TECHNICAL MANUAL

AVIATION UNIT AND INTERMEDIATE MAINTENANCE MANUAL

HELICOPTER, OBSERVATION OH-6A

This manual supersedes TM 55-1520-214-20, dated 3 May 1974 and TM 55-1520-214-34, dated 17 May 1974, including all changes.

This copy is a reprint which includes current pages from Changes 1 through 25

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31 DECEMBER 1976

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CHANGE
NO. 25

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---	1-36.1/1-36.2

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- - -	1-42A/1-42B
9-13 and 9-14	9-13 and 9-14

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Warning	a and b	a thru d
Preface	P-1 thru P-4	P-1 thru P-4
Chapter 1	1-3 and 1-4	1-3 and 1-4
	1-5 thru 1-16	1-5 thru 1-12A/1-12B
	— — —	1-13 thru 1-16
	1-21 thru 1-24	1-21 thru 1-24
	1-27 thru 1-32A/1-32B	1-27 thru 1-32
	1-33 thru 1-43/1-44	1-33 thru 1-45/1-46
Chapter 2	2-13 and 2-14	2-13 and 2-14
	2-35 thru 2-38	2-35 thru 2-38
	2-63 and 2-64	2-63 and 2-64
	2-71 thru 2-74	2-71 thru 2-74
	2-81 thru 2-84	2-81 thru 2-84
	2-87 thru 2-90	2-87 thru 2-90
	2-103 and 2-104	2-103 and 2-104
	2-107 and 2-108	2-107 and 2-108
Chapter 3	3-7 and 3-8	3-7 and 3-8
Chapter 4	4-7 and 4-8	4-7 and 4-8
	4-9 and 4-10	4-9 and 4-10
	4-11 and 4-12	4-11 and 4-12
	4-47 and 4-48	4-47 and 4-48
	4-73 and 4-74	4-73 and 4-74
	4-77 thru 4-80	4-77 thru 4-80
Chapter 5	5-1 thru 5-8	5-1 thru 5-8
	5-19 and 5-20	5-19 and 5-20
	5-27 and 5-28	5-27 and 5-28
	5-35 thru 5-38	5-35 thru 5-36B
	— — —	5-37 and 5-38
	— — —	5-40A/5-40B
	5-43 thru 5-48	5-43 thru 5-46A/5-46B
	— — —	5-47 and 5-48
	5-59 and 5-60	5-59 and 5-60

	Remove pages	Insert pages
Chapter 6	6-3 thru 6-6	6-3 thru 6-6
	6-9 and 6-10	6-9 and 6-10
	6-13 and 6-14	6-13 and 6-14
	6-15 and 6-16	6-15 and 6-16
	6-19 thru 6-22	6-19 thru 6-22
	6-25 thru 6-30	6-25 thru 6-30B
	6-31 and 6-32	6-31 and 6-32
Chapter 8	8-13 thru 8-22	8-13 thru 8-14A/8-14B
	– – –	8-15 thru 8-20D
	– – –	8-21 and 8-22
Chapter 9	9-7 and 9-8	9-7 and 9-8
Chapter 10	10-7 and 10-8	10-7 and 10-8
	10-11 thru 10-14	10-11 thru 10-14
	10-29 thru 10-47	10-29 and 10-30
Chapter 11	11-3 and 11-4	11-3 and 11-4
	11-11 thru 11-18	11-11 thru 11-18
	11-23 thru 11-26	11-23 thru 11-26
	11-45 and 11-46	11-45 and 11-46
	11-61 and 11-62	11-61 and 11-62
	11-67 and 11-68	11-67 and 11-68
Chapter 17	17-1 and 17-2	17-1 and 17-2
Appendix A	A-1 thru A-3/A-4	A-1 thru A-4
Appendix C	C-1 and C-2	C-1 and C-2
Appendix D	D-1 thru D-4	D-1 thru D-4
	D-19	D-19 thru D-21/D-22
Appendix E	E-3 and E-4	E-3 and E-4
	E-7 and E-8	E-7 and E-8
	E-11 and E-12	E-11 and E-12
Index	Index 1 thru Index 8	Index 1 thru Index 8
Fold-Outs	FO-6	FO-6
	FO-14	FO-14

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NO. 20

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Warning	a and b	a and b
Chapter 1	1-3 and 1-4	1-3 and 1-4
	1-9 and 1-10	1-9 and 1-10
		1-41A thru 1-41C/1-41D
Chapter 5	5-53 and 5-54	5-53 and 5-54
	5-54A and 5-54B	5-54A and 5-54B

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Chapter 1	1-1 and 1-2 1-29 thru 1-32 1-35 and 1-36	1-1 and 1-2 1-29 thru 1-32 1-35 and 1-36

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	Warning	a and b
Chapter 1	1-3 and 1-4	1-3 thru 1-4A/1-4B
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Chapter 1	1-9 and 1-10	1-9 and 1-10
	1-15 and 1-16	1-15 and 1-16
	1-29 thru 1-32	1-29 thru 1-32
	1-35 thru 1-38	1-35 thru 1-38
	1-43/1-44	1-43/1-44
Chapter 4	4-31 and 4-32	4-31 and 4-32
Chapter 6	6-14A/6-14B	6-14A/6-14B
	6-17 and 6-18	6-17 and 6-18
Appendix B	B-13 and B-14	B-13 and B-14

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Chapter 1	1-27 and 1-28	1-27 and 1-28

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Change No. 13 will be forthcoming.

TM 55-1520-214-23
C 14

CHANGE
No. 14

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Chapter 5	5-57 thru 5-58C/5-58D	5-57 thru 5-58C/5-58D

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Chapter 1	1-9 and 1-10	1-9 and 1-10
	1-29 thru 1-32	1-29 thru 1-32
	1-35 and 1-36	1-35 and 1-36
	1-41 and 1-42	1-41 and 1-42
Chapter 2	2-23 and 2-24	2-23 and 2-24
	2-103 and 2-104	2-103 and 2-104
Chapter 3	3-7 and 3-8	3-7 and 3-8
Chapter 5	5-5 and 5-6	5-5 and 5-6
Chapter 6	6-25 and 6-26	6-25 and 6-26
Chapter 9	9-11 and 9-12	9-11 and 9-12.1/9-12.2
	9-15 thru 9-18	9-15 thru 9-18
	9-49 thru 9-52	9-49 thru 9-52
Appendix A	A-1 thru A-3/A-4	A-1 thru A-3/A-4
Appendix B	B-13 and B-14	B-13 and B-14
Appendix E	E-1 thru E-8	E-1 thru E-8
	FO-4	FO-4
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Chapter 10	10-1 and 10-2	10-1 thru 10-2A/10-2B

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Chapter 1	1-37 and 1-38	1-37 and 1-38
	1-43/1-44	1-43/1-44
Chapter 5	5-49 thru 5-52	5-49 thru 5-52A/5-52B
	5-53 and 5-54	5-53 thru 5-54B
	5-55 thru 5-58	5-55 thru 5-58C/5-58D
	5-59 and 5-60	5-59 thru 5-60A/5-60B
	5-61 thru 5-65	5-61 thru 5-66
Chapter 9	9-9 and 9-10	9-9 and 9-10
INDEX	INDEX-1 and INDEX-2	INDEX-1 and INDEX-2
	INDEX-7 and INDEX-8	INDEX-7 and INDEX-8
	FO-1	FO-1
	FO-9	FO-9

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Chapter 4	4-10A thru 4-10C/4-10D	4-10A thru 4-10C/4-10D

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HELICOPTER, OBSERVATION OH-6A

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Chapter 1	1-43	1-43/1-44
Chapter 6	6-13 and 6-14	6-13 thru 6-14A/6-14B

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Preface	P-3 and P-4	P-3 and P-4
Chapter 10	10-1 thru 10-4	10-1 thru 10-4

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Chapter 1	1-9 thru 1-12	1-9 thru 1-12
	1-29 and 1-30	1-29 and 1-30
	1-33 and 1-34	1-33 and 1-34
	1-39 and 1-40	1-39 and 1-40
Chapter 2	2-5 and 2-6	2-5 thru 2-6A/2-6B
	2-11 and 2-12	2-11 and 2-12
	2-39 and 2-40	2-39 thru 2-40A/2-40B
	2-73 and 2-74	2-73 and 2-74
	2-79 and 2-80	2-79 and 2-80
Chapter 4	4-3 and 4-4	4-3 and 4-4
	---	4-8A/4-8B
	4-9 thru 4-12	4-9 thru 4-12
	4-83 and 4-84	4-83 and 4-84
Chapter 5	5-19 and 5-20	5-19 and 5-20
Chapter 6	6-9 and 6-10	6-9 and 6-10
Chapter 8	8-1 and 8-2	8-1 and 8-2
	8-7 and 8-8	8-7 and 8-8
Chapter 9	9-49 and 9-50	9-49 and 9-50
Chapter 10	10-19 and 10-20	10-19 and 10-20
	10-31 and 10-32	10-31 and 10-32
	10-43 and 10-44	10-43 and 10-44
Chapter 13	13-1 and 13-2	13-1 and 13-2
	13-5 and 13-6	13-5 and 13-6
	13-11	13-11/13-12
Appendix A	A-1 and A-2	A-1 and A-2
Appendix B	B-1 and B-2	B-1 and B-2
	B-17 thru B-27/B-28	B-17/B-18
Foldouts	FO-10	FO-10
	FO-18	FO-18

2. New or changed text material is indicated by a vertical bar in the margin. An illustration change is indicated by a miniature pointing hand.

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Chief of Staff

Official:

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Major General, United States Army
The Adjutant General

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Aviation Unit and Intermediate
Maintenance Manual

HELICOPTER, OBSERVATION OH-6A

TM 55-1520-214-23, 31 December 1976, is changed as follows:

1. Remove and insert pages as indicated below.

	Remove pages	Insert pages
Chapter 1	1-31 and 1-32	1-31 thru 1-32A/1-32B

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	Remove pages	Insert pages
Chapter 1	1-31 and 1-32	1-31 and 1-32

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C 3

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TM 55-1520-214-23, 31 December 1976, is changed as follows:

1. Warning page is changed as follows.

2. Remove and insert pages as indicated below.

	Remove pages	Insert pages
Chapter 9	9-9 and 9-10	9-9 and 9-10

3. New or changed text material is indicated by a star preceeding and following the changed material. An illustration change is indicated by a miniature pointing hand.

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TM 55-1520-214-23
C 2

CHANGE No. 2

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Aviation Unit and Intermediate
Maintenance Manual

HELICOPTER, OBSERVATION OH-6A

TM 55-1520-214-23, 31 December 1976, is changed as follows:

1. Remove and insert pages as indicated below.

	Remove pages	Insert pages
Chapter 4	4-9 and 4-10	4-9 thru 4-10C/4-10D

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URGENT

CHANGE No. 1

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Aviation Unit and Intermediate
Maintenance Manual

HELICOPTER, OBSERVATION OH-6A

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	Remove pages	Insert pages
Appendix B	B-1 thru B-4	B-1 thru B-4 B-17 thru B-27/B-28

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Major General, United States Army
The Adjutant General

DISTRIBUTION:
To be distributed in accordance with DA Form 12-31 (qty rqr block no. 192) Organizational Maintenance Requirements for OH-6 aircraft.

WARNING PAGE

Personnel performing instructions involving operations, procedures, materials, and practices which are included or implied in this technical manual shall observe the following instructions. Disregard of these warning and precautionary information can cause serious injury or death. Refer to FM 21-11 for first aid data to treat injuries resulting from working on the helicopter.

Warnings, cautions, and notes emphasize important and critical instructions.

WARNING

An operating procedure or practice which, if not correctly followed, will result in personnel injury or loss of life.

CAUTION

An operating procedure or practice which, if not strictly observed, will result in damage or destruction of equipment.

NOTE

An operating procedure or condition which it is essential to highlight.

STARTING AIRCRAFT

Minimum rotor clearance is low enough to cause injury to personnel. Coordinate all cockpit actions with ground observer. Ensure that rotor and blast areas are clear and fire guard is posted. Secure safety belts and shoulder harness in unoccupied seats to prevent fouling controls.

GROUND OPERATION

The engine will be started and operated only by authorized personnel. Refer to AR 95-1.

GROUNDING AIRCRAFT

All metal apparatus shall be grounded to avoid the danger to igniting test fluid fumes or creating electrical shock.

HIGH VOLTAGE

Before removing igniter, be sure ignition system has been OFF for at least 5 minutes. Dissipate all energy stored in the condenser by grounding the igniter lead to the engine using an insulated screwdriver. Be sure all electrical power is OFF before disconnecting static inverters, radio equipment or other electrical components.

Electrical Equipment Maintenance. Do not wear rings, watches, or metal jewelry when working around electrical equipment. Serious burns can result.

Dangerous Voltages Exist in the Electronic Equipment. Be careful when working on the 150- and 300-volt dc circuits and on the ac generator 115- and 200-volt ac outputs. Serious burns can result.

Dangerous Voltages may Exist at Antenna Terminals. Be careful when working near the antenna or the antenna terminals. Radio-frequency (rf) high voltages exist at these points when transmitters are operating. Contact with radiating antennas cause serious rf burns.

HANDLING HYDRAULIC FLUID (MIL-H-5606)

When handling hydraulic fluid (MIL-H-5606) observe the following:
— Prolonged contact with liquid or mist can irritate eyes and skin.
— After any prolonged contact with skin, immediately wash contact area with soap and water. If liquid contacts eyes, flush them immediately with clear water.
— If liquid is swallowed, do not induce vomiting; get immediate medical attention.
— Wear rubber gloves when handling liquid. If prolonged contact with mist is likely, wear an appropriate respirator.
— When fluid is decomposed by heating, toxic gases are released.

HANDLING FUEL

Dangerous Fuel Handling. Incorrect fuel handling causes fire hazards. Ground the helicopter when fueling or defueling.

ACIDS

DANGEROUS CHEMICALS ARE USED IN NICKEL-CADMIUM BATTERIES. The electrolyte used in nickel-cadmium batteries contains potassium hydroxide (KOH), which is a caustic chemical agent. Serious and deep burns of body tissue will result if the electrolyte comes in contact with the eyes or any part of the body. Use rubber gloves, rubber apron, and protective eye covering when handling the battery. If accidental contact with the electrolyte is made, use ONLY clean water and immediately (seconds count) flush contaminated area. Continue flushing with large quantities of clean water. Seek medical attention immediately. Before removing or installing the battery, insure that the battery switch is off and the battery has cooled down if overheated. Removal or installation of the battery connector while the battery is under load may result in explosion, electrical arcing and possible severe burns to personnel.

Acids and Alkalines. Do not add water to acids. A violent action will result. Acids should be added to water in small quantities. Ruststripper is an alkaline solution. Avoid contact with the skin. Wear protective clothing. Wash thoroughly after using.

ROTOR BLADES

Before approaching empennage, be certain to notify operator at controls of helicopter that personnel will be in the tail rotor area. Approach the empennage from the horizontal stabilizer side only.

NOISE LEVEL

Sound pressure levels in this aircraft during some operating conditions exceed the Surgeon General's hearing conservation criteria, as defined in TB MED 251. Hearing protection devices, such as the aviator helmet or ear plugs, are required to be worn by all personnel in and around the aircraft during its operation.

RADIATION

Some instruments contain radioactive material. (See TB 55-1500-314-25.) Do not try to disassemble these instruments. They present no radiation hazard unless seal is broken. If you think seal is broken, do not remove instrument from aircraft until you consult Base Radioactive Protection Officer (AR 40-15). Use a beta-gamma rad ac meter AN/PDR-27 or equivalent to determine if instrument contains radioactive material (radium).

FIRE EXTINGUISHER

Avoid repeated or prolonged exposure to high concentration of bromochloromethane (CB) or decomposition products. CB is a narcotic agent of moderate intensity but prolonged duration. It is less toxic than carbon tetrachloride, methylbromide, or products of combustion. Take normal precautions while using bromochloromethane. Use oxygen masks when available.

Monobromotrifluoromethane (CF3Br) is highly volatile but is not easily detected by its odor. Although nontoxic, it is about the same as other freons and carbon dioxide, causing danger to personnel primarily by reduction of oxygen available for proper breathing. Do not allow the liquid to come into contact with your skin. It may cause frostbite or low temperature burns.

ARMAMENT

Loaded weapons, or weapons being loaded or unloaded, shall be pointed in a direction which offers the least exposure to personnel or property in the event of accidental firing. Personnel should remain clear of hazardous area of all loaded weapons.

ANY ROTATION OF THE GUN ARMAMENT SUBSYSTEM BARRELS WILL CAUSE THE GUN TO FIRE. Upon landing, immediately alert personnel to probable presence of live rounds in the gun. Summon armament repairman to clear weapon.

CARBON MONOXIDE

Toxic carbon monoxide fumes may be present inside the helicopter whenever the APU or engines are operating with the cargo ramp open. Ventilate the cockpit.

TOXIC CHEMICALS

Solvent and Cleaning Solutions. These materials are generally toxic and many (toluene, benzene, xylene, methyl-ethyl-ketone, perchlorethylene, naphtha, trichloroethylene) are highly flammable. Work in a well-ventilated area away from open flames. Avoid inhaling fumes and prolonged contact with the skin. Wear protective clothing and goggles. Wash thoroughly after using.

Windshield Repellant. Do not let windshield rain repellant contact open flame. Deadly hydrogen fluoride gas could be generated. Wash hands with soap and water after handling repellant.

Antiseize Compounds. Some antiseize compounds are irritants. Avoid inhaling fumes and contact with the skin. Wear protective clothing. Wash thoroughly after using.

Paints, Varnishes, Dopes, Thinners, Lubricants, and Fuels. These materials are generally highly flammable and may be irritants. Work in a well-ventilated area away from open flames. Avoid inhaling fumes and prolonged contact with the skin. Wash thoroughly after using.

Epoxy Resins, Cements, and Adhesives. These materials may contain toxic or irritating substances. They may also be flammable. Work in a well ventilated area away from open flames. Wear protective clothing. Avoid contact with the skin. Wash thoroughly after using.

COMPRESSED AIR

Do not use more than 30 psi compressed air for cleaning purposes. Debris trajected under pressure can cause injury to eyes. Use source of compressed air under 30 psi and eye protection to prevent injury to personnel.

FOREIGN OBJECTS

Make sure area is clear of foreign objects before closing access doors, panels, and fairings. If the area is not cleared, damage to components and systems could result in personal injury or death.

Aviation Unit and Intermediate Maintenance Manual

HELICOPTER, OBSERVATION OH-6A

PREFACE

P-1. GENERAL.

P-2. Contents. a. This manual is the official document for Aviation Unit and Intermediate Maintenance of Army Model OH-6A Helicopters.

b. The purpose of this manual is to familiarize you with the maintenance functions to be performed at the Aviation Unit and Intermediate Maintenance levels. The Table of Contents for this manual is provided to assist in determining the chapter in the manual in which individual functions are covered. This manual provides all essential information for personnel to accomplish Aviation Unit and Intermediate Maintenance on the complete airframe, its components, and systems, excluding armament and avionics subsystem as indicated for Aviation Unit and Intermediate Maintenance activities in the Maintenance Allocation Chart (MAC). (Refer to Appendix B.)

P-3. QUALITY ASSURANCE.

P-4. Quality Assurance/Quality Control (QA/QC). Personnel will assure proper maintenance has been performed by verifying dimensions and tolerances contained throughout this technical manual have been complied with.

P-5. DESCRIPTION-ARMY MODEL OH-6A HELICOPTER.

P-6. General Features. The Army Model OH-6A Helicopter is a turbine powered rotary-wing aircraft constructed primarily of aluminum alloy. The main rotor is a four-bladed, fully articulated type and the tail rotor is a two-bladed anti-torque rotor of the semi-rigid type. The missions for which the aircraft are employed include visual observation, target acquisition, reconnaissance, and command and control. The aircraft is readily adaptable to utility tasks at the combat company level without the use of special kits or special attachments. The aircraft consists of two main structural sections, the fuselage and the tailboom. See figure P-1 for major components.

P-7. Fuselage. The fuselage (incorporating a central framework consisting of a mast support structure, two bulkheads, and a center beam) is a semi-monocoque structure that is divided into three main sections. The forward section includes a pilot compartment and a cargo and/or passenger compartment. The pilot compartment is normally equipped with seating for the pilot and copilot/observer. The seat on the right side of the pilot's compartment (when looking forward) is the pilot's seat (command position). The cargo compartment in back of the pilot compartment contains provisions for installation of two additional passenger seats. The cargo compartment seats may be easily folded out of the way or completely removed for the accommodation of cargo. Standard torso range-extension tanks may be installed in the cargo compartment seats. The aft section includes the structure for tailboom attachment and houses the engine. The lower section is divided by the center beam and provides a housing for the two fuel cells.

P-8. Tailboom. The tailboom is a monocoque structure of aluminum alloy skin. The tailboom serves as the supporting structure to which the horizontal stabilizer, the upper and lower vertical stabilizers, and the tail rotor transmission and tail rotor are attached. In addition, the tailboom houses the tail rotor transmission drive shaft, and the tail rotor blade angle control rod.

P-9. Main Rotor Group. The main rotor group consists of four rotor blades, a fully articulated main rotor hub assembly that incorporates offset flapping hinges, a scissors assembly, and a swashplate and associated mixer control mechanisms. The main rotor blades are secured to the rotor hub assembly by quick-release lever type pins that permit easy and fast removal of the blades.

P-10. Landing Gear. The landing gear is the horizontal, skid-type gear, attached to the fuselage at 12 points. The landing gear is nonretractable. The struts, from the fuselage to the skids, are covered with aerodynamic fairings. Hydraulic dampers, between the struts and structure, act as shock absorbers to cushion landings.

P-11. HELICOPTER SERIES NUMBERS.

P-12. Explanation of Helicopter Series Numbers. *a.* Although all OH-6A aircraft are functionally similar, physical differences in avionics equipment, electrical harnesses and panels, instrument panel, engine air intake filters, etc, occasionally require separate maintenance instructions and illustrations for each of the alternate arrangements or configurations.

b. To readily distinguish between the various equipment configurations without repetitious review of helicopter serial number listings, the manufacturing series numbers (or production lot numbers) are included in paragraph titles, figure titles and/or table titles where necessary to highlight applicability of the information.

c. If no aircraft series or serial number limitations are included in a given paragraph, figure or table, and the differences are not obvious, the information shall be considered applicable for all aircraft. Minor differences not relative to all aircraft in a series are separately noted where they apply.

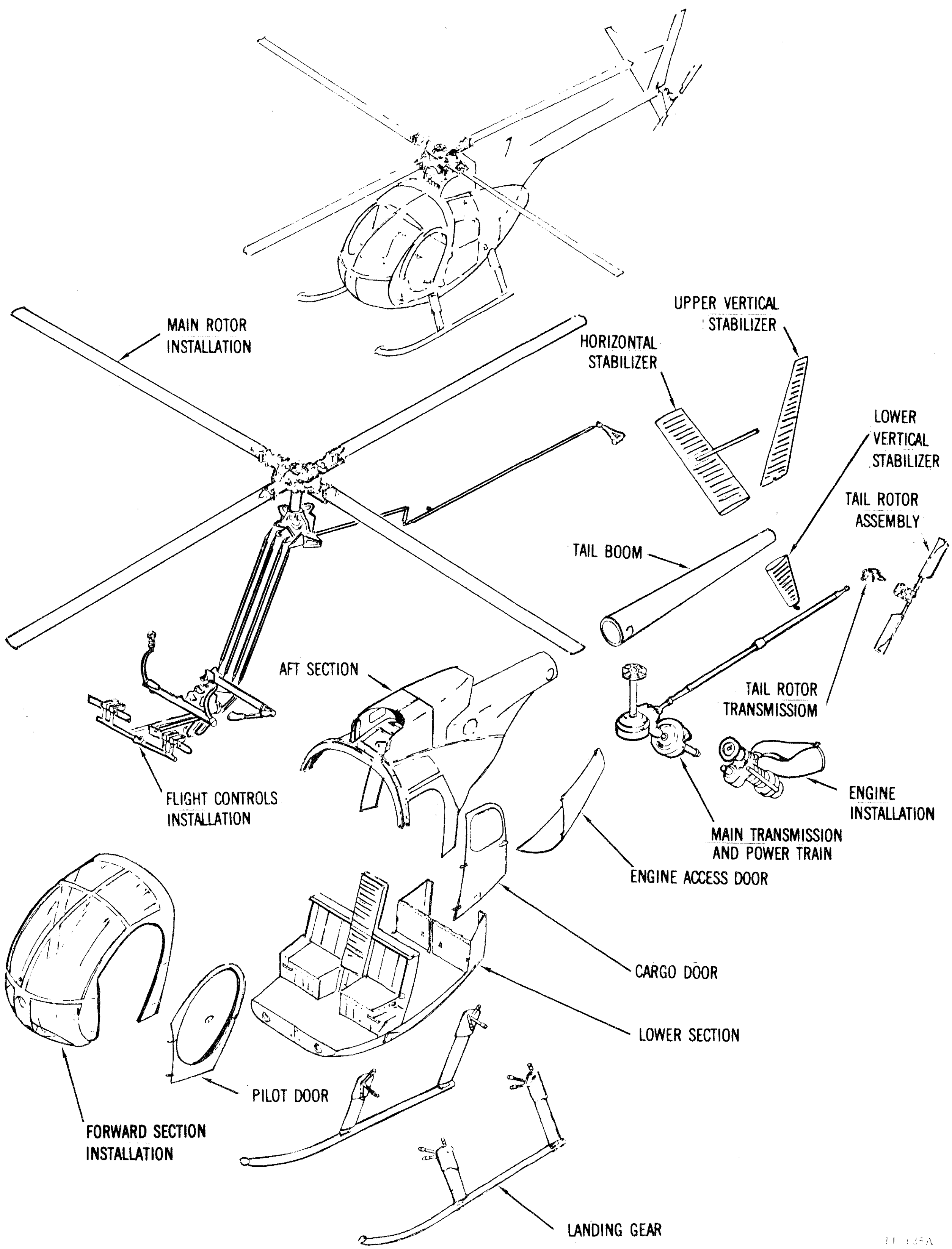


Figure P -1. Major Components

d. The following list provides a cross-index of OH-6A serial numbers with the corresponding aircraft series numbers used throughout the manual.

Aircraft Serial No.	Aircraft Series
65-12916 thru 65-13003	1
66-7775 thru 66-7899	1
66-7900	2
66-7901 thru 66-7920	1
66-7921 thru 66-7924	2
66-7925 thru 66-7938	1
66-7939	2
66-7940 thru 66-7942	1
66-14376 thru 66-14419	1
66-17750	2
66-17751 thru 66-17833	1
67-16000 thru 67-16126	1
67-16127 thru 67-16686	2
68-17140 thru 68-17369	3
69-15960 thru 69-16075	3

P-13. REPORTING OF ERRORS.

P-14. Recommended Changes.

You can help improve this manual. If you find any mistakes or if you know of a way to improve the procedure, please let us know. Mail your letter, DA Form 2028 (Recommended Changes to Publications and Blank Forms), or DA Form 2028-2 located in the back of this manual direct to: Commander, US Army Aviation Systems Command, ATTN: DRSAV-MPSD, 4300 Goodfellow Blvd., St. Louis, MO 63120. A reply will be furnished direct to you.

P-15. DESTRUCTION TO PREVENT ENEMY USE.

P-16. Destruction Procedures. Refer to TM 750-244-1-5 for procedures dealing with the destruction of aircraft and associated equipment to prevent enemy use.

P-17. MAINTENANCE OF FORMS AND RECORDS.

P-18. Record Keeping. Maintenance forms, records, and reports which are to be used by maintenance personnel at all maintenance levels are listed in and prescribed by TM 38-750 and TB 55-1500-307-24.

P-19. AUTHORITY FOR SUBSTITUTION.

P-20. Substitution Control. Substitution or interchange of items of materiel for maintenance of Department of the Army aircraft shall not be authorized, nor shall orders be issued for shipment. Substitution or interchangeability shall be authorized only by U.S. Army Aviation Systems Command.

P-21. SPECIAL TOOLS AND EQUIPMENT.

P-22. Special Tools and Equipment Lists. Aviation Unit (AVUM) and Intermediate maintenance (AVIM) special tools and equipment will be found in TM 55-1520-214-23P RPSTL manual. Special tools and test equipment used in accomplishing complex tasks are listed in this manual.

P-23. CALIBRATION.

P-24. Calibration Checks. Aircraft components, accessories, and instruments requiring calibration are listed in the Maintenance Allocation Chart (MAC), appendix B. Special tools and test equipment shall be calibrated as specified in TB 750-25.

P-25. STORAGE.

P-26. Storage Procedures. Storage of aircraft and equipment shall be accomplished as specified in TM 740-90-1, Appendix E for storage of aircraft.

P-27. DESIGNATOR SYMBOLS.

P-28. Designator System. Designator symbols are not used. Effectivity of equipment configurations is denoted according to aircraft series, as listed in paragraph P-11.

P-29. EXPLANATION OF CHANGE SYMBOLS.

P-30. Use of Change Symbols. Changes, except as noted below, to the text and tables, including new material on added pages, are indicated by a vertical line in the outer margin extending close to the entire area of the material affected; exception: pages with emergency markings, which consist of black diagonal lines around three edges, may have the vertical lines or change symbol placed along the inner margins. Symbols show current changes only. A miniature pointing hand symbol is used to denote a change to an illustration. However, a vertical line in the outer margin, rather than miniature pointing hands, is utilized when there have been extensive changes made to an illustration. Change symbols are not utilized to indicate changes in the following:

a. Introductory material.

b. Indexes and tabular data where the changes cannot be identified.

c. Blank space resulting from the deletion of text, an illustration, or a table.

d. Correction of minor inaccuracies, such as spelling, punctuation, relocation of material, etc., unless such correction changes the meaning of instructive information and procedures.

P-31. EXPLANATION OF MAINTENANCE LEVELS AND PERSONNEL TITLES.

P-32. Maintenance Level Limitations. Unless otherwise noted by an "(AVIM)" at start of paragraph, all procedures apply to Aviation Unit Maintenance. All procedures can be accomplished by an airframe mechanic (MOS 67V) unless otherwise noted in a pre-maintenance table.

P-33. Engineering Authorization. All requests for engineering authorization, when required by this manual will be forwarded to USAAVSCOM, ATTN: DRSAV-MEC, 4300 Goodfellow Blvd., St. Louis, MO 63120. Urgent requests shall be clearly identified to insure priority handling and response. The requests shall include detailed information on the problem, e.g., sketches, photographs, dimensional data, etc., to assist in the evaluation and prompt reply.

CHAPTER I

AIRCRAFT GENERAL

SECTION I SERVICING

1-1. SERVICING.

1-2. General — Servicing. Servicing of the aircraft includes replenishing of fuel, changing or replenishment of oil, and other such maintenance functions. Fuels, oils and other servicing materials are listed in table 1-1 and TM 55-1520-214-10. The locations of servicing points are shown in figure 1-1.

CAUTION

Use extreme care when applying any type of lubricant (grease, oil, dry-film, etc) in the vicinity of teflon bearings. Most lubricants will form a dirt retaining film or have otherwise detrimental effects that can cause rapid deterioration of the bearing surfaces.

1-3. FUEL SYSTEM.

1-4. General — Servicing Fuel System. The fuel system has two fuel cells that are interconnected for simultaneous flow and venting. The fuel gravity filler cap is on the right side of the fuselage aft of the cargo compartment door. On aircraft with a crash-resistant (CR) fuel system, a closed circuit fuel filler cap is located directly under the forward end of the right cargo compartment door sill. On these aircraft, the gravity fill cap must be removed and the attaching cable hooked into a notched tab to hold open a flapper valve in the filler neck prior to fueling.

1-5. Servicing Precautions.

CAUTION

Turn off electrical switches and disconnect external power from the aircraft.

a. The aircraft must be electrically grounded prior to defueling. The possibility of static discharge (from difference in electrical potential) and presence of fuel vapors always present fire and explosion hazards.

b. The refueling vehicle should be parked a minimum of 20 feet from the aircraft during the fueling operation. Before starting the fueling operation, always ground the fueling nozzle or fuel truck to the GROUND HERE receptacle (fig. 1-1) or to another bare metal location.

c. Refuel aircraft with correct fuel (C51) as soon after landing as possible to prevent moisture condensation and to keep the aircraft as heavy as possible in case of winds.

d. Keep fueling nozzle free of all foreign matter.

e. Check filler cap(s) for security after fueling.

1-6. Filling — Gravity Fuel System. *a.* Remove filler cap aft of right cargo compartment door.

b. If required, pull filler cap cable and notch to open flapper valve.

c. Refuel aircraft.

1-7. Filling — Closed Circuit Fuel System. *a.* Remove filler cap below right cargo compartment door sill.

b. Refuel aircraft using equipment with compatible hose connections.

NOTE

The fuel venting system may cause an air pressure buildup in the fuel cells resulting in damage to the cargo decks or a short fuel load (one which does not fully fill both fuel cells before the closed-circuit fuel nozzle turns off). When fueling into the closed-circuit receiver, one of two procedures must be used; either remove the gravity fill cap and assure that the flapper valve is open or run the fuel nozzle in increments of 30 seconds on and 30 seconds off until the fuel cells are full.

c. Reinstall filler cap. Be sure that the cap retention cable is coiled and positioned inside the receiver well so that no interference with locking mechanism and sealing ring occurs when the cap is installed.

1-8. Draining — Fuel System. Fuel draining should be accomplished with the aircraft as level as possible. The fuel system may be defueled in two ways. One

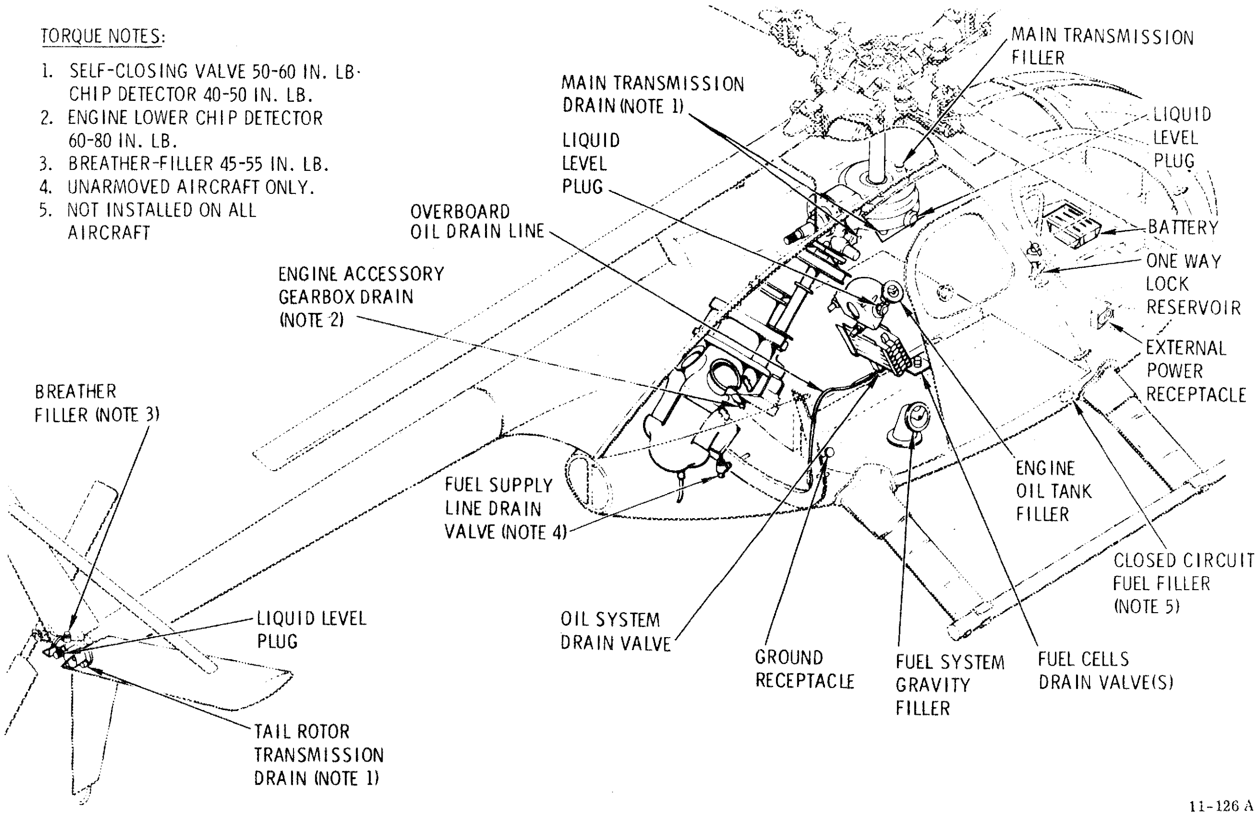


Figure 1-1. Servicing Points.

method is to defuel through the filler port by using a pump. The second method is to open the system drain valves (fig. 1-1).

a. On aircraft without closed circuit refueling provisions, the drain valves are located on the fuselage underside and in the engine compartment. (Armored aircraft have only the main fuel cells drain valve.) The drain valves are spring-loaded in the closed position. The valves are opened by depressing the plunger with a screw driver and rotating the plunger either clockwise or counter-clockwise to the stop. Be sure to close and lockwire the drain valves after defueling.

b. On aircraft with closed circuit refueling provisions, two adjacent drain valves (left and right fuel cell) are located on the fuselage underside. The valves are spring-loaded in the closed position. Rotate the valve center slot counter-clockwise and hold open for drainage.

NOTE

For rapid drainage, remove the drain valves and preformed packings (O-rings), using a wide-blade screwdriver. Reinstall drain valves with new O-rings. Lubricate valve threads and O-ring with petrolatum (C73) before installation.

1-9. ENGINE OIL SYSTEM.

1-10. Filling — Engine Oil System. The engine oil filler cap is on the right side of the fuselage. A liquid level plug for checking the oil level in the tank is visible through a transparent window near the filler.

a. Check the oil level.

b. Replenish with correct oil (C67) until the level reaches FULL on the plug.

c. Make certain that the oil tank filler cap is securely tightened immediately after servicing.

1-11. Draining — Engine Oil System. *a.* Remove sound insulation from cargo compartment aft bulkhead right access door and remove the door.

b. Place a suitable container under the overboard oil drain line where it exits the fuselage underside at the firewall.

c. Remove cap from oil tank filler (fig. 1-1). Pull out knurled spring-loaded button to open valve in engine oil drain line just below engine oil cooler. Rotate button and valve poppet so that locking pin rests on shoulders of pin slot.

d. After draining the oil from the tank, reinstall the filler cap and close oil drain valve; ensure that poppet pin is in stop slot.

e. Install access door and sound insulation.

f. To drain approximately 1/2 pint of residual oil from engine accessory gearbox, remove wiring plug and the lower chip detector (fig.1-1). Use a suitable container to catch the oil. Check that detector packing is serviceable (replace if necessary), reinstall detector **(60 TO 80 INCH-POUNDS TORQUE)**, reconnect wiring plug and secure with 0.032-inch lockwire (C57).

NOTE

The engine oil filter should be cleaned whenever the oil is changed (TM 55-2840-231-23.) If the two oils are mixed refer to TM 55-2840-231-23 for oil changeover procedures.

Lubricating oil made to MIL-L-7808 by Shell Oil Company under their part number 307, qualification number 7D-1 shall NOT be used in OH-6A engine or aircraft systems. It contains additives which are harmful to seals in the system.

1-12. Combining MIL-L-23699 and MIL-L-7808 Oils — Engine Oil System.

CAUTION

Do not mix MIL-L-23699 oil with MIL-L-7808 oil except in case of an emergency. Operation is limited to 6 hours.

If the two oils are mixed refer to TM 55-2840-231-23 for oil changeover procedure.

1-13. MAIN TRANSMISSION.

1-14. Filling — Main Transmission. Transmission (gearbox) oil should be replaced with new oil whenever it is drained from the gearbox.

a. Check transmission oil level in liquid level plug (fig. 1-1).

b. Replenish with correct oil (C67) until the level reaches full mark on the plug.

c. Fill main transmission by lifting breather-filler cap and inserting spout of oil can into opening. Check that spring-loaded cap closes when oil can spout is removed.

1-15. Draining — Main Transmission. *a.* Remove, in order, sound insulation, transmission access cover (para 2-11), transmission drain assembly (para 2-14) and main transmission cover (para 2-20).

b. Position a suitable (minimum 4-quart/4-liter) container under chip detectors (fig. 1-1).

c. Remove wire leads, lockwire, chip detectors, and self-closing valves.

d. If damaged, replace packings used with chip detectors and self-closing valves.

e. After oil has drained, install self-closing valves **(50 TO 60 INCH-POUNDS TORQUE)** and chip detectors **(40 TO 50 INCH-POUNDS TORQUE)**. Secure valve to gearbox and detector to valve with 0.032-inch lockwire (C57). Reconnect wire leads.

f. Remove, inspect, clean and reinstall transmission oil filter (chapter 6).

g. Reinstall, in order, the main transmission cover, the drain assembly, the main gearbox access cover, and the sound insulation.

1-16. TAIL ROTOR TRANSMISSION.

1-17. Filling — Tail Rotor Transmission. Tail rotor transmission oil should be replaced with new oil whenever it is drained from the gearbox.

a. Check tail rotor transmission oil level in liquid level plug (fig. 1-1).

b. Replenish with correct oil (C67) until the oil level reaches the full mark on the liquid level plug.

c. Fill tail rotor transmission by removing lockwire, unscrewing breather-filler (fig. 1-1) and pouring oil into transmission. Check that filler O-ring packing is serviceable (replace if necessary), reinstall breather-filler **(45 TO 55 INCH-POUNDS TORQUE)**, and secure with 0.032-inch lockwire (C57).

NOTE

Breather-filler plugs that have the threaded insert are installed with the breather hole rearward (chapter 6).

1-18. Draining — Tail Rotor Transmission. *a.* Position suitable container under the chip detector (fig.1-1).

b. Remove wire lead, lockwire, chip detector, and self-closing valve.

c. If damaged, replace packings used with chip detector and self closing valve.

d. After oil has drained, install self-closing valve **(50 TO 60 INCH-POUNDS TORQUE)** and chip detector **(40 TO 50 INCH-POUNDS TORQUE)**. Lockwire valve to gearbox and detector to valve with 0.032-inch lockwire (C57). Reconnect wire lead.

e. Wipe dry any oil spillage with a clean cloth moistened by solvent (C94).

1-19. ONE-WAY LOCK.

WARNING

Prolonged contact with hydraulic fluid (MIL-H-5606) liquid or mist can irritate eyes and skin. After any prolonged contact with skin, immediately wash contacted area with soap and water. If liquid contacts eyes, flush immediately with clear water. If liquid is swallowed, do not induce vomiting; get immediate medical attention. Wear rubber gloves when handling liquid. If prolonged contact with mist is likely, wear an appropriate respirator. When fluid is decomposed by heating, toxic gases are released.

1-20. Servicing — One-Way Lock. Remove the screws and washers from the pilot's collective stick cover and lift cover to check the oil level. If oil level in reservoir (fig. 1-1) is low, lift filler cap and add the correct oil (C48 or C48A) as needed. Reinstall pilot's collective stick cover.

NOTE

If oil level is consistently found to be low, the one-way lock should be replaced.

1-21. BATTERY.

1-22. Handling and Servicing Precautions — Battery.

WARNING

The electrolyte used in nickel-cadmium batteries contains potassium hydroxide, a caustic chemical agent. Serious burns will result if the electrolyte contacts the skin. Explosive gasses may be released from the battery during charging. Before removing the battery from the aircraft, make sure that the power selector switch is at the OFF position. Removal or installation of the battery connector while the battery is under load may result in explosion, electrical arcing and possible severe burns to personnel.

WARNING

Attempts to remove the battery immediately following a battery overtemperature condition may result in serious burns to the hands or possible explosion hazards. Battery overtemperature may be identified by electrolyte spewing or heavy fumes. If this condition is detected, allow the battery to thoroughly cool (approximately one hour) prior to removal.

a. Nickel-cadmium batteries provide numerous advantages over lead-acid batteries. However, because of important handling and servicing differences between nickel-cadmium batteries and the more familiar lead-acid batteries it is extremely important to understand and observe the following precautions before performing any nickel-cadmium battery handling or servicing operations.

b. Satisfactory battery operation is largely dependent upon proper operation of the aircraft voltage regulator. Battery problems may often be prevented or eliminated by maintaining the voltage regulator setting at the precise voltage specified. Refer to inspection requirements in this chapter.

c. Nickel-cadmium batteries contain an electrolyte mixture of potassium hydroxide and distilled water. Chemically, this is just the opposite of an acid. Take every possible step to keep the nickel-cadmium battery as far away as possible from the lead-acid type of battery. Do not use the same tools and materials (screwdrivers, wrenches, syringes, gloves, apron, etc) for both types of batteries. Anything associated with the lead-acid battery, even the air, must never come in contact with the nickel-cadmium battery or its electrolyte. Even a trace of sulphuric acid fumes from a lead-acid battery may result in damage to the nickel-cadmium battery.

d. A low electrolyte level does not necessarily indicate that water must be added to the cells. The electrolyte level in the cells will vary, depending upon the battery's state of charge. Refer to 1-23 below before adjusting the electrolyte level.

e. During operation of the battery, some water is lost from the electrolyte as a result of normal gassing, venting or overcharging. This loss should be replaced with pure distilled water only; do not use potassium hydroxide solution. Refer to 1-23 below before adding distilled water.

f. The state of charge of a nickel-cadmium battery cannot be determined by either the specific gravity of

the electrolyte or the battery voltage. The specific gravity will remain the same whether the battery is charged or discharged, and the voltage will not change appreciably until the battery is almost completely discharged.

g. If sulphuric acid has inadvertently mixed with the electrolyte in the battery, the upper areas of the cells will appear greenish in color. In such cases, the battery must be replaced.

1-23. Servicing — Battery. Battery servicing consists of replenishing any electrolyte water that may have been lost through normal gassing, venting or overcharging. The lost water should be replaced with pure distilled water only. Never use potassium hydroxide solution.

CAUTION

The electrolyte level should be checked only after the battery has been fully charged and then allowed to rest (stand idle) for a period of 30 to 60 minutes. If the aircraft has operated continuously for a minimum of 1 hour or more the battery may be considered fully charged.

a. Set power selector switch at OFF.

b. Raise pilot compartment floor left door and remove battery cover (fig. 1-1).

c. Inspect and service battery in accordance with TM 11-6140-203-15-2.

NOTE

The battery does not have to be removed for servicing at the weekly or 25-hour special inspection interval.

d. Reinstall filler vent caps.

e. Reinstall battery cover, mounting screws and washers; close and latch access door.

1-24. CLEANING.

1-25. General — Cleaning. General cleaning of oil and dirt deposits from the aircraft and its components may be accomplished by using dry-cleaning solvent (C94), or a solution of detergent soap (C35) and water. (Refer to TM 55-1500-204-25/1 and TM 55-1500-333-24 for additional information.) Any exceptions that must be observed are specified in the following cleaning paragraphs.

1-26. Cleaning — Fuselage Interior and Upholstery.
a. Clean dirt or dust accumulations from floors and other metal surfaces with a vacuum cleaner or small hand brush.

b. Sponge soiled upholstery with a mild soap and lukewarm water solution. Avoid complete soaking of the upholstery. Wipe solution residue from upholstery with a cloth dampened by clean water.

c. Remove imbedded grease or dirt from upholstery by sponging or wiping with an upholstery cleaning solvent.

1-27. Cleaning — Airframe Exterior and Rotor Blades.

CAUTION

Use care to prevent scratching of the aluminum skin when cleaning main rotor blades. Never use volatile solvents or abrasive materials. Never apply bending loads to blades or blade tabs during the cleaning process.

a. Wash the aircraft exterior, including fiberglass components and rotor blades, when necessary by using a solution of clean water and mild soap.

NOTE

Avoid directing water toward the engine air intake area and the instrument static port in the aft fairing. Use covers, as necessary.

b. Clean those surfaces that are stained with fuel or oil by initial wiping with a soft cloth dampened by solvent (C94) followed by washing with clean water and mild soap.

c. Rinse washed areas with clean water and dry with a soft cloth.

1-28. Cleaning — Transparent Plastic.

CAUTION

Do not use naphtha on polycarbonate canopy. Use of this material will cause damage to the canopy.

a. Clean the outside surfaces of plastic panels by rinsing with clean water and rubbing lightly with palm of hand.

b. Use a detergent (C35) and water solution to remove oil spots and similar residue.

CAUTION

Never attempt to dry plastic panels with a dry cloth. To do so will cause any abrasive particles lying on the plastic to scratch or dull the surface. Wiping with a dry cloth will also build up an electrostatic charge that will attract dust particles from the air. Use care when cleaning around the metal

foil tape elements of the fm homing antenna.

c. After dirt is removed from surface of plastic, rinse with clean water and air-dry, or dry with a soft, damp chamois.

d. Clean the inside surfaces of plastic panels by dusting the surfaces with a soft, clean cloth, saturated with clean water. DO NOT USE DRY CLOTH, wipe carefully with a soft, damp cloth or sponge. Keep the cloth or sponge free from grit by rinsing frequently in clean water.

1-29. Cleaning — Fiberglass Components. Cleaning of fiberglass components is performed the same as for the aircraft exterior (para 1-26).

1-30. Cleaning — Battery and Electrolyte Spillage. *a.* Set the power selector at OFF.

WARNING

Electrolyte is a strong alkaline solution and is harmful to the skin and clothing. Wear protective clothing that is used exclusively for servicing nickel-cadmium batteries. Neutralize and flush electrolyte from the skin or hands as described below.

Where there is evidence of spewed or spilled battery electrolyte, flush off the surface immediately with water (cold if possible); then neutralize with citric acid solution mixed 14 parts water to one part citric acid (C23). Follow by flushing with chromium triox (C22); then thoroughly flush with clean water (cold if possible).

b. Unlatch and raise pilot compartment floor left side access door. Remove four screws and washers securing battery cover and remove cover.

c. Use a clean cloth dampened by clean water to remove any accumulation of dust, dirt, or white powder (potassium carbonate).

d. If battery is unusually dirty or shows evidence of caked crystals around the cells it should be removed from the aircraft for further cleaning with a nylon (or other non-metallic) brush and clean running water. For additional cleaning instructions refer to TM 11-6140-203-14-2.

e. Dry the top of the battery thoroughly with a clean cloth.

f. Reinstall battery cover, mounting screws and washers.

g. Lower and latch the floor access door.

1-31. CORROSION CONTROL.

1-32. General — Corrosion Control. The airframe is fabricated mainly of aluminum and some magnesium alloys, with selective use of stainless steel and titanium, and should be checked regularly for any signs of corrosion, especially at points of dissimilar and overlapping metal contact. Corrosion of dissimilar metals is the result of several conditions; lack of sufficient insulation in the areas of metal contact, tears or punctures in the metal itself, and areas where the protective finishes have been scuffed, scratched, chipped, or worn away. Inspections and maintenance precautions that should be performed to inhibit the start of corrosive action are outlined in paragraph 1-33. Common types of corrosion that may be encountered are described in paragraphs 1-39 through 1-41. Restoration procedures for marred but uncorroded surfaces, as well as surfaces on which corrosion is found, are given in paragraph 1-42.

Refer to TM 55-1500-204-25/1 for additional information.

1-33. STANDARD PRACTICES FOR CORROSION PREVENTION.

1-34. Corrosion Inspection — Interior Metal Surfaces. *a.* Inspect primer-painted surfaces for scratches and other damage.

b. Inspect finish-painted (color coated) surfaces for condition of finish.

c. Inspect areas of metal overlap (faying surfaces) for evidence of corrosion.

d. Inspect the attachment area of bolts, screws, and other fasteners for corrosion.

1-35. Corrosion Inspection — Exterior Metal Surfaces. *a.* Inspect finish for scratches, cracks, peeling, fading, or other damage, particularly around bolts, screws, and other fasteners.

b. Inspect normally sealed seams and joints for loose or missing sealing compound.

c. Inspect exposed skin edges for condition of corrosion-protective finish or sealing compound and for evidence of corrosion.

d. Inspect areas of metal overlap for evidence of corrosion.

1-36. Insulation — Magnesium Alloys Against Corrosion. To prevent galvanic corrosion between magnesium and any dissimilar metals:

a. Coat contacting surfaces with a layer of sealing compound (C89), in addition to the primer (C78) or (C79).

CAUTION

Do not use steel washers during the following step:

b. Apply primer (C79) on the attaching hardware before installation.

1-37. Application — Sealing Compound. Use sealing compound (C89) to replace loose or missing sealant on exterior surfaces. Sealant is used to fill seams and joints that might trap water. Apply sealant as follows:

a. Check that seam or joint is clean and free of foreign matter and moisture.

b. Apply sealant with a putty knife or similar tool.

c. Force the sealant well down into the seam to eliminate any air pockets.

d. Fillet the sealant to give the joint or seam a smooth appearance.

1-38. Removal — Salt Deposits. To inhibit corrosion, aircraft operating over salt water and those that come in contact with salt water or spray should be washed with fresh water as frequently as possible.

1-39. Corrosion on Magnesium Alloys.

CAUTION

Bare magnesium alloys, when exposed to salt-laden air, will corrode very rapidly. Adequate protective finishes must at all times be maintained on magnesium.

Corrosion will not normally be present on painted, treated or protected surfaces. Corrosion will attack magnesium when nicks or scratches through the surfact protection expose the metal to moisture or air. Corrosion is present if the following conditions are in evidence:

a. Whitish powdered deposits.

b. Zinc chromate discoloration over an area.

c. Blistering or cracking of the finish coating.

1-40. Corrosion of Aluminum Alloys. Corrosion will not normally be present on aluminum surfaces that have a chemical protective finish; however, because moisture can permeate paint that is nicked or scratched, corrosion might attack the metal even though it is painted. In such cases, the affected areas will generally be characterized by:

a. A scaly or blistered appearance of the finish surface.

b. A dulling and pitting of the area.

c. Whitish powdered deposits.

1-41. Corrosion on Alloy Steels. Corrosion (rust) will not normally be present on steel surfaces that have been painted; however, surfaces may corrode where moisture has permeated the paint. Such corrosion will be characterized by:

a. A reddish or brownish blistered appearance in the corroded area.

b. Blistering of the painted surfaces.

1-42. CORROSION TREATMENT.

1-43. Temporary Anti-corrosion Measures. The temporary anti-corrosion measures outlined here are to be used only in cases where the proper materials or equipment are not available.

WARNING

These temporary anti-corrosion measures apply to the airframe only. If a part of the structure is corroded too badly to withstand normal loads before the aircraft can reach a repair station, metal patches will have to be installed before the aircraft is flown.

a. Examine part of area in question for extent of corrosion.

b. Remove loose paint and powdery products of corrosion by scraping with a sharp phenolic scraper, or brushing the area with a heavy fiber brush.

c. Wash off the affected areas with mild soap and clean fresh water; rinse thoroughly.

d. If protective paint coatings are not available, liberally apply corrosion-preventive compound (C30) or any available grease to affected areas.

1-44. Exterior Surface Corrosion Touchup Treatment — Magnesium Alloy. *a.* Wash affected area with a solution of mild soap and clean fresh water. Rinse area with clean water and wipe dry by using a clean soft lint-free cloth.

b. Use thinner (C108) on damaged area to remove any grease and old paint.

c. Apply chrome pickle solution (C21) by swabbing exposed area for 3 to 10 minutes.

d. Using a clean cloth soaked in clean fresh water, thoroughly rinse area where solution was applied. Allow area to thoroughly dry.

e. Apply paint finish touchup (para 1-47).

1-45. Exterior Surface Corrosion Touchup Treatment — Aluminum Alloy.

NOTE

If there is any question of whether or not the protective coating is removed, it should always be assumed that bare metal is exposed.

a. Wash affected area with a solution of mild soap and fresh water. Rinse area with clean water and wipe dry with a clean soft lint-free cloth.

b. Using a swab, liberally apply chemical film solution (C20).

c. Allow solution to remain on surface for 1 to 3 minutes, or until surface becomes amber to brown in color.

NOTE

Avoid letting the chemical mixture dry on the surface. If it has dried, rewet the surface with the solution.

d. Rinse treated surface thoroughly with clean water. After rinsing, wipe off excess moisture with a clean lint-free cloth. If dry compressed air is available, blow any moisture from joints or crevices and allow to dry completely at room temperature for approximately 1 hour.

e. Apply paint finish touchup (para 1-47).

1-46. Exterior Surface Corrosion Touchup Treatment — Steel Alloy. *a.* Remove loose paint and corrosion products by scraping area with a sharp phenolic scraper, brushing with a heavy fiber brush, and light sanding with grade 400 or finer abrasive paper (C3) and (C4).

b. Wash off the area with mild soap and clean fresh water; rinse thoroughly.

c. Treat surface with surface cleaner (C74) or equivalent.

d. Allow solution to remain on surface for approximately 5 minutes. Keep surfaces wet.

e. Rinse thoroughly with clean water. Dry with a clean lint-free cloth and then allow to air-dry completely.

f. Apply paint finish touchup (para 1-47).

1-47. PAINT FINISH.

1-48. General — Paint Finish. All surfaces of the aircraft are prime-painted with one coating of yellow epoxy primer, and finish-painted with either one or two color coats of acrylic lacquer. Nonvisible interior surfaces have only one finish coat of acrylic lacquer that is green in color. Personnel areas, visible interior areas, and the exterior have two color coats. Refer to TB 746-93-2 for touchup and complete painting procedures.

CAUTION

If paint remover is used in the vicinity of drive shaft couplings, be sure the couplings are completely masked and covered. If paint remover contacts the coupling diaphragms, rust spots will develop and coupling replacement will be required.

1-49. LIST OF CONSUMABLE MAINTENANCE SUPPLIES AND MATERIALS.

1-50. General — List of Consumable Maintenance Supplies and Materials. Consumable maintenance supplies and materials are listed in table 1-1 in alphabetical order. Each consumable also has an item number assigned for ease of location and reference. When an item number is unknown, you may locate any consumables used within this manual through its alphabetical arrangement. When an item number is referenced in the manual, you may locate the item through its C designator and item number. C designators are used only with consumable maintenance supplies and material. Consumable maintenance supplies and materials tables are found only in this chapter, therefore the table number will not be referenced in the text.

1-51. NATIONAL SUPPLY CLASS (NSC) REFERENCE.

1-52. General — NSC Reference. National supply class references are listed in table 1-2.

1-53. SPECIAL TOOLS AND TEST EQUIPMENT.

1-54. General — Special Tools and Test Equipment. Special tools and test equipment are listed in table 1-3 in alphanumeric order. Each tool or piece of test equipment has an item number assigned for ease of location and reference. When an item number is unknown, you may locate special tools and test equipment through alphanumeric arrangement within the table. When an

Table 1-1. Consumable Maintenance Supplies and Materials.

The supplies and materials listed in this table are required for maintenance support of this equipment and are authorized to be requisitioned by SB700-50.

Item No.	Description	Ref. No. & (FSCM)	NSN
1	ABRASIVE PAPER, SILICON CARBIDE, grade 180	P-P-101	5350-00-721-8117
2	ABRASIVE PAPER, SILICON CARBIDE, grade 280	P-P-101	5350-00-224-7205
3	ABRASIVE PAPER, SILICON CARBIDE, grade 400	P-P-101	5350-00-224-7201
4	ABRASIVE PAPER, SILICON CARBIDE, grade 600	P-P-101	5350-00-224-7215
5	ADHESIVE	RTV 731	8040-00-933-9563
5 A	ADHESIVE	MMM-A-132 TY I, CL I	8040-00-923-2198
6	ADHESIVE	MMM-A-138	8040-00-145-0303
7	ADHESIVE	8089ABX	8040-00-828-4936
8	ADHESIVE	MIL-A-8576	8040-00-266-0815
9	ADHESIVE	MMM-A-134 Ty 2	8040-00-691-1322
9 A	ADHESIVE	EC2216 B/A	8040-00-145-0019
10	ADHESIVE, BONDING, VULCANIZED (Synthetic rubber to seal)	MIL-A-1154C	8040-00-165-8614
11	ADHESIVE, POLYURETHANE	Stabond U136 AC AAAA	8040-00-224-4673
12	ADHESIVE, SILICONE	MIL-A-46106	8040-00-924-8827
13	ADHESIVE, SILICONE RUBBER	Silastic 140 RTV	8040-00-701-9546
14	ANTI-SEIZE COMPOUND	MIL-T-5544B	8030-00-087-8630
15	BARRIER MATERIAL, grease proof	MIL-B-121 Ty 1, Gr A, CL 2	8135-00-753-4661
16	BARRIER MATERIAL, WATER VAPORPROOF, flexible	MIL-B-131 CL 1	8135-00-226-0839
17	BRAZING ALLOY, SILVER BASE	QQ-B-654 No. 7	
18	CELLULOSE TAPE	MIL-T-18833	1375-00-609-2421
19	CEMENT, RUBBER	MIL-A-5092 Ty 3	8040-00-152-0063
20	CHEMICAL FILM	MIL-C-5541 Ty II, Gr B, Cl 1	8030-00-811-3723
21	CHROME PICKLE SOLUTION	MIL-M-3171 Ty 1 FED	8030-00-050-9043
22	CHROMIUM TRIOX	OC 303 Ty 2	6810-00-264-6517
23	CITRIC ACID	OA 76	6810-00-275-1215

Table 1-1. Consumable Maintenance Supplies and Materials. (cont)

Item No.	Description	Ref. No. & (FSCM)	NSN
24	CLOTH, ABRASIVE, ALUMINUM OXIDE (gr 200)	P-C-451	5350-00-246-0330
24A	CLOTH, ABRASIVE, ALUMINUM OXIDE (gr 400)	PC 451	5350-00-865-5700
25	CLOTH, ABRASIVE, crocus gr	P-C-458	5350-00-221-0872
26	Deleted		
27	COMPOUND, ALKALINE WATERBASE	MIL-C-25769	6850-00-935-0995
28	COMPOUND, ANTI-SEIZE, LEAD PLATE NO. 250	MIL-A-907	8030-00-597-5367
29	CORROSION PREVENTIVE COMPOUND	MIL-C-11796 Cl 3	8030-00-231-2353
30	CORROSION PREVENTIVE COMPOUND	MIL-C-16173C Gr 1	8030-00-231-2345
31	CORROSION PREVENTIVE COMPOUND	MIL-C-16173C Gr 2	8030-00-118-0666
32	CORROSION PREVENTIVE CONCENTRATE	Brayco 599	6850-00-142-9582
32A	CORROSION REMOVING COMPOUND	MIL-C-10518 Type II	6850-00-174-9672
33	CUSHIONING MATERIAL, PACKAGING, cellulose wadding, water resistant, low absorbency	PPP-C-843, Ty II, C1 B	8135-00-664-6958
34	CORD, NYLON, 12, NATURAL	MIL-C-5040 Ty 1	4020-00-240-2154
35	DETERGENT, GENERAL PURPOSE	MIL-D-16791 Ty 1	7930-00-985-6911
36	DETERGENT, SURFACE CLEANER	MIL-D-26549	6850-00-597-1528
37	DIELECTRIC COMPOUND, SILICONE BASE, HIGH TEMPERATURE	MIL-S-8660B	6850-00-880-7616
38	DICHLOROMETHANE, technical	MIL-D-6998 Gr B	6810-00-244-0290
39	FIBERGLASS REPAIR KIT	PN S1607-7021	1560-00-856-9222
39A	FIBERGLASS	MIL-C-9084	8305-00-530-0109
40	DELETED		
41	FILLER, RESIN	MMM-A-132 Ty 1, Cl 1 Ren Plastics RP1257-3 (02684)	8030-00-891-3113
42	FLUX, BRAZING PASTE	O-F-499 Ty B	3439-00-640-3713
43	FUEL CELL REPAIR KIT	RK10-34	2910-00-437-0588
44	GASKET MATERIAL (adhesive one side only)	3M Scotchfoam 4304 (76381)	9330-00-242-6229
45	GREASE, AIRCRAFT HIGH TEMPERATURE	MIL-G-25537	9150-00-478-0055

Table 1-1. Consumable Maintenance Supplies and Materials (cont)

Item No.	Description	Ref. No. & (FSCM)	NSN
46	GREASE, AIRCRAFT AND INSTRUMENT	MIL-G-23827	9150-00-985-7245
47	GREASE, BALL AND ROLLER BEARING	MIL-G-18709A (Navy Amend 3)	9150-00-526-4205
47A	GREASE, MOBIL 28	MIL-G-24508	9150-00-149-1593
48	HYDRAULIC FLUID, PETROLEUM BASE	MIL-H-5606	9150-00-252-6883
49	INSULATION SLEEVING, ELECTRICAL, (size as applicable)	MIL-I-631, Ty F, Form U, Gr A, Cl 1, Cat 1	5970-00-284-8640
50	INSULATION SLEEVING, ELECTRICAL, FIBERGLASS (size as applicable)	MIL-L-3190 C1 HA-1	5920-00-250-3026
51	JET FUEL, GRADE JP-4	MIL-T-5624	9130-00-256-8617
52	LACING CORD, HIGH TEMPERATURE	Warren Wire Co T-3333	4020-00-807-4491
53	LACQUER	TT-L-32	8010-00-166-3164
54	LACQUER, GULL GRAY, FED STD 595, COLOR NO. 36231	MIL-L-19538	8010-00-526-3296
55	LACQUER, WHITE, FED STD 595, COLOR NO. 17875	MIL-L-81352	8010-00-935-6608
56	LOCKWIRE, CRES, 0.020 IN. DIA	QQ-W-423, Comp 302, Cond A	9505-00-596-5101
57	LOCKWIRE, CRES, 0.032 IN. DIA	QQ-W-423, Comp 302, Cond A	9505-00-293-4208
58	LOCKWIRE, CRES, 0.042 IN. DIA	Comp 304, Cond A	9505-00-804-3814
59	DELETED		
60	LUBRICANT	VV-G-632 Gr 2	9150-00-753-4649
61	LUBRICANT, CORROSION INHIBITING	VV-L-800	9150-00-231-6689
62	LUBRICANT, GRAPHITE	SS-G-659A	9620-00-233-6712
63	LUBRICANT, GREASE	MIL-G-81322	9150-00-944-8953
64	LUBRICANT, SOLID FILM	Electrofilm, Inc., Lubri-Bond A (85932)	9150-00-754-0064
65	LUBRICANT, SOLID FILM	MIL-L-46010	9150-00-948-6912
66	LUBRICANT, SOLID FILM	MIL-L-8937	9150-00-985-7255
67	LUBRICATING OIL, AIRCRAFT TURBINE ENGINE, SYNTHETIC	MIL-L-23699	9150-00-985-7099
68	LUBRICATING OIL, jet engine	MIL-L-6081 GR 1010	9150-00-273-2388
68A	METAL CONTAINER	MIL-C-10578 Ty 1	6850-00-174-9670
69	METHYL ETHYL KETONE, TECHNICAL		

Table 1-1. Consumable Maintenance Supplies and Materials (cont)

Item No.	Description	Ref. No. & (FSCM)	NSN
70	NAPHTHA, ALIPHATIC	TT-N-95	6810-00-238-8119
71	NITRIC ACID, TECHNICAL	O-N-350	6810-00-222-9655
72	PAINT REMOVER	TT-R-248	8010-00-943-7126
73	PETROLATUM, TECHNICAL	VV-P-236	9150-00-250-0926
74	DELETED		
75	POLISHING KIT, PLEXIGLASS	P-P-560	7930-00-634-5340
76	PRESERVATIVE OIL, HYDRAULIC	MIL-H-6083	9150-00-159-4472
77	PRIMER	EXB 576-6	8040-00-104-5263
78	PRIMER, CATALYZED EPOXY (YELLOW)	MIL-P-23377B	8010-00-082-2450
79	PRIMER COATING	MIL-P-8585	8010-00-297-0593
80	PRIMER, EPOXY (YELLOW)	MIL-P-23377	8010-00-082-2450
81	PRIMER, SILICONE	MIL-S-8660	6850-00-924-2827
81A	PRIMER, ZINC CHROMATE	MIL-P-8585	8030-00-297-0593
82	PROPANOL 2, TECHNICAL (ISOPROPYL ALCOHOL)	TT-I-735 Gr B	6810-00-855-6160
83	PUTTY, ZINC CHROMATE	MIL-P-8116	8030-00-145-0300
83A	RELEASE AGENT	TC7-527 EPD Industries, Long Beach, CA	
84	DELETED		
85	ROD, WELDING	Armco Steel 21-6-9	3439-00-134-9209
86	ROD, WELDING	QQ-R-566 CFL-S-RA12	3439-00-268-9654
87	RUST INHIBITOR, SPRAY	WD 40 Rocket Chem Corp	8030-00-838-7789
87A	SEALANT	MIL-S-8802	8030-00-753-5006
88	SEALANT, SILICONE RUBBER	Dow Corning RTV 731	8040-00-842-6380
89	SEALING COMPOUND	MIL-S-8802	8030-00-753-4596
90	SEALING, LOCKING AND RETAINING COMPOUNDS; single component	MIL-S-22473 Gr A	8030-00-081-2338
91	SEALING, LOCKING AND RETAINING COMPOUNDS; SINGLE COMPONENT (surface primer)	MIL-S-22473 Gr T	8030-00-082-2508
92	SOAP, TOILET	P-S-620, Type II	8520-00-531-6484

Table 1-1. Consumable Maintenance Supplies and Materials (cont)

Item No.	Description	Ref. No. & (FSCM)	NSN
93	SOLDER, TIN ALLOY	QQ-S SN60WRP2	3439-00-224-3567
94	SOLVENT, DRY CLEANING	P-D-680 Ty 1	6850-00-264-9038
95	SOLVENT, TOLUENE	TT-T-548	6810-00-281-2002
96	SOLVENT, TRICHLOROETHYLENE	O-T-620	6810-00-664-0387
97	SOLVENT, XYLENE	TT-X-916	6810-00-598-6600
98	TAPE	3M 850	7510-00-194-6643
99	TAPE, ALUMINUM FOIL	3M 425	7510-00-864-8803
100	TAPE, PLASTIC ELECTRICAL	MIL-I-24391	5970-00-419-4291
101	TAPE, PRESSURE SENSITIVE	3M 471	7510-00-142-9840
102	TAPE, PRESSURE SENSITIVE	PPP-T-60 Cl 4	8130-00-181-7610
103	TAPE, PRESSURE SENSITIVE	PPP-T-66 Ty 1, Cl B	7510-00-145-0002
104	TAPE, PRESSURE SENSITIVE ADHESIVE (MASKING)	UU-T-106A	7510-00-266-6712
105	TAPE, PRESSURE SENSITIVE ADHESIVE, WATER RESISTANT	PP-T-76	7510-00-297-6655
106	TAPE, PRESSURE SENSITIVE	455-3W	9330-00-664-4893

Table 1-1. Consumable Maintenance Supplies and Materials. (cont)

Item No.	Description	Ref. No. & (FSCM)	NSN
107	TAPE, PRESSURE SENSITIVE, WATERPROOF, FOR PACKING	PPP-T-60 Ty I, Cl 1	7510-00-266-5006
108	THINNER, ACRYLIC-NITROCELLULOSE LACQUER	MIL-T-19544	8010-00-527-2897
109	THINNER, PAINT, MINERAL SPIRITS	TT-T-291A	8010-00-242-2089
110	THREAD COMPOUND, ANTI-SEIZE AND SEALING	MIL-T-5542	8030-00-530-5234
111	TRICHLOROETHYLENE, TECHNICAL	O-T-634 Ty 2	6810-00-184-4800
112	TWINE, NYLON	MIL-T-713 Ty P, Cl 2	4020-00-202-1924
113	VARNISH	MIL-V-173 Comp I	5970-00-548-9520
114	WEIGHT, LEAD PACKING (NO. 66C38599)	MIL-W-27888	1670-00-375-9134

item is referenced in the manual, you may locate the item through its T designator and item number. T designators are used only with special tools and test equipment. The special tools and test equipment table is found only within this chapter; therefore the table number will not be referenced within the text. A complete listing of all special tools and test equipment authorized for use to perform maintenance on OH-6A aircraft/accessories are contained in the aircraft parts manual.

1-55. SUPPORT EQUIPMENT.

1-56. General — Support Equipment. Support equipment is listed in table 1-4 in alphanumeric order. Each item of support equipment has an item number assigned for ease of location and reference. When an item number is unknown, you may locate support equipment through alphanumeric arrangement within the table. When an item is referenced in the manual, you may locate the item through its S designator and item number. S designators are used only with support equipment. The support equipment table is found only within this chapter; therefore the table number will not be referenced within the text.

Table 1-2. National Supply Class Reference.

Nomenclature	National Supply Class
Abrasives, grain, cloth, paper	5350
Acid	6810
Adhesives	8040
Aluminum or steel wool	5350
Anti-icing fluid	6850
Antiseize compounds	8030
Bags	8105
Beeswax	9610
Carbon removing compound	6850
Castor oil	9150
Cement	8040
Chemicals	6810
Cleaning cloths	7920
Cleaning Compounds	6850
Cleaning compounds	7930
Core material	5680
Corrosion preventives	6850
Corrosion removing compounds	6850
Deodorant, general purpose	6840
Desiccant	6850
Detergent	7930
Distilled water	6810
Dopes	8010
Dry cleaning solvent	6850
Dyes	6820
Fiberglass repair kit	1560
Fuel oils	9140
Gases, compressed and liquefied	6830
Graphite	9620
Greases	9150
Hydraulic fluid	9150
Inhibitors	6850
Ink	7510
Inspection penetrant kit	6850

Nomenclature	National Supply Class
Inspection penetrant remover	6850
Insulating compound, electrical	5970
Lacquer and enamel	8010
Layout dye	6850
Lubricating oil	9150
Paint	8010
Paint remover and thinner	8010
Paraffin	9160
Petrolatum	9150
Plastic molding material and compounds	9330
Plexiglas repair kit	1560
Polishing cloths	7920
Polishing compounds	7930
Preservative compounds	8030
Primer, coating and paint	8010
Rags, wiping	7920
Sacks	8510
Sealing compounds	8030
Soap, toilet	8520
Solder	3439
Steel plate, sheet and strip	9515
Tape, electrical	5970
Tape, identification	7690
Tape, luminous and reflective	9390
Tape, paper, acetate fiber and gummed	8135
Tape, pressure-sensitive adhesive	7510
Tape, rubber, adhesive and cloth-coated	9320
Towel, paper and machinery wiping	7920
Varnish	8010
Varnish, electrical insulating	5970
Walkway compounds, nonslip	5610
Water repellant kit	6850
Wax, microcrystalline	9160

Table 1-3. Special Tools and Test Equipment.

ITEM No.	Part No.	Nomenclature	Usability Code Calibration	Figure Reference
1	BH112JA36	Cal Analyzer	T/AD	
2	MIL-S-6180B (NSN 5130-00-712-4855)	Tool Kit	RP	
3	PN-901	Voltmeter (0.5% full scale accuracy)	T/AD	
4	TS-352 B/U	Multimeter	T/AD	
5	TS-443/u	Voltmeter (0.25% full scale accuracy)	T/AD	
6	VS5236	Wrench	R/IN	
7	369A1400-30202	Damper Assembly Holding Fixture	T/AD	5-10
8	369A2000-80902	Build-up Aligning Tool	I	4-5
9	369A5002	Lifting Bracket	R/IN	1-3
10	369A6001-50506	Drill jig	RP	
11	369A8009	Engine Hoist	R/IN	
12	369A8100-80902-206	Spanner Wrench	R/IN	
13	369A8100-80902-9	Spanner Wrench	R/IN	
14	369A9825	Pivot Bearing Adapter Wrench	R/IN	
15	369A9904	Adapter Assembly	R/IN	
16	369A9920	Bracket Assembly-Accelerometer MTG, Tail Rotor Balancing	I/AD	5-25
17	369A9925	Light, Strobe Strobe Light Instl &	I/AD	5-24
18	369A9926	Assy, High Intensity-Blade Tracking	I/AD	5-24
19	369A9927	Collective Rigging Fixture	I/AD	11-2
20	369A9928-5	Cyclic Lateral Rigging Fixture	I/AD	11-2
21	369A9929-5	Cyclic Longitudinal Rigging Fixture	I/AD	11-2
22	369A9930	Master Rigging Plate	I/AD	11-2
23	369A9931	Tail Rotor Swashplate Rigging Tool	I/AD	11-25
24	369A9932	Hub Puller, Main Rotor	R/IN	5-4
25	369A9933	Main Rotor Hub Driver	IN	
26	369A9934	Main Rotor (Face Socket) In Wrench	R/IN	
27	369A9936	Collective Bungee Installation Tool	R/IN	11-9, 11-12
28	369A9937	Torque Wrench Adapter Torque	R/IN	5-21
29	369A9949	Wrench Adapter	R/IN	
30	369A9957	Torque Wrench Adapter	R/IN	6-3
31	369A9958	Tool Assembly, Tab Binding	AD	5-2,5-14
32	369A9968	Pitot Tube Angularity Template	I/AD	8-5
33	369A9979	Protractor Assy, Tail Rotor Balancing	I/AD	
34	369A9985	Bungee Compression Tool (Rod and Channel)	R/IN	
35	369A9993-601	Main Rotor (Face Socket) Wrench	R/IN	

Table 1-3. Special Tools and Test Equipment. (cont)

ITEM No.	Part No.	Nomenclature	Usability Code Calibration	Figure Reference
36	369A9998	Bearing Installation and Removal Tool	R/IN	2-26
37	369A9999	Tail Rotor Balance Kit, Assy	I/AD	5-24
38	6795579	Engine Turnover Stand	R/IN	4-2
39	6796963	Engine Assembly Lift	R/IN	
40	B4591	Kit, Vibrex Balancing	IN/AD	5-14A
41	10ARAX6	Wrench Torque	R/IN	

USABILITY CODES

R – Removal
D – Disassembly
I – Inspection
RP – Repair/Replace
T – Testing
A – Assembly
IN – Installation
AD – Adjustment
S/P – Storage/Preservation

Table 1-4. Support Equipment.

Item No.	Part No.	Nomenclature	Figure Reference
1	AN 8015-2	Mooring kit	1-7
2	369ASK1970	Exhaust stack cover	1-7
3	369A2010	Jack Fitting	1-3, 1-7
4	369A4023	Rotor hub and engine inlet cover	1-7
5	369A4025	Blade cover	1-5
6	369A4026	Pitot tube cover, nonheated pitot tube	1-7
7	369A4027	Blade sock Main rotor maintenance plat-	1-5
8	369A9810	Main rotor maintenance platform	1-4
9	369A9905	Ground handling wheels	1-6
10	369A9906	Jack handle	1-6
11	369A9918	Blade rack	1-5
12	369H4009	Pitot tube cover, heated pitot tube	1-7
13	A338	Fuel cell leak check manometer	

SECTION II LUBRICATION

1-57. LUBRICATION.

1-58. General — Lubrication. All lubrication of the aircraft is accomplished through servicing (section I) or during maintenance of disassembled components.

CAUTION

Use extreme care when applying any type of lubricant (grease, oil, dry-film, etc) in the vicinity of teflon bearings. Most lubricants will form a dirt retaining film or have otherwise detrimental effects that can cause rapid deterioration of the bearing surfaces.

SECTION III HANDLING, JACKING, MOORING, HOISTING AND SLING LOADING.

1-59. GROUND HANDLING.

1-60. General — Ground Handling. Ground handling procedures for the aircraft include hoisting, jacking, installing round handling wheels, towing, parking, mooring and leveling procedures. The following information will aid ground handling personnel.

Table 1-5. Premaintenance Requirements for General Handling of Aircraft.

Condition	Requirements
Special Tools	(T15)
Support Equipment	(S1) (S2) (S3) (S4) (S5) (S6) (S7) (S8) (S9) (S10) (S11) (S12)
Minimum Personnel Required	Two
Consumable Materials	(C33) (C45) (C105)

1-61. Aircraft Dimensions. The principal dimensions of the aircraft are shown in figure 1-2.

1-62. Leveling. Leveling (fig. 1-3) is accomplished by causing a plumb bob to intersect register marks enscribed on the target plate located on the cargo compartment floor.

a. Suspend plumb bob from a line attached to the support clip located on the upper right edge of the controls tunnel (fig. 1-3).

b. Raise the aircraft from the ground (para 1-69).

c. Adjust side jacks to obtain lateral level. Adjust tailboom jack to obtain longitudinal level.

d. Recheck lateral and longitudinal levels until the plumb bob exactly intersects the register marks enscribed on the target plate.

1-63. Main Rotor Maintenance Platform. A main rotor maintenance platform (S58) can be used on aircraft equipped with a fitting installed in the aft edge of the cargo door frame (fig. 1-4). The platform support arm mounts in the aircraft jack fitting. When not in use, the platform and support arm may be stowed and secured with a lockpin.

1-64. Folding Main Rotor Blades. *a.* Locate aircraft slightly more than rotor span from other aircraft or vehicles, parking the aircraft on the most level ground available so that the load is balanced as much as possible.

CAUTION

During blade folding, use adequate covering over engine air inlet fairing opening to prevent entry of foreign objects into air intake. Install exhaust covers (S2) on aircraft with upward exhausts.

b. Disconnect wire antenna (as applicable) at boom attach fitting and coil while blades are folded.

c. Attach blade rack (S11) to the tailboom (fig. 1-5).

d. Position main rotor so that one blade is centered fore and aft over the tailboom.

e. Wrap a 4-inch strip of cushioning material (C33) around each rotor blade at the point where vibration absorbers contact blade when absorber is moved fully upward. Lift vibration absorbers upward to contact the cardboard, wrap tape (C105) or equivalent, around absorbers and the cardboard wrapped blade to hold absorbers.

f. Install four blade covers (S5).

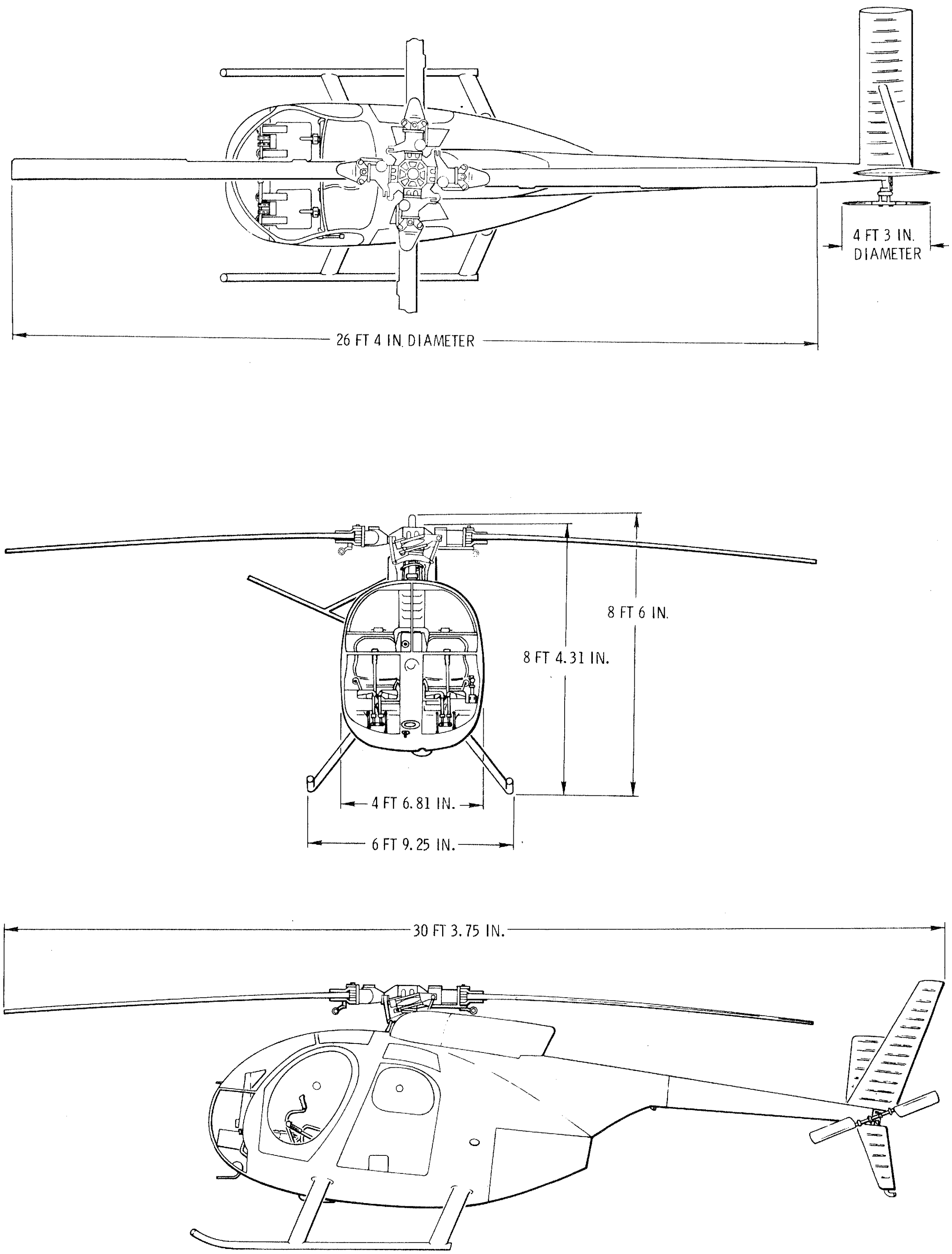


Figure 1-2. Principal Dimensions.

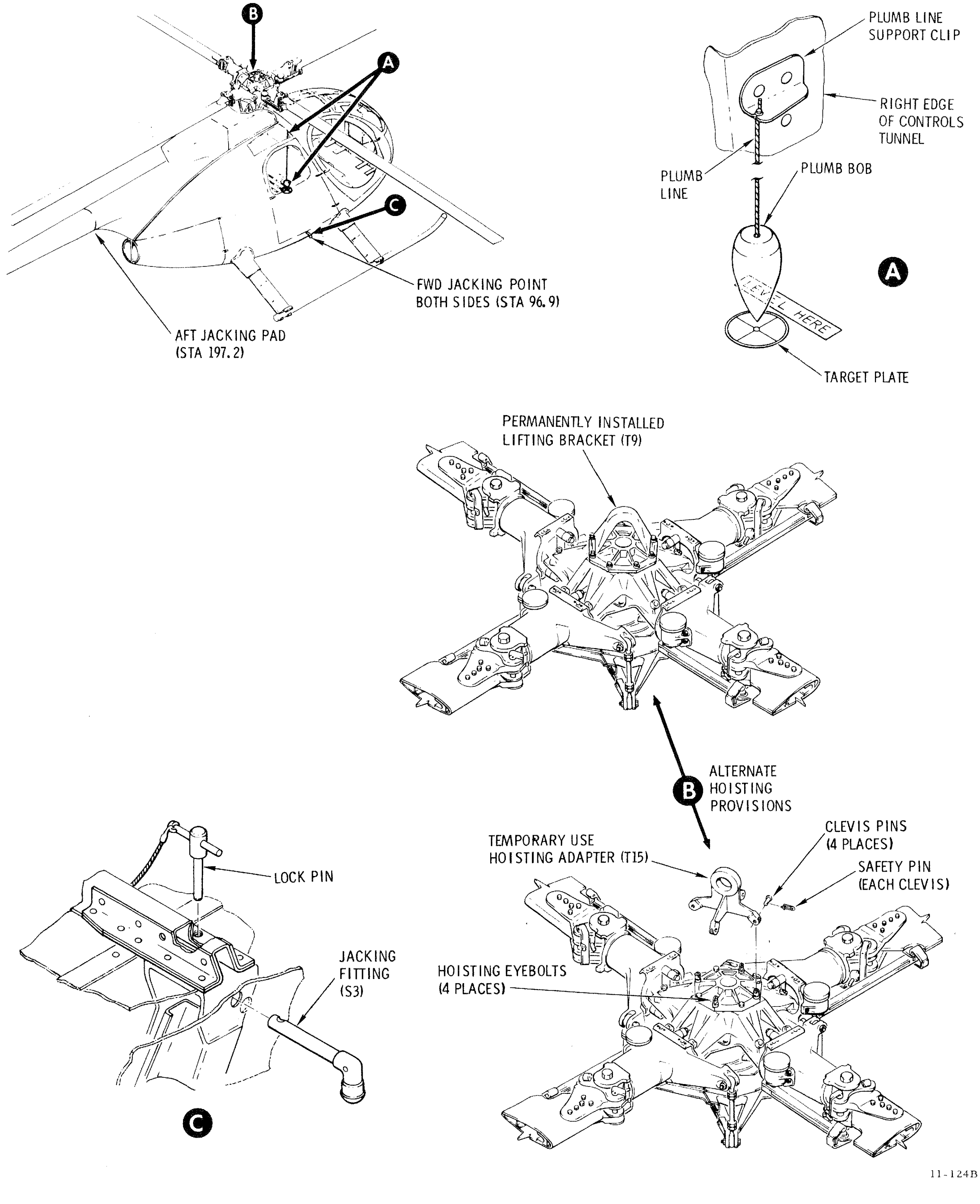


Figure 1-3. Hoisting, Jacking and Leveling.

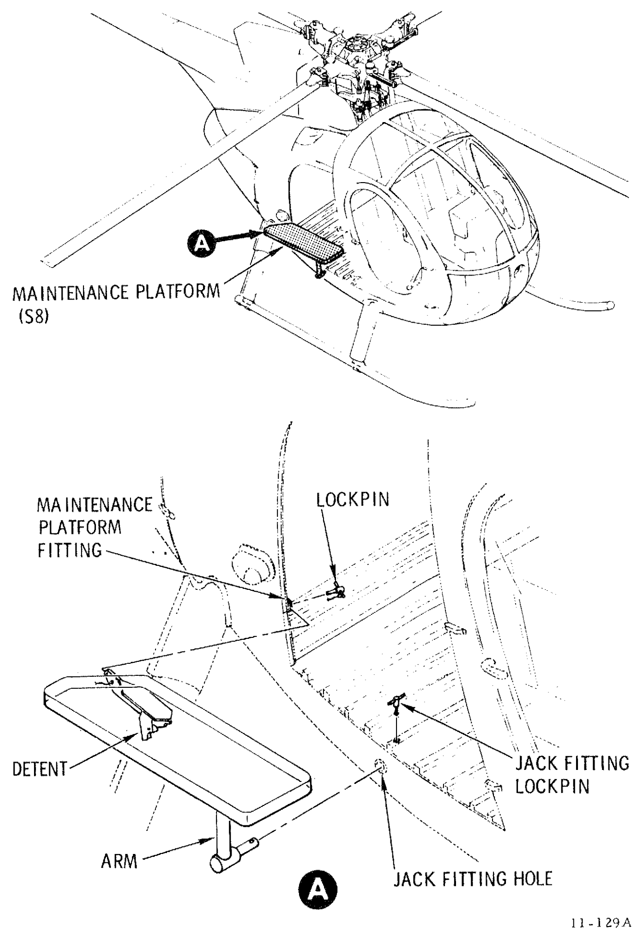


Figure 1-4. Main Rotor Maintenance Platform.

g. Lock control sticks (by applying friction) in the following positions:

(1) Collective pitch — mid position.

(2) Longitudinal cyclic — full aft.

(3) Lateral cyclic — two-thirds of total stick travel from full left toward full right.

CAUTION

During blade folding, continually monitor stick positions to make sure no change occurs. Serious blade damage can result from improper stick placement.

h. Secure trailing blade to blade rack central holddown position.

CAUTION

When folding a main rotor blade, use care to prevent scratching of the blade by the blade attaching pins securing other blades. Avoid abrading blades against one another or against any other surfaces during the folding procedure. During the folding operation, have an assistant positioned at the outboard end of the blade.

i. Fold left blade as follows:

(1) Remove attach pin from damper arm and move the arm away from the blade; replace pin in damper arm.

(2) Remove blade attach pin from trailing edge of blade at lead-lag link.

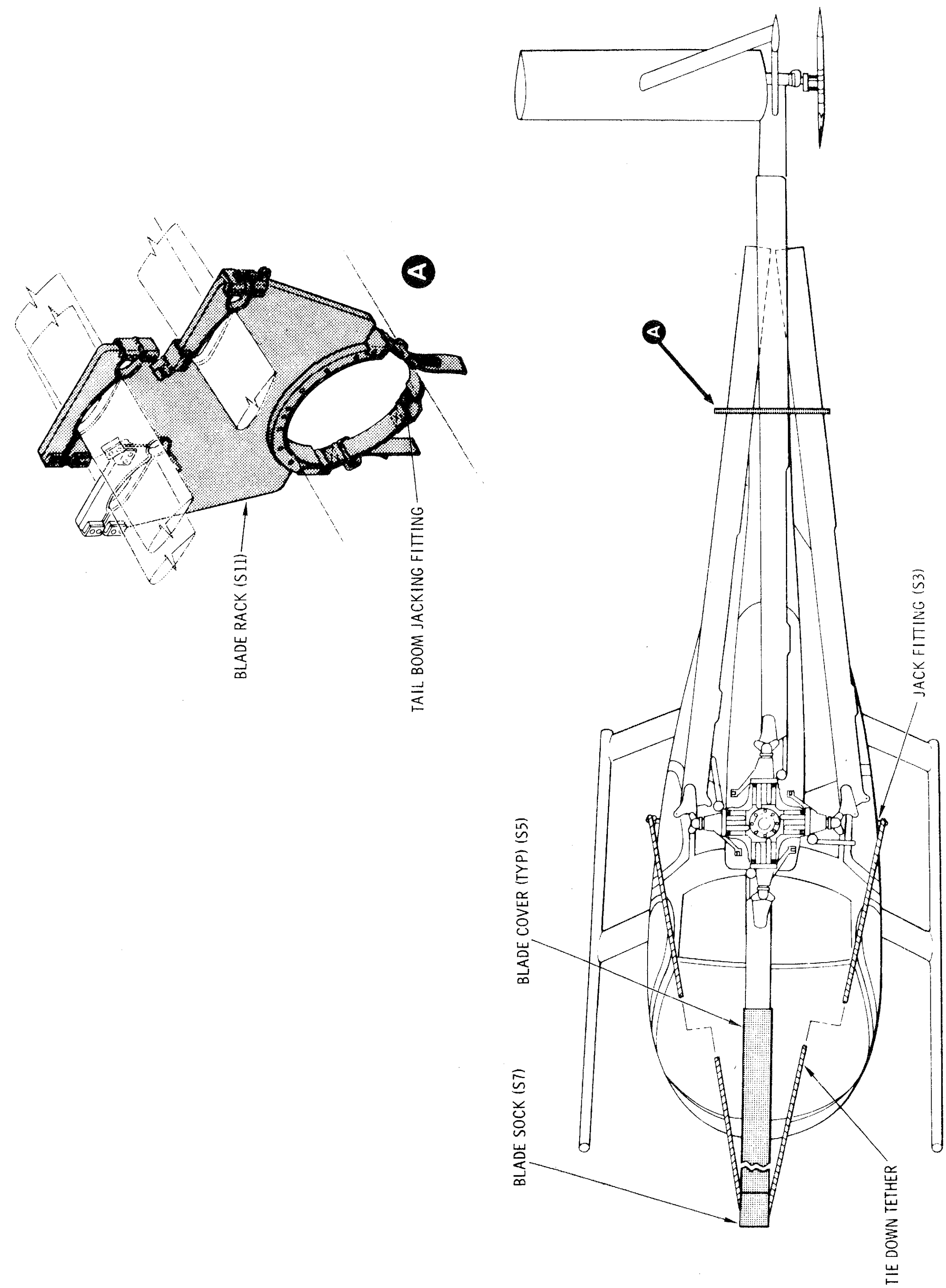


Figure 1-5. Folding and Stowing of Main Rotor Blades.

CAUTION

When performing the next step, unlock but do not remove the blade attach pin that secures the leading edge of the blade to the main rotor hub.

(3) Unlock the trailing edge blade, attach pin that secures the blade to the lead-lag link and position the cam handle vertically above the pin.

(4) Have the assistant rotate the blade to the proper rack position (toward tailboom); secure blade to blade rack. Reinstall blade attach pin to its original position in the lead-lag link on the main rotor hub.

j. Install padding between the damper arms and blades to prevent marring of the blades. Tape or tie padding in place.

k. Secure the forward blade with a blade sock (S7) as shown in figure 1-5.

NOTE

If the main rotor blades are to remain folded for a considerable length of time, the main rotor hub should be covered with a suitable weather-resistant cover (S4).

1-65. Unfolding Main Rotor Blades. *a.* Locate aircraft on most level ground available so that the load is balanced as much as possible.

CAUTION

During blade unfolding, use adequate covering over engine air inlet fairing opening to prevent entry of foreign objects into air intake. Install exhaust covers (S2) on aircraft with upward exhausts.

b. Remove padding between damper arms and blades.

c. Check that the locked positions of the control sticks are as follows:

(1) Collective pitch — mid position.

(2) Longitudinal cyclic — full aft.

(3) Lateral cyclic — two thirds of the total stick travel from full left toward full right.

CAUTION

During blade unfolding, continually monitor stick positions to make sure no change occurs. Serious blade damage can result from improper stick placement. Use care to prevent any scratching or marring of the blades by the blade attaching pins securing other blades. Avoid abrading blades against one another or against any other surface during the unfolding procedure. During unfolding operation, have an assistand positioned at outboard end of the blade.

d. Unfold right blade as follows:

(1) Release blade from rack (fig. 1-5).

(2) Remove blade attach pin from leading edge of lead-lag link for right blade.

(3) Lift right blade and rotate it forward to engage lead-lag link.

(4) Lift blade to align link and blade fitting holes and insert blade attach pin.

(5) Remove attach pin from damper arm, engage damper arm with trailing edge of right blade and insert attach pin.

e. Unfold left blade as follows:

(1) Release blade from rack (fig. 1-5).

(2) Remove blade attach pin from trailing edge of lead-lag link for left blade.

(3) Lift left blade and rotate it forward to engage lead-lag link. Lift blade to align link and blade fitting holes, and insert blade attach pin.

(4) Remove attach pin from damper arm, engage damper arm with trailing edge of left blade and insert attach pin.

f. Lock all attaching pins and check that the locking force is correct. **APPROXIMATE FORCE REQUIRED TO CLOSE HANDLE IS 25 — 35 POUNDS (MAXIMUM HAND PRESSURE).** Adjust pins having incorrect locking force.

(1) Remove pin and adjust by turning small nut at pin end. Do not adjust nut with pin installed.

(2) Reinstall and check locking action until handle closing is the maximum at which the safety latch can be forced over the nut **(25 — 35 POUNDS).**

g. Release trailing blade from blade rack; then remove blade rack from tailboom.

h. Remove blade sock and blade covers.

i. Reinstall any disconnected antennas.

j. Remove tape and cardboard from vibration absorber and check absorber for freedom of movement.

k. Loosen friction on the controis and check movement of controls through complete range of travel.

1-66. Towing. The aircraft may be towed either manually or by a vehicle.

CAUTION

If aircraft is to be towed or transported with the blades folded, the vibration pendulums must be secured with tape or cord to prevent them from striking adjacent blades as a result of bouncing.

Aircraft must not be towed unless flight control access door and fuel cell access doors are securely installed.

1-67. Installation — Ground Handling Wheels. (See fig. 1-6.) *a.* Position ground handling wheel assembly (S9), over the skid tube at the location of the skid fittings.

b. With ground handling wheels in the retracted position, align and engage the skid fittings.

c. Install jack handle (S10) in the wheel assembly socket, install lock pin, and rotate handle downward to lower the wheels and raise the aircraft.

WARNING

Hold downward pressure on jack handle until extended lock snaps into locked position.

d. Check that the extend lock is engaged,. release downward pressure and remove jack handle.

e. Install second ground handling wheel assembly on the other skid tube.

NOTE

Ground handling wheel tire pressure should be checked at regular intervals for 80-90 psi and the wheel bearing lubricated with grease (C45).

1-68. Manual Towing. Once the ground handling wheels are installed (para 1-67 above), the aircraft may be moved by hand; balance at the tailboom and push from the rear fuselage portion of the airframe.

1-69. Tug (vehicle) Towing.

CAUTION

The aircraft must be towed at a slow speed, not exceeding 5 mph, except under extreme emergency conditions. Do not allow the front end of the skid tubes to drag on the ground. Avoid sudden stops and starts and short turns which could cause the aircraft to turn over. Allow the inside wheel to rotate (not pivot) while aircraft is being turned. The proper minimum turning radius is approximately 20 feet.

a. Install ground handling wheels (para 1-67 above).

b. Have an assistant lower the tailboom slightly.

c. Position and secure a tow bar at the front end fittings of the wheel jacks.

d. Connect the tow bar to a suitable vehicle.

1-70. Removal — Ground Handling Wheels. *a.* Install jack handle in the wheel assembly socket and install the lock pin. Apply a downward pressure on the jack handle and manually release the extend lock.

WARNING

Keep a firm grip on the jack handle and keep all parts of the body clear of jack handle path of movement.

b. Apply an upward pressure; then rotate jack handle to raise the wheels and lower the aircraft.

c. Remove the ground handling wheel assembly from each skid fitting.

1-71. Jacking.

CAUTION

Aircraft must not be jacked unless flight control access door and fuel cell access doors are securely installed.

a. Provisions for jacking the aircraft (fig. 1-3) are provided by two forward (side) jacking point fittings (S3) and an aft jacking pad.

b. Install jacking fittings (S3) in the fuselage jacking points (fig. 1-3). Secure the jacking fittings with the locking pins that are secured to the fuel cell access doors.

c. Place suitable jacks under jacking fittings, and under aft jacking pad.

d. Raise the aircraft to the desired height.

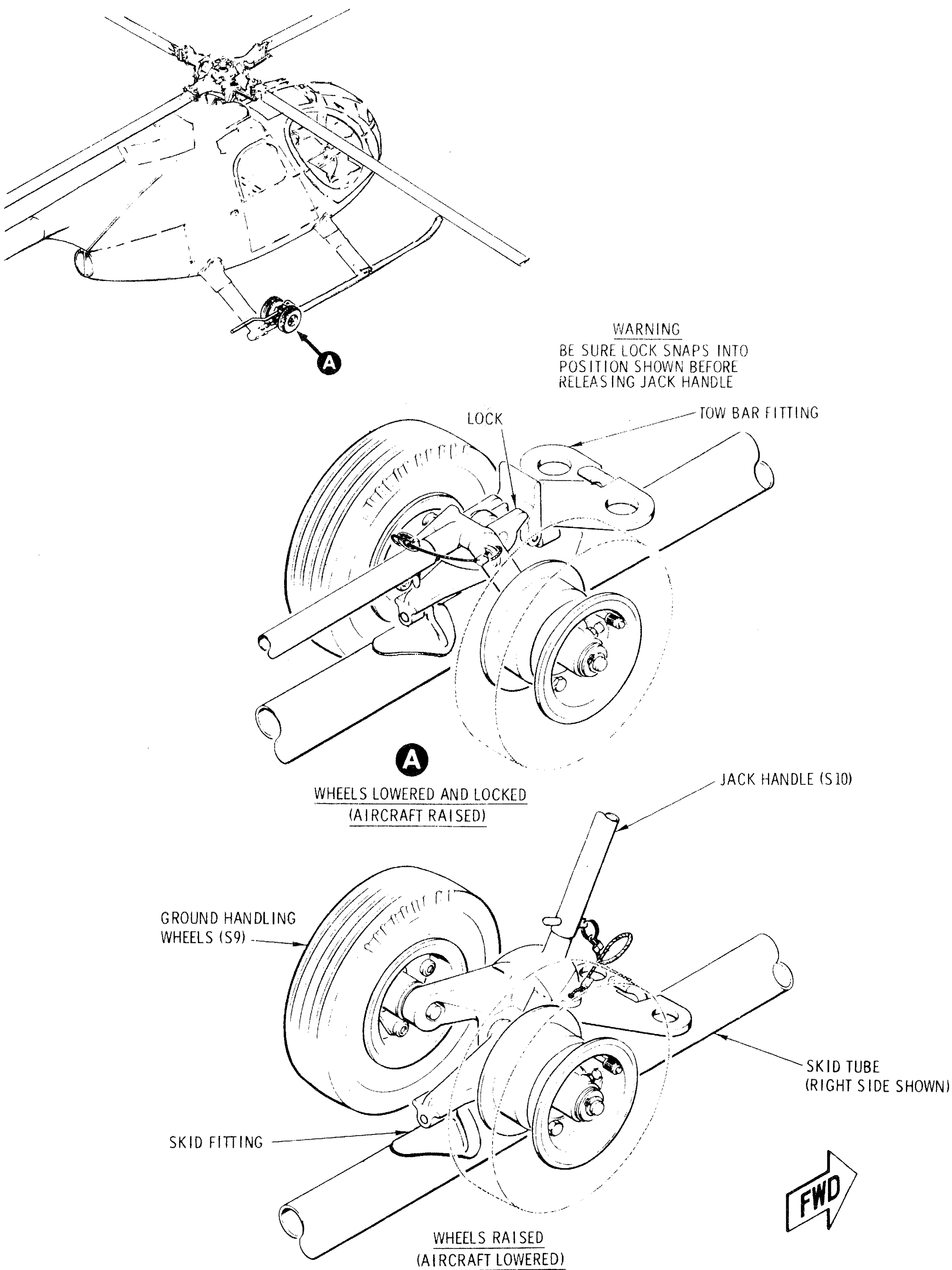


Figure 1-6. Ground Handling Wheels.

CAUTION

When the aircraft is jacked from one side only, a cushioned saddle-type support should be placed under the tailboom at the boom jacking location for extra stability. The landing gear skid on the lifted side should not be more than 6 inches off the ground. Take precautions to avoid bumping or dislodging the aircraft while it is being supported by jacks.

1-72. Parking. To park the aircraft for short periods (for example, between closely scheduled flights), proceed as follows:

CAUTION

To prevent rotor damage from blade flapping (droop stop pounding) as a result of air turbulence from other aircraft landing, taking off, or taxiing, or sudden wind gusts, rotor blades should be secured whenever aircraft is parked.

a. Locate the aircraft slightly more than rotor span from nearby objects, and on the most level ground available.

b. Apply friction to lock the cyclic stick so that the friction control knob is positioned on the lateral and longitudinal travel stop guides as follows: neutral laterally (center of slot), and one-third from rear of longitudinal slot.

NOTE

If not already accomplished, apply index paint marks on guide edges as permanent neutral position locator.

c. Secure the rotor blades as follows:

(1) Turn main rotor blades until they are at a 45-degree angle to the fuselage centerline (fig. 1-7).

(2) Install blade socks (S7) on all blades (fig. 1-7).

CAUTION

Take up slack in tethers but do not exert bending loads on blades.

(3) Secure blade sock tethers to fuselage jacking fittings.

d. Attach static ground wire to GROUND HERE receptacle as shown in figure 1-1.

1-73. Mooring. Local regulations or procedures supplement the following instructions. It is a command responsibility to provide maximum security for the aircraft consistent with actual weather and climatic conditions that prevail. Mooring is defined as a condition under which the aircraft is secured to the ground in the parked condition. The aircraft may be moored with the main rotor blades either extended or folded, depending on environmental requirements.

a. Secure all fuselage doors.

b. Install pitot tube cover (S6) or (S12) on pitot tube.

c. Install exhaust covers (S2) on aircraft with upward exhausts.

d. To moor with main rotor blades extended: Secure the blades to the jacking fittings as shown in figure 1-7, and anchor the aircraft by using standard Army mooring kit (S1).

e. To moor with main rotor blades folded: Fold main rotor blades (para 1-64) and anchor aircraft by using standard Army mooring kit (S1).

f. When wind velocities exceed 25 knots (or are forecasted to exceed 25 knots) accomplish the following additional mooring procedures:

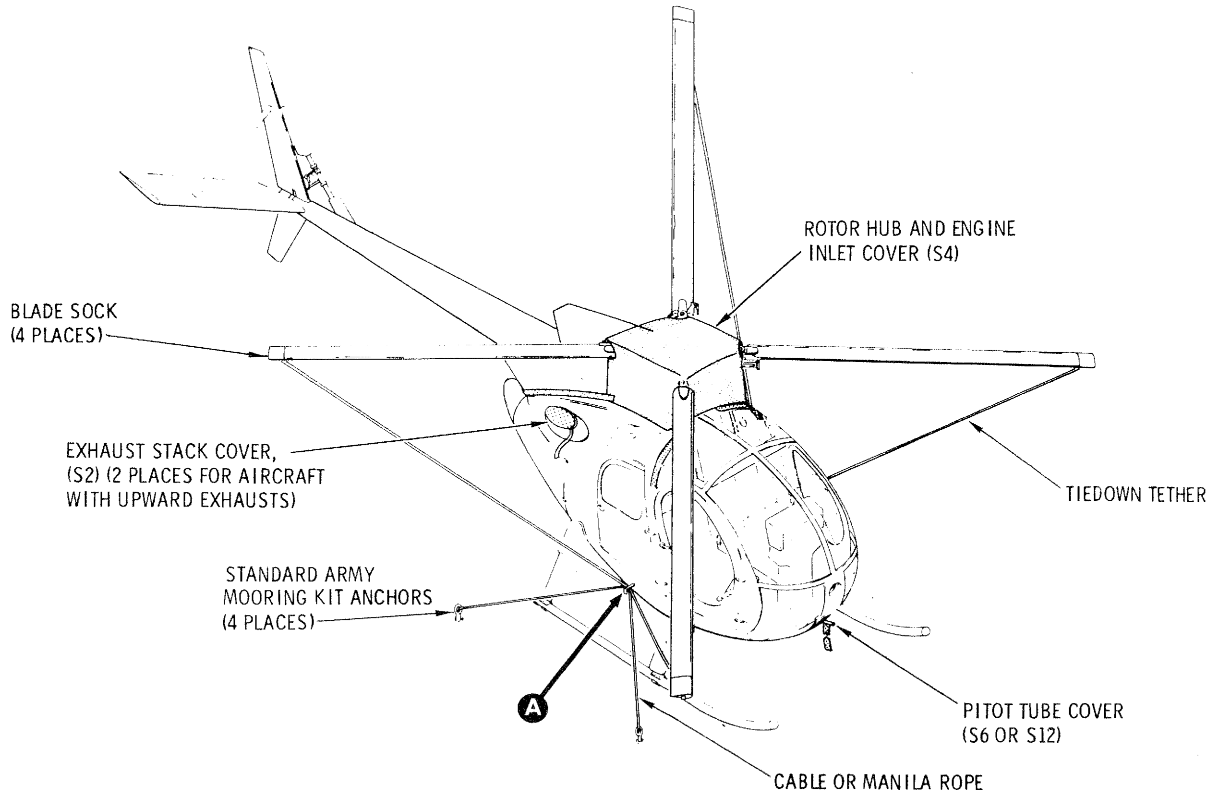
(1) Fill fuel tank.

(2) Apply friction to lock the cyclic and collective sticks.

g. Evacuate the aircraft to a safe area, such as a hanger or windbreak, whenever adverse or extreme weather conditions are forecasted or anticipated.

CAUTION

Whenever severe storm conditions (hurricanes, etc), or wind velocities higher than 40 knots are forecast, the aircraft should be evacuated to a safer area, if possible. The omission of additional references to specific precautions in this manual shall not be construed as relieving personnel of the



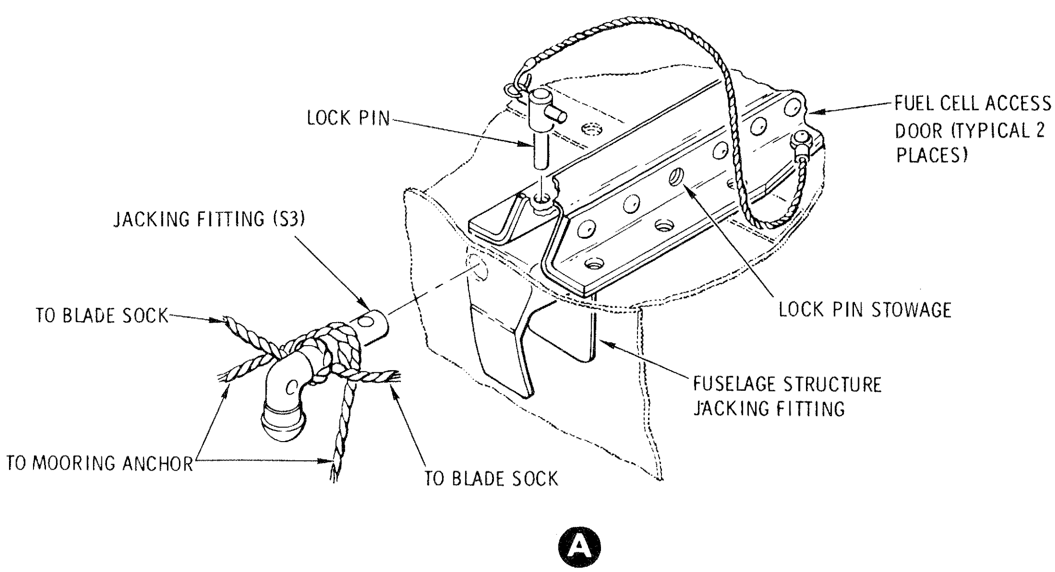


Figure 1-7. Securing Main Rotor Blades.

responsibility for performing any operation deemed necessary to provide maximum security for the aircraft.

1-74. Hoisting. Observe the following precautions (para 1-75) during any hoisting operation.

1-75. Hoisting Precautions. *a.* Use a hoist cf no less than 3500-pound capacity when hoisting the complete helicopter.

b. Use hoisting equipment of sufficient capacity (minimum twenty-percent over-rate) to hoist the heavier components if handled separately. (Table 1-6 lists approximate weights.)

CAUTION

Any time work is being performed near the engine air inlet, use care to prevent entry of foreign objects. On aircraft without a filter installed, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove covers until work is completed and any debris is thoroughly cleaned out of the area. After removing covers, verify that area around base of mast, inlet plenum, and entire plenum chamber is free of foreign material and reinstall plenum access doors.

1-76. Hoisting Operation.

Table 1-6. Approximate Hoisting Weights of Components.

Tailboom	14 lb
Main Rotor hub	56 lb
Main transmission	78 lb
Engine (built up)	180 lb
Aircraft (less engine)	980 lb
Aircraft (less main rotor hub, swashplate, scissors, and rotor blades)	970 lb
Aircraft (complete)	1160 lb

NOTE

On aircraft equipped with permanently installed lifting bracket illustrated in figure 1-3, special tool (T15) is not required, and steps a and b below are unnecessary.

a. Install hoisting adapter (T15) on main rotor hub so that the hoisting eyebolts fit into the slots on the hoisting adapter (fig. 1-3).

b. Install the four clevis pins.

c. Attach cable from overhead hoist to the adapter eye.

d. Secure a line to the tailboom. Have an assistant hold the line to keep the aircraft from swinging.

e. Hoist slowly and smoothly to maintain a steady lifting force.

1-77. Sling Loading (Airlift Recovery). For airlift recovery of the aircraft, use the belly band procedure described in ST 55-413-2 until publication of FM 55-413.

CAUTION

Lifting bracket (T9) shown in figure 1-3 shall not be used for aircraft (airlift) recovery.

1-78. Application of External Power. An external power receptacle is located at the right side of the pilot's compartment seat structure (fig. 1-1). A source of external power capable of 350 to 450 amperes and 28 volts is recommended for starting the engine; however, limits of 300 to 750 amperes and 28 volts are allowable for starting the engine.

CAUTION

Before connecting external power, make certain that the aircraft main electrical power selector switch is at the OFF position. After power is connected to the receptacle, the power switch must be at EXT position to connect external power to the aircraft electrical system.

NOTE

External power units with ampere ratings less than that specified above must be used with a battery (24 v) wired in parallel to provide amperage required. The positive (+) and negative (-) voltage terminals are clearly marked on the base of the receptacle to prevent reversing of polarity if a standard auxiliary power plug is not available.

SECTION IV INSPECTION REQUIREMENTS

1-79. INSPECTION REQUIREMENTS.

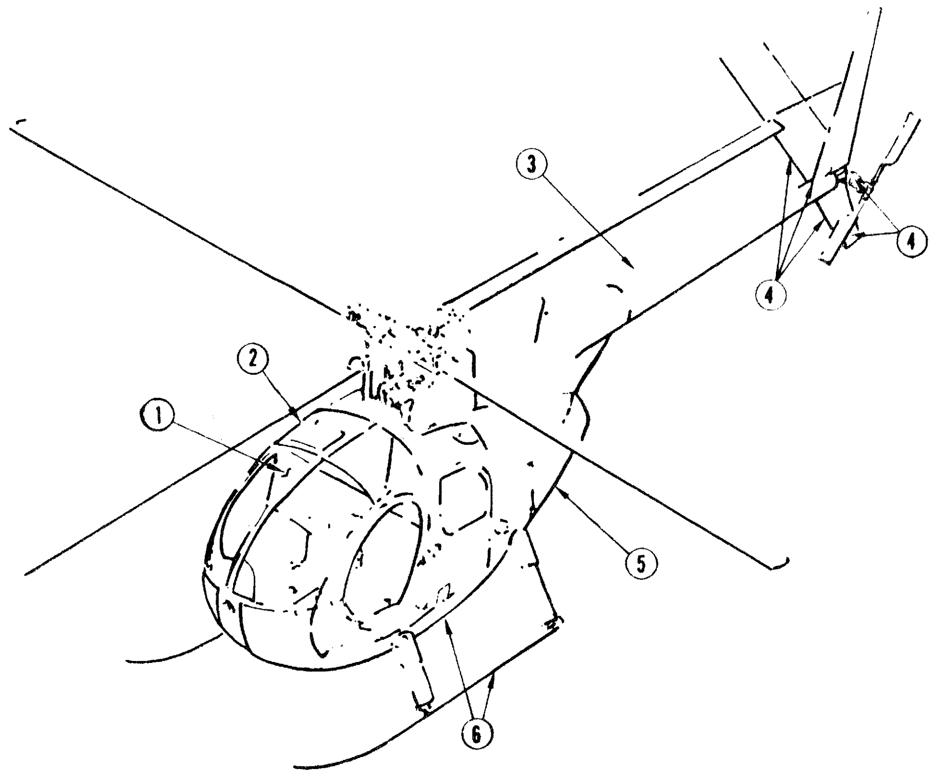
1-80. General — Inspection Requirements. This section contains complete requirements for special inspections, overhaul and retirement schedule, and standards of serviceability applicable to the OH-6A aircraft. The inspections prescribed in this section shall be accomplished at specified periods by aviation unit maintenance activities with the assistance of intermediate maintenance activities when required. Complete Daily, Intermediate, and Periodic inspections are contained in Preventive Maintenance Services checklists (TM 55-1520-214-PMS). In order to arrange inspection requirements as nearly as possible according to the manner in which work will be assigned, requirements are divided into groups under area headings (fig. 1-8). This manual pertains to all OH-6A series aircraft, and may therefore contain inspection requirements applicable to specific equipment not installed in individual aircraft. When this situation is encountered, requirements that are not applicable should be disregarded. Refer to TM 38-750 for applicable forms, records, and work sheets.

NOTE

During inspection of teflon lined bearings, refer to TM 55-1500-322-24 (Maintenance of Aeronautical Antifriction Bearings).

1-81. Standards of Serviceability. Standards of serviceability to be utilized in the day-to-day inspection and maintenance of the aircraft can be found as fits, tolerances, wear limits, and specifications in the aircraft maintenance manuals. Standards of serviceability for transfer of aircraft are contained in TM 55-1500-326-24.

1-82. Special Inspection. This supplements the scheduled inspections as outlined in the Preventive Maintenance Services, TM 55-1520-214 PMS, to include inspection of items which are required to be inspected at intervals not compatible with airframe operating time or airframe inspection intervals. Special inspection is as follows:



AREA NO. 1 CANOPY AND PILOTS COMPARTMENT

AREA NO. 2 RH FUSELAGE (FORWARD CENTER AND AFT SECTIONS) FRONT PART OF A - FRAME (TUNNEL AREA) RH MAIN ROTOR AND RH LANDING GEAR

AREA NO. 3 FUSELAGE BOOM

AREA NO.4 STABILIZER TAIL ROTOR TRANSMISSION AND TAIL ROTOR

AREA NO. 5 ENGINE COMPARTMENT

AREANO. 6 LH FUSELAGE (FORWARD CENTER AND AFT SECTIONS) FRONT PART OF A - FRAME (TUNNEL AREA) LH MAIN ROTOR AND LH LANDING GEAR

Figure 1-8. Area Diagram.

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AREA NO.	SPECIAL INSPEC-TION NO.	REQUIRE-MENT EVERY	ITEM	STATUS	RECORD-ED ON WORK SHEET
	1,		**HARD LANDING OR AFTER MAJOR DAMAGE**		
			Accomplish the following:		
5			a. Visually inspect engine mounts, mounting pads and firewall for damage and distortion. Inspect all suspected parts by the magnetic particle or flourescent penetrant methods, as applicable.		
2 & 6			b. Inspect landing gear and landing gear components and attaching areas for breaks, cracks, or other damage or distortion.		
2 & 6			c. Inspect fuselage fittings for bends and cracks.		
2.5 & 6			d. Inspect fuel cells, supports, tubing, hoses and breakaway fittings for leaks, cracks and security.		
5			e. Inspect engine chip detectors for metal particles (TM 55-2840-231-23).		
5			f. Inspect engine oil tank, supports, tubing and hoses for leaks, cracks and security.		
2 & 6			g. Inspect main transmission chip detectors for metal particles. Inspect main transmission drive shaft for distortion, breaks, cracks or other damage. Perform an alignment check of the main transmission drive shaft.		
5			h. Inspect engine accessory drive housing for cracks (TM 55-2840-231-23).		
5			i. Check all engine accessories for cracked flanges, loose bolts and nuts, connections, and general condition.		
2 & 6			j. Inspect tunnel area A-frame for distortion buckling or any other damage.		
3 & 4			k. Inspect tail boom, tail rotor drive shaft couplings and damper (shaft removed), tail rotor transmission and tail rotor for distortion, loose mounts or attaching parts, buckling, breaks, or other damage.		
2 & 5			l. Inspect transmission oil filter and engine oil and fuel filters for loose bolts and damaged filter elements (TM 55-2840-231-23).		
5			m. Check air, oil and fuel hose connections for tightness.		
All Areas			n. Inspect all flight and engine control system push-pull tubes, links, bell-cranks, and bearings for bends, cracks, security, and freedom of movement.		
2 & 6			o. If hard landing is made to low rpm inspect rotor blades for bending, cracks, or wrinkles.		
2 & 6			p. Inspect rotor blade droop stops for damage.		
2 & 6			q. Inspect accessible areas of main rotor mast assembly and transmission attachment area.		
1, 2, 5 & 6			r. On aircraft equipped with armor, inspect armor for security of attachment, buckling, and distortion.		
2			s. Inspect keep beam in cargo compartment for buckling, loose or missing rivets, and tears or cracks in web or stiffeners.		

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
			NOTE **Sudden stoppage is defined as an abnormal deceleration of the drive system caused by the main rotor or tail rotor blades striking something which causes rapid deceleration.** **Sudden stoppage could be further defined as an instantaneous shock load applied to the drive train and rotor system. Shock loads result from blades striking an object.**		
2 & 6	2.		**MAIN ROTOR BLADE STRIKE (MAJOR VISIBLE DAMAGE) OR AFTER SUDDEN STOPPAGE.** Inspect the following: a. If the main rotor blades contact an object while rotating, inspect the rotor blades for security of all bonds and visible damage. If the bond is separated in any area, or dents and scratches are in excess of allowable limits, replace blades.		
5			b. Inspect engine mounts for security, cracks or misalignment.		
2 & 6			c. Inspect transmission oil filter and each chip detector for metal particles. d. Inspect compressor rotor and stator blades and turbine blades (TM 55-2840-231-23) for foreign object damage.		
2 & 5			e. On aircraft without engine air filter, inspect engine air inlet for foreign objects; then motor engine and check for unusual noise.		
2, 5 & 6			f. Reinspect chip detectors after 8 hours of engine operation.		
2, 3, 4			g. Inspect all power train drive shafts and drive shaft couplings (tail rotor drive shaft removed).		
4			h. Inspect tail rotor assembly.		
2 & 6			i. Inspect complete main rotor assembly for evidence of damage.		
	3.		**TAIL ROTOR BLADE STRIKE** Accomplish the following: a. Inspect tail rotor transmission and main transmission chip detectors for metal particles.		
2, 4, & 6			b. Inspect tail rotor drive shaft (shaft removed), damper and couplings for distortion, breaks, cracks or other damage. Shaft or damper distortion is cause for replacement of tail rotor transmission.		
4			c. Inspect tail rotor transmission for TIR of output level pinion gear shaft cracks (5 power magnifying glass required) in mounting flanges and output shaft housing support junction area.		

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET	
4			d. Inspect aft frame of tail boom for cracks and boom skin for loosened or popped rivets.			
	4.		**POWER TRAIN SUDDEN STOPPAGE**			
			Accomplish the following:			
			NOTE: Power train sudden stoppage will be indicated by signs of impact between main rotor lead-lag arms and lag stop or by compression damage to trailing edge of rotor blades immediately outboard of damper attachment point.			
2 & 6			a. If damper arm has contacted damper stop, replace main rotor hub, main rotor drive shaft, main drive shaft, and main transmission; inspect tail rotor drive shaft and couplings (shaft removed) for damage.			
2 & 6			b. If damper arm has not contacted damper stop, visually inspect main rotor hub for damage, paying particular attention to droop stop mechanism and strap packs.			
3			c. Inspect aircraft structure forward of tailboom attachment for evidence of sheet metal yielding or buckling, paying particular attention to signs of buckling at right-hand side of aircraft forward of tailboom attachment.			
5			d. Inspect engine mounts for security, cracks, or alignment.			
2 & 5			e. Inspect compressor rotor and stator blades and turbine blades (TM 55-2840-231-23) for foreign object damage.			
2 & 5			f. On aircraft without engine air filter, inspect engine inlet for foreign objects, then motor engine and check for unusual noise.			
2, 5, & 6			g. Reinspect chip detectors after 8 hours of engine operation.			
4			h. Inspect tail rotor assembly.			
			i. Inspect overrunning clutch subassembly for damage.			
2	5.		**MAIN ROTOR OVERSPEED IN EXCESS OF LIMIT SPECIFIED**			
			(TM 55-1520-214-10.) Accomplish the following:			
			a. If overspeed is between 514 and 540 rpm:			
			(1) Remove main rotor blades: Inspect for visible damage and distortion; vibration absorbers and tip caps for security; all bond lines for evidence of separation; root doublers and attachment fittings for security. Replace any blade with visible bond line separation.			
			(2) Inspect tail rotor blades and hub for visible damage; freedom of movement and security.			
			(3) Refer to main transmission overspeed inspection requirements			
			b. If overspeed is over 540 rpm.			
			(1) Remove main rotor hub assembly and tail rotor assembly and tag for overhaul.			
			(2) Replace main rotor blades.			
			(3) Refer to main transmission overspeed inspection requirements.			

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
2	6.		**MAIN TRANSMISSION OVERSPEED IN EXCESS OF 514 AND 540 ROTOR RPM** Accomplish the following: a. If overspeed is between 514 and 540 rpm: Inspect main rotor transmission and tail rotor transmission chip detectors for metal particles. b. If overspeed is over 540 rpm: (1) Remove main transmission and tag for overhaul. (2) Remove tail rotor transmission and tag for overhaul.		
2	7.		**MAIN TRANSMISSION OVERTORQUE** (TM 55-1520-214-10.) Accomplish the following: a. If overtorque does not exceed transient limits: Inspect main rotor transmission chip detectors for metal particles. b. If overtorque exceeds transient limits: Remove main transmission and tag for overhaul.		
4	8.		**UPON REPLACEMENT OF TAIL ROTOR** Inspect the following: Tail rotor control assembly.		
5	9.		**IF ENGINE OIL TEMPERATURE EXCEEDS 225°F WITHOUT ANY CHANGE IN NORMAL OIL PRESSURE** Refer to TM 55-2840-231-23 a. Ground run aircraft for 10 minutes. Obtain as high a power level as possible without liftoff. b. Reinspect the chip detectors. If the chip detectors are free of particles, reinspect after 5 hours of operation.		
5	10.		**IF ENGINE OIL TEMPERATURE EXCEEDS 225°F WITH A CHANGE IN NORMAL OIL PRESSURE.** Refer to TM 55-2840-231-23.		
5	11.		**FUEL FILTER CAUTION LIGHT INDICATION** Accomplish the following: a. Replace fuel pump filter element. b. Clean fuel control filter assembly (see TM 55-2840-231-23).		
2, 4, 5 & 6	12.		**CHIP DETECTOR LIGHT INDICATION** Accomplish the following: a. On series 1 and 2 aircraft, remove chip detector leads, in turn, or use chip detector switch (as applicable) to determine which chip detector caused the light to go on.		

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			b. Remove and inspect the applicable engine (TM 55-2840-231-23) or transmission chip detector for metal particles. c. Clean the engine oil filter if an engine detector cause the light (TM 55-2840-231-23). d. Clean transmission oil filter if transmission chip detector caused light to go on. e. Request special oil analysis in accordance with TB 43-0106.		
2 & 5	13.		**ENGINE OVERSPEED OR OVERTORQUE IN EXCESS OF LIMITS SPECIFIED (TM 55-1520-214-10)** Accomplish the following: Remove the engine and tag for overhaul (TM 55-2840-231-23).		
2	14.		**AFTER EMERGENCY FUEL USAGE** Emergency fuel usage is limited to an accumulative total of 6 hours. When this limit is exceeded, remove the engine and return it for overhaul. Any mixture of emergency fuel to regular fuel greater than 1% of emergency fuel must be considered emergency fuel usage. Defuel the tank completely. Open the tank sump and drain the remaining fuel into a suitable container. No hot end inspection or operational check is required.		
	15.		**AFTER COMPRESSOR STALL** Refer to TM 55-2840-231-23 and TM 55-2840-241-23.		
5	16.		**ENGINE CHANGE OR REINSTALLATION** Accomplish the following: a. Inspect all fuel and oil system components, lines, hoses and connections for leaks, chafing, and security. Inspect metal line for dents and cracks. b. Perform test adjustment of TOT indicating system. c. Engine vibration test required.		
4	17.		**ENGINE PERFORMANCE CHECK REQUIRED** Refer to TM 55-1520-214-MTF. a. After installation of engine. b. Any time poor performance is noted. c. Removal/Replacement, Repair Adjustment of fuel control, Governor, Double check valve, compressor. d. Removal/Installation of reverse flow inlets. e. Any time a pneumatic line is loosened. f. Calibration/Replacement of TOT indicating system. Replacement of FAT gauge for error.		

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5	18.		**INSTALLATION OF AN ENGINE, ENGINE COMPONENTS, MAIN TRANSMISSION OR TAIL ROTOR TRANSMISSION** Accomplish the following: a. Operate engine and check for leaks. b. Shut down and recheck for leaks. c. Inspect firewall bulkhead for damage and distortion evidenced by buckling, cracks, etc.		
2 & 5	19.		**ENGINE CHANGE AS A RESULT OF INTERNAL ENGINE FAILURE** Accomplish the following: a. Remove, clean, and reinstall oil tank, replace oil cooler. b. Flush all lines, fittings and components.		
5	20.		**OVERTEMPERATURE OPERATION** Inspection according to TM 55-2840-231-23 is required. NOTE: Perform adjustment/test of temperature system prior to any engine maintenance action. a. If temperature exceeded 749°C for more than 10 seconds any time during start. b. If temperature was above 927°C any time during start. c. If temperature was 749°C for more than 6 seconds or exceeded 843°C any time during power transient. d. If temperature exceeded 693°C for 5 minutes or more continuous operation.		
	21.		**SUSPECTED FOREIGN OBJECT DAMAGE** Accomplish the following:		
2 & 5			a. Inspect compressor rotor blades and stator vanes and turbine blades (TM 55-2840-231-23).		
5			b. Motor engine and listen for unusual noises. Observe starter limitations.		
5	22.		**ENGINE FLAMEOUT DUE TO POSSIBLE SNOW, ICE, OR WATER IN INGESTION** Accomplish the following according to TM 55-2840-231-23. a. Obtain access to the compressor inlet and inspect vanes of compressor front support and first stage compressor blades for mechanical damage, distortion, or bending. b. Remove and replace compressor if damage noted above is found.		

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5	23.		**IF AIRCRAFT HAS BEEN SUBJECTED TO SALT WATER, SALT WATER SPRAY OR OPERATED WITHIN 10 MILES OF SALT WATER** Accomplish the following: a. Wash engine internally using the procedures outlined in the Compressor Cleaning to Remove Salt Water Contamination section of TM 55-1520-231-23. b. Wash entire aircraft as prescribed by TM 55-1500-333-24.		
	23.1		**IF AIRCRAFT IS OPERATED WITHIN 200 MILES OF VOLCANIC ACTIVITY** Accomplish the following: Water wash engine using the procedures outlined in the Compressor Cleaning to Remove Salt Water Contamination Section of TM 55-1520-231-23.		
2 & 4	24.		**FIRST FLIGHT FOLLOWING ENGINE, MAIN TRANSMISSION, OR TAIL ROTOR TRANSMISSION CHANGE OR REINSTALLATION** Accomplish the following after the first flight as applicable: a. Inspect fuel and oil lines for leakage and security. b. Inspect chip detectors for metal particles. c. Inspect electrical connections for security. d. Inspect all components and engine mounting points for security.		
	25.		**WHEN EQUIPMENT REPLACEMENT, RELOCATION OR MODIFICATION MIGHT RESULT IN COMPASS DEVIATION** Accomplish the following: Check magnetic standby and remote compass indicators for a correct reading on all cardinal headings. If necessary, recompensate.		

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	26.		**AIRCRAFT AIRSPEED BEYOND 10% ABOVE VNE LIMITS** (TM 55-1520-214-10) Accomplish the following:		
2 & 6			a. Remove main rotor blades. Inspect for visible damage and distortion; vibration absorbers and tip caps for security; all bond lines for evidence of separation; root doublers and attachment fittings for security. Replace any blade with visible bond line separation.		
4			b. Inspect tail rotor blades and hub for visible damage; freedom of movement and security.		
4			c. Inspect horizontal and vertical stabilizers; mounting fittings for cracks; retorque mounting bolts; skin for damage and loose rivets; strut fittings lugs for cracks, strut for obvious damage and security.		
3			d. Inspect tail boom for visible deformation, loose or missing rivets, cracks and security, attaching points to fuselage for cracks and security.		
1			e. Inspect canopy glass for security.		

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
1	27.		**INSTALLATION OF NEW BATTERY** Every time a new nickel-cadmium battery is installed, the following will be performed before installation: Prepare for service according to TM 11-6140-203-14-2.		
2 & 6	28.		**MAIN TRANSMISSION DRIVE SHAFT REMOVAL** Accomplish the following: a. Check oil level in overrunning clutch.		
2 & 6			b. When main transmission oil is drained for any reason, remove sight GAGE, rotate main rotor and using an adequate light visually inspect to determine that all sixteen (16) ring gear bolts and securing lockwire are intact and that no cracks are evident on visible areas of the ring gear or gear shaft.		
2 & 6	29.		**NEW OR REPLACEMENT MAIN ROTOR HUB INSTALLATION** Accomplish the following: a. After installation of the main rotor blades inspect the blade droop angle. b. Inspect under flexible boot for excessive grease that could contaminate teflon bearing after 5 to 10 hours of flight.		
2	30.		**SWASHPLATE SUPPORT BEARING (369A7003 or 369A7003-3) REPLACEMENT** Accomplish the following: Inspect swashplate with fluorescent penetrant.		
2 & 6	31.		**WHEN MAIN ROTOR BLADES HAVING RESTRICTED SERVICE REPAIRS ARE INSTALLED** Accomplish the following: a. Perform a daily inspection of all repaired areas for cracks. Replace any blade that has cracks progressing from a repaired area. b. Perform a fluorescent penetrant inspection, as required.		
All Areas	32.		**INVENTORY AIRCRAFT UPON TRANSFER AND UPON RECEIPT OF AN AIRCRAFT; UPON EXPIRATION OF TWELVE MONTHS ELAPSED TIME SINCE LAST INVENTORY; AND UPON PLACING AIRCRAFT IN STORAGE AND UPON REMOVING AIRCRAFT FROM STORAGE.** (Aircraft need not be inventoried while in storage.)		
All Areas	33.		**AFTER AN AIRCRAFT HAS BEEN IDLE FOR SEVEN (7) CONSECUTIVE DAYS AND ALL SEVEN (7) CONSECUTIVE DAY PERIODS THEREAFTER – PERFORM A DAILY INSPECTION TO INCLUDE AN ENGINE RUN-UP**		

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
All Areas	34.		**LIGHTNING STRIKE INSPECTION** a. General Requirement whenever the aircraft is struck by lightning: (1) Inspect the fuselage interior and exterior, the landing gear, the rotor systems and ground wire connection for burn marks, cracks, pitting or other signs of high temperature stress, to determine the lightning entry and exit points. (2) Trace the path of the lightning strike to the extent possible using a magnetometer. (3) Check the magnetic compass for accuracy (the degree of inaccuracy may serve as an indicator of the severity of the strike). (4) Inspect all wiring. (5) Inspect antenna for burns and pitting. (6) Inspect all electrically operated components and lighting systems for damage. (7) Inspect communications and navigation equipment for damage. (8) If the preceding steps (1) through (7) reveal major damage has occurred, proceed as follows: (a) Bench test all avionics and electrical systems and components. (b) Perform a continuity check on all wiring and cables. (c) Perform a Voltage Standing Wave Ratio (VSWR) check on all antennas, antenna cables, and connectors. (9) Perform specific inspections/replacements as required. (10) Perform a ground run operational check on the aircraft. Functionally check the flight control system, and all avionics, electrical, lighting, communication, and navigation systems. (11) Repair any damage and replace damaged components as required using standard maintenance practices. b. Specific Requirements whenever lightning strike is evident on main rotor system: (1) Inspect blades for damage such as burns, pitting, skin separation, etc. If damage is evident, locally scrap damaged blade(s). (2) Remove hub assembly and return for overhaul. (3) Inspect all bearings in the fixed and rotating control system located on the main rotor mast.		

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			(4) Remove transmission assembly and return for overhaul.		
			(5) Inspect Main Rotor Mast and drive shaft for evidence of burns.		
			(6) Check drive shafts for residual magnetism. If magnetized or damaged replace drive shafts and remove engine and return for overhaul.		
			c. Specific Requirements whenever lightning strike is evident on tail rotor system:		
			(1) Inspect blades for damage such as burns, pitting, skin separation, etc. If damage is evident, locally scrap damaged blade(s).		
			(2) Tail rotor assembly return to overhaul.		
			(3) Remove and condemn pitch change links, and pitch change assembly.		
			(4) Inspect bellcrank and control rod for any indications of arcing. Replace as necessary.		
			(5) Remove tail rotor gearbox and return for overhaul.		
			(6) Inspect tail rotor driveshaft and driveshaft dampner for magnetism and/or burns.		
			(7) Check Oil Cooler Blower Assembly, Overrunning Clutch, and Tail Rotor Driveshaft Couplings for residual magnetism. Replace as necessary.		
			(8) Inspect Oil Cooler Assembly for damage. Replace as necessary.		
			(9) If previous drive train items show magnetism, remove transmission and return for overhaul.		
			(10) If the Overrunning Clutch Assembly shows magnetism remove Engine Assembly and return for overhaul.		
			(11) Inspect engine mounts, and fittings for damage. Replace as necessary.		
1	35.		**25 ± 2.5 HOURS OR 30 DAYS, WHICHEVER OCCURS FIRST**		
			Accomplish the following:		
			Perform preventive maintenance checks and services on the nickel-cadmium battery. (Refer to TM 11-6140-203-14-2.)		
5	36.		**50 ± 5 HOURS, ACCOMPLISH THE FOLLOWING**		
			Clean engine compressor (TM 55-2840-231-23).		

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
5	37.		**EVERY 90 DAYS** Accomplish the following: Inspect Power Coupling Nut on all T63-700/5A Engines in accordance with TM 55-2840-231-23. This inspection is not required if nut part number 6890531 is installed. Engines having this nut installed will be identified with a suffix "C" after the serial number. (Example AE400100ABC, AE402603BC.)		
2	38.		**100 ± 10 HOURS** Accomplish the following:		
2 & 6			Every 100 hours, remove controls support Bracket Assembly and penetrant inspect critical area of bracket and matching cap for cracks. Inspect inboard collective torque tube bearing for binding or roughness. This inspection not required on aircraft verified to have steel strap installed on serviceable bracket assembly per MWO 55-1520-214-30-18. When steel strap is installed, visually inspect every 300 hours.		
	39.		**100 ± 10 HOURS OR 120 CALENDAR DAYS, WHICHEVER OCCURS FIRST** Accomplish the following: a. Perform preventive maintenance checks and services on the nickel-cadmium battery (TM 11-6140-203-14-2). b. Check voltage regulator setting; adjust for temperature as required. Refer to TM 55-1500-204-25/1.		
5	40.		REQUIREMENT EVERY 150 HOURS ± 15 HOURS, ACCOMPLISH THE FOLLOWING: a. Remove, inspect, clean and reinstall engine oil filter (TM 55-2840-231-23). Drain oil systen and refill. (Refer to paragraph 1-10.) b. (Applicable to aircraft with armor provisions installed.) Perform an operational check of oil cooler bypass switch and solenoid valve according to paragraph 4-169. c. Apply a film of lubricant (C63) to starter-generator drive shaft splines. d. Perform an engine deceleration check (TM 55-2840-231-23).		

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AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
			NOTE **Above check is only required when using alternate fuel.**		
			e. Remove landing gear strut fairings and inspect struts for cracks or loose rivets. (Not applicable to one piece struts.)		
			f. Inspect and repack overrunning clutch bearings. (Refer to paragraph 6-30.1.)		
			REQUIREMENT EVERY 150 HOURS ± 15 HOURS OR 12 CALENDAR MONTHS, WHICHEVER OCCURS FIRST.		
			a. Remove, clean and inspect transmission oil filter. (Refer to paragraph 1-15.)		
			b. Drain oil from transmission and refill. (Refer to paragraph 1-13.) Also remove oil sight gauge and inspect ring gear bolts. (Refer to SOF OH-6A-80-5.)		
			c. Drain oil from tail rotor transmission and refill. (Refer to paragraph 1-16.)		
			REQUIREMENT EVERY 150 HOURS ± 15 HOURS OR 6 CALENDAR MONTHS; WHICHEVER OCCURS FIRST.		
			Remove main rotor blade pins and damper attach pin and accomplish the following:		
			a. Visually inspect for excessive wear, evidences of corrosion or cracking. Any evidences of excessive wear, corrosion or cracking requires replacement of the attach pin.		
			b. Using a 5X magnifying glass, inspect area of cam locking lever at top attaching point for cracks. Any evidence of cracking requires replacement of the attach pin.		
			c. Accomplish corrosion control lubrication of pivoting surface.		
			d. Reinstall attach pins.		

AIRCRAFT INSPECTION CHECKSHEET	TYPE OF INSPECTION SPECIAL	PAGE NO.	NO. OF PAGES
AIRCRAFT AND SERIAL NO.	INSPECTION NO.	DATE OF INSPECTION	

AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
1	41.		**6 MONTHS** Accomplish the following: Inspect portable fire extinguisher (TM 55-1500-204-25/1).		
5	42.		**EVERY 300 ± 30 HOURS OR 12 CALENDAR MONTHS, WHICHEVER OCCURS FIRST (TM 55-2840-231-23).** Accomplish the following: a. Replace the fuel pump filter element. b. Clean the fuel control filter assembly.		
2 & 6			c. Inspect Vibration Absorber bracket for cracks and breaks. d. Free air temperature gauge — Inspect for obvious damage and test in accordance with TM 55-1500-204-25/1, or replace if required.		

AIRCRAFT INSPECTION CHECKSHEET	TYPE OF INSPECTION SPECIAL	PAGE NO.	NO. OF PAGES
AIRCRAFT AND SERIAL NO.	INSPECTION NO.	DATE OF INSPECTION	

AREA NO.	SPECIAL INSPECTION NO.	REQUIREMENT EVERY	ITEM	STATUS	RECORDED ON WORK SHEET
1	43.		**12 MONTHS** Accomplish the following: Check magnetic standby compass for discoloration of liquid and proper calibration. If necessary, recompensate. (Refer to TM 55-1500-204-25/1.) Check remote compass indicator for calibration. If necessary, recompensate. (Refer to TM 55-1500-204-25/1.)		
1	44.		**24 MONTHS OR NEAREST SCHEDULED INSPECTION** Accomplish the following: a. Perform functional check of pitot static system and instruments. b. For encoding altimeters, after the 24 month test or replacement of the altimeter, check mode C (altitude) test using the appropriate transponder test procedure in TM 11-6625-667-12 or TM 11-4920-296-14&P. c. Replace nylon cloth, Raschel Knit, seat covers (pilot and co-pilot). Refer to TM 55-1500-204-25/1.		
All	45.		**30 MONTHS** Weigh aircraft and perform weight and balance records check (Refer to TM 55-1520-214-10 and TM 55-405-9).		
1	46.		**36 MONTHS** Accomplish the following: Inspect crashworthy fuel cells. (Refer to paragraph 10-2A.)		
	47.		**PITOT STATIC FUNCTION CHECK** A functional check of Pitot Static System and Pitot Instruments will be performed following any opening and closing of the Pitot Static System, except for the use of system drain.		

SECTION V OVERHAUL AND RETIREMENT SCHEDULE

1-83. OVERHAUL AND RETIREMENT SCHEDULE.

WARNING

TM 55-1500-328-25 should be refered to concerning mutilation/destruction of items when they have reached the established life expectancy (finite life) before the items are forwarded for property disposal.

1-84. General - Overhaul and Retirement Schedule. This section lists (table 1-7) units of operating equipment that are to be overhauled or retired at the period specified. Removal of equipment for overhaul may be accomplished at the inspection nearest the time when overhaul is due unless otherwise specified in TM 55-1500-328-25. Upon replacement of items listed in this chapter, all applicable forms, records and worksheets will be completed and updated as required (TM 38-750).

1-85. Overhaul Interval. The maximum authorized operating time of parts prior to removal for overhaul at level authorized in accordance with the Maintenance Allocation Chart.

1-86. Retirement Interval. The operating time specified for removal, condemnation, and disposal of parts in accordance with applicable directives.

Table 1-7. Overhaul and Retirement Schedule.

Area (fig. 1-8)	Overhaul Interval (hr)	Retirement Interval (hr)	Item	Part Number
2	750		Main transmission assy	369A5100 369A5100-601 369A5100-603 369A5100-605 369A5100-607 369A5100-609
2	750		Transmission lube pump	369A5264
2		1,200	Swashplate control bearing assy	369A7003 369A7003-3
2 & 6		5,760	Main rotor blade attach pin	369A1004 369A1004-3
2 & 6		1,570	Main rotor blade assy	369A1100
2 & 6		5,710	Main rotor mast assy	369A2014 369A2014-601
2 & 6		1,990 3,900	Main rotor drive shaft	369A5520 369A5500
2 & 6		3,700	Interconnect center shaft	369A5510
2 & 6	1,200		Main rotor hub assy	369A1200 369A1200-3 369A1200-615 369A1200-617
3		8,730	Tail rotor drive shaft assy	369A5518 369A5518-601
3		2,177	Tail boom assy	369A3500 369A3500-601 369A3500-603 369A3500-605 369A3500-615 369A3500-617 369A3500-619
4		3,000	Horizontal stabilizer assy	369A3600 369A3600-601 369A3600-603 369A3600-605
4		3,170	Upper vertical stabilizer assy	369A3625 369A3625-601 369A3625-603
4	500 500 1,500 750		Tail rotor transmission	369A5400 369A5400-601 369A5400-605 369A5400-607
4	600		Tail rotor assy	369A1600-3 369A1600-5 369A1600-7 369A1600-9

Table 1-7. Overhaul and Retirement Schedule. (cont)

Area (fig. 1-8)	Overhaul Interval (hr)	Retirement Interval (hr)	Item	Part Number
	600		Hub and blade assy	369A1600-21 369A1600-907
	600		Hub and Blade Assy	369A1620-603
	600			369A1620-609
4		7,080	Tail rotor shaft coupling	369A5501 19E111-1A
5	750		Turboshaft engine (Allison)	6852600
	750			6874201
5	1,000		Turboshaft engine (Suffix "B" serial numbers only) (Allison)	6874201
4		5600	Tail Rotor Blade Assy	369A1613-7
4		5100	Tension-Torsion Stray Assy	369A1706 369A1706-5d

CHAPTER 2

AIRFRAME

SECTION I FUSELAGE

2-1. FUSELAGE.

2-2. Description — Fuselage. The fuselage is divided into three major sections; forward section, lower section, and aft section.

a. Forward Section. The fuselage forward section includes the canopy installation and pilot's door frames.

b. Lower Section. The fuselage lower section includes the pilot's seat support structure, pilot's compartment floor, cargo compartment floor, underfloor electronics compartments and battery area, floor and seat support bulkheads and associated structure, the center beam assembly, landing gear fittings, and the fuel tank support structure.

c. Aft Section. The fuselage aft section includes the main rotor mast support structure, cargo door frames, engine compartment, engine air inlet (plenum chamber) installation, engine inlet aft fairing, firewall installation, and the boom fairing.

2-3. NON-STRESSED ACCESS DOORS AND COVERS.

2-4. Description — Non-Stressed Access Doors and Covers. Removable access doors and covers (fig. 2-1, sh 1 thru 4) are provided for servicing, inspection, removal, installation, and adjustment of components. Except for stress areas, the access provisions provided in the outer fuselage, the pilot's compartment, the cargo compartment and aft bulkhead stations have turnlock type quick-opening fasteners. Screws are used to secure access doors in stress areas. Liquid level plugs allow "sight" inspection of the lubricant levels of the main transmission, engine oil tank, and tail rotor transmission.

CAUTION

Any time maintenance work is to be performed near the engine air inlet, use care to prevent entry of foreign objects that might later be sucked into the compressor. On aircraft without a barrier filter installed, tape covers of cardboard or other suitable material in place over engine inlet screen in the plenum chamber. Plug the oil cooler air inlets. Covers should not be removed until work is completed and debris is thoroughly cleaned out of the area.

2-5. Inspection — Non-Stressed Access Doors and Covers. Inspect non-stressed access doors and covers for the following:

a. Cracks and other visible damage.

b. Turnlock fasteners and receptacles for proper fastening.

c. Cover gaskets, if applicable, for deterioration or missing sections.

2-6. Repairs — Non-Stressed Access Doors and Covers. Refer to paragraph 2-7.

2-7. Repair and Replacement — Non-Stressed Sheet Metal Damage. Non-stressed sheet metal members consist primarily of hinged covers, access covers and doors (except the fuel cell access and controls access doors which are stressed). Guidelines defining repair of damage are described in paragraph 2-269 and as follows.

2-8. Negligible Damage — Non-Stressed Sheet Metal. Small dents, scratches, nicks, and light corrosion deposits are considered negligible damage. Cracks that do not exceed 0.25 inch in length, are less than one-fourth the width of the damaged component, are removed at least 1 inch from the end of the damaged component or an attachment point, may also be considered negligible damage.

a. Replace defective rivets, nutplates, or fasteners.

b. Replace defective gaskets, as applicable.

c. Smooth-countour dents that do not exceed 0.12-inch depth do not require repair if they will not damage or deform mating structure.

2-9. Patch or Insertion Repair — Non-Stressed Sheet Metal. Holes and cracks in non-stressed sheet metal may be patched if they do not exceed the following percentages of total surface area:

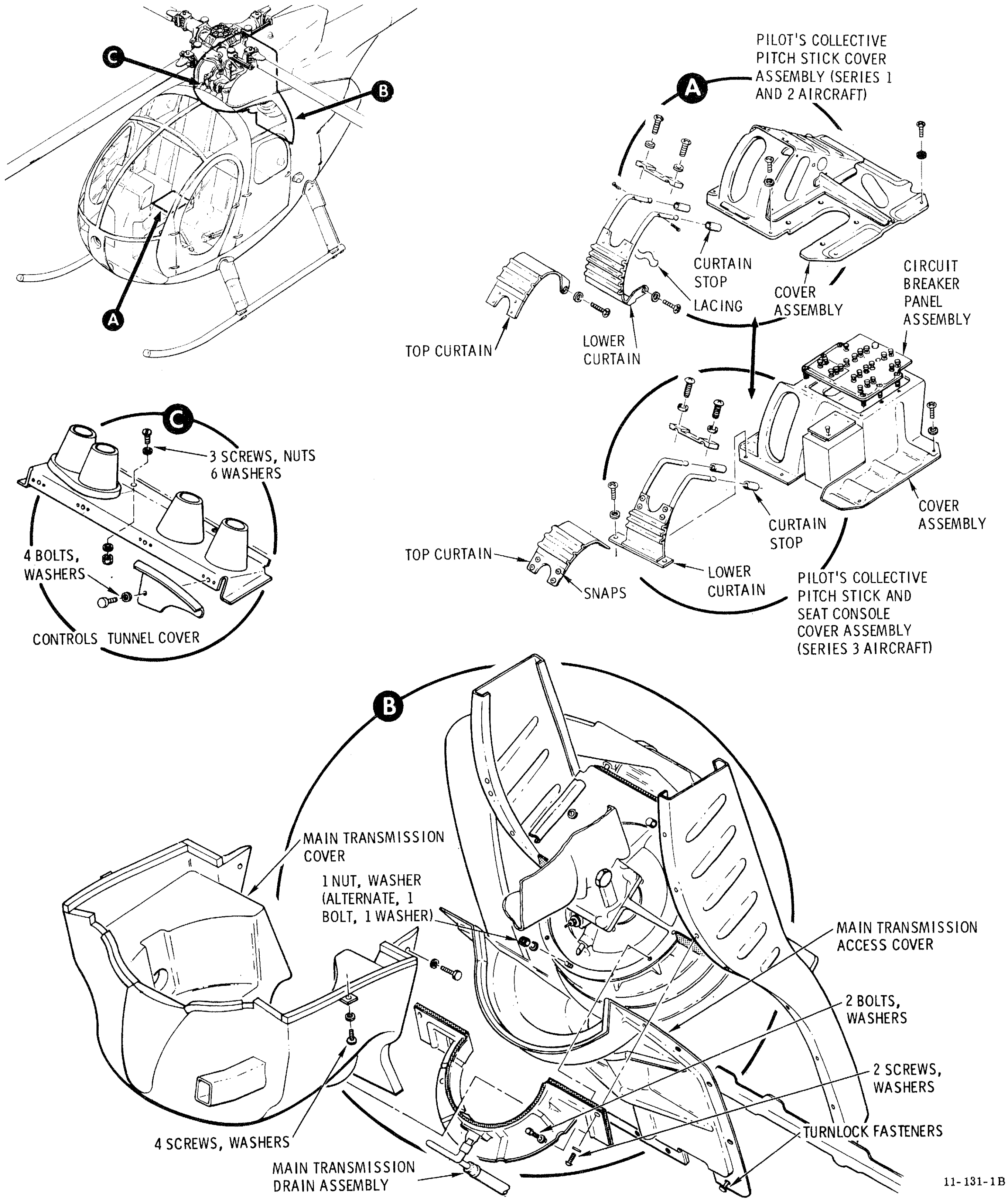


Figure 2-1. Access and Inspection Provisions. (sheet 1 of 4)

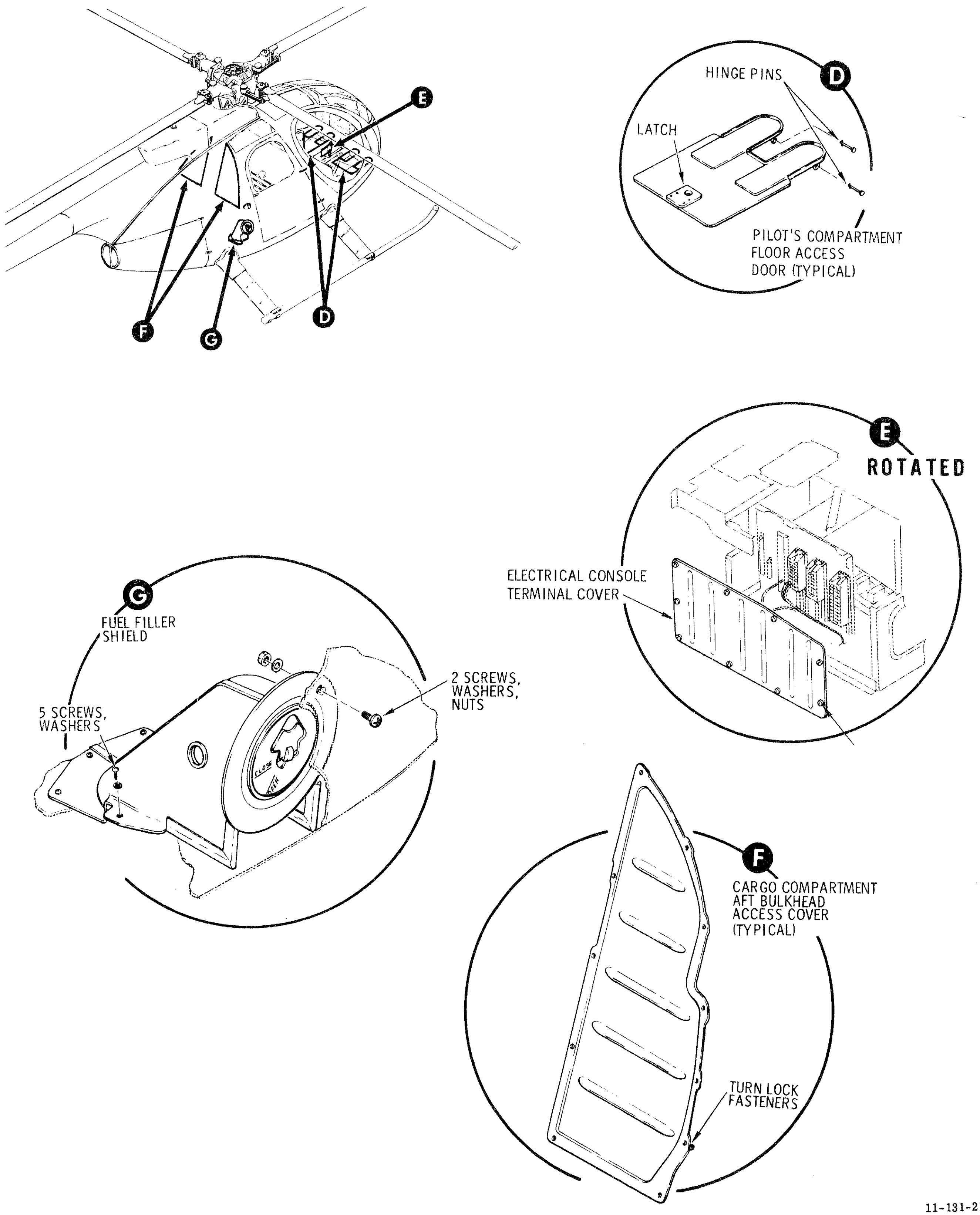


Figure 2-1. Access and Inspection Provisions. (sheet 2 of 4)

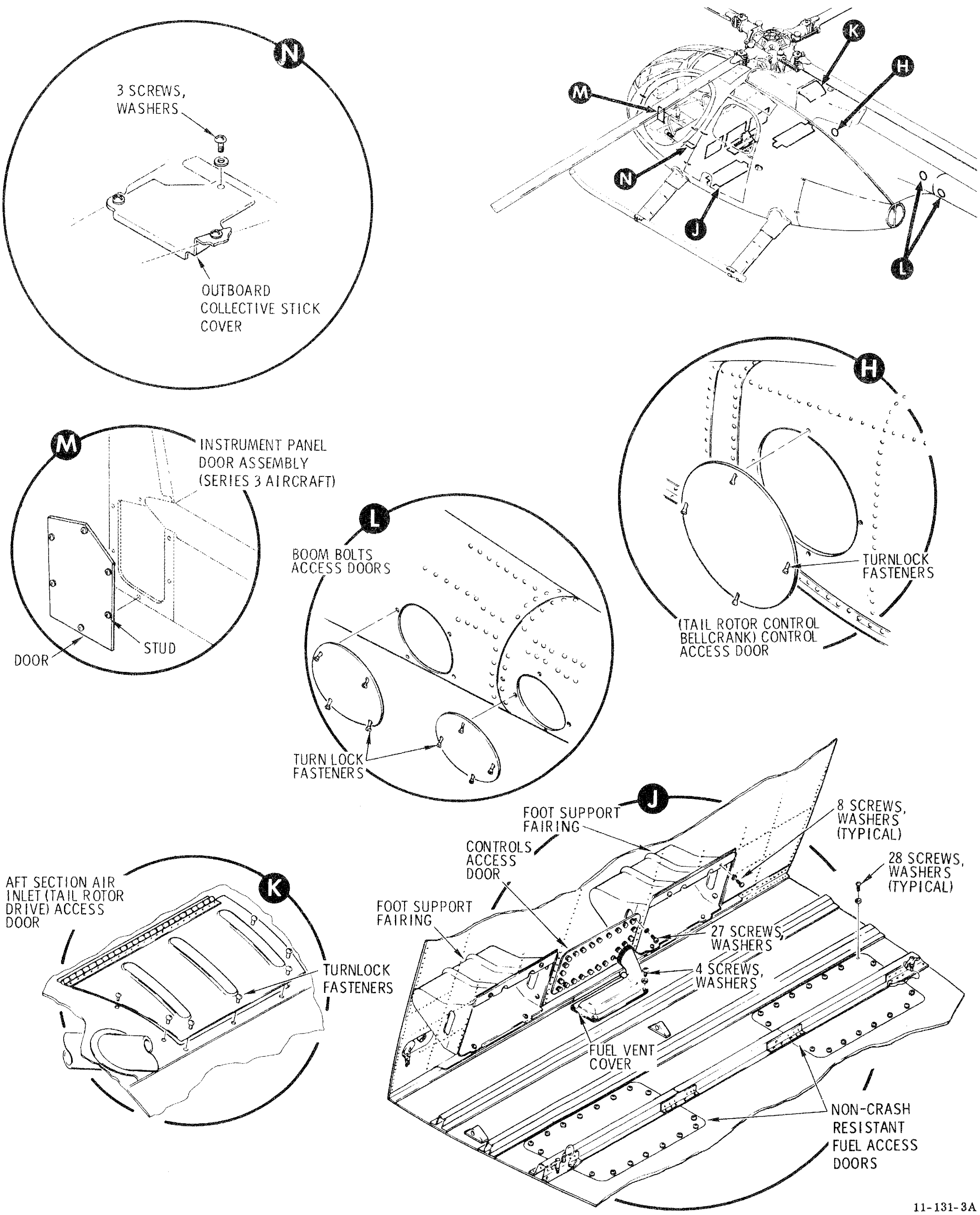


Figure 2-1. Access and Inspection Provisions. (sheet 3 of 4)

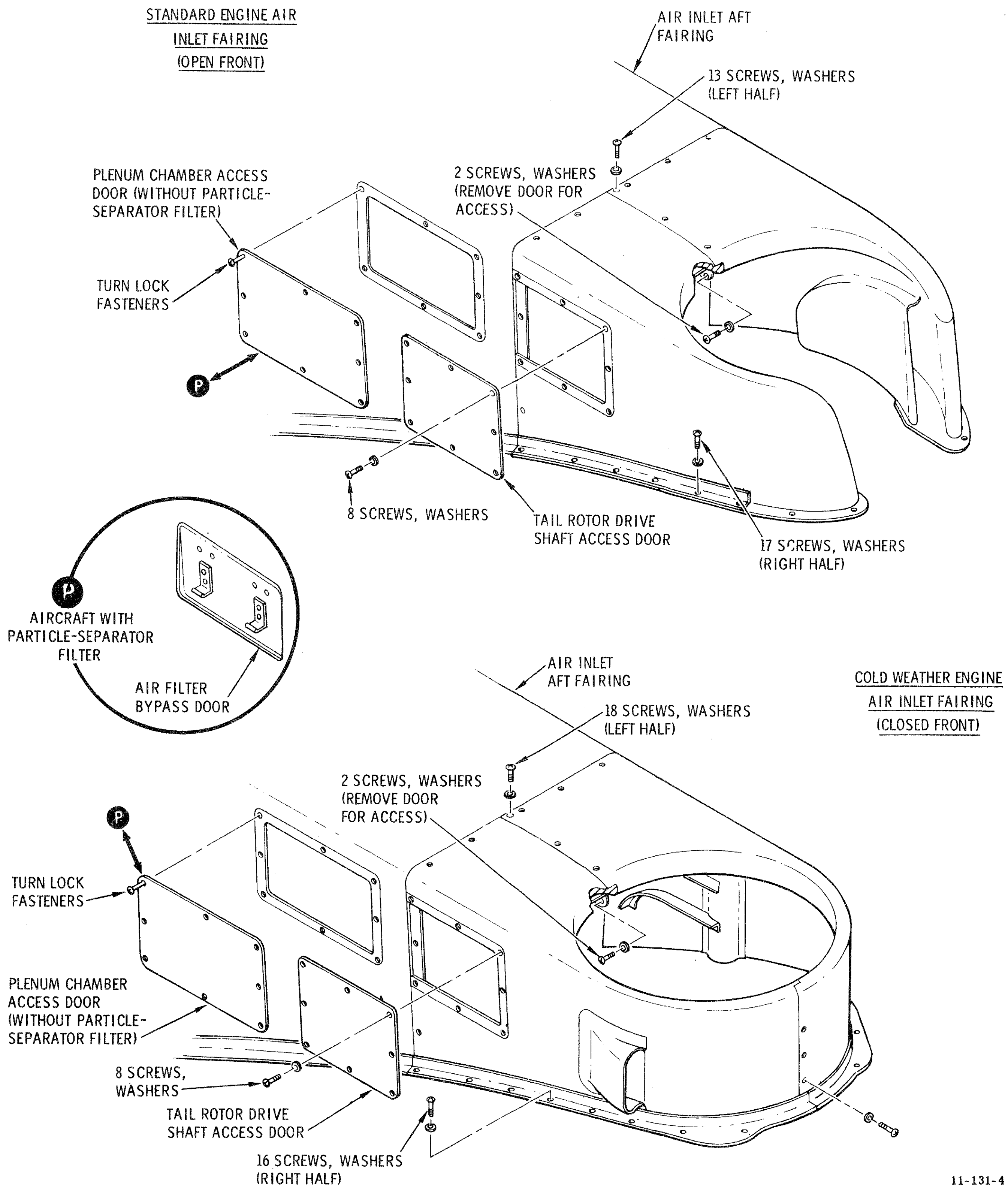


Figure 2-1. Access and Inspection Provisions. (sheet 4 of 4)

a. A total of 10 percent of parts with less than 40 square inches.

b. A total of 15 percent of parts with less than 400 square inches and more than 40 square inches.

c. A total of 25 percent of parts with more than 400 square inches.

2-10. Replacement — Non-Stressed Sheet Metal. Damage that exceeds the limits for repair by patch or insertion requires replacement of the part.

2-11. MAIN TRANSMISSION ACCESS COVER.

2-12. Description — Main Transmission Access Cover. The main transmission access cover (fig. 2-1, sh 1) provides access to the main transmission, oil cooler blower and engine-to-transmission drive shaft. A silicone rubber-sponge gasket along the upper edge of the fiberglass cover provides a water and dust seal. Turnlock fasteners secure the cover to the structure. The cover must be removed before the main transmission drain assembly or the transmission cover can be removed.

2-13. Removal — Main Transmission Access Cover. *a.* Remove insulation (para 2-175).

b. Disengage turnlock fasteners and remove cover.

2-14. Inspection — Main Transmission Access Cover. *a.* Inspect cover for cracks or other visible damage.

b. Inspect turnlock fasteners and receptacles for proper fastening action.

c. Inspect cover gasket for deterioration or missing sections.

2-15. Repair — Main Transmission Access Cover. *a.* Refer to TM 55-1500-204-25/1 for replacement of turnlock fasteners, and patching of plastic sections.

b. Replace silicone rubber gaskets according to paragraph 2-171.

NOTE

Before installing replacement turnlock fastener studs and washers, ensure that the aluminum reinforcing strip is securely bonded to the external side of the access door at the stud hole.

c. Seam separations or small areas of rip damage may be repaired by cleaning the damaged area with naptha (C70), injecting dichloromethane (C38) into the void area and clamping together under light pressure.

d. Replace the defective turnlock fastener reinforcing strips according to steps *e* through *g* below.

e. Ensure mating surfaces of reinforcing strip and access door are fitted to make full surface contact.

f. Lightly abrade mounting surface on access door with grade 180 abrasive paper (C1).

g. Install replacement reinforcing strips (para 2-173).

h. Repair cracks in polycarbonate material as follows:

(1) Stop drill at each end of crack using a number 40 drill.

(2) Lightly sand-area to be repaired with sandpaper (C3) about 0.750 inch beyond crack for adhesive application.

(3) Wipe area with dry, clean cloth to remove residue.

(4) Cut section of fiberglass (39A) to extend a minimum of 0.500 inch around crack or area to be repaired.

(5) Apply one brush coat of urelane adhesive (C7) to repair area and lay fiberglass over repair area.

(6) Rub patch lightly to assure adhesion to repair area and brush another coat of adhesive over area.

(7) Allow to cure for 48 hours. (Handling strength is developed in 24 hours). Accelerated cure time will be two hours under a heat lamp at 71°C (160°F).

2-16. Installation — Main Transmission Access Cover. (With main transmission cover and drain assembly already installed.) *a.* Position access cover and secure by engaging turnlock fasteners.

b. Reinstall insulation (para 2-175).

2-17. MAIN TRANSMISSION DRAIN ASSEMBLY.

2-18. Description — Main Transmission Drain Assembly. The main transmission drain assembly (fig. 2-1, sh 1) is a clear polycarbonate plastic yoke-type collector that provides an overboard drain for water or oil entering the main transmission cover, and for any oil seepage from the transmission input pinion oil seal. Foam gaskets cushion and seal the mounting edges. The drain assembly must be removed before the main transmission cover can be removed.

2-19. Removal — Main Transmission Drain Assembly. (With main transmission access cover removed; para 2-8.) *a.* Remove the two screws, two bolts and washers securing the drain assembly to the structure, transmission shroud mount and blower scroll.

b. Remove the bolt and washer (or the one nut and washer from the scroll stud) under the drain outlet tube. Lift drain assembly from shroud mount.

c. Unless replacing drain assembly, leave drain hose lockwired and attached.

2-20. Inspection — Main Transmission Drain Assembly. Inspect drain assembly for cracks, gaskets for deterioration, and outlet tube for internal obstruction.

2-21. Repair — Main Transmission Drain Assembly. Refer to paragraph 2-15.

2-22. Installation — Main Transmission Drain Assembly. (With main transmission cover already installed.) *a.* Position drain assembly on the transmission shroud. Check that the short plastic drain tube from the transmission input pinion seal drain port enters the drain assembly outlet tube port.

b. Install the two screws, three bolts (or two bolts and nut) and washers that secure the drain assembly to the structure, shroud and scroll (fig. 2-1, sh 1). **DO NOT OVERTIGHTEN THE BOLTS AND/OR NUT (10 INCH-POUNDS MAXIMUM).**

c. If drain assembly is a replacement, connect the flexible drain hose and secure it with two wraps of lockwire.

2-23. MAIN TRANSMISSION COVER.

2-24. Description — Main Transmission Cover. The main transmission cover (fig. 2-1, sh 1) is a polycarbonate plastic form that essentially matches the transmission housing contour. The plastic form has a permanently bonded insulation blanket cover with a fiberglass core and flexible vinyl exterior. Plastic foam gaskets cushion the cover surfaces that mate with the adjacent structure. When installed, there is space between the cover and transmission to allow inlet air flow for transmission cooling. A yoke-type drain assembly fits around the lower end of the cover. A flexible hose connects to the drain assembly outlet tube and pipes overboard any water or oil that collects in the cover.

2-25. Removal — Main Transmission Cover. *a.* Detach the main transmission drain assembly (para 2-17).

b. Release nylon fastener tape hook from mating nylon fastener tape pile at aft edges of cover.

c. Remove four screws and washers from cover.

d. Remove bolt from heater duct flange at lower left aft inside corner of cover if heating system ducting is installed; then lower the cover to remove it.

2-26. Inspection — Main Trainsmission Cover. Inspect foam gaskets and cover blanket for deterioration, and plastic cover for cracks.

2-27. Repair — Main Transmission Cover. Refer to paragraph 2-15.

2-28. Installation — Main Transmission Cover. *a.* Position cover over transmission.

b. Install heater duct bolt at lower left aft inside corner of cover if heating system ducting is installed.

c. Install the four screws and washers fingertight. Check cover for proper fit and that liquid level plug is visible.

d. Tighten screws.

e. Set the nylon fastener tape hook to the mating fastener tape pile by using hand pressure.

f. Attach the main transmission drain assembly (para 2-17).

2-29. CARGO COMPARTMENT AFT BULKHEAD ACCESS COVERS.

2-30. Description — Cargo Compartment Aft Bulkhead Access Covers. The cargo compartment aft bulkhead access covers (fig. 2-1, sh 2) enclose openings to the fuselage spaces at either side of the engine air inlet plenum chamber. The right side cover provides access to the oil tank, oil cooler and oil system drain valve. The left side cover provides access to elements of the cabin heating installation. Turnlock fasteners secure the outer edge of each cover to the structure.

2-31. Removal — Cargo Compartment Aft Bulkhead Access Covers. *a.* Remove insulation (para 2-175).

b. Release turnlock fasteners and lift cover from structure.

2-32. Inspection — Cargo compartment Aft Bulkhead Access Covers. *a.* Inspect turnlock fasteners and receptacles for condition.

b. Inspect cover for corrosion and cracks.

2-33. Repair — Cargo Compartment Aft Bulkhead Access Covers. Refer to paragraph 2-7.

2-34. Installation — Cargo Compartment Aft Bulkhead Access Covers. *a.* Position the cover over the opening in the structure and engage the turnlock fasteners.

b. Reinstall insulation (para 2-175).

2-35. CARGO COMPARTMENT FOOT SUPPORT FAIRINGS.

2-36. Description — Cargo Compartment Foot Support Fairings. The two foot support fairings (fig. 2-1, sh 3) are fiberglass assemblies, each having two small polycarbonate plastic windows. The outboard windows provide a means for inspection of the forward landing gear damper assemblies without removal of the fairings. The inboard windows (as installed) are not for inspection but permit left and right side interchangeability of the fairings. Removal of the fairings permits access to the forward landing gear struts and dampers, lower elements of the underseat installation of the engine and flight control systems, and portions of the electrical wiring routed beneath the pilot's seat structure.

2-37. Removal — Cargo Compartment Foot Support Fairings. Remove eight screws and washers to release each foot support fairing.

2-38. Inspection — Cargo Compartment Foot Support Fairings. Inspect for cracks, breaks, and separation of the plastic windows.

2-39. Repair — Cargo Compartment Foot Support Fairings. Refer to paragraph 2-244.

2-40. Installation — Cargo Compartment Foot Support Fairings. Position each support fairing and secure with screws and washers.

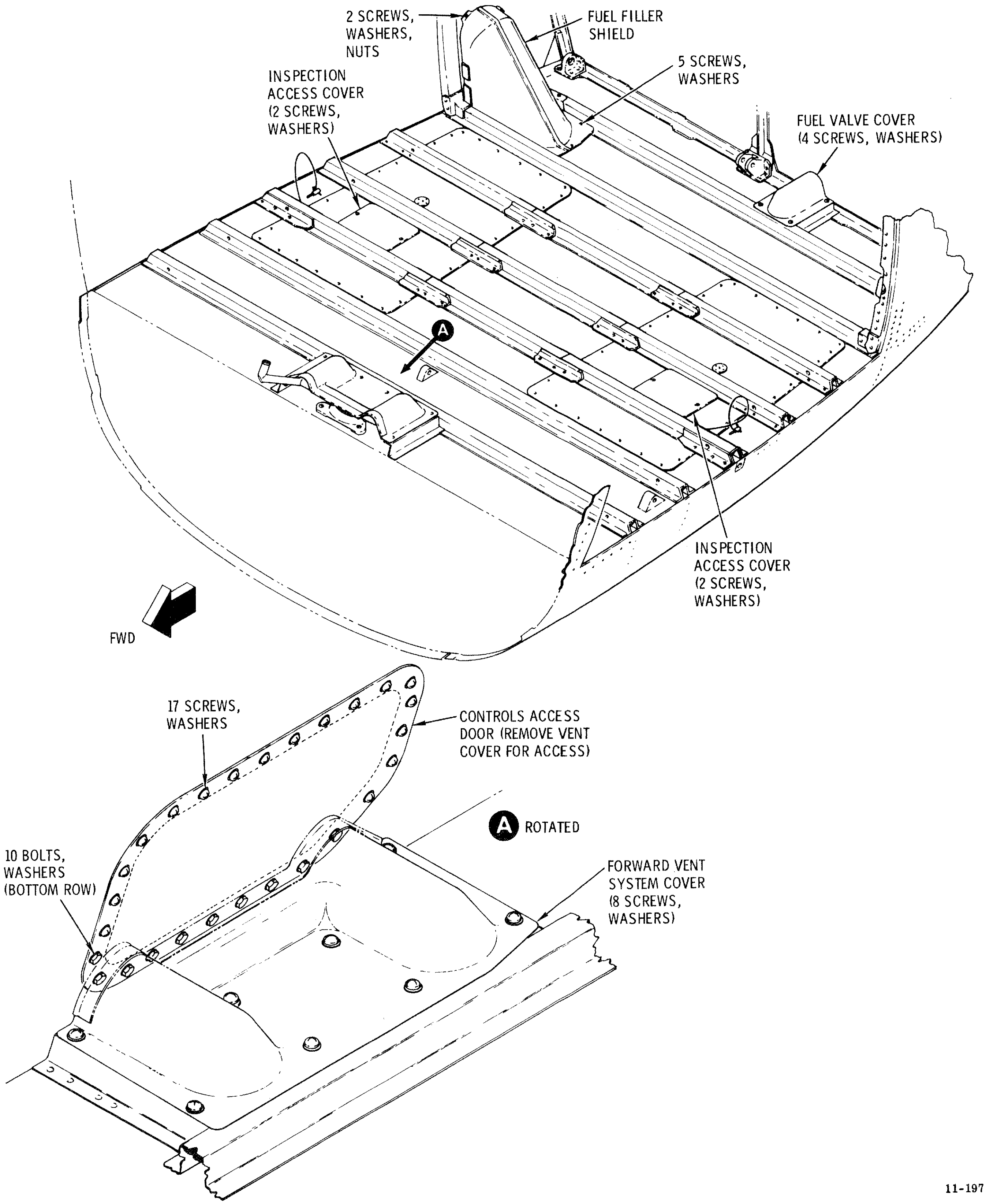


Figure 2-2. Access and Inspection Provisions (Crash-resistant Fuel System).

2-41. FUEL FILLER SHIELD.

2-42. Description — Fuel Filler Shield. The polycarbonate plastic fuel filler shield (fig. 2-1, sh 2 or fig. 2-2) protects the right fuel cell filler extension against possible damage from cargo impact, and also covers the cargo floor opening for the filler.

2-43. Removal — Fuel Filler Shield. *a.* Remove five screws and washers that secure the fuel filler shield base to the cargo floor.

b. Remove the two nuts, screws and washers that secure the shield tabs to the fuselage skin and fuel filler cap to release the shield.

NOTE

Use care not to dislodge or damage the sealing grommet (if installed) that fits around the LONG-RANGE TANK CONN fitting opening in the shield.

2-44. Inspection — Fuel Filler Shield. Inspect shield for cracks and breaks and condition of grommet.

2-45. Repair — Fuel Filler Shield. Refer to paragraph 2-15.

2-46. Installation — Fuel Filler Shield. *a.* Position the fuel filler shield and install the two screws, washers and nuts that secure the shield tabs to the fuselage skin and fuel filler cap; do not tighten.

b. Install the five screws and washers that secure the shield base and tighten screws.

c. Tighten nuts that secure shield tabs.

d. Check that the LONG-RANGE TANK CONN fitting grommet (if installed) provides a tight seal between the shield and fitting. Replace grommet if deteriorated.

2-47. FUEL VENT COVER.

2-48. Description — Fuel Vent Cover. The formed aluminum fuel vent cover (fig. 2-1, sh 3, or fig. 2-2) provides access to the fuel cell vent system crossover fitting that interconnects the two cells.

2-49. Removal — Fuel Vent Cover. Remove either four or six screws and washers to release the fuel vent cover.

2-50. Inspection — Fuel Vent Cover. Check for cracks and other visible damage.

2-51. Repair — Fuel Vent Cover. Refer to paragraph 2-7.

2-52. Installation — Fuel Vent Cover. Check that area to be covered is clean. Position cover and secure with screws and washers.

2-53. FUEL VALVE COVER (CRASH-RESISTANT FUEL SYSTEM).

2-54. Description — Fuel Valve Cover (Crash-Resistant Fuel System). The fuel valve cover (fig. 2-2) protects the fuel outlet valve used with crash resistant fuel systems. The aluminum cover is located on the cargo compartment floor at the left rear bulkhead. Removal of the cover provides access to the fuel outlet valve and allows for inspection of the left fuel cell aft vent fitting.

2-55. Removal — Fuel Valve Cover (Crash-Resistant Fuel System). Remove four screws and washers to release cover.

2-56. Inspection — Fuel Valve Cover (Crash-Resistant Fuel System). Check for cracks and other visible damage.

2-57. Repair — Fuel Valve Cover (Crash-Resistant Fuel System). Refer to paragraph 2-7.

2-58. Installation — Fuel Valve Cover (Crash-Resistant Fuel System). Position cover and secure with four screws and washers.

2-59. PILOT'S COMPARTMENT FLOOR ACCESS DOORS.

2-60. General — Pilot's Compartment Floor Access Doors. The pilot's compartment floor access doors (fig. 2-1, sh 2) are fiberglass with stainless steel heel strips bonded in place. A flush-mounted latch and hinges secure each door in place.

2-61. Removal — Pilot's Compartment Floor Access Doors. Remove either floor access door by releasing the flush-mounted latch and removing the two hinge pins that secure the door hinges at the forward end.

2-62. Inspection — Pilot's Compartment Floor Access Doors. *a.* Inspect doors for cracks and other visible damage.

b. Check security of heel strip bonding.

2-63. Repair — Pilot's Compartment Floor Access Doors. Refer to paragraph 2-7.

2-64. Installation — Pilot's Compartment Floor Access Doors. Position door and install hinge pins through hinges and antitorque pedal mounting bracket.

2-65. PILOT'S COLLECTIVE PITCH STICK COVER.

2-66. Description — Pilot's Collective Pitch Stick Cover. The pilot's collective pitch stick cover (fig. 2-1, sh 1) provides primary access to the upper elements of the underseat installation of engine and flight controls systems. The cover supports the circuit breaker assembly and forms a protective guard for the collective friction mechanism on the collective pitch stick. A flexible, two-part curtain enclosure on the cover is cord-laced or snap-fastened around the stick to prevent foreign matter from entering the underseat controls area. The stick cover is made of formed and spot-welded sheet aluminum and attached to the seat structure with five screws.

2-67. Removal — Pilot's Collective Pitch Stick Cover. *a.* Set the power selector switch at OFF.

b. Unfasten the circuit breaker assembly and tie it out of the way.

c. Untie or unsnap upper curtain from lower curtain and slide curtains away from the stick (fig. 2-1, sh1).

d. Remove the five cover attaching screws and washers to release the cover from the seat structure.

e. If required, remove the flexible closure from the cover according to *f* and *g* below.

f. Remove support tube clamp and four screws and washers that attach closure to cover; remove closure from cover.

g. Remove top curtain, then lower curtain, from curtain support tubes.

2-68. Inspection — Pilot's Collective Pitch Stick Cover. *a.* Inspect cover for obvious damage.

b. Inspect the flexible closure for cuts, holes, deterioration, defective cord or snap fasteners, and freedom of movement on curtain support tubes.

2-69. Repair — Pilot's Collective Pitch Stick Cover. Refer to paragraph 2-7.

2-70. Installation — Pilot's Collective Pitch Stick Cover. *a.* Set the power selector switch at OFF.

b. Position the cover over the collective pitch stick. Locate circuit breaker assembly wiring in cutout and secure cover to structure with five screws and washers. The longer screw is used at the front (fig. 2-1, sh 1).

c. If removed, reinstall the flexible cover according to *d* through *f* below.

d. Place lower curtain, then top curtain, on curtain support tubes.

e. Install curtain stops, cotter pins, lower curtain and then upper curtain on curtain support tubes.

f. Place enclosure on cover; secure enclosure to cover with support tube clamp and four screws and washers. Bottom edge of lower curtain must be tucked under support.

CAUTION

Closure lacing cord must be tight so that curtains will not fold inward through full travel of the collective stick. Folding inward can result in entanglement with the collective stick friction gear mechanism.

g. Close upper and lower curtains around pitch stick and connect the curtains together with the cord lacing or snap fasteners. Lacing (1/8 by 4-inch braided nylon) cords must be tied in a square knot.

h. Position and fasten circuit breaker assembly.

2-71. OUTBOARD COLLECTIVE STICK COVER.

2-72. Description — Outboard Collective Stick Cover. The small cover installed at the left side of the seat structure (fig.2-1, sh 3) keeps the left side seat belt from fouling the aft end of the collective stick, and shields the electrical wiring where it connects to the auxiliary circuit receptacles mounted in the left corner of the bulkhead.

2-73. Removal — Outboard Collective Stick Cover. Remove three screws and washers to release cover.

2-74. Inspection — Outboard Collective Stick Cover. Check for cracks and other visible damage.

2-75. Repair — Outboard Collective Stick Cover. Refer to paragraph 2-7.

2-76. Installation — Outboard Collective Stick Cover. Check that there are no foreign objects in the area shielded by the cover, position cover and secure with screws and washers.

2-77. ELECTRICAL CONSOLE TERMINAL BLOCK ACCESS COVER.

2-78. General — Electrical Console Terminal Block Access Cover. The terminal block access cover (fig. 2-1, sh 2) provides access to the electrical console terminal blocks for inspection and repair.

2-79. Removal — Electrical Console Terminal Block Access Cover. Release turnlock fasteners and lift access covers from console.

2-80. Inspection — Electrical Console Terminal Block Access Cover. Inspect turnlock fasteners and receptacles for proper fastening action.

2-81. Repair — Electrical Console Terminal Block Access Cover. Refer to paragraph 2-7.

2-82. Installation — Electrical Console Terminal Block Access Cover. Position access cover over receptacles and engage turnlock fasteners.

2-83. CONTROLS TUNNEL COVER.

2-84. Description — Controls Tunnel Cover. Four tunnel-routed control rods exit the tunnel area through a cover mounted at the top of the station 78.50 canted frame. The cover or cover boots must be removed before any of the control rods are removed. The cover has four naugahyde boots, two of which are sewed together while the other two are individual. All boots are secured to the cover with self-clinching nylon straps. (See fig. 2-1, sh 1.)

2-85. Removal — Controls Tunnel Cover. The right and left engine air inlet forward fairings (fig. 2-1, sh 4) should be removed (para 2-101) for best access to the controls tunnel cover.

a. Remove the three screws, nuts and six washers, and the four bolts and washers from controls cover.

b. Remove the cotter pin, nut, bolt and washers that secure the upper end of each tunnel-routed control rod. Disengage the rod ends from the bellcranks.

c. Lift controls cover over rod ends.

2-86. Inspection — Controls Tunnel Cover. *a.* Check boots for control binding.

b. Check for tears, breaks, and damaged or missing straps.

c. Inspect for damaged or loose boot stitching.

2-87. Repair — Controls Tunnel Cover. *a.* Repair all loose or damaged stitching on boots by hand-stitching with nylon cord (C34).

b. Remove damaged boots by loosening nylon strap. Replace damaged or defective straps.

c. Install naugahyde boots over flared end of tubular mounts on cover shelf. Secure with straps.

2-88. Installation — Controls Tunnel Cover. *a.* Place controls tunnel cover over the four control rods, with double boot section fitting over the two right-hand control rods.

b. Attach each control rod end to its bellcrank with bolt (head to left), two washers, nut and cotter pin.

c. Install controls tunnel cover with the three screws, nuts and six washers aft, and the four bolts and washers forward. Tighten bolts and screws evenly.

d. If removed, reinstall right and left engine inlet fairings (para 2-101).

e. Check that control rods move freely in the cover boots.

2-89. FAIRING ACCESS DOORS AND AFT SECTION AIR INLET ACCESS DOOR.

2-90. Description — Fairing Access Doors and Aft Section Air Inlet Access Door. Two removable doors are installed on the right side of the air inlet front and aft fairings (fig. 2-1, sh 4). On aircraft equipped with an inertial particle-separator air filter, the aft fairing contains a filter bypass door (fig.2-1, sh 4). The aft section air inlet (tail rotor drive) access door (fig. 2-1, sh 4) is hinge-mounted to the structure and is constructed of aluminum. The door provides access to the front end of the tail rotor drive shaft, and limited access to the accessories mounted on the aft end of the main transmission. On aircraft equipped with a barrier filter, the tail rotor drive shaft access door is accessible when the filter element is removed and the hinged filter bypass door and frame assembly is moved forward. The plenum chamber access and tail rotor drive shaft doors installed on fairings of series 1 and 2 aircraft are made of fiberglass and reinforced with polyurethane-foam-filled stiffeners. The filter bypass door installed on the aft fairing of series 3 aircraft is hinge-mounted and latched to the aft fairing, and is opened with the BYPASS AIR CONTROL release handle located overhead in the flight compartment.

2-91. Removal — Fairing Access Doors and Aft Section Air Inlet Access Door. *a.* Release turnlock fasteners and lift plenum chamber access (aft) door from fairing.

b. Remove the eight screws and washers and lift tail rotor drive access (forward) door from fairing.

c. Release turnlock fasteners and raise hinged aft section air inlet door for access. On aircraft equipped with an engine barrier filter, remove filter element (chapter 4) and fold filter bypass door and frame portion of filter assembly forward to expose access door.

d. On series 3 aircraft, open filter bypass door on aft fairing by pulling BYPASS AIR CONTROL release handle forward.

2-92. Inspection — Fairing Access Doors and Aft Section Air Inlet Access Door. *a.* Inspect turnlock fasteners and receptacles for proper fastening action.

b. Inspect fiberglass doors for structural damage such as cracked or frayed glass cloth surfaces.

c. Inspect interior door for proper fit when closed, and hinge halves for security.

2-93. Repair — Fairing Access Doors and Aft Section Air Inless Access Door. Refer to paragraph 2-7 for repair of aluminum doors and paragraph 2-244 for repair of fiberglass doors.

2-94. Installation — Fairing Access Doors and Aft Section Air Inlet Access Door.

CAUTION

Check that all areas of the air inlet are clean, that all debris is removed and that all protective covers are removed. Engine damage could result if these precautions are not always taken.

a. Close hinged aft section air inlet access door and engage the turnlock fasteners. On aircraft equipped with an engine barrier filter, fold filter bypass door and frame into place and install filter element.

b. Position plenum chamber access door and engage turnlock fasteners.

c. Position tail rotor drive shaft access door and secure with screws and washers.

d. Secure particle separator air filter bypass door by using door pulls to close and latch door on aft fairing.

2-95. BOOM BOLTS ACCESS DOORS AND TAIL ROTOR CONTROL BELLCRANK ACCESS DOOR.

2-96. Description — Boom Bolts Access Doors and Tail Rotor Control Bellcrank Access Door. The forward and aft boom bolts access door (fig. 2-1, sh 3) provide access to the bolts that secure the tailboom to the fuselage aft section. These doors are also removed for access to perform removal and installation of the tail rotor drive shaft and/or tailboom (tail rotor blade angle) control rod, and to check drive shaft damper friction or replace the damper. The tail rotor control bellcrank access door provides primary access to the station 142 bellcrank link between the station 100 tail rotor control rod and the tailboom control rod.

2-97. Removal — Boom Bolts Access Doors and Tail Rotor Control Bellcrank Access Door. Release turnlock fasteners and lift boom bolts or controls access door from fuselage.

2-98. Inspection — Boom Bolts Access Doors and Tail Rotor Control Bellcrank Access Door. Inspect turnlock fasteners and receptacles for proper fastening action.

2-99. Repair — Boom Bolts Access Doors and Tail Rotor Control Bellcrank Access Door. Refer to paragraph 2-7.

2-100. Installation — Boom Bolts Access Doors and Tail Rotor Control Bellcrank Access Door. Position boom bolts or controls access door and engage turnlock fasteners.

2-101. ENGINE AIR INLET FRONT FAIRINGS.

2-102. Description — Engine Air Inlet Front Fairings. Ambient air enters the engine inlet and the engine oil cooler through the removable engine air inlet front fairings (fig. 2-1, sh 4). The fairings are in two halves constructed of fiberglass. The fairings provide access to the main rotor mixer controls, the main rotor mast, its base and supporting structure, and the engine air filter (when installed). Either one of two fairing types may be installed on the aircraft: the standard open-front fairing; or the closed-front fairing, as part of the snow ingestion/cold weather kit.

2-103. Removal — Engine Air Inlet Front Fairings. Either half of the fairing may be independently removed. For the standard open-front fairings, a total of 33 screws with washers secure the fairing halves together, and to the structure. For the closed-front fairings, a total of 36 screws with washers secure the fairing halves together, and to the structure. Both the open-front and the closed-front fairings have an access door in the right half, secured by eight screws and washers (three of which secure the front fairing to the aft fairing).

a. Remove the eight screws and washers from the access door, and remove the door.

b. Remove the screws and washers that join the two fairing halves together.

c. Remove the screws and washers that attach the fairing halves (each, or both, as necessary) to the fuselage and aft fairing. Remove front fairing halves.

NOTE

Either the standard open-front or the closed front fairings have been fitted to the particular aircraft as a matched set and should be so identified.

2-104. Inspection — Engine Air Inlet Front Fairings. Inspect the engine air inlet fairings for structural damage.

2-105. Repair — Engine Air Inlet Front Fairings. Refer to paragraph 2-7 for sheet metal repairs and to paragraph 2-224 for repair of fiberglass.

2-106. Installation — Engine Air Inlet Front Fairings. *a.* Position the engine air inlet fairings on the fuselage and align attachment holes on fuselage and aft fairing.

b. Install attaching washers and screws in mounting flanges and at fairing mating attachments.

2-107. ACCESS DOORS AND COVERS — STRESSED (GENERAL).

2-108. Description — Access Doors and Covers — Stressed (General). The fuel cell access doors and the controls access door (fig. 2-1, sh 3 or fig. 2-2) are stressed sheet metal. Removal, inspection, repair and installation of these doors are covered in paragraphs 2-109 through 2-122 below.

2-109. FUEL CELL ACCESS DOORS (NON-CRASH RESISTANT FUEL SYSTEM ONLY).

2-110. Description — Fuel Cell Access Doors (Non-Crash Resistant Fuel System Only). The fuel cell access doors (fig. 2-1, sh 3) are stiffener-reinforced aluminum plates that form a portion of the cargo floor. The left access door provides access to the fuel quantity transmitter (tank unit) and fuel shutoff valve, and the fuel cell cover (for access to the engine starting pump) as well as the left fuel cell. A quick-release lock pin is secured with a four-inch lanyard to the outboard edge of each door. The pins retain the removable jacking fittings that are used for jacking, parking, and mooring the aircraft.

CAUTION

These are stressed doors. The aircraft must never be flown jacked or towed with either door removed.

2-111. Removal — Fuel Cell Access Doors (Non-Crash Resistant Fuel System Only). Remove the 28 retaining screws and washers and lift out door.

2-112. Inspection — Fuel Cell Access Doors (Non-Crash Resistant Fuel System Only). Refer to paragraph 2-264.

2-113. Repair — Fuel Cell Access Doors (Non-Crash Resistant Fuel System Only). Refer to paragraph 2-264 for general repairs. Replace a defective or broken lanyard as follows:

a. Disengage lanyard from ring or lockpin.

b. Remove rivet that secures lanyard to door stiffener.

c. Rivet replacement lanyard to clip and reinforcement of plate.

d. Attach ring of lockpin to replacement lanyard.

2-114. Installation — Fuel Cell Access Doors (Non-Crash Resistant Fuel System Only). *a.* Position door over opening and secure in place with retaining screws and washers.

b. Stow quick-release lock pin in its hole.

2-115. FUEL CELL ACCESS DOORS (CRASH-RESISTANT FUEL SYSTEM ONLY).

2-116. Description — Fuel Cell Access Doors (Crash-Resistant Fuel System Only). The CR fuel cell access doors (fig. 2-2) are essentially the same as the NCR doors described in paragraph 2-109 except that the doors are larger. In addition, a small covered inspection/access hole is provided. Because the CR fuel cells are attached to the doors, door removal and installation is accomplished according to access door removal procedures in chapter 10. Refer to paragraph 2-113 for door repair.

2-117. FUEL CELLS INSPECTION ACCESS COVER (CRASH-RESISTANT FUEL SYSTEM ONLY).

2-118. Description — Fuel Cells Inspection Access Cover (Crash-Resistant Fuel System Only). The CR inspection access cover (fig. 2-2) is a square aluminum plate located near the center of each fuel cell access door used with crash-resistant fuel systems. The right cover is removed for visual inspection of the right fuel cell upper surfaces. The left cover is removed for inspection of the left cell cover area and for adjustment of fuel shutoff control rigging.

2-119. Removal — Fuel Cells Inspection Access Cover (Crash-Resistant Fuel System Only). Remove two screws and washers and remove cover.

2-120. Inspection — Fuel Cells Inspection Access Cover (Crash-Resistant Fuel System Only). Refer to paragraph 2-7 for inspection and repair.

2-121. Installation — Fuel Cells Inspection Access Cover (Crash-Resistant Fuel System Only). Position cover over round inspection hole and attach with two screws and washers.

2-122. CONTROLS ACCESS DOOR.

2-123. Description — Controls Access Door. The controls access door (fig. 2-1 or 2-2) is an aluminum plate that provides primary access to the lower end of the tunnel-routed flight control push rods as well as other elements of the control system.

CAUTION

The controls access door is a stressed door. The aircraft must never be flown, jacked or towed with this door removed.

2-124. Removal — Controls Access Door. *a.* On aircraft with non-crash-resistant fuel system (fig. 2-1, sh 3) remove 27 screws and washers to release cover.

b. On aircraft with crash-resistant fuel system (fig. 2-2) remove forward vent system cover (para 2-47); then remove 17 upper screws, 10 lower bolts and washers to release cover.

2-125. Inspection — Controls Access Door. Refer to paragraph 2-264.

2-126. Repair — Controls Access Door. Refer to paragraph 2-264.

2-127. Installation — Controls Access Door. *a.* On aircraft with crash-resistant fuel system (fig. 2-2) position door (bevel at lower right corner) and install 10 lower bolts and washers and 17 upper screws and washers as shown in figure 2-2; then reinstall forward vent system cover (para 2-47).

b. On aircraft with non-crash-resistant fuel system (fig. 2-1, sh 3), position door (bevel at lower right corner) and install 27 screws and washers.

2-128. ENGINE ACCESS DOORS.

2-129. Description — Engine Access Doors. The engine access doors are stamped and bonded aluminum alloy structures that form the fuselage contour below the aft section engine compartment. Figure 2-3 shows series 1 and 2 aircraft access doors having original hinge and latching configuration. Figure 2-4 shows access doors with adjustable door hinges and latches that are installed on all series 3 aircraft and on series 1 and 2 aircraft that have been modified. Figure 2-5 shows engine access doors used on aircraft with the upward exhaust system. These doors are closed at the aft end with a cone shaped fairing, one-half of which is attached to each door. A fourth latch is added at the aft end of the cone. Vertical adjustment can be made to the doors by positioning of washers at the hinge points. Fore and aft adjustment of the doors is made possible by slotted holes in the hinges and serrated mating surfaces between the hinges and doors. Lateral adjustment is obtainable by installing laminated shims under the door hinges. Door alignment is provided by V-type striker blocks riveted to the door frames next to the lower latch.

2-130. Inspection — Engine Access Doors. *a.* Check hinges and latch hardware for looseness, cracks or damage.

b. Check the outside edge of door for security of bond between outer and inner skin. Any separation requires repair.

c. Inspect the door for corrosion, distortion, breaks or cracks, and condition of abrasion strip tape along upper inside edge of door.

d. On aircraft with upward exhausts, inspect tail cone seals.

2-131. Latch Adjustment — Engine Access Doors. On series 3 aircraft and series 1 and 2 aircraft with the modified engine access doors, adjust the latches as follows: (See fig. 2-4.)

a. Remove lockwire from threaded (adjustment) end of hooks.

b. Loosen all hooks until striker blocks will engage in light contact with no door deflection when the hooks are latched.

c. Unlatch the doors and tighten all hooks four to five turns. Check door latching and make additional minor adjustments as required.

d. Install 0.032-inch lockwire (C57) in hook threaded end in a manner to permit latching and unlatching without interference.

2-132. Door Position Adjustment (Series 3 and Modified Series 1 and 2 Aircraft) — Engine Access Door. Adjust door position by loosening the three screws in aft hinge halves and changing hinge position on serrated plate as required. Tighten hinge screws.

2-133. Removel (Unmodified Series 1 and 2 Aircraft) — Engine Access Door. *a.* Release all door latches.

b. Remove door hinge pivot bolts (fig. 2-3) and lift door from fuselage hinge halves.

2-134. Removal (Series 3 and Modified Series 1 and 2 Aircraft) — Engine Access Door. *a.* Release all door latches.

b. Index mark the shim plate, serrated plate, aft hinge halves and door (fig. 2-4) so that doors can be replaced in the same position.

c. Remove three screws and washers in the aft hinge halves. Two nuts and one rivnut are used. Tie parts together at each hinge point so that they do not become mixed.

NOTE

The hinges pivot on steel spacers and nylon shims which are not as easily removed as the entire aft hinge halves.

d. Lift doors free of fuselage.

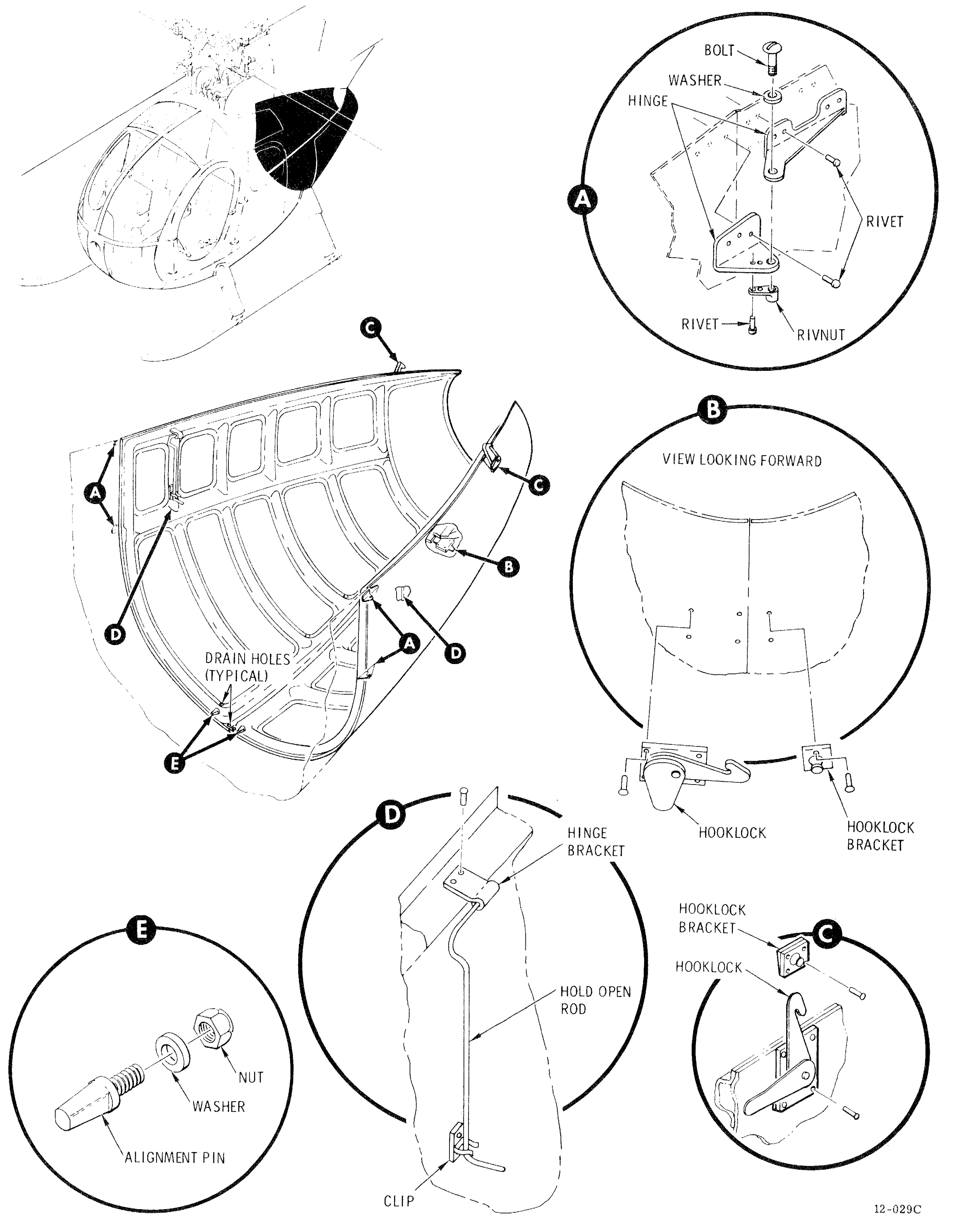


Figure 2-3. Engine Access Doors (Unmodified Series 1 and 2 Aircraft).

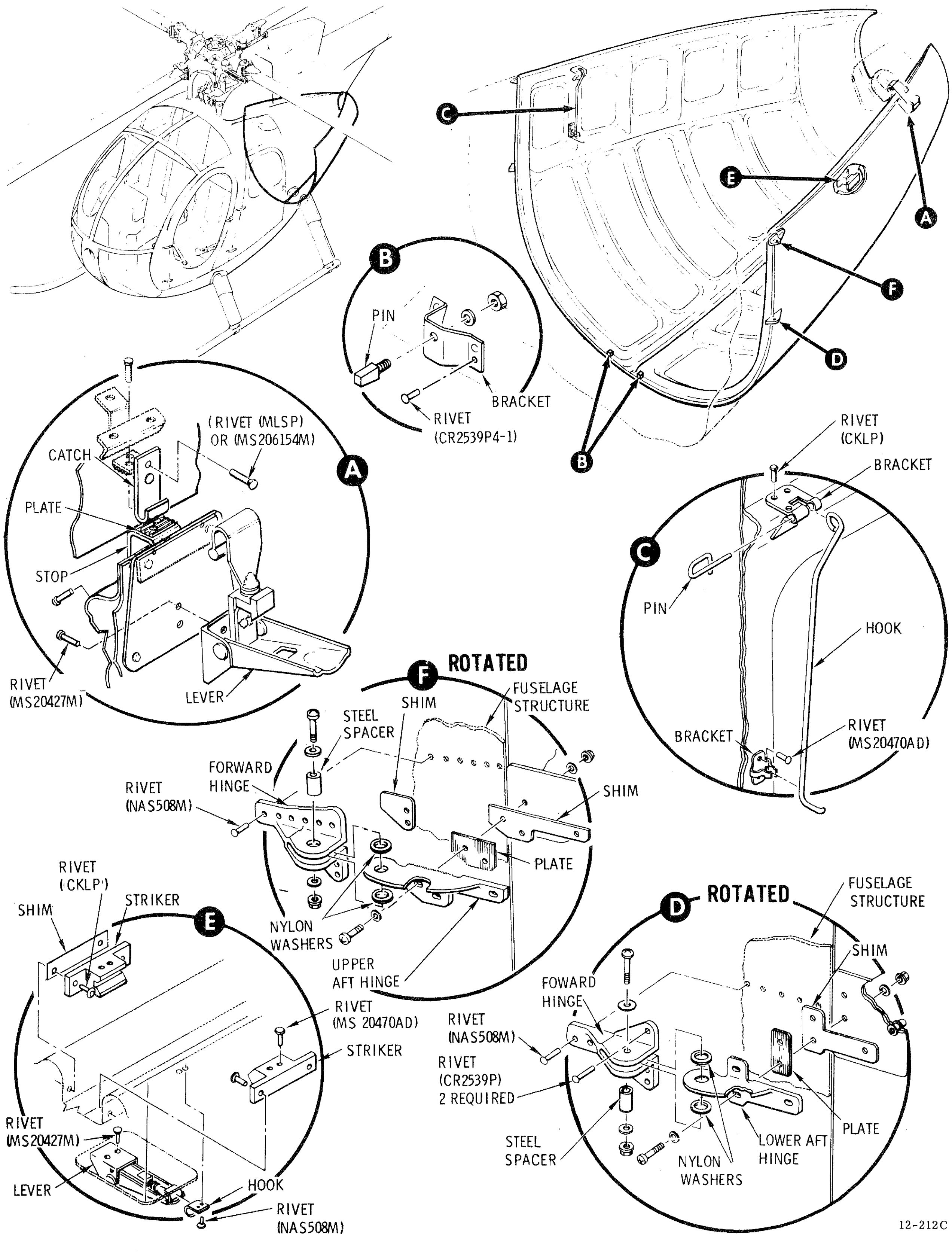


Figure 2-4. Engine Access Doors (Series 3 Aircraft and Modified Series 1 and 2 Aircraft).

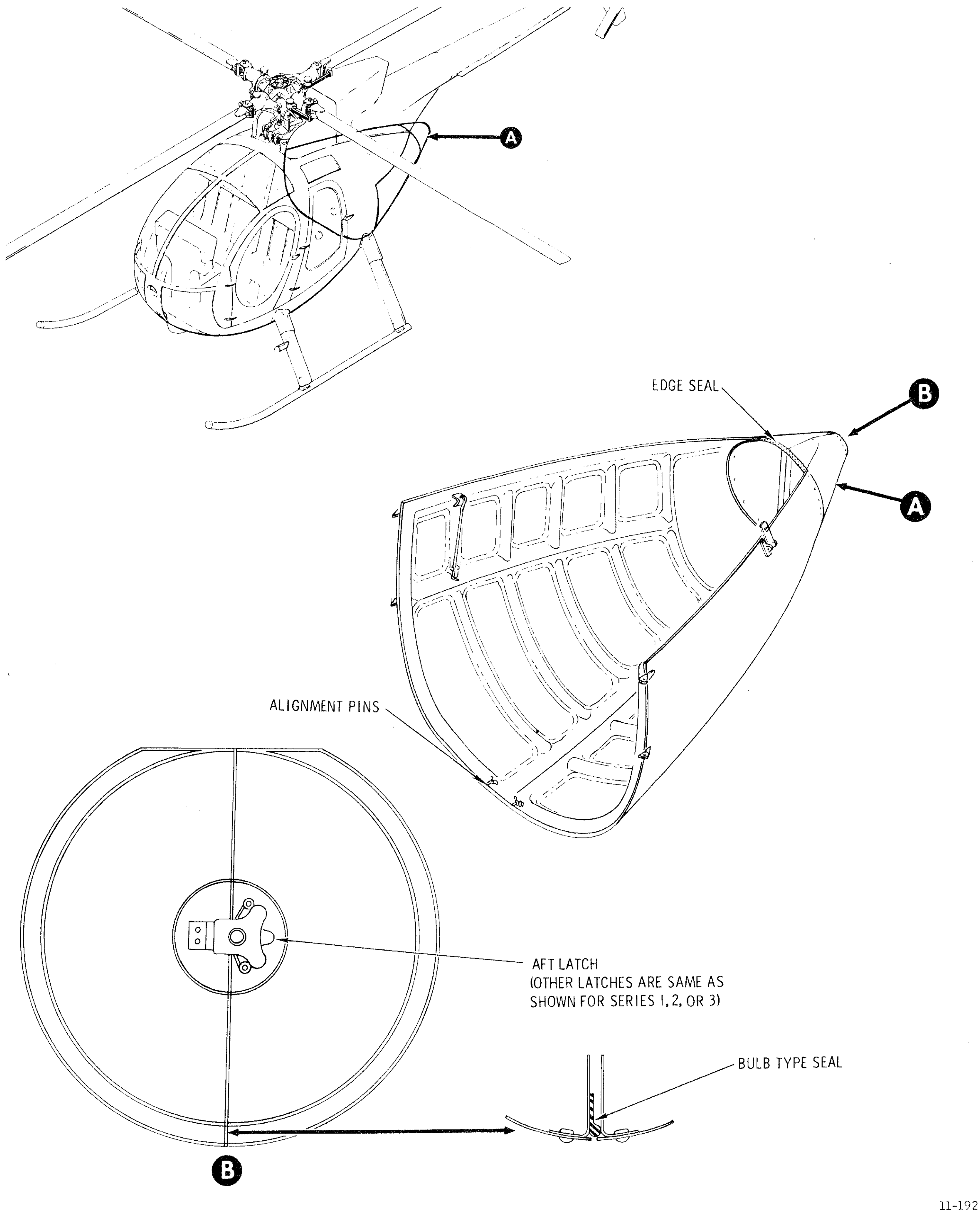


Figure 2-5. Engine Access Doors for Aircraft with Upward Exhausts.

2-135. Repair — Engine Access Door. *a.* Refer to paragraph 2-7 for patch and insertion repair procedures. Refer to table 2-3 for repair material.

b. (See fig. 2-3.) Improved fit of doors on aircraft without adjustable hinges may be obtained by shimming as follows:

(1) Remove engine access doors from fuselage (para 2-133).

CAUTION

Rivets must be carefully drilled out. Removing by the standard method of drilling and drift punching could result in damage to the internal structural bonding and surrounding material.

(2) Remove lower left and right hinges from engine access doors. Do not remove upper hinges.

(3) Remove all four fuselage hinges.

NOTE

Installation of shims must be accomplished as a full set (under each hinge indicated in (2) and (3) above) to provide proper alignment.

(4) Locate rivet holes in shims by using removed hinges as templates; drill and deburr.

(5) Paint mating surfaces with primer (C79). Use NAS508M4 rivets to install hinges, with shims between hinge and skin.

(6) Install engine access doors and check for fit and no preloading at hinge attach points.

2-136. Replacement — Engine Access Door Hinges. On aircraft engine access doors equipped with adjustable door hinges, replace hinges halves as follows:

a. Remove access door from fuselage structure (para 2-134).

CAUTION

Rivets must be carefully drilled out. Removal by the standard method of drilling and drift punching could result in damage to the internal structure and surrounding material.

b. Remove attaching hardware that secures defective hinge half to door. Drill out rivets that secure defective half to fuselage structure. (See fig. 2-4.)

c. Install replacement hinge half on engine access door and temporarily locate hinge serrated surface on hinge adjustable plate. Install hinge attaching hardware but do not tighten.

d. Locate existing rivet holes in fuselage structure for replacement hinge half. Match hole pattern in replacement hinge and drill out holes. Install hinge and shim on fuselage structure and secure with replacement rivets (details D and F, fig. 2-4).

e. Mount door on fuselage structure and install hinge pin hardware. Close all door latches. Check door for alignment and preloading at hinge points.

f. Complete the installation as follows. Make vertical adjustments to obtain proper door fit by installing shimming washers, as required, at hinge pinning points. Install shim(s) under door hinge to obtain lateral adjustment. Use slotted holes in hinge and serrated surfaces to obtain forward and aft adjustment. When adjustments are completed, tighten screws that secure hinge to access door.

g. Refinish area according to chapter 1.

2-137. Replacement — Engine Access Door Latches. On engine access doors having adjustable door latches, replace defective latches or latch hooks as follows:

CAUTION

Rivets must be carefully drilled out. Removal by the standard method of drilling and drift punching could result in damage to the internal structure and surrounding material.

a. Drill out rivets and remove defective latch or latch hook from engine access door or fuselage structure (fig. 2-4). Replace defective door stops or serrated adjustment plates, if required, by drilling out attaching rivets. Refer to paragraph 2-7 for patch and insertion repair procedures.

b. Locate existing rivet holes and install replacement latch, latch hook, door stop, or adjustment plate. Secure with replacement rivets (details A and E, fig. 2-4).

c. Adjust latches as required for proper tension of of latch lever to hook without deflecting the door (para 2-131).

d. Refinish area (except latch lever assembly) according to chapter 1.

2-138. Replacement — Engine Access Door Abrasion Strip and Seals. *a.* Replace worn abrasion strip tape by cleaning the contact area with naphtha (C70). Allow to air-dry a minimum of 30 minutes.

b. Install a 0.50 x 40.0 inch strip of polyurethane pressure tape (C106) to the clean dry surface. Apply hand pressure to ensure firm contact and trim as required. Replace worn and broken tail cone.

c. Repair seals according to procedure in paragraph 2-168.

2-139. Installation (Unmodified Series 1 and 2 Aircraft) — Engine Access Door. *a.* Lift engine access door into position and install hinge pivot bolts (fig. 2-3).

b. Close and latch doors. Check for firm fit and check that there is no door deflection when the hooks are latched.

c. For doors with adjustable latches, adjust as required (para 2-131).

2-140. Installation (Series 3 and Modified Series 1 and 2 Aircraft) — Engine Access Door. *a.* Lift door into position and assemble the marked aft hinge half parts (fig. 2-4).

b. Align index marks on shim, serrated plate, and hinge with marks on door and then tighten the attaching screws.

c. Close and latch doors. Check for firm fit without deformation of doors.

d. Adjust door latches or door position according to para 2-131 or 2-132, if required.

2-141. CARGO DOORS.

2-142. Description — Cargo Doors. The two cargo doors are similar except for an armament door at the lower edge of the left door. (See fig. 2-6.) Each cargo door is a bonded aluminum alloy frame containing a large polycarbonate plastic window. The door latching mechanism consists of four lever-type latches that are cable interconnected. A drain hole is located in the forward lower corner of the door outer skin panel. The outward swing of the door is limited by a rubber stop within each hinge bracket in the fuselage structure. Each door is equipped with a jettison mechanism for emergency jettisoning of the complete door assembly. Extruded rubber seals or an extruded aluminum alloy seal depressor on the fuselage door frame provide a weatherstrip seal with mating extruded rubber seals bonded on the door inner frames. If so modified each cargo door is equipped to lock from the inside and snap vents are reworked to prevent removal. (See fig. 2-7.) A short looped cable, secured at one end, may be pushed up on one locking lever to prevent rotation of all levers to the unlocked position. Snap vents are equipped with an aluminum tube that extends beyond the vent diameter to prevent removal.

NOTE

If cargo door handle cannot be rotated, the door lock may be in place. Open pilot's door; then reach over compartment rear bulkhead and slip looped cable down on upper forward locking lever.

2-143. Inspection — Cargo Door. *a.* Inspect plastic windows for cracks, scratches, crazing, and discoloration that would render the window unserviceable.

b. Inspect skins for cracks, wrinkled areas, dents, scratches, signs of corrosion, and general condition of painted surfaces.

c. Ensure that drain holes in lower corners of the outer skin panel are open.

d. Inspect door handles, latch levers, striker plates and jettison mechanisms for overall wear. Check lever actuating cable assembly for proper tension. Inspect visible portions of cable for excessive wear.

e. Inspect door hinges for secure attachment to door and for cracks.

f. Inspect all stricker plates for excessive wear. Inspect door frame for dents and deformation.

g. Inspect weatherstrip compression seal for cuts, wear, deterioration, and for secure bond to door frame. (Refer to para 2-168 for repair or replacement of rubber door seals.)

h. Inspect door frame anti-chafing tape for serviceability. Remove defective tape where necessary, clean frame surface with methyl ethyl ketone (C69), and replace with new tape (C106).

i. If installed, check door locking cable for condition, check locking adjustment, para 2-146.

2-144. Adjustment (Operational Check) — Cargo Door Latching. *a.* Open door and check for smooth operation of handles and latch mechanism. Check that decals indicate correct direction, check condition, and security. Check that door stop and springs function properly.

NOTE

When opening door, apply light inward force on the door. This will reduce wear of the latching mechanism by overcoming the door seal pressure.

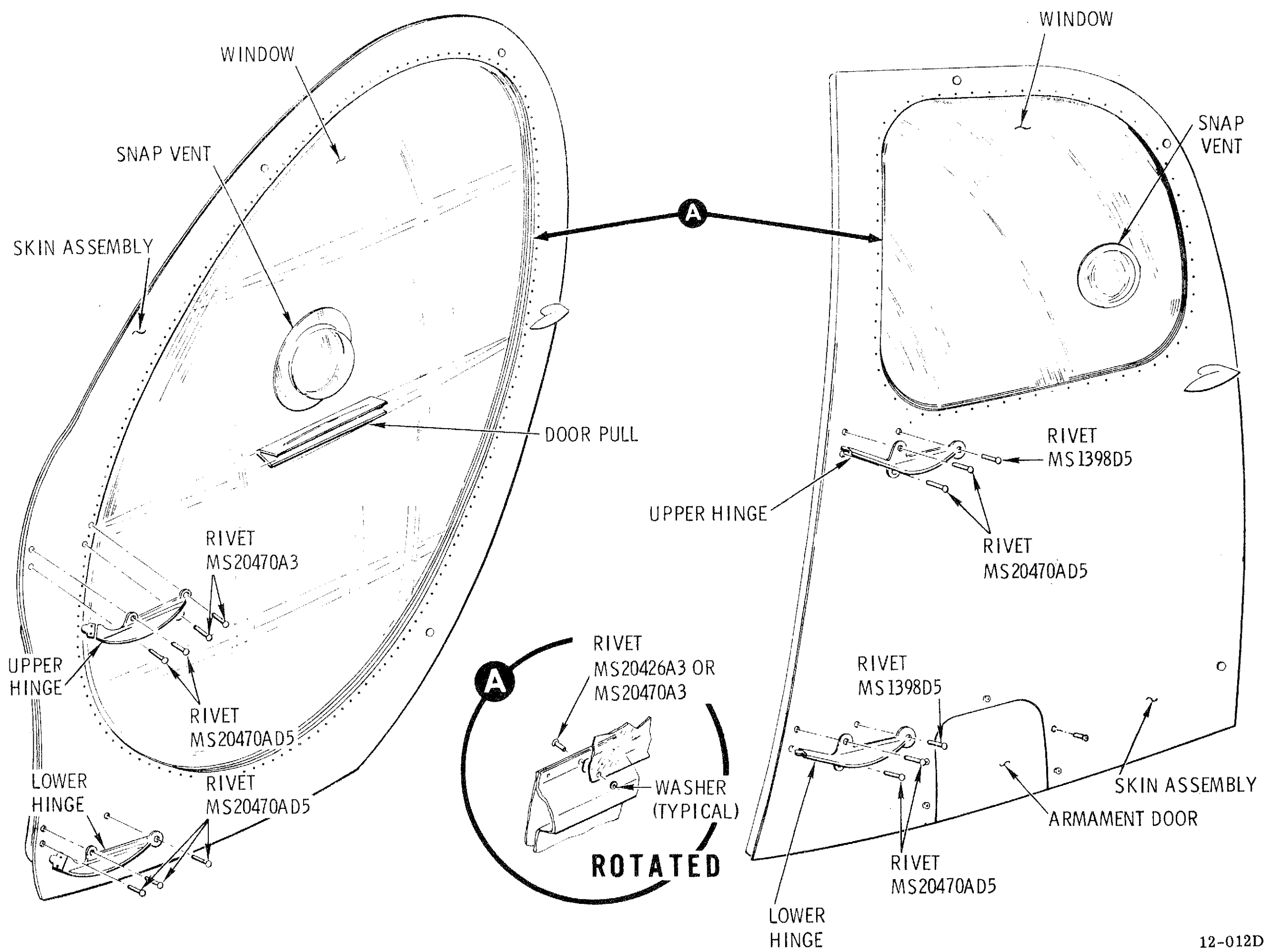


Figure 2-6. Pilot and Cargo Doors.

b. Close door from inside and check latching mechanism. (See fig. 2-6.) Check for lack of tension in cable assembly that actuates the three secondary latch levers. Remove slack by the tension adjustor.

c. Inspect the primary latch lever on the inside latch handle to ensure that it is within the detent position of the main striker plate.

2-145. Adjustment (Operational Check) — Cargo Door Seal Compression. *a.* Open door.

b. Hold a strip of heavy paper against the door seal so that the strip extends approximately 0.25 inch beyond the seal toward the outside of the doorway.

c. While holding the paper strip in place, close and latch the door.

d. Attempt to pull the paper strip from between the door frame and the seal. The strip should not pull out with less than a moderately heavy pull.

NOTE

Where the strip can be withdrawn with a light pull, the fit of the door against the frame is not tight enough to provide an adequate seal. This condition may be due to deterioration of the weatherstrip compression seal, deformation of the door frame, worn hinges, or bent hinge pins.

e. Repeat *a* through *d* at approximately 1-foot increments along the length of the door seal.

2-146. Adjustment (Operational Check) — Cargo Door Locks. *a.* Close and latch the door.

b. Inside the cargo compartment, loop the cable over the upper forward locking lever.

c. Check that the door handle cannot be rotated and the cable remains tight and in place.

d. Slip looped cable up; then open door and check

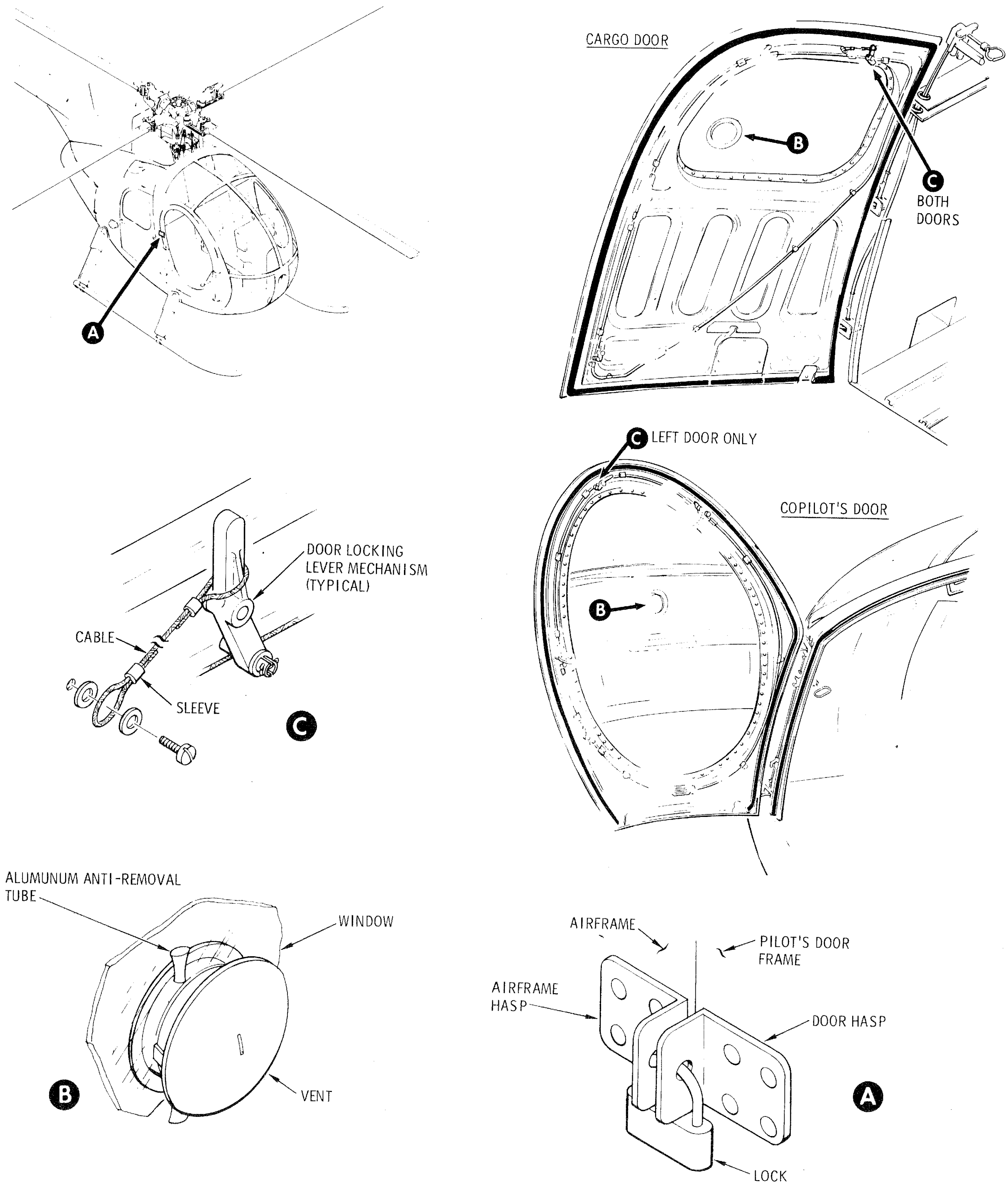


Figure 2-7. Door Locking Devices.

that the cable is free and does not interfere with door operation.

e. If cable is loose, remove loop sleeve and readjust using a new swedged sleeve or replace the cable.

2-147. Removal — Cargo Door. (See fig. 2-6.) *a.* Open the door and pull the hinge pins by the pin tabs.

b. Hold door in alignment so that hinges do not bind, and slide door hinges from hinge sockets to remove door.

2-148. Repair — Cargo Door Frame. Replace complete door frame if repair cannot be performed according to paragraph 2-269. Refer to table 2-3, items 2 and 8 for repair materials.

2-149. Repair — Cargo Door Hinges. (See fig. 2-6.) Replace damaged door hinges. Disassemble only as required for repair or replacement of parts.

a. Remove the door and drill out hinge attaching rivets.

b. Position replacement hinges at basic body attach points and secure with door hinge pins.

c. With damaged hinges removed, install door in closed position. Retain in place by latching the door.

d. Mark the hinge rivet attachment locations through the existing holes in the bonded door skin. Locate and mark the blind rivet attachment points (cargo door only).

e. Remove the door hinges and check for proper edge distance.

f. Drill out the located holes. Use a No. 41 drill at the aft rivet points of the pilot's door upper hinge. Use a No. 21 drill at all other hinge rivet points.

g. On pilot's door, install window attaching rivets with washers under bucked heads.

h. Install two mechanically expanded rivets at cargo door hinge aft attachment points.

i. Install rivets at remaining attachment points.

j. Paint as required.

k. Install doors and perform an operational check (para 2-144).

2-150. Repair — Cargo Door Windows. Refer to paragraph 2-228 for serviceability criteria and repair of plastics. Replace window if damaged beyond practical repair. Replace window as follows:

a. If a snap vent anti-removal device is installed, remove the tube. Use soft-nosed pliers and form one tube end into a round shape; then carefully withdraw the tube.

b. Squeeze snap vent into oval shape and remove from window.

c. Drill out rivets that attach window to door structure and remove window.

d. If door frame or window is to be reused, carefully scrape off sealant around edge of window frame by using a wooden or plastic scraper.

e. Clamp replacement window in position in door frame and drill No. 42 rivet holes using holes in door frame for pattern.

f. Remove window and spread a continuous bead of sealing compound (C89) around frame.

g. Install window with rivets. Use washers, as required, between formed rivet heat and plastic window.

h. Squeeze snap vent into oval shape and reinstall.

i. If snap vents are so modified, install anti-removal tube. Use soft-nosed pliers and form both tube ends into an oval shape to prevent tube removal.

2-151. Repair — Cargo Door Handle and Locking Mechanism. Replace loose lock lever bushings (6 and 9, fig. 2-8). Replace complete cable (15) and conduit as an assembly if either is damaged. Disassemble only as required for repair or replacement of parts.

a. Remove cotter pins (12) from cable retaining bushings (8) at each of the four locking levers (10 and 16). Remove spacers (21) and disengage cable.

b. Drill out rivets in all clips (22) that attach cable (15) and nylon conduit assembly to door frame.

c. Push out roll pin(s) (7), remove inner door handle (14), outer door handle (17), and locking lever (16). Retain associated hardware.

d. Push out roll pin(s) (7) in each of the three remaining locking levers (10) and remove by withdrawing pins (5 and 13).

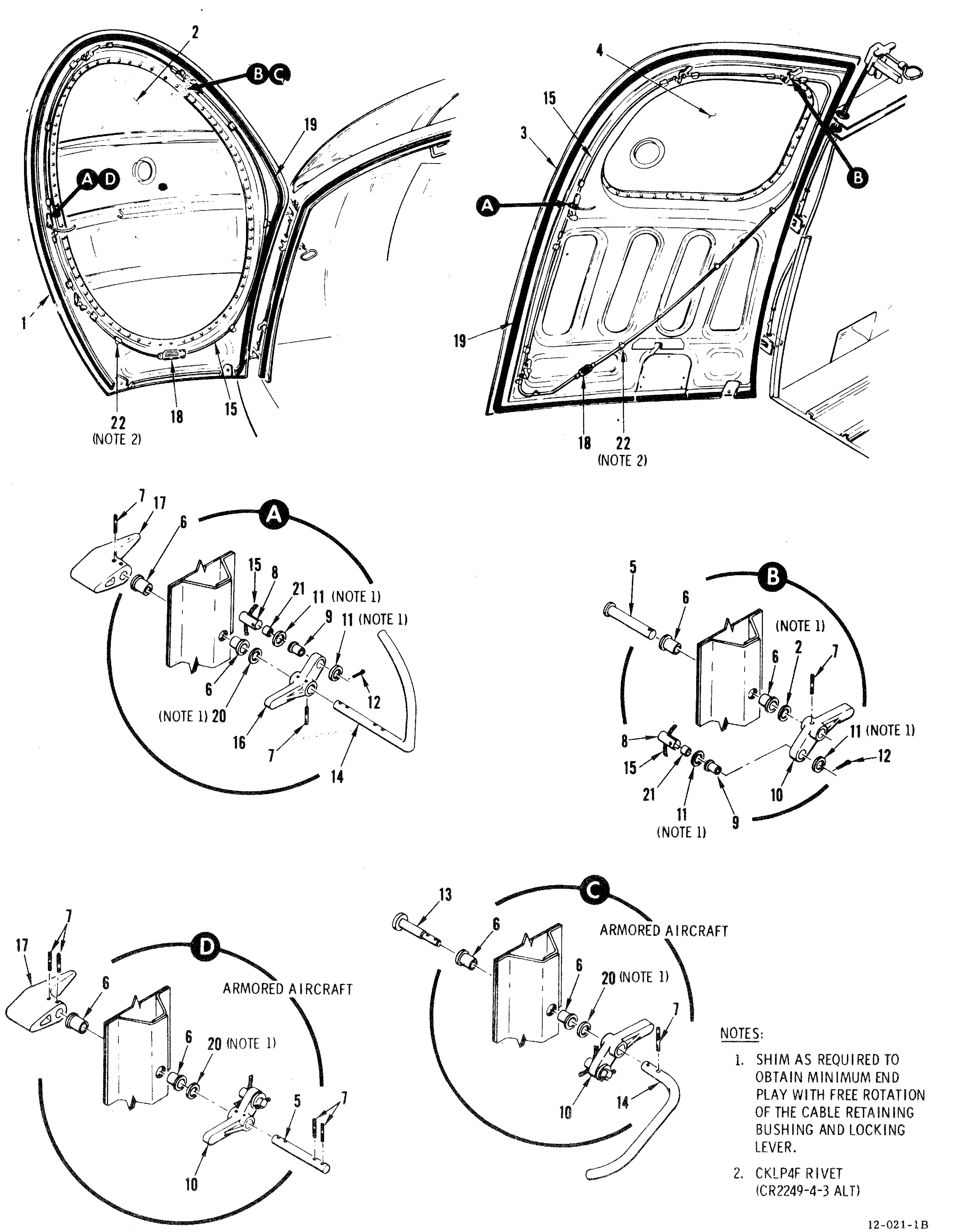
e. Remove bushings (6) from door frame, and bushing (9) from locking levers (10).

f. Install bushings (6) in door frame and install three locking levers (10) with pins (5) or (13) and roll pins (7). Shim with washers (20) to get free rotation and minimum end play between bushing (6) and locking lever (10).

g. Install inner door handle (14), outer door handle (17), and locking lever (16) with four roll pins (7). Using washers (20) between locking lever (16) and bushing (6), shim to get free rotation with minimum end play.

h. Install replacement conduit and cable assembly (15) with mechanically expanded rivets through clips (22). See figure 2-8 for rivet type and location. Use existing holes in old door frame.

i. Position all locking levers (10 and 16) and door handle in unlatched positions and install cable (15), retaining bushings (8), and washers (11). Use washers



NOTES:

1. SHIM AS REQUIRED TO OBTAIN MINIMUM END PLAY WITH FREE ROTATION OF THE CABLE RETAINING BUSHING AND LOCKING LEVER.
2. CKLP4F RIVET (CR2249-4-3 ALT)

Figure 2-8. Pilot and Cargo Door Details.

(11) to obtain minimum end play with free rotation of the cable retaining bushing. Install spacers (21) and cotter pin (12).

j. Check latch assembly for free movement and locking levers for proper positioning. Adjust cable tension at turnbuckle (18) as required.

k. If used, reinstall door locking cable as shown in figure 2-7.

l. Install door and perform an operational check (para 2-144 through 2-146).

2-152. Replacement — Cargo Door Latch Cable. If cable and conduit assembly cannot be obtained, cable (15, fig. 2-8) may be replaced as follows:

a. Remove pilot/cargo door from aircraft.

b. Remove cotter pins (12) and washers (11) from cable retaining bushing (8) at each of the four locking levers (10 and 16). Remove spacers (21) and disengage cable. Retain the associated hardware.

c. Do not remove rivets, clips (22) or conduit from door frame when replacing cable.

d. Remove cable (15) from conduit by cutting cable (15) at several points. Pull cable from plastic conduit. Do not cut conduit during cable removal.

e. Install new cable (MIL-C-5424) by threading end into conduit. Stop at each latch to thread on a ball.

NOTE

Do not swage balls on cable at this time.

f. Continue threading cable through conduit until cable and unswaged balls are positioned around the circumference of the door frame.

g. Mark the position on cable for each ball where the door latches will close on the striker plates of the door frame.

h. At position marked on cable in step *g* swage balls with swaging tool kit (T2) or an equivalent tool.

i. Pull cable tight around door.

j. Swage both ends of cable together with terminal splice (swedged sleeve).

k. Position all locking levers (10 and 16) and door handle in unlatched positions and reinstall retained associated hardware from step *b* onto cable (15). Use washers (11) to obtain minimum end play with free rotation of the cable retaining bushing (8). Install spacers (21) and cotter pins (12).

l. Check latch assembly for free movement and locking levers for proper positioning. Adjust cable tension at turn-buckle (18) as required.

m. Install door and perform operational check (para 2-144 through 2-146).

2-153. Repair — Cargo Door Seals. Refer to paragraph 2-168.

2-154. Repair — Rubber Cargo Door Stop. Replace door stop if damaged. Remove and re-bond if partially separated.

a. Remove partially separated or damaged door stops (fig. 2-9) by carefully prying bonded rubber stop out of the body hinge recess.

b. Bond the stop horizontally on the forward side of the hinge recess. (Refer to paragraph 2-173 for application of general use non-structural bonding adhesive.)

2-155. Replacement — Emergency Release Cargo Door Jettison Assembly. No repair of cable is permitted.

a. Drill out handle support rivets (fig. 2-9).

b. Carefully pull away rubber door seal aft of the center clip and drill out all remaining upper and lower clip rivets.

c. Work release handle through the bulkhead and remove grommets.

d. Remove the emergency release assembly. Remove clips and straighten tubes as required.

e. Install the replacement emergency release assembly by feeding the jettison door pins through the bulkhead channel. Feed the handle through the station 75 bulkhead.

f. Install two grommets and form the tubes to contour.

g. Install clips and brackets as shown in figure 2-9.

h. Install rivets at clip locations as shown in figure 2-9.

i. Reseal cargo door. (Refer to paragraph 2-168 for repair or replacement of extruded rubber seals.)

j. Install door and perform operational check of the door release mechanism.

2-156. Replacement — Cargo Door Hinge Spring. No repair of hinge spring is permitted.

a. Remove the door (para 2-147).

b. Remove two cotter pins (fig. 2-9), and pins located on the body hinge recess.

c. Remove stop spring.

d. Replace the stop spring with its longest flat area outboard and aft in the body hinge recess.

e. Install pins and cotter pins. Pins may be installed with heads down if interference prevents installation with cotter pin hole down.

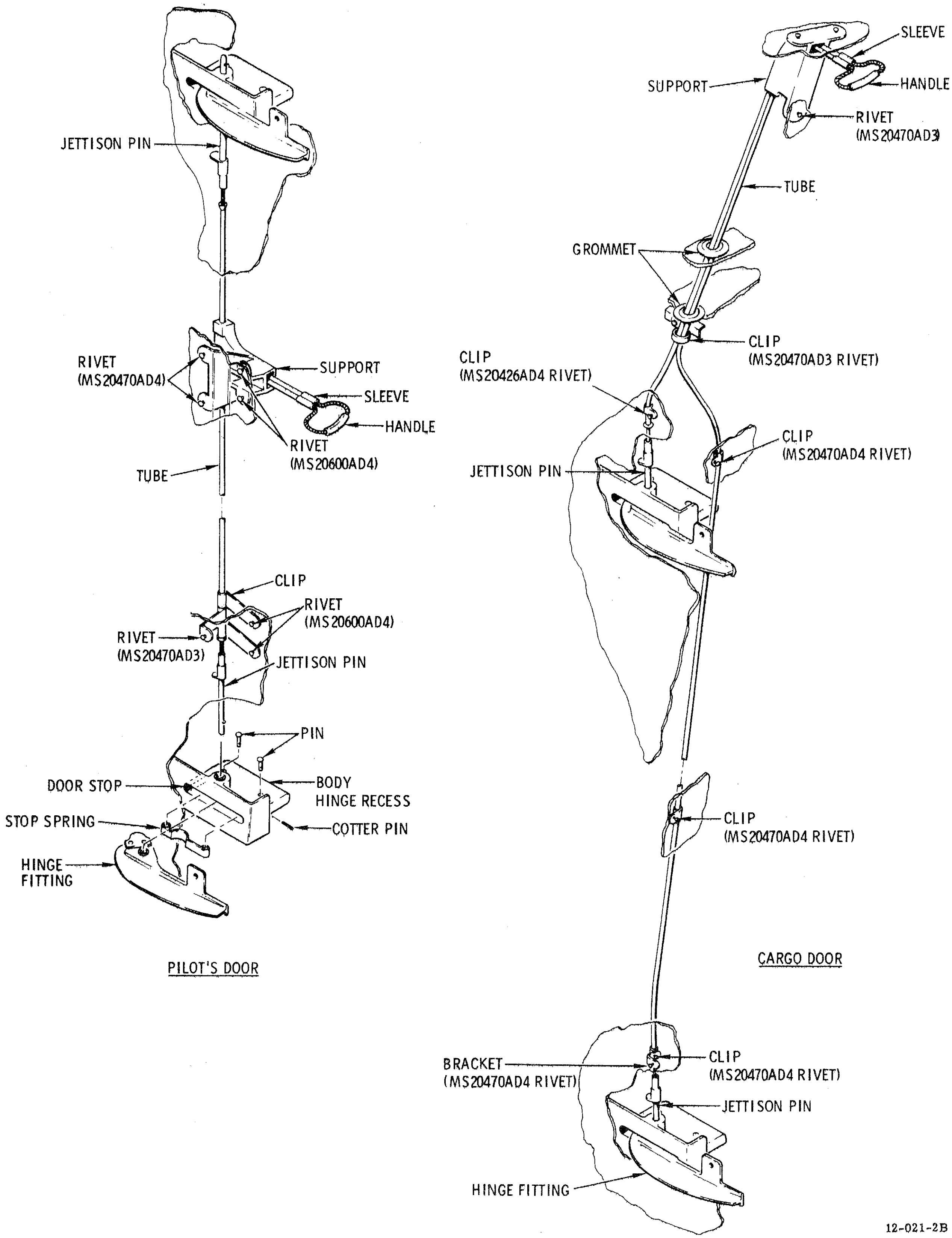


Figure 2-9. Pilot and Cargo Door Jettison Mechanism.

f. Cover exposed sharp edges of cotter pins with sealer (C89).

2-157. Installation — Cargo Door. (See fig. 2-9.) *a.* Lift door into position so that hinges are aligned, and engage door hinges with fuselage hinge sockets.

NOTE

Door must be held open while inserting hinge pins to keep pin holes aligned for pin insertion.

b. Insert pins through upper and lower hinges until spring-loaded ball detent in pin emerges past the lower side of the hinge socket. Close and latch the door.

2-158. ARMAMENT DOOR.

2-159. Description — Armament Door. The aluminum alloy armament door (fig. 2-6) is mounted in the lower edge of the left cargo door. A stiffener clips to the cargo door frame. A section of rubber door seal is bonded to the lower edge.

2-160. Removal — Armament Door. (See fig. 2-6.) *a.* Open the left cargo door.

b. Remove five screws and washers that secure armament door to cargo door.

c. Push the armament door stiffener to unclip from cargo door and remove armament door.

d. Close the cargo door.

2-161. Inspection — Armament Door. *a.* Inspect for corrosion, distortion, or cracks.

b. Check for loose rivets or nutplates.

c. Inspect stiffener for condition of weld.

d. Inspect door seal and anti-chafing tape for serviceability.

2-162. Repair — Armament Door Nutplate/Rubber Seal Strip. Replace a defective nutplate or rubber seal strip as follows:

a. Remove defective seal strip and nutplates from door as required. Replace plastic tape seal strip, if installed, when necessary.

b. Clean door surface with solvent (C94), and dry (with low pressure compressed air, if available).

c. Use MS20426AD3 rivets to replace nutplate.

d. Cut holes through replacement seal strip to match existing holes and install strip. (Refer to paragraph 2-168 for repair or replacement of extruded rubber seals.)

2-163. Installation — Armament Door. (See fig.2-6.) *a.* Open cargo door.

b. Position armament door against inside of the cargo door. Clip stiffener to cargo door frame and secure with screws and washers.

c. Close cargo door.

2-164. PILOT'S DOORS.

2-165. Description — Pilot's Doors. Each pilot's door is a bonded aluminum alloy frame containing a large polycarbonate plastic window with three lateral plastic stiffeners. (See fig.2-6.) The door latching mechanism consists of four lever-type latches that are cable interconnected. The inside latch handle is normally located at the rear edge of the door. On aircraft with armor installed, the inside latch handle is located at the upper forward edge of the door frame and a plastic door pull is bonded to the middle stiffener of the window. If so modified, the copilot's door is equipped to lock from the inside the same as the cargo doors. (See fig. 2-7.) The pilot's door is equipped with an external locking device consisting of two flanges, one on the door and one on the door frame, to accommodate a padlock. The door window snap vents on both pilot's doors are equipped with the anti-removal device the same as that on the cargo doors. Installation and jettisoning features of both doors are comparable to those of the cargo doors. Refer to paragraph 2-141 for operational check, removal, inspection, and installation.

2-166. Repair — Pilot's Doors. (See fig. 2-8 and 2-9.) The emergency release jettison mechanism, the latching system, and the adjustment of the latch actuating cable are comparable to that of the cargo doors (para 2-141). Except for replacement of the jettison release, repair of the pilot's doors and windows is outlined under cargo door information (para 2-141). (Refer to table 2-3, items 3 and 8 for repair material.)

2-167. Replacement — Emergency Release Pilot's Door Jettison Assembly. No repair of cable is permitted.

a. Drill out four rivets that attach the door jettison handle support (fig.2-9).

b. Drill out three rivets that attach the tube support clip.

c. Remove the handle, support tube and clip as a unit.

d. Install by placing emergency release assembly in position with jettison pins inserted in door hinge pivot holes.

e. Align the jettison handle support with existing holes and install two rivets, bucked head inboard, and two mechanically expanded rivets as shown in figure 2-9.

f. With the tube firmly seated in the jettison handle support, position the clip at the lower flared end of the tube. Check for matching door frame hole alignment.

NOTE

If tube clip mounting holes fall approximately 2 inches above the existing door frame holes, proceed with h below.

g. Position clip as described in *f* above. Install rivet with manufactured head inboard. Install mechanically expanded rivets as shown in figure 2-9.

h. Lay a straightedge across the center line. Position the clip at the lower flared end of the tube. Locate and mark the clip mounting holes on the centerline. Drill two No. 30 holes for inboard rivets and one No. 41 hole for the outboard rivet.

i. Proceed with *g* above.

j. Plug any open holes with rivets.

k. Install door and perform operational check of the door release mechanism.

2-168. REPAIR OR REPLACEMENT — RUBBER EXTRUSION SEALS.

2-169. Replacement — Rubber Extrusion Seals. *a.* Carefully pull or scrape away damaged seal and remove all traces of adhesive from seal mounting surface by using a cloth dampened with methyl ethyl ketone (C69).

b. Restore chemical film protection to any base metal exposed during cleaning process by using chemical film (C20).

c. Clean the seal contact area with naphtha (C70) and allow to air-dry a minimum of 30 minutes.

d. Remove all traces of talc, grease, oil, and dust from rubber seal by using naphtha (C70). Allow to air-dry a minimum of 30 minutes prior to application of adhesive primer.

e. Thoroughly mix primer (C77). Brush a thin coat of primer on the rubber bonding surfaces and air-dry for a minimum of 30 minutes.

f. Cure primer-coated seal in a circulating air oven for 50-70 minutes at 145-175°F (63-79° C).

g. Thoroughly stir the adhesive (C10).

h. Apply one uniform brush coat of adhesive to the seal contact area rubber seal.

NOTE

Do not overbrush. Allow to air-dry for a minimum of 20 minutes.

i. Apply a second uniform brush coat to the seal contact area and rubber seal. Allow to dry approximately 5 minutes. Test tack with a knuckle. Adhesive is ready for mating when adhesive no longer transfers.

j. Carefully align seal with the mounting surface and press firmly together. Allow adhesive to cure a minimum of 4 hours before applying any load.

2-170. Repair — Partially Separated Rubber Extrusion Seals. *a.* Clean the separated area with naphtha (C70) and allow to dry for a minimum of 20 minutes.

b. Thoroughly stir adhesive (C10).

c. Apply one uniform brush coat of adhesive to both the rubber seal and mating surface contact area. Allow to dry for 5 minutes and press mating surfaces together.

2-171. REPAIR OR REPLACEMENT OF SILICONE RUBBER GASKETS.

2-172. Repair or Replacement — Silicone Rubber Gaskets. *a.* Carefully pull or scrape away damaged gasket.

b. Remove all cement and foreign material from gasket mounting surface with a cloth dampened by methyl ethyl ketone (C69).

c. Using masking tape (C10), mask area to restrict cement to mounting surface.

d. Evenly spread a thin layer of adhesive (C13) on one of the surfaces to be bonded.

e. Carefully align gasket with mounting surface and press firmly and evenly to displace any air bubbles. Do not force adhesive out of the joint.

f. Allow bond to set up for 24 hours at room temperature. Maximum bond strength will be reached in 72 hours.

2-173. APPLICATION OF GENERAL NON-STRUCTURAL BONDING ADHESIVE.

2-174. General — Application of General Non-Structural Bonding Adhesive. Except as noted otherwise, this method may be utilized at all levels of maintenance as a general purpose replacement for repair of rubber and non-structural bonded materials.

a. Clean mating surfaces with a cloth dampened by methyl ethyl ketone (C69).

b. Restore chemical film protection to any base metal exposed during cleaning process by using chemical film (C20).

NOTE

When mixing adhesive, use a wax-free cup.

c. Using equal parts by volume of adhesive parts A and B (C7), prepare a mixture of adhesive. Mix until color becomes uniform light green.

d. Apply an even, thin coating of adhesive to mating surfaces.

e. Carefully position surfaces to be mated. Apply sufficient pressure to assure good contact until bonding cures; 8 hours at ambient (room) temperature, or 2 hours at 150°F (66°C).

2-175. SOUND INSULATION BLANKETS.

2-176. Description — Sound Insulation Blankets. Sound insulation blankets are installed across the aft section of the passenger-cargo compartment at station 124.00 (fig. 2-10). The insulation is removed by sections to permit removal of the main gearbox access cover and cargo compartment left and right side aft bulkhead access covers. The blanket sections consist of a double fiberglass core backed with netting and faced with vinyl-coated nylon fabric. The blankets are attached to the compartment structure with nylon fastener tape hook and pile devices located on mating surfaces. Cutouts are provided in the lower section for troop seat attach fittings, the cargo compartment heat duct, and for avionics components (on Series 3 aircraft). The main transmission insulation cover is bonded to and removed with the main transmission cover. Head pads, web pads and pad fillets are bonded to the upper aft section of the cargo compartment structure. Additional insulation blankets are nested on the inside of each aft bulkhead channel.

2-177. Removal — Sound Insulation Blankets. (See fig. 2-10.) *a.* Remove troop seats from cargo compartment (para 2-196).

b. Remove nylon fastener tape by pulling from adjoining sections.

c. Carefully separate nylon fastener tape hook and pile and release tape edging as required to free insulation blanket sections.

d. Remove non-fixed plugs.

e. Remove main transmission cover (with bonded insulation) blanket) as an assembly (para 2-23).

f. Remove upper and lower (non-fixed) insulation blankets from inner frames.

NOTE

Pads and piping are bonded to the airframe and should not require removal.

2-178. Inspection — Sound Insulation Blankets. *a.* Inspect blankets for holes, tears, cuts and serviceability.

b. Check binding and doublers for loose or torn fabric.

c. Inspect blankets and mating surfaces for loose, torn or missing hook and pile fastener tape.

d. Inspect heat pads and piping for security of bond to structure and for tears, etc.

2-179. Repair — Sound Insulation Blankets. *a.* Patch holes, tears, and cuts with general purpose adhesive-backed cloth tape.

b. Repair loose bindings, doublers, padding and nylon fastener tape according to *c* through *f* below.

c. Remove all dirt, dust, oil and grease by wiping with a cloth dampened by isopropyl alcohol (C82).

NOTE

Loosened nylon fastener tape might be reactivated for adhesion by wiping the original adhesive film with methyl ethyl ketone (C69).

d. Apply a thin even coat of cement (C19) to each mating surface.

e. Allow to dry from 3 to 5 minutes until the cement has strong tack.

f. Align the mating surfaces and press together.

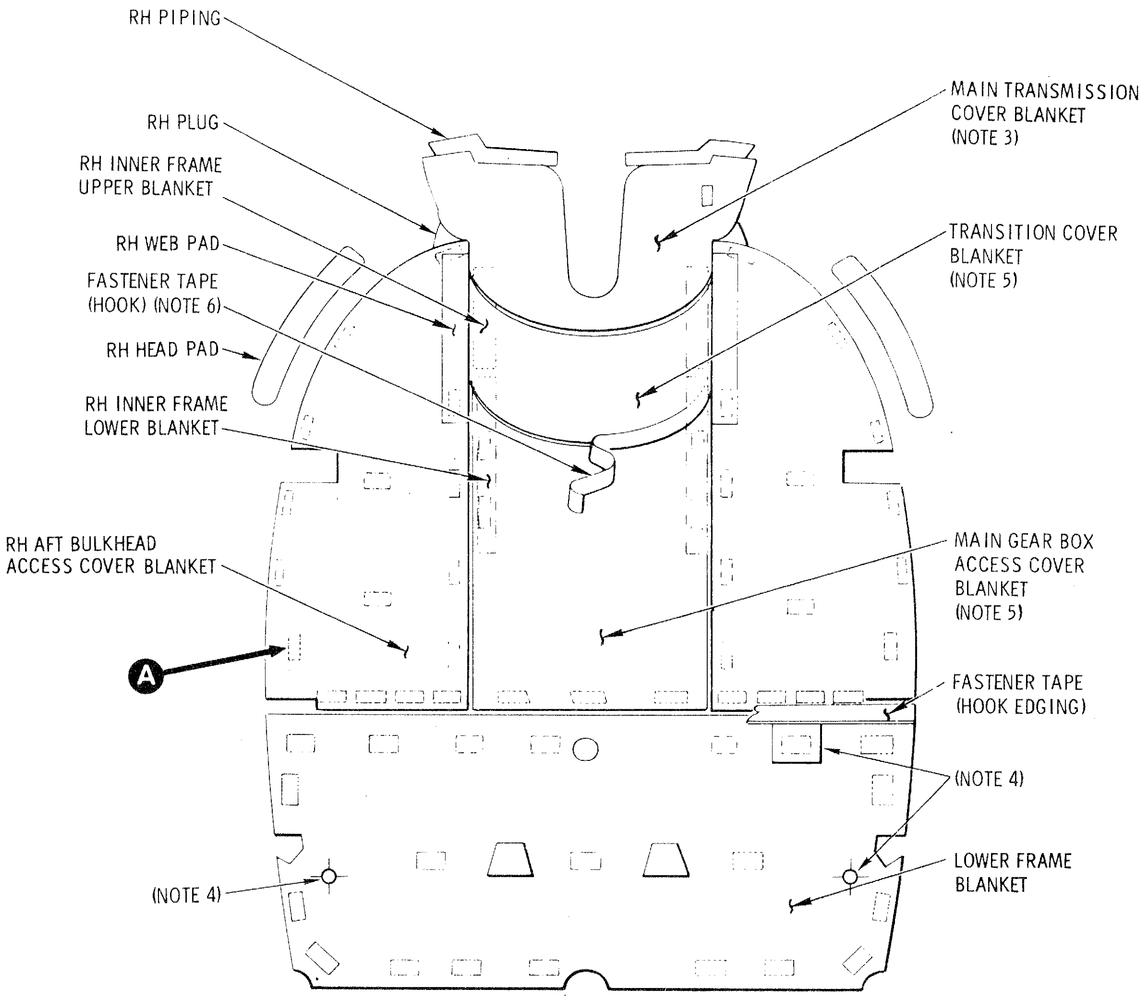
g. Defective nylon fastner tape may be replaced using the removed velcro fastener as a pattern. Cut new hook and pile fastener tape to size, activate adhesive backing by wiping with methyl-ethyl-ketone (C69) and press in place. Allow 3 to 5 minutes for drying time before installing insulation blanket to aircraft.

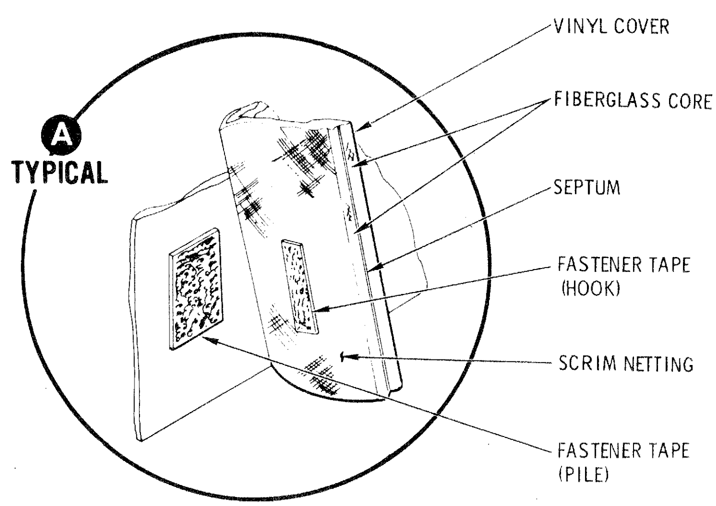
2-180. Installation — Sound Insulation Blankets. (See fig. 2-10.)

CAUTION

Check that all nylon fastener tape is clean and not clogged with foreign matter. Clean as necessary. Replace fastener tape (pile) that does not provide positive attachment.

a. Install main transmission cover (para 2-23).





NOTES:
1. USE INSULATION BLANKET AS A TEMPLATE FOR LOCATION OF LOOSE OR MISSING BULKHEAD FASTENERS.
2. RIGHT SIDE (RH) DETAILS IDENTIFIED ARE TYPICAL OF LEFT SIDE DETAILS.
3. PART OF MAIN TRANSMISSION COVER.
4. CUTOUTS FOR RADIO WIRE HARNESS ON SERIES 3 AIRCRAFT ONLY.
5. TRANSITION AND MAIN GEAR BOX ACCESS COVER BLANKETS ARE A ONE-PIECE ASSEMBLY IN ALTERNATE CONFIGURATION.
6. NOT REQUIRED WITH ONE-PIECE BLANKET ASSEMBLY.

11-132B

Figure 2-10. Sound Insulation Blankets.

b. Install upper and lower (non-fixed) insulation blankets into inner frames.

c. Press insulation blanket sections into place on the bulkhead tape piles.

d. Install nylon fastener tape (hook and pile), and set with hand pressure.

e. Install troop seats (para 2-196).

2-181. PILOT'S SEATS AND SEAT ARMOR

2-182. Description — Pilot's Seats and Seat Armor. Two-part seats are installed in the pilot's compartment. (See fig. 2-11.) Each two-part seat consists of aluminum tubing (0.80-inch-diameter) with nylon mesh covering and rubber pads cemented to the frame mounting plates. Seat bottoms are secured to the top of the pilot's seat structure. On aircraft without armor installed, the seatbacks are mounted on the station 78.50 canted bulkhead. On aircraft requiring armor installation, the seatback is replaced by a reinforced ceramic tile, winged plate. The plate is mounted to bulkhead attaching lugs with four straight pins and cotter pins. The pilot's armor wing rests on a support cushion at its lower forward edge. Seatback cushions are attached to the plate with velcro hook and pile fasteners.

2-183. Removal — Pilot's Seats and Seat Armor. (See fig. 2-11.) *a.* Clear seat of seat belt and shoulder harness.

b. On aircraft without armor installed, remove attaching screws, seatback, and seat bottom.

WARNING

If the aircraft is to be operated with part or all of the seatback armor panels removed, the engine fuel controls armor may also have to be removed. A weight and balance check must always be made to ensure that center of gravity limits will not be exceeded.

c. On aircraft with armor installed, remove seat bottom as above. Remove cotter pins, straight pins and seatback armor.

d. Release nylon fastener tape and remove backrest cushions.

2-184. Inspection — Aircraft Pilot's Seats Without Armor Installation. *a.* Inspect seat mounting plates and nutplates for security and rubber pads for excessive wear.

b. Inspect seat frame tube members for distortion, cracks, corrosion, and paint deterioration.

c. Inspect nylon mesh webbing for damage or excessive wear and installation date (TM 55-1500-204-25/1).

d. Inspect nutplates and rivets for security.

2-185. Inspection — Aircraft Pilot's Seats With Armor Installation. *a.* Inspect seat bottoms (para 2-184).

b. Inspect armor mounting pins and attach fittings for security and condition.

c. Inspect armor for bullet and shrapnel strikes.

d. Inspect armor mounting brackets for breaks, cracks and distortion.

e. Inspect backrest cushion for cuts, tears and security of attachment. (Refer to para 2-179 for repairs.)

f. Inspect pilot armor rubber edge guard for condition and security of attachment. (Refer to para 2-168 for repair or replacement of rubber seals.)

g. Inspect pilots armor wing support bracket for cracks, security of attachment and distortion.

2-186. Repair — Pilot's Seats Armor. (See fig. 2-11.) *a.* Replace damaged seatback armor panels. No repairs are permitted on armor panels other than replacement of attaching fittings.

b. Replace damaged or distorted armor bracket (detail B, sh 1) or armor attach fitting (details E, F, G, sh 2).

c. Replace damaged armor attach fitting nutplates.

2-187. Repair — Pilot's Seats. (See fig. 2-11.) *a.* Remove damaged rubber pads from seat leg mounting plate.

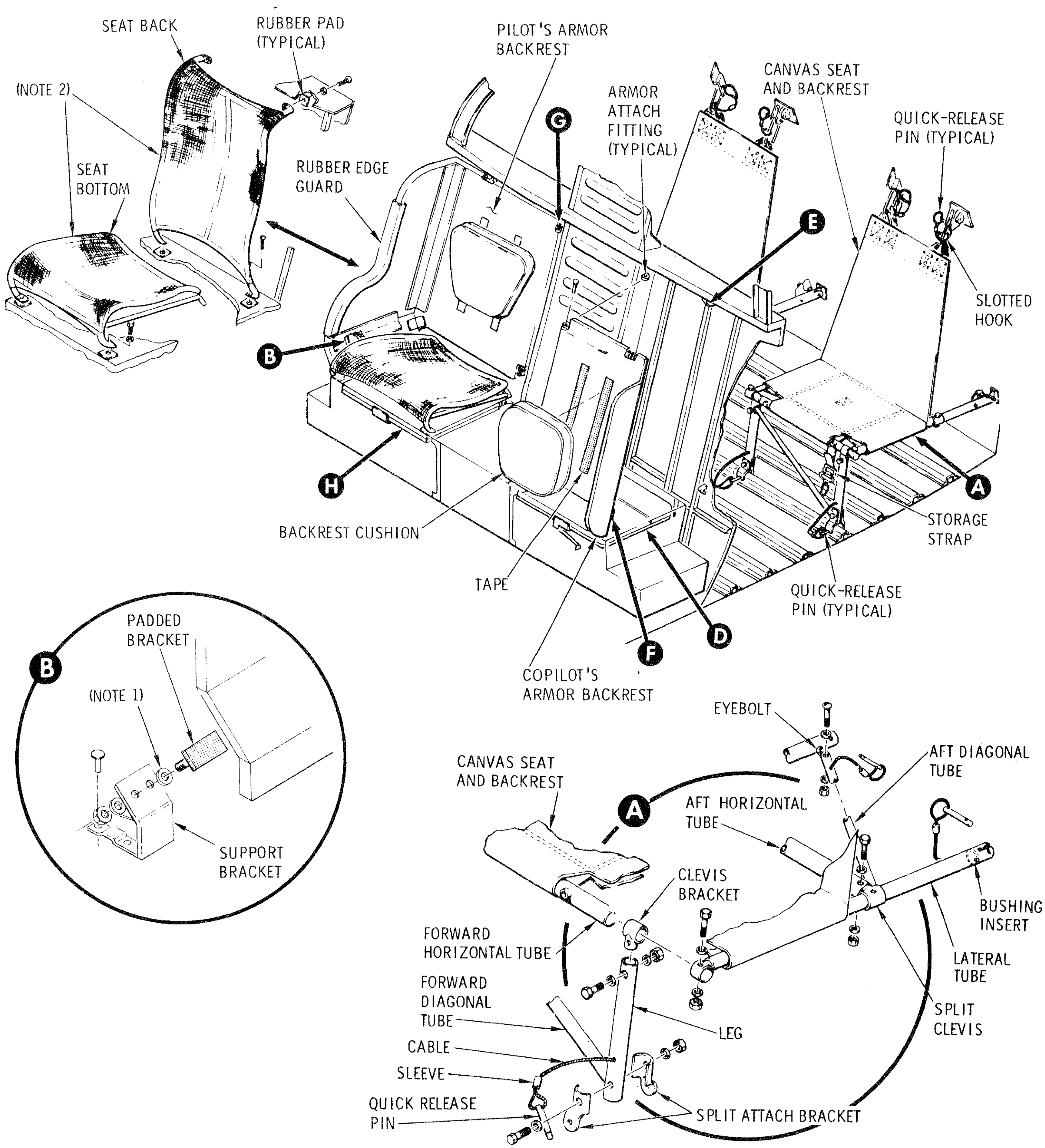
b. Clean seat leg mounting plate with solvent (C94).

c. Apply cement (C19) and attach new rubber pad to seat leg mounting plate.

d. Repair frayed or torn nylon webbing or replace if damaged beyond repair. (Refer to TM 55-1500-204-25/1.)

e. Remove minor faults and imperfections in tube members by polishing to restore original finish. Repair any damage not removable by light polishing by splicing of damaged tube sections (TM 55-1500-204-25/1).

2-188. Installation — Aircraft Pilot's Seats Without Armor Installation. (See fig. 2-11.) *a.* Clear seat structure and bulkhead mounting areas of seat shoulder harness and any foreign material.



NOTES:
1. SHIM WASHERS (THK CRES) AS REQD FOR BRACKET-TO-ARMOR CONTACT: FOUR MAXIMUM PER STUD.
2. INSTALLATION DATE STENCIL (1/2 INCH LETTERING) ON NYLON CLOTH UNDER OUTBOARD SIDE PARALLEL WITH SEAT TUBING. INSPECT WITH MIRROR.

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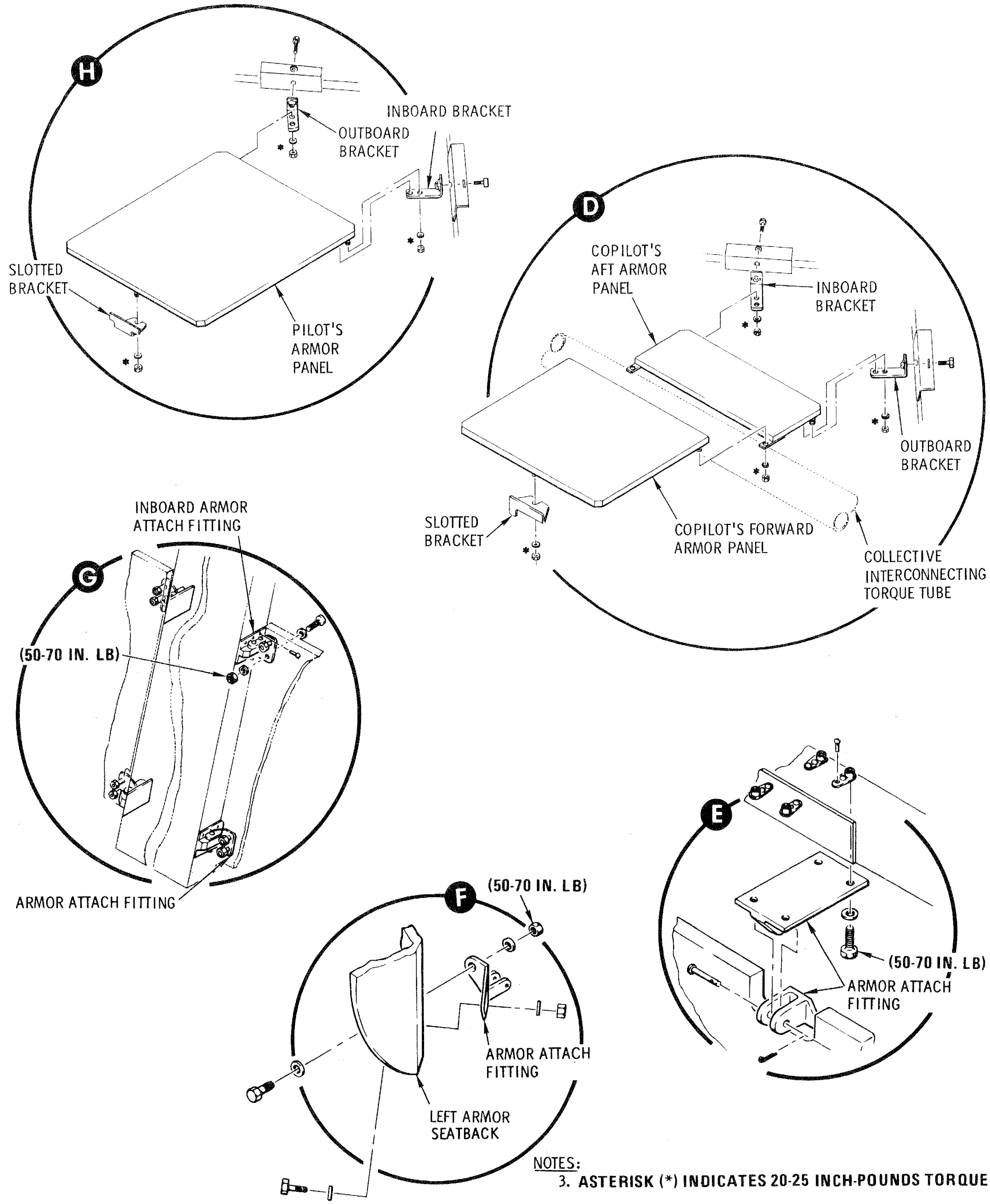
Figure 2-11. Seats and Seat Armor. (sheet 1 of 2)



NOTES:
3. **ASTERISK (*) INDICATES 20-25 INCH-POUNDS TORQUE.**

11-041-2B

Figure 2-11. Seats and Seat Armor. (sheet 2 of 2) *b.* Install six screws and firmly secure seat bottom and seatback.

2-189. Installation — Aircraft Pilot's Seats With Armor Installation. (See fig. 2-11.)

WARNING

If the aircraft is to be operated with part or all of the seatback armor panels installed, the engine fuel controls armor may also have to be installed. A weight and balance check must always be made to ensure that center of gravity limits will not be exceeded.

a. Position armor seatback panel. Install four straight pins and secure with cotter pins.

NOTE

Tilting copilot's seatback armor forward provides greater access for attachment of fittings at lower outboard side. These fittings are to be attached first. Tilting also requires temporary removal of two inertia reel control clamps to allow clearance in the armor tilt path.

b. Observe and note fit of pilot's seatback armor between lower wing and padded bracket.

c. Use washers as required (four thick washers maximum) to position padded bracket for contact with underside of lower armor wing. Install two washers and locking nuts to secure padded bracket.

NOTE

Shimming washers may be used in place of the two nuts next to the bracket pad when these nuts do not allow enough downward adjustment. No more than four shimming washers are to be used on each stud of the bracket for upward adjustment.

d. Position backrest cushions and firmly press mating nylon fastener tape (hook and pile) together.

e. Install seat bottom.

2-190. PILOT'S UNDERSEAT ARMOR.

2-191. Description — Pilot's Underseat Armor. On aircraft requiring armor installation, aluminum alloy armor support brackets (fig. 2-11) are riveted to the seat support structure beneath the pilot and copilot positions to support arm panels.

2-192. Removal — Pilot's Underseat Armor Panels. (See fig. 2-11.)

WARNING

If the aircraft is to be operated with part or all of the underseat armor removed, the engine fuel controls armor may also have to be removed. A weight and balance check must always be made to ensure that center of gravity limits will not be exceeded.

a. Remove foot support fairings (para 2-35) as applicable.

b. For copilot's armor removal, lay enough padding on collective interconnecting torque tube (fig.2-11) to prevent damage of tube.

c. Remove nut and washer at forward slotted bracket. Armor panel will remain supported by slotted brackets.

d. Support armor panel and remove bolts and washers from aft (inboard and outboard) mounting brackets.

e. On copilot's armor panel only, remove the two nuts and washers that join the forward and aft panel sections. Raise the forward panel far enough to get clearance and remove the aft panel.

f. Remove armor panel supported by slotted bracket.

g. Remove attaching nuts and washers to remove aft mounting brackets, as required.

h. Remove torque tube padding.

2-193. Inspection — Pilot's Underseat Armor Panels. *a.* Inspect armor panel for bullet and shrapnel strikes, and for loose or damaged mounting studs.

b. Inspect armor panel for secure attachment, and mounting brackets for breaks, cracks and distortion.

2-194. Repair — Pilot's Underseat Armor Panels. No repairs are permitted on armor panels other than replacement of attaching fittings.

2-195. Installation — Pilot's Underseat Armor Panels. (See fig. 2-11.)

WARNING

If the aircraft is to be operated with

part or all of the underseat armor installed, the engine fuel controls armor may also have to be installed. A weight and balance check must always be made to ensure that center of gravity limits will not be exceeded.

a. Remove foot support fairing (para 2-35) as applicable.

b. When installing copilot's armor, lay enough padding on collective interconnecting torque tube to prevent damage to tube.

c. If not already installed, attach aft mounting brackets to armor panel with nuts and washers. **TORQUE NUTS TO 20 — 25 INCH-POUNDS.**

d. Insert armor panel through foot fairing opening so that single mounting stud is down and in front. Place forward stud into slotted bracket.

e. On copilot side only, insert armor panel aft section so that panel joint brackets are forward and install two new nuts with washers. **TORQUE NUTS TO 20 — 25 INCH-POUNDS.**

f. Align the two aft (inboard and outboard) mounting brackets with mating holes in structure bracket and install two bolts with washers.

NOTE

Check that each aft corner of armor panel clears end of mounting bracket bolts by a minimum of 0.12 inch. Ensure that armor does not contact underseat equipment such as collective interconnecting torque tube, electrical units and wiring.

g. Secure panel to slotted bracket with washer and new nut. **TORQUE NUT TO 20 — 25 INCH-POUNDS.**

h. Remove torque tube padding and reinstall foot support fairings.

2-196. TROOP SEATS.

2-197. Description — Troop Seats. Two troop seats are installed side by side in the cargo compartment and are fastened by quick-release pins to fittings on the station 124.00 bulkhead. (See fig. 2-11.) Each troop seat consists of a tubular frame that supports the seat portion of the canvas seat and backrest, two legs, mounting brackets, two backrest attach fittings, and six quick-release pins. Each troop seat can be folded to allow use of the entire cargo floor space.

2-198. Removal — Troop Seats. Pull out six quick-release pins and remove troop seat from fuselage.

2-199. Disassembly — Troop Seats. (See fig. 2-11.) Disassemble (seat removed) only as required to remove or replace defective parts.

a. Remove bolts that attach forward diagonal tube (detail A, sh 1) to legs.

b. Remove bolt that attaches legs to clevis brackets.

c. Remove bolts that attach split attach brackets to legs.

d. Remove bolts from forward horizontal tube; then rotate tube and slide it out of canvas seat loop.

e. Remove aft diagonal tube attaching bolts.

f. Spread left and right tubes and slip split clevis out of aft lateral tube. Slowly withdraw tubes from seat loops.

g. Slide aft horizontal tube out of seat loop.

h. Remove quick-release pin sleeve and disconnect cable and quick-release pin.

2-200. Inspection — Troop Seats. *a.* Inspect clevis brackets and split support bracket for security, cracks and corrosion. Inspect quick-release pins for distortion, corrosion, or any visible damage.

b. Inspect horizontal and lateral tubes and forward and aft diagonal tubes for security of all members. Inspect for distorted, cracked, broken, and corroded members, and paint deterioration.

c. Inspect canvas seat and bracket for damage and excessive wear (TM 55-1500-204-25/1).

d. Inspect backrest slotted hooks for breaks, cracks, or distortion, and security.

e. Inspect troop seat brackets for wear. (Refer to para 2-201 for repair of brackets.)

2-201. Repair — Troop Seats and Brackets. (See fig. 2-11 and 2-12.) *a.* Rework all worn areas to remove a sharp edge. Reworked areas should be well radiused into the surrounding area.

b. Brackets not within the limits shown on figure 2-12 after rework must be replaced.

c. Replace defective quick-release pins.

d. Replace warped tubes.

e. Replace cracked, broken, deformed, corroded, or otherwise damaged brackets, clevises, or slotted hooks.

f. Replace canvas seat and backrest, if damaged beyond repair.

g. Remove minor faults and imperfections in tube members by polishing to restore original finish. Repair any damage not removable by light polishing by splicing of damaged tube sections (TM 55-1500-204-25/1).

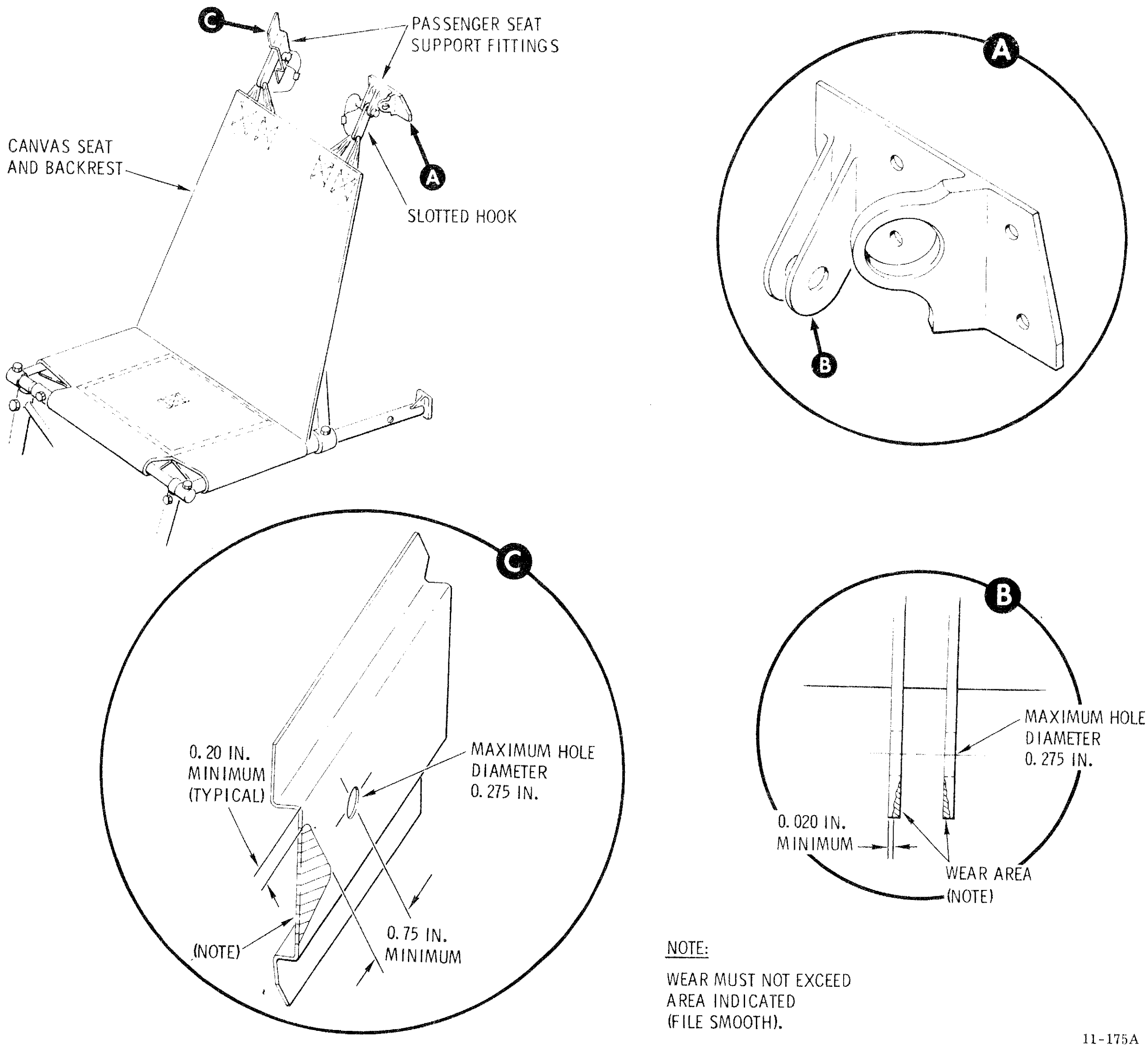


Figure 2-12. Repair Limits for Troop Seat Brackets.

2-202. Reassembly — Troop Seats (See fig. 2-11.) *a.* Slide aft horizontal tube into aft seat loop.

b. With clevis installed on horizontal tube, slip left and right lateral tubes into corresponding seat loops and seat clevises in aft horizontal tube. Install bolts.

c. Slide forward horizontal tube through lateral tube, forward clevis and forward seat loop.

d. Assemble legs, forward diagonal tube and split attach brackets. Install bolts.

e. Mount legs to clevises and install bolts.

f. Install upper slotted hooks to mounting straps of canvas seat back.

g. Install seat.

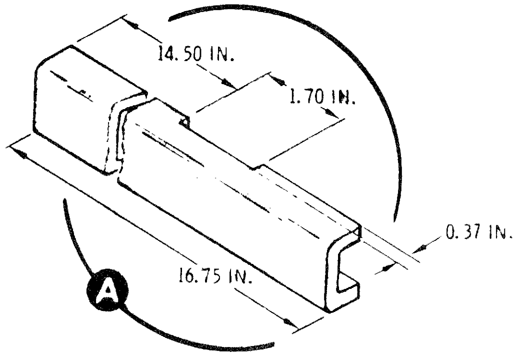
2-203. Installation — Troop Seats. *a.* Using two quick-release pins, secure horizontal tubes to fittings on the station 124.00 bulkhead.

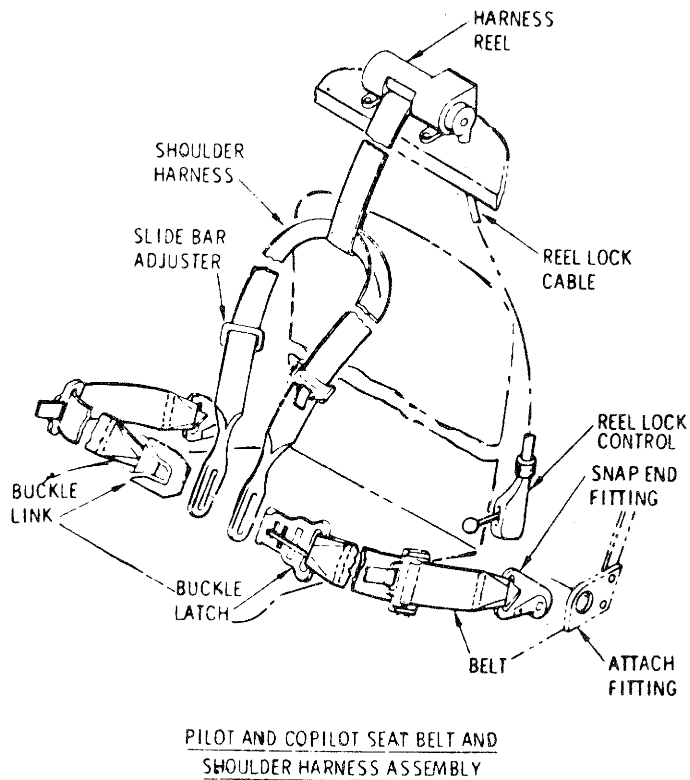
b. With two quick-release pins, secure backrest slotted hooks to fittings on the station 124.00 canted frame.

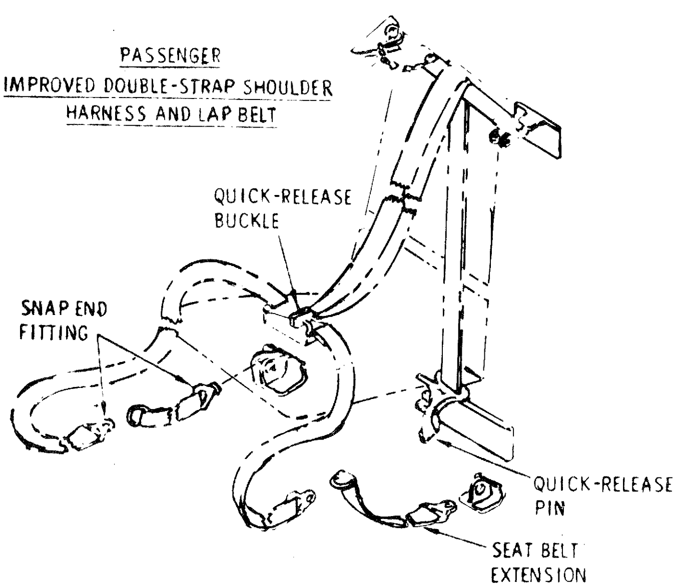
c. Using quick-release pins, secure leg brackets to channel on the cargo compartment floor.

2-204. SEAT BELTS, SHOULDER HARNESSES, AND CRASH PADS.

2-205. Description — Seat Belts, Shoulder Harnesses, and Crash Pads. Seat belts and shoulder harnesses are provided for the pilot, copilot, and two







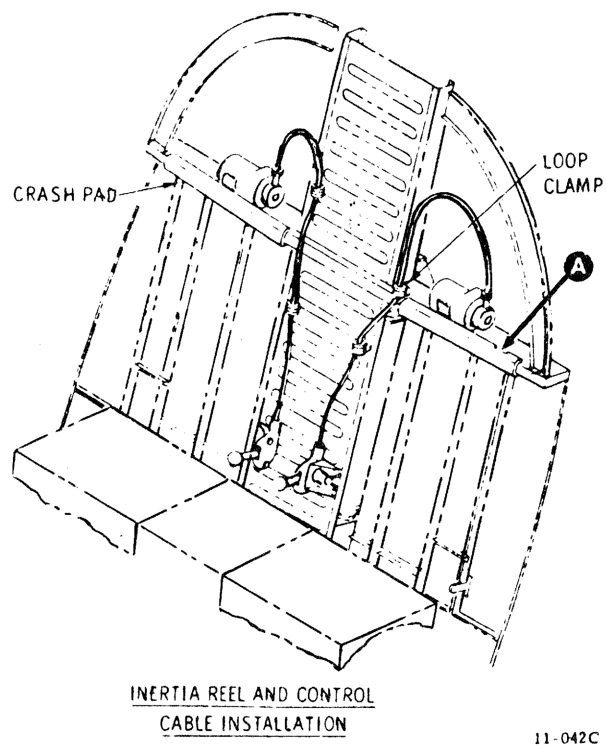


Figure 2-13. Seat Belt,Shoulder Harness,and Crash Pads.

passengers. (See fig. 2-13.) Each seat belt/harness combination consists of two nylon-webbing belts and a nylon-webbing shoulder harness. Fach half of the pilot's seat belt incorporates a length adjuster. Adjusters are also built into all of the shoulder harnesses. The pilot's/copilot's seat belts fasten by means of a link and latch type buckle. The pilot's/copilot's shoulder harness attach to inertial reels mounted on the canted bulkhead behind the seat. On aircraft not modified for the improved seat belt/harness for the passengers, the seat belt/harness incorporates snap end fittings for ready installation and removal, and has a quick-release friction type buckle. On aircraft so modified, the seat belt incorporates seat belt extensions and snap end fittings, and the shoulder harness is attached by a quick-release pin at the firewall bulkhead. The seat belt and harness join together in a snap-over type buckle. All belts are attached to bulkhead fittings with snap end fittings. Harness adjuster springs located on all shoulder harness and seat belts except pilot and co-pilot shoulder harness.

2-206. Removal — Troop Standard Harness and Pilot's Belt. Remove seat belts and shoulder harness by unfastening one harness and two belt snap end fittings from attachment fittings at the bulkhead.

2-207. Removal — Troop Positive Restraint Harness. *a.* Remove seat belt extensions by unfastening extension snap-end fittings, one at each side of the seat, from bulkhead attachment fittings. Belt extensions may be unsnapped from the seat belt end snaps.

b. Remove the shoulder harness by removing the quick-release pin at the floor attachment fitting under the seat. (If necessary, remove quick-release pin at inboard support beam fitting to free harness for removal. Remove harness, re-position support beam, and attach quick-release pin at inboard support beam fitting.)

2-208. Inspection — Seat Belts, Shoulder Harnesses, and Crash Pads. *a.* Inspect seat belts and nylon webbing shoulder harnesses for installation date, worn or frayed condition and loose stitching (TM 55-1500-204-25/1.)

b. Inspect snap end fittings for cracking, wear, or deformation.

c. Inspect harness adjusters for ease of operation.

d. Inspect buckle links and latches for wear, deformation, and correct latching and release.

e. Inspect all seat belt attachment fittings for wear and deformation.

f. Inspect crash pads for serviceable condition. (Refer to para 2-169 for rebonding or replacement of crash pads.)

2-209. Cleaning and Repair — Seat Belts and Shoulder Harnesses. Refer to TM 55-1500-204-25/1.

2-210. Repair — Crash Pads. Damaged crash pads cannot be repaired, but must be replaced. Remove damaged crash pads. See figure 2-13 for crash pad installation. Refer to paragraph 2-173 for replacement of crash pads.

2-211. Installation — Troop Standard Harness and Pilot's Belt. Install seat belts and shoulder harness by fastening one harness and two belt snap end fittings to bulkhead attachment fittings.

2-212. Installation — Troop Positive Restraint Harness. *a.* Pass double strap shoulder harness over bulkhead support beam above the seat back. (It may be necessary to swing support beam out from bulkhead by removing the quick-release pin from the inboard support beam fitting. After harness is positioned over the support beam, swing support beam back into place, and reinstall quick-release pin.) Attach harness end fitting to floor attachment fitting under the seat, using the quick-release pin at that position.

b. Snap seat belt extension onto seat belt end snaps. Fasten belt extensions to bulkhead attachment fittings under the seat, using extension snap-end fittings.

2-213. PILOT'S INERTIA REEL AND SHOULDER HARNESS.

2-214. Description — Pilot's Inertia Reel and Shoulder Harness. The inertia reel assembly consists of a harness reel, and a reel lock cable and control (fig.2-14). On aircraft without armor installation the reel lock cable is routed directly downward from the reel to the reel lock control located to the left of each pilot's seat. Identical inertial reel assemblies are mounted on the canted bulkhead behind the pilot and copilot. On aircraft with armor installation, the control cables loop up and then down to the reel lock controls at the pilot's left and the copilot's right. The shoulder harness is secured to the inertia reel by a web retaining insert that fits flush into the main shaft groove. The main shaft is spring-loaded counterclockwise. The power spring is installed within the inertia reel case and is contained by the spring shield and retaining ring. A hexagonal socket is provided in the main shaft to assist in shoulder harness removal and installation, and covered by an access tab mounted on the spring shield.

2-215. Operational Check — Pilot's Inertia Reel and Shoulder Harness. *a.* Move lever of reel lock control to MANUAL LOCK. Check that the harness strap cannot be pulled out of harness reel.

b. Move control lever to AUTOMATIC LOCK. Check that harness reeling is unrestrained into and out of reel.

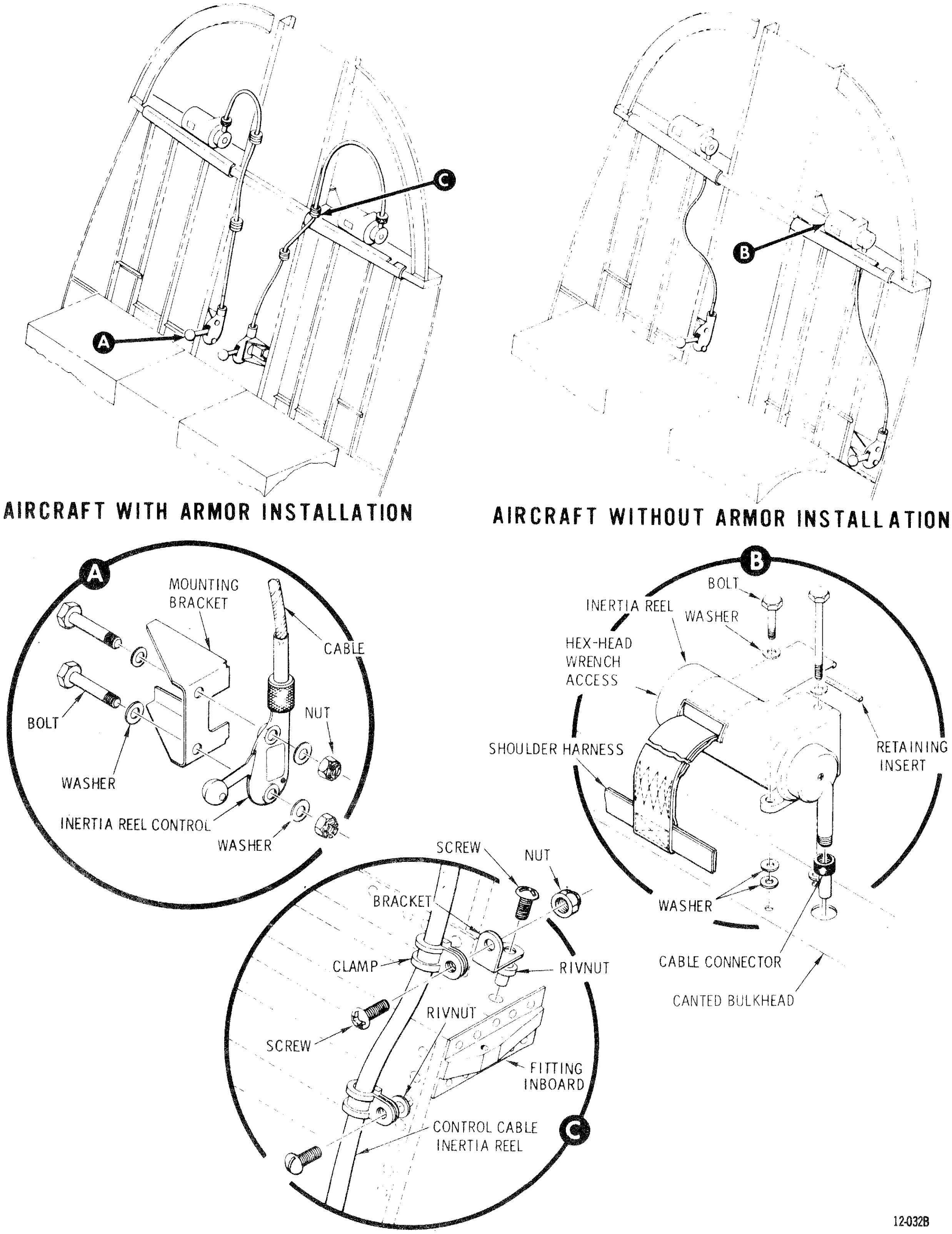


Figure 2-14. Inertia Reel and Manual Control

c. With the control lever at AUTOMATIC LOCK, gently pull the harness strap out of harness reel; then jerk on the strap. Check that the sudden jerk results in automatic locking of the harness strap, limiting the forward travel of the strap to no more than an additional 0.5 inch.

d. Release tension. Check that strap automatically winds back into harness reel.

2-216. Removal — Pilot's Shoulder Harness. (See fig. 2-14.) *a.* Pull shoulder harness fully outward from inertia reel.

CAUTION

Use care to prevent the Allen wrench from slipping out of the main shaft socket while the power spring is in tension.

b. Remove access tab and insert a 7/32-inch hex-head wrench into main shaft socket to hold reel in position.

c. Push end of shoulder harness from main shaft, and remove web retaining insert.

d. Remove shoulder harness from inertia reel while maintaining tension on Allen wrench.

e. Slowly release tension on hex head wrench allowing power spring to unwind fully.

2-217. Removal — Pilot's Inertia Reel. (See fig. 2-14.) *a.* Remove 4 bolts and 12 washers that attach reel to structure.

b. Move reel upward until knurled cable connector is accessible.

c. Loosen knurled connector and lift cable from recess in control shaft.

d. Remove inertia reel.

2-218. Inspection — Pilot's Inertia Reel and Harness Assembly. *a.* Check harness reel for security of installation. Inspect reel housing and control head for dents, breaks or cracks, corrosion, or any other visible damage.

b. Inspect the reel lock cable sheath for damage. Check that cable knurled nuts are tight at the control head of the reel and at the reel lock control.

c. On aircraft equipped with armor, inspect the control cable clamps for security on the canted bulkhead (tunnel).

d. Inspect harness strap of reel for installation date, worn or frayed condition and loose stitching (TM 55-1500-204-25/1).

2-219. Cleaning and Repair — Pilot's Shoulder Harness. Refer to TM 55-1500-204-25/1.

2-220. Repair — Pilot's Inertia Reel. *a.* Polish out minor nicks and scratches from aluminum and steel parts with grade 400 abrasive paper (C3).

b. Restore protective finish to aluminum parts by applying chemical film (C20).

c. If necessary, renew finish on case with a wash coat of primer (C79) and two coats of lacquer (C54).

d. Refer to TM 55-1500-204-25/1 for internal repairs.

2-221. Installation — Pilot's Inertia Reel. (See fig. 2-14.) *a.* Install cable end retainer in inertia reel control shaft and tighten knurled connector.

b. Place inertia reel in mounting position on canted bulkhead.

c. Install mounting bolts with two washers (one thick and one thin) under inertia reel and one (thin) washer under the bolt head at each of the four mount points.

d. Install shoulder harness (para 2-222).

2-222. Installation — Pilot's Shoulder Harness. (See fig. 2-14.) *a.* Temporarily install shoulder harness through main shaft and turn shaft clockwise with 7/32-inch hex-head wrench until harness is wound on main shaft. Note number of turns made with hex-head wrench.

b. Unwind inertia reel slowly with hex-head wrench and remove shoulder harness.

CAUTION

Prevent hex-head wrench from slipping out of main shaft socket while power spring is in tension. Sudden slippage could damage the inertia reel mechanism.

c. Insert hex-head wrench into main shaft socket; turn counterclockwise by the number of turns noted in *a* above plus approximately one more turn, until main shaft is aligned with insertion end of shoulder harness.

d. Push shoulder harness through main shaft while maintaining tension on hex-head wrench, and install web retaining insert into shoulder harness.

e. Pull shoulder harness back through main shaft until web retaining insert is seated flush in main shaft groove; then hold. Remove hex-head wrench and slowly allow shoulder harness to wind on main shaft.

f. Install access tab.

g. Perform operational check (para 2-215).

2-223. INERTIA REEL MANUAL CONTROL.

2-224. Removal — Inertial Reel Manual Control. (See fig. 2-14.) *a.* Place control lever in forward (locked) position to move drive control shaft toward cable attachment end of control case. Loosen knurled cable connector and lift cable retaining ring from recess in drive control shaft.

b. On aircraft with armor installation, remove screws that attach cable clamps to structure and remove cable.

CAUTION

Do not attempt to disassemble cable.

c. Remove two nuts, bolts, and washers that attach control unit to structure.

d. Remove inertia reel control unit.

2-225. Inspection — Inertia Reel Manual Control. *a.* Check manual control for security of installation. Inspect control housing for dents, breaks or cracks, corrosion, or any other visible damage.

b. Inspect the control cable sheath for damage. Check that cable knurled nuts are tight at both ends.

2-226. Repair — Inertia Reel Manual Control. *a.* Polish out minor nicks and scratches from aluminum and steel parts with grade 400 abrasive paper (C3).

b. Restore protective finish to aluminum parts by applying chemical film (C20).

c. If necessary, renew finish on control case with a wash coat of primer (C79) and two coats of lacquer (C54).

2-227. Installation — Inertia Reel Manual Control. (See fig. 2-14.) *a.* Install inertia reel control unit on structure by using two bolts, washers, and nuts.

b. Place control lever in the forward (locked) position.

c. Install retainer in recess in the control shaft; then tighten down knurled cable connector.

d. Operate control lever through several cycles to assure free movement.

e. Perform operational check (para 2-215).

2-228. CANOPY AND WINDOW INSTALLATION.

2-229. Description — Canopy and Window Installation. The canopy installation forms the forward part of the fuselage airframe enclosing the pilot's compartment. Windows include the polycarbonate plastic 0.040-inch thick pilot's and cargo door windows and 0.080-inch thick aft compartment windows. The canopy installation consists mainly of three sections; lower windshield, center windshield, and upper windshield. Each windshield consists of symmetrical left and right panels extending from the windshield centerline to the edges of the pilot's door frames. The center and lower windshield sections are clear acrylic or polycarbonate. The upper section is tinted a smoke-gray. On aircraft serial No. 65-12916 through 65-12934, all original canopy windshield panels are heat-resistant (cast) acrylic. On subsequent aircraft, the center and lower canopy windshield panels may be either stretched acrylic or polycarbonate. The two center windshield panels are approximately 0.080-inch thick and all other panels are approximately 0.060-inch thick.

NOTE

Polycarbonate windshields can be identified by the lack of geon attachment strips.

2-230. Inspection — Canopy and Window Installation. (See fig. 2-15.) *a.* Inspect for cracks, scratches, nicks, crazing and discoloration. Cracks are not permitted without repair. Scratches, nicks, crazing, and discoloration, or repair thereof, must not restrict operator vision.

b. Inspect windshield panels for a good seal around all edges, and security of the fairing tape.

2-231. Serviceability Criteria — Canopy and Window Installation. Refer to figure 2-15 for classification of damage and repair or replacement criteria for both types of plastic.

2-232. Repair — Canopy and Window Installation. Acrylic panels that are damaged beyond negligible limits may be repaired by using the method appropriate for the type of damage. Typical repairs and detailed procedures will be found in TM 55-1500-204-25/1. Both the stretched acrylic and cast acrylic panels are repaired by using the same methods and repair materials. The materials are cellulose tape (C18) for masking and adhesive (C8) for patching. These materials and the "glue method" of cementing should be used for all patch type repairs.

a. Repair Polycarbonate windows and/or windshields as follows:

(1) Stop drill at each end of crack using a number 40 drill.

(2) Lightly sand area to be repaired with sandpaper (C3) about 0.750 inch beyond crack for adhesive application.

(3) Wipe area with dry, clean cloth to remove residue.

(4) Cut section of fiberglass (C39A) to extend a minimum of 0.500 inch around crack or area to be repaired.

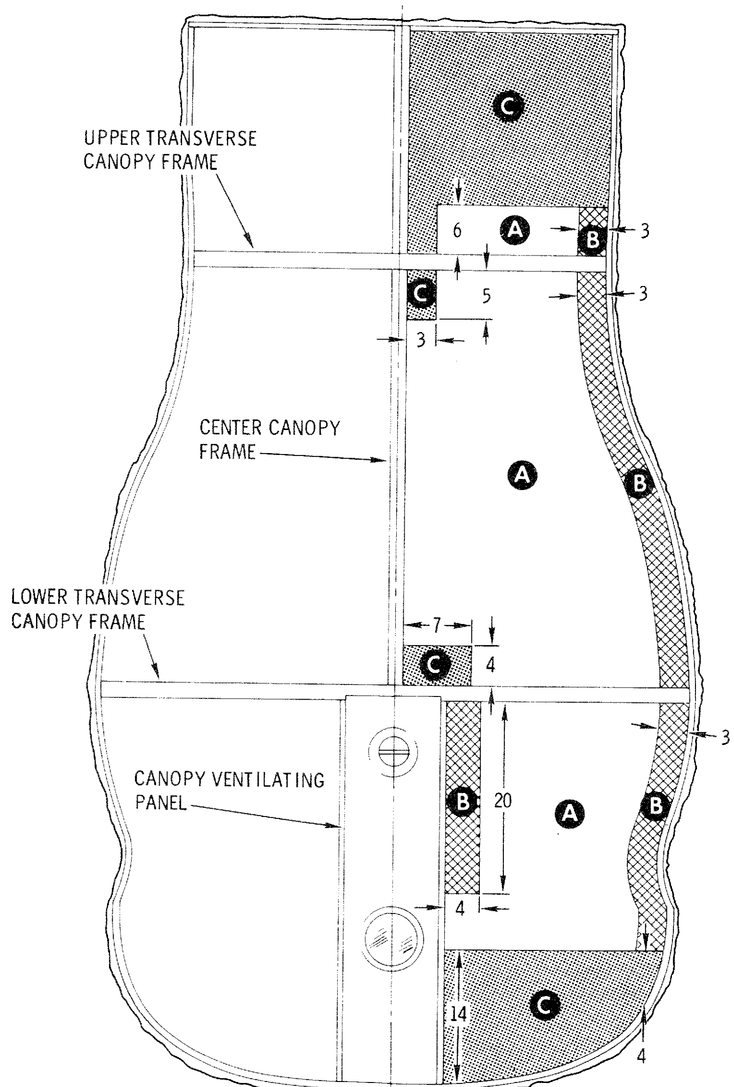
(5) Apply one brush coat of urelane adhesive (C7) to repair area and lay fiberglass over repair area.

(6) Rub patch lightly to assure adhesion to repair area and brush another coat of adhesive over area.

(7) Allow to cure for 48 hours. (Handling strength is developed in 24 hours.) Accelerated cure time will be two hours under a heat lamp at 71°C (160°C).

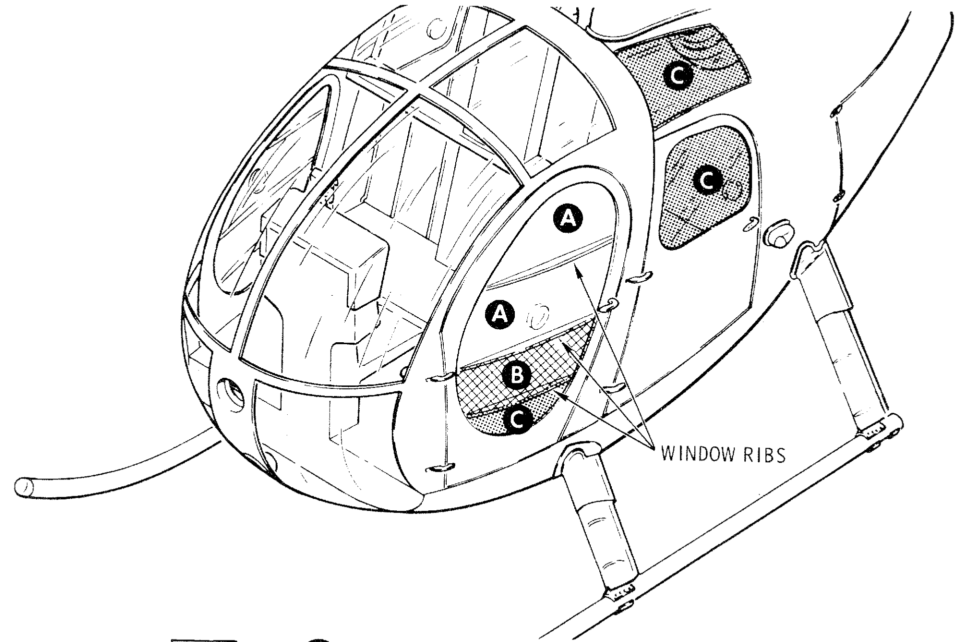
b. Repair water leaks by using a small bead of sealing compound (C89) on the affected area.

2-233. Replacement — Canopy Panels. (See fig. 2-16.)



CANOPY WINDSHIELD AREAS
FLAT PLANE VIEW

LEFT SIDE SHOWN
(ALL DIMENSIONS ARE IN INCHES)



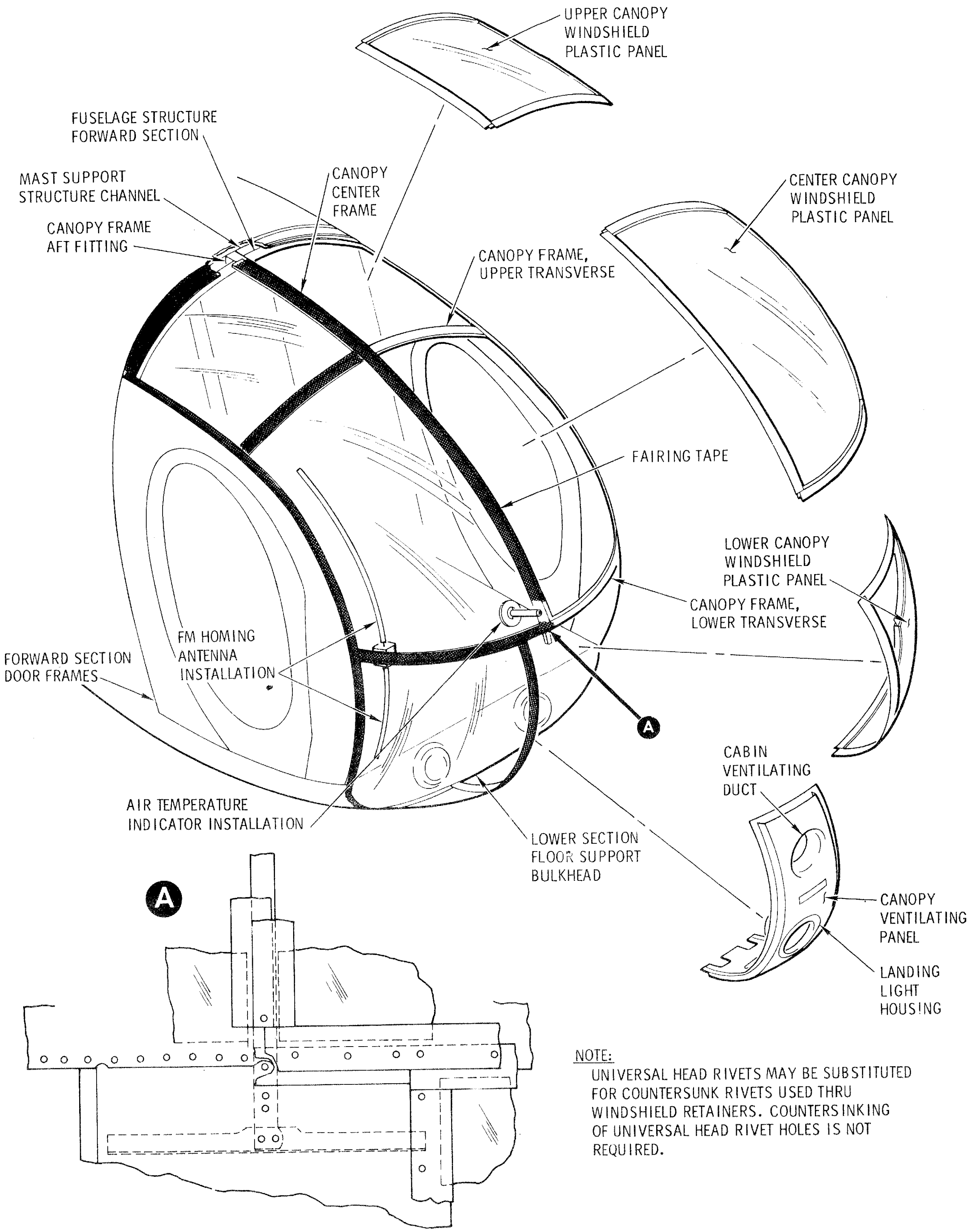
AREA A — SCRATCHES AND PITS MAY BE POLISHED OUT TO THE EXTENT THAT VISION IS NOT DISTORTED. DISTORTION OF VISION IS CAUSE FOR REPLACEMENT. CRACKS, HOLES OR OTHER DAMAGE MAY BE TEMPORARILY REPAIRED, IF VISION OF CREW MEMBERS WILL NOT BE IMPAIRED, BY STOP DRILLING, PATCHING OR OTHER APPROVED METHODS (REFER TO TM 55-1500-204-25/1), BUT WINDOW MUST BE REPLACED AT THE EARLIEST OPPORTUNITY.

AREA B — SCRATCHES AND PITS ARE PERMITTED IN THIS AREA PROVIDED THEY ARE NOT SO NUMEROUS OR FORM SUCH A PATTERN AS TO BE OBJECTIONABLE TO THE VIEWER. CRACKS, HOLES OR OTHER DAMAGE MAY BE REPAIRED BY STOP DRILLING, PATCHING OR OTHER APPROVED METHODS (REFER TO TM 55-1500-204-25/1), AS LONG AS SUCH REPAIRS PRESENT NO OBJECTIONABLE PATTERN TO THE VIEWER.

AREA C — SCRATCHES AND PITS ARE PERMITTED IN THIS AREA, PROVIDING THE STRUCTURAL INTEGRITY OF THE WINDOW IS NOT IMPAIRED. CRACKS, HOLES OR OTHER DAMAGE MAY BE REPAIRED BY STOP DRILLING, PATCHING OR OTHER APPROVED METHODS (REFER TO TM 55-1500-204-25/1) PROVIDED STRUCTURAL INTEGRITY IS NOT IMPAIRED.

11-185

Figure 2-15. Canopy and Window Serviceability Criteria.



12-030B

Figure 2-16. Canopy Installation.

NOTE

The individual center and lower windshield panels of cast acrylic material should be replaced with cast acrylic spares, if available. If only stretched acrylic or polycarbonate replacement panels are available, all four cast panels must be changed to the stretched acrylic or polycarbonate type. Stretched acrylic and polycarbonate panels are interchangeable on an individual basis.

a. Accomplish *b* through *e* before windshield replacement, as appropriate to the windshield panel being replaced.

b. Remove fm homing antenna components (TM 11-1520-214-20 or -20-1), as applicable.

c. Remove magnetic compass and outside air temperature indicator from the canopy.

d. Protect torque pressure and static tubes to prevent drilling damage.

e. Fashion a cloth cover above the cockpit instrument panel, the console, and over pilot's seats to catch removed rivets in areas where rivets cannot be drilled out from the inside.

CAUTION

The plastic canopy panels become brittle when cold and could crack during handling. The following procedures should be performed under warm room temperature conditions (70°F or higher).

NOTE

The left center windshield inboard retainer overlaps the right center windshield inboard retainer; these inboard retainers are riveted to the canopy frame. The aft (upper) retainers of the center windshield assemblies overlap the forward retainers of the upper windshield, and are riveted to the canopy frame. The lower retainers of the center windshield overlap the upper retainers of the lower windwhield as well as the upper joggled edge of the fiberglass canopy panel.

f. Remove fairing tape and, in order, drill out rivets that attach: windshield outboard retainer to door frame; windshield inboard retainers to centerline canopy frame; aft (upper) retainer of windshield to canopy frame; and forward (lower) retainer of windshield to canopy frame. Slide windshield panel free, as applicable; push up and out to remove.

g. Lay replacement windshield panel against canopy frame. Position panel for correct overlap. Panels with retaining strips should be positioned so that drill holes are through the retainer.

h. Mark and trim windshield to fit canopy opening.

i. Check window contour for fit at canopy ventilating panel upper corner. Install tapered shim, as required.

NOTE

Canopy panels may be replaced by using screws (MS35218-27), nuts (MS20364D632A) and washers (AN960PD6L) in place of rivets.

j. Drill holes using the canopy frame as a guide. Install fasteners to hold panels in position. Countersink holes for flush head rivet installation as required.

k. Remove windshield from frame and apply a bead of sealing compound (C88), to lip of retainers for stretched acrylic panels and for polycarbonate panels.

l. Enlarge all rivet holes in polycarbonate panels to 0.031 inch oversize. Use washers under both sides of attachment rivets.

m. Use masking tape on universal rivet sets to prevent marking panels; rivet replacement windshield panels to the door frame and canopy frames. MS20470A4 and MS20426A3 rivets are used in most places.

NOTE

Care should be exercised during rivet installation to drive rivets as lightly as possible. Over-driving of the rivet can cause star crazing. Star crazing within 0.50 inch of rivet is not cause for replacement of panel.

n. Remove excess sealant around retainers by using a scraper made of wood or plastic.

o. Clean acrylic retainer surfaces with naphtha (C70). Polycarbonate retainers are cleaned with isopropyl alcohol (C82). Install 2-inch wide tape (C103) over retainers and rivet patterns.

p. Install magnetic compass and outside air temperature indicator, as applicable.

q. Install fm homing antenna components (TM 11-1520-214-20 or -20-1), as applicable.

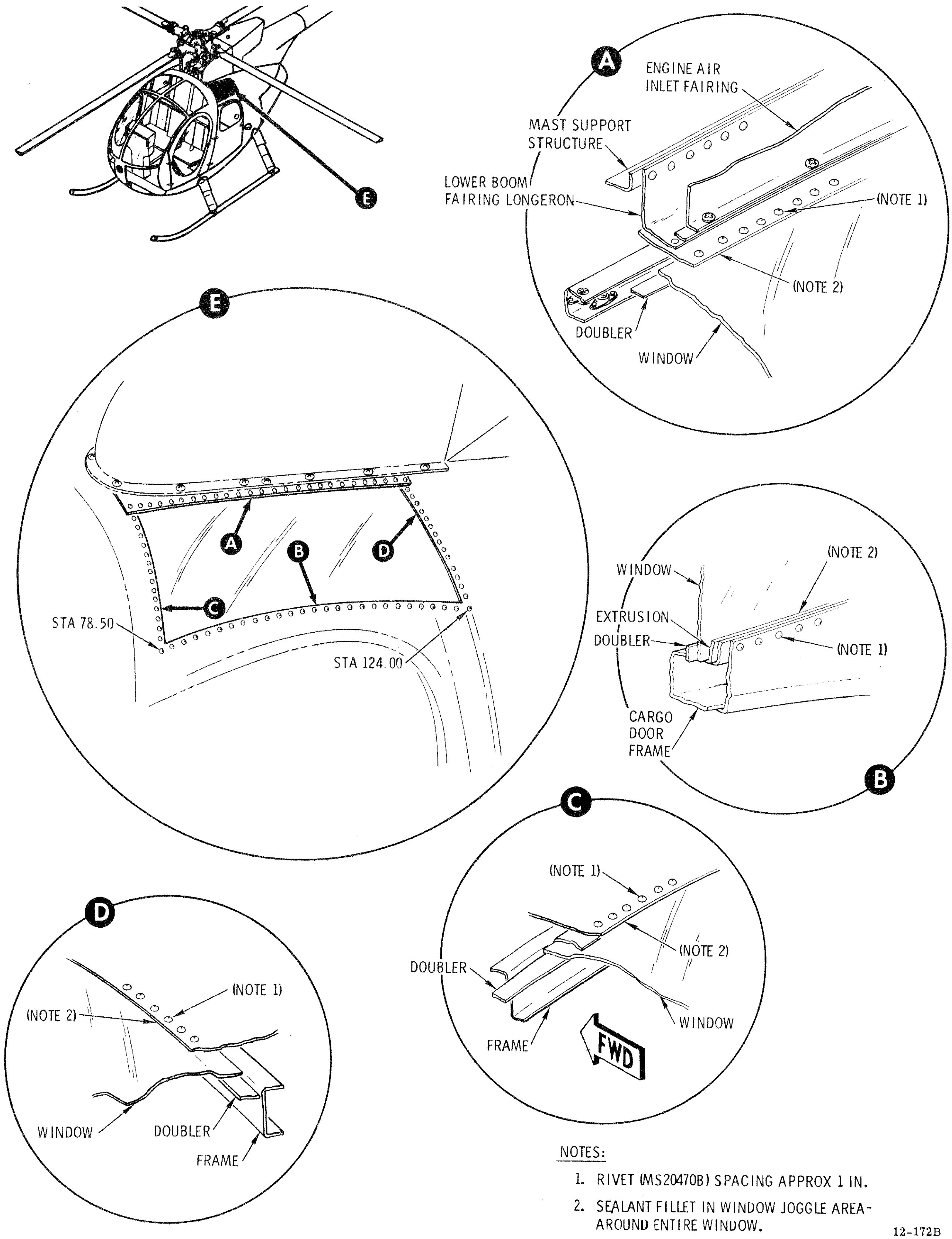


Figure 2-17. Aft Compartment Window Installation.

2-234. Replacement — Aft Windows. (See fig. 2-17.) *a.* Drill out rivets that secure window to the cargo door frame, canted station 78.50, and canted bulkhead station 124.00. Remove doublers.

b. Remove engine air inlet forward fairing if necessary to clear upper rivet pattern (para 2-101).

c. Drill out rivets that hold window and doubler to fuselage.

d. Break sealant bond between window and structure; remove window from inside cargo compartment.

e. Remove any sealant residue from structure by peeling off the particles.

f. Remove the protective covering from edges of the replacement window.

CAUTION

Handle window with care to avoid scratching the polycarbonate plastic.

g. Clean the window edges and structural framework mounting surfaces with nonabrasive soap or detergent and wipe with clean damp cloth.

h. Trim and fit window prior to drilling by using old window as pattern, if possible.

i. Lay replacement window in position and drill rivet holes in the window by using the existing structure rivet hole pattern as a template.

CAUTION

Use care when drilling and riveting to avoid damaging the instrument lines and cables that run alongside the mast support structure and through the substructure near the forward edge of the window.

j. Rivet window and doublers in place.

k. Apply sealing compound (C89), to fillet area around entire window.

2-235. CANOPY ENCLOSURE SUPPORTING STRUCTURE.

2-236. Description — Canopy Enclosure Supporting Structure. The frame to which the windshield and the cabin ventilating panel are mounted (fig.2-16) consists of a longitudinal frame located on the fuselage centerline, a transverse upper frame located at canted bulkhead station 74.25, and a transverse lower frame located at waterline 38.64. The canopy is secured to the fuselage structure by: the canopy frame aft fitting riveted to the mast support structure channel; the upper windshield retainers riveted to the fuselage structure forward section weld; the windshield outboard retainers riveted to the left and right section door frames; and the lower windshield retainers and lower edge of the cabin ventilating panel riveted to the floor bulkhead.

2-237. Inspection — Canopy Enclosure Supporting Structure. Inspect the supporting structural elements and canopy framework for cracks, deformations, dents, signs of corrosion, and overall general condition.

2-238. Replacement — Canopy Enclosure Supporting Structure. *a.* Remove the canopy windshields in the following order: center canopy windshields, upper canopy windshields, lower canopy windshields (para 2-233), and the cabin ventilating panel (para 2-239).

b. Remove fasteners, as required, and disconnect: cabin heat and defogging outlets; engine anti-ice and cabin heat control levers; instrument panel braces; magnetic compass support bracket; and the engine torque pressure and instrument static lines.

c. Protect all disconnected components against drill damage and drill out rivets that attach canopy frame to structure.

d. Push canopy frame up and out to remove.

e. Install fasteners to attach centerline frame to the fuselage structure.

NOTE

MS20426A rivets are used in most locations.

f. Install fasteners to attach upper and lower (transverse) canopy frames to left and right door frames.

g. Complete installation of canopy frame by riveting fastener-held locations.

h. Install canopy ventilating panel (para 2-239).

i. Install, in order: lower canopy windshield, upper canopy windshields, and center canopy windshields (para 2-233).

2-239. CABIN VENTILATING PANEL.

2-240. Description — Cabin Ventilating Panel. The ventilating (canopy) panel (fig. 2-16) is fabricated from several laminations of fiberglass, fiberglass reinforcements, and aluminum inserts. Classification of damage and repair or replacement criteria are as outlined below and in paragraph 2-244.

2-241. Negligible Damage — Cabin Ventilating Panel. Refer to paragraph 2-244.

2-242. Repair — Cabin Ventilating Panel. Repair heavy scratches, cracks or punctures, and delamination beyond negligible limits but confined to small areas within the perimeter defined by panel doublers, according to paragraph 2-244.

2-243. Replacement — Cabin Ventilating Panel. (See fig. 2-16.) *a.* Disconnect ventilating control valve (chapter 13), and remove landing light lamp and pitot tube.

b. Using the following sequence, drill out rivets attaching:

(1) Lower windshield to canopy panel.

(2) Center windshield retainers over lower windshield retainers to canopy frame.

(3) Lower corners of center windshield retainers to canopy frame. (Insert a wedge between each lower windshield retainer and canopy frame.)

(4) Canopy panel to underside of flange of the canopy frame; use care to prevent drilling through wedged upper windshield retainers. (Drill from inside the cabin.)

(5) Lower flange of canopy panel to floor bulkhead.

(6) Centerline portion of the canopy panel to the lower portion of canopy frame.

c. Push the panel lower edge away from flange of floor bulkhead, slide panel downward, and remove from canopy.

d. Position replacement canopy panel in canopy with upper left, and right mounting flanges under the canopy frame and the lower windshield inboard retainers. Seat lower mounting flange of the panel inside contoured flange of floor bulkhead.

e. Rivet components in the following sequence:

(1) Centerline portion of canopy panel to lower portion of canopy frame.

(2) Lower mounting flange of panel to contoured flange of floor bulkhead.

(3) Upper mounting flange of panel to canopy frame.

(4) Lower windshield retainers to canopy frame. (Remove wedges from under upper windshield retainers.)

(5) Lower windshield retainers to canopy panel.

(6) Lower half of center windshield retainers to canopy frame.

f. Install ventilating control valve (chapter 13).

g. Install landing light lamp and pitot tube.

2-244. TYPICAL FIBERGLASS REPAIRS.

2-245. General — Typical Fiberglass Repairs. Classification of damage and repair or replacement criteria for laminated fiberglass components are as given below, and in paragraph 2-239.

2-246. Negligible Fiberglass Damage. Minor defects or imperfections that are obviously not indications of impending structural failure are considered negligible, and do not require repair. Examples of negligible damage are:

a. Light scratches less than 0.005 inch deep.

b. Delamination in not more than one area, and at least 3.50 inches away from edge of component, structural member, or rivet pattern.

c. Minor nicks and scratches that do not penetrate the fiberglass laminations and do not intersect rivet or screw holes.

NOTE

Fiberglass components should be replaced when damage exceeds 20 percent of the total component area, when structural members bonded within the fiberglass are cracked or badly distorted, or when large sections of sharp compound-curved areas are severely damaged.

2-247. Fiberglass Repair — General. A typical fiberglass repair is shown in figure 2-18. The repair procedures apply to fiber-laminate materials (fiberglass) conforming to Military Specification MIL-C-9084, Type VIII (No. 181). Apply fiberglass finish paint as specified in chapter 1. Relative humidity and ambient temperature should approximate the following for best repair results.

HUMIDITY (PERCENT)	AMBIENT TEMPERATURE (DEGREES F)	(DEGREES C)
75	70	21.1
71	75	23.9
68	80	26.7
65	85	29.4
61	90	32.2
58	95	35.0

2-248. Fiberglass Repair — Temporary. If patch or insertion repairs are not feasible, temporary repairs may be made by applying catalyzed resin (C39) to an aluminum patch 0.025-inch thick and installing the temporary patch over damaged area. Permanent types of repair or replacement of fiberglass should be accomplished as soon as possible.

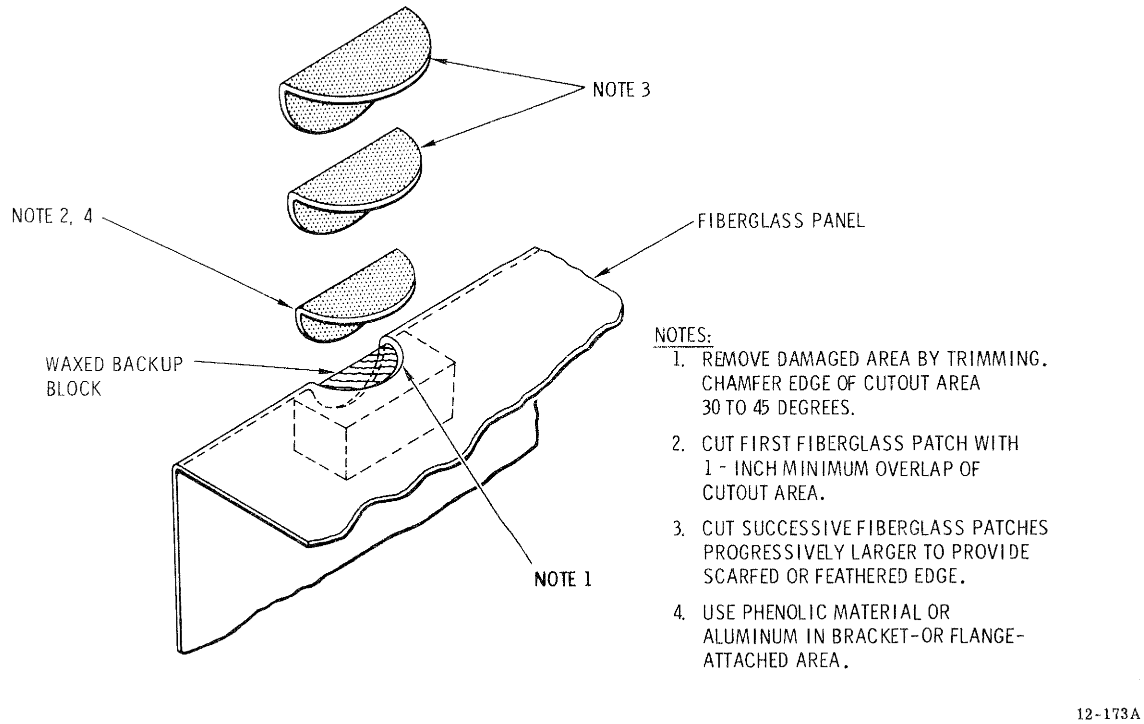


Figure 2-18. Typical Laminated Fiberglass Repair.

NOTE

Do not use temporary aluminum patches where antennas are installed.

2-249. Fiberglass Repair — Delaminations. Work catalyzed resin (C39) between plies, apply pressure to force plies together, and cure according to container instructions.

2-250. Fiberglass Repair — Small Nicks, Scratches, or Pores. Small nicks, scratches or pores may be repaired by working catalyzed resin (C39) into the discontinuity, and curing according to container instructions.

WARNING

Sanding on glass cloth reinforced laminates produces fine dust that may cause skin irritations. Breathing of this dust may be harmful. Observe necessary protective measures.

CAUTION

When maintenance work is being performed hear the engine air inlet, use care to prevent entry of foreign objects. Tape covers of cardboard or other suitable material in place over the engine inlet screen and oil cooler air inlets. Do not remove covers until work is completed and debris is thoroughly cleaned out of the area. After removing covers, verify that area around base of mast, inlet to plenum, and entire plenum chamber is free of foreign material. Install plenum access doors. When performing work above or near upward exhausts, install exhaust covers.

NOTE

Fiberglass repair kit (C39) contains the repair materials itemized in the following procedures.

When accomplishing repair in an area secured by rivets or screws always laminate or bond a metal doubler washer at each attachment point (hole) for reinforcement.

2-251. Fiberglass Repair — Holes or Cracks. *a.* Cut away damaged area to form a square, rectangular or circular opening (fig. 2-18).

b. Trim 30- to 45-degree chamfers at edges on both sides of repair area.

c. Using grade 400 abrasive paper (C3), sand damaged area (both sides, when possible) to form beveled surfaces.

d. Thoroughly clean surface of repair area with a clean, lint-free cloth moistened with methyl ethyl ketone (C69) until resin or cloth is exposed; allow to dry completely.

e. Dry repair area with a clean, lint-free cloth.

f. Cut layers of fiberglass cloth (C39) to uniformly fill repair area. After completely filling repair area, cut a final layer of cloth to completely overlap entire repair area. Remove all cloth patches, maintaining relative placement order of each cloth.

NOTE

Each layer of fiberglass cloth in kit (C39) is approximately 0.010-inch thick after curing. Use sufficient number of layers to at least equal the original thickness of the material being patched. Warming the cloth provides greater flexibility. Do not use a hot iron for warming the cloth.

WARNING

Use protective clothing during the preparation and use of resin mixture. Adequate ventilation must be provided.

g. Prepare resin (C39) according to container instructions. Stir thoroughly to ensure a uniform mixture. Avoid any whipping or beating that might produce air bubbles in the mixture.

NOTE

The resin and catalyst should not be mixed until required because the mixture has a limited pot life. Warming the mixture reduces viscosity and may facilitate handling, but shortens pot life of mixture. In all cases, refer to container instructions on use of resin. Processing and repair work should be performed in an area sheltered from adverse weather conditions and free from dust.

h. Place wax backup block against surface of repair area opposite of working (patch) surface.

i. Install first layer of cloth and brush prepared resin over repair area and cloth, thoroughly impregnating cloth. Scrape away any excess resin. Repeat procedure until repair area is completely filled. Smooth the surface, blending patch into surrounding area, and remove any air bubbles by applying slight pressure when necessary.

j. Cure at room temperature for 24 hours or allow to stand at room temperature (approx 70°F) for one hour, followed by heating at 160°F (71.1°C) for one hour.

2-252. FUSELAGE BODY STRUCTURE.

2-253. Description — Fuselage Body Structure. The body group is an assembly of three major elements: the forward fuselage, lower fuselage and aft fuselage. The tailboom and empennage are discussed further in section 2 of this chapter. (See fig. 2-19.) The airframe primary and secondary structure consists of all metal, metal and fiberglass, and transparent plastic components. The major bulkheads and structural members are shown in figure 2-20.

2-254. Fuselage Body Structure — Metal. The aircraft basic body and fuselage boom are conventional, all metal, riveted structures incorporating formed aluminum alloy, stainless steel and titanium bulkheads, canted frames, channel members, beams, structure rings, ribs, stiffeners, doublers, longerons, and stringers. All stressed skin panels are either smooth or beaded. The stabilizers are all-metal airfoils.

2-255. Fuselage Body Structure — Metal and Fiberglass. The tail rotor blades are fiberglass and stainless steel. The main transmission access door is fiberglass. The compartment access doors are metal or sheet aluminum and glass cloth. The engine inlet fairing is sheet aluminum and glass cloth.

2-256. Fuselage Body Structure — Transparent Plastic. The upper, center, and lower canopy windshields are acrylic plastic. The aft windows, the windows in the passenger-cargo and pilot's compartment doors, main transmission cover, and fuel inlet shield are clear plastic (polycarbonate).

2-257. AIRFRAME STATION LOCATIONS.

2-258. General — Airframe Station Locations. Figure 2-21 identifies major fuselage, boom, and empennage stations.

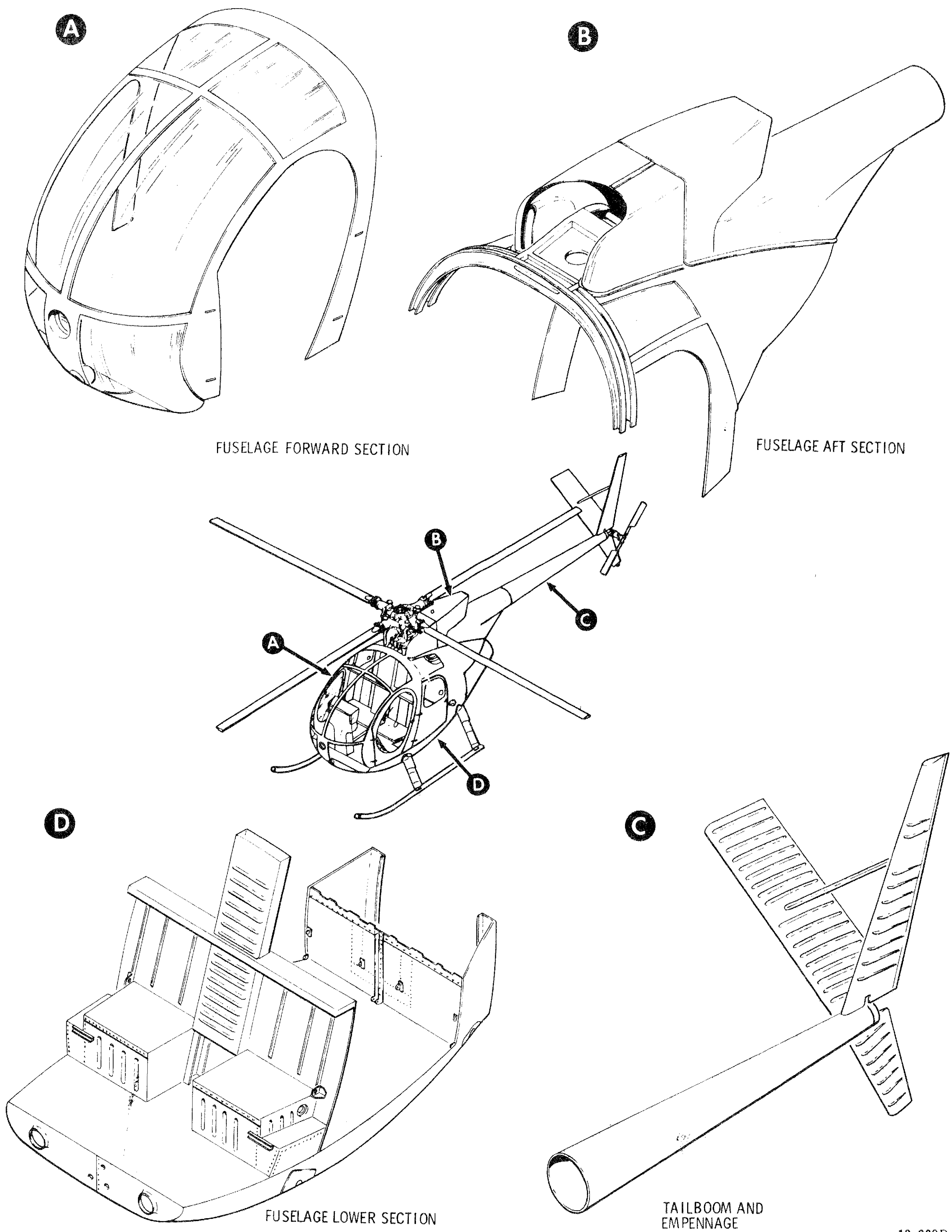
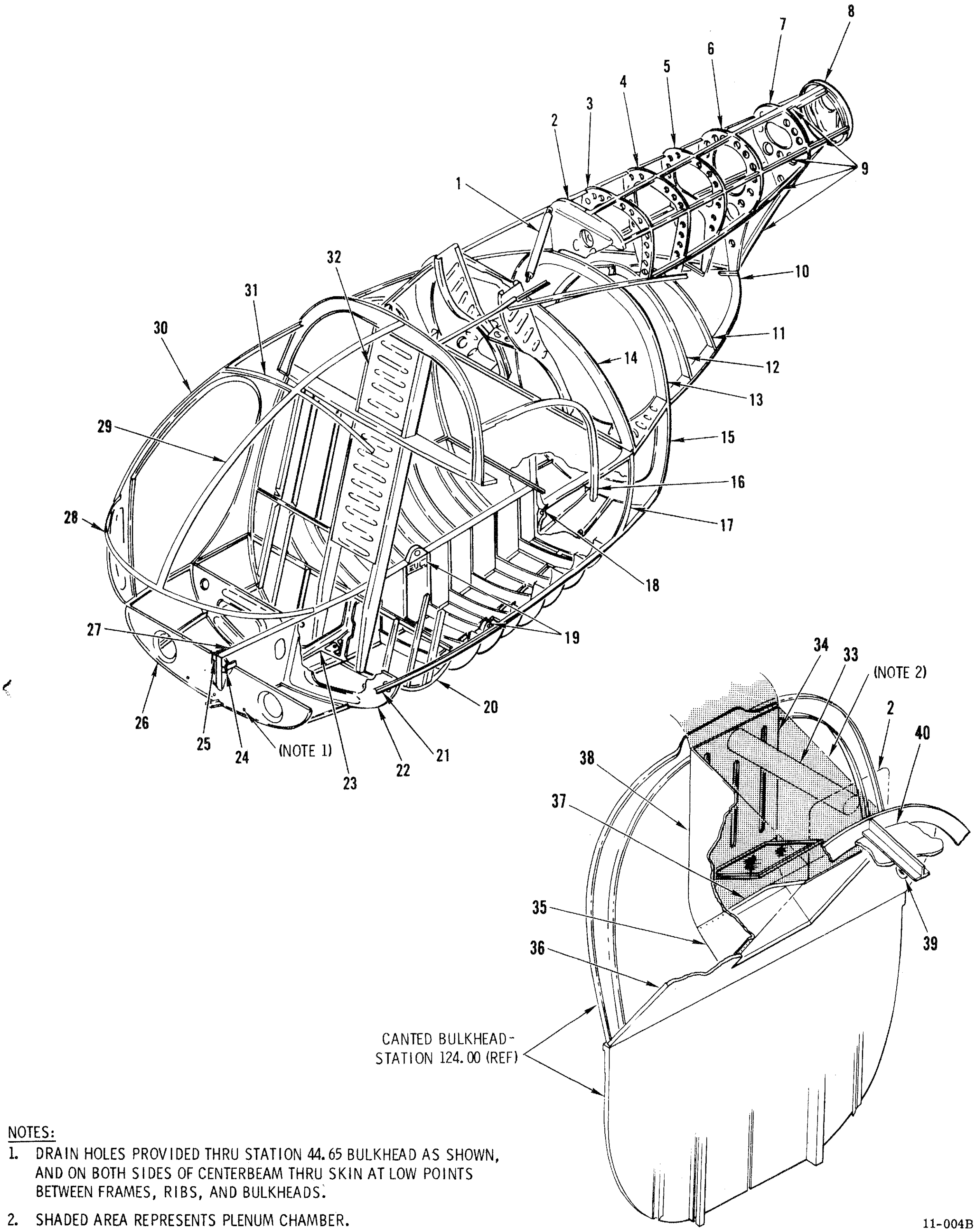


Figure 2-19. Body Group Sections.



NOTES:

1. DRAIN HOLES PROVIDED THRU STATION 44.65 BULKHEAD AS SHOWN, AND ON BOTH SIDES OF CENTERBEAM THRU SKIN AT LOW POINTS BETWEEN FRAMES, RIBS, AND BULKHEADS.
2. SHADED AREA REPRESENTS PLENUM CHAMBER.

Figure 2-20. Body Group Major Structural Sections.

Key to Figure 2-20.

1. Aft section strut
2. Station 137.50 boom fairing ring
3. Station 146.62 boom fairing ring
4. Station 155.75 boom fairing ring
5. Station 164.76 boom fairing ring
6. Station 174.00 boom fairing ring
7. Station 185.89 boom fairing ring
8. Station 197.78 boom fairing canted frame fitting
9. Boom fairing longerons
10. Waterline 34.96 rib
11. Station 155.76 aft section ring
12. Station 146.62 aft section ring
13. Station 137.50 upper aft section ring
14. Station 124.00 upper canted frame
15. Station 137.50 lower section ring
16. Cargo door frames
17. Station 124.00 lower section frame
18. Aft landing gear fitting
19. Armament support fitting assembly
20. Station 78.50 lower canted frame
21. Floor support longerons
22. Pilot's seat structure support bulkhead
23. Forward landing gear fitting
24. Lower bracket support fitting
25. Upper bracket support fitting
26. Pilot's floor support bulkhead
27. Center beam assembly
28. Canopy lower frame
29. Canopy center frame
30. Pilot's door frame
31. Canopy upper frame
32. Station 78.50 upper canted frame
33. Tail rotor shaft fairing
34. Forward engine air inlet panel
35. Lower engine air inlet panel
36. Forward firewall
37. Engine air inlet pan
38. Engine air inlet side channel
39. Engine hoist fitting
40. Aft section upper firewall

2-259. SUPPORT OF STRUCTURE DURING REPAIRS.

2-260. Support — Assembled Aircraft. Refer to paragraph 1-71 for jacking and paragraph 1-74 for hoisting instructions.

2-261. Support — Disassembled Aircraft. Support the aircraft structure with padded cradles (fig. 2-22). Position the cradles exactly under the structural bulkheads.

CAUTION

Placing supports at locations other than directly under a structural bulkhead will result in damage to the aircraft skin panels.

2-262. CLASSIFICATION OF FUSELAGE DAMAGE AND TYPES OF REPAIR.

2-263. General — Classification of Fuselage Damage and Types of Repair. Examine all damage thoroughly, regardless of how trivial it may seem. Investigate for cracks, breaks, scratches, nicks, dents, depressions, punctures, worn spots, chips, and elongated bolt holes, especially after impact. Examine all parts of the structure, as well as the locally affected section, for misalignment, distortion, and other damage. After the extent of damage has been determined, the damage to each part of the affected structure should be examined carefully and classified as: accept as is; negligible damage; damage repairable by patching; or damage requiring replacement of parts. Special jigs or holding fixtures may be required to hold critical dimensions during replacement of fittings or repair of major structural damage. Availability of special jigs or holding fixtures should be considered before attempting major repairs.

NOTE

Excessively damaged castings or forgings, such as hinge brackets, fittings, bolts, or hinge pins, should be replaced. When considering a repair, whether it be on a small part or a complete component, the time involved and the approximate cost should be taken into account, as replacement may be advantageous both in cost and in time involved.

2-264. STRESSED SHEET METAL PANELS (AVIM).

2-265. General — Stressed Sheet Metal Panels (AVIM). Stressed sheet metal panels consist primarily of the fuselage skins (fig. 2-23) and bulkhead webs. The most highly stressed skin sections are those that form the cylindrically tapered tailboom assembly (section 2). Classification of damage, guidelines defining the extent of damage requiring complete replacement, and basic repair methods applicable to stressed sheet metal panels except the tailboom are outlined as follows:

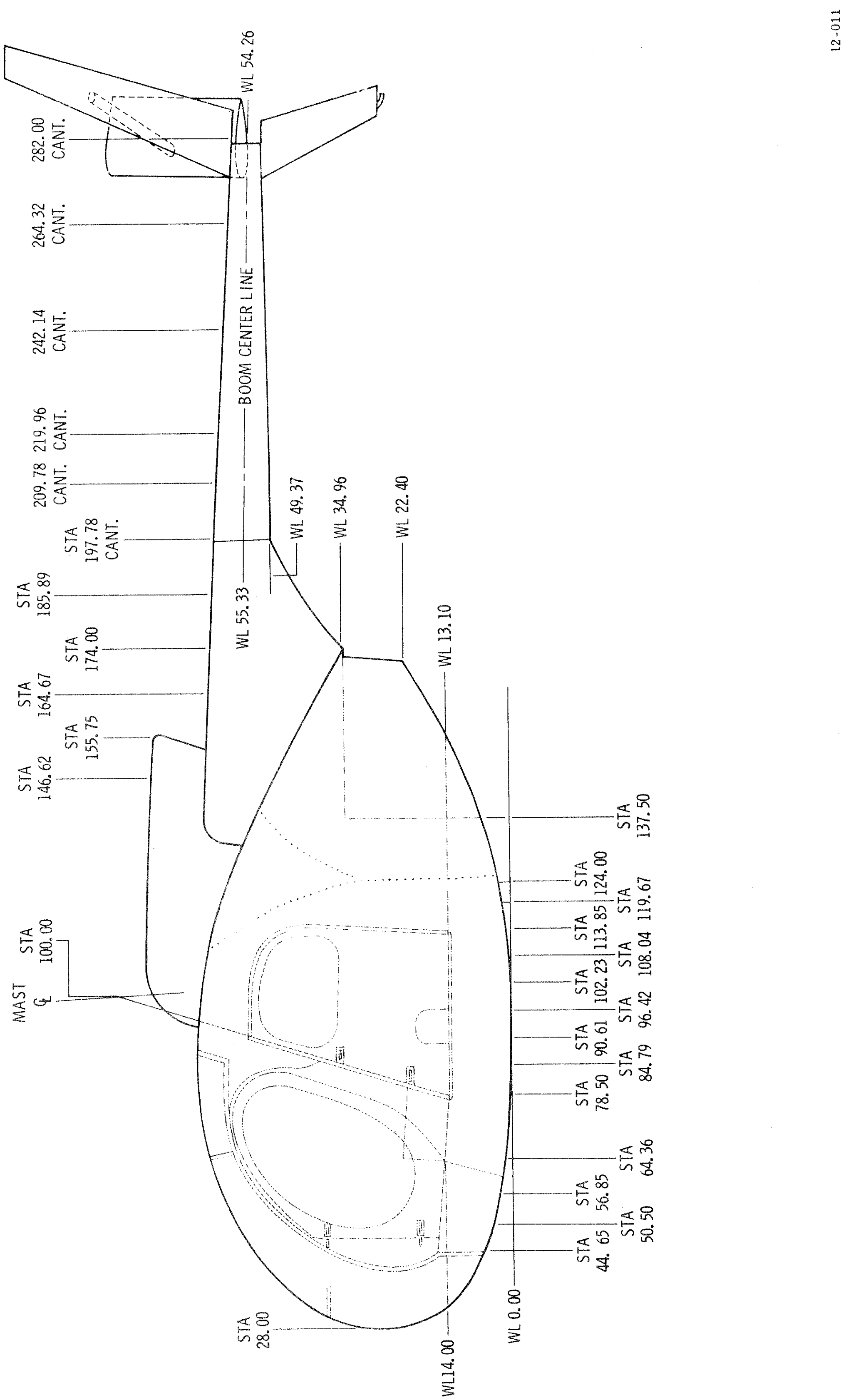


Figure 2-21. Station Diagram.

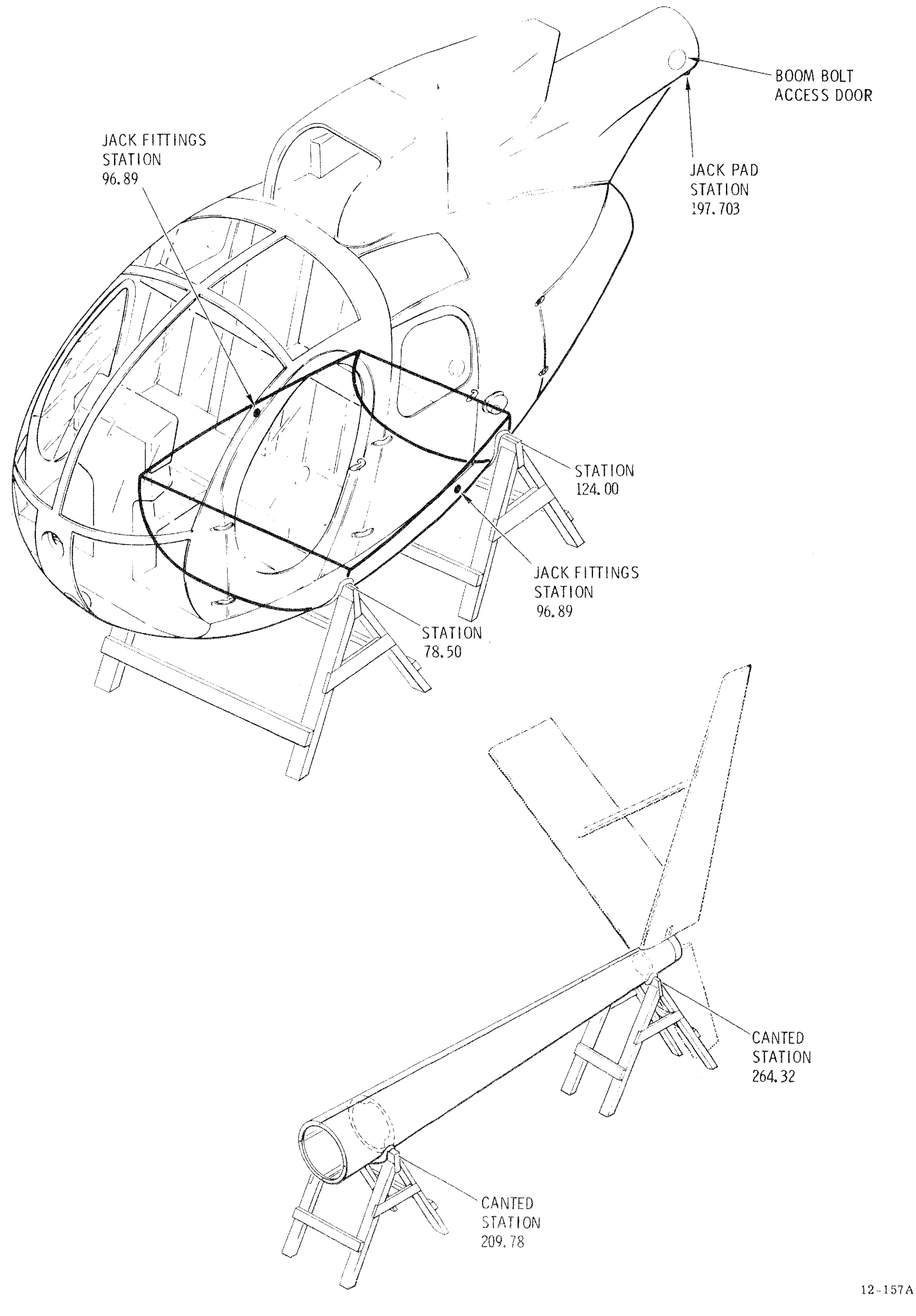
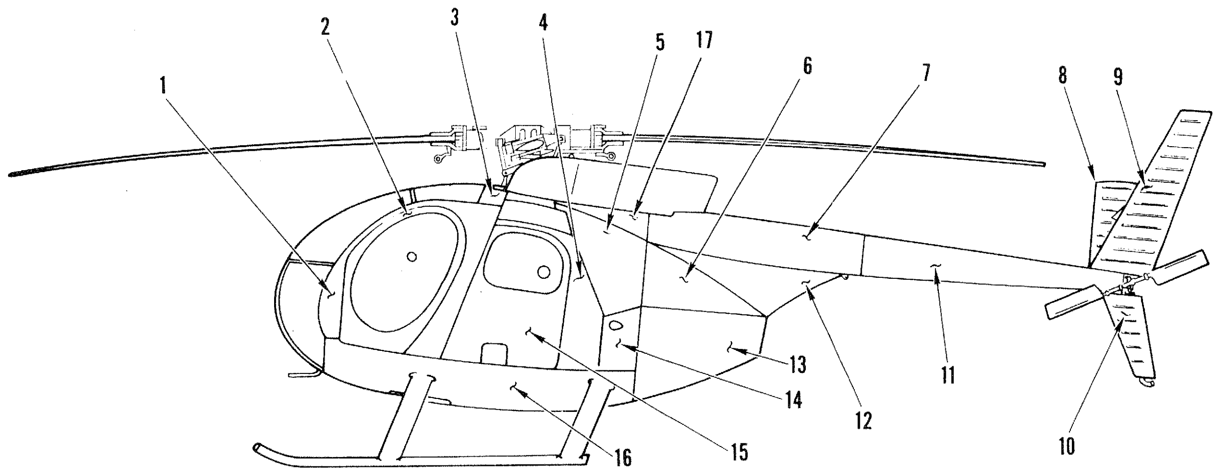


Figure 2-22. Support of Disassembled Aircraft.



CAUTION: DO NOT INSTALL CONDITION O MATERIAL. HEAT TREAT PARTS TO CONDITION INDICATED AFTER FORMING.

INDEX	NOMENCLATURE	NO. OF SKINS	GAGE AND DIMENSION	COMMERCIAL ALLOY	MIL-STANDARD
1.	PILOT'S DOOR FRAME				
	INNER SKIN	2 EA.	0.032 X 50.00 X 60.00	AL ALY 2024-T42	QQ-A-250/5
	OUTER SKIN	2 EA.	0.020 X 50.00 X 60.00	AL ALY 2024-T42	QQ-A-250/5
2.	PILOT'S DOOR				
	INNER SKIN	2 EA.	0.025 X 48.00 X 60.00	AL ALY 6061-T3	QQ-A-250/11
	OUTER SKIN	2 EA.	0.020 X 48.00 X 60.00	AL ALY 2024-T3	QQ-A-250/5
3.	FUSELAGE FWD UPPER SECTION SKIN	1 EA.	0.016 X 7.00 X 46.00	AL ALY 2024-T3	QQ-A-250/5
4.	CARGO DOOR AFT SECTION FRAME SKIN	2 EA.	0.020 X 45.00 X 45.00	AL ALY 7075-T6	QQ-A-250/13
5.	FUSELAGE AFT SECTION SKIN	2 EA.	0.016 X 26.00 X 40.00	AL ALY 2024-T3	QQ-A-250/4
6.	FUSELAGE AFT SECTION SKIN	2 EA.	0.016 X 39.00 X 32.00	AL ALY 2024-T3	QQ-A-250/4
7.	BOOM FAIRING AFT SECTION UPPER SKIN	2 EA.	0.016 X 46.00 X 63.00	AL ALY 2024-T3	QQ-A-250/5
8.	HORIZONTAL STABILIZER SKIN	1 EA.	0.012 X 35.00 X 64.00	AL ALY 2024-T42	QQ-A-250/5
9.	UPPER VERTICAL STABILIZER SKIN	1 EA.	0.012 X 32.00 X 56.00	AL ALY 2024-T42	QQ-A-250/5
10.	LOWER VERTICAL STABILIZER SKIN	1 EA.	0.012 X 36.00 X 30.00	AL ALY 2024-T42	QQ-A-250/5
11.	BOOM SKIN	2 EA.	0.032 X 45.00 X 85.00	AL ALY 2024-T3	QQ-A-250/5
12.	BOOM FAIRING AFT SECTION SKIN	2 EA.	0.016 X 17.00 X 53.00	AL ALY 2024-T3	QQ-A-250/5
13.	ENGINE ACCESS DOOR				
	INNER SKIN	2 EA.	0.020 X 41.75 X 35.50	AL ALY 6061-T6	QQ-A-250/11
	OUTER SKIN	2 EA.	0.016 X 41.75 X 35.50	AL ALY 2024-T3	QQ-A-250/5
14.	LOWER SECTION FUSELAGE STRUCTURE SKIN	2 EA.	0.012 X 24.00 X 26.00	TITANIUM UNALLOYED	MIL-T-9046 CLASS 6
15.	CARGO DOOR INNER SKIN	2 EA.	0.032 X 40.00 X 45.00	AL ALY 6061-T6	QQ-A-250-11
	OUTER SKIN	2 EA.	0.016 X 40.00 X 45.00	AL ALY 2024-T42 OR AL ALY 2024-T3	QQ-A-250/5 OR QQ-A-250/5
16.	LOWER SECTION FUSELAGE STRUCTURE BOTTOM SKIN	2 EA.	0.016 X 44.00 X 98.00	AL ALY 2024-T42	QQ-A-250/5
17.	ENGINE AIR INLET PANEL	2 EA.		AL ALY 2024-T42	QQ-A-250/5

12-002C

Figure 2-23. Skin Plating Diagram.

NOTE

A skin panel is defined as skin area bounded by three or more structural members (stiffeners, longerons, stringers, etc.).

2-266. Negligible Damage — Stressed Sheet Metal Panels (AVIM). None. All repairable damage shall be repaired upon detection. Cracks, tears, or punctures in stressed sheet metal panels that do not exceed 0.160-inch diameter and can be removed by drilling out with a No. 20 or smaller diameter drill do not require structural doublers. Install a rivet of appropriate diameter to fill the hole.

2-267. Patch Repair — Stressed Sheet Metal Panels (AVIM). Cracks, tears, or punctures in stressed sheet metal panels exceeding 0.160-inch diameter are to be patched provided the damage area does not exceed a total of 25 percent of the panel area (previously repaired areas included). The patch shall be applied to the convex surface of the stressed panel. Patch with a single row of rivets, using 6 to 8 diameter spacing.

2-268. Replacement — Stressed Sheet Metal Panels (AVIM). Damage that exceeds the limits for repair by patching requires replacement of the panel.

NOTE

Except for tailboom skin (section 2), excessive damage to the fuselage skin supporting structure (frames, longerons, stringers, etc.) must occur before replacement of a complete skin panel becomes necessary; however, any portion of the skin panel can be replaced.

2-269. NON-STRESSED SHEET METAL PANELS.

2-270. General — Non-Stressed Sheet Metal Panels. Non-stressed sheet metal panels consist primarily of hinged covers and doors (except the fuel cell access doors and controls access door which are stressed). Damage repair definitions are as follows.

2-271. Negligible Damage — Non-Stressed Sheet Metal Panels. Small dents, scratches, nicks, and light corrosion deposits are considered negligible damage. Cracks that do not exceed 0.25-inch in length, are less than one-fourth the width of the damaged component, and are removed at least 1.0 inch from the end of the damaged component or at an attachment point, may be considered negligible damage.

2-272. Damage Repairable by Patching — Non-Stressed Sheet Metal Panels. Damage exceeding that determined as negligible must be repaired by patching, or the section must be replaced. Patches can be applied to a damaged area, in many instances, provided the damage is first trimmed to a suitable shape, and the repair patch cut with sufficient overlap to allow proper edge distance for attachment rivets. An overlay patch may be used if it does not restrict a moving component.

2-273. Repair Guidelines — Non-Stressed Sheet Metal Panels. Refer to paragraph 2-7 for additional non-stressed sheet metal repair information.

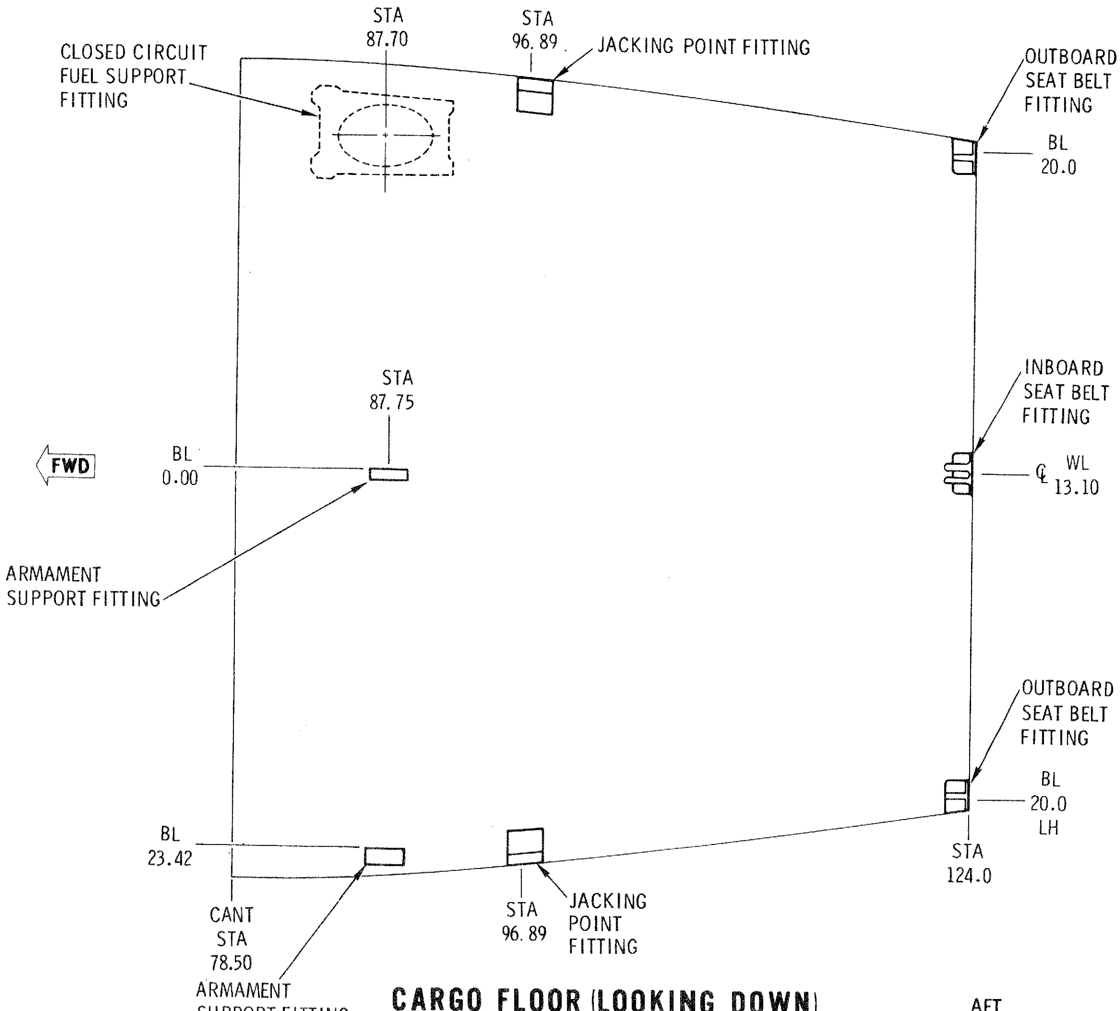
2-274. Replacement — Non-Stressed Sheet Metal Panels. Damage that exceeds the limits for repair requires replacement of the part.

2-275. FUSELAGE CAST AND FORGED FITTINGS (AVIM).

2-276. General — Fuselage Cast and Forged Fittings (AVIM). See figure 2-24, sheet 1 through 4, for location and identification of fuselage fittings. Classification of damage, and repair or replacement criteria applicable to cast and forged fittings listed in table 2-1 are provided in paragraphs 2-277 and 2-282. The fittings listed in this table are highly-stressed structural parts. Any damage in excess of negligible limits (para 2-277) requires replacement of the damaged part or major assembly containing the part. Any crack, regardless of length, requires replacement of the part.

Table 2-1. Highly-Stressed Structural Fittings.

Fitting	Part Number
Mast fitting	369A2016
Cargo floor armament support fitting assembly	369A2511
Mast support structure fitting	369A3027
Boom attach fuselage fitting	369A3030
Fuselage attach boom fitting	369A3510
Stabilizer and tail rotor transmission mounting frame	369A3503
Horizontal stabilizer center boom attach fitting	369A3601
Upper vertical stabilizer center boom attach fitting	369A3626
Landing gear strut	369A6001



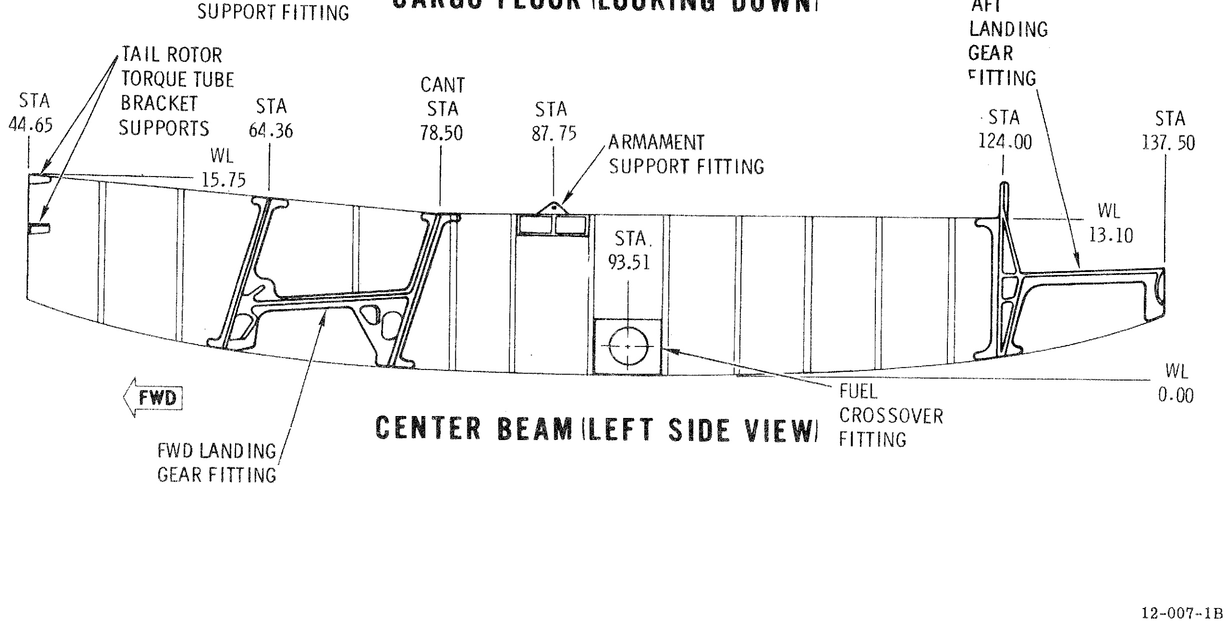
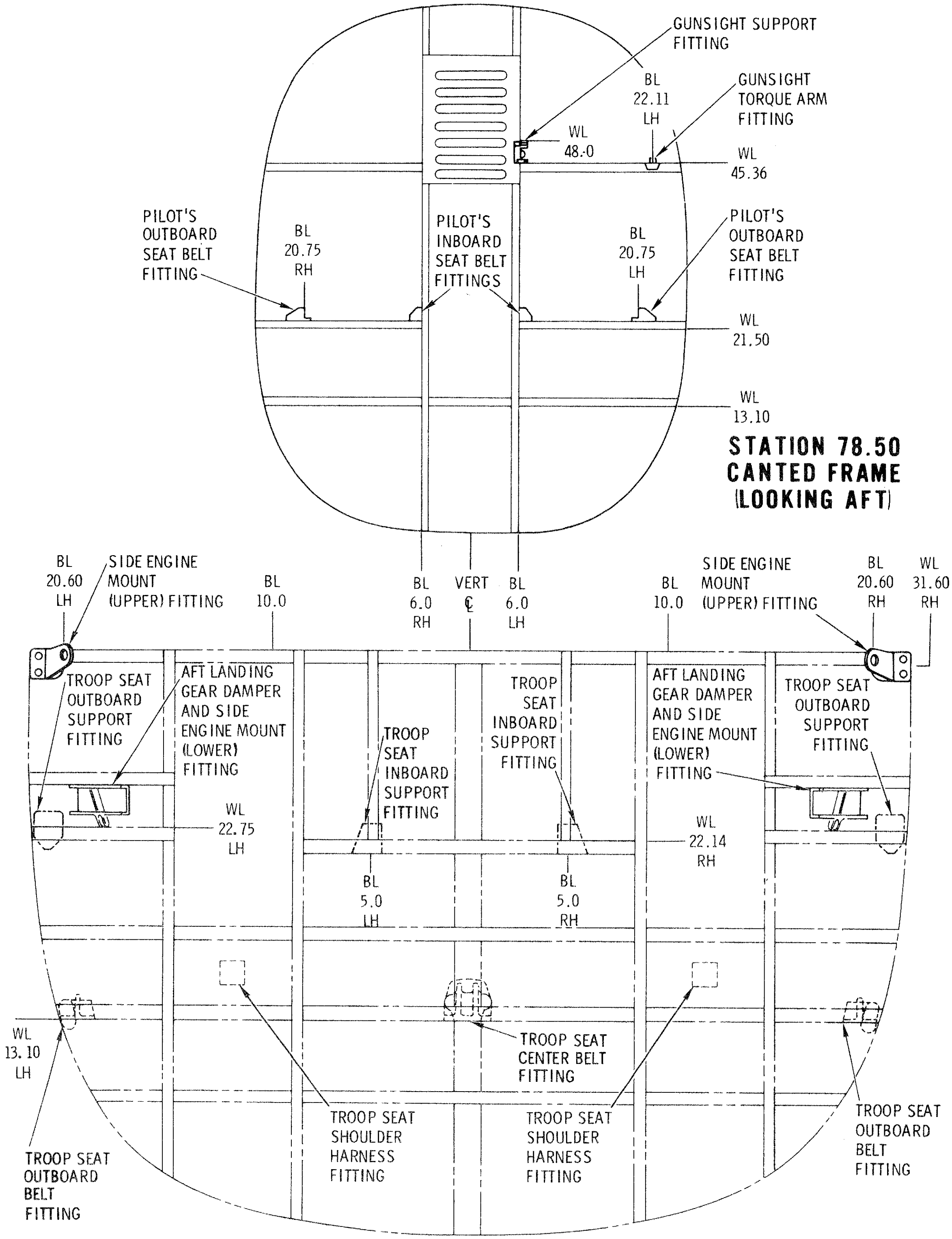


Figure 2-24. Body Group Fittings. (sheet 1 of 4)



STATION 124.00 LOWER SECTION FRAME (LOOKING FORWARD)

12-007-2C

Figure 2-24. Body Group Fittings. (sheet 2 of 4)

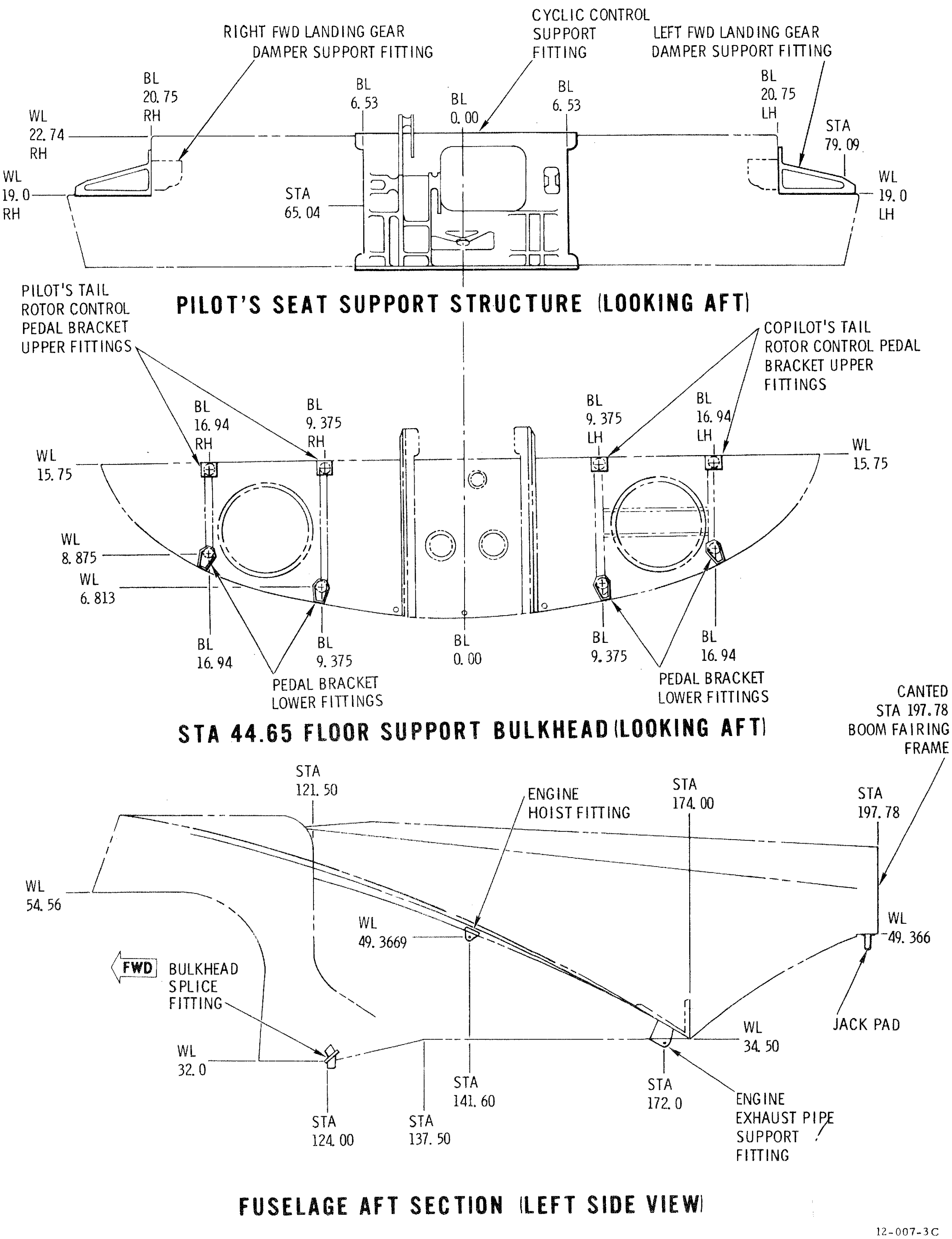
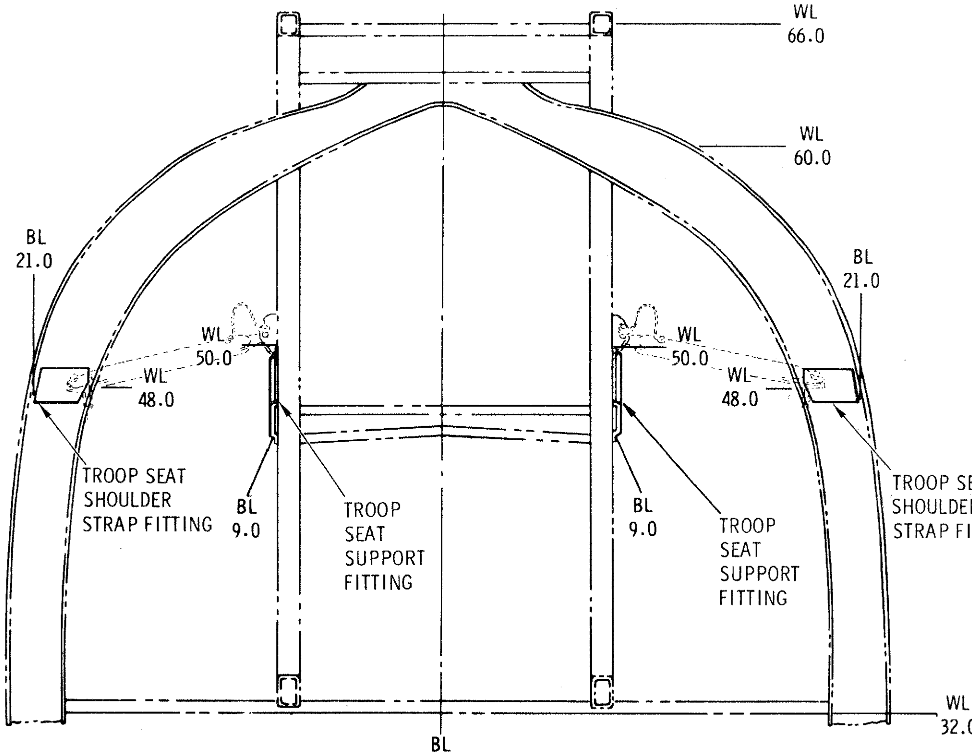
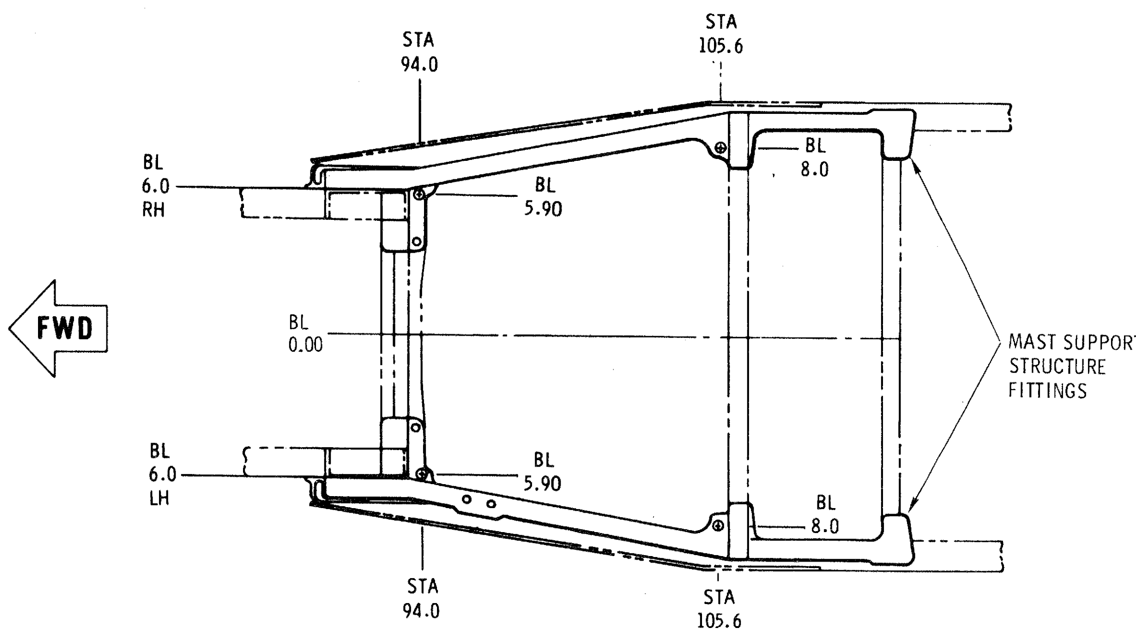


Figure 2-24. Body Group Fittings. (sheet 3 of 4)



STATION 124.00 CANTED FRAME (LOOKING AFT)



MAST SUPPORT STRUCTURE FITTINGS (LOOKING DOWN)

12-007-4B

Figure 2-24. Body Group Fittings. (sheet 4 of 4)

NOTE

Special jigs or holding fixtures may be required to hold critical dimensions during replacement of fittings. Availability of special jigs or holding fixtures should be considered before attempting major repairs.

2-277. Classification of Fuselage Cast and Forged Fitting Damage — General (AVIM).

2-278. Negligible Damage — Fuselage Cast and Forged Fittings (AVIM). Longitudinal scratches, nicks, corrosion, or dents shall not exceed 0.010-inch depth or 15 percent of the length of the fitting. Transverse scratches, nicks, or dents shall not exceed 0.010-inch depth or 15 percent of the thickness of the fitting. The preceding damage limits apply after polishing or repair is accomplished. All repaired surfaces shall be treated with chemical film (C20).

2-279. Patch Repair — Fuselage Cast and Forged Fittings (AVIM). A typical temporary patch repair for fittings is described in paragraph 2-301. Temporary repairs will be replaced at next scheduled maintenance period (PMP or Depot Maintenance). DA form 2408-13 entry is required on all temporary repairs. Temporary repairs require daily inspection until replaced.

2-280. Reparable Replacement — Fuselage Cast and Forged Fittings (AVIM). Replacement of bushings, bearings, or inserts of a dissimilar metal which form a part of the fitting are considered replacement repairs.

2-281. Non-Reparable Replacement — Fuselage Cast and Forged Fittings (AVIM). Damage to a fitting not reparable by temporary patch or insertion repair requires replacement of the entire fitting. (Refer to TM 55-1520-204-25/1 series manuals when replacement of lock-bolts is required.)

2-282. Repair or Replacement — Forward and Aft Landing Gear Fittings (AVIM). Replacement procedures for landing gear fitting bearings are given in paragraph 2-298. Cracks in the webs and flanges of the landing gear fittings are not reparable except for the temporary type repairs described in paragraph 2-301.

2-283. CARGO FLOOR.

2-284. General — Cargo Floor. The cargo floor provides a multi-use area for combination of cargo, passengers, and armament. (Refer to fig. 2-24 for fitting location.) Tie-down or mounting hard points for each use are provided on the floor within the compartment.

2-285. Cargo Tiedown Fittings — Cargo Floor. Twelve 0.186-inch diameter holes, two per stiffener, located 2.25 inches inboard from each side of the fuselage, provide attach points for cargo tiedown fittings. The cargo tiedown fittings are formed of 0.250-inch diameter, U-shaped, steel rods having a lanyard-attached quick-release pin to secure the fitting to the cargo floor. Cargo tiedown fittings can be installed at any of the 12 locations, as required for load arrangement.

2-286. Passenger Seat Belt Attachment Fittings — Cargo Floor. Seat belt attachment fittings are mounted in the cargo floor, against the aft bulkhead. The inboard fitting is a two-lug, aluminum alloy forging with a 0.25-inch cotter-pin-secured flathead pin for belt snap end fitting attachment. the outboard fittings are single-lug, aluminum alloy forgings with 0.25-inch diameter holes for the belt snap end fittings.

2-287. Armament Support Fittings — Cargo Floor. The outboard fitting just inside the left cargo door, and the inboard fitting of the center beam assembly, are single-lug, aluminum alloy forgings with 0.375-inch diameter bushed holes.

2-288. Fuel Cell Tiedown Eyelets — Cargo Floor. Twelve grommet-lined tiedown eyelets in aluminum alloy brackets on the underside of the floor provide lacing support for the fuel cells.

2-289. Jacking Point Fittings — Cargo Floor. Each fitting is a machined, aluminum alloy forging with a 0.702-inch diameter bore exposed through a cutout in the skin just below the edge of each cargo door.

2-290. Inspection — Cargo Floor. *a.* Inspect cargo floor for dents, cracks or bent members. If severe dents, cracks, or punctures are found, remove appropriate fuel access cover and inspect fuel cell.

b. Inspect all fittings for corrosion, cracked or broken lugs, and loosened rivets.

c. Inspect cargo floor for evidence of water accumulation. Check that drain holes (if present) aft of cargo doors are clear and free of obstruction

2-291. General Repair — Structural Support and Attachment. Refer to paragraph 2-264 for general repair information and TM 55-1500-204-25/1 for structural support and attachment repair and replacement. Refer to table 2-3, and note items 2, 5, 11, and 13, and items 1 and 2, table 2-4, as repair material.

2-292. Repair — Cargo Floor Drainage. Drain holes may be drilled at floor level on both sides of the aircraft to provide drainage through the exterior skin as follows:

a. Locate holes on the cargo compartment side of the exterior skin 2.00 inches forward of the station 124.00 lower section just above the cargo floor. Be sure that drilling will not damage floor and that holes are no more than 0.04-inch above the floor.

b. Drill a 0.25-inch hole and deburr.

c. Treat the area with aluminum surface protection (para 1-42 and 1-47) and touch up paint.

2-293. Replacement — Armament Support Bushing. *a.* Remove fuel access doors.

b. Remove lockbolts that attach armament support fitting to center beam; lift out fitting.

NOTE

The 0.375-inch diameter center hole of the support bushing may be located off-center in relation to the support log. Off-center bushings are installed with the thin wall up. Check bushing bore for concentricity to determine bushing wear.

c. Note location of bushing hole center and replace only with like bushing; press worn bushing out of the fitting.

d. Inspect support fitting for cracks and scoring of bore.

e. Clean the ID of the armament support fitting lug and the OD of the replacement bushing by wiping with methyl ethyl ketone (C69) and a clean cloth a minimum of three times.

f. Coat the ID of the support fitting lug and the OD of the replacement bushing with locking compound primer (C91) and allow to air-dry for 10 minutes.

g. Apply locking compound (C90) sparingly to the previously coated surfaces by using a clean applicator.

h. Press the replacement bushing into the armament support fitting lug. Off-center replacement bushings are installed with the thin wall section next to the rounded upper edge of the lug. Wipe away any compound that may have worked into the bushing ID.

i. Install armament support fitting in center beam assembly; install eight lockbolts.

j. Install fuel cell access doors.

2-294. CENTER BEAM.

2-295. Description — Center Beam. The lower section center beam assembly (27, fig. 2-20) is a primary structural member of the aircraft. The beam is a riveted and bolted assembly of aluminum alloy webbing, stiffeners, and doublers. Forged aluminum alloy landing gear fittings are mounted at tht forward and aft ends of the center beam. Each fitting contains four swivel bearings for attachment of the landing gear braces and struts. The forward fittings have two additional bearing attachment points for the longitudinal and lateral cyclic trim actuators.

2-296. Inspection — Center Beam. (See fig. 2-20.) *a.* Inspect the forward and aft landing gear fittings (18 and 23) for cracks and distortion. A distorted fitting usually indicates yielding due to excessive load. Perform dye penetrant inspection of questionable fittings according to TM 55-1500-204-25/1. Inspect the fitting bearings for scuffing or scoring on the ball surfaces caused by binding.

b. Inspect the armament support fittings (19) for secure attachment and for cracks and distortion. Inspect the bushings for secure fit and bores for elongation.

c. Inspect the tail rotor pedal torque tube bracket support fittings (24 and 25) for secure attachment; check for cracks and deformation.

NOTE

If there is any indication of center beam damage, remove fuel cells and inspect the center section of the beam (fig. 2-25).

d. Remove right fuel access door.

e. Remove necessary fuel cell attaching hardware and pull the fuel cell assembly from the keel beam.

f. Inspect the keel beam upper flange for indications of web or stiffener buckling between stations 95 and 115.

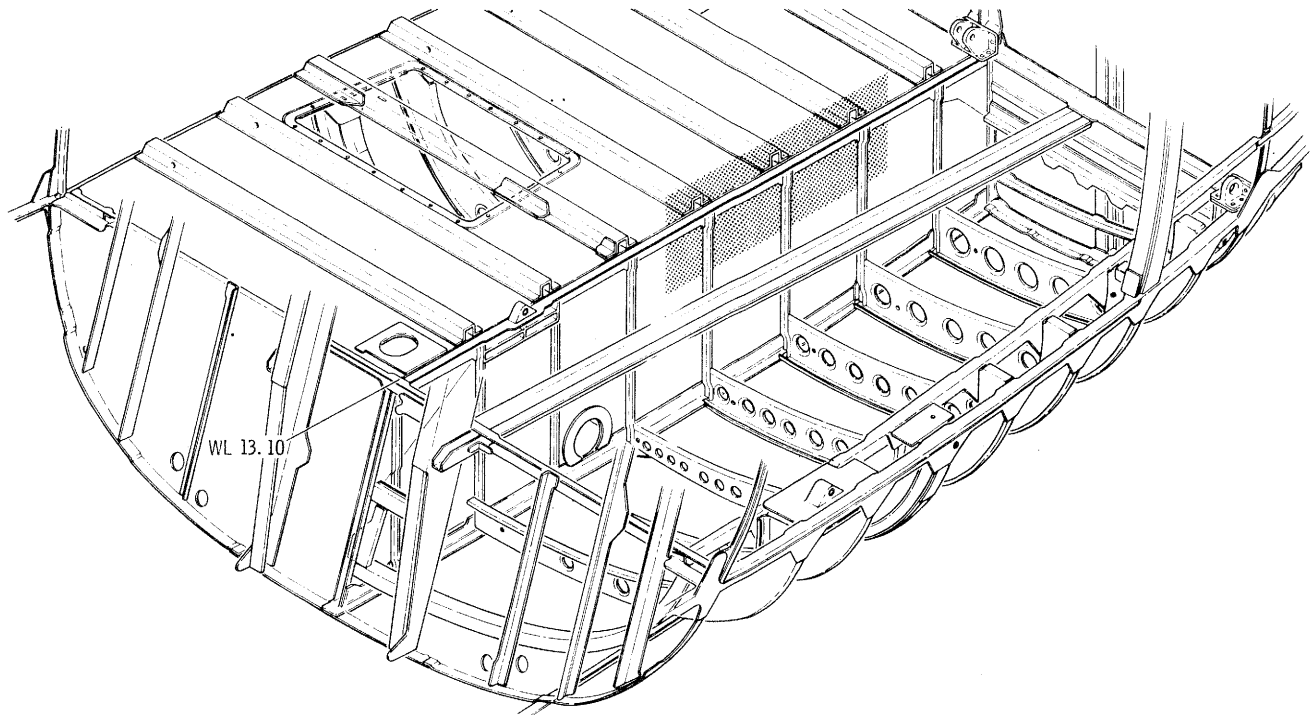
(1) Minor buckling of the keel beam is acceptable provided no top stiffener rivets have pulled out and no cracks or tears exist in the web or stiffeners.

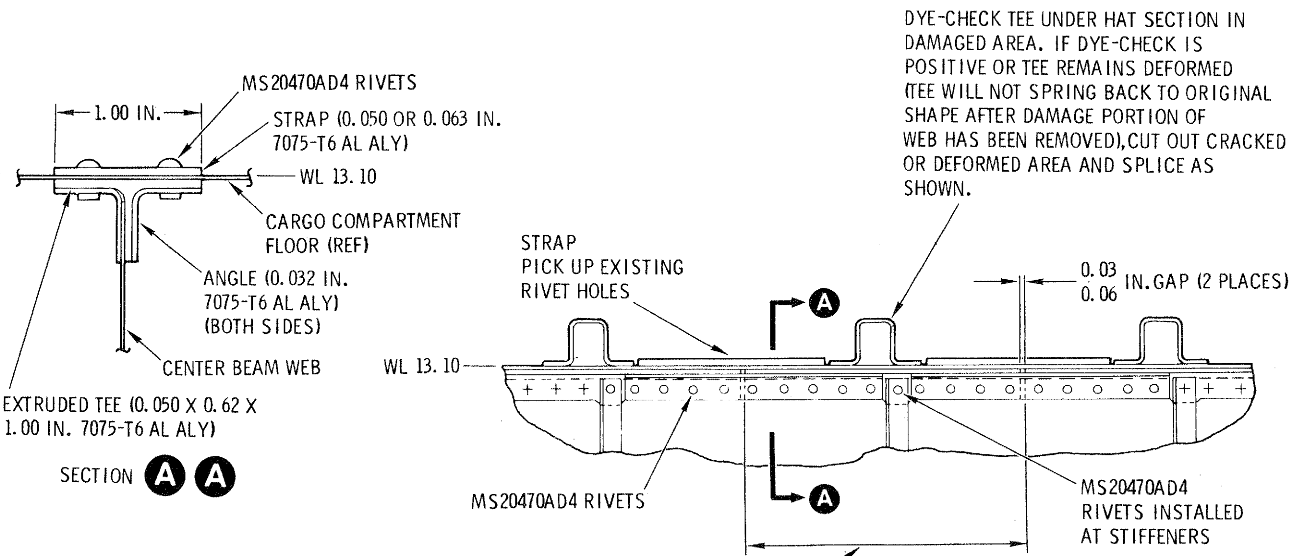
(2) If any top stiffener rivets have pulled out and/or cracks or tears are evident in the web or stiffeners the aircraft must be restricted to limited service until repairs can be accomplished.

(3) Aircraft in limited service should have cargo compartment loading restricted to 400 pounds maximum.

g. After completion of inspection reinstall hardware removed in *d* and *e* above.

2-297. Repair — Center Beam (AVIM). Scratches, nicks, light corrosion deposits, and smooth contour dents that blend smoothly into surrounding surfaces are considered negligible damage. Small, smooth, isolated dents may be classified as negligible, and if in heat treated material should be left as is. Nicks and scratches, if small, isolated, and free of sharp edges, may also be classified as negligible. Touch up dented. scratched, or broken areas to prevent corrosion (chapter 1). Materials used in all repairs and reinforcements (items 2, 5, 11 and 13, table 2-3, and items 3, 4, and 5, table 2-4) shall be selected according to TM 55-1500-204-25/1. Refer to figure 2-25.





CUT OUT DAMAGED AREA OF TEE. CUT HALFWAY BETWEEN HAT SECTIONS AND BETWEEN TOP AND SIDE RIVETS. REPLACE DAMAGED PORTION OF TEE WITH EXTRUDED TEE SECTION (LENGTH AS REQD). ADD ANGLES OVERLAPPING RIVETS EITHER SIDE OF JOINT. ADD STRAP OVER TOP OF JOINT. IF EXTRUSION NOT AVAILABLE SEE ALTERNATE TEE REPAIR METHOD (SHEET 2 OF 2).

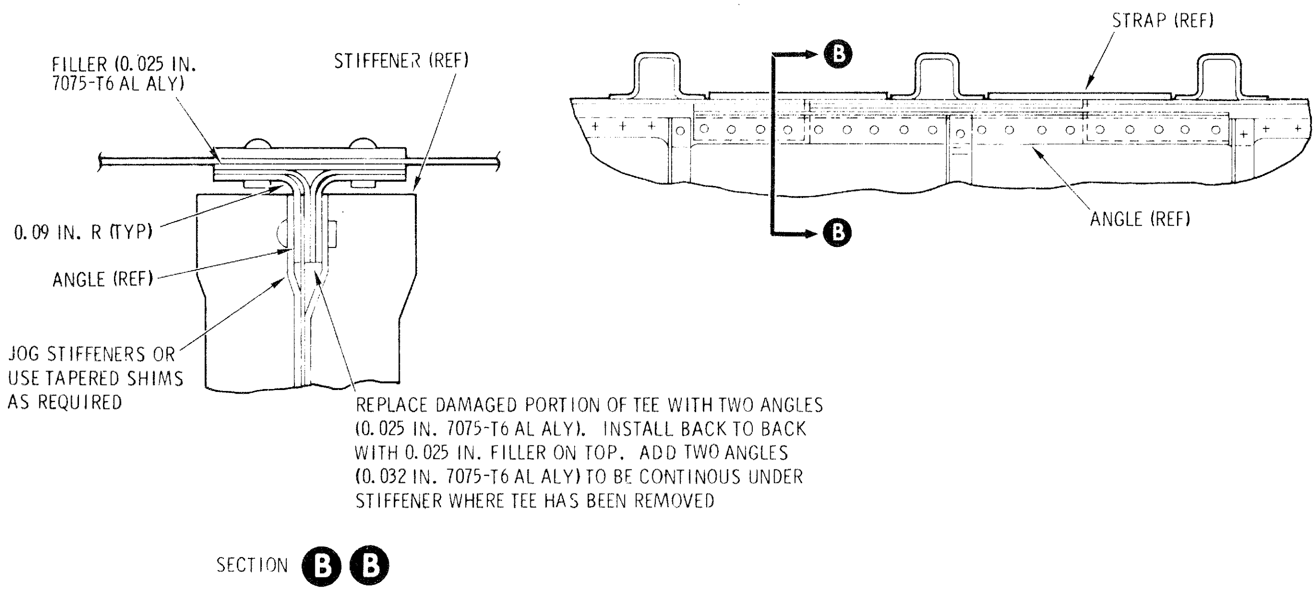
NOTES:
1. (+) INDICATES EXISTING RIVET LOCATION.
2. (O) INDICATES NEW RIVET LOCATION.
3. OBSERVE THE FOLLOWING SHEET METAL PRACTICE:
RIVET EDGE DISTANCE 2 X DIA.; LOCATE RIVET MINIMUM OF 0.25 INCH FROM ADJACENT STRUCTURE AND OR JOGGLE BEND.

REPAIR OF CRACKED OR DEFORMED TEE

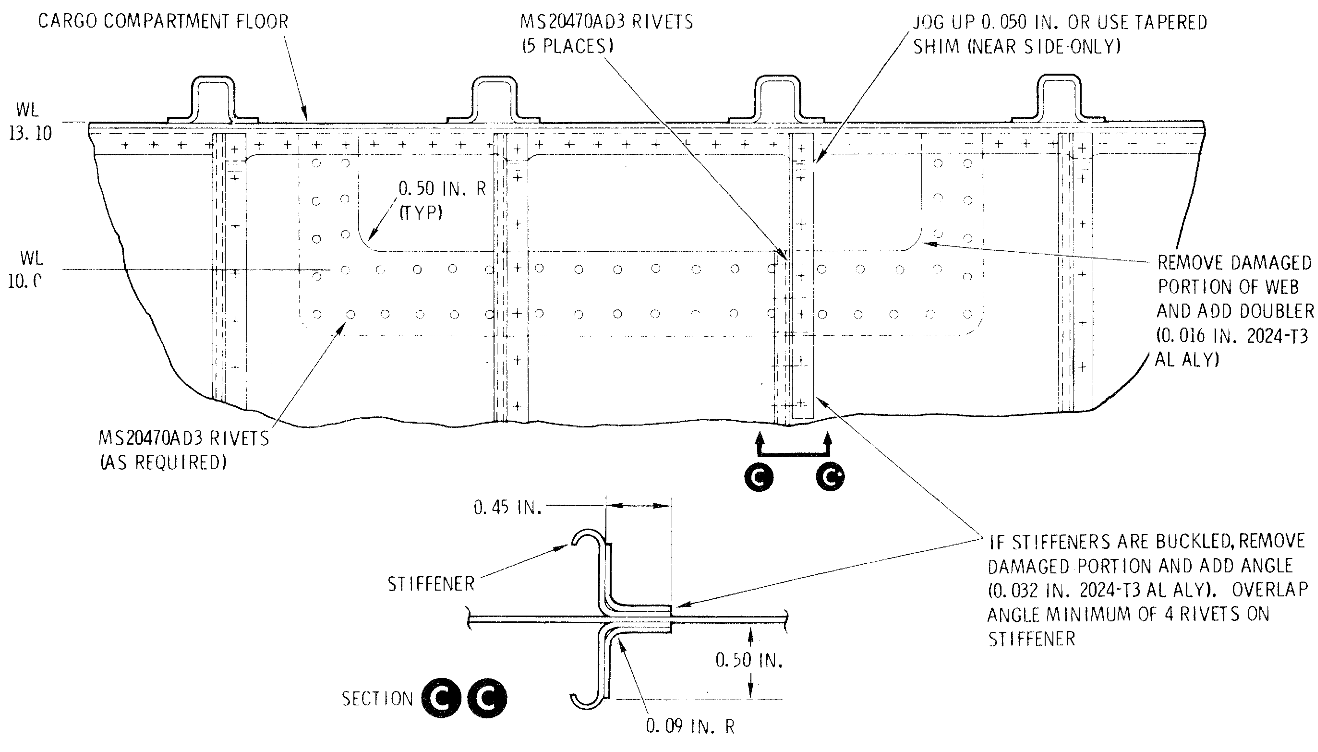
VIEW LOOKING INBOARD LEFT SIDE

12-170-1

Figure 2-25. Center Beam Repair. (sheet 1 of 2)



TEE REPAIR ALTERNATE METHOD



REPAIR OF BUCKLED WEB AND STIFFENERS

VIEW LOOKING INBOARD LEFT SIDE

12-170-2

Figure 2-25. Center Beam Repair. (sheet 2 of 2)

NOTE

Straightening of dents will disturb molecules in heat treated metal and may cause internal cracks.

When necessary, place fuselage structure in a suitable support as shown in figure 2-22.

a. Gain access to the damaged part by removing adjacent parts, opening up skins or by cutting an access hole, whichever is more feasible.

b. Drill out rivets or remove hardware that secures damaged part to structure; remove damaged part.

c. Position replacement part to structure and clamp securely in place.

d. Pick up rivet or hardware hole locations from installed structure and drill rivet or hardware holes in replacement part.

e. Install rivet or hardware to secure replacement part to structure.

f. Install parts removed to gain access; repair access hole by patching. Refer to figure 2-23 and table 2-3 for repair material.

Table 2-2. Premaintenance Requirements for Removal and Installation of Landing Gear Bearings.

Conditions	Requirements
Special Tools	(T36)

2-298. FORWARD AND AFT LANDING GEAR FITTINGS AND BEARINGS

2-299. Inspection — Forward and Aft Landing Gear Fittings and Bearings. *a.* Inspect the landing gear fittings for cracks in the webs and flanges and evidence of wear or damage in the bearing areas.

b. Inspect bearings in landing gear fittings for evidence of wear and corrosion. Wear tolerances, radial and axial, of 0.0040 shall not be exceeded.

2-300. Replacement of Landing Gear Attach Bearings (AVIM). *a.* Install detail A of removal tool (T36) as shown in figure 2-26 with the bolt head on the side of the bearing seat.

b. With the bolt shank centered in the bearing bore, tighten the nut and remove the bearing.

c. Clean the ID of the bearing recess in the landing gear attach fitting and the OD of the replacement bearing by wiping with methyl ethyl ketone (C69) and a clean cloth a minimum of three times.

d. Coat the ID of the attach fitting bearing recess and the OD of the replacement bearing with locking compound primer (C91) and allow to air-dry for 10 minutes.

e. Apply locking compound (C90) to the previously coated surfaces by using a clean applicator.

f. Install the bearing by using detail B of installation tool (T36) as shown in figure 2-26. Make sure that the bearing enters the bore straight and that the bolt is centered in the bore.

g. Remove excess sealant. Apply a small fillet of sealant to the parting surfaces of the bearing and housing.

h. Allow the sealant to harden a minimum of 24 hours before installing the landing gear.

i. After the sealant has hardened, visually check the bearing for proper seating and free movement of the bearing inner race.

2-301. Repair or Replacement of Landing Gear Attach Fittings (AVIM). *a.* Cracks in the webs and flanges of the landing gear fittings are not repairable except for the temporary type repairs shown in figure 2-27.

b. Refer to paragraph 2-275 for information on replacement of fuselage cast and forged fittings.

2-302. NON-CRASH-RESISTANT (NCR) FUEL CELL SUPPORT SKINS.

2-303. General — Non-Crash-Resistant (NCR) Fuel Cell Support Skins. Cleaning and inspection of support skins can only be accomplished with fuel cells removed. (See fig. 2-28.)

2-304. Cleaning — NCR Fuel Cell Support Skins. *a.* Dry the support skins with a clean cloth, as necessary.

b. Wipe skin surfaces with clean cloths wet with isopropyl alcohol (C82); then wipe with a clean, dry cloth until dry.

2-305. Inspection — NCR Fuel Cell Support Skins. *a.* Check security of fiberglass skin attachment to support angles and brackets.

b. Inspect fiberglass skin for distortion, breaks, or cracks.

c. Inspect anti-abrasion tape for secure adhesion over rivet heads and all sharp edges and lap joints.

2-306. Repair — NCR Fuel Cell Support Skins. (See fig. 2-28.) (Refer to figure 2-23 and table 2-3 for repair materials.) *a.* Disconnect all external electrical power.

b. Open all circuit breakers and set power selector switch to OFF.

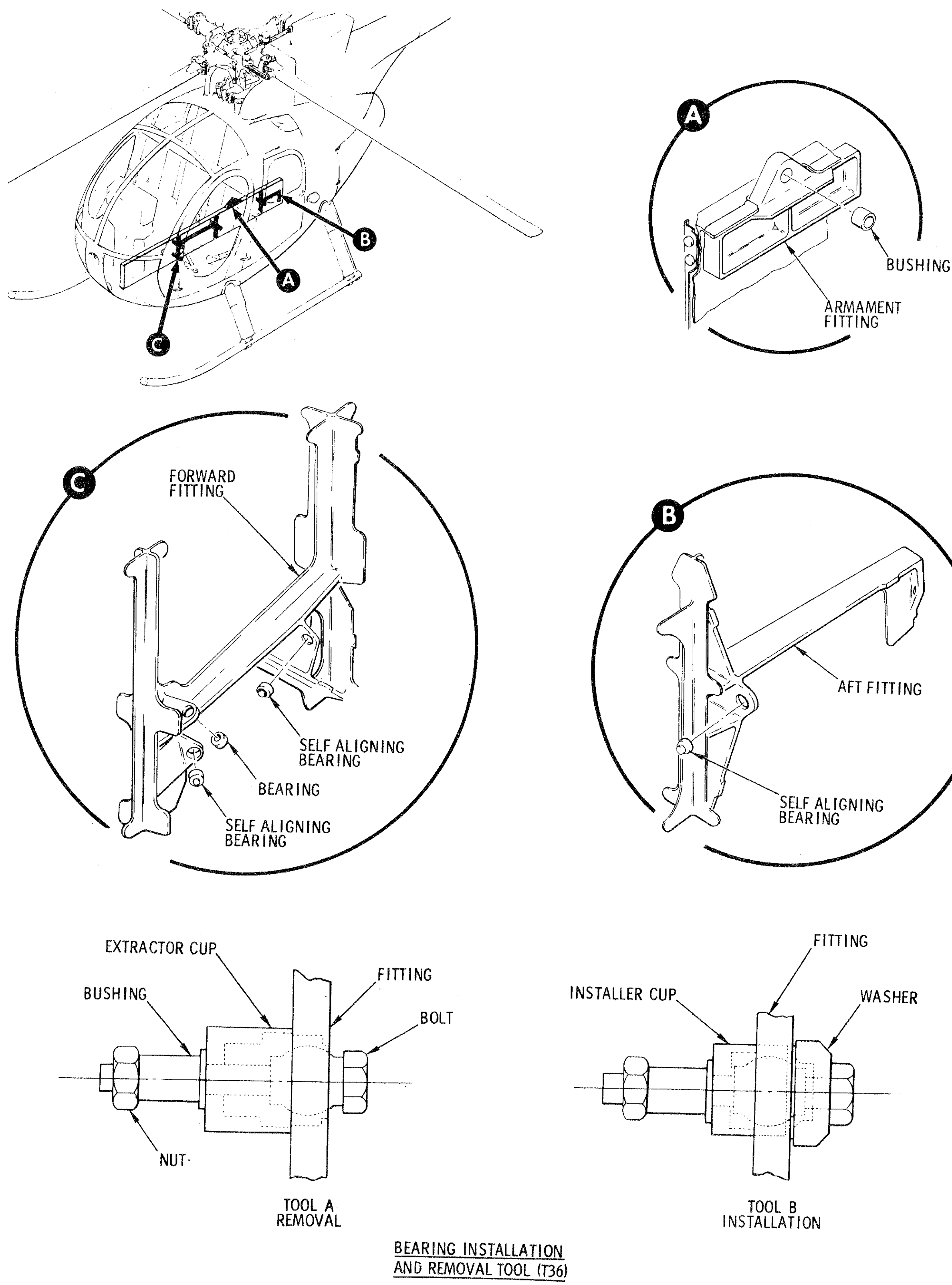


Figure 2-26. Center Beam Fitting Bearing Replacement.

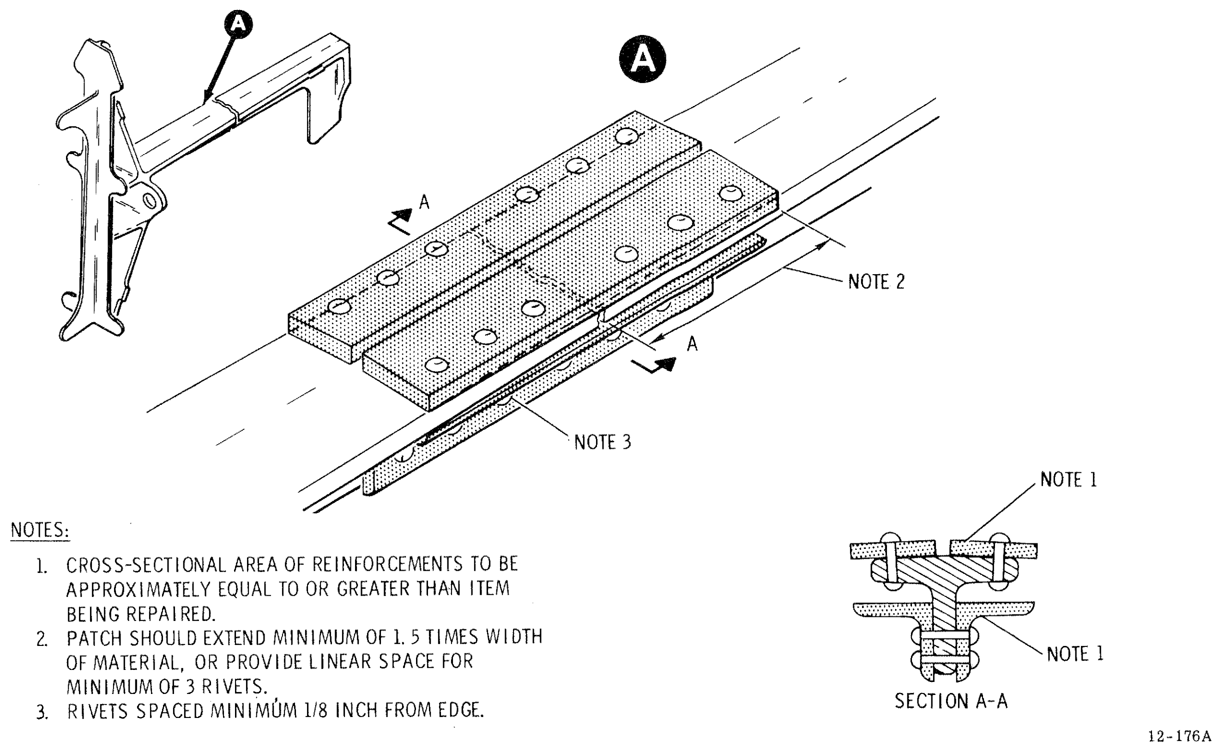


Figure 2-27. Landing Gear Fitting Temporary Repair.

c. Remove fuel cell access door.

d. Remove fuel cell (chapter 10).

e. Determine location of area requiring repair.

CAUTION

Check beneath cargo floor to make sure major structural elements are not affected by cutting operation.

NOTE

When damaged area of skin is not easily accessible through fuel cell access opening, continue according to f through h below; otherwise, proceed with step i.

f. Cut cargo floor in repair area, cutting longitudinally (parallel with aircraft centerline) 2 to 3 inches away from the aircraft center beam, through one channel, and extend cut to flange edge of adjacent forward and aft channels.

g. Cut along adjacent channel flange edges to outboard edge of cargo floor.

h. Drill out rivets on outboard edge of cargo floor between adjacent channels in cutout area and remove floor panel.

i. Repair of the fuel cell support skin requires an aluminum patch equal to, or greater than, the thickness needing repair. Overlap the edges of the aluminum patch approximately one inch with a fiberglass patch.

j. When the fiberglass skin is damaged (cracked, etc.) in an area that is attached to a supporting rib, flange, or bracket, drill the attaching rivets out of the damaged area, and stop-drill crack ends.

WARNING

During repair, do not use explosive-type rivets as they constitute a serious fire hazard because of the possible concentration of fuel vapor.

NOTE

Rivets need not be used to secure patch to skin provided the patch is clamped in place under a contact pressure of 10 to 30 psi during the curing period.

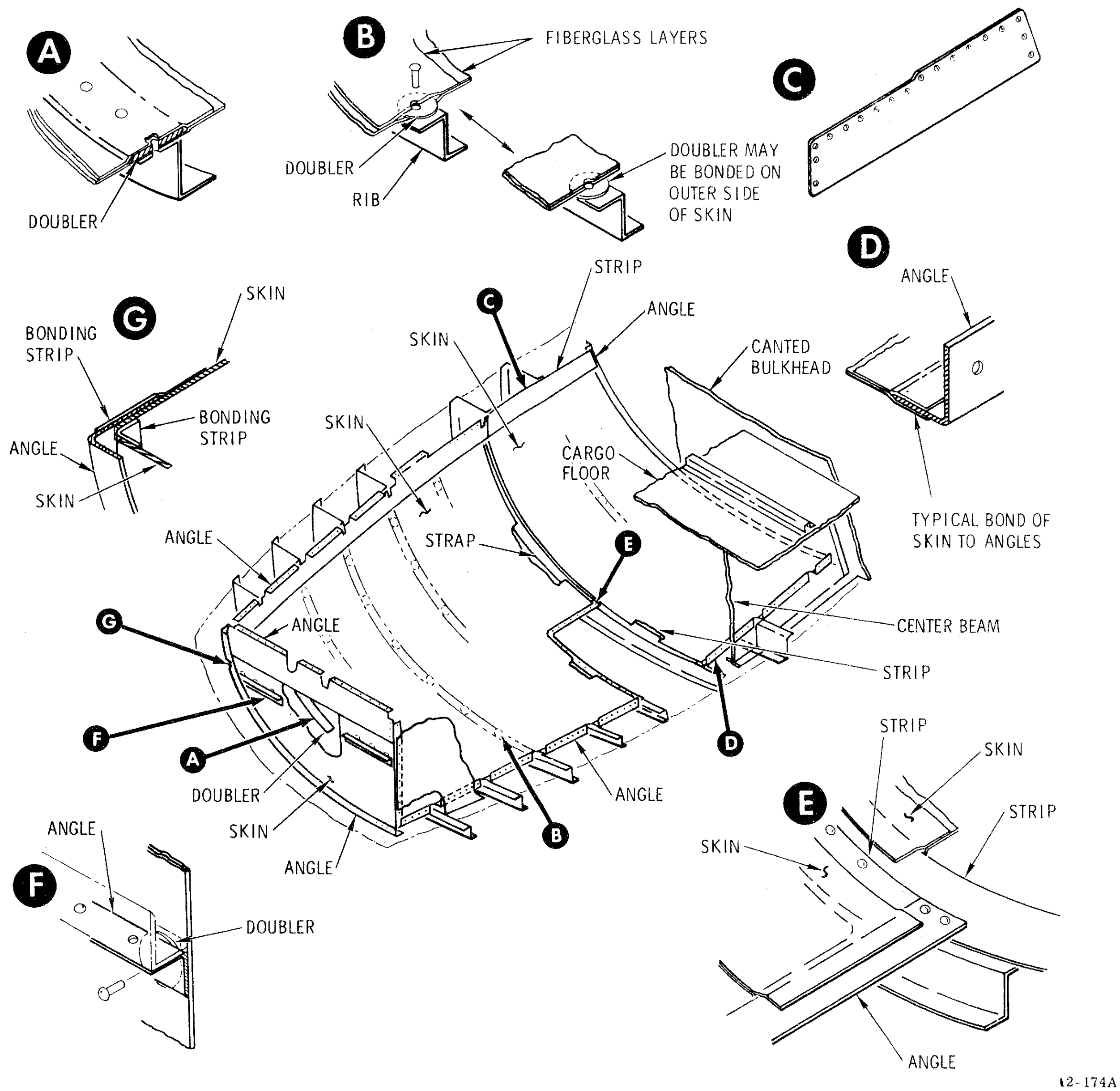


Figure 2-28. NCR Fuel Cell Support Skin Installation.

k. Install patch according to *i* above. Pick up existing rivet hole size and pattern.

l. Secure patch in position with mechanically expanded rivets. Cover rivet heads with tape (C106) to prevent chafing of fuel cell skin.

m. Obtain necessary parts of proper dimensions and material (same as or stronger than original) to accomplish repair of cutout cargo floor section.

(1) Install new channel to rest under channel. Rivet channel in place by using same rivet size and spacing as adjacent areas.

(2) Rivet angles to flanges of adjacent forward and aft channels; check for proper level of cargo floor.

(3) Install cutout section of cargo floor and rivet in place, picking up existing cargo floor rivet pattern.

n. Install fuel cell (chapter 10).

o. Install fuel cell access door.

2-307. Replacement — NCR Fuel Cell Support Skins. Refer to paragraph 2-244 for fiberglass replacement criteria.

2-308. CRASH-RESISTANT (CR) FUEL CELL BACKUP LINERS.

2-309. General — Crash-Resistant (CR) Fuel Cell Backup Liners. Cleaning and inspection of backup

CAVITY FILLER

	LEFT SIDE 369A2639	RIGHT SIDE 369A2639
1	-3	-4
2	-5	-6
3	-7	-8
4	-9	-10
5	-11	-12
6	-13	-14
7	-15	-16
8	-17	-18

NOTES:

1. FOR LINER REPLACEMENT, USE EXISTING LINER AS TEMPLATE FOR CUTOUTS AND HOLE (0.250 IN. DIA) LOCATION.
2. COVER EXPOSED EDGES AND SCREW HEADS WITH TAPE (C107).

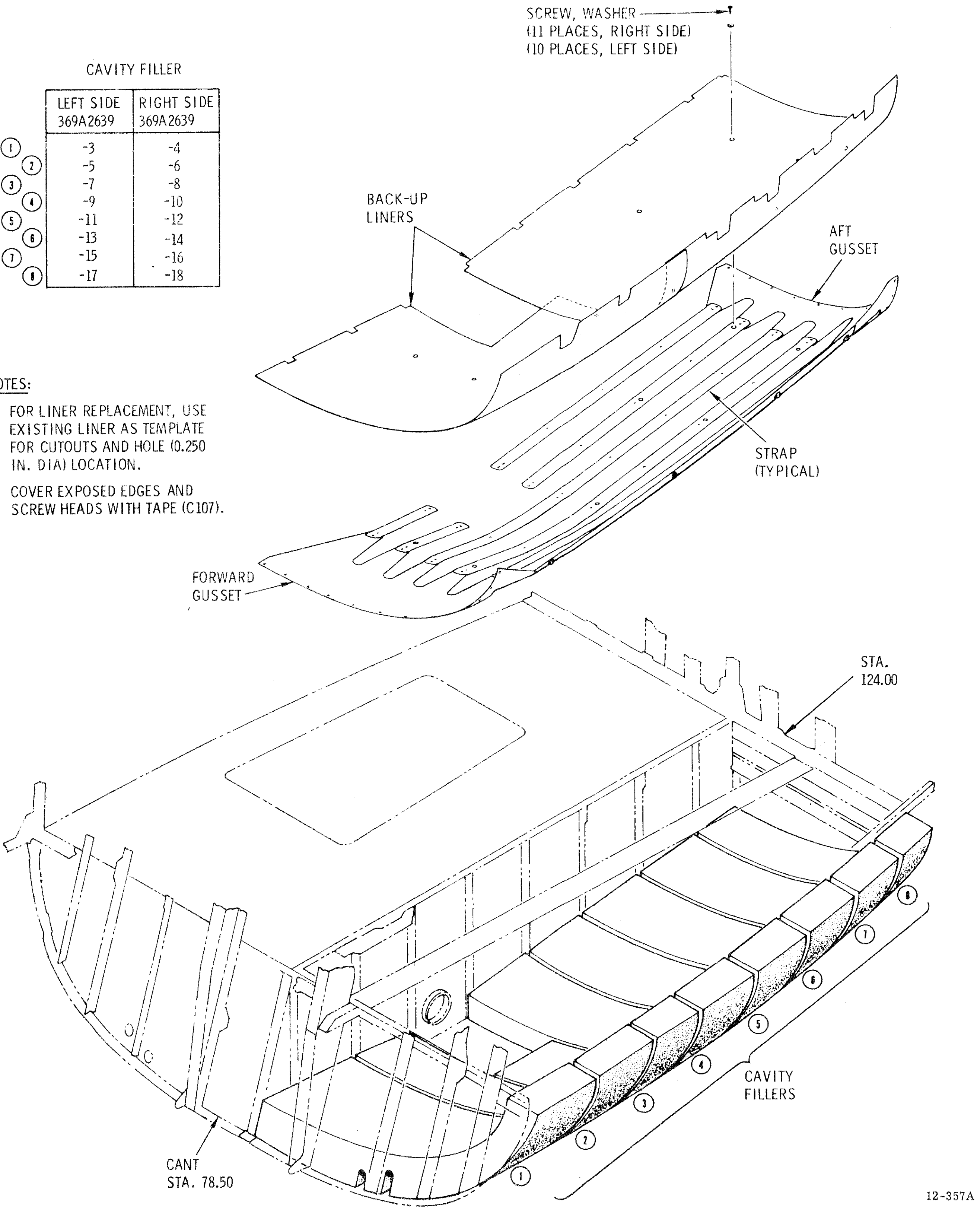


Figure 2-29. CR Fuel Cell Support Installation.

liners can only be accomplished with fuel cells removed. (See fig. 2-29.)

2-310. Cleaning — CR Fuel Cell Backup Liners. *a.* Remove talc or other foreign material from backup liners with a clean cloth.

b. Wipe liner surfaces with clean cloths wet with isopropyl alcohol (C82); then wipe with a clean, dry cloth until dry.

2-311. Inspection - CR Fuel Cell Backup Liners. *a.* Check for security of backup liner and that attachment screws are in place and secure.

b. Inspect backup liner for distortion, breaks or cracks.

c. Inspect anti-abrasion tape for secure adhesion over screw heads and exposed edges of liner cutouts.

d. Check visible fiberglass at forward and aft gussets for condition.

2-312. Removal — CR Fuel Cell Backup Liners. (See fig. 2-29.) *a.* Remove fuel cells (chapter 10).

b. Remove abrasion tape covering screw heads and exposed edges of liners.

c. Remove attachment screws and washers; then roll liner and remove through floor opening.

2-313. Removal — CR Fuel Cell Cavity Fillers. *a.* Identify strap locations for reinstallation.

b. With backup liner removed, drill out rivets and remove sufficient straps and/or forward and aft gussets to release cavity filler blocks.

c. Lift filler block out from between aircraft ribs. Note that filler may not slide out in lateral direction due to aircraft conduit running through notches at the underside of filler blocks.

2-314. Repair — CR Fuel Cell Backup Liners. No repair of backup liners is authorized. Replace any damaged liner.

2-315. Installation — CR Fuel Cell Cavity Fillers. *a.* Install correct part number (dash number) filler block(s) in locations shown in figure 2-29.

b. Reinstall straps and/or gussets in original locations with same size and type rivets.

c. Install backup liner.

2-316. Installation — CR Fuel Cell Backup Liners. *a.* With cavity fillers and straps installed, roll liner and position in fuel cell as shown in figure 2-29.

NOTE

If a replacement liner is being installed, use the old liner as a template to locate attachment holes and cutouts. Drill and trim replacement liner before installation.

b. Install liner attachment screws and washers.

2-317. FLOOR SUPPORT LONGERONS.

2-318. Description — Floor Support Longerons. The two floor support longerons (21, fig. 2-20) are structural members routed from the pilot's floor support bulkhead (26), along the fuselage contour back to the lower section ring (15). The longerons are formed and chemically milled extrusions that provide the outboard support for the floor structure.

2-319. Inspection — Forward Ends of Floor Support Longerons. Open the pilot's compartment floor access doors, inspect accessible length of the floor longerons for secure installation, corrosion, distortion, and breaks or cracks. Close access doors after inspection.

2-320. Inspection — Aft Ends of Floor Support Longerons. Open the engine compartment access doors. Inspect accessible length of the floor longerons for condition as in paragraph 2-319. After inspection, close engine access doors.

2-321. Repair or Replacement — Floor Support Longerons (AVIM). The left and right floor support longerons are chemically milled 2014-T6 aluminum extrusions riveted along the fuselage mold line, to bulkhead station 44.64, to ring station 137.50, and to other structural members of the fuselage lower section. The outboard edges of the pilot's floor and cargo floor are riveted to the horizontal leg of the extrusions. The fuselage lower section skin panels are riveted along the lower edge of the vertical web. Structural repair of the floor support longerons requires removal of the fuselage lower section skin panel from the damaged section of the longeron. Insert or overlay repairs to damaged areas of the floor support longerons are limited to 4 inches after cleanup and must be made according to TM 55-1500-204-25/1 repair instructions for stressed areas. Cracks may be repaired as follows:

NOTE

Replacement of floor support longerons is not recommended as a maintenance function as it would involve removal of the entire lower section from the fuselage. Refer to table 2-4, items 6 and 9, for repair materials.

a. Remove fuselage skin panel from section of floor support longeron where damage is located. Refer to paragraph 2-264.

b. Stop-drill ends of cracks.

CAUTION

The patch angle shall be at least 2.0 inches longer than the length of the crack, and the angle sides shall match the cross-sectional dimensions of the extrusion webs at the damaged area.

c. Form an angle from item 12, table 2-3, to match cross-sectional inside contour of longeron extrusion at damaged area.

d. Rivet patch angle to inside of longeron extrusion webs.

e. Rivet removed fuselage skin panel in place.

2-322. INSTRUMENT PANEL, HOOD, AND CONSOLE.

2-323. Description — Instrument Panel, Hood, and Console. The instrument panel is a box structure formed of sheet aluminum panels. The frame of the panel is formed of riveted aluminum angle. The panel face and side panels are riveted to the angle framework. The top of the instrument panel is shaded with a formed thermoplastic hood, stiffened by aluminum tubing bonded to the edge. The sheet aluminum electrical control console assembly provides support for the intercom, radio and electrical system switch controls, and the fuel shutoff valve control. The console structure supports most of the terminal blocks for interconnecting the various electrical and electronic systems. The terminal blocks are shielded by a cover attached with turnlock fasteners.

2-324. Inspection — Instrument Panel, Hood, and Console. *a.* Check hood mounting for secure condition.

b. Check instrument panel and console attachment to fuselage structure for cracks, corrosion and loosened rivets.

c. Check console terminal block cover fasteners for positive locking, and cover for corrosion, dents and cracks.

2-325. Repair — Instrument Panel, Hood, and Console. Refer to paragraphs 2-7 and 2-269.

2-326. PILOT'S COMPARTMENT FLOOR.

2-327. Description — Pilot's Compartment Floor. The pilot's compartment floor is a structure of aluminum alloy formed channels, intercostals, clips, gussets, and skin. The pilot's seat support structure attaches to the aft end of the floor. Tail rotor control support fittings are at the forward end of the floor, and there are openings on each side for access to the underfloor compartments.

2-328. Inspection — Pilot's Compartment Floor *a.* Inspect floor for cracks, dents, gouges, and signs of excessive wear.

b. Inspect tail rotor control support fittings for cracks, loosened attachment, and wear or elongation of bolt holes (when torque tube and mounting brackets are removed.)

2-329. Repair — Pilot's Compartment Floor. Refer to paragraphs 2-7 and 2-269 for general repair information and TM 55-1500-204-25/1 for structural support and attachment repair and replacement. Table 2-4, items 1 and 2 are listed as structural repair materials.

2-330. ELECTRONIC COMPARTMENTS.

2-331. Description — Electronic Compartments. The electronic compartments are below the fuselage floor level, on both sides of the center beam and forward of the fuel cells. The compartments primarily contain avionics equipment, the battery, and electrical control components. Access to the units is provided by floor access doors on each side of the electrical control console.

2-332. Inspection — Electronic Compartments. *a.* Inspect individual structure components and mounts for cracks, breaks and stripped threads.

b. Inspect electrical harnesses and wiring for cracked or frayed insulation and grounding terminals for corrosion.

c. Inspect disconnect plugs for corrosion, and signs of arcing or burning.

d. Inspect battery area for evidence of battery leakage and corrosion on compartment structure. Refer to paragraph 1-30 for cleaning up spilled or spewed electrolyte.

2-333. Repair — Electronic Compartments. Refer to paragraphs 2-7 and 2-269 for general repair information and TM 55-1500-204-25/1 for structural support and attachment repair and replacement. Refer to table 2-3, item 2 for repair material.

2-334. PILOT'S SEAT SUPPORT STRUCTURE.

2-335. Description — Pilot's Seat Support Structure. The pilot's seat support structure is formed of aluminum alloy ribs and sheet. The seat structure incorporates forged and machined aluminum alloy fittings for seat belt attachment, landing gear damper attachment, and flight and engine control components. On aircraft requiring armor installation, aluminum alloy armor support brackets are riveted to the seat support structure beneath the pilot's and copilot's seat positions.

2-336. Inspection — Pilot's Seat Support Structure. *a.* Inspect the seat support structure for cracks, dents, and loose attaching hardware.

b. Inspect attachment fittings, forgings, and brackets for cracks and wear.

c. Inspect electrical harnesses and wiring for cracked or frayed insulation and grounding terminals for corrosion.

2-337. Repair — Pilot's Seat Support Structure. Refer to paragraphs 2-7 and 2-269 for general repair information and TM 55-1500-204-25/1 for structural support and attachment repair and replacement. Refer to table 2-4, item 16 for repair material.

2-338. FIREWALL INSTALLATION.

2-339. Description — Firewall Installation. The firewall installation (fig. 2-30) consists of two separate units; the forward section (canted) firewall as described in paragraph 2-340, and the aft section (upper) firewall as described in paragraph 2-344.

2-340. FORWARD CANTED FIREWALL.

2-341. Description — Forward Canted Firewall. The semicircular forward firewall (fig. 2-30) is in front of and perpendicular to the engine centerline. This canted firewall is a stamped semi-circular panel fabricated from 0.012- and 0.016-inch corrosion-resistant stainless steel sheet. The firewall contains circular cutouts for the engine drive shaft and engine oil cooler lines, and a reinforced rectangular opening for the engine oil cooler. The forward firewall is riveted to the fuselage structure with monel rivets.

2-342. Inspection — Forward Canted Firewall. *a.* Inspect for punctures and corrosion.

b. Inspect for a complete seal around the openings for the engine drive shaft, engine oil cooler lines, and engine oil cooler.

2-343. Repair — Forward Canted Firewall. Repair the forward firewall as described in paragraphs 2-262 and 2-269, and according to TM 55-1500-204-25/1.

a. Stop-drill cracks.

b. Trim holes to smooth edges and patch, using the same or one gage heavier material conforming to item 14, 15, or 16, table 2-3, and monel rivets. A typical repair is shown in figure 2-31.

2-344. AFT UPPER FIREWALL.

2-345. Description — Aft-Upper Firewall. Aft upper firewall blanket (fig. 2-30) extends back from the circular edge of the canted firewall in the form of a tapered shell. The blanket surrounds the upper portion of the engine combustion chamber and exhaust pipe assembly, to contain engine heat and any fire that might develop within the engine compartment. This upper firewall is fabricated from 0.0015-inch, type 321 rigidized corrosion-resistant stainless steel sheet and is covered with a ceramic fiber blanket. Flanges at the forward and lower edges of the upper firewall are covered with heat-resistant tape. Pronged fasteners attach the upper firewall to three vertical fuselage rings and to a horizontal rib.

2-346. Inspection — Aft Upper Firewall. *a.* Inspect aft firewall for security of attachment, punctures, and corrosion.

b. Inspect ceramic fiber blanket for attachment and general condition.

2-347. Removal — Aft Upper Firewall. *a.* Remove heat-resistant tape covering fasteners and along forward edges that join with the station 124 forward canted firewall.

b. Remove tailpipe assembly (chapter 4).

c. Detach compressor cooling air duct support bracket. Remove upper firewall blanket by pulling out buttonhead fasteners that attach blanket to fuselage rings and to waterline 34.96 rib.

2-348. Repair — Aft Upper Firewall (AVIM). (See fig. 2-30.) *a.* Remove upper aft firewall (para 2-347).

b. Remove tape and ceramic fiber blankets from firewall blanket.

c. Use a stiff bristle brush wet with solvent (C94) to accomplish preliminary cleaning.

d. Wipe with a clean, lint-free dry cloth.

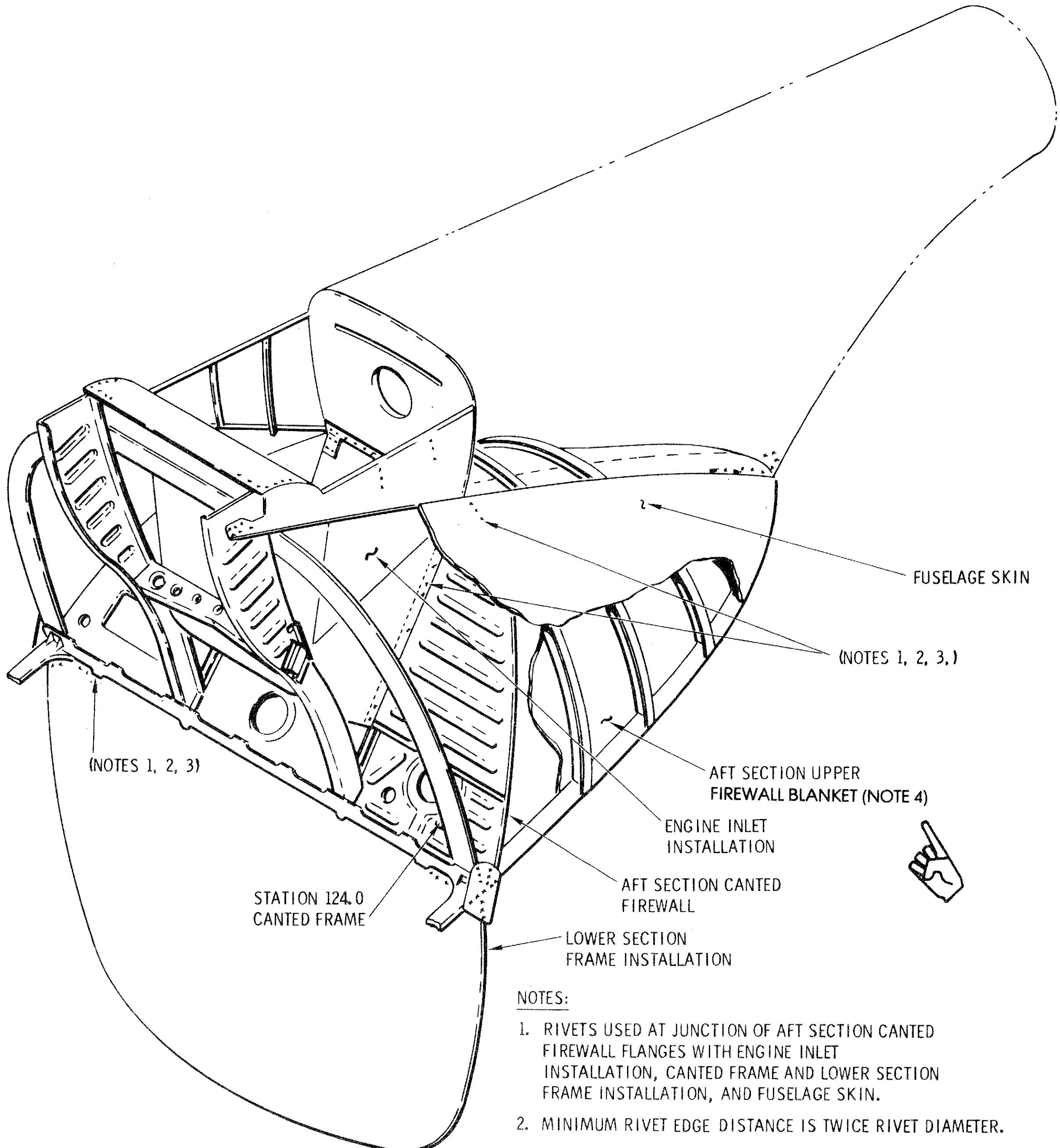
e. Smooth out the waffle pattern in the area to be repaired.

f. Prepare a cleaning solution as follows: 1- to 3-percent hydroflouric acid (C36) and 18- to 30-percent nitric acid (C71) by volume at 75 to 140°F temperature.

g. Cut a suitable repair patch from a damaged upper firewall blanket or use Type 321 stainless steel sheet of 0.0015 to 0.0018-inch thickness.

h. Contour the patch to match the contour of the firewall blanket.

i. Using a stiff bristle brush with the cleaning



NOTES:

1. RIVETS USED AT JUNCTION OF AFT SECTION CANTED FIREWALL FLANGES WITH ENGINE INLET INSTALLATION, CANTED FRAME AND LOWER SECTION FRAME INSTALLATION, AND FUSELAGE SKIN.
2. MINIMUM RIVET EDGE DISTANCE IS TWICE RIVET DIAMETER.
3. RIVET SPACING APPROXIMATELY 5/8 INCH. PICK UP FROM EXISTING PATTERN. SAME SPACING AS ORIGINAL FOR MONEL RIVETS.
4. AFT SECTION UPPER FIREWALL IS A REMOVABLE ASSEMBLY. WITHDRAW BUTTONHEAD FASTENERS FROM INSIDE SURFACE OF FIREWALL BLANKET TO REMOVE.

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Figure 2-30. Firewall Installation.

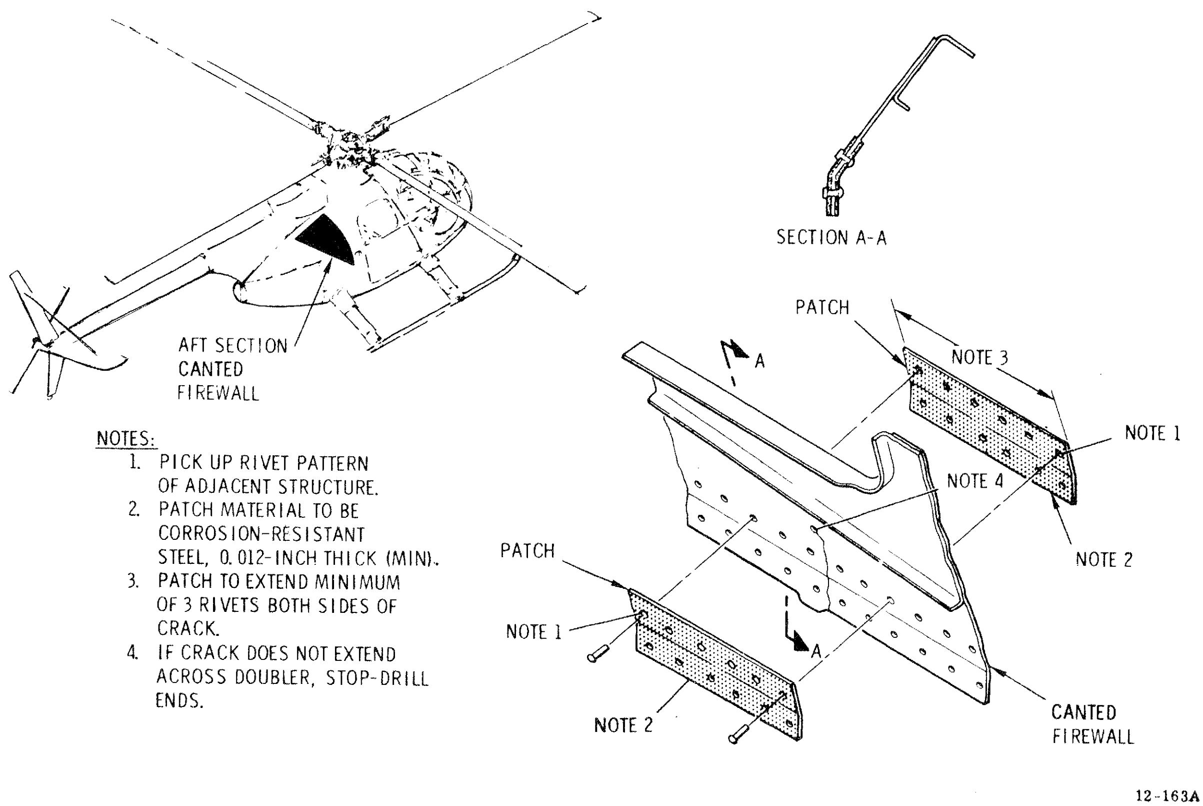


Figure 2-31. Typical Aft Section Firewall Repair.

solution prepared in *f* above, clean the inner and outer surfaces of the stainless steel patch and the firewall area to be repaired.

j. Rinse patch and firewall thoroughly with clean water, then air-dry.

k. Coat the surfaces to be joined with a thin even coating of silver alloy brazing flux (C42).

l. Use a suitable device to hold the patch in place during the brazing operation.

m. Braze patch in place with grade 4 silver brazing alloy (C17) using a suitable torch to heat the patch area to a temperature moderately above the flow point (1295°F) of the brazing alloy.

CAUTION

Do not overheat and burn through the extremely light gage (0.0015-inch thick) firewall blanket.

n. Allow joint to cool for at least 60 seconds before removing holding device.

o. Remove flux by immersing repaired area in water at 160 to 212°F for 40 minutes.

p. Follow *o* with a thorough rinse in clean, running water and air-dry, or wipe dry with a clean, dry cloth.

q. Install ceramic blanket on outer surface of firewall blanket.

NOTE

No unrepaired damage to firewall blanket is permissible. If firewall blanket cannot be repaired, it must be replaced.

r. If inspection reveals damage in the engine compartment rib area, repair according to TM 55-1500-204-25/1 and figure 2-32. Refer to table 2-3, item 4 for repair material.

2-349. Installation — Aft Upper Firewall. *a.* If a repaired firewall blanket, replace damaged or corroded fasteners.

b. If a new firewall blanket, trim blanket to fit installation. Allow suffcient trim excess for folding back the edge to provide double thickness at the attachment points.

c. Pierce fastener holes in new firewall blanket to align with holes in fuselage rings and waterline 34.96 rib. Install new buttonhead fasteners.

d. Place firewall blanket in position. Secure in place

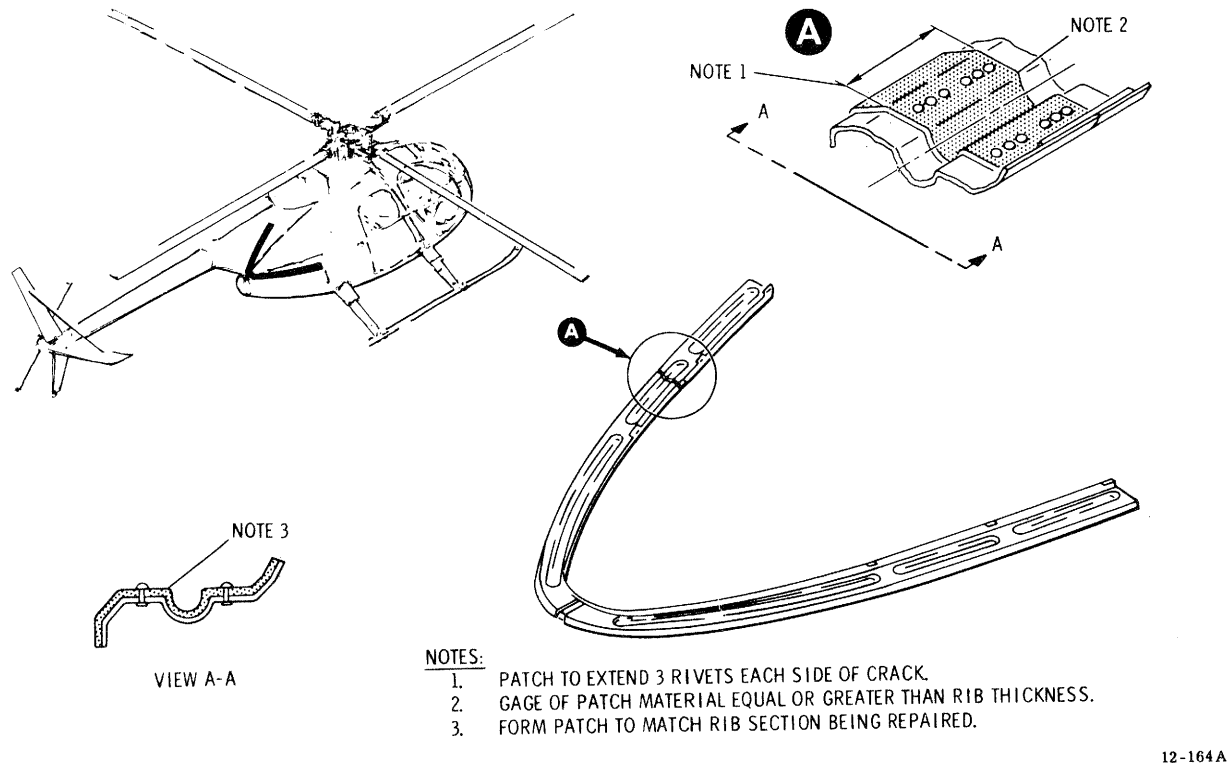


Figure 2-32. Typical Engine Compartment Rib Repair.

by pressing fasteners into holes in fuselage rings and waterline 34.96 rib.

e. Attach cooling air duct support bracket.

f. Clean the forward and lower flanges and fastener lines with solvent (C96); then apply tape (C99).

g. Install tailpipe assembly (chapter 4).

2-350. ENGINE HOIST FITTING.

2-351. Description — Engine Hoist Fitting. The hoist fitting (39, fig. 2-20) is die-forged and machined aluminum, riveted to an intercostal between two of the aft section upper structural rings. The engine hoist is attached to the fitting lug that projects through the upper firewall and into the engine compartment.

2-352. Inspection — Engine Hoist Fitting. *a.* Remove the edge tape, and enough of the upper firewall fasteners to permit access for inspection.

b. Inspect for loosened rivets.

c. Inspect for elongated or deformed lifting eye.

d. Inspect for cracks or chipping.

2-353. Repair — Engine Hoist Fitting (AVIM). Repair of the engine hoist fitting is restricted to general negligible damage as defined in paragraph 2-275. Any damage exceeding these limits or any crack, regardless of length, requires replacement of the fitting.

2-354. STATION 124.00 CANTED FRAME AND LOWER SECTION FRAME.

2-355. Description — Station 124.00 Canted Frame and Lower Section Frame. The station 124.00 upper and lower section frames (14 and 17, fig. 2-20) are primary structural members joined together at waterline 32.00 by a Y-section. The forward-canted frame is formed of channel and sheet titanium. Passenger seat, shoulder harness, and engine mount support -fittings are secured to the canted frame. The lower section frame is formed of two crescent-shaped aluminum channel members.

2-356. Inspection — Station 124.00 Canted Frame and Lower Section Frame. *a.* Inspect frame stiffeners for corrosion, distortion, cracks and secure installation.

b. Inspect frame members for corrosion, distortion, and breaks or cracks.

c. Deleted.

2-357. Repair — Station 124.00 Canted Frame and Lower Section Frame (AVIM). Refer to paragraph 2-275 for repair and replacement criteria for fittings.

a. Accomplish repair of the station 124.00 canted frame according to figure 2-33, using item 13, table 2-3 as a replacement for 2024-T3 sheet material.

b. Accomplish aft bulkhead channel repairs according to figure 2-34. Use item 13, table 2-3.

2-358. ENGINE AIR INLET (PLENUM CHAMBER).

2-359. Description — Engine Air Inlet (Plenum Chamber). The engine air inlet plenum chamber is just below the engine air inlet aft fairing. (See fig. 2-20.) On modified series 1 and 2 aircraft, an engine barrier filter is installed above the plenum chamber to provide air filtration and prevent foreign object entry. On series 3 aircraft, an inertial particle separator air filter is installed above the plenum chamber and is contained within the air inlet aft fairing. The basic air chamber is formed by a forward panel, two side panels, and a pan that extends between the lower edges of the side panels. The pan contains an opening for entry of the engine compressor bellmouth into the air chamber. The boom fairing ring at station 137.50 forms the rear wall of the plenum chamber. A diagonal strut passes through a cutout in the upper left side of the forward panel. The strut braces the structure opening from in back of the left mast support fitting to the boom fairing ring. The tail rotor drive shaft fairing passes through the chamber. Angle clips, each with a nutplate, provide attach points for either a single or double engine air shield screen.

2-360. Inspection — Engine Air Inlet. *a.* Open two access doors on right side of engine air inlet fairing, and open the engine compartment access doors.

b. Inspect all panels for evidence of corrosion, for rivet security, and for punctures.

c. Inspect the engine air shield mounting clips for secure attachment.

d. Inspect the aft section strut for rivet corrosion, and for edge clearance where it passes through the cutout in the forward panel.

e. Inspect the tail rotor drive shaft tube fairing for dents, buckled or wrinkled areas, and signs of corrosion.

f. Close the engine compartment access doors and two air inlet access doors.

2-361. Repair — Engine Air Inlet. Classification of damage and repair or replacement criteria are as outlined below and in paragraph 2-269.

2-362. Patch Repair — Engine Air Inlet. Patch cracks, tears, and holes in the outer fiberglass skin of the inlet fairing when the damage does not exceed the limits described in paragraph 2-269. The inner surface fiberglass skin is not repairable by patch methods.

2-363. Insertion Repair — Engine Air Inlet. Insertion repairs may be made to those areas of the inlet fairing where no polyurethane foam filler is present between the inner and outer fiberglass skins.

2-364. Fiberglass-to-Metal Bonding Repair — Engine Air Inlet. Damaged bonding between inlet fairing fiberglass skin and metal plates is repaired as described in paragraph 2-244.

2-365. AFT SECTION STRUT.

2-366. Description — Aft Section Strut. The aft section strut (1, fig. 2-20) consists of a tube riveted to strut fittings. The strut is approximately 28 inches long, is diagonally attached by lockbolts to the aft end of the left side mast support structure at the canted firewall (14), and to a longeron and angle in the upper right corner of the boom fairing ring (2).

2-367. Inspection — Aft Section Strut. *a.* Remove the plenum chamber access door (para 2-89).

b. Inspect strut for secure attachment to left side mast support structure at canted firewall (14, fig. 2-20) and to boom fairing ring (2).

c. Inspect tube and end fittings for cracks, and for loose or missing rivets.

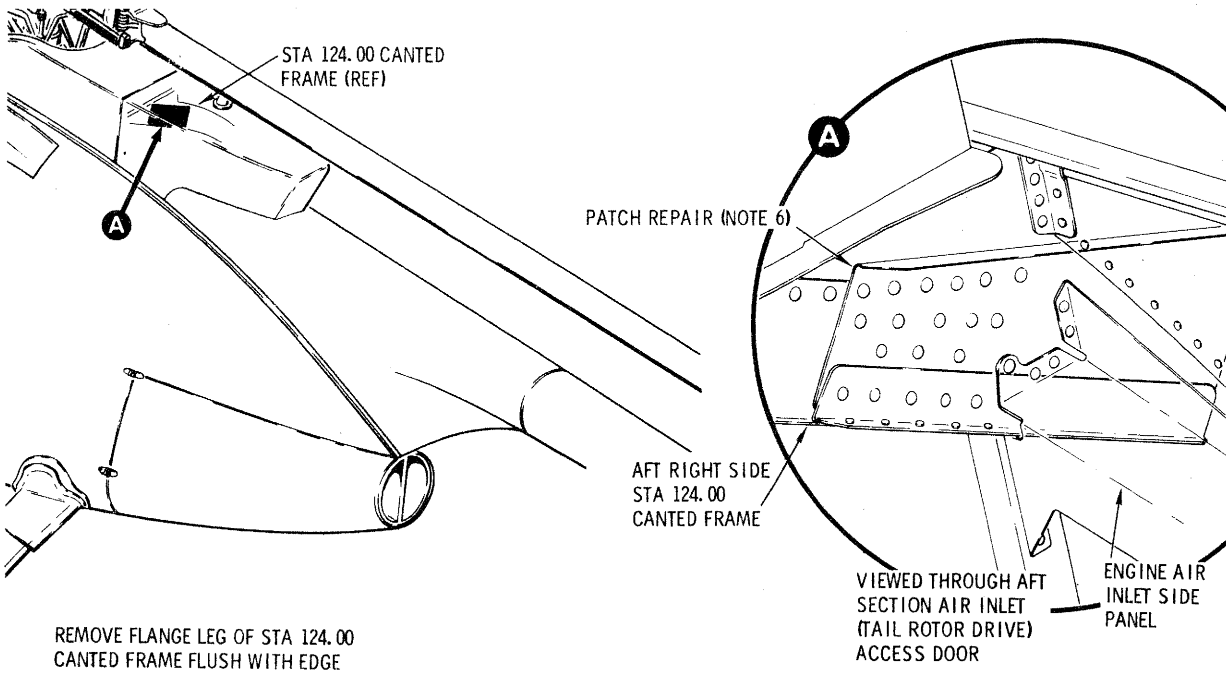
d. Inspect for scratches through protective coating, and distortion, bending, dents or other surface damage.

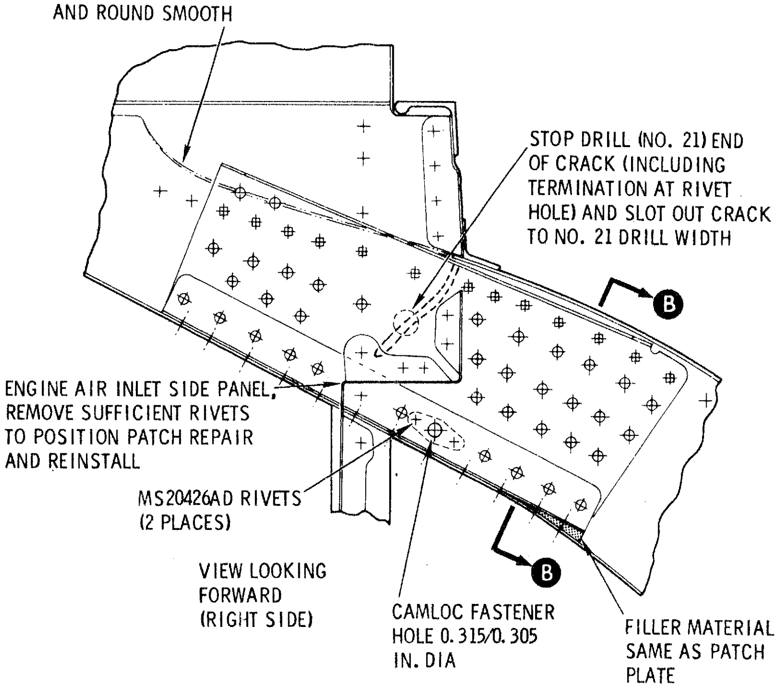
2-368. Repair — Aft Section Strut. Permissible repair of the strut is limited to smoothing out of minor dents, scratches, or nicks. If inspection reveals that the strut is badly damaged, replace the strut as follows.

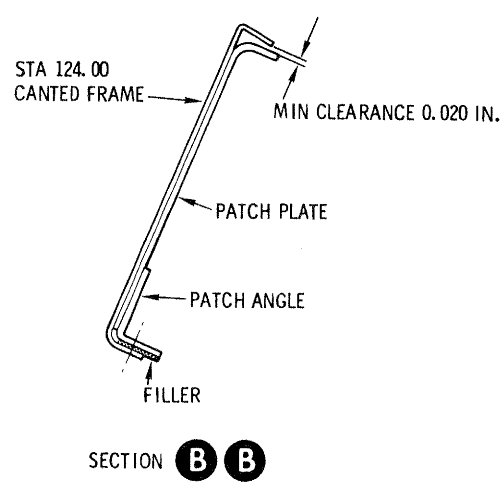
CAUTION

When maintenance work is being performed near the engine air inlet, use care to prevent entry of foreign objects. Tape covers of cardboard or other suitable material in place over the engine air inlet and oil cooler air inlets. Do not remove covers until work is completed and debris is thoroughly cleaned out of the area. After removing covers, verify that area around base of mast, inlet to plenum, and entire plenum chamber is free of foreign material. Install plenum access doors. When performing work above or near upward exhausts, install exhaust covers.

a. Remove the following parts and assemblies, in sequence:







NOTES:

1. RIVET LENGTHS ARE AS REQUIRED.
2. (+) REPLACE EXISTING RIVETS (MS20470AD5).
3. (⊕) ADD MS20470AD4 RIVETS. MAINTAIN 2X DIA EDGE DISTANCE AND 5-6X DIA SPACING.
4. (¦) ADD MS20470AD3 RIVETS THROUGH UPSTANDING REPAIR ANGLE LEG. MAINTAIN 2X DIA EDGE DISTANCE ON REPAIR ANGLE, 1.5X DIA EDGE DISTANCE ON THE CANTED FRAME, AND 7X DIA SPACING.
5. (⊕) MS20470AD4 RIVETS REPLACE EXISTING RIVETS.
6. RIGHT SIDE REPAIR SHOWN. DETAILS ARE TYPICAL FOR LEFT SIDE REPAIR.

12-329-1

Figure 2-33. Station 124.00 Canted Frame Patch Repair. (sheet 1 of 2)

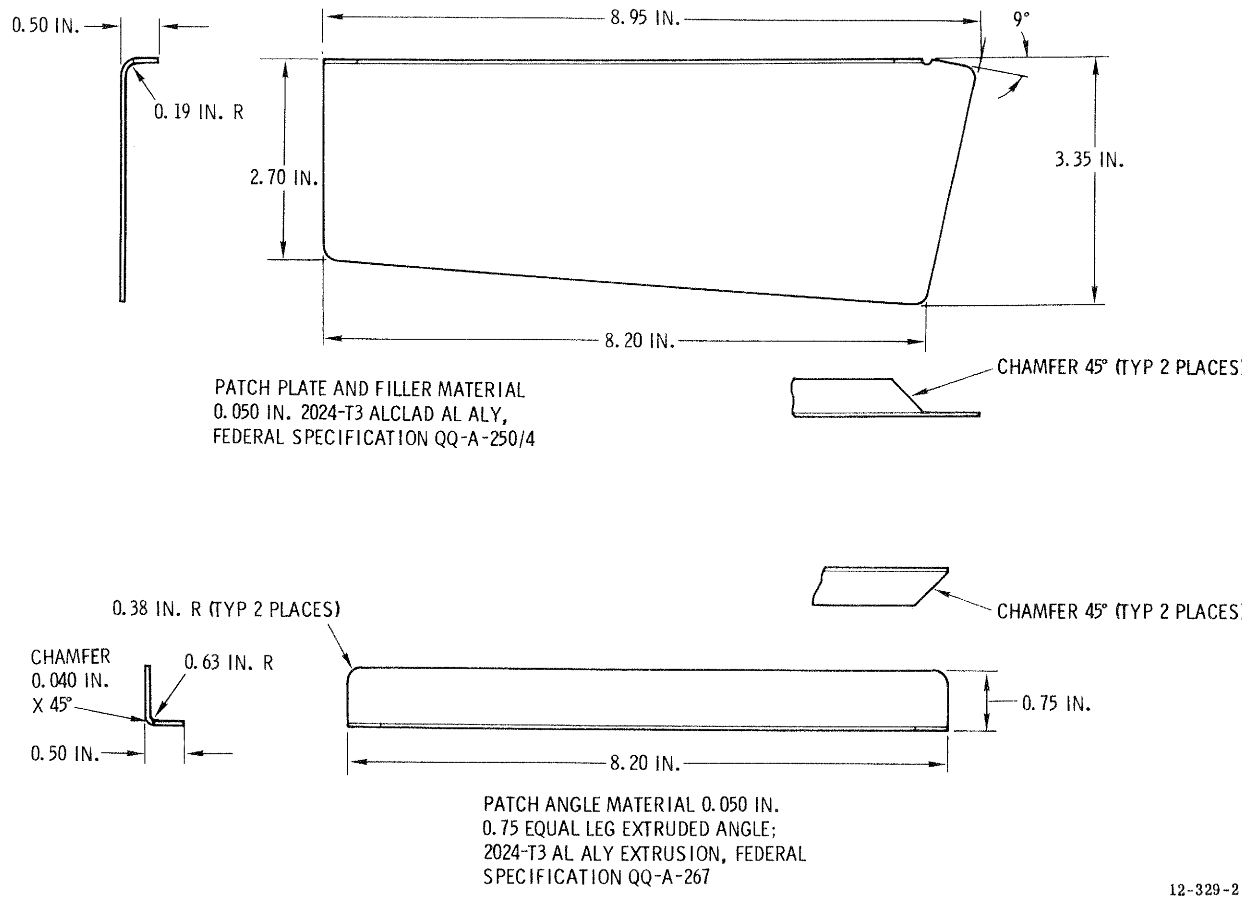


Figure 2-33. Station 124.00 Canted Frame Patch Repair. (sheet 2 of 2)

(1) Forward engine inlet fairing installation.

(2) Aft engine inlet fairing, and air filter installation (as applicable).

(3) Aft engine inlet door installation.

CAUTION

Before proceeding with replacement of the strut, install a support jack at the boom fairing jack pad. Structural damage may result if strut removal is attempted without proper support of fuselage aft section and boom weight.

b. Remove strut by removing the lockbolt collars and pins.

c. Install aft section strut replacement with lockbolts.

NOTE

The left segment of the aft engine inlet door (with oblong cut-out) must be pulled over the strut before installation.

d. Reinstall the following parts and assemblies in sequence:

(1) Aft engine inlet door installation.

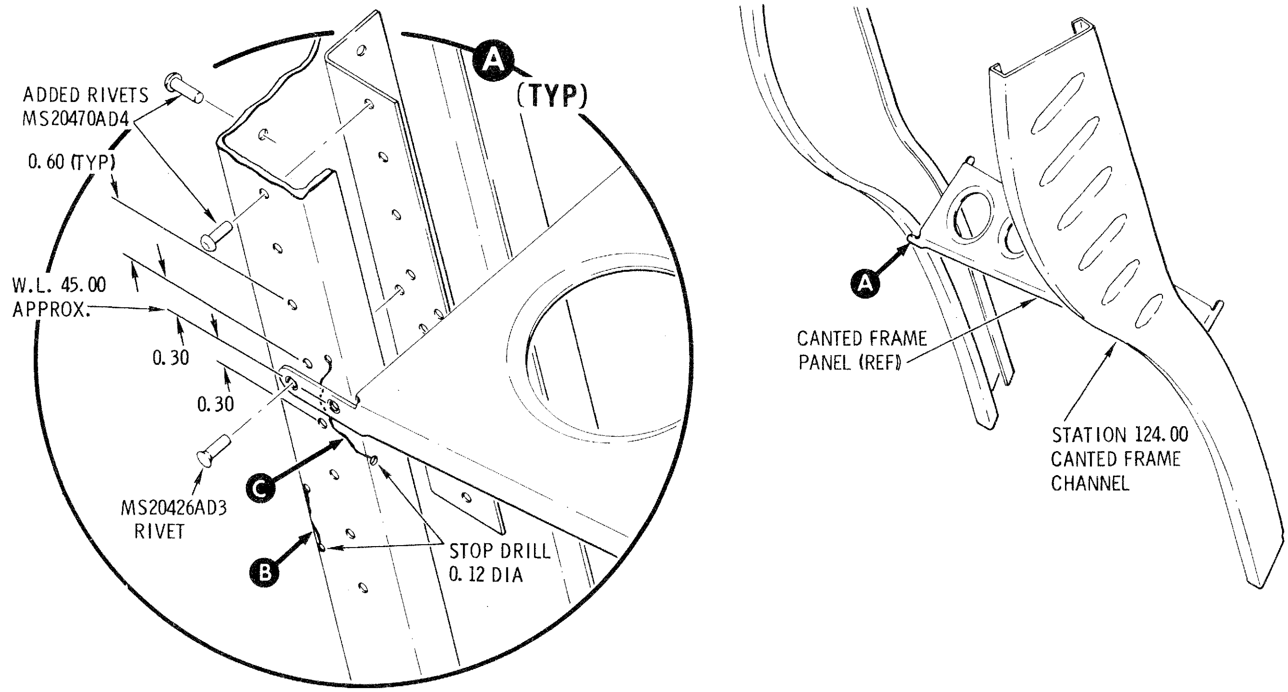
(2) Aft engine inlet fairing, and air filter installation (as applicable).

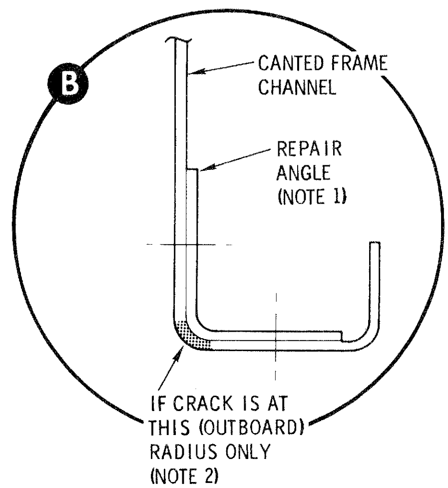
(3) Forward engine inlet fairing installation.

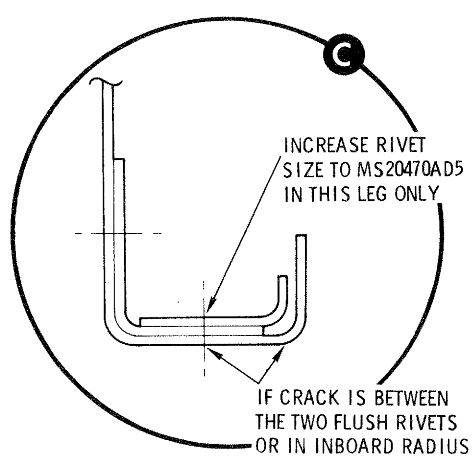
e. Remove support jack from boom fairing jack pad.

2-369. TAIL ROTOR DRIVE SHAFT FAIRING (AVIM).

2-370. General — Tail Rotor Drive Shaft Fairing (AVIM). Fairing assemblies installed on series 1 and 2 aircraft should be replaced with fairing installation 369A3021-615, shown in figure 2-35, to provide







NOTES:

1. REPAIR ANGLE MATL:
   0.051 THICK, 7075-0 FOR FORMING. HEAT TREAT TO -T6 AFTER FORMING. RIVET EDGE DISTANCE REQD IS 2X DIA.
2. IF CRACK EXTENDS INTO OUTBOARD RADIUS, USE TWO VERTICAL ROWS OF MS20470AD4 RIVETS IN OUTBOARD LEG.

12-328

Figure 2-34. Aft Bulkhead Channel Repairs.

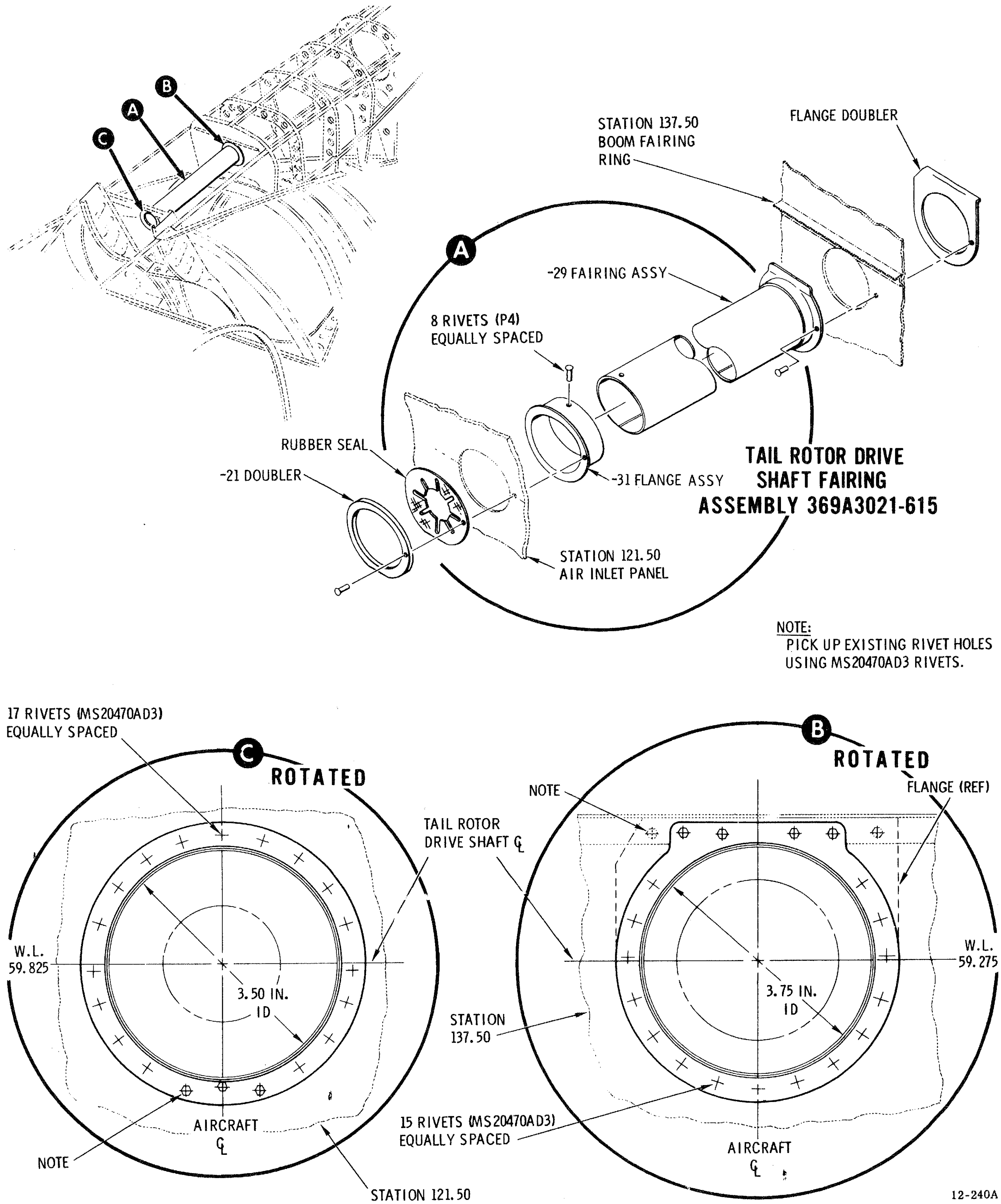


Figure 2-35. Tail Rotor Drive Shaft Fairing Repair and Replacement.

additional tail rotor drive shaft-to-fairing clearance. These fairing and flange assemblies should also be used for series 3 aircraft repair.

a. Remove engine air inlet front fairings (para 2-101) and tail rotor control bellcrank access door (para 2-95).

b. On series 1 and 2 aircraft with an engine air barrier filter, remove filter element and open the aft section engine air inlet door.

c. On series 3 aircraft with an engine air particle separator filter installed, remove filter separator as required to gain access to the tail rotor drive shaft fairing. Open the aft section engine air inlet door.

d. Remove anti-icing cable strap at center of fairing tube.

e. Remove engine air shield screen and tape a suitable covering over the engine air intake.

f. Remove rivets that secure the tail rotor drive fairing assembly to station 137.50 and 121.50 ring bulkheads and remove fairing, flanges, seal, and doublers. Retain the seal.

g. Enlarge station 137.50 ring drive shaft hole to 3.75-inch diameter and station 121.50 ring drive shaft hole to 3.50-inch diameter. Maintain hole concentricity around existing hole.

h. Position replacement fairing assembly and forward flange at bulkhead holes and locate existing rivet holes wherever possible. Drill and install fasteners.

i. Locate rivet holes as shown in figure 2-35; position doublers and drill.

j. Position rubber shield between station 121.50 doubler and air inlet panel; run drill through existing rivet holes and install rivets.

k. Position flange and fairing assemblies as shown in figure 2-35. Locate and drill eight equally spaced holes and install rivets.

l. Install all rivets at boom fairing ring and inlet panel.

m. Thoroughly clean work area and plenum chamber prior to removal of engine intake covering.

n. Reinstall assemblies removed in *a* through *e* above.

2-371. POSITION LIGHT SUPPORTS.

2-372. Inspection — Position Light Supports. Inspect for skin damage according to paragraph 2-262, and determine extent of repair necessary.

2-373. Repair — Position Light Supports. *a.* After extent of skin damage has been determined (para 2-372), repair local damage according to paragraph 2-262.

b. Form two 1.00 by 7.25-inch stiffeners from item 27, table 2-4.

c. Install stiffeners according to dimensions shown in figure 2-36.

2-374. STATION 78.50 CANTED FRAME.

2-375. Description — Station 78.50 Canted Frame. The station 78.50 aft-canted frame (bulkhead) establishes the forward portion of the cross-sectional form of the aircraft. The canted frame, constructed of formed sheet metal webs, stiffeners, and doublers, is a primary structural member of the aircraft. Three cutouts, at cargo floor level, provide access to the landing gear dampers, other underseat components and the four flight control push rods that are routed upward through the canted tunnel. A cast aluminum gunsight support fitting contains two concentrically aligned, press-fit bearings and a gunsight support nut that is secured with a roll pin. A forged aluminum gunsight torque arm fitting has two lugs with 0.25 inch holes for attachment of the torque arm from the armament subsystem. The pilot's and copilot's seat belt fittings are also located on the station 78.50 canted frame at waterline 21.50.

2-376. Inspection — Station 78.50 Canted Frame. *a.* Inspect for loose, sheared, or missing rivets.

b. Inspect sheet metal for cracks, buckled webs, and twisted or otherwise deformed parts.

c. Check fittings for secure attachment.

d. Inspect gunsight support fitting for cracks and chipping, loose or damaged bushing, defective threads, and loose or damaged rollpin.

e. Inspect all parts for corrosion.

f. Deleted.

2-377. Repair — Station 78.50 Canted Frame (AVIM). Accomplish repair of the station 78.50 canted frame according to paragraph 2-262 and TM 55-1500-204-25/1. Refer to paragraph 2-275 for repair and replacement criteria for fittings.

2-378. MAST SUPPORT STRUCTURE.

2-379. Description — Mast Support Structure. The mast support structure (fig. 2-37) consists of two mast support fittings, stiffener angles, a panel riveted to the upper-rear surface of the mast support fittings, a pan, a doubler riveted to the lower-forward surfaces of the mast support fitting, and two channels. (See fig. 2-37.) The two mast support fittings are machined aluminum alloy forgings, channel-shaped in cross section. Two bolt holes in each mast support fitting provide the four attach points for mounting the main rotor mast support base. Three additional holes through the forward end of

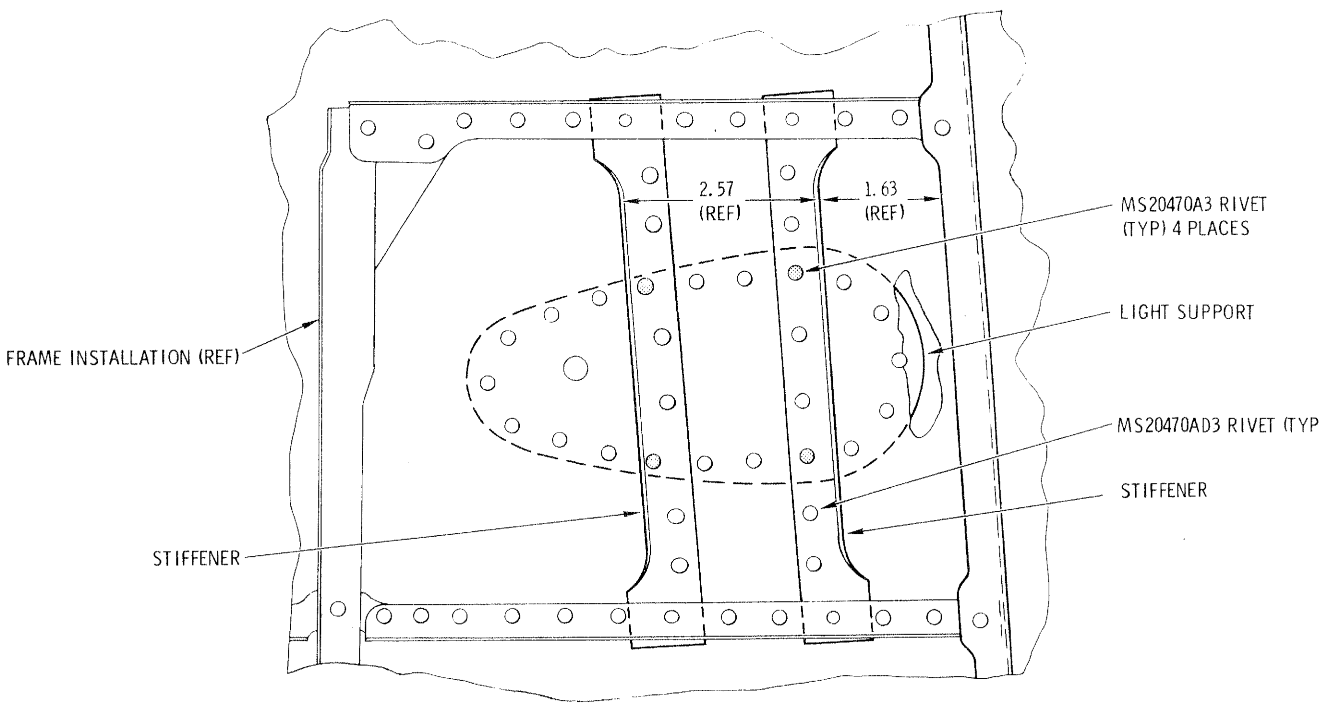


Figure 2-36. Position Light Support Repair.

the left mast support fitting provide attach points for mounting the tail rotor control rod support bracket. The forward ends of the mast support fittings are riveted to the upper ends of two channels on the aft surface of station 78.50 canted bulkhead. The aft ends of the mast support fittings are riveted to the upper ends of two channels on the forward surface of station 124.00 canted bulkhead.

CAUTION

Any time maintenance work is being performed near the engine air inlet, use care to prevent entry of foreign objects. On aircraft without an air filter, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove covers until work is completed and debris is thoroughly cleaned out of the area. After removing covers, verify that area around base of mast, inlet to plenum, and entire plenum chamber is free of foreign material.

2-380. Inspection — Mast Support Structure. Accomplish the mast support inspection as follows:

a. Remove left and right forward ends of engine air inlet fairing (para 2-101).

b. Remove main transmission cover and main transmission access cover (para 2-11 and 2-23).

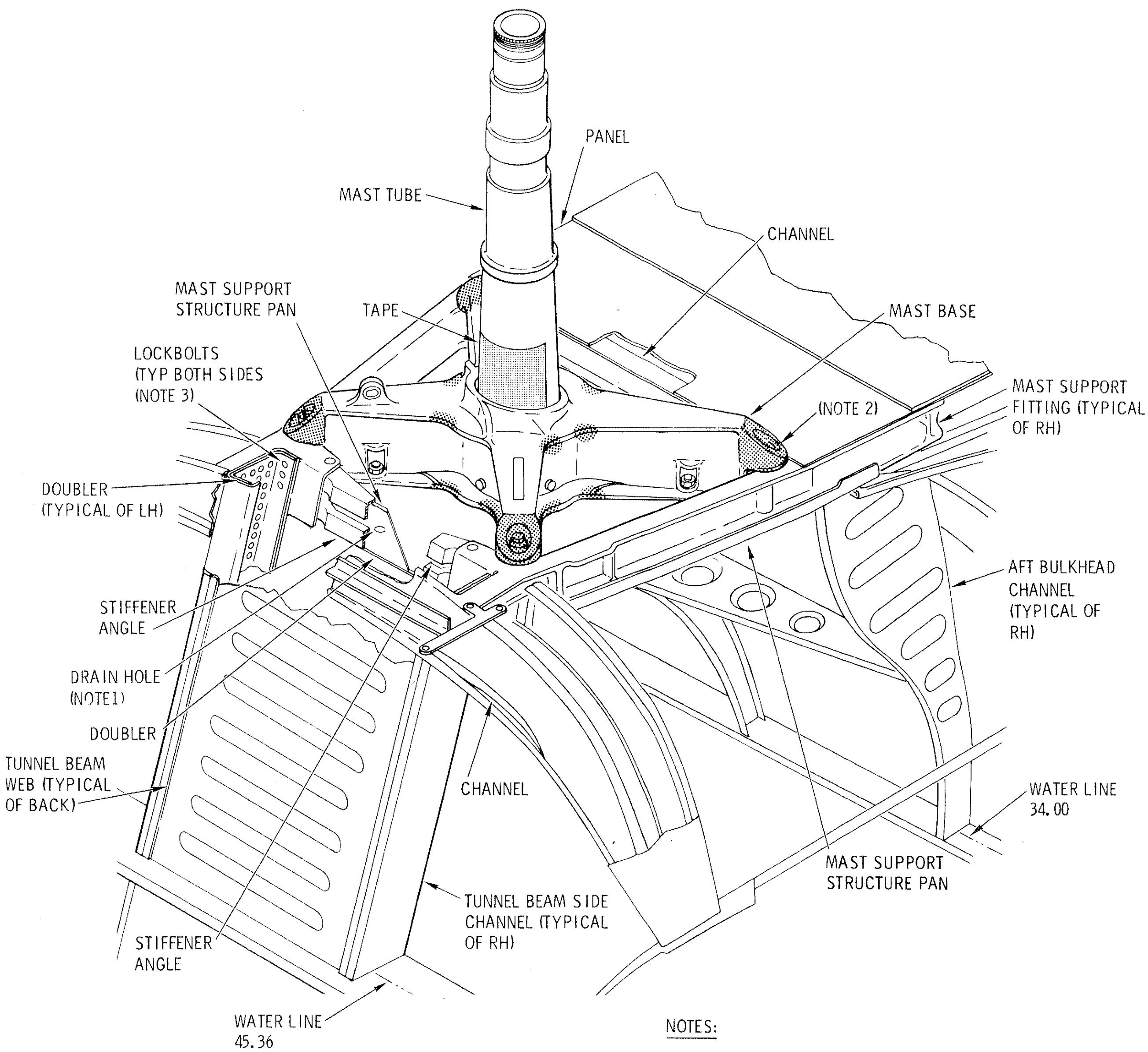
c. Detach control tunnel cover from structure. Slide cover up on control rods and secure it out of the way.

d. Using a bright light and a 5-power magnifying glass, carefully inspect the following areas for evidence of cracks. (See fig. 2-37.)

(1) Inspect the side channels and fore and aft webs of the controls tunnel, from the mast support fitting down to the top of the pilot's seatback (canted bulkhead waterline 45.36) with particular attention to lockbolts identified in figure 2-37.

(2) Inspect as much as is visible of the mast support fitting, mast base, and mast tube, with particular attention to the base attachment areas and shaded areas.

(3) Inspect both sides of the channels on the aft canted bulkhead (station 124), from the mast support fitting down to waterline 34, with particular attention to points of attachment.



NOTES:

1. SECOND HOLE IN PAN AFT OF MAST BASE.
2. SHADED SPOTS REQUIRE CLOSE INSPECTION.
3. USE A MIRROR AND FLASHLIGHT FOR CLOSE INSPECTION OF LOCKBOLTS AND ATTACHMENT AREA.

11-085B

Figure 2-37. Mast Support Structure.

(4) Clean any questionable area; use fluorescent penetrant to determine if a crack does exist. If a crack is found, the affected part must be replaced.

e. Inspect mast support structure for corrosion, loose boles and rivets, and general condition of finish.

f. Check that the two 0.25 inch drain holes in the pan are not plugged.

g. Reinstall left and right forward ends of engine air inlet fairing (para 2-101).

h. Reinstall main transmission cover and main gearbox access cover (para 2-11 and 2-23).

2-381. General Repair — Mast Support Structure (AVIM). Make general repairs to the mast support structure according to paragraph 2-262 and TM 55-1500-204-25/1.

2-382. Repair — Station 78.50 to Mast Support Fitting — Loose Lock Bolts (AVIM). (See fig. 2-37.) Affected lockbolts are located at butt line 6.00 and station 94.00 at the top end of the canted station 78.50 channels at the channel-to-mast support fitting attach points. Lockbolts are installed through left and right side channels.

a. Remove engine air inlet front fairings (para 2-101).

b. Install a cover in plenum chamber to prevent foreign objects from entering engine air inlet.

c. Check for loose of missing lockbolts and/or lockbolt collars. Remove defective lockbolt.

d. Inspect area for cracks, using a 5-power magnifying glass. If any cracks are found, repair according to procedures in paragraph 2-269 and TM 55-1500-204-25/1.

e. If no cracks are found but the original lockbolt (NAS 1425-3) was loose, check that lockbolt hole can be cleaned up to 0.188 to 0.189-inch diameter and install next size lockbolt (NAS 1426-3); alternatively, replace with a bolt as follows.

f. Drill out hole using a size "D" (0.246) drill and ream to 0.250 inch.

g. Install a bolt (AN174H6A), two washers (AN960PD416) and nut (MS21042L4). **TORQUE BOLT TO 100 — 125 INCH-POUNDS.**

h. Remove plenum chamber cover and install engine air inlet front fairings (para 2-101).

2-383. BOOM FAIRING.

2-384. Description — Boom Fairing. The boom fairing is fabricated from a formed ring, aluminum alloy bulkheads, extruded longerons and flush-riveted aluminum alloy skin. A forged aluminum alloy frame fitting (8, fig. 2-20) with four boom attachment holes, is installed at station 197.78. The frame fitting mounts a graphite-filled teflon damper for tail rotor drive shaft support.

2-385. Inspection — Boom Fairing. *a.* Inspect fairing skin for cracks, dents and abraision and corrosion.

b. Inspect all rivet patterns for pulled or loosened condition.

c. Inspect boom attach frame fitting (8, fig. 2-20) for cracks.

d. Inspect tail boom attachment bolts for security of attachment.

e. Inspect access doors and door fasteners for serviceable condition.

2-386. General Repair — Boom Fairing. Make general repairs to the boom fairing according to paragraph 2-262 and TM 55-1500-204-25/1. Refer to table 2-3, item 2 for general repair material and table 2-4 for structural repair material.

2-387. Repair — Tail Rotor Control Rod Grommets. Replace defective tail rotor control rod grommets as follows. If tail rotor control rod grommet mounting hole in damper support bracket is enlarged, refer to figure 2-38 for doubler installation.

a. Remove tail rotor control rod (chapter 11).

b. Remove engine tailpipe assembly (chapter 4).

c. Remove aft section upper firewall shell (para 2-344).

d. Replace tail rotor control rod grommets by removing snap-on ring and pulling grommet from hole. Insert new grommet and press on new snap-on ring.

2-388. Jack Pad Fitting — Boom Fairing. The boom fairing jack pad fitting is a cast steel, domed cylinder on the underside of the boom fairing at fuselage station 197.187. Four rivets attach the jack pad flange, through the boom fairing skin, to the canted frame fitting at station 197.78. (See fig. 2-38.)

2-389. Inspection — Jack Pad Fitting. Check the jack pad fitting for secure installation, corrosion, and loosened or missing rivets. Inspect areas surrounding the jack pad for buckling, cracks or any other evidence of strain.

2-390. Replacement — Jack Pad Fitting. Drill out four rivets (fig. 2-38) to remove damaged fitting. Rivet new fitting in place.

2-391. Repair — Tail Rotor Drive Shaft Damper Support Bracket. The damper support bracket is alclad sheet, riveted to the boom fairing frame fitting (fig. 2-38). Cracks up to 0.130 inch are permissible if drilled out immediately, and inspected periodically for extension. Cracks that continue to extend, or breaks and

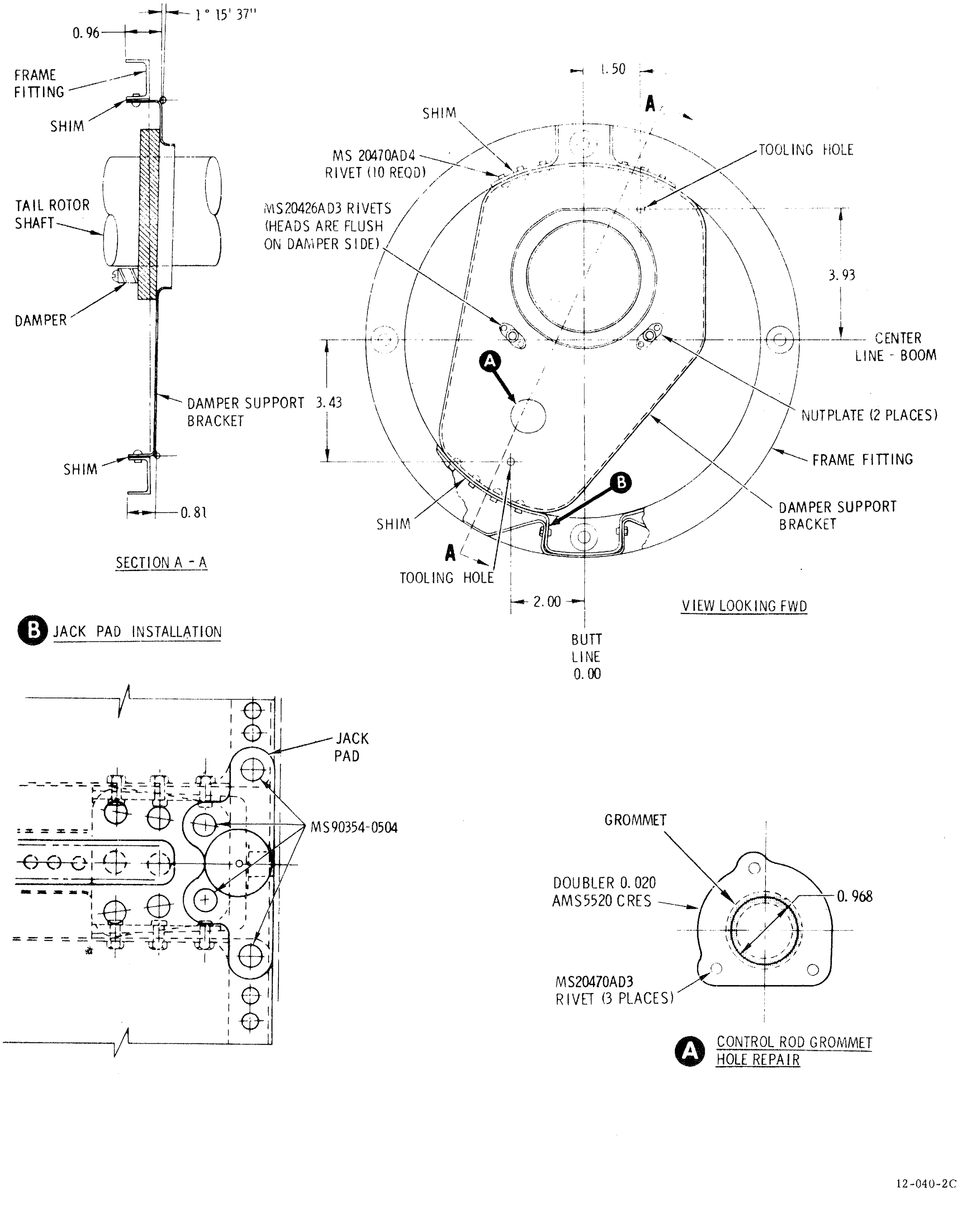


Figure 2-38. Boom Fairing Repairs.

holes may be patched. Areas that cannot be safely patched require replacement of the complete support bracket, as follows.

a. Remove tailboom (para 2-432) and place in suitable repair cradles.

b. Remove drive shaft damper (chapter 6).

c. Drill out rivets that secure bracket to boom fairing frame fittings, remove bracket and discard if unserviceable.

d. If undamaged, retain upper and lower alclad shims located between bracket and frame, and control rod grommet for reuse.

e. Clamp replacement bracket in position on frame fitting. Use care to align bracket according to the dimensions shown in figure 2-38.

f. Drill holes for damper support bracket attachment and install rivets.

g. Reinstall tailboom (para 2-432).

2-392. Repair — Boom Fairing Bellcrank Support (AVIM). The boom fairing bellcrank support consists of two aluminum alloy sheets mounted between stations 137.50 and 146.62 (inside the tail rotor control bellcrank access door). Repair cracks according to paragraph 2-269. Cracks that continue to grow and breaks or holes shall be patched. Areas that cannot be safely patched shall be repaired by partial or complete replacement.

CAUTION

Use flush rivets where clearance for bellcrank movement is required.

2-393. AFT LANDING GEAR DAMPER AND SIDE ENGINE MOUNT (LOWER) FITTING.

2-394. General — Aft Landing Gear Damper and Side Engine Mount (Lower) Fittings. The side engine mount (lower leg) fittings (fig. 2-24, sh 2) include the landing gear aft damper attach points. Any damage in excess of negligible limits given in paragraph 2-275, or any crack, regardless of length, requires replacement of the fitting.

2-395. Replacement — Aft Landing Gear Damper and Side Engine Mount (Lower) Fitting (AVIM).

NOTE

Build-up aligning tool (T8) is required.

a. Remove engine (chapter 4). Remove damage engine mount fitting.

b. Jack aircraft until landing gear skid tubes just clear the ground.

CAUTION

Remove and reinstall only one aft landing gear support fitting at a time to ensure proper alignment. Replacement fittings are supplied undrilled. Complete the following procedure in its entirety before removal of the opposite fitting.

c. Disconnect landing gear damper upper attachment bolt and remove electrical equipment from support fitting, if installed.

d. Using a pencil, trace around sheet metal and edges of fitting at bulkheads to assist in positioning of replacement fitting.

NOTE

During fitting removal, drill out rivets so that sheet metal stiffeners remain attached to the fuselage wherever possible. Mark and retain any shims for reinstallation.

e. Remove damaged fitting by drilling out attachment rivets.

f. Using the pencil outline traced on bulkheads and sheet metal, position the undrilled replacement fitting. Mark the rivet hole locations and check for rivet hole edge distance equivalent to the damaged fitting.

g. With the fitting in place pick up attachment holes and install monel rivets.

h. Reinstall landing gear damper and remove jacks. reinstall engine and engine mount. The engine mount lower attachment hole must remain unattached because the replacement support fitting attachment point is undrilled.

i. Locate and drill the engine mount-to-support fitting attachment hole as follows:

(1) Perform an engine-to-transmission alignment check (chapter 4).

NOTE

If alignment is not correct, place a jack or similar type equipment under engine and raise until correct engine-to-transmission alignment is obtained.

CAUTION

Enlargement of upper engine mount lower attachment hole will occur if drill bushing is not used.

(2) Insert a drill bushing (0.250 OD/0.1935 ID) in upper engine mount lower attachment hole. Using a No. 10 (0.1935-inch diameter) drill, drill through bushing to add upper engine mount lower attachment hole to oleo support fitting.

(3) Temporarily disconnect the upper engine mount at engine attach fitting (side) and swing the upper engine mount clear of the oleo support fitting hole. (Refer to chapter 4.)

(4) Using No. D (0.246-inch diameter) drill, enlarge the upper engine mount lower attachment hole drilled in the oleo support fitting. **REAM THE HOLE TO 0.250-INCH DIAMETER.**

(5) Reinstall engine mount and attach at oleo support fitting (chapter 4).

j. Recheck engine-to-transmission alignment.

k. If aft oleo landing gear damper support fitting on opposite side of aircraft is to be replaced, repeat entire procedure.

2-396. STRUCTURAL RINGS.

2-397. Description — Structural Rings. Structural rings are shown in figure 2-20. Station 137.50 aluminum boom fairing ring (2), the forward bulkhead of the boom fairing, supports the aft end of the tail rotor drive shaft fairing. At the lower end of the boom fairing, the canted firewall and engine inlet are attached. Station 137.50 corrosion-resistant-steel upper and lower section rings (13 and 15) attach to waterline 34.96 rib (10) and form the bulkhead structure between waterline 0.00 and boom fairing ring (2). Station 146.62 and 155.75 aluminum aft section rings (11 and 12) form the intermediate vertical rib structure between boom fairing rings (3 and 4) and the corrosion-resistant-steel rib (10) at waterline 34.96. Station 146.62, 155.75, 164.87, 174.00, and 185.89 aluminum alloy boom fairing rings (3, 4, 5, 6, and 7) form the upper vertical rib structure to which the boom fairing extruded aluminum longerons (9) are riveted. The bracket structure for the tail rotor control bellcrank is attached between rings (2 and 3).

2-398. Inspection — Structural Rings. *a.* Inspect for cracks, and loosened or missing rivets.

b. Inspect for scratches, nicks, and dents.

c. Inspect for distortion, bends, and other surface damage.

2-399. Repair — Structural Rings (AVIM). Repair structural rings according to paragraphs 2-262 and 2-264 and TM 55-1500-204-25/1.

2-400. SHEET STOCK STRUCTURAL REPAIR MATERIALS.

2-401. General — Sheet Stock Structural Repair Materials. Sheet stock materials used for most standard repairs are provided in table 2-3. Refer to TM 55-1520-214-23P for alternate or next gage material.

2-402. STANDARD STOCK EXTRUSION AND FORMED SHAPE REPAIR MATERIALS.

2-403. General — Standard Stock Extrusion and Formed Shape Repair Materials. Standard stock extrusions and formed shape materials for structural repairs are provided in figures 2-39 through 2-58. Table 2-4 provides an index listing of materials with figure references. When equivalent sections can be formed from sheet or other stock, this information is given as an alternate.

SECTION II EMPENNAGE

2-404. TAIL SURFACES (EMPENNAGE).

2-405. Description — Tail Surfaces (Empennage). The empennage tail surfaces (fig. 2-59) consist of upper and lower vertical stabilizers and a horizontal stabilizer mounted at an upward angle of 25 degrees. A strut connects the horizontal stabilizer and the upper vertical stabilizer at their approximate midpoints. Tail surfaces are bolted to fittings on the aft end of the tailboom and are not adjustable.

2-406. Inspection — Upper and Lower Vertical Stabilizers. *a.* Inspect the silicone rubber seals (when installed) at the root ribs for secure bonding.

b. Inspect the metal skin for holes, cracks, loose rivets, or corrosion.

Table 2-3. Structural Repair Materials.

Item No.	Description	Thickness (inches)	Ref. No. & FSCM	NSN
1	Aluminum alloy sheet	0.016	2024-T3 QQ-A-250/4	9535-00-167-2172
2	Aluminum alloy sheet	0.016	2024-T3 QQ-A-250/5	9335-00-084-4450
3	Aluminum alloy sheet	0.020	2024-T3 QQ-A-250/5	9335-00-084-4484
4	Aluminum alloy sheet	0.025	2024-T3 QQ-A-250/5	9335-00-084-4533
5	Aluminum Alloy Sheet	0.032	2024-T3 QQ-A-250/5	9335-00-086-9729
6	Aluminum alloy sheet	0.040	2024-T3 QQ-A-250/5	9335-00-086-4551
7	Aluminum alloy sheet	0.020	6061-T6 QQ-A-250/11	9535-00-084-4415
8	Aluminum alloy sheet	0.032	6061-T6 QQ-A-250/11	9535-00-085-4133
9	Aluminum alloy sheet	0.020	7075-T6 QQ-A-250/3	9535-00-086-9808
10	Aluminum alloy sheet	0.025	7075-T6 QQ-A-250/3	9535-00-086-9864
11	Aluminum alloy sheet	0.032	7075-T6 QQ-A-250/3	9535-00-249-5811
12	Aluminum alloy sheet	0.040	7075-T6 QQ-A-250/3	9535-00-084-4581
13	Aluminum alloy sheet	0.050	7075-T6 QQ-A-250/3	9535-00-086-9465
14	Corrosion resistant steel sheet	0.012	MIL-S-5059-301CA	9515-00-529-0434
15	Corrosion resistant steel sheet	0.016	MIL-S-5059-301CA	**9515-00-203-5899**
16	Corrosion resistant steel sheet	0.018	MIL-S-6721 COMPT1	**9515-00-683-9284**
17	Titanium unalloyed sheet	0.012	MIL-T-9046c16	9535-00-596-2093

Table 2-4. Standard Stock Structural Repair Materials.

Item No.	Description	Figure Reference
1	Extruded Angle, Equal Leg, Al Aly	2-39
2	Extruded Angle, Unequal Leg, Al Aly	2-40
3	Extruded Bulb Angle, Equal Thickness, Al Aly	2-41
4	Extruded Tee, Al Aly	2-42
5	Extrusion, Al Aly	2-43
6	Extruded Tee, Bulb, Al Aly	2-44
7	Extruded Open Angle, Al Aly	2-45
8	Tee, Bulb, Unequal Thickness, Al Aly	2-46
9	Extrusion, Al Aly	2-47
10	Extruded Open Angle, Unequal Legs & Thk, Al Aly	2-48
11	Extrusion, Al Aly	2-49
12	Tee, Bulb, Equal Thickness, Al Aly	2-50
13	Extrusion, Magnesium	2-51
14	Extrusion, Al Aly	2-52
15	Extrusion, Tee, Unequal Legs, Al Aly	2-53
16	Extrusion, Unequal Legs, Al Aly	2-54
17	Extruded Open Angle, Equal Thickness Legs, Al Aly	2-55
18	Extruded Tee, Unequal Legs, Al Aly	2-56
19	Extruded Zee, Unequal Legs, Al Aly	2-57
20	Extruded Channel, Al Aly	2-58

c. Inspect attach fittings for cracks, loose rivets, or corrosion. Check that drain holes in lower stabilizer bottom rib are open.

d. Inspect lower stabilizer attachment bolts for looseness and rotation.

e. Inspect upper vertical stabilizer attach bolts for looseness and rotation.

f. Inspect tail skid for cracks, deformation, and condition of plastic sleeve between tube and bottom rib. Check security of plastic plug in end of tube (when installed).

2-407. Inspection — Horizontal Stabilizer. *a.* Inspect the silicone rubber seals (when installed at the root rib for secure bonding.

b. Inspect metal skin for holes, cracks, or corrosion.

c. Inspect attach fittings for cracks, loosened rivets, or corrosion. Inspect for relative motion at the attach points.

d. Inspect attaching bolts for looseness and rotation.

e. Inspect outboard steel plate for loosened rivets.

f. Inspect the upper surface of stabilizer for cracks around the beaded areas between the strut attach point and the end of the stabilizer. Any evidence of cracking is cause for stabilizer replacement.

g. Inspect stabilizer spar rivet line for loose rivets. Any loose rivets must be replaced with new rivets of same size and material.

2-408. Inspection — Stabilizer Strut. *a.* Inspect the strut for holes, cracks, corrosion, and secure attachment.

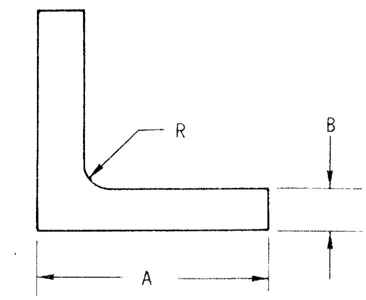
b. Inspect cushion seals at each end of strut for deterioration and secure bonding.

2-409. HORIZONTAL STABILIZER.

2-410. Description — Horizontal Stabilizer. The horizontal stabilizer (fig. 2-59) consists essentially of a forged aluminum alloy center boom attach fitting (I-beam shaped spar), a forged aluminum leading edge attach fitting, four formed sheet metal ribs, and a beaded aluminum alloy skin, riveted together to form an airfoil assembly. A lug at the spar midpoint extends through the upper skin for strut attachment. A cadmium-plated steel plate is riveted to the external surface of the outboard rib on aircraft equipped with armament. (Refer to para 2-404 for inspection.)

2-411. Removal — Horizontal Stabilizer. (See figure 2-59.) *a.* Remove tail rotor transmission and drive shaft as an assembly to provide access to stabilizer attach nuts (chapter 6).

STANDARD STOCK

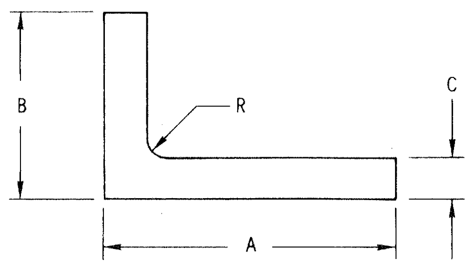


DIE NO.	DIMENSIONS A	B	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
6240	0.562	0.040	0.040	2024-T4 (QQ-A-200/3)	HS1003DD22	Aft section, top longeron; aft section fuselage structure; aft section, cargo door frame.
PA3070 (Pioneer (Aluminum)	0.750	0.062	0.125	7075-T6 (QQ-A-200/11)	HS1003AA40B	Lower section, cargo floor; fuselage structure, fuel access door.
AND10133-0601	0.750	0.062	0.094	7075-T6 (QQ-A-200/11)	HS1003AA6B	Lower section, pilots floor.
ALTERNATE STOCK						
Refer to TM 55-1500-204-25/1 for setback and bend allowance information	0.562	0.040	0.040	7178-0 (QQ-A-250/14) 7075-0 (QQ-A-250/13) 2014-0 (QQ-A-250/3)	For HS1003DD22	(Heattreat all materials to T6 condition after forming.)
	0.750	0.062	0.125 or 0.094	7178-0 (QQ-A-250/14)	For HS1003AA40B and HS1003AA6B	(Heattreat 7178 and 2014 to T6; 2024 to T3 after forming.)
		0.071		2014-0 (QQ-A-250/3)		
		0.80		2024-0 (QQ-A-250/5)		

12-177

Figure 2-39. Extruded Angle, Equal Leg, Aluminum Alloy.

STANDARD STOCK

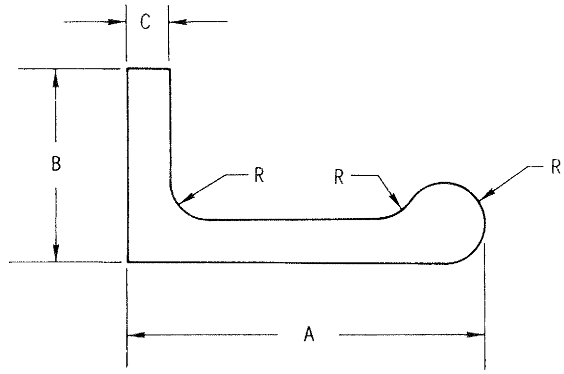


DIE NO.	DIMENSIONS A	B	C	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
	0.750	0.650	0.050	0.125	7075-T6 (QQ-A-200/11)	HS1004AA69B	Lower section, cargo floor; fuselage structure, fuel access door.
	2.250	0.750	0.060	0.060	7075-T6 (QQ-A-200/11)	HS1004AA70B	Lower section, cargo floor.
PA6924 (Pioneer Aluminum)	1.500	1.250	0.063	0.156	7075-T6 (QQ-A-200/11	HS1004AA71B	Lower section, sta 124 frame.
PA4119	1.250	0.875	0.070	0.188	7075-T6 (QQ-A-200/11)	HS1004AA72B	Lower section, sta 124 frame.
ALTERNATE STOCK							
Refer to TM 55-1500-204-25/1 for setback and bend allowance information.	0.750	0.650	0.050	0.125	7178-0 (QQ-A-250/14)	For HS1004AA69B	(NOTE: Heattreat 7178 and 2014 to T6; 2024 to T3 after forming.)
			0.060		2014-0 (QQ-A-250/3)		
			0.071		2024-0 (QQ-A-250/5)		
	2.250	0.750	0.060	0.060	7178-0 (QQ-A-250/14)	For HS1004AA70B	(See note)
			0.071		2014-0 (QQ-A-250/3)		
			0.080		2024-0 (QQ-A-250/5)		
	1.500	1.250	0.063	0.156	7178-0 (QQ-A-250/14)	For HS1004AA71B	(See note)
			0.071		2014-0 (QQ-A-250/3)		
			0.080		2024-0 (QQ-A-250/5)		
	1.250	0.875	0.070	0.188	7178-0 (QQ-A-250/14)	For HS1004AA72B	(See note)
			0.080		2014-0 (QQ-A-250/3)		

12-178

Figure 2-40. Extruded Angle, Unequal Leg, Aluminum Alloy.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
AND10135-0501	0.625	0.562	0.051	0.075	2024-T4 (QQ-A-200/3)	HS1005DD22-( )	Lower section, center beam.

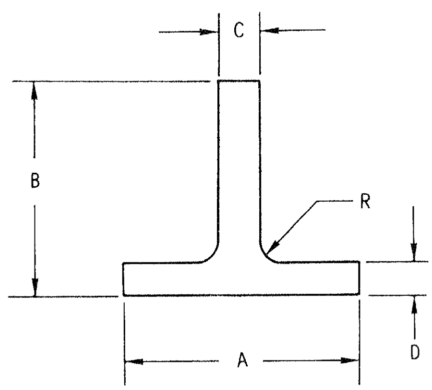
ALTERNATE STOCK

No equivalent

12-179

Figure 2-41. Extruded Bulb Angle, Equal Thickness, Aluminum Alloy.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	1.250	0.888	0.050	0.050	0.062	7075-T6 (QQ-A-200/11)	HS1006AA27B	Lower section, center beam.
–	2.750	1.000	0.062	0.062	0.094	7075-0 (QQ-A-200/11)	HS1006AA29A	Lower section, sta 124 frame.

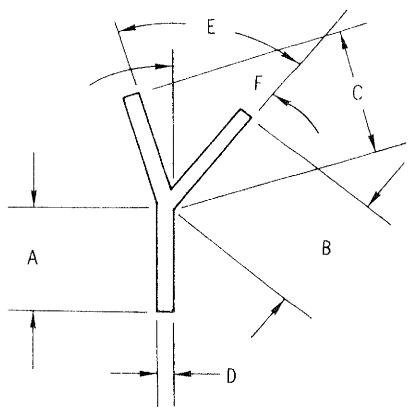
ALTERNATE STOCK

No equivalent

12-180

Figure 2-42. Extruded Tee, Aluminum Alloy.

STANDARD STOCK



DIE NO.	A	B	C	D	E	F	MATERIAL (SPEC)	PART NO.	
--	1.06	0.82	0.94	0.050	52°3'	35°9'	7075-T6 (QQ-A-200/11)	HS1011-( )	Sta 124 canted frame.

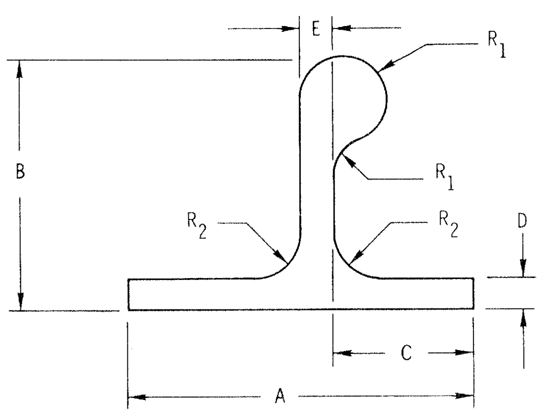
ALTERNATE STOCK

No equivalent

12-181

Figure 2-43. Extrusion, Aluminum Alloy.

STANDARD STOCK



DIE NO.	A	B	C	D	E	R_1	R_2	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
	0.700	0.700	0.460	0.040	0.040	0.075	0.040	7075-T6 (QQ-A-200/11)	HS1012AA4B	Lower section, lower longeron.

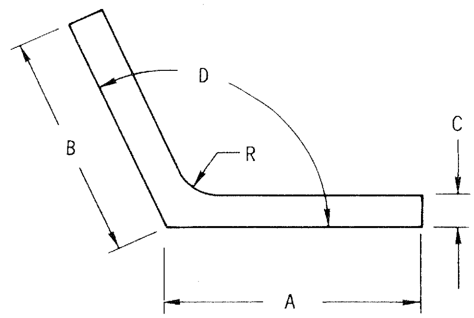
ALTERNATE STOCK

No equivalent

12-182

Figure 2-44. Extruded Tee, Bulb, Aluminum Alloy.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	1.000	1.000	0.062	94°0'	0.062	2014-T4 (QQ-A-200/2)	HS1019AB9	Lower section fuselage structure
ALTERNATE STOCK								
Refer to TM 55-1500-204-25/1 for setback and bend allowance information.	1.000	1.000	0.062 0.062 0.071	94°0'	0.062	7178-0 (QQ-A-250/14) 7075-0 (QQ-A-250/13) 2024-0 (QQ-A-250/5)	For HS1019AB9	(Heattreat 7178 and 7075 to T6; 2024 to T3 after forming.)

12-183

Figure 2-45. Extruded Open Angle, Aluminum Alloy.

b. Remove screws and washers (detail B) that attach access plates at each end of the strut.

c. Remove bolt and washer from each end of strut.

d. At root of horizontal stabilizer (detail C), loosen bolts and nuts far enough to allow the strut to clear the stabilizer attach fittings.

e. Remove the strut.

f. Disconnect taillight wiring, and remove bonding jumper (if installed). Remove three bolts, washers, nuts and bushing that attach the horizontal stabilizer. Remove horizontal stabilizer.

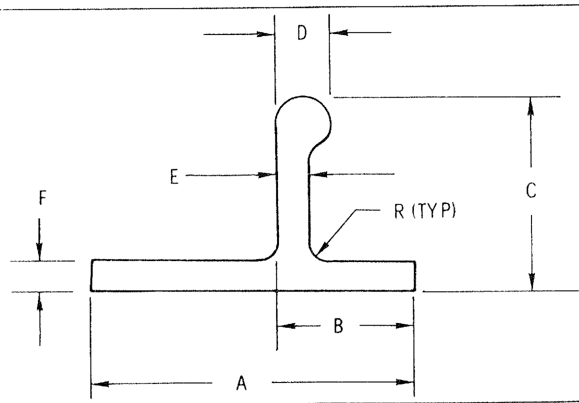
2-412. Repair — Horizontal Stabilizer. Refer to paragraph 2-429 for structural repair. Refer to table 2-3, item 2 for repair material.

NOTE

Deteriorated seals at root edges should be removed and discarded. The seals need not be replaced; however, there should be at least 0.10-inch clearance between the stabilizer skin and adjacent sheet metal. Trim skin as necessary to get clearance and apply protective finish.

2-413. Installation - Horizontal Stabilizer. (See fig. 2-59.)

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	E	F	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
AND10141-1401	1.500	0.775	0.875	0.188	0.050	0.060	0.094	7178-T6 (MIL-A-9186)	HS1030AC2B	Top center boom fairing longeron

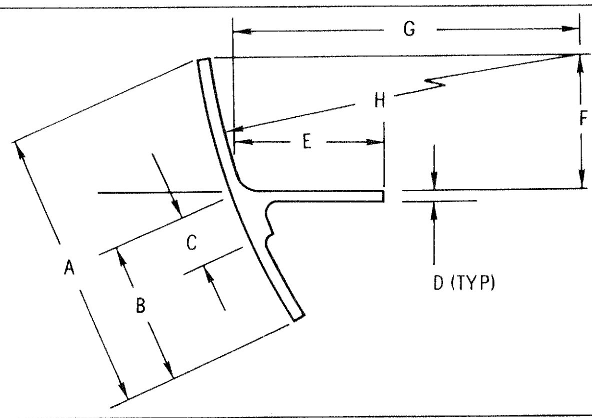
ALTERNATE STOCK

No equivalent

12-184

Figure 2-46. Tee, Bulb, Unequal Thickness.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	E	F	G	H	MATERIAL (SPEC)		INSTL OR ASSY
–	5.00	2.60	0.50	0.064	1.62	7.40	26.05	27.00R	2014-T6 (QQ-A-200/2)	HS4004	Longeron instl (lwr sect floor)

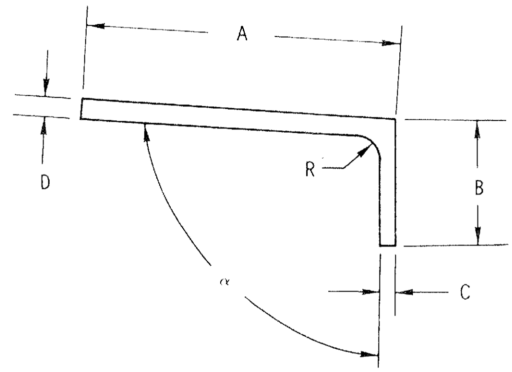
ALTERNATE STOCK

No equivalent

12-185

Figure 2-47. Extrusion, Aluminum Alloy.

STANDARD STOCK



DIE NO.	A	B	C	D	R	a	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	1.80	0.54	0.040	0.062	0.062	95°19'	7075-T6 (QQ-A-200/11)	HS1118-1( )	(Heattreat 7178 and 2014 ring.)

ALTERNATE STOCK

DIE NO.	A	B	C	D	R	a	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
Refer to TM 55-1500-204-25/1 for setback and bend allowance	1.80	0.54	0.062 0.071 0.080	0.062 0.071 0.080	0.062	95°19'	71780 (QQ-A-250/14) 2014-0 (QQ-A-250/3) 2024-0 (QQ-A-250/5)	For HS1118-1( )	(Heattreat 7178 and 2014 to T6; 2024 to T3 after forming.)

12-186

Figure 2-48. Extruded Open Angle, Unequal Legs and Thickness.

CAUTION

Be sure that countersunk washers are installed according to the following instructions. If washers are installed backwards, structural failure may result due to insufficient surface in bearing and the spreading or cracking of washers with resultant loss of clamp-up torque.

a. Place horizontal stabilizer in position on tailboom. Install one countersunk washer (detail C) and bushing on bolt; countersunk side of washer must face bolthead. Insert bolt in forward mounting hole. Do not tighten.

b. Install two countersunk washers on two external wrenching bolts with countersunk side of washers facing the bolthead. Insert external wrenching bolts through aft mounting holes and install two new nuts and washers. Do not tighten.

c. Position strut between horizontal and upper vertical stabilizers. Install washers (detail B) and bolts through access holes at each end of strut. Do not tighten.

d. **TORQUE FORWARD MOUNTING BOLT TO 50 — 70 INCH-POUNDS.**

NOTE

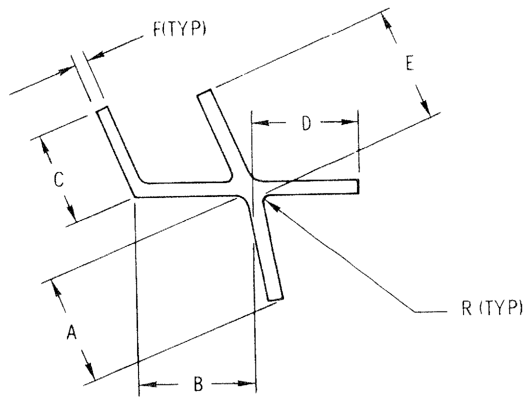
Torque values for nuts may be applied to the corresponding bolthead at the high limit of the tolerance when the nut is not accessible.

e. **TORQUE TWO AFT NUTS TO 380 — 410 INCH-POUNDS.**

f. **TIGHTEN STRUT BOLTS TO 50 — 70 INCH-POUNDS.**

g. Position access plates at each end of strut and attach with screws and washers.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	E	F	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	0.90	0.85	0.66	0.75	0.75	0.040	0.06	2024-0 (QQ-A-200/3)	HS4005A	Lower section, sta 78.50 canted frame.

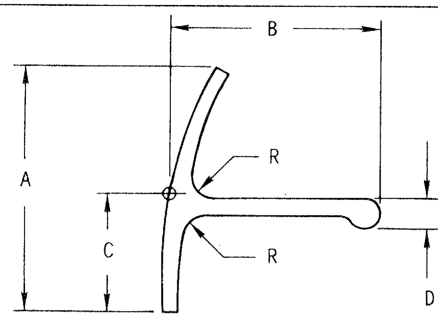
ALTERNATE STOCK

No equivalent

12-187

Figure 2-49. Extrusion, Aluminum Alloy.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	1.375	1.000	0.700	0.125	0.060	7075-T4 (QQ-A-200/11)	HS4006AA1-( )	Boom fairing, upper longeron.

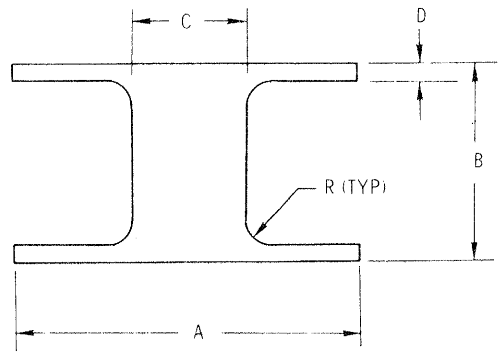
ALTERNATE STOCK

No equivalent

12-188

Figure 2-50. Tee, Bulb, Equal Thickness.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	1.30	0.767	0.500	0.050	0.010	AZ31B-F (QQ-M-31B)	HS4011	Pilot's and cargo doors.
	1.30	0.783	0.520	0.060	0.030	AZ31B-F (QQ-M-31B)	HS4011	Pilot's and cargo doors.

ALTERNATE STOCK

No equivalent

12-189

Figure 2-51. Extrusion, Magnesium.

h. Connect tail light wiring.

i. Attach bonding jumper, as required.

j. Reinstall tail rotor drive shaft, gearbox and tail rotor assembly (chapter 6).

2-414. UPPER VERTICAL STABILIZER.

2-415. Description — Upper Vertical Stabilizer. The upper vertical stabilizer (fig. 2-59) is an airfoil consisting of beaded aluminum alloy sheet metal skin supported by a forged aluminum center boom attach fitting, a forward boom aluminum attach fitting and five formed sheet metal ribs. The airfoil on most upper stabilizers has a 5-degree twist which improves tail rotor pedal neutral position during cruise flight. A lug at the boom fitting midpoint extends through the right side skin for strut attachment. (Refer to para 2-404 for inspection.)

2-416. Removal — Upper Vertical Stabilizer. (See fig. 2-59.) *a.* Remove strut (para 2-424).

b. Detach bond jumper (if installed).

c. Remove forward attach bolt, washers and bushing (detail A).

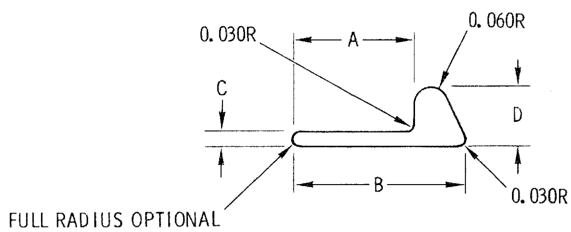
d. Remove two aft bolts, countersunk washers, washers and nuts that secure upper vertical stabilizer to tailboom structure; remove stabilizer.

2-417. Repair — Upper Vertical Stabilizer. Refer to paragraph 2-429 for structural repair. Refer to table 2-3, item 2 for repair material.

NOTE

Deteriorated seals at root edges should be removed and discarded. The seals need not be replaced; however, there should be at least 0.10-inch clearance between the stabilizer skin and adjacent sheet metal. Trim skin as necessary to get clearance and apply protective finish.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	0.50	0.68	0.040	0.22	2014-0 (QQ-A-200/2)	HS4010-1A-( )	Cargo door frame. Lower section station 78.50 canted frame. (Heattreat 2014-0 to T6 after forming.)

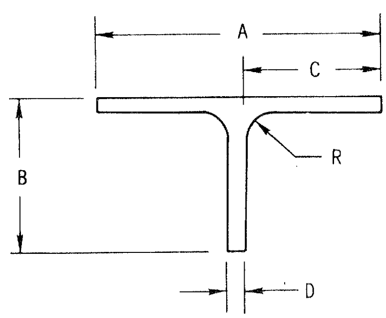
ALTERNATE STOCK

No equivalent

12-200

Figure 2-52. Extrusion,Aluminum Alloy.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	R	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	2.700	0.750	1.280	0.050	0.080	7075-T6 (QQ-A-200/11)	HS4013-1	Aft section, boom fairing
	2.700	1.280	0.85	0.050	0.080	7075-T6 (QQ-A-200/11)	HS4013-3	Aft section, boom fairing

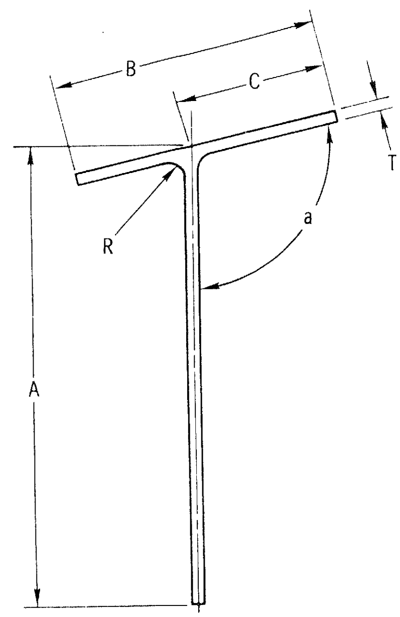
ALTERNATE STOCK

No equivalent

12-190

Figure 2-53. Extrusion, Tee, Unequal Legs, Aluminum Alloy.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	C	D	T	a	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
–	2.25	1.30	0.72	0.06	0.050	1.03°	7055-T6 (QQ-A-277)	HS4003	Pilot's seat structure (lower section).

ALTERNATE STOCK

No equivalent

12-023

Figure 2-54. Extrusion, Unequal Legs.

2-418. Installation — Upper Vertical Stabilizer. (See fig. 2-59.)

CAUTION

Be sure that countersunk washers are installed according to the following instructions. If washers are installed backwards, structural failure may result due to insufficient surface in bearing and the spreading or cracking of washers with resultant loss of clamp-up torque.

a. Place upper vertical stabilizer in position on tailboom. Install forward attach bolt, countersunk washer and bushing (detail A). Countersunk side of washer must face bolt head. Do not tighten.

b. Install aft external wrenching bolts with countersunk washers through mounting holes. Countersunk side of washer must face bolt head. Install two new nuts and washers; do not tighten.

c. Position strut between horizontal and upper vertical stabilizers. Install washers and bolts through access holes at each end of strut (detail B).

d. **TORQUE FORWARD ATTACH BOLT TO 50 — 70 INCH-POUNDS (DETAIL A).**

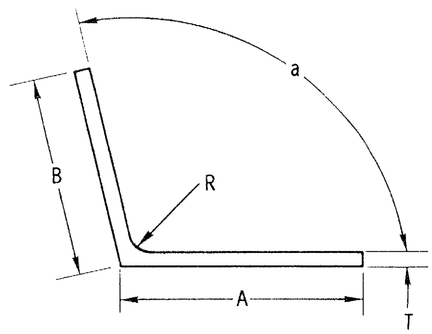
NOTE

Torque values for nuts may be applied to the corresponding bolthead at the high limit of the tolerance when the nut is not accessible.

e. **TORQUE TWO NUTS TO 170 — 200 INCH-POUNDS.**

f. **TIGHTEN STRUT BOLTS TO 50 — 70 INCH-POUNDS.**

STANDARD STOCK



DIE NO.	A	B	R	T	a	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
PA1706 (Pioneer Aluminum)	1.170	0.980	0.062	0.060	103°30'	(QQ-A-200/11)	369A2516-33	Sta. 78.50, lower

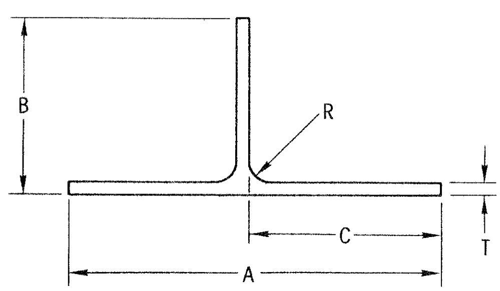
ALTERNATE STOCK

No equivalent

12-024

Figure 2-55. Extruded Open Angle, Equal Thickness Legs.

STANDARD STOCK



DIE NO.	A	B	C	R	T	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
16102 (Reynolds)	3.500	1.500	2.500	0.150	0.063	7075-T6 (QQ-A-277)	HS1021-2	Sta. 124 lower

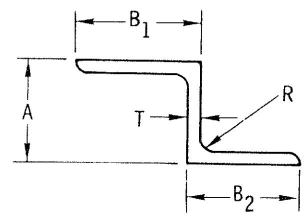
ALTERNATE STOCK

No equivalent

12-025

Figure 2-56. Extruded Tee, Unequal Legs.

STANDARD STOCK



DIE NO.	A	B_1	B_2	R	T	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
—	0.500	0.660	0.620	0.060	0.040	7075-0 (QQ-A-250/11)	HS4022	Sta. 124 lower.

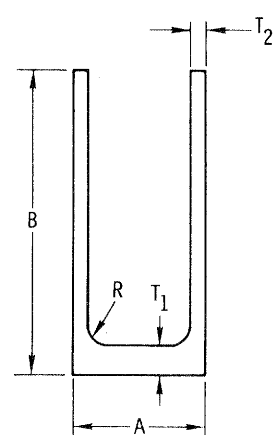
ALTERNATE STOCK

No equivalent

12-026

Figure 2-57. Extruded Tee, Unequal Legs.

STANDARD STOCK



DIE NO.	DIMENSIONS A	B	R	T_1	T_2	MATERIAL (SPEC)	PART NO.	INSTL OR ASSY
—	0.644	1.500	0.090	0.130	0.072	2024-0 (QQ-A-267)	HS1449-1	Pilot's floor (lower section.

ALTERNATE STOCK

No equivalent

12-027

Figure 2-58. Extruded Channel.

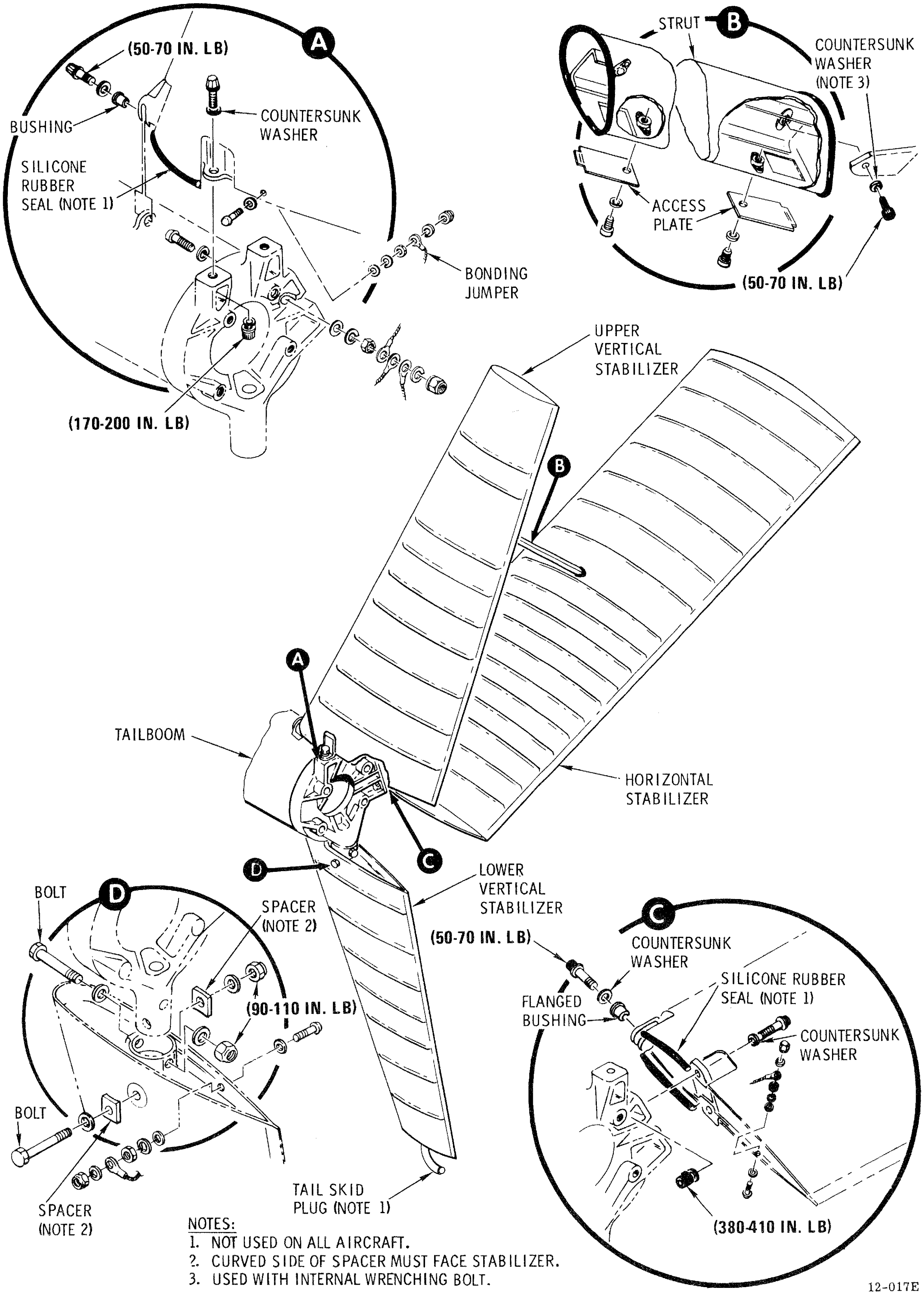


Figure 2-59. Tail Surfaces (Empennage).

g. Position access plates at each end of strut and attach with screws and washers.

h. Attach bonding jumper, as required.

2-419. LOWER VERTICAL STABILIZER.

2-420. Description — Lower Vertical Stabilizer. The lower vertical stabilizer (fig. 2-59) is an airfoil consisting essentially of beaded aluminum alloy sheet metal skin, ribs, spars, an upper fitting, and a tail skid. Two channel-shaped sheet metal spars run from the root to the tip of the stabilizer, between the ribs. The tail skid is a formed tube that tapers from its attach point in the root rib to its curved end just below the bottom rib. The upper end of the tube attaches the stabilizer to the stabilizer and gearbox mounting frame. (Refer to para 2-404 for inspection.)

2-421. Removal — Lower Vertical Stabilizer. (See fig. 2-59.)

a. Detach bond jumper (if installed).

b. Remove two nuts, washers, lower spacers and bolts (detail D).

c. Remove lower vertical stabilizer from boom.

2-422. Repair — Lower Vertical Stabilizer. Refer to paragraph 2-429 for structural repair. Refer to table 2-3, item 2 for repair material.

2-423. Installation — Lower Vertical Stabilizer. (See fig. 2-59.) *a.* When stabilizers (369A3650) are used, apply a coating of corrosion preventive compound (C31), or lubricant (C60) on the gearbox mounting frame post. When stabilizers (369A3650-601, -603, -605) are used, apply zinc chromate putty (C83) on the gearbox mounting frame post. Ensure that fiberglass liner installed in the stabilizer fitting of stabilizers (369A3650-601, -603, -605) is secure and not damaged.

b. Position stabilizer on mounting frame post and align holes in stabilizer and mounting frame.

c. Install bolt, two washers and nut at the upper attachment hole. Install a bolt, two washers and curved spacers (curved side against stabilizer) and nut as shown in detail D. **TORQUE NUTS TO 90 — 110 INCH-POUNDS.**

d. Install bonding jumper, as required.

2-424. STABILIZER STRUT.

2-425. Description — Stabilizer Strut. The strut (fig. 2-59) is a shaped airfoil section connected between the lugs of the upper vertical and horizontal stabilizer spars. A seal at the edges of each end of the strut forms a protective cushion for the skin surface of the stabilizers at the strut attach points. (Refer to para 2-404 for inspection.)

2-426. Removal — Stabilizer Strut. (See fig. 2-59.) *a.* Remove access plates (detail B) at each end of strut by removing screws and washers.

b. Remove mounting bolt and washer at each end of strut.

c. Loosen upper vertical stabilizer bolts and nuts far enough to allow the strut to clear the stabilizer attach fittings.

d. Remove the strut.

2-427. Repair — Stabilizer Strut. No structural repair is authorized for the stabilizer strut. Replace extruded rubber seals if deteriorated. Refer to paragraph 2-168 for repair or replacement of rubber extrusion seals.

2-428. Installation — Stabilizer Strut. (See fig. 2-59.) *a.* Loosen upper vertical stabilizer bolts and nuts far enough to allow positioning of strut on attach fittings of horizontal and upper vertical stabilizers.

b. Position strut and install bolt and washer at one end of the strut only.

c. **TORQUE UPPER VERTICAL STABILIZER FORWARD BOLT TO 50 — 70 INCH-POUNDS. TORQUE REMAINING TWO NUTS TO 170 — 200 INCH-POUNDS.**

CAUTION

The combined (total) deflection of horizontal and upper vertical stabilizers needed to get alignment for attachment of strut-to-stabilizer fittings must not exceed 0.25 inch. If misalignment exceeds 0.25 inch, replace the stabilizers one at a time to find cause of misalignment. Replace defective part.

d. Align strut with stabliizer hole and insert remaining bolt and washer. **TORQUE BOLT TO 50 — 70 INCH-POUNDS.**

e. Place access plates in position at each end of strut and secure with screws and washers.

2-429. CLASSIFICATION OF STABILIZER DAMAGE AND TYPES OF REPAIR.

2-430. General — Classification of Stabilizer Damage and Types of Repair. All repairable damage shall be repaired upon detection. Cracks in stabilizer skin are permissible up to 0.130 inch if drilled out and if not in a bead area. Check for extension of cracks. Any crack longer than the specified dimensions shall be stop-drilled and patched. Repair beaded skin area damage by patching according to TM 55-1500-204-25/1.

2-431. Stabilizer Attach Fittings. The upper vertical stabilizer and the horizontal stabilizer have attachment fittings which secure them at their leading edge and center on the stabilizer and tail rotor transmission mounting frame. The lower vertical stabilizer attaches to a stub fitting on the stabilizer and tail rotor transmission mounting frame. Refer to paragraph 2-446 for damage classification and repair criteria for stabilizer fittings.

2-432. TAILBOOM.

2-433. Description — Tailboom. The tailboom assembly (fig. 2-60), a monocoque structure of aluminum skin over forged aluminum frames, houses the tail rotor drive shaft and tail rotor control rod, and supports the horizontal and vertical stabilizer tail surfaces. The major fittings are the station 197.78 frame fitting, a stabilizer and gearbox mounting frame. Two lugs on the stabilizer leading edge frame support the forward ends of the horizontal and upper vertical stabilizers.

2-434. Inspection — Tailboom (Installed) *a.* Remove boom bolts access door and inspect interior of boom for moisture and corrosion.

b. Check attaching bolts and nuts in boom canted station 197.78 frame fitting for security.

c. Inspect boom canted station 197.78 frame fitting for cracks and bond jumper for security and indication of corrosion.

d. Inspect tail rotor control rod grommets in boom frames for signs of deterioration.

e. Inspect all boom assembly frames for cracks and distortion.

f. Inspect boom exterior for loosened or missing rivets.

g. Inspect exterior surfaces for wrinkles that would indicate overstress.

h. Inspect for external corrosion.

i. Inspect for bare metal areas on otherwise protected surfaces.

j. Inspect two lugs of the forward stabilizer mounting frame for cracks and elongated holes.

k. Check boom stabilizer and gearbox mounting frame for security of attachment.

2-435. Removal — Tailboom. (See fig. 2-60.) *a.* Disconnect tail light and chip detector connectors.

b. Remove tail rotor transmission (chapter 6).

c. Remove tail rotor control rod (chapter 11).

NOTE

Removal of stabilizers is not required if assembly is suitably cradled following removal. If stabilizers are to be disconnected, removal is best accomplished before the tailboom is removed.

d. If required, remove stabilizers.

e. Position suitable cradles under tailboom to hold boom in alignment with boom fairing.

f. Open boom bolts access doors.

g. If installed, disconnect adf sense antenna wire from clip on tailboom, coil the antenna, and tape coil to the forward attachment point.

h. If installed, disconnect No. 2 fm antenna splice. Remove whip antenna, module and antenna line. (Refer to TM 11-1520-214-20-1.)

i. Disconnect bond jumper at tailboom skin.

j. Disconnect taillight and chip detector splices.

CAUTION

To avoid damage, ensure that the boom is properly supported before removing attachment bolts.

k. Remove nuts, washers, and bolts attaching boom.

2-436. Inspection — Tailboom (Removed). *a.* Inspect fuselage attach frame fitting for cracks, distortion, or elongated bolt attach holes.

b. Inspect stabilizer and gearbox mounting frame for cracks, distortion, or elongated bolt attachment holes.

c. Inspect stabilizer forward mounting frame for cracks, distortion, or elongated bolt holes.

d. Perform fluorescent penetrant inspection of tailboom fitting within 0.5 inch of the bolt holes for cracks if condition is questionable.

e. Inspect the tailboom and boom fairing boom bolt holes for elongation. Maximum allowable hole diameter is 0.330 inch.

f. Visually check nuts and bolts for defective threads; countersunk washers for cracks.

g. Perform magnetic particle inspection of tailboom attachment bolts.

2-437. General Repair — Tailboom (AVIM). The two skins that form the monocoque tailboom assembly are highly stressed panels. All cracks must be stop-drilled and patched immediately. Permissible repairs are as

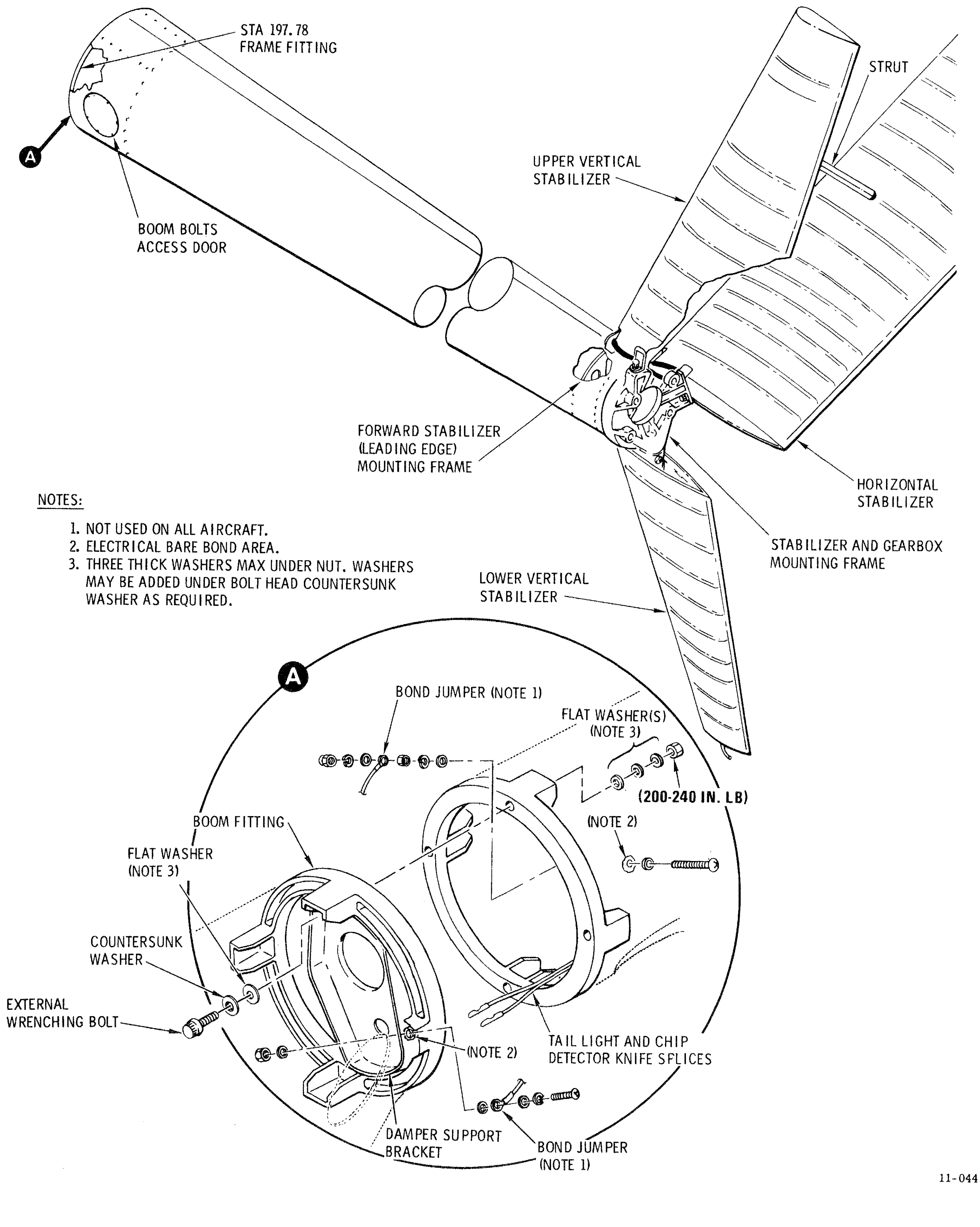


Figure 2-60. Tailboom Assembly.

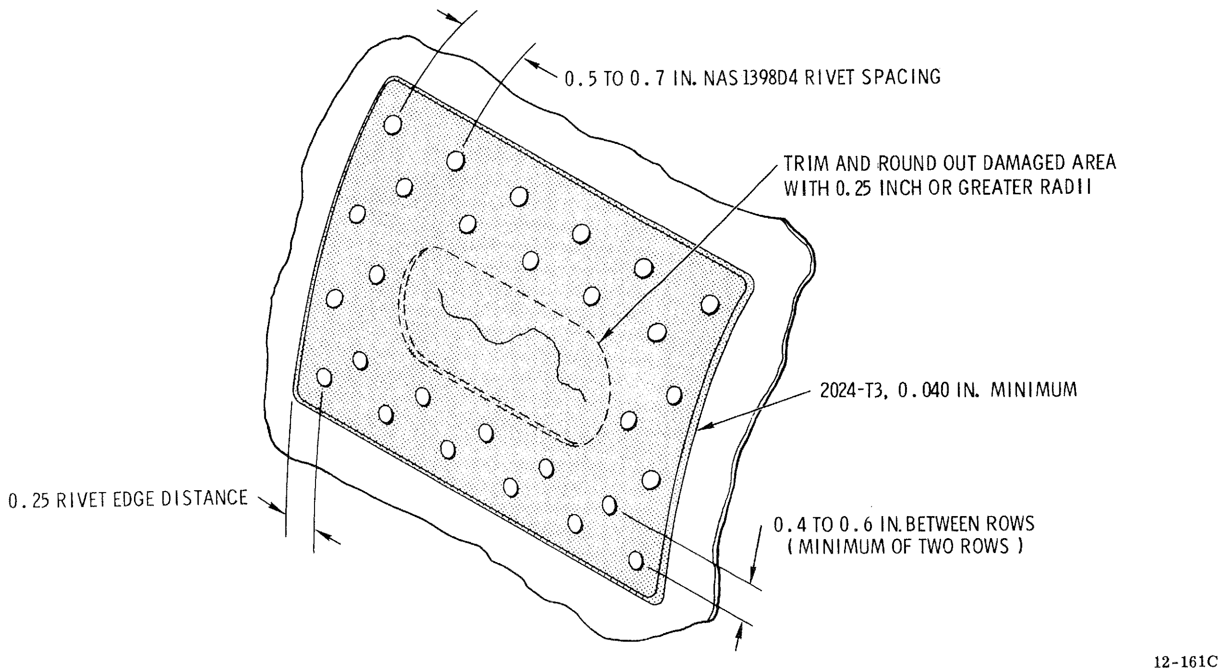


Figure 2-61. Typical Tailboom Skin Patch Repair.

given in paragraph 2-264 but with the following differences:

a. The damaged area shall not exceed 10 percent of the panel area (previously repaired areas included).

b. The patch material shall be 2024-T3 aluminum alloy of 0.040-inch minimum thickness. A typical tailboom patch repair is shown in figure 2-61. Refer to table 2-3, item 6 for repair material.

c. No insertion type patches are allowable.

2-438. Repair — Defective Tail Rotor Control Grommets and Doublers (AVIM). Tailboom grommets may be replaced without, sheetmetal work by use of the locally fabricated rods and adapter shown in figure 2-62.

NOTE

If grommet holes are worn so that grommet ring will pass through the hole, a doubler must be installed as shown in figure 2-62. During grommet installation reverse the ring to the doubler side.

a. Replace front frame grommet by removing snap-on retaining ring and pulling grommet from hole. Insert new grommet and press on snap-on retaining ring.

b. Middle frame grommet replacement:

(1) Place new grommet over the end of the 60-inch wood rod. Insert rod through the end of the tailboom and position grommet in mounting hole.

(2) Secure the snap-on retaining ring in the adapter recess. Insert the 84-inch threaded rod through the forward grommet and screw adapter on the rod.

(3) Have an assistant hold the replacement grommet in position with the wood rod. Move the metal rod aft to position the retaining ring.

(4) Tap the metal rod lightly to seat the retaining ring.

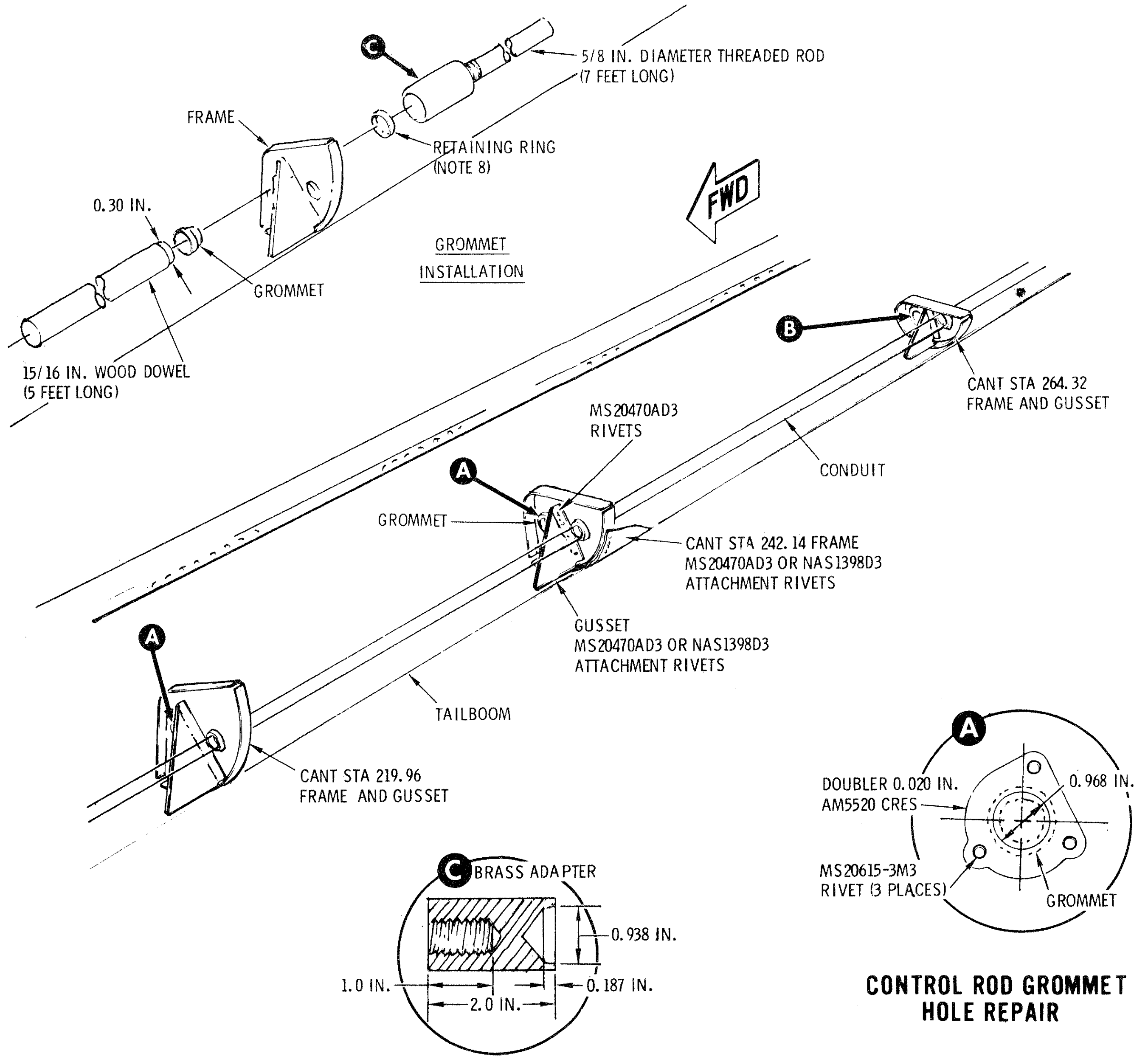
(5) Remove the rods and adapter. Inspect the grommet to verify proper seating and installation.

c. Aft frame grommet replacement: Replace by use of metal rod and adapter only.

(1) Have an assistant hold grommet in place.

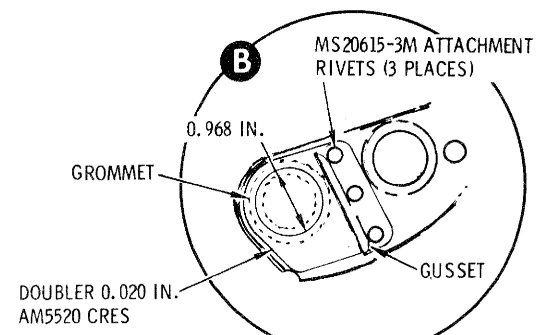
(2) Thread adapter and retaining ring on the metal rod and insert through the boom, by-passing the grommet frames.

(3) Seat ring; then inspect the ring for proper seating and installation.



NOTES:

1. FLATTEN FLARE ON END OF CONDUIT.
2. REMOVE RIVETS FROM FRAME AND GUSSET.
3. SLIDE OUT CONDUIT UNTIL FRAME IS FREE.
4. POSITION REPLACEMENT FRAME, LOCATE AND DRILL UPPER AND LOWER HOLES AND INSTALL FASTENERS. CUT ACCESS HOLE, IF REQUIRED.
5. USE TAILBOOM CONTROL ROD TO CHECK ALIGNMENT.
6. MAINTAINING EDGE DISTANCE, LOCATE AND INSTALL REMAINING RIVETS.
7. REPOSITION CONDUIT AND FLARE CONDUIT END.
8. ON AIRCRAFT WITH A GROMMET HOLE DOUBLER INSTALLED, PLACE RETAINING RINGS ON THE DOUBLER SIDE.



12-289B

Figure 2-62. Tailboom Frame and Gusset Repair.

2-439. Repair — Tailboom Sheet Metal Frame Members (AVIM). The sheet metal push-pull rod supports (fig. 2-62) mounted at station 219.96, 242.14, and 264.32 are 0.016-inch aluminum sheet. The flange of each support is formed to match the boom contour and is riveted to the inner surface of the skin. The canted frame that is riveted at station 209.78 is 0.032-inch aluminum sheet. Refer to paragraph 2-441 and see figure 2-62 for repair information.

NOTE

Holes as large as 3 by 4 inches may be cut in the tailboom skin for access to push-pull rod supports (refer to paragraph 2-441 for limitations). Refer to figure 2-61 for contoured panel skin patch installation.

2-440. Installation — Tailboom. *a.* Position boom in cradles so that mating bulkheads are flush.

CAUTION

Be sure that countersunk washers are installed as specified in *b* below. If washers are installed backwards, structural failure might result due to insufficient surface in bearing and the spreading or cracking of washers with resultant loss of clamp-up torque.

b. Slide countersunk washers on external wrenching bolts with countersunk side facing bolt heads.

c. Install bolts through aft section fuselage structure into boom assembly. Install flat washers on each bolt as required for proper bolt grip (fig. 2-60). Do not use more than three thick washers under each nut.

d. INSTALL NUTS AND TORQUE TO 200 — 240 INCH-POUNDS USING WRENCH (T40).

e. Install bond jumper (fig. 2-60).

f. Connect taillight and chip detector splices.

g. Reinstall adf sense antenna and No. 2 fm antenna as required (TM 11-1520-214-20-1).

h. If removed, install stabilizers.

i. Install tail rotor transmission (chapter 6).

j. Install tail rotor control rod (chapter 11).

k. Check rigging of tail rotor (chapter 11).

2-441. CLASSIFICATION OF TAILBOOM DAMAGE AND TYPES OF REPAIR (AVIM).

2-442. Negligible Damage — Tailboom. None. All reparable damage shall be repaired upon detection. Small cracks or punctures in sheet metal that do not exceed 0.160-inch can be repaired by drilling out with a No. 20 or smaller diameter drill and installing a rivet of appropriate size to fill the hole.

2-443. Patch Repair — Tailboom. Damage to sheet metal that exceeds 0.160-inch length is to be patched provided the damage area does not exceed a total of 25 percent of the panel area (previously repaired areas included). The patch material shall be 2024-T3, 0.040-inch minimum thickness (item 6, table 2-3), applied to the outer surface of the boom.

2-444. Repair of Transverse Crack — Tailboom. Patch with a staggered triple row of rivets, using 8 to 10 diameter spacing for the inner and outer rows, and 6 to 8 diameter spacing for the center row.

2-445. Repair of Longitudinal Crack — Tailboom. Patch with a single row of rivets or using 8 to 10 diameter spacing.

2-446. TAILBOOM FITTINGS.

2-447. General — Tailboom Fittings (AVIM). See figure 2-63 for location and identification of tailboom fittings. The tailboom fittings are listed by part number in table 2-1, as highly-stressed structural parts. Refer to paragraph 2-275 for classification of damage and repair criteria.

2-448. Stabilizer and Tail Rotor Transmission Mounting Frame — Tailboom. The stabilizer and tail rotor transmission mounting frame (fig. 2-63) is an aluminum alloy forging that supports the tail rotor gearbox and the three stabilizers. The gearbox mounting frame is attached to the boom skins. Four bracket assemblies, equipped with nutplates and attached to the boom structure for mounting of the tail rotor gearbox, are also riveted to the gearbox mounting frame. Any damage other than that classified as negligible in paragraph 2-275, and reparable as such, is cause for replacement of the frame assembly.

NOTE

Special jigs or holding fixtures may be required to hold critical dimensions during replacement of fittings. Availability of special jigs or holding fixtures should be considered before attempting major repairs.

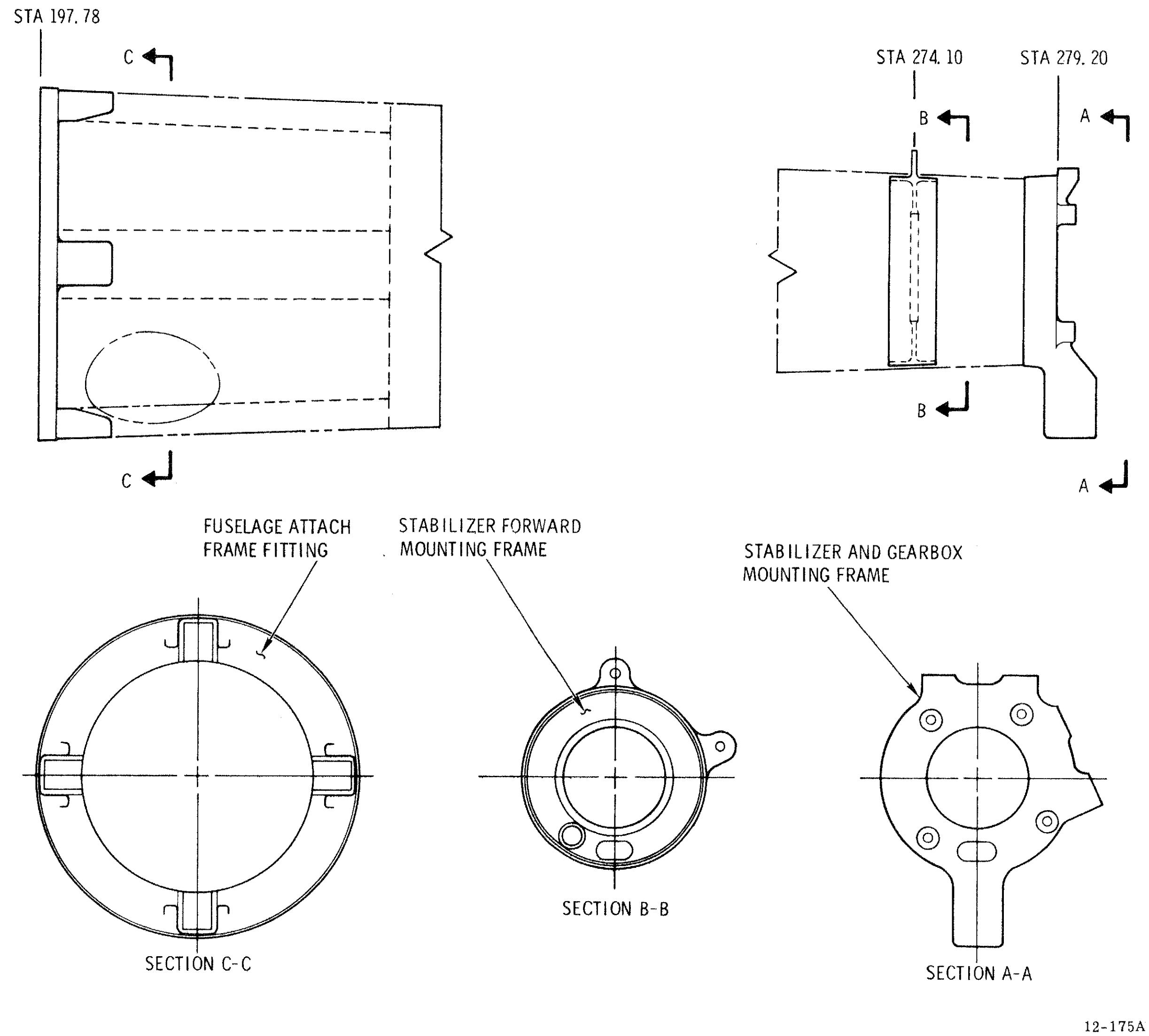


Figure 2-63. Tailboom Fittings.

2-449. Stabilizer Forward Mounting Frame — Tailboom. The stabilizer forward mounting frame (fig. 2-63) is a forged aluminum alloy frame that is riveted at station 274.10. The frame is a cylindrical-shape fitting, approximately 6.5 inches in diameter, with lugs for stabilizer leading edge attachment. Minor surface defects on stabilizer lugs, such as scratches, nicks, dents, burrs, and light corrosion deposits may be repaired by polishing with abrasive cloth (C24). Scratches, nicks, and dents in excess of 0.032 inch deep, or 0.010 inch deep within 0.125 inch of bolt hole require replacement of frame.

See NOTE (2-448) above.

2-450. Fuselage Attach Frame Fitting — Tailboom. The boom canted station 197.78 frame fitting (fig. 2-63) is a 12-inch diameter, aluminum alloy forging that mates with the fuselage boom fairing frame fitting. Any damage other than that classified as negligible in paragraph 2-275, and reparable as such, is cause for replacement of the fitting. See NOTE in paragraph 2-448.

CHAPTER THREE

ALIGHTING GEAR

SECTION I SKIDS AND STRUTS

3-1. LANDING GEAR.

3-2. Description — Landing Gear Assembly. The landing gear (fig. 3-1) consists of two strut-mounted, shock-dampened skids aligned longitudinally along the aircraft fuselage. The landing gear skids are either bolt-mounted or riveted to the landing gear struts. Both right and left skids pivot as the damper assemblies extend and retract. Either four or five replaceable abrasion strips are installed on each skid to retard skid wear on hard surfaces. Fairings on each strut reduce aerodynamic drag during flight. Landing gear braces are attached between the landing gear struts and fuselage center beam. The braces prevent shearing of the struts, and keep the struts in alignment with the landing gear. The landing gear struts are aluminum alloy forgings attached between the skids and fuselage center beam. Removal, repair, and installation procedures are identical for both sides of the landing gear installation.

Table 3-1. Premaintenance Requirements for Removal of Landing Gear.

Conditions	Requirements
Support Equipment	Jacks or Hoist
Minimum Personnel Required	Two; Three for Jacking

3-3. Removal — Landing Gear Assembly. *a.* Jack up the aircraft until the landing gear dampers are fully extended. Place supports beneath the skid tubes at the strut locations.

b. Gain access to the forward brace, strut, and damper at fuselage attachment points by opening the compartment access doors and foot support fairings, respectively.

c. Gain access to aft landing gear attachment points by opening the engine access doors.

WARNING

Exercise extreme care during bladder-type damper removal if the damper extends beyond 12.25 inches (bolt center to bolt center). This condition indicates internal failure of the damper that will result in a sudden release of internal pressure and explosion of the bladder.

d. Disconnect lower end of landing gear dampers (fig. 3-2). Remove clamp that attaches bonding jumper to strut.

e. On series 3 aircraft, remove airborne switch (fig. 3-3) from landing gear strut. (Refer to TM 11-1520-214-20-1.)

f. Pull fairing fillet downward against spring tension and secure with tape (C105).

g. Remove cotter pins, nuts, washers and bolts that attach braces to struts and support fittings, and struts to support fittings on left side of fuselage. Remove braces.

h. Remove supports from beneath left skid and carefully remove left struts, skid and fairings as an assembled unit.

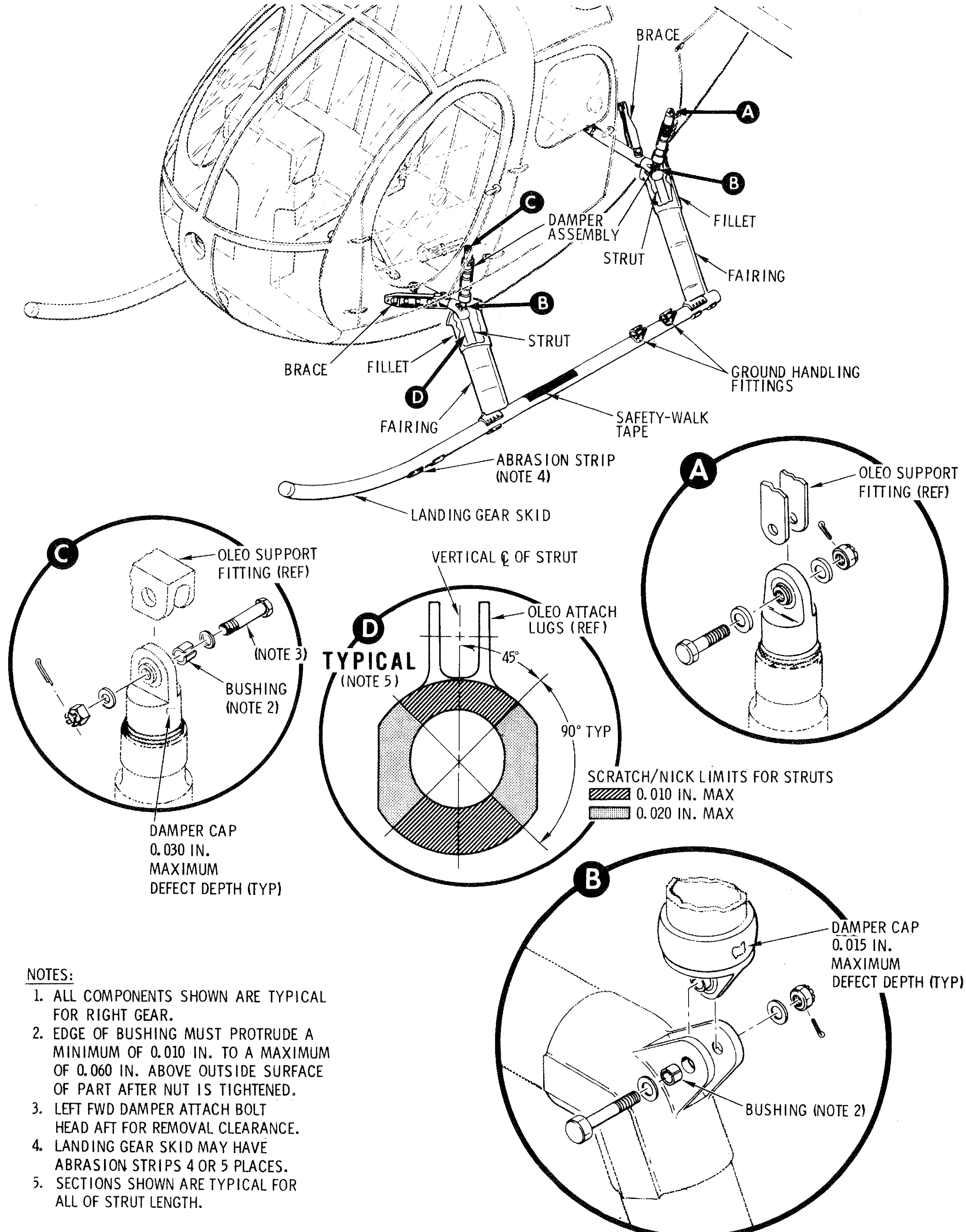
i. Repeat the sequence to remove the right landing gear.

3-4 Inspection — Landing Gear Assembly. (Refer to table 3-2.) Jack up the aircraft with the skid completely off the ground to perform the following inspection.

a. Inspect abrasion strips, struts, braces, and fairings for dents, cracks, loosened components or rivets, missing cotter pins, and loosened nuts. Repair or replace unserviceable parts.

NOTE

Abrasion strips with raised cobalt-barium inserts should not be mixed with smooth steel abrasion strips.



NOTES:

1. ALL COMPONENTS SHOWN ARE TYPICAL FOR RIGHT GEAR.
2. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
3. LEFT FWD DAMPER ATTACH BOLT HEAD AFT FOR REMOVAL CLEARANCE.
4. LANDING GEAR SKID MAY HAVE ABRASION STRIPS 4 OR 5 PLACES.
5. SECTIONS SHOWN ARE TYPICAL FOR ALL OF STRUT LENGTH.

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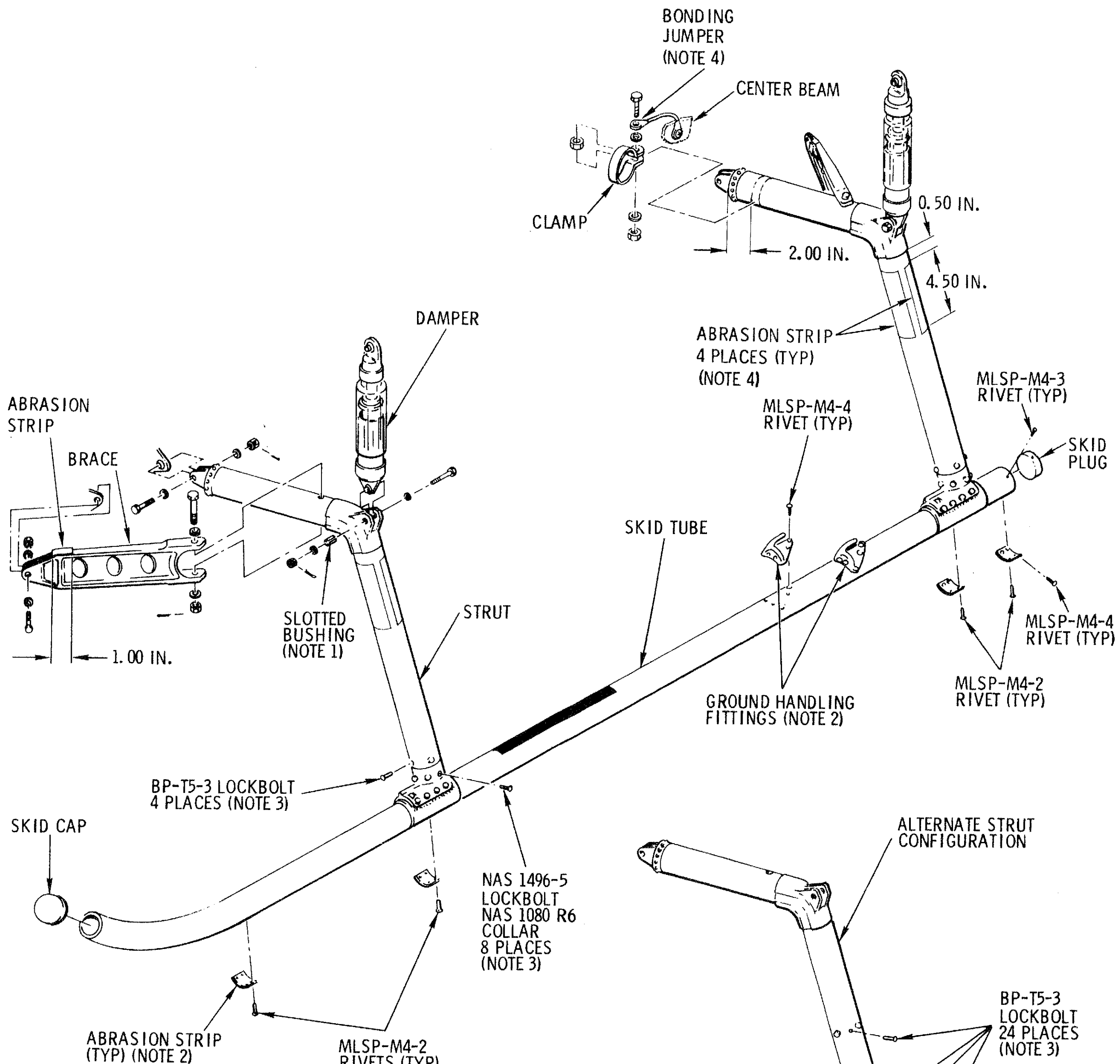
Figure 3-1. Landing Gear.



NOTES:
1. EDGE OF BUSHING MUST PROTRUDE FROM A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. SEAL PARTING EDGE WITH A FILLET OF SEALING COMPOUND (C89).
3. INSTALL RIVETS WITH WET PRIMER (C79).
4. NOT USED ON ALL AIRCRAFT.
5. BP-T6-4 LOCKBOLTS MAY BE USED AS ALTERNATE IF REQUIRED BY TOLERANCE BUILDUP.
6. BP-T8 RIVETS ARE AUTHORIZED IN LIEU OF BP-T6 RIVETS FOR USE WITH MODEL 353 RIVET GUN AND NOSE ADAPTER 99-591.
7. JO-BOLTS PP200 OR PP260 MAY BE USED AS SUBSTITUTE FOR BP-T6 OR BP-T8 LOCKBOLTS.

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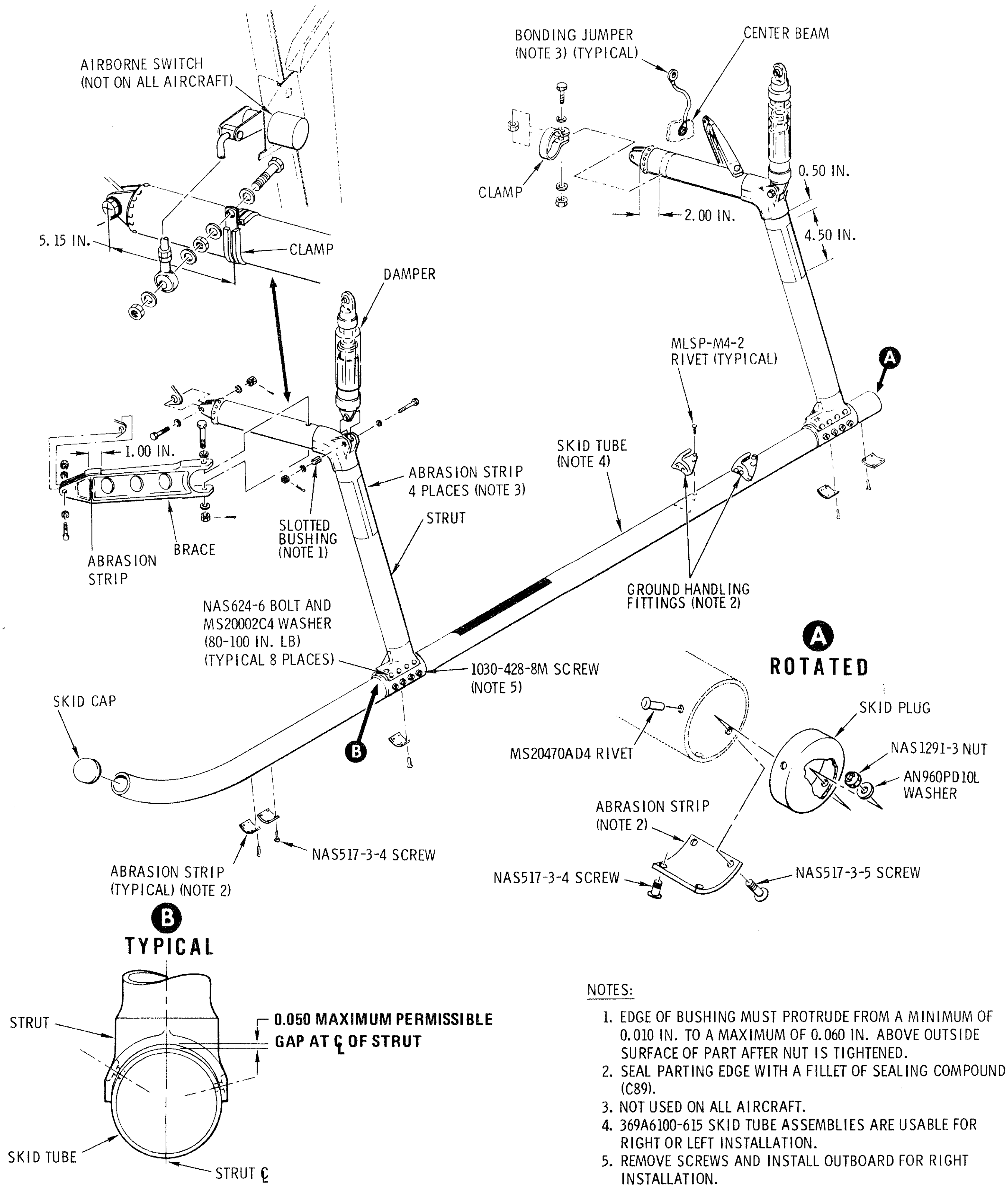
Figure 3-2. Landing Gear (Lockbolt-mounted skids) Repair and Replacement.



NOTES:

1. EDGE OF BUSHING MUST PROTRUDE FROM A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. SEAL PARTING EDGE WITH A FILLET OF SEALING COMPOUND (C89).
3. NOT USED ON ALL AIRCRAFT.
4. 369A6100-615 SKID TUBE ASSEMBLIES ARE USABLE FOR RIGHT OR LEFT INSTALLATION.
5. REMOVE SCREWS AND INSTALL OUTBOARD FOR RIGHT INSTALLATION.

12-216D

Figure 3-3. Landing Gear (Machine bolt-mounted skids) Repair and Replacement.

b. Inspect skids for dents and depressions in excess of 0.200 inch in depth, and for punctures. Perform dye-penetrant inspection of areas adjacent to punctures according to TM 55-1500-204-25/1.

c. Inspect from the aft end of the skid tube to a point 10 inches forward of the aft strut for scratches and nicks that will exceed a length of 0.25 inch and a depth of 0.15 inch when repaired.

d. Inspect from the forward end of the skid tube to 10 inches forward of the aft strut for scratches and nicks that will exceed a length of 0.25 inch and a depth of 0.020 inch when repaired.

e. Inspect ground handling fittings for cracks, breaks, loose rivets and corrosion.

f. Inspect the four skid-to-foot and foot-to-strut attachments for loosened lockbolts and relative motion between connecting parts. If the inspection reveals that the lockbolts are loose, replace with original size (if holes are not elongated) or next larger size lockbolts.

g. Remove foot support fairings (chapter 2) and open engine access doors.

h. Inspect dampers for oil leakage. Hydraulic fluid leakage from any of the landing gear dampers is not permissible. When leakage is observed, the damper assembly must be replaced.

NOTE

It is normal for a thin hydraulic oil film to remain on the damper piston as a result of wiping contact with the piston seal. Newly installed dampers may also exhibit slight oil seepage caused by oil trapped in the end cap threads during damper assembly. Neither of these should be construed as damper leakage or cause for damper replacement.

i. Check dampers for proper extension (para 3-16), and general condition.

j. Reinstall foot support fairings and close engine access doors.

3-5. Maximum Damage Limits — Landing Gear Assembly. Refer to table 3-2 for landing gear component maximum damage limits.

3-6. Repair — Landing Gear Assembly. *a.* Repair scratches and nicks by smoothing sharp edges.

b. Drill (0.25 inch maximum) out dents, and nicks to the nearest mechanically expanded rivet size when such damage exceeds depth limits. Clean out puncture holes in the same manner. For damage exceeding 0.25-inch diameter refer to paragraph 3-13.

c. Apply zinc chromate putty (C83) to edges of drilled repair holes and install the appropriate size rivet to close the hole.

d. Replace unserviceable skid plug as follows:

(1) (See fig. 3-3.) Drill out rivets attaching skid plug. On aircraft with machine bolt-attached skids, drill out attaching rivets and remove attaching screws. Remove plug by drilling a hole in its center and using a suitable puller.

(2) Apply a thin layer of zinc chromate putty (C83) to the mating surfaces of the new plug and skid tube to ensure a watertight fit.

(3) Rivet new skid plug through aft abrasion strip to tube. (See fig. 3-2.) On aircraft with machine bolt-attached skids, secure skid plug with three rivets and two screws, washers, and nuts (fig. 3-3).

e. Replace a worn abrasion strip as follows:

(1) (See fig. 3-2 and 3-3.) Remove screws or drill out rivets (as appropriate) that attach abrasion strip and remove strip.

(2) Completely remove any zinc chromate putty residue. Apply zinc chromate primer (C79) to any bare metal.

(3) Coat new rivets with zinc chromate primer (C79). Install rivets or screws (as appropriate) to attach the replacement abrasion strip to the tube while primer is still wet. A maximum centerline gap of 0.100 inch is permissible on the aft edge of the forward abrasion strip. A maximum centerline gap of 0.030 inch is permissible on the forward edge of the forward strip and the forward edge and aft edges of the other strips.

(4) After mounting, seal all of the parting edge of the abrasion strip next to the skid tube. Apply an approximate 0.06-inch bead (fillet) of sealing compound (C89) to ensure a watertight assembly.

f. Replace a damaged skid cap as follows:

(1) (See fig. 3-2 and 3-3.) Remove skid cap.

(2) Using isopropyl alcohol (C82), clean bonding area.

(3) Bond new cap to skid tube with sealing compound (C89).

g. Replace an unserviceable ground handling fitting according to *e* above.

NOTE

As installed, the longitudinal centerline of the fitting hook should be approximately 45 degrees outboard from the centerline of the landing gear strut. Both fittings must be in line.

h. Replace worn non-slip, black safety walk tape

Table 3-2. Maximum Damage Limits for Landing Gear Components.

Component	Dents (inches)	Nicks (inches)	Scratches (inches)	Cracks (inches)	Holes (inches)
Skid tube (Note 1)	0.200	0.010, from aft side of aft strut to 10 in. forward of the strut, with cleanup not exceeding 0.015; 0.015 if forward of that point to start of curved section, with cleanup not exceeding 0.020 (Note 2)	(Same as for nicks)	No cracks allowed	0.250 (Note 3)
Strut	0.060	0.010/0.020 (see fig. 3-1)	0.010/0.20 (see fig. 3-1)	No cracks allowed	No holes allowed
Brace	0.040	0.010	0.005	No cracks allowed	No holes allowed
Damper assembly	0.060	0.010 (see fig. 3-1 for caps)	0.010 (see fig. 3-1 for caps)	No cracks allowed	No holes allowed
Ground handling fittings	0.010	0.010	0.010	No cracks allowed	No holes allowed

NOTES:

1. Damage exceeding these limits may be repaired by structural repair (para 3-13).
2. Repair of minor skid tube damage aft of the rear strut and on the forward (curved) section is not required but surface finish must be restored.
3. Hole must be plugged with correct size mechanically expanded rivet.

on top surface of skid tube when the tape grit is worn away. Install new tape as follows:

(1) Carefully pull or scrape away damaged tape. Remove all adhesive residue from skid tube by wiping with a cloth wetted by naptha (C70).

(2) Carefully align new tape with mounting surface and press firmly into place. Expel air bubbles while pressing down the tape.

(3) Touch up paint as required (TB 746-93-2).

i. Replace defective or badly damaged fairing (para 3-27).

j. Replace fairing fiberglass damage, such as small tears or punctures that do not impair the telescoping action of the fairing (refer to chapter 2).

3-7. Installation — Landing Gear Assembly. *a.* Gain access to landing gear at fuselage attachment points by opening or removing compartment access doors, engine access doors, and foot support fairings.

b. With aircraft supported by jacks, position left landing gear to align the struts with their openings in the fuselage structure.

NOTE

Ensure that the fairing fillet guide pins are engaged with the fairing guide holes before positioning the struts for attachment to the structure.

c. Lift and carefully position the left landing gear; then place supports beneath the gear to hold it in place.

d. Align lower bearings of dampers with mating holes in strut (fig. 3-2) and install bolt, slotted bushing, two washers, nuts, and cotter pins to secure each damper.

e. Align struts with mating bearings in fuselage support fittings and install bolt, two washers, nut, and cotter pin, to secure each strut.

f. Align inboard end of each brace with mating bearings in fuselage support fitting and install bolt, two washers, nuts, and cotter pin to secure. If applicable, install bonding jumper with clamp and attaching hardware.

CAUTION

Do not overtighten brace to strut hardware. Maximum permissible torque of nut is finger-tight, then backed off to nearest castellation for installation of cotter pin.

g. On series 3 aircraft, install airborne switch on landing gear strut. (Refer to TM 11-1520-214-20-1.)

h. Align outboard end of each brace with mating hole in strut and install bolt, two washers, nut, and cotter pin to secure each brace.

i. Repeat above steps to install the right landing gear.

j. Remove supports, lower aircraft, and remove jacks.

k. Close all access doors and install foot support fairings.

3-8. LANDING GEAR SKID AND STRUTS.

3-9. Description — Landing Gear Skid and Struts. The landing gear skids are constructed of tubular aluminum alloy. The skids are attached to the forward and aft struts by lockbolts or standard thread machine bolts. (See fig. 3-2 and 3-3.) The struts are aluminum alloy forgings. The abrasion strips on the machine bolt-attached strut are secured by screws. Machine bolt-attached skids are interchangeable between the right and left sides.

Table 3-3. Premaintenance Requirements for Removal and Installation of Skids.

Conditions	Requirements
Special Tools	(T40)
Minimum Personnel Required	Two; Three for Jacking
Consumable Materials	(C8) (C71) (C80)

3-10. Removal — Landing Gear Skid and Struts (AVIM). *a.* Jack up the aircraft until landing gear dampers are fully extended. Place supports beneath the skid tubes at the strut locations.

b. Remove fairings (para 3-27).

c. On aircraft with lockbolt-attached skids (fig. 3-2) remove lockbolts that attach skid to forward and aft struts. (Refer to TM 55-1500-204-25/1 for lockbolt replacement.) On aircraft with machine bolt-attached skids (fig. 3-3) remove bolts and washers that attach skid tube to forward and aft struts.

3-11. Repair — Landing Gear Skid and Struts (AVIM). *a.* Repair scratches and nicks by smoothing sharp edges.

b. Repair bolt holes in strut fittings, either inboard or outboard, as follows:

(1) Enlarge the bolt hole to a size U drill diameter.

(2) Ream the drilled hole to a diameter of 0.375-0.376 inch. Chamfer both ends of hole to 0.015 inch by 45 degrees.

(3) Check that edge distance is not less than 0.375 inch.

(4) Insert a NAS75-4 press-fit bushing of correct length coated with wet zinc chromate primer (C79).

(5) Inspect reworked hole and bushing by using dye-penetrant.

c. Repair bolt holes that exceed 0.440 inch in diameter at large end of brace attached to strut as follows:

(1) Enlarge the bolt holes to a diameter of 0.5625 inch using a 9/16 inch drill.

(2) Ream the drilled hole to a diameter of 0.5631 to 0.5638 inch. Chamfer both ends of hole to 0.015 inch by 45 degrees.

(3) Check that edge distance is not less than 0.375 inch.

(4) Insert a NAS75-7 press-fit bushing of correct length coated with wet zinc chromate primer (C79).

(5) Inspect reworked holes after bushings are installed by using fluorescent penetrant.

d. Repair the bolt holes located at the small end of the brace that attaches to the landing gear attach fitting as described in *b* above.

e. Replace any strut that is cracked.

f. Replace struts that have brace attaching bolt holes exceeding 0.440 inch in diameter.

g. Replace struts that are dented or scratched beyond the allowable limits of table 3-2.

h. Repair strut scratches and dents that do not exceed limits by sanding the affected area to blend smoothly with surrounding surface area, then treat exposed surface with chemical film (C20). Scratch areas caused by rubbing of fairing guides may be further protected by placing 0.50 inch by 4.50 inch strips (maximum of four per side) of tape (C101) vertically on wear points.

3-12. Installation — Landing Gear Skid and Struts (AVIM). If machine bolt-attached skids are to be used with the lockbolt style struts, perform *e* through *m* following to enlarge the strut holes. When bolt style strut is to be used with lockbolt style skid, use 0.250-inch lockbolts with a washer to protect counterbore surface on strut foot.

a. Completely remove any zinc chromate putty residue from strut.

b. Apply a thin coating of zinc chromate primer (C79) to strut and skid tube mating surfaces.

c. Align mating parts and install lockbolts (fig. 3-2) with zinc chromate primer (C79), or secure with machine bolts and washers (fig. 3-3).

d. Seal edges of landing gear strut foot with 0.06-inch fillet of sealing compound (C89). Seal the two unused holes on the top of each strut foot (if present) with silicone rubber (C13).

e. To enlarge strut foot mounting holes for machine bolt-attached skids, place the -1 detail of drilling jig (T10) on the left strut forward foot. Use the small diameter alignment pin to locate the existing lockbolt hole.

f. Clamp the tool to strut with "C" clamps in three places.

g. Insert the drill bushing in the jig and drill a 0.250-inch hole in the strut by using a flat end (bottom) drill (size F). Install the large diameter alignment pin in the drilled hole. Use the bushing and drill the remaining holes.

h. Remove drill jig from strut. Deburr drilled holes.

i. Enlarge counterbore of mounting holes on face of to 0.687-inch diameter, flush with or 0.020 inch above original depth (0.090-inch minimum thickness remaining). Maintain the 0.030- to 0.060-inch corner radius.

j. Apply chemical film protection on exposed aluminum surfaces (chapter 1).

k. Repeat *e* through *j* above for left aft strut foot.

l. Perform *e* through *k* above using the -2 detail drilling jig for the right struts as necessary.

m. Perform *b* through *d* above for mounting skid tube assemblies.

3-13. Structural Repair — Skid Tube (AVIM). Dents, scratches, nicks, cracks and holes exceeding the normal limits described in table 3-2 may be repaired by splice plate or insertion repair. (See fig. 3-4.) Final determination of the type of repair to be made must be decided by the repair facility.

3-14. Splice Plate Repair — Skid Tube (AVIM). Splice plate repairs are suitable for dents, nicks, cracks and single penetrations requiring 30% or less metal removal from tube diameter. Plate overlap must meet minimums shown in figure 3-4.

3-15. Insertion Repair — Skid Tube (AVIM). Multiple penetrations and damage are not reparable by splice plate. Insertion repair must meet minimums shown in figure 3-4.

3-16. LANDING GEAR DAMPER ASSEMBLIES.

3-17. Description — Landing Gear Damper Assemblies. The landing gear damper assemblies are hydraulic units charged with nitrogen. The dampers must be replaced if damage or loss of pneumatic pressure (nitrogen gas) or hydraulic oil occurs. The forward damper assemblies are attached to the oleo attachment fittings located on the outboard side of the pilot's seat structure. The aft damper assemblies are attached to the lower section oleo support fittings in the engine compartment. The damper assemblies are approximately 12.21 inches long when extended and 8.96 inches long when compressed. There are two basic types of landing gear dampers which can be installed on the helicopter: (1) A 369A6300 series bladder-type damper assembly consisting of a barrel, upper and lower mounting cap, and internal rubber bladder, and a piston; or (2) a 369A6350 poppet-type damper assembly consisting of a barrel, upper and lower mounting cap, main poppet, rebound poppet and a piston. The two damper types are interchangeable individually or in sets of four.

NOTE

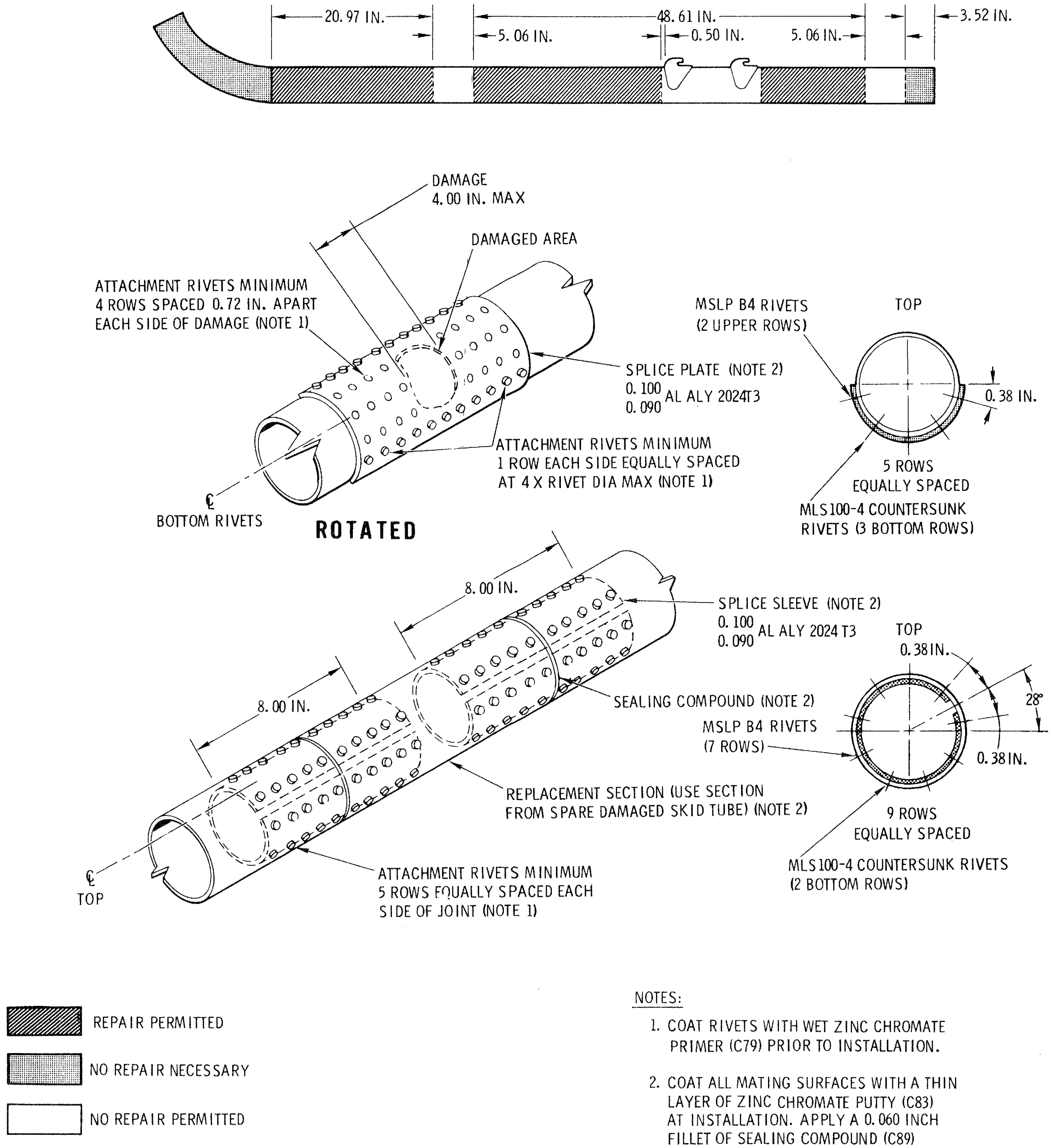
Dampers may be visually identified as follows: (1) Bladder-type (369A6300 series) damper barrels are less than 2 inches in diameter and have a transparent plastic cover; (2) poppet-type (369A6350) damper barrels are more than 2 inches in diameter and do not have a cover. (See fig. 3-5.)

3-18. Inspection — Landing Gear Damper Assemblies. Inspect forward and aft landing gear dampers as follows:

NOTE

Hoist or jack the aircraft completely off the ground prior to steps a through c below.

a. Inspect dampers for evidence of bearing looseness, defects in upper and lower caps (details B and C, fig. 3-1), loose or cracked caps security of attaching hardware, dents, scratches or cracks (table 3-2), and condition of transparent damper covers, if installed.



NOTES:

1. COAT RIVETS WITH WET ZINC CHROMATE PRIMER (C79) PRIOR TO INSTALLATION.
2. COAT ALL MATING SURFACES WITH A THIN LAYER OF ZINC CHROMATE PUTTY (C83) AT INSTALLATION. APPLY A 0.060 INCH FILLET OF SEALING COMPOUND (C89) TO SEAL EXTERIOR MATING SURFACES.

12-296C

Figure 3-4. Skid Tube Structural Repair.

b. Inspect dampers for oil leakage (para 3-3).

WARNING

Internal failure of the bladder-type damper can result in a sudden release of pressure and explosion of the bladder.

c. Inspect for damper internal failure by checking for clearance between the bottom of the landing gear strut and the fuselage and measuring the length of the extended damper. If the rear strut is touching the fuselage, or if the length of the damper, measured between the end cap flat surfaces, is 11.25 inches (12.25 inches between bolt centers) or more it must be replaced (para 3-20).

d. Perform a damper extension check, (para 3-19).

Table 3-4. Premaintenance Requirements for Damper Extension Check.

Conditions	Requirements
Support Equipment	Hoist
Minimum Personnel Required	Two

3-19. Damper Extension Check — Landing Gear Damper Assemblies.

CAUTION

A flashlight and inspection mirror should be used during the following checks to eliminate parallax errors; the damper reference points must be viewed at right angles to the damper axis.

a. Perform a quick check inspection of the dampers as directed in table 3-5. If this inspection reveals that one of the dampers is compressed to less than the minimum dimensions specified in table 3-5 and illustrated in figure 3-5, perform the comprehensive check as specified in the table. In each case, damper checks should be repeated after slight upward and downward rocking of the tailboom to ensure that the landing gear is free from any other friction or sideloads.

NOTE

If bladder-type dampers do not have painted color bands as shown in figure 3-5, equivalent red and yellow bands or reference markings shall be applied to the transparent damper covers as shown in figure 3-6. The green band located

Table 3-5. Damper Extension Check.

Damper Location	Bladder-Type (fig. 3-5)	Poppet-Type (fig. 3-5)
QUICK CHECK: (Notes 1, 2)		
Aft Damper	0.25 in. at location A	None required
Forward Damper	0.50 in. at location A	None required
COMPREHENSIVE CHECK: (Notes 1, 3)		
Aft Damper	At location B, lower edge of red band above upper edge of green band	Less than 1.54 in. at location D
Forward Damper	At location C, lower edge of yellow band above upper edge of green band	Less than 1.96 in. at location D

NOTES:

1. Condition checks are valid only when ambient temperature and aircraft systems are at 25°F (-4°C) or higher.
2. Aircraft condition for quick check:
   a. Aircraft empty, fuel tanks full.
   b. Skid runners resting on level surface.
3. Aircraft condition for comprehensive check:
   a. Aircraft resting on greased steel plates.
   b. Aircraft empty, fuel tanks full (gross weight 1630 ±100 lb).
   c. Approximate center of gravity 104.7 inches.

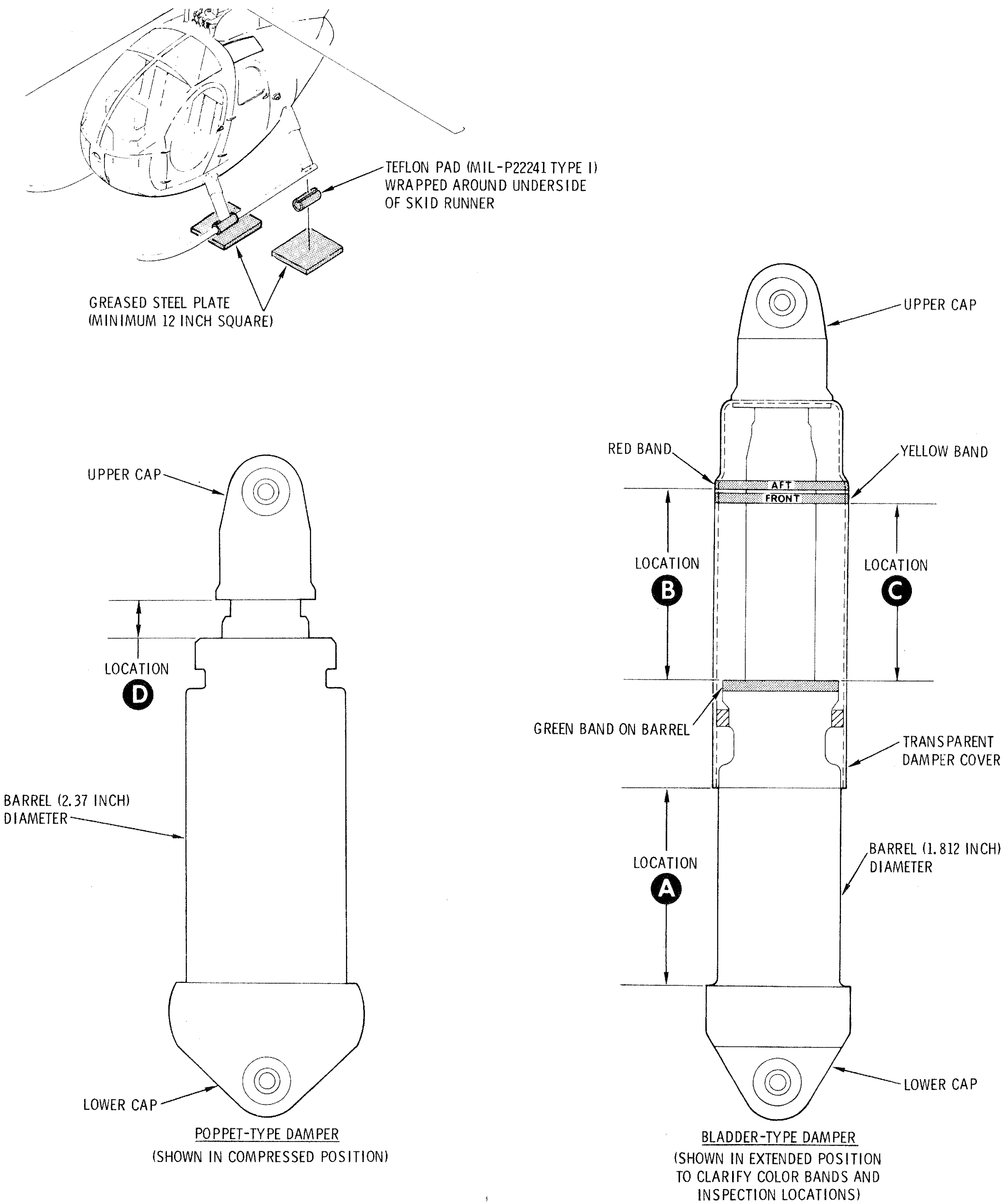


Figure 3-5. Landing Gear Damper Inspection Methods.

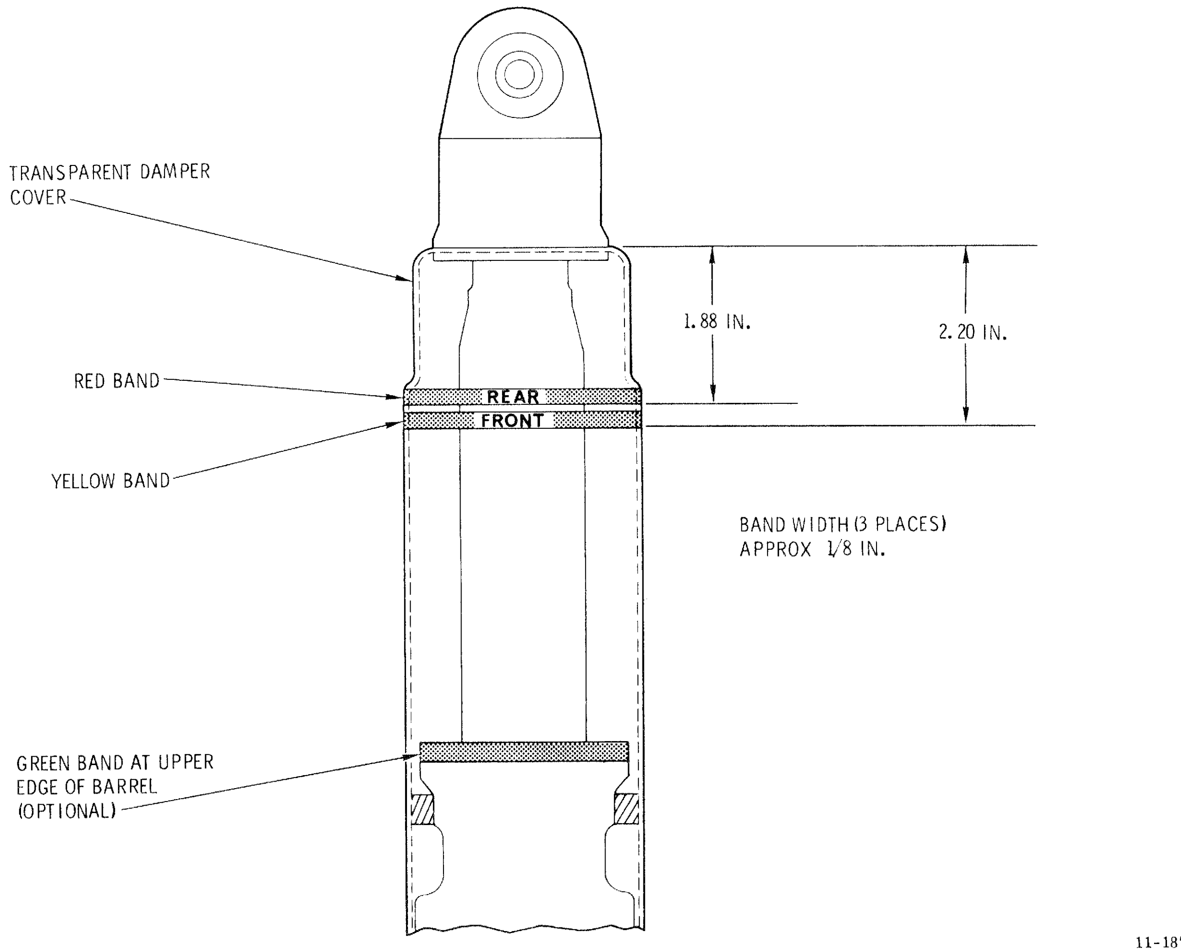


Figure 3-6. Landing Gear Damper (Bladder-type) Reference Markings.

around the upper edge of the damper barrel is an optional convenience. The upper corner of the barrel is equivalent to the upper edge of the green band.

b. If any bladder-type damper is compressed to less than the limits specified in table 3-5, it is an indication that the damper has lost part of its nitrogen charge or hydraulic oil and the damper must be replaced.

NOTE

The checks outlined in table 3-5 are only valid for the conditions stated. Failure to place the aircraft on greased plates or deviation from the specified aircraft weight, temperature and center of gravity conditions will produce inaccurate and misleading results.

3-20. Damper Removal and Installation — Landing Gear Damper Assemblies.

Table 3-6. Premaintenance Requirements for Damper Removal and Installation.

Conditions	Requirements
Support Equipment	Hoist or Jacks
Minimum Personnel Required	Two; Three for Jacking

3-21. Preparation for Damper Removal — Landing Gear Damper Assemblies.

WARNING

Check that all electrical power is OFF. This precaution will prevent personal injury or aircraft damage which could result from body or tool contact with electrical terminals.

CAUTION

Make certain the aircraft is jacked evenly in a level area so that jacks cannot slip and damage the aircraft structure.

a. Jack the aircraft until the landing gear skids just clear the ground.

b. Remove foot support fairings (chapter 2) for forward damper access or open engine compartment doors for aft dampers.

WARNING

Use extreme care during bladder-type damper removal if the damper is extended beyond 11.25 inches between the flat surfaces of both end caps (12.25 inches, bolt center to bolt center). This condition indicates an internal failure of the damper that will result in a sudden release of internal pressure and explosion of the bladder. Before the damper is released from the fuselage fittings the internal pressure must be relieved, *c* below.

c. Dampers which show evidence of, or are suspected of, internal failure must have the pressure relieved (1) through (3) below before removal.

(1) Remove any electronic equipment in the immediate area of the damaged damper. Cover all electrical connections with plastic material.

(2) Wrap the damaged damper with 1-inch-thick polyfoam (or material of equivalent absorption property) to minimize spraying of hydraulic oil.

WARNING

Protective glasses and heavy gloves must be worn when drilling the pressure relief hole in the damaged damper. The spray of hydraulic oil under pressure can result in injury to personnel.

(3) Drill a 0.059-inch hole (size 53 drill) in the body of the damper. Keep the drill bit in the hole until the hydraulic oil stops flowing.

3-22. Removal — Forward Damper. *a.* Prepare for forward damper removal according to paragraph 3-21. Observe applicable warnings and caution.

b. Remove hardware that attaches forward damper assemblies to fuselage fittings and landing gear struts.

3-23. Removal — Aft Damper. *a.* Prepare for aft damper removal according to paragraph 3-21. Observe applicable warnings and caution.

b. Remove hardware that attaches aft damper to fuselage fittings and landing gear struts.

3-24. Replacement — Bladder-type Damper Plastic Cover.

NOTE

When replacing a damper assembly plastic cover, only the upper end of the damper must be detached.

a. Remove plastic tape used to compress cover tangs.

b. Carefully but firmly press cover toward top end to disengage tangs from bearing cap groove.

c. Install a replacement by reversing the procedure.

d. Secure the cover tangs in place by wrapping several turns of tape (C100) around tanged end of cover.

3-25. Installation — Aft Damper.

NOTE

Before installing a poppet-type damper at the aft right position on series 1 aircraft, check that the aircraft has been modified with the improved engine-out warning and automatic restart system. If a series 1 aircraft has not been so modified, the aft right side poppet-type damper will interfere with the start relay.

a. Position lower end of replacement damper assembly in attachment lugs of landing gear strut. Install bolt, flat washers and nut. Tighten nut and install new cotter pin.

b. Position upper end of damper assembly between oleo attachment fitting lugs. Install bolt with flat washer from front of strut through damper assembly; install washer and nut. Tighten nut and install new cotter pin. (See detail A, fig. 3-1.)

3-26. Installation — Forward Damper. *a.* Position lower end of replacement damper assembly in attachment lugs of landing gear strut. Install bolt, flat washers and nut. Tighten nut and install new cotter pin.

b. Position upper end of damper assembly between oleo attachment fitting lugs. Install bolt with flat washer from front of strut through damper assembly; install washer and nut. Tighten nut and install new cotter pin. (See detail C, fig. 3-1.)

3-27. LANDING GEAR FAIRING ASSEMBLY.

3-28. Description — Landing Gear Fairing Assembly. Each landing gear fairing assembly (fig. 3-7) is a two-piece, spring-loaded, telescoping fiberglass airfoil that is contour-fitted to the fuselage and skid tube. The fillet has a teflon strip bonded around the contact surface of its fuselage-contour-matching flange. The rubbing plate is color impregnated fiberglass, bonded and riveted to the fuselage adjacent to the fairing. The rubbing plate does not require painting. The fairing telescopes inside the fillet to allow movement of the strut as the landing gear dampers compress or extend. The upper guide is an assembly of two unattached springs fitted over two 1/4-inch pins that engage matching holes in the lower guide. The sliding surfaces (ID) of both parts of the upper guide are lined with a teflon pad to reduce friction during movement between the guide halves and strut.

3-29. Removal — Landing Gear Fairing Assembly. (See fig. 3-7.) The following removal instructions are typical for all four fairing assemblies. If the fairing is to be replaced or removed for repair, the fillet as well as the fairing must be removed.

a. Remove the four rivets from the trailing edge of the fillet.

b. Drill out one of the six rivets that secure the fillet to the upper guide. Replace the rivet with a sheet metal hole fastener. Repeat this step on the other five rivets.

c. Open electronic compartment or engine compartment access doors for access to the strut cutout in the fuselage skin. Have an assistant push down on the upper guide with a suitable tool (wood dowel or equivalent), to relieve the spring tension.

d. With spring tension off the guide, take out the six fasteners. Slowly relax the dowel pressure on the guide until the springs reach maximum travel.

e. Carefully spread fillet at trailing edge and remove in a forward direction.

f. Remove eight or nine rivets (as applicable) that secure trailing edge of fairing, and eight rivets that secure fairing to the two strut brackets.

g. Use a putty knife or similar thin-bladed tool to carefully pry apart the bonding and separate the upper 5 inches of the trailing edge; then separate the fairing from the flange of the lower guide.

h. Carefully spread trailing edge of fairing and remove in a forward direction.

3-30. Inspection — Landing Gear Fairing Assembly. (See fig. 3-7.) *a.* Inspect fairing support brackets for security of attachment to strut, cracks and deformation.

b. Inspect fairing-to-skid seal for security of bond.

c. Inspect the fuselage rubbing plate for security of bond and excessive abrasion.

3-31. Repair — Landing Gear Fairing Assembly. (See fig. 3-7.) *a.* Replace a damaged fairing bracket or lower guide. Remove the rivets, locate the new part, and install mechanically expanded rivets per TM 55-1500-204-25/1. Use next larger size rivets if mating holes in strut are enlarged.

b. Replace a damaged guide/pin assembly or guide half if guide is cracked, pins are bent or badly worn,, or teflon pad is loose or badly worn.

c. Replace springs that are badly worn, rusty, or **REPLACE IF FREE LENGTH IS LESS THAN 8.72 INCHES.**

d. Replace fillet if the teflon strip is loose or badly worn.

e. Replace a damaged strut bracket. Remove the rivets, locate the new part, and install mechanically expanded rivets per TM 55-1500-204-25/1. Use next larger size rivets if mating holes in strut are enlarged.

f. Repair an unbonded section of the fairing-to-skid seal (chapter 2).

g. Refer to chapter 2 for typical fiberglass repairs.

3-32. Installation — Landing Gear Fairing Assembly. (See fig. 3-7.) The following installation instructions are typical for all four fairing assemblies. Where riveting is required, use rivets of the type shown in figure 3-7.

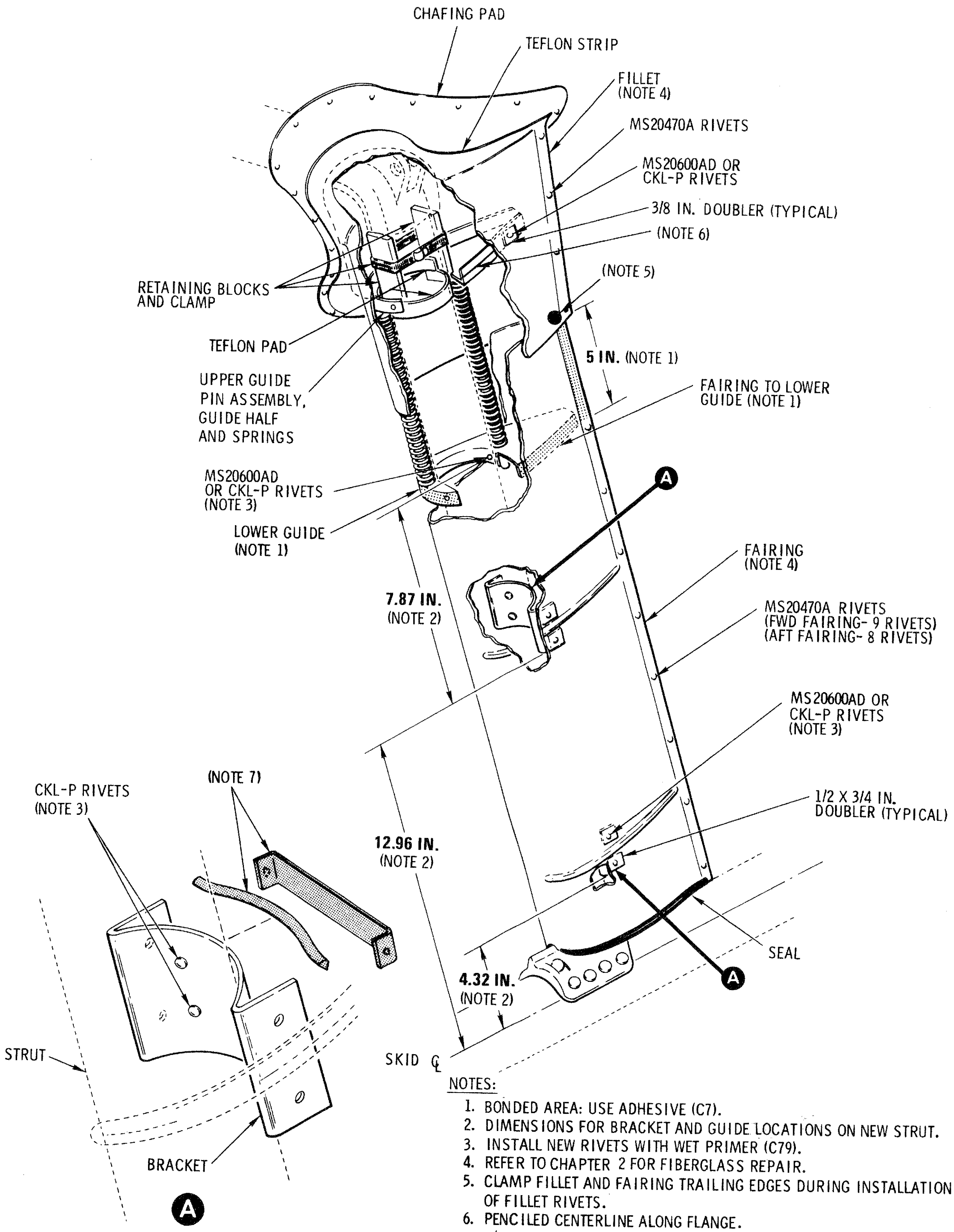
a. Assemble new upper guide, new pin assembly, new guide half, new springs and new lower guide on the landing gear strut. Install two rivets to secure lower guide to strut.

b. Push guide pins into the matching holes of the lower guide. Wedge a temporary holding device between the upper guide and the strut to keep the pins engaged.

NOTE

Use two small wood blocks or any similar suitable means to keep the springs in compression. The device must be small enough to be removed through the strut cutout in the skin after the fillet is assembled.

c. Install blind hole transfer punches (four places)



NOTES:

1. BONDED AREA: USE ADHESIVE (C7).
2. DIMENSIONS FOR BRACKET AND GUIDE LOCATIONS ON NEW STRUT.
3. INSTALL NEW RIVETS WITH WET PRIMER (C79).
4. REFER TO CHAPTER 2 FOR FIBERGLASS REPAIR.
5. CLAMP FILLET AND FAIRING TRAILING EDGES DURING INSTALLATION OF FILLET RIVETS.
6. PENCILED CENTERLINE ALONG FLANGE.
7. 3/16 INCH WELDING ROD OR OTHER SUITABLE DEVICE (4 PLACES).

12-244B

Figure 3-7. Landing Gear Fairing — Repair and Replacement.

or fabricate suitable tools as shown in figure 3-7. Install between opposing holes in the two strut brackets.

d. Carefully spread trailing edge of new fairing and position the fairing on the two strut brackets and the lower guide. **THERE SHOULD BE 0.020 - 0.080 INCH COMPRESSION OF THE FAIRING SEAL AGAINST THE SKID TUBE WHEN THE FAIRING IS CORRECTLY POSITIONED.** Have an assistant hold the fairing in this position.

e. Back up the fairing with a fiber block at the transfer punch location. Using a plastic hammer, strike the opposite side of the fairing hard enough to transfer the hole centers.

f. Remove fairing and check that all the transfer marks appear within the outline of the small laminated doublers. Drill out the located holes.

g. Carefully bond the fairing to the lower guide (chapter 2). (See fig. 3-7.)

h. Install eight rivets to secure fairing to the two strut brackets, and eight or nine rivets (as applicable) to secure trailing edge together.

i. Using a pencil, draw a continuous line along the horizontal center of the upper guide flange.

j. Drill a No. 40 hole in the center of each 0.38-inch-square doubler in the fillet. (There are six doublers.)

k. Carefully spread trailing edge of fillet and position about upper guide. Clamp the lower end of the fillet trailing edge to the upper end of the fairing trailing edge. Clamping will prevent fillet movement when the fillet rivets are installed.

l. Have an assistant remove the temporary holding device and push down on the upper guide with a suitable tool (wood dowel, etc.). Have the guide pushed down to a point where the pencil line drawn on the guide flange is visible through one of the No. 40 holes in the fillet. Match-drill the guide and secure the fillet and guide with a hole fastener. Repeat the procedure for all six rivet locations. Remove the pushing tool from the strut opening.

m. Remove one fastener at a time, enlarge the hole to rivet diameter and install a mechanically expanded rivet.

n. Install four rivets to secure the fillet trailing edge together.

o. Check for smooth telescoping action of fairing into fillet by manually sliding the fillet up and down several times.

p. Close access doors.

CHAPTER 4

POWER PLANT

SECTION I GENERAL

4-1. PURPOSE.

4-2. General. This chapter provides maintenance instructions for powerplant related systems. This includes engine-to-airframe related troubleshooting, engine mounts, air induction, exhaust, oil, cooling, accessories and troubleshooting of the power controls.

4-3. SCOPE.

4-4. General. This chapter contains data pertaining to engine related systems according to the Maintenance Allocation Chart. It does not contain maintenance instructions for the engine itself. Maintenance instructions for the T63-A-5A and T63-A-700 engines are contained in TM 55-2840-231-24.

SECTION II POWER PLANT

4-5. ENGINE ASSEMBLY.

4-6. Description — Engine Assembly. This section contains applicable maintenance information for the T63-A-5A and alternate T63-A-700 engine assemblies. The two engines are dynamically, functionally, and operationally identical. Some T63-A-700 engines modified by an MWO have a fuel control heater which is not installed on T63-A-5A engines. The few hardware modifications which must be performed before installing the T63-A-700 engine are contained in paragraph 4-15. Maintenance information for compressor armor and engine mounts is also provided. Use 0.20-inch stainless steel lockwire (C56) unless otherwise specified. Double strand lockwire all drilled bolts, plugs, and screws, except those locked with self-locking nuts or lockwashers. Lockwire bolts in pairs where possible. When reassembling, be sure to safety wherever lockwire was removed. Do not use zinc lockwire. Do not reuse lockwire, cotter pins, ring seals, lip seals, composition gaskets, and split or tab washers.

4-7. Maintenance — Engine Assembly. Refer to TM 55-2840-231-24 for engine maintenance, troubleshooting and permissible oil leakage information. Refer to table 4-1 for additional engine-to-airframe related troubleshooting.

4-8. Troubleshooting — Engine Assembly. Engine malfunctions may be obvious, or they may be of a nature which is not obvious but can cause considerable damage to the engine if not corrected. It is essential that maintenance personnel have thorough knowledge of turbine outlet temperatures, fuel pressures, oil pressures and other important details of normal engine performance in order to recognize engine malfunctions if they occur. (Refer to TM 55-1520-214-10.) Malfunction correction may require simple repair of a faulty installation, replacement of an assembly or part, or removal of the engine for inspection and repair. Refer to appendix B for corrective action authority.

4-9. Operational Check — Engine Assembly. In addition to engine runup performed to check systems operation, test run the engine after the compressor, turbine, combustion section, fuel control, governor, fuel pump, thermocouple or interstage bleed valve has been removed or replaced. Refer to TM 55-1520-214-10 for operating instructions and limits. Make note of all incidents of the run such as leaks, abnormal vibration or noises, and/or any irregular functioning of engine equipment. Also note that the following items are within operating limits:

Turbine outlet temperature

Output shaft torque

Oil pressure

Gas producer N1 speed

Power turbine N2 speed

4-10. Removal and Installation — Engine Assembly. Removal and installation of either the T63-A-5A or T63-A-700 engine assemblies in the following paragraphs are identical. Minor hardware changes and modifications are noted as applicable.

Table 4-1. Troubleshooting of the Engine.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Engine fails to start.**

STEP 1. Check for faulty ignition switch or wiring (para 9-80).

If defective, replace or repair switch or wiring (para 9-80).

2. **Engine fails to light off when hot.**

STEP 1. Check for correct alignment of engine gear case cooling air duct causing the engine fuel pump to overheat and fuel vaporization.

If the gear case cooling air duct is misaligned, reposition air duct as shown in figure 4-6.

3. **Engine lights off but will not accelerate to idle speed in 45 seconds (ammeter shows a positive charge).**

STEP 1. Check for generator on during start sequence.

If generator was on during start sequence, restart with generator off.

STEP 2. Check for defective voltage regulator (para 9-10).

If the voltage regulator is defective, replace voltage regulator (para 9-10).

4. **Acceleration temperature too high during start sequence.**

STEP 1. Check for obstructed air inlet (para 4-52).

If air inlet is obstructed, clear obstruction from air inlet and clean air inlet system.

STEP 2. Check for clogged air inlet filter.

If the air inlet filter is clogged, dirty or damaged, clear and clean or replace air inlet filter (para 4-77 or 4-105).

5. **Low power with high turbine air temperature (TOT).**

STEP 1. Check for defective BYPASS AIR caution light (para 4-138).

If BYPASS AIR caution light is defective, replace light.

STEP 2. Check for clogged, dirty, or damaged air filter.

If air filter is clogged, dirty, or damaged, clear and clean or replace air filter (para 4-77 or 4-105).

6. **Low measured TOT at normal or high power setting.**

STEP 1. Check for defective or out of calibration TOT indicating system (chapter 8).

If the TOT indicating system checks within tolerance (chapter 8) and indicator still reads low; refer to TM 55-2840-231-24 for further troubleshooting and corrective action.

7. **Engine instability above idle speed.**

STEP 1. Check for engine-to-transmission misalignment (para 4-47).

If a misalignment of the engine to transmission exists, align engine-to-transmission (para 4-47).

Table 4-2. Premaintenance Requirements for Removal of the Engine.

Conditions	Requirements
Special Tools	(T11) (T38)
Minimum Personnel Required	Two (MOS 68B & 67V)

4-11. Disconnection for Removal — Engine Assembly. See figure 4-1 for the location of points to be disconnected before engine removal from the aircraft.

a. Set power selector switch at OFF and disconnect external power.

b. Remove main transmission drive shaft (chapter 6).

c. Remove fuel controls armor, if installed (para 4-35).

d. Open engine access doors and remove engine tailpipes (para 4-156).

e. Remove lockwire and engine electrical harness plug from the receptacle mounted on the right side oleo structural fitting.

CAUTION

Cap all disconnected lines and fittings. Do not use tape to cover fuel and oil openings. Tape adhesive is soluble in fuel or oil and can cause contamination.

NOTE

As an aid for reassembly, attach identifying tags to all electrical wires and fuel and oil hoses before disconnecting.

f. Remove chromel and alumel nuts and thermocouple leads from terminal studs.

g. Disconnect leads from terminals E, C, and B of starter-generator. Disconnect bond jumper at engine compartment firewall (if installed).

h. Remove loop clamp on right engine mount to release engine harness.

i. Remove combustion chamber drain line from burner drain valve.

j. Disconnect fuel inlet line at pump.

NOTE

The fuel inlet line is disconnected at the firewall on aircraft equipped with fuel controls armor.

k. Disconnect engine fuel pump seal drain line.

l. Disconnect cabin air outlet tube (or hose) at firewall fitting.

m. Loosen compressor cooling air duct clamp at firewall coupling, disconnect support bracket at hoisting eye and remove duct from engine compartment.

n. Disconnect oil line from torquemeter fitting.

o. Disconnect N2 control rod by removing cotter pin, nut, bolt, and two washers. Tie or tape control rod out of the way against the firewall.

p. Disconnect anti-icing control valve lever by removing cotter pin, loosening nut on adapter and detaching control cable.

q. Disconnect N1 control rod by removing cotter pin, nut, two washers, and bolt. Tape or tie control rod out of the way against the firewall.

NOTE

Remove N1 control rod completely on aircraft equipped with fuel controls armor.

r. Disconnect accessory drive overboard vent.

s. Disconnect oil line from engine oil pressure sender.

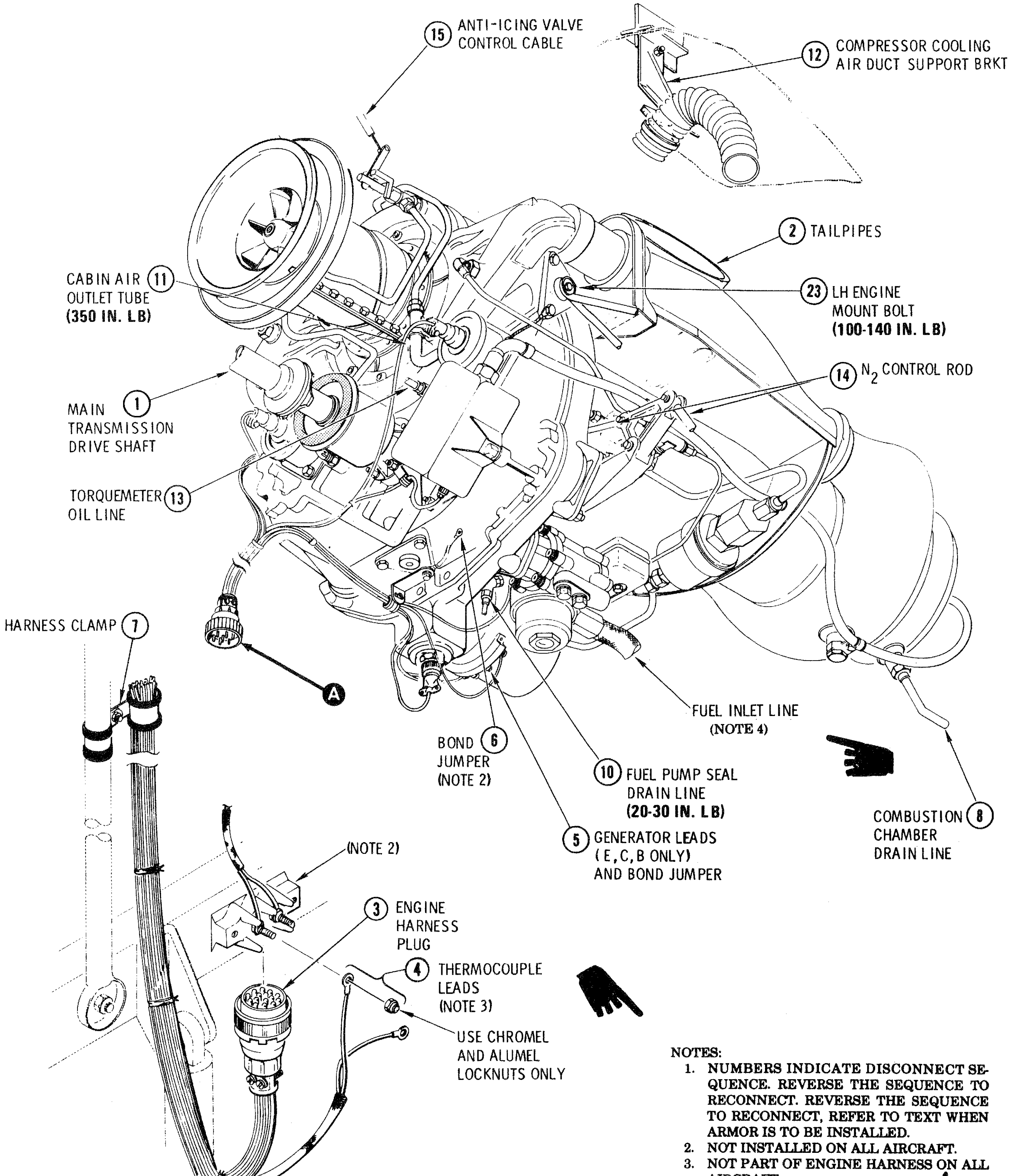
CAUTION

Remove the oil inlet and outlet lines, *t* and *u* below, only at the engine. Removal at the oil cooler end will damage the cooler frame if the frame oil port boss is not supported with a wrench. For the same reason, do not apply extreme bending loads on the oil lines when they are disconnected from the engine.

t. Disconnect engine oil inlet line.

u. Disconnect engine oil outlet line.

v. Disconnect engine oil vent line.



NOTES:

1. NUMBERS INDICATE DISCONNECT SEQUENCE. REVERSE THE SEQUENCE TO RECONNECT. REVERSE THE SEQUENCE TO RECONNECT, REFER TO TEXT WHEN ARMOR IS TO BE INSTALLED.
2. NOT INSTALLED ON ALL AIRCRAFT.
3. NOT PART OF ENGINE HARNESS ON ALL AIRCRAFT.
4. REFER TO PARAGRAPH 10-55 (Fig. 10-8) AND PARAGRAPH 10-144 (Fig. 10-16) OF THIS MANUAL FOR PROPER FUEL LINE FITTING TORQUE.

Figure 4-1. Engine Disconnect/Reconnect Points. (sheet 1 of 2)

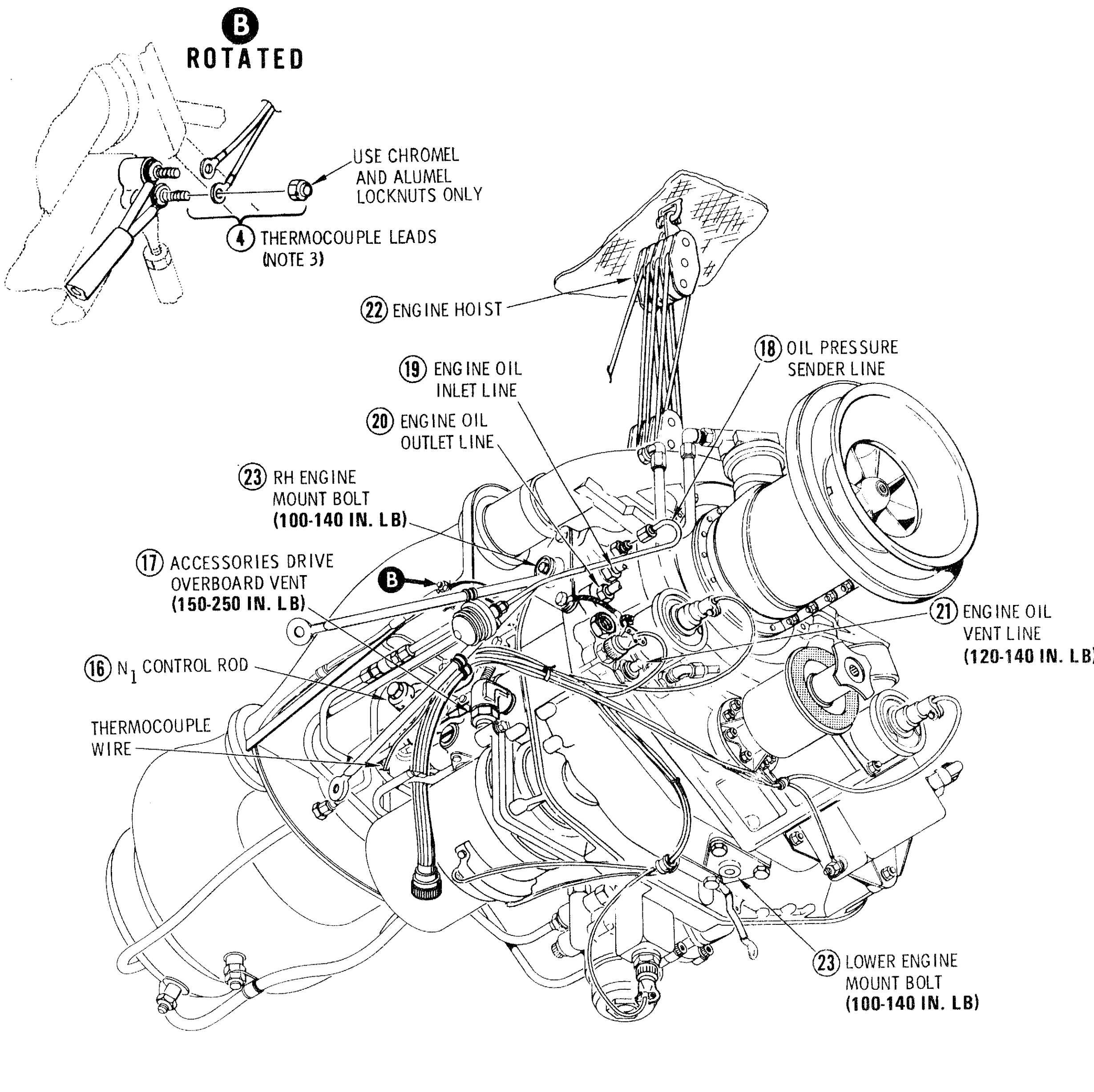


Figure 4-1. Engine Disconnect/Reconnect Points. (sheet 2 of 2)

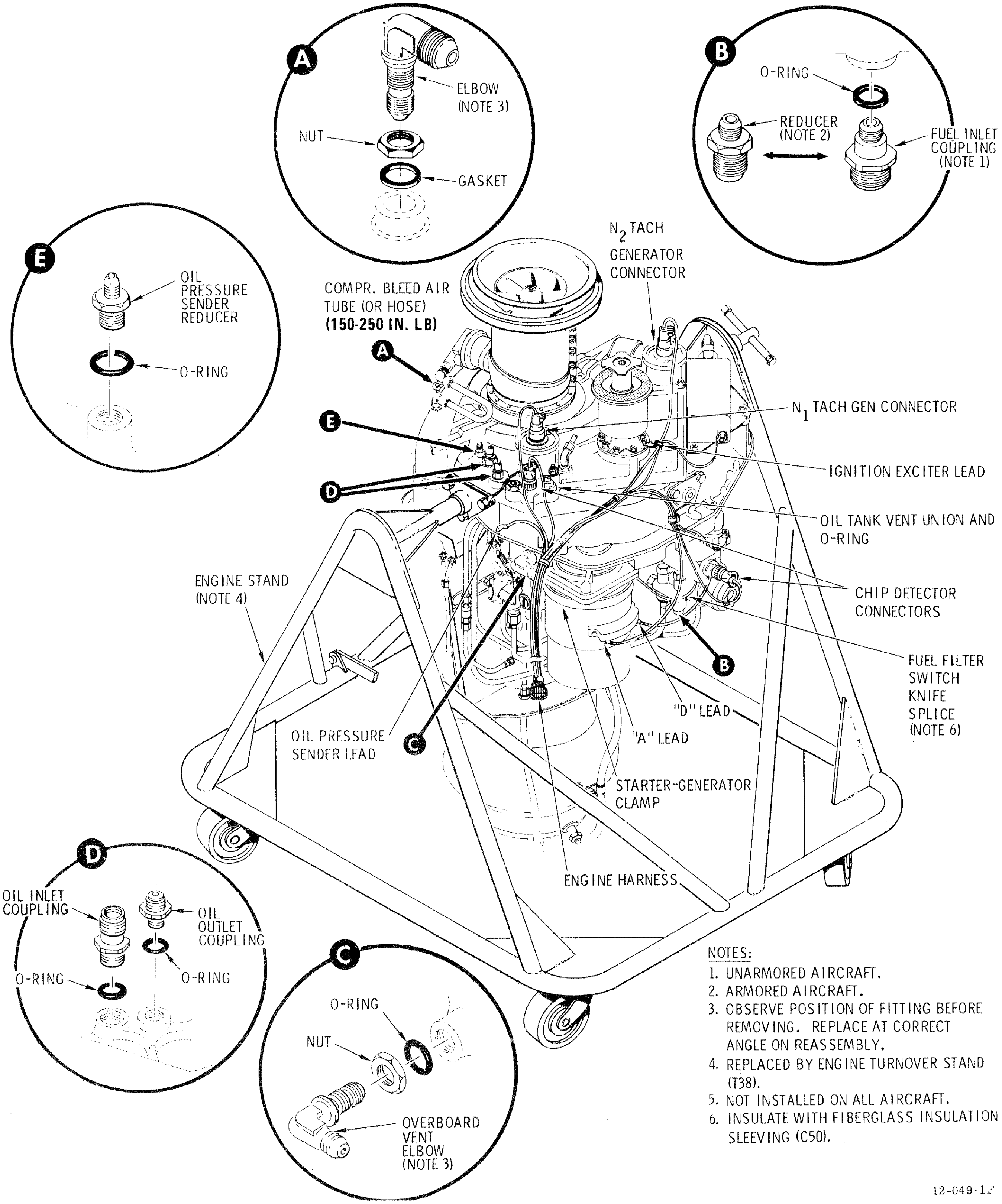


Figure 4-2. Engine Accessory Replacement. (sheet 1 of 2)

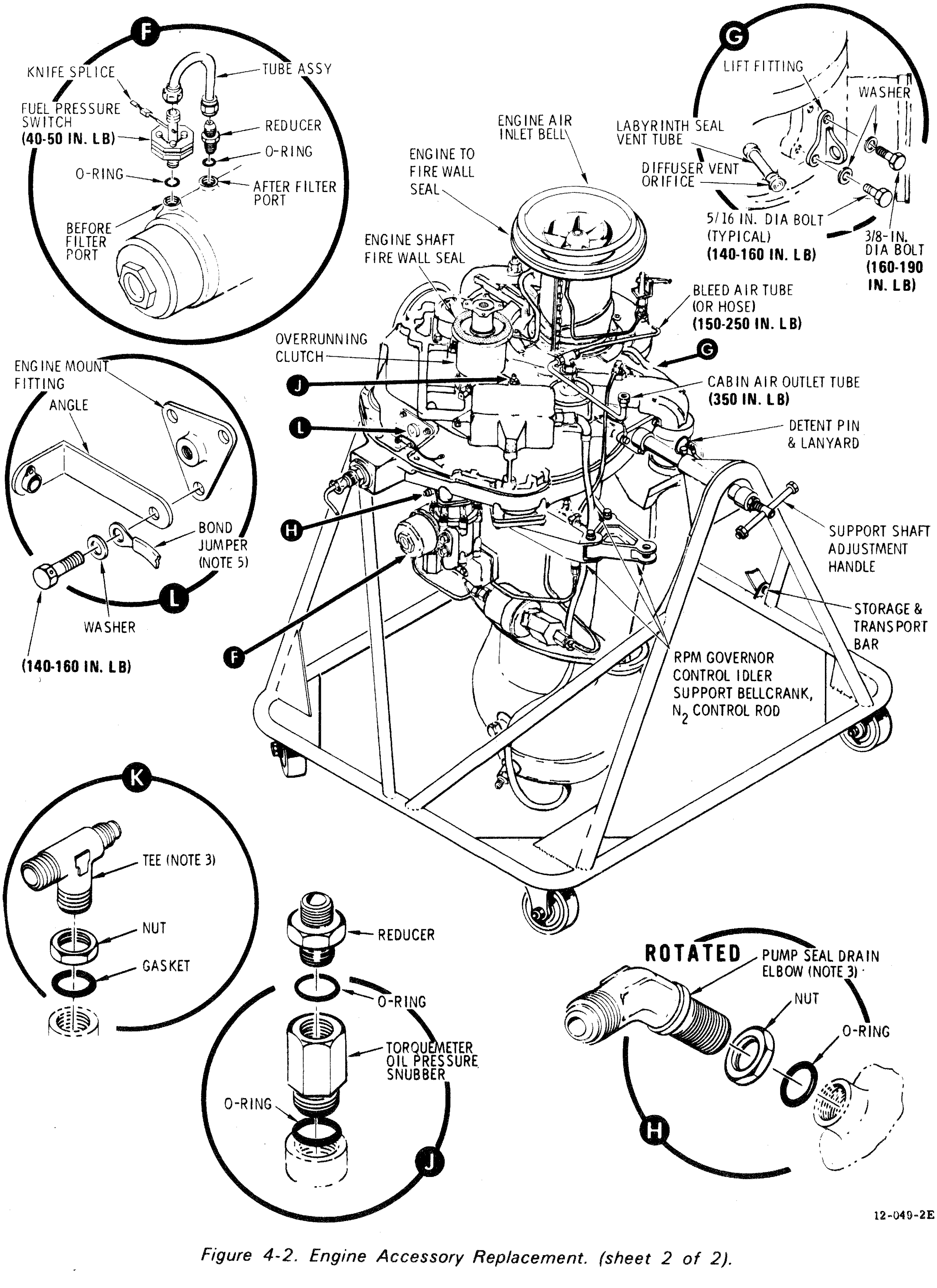


Figure 4-2. Engine Accessory Replacement. (sheet 2 of 2).

4-12. Disconnected Engine Removal — Engine Assembly. (See figure 4-2.) *a.* Install engine hoist (T11) between hoisting eye fitting in structure above engine and the hoist fitting provided on engine.

b. Have an assistant apply tension on hoist line.

CAUTION

Provide sufficient support during side bolt removal to eliminate bolt loading in mount bushing. Damage to bushings can be caused by loading bolt threads.

c. Remove bolt and washer from bottom engine mount. Maintain downward pressure at rear of engine during removal of engine mount bolts.

d. Remove two bolts and washers from the left and right engine mounts.

CAUTION

Do not use fuel or oil lines to handle engine during removal. Take care not to contact adjoining structure or components while lowering engine. Use particular care to prevent damage to the No. 1 bearing oil return line on the underside of the compressor.

e. Slowly lower engine from aircraft while supporting engine from both sides.

f. Remove bolts, bond jumpers (if installed) and three mount fittings from engine (detail L, fig. 4-2, sh 2).

g. Remove hoist from engine lifting eye and install engine in turnover stand (T38). (Refer to TM 55-2840-231-24.)

Table 4-3. Premaintenance Requirements for Removal of Accessories.

Conditions	Requirements
Special Tools	(T39)
Minimum Personnel Required	One (MOS 68B)

4-13. Removal of Accessories — Engine Assembly. (See fig. 4-2.) *a.* Disconnect A and D leads from starter-generator and remove starter-generator (chapter 9).

b. Disconnect electrical connectors from both chip detectors.

c. Disconnect both tachometer generator connectors, ignition exciter lead, and fuel pressure switch splice.

d. Disconnect thermocouple leads from engine, and remove entire engine wiring harness.

e. Remove fuel inlet coupling (or reducer) and preformed packing (O-ring) (detail B, sh 1).

f. Remove oil inlet and outlet couplings with O-rings (detail D, sh 1).

g. Remove elbow, nut, and O-ring (detail C, sh 1).

h. Loosen both ends of compressor bleed air tube (or hose) and detach from engine.

i. Remove elbow, nut, and gasket from the compressor scroll (detail A, sh 1).

j. Remove oil pressure sender reducer and O-ring (detail E, sh 1).

k. Remove both tachometer generators (chapter 8).

l. Remove tube assembly, reducer, fuel pressure switch, and two O-rings from fuel pump assembly (detail F, sh 2).

m. Remove engine air inlet bell, and engine-to-firewall seal (para 4-69).

n. Remove three bolts, washers, and engine hoist fitting (detail G, sh 2).

o. Remove elbow, nut, and packing from the fuel pump seal drain port (detail H, sh 2).

p. Remove rpm governor control idler support, bellcrank, and short N2 control rod as an assembly. Install vacuum/hydraulic pump pad cover.

q. Remove oil pressure snubber, reducer, and two O-rings (detail J, sh 2).

r. Remove engine shaft firewall seal (para 4-71).

s. Remove the overrunning clutch (chapter 6). Install pad cover.

t. Remove cabin air outlet tube.

u. Remove tee, nut, and gasket from compressor scroll (detail K, sh 2).

NOTE

Step v below applies only to aircraft with the automatic restart system.

v. Remove and retain ignition igniter (Allison 6843984, Champion FHE 161-9, or AC 5611588) and ignition exciter (Allison 6870885, GLA 43754, or

Bendix 6870891). These units will be reused on the replacement engine.

w. Remove engine lift (T39), if installed.

x. Remove the power turbine governor and gas producer levers. These, with the attaching nuts will be used on the replacement engine.

Table 4-4. Premaintenance Requirements for Installation of Accessories.

Conditions	Requirements
Special Tools	(T38) (T39)
Minimum Personnel Required	One (MOS 68B)
Consumable Material	(C57)

4-14. Installation of Accessories — Engine Assembly. (See fig. 4-2.) *a.* Install engine in engine stand (T38). (Refer to TM 55-2840-231-24.)

b. Install gasket, nut, and tee in compressor scroll (detail k, sh 2); secure tee with nut.

c. Hand-tighten cabin air outlet tube to scroll-mounted tee.

d. Install overrunning clutch (chapter 6).

e. Install engine shaft firewall seal (para 4-76).

f. Install gasket, nut and elbow in compressor scroll (detail A, sh 1).

g. Position compressor bleed air tube between scroll-mounted tee and elbow. **TORQUE THE TUBE NUTS TO 150-250 INCH-POUNDS.**

h. Install two O-rings, reducer, and torquemeter oil pressure snubber (detail J, sh 2).

i. Install rpm governor control idler support, bellcrank, and short N2 control rod as an assembly (para 4-215).

NOTE

Only high temperature all-metal nuts (not the fiber insert type) are to be used for anchoring governor idler support bracket to engine accessory pad.

j. Install O-ring, nut, and elbow in fuel pump seal drain port (detail H, sh 2).

k. Remove lift (T39), if installed. Install engine hoist fitting by using three washers and bolts (detail G, sh 2). **TORQUE THE 3/8-INCH-DIAMETER BOLT TO 160-190 INCH-POUNDS. TORQUE THE TWO** 5/16-INCH-DIAMETER BOLTS TO 140-160 INCH-POUNDS.

l. Install engine-to-firewall seal and engine air inlet bell (para 4-74).

m. Install packing and fuel pressure switch in the before-filter port of the fuel pump (detail F, sh 2). **TORQUE SWITCH TO 40-50 INCH-POUNDS.** Install O-ring and reducer in the after-filter port. Install tube assembly.

n. Install tachometer generators (chapter 8)

o. Install O-ring and oil pressure sender reducer detail E, sh 1).

p. Install O-ring, nut and elbow for accessories drive overboard vent (detail C, sh 1).

q. Install oil inlet and outlet couplings and O-rings (detail D, sh 1).

r. Install fuel inlet coupling (or reducer) and O-ring (detail B, sh 1).

s. Install starter-generator (chapter 9).

NOTE

Step t below applies only to aircraft with the automatic restart system.

t. Check that ignition igniter (Allison 6843984, Champion FHE 161-9, or AC 5611588) and ignition exciter (Allison 6870885, GLA 43754, or Bendix 6870891) are installed.

u. Install engine harness. Connect ignition exciter lead, starter-generator A and D leads, thermocouple leads (if applicable) and fuel pressure switch splice. Install fiberglass sleeving (C50) over knife splice and secure with high temperature lacing cord (C52). Install electrical connectors to tach generators and chip detectors. Secure connectors with 0.032-inch lockwire (C57).

v. Install power turbine governor and gas producer levers removed from the previously installed engine. Reuse the attaching nuts. Instructions for installation and adjustment of the levers are contained in paragraph 4-215.

4-15. Installation — T63-A-700 Engine Accessories. Before installing a T63-A-700 engine in the aircraft, accomplish the modification described in steps *a* through *e* below.

a. The combustion chamber drain valve assembly (furnished with engine) must be relocated to the aft port of the outer combustion case. (The drain valve differs from that used on the -5A as it is 0.334 inch longer, thus extending the attaching drain tube an additional 0.375 inch through the engine access doors.) Remove the MS9015-03 plug from the aft port of the outer combustion case and install the drain valve (from the forward port) in the aft port. Then install the special double-ended Allison (6854519) plug (removed from the forward port of the -5A or -700 engine previously installed in the aircraft) in the forward port of the outer combustion case. The double ended plug provides attachment of the bracket that supports the igniter lead. Be sure to use an MS9387-03 O-ring with the drain valve assembly and plug.

b. Remove the power turbine governor and gas producer levers from the previously installed engine and install them on the T63-A-700 engine. Instructions for installation and adjustment of the levers are contained in paragraph 4-215. Replace the attaching nuts removed from the previously installed engine.

c. If the engine has a hose assembly (compressor bleed-to-shutoff valve) routing which causes an interference with the engine "oil out" hose assembly, the restraining clamp on the compressor bleed-to-shutoff valve hose may be relocated and the hose assembly rerouted to eliminate this interference during installation (para 4-16).

d. Check to see that bleed valve jet assembly (6875147) is installed on the T63-A-700 engine. All T63-A-5A engines and most T63-A-700 engines have this new valve assembly installed; however, some T63-A-700 engines have the old MS24394C4 elbow tube installed at this location. If required, replace the old configuration valve assembly on the T63-A-700 engine with the new configuration valve assembly from the engine being removed. If the part number is not visible on the valve assembly on the T63-A-700 engine, its configuration can be determined by removing the valve from the engine and checking for a jet assembly installed inside the valve. Only the new configuration valve assemblies (6875147) have these jets installed.

d.1. If the engine has two accumulators installed (figure 4-2A, sheet 1 of 2), remove the second accumulator as follows:

(1) Loosen clamp (1, figure 4-2A, sheet 1 of 2).

(2) Disconnect flexible hose (2) at the union (3), hold union with back up wrench and unscrew the nut.

(3) Remove union (3). Hold back up wrench on hexagonal surface on the accumulator adjacent to the union (3) and unscrew the union. Discard the o-ring (4).

(4) Remove clamp assy (5) from the fire shield and accumulator.

(5) Hold elbow (8) with back up wrench and remove the accumulator (6) by turning on hexagonal surface adjacent to the union. Discard o-ring (7).

(6) Install clamp assy (5 , figure 4-2A, sheet 2 of 2), on double check valve (12). Attach clamp to fire shield using bolt, spacer, washer and nut. Tighten clamping nut to 35 - 40 inch-pounds.

(7) Place new o-ring (4) on union (3). Do not lubricate. Install union (3) in elbow (8). Hold elbow with back up wrench. Tighten union 80 - 120 inch-pounds.

(8) Attach hose and nut (2) to union (3). Hold union with back up wrench and tighten nut 80 - 120 inch-pounds.

(9) Adjust clamp assembly (1) and tighten clamping nut 35 - 40 inch-pounds.

(10) Return removed accumulator with the engine which is to be returned for overhaul.

e. After the installations have been performed, the T63-A-700 can be installed in the aircraft according to instructions contained in paragraph 4-16.

f. Verify that the fuel control heating system is installed according to MWO 55-2840-231-30/3.

Table 4-5. Premaintenance Requirements for Installation of the Engine.

Conditions	Requirements
Special Tools	(T11) (T38)
Minimum Personnel Required	Two (MOS 67V & 68B)
Consumable Materia'	(C57)

4-16. Installation in Mounts — Engine Assembly. (See fig. 4-2.) *a.* Position engine stand (T38) as close as possible to aircraft, with engine access doors open.

CAUTION

Do not grasp fuel or oil lines during engine installation. Use particular care not to contact adjoining structure or components. Make certain that engine is tilted so that inlet bell, firewall seal and compressor oil line clear the compressor armor (if installed).

b. Remove the engine from the stand and position under the engine compartment.

c. Install engine hoist (T11) between hoist fitting located on airframe above engine and on hoist fitting of engine.

d. Support replacement engine in position for access to engine mount fitting locations.

e. Install the three engine mount fittings including bonding jumpers and wiring harness clips (if applicable), by using nine washers and bolts (detail L, sh 2). **TORQUE TO 140 TO 160 INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

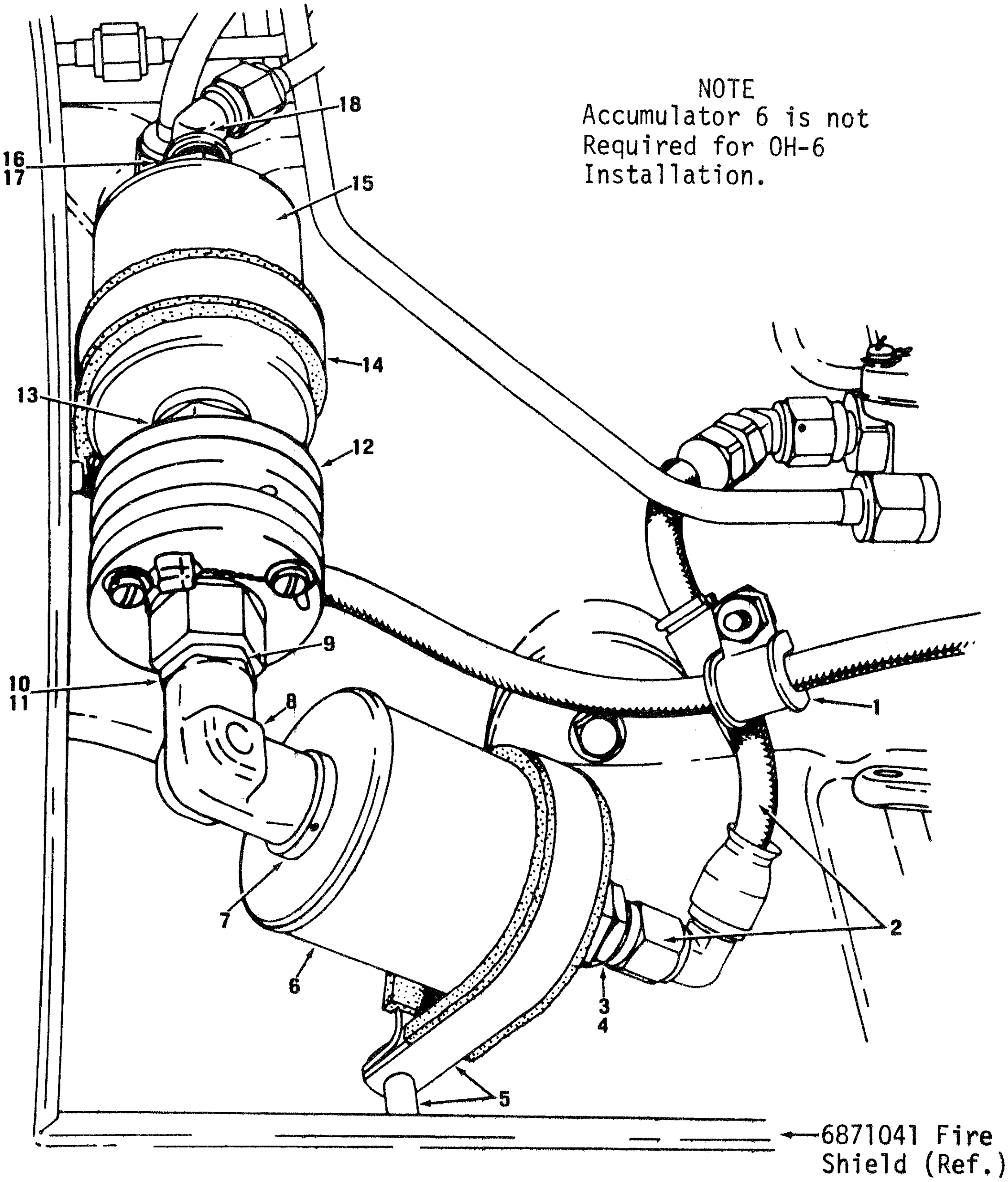


Figure 4-2A. OH-6 Accumulator and Diaphragm Type Double Check Valve Installation (sheet 1 of 2).

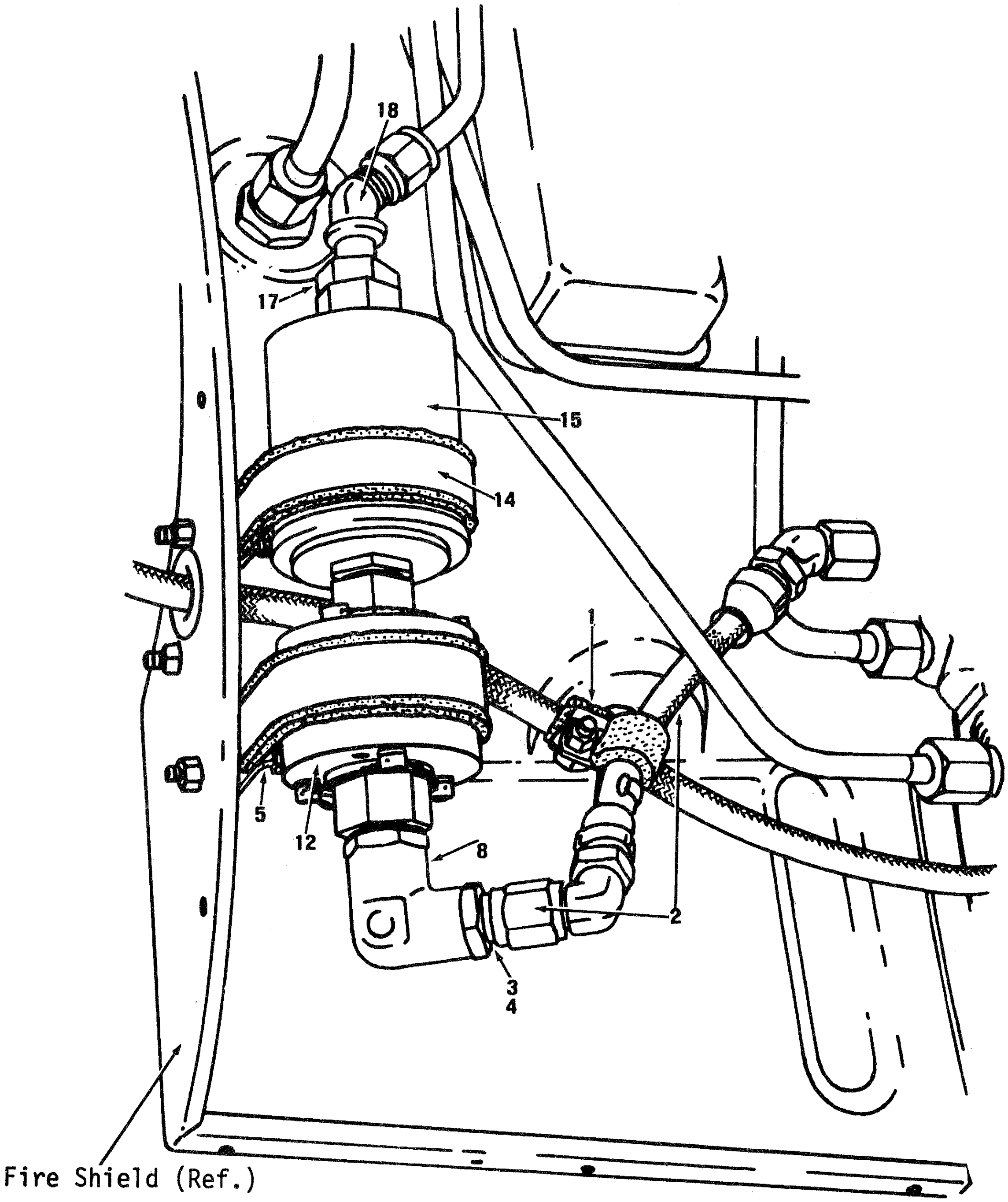


Figure 4-2A. OH-6 Accumulator and Diaphragm Type Double Check Valve Installation (sheet 2 of 2).

Key to Figure 4-2A.

1. Clamp Assy.
2. Hose and Nut
3. Union
4. O-Ring
5. Clamp Assy.
6. Accumulator
7. O-Ring
8. Elbow
9. O-Ring
10. Union
11. O-Ring
12. Double Check Valve
13. O-Ring
14. Clamp
15. 6875224 Accumulator (Ref.)
16. O-Ring
17. Nut
18. Elbow

f. Have an assistant slowly raise the engine while necessary pressure is applied downward at rear of engine.

NOTE

See figure 4-1 while accomplishing the sequence below. Reconnect the engine by reversing the number sequence shown.

g. Align left and right engine mounts with mount fittings and install bolts and washers. **TORQUE BOLTS TO 100 - 140-INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

h. Align hole in lower engine mount with mount fitting and install bolt and washer. **TORQUE BOLT TO 100 - 140-INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

NOTE

Check to ensure that the engine air inlet firewall seal is flush with the firewall.

i. Remove engine hoist from aircraft.

4-17. Installation — Engine Connections. (See fig. 4-1.) *a.* Connect engine oil vent line. **TORQUE THE LINE NUT TO 120 - 140 INCH-POUNDS.**

CAUTION

Do not overtorque engine oil inlet or outlet couplings; damage to the self-closing valve in the coupling may result. Do not apply extreme bending loads on the oil outlet line. Such loads will be transferred to the oil cooler

inlet port boss and can cause the cooler frame to fracture.

b. Connect engine oil outlet line. Hand-tighten until snug; then **WRENCH-TIGHTEN APPROXIMATELY 1/4 TURN** or until definite resistance is felt.

c. Connect engine oil inlet line. Hand-tighten until snug; then **WRENCH-TIGHTEN APPROXIMATELY 1/4 TURN** or until definite resistance is felt.

d. Connect oil line from oil pressure sender clamped on right engine mount.

e. Connect accessories drive overboard vent. **TORQUE THE TUBE NUT TO 150 - 250 INCH-POUNDS.**

f. Connect N1 control rod to lever with bolt, two washers, nut, and new cotter pin. Install the bolt with head outboard, one washer under the nut and one washer under the head.

NOTE

If fuel controls armor is to be installed the N1 control rod is not connected until after the armor installation.

g. Install anti-icing valve control cable in lever adapter. Make an operational check of the cable installation (para 4-146).

h. Connect short N2 control rod to governor lever with bolt (head up), two washers, nut, and new cotter pin. Connect long N2 control rod with bolt (head forward), two washers, nut and new cotter pin.

i. Connect oil line to torquemeter fitting below N2 tach generator.

j. Install compressor cooling air duct support bracket on hoisting eye above engine. Connect and tighten flexible duct on firewall coupling.

k. Connect cabin air outlet tube to firewall fitting. **TIGHTEN BOTH TUBE NUTS TO 350 INCH-POUNDS.**

l. Connect engine fuel pump seal drain line. **TORQUE TUBE NUT TO 20-30 INCH-POUNDS.**

m. Connect fuel inlet line to fuel pump. Torque fuel line to pump fitting according to the fuel system installed. Refer to paragraph 10-55 (fig. 10-8) and paragraph 10-144 (fig. 10-16).

CAUTION

A twisted self-sealing (armored) fuel hose will result in a partial blockage in the line and cause flameout at high power settings. The self-sealing outer hose prevehts visual detection of a twisted inner hose. Care should be taken during installation, *t* following, not to twist the couplings 180° from their original position.

NOTE

If fuel controls armor is to be installed, t below, the 90-degree elbow of the fuel inlet line must be angled to provide 0.13-inch clearance between the line and the hole cutout in the armor.

n. Install combustion chamber drain line on drain valve.

NOTE

Check that the burner drain valve and plug are correctly installed in the combustion chamber. Each part must be installed with an O-ring, with the drain valve in the lower position.

o. Secure engine harness to engine mount with cushioned loop clamp. Connect oil pressure sender lead.

p. Install generator leads E, C, and B on starter-generator and secure with washers and nuts.

q. Position thermocouple leads on terminal studs and secure with chromel and alumel locknuts.

r. Connect engine harness plug to receptacle and secure with lockwire. If installed, connect bond jumper terminal to GRD on engine compartment firewall.

s. Install engine exhaust tailpipes (para 4-160).

t. Install fuel controls armor, if required (para 4-39).

u. Install main transmission drive shaft (Chapter 6).

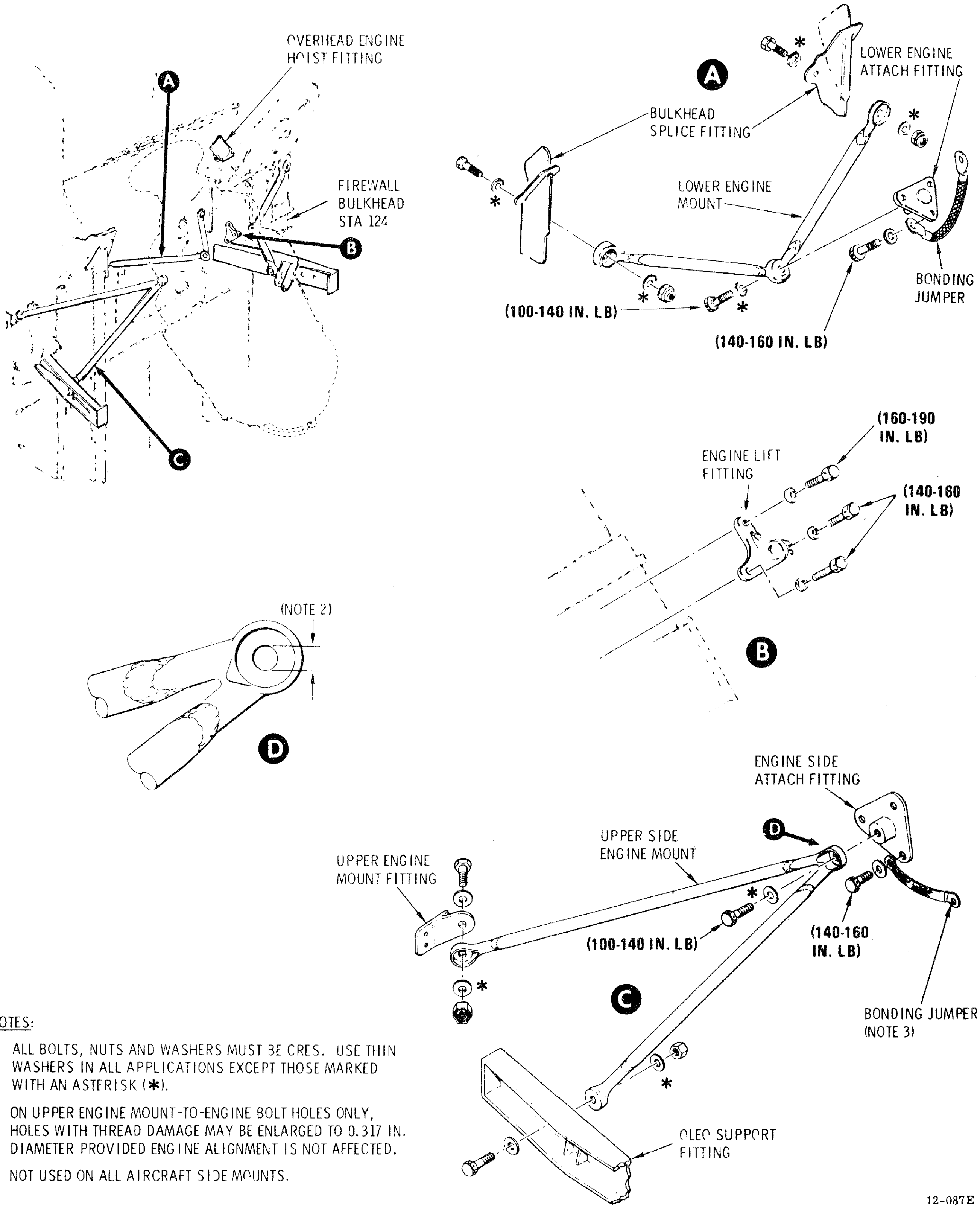
v. Install heating system and oil cooler access doors, main transmission cover assembly, and main gearbox access door. Install sound insulation.

w. Fill engine oil system.

x. Adjust gas producer (N1) and power turbine (N2) engine power controls (para 4-220 and 4-222).

y. Check overall condition of engine. Open fuel shutoff valve and operate electric fuel pump while making an engine fuel system leakage check.

z. Install the diffuser vent orifice and perform engine operational check (TM 55-2840-231-23). Correct malfunctions, if any, as necessary.



NOTES:

1. ALL BOLTS, NUTS AND WASHERS MUST BE CRES. USE THIN WASHERS IN ALL APPLICATIONS EXCEPT THOSE MARKED WITH AN ASTERISK (*).
2. ON UPPER ENGINE MOUNT-TO-ENGINE BOLT HOLES ONLY, HOLES WITH THREAD DAMAGE MAY BE ENLARGED TO 0.317 IN. DIAMETER PROVIDED ENGINE ALIGNMENT IS NOT AFFECTED.
3. NOT USED ON ALL AIRCRAFT SIDE MOUNTS.

Figure 4-3. Engine Mounts and Fittings.

4-18. ENGINE MOUNTS AND FITTINGS.

4-19. Description — Engine Mounts and Fittings. The engine mount installation consists of one lower (center) and two upper (side) engine mounts, six structural attach fittings, three engine attach fittings, and associated attaching hardware. (See fig. 4-3.) Each of the three mounts is a tubular steel, V-shaped, welded assembly. The forward ends of the lower mount are bolted to the two bulkhead splice fittings. The aft end of the lower mount is bolted to the engine attach fitting on the underside mounting pad of the power and accessories gearbox. The two upper mounts are symmetrically opposite assemblies. The longer ends of the upper mounts are bolted to mount fittings at the outboard edges of station 124.00 bulkhead and the shorter ends are bolted to the oleo support fittings. The aft ends of the left and right upper mounts are bolted to the engine attach fittings on the side mounting pads of the power and accessories gearbox. The three engine fittings (fig. 4-3) bolted to the power and accessory gearbox are machined aluminum alloy forgings, each containing a steel threaded insert. The engine lift fitting is an anodized machined aluminum alloy forging. The lift fitting has an eye for attachment of the engine hoist (T11) used for engine removal and installation. The lift fitting is mounted on the upper side of the power and accessory gearbox housing just aft of the engine air inlet scroll assembly.

4-20. Inspection — Installed Engine Mounts and Fittings. *a.* Check all engine mount assemblies for general condition of surface cadmium plating where paint is chipped away.

b. Check all engine mount tubes for straightness. Hold a steel straightedge against the surface of tubes to reveal any warps or bends in the tubes.

c. Visually check all welded joints for cracks and evidence of corrosion.

d. Check engine mount tubes and end fittings for cracks, nicks, dents, scratches, and evidence of corrosion.

e. Check engine mount attaching bolts, engine attach fitting bolts and bonding jumper for evidence of tool damage and corrosion.

f. Visually check flanges and bosses of engine attach fittings for deformation, cracks, nicks, dents, and corrosion.

g. Check engine lift fitting for condition of anodized surface, corrosion, wear in lifting eye, cracks, nicks, dents, and scratches.

Table 4-6. Premaintenance Requirements for Removal of Lower Engine Mounts.

Conditions	Requirements
Special Tools	(T11)
Minimum Personnel Required	Two (MOS 67V & 68B)

4-21. Removal — Lower Engine Mount.

CAUTION

To avoid disturbing security of the engine installation, do not loosen and/or remove more than one engine mount at any one time. Removal of the engine does not require removal of the engine mounts. Do not use open end wrenches when removing engine attach fittings and attaching hardware. Use box end and socket wrenches to avoid tool damage to hardware and engine mount fittings.

a. Remove the bolt and nut that attach the compressor air duct support bracket to overhead engine hoist fitting.

b. Install engine hoist (T11) between overhead hoist fitting and engine lift fitting (detail C, fig. 4-3). Apply enough tension with hoist to relieve stress from lower mount (detail B). Engine loading will be negative when bolts that secure the engine mount to the airframe turn freely with attaching nuts backed off one-half turn.

c. Release cargo compartment lower isolation blanket enough to gain access to the bolt heads protruding through firewall at bulkhead splice fittings (detail A).

d. At left and right forward ends of lower engine mount, remove two bolts, four washers, and two nuts at attach points. Discard the two nuts.

e. Remove engine attach bolt and washer from engine attach fitting (lower).

f. Remove three bolts, washers, bonding jumper (if installed) and fitting from mounting pad of power and accessory gearbox.

4-22. Removal — Upper Engine Mount. *a.* Remove engine oil pressure sender clamps and engine harness clamp from right side engine mount. (See detail A, fig. 4-1, sh 1.)

b. Remove bolts, washers and nuts from attach points of upper engine mount (detail B, fig. 4-3) at upper mount fitting and oleo support fitting. Discard nuts.

c. Remove engine attach bolt and washer to free aft end of mount from engine attach fitting (side) on pad of power and accessory gearbox.

4-23. Removal — Engine Mount Fitting. (See fig. 4-3.) Remove three bolts, washers, and engine attach fitting (lower or side) from mounting pad of power and accessory gearbox. (See detail A or B.) Remove bond jumper (if installed) from fitting.

4-24. Removal — Engine Lift Fitting. Remove engine fitting (detail C, fig. 4-3) by removing three bolts and washers.

4-25. Inspection — Engine Mounts and Fittings. *a.* Magnetic-particle inspect engine mount assemblies according to TM 55-1500-204-25/1 if condition is questionable. If cracks are found, the mount shall be considered unserviceable.

b. Check engine mount-to-engine bolt holes for thread damage. Determine if upper engine mount-to-engine holes have been opened to **0.317-INCH DIAMETER.** If both left and right mounts are reworked, engine-to-transmission alignment must be checked (para 4-47).

c. Check engine mount attaching bolts for damaged threads. Replace defective bolts.

d. Check flanges and bosses of engine attach fittings for flatness, cracks, and corrosion. Replace defective fittings.

e. Check holes in mount flanges for elongation and deformation. Replace defective mounts.

f. Check threaded steel insert for damaged threads, corrosion, and for full bottoming to seat of insert hole.

g. Check engine lifting eye for flatness, wear, and corrosion.

4-26. Repair — Engine Mounts. No straightening or weld repair of engine mounts is authorized.

4-27. Installation — Engine Mount Fitting. (See fig. 4-3.) *a.* Install engine attach fitting (lower or side) by inserting chamfered side of fitting into mounting pad recess. Align holes in fitting flange with three threaded holes in mounting pad.

b. Install bonding jumper, as required, and three bolts with washers. **TORQUE BOLTS TO 140 - 180 INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

4-28. Installation — Lower Engine Mount. *a.* Place ends of lower engine mount (detail A) against bulkhead fittings.

b. Install two bolts, four washers, and two new nuts at bulkhead attach points.

CAUTION

Do not tighten bolts installed in *b* above until the engine attach bolt is tightened according to *c* below.

c. Install engine attach bolt and washer that secure mount to fitting. **TORQUE BOLT TO 100-140 INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

d. Tighten bulkhead fitting bolts and nuts.

Table 4-7. Premaintenance Requirements for Installation of Upper Engine Mounts.

Conditions	Requirements
Special Tools	(T11)
Minimum Personnel Required	Two (MOS 67V & 68B)
Consumable Material	(C57)

4-29. Installation — Upper Engine Mount. (See fig. 4-3.) Inspect engine as described in step *b* of para 4-25 before installation.

a. Place short end of upper engine mount (detail B) against oleo support fitting and long end against upper engine mount fitting.

CAUTION

Do not tighten bolts installed in *b* below until the engine attach bolt is tightened according to *c* below.

b. Install bolt, two washers, and new nut to fasten mount to structure.

c. Install engine attach bolt and washer that secure mount to fitting. **TORQUE BOLT TO 100-140 INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

d. Tighten upper mount fitting and oleo support fitting nuts and bolts.

e. Remove engine hoist (T11) and reinstall the compressor air duct support bracket in the overhead hoist fitting.

f. Reinstall engine oil pressure sender clamps and engine harness clamp on right side engine mount. (See detail A, fig. 4-1, sh 1.)

4-30. Installation — Engine Lift Fitting. Install engine lift fitting (detail C, fig. 4-3) by installing two 5/16 inch-diameter (side) bolts and washers, and one 3/8 inch-diameter (front) bolt and washer. **TORQUE SIDE BOLTS TO 140-160 INCH-POUNDS, AND FRONT BOLT TO 160-190 INCH-POUNDS.**

4-31. ENGINE ARMOR KIT.

4-32. Description — Engine Armor Kit. The engine armor kit installation consists of an engine fuel controls armor assembly. (See details A and B, fig. 4-4.) The box-shaped fuel controls armor is mounted below the engine on two support arms. The box shape provides protection for the engine fuel pump, gas producer fuel control, power turbine governor and most of the related interconnecting lines. The mounting position also gives protection to the lower section of the power and accessories gearbox. The fuel system armor is fabricated of reinforced ceramic tile. The armor unit has a large rectangular opening in the lower right side for installation and operational movement of the N1 gas producer control rod. The lower left side has an oblong hole for entry of the fuel supply line. A 0.38-inch hole is centered in the bottom of the rear panel to drain any fuel or oil leakage. The channel-shaped compressor armor is mounted in the air inlet recess and provides protection for the bottom and sides of the compressor. Replacement of compressor armor is outlined in paragraph 4-46. The fuel controls armor is interchangeable among aircraft equipped with the installation provisions.

WARNING

If the aircraft is to be operated with the engine fuel controls armor removed, part or all of the crew armor must also be removed, depending on mission requirements and aircraft loading. A weight and balance check must always be made to ensure that center of gravity limits will not be exceeded.

4-33. FUEL CONTROLS ARMOR.

4-34. Inspection — Fuel Controls Armor Installed. *a. Check* for any evidence of cracks; rigidity and security of mounting to the right and left support (detail B, fig. 4-4); signs of fuel or oil leakage at the drain hole; and any evidence of a bullet or shrapnel strike.

NOTE

The fuel controls armor must be completely removed and a thorough inspection made of the engine (TM 55-2840-231-24) when there is evidence of fuel or oil leakage at the drain hole.

b. With armor installed:

(1) Inspect armor supports (detail B) for security of mounting to the aircraft structure, abnormal bends in the support tubing, and cracks.

(2) Inspect the armor installation for a minimum clearance of 0.13-inch between all surfaces and edges of the armor and the engine, the armor and the fuel inlet line. Check that clearance also exists between the left support and the engine gearcase cooling duct clamped to the firewall.

(3) While an assistant rotates the throttle grip through full travel on the pilot's collective pitch stick, check that there is no interference between the armor and the gas producer control rod.

4-35. Removal — Fuel Controls Armor. (See detail B, fig. 4-4.) *a.* Without loosening rod end bearings, disconnect both ends of the gas producer control rod. Carefully remove rod from between armor and engine.

b. With a container in place to catch trapped fuel, disconnect lower end of engine fuel supply hose. Drain fuel from line into container. Cap fuel line fitting.

c. Remove the bolts, washers, and nuts that secure the right and left supports (detail B) to the firewall.

d. While an assistant supports the armor unit, remove the bolts, washers and nuts that secure the supports to the structural ring. Using one man to support each side, slowly start to lower the armor unit. As the armor assembly is lowered, feed the fuel line through the hole in the left side of the armor. Continue lowering downward and aft until armor assembly clears the engine and structure.

e. If aircraft is to be operated without fuel controls armor installed, remove cap and reconnect lower end of fuel line. **TORQUE NUT TO 270 - 325 INCH-POUNDS.**

4-36. Disassembly — Fuel Controls Armor. Disassemble the fuel controls armor by removing the bolts, washers and nuts attaching the right and left supports to the armor unit. Observe that the two forward bolts are longer than the others.

4-37. Repair or Replacement — Fuel Controls Armor. *a.* Replace the armor if it is cracked or has received a strike.

b. (AVIM) Repair cracked supports (detail B, fig.

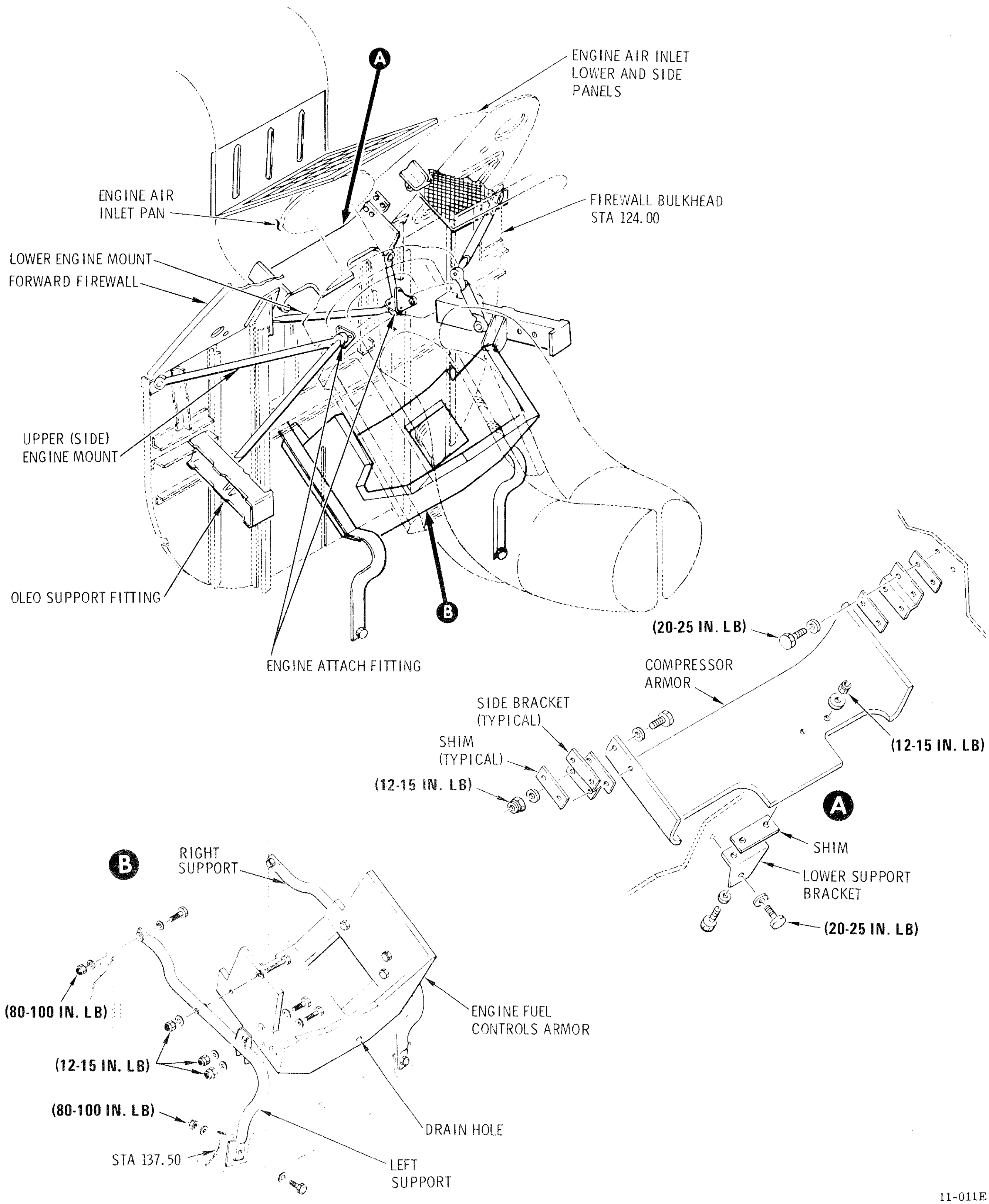


Figure 4-4. Engine Armor Kit Installation.

4-4) by inert arc welding according to TM 55-1500-204-25/1. Splices may be used in repair of cracked or kinked tubes provided the original tube angularity is maintained or restored. The supports are fabricated of Type 321 or 347 corrosion-resistant steel tube with 0.020-inch wall.

4-38. Assembly — Fuel Controls Armor. (See detail B, fig. 4-4.) Assemble by attaching the right and left supports to the armor unit with bolts, washers, and nuts. Install the two longer bolts at the forward attach points. **TORQUE THE NUTS TO 20-25 INCH-POUNDS.**

4-39. Installation — Fuel Controls Armor. (See detail B, fig. 4-4.) *a.* With the help of an assistant, slowly lift assembled armor upward and forward into position under engine.

b. As the armor is lifted, route the fuel line out through the hole in the left side of the armor. Continue lifting until armor attach points are in line and install the lower bolts, washers and nuts. **TORQUE THE NUTS TO 50-70 INCH-POUNDS.**

c. Continue to support the back of the armor and install the bolts, washers and nuts that attach the forward ends of the support to the firewall. **TORQUE THE NUTS TO 80 - 100 INCH-POUNDS.**

d. Uncap the fuel line and connect the lower end of the hose to the engine compartment bulkhead union. **TORQUE HOSE NUT TO 270 - 325 INCH-POUNDS.**

CAUTION

A twisted fuel hose will result in a partial blockage in the line, causing flame-out at high power settings. The self-sealing hose prevents visual detection of a twisted inner hose. Care should be taken during installation not to twist the coupling from their original position.

e. Carefully guide gas producer rod through right side armor cut-out and into place. Check that both rod ends are correctly aligned and install the attaching hardware. Secure nuts with new cotter pins.

f. Make a thorough inspection of the installation. (Refer to *a* above.) Check for a 0.13 inch minimum clearance between compressor armor and the torquemeter oil pressure line.

4-40. COMPRESSOR ARMOR.

4-41. Description — Compressor Armor. The compressor armor is fabricated of two steel plates fused together to form a single laminated thickness. Compressor armor is interchangeable among aircraft equipped with the installation provisions. (See detail A, fig. 4-4.)

4-42. Inspection — Compressor Armor Installed.

a. Check compressor armor for any evidence of cracks, security of support brackets and mounting to the structure, and clearance on all sides. Check for any evidence of a bullet or shrapnel strike. Armor shall be replaced if it is cracked or has received a strike.

b. Check that a minimum clearance of 0.13 inch exists between the compressor armor and the torquemeter oil pressure line.

4-43. Removal — Compressor Armor. *a.* Remove the engine (para 4-10).

b. Insert suitable wood blocking (approx. 2 in. × 2 in. × 0.38 inch thick) between the bottom of the compressor armor and the air inlet panel. (Blocking will support armor when hardware is removed.)

c. Remove attach bolt from lower support bracket (detail A, fig. 4-4). Remove the two (upper) bolts from each side bracket.

NOTE

The lower and side brackets must be kept with the compressor armor to which they are attached to ensure armor interchangeability.

d. Slide armor from air inlet recess and remove wood blocks. Tie shims to the compressor armor brackets to prevent loss. Do not disassemble side brackets from armor unless brackets are questionable.

4-44. Installation — General. The engine must be removed for compressor armor installation. If the armor to be installed was previously removed from the same aircraft, use the sequence in paragraph 4-45. If replacement armor is to be installed, use the sequence in paragraph 4-46.

4-45. Installation — Compressor Armor. *a.* Place compressor armor in approximate mounting position in air inlet recess. Support armor on wood blocks (approximately 2 in. × 2 in. × 0.38 inch thick).

b. Align shims (detail A, fig. 4-4) between side brackets and matching holes. Start bolts and washers into the four nutplates.

c. Install bolt and washer to secure lower support bracket. **TORQUE BOLT TO 20 - 25 INCH-POUNDS.**

d. **TORQUE THE FOUR SIDE BOLTS TO 20 - 25 INCH-POUNDS.** Remove wood blocks.

e. Check armor for rigidity and clearance on all sides. Ensure that a minimum of 0.13 inch clearance exists between torquemeter oil pressure line and armor.

f. Install engine (para 4-16).

4-46. Installation — New Armor, or Armor Originally Fitted to Another Aircraft. *a.* Install wood blocks (2 in. × 2 in. × 0.38 inch thick) on the horizontal panel of the air inlet recess. Secure with general purpose masking tape.

b. Place compressor armor in approximate mounting position, resting on the two blocks. Hold armor in place.

c. Move armor until right side bracket (detail A, fig. 4-4) is flush against right vertical firewall panel. Align bracket holes with nutplate holes.

d. While holding armor in this position, measure any gap between left side bracket and vertical firewall panel. Shims are required if gap is 0.012 inch or more.

NOTE

Check that not more than five shims are installed between each bracket and the armor. Additional shimming must be between the brackets and firewall panels.

e. Install 0.012-inch-thick shims to eliminate gap between bracket and firewall panel. Use approximately equal division of shims between sides if more than one shim is required.

f. If drilled shims are unavailable remove armor from recess. Fabricate the necessary shim thickness from aluminum alloy sheet stock. Finish dimensions are 0.012 inch × 0.70 inch × 2.12 inches. Material shall be 2024-T3 conforming to Federal Specification QQ-A-250/5. Drill the shims with a No. 7 drill to match the armor and mounting hole patterns.

g. Remove wood blocks. Reposition armor in firewall recess. Use shims between each bracket and firewall side panels, as required. Install bolts and washers to secure each side bracket and shim(s) to firewall. **TORQUE BOLTS TO 20 - 25 INCH-POUNDS.**

h. Observe whether hole in vertical portion of lower bracket is vertically aligned with nutplate hole. If not aligned continue with *i* below. If aligned, secure lower bracket with bolt and washer. **TORQUE BOLT TO 20 - 25 INCH-POUNDS.**

i. If holes in lower bracket and firewall are not vertically aligned, remove bracket from armor and install on firewall with bolt and washer. **TORQUE BOLT TO 20 - 25 INCH-POUNDS.**

j. Measure any gap existing between lower bracket and underside of armor. Determine the number of shim(s) required. Shim(s) are required if gap is 0.012 inch or more.

k. Install 0.012-inch-thick shim(s) to eliminate gap between bracket and armor.

l. If drilled shims are unavailable, fabricate the necessary shim thickness according to *f* above, except for finish size dimensions. Dimensions are 0.012 inch × 0.80 inch × 1.50 inches.

m. Insert required shim(s) between bracket and armor. Use two bolts, two washers, and two nuts to secure bracket to armor. **TORQUE NUTS TO 12 - 15 INCH-POUNDS.**

n. Check armor for rigidity and clearance on all sides. Ensure that a minimum of 0.13-inch clearance exists between torquemeter oil pressure line and armor.

o. Install engine (para 4-16).

Table 4-8. Premaintenance Requirements for Engine-to-Transmission Alignment.

Conditions	Requirements
Special Tools	(T8)
Minimum Personnel Required	Two
Consumable Materials	(C14) (C63)

4-47. Engine-to-Transmission Alignment. If misalignment of the engine to the transmission is suspected, check for proper alignment as follows. (See fig. 4-5.)

a. Check aligning tool (T38) to ensure that the mounting flanges and shaft are not defaced or distorted.

b. With drive shaft removed, attach aligning tool to transmission input shaft coupling with four bolts, eight washers, and four 1/4-28 nuts. **TORQUE BOLTS TO 50 - 70 INCH-POUNDS.**

c. Mount a dial indicator to contact the OD of the tool lower flange. Check that there are no obstacles near the main rotor blades, if installed. Have an assistant rotate the rotor drive system by hand-turning the tail rotor. The total indicated runout (TIR) at the lower end of the tool must not exceed 0.100 inch. (See detail A, fig. 4-5.)

d. If the TIR is more than 0.100 inch, remove tool from coupling. Remove coupling bolt from coupling using care to retain coupling shims. Rotate coupling to a different spline position. Coat coupling splines with lubricant (C63) and coupling bolt threads with antiseize compound (C14) before reassembly. Install coupling and bolt. **TORQUE COUPLING BOLT TO ACTUAL DRAG TORQUE PLUS 250 - 300 INCH-POUNDS.**

e. Repeat *b* and *c* above. If necessary, make several attempts to improve the alignment by further

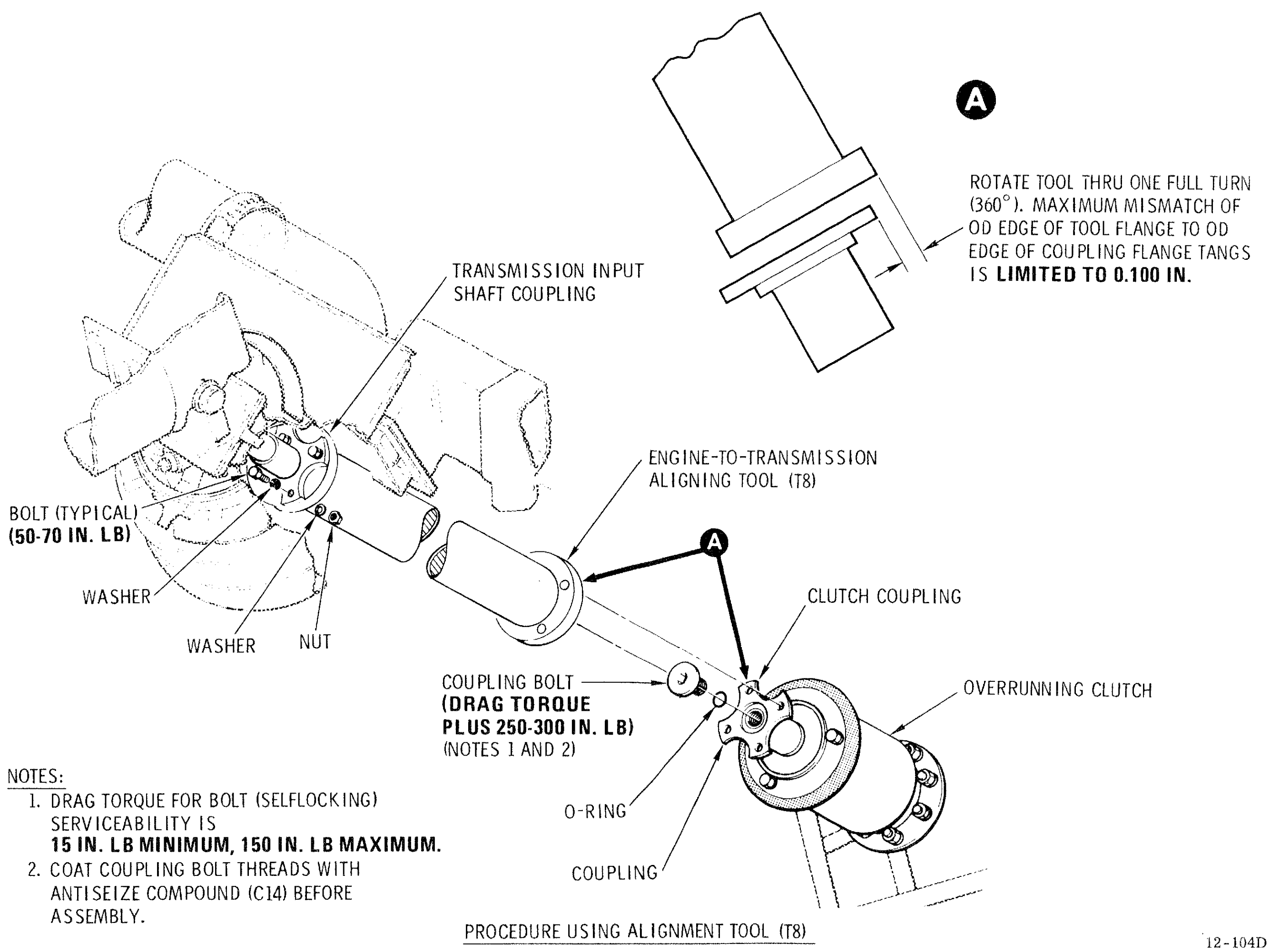


Figure 4-5. Engine-to-Transmission Alignment Check.

repositioning of the coupling. If TIR at the tool lower flange cannot be improved to indicate within the limit, remove the tool and replace the coupling.

f. With the new coupling installed, check TIR of the tool lower flange according to *b* through *d* above. If within the limit, continue with *g* below. If not within the limit, refer to *j* below.

g. Matchmark (pencil, etc) one of the four flange tangs of the clutch coupling to a starting point on the aligning tool flange. Have assistant rotate the drive system to turn the tool in 90-degree increments, through one complete revolution.

h. Measure the mismatch of the tool flange at the edge of each of the four tangs. The mismatch must be 0.100 inch or less for the engine-to-transmission drive alignment to be acceptable (detail A, fig. 4-5). If mismatch is not within the limit, refer to *j* below.

i. Remove tool from the transmission coupling and repeat *b* through *h* above with the aligning tool attached to the clutch coupling.

NOTE

If the clutch coupling is removed, ensure that the coupling shims remain installed on the clutch spline and that the packing remains on the coupling bolt.

j. Mismatch limits are the same for both points of attachment and must be met at both ends of the tool. The alignment of the transmission input shaft coupling and clutch coupling is unacceptable if limits are exceeded. Remove aligning tool after obtaining alignment.

SECTION III COOLING SYSTEM

4-48. COOLING SYSTEM INSTALLATION.

4-49 Description — Cooling System Installation. The engine cooling system consists of a ram air scoops built into the air inlet fairing, an oil cooler fan mounted on the main transmission input shaft, an oil cooler duct, and engine gearcase and compressor cooling ducts (fig. 4-6). Air enters either an external scoop or two inside scoops on the inlet fairings and flows to the oil cooler. Ambient air from the oil cooler blower is directed to and through the engine oil cooler, by the oil cooler duct, into the engine compartment. The oil cooler blower also supplies a constant ambient air flow through the engine gearcase and compressor cooling ducts into the lower and upper areas of the engine compartment. Two duct outlets direct the air upon the engine gearcase and the compressor section. All cooling air then exhausts from the engine compartment through a gap between the exhaust tailpipes and the engine access doors. A special engine gearcase cooling duct outlet is installed in armored aircraft. The duct is tilted left at a 15-degree angle to produce best possible cooling of the gearcase with the fuel controls armor installed.

4-50. Inspection — Cooling System Installation. Visually check for general condition, security of clamps, tubes and ducts, and for fraying or worn spots on tubing.

4-51. Repair — Cooling System Installation. Repair frayed or worn areas of tubes and ducts using tape (C103). Replace any tubes or ducts damaged beyond repair with the same size as that removed.

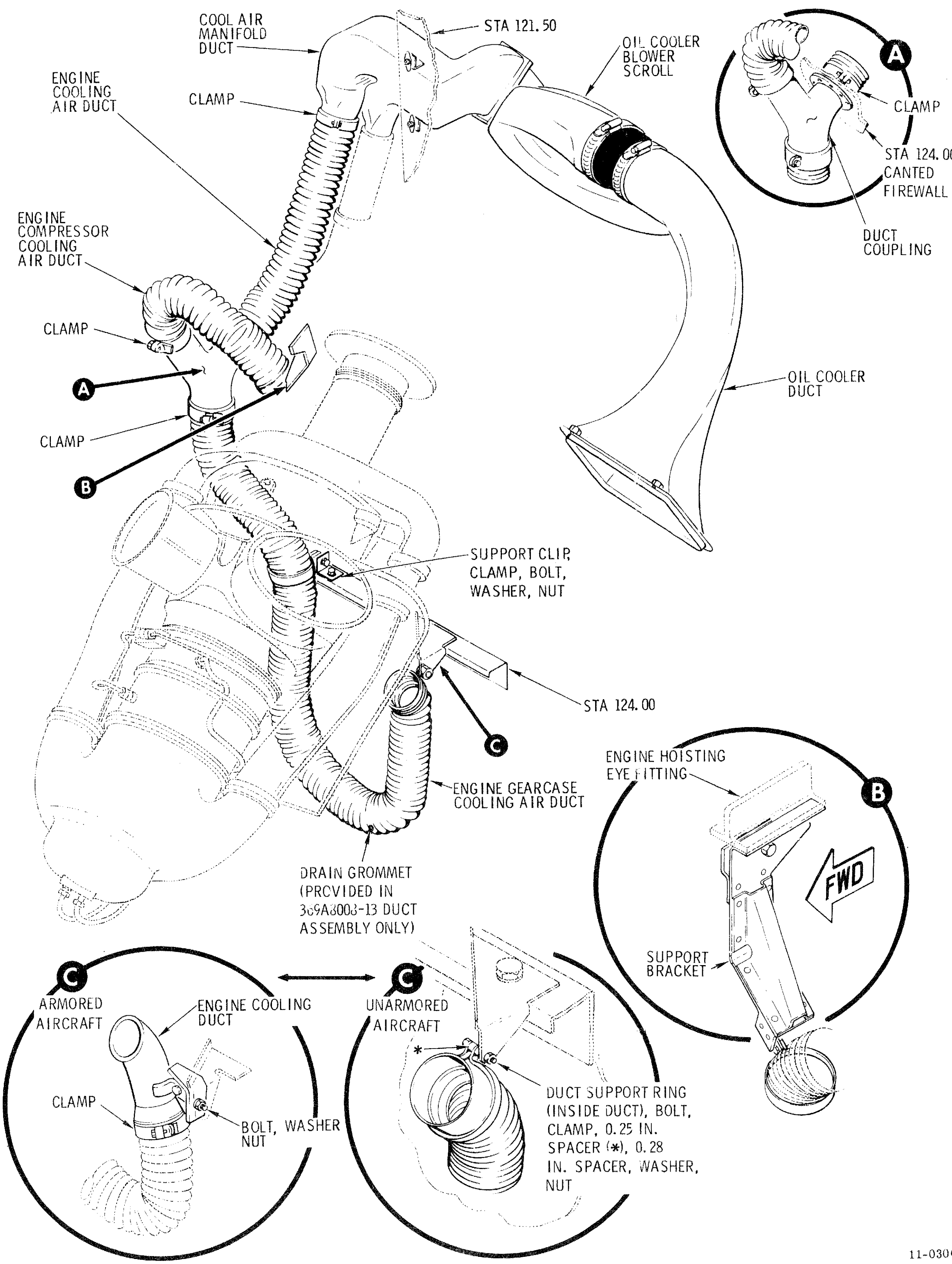


Figure 4-6. Engine Cooling System Installation.

SECTION IV AIR INDUCTION SYSTEM

4-52. AIR INDUCTION SYSTEM INSTALLATION.

4-53. Description — Air Induction System Installation. The air induction system consists of an air inlet installation (plenum chamber), and air filter system, engine air shield screen, engine air inlet bell, engine-to-firewall seal, and an engine air inlet-to-firewall seal. Air filtration is provided by either a barrier filter system or an inertial particle separator installation. Both filter systems may have been modified with a snow ingestion/cold weather kit.

4-54. Description — Engine Air Inlet Barrier Filter Installation. (See fig. 4-7, sh 1). The engine air inlet barrier filter is installed over the plenum chamber in series 1 and 2 aircraft. The unit provides for filtration of air flowing into the engine air intake and protects against foreign object entry. The filter installation consists of a filter element, a plastic filter frame for containing the element, a metal bypass door, pressure sensing switch, bypass air indicating system, and a bypass air control installation. The bypass door located on the front of the filter frame can be opened to permit air to bypass the filter element. A bypass air control release handle controls the opening of the bypass door and is located overhead in the flight compartment. The pressure switch located downstream of the filter element actuates the BYPASS AIR caution light located on the instrument panel.

4-55. Description — Engine Air Inlet Inertial Particle Separator Air Filter Installation. (See fig. 4-7, sh 2.) The inertial particle separator installation used on series 3 aircraft is located and contained within the engine air inlet aft fairing, the unit consists of a filtering system and particle separator, a bypass air control installation, filter bypass air indicating system, and an engine bleed air powered scavenging system. Atmospheric air enters the air filter tubes contained within the filter separator envelope. (See fig. 4-9.) Swirl guides within each of the filter tubes cause the inlet air to be swirled within the filter tubes. Heavier air concentrations (contaminants) are separated from the inlet air as a result of the swirling action and are forced into the separator portion of the air filter. Contaminants collected within the separator portion are withdrawn and discharged overboard through the ejector manifold nozzle and ejector tubes when the scavenge air system is placed into operation. The engine bleed air scavenging system taps bleed air at the cabin heat control and mixing valve. Operation of the bleed air scavenging system is controlled by an engine bleed air shutoff valve and operated by the SCAV AIR-OFF switch. Normally, air enters the inlet fairing, passes through the engine air filter filtering system and then flows into the plenum chamber above the engine air inlet bell and to the compressor inlet. Should the filtering system become clogged, the filter unit has provisions for bypassing the filtering system to allow inlet air to flow directly into the plenum chamber. The installation consists of a hinged bypass door mounted to the aft fairing, a series of pulleys, and a bypass door release cable assembly. A pull handle for the bypass door release is located overhead in the flight compartment. The air induction system is sealed by the engine-to-firewall seal which is springloaded against the engine side of the firewall seal bolted on the front of the over-running clutch at the firewall.

4-56. Description — Snow Ingestion/Cold Weather Kit. On aircraft with a barrier filter, a drain diverter is attached below each of the five filter pleats. The diverters drain filter moisture (water) to a natural trough on the plenum chamber forward face. The water then drains overboard through a drain tube at the right side of the chamber. On aircraft with an inertial particle-separator filter, a lipped seal around the separator lower edge collects water and provides a flow path to the drain tube. A drain hole in the filter element is also provided. Both systems tap-off compressor bleed (hot) air to melt snow and ice and clear the drain system. The hot air tube (fig. 4-10) starts at the cabin heat valve and is routed across the plenum chamber face to the overboard drain. An additional engine air shield screen is installed in both systems to completely screen the engine air inlet bell from large ice or snow formations.

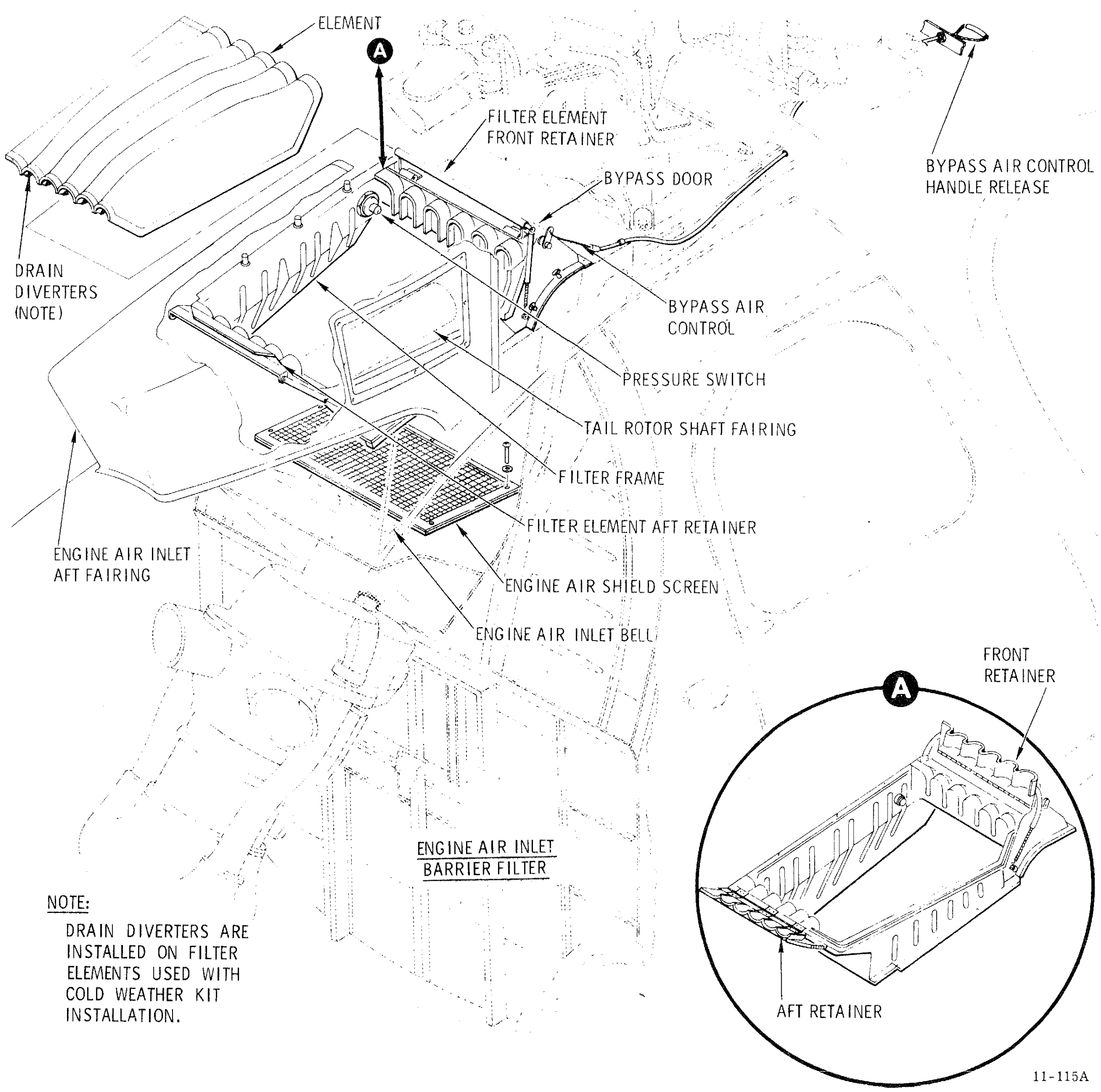
4-57. ENGINE AIR INLET AFT FAIRING.

4-58. Description — Engine Air Inlet Aft Fairing. The engine air inlet aft fairing is a fiberglass structure that covers the plenum chamber. The tail rotor drive shaft access door and plenum chamber access door are on the right side of the fairing on aircraft without the particle-separator filter. Aircraft with the particle-separator have an air filter bypass door instead of the plenum chamber access door, and a filter ejector overflow opening on the left side of the fairing. Two drain holes may be located in the lower aft corners of the fairing.

4-59. Inspection — Engine Air Inlet Aft Fairing. *a.* Check the engine air inlet aft fairing for structural damage. Check for cracked or frayed glass cloth surfaces.

b. Check access doors and opening screen for damage, proper fit, and security of mounting.

c. Check that fairing drain holes, if present, are open and free of foreign matter.



NOTE:

DRAIN DIVERTERS ARE INSTALLED ON FILTER ELEMENTS USED WITH COLD WEATHER KIT INSTALLATION.

Figure 4-7. Air Induction System. (sheet 1 of 2)

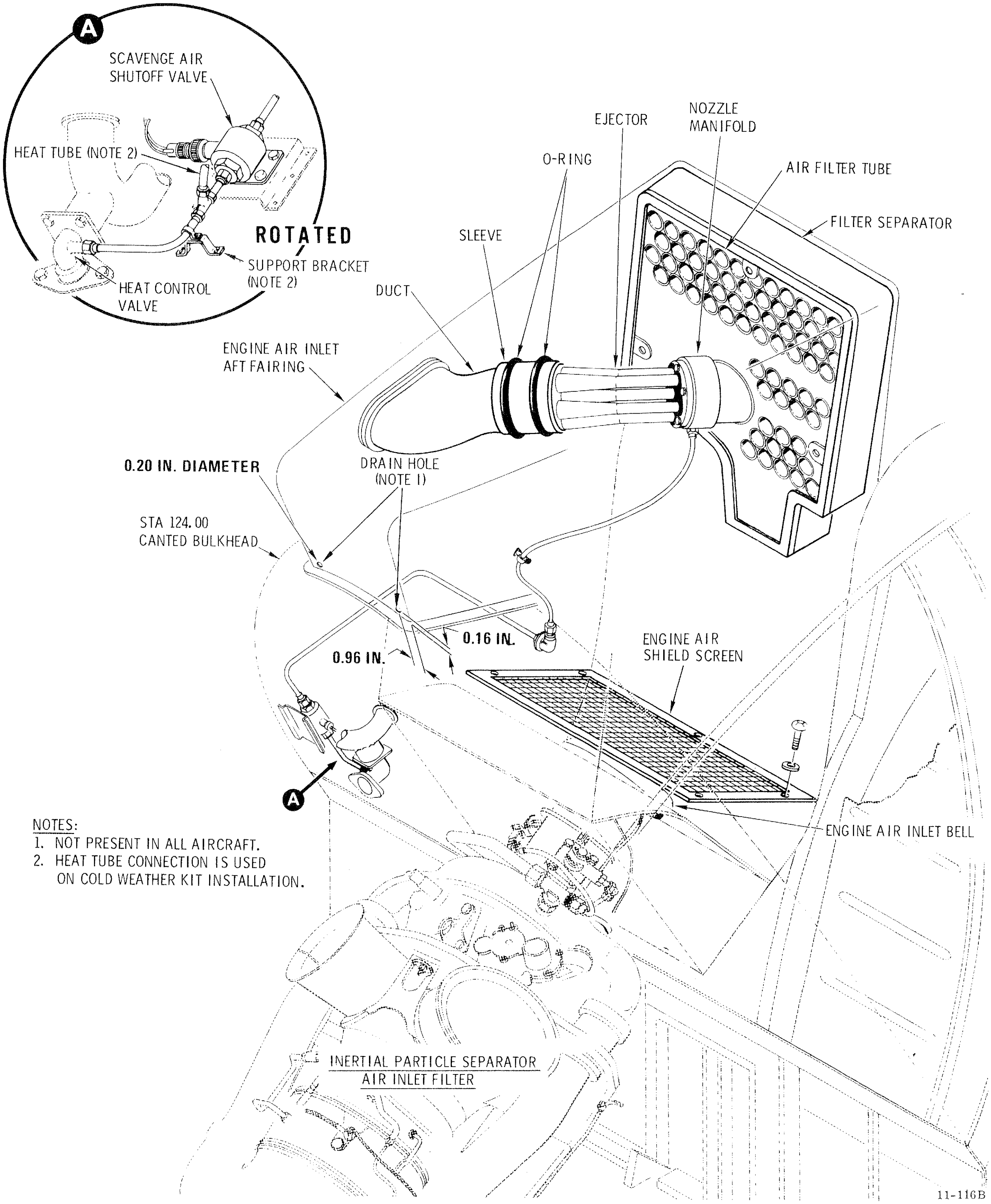
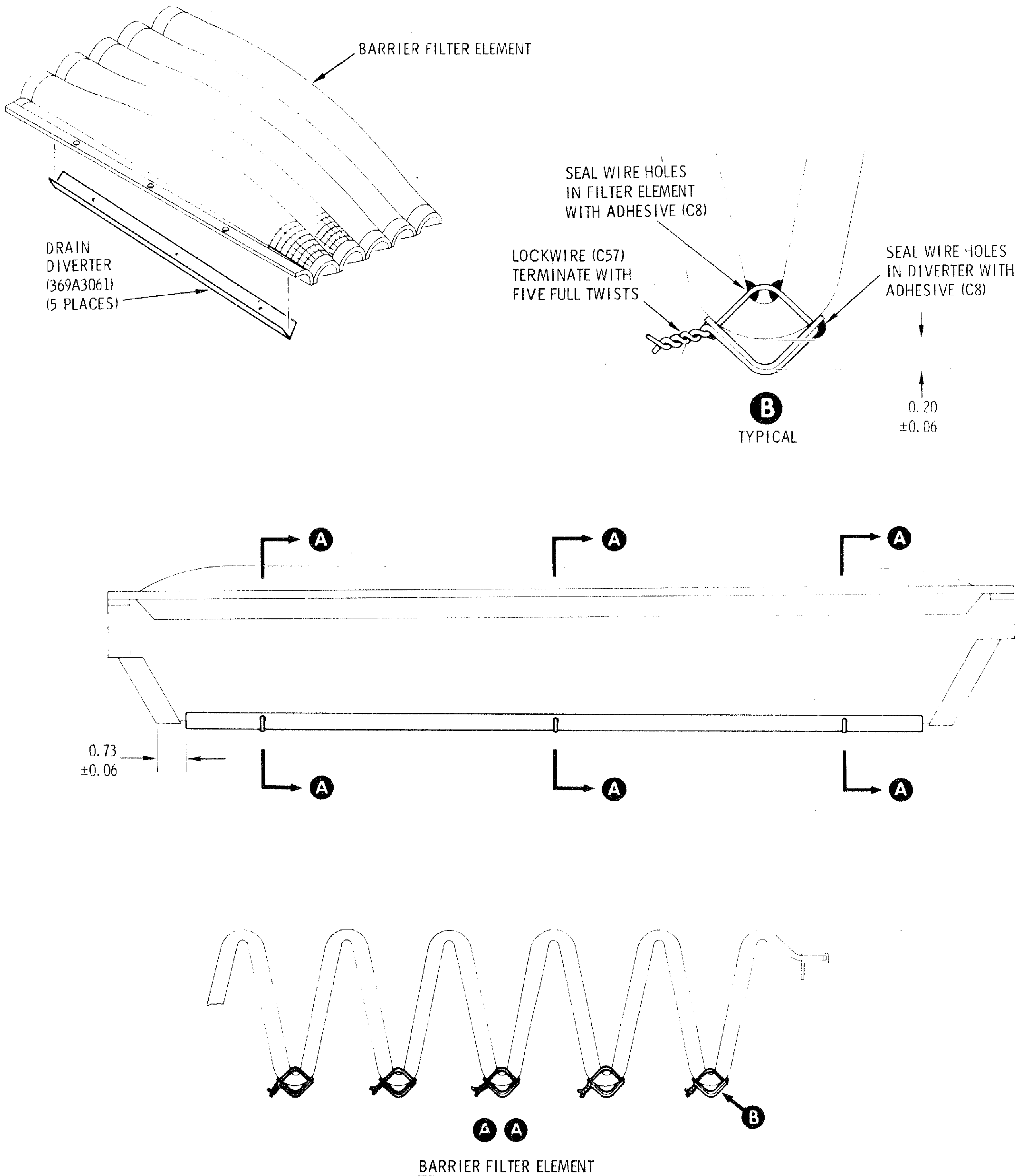


Figure 4-7. Air Induction System. (sheet 2 of 2)



NOTE:
ALL DIMENSIONS IN INCHES.

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Figure 4-8. Rework of Replacement Filter Elements. (sheet 1 of 2)

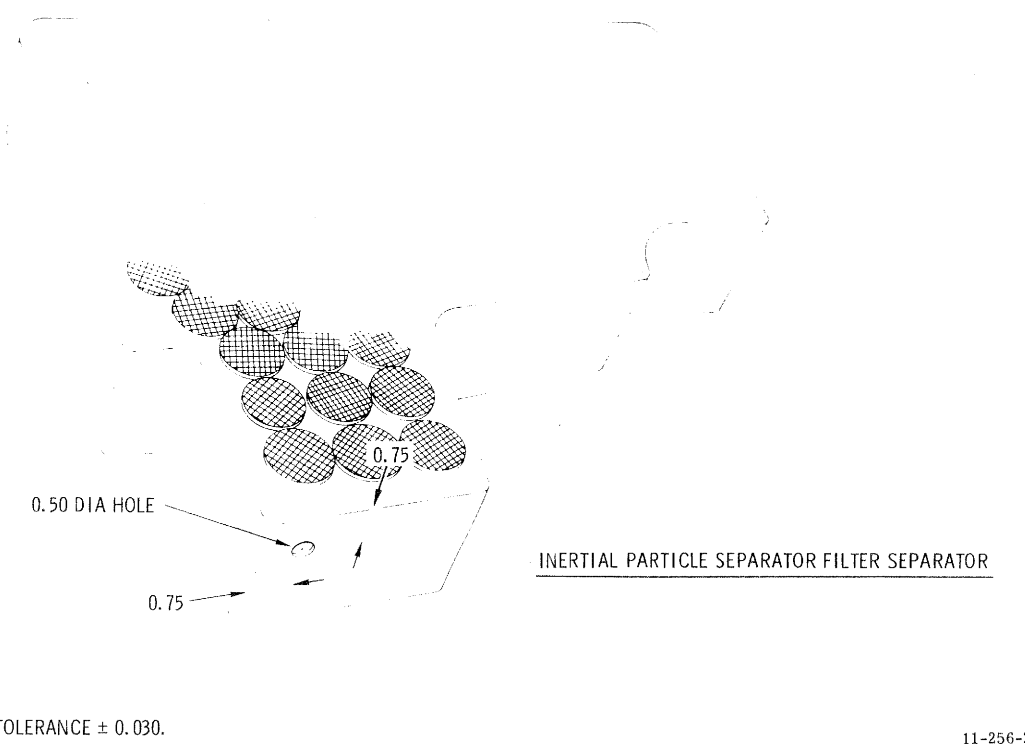


Figure 4-8. Rework of Replacement Filter Elements. (sheet 2 of 2)

4-60. Repair — Engine Air Inlet Aft Fairing. Repair aft fairing using fiberglass repair procedures described in chapter 2.

4-61. ENGINE AIR SHIELD SCREEN ASSEMBLY.

4-62. Description — Engine Air Shield Screen Assembly. The engine air shield assembly consists of either a single aluminum alloy framed screen (fig. 4-7, sh 1 or sh 2) or a dual screen installation (fig. 4-10). Dual screens are used on aircraft equipped with a cold weather kit. The forward screen is then teflon coated.

CAUTION

Use care to avoid dropping screws, washers, or any foreign material into engine air intake. The engine can be badly damaged by the entry of such objects.

4-63. Removal — Engine Air Shield Screen Assembly. *a.* Open plenum chamber access door. On aircraft with inertial particle-separator air filter, open filter bypass door.

b. On aircraft with cold weather kit, remove four screws and washers; then remove aft screen.

c. Remove forward engine air shield screen by removing attaching hardware. Remove from plenum chamber.

d. Tape a cover of cardboard or other suitable material and install over the engine air intake.

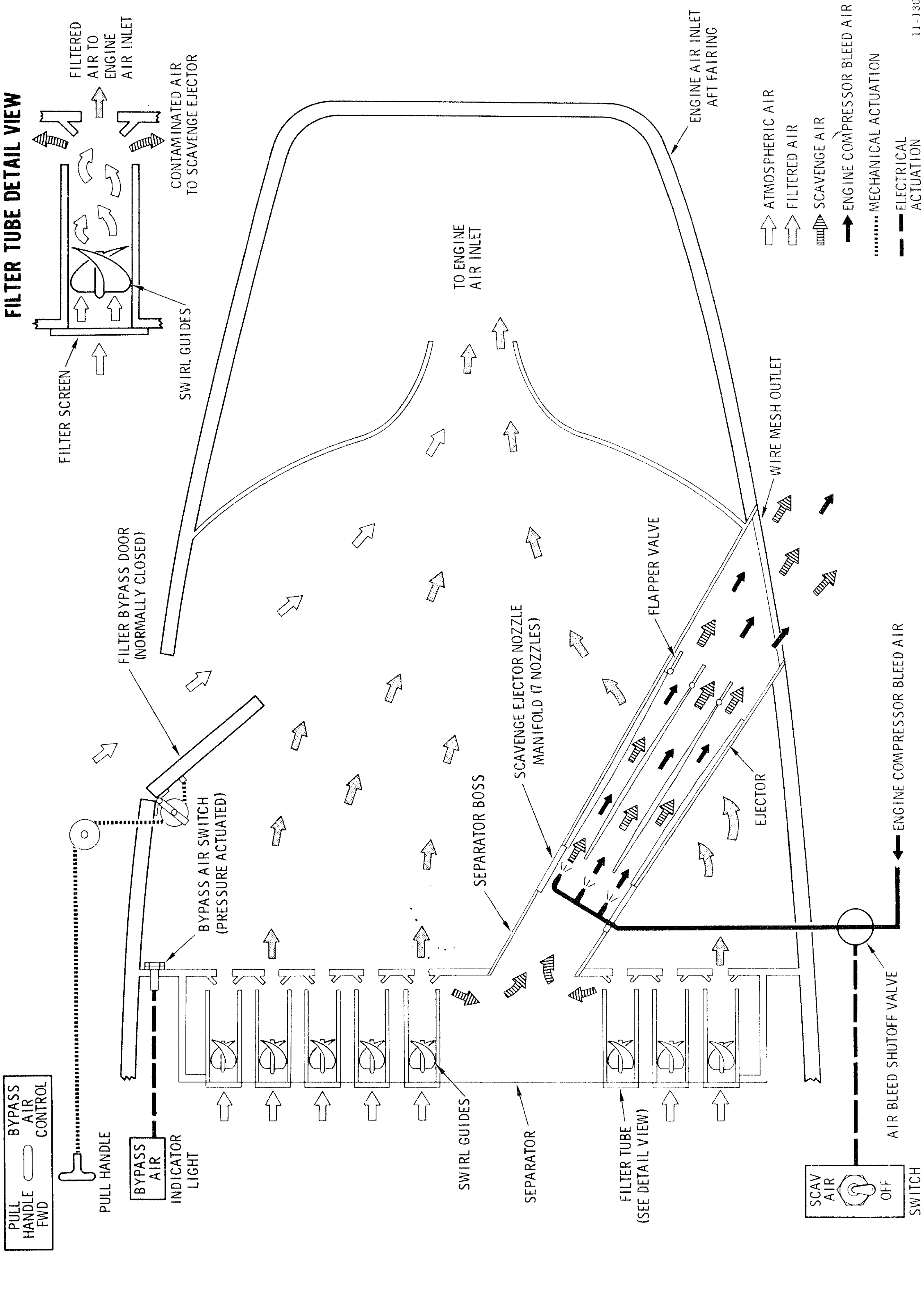


Figure 4-9. Simplified Schematic of Inertial Particle Separator Air Filter.

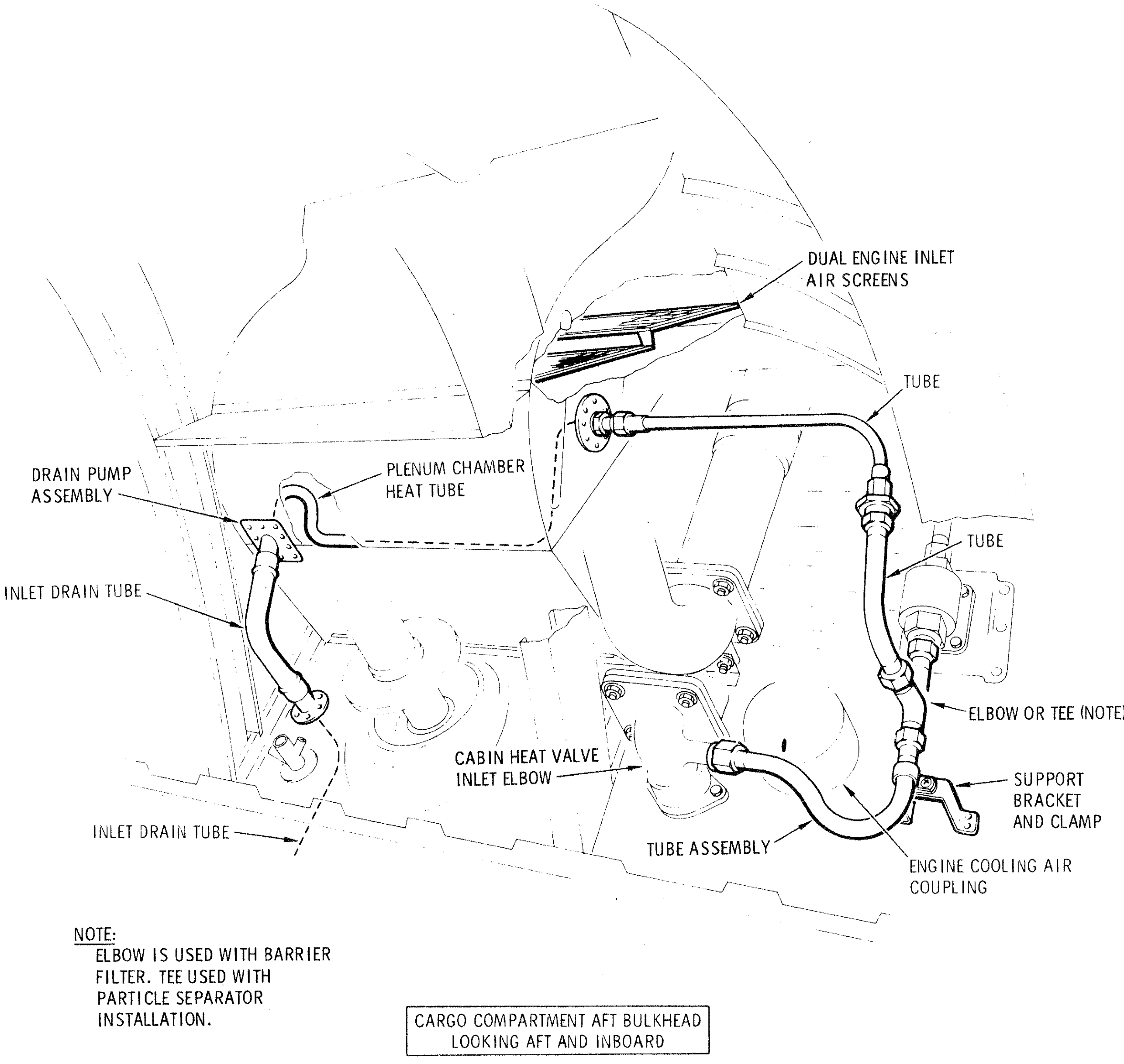


Figure 4-10. Cold Weather Heat Tube Installation.

4-64. Cleaning — Engine Air Shield Screen Assembly. Clean the engine air shield screen in solvent (C94) and allow to air dry.

4-65. Inspection — Engine Air Shield Screen Assembly. *a.* Check engine air shield screen for breaks, tears, corrosion, or other damage. Replace a damaged unit.

b. Replace a teflon coated screen if coating is chipping or peeling.

4-66. Installation — Engine Air Shield Screen Assembly.

CAUTION

Prior to removal of engine air inlet cover for screen installation, check the entire plenum area for any foreign material that could fall into engine. The engine can be badly damaged by the entry of such objects.

a. Remove taped cover from engine air inlet.

b. On aircraft without cold weather kit, position screen in plenum chamber and attach with five screws and washers.

c. On aircraft with cold weather kit, position forward teflon coated screen in plenum chamber and install three forward screws with washers. Position aft screen and secure with four screws and washers. The two center screws attach both screens.

d. Inspect plenum inlet, base of mast, and entire plenum chamber for foreign material.

e. Close plenum chamber access door. On aircraft with inertial particle-separator air filter, close filter bypass door.

4-67. ENGINE AIR INLET BELL AND ENGINE FIREWALL SEALS.

4-68. Description — Engine Air Inlet Bell and Engine Firewall Seals. The engine air inlet bell (fig. 4-11) is a molded nylon assembly with the bell and mounting flange formed as a unit. The engine air inlet bell provides a method of collecting and directing air from the plenum chamber to the engine compressor. The engine-to-firewall seal is a stamped steel assembly incorporating three flat springs, riveted to the inside diameter, for mounting to the engine air inlet bell mounting flange. The engine firewall seal seals the engine compressor to the plenum chamber. The engine air inlet firewall seal is a high temperature, all synthetic seal. The engine air inlet firewall seal provides a pliable surface for the rim of the engine-to-firewall seal to press against when the engine is installed. The engine shaft firewall seal assembly is a stamped steel backing with a seal that is bonded to the cupped diameter of the backing.

4-69. Removal — Engine Air Inlet Bell and Engine-to-Firewall Seal. *a.* Remove engine (para 4-10).

b. Remove five screws, washers, and nuts (fig. 4-11).

c. Remove engine air inlet bell.

d. Pull engine-to-firewall seal from bell.

4-70. Removal — Engine Air Inlet Firewall Seal. *a.* Remove engine (para 4-10).

b. Remove engine air inlet firewall seal (fig. 4-11).

c. Remove residue from firewall by using methyl ethyl ketone (C69); carefully scrape firewall as necessary.

4-71. Removal — Engine Shaft Firewall Seal. *a.* Remove engine (para 4-10).

b. Remove three bolts, six washers and three nuts.

c. Remove engine shaft firewall seal and backing (fig. 4-11).

4-72. Inspection — Engine Air Inlet Bell and Engine Firewall Seals. *a.* Check engine air inlet bell for cracks, breaks, or other damage.

b. Check engine-to-firewall seal for weak springs, cracks, breaks, dents, corrosion, or other damage.

c. Check engine shaft firewall seal backing for cracks, breaks, dents, corrosion, or other damage.

d. Check engine shaft firewall seal for brittleness, breaks or other damage.

e. Check engine air inlet firewall seal for cracks, breaks, dents, or excessive damage.

f. Replace a defective engine air inlet bell, engine-to-firewall seal, engine air inlet firewall seal, and engine shaft firewall seal.

4-73. Repair — Engine Air Inlet Bell and Engine Firewall Seals. *a.* To repair engine-to-firewall seal, replace any damaged or broken springs.

b. To repair engine shaft firewall seal, replace seal. Bond new seal to backing with adhesive (C9).

c. Repair of the engine air inlet bell is not authorized.

4-74. Installation — Engine Air Inlet Bell and Engine-to-Firewall Seal. *a.* Attach engine-to-firewall seal to engine air inlet bell by lifting leaf springs on seal and positioning over bell flange.

b. Place engine air inlet bell and engine-to-firewall seal into position on compressor inlet mounting flange and align bolt holes in engine air inlet bell mounting

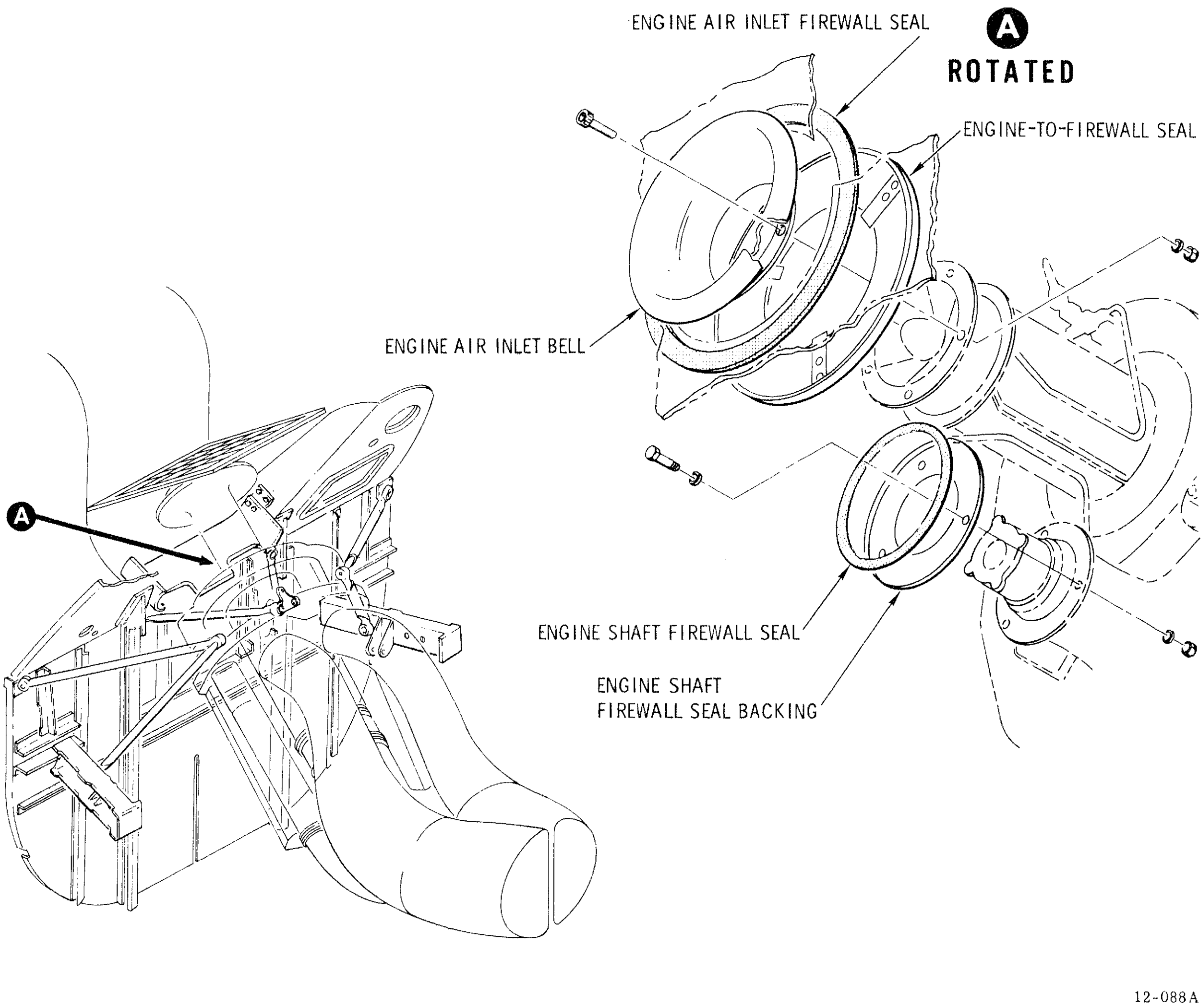


Figure 4-11. Air Induction System Components.

flange with bolt holes in mounting flange of compressor air inlet.

c. Install five screws, washers, and nuts; tighten nuts to secure.

d. Install engine (para 4-16).

4-75. Installation — Engine Air Inlet Firewall Seal. *a.* Using adhesive (C7), bond engine air inlet firewall seal into place on firewall.

b. Install engine (para 4-16).

4-76. Installation — Engine Shaft Firewall Seal Assembly. *a.* Place engine shaft firewall seal assembly into position and align holes with mounting holes in flange of overrunning clutch.

b. Install three bolts, six washers, and three nuts; tighten nuts to secure.

4-77. ENGINE BARRIER FILTER.

4-78. Description — Engine Barrier Filter. The barrier filter assembly consists primarily of a filter element, plastic filter element frame, and a metal bypass door and frame assembly. (See fig. 4-7, sh 1.) The filter element is secured in place by front and rear retainers on the filter assembly. Two configurations of filter element retainers are installed on the basic filter assembly. The filter frame is secured to the top of the air inlet installation above the plenum chamber by mounting bolts on the sides and rivets at the front and rear. The bypass door and frame assembly forms the front portion of the filter assembly and is separable from the filter frame. The bypass door frame contains a lever mechanism to allow opening of the bypass door when the bypass air control release handle is pulled. The bypass frame is hinge-mounted to the fuselage structure and can be folded forward for access to the engine air inlet access door. The filter assembly accommodates a pressure sensing switch and the tail rotor control rod at stations 113.22 and 137.50. Rubber boots are installed over the control rod where it enters and exits the filter frame to provide plenum chamber sealing.

4-79. Inspection — Engine Barrier Filter. *a.* Remove plenum chamber access door.

b. Remove engine air inlet front fairings (chapter 2).

c. Check that filter element is not clogged or contaminated. Check element for cuts, breaks, and for damaged areas. Check that filter element is properly installed and that no openings exist around sides of filter frame.

d. Check bypass air control release cable for security of attachment to bypass door lever. Check bypass door for operation.

4-80. ENGINE BARRIER FILTER ELEMENT.

4-81. Description — Engine Barrier Filter Element. The filter element is constructed of pleated fiberglass screen covering and an inner and outer filter medium. The element is held in place by retainers located fore and aft on the filter assembly. The filter element used on aircraft equipped with a cold weather kit has an aluminum moisture diverter lock wired to the bottom of each of the five pleats. (See fig. 4-7, sh 1.)

CAUTION

Install protective cover in the plenum chamber to prevent foreign objects from entering engine air inlet.

4-82. Removal — Engine Barrier Filter Element. *a.* Remove plenum chamber access door (chapter 2).

b. Release filter element retainers on forward and aft ends of filter assembly to free element. (See fig. 4-7.)

c. Move filter element forward to compress pleats and rotate through access door.

4-83. Cleaning — Engine Barrier Filter Element. *a.* If clean low pressure compressed air is available, direct air through filter element underside. To prevent damaging the element material, do not place hose nozzle directly against filter element. Continue to direct air through element until clean.

b. If no compressed air is available, turn element upside down and tap against a flat or inclined surface. Use care to avoid damaging the element corners.

c. If washing facilities are available, clean as follows: Wash in warm water and detergent (C35). Rinse in clean water flushing the filter from the underside out. Shake out and allow to air dry.

4-84. Inspection — Engine Barrier Filter Element. *a.* Inspect elements for cuts, holes, breaks and sagging. Check element fiberglass covering for security and for unraveling. Replace the element for any such damage or deterioration.

b. On cold weather kit elements, check that diverters are secure. If required, tighten or replace 0.032-inch lockwire (C57) and reseal wire holes with adhesive (C8).

4-85. Rework — Engine Barrier Filter Element. On aircraft equipped with a cold weather kit, replacement filter elements must have drain diverters installed. If required, rework filter elements as shown in figure 4-8, sheet 1.

NOTE

Modified filter is usable with summer and winter kits.

4-86. Installation — Engine Barrier Filter Element. *a.* Check for drain diverters used on aircraft equipped with a cold weather kit (para 4-85).

b. Clean exposed areas of filter frame with a cloth dampened by solvent (C94).

c. Compress filter element and rotate through plenum chamber access door and install in filter frame. Ensure that the installed element forms a tight seal on all edges (chapter 2).

d. Close plenum chamber access door. (See chapter 2.)

4-87. BARRIER FILTER PRESSURE SWITCH.

4-88. Description — Barrier Filter Pressure Switch. The barrier filter pressure switch is located on the side of the barrier filter frame and is used to sense engine air inlet pressure on the downstream side of the filter element. (See fig. 4-7, sh 1.) The switch housing contains a plenum chamber sensing port and an atmosphere sensing port. When a pressure differential is sensed, the switch closes and actuates the BYPASS AIR caution light. The switch is designed to actuate at a pressure of 3.0 ±0.25 in. water.

4-89. Inspection — Barrier Filter Pressure Switch. *a.* Check switch housing for cracks and breaks. Check condition of electrical leads and for damaged ground terminal.

b. Check that switch housing filter is not clogged or damaged. Check that switch plenum chamber sensing port is not clogged or damaged.

Table 4-9. Premaintenance Requirements for Testing Barrier Filter Pressure Switch.

Conditions	Requirements
Support Equipment	(S13) with a suitable vacuum source.
	Multimeter (T4)

4-90. Troubleshooting — Barrier Filter Pressure Switch. Refer to table 4-10.

4-91. Test — Barrier Filter Pressure Switch. The following test procedure may be performed with the pressure switch either removed or installed.

a. Attach a manometer pressure hose to the pressure-sensing port of the pressure switch.

b. If the switch is being tested in the installed condition, set the power selector switch at BATT to apply power to the BYPASS AIR indicator light circuit. If the switch is being bench tested, attach the test leads of a multimeter (T4) or similar continuity tester to the wire terminals of the two electrical leads extending from the switch. At this point the switch contacts should be open (BYPASS AIR indicator lamp out, or no continuity).

c. Slowly apply low-vacuum pressure to the switch pressure-sensing port while observing the manometer and BYPASS AIR indicator light or continuity tester. **THE SWITCH CONTACTS SHOULD CLOSE WHEN THE VACUUM REACHES 3.0 ±0.25 INCHES WATER** and should open when vacuum pressure is released. Repeat the check several times to verify consistent switch operation. Failure to operate consistently within the prescribed pressure range indicates a faulty switch and is cause for replacement.

4-92. Removal — Barrier Filter Pressure Switch. *a.* Set power selector switch at OFF and disconnect external power.

b. Remove engine air inlet front fairing and plenum chamber access door (chapter 2).

c. Remove filter element from barrier filter frame (para 4-80).

Table 4-10. Troubleshooting of the Filter Bypass Air Indicator System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **BYPASS AIR caution light erratic or inoperative.**

STEP 1. Check for defective bypass air caution light (chapter 9).

If the caution light is defective, replace caution light.

STEP 2. Check for loose, broken, shorted, or defective electrical connectors or wiring at pressure switch and caution light. Check wiring harness between pressure switch and caution light.

If electrical connectors or wiring are found to be defective, repair or replace defective connector or wiring (chapter 9).

STEP 3. Check for defective pressure switch at barrier filter (para 4-87).

If the pressure switch is found defective, replace pressure switch.

2. **BYPASS AIR caution light illuminated when filter is not clogged.**

STEP 1. Check for a short circuit in caution light to pressure switch wiring.

If a short circuit is found, repair electrical wire(s) as necessary (chapter 9).

STEP 2. Check for defective pressure switch.

If pressure switch is found to be defective, replace pressure switch (para 4-87).

CAUTION

Install protective cover in the plenum chamber to prevent foreign objects from entering engine air inlet.

d. Disconnect pressure switch knife splice lead and ground terminal connection.

e. Remove nut with O-ring and remove switch from filter frame (fig. 4-7).

4-93. Installation — Barrier Filter Pressure Switch. *a.* Position pressure switch in filter frame and install O-ring and mounting nut.

b. Connect pressure switch knife splice lead and ground terminal connnection.

CAUTION

Remove protective cover from plenum chamber using care to prevent entry of foreign materials.

c. Install filter element in filter frame (para 4-86).

d. Install engine air inlet front fairing and plenum chamber access door (chapter 2).

4-94. BARRIER FILTER BYPASS AIR CONTROL INSTALLATION.

4-95. Description — Barrier Filter Bypass Air Control Installation. The bypass air control installation consists of a cable and housing assembly with a pull handle (fig. 4-12). The cable end is attached to the bypass door release lever at the filter frame assembly. The BYPASS AIR CONTROL release handle is located overhead in the pilot's compartment near the cabin heat and engine anti-icer control knobs. Clamps secure the cable housing where routed through the fuselage structure.

4-96. Removal — Barrier Filter Bypass Air Control Installation. (See fig. 4-12.) *a.* Remove engine air inlet right front fairing (chapter 2).

b. Disconnect bypass air control cable from bypass door release lever.

c. Remove clamps and attaching hardware that secure cable housing to fuselage structure. Unfasten straps that secure cable housing in pilot's compartment.

d. Remove the two screws that secure release assembly to canopy frame. Pull release assembly through grommet and remove from aircraft.

4-97. Inspection — Barrier Filter Bypass Air Control Installation. *a.* Check release cable and housing for kinking and cuts. Check release cable pull handle for free movement within cable housing.

b. Check grommet for condition.

4-98. Installation — Barrier Filter Bypass Air Control Installation. (See fig. 4-12.) *a.* Thread release cable and housing assembly through grommet.

b. Install release assembly handle mounting bracket to canopy frame and secure with two screws.

c. Install clamps to secure cable housing to fuselage structure. Fasten straps to secure cable housing overhead in pilot's compartment.

d. Pull cable aft to remove all slack. Secure thimble and spacer with screw, two washers, and nut. Overlap and crimp cable end to cable with a sleeve.

e. Pull BYPASS AIR CONTROL handle and check that filter bypass door opens.

f. Close bypass door and install engine air inlet right front fairing (chapter 2).

4-99. BARRIER FILTER FRAME ASSEMBLY.

4-100. Removal — Barrier Filter Frame Assembly. *a.* Set power selector switch at OFF and disconnect external power.

b. Remove engine air inlet from fairing (chapter 2).

c. Remove filter element (para 4-82).

CAUTION

Install protective cover in the plenum chamber to prevent foreign objects from entering the engine air inlet.

d. Remove main transmission cover (chapter 2).

e. Disconnect pressure switch knife splice connection and ground terminal connection (fig. 4-7, sh 1).

f. Remove tail rotor control rod (chapter 11).

g. Unfasten the two studs that lock the bypass door frame to fuselage structure. Separate bypass door frame from filter frame and fold forward. Remove eight nuts, washers, spacers, and bolts that secure filter frame. Drill out rivets that secure the front and rear of filter frame. Remove sealant from inlet mounting surfaces if necessary to free filter frame. Remove filter frame from air inlet installation.

h. To remove filter bypass door and frame, drill out rivets that secure hinge to fuselage structure. Disconnect bypass air control cable from bypass door lever. Remove filter bypass door and frame assembly with hinge shim.

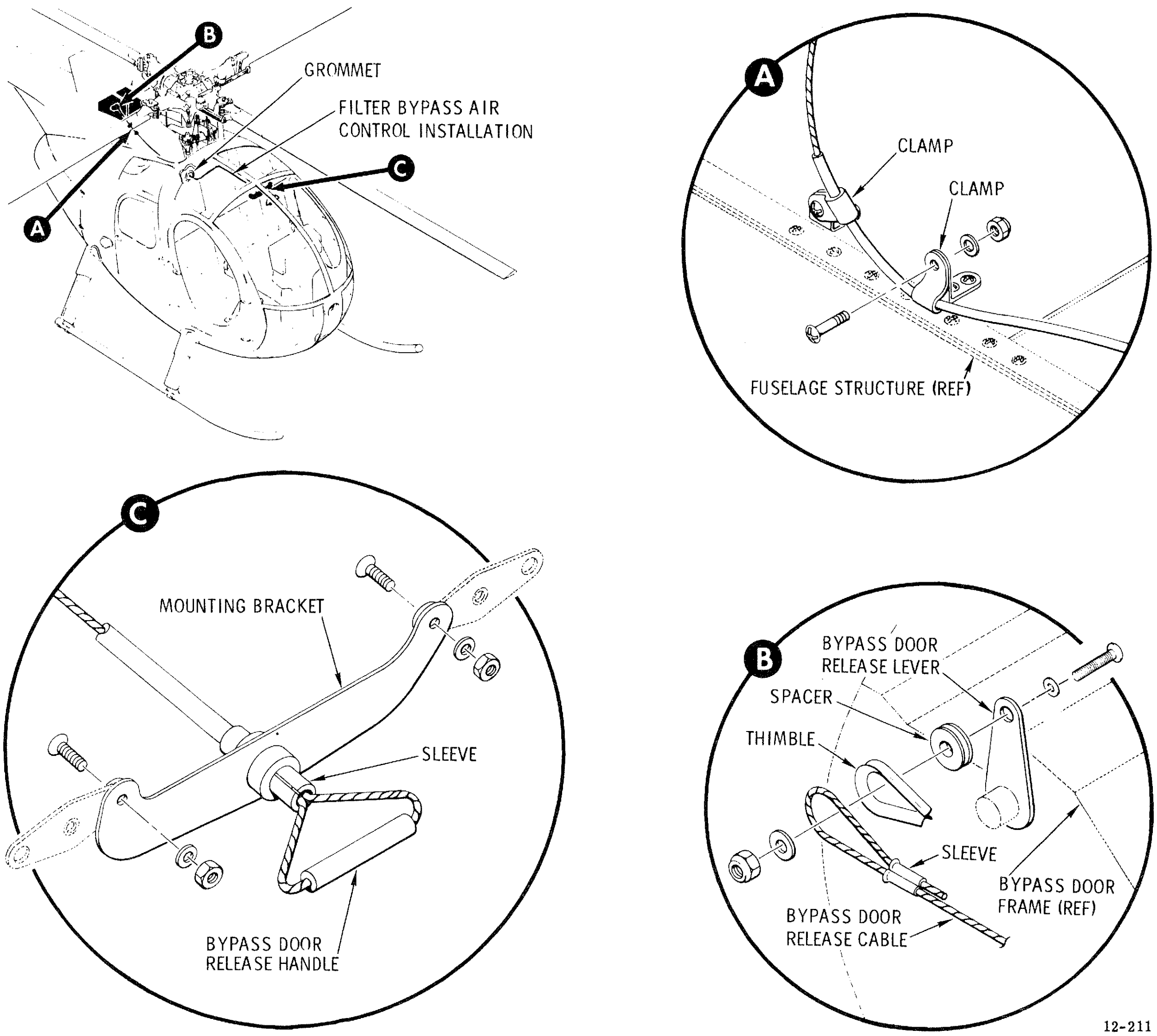


Figure 4-12. Barrier Filter Bypass Air Control Installation.

4-101. Cleaning — Barrier Filter Frame Assembly. *a.* Remove heavy dirt accumulations from frame assembly with a bristle brush.

b. Clean surfaces of frame assembly with a soft cloth dampened with solvent (C94).

4-102. Inspection — Barrier Filter Frame Assembly. (See fig. 4-7, sh 1.) *a.* Check plastic filter frame for cracks, holes and distortion. Check for cracks and breaks in frame at mounting holes.

b. Check extension spring and sleeving of element aft retainer for condition and security. When filter assembly with hinged retainers is installed, check aft hinge retainer, spring, and sleeving for condition and security.

c. Check bypass door and frame assembly for corrosion, dents, and distortion. Check bypass door release lever for security and operation.

d. Check the element retainer on bypass door frame for secure attachment and operation. When filter assembly with hinged element retainer is installed, check hinged element retainer, spring, and sleeving for condition.

4-103. Repair or Replacement — Barrier Filter Frame Assembly. *a.* For patch repairs to filter frame, refer to chapter 2.

b. Replace broken element retainer extension springs and defective sleeving.

c. Replace missing or broken locking studs on bypass door frame.

4-104. Installation — Barrier Filter Frame Assembly. (See fig. 4-7, sh 1.) *a.* Position filter frame over plenum chamber and align with existing bolt and rivet holes. Install eight nuts, washers, spacers, and bolts. Rivet front and aft portions of filter frame in place with MS20600AD4W replacement rivets.

b. Position bypass door and frame assembly with shim in place and align with existing rivet holes. Rivet frame assembly in place with MS20600AD4W replacement rivets. Insert filter bypass frame in filter frame and secure with two locking studs.

c. Install tail rotor control rod and boot (chapter 11).

d. Install bypass door release cable on bypass door release lever.

e. Connect pressure switch knife splice and ground terminal connections.

f. Install main transmission cover (chapter 2).

CAUTION

Remove protective cover from plenum chamber.

g. Install filter element (para 4-86).

h. Seal all openings at base of the inlet portion of frame assembly with sealant (C89).

i. Seal all openings between the inlet portion of frame assembly sides and the plenum chamber with tape (C106). Check that tape is applied externally and cannot enter engine inlet.

j. Install engine air inlet front fairing (chapter 2).

4-105. INERTIAL PARTICLE SEPARATOR AIR FILTER.

4-106. Description — Inertial Particle Separator Air Filter. The air filter assembly consists of a series of screened tubes contained within the particle-separator envelope, a scavenge air ejector nozzle manifold, and an ejector tube. (See fig. 4-13.) The filter separator is shaped to fit the front portion of the aft fairing and is secured to the fairing with three mounting bolts. Each of the filter tubes contained in the separator envelope consists of a filter screen, an air swirl guide, and an opening to the separator section. The ejector nozzle manifold and ejector tube are mounted aft of the filter separator exit boss. The nozzle manifold contains a bleed air fitting connection and seven jets for the bleed air scavenging system. The ejector tube consists of a mounting flange with seven tubes and seven hinged, air-operated flapper valves. Contaminants are directed overboard through filter duct and a screened opening in the left side of the aft fairing. On aircraft equipped with a cold weather kit, a drain hole is provided at the bottom of the assembly. When the filter duct is removed from the filter assembly, a protective cover may be installed in the opening in the left side of the aft fairing.

4-107. Removal — Inertial Particle Separator Air Filter. *a.* Remove engine air inlet front fairings (chapter 2).

b. Open filter bypass door.

CAUTION

Install protective cover in the plenum chamber to prevent foreign objects from entering engine air inlet.

NOTE

Prior to removing ejector tube from nozzle, paint a mark across flange of ejector and nozzle manifold to aid repositioning of ejector tube flapper valves during installation.

c. (See fig. 4-13.) Remove three bolts that secure separator to aft fairing. Remove six bolts that secure ejector tube and nozzle manifold to separator. Tip the top of separator outward and remove from aft fairing.

d. Remove O-ring from ejector and slide onto sleeve. Disconnect bleed air tube from nozzle manifold. Remove ejector tube and nozzle manifold from aft fairing.

e. Remove aft O-ring from sleeve and slide onto the filter duct. Remove sleeve from duct.

f. Remove the duct and screen by removing the eight screws and washers that secure them to the aft inlet fairing.

4-108. Inspection — Inertial Particle Separator Air Filter. *a.* Check filter screens for damage and secure attachment. Check air swirl guides for damage and for broken or missing vanes. Up to five damaged swirl guide tubes may be blocked off *e* below. Inspect particle separator openings in filter tubes for clogging.

b. Check separator for cracks, holes and distortion. Evidence of cracking may require realignment of the

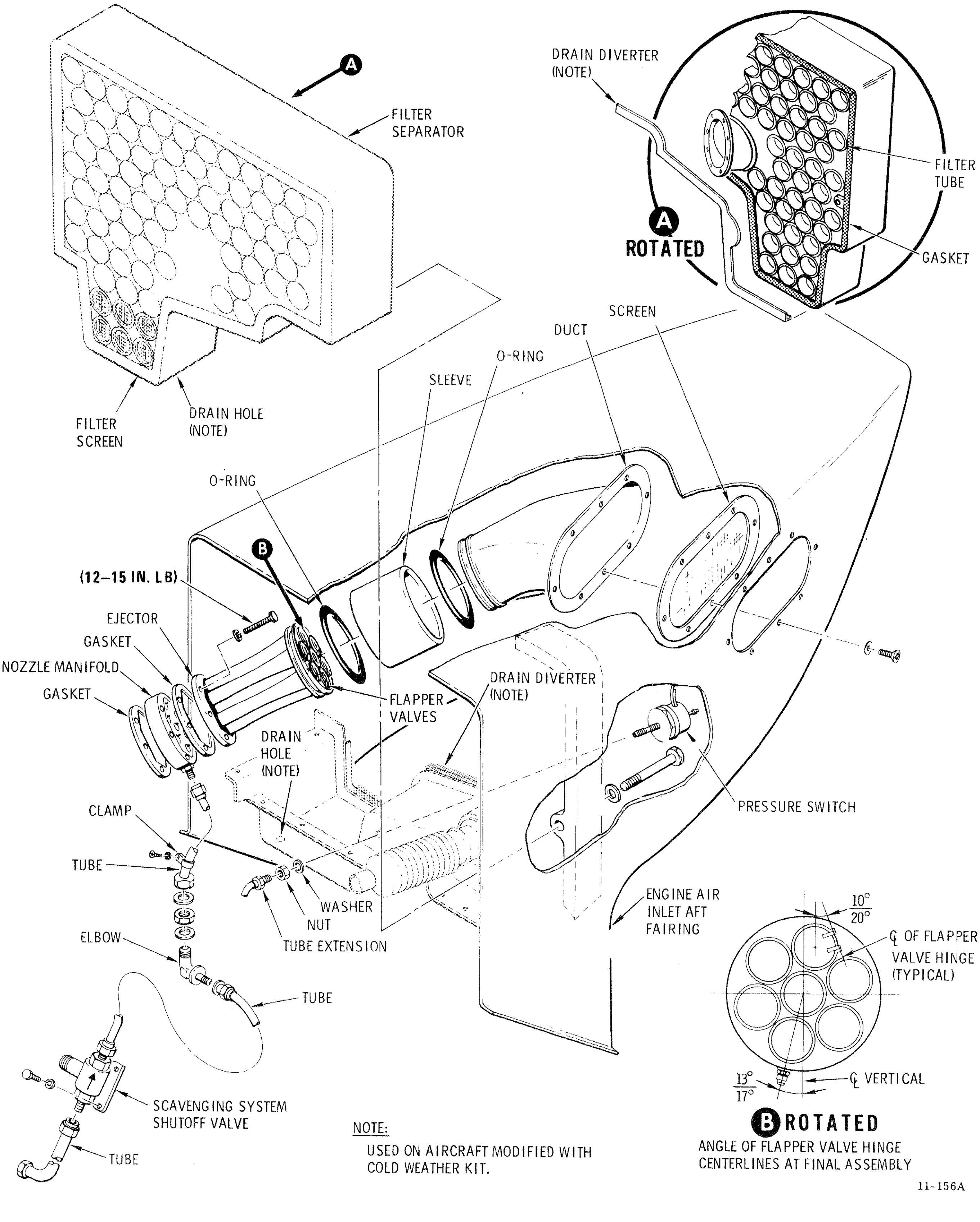


Figure 4-13. Inertial Particle-Separator Air Filter Assembly.

filter. Check that gasket installed on edge of separator is securely attached and not damaged.

c. Check bypass air control installation for security and bypass door for operation. Check hinges and pulleys for loose or missing hardware. Check door gasket for security of attachment and for cuts and deterioration.

d. Check rubber sleeving and O-rings for cuts, holes, and deterioration.

e. Check ejector tube for cuts, breaks, and distortion. Check condition of flange area. Check condition of flapper valves and that valves swing freely on hinges. Ejectors with missing flapper valves may be continued in operation until replacements are installed.

f. Check nozzle manifold for breaks, cracks, and distortion. Check fitting for crossed or stripped threads. Check that the seven nozzles are not plugged, bent, or damaged.

g. Check that air filter overboard vent screen is not damaged or clogged.

4-109. Cleaning — Inertial Particle Separator Air Filter. *a.* Remove filter tube screens and clean with a soft brush to remove dirt accumulations.

b. Immerse separator in a solution of detergent soap (C35), and allow to soak approximately 15 minutes. Flush out with clear water. Allow filter assembly to drain and air-dry thoroughly. Reinstall filter tube screens.

c. Clean nozzle manifold with high pressure compressed air. Use care to prevent damaging the nozzles.

4-110. Alignment and Surface Flatness Repair — Separator and Duct. (See fig. 4-13.) *a.* Remove the filter separator.

b. Using plywood or phenolic sheet of 0.5-inch minimum thickness, locally manufacture a flat template surface. Use the filter assembly as a guide to draw and then cut out the template.

c. Position the template against the engine air inlet aft fairing-to-filter mounting surface and install three bolts and nuts. Tighten fingertight so that template is held firmly in place without twisting.

d. Use a thickness gage and check for gap(s) at the three mounting bolt attach points. **GAP SHOULD NOT EXCEED 0.030 INCH.**

e. Eliminate excessive gap by bonding an equivalent thickness shim at engine air inlet fairing bolt attach point. Shim may consist of aluminum sheet stock bonded to the fairing with catalyzed resin (C39) or layers of fiberglass cloth installed according to fiberglass repair procedures, chapter 2.

f. Reinstall the filter separator.

g. With the rubber sleeve removed, use a straightedge and check for misalignment between the ejector and ejector overboard duct. **MAXIMUM MISALIGNMENT IS 0.25 INCH.**

h. If duct and ejector are misaligned, install tapered shim(s) between the duct and screen rubber rim picking up at least two attachment screws.

NOTE

Existing screen and duct attachment screws (MS27039C1-11) are long enough to allow a 0.25-inch-thick shim addition.

i. Reinstall the duct, screen and shim(s) with several screws and recheck duct for misalignment. If alignment is correct, *g* above, reinstall sleeve and continue with installation.

4-111. Repair — Air Filter Separator. Repairs to the air filter separator are accomplished using standard laminated fiberglass methods and fiberglass repair kit (C39) as described in chapter 2. Take care that no structural interference or potential foreign object damage to the engine can occur as a result of filter repair.

a. Rebond loose or missing fiberglass attachment bolt spacers.

b. Rebond seams that join the front and back walls to the center section.

c. Rebond manifold nozzle attachment tube.

d. Patch repair or fill any other damaged or cracked areas.

e. Replace missing or damaged guide tubes by pulling loose and rebonding in place. Up to five swirl guide tubes may be temporarily blocked out by installing an aluminum patch, attached with a minimum of three mechanically expanded rivets, over the aft side of the swirl guide hole.

f. Repair or replace worn or loose gasket on separator. For gasket bonding information, refer to general use and application of nonstructural bonding adhesives (chapter 2).

4-112. Rework — Filter Separator. On aircraft equipped with a cold weather kit, replacement filter separators must have a lower drain hole. If required drill hole as shown in figure 4-8, sheet 2.

4-113. Installation — Inertial Particle Separator Air Filter. (See fig. 4-13.) *a.* Align mounting holes of duct and screen. Secure duct and screen to aft fairing with eight mounting screws and washers. **SCREW TORQUE NOT TO EXCEED 11 INCH-POUNDS.**

b. Check filter separator lower section for a drain hole used on aircraft equipped with a cold weather kit (para 4-112).

c. Install filter separator into front of aft fairing and align mounting holes. Apply a coat of anti-seize compound (C14) to bolt threads. Secure filter separator with three mounting bolts and washers; **TORQUE BOLTS TO 60-80 INCH-POUNDS.**

d. Preassemble sleeve and one O-ring on the grooved end of the ejector. Install one O-ring on end of installed duct.

e. Position nozzle manifold in place and connect bleed air tube to nozzle manifold fitting. Apply a coat of anti-seize compound (C14) to fitting threads before installation.

f. Install ejector in place using serviceable gaskets. Position ejector so paint stripe marks and bolts in nozzle manifold align.

NOTE

To aid installation when replacement ejector is used, position ejector tubes so flapper valves are as shown in figure 4-13 and mark uppermost tube. Position flapper valves as shown for final assembly.

g. Apply a coating of anti-seize compound (C14) to ejector mounting bolts. Secure ejector and nozzle to separator with six bolts and washers. **TORQUE BOLTS TO 12-15 INCH-POUNDS.**

h. Slide sleeve over the end of the installed duct and secure with O-rings.

CAUTION

Remove protective covering installed in plenum chamber.

i. Close filter bypass door.

j. Install engine air inlet front fairings (chapter 2).

4-114. BYPASS AIR CONTROL INSTALLATION.

4-115. Description — Bypass Air Control Installation. The bypass air control installation for the inertial particle-separator air filter includes the filter bypass door, bypass air control release, a door operating cable and stop, and a series of pulleys installed on the inside of the aft fairing. (See fig. 4-14, sh 1 and 2.) The BYPASS AIR CONTROL release handle is used to unlatch and open the bypass door and is located overhead in the pilot's compartment adjacent to the cabin heat and engine anti-icer control knobs. A cable connector located at station 130.00 joins the release cable to the door operating cable and provides a disconnect point.

4-116. Removal — Bypass Air Control Installation. *a.* Remove main transmission cover (chapter 2).

b. Remove engine air inlet left front fairing (chapter 2).

CAUTION

Install protective cover in the plenum chamber to prevent foreign objects from entering engine air inlet.

c. (See fig. 4-14., sh 1.) Disconnect the bypass door release assembly cable from the bypass door operating cable at the connector. Remove connector from installation.

d. Remove screws that secure release assembly bracket overhead in the pilot's compartment. Remove clamps and straps as necessary to free release assembly cable housing. Pull release assembly cable forward through grommets in structure and remove from aircraft.

e. Remove sleeve that secures cable to bypass door cable control crank (detail C, sh 2). Remove door operating cable with cable stop from pulleys and remove from fairing.

4-117. Inspection — Bypass Air Control Installation. *a.* Check door operating cable for breaks, cuts, and kinks. Check that cable stop terminals are swaged and secure. Check cable stop for damaged and stripped threads.

b. Check bypass door release assembly for breaks, cuts, and kinks in cable and cable housing. Check terminal stop on cable end for secure attachment.

c. Check door operating cable pulleys on aft fairing for rotation and secure attaching hardware.

CAUTION

Loose, missing or improperly installed filter bypass door and latching mechanism hardware can cause foreign object damage to the engine. Perform a thorough inspection and accomplish repairs per paragraph 4-118.

d. Inspect filter bypass door hinges for secure attachment and hinge pins for wear or looseness.

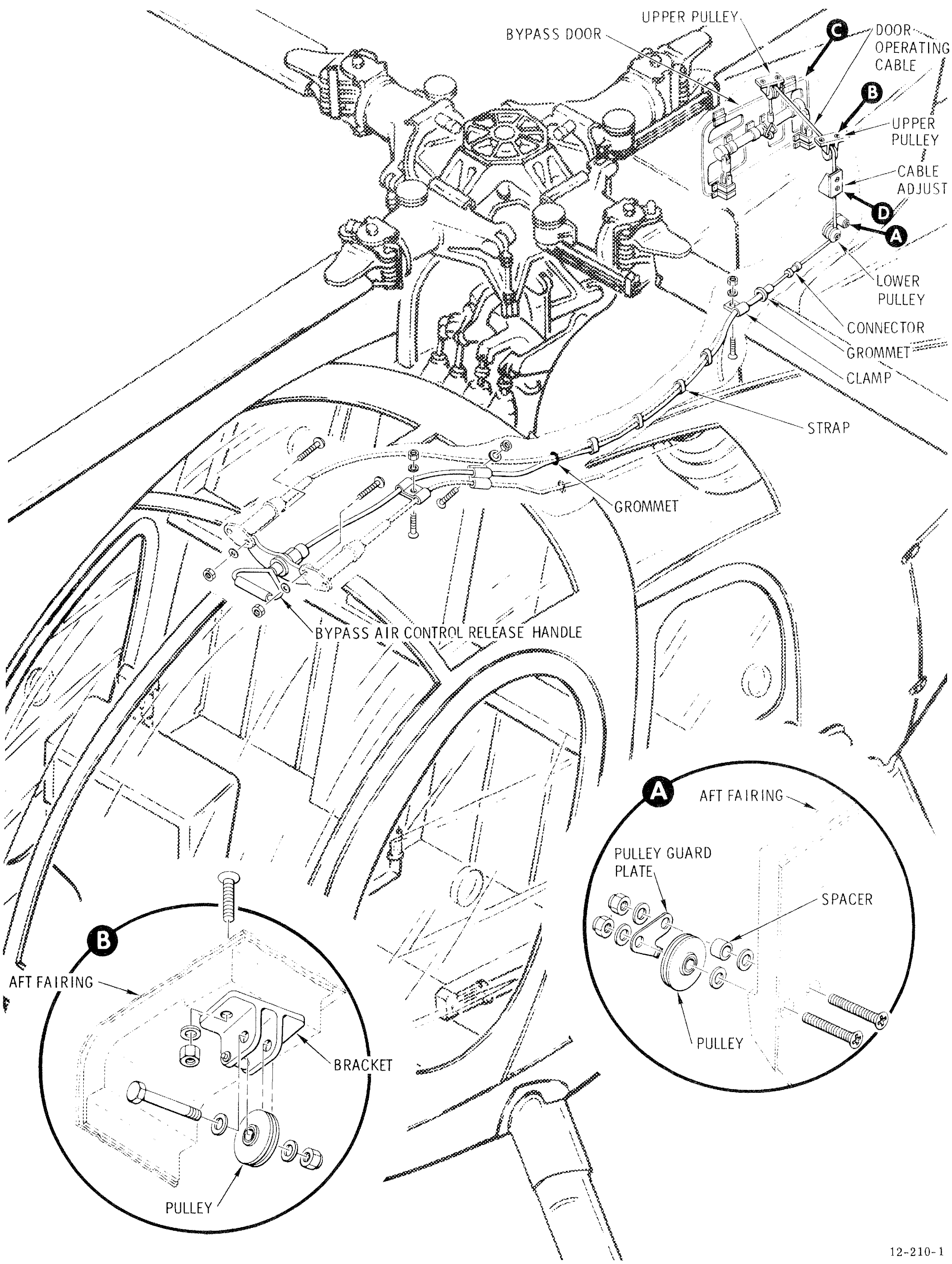


Figure 4-14. Bypass Air Control Installation, Inertial Particle Separator Air Filter. (sheet 1 of 2)

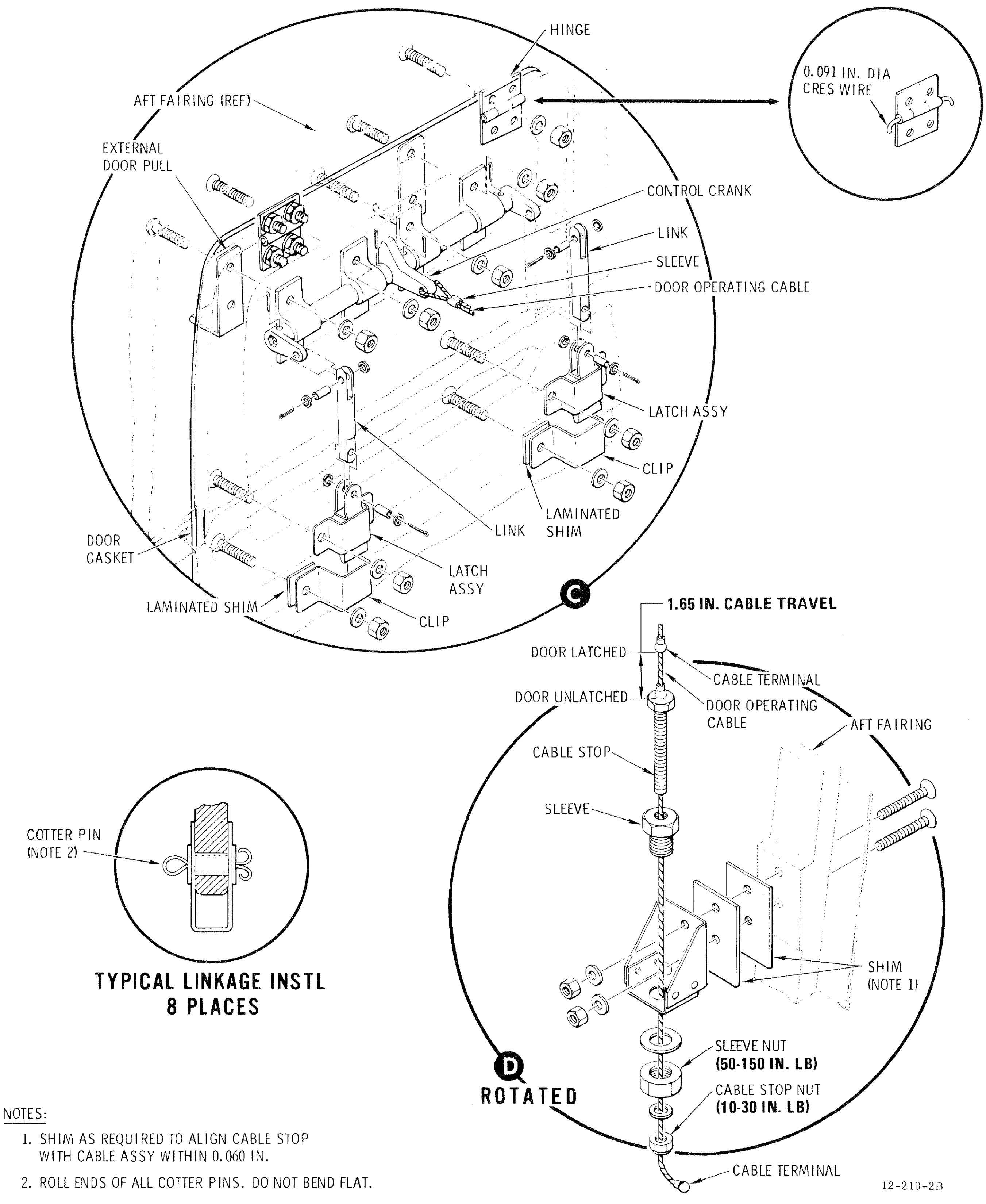


Figure 4-14. Bypass Air Control Installation, Inertial Particle Separator Air Filter. (sheet 2 of 2)

e. Check for loose or missing door control crank and latching mechanism attaching hardware.

f. Check that all eight door mechanism cotter pins are rolled as shown in figure 4-14, sheet 2.

g. Check bypass door control crank and latching mechanism for operation and door for firm closure.

4-118. Repair — Bypass Air Control Installation. *a.* Replace unrolled cotter pins with new cotter pins. Roll ends of pins as shown in figure 4-14, sheet 2.

b. Replace loose or missing door hinge pins with 0.091-inch-diameter corrosion resistant steel wire (C59) of suitable length. Bend over at both ends after installation as shown in figure 4-14, sheet 2.

c. Adjust door clip laminated shim thicknesses as required so that door latching mechanism engages clip and door is held firmly closed without binding when latched.

d. Spray filter bypass door gasket on aft fairing and bypass door edge with dry lubricant (C64).

4-119. Installation — Bypass Air Control Installation. (See fig. 4-14.) *a.* Mount bypass air control release assembly bracket to canopy frame and install two attaching screws, washers, and nuts.

b. Thread bypass air control release assembly housing and cable through grommets in fuselage structure (sh 1).

c. Install door operating cable assembly on aft fairing pulleys.

d. Assemble door operating cable stop, stop sleeve, and stop sleeve nut on fairing bracket (detail D, sh 2).

e. Install connector to join bypass air control release cable with door operating cable.

f. Connect end of door operating cable to bypass cable control crank. With door fully latched, install sleeve on cable overlap; check that cable overlap will allow a cable travel of approximately 1.65 inches and then crimp sleeve. (See detail D, sh 2.) With cable terminal at rest on cable stop, bypass door should be opened a minimum of 45 degrees. Make final adjustment of cable travel at cable stop and sleeve. When final adjustment of cable travel is made, **TORQUE SLEEVE NUT TO 50-150 INCH-POUNDS. TORQUE CABLE STOP NUT TO 10-30 INCH-POUNDS.**

g. Install clamps and fasten straps to secure release cable housing (sh 1).

CAUTION

Remove protective cover installed in plenum chamber using care to prevent entry of foreign material.

h. Install main transmission cover (chapter 2).

i. Install engine air inlet left front fairing (chapter 2).

4-120. AIR FILTER BYPASS DOOR.

4-121. Removal — Air Filter Bypass Door. *a.* Remove engine inlet front fairings (chapter 2).

b. Disconnect bypass door operating cable (para 4-116). Cable may be removed as part of the door or disconnected by removing the cable sleeve.

c. Remove hinge pins and remove door assembly.

4-122. Disassembly — Air Filter Bypass Door. Disassemble door latching mechanism as shown in figure 4-14, sheet 2. Disassemble only as required for repair or replacement of damaged part.

4-123. Inspection — Air Filter Bypass Door. *a.* Check latching mechanism and hinges (para 4-117).

b. Check fiberglass door for structural damage such as cracked or frayed glass cloth surfaces. Repair using fiberglass repair procedures described in chapter 2.

4-124. Assembly — Air Filter Bypass Door. Assemble door latching mechanism as shown in figure 4-14, sheet 2.

4-125. Installation — Air Filter Bypass Door. *a.* Install latching mechanism and hinges as shown in figure 4-14, sheet 2.

b. Position door on aft fairing and install 0.091-inch-diameter wire (C59) hinge pins through hinges. Bend wire over at both ends after installation as shown in figure 4-14, sheet 2.

c. Reconnect door operating cable and adjust control release mechanism (para 4-119).

4-126. AIR FILTER SCAVENGING SYSTEM SHUTOFF VALVE.

4-127. Description — Air Filter Scavenging System Shutoff Valve. The air filter scavenging system shutoff valve is mounted on the forward side of the aft canted bulkhead (sta 124.00). The valve controls the flow of engine bleed air to the air filter scavenging system. (See fig. 4-13.)

4-128. Removal — Air Filter Scavenging System Shutoff Valve. *a.* Set power selector switch at OFF and disconnect external power.

b. Disconnect electrical plug from valve receptacle.

c. Disconnect tubing from valve inlet and outlet ports.

d. Remove two bolts and washers that secure valve to bulkhead. Remove valve.

4-129. Cleaning and Flushing — Air Filter Scavenging System Shutoff Valve. *a.* Disconnect scavenging system shutoff valve inlet and outlet tubes (fig. 4-13).

b. Provide for water flushing of fitting by attaching suitable hoses to valve inlet and outlet ports. Do not allow water to enter the engine.

c. Connect external power source to aircraft and set power selector switch at EXT.

d. Actuate the SCAV AIR switch. Have an assistant cycle the switch several times while flushing the valve poppet seat with a stream of clear water.

NOTE

The valve may also be removed from the aircraft for flushing. If removed, actuate the valve, using a 24 vdc power source connected to the valve electrical connector pins.

e. Using high pressure compressed air, thoroughly dry out valve passage.

f. Reinstall valve and electrical connector, if removed, and connect inlet and outlet tubes.

4-130. Inspection — Air Filter Scavenging System Shutoff Valve. (See fig. 4-13.) *a.* Check valve for cracks and dents.

b. Check valve ports for damage and clogging; check fittings for stripped or crossed threads.

c. Check valve electrical receptacle for damage to pins.

4-131. Installation — Air Filter Scavenging System Shutoff Valve. (See fig. 4-13.) *a.* Apply a coating of antiseize compound (C14) to threads of valve mounting bolts. Position valve on aft bulkhead with flow arrow pointing up, and install mounting bolts with washers; **TORQUE BOLTS TO 22 - 27 INCH-POUNDS.**

b. Apply a coating of antiseize compound (C14) to threads of valve port fittings and connect tubes to shutoff valve.

NOTE

Before connecting wire harness to valve, check for moisture. Inspect O-ring seal in connector for condition; replace if necessary.

c. Connect electrical wiring harness to valve receptacle.

4-132. INERTIAL PARTICLE AIR FILTER PRESSURE SWITCH.

4-133. Description — Inertial Particle Air Filter Pressure Switch. The air filter pressure switch is located just below the filter separator envelope and is mounted to the filter sealing panel. (See fig. 4-13.) The switch is used to sense air inlet pressure on the outlet side of the air filter. The switch consists of a housing, a plenum chamber sensing port, and an atmosphere sensing port. A tube extension is installed on the atmosphere sensing port. The switch operates on 28 vdc and has two electrical leads. When a pressure differential is sensed, the switch closes and actuates the BYPASS AIR caution light on the annunciator panel.

4-134. Inspection — Inertial Particle Air Filter Pressure Switch. *a.* Check switch housing for cracks and breaks. Check electrical wiring for cuts, fraying, and condition.

b. Check that switch plenum chamber sensing port and atmosphere sensing port are not damaged or clogged.

4-135. Test — Inertial Particle Air Filter Pressure Switch. Test the air filter pressure switch according to paragraph 4-91 with the following exception: The switch contacts should close when the vacuum reaches 4.0 ±0.04 inches water and should open when the vacuum is decreased to 2.0 inches water.

4-136. Removal — Inertial Particle Air Filter Pressure Switch. *a.* Set power selector switch at OFF and disconnect external power.

b. Remove engine air inlet front fairing.

c. Open air filter bypass door.

CAUTION

Install protective cover in the plenum chamber to prevent foreign objects from entering engine air inlet.

d. Disconnect pressure switch electrical leads from wiring harness.

e. Disconnect tube extension from atmosphere sensing port and remove mounting nut and washer that secure switch to filter sealing panel. Remove switch from panel.

4-137. Installation — Inertial Particle Air Filter Pressure Switch. *a.* Position pressure switch in filter sealing panel and install washer and mounting nut. Connect tube extension to atmosphere sensing port.

b. Connect pressure switch electrical wiring.

CAUTION

Remove protective cover from plenum chamber.

c. Close air filter bypass door.

d. Install engine air inlet front fairing.

4-138. FILTER BYPASS AIR CAUTION LIGHT.

4-139. General — Filter Bypass Air Caution Light. Perform an operational check of the caution light pressure switch according to paragraph 4-87 with the following exception: The BYPASS AIR caution **LIGHT SHOULD ILLUMINATE WHEN THE VACUUM REACHES 4.0 ±0.4 IN. WATER AND SHOULD GO OUT WHEN THE VACUUM IS DECREASED TO 2.0 IN. WATER.**

4-140. ANTI-ICING SYSTEM.

4-141. Description — Anti-icing System. An anti-icing system (fig. 4-15) is provided only for the engine compressor inlet. The system consists of a manually operated push-pull control cable connected to an anti-icing valve lever located on the engine compressor scroll. The anti-icing shutoff valve controls the compressor discharge air routed to the compressor inlet and front bearing support hub. There is no electrical control nor intermediate heat control position.

4-142. Troubleshooting — Anti-icing System. Refer to table 4-11.

4-143. Inspection — Anti-icing System. *a.* Check anti-icing valve control lever and cable for secure attachment to canopy frame.

b. Check cable for kinks, crushed sleeve, corrosion, and secure attachment.

c. Open engine access door and check cable wire for secure attachment to valve lever adapter.

d. Check compressor discharge air lines connected to anti-icing air shutoff valve on engine for cracks and secure hex nuts. Close engine access door.

4-144. ANTI-ICING VALVE CONTROL LEVER AND CABLE.

4-145. Description — Anti-icing Valve Control Lever and Cable. The anti-icing valve control (fig. 4-15) consists of a flexible, enclosed, wire-type cable assembly bonded to a control housing. The control housing, with cable actuating knob, is attached to the overhead canopy frame. Movement of the control lever rearward to the latch notch opens the anti-icing valve completely. Releasing the control lever from the latch notch results in spring-loaded return of the lever to the forward (OFF) position. Control lever piston travel in the housing is approximately 1.50 inches from the open to closed positions. Cable routing is on the left side of the main rotor mast support structure alongside the heating system control valve cable and torquemeter oil pressure line as far as the plenum chamber. At this point the anti-icing cable is routed downward through the firewall into the engine compartment.

4-146. Operational Check — Anti-icing Valve Control Lever and Cable. *a.* Move control lever knob to full aft (anti-icing air valve open) position and latch it in housing notch.

b. Open engine access door.

c. Check that anti-icing valve lever is positioned at aft travel limit, and that cable wire does not appear to be distorted by overtravel.

d. Release control lever knob from housing notch; lever piston return spring should return lever knob to housing forward stop without manual assistance. If knob does not return, the valve lever adapter nut may be overtightened; refer to *g* below.

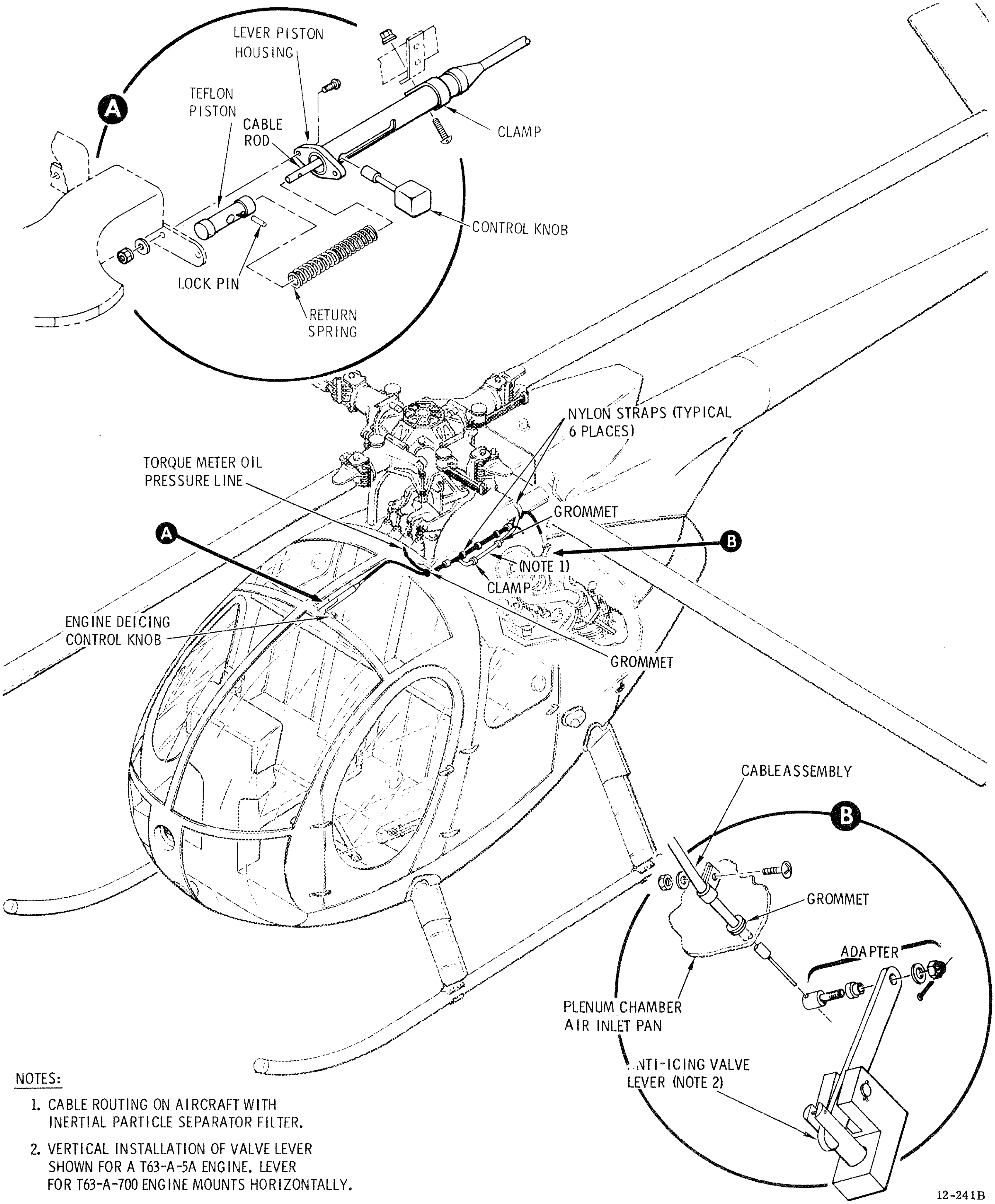
e. Recheck position of anti-icing valve lever; lever should be positioned at forward travel limit.

f. If anti-icing valve lever does not have sufficient travel or the cable appears to distort when actuated, adjust cable wire for correct lever stroke.

g. When control cable operation is stiff, isolate the trouble as follows. Remove the cable wire from the valve lever adapter and check for binding during movement of control lever to latch, release, and the spring-loaded return. Manually actuate the anti-icing valve lever to check freedom of movement. Reinstall cable wire and torque adapter nut to 10 inch-pounds maximum. Secure nut with new cotter pin.

h. Close engine access door.

4-147. Removal — Anti-icing Control Lever and Cable. (See fig. 4-15.) *a.* On aircraft without a filter, or with the barrier filter, remove plenum chamber access door and left half of engine air inlet forward fairing (chapter 2). Remove the filter element from barrier filter. On aircraft with an inertial particle separator filter, open the filter bypass door to gain access to plenum



NOTES:

1. CABLE ROUTING ON AIRCRAFT WITH INERTIAL PARTICLE SEPARATOR FILTER.
2. VERTICAL INSTALLATION OF VALVE LEVER SHOWN FOR A T63-A-5A ENGINE. LEVER FOR T63-A-700 ENGINE MOUNTS HORIZONTALLY.

Figure 4-15. Engine Anti-icing Valve Control Cable.

Table 4-11. Troubleshooting of the Anti-icing Control System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Activation of anti-icing air shutoff valve does not restore normal engine operation when compressor icing occurs.**

STEP 1. Check for a loose cable wire in anti-icing air valve lever adapter or adapter stud sheared or stripped.

If either defects are found, tighten or replace adapter with cable and valve in correct position (para 4-144).

STEP 2. Check for a defective or stuck anti-icing air valve.

If the valve is defective or stuck, replace the valve (TM 55-2840-231-24).

STEP 3. Check for lack of anti-icing air.

If the anti-icing air is insufficient, check engine air lines for leaks.

2. **Release of anti-icing valve control lever to OFF (fwd) position does not return anti-icing valve lever to forward position. (Inflight symptom is continuation of a 10% reduction in available power or torque.)**

STEP 1. Same as STEPS 1 and 2 above.

STEP 2. Check for a sheared cable to control lever piston retaining pin (inside lever housing) (fig. 4-15).

If the retaining pin is found to be sheared, replace damaged part(s).

STEP 3. Check for an overtightened adapter nut.

If the nut is found to be overtightened, loosen adapter nut and retorque to not more than 10 inch-pounds.

3. **Anti-icing valve control lever difficult to move.**

STEP 1. Check for control cable sleeve kinked or one or more routing bend radii less than a 3-inch minimum. Check cable operation with cable wire detached from shutoff valve lever adapter (para 4-144).

If the cable is found kinked, replace cable.

If sharp bends are found, remove sharp bends in cable routing.

chamber, and remove main transmission cover (chapter 2).

b. Remove two screws, washers, and nuts that attach cable lever piston housing to canopy structure.

c. Remove clamps and straps that attach cable to structure or adjacent cabling along its full length.

d. Loosen the nut of adapter at the anti-icing valve on the engine.

e. Remove control cable.

4-148. Disassembly — Anti-icing Control Lever and Cable. (See fig. 4-15.) The control assembly is normally replaced as a unit. However, disassembly to the extent shown on figure 4-15 can be accomplished for inspection purposes or lever mechanism parts replacement.

a. Move control knob to full extent of forward travel.

b. Pull control knob shaft out of piston.

c. Pull piston out of housing.

d. Remove lock pin from piston to free cable and piston.

4-149. Inspection — Anti-icing Valve Control Lever and Cable. *a.* Check cable for kinks, crushed sleeve and corrosion *b.* Check piston for wear or elongation of knob shaft or lock pin holes.

c. Check return spring for kinks or breaks.

d. Check valve lever adapter for excessive wear.

4-150. Repair — Anti-icing Valve Control Lever and Cable. No repairs are recommended except for replacement of lever mechanism parts.

4-151. Reassembly — Anti-icing Valve Control Lever and Cable. (See fig. 4-15.) *a.* Extend cable rod through housing so parts can be assembled.

b. Apply a thin film of grease (C46) on a serviceable spring. Slide spring on cable rod and into housing.

c. Fit serviceable piston on cable rod and secure with lockpin.

d. Push piston into housing until large hole in piston is aligned with large hole in end of housing slot.

e. Insert control knob rod through housing slot and into piston.

f. Move control knob to detent until housing is installed.

4-152. Installation — Anti-icing Valve Control Lever and Cable. (See fig. 4-15.) *a.* Route the cable assembly into position. Be sure the bulkhead grommets are in place.

b. Attach control lever piston housing to canopy structure bracket with two screws, washers, and nuts.

c. Install clamps and straps that attach cable to structure or adjacent cabling.

NOTE

The clamp nearest to the valve should not be tightened until after travel is checked.

d. Adjust cable wire in valve lever adapter and perform an operational check (para 4-146).

e. When assured that control will operate the valve through full range of travel, tighten the clamp nearest the valve.

f. Reinstall barrier filter, element, plenum chamber access door, main transmission cover, left half of engine air inlet forward fairing, and filter bypass door, as applicable (chapter 2).

SECTION V EXHAUST SYSTEM

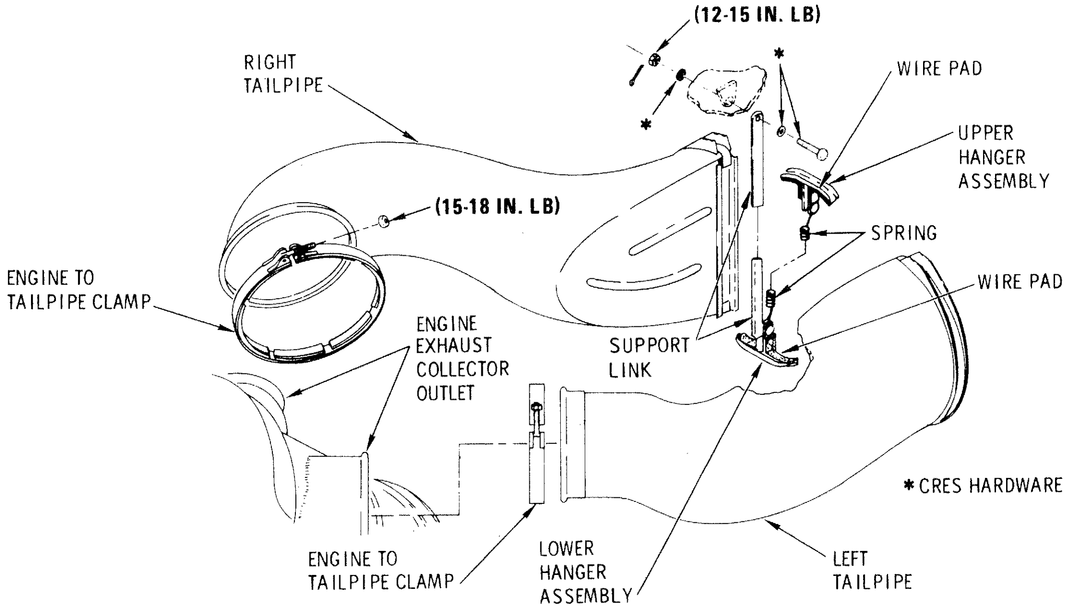
4-153. EXHAUST SYSTEM INSTALLATION.

4-154. Description — Exhaust System Installation. Two different types of exhaust systems are used. (See fig. 4-16.) The aft exhaust system directs exhaust gasses aft through the ends of the engine access doors. The system consists of two engine tailpipe clamp assemblies, right and left tailpipe assemblies, and cushioned support hangers. The engine tailpipe assembly is secured at the engine exhaust collector outlets by the tailpipe clamps, and supported at the exhaust opening by a cushioned hanger assembly. The lower hanger is suspended from a structure fitting above the tailpipes. The upper and lower hangers are spring-loaded together. Inconel wire mesh cushions and the joining spring provide flexibility between the aft ends of the tailpipes and the structure. The aft exhaust system is accessible by opening the two engine compartment access doors. The upward exhaust system consists of two stainless steel pipes which direct exhaust gasses upward from the engine through streamlined fiberglass fairings on each side of the fuselage above the engine access doors. The opening at the aft end of the engine access doors is closed by a cone shaped plug, constructed in halves, attached to each door. The upward exhaust pipes are attached to the engine with V-type clamps. No other support is required. The upper, exposed ends of the pipes are equipped with screens welded in place to prevent entry of foreign material. Fiberglass covers are also provided and should be used during maintenance activities and when the aircraft is moored (chapter 1). The following maintenance functions apply to either type exhaust system.

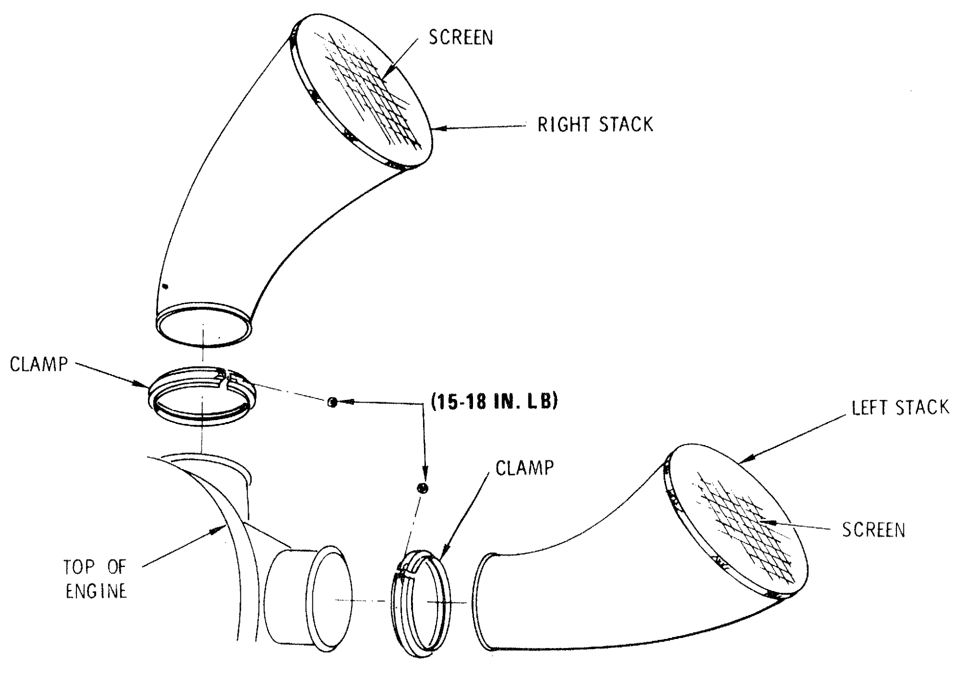
4-155. Test (Operational Check) — Exhaust System Installation. Ground run the engine for 5 minutes according to TM 55-1520-214-10; check exhaust system for leaks as evidenced by exhaust residue.

4-156. Removal — Aft Exhaust Tailpipes. (See fig 4-16.) *a.* Remove tailpipe clamp nuts and clamps. Have an assistant support the tailpipes until aft attachment is released.

b. Remove cotter pin, nut, washer, and bolt attaching lower tailpipe hanger to aircraft structure.



AFT EXHAUST SYSTEM



UPWARD EXHAUST SYSTEM

11-014C

Figure 4-16. Exhaust Systems.

CAUTION

Whenever tailpipes are removed, cover exhaust collectors with barrier material (C16) to prevent foreign objects from entering power turbine section of engine.

c. Remove left and right tailpipes as an assembly.

d. Remove spring, and upper and lower hangers; then separate the tailpipes.

4-157. Removal — Upward Exhaust Stacks. *a.* Remove stack clamp nuts and clamps (fig. 4-16).

b. Remove exhaust stacks by sliding upward through fuselage fairings.

4-158. Inspection — Exhaust System Installation. (See fig. 4-16.) *a.* Check the tailpipe support link (aft system) for cracks, breaks, corrosion or other damage, and secure attachment.

b. Check the engine tailpipes for large dents, holes, cracks, cracked or open seams, damaged flanges or fittings, corrosion or other damage, and secure attachment.

c. Check engine exhaust collector support for cracks (TM 55-2840-231-24).

d. Check upper and lower hanger assemblies (aft system) and spring for damage, corrosion and secure attachment.

e. Check visible portion of aft system inconel wire pads on hanger assemblies for wear, tears, and for secure spotwelds.

f. Check upward system protective screens and fuselage fairings for breaks or burning.

4-159. Repair — Exhaust System Installation. Repair either the aft exhaust tailpipes or upward exhaust stacks as follows:

a. Repair dents or bends by using a form block and suitable mallet.

b. Remove or smooth scratches and nicks by filing.

c. Repair cracks or breaks by stop-drilling a 0.040-inch-diameter hole at ends of the cracks and/or breaks to prevent further progression; then weld as described below:

(1) Using a stainless steel brush, thoroughly clean at least 1 inch on all sides of the cracked/broken area to remove all carbon from both inner and outer surfaces and from the crack.

NOTE

Weld repairs requiring insertion or overlay patching are not allowable.

(2) (AVIM) Weld by using inert arc method with rod (C85) that is designed for use on the CRES 21-6-9 tailpipe material. During welding, continuously flush the back-side of the tailpipe with an inert gas. If possible, the section being welded should rest on a brass bar to remove excessive heat build-up. For best results, use dc powered, tungsten tip, heli-arc welding equipment.

(3) Replace broken, excessively worn or torn hanger assemblies and springs on aft exhaust systems.

4-160. Installation — Aft Exhaust Tailpipes. (See fig. 4-16.) *a.* Remove protective covers from engine exhaust ports.

CAUTION

During tailpipe positioning and attachment, *b* through *e* below, an assistant must support the aft ends of the exhaust tailpipes while the tailpipes are being secured in place. Do not allow the aft ends of the tailpipes to become displaced from their proper mounting locations; to do so may damage the forward mounting flanges or result in a poorly mated clamp joint.

b. Position the forward end of each tailpipe over its respective engine exhaust outlet and install the two engine-to-tailpipe clamps. Position the clamp joints at the upper or lower surfaces of the tailpipes and partially tighten the clamps; final tightening will be accomplished in a step below.

c. Taking care not to displace the aft ends of the tailpipes, install the upper and lower hanger assemblies as shown. Make certain that the hanger assembly saddles are properly positioned between the locating flanges at the aft ends of the tailpipes.

NOTE

When assembling the tailpipe and hanger assemblies use care not to tear or otherwise damage the inconel wire pads that serve as cushions between the hangers and tailpipes.

d. Secure the tailpipes and hanger assemblies together by installing the spring over the rivets connecting each pair of hanger clips.

NOTE

Attach the upper spring hook so that the four captive washers on the rivet are divided equally on either side of the spring hook (two washers separating the hook from each hanger clip). Attach the lower spring hook between the left side of the lower hanger support link and the adjacent clip.

e. Attach the vertical link of the lower hanger assembly on the support fitting that extends from the structure by using CRES hardware (bolt, two washers, and nut). Tighten the nut only only finger-tight.

f. **TORQUE THE TWO ENGINE-TO-TAILPIPE CLAMP NUTS TO 15-18 INCH-POUNDS.**

NOTE

Using a plastic or rawhide mallet, tap lightly around the outside of both clamps to ensure that they are properly seated; then retorque the clamp nuts.

g. **TORQUE THE SUPPORT FITTING BOLT AND NUT TO 12-15 INCH-POUNDS** and safety with cotter pin. Check for leaks by performing an engine runup and checking for exhaust residue.

4-161. Installation — Upward Exhaust Stacks. *a.* Slip stacks into position through fuselage fairings (fig. 4-16).

b. Install stack clamps. Check that stacks are properly positioned in fairings and then **TORQUE CLAMP NUTS TO 15-18 INCH-POUNDS.**

SECTION VI OIL SYSTEM

4-162. OIL SUPPLY SYSTEM.

4-163. Description — Oil Supply System. The oil supply system includes the oil tank and cooler, oil cooler duct, oil temperature and pressure senders, check and drain valves and associated pressure and drain hoses and tubes that interconnect to the engine internal lubrication system (TM 55-2840-231-24). Oil flows from the engine through a check valve (fig. 4-17) at the inlet port of the oil cooler. The circulating oil is cooled to approximately 185° F by air blown through the cooler by the oil cooler blower that is driven by the main transmission drive shaft. The oil cooler contains a thermostat for bypassing oil around the cooler until the engine oil reaches operating temperature, and a pressure relief bypass valve. Oil then flows from the cooler into the oil tank from which the oil is pumped back to the engine internal lubricating system. Aircraft with armor provisions have a self-sealing oil tank and an oil cooler bypass system. (See fig. 4-18 and 4-19.)

4-164. Troubleshooting — Engine Internal Oil System. Refer to TM 55-2840-231-24 for engine internal oil system troubleshooting.

4-165. Troubleshooting — Aircraft Oil System. Refer to table 4-12.

4-166. OIL TANK.

4-167. Description — Oil Tank. In unarmored aircraft, the engine oil tank is constructed of two aluminum alloy halves welded together to form a sphere. (See fig. 4-17.) Space is included to allow for expansion of metal, oil, and air. Welded fittings for the vent tube, inlet line, outlet line, sight plug, and filler neck complete the assembly. In armored aircraft, the oil tank is fabricated of aluminum alloy shells, fittings and bosses welded into a tank assembly. (See fig. 4-18.) This assembly is completely coated with approximately 0.38 inch of a compound that provides for self-sealing of ruptures caused by armor piercing projectiles. The self-sealing tank is also bonded to the mounting support cradle. An oil cooler bypass line is connected to a bypass fillpipe fitting at the engine side of the tank. The bypass fillpipe extends through the tank to a point adjacent to the normal fillpipe inlet from the oil cooler. A low level warning switch is installed in the top of the tank. The switch is normally open. When the oil quantity drops below the low level operationg limit (system approximately 1 quart low), a float on the switch probe causes closure of the switch contacts. The closed contacts complete a ground circuit to a relay. On series 1 and 2 aircraft, the relay connects power from the CYCLIC TRIM circuit breaker and energized the OIL CLR BYPASS caution light and the oil cooler bypass valve. On series 3 aircraft, the relay supplies power from the OIL CLR-SCAV AIR circuit breaker which energizes the ENG OIL, LOW BYPASS MODE caution light. The bypass valve is mounted at the right side of the engine compartment. When energized, the valve diverts engine oil so that it flows directly to the oil tank bypassing the oil cooler. Oil flow from the tank back to the engine is normal. In both configurations, a replaceable sight plug provides for visual checking of the oil level within the tank.

4-168. Inspection — Oil Tank. *a.* Check oil tank for cracks, corrosion, distortion, obvious damage, leaks, and secure mounting.

b. Check all associated hoses, lines and fittings for oil leaks and secure attachment.

Table 4-12. Troubleshooting of the Engine Oil System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **No oil pressure.**

STEP 1. Check engine oil tank for correct quantity.

If the engine oil tank requires servicing, service oil tank (chapter 1).

STEP 2. Check for defective oil pressure sender.

If the oil pressure sender is found to be defective, replace oil pressure sender (para 4-202).

STEP 3. Check for defective engine oil pressure indicator.

If the engine oil pressure indicator is found to be defective, replace the indicator (chapter 8).

STEP 4. Check for clogged oil filter element (TM 55-2840-231-24).

STEP 5. Check for restricted, clogged, or damaged oil lines.

If an oil line is found to be restricted, clogged or damaged, clean or replace oil line.

2. **Low oil pressure.**

STEP 1. Accomplish inspections for no oil pressure malfunction.

STEP 2. Check for improper type or contaminated oil.

If engine oil is found to be of the improper type or contaminated, drain, flush and service oil systems, (chapter 1).

STEP 3. Check for improperly adjusted oil pressure regulator valve (TM 55-2840-231-24).

3. **Oil pressure drops off severely.**

STEP 1. Check engine oil tank for low oil quantity.

If the engine oil tank requires servicing, service oil tank (chapter 1).

STEP 2. Check for defective engine oil pressure indicator.

If the engine oil pressure indicator is found to be defective, replace the indicator (chapter 8).

STEP 3. Check for defective oil pressure sender.

If the oil pressure sender is found to be defective, replace oil pressure sender (para 4-202).

STEP 4. Check oil pressure regulator valve for sticking or broken spring (TM 55-2840-231-24).

STEP 5. Check for defective oil pump (TM 55-2840-231-24).

4. **Excessive oil pressure fluctuation.**

STEP 1. Check for defective engine oil pressure indicator.

If the engine oil pressure indicator is found to be defective, replace the indicator (chapter 8).

STEP 2. Check for damaged oil lines and oil leaks.

If oil lines are damaged, replace the lines. If oil leak exists, repair or replace defective oil line(s).

STEP 3. Check for clogged oil filter element (TM 55-2840-231-24).

STEP 4. Check for improperly adjusted oil pressure regulator valve (TM 55-2840-231-24).

STEP 5. Check for defective oil pump (TM 55-2840-231-24).

5. **High oil pressure.**

Table 4-12. Troubleshooting of the Engine Oil System. (cont)

STEP 1. Check for defective engine oil pressure indicator.

If the engine oil pressure indicator is found to be defective, replace the indicator (para 8-17).

STEP 2. Check for defective oil cooler bypass valve.

If the bypass valve is found to be defective, replace bypass valve (para 4-188).

STEP 3. Check for improperly adjusted oil pressure regulator valve (TM 55-2840-231-24).

6. High oil temperature with normal oil pressure (armored aircraft).

STEP 1. Check for leaking oil cooler causing actuation of cooler bypass valve.

If the oil cooler is leaking, replace oil cooler (para 4-181).

STEP 2. Check for defective oil cooler bypass valve.

If the bypass valve is found to be defective, replace bypass valve (para 4-188).

STEP 3. Check for defective low level warning switch.

If the low level warning switch is found to be defective, replace the switch (fig. 4-18).

7. Abnormal oil temperature indication.

STEP 1. Check for defective oil pressure sender.

If the oil pressure sender is found to be defective, replace oil pressure sender (para 4-202).

STEP 2. Check for defective engine oil temperature indicator.

If the oil termperature is found to be defective, replace the indicator (chapter 8).

8. Oil consumption exceeds 6.50 fluid oz (0.05 gallon) per hour.

STEP 1. Check for oil leaks at loose fittings or connections.

If fittings or connections are loose, tighten fittings or connections as required.

STEP 2. Check for clogged or restricted vent lines.

If vent lines are found to be clogged or restricted, clear or replace vent lines.

STEP 3. Check for defective internal engine oil seals (TM 55-2840-231-24).

9. OIL CHIPS or ENG CHIPS caution light illuminates.

STEP 1. Check for metallic chips in lubrication system (TM 55-2840-231-24).

10. OIL CLR BYPASS caution light illuminates with no oil system leaks (armored aircraft).

STEP 1. Check engine oil tank for insufficient oil.

If the engine oil tank requires servicing, service oil tank (chapter 1).

STEP 2. Check for defective low level warning switch in oil tank.

If the low level warning switch is found to be defective, replace the switch (fig. 4-18).

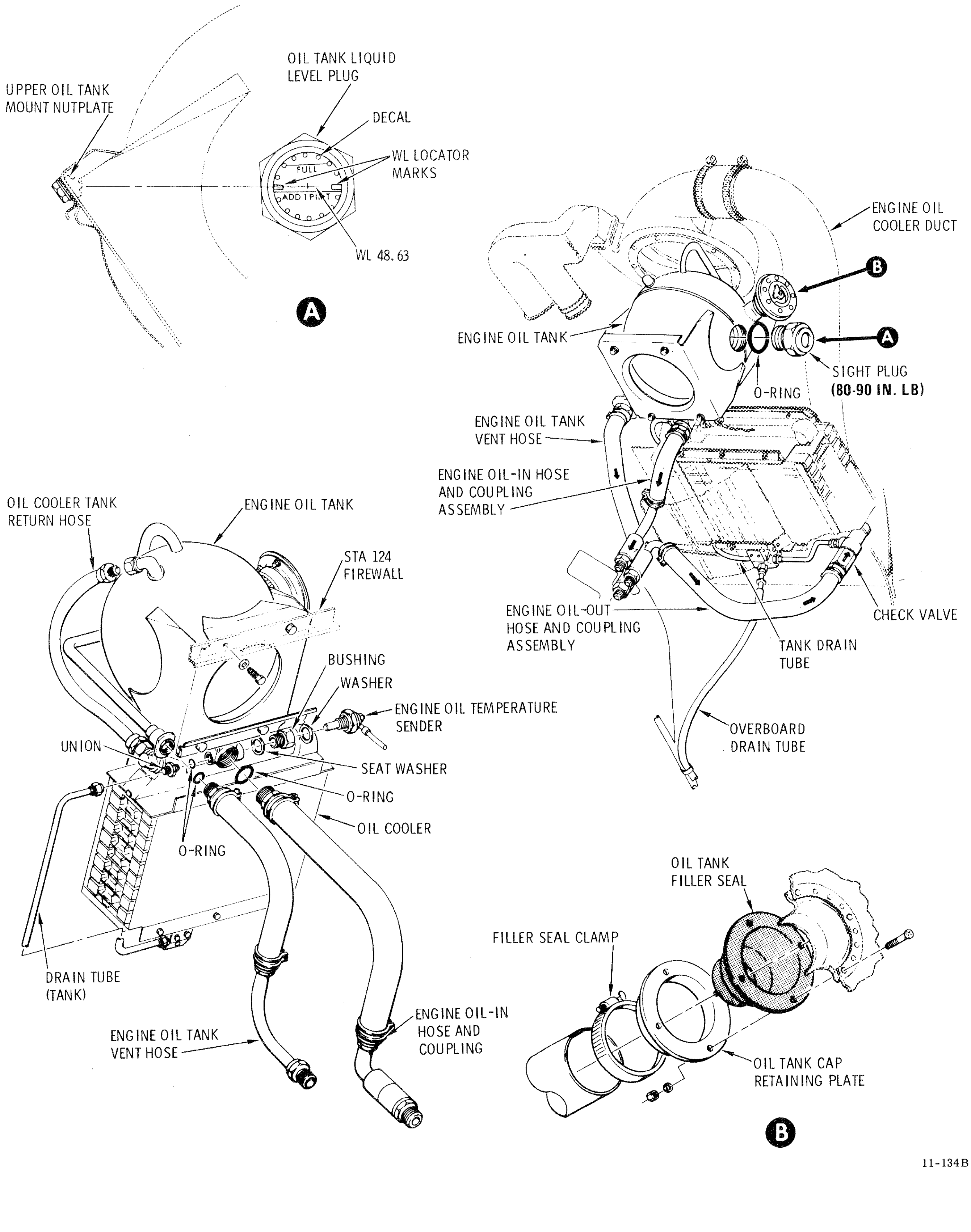


Figure 4-17. Engine Oil Tank (Unarmored Aircraft).

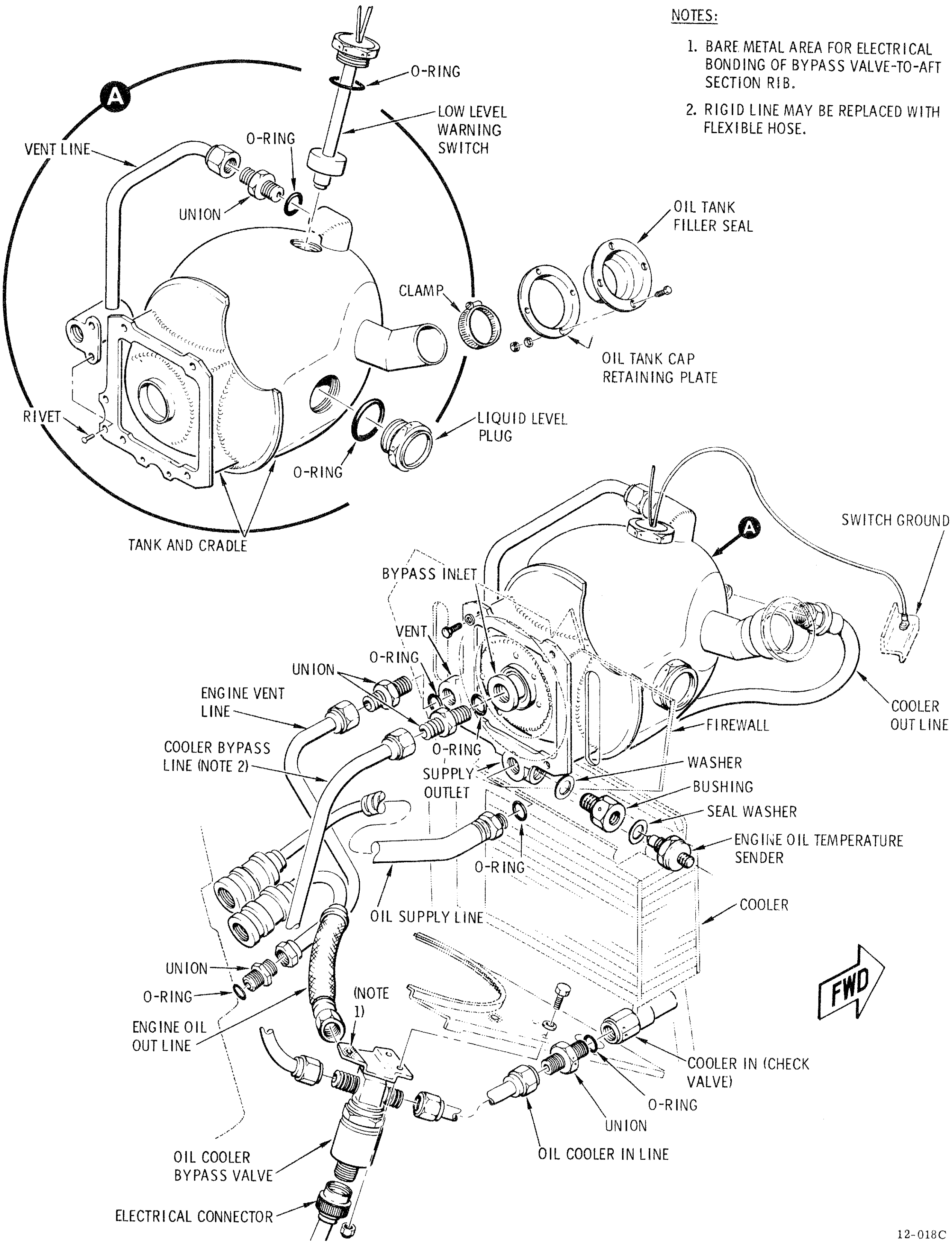


Figure 4-18. Engine Oil Tank (Armored Aircraft).

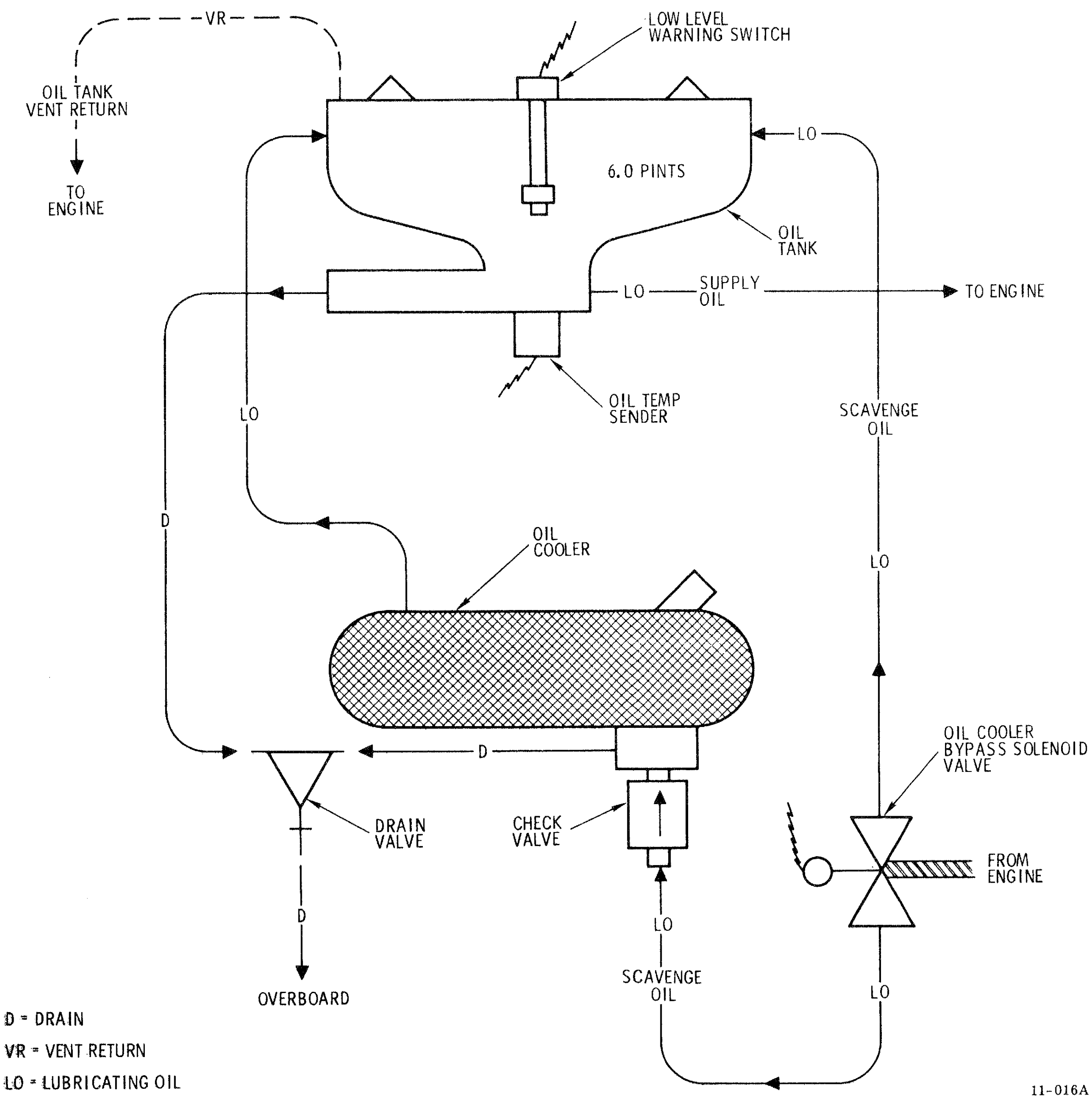


Figure 4-19. Engine Oil Supply System Schematic (Armored Aircraft).

4-169. Test (Operational Check) — Oil Tank. Check aircraft with armor provisions for correct bypass oil level switch operation (Special Inspection, chapter 1) as follows.

a. Check that oil level is at the full line. Drain oil from tank until oil cooler bypass caution light and bypass valve are energized. Measure the oil drained. With the switch closed and the caution light and bypass valve energized, from 32 to 60 ounces of oil should have been drained. If drained oil is not within the specified range the switch is faulty and should be replaced.

b. Return drained oil to tank and repeat check if necessary.

4-170. Removal — Oil Tank (Unarmored Aircraft). (See fig. 4-17.) During removal, cap or plug all open fittings, lines or ports.

a. Remove the right side troop seat, sound insulation, and oil cooler access door in the cargo compartment.

b. Drain the oil system (chapter 1).

c. Remove the oil cooler duct for access to oil tank fittings (para 4-177).

d. Open the engine compartment access doors for access to the oil hoses.

e. Disconnect the oil tank vent hose assembly at the oil tank support by loosening the clamp that secures the fire sleeve. Slide the clamp and fire sleeve back to clear the fitting.

f. Disconnect the oil-in hose and coupling assembly at oil tank outlet fitting by loosening clamp that secures the fire sleeve. Slide the clamp and fire sleeve back to clear the fitting.

g. Disconnect the oil cooler tank return hose at oil tank inlet fitting.

h. Disconnect oil tank drain tube at the tank outlet fitting. Allow tube to drain; then cap open end of fitting.

i. Remove the oil tank filler seal clamp; then remove the four nuts, washers, and screws that secure the oil tank cap retaining plate and oil tank filler seal. Lift the oil tank cap retaining plate and oil tank filler seal from position.

j. Remove four bolts and washers that secure the oil tank to the firewall and remove the oil tank.

k. Remove the oil tank liquid level plug and gasket as necessary.

l. Remove oil temperature sender, seat washer, and bushing.

m. Remove oil supply line union and O-ring.

4-171. Removal — Oil Tank (Armored Aircraft). (See fig 4-18.) During removal, cap or plug all open fittings, lines or ports.

a. Remove the right side troop seat, sound insulation, and oil cooler access door in the cargo compartment.

b. Drain the oil system (chapter 1).

c. Remove the oil cooler duct for best access to oil tank fittings (para 4-179).

d. Open the engine compartment access doors for access to the oil hoses.

e. Disconnect engine oil vent line (fig 4-18) oil cooler bypass line (or flexible hose), and the engine oil-in hose and coupling assembly. Remove unions and O-rings; discard used O-rings.

f. In the cargo compartment, disconnect the low level switch ground connection and wire knife splice. Disconnect engine oil temperature sender wire by removing nut and washers.

g. Disconnect the engine oil cooler outline at the oil tank fitting.

h. Disconnect the oil tank drain tube at the tank fitting.

i. Remove the oil tank filler seal clamp; then remove the four nuts, washers and screws that secure the oil tank cap retaining plate and oil tank filler seal. Lift the oil tank cap retaining plate and oil tank filler seal from position.

j. Remove the four bolts and washers that secure the oil tank to the firewall and remove the oil tank.

k. Remove the oil tank liquid level plug and O-ring, if necessary.

l. Remove the oil tank low level warning switch and O-ring.

m. Loosen the oil tank vent line and remove the union and O-ring.

NOTE

Do not remove vent line and boss assembly from the tank unless condition warrants replacement. The boss of the assembly is riveted to the tank cradle.

n. Remove engine oil temperature sender, washer, busing and washer.

4-172. Cleaning — Oil Tank. Clean the interior of the engine oil tank by agitation with mineral spirits (C109) or solvent (C94) until cleaning solution shows no signs of particles or oil traces.

4-173. Repair — Oil Tank (Unarmored Aircraft). *a.* (AVIM) Weld all cracks, open seams, loose fittings, and tubes with welding rod (C86).

b. Replace all damaged or unserviceable nutplates.

c. Remove or smooth scratches and nicks by filling.

d. Replace a defective oil tank, sight plug, hose, tube, or fitting.

4-174. Repair — Oil Tank (Armored Aircraft). *a.* Replace damaged, unserviceable nutplates.

b. Replace damaged, unserviceable components such as vent tube assembly, sight plug, float switch, and oil temperature sender.

c. Repair separation between tank cradle and self-sealing tank as follows:

NOTE

The self-sealing material on the tank cannot be removed or repaired. The tank is not repairable by welding. Replace unserviceable tanks.

(1) Carefully scrape away any residual bonding material at tank-to-cradle separation.

(2) Clean surfaces with a cloth dampened with naphtha and allow to dry for a minimum of 20 minutes.

(3) Using two-part adhesive (C11) mix 100 parts resin with 22 parts catalyst by weight. Mixed adhesive has a 1 to 1-1/2-hour working life.

(4) Completely fill the separation (gap) with adhesive and allow to cure for approximately 24 hours at ambient (room) temperature.

4-175. Installation — Oil Tank (Unarmored Aircraft). (See fig. 4-17.) *a.* Install new O-ring on tank drain tube union and install union.

b. Coat liquid level plug threads with lubricating oil (C67); then install with new gasket. **TORQUE PLUG TO 80-90 INCH-POUNDS.**

c. Install oil temperature sender bushing using a new washer and secure with 0.032-inch lockwire (C57).

d. Install oil temperature sender using a new washer. **TORQUE TEMPERATURE SENDER TO 100-150 INCH-POUNDS.**

e. Position the engine oil tank on firewall and install four washers and bolts. Tighten the bolts.

f. Install the oil tank filler seal in the oil tank cap retaining plate and secure with four bolts, washers, and nuts. Tighten nuts.

g. Ensure that oil tank filler seal neck is properly mated over oil tank filler neck. Install oil tank seal clamp.

NOTE

Before performing h through k below, remove protective caps or plugs from hose or tube assemblies and fittings or ports.

h. Connect tank drain tube to oil tank outlet fitting and tighten.

i. Connect oil cooler tank return hose to oil tank inlet fitting and tighten.

j. Connect oil-in hose and coupling assembly to oil tank outlet fitting and tighten.

k. Connect oil tank vent hose assembly to fitting provided on oil tank support and tighten.

l. After installing oil-in hose and coupling assembly, and the oil tank vent hose assembly, pull fire sleeves into position over ends of respective hoses and tighten clamps.

m. Install oil cooler duct (para 4-180).

n. Fill oil system with oil (C67).

o. Ground run the engine and check oil tank and related parts for leaks.

4-176. Installation — Oil Tank (Armored Aircraft). (See fig 4-18) *a.* Coat threads of the oil tank liquid level plug with lubricating oil (C67). Install new O-ring and liquid level plug in oil tank opening. **TORQUE LIQUID LEVEL PLUG TO 80-90 INCH-POUNDS.** Secure with 0.032-inch lockwire (C57).

b. Coat threads of the low level warning switch with lubrication oil (C67). Using a new O-ring, install O-ring and low level warning switch in oil tank opening. Secure with 0.032-inch lockwire (C57).

c. Install engine oil temperature sender bushing using a new washer. Secure with 0.032-inch lockwire (C57).

d. Install engine oil temperature sender using a new seal washer. **TORQUE SENDER TO 100-150 INCH-POUNDS.**

e. Install vent line union using new O-ring and tighten union. Connect oil tank vent line and tighten line nut.

f. Position and align the engine oil tank with firewall and install four bolts with washers.

g. Install the oil tank filler seal in the oil tank cap retaining plate and secure with four bolts, washers, and nuts.

h. Ensure that oil tank filler seal neck is properly mated over oil tank filler neck. Install oil tank seal clamp.

NOTE

Remove protective caps or plugs from hose or tube assemblies and fittings and ports.

i. Install unions in vent and bypass ports using new O-rings and connect vent and bypass lines.

j. Connect engine oil-in hose using new O-ring. Pull fire sleeves into position over hoses and tighten clamps.

k. In the cargo compartment, connect oil cooler out line to tank return fitting.

l. Connect low level switch ground terminal to structure using screw, nut and washer. Connect knife splice, slip fiberglass sleeving over the connected splice, and tie in place.

m. Connect oil temperature sender terminal with a washer and nut.

n. Install oil cooler duct (para 4-180).

o. Fill oil system with oil (C67).

p. Ground run the engine and check oil tank, lines and fittings for leakage.

4-177. OIL COOLER DUCT.

4-178. Description — Oil Cooler Duct. The engine oil cooler duct is a plastic assembly that directs cooling air from the oil cooler blower to the engine oil cooler.

4-179. Removal — Oil Cooler Duct. (See fig 4-20.) *a.* Remove right side troop seat, sound insulation, and oil cooler access door in cargo compartment.

b. Remove the two clamps and air duct connector.

c. Remove the four nuts, washers, and bolts that secure the engine oil cooler duct to the oil cooler. Remove the engine oil cooler duct.

NOTE

The four spacers may or may not be bonded to the duct flange. Retain unbonded spacers for reinstallation of duct.

4-180. Installation — Oil Cooler Duct. (See fig. 4-20.) *a.* Position oil cooler duct at oil cooler. If the four spacers are not bonded to the duct flange, install spacers at each attaching bolt location.

CAUTION

Overtightening bolts can crack the duct flange. Tighten nuts only to the point of firm contact.

b. Install the four attaching bolts, washers, and nuts.

c. Install the air duct connector and clamps. Tighten the clamps.

4-181. OIL COOLER.

4-182. Description — Oil Cooler. The engine oil cooler assembly (fig. 4-20) is a radiator-type heat exchanger that is constructed of aluminum alloy tubes welded to a frame. Engine outlet oil enters the cooler through an inlet fitting at the lower right corner, circulates through the tubes, and exits through an outlet fitting at the upper left corner of the cooler. In armored aircraft, a bypass valve is installed between the engine and the oil cooler. (See fig. 4-18.) If the oil cooler is ruptured it is bypassed and isolated from oil system pressure (para 4-162). The cooler bypass thermostat starts to close at approximately 140°F (60°C) and is fully closed at 180°F (82°C). The bypass pressure relief valve "cracks" at 25 psig and is fully opened at 30 psig (maximum). The bypass valve provides 3.26 gpm flow at temperatures down to -25°F (-32°C).

4-183. Inspection — Oil Cooler. Check oil cooler for cracks, corrosion, damaged or bulged plates, broken welds, foreign matter clogging, oil leaks, and secure mounting. Check all associated hoses and fittings for oil leaks and secure attachment.

Table 4-13. Premaintenance Requirements for Removal of Oil Cooler.

Conditions	Requirements
Special Tool	(T6)
Minimum Personnel Required	Two

4-184. Removal — Oil Cooler. (See fig. 4-20.) During removal, cap or plug all open fittings, lines or ports.

a. Remove the right side troop seat, sound insulation and oil cooler access door in the cargo compartment.

b. Drain the oil system (chapter 1).

c. Disconnect the oil cooler tank return hose from the oil cooler. Allow oil to drain; then cap hose connection.

d. Remove engine oil cooler duct (para 4-177).

e. Remove the drain tube, union, and O-ring.

f. Open engine compartment access door.

CAUTION

Have an assistant hold pressure with tool (T6) across the flats of the oil cooler inlet boss during performance of *g*, *h*, and *i* below to prevent twisting of the boss.

g. Disconnect oil hose connected to the check valve at the oil cooler inlet port.

h. In armored aircraft: remove union and packing in check valve.

i. Remove the check valve and O-ring.

j. Remove the screws and washers securing the plastic deflector to the mounting flange. Remove deflector.

k. Remove the six nuts, washers, and bolts that secure the oil cooler to the firewall structure and remove the oil cooler.

NOTES:

1. ENGINE OIL OUT HOSE SHOWN IS USED IN UNARMORED AIRCRAFT. IN ARMORED AIRCRAFT, A SIMILAR LINE CONNECTS TO THE BYPASS VALVE OUT FITTING.
2. SPACERS MAY BE BONDED TO OR BE SEPERATE FROM THE OIL COOLER DUCT.

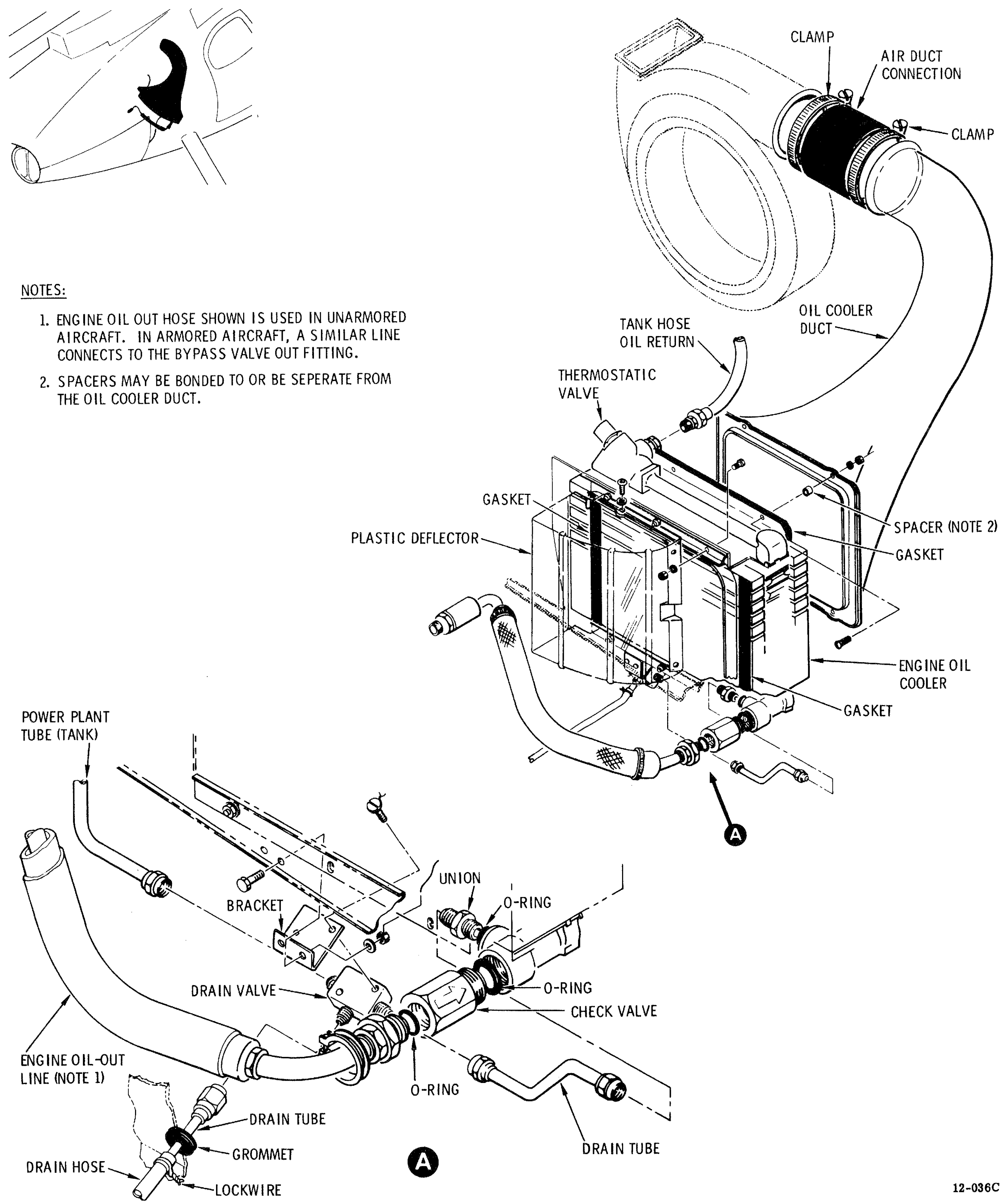


Figure 4-20. Oil Cooler and Duct Assembly.

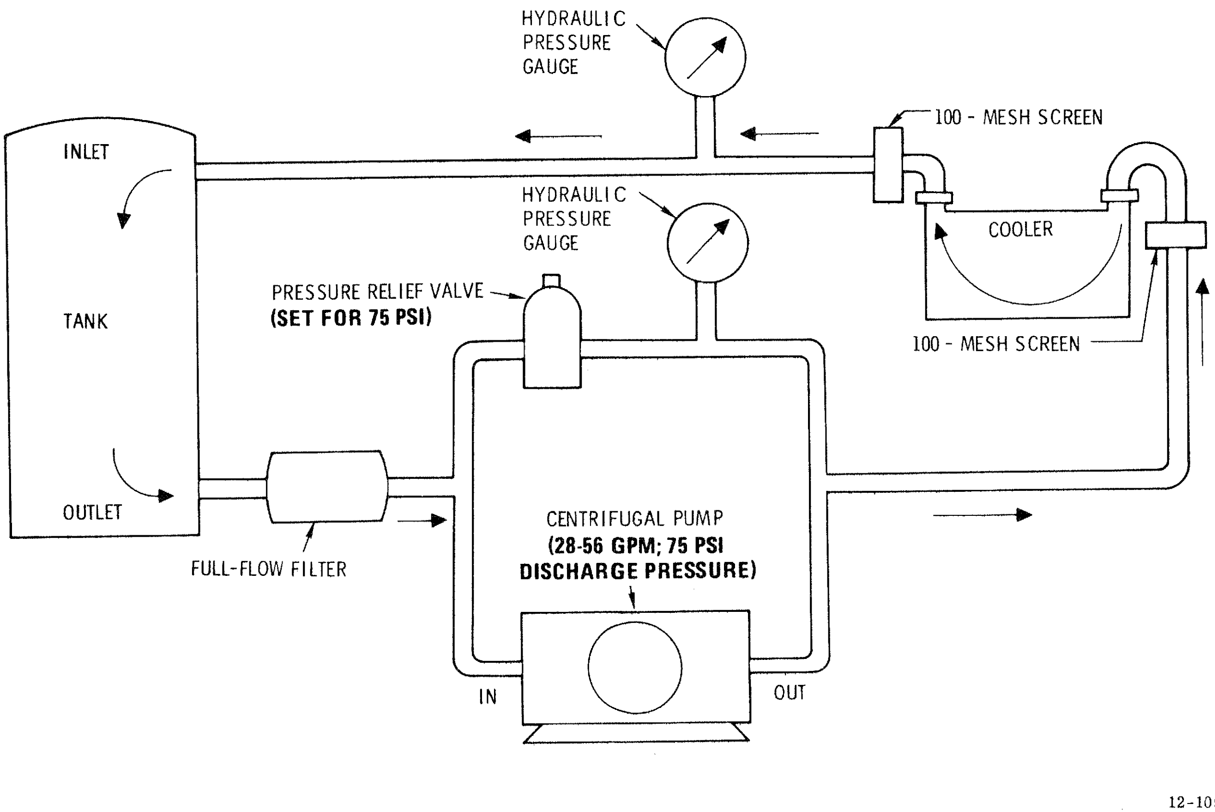


Figure 4-21. Oil Cooler Cleaning Setup.

4-185. Cleaning — Oil Cooler (AVIM). Establish cleaning setup as shown in figure 4-21.

a. Plug oil cooler inlet port.

b. Remove thermostatic valve (fig. 4-20) and plug openings in oil cooler.

c. Plug oil cooler outlet port.

d. Clean exterior of oil cooler with steam.

CAUTION

Use only cleaning solutions recommended for use on aluminum. Many solutions satisfactory for cleaning cooper or copper nickel are highly corrosive to aluminum and, if used, will result in the destruction of the oil cooler assembly. If equipment has been previously used with any other cleaning solution, it should be thoroughly washed and flushed with the recommended solution.

e. Use solvent (C94) to clean oil cooler interior.

f. Connect hose from discharge side of pump to oil cooler outlet to provide a cleaning solution flow opposite to direction of normal flow.

g. Connect hose from storage tank to inlet port of oil cooler.

h. Start centrifugal pump and allow to flush for 30 minutes or until the cleaning solution appears clean after flowing through oil cooler.

i. Reverse hoses on oil cooler and flush for approximately 15 minutes in opposite (normal flow) direction.

j. Drain oil cooler.

k. Flush oil cooler with compound (C27) in reverse direction for 30 minutes.

l. Reverse hoses and flush oil cooler for 10 to 15 minutes in normal oil flow direction.

m. To rinse, flush oil cooler with solvent (C94) for 10 minutes.

n. Check filter screens for metallic particles. Dispose of the oil cooler if metallic particles still appear in filter screens.

o. If metal particles are found, initiate oil analysis procedure.

p. Continue cleaning operations. Use solvent (C94) and dry with clean, filtered, low pressure, compressed air.

4-186. Repair — Oil Cooler. *a.* Replace a defective thermostatic valve (fig. 4-20).

b. Straighten bent cooling fins with a pair of duck-bill pliers ground to fit between cooling tubes.

c. (AVIM) Weld all small holes, cracked seams, or loose fittings with welding rod (C86). Refer to TM 55-1500-204-25/1 for welding practices and procedures.

d. Smooth scratches and nicks by filing.

Table 4-14. Premaintenance Requirements for Installation of Oil Cooler.

Conditions	Requirements
Special Tool	(T6)
Minimum Personnel Required	Two
Consumable Material	(C67)

4-187. Installation — Oil Cooler. (See fig 4-20.) *a.* Place oil cooler in position on the firewall. Install the drain tube union using a new O-ring. Install the drain tube between union and drain valve.

b. In the engine compartment: install six bolts, washers and nuts to attach cooler to firewall.

c. Remove protective caps from cooler and check valve. Coat threads of check valve with lubrication oil (C67).

CAUTION

Have an assistant hold pressure with tool (T6) across the flats of the oil cooler inlet boss during performance of *d*, *e* and *f* below to prevent twisting of the boss. Install check valve with arrows pointing toward oil cooler.

d. Install check valve with a new O-ring.

e. In unarmored aircraft: install oil-out line with a new O-ring.

f. In armored aircraft: install union and new O-ring in the check valve; then connect oil cooler in line.

g. Install plastic deflector on oil cooler mounting flange and secure with screws and washers.

h. In the cargo compartment: remove protective caps from the oil cooler port, and oil tank return hose. Install hose.

i. Fill oil system with oil (C67).

j. Ground run the engine and inspect oil cooler and related parts for oil leaks.

k. Close oil cooler access door and reinstall sound insulation, and troop seat.

4-188. OIL SUPPLY SYSTEM DRAIN VALVE, CHECK VALVE, AND OIL COOLER BYPASS VALVE.

4-189. Description — Oil Supply System Drain Valve, Check Valve, and Oil Cooler Bypass Valve. The oil supply system drain valve, a three port plug-type valve, is mounted on a bracket beneath the engine oil cooler. (See fig. 4-20.) Three oil tubes are connected to the drain valve: one from the engine oil tank, one from the engine oil cooler, and the third provides an overboard drain. The drain valve is spring-loaded in the closed position. When the valve is opened, all ports are open to each other; when closed, all ports are closed. The oil system check valve is a one-way, ball-type valve mounted at the inlet port of the oil cooler. The check valve functions to prevent hot oil from flowing back into the engine when the engine is not operating. In armored aircraft (fig. 4-18) a three-way, two-position, solenoid operated bypass valve is installed in the oil system. The valve is mounted on the fuselage structure just aft of the oil cooler deflector in the engine compartment. It is connected in the engine oil-out line, between the engine and oil cooler. When the valve coil is energized, oil flow is from the IN port to the BYPASS port (oil tank). When deenergized, oil flow is from the IN port to the OUT port (oil cooler).

4-190. Inspection — Oil Supply System Drain Valve, Check Valve, and Oil Cooler Bypass Valve. Check the oil system drain valve, check valve and bypass valve for cracks, corrosion, secure attachment, damage and leaks. Check connecting hoses and tubing for secure attachment, damage and leaks at connections.

Table 4-15. Premaintenance Requirements for Removal of the Oil Supply System Drain Valve and Check Valve.

Conditions	Requirements
Special Tool	(T6)
Minimum Personnel Required	Two

4-191. Removal — Oil System Drain Valve and Check Valve. (See fig. 4-20.) The following procedures apply to both armored and unarmored aircraft.

a. Drain the oil system (chapter 1).

CAUTION

Have an assistant hold pressure with tool (T6) across the flats of the oil cooler inlet boss during removal of oil-out hose or check valve to prevent twisting of the boss.

b. In unarmored aircraft: Disconnect the engine oil-out hose and coupling assembly at inlet port of the check valve and allow to drain. Remove O-ring.

c. In armored aircraft: Disconnect oil line connected to union in the check valve. Remove union and O-ring in check valve.

d. Remove the check valve and O-ring.

e. Plug opening at inlet port of oil cooler and cap open end of oil out (or bypass valve) hose and coupling assembly.

f. Disconnect the tank drain tube, drain hose, and drain tube at the drain valve. Cap or plug all open lines or ports.

g. Remove the two nuts, washers, bolts and brackets with drain valve attached. Remove lockwire and the two screws securing drain valve to the bracket.

4-192. Removal — Oil Cooler Bypass Valve (Armored Aircraft). (See fig 4-18.) *a.* Open the engine compartment doors for access.

b. Remove lockwire and disconnect the electrical connector from the oil cooler bypass solenoid valve.

c. Disconnect the oil lines connected to the bypass valve IN, OUT, and BYPASS ports. Cap or plug all open lines or ports.

d. Remove the three nuts, bolts, and washers that attach valve to the structure. Remove the valve.

4-193. Test — Oil Cooler Bypass Valve (AVIM). *a.* Make test setup as shown in figure 4-22.

b. Connect pressure line from test stand to the bypass valve IN port.

NOTE

Use only hydraulic fluid (C48) or (C76) in the test stand.

c. Install a cap on bypass valve out port. Set the power selector switch at OFF.

d. Raise test stand pressure to 30 psig and check for leakage at the BYPASS port. Leakage shall not exceed two drops per minute.

e. Reduce test stand pressure to zero. Remove the cap from the OUT port and install it on the BYPASS port.

f. Close the switch and energize the bypass solenoid; then raise test stand pressure to 5 psig.

g. Check for leakage at the bypass valve OUT port. Leakage shall not exceed two drops per minute.

h. Reduce test stand pressure to zero. Open the switch to de-energize the bypass solenoid.

i. Connect test stand return line to the bypass valve OUT port. Check that restrictor valve is open.

j. Raise test stand pressure to **30 PSIG AT 2.85 GPM. PRESSURE DROP AS READ ON RETURN LINE GAGE SHALL NOT EXCEED 6 PSI (24 PSI MINIMUM READING).**

k. Reduce test stand pressure to zero. Remove return line from OUT port and connect to BYPASS port. Energize the bypass solenoid by closing power switch.

l. Raise test stand pressure to **30 PSIG AT 2.85 GPM. THE PRESSURE DROP AS READ ON RETURN LINE GAGE SHALL NOT EXCEED 8 PSI (22 PSI MINIMUM READING).**

m. Reduce test stand pressure to zero and deenergize the solenoid valve.

n. Remove return line from BYPASS port and connect to the OUT port. Connect a third line from the BYPASS port to the tee N return line.

o. Raise test stand pressure and establish a flow rate of 2.85 gpm at 30 psig by partially closing restrictor valve and adjusting test stand pressure as required.

p. Energize the bypass solenoid to divert flow to the BYPASS port; then deenergize the solenoid. Repeat valve operation through 10 cycles.

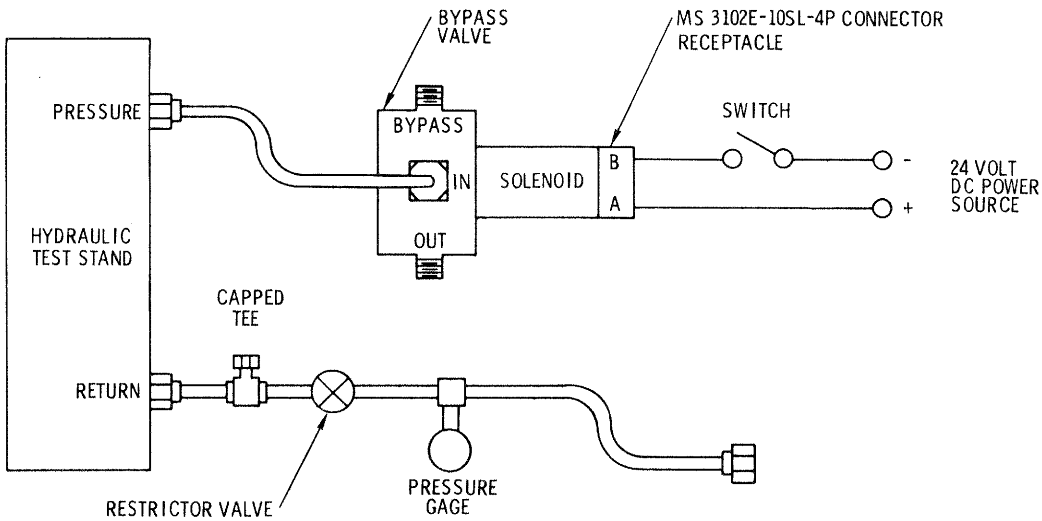
q. Reduce test stand pressure to zero and shut down. Disconnect electrical power and remove test equipment from valve.

r. Flush the valve thoroughly with solvent (C94) and dry the exterior with a clean cloth.

Table 4-16. Premaintenance Requirements for Installation of the Oil System Drain Valve and Check Valve.

Conditions	Requirements
Special Tool	(T6)
Minimum Personnel Required	Two
Consumable Material	(C57) (C67)

4-194. Installation — Oil System Drain Valve and Check Valve. (See fig. 4-20.) *a.* Install a new O-ring on the check valve.



TEST SETUP COMPONENTS

HYDRAULIC TEST STAND, 10 GPM AT 3000 PSI, CONTROLLABLE
PRESSURE GAGE, 60 PSI, LABORATORY TYPE
POWER SUPPLY, 24 VOLTS, DIRECT-CURRENT
SPST SWITCH AND WIRING AS REQUIRED
VALVE, HYDRAULIC FITTINGS, AND HOSES AS REQUIRED

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Figure 4-22. Oil Cooler Bypass Valve Test Setup.

CAUTION

Have an assistant hold pressure with tool (T6) across the flats of the oil cooler inlet boss during installation of oil-out hose or check valve to prevent twisting of the boss. Install check valve with arrows pointing towards oil cooler.

b. Coat threads of check valve with lubricating oil (C67); install male end of valve in cooler port and tighten.

c. In unarmored aircraft: Remove cap from end of engine oil-out line.

d. In armored aircraft: Install a new O-ring on the union and install the union in check valve. Remove cap from end of bypass oil line.

e. Connect oil line to union or check valve as applicable. Tighten coupling nut.

f. Position drain valve on the mounting bracket, install two screws and tighten. Secure screws with 0.032-inch lockwire (C57).

g. Install the bracket on oil cooler flange with two bolts and nuts.

h. Remove caps and connect tube assemblies to the drain valve, and tighten to secure.

i. Ground run the engine and check oil supply system check valve, drain valve, and associated plumbing for oil leaks.

4-195. Installation — Oil Cooler Bypass Valve. (See fig. 4-18.) Maintain electrical bond surfaces. Refer to TM 55-1500-204-25/1.

a. Remove protective caps from valve and lines.

b. Hold the BYPASS valve in the correct relative position. Connect, but do not tighten, the corrosion-resistant steel lines from bypass port on oil tank to BYPASS port on valve, and from the oil cooler check valve to the OUT port on the valve.

c. Align valve mounting holes with holes in structure. Install three bolts, nuts and washers. Tighten securely.

d. Tighten the line connections at bypass valve OUT and BYPASS ports.

e. Connect and tighten the oil-out line to the bypass valve IN port.

f. Remove lockwire from two of the four screws that attach valve body to its mounting bracket. Connect electrical connector and two screwheads together with 0.032-inch lockwire (C57).

4-196. OIL TEMPERATURE SENDER.

4-197. Description — Oil Temperature Sender. The oil temperature sender, mounted at the bottom of the oil tank and electrically connected to the ENG OIL TEMP indicator on the instrument panel, senses the temperature of the oil supplied from the oil tank to the engine. (See fig. 4-17 or 4-18.) Oil passing around the sender bulb causes the internal circuit resistance of the sender to vary in proportion to oil temperature. This resistance variation results in increased or decreased current flow through the ENG OIL TEMP indicator and causes proportional movement of the indicator pointer.

4-198. Test (Operational Check) — Oil Temperature Sender. Start and operate engine according to TM 55-1520-214-10. Check oil temperature sender for oil leaks and indicating circuit for proper operation. Replace defective units.

4-199. Removal — Oil Temperature Sender. (See fig. 4-17 or 4-18.) *a.* Drain the oil system (chapter 1).

b. Disconnect electrical wire and tape loose end.

c. Remove the engine oil temperature sender and seat washer.

d. Cap or plug hole in tank fitting.

4-200. Inspection — Oil Temperature Sender. Check the engine oil temperature sender for cracks, corrosion, damage, obvious oil leaks and secure electrical connection. Replace a defective oil temperature sender.

4-201. Installation — Oil Temperature Sender. (See fig. 4-17 or 4-18.) *a.* Remove plug or cap from tank fitting.

b. Position the seat washer, install the sender into fitting bushing; **TORQUE SENDER TO 100 - 150 INCH-POUNDS.**

c. Remove tape from electrical wire and connect wire to terminal.

d. Fill the oil system (chapter 1).

4-202. OIL PRESSURE SENDER.

4-203. Description — Oil Pressure Sender. The oil pressure sender, mounted on the right side engine mount and electrically connected to the ENG OIL PRESS indicator on the instrument panel, senses oil pressure between the pressure regulating valve and the engine internal lubrication system. (See fig. 4-23.) Oil pressure variations at the oil pressure sender orifice cause resistance variations proportional to the pressure variations. The resistance changes result in increased or decreased current flow through the ENG OIL PRESS indicator and cause proportional movement of the indicator pointer.

4-204. Test (Operational Check) — Oil Pressure Sender. Start and operate engine according to TM 55-1520-214-10. Check oil pressure sender and pressure line for oil leaks, and indicating circuit for proper operation. Replace defective units.

CAUTION

Do not adjust engine oil pressure until integrity of electrical system has been verified.

NOTE

Engine oil pressure may be verified by using a Gage, Oil Pressure, Direct Reading, 0-200 lb GGG-76, Class 1 Type A.

4-205. Removal — Oil Pressure Sender. (See fig. 4-23.) *a.* Remove pressure tube nut from pressure sender connector. Provide container to catch any residual oil spillage from pressure line; plug line fitting.

b. Disconnect electrical wire and tape loose end.

c. Remove the two bolts, washers, and nuts securing the sender and pressure line connector clamps to the clamps on the engine mount.

d. Remove engine oil pressure sender.

e. Remove connector from sender.

4-206. Inspection — Oil Pressure Sender. Check the oil pressure sender for cracks, corrosion, damage, secure mounting, and obvious oil leaks. Check for secure electrical connection.

4-207. Installation — Oil Pressure Sender. (See fig. 4-23.) *a.* Apply graphite lubricant (C62) on sender pipe threads.

b. Install and tighten female connector on the sender.

c. Install clamp on sender case and clamp around hex of connector. Attach clamps to engine mount clamps with bolts, washers and nuts. Do not tighten.

d. Remove plug from pressure line. Align sender connector with tube nut, connect and tighten line. Check both ends of pressure line for secure attachment. Tighten clamp bolts.

e. Remove tape from electrical wire and connect wire to terminal.

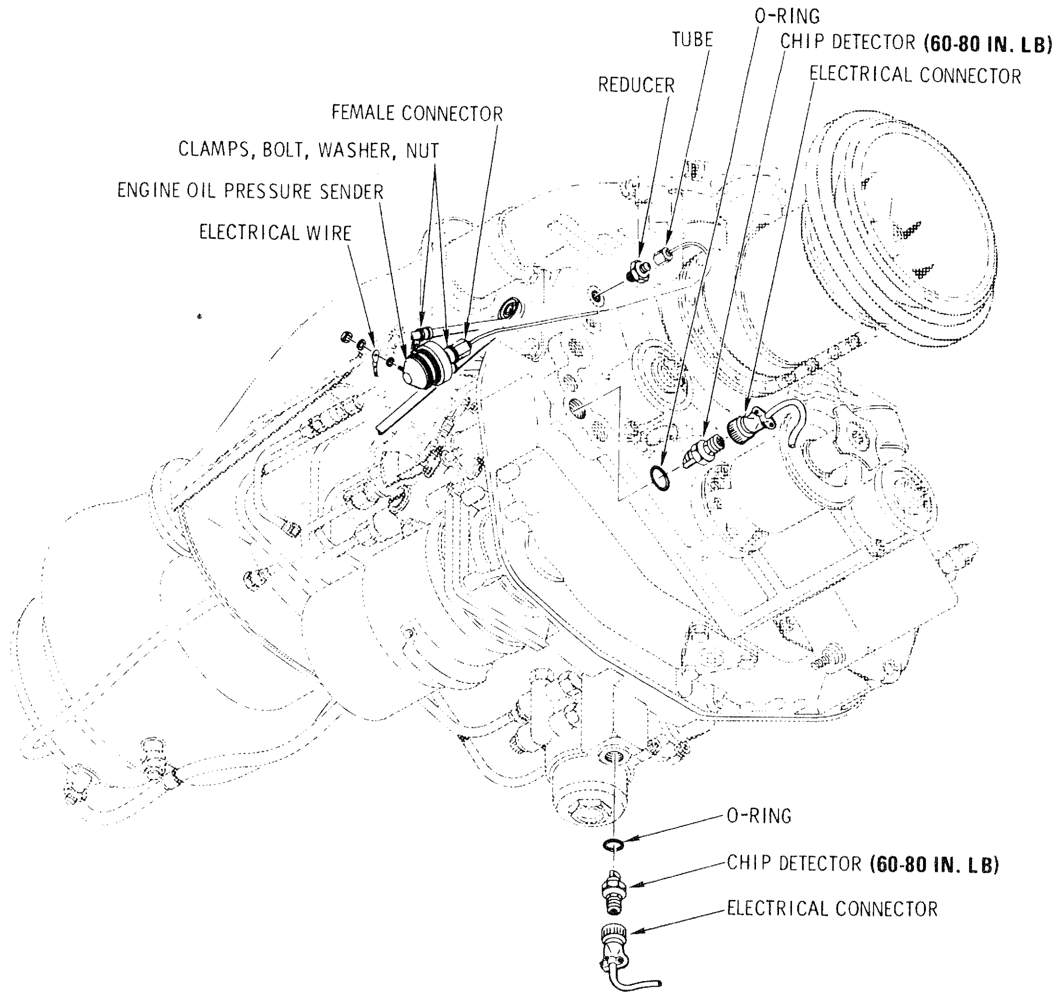


Figure 4-23. Engine Oil Pressure Sender and Chip Detector.

4-208. OIL SUPPLY SYSTEM HOSES, FITTINGS, AND TUBING.

4-209. Description — Oil Supply System Hoses, Fittings, and Tubing. The hoses and tubing used in the oil supply system are lightweight assemblies incorporating permanent fittings. The hoses located near the engine are provided with fire shields. In armored aircraft the oil cooler bypass line is either a corrosion-resistant steel tube or a flexible hose. Hose and tube assemblies are located and connected as shown in figures 4-17 and 4-18.

4-210. Inspection (Installed) — Oil Supply System Hoses, Fittings, and Tubing. *a.* Check oil lines for kinks, uniformity of diameter, breaks, and freedom from interference with adjoining structure or other components. Replace defective oil lines.

b. Check fittings and hardware for cracks, crossed threads, obstructions in openings, burrs or other damage. Replace all damaged fittings. Replace all seals, O-rings, selflocking nuts, cotter pins, and lockwire when removed from a unit.

4-211. Removal — Oil Supply System Hoses, Fittings, and Tubing. When removing an oil supply system hose or tube that is connected to lower end of oil cooler or oil tank, drain the oil from the cooler or tank before disconnecting the hose or tube. Tag or otherwise identify all fittings, tubing and hardware to aid reassembly.

a. Disconnect lower end of oil line. Drain residual oil in line into suitable container.

b. Disconnect upper end of oil line and remove line.

c. Cap oil lines and all open lines to prevent entry of foreign matter.

4-212. Cleaning — Oil Supply System Hoses, Fittings, and Tubing. Clean hoses, fittings and tubing by flushing with solvent (C94) and air-dry or blow dry.

4-213. Inspection (Removed) — Oil Supply System Hoses, Fittings, and Tubing. Inspect and reject oil tubes for the following:

a. Kinks or dents that could obstruct oil flow. **DENTS ARE ALLOWABLE UP TO 0.015 INCH DEEP.**

b. Cracked or broken tubes or coupling nuts.

c. Cross-threaded, crushed, or otherwise damaged coupling nuts.

d. **CHAFING WITHIN CLAMP AREAS IN EXCESS OF 0.010 INCH.** No chafing is allowed at or near the flared tubing end.

e. **NICKS THAT ARE DEEPER THAN 0.010 INCH.**

4-214. Installation — Oil Supply System Hoses, Fittings, and Tubing. *a.* Remove caps or plugs from fittings, oil hoses and tubes.

b. Install lower end of oil line but do not tighten.

c. Install upper end of oil line.

NOTE

During tightening, rotate coupling nut of hose or tubing with one wrench while holding the fitting to which it is being attached securely with another wrench. This procedure will prevent twisting and possible deformation of the hose or tubing.

d. Tighten No. 4 and No. 5 coupling nuts in accordance with TM 55-1500-204-25/1.

e. Refill oil system with lubricating oil (C67).

SECTION VII POWER CONTROLS

4-215. POWER CONTROLS.

4-216. Description — Power Controls. The power controls consist of the gas producer (N1) controls and the power turbine (N2) governor controls. (See fig. 4-24.) The mechanically (throttle) operated gas producer controls actuate the gas producer control lever that schedules the quantity of fuel metered to the engine. The electrically (beeper switch) operated governor trim actuator and the mechanically (collective) operated droop compensator change the position of the power turbine governor control lever that regulates the operating rpm of the engine. The N2 trim actuator, a linear electro-mechanical actuator, allows N2 speed to be varied over a range of approximately 94 to 105 percent. Any change in actuator length or in collective pitch actuates the governor linkage and establishes a new power output demand on the engine. (For all practical purposes, main rotor speed is held constant.) Operating stroke of the actuator ram is approximately 1.00 inch and travel time from full retraction to full extension is approximately 7 seconds. The actuator is lubricated by the manufacturer and no further lubrication is required for the life of the actuator. The control rod assemblies consist of aluminum alloy tubes or solid rods with spherical bearing rod ends. The bellcranks and supports all are cast magnesium, each bellcrank containing a roller bearing.

4-217. Inspection — Power Controls. *a.* Check all control rods, bellcranks, supports, and links for worn or binding bearings, cracks, stripped threads, deformed pivot lugs, corrosion, or other obvious damage.

b. Check all switches, wires, connectors, and insulation. Check trim actuator bond jumper (if installed) for secure attachment and signs of corrosion.

4-218. Troubleshooting — Power Controls. Refer to table 4-17.

4-219. GAS PRODUCER (N1 FUEL CONTROL) LINKAGE RIGGING.

4-220. Adjustment — Gas Producer (N1 Fuel Control) Linkage Rigging. (See fig. 4-25.)

CAUTION

When tightening a jam nut to secure a control rod end, always hold the rod end with a wrench to prevent jamming of the bearing against the fitting. Check that all misalignment is divided equally between bearings of each control rod assembly.

NOTE

Adjustment steps a through d of this procedure are necessary only when; the underseat linkage or the fuselage-routed control rod has been removed and reinstalled or replaced; or, with governo,

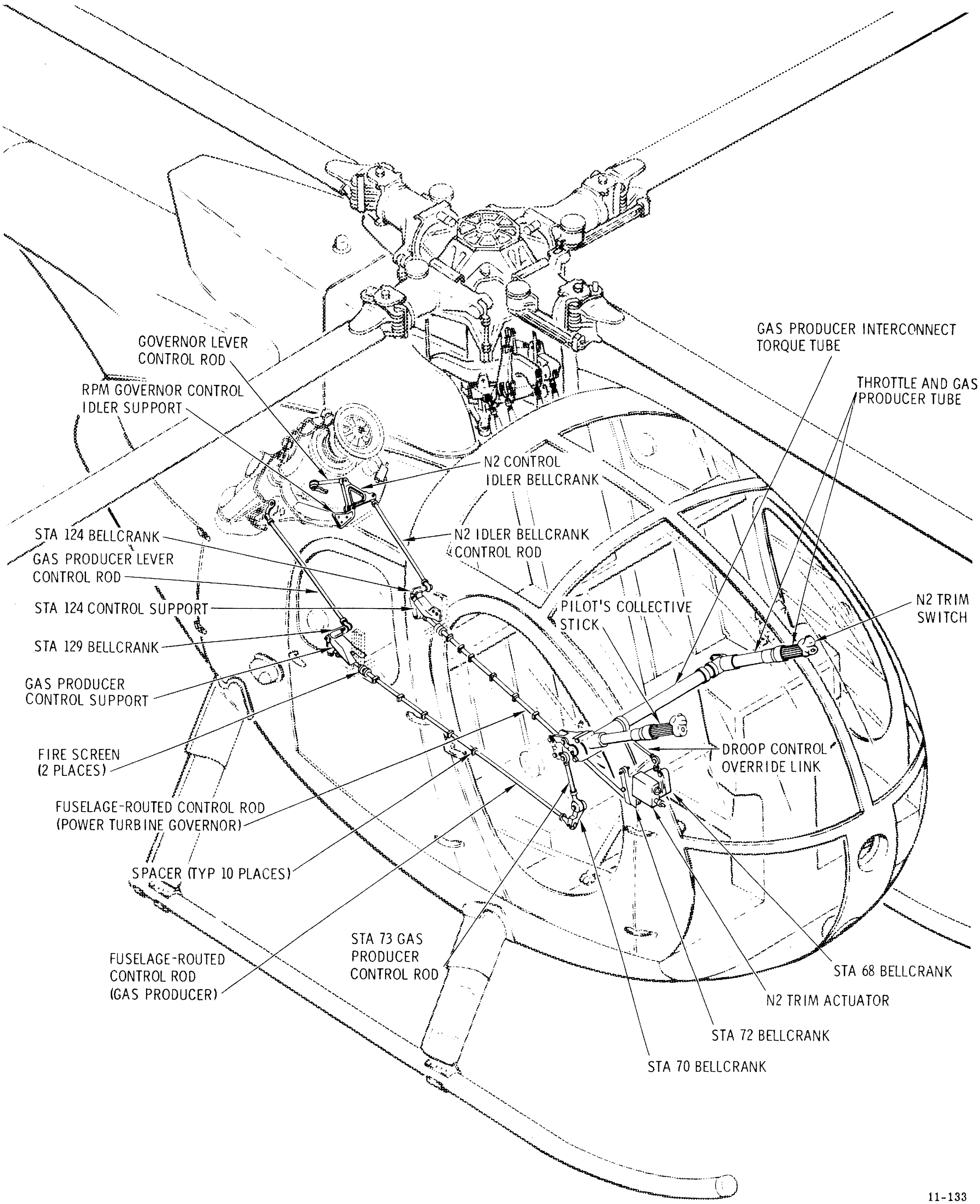


Figure 4-24. Engine Power Controls System.

Table 4-17. Troubleshooting of the Power Controls.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Movement of control results in springy feeling; response lags behind control movement.**

STEP 1. Check for loose gas producer interconnect torque tube in collective stick gearshafts.

If the gas producer interconnect torque tube is found to be loose, tighten pipe plug in the torque tube for zero backlash (para chapter 11).

STEP 2. Check for worn rod ends.

If a rod end is found to be worn, replace worn rod end.

STEP 3. Check for worn bellcrank bearings.

If a bellcrank bearing is found to be worn, replace bellcrank.

2. **Control action stiff.**

STEP 1. Check for binding rod ends at connection fittings.

If the rod ends are found to be binding, realign rod end bearings.

3. **Low engine power; improper idling speed; improper acceleration; variable power output at constant setting.**

STEP 1. Check for incorrectly adjusted gas producer or power turbine governor controls (para 4-215).

If the gas producer or power turbine governor controls are found to be incorrectly adjusted, adjust the gas producer or power turbine governor controls (para 4-215).

NOTE

If the gas producer (throttle) controls are adjusted, the N2 disable switch must be readjusted (para 4-244).

STEP 2. Check for a defective gas producer control assembly or power turbine governor (TM 55-2840-231-24).

STEP 3. Check for defective engine (TM 55-2840-231-24).

(2524247) installed, if there is less than 0.030-inch clearance between the lower firewall bellcrank and the support when the governor lever is at the maximum stop. Otherwise, perform adjustment of power turbine governor controls starting with step e.

a. Position the collective stick and throttle at mid-travel.

b. Adjust the length of the station 73 control rod so that the long arm of station 70 bellcrank is vertical.

c. Adjust the length of the fuselage-routed gas producer control rod so that the bolt center in the lower arm of station 129 bellcrank is 5.00 ±0.03 inches from the firewall.

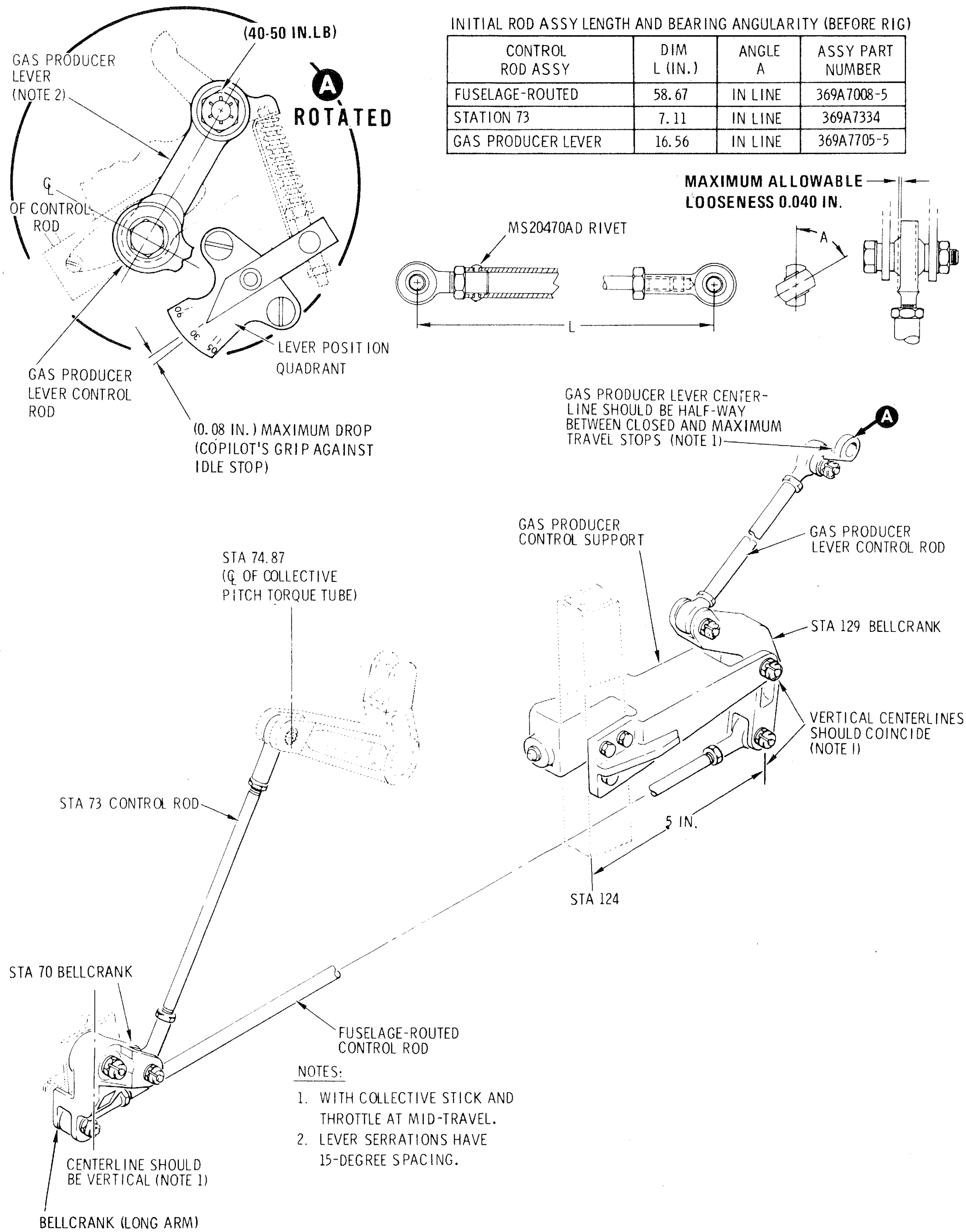
CAUTION

When repositioning the gas producer lever in *d* below, DO NOT EXCEED THE LEVER NUT TORQUE LIMIT OF 40 - 50 INCH-POUNDS. Overtorquing will cause binding of the lever.

d. With the engine compartment gas producer control rod length set to 16.56 inches, reestablish the gas producer lever mid-position so that lever centerline is half-way between the closed and maximum travel stops; lever serrations have 15 degree spacing. (The

INITIAL ROD ASSY LENGTH AND BEARING ANGULARITY (BEFORE RIG)

CONTROL ROD ASSY	DIM L (IN.)	ANGLE A	ASSY PART NUMBER
FUSELAGE-ROUTED	58.67	IN LINE	369A7008-5
STATION 73	7.11	IN LINE	369A7334
GAS PRODUCER LEVER	16.56	IN LINE	369A7705-5



NOTES:

1. WITH COLLECTIVE STICK AND THROTTLE AT MID-TRAVEL.
2. LEVER SERRATIONS HAVE 15-DEGREE SPACING.

12-111B

Figure 4-25. Gas Producer Controls Rigging.

closed stop position is parallel to the engine compartment door frame within ±5 degrees.)

e. Lower collective stick to full down.

f. Rotate pilot's throttle counterclockwise to maximum rpm, then clockwise to idle.

g. With pilot's throttle held at idle, adjust upper rod end of gas producer lever control rod until pointer is positioned at 30 to 31 degrees on quadrant (detail A).

h. Release the pilot's throttle and check the gas producer pointer with the copilot's throttle held lightly but firmly against the pilot's idle stop **(NOT MORE THAN 10 INCH-POUNDS TORQUE).** The pointer must not drop more than 0.08 inch below the 30-degree mark on the quadrant.

NOTE

As a final step of adjustment, the N2 disable switch mounted on the base of the pilot's collective pitch stick must be adjusted. Refer to paragraph 4-241 for aircraft with automatic restart system.

CAUTION

Do not alter gas producer lever position or control linkage to adjust idle speed. Idle speed will be checked during flight test and, if necessary, adjusted by the fuel control idle screw, not by the linkage. Check extreme travel position of the linkage to ensure that gas producer lever hits the fuel control stops before the throttle grip stops make contact. There must be no interference or binding in the linkage.

i. Raise pilot's throttle idle setting, if required, but do not exceed the limit specified in *r* below.

j. Rotate pilot's throttle counterclockwise to maximum rpm.

k. Observe gas producer lever to ensure it strikes the maximum travel stop.

l. Raise collective stick to full up.

CAUTION

Do not change the minimum and maximum stops. These stops are flow bench settings and are not field adjustable.

m. If gas producer lever does not strike maximum travel stop, check gas producer controls for binding.

n. Rotate pilot's throttle fully clockwise to fuel off. Observe gas producer lever to ensure it strikes minimum travel stop (closed position).

o. Lower collective stick to full down. Observe gas producer lever to ensure it does not move from the minimum travel stop. Recheck the static rigging for correct adjustment from the copilot's side.

p. At the full closed and full open positions of the throttle, check all the movable linkage for clearance with supporting or adjacent parts. Check that control rod bearings are not jammed when the linkage is in the extreme control positions.

q. Ground run the engine at idle with main rotor blades in flat pitch.

r. With pilot's throttle at idle, observe N1 tachometer indicator to ensure N1 speed stabilizes. Adjust N1 speed with the fuel control idle speed adjustment screw (TM 55-2840-231-24) if required to obtain 62-65 percent.

CAUTION

Be sure to check N1 idle speed with the copilot's throttle after completing adjustment of idle for the pilot's throttle. N1 speed must not drop below 62 percent when the copilot's throttle is set to idle. Refer to *h* and *i* above.

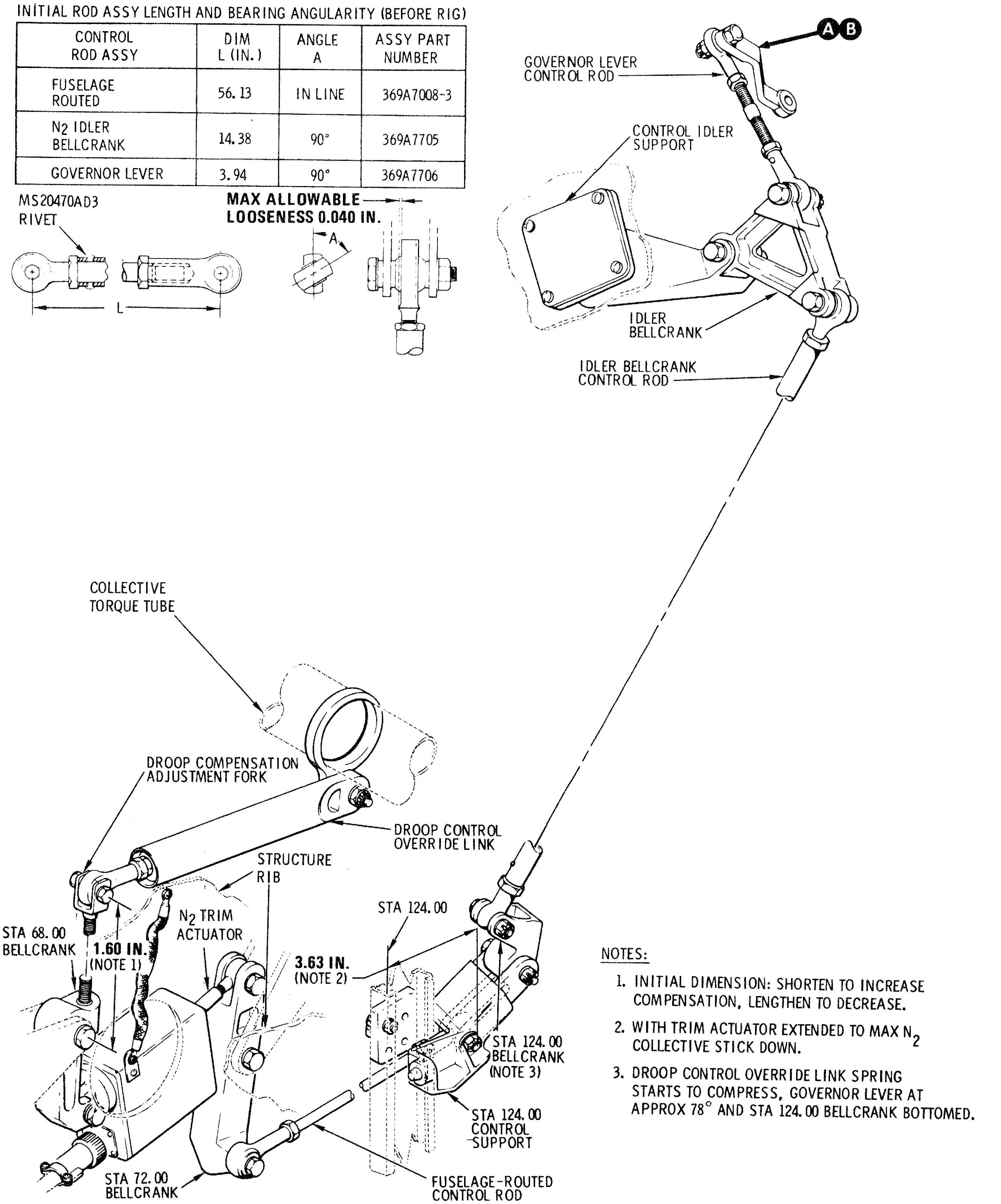
s. If idle rpm does not stabilize and/or cannot be adjusted within limits in *r* above, proceed with engine troubleshooting in TM 55-2840-231-24.

4-221. POWER TURBINE GOVERNOR (N2 RPM) CONTROL LINKAGE RIGGING.

4-422. Adjustment — Power Turbine Governor (N2 RPM) Control Linkage Rigging. (See fig. 4-26.)

INITIAL ROD ASSY LENGTH AND BEARING ANGULARITY (BEFORE RIG)

CONTROL ROD ASSY	DIM L (IN.)	ANGLE A	ASSY PART NUMBER
FUSELAGE ROUTED	56.13	IN LINE	369A7008-3
N2 IDLER BELLCRANK	14.38	90°	369A7705
GOVERNOR LEVER	3.94	90°	369A7706

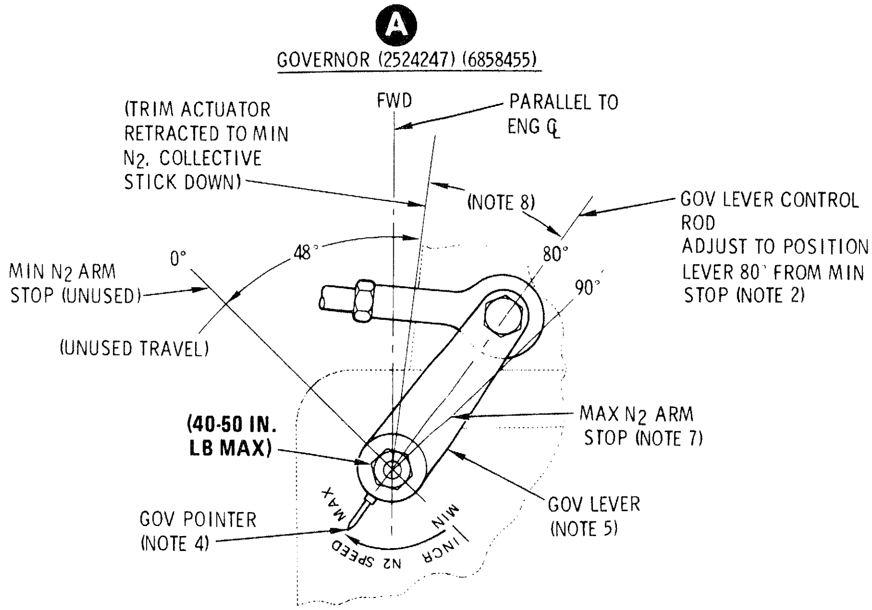


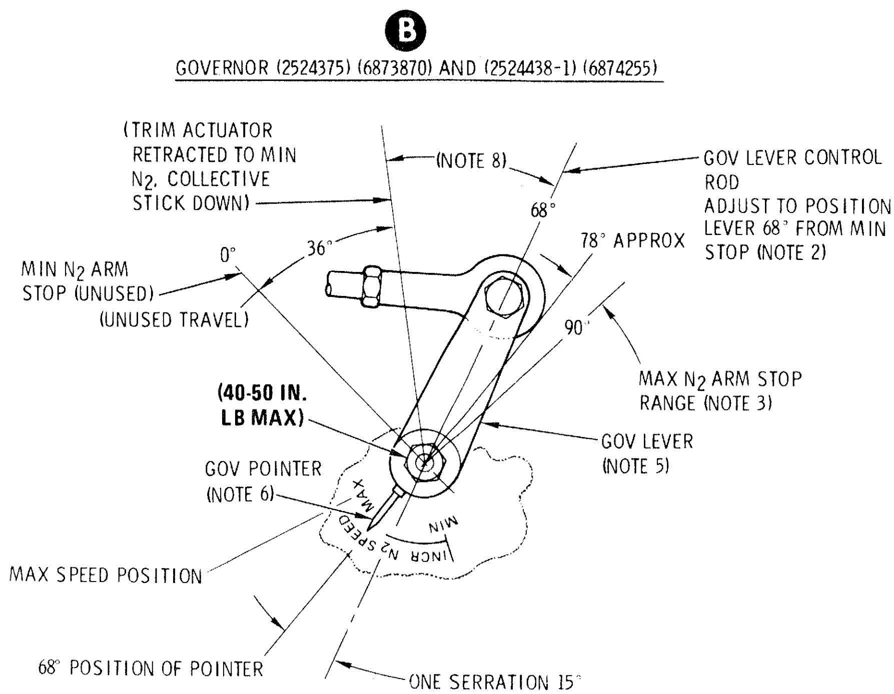
NOTES:

1. INITIAL DIMENSION: SHORTEN TO INCREASE COMPENSATION, LENGTHEN TO DECREASE.
2. WITH TRIM ACTUATOR EXTENDED TO MAX N_2 COLLECTIVE STICK DOWN.
3. DROOP CONTROL OVERRIDE LINK SPRING STARTS TO COMPRESS, GOVERNOR LEVER AT APPROX 78° AND STA 124.00 BELLCRANK BOTTOMED.

12-207-1B

Figure 4-26. Power Turbine Governor Controls Rigging. (sheet 1 of 2)





NOTES: (CONT)

4. WITH LEVER AT 80°, POINTER MUST ALIGN WITH GOVERNOR CENTERLINE.
5. LEVER SERRATIONS HAVE 15° SPACING.
6. WITH LEVER AT 68°, POINTER TIP MUST ALIGN BETWEEN LETTERS P & E OF WORD SPEED.
7. DROOP CONTROL OVERRIDE LINK SPRING STARTS TO COMPRESS.
8. TRAVEL LIMIT OF TRIM ACTUATOR IS 32 DEGREES.

12-207-2C

Figure 4-26. Power Turbine Governor Controls Rigging. (sheet 2 of 2)

CAUTION

When tightening a jam nut to secure a control rod end, always hold the rod end with a wrench to prevent jamming of the bearing against the fitting. Check that all misalignment is divided equally between bearings of each control rod assembly.

NOTE

Adjustment steps a through d of this procedure are necessary only when; the underseat linkage or the fuselage-routed control rod has been removed and reinstalled or replaced; or, with governor (2524247) installed, if there is less than 0.030-inch clearance between the lower firewall bellcrank and the support when the governor lever is at the maximum stop. Otherwise, perform adjustment of power turbine governor controls starting with step e.

a. Check that droop control override link total end play does not exceed 0.015 inch.

b. Check that the droop compensation adjustment fork is adjusted so that the dimension between the centerlines of the station 68 bellcrank pivot bolt and the fork is 1.60 ±0.03 inches (sh 1).

c. Connect an external power source. Check that the N2 trim actuator is adjusted so that the dimension between attach bolt centers is 5.47 ±0.03 inches with actuator at maximum extension.

d. With the pilot's collective stick full down and the N2 trim actuator at maximum extension, adjust the length of the fuselage-routed governor, adjust the length of the fuselage-routed governor control rod so that the bolt center in the lower arm of station 124 bellcrank is 3.63 ±0.03 inches (sh 1) from the firewall.

e. Check that the engine compartment N2 bellcrank control rod length is set to 14.38 inches (sh 1).

f. Check position of governor lever and pointer. On governor (2524247), the lever pointer must align with the lever centerline (detail A, sh 2). On governors (2524438-1 and 2524375) the lever point must be positioned one serration clockwise (away from engine) from lever centerline (detail B, sh 2). Reposition lever and pointer, if necessary, starting with both the lever and governor shaft at the minimum stop; lever serrations have 15-degree spacing. **DO NOT EXCEED THE LEVER NUT TORQUE LIMIT OF 40-50 INCH-POUNDS** or the lever will bind.

NOTE

Approximately 48 degrees of governor lever travel from the minimum N2 stop is unused on governor (2524247); 36 degrees is unused on governors (2524438-1 and 2524375).

g. Adjust governor lever control rod length until the governor control lever is 80 degrees from lever minimum stop (detail A, sh 2), or 68 degrees from minimum stop (detail B, sh 2), as applicable. The 80-degree position is the center of the letter D in the word SPEED embossed on the power turbine governor. The 68-degree position is when the pointer tip is between the letters P and E in the word SPEED.

h. Raise the pilot's collective stick to approximately one-third of full up travel and note below.

(1) With governor (2524247) installed (detail A): When the N2 trim actuator is fully extended, the governor lever may hit the maximum travel stop when the collective stick is raised to one-third travel. At this stick position, the droop control override link spring starts to compress and spring compression continues to increase until the stick reaches full up travel.

(2) With governors (2524438-1 or 2524375) installed (detail B): When the N2 trim actuator is fully extended and the collective stick is raised to one-third travel, check for one of two conditions — that the governor lever hits the maximum travel stop without any interference or binding in the linkage, or that the pointer reaches the N2 maximum speed position (approximately 78°) before the linkage (lower arm of sta 124 bellcrank) bottoms out.

i. Lower collective stick to full down and decrease N2 trim to minimum. Check that the governor lever is at the approximate midpoint (48 or 36 degrees, as applicable) between the minimum and maximum stops.

j. At the full up and full down positions of the collective stick, check all movable linkage for clearance with supporting or adjacent parts. (Note the exception in *h* above.) Check that the control rod, N2 actuator and override link bearings are not jammed when the linkage is in the extreme control positions.

k. Ground run engine at idle with the main rotor blades in flat pitch.

l. Rotate pilot's throttle counterclockwise to maximum rpm.

m. Decrease N2 trim to minimum.

n. Observe N2 tachometer pointer for 100 percent or less when rotor rpm pointer is superimposed on N2 pointer, and N1 tachometer pointer is stabilized.

o. With collective stick down, increase N2 trim to maximum.

p. Observe N2 tachometer pointer for 104 percent minimum, 105 percent maximum, when rotor rpm pointer is superimposed on N2 pointer and N1 tachometer pointer is stabilized.

CAUTION

Make all engine compartment control rod adjustments with the engine shut down.

q. Stop the engine. Adjust the length of the governor lever control rod until N2 trim is within the limits given in *n* and *p* above. Adjust one end not more than one turn at a time.

r. Restart the engine, actuate N2 trim for 101 percent with main rotor blades in flat pitch, and lift off and hover.

s. Observe N2 tachometer pointer when pointer of N1 tachometer is stabilized. N2 pointer indication (droop compensation) should be 1.50 to 2.0 percent above N2 setting in *r* above.

t. Land and shut down engine.

u. If droop compensation does not occur as specified in *s* above perform the following:

(1) Recheck power turbine governor (N2 rpm) control linkage adjustment *a* through *j* above.

(2) Check gas producer (N1 fuel control) linkage adjustment, paragraph 4-220 above.

WARNING

Any change to the initial droop compensation fork length established in *b* above may result in interference between the N2 trim actuator case and the fork threaded end. Operate the collective through full travel after adjustment and check for clearance between movable linkage and adjacent parts.

(3) If N1 and N2 linkages are correctly adjusted, an additional adjustment may be made at the droop compensation adjustment fork (fig. 4-26, sh 1). Shorten the droop compensation adjustment fork to increase compensation. Three to five turns of the droop compensation fork may be necessary to change the droop compensation approximately 0.5 percent.

4-223. GAS PRODUCER (N1 FUEL CONTROL) LINKAGE.

4-224. Removal — Gas Producer (N1 Fuel Control) Linkage. (See fig. 4-27.) Disassemble the linkage only to the extent necessary to remove worn or damaged parts, or to perform repairs.

a. Remove cotter pins, nuts, washers and bolts, as necessary, to disconnect the appropriate control rod, bellcrank or support.

b. Remove the control support from the firewall by removing the three standard bolts, washers and nuts from the vertical stiffener; then remove the pull-type lockbolt from the firewall according to *c* and *d* below.

c. Locate the lockbolt installation at the floorline of the cargo compartment. Split the collar axially with a sharp cold chisel. Use a backup bar on the collar opposite to the side being split, to prevent deformation or breakout of the hold.

d. Drive the lockbolt pin out with a drift.

NOTE

If the lockbolt is carefully removed, another lockbolt of the same diameter may be used as a replacement. If the hole is enlarged to more than 0.187 inch, an oversize lockbolt must be installed.

4-225. Inspection — Gas Producer (N1 Fuel Control) Linkage. *a.* Check rod end bearings for binding and excessive wear **(0.040-INCH MAXIMUM AXIAL LOOSENESS).**

b. Check control rod for surface damage and evidence of bending.

c. Check for loose rivet at fixed rod end (lower end).

d. Check bearings in bellcranks for binding.

e. Perform a fluorescent penetrant inspection TM 55-1500-204-25/1 on any questionable part.

4-226. Repair — Gas Producer (N1 Fuel Control) Linkage. *a.* Perform a straightness check on a control rod that appears bent or bowed. The total length of gas producer control rods (excluding rod ends) must be straight within the following tolerances, with **STRAIGHTNESS VARIATION LIMITED TO A MAXIMUM OF 0.010 INCH IN EACH FOOT OF LENGTH.**

(1) Fuselage-routed control rod: 0.050 inch.

(2) Station 73 control rod: 0.010 inch.

(3) Gas producer lever control rod: 0.020 inch.

WARNING

A dye-check for cracking shall always be performed after cold-straightening

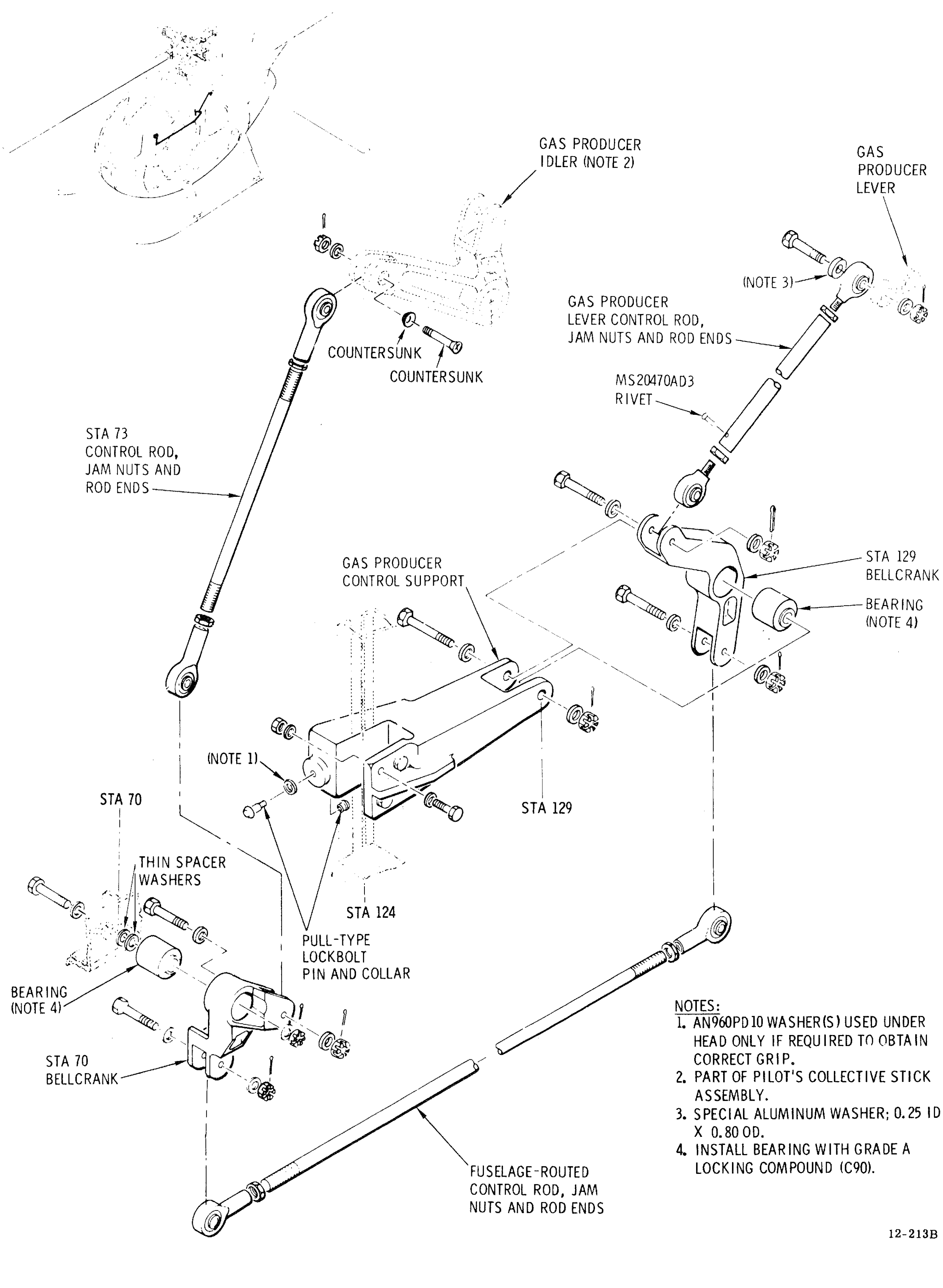


Figure 4-27. Gas Producer Control Linkage Disassembly.

(TM 55-1500-204-25/1). Replace a cracked rod, or a cracked or bent rod end.

b. Cold-straighten a bent rod that is not within tolerance, *a* above, provided there are no nicks or sharp dents in the bend length. Do NOT use the rod ends to support the rod during the straightening process.

CAUTION

Use care when drilling to remove or install a riveted rod end; the rod end is steel and the rod is aluminum. Tighten the jam nut before drilling and riveting.

c. **REPLACE A CONTROL ROD END IF BEARING AXIAL PLAY IS MORE THAN 0.040 INCH.** Set initial control rod length and bearing angularity as shown in figure 4-26.

d. Replace unserviceable bellcrank bearings. Install a new bearing of the correct type with surface primer and grade A locking compound (C91) and (C90), according to container instructions.

e. Replace bellcranks or control supports for distortion, cracks, or elongated holes.

4-227. Installation — Gas Producer (N1 Fuel Control) Linkage. (See fig. 4-27.) *a.* Position the control support on the firewall. Align the three standard bolt holes and the lockbolt hole with the matching holes. Install but do not tighten the three standard bolts, washers, and nuts.

NOTE

If the lockbolt hole diameter in the firewall does not exceed 0.187 inch, a standard diameter lockbolt pin can be used. If the hole is oversize and can be cleaned up to 0.201 inch maximum, use an oversize 0.187-inch lockbolt.

b. Install the replacement lockbolt pin and collar; best fit results when the compressed air line pressure to the pulling gun is 90 to 125 psi. Use up to three AN960PD10 washers under the pin head, if necessary, to obtain correct lockbolt grip.

c. Visually check the lockbolt installation. No measurable gap is permissible under either the pin head or collar. If the pin grip length or completeness of the collar swaging is questionable, use a lockbolt inspection gage or an equivalent inspection gage per NAS1563 to check for proper installation. Replace an incorrectly installed lockbolt.

d. Tighten the three nuts that attach the control support to the firewall.

CAUTION

When tightening a jam nut to secure a control rod end, always hold the rod end with a wrench to prevent jamming of the bearing against the fitting. Check that all misalignment is divided equally between the bearing of each control rod assembly.

NOTE

Before installing any control rod, measure the rod assembly for correct length (fig. 4-26).

e. Install control rod or bellcrank with bolt, washers, nut, and new cotter pin as shown in figure 4-27.

4-228. POWER TURBINE (N2 RPM) CONTROL LINKAGE.

4-229. Removal — Power Turbine (N2 RPM) Control Linkage. Remove control rods, bellcranks or supports according to paragraph 4-224 and figures 4-28 and 4-29. There are two exceptions: removal of the trim actuator and station 72 bellcrank as a unit, and removal of the droop control override link assembly.

NOTE

Only high temperature metal nuts (not fiber insert type) are to be used for anchoring control idler support (fig. 4-29) to engine gearcase accessory pad.

a. (See fig. 4-28.) Remove the droop control override link by disconnecting the link housing end fitting from the collective torque tube, and the rod end from the droop compensation fork.

b. Remove the N2 trim actuator and station 72 bellcrank as a unit according to *c* through *f* below.

c. Disconnect the bonding jumper and electrical connector from the actuator.

d. Disconnect the actuator from station 68 bellcrank.

e. Disconnect station 72 bellcrank from the fuselage-routed control rod end.

f. Remove the pivot bolt that attaches station 72

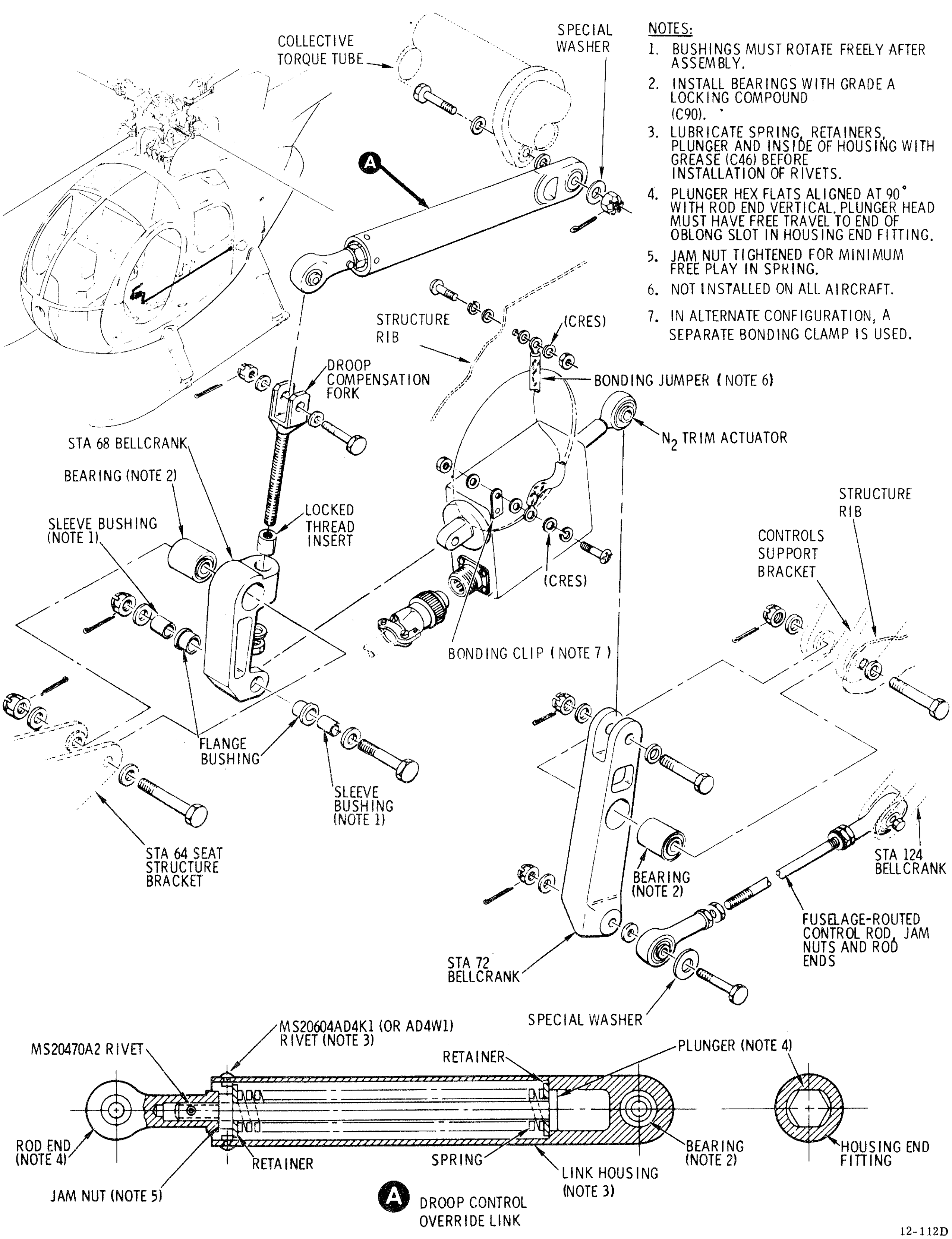


Figure 4-28. Governor Control Linkage (Forward of Engine Compartment) Disassembly.

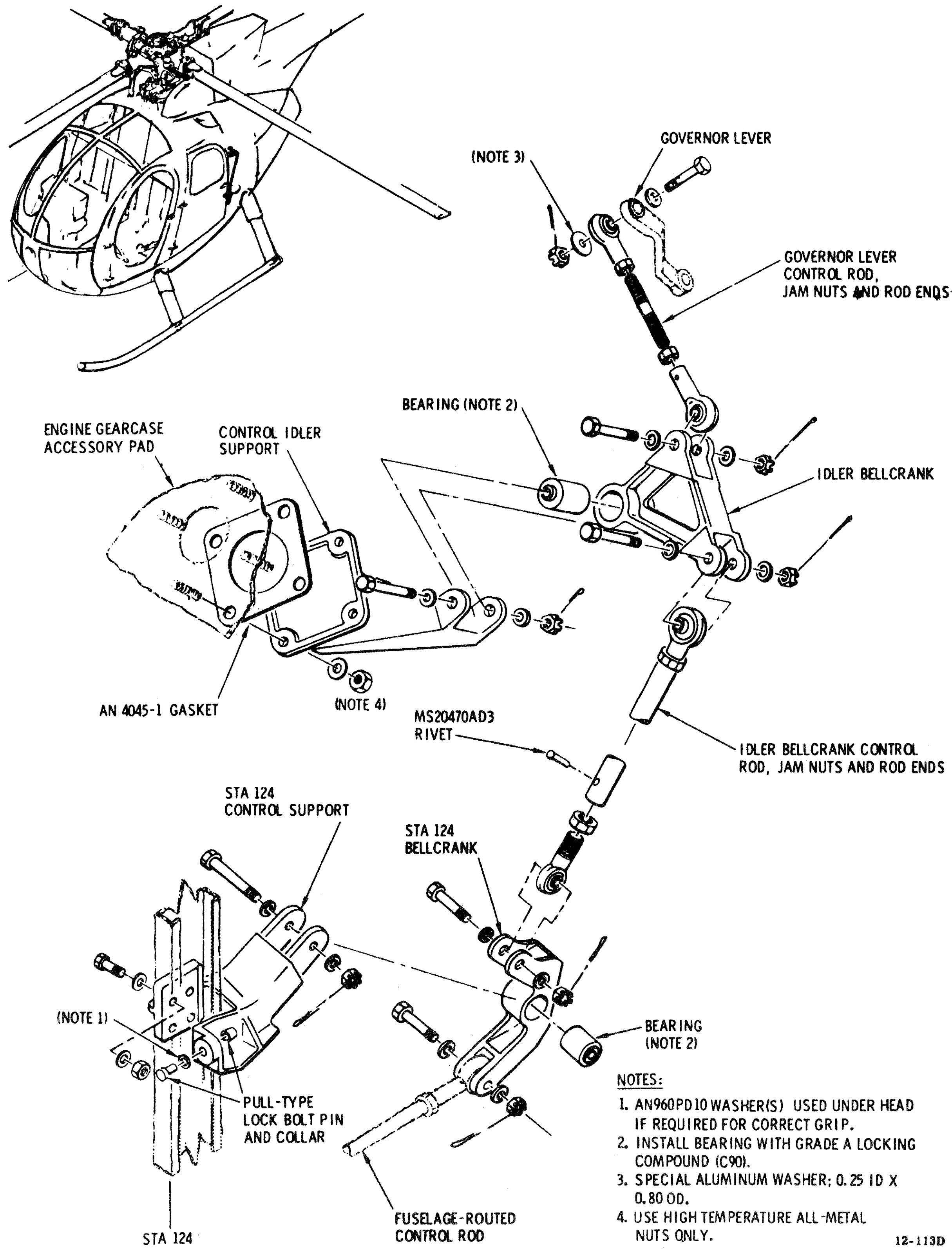


Figure 4-29. Governor Control Linkage (Engine Compartment) Disassembly.

bellcrank to structure rib and controls support bracket; remove actuator and bellcrank.

g. Disconnect the trim actuator from station 72 bellcrank. Use care to keep sleeved bushings with bellcrank.

4-230. Inspection — Power Turbine (N2 RPM) Control Linkage. *a.* Check rod end bearings for binding and excessive wear **(0.040-INCH MAXIMUM AXIAL LOOSENESS).**

b. Check control rod for surface damage and evidence of bending.

c. Check for loose rivet at fixed rod end (lower end).

d. Check bearings in bellcranks for binding.

e. Perform a fluorescent penetrant inspection (TM 55-1500-204-25/1) on any questionable part.

4-231. Repair — Governor Control Rods and Bellcranks. *a.* Perform a straightness check on a control rod that appears bent or bowed. The total length of governor control rods (excluding rod ends) must be straight within the following tolerances, with **STRAIGHTNESS VARIATION LIMITED TO A MAXIMUM OF 0.010 INCH IN EACH FOOT OF LENGTH.**

(1) Fuselage-routed control rod: 0.050 inch.

(2) Governor lever control rod: 0.010 inch.

(3) N2 idler bellcrank control rod: 0.020 inch.

b. Rods which exceed the straightness Limitations listed above will be replaced.

CAUTION

Use care when drilling to remove or install a riveted rod end; the rod end is steel and the rod is aluminum. Tighten the jam nut before drilling and riveting.

c. **REPLACE A CONTROL ROD END IF BEARING AXIAL PLAY IS MORE THAN 0.040 INCH.** Set initial control rod length and bearing angularity as shown in figure 4-26.

d. Replace unserviceable bellcrank bearings. Install a new bearing with surface primer and grade A locking compound (C91) and (C90), according to container instructions.

e. Replace bellcranks or control supports for distortion, cracks or elongated holes.

4-232. Repair — Droop Control Override Link Assembly. (See fig. 4-28.) *a.* Replace an unserviceable link housing end fitting bearing (detail A). Install a new bearing with surface primer and grade A locking compound (C91) and (C90), according to container instructions.

b. Replace an unserviceable rod end or broken spring according to *c* through *i* below.

c. Remove the four rivets from rod end of housing.

d. Drill out rod end rivet and disassemble spring assembly. Discard rod end and plunger.

e. Assemble new plunger, retainers, spring and jam nut on plunger threads until there is minimum free play of the spring.

f. Install replacement rod end to point of contact with jam nut. Check alignment of the flats on the plunger head and the rod end; with rod end vertical, two of the flats must be perpendicular at 90 degrees (detail A). Tighten jam nut against rod end. Check that there is no more than 0.010 inch free play in spring.

NOTE

When the override link assembly is installed between the droop compensation fork and the collective torque tube, the plunger must be free to travel to the end of the oblong slot in the link housing end fitting without binding.

g. Using a No. 50 drill, drill through rod end and plunger at the witness hole of the rod end. Install an MS20470A2 rivet of correct length.

h. Lubricate the interior of the housing, the spring retainers and plunger with grease (C46). Install the spring assembly in the housing so that plunger hex enters end fitting slot and rod end aligns vertically with housing bearing (detail A, fig. 4-28).

i. Complete the link assembly by installing the four retention rivets (MS20604AD4K1 or W1) in the housing (detail A).

4-233. Installation — Power Turbine (N2 RPM) Control Linkage. (See fig. 4-28 and 4-29.) *a.* Check a replacement trim actuator for correct **MAXIMUM EXTENDED LENGTH OF 5.47 ±0.03 INCHES BETWEEN ATTACH BOLT CENTERS.** If dimension is not correct, test actuator for proper operation (para 4-237).

b. Assemble original or replacement trim actuator to original or replacement station 72 bellcrank with a bolt, two washers, nut and new cotter pin.

c. Check that both sleeve bushings are in place in station 68 bellcrank arm before connecting the actuator; then install assembled unit in controls support bracket and connect linkage as shown in figure 4-28. The sleeve bushings in station 68 bellcrank arm must rotate freely, without any binding, after the actuator is attached and the connecting hardware is tightened.

d. Connect replacement droop control override link to collective torque tube. Push on rod end and check that the link assembly plunger head is free to slide back and forth in the link housing end fitting. Connect rod end to droop compensation fork; the rod end should align with the fork.

4-234. RPM GOVERNOR (N2) TRIM ACTUATOR.

4-235. Removal — Rpm Governor (N2) Trim Actuator. Refer to paragraph 4-229.

4-236. Repair and Adjustment — Rpm Governor (N2) Trim Actuator. Repair of the N2 trim actuator is limited to replacement of an unserviceable rod end in the actuator ram. With the ram at full extension, set the distance between attach bold centers to 5.47 inches before reinstalling the actuator.

4-237. Test — Rpm Governor (N2) Trim Actuator. *a.* Connect the N2 trim actuator to test harness and equipment shown in figure 4-30.

b. Turn on dc power and adjust output to 25.75 to 26.25 volts.

c. With actuator ram approximately halfway between stops, set up dial test indicator and measure the ram end play. **THE END PLAY MUST NOT EXCEED 0.005-INCH TIR WHEN MEASURED UNDER A 10-POUND REVERSING LOAD.**

d. Position test switch to RETRACT and allow the actuator ram to fully retract. Using the end of the ram sleeve as a reference point, measure the portion of ram that remains out of the sleeve to the nearest 0.016 inch. Record this measurement for use in *e* below.

e. Position test switch to EXTEND and run ram to the extend stop. Measure the length of the extended ram. Subtract the measured result of *d* above, from this value. The result should be 0.97 to 1.03 inches.

f. The operating current under no-load conditions, at 26.0 vdc input should be between 0.60 to 1.25 amperes when running and 2.0 amperes maximum when stalled.

g. Operate the actuator motor in both extend and retract direction. Overtravel must not exceed 0.020 inch maximum when power is turned off.

h. Reduce the voltage input to 21 volts. The actuator ram must retract and extend to the stops without binding on the stops or between the stops.

i. Increase the input voltage to 28 volts. The actuator ram must retract and extend to the stops without binding on the stops or between stops.

j. Run the actuator to the fully extended position. The overall length of the extended actuator should be 5.47 inches between attach bolt centers. Turn power off and disconnect the actuator.

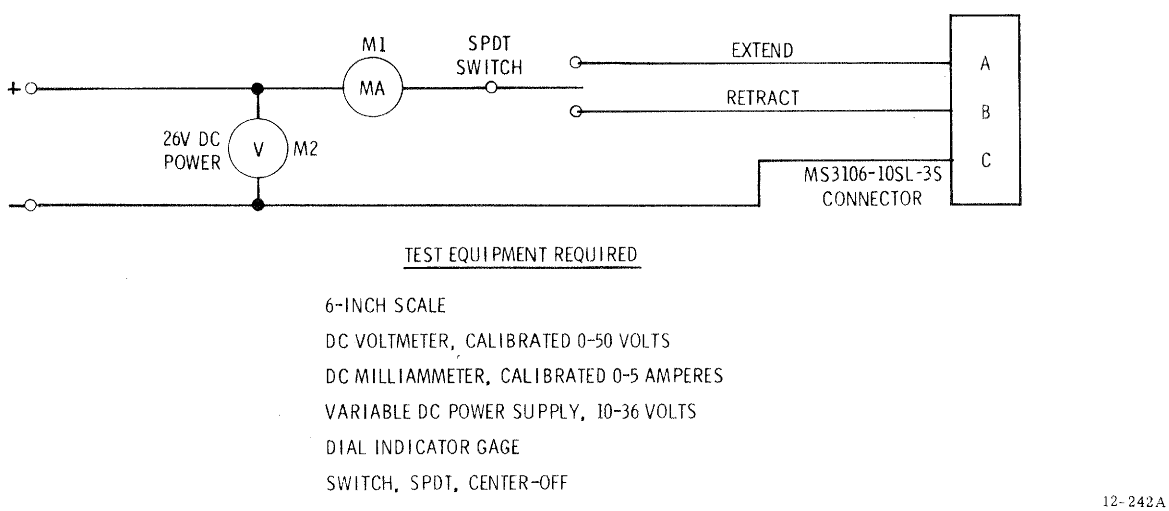


Figure 4-30. N2 Trim Actuator Test Hookup.

4-238. Installation — Rpm Governor (N2) Trim Actuator. Refer to paragraph 4-228.

SECTION VIII IGNITION SYSTEM

4-239. IGNITION SYSTEM.

4-240. General — Ignition System. Refer to TM 55-2840-231-24 for description and maintenance of the engine ignition system. Refer to paragraphs 4-241 through 4-250 for the engine ignition automatic restart system.

Table 4-18. Premaintenance Requirements for Maintenance of the Ignition System.

Conditions	Requirements
Test Equipment	(T4)
Minimum Personnel Required	Two (MOS 67V & 68F)
Consumable Materials	(C12) (C96)

4-241. Automatic Restart System Description — Ignition System. The automatic restart system (fig. 4-31) is provided to automatically energize the engine ignition system when an engine out condition is sensed by the engine power out warning unit. Engine power out warning and reignition occurs when engine N2 rpm decreases to 95 ± 1 percent or below during flight with the throttle at full open (governed) position. At other throttle positions the N2 sensing is disabled by a switch mounted on the base of the pilot's collective pitch stick. A reignition time delay of 3.5 ±0.5 seconds is provided after returning the throttle to the full open (governed) position. This prevents erroneous engine out warning and reignition while rpm is increasing to above 95 percent N2.

Engine power out warning and reignition also occurs when engine N1 rpm decreases to 55 ±5 percent or below.

NOTE

Reignition is automatically limited to a duration of 5 seconds to prevent excessive surge and to eliminate danger of post-crash fires.

An indicator light, mounted on the underside of the instrument panel hood, illuminates to indicate that reignition has occurred. The system can be reset and the light extinguished by pressing the indicator face. Reignition system power is supplied through a 5-ampere circuit breaker (AUTO RE-IGN) mounted near the switch panel. System ground is provided by the generator switch and reignition does not occur unless the generator switch is at GEN position. The system includes an isolation diode CR2, mounted near the starter relay, which allows the engine ignition system to be energized without operating the starter. Normal starting and ignition, using the start button, remains unchanged. The engine ignition exciter used with this system is Allison 6870885 (GLA43754) or Bendix 6870891. The igniter is Allison 6843984, Champion FHE161-9, or AC5611588. Refer to chapter 9 for information concerning the engine power out warning unit used with this system.

4-242. Test (Operational Check) — Automatic Restart System. *a.* Open the engine access doors and have an assistant prepare to listen for actuation of the engine ignition exciter.

b. Connect external electrical power and place power selector switch at EXT. The MASTER CAUTION, ENGINE OUT, and TRANS OIL PRESS warning lights will flash.

c. Check that RE-IGN circuit breaker is engaged and actuate generator switch to GEN. An audible warning tone will be heard in headsets, the RE-IGN indicator will illuminate, and a buzzing sound will be heard at the engine ignition exciter.

d. Check that ignition exciter continues to buzz for 5 seconds and then stops.

e. Switch generator OFF. The audible warning tone will stop and the reignition indicator light will extinguish. Close engine access doors.

NOTE

The following steps are to be performed only after maintenance involving change of the engine power out warning unit or when required for troubleshooting the complete automatic restart system.

f. Make a normal engine start in accordance with TM 55-1520-214-10. Place generator switch at GEN after engine stabilizes at idle.

g. With collective pitch full down, twist throttle smoothly to full open position. Note that reignition indicator should not illuminate if N2 rpm reaches 95 percent or more within 3.5 seconds after throttle is opened.

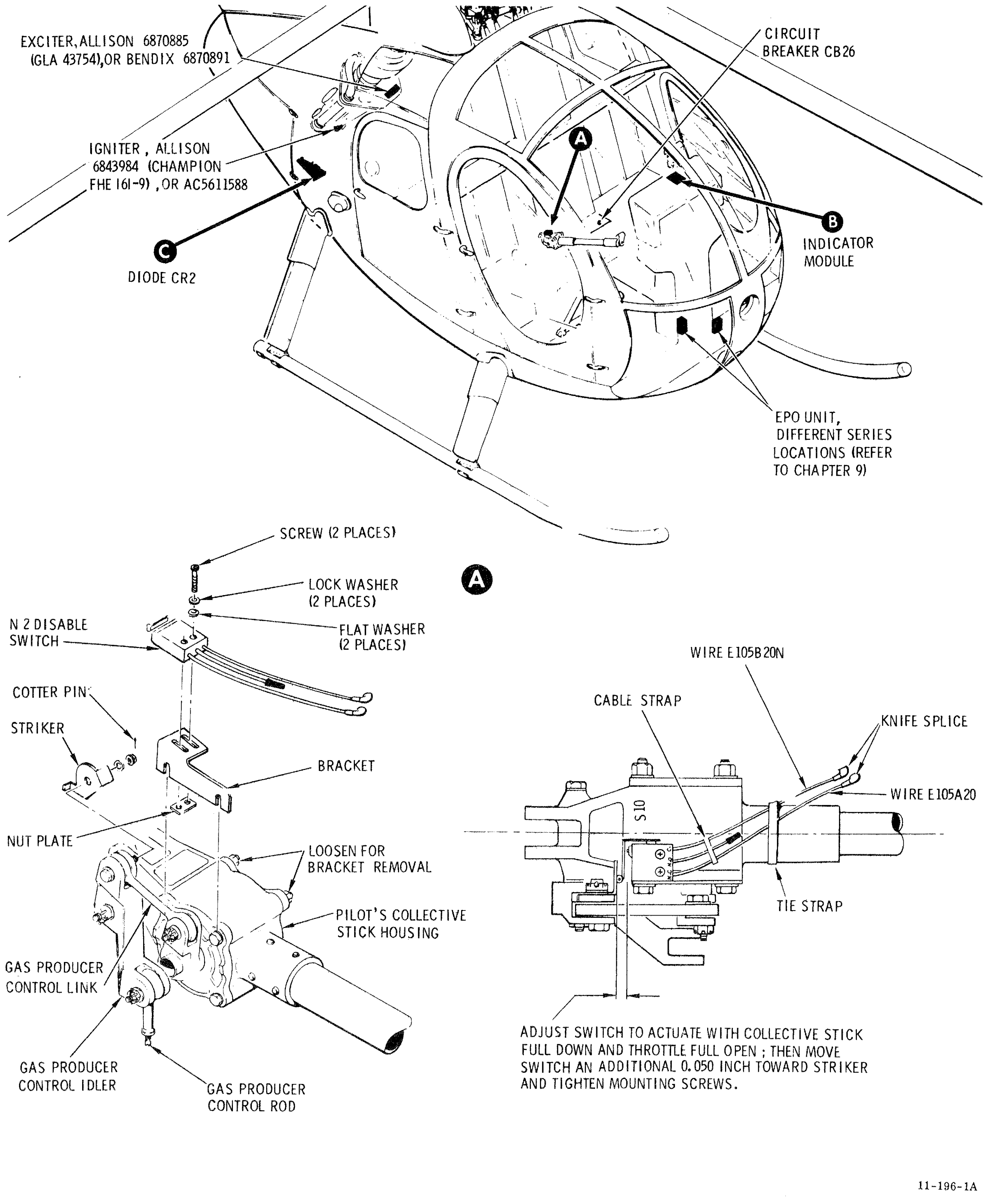
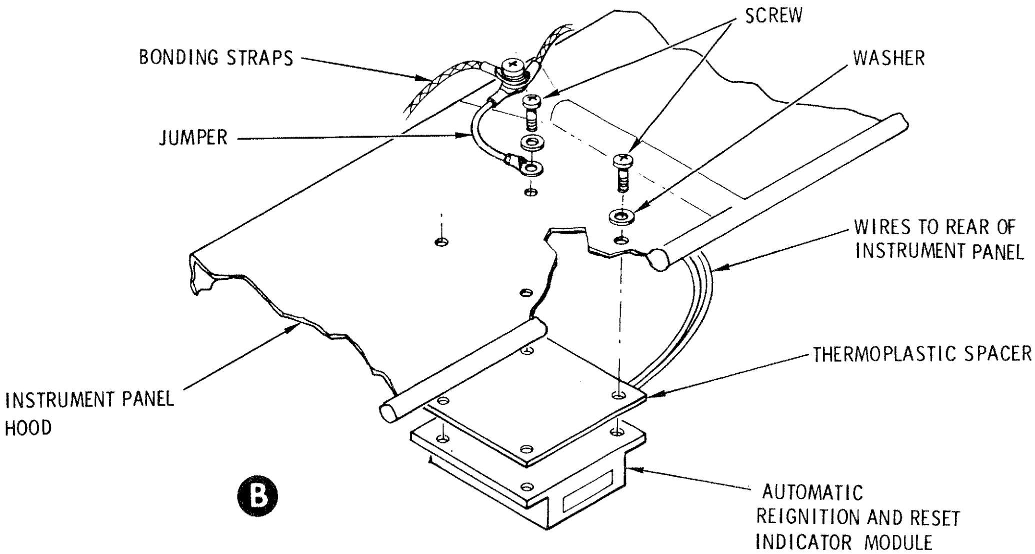
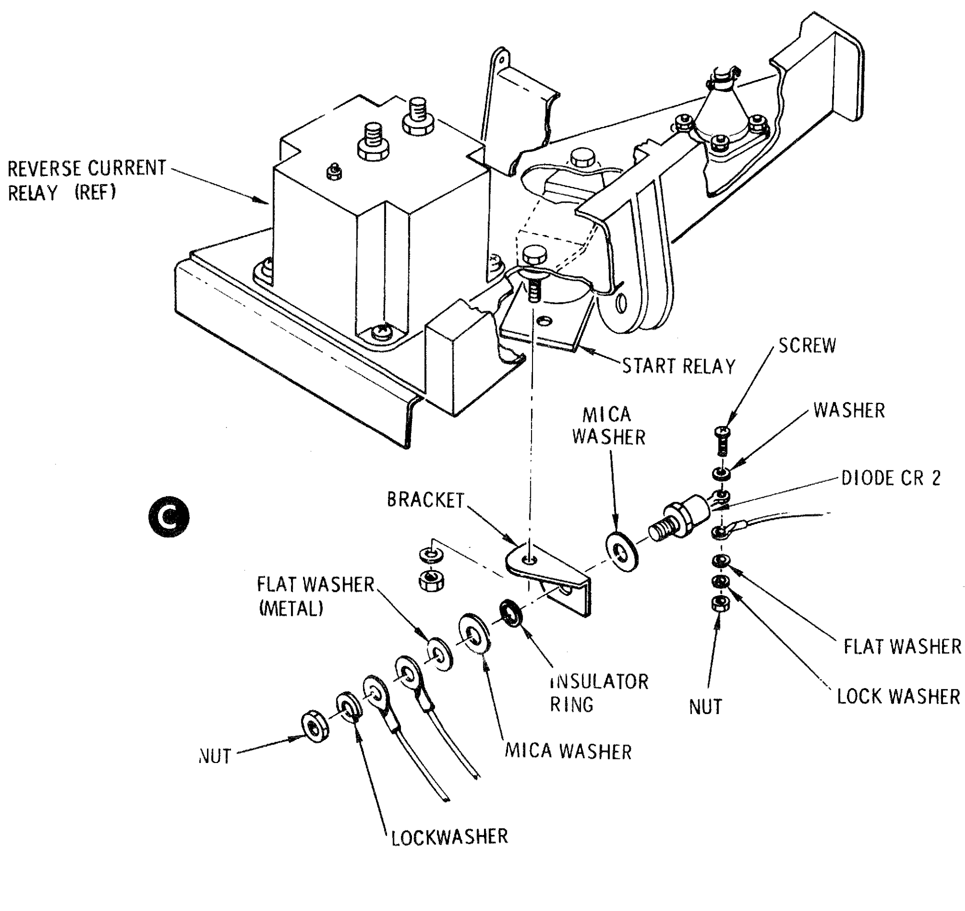


Figure 4-31. Automatic Restart System. (sheet 1 of 2)





11-196-2

Figure 4-31. Automatic Restart System. (sheet 2 of 2) *h.* Pull fuel shutoff valve to off position and allow engine flameout. As engine N2 rpm decreases to 95 percent the reignition indicator (RE-IGN) should illuminate.

CAUTION

Fuel valve must remain off until engine is completely stopped.

i. Twist throttle to idle cutoff position and push the RE-IGN indicator to reset the system.

j. Perform another start *f* above and allow engine to stabilize at idle (65 percent rpm or higher). Twist the throttle to idle cutoff and observe that RE-IGN light illuminates as N1 falls through 55 percent rpm. Complete the shutdown and remove electrical power.

4-243. Troubleshooting — Automatic Restart System. Refer to table 4-19.

4-244. N2 DISABLE SWITCH.

4-245. Adjustment — N2 Disable Switch. Adjust N2 disable switch with the collective pitch stick fully down and with throttle fully open (gas producer N1 lever on the open stop). Loosen switch mounting screws and move switch forward in mounting slots. Move switch aft, toward the striker place, until an actuating click is heard and then continue moving the switch aft an additional 0.050 inch. Tighten switch mounting screws.

NOTE

If switch actuating click cannot be heard, use a multimeter (T4) connected between splices SP-9 and SP-10 to determine actuation point.

4-246. Removal — N2 Disable Switch. (See fig. 4-31.) *a.* Gain access to the base of the pilot's collective pitch stick by removing cover. (Refer to chapter 2.) It will also be necessary to remove the lower portion of both pilot's seats.

b. Remove N2 disable switch as shown in exploded view on figure 4-30. Remove assembled switch and bracket for best access.

4-247. Installation — N2 Disable Switch. (See fig. 4-31.) *a.* Assemble N1 disable switch and support bracket as shown in figure 4-31; then install bracket and tighten collective stick housing bolts.

b. Install electrical wires and tie straps. Refer to appendix F for restart system wiring diagrams.

c. Check that tie straps are properly installed on switch wires to prevent interference with collective pitch and throttle movement.

d. Adjust disable switch according to paragraph 4-245.

e. Reinstall cover over base of collective pitch stick.

4-248. ISOLATION DIODE CR2.

4-249. Removal — Isolation Diode CR2. (See fig. 4-31.) *a.* Remove bolt attaching diode mounting bracket at base of starter relay.

b. Using a knife blade or similar tool, scrape away sealant for access to diode mounting stud and electrical terminals.

c. Remove diode from mounting bracket and disconnect electrical terminals. Retain all attaching parts.

4-250. Installation — Isolation Diode CR2. (See fig. 4-31.) *a.* Install diode in mounting bracket being sure that the diode stud is insulated from electrical contact with the bracket by installing the parts in the sequence shown in figure 4-31. Using a multimeter (T4), check that resistance between diode body and bracket is infinite.

b. Connect wire from start relay to small terminal on diode.

c. Install bolt to attach diode mounting bracket to base of start relay.

d. Perform operational check (para 4-242). Perform an engine start (TM 55-1520-214-10) to determine that normal start system functions properly.

e. Clean diode and mounting bracket with solvent (C96). Apply silicone adhesive (C12) to completely cover the diode and wire terminals. Allow adhesive to cure for 24 hours at room temperature.

Table 4-19. Troubleshooting of the Engine Automatic Restart System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Automatic engine restart functions normally, no RE-IGN indication.**

STEP 1. Check for defective indicator lamps.

If a lamp is defective, replace the lamp (chapter 9).

2. **Regular engine start is normal. Automatic engine restart does not occur when power decreases to reignite level; reignition indicator does not light. Engine power out warning normal.**

STEP 1. Check that RE-IGN circuit breaker is set (fig. 4-31).

If the generator switch is at OFF, place the switch ON.

STEP 2. Check that the generator switch is at ON (chapter 9).

If the circuit breaker is disengaged depress the circuit breaker.

STEP 3. Check for defective electrical connectors and wiring (appendix F).

If an electrical connector or the wiring is found to be defective, repair or replace defective connector or wiring.

STEP 4. Check for a defective reignition indicator module (fig. 4-31).

If the reignition indicator module is found to be defective, replace the indicator module.

STEP 5. Check for no reignition output from engine power out warning unit at connector J21 pin U.

If the engine power out warning unit is found defective refer to chapter 9.

3. **Same symptom as malfunction 2, above, except engine power out warning does not occur.**

STEP 1. Check for defective engine power out warning unit (chapter 9).

If the engine power out warning unit is defective, replace the unit.

4. **Reignition indicator light does not extinguish, on push to reset, with engine power normal.**

STEP 1. Check for defective reignition indicator module (para 4-241).

If the reignition indicator module is defective, replace the module.

5. **Engine power out warning and reignition occurs at power settings above 55 percent N1 and below 95 percent N2.**

STEP 1. Check for defective or improperly adjusted N2 disable switch (para 4-244).

If the N2 disable switch is defective or improperly adjusted, replace or adjust the switch (para 4-244).

6. **Engine power out and reignition occurs at power settings above 95 percent N2.**

STEP 1. Check for defective engine power out warning unit.

If the engine out warning unit is defective, replace the unit.

7. **Engine ignition does not occur during normal start attempt.**

STEP 1. Check for defective diode CR2 (fig. 4-31).

If diode CR2 is defective, replace the diode.

STEP 2. Check for defective exciter or igniter (TM 55-2840-231-24).

SECTION IX QUICK CHANGE ASSEMBLY

4-251. QUICK CHANGE ENGINE UNIT.

4-252. General — Quick Change Engine Unit. Replacement engines may be built-up as quick change engine units and preserved according to the following paragraphs. Use of a quick change unit depends on availability of sufficient spare components and fittings and the requirement for a slightly reduced aircraft down time.

4-253. Accessories Installation — Quick Change Engine Unit. (Refer to para 4-14.)

4-254. T63-700 Engine Accessories Installation — Quick Change Engine Unit. (Refer to para 4-15.)

4-255. Preservation — Quick Change Engine Unit. Prepare the engine for long term storage as described in TM 55-2840-231-24. Provide for corrosion protection of engine accessories as described in chapter 1 of this manual.

CHAPTER 5

ROTORS

SECTION I MAIN ROTOR HUB AND BLADES

5-1. MAIN ROTOR HUB AND BLADES.

5-2. Description — Main Rotor Hub and Blades. The main rotor (fig. 5-1) is located close to the center of the helicopter cg range. The rotor controls the helicopter lateral and longitudinal motion and the lifting force. The helicopter rate of ascent or descent is controlled by the collective pitch control system. Horizontal motion is controlled by the cyclic system. The main rotor is fully articulated with offset flapping hinges. The rotor consists primarily of four removable rotor blades attached to the rotor hub pitch housings. A central hub supports the pitch housings, cross connected retention straps, and the associated pitch change control mechanism.

CAUTION

When work is performed near the engine air inlet, use care to prevent entry of foreign objects. On aircraft without a filter, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove covers until work is completed and check that area around base of mast, inlet to plenum, and entire plenum chamber is free of foreign material. Install plenum access doors.

Install exhaust covers on aircraft with upward exhausts when work is performed above or near exhaust outlets.

5-3. Troubleshooting — Main Rotor Hub and Blades. Troubleshooting of the main rotor system includes three areas of investigation: (1) investigation of operational vibration problems originating with the main rotor assembly (table 5-1); (2) investigation of symptoms at the controls that can be recognized (table 11-1); and (3) isolation of an unusual controls malfunction that will require disconnection or removal of control components. Troubleshoot by investigating until the component that is causing malfunction has been located.

SECTION II MAIN ROTOR HUB

5-4. MAIN ROTOR HUB.

5-5. Description — Main Rotor Hub. The main rotor hub assembly (fig. 5-1) consists of a central hub, and four identical pitch housing assemblies that are 90-degrees apart and slightly offset. Lead-lag links; a blade damper and damper arm; a droop stop striker strip and spacer, and a pitch control bearing assembly are combined with each pitch housing to produce the pivoting axis, blade flapping stop contact surfaces, and lead-lag hinge for the rotor blades. Two laminated retention strap assemblies that are both vertically and torsionally flexible, extend through the pitch housings and connect to the lead-lag links. A lower shoe assembly, attached to the central hub, contains a droop stop ring and droop restrainers that support the blades at rest and distribute droop loads at low blade rpm. Hub assemblies have balancing hardware installed in the hollow cores of the lead-lag bolts. Balance washers are added or removed when main rotor balancing is required. Refer to paragraph 5-53A for balance instructions.

5-6. Inspection — Main Rotor Hub. *a.* Inspect all accessible areas of main rotor hub and subassemblies for evidence of cracks, scratches, nicks and any other physical damage or deformation. **DAMAGE THAT EXCEEDS 0.010 INCH IN DEPTH IS NOT ALLOWED** unless removed without leaving abrupt changes in surface contour. **NO CRACKS OF ANY KIND ARE ALLOWED.**

b. Check that blade attach pins and damper attach pins are locked and not loose (para 5-73).

c. Check for gap between damper and pitch housing **0.002-INCH MINIMUM.** If gap is insufficient,

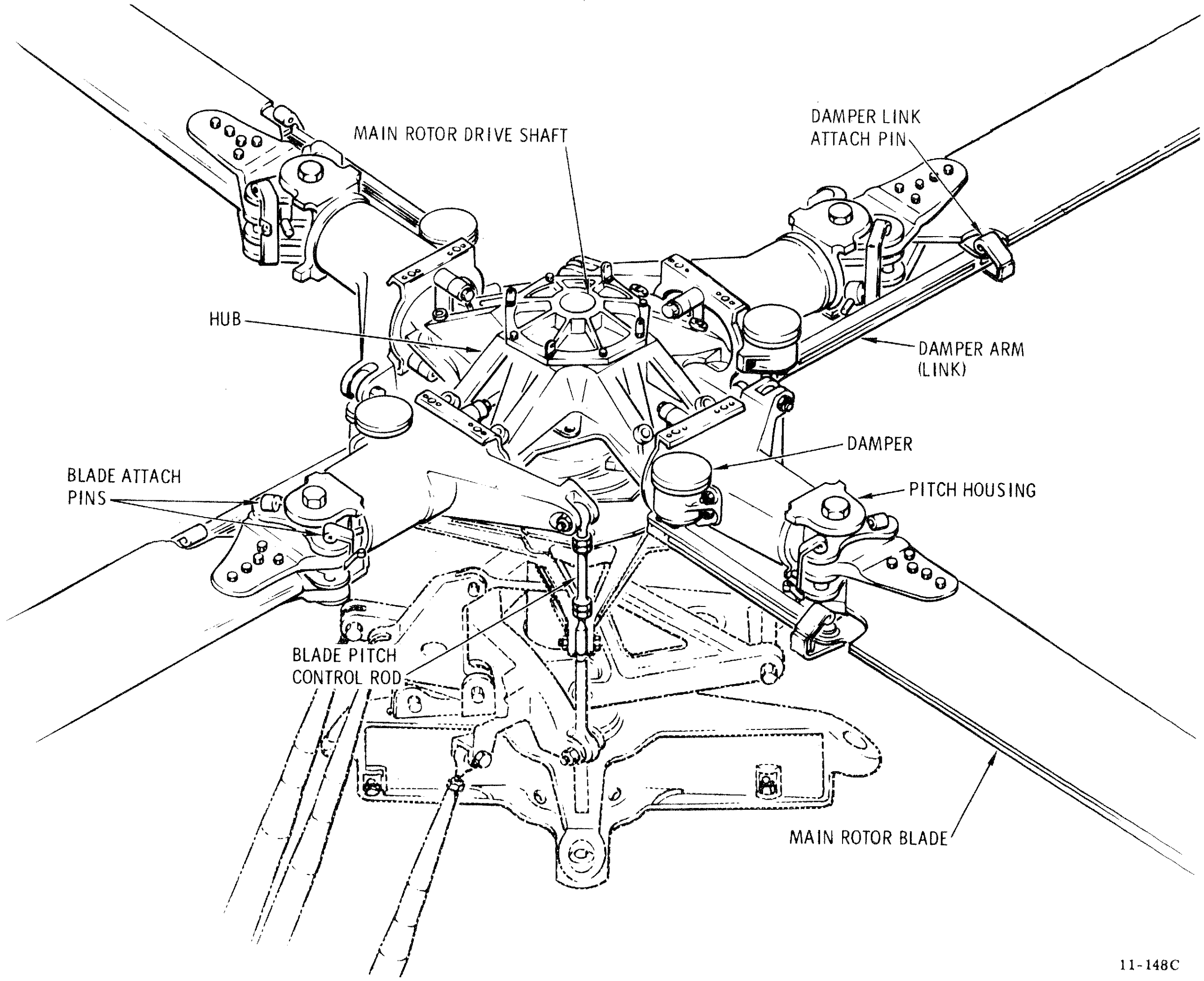


Figure 5-1. Main Rotor Hub and Blades

remove main rotor damper (para 5-30) and rework pitch housing by smooth blending into the surrounding area (para 5-33).

d. Inspect main rotor damper for hydraulic fluid leakage (none permitted). If leakage is noted, repair or replace damper.

5-7. Cleaning — Main Rotor Hub.

CAUTION

Do not allow solvent to enter inside of hub and contaminate bearing lubricant. Bearings are cleaned and lubricated only during hub assembly overhaul.

Clean the main rotor with a clean, dry cloth moistened with solvent (C94). Blow out all crevices and holes with dry, filtered, low-pressure compressed air.

Table 5-1. Troubleshooting the Main Rotor Hub and Blades.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **A one-per-revolution lateral beat is noticed during flight at high power settings. Lateral beat is also noted in autorotation and in turns.**

STEP 1. Check for correct damper torque (para 5-29).

If damper torque is incorrect, readjust torque within limits (para 5-38).

STEP 2. Remove dampers (para 5-30) and check for correct phasing.

If damper phasing is incorrect, adjust phasing (para 5-38).

STEP 3. Check main rotor balance.

If main rotor is not in balance, readjust according to paragraph 5-53A.

2. **A one-per-revolution lateral beat is noticed during warmup or shutdown.**

STEP 1. Perform steps 1, 2 and 3 in Malfunction No. 1 as required.

3. **A lateral beat occurs at low (idle) rotor rpm. When operating at normal (flight) rotor rpm, the lateral beat becomes a vertical beat.**

STEP 1. Perform steps 1, 2 and 3 in Malfunction No. 1 as required.

4. **When operating, a four-per-revolution or medium frequency beat is noticed.**

STEP 1. Check that damper is not sticking by moving blade; then check damper low range torque.

If damper is sticking or low range torque is incorrect, adjust low range torque (para 5-29). Repair a defective damper.

STEP 2. Remove dampers and check for correct phasing.

If damper phasing is incorrect, adjust phasing (para 5-38).

STEP 3. Check blade vibration absorbers for freedom of motion, condition and security.

If defective, repair or replace absorber (para 5-77).

STEP 4. Check for worn rotor hub feathering bearings.

If bearings are worn, replace bearings (para 5-13).

5. **A one-per-revolution vertical beat is present throughout all flight maneuvers. The beat becomes heavier at higher airspeeds.**

STEP 1. Check main rotor blade track.

If blades are not in track, readjust according to paragraph 5-43).

6. **When operating, a continuous one-per-revolution lateral beat is present.**

STEP 1. Check main rotor blade tip weights for security.

If tip weights are not secure, repair as necessary (para 5-65).

If tip weights are missing, replace the blade (para 5-54).

STEP 2. Check main rotor balance.

If main rotor is not in balance, readjust according to paragraph 5-53A.

7. **When operating, a lateral feedback (beat) is noted in the cyclic stick. No longitudinal feedback is detected.**

STEP 1. Check main rotor blade track.

If blades are not in track, readjust according to paragraph 5-43).

Table 5-2. Premaintenance Requirements for Removal of Main Rotor Head.

Conditions	Requirements
Special Tools	(T26 or T35)(T24)(T15)
Support Equipment	Hoist
Minimum Personnel Required	Two

5-8. Removal — Main Rotor Hub. *a.* Remove four main rotor blades (para 5-54).

b. Remove main rotor drive shaft (para 6-7).

c. (See fig. 5-2.) Disconnect flexible boot from lower end of hub outer liner.

d. Disconnect scissors assembly from main rotor hub (para 11-17).

e. Disconnect each pitch change rod from pitch housing (fig. 5-2).

NOTE

Color code each pitch change rod to match the pitch housing and blade to which it was attached. The main rotor can be reinstalled in the same position.

f. Cut lockwire and remove two screws securing retainer to locknut. Remove retainer.

NOTE

If main rotor wrench (T26) is not available, tool (T35) may be used if all tang slots are widened to 0.344 inch.

g. Remove locknut by using main rotor wrench (T35).

CAUTION

Use care to ensure proper installation and use of hub puller (T24) to prevent damaging main rotor hub assembly.

h. (See fig. 5-4.) Break loose main rotor hub by using hub puller (T24), then remove hub puller.

CAUTION

Do not lift the main rotor hub by using the pitch change housings for lifting points. The strap packs and/or the upper hub casting may be damaged.

i. Temporarily reinstall bracket (on aircraft so equipped) or eyebolts and lifting adapter (T15) on main rotor hub. Take care to tighten the four lifting bracket mounting bolts (or eyebolts) evenly and only finger tight. Do not wrench-tighten the bolts with the main rotor drive shaft removed.

j. Attach an overhead hoist to the lifting bracket or lifting adapter eye and lift the hub free from the main rotor mast.

k. Position hub in a suitable working area. Remove lifting bracket or lifting adapter and eyebolts from rotor hub.

5-9. Inspection — Main Rotor Hub Assembly. *a.* Inspect pitch housings (fig. 5-2 and 5-5) for scratches, nicks and cracks. No cracks are permissible. Scratches and nicks must be cleaned up before measuring the depth of the damage to determine housing serviceability. Accomplish repair according to paragraph 5-10.

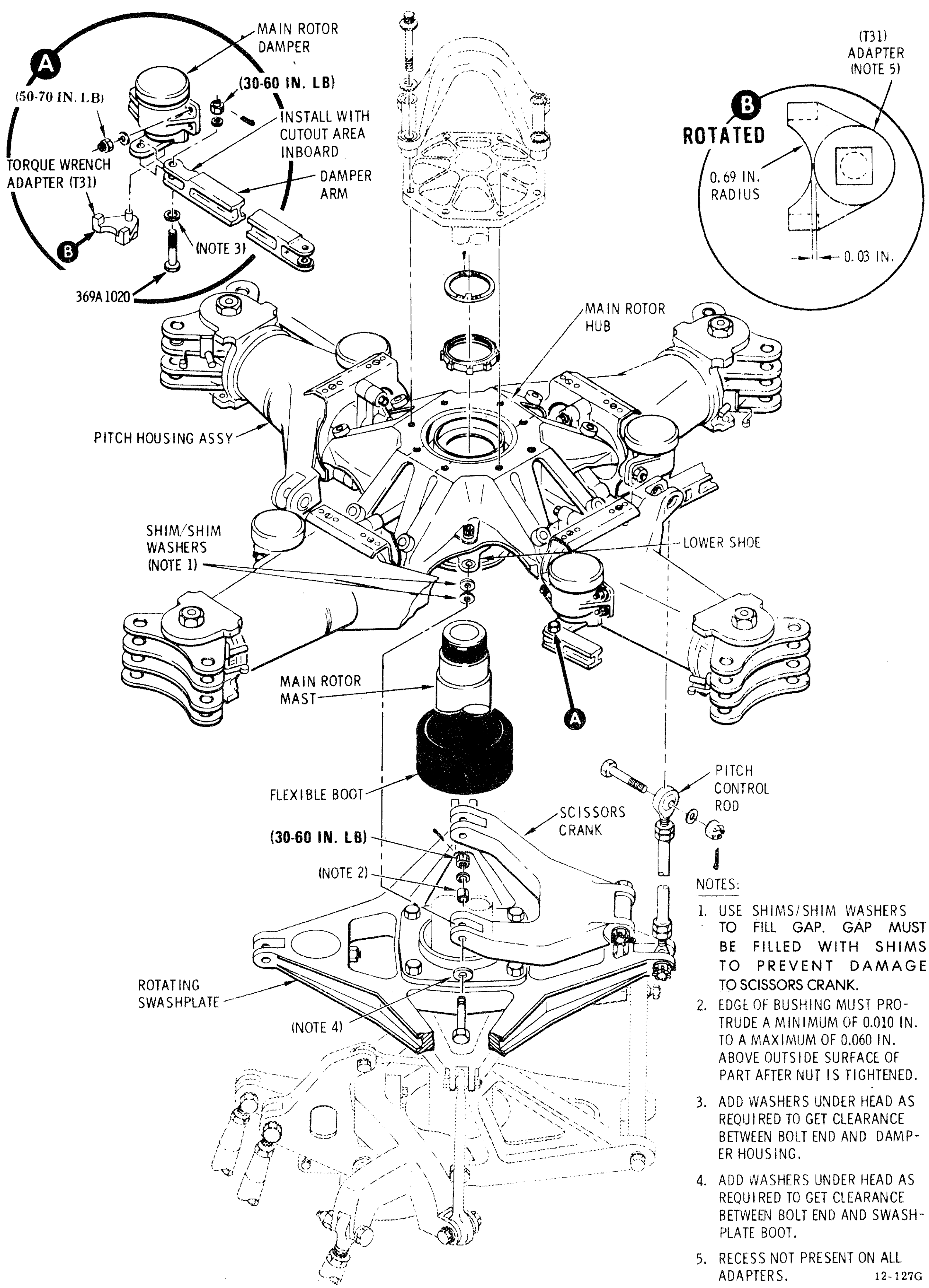
b. Inspect sleeve bushing for snug fit in pitch housing arm clevis lug; the lug must not show evidence of wear caused by bushing movement. **MAXIMUM ALLOWABLE DIAMETER OF THE HOLE IN THE BUSHED LUG IS 0.500 INCH; FOR THE UNBUSHED LUG AND THE BUSHING ID, NOT MORE THAN 0.313 INCH.** Inspect inner surfaces of all pitch housing arm clevis lugs for chafing caused by misaligned pitch control rod end bearings. Chafing will be in the form of crescent-shaped grooves. If chafing wear is found, accomplish repair according to paragraph 5-10. The chafed area must be reworked before measuring the depth of the damage.

c. Inspect the droop stop ring for corrosion, dents, and scratches. Repair defects according to paragraphs 5-20 and 5-21. To inhibit corrosion, spray droop stop ring, rollers and pitch bearings with rust inhibitor (C87).

d. Inspect droop stop roller for flat spots, pit marks, and looseness on roller shaft.

e. Inspect striker plate for dents (brinelling) and pit marks.

f. Press each pitch housing downward several times and check for evidence of a binding pitch bearing or droop restrainer follower. Inspect visible portion of droop restrainer for cracks, and the follower spring for breaks.



NOTES:

1. USE SHIMS/SHIM WASHERS TO FILL GAP. GAP MUST BE FILLED WITH SHIMS TO PREVENT DAMAGE TO SCISSORS CRANK.
2. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
3. ADD WASHERS UNDER HEAD AS REQUIRED TO GET CLEARANCE BETWEEN BOLT END AND DAMPER HOUSING.
4. ADD WASHERS UNDER HEAD AS REQUIRED TO GET CLEARANCE BETWEEN BOLT END AND SWASHPLATE BOOT.
5. RECESS NOT PRESENT ON ALL ADAPTERS.

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Figure 5-2. Main Rotor Hub Installation.

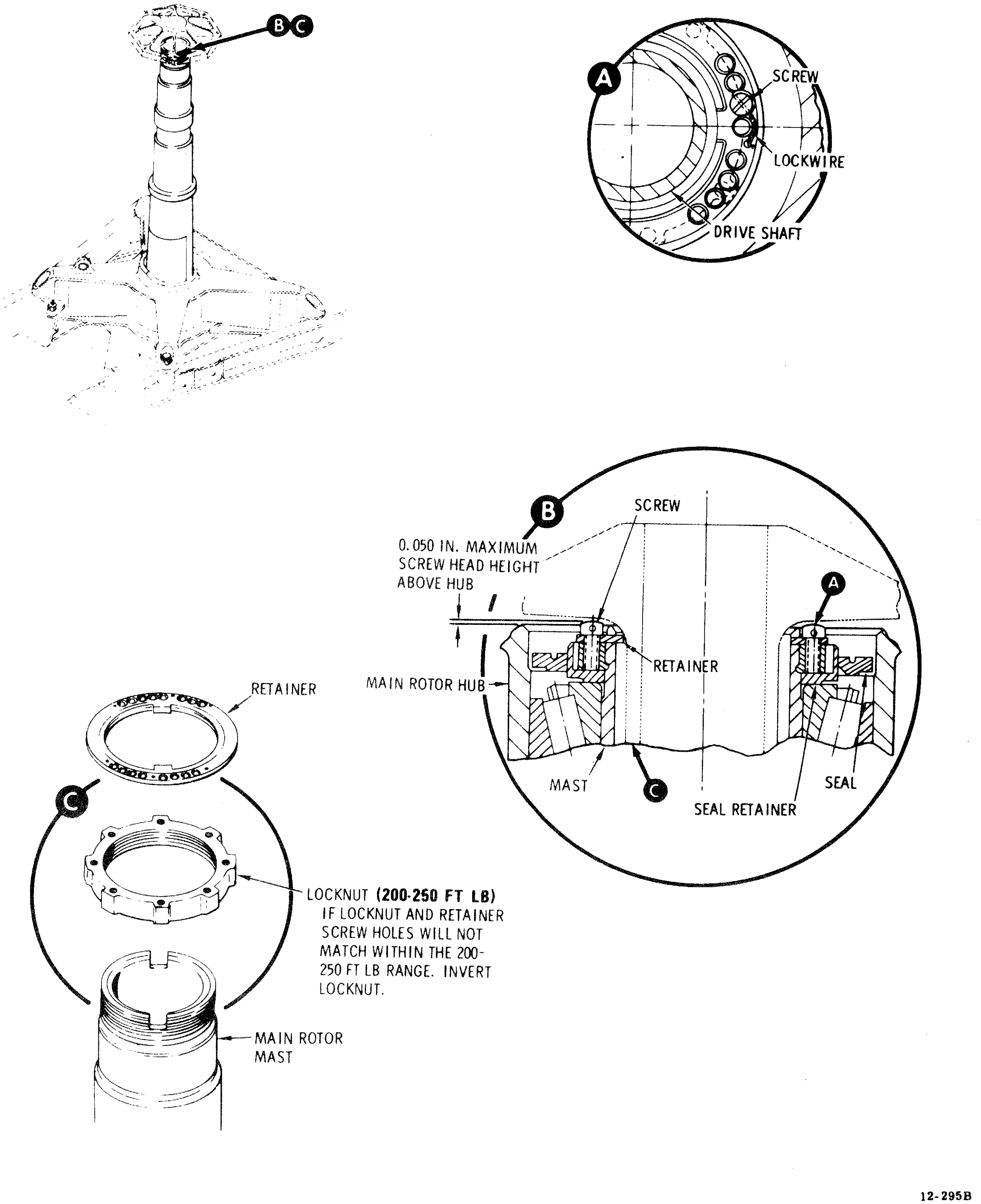


Figure 5-3. Main Rotor Mast Locknut Installation.

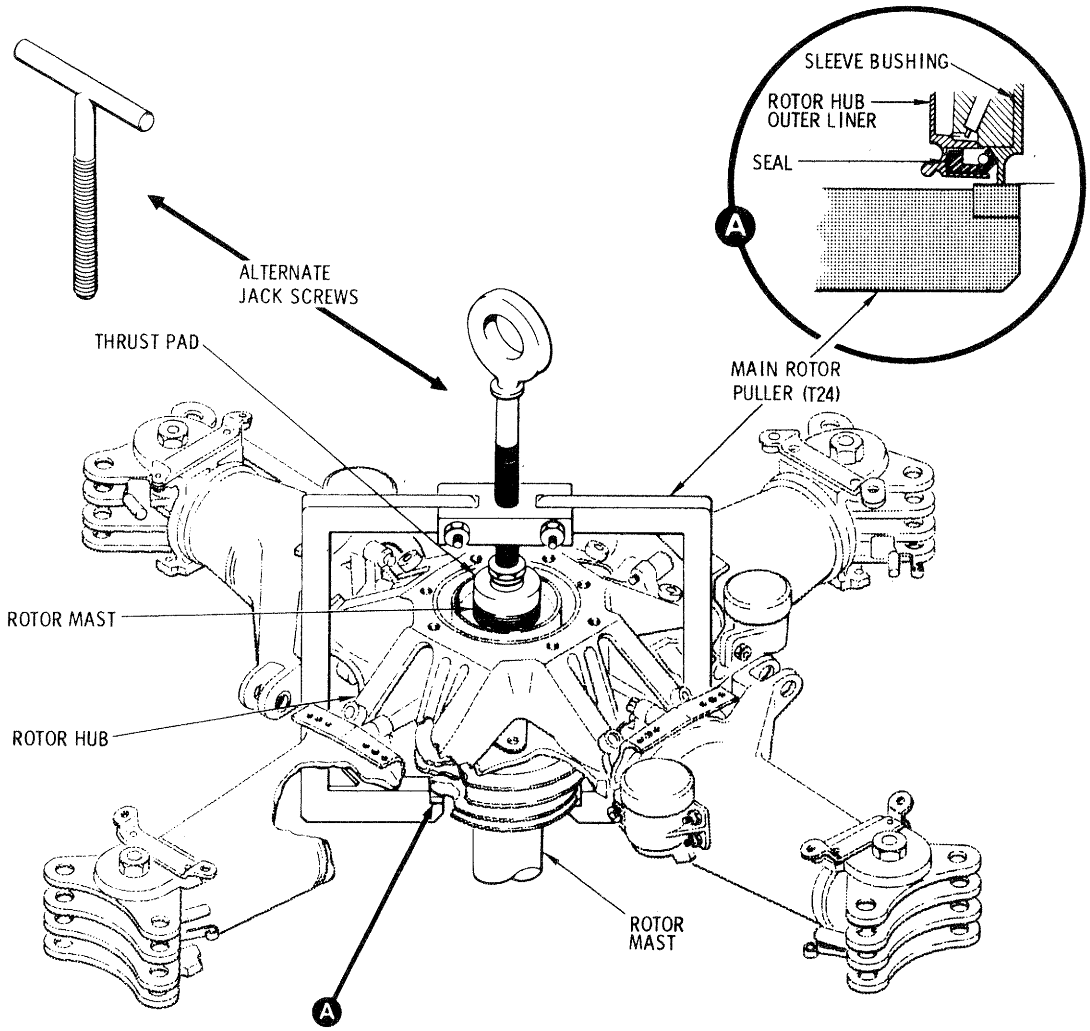


Figure 5-4. Pulling Hub Assembly Off Mast.

NOTE

Do not remove tetrafluorethylene (TFE) debris which works out of pitch bearing edge. The debris is normal and helps lubricate bearing. Removal of debris will increase bearing wear rate.

g. Inspect lead-lag link stop for broken spring cracks, breaks and visible bond line cracks. Inspect for 0.0000 axial and 0.0015 radial play as measured from the center leading lag bolt to outside edge of the link.

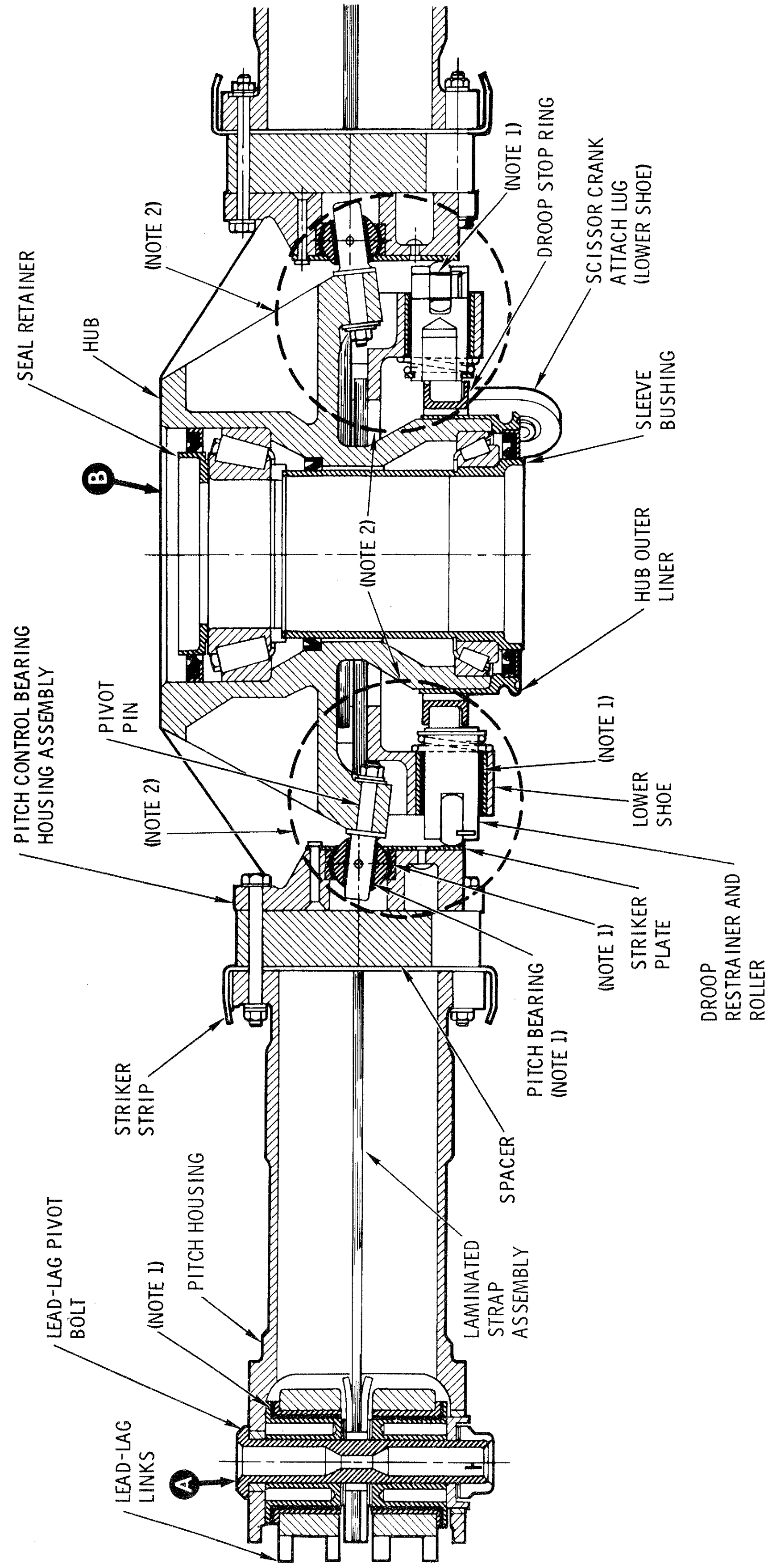
h. Inspect ID of lead-lag bolts for presence of corrosion. Bolts having evidence of corrosion should get the preventive treatment described in paragraph 5-16.

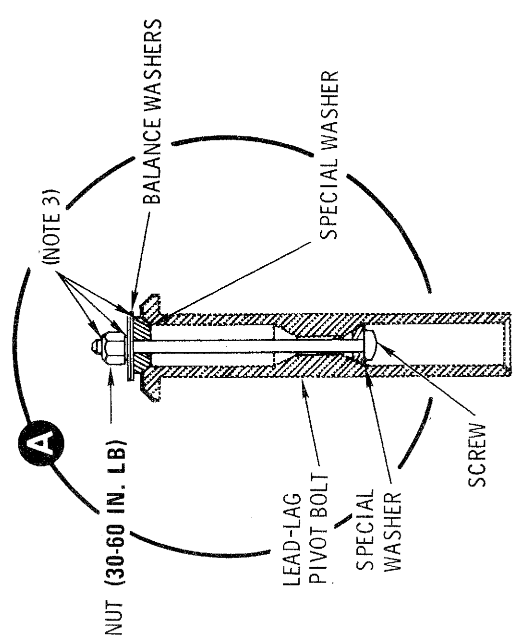
NOTE

Lead-lag bolts in hub assemblies contain balancing hardware. Remove the balancing hardware to perform the inspection. Each set of removed hardware should be tagged or color-coded to ensure correct reinstallation.

i. Inspect each striker strip for cracks, deformation, loose nutplates, and badly worn contact surfaces.

j. Inspect all parts of main rotor hub for cracks, breaks, scratches and nicks. Any evidence of damage





NOTES:

1. HEAVY LINES INDICATE REINFORCED TEFLON BEARING LININGS (TYPICAL).
2. RELATIVE POSITIONS OF PITCH LINE AND DROOP STOP DETAILS STAGGERED BY 1/4 INCH (0.25) TO MATCH OFFSET BUILDUP OF LAMINATED STRAP ASSEMBLIES.
3. **USE SEALANT (C89) AT JOINTS SHOWN. AT LEAST TWO THREADS MUST SHOW THROUGH NUT AFTER TORQUING.**

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Figure 5-5. Hub Assembly Cross Section (sheet 1 of 2)

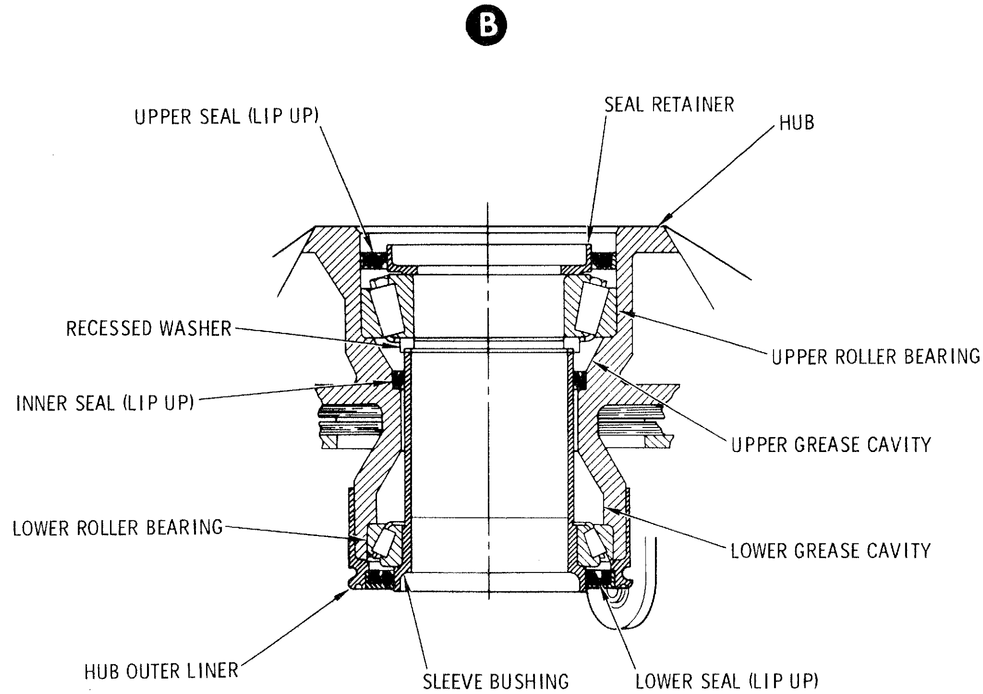


Figure 5-5. Hub Assembly Cross Section. (sheet 2 of 2)

that cannot be repaired as described in paragraph 5-10, requires that the main rotor hub be sent to overhaul.

k. Every 600 hours, inspect the main rotor retention straps (fig. 5-6) as follows.

(1) Check for breaks in the strap pack laminations at the inboard and outboard ends of the pitch housing where the straps are accessible to view. Check both the lead and lag legs of the straps at these locations.

CAUTION

Do not pry at the strap pack with a sharp or hard-edged tool. If edges become nicked the rotor hub assembly must be sent to overhaul.

(2) If no obvious strap failure is noted, use a blunt-nosed wooden or phenolic pin (pencil size with 0.06-inch-radius point) to probe at the upper and lower strap laminations at the inboard end of the blade pitch housings. A failed strap, either in the lead or lag leg of the pack, will move away from the other straps. If the upper and lower straps laminations remain in tension when probed, no failure has occurred.

(3) If a single strap failure is noted during inspection, rotor operation can still be continued. However, a special inspection shall be performed again in 300 hours, and recorded on appropriate forms. When three straps to any one blade are broken (at one side of the rotor hub) the rotor hub assembly must be sent to overhaul.

NOTE

Ends of permissible broken straps must be taped according to paragraph 5-17 to prevent scratching of adjacent straps.

l. Visually inspect the outboard ends of the main rotor retention straps for gaps between the pack laminations (fig. 5-6).

(1) A single straight lamination gap **NOT IN EXCESS OF 0.030-INCH** is permissible within the pack or next to either the upper or lower outer shoe.

NOTE

On some strap packs the teflon may appear wrinkled or extend past the end of the laminates obstructing the view.

When this condition exists, visual inspection should be made from the leading and trailing edges of the strap pack.

(2) Retention strap packs showing a fanning or bowing of the laminations indicate an improper condition and the hub must be replaced.

5-10. Repair — Main Rotor Hub Pitch Housing Assembly.

CAUTION

During repair on aircraft, use covering over engine air inlet fairing opening to prevent entry of foreign objects into air intake. When reconnecting a pitch control rod, be sure that the rod ends are centered in the swashplate and housing arm clevis lugs. Realign, if necessary, and hold rod to prevent turning while tightening locknuts.

a. Use grade 320 abrasive cloth (C24) to polish smooth all scratches, nicks and chafing wear in the pitch housing.

b. After smoothing (removal of all sharp or raised edges), the **REPAIR DEPTH MUST NOT EXCEED 0.010 INCH** in any area of the housing except the inner surfaces of the arm clevis lugs.

c. A maximum (repaired) depth of 0.050 inch is acceptable in the clevis lug with the sleeve bushing. On the opposite (counterbored) lug, a maximum (repaired) depth of 0.010 inch is permissible in the area of the counterbore, and a maximum of 0.050 inch in the area outside the counterbore.

d. Touch up all repaired areas with chemical film (C21), followed by primer (C79).

e. Remove main rotor hub assembly for overhaul if repair limits are exceeded.

5-11. Repair — Main Rotor Hub Pitch Housing Attaching Parts (AVIM). Repair pitch housing attaching parts according to paragraphs 5-12 through 5-18.

5-12. Replacement — Pitch Control Bearing Housing Assembly, Spacer or Striker Strip (AVIM). (See fig. 5-7.)

CAUTION

Use care during removal of parts from around the strap pack. Any nicks or scratches on the straps require scrapping of the strap pack. Do NOT remove lead-lag hinge bolt to remove pitch control bearing housing assembly.

a. Support blade and pitch housing from beneath. Remove three nuts, six washers, and three bolts that secure bearing housing.

NOTE

Check if washers are installed between the spacer and striker strip. These washers must be reinstalled to establish correct static droop angle between the pitch housing and hub.

b. Carefully remove spacer by sliding it downward past striker strip. Retain exact number of droop shim washers, if installed.

c. Slide bearing housing assembly off pivot pin.

d. Inspect pivot pin for serviceability (para 5-14). Unserviceable pivot pins on 369A1200-617 main rotor hubs may be replaced as follows.

(1) Replace pivot pin using pitch housing stud wrench (T14).

(2) **TORQUE PIN TO 200-220 INCH-POUNDS** with vertical edges of pin flange positioned parallel to vertical lug edges.

e. Remove and discard pitch bearing if defective (para 5-13). **TOTAL RADIAL LOOSENESS OF ASSEMBLED BALL AND PIN MUST NOT EXCEED 0.010 INCH. MINIMUM ALLOWABLE PIN DIAMETER IS 0.433 INCH.**

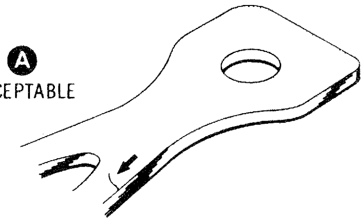
f. Replace a flanged one-piece or two-piece striker strip if it is cracked or the flapping stop contact areas are worn through the hard anodized surface. One-piece striker strips without flanges for fairing attachment are not replaceable except at overhaul.

g. Position pitch control bearing housing on pivot pin.

CAUTION

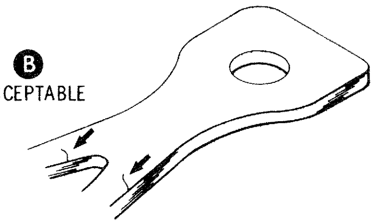
If droop shim washers were removed from between the spacer and striker strip the exact thickness removed must be reinstalled. There must be an

A
ACCEPTABLE



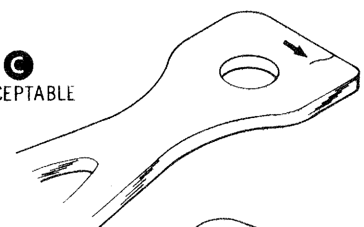
A IF A CRACK IS LOCATED IN ONE STRAP (LAMINATE) ON EITHER THE LEAD OR LAG LEG OF A STRAP PACK, THE COMPLETE LAMINATE HAS NOT FAILED AND THE BREAK IN THE LEG IS ACCEPTABLE.

B
ACCEPTABLE



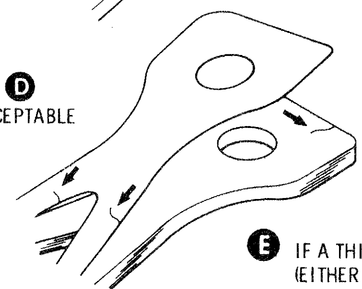
B IF ANY CRACK IS LOCATED ON THE SAME LAMINATE ON BOTH THE LEAD AND THE LAG LEGS OF ANY ONE SIDE (BRANCH) OF THE ROTOR HUB, THAT LAMINATE HAS FAILED; BUT THE FAILURE IS ACCEPTABLE.

C
ACCEPTABLE



C IF ANY CRACK IS LOCATED BEYOND THE JOINT OF THE STRAP PACK LEGS, THAT LAMINATE HAS FAILED; BUT THE FAILURE IS ACCEPTABLE.

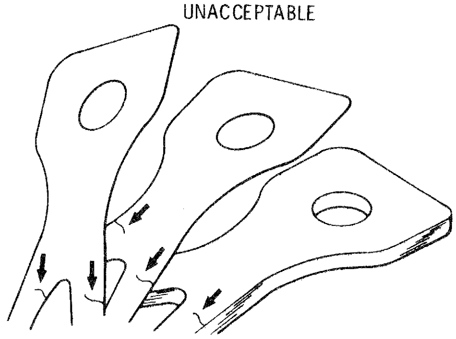
D
ACCEPTABLE



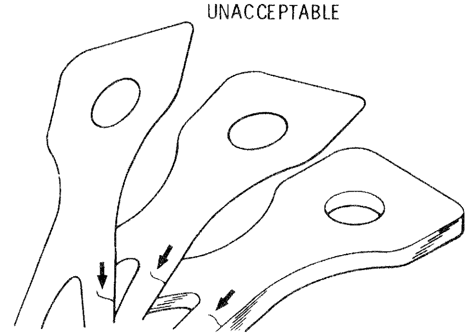
D ANY TWO LAMINATES FAILED AS DESCRIBED IN B AND/OR C ARE ALLOWED ON EACH SIDE OF THE ROTOR HUB (TOTAL OF EIGHT PER HUB ASSEMBLY).

E IF A THIRD LAMINATE CRACKS IN THE SAME PACK (EITHER LEAD OR LAG LEG OF JOINT) AT ANY ONE SIDE OF THE HUB, THE STRAP PACK MUST BE REPLACED.

E
UNACCEPTABLE



E
UNACCEPTABLE



NOTES:

1. ANY CRACK IN A STRAP (LAMINATE) IS CONSIDERED AS A BREAK. THE LAMINATION HAS NOT FAILED UNLESS BOTH LAMINATION LEGS ON ONE SIDE OF THE ROTOR HUB ARE BROKEN OR THE CRACK (BREAK) IS LOCATED ABOVE THE POINT WHERE THE LEGS JOIN.
2. LAMINATIONS SHOWN SEPARATED ONLY TO DEPICT POSSIBLE CRACK LOCATIONS. SEE TEXT FOR INSPECTION METHODS (VISIBLE LOCATIONS ONLY).

12-290-1

Figure 5-6. Strap Pack Lamination Inspection. (sheet 1 of 2)

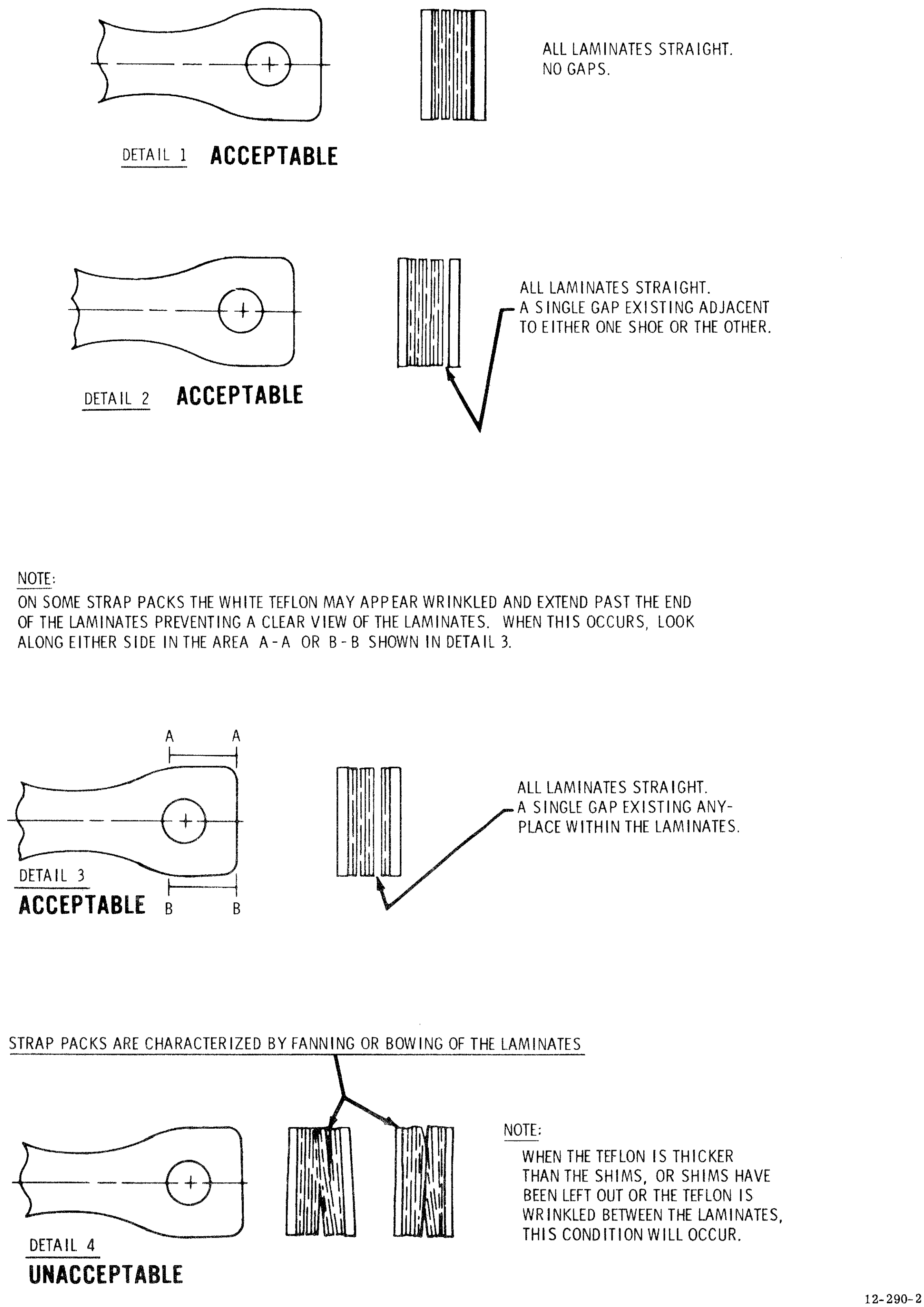
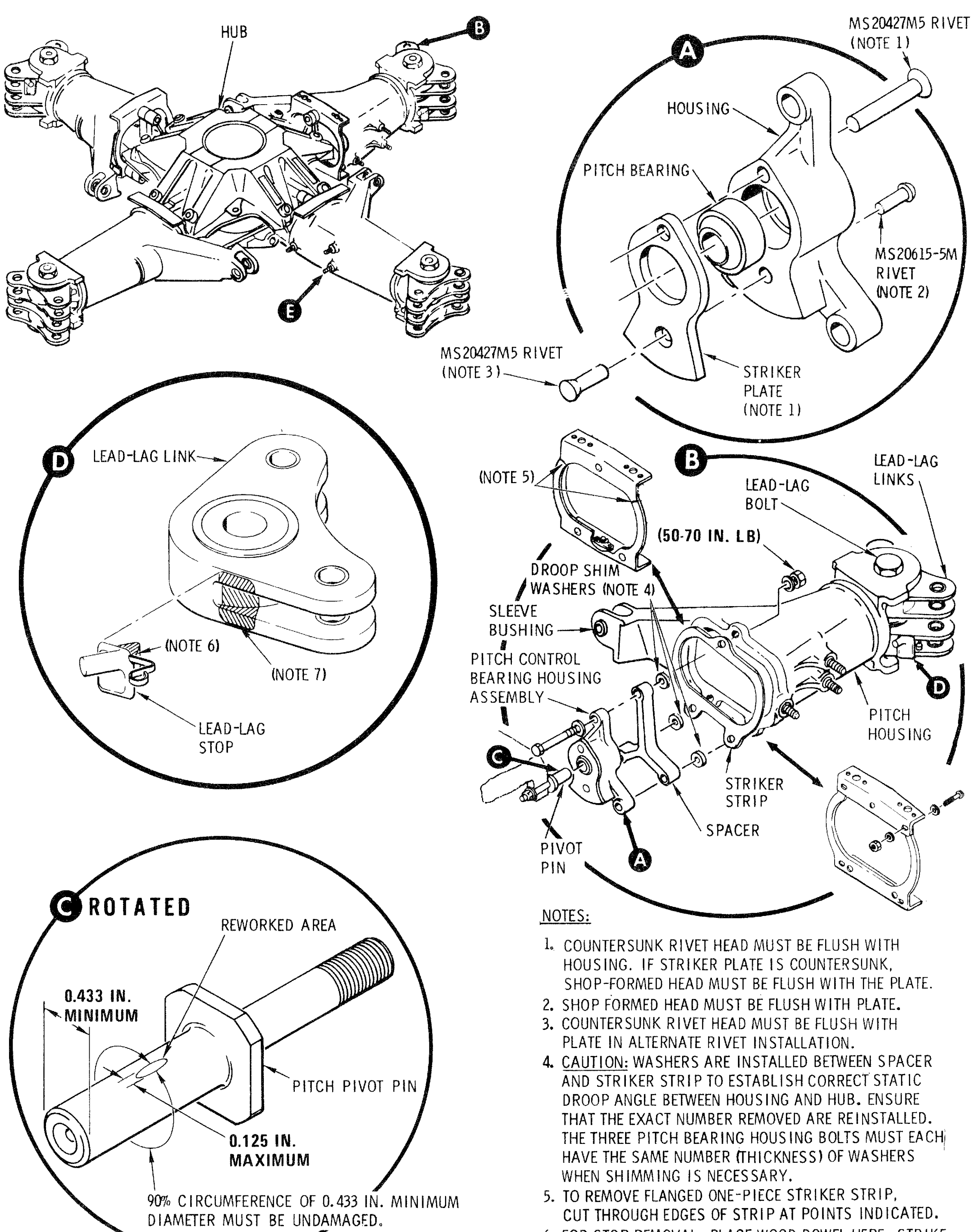


Figure 5-6. Strap Pack Lamination Inspection. (sheet 2 of 2)



NOTES:

1. COUNTERSUNK RIVET HEAD MUST BE FLUSH WITH HOUSING. IF STRIKER PLATE IS COUNTERSUNK, SHOP-FORMED HEAD MUST BE FLUSH WITH THE PLATE.
2. SHOP FORMED HEAD MUST BE FLUSH WITH PLATE.
3. COUNTERSUNK RIVET HEAD MUST BE FLUSH WITH PLATE IN ALTERNATE RIVET INSTALLATION.
4. CAUTION: WASHERS ARE INSTALLED BETWEEN SPACER AND STRIKER STRIP TO ESTABLISH CORRECT STATIC DROOP ANGLE BETWEEN HOUSING AND HUB. ENSURE THAT THE EXACT NUMBER REMOVED ARE REINSTALLED. THE THREE PITCH BEARING HOUSING BOLTS MUST EACH HAVE THE SAME NUMBER (THICKNESS) OF WASHERS WHEN SHIMMING IS NECESSARY.
5. TO REMOVE FLANGED ONE-PIECE STRIKER STRIP, CUT THROUGH EDGES OF STRIP AT POINTS INDICATED.
6. FOR STOP REMOVAL, PLACE WOOD DOWEL HERE. STRIKE SHARPLY WITH HAMMER TO BREAK STOP BOND.
7. BOND ALL STOP-TO-LINK SHADED CONTACT SURFACES WITH ADHESIVE (C9).

12-129-1

Figure 5-7. Main Rotor Hub Component Repair. (sheet 1 of 2)

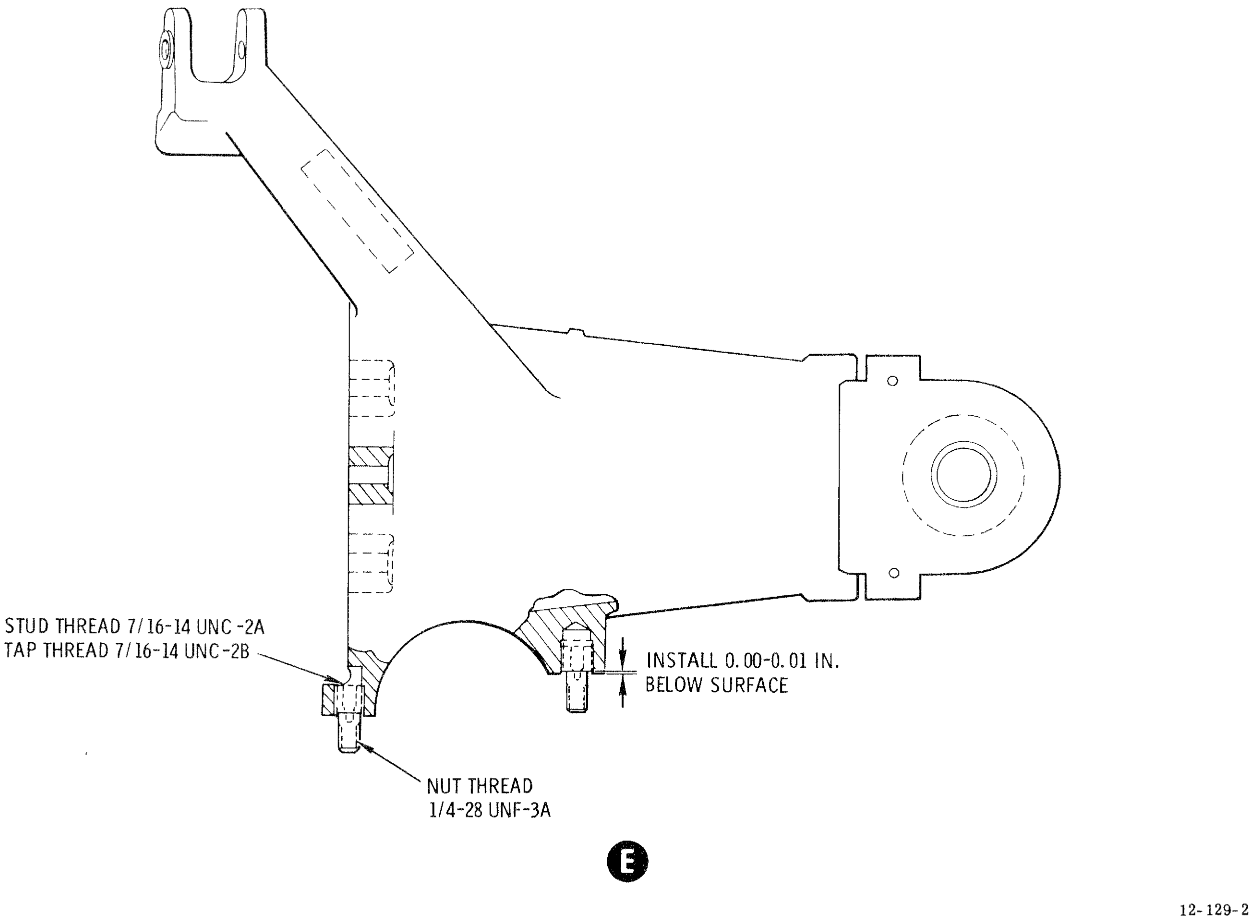


Figure 5-7. Main Rotor Hub Component Repair. (sheet 2 of 2)

equal number (thickness) of washers on each of the three bolts. These washers establish the correct static droop between the pitch housing and the hub.

h. Slide spacer into position between bearing housing assembly and striker strip on pitch housing. Install exact thickness of droop shim washers removed at time of disassembly. Align holes for the three bolts.

i. Install three bolts, six washers, and three nuts. **TORQUE NUTS TO 50-70 INCH-POUNDS.**

j. Seal all the parting lines (joints) of the assembly with a bead of sealing compound (C89).

5-13. Replacement — Pitch Control Bearing (AVIM). (See fig. 5-7.) *a.* Drill out rivets to remove striker plate. **DISCARD STRIKER PLATE IF BRINELLING OR PITTING EXCEEDS 0.030-INCH DEPTH.**

b. Press bearing from housing bore. Clean any primer residue from housing bore with methyl ethyl ketone (C69).

c. Rework minor surface abrasion in housing bore by polishing with crocus cloth (C24). Restore chemical film (C20), where removed.

d. Apply one thin coat of primer (C79) to the bearing OD and the housing bore. Press bearing into bore while primer is still wet. Wipe away any excess primer, using care to keep it out of the bearing races. Check teflon linings of bearing after installation to determine that no damage has occurred from the pressing operation.

NOTE

The outside edge of bearing outer race must be at least flush with the face of the bearing housing. However, the bearing can be recessed up to 0.015 inch.

e. Install striker plate with one MS20427M5 rivet (upper hole) and one MS20615-5M rivet (lower hole). If upper hole of striker plate is countersunk, rivet must be installed with shop-formed head flush with the plate. The shop-formed head of the MS20615-5M rivet must be flush with the plate. An MS20427M5 rivet may be used as an alternate for the MS20615-5M rivet. When the alternate is used, install the rivet from the plate side of the assembly. The heads of all countersunk rivets must be completely flush.

5-14. Repair — Pitch Bearing Pivot Pin (Stud) (AVIM). (See detail C, fig. 5-7.) Pitch bearing pivot pins should be inspected for indications of wear or grooving. Wear or grooving must be cleaned up as specified below before measurement is made to determine hub serviceability.

a. Use a fine India stone or equivalent to clean up all wear spots. Blend smoothly into surrounding area and restore finish to equal the original.

b. If 90 percent of the pin minimum diameter (0.433) is good after rework the pin is serviceable within the following limits. (See detail C, fig. 5-7.)

(1) The reworked area may extend the length of the pin but **MUST NOT EXCEED 0.125 INCH IN WIDTH.**

(2) Several places may be reworked along the diameter of the pin provided the **WIDTH OF THE REWORKED AREAS DOES NOT TOTAL MORE THAN 0.125 INCH.**

(3) **TOTAL RADIAL LOOSENESS** of assembled pitch control bearing and pivot pin **MUST NOT EXCEED 0.010 INCH.**

c. If pivot pin does not meet the above serviceability standards, proceed as follows.

(1) On 369A1200-617 main rotor hub assemblies, replace the pivot pin (para 5-12).

(2) On all other main rotor hub assemblies, replace the hub.

5-15. Replacement — Lead-Lag Link Blade Stop (AVIM). (See fig. 5-7.) Replace a blade stop if it is broken, cracked, or has a broken spring.

CAUTION

The lead-lag link assembly must not be removed from the pitch control housing for this repair. The hinge bolt cannot be retorqued without a special adapter.

a. Provide a backup support for the link assembly. Use a 1/2-inch wood dowel and a hammer to drive stop from link. Strike the dowel sharply to break the adhesive bond.

b. Using a sharp-edged metal scraper, carefully scrape any adhesive residue from the link. If scraper penetrates through the paint and chemical surface film, the surface must be refinished. **THE DEPTH OF GOUGES OR NICKS IS LIMITED TO 0.010 INCH MAXIMUM AFTER REWORK.** Rework by smooth blending into the surrounding area with grade 320 abrasive cloth (C24). Final polish with crocus cloth (C25). Restore chemical film protection (C20).

c. Clean the bond area of all contact surfaces by wiping with phosphoric acid (C74). Flush-wipe the cleaned surface four times with a mixture of equal parts of distilled water and isopropyl alcohol (C82) to remove all traces of the phosphoric acid. Rinse the cleaned surface with tap water, followed by a rinse of distilled water until the surface is "waterbreak" free. Dry for 30 minutes minimum at 150°F.

d. Check the fit of the stop in the link. The flange and radius contact surfaces of the **STOP MUST MATE WITH THE LINK WITHIN 0.010 INCH. MAXIMUM CLEARANCE BETWEEN THE STOP AND INSIDE SURFACES OF THE LINK EARS IS LIMITED TO 0.020 INCH.** This tolerance applies to either side of the stop.

e. Prepare a mixture of epoxy adhesive (C9). Apply a uniform coating of the mixed adhesive to all contact surfaces. Hand-press the stop into place between link ears until all mating surfaces are in firm contact. Apply a suitable clamping device so that contact is maintained. Cure for a minimum of 8 hours at room temperature.

5-16. Corrosion Control — Main Rotor Hub Lead Lag Bolt. (See fig. 5-5.)

CAUTION

Do not disturb the torque on the lead-lag bolts. A special adapter is needed to retorque the lead-lag bolts.

a. Remove and tag or color-code each set of balance hardware that is installed.

b. Remove corrosion with grade 180 abrasive paper (C1) and finish with grade 400 abrasive cloth (C24).

c. Swab cleaned surface with methyl ethyl ketone (C69) and apply unthinned chromate primer (C79).

d. Reinstall balance hardware that was removed.

5-17. Repair — Taping Broken Retention Strap Ends.

CAUTION

Be careful not to bend broken ends excessively or scratch adjacent straps.

a. Carefully wipe ends of broken strap with a clean, soft cloth moistened with solvent (C94).

b. Use a mild blast of filtered air to dislodge any foreign particles between broken strap ends and adjacent strap.

c. Carefully tape broken ends of strap with tape (C100).

5-18. Replacement — Main Rotor Hub Damper Attaching Studs. (See fig. 5-7, sh 2.) *a.* Prior to removing a stud, cut off the nut end of the stud at a point approximately 0.06-inch from the housing boss. There is an internal pilot hole for the removal drill.

b. Use a 11/32-inch (0.344 in.) drill and drill to a depth of 0.19-inch to remove stud kees. Break off kees by deflecting them inward. Unscrew remainder of stud with an easy-out type tool.

c. Correct any minor thread imperfections in the parent metal. Use a 7/16-14 UNC-2B bottoming tap for studs.

d. Apply a coating of primer (C79) to the stud end. Turn into the threaded bore by hand or by using applicable insertion tool. Check that the stud stops at the correct depth of 0.00 - 0.01 inch below the boss surface. (See fig. 5-7, sh. 2.)

NOTE

Before driving in the kees of a stud, check its parallelism with the other studs to reveal any possible interference with the damper housing mounting holes. A spare damper housing or assembly will serve as suitable checking device.

e. Check that the stud kee position is rotated approximately 1/8 turn from the original kee grooves. Use an applicable installation tool and drive in the kees until 0.01 - 0.03-inch below the boss surface. (An old damper housing may be used to press stud kees most of the way in to ensure stud to damper hole alignment. If this method is used, tighten down evenly on all three studs.)

5-19. Inspection — Main Rotor Hub Roller Bearing. (See figure 5-5.) *a.* Remove seal retainer from recess in top of hub.

b. Remove upper seal with a standard seal puller, or by carefully prying with a flatbladed tool. Discard seal.

c. Remove upper roller bearing cone by hand.

d. Remove recessed washer from top of sleeve bushing.

CAUTION

Do not interchange or replace recessed washer. The original washer MUST be reinstalled to obtain correct bearing preload. The recessed washer is custom fitted to each hub during overhaul.

e. Use light pressure and push sleeve bushing from the hub. As the bushing is pressed out the lower roller bearing cone will force the lower seal from the bottom of the hub bore. Discard seal.

f. Carefully remove the inner seal by grasping seal lip with longnose pliers and pulling the seal from hub bore. Discard seal.

NOTE

To prevent damage, make sure that the main rotor hub laminated strap assemblies and pitch bearings (fig. 5-5) are protected from solvent used in the following step.

g. Using solvent (C94) and a soft bristle brush, clean bearings, sleeve-bushing and main rotor hub bore and allow to air dry. Immediately apply a light coat of lubricant (C61) to roller bearing cups and cones.

h. Inspect, without further disassembly, the rolling surface of upper and lower roller bearing cups and cones for flat spots, scoring, pitting, grooving, roughness, and heat discoloration. No defects are allowed.

i. Install new center seal into center of hub by hand pressing into place. Check that the seal lip is up as shown in figure 5-5, sheet 2.

j. With the aid of an assistant, turn the hub assembly upside down on the bench.

k. Hand pack the lower cavity of the hub with grease (C47). Pack the lower roller bearing with grease and install the sleeve bushing with cone in hub bore. Fill remaining cavity up to level of seal with the same grease.

l. Position lower seal with lip towards hub center as shown in figure 5-5, sheet 2. Press in or tap lightly until seal is seated. Wipe away any excess grease.

m. With the aid of an assistant, turn the hub assembly right side up on the bench.

n. Hand pack upper cavity of the hub with grease (C47).

CAUTION

Do not interchange or replace recessed washer. The original washer must be reinstalled to obtain correct bearing preload.

o. Place original recessed washer on top of sleeve bushing with recess down as shown in figure 5-5, sheet 2.

p. Hand pack upper bearing cup and cone with grease (C47) and install cone in cup. Fill remaining cavity up to level of upper seal with the same grease.

NOTE

Do not reinstall upper seal and retainer at this time. Protect hub bore from contamination and retain new upper seal and retainer for reinstallation after seating main rotor hub on mast (para 5-23).

5-20. Repair — Main Rotor Hub Droop Stop Restrainer and Roller Repair. Replace a defective droop stop roller (fig. 5-8). If **CLEARANCE** between roller bearing liner and shaft **IS MORE THAN 0.015 INCH**, replace the worn part. **ROLLER SHAFT OD MUST NOT BE LESS THAN 0.437 INCH.** A defective follower or spring is replaceable after removal of the droop stop ring (para 5-22).

a. Remove pitch control bearing housing assembly that contacts the lower shoe roller to be removed (para 5-12).

NOTE

Only one roller of each opposing pair of droop stop rollers can be removed at one time. One droop restrainer must be pressed against the droop stop ring to force the opposite restrainer out and expose the roller shaft for removal. The same condition pertains during installation.

b. Press down on pitch housing that is opposite to the roller to be removed. Remove cotter pin and shaft.

c. Remove droop stop roller.

d. To install replacement roller, press down on pitch housing at the opposite side of the hub from the roller to be installed. Install roller, shaft, and new cotter pin.

e. Reinstall pitch control bearing housing assembly (para 5-12).

5-21. Repair — Main Rotor Hub Droop Stop Ring. *a.* The repair depth limit for corrosion nicks or scratches in the droop stop ring is **0.007 INCH FOR ALL SURFACES EXCEPT THE EDGES OF THE RING OD.** The depth limit for the **RING OD EDGES IS 0.030 INCH.** (See detail C, fig. 5-8.) All reworked areas must be blended smoothly with a 15 to 1 ratio into the surrounding area.

b. Touch up repaired areas of cadmium-plated rings, except the channel, with chemical film (C20), followed by primer (C79). Repairs in the channel of cadmium plated rings and in all areas of stainless steel rings should be sprayed with dry film lubricant (C65) only.

c. If the repair limits for the droop stop ring have been been exceeded, the ring may be replaced according to paragraph 5-22.

5-22. Replacement — Main Rotor Hub Droop Stop Ring. *a.* Turn the hub upside down (fig. 5-8). Support the hub so that the pitch housing will unload the cam followers and provide maximum clearance between the striker plates and droop stop rollers.

b. Release the retaining (snap) ring of each droop stop restrainer from its groove. Move the retaining ring flush against the T-head to provide additional clearance and reduce the spring tension.

c. Remove the four droop stop rollers (para 5-20).

d. Starting from either scissors crank attach lug, number the four droop restrainers in a clockwise direction for identification during the replacement procedure (fig. 5-8). This number code should be placed on the outboard (roller) upper end of follower. Guard against accidental removal of the codes during the remaining steps.

NOTE

Designation of upper and lower droop restrainer followers is in relation to the hub as it sits, not as it is installed on the aircraft. Notice that the odd numbered (1 and 3) followers are in the upper position and the even numbered (2 and 4) followers are in the lower position.

e. Push the droop stop ring toward the number 1 and 2 followers. Pull the number 3 and 4 follower T-heads from the droop stop ring channel. When the T-heads are free of the ring channel they should be turned perpendicular (vertical to the channel.)

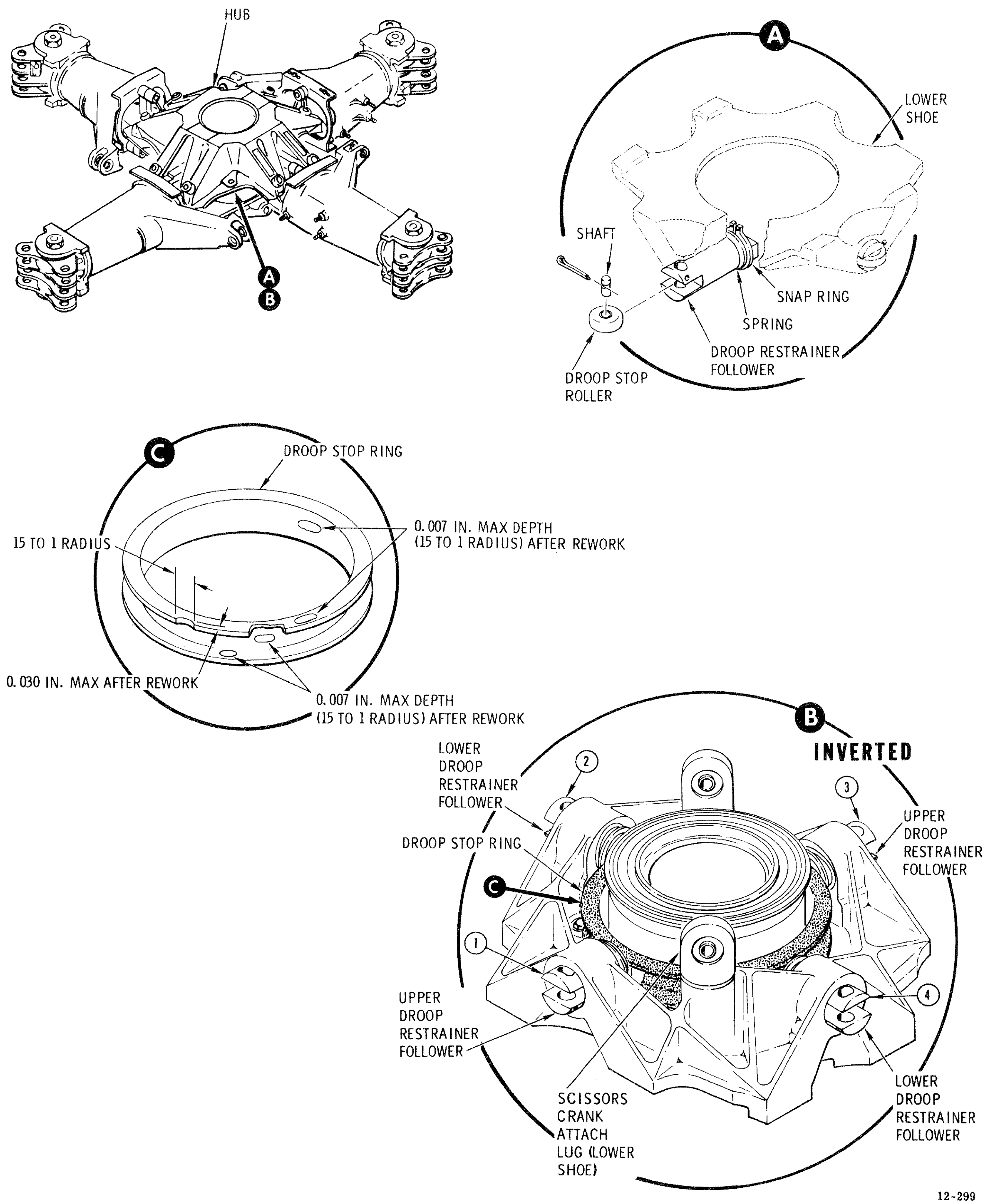


Figure 5-8. Droop Stop Repair.

NOTE

The followers can be easily rotated by the use of a non-metallic drift in the hole for the droop stop roller shaft.

f. Tilt the ring up on the number 3 and 4 followers as far as possible.

g. While pulling upward on the ring, turn the number 4 follower to rotate the T-head under the ring.

h. Continue pulling upward to the ring and turn the number 3 follower to rotate the T-head under the ring.

i. Withdraw the number 1 and 2 T-heads from the ring channel.

j. Remove the ring from the hub assembly.

k. Prepare for installation of a serviceable droop stop ring by orientating the number 4 follower for incorrect (upper) position. Orient the number 1 and 3 followers for correct (upper) position. Orient the number 2 follower for correct (lower) position.

l. Pull the number 1 and 2 followers outboard and engage the number 1 and 2 T-heads in the new droop stop ring channel.

m. Rotate the number 3 follower in the direction that has the greatest amount of clearance between the T-head and ring. Continue the rotation of the number 3 follower until the T-head is in the incorrect (lower) position and engages the ring channel.

n. Rotate the number 4 follower 180° which will position the T-head in the correct (lower) position and engage the ring channel.

o. Pull the number 3 follower outboard to disengage the T-head from the ring channel. Rotate the number 3 follower 90° until the T-head is perpendicular (vertical) to the ring channel.

p. Push down on the ring. Rotate the number 3 follower 90° until the T-head is in the correct (upper) position and engages the ring channel.

q. Check the number 1 and 3 followers for correct (upper) orientation. Check the number 2 and 4 followers for correct (lower) orientation. Check the droop stop ring for correct (level) installation.

r. Compress the droop restrainer spring and install the retaining (snap) ring in its groove on each follower.

s. Reinstall the droop stop rollers (para 5-20).

Table 5-3. Premaintenance Requirements for Installation of Main Rotor Head.

Conditions	Requirements
Special Tools	(T26 or T35) (T24) (T15) (T25)
Support Equipment	Hoist
Minimum Personnel Required	Two
Consumable Materials	(C31) (C60) (C47) (C58)

5-23. Installation — Main Rotor Hub.

CAUTION

The main rotor hub is a highly stressed component. Use extreme care to prevent it from striking any object. Any impact damage may require replacement of the main rotor hub.

a. Temporarily install lifting bracket (on aircraft so equipped) or eyebolts and lifting adapter (T15) on main rotor hub, taking care to tighten the bolts evenly and only finger tight.

b. Attach a suitable overhead hoist to the lifting bracket eye or lifting adapter eye.

c. Check that rotor mast is clean. Hoist main rotor hub and position it over the mast; then lower the hub onto mast.

NOTE

If the upper seal and seal retainer has been installed and if the hub seats itself on the mast far enough to expose the upper mast threads, accomplish step f below and then proceed directly to step k. If the hub does not seat itself on the mast, proceed as directed in steps d through j.

d. Hoist the main rotor hub off the mast and remove the seal retainer and the upper seal (fig. 5-5, sh 2). Discard the removed seal; removal causes permanent seal damage.

e. Lower hub onto mast.

f. Remove hub lifting adapter, hoist and eyebolts from hub.

g. Seat main rotor hub on mast by using main rotor hub driver (T25).

h. Using grease (C47), hand pack the exposed surface of the upper roller bearing cone (fig. 5-5, sh 2) to the level of the bore step against which the upper seal will be installed.

CAUTION

During the following step, make certain that seal lip faces upward.

i. Place a new upper seal (fig. 5-5, sh 2) inside the hub bore and seat it against the bore step by carefully and evenly tapping around the seal outside edge with a plastic mallet. Take care not to damage seal lip or deform seal case.

j. Install seal retainer (fig. 5-5, sh 2) inside seal and against bearing. Make sure that the flat side of the retainer faces downward (against bearing cone).

CAUTION

Do not apply excessive lubricant in *k* below. Excess can transfer down the mast to the swashplate and cause swashplate spherical bearing lining to be damaged.

k. Apply a light coat of lubricant (C60), to exposed threads on mast.

NOTE

Be sure that seal is properly positioned against retainer and that retainer does not pinch seal. Check by inserting a loop of 0.042-inch-diameter lockwire (C58), round feeler gage, or other suitable tool without sharp edges, between seal lip and retainer.

l. Install locknut, using wrench (T35) and **TORQUE NUT TO 200 FOOT-POUNDS.** Apply a coat of corrosion preventive compound (C31) to the screw holes in the mast nut.

m. Place retainer on nut and check retainer-to-locknut screw hole alignment. **INCREASE LOCKNUT TORQUE TO NOT MORE THAN 250 FOOT-POUNDS** to align screw holes in retainer and locknut.

NOTE

If holes in retainer and locknut cannot be aligned in the 200 to 250 foot-pound torque range, remove retainer and locknut and invert locknut.

n. Secure retainer to locknut with two screws; secure screws to retainer (fig. 5-3) with 0.032-inch lockwire (C57).

CAUTION

Ensure that no washers are used under screwheads and screwheads are free of burrs.

o. Place a straightedge across hub upper surface. **CHECK THAT SCREWHEADS DO NOT PROJECT MORE THAN 0.050 INCH ABOVE HUB UPPER SURFACE (FIG. 5-3).** This will provide adequate clearance between screwheads and drive shaft flange underside.

p. Install four pitch change rods to pitch housings (para 11-15).

NOTE

Be sure pitch change rods are reinstalled in the same positions from which removed.

CAUTION

Scissors crank must be positioned as shown in figure 5-2 with hole in crank web down and decal, if present, up.

q. Connect scissors assembly to main rotor hub (para 11-17).

r. Connect flexible boot to lower end of rotor hub liner.

s. Install main rotor drive shaft (para 6-10).

t. Install four main rotor blades (para 5-72).

u. Perform main rotor blade droop angle inspection (para 5-24) if rotor hub is new or a replacement.

Table 5-4. Premaintenance Requirements for Droop Angle Inspection.

Conditions	Requirements
Special Tools	(T19) (T20) (T21)
Support Equipment	Propeller Protractor
	Maintenance Stand

5-24. Inspection — Main Rotor Hub Droop Angle. Inspect droop angle of all four blades whenever a new or replacement main rotor hub has been installed or whenever excessive droop is suspected. **DROOP ANGLE OF ALL FOUR BLADES MUST BE BETWEEN 5 AND 6 DEGREES.** Inspect as follows:

a. Install cyclic lateral rigging fixture (T20), cyclic longitudinal rigging fixture (T21), and collective rigging fixture (T19), according to chapter 11.

b. Remove three of the blades (para 5-54). Position main rotor so that blade to be inspected is over the tailboom.

c. Place a propeller protractor on the top center of the main rotor drive shaft. Adjust protractor to the zero setting.

d. Place the protractor on the machined surface of the outboard end of the aft blade pitch housing, alongside the lead-lag bolt head. Measure and record the static droop angle.

NOTE

When checking main rotor blade droop angle, only the blade being checked should be installed. The others must be removed per blade removal instructions described in paragraph 5-54.

e. Repeat *a* through *d* above for the remaining blade positions.

f. **THE MAXIMUM ALLOWABLE STATIC DROOP ANGLE IS 6 DEGREES.** If the measured droop angle exceeds 6 degrees, inspect the striker plate and roller for excessive wear (para 5-20) and adjust the droop angle (para 5-25).

5-25. Adjustment — Main Rotor Hub Static Droop Angle. Whenever a new or replacement main rotor hub is installed, inspect and measure the static droop angle of all four rotor blades (para 5-24). **IF STATIC DROOP ANGLE EXCEEDS 6 DEGREES OR IS LESS THAN 5 DEGREES, ADJUST AS FOLLOWS:**

a. Remove main rotor blades (para 5-54).

b. (See fig. 5-7.) Use AN960C416, C416L, PD416 or PD416L flat washers to adjust the spacing between spacer and striker strip. Any one type, or combination, of the washers specified may be used; however, an identical washer selection (thickness) must be installed on each of the three bolts that secure the pitch control bearing assembly to the pitch housing.

c. Remove nuts and washers and separate spacer from striker strip. The use of one thick (0.016-inch) washer will raise the static droop angle approximately one-half a degree. Add sufficient washers to adjust droop angle to within the range of 5 to 6 degrees. Reinstall nuts and washers.

d. If more than 0.063-inch spacing is required, inspect striker plate and droop stop roller for excessive wear. Replace as required (para 5-20 and 5-22).

e. Repeat the inspection for static droop angle (para 5-24) to recheck the droop angle.

f. Reinstall blades. Check track of main rotor blades following installation of removed or replacement parts (para 5-43).

5-26. MAIN ROTOR DAMPER.

5-27. Description — Main Rotor Damper. A main rotor damper is mounted on each pitch housing of the rotor hub assembly (fig. 5-1). The damper is connected to the inboard trailing edge of the associated main rotor blade by a damper arm. Each damper limits blade movement on the lead-lag axis and absorbs lateral vibrations that may occur in the main rotor blade. The damper contains spring-loaded friction plates and associated parts in an oil filled housing. The damper functions as a rotary friction damper with either three or four consecutive separate torque stages through a specific range. A damper torque adjustment bolt retains the components inside the housing and provides the means for torque adjustment. The damper operates as a sealed unit and does not require regular servicing.

5-28. Inspection — Main Rotor Damper. *a.* Inspect each damper arm for cracks according to *b* through *f* below.

b. Remove attach pin from damper link.

c. Have an assistant move rotor blade forward into maximum lead position; then swing damper link outward for clearance.

d. Visually inspect the outboard clevis of damper arm (link), both top and bottom lugs, for any cracks that might extend from the bushing holes toward the edges of the lugs. Use a 5-power magnifying glass (minimum) to determine the presence of any cracks. Removal of paint is not necessary.

e. Repeat *b* through *d* above on remaining damper arms.

f. If cracks are found, replace damper arm with a serviceable arm that has been inspected according to *d* above.

g. When trouble has been experienced with rotor vibration, damper torque must be checked (para 5-29).

h. Reconnect damper arm to rotor blade with attach pin. Lock the pin and check tightness (para 5-73).

i. Inspect each main rotor damper for cracks, breaks, leaks, secure lockwire and secure attachment.

Table 5-5. Premaintenance Requirements for Checking Damper Torque.

Conditions	Requirements
Special Tools	(T29)
Support Equipment	Torque wrench (dial indicating type)
Minimum Personnel Required	Two

5-29. Torque Check — Main Rotor Damper (Installed). *a.* If recessed torque wrench adapter (T29) (fig. 5-2) is to be used disconnect damper arm (link) at blade and rotate link away from pitch housing. (See fig. 5-1 and 5-2.) If unrecessed adapter is to be used disconnect damper arm (link) from damper. Attach adapter to damper.

b. Using a dial indicator type torque wrench, move the arm slowly through the first stage travel. **MINIMUM TORQUE REQUIRED TO MOVE ARM MUST NOT BE LESS THAN 265 INCH-POUNDS. MAXIMUM TORQUE REQUIRED TO MOVE THE ARM MUST NEVER EXCEED 385 INCH-POUNDS.** If torque is outside of limits damper must be adjusted (para 5-38).

c. Reconnect damper arm (link).

5-30. Removal — Main Rotor Damper. (See fig. 5-2.) *a.* Unlock damper arm attaching pin, but do not remove.

CAUTION

While removing the bolt in next step, special care must be taken to hold the blade to prevent it from swinging freely and causing damage to the hub assembly. An assistant should hold the free-swinging blade while the damper is being replaced.

b. Remove cotter pin, nut, washer, and bolt that attaches damper arm to damper.

c. Swing damper arm aside.

d. Remove three nuts and washers that mount damper, and remove damper.

5-31. Disassembly, Inspection, and Repair — Main Rotor Damper. Refer to paragraphs 5-33 through 5-39 for tear-down inspection and permissible damper repairs.

5-32. Installation — Main Rotor Damper. (See fig. 5-2.)

NOTE

For maximum effectiveness, mixing of different part number damper assemblies on installation should be avoided whenever possible. Dampers should be installed so that torques of opposing dampers are within 40 inch-pounds of each other. A spread of more than 40 inch-pounds may cause a lateral vibration.

a. Position main rotor damper on studs of pitch housing.

NOTE

Slight looseness of damper attaching studs is allowable. The maximum displacement of the threaded end (tip) of the stud is limited to 0.001 inch. The large portion of the stud must be below the mounting surface of the pitch housing. Replace damper attaching studs not meeting these requirements.

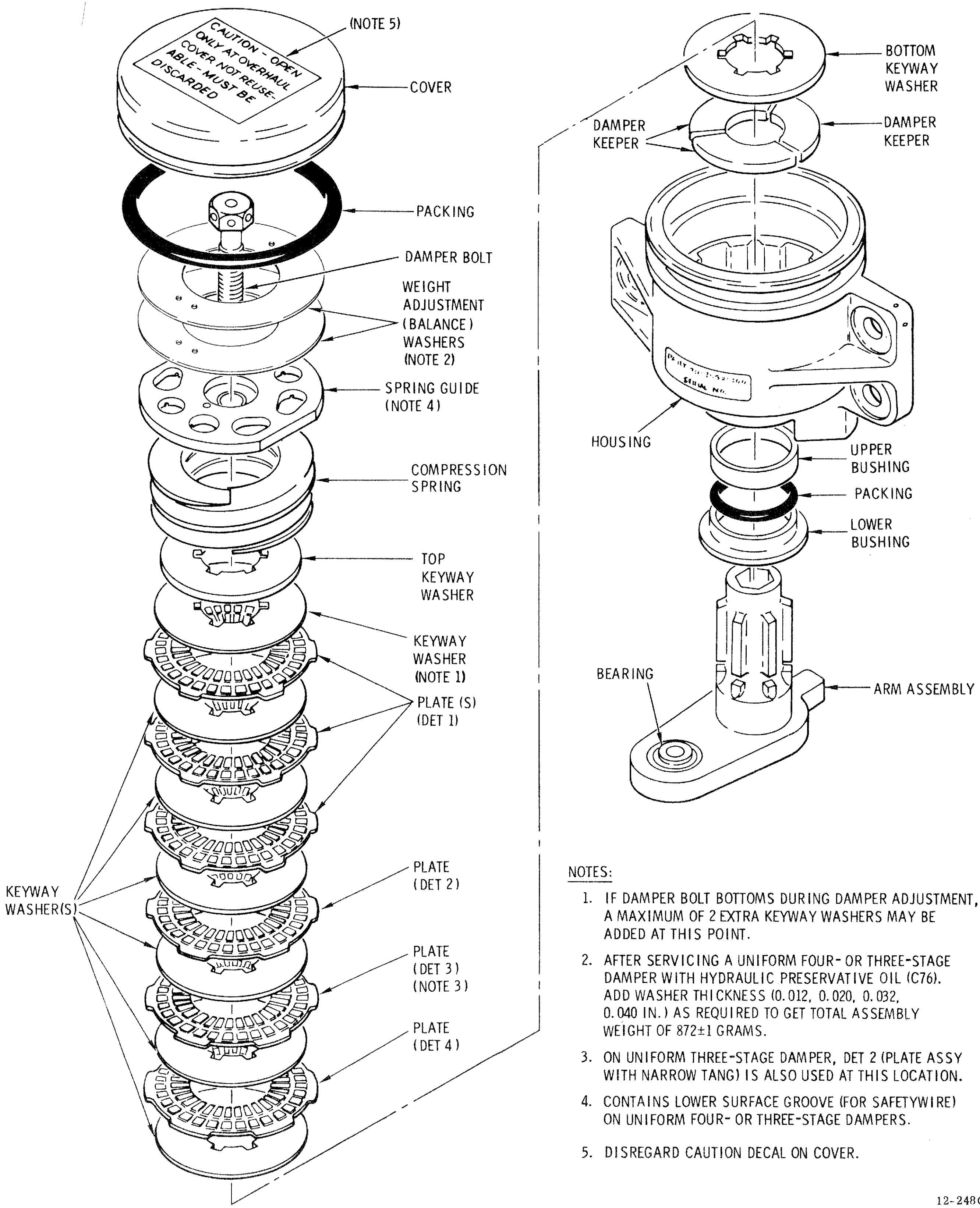
b. Install three washers and nuts. **TORQUE NUTS TO 50-70 INCH-POUNDS.**

c. Inspect for a minimum of 0.002-inch gap between damper and pitch housing. If gap is insufficient, rework pitch housing (para 5-10),

d. Align damper arm with damper; install bolt with head down, washer, and nut. Install washers under bolt head as required to provide additional clearance between bolt end and damper housing. TORQUE NUT TO 30-60 INCH-POUNDS. Install new cotter pin.

e. Install damper arm attach pin and lock in place. (Refer to para 5-73 for correct adjustment of damper arm (blade) attach pins).

5-33. Repair — Main Rotor Damper (AVIM). The following procedures apply to all damper configurations unless otherwise indicated. Data on damper configurations differences are noted in the text and on illustrations.



NOTES:

1. IF DAMPER BOLT BOTTOMS DURING DAMPER ADJUSTMENT, A MAXIMUM OF 2 EXTRA KEYWAY WASHERS MAY BE ADDED AT THIS POINT.
2. AFTER SERVICING A UNIFORM FOUR- OR THREE-STAGE DAMPER WITH HYDRAULIC PRESERVATIVE OIL (C76). ADD WASHER THICKNESS (0.012, 0.020, 0.032, 0.040 IN.) AS REQUIRED TO GET TOTAL ASSEMBLY WEIGHT OF 872±1 GRAMS.
3. ON UNIFORM THREE-STAGE DAMPER, DET 2 (PLATE ASSY WITH NARROW TANG) IS ALSO USED AT THIS LOCATION.
4. CONTAINS LOWER SURFACE GROOVE (FOR SAFETYWIRE) ON UNIFORM FOUR- OR THREE-STAGE DAMPERS.
5. DISREGARD CAUTION DECAL ON COVER.

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Figure 5-9. Rotor Blade Damper Assembly.

NOTE

The following reference names are used to indicate damper differences.

Damper 369A1400 is the "basic four-stage damper." This damper is not weight controlled.

Damper 369A1400-601 is the "uniform four-stage damper." This damper is identical to the 369A1400 damper except that its total weight is adjusted to 872±1 grams for weight uniformity. Weight adjustment is made with balancing washers that are secured to a grooved guide. Damper 369A1400-603 is the "uniform three-stage damper." This damper is weight controlled the same as the -601 but has only three friction stages and uses a special plate assembly retaining bolt.

5-34. Removal For Repair — Main Motor Damper. Refer to paragraph 5-30.

5-35. Disassembly — Main Rotor Damper (AVIM). (See fig. 5-9.) *a.* Cut and remove lockwire securing cover.

CAUTION

Damage to damper housing and distortion of the cover will result if removal is not carefully performed.

b. Using phenolic or hardwood wedge, gently pry cover from damper housing.

c. Drain oil from assembly into suitable container.

d. Remove packing and discard.

e. Cut lockwire from bolt and remove bolt by turning counterclockwise.

NOTE

Weight adjustment washers may be in place on top of the spring guide. It is not necessary to remove these washers to disassemble the damper.

f. Lift guide, spring, and keyway washer from housing.

g. Remove all washers and plates as a unit stack. Code the stack so it can be reassembled in the same buildup sequence after inspection.

h. Slide three damper keepers out of groove in arm assembly; remove keepers.

i. Remove arm assembly and packing from the housing and discard packing.

j. Clean all components, except arm assembly, with solvent (C94). Use a bristle brush as required and wipe dry with a soft lint-free cloth.

5-36. Inspection — Main Rotor Damper (AVIM). *a.* Inspect the spring guide for nicks, scratches and wear. Minor damage may be reworked by polishing with crocus cloth (C25) as long as OD of 2.125 inches is maintained.

b. Inspect compression spring for visible damage.

c. Inspect all keyway washers for nicks, burrs, scratches, corrosion (none allowed) and worn splines. **SPLINE WIDTH SHALL NOT BE MORE THAN 0.221 INCH IF WEAR IS EVIDENT.** Minor nicks, burrs and scratches may be reworked by polishing.

d. Inspect plates for scratches, nicks and voids in clutch facing (none allowed). V-grooves in facings shall not be completely worn away. **SPLINE WIDTH WEAR IS LIMITED TO 0.135 INCH FOR DETAIL 1 (FIG. 5-9), 0.254 INCH FOR DETAIL 2, 0.319 INCH FOR DETAIL 3, AND 0.492 INCH FOR DETAIL 4.** On uniform three-stage dampers, detail 3 has been eliminated and two each of detail 2 has been installed.

e. Inspect the three damper keepers for nicks, burrs, scratches and corrosion (light corrosion may be removed). Minor nicks, burrs and scratches may be reworked by polishing.

f. Inspect arm assembly for burrs, nicks, scratches and corrosion (machined and forged surfaces may have light surface corrosion removed). **SPLINE WIDTH WEAR IS LIMITED TO 0.212 INCH.** Inspect bushing surface area for wear; minimum diameter shall not be less than 0.878 inch after polishing.

g. Inspect the arm assembly bearing for looseness, binding (high rotational drag is normal), and galling or scoring in bore. **BEARING RADIAL PLAY IS LIMITED TO 0.010 INCH MAXIMUM; AXIAL PLAY IS LIMITED TO 0.020 INCH MAXIMUM.**

h. Inspect bushings for nicks, burrs and scratches. **INSIDE DIAMETER WEAR SHALL NOT EXCEED 0.887 INCH (LOWER BUSHING) AND 0.885 INCH (UPPER BUSHING.)**

i. Inspect the housing for cracks, worn or elongated mounting bolt holes (no cracks allowed) and corrosion (light corrosion may be removed). Restore chemical film protection.

j. Inspect the cover for distortion and cracks (none allowed).

NOTE

Any damper that fails to meet the above inspection requirements will be reassembled and returned for overhaul.

5-37. Preliminary Reassembly — Main Rotor Damper (AVIM).

NOTE

Apply a film of oil (C76) to all parts before assembly.

a. Install a new packing into place between upper and lower bushings as shown in figure 5-9.

b. Install arm assembly into the housing.

c. Position the three damper keepers and slide them into place in arm groove. Install bottom keyway washer with recess down to retain keepers.

d. Install the six plates, seven spacing keyway washers and top keyway washer in the same sequence as removed in disassembly.

NOTE

In the stack-up of plates (details 1 through 4, fig. 5-9) the plate with the widest splines is on the bottom with successively narrower splined plates towards the top.

e. Install the compression spring and spring guide. Install bolt; screw in until fit is a loose fingertight.

NOTE

The spring guide is to be installed with its flats perpendicular to the damper shaft arm. Hold the guide in this position while tightening the bolt. Tighten the bolt until the upper surface of the spring guide is flush with the housing.

Table 5-6. Premaintenance Requirements for Damper Adjustment, Phasing, and Reassembly.

Conditions	Requirements
Special Tools	(T7)
Consumable Materials	(C48) or (C76)

5-38. Adjustment and Phasing — Main Rotor Damper (AVIM). *a.* Clamp damper assembly holding fixture (T7) into a bench vise.

b. Loosen socket-head screw (fig. 5-10) that secures arm phasing plate.

c. Bolt damper housing in an upside down position on the fixture.

d. Install large diameter pin of torque adapter into matching hole of damper arm and swing adapter into position to align T-pin holes with hole in arm bearing. Install 0.25-inch diameter T-pin and pin spacer as shown in figure 5-10. Move damper arm to left until it contacts the lag stop of the damper housing. Shift the arm phasing plate so that plate leg ''X'' just touches the adapter pin with damper arm at lag stop. Tighten socket-head screw.

e. Move damper arm to right until adapter pin contacts phasing plate leg ''Y''.

f. Install dial indicating torque wrench on torque adapter and check torque required to move through first stage travel by slowly working damper arm from plate leg ''Y'' stop until second stage is felt. (First-stage travel is approximately 5 degrees for basic four-stage or uniform four-stage and 10 degrees for uniform three-stage damper assembly).

NOTE

As the damper arm is moved, the first-stage travel limit will be felt immediately. This is because twice the amount of torque is required to move into the second stage.

g. While in the first stage travel limits cycle the arm several times; then adjust damper bolt until torque required to move the arm falls within the range of **300-325 INCH-POUNDS.**

NOTE

Before each adjustment of the bolt arm, make sure the adapter pin (fig. 5-10) is at the ''Y'' leg position on phasing plate. If damper bolt bottoms before correct torque is reached, add an extra keyway washer.

h. After obtaining correct torque, position damper arm so that adapter pin is at the ''Y'' leg position on phasing plate. The damper is now correctly phased.

i. Remove T-pin, torque adapter and damper assembly from fixture.

5-39. Final Reassembly. — Main Rotor Damper (AVIM). Accomplish final reassembly of damper only after first-stage torque has been adjusted to 300-325 inch-pounds and the damper arm is correctly phased.

a. Secure head of bolt to the spring guide (fig. 5-9) with lockwire. Bend pigtail of lockwire down so that it will clear damper cover when installed.

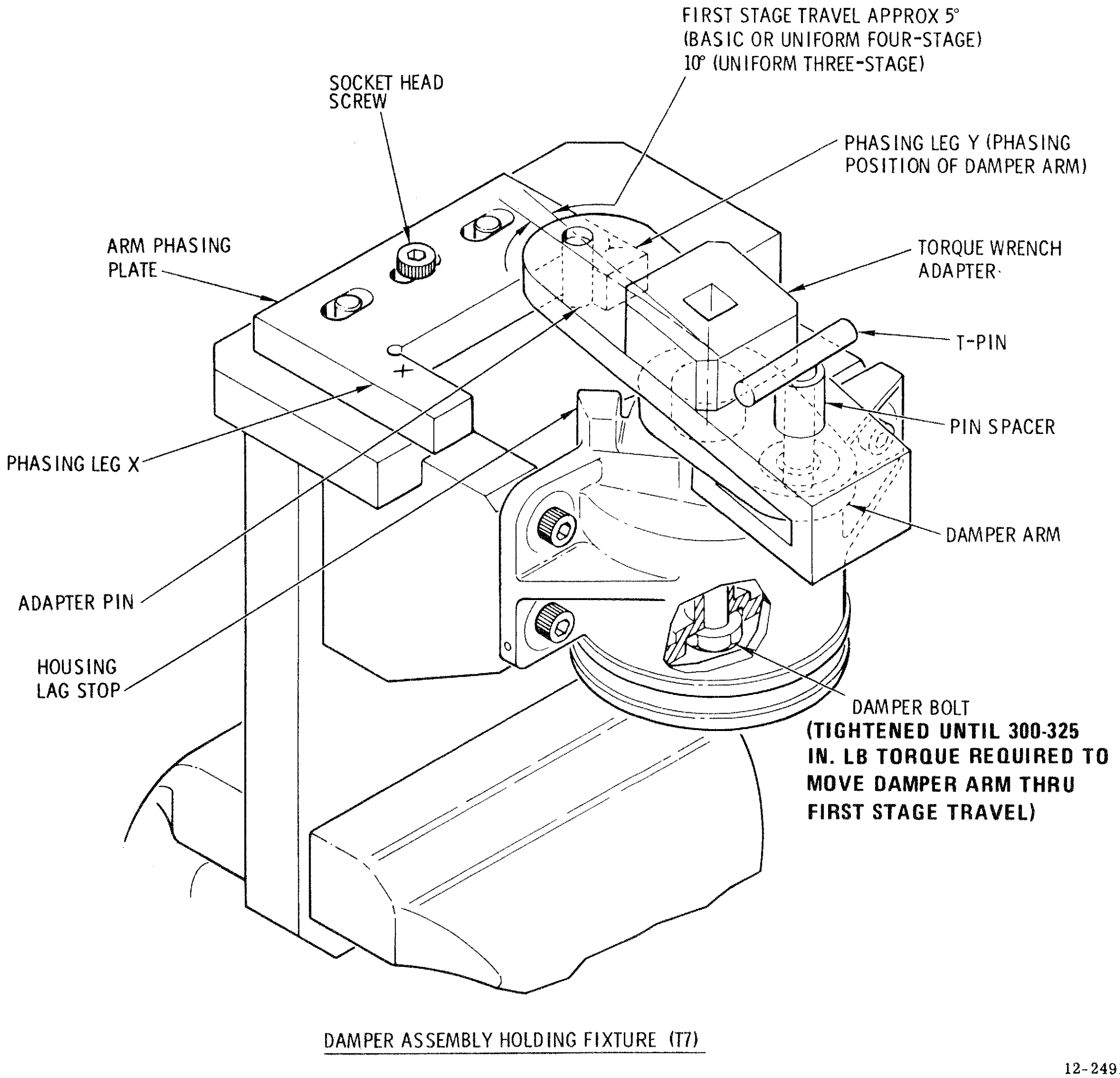


Figure 5-10. Damper Torque and Phasing Adjustment.

b. Apply a thin film of oil preservative (C76) to new cover packing and install in housing packing groove.

c. Service damper assembly with hydraulic preservative oil (C76) or hydraulic oil (C48). Fill housing until oil just covers the top surface of the spring guide.

d. Weight controlled damper assemblies (uniform three- or four-stage dampers only) must be weighed by the following procedure:

(1) Place the damper and cover on a suitable scale.

(2) Add weight adjustment washer thickness (0.012, 0.020, 0.032 or 0.040 inch) as required to bring the weight of the damper to 872±1 grams. Minor weight adjustments may be made by addition or subtraction of damper oil. Removal of oil may not uncover the top of the spring guide.

(3) Install two loops of 0.020-inch lockwire (C56) through the grooves on the underside of the spring guide. Before final balance, lockwire the balance washer(s) in place. Bend twisted pigtails of lockwire down to avoid interference with installed damper cover.

e. Set up damper assembly on a hand-operated arbor press, damper opening up.

NOTE

Lockwire must be installed in cover before pressing cover into place. The packing compresses to block the lockwire hole once the cover is installed.

NOTE

As the damper cover is seated in the following step, air may be trapped in the damper body. Use a feeler gauge or equivalent to gently work under the cap edge to relieve this pressure.

f. Install a piece of 0.020-inch lockwire (C56) through hole in edge of a damper cover. Twist lockwire approximately 5 turns. Locate cover squarely on housing and align lockwire hole in cover with lockwire hole in housing flange. Using an adapter from the arbor press, larger in diameter than the cover, press cover into place. Complete lockwiring of cover to hole in edge of housing mounting flange.

5-40. Installation — Repaired Main Rotor Damper. Refer to paragraph 5-32.

SECTION III MAIN ROTOR BLADES

5-41. MAIN ROTOR BLADES.

5-42. Description — Main Rotor Blades. Each of the four main rotor blades (fig. 5-11) is a balanced airfoil with a wrap-around aluminum alloy skin. The skin is bonded to an extruded aluminum alloy spar. One type blade (369A1100-601) has a bonded metal abrasion strip on the leading edge, the other type (369A1100) does not. Blade types may be intermixed without regard to part number. An upper and lower root fitting provide for blade attachment. A vibration absorber is installed on the blade lower inboard end. Two balance weights are installed in the tip end of each blade. A removable forward outboard tip cap is replaced with a tracking cap when blade tracking is performed. The blade trailing edge tab may be bent up or down. Bending the tab corrects the diving and climbing differences in individual blade during flight.

NOTE

The short inboard blade tab (fig. 5-14) is never used and has been deleted from later production blades. Blades may be interchanged without regard for the presence or absence of the inboard tab.

5-43. General — Main Rotor Blade Tracking. *a.* Tracking of the main rotor blades is accomplished with tracking tip cap reflectors and a strobe light. The tip caps are temporarily attached to the tip of each blade. The high-intensity strobe light flashes in-time with the rotating blades. The strobe light operates from the aircraft electrical power supply. By observing the reflected tip cap image, it is possible to view the track of the rotating blades. Tracking is accomplished in a sequence of four separate steps; ground tracking, hover verification, forward flight tracking and autorotation rpm adjustment.

b. The tracking procedure varies when tower-tracked blades are installed. Tower-tracked blades have been whirled and tracked to master blades. The blade tabs have been adjusted and pitch control rod length has been determined. A tower-tracked blade can be identified by a decal on the underside of the blade. The decal gives the basic length of the blade attaching pitch control rod. The decal shows the basic length for a blade attached to either the upper or lower strap pack span of the hub.

c. Before attempting to track the rotor blades, it is important to read and thoroughly understand the tracking sequence and interrelationship of the various adjustments. A review of the following should prove helpful.

(1) Install the tracking strobe light, blade tip cap reflectors and related equipment according to paragraph 5-44.

(2) Refer to table 5-7 for a condensed summary of the proper sequence for blade tracking.

(3) Ground tracking basically involves track-observation and adjustment at idle rpm and at flight rpm. Track at idle rpm is adjusted by the pitch control rods connecting the rotating swashplate and the blades. Ground track at flat pitch and flight rpm is corrected by blade tab adjustment.

(4) Hover track verification is a check for track variation that might occur between high rpm (flat pitch) ground track check and hovering. NO TRACK ADJUSTMENTS ARE MADE ON THE BASIS OF TRACK OBSERVATIONS DURING HOVERING. However, track variations should be noted and recorded for reference use during the check of track in forward flight.

(5) Forward flight tracking requires track observation during the following airspeeds and maneuvers. Flight track is corrected by making blade tab adjustments ONLY.

(a) Flight at 0 to 100 knots.

(b) Forty-five degree banked turns at 80 to 100 knots.

(c) Flight at 100 to 120 knots.

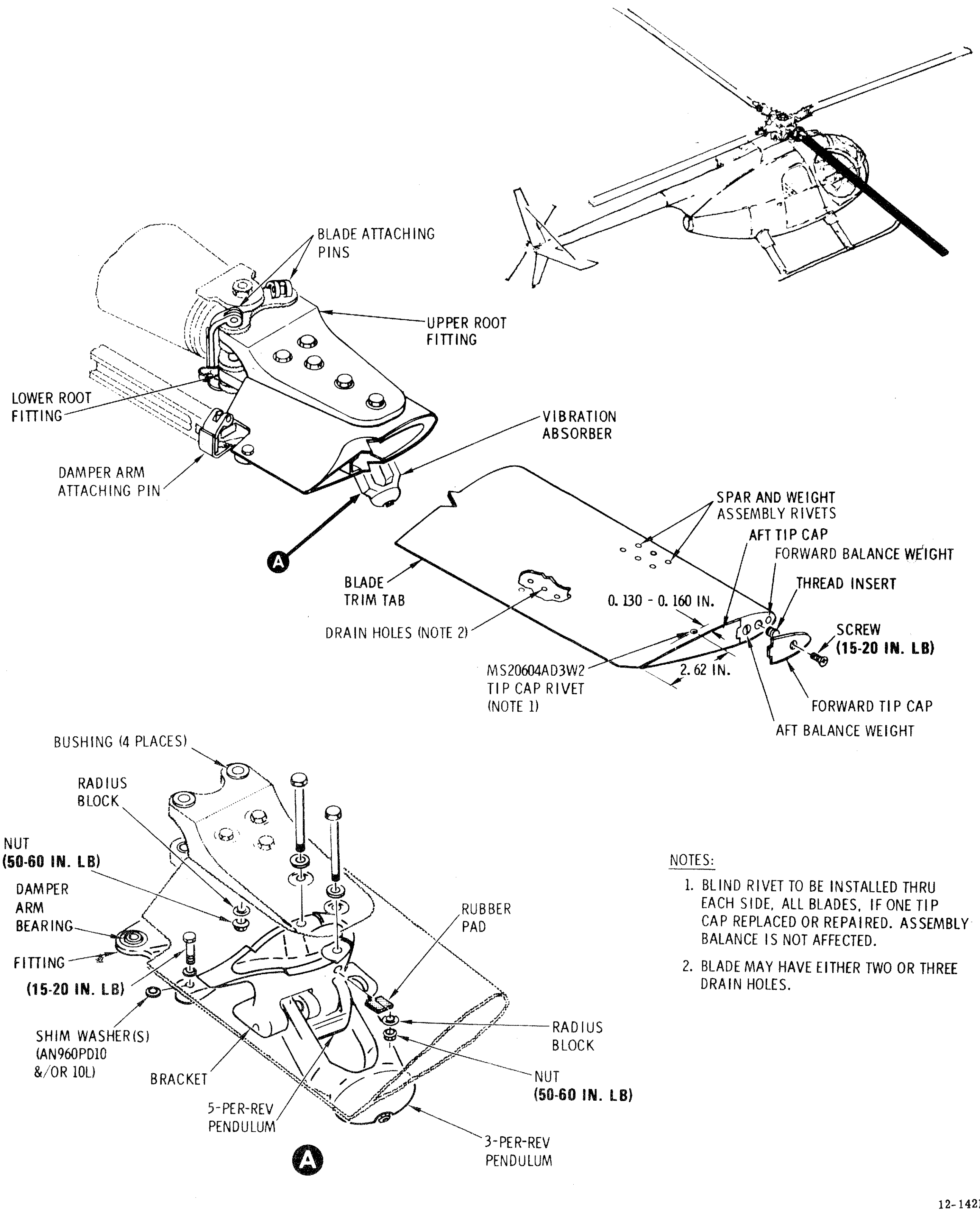


Figure 5-11. Main Rotor Blade Assembly.

Table 5-7. Blade Tracking Summary Procedure.

NOTE

The 0.5 inch-diameter tracking tip cap reflector size should be used as a guide for estimating track accuracy. For example: a reflector image displaced a half-diameter upward or downward indicates that the corresponding blade tip is approximately 0.25 inch out of track; one full reflector diameter indicates 0.5 inch out of track, etc. (See fig. 5-12.)

Observe ground track (para 5-48)		
SATISFACTORY	UNSATISFACTORY	Adjust pitch control rods (para 5-46 and/or blade tabs (para 5-47 as required, and repeat track observation.
Perform hover verification (para 5-49)		
SATISFACTORY	UNSATISFACTORY	Record track variation and proc with forward flight tracking.
Perform forward flight tracking (para 5-50)		
SATISFACTORY	UNSATISFACTORY	Adjust blade tabs (5-47) and repeat forward flight track
Perform autorotation rpm check (para 5-51)		
SATISFACTORY	UNSATISFACTORY	Adjust autorotation rpm (para 5-53) and repeat rpm check.
Remove tracking equipment and return aircraft to service.		

(6) Obtaining correct autorotation rpm consists of checking the main rotor rpm during stabilized autorotation; then adjusting rpm to specified limits. The autorotation rpm check must be made to ensure that track adjustments have not changed the rotor performance required for safe power-off landings.

Table 5-8. Premaintenance Requirements for Tracking Main Rotor Blades.

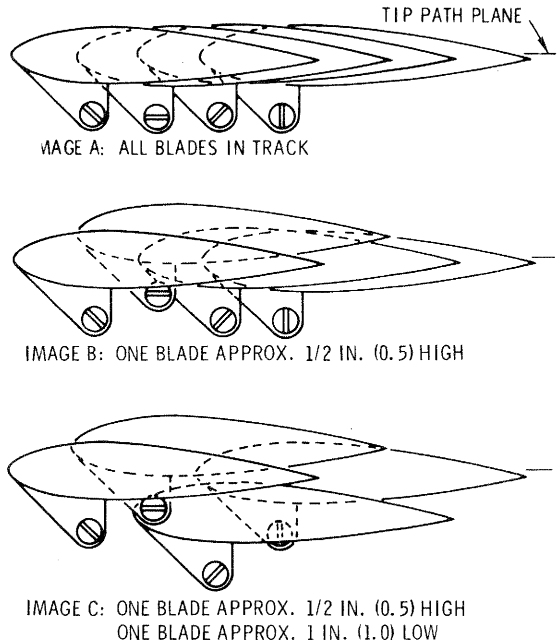
Conditions	Requirements
Special Tools	(T18)
Consumable Materials	(C103)

5-44. Installation — Main Rotor Blade Tracking Equipment. Prepare for main rotor blade tracking (fig. 5-13) by installing strobe light components furnished with the strobe light installation kit (T18). Install reflector type tip caps before tracking the rotor blades.

NOTE

The aircraft must be equipped with a set of unmixed tracking interrupters (369A9946-3, -5, -7 and -9, or 369A9946-23, -25, -27 and -29) installed on the rotating swashplate. A support bracket (369A9943) for the magnetic pickup is also installed on the stationary swashplate. (See fig. 5-13.) The duplicate set of interrupters and pickup bracket included in the strobe light installation kit should not have to be used unless existing parts are unserviceable. See figure 5-13 when replacement is necessary.

a. Install strobe light container on cargo floor. Use



CORRECTIVE ACTION

CONDITION	GROUND IDLE RPM	HIGH RPM AND FORWARD FLIGHT
IMAGE A	NONE REQUIRED	NONE REQUIRED
IMAGE B	SHORTEN PITCH CONTROL ROD (2ND BLADE)	MOVE TAB DOWNWARD (2ND BLADE)
IMAGE C	SHORTEN PITCH CONTROL ROD (2ND BLADE); LENGTHEN CONTROL ROD (3RD BLADE)	MOVE TAB DOWNWARD (2ND BLADE);MOVE TAB UPWARD (3RD BLADE)

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Figure 5-12. Typical Track Conditions and Adjustment.

four adjustable straps for tiedown. Straps should extend diagonally outward from container to tiedown fittings attached to the cargo floor.

NOTE

There are three Strobex Blade Trackers which can be used with the OH-6A: the 135M, the 135M5, and the 135M9. The instructions below are used with the 135M or the 135M5. When using the 135M9, the vernier on the back of the hand-held lamp must be turned full counterclockwise to deactivate the oscillator. With the vernier knob thus adjusted, operation of the 135M9 is exactly like that of the 135M or the 135M5 and the following instructions apply.

b. Remove the strobe light from container. Pass the light through left opening in the forward bulkhead to observer's seat.

c. Route all three electrical cables from the container through cutout opening in container; then close cover.

d. Attach 24-vdc power cord connector to the utility power receptacle. The receptacle is located at lower left corner of the forward canted bulkhead.

CAUTION

When work is being performed near the engine air inlet, use care to prevent entry of foreign objects. On aircraft without a filter installed, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove covers until work is

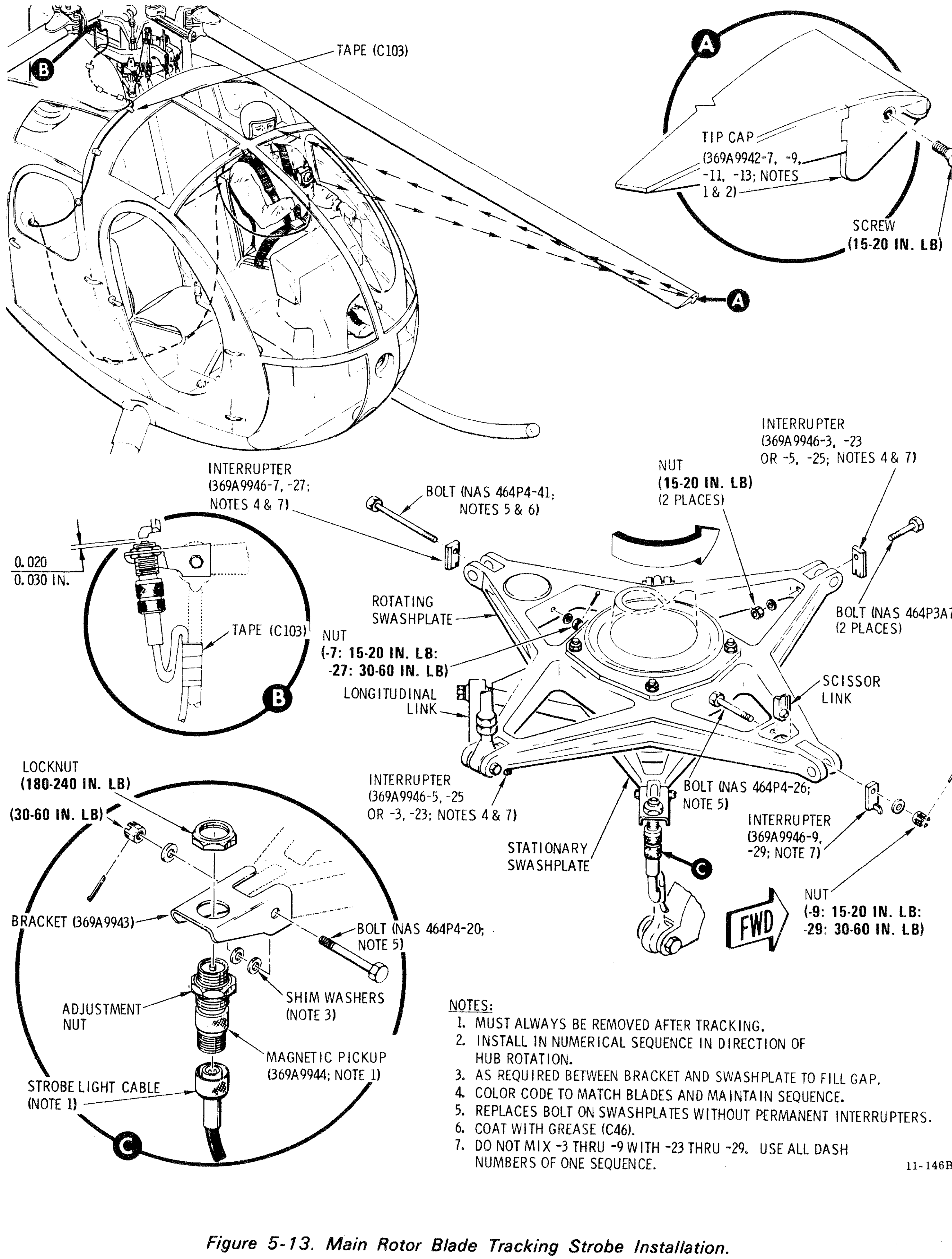


Figure 5-13. Main Rotor Blade Tracking Strobe Installation.

completed and debris is thoroughly cleaned out of the area. After removing covers, check that area around base of mast, inlet to plenum, and entire plenum chamber is clean.

e. Route triggering cable through the cargo door and up around engine air inlet. Use strips of pressure-sensitive tape (C103) to attach cable to fuselage and air intake.

f. Install magnetic pickup (369A9944) in bracket on stationary swashplate. Do not tighten top nut that secures pickup at this time. Plug the triggering cable into pickup and secure cable. Be sure that the cable will not interfere with extreme control movements.

CAUTION

GAP MUST NOT BE LESS THAN 0.020 INCH. Swashplate must be rotated to check the clearance between each interrupter and the pickup. DO NOT USE OVER 240 INCH-POUNDS TORQUE IN TIGHTENING NUT ON PICKUP.

g. **ADJUST GAP BETWEEN INTERRUPTERS AND PICKUP TO 0.020 TO 0.030 INCH. TORQUE TOP NUT SECURING PICKUP TO 180-240 INCH-POUNDS.**

h. Remove existing forward tip caps from rotor blades and install reflector type tip caps. Start with the 369A9942-7 cap, followed by the -9, -11 and -13 caps in dash number order and in direction of hub rotation. **TORQUE ATTACHING SCREWS TO 15 TO 20 INCH-POUNDS.** Take care not to overtighten screws.

i. To operate the strobe, set the auxiliary tank (or utility power) switch to AUX TANK (or UTILITY POWER). The rotor must be turning to trigger the lamp.

NOTE

Strobe light operation can be checked with no signal input (rotor not turning) by triggering the lamp several times. This action will flash the light. Do not hold the lamp trigger for long periods when rotor is not turning; to do so may damage lamp circuitry.

Table 5-9. Premaintenance Requirements for Blade Track Adjustment.

Conditions	Requirements
Special Tools	(T31)

5-45. Adjustment — Main Rotor Blade Track. Adjustment of pitch control rods will affect blade track at all rotor speeds. However, adjust blades only when necessary to establish acceptable track at ground idle speed; the blade tabs are used for all other track corrections.

NOTE

If pitch control rods are badly out of adjustment, or if the rod end bearings have been replaced, set the length of the affected pitch control rod(s). For blades with a tower-tracking data decal, set pitch control rods (between rod end bearing centerlines) to the applicable strap pack pitch link length specified on the decal of the mating blade. For blades without a decal, set pitch control rods to 6.60 inches (between rod end bearing centerlines) for the two blades attached to the hub upper strap pack, and 6.35 inches for the two blades attached to the lower strap pack. Observe the ground track (para 5-48) before making additional adjustment.

5-46. Adjustment — Main Rotor Blade Pitch Control Rod. Repeat this adjustment procedure as necessary to establish ground idle track.

a. Remove lockwire from both ends of pitch control rod and loosen the rod end jamnuts.

NOTE

The upper jamnut and rod end have left-hand threads.

b. To lower a blade tip, shorten the pitch control rod assembly by turning the rod in a counterclockwise direction as viewed from below. To raise a blade tip, lengthen the pitch control rod assembly by turning in a clockwise direction. One-sixth turn of the rod (one flat) will raise or lower the blade tip approximately 0.25 inch.

CAUTION

After adjusting pitch control rods, accomplish the following.

c. Check that the rod end threads are engaged far enough to block the rod body inspection (witness) hole. Check upper jamnut positioning for possible contact with the pitch housing in the full up travel position. Ensure that at least four threads are exposed.

d. After adjusting the pitch control rod length, center each rod end in its fitting and hold while tightening the jamnuts. Safety with lockwire.

5-47. Adjustment — Main Rotor Blade. Once ground idle track is obtained, all remaining tracking correction is accomplished by VERY SLIGHT bending of the various blade tab zones with tab bending tool (T31). Different zones of the tabs are used to adjust blade track at different airspeeds. (See fig. 5-14.) In general, tab Zone A is used for high rpm, flat pitch ground tracking (103% N2) and Zones C, D, and E are used for tracking at the higher airspeeds. Zone B is used to supplement Zone A track correction when maximum tab (5 degrees) has been applied to Zone A; Zone B may also be used, if necessary, for correction in the 0 to 100 knot airspeed range.

CAUTION

Restrict bending to very small increments so that the bonded trailing edge joint between the upper and lower skins will not be damaged. ALL TABS MUST NEVER BE DISPLACED MORE THAN 5 DEGREES ABOVE OR BELOW THE NEUTRAL POSITION (parallel to the chordline).

a. To lower the blade tip that tends to climb during ground tracking at high rpm or during forward flight, bend the appropriate tab section slightly downward; to raise the tip of a blade that tends to descend; bend the tab slightly upward. If only slight track correction is necessary, limit tab bending to the width of the bending tool. If more correction is necessary, bend a slightly wider section of the tab. Avoid excessive rebending of tabs by using small adjustments until the necessary result is obtained.

NOTE

Tab zones on the same blade can require bending in opposite directions. For example, after bending tab Zone C downward to get good tracking at 60 to 90 knots; it might become necessary to bend tab Zone E upward to correct track at redline airspeed. In any case, do not use larger tab corrections than are actually necessary.

b. Each time blade tabs are adjusted, recheck the ground idle track and readjust if necessary (para 5-48).

c. After completion of forward flight tracking, check autorotation rpm (para 5-51).

5-48. Ground Tracking — Main Rotor Blades. For best results, tracking should be performed under calm air conditions. Wind velocity should not exceed 6 knots during preliminary adjustments nor 3 knots for the final adjustment. Accurate adjustment of the initial ground track is very important. In most instances, forward flight tracking problems can be avoided or greatly reduced by setting the initial track as nearly perfect as possible. The tolerance specified in the following instructions is the minimum permissible deviation rather than the desired goal.

NOTE

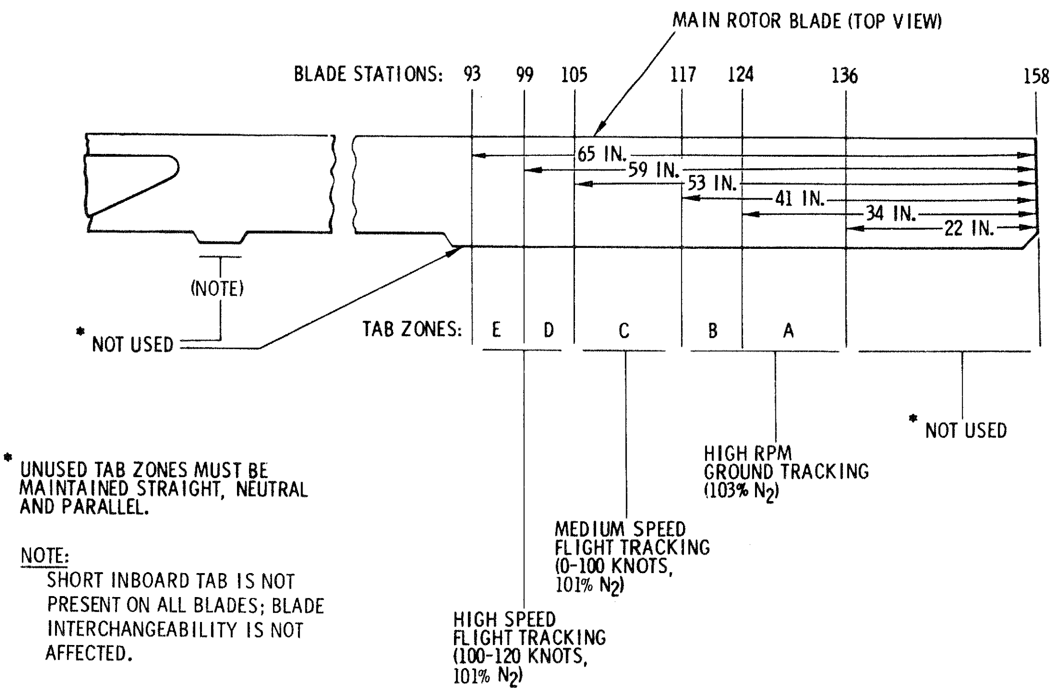
Both tower-tracked and previously used blades have been pretracked. Tower-tracked blades have a decal on the underside of the blade. Do not adjust tab zones between blade station 93 and 136 (fig. 5-14) until tracking observations indicate need for adjustment. If replacement tower-tracked blades or rotor hub have been installed, or if a replacement pitch control rod assembly or rod end bearing has been installed, set the length of each pitch control rod assembly according to paragraph 5-45 before performing ground tracking.

a. Before tracking blades that are NOT new, check that the normally straight (not to be used) tab areas shown in figure 5-14 are in the neutral position (centered on chordline) and straight.

b. Load aircraft to a gross weight of 1900 to 2200 pounds.

c. With collective pitch stick full down, operate engine for a brief period at 100% N2 and then reduce rpm to ground idle (approximately 62% N1). This will ensure that the blade dampers are correctly positioned.

d. Observe tracking tip cap reflector image by directing strobe light beam toward blade tip path; and



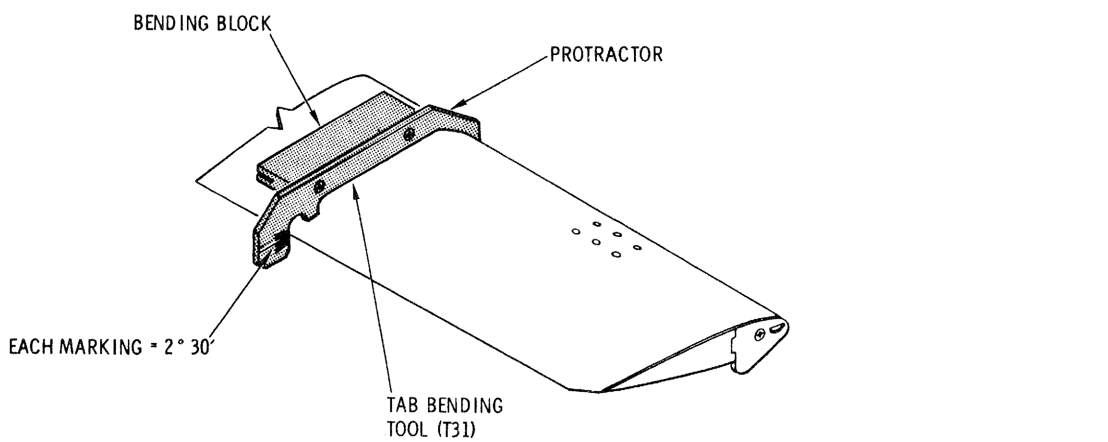


Figure 5-14. Blade Tab Angle Adjustment.

then beam slowly back and forth until reflector images can be seen clearly. The tracking image should appear directly in front of the aircraft or slightly off the aircraft centerline.

e. If blades are in track, the tip cap reflector image pattern should resemble Image A, figure 5-12 (none of the blade tips more than 1/4 of one tracking reflector diameter (1/8 inch) above or below the adjacent reflectors).

(1) If blades are in track, proceed with *f* below.

(2) If blades are out of track, adjust pitch control rods *a* and repeat *c* through *e* above.

f. With collective pitch stick full down, increase engine speed to 103% N2; then observe tip cap reflector image to see if blade track has changed the ground idle track.

(1) When all four blades are in track within 1/4 of one reflector diameter (1/8 inch), the ground track is good; proceed with hover verification (para 5-49).

(2) When a blade is out of track, adjust blade tab Zone A (para 5-47) until the high rpm ground track is within tolerance.

NOTE

Increased rotor stabilization may be obtained by minimizing blade climb between ground idle and 103% N2 by adjustment in tab Zone A.

5-49. Hover Track Verification — Main Rotor Blades. Hover track verification must be performed after ground tracking and before forward flight tracking. Verification is only a CHECK of the hover track. DO NOT adjust pitch control rods or blade tabs because of the track picture observed during hovering. Tab adjustments often cause variation between ground track and hover track. A large track variation may indicate that one or more blades is beyond its chordwise balance tolerance. However, this can only be determined during forward flight tracking. Perform hover track verification as follows:

NOTE

The collective pitch stick may be "heavy" after tracking reflectors are mounted on the blade tips. This condition is not unusual and may be disregarded.

a. Verify the aircraft gross weight (1900 to 2200 pounds).

b. With collective pitch stick full down, increase N2 to 101%.

c. Observe tracking reflector images to verify that ground track is within limits.

d. With aircraft in a stable hover, observe the reflector images. If one or more blades are out of track, record the condition for reference during forward flight tracking.

e. Proceed with forward flight tracking (para 5-50).

5-50. Forward Flight Tracking — Main Rotor Blades. Forward flight tracking should be performed whenever vertical flight vibrations indicate that the blades may be out of track.

a. Verify the aircraft gross weight (1900 to 2200 pounds).

b. Perform flight tracking from hover up to 100 knots at 101% N2. If flight track varies from hover track more than 1/2 inch, bend tab Zone C (5 degrees maximum) to limit the variation to 1/2 inch. If variation is still excessive, bend tab Zone B (5 degrees maximum). Refer to paragraph 5-47.

c. Check autorotation rpm (para 5-51); rpm must not be less than 450.

d. Perform a series of 45-degree banked turns at 80 to 100 knots and observe track change from level flight. If any blade climbs or dives more than 1 inch out of track with the others, the chordwise balance (center of gravity) of that blade is beyond its tolerance and the blade must be replaced. (Hover track variations of this type that do not repeat during the banked-turn maneuver may be ignored).

e. Perform flight tracking at 100 to 120 knots and 101% N2. If necessary, adjust tab Zone D (5 degrees maximum) to limit track variation to 1/2 inch or less and to minimize excessive vertical (1-per-rev) vibration. If variation is excessive, bend tab Zone E (5 degrees maximum). Refer to paragraph 5-47.

NOTE

When deciding whether blade track is acceptable, the overall vibration level of the aircraft should be the determining factor. Some combinations of rotor blades might produce a higher 4-per-revolution vibration as the blade tips are brought into close track; in such cases, the lowest vibration level is preferred, even though the observed blade track may be beyond the specified tolerances.

f. After flight tracking is completed, perform an autorotation rpm check (para 5-51).

NOTE

Correction may be made for cyclic feedback after blade track and vibration level are acceptable. The correction is made by separating the blade track between the blades attached to the hub upper strap pack and those attached to the lower strap pack by up to 1/4 inch (the amount of strap pack offset).

5-51. Check — Main Rotor Autorotation Rpm. An autorotation rpm check is required after each blade tracking operation and whenever the rpm is outside the limits given in table 5-10. Check rotor rpm according to paragraph 5-52 and make adjustments according to paragraph 5-53.

5-52. Check — Main Rotor Autorotation Rpm. *a.* Load aircraft to a gross weight of 1900 to 2200 pounds.

b. Perform a practice autorotative descent according to TM 55-1520-214-10, taking care not to allow rpm to exceed the rotor speed limitations.

c. During autorotative descent, take careful note of stabilized autorotative rpm at one of the gross weight/density altitude combinations given in table 5-10.

d. After landing, compare observed rpm with the values given in table 5-10. If observed rpm is within the limits given in the table, rpm setting is correct. If limits were exceeded, make corrective adjustments according to paragraph 5-53 until rpm falls within limits.

NOTE

When the rotor track autorotation rpm adjustments have been satisfactorily accomplished, remove tracking equipment that must not remain installed (fig. 5-13) before returning aircraft to service. Take care to properly tighten and safety all bolts after they have been reinstalled.

5-53. Adjustment — Main Rotor Autorotation Rpm. *a.* Hold the lower end of collective pitch mixer control rod (fig. 11-1) with a wrench to restrain it against rotation and loosen the upper rod end jamnut.

b. Remove bolt that attaches mixer control rod upper rod end to bellcrank and shorten length by turning rod end clockwise (as viewed from above) to decrease rotor rpm, or counterclockwise to increase rpm. Each 1/2 turn of the rod end will change rotor speed by approximately 6 rpm.

CAUTION

After adjusting collective pitch mixer control rod, check to make sure that the threads of the rod are engaged far enough to block the rod body inspection (witness) hole.

c. After adjusting collective pitch mixer control rod length, install bolt and check for control interference as described in *d* and *e* below.

d. Check for interference between the lateral bellcrank and longitudinal pitch idler (fig. 11-1) of the mixer controls by positioning the collective stick full up and the cyclic stick at the aft stop. If interference exists, lengthen the collective pitch mixer control rod enough to eliminate the interference; then lengthen the four blade pitch control rods equally by the same amount.

e. Check for interference between the rotating swashplate and longitudinal pitch idler (fig. 11-1) of the mixer controls by positioning the collective stick on the downstop and the cyclic stick full forward and full left. If less than 0.150-inch clearance exists, shorten the collective pitch mixer control rod enough to establish clearance (0.250 inch preferred); then shorten the four blade pitch control rods equally by the same amount.

f. After accomplishing checks and adjustments in *c* through *e* above, install nut, washer and new cotter pin. Tighten the rod assembly jamnut while holding rod end centered in the bellcrank fitting.

g. If more adjustment is needed, change the length of all four pitch control rods as necessary to obtain additional autorotation rpm. Be sure to lengthen or shorten all four rods exactly the same or blade track will be changed. One flat (1/6 turn) of the control rod body will cause approximately 8 rpm change in autorotation rpm. Refer to paragraph 5-46.

Table 5-9A. Premaintenance Requirements for Balancing Main Rotor Hub and Blades.

Condition	Requirements
Special Tools	(T40)
Consumable Materials	(C103)

5-53A. General — Main Rotor Hub and Blade Balancing.

a. Main rotor balance is accomplished using instrumentation that locates and measures vibrations due to main rotor imbalance. Data provided by the instrumentation is plotted on a chart designed to indicate how much weight shall be added to or subtracted from screws installed through the hollow lead-lag pivot bolts (fig. 5-5).

NOTE

Main rotor blades must be in track before attempting to balance the main rotor.

b. The balancing kit (T40) contains a procedural checklist, charts and all equipment necessary to accomplish main rotor balance.

c. Installation of main rotor balance equipment (fig. 5-14A) should only be accomplished using the balance checklist provided in balance kit (T40) and instructions provided in TM 55-4920-402-13&P.

CAUTION

When working near engine air inlet, care should be taken to prevent foreign objects dropping into engine inlet.

Observe all NOTES and CAUTIONS in balance kit checklist and TM 55-4920-402-13&P, relating to the installation and security of equipment, including cables, fastened to the exterior of the helicopter.

d. Balance the main rotor according to instructions in the balance kit checklist. Acceptance criteria for balance and vibration are contained on each balance chart. Figure 5-5 shows balancing hardware is installed and sealed at the lead-lag pivot bolts. Table 5-9B lists approved balancing hardware and gives the weight of each item. The balance charts indicate an option of adding or subtracting weight. Keep overall weight at a minimum. Always remove weight when possible.

Table 5-9B. Main Rotor Balancing Hardware

Hardware	Unit Weight (Grams)
NAS603-56 Screw	11.10
NAS603-64 Screw	12.25
MS21042-3 Nut	1.90
AN960C10 Washer	0.63
AN960C10L Washer	0.32
AN970-3 Washer	4.33
HS1554 Washer	1.00
HS1555 Washer	2.80

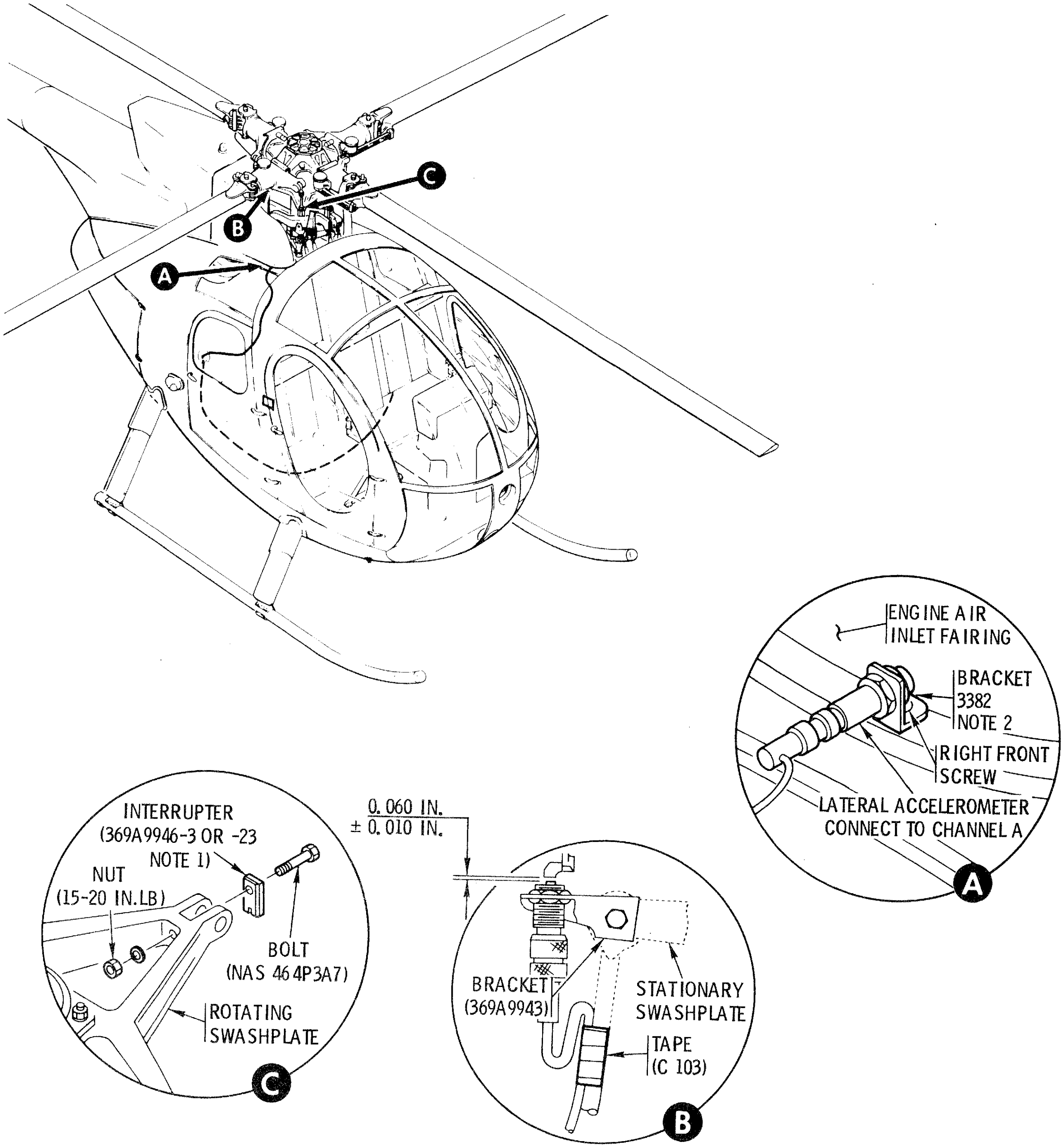
5-54. Removal — Main Rotor Blade. *a.* Using masking tape and/or grease pencil, mark each blade and its respective pitch housing links so that blades can be reinstalled in the same relative positions.

b. Have an assistant relieve load on blade attachment points by supporting blade tip from below. See figure 5-11.

c. Unlock and remove damper arm attaching pin and blade attaching pins.

d. Slide blade from lead-lag links.

5-55. Inspection — Main Rotor Blade (General). The inspection requirements for main rotor blades are divided into two groups; normal inspection criteria (para 5-56), and restricted service life inspection criteria (para 5-57). Blade damage and repairs that do not exceed the limits for normal operation (described in para



NOTES:
1. REPLACE INTERRUPTER 369A9946-3 OR -23 WITH VIBREX BALANCE KIT DOUBLE INTERRUPTER NO. 3379.
2. INSTALL WITH VIBREX BALANCE KIT BRACKET NO. 3382.

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Figure 5-14A. Main Rotor Blade Balance Equipment Installation

Table 5-10. Autorotation Rpm Chart.

Gross Wt (lb)	Sea Level	1000 ft	2000 ft	3000 ft	4000 ft	5000 ft
	Stabilized Autorotation Rpm at Density Altitude					
1900	447-457	454-464	461-471	468-478	475-485	482-492
2000	459-469	466-476	473-483	480-490	487-497	494-504
2100	471-481	478-488	485-495	492-502	499-509	
2200	483-493	490-500	497-507	504-514		

NOTE:
1. Chart values based upon 15°C FAT. At sea level, 8°C temperature change is equal to 1000 ft change in density altitude.
2. Perform autorotation rpm checks at gross weight/density altitude combinations for which rpm values are given. Blank spaces indicate that application of collective pitch may be necessary to avoid rotor overspeed.

5-56) permit the blade(s) to be continued in service without restrictions. Damage and repair that are within the limits given in restricted service life inspection criteria (described in para 5-57) impose additional daily inspection requirements and reduce the service life of the blades.

5-56. Inspection — Main Rotor Blade (Normal Criteria). *a.* Inspect skin for evidence of cracks and holes. Cracks or holes in blade skin, regardless of location, shall be cause for rejection of the blade (except for damage within the restricted service life criteria of para 5-57). **SCRATCHES, DENTS, NICKS AND OTHER SURFACE DEFECTS IN THE BLADE SKIN ARE LIMITED ACCORDING TO *b* and *c* BELOW.**

CAUTION

The repairable limit dimensions presented in the following procedures apply to surfaces that have not been repaired before. Ensure that material has not been removed from damaged areas before determining that the damage depth limit has not been exceeded.

NOTE

Use a dial indicator to check depth of blade dents and scratches.

b. Limitations for surface scratches, nicks, and gouges in seven areas of the blade skin are described below. See figure 5-15 for area location. Scratches that do not penetrate the clad surface are acceptable without rework in all areas except area 5.

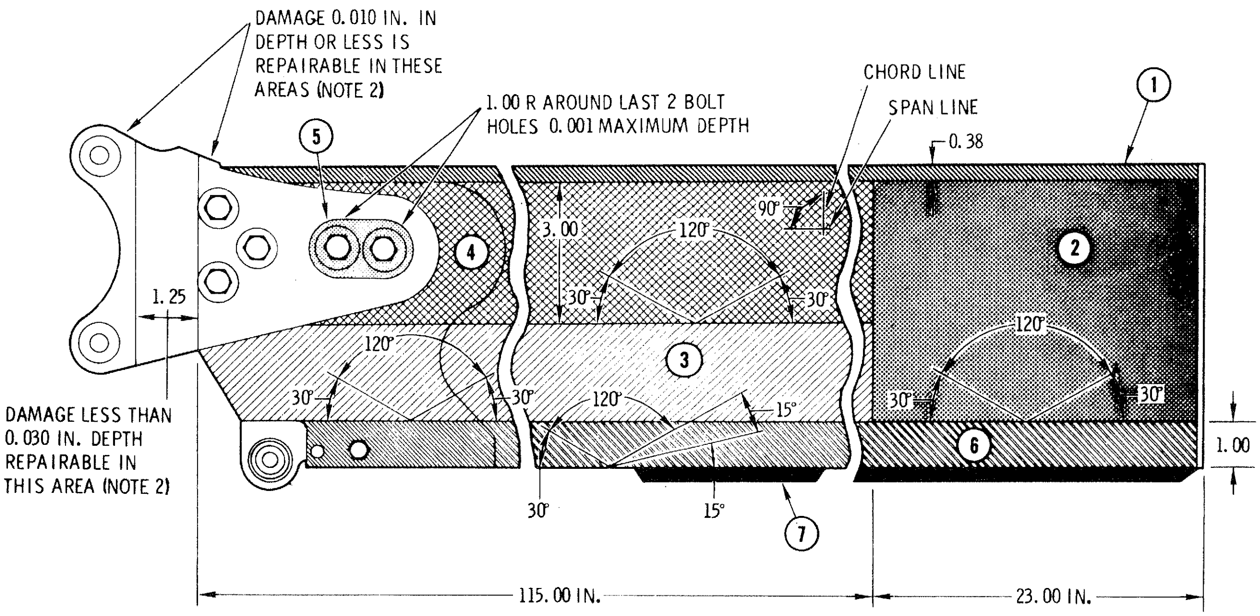
(1) AREA 1: Minor scratches, nicks, and gouges without skin penetrations are acceptable without repair on blades not equipped with the metal abrasion strip. On blades with the metal abrasion strip (fig. 5-16A) inspect the strip for erosin. Abrasion strip erosion may extend the length of the abrasion strip, but may not exceed 0.40-inch width at any point. Skin erosion under a worn abrasion strip may not exceed 0.30-inch width over a 9.0-inch length measured from the blade tip. Inspect for adhesive erosion at the interface with the abrasion strip. If adhesive erosion is found, repair as instructed in paragraph 5-67A. Skin deformation with holes is cause for replacedment. On blades not equipped with the metal abrasion strip, leading edge erosion may be repaired if the skin is not eroded through. On all blade types, if the skin is eroded through, no matter how slightly, the blade must be immediately replaced.

(2) AREA 2: Scratches to 0.005 inch deep if oriented 0 to 30 degrees from spanline and to 0.003 inch deep if oriented between 30 and 90 degrees from spanline are acceptable without repair. Nicks and gouges not exceeding 0.003 inch in depth are acceptable without repair and 0.005 inch with repair.

(3) AREA 3: Scratches to 0.005 inch deep if oriented 0 to 30 degrees from spanline and 0.003 inch deep if oriented between 30 and 90 degrees from spanline shall be removed. No sharp nicks or gouges requiring more than 0.005 inch removal of skin surface are permissible.

(4) AREA 4: Scratches to 0.003 inch deep if oriented 0 to 30 degrees and to 0.002 inch deep if oriented between 30 and 90 degrees from spanline shall be removed. No sharp nicks or gouges requiring more than 0.003 inch removal of skin surface are permissible.

(5) AREA 5: Scratches exceeding a depth of 0.001 inch shall be cause for rejection. Remove scratches to a depth of 0.001 inch. The area inside of the outboard two root fitting bolt holes (includes both upper and lower surfaces) and including the skin surface adjacent to these bolt holes within a radius of 1.00



SCRATCHES (NOTE 1)				GOUGES AND NICKS (NOTE 1)	
AREA	ORIENTATION FROM SPANLINE	MAXIMUM DEPTH WITHOUT REPAIR	MAXIMUM DEPTH WITH REPAIR (NOTE 4)	MAXIMUM DEPTH WITHOUT REPAIR	MAXIMUM DEPTH WITH REPAIR (NOTE 4)
1	0°-90°	MINOR WITHOUT SKIN PENETRATION	MINOR WITHOUT SKIN PENETRATION	MINOR WITHOUT SKIN PENETRATION	MINOR WITHOUT SKIN PENETRATION
2	0°-30° 30°-90°	0.005 0.003	(NOTE 4) (NOTE 4)	0.003 0.003	0.005 0.005
3	0°-30° 30°-90°	NONE NONE	0.005 0.003	NONE NONE	0.005 0.005
4	0°-30° 30°-90°	NONE NONE	0.003 0.002	NONE NONE	0.003 0.003
5 (NOTE 2)	0°-90°	NONE	0.001 (NOTE 3)	NONE	0.001 (NOTE 3)
6	0°-15° 15°-30° 30°-90°	0.005 0.003 NONE	(NOTE 4) 0.005 0.003	NONE NONE NONE	0.005 0.005 0.005
7	0°-90°	(NONE)	0.005 (NOTE 3)	NONE	0.005 (NOTE 3)

NOTES:

1. ALL DIMENSIONS ARE IN INCHES.
2. DAMAGE LIMITS APPLY TO B⁻⁻H UPPER AND LOWER ROOT FITTINGS.
3. INSPECTION UNDER 5 X MAGNIFICATION (MINIMUM) REQUIRED.
4. REFER TO INSPECTION CRITERIA FOR REDUCED LIFE BLADE REPAIR LIMITS.

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Figure 5-15. Main Rotor Blade Normal Damage and Repair Limits for Scratches, Gouges, and Nicks.

inch from each hole shall be free of scratches, nicks, or gouges as would be detected under 5 × magnification (minimum).

(6) AREA 6: Scratches to 0.005 inch deep and oriented 0 to 15 degrees from spanline are acceptable without repair. Scratches to 0.005 inch deep and oriented from 15 to 30 degrees from spanline shall be removed. Scratches 0.003 inch deep and oriented from 15 to 30 degrees from spanline are acceptable without repair. Scratches to 0.003 inch deep and oriented from 30 to 90 degrees from spanline shall be repaired. Nicks and gouges not exceeding 0.005-inch depth shall be repaired.

(7) AREA 7: Scratches, nicks, or gouges up to 0.005 inch deep and detectable with 5 × magnification are not acceptable without repair. Defects beyond 0.005-inch depth shall be cause for rejection.

c. Depth limitations for surface dents or depressions in eight areas of the blade skin are described below. See figure 5-16 for area location. A dent or depression is defined as a smooth depression or discontinuity with no sharp changes in section.

(1) AREA A: Dents and depressions exceeding 0.015 inch without sharp changes in section to a maximum of 0.062 inch shall be repaired.

(2) AREA B: Dents and depressions to a maximum of 0.010 inch are acceptable without repair. No repairs are permitted in this area.

(3) AREA C: No repairs permitted.

(4) AREA D: Dents and depressions exceeding 0.005 inch without sharp change in section to a maximum of 0.030 inch shall be repaired.

(5) AREA E: Dents and depressions exceeding 0.010 inch without sharp change in section to a maximum of 0.040 inch shall be repaired.

(6) AREA F: Dents and depressions exceeding 0.010 inch without sharp change in section to a maximum of 0.040 inch shall be repaired.

(7) AREA G: No repairs permitted.

(8) AREA H: Dents and depressions exceeding 0.010 inch without sharp changes in section to a maximum of 0.040 inch shall be repaired.

d. Inspect surface areas of upper and lower root fittings (other than Area 5) for evidence of nicks, scratches, and wear spots. Nicks, scratches, and wear spots deeper than 0.010 inch in the attachment lug area and the bolt hole area are not repairable. **DAMAGE 0.010 INCH DEEP OR LESS SHALL BE REPAIRED.**

e. (See fig. 5-11.) Inspect the four lead-lag link attachment bushings for security, and evidence of cracks. No bushing looseness or cracks allowed.

f. Inspect damper arm bearing for binding, evidence of galling or scoring in bore and wear. No radial play is permissible. **MAXIMUM AXIAL PLAY IS 0.015 INCH.**

g. Inspect all bonded areas for evidence of separation.

(1) If evidence of bonding separation is noted around the root fittings or root doublers the blade must be replaced. If there is doubt as to whether a visible crack is in the paint or bond, leave the blade on and check after each flight for growth (an indication of bonding failure).

(2) If there appears to be separation of the trailing edge bond, lightly probe the joint with a 0.001- to 0.002-inch feeler gauge. **IF THE FEELER CAN BE INSERTED TO A DEPTH OF 0.5 INCH OR MORE, EITHER ABOVE OR BELOW THE VEE INSERT, EXCESSIVE SEPARATION IS EVIDENT AND THE BLADE MUST BE REPLACED.**

NOTE

The trailing edge structural bond line to the vee strip insert starts 0.25 inch in (chordwise) from the trailing edge joint; therefore the above tolerance allows 0.25 inch separation. Be sure that measurement is taken from the trailing edge joint, not the tab trailing edge.

h. Ensure that vent holes are open.

NOTE

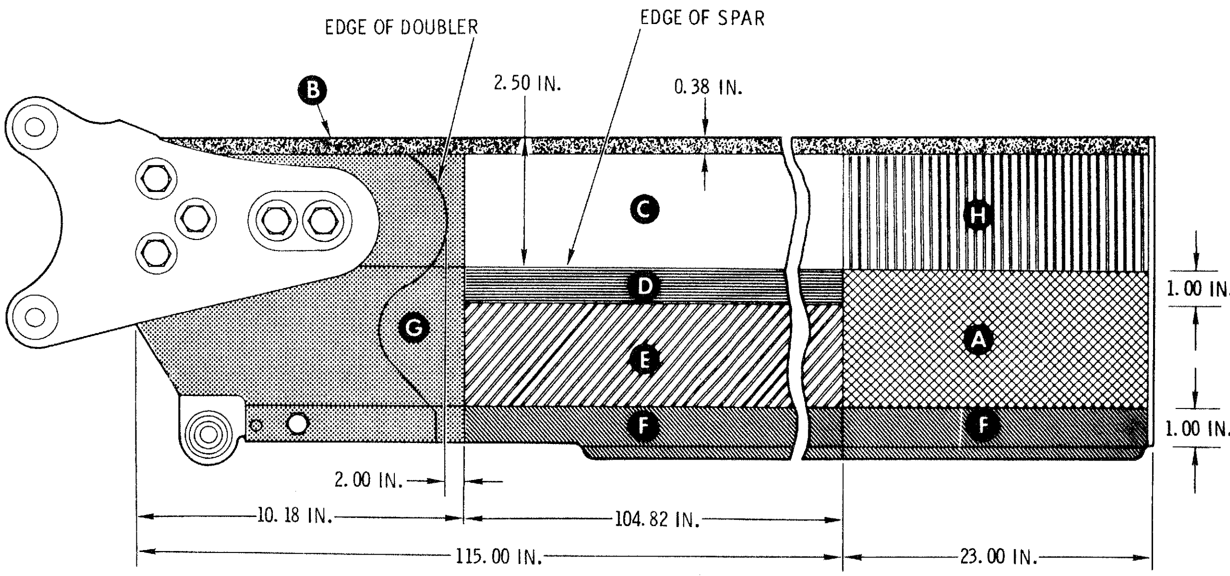
Two or three vent holes are located in the lower blade skin 5.50 inches aft of the leading edge with the first hole 7.50 inches inboard from the tip.

i. Inspect bolts that secure upper and lower root fittings for looseness by attempting to turn bolts with fingers. **IF A BOLT IS FOUND LOOSE, RETORQUE TO 50-60 INCH-POUNDS.** Inspect the six rivets that secure the blade skin to the spar near the tip. If any evidence of insecure attachment or looseness of the rivets is found, the blade must be replaced.

j. Inspect vibration absorber pad on underside of blade for secure bond. Refer to paragraph 5-69 for repair or installation of absorber pad.

k. Inspect forward tip cap screw insert for security and thread damage.

l. When continuous one-per-rev lateral vibration occurs, inspect forward and aft balance weights for security.



DENTS AND DEPRESSIONS (NOTE 1)					
AREA	MAXIMUM DEPTH WITHOUT REPAIR	MAXIMUM DEPTH WITH REPAIR (NOTE 2)	MAXIMUM ALLOWED AREA	MAXIMUM NUMBER OF DEFECTS	MINIMUM DISTANCE BETWEEN DEFECT CENTERS
A	0.015	0.062	1.5 x 1.5	1	NONE
B	0.010	(NOTE 2)	0.25 x 0.25	2	18 INCHES
C	NONE	NONE	NONE	NONE	NONE
D	0.005	0.030	1.5 x 1.5	2	18 INCHES
E	0.010	0.040	1.5 x 1.5	2	18 INCHES
F	0.010	0.040	1.5 x 1.5	3	18 INCHES
G	NONE	NONE	NONE	NONE	NONE
H	0.010	0.040	1.5 x 1.5	1	NONE

NOTES:

1. DIMENSIONS ARE IN INCHES.
2. REFER TO INSPECTION CRITERIA FOR RESTRICTED SERVICE LIFE BLADE REPAIR LIMITS.

11-173 A

Figure 5-16. Main Rotor Blade Normal Damage and Repair Limits for Dents and Depressions.

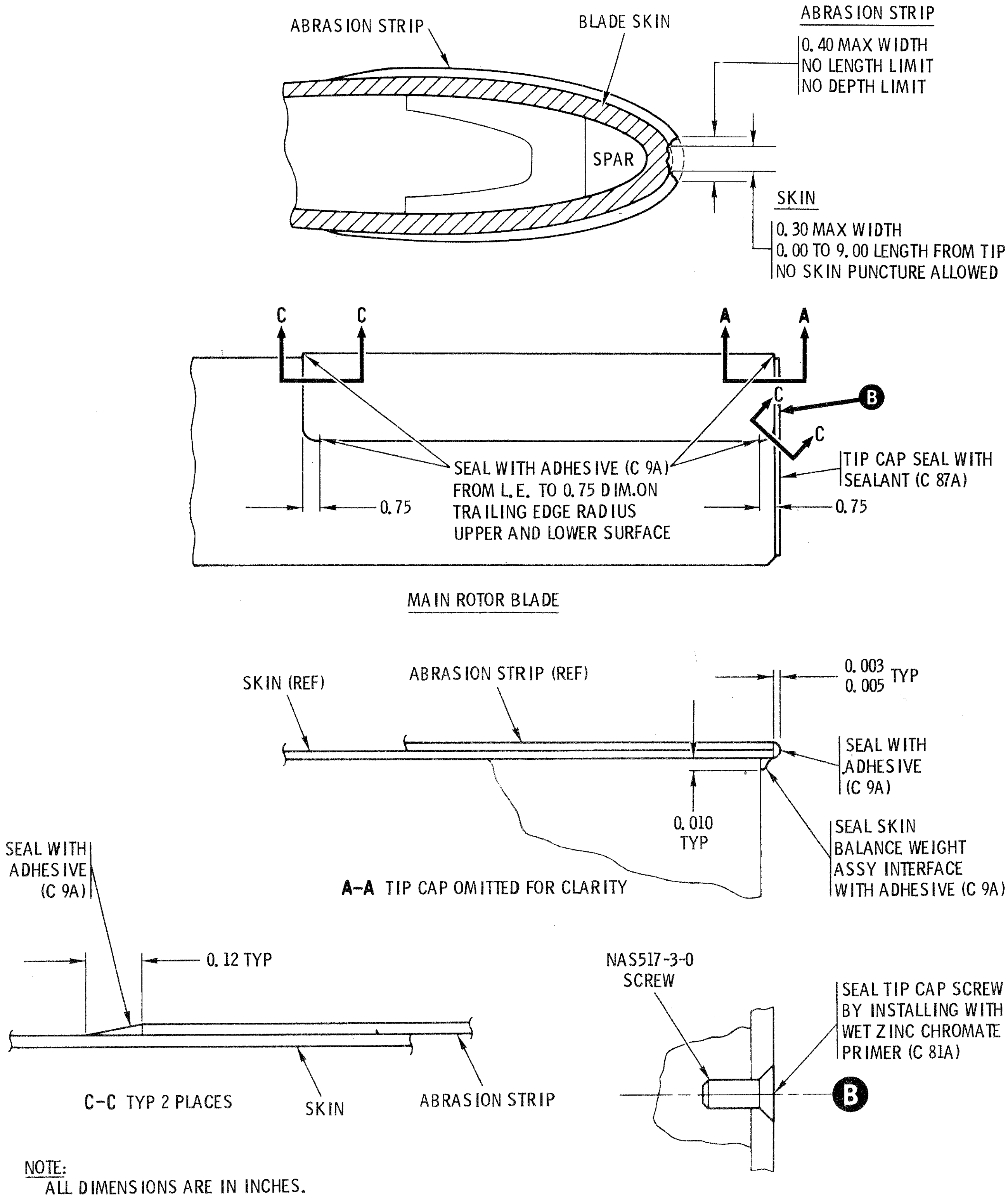


Figure 5-16A Inspection and Repair Main Rotor Blade Metal Abrasion Strip

NOTE

Weights are normally recessed 0.050 inch into threaded weight assembly tip. Security of the weights may be checked by using a torque wrench with a screwdriver socket in the slot of the weight to detect if the weight can be rotated by less than 20 inch-pounds. If a torque wrench is not available, check weight security by fitting a small coin (dime) in the weight slots, and applying the maximum force that can be exerted by only the index finger and thumb.

5-57. Inspection — Main Rotor Blade (Restricted Service Life Criteria). The following paragraphs define limits for restricted service life criteria. **BLADES DAMAGED BEYOND THE LIMITS DESCRIBED IN PARAGRAPHS 5-58 THROUGH 5-61 SHALL BE REPLACED.**

CAUTION

Restricted service life inspection criteria does not extend the normal retirement schedule for main rotor blades (table 1-10). Blades damaged beyond the restricted service criteria defined below will be disposed of locally. Daily inspection is required for all repaired or affected areas that fall within these criteria. (See fig. 5-17 for area locations.) Evidence of cracks in skin areas already repaired will be cause for replacement of affected blade.

5-58. Blade Inspection Criteria — 50-Hour Restricted Service Life.

NOTE

Refer to paragraph 5-61 for extension of restricted service life.

a. Penetrations (holes) — blade area B. Four penetrations through one or both sides (eight holes, four per side) with a minimum of 20 inches between hole centers. Within the above limits one hole may remove a maximum of 0.20 inch from the trailing edge of the spar (fig. 5-17). Maximum hole dimensions after cleanup (removing torn metal, deburring, and rounding of corners) are limited to 3.00 inches spanwise by 1.25 inches chordwise.

b. Scratches, nicks, gouges and dents — blade area C. A dent not exceeding 0.100 inch in depth and 3 inches spanwise length by 1.25 inches chordwise length after cleanup. Scratches, nicks or gouges, oriented 0° to 30° spanwise not exceeding 1.50 inches in length and 0.010 inch in depth, or oriented 30° to 90° chordwise not exceeding 0.75 inch in length and 0.006 inch in depth.

5-59. Blade Inspection Criteria — 100-Hour Restricted Service Life.

NOTE

Refer to paragraph 5-61 for extension of restricted service life.

a. Penetrations (holes) — blade area A. Limit of one hole removing a maximum of 0.20 inch from the trailing edge of the spar (fig. 5-17). Two penetrations through one or both sides (total of four holes) with a minimum of 10 inches between centers. Maximum hole dimensions after cleanup are limited to 3.00 inches spanwise by 1.25 inches chordwise.

b. Penetrations (holes) — blade area B. Four penetrations through one or both sides of the blade (total of eight holes) with a minimum of 20 inches between the centers. Maximum hole dimensions after cleanup are limited to 3.00 inches spanwise and 1.25 inches chordwise. Hole cleanup may include removal of a maximum of one-half the chordwise width of the center channel and/or trailing edge angle (fig. 5-17).

5-60. Blade Inspection Criteria — 300-Hour Restricted Service Life.

NOTE

Refer to paragraph 5-61 for extension of restricted service life.

a. Penetrations (holes) — blade area A. Two penetrations through one or both sides (four holes, two per side) with a minimum of 10 inches between hole centers. Hole cleanup may include removal of a maximum of one-half of the chordwise width of the center channel and/or the trailing edge angle (fig. 5-17).

b. Scratches, nicks, gouges, and dents — blade area A. Dents not exceeding 0.250 inch in depth and 3 inches spanwise length by 1.25 inches chordwise length after cleanup. Scratches, nicks or gouges, oriented 0° to 30° spanwise and not exceeding 1.50 inches in length and 0.025 inch in depth, or oriented 30° to 90° chordwise not exceeding 0.75 inch in length and 0.025 inch in depth.

c. Scratches, nicks, gouges and dents — blade area B. Dents not exceeding 0.150 inch in depth and 3 inches spanwise length by 1.25 inches chordwise

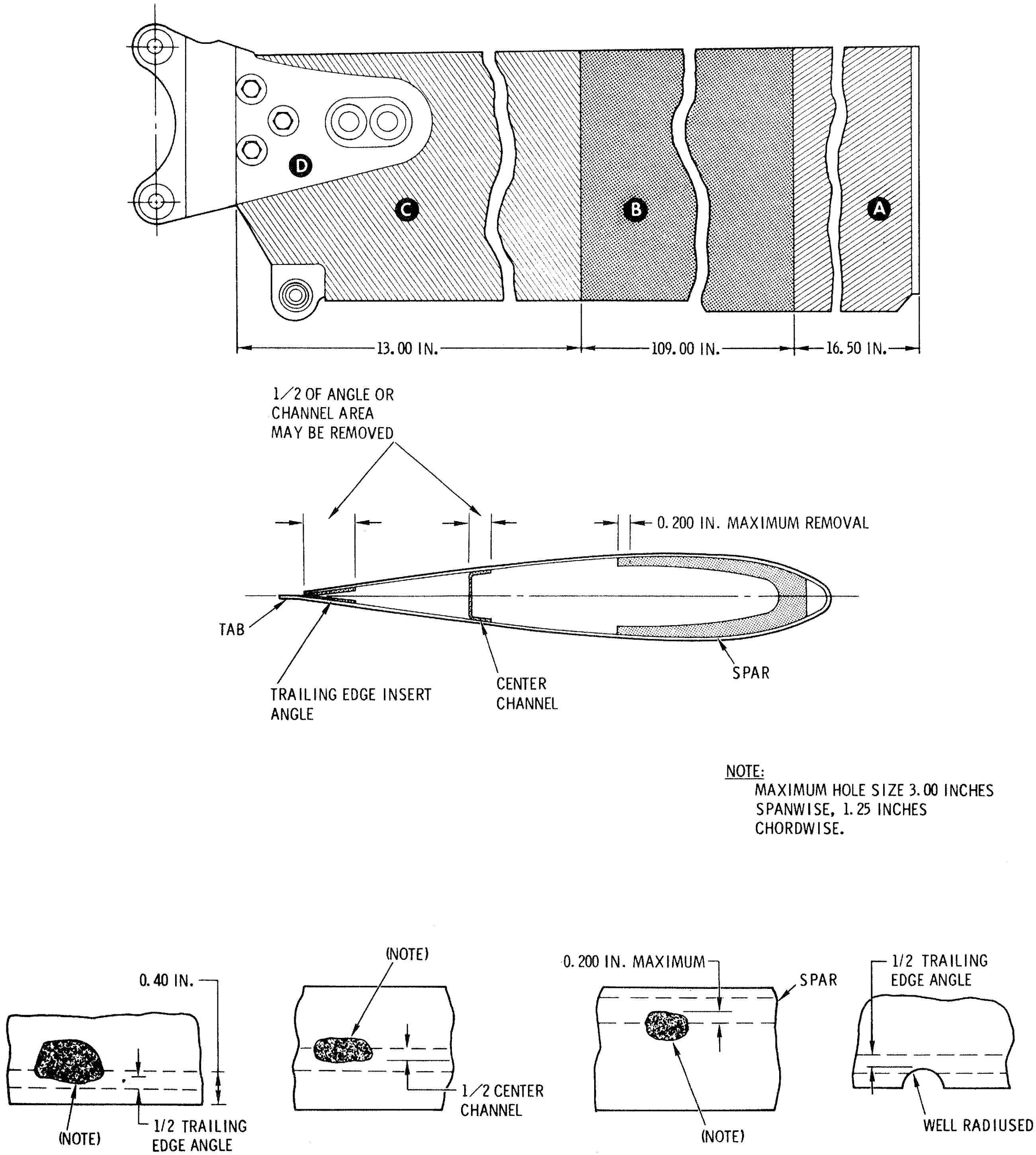


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Figure 5-17. Main Rotor Blade Restricted Service Damage and Repair Limits. (sheet 1 of 2)

PENETRATIONS (HOLES) (NOTE 1)					
BLADE AREA	NUMBER OF PENETRATIONS PER SIDE	MAXIMUM AREA OF DAMAGE (NOTE 2)	MINIMUM DISTANCE BETWEEN CENTERS	ALLOWABLE STRUCTURAL DAMAGE (NOTE 2)	INITIAL RESTRICTED SERVICE LIFE LIMIT (HRS)
A	2	3.00 X 1.25	10	SKIN DAMAGE AND LIMIT OF ONE HOLE REMOVING A MAX OF 0.200 IN. OF SPAR TRAILING EDGE (NOTE 4)	100
A	2	3.00 X 1.25	10	SKIN DAMAGE AND MAX OF ONE-HALF THE CHORDWISE WIDTH OF THE CHANNEL OR TRAILING EDGE ANGLE	300
B	4	3.00 X 1.25	20	SKIN DAMAGE AND LIMIT OF ONE HOLE REMOVING A MAX OF 0.200 IN. OF SPAR TRAILING EDGE	50
B	4	3.00 X 1.25	20	SKIN DAMAGE AND MAX OF ONE-HALF THE CHORDWISE WIDTH OF THE CHANNEL OR TRAILING EDGE ANGLE	100
C	NONE	NONE	NONE	NONE	NONE
D	NONE	NONE	NONE	NONE	NONE

SCRATCHES, NICKS AND GOUGES (NOTE 1)				
BLADE AREA	ORIENTATION FROM SPANLINE	MAXIMUM LENGTH (NOTES 2, 3)	MAXIMUM DEPTH (NOTES 2, 3)	INITIAL RESTRICTED SERVICE LIFE LIMIT (HRS)
A	0° - 30° 30° - 90°	1.50 0.75	0.025 0.025	300 300
B	0° - 30° 30° - 90°	1.50 0.75	0.015 0.015	300 300
C	0° - 30° 30° - 90°	1.50 0.75	0.010 0.006	50 50
D	NONE	NONE	NONE	NONE

DENTS (NOTE 1)			
BLADE AREA	MAXIMUM DEPTH (NOTES 2, 3)	MAXIMUM AREA OF DEPRESSION (NOTE 2)	INITIAL RESTRICTED SERVICE LIFE LIMIT (HRS)
A	0.250	3.00 X 1.25	300
B	0.150	3.00 X 1.25	300
C	0.100	3.00 X 1.25	50
D	NONE	NONE	NONE

NOTES:

1. ALL DIMENSIONS SHOWN ARE IN INCHES.
2. LIMIT IS AFTER CLEANUP.
3. DAMAGE THAT EXCEEDS THESE LIMITS BUT NOT THE LIMITS FOR HOLES MAY BE REPAIRED AS HOLES, WITH THE SAME SERVICE LIFE LIMITS AS HOLES.
4. DAMAGE DOES NOT INCLUDE BLADE TIP WEIGHT.

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Figure 5-17. Main Rotor Blade Restricted Service Damage and Repair Limits. (sheet 2 of 2) length after cleanup. Scratches, nicks or gouges, oriented 0° to 30° spanwise and not exceeding 1.50 inches in length and 0.015 inch in depth, or oriented 30° to 90° chordwise not exceeding 0.75 inch in length and 0.015 inch in depth.

5-61. Blade Inspection Criteria Extension — Restricted Service Life. Main rotor blades within the restricted service life criteria may have that life extended if the following inspection procedures can be successfully accomplished.

NOTE

The blade life changes specified in the following criteria do not extend the normal retirement schedule for main rotor blades (table 1-10).

a. When the restricted service life limit of a repaired blade has been reached the area around the repair is to be thoroughly fluorescent-penetrant inspected for cracks. If no crack has developed, the restricted blade life may be extended, with daily inspections, for 50 hours. At the end of this 50-hour extension another fluorescent-penetrant inspection must be performed. If a crack develops the blade must be replaced.

b. With daily inspections, 50-hour fluorescent-penetrant inspections and no cracks, the restricted service life of a repaired blade may be extended to the following absolute limits.

(1) 50-hour initial limit to 100 hour absolute limit.

(2) 100-hour initial limit to 200 hour absolute limit.

(3) 300-hour initial limit to 600 hour absolute limit.

Table 5-11. Premaintenance Requirements for Main Rotor Blade Repair.

Conditions	Requirements
Minimum Personnel Required	One
Consumable Materials	(C1) (C3) (C4) (C7) (C9A) (C20) (C24) (C25) (C40) (C41) (C69) (C70) (C72) (C79) (C83A) (C87A) (C90) (C91) (C94) (C96) (C98) (C106)

5-62. Repair — Main Rotor Blade.

5-63. Repair — Main Rotor Blade Nicks, Scratches, and Wear Spots.

CAUTION

Remove only those nicks, scratches or wear spots that are within the reparable limits in paragraph 5-56*b*.

a. Using grade 400 and 600 abrasive paper (C3) and (C4), finish not coarser than grade 400 abrasive cloth (C24), remove nicks, scratches, and wear spots from upper and lower root fittings, and from the blade skin.

b. Use finer grades, as necessary, to restore surface roughness to the original finish. Remove material in such a manner that no abrupt changes occur in surface contours.

c. Apply chemical film treatment (C 20) to the repaired surface.

5-64. Repair — Main Rotor Blade Dents and Depressions.

CAUTION

Repair only those dents and depressions that are within the reparable limits of step *c* of paragraph 5-56.

a. Use paint remover (C72) to remove paint from surface area to be repaired.

b. Wipe away all residue with a clean cloth dampened by solvent (C96 or C70). Allow to air-dry for a minimum of 15 minutes.

c. Mask edges of repair area with one layer of tape (C98).

CAUTION

Do not cut tape after it has been applied to the blade.

d. Mix filler (C40), three parts "A" and two parts "B" by weight. Mix thoroughly until the mixture is dark red in color. An alternate filler (C41) may be used if equal parts "A" and "B" by weight are mixed.

e. Allow filler to cure for a minimum of 24 hours at room temperature.

f. Smooth the filled area with grade 400 abrasive cloth (C24). Limit smoothing to the masked area.

g. Remove the tape and inspect the alclad coating of the area around the repair. Penetration of the coating is cause for repair under the restricted service life criteria (para 5-57).

h. Clean repaired area with a cloth dampened by solvent (C96 or C70).

i. Touch-up edge of repaired area with chemical film treatment (C20).

5-65. Repair — Loose Forward and Aft Main Rotor Blade Balance Weights. Reinstall loose forward and aft balance weights as follows.

a. Unscrew loose weight until it projects approximately one-half inch and remove old accumulations of powdered compound.

b. Apply primer (C91) and allow to dry for 5 minutes; then apply compound (C90) and thread the weight back into the blade until the slotted end of the weight is recessed 0.050 inch into the threaded section. Wipe off excess compound.

CAUTION

Allow the sealant to cure for a minimum of 12 hours. If a faster cure is required, a complete cure can be obtained by allowing the parts to set for 30 minutes at room temperature and then heating for 30 minutes at approximately 212°F.

c. If locking compound is not available, thread the weight into normal position; then center-punch the end of the weight into the mating threads at three evenly spaced points to prevent rotation.

5-66. Repair or Replacement — Loose or Missing Main Rotor Blade Aft Tip Cap.

NOTE

Replacement tip caps are supplied with bonding surface pretreated (coated) with nylon primer.

a. Lightly abrade the blade and tip cap mating surfaces with grade 180 abrasive paper (C1). Wipe away residue with cloth dampened in solvent (C94) and allow tip cap to air-dry at room temperature for a period of 30 minutes.

b. Mix two-part bonding adhesive (C7) in equal proportions by weight.

c. Apply bonding adhesive to previously prepared tip cap and mating surfaces of rotor blade tip.

CAUTION

When clamping pressure is applied, be careful not to deform the airfoil or squeeze out all the adhesive.

NOTE

A clamping device may be fabricated by bonding 0.5-inch foam rubber to suitably shaped blocks of wood. Use clamps to provide pressure.

d. Install tip cap in blade tip, apply uniform clamping pressure to the joint, and wipe away excess adhesive. Allow adhesive to cure for 8 hours at room temperature or 2 hours at 150°F.

e. Drill two No. 40 holes, one through each side of blade tip and tip cap, as shown in figure 5-11.

f. Install mechanically expanded rivets (MS20604AD3W2).

NOTE

Tip cap replacement or repair will not affect balance.

5-67. Repair — Main Rotor Blade Forward Tip Cap Insert. Replace a loose or stripped forward cap insert. Use a self-locking insert NAS1394 of correct size for replacement, and install with wet primer (C79).

5-67A. Repair — Main Rotor Blade Abrasion Strip.

CAUTION

Repair only that erosion within reparable limits of step b of paragraph 5-56.

a. Remove screw and tip cap (fig. 5-11) from main rotor blade.

b. Remove any loose or cracked sealant or adhesive from areas of main rotor blade (fig. 5-16A).

c. Clean areas to be sealed with clean cloth and MEK (C69). Wipe surface dry before solvent evaporates.

d. Prepare adhesive (C9A) according to container instructions.

e. Apply bead of adhesive to fill area between edges of abrasion strip and blade skin (fig. 5-16A). Ensure no gaps or ridges are visible in adhesive.

f. Allow adhesive to cure according to container instructions.

g. Apply release agent (C83A) to inside of tip cap according to container instructions.

h. Prepare sealant (C87A) according to container instructions. Apply a thin layer of sealant to mating surfaces.

i. Apply zinc chromate primer (C81A) to tip cap screw. Install tip cap.

j. Remove excess sealant with clean cloth.

k. Should sealant become cracked or erroded, reapply as necessary.

5-68. Erosion Protection — Main Rotor Blade Leading Edge Not Equipped With Metal Abrasion Strip. Blades subjected to operation in a highly abrasive environment and not equipped with an abrasion strip, should have the outer three feet of each blade leading edge covered by protective tape so that blade life will not be reduced by excessive erosion. Replace the tape when it becomes abraded.

WARNING

Do not cut tape after it has been applied on the blade.

a. Remove existing tape. Wipe or wash off any dust, foreign matter or adhesive residue. Smooth the bare metal surfaces and feather the edges of painted surfaces with crocus cloth (C25).

b. Clean leading edge surface with a cloth and naptha (C70), to remove any wax or oil. Wipe dry with clean cheesecloth.

c. Cut tape (C106) to 3-foot lengths.

d. Remove backing from tape.

e. Starting at the tip end, apply tape along the leading edge of the blade so that 0.5 inch of tape is on top of the blade and 1.5 inches are on the bottom.

f. Work the tape with roller, spatula or similar tool from the leading edge toward the trailing edge until positive contact is made over the whole tape area. If air bubbles are trapped, lift tape and release air; then reaffix tape. Allow 2 hours for tape to set.

NOTE

Properly applied tape should show no evidence of air bubbles.

5-69. Replacement — Main Rotor Blade Vibration Absorber Pad.

NOTE

The vibration absorber pad bonded to the blade skin is identical to rubber pads bonded to the vibration absorber assembly.

a. Raise the 3-per-rev pendulum to locate the contact point for pad attachment.

b. Clean the blade contact area with naptha (C70) and allow to air-dry for 30 minutes.

c. Remove backing from the rubber pad.

d. Position the pad so that the narrow edge is parallel to the blade leading edge; using hand pressure, press pad firmly in place.

5-70. Repair — Main Rotor Blade Vibration Absorber. Refer to paragraph 5-83.

5-71. Repair — Main Rotor Blade with Restricted Service Life. (See fig. 5-18.)

CAUTION

Repair only those holes that are within the reparable limits of paragraph 5-57.

a. Radius all sharp notches, tears, and cracks. Deburr all damaged area edges.

b. Repair holes shall either be circular or elongated in a spanwise direction. Maximum hole size is 3.00 inches spanwise by 1.25 inches chordwise.

c. Cut a doubler for each hole, using 0.020- to 0.040-inch 2024-T32 aluminum alclad sheet. Doubler should extend beyond the hole edges by a minimum of 0.375 to a maximum of 0.500 inch.

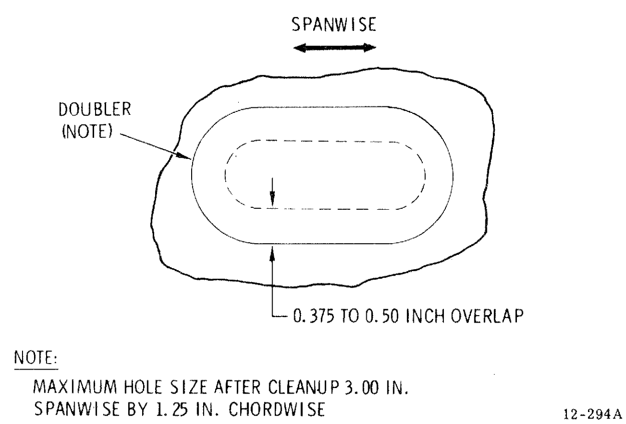


Figure 5-18. Restricted Service Life Blade Repair.

d. Lightly abrade the mating surfaces of the blade and doubler with grade 180 abrasive paper (C1). Wipe away residue with cloth dampened by solvent (C94).

e. Mix two-part bonding adhesive (C7) in equal proportions by weight.

f. Apply bonding adhesive to prepared surface of blade and doubler.

CAUTION

When clamping pressure is applied, be careful not to deform the airfoil or squeeze out all the adhesive.

NOTE

A clamping device may be fabricated by bonding 0.5-inch foam rubber to suitably shaped blocks of wood. Use clamps to provide pressure.

g. Position doubler on blade and wipe away excess adhesive. Apply uniform clamping pressure to joint. Allow adhesive to cure for 8 hours at room temperature.

5-72. Installation — Main Rotor Blade. *a.* With an assistant at the blade tip, position main rotor blade in its (marked) pitch housing linkage. (See fig. 5-11.) Install attaching pins.

NOTE

Blade types may be intermixed without regard to part number.

CAUTION

If a new blade is to be installed, or if blades are not reinstalled in same positions from which removed or are installed on a new rotor hub, the blade and damper attaching pins must be adjusted (para 5-76). It will also be necessary to track the blades.

b. Align damper arm with main rotor blade; install damper attaching pin.

c. Lock all attaching pins (para 5-76).

d. Remove blade-to-pitch housing link markings.

5-73. MAIN ROTOR BLADE AND DAMPER ATTACH PINS.

5-74. Inspection — Main Rotor Blade and Damper Attach Pins. *a.* Check the attach pin lever cam for excessive wear. Cam lobe wear is limited to the hard anodized surface.

b. Inspect for evidence of corrosion between the pivot (outside edge of barrel nut) and the bore in the cam handle. Inspect attach pin rivets for corrosion and security. Any evidence of corrosion requires replacement of attach pin.

c. Visually inspect rotating areas and cam contact surface for cracks. Using a 5x magnifying glass, inspect area of cam locking lever at top attaching point for cracks. Any evidence of cracking requires replacement of the attach pin.

5-75. Corrosion Control — Main Rotor Blade and Damper Attach Pins. *a.* Relieve load on blade attachment points by supporting blade tip from below. Unlock and remove the attach pin.

b. Lubricate pivoting surfaces with clean oil (C67).

c. Rotate pin cam handle back and forth on barrel nut several times until oil gets into all sections of the pin. Remove excess oil with a clean cloth.

d. Reinstall the attach pin.

CAUTION

The following must be accomplished after changing an attaching pin, a main rotor hub or a blade.

5-76. Adjustment — Main Rotor Blade and Damper Attach Pins. *a.* Have an assistant support the blade tip to establish alignment of holes in the blade root fittings and lead-lag links. Install the attach pin.

b. Check locking force required. **FORCE REQUIRED TO CLOSE HANDLE IS 25-35 POUNDS** (maximum hand force).

c. Adjust pin having incorrect locking force. Remove pin and adjust by turning small hex nut at end of pin. Do not adjust nut with pin installed. Reinstall and check locking action.

NOTE

After adjustment, the installed length of the pins, measured from the face of the thrust washer to the outer edge of the last bushing, should be 0.99 to 1.04 inches for the damper pins and 2.84 to 2.89 inches for the main rotor blade attachment pins.

5-77. VIBRATION ABSORBER.

5-78. Description — Vibration Absorber. One vibration absorber is installed on the lower inboard end of each main rotor blade (fig. 5-11). Each vibration absorber consists of two pendulums that pivot about a common axis. These pendulums are tuned to cancel out the first and second harmonic beats of the natural vibration frequency of each individual blade. The largest pendulum counteracts any 3-per-revolution vibrations; the smallest pendulum counteracts any 5-per-revolution vibrations.

5-79. Removal — Vibration Absorber (AVIM). *a.* Remove bolt and washer at trailing edge of blade; remove shim washer(s) if installed (fig 5-11).

NOTE

Shim washers are installed between vibration absorber and main rotor blade as required. The washers compensate for minor variations in blade contour.

b. Remove two nuts, radius blocks, washers, and bolts that attach absorber to blade.

c. Remove vibration absorber.

5-80. Disassembly — Vibration Absorber (AVIM). *a.* Remove rivet that secures pivot shaft to bracket by drilling off head and pressing rivet out (fig. 5-19).

b. Remove pivot shaft by inserting a pin through the hole in the forward end of the bracket and pushing shaft out of bracket.

c. Remove 5-per-revolution pendulum, and 3-per-revolution pendulum.

d. Do not remove shaft bearings or thrust washers unless replacement is necessary.

5-81. Cleaning — Vibration Absorber (AVIM). Clean all parts of the vibration absorber using a clean cloth

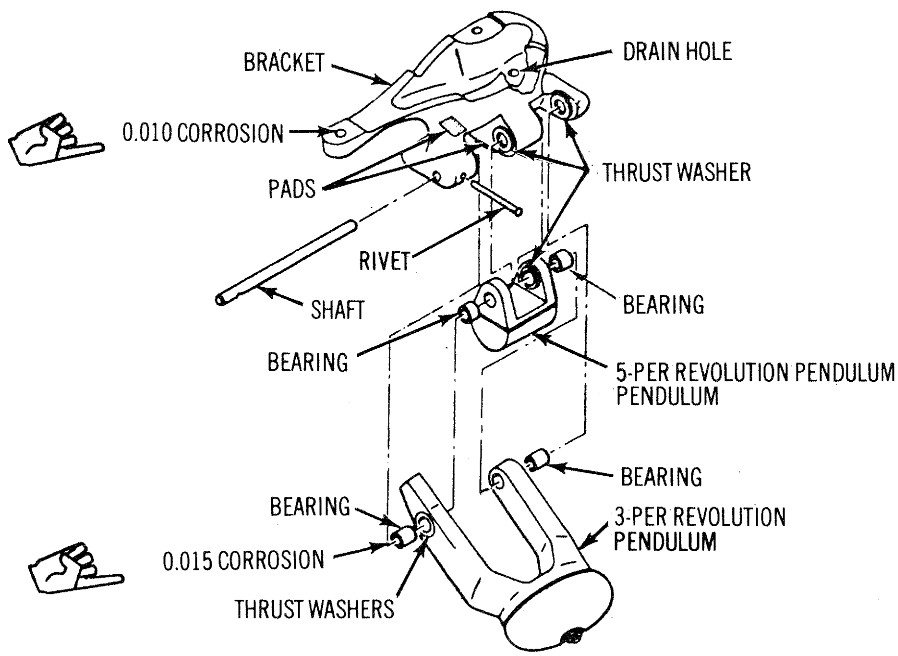


Figure 5-19. Vibration Absorber.

saturated with solvent (C1). Use dry, filtered, low-pressure compressed air to blow out bearing bores and drain hole in bracket (fig. 5-19).

5-82. Inspection — Vibration Absorber (AVIM). *a.* Inspect thrust washers bonded to bracket and pendulums for evidence of damaged teflon surfaces, wear and failure of bond.

b. Inspect bores of bearings for evidence of galling, scoring, and wear. **MAXIMUM ALLOWABLE BORE DIMENSIONS IS 0.268 INCH.**

c. Inspect pendulums for evidence of cracks and breaks.

d. Inspect bracket for cracks and breaks.

e. Inspect shaft for cracks, distortion and wear.

f. Inspect bracket for presence of ballast weights. Evidence of lost ballast will require weighing of complete vibration absorber.

5-83. Repair — Vibration Absorber (AVIM). Repair of the vibration absorber is limited to disassembly and replacement of pivot shaft, bearings, thrust washers, and rubber pads on bracket. The pendulum weights are not directly interchangeable between absorbers and the weight shims must not be altered because the weights and the assembly have been tuned.

a. Use press to remove and install new pivot bearings into pendulums.

b. Using adhesive (C7), install new, or rebond loosened thrust washers to bracket or pendulums. Follow container instructions for adhesive application.

c. Using silicone rubber adhesive (C13), install new, or rebond loosened 3-per-rev and 5-per-rev pads to bracket. Follow container instructions for adhesive application. After bonding, seal the edges of the pad with rubber cement (C19). Ensure that center hole in pad at 5-per-rev location does not block the drain hole through the bracket.

5-84. Reassembly — Vibration Absorber (AVIM). (See fig. 5-19.)

CAUTION

During reassembly, do not allow zinc chromate primer to get on the fabroid bearings.

a. Using zinc chromate primer (C79), coat cavity in bracket where undrilled end of shaft will seat.

b. Align bores of pendulums with bracket.

c. Install shaft so that rivet hole in shaft aligns with matching hole in bracket while primer is still wet.

d. Install new rivet and flush the shop-formed end.

e. Coat open hole in bracket with primer. Check that the pendulums rotate freely on the shaft without binding and that the shaft is free of primer.

5-85. Weighing — Vibration Absorber (AVIM). *a.* Add lead weights (C114) as required to establish total weight of 1360 ±0.1 grams.

b. Bond lead weight into recess cup area of bracket with adhesive (C77).

c. Assure drain hole is open through the bonding material and pad on the bracket.

5-86. Installation — Vibration Absorber (AVIM). (See fig. 5-11.) *a.* Apply zinc chromate primer (C79) on the two long bolts.

b. Align vibration absorber with holes in main rotor blade. Install two long bolts and washers while primer is still wet; then install radius blocks and nuts. Check that the shorter length bolt is in outboard hole. **TORQUE BOLTS TO 50-60 INCH-POUNDS.**

CAUTION

Improper shimming and torquing can result in damage to the vibration absorber bracket.

c. If a gap exists between the trailing edge of vibration absorber and main rotor blade, shim with washers. If no gap exists, install trailing edge bolts and washer without shim washers. Apply zinc chromate primer to bolt and install bolt and washer while primer is still wet. **TORQUE BOLT TO 15-20 INCH-POUNDS.**

SECTION IV MAIN ROTOR CONTROLS

(Refer to chapter 11)

SECTION V TAIL ROTOR AND CONTROL SYSTEM

5-87. TAIL ROTOR AND CONTROL SYSTEM.

5-88. Description - Tail Rotor and Control System. The tail rotor is mounted on the tail rotor transmission at the end of the tailboom. The tail rotor counteracts main rotor torque and controls the heading of the helicopter. The tail rotor installation (fig. 5-20, sheets 1 and 2) consists of a pitch control assembly, drive fork, two pitch control links, and two blade assemblies. Either two fiberglass blades (sheet 1) or two metal blades (sheet 2) may be installed. The metal blade installation can be equipped with either spherical or elastomeric drive fork-to-hub bearings. Otherwise both tail rotor installations are identical. The blade assemblies telescope over a hub and are bolted to an interconnecting tension-torsion strap assembly within the hub. Blade angle is controlled by the pitch control assembly consisting of links connecting the pitch control arms to a swashplate that slides axially on the tail rotor transmission output shaft. Movement of the swashplate is controlled through a series of bellcranks and rod assemblies connected to the pilot's foot pedals.

5-89. Troubleshooting — Tail Rotor and Control System. Troubleshooting information is divided into: (1) Investigation of operational vibration problems originating with the tail rotor assembly, or symptoms that can be recognized (table 5-12); and (2) isolation of an unusual controls malfunction (chapter 11).

Table 5-13. Premaintenance Requirements for Removal of Tail Rotor Assembly.

Conditions	Requirements
Special Tools	(T28)

5-90. Removal — Tail Rotor Assembly. (See fig. 5-21.)

CAUTION

Do not remove hub-to-drive fork hinge bolt (369A1602) during removal of the tail rotor assembly. It is possible to damage the strap pack during removal and reinstallation of bolt.

a. Disconnect station 282 bellcrank from transmission at pivot point so that bellcrank pin is disengaged from bearing in pitch control housing.

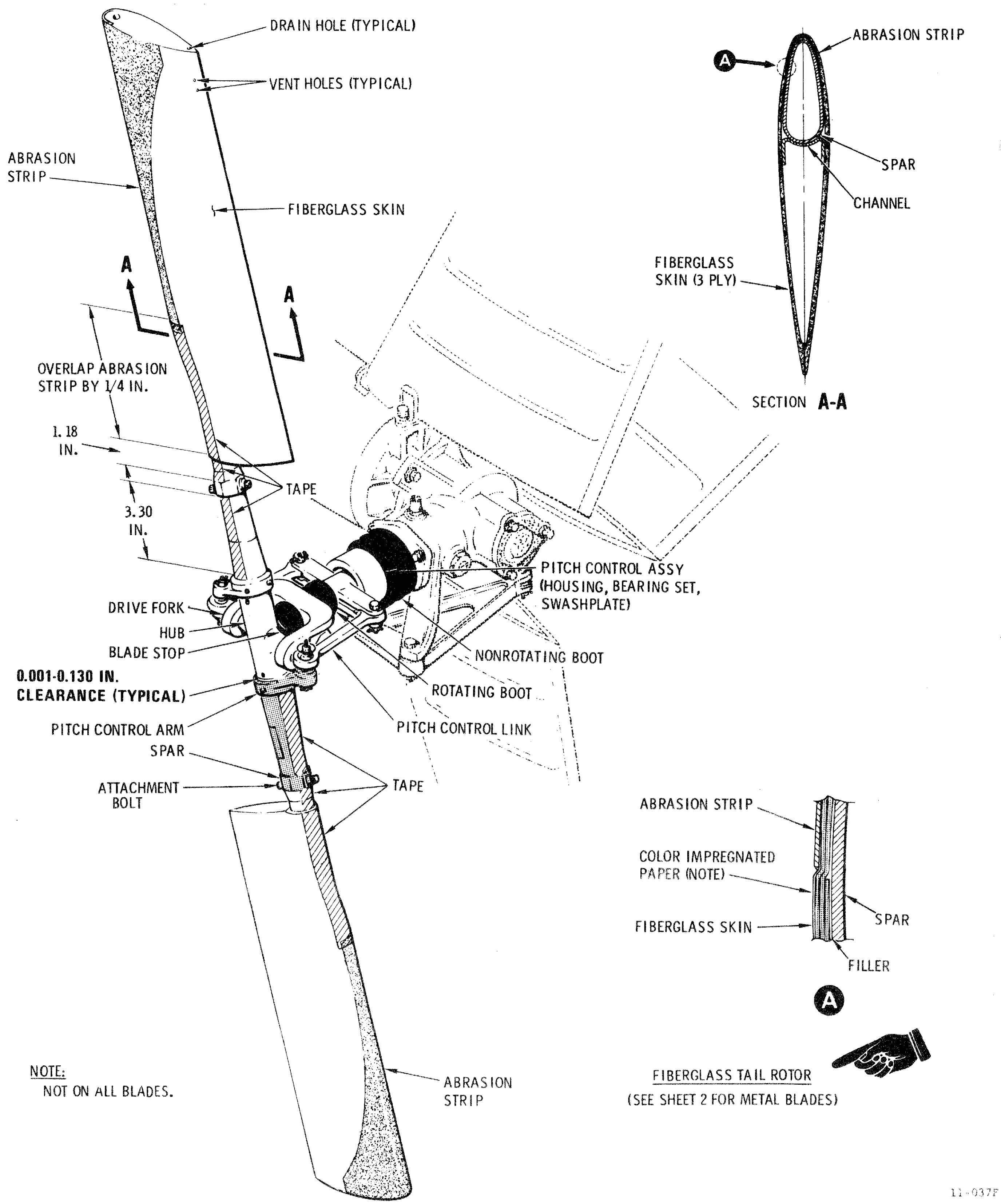


Figure 5-20. Tail Rotor Installation. (Sheet 1 of 2)

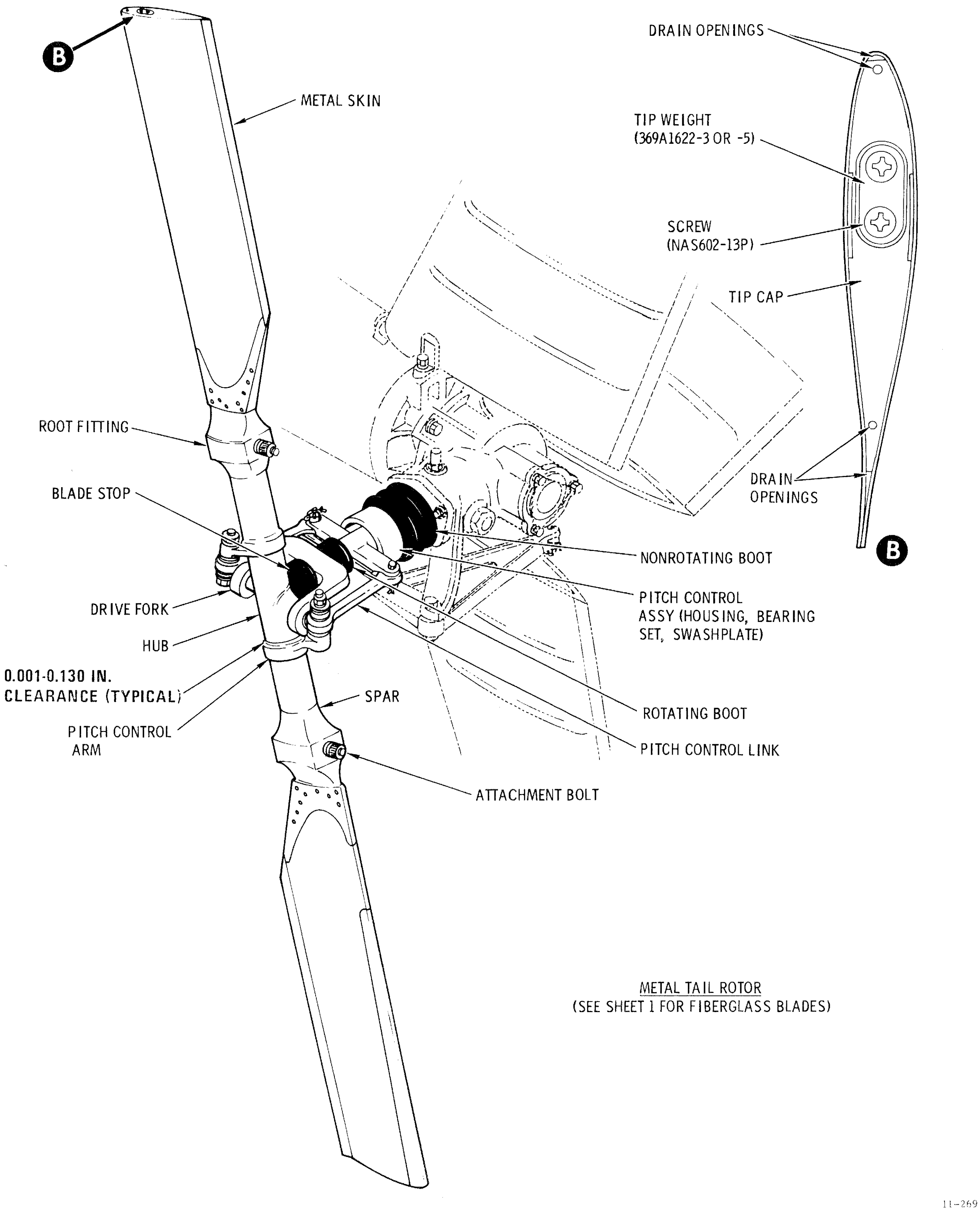
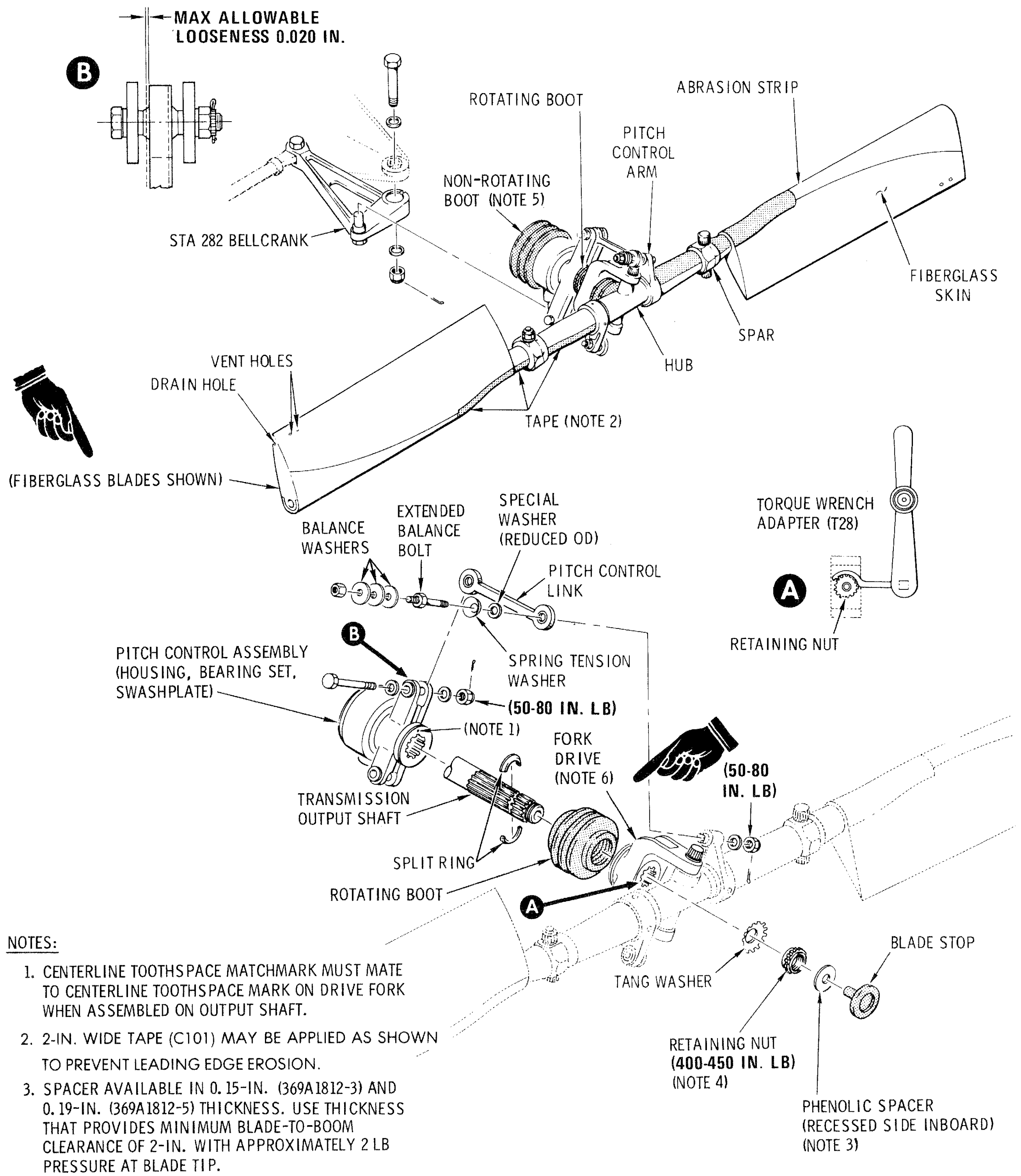


Figure 5-20. Tail Rotor Installation. (Sheet 2 of 2)



NOTES:

1. CENTERLINE TOOTHSPACE MATCHMARK MUST MATE TO CENTERLINE TOOTHSPACE MARK ON DRIVE FORK WHEN ASSEMBLED ON OUTPUT SHAFT.
2. 2-IN. WIDE TAPE (C101) MAY BE APPLIED AS SHOWN TO PREVENT LEADING EDGE EROSION.
3. SPACER AVAILABLE IN 0.15-IN. (369A1812-3) AND 0.19-IN. (369A1812-5) THICKNESS. USE THICKNESS THAT PROVIDES MINIMUM BLADE-TO-BOOM CLEARANCE OF 2-IN. WITH APPROXIMATELY 2 LB PRESSURE AT BLADE TIP.
4. TORQUE NUT TO 550-600 IN. LB; BACK OFF TO 200-250 IN. LB. THEN FINAL TORQUE TO 400-450 IN. LB.
5. TWO DRAIN/VENT HOLES IN FOLDS AT BOTTOM.
6. APPLY LUBRICANT (C63) TO INNER SPLINES OF DRIVE FORK ONLY.

Figure 5-21. Tail Rotor and Pitch Control Assemblies.

Table 5-12. Troubleshooting the Tail Rotor and Control System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **During aircraft operation, a high frequency vibration (buzzing) is felt in the tail rotor pedals.**

STEP 1. Check that fiberglass leading edge tape is not worn or lost from the blade.

If fiberglass blade tape is damaged or lost, replace according to paragraph 5-99.

STEP 2. Check for excessive looseness in tail rotor fork bearing and for play between tail rotor hub and fork.

If looseness or play is evident on the tail rotor hub area, replace the tail rotor assembly (para 5-90).

STEP 3. Check for worn pitch control link bearings.

If bearings are worn beyond 0.020 inch, replace the control links (para 5-102).

STEP 4. Check for loosened tail rotor assembly retaining nut or loose stabilizer strut bolts.

If nut or bolts are loose, retorque tail rotor nut (para 5-100) and stabilizer chapter 2.

STEP 5. Check tail rotor for separation of leading edge abrasion strip.

If the tail rotor leading edge abrasion strip on fiberglass blade has any separation, replace the tail rotor assembly (para 5-90). If on a metal blade, replace the blade (para 5-99D).

STEP 6. Check for excessive wear in pitch control assembly bearing (fig. 5-22).

If wear is evident in pitch control bellcrank bearing, replace the pitch control assembly (para 5-90).

STEP 7. Check for excessive wear (wobble) in swashplate liners as shown in figure 5-22.

If wobble exceeds the 3 pound limit, replace the swashplate or pitch control assembly (para 5-90).

STEP 8. Check that tail rotor balance is within limits (para 5-103 or 5-104).

If tail rotor cannot be balanced to an acceptable level, replace tail rotor assembly (para 5-90).

b. Pull beaded end of stationary boot out of inboard groove of pitch control assembly.

c. Straighten tabs on tang washer. Using torque wrench adapter (T28), loosen retaining nut and slide tail rotor outward on shaft to remove blade stop and phenolic spacer; then remove nut.

CAUTION

Tail rotor tang washers shall not be reused.

d. Remove and discard tang washer.

e. Place a grease pencil mark on the drive fork next to the keyway on the transmission output shaft. The mark aids reinstallation of the tail rotor at the same position.

f. Slide tail rotor assembly (including pitch control assembly) off shaft; catch or remove split rings from shaft.

g. Remove hub and blade assembly from the pitch control assembly, if required. Separate by removing pitch control link bolts at the hub end.

NOTE

The hub and blade assembly is a separately replaceable item.

5-91. Component Inspection — Tail Rotor Installation. *a.* Check that no play exists between tail rotor hub and drive fork.

b. Check hub-to-drive fork bearings for radial looseness 0.008 INCH MAXIMUM.

c. Inspect fork and hub for scratches.

nicks, dents, cracks, corrosion and similar surface defects. No cracks are allowable. Scratches and nicks that do not exceed 0.005 inch are permissible with rework. Polish out and blend to contour with grades 180 through 600 abrasive paper (C1), (C2), (C3), and (C4).

d. Inspect all bolts and nuts for secure attachment.

e. Inspect blade stop and boots for deterioration and secure attachment. With antitorque pedals in neutral position, apply approximately 2 pounds against blade tip and check for a minimum of 2 inches clearance between tip and tailboom.

f. Accomplish fiberglass tail rotor blade inspection (para 5-92) or metal tail rotor blade inspection (para 5-92A).

g. Accomplish pitch control assembly inspection (para 5-93).

5-92. Inspection - Fiberglass Tail Rotor Blade. (See fig. 5-20, sh 1.) The fiberglass skins of tail rotor blades are color-finished in two ways; they are either painted or a limited number have a layer of color-impregnated paperbonded directly to the outer ply of the skin. The bonding process for colored paper can and does produce some creases or wrinkles, as well as small blisters (voids) that are repaired before blades are put in service. When inspecting blade skins during inspection below, be sure that what might appear to be a crack is not just a crease line or the result of a repaired blister. To recognize the difference will help prevent otherwise serviceable blades from being removed from service.

WARNING

Inspection of the tail rotor blade should be extremely thorough. Particular care should be taken when inspecting any blade trailing edge or abrasion strip deficiencies. Do not attempt repairs beyond those permitted herein. Tail rotor failure and possible injury can result.

a. Visually inspect blade airfoil contour for evidence of collapse or major deformation. Minor deformation is allowed. If either or both of the blades show signs of a collapsed airfoil, replace the tail rotor.

b. Inspect leading edge abrasion tape for condition and secure attachment.

c. Inspect stainless steel leading edge abrasion strips for cracks, scratches, nicks, cracks between the abrasion strip and the fiberglass, and for parting of the abrasion strip from the blade. Limits for the stainless abrasion strip are as follows:

(1) No cracks are allowable.

(2) Scratches and nicks requiring no more than 0.010 inch removal of material in cleanup are permissible. Polish out and blend to contour with grades 180 through 600 abrasive paper (C1), (C2), (C3), and (C4).

(3) Splitting at the tip end of the abrasion strip is allowable **IF LENGTH DOES NOT EXCEED 0.5 INCH.**

CAUTION

When making the inspection in (4) below, do not pry under the abrasion strip. To do so might cause additional separation and require replacement of the blade.

(4) Parting of stainless steel abrasion strip bonding from the fiberglass skin is allowable with the following limits: **SPANWISE LENGTH SHALL NOT EXCEED 0.50 INCH; CHORDWISE DEPTH SHALL NOT EXCEED 0.05 INCH; A MINIMUM OF 1.00 INCH OF GOOD BOND HAVING NO SEPARATION MUST EXIST BETWEEN SEPARATED AREAS.**

CAUTION

When making the inspection in *d* below, use a 5X magnifying glass to assure there is a bond separation rather than a separation of the outer paper from the fiberglass. Use extreme care in measuring suspected trailing edge separations to avoid additional damage to tail rotor blade.

d. Inspect the tail rotor blade trailing edges for separation. Use a 0.001- to 0.002-inch feeler gage to measure the separation depth. **TRAILING EDGE SEPARATIONS ARE ALLOWABLE WITH THE FOLLOWING LIMITS:**

(1) Spanwise length shall not exceed 0.50 inch.

(2) Chordwise penetration shall not exceed 0.05 inch.

(3) A minimum of 2.00 inches of good bond having no separation must exist between separated areas.

(4) Bubbles or voids in the trailing edge not exceeding the above criteria are acceptable and considered only appearance items. Paragraph 5-94 details repair procedures for allowable trailing edge separation voids and/or paper separation.

CAUTION

When making the inspection in *e* below, do not allow solvent remover to soak in blade crevices as the bond can be damaged. If more than 2 square inches of paint are removed, tail rotor balance can be affected.

e. Check for paint cracks between the abrasion strip and fiberglass skin. If questionable, strip paint down to primer by using solvent remover (C69 or C72). Inspect stripped area with a 5-power magnifying glass. Spanwise paint cracks between the fiberglass skin and trailing edge of abrasion strip are allowable for the full span of the blade.

f. Inspect blade fiberglass skin for damage. No cracks are permissible. **MINOR SCRATCHES THAT DO NOT EXCEED 0.005 INCH DEPTH (ANY LENGTH) ARE PERMISSIBLE WITHOUT REWORK. SCRATCHES THAT DO NOT EXCEED 0.010 INCH DEPTH AND 1.5 INCH LENGTH ARE ACCEPTABLE IF REWORKED.** (Paragraph 5-97 gives instruction for fiberglass repair.)

g. Inspect for clogged vent holes in trailing edge, and tip cap drain hole; if clogged, gently clear with a blunt instrument. Inspect tip caps for secure attachment. Repair tip cap with loose or missing rivets, or unsealed rivet cores according to paragraph 5-96.

h. Inspect blade spars between attachment bolt and pitch control arm for scratches, nicks, dents, cracks, and similar surface defects. No cracks are allowed. Scratches and nicks that do not exceed 0.005 inch are permissible with rework. Polish out and blend to contour with grades 180 through 600 abrasive paper (C1), (C2), (C3), and (C4).

5-92A. Inspection - Metal Tail Rotor Blade. (See fig. 5-20, sh 2.) Refer to figure 5-21A for damage and repair areas.

WARNING

Inspection of the tail rotor blade should be extremely thorough. Particular care should be taken when inspecting any blade deficiencies. Do not attempt repairs beyond those permitted herein. Tail rotor failure and possible injury can result.

a. Inspect the blade skin for evidence of cracks and holes. Cracks or holes in blade skin, regardless of location, shall be cause for rejection of the blade. SCRATCHES, DENTS, NICKS, GOUGES, AND OTHER SURFACE DEFECTS IN THE BLADE ARE LIMITED ACCORDING TO b and c below.

NOTE

Use a dial indicator to check depth of blade dents and scratches.

b. Limitations for surface scratches, nicks, gouges, and pits in three areas of the blade are described below. See figure 5-21A for area location. SCRATCHES THAT DO NOT PENETRATE THE CLAD SURFACE ARE ACCEPTABLE WITHOUT REWORK IN ALL AREAS.

(1) AREA A: Acceptable with repair. Scratches, nicks, gouges, or pits to 0.003-inch deep if oriented more than 15 degrees from span line; or 0.006-inch deep if less than 15 degrees from spanline.

(2) AREA B: Acceptable with repair. Scratches, nicks, gouges, or pits to 0.006 inch deep in any direction.

c. Limitations for dents in two area of the blade are described below. See figure 5-21A for area location. A dent or depression is defined as a smooth depression or discontinuity with no sharp changes in section.

(1) AREA A: Acceptable without repair. Dents to a maximum depth of 0.020 inch. No dents with sharp contour changes are allowed.

(2) AREA B: Acceptable without repair. Dents to a maximum depth of 0.050 inch. No dents with sharp contour changes are allowed.

d. Inspect all exposed blade edges for possible separation. No separations are allowed.

e. Inspect blade leading edge for erosion in AREA B, figure 5-21A. If erosion has not effected blade performance or caused tail rotor vibration, polish repair the area (para 5-99A).

f. Inspect blade spars in AREA C, figure 5-21A for nicks, cracks, and similar surface defects . No cracks are allowed. Scratches and nicks that do not exceed 0.002-inch after rework are permissible.

g. Check that tip weights and attaching hardware are secure .

h. Inspect for clogged drain openings at blade tip. If clogged, gently clear with a blunt instrument.

i. Inspect the tail rotor blade leading edge abrasion strip/airfoil bond line for cracks/chipping or other evidence of abrasion strip separation. If cracking or chipping is observed, use a 5x magnifying glass and closely inspect a ling bond line and at blade tip for any debonding between epoxy adhesive and abrasion strip. Any separation between bonding adhesive and abrasion strip is cause for removal of blade form service. Small blow holes in the epoxy adhesive are not considered as adhesive separation.

NOTE

Paint chipping or cracking can be the normal result of service time or operating environment, but requires a closer examination using a 5x magnifying glass, to inspect for possible discrepancies.

5-93. Inspection — Tail Rotor Pitch Control Assembly. *a.* Inspect the pitch control assembly for evidence of binding by hand-turning a few turns while listening for unusual sounds.

NOTE

When the tail rotor blades are displaced from maximum to minimum pitch angles, an audible snapping noise may be heard. This is a normal condition caused by the strap pack when it is not under a centrifugal load.

b. Move (teeter) the blades back and forth to check for evidence of binding.

c. If tail rotor control system drag or friction has been reported or is suspected, use a spring scale as shown in figure 5-22 to measure the drag (sliding friction) of the pitch control assembly on the output shaft. Note that the pitch links, station 282 bellcrank and the rubber boots must be detached from the assembly during the drag check. If drag exceeds 3 pounds, remove the pitch control assembly and clean the swashplate and output shaft splines.

d. Inspect for surface defects such as nicks, scratches, dents or corrosion. Defects may be reworked by abrasive polishing and blending to a depth of 0.015 inch on machined surfaces, or a depth of 0.020 inch on cast or forged surfaces. After repair, apply corrosion protective treatment to the magnesium alloy pitch control bearing housing or aluminum alloy swashplate as directed in chapter 1.

e. Inspect for brinelling wear of the inner surfaces of the swashplate clevis ears. **WEAR UP TO 0.003 INCH-DEPTH IS PERMISSIBLE WITHOUT REWORK OR REPAIR. WEAR UP TO 0.015 INCH-DEPTH IS REPARABLE ACCORDING TO *d* ABOVE.**

f. Inspect for slippage between the pitch control bearing inner race and swashplate. Check external slippage mark. If bearing slippage has occurred replace the pitch control assembly and accomplish the following.

(1) Paint a slippage mark across the outboard end of the replacement pitch control bearing inner race and the swashplate, midway between the connecting link ears.

(2) Allow paint to dry.

(3) Perform a ground runup of the aircraft and while running, cycle the tail rotor control through a full pedal travel a minimum of six cycles. Shut down the aircraft.

(4) Inspect the slippage mark. If there is any evidence of bearing slippage, replace the pitch control assembly.

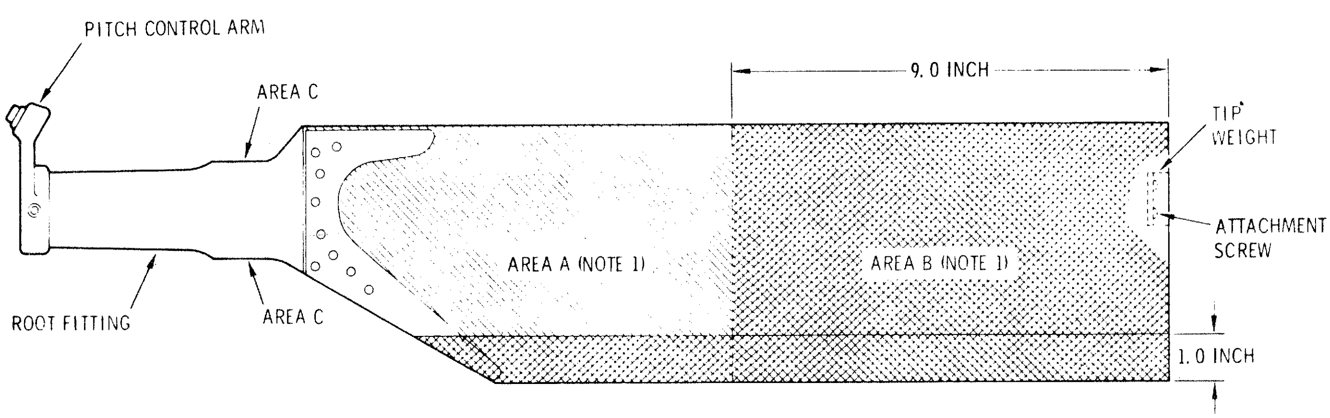
g. Inspect for pitch control wobble on the transmission output shaft. **WOBBLE UP TO 0.020 INCH, AS SHOWN IN FIGURE 5-22, IS PERMISSIBLE.** If a greater amount of wobble is present replace the swashplate or pitch control assembly, as applicable.

NOTE

Wobble may be accurately measured by attaching a dial indicator support to the tail rotor drive fork, with the indicator probe in contact with the swashplate clevis ear location shown in the illustration. Take care not to allow the pitch control assembly to slide on the output shaft while measuring wobble.

h. Inspect pitch control assembly for loose, rough or binding bearings; bolts and bolt holes for wear and condition; swages and/or bearing retainer for looseness and damage; arms for bends and deformation.

i. Inspect swashplate (fig. 5-21) liners (1 splined, 1 smooth) and mating surfaces of tail rotor transmission output shaft for dirt deposits or other contamination. Dirt or dried grease deposits at these locations can cause excessive drag or friction in the tail rotor controls.



NOTES:

1. INSPECTION AREA BOTH SIDES.
2. FOR BLADE INSPECTION CRITERIA, REFER TO TEXT.

11-270

Figure 5-21A. Metal Tail Rotor Blade Damage and Repair Areas.

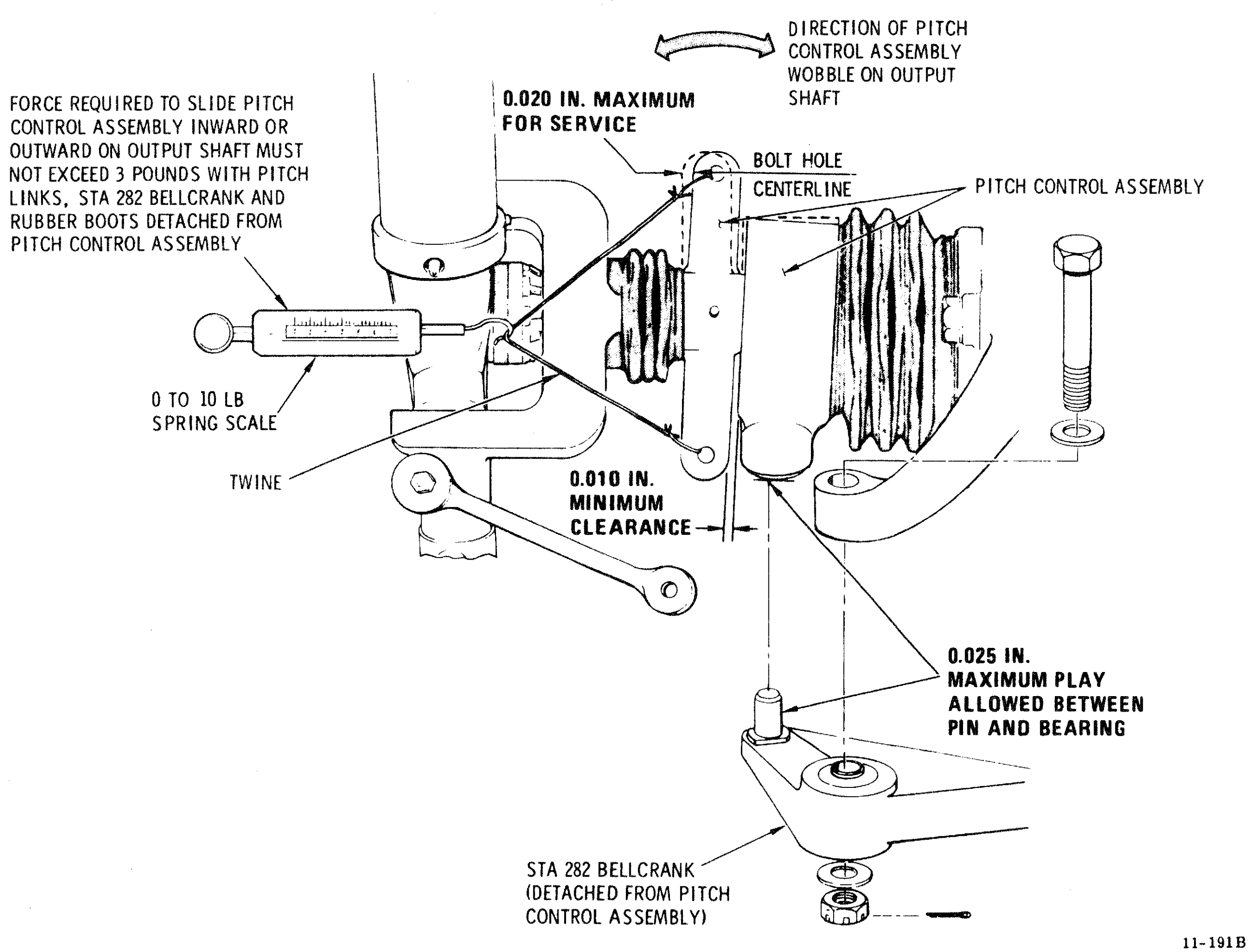


Figure 5-22. Tail Rotor Pitch Control Assembly Inspection.

Use solvent and abrasive cloth (C25) to remove deposits or contamination.

j. Inspect pitch control non-metallic liners for looseness, damage and wear. Maximum ID of inboard smooth liner is 0.9145 inch. Maximum ID of outboard spline liner, measured between two 0.0864-inch-diameter pins (No. 44 drill, shank free of burrs) placed 180 degrees apart, is 0.725 inch.

k. Inspect pitch control links and bearings for bending, cracks, and bearing axial play. **MAXIMUM ALLOWABLE AXIAL PLAY OF BEARING IS 0.020 INCH.**

Table 5-14. Premaintenance Requirements for Repair of Fiberglass Tail Rotor Blades.

Conditions	Requirements
Minimum Personnel Required	One MOS (68E)
Consumable Materials	(C3)(C6)(C7)(C39) (C74)(C79) (C89)(C90)(C91) (C94)(C98)

5-94. Repair - Fiberglass Tail Rotor Blade. Do not attempt repairs other than those permitted herein. The tail rotor operates at high rpm. Use particular care when evaluating the acceptability of any blade trailing edge or abrasion strip deficiencies. For repair of metal tail rotor blades, refer to paragraph 5-99A

WARNING

Unauthorized repairs can exceed balance limits or result in tail rotor failure and possible injury or loss of life.

5-95. Repair - Fiberglass Tail Rotor Blade Scratches, Nicks, and Gouges. *a.* Repair fiberglass skin by filling the defect with epoxy (C39), blending the surface to match contour, and painting the repaired surface.

NOTE

If more than 2 square inches of surface is painted, tail rotor balance may be affected. If epoxy primer coats are removed, the blade must be refinished and balanced.

b. Repair the stainless steel leading edge abrasion strip by polishing out the defect to blend with surrounding contour. Polish with grade 400 abrasive paper (C3).

c. Repair the fork, hub, and blade spars by using grade 400 abrasive paper (C3) to round out and blend the defect. Apply exterior surface touchup treatment and paint touchup.

5-96. Repair - Fiberglass Tail Rotor Blades Tip Cap (AVIM). Tip caps with bonding separation and loose or missing rivets should be repaired as follows.

a. Blade tip caps with bonding separation may be repaired by working adhesive (C7) in separation and applying uniform pressure. Allow adhesive to cure for 8 hours at room temperature or 2 hours at 150°F.

b. Blade tip caps with rivets only at point B (fig. 5-23) should be repaired according to *d* through *h* below.

c. Blade tip caps with rivets installed at points A and C (fig. 5-23) should have loose or missing rivets replaced according to *d* and *f* through *h* below, as required.

NOTE

When rivets are installed in one tail rotor blade tip, an equal number of rivets must also be installed at the opposite blade tip to maintain proper tail rotor balance.

d. Replace loose or missing rivets installed at point B with mechanically expanded rivets (MS20605AD3C2).

e. Drill four holes (No. 30 drill), two through each side of blade tip cap, as shown in figure 5-23. Countersink the holes 100° by 0.042 inch.

f. Measure the thickness of the blade tip and tip cap.

g. If material thickness is between 0.063 and 0.125 inch, install four mechanically expanded rivets (MS20605AD4C3).

NOTE

Blades with four rivets installed do not require the addition of two rivets located at point B (fig. 5-23).

h. Seal all rivet core holes with sealing compound (C89). Smooth surface of sealing compound after application. Allow 3 hours curing time.

i. If tip cap does not have a retention screw installed (fig 5-23) perform *j* through *m* below.

j. Lay out the retention screw location, using the dimensions given on figure 5-23. Use a No. 36 tap drill to drill a 0.500-inch-deep hole.

k. Use a 6-32 UNC-2B tap to thread the screw hole of a depth of 0.400 inch. Clean screw hole to a depth of 0.400 inch. Clean screw hole with solvent (C94).

l. Use phosphoric acid (C74) to clean retaining screw (NAS1635-6LL6) and washer (AN960C6).

NOTE

If screw is not installed within 8 hours after cleaning, it must be recleaned prior to installation.

m. Coat threads of screw with primer and locking compound (C91 or C90) and install washer and screw. Paint screwhead/washer with primer (C79). Allow locking compound to cure for 12 hours at room temperature.

n. Repeat *j* through *m* above for other tail rotor tip cap.

5-97. Repair - Skin Under Fiberglass Tail Rotor Blade Leading Edge Tape (AVIM). *a.* Remove the leading edge abrasive tape (fig. 5-20, sheet 1).

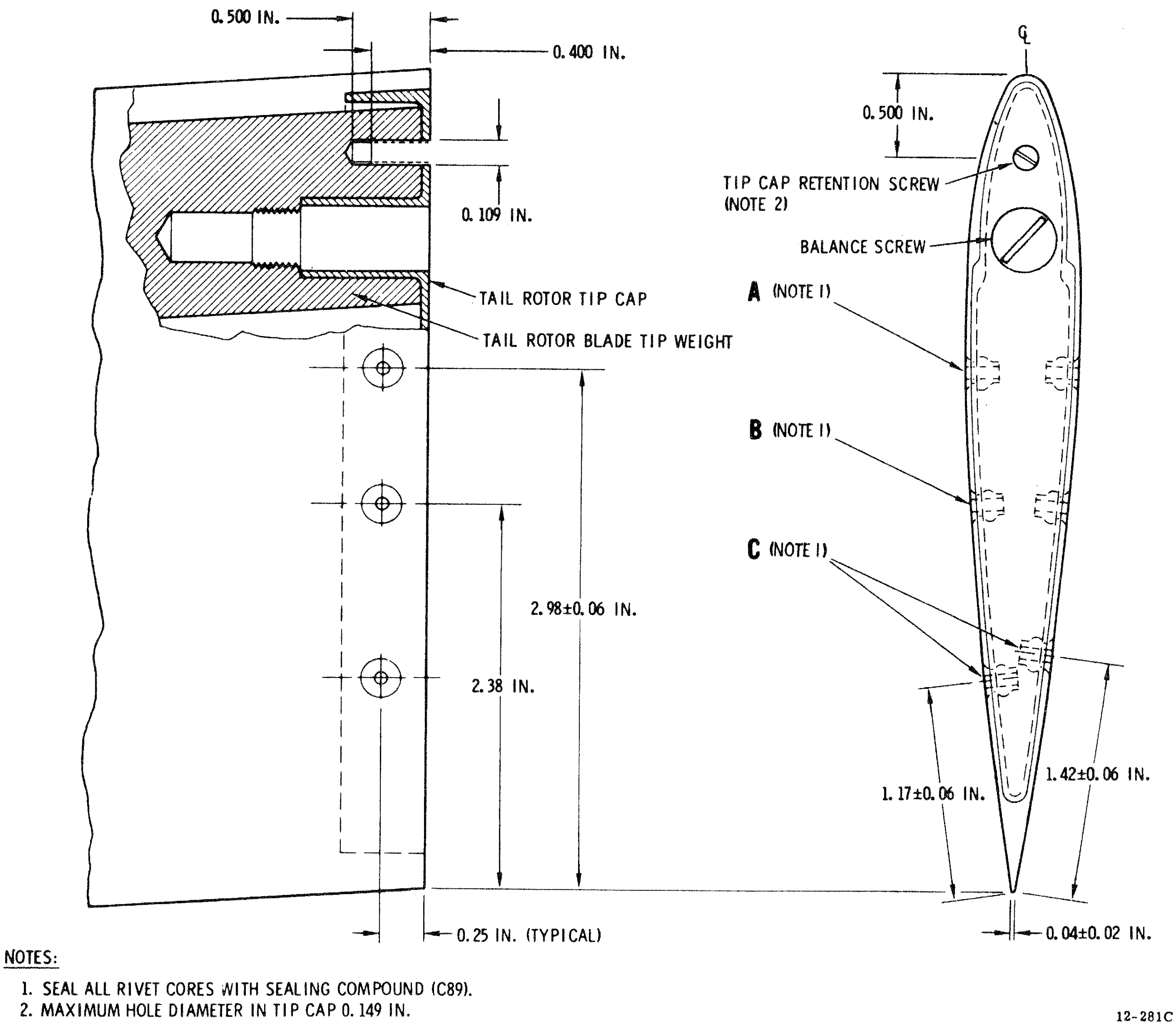


Figure 5-23. Fiberglass Tail Rotor Tip Cap Repair.

CAUTION

Rework is limited to a single ply of fiberglass. Rework is not allowed on root end of spar.

b. Repair the fiberglass by using fiberglass repair kit (C39), and blending the surface to match contour.

c. Repair the abrasive tape, (para 5-99).

5-98. Repair - Blistering or Separation on Paper Covered Fiberglass Tail Rotor Blades (AVIM). *a.* Puncture blister in two locations with separation between punctures approximately equal to the diameter of the blister.

b. Mix adhesive (C6) 100 parts A to 16 parts B by weight. Inject adhesive into one of the punctures until it is forced out of the second puncture. Apply adhesive under separation and flatten the area with finger pressure to squeeze out excess.

c. Allow to cure 18 to 24 hours at room temperature.

5-99. Replacement - Fiberglass Tail Rotor Blade Leading Edge Abrasive Tape. (See fig. 5-20, sheet 1). *a.* Remove old tape and clean area with solvent (C94).

CAUTION

Do not cut tape after it has been installed on blades.

b. Cut 2-in-wide tape (C101) to length as follows.

(1) Spar inboard length is 3.30 inches.

(2) Spar outboard length is 1.18 inches.

(3) Tape length at blade root is to overlap abrasion strip by 0.25 inch.

c. Apply tape on centerline of blade and spar; press down firmly, eliminating all air pockets.

Table 5-14A. Premaintenance Requirements for Repair of Metal Tail Rotor Blades.

Conditions	Requirements
Minimum Personnel Required	One MOS (68E)
Consumable Materials	(C1) (C2) (C3) (C4)

5-99A. Repair - Metal Tail Rotor Blade. Do not attempt repairs other than those permitted herein. The tail rotor operates at high rpm. For repair of fiberglass tail rotor blades, refer to paragraph 5-94.

WARNING

Unauthorized repairs can exceed balance limits or result in tail rotor failure and possible injury or loss of life.

5-99B. Repair - Metal Tail Rotor Blade Scratches, Nicks, and Gouges.

a. Repair the stainless steel leading edge by polishing out the defect to blend with surrounding surface. Polish with grade 400 abrasive paper (C3).

b. Repair the fork, hub, and blade spars by using grade 400 abrasive paper (C3) to round out and blend the defect . Apply exterior surface touchup treatment and paint touchup.

NOTE

If more than 2 square inches of surface is painted, tail rotor balance may be affected. If epoxy primer coats are removed, the must be refinished and balanced.

c. Repair scratches and gouges on the blade surface or spar that do not exceed the repair limits shown in figure 5-21A by polishing and blending to surrounding area. Use grades 180 through 600 abrasive paper (C1), (C2), (C3), and (C4). Apply exterior surface touchup treatment and paint touchup.

5-99C. Repair - Metal Tail Rotor Blade Tip Cap. No repairs are allowed except for replacement of tip weights and attaching hardware. Tail rotor balance shall be required if tip weights are missing or changed.

Table 5-14B. Premaintenance Requirements for Metal Tail Rotor Blade Replacement.

Conditions	Requirements
Consumable Materials	(C15) (C79)

5-99D. Replacement - Metal Tail Rotor Blade.

A new replacement metal blade is prebalanced with tip weights. This blade can be installed on a tail rotor assembly even though the opposite blade may have some erosion or paint wear that would cause minor balance differences. After installation, the assembly must be checked for balance (para 5-110A)and any weight differences adjusted.

CAUTION

The metal tail rotor assembly blade overhaul and retirement schedule (table 1-7) must always be considered when replacing a tail rotor component. THE TAIL ROTOR ASSBMBLY COMPONENT WITH THE HIGHEST TIME (LEAST SERVICE LIFE) LIMITS THE ASSEMBLY.

NOTE

Always record the number and location of existing balance washers when replacing a blade. This record aids in balance troubleshooting.

a. At the blade pitch control assembly (fig. 5- 23A), remove cotter pin, nut, and washer. Remove the bolt with washer and disconnect the pitch control link.

NOTE

Outboard blade attaching bolts and bushings may be reused. It is necessary to know which outboard bushing was under the bolt head.

b. Remove the outboard blade attachment nut. Discard; then push the bolt out.

c. Pull the blade off the hub trunnion. Remove the blade bushing and crush washer installed in the hub at the attachment holes. Identify bushing removed from under bolt head. Discard the crushed washer.

CAUTION

Do not disassemble strap shoes from end of strap pack protruding from trunnion. Strap pack parts are not individually replaceable and must remain assembled. Avoid damaging strap pack. Scratches or nicks on strap laminates are cause for replacement of the entire tail rotor blade assembly.

d. If strap pack is to remain exposed for any length of time, wrap exposed end of strap pack with barrier material (C15) or other similar non- abrasive material to protect from damage.

e. Before installation of replacement blade assembly, remove strap pack protective covering and inspect the visible strap pack. Check for scratches, nicks, and damage to the strap pack laminates and strap shoes; none is allowed.

(1) Visually inspect the bushing and bolt for evidence of interference in the radius area.

(2) Inspect outboard retaining bushings for a 0.070 inch radius or countersink on the inside diameter at the flanged end.

(3) Visually inspect the bushing. The outside diameter of the radius or countersink area must be at least 0.489 inch to clear the bolt head radius. On a properly manufactured bushing, the inside diameter edge radius or chamfer will be 0.060 inch minimum. An accurate measurement is not required. The visual inspection is used to detect obvious discrepancies.

(4) Insert a new bolt through the original bushing using hand pressure only. There must be no clearance between the bolt head and the bushing flange.

(5) If the bushing used under the bolt head has an adequate radius or countersink and no clearance is visible, then the original bolt and bushing may be reused if they are otherwise serviceable and protective plating is not worn through.

(6) If the bushing under the bolt head does not have an adequate radius or countersink and clearance is visible the original bushing and bolt must be replaced prior to furhter operation.

NOTE

Inspection procedures prior to installation of new outboard blade attaching bolts and bushings must be accomplished as described in paragraph 5-99De(1), (2), (3), and (4).

f. Slide blade on hub with blade leading edge facing in counterclockwise direction (viewing hub as in figure 5-23A). Use care to keep blade correctly aligned so that hub trunnion slides into blade pitch bearings. Do not use force.

g. Aline bolt holes in root fitting with hole through bushing in outboard shoes of strap pack.

h. Coat contacting surface of crush washers, bushings, bolt and nut with chromate primer (C79) and immediately, while wet, perform steps i and j below.

CAUTION

Do not permit zinc primer on the threads of the bolt or nut. The primer can act as a lubricant and cause incorrect torque applications.

i. Assemble and install new bolt, bushings and crush washers in sequence shown in figure 5-23A. Make certain that direction of bolt is opposite to that of opposing blade. Do not force bolt; it must have an easy but snug fit through blade, shoes and bushings.

CAUTION

During the next step, take care not to exceed the specified 650 inch-pound torque limit. If the outboard blade attachment bolt is overtorqued, replace the bolt, bushings, crush washers and nut. The blade attachment bolt, bushings, crush washers and nut cannot be lossened and retightened after they have once been torqued; replace crush washers and selflocking nut whenever the nut is lossened or removed for any reason

j. Install nut and TORQUE TO 600 TO 650 INCH-POUNDS while zinc chromate primer is wet.

j.1. Bushings removed from under the nut, if otherwise serviceable and protective plating is not worn through, may be reused under the nut. The radius or countersink is not required under the nut.

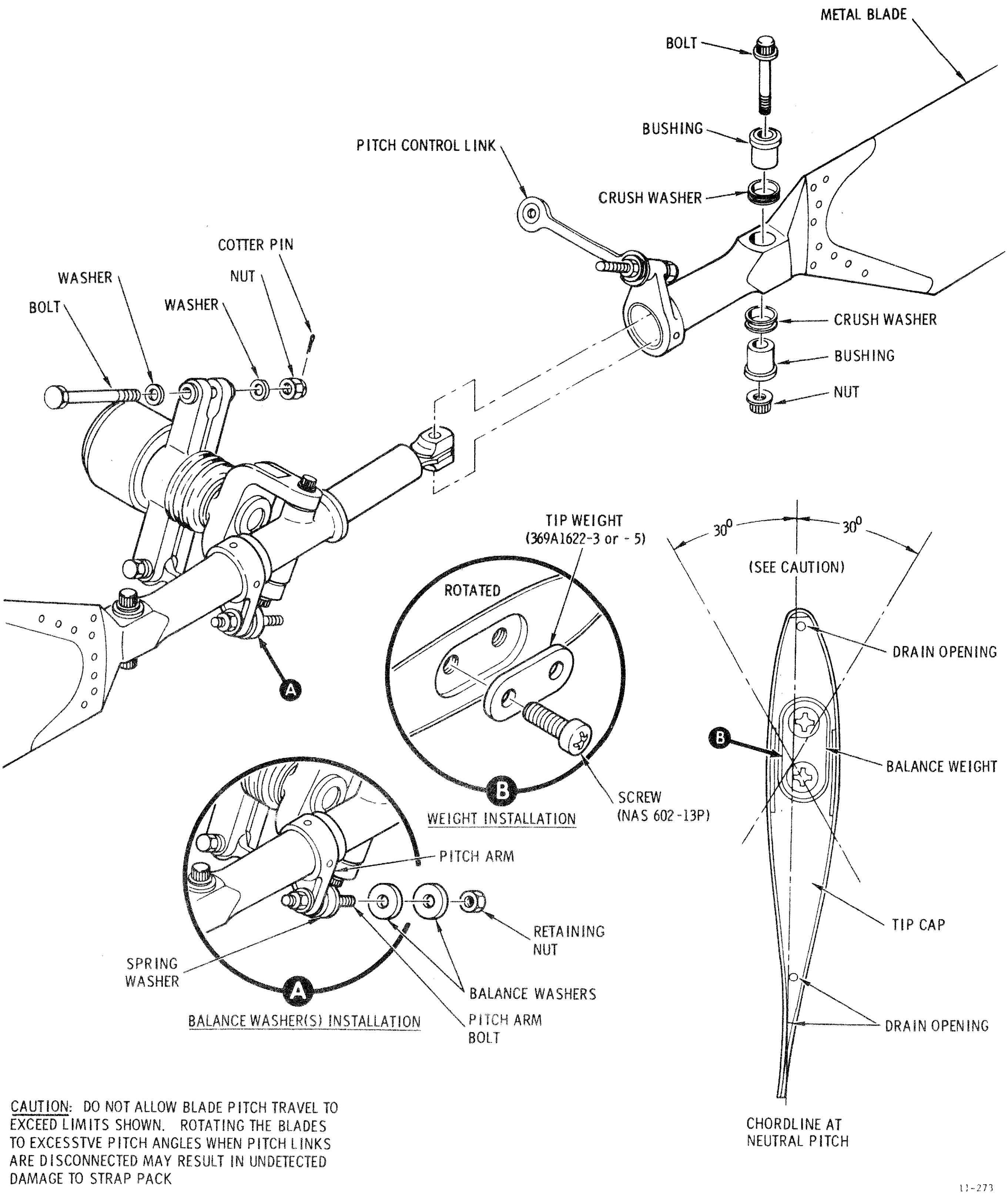


Figure 5-23A. Metal Tail Rotor Blade - Replacement.

CAUTION

After blade is secured to strap pack assembly, do not allow blade pitch travel from neutral to exceed 30 degrees in either direction (fig. 5-23A). Rotating blade to excessive pitch angles may result in undetected damage to strap assembly.

k. At the blade pitch control assembly, position the pitch control link. Install the bolt with washers and nuts. TORQUE NUT TO 50-80 INCH- POUNDS AND INSTALL NEW COTTER PIN.

l. Check the tail rotor assembly for balance (para 5-110A).

5-100. Repair — Tail Rotor Pitch Control System. Repair the tail rotor pitch controls attached at the tail rotor as described in paragraphs 5-101 and 5-102.

5-101. Repair — Tail Rotor Pitch Control Assembly. *a.* Measure the clearance between the swashplate and housing (fig. 5-22). If a minimum clearance of 0.010 inch is not present, the swashplate is unserviceable and must be replaced.

b. Replace a swashplate that exceeds the serviceability limits described in paragraph 5-93.

5-102. Replacement — Tail Rotor Pitch Control Link. Before removal of pitch control links, color code the pitch control arms, bolts, and all washers used at each arm. An identical bolt and the same combination of washers must be reassembled in the positions from which removed or tail rotor balancing will be required.

a. Disconnect ends of pitch control link at swashplate and at pitch control arm, by removing the nuts, washers, and bolts.

b. Insert either end of replacement pitch control link between lugs of swashplate.

c. Install bolt at swashplate with two washers and nut; **TORQUE NUT TO 50 — 80 INCH-POUNDS AND INSTALL NEW COTTER PIN.**

d. Pull pitch control assembly inboard or outboard as required to align pitch control link bearing with pitch control arm bushing.

NOTE

Assemble the extended balance bolt installed in the following step with the spring tension washer on the bolt so that the concave (dished) surface is against the bolt hex portion; then install the reduced OD special washer.

e. Install washer(s), extended balance bolt and nut as shown in figure 5-21. **TORQUE NUT TO 50 — 80 INCH-POUNDS** and install new cotter pin.

Table 5-15. Premaintenance Requirements for Installation of Tail Rotor Assembly.

Conditions	Requirements
Special Tools	(T28)
Consumable Materials	(C3)(C63)

5-103. Installation — Tail Rotor Assembly. (See fig. 5-21.) *a.* Pull beaded end of rotating boot from groove of drive fork and position tail rotor assembly in line with transmission output shaft. Apply lubricant (C63) to inner splines of drive fork only.

b. Rotate tail rotor back and forth slightly until splined swashplate of pitch control assembly engages spline of shaft.

NOTE

Installing tail rotor blades at 90° to the high and low extremes of transmission output shaft runout, using marks applied at time of inspection (chapter 6) reduces chances of high frequency vibration.

c. Slide tail rotor assembly onto shaft. Keep the drive fork-to-hub pivot bolt aligned with lugs of swashplate. Locate tail rotor assembly on shaft so that approximately two threads at end of shaft are exposed outboard of drive fork.

NOTE

Assembling the tang washer, nut, phenolic washer and rubber stop together makes d and e below easier to perform. Marking the outer tang next to the inner key tang also simplifies alignment of the key tang in the shaft groove. Teeter the blades to start the tang washer on the shaft.

d. Slip new tang washer over exposed threads; then install nut by two full turns.

CAUTION

Do not force blade stop into position with a screwdriver; this can crack the stop.

NOTE

The phenolic spacer comes in two thicknesses. Install the thickness (0.15- or 0.19-inch) that will keep the tail rotor blade tips at least 2 inches from the boom with approximately 2 pounds pressure applied at the bladetip. Pedals must be placed in neutral (centered) position for this check.

e. Slide tail rotor assembly outboard against nut. Install phenolic spacer (recessed side facing inboard) and rubber stop.

f. Fold back rotating boot to expose groove in shaft. Groove must be clean and free of paint and foreign matter.

NOTE

Split rings with a full chamfer (sharp edge) must be replaced with split rings having a land (flat edge) and chamfer. Check split rings for uniformity of thickness and chamfer. Rings must be used in matched (two-piece) sets. Widths of the flat outer surfaces must be the same. If the chamfer surface is rough, polish with grade 400 abrasive paper (C3).

g. Check tail rotor fork conical split ring seat for yellow spline alignment dot. If dot is on conical surface, remove and repaint on flat surface adjacent to original dot. Insert split ring so that chamfered edges mate with beveled seat in bore of drive fork.

NOTE

Installing the split ring with its center gaps aligned with the blade assembly centerline reduces the chances for high frequency vibration.

h. While holding split ring in place, slide tail rotor assembly inboard; check that chamfered split ring remains completely nested in drive fork bevel. Check that the fork contacts both split rings. **REPLACE THE SPLIT RINGS IF A GAP OF MORE THAN 0.002 INCH EXISTS** between the fork and one split ring with the other in contact.

CAUTION

- **When torquing retaining nut, be sure that torque wrench and adapter form a right angle as shown (detail A); otherwise, torque reading will be in error.**
- Tang washer movement of more than one Tang width is an indication that the Tang washer inner key has sheared. This condition requires replacement of the washer and reassembly of parts according to the initial installation procedures. To prevent washer movement a nonmetallic rod may be inserted between Tang washer and supported on the fork drive during installation of the nut.

i. Using torque wrench adapter (T28), torque nut to 550 — 600 inch-pounds. Back off the nut to approximately 200 — 250 inch-pounds; then **FINAL TORQUE TO 400 — 450 INCH-POUNDS.**

CAUTION

Tail rotor control pedals, in the following step, must be positioned in neutral. If the foot pedals are in the extreme right or left position, the pitch control links and the stop may be damaged by forcing the blade about the teetering axis against the stop.

j. Install beaded end of non-rotating boot (drain/vent holes down) into groove of pitch control housing. Install beaded end of rotating boot into groove of drive fork.

k. Position station 282 bellcrank so that pivot pin slips into bearing of pitch control assembly.

l. Rotate bellcrank back and forth as required to align bearing with gearbox arm. Install bolt, two washers, nut, and new cotter pin.

m. With the rotor control pedals in neutral, check blade-to-boom clearance at maximum teeter position. Refer to NOTE above step *e*. Bend tab on tang washer to lock retaining nut.

CAUTION

One-half spline mis-alignment between pitch control swashplate and hub drive fork is possible. Perform the following check.

n. Position tail rotor pedals in neutral and observe whether the hub drive fork, the pitch links, and the swashplate are in a line parallel to the gearbox output shaft centerline. If incorrectly assembled, the swashplate will be misaligned approximately 10 degrees from the drive fork. To correct this condition, disconnect the pitch links, remove and rotate the tail rotor 180 degrees; reinstall and reconnect the links.

o. Inspect to determine that there will be no relative motion between the inner race of the pitch control bearing and the tail rotor swashplate (fig. 5-22).

p. Check rigging of tail rotor controls after installation of any removed or replaced parts (chapter 11).

5-103A. Balance - Tail Rotor Assembly.

Either fiberglass or metal tail rotor blade assemblies may be installed on the helicopter (See fig. 5-20). Balance and vibration analysis is accomplished using different equipment for each type. For balancing fiberglass tail rotor blade assemblies, refer to paragraphs 5-104 through 5-110. For metal tail rotor blade assemblies, refer to paragraph 5-110A.

5-104. Adjustment - Fiberglass Blade, Tail Rotor Balance (Without Balancing Kit). Evaluate tail rotor high frequency and correct slight imbalance as follows:

CAUTION

Do not use this procedure for balance adjustment of metal tail rotor blade assemblies. For metal tail rotor blade balance, refer to paragraph 5-110A.

a. With tail rotor operating at normal flight rpm, feel the control pedals for the presence of high frequency vibration (buzz).

WARNING

Before approaching empennage, make certain to notify operator at the flight controls that personnel will be in the tail rotor area. Approach the empennage from the horizontal stabilizer side only.

b. If vibration is noticeable, observe the outboard tip of the horizontal stabilizer. Determine if stabilizer buzz is in excess of 0.06 inch.

NOTE

Blurry appearance of lower vertical or horizontal stabilizer at 100 to 103% N2, with tail rotor pedals in neutral, are typical indications of high frequency vibration. High frequency vibrations can also be produced by components in the power plant and power train installations. (Refer to troubleshooting tables in chapters 4 and 11.)

c. If stabilizer buzz appears excessive (cannot be damped easily by holding tip with one hand), or pedal buzz feels excessive, tail rotor balance should be improved by performing the actions outlined in the following steps.

d. Attempt imbalance correction by repositioning tail rotor on drive shaft, *e* through *h* below.

e. Remove tail rotor and pitch control assembly, rotate 45 degrees from original index position, and reinstall.

f. With tail rotor operating at normal flight rpm, determine if the vibration is improved to an acceptable level.

g. If vibration level is increased, reposition tail rotor 45 degrees in the opposite direction from the original index position.

h. If vibration level is still unacceptable, return tail rotor to the original index position and proceed with *i* below.

i. Attempt rebalancing of the tail rotor by addition of tape, *j* through *m* below.

j. Select either blade. Assuming that it is the light blade, wrap two layers of pressure-sensitive abrasion tape (C103) around the shank of the blade.

k. Operate the tail rotor at normal flight rpm and evaluate the vibration level.

l. If vibration level has decreased, continue to add tape until the vibration level is acceptable.

m. If vibration level has increased, remove the tape and evaluate the opposite blade in the same manner.

n. If an acceptable vibration level can be produced according to *e* through *m* above, continue tail rotor in service with tape installed.

o. If vibration level is unacceptable after attempting correction *e* through *m* above, tail rotor assembly must be either dynamically balanced (para 5-105) or replaced.

Table 5-16. Premaintenance Requirements for Balance Adjustment With Balancing Kit.

Conditions	Requirements
Special Tools	(T18)(T37)
Consumable Materials	(C55)(C103)

5-105. Adjustment - Fiberglass Blade, Tail Rotor Balance (With Balancing Kit). Tail rotor balancing may be accomplished by strobe light observation of the tail rotor during ground run. A vibration pickup (accelerometer), mounted on the tail rotor gearbox, causes the strobe light to flash in each vibration cycle. The flashing light stops the rotating blade image, as viewed by the operator. The vibrations measured in mils on an analyzer dial, and the tail rotor angle measured with a protractor mounted on the strobe light, are recorded by the operator. These data are applied to tables showing location and weight required to restore tail rotor balance.

CAUTION

Do not use this procedure for balance adjustment of metal tail rotor blade assemblies. For metal tail rotor blade balance, refer to paragraph 5-110A.

NOTE

The detailed dynamic balancing instructions outlined in adjustment paragraphs 5-107 through 5-110 below supersede those instructions contained in both the analyzer and the tracking kits.

5-106. Installation - Tail Rotor Vibration Analysis Kit (Fiberglass Blades). (See fig. 5-24).

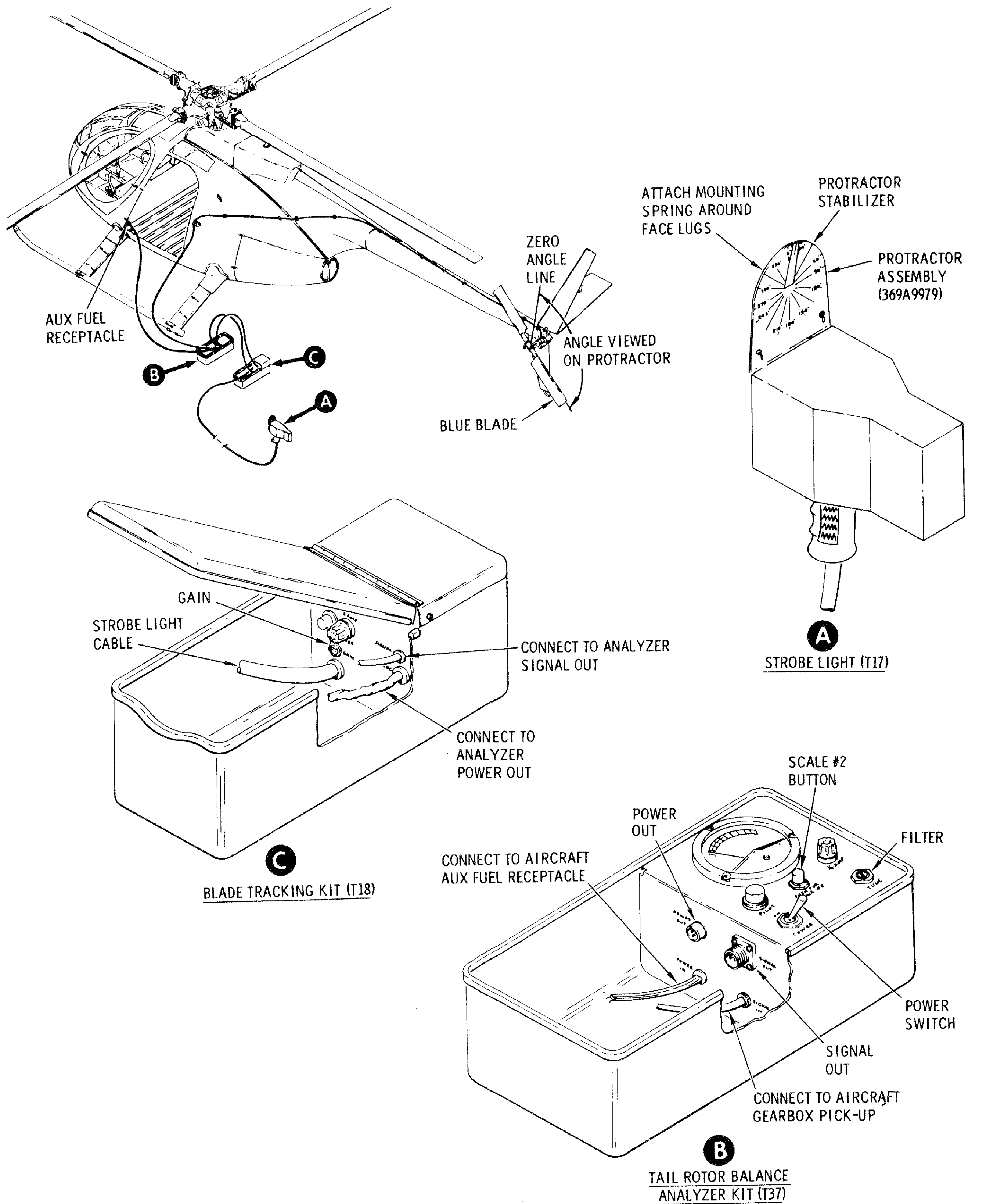
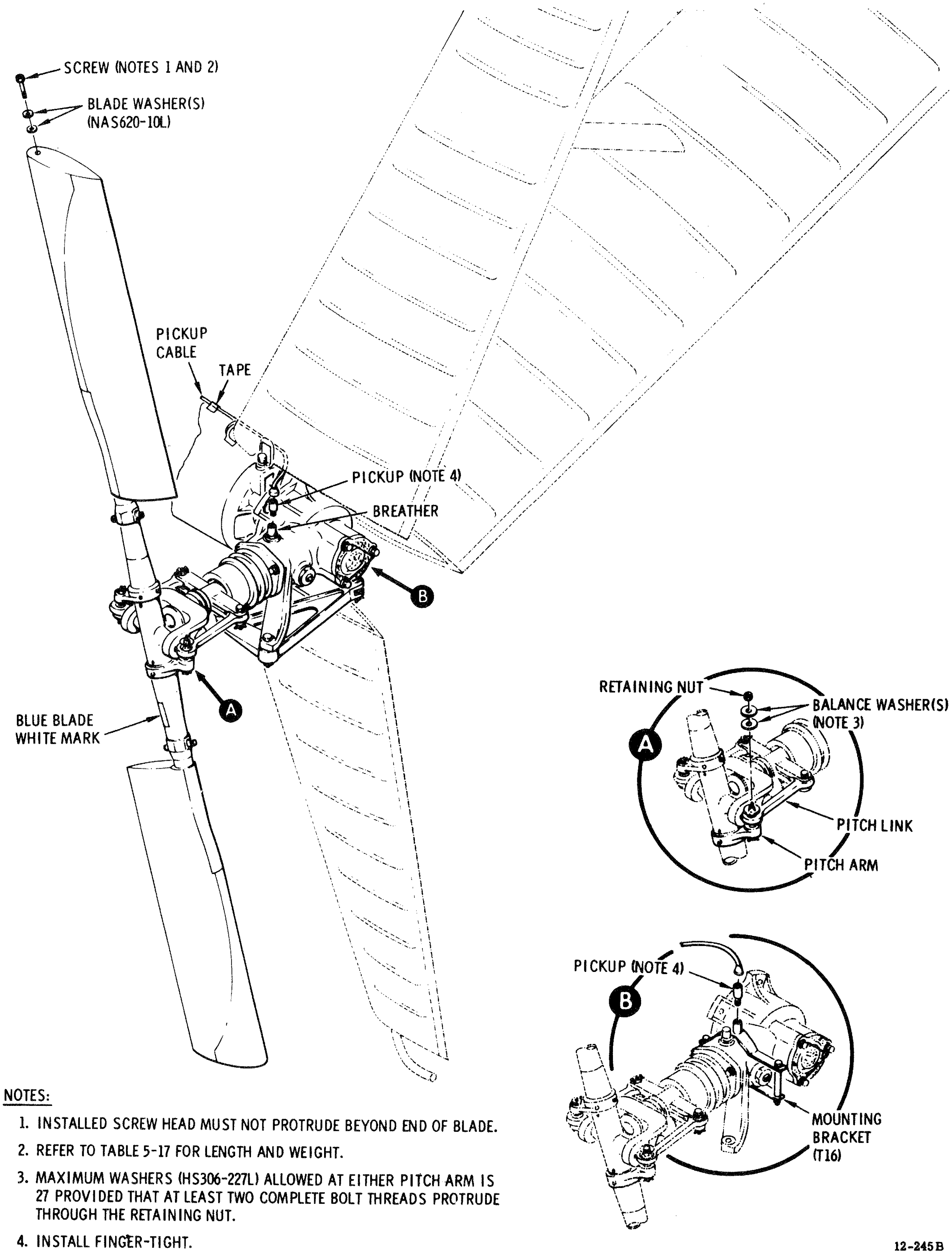


Figure 5-24. Tail Rotor Vibration Analysis Equipment Installation. (Fiberglass Blades)



NOTES:

1. INSTALLED SCREW HEAD MUST NOT PROTRUDE BEYOND END OF BLADE.
2. REFER TO TABLE 5-17 FOR LENGTH AND WEIGHT.
3. MAXIMUM WASHERS (HS306-227L) ALLOWED AT EITHER PITCH ARM IS 27 PROVIDED THAT AT LEAST TWO COMPLETE BOLT THREADS PROTRUDE THROUGH THE RETAINING NUT.
4. INSTALL FINGER-TIGHT.

12-245B

Figure 5-25. Tail Rotor Vibration Analysis. (Fiberglass Blades)

NOTE

Tool number prefix 369A has been omitted from kit details listed in the following instructions for brevity.

a. On tail rotor gearbox without breather-filler threaded for vibration pickup, install pickup mounting bracket assembly (9920) as shown in detail B, figure 5-25; then install vibration pickup on mounting bracket.

b. On tail rotor gearbox having a breather-filler with a threaded hole, install pickup on breather (fingertight).

c. Connect vibration pickup cable to pickup. Route cable forward along right side of tailboom (and under the adf sense antenna as applicable). Secure with tape (C103).

CAUTION

Weight cable on ground to prevent possible fouling of tail rotor.

d. Connect components of the vibration analyzer kit (T37) and blade tracking kit (T18) as shown in figure 5-24.

e. Attach protractor assembly (9979) to strobe light.

f. If not already marked, identify blue blade with paint (C55) using a 3/8- by 3/4-inch stencil. (See fig. 5-25.)

g. Set strobe light ON-OFF switch at ON.

h. Adjust analyzer indicator needle to zero, if necessary.

5-107. Preparation - Tail Rotor Balancing (Fiberglass Blades). (See fig. 5-24). Ground run engine at 103% N2 with main rotor blades in flat pitch and tail rotor pedals in neutral. With AUX TANK (or UTILITY POWER, as applicable) switch ON, adjust analyzer and strobe as follows:

a. Adjust analyzer filter (TUNE) until the meter reads maximum.

b. In tracking strobe case, adjust GAIN control counterclockwise until strobe light will flash; then rotate 1/8-turn more.

NOTE

Do not operate the strobe light with the GAIN too high, as this may cause the light to flash erratically and indicate false angles.

5-108. Analysis - Tail Rotor Vibrations (Fiberglass Blades). (See fig. 5-24). *a.* Stand approximately 10 feet away from the tail rotor and 90 degrees to the axis of rotation.

b. Direct the strobe light toward the rotating tail rotor and align the leading edge of the protractor stabilizer with the leading edge of the aircraft upper vertical stabilizer; then read the angle on the protractor between the blue blade leading edge and the 0 degree centerline. Record reading.

NOTE

Erratic strobe light flashing is expected at a meter reading of 1-1/2 mils or lower and is an indication of good balance.

c. Read the value on analyzer upper dial. If analyzer meter needle reads above 10, depress SCALE #2 button and read the lower dial.

d. If meter reading is 2 or less (strobe light flashing erratic), balance is acceptable.

e. If meter reading is more than 2, record reading, shut down engine and proceed with *f* or *g* below, as appropriate.

f. When meter reading is more than 2, but less than 20:

(1) Use the figure 5-26 chart to determine number and location of balance washers required.

(2) Install or remove washers on the specified blade tip and/or pitch arm bolt. Refer to paragraphs 5-109 and 5-110 for weight removal and adjustment procedures.

(3) Repeat vibration analysis.

g. When meter reading is more than 20, inspect tail rotor installation as follows:

(1) Remove tail rotor assembly (para 5-90) and inspect split ring (fig. 5-21). Replace if damaged or mismatched.

(2) With tail rotor removed, check TIR of tail rotor gearbox output shaft with dial indicator contact in split ring groove of the gear shaft. **RUNOUT IS LIMITED TO 0.005-INCH TIR.** Replace tail rotor gearbox if beyond limit. Refer to chapter 6 for gearbox replacement.

(3) Repeat vibration analysis if gearbox or split ring is defective.

h. When meter reading is more than 20 (tail rotor and gearbox installation inspection satisfactory), replace tail rotor.

5-109. Balance - Fiberglass Tail Rotor Blade Tip. (See fig. 5-25). A weight increase at a light blade tip may be

Washer locations	Angle (degrees)	Meter reading (mils) 2	4	6	8	10	12	14	16	18
Blue tip / Green pitch arm	0	1/2	1/4	1/7	1/10	1/13	1/15	1/18	1/22	1/23
	15	1/2	1/4	2/7	2/9	3/12	3/14	3/17	4/20	5/21
	30	1/2	2/4	3/7	3/8	4/9	5/11	6/12	7/14	7/17
Blue tip Only	45	1	2	3	4	5	6	7	7	8
	60	1	2	3	4	5	6	7	8	9
	75	1	2	3	4	5	6	7	8	9
	90	1	2	3	4	5	6	6	7	8
	105	1	2	3	3	4	4	5	7	7
	120	1	1	2	3	3	3	4	5	6
	135	1	1	2	2	2	2	3	3	3
Blue tip / Blue pitch arm	150	0/2	1/4	1/6	1/10	1/12	1/14	1/17	1/20	1/23
	165	0/2	0/4	0/7	0/10	0/13	0/15	0/18	0/20	0/23
Green tip / Blue pitch arm	180	1/2	1/4	1/7	1/10	1/13	1/15	1/18	1/22	2/23
	195	1/2	1/4	2/7	2/9	3/12	3/14	3/17	4/20	5/21
Green tip Only	210	1	2	3	3	4	5	6	7	7
	225	1	2	3	4	5	6	7	7	8
	240	1	2	3	4	5	6	7	8	9
	255	1	2	3	4	5	6	7	8	9
	270	1	2	3	4	5	6	6	7	8
	285	1	2	3	3	4	4	5	7	7
	300	1	1	2	3	3	3	4	5	6
	315	1	1	2	2	2	2	3	3	3
Green tip / Green pitch arm	330	0/2	1/4	1/6	1/10	1/12	1/14	1/17	1/20	1/23
	345	0/2	0/4	0/7	0/10	0/13	0/15	0/18	0/20	0/23

Use of table

For any angle and meter reading combination, the value above the diagonal line in the table is the number of small washers (NAS620-10L) to be added to the blade tip and the value below the diagonal line is the number of large washers (HS306-227L) to be added to the pitch arm. The maximum number of washers allowed at either pitch arm is 27 provided that two complete bolt threads protrude through the retaining nut. The tip screw head must not protrude beyond the blade tip.

Example: Angle - 195 degrees
Meter reading - 8

Read down the angle column to 195 degrees, then right to point of intersection at meter column 8. The numerals 2/9 will appear in this block, which indicates number of small large washers required.

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Figure 5-26. Tail Rotor Balance Weight Location Chart. (Fiberglass Blades)

obtained by equivalent washer removal from the opposite blade. Always remove washers from opposite tip, if installed, and subtract from the weight value to be added before adding more weight. Adjust tip weight as follows:

a. Remove blade tip screw; select washer and screw combination determined from the figure 5-26 chart and table 5-17.

Table 5-17. Fiberglass Blade Tail Rotor Balance Equivalent Weight Chart.

AT BLADE TIPS			
	Screws (LENGTH) *(In.)*		Weight *Value**
			(Minus) 4-1/2
NAS1351-3-6P	3/8		
NAS1351-3-12P	3/4		0
NAS1351-3-14P	7/8		1-1/2
NAS1351-3-16P	1		3
	Washer (THICKNESS) *(In.)*		Weight Value *(Fig. 5-26 Chart)*
Steel			
NAS620-10L	0.032		1
AT PITCH ARM BOLTS			
	Washers THICKNESS *(In.)*	OD *(In.)*	Weight Value *(Fig. 5-26 Chart)*
Steel			
HS306-227L	0.016	0.800	1

The "Weight Value" is the equivalent value of washers for a change from the installed to the listed screw. Example: A change from a NAS1351-3-12P to a 16P is equal to three washers in the figure 5-26 chart. Measure installed screw or bolt length for weight value determination.

NOTE

If not already accomplished, replace existing screw with screw (NAS1351-3) provided in analyzer kit. With washers stacked on screw, the remaining threads must not exceed 0.75 inch nor be less than 0.31 inch. Each blade tip must have a screw installed. Screw length may be changed to increase or decrease tip weight and/or maintain screw length tolerance. A longer or shorter screw can replace washers. The screw Weight Value column in table 5-17 shows the equivalent number of washers that must be added or taken away to maintain the same weight or to change weight at the blade tip screw. A selection of screws is provided in the vibration analyzer kit.

b. Install screw and washer combination. Installed screw must not protrude beyond blade tip.

5-110. Balance - Fiberglass Tail Rotor Blade Pitch Arm. (See fig. 5-25.) A weight increase at a light pitch arm may be obtained by equivalent weight removal from the opposite pitch arm. Always remove washers from opposite pitch arm, if installed, and subtract from the weight value to be added before adding more weight. Adjust pitch arm weight as follows:

a. Remove balance washer retaining nut (detail B) from extended balance bolt and install or remove washers specified in the figure 5-26 chart. A maximum of 27 washers may be used on either pitch arm bolt provided that at least two complete bolt threads protrude through the nut.

b. If a tail rotor has the maximum number of washers allowed on one pitch arm and the opposite pitch link appears larger, exchange one link for the other and repeat vibration analysis.

5-110A. Balance and Vibration Analysis - Metal Blade Tail Rotor Assembly.

Balancing of tail rotor assemblies with metal blades is accomplished by use of the Chadwick-Helmuth balancing kit. The carrying case of this kit contains all instrumentation, balance charts and miscellaneous items needed to balance the tail rotor. Also included is a track and balance handbook for use with the equipment to correct balance when such can be obtained by addition or subtraction of weight at pitch-arm studs or at blade tips. Since vibration reduction by weight adjustment is dependent on proper mechanical condition of the tail rotor and tail rotor drive system, troubleshooting information in paragraph 5-89 should be used with balancing kit instructions. Acceptance criteria for balance and vibration are contained in the balancing kit and on each balance chart. For specific instructions on use of the balancing hardware, refer to a and b below.

a. Balance at Blade Pitch Arm. (See figure 5-23A.) Weight increase at light pitch arm may be obtained by removal of equivalent washer weight from opposite pitch arm. Always remove washers from opposite pitch arm, if installed, and subtract from weight to be added before adding more weight . For washer data, refer to table 5-18. Maximum washer weight allowed at either pitch arm bolt is 26.91 grams (23 washers).

NOTE

There is possibility of slight weight variation between pitch control links. If tail rotor has maximum balance washer weight allowed on one pitch arm, compare the two links. If pitch link opposite the weight requirement appears larger, exchange one link for the other and repeat vibration analysis.

b. Balance at Blade Tip. When balancing procedures indicate that weight should be added to a tip it is preferable (if possible) to remove an equivalent weight from the opposite tip, to keep overall weight to a minimum. Installation of tip weights is not mandatory. However, open screw holes are not permitted; screws must be installed. Shorter than normal screws may be used for balance if minimal thread engagement (5/16-inch) exists.

(1) Remove tip-weight screws and weights (fig. 5-23A). Select balancing hardware indicated by balancing procedure.

(2) Install combination of weights required. Maximum weight permitted is twenty-four grams at each tip. For balance hardware data, refer to table 5-18.

5-111. Rigging Check — Tail Rotor Controls. Check rigging of tail rotor controls following installation of any removed or replaced parts. (Refer to chapter 11.)

Table 5-18. Metal Blade Tail Rotor Balance Weight - Value Chart

At Blade Tip (Note 4)			At Blade Pitch Arm (Note 3)			
Screw (Note 5)			Washer (Note 1, 2)			
Part Number	Length (Inch)	Weight (Grams)	Part Number	Thickness (Inch)	Weight (Grams)	OD (Inch)
NAS1352-08-12P	0.75	2.22	HS306-227L	0.016	1.17	0.800
NAS1352-08-14P	0.875	2.44				
	Weight					
	Thickness (Inch)	Weight (Grams)				
369A1622-3	0.016	0.29				
369A1622-5	0.036	1.76				

Notes:
1. Used on balance bolt.
2. Maximum of 23 washers is permitted on each balance bolt.
3. Minimum of two threads must extend past nut securing balance washers on balance bolt.
4. Maximum weight of weight plus screws is 24 grams.
5. Minimum screw thread engagement is 5 16 inch.

CHAPTER 6

DRIVE TRAIN SYSTEM

SECTION I GENERAL

6-1. POWER TRAIN SYSTEM.

6-2. Description — Power Train System. The power train system, starting at the engine power takeoff pad, consists primarily of the overrunning clutch, drive shaft, and transmission shown in figure 6-1. At 100 percent, N2, engine power output through the clutch to the main transmission is 6000 rpm; main transmission output to the main rotor is 470 rpm. Main transmission output through the tail rotor drive shaft to the tail rotor transmission is 2045 rpm; tail rotor transmission output to the tail rotor is 3018 rpm. Rotation of the components is in the direction shown in figure 6-1.

6-3. Troubleshooting — Power Train System. Refer to table 6-1.

SECTION II MAIN DRIVE SHAFT

6-4. MAIN ROTOR DRIVE SHAFT.

6-5. Description — Main Rotor Drive Shaft. The main rotor drive shaft is a shot-peened, nitrided steel-alloy forging having a spline coupling at one end that mates with the main transmission output shaft (fig. 6-2), and an octagonally shaped head that attaches to the main rotor hub with four bolts and four hoisting eyebolts.

6-6. Inspection — Main Rotor Drive Shaft (Installed). Check exposed head of drive shaft for scratches, cracks, breaks, loosened attaching bolts, and any indication of stress damage.

6-7. Removal — Main Rotor Drive Shaft.

CAUTION

The main rotor drive shaft is a highly stressed part. Do not allow tools to strike the shaft, or the shaft to strike any object. Any impact damage may require replacement of the drive shaft.

a. Remove the nuts, washers, bolts, and lifting bracket or hoisting eyebolts (fig. 6-3).

CAUTION

Any time the drive shaft is removed, cover the opening in the top of the main rotor hub to prevent possible entry of foreign matter into the hub, mast and the transmission.

b. Lift drive shaft clear of main rotor hub.

6-8. Inspection — Main Rotor Drive Shaft (Removed). *a.* Check all surfaces of drive shaft for dents, nicks, scratches, and evidence of deformation. No damage of any kind is permissible in the area within 3.6 inches of the transmission end of the shaft. Dents less than 0.005-inch deep and 0.50-inch diameter are permissible in other areas. A dent is defined as a smooth depression of the surface with no sharp edges or abrupt changes in contour.

b. Check all external surfaces of shaft for corrosion.

c. Check for spline wear (0.005 inch maximum). Wear is measured by placing two 0.108 inch pins in opposite spline grooves; minimum diameter over the pins is 2.265 inches.

d. If shaft condition is questionable, refer to AVIM for magnetic particle inspection according to TM 55-1500-204-25/1. No surface or subsurface cracks are permissible.

e. Check entire shaft for presence of oil coating.

6-9. Repair — Main Rotor Drive Shaft. No repair of the drive shaft is permitted except for the removal of

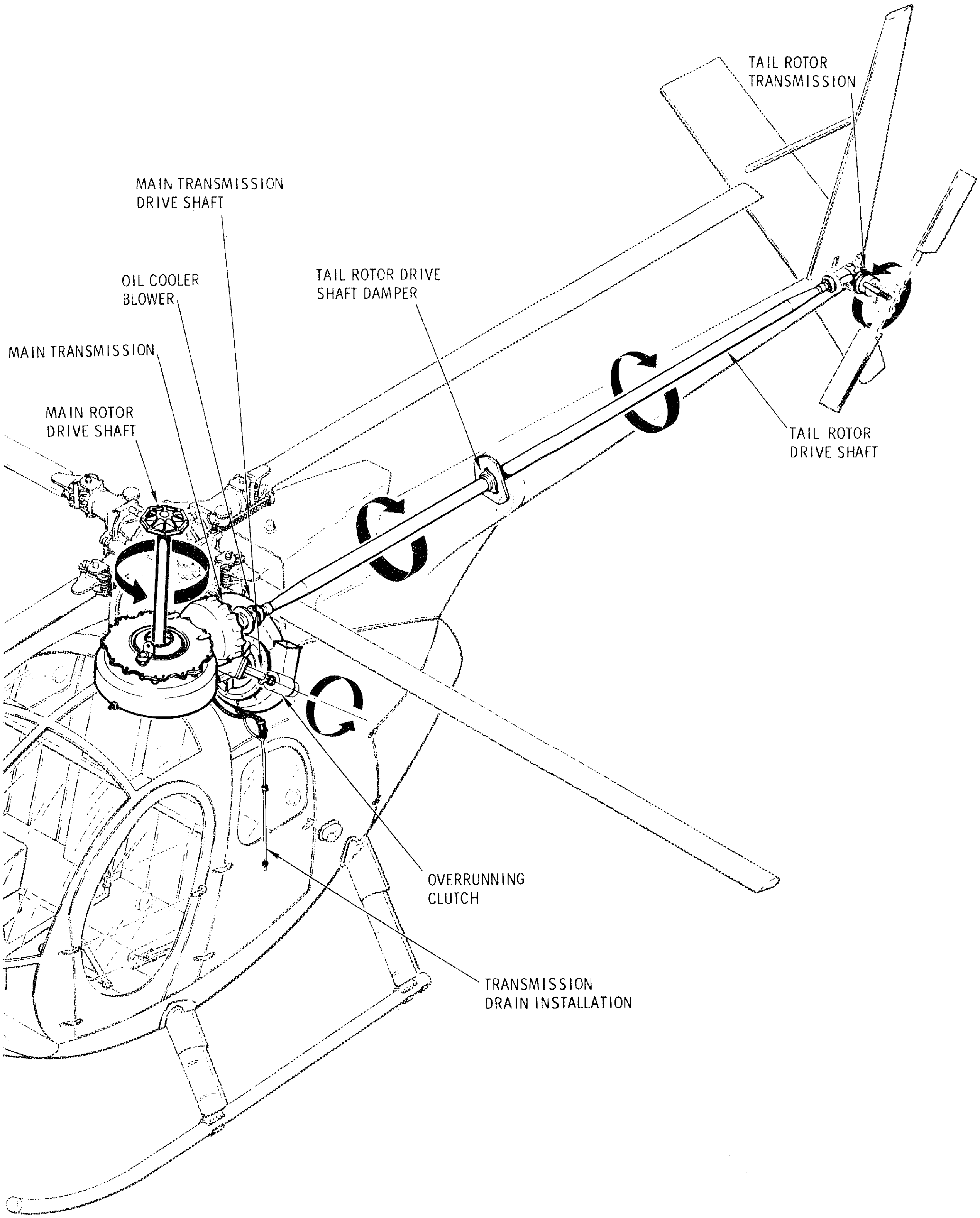


Figure 6-1. Power Train System.

Table 6-1. Troubleshooting of the Power Train System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Main rotor does not rotate.**

STEP 1. Check to make sure that the N2 turbine is not turning (TM 55-2840-231-24).

STEP 2. Check for defective overrunning clutch.

If the overrunning clutch is found defective, replace the overrunning clutch (para 6-27).

STEP 3. Check for defective main transmission.

If the main transmission is found defective, replace the main transmission (para 6-35).

2. **Main transmission drive shaft vibrates.**

STEP 1. Check for loose attaching bolts on main transmission drive shaft (para 6-13).

If the attaching bolts are found to be loose, retorque the bolts (para 6-17).

STEP 2. Check for dented or bent drive shaft (para 6-13).

If the drive shaft is found to be dented or bent, check the drive shaft for serviceability limits. Repair or replace the shaft as necessary (para 6-11).

STEP 3. Check for loose or incorrectly shimmed couplings (para 6-17).

If the couplings are found to be loose or incorrectly shimmed, check couplings for not less than 0.010 inch shimming. Adjust shims as required to maintain correct installation fit of driveshaft. Check coupling bolt for correct torque (para 6-17). Install the correct number and size of shims under couplings (para 6-17) as necessary.

STEP 4. Check for cracks in the diaphragm discs or welds.

If cracks are found in diaphragm discs or weld areas replace the drive shaft (para 6-14).

3. **Excessive noise or high frequency vibration in oil cooler blower assembly.**

STEP 1. Check for cracked or loose rivets on the impeller or a damaged scroll.

If the impeller or scroll is found defective, replace that particular item (para 6-19).

4. **Overrunning clutch vibrates or will not free-wheel.**

STEP 1. Check for a defective clutch.

If the overrunning clutch is found to be defective, replace the clutch (para 6-32).

5. **Main transmission requires frequent oil servicing.**

STEP 1. Check for oil leakage at the transmission input or output seals.

If oil leakage rate exceeds specified limits (para 6-104), replace the transmission (para 6-38).

STEP 2. Check for oil leakage around the accessory drive cover.

If oil leakage is found around the accessory drive cover, replace the transmission (para 6-38).

6. **Oil temperature warning light comes on.**

STEP 1. Check for a defective oil temperature sender.

If the oil temperature sender is defective, replace the oil temperature sender (para 6-53).

Table 6-1. Troubleshooting of the Power Train System. (cont)

STEP 2. Check for a blockage of the cooling air.

If a blockage is found, clear the blockage and clean the immediate area.

STEP 3. Check for low oil level or oil above full level in main transmission.

If the oil level is incorrect, establish correct oil level (chapter 1).

STEP 4. Check for a defective lubrication pump.

If the lubrication pump is found to be defective, replace the pump (para 6-46).

STEP 5. Check for defective bearings or gears.

If the bearings or gears are found to be defective, replace the transmission (para 6-38).

7. **Excessive noise in main transmission.**

NOTE

Noise that signals a defective transmission should not be confused with normal gear backlash noise usually audible at rundown or low rpm, low power conditions.

STEP 1. Check transmission oil filter and chip detector for contamination.

If contamination is found, replace the transmission (para 6-38).

8. **Oil pressure warning light on (oil pressure low). Light should go out at approximately 55 percent N2.**

STEP 1. Check for low oil level in main transmission.

If the main transmission oil level is low, service the transmission (chapter 1).

STEP 2. Check for a defective oil pressure sender.

If the oil pressure sender is found to be defective, replace the sender (para 6-53).

STEP 3. Check for a defective lubrication pump.

If the lubrication pump is found to be defective, replace the pump (para 6-46).

STEP 4. Check for a clogged filter element.

If the filter element is found to be clogged, remove, clean, and replace filter element (para 6-45).

STEP 5. Check for an internal lubrication system failure.

If the internal lubrication system is found to have failed, replace the transmission (para 6-38).

9. **Tail rotor drive shaft or damper vibrates at high frequency (100 percent N2).**

STEP 1. Check the damper alignment.

If the damper is found to be out of alignment, adjust damper friction (para 6-85).

STEP 2. Check for a defective damper.

If the damper is found to be defective, replace the damper (para 6-86).

STEP 3. Check for a bent or dented tail rotor drive shaft.

If the tail rotor drive shaft runout is beyond limits, replace the tail rotor drive shaft (para 6-14).

STEP 4. Check for loss of balance weights or defective tail rotor drive shaft.

If balance weights are missing or the tail rotor drive shaft is defective, replace the tail rotor drive shaft (para 6-14).

STEP 5. Check for tail rotor out of balance.

If the tail rotor is found to be out of balance, rebalance or replace the tail rotor (chapter 8).

Table 6-1. Troubleshooting of the Power Train System. (cont).

STEP 6. Check for loose or incorrectly shimmed couplings.

If a coupling is loose or incorrectly shimmed, check coupling for not less than 0.010 inch shimming. Adjust shims as required. Retorque coupling bolt(s), (para 6-17).

STEP 7. Check for water in the tail rotor transmission coupling diaphragm.

If water is found in the tail rotor transmission coupling diaphragm, drain the water. Replace coupling if it is corroded (para 6-76).

10. Tail rotor transmission output shaft vibrates.

STEP 1. Check for bent or excessive tail rotor transmission output shaft run-out.

If the tail rotor transmission output shaft is bent or the run-out is excessive, replace the tail rotor transmission (para 6-93).

11. Excessive noise in tail rotor transmission.

STEP 1. Check chip detector for contamination.

If contamination is found, replace the transmission (para 6-93).

12. Chip detector caution light on.

STEP 1. Check for contamination of the transmission oil.

If metal particles are found refer to paragraph 6-54 for inspection and maintenance procedures.

STEP 2. Check for a defective chip detector.

If the chip detector is found to be defective, replace the chip detector.

minor scratches or dents and light surface corrosion by using the procedures outlined below.

CAUTION

Do not allow hands or fingers to contact any part of the shaft after degreasing and cleaning steps.

a. Degrease affected area of the drive shaft with a clean cloth saturated by solvent (C96).

CAUTION

The metal conditioner used in *b* below will irritate hands on repeated exposure. Protective rubber gloves should be worn.

b. This step is necessary for corrosion removal only. For scratch and dent removal proceed with *c* below. Using a clean cloth or brush, apply a diluted solution (1 part to 3 parts water) of metal conditioner (C68A) to corroded area. Keep wet with solution for 10 minutes, or until corrosion appears to be removed. Wipe clean, inspect, and repeat as necessary until there is no further evidence of corrosion.

c. Rinse with water and dry with a clean cloth.

d. Lightly abrade the affected surface with grade 400 abrasive cloth (C24) to remove the blemish. If metal removal exceeds the depth for the specified area listed below the shaft is unserviceable and must be replaced. (See fig. 6-3.)

(1) *Area 1:* Removal in excess of 0.001 inch from the surface between 3.6 to 6.0 inches of the transmission end of the shaft.

(2) *Area 2:* No damage or rework is allowed. Replace shaft.

(3) *Area 3:* Removal in excess of 0.005 inch from the surface of the shaft between areas 1 and 2.

(4) *Area 4:* No damage or rework is allowed. Replace shaft.

e. Repeat *a* through *c* above.

f. Immediately apply a coating of preservative oil (C61). Oil should cover the entire shaft after rework.

Table 6-2. Premaintenance Requirements for Installation of Main Rotor Drive Shaft.

Conditions	Requirements
Special Tool	(T30)

6-10. Installation — Main Rotor Drive Shaft. (See fig. 6-3.)

WARNING

Before installation of main rotor drive shaft, verify that the main rotor hub retaining nut is secured and safetied according to paragraph 5-23.

a. Check that shaft is coated with corrosion preventive (para 6-9). Position drive shaft over main rotor hub with gear coupling down.

b. Slowly and carefully lower drive shaft through main rotor hub and mast until gear meshes with main transmission internal gear teeth.

c. Rotate drive shaft head until holes align with main rotor hub.

d. On aircraft equipped with lifting bracket, install bracket, four attaching bolts and countersunk washers (beveled sides next to boltheads). Install one flat washer and nut on each bolt but do not tighten bolts.

e. On aircraft equipped with hoisting eyebolts, install four eyebolts and countersunk washers (beveled sides next to boltheads) at every second bolthole location. Install two flat washers and nut on each bolt but do not tighten nuts.

f. Install four remaining bolts in the same manner, using two flat washers under each nut.

g. Align hoisting eyebolts so that a centerline through the eyebolt flats intersects the drive shaft axis.

h. **TORQUE ALL NUTS TO 120 TO 140 INCH-POUNDS** using torque wrench adapter (T30) as shown in figure 6-3.

i. After installation of main rotor drive shaft, apply sealant C88 where drive shaft meets main rotor hub.

6-11. MAIN TRANSMISSION DRIVE SHAFT.

6-12. Description — Main Transmission Drive Shaft. The main transmission drive shaft (fig. 6-2) is located behind the sound insulation and access cover at the center of the cargo compartment aft bulkhead. The drive shaft is a dynamically prebalanced steel shaft equipped with a flexible diaphragm-type joint and mounting flange at each end. The shaft interconnects the overrunning clutch and the transmission input shaft.

6-13. Inspection — Main Transmission Drive Shaft (Installed). *a.* Remove the sound insulation and access cover from the cargo compartment aft bulkhead.

b. Inspect the drive shaft diaphragms for dents, cracks, scratches, nicks, corrosion, and evidence of joint separation from the shaft or at the outside diameter edges. Evidence of any such defects on either diaphragm requires removal of the shaft from service.

c. Inspect the drive shaft tube between diaphragms for dents, scratches, cracks, or corrosion pits.

d. Measure depth and diameter of dents. Dents that blend smoothly into surrounding surface area with no sharp change in contour and do not exceed 0.015-inch depth are acceptable. Dents that are not within these limits require removal of the shaft from service.

e. Measure depth of cracks, nicks, corrosion pits, or scratches; length and direction are not critical. **MAXIMUM DEPTH ALLOWED BEFORE REWORK IS 0.003-INCH.** The shaft must be removed to perform rework.

f. Check suspected cracks by using dye penetrant. If a crack indication appears, remove the black (phenolic compound) coating from the area by using grade 400 abrasive paper (C3).

NOTE

The drive shaft is coated with a special phenolic thermosetting compound. It is a brittle coating that may indicate a crack that does not penetrate the tube.

g. Reinspect the questionable area by using dye penetrant. If condition remains questionable, remove shaft for magnetic particle inspection. If crack does not reappear, polish the surface with crocus cloth (C25) to restore the original shaft finish, and touch up the polished area with primer coating (C79).

h. Check attaching bolts for looseness.

i. Install access cover and sound insulation.

6-14. Removal — Main Transmission Drive Shaft. (See fig. 6-4.) *a.* Remove sound insulation and main gearbox access door over main transmission drive shaft in the cargo compartment (chapter 2).

b. Remove eight bolts and washers that secure shaft to couplings.

c. Remove drive shaft; use care to prevent shaft from striking any object.

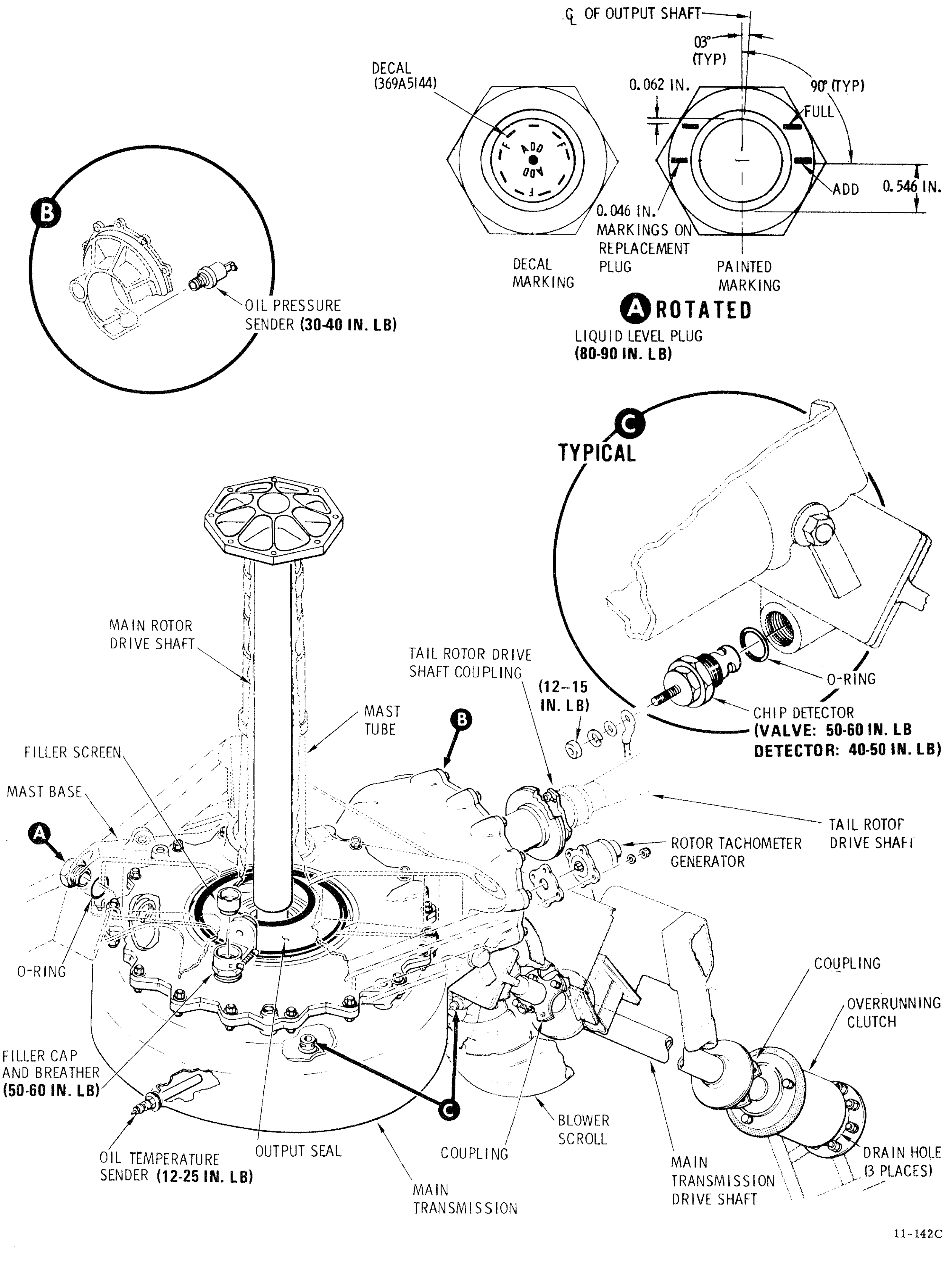
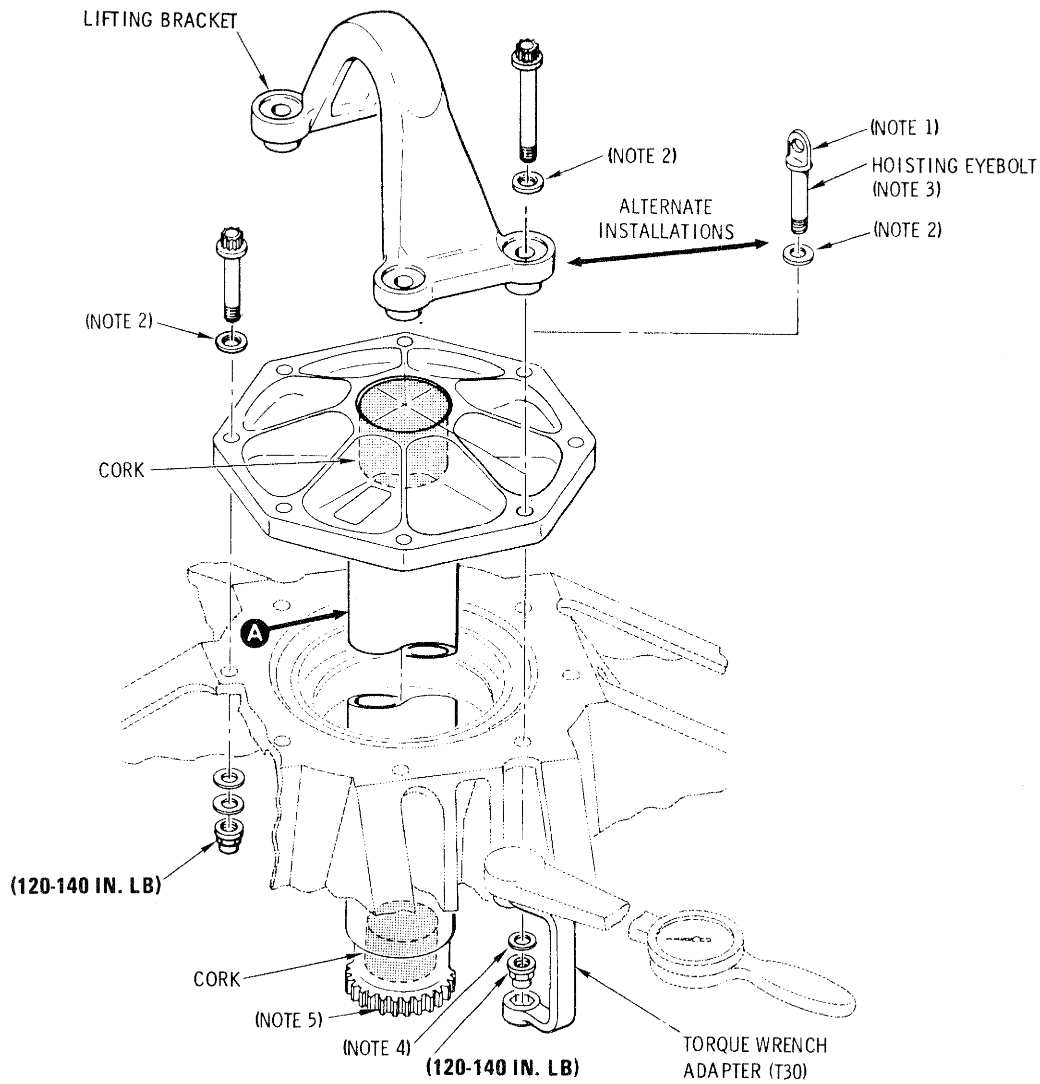
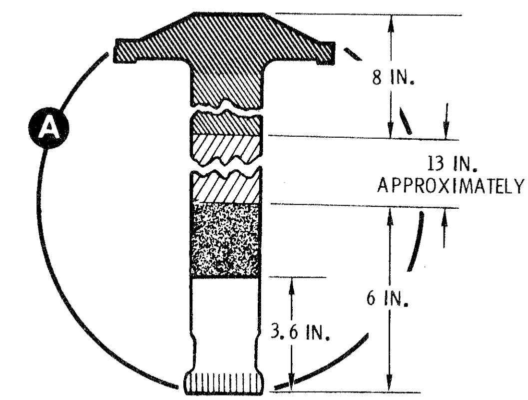


Figure 6-2. Main Transmission Installation.





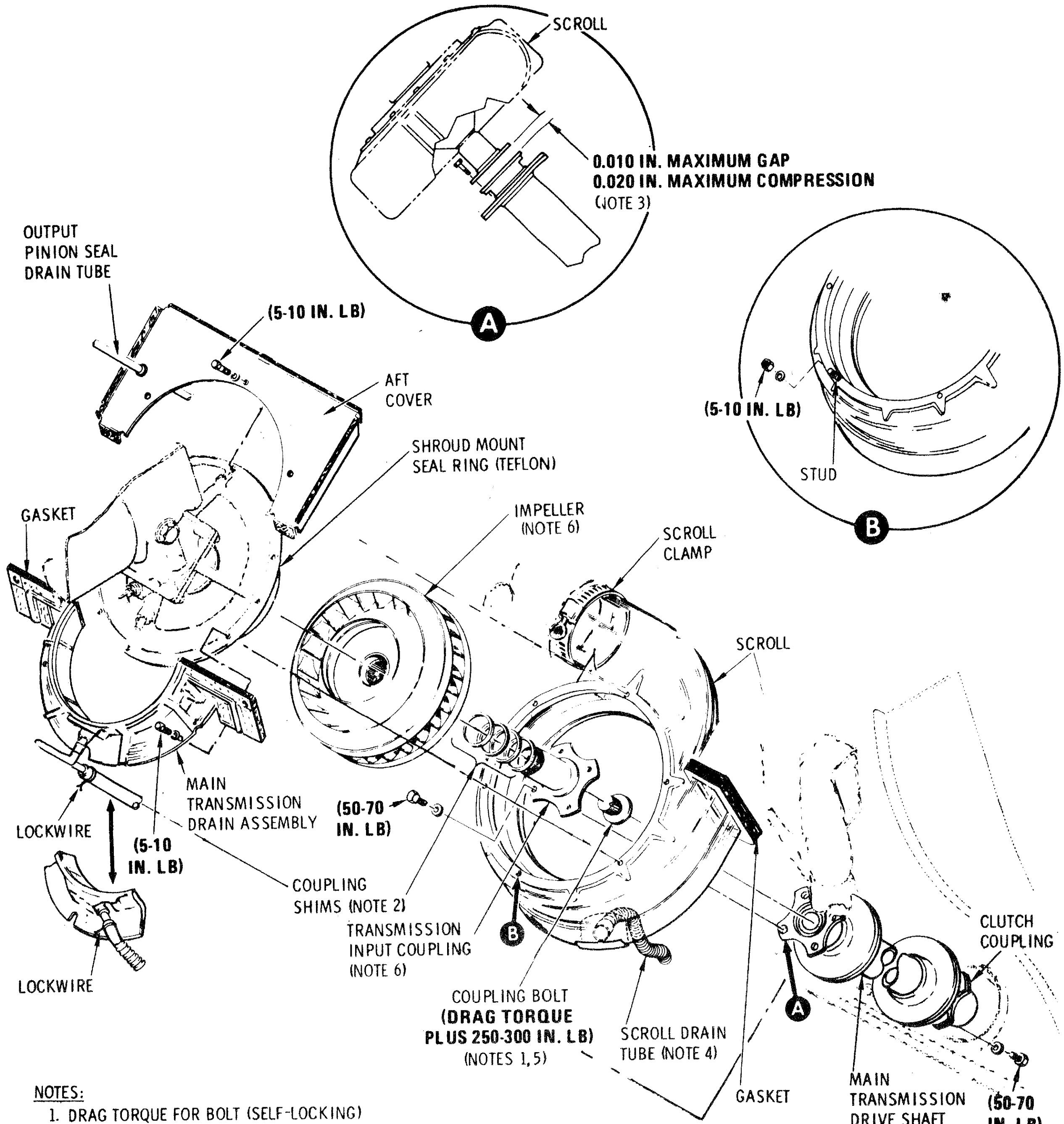
AREA 1: MAXIMUM PIT DEPTH 0.001 INCH.
AREA 2: NO DAMAGE PERMISSIBLE.
AREA 3: MAXIMUM PIT DEPTH 0.005 INCH.
AREA 4: NO DAMAGE PERMISSIBLE (NOTE 5).

NOTES:

1. FLAT OF EYEBOLT HEADS TO INTERSECT DRIVE SHAFT AXIS.
2. INSTALL WITH COUNTERSINK NEXT TO EACH BOLTHEAD. OTHER WASHERS ARE STANDARD FLAT.
3. REPLACE ORIGINAL EYEBOLTS (1.235 IN. SHANK) WITH 369A1010-3 EYEBOLTS (1.360 IN. SHANK) WHEN AVAILABLE. DISCARD OLD EYEBOLTS.
4. INSTALL ONLY ONE FLAT WASHER UNDER NUTS AT THE FOUR LIFTING BRACKET BOLT LOCATIONS AND TWO WASHERS UNDER FOUR REMAINING NUTS.
5. TO CHECK FOR SPLINE WEAR, PLACE TWO 0.108 IN. PINS IN OPPOSITE SPLINE GROOVES; MINIMUM DIAMETER OVER THE PINS IS 2.265 IN.

12-130F

Figure 6-3. Main Rotor Drive Shaft.



NOTES:

1. DRAG TORQUE FOR BOLT (SELF-LOCKING) SERVICEABILITY IS 15 IN. LB MINIMUM, 150 IN. LB MAXIMUM.
2. CORROSION RESISTANT STEEL SHIM; 0.002-IN. LAMINATIONS. CAUTION: ID MUST NOT BE MORE THAN 1.20 IN.
3. CHECK WITH LOWER END OF SHAFT ATTACHED.
4. SCROLL DRAIN INSTALLATION NOT INSTALLED ON ALL AIRCRAFT.
5. COAT COUPLING BOLT THREADS WITH ANTISEIZE COMPOUND (C14) BEFORE ASSEMBLY.
6. COAT COUPLING AND IMPELLER SPLINES WITH LUBRICANT (C63).

Figure 6-4. Main Transmission Drive Shaft and Oil Cooler Blower Replacement

6-15. Inspection — Main Transmission Drive Shaft (Removed). *a.* Check the drive shaft diaphragms for dents, cracks, scratches, nicks, corrosion and evidence of joint separation from the shaft or at the outside diameter edges. Evidence of any such defects on either diaphragm requires replacement of the shaft.

b. Check the drive shaft tube between diaphragms for dents scratches, cracks, or corrosion pits. Determine serviceability according to *c* through *e* below.

c. Measure depth and diameter of dents. Dents that blend smoothly into surrounding surface area with no sharp change in contour and that do not exceed 0.015-inch depth are acceptable. Dents that are not within these limits require replacement of the shaft.

d. Measure depth of cracks, nicks, corrosion pits, or scratches; length and direction are not limited. **MAXIMUM DEPTH ALLOWED BEFORE REWORK IS 0.003 INCH.**

e. Check suspected cracks using fluorescent-penetrant according to TM 55-1500-204-25/1. If a crack indication appears, remove the black (phenolic compound) coating from the area with grade 400 abrasive paper (C3) and crocus cloth (C25) according to paragraph 6-16.

NOTE

The drive shaft is coated with a special phenolic thermosetting compound. It is a brittle coating that may indicate a crack that does not penetrate the tube.

f. Reinspect the suspected area using magnetic-particle or fluorescent-penetrant methods (TM 55-1500-204-25/1). If crack does not reappear, touch up cleaned area with primer coating (C79) and return part to service.

NOTE

Perform inspections g and h below only if shaft damage is suspected.

g. Check shaft tube for out-of-round conditions. **OUT-OF-ROUND SHALL NOT EXCEED 0.060-INCH TIR.**

h. Check shaft for straightness. **SHAFT SHALL BE WITHIN 0.030-INCH TIR** at all locations relative to the centerlines of the two flange mounting bolt patterns.

i. Check all attaching hardware for stripped or crossed threads, and corrosion. Discard unserviceable hardware. Check nutplates for drag torque.

6-16. Repair — Main Transmission Drive Shaft. *a.* Repair of the drive shaft coupling diaphragms is not permissible.

b. Repair all shaft tube damage that is **NOT DEEPER THAN 0.003 INCH.** Completely remove the defect. Maintain a smooth transition into the surrounding surface. Use grade 400 wet or dry abrasive paper (C3) for preliminary finishing, followed by polishing with crocus cloth (C24). Restore the surface until it equals the original finish of the shaft. After rework, check the shaft tube wall thickness in the repair area. Wall thickness shall not be less than 0.025 inch. Apply primer coating (C79) for corrosion protection.

6-17. Installation — Main Transmission Drive Shaft. (See fig. 6-4.) *a.* Assure that coupling bolt threads are coated with anti-seize compound (C14) and **BOLT IS TORQUED TO ACTUAL DRAG TORQUE PLUS 250-300 INCH-POUNDS.**

NOTE

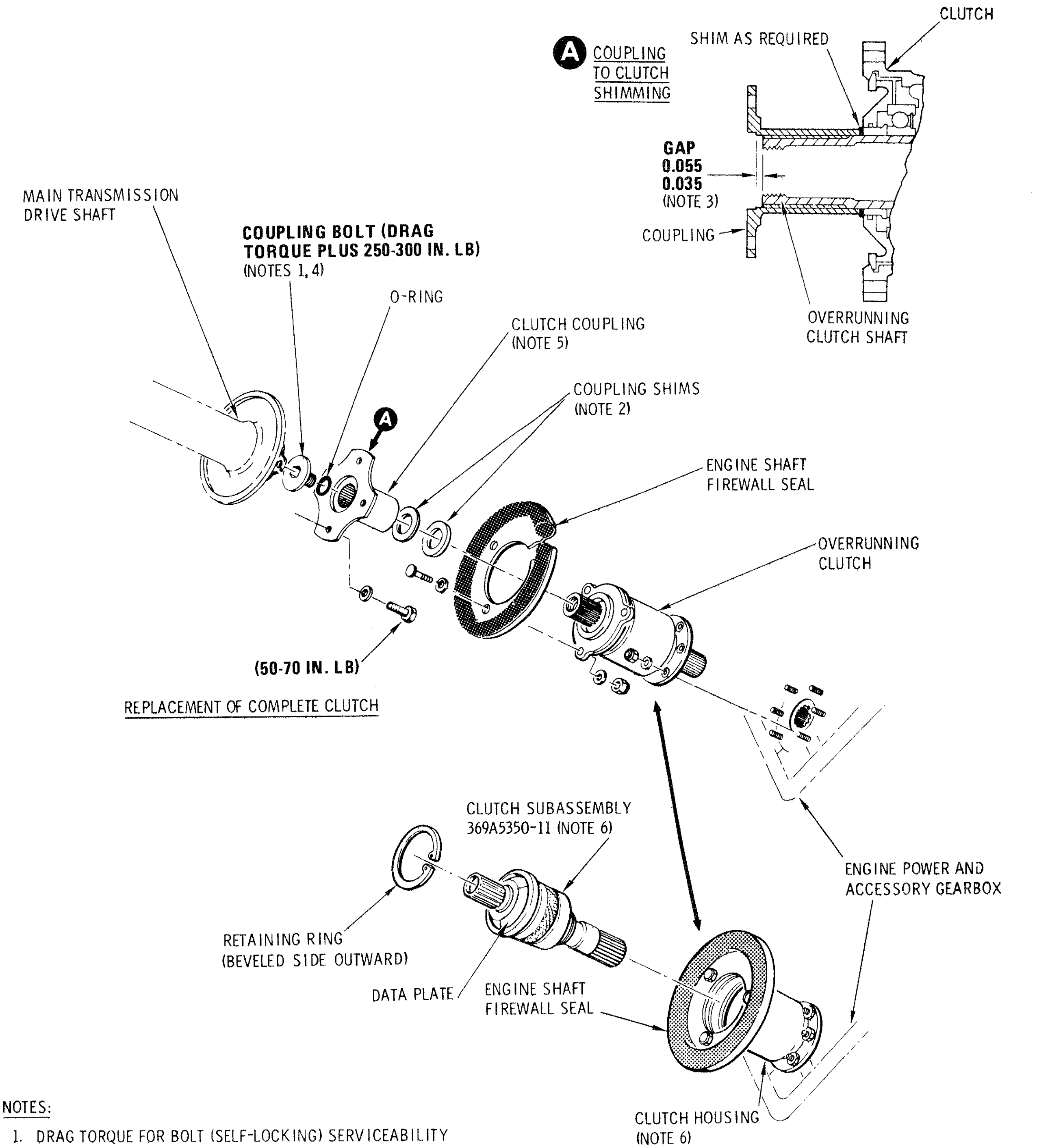
If a bolt is not seated and/or existing torque is found to be less than 250 inch-pounds, check to be sure that the bolt's self-locking drag torque is not less than 15 or more than 150 inch-pounds before retorquing bolt.

b. Position main transmission drive shaft between couplings. **COMPRESSION OF THE DIAPHRAGMS (STATIC STATE) IS LIMITED TO 0.020 INCH.** If shaft diaphragm compression is excessive, continue with *c* below. If diaphragm compression is within limits or if there is a gap between the shaft flanges or the coupling flanges, continue with *d* below.

CAUTION

Do not use clutch coupling shims under the transmission coupling. The OD of these shims is approximately the same; however, the ID of the clutch coupling shims (1.260 to 1.270 inches) is approximately 0.070 inch larger than the transmission coupling shims ID. The difference in ID can cause improper seating of the shim resulting in misalignment and cocking of the coupling.

c. Remove the bolt that secures the transmission coupling and remove coupling. Remove sufficient shims to eliminate excessive diaphragm compression, retaining a **MINIMUM OF 0.010 INCH IN SHIMS THAT MUST REMAIN INSTALLED UNDER COUPLING.**



NOTES:

1. DRAG TORQUE FOR BOLT (SELF-LOCKING) SERVICEABILITY IS 15 IN. LB MINIMUM, 150 IN. LB MAXIMUM.
2. CORROSION RESISTANT STEEL SHIM; 0.002-IN. LAMINATIONS.
3. USE FEELER GAGE TO CHECK GAP DIMENSION.
4. COAT COUPLING BOLT THREADS WITH ANTISEIZE COMPOUND (C14) BEFORE ASSEMBLY.
5. COAT COUPLING-TO-CLUTCH SPLINES WITH LUBRICANT (C63).
6. THE CLUTCH SUBASSEMBLY SERIAL NUMBER IDENTIFIES THE COMPLETE REPAIRED CLUTCH. REFER TO TEXT.

Figure 6-5. Overrunning Clutch Repair or Replacement.

Coat bolt threads with anti-seize compound (C14) and coupling splines with lubricant (C63). Reinstall coupling and bolt. **CHECK THAT BOLT DRAG TORQUE IS NOT LESS THAN 15 OR MORE THAN 150 INCH-POUNDS. TORQUE BOLT TO ACTUAL DRAG TORQUE PLUS 250-300 INCH-POUNDS.**

NOTE

Transmission input coupling shims are laminated stock. Each lamination is 0.002 inch thick. Peel away as required.

d. Attach drive shaft lower flange to clutch coupling with four bolts and washers. **TORQUE BOLTS TO 50 — 70 INCH-POUNDS.**

e. Align the drive shaft and transmission coupling flanges. **ASSURE THAT SHAFT DIAPHRAGM COMPRESSION DOES NOT EXCEED 0.020 INCH** or that gap between drive shaft and coupling flanges **DOES NOT EXCEED 0.010 INCH.** Attach drive shaft upper flange to transmission coupling with four bolts and washers. **TORQUE BOLTS TO 50 — 70 INCH-POUNDS.**

f. Install access door and sound insulation over main transmission drive shaft.

6-18. OIL COOLER BLOWER.

6-19. Description — Oil Cooler Blower. The blower (fig. 6-4) consists of an impeller mounted directly on the transmission input pinion gear shaft, within a polycarbonate plastic scroll-type enclosure that is attached to the transmission housing. The impeller, a riveted and brazed assembly of aluminum blades and a casting, is dynamically balanced at 6000 rpm. The blower draws cooling air over the main transmission in addition to supplying forced ambient air to the engine oil cooler, compartment heating system, and the engine area. Chapter 4 provides information on the oil cooler ducting.

6-20. Inspection — Oil Cooler Blower (Installed). *a.* Check for damage such as cracks, holes, crazing, or parting of joints.

b. Check for security of attaching bolts around the lower flange of the main transmission.

c. Check for scroll wear caused by impeller rubbing.

6-21. Removal — Oil Cooler Blower. (See fig. 6-4.) *a.* Remove main transmission drive shaft (para 6-14).

b. Remove scroll drain tube from scroll, if installed.

c. Loosen scroll clamp.

d. Remove bolts and washers that attach scroll to main transmission. (Remove nut from scrolls fitted with stud.)

e. Remove drain tubes, scroll, aft cover, and main transmission drain assembly.

f. Remove coupling bolt, transmission input coupling, shim(s), and impeller. Retain shim(s) with coupling for reuse.

6-22. Inspection — Oil Cooler Blower (Scroll Removed). *a.* Check for surface damage. Scratches and nicks that do not exceed 0.012 inch in depth are repairable.

b. Check for, and measure length of cracks. Cracks that do not exceed 3.0 inches in length, do not extend to the hardware mounting holes, and will not impair the function of the scroll are repairable.

c. Check ribs and seams for separation, and threaded inserts (and stud, as applicable) for stripped threads or looseness in flange.

NOTE

There may be small irregular-shaped voids in the cement between the cemented pieces. Such voids shall not be considered as delamination (separation).

d. Check rectangular gasket for deterioration.

e. Check for scroll wear caused by impeller rubbing.

f. Check teflon seal ring (fig. 6-4) that is bonded to the transmission shroud and bearing retainer mount. Seal ring must be completely bonded around the mount perimeter.

6-23. Inspection — Oil Cooler Blower (Impeller Removed). *a.* Check for cracks and separated or deformed vanes. A cracked impeller, or one having damaged vanes, shall be replaced.

b. Check for surface damage. Scratches and nicks not exceeding 0.006-inch depth are repairable.

6-24. Repair — Oil Cooler Blower Scroll. *a.* Repair cracks that do not exceed 3 inches in length, do not extend to the hardware mounting holes, and will not impair the function of the scroll. Repair by bonding a patch according to chapter 2 instructions.

b. Repair scratches and nicks not exceeding 0.012-inch depth by sanding the affected area to blend smoothly with surrounding surface area. Use grade 280 abrasive paper (C2).

c. Replace inserts that are stripped or loose. Use self-locking inserts NAS1394-3L for replacement.

d. Repair small areas of rib or seam separation by injecting dichloromethane (C38) into void area and cementing together under light pressure. Repair deteriorated or otherwise damaged gasket (fig. 6-4) by replacing with new gasket material (C44).

e. Repair a teflon seal ring that is not completely secure around outside of transmission shroud and bearing retainer mount by bonding loose area with a mixture of 100 parts of resin (C84) to 74 parts of activator (C5). Cure at 150°F for 1 hour or 8 hours at room temperature.

6-25. Repair — Oil Cooler Blower Impeller. *a.* Replace an impeller that is cracked, has damaged vanes, or loose rivets securing adapter to impeller or balance weights to impeller.

b. Repair scratches and nicks not exceeding 0.006-inch depth by sanding. Use grade 280 to 400 abrasive paper (C2) and (C3) to blend defect into surrounding surface area.

6-26. Installation — Oil Cooler Blower. (See fig. 6-4.)

NOTE

Paragraph 6-55 contains initial installation instructions for the scroll drain tube.

a. Coat transmission input shaft with lubricant (C63) and slide impeller on shaft.

b. Install retained shim(s), transmission input coupling, and coupling bolt. (Shim thickness shall not be less than 0.010 inch.) **CHECK THAT BOLT SELF-LOCKING DRAG TORQUE IS NOT LESS THAN 15 INCH-POUNDS. TORQUE BOLT TO 250-300 INCH-POUNDS ABOVE DRAG TORQUE.**

c. Position scroll to engage oil cooler and heater system ducting. Position aft cover and yoke-type drain assembly. Install bolts and washers to secure scroll to transmission. (Use nut on scrolls fitted with stud.) Tighten the bolt (and stud nut, as applicable); **TORQUE SHOULD NOT EXCEED 5-10 INCH-POUNDS.**

d. Tighten oil cooler duct clamp and connect output seal drain tube. Secure lower end of drain tube with a double wrap of lockwire.

e. Turn impeller and check for clearance with scroll.

f. Install main transmission drive shaft (para 6-17).

SECTION III CLUTCHES

6-27. OVERRUNNING CLUTCH.

6-28. Description — Overrunning Clutch. The overrunning clutch (fig. 6-2) transmits power from the engine to the main transmission drive in a clockwise direction. The purpose of the clutch is to disengage the engine from the remainder of the drive system (allow free-wheeling) in case of engine failure and during autorotations. The clutch contains a sprag unit that disengages automatically when N2 rpm is less than the corresponding main rotor rpm.

6-29. Inspection — Overrunning Clutch (Installed). (See fig. 6-5.) *a.* Remove the sound insulation and access cover in cargo compartment, and open engine access doors.

b. Visually inspect case and mounting flanges of clutch for evidence of cracks, corrosion, or excessive heat.

c. Check attaching bolts and nuts for looseness. Check for looseness of the clutch coupling that would indicate looseness of the coupling retaining internal wrenching bolt.

d. Check that oil drain holes are clean and free of obstruction.

e. Check for evidence of oil leakage or seepage. If leakage is evident, check the oil level.

f. Reinstall access cover, sound insulation, and close engine access doors.

6-30. Inspection — Overrunning Clutch Oil Level. Check clutch oil level whenever leakage is noted or the main transmission drive shaft or engine is removed for maintenance.

a. Remove main transmission drive shaft (para 6-14).

b. (See fig. 6-4.) Remove coupling bolt, preformed packing (O-ring), clutch coupling, and shim(s).

c. On aircraft with clutch housing drain holes, check that the three drain holes are clean and free of obstruction. Oil leakage may indicate engine power output seal leakage if clutch oil level is checked and found within limits.

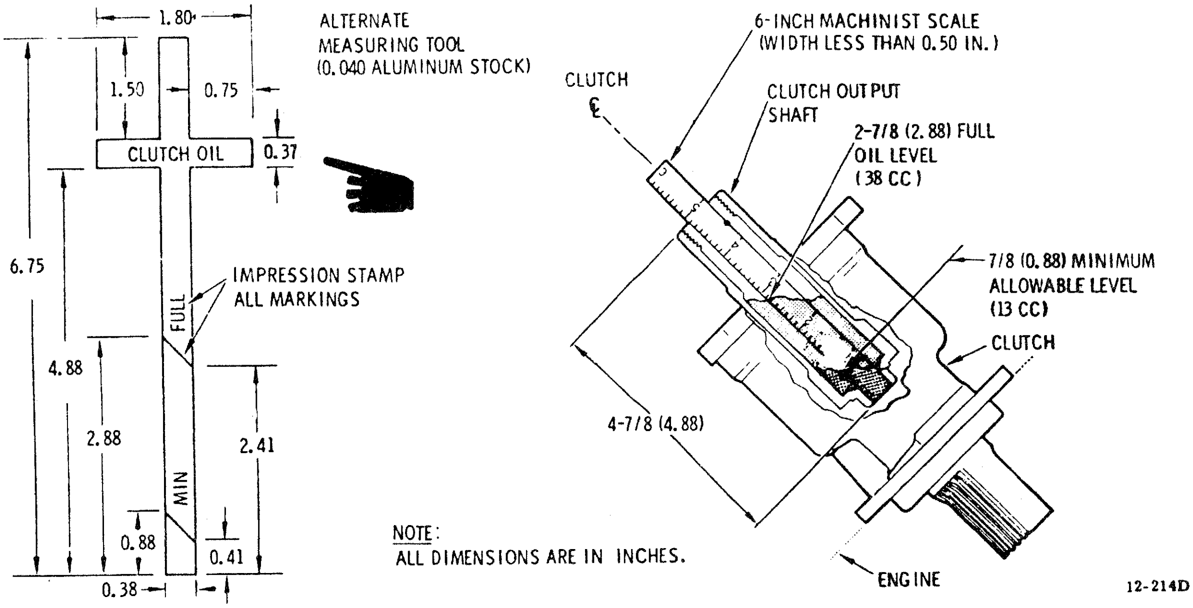


Figure 6-6 Overrunning Clutch Oil Level Check.

NOTE

To measure the clutch oil level as directed below, reduce the width of 1/2-inch scale, as required, to permit the end of the scale to bottom out in the clutch. (See fig. 6-6.) As an alternate, fabricate the measuring tool shown.

d. Slowly insert a CLEAN machinist's 6-inch scale (1/2 (0.50)-inch width) into the center of the clutch until it bottoms. Scale has reached the bottom of the clutch when it indicates 4-7/8 (4.88)-inches (fig. 6-6).

e. After the scale has been inserted 4-7/8 inches remove it and check the oil level reading. Repeat the measurement a minimum of three times.

(1) Full oil level (38 cc) reading will be 2-7/8 (2.88)-inches on the LOWER edge of the scale.

(2) Minimum allowable oil level (13 cc of trapped oil) reading will be 7/8 (0.88)-inch on the lower edge of the scale.

f. If oil level is less than the minimum acceptable quantity (13 cc) the clutch must be returned for overhaul and a serviceable unit installed.

g. Fill the clutch with lubricating oil (C67). Do not overfill. Recheck the oil level, *e* above, after servicing. (See fig. 6-6.)

CAUTION

When installing the clutch coupling bolt in *h* below, the INSTALLATION TORQUE ON THE BOLT MUST NOT BE LESS THAN 250 INCH-POUNDS. Torquing to a lower value will reduce clutch bearing clamp-up and possibly lead to bearing race spinning.

h. **SHIM COUPLING SO THAT THERE IS 0.035-0.055 INCH O-RING GAP** from end of clutch shaft to face of coupling recess (surface that bolt head contacts). Measure gap with feeler gage (fig. 6-5). Coat bolt threads with antiseize compound (C14). Coat clutch splines with lubricant (C63). Install shims(s), clutch coupling and bolt with O-ring. **DRAG TORQUE FOR SELF-LOCKING BOLT SERVICEABILITY IS 15 INCH-POUNDS MINIMUM, 150 INCH-POUNDS MAXIMUM. TORQUE BOLT TO ACTUAL DRAG TORQUE PLUS 250-300 INCH-POUNDS.**

i. Reinstall main transmission drive shaft (para 6-17).

6-30.1. Inspection-Overrunning Clutch Front Bearing.

6-30.2 Disassembly of Clutch Assembly.

a. Remove main transmission drive shaft, P/N 369A5510 (para. 6-14).

b. Remove retainer ring, P/N N5002-315, from housing.

c. Remove clutch subassembly, P/N 369A5350-11 (para. 6-31).

d. Carefully pry retainer, P/N 369A5366, loose and remove from shaft.

e. Use bearing puller, NSN 5120-00-924-7715, to remove bearing carrier, P/N 369A5355, along with bearing, P/N 369A5361, and outboard sleeve spacer, P/N 369A5376, from output shaft.

6-30.3. Inspection.

a. Using solvent, dry cleaning, (C94), and a soft bristle brush, clean bearing and allow to air dry.

NOTE

Thorough flushing of bearing and retainer of all existing grease is necessary to prevent possible mixing of dissimilar greases when bearing is repacked with Mobil 28.

b. Immediately apply a light coat of lubricant, corrosion inhibiting, (C61), to the bearing.

NOTE

Do not attempt to disassemble the P/N 369A5361 bearing.

c. Inspect the bearing, P/N 369A5361, for roughness of operation and/or any other signs of damage such as heat discoloration scoring, pitting, or flat spots.

NOTE

Do not substitute other greases for that specified.

d. If the bearing is found in satisfactory condition, reclean with solvent' redry and repack bearing to 50 percent capacity with Mobil 28 Grease (C-47A) only. If the bearing is unserviceable, it must be replaced.

6-30.4. Reassembly.

a. Check to see that seal groove in clutch retainer, P/N 369A5366 and 369A5367, has been thoroughly cleaned prior to installation.

b. Install new "0" ring packing, P/N M83248-1-026 and M83248-1-035, in retainer, P/N 369A5366, and in sleeve spacer, 369A5367. To aid installation of "O" ring, packing lubricate with petrolatum, technical, (C73), prior to installing.

CAUTION

To prevent damage to the clutch assembly, the grease packed side must face up on the assembly.

c. Press bearing, P/N 369A5361, into bearing carrier, P/N 369A5355, with grease packed side up.

d. Press carrier, P/N 369A5355, on clutch inner race seats against inner sleeve spacer, 369A5367.

e. Install outer sleeve spacer, P/N 369A5367, with flange against bearing, P/N 369A5361.

f. Fill void of retainer, P/N 369A5366, 50 percent full with Mobil 28 grease. Then install retainer on outer sleeve spacer, P/N 369A5367.

g. Carefully insert subassembly into clutch housing and secure with retainer ring, P/N N5002-315, using care to assure that the flat side of the retaining ring faces in and the chamfered side out on the clutch.

h. Service clutch and reinstall main transmission drive shaft (para 6-31).

6-31. Subassembly Replacement — Overruning Clutch. (See fig. 6-5.) If the clutch housing is serviceable, the internal subassembly of a defective overrunning clutch may be removed as a unit; then replaced with the internal subassembly of a serviceable clutch

without removing the housing from the engine.

a. Remove main transmission drive shaft (para 6-14).

b. Remove coupling bolt, O-ring, clutch coupling and coupling shim(s). Retain shim(s) for possible reuse. Reinstall coupling bolt and O-ring to prevent spillage of lubricating oil from clutch subassembly during final steps of removal.

c. Remove retainer ring from clutch housing.

d. Lift out clutch subassembly from housing. If subassembly does not have a data plate attached to clutch retainer, transcribe clutch serial number and all overhaul data from data plate on housing to a tag; then attach tag to internal components subassembly. (It is not necessary to fabricate data plate for subassembly being turned in for overhaul, however, tag containing necessary data must be attached to subassembly.)

e. Place defective clutch subassembly in new housing, from which serviceable replacement subassembly was obtained for shipment to overhaul facility.

f. Drain preservative oil from the replacement clutch subassembly. Add 38 cc (1.28 oz) of lubricating oil (C67). Temporarily install coupling bolt and O-ring to prevent spillage.

g. Coat lower clutch splines with lubricant (C63) and install new clutch subassembly into housing on engine.

h. Install retainer ring with bevelled side outward.

i. Remove coupling bolt and O-ring and install coupling shim(s) and clutch coupling. Coat coupling bolt threads with antiseize compound (C14) and reinstall coupling bolt with new O-ring. (Refer to step *h* of para 6-30).

j. Coat clutch upper splines with lubricant (C63).

k. Install main transmission drive shaft (para 6-17).

6-32. Removal — Overrunning Clutch (Complete Clutch Assembly). (See fig. 6-5.) *a.* Remove engine (chapter 4).

b. Remove the bolts, washers, nuts, and engine shaft firewall seal from clutch.

NOTE

Removal of the clutch coupling from the clutch in c below is unnecessary unless clutch is being replaced.

c. Remove coupling bolt, O-ring, clutch coupling, and coupling shim(s). Retain shim(s) with coupling for reuse. Reinstall bolt and packing to prevent spillage of lubricating oil from housing during final steps of removal.

d. Remove nuts and washers that secure overrunning clutch; remove clutch.

e. If a clutch is being replaced install a spare coupling bolt and O-ring or suitable plug in the output shaft (clutch inner race bore) to prevent contamination during clutch handling, shipping, or storage.

NOTE

The operating lubricant is an approved preservative for shipping or storage.

f. Wrap clutch in barrier material (C15), to protect splined areas of race shafts.

6-33. Repair — Overruning Clutch. No field repair except replacement of clutch subassembly (para 6-31) is allowable.

6-34. Installation — Overruning Clutch. (See fig. 6-5.) *a.* If clutch is new, remove tag, drain residual (trapped) preservative oil; then temporarily install coupling bolt and O-ring.

NOTE

Trapped oil can be removed by inverting the clutch a minimum of three times.

b. Coat clutch splines with lubricant (C63). Insert overrunning clutch outer race spline into engine and install six washers and nuts.

c. Remove coupling bolt and install coupling shim(s) and clutch coupling. Reinstall coupling bolt with new O-ring. (Refer to step *h* of para 6-30).

d. Install engine shaft firewall seal, three bolts, six washers, three nuts and tighten.

e. Reinstall engine (chapter 4).

SECTION IV MAIN TRANSMISSION

6-35. MAIN TRANSMISSION.

6-36. Description — Main Transmission. The main transmission (fig. 6-7) is mounted on the main rotor mast support structure. The transmission is basically a two-stage, speed reduction unit, utilizing the first reduction stage for the tail rotor drive system and accessory drive trains, and the second stage to further reduce rpm for the main rotor. All of the gears are spiral-bevel type, except for the accessory drive gears which are spur type. The transmission housing is magnesium alloy. The input pinion drives the input bevel gear which is concentrically mounted on the tail rotor output pinion. The output pinion simultaneously drives the output bevel gear, main rotor drive shaft, tail rotor drive shaft and accessory drive train. The accessory gear train drives (at 4200 rpm) both the rotor tachometer generator and the transmission oil pump which are mounted on drive pads at the back of the transmission. Transmission cooling is accomplished by a combination of the lubrication oil, air drawn over the housing by the oil cooler blower, and raw air routed to the gearbox by the two side ducts in the forward end of the air inlet fairing.

6-37. Inspection — Main Transmission. *a.* Remove the sound insulation and transmission access covers (chapter 2).

b. Check transmission for oil leaks, cracks, corrosion, secure electrical connections, and proper oil level. Evaluate leakage, if any, according to paragraph 6-104.

c. Check the four mounting flanges on main transmission housing for corrosion and cracks. Check that transmission mounting studs and nuts are secure. This should be accomplished from the fuselage interior as well as the exterior at the main rotor mast base.

d. (Refer to para 6-54, chip detector inspection.) Remove lockwire, electrical wire, and chip detectors from self-closing valves. Check for presence of foreign matter such as dirt and metal particles. If metal particles are present, drain oil and observe condition. If condition is questionable, check chip detectors and filter after 5 hours of flight. If still questionable, check electrical system for proper operation (chapter 9). Clean oil filter (para 6-45). If no metal particles are present, wipe detectors clean, and reinstall (fig. 6-2). Secure chip detectors with 0.032-inch lockwire (C57).

e. Check that all safety wiring is intact and secure.

f. Reinstall transmission access covers and sound insulation (chapter 2).

6-38. Removal — Main Transmission. (See fig. 6-7.) *a.* Remove sound insulation, transmission cover, main gearbox access door, and oil cooler access door (chapter 2).

b. Remove main transmission drive shaft (para 6-14).

c. Remove oil cooler blower (para 6-21).

d. Remove tail rotor drive shaft (para 6-72).

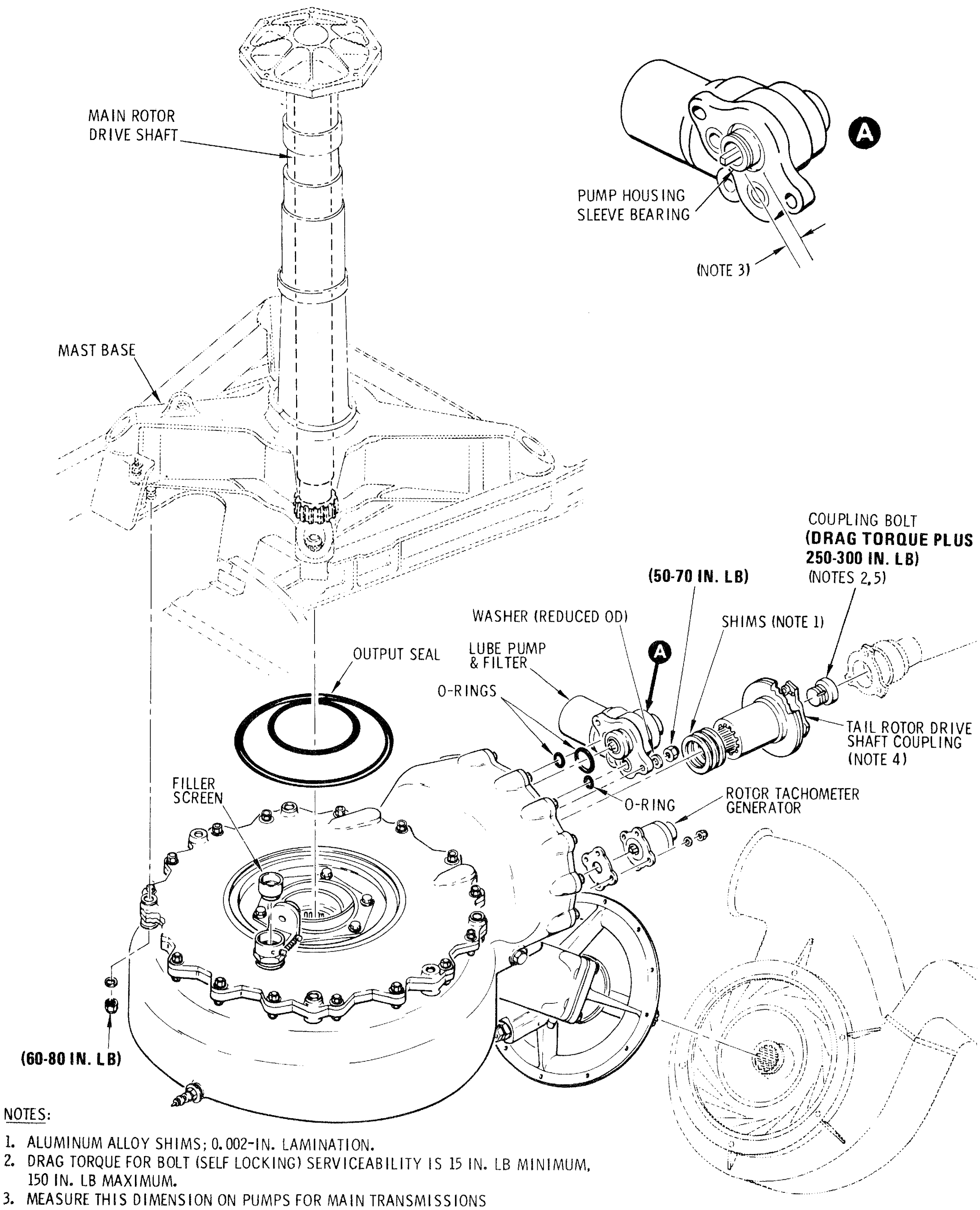
e. Disconnect wiring from rotor tachometer generator, two chip detectors, and oil pressure and temperature switches. Disconnect rotor tachometer generator bonding jumper (if installed).

f. Drain oil from transmission.

g. Have an assistant support the transmission; then remove the four mounting nuts and washers.

CAUTION

Lower the transmission with extreme care to prevent contact between the tail rotor drive coupling and the surrounding structure. Any dents, nicks or scratches on the coupling diaphragm requires replacement of the coupling. Use care to protect temperature sender and chip detector terminal studs from damage during handling



NOTES:

1. ALUMINUM ALLOY SHIMS; 0.002-IN. LAMINATION.
2. DRAG TORQUE FOR BOLT (SELF LOCKING) SERVICEABILITY IS 15 IN. LB MINIMUM, 150 IN. LB MAXIMUM.
3. MEASURE THIS DIMENSION ON PUMPS FOR MAIN TRANSMISSIONS (369A5100).DIMENSION ON REPLACING PUMP MUST NOT BE MORE THAN ON THE REMOVED PUMP.
4. COAT COUPLING SPLINES WITH LUBRICANT (C63) BEFORE ASSEMBLY.
5. COAT COUPLING BOLT THREADS WITH ANTISEIZE COMPOUND (C14) BEFORE ASSEMBLY.

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Figure 6-7. Main Transmission and/or Lube Pump Replacement.

and when placing transmission on any surface. Use suitable cover to prevent entry of contamination at main rotor drive shaft opening.

h. With help from assistant, carefully lower main transmission from mounting studs.

i. Keep output seal with transmission by taping or tying in place to prevent loss during handling, shipping or storage. Inspect seal for permanent buckling or dents that could permit water entry.

6-39. Disassembly — Main Transmision. (See fig. 6-7 and 6-8.) *a.* Remove tachometer generator (chapter 8).

b. Remove coupling bolt, tail rotor drive shaft coupling, and coupling shims. Retain shims with coupling for reuse.

c. Install suitable covers and plugs to protect seal ports, stud threads, and the tachometer generator pad.

NOTE

Further disassembly is not required. (Refer to para 6-56 for transmission drain tubing information.)

6-40. Repair — Main Transmission. Field repair of the main transmission is limited to replacement of electrical components, the breather-filler, externally accessible O-rings, oil pump and repair of surface damage. Housing external surface damage is repairable within the limits outlined as follows.

a. If the depth of a depression does not exceed 0.03 inch and the area is no greater than 0.6 square inch, repair according to steps *c*, *d* and *e*.

b. If the depth of a depression exceeds 0.03 inch but does not exceed 0.1 inch and the area is no greater than 0.6 square inch, repair according to steps *c* through *f*.

NOTE

Any evidence of lubricating oil leakage in a repaired area requires replacement of the transmission.

c. Clean area with solvent (C94), and remove sharp edges with grade 280 abrasive paper (C2).

d. Brush coat primer (C79) on the repaired surface.

e. Fill depression with epoxy adhesive (C7) and blend to surrounding surfaces.

f. Touch up the reworked area with paint (chapter 1).

6-41. Assembly — Main Transmission. (See fig. 6-7 and 6-8.)

NOTE

Refer to (para 6-17) for shimming the transmission input shaft coupling and to (para 6-75) for shimming the tail rotor drive shaft coupling after the transmission is installed.

a. Install one preliminary 0.010-inch coupling shim on transmission output (tail rotor drive) gear shaft to prevent coupling from bottoming against end of gear shaft.

b. Coat coupling splines with lubricant (C-63) and coupling bolt threads with antiseize compound (C-14) before a ssembly. Install tail rotor drive shaft coupling and coupling bolt. DRAG TORQUE FOR COUPLING BOLT SERVICEABILITY IS 15 INCH-POUNDS MINIMUM, 150 INCH-POUNDS MAXIMUM. TORQUE COUPLING BOLT TO ACTUAL DRAG TORQUE PLUS 250-300 INCH POUNDS.

c. Install tachometer generator (chapter 8).

d. Install transmission input gear shaft seal drain tube if not already in place (para 6-61).

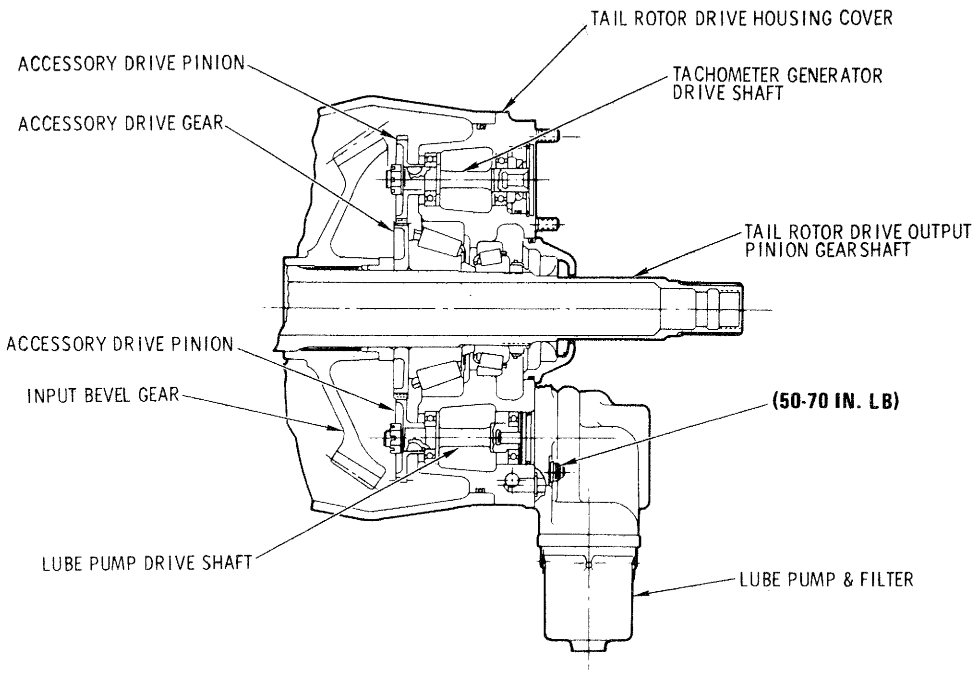
6-42. Installation — Main Transmission. (See fig. 6-7.) *a.* Accomplish build-up described in paragraph 6-41 on replacement transmissions.

NOTE

Transmission thermoswitch (oil temperature sender) may be found taped to shroud mount. If so, remove plug from thermoswitch bore, apply a light coat of antiseize compound (C14) to the switch threads, install thermoswitch and **TORQUE SWITCH TO 12-25 INCH-POUNDS.**

b. Apply petrolatum (C73) to outer edge of output seal. Press seal firmly into recess in top of transmission housing.

c. With help from an assistant, carefully lift and position transmission and blower in place on mounting studs.



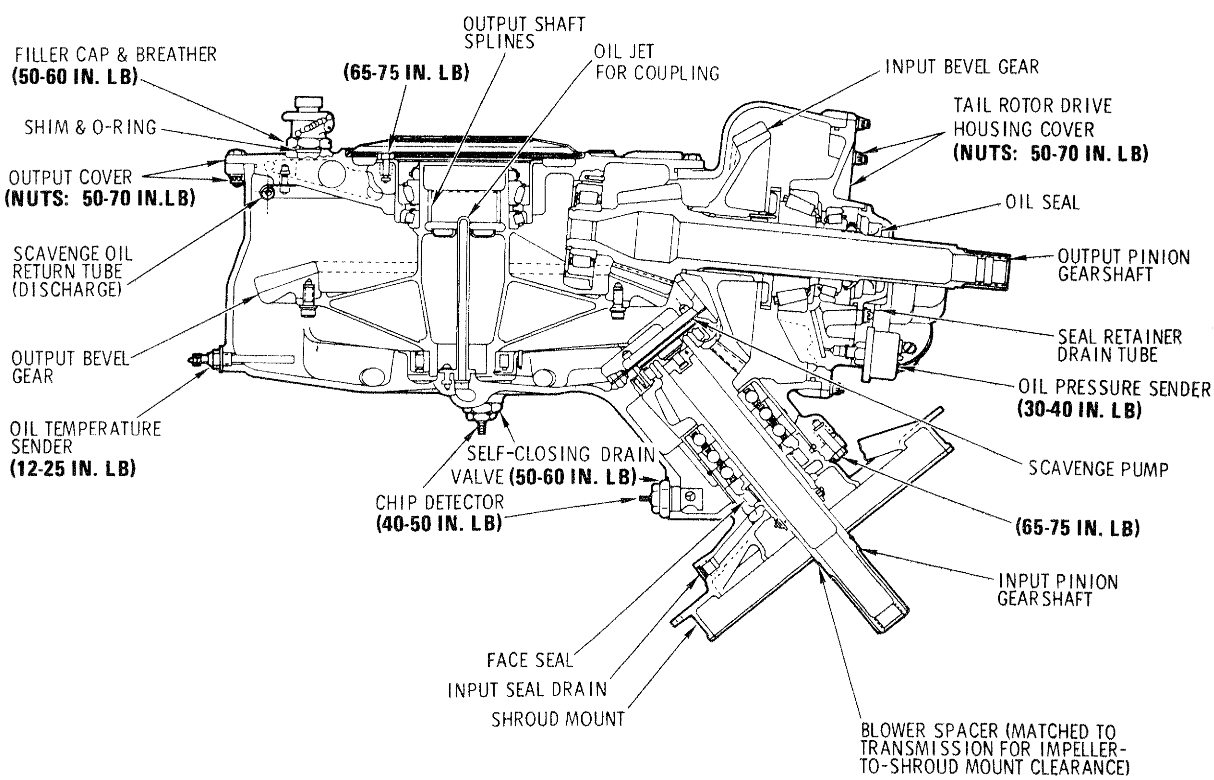


Figure 6-8. Main Transmission Cross-Section Views.

CAUTION

If main rotor drive shaft is installed, make sure that drive shaft gear coupling is properly meshed before installing transmission mounting nuts.

d. Install four washers and nuts; **TORQUE NUTS TO 60-80 INCH-POUNDS.**

e. Connect wiring to tachometer generator, two chip detectors, and oil pressure and temperature senders. Connect tachometer generator bonding jumper (if installed).

f. Service transmission with lubricating oil (chapter 1).

g. Install tail rotor drive shaft (para 6-75).

h. Install oil cooler blower (para 6-26).

i. Install main transmission drive shaft (para 6-17).

j. Ground run aircraft according to TM 55-1520-214-10 and inspect main transmission for correct warning light operation and oil leaks. (Refer to para 6-104 for oil leakage limits and table 6-1 for troubleshooting procedures.)

k. Install transmission cover, oil cooler access door, sound insulation, and main gearbox access door (chapter 2).

6-43. MAIN TRANSMISSION LUBRICATION PUMP.

6-44. Description — Main Transmission Lubrication Pump. The main transmission has a self-contained oil system. A lubrication pump (fig. 6-9) draws oil from the main sump of the transmission housing and provides oil flow to the oil pressure sender, bearings, gears and the output gear coupling for the main rotor drive shaft. The lube pump is a positive-displacement, vane-type with a non-adjustable discharge pressure of 25 to 50 psi. It is driven at 4200 rpm by the accessory drive gear attached to the tail rotor output gear shaft. A relief valve that cracks open above 60 psi, with full-flow bypass at 75 psi maximum (369A5100-601 and -603 transmissions), or 100 psi maximum (369A5100-605 and subsequent transmissions) and a filter are integral parts of the lubrication pump housing assembly. A vane-type scavenge pump, with a discharge pressure of 10 to 20 psi, is mounted on the upper end of the transmission input pinion. The scavenge pump draws oil from the input gear shaft sump and pumps the oil to a discharge tube mounted inside the top of the transmission housing. The tube is perforated to direct oil against the interior sides of the transmission housing to produce cooling as the oil drains down to the main sump. Externally mounted oil system accessories include a combined breather-filler, a sight type liquid level plug, two chip detectors, and oil pressure and temperature warning senders (fig. 6-2). Refer to chapter 1 for transmission servicing information.

6-45. Inspection — Main Transmission Lubrication Pump Oil Filter. *a.* Remove, in order, sound insulation, gearbox access cover, transmission drain assembly, and main transmission cover. Refer to chapter 2.

b. Remove lockwire from lubrication pump filter housing (fig. 6-9).

c. Position a container or cloth to catch residual oil. Loosen and remove filter housing by turning it counterclockwise.

d. Remove filter element, and element preformed packing (O-ring) from pump housing. Discard O-ring.

NOTE

There are two configurations of the transmission lubrication pump oil filter. If the part number is 369A5264 a spring tension washer is required.

e. Remove spring washer and housing O-ring. Discard O-ring.

f. Check filter element for metal particles. If metal particles are present, remove main transmission chip detectors (para 6-54) and inspect for other evidence of internal failure in the gearbox.

NOTE

Use ultrasonic cleaning equipment if available.

g. Clean the filter element and housing with solvent (C94). Agitate element in solvent until solvent is clear and then let element air dry.

h. Check element for tears, cracks or dents that would make it unserviceable. Replace element if condition is questionable or if filter end flanges are cracked. Install new O-rings.

i. Lubricate replacement O-rings with transmission oil and install on end of filter and on filter housing.

j. Install tension washer in filter housing and install element as shown in figure 6-9. Check that the O-ring on the element will seat properly in the pump housing when element is installed.

k. Turn filter housing clockwise and tighten securely. Safety the filter housing to the pump housing using 0.032-inch lockwire (C57).

l. Replenish transmission oil supply (chapter 1) as necessary; then perform ground runup of aircraft and check joint line for oil leakage.

m. Reinstall, in order, the main transmission cover, the drain assembly, the main gearbox access cover and sound insulation.

6-46. Removal — Main Transmission Lubrication Pump. (See fig. 6-7 and 6-8.)

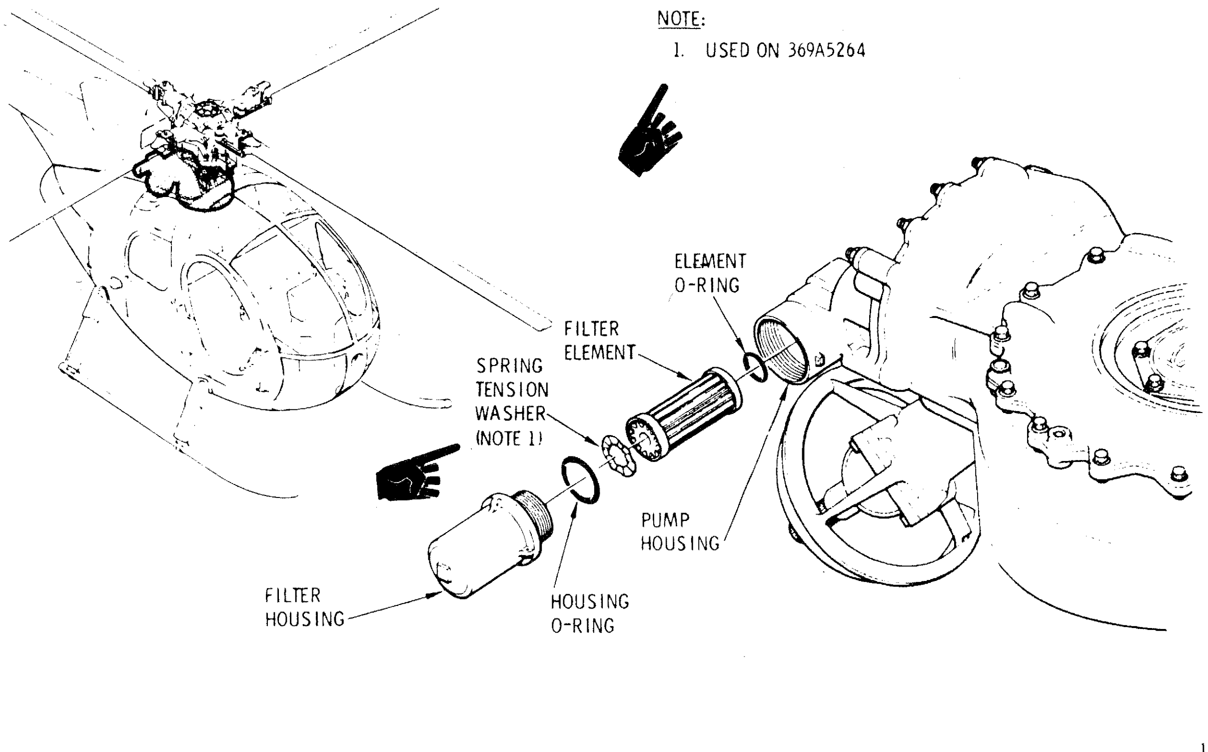


Figure 6-9. Transmission Lubrication Pump Oil Filter.

a. Remove sound insulation, transmission cover, and main gearbox access door (chapter 2).

b. Remove three nuts and washers that secure pump and filter to transmission housing accessory cover.

c. Remove pump and discard the three O-rings. Plug inlet and discharge ports in pump housing, and the drive bore in housing to keep out foreign matter.

CAUTION

Main transmissions (369A5100) incorporate bearing shims on the lubrication pump drive shaft to control end play. If the sleeve bearing of a replacement pump is longer than on the removed pump the pump drive shaft bearings can be preloaded to cause bearing failure. If transmission is identified with (369A5100-601 or subsequent dash numbers) it has been modified and the special measurement is not required when changing the pump.

Main transmissions (369A5100-605) incorporate a high capacity oil system. Only oil pump (369A5264) must be used for replacement.

d. Check part number of transmission on the identification plate next to liquid level plug. If part number is 369A5100, measure the distance that the sleeve bearing extends from the mounting face of the pump housing. (See detail A, fig. 6-7.) Make the same measurement on the replacement pump. The dimension on the new pump must not be more than on the removed pump.

NOTE

If the sleeve bearing dimension is greater than that of the old pump, return the replacement pump to supply and request a different serial number. If three attempts fail to produce a suitable pump, replace the main transmission.

6-47. Inspection — Main Transmission Lubrication Pump (General). *a.* Inspect for external leaks around mounting pad and ports, and for cracks or corrosion.

b. Check that all lockwire is intact.

c. Check senders, tachometer, and electrical connections for secure attachment.

6-48. Installation — Main Transmission Lubrication Pump. (See fig. 6-7 and 6-8.) *a.* Remove plugs from pressure ports in pump and transmission.

b. Install new O-rings in port and drive seal recesses of pump. Apply a light coating of petrolatum (C73) to both the O-ring and mating bore in transmission to prevent damage to O-ring.

c. Align square drive of pump with pump drive shaft in transmission and carefully press pump into place on mating surface.

d. Install three washers and nuts; **TORQUE NUTS TO 50-70 INCH-POUNDS.**

e. Ground run aircraft according to TM 55-1520-214-10 and inspect pump parting surfaces for leaks. Check pressure warning light for proper operation. If leaks in excess of limits described in paragraph 6-104 are detected, remove pump and replace O-rings. Recheck for leaks. Continued excessive leakage requires pump replacement.

f. Install transmission cover, main gearbox access door, and sound insulation (chapter 2).

6-49. MAIN TRANSMISSION EXTERNAL COMPONENT REPAIRS.

6-50. Liquid Level (Sight) Plug — Main Transmission External Components. (See fig. 6-2.) *a.* Drain oil from transmission until oil level is well below edge of sight plug and port.

b. Remove lockwire from sight plug.

c. Remove sight plug by unscrewing.

d. Remove and inspect O-ring. Install new O-ring on sight plug, if required.

NOTE

When a new sight plug is being installed, the liquid level markings are added f below after the plug is installed and torqued. Coat new sight plugs that are not identified with an "X" after the part number with silicone primer to prevent clouding of sight glass. Refer to g below for coating application.

e. Install sight plug. **TORQUE SIGHT PLUG TO 80-90 INCH-POUNDS**, and install lockwire.

f. Oil level markings may be applied on newly installed sight plugs as follows:

(1) Using a machinists level and scale, scribe an ADD line horizontally on both outer faces of the sight plug as shown in detail A. Do NOT mark the window.

(2) Scribe the FULL line horizontally on both outer edges of the sight plug as shown in detail A. Do NOT mark the window.

CAUTION

Application of any cleaning material other than soap and water to the sight plug window may cause it to craze.

(3) Using white paint (C55), paint four lines with the edges touching the scribe marks as shown in detail A. Do NOT paint the surface of the window.

g. A new P/N S51H sight plug may be coated as follows:

(1) Fill a clean container with silicone primer (C81) deep enough to cover the sight plug.

(2) Dip plug in primer long enough to coat inner face of window. Allow to air-dry.

(3) Reidentify plug by adding an "X" after the part number.

h. Apply decal markings to the liquid level plug as follows:

NOTE

The decal marker (fig. 6-2) may be installed on the plug prior to installation of the plug in the transmission since rotation of the bulls-eye type decal does not affect readability.

(1) Clean the exterior surface of the plug window with naptha (C70).

(2) Place the decal, backside down, on a cloth or sponge saturated with water. Allow the cellophane backing on the decal to soften for two to five minutes.

(3) Peel cellophane backing from decal using care to avoid touching adhesive back or tearing the decal.

(4) Place decal on plug window with adhesive side down and press out all air bubbles.

6-51. Chip Detectors — Main Transmission External Components. Refer to main transmission draining (chapter 1) for removal and installation instructions. Refer to chapter 9 for functional test of detector circuit and to paragraph 6-54 for inspection.

6-52. Rotor Tachometer Generator — Main Transmission External Components. Refer to chapter 8.

6-53. Oil Pressure and Temperature Senders — Main Transmission External Components. (See fig. 6-2.)

NOTE

When oil temperature sender is to be replaced, drain oil from transmission.

a. Cut lockwire from temperature sender.

b. Remove electrical wire from terminal post.

c. Unscrew sender (fig. 6-2). Catch any residual oil from the sender port in an absorbent cloth.

d. Apply antiseize compound (C28) sparingly to only the threads of the replacement sender.

e. Install oil pressure sender; **TORQUE TO 30-40 INCH-POUNDS.**

f. Install oil temperature sender; **TORQUE TO 12-25 INCH-POUNDS.**

g. Lockwire the temperature sender hex to adjacent lug on transmission housing using 0.032-inch lockwire (C-57).

h. Connect electrical wire to terminal post.

i. Service transmission (chapter 1).

6-54. Chip Detector (Caution Light ON) Inspection — Main Transmission External Components.

NOTE

Refer to the chapter 1 special inspection requirements before proceeding with the following steps.

Remove and inspect chip detector magnetic plugs and oil filter. Steel fuzz characterized by fine hair-like particles is the result of normal wear and does not indicate a problem.

a. **IF GRANULAR STEEL OR BRASS PARTICLES ARE PRESENT, REPLACE THE TRANSMISSION.** These particles usually indicate an internal failure.

b. **IF STEEL SPLINTERS OR FLAKES EXCEEDING 1/16 (0.062) × 3/16 (0.187) INCH ARE PRESENT, REPLACE THE TRANSMISSION.**

c. **IF MORE THAN 10 STEEL OR ALUMINUM FLAKES OR GRANULAR PARTICLES OF APPROXIMATELY 1/16 × 1/16 INCH SIZE ARE PRESENT, REPLACE THE TRANSMISSION.**

d. If the particles found are smaller or fewer in number than the limit in *c* above, clean the main transmission filter and reinstall, **CHECK THE CHIP DETECTORS AGAIN AFTER FIVE (5) HOURS OF FLIGHT;** if more particles are found, replace the transmission.

6-55. MAIN TRANSMISSION DRAINS AND TUBING.

6-56. Description — Main Transmission Drains and Tubing. The transmission drain installation (fig. 6-10) consists of a drain line from the output pinion (tail rotor) gearshaft seal retainer, one from the main transmission yoke type drain assembly (input pinion seal drain) and, if the oil cooler blower scroll is fitted with a drain port, one from the blower scroll. The drains merge through a two- or three-port manifold fitting in the forward firewall, and into a single drain tube that is routed down the engine compartment bulkhead and overhead.

6-57. Inspection — Main Transmission Drains and Tubing (Installed). *a.* Inspect flexible tubing for unserviceable wear, cuts or breaks.

b. Inspect grommets, chafe strip, nylon straps and lockwire for secure installation. (Refer to chapter 2 for additional information on the main transmission drain assembly.)

6-58. Removal — Main Transmission Drains and Tubing. (See fig. 6-10.) *a.* If a scroll drain tube is installed, use steps *b* through *d* to remove scroll drain tube from scroll.

b. Detach transmission drain tubes from the firewall draining fitting.

c. If not already removed, detach drain tubes from the transmission drain assembly.

d. Release supporting nylon straps and remove drain tubes.

e. Remove attaching clamp on engine compartment lower firewall drain tube and remove drain tube.

NOTE

Remove main transmission input pinion flex-drain and bushing (detail A) whenever main transmission is replaced. The flex-drain and bushing are not normally part of a spare transmission assembly.

f. Grasp the flex-drain and pull out to release the bond. Insert a 5/16(0.312)- or 3/8(0.375)-inch tap or easy-out into tube ID and rotate to pull out bushing.

6-59. Inspection — Main Transmission Drains and Tubing (Removed). Check transmission yoke-type drain assembly and tubing for serviceability, and repair as necessary.

6-60. Repair — Main Transmission Drains and Tubing. *a.* Repair cracks in the yoke-type drain assembly according to repair instructions for polycarbonate plastic in chapter 2.

b. Repair deteriorated or otherwise damaged mounting gasket segments on the yoke-type plastic drain assembly by replacing with new gasket material (C44).

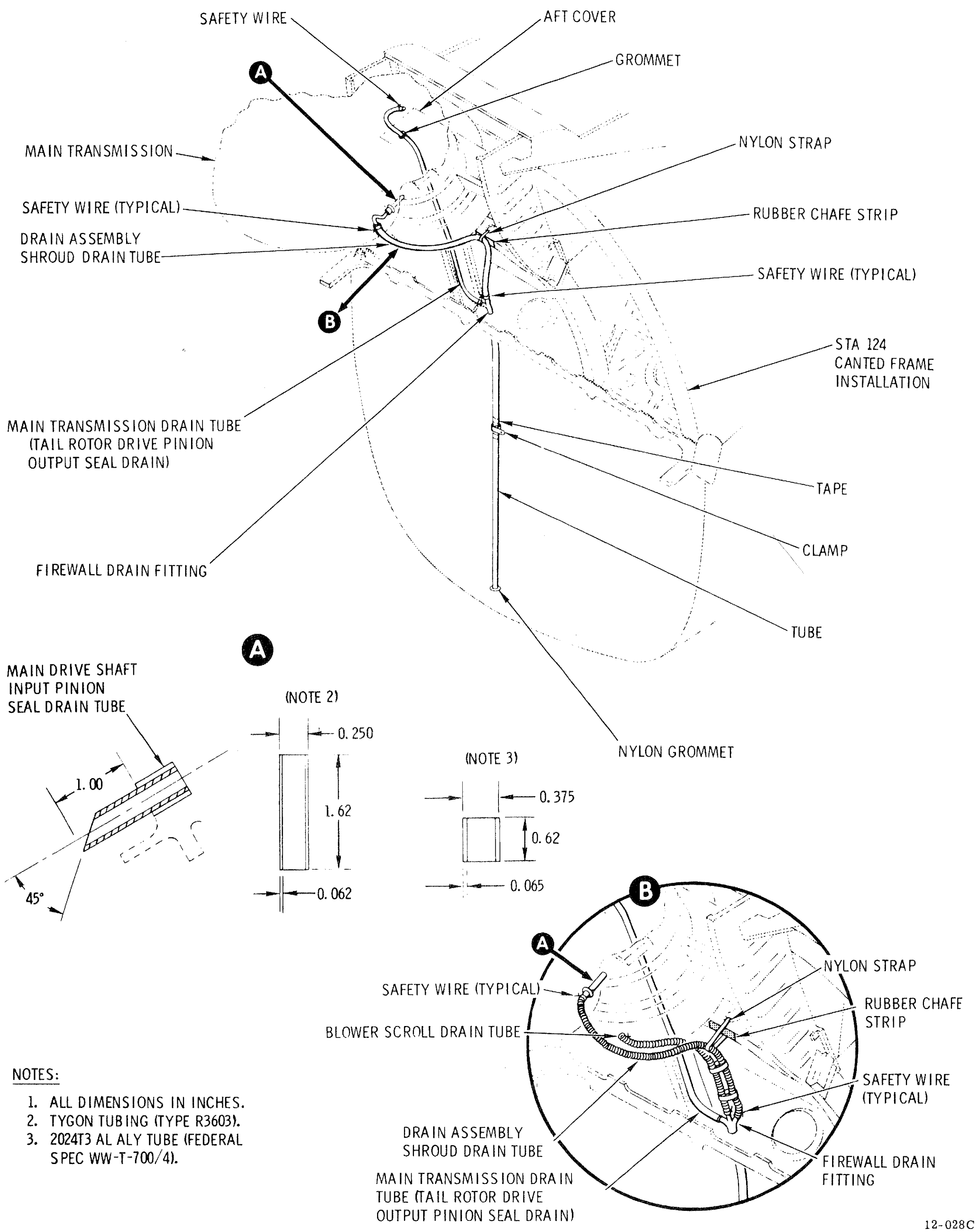


Figure 6-10. Main Transmission Drain Tubing Installation.

c. Repair damaged flexible drain tubing with tape (C18).

6-61. Installation — Main Transmission Drains and Tubing. *a.* If a scroll drain is installed use steps *e* through *g* to install drain tube. For initial installations, replace the two-way firewall fitting with a three-way fitting.

b. On a new main transmission installation, clean mating surface of flex-tube, bushing, and transmission drain outlet with solvent (C94).

NOTE

A replacement main transmission flex-tube and bushing may be fabricated using materials and dimensions shown in detail A, figure 6-10.

c. Bond flex-tube to bushing and bushing to main transmission with sealing compound (C89), according to container instructions.

d. Remove excess sealing compound and scarf flex-tube at 45 degrees (detail A, fig. 6-10).

e. Position transmission drain assembly and install bolts. Ensure that main transmission flex-drain will drain directly into mating drain assembly port.

f. Attach upper ends of drain tubing to transmission drain assembly; secure with double wraps of lockwire.

g. Route drain tubing and secure with nylon straps. Position tubing so that no drainage traps are formed.

h. Secure lower ends of drain tubing to firewall fitting with double wraps of lockwire.

i. Position engine compartment firewall drain tube; wrap tube with one layer of pressure-sensitive polyurethane tape (C105) at clamp-attach point, and install clamp.

6-62. MAIN ROTOR MAST.

6-63. Description — Main Rotor Mast. The main rotor mast (fig. 6-11) is a machined steel tube joined with a forged aluminum alloy base having four legs. The tube and mast base are fitted by differential temperature, and are secured together as a one-piece assembly by the interference fit and four hi-shear rivets. The mast assembly base supports the main transmission and the mounting bracket for cyclic and collective mixer control bellcranks. The mast assembly tube provides the structure on which the main rotor hub is mounted, the axis for stationary swashplate position and main rotor hub rotation, and the housing for the main rotor drive shaft.

6-64. Inspection — Main Rotor Mast (Installed). *a.* Check all visible areas of mast for cracks, nicks, scratches, corrosion and evidence of impact damage. Figure 6-12 contains inspection and repair criteria.

b. Refer to chapter 2 for 300-hour inspection requirement.

6-65. Removal — Main Rotor Mast.

CAUTION

The main rotor mast is a highly stressed part. Do not allow tools to strike the mast, or the mast to strike any object. Any impact damage may require replacement of the mast.

a. Remove main rotor hub (chapter 5).

b. Remove main rotor controls (chapter 11).

c. Remove main transmission (para 6-38). Transmission removal is required for access to mounting studs.

NOTE

The nuts securing the studs have left-hand threads.

d. (See fig. 6-11.) Remove four mast base nuts, eight washers, and four bolts.

e. Remove the four main transmission mounting studs, washers, and nuts.

f. Lift main rotor mast from supporting structure.

6-66. Inspection — Main Rotor Mast (Removed). (See fig. 6-12.) *a.* Check all areas of main rotor mast for cracks, nicks, scratches, and evidence of impact damage. Check hub bearing surfaces for scoring and galling.

b. Check threads and serrations for damage.

c. Check for four rivets that secure the mast base to the mast tube for security.

d. Check that the tape on forward edge of the mast tube is secure and undamaged. Replace with tape (C101). Tape is installed on the front 180 degrees of lower mast tube, 5.50 inches up from the mast base. (See fig. 6-12.) Surface should be cleaned with naphtha (C70) before installation of new tape.

e. Check mast base for cracks, nicks, scratches and corrosion

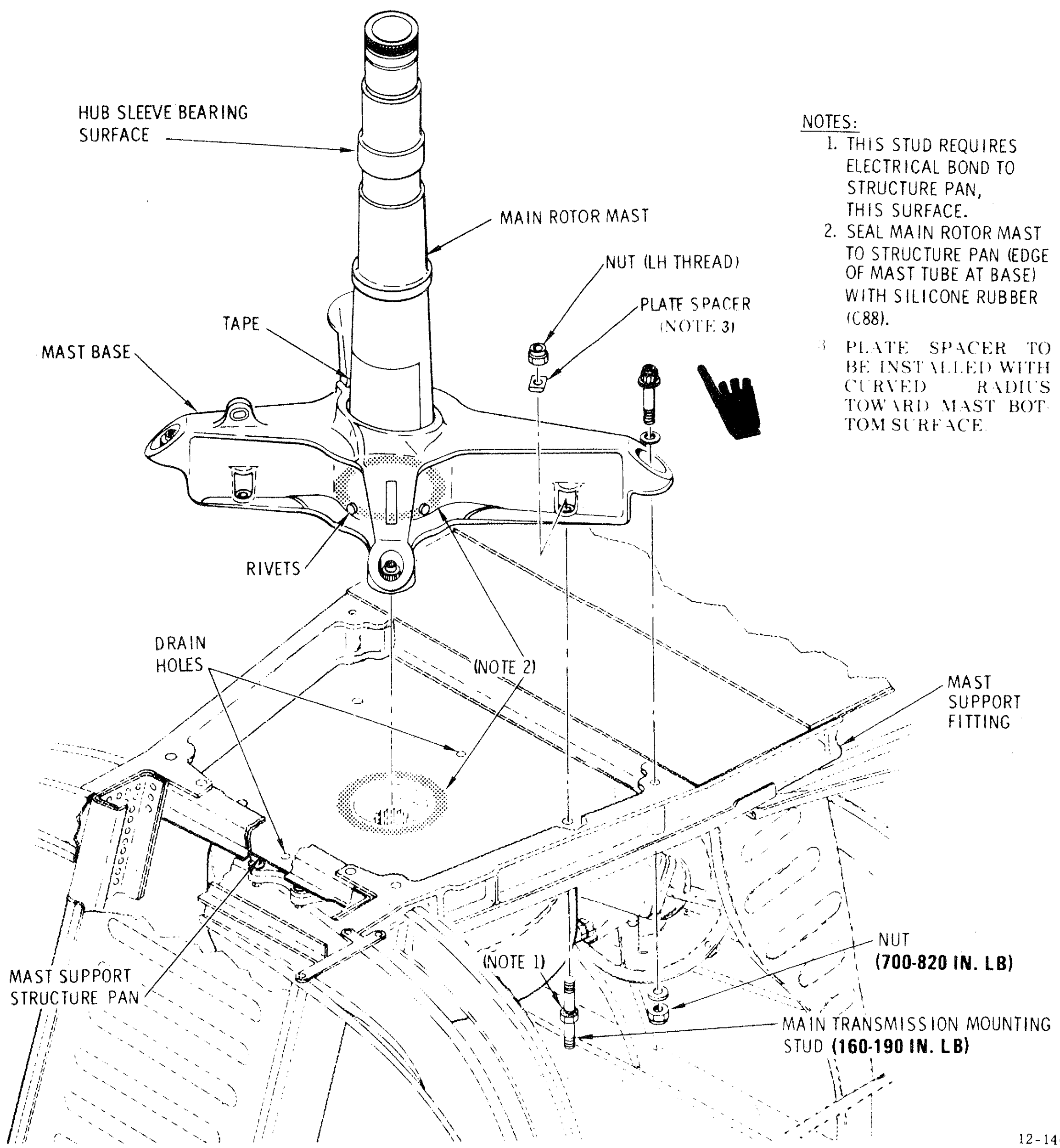


Figure 6-11. Main Rotor Mast Installation.

6-67. Repair — Main Rotor Mast. See figure 6-12 for inspection and repair criteria for the mast tube. Refer to chapter 2 for permissible repair of mast base.

6-68. Installation — Main Rotor Mast. (See fig. 6-11.) *a.* Scrape off any sealant residue and apply silicone rubber (C88) on inside base edge of main rotor mast tube. (This will seal base to mast support structure and prevent entry of water or leakage of oil.)

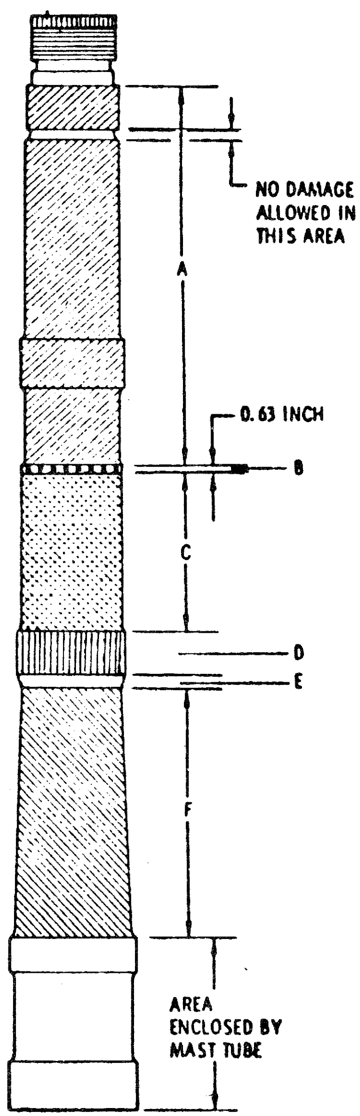
b. Position main rotor mast so that holes in base align with holes in mast support structure.

c. Install the four holddown bolts, eight washers, and four nuts. **TORQUE NUTS TO 700-820 INCH-POUNDS.**

d. Check underside of mast support structure at the left aft stud hole location. The stud-to-pan doubler surface must be clean to bear metal for electrical bonding. Install four transmission mounting studs, plate spacers, with radius side down, and nuts. **TORQUE STUDS TO 160-190 INCH-POUNDS.**

e. Using a 0.001-inch feeler gauge, check for gap between self-locking nuts and plate spacers. No gap is allowed.

f. If gap is observed, remove nut and replace with new self-locking nut; this applies to all nuts that are not flush against the mating spacer. **TORQUE STUDS TO 160-190 INCH-POUNDS** and repeat inspection, step



AREA	MAXIMUM SERVICEABLE LIMITS	CORRECTIVE ACTION	MAXIMUM REPAIRABLE LIMITS	CORRECTIVE ACTION
A	0.010 INCH 0.125 INCH WIDE CORROSION & SCRATCHES	POLISH AREA SMOOTH WITH ABRASIVE CLOTH (ITEMS 48 & 73, TABLE 2-2) IF CAD PLATING HAS BEEN PENETRATED TREAT REWORK AREA WITH PRIMER (ITEM 1)	0.125 INCH TO 1/3 CIRCUMFERENCE 0.010 INCH DEEP CORROSION & SCRATCHES	CAD PLATE PER HP4-113 (QQ-P-416) THEN PRIME WITH (ITEM 1)
B	0.0035 INCH DEEP 0.125 INCH WIDE CHIPPING OF NICKLE PLATE	POLISH AREA SMOOTH WITH ABRASIVE CLOTH (ITEMS 48 & 73, TABLE 2-2) TREAT REWORKED AERA WITH PRIMER (ITEM 1)	0.125 INCH TO 1/3 CIRCUMFERENCE 0.010 INCH DEEP CHIPPING OF NICKLE PLATE	NICKLE PLATE PER HP4-113 (QQ-N-290) THEN PRIME WITH (ITEM 1)
C	0.0035 INCH DEEP 0.125 INCH WIDE CHIPPING OF NICKLE PLATE	POLISH AREA WITH ABRASIVE CLOTH (ITEMS 48 & 73, TABLE 2-2) TO REMOVE BURRS AND SHARP EDGES ONLY.	0.125 INCH TO 1/3 CIRCUMFERENCE 0.010 INCH DEEP CHIPPING OF NICKLE PLATE	NICKLE PLATE PER HP4-113 (QQ-N 290)
D	0.020 INCH DEEP 0.125 INCH WIDE CORROSION & SCRATCHES	POLISH AREA SMOOTH WITH ABRASIVE CLOTH (ITEMS 48 & 73, TABLE 2-2)	0.020 INCH DEEP 0.125 INCH TO 1/3 CIRCUMFERENCE CORROSION & SCRATCHES	CAD PLATE PER HP4-113 (QQ-P-416)
E	0.010 INCH DEEP 0.125 INCH WIDE CORROSION & SCRATCHES	POLISH AREA SMOOTH WITH ABRASIVE CLOTH (ITEMS 48 & 73, TABLE 2-2) TREAT REWORKED AREA WITH PRIMER (ITEM 17) AND TOP COAT OF LACQUER (ITEM 50)	0.010 INCH DEEP 0.125 INCH TO 1/3 CIRCUMFERENCE CORROSION & SCRATCHES	CAD PLATE PER HP4-113 (QQ-P-416) THEN PRIME WITH (ITEM 17) AND TOP COAT OF LACQUER (ITEM 50)
F	0.010 INCH DEEP 0.125 INCH WIDE CORROSION & SCRATCHES	REMOVE DAMAGE OR DETERIORATED TAPE POLISH CORROSION SPOTS WITH ABRASIVE CLOTH (ITEMS 48&73, TABLE 2-2) TREAT REWORKED SURFACES WITH PRIMER (ITEMS 17) AND A TOP COAT OF LACQUER (ITEM 50) REPLACE TAPE (ITEM 68)	0.010 INCH DEEP 0.125 INCH TO 1/3 CIRCUMFERENCE CORROSION & SCRATCHES	CAD PLATE PER HP4-113 (QQ-P-416) TREAT REWORKED SURFACES WITH PRIMER (ITEM 17) AND A TOP COAT OF LACQUER (ITEM 50) REPLACE TAPE (ITEM 68)

Figure 6-12 Main Rotor Mast Inspection and Repair Criteria

SECTION V TAIL ROTOR DRIVE SHAFT

6-69. TAIL ROTOR DRIVE SHAFT.

6-70. Description — Tail Rotor Drive Shaft. The tail rotor drive shaft (fig. 6-13) interconnects the main transmission and the tail rotor (transmission) gearbox. The shaft is a dynamically balanced and positively damped aluminum alloy tube, approximately 13 feet long, that rotates at 2045 rpm (100% N2). Identical mounting flanges, riveted to each tapered end of the tube, connect the shaft to the flexible joint couplings on the transmissions. A damper located near the center of the shaft maintains a minimum vibration level in the tail rotor drive system.

6-71. Inspection — Tail Rotor Drive Shaft (Installed). *a.* Open access doors on the air inlet fairing and tailboom. (Refer to chapter 2.)

b. (See fig. 6-13.) Check visible portions of drive shaft for dents, scratches, cracks, corrosion, evidence of torsional buckling or shaft bending, or bulkhead interference. Check for 0.25-inch minimum clearance around drive shaft at shaft fairing opening (station 137.50) while manually rotating drive system. Less than minimum clearance is a possible indication of shaft distortion. Scratches in the shaft section that passes through the plenum chamber fairing tube can indicate possible contact with the tube edges at the bulkhead openings.

c. Check that coupling attachment at main transmission is secure.

d. Check for evidence of shaft damper sleeve bond failure (sleeve shifting) from excessive heat or loads.

e. Close all access doors.

6-72. Removal — Tail Rotor Driveshaft. (See figure 6-13.) *a.* Remove (or open) tail rotor driveshaft access doors.

b. Remove three bolts and washers that secure shaft to output gearshaft coupling on main transmission.

c. Disconnect chip detector wiring from tail rotor transmission.

d. Detach station 282 bellcrank from tail rotor transmission.

e. Remove lockwire, four bolts, and washers that attach tail rotor gearbox to tailboom.

f. Remove mount bolts one at a time and install workaid bolts (P/N AN4H77A) (fig. 6-13A).

WARNING

Application of additional loads to tail rotor transmission in extended position could cause damage to gearbox mounting frame or nutplate connection.

g. Carefully slide tail rotor gearbox to outboard end of workaid bolts.

CAUTION

While removing the driveshaft from the tail rotor gearbox coupling, the shaft will rest on the bulkhead forward of station 142 bellcrank. A cushion material of sort will stop the shaft from rubbing the bulkhead. Remove after use.

h. While holding the output shaft, remove three bolts and washers that attach driveshaft to tail rotor gearbox input coupling.

i. Slide driveshaft forward and install workaid (fig. 6-13B) to tail rotor input coupling using driveshaft attaching bolts (P/N NAS464P4A4).

j. With workaid (fig. 6-13A) remove the hex bolt holding the driveshaft coupling to the tail rotor gearbox.

k. Remove workaid bolts and tail rotor gearbox from tailboom.

l. With aid from an assistant, remove driveshaft from tailboom.

6-73. Inspection — Tail Rotor Drive Shaft (Removed). *a.* When there has been abnormal shaft vibration, check damper friction (para 6-84) and shaft straightness or runout. When rotated between centers, or with ends plugged and rolled over-rollers, **THE RUN-OUT OF THE LARGE DIAMETER MUST NOT EXCEED 0.060-INCH TIR** (permissible dents excluded).

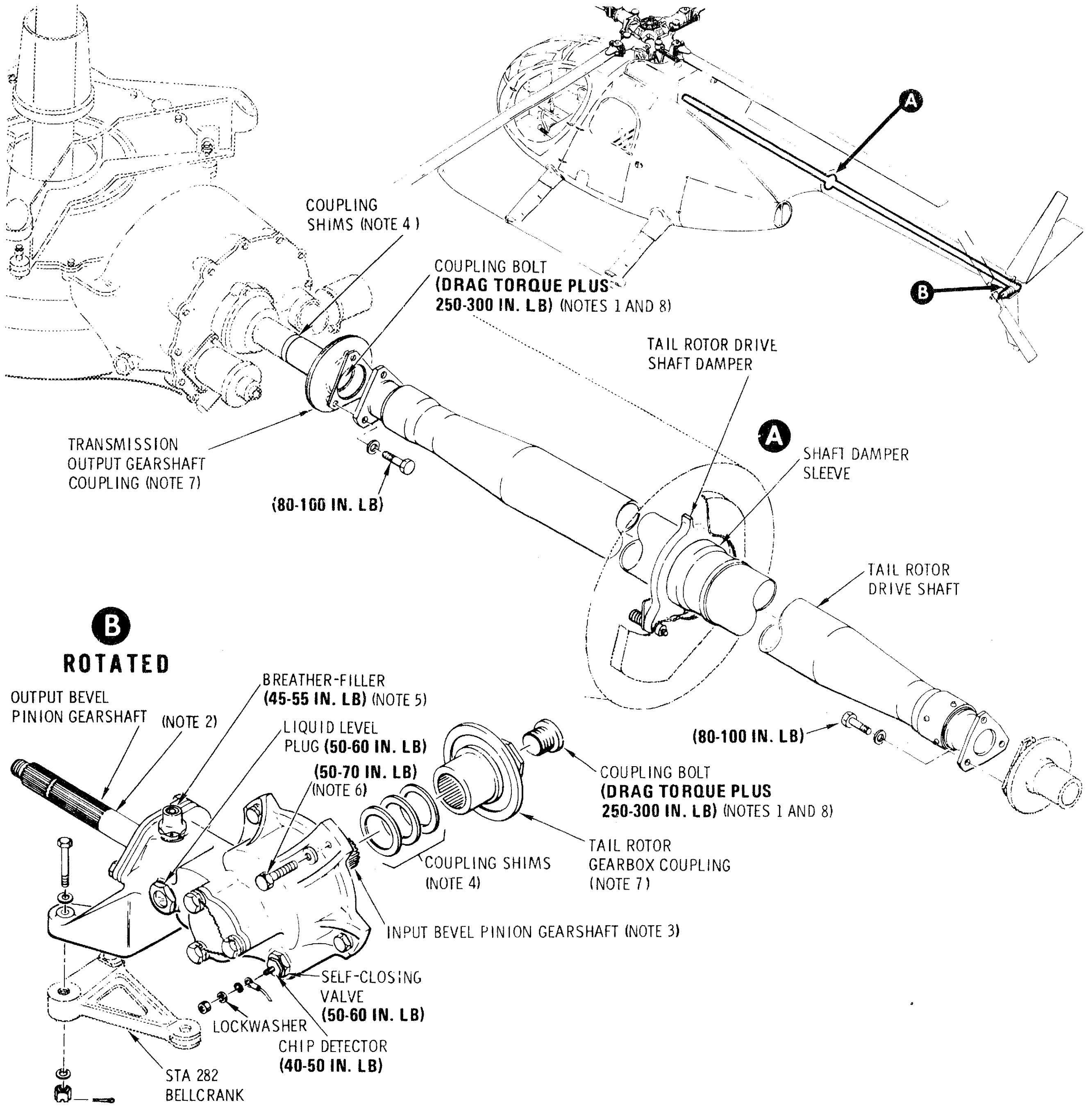
b. Check for scratches in the shaft section that passes through the plenum chamber fairing tube and bulkheads. Such scratches indicate possible contact with the tube edges in the bulkhead openings.

c. Check for evidence of torsional buckling or shaft bending as a result of a blade strike or hard ground contact by the lower stabilizer.

d. Check shaft (damper) sleeve (fig. 6-13) for security of bond and evidence of slippage (sleeve shifting) from excessive heat or loads.

e. Check for corrosion.

f. Check attaching hardware for stripped or crossed



NOTES:

1. DRAG TORQUE FOR BOLT (SELF-LOCKING) SERVICEABILITY IS 15 IN.LB MINIMUM, 150 IN. LB MAXIMUM.
2. RUNOUT (TIR) NOT MORE THAN 0.005 INCH AT THIS POINT.
3. AXIAL PLAY NOT MORE THAN 0.005 IN. WHEN OUTPUT SHAFT IS MOVED IN AND OUT.
4. ALUMINUM ALLOY; 0.002-IN. LAMINATIONS.
5. BREATHER MUST BE INSTALLED WITH HOLE ORIENTED REARWARD WITHIN ±85° OF PARALLEL WITH THE AIRCRAFT CENTERLINE (INPUT SHAFT).
6. REPLACE AN174H-17A BOLT WITH NAS 1304-24H BOLT AT FIRST OPPORTUNITY.
7. COAT COUPLING SPLINES WITH LUBRICANT (C63) BEFORE ASSEMBLY.
8. COAT COUPLING BOLT THREADS WITH ANTISEIZE COMPOUND (C14) BEFORE ASSEMBLY.

12-140J

Figure 6-13. Tail Rotor Drive Shaft, Damper and Tail Rotor (Transmission) Gearbox.

threads and nutplates for drag torque. Discard unserviceable hardware.

6-74. Repair — Tail Rotor Drive Shaft. *a.* Replace shafts for bending, defective damper sleeve, or for scratches, nicks, dents and corrosion that exceed limits, *c* and *d* below. The tail rotor drive shaft damper sleeve repair and rework limits are the same as for the tail rotor drive shaft.

NOTE

If shaft is replaced for torsional buckling, replace and condemn the main transmission output gearshaft coupling, the tail rotor gearbox coupling, and both coupling bolts (fig. 6-13).

b. **NICKS AND SCRATCHES NO DEEPER THAN 0.001 INCH ARE ALLOWABLE WITHOUT REWORK BUT REQUIRE FINISH PROTECTION WITH CHEMICAL FILM (C20).**

CAUTION

Any section of the shaft that appears to be scratched, nicked or corroded in a previously reworked area shall not be repaired a second time. Remove such a shaft from further service.

c. Nicks, scratches and corrosion that do not exceed the limits described below are allowable with rework. Completely remove, smooth out and blend into surrounding material with grade 400 abrasive paper (C3) followed by polishing with crocus cloth (C25). Apply finish protection with chemical film (C20).

(1) SCRATCHES 0.007 INCH DEEP TO A MAXIMUM LENGTH OF 1.0 INCH, AT ANY RANDOM ANGLE.

(2) SCRATCHES 0.010 INCH DEEP TO A MAXIMUM LENGTH OF 0.25 INCH, AT ANY RANDOM ANGLE.

(3) CIRCUMFERENTIAL SCRATCHES 0.004 INCH DEEP.

d. **SMOOTHLY CONTOURED DENTS ARE ALLOWABLE WHEN THE RATIO OF DENT DIAMETER TO DENT DEPTH IS AT LEAST 15:1, THAT IS, WHEN THE DEPTH IS 0.040 INCH, THE MINIMUM ACCEPTABLE DIAMETER WOULD BE 0.60 INCH. THE MAXIMUM ACCEPTABLE DEPTH OF A DENT IS 0.040 INCH. DENTS THAT RAISE MATERIAL REQUIRE SHAFT REPLACEMENT.**

6-75. Installation — Tail Rotor Driveshaft. (See fig. 6-13.)

CAUTION

Replacement of the main transmission, tail rotor driveshaft, tail rotor transmission or tail boom assembly requires mandatory reshimming of both the main transmission and tail rotor transmission couplings.

a. Remove any existing shims and install one 0.0101 inch shim at both the main transmission and tail rotor transmission couplings. Coat coupling bolt threads with antiseize compound (C14). TORQUE COUPLING BOLTS TO ACTUAL DRAG TORQUE PLUS 250-300 INCH POUNDS. DRAG TORQUE FOR COUPLING BOLT SERVICEABILITY IS 15 INCH POUNDS MINIMUM, 150 INCH POUNDS MAXIMUM.

b. With help from an assistant, guide driveshaft carefully through tail boom and damper into position.

NOTE

Install tail rotor drive shaft with data plate forward.

c. Using bolts (P/N AN4H77A) as a workaid, install bolts through tail rotor gearbox mount holes. Install tail rotor gearbox in place. Start bolts (do not overtighten bolts as they may break nutplates loose in tailboom).

WARNING

Application of additional loads to tail rotor transmission in extended position could cause damage to gearbox mounting frame on nutplate connection.

d. Carefully pull tail rotor gearbox to outboard end of bolts, (P/N AN4H77A), to gain access to the end of the driveshaft and coupling on tail rotor gearbox.

e. Install three bolts and washers at forward end of shaft. TORQUE BOLTS 80 TO 100 INCH POUNDS.

CAUTION

While fastening the driveshaft to the tail rotor gearbox coupling, the shaft will ride on the bulkhead just forward of station 142 bellcrank. A cushion material of sort will stop the shaft from rubbing the bulkhead. Remove after use.

f. Carefully slid tail rotor gearbox forward into place against tail boom. Remove left top and right bottom long bolts. Install two bolts (P/N NAS464P4A4) used to mount tail rotor gearbox to tailboom attaching points. TORQUE BOLTS 50-70 INCH POUNDS.

g. Align the forward end of the driveshaft with the coupling on the main transmission output gearshaft. Partially install two bolts as alignment pins but do not tighten to obtain tension.

h. Use a feeler gage to measure the gap at each bolt hole between the main transmission coupling and the tail rotor driveshaft. Average the three readings. Record the average gap.

i. Remove two alignment bolts from coupling. Remove the two mount bolts holding tail rotor gearbox to the tail boom. Reinstall the two workaid bolts.

j. Carefully slide the tail rotor gearbox to the end of bolts again, to gain access to coupling and driveshaft.

k. Holding output shaft, remove bolts holding driveshaft to coupling. Carefully slid driveshaft forward.

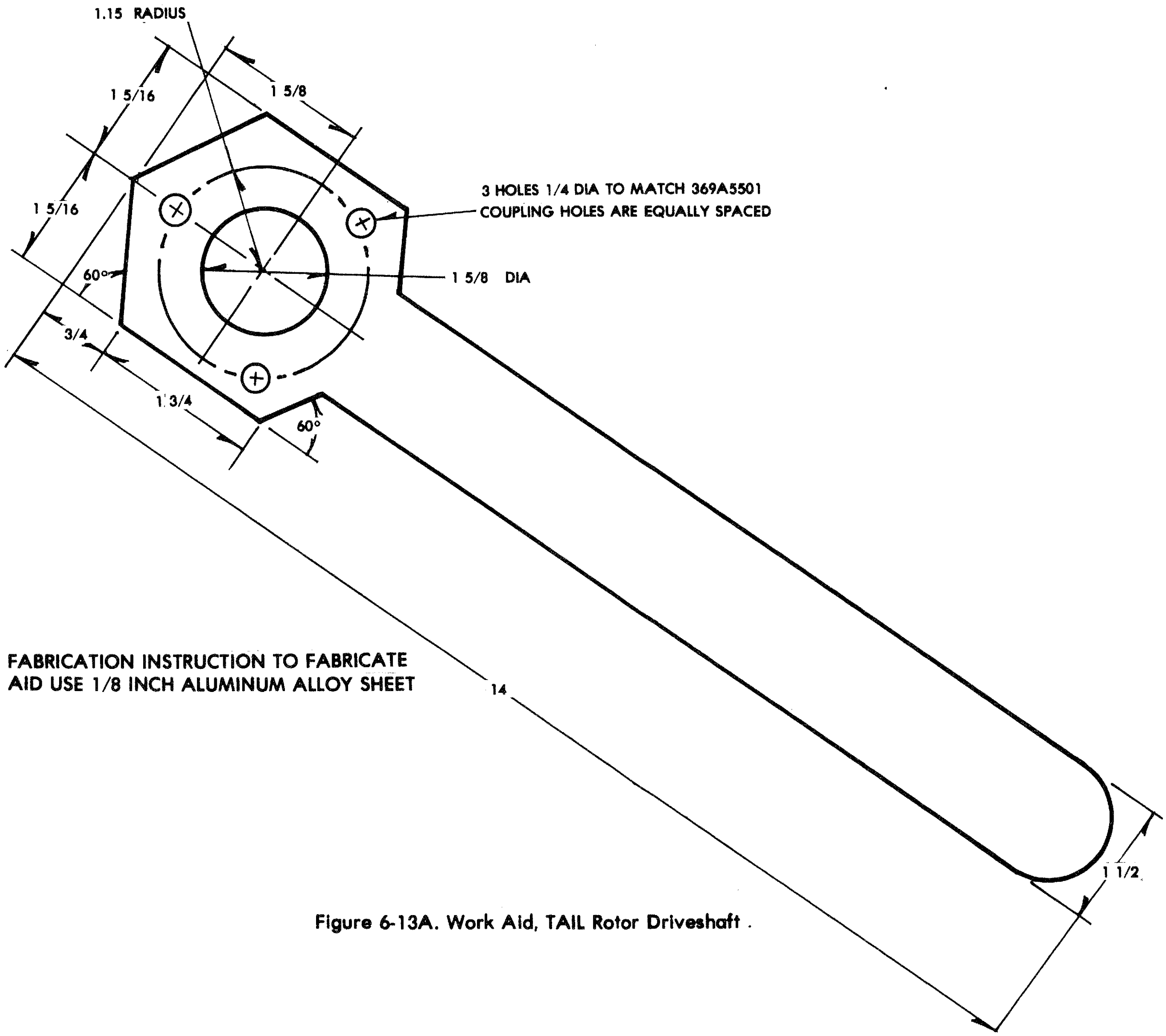


Figure 6-13A. Work Aid, TAIL Rotor Driveshaft .

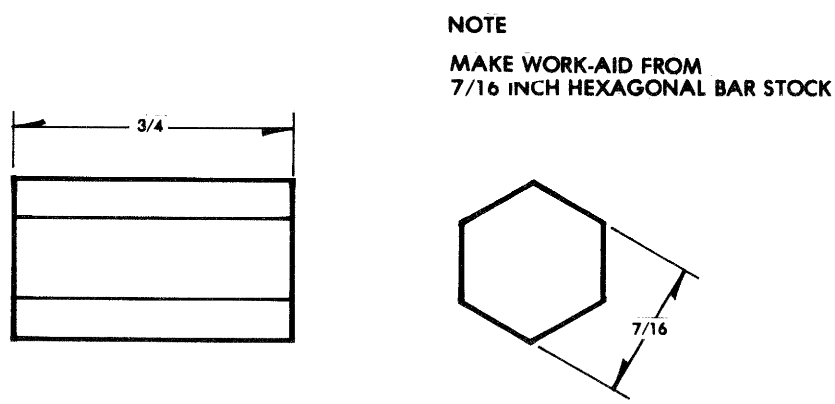


Figure 6-13B. Work Aid, Tail rotor Driveshaft *l.* Using bolts from coupling, fasten workaid (fig. 6-13B) to coupling on tail rotor gearbox.

m. Using workaid (fig. 6-13A), remove the hex bolt holding the driveshaft coupling to the tail rotor gearbox.

n. Select a shim thickness to fill the measured gap. NO GAP (COUPLING TENSION) IS ALLOWABLE, BUT THERE MAY BE UP TO 0.005 INCH COUPLING COMPRESSION.

o. Divide shims equally between the main transmission and tail rotor gearbox couplings. Install required shims. Coat coupling splines with lubricant (C63) and coupling bolt threads with antisieze compound (C14). Using workaids reinstall coupling and through bolt. TORQUE COUPLING THROUGH BOLT TO ACTUAL DRAG TORQUE PLUS 250-300 INCH POUNDS.

p. Align shaft to coupling on tail rotor gearbox. TORQUE BOLTS TO 80-100 INCH POUNDS.

q. Carefully slide tail rotor gearbox forward. Install upper left and lower right mount bolts. Torque bolts 50 to 70 inch pounds. Recheck for gap between main transmission coupling and driveshaft.

r. Apply primer (C79) to the four gearbox mounting bolts. Remove long bolts and install bolts with washers while the primer is still wet. TORQUE BOLTS TO 50-70 INCH POUNDS.

s. Install three bolts and washers at forward end of shaft. TORQUE BOLTS TO 80-100 INCH-POUNDS. Lockwire the four gearbox mounting bolts in pairs with 0.032 inch lockwire (C57).

t. Connect the electrical wire to the gearbox chip detector.

u. Insert pin of station 282 bellcrank into bearing in tail rotor pitch control. Pivot bellcrank to align with mating hole in tail rotor transmission and install bolt, two washers, nut and nut cotter pin.

v. Slowly rotate shaft and CHECK FOR NOT LESS THAN 0.25 INCH CLEARANCE BETWEEN SHAFT OF AND THE FAIRING TUBE AT STATION 137.5. While turning, also check to ensure shaft is not bent. Install all access doors and covers.

6-76. TAIL ROTOR DRIVE SHAFT COUPLINGS.

6-77. Removal — Tail Rotor Drive Shaft Couplings. (See fig. 6-13.) *a.* Remove tail rotor drive shaft (para 6-72).

b. Remove rear coupling from tail rotor transmission by removing attaching bolt. (See fig. 6-13.) Retain shims with coupling for reinstallation.

c. Remove front coupling from main rotor transmission by removing attaching bolt. (See fig. 6-7.) Retain shims with coupling for reinstallation.

6-78. Inspection — Tail Rotor Drive Shaft Couplings. *a.* Check coupling diaphragm for dents, cracks, nicks, rust spots and joint separation at the weld junction. If any such defect is evident the coupling must be replaced.

b. Check coupling ends for scratches, nicks, dents, cracks and corrosion pits. No cracks are allowed. Maximum depth of other defects before rework is 0.005 inch.

c. Measure the overall length of the couplings at three different points. If the average of the readings exceeds 2.541 inches, the coupling must be replaced.

6-79. Repair — Tail Rotor Drive Shaft Couplings. *a.* No repairs to the coupling diaphragms are permissible.

b. Repair damage to the ends of the coupling that is no deeper than 0.005 inch. Use abrasive paper grades 400 and 600 (C3) and (C4) and crocus cloth (C25) to completely remove and polish out the defect. **MAXIMUM DEPTH AFTER REWORK IS 0.010 INCH.** Apply primer (C79) to repaired area for corrosion protection.

6-80. Installation — Tail Rotor Drive Shaft Couplings. Install tail rotor couplings and tail rotor drive shaft (para 6-69).

6-81. TAIL ROTOR DRIVE SHAFT DAMPER.

6-82. Description — Tail Rotor Drive Shaft Damper. The tail rotor drive shaft damper is a graphite-filled teflon plate that controls and limits deflection of the tail rotor drive shaft about its approximate midpoint at all drive system speeds. The damper is spring-loaded against a structural support bracket mounted on the aft section tailboom fairing. Maximum possible shaft deflection in one direction is limited to less than 0.50 inch at the extreme throw positions of the damper.

6-83. Troubleshooting — Tail Rotor Drive Shaft Damper. If there is excessive damper vibration during acceleration from ground idle, the probable cause is a defective or loose damper. Friction adjustment or damper replacement is required.

6-84. Inspection — Tail Rotor Drive Shaft Damper (Installed). *a.* Remove boom bolts access doors.

b. Inspect the tail rotor drive shaft damper and support structure for broken, cracked, or bent parts.

c. Displace the damper so that it touches the tail rotor drive shaft.

d. Use a wire gage to measure the damper to shaft clearance 180 degrees from the contact point. **MINIMUM ACCEPTABLE SHAFT CLEARANCE IS 0.020 INCH.**

e. Repeat steps *c* and *d* above at 90 degrees intervals from the initial check until clearance on each side (360 degrees) of damper has been checked.

f. Dampers not meeting the minimum clearance of 0.020 inch on each side are to be replaced.

g. Close access doors.

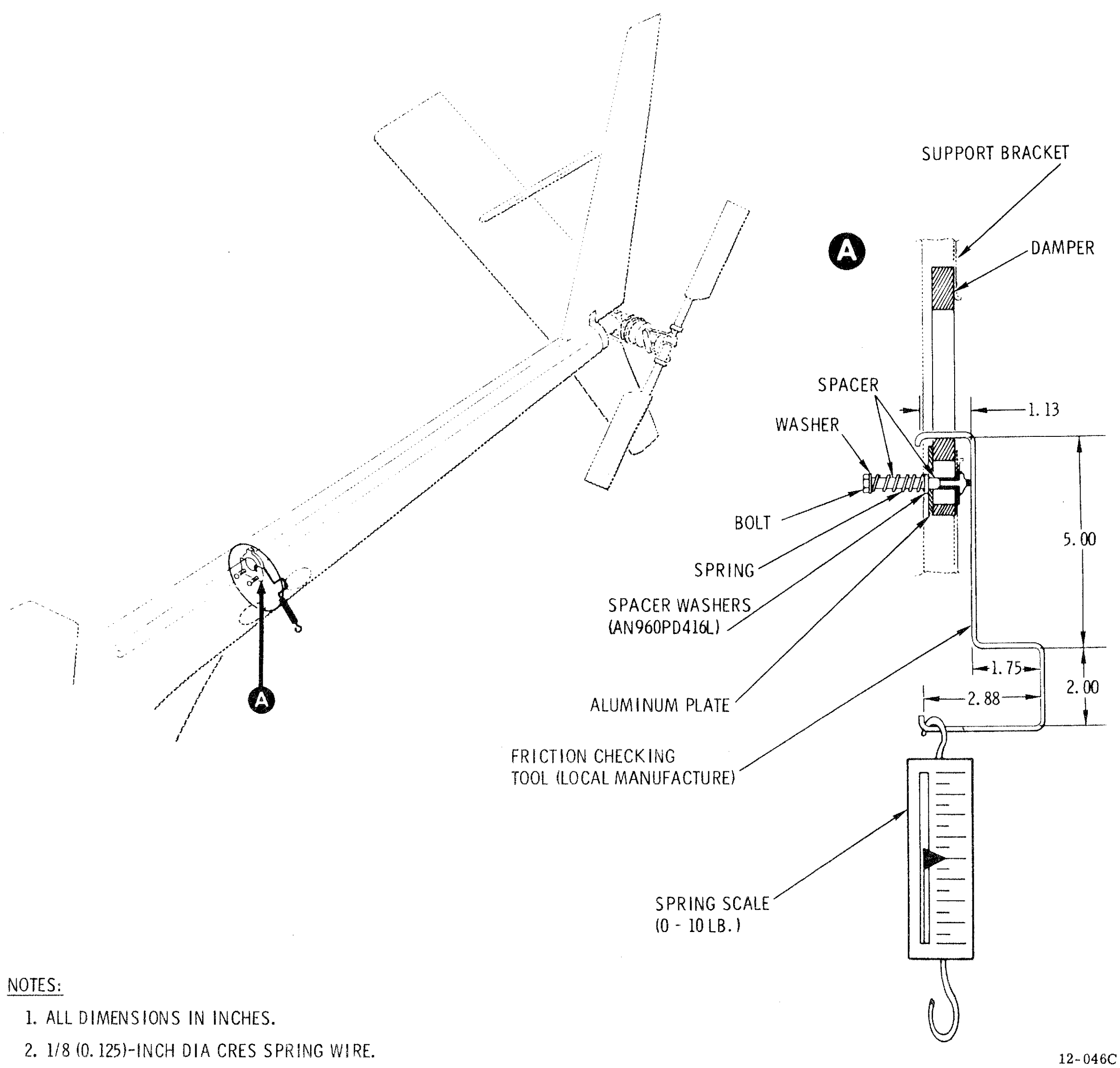


Figure 6-14. Tail Rotor Drive Shaft Damper Friction Check and Adjustment.

6-85. Adjustment — Tail Rotor Drive Shaft Damper. (See fig. 6-14.) *a.* Remove tail rotor drive shaft (para 6-72).

b. Using a 0-10 pound spring scale, measure the force required to move the damper radially on bulkhead. Pressure friction must be adjusted so that a pull of 1.75-2.25 pounds is required to slide the damper between the plate and support bracket.

c. To increase friction, add spacer washers between the springs and the plate. To decrease friction, remove washers. (Add or remove the same number of washers at each pressure point.)

NOTE

A minimum of one washer must always remain between each spring and the plate.

d. Install tail rotor drive shaft (para 6-75).

6-86. Removal — Tail Rotor Drive Shaft Damper. (See fig. 6-14.) *a.* Remove boom bolts access doors.

b. Remove tail rotor drive shaft (para 6-72).

c. Remove the two bolts, washers, springs, shim washers, plate and spacers, and damper.

6-87. Inspection — Tail Rotor Drive Shaft Damper (Removed). Inspect the inside diameter of the damper for excessive wear. If the diameter is more than 3.035 inches, the damper must be replaced.

6-88. Repair — Tail Rotor Drive Shaft Damper. Repair of the tail rotor drive shaft damper is limited to replacement of defective parts. (Refer to chapter 2 for replacement or repair of tail rotor drive shaft support bracket.)

6-89. Installation — Tail Rotor Drive Shaft Damper. (See fig. 6-14.)

CAUTION

Make certain that the two bolts and washers under the boltheads are seated against the spacers when tightened. The bolts will wear rapidly if the clamp-up is not solid.

a. Position damper over mounting holes in support bracket. Install plate and spacers with bolts, washers, springs, and spacer washers.

b. Adjust damper friction, paragraph 6-85.

c. Reinstall tail rotor drive shaft (para 6-75).

d. Install access doors.

SECTION VI INTERMEDIATE GEAR BOX

(Not Applicable)

SECTION VII TAIL ROTOR GEAR BOX

6-90. TAIL ROTOR (TRANSMISSION) GEARBOX

6-91. Description — Tail Rotor (Transmission) Gearbox. The tail rotor (transmission) gearbox is a right-angle, speed-increasing transmission (2045 to 3018 rpm at 100% N2) having a magnesium alloy housing. (See fig. 6-13.) The transmission drive consists of a pair of spiral-bevel gears that are splash lubricated. An oil liquid level sight plug and magnetic chip detector are located on the aft end of the transmission and a breather-filler is located on top. Refer to chapter 1 for transmission servicing information.

6-92. Inspection — Tail Rotor (Transmission) Gearbox (Installed). *a.* Check the transmission for leaks, cracks, or corrosion. Evaluate oil leakage according to paragraph 6-104.

NOTE

Operation in heavy rain may cause water to enter the gearbox through the breather-filler. If water contamination is suspected, drain gearbox oil and replace with new oil (chapter 1).

b. Check that bolts are secure with lockwiring intact.

c. (Refer to paragraph 6-54 and table 6-1, chip detector caution light on.) Remove lockwire, electrical wire and chip detector from self-closing valve. Check for presence of foreign matter such as dirt or metal particles. If metal particles are present, drain oil and observe condition. If no metal particles are present, wipe detector clean and reinstall (fig. 6-13). Lockwire chip detector.

6-93. Removal — Tail Rotor (Transmission) Gearbox. *a.* Remove tail rotor from tail rotor gearbox (para 5-90).

b. Remove tail rotor drive shaft; remove gearbox from shaft (para 6-69).

6-94. Disassembly — Tail Rotor (Transmission) Gearbox. (See fig. 6-13.) Remove coupling bolt, gearbox coupling, and coupling shims from input bevel pinion gearshaft. Retain shims for reuse.

NOTE

Further disassembly is not required as a replacement transmission is equipped with all other accessories.

6-95. Inspection — Tail Rotor (Transmission) Gearbox (Removed). (See fig. 6-13.) *a.* Check the input bevel gearshaft for axial play by moving the input shaft in and out. The gearbox must be cold. **AXIAL PLAY IS LIMITED TO 0.005 INCH MAXIMUM.**

b. Check TIR with dial indicator contact in split ring groove of the output bevel pinion gearshaft. **RUNOUT IS LIMITED TO 0.005-INCH TIR.** No axial play is permissible.

c. Check for high and low runout at the split ring groove on the output shaft. Mark the high and low extremes on the outer end of the shaft with a grease pencil.

NOTE

Marking the shaft to indicate the runout extremes will provide a guide for reinstallation of the tail rotor (para 5-103) so that chances of high frequency vibration are reduced along with a lessening of the need for tail rotor balancing.

d. Inspect all scoring or scratching of the output shaft to determine if the marks penetrate through the cadmium plating and into the steel shaft. **PENETRATION OF THE STEEL SHAFT IS ALLOWABLE TO A MAXIMUM DEPTH OF 0.005 INCH IF THE DAMAGED AREA IS REPAIRED ACCORDING TO PARAGRAPH 6-96.**

NOTE

The output shaft may have indications of scoring or scratching that do not penetrate the cadmium plating. If inspection reveals that the steel is not penetrated do NOT rework the area. To polish out such marks will only remove additional cadmium, leaving the shaft susceptible to corrosion.

6-96. Repair — Tail Rotor (Transmission) Gearbox. (See fig. 6-15.) Repair of the tail rotor gearbox is limited to repair of the output pinion gearshaft, and replacement of; input or output pinion gearshaft seals; liquid level plug; chip detector; breather filler (or); the complete gearbox assembly. For replacement of the liquid level plug, chip detector, or breather-filler refer to paragraph 6-98.

a. Scratches, corrosion or score marks that penetrate the steel output shaft between the seal lip contact area and splines are allowable with rework provided the repaired area is not reduced more than 0.005 inch below the surrounding area.

b. Smooth out and blend the defect into surrounding material with grade 400 wet or dry abrasive paper (C3) and then polish with crocus cloth (C25).

c. Clean shaft with thinner (C109) and apply a light coat of primer (C79) for corrosion protection.

6-97. Installation — Tail Rotor (Transmission) Gearbox.

CAUTION

Do not carry or otherwise support the gearbox by the coupling, as the coupling diaphragm will buckle if excessively deflected. During performance of all maintenance on the gearbox, use extra care to keep contaminants such as paint, dirt, etc. from the areas around the input and output shaft seals.

a. Install tail rotor drive shaft and gearbox (para 6-75).

b. Install tail rotor on tail rotor gearbox (chapter 5).

c. Drain any preservative oil residue and service gearbox with lubricating oil (chapter 1).

6-98. TAIL ROTOR TRANSMISSION EXTERNAL COMPONENT REPAIRS.

6-99. Liquid Level (Sight) Plug — Tail Rotor Transmission External Components. (See fig 6-15.) *a.* Drain oil from transmission until oil level is well below edge of sight plug and port.

b. Remove lockwire from sight plug.

c. Remove sight plug by unscrewing.

d. Remove and inspect O-ring. Install new O-ring on sight plug, if required.

NOTE

When a new sight plug is being installed, the liquid level markings are added (para 6-101) after the plug is installed and torqued. Coat new sight plugs that are not identified with an "X" after the part number with silicone primer to prevent clouding of sight glass. Refer to paragraph 6-100 below for coating application.

e. Install sight plug. **TORQUE SIGHT PLUG TO 50 - 60 INCH-POUNDS,** and safety with 0.032-inch lockwire (C57).

6-100. Application of Sight Plug Silicone Coating — Tail Rotor Transmission External Components. A new sight plug (S53H) may be coated as follows:

a. Fill a clean container with silicone primer (C81) deep enough to cover the sight plug.

b. Dip plug in primer long enough to coat inner face of window. Allow to air-dry.

c. Reidentify plug by adding an "X" after the part number.

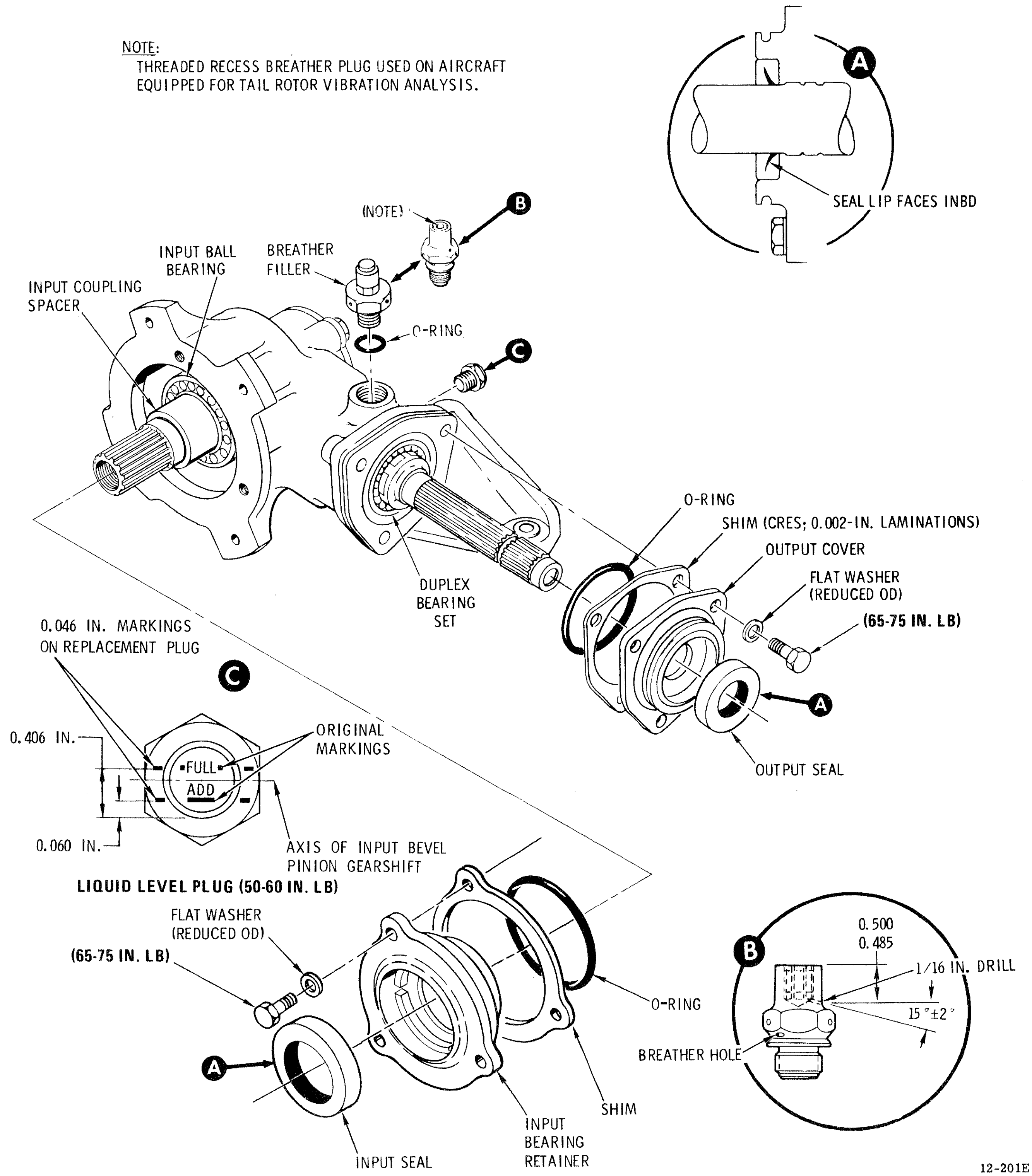


Figure 6-15. Tail Rotor Transmission Repairs.

6-101. Application of Markings-Liquid Level (Sight) Plug — Tail Rotor Transmission External Components. (See fig. 6-15.) Oil level markings may be applied on newly installed sight plugs as follows:

a. Using a machinists level and scale, scribe an ADD line horizontally on both outer faces of the sight plug, above the lower edge of the window as shown in detail A. Do NOT mark the window.

b. Scribe the FULL line horizontally on both outer edges of the sight plug, above the lower edge of the window as shown in detail A. Do NOT mark the window.

CAUTION

Application of any cleaning material other than soap and water to the sight plug window may cause it to craze.

c. Using white paint (C-55), paint four lines with the edges touching the scribe marks as shown in detail A. Do NOT paint the surface of the window.

6-102. Chip Detector — Tail Rotor Transmission External Components. Refer to tail rotor transmissions draining, removal and installation instructions. Refer to chapter 9 for functional test of detector circuit. Refer to paragraph 6-54 for inspection.

6-103. Breather-Filler — Tail Rotor Transmission External Components. *a.* Remove lockwire from breather-filler.

b. Remove breather-filler by unscrewing.

c. Remove and inspect the O-ring. Replace unserviceable O-ring.

d. Use step *f* below to install non-threaded insert breather-filler. Use steps *e* and *f* below to install threaded insert breather-fillers.

e. Locate the breather hole on threaded insert breather-fillers (between the flange and the flats). This type breather-filler must be installed with the breather hole oriented rearward with ±85 degrees of parallel with the aircraft centerline (input shaft). If proper orientation of the breather-filler cannot be obtained, one or two AN960PD1016L washers may be added between the O-ring and the breather-flange (one washer will change the breather hole orientation approximately 100 degrees).

f. **TORQUE THE BREATHER-FILLER TO 45 - 55 INCH-POUNDS.** Secure the breather-filler with 0.032-inch lockwire (C57).

6-104. MAIN AND TAIL ROTOR TRANSMISSION OIL LEAKAGE CRITERIA.

6-105. General — Oil Leakage Criteria. Oil leakage, seepage or capillary wetting at the oil seals or assembly joint lines of the main transmission or tail rotor transmission are permissible if the leakage rate does not exceed 2 cc per hour (one drop per minute). An acceptable alternate rate of leakage from either transmission is if oil loss is not more than from the full to the add mark on the sight gage within 25 flight hours. Excessive leakage must be repaired or the transmission replaced.

NOTE

On transmission input and output pinion gear oil seals with less than 2 hours of operation, some seepage or wetting of adjacent surfaces is normal until the seal is wetted and worn-in (seated). If seepage continues at a rate of one drop per minute or less, the seal may be continued in service. Check transmission oil level and observe seepage rate after every 2 hours of operation. Shorter inspection periods may be required if seal leakage appears to be increasing.

6-106. TAIL ROTOR TRANSMISSION OIL SEAL REPLACEMENT.

6-107. Output Gearshaft Oil Seal — Tail Rotor Transmission Seal Replacement. (See fig 6-15.)

NOTE

Before proceeding with the following steps, verify that output gearshaft radial and axial play do not exceed limits specified in paragraph 6-95.

a. Drain oil from transmission (chapter 1).

NOTE

To maintain the best tail rotor balance possible, mark or index the output shaft for reassembly reference.

b. Remove tail rotor assembly (chapter 5).

CAUTION

Press firmly against output shaft as output cover is removed to prevent unseating of the duplex bearing set. Replacement of the output gear shaft oil seal should only be performed under carefully controlled conditions.

c. Remove three bolts, washers, and output cover from gearbox.

d. Remove laminated shim. Retain shim with cover and use care to keep them from becoming damaged.

e. Check O-ring for cuts, breaks or swelling. Discard O-ring if it is defective.

f. Press seal from output cover and discard.

g. Apply sealing compound (C89), to outside diameter (OD) of seal. Using an arbor press, press seal into output cover so that lip direction will be toward gears (detail A). Do not allow seal to cock in cover bore during installation.

NOTE

Zinc chromate primer (C79) may be used on seal OD if sealing compound is not available. Press seal into place while primer is still wet.

h. Inspect circumference of gearbox output shaft for scratches, corrosion and general condition of finish where the seal lip makes contact.

CAUTION

Do not use an axial or diagonal (helical) motion when polishing the shaft. Use a polishing motion that is rotational and at right angles to the shaft centerline. A finish lay that is axial or helical to the direction of shaft rotation will cause excessive seal wear.

i. Using crocus cloth (C25), polish out any indications of corrosion or scratches. Reworked finish must be equal to or better than original finish.

j. Wrap one layer of cellulose tape (C18) over the full length of the output shaft splines.

k. Install new O-ring, if required. Lubricate the tape, packing and mating bore, and seal lip with petrolatum (C73).

l. Very carefully install shim and output cover; the sharp edge of the seal lip must not be damaged.

CAUTION

Do not substitute standard AN960 flat washers for the NAS620A416L washers used under the output cover bolts. The NAS washers have a reduced OD to prevent entry of the washer edges into adjacent radii of the cover. Washer substitution will produce false readings when torquing the cover bolts, and the interference pressure may crack the cover.

m. Install the cover bolts and washers; **TORQUE BOLTS TO 65-75 INCH-POUNDS.** Secure all bolts together with 0.032-inch lockwire (C57).

n. Check output shaft for axial play; no axial play is permissible. If play is detected, remove output cover and shim. Shim laminations are 0.002 inch thick. Peel away laminations as required to produce 0.001- to 0.002-inch clamp-up pressure on the outer races of the output shaft duplex bearing set. Reinstall shim and cover, steps *l* and *m* above, and lockwire all boltheads together.

o. Remove tape and wipe excess petrolatum from shaft. Do NOT wipe petrolatum from seal rubber as it protects the seal from drying out.

p. Reinstall tail rotor assembly (chapter 5), and service the gearbox (chapter 1).

6-108. Input Gearshaft Oil Seal — Tail Rotor Transmission Seal Replacement. (See fig. 6-16.) *a.* Remove tail rotor transmission (para 6-93).

b. Drain oil from transmission (chapter 1).

c. Match mark the input bearing retainer and gearbox housing for reassembly reference.

CAUTION

Do not allow the input shaft to move outward. Excessive outward movement of the input pinion shaft will result in mislocation of the anular contact bearing balls which will require that the transmission be returned to overhaul.

d. Position the transmission on a suitable work bench with the input pinion shaft pointing upward. Remove three bolts and washers securing the input bearing retainer assembly (fig. 6-15.) While holding moderate downward pressure on the input pinion gearshaft, carefully lift off the input bearing retainer and shim. Retain shim with housing. Cover the gearshaft and housing bore with a clean lint-free cloth to prevent entry of foreign material.

e. Using an arbor press and suitable adapters, press seal from retainer assembly. Clean seal area with methyl ethyl ketone (C69) to remove all sealing compound residue.

f. Apply a light coat of sealing compound (C89) to the OD of seal. Using an arbor press, press seal into retainer assembly with the seal lips facing inboard (detail A).

g. Inspect O-ring for cuts, breaks or swelling. Discard and replace O-ring if it is defective. Lubricate the seal lip and O-ring with petrolatum (C73). Install O-ring.

h. Inspect circumference of input coupling spacer for scratches, corrosion and general condition finish where the seal lip makes contact.

i. Using crocus cloth (C25), polish out any indications of corrosion or scratches. Polish with a motion that is in the direction of shaft rotation. Rework finish must be equal to or better than original finish.

WARNING

Do not alter input pinion shim thickness.

j. Carefully install the retainer and shim with the match marks aligned.

k. Reinstall retainer washers and bolts; **TORQUE BOLTS TO 65-75 INCH-POUNDS.** Secure all bolts together with 0.032-inch lockwire (C57).

l. Reinstall tail rotor transmission (para 6-97) and service (chapter 1).

CHAPTER 7
HYDRAULIC AND PNEUMATIC SYSTEMS

(Not Applicable)

CHAPTER 8

INSTRUMENTS

SECTION I INTRODUCTION

8-1. General Information — Instruments. This chapter provides all essential information for maintenance personnel to perform maintenance on the instruments and associated electrical components, according to the Maintenance Allocation Chart.

8-2. Description — Instruments. All of the instruments covered in this chapter are located on the instrument panel (fig. 8-1) excepting the outside air temperature thermometer which is inserted through the canopy and the magnetic compass which is mounted on the canopy frame above the instrument panel. Refer to chapter 9 for maintenance information pertaining to instrument lighting and caution and warning indicator lights. Refer also to chapter 9 for maintenance information pertaining to electrical power (ac or dc) provided to instrument systems requiring an external power source. Components of some instrument systems are covered in other chapters of this manual or in other manuals and in such cases reference is made to the applicable chapter or manual. Appendix F contains wiring diagrams for all instrument systems employing an electrical function excepting the homing, heading, and bearing indicator which is covered in TM 11-1520-214-20 and TM 11-1520-214-20-1.

WARNING

Radioactive Materials. Those instruments having radioactive self-luminous instrument dials will have a radioactive warning legend preceding the maintenance instructions. If such an instrument is broken or becomes unsealed, avoid personal contact with the instrument. Use forceps or gloves made of rubber or polyethylene to pick up contaminated material. Place the material and the gloves in a plastic bag and dispose of as radioactive waste in accordance with AR 755-15 and TM 3-261. (Refer to TB 55-1500-314-25.) Repair procedures shall conform to requirements of AR 700-52.

SECTION II ENGINE INSTRUMENTS

8-3. ENGINE INSTRUMENTS.

8-4. Description — Engine Instruments. All indicators displaying engine, engine accessory, or transmission functions are mounted on the instrument panel shown in figure 8-1. Refer to chapter 9 for caution and warning light system maintenance instructions. Refer to appendix A for reference to engine instrument overhaul publications.

8-5. N1, N2, AND ROTOR TACHOMETER INDICATING SYSTEMS.

8-6. Description — N1, N2, and Rotor Tachometer Indicating Systems. The N1 tachometer indicator (fig. 8-1) indicates the speed of the gas producer turbine in percent of rpm. The N2 and rotor tachometer indicator (fig. 8-1) provides two indications: The actual rpm of the main rotor is indicated on the inner scale with the small (R) pointer; the engine power turbine speed is indicated in percent of rpm on the outer scale with the long pointer. Three 3-phase, 2-pole, synchronous tachometer generators produce the voltages that energize the power turbine rpm (N2), gas producer turbine rpm (N1), and the main rotor rpm (NR) indicators. The N2 and N1 tachometer generators are located on the front of the engine accessories gearcase, and the NR tachometer generator is located by the tail rotor drive output shaft on the main transmission accessory section (fig. 8-2). The tachometer generators are independently connected to the indicators; there is no connection with the main electrical system.

Table 8-l. Premaintenance Requirements for Maintenance of Tachometer Indicating Systems.

Conditions	Requirements
Minimum Personnel Required	Two (MOS 67V & 68F)
Consumable Materials	(C63)

8-7. Inspection — N1, N2, and Rotor Tachometer Indicating Systems. (See fig. 8-1 and 8-2.) Refer to wiring diagram in appendix F.

a. Check that indicator pointer is in the zero position with engine inoperative. If not, replace the indicator.

b. Examine the indicator cases (fig. 8-1) for damage, corrosion, and cracked or broken glass.

c. Examine the indicator faces for legibility of markings and for cracked or peeling paint on the pointers. Verify that the range markings are legible and correctly placed. Refer to TM 55-1520-214-10 for location. Refer to TM 55-6600-200-20 for marking application.

d. Examine wiring (appendix F) from the indicators to the tachometer generators for secure connections and for damage. Examine the bonding wires for secure connection and absence of corrosion.

e. Examine the tachometer generators (fig. 8-2) for damage and security of mounting.

8-8. Troubleshooting — N1, N2 and Rotor Tachometer Indicating Systems. Refer to table 8-2 and the wiring diagram in appendix F.

8-9. N1, N2, AND ROTOR TACHOMETER GENERATORS.

8-10. Removal — N1, N2, and Rotor Tachometers Generators (Typical). (See fig. 8-2.) *a.* Detach the electrical plug from the tachometer receptacle and cap both connectors to exclude contamination.

b. Remove the four nuts, washers, and the ground wire (if used), and extract the generator (pull it straight out).

8-11. Inspection — N1, N2, and Rotor Tachometer Generators (Typical). (See fig. 8-2.) *a.* Examine the gasket for damage. Discard it if it is damaged,

b. Examine the driveshaft keyway for excessive wear.

c. Verify that the driveshaft rotates freely.

8-12. Test — N1, N2, and Rotor Tachometer Generaors (Typical) (AVIM). (See fig. 8-3.) *a.* Mount the tachometer generator on the test stand.

b. Start the tachometer test stand. Set the speed control to zero rpm. Turn the strobe light switch to ON.

c. Connect the tachometer generator to the test setup for 20-ohm load test.

d. Increase the tachometer generator speed to 1000 rpm. Check the phase-to-phase voltage: TP1 to TP2, TP1 to TP3 and TP2 to TP3. The voltage shall be 3.0-4.0 vac for each measurement.

e. Reduce the tachometer generator speed to zero, disconnect the 20-ohm test setup, and connect the 40-ohm load test setup.

f. Increase the tachometer generator speed to 3600 rpm. Check the phase-to-phase voltage: TP1 to TP2, TP1 to TP3, and TP2 to TP3. The voltage shall be 17.5-18.5 vac for each measurement.

g. Increase the tachometer generator speed to 4200 rpm. Check phase-to-phase voltage: TP1 to TP2, TP1 to TP3, and TP2 to TP3. The voltage shall be 20.0-22.0 vac for each measurement.

h. Reduce the tachometer generator speed to zero and disconnect the 40-ohm test setup.

i. Increase the tachometer generator speed to 3600 rpm. Check the phase-to-phase voltage: pin A to ground, pin B to ground, and pin A to pin B. The voltage shall be 25.0 to 31.0 vac for each measurement.

8-13. Installation — N1, N2, and Rotor Tachometer Generators (Typical). (See fig. 8-2.) *a.* Apply a light coat of lubricant (C63) to the generator.

b. Install the gasket on the mounting pad studs.

c. Install the generator on the mounting pad studs, making sure the gasket is flat against the mounting pad and the generator base is flat against the gasket.

d. Attach the grounding wire (if one was previously installed) to one of the studs and secure the generator with four corrosion-resistant washers and nuts.

e. Remove the protective caps from the connectors and join and secure the plug and receptacle. (Fig 4-1, Sheets 1 and 2, Note 3).

8-14. N1 TACHOMETER INDICATOR.

8-15. Removal — N1 Tachometer Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Loosen the clamp on the electrical connector hood. Unscrew the hood and uncouple the plug lock spring from the receptacle lock ring. Rotate the lock spring in either direction. Remove the connector and the bonding jumper (if used) from the indicator. Install dust covers on the electrical connectors.

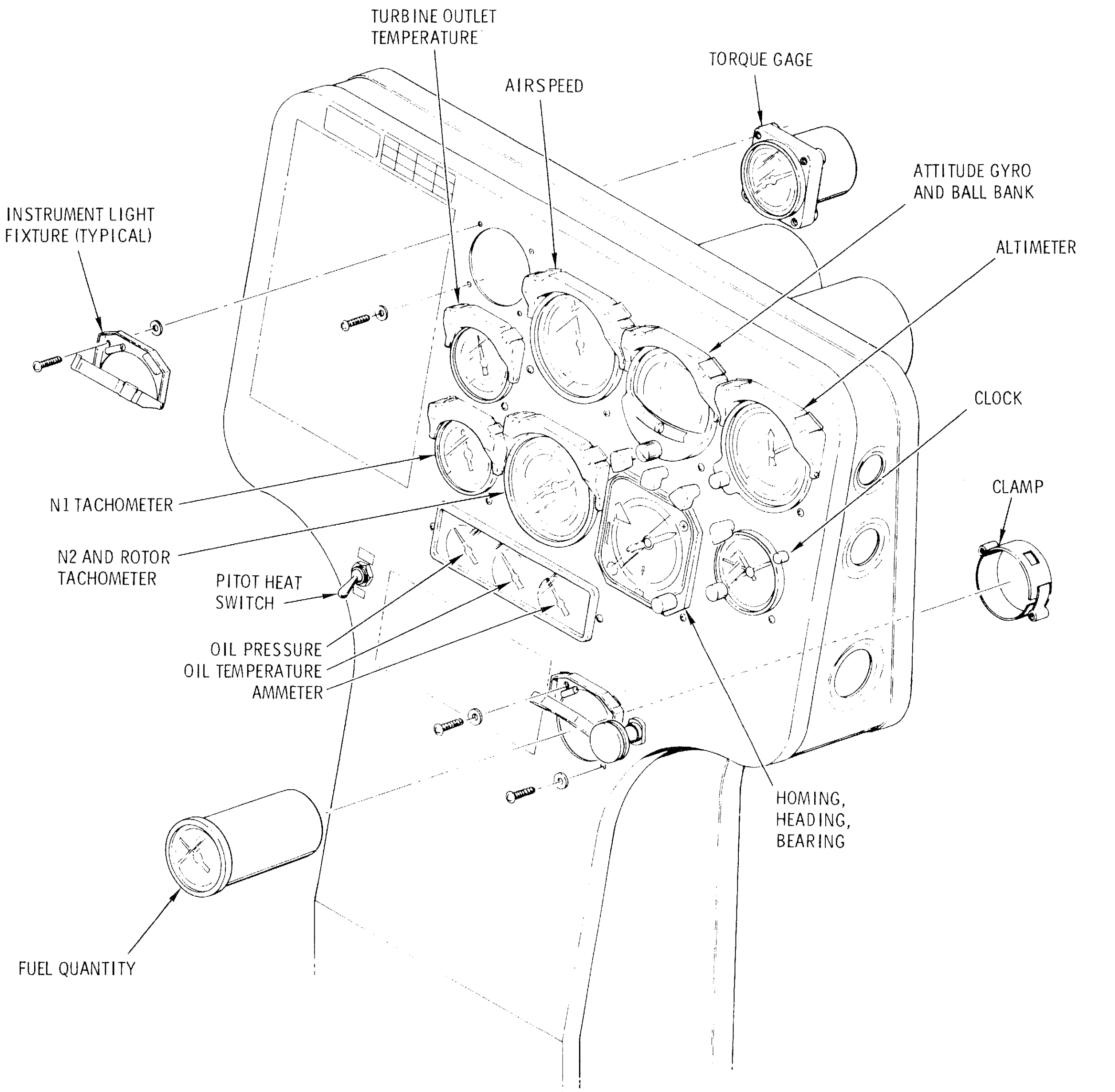


Figure 8-1. Instruments — Panel-Mounted (Typical).

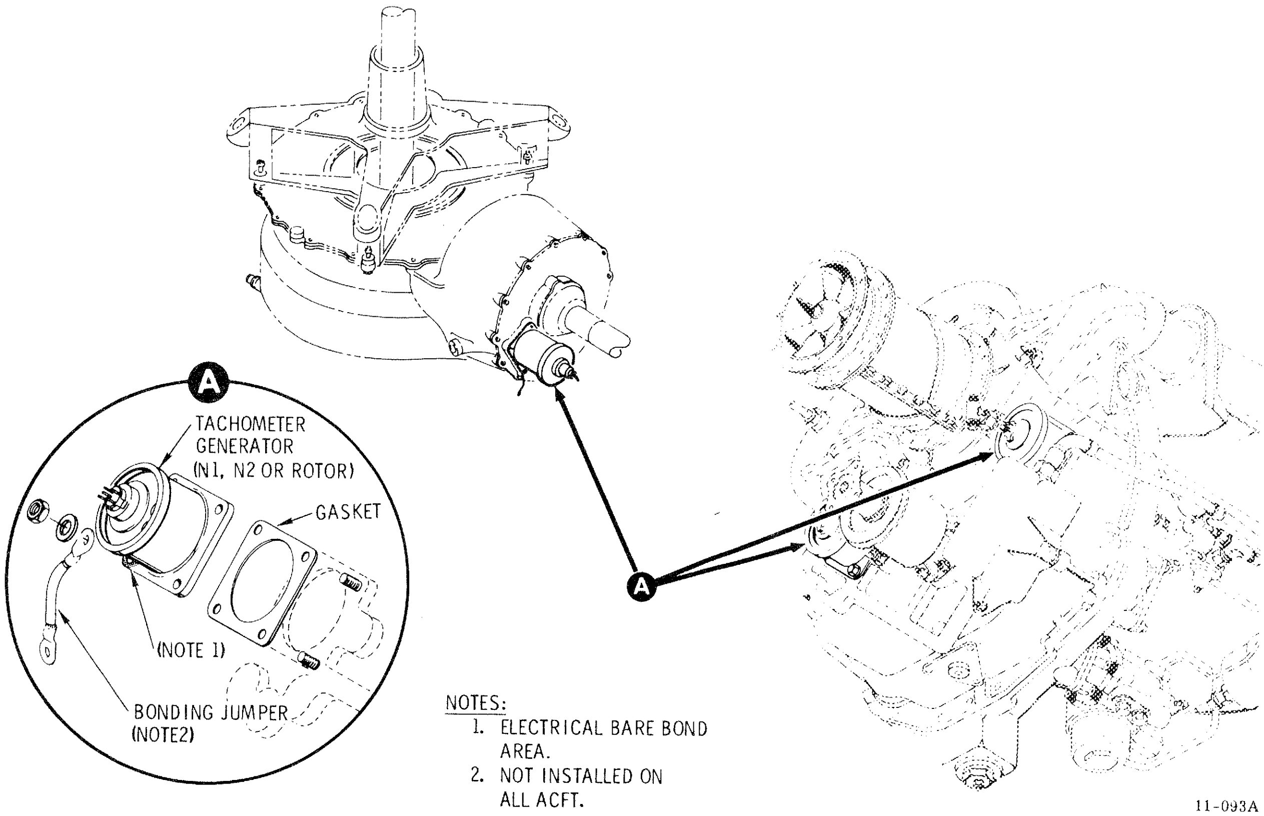


Figure 8-2. Tachometer Generators.

c. Open the instrument light fixture, swing out the lampholder, and remove the 6-32 screw that secures the light fixture.

d. Loosen (but do not remove) the 10-32 cinching screw in the right lower corner of the clamp.

e. Loosen the upper left 8-32 holding screw and slide the indicator outward, away from the panel face and out of the clamp. The clamp need not be removed.

8-16. Calibration Data — N1 Tachometer Indicator (AVIM).

Tachometer Generator Output (cps)	N1 Indicator (percent)
3.54	5 ± 0.59
35.08	50 ± 0.59
70.1	100 ± 0.95

NOTE

Within the range of 14.2 percent to full scale, the maximum pointer oscillation shall be 0.48 percent.

8-17. Installation — N1 Tachometer Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Slide the indicator into the face side of the instrument panel and through the supporting clamp.

c. Align the instrument and tighten the upper left 8-32 holding screw.

d. Tighten the lower right 10-32 cinching screw.

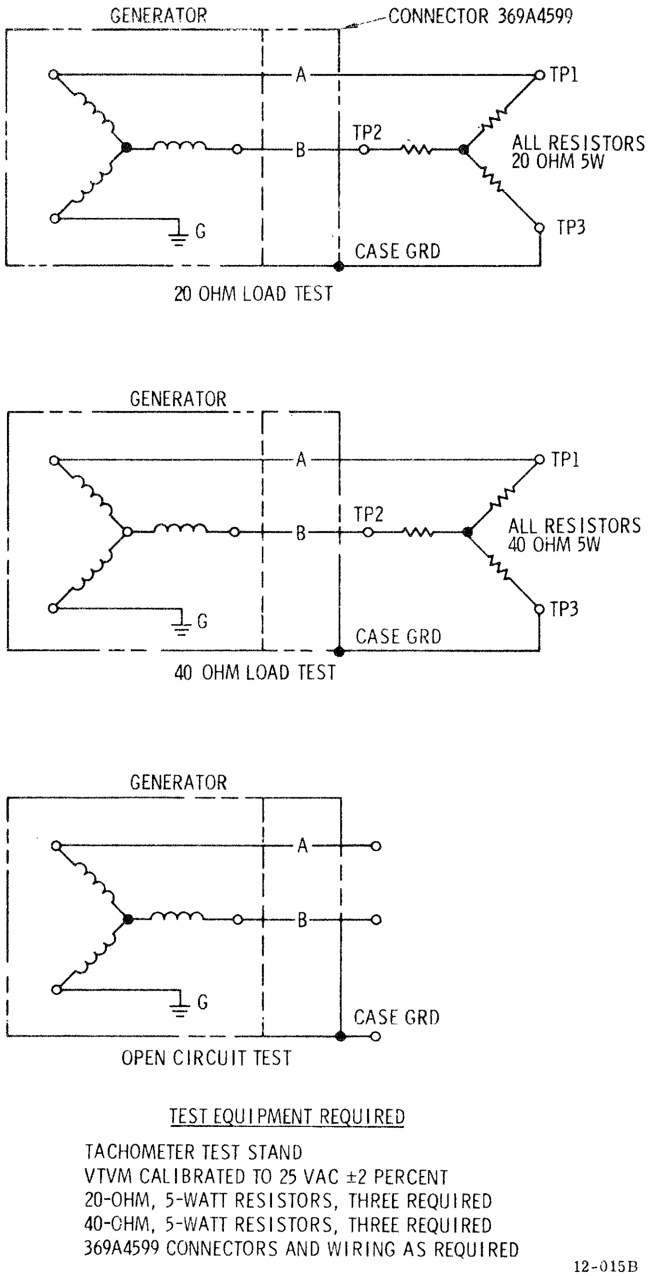


Figure 8-3. Tachometer Generator Test Setup (Typical).

e. Secure the instrument light fixture to the instrument panel with a 6-32 screw.

f. Join the electrical plug and receptacle, couple the plug lock spring and the receptacle lock ring, install the bonding jumper (if used), hand-tighten the hood to secure the lock spring, and tighten the hood clamp screws.

8-18. N2 AND ROTOR TACHOMETER INDICATOR.

8-19. Removal — N2 and Rotor Tachometer Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Loosen the clamp on the connector hood. Unscrew the hood and uncouple the plug lock spring from the receptacle lock ring. Rotate the lock spring in either direction. Remove the connector and the bonding jumper (if used) from the instrument. Install dust covers on the electrical connectors.

c. Remove the four screws, four nuts, and two washers that secure the light fixture and the instrument to the instrument panel.

8-20. Calibration Data — N2 and Rotor Tachometer Indicator (AVIM).

Tachometer Generator rpm	Instrument Indication
2053 - 2137	(N2) 50
4574 - 4642	(N2) 110

NOTE

N2 pointer shall not oscillate more than 0.6 percent.

2639 - 2723	(Rotor) 300
4166 - 4234	(Rotor) 470

NOTE

Rotor pointer shall not oscillate more than 2.2 rpm.

8-21. Installation — N2 and Rotor Tachometer Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Secure the instrument to the instrument panel with four screws, four nuts, and two washers.

c. Join the electrical plug and receptacle, couple the plug lock spring and the receptacle lock ring, install the bonding jumper (if used), hand-tighten the hood to secure the lock spring, and tighten the hood clamp screws.

8-22. TURBINE OUTLET TEMPERATURE INDICATING SYSTEM.

8-23. Description — Turbine Outlet Temperature Indicating System. The turbine outlet temperature indicating system has four chromel-alumel single junction thermocouples (refer to TM 55-2840-231-24) in the gas producer turbine outlet and an associated integral harness. The voltages of the four thermocouples are

Table 8-2. Troubleshooting the N1, N2, and Rotor Tachometer Indicating Systems.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Tachometer indicator pointer fluctuates or remains at zero.**

STEP 1. Using a jumper wire, electrically connect the corresponding tachometer generator case to any electrically grounded portion of the aircraft.

If the instrument indicates properly, remove and install the tachometer generator ground wire. Refer to paragraph 8-9. Refer to TM 55-1500-204-25/1 for grounding surface preparation. If no ground wire was previously installed, install one.

STEP 2. Using a jumper wire, electrically connect the tachometer indicator plug ground wire to an additional grounding location.

If the instrument indicates properly, repair the defective ground connection. Refer to TM 55-1500-204-25/1 for method.

STEP 3. Examine all wires and connections from the tachometer generator to the indicator for continuity and absence of shorts.

If the wires and connectors are not at fault, replace the tachometer generator (para 8-9).

If replacement of the tachometer generator fails to correct the malfunction, replace the indicator (para 8-18).

electrically averaged in the assembly and delivered by the assembly lead to an engine terminal block (refer to chapter 4) for attachment to the airframe harness. The airframe harness extends from the terminal block to the turbine outlet temperature indicator on the instrument panel (fig. 8-1). The indicator is calibrated in major units from 0-12, read as degrees centigrade times 100 (0-1200 degrees centigrade). On aircraft with an adjustable resistor in the airframe harness, the resistor is mounted either on the right side of the center beam below the pilot's floor or in back of the instrument panel.

8-24. Inspection — Turbine Outlet Temperature Indicating System. (See fig. 8-1.) Refer to wiring diagram in appendix F.

a. Examine the airframe harness connections for security.

b. Examine the airframe harness for security and damage.

c. Examine the instrument case and glass for damage.

d. Examine the instrument face for legibility of markings and for cracked or peeling paint on the pointer. Verify that range markings are legible and correctly placed. Refer to TM 55-1520-214-10 for location. Refer to TM 55-6600-200-20 for marking application.

8-25. Troubleshooting — Turbine Outlet Temperature Indicating System. Refer to table 8-4 and the wiring diagram in appendix F.

8-26. TURBINE OUTLET TEMPERATURE INDICATOR.

8-27. Removal — Turbine Outlet Temperature Indicator. Removal procedures for the turbine outlet temperature indicator are identical to those for the N1 tachometer indicator (para 8-15) excepting the electrical connections:

a. Disconnect the airframe harness leads from the indicator.

b. Short the indicator circuit electrically connecting the two terminal posts with safety wire (or the equivalent) to prevent extreme pointer movement during handling.

Table 8-3. Premaintenance Requirements for Adjusting the Turbine Outlet Temperature Indicator.

Conditions	Requirements
Test Equipment	(T1)
Minimum Personnel Required	One (MOS 68F)
Consumable Materials	(C13)

Table 8-4. Troubleshooting the Turbine Outlet Temperature Indicating System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Indicator fails to respond to turbine outlet heat, or the indication is intermittent.**

STEP 1. Verify that all electrical connections, from the terminal block on the engine to the indicator, are secure; and that the wiring and the resistor are free of breaks and shorts.

If the malfunction persists, replace the indicator (para 8-26).

2. **Erroneous indications are suspected.**

STEP 1. Verify that all electrical connections, from the terminal block on the engine to the indicator, are secure, and that the wiring and the resistor are free of breaks and shorts.

If the malfunction persists, adjust the indicator (para 8-26).

If the indicator cannot be adjusted, replace it (para 8-26).

STEP 2. Adjust the airframe harness resistance. Refer to paragraph 8-31.

If the malfunction persists, test the thermocouple harness. Refer to TM 55-2840-231-24.

8-28. Adjustment — Turbine Outlet Temperature Indicator. The following procedure is required after wire leads have been repaired or replaced, at engine change, prior to turbine over-temperature inspection, after excessive turbine outlet temperature, and at each periodic inspection.

a. Set the degree scale on the Jet Cal analyzer (T1) to indicate a test temperature of 750°C. The temperature variation between the analyzer setting and the turbine outlet temperature indicator must be within plus or minus 5°C. Turbine outlet temperature indicators may be adjusted within limits by turning the zero adjusting screw, located on the back of the indicator.

b. Increase the degree scale of the analyzer to indicate a test temperature of 1000°C. The temperature variation between the analyzer setting and the turbine outlet temperature indicator must be within plus or minus 20°C.

NOTE

If the turbine outlet temperature indicator reading is in excess of allowable tolerance, replace indicator.

c. After completion of an adjustment, apply a small amount of silicone rubber adhesive (C13) to the sides of the indicator zero adjustment screw.

8-29. Test — Turbine Outlet Temperature Indicator. Using the Jet Cal analyzer (T1) perform an operational test on the turbine outlet temperature indicator using instructions outlined in TM 55-4920-244-14.

8-30. Installation — Turbine Outlet Temperature Indicator. (See fig. 8-1.) Installation procedures for the turbine outlet temperature indicator are identical to those for the N1 tachometer indicator (para 8-14) excepting electrical connections. If the terminals of the turbine outlet temperature indicator were shorted per instructions in paragraph 8-27, remove the shorting wire after the instrument is secured to the panel, but prior to installing the harness leads. On aircraft having white and green wires, white lead goes on the (+) terminal, the green lead on the (-) terminal. On aircraft having yellow and red wires, the yellow lead goes on the (+) terminal and the red lead goes on the (-) terminal.

8-31. TURBINE OUTLET TEMPERATURE INDICATOR HARNESSES.

8-32. Description — Turbine Outlet Temperature Indicator Harnesses. The airframe harness is routed from a terminal block on the engine (refer to chapter 4) to the turbine outlet temperature indicator (para 8-25). It is essential that the resistance between the two wires in the harnesses be maintained at 8 ± 0.05 ohms. On aircraft not having a wire-wound resistor in the circuit, a greater than optimum resistance may be corrected by replacing terminals and shortening the airframe harness wires, provided such shortening does not reduce the harness length by more than 12 inches. A less-than-optimum resistance indicates that the airframe harness shall be replaced. The adjustment procedures that follow apply to aircraft that have a wire-wound resistor in the circuit. A test and an adjustment is required after harness wires have been replaced or repaired, at engine change, prior to overtemperature inspection (refer to TM 55-2840-231-24), after excessive turbine outlet temperature, and at each periodic inspection. The thermocouple harness is integral to the engine (refer to TM 55-2840-231-24).

Table 8-5. Pre-maintenance Requirements for Adjusting the Turbine Outlet Temperature Indicator Harness Resistance.

Conditions	Requirements
Test Equipment	(T1)
Minimum Personnel Required	One (MOS 68F)
Consumable Materials	(C93)

8-33. Adjustment — Turbine Outlet Temperature Indicator Harnesses. Harness resistance adjustment is done by altering a resistor. On series 1 and 2 aircraft, the resistor (series 1 aircraft: R22. Series 2 aircraft: R200) is located on the center beam near the aft end of the right electronics compartment. On series 3 aircraft, the resistor R200 is located on the back of the instrument panel.

a. Disconnect the airframe harness leads from the rear of the turbine outlet temperature indicator and connect the Jet Cal analyzer (T1) between the leads. Measure the circuit resistance. The meter reading should be 8 ±0.05 ohms.

b. If the reading is too high, unsolder one of the resistor leads from its terminal lug. Unwind and shorten the resistor wire until the correct reading is obtained with the resistor wire connected to the lug. Connect the resistor to the terminal lug using solder (C93).

c. If the reading is too low, unsolder both leads of the resistor in use and stow by wrapping the ends on the adjacent post. Unwrap the wire ends of the spare resistor and solder (C93) the inside end to one lug. Adjust resistance by shortening the outside end of the resistor winding until the correct meter reading is obtained as in *a* above.

NOTE

Normally, only one resistor provides the necessary circuit resistance. However, should the need arise, both resistors could be adjusted accordingly.

c. Disconnect the Jet Cal analyzer (T1) and reconnect the airframe harness leads to the turbine outlet temperature indicator. On aircraft having white and green wires, the white lead goes on the (+) terminal, the green lead on the (–) terminal. On aircraft having yellow and red wires, the yellow lead goes on the (+) terminal and the red lead goes on the (–) terminal.

8-34. Test — Thermocouple Harness. Test the thermocouple harness (the harness interconnecting the four thermocouples) as outlined in TM 55-2840-231-24.

8-35. ENGINE TORQUE INDICATING SYSTEM.

8-36. Description — Engine Torque Indicating System. The torque gage (fig. 8-1) indicates engine torque supplied to the engine output shaft. The torque gage is a direct-reading instrument that indicates in pounds-per-square-inch, equivalent to the amount of pressure (torque) exerted on the accessory gear train by the turbine output shaft. Oil is routed from the torquemeter pressure port of the engine to the gage. The gage pointer is driven by a bourdon tube mechanism.

8-37. Troubleshooting — Engine Torque Indicating System. Refer to table 8-6.

8-38. System Bleeding — Engine Torque Indicating System. After loosening or replacing any plumbing component in the engine torque indicating system, bleed the system as follows.

a. Start and operate the engine (refer to TM 55-1520-214-10).

b. Wrap cloth around the fitting at the back of the torque gage.

c. Loosen, but do not remove, the nylon tube connector at the back of the torque gage.

d. Allow engine oil to drip into the cloth until no air bubbles are visible in the nylon tube (inspect all accessible length of the tube).

e. Tighten the tube connector at the back of the torque gage and wipe up any spilled oil.

f. Shut down the engine (TM 55-1520-214-10).

g. Check the engine oil level and replenish if necessary (refer to chapter 1).

Table 8-6. Troubleshooting the Engine Torque Indicating System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Gage pointer reacts sluggishly to obvious changes in output torque.**

STEP 1. Inspect the plumbing, from the engine to the gage, for leaks and kinked tubing.

If no leaks or tubing distortions exist, bleed the plumbing (para 8-38).

If the malfunction persists, replace the gage (para 8-39).

2. **Gage pointer remains at zero during engine operation.**

STEP 1. Look for a major leak or a severe kink in the plumbing between the engine and the gage.

If no leak or kink exists, bleed the plumbing (para 8-38).

If an attempt to bleed the plumbing reveals a scarceness or absence of oil in the plumbing, investigate oil stoppage at the reducer or snubber at the engine torquemeter pressure port. Refer to chapter 4.

If the malfunction persists, replace the gage (para 8-39).

3. **Incorrect or oscillating gage indication.**

STEP 1. Inspect the plumbing, from the engine to the gage, for leaks.

If no leaks exist, bleed the plumbing (para 8-38).

If bleeding the plumbing fails to correct the malfunction, replace the gage (para 8-39).

8-39. ENGINE TORQUE GAGE.

8-40. Inspection — Engine Torque Gage. (See fig. 8-1.) *a.* Inspect the torque gage case for dents and glass for cracks.

b. Inspect the torque gage fitting for damage that could prevent a leak-tight connection.

c. Inspect markings and numerals for legibility, and the pointer for cracked or peeling paint.

d. Inspect the range markings for legibility and correct limits. Refer to TM 55-1520-214-10 for location. Refer to TM 55-6600-200-20 for marking application.

8-41. Removal — Engine Torque Gage. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Disconnect the nylon tube from the torque gage and cap the gage fitting and the tube.

c. Open the hood of the light fixture, swing out the lamp holders, and remove the mounting screws, washers, and the gage.

8-42. Installation — Engine Torque Gage. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Position the gage and the light fixture on the instrument panel and secure them with mounting screws and washers.

c. Close the light fixture.

NOTE

Alternate fittings, provided with either an 11/16-inch or a 1/2-inch hex, may be installed in the gage. If replacing a gage with a fitting having a 1/2-inch hex, be sure to use a washer seal under the hex in addition to the preformed packing (O-ring) gasket.

d. Remove caps from the tube and the torque gage fittings and connect the tube to the gage.

8-43. ENGINE TORQUE INDICATING SYSTEM PLUMBING.

8-44. General — Engine Torque Indicating System Plumbing. Three types of tubing are used to route oil from the engine to the torque gage on the instrument panel: corrosion-resistant steel, aluminum alloy, and nylon. Installation details are shown in figure 8-4.

8-45. ENGINE TORQUE INDICATING SYSTEM NYLON TUBE.

8-46. Removal — Engine Torque Indicating System Nylon Tube. (See fig. 8-4.) *a.* Set the power selector switch at OFF.

b. Disconnect the tube from the fitting on the torque gage.

c. Cap the openings in the torque gage and the tube.

d. Remove the tube by releasing or cutting the nylon straps along the center canopy frame and disconnecting the tube nut at the upper frame of canted station 78.50 (details A, B, and C, figure 8-4). Cap both tubes.

8-47. Installation — Engine Torque Indicating System Nylon Tube. (See fig. 8-4.) *a.* Set the power selector switch at OFF.

b. Remove protective caps; then connect the tube to the torque gage.

c. Route the tube along the center canopy frame and connect the tube to the bulkhead fitting at the upper frame of canted station 78.50. Secure the tube in place alongside the pitot static tube, using the canopy frame slots and nylon straps, or tying twine (C112) if straps are not available.

d. Bleed the system (para 8-38).

8-48. ENGINE TORQUE INDICATING SYSTEM ALUMINUM TUBE.

8-49. Removal — Engine Torque Indicating System Aluminum Tube. (See fig. 8-4.) *a.* Remove the left forward air intake fairing and the left aft bulkhead access cover in the cargo compartment.

b. Remove the screws, clamps, nuts, and washers that secure the tube along the left side of the mast support structure and at the left side of the air inlet (plenum) chamber (details D, E, and F. fig. 8-4).

c. Disconnect the tube nuts at the bulkhead fittings. Cap the ends of the tube.

8-50. Installation — Engine Torque Indicating System Aluminum Tube. (See fig. 8-4.) *a.* Position the tube and connect both ends to bulkhead fittings.

b. Wrap the tube where it is routed past the aft corner of the mast support structure and into plenum chamber, using tape (C106) to prevent chafing between the tube and the structure.

c. Secure the tube along the left side of the mast support structure and at the left side of the plenum chamber with screws, clamps, nuts, washers, and standoff (details D, E, and F, fig. 8-4).

d. After bleeding the system (para 8-38) install the left forward air intake fairing and the left aft bulkhead access cover in the cargo compartment.

8-51. ENGINE TORQUE INDICATING SYSTEM STEEL TUBE.

8-52. Removal — Engine Torque Indicating System Steel Tube. (See fig. 8-4.) *a.* Gain access through the engine compartment left access door.

b. Remove the screw, clamp, nut, and washer (detail E, fig. 8-4).

c. Disconnect the tube nuts at the plenum chamber pan fitting and at the engine torquemeter fitting. Install protective cap on fitting.

8-53. Installation — Engine Torque Indicating System Steel Tube. (See fig. 8-4.) *a.* Position the tube; then remove protective caps and connect it to plenum chamber pan fitting and the engine torquemeter fitting.

b. Secure the tube to the left side of the plenum chamber with screw, clamp, washer and nut (detail E, fig. 8-4).

c. After bleeding the system (para 8-38) secure the engine compartment access door.

8-54. INSTRUMENT CLUSTER (ENGINE OIL TEMPERATURE, ENGINE OIL PRESSURE, AMMETER).

8-55. Description — Instrument Cluster (Engine Oil Temperature, Engine Oil Pressure, Ammeter). The instrument cluster three-pack (fig. 8-1) is an instrument assembly with indicators that register generator output in amperes, engine oil pressure in pounds per square inch, and engine oil temperature in degrees centigrade. The aircraft electrical system voltage of 28 vdc is reduced to a nominal 14.5 vdc by resistor R2 for operation of the engine oil pressure and the engine oil temperature indicators. Resistor R2 is mounted inside the electrical console on series 1 aircraft, under the pilot's right side on series 2 aircraft, and in back of the instrument panel on series 3 aircraft. The senders for the engine oil pressure and engine oil temperature indicators each function as a variable resistance leg of

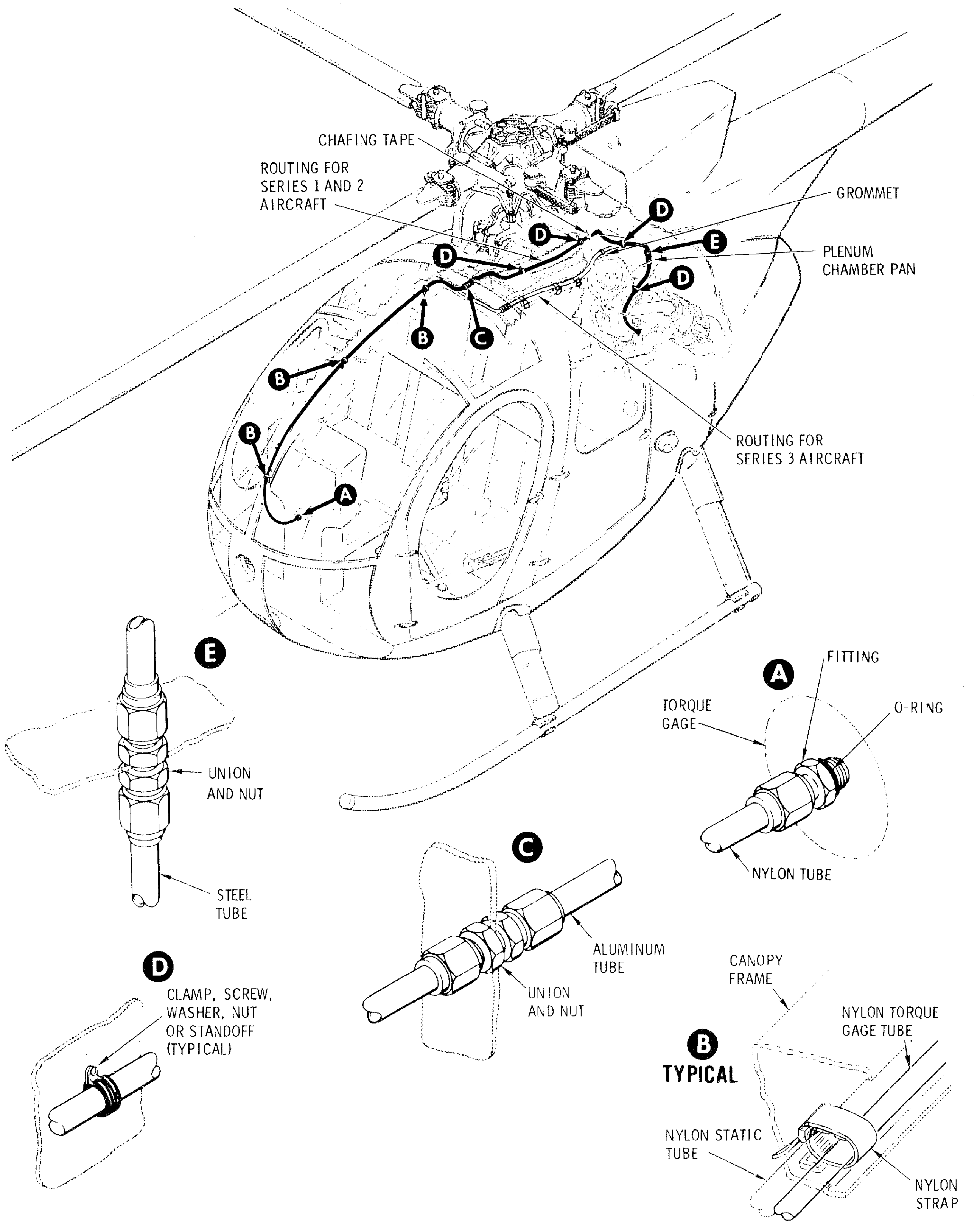


Figure 8-4. Engine Torque Indicating System Plumbing.

the indicating system bridge circuit. As pressure or temperature change, the sender resistance changes and causes an imbalance in current flow through the indicator windings to produce pointer movement. Ammeter pointer deflection is a result of current flow through resistor R16. Resistor R16 is mounted next to the reverse current relay in the engine compartment on series 1 aircraft, under the pilot's right side floor next to the seat bulkhead on series 2 aircraft, and in back of the instrument panel on series 3 aircraft. Resistor R16 is installed in the low voltage output (ammeter) circuit of the starter-generator to cause ammeter pointer movement that is proportional to the main power output of the generator or to the power consumption of the starter. Panel light brilliance is controlled by the indicator light dimmer assembly. Resistor R17 at the back of the cluster is used to keep cluster panel light brilliance about the same as other panel lights. The oil pressure and temperature senders are discussed in chapter 4 as part of the oil system.

8-56. Inspection — Instrument Cluster (Engine Oil Temperature, Engine Oil Presure, Ammeter). (See fig. 8-1.) *a.* Examine the instrument cases for damage and the glass for cracks.

b. Inspect the electrical terminals and studs for secure attachment.

c. Inspect the light sockets in the mounting plate for secure attachment.

d. Inspect range markings for legibility and correct placement. Refer to TM 55-1520-214-10 for location. Refer to TM 55-6600-200-20 for marking application.

8-57. Troubleshooting — Instrument Cluster (Engine Oil Temperature, Engine Oil Pressure, Ammeter). Refer to chapter 9 for troubleshooting the ammeter circuit. Refer to table 8-7 and the wiring diagram in appendix F to troubleshoot the oil system indicator circuits.

8-58. Removal — Instrument Cluster (Engine Oil Temperature, Engine Oil Pressure, Ammeter). (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Disconnect the electrical connector in the cluster harness as follows. Loosen the wiring clamp of the connector hood, hand-loosen the hood, and uncouple the plug lock spring from the receptacle lock ring by turning the lock spring either way.

CAUTION

Use care during removal of the instrument cluster to avoid damaging adjacent electrical units.

c. Remove the two screws, nuts, and washers that secure the instrument cluster to the instrument panel and remove the cluster.

d. If the cluster is to be replaced by another cluster, remove the wiring harness for use on the replacement cluster.

8-59. Installation — Instrument Cluster (Engine Oil Temperature, Engine Oil Pressure, Ammeter). (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. If installing a cluster other than the one removed, attach the harness wiring to the proper terminal studs (refer to wiring diagram in appendix F).

CAUTION

Use care during installation of the instrument cluster to avoid damaging adjacent electrical units.

c. Mount and secure the cluster with two screws, washers, and nuts.

d. Join the harness electrical connectors and secure as follows. Turn the plug lock spring to engage the receptacle lock ring, hand-tighten the hood to secure the lock spring, and tighten the hood clamp screws.

Table 8-7. Troubleshooting the Instrument Cluster (Engine Oil Temperature and Engine Oil Pressure).

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Instrument indicates erroneously, intermittently, or not at all.**

STEP 1. Examine wiring connections from power source to sender to instrument for security. Examine wiring for breaks and shorts.

If wires and connections are not at fault, replace the sender. Refer to chapter 5.

If replacement of the sender fails to correct the malfunction, replace the instrument cluster (para 8-54).

SECTION III FLIGHT INSTRUMENTS

8-60. Reference Information — Flight Instruments. Refer to TM 55-1500-204-25/1 for general maintenance instructions pertaining to flight instruments and the pitot-static system. Refer to appendix A for other publications applicable to the overhaul of flight instruments.

8-61. PITOT-STATIC SYSTEM.

8-61A. Description — Pitot-Static System. The pitot-static system (fig. 8-5) consists of a pitot tube assembly, a static port installation, tubing, and fittings. The pitot tube, mounted near the centerline of the canopy panel, is routedc to the airspeed indicator. The static port located in the aft section of the air inlet aft fairing, is connected by a nylon tube to the altimeter. A static crossover tube connects the altimeter and the airspeed indicator. On aircraft with a heated pitot tube, the pitot-heat switch-circuit breaker is located on the instrument panel near the engine oil pressure indicator. (See fig. 8-1.)

CAUTION

Do not blow air through the pitot-static system lines without disconnecting the altimeter and airspeed indicator. Instantaneous air pressure may result in distortion or rupture of the instrument diaphragm.

Table 8-8. Premaintenance Requirements for Troubleshooting of Heated Pitot Tube.

Conditions	Requirements
Minimum Personnel Required	One (MOS 68F)

8-62. Functional Check — Altimeter, and Airspeed — Pitot Static System. (Removing the instruments from the helicopter is not required for performing the functional check.)

CAUTION

Do not apply suction to Pitot lines or pressure to Static lines; except as instructed in step c.

NOTE

Assure Pitot Static Tester has a current calibration lable (DA Form 80). Use appropriate power supply in accordance with Pitot Static Tester requirements.

a. Pitot Line Leak Check. See figure 8-5A.

(1) Seal pitot tube drain holes air tight with pressure sensitive taqpe (C102).

(2) Hook up Pitot Static Tester, Airspeed Outlet, to Pitot System. Close Pitot Static Tester pressure down valve.

CAUTION

The valves on the test set are sensitive and should be operated slowly and with care to avoid possible damage to instrument. The instrument is easily pegged and damaged by improper manipulation of the valves on this test set.

(3) Slowly apply pressure to pitot line until the airspeed indicator reads 100 knots.

(4) Tap instrument to remove friction effects. When indicator pointer drops more than 10 mph (8.7 knots) in one minute, a leak is indicated. Slowly decrease pressure to return tester airspeed indicator to zero, repair any faults if necessary and repeat above steps.

b. Airspeed Indicator Functional Check.

(1) Slowly apply pressure to pitot line to obtain airspeed readings in table 8-11A; indicator should be gently tapped prior to reading; check need not exceed 120 knots.

(2) If readings are not within tolerance of table 8-11A, slowly relieve pressure on pitot line until airspeed indicator reads zero. Replace indicator with serviceable one and repeat steps b (1) and (2).

(3) Airspeed indicators that fail check shall be turned in for overhaul.

CAUTION

Hooking up the pitot line to the tester vacuum source will only be done during the following check to equalize the pressure in the airspeed indicator case to prevent damage to its diaphragm.

CAUTION

Assure that both aircraft pitot and static lines are connected to airspeed indicator to prevent possible damage to airspeed indicator.

c. Static Line Leak Check.

(1) Tape all unused static ports.

(2) Hook up Pitot Static Tester, rate of climb and altimeter outlet to Pitot and Static System. See figure 8-5B.

(3) Adjust test set and aircraft altimeter barometric scales to read 29.92, gently tap altimeters, and check to insure that aircraft altimeter reads within 70′ of test set altimeter and calibration data card located across top of tester outlet valve cover.

(4) If aircraft altimeter failes to meet this check, remove and install a serviceable instrument and repeat step c.(3).

(5) Slowly apply vacuum to pitot and static lines (fig. 8-5B) until the altimeter reads 1000′ above reading obtained from step (3). Close vacuum source and gently tap altimeter at same time until rate of climb indicator stabilizes at zero.

(6) After stabilization, the altimeter should not drop more than 100 feet in one minute. Slowly decrease vacuum until rate of climb stabilizes at zero. Repair faults if necesssary and repeat this step.

d. Altimeter Function Check.

(1) Apply 28 volt power to aircraft electrical system to provide electrical power to counter drum altimeter vibrators.

(2) Slowly apply vacuum to aircraft system at a rate not to exceed 3,000 feet per minute. Continue to apply vacuum until altimeter readings reach the next higher reading on table 8-11B above altimeter altitude indicated when barometric scale is set at 29.92.

(3) Close vacuum valve and keep altimeter at this setting for at least one minute, but not more than five minutes, then gently tap three pointer type altimeters before reading. Assure that vibrators are operative on counter drum altimeters before readings are compared to tolerance. Refer to table 8-11B.

(4) Compare the readings of the installed instruments against readings on the tester and calibration data card at the check points. Refer to table 8-11B.

NOTE

Altimeters that do not pass the performance requirements of table 8-11B shall be removed from aircraft and shop tested in accordance with the test procedures printed in TM 55-1500-204-25/1 to verify malfunctions. Install serviceable altimeter and repeat steps (1) thru (4).

(5) Remove Pitot Static Tester and all tape from pitot/static lines and openings.

NOTE

Do not adjust altimeter through disengagement of the baroset locking screw (commonly called rezeroing) while the altimeter is installed in helicopter.

8-63. Troubleshooting — Pitot-Static System. Refer to paragraphs 8-81 and 8-88 for pitot-static system malfunctions other than pitot-tube heating. For pitot-tube heater circuit malfunctions, refer to table 8-10 and the wiring diagram in appendix F.

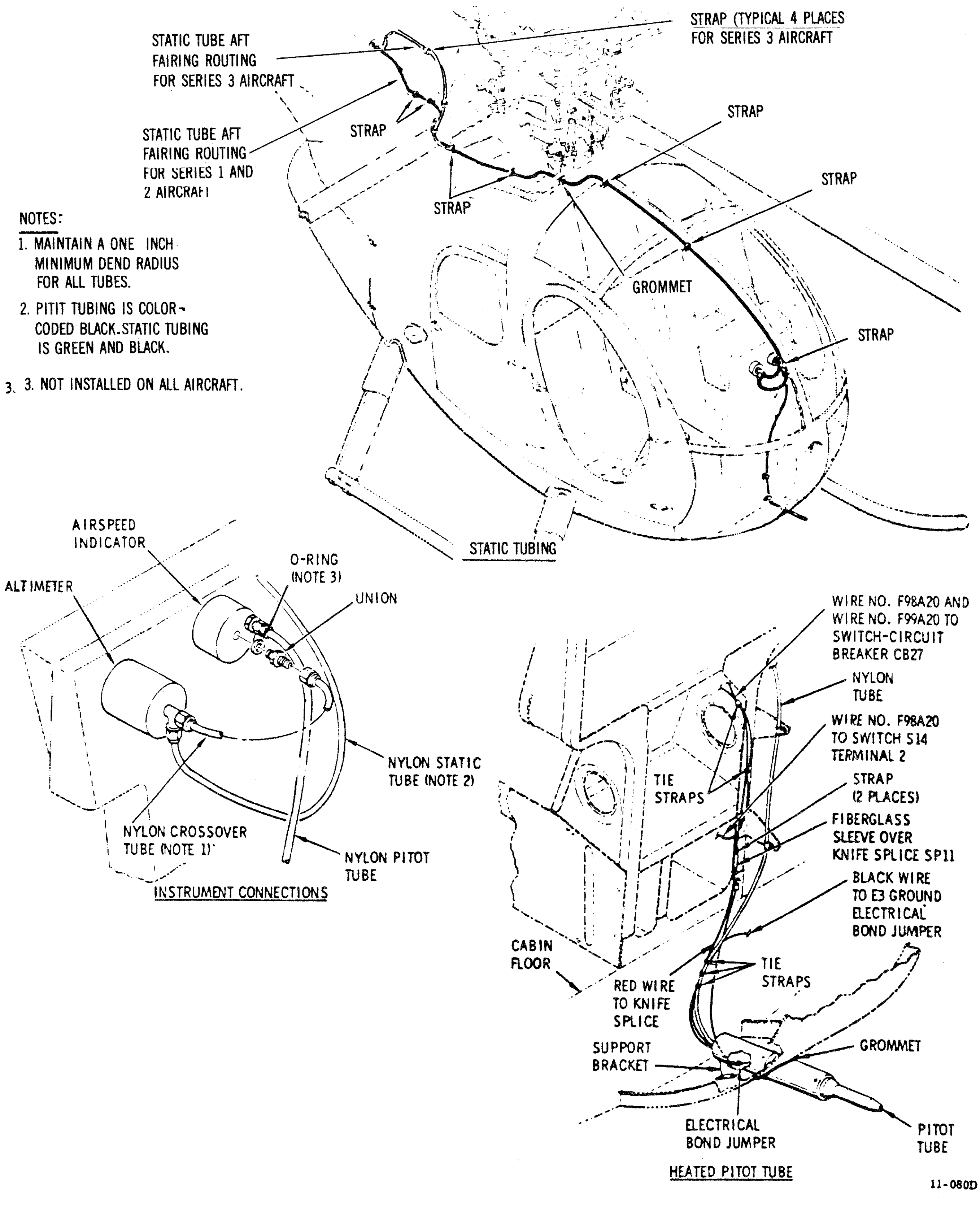


Figure 8-5. Pitot-Static System.

Table 8-10. Troubleshooting the Pitot-Tube Heating Circuit.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Pitot tube fails to heat when the selector switch is set at EXT or BAT and the pitot heat switch-circuitbreaker is set at PITOT HEAT.**

STEP1. Examine wire connections at the power selector switch, the pitot heat switch-circuitbreaker CB27, the knife splice SP11, and at grounding terminal E3 for security. Examine the positive wires for breaks and shorts. Examine the ground wire (the black wire terminating at E3) for breaks.

If the wiring and connections are not at fault, jumper the terminals of the pitot heat switch-circuitbreaker CB27.

If jumpering the pitot heat switch-circuitbreaker CB27 eliminates the malfunction, replace the switch-circuitbreaker.

If jumpering the pitot heat switch-circuitbreaker CB27 fails to correct the malfunction, replace the pitot tube assembly (para 8-75).

8-64. Adjustment — Pitot-Static System. No adjustment is required for a heated pitot tube.

8-65. STATIC TUBES AND PORT.

8-66. Description — Static Tubes and Port. The aft fairing routing for series 1 and 2 aircraft differs from that of series 3 aircraft as shown in figure 8-5. All static tubing is color-coded green and black.

8-67. Inspection — Static Tubes and Port. (See fig. 8-5.) *a.* Examine tubing for any damage that could cause a leak.

b. Examine tubing for security of attachment by straps and for tight bends **(MINIMUM BEND RADIUS SHALL BE ONE INCH).**

c. Examine tubing connections for security of attachment at the static port and at the altimeter and airspeed indicator.

d. Examine the static port for foreign matter.

8-68. Removal — Static Tubes and Port. (See fig. 8-5.)

CAUTION

When maintenance work is being performed near the engine air inlet, use care to prevent entry of foreign objects. On aircraft without a filter installed, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove covers until work is completed and debris is thoroughly cleaned out of the area. After removing covers, verify that the area around the base of the mast, the inlet to the plenum, and the entire plenum chamber is free of foreign material. Install the plenum access doors.

a. On aircraft with an engine barrier filter, remove the plenum chamber access door. On aircraft with an inertial particle-separator air filter, open the bypass door on the engine air inlet aft fairing (refer to chapter 2).

b. Remove the static tube from the air inlet aft fairing by breaking the adhesive bondline.

c. Remove the nylon straps that attach the static tube along its entire length.

d. Disconnect the static tube from the altimeter fitting. Cap the fitting.

e. Pull the static tube out of the bulkhead grommets.

f. Remove the crossover tube from the altimeter and the airspeed indicator fittings. Cap the fittings.

8-69. Cleaning — Static Tubes and Port. (See fig. 8-5.) *a. Static Port Tube.*

(1) Disconnect the tube at the altimeter and cap the altimeter fitting.

(2) Apply pressurized, dry, filtered air to the tube until contamination is removed from the tube and the static port.

(3) Connect the tube to the altimeter fitting.

b. Crossover Tube.

(1) Disconnect the tube from the altimeter and the airspeed indicator. Cap the instrument fittings. Remove the tube from the aircraft.

(2) Apply pressurized, dry, filtered air to one end of the tube until contamination is removed.

(3) Connect the tube to the airspeed indicator and the altimeter.

8-70. Installation — Static Tubes and Port. (See fig. 8-5.)

CAUTION

When maintenance work is being performed near the engine air inlet, use care to prevent entry of foreign objects. On aircraft without a filter installed, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove covers until work is completed and debris is thoroughly cleaned out of the area. After removing covers, verify that the area around the base of the mast, the inlet to the plenum, and the entire plenum chamber is free of foreign material. Install the plenum access doors.

a. Route the static tube through grommets in the canted bulkhead and place the aft end of the tube in the fairing port.

b. Attach the static tube along its length to structure with nylon straps.

c. Connect the static tube to the altimeter fitting.

d. Prime and bond the static tube to the air inlet fairing with adhesive (para 2-173). The end of the tube must be flush with the fairing face.

e. Attach the crossover tube to the altimeter and airspeed indicator fittings.

f. On aircraft with an engine barrier filter, install the plenum chamber access door. On aircraft with an inertial particle-separator air filter, close the filter bypass door (chapter 2).

8-71. PITOT TUBE ASSEMBLY (NON-HEATED).

8-72. Cleaning — Pitot Tube Assembly (Non-heated). (See fig. 8-5.) *a.* Disconnect the pitot tube from the airspeed indicator and cap the indicator fitting.

b. Apply pressurized, dry, filtered air to the disconnected end of the tube until contamination is removed.

c. Connect the pitot tube to the airspeed indicator.

8-75. PITOT TUBE ASSEMBLY (HEATED).

8-76. Description — Pitot Tube Assembly (Heated). The heated pitot tube assembly comprises a protruding tube, an electrical heater, a nylon tube for connection to the altimeter, a ground wire terminating at a lug, and a wire terminating at a knife splice SP11. The assembly is secured to a support bracket by four screws and washers. A grommet in the canopy opening seals the juncture of canopy, bracket, and pitot tube assembly.

8-77. Cleaning — Pitot Tube Assembly (Heated). Cleaning of the heated pitot tube and the nylon tube procedures. Refer to paragraph 8-72.

8-78. Removal — Pitot Tube Assembly (Heated). (See fig. 8-5.) *a.* Open the left access door on the pilot's floor.

b. Disconnect the nylon tube from the airspeed indicator and cap the indicator fitting.

c. Remove two clamps, screws, washers, and nuts that attach the nylon tube to the angle and to the station 44.65 bulkhead.

d. Disconnect the pitot tube heater ground wire (black) from aircraft structure ground terminal E3.

e. Loosen two tie straps and slide the fiberglass sleeve upward from the knife splice at the aft end of the pitot tube heater red wire and separate the knife splice.

f. Remove the strap that ties the nylon tube, the red wire, and the black wire to the grounding jumper that connects the pitot tube assembly support bracket to the aircraft structure. Do not remove the jumper.

g. Break the safety wire and remove the four screws and washers that secure the pitot tube assembly to the support bracket.

h. Cut the sealant between the grommet and the pitot tube assembly and pull the pitot tube assembly forward, out of the support bracket and through the canopy grommet.

8-79. Installation — Pitot Tube Assembly. (See fig. 8-5.) *a.* Cap the end of the nylon tube to preclude entry of contamination and insert the nylon tube and two heater wires aft into the grommet and through the bracket.

b. Pull the wires and the nylon tube aft and upward and push the solid portion of the pitot tube assembly into place so that the four attaching holes align with those in the support bracket, making sure the pitot tube moisture drain hole (located just forward of the necked-down portion of the tube) faces downward.

c. Secure the pitot tube assembly with four screws and washers. Secure the screws with lockwire.

d. Connect the nylon tube to the airspeed indicator.

e. Connect the pitot tube heater black wire to aircraft structure ground terminal E3. Refer to TM 55-1500-204-25/1 for grounding surface preparation.

f. Connect the pitot tube heater red wire to the knife splice SP11 on wire No. F99A20 (refer to the wiring diagram in appendix F).

g. Slide the fiberglass sleeve over the knife splice and secure the sleeve with two tie straps.

h. Adjust the nylon tube so that the **MINIMUM BEND RADIUS AT ANY POINT IS ONE INCH** and secure the red and black wires to the nylon tube with two tie straps. With a single tie strap, secure the red and black wires and the nylon tube to the jumper wire that connects the pitot tube assembly support bracket to the aircraft structure.

i. Seal the juncture of the pitot tube assembly and the canopy grommet with silicone rubber adhesive (C13).

j. Close the left access door on the pilot's floor.

8-80. Test — Pitot Tube Assemmbly (Heated) (Operational Check).

WARNING

Exercise caution when touching the pitot tube with fingers as serious burns may result.

a. Set the power selector switch at EXT.

b. Set the pitot heat switch at PITOT HEAT.

c. Check for heat rise by touching the pitot tube lightly with fingers.

CAUTION

At temperatures above freezing, do not leave the pitot heat switch set at PITOT HEAT for longer than two minutes. Excessive heat can damage the pitot tube assembly and adjacent parts of the aircraft.

d. If a heat rise is obtained or if two minutes have elapsed, set the pitot heat switch at OFF and set the power selector switch at OFF. If no heat rise is obtained, refer to paragraph 8-63 troubleshooting.

8-81. ALTIMETER.

8-82. Description — Altimeter. The altimeter (fig. 8-1) presents aircraft altitude in feet above sea level under standard conditions of atmospheric pressure and temperature. The range of the internal mechanism is 0-50,000 feet. Three pointers, concentrically arranged, indicate altitude in increments of 100, 1000, and 10,000 feet. A small scale, located between the 2 and 3 index markings on the instrument face is calibrated in inches of mercury. The scale is rotated by a knob at the lower left of the instrument to equal the ambient barometric pressure. After this adjustment is made, the altimeter will indicate the actual height of the aircraft in feet above sea level.

8-83. Inspection — Altimeter. (See fig. 8-1.) *a.* Inspect the instrument case for dents and the cover glass for slippage and cracks.

b. Inspect the fitting at the back of the instrument for damaged threads that could prevent a leak-free connection. See figure 8-5.

c. Inspect the index markings and numerals on the dial face for discoloration and legibility. Inspect the pointers for cracked or peeling fluorescent paint.

8-84. Deleted.

8-85. Troubleshooting — Altimeter. Refer to table 8-11.

8-86. Removal — Altimeter. (See fig. 8-1.)

WARNING

Instrument contains radioactive material (Refer to para 8-2.)

a. Set the power selector switch at OFF.

b. Pull down on the hood of the instrument light fixture, swing out the lampholders, and remove the mounting screws and washers.

c. Remove screws and loosen the mounting strap on the back side of the panel. Disconnect the static air tubes. See figure 8-5.

d. Remove the altimeter. Cap the altimeter fitting openings and the ends of the disconnected tubes.

8-87. Installation — Altimeter. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

CAUTION

Use antiseize compound sparingly; remove all excess to prevent contamination of interior of instrument and lines. Never apply antiseize compound to female fitting threads.

b. Coat male fitting threads lightly with antiseize compound (C110).

c. Position instrument and the light fixture on the instrument panel and secure them with mounting screws and washers.

d. Close the light fixture.

e. Install the mounting support strap and attaching screws.

f. Connect the static air tubes. See figure 8-5.

g. Perform a functional check whenever an instrument in the pitot/static system is replaced or reinstalled. Refer to paragraph 8-62.

8-88. AIRSPEED INDICATOR.

8-89. Description — Airspeed Indicator. The airspeed indicator (fig. 8-1) is a pressure-sensitive unit actuated by the differential between impact and static air pressures. The impact (pitot) pressure is supplied by the pitot tube to a pressure-sensitive diaphragm in the indicator case. Static air pressure is supplied to the inside of the indicator case.

8-90. Inspection — Airspeed Indicator. (See fig. 8-1.) *a.* Examine the instrument case for damage and the cover glass for damage and slippage (note white slippage mark at the bottom of the glass).

b. Examine the two fittings at the back of the instrument for damaged threads that could prevent a leak-free connection. See figure 8-5.

c. Examine the markings on the dial for discoloration and legibility. Inspect the pointer for cracked or peeling fluorescent paint.

d. Examine the range markings for legibility and correct placement. Refer to TM 55-1520-214-10 for location. Refer to TM 55-6600-200-20 for marking application.

8-91. Troubleshooting — Airspeed Indicator. Refer to table 8-11.

8-92. Removal — Airspeed Indicator. (See fig. 8-1.)

WARNING

Instrument contains radioactive material (Refer to para 8-2).

Table 8-11. Troubleshooting the Altimeter and the Airspeed Indicator.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Altimeter indicates erroneously, intermittently, or not at all. The airspeed indicator functions correctly.**

STEP 1. Check that airspeed indicator functions correctly.

If airspeed indicator functions, replace the altimeter.

2. **Airspeed indicator functions erroneously, intermittently, or not at all whereas the altimeter functions correctly.**

STEP 1. Examine the pitot tube plumbing from the instrument to the forward end of the tube for security of connections and absence of tight bends or kinks, and, for a non-heated tube, check the adjustment (para 8-64).

If the fault is not the connections, the tube routing, nor in the tube adjustment, clean the pitot tube and plumbing to remove any potential obstruction. Refer to paragraph 8-72.

If the malfunction persists, inspect and clean the crossover tube between the altimeter and the airspeed indicator. Refer to paragraph 8-65.

If the malfunction persists, replace the airspeed indicator (para 8-88).

3. **Both the altimeter and the airspeed indicator function erroneously or intermittently.**

STEP 1. Inspect the static tubes and the static port. Refer to paragraph 8-67.

If the malfunction persists after inspection and any remedial maintenance performed as a consequence of inspection, clean all static tubing and the port to remove potential obstructions. Refer to paragraph 8-69.

---

a. Set the power selector switch at OFF.

b. Pull down on the hood of the instrument light fixture, swing out the lampholders, and remove the mounting screws and washers.

c. Remove screws and loosen the mounting strap on the back of the panel. Disconnect the pitot-static air tubes. See figure 8-5.

d. Remove the airspeed indicator. Cap the two fittings at the back of the instrument and cap the two disconnected tubes.

8-93. Installation — Airspeed Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

CAUTION

Use antiseize compound sparingly; remove all excess to prevent contamination of interior of instrument and lines. Never apply antiseize compound to female fitting threads.

b. Coat the airspeed indicator male fitting threads lightly with antiseize compound (C110).

c. Position the instrument and the light fixture on the instrument panel and secure them with mounting screws and washers.

d. Close the light fixture.

e. Install the mounting support strap and the attaching screws.

f. Connect the pitot-static air tubes. Be sure the tubes are connected to the proper fittings. See figure 8-5.

g. Perform a functional check whenever an instrument in the pitot/static system is replaced or reinstalled. Refer to paragraph 8-62.

TABLE 8-11A

AIRSPEED INDICATOR TOLERANCE

AIRSPEED CHECK POINTS (KNOTS)	MS 28045 10 TO 150 (KNOTS)	MS 28021 20 TO 250 (KNOTS)	MS 28048 40 TO 400 (KNOTS)
40	3	5	4
60	3	3	3
80	3	3	3
100	3	4	3
120	3	4	3

TABLE 8-11B

ALTIMETER SCALE ERROR TOLERANCE

ALTITUDE (FEET)	TOLERANCE (FEET)
0	± 70
500	± 70
1,000	± 70
2.000	± 70
3.000	± 70
5,000	±100
10,000	± 130
15,000	± 140

8-94. ATTITUDE GYRO AND BALL BANK INDICATOR.

8-95. Description — Attitude Gyro and Ball Bank Indicator. The attitude gyro and ball bank indicator (fig. 8-1) shows aircraft flight attitude relative to the surface of the earth by means of a horizon bar, a turn index mark, and a reference airplane model. The attitude indication is a result of internal gyro reaction to pitch and roll. The gyro is electrically driven by the 115-vac, 400-Hz, static inverter and will operate when the power selector switch and inverter switch are on. The ball bank slip indicator in the lower part of the instrument face is a conventional ball and tube that shows a skidding or slipping flight attitude when ball is off center.

Table 8-12. Premaintenance Requirements for Troubleshooting Attitude Gyro and Ball Bank Indicator.

Conditions	Requirements
Test Equipment	(T4)
Minimum Personnel Required	One (MOS 68F)

8-96. Troubleshooting — Attitude Gyro and Ball Bank Indicator. Refer to the wiring diagram in appendix F and make sure that wiring to the inverter and between the inverter and the instrument is free of breaks and shorts and that power is being delivered to the gyro connector on the instrument. Series 1 aircraft have two inverters. Refer to chapter 9 for maintenance references pertaining to the alternating-current power distribution system.

8-97. Removal — Attitude Gyro and Ball Bank Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Loosen the clamp on the connector hood. Unscrew the hood and uncouple plug lock spring from the receptacle lock ring; rotate the lock spring in either direction. Remove the connector and the bond jumper (if used) from the indicator.

c. Open the instrument light fixture, swing out the lampholders, and remove the mounting screws and washers.

d. Remove the screws that secure the mounting support strap at the back of the panel.

NOTE

Turn the caging lockscrew at the back of the instrument to avoid damaging the instrument after removal.

e. Remove the indicator.

f. Install dust covers on the electrical connectors.

8-98. Inspection — Attitude Gyro and Ball Bank Indicator. (See fig. 8-1.) *a.* Examine the instrument case for dents and the glass for cracks.

b. Inspect the electrical connector for bent pins, corrosion, and signs of arcing.

c. With the indicator in a normal operating position, inspect the ball bank fluid for visible bubbles. Tilt the ball bank indicator to each side and inspect the glass ball for unrestricted movement and no sticking at either end of the tube.

NOTE

At average temperatures, a small bubble is normal. It allows for liquid expansion at higher temperatures.

8-99. Installation — Attitude Gyro and Ball Bank Indicator. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Remove dust covers from the electrical connectors.

c. Check that the instrument flange has mounting nuts. If missing from indicators with four-hole mounting, install NAS487-21 instrument mounting nuts. Install the instrument and light fixture in the panel and secure them with mounting screws.

NOTE

Unlock the caging mechanism by turning the lock-screw at the back of the instrument.

d. Close the instrument light fixture.

e. Install the mounting support strap and attaching screws. Install the electrical plug to the receptacle; couple the plug lock spring and receptacle lock ring. Install the bond jumper (if used). Hand-tighten the hood to secure the lock spring; then tighten the hood clamp screws.

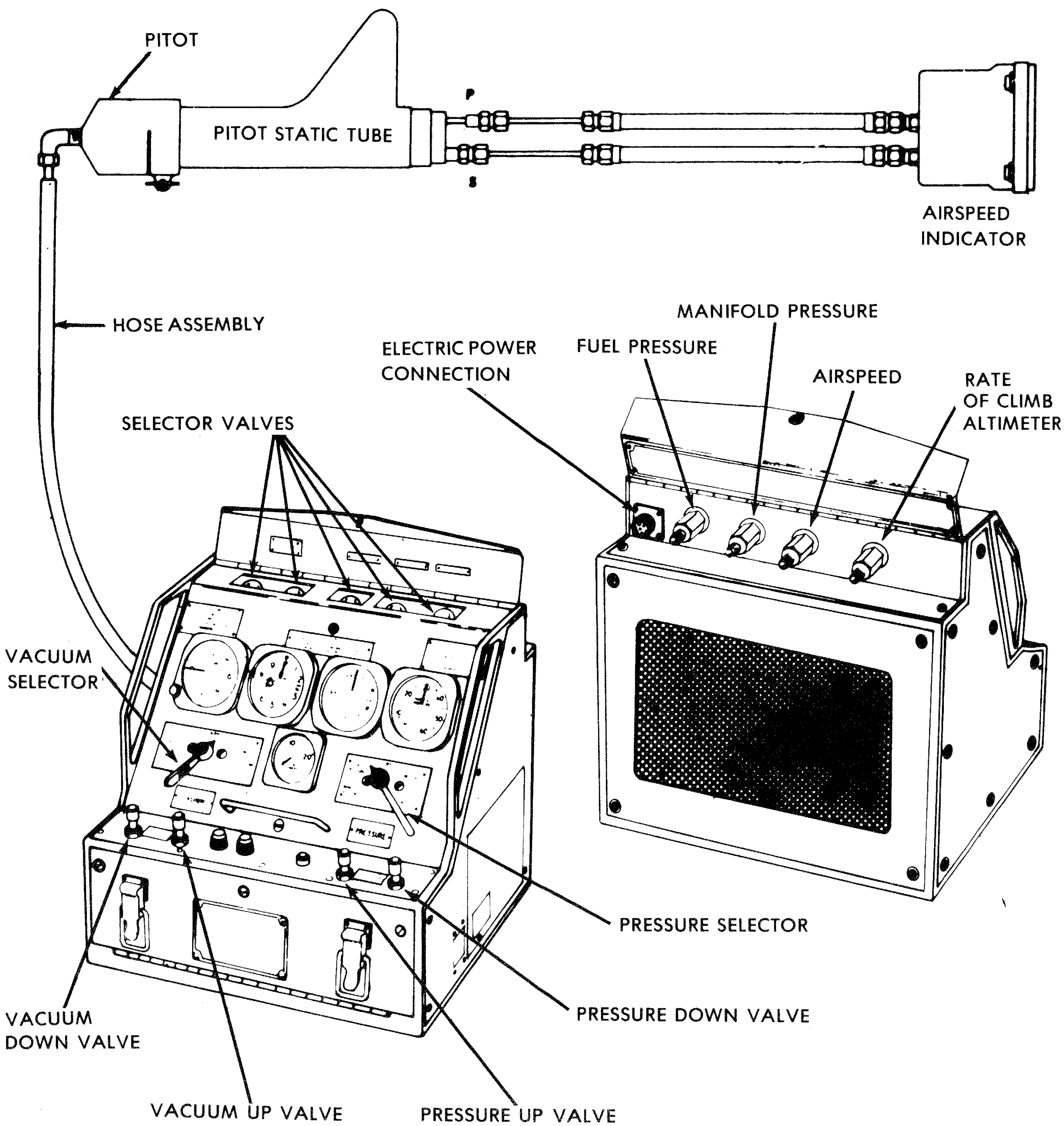


Figure 8-5A. Connection for Pitot Leak Check

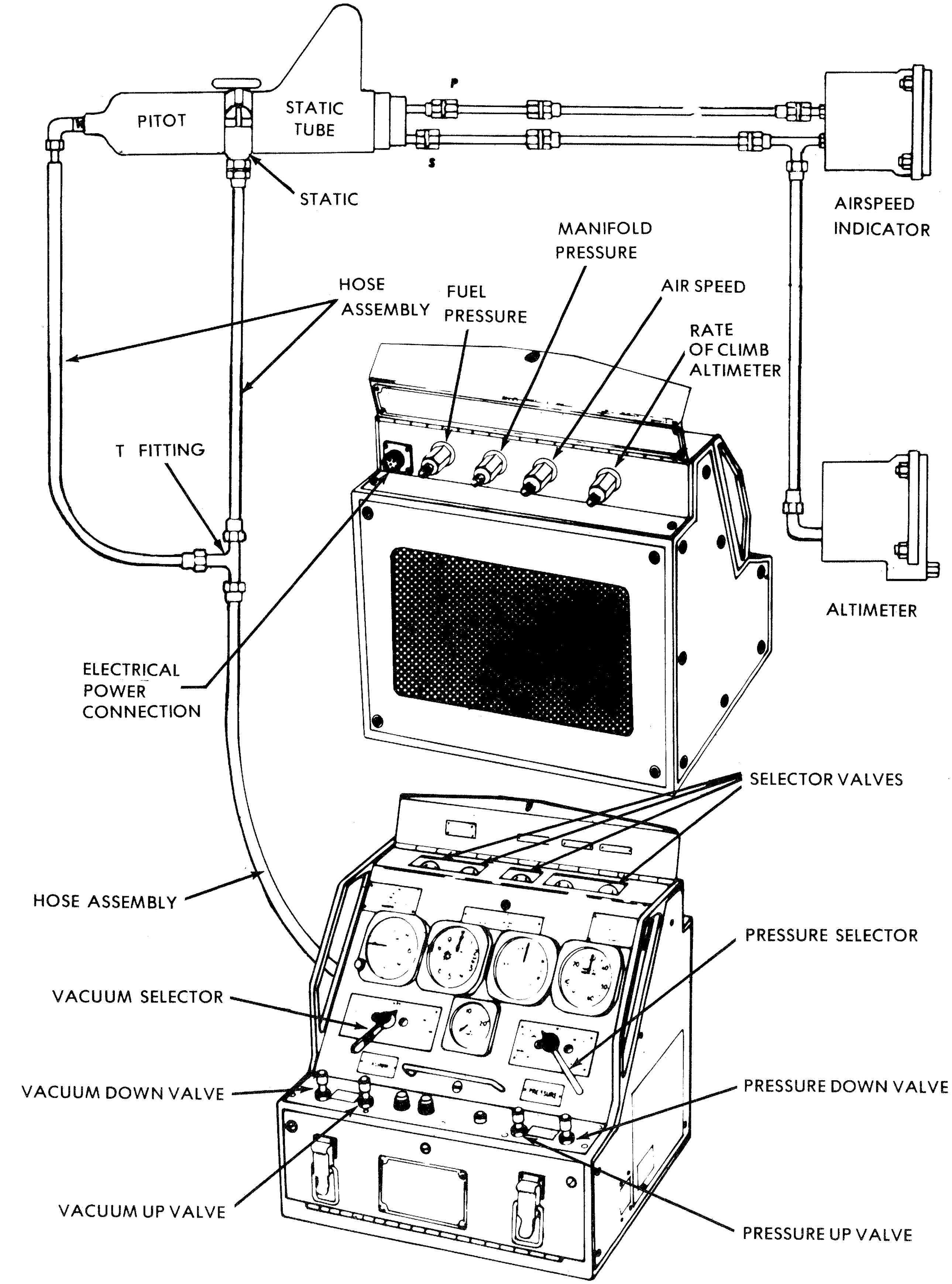


Figure 8-5B. Connections for Static Leak Check

SECTION IV NAVIGATION INSTRUMENTS

8-100. Reference Information — Navigation Instruments. Refer to TM 55-1500-204-25/1 for general maintenance instructions on the navigation instruments. Refer to appendix A for other publications applicable to the overhaul of navigation instruments. Maintenance of navigation instruments associated with the avionics installation is covered in the Electronics Equipment Configuration manuals (TM 11 Series).

8-101. HOMING, HEADING, BEARING INDICATOR.

8-102. Description — Homing, Heading, Bearing Indicator. The homing, heading, and bearing indicator (fig. 8-1) provides precise visual directional references for flying a straight course and making precision turns, and allows homing on a known fm station with visual indications for guidance to the station and when passing over the station.

NOTE

Failure of the inverter will cause loss of compass bearing card segment of the indicator. Refer to TM 11-1520-214-20 or TM 11-1520-214-20-1 for instrument information.

8-103. Inspection — Homing, Heading, Bearing Indicator. (See fig. 8-1.) *a.* Inspect the instrument case for dents and the glass for cracks.

b. Inspect the electrical connector for secure connection and signs or arcing. Inspect the bonding lead (if used) for secure attachment and corrosion.

c. Inspect the support strap and the mounting clamp for secure attachment to instrument panel.

8-104. Troubleshooting — Homing, Heading, Bearing Indicator. Refer to TM 11-1520-214-20 or TM 11-1520-214-20-1.

8-105. MAGNETIC COMPASS.

8-106. Description — Magnetic Compass. The magnetic compass, mounted on the canopy frame above the instrument panel, continuously indicates the heading of the aircraft relative to the earth's magnetic field. The compass consists of a liquid-filled case in which a float assembly, incorporating a compass card and magnets, pivots on a jewel post. The compass card is graduated in 5-degree increments. The card is read against a fixed vertical white line (lubber's line) on the face of the instrument. A compass correction card is located next to the magnetic compass.

8-107. Adjustment — Magnetic Compass. Refer to TM 55-1500-204-25/1 to compensate the compass *(record deviation in 45-degree increments — as listed on the compass correction card).*

8-108. Inspection — Magnetic Compass. *a.* Verify the absence of air bubbles.

b. Inspect the index markings and numerals for legibility.

c. Inspect the case for damage and the window for cracks or discoloration.

8-109. Test — Magnetic Compass (Operational Check). Perform an operation check and adjust compass per paragraph 8-107.

8-110. Troubleshooting — Magnetic Compass. If adjustment (compensation) cannot be obtained (refer to para 8-107), replace the compass (refer to para 8-111 and 8-112).

8-111. Removal — Magnetic Compass. *a.* Remove the light wiring connector.

b. Remove the mounting screws, nuts, washers, and the Vne card bracket from the mounting bracket; remove the compass.

8-112. Installation — Magnetic Compass. *a.* Place the compass in the mounting bracket.

b. Secure the compass and the Vne card bracket with brass mounting screws, nuts, and washers.

c. Install the light wiring connector.

SECTION V MISCELLANEOUS INSTRUMENTS

8-113. CLOCK.

8-114. Description — Clock. The clock (fig. 8-1) is a 22-jewel movement, elapsed-time, spring-driven, 8-day timepiece. The clock is stem-wound by a knob on the lower left side of the dial. The knob, when pulled and turned, also adjusts the hands. The elapsed time hands, controlled by a knob at the upper right side of the dial, record elapsed time in minutes and seconds for up to one hour. The elapsed time control knob can be pushed in to cause any of three timing actions that start with the hands set at 12 o'clock; the first push starts the hands; the second push stops the hands; and the third push resets the hands at 12 o'clock.

8-115. Inspection — Clock. (See fig. 8-1.) *a.* Inspect the clock case for dents and the glass for cracks.

b. Inspect the dial index markings and numerals for legibility, and the clock hands for cracked or peeling paint.

c. Check the winding knob by turning it clockwise and setting the knob by pulling and turning it in either direction.

d. Check the elapsed time knob setting sequence by pushing to start, stop, and reset.

8-116. Removal — Clock. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Unscrew the threaded plastic sleeve connecting the wire to the instrument post light. Loosen and remove the small hex nut that secures the post light and the clock to the panel; then remove the lockwasher, spacer washers, and the post light.

c. Remove the remaining screw and washer, and then the clock.

8-117. Installation — Clock. (See fig. 8-1.) *a.* Set the power selector switch at OFF.

b. Mount the clock in the panel and attack it with the lower screw and washer.

c. Install the post light in the panel and the clock, turn the lamp hood to face the lower mounting screw, and secure it with spacer washers, lockwasher, and hex nut; do not overtighten.

d. Tighten the lower screw, and fasten the wire to the post light by screwing the plastic sleeve clockwise until it is fingertight.

8-118. OUTSIDE AIR TEMPERATURE THERMOMETER.

8-119. Description — Outside Air Temperature Thermometer. The outside air temperature thermometer (fig. 8-6) consists of a bimetal element (enclosed in a stainless steel thermometer stem) attached through a stem to a dial-type temperature indicator. The thermometer is installed through the right side of the canopy just forward of the instrument panel and is secured by a sun shield. The temperature dial has a centigrade scale, calibrated from −75°C to 50°C. Movement of the pointer is caused by a small helical spring that consists of two dissimilar alloys welded together. These alloys, having different coefficients of expansion, cause the spring to coil more tightly, or uncoil, from changes in temperature.

8-120. Inspection — Outside Air Temperature Thermometer. (See fig. 8-6.) *a.* Examine the case and the glass for damage.

b. Inspect the pointer for a correct indication of a known temperature reading. Inspect the dial index markings and numerals for legibility.

c. Inspect the rubber mounting washers for cracks from hardening.

8-121. Removal — Outside Air Temperature Thermometer. (See fig. 8-6.)

WARNING

Instrument contains radioactive material (refer to para 8-2).

a. Hold the indicator dial firmly and remove the sunshield from the thermometer stem.

b. Remove the dished washer and the outer rubber mounting washer from the stem; then remove the indicator from the canopy.

8-122. Installation — Outside Air Temperature Thermometer. (See fig. 8-6.) *a.* Remove the sunshield, dished washer and outer rubber washer from the indicator assembly and install the remainder of the assembly in the canopy.

b. Rotate the indicator dial for the correct reading position, and install the outer rubber washer, the dished washer, and the sunshield. Securely hand-tighten the sunshield.

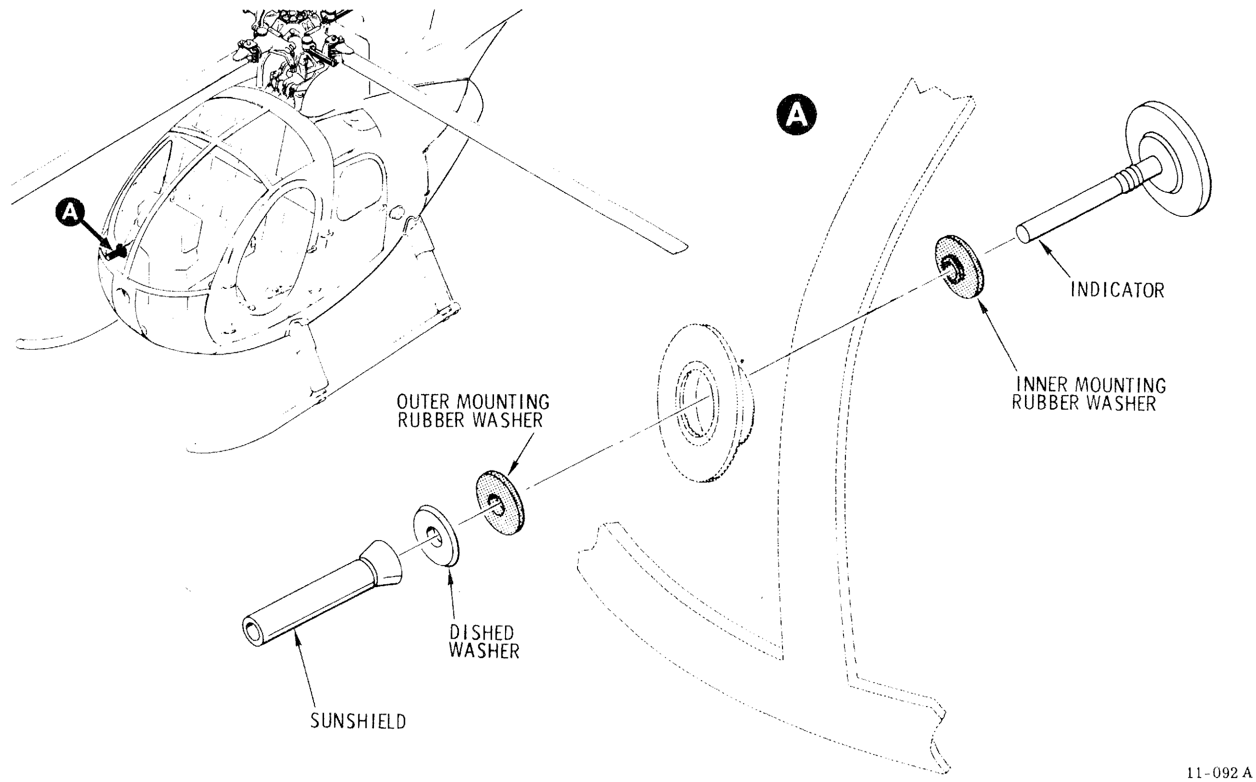


Figure 8-6. Outside Air Temperature Thermometer.

SECTION VI PANELS

8-123. INSTRUMENT AND INDICATOR PANELS.

8-124. References — Instrument and Indicator Panels. Refer to chapter 9 for maintenance instructions pertaining to panels and components thereof not covered in this chapter.

CHAPTER 9

ELECTRICAL SYSTEMS

SECTION I GENERAL INFORMATION

9-1. ELECTRICAL SYSTEMS.

9-2. General — Electrical Systems. Power loading charts and detail system wiring diagrams are contained in appendix F. Aviation Unit Maintenance activities shall request Intermediate Maintenance assistance for electrical system repairs in accordance with the maintenance allocation chart, appendix B. Refer to chapter 8 for information on electrically operated instruments. Refer to system coverage in other chapters of this manual for information on sending devices associated with indicator lights on the instrument panel. Refer to chapter 10 for information on the electric fuel pump. Electrical item reference designators are used to correlate information in this chapter with the wiring diagrams in appendix F.

NOTE

All electrical systems troubleshooting and maintenance in this chapter require the assistance of an Electrical Repairman (MOS 68F).

9-3. Repair or Replacement — Electrical Wiring Harness. Components of the electrical system are connected by wiring harnesses, with wire identification as shown in appendix F. Wires in a harness are terminated with terminal lugs, electrical connectors (plugs and receptacles), or snap-in contacts. Wiring harnesses are held in place by nylon self-clinching loop straps and nylon tying twine. Nylon caterpillar-type grommets provide insulation where harnesses pass through structural openings. Each electrical wire of a harness is identified by a letter-number code (with the exception of a few color-coded lengths that are part of a specific component or assembly). The following is a typical procedure.

a. Set the power selector switch at OFF and perform the following steps as applicable.

b. Remove the access door.

c. Open panels by removing the fasteners that secure the panels to the structure.

d. Remove the component(s) blocking access. (Refer to component coverage elsewhere in this manual.)

e. Disconnect the tag-identify wire terminal(s) at applicable panel.

f. Disconnect wiring harness connector (plug or receptacle) and tag-identify any single terminal connections.

NOTE

Observe the method of wiring harness routing, clamping, and tying before removal to aid reinstallation.

g. Cut tying twine and cut or loosen nylon straps as necessary to permit removal of the harness or individual wire(s).

h. Route replacement harness or wire(s) into position.

i. Install harness connectors (receptacle, plug or inserts). Refer to paragraph 9-4.

j. Connect any harness single-terminal leads.

k. Check the operation of all circuits affected by harness or wire replacement. Refer to applicable wiring diagrams.

l. Replace tying twine (C112) and reinstall nylon straps if available.

NOTE

Substitute tying twine if nylon straps are unavailable. If wiring is near the engine, use high temperature lacing cord (C52).

m. Install component(s) removed to gain access. Refer to component coverage elsewhere in this manual.

n. Close the panels and reinstall the fasteners.

o. Install the access door(s).

9-4. Description — Electrical Connectors. Plugs, receptacles and terminal block snap-in type contacts provide the means by which electrical harness assemblies are connected to individual panels and electrical components. Each plug/receptacle combination is secured with lockwire, or otherwise mechanically locked, to prevent opening of the connection due to vibration. Terminal block snap-in type contacts physically and electrically connect individual harness leads (wires) to terminal block modules.

NOTE

Contacts and tools listed in the first two entries (marked by asterisk) below are provided in Electronics Equipment Maintenance Kit, MK-693/A.

Nomenclature	Application or Use
*Hand-operated crimping tool Hytool M10S: Die-Set S5: Stop bushing SL-26 (Burndy, Norwalk, Connecticut); or equivalent.	*Manual crimping of Hytip electrical contacts on electrical wire leads. (AWG 22-20.)
*Hytip insertion-extraction tool RXT20-5 (Burndy); or equivalent.	*Installing and removing Hytip electrical contacts of terminal block modules.
Hand-operated crimping tool Hytool M8ND (Catalog No. Y14MRC): Die-Sets N22PCT-1 and -2, and N14HCT (Burndy): or equivalent.	Manual crimping of Minilok (Hytip) and Modulok terminal block contact tips on electrical wire leads. (AWG 22-14.)
Hand-operated crimping tool (Catalog No. 107-0970) with positioners (Catalog No. 107-0976 and 107-0977) Winchester Electronics Division of Litton Industries Main St. and Hillside Ave., Oakville, Connecticut, 06779; or equivalent.	Hand crimping of removable contacts used in rectangular connectors.
Removable contact removal tool (Catalog No. 107R1001) and insertion tool (Catalog No. 107-1015) Winchester Electronics; or equivalent.	Removing and inserting removable contacts of rectangular connector inserts.

9-5. Inspection — Electrical System. (See fig. 9-1.) *a.* Inspect electrical connectors for cracks, distortion, and damaged threads.

b. Inspect connector pins, sockets, and lugs for deformation, corrosion, signs of arcing, and damaged insulation.

c. Inspect electrical wires for broken strands and damaged insulation.

9-6. Repair or Replacement — Plug/Socket Connectors. (See fig. 9-1.) *a.* Set the power selector switch at OFF.

b. Remove any connector bonding or lockwire.

c. Disconnect the electrical connector from the mating plug or receptacle by using the applicable following steps:

(1) Loosen the strain-relief clamp on the connector hood or adapter.

(2) On circular type connectors, loosen the hood and disengage the plug lock spring from the receptacle lock ring by rotating the lock spring in either direction.

(3) Disassemble the shell or hood from the electrical connector to expose the terminals.

(4) Disconnect crimp-type removable contacts by using the appropriate contact removal tool.

d. Slide the insulation sleeves back over the wires and unsolder the wires from the contact inserts, or cut off crimp-type contacts as necessary.

e. Solder or crimp the electrical wires to the insert contacts and remove identifying tags, if used. Use a tin-alloy solder (C93) for all soldering operations, and the appropriate crimping tool for replacement of removable crimp-type contacts.

f. Install crimp-type contacts by using the appropriate contact insertion tool.

g. Check for short circuits. Ensure that resistance between pins is infinite except where pins are interconnected. Refer to wiring diagrams, appendix F.

h. Join and secure the connector halves.

(1) On circular type connectors, rotate the plug lock spring to engage the receptacle lock ring.

(2) Tighten the hood or adapter strain-relief clamp on the connector and tighten the clamp screws.

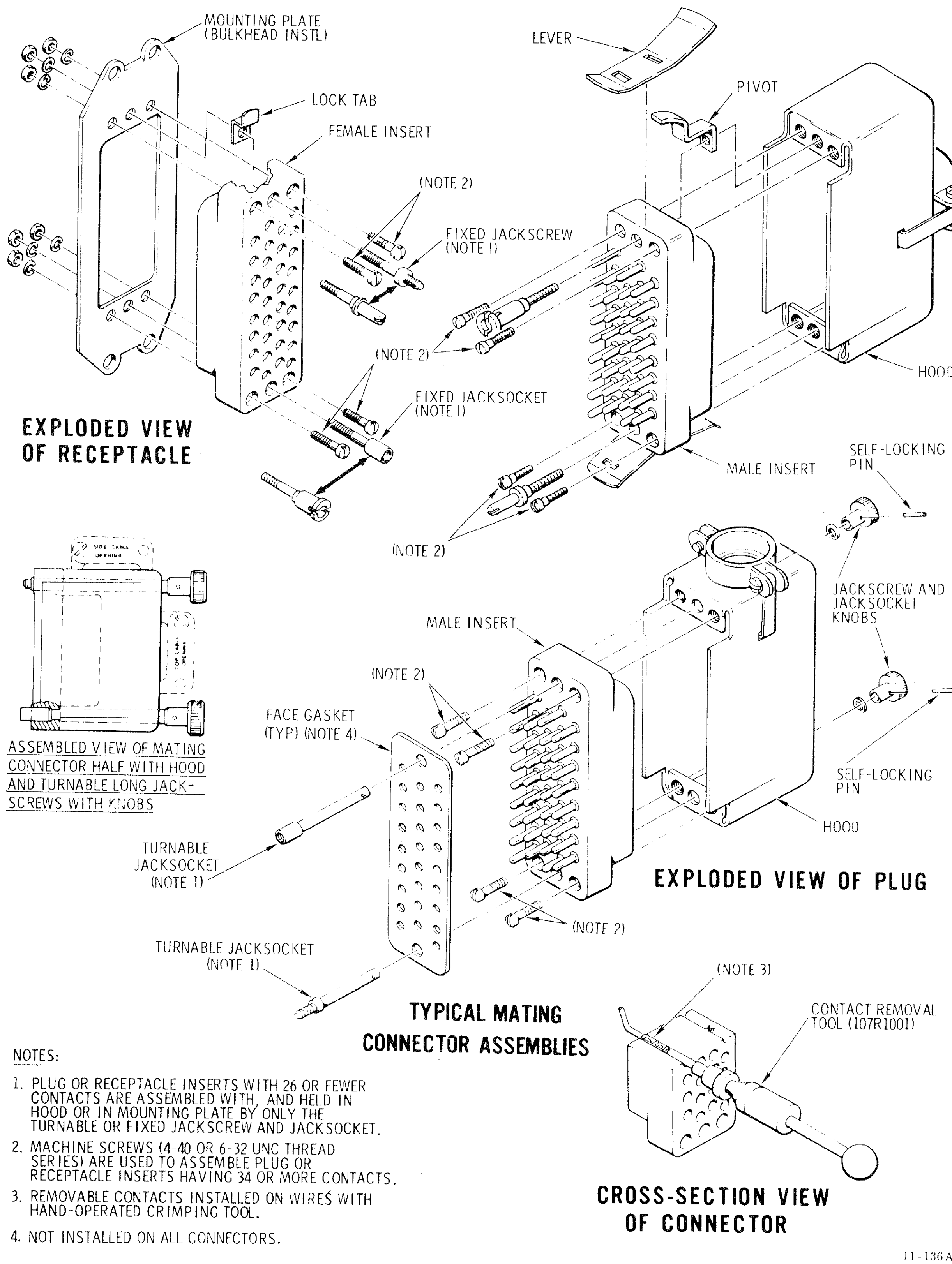
(3) Use lockwire and attach connector bonding, as applicable.

9-7. Replacement — Minilok (Hytip) and Modulok Terminal Block Contact Tips. (See fig. 9-2 and 9-3.) *a.* Set the power selector switch at OFF.

b. Use a Burndy insertion/extraction tool to pull a Hytip contact from a Minilok module.

c. Modulok modules have either two-tier or four-tier spring-loaded connectors. Using a standard screwdriver for recessed head screws, twist the locking screw counterclockwise to the quick-disconnect position of the screw and spring mechanism to release the contact tips. Pull the tips from spring-loaded socket connectors.

d. Strip either 0.17 or 0.20 inch of insulation from end of wire, as applicable. See figure 9-2.

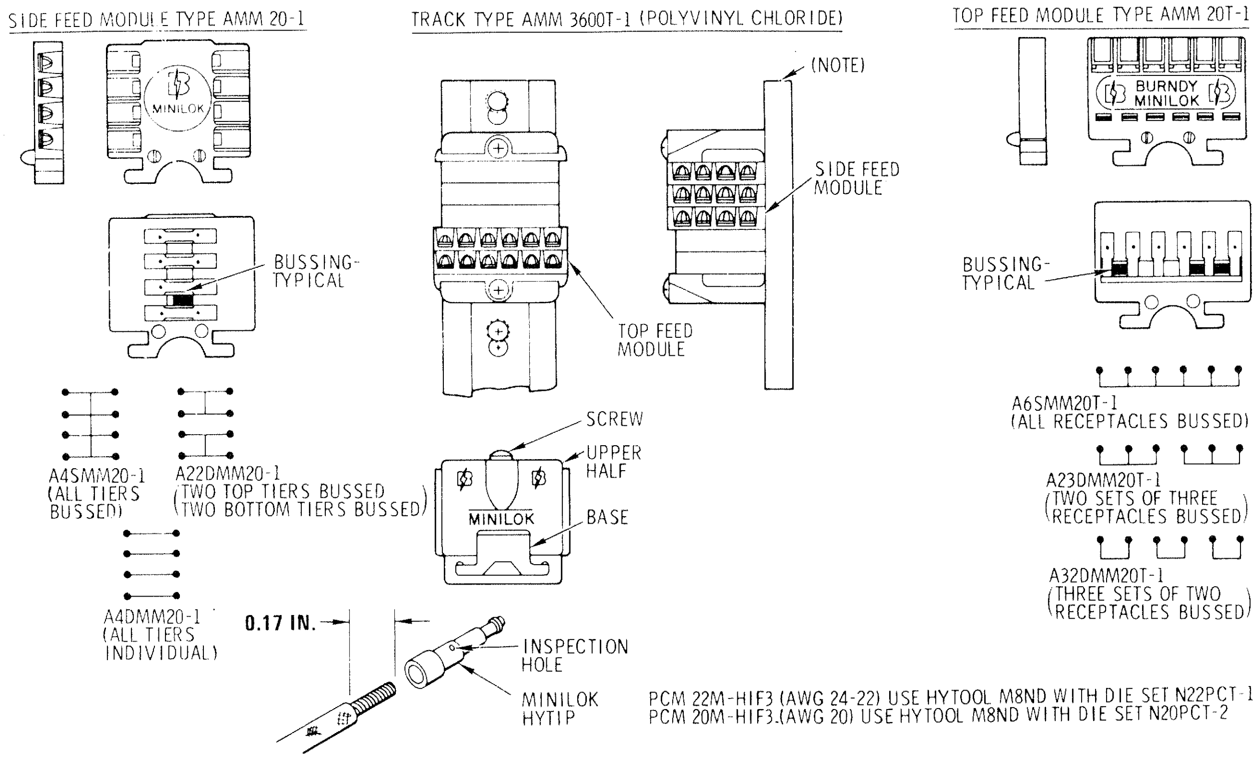


NOTES:

1. PLUG OR RECEPTACLE INSERTS WITH 26 OR FEWER CONTACTS ARE ASSEMBLED WITH, AND HELD IN HOOD OR IN MOUNTING PLATE BY ONLY THE TURNABLE OR FIXED JACKSCREW AND JACKSOCKET.
2. MACHINE SCREWS (4-40 OR 6-32 UNC THREAD SERIES) ARE USED TO ASSEMBLE PLUG OR RECEPTACLE INSERTS HAVING 34 OR MORE CONTACTS.
3. REMOVABLE CONTACTS INSTALLED ON WIRES WITH HAND-OPERATED CRIMPING TOOL.
4. NOT INSTALLED ON ALL CONNECTORS.

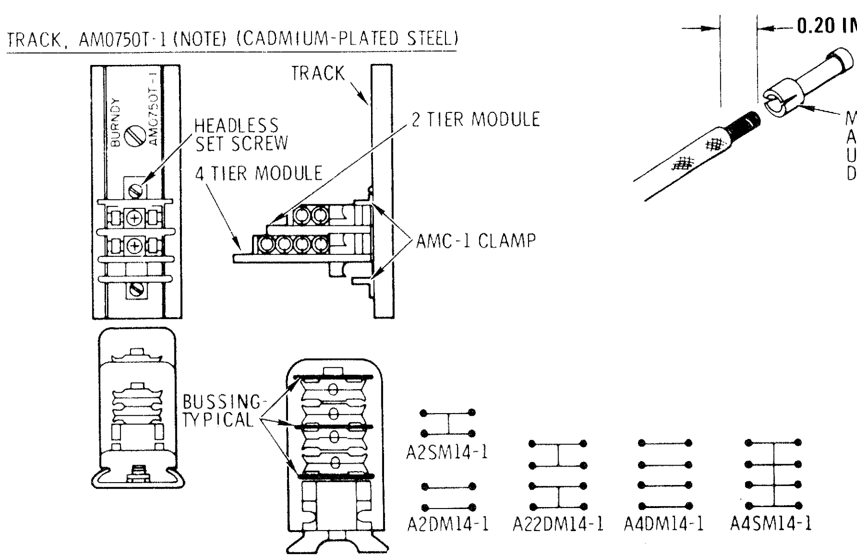
11-136A

Figure 9-1. Rectangular Connector Repair and Replacement.



TYPICAL INSTALLATION OF MINILOK TERMINAL BLOCK MODULES

REF TB1 (SERIES 1 & 2 ACFT) REF TB100, TB200, TB300, TB350, TB400 TB700, TB800 (SERIES 3 ACFT)



NOTE:

THE FOUR MIDDLE NUMERICAL DIGITS IDENTIFY LENGTH IN INCHES. AMM TRACK COMES IN 36 INCH LENGTHS AND IS CUT TO SIZE; AM TRACK CAN BE ORDERED IN ANY PRECUT LENGTH UP TO 32 INCHES (EG, 0750 = 7.50 INCHES)

TYPICAL INSTALLATION OF MODULOK TERMINAL BLOCK MODULES

REF TB500 (SERIES 3 ACFT)

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Figure 9-2. Terminal Block Identification and Replacement.

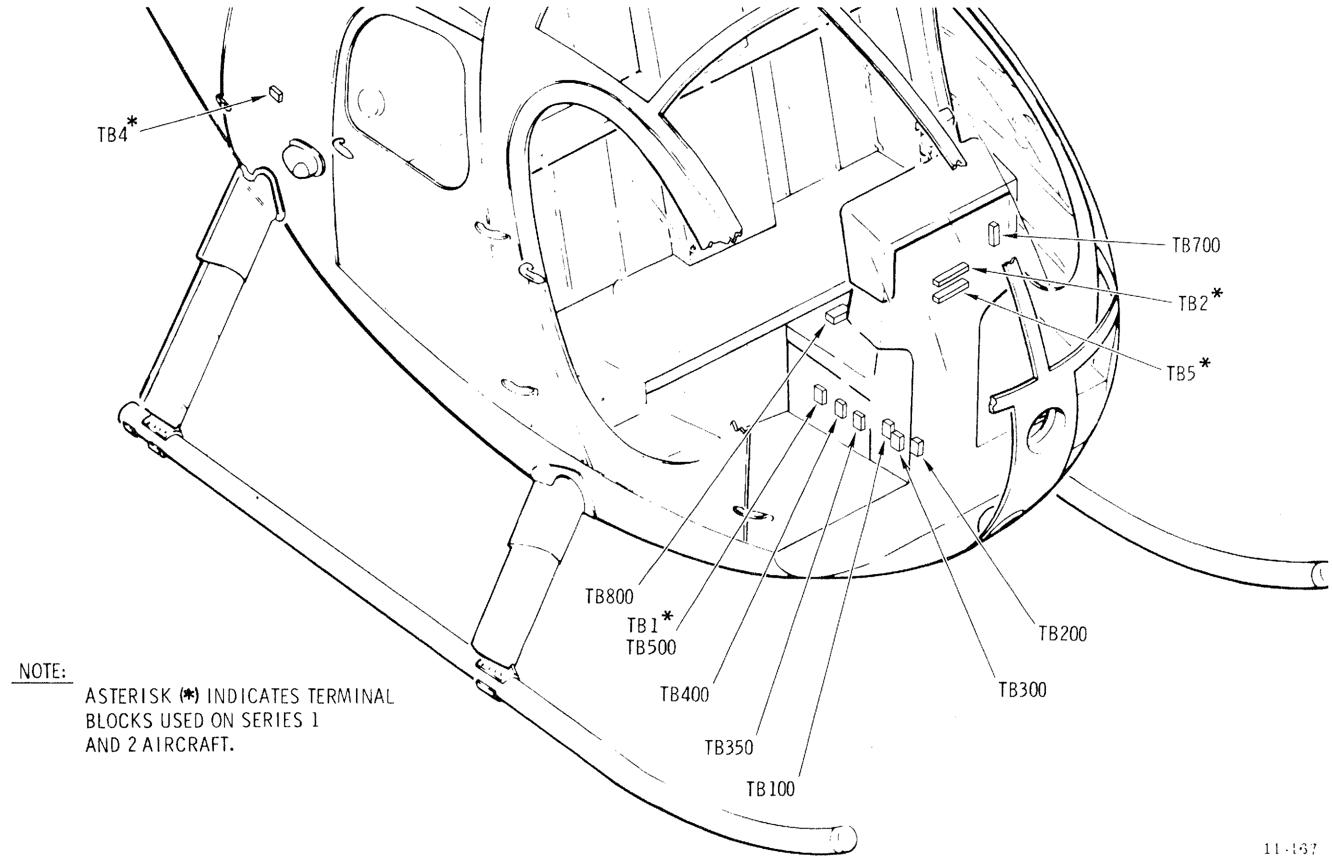


Figure 9-3. Terminal Block Locations.

e. Insert the stripped end of the wire into the snap-in contact. On Hytip (Minilok) contacts, be sure that the stripped end of the wire extends past the inspection hole. Make sure that the stripped end of the wire is inserted as far as possible into the contact tip and that the wire insulation extends approximately 0.06 inch into the contact shoulder (Minilok), or to the end of the contact shoulder notch (Modulok).

f. Using the Burndy manual crimping tool and correct die-set, crimp the contact on the wire.

g. Visually inspect the contact tip on the wire. Make sure that it is firmly attached to the wire and that the stripped end of the wire blocks the inspection hole in the Hytip contact.

h. Use a Burndy insertion/extraction tool to push Hytip snap-in contact into the proper terminal block module of the terminal block assembly. Refer to the applicable wiring diagram in appendix F.

i. Plug Modulok contact tips into spring-loaded socket connectors and twist the locking screw clockwise to the locked position (spring fully compressed).

NOTE

Modulok modules may be set in either the quick-disconnect position or the locked position; the spring-loaded socket connectors exert uniform pressure in either position. The quick-disconnect position should be used only to facilitate circuit checkout.

9-8. Replacement — Minilok and Modulok Terminal Blocks. *a.* Set the power selector switch at OFF.

NOTE

To aid reassembly, tag and identify all wires that are to be disconnected from the terminal block module to be replaced.

b. Remove contact tips according to paragraph 9-7.

c. Loosen the screw that secures the end clamp holding the modules on the track. Slide the end clamp away from the modules and separate the module to be replaced by unsnapping it from adjoining modules.

d. With modules separated for clearance from one another, remove the module by turning it 90 degrees and unsnapping it from the track.

e. Insert the replacement module into mounting position between or next to the correct adjoining modules, rotate it 90 DEGREES and snap it into place. Press the modules together.

f. Slide the end clamp into place against the outside module. While pressing and holding the clamp against the outside module, tighten the end clamp screw.

g. Install contact tips according to paragraph 9-7. Remove identifying tags from wires.

9-9. Maintenance — Electrical Bonding and Grounding Connections. Refer to TM 55-1500-204-25/1.

SECTION II DIRECT CURRENT POWER DISTRIBUTION SYSTEM

9-10. DIRECT CURRENT POWER DISTRIBUTION SYSTEM.

9-11. Description — Power Supply and Distribution. The electrical system is supplied by a 28-volt direct-current, 150-ampere, engine-driven generator, in conjunction with a 24-volt nickel-cadmium battery, and incorporates an external power receptacle. The system is a single-wire type installation with aircraft structure used as the ground return. Control of the electrical system, exclusive of avionics equipment controls, is provided by the switches and circuit breakers on the electrical control console and circuit breaker panel. All circuits of the electrical system are protected by push-to-reset or switch-type circuit breakers.

9-12. Test (Operational Check) — Direct Current Power Distribution System. The operational checkout procedure for the dc power distribution system concerns the functioning of electrical components used to start the engine (start switch, starting pump, start relay, and engine igniter) and generator voltage control devices (voltage regulator, overvoltage relay, and reverse current relay). The operation of these components is checked as a part of the aircraft ground run procedure. (Refer to TM 55-1520-214-10.)

9-13. Troubleshooting — Direct Current Power Distribution System. Refer to table 9-1 and the wiring diagram in appendix F. Prior to using table 9-1, make sure that applicable circuit breakers are reset and that warning and caution indicator lamps are functional; use the wiring diagram to check for electrical breaks, shorts, and insecure connections.

9-14. EXTERNAL POWER RECEPTACLE.

9-15. Inspection — External Power Receptacle. (See fig. 9-4.) *a.* Inspect the terminal pin support for cracks or other damage.

b. Inspect the receptacle platenuts for secure attachment.

c. Inspect the terminals for any loose, cracked, or burned condition.

d. Inspect the wiring for loose or corroded connections and worn or frayed insulation.

9-16. Removal — External Power Receptacle. (See fig. 9-4.) *a.* Remove the two attaching screws and pull the receptacle outward.

NOTE

It may be necessary to disconnect the ground cable and harness clamps to obtain sufficient slack for receptacle removal. If so, gain access through the right pilot's floor access door.

b. Disconnect the cables and remove the receptacle.

9-17. Installation — External Power Receptacle. (See fig. 9-4.) *a.* Attach the cables to the receptacle terminals and secure them with terminal nuts.

Table 9-1. Troubleshooting the Direct-Current Power Distribution System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **No power is available to any electrical components when the power selector switch is set at BAT. (Power is available from an external source when the switch is set at EXT.)**

STEP 1. Determine the condition of the battery per TM 11-6140-203-15-2.

If the battery is defective, replace it per paragraph 9-18.

STEP 2. Test the power selector switch (S14) by temporarily reversing the wire connections on terminals 1 and 3. Refer to paragraph 9-50 for gaining access to the switch.

If the switch fails to conduct power when set at EXT, replace the switch (para 9-50).

2. **Weak or discharged battery.**

STEP 1. Adjust the voltage regulator to establish the correct charging rate (para 9-31).

If the voltage regulator cannot be adjusted for optimum charging rate, replace it. Refer to paragraph 9-31.

STEP 2. Charge the battery per TM 11-6140-203-15-2 or by aircraft engine operation.

If the battery is not chargeable or will not retain a charge, replace it. Refer to paragraph 9-18.

3. **Excessive corrosion on the battery terminals.**

STEP 1. Adjust the voltage regulator to establish the correct charging rate. Refer to paragraph 9-31. (Corrosion can be the result of an excessive charging rate.)

If the voltage regulator cannot be adjusted for optimum charging rate, replace it. Refer to paragraph 9-31.

STEP 2. Ascertain whether the aircraft battery has been used extensively, instead of an external power source, to start the engine. (An excessive discharge rate can cause corrosion on the battery terminals.)

If excessive use of the aircraft battery for the purpose of starting the engine is occurring, substitute external power whenever possible.

4. **Battery overheats and/or discharges smoke.**

STEP 1. Replace battery (para 9-18) and then adjust the voltage regulator to establish the correct charging rate (para 9-31).

If the voltage regulator cannot be adjusted for an optimum charging rate, replace it. Refer to paragraph 9-31.

STEP 2. Examine the battery for loose terminal links, straps, and screws (para 9-18).

If looseness exists, make repairs per TM 11-6140-203-15-2. If the battery continues to overheat and/or discharge smoke, replace it. Refer to paragraph 9-18.

5. **Starter-generator fails to supply battery-charging power with the engine operating.**

STEP 1. Using a multimeter (T4) set to the 30 volt DC scale, attach the (+) lead to the reverse current relay GEN terminal and the (-) lead to the BAT terminal. (See fig. 9-14.) With the engine operating observe the voltmeter scale.

If a voltage reading (approximately 2 to 4 volts) is noted, replace the voltage regulator (para 9-45).

Table 9-1. Troubleshooting the Direct-Current Power Distribution System. (cont)

STEP 2. Jumper the SW terminal to the GEN terminal on the reverse current relay K300. See figure 9-14.

If the instrument panel DC AMPS meter indicates a charge, adjust or replace the reverse current relay (para 9-31).

If the instrument panel DC AMPS meter remains at zero, replace the starter-generator (para 9-22).

6. **Starter-generator field fuse XF1 on series 1 and 2 aircraft or the starter-generator field circuit breaker CB31 on series 3 aircraft blows repeatedly.**

STEP 1. Adjust the voltage regulator for an optimum charging rate (para 9-31).

If the voltage regulator cannot be properly adjusted, replace it. Refer to paragraph 9-31.

If regulator adjustment or replacement fails to correct the trouble, replace the starter-generator (para 9-22).

7. **Instrument panel DC AMPS meter indicates a discharge with the engine not operating and the generator switch set at ON.**

STEP 1. Disconnect wires and bus bar from terminals A1 and A2 of the start relay K301. See figure 9-14. Use a multimeter in the ohmmeter mode to measure the resistance between the terminals.

If the resistance indication is less than infinite, replace the start relay (para 9-85).

If the resistance indication is infinite, replace the reverse current relay K300 (para 9-45).

8. **GENERATOR OUT (series 1 and 2 aircraft) or DC GEN (series 3 aircraft) caution light on the instrument panel is lighted while the engine is operating and the generator switch is set at ON.**

STEP 1. Jumper between the SW terminal and the GEN terminal of the reverse current relay K300. See figure 9-14. Momentarily jumper terminal A to terminal B on the starter-generator.

If the caution light goes out, adjust or replace the voltage regulator (para 9-31).

STEP 2. Jumper terminal B of the starter-generator to the BAT terminal of the reverse current relay K300. See figure 9-14. Momentarily jumper terminal A to terminal B on the starter-generator.

If the instrument panel DC AMPS meter indicates a charge of more than 25 amperes, replace the reverse current relay (para 9-45).

If less than 25 amperes charge is indicated on the instrument panel DC AMPS meter, replace the starter-generator (para 9-22).

9. **Fluctuating charge on instrument panel DC AMPS meter when engine rpm is constant.**

STEP 1. Jumper the SW terminal and the GEN terminal of the reverse current relay K300. See figure 9-14. Momentarily jumper terminal A to terminal B on the starter-generator.

If the instrument panel DC AMPS meter indicates an increased charge without fluctuation while both jumpers are installed, adjust or replace the voltage regulator. Refer to paragraph 9-31.

STEP 2. Jumper the GEN and BAT terminals of the reverse current relay K300. See figure 9-14.

If the instrument panel DC AMPS meter indicates steadily, replace the reverse current relay. Refer to paragraph 9-45.

If the instrument panel DC AMPS meter continues to fluctuate, replace the starter-generator (para 9-22).

10. **Instrument panel DC AMPS meter indicates an excessive charge during engine operation.**

STEP 1. Disconnect resistor R1 (refer to table 9-3) and temporarily install a new resistor.

Table 9-1. Troubleshooting the Direct-Current Power Distribution System. (cont)

If the instrument panel DC AMPS meter indicates zero and the GENERATOR OUT (series 1 and 2 aircraft) or the DC GEN (series 3 aircraft) caution light comes on, permanently replace the resistor.

If the instrument panel DC AMPS meter continues to indicate an excessive charge, replace the overvoltage relay and adjust or replace the voltage regulator. Refer to paragraphs 9-31 and 9-38.

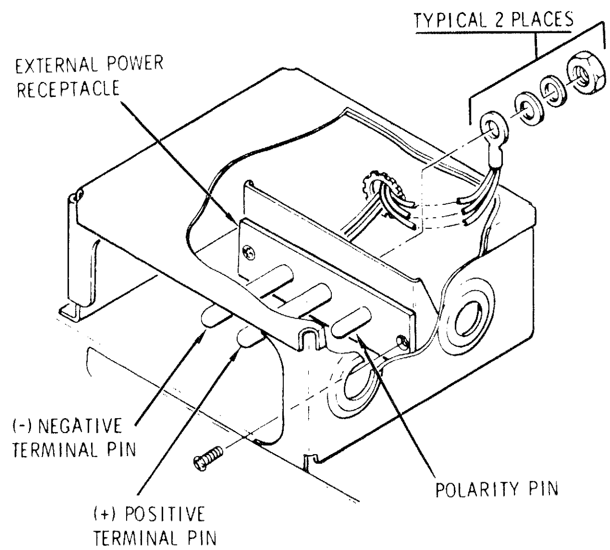


Figure 9-4. External Power Receptacle.

NOTE

The center terminal is the positive (+) terminal. The aft terminal is the negative (-) or ground terminal. There is no wire attachment to the polarity pin.

b. Secure the receptacle with two screws.

c. Re-install any cabling that may have been loosened to facilitate receptacle removal. Refer to paragraph 9-3.

9-18. BATTERY.

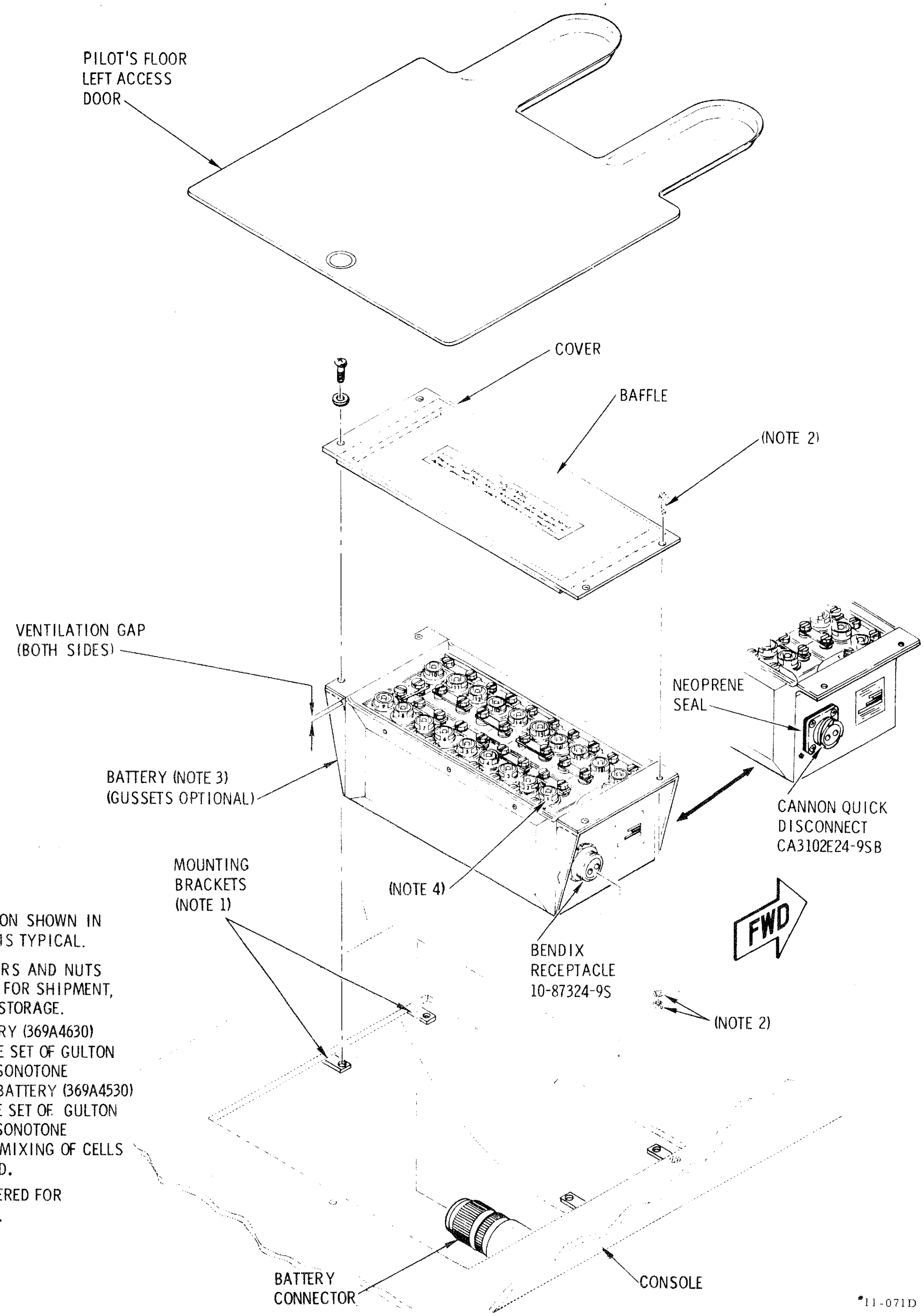
9-19. Description - Battery (BB-641/A or BB-678/A). The battery is a rechargeable, nominal 24-volt, 13-ampere-hour, 19-cell, nickel-cadmium battery capable of starting the engine in temperatures from -25°F to +130°F (-31.6°C to +54.4°C). See figure 9-5. The battery is connected to the electrical system by a two-pin, keyed, male connector receptacle located at the inboard end of the battery case. The 19 identical and individually replaceable battery cells are contained in separate molded nylon cases and interconnected by bus bars. The cells are equipped with removable filler cap vent plugs that allow gasses from the cell to escape through the vent when the battery is in, or near, the upright position. The filler cap vent plugs also prevent electrolyte spillage when the battery is tipped or inverted. A mixture of potassium hydroxide and water is used as the electrolyte; however, only pure distilled water is used for servicing. The battery case, cover, and end plates are constructed from epoxy-impregnated fiberglass. The battery is secured to brackets on the pilot's compartment floor structure by four screws and washers installed through the cover and battery case. Gasses generated within the battery are vented from the cells to the inside of the cover and out through the gap between each upper side edge of the case and cover. Servicing precautions, servicing, and cleaning instructions for the battery are provided in TM 11-6140-203-15-2.

WARNING

Corrosive Battery Electrolyte (Potassium Hydroxide). Wear rubber gloves, apron, and face shield when handling leaking batteries. If potassium hydroxide is spilled on clothing, or other material wash immediately with clean water. If spilled on personnel, immediately start flushing the affected area with clean water. Continue washing unitl medical assistance arrives.

WARNING

Attempts to remove the battery immediately following a battery overtemperature condition may result in serious burns to the hands or possible explosion hazards. Battery overtemperature may be identified by electrolyte spewing or heavy fumes. If this condition is detected, allow the battery to thoroughly cool (approximately one hour) prior to removal.



NOTES:

1. BATTERY LOCATION SHOWN IN FLOOR OPENING IS TYPICAL.
2. SCREWS, WASHERS AND NUTS SHOULD BE USED FOR SHIPMENT, HANDLING AND STORAGE.
3. CAUTION: BATTERY (369A4630) USES A COMPLETE SET OF GULTON (13040) CELLS OR SONOTONE (25468-3) CELLS. BATTERY (369A4530) USES A COMPLETE SET OF GULTON (11570) CELLS OR SONOTONE (25468-1) CELLS. MIXING OF CELLS IS NOT PERMITTED.
4. CELLS ARE NUMBERED FOR REFERENCE ONLY.

Figure 9-5. Battery Replacement.

9-20. Removal — Battery. (See fig. 9-5.) *a.* Set the power selector switch at OFF.

b. Raise the pilot's floor left access door by lifting the rear latch. For better access, hand-press the access door pivot pins from the hinge points and remove door.

WARNING

The electrolyte used in nickel-cadmium batteries contains potassium hydroxide (KOH), which is a caustic chemical agent. Serious and deep burns of body tissue will result if the electroltye comes in contact with the eyes or any part of the body. Use rubber gloves, rubber apron, and protective eye covering when handling the battery. If accidental contact with the electrolyte is made, use ONLY clean water and immediately (seconds count) flush contaminated areas. Continue flushing with large quantities of clean water. Seek medical attention immediately. Before removing or installing the battery, insure that the battery switch is off and the battery has cooled down if overheated. Removal or installation of the battery connector while the battery is under load may result in explosion, electrical arcing and possible severe burns to personnel.

c. Disconnect the battery plug.

d. Remove the four screws and washers that secure the battery to the mounting flanges.

CAUTION

Avoid tilting the connector-end of the battery downward during removal and when handling after removal. Any water or electrolyte in the battery case will collect at the connector leads and shorting may occur.

e. Lift the battery and the cover from the electronic compartment. Secure the cover to the battery case. Place the battery in a protected area away from any lead-acid type battery.

9-21. Installation — Battery. (See fig. 9-5.) *a.* Remove handling hardware from the cover, if used. Lower the battery and cover into place on the mounting brackets.

CAUTION

Take every possible step to keep the nickel-cadmium battery as far away as possible from the lead-acid type of battery. Do not use the same tools and materials (screwdrivers, wrenches, gloves, apron, etc.) for both types of batteries. Anything associated with the lead-acid battery, even the air, must never come in contact with the nickel-cadmium battery or its electrolyte. Even a trace of sulfuric acid fumes from a lead-acid battery may result in damage to the nickel-cadmium battery. If sulfuric acid has been in advertently mixed with the electrolyte in the battery, the upper areas of the cells will appear greenish in color. In such cases, the battery must be replaced.

b. Secure the battery and the cover to the mounting brackets with four mounting screws and washers.

c. Apply a generous coating of dielectric compound (C37) to the connector pins and sockets and also to the connector plug and receptacle threads. All surfaces should be coated. Connect the battery plug and wipe away excess compound.

d. Lower and latch the floor access door. If the door was removed, reinstall pivot pins at the hinge and latch the door.

9-22. STARTER-GENERATOR.

9-23. Description — Starter-Generator. A combined starter and generator G300 is used to start the engine and to provide 28 vdc power for operation of the electrical system. The starter-generator, a self-cooled unit, is clamp-attached to a quick-disconnect mounting flange on the engine power and accessory gearbox. See figure 9-6. A shear point is incorporated in the generator drive shaft to protect the engine drive from excessive torque loads. The generator system consists of the starter-generator, reverse current relay, voltage regulator, overvoltage relay, generator caution light, and the ammeter. Generator operation is controlled by the generator switch. At flight idle rpm and above, the voltage regulator automatically maintains the correct generator output voltage by varying the generator field current.

The generator field circuit is protected by a 15-ampere fuse in back of the instrument panel on series 1 and 2 aircraft or a circuit breaker in the circuit breaker panel on series 3 aircraft. The circuit protection is located between the voltage regulator and the generator to protect the generator against a regulator failure which could permit excessive current flow through the generator field. When an overvoltage condition occurs, the overvoltage relay is energized by the voltage regulator. The overvoltage relay opens the switch circuit of the reverse current relay to remove generator output from the bus. The reverse current relay prevents the battery from discharging through the generator when the output voltage falls below battery voltage. Electrical systems operate from the battery when the generator switch is at OFF and the power selector switch is at BAT. A terminal block on the starter-generator provides for connection of electrical wiring and houses a radio-frequency-interference filter. Access to the brushes in the starter-generator is provided by a brush cover. When performing as a generator, the starter-generator has a maximum continuous duty rating of 30 volts, 150 amperes, over a range of 7200 to 13,000 rpm.

9-24. Removal — Starter-Generator. (See fig. 9-6.)
a. Set the power selector switch at OFF.

b. Disconnect electrical wiring from the starter-generator terminal block; insulate the wiring terminals. Reinstall nuts and washers on the studs.

c. Loosen the mounting clamp that secures the starter-generator to the mounting flange and remove the starter-generator.

d. Install a suitable protective closure over the opening in the mounting flange.

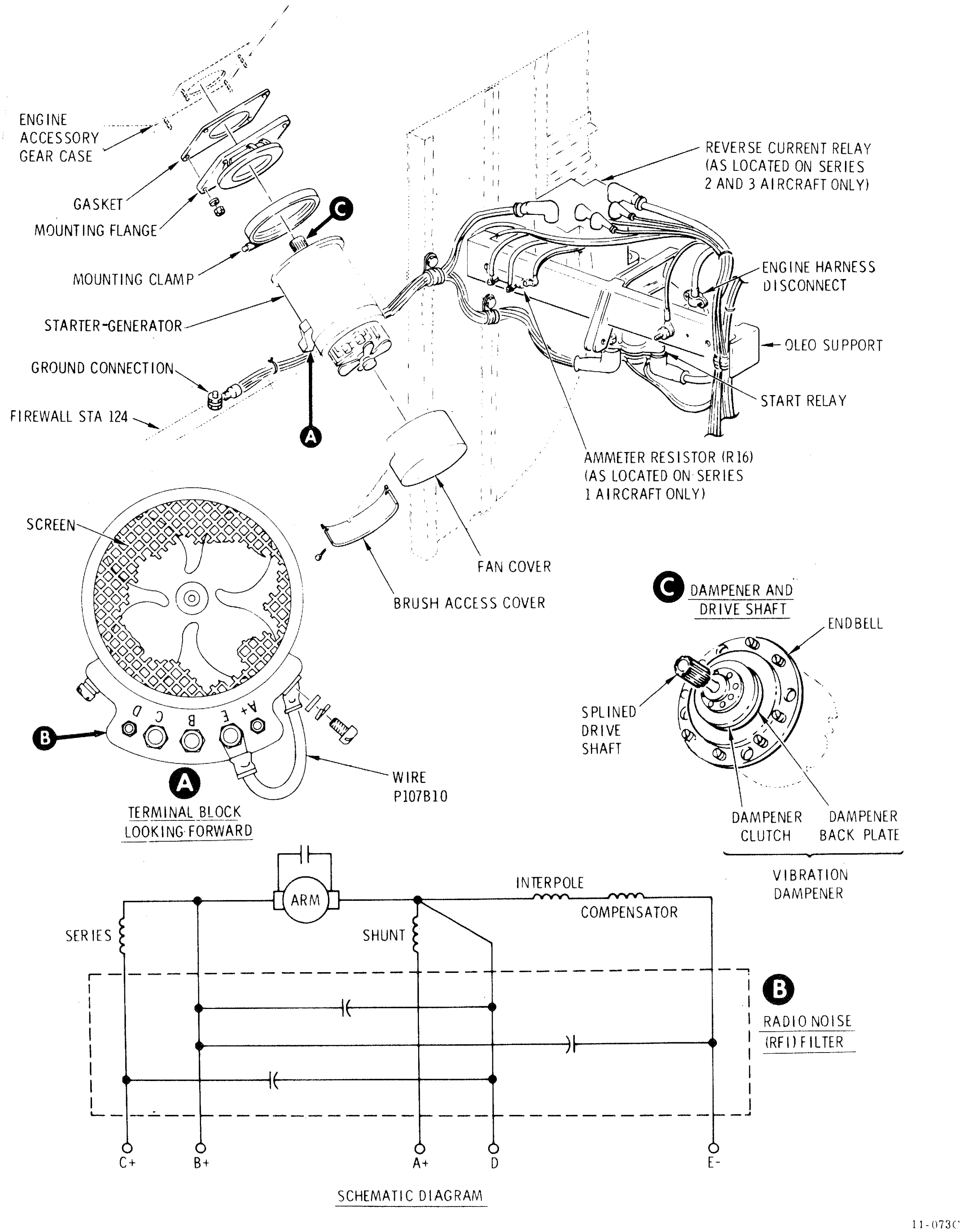


Figure 9-6. Starter-Generator Replacement.

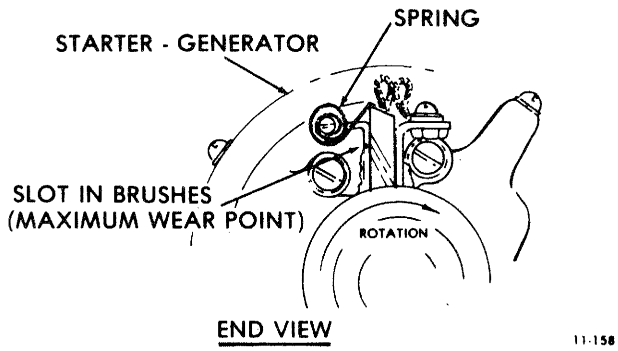


Figure 9-7. Starter Generator Brush Wear Limits.

9-25. Inspection — Starter-Generator. (See fig. 9-6.) a. Remove the brush cover and fan cover. Inspect for oil, dirt, or other foreign material that has entered the unit. Remove material with a vacuum cleaner only.

CAUTION

Do not use compressed air for internal cleaning of starter generator.

b. Inspect the commutator for pitting, scoring, or burned areas.

c. Check the brushes for cracks, chips, frayed leads, and loose shunt connections. Replace the generator if remaining allowable wear on brushes might be exceeded before the next scheduled inspection. See figure 9-7.

d. Check the brush holder and end bell for cracks and warpage.

e. Check the fan for looseness on the hub, and for any cracked or bent condition.

f. Inspect the terminal block for cracks, breaks, loose or damaged terminal studs, and for loose electrical connections.

g. Inspect the starter-generator housing for any signs of damage, and the inlet and outlet blower screens for clogging.

h. Inspect the mounting clamp and the mounting flange for condition, and the mounting flange hardware for secure attachment.

i. Inspect the dampener clutch. See figure 9-6. Generator replacement is required if the following conditions are not met.

(1) Inspect the dampener clutch for cracks, warpage, loose rivets, and excessive wear on the spring surface that mates with the dampener back plate. **MINIMUM SPRING THICKNESS IS 0.015 INCH.**

(2) Make certain that the dampener clutch exerts pressure against the dampener back plate. Weak pressure or gap between these parts is not allowed.

(3) Inspect the dampener back plate for cracks, and the bronze friction facing for wear. **MINIMUM FACING THICKNESS IS 0.010 INCH.**

j. Check splines on the drive shaft for evidence of excessive wear.

Table 9-2. Premaintenance Requirements for Testing of Starter-Generator.

Conditions	Requirements
Test Equipment	(T4)

9-26. Test (AVIM) — Starter-Generator. Assemble a test setup in accordance with figure 9-8. Test the generator-starter by performing the following steps.

a. Connect the starter-generator and test equipment as shown in figure 9-8. Mount the starter-generator securely in suitable vee blocks.

CAUTION

Be sure switch S1 is set at OFF. Do not exceed 14-16 vdc input; starter-generator may be damaged.

b. Start the variable dc generator and adjust the output at 14-16 volts.

c. Connect the 24-vdc power supply.

d. Using a multimeter (T4), adjust the 25-ohm potentiometer so that the terminals that will be attached to A + and B + on the starter-generator terminal block provide a 3-ohm resistance.

e. Connect the 25-ohm potentiometer into the test circuit.

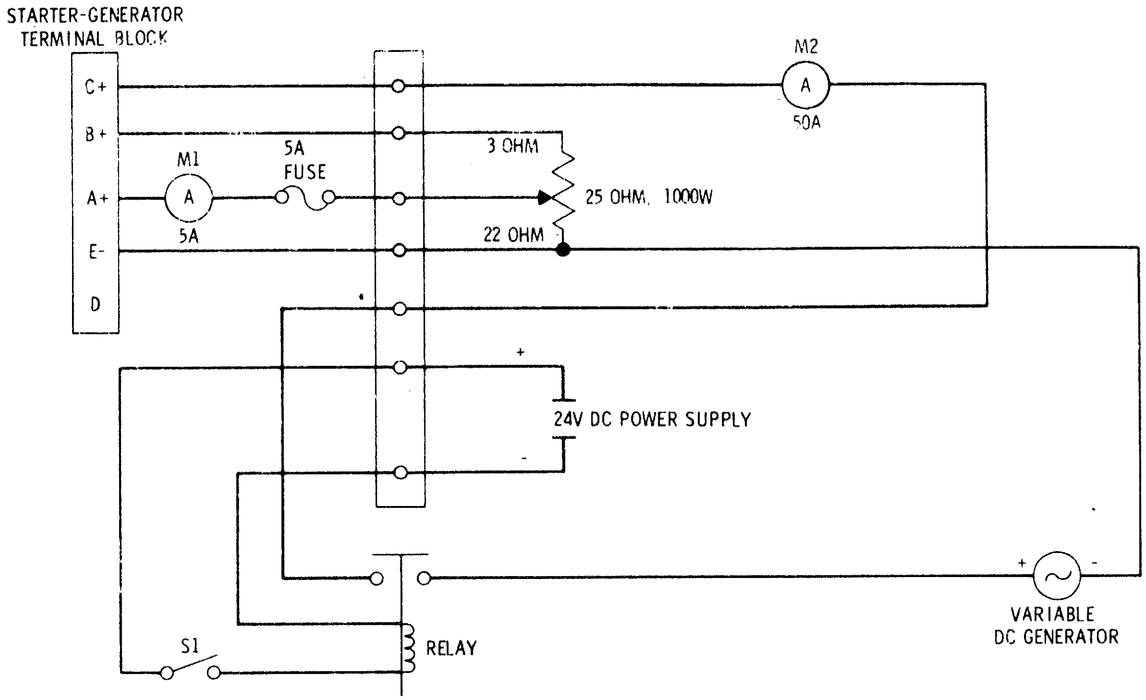
f. Set switch S1 momentarily ON and observe the direction of starter-generator shaft rotation. Rotation should be counterclockwise when viewing the drive-end of the starter-generator.

g. Set switch S1 at ON. The starter-generator drive shaft shall rotate freely and generator output shall be indicated on meter M1.

h. Adjust the 25-ohm potentiometer until meter M1 indicates approximately 1 ampere. Meter M2 should indicate 25 to 35 amperes at this time.

i. Observe the degree of sparking at the starter-generator commutator. If the brushes are seated properly, the sparking should be no more than pinpoints.

j. Turn off all electrical power and inspect the brushes for proper seating. Brush faces should be 90 percent seated in the axial direction, and 100 percent of the axial contact should show seating contact in the direction of rotation. Blow out all brush dust with clean, dry, low pressure compressed air.



TEST SETUP

TEST SETUP COMPONENTS

VARIABLE VOLTAGE D-C GENERATOR, 150 AMPERE CAPACITY

D-C AMMETER M1, 5 AMPERES FULL SCALE

D-C AMMETER M2, 50 AMPERES FULL SCALE

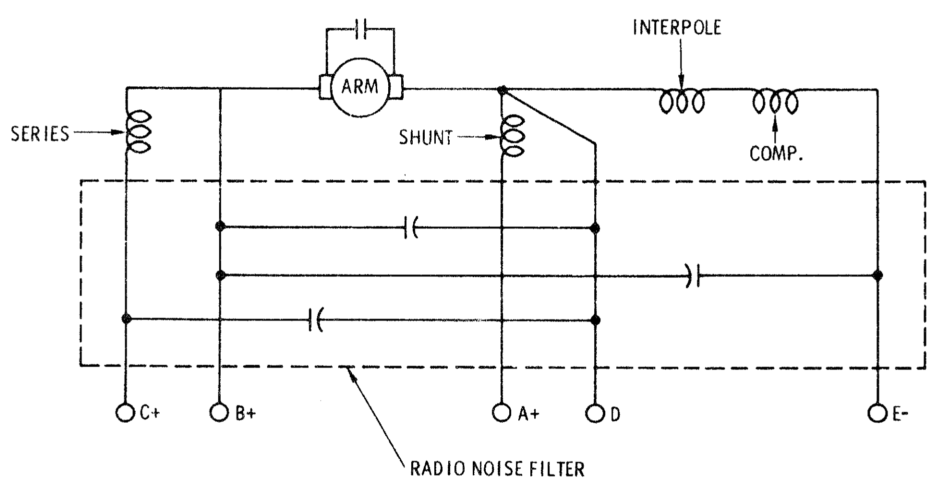
RELAY, 200-AMPERE, SINGLE-POLE, SINGLE-THROW (AN3371-2)

SWITCH S1, SPST, 10 AMPHERES AT 28 VOLTS, DIRECT CURRENT

POTENTIOMETER, 25 OHMS, 1000 WATTS

POWER SUPPLY, 10-36 VOLTS, DIRECT-CURRENT

5-AMPERE FUSE



SCHEMATIC

12-158A

Figure 9-8. Starter-Generator Schematic Diagram and Test Setup.

k. Reconnect electrical power and set switch S1 at ON.

l. Adjust the 25-ohm potentiometer until meter M1 indicates 4 amperes. At this time, meter M2 shall indicate 25 to 35 amperes.

m. Turn all power OFF and disconnect the starter-generator. Remove it from vee blocks and re-install the brush cover.

9-27. Repair — Starter-Generator. An unserviceable starter-generator shall be replaced.

9-28. Installation — Starter-Generator. (See fig. 9-6.) *a.* Set the power selector switch at OFF.

b. Remove the protective cover from the opening in the starter-generator mounting flange.

c. Apply a film of lubricant (C63) to the starter-generator drive shaft splines.

d. Align the screw heads in the endbell with cut-outs in mounting flange. Hold the starter-generator firmly against the flange.

e. Place the V-band mounting clamp on the starter-generator; then latch the T-bolt and tighten the nut snugly.

f. Tap the mounting clamp in several places with a soft mallet and **TORQUE THE NUT TO 45-55 INCH POUNDS.**

g. Install electrical leads, washers and nuts. Tighten nuts and remove identifying tags.

9-29. GENERATOR FIELD.

9-30. Flashing — Generator Field. Generator flashing is performed when the starter-generator fails to produce adequate output voltage due to loss of residual magnetism or reversed field polarity. This condition is indicated by a zero or negative DC AMPS meter indication with the DC GEN or GENERATOR OUT caution light on. See figures 9-6 and 9-14 and the applicable wiring diagram in appendix F to locate wires and terminals referred to in the following procedure.

CAUTION

Do not flash the field while the starter-generator is in operation as damage to the voltage regulator may result.

a. Disconnect the external field wire from generator field stud A +, and the ammeter wire from stud D.

b. Connect one end of a jumper wire to the generator field stud A +.

c. Connect a second jumper wire to the BAT terminal of the reverse current relay.

d. Set the generator switch at OFF and set the power selector switch at BAT.

e. Momentarily touch the ends of the jumper wires together several times but do not exceed a total contact time of more than five seconds.

f. Set the power selector switch at OFF.

g. Disconnect the jumper wires and reconnect the external field wire to terminal stud A + and the ammeter wire to terminal stud D.

h. Operate the engine for a generator check. If flashing the field fails to correct generator polarity, check for incorrect connections or a defective field circuit.

9-31. VOLTAGE REGULATOR.

9-32. Description — Voltage Regulator. The voltage regulator maintains a constant generator output voltage under varying speed and load conditions. The regulator has an adjustment range of 26 to 30 volts and contains an overvoltage circuit.

9-33. Inspection — Voltage Regulator. (See fig. 9-9 and 9-10.) *a.* Inspect the regulator for cracks, breaks, dents, corrosion, and signs of arcing.

b. Inspect the plug for broken or deteriorated insulation, and the receptacle for broken or bent pins, and corrosion.

c. Check the regulator setting at intervals specified in the Special Inspection Checksheet, chapter 1.

Table 9-3. Premaintenance Requirements for Adjusting the Voltage Regulator.

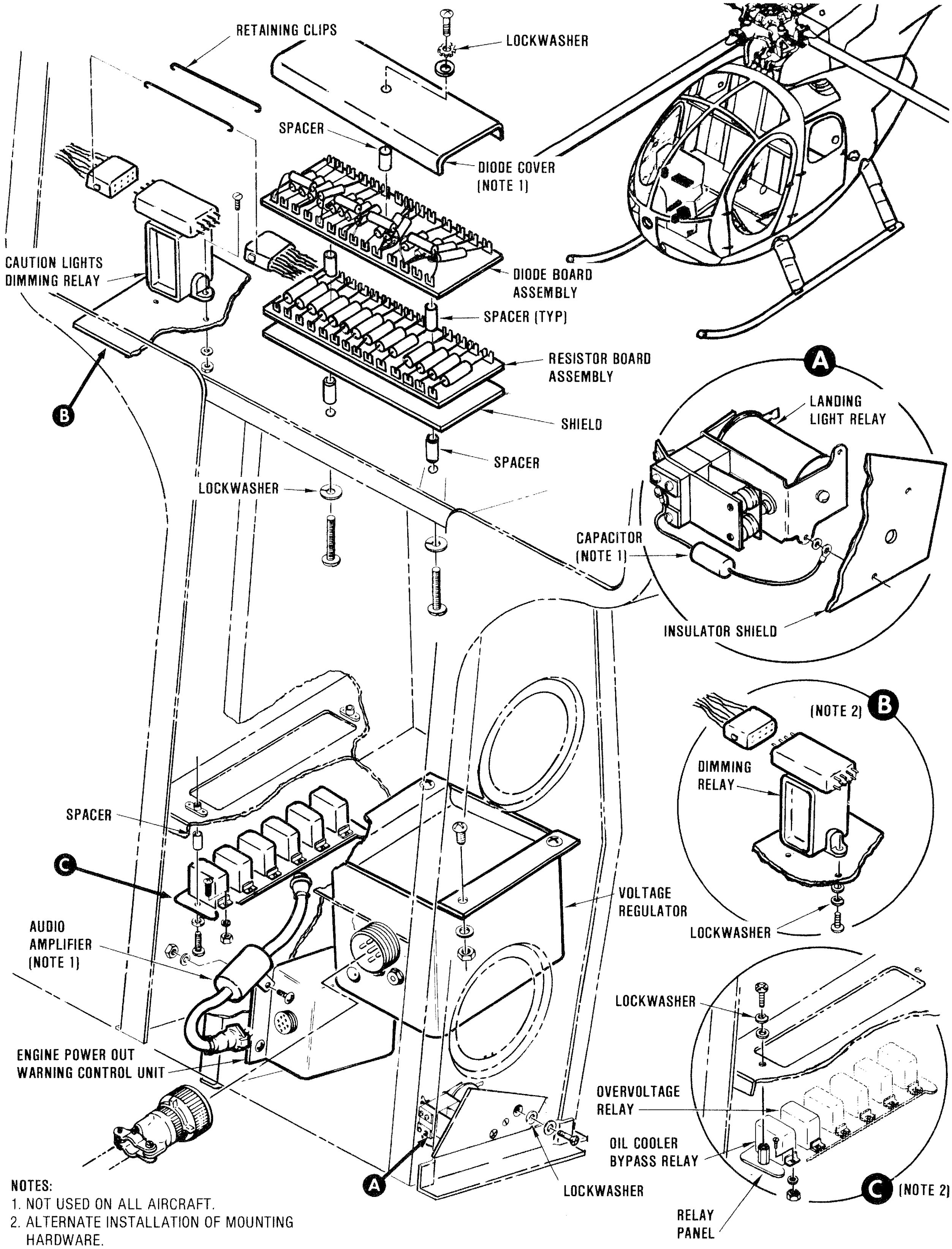
Conditions	Requirements
Test Equipment	(T3) or (T5)
Minimum Personnel Required	Two

9-34. Adjustment - Voltage Regulator. Check the voltage regulator setting at the intervals specified in the Special Inspection Checksheet, chapter 1, and adjust. Refer to TM 55-1500-204-25/1.

a. Deleted.

b. Start the engine, keep the collective pitch full down, and allow N1 idle speed to stabilize according to TM 55-1520-214-10.

c. Turn off unnecessary electrical loads.



NOTES:
1. NOT USED ON ALL AIRCRAFT.
2. ALTERNATE INSTALLATION OF MOUNTING HARDWARE.

Figure 9-9. Instrument Panel Electrical Component Replacement (Series 1 and 2 Aircraft).

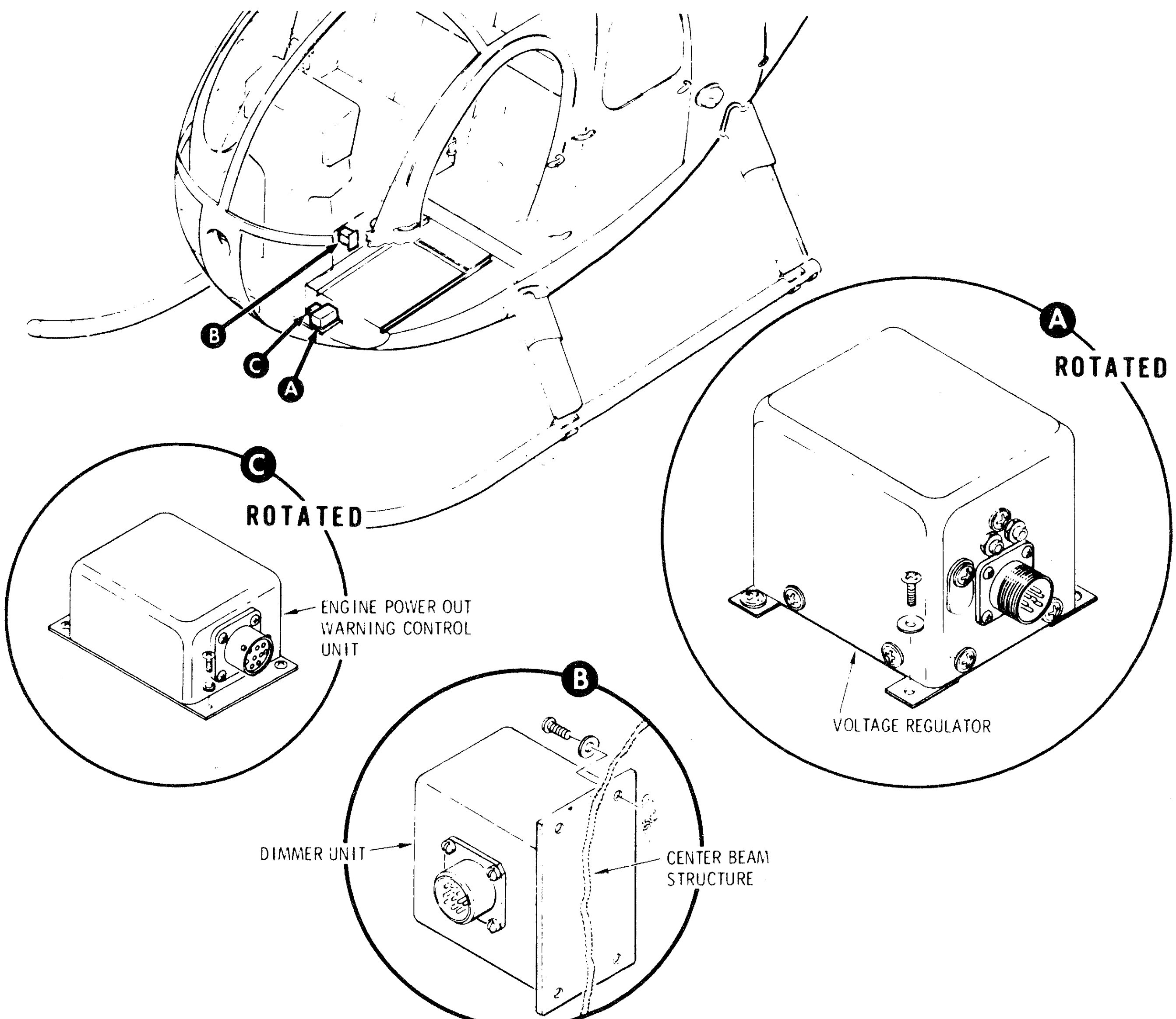


Figure 9-10. Underfloor Electrical Components (Series 3 Aircraft).

d. Increase N1 rpm to 68-70 percent and check the voltmeter reading. If the reading is not within the applicable range listed below, open the adjustment screw cover on the voltage regulator, insert a small screwdriver tip, and turn the adjustment screw slowly for voltage change. Increments of 0.05 volt are possible.

Ambient ground level temperature	Voltage regulator setting (volts)
Winter 32°F or lower	28.5 ±0.2
Summer 80°F or higher	27.0 ±0.2
Fall Spring 32°F to 80°F	28.0 ±0.2

NOTE

To preclude frequent voltage regulator setting changes, the ambient ground level temperature must be considered as a mean (average) value. For example, if the voltage regulator is checked during the month of December in the northeastern United States and the ambient temperature at that time of day is 82°F, it is doubtful if that temperature would be a mean value, but rather a daily high; therefore, maintenance personnel must consider the mean or average weekly ambient temperature rather than the ambient temperature on a single day.

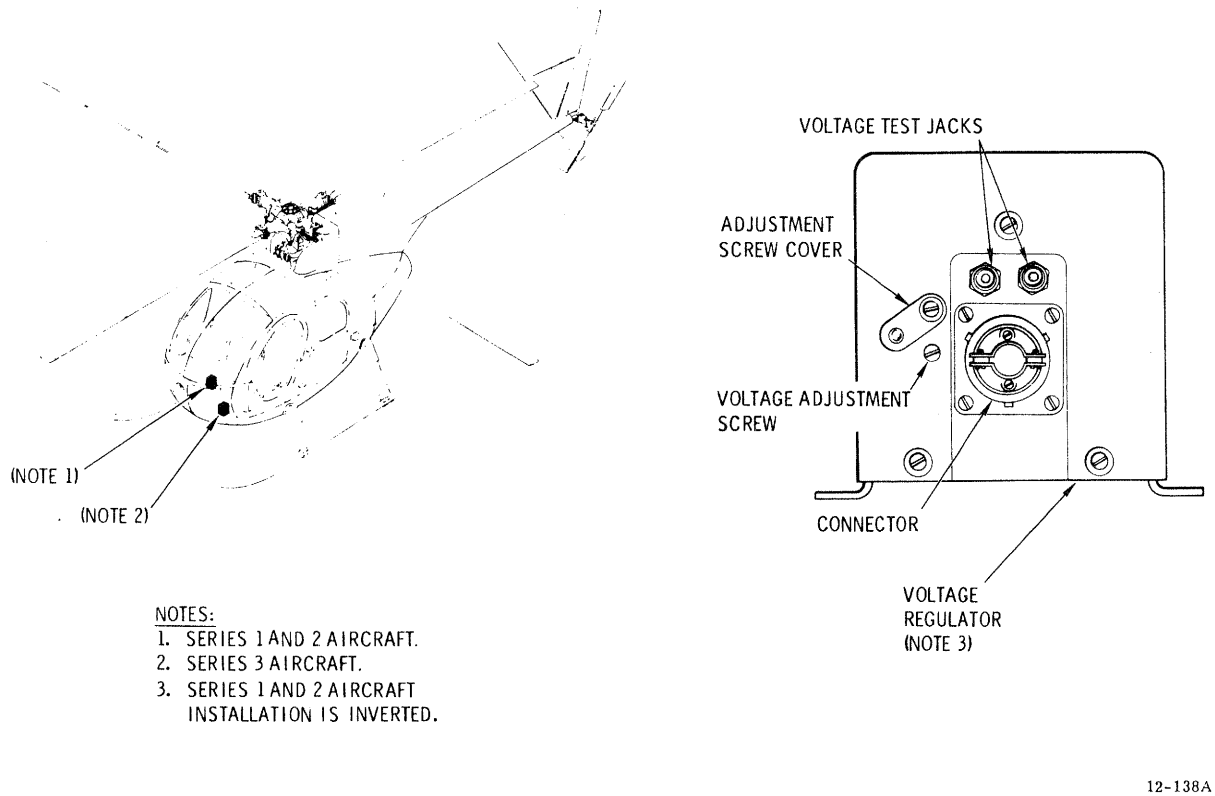


Figure 9-11. Voltage Regulator Adjustment Provisions.

e. While maintaining full down collective pitch, increase N1 rpm to approximately 77-80 percent.

f. Check the voltmeter to see if the voltage remains within the applicable range given above. If necessary, readjust the voltage to remain within the applicable limits at the engine speeds specified in steps *d* and *e* above.

g. Rapidly increase the electrical loads by turning on all radio and electrical equipment possible. Except for momentary fluctuations, the regulator must maintain generator output indicated on meter within range. If the electrical loads cause the voltage to vary beyond acceptable limits, replace the voltage regulator and repeat steps *a* through *f*.

h. Turn off electrical loads, reduce N1 rpm to 68-70 percent, and repeat step *g*. Replace the regulator if the voltage setting varies beyond acceptable limits with an increase in electrical load.

i. Shut down the engine (TM 55-1520-214-10).

j. Deleted.

k. Secure the voltage regulator adjustment screw cover with the retaining screw.

9-35. Removal — Voltage Regulator. (See fig. 9-9 and 9-10.) *a.* Set the power selector switch at OFF.

b. Disconnect the electrical connector at the back of the instrument panel or in the left-side electronics compartment.

c. Remove four mounting screws and washers and lift out the voltage regulator.

9-36. Test (AVIM). — Voltage Regulator. (See fig. 9-12.) *a.* Assemble a test setup as shown in figure 9-12.

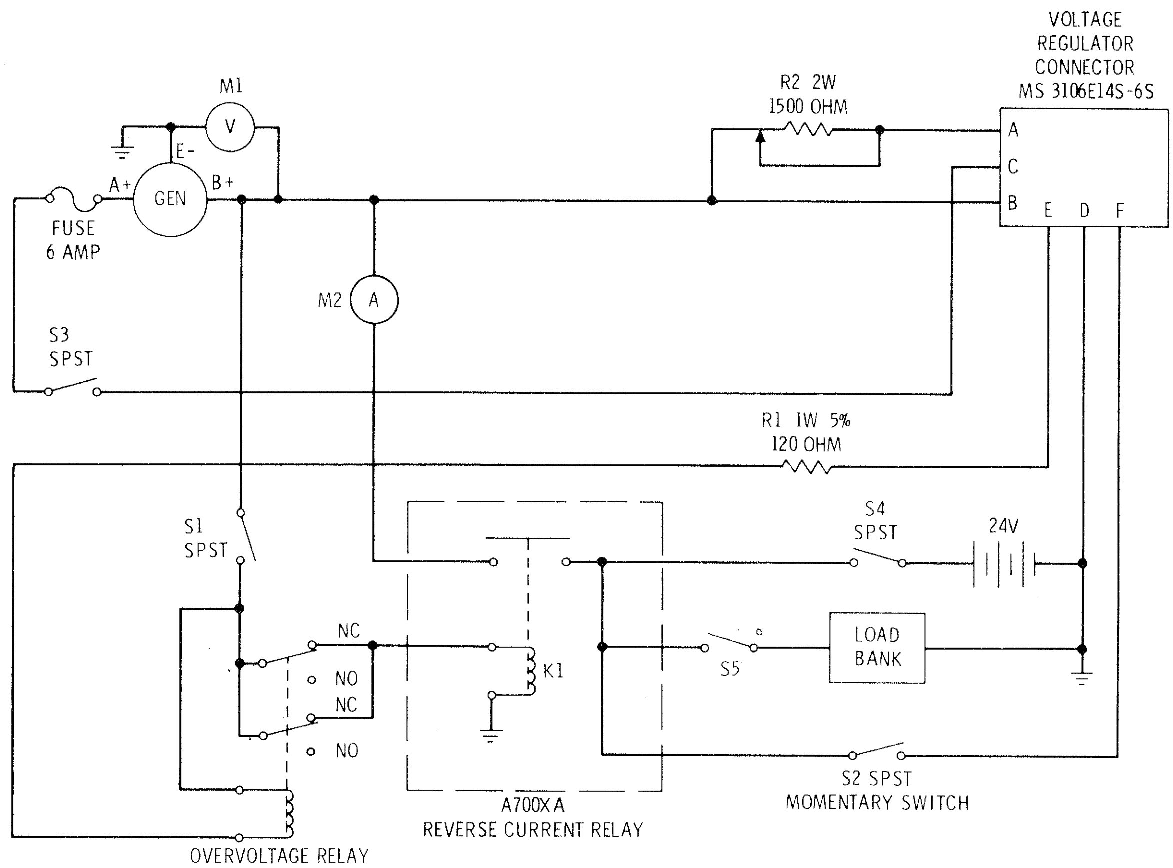
b. Mount the generator on the vari-drive.

c. Connect the voltage regulator into the test circuit as shown in figure 9-12.

d. Adjust potentiometer R2 fully counterclockwise while the load bank is off (switch S5 off). Set switch S1 at off and S3 and S4 at on.

e. Hold the momentary contact switch S2 at on, start the vari-drive and adjust it to bring the generator speed to 7200 rpm or above. Voltage shown on meter M1 must not exceed 10.75 volts; this indicates that the voltage regulator is in the simulated start mode.

f. Release switch S2. The regulator must regulate



TEST SETUP COMPONENTS

VARI-DRIVE, 3000 - TO 12,000 - RPM

DC GENERATOR, 28 - VOLT, 369A4550, OR EQUIVALENT

NI - CAD BATTERY, 24 - VOLT, 369A4530 OR EQUIVALENT

AMMETER (M2), 200 AMPERES FULL SCALE

DC VOLTMETER (M1), CALIBRATED WITHIN 0.25%, AT 0 - 50 VOLTS

LOAD BANK, 0 - 150 AMPERES

OVERVOLTAGE RELAY, 369A4557, OR EQUIVALENT

REVERSE CURRENT RELAY, A700XA, OR EQUIVALENT

SWITCHES, FUSE, RESISTANCES, AND WIRING AS SHOWN

12-154B

Figure 9-12. Voltage Regulator Test Setup.

voltage at between 26 and 30 vdc as indicated on meter M1.

g. Check that full range of voltage adjustment potentiometer on voltage regulator is 26 to 30 vdc as indicated on meter M1. Readjust the potentiometer to 27.6 volts.

h. Adjust the generator speed to 10,750 rpm. Set switch S1 at on and adjust the potentiometer on the voltage regulator to 27.8 vdc as indicated on meter M1.

i. Set switch S5 at on and adjust the load bank to a 100-ampere load as indicated on meter M2; turn switch S1 on and then off. The voltage regulator shall regulate voltage under load surge conditions.

j. Decrease the generator speed to 7200 rpm and apply a 100-ampere load. Set switch S1 at on and then at off. The voltage regulator shall regulate voltage under load surge conditions.

CAUTION

Do not maintain voltage applied in step *k* for more than 15 seconds.

k. Adjust the load bank to a 10 to 20 ampere-load. Set switch S4 at off. Adjust potentiometer R2 clockwise until the overvoltage relay operates. This shall occur at 31.5 to 34.5 volts as indicated on meter M1. Return the potentiometer to the full counterclockwise position.

l. Reset the overvoltage relay by setting switch S1 at OFF and then at ON.

m. Turn switches OFF in the following order: S1, S3, S5, and S4. Disconnect the voltage regulator.

9-37. Installation — Voltage Regulator. (See fig. 9-9 and 9-10.) *a.* Set the power selector switch at OFF.

b. Secure the voltage regulator to the structure with four screws and washers.

c. Attach the electrical connector.

d. Perform an operational check of the regulator.

9-38. OVERVOLTAGE RELAY.

9-39. Description — Overvoltage Relay. The overvoltage relay protects the 28 vdc system from an excessive voltage condition. With the generator switch on, the normally closed contacts of the overvoltage relay apply generator output to the SW (switch) circuit of the reverse current relay. When an overvoltage occurs in the range of 31.5 to 34.5 volts, the overvoltage relay energizes, removing power from the switch circuit of the reverse current relay to remove generator output from the electrical system.

CAUTION

No attempt should be made to adjust the overvoltage circuit in the aircraft. Adjustment is performed only during bench test of the voltage regulator. Refer to paragraph 9-31.

9-40. Inspection — Overvoltage Relay. (See fig. 9-9 and 9-13.) *a.* Inspect the case for damage, deterioration, and corrosion.

b. Inspect for bent, cracked, or broken terminal hooks or pins and for signs of arcing.

9-41. Removal — Overvoltage Relay (Series 1 and 2 Aircraft). (See fig. 9-9.) *a.* Set the power selector switch at OFF.

b. Gain access to the overvoltage relay behind the instrument panel structure.

c. Unsolder electrical wiring from the relay terminal hooks.

d. Remove screws, nuts, and washers that attach the overvoltage relay to the relay panel, and remove the relay.

9-42. Removal — Overvoltage Relay (Series 3 Aircraft). (See fig. 9-13.) *a.* Set the power selector switch at OFF.

b. Remove screws that attach the switch panel to the console and lift the panel out far enough to gain access to the overvoltage relay attaching screws.

c. Remove the two screws and the washers that attach the relay to its socket mounting studs and unplug the relay.

NOTE

The relay socket hardware does not have to be loosened or removed.

9-43. Installation — Overvoltage Relay (Series 1 and 2 Aircraft). (See fig. 9-9.) *a.* Set the power selector switch at OFF.

b. Secure the overvoltage relay to the panel with screws, nuts, and washers.

c. Solder wires to the relay terminal hooks.

9-44. Installation — Overvoltage Relay (Series 3 Aircraft). (See fig. 9-13.) *a.* Set the power selector switch at OFF.

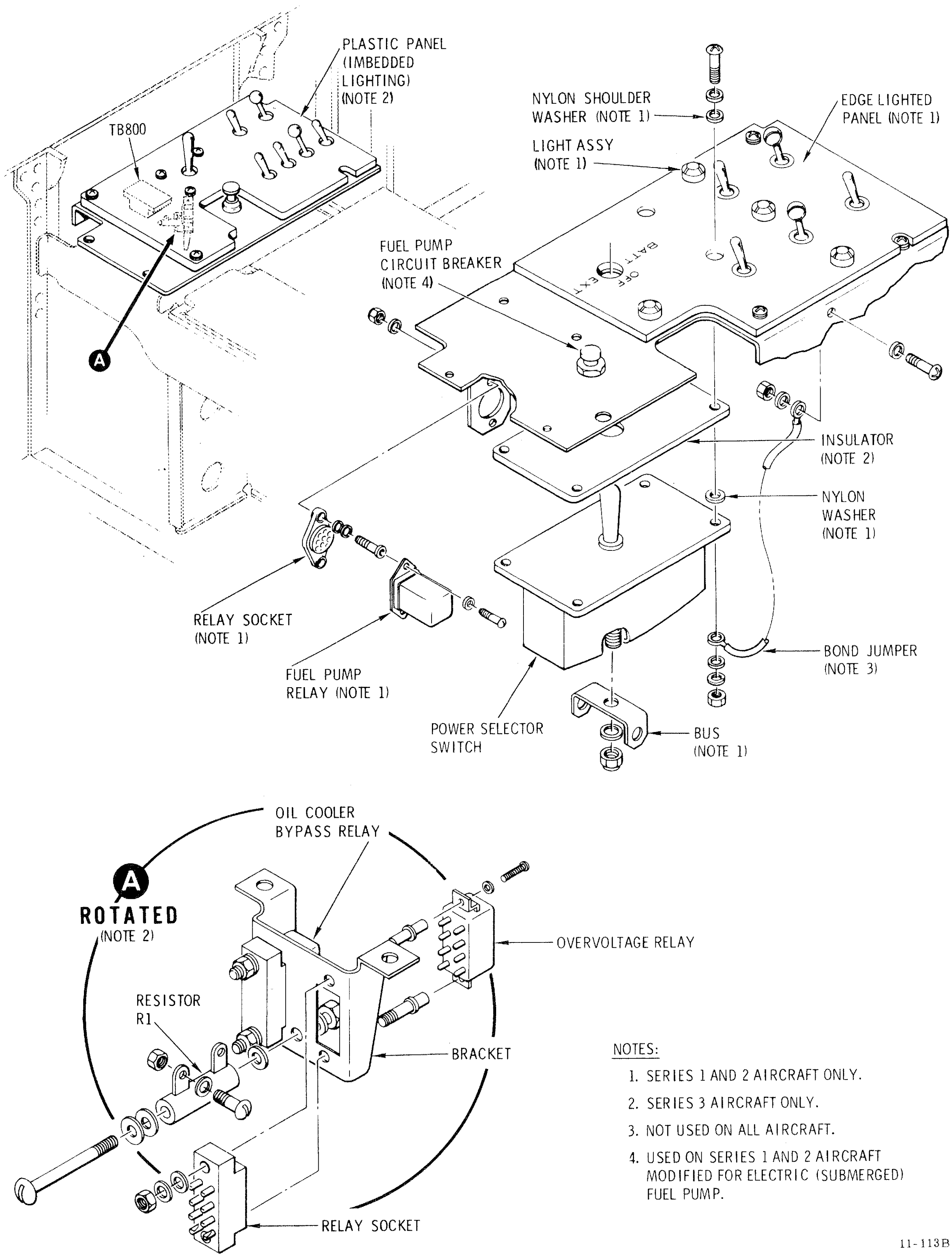
b. Insert the overvoltage relay in the socket and secure the relay with two screws and two washers.

c. Position the switch panel and secure it with attaching screws.

9-45. REVERSE CURRENT RELAY.

9-46. Description — Reverse Current Relay. The reverse current relay, mounted on the oleo support fitting at the right side of the engine compartment, opens and closes the circuit to the battery and the 28-vdc bus from the generator. The relay isolates the generator from the 28-vdc bus (and battery) when the generator switch is OFF, an overvoltage condition exists, or when the battery voltage is greater than the generator voltage and current flows in reverse (battery to generator).

9-47. Inspection — Reverse Current Relay. (See fig. 9-14.) *a.* Examine the housing for dents, cracks, or corrosion at bonding locations.



NOTES:

1. SERIES 1 AND 2 AIRCRAFT ONLY.
2. SERIES 3 AIRCRAFT ONLY.
3. NOT USED ON ALL AIRCRAFT.
4. USED ON SERIES 1 AND 2 AIRCRAFT MODIFIED FOR ELECTRIC (SUBMERGED) FUEL PUMP.

11-113B

Figure 9-13. Electrical Console Switch Panel.

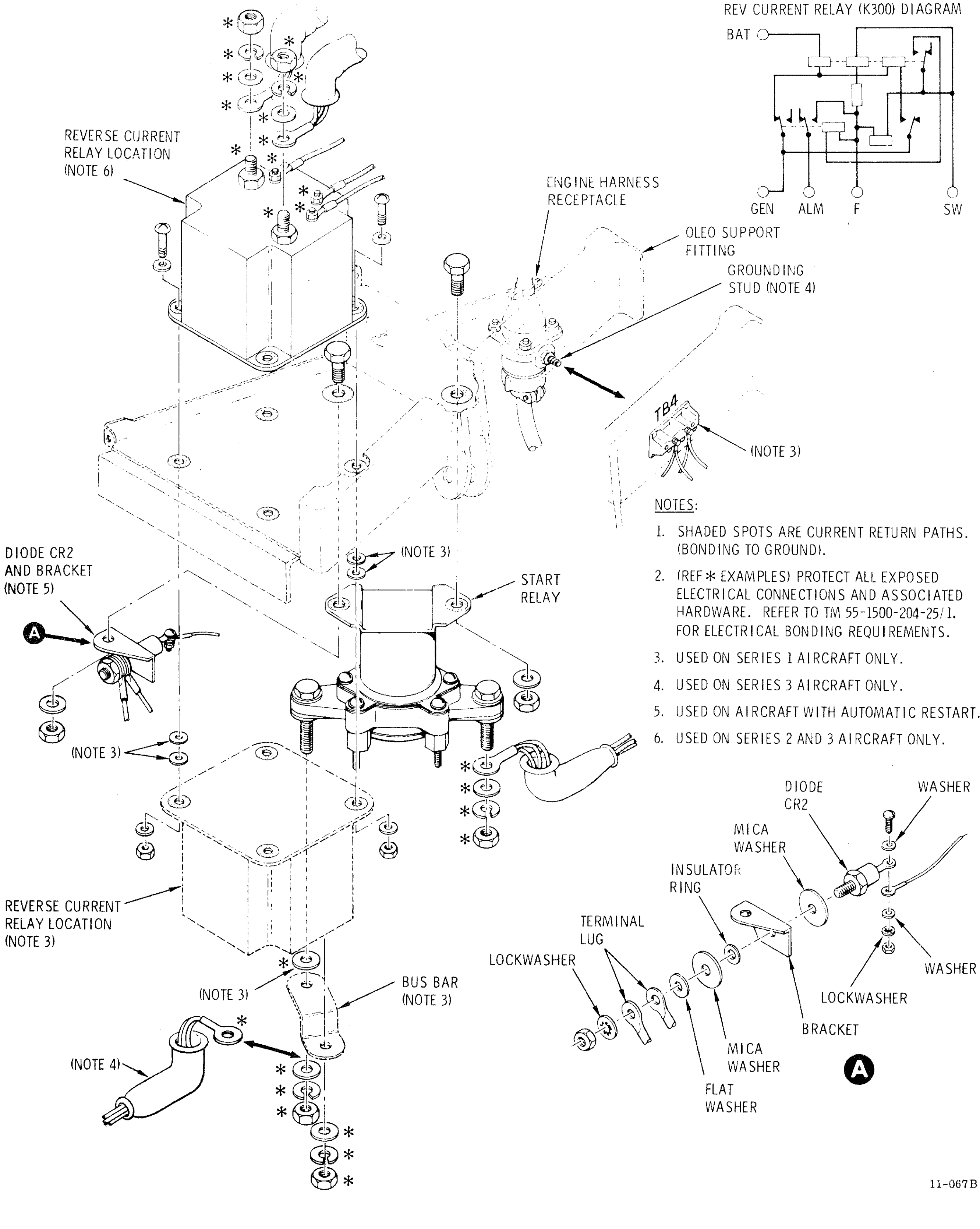


Figure 9-14. Relay and Diode Installation — Engine Compartment.

b. Examine terminal studs for corrosion and signs of arcing.

9-48. Removal — Reverse Current Relay. (See fig. 9-14.) *a.* Set the power selector switch at OFF.

b. Remove wires from the relay terminal studs. (Retain nuts and washers by re-installing them on the studs after wire removal.)

c. Insulate the terminal of the wire removed from the BAT stud to preclude any chance of shorting.

d. Remove the relay mounting screws, nuts, and washers, and remove the relay.

9-49. Installation — Reverse Current Relay. (See fig. 9-14.) *a.* Set the power selector switch at OFF.

b. Prepare the bonding surfaces on the relay and structure at the locations shown in figure 9-14. Refer to TM 55-1500-204-25/1 for methods.

c. Secure the relay to the structure with mounting screws, nuts, and washers.

d. Install wires on the relay terminal studs.

e. Apply lacquer (C53) to all electrical connections, including those surfaces left exposed by bonding preparation in step *b* above.

9-50. POWER SELECTOR SWITCH.

9-51. Description — Power Selector Switch. The power selector switch, marked BATT-OFF-EXT, is a single-pole three-position toggle switch with a continuous current capacity of 175 amperes at 28 vdc. The switch installation has two configurations. In the first configuration, the switch case is mounted directly to the edgelighted switch panel plate with electrical bonding between the case and plate. In the second configuration, the switch case is electrically isolated from the panel plate with bonding provided through a jumper wire to the console structure. The edgelighted switch panel and panel plate are both slightly modified for the isolated switch configuration; therefore, a replacement for either part must match the particular power switch configuration. See figure 9-13.

9-52. Removal — Power Selector Switch. (See fig. 9-13.) *a.* Remove external power (fig. 9-4).

b. Disconnect the battery (fig. 9-5).

NOTE

The isolated switch configuration can be identified by checking the edgelighted panel. If the heads of the left two screws that secure the switch to the plate are visible (holes cut through the edgelighted panel), the switch is isolated.

c. If the switch is not isolated (left two screw heads not visible), remove the edgelighted panel by removing, in order, the lamp covers and lamps, the panel attaching screws, and the panel.

NOTE

It is not necessary to remove the edgelighted panel for switch removal in the isolated-switch configuration.

d. Cut as many nylon straps as necessary to separate the switch panel harness wiring from the wiring routed to the circuit breaker panel.

e. Remove screws that secure the metal switch panel to the console. Carefully raise the right edge of the panel (looking forward) about three to four inches. Do not use undue force; cut additional nylon straps as necessary.

f. Working from right to left, slide back the rubber boot from each switch terminal stud and disconnect the wire terminals from the studs. Tag the wires to identify post location. On series 1 and 2 aircraft, remove the U-shaped bus bar.

g. Remove the four screws that secure the switch to the panel. See figure 9-13 for differences in method of attachment (jumper, insulator, nylon washers).

h. Turn the switch on its side to clear the toggle from the plate hole, and hold it in this position while withdrawing it toward the right.

9-53. Inspection — Power Selector Switch. Inspect an uninstalled switch for signs of arcing or burn-through, cracks, and general condition. Examine the terminal stud threads for usability.

9-54. Installation — Power Selector Switch. (See fig. 9-13.) *a.* Remove external power (fig. 9-4).

b. Disconnect the battery. See figure 9-5.

c. On those aircraft using a bonding jumper prepare the electrical bonding surfaces indicated in figure 9-13. Refer to TM 55-1500-204-25/1 for method.

d. Place the switch on its side (toggle aft) and slide it under the metal panel. Rotate the switch into position. For installations using an insulator plate, see figure 9-13 for sequence of installation.

e. Secure the switch to the metal panel as shown in figure 9-13, making certain that on series 1 and 2 aircraft, the nylon washers are installed as shown; also, that the longest of the four attaching screws is used to connect the jumper on those installations where a jumper is used.

f. Remove the temporary identification tag and secure the wire terminal on the left (looking forward) switch terminal stud, using washer, lockwasher, and nut. Slide the rubber boot over the switch terminal stud.

g. On series 1 and 2 aircraft, install the U-shaped bus bar on the center terminal stud of the switch as shown in figure 9-13, aligning the length of the bus bar forward and aft. Install wiring as described in step *f* above.

h. On series 3 aircraft, secure wires to the center terminal stud of the bus as described in step *f* above.

i. Secure wire terminal to the right (looking forward) switch terminal stud as described in step *f* above.

j. Position the switch panel on the console and secure with screws as shown in figure 9-13.

CAUTION

Severe electrical arcing between the power selector switch wire bundle and flight controls may occur unless step *k* below, is precisely performed. If unrecorded maintenance in this area is suspected, check for adequate clearance of all wire bundles. The entire area should be inspected for wire clearance after any maintenance action requiring the switch panel to be removed or lifted up from its permanent position on the console.

k. Secure the wiring harness routed under the switch panel and through the console. Inspect under the panel, particularly in the area where the battery cable connects to the power selector switch. Make certain that the terminals are covered with rubber boots and that the wire bundles are routed away from the terminals at 90 degrees (right angles). Be sure that there is enough clearance between wire bundles and the cyclic torque tube when the cyclic stick is in its extreme positions. Wiring may touch only structure edges that have nylon insulating strips.

l. On series 1 and 2 aircraft, secure the edgelighted panel by installing, in order, the panel, attaching screws, lamps, and lamp covers.

m. Operate the fuel shutoff valve control to be sure there is no binding.

n. Set the power selector switch at OFF.

o. Connect the battery. See figure 9-5.

9-55. Test (Operational Check) — Power Selector Switch. *a.* With external power source disconnected from the aircraft set the power selector switch at BAT and turn on any of the aircraft electrical equipment, such as lights, to verify that battery power is available to the direct-current distribution system.

b. Set the power selector switch at BAT and then at EXT and verify that equipment turned on in step *a* above ceases to function in both positions.

c. Set the power selector switch at OFF.

d. Turn off aircraft electrical equipment that was turned on in step *a* above.

e. Connect external power to the aircraft. See figure 9-4.

f. Set the power selector switch at EXT and turn on any of the aircraft electrical equipment, such as lights, to verify that external power is available to the direct-current distribution system.

g. Turn off the aircraft electrical equipment turned on in step *f* above.

h. Set the power selector switch at OFF.

i. Remove external power from the aircraft. See figure 9-4.

9-56. CIRCUIT BREAKER PANEL.

9-57. Description — Circuit Breaker Panel. The edgelighted circuit breaker panel, located on the pilot's collective pitch stick cover, is a removable assembly that supports pushbutton-type circuit breakers. Thirteen breakers are used in series 1 and 2 aircraft and 23 breakers (plus provisions for five more) are used in series 3 aircraft. (See fig. 9-15 and 9-16.) The panel in series 1 and 2 aircraft contains three removable panel lamps whereas the series 3 aircraft panel has imbedded (nonreplaceable) lamps. The series 3 aircraft panel assembly also includes a standoff-mounted bracket that supports the electric fuel pump relay and landing light relay. The circuit breakers are normally depressed (button in). When a circuit is overloaded, the circuit breaker button pops out and opens the circuit.

9-58. Inspection — Circuit Breaker Panel. (See fig. 9-15 and 9-16.) *a.* Set the power selector switch at OFF.

b. Inspect plastic panel for damage and legible placarding.

c. Inspect for broken wires, frayed or broken insulation, loose connections, and burned terminals.

d. On series 1 and 2 aircraft, inspect the circuit breaker housing for cracks, breaks, or distortion.

e. On series 3 aircraft, inspect the bracket-mounted relays for signs of arcing and secure attachment.

f. Inspect brass bus bars for signs of arcing, corrosion and secure attachment.

9-59. Removal — Circuit Breaker (Typical). (See fig. 9-15 and 9-16.) *a.* Set the power selector switch at OFF.

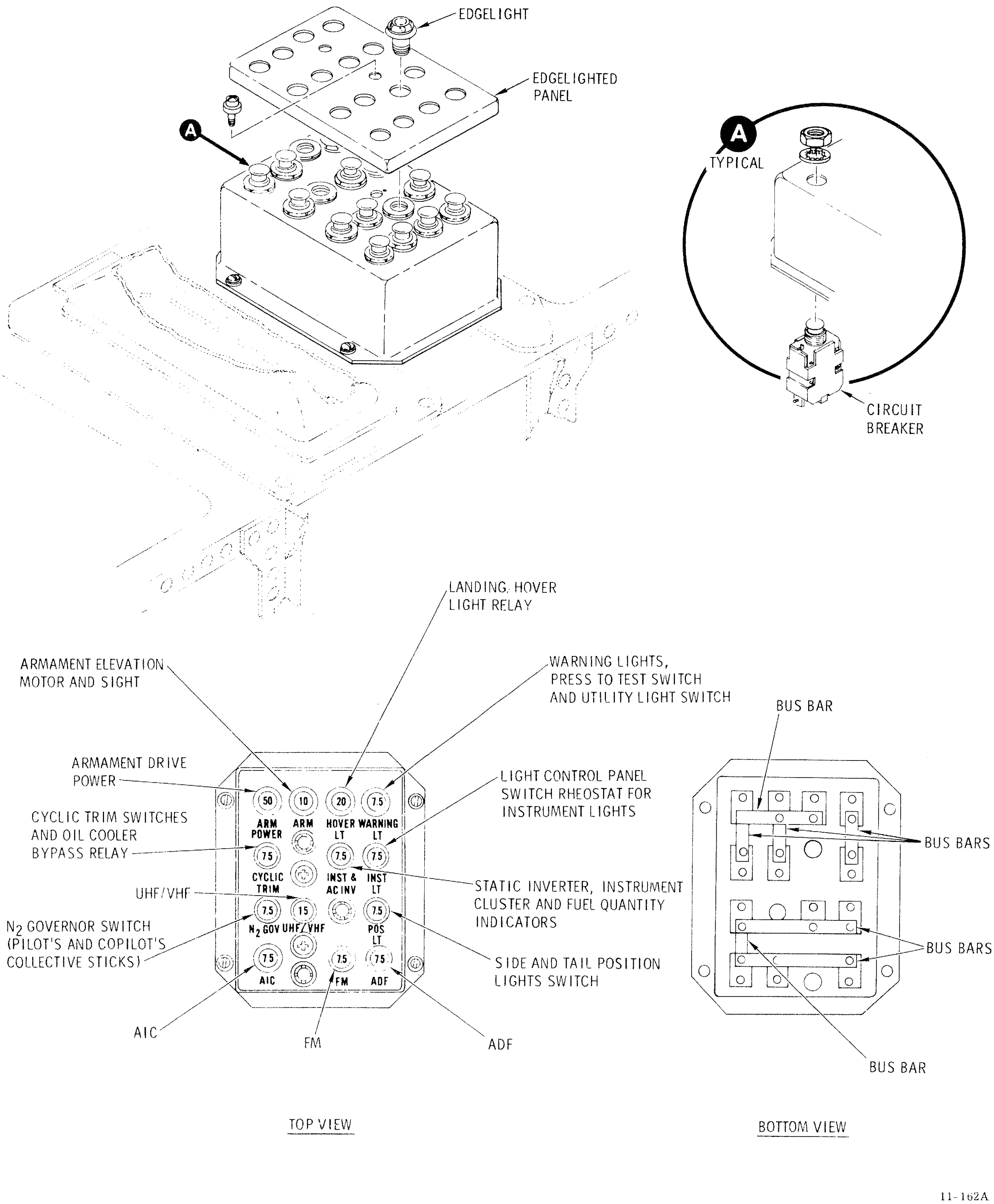


Figure 9-15. Circuit Breaker Panel (Series 1 and 2 Aircraft).

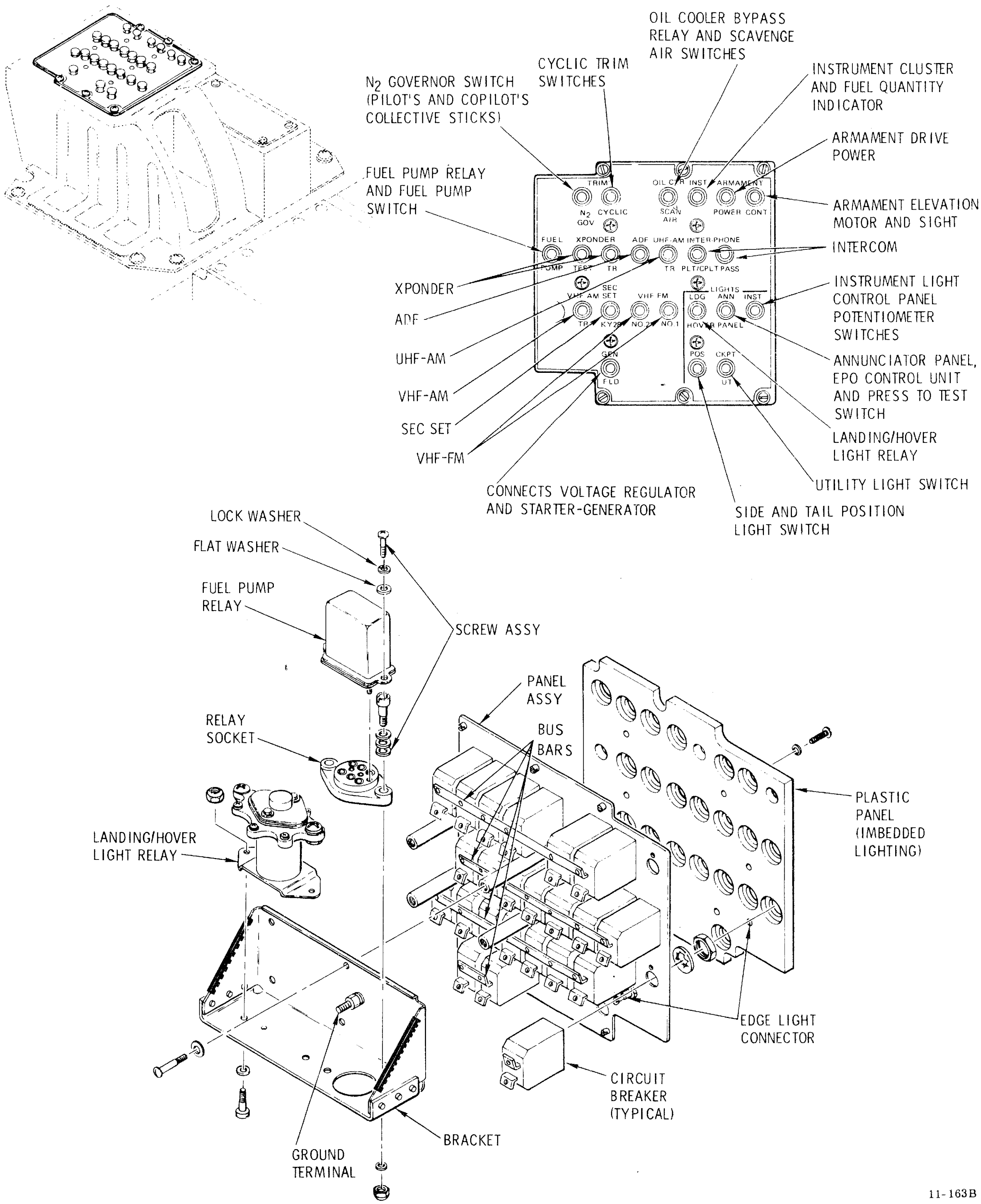


Figure 9-16. Circuit Breaker Panel (Series 3 Aircraft).

b. Release the turnlock fasteners that attach the circuit breaker panel assembly to the collective stick cover. Lift the panel assembly out to expose the circuit breaker electrical connectors.

c. Remove the plastic panel attaching screws and lift off the panel.

d. Disconnect the wiring and bus bar(s) from the circuit breaker. Remove the nut, lockwasher, and circuit breaker.

9-60. Installation — Circuit Breaker (Typical). (See fig. 9-15 and 9-16.) *a.* Set the power selector switch at OFF.

b. Secure the circuit breaker in place with lockwasher and nut.

c. Connect the wiring and bus bar(s).

d. Secure the plastic panel to the panel assembly with screws and washers.

e. Place the circuit breaker panel assembly in the collective stick cover and secure it with the turnlock fasteners.

9-61. Removal and Installation — Lamp (Typical). (See fig. 9-15 and 9-16.) *a.* On series 1 and 2 aircraft, unscrew the lamp cover, remove the lamp, and install another lamp, base inward. Install and finger-tighten the lamp cover.

b. On series 3 aircraft, remove the plastic panel by taking out six attaching screws and washers. Install another plastic panel.

9-62. ELECTRICAL CONSOLE SWITCH PANEL.

9-63. Description — Electrical Console Switch Panel. The electrical console switch panel is located on the electrical console just aft of the light control panel. See figure 9-13. In addition to switches, the panel contains the fuel valve control. Some of the switches are of the circuit-breaker type that snap to the OFF position when overloaded. All of the switches are individually replaceable. On series 1 and 2 aircraft, the plastic (outermost) panel is edgelighted by individually replaceable lamps whereas on the series 3 aircraft the plastic panel has imbedded lights that are not replaceable and defective lighting can only be corrected by installing another plastic panel. On series 1 and 2 aircraft, the switch panel assembly supports the electric fuel pump relay on those aircraft that have such a pump. On series 3 aircraft, the panel assembly supports the overvoltage relay, the overvoltage relay sensing circuit resistor, the oil cooler bypass relay, and a wiring terminal block TB800.

9-64. Removal — Electrical Console Switch Panel. (See fig. 9-13.) *a.* Set the power selector switch at OFF.

b. On series 1 and 2 aircraft, unscrew the lamp covers and remove the edgelight lamps.

c. Remove the screws that attach the plastic panel to the metal panel, and remove the plastic panel.

NOTE

On series 3 aircraft, the switch panel assembly and its harness can be completely removed by disconnecting electrical connector P118 if the wires to the power selector switch are removed. Refer to paragraph 9-50.

d. On series 1 and 2 aircraft, cut nylon straps as necessary to separate the switch panel harness from adjoining harness and obtain slack and flexibility in the wiring.

e. Remove the screws that secure the switch panel to the console. Carefully lift the panel as far out as possible to gain access to panel components.

f. For complete removal of the metal switch panel:

(1) Remove the power selector switch. Refer to paragraph 9-52.

NOTE

If the metal switch panel has a cutout at the forward edge, the fuel valve control cable need not be removed.

(2) Remove the fuel valve control cable assembly. Refer to chapter 10.

(3) On series 1 and 2 aircraft, disconnect wiring to switch panel components and tag each wire to identify its termination.

(4) On series 1 and 2 aircraft, remove the electric fuel pump relay if one is installed. Refer to paragraph 9-223.

(5) On series 3 aircraft, remove the overvoltage relay (para 9-42), the oil cooler bypass relay (para 9-227), wiring terminal block 800 (fig. 9-13), and the R1 overvoltage relay sensing circuit calibration resistor (para 9-74).

(6) Remove remaining switches and edgelight sockets (series 1 and 2 aircraft) by removing nuts and lockwashers.

(7) On series 1 and 2 aircraft, remove the electric fuel pump circuit breaker (if one is installed) by removing a nut and lockwasher.

9-65. Inspection — Electrical Console Switch Panel. (See fig. 9-13.) *a.* Examine the plastic panel for breaks, cracks, and unclear placarding.

b. Examine switches for cracks, signs of arcing, and loose terminals.

c. On series 1 and 2 aircraft, if an electric fuel pump circuit breaker has been installed, examine it for cracks, signs of arcing, and loose terminals.

9-66. Installation — Electrical Console Switch Panel. (See fig. 9-13.) Perform the following steps as applicable, depending on the extent to which the switch panel assembly has been removed and components detached from it.

a. Disconnect external power. See figure 9-4.

b. Disconnect the battery. See figure 9-5.

c. Install all switches excepting the power selector switch with lockwashers and nuts.

d. On series 1 and 2 aircraft, install the edgelight sockets with lockwashers and nuts.

e. On series 1 and 2 aircraft, install the electric fuel pump circuit breaker.

f. On series 1 and 2 aircraft, install the electric fuel pump relay if provisions exist for it. Refer to paragraph 9-224.

g. On series 3 aircraft, install the overvoltage relay (para 9-44), the oil cooler bypass relay (para 9-227), terminal block 800 (fig. 9-13), and the R1 overvoltage relay sensing circuit calibration resistor (para 9-74).

h. Connect wiring to the electrical components installed in steps *c* through *g* above.

i. Install the power selector switch as set forth in paragraph 9-54 and connect wires and cables to it.

j. On series 3 aircraft, join and secure electrical connectors J118 and P118.

k. If removed, install the fuel valve cable control assembly. Refer to chapter 10.

l. Secure the metal switch panel to the console with screws.

m. Secure the plastic panel to the metal panel with screws and washers.

n. On series 1 and 2 aircraft, insert edgelight lamps and install covers.

o. Arrange, tie, and secure harness and wiring with nylon straps to prevent chafing and interference. (Wiring shall not touch structure edges that are without nylon insulation strips. Wiring shall clear flight controls in all positions.) Refer to paragraph 9-3.

p. Set all switch panel switches at OFF.

q. Connect the battery. See figure 9-5.

r. Connect external power (fig. 9-4), set the power selector switch at EXT and perform an operational check of those circuits affected by removal and installation of components.

s. Set the power selector switch at OFF.

9-67. ELECTRICAL CONSOLE LIGHT CONTROL PANEL.

9-68. Description — Electrical Console Light Control Panel. The light control panel is located on the electrical console and has an ash receiver at the forward end. On series 1 and 2 aircraft the four variable resistors are switch-rheostats whereas on series 3 aircraft they are potentiometers. On series 3 aircraft the FLIGHT potentiometer actuates a microswitch S6 in the annunciator panel bright-dim control circuit. See figure 9-17.

9-69. Removal — Electrical Console Light Control Panel. (See fig. 9-17.) *a.* Set the power selector switch at OFF.

b. On series 1 and 2 aircraft, unscrew the four lamp covers and remove the edgelight lamps.

c. Loosen setscrews and remove the four switch-rheostat (potentiometer) knobs.

d. Remove the screws and washers that secure the plastic panel to the plate, and remove the plastic panel.

e. Remove the screws that secure the plate to the console.

f. On series 3 aircraft, detach the electrical connector P115 and remove the remainder of the light control panel assembly.

g. On series 1 and 2 aircraft, perform the following steps.

(1) Cut twine and nylon straps as necessary to separate light control panel harness from adjoining harness and obtain slack and flexibility in the wiring.

(2) Disconnect wires from the switch, the switch-circuitbreaker, the four switch-rheostats, and the four edgelight sockets. Tag each wire to identify its termination.

(3) Remove the remainder of the light control panel assembly from the console.

9-70. Disassembly — Electrical Console Light Control Panel. (See fig. 9-17.) Remove the nuts and lockwashers that secure the switch, the switch-circuit-breaker, the four switch-rheostats (potentiometers), and, on series 1 and 2 aircraft, the four edgelight sockets.

NOTE

On series 3 aircraft, removal of the FLIGHT potentiometer also releases a bracket that supports a microswitch. Do

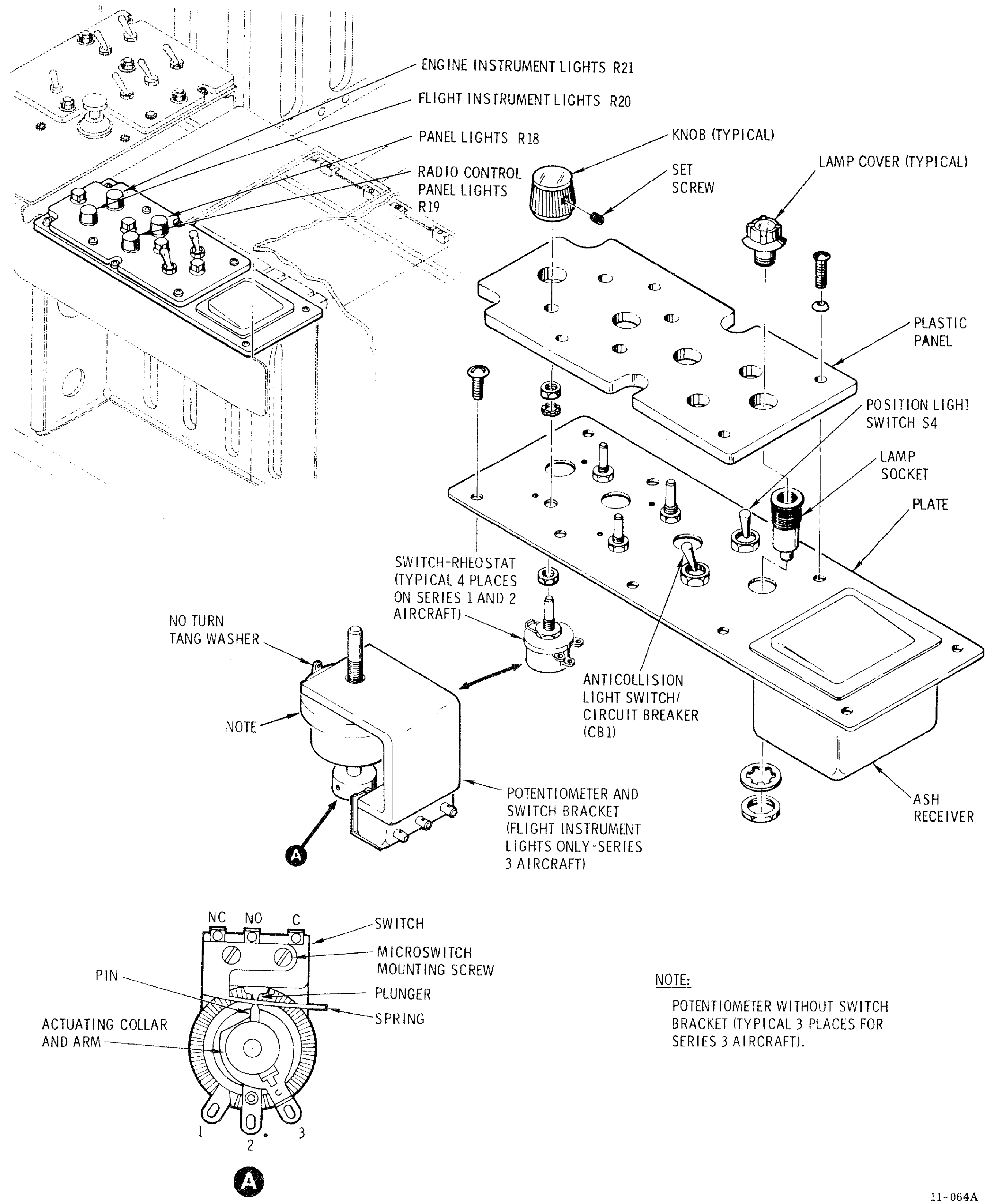


Figure 9-17. Electrical Console Light Control Panel.

not change the switch position in the bracket unless replacement of either the switch or the potentiometer is intended. Refer to paragraph 9-72 for assembly instructions.

9-71. Inspection — Electrical Console Light Control Panel. (See fig. 9-17.) *a.* Examine the plastic panel for breaks, cracks, and unclear placarding.

b. Examine switches, switch-rheostats (potentiometers), and the edgelight sockets for cracks, signs of arcing, and loose terminals.

c. Examine switch-rheostat (potentiometer) knobs for cracks and breaks, especially in the setscrew area.

9-72. Assembly — Electrical Console Light Control Panel. (See fig. 9-17.) *a.* On series 1 and 2 aircraft, secure the switch, the switch-circuitbreaker, the four edgelight sockets, and the four switch-rheostats (potentiometers) to the plate with lockwashers and nuts.

b. On series 3 aircraft, perform step *a* except for installation of the edgelight sockets and for installation of the FLIGHT potentiometer which requires the following procedure.

(1) Secure the FLIGHT potentiometer R20 and the microswitch S6 bracket to the plate with a lockwasher and a nut.

(2) Check the extent to which the no-turn tang washer of the potentiometer extends. If it will interfere with the plastic panel, trim it to clear.

(3) Turn the potentiometer to the OFF position, loosen the screws that attach the microswitch to the bracket, and loosen the actuating collar setscrew.

(4) Locate the microswitch so that it is actuated to the on position by the arm of the actuating collar; then, tighten the microswitch attaching screws and the actuating collar setscrew.

(5) Operate the potentiometer several times to make sure that when the potentiometer is at the OFF position the microswitch is actuated to its on position.

(6) Secure nuts on the microswitch attaching screws and on the potentiometer actuating collar setscrew with locking compound (C45). Keep locking compound away from the microswitch plunger.

9-73. Installation — Electrical Console Light Control Panel. (See fig. 9-17.) *a.* On series 3 aircraft:

(1) Secure the plate to the console with screws.

(2) Secure the plastic panel to the plate with washers and screws.

(3) Install the four switch-rheostat (potentiometer) knobs and tighten the setscrews.

(4) Join and secure harness connectors P115 and J115. (Refer to paragraph 9-4.)

(5) Arrange and secure harness and wires. (Refer to paragraph 9-3.)

b. On series 1 and 2 aircraft:

(1) Connect wires to panel components. (Refer to wiring diagrams in appendix F.)

(2) Perform step *a*, substeps (1) through (3) above.

(3) Arrange and secure harnesses and wires. (Refer to paragraph 9-3.)

(4) Insert the four edgelight lamps and install the lamp covers.

c. Connect external power (fig. 9-4), set the power selector switch at EXT, and perform an operational check of circuits affected by the maintenance action.

d. Set the power selector switch at OFF.

9-74. FIXED RESISTORS.

9-75. Description — Fixed Resistors. Fixed resistors in the direct-current power distribution system are described and located in table 9-4. Refer also to wiring diagrams in appendix F.

9-76. Removal — Fixed Resistors. *a.* Set the power selector switch at OFF.

b. Locate resistor by referring to table 9-4.

c. Gain access to resistor and cut nylon straps or twine as necessary to detach resistor.

d. On series 1 and 2 aircraft, use a pencil-type soldering iron to unsolder the resistor.

e. On series 3 aircraft, remove nuts, washers, and screws from the resistor terminals.

9-77. Installation — Fixed Resistors. (Refer to table 9-4.) *a.* Set the power selector switch at OFF.

b. On series 1 and 2 aircraft, use a pencil-type soldering iron and tin-alloy solder (C93) to electrically connect the resistor.

c. On series 3 aircraft, electrically connect the resistor with screws, washers, and nuts.

d. Arrange wiring or harness and secure (refer to paragraph 9-3).

e. Connect external power (fig. 9-4), set power selector switch at EXT, and perform an operational check of the affected circuit.

f. Set the power selector switch at OFF.

Table 9-4. Fixed Resistors—Direct-Current Power Distribution System.

Reference Designator	Circuit and Purpose	Aircraft Series	Location
R1	In series with the overvoltage relay K103 coil so as to require an overvoltage to actuate the relay.	1 and 2 3	On the right side of the electrical console structure (at bottom of TB5). On the electrical console switch panel (near the power selector switch).
R2	Reduces 24 vdc to 12 vdc for oil temperature indicator M6 and oil pressure indicator M7.	1 2	Aft of resistor board TB5 on the electrical console. Under the pilot's floor skin, right side, aft.
		3	Near R16 on the instrument panel structure.
R3	Reduces voltage for position light dimming.	1 and 2 3	Same as for R2. Same as for R2, aircraft series 2.
R16	In series with ammeter M4 in starter-generator circuit to provide voltage reduction proportional to starter-generator output or current draw.	1 2 3	Right side of engine compartment, on damper support. Same as for R2. Same as for R2.
R17	In series with the oil temperature M6, oil pressure M7, and ammeter M4 panel lights to reduce brilliance.	1,2,3	In the 3-pack cluster enclosure on the instrument panel.

SECTION III ALTERNATING-CURRENT POWER DISTRIBUTION SYSTEM

9-78. STATIC INVERTER ALTERNATING-CURRENT SYSTEM.

9-79. Description — Static Inverter Alternating-Current System. The alternating-current system consists of either one or two static (solid state) inverters, a waveshape filter (series 1 and 2 aircraft) and the inverter circuit breaker as power source components. On aircraft with two inverters, the control switch is a three-position type marked INVERTER 1-2-OFF. On aircraft with a single inverter the switch is marked INVERTER-OFF. Either the single inverter system or the No. 1 inverter of the two-inverter system will provide power for both the gyro of the attitude indicator and the directional gyro of the heading reference facility (homing, heading, bearing indicator). On series 1 and 2 aircraft, power to the directional gyro is fed through a waveshape filter (not a separate unit on series 3 aircraft). Failure of the inverter will cause power OFF flags to appear in the heading-bearing and attitude indicators. With the two-inverter system, the No. 2 inverter can be selected after failure of No. 1 and power will then be provided for operation of the attitude indicator only. The AC BUS caution light is turned on when the No. 2 inverter is selected. Aircraft with the single inverter system do not have the AC BUS caution light or any backup source of ac power. Refer to TM 11-1520-214-20 for maintenance of the inverter and waveform filter as installed in series 1 and 2 aircraft. Refer to TM 11-1520-214-20-1 for maintenance of the inverter as installed in series 3 aircraft. Refer to appendix F for wiring diagrams.

SECTION IV STARTING SYSTEM

9-80. STARTING SYSTEM.

9-81. Description — Starting System. The starting system consists of the starter-generator G300, the start relay K301, the START switch S106 on the pilot's collective stick, the ignition keylock switch S500(KL) located on the right side of the pilot's seat structure, and associated wiring (refer to wiring diagrams in appendix F). There are no circuit breakers in the starting system: power is taken directly from the power selector switch S14. After the ignition key is turned on, pressure on the START switch button causes the start relay to connect power to terminal C of the starter-generator.

9-82. Troubleshooting — Starting System. Refer to table 9-5 and the wiring diagram in appendix F. Use the wiring diagram to check for electrical breaks, shorts, and insecure connections.

9-83. Starter-Generator. Refer to paragraph 9-22 for maintenance instructions for the starter-generator.

9-84. Start Switch. Refer to paragraph 9-214 for inspection, removal, and installation of the START switch on the pilot's collective stick.

9-85. Start Relay. The start relay is mounted aft of the reverse current relay on the oleo support fitting at the right side of the engine compartment.

a. *Inspection.* (See fig. 9-14.) Examine the start relay for dents, cracks, corrosion, and signs of arcing.

b. *Removal.* (See fig. 9-14.)

(1) Set the power selector switch at OFF.

(2) Slide the rubber boots back from the relay terminals and disconnect the wires. To prevent accidental shorting, insulate and tie back the wire terminals that were attached to relay terminals X2 and A1. Tag-identify all wires.

(3) Remove the bolts, nuts, and washers that secure the relay to the oleo support.

NOTE

On aircraft with the automatic restart system, the bracket holding the CR2 diode is attached by one of the relay mounting bolts. It is not necessary to remove the wire from the diode.

c. *Installation.* (See fig. 9-14.)

(1) Prepare those relay bonding surfaces shown in figure 9-14 per TM 55-1500-204-25/1.

(2) Secure relay (and CR2 diode, if the aircraft has a restart system) to the oleo support with bolts, washers, and nuts.

(3) Connect wires with washers and nuts. Refer to wiring diagram in appendix F if there is any question of wire-to-terminal identity.

Table 9-5. Troubleshooting the Starting System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Starter-generator fails to function as a starter when the START switch on the pilot's collective pitch stick is pressed.**

STEP 1. Set the power selector switch at OFF and temporarily jumper terminal A1 to A2 on the start relay K301. See figure 9-14. Then, momentarily set the power selector switch at EXT.

If the starter-generator fails to function as a starter, replace it. Refer to paragraph 9-22.

STEP 2. Set the power selector switch at OFF and temporarily jumper terminal X1 to terminal A1 on the start relay K301. See figure 9-14. Then, momentarily set the power selector switch at EXT.

If the starter-generator fails to function as a starter, replace the start relay (para 9-80).

STEP 3. Set the power selector switch at OFF and temporarily jumper the terminals of the START switch on the pilot's collective pitch stick. (To expose the switch terminals, remove the self-tapping screw in the switch flange and pull the switch and wiring from the stick grip as shown in figure 9-27.) Then, momentarily set the power selector switch at EXT.

If the starter-generator fails to function as a starter, replace the ignition keylock switch (para 9-80).

If the starter-generator functions as a starter, replace the START switch (para 9-214).

(4) Apply a coating of lacquer (C53) to all exposed electrical connections. Allow the lacquer to dry, then slide the rubber boots over the relay studs.

9-86. Ignition Keylock Switch. The ignition keylock switch S500(KL) is located on the right side of the pilot's seat support structure, just aft of the external power receptacle (fig. 9-28). Electrically, the switch is inserted between the 28-vdc power source and the START switch on the pilot's collective pitch stick, thus disabling both the starter and ignition functions when the switch is off.

a. Inspection. Check the switch for security of attachment and for secure wire connections. Make sure that the key slot is vertical when the key is in the off position.

b. Removal.

(1) Set the power selector switch at OFF.

(2) Gain access to the switch by removing the cargo compartment right foot fairing.

(3) Remove the nut and lockwasher from the switch and withdraw the switch from the hole in the structure.

(4) Unsolder the wires from the switch.

c. Installation.

(1) Set the power selector switch at OFF.

(2) Insert the two wires through the nut and lockwasher, in that order.

(3) Solder wires to the switch terminals and apply sufficient dielectrical compound (C37) to assure moisture-proofing of connections.

(4) Insert the switch into the hole in the seat support structure and secure it with the nut and lockwasher, making sure that the switch is placed so that in the off position the key slot is vertical and the on position is 60 degrees clockwise of vertical.

(5) Install the right foot fairing in the cargo compartment.

SECTION V IGNITION SYSTEM

9-87. Ignition System. Refer to chapter 4 for maintenance of the ignition system excepting the ignition keylock switch which affects both the starter and ignition systems and is covered in section IV of this chapter.

SECTION VI LIGHTING PROVISIONS

9-88. EXTERIOR LIGHTING SYSTEM.

9-89. Description — Exterior Lighting System. The exterior lighting system consists of the landing/hover light, the side and tail position lights, the upper and lower anticollision lights, associated electrical equipment, and wiring. Refer to paragraph 9-214 for maintenance of the landing/hover light switch on the pilot's collective pitch stick. Refer to paragraph 9-67 for maintenance of the position light switch and anticollision light switch-circuitbreaker. Refer to paragraph 9-56 for circuitbreaker maintenance. Refer to paragraph 9-74 for position light dimming resistor maintenance. Refer to appendix F for wiring diagrams.

9-90. Test (Operational Check) — Exterior Lighting System. (See fig. 9-17 and 9-27.) *a.* Set the power selector switch at EXT.

CAUTION

To avoid overheating the lamp, do not leave the landing/hover light on for longer than is needed to verify its operation. (Maximum time on: 60 seconds.)

b. Press the LDG LT switchbutton on the pilot's collective pitch stick. The landing/hover light should come on.

c. Release the LDG LT switchbutton. The landing/hover light should go out.

d. Set the POS LT switch on the electrical console light control panel at BRT. The side and tail position lights should illuminate brightly. Set the switch at DIM and the brightness should be reduced.

e. Set the POS LT switch at OFF. The side and tail position lights should go out.

f. Set the anticollision light switch-circuitbreaker on the electrical console light control panel at ANTI-COLLISION. The upper and lower anticollision lights should display 40 to 65 flashes per minute.

g. Set the anticollision light switch-circuitbreaker at OFF.

h. Set the power selector switch at OFF.

9-91. Troubleshooting — Exterior Lighting System. Refer to table 9-6 and the wiring diagram in appendix F. Prior to using table 9-6, make sure that circuit breakers are reset and that lamps are functional; use the wiring diagram to check for electrical breaks, shorts, and insecure connections.

9-92. ANTICOLLISION LIGHTS.

9-93. Description — Anticollision Lights. The anticollision lights are mounted on the engine air inlet aft fairing and lower fuselage skin. See figure 9-18. Each light consists of a double-contact, bayonet-base lamp, lamp base, socket, lens, and split ring retainer. The lens and base are sealed by a gasket when the split ring retainer is tightened down. The 28-vdc power from the anticollision light flasher is connected to both anticollision light circuits through knife splices. The anticollision light flasher has a flashing rate of 40-65 flashes per minute. Upper and lower anticollision lights can flash simultaneously or alternately, depending on the type of flasher installed.

Table 9-6. Troubleshooting the Exterior Lighting System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Both anticollision lights fail to function when the anticollision light switch-circuitbreaker on the electrical console light control panel is set at ANTI-COLLISION.**

STEP 1. Set the power selector switch at OFF and disconnect the two yellow wires and the red wire of the flasher at the splices. Refer to paragraph 9-98. Connect a jumper between the two wires that were disconnected from the yellow wires. Connect another jumper between the wires already jumpered and the wire that was disconnected from the red wire. Then, momentarily set the power selector switch at EXT.

If both anticollision lights come on (steadily; not flashing) replace the flasher (para 9-98).

STEP 2. Gain access to the anticollision light switch-circuitbreaker. Refer to paragraph 9-67. Jumper across the terminals of the switch-circuitbreaker CB1, and momentarily set the power selector switch at EXT.

If both anticollision lights flash continuously, replace the switch-circuitbreaker (para 9-67).

2. **All position lights fail to function when the POS LT switch on the electrical console light control panel is set at DIM.**

STEP 1. Set the POS LT switch at BRT.

If all position lights function, replace resistor R3 (para 9-74).

3. **All position lights fail to function when the POS LT switch on the electrical console light control panel is set at BRT.**

STEP 1. Set the POS LT switch at DIM.

If all position lights function, replace the POS LT switch. Refer to paragraph 9-67.

4. **Landing/hover light fails to function when the LDG LT switch-button on the pilot's collective pitch stick is pressed.**

STEP 1. Set the power selector switch at OFF and jumper terminal X2 of the landing/hover light relay to ground. (Refer to paragraph 9-121 to gain access to the relay.) Then, momentarily set the power selector switch at EXT.

If the landing/hover light comes on, replace the LDG LT switch on the pilot's collective pitch stick. Refer to paragraph 9-214.

If the landing light remains inoperative, replace the landing/hover light relay (para 9-121).

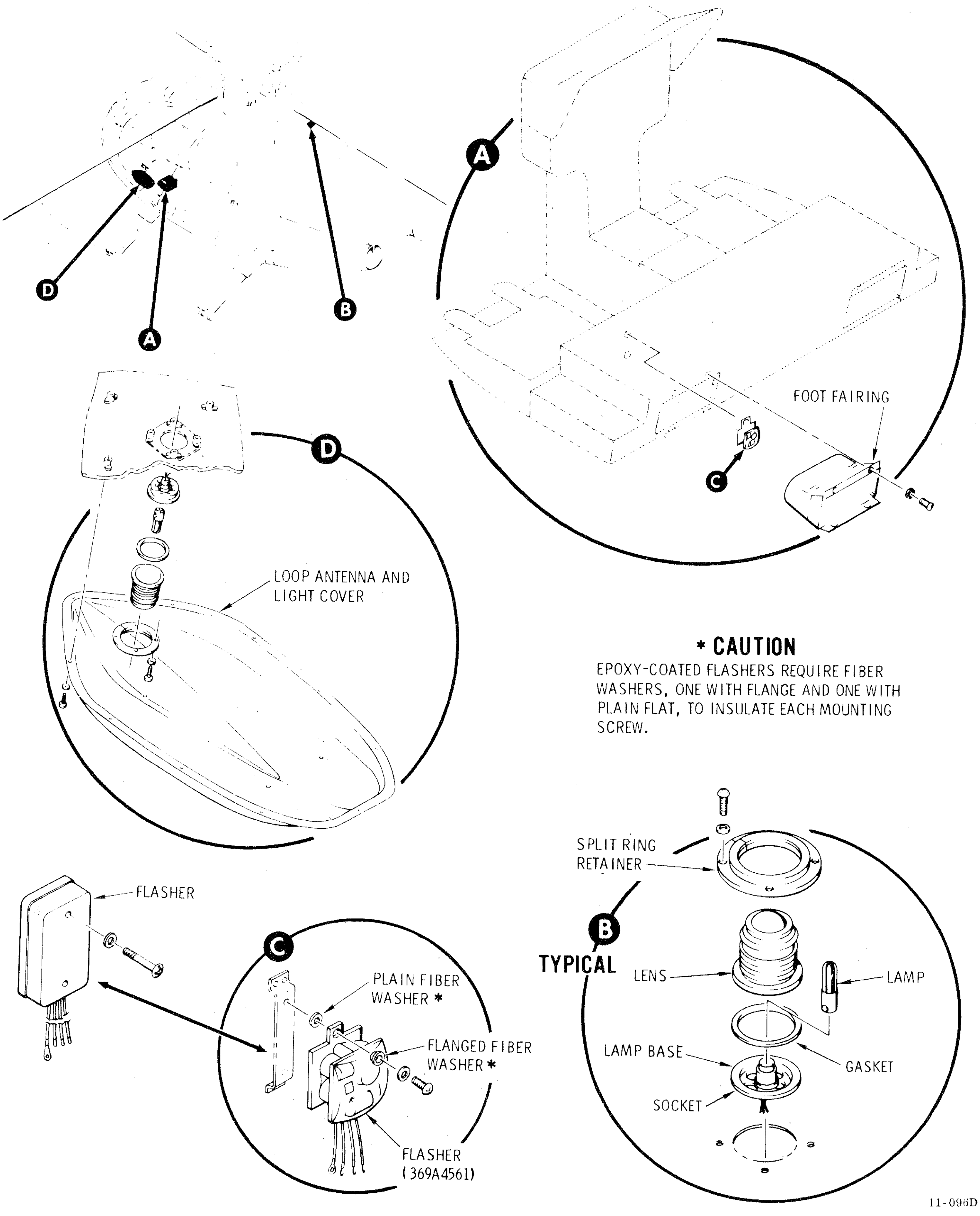


Figure 9-18. Anticollision Light and Flasher Replacement.

CAUTION

Any time maintenance work is being performed near the engine air inlet, as in the base of the upper anticollision light, use care to prevent the entry of foreign objects. On aircraft without an air filter, tape covers of cardboard or other suitable material in place over the engine inlet screen. Plug the oil cooler air inlets. Do not remove the covers until work is completed and debris is thoroughly cleaned out of the area. After removing the covers, verify that the area around the base of the mast, the inlet to the plenum, and the entire plenum chamber are free of foreign material.

9-94. Removal — Anticollision Lights. (See fig. 9-18.) Perform the following steps as applicable.

a. Set the power selector switch at OFF.

b. If a lower light is to be removed, take out the screws that secure the red plastic loop antenna cover, and remove the cover.

c. Disconnect wires at the following locations.

(1) Series 1 aircraft, upper light: P201 and P120.

(2) Series 1 aircraft, lower light: P113 and P121.

(3) Series 2 aircraft, upper light: SP121, SP129, E51.

(4) Series 2 aircraft, lower light: SP116, E32.

(5) Series 3 aircraft, upper light: SP199, SP201.

(6) Series 3 aircraft, lower light: SP113, E33.

d. Remove the four screws and washers from the split ring retainer and remove the retainer.

e. Pull out the assembled lens, gasket, and base.

f. Separate the lens from the base at the gasket and remove the lamp.

9-95. Inspection — Anticollision Lights. (See fig. 9-18.) *a.* Examine the lens for cracks.

b. Examine the gasket for breaks and for brittleness.

c. Examine the lamp socket interior for dirt, corrosion, and any other cause of poor electrical contact.

d. Examine wiring for damage and secure connections.

9-96. Installation — Anticollision Lights. (See fig. 9-18.) *a.* Set the power selector switch at OFF.

b. Prepare all electrical contacting surfaces, including ground connections. Refer to TM 55-1500-204-25/1 for methods.

c. Secure the lamp in the socket.

d. Assemble the lens, gasket, and base; and place the assembly on the structural cutout.

e. Install the split ring retainer with four screws and washers.

f. Connect wires at the following locations.

(1) Series 1 aircraft, upper light: P201 and P120.

(2) Series 1 aircraft, lower light: P113 and P121.

(3) Series 2 aircraft, upper light: SP121, SP129, E51.

(4) Series 2 aircraft, lower light: SP116, E32.

(5) Series 3 aircraft, upper light: SP199, SP201.

(6) Series 3 aircraft, lower light: SP113, E33.

g. After installing a lower light, secure the red plastic loop antenna cover with screws.

9-97. Lamp Replacement — Anticollision Lights. Refer to applicable steps in paragraphs 9-94 and 9-96 above.

9-98. ANTICOLLISION LIGHT FLASHER.

9-99. Description — Anticollision Light Flasher. The static (solid state) anticollision light flasher is mounted within the structure below the pilot's seat. See figure 9-18. The flasher energizes the anticollision lights at 40 to 65 flashes per minute. The flasher operates continuously when the anticollision switch-circuit-breaker is set at ANTI-COLLISION. Two types of flashers are in use. One is epoxy-insulated whereas the other is case-contained. The epoxy-insulated type must have its base insulated from the airframe. Flashing of the upper and lower lights can be simultaneous or alternating.

9-100. Removal — Anticollision Light Flasher. (See fig. 9-18.) *a.* Set the power selector switch at OFF.

WARNING

The transistor mounting base of a flasher that is coated with epoxy insulation is at +28 vdc potential. Use care not to short it to structural ground with tools or other metal objects. Always be sure that the screws used to attach this type of flasher are insulated from the structure. (The case-contained flasher does not require base insulation.)

b. Remove the left foot fairing from the canted bulkhead.

c. Disconnect the flasher ground lead and the wiring splices. Tag-identify the leads, if necessary, for later use.

d. Remove the two screws that secure the flasher to the seat structure. (Do not lose the four insulating washers used with an epoxy-coated flasher.) Lift out the flasher.

9-101. Inspection — Anticollision Light Flasher. (See fig. 9-18.) Examine the flasher and wiring for damage and corrosion. Look for signs of burning or deformation of the flasher due to overheating.

9-102. Installation — Anticollision Light Flasher. (See fig. 9-18.) *a.* Set the power selector switch at OFF.

b. Prepare electrical contacting surfaces, including the ground connection. Refer to TM 55-1500-204-25/1 for methods.

WARNING

The transistor mounting base of a flasher that is coated with epoxy insulation is at +28 vdc potential. Use care not to short it to structural ground with tools or other metal objects. Always be sure that the screws used to attach this type of flasher are insulated from the structure. (The case-contained flasher does not require base insulation.)

c. Connect the wire splices and the ground terminal.

d. Secure the flasher to the seat structure with two screws and washers. Be sure to install fiber washers as shown in figure 9-18 when installing an epoxy-coated flasher.

e. Install the left foot fairing on the canted bulkhead.

9-103. SIDE POSITION LIGHTS.

9-104. Description — Side Position Lights. The side postion lights (fig. 9-19) are identical units that differ only in lens color. The right lens is green, the left red. The side position light supports are riveted to the fuselage skin, aft of the cargo doors. Each light is mounted by three screws. Electrical ground is assured by a strap riveted at one end to the fuselage structure and at the other end to one of the platenuts by which the lamp base is secured.

9-105. Removal — Side Position Lights. (See fig. 9-19.) *a.* Set the power selector switch at OFF.

b. Remove the screw that secures the lens cover to the lamp base.

c. Remove the lens, lens gasket, and the lamp.

d. Remove the three screws that secure the lamp base to the structure.

e. Disconnect the wire.

9-106. Inspection — Side Position Lights. (See fig. 9-19.) *a.* Examine the lens for cleanliness and cracks.

b. Examine the gasket for breaks and for brittleness.

c. Examine the lamp socket for dirt, corrosion, or any other cause of poor electrical contact.

d. Examine the wire for damage.

9-107. Installation — Side Position Lights. (See fig. 9-19.) *a.* Set the power selector switch at OFF.

b. Prepare all electrical contacting surfaces, including ground connections. See figure 9-19 for ground areas. Refer to TM 55-1500-204-25/1 for methods.

c. Connect the wire.

d. Install the base with three screws.

e. Insert the lamp, position the gasket and the lens, and secure by installing the lens cover with one screw.

9-108. Lamp Replacement — Side Position Lights. Refer to applicable steps in paragraphs 9-105 and 9-107.

9-109. TAIL POSITION LIGHT.

9-110. Description — Tail Position Light. The tail position light is located at the lower aft corner of the horizontal stabilizer. See figure 9-19. The single wire to the light is routed through conduit in the tailboom. Operation is simultaneous with the side position lights.

9-111. Removal — Tail Position Light. (See fig. 9-19.) *a.* Set the power selector switch at OFF.

b. Detach the electrical connector.

c. Remove the screws, nuts, and washers that secure the light assembly to the horizontal stabilizer bracket, and remove the light assembly.

d. Separate the lens from the base at the gasket and remove the lamp.

9-112. Inspection — Tail Position Light. (See fig. 9-19.) *a.* Examine the lens for cleanliness and for cracks.

b. Examine the gasket for breaks and for brittleness.

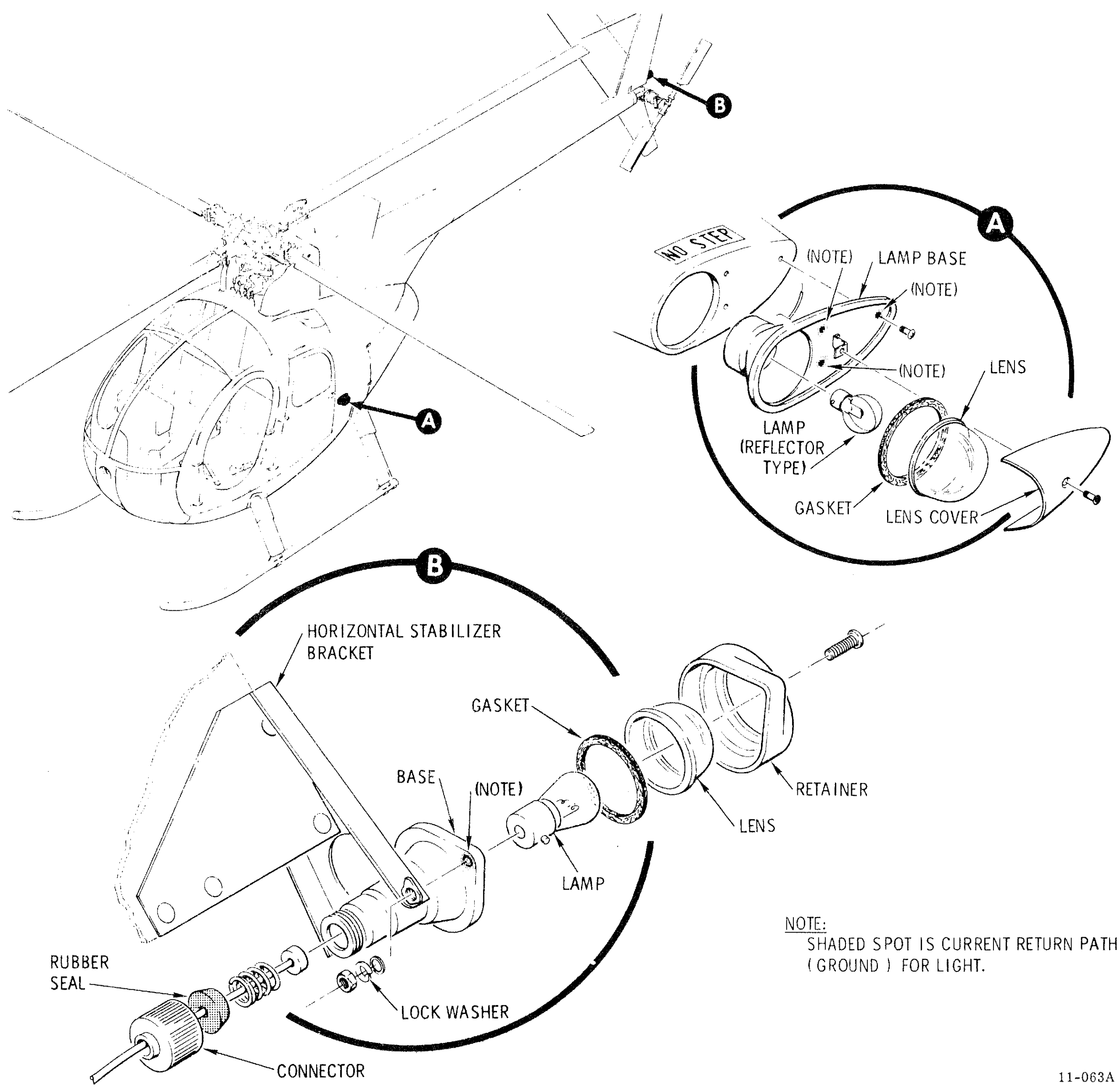


Figure 9-19. Position Light Replacement.

c. Examine the lamp socket for dirt, corrosion, or any other cause of poor electrical contact.

d. Examine the wire for damage.

9-113. Installation — Tail Position Light. (See fig. 9-19.) *a.* Set the power selector switch at OFF.

b. Prepare all electrical contacting surfaces, including the ground connections. See figure 9-19 for ground areas. Refer to TM 55-1500-204-25/1 for methods.

c. Screw the electrical connector on about half way.

d. Install the lamp in the base and then completely screw on the electrical connector finger tight.

e. Position the gasket, lens, and retainer in the order shown in figure 9-19 and install the mounting screws, washers, and nuts.

CAUTION

When removing or installing a tail position light lamp, be sure to first loosen the connector on the lamp base. If the wire is not free to move through the connector seal, the lamp may break.

9-114. Lamp Replacement — Tail Position Light. To replace a lamp, refer to applicable steps in paragraphs 9-111 and 9-113.

9-115. LANDING/HOVER LIGHT.

9-116. Description — Landing/Hover Light. The landing/hover light is an assembly containing a flush-mounted, clear glass, and a sealed-beam lamp. It is recessed in a canopy panel housing that protects the screw terminal base and back of the lamp. Two types of installation are in service; one is nonadjustable, the other adjustable. See figure 9-20. In the nonadjustable installation, the lamp angle is fixed by the canopy panel housing position. In the adjustable installation, the lamp is mounted in a hinged support that fits in the canopy panel housing. The support pivots at the bottom and is bolted to the upper support that contains the holes for angle adjustment. Power from the landing light circuit breaker (fig. 9-15 and 9-16) is controlled by the hover light relay (para 9-121) and the landing/hover light switch on the pilot's collective pitch stick.

9-117. Hinged Lamp Adjustment — Landing/Hover Light. (See fig. 9-20.)The lamp must be removed from its support ring to change the lamp angle.

a. Remove the retainers; lift out and lower the lamp until it is suspended by wiring.

b. Remove the upper support bolt, reposition the lamp to the required angle and reinstall the bolt.

c. Reinstall the lamp in the support ring and secure it with retainers.

9-118. Removal — Landing/Hover Light. (See fig. 9-20.) *a.* Set the power selector switch at OFF.

b. On an adjustable light, remove the lamp retainers; on a nonadjustable light, remove the retaining ring.

c. Lift out the lamp and disconnect the two wires.

d. On an adjustable light, remove the adjustment bolt, nut, and washers.

e. On an adjustable light, remove the pivot pin, washers, and shim washers (if any) noting the quantity and location of any shim washers that may have been used.

9-119. Inspection — Landing/Hover Light. (See fig. 9-20.) Look for corrosion, dirt, and any other cause of poor electrical contact at wire connections, including the ground connection.

9-120. Installation — Landing/Hover Light. (See fig. 9-20.) *a.* Set the power selector switch at OFF.

b. Prepare all electrical contacting surfaces, including the ground connections. Refer to TM 55-1500-204/25/1 for methods.

c. On an adjustable light, secure the lamp support with the pivot pin, using as many shim washers as necessary to prevent sideplay between the lamp support and the hinge. Select the desired angle and install the lamp adjustment bolt, washers, and nut.

d. Connect the wiring.

e. For an adjustable light, place the lamp in the lamp support so that the locating lug is in the detent; then secure the lamp by installing the retainers.

f. For a nonadjustable light, place the lamp so that the locating lug is in the support detent; then secure the lamp by installing the retaining ring.

9-121. LANDING/HOVER LIGHT RELAY.

9-122. Description — Landing/Hover Light Relay. The landing/hover light relay is energized through the landing/hover light switch on the pilot's collective pitch stick and switches 28 vdc to the landing/hover light. On series 1 and 2 aircraft, the relay is within the instrument panel structure on the lower left side. See figure 9-9. On series 3 aircraft, it is on the underside of the circuit breaker panel (fig. 9-16). On some series 2 aircraft, a capacitor C1 is connected between relay terminal A1 and one of the relay mounting screws.

9-123. Removal — Landing/Hover Light Relay. (See fig. 9-9 and 9-16.) *a.* Set the power selector switch at OFF.

b. On series 3 aircraft, lift out the circuit breaker panel to the extent necessary to gain access to the relay. Refer to paragraph 9-56.

c. Disconnect all wires from the relay and tag-identify them.

d. Remove the relay attaching hardware. (On series 1 and 2 aircraft, a phenolic-linen shield is inserted between the relay base and the structure.)

9-124. Inspection — Landing/Hover Light Relay. (See fig. 9-9 and 9-16.) Look for dirt, corrosion, loose relay terminals, damaged wire, and signs of arcing or overheating.

9-125. Installation — Landing/Hover Light Relay. (See fig. 9-9 and 9-16.) *a.* Set the power selector switch at OFF.

b. On a series 2 aircraft having a capacitor, prepare the structure and relay base for bonding at the mounting screw holes per TM 55-1500-204-25/1.

c. Secure the relay to the mounting surface with screws, lockwashers, and nuts. On series 1 and 2 aircraft, insert the phenolic-linen shield between the relay and the structure. On series 2 aircraft having a capacitor, insert one of the capacitor wire terminals between the relay base and the shield.

d. Connect all wires to the relay. (On series 2

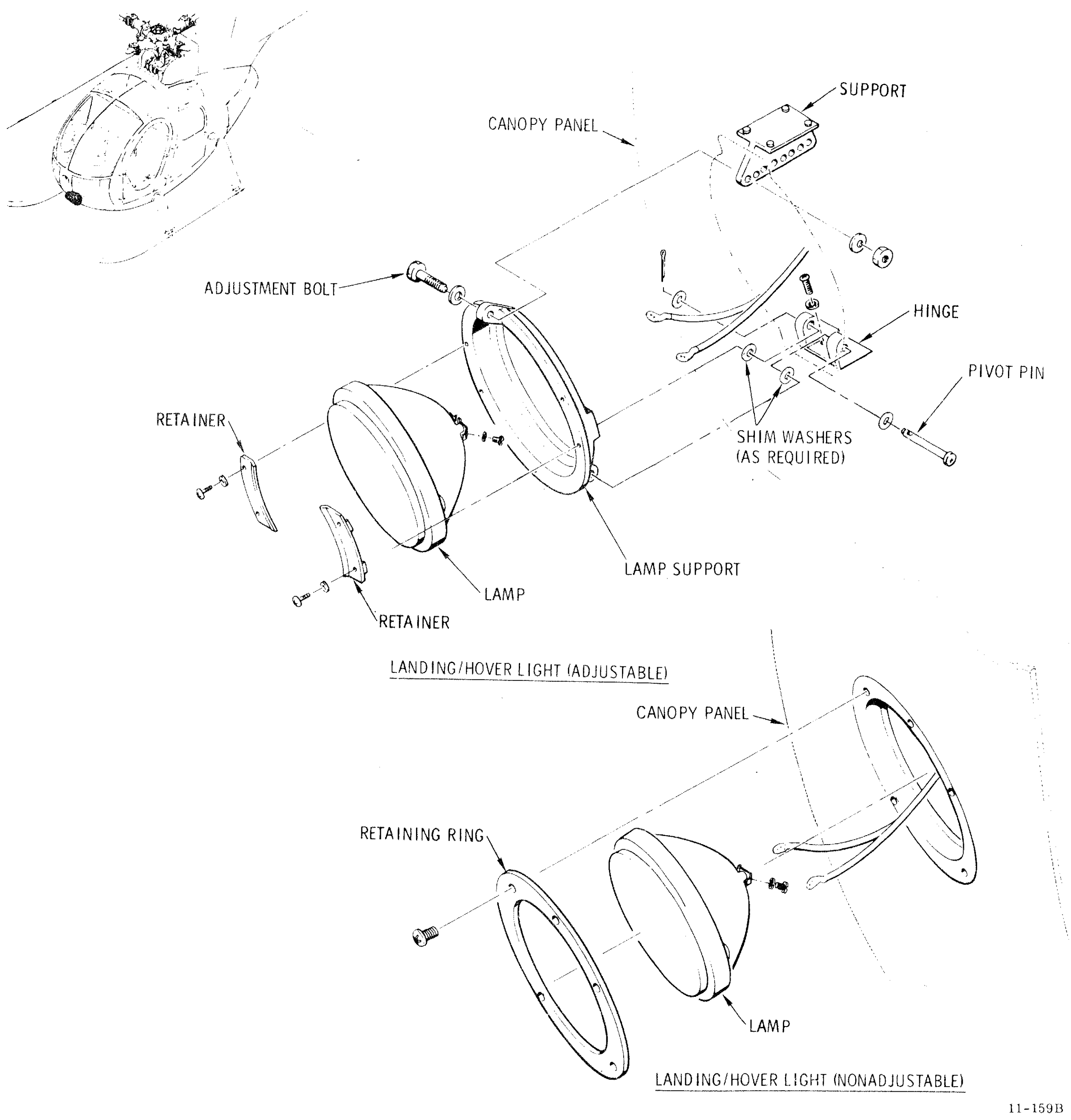


Figure 9-20. Landing/Hover Light Replacement.

aircraft having a capacitor, connect the remaining capacitor wire terminal to terminal A1 of the relay.)

e. On series 3 aircraft, position and secure the circuit breaker panel. Refer to paragraph 9-56.

9-126. INTERIOR LIGHTING SYSTEM.

9-127. Description — Interior Lighting System. The interior lighting system includes instrument lights, a utility light, and panel lights (excepting avionics control panel lights which are covered in TM 11-1520-214-20 and TM 11-1520-214-20-1).

9-128. Test (Operational Check) — Interior Lighting System. (See fig. 9-17.)

NOTE

Unless performed at night, a hood will probably be required to perform an adequate check of edge-lighted panel lamp operation.

a. Set the power selector switch at EXT.

b. Rotate the SW PANEL control on the light control panel (fig. 9-17) slowly clockwise and observe that the armament control panel lights (fig. 9-28), the switch panel lights (fig. 9-13), the light control panel lights, and the circuit breaker panel lights (fig. 9-15 and 9-16) increase in brightness until full brightness is obtained at the full clockwise rotation; then, slowly rotate the control counterclockwise and verify that the lights gradually dim until they are off at the full counterclockwise knob rotation.

c. Rotate the ENGINE control on the light control panel slowly clockwise and observe that lights for the TORQUE PSI, N2 and ROTOR, TOT, N1, and LBS FUEL indicators (refer to chapter 8) increase in brightness until full brightness is obtained at the full clockwise rotation; then, slowly rotate the control counterclockwise and verify that the lights gradually dim until they are off at the full counterclockwise knob rotation.

d. Rotate the FLIGHT control on the light control panel slowly clockwise and observe that lights for the KNOTS, ALT, clock, compass, attitude and ball bank, and heading and bearing indicators (refer to chapter 8) increase in brightness until full brightness is obtained at the full clockwise knob rotation; then, slowly rotate the control counterclockwise and verify that the lights gradually dim until they are off at the full counterclockwise knob rotation.

e. Rotate the control on the back of the utility light (fig. 9-21) slowly clockwise and observe that the light intensity increases until full brightness is obtained at the full clockwise rotation; then, slowly rotate the control counterclockwise and verify that the light gradually dims until it is off at the full counterclockwise knob rotation. Rotate the control slightly clockwise so that the light is noticeably dim and then press the override button. Verify that depression of the override button causes the light to come on at full brilliance.

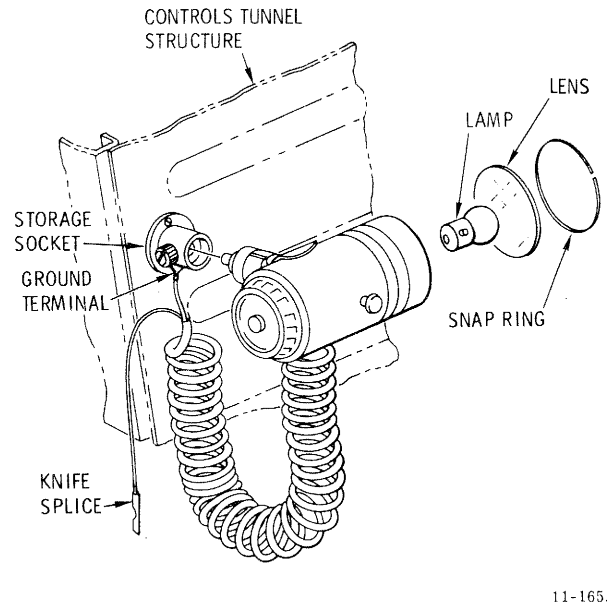


Figure 9-21. Utility Light.

9-129. Troubleshooting — Interior Lighting System. Refer to table 9-7 and the wiring diagram in appendix F. Prior to using table 9-7, make sure that circuit breakers are reset and that lamps (including the imbedded type) are functional; use the wiring diagram to check for electrical breaks, shorts, and insecure connections.

9-130. PANEL LIGHTING.

9-131. Description - Panel Lighting. Individual panel lights or plastic cover panels with imbedded lamp elements provide illumination for the switch panel, the light control panel and the circuit breaker panel assembly. (The radio control panels and armament control panel are illuminated in the same way.) The removable light assemblies consist of miniature sockets with wiring connected in the normal manner. Each plastic panel with imbedded lamp elements is connected to the wiring circuit by a flush-mounted connector that engages its mating half when the panel is installed. Lamp intensity is regulated by controls on the electrical console light control panel. See figure 9-17. For lamp replacement procedures refer to paragraph 9-56 (circuit

Table 9-7. Troubleshooting the Interior Lighting System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **On series 1 and 2 aircraft, the switch panel lights, the light control panel lights, the armament control panel lights, and the circuit breaker panel lights fail to function when the PANEL control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Rotate the ENGINE control on the electrical console light control panel clockwise.

   If the engine instrument lights come on and increase in brilliance as the control is rotated, replace the PANEL control R18. Refer to paragraph 9-67.

2. **On series 3 aircraft, the switch panel lights, the light control panel lights, the armament control panel lights, and the circuit breaker panel lights fail to function when the PANEL control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Replace the dimmer unit DS119 per paragraph 9-138.

   If replacement of the dimmer unit fails to correct the malfunction, replace the PANEL control R18. Refer to paragraph 9-67.

3. **On series 1 and 2 aircraft, the communication and navigation control unit lights fail to function when the RADIO control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Rotate the ENGINE control on the electrical console light control panel clockwise.

   If the engine instrument lights come on and increase in brilliance as the control is rotated, replace the RADIO control R19. Refer to paragraph 9-67.

4. **On series 3 aircraft, the communication and navigation control unit lights fail to function when the RADIO control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Replace the dimmer unit DS119. Refer to paragraph 9-138.

   If replacement of the dimmer unit fails to correct the malfunction, replace the RADIO control R19. Refer to paragraph 9-67.

5. **On series 1 and 2 aircraft, the flight instrument lights fail to function when the FLIGHT control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Rotate the RADIO control on the electrical console light control panel clockwise.

   If the communication and navigation control unit lights come on and increase in brilliance as the control is rotated, replace the FLIGHT control R20. Refer to paragraph 9-67.

6. **On series 3 aircraft, the flight instrument lights fail to function when the FLIGHT control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Replace the dimmer unit DS119. Refer to paragraph 9-138.

   If replacement of the dimmer unit fails to correct the malfunction, replace the FLIGHT control R20. Refer to paragraph 9-67.

7. **On series 1 and 2 aircraft, the engine instrument lights fail to function when the ENGINE control on the electrical console light control panel is rotated clockwise.**

   STEP 1. Rotate the FLIGHT control on the electrical console light control panel clockwise.

   If the flight instrument lights come on and increase in brilliance as the control is rotated, replace the ENGINE control R21. Refer to paragraph 9-138.

Table 9-7. Troubleshooting the Interior Lighting System. (cont)

8. On series 3 aircraft, the engine instrument lights fail to function when the ENGINE control on the electrical console light control panel is rotated clockwise.

STEP 1. Replace the dimmer unit DS119. Refer to paragraph 9-138.

If replacement of the dimmer unit fails to correct the malfunction, replace the ENGINE control R21. Refer to paragraph 9-138.

breaker panel), paragraph 9-62 (switch panel), paragraph 9-67 (light control panel), and paragraph 9-235 (armament control panel).

9-132. INSTRUMENT LIGHTING.

9-133. Description — Instrument Lighting. Individual lights or fixtures are used to illuminate the instrument dial faces. The engine instrument cluster 3-pack and the magnetic compass each contains its own internal lighting provisions. All other instruments are lighted by externally mounted fixtures and post (bolt) lights that are hooded to direct the light to the face of the instrument. Each instrument light fixture, installed over the upper portion of the instrument, consists of a hinged cover, two red-filter lamp holders, a sleeved wire lead, and a back plate. Each of the instrument bolt lights, for the heading, bearing, homing indicator and clock, consists of a removable lamp hood with a red filter, a midget flange base with separate backing plate, and a single terminal connection behind the panel. Power to all instrument lights is controlled by the light control panel switch-rheostats on series 1 and 2 aircraft or potentiometers and dimmer unit on series 3 aircraft. See figure 9-17.

9-134. Removal — Instrument Lighting. (See fig. 9-22.) *a.* Set the power selector switch at OFF.

b. At the back of the instrument panel, unsolder or unplug the wire, as applicable.

c. Remove the attaching hardware and the light fixture from the instrument panel. Note the location of spacer washers used to position the bolt lights. The attaching hardware for most light fixtures also partially secures the instrument to the instrument panel.

9-135. Inspection — Instrument Lighting. (See fig. 9-22.) *a.* Check security of the hood, lamp holder, back plate, and mounting plate, as applicable.

b. Inspect the wire connection to the insulated terminal plate. Check for any damage to the wire insulation.

c. Check the cover surfaces for corrosion and dirt.

d. Inspect the interior of the lamp holders for corrosion, dirt, or any other cause of poor electrical contact. Inspect each red filter for discoloration and cracks.

e. Inspect areas of contact with the instrument panel for unpainted, clean, and uncorroded surface to ensure good ground connection.

9-136. Installation — Instrument Lighting. (See fig. 9-22.) *a.* Set the power selector switch at OFF.

b. Insert the light fixture wire through the hole in the instrument panel and secure the instrument light fixture to the panel with mounting hardware. Make sure that the washers used to space the light mount are correctly located.

c. Plug in or solder, as applicable, the light fixture wire.

9-137. Lamp Replacement — Instrument Lighting. *a.* Set the power selector switch at OFF.

b. For hooded or capped lamps, pull the hood or cap straight off. For other means of lamp access, see figure 9-22.

c. Remove the lamp and replace with another.

d. Press on the hood or cap, or otherwise restore the original condition of the holder as shown in figure 9-22.

9-138. DIMMER UNIT (SERIES 3 AIRCRAFT).

9-139. Description — Dimmer Unit (Series 3 Aircraft). The dimmer unit DS119 is a solid-state device that regulates the voltage applied to the instrument lights, radio lights, and to the edgelighted panel lamps. The dimmer unit is mounted on the center beam in the right electronics compartment. The four potentiometers on the light control panel vary the bias voltage to transistors in the dimmer unit. As potentiometer resistance is increased (clockwise knob rotation), transistor bias voltage is decreased and more voltage is passed to the lamps. Maximum voltage (28 vdc) is available to

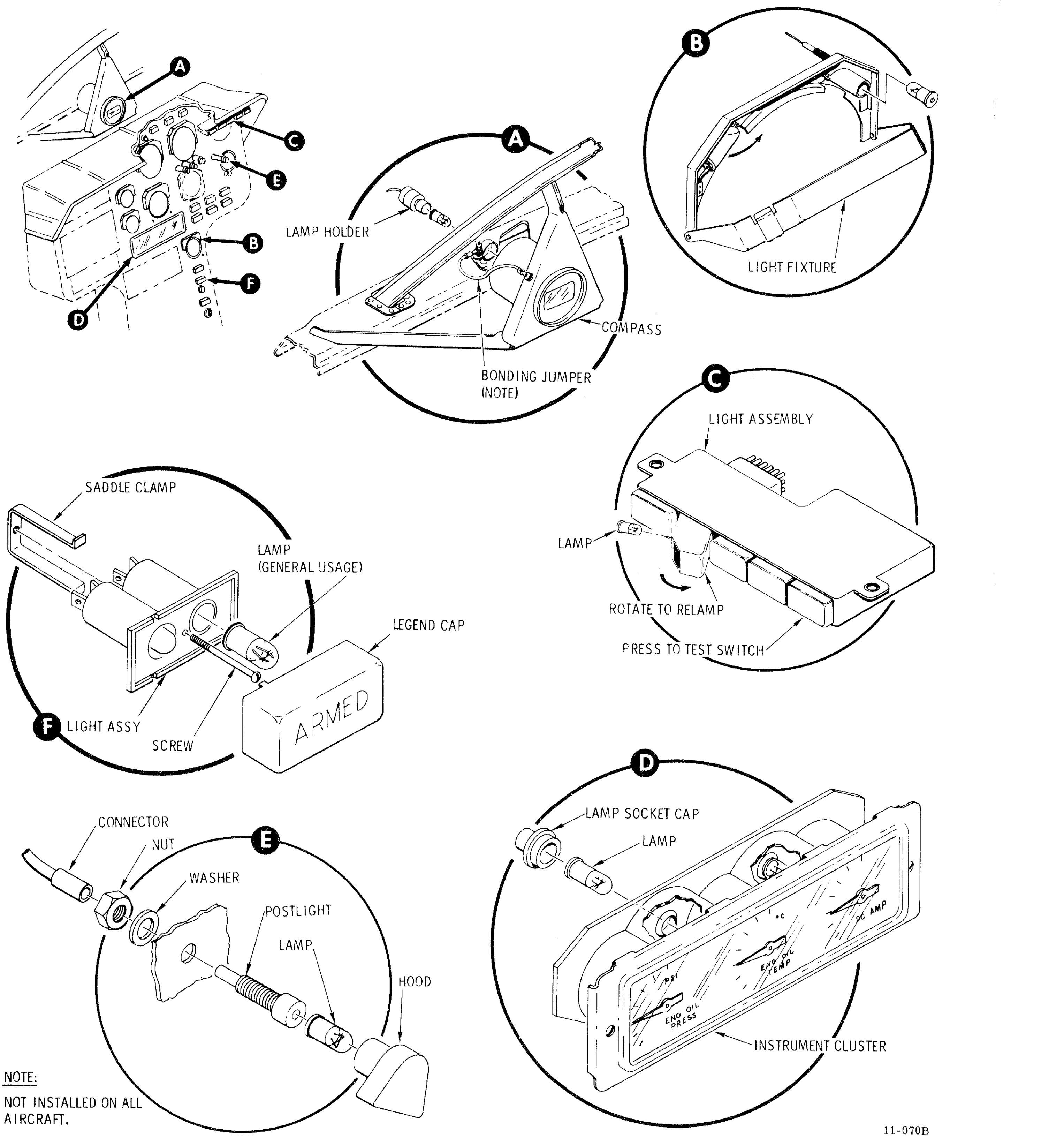


Figure 9-22. Lamp and Fixture Replacement — Instrument Panel.

associated lamps when a potentiometer is set for maximum resistance (knob at BRT position).

9-140. Inspection — Dimmer Unit (Series 3 Aircraft). (See fig. 9-10.) Examine the dimmer unit for secure attachment, corrosion, damage, and security of the electrical plug P104. Look for damaged insulation on wires leading from the plug.

9-141. Removal — Dimmer Unit (Series 3 Aircraft). (See fig. 9-10.) *a.* Set the power selector switch at OFF.

b. Open the pilot's floor right access door.

c. Disconnect the electrical plug P104.

d. Remove the screws and washers that secure the dimmer unit to the center beam.

9-142. Installation — Dimmer Unit (Series 3 Aircraft). (See fig. 9-10.) *a.* Set the power selector switch at OFF.

b. Secure the dimmer unit to the center beam with screws and washers.

c. Connect the electrical plug P104. Refer to paragraph 9-4.

d. Close the pilot's floor right access door.

9-143. UTILITY LIGHT.

9-144. Utility Light — Description. The utility light consists of an adjustable fixture, a switch-rheostat, and a coiled extension cord. See figure 9-21. Rotation of the front section selects white spot, white flood, red flood, or red spot. The color setting cannot be changed accidentally as the lock button on the side of the fixture must be depressed simultaneously while turning the front section. The light fixture is normally stowed in a socket mounted on the right side of the controls tunnel behind the pilot's seat. An alternate socket is on the left edge of the tunnel so the utility light may be used by passengers in the cargo compartment. The switch-rheostat has an override pushbutton that provides full lamp output regardless of knob setting.

9-145. Inspection — Utility Light. (See fig. 9-21.) *a.* Examine the lens for damage and secure attachment.

b. Examine the light housing for damage and corrosion. Verify that the housing swivels without excessive effort and that the front section rotates easily.

c. Examine the electrical splice and the ground terminal for secure electrical connection.

d. Look for damage to insulation on the cord and wires.

9-146. Removal — Utility Light. (See fig. 9-21.) *a.* Set the power selector switch at OFF.

b. Separate the knife splice and disconnect the ground terminal.

c. Pull the light fixture from the storage socket. If difficulty is encountered, loosen the socket setscrew.

9-147. Installation — Utility Light. (See fig. 9-21.) *a.* Set the power selector switch at OFF.

b. Insert the light fixture into the storage socket, tightening the setscrew, if necessary, to provide the proper degree of retention.

c. Secure and insulate the electrical splice.

d. Connect the ground wire terminal to one of the storage socket attaching screws. Refer to TM55-1500-204-25/1 for grounding methods.

9-148. Lamp Replacement — Utility Light. (See fig. 9-21.) *a.* Remove the clear lens by removing the snap ring.

b. Replace the lamp.

c. Install the clear lens and secure it with the snap ring.

SECTION VII CAUTION AND WARNING EQUIPMENT

9-149. CAUTION AND WARNING INDICATING SYSTEM (SERIES 1 AND 2 AIRCRAFT).

9-150. Description — Caution and Warning Indicating System (Series 1 and 2 Aircraft). Legend-type caution (yellow) lights are clustered below the homing-heading-bearing indicator on the instrument panel. Legend-type warning (red) lights are located on the right side of the instrument panel just beneath the hood. Armament legend-type caution lights are located on the instrument panel below the fuel quantity indicator. All legend caps have locator pins to prevent mismatch. A lamp test pushbutton (LIGHT TEST) is located to the left of the warning lights near the instrument panel hood. When the LIGHT TEST pushbutton is pressed, a 28 vdc circuit is completed through diodes to illuminate all caution and warning lights. Refer to paragraph 9-162. Dimming of the caution lights for night flying is controlled by the FLIGHT switch-rheostat on the electrical console light control panel. When this FLIGHT knob is rotated completely to OFF, the dimming relay causes resistors to be bypassed in the light circuit.

Refer to paragraphs 9-162 and 9-167. Engine and transmission warning lights flash on and off. The flashing is produced in the engine power out warning control unit. Refer to paragraph 9-183. The caution lights do not flash.

9-151. Test (Operational Check) — Caution and Warning Indicating System (Series 1 and 2 Aircraft).

NOTE

The following procedure reveals the functional state of a single fault input circuit. It also indicates whether or not circuits common to all caution lights function correctly. A simulation of other fault signals can be made by applying the following jumper-to-ground procedure to any selected negative-input caution light circuit. Refer to appendix F for wiring diagrams.

a. Set the power selector switch at EXT.

b. Momentarily press the LIGHT TEST pushbutton and verify that all caution and warning lights come on at full brilliance.

c. Set the power selector switch at OFF.

d. Using a jumper lead, ground the terminal of the wire that connects to the tail rotor transmission chip detector S400.

e. Rotate the FLIGHT switch-rheostat on the light control panel completely to OFF.

f. Set the power selector switch at EXT.

g. Verify that the OIL CHIPS caution light is on at full brilliance.

h. Rotate the FLIGHT switch-rheostat to any position excepting the OFF position.

i. Verify that the OIL CHIPS caution light is on at partial brilliance.

j. Set the power selector switch at OFF.

k. Remove the jumper and reconnect the wire to the chip detector.

9-152. Troubleshooting — Caution and Warning Indicating System (Series 1 and 2 Aircraft). Refer to table 9-8 and wiring diagrams in appendix F. Prior to using table 9-8, make sure that the WARNING LT circuit breaker CB7 is reset and that all lamps are functional. Use the wiring diagrams to check for electrical breaks, shorts, and insecure connections.

9-153. CAUTION AND WARNING INDICATOR LIGHTS (SERIES 1 AND 2 AIRCRAFT).

9-154. Inspection — Caution and Warning Indicator Lights (Series 1 and 2 Aircraft). Look for dirt and corrosion on each light assembly and examine each legend cap for cracks, chipping, and discoloration.

NOTE

Each lamp housing and legend cap is a matched assembly. If a housing has the wrong cap, the cap will not fit flush.

9-155. Removal (Typical) — Caution and Warning Indicator Lights (Series 1 and 2 Aircraft). *a.* Set the power selector switch at OFF.

b. Unsolder the wires from the light assembly and tag-identify each wire as to whether it attaches to light terminal 1 or 2.

c. Pull off the legend cap (pull it straight out).

d. Remove the attaching hardware and the remainder of the light assembly.

9-156. Installation (Typical) — Caution and Warning Indicator Lights (Series 1 and 2 Aircraft). *a.* Place the lamp housing in the panel.

b. Secure the lamp housing with attaching hardware.

c. Solder wires to the light housing terminals per the identifying tags or by referring to the wiring diagram in appendix F.

9-157. Lamp Replacement (Typical) — Caution and Warning Indicator Lights (Series 1 and 2 Aircraft). To change a lamp, pull the legend cap straight out from the housing.

9-158. OIL CHIPS TEST SWITCH (SERIES 2 AIRCRAFT).

9-159. Description — Oil Chips Test Switch (Series 2 Aircraft). On aircraft with an OIL CHIPS TEST switch in the lower right corner of the instrument panel, an oil chips fault can be isolated to the tail rotor transmission, the main transmission, or to the engine. The 3-position toggle switch is spring-loaded to the center (unmarked) position.

9-160. Test (Operational Check) — Oil Chips Test Switch (Series 2 Aircraft). *a.* Set the power selector switch at EXT.

b. Jumper-ground the wire connected to the tail rotor transmission chip detector. The oil CHIPS caution light should come on. Hold the test switch at TAIL

Table 9-8. Troubleshooting the Caution and Warning Light System (Series 1 and 2 Aircraft).

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **All caution lights fail to be dimmed when the FLIGHT control on the electrical console light control panel is rotated clockwise away from OFF.**

   STEP 1. Replace the dimming relay (para 9-167).

   If replacement of the dimming relay fails to correct the malfunction, replace the FLIGHT control R20. Refer to paragraph 9-67.

2. **One of more (but not all) of the caution lights fails to be dimmed when the FLIGHT control on the electrical console light control panel is rotated clockwise to any position excepting OFF.**

   STEP 1. Replace the dimming relay (para 9-167).

   If replacement of the dimming relay fails to correct the malfunction, replace the resistor board TB5. Refer to paragraph 9-162.

3. **None, or fewer than all, of the caution or warning lights functions when the LIGHT TEST pushbutton on the instrument panel is pressed.**

   STEP 1. Set the power selector switch at OFF. Jumper the LIGHT TEST switch S1 terminals as follows: A1 to B1; A2 to B2. Momentarily set the power selector switch at EXT.

   If all of the caution and warning lights come on, replace the LIGHT TEST switch S1.

   If fewer than all of the caution and warning lights come on, replace the diode board TB2. Refer to paragraph 9-162.

4. **XMSN OIL TEMP, XMSN OIL PRESS, or ENGINE OUT warning light fails to flash on and off when performing an operational check as outlined in paragraph 9-149.**

   STEP 1. Replace the engine power out warning control unit (para 9-189).

   If replacement of the engine power out warning control unit fails to correct the malfunction, check continuity through the audio amplifier and replace the amplifier (para 9-196) or defective connectors (para 9-4).

5. **OIL CHIPS caution light fails to function when performing an operational check as outlined in paragraph 9-149.**

   STEP 1. On terminal board TB1-14, jumper terminal A to ground.

   If the caution light comes on when the detector is shorted, replace the OIL CHIPS TEST switch (para 9-158).

TRANS. If the oil CHIPS light goes out the circuit functions properly. Remove the ground connection.

c. Jumper-ground, in turn, the wires connected to both chip detectors of the main transmission. The oil CHIPS light should come on. Hold the switch at MAIN TRANS to check both detectors. The light will go out if the chip detector circuit is functioning properly. Remove the ground connection.

d. Remove the plugs from both engine chip detectors. Ground, in turn, the plug pins. The oil CHIPS light should come on and stay on with switch held at either transmission switch position. Reconnect and lockwire the chip detector plugs.

e. Set the power selector switch at OFF.

9-161. Troubleshooting — Oil Chips Test Switch (Series 2 Aircraft). Refer to table 9-8.

9-162. DIODE AND RESISTOR BOARDS (SERIES 1 AND 2 AIRCRAFT).

9-163. Description — Diode and Resistor Boards (Series 1 and 2 Aircraft). Diodes and resistors in the caution and warning light system are mounted on terminal boards centrally located in back of the instrument panel. See figure 9-9. Resistors on terminal board TB5 are switched into the caution light circuits by the dimming relay (para 9-167) whenever the FLIGHT switch-rheostat on the light control panel is in any position excepting OFF. Diodes on terminal board TB2 isolate the normal caution and warning light circuits from the lamp test circuit.

9-164. Inspection — Diode and Resistor Boards (Series 1 and 2 Aircraft). (See fig. 9-9.) Inspect the diode and resistor boards for broken wires and connections and for evidence of overheated or damaged diodes or resistors. Check the terminal boards for cracks, breaks, and secure attachment.

9-165. Removal — Diode and Resistor Boards (Series 1 and 2 Aircraft). (See fig. 9-9.) *a.* Set the power selector switch at OFF.

b. Remove the cover (if installed) from the diode board.

c. Remove the screws, washers, and spacers that secure the diode and resistor boards (and the shield) to the instrument panel.

d. Cut nylon straps to the extent necessary to gain slack in the wiring.

e. Lift the diode board and turn it on edge for access to soldered wire connections.

NOTE

On series 2 aircraft, unsoldering can be done outside the aircraft by disconnecting connectors P10, P11, and P158. The diode and resistor boards, together with the harness, can then be removed as a unit. However, on aircraft with a BYPASS AIR caution light, one wire must be unsoldered from diode CR25 before the diode board becomes removable.

f. Use a pencil-type soldering iron to unsolder wires from the diode and/or resistor board.

9-166. Installation — Diode and Resistor Boards (Series 1 and 2 Aircraft). (See fig. 9-9.)

NOTE

On series 2 aircraft, soldering can be done outside the aircraft, if the diode and resistor boards and the harness were removed as a unit (refer to note under para 9-165).

a. Position the resistor board. Use a pencil-type soldering iron to solder wires to the terminals. Use the wiring diagram in appendix F as a reference.

b. Position the diode board. Use a pencil-type soldering iron to solder wires to the terminals. Use the wiring diagram in appendix F as a reference.

c. Secure the shield, the diode board, and the resistor board to the instrument panel with spacers, washers, and screws. See figure 9-9 for sequence.

d. Plug in the electrical connectors. Arrange and tie the harness or wires. Refer to paragraph 9-3 for materials and methods.

e. Install the cover (if previously installed) on the diode board with screw, washers, and spacer as shown in figure 9-9.

f. Test all circuits affected by the maintenance action. Refer to paragraph 9-149.

9-167. DIMMING RELAY (SERIES 1 AND 2 AIRCRAFT).

9-168. Description — Dimming Relay (Series 1 and 2 Aircraft). The dimming relay K104 is mounted on the instrument panel behind the caution lights. See figure 9-9. The relay has two wiring connectors, P10 and P11, each containing 12 contacts. Each contact is held in place by retaining clips. When the FLIGHT switch-rheostat on the light control panel is set at OFF, the relay acts to convey fault signals directly to the caution lights. When the FLIGHT switch-rheostat is in any position excepting OFF, the relay is de-energized and fault signals are conveyed through resistors on the resistor board (para 9-162) to the caution lights which then illuminate at partial brilliance. Warning lights are not dimmed.

9-169. Inspection — Dimming Relay (Series 1 and 2 Aircraft). (See fig. 9-9.) *a.* Examine the relay for secure attachment, and that both connector plug clips are in place.

b. Examine adjacent wiring for damaged insulation.

c. Examine the relay and the electrical connectors for corrosion and evidence of overheating.

9-170. Removal — Dimming Relay (Series 1 and 2 Aircraft). (See fig. 9-9.) *a.* Set the power selector switch at OFF.

b. Unfasten the retaining clips that secure the two connectors and unplug the connectors.

c. Remove the relay attaching hardware.

9-171. Installation — Dimming Relay (Series 1 and 2 Aircraft). (See fig. 9-9.) *a.* Position the relay and secure it to the instrument panel. (Some relays have platenuts installed while others do not. If a relay has platenuts, use a flat washer and a lockwasher under each screw head. Use only one flat washer under each nut if separate nuts are to be installed.)

b. Plug in and secure connectors P10 and P11 with clips.

9-172. CAUTION AND WARNING INDICATING SYSTEM (SERIES 3 AIRCRAFT).

9-173. Description — Caution and Warning Indicating System (Series 3 Aircraft). Legend-type caution (yellow) lights are clustered below the clock on the instrument panel. Legend-type engine warning (flashing red) lights are located on the right side of the instrument panel just below the hood. Armament legend-type caution lights are located on the instrument panel below the fuel quantity indicator. All legend caps have locator pins to prevent mismatch. A pushbutton (PRESS TO TEST) is located to the right of the engine warning light cluster, under the instrument panel hood. When the PRESS TO TEST pushbutton is held in, all caution and warning lights come on at full brilliance for the purpose of checking the lamps and circuits. All caution lights are automatically dimmed when the FLIGHT control on the electrical console light control panel is in any position excepting OFF. Any caution-type signal energizes both the specific caution light and the flashing MASTER CAUTION light-switch (adjacent to the PRESS TO TEST pushbutton) until the MASTER CAUTION light-switch is pressed, whereupon signal conditioning and logic circuits within the annunciator panel turn off and reset master caution flashing circuits and the MASTER CAUTION light-switch stops flashing. The caution light remains lit, however, until there is no longer a fault input signal.

CAUTION

Continuous flashing of the MASTER CAUTION light-switch shall be limited to 10 minutes. Excessive uninterrupted flashing can result in damage to components of the flashing circuits in the annunciator caution light assembly.

Flashing of the engine out, transmission oil pressure and transmission warning lights is produced in the engine power out warning control unit.

9-174. Test (Operational Check) — Caution and Warning Indicating System (Series 3 Aircraft). An operational check of the caution and warning indicating system shall be made after replacement or repair of the annunciator caution light assembly, the master caution and warning light assembly, the engine power out warning control unit, or the armament control panel.

NOTE

The following procedure reveals the functional state of two fault input circuits. It also indicates whether or not circuits common to all caution and warning lights function correctly. A simulation of other fault signals can be made by applying the following jumper-to-ground procedure to any selected negative-input caution or warning light circuit. Refer to appendix F for wiring diagrams.

a. Set the FLIGHT control on the electrical console light control panel at OFF.

b. Using a jumper, ground the terminal of the wire that connects to the tail rotor transmission chip detector S400.

c. Set the power selector switch at EXT.

d. Verify that the TAIL CHIPS caution light on the annunciator caution light assembly is on and that the MASTER CAUTION light-switch is flashing on and off.

e. Press and release the MASTER CAUTION light-switch and verify that the MASTER CAUTION light is out and that the TAIL CHIPS light remains on.

f. Using a second jumper lead, ground the terminal stud of the engine chip detector to produce a fault input for the ENGINE CHIPS caution light.

g. Verify that the ENGINE CHIPS and MASTER CAUTION lights are flashing and the TAIL CHIPS light remains on.

h. Press and release the MASTER CAUTION light switch and verify that the MASTER CAUTION light is out and that the ENGINE CHIPS and TAIL CHIPS caution lights are on.

i. Disconnect the wire from the transmissiion oil pressure switch. Verify that the TRANS OIL PRESS light goes out.

j. Reconnect the wire to the transmission oil pressure switch. Verify that the TRANS OIL PRESS light is on.

k. Rotate the FLIGHT switch-rheostat to any position except OFF and verify that the TAIL CHIPS caution light is reduced to partial brilliance.

l. Set the power selector switch to OFF.

m. Remove jumpers from tail rotor transmission and engine chip detectors.

n. Set the power selector switch to EXT.

o. Verify that all caution and warning lights are out.

p. Set the power selector switch to OFF.

9-175. Troubleshooting — Caution and Warning Indicating System (Series 3 Aircraft). Troubleshoot the caution and warning indicating system per table 9-9. Perform operational checks (para 9-174) as necessary to facilitate troubleshooting. Use the information in table 9-10 to determine the printed circuit board associated with each caution or warning light, the type of fault input signal, and corresponding connector pins. Before using table 9-9, make sure the FLIGHT control on the electrical console light control panel is set at OFF excepting when the troubleshooting procedure specifically requires it to be otherwise.

9-176. ANNUNCIATOR CAUTION LIGHT ASSEMBLY (SERIES 3 AIRCRAFT).

9-177. Description — Annunciator Caution Light Assembly (Series 3 Aircraft). The annunciator caution light assembly DS118 is located in the right lower corner of the instrument panel. See figure 9-23. None of the caution lights flashes, but the assembly contains electrical components for the flashing and reset circuits affecting (and affected by) the MASTER CAUTION light-switch on the master caution and warning light assembly. The annunciator caution light assembly also contains electrical components relative to the circuit and lamp tests operated by the PRESS TO TEST pushbutton on the master caution and warning light assembly.

9-178. Inspection — Annunciator Caution Light Assembly (Series 3 Aircraft). (See fig. 9-23)

CAUTION

Do not close a legend cap unless the cap recess is free of foreign matter. Use compressed air to blow out any accumulation of dust, dirt, grit or foreign matter before attempting to close an indicator legend cap.

a. Inspect all legend caps for cracks, chipping, and discoloration.

b. Check the panel for corrosion, dirt, and damage.

c. Examine the electrical connector for security of attachment and look for damaged insulation on adjacent wires.

9-179. Removal — Annunciator Caution Light Assembly (Series 3 Aircraft). (See fig. 9-23.) *a.* Set the power selector switch at OFF.

b. Unplug the electrical connector P15 at the back of the assembly.

c. Remove the screws and washers that secure the light panel to the instrument panel.

d. If installation of another assembly is intended, remove the clip-type nuts from the light panel flanges and save them for use in installation.

9-180. Repair (AVIM) — Annunciator Caution Light Assembly (Series 3 Aircraft). (See fig. 9-24.) Use the following procedures as applicable for disassembly, repair, or replacement of unserviceable parts.

a. To gain access for part removal, repair or replacement:

(1) Remove the six flathead screws that secure the cover to the housing and remove the cover.

(2) Remove the eight flathead screws that attach the light subassembly housing to the terminal board assembly housing.

(3) Hold the annunciator in a horizontal position and carefully separate the light assembly from the terminal board assembly.

Table 9-9. Troubleshooting the Caution and Warning Light System (Series 3 Aircraft).

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Caution lights fail to dim when the PRESS TO TEST pushbutton is pressed and held, and the FLIGHT control is moved clockwise away from OFF.**

STEP 1. Adjust or replace the FLIGHT control microswitch S6. Refer to paragraph 9-67.

If adjustment or replacement of the microswitch fails to correct the malfunction, replace printed circuit board 81-0786-1 (the top board nearest the annunciator assembly top cover). Refer to paragraph 9-176.

If the replacement of circuit board 81-0786-1 fails to correct the malfunction, replace the annunciator caution light assembly (para 9-176).

2. **One or more (but not all) warning or caution indicators fails to function when the PRESS TO TEST pushbutton is pressed.**

STEP 1. Observe whether or not the MASTER CAUTION switch-indicator is flashing.

If the MASTER CAUTION switch-indicator is flashing, replace the printed circuit board(s) corresponding to the non-functioning indicator(s). Refer to table 9-10 and paragraph 9-176.

3. **When the PRESS TO TEST pushbutton is pressed, all indicators function excepting the MASTER CAUTION switch-indicator.**

STEP 1. Replace the master caution and warning light assembly (para 9-183).

If replacement of the master caution and warning light assembly fails to correct the malfunction, replace annunciator caution light assembly printed circuit board 81-0790-1 (para 9-176).

4. **When the PRESS TO TEST pushbutton is pressed, one or more caution indicators lights dimly and/or the MASTER CAUTION switch-indicator flashes dimly.**

STEP 1. Determine whether or not all caution indicators function dimly.

If all indicators are dimmed, adjust or replace the FLIGHT control microswitch S6. Refer to paragraph 9-67.

If one of more indicators functions at full brilliance, replace the annunciator caution light assembly printed circuit board(s) corresponding to the dim indicator(s). Refer to table 9-10 and paragraph 9-176.

5. **All caution indicators, including the MASTER CAUTION switch-indicator, fail to function when the PRESS TO TEST switch is pressed.**

STEP 1. Replace the master caution and warning light assembly. Refer to paragraph 9-183.

If replacement of the master caution and warning light assembly fails to correct the malfunction, replace annunciator caution light assembly printed circuit board 81-0790-1 (para 9-176).

If replacement of printed circuit board 81-0790-1 fails to correct the malfunction, replace printed circuit board 81-0786-1 (the top board nearest the annunciator assembly top cover). Refer to paragraph 9-176.

6. **When performing an operational check as outlined in paragraph 9-174, the specific fault indicator functions, but the MASTER CAUTION switch-indicator fails to flash.**

STEP 1. Press the PRESS TO TEST pushbutton.

If the MASTER CAUTION switch-indicator again fails to flash, replace annunciator caution light assembly printed circuit board 81-0790-1. Refer to paragraph 9-176.

7. **The MASTER CAUTION switch-indicator flashes or lights steadily in the absence of a fault signal.**

Table 9-9. Troubleshooting the Caution and Warning Light System (Series 3 Aircraft). (Cont)

STEP 1. Replace annunciator caution light assembly printed circuit board 81-0790-1. Refer to paragraph 9-176.

If replacement of printed circuit board 81-0790-1 fails to correct the malfunction, replace printed circuit board 81-0786-1 (the board nearest the annunciator assembly top cover). Refer to paragraph 9-176.

8. When performing an operational check as outlined in paragraph 9-174, the simulated fault signal fails to cause the specific caution indicator to function, but the MASTER CAUTION switch-indicator flashes.

STEP 1. Temporarily remove the jumper used to simulate the fault signal.

If the MASTER CAUTION switch-indicator ceases to flash, replace the printed circuit board corresponding to the inoperative indicator. Refer to table 9-10 and paragraph 9-176.

9. The MASTER CAUTION switch-indicator flashes and one of the caution indicators functions without a fault signal being applied.

STEP 1. Replace the annunciator caution light assembly printed circuit board corresponding to the specific fault indicator. Refer to table 9-10 and paragraph 9-176.

If replacement of the printed circuit board determined by using table 9-10 fails to correct the malfunction, replace printed circuit board 81-0790-1. Refer to paragraph 9-176.

10. When performing an operational check as outlined in paragraph 9-174, the simulated fault signal causes the specific fault indicator to function and the MASTER CAUTION switch-indicator functions. However, the MASTER CAUTION switch-indicator continues to flash on and off after having been pressed.

STEP 1. Replace annunciator caution light assembly printed circuit board 81-0790-1 (para 9-176).

If replacement of printed circuit board 81-0790-1 fails to correct the malfunction, replace the printed circuit board corresponding to the specific fault indicator. Refer to table 9-10 and paragraph 9-176.

11. When performing an engine or transmission warning light operational check, as outlined in paragraph 9-174, the warning light does not go out.

Step 1. Replace the engine power out warning control unit. Refer to paragraph 9-189.

12. When performing an engine or transmission warning light operational check as outlined in paragraph 9-174, the simulated fault signal causes the specific fault indicator to flash dimly instead of brightly.

Step 1. Replace the engine power out warning control unit. Refer to paragraph 9-189.

13. The ENGINE OUT warning indicator flashes in the absence of a fault signal.

Step 1. Replace the engine power out warning control unit. Refer to paragraph 9-189.

Table 9-10. Troubleshooting Data—Caution and Warning Light System (Series 3 Aircraft).

Indicator light or control	Fault input signal	Connector number and pin letter: Master caution and warning light assem P14	EPO warning control unit P21	Annunciator caution light assem P15	Printed circuit board
ENG CHIPS	Grd	-	-	Z	81-0786-1***
FUEL LOW	Grd	-	-	D	81-0786-1***
TRANS CHIPS	Grd	-	-	J	81-0786-3
FUEL FILTER	Grd	-	-	B	81-0786-3
TAIL CHIPS	Grd	-	-	F	81-0786-1
ENGINE OIL LOW BYPASS MODE	Grd	-	-	H	81-0786-1
BYPASS AIR	Grd	-	-	h	81-0786-1***
IFF	+28 vdc	-	-	C	81-0788-1
DC GEN 81-0786-1	Grd	-	-	A	
TRANS OIL TEMP	Grd	N	-	-	-
TRANS OIL PRESS	Grd	M	-	-	-
ENGINE OUT	Low or no engine tach gen output	L	A or L	-	-
MASTER CAUTION		C	-	X	81-0790-1
PRESS TO TEST*	Grd**	R	-	R	81-0786-1***
PRESS TO TEST*	+28 vdc**	B	-	S	81-0790-1
GUN NOT CLEARED	Grd	-	-	c	81-0788-1
AMMO LO	Grd	-	-	L	81-0788-1
ARMED 81-0790-1	+28 vdc	-	-	M	
Dim-Brt*	+28 vdc** (brt)	-	-	U	81-0786-1***
Reset*	+28 vdc**	-	-	V	81-0788-1

*Function (not an indicator).

**Function input signal (not a fault signal).

***Top PC board nearest the assembly top cover.

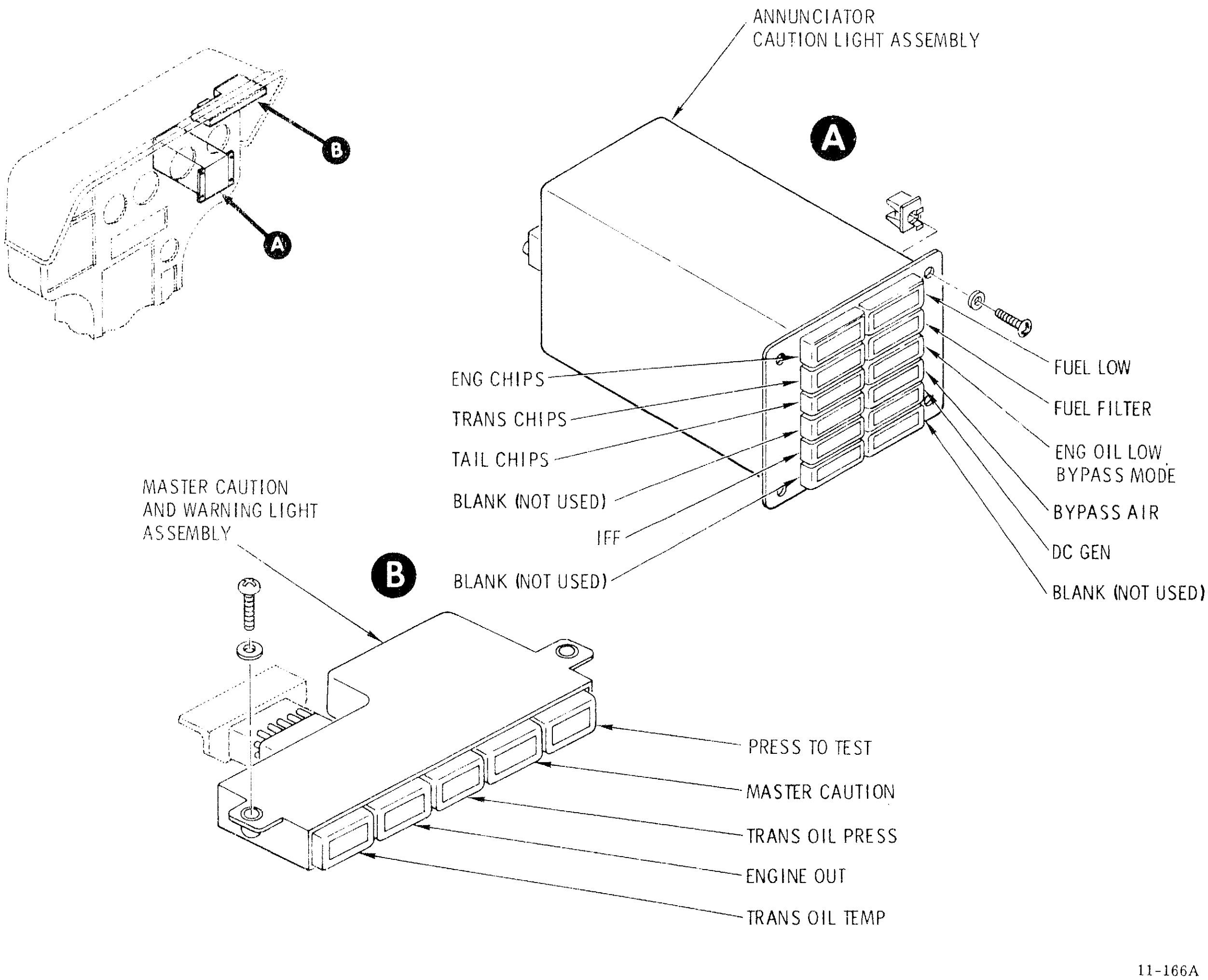


Figure 9-23. Annunciator Panel and Master Caution and Warning Light Panel.

NOTE

The five printed circuit (PC) boards should remain with either the light assembly or with the terminal board assembly when the two housing parts are separated. However, it is possible that the PC boards may pull loose at both ends. Be careful not to drop the PC boards.

b. To replace a defective PC board:

(1) Carefully pull one PC board at a time from its plugged-in position. Code each board by felt pen or similar method to aid correct installation.

(2) Remove as many PC boards as necessary to accomplish the replacement.

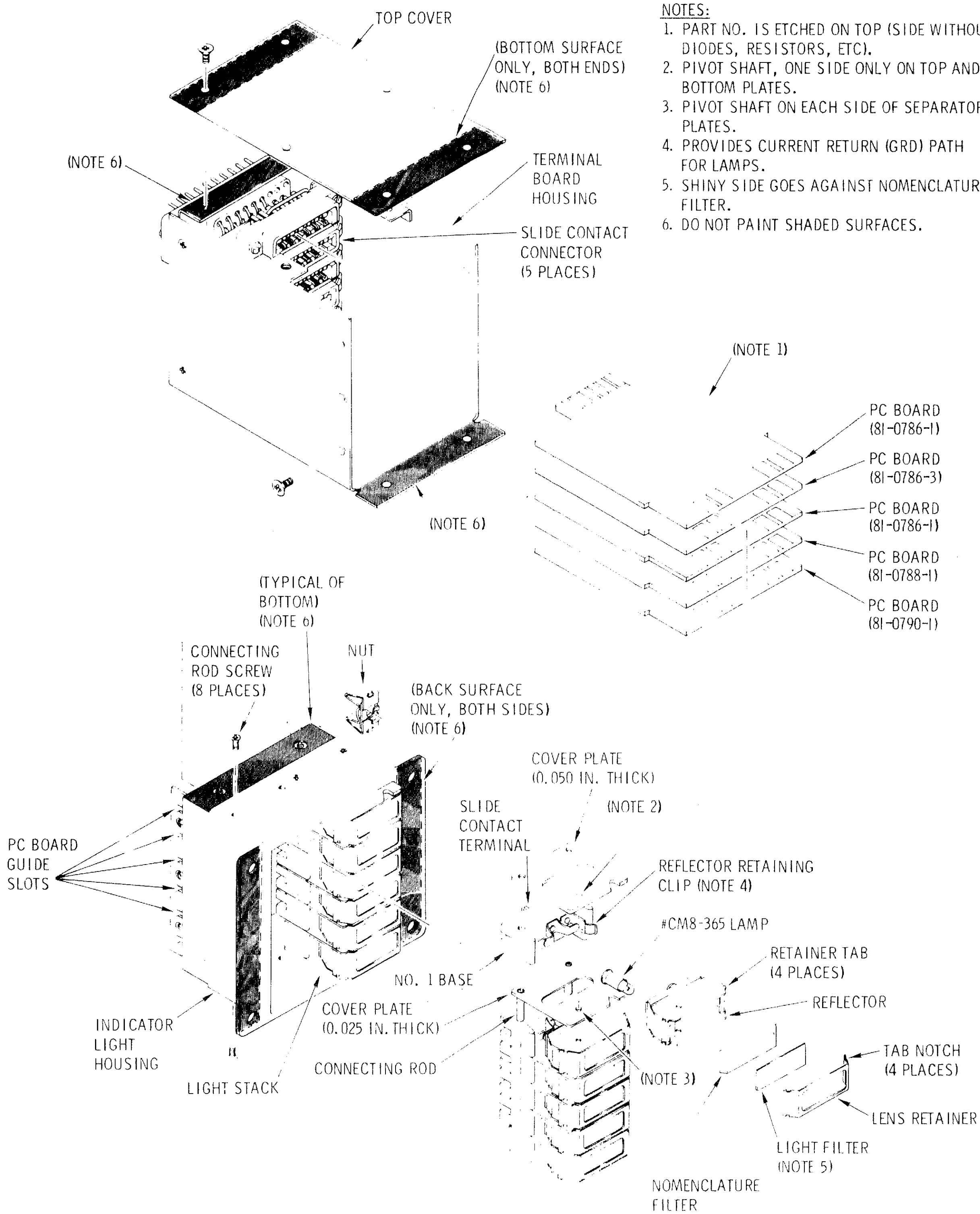
(3) Install serviceable boards in the light assembly. See figure 9-24 for correct location.

c. To remove either stack of six lights from the light assembly housing:

(1) Remove two flathead screws from both the top and bottom of the light assembly to release the two connecting rods on which the six lights are stacked.

(2) Slowly push the stack from the housing. Hold the stack intact to avoid dropping any parts.

d. To replace indicator light parts in a stack:



NOTES:
1. PART NO. IS ETCHED ON TOP (SIDE WITHOUT DIODES, RESISTORS, ETC).
2. PIVOT SHAFT, ONE SIDE ONLY ON TOP AND BOTTOM PLATES.
3. PIVOT SHAFT ON EACH SIDE OF SEPARATOR PLATES.
4. PROVIDES CURRENT RETURN (GRD) PATH FOR LAMPS.
5. SHINY SIDE GOES AGAINST NOMENCLATURE FILTER.
6. DO NOT PAINT SHADED SURFACES.

Figure 9-24. Annunciator Caution Light Assembly (Series 3 Aircraft).

(1) With the stack upright, remove the top cover-plate from the two connecting rods.

(2) Lift out the top indicator light reflector assembly.

(3) Slide the indicator light base from the two connecting rods. Check that the contact springs are intact on the base and are not broken. Replace the base if the contacts are broken.

(4) Remove additional cover plates, reflector assemblies, and bases in the same way. Replace cover plates if the pivot shafts are broken.

(5) Assemble the stack as shown in figure 9-24.

e. To replace the indicator light assembly parts:

(1) Indicator lamps are held in place by spring friction. Remove by direct pull on lamp base. Replace burned out lamps.

(2) Using fingernail, release the lens retainer from the reflector assembly. Remove the light grey Mylar filter and the legend filter (lens) from the reflector. Replace cracked filters. Be sure not to mix legend filters.

(3) Assemble the light assembly parts as shown in figure 9-24. When assembling the lens retainer and filters with the reflector, be sure that all four lens retainer notches snap into place on the reflector tabs.

f. Assemble the light assembly containing plugged-in PC boards with the terminal board assembly housing as follows:

(1) Be sure all PC boards are aligned in the mating slide contact connectors and carefully push the housing halves together.

(2) Install the eight flathead screws in the housing but do not tighten them.

(3) Install the cover with the six flathead screws and tighten the screws. Then tighten the remaining eight housing screws.

(4) Apply paint finish touchup as required (TB 746-93-2).

9-181. Installation — Annunciator Caution Light Assembly (Series 3 Aircraft). (See fig. 9-23.) *a.* Install clip nuts on the light panel flanges.

b. Position the annunciator caution light assembly in the instrument panel and secure it with screws and washers.

c. Plug in the electrical connector P15.

d. If installation has been preceded by exchange or repair of the annunciator caution light assembly, perform an operational check as outlined in paragraph 9-172.

9-182. Replacement — Lamp (Typical) — Annunciator Caution Light Assembly (Series 3 Aircraft). (See fig. 9-18.)

CAUTION

Positioning (closing) of the indicator legend cap without the lamp in place can cause the lamp retainer spring to break and result in internal (short-circuiting) damage to the annunciator caution light assembly. Burned out lamps should be left installed until replacement lamps are available.

a. Rotate the indicator legend cap by pressing on the left end of the cap.

b. Remove and replace the lamp.

c. Rotate the cap back into place.

9-183. MASTER CAUTION AND WARNING LIGHT ASSEMBLY (SERIES 3 AIRCRAFT)

9-184. Description — Master Caution and Warning Light Assembly (Series 3 Aircraft). The master caution and warning light assembly DS101 is located just under the hood on the right side of the instrument panel. The assembly contains three warning (red) legend-type lights, the yellow MASTER CAUTION light-switch (the switch function resets flashing circuits in the annunciator caution light assembly), and a non-lighted pushbutton switch (PRESS TO TEST).

9-185. Inspection — Master Caution and Warning Light Assembly (Series 3 Aircraft). (See fig. 9-22 and 9-23.) *a.* Examine the legend caps for cracks, chipping, and discoloration.

b. Check for dirt and corrosion in the lamp compartments.

c. Examine electrical connectors P14 and J14 for damage and secure installation. Inspect wire insulation for damage.

9-186. Removal — Master Caution and Warning Light Assembly (Series 3 Aircraft). (See fig. 9-23.) *a.* Set the power selector switch at OFF.

b. Detach the light assembly from the hood by removing two screws and washers.

c. Carefully extract the light assembly from the bracket-mounted electrical connector P14.

9-187. Installation — Master Caution and Warning Light Assembly (Series 3 Aircraft). (See fig. 9-23.) *a.* Set the power selector switch at OFF.

b. Securely insert the light assembly into the bracket-mounted electrical connector P14.

c. Attach the light assembly to the hood with two screws and washers. Be sure that the outboard screw also passes through the ground strap on the underside of the hood. Refer to TM 55-1500-204-25/1 for electrical bonding methods.

d. If installation has been preceded by exchange of the master caution and warning light assembly, perform an operational check as outlined in paragraph 9-172.

9-188. Lamp Replacement — (Typical) — Master Caution and Warning Light Assembly (Series 3 Aircraft). (See fig. 9-22.) *a.* Rotate the indicator legend cap by pressing on the right end of the cap.

b. Remove and replace the lamp.

c. Rotate the cap back into place.

9-189. ENGINE POWER OUT WARNING CONTROL UNIT.

9-190. Description — Engine Power Out Warning Control Unit. The engine power out (EPO) warning control unit is located within the base of the instrument panel structure or in the left electronics compartment. For series 1 and 2 aircraft, see figure 9-9. For series 3 aircraft, see figure 9-10. One of two different types of EPO units may be installed. EPO unit 369A4640 is used prior to incorporation of the automatic restart system (chapter 4) and EPO unit 369A4614 is required when the automatic restart system is installed. Both units monitor engine N1 rpm and main transmission oil temperature and pressure. EPO unit 369A4614 also monitors engine N2 rpm. Both units contain two flashing circuits, one for the engine out warning light and the other for the main transmission temperature and oil pressure warning lights. The units also contain an audio warning signal tone generator. Both units provide a flashing engine power-out signal and a warning tone in the headsets when N1 falls below 55 percent. EPO unit 369A4614 also provides the same engine-out warning signals when engine N2 rpm falls below 95 percent with the throttle open. EPO unit 369A4614 also provides a signal to energize the engine ignition system when engine power out (N1 or N2 low rpm) is sensed. The reignition signal continues for five seconds and then cuts off. The audible warning, for both units, is shut off when the generator switch is OFF, although the warning lights continue to flash. Reignition is also shut off when the generator switch is OFF. Reignition operation is indicated by the RE-IGN PRESS TO RESET light-switch mounted on the underside of the instrument panel hood. This light remains on until the indicator face is pushed for reset. Engine power out and transmission lights flash alternately at approximately 150 times per minute when the engine is shut down.

9-191. Inspection — Engine Power Out Warning Control Unit. For series 1 and 2 aircraft, see figure 9-9. For series 3 aircraft, see figure 9-10.

a. Check the unit for damage and for secure attachment.

b. Examine adjacent wires for damaged insulation.

c. Examine electrical connectors J21 and P21 for proper engagement and security.

9-192. Removal — Engine Power Out Warning Control Unit. For series 1 and 2 aircraft, see figure 9-9. For series 3 aircraft, see figure 9-10.

a. Set the power selector switch at OFF.

b. Disconnect the electrical connector P21 from the unit.

c. Remove the hardware that attaches the unit to the aircraft structure.

9-193. Installation — Engine Power Out Warning Control Unit. For series 1 and 2 aircraft, see figure 9-9. For series 3 aircraft, see figure 9-10.

a. Set the power selector switch at OFF.

b. Secure the unit to the instrument panel support or to the underfloor mounting bracket, as applicable.

c. Join and secure electrical connectors J21 and P21.

d. Perform an operational check of the automatic restart system. Refer to chapter 4.

9-194. BEEP-TONE AMPLIFICATION — ENGINE POWER OUT WARNING

9-195. Description — Beep-Tone Amplification — Engine Power Out Warning. On series 1 and 2 aircraft that do not have an automatic engine restart system, a small audio signal amplifier AR1 is inserted between the engine power out warning control unit and the pilot's headphones. On series 3 aircraft and all other aircraft having an automatic engine restart system, amplification takes place in the intercommunication control units. Refer to TM 11-1520-214-20-1 and TM 11-1520-214-34-1.

9-196. ENGINE POWER OUT WARNING AUDIO AMPLIFIER (AR1)

9-197. Description — Engine Power Out Warning Audio Amplifier (AR1). The audio amplifier is held by a clip to one of the screws that attach the engine power out warning control unit to the aircraft structure. See figure 9-9. It is a dual-channel amplifier that increases beep-tone strength from the engine power out warning control unit for delivery to the pilot's headsets.

9-198. Removal — Engine Power Out Warning Audio Amplifier (AR1). (See fig. 9-9.) *a.* Set the power selector switch at OFF.

b. Electrically disconnect the amplifier at connectors P2 and P13.

c. Remove the nut, washer, and screw that secure the amplifier (and the engine power out warning control unit) to the aircraft structure.

9-199. Installation — Engine Power Out Warning Audio Amplifier (AR1). (See fig. 9-9.) *a.* Set the power selector switch at OFF.

b. Electrically connect the amplifier at connectors P2 and P13.

c. Secure the amplifier to the mounting flange of the engine power out warning control unit with a screw, washer, and nut.

d. Perform an operational check of the automatic restart system. Refer to chapter 4.

SECTION VIII MISCELLANEOUS EQUIPMENT

9-200. CYCLIC STICK GRIP.

9-201. Description — Cyclic Stick Grip. The cyclic stick grip (fig. 9-25) contains the cylic trim switch, the armament elevation/depression switch, the armament trigger switch, and the RADIO-ICS switch. On series 1 aircraft, the wiring to the RADIO-ICS switch and the cyclic trim switch disconnects at plug P130 (pilot's stick) and plug P110 (co-pilot's stick). On series 1 aircraft, wiring for the armament switches disconnects at plug P131 (pilot's stick) and plug P132 (co-pilot's stick). On series 2 and 3 aircraft, all wiring to each stick is disconnected at a single plug.

9-202. Removal — Cyclic Stick Grip. (See fig. 9-25.) *a.* Disconnect electrical contacts from plug(s). Refer to paragraph 9-4 for method.

b. Remove the clamp that secures the wiring to the cyclic stick socket.

c. Remove the screw and the washer that secure the grip to the cyclic stick and separate the grip from the stick.

d. Tie twine (C112) to each wire bundle to aid reassembly. Push the wiring into the exit holes in the stick while pulling the grip and the wiring from the stick. Remove the grip and the wiring, but leave the two lengths of twine in the stick.

9-203. Disassembly — Cyclic Stick Grip. (See fig. 9-25.) *a.* Remove the six pan-head screws that secure the grip cap, and lift the cap from the grip.

b. Press the pin from the button on the armament elevation/depression switch and remove the button.

c. Remove the trigger pin from the armament trigger switch, and loosen the switch.

d. Remove the two pan-head screws that secure the RADIO-ICS switch to the grip, and slide the switch out from the recess in the grip.

e. Remove the truss-head screw that secures the (spare) pushbutton switch that is vertically inserted into the side of the grip. Push the switch out of the grip.

f. Remove insulation to the extent necessary to unsolder wires from the switches.

g. Carefully remove the pin that secures the trigger guard to the armament trigger switch. Note how the return spring is installed and take care not to lose it.

9-204. Cleaning — Cyclic Stick Grip. Clean all parts exposed in disassembly with dry compressed air at about 15 psi. New parts should be cleaned prior to assembly. Dirt or grease may be removed with solvent (C107) applied sparingly with a clean cloth.

9-205. Inspection — Cyclic Stick Grip. Inspect all switches and the grip for damage and deterioration. Examine wiring for damaged insulation and secure terminals. Check switches and wiring for continuity.

9-206. Repair — Cyclic Stick Grip. Do not attempt to repair any parts of the grip assembly. All defective parts shall be replaced. All switch assemblies shall be replaced as assemblies and soldered to the proper wires unless complete assemblies with wires attached are available. All wires replaced shall be of the type, size, and color specified in figure 9-26.

9-207. Assembly — Cyclic Stick Grip. (See fig. 9-25.) *a.* Install the trigger guard and return spring on the trigger switch by inserting the pin. Refer to paragraph 9-203.

b. Secure the button to the armament elevation/depression switch by inserting the pin.

c. Slip an insulating sleeve onto each wire and solder wiring to switches. Refer to wiring diagram in appendix F. Pull sleeves up over soldered connections.

d. Insert the (spare) pushbutton switch in the side recess of the grip and secure it with the truss-head screw. This switch is a normally-open type; do not install a normally closed switch at this location.

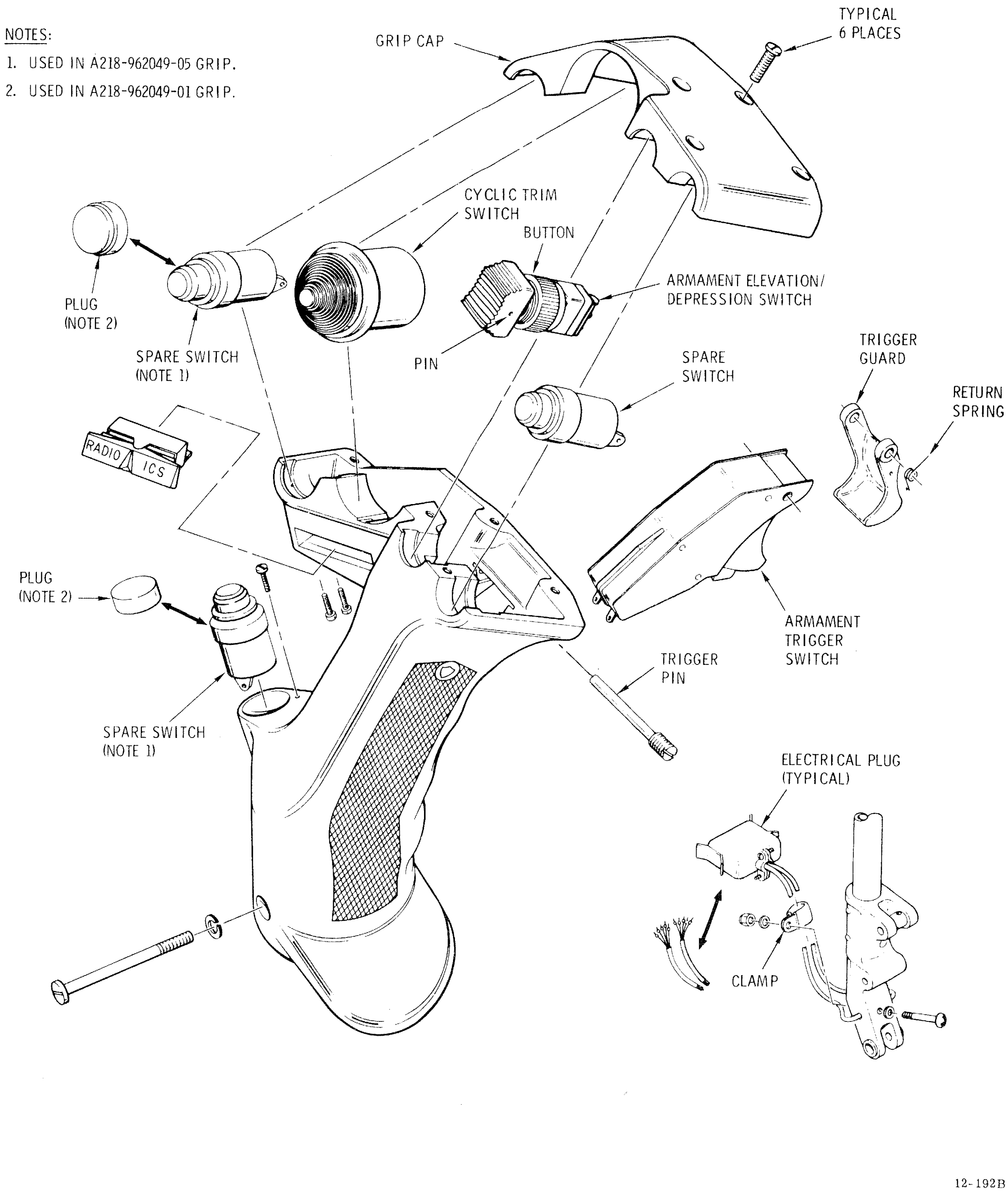


Figure 9-25. Cyclic Stick Grip (Typical).

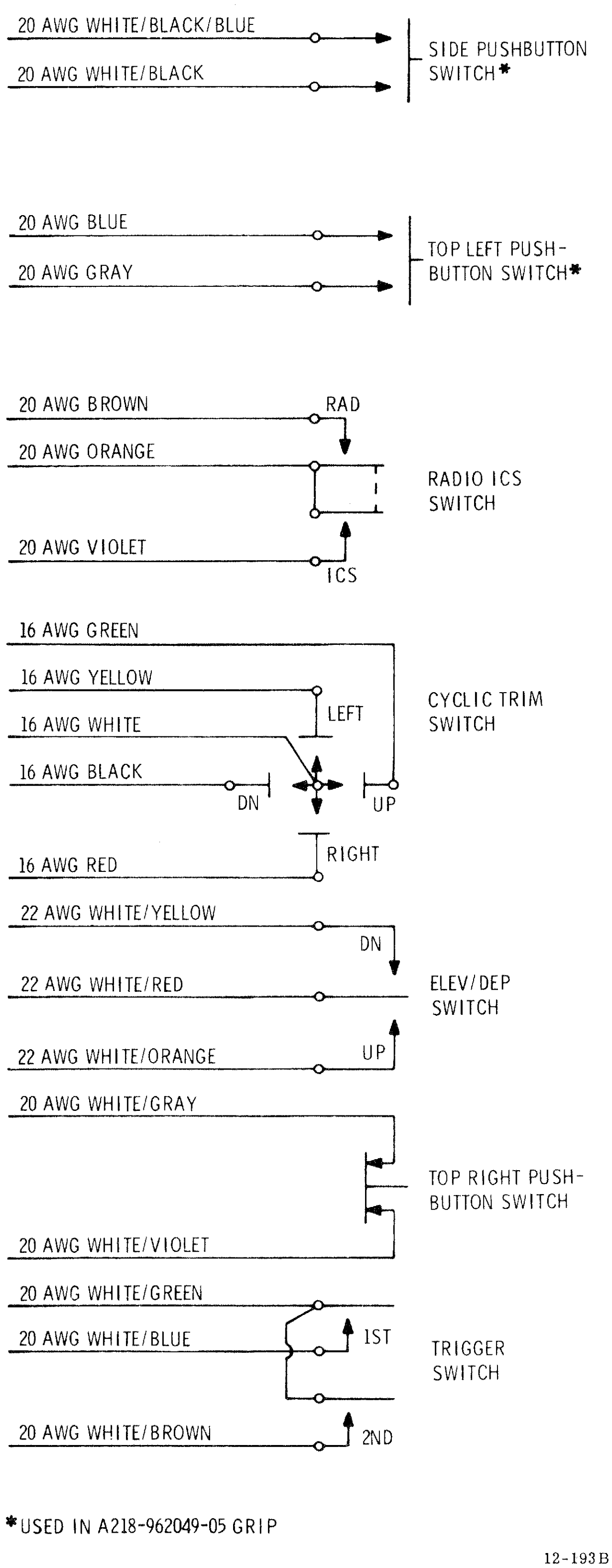


Figure 9-26. Cyclic Stick Grip Switch-Wiring Diagram.

e. Insert the armament trigger switch in the grip and secure it with the trigger pin.

f. Insert the RADIO-ICS switch into the grip and secure it with two pan-head screws.

NOTE

In the following step, make sure that the two spare pushbutton switches are properly placed. The left is a normally-open type. The right is a normally-closed type. See figure 9-26 for electrical characteristics of all spare switches.

g. Install the two spare pushbutton switches, the cyclic trim switch, and the armament elevation/depression switch in the grip and secure them by temporarily installing the grip cap with six pan-head screws. Note that the cyclic trim switch has a small projection or key that must fit into a keyway in the grip, and that the armament elevation/depression switch is rotated so that operation is vertical.

h. So as not to loosen the grip cap sufficiently to disturb switch security, remove the six pan-head screws one-by-one, lightly coating the threads of each screw with varnish (C113) before re-installing. Use a clean cloth to wipe the varnish on the threads.

9-208. Installation — Cyclic Stick Grip. (See fig. 9-25.) *a.* Install an 8-inch length of AWG Size No. 3 electrical insulation sleeving (C49) on each wire bundle and push each sleeve up into the grip.

b. Route the sleeved wire bundles (with twine previously attached; refer to paragraph 9-202) through the stick and out through the wiring exit holes. Push wiring at the grip-end while pulling on twine or wire bundles.

c. Install an additional eight inches of sleeving (refer to step *a* above) on each bundle to additionally protect the bundles as they pass through the stick exit holes.

d. Secure the grip to the stick with the screw and washer shown in figure 9-25.

e. Push as much wire as possible back into the stick and secure the two wire bundles to the stick socket with the clamp, screw, washer, and nut shown in figure 9-25.

f. Connect the wire contacts to the plug(s). If new wires are being installed, install contacts on wires. Refer to paragraph 9-4 for methods. Refer to appendix F for the wiring diagram.

9-209. Test — Cyclic Stick Grip. Refer to the wiring diagram in appendix F and see figure 9-26. Use a multimeter (T4) or a 28-vdc power supply and a test lamp to check switch operation and electrical continuity to and from the connector(s) near the stick socket. Note that the upper right pushbutton switch is normally closed. Check for no continuity (infinite resistance for the ohmmeter function of a multimeter) between each wire and aircraft ground.

9-210. RPM GOVERNOR (N2) TRIM ACTUATOR.

9-211. General — Rpm Governor (N2) Trim Actuator. Refer to chapter 4 for maintenance instructions pertaining to the rpm governor (N2) trim actuator.

9-212. CYCLIC TRIM ACTUATORS.

9-213. General — Cyclic Trim Actuators. Refer to chapter 11 for maintenance instructions pertaining to the cyclic trim actuators.

9-214. COLLECTIVE PITCH STICK SWITCHES.

9-215. Description — Collective Pitch Stick Switches. The collective pitch sticks contain the START, N2 GOV, and LDG LT switches. The pilot's stick (fig. 9-27) contains all three; the co-pilot's stick contains only the N2 GOV switch. Wires from the switches are routed inside the stick and emerge through holes at the aft end of the stick. On series 3 aircraft, all wires from the pilot's stick terminate in a plug P109 whereas in series 1 and 2 aircraft they mix with other wiring in harnesses. Wires emerging from the co-pilot's stick terminate in a plug P111 on all aircraft.

9-216. Inspection — Collective Pitch Stick Switches. Inspect the switches for damage or deterioration, security of wire terminals, and for damaged wire insulation. Check knobs and switches for secure attachment.

9-217. Removal — Collective Pitch Stick Switches. (See fig. 9-27.) *a.* Set the power selector switch at OFF.

b. Cut the twine or nylon strap that secures wiring to the stick.

c. Disconnect ground connections, if there are any.

d. Loosen the two setscrews that secure the grip (switch housing) to the stick and push wires toward the grip while sliding the grip from the end of the stick.

e. Remove the four screws and washers that hold the switchplate to the grip.

f. Remove the setscrews that secure the LDG LT and N2 GOV knobs, and remove the knobs.

g. Remove the panel nuts, lockwashers, switchplate, and check nuts from the LDG LT and N2 GOV switches.

h. Remove the self-tapping screw that secures the flange of the START switchbutton assembly and extract the START switch.

i. Unsolder wires from the terminals of the switches. Tag-identify the wires.

9-218. Installation — Collective Pitch Stick Switches. (See fig. 9-27.) When installing a replacement pilot's collective stick assembly as detailed in chapter 11, it may be that the stick assembly wires are numbered differently from those of the stick assembly that was removed. If the replacement stick wire numbers do not correspond with either the wiring diagram in appendix F or those in the removed stick, connect the wires as indicated in table 9-11.

a. Set the power selector switch at OFF.

b. Solder wires to switch terminals. Use previously attached tags (para 9-217), the wiring diagram in appendix F, or data in table 9-11 as applicable.

c. Install the LDG LT and N2 GOV switches, knobs, and switchplate in the sequence shown in figure 9-27.

d. Install the START switch and secure it with the self-tapping screw.

e. Carefully pull wires aft while inserting the stick in the grip (switch housing). Make sure that the idle stop release ring and spring are installed as shown in figure 9-27. Secure the grip to the stick with two setscrews.

f. Use twine (C112) or tape to secure wiring to the stick at a point about one inch from the exit holes. (Nylon strap may be used if available.)

g. Connect grounding terminals (if any). Refer to TM 55-1500-204-25/1 for surface preparation.

9-219. Test — Collective Pitch Stick Switches. Operationally check the switches and the idle stop by checking the landing light function and by performing the engine starting and runup procedures as detailed in TM 55-1520-214-10.

9-220. ELECTRIC FUEL PUMP RELAY.

9-221. Description — Electric Fuel Pump Relay. On series 1 and 2 aircraft delivered with or modified for the electric fuel pump, the fuel pump relay is mounted on a support bracket attached to the right underside of the electrical console switch panel. See Figure 9-13. On series 3 aircraft, the relay is mounted on a bracket attached to the underside of the circuit breaker panel. See figure 9-16. The relay has enclosed contacts, and plugs into a socket that is secured to the mounting bracket.

9-222. Inspection — Electric Fuel Pump Relay. For series 1 and 2 aircraft, see figure 9-13. For series 3 aircraft, see figure 9-16.

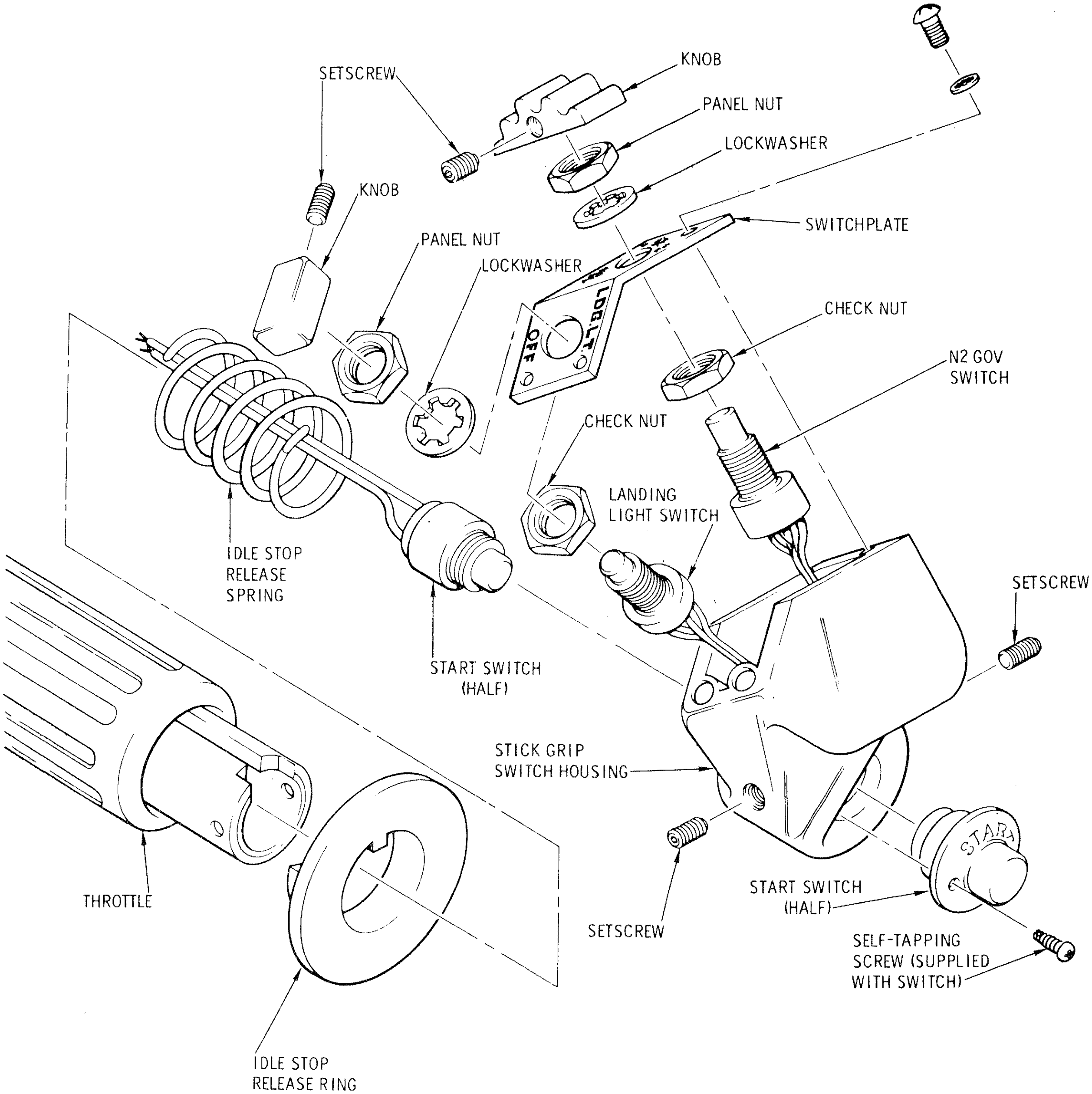


Figure 9-27. Collective Pitch Stick Switches.

Table 9-11. Pilot's Collective Pitch Stick Electrical Wiring—Replacement Data.

Replacement Collective Pitch Stick Assy Wire No.	From	Connect to Series 1 Aircraft	Series 2 Aircraft	Series 3 Aircraft
P111D20	START switch	TB1-13C	P152 R	P109 B
P112A20	START switch	TB1-13E	P152 L	P109 A
L104C20	LDG LT switch	TB1-12C	P152 N	P109 C
L113A20N	LDG LT switch	Ground terminal	Ground terminal	Ground terminal
C108G20	N2 GOV switch	TB1-11C	P152 B	P109 F
C109F20	N2 GOV switch	TB1-10G	P152 S	P109 H
C110F20	N2 GOV switch	TB1-11F	P152 K	P109 E

a. Inspect the relay case for damage and for evidence of overheating.

b. Examine the electrical connectors for corrosion, security of attachment, and evidence of arcing.

9-223. Removal — Electric Fuel Pump Relay. For series 1 and 2 aircraft, see figure 9-13. For series 3 aircraft, see figure 9-16.

a. Set the power selector switch at OFF.

b. For series 1 and 2 aircraft, gain access to the relay by using procedures detailed in paragraph 9-62. For series 3 aircraft, gain access to the relay by using procedures detailed in paragraph 9-56.

c. Remove the two screws and washers that attach the relay to the socket and unplug the relay. It is not necessary to remove the socket.

9-224. Installation — Electric Fuel Pump Relay. For series 1 and 2 aircraft, see figure 9-13. For series 3 aircraft, see figure 9-16.

a. Set the power selector switch at OFF.

b. Plug in the relay and secure it with two screws and washers.

c. For series 1 and 2 aircraft, re-install the electrical console switch panel using procedures detailed in paragraph 9-62. For series 3 aircraft, re-install the circuit breaker panel by using procedures detailed in paragraph 9-56.

9-225. FUEL FILTER PRESSURE SWITCH (TEST)

9-226. Description — Fuel Filter Pressure Switch (Test). The fuel filter pressure switch S303 mounted on the engine-driven fuel pump senses the differential fuel pressure across the fuel filter. If the differential pressure exceeds 0.9 ±0.1 of an inch of mercury, the switch initiates a signal to the FUEL FILTER caution light on the instrument panel. Excepting the test below, all other maintenance data pertaining to the switch is in chapter 4. Test the switch as follows.

a. Connect the positive lead to a 28-vdc power source to one lead of a test lamp.

b. Connect the negative lead of the 28-vdc power source to the fuel filter pressure switch case.

c. Connect the remaining lead of the test lamp to the knife splice on the pressure switch.

d. Connect a source of controlled and monitored pressure to the larger of the two switch ports (the port farthest from the wire). Leave the other (smaller) switch port open.

e. Slowly increase the pressure until the test lamp comes on. The reading on the pressure indicator shall be 0.9 ±0.1 of an inch of mercury at the moment the test lamp lights.

f. Allow the pressure to slowly decrease. The test lamp shall go out when pressure has decreased to 0.4 of an inch of mercury.

9-227. OIL COOLER BYPASS RELAY.

9-228. Description, Inspection, Removal, and Installation — Oil Cooler Bypass Relay. The oil cooler bypass relay energizes the bypass solenoid in the engine compartment on aircraft delivered with or modified for the armored oil system. On series 1 and 2 aircraft, the bypass relay is mounted on the radio relay panel inside the lower central area of the instrument panel.

See figure 9-9. On series 3 aircraft, the bypass relay is mounted alongside the overvoltage relay on the underside of the electrical console switch panel. See figure 9-13. Inspection, removal, and installation procedures are identical to those for the overvoltage relay detailed in paragraph 9-38.

9-229. ATTITUDE GYRO.

9-230. General — Attitude Gyro. Maintenance instructions pertaining to the attitude gyro are detailed in chapter 8.

9-231. DIRECTIONAL GYRO.

9-232. General — Directional Gyro. Maintenance instructions pertaining to the directional gyro are detailed in chapter 8.

9-233. ARMAMENT CONTROLS.

9-234. General — Armament Controls. The armament trigger switches and elevation/depression switches are on the pilot's and co-pilot's cyclic stick grips. Refer to paragraph 9-200. All other armament controls are on the armament control panel.

9-235. ARMAMENT CONTROL PANEL

9-236. General — Armament Control Panel. The armament control panel is located on the instrument panel just below the fuel quantity indicator. See figure 9-28. The face of the panel is plastic and has imbedded lighting provided by a light-circuit board sandwiched between the plastic panel and a panel backing plate. See figure 9-29. The imbedded lights are controlled from the electrical console light control panel (para 9-67) by the SW PANEL switch-rheostat on series 1 and 2 aircraft, or by the PANEL potentiometer on series 3 aircraft. Three legend-type indicator lights, the armed-SAFE switch, and the SYSTEM MODE selector switch are mounted on the backing plate so as to project through the light-circuit board and the plastic panel. The armed-SAFE switch locks in the SAFE position so that the toggle must be pulled out to unlock it before it can be moved upward to arm the system. Maintenance instructions pertaining to the ARMED warning light, the GUN NOT CLEARED caution light, and the AMMO LOW caution light are detailed in paragraphs 9-149 and 9-153 for series 1 and 2 aircraft; in paragraph 9-172 for series 3 aircraft. To troubleshoot the armament circuits, refer to TM 9-1005-298-12 and TM 9-1005-298-34.

9-237. Removal — Armament Control Panel. (See fig. 9-29.) *a.* Set the power selector switch at OFF.

b. Disconnect electrical connector P12 located behind the instrument panel.

c. Remove the six screws that attach the armament

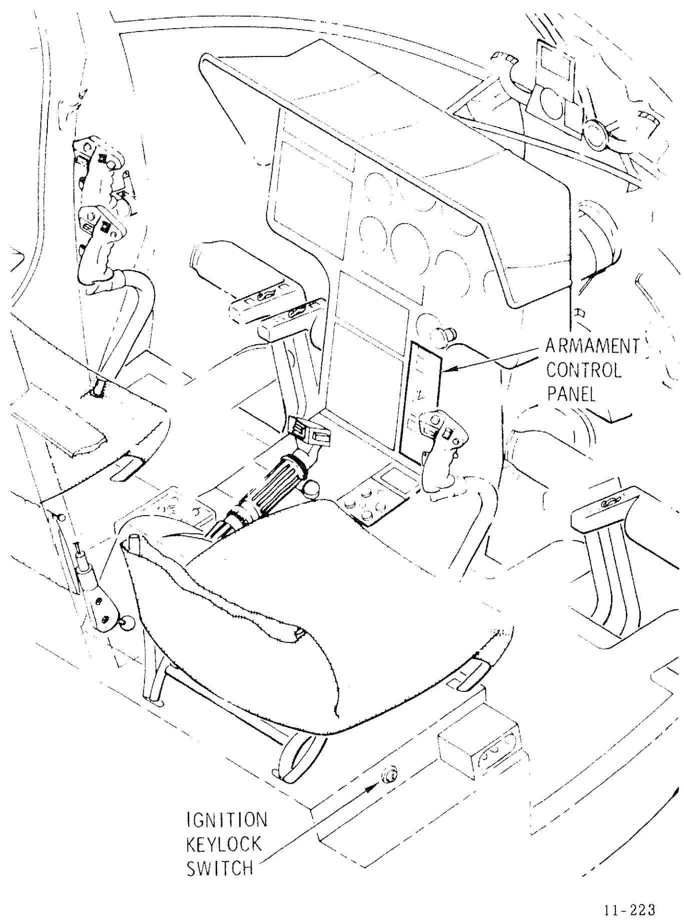


Figure 9-28. Armament Control Panel and Ignition Keylock Switch Locations.

control panel to the instrument panel. Extract the panel and electrical harness.

9-238. Repair — Armament Control Panel. (See fig. 9-29.) Unsolder wiring from switches, indicator lights, and the light-circuit board connector only to the extent necessary to replace defective parts. Lamps (imbedded) in the light-circuit board are not replaceable: replace the light-circuit board. Replace warning and caution light assemblies and lamps as shown in figure 9-22. Note that legend caps are pulled straight out to remove, and are individually indexed so they cannot be mixed.

9-239. Inspection — Armament Control Panel. (See fig. 9-29.) *a.* Examine plastic parts for damage and deterioration.

b. Examine lamp receptacles for dirt and corrosion.

c. Examine electrical connections for security and corrosion.

d. Make sure the armed-SAFE switch locks securely in the SAFE position.

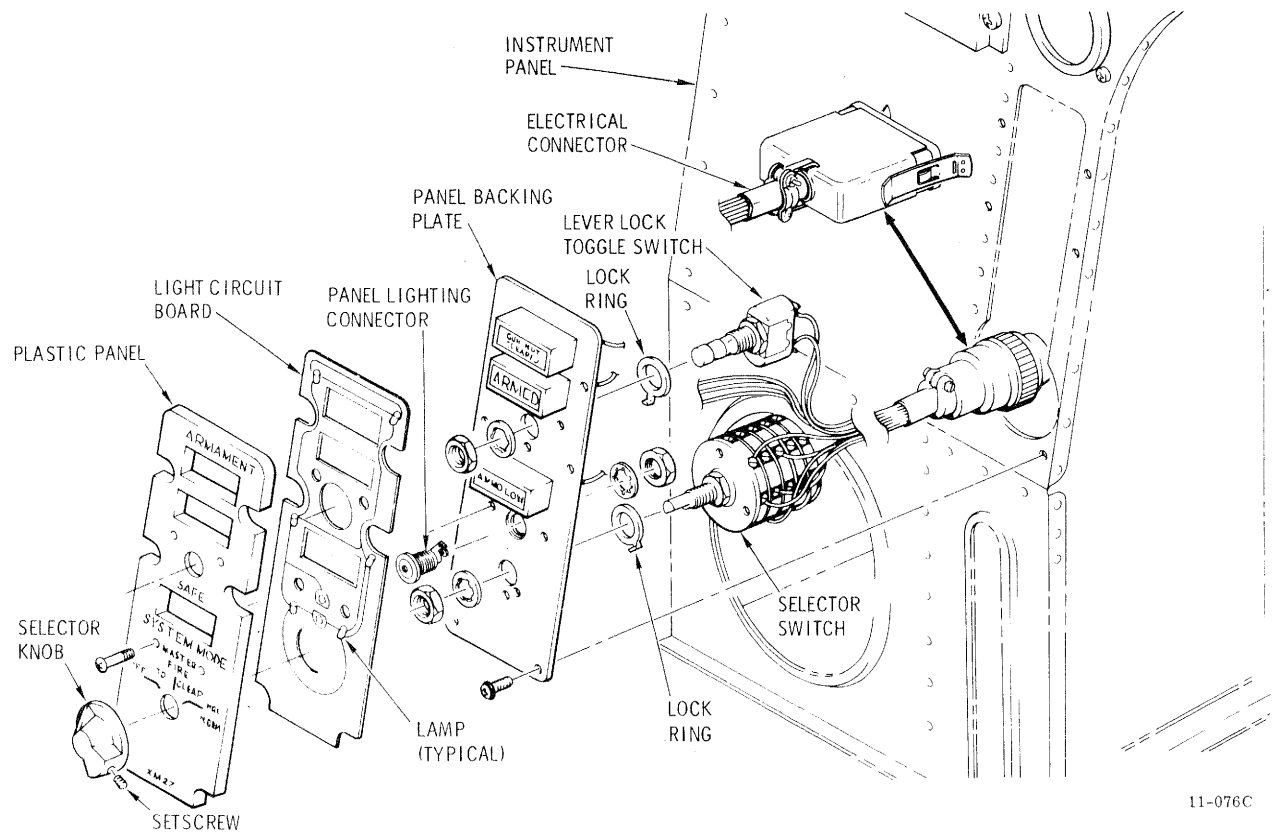


Figure 9-29. Armament Control Panel Replacement.

e. Examine wires for damaged insulation.

9-240. Installation — Armament Control Panel. (See fig. 9-29.) *a.* Set the power selector switch at OFF.

b. Position the armament control panel on the instrument panel and secure it with six screws.

c. Connect armament control panel harness connector J12 to harness connector P12 behind the instrument panel.

CHAPTER 10

FUEL SYSTEM

SECTION I CRASH-RESISTANT FUEL CELLS

NOTE

This section of this manual is applicable only to aircraft which have been modified to incorporate the Crash-Resistant (CR) Fuel System per MWO 55-1520-214-50-6.

For information relative to aircraft which have not been modified per MWO 55-1520-214-50-6, refer to Sections III and IV.

Major paragraphs, figures and tables of applicable sections in this chapter are coded "(CR)" or "(NCR)" in order to clearly distinguish between the two alternate fuel systems.

10-1. CR FUEL CELLS.

10-2. Description — CR Fuel Cells. The CR fuel cells (fig. 10-1) are supported within the aircraft structure by contoured back-up liners installed over the airframe ribs. Foam filler blocks located between the rib cavities reduce air space where fuel fumes might otherwise accumulate. Each cell is secured to the underside of the cargo compartment floor by three hanger assemblies attached to the cargo compartment fuel access door. Additional support is provided at valve-to-structure attachment points. The right cell has openings for the gravity fill valve, closed circuit fuel receiver, crossover coupling, drain valve, vents and cell cover. The left cell has openings for the fuel outlet valve, crossover coupling, drain valve, vents and cell cover. The submerged fuel shutoff valve and tank unit, located in a covered well, are mounted under the left cell cover.

10-2A. Inspection Procedures.

a. Prepare the aircraft for defueling in accordance with TM 55-1500-204-25/1.

b. Remove five gallons of fuel from the aircraft gravity filler neck to prevent spillage when access panels are opened.

c. Remove the fuel quantity transmitter NSN 6680-00-410-5912, P/N 369A4245, tank unit adapter, and cover in accordance with TM 1520-214-23, Chapter 10.

d. Inspect fuel for foreign materials such as lumps of spongy slimy natural rubber which has been activated by fuel, and remove all large pieces.

e. Defuel the aircraft in accordance with TM 55-1500-204-25/1.

f. Remove the access doors on both fuel cells in accordance with TM 55-1520-214-23.

g. Inspect the insides of both fuel cells for cell activation and/or damage to the inner cell walls.

(1) Cell activation will appear as a bulge in the wall of a fuel cell which is moist with fuel and spongy to the touch.

(2) Any bulge in the fuel cell wall larger than one inch in height and three inches in diameter is suspect.

(3) Both visual and touch methods of inspection should be used to confirm suspected areas.

(4) Activation of a fuel cell is not reparable.

h. Replace defective fuel cells in accordance with TM 55-1520-214-23, Chapter 10.

10-3. Removal — CR Fuel Cells (General). Before maintenance is performed on fuel system components (except fuel shutoff valve control), accomplish the following when indicated in the applicable paragraph.

a. Check that all electrical power is OFF; disconnect external power and the battery.

b. Ensure that aircraft is electrically grounded.

c. Drain all fuel from the fuel cells (chapter 1).

WARNING

Purging of fuel cells requires special safety precautions.

d. Purge fuel cells according to TM 55-1500-204-25/1 or appendix E.

NOTE

Refer to TM 55-1500-204-25/1 for supplemental information on fuel cells.

10-4. Removal — Right CR Fuel Cell. *a.* Remove gravity fill valve (para 10-78).

b. Remove closed circuit fuel receiver (para 10-71).

c. Remove right side aft vent system (para 10-64).

d. Remove crossover fitting (para 10-57).

e. Remove drain valve and O-ring using a wide-blade screwdriver.

f. Fold and remove right fuel cell (para 10-10).

10-5. Removal — Left CR Fuel Cell. *a.* Remove tank unit (para 10-27).

b. Remove Fuel shutoff valve (para 10-46).

c. Remove fuel pump (para 10-51).

d. Remove fuel supply hoses (para 10-54).

e. Remove left side aft vent system (para 10-64).

f. Remove crossover fitting (para 10-57).

g. Remove drain valve and O-ring using a wide blade screwdriver.

h. Fold and remove left fuel cell (para 10-10).

10-6. Inspection - CR Fuel Cell. If the fuel cell exhibits defects or damage, contact USATSARCOM (Refer to para P-33).

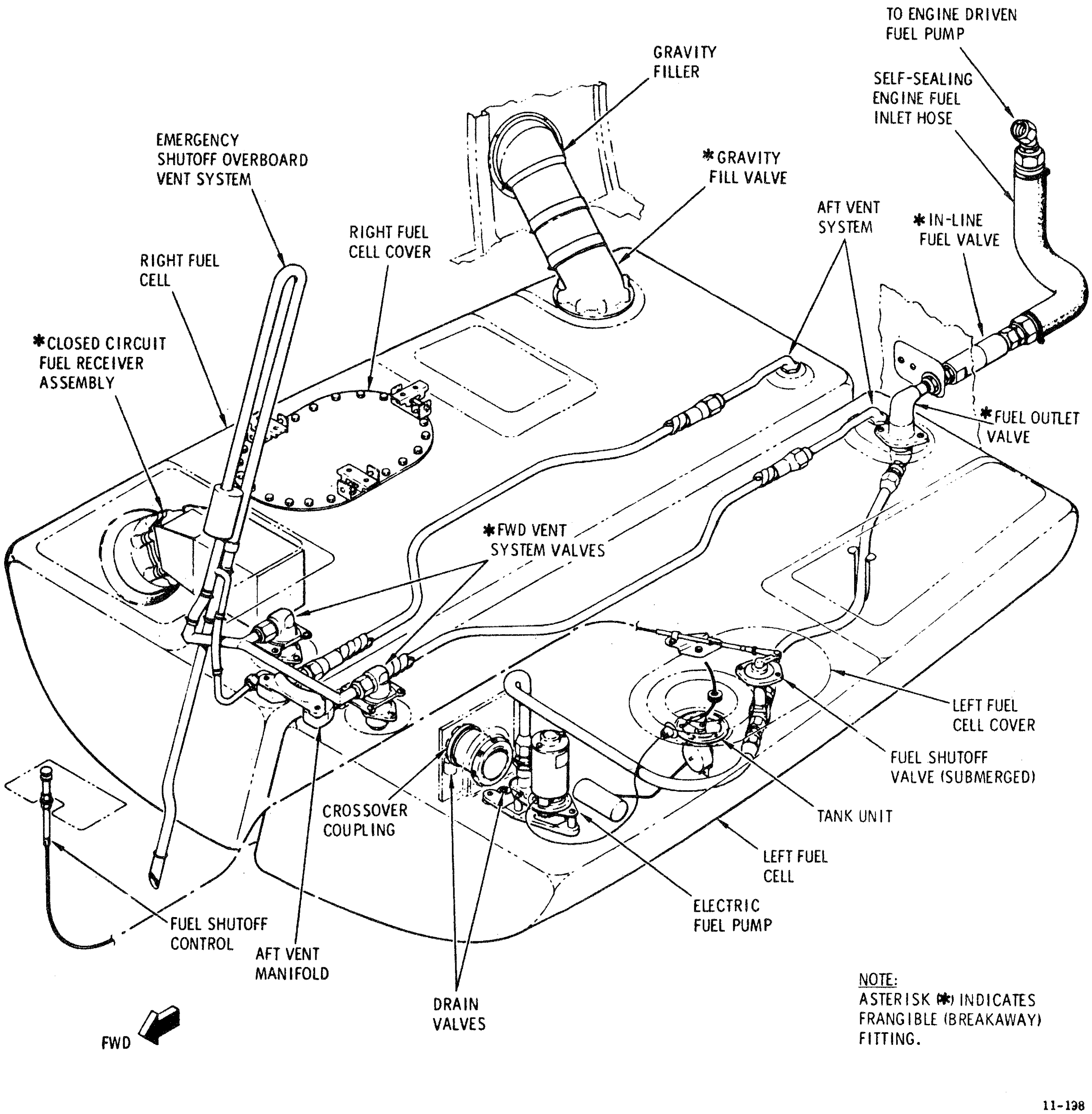


Figure 10-1. CR Fuel System.

10-7. Repair - CR Fuel Cells. Before attempting any fuel cell repair, contact USATSARCOM (Refer to para P-33).

10-8. Installation — Right CR Fuel Cell. *a.* Fold and install right fuel cell (para 10-10).

NOTE

Do not install fuel cell access door until step h below has been completed.

b. Before installation, lubricate drain valve O-ring and valve threads with petrolatum (C73). Then install drain valve with O-ring using a wide-blade screwdriver.

c. With left cell in place, install crossover fitting (para 10-59).

d. Install right side aft vent system (para 10-67).

e. Install closed circuit fuel receiver (para 10-74).

f. Install gravity fill valve (para 10-79).

g. Check that protective caps on access door nutplates under cargo floor that might contact fuel cell are in place. Install new caps, if required, and puncture caps to relieve air pressure.

h. Protect top of fuel cell in areas of possible chafing with tape (C102).

i. Pressure check complete fuel system (para 10-11).

10-9. Installation — Left CR Fuel Cell. *a.* Fold and install left fuel cell (para 10-10).

NOTE

Do not install fuel cell access door until step j below has been completed.

b. Install drain valve with O-ring using a wide-blade screwdriver. Lubricate O-ring and valve threads with petrolatum (C73) before installation.

c. With right cell installed, install crossover fitting (para 10-59).

d. Install left side aft vent system (para 10-67).

e. Install fuel supply hoses (para 10-55).

f. Install fuel pump (para 10-52).

g. Install fuel shutoff valve (para 10-48).

h. Install tank unit (para 10-30).

i. Check that protective caps on access door nutplates, under cargo floor, that might contact fuel cell, are in place. Install new caps, if required, and puncture caps to relieve air pressure.

j. Protect top of fuel cell in areas of possible chafing with tape (C102).

k. Pressure check complete fuel system (para 10-11).

10-10. CR Fuel Cell — Folding. Each fuel cell must be folded for either removal from, or installation in, the aircraft. The folding and restraining procedure is identical in either case. Due to the size of the fuel cell opening in the cargo compartment floor, instructions must be carefully followed to prevent damage to the cell and aircraft structure surrounding the cell.

a. Folding — CR Fuel Cell. (See fig. 10-2.)

NOTE

To facilitate folding, the fuel cells should be warmed to not more than 150°F.

(1) Flatten the fuel cell by tucking the forward and aft ends inward in accordion fashion to shorten the tank outline as much as possible; see figure 10-2. Use a nylon or canvas strap installed lengthwise around the tank to hold it in the flattened condition. Tie strap at end of cell.

NOTE

For installation, note all tie straps should be tied using a single bow-type half-hitch that can be pulled free after the tank is installed.

(2) Make a lengthwise fold in the cell inboard of the large fitting in the top of the cell (just outboard of the engine fuel outlet fitting in the left cell). The fold (crease) will be downward looking at the top of the cell. Install four crosswise straps to hold the cell in this condition.

NOTE

Cargo tiedown straps with ratchet type tension units may be used to fold the cells and hold them while the tie straps are being installed.

(3) Tie a puller strap to the crosswise strap around the aft end of the cell. This step is particularly necessary for the left cell.

(4) Remove the lengthwise strap installed in step a above and then fold the aft 1/4 of the cell downward at 90 degrees to the lengthwise fold. Tie this fold down securely with a strap.

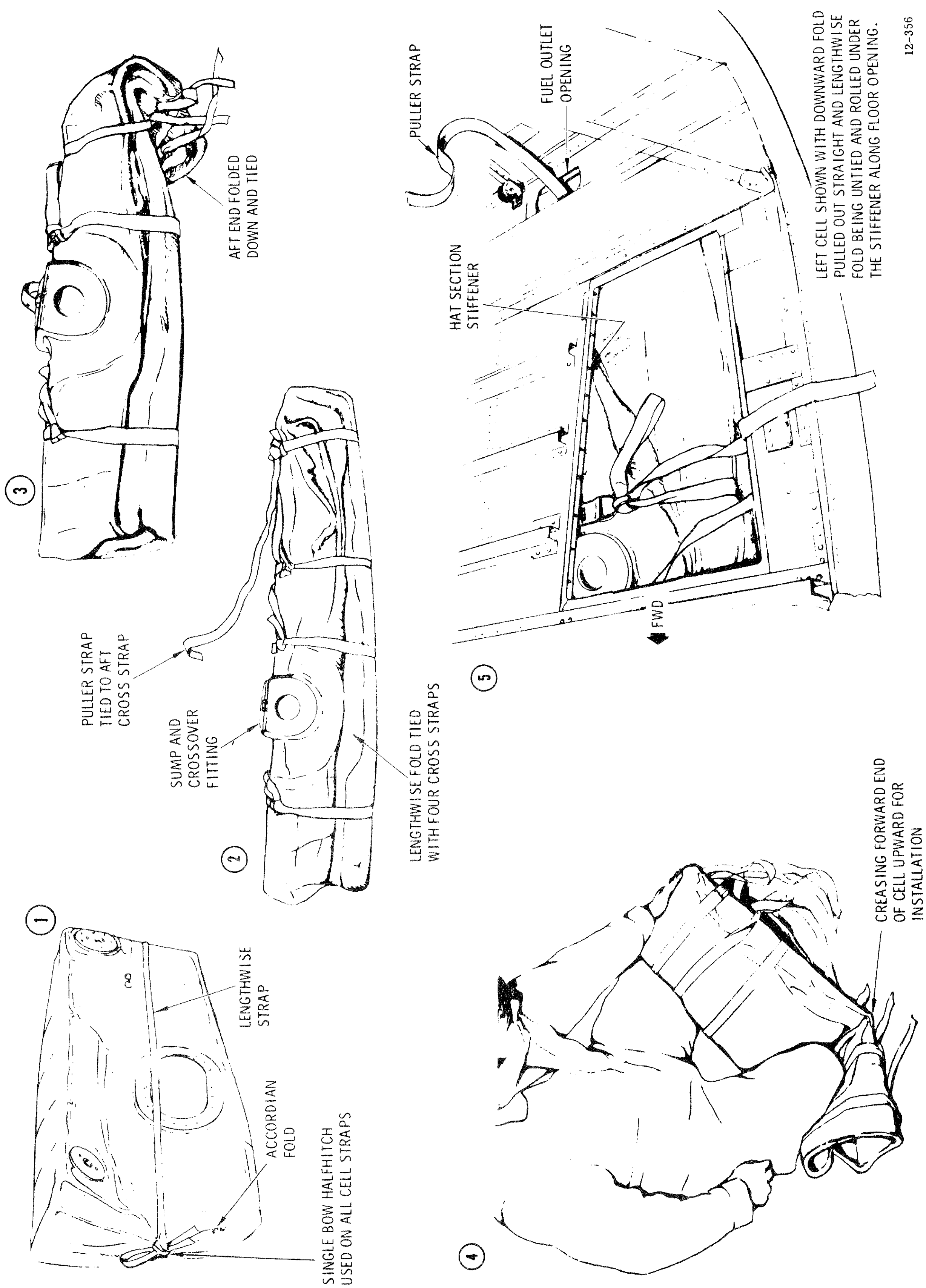


Figure 10-2. Folding CR Fuel Cells.

NOTE

The ends of the cell, particularly the forward end, must remain tucked inward to reduce package length as much as possible.

b. Removal — Folded CR Fuel Cell.

(1) Tape the edges of the floor openings to prevent damage to the cells during removal.

(2) Dust the accessible areas of the fuel cell and cell compartment with talc or soapstone.

(3) Work a long pull strap through the cell aft openings.

(4) Work the folded cell around until crossover opening is facing upward.

(5) Using straps with ratchet type tension units, pull the cell as far forward as possible.

(6) Work the folded aft end of the cell up and out through the floor opening.

NOTE

A heat lamp or heat gun (blower) may be used to warm corners and creases of the cell that resist manipulation. Heat only until warm to touch, **NOT TO EXCEED 150 DEGREES F.**

(7) Withdraw the cell rearward, bending the cell as it passes through the floor opening.

c. Installation — Folded CR Fuel Cell.

(1) Cover exposed edges of fuel compartment backup liner and attachment screw heads with tape (C103).

(2) Check that fuel cell compartment is clean and free of foreign material.

(3) Tape the edges of the floor openings to prevent damage to the cells during installation.

(4) Dust the interior of the cell area (backup liner) and all corners and folds with talc or soapstone.

(5) Bend the front 1/4 of the cell upward to create a flexible crease and then slide forward end of the cell downward and forward through the floor opening.

NOTE

Be sure cell is full forward by checking position through the forward vent opening in the floor.

(6) Route the puller strap through the engine fuel outlet opening in the floor for the left cell, or through the gravity fill opening for the right cell.

(7) Push folded aft end of the cell downward until it is almost clear of the floor opening and then remove the strap holding the fold. Unfold the aft end of the cell into proper position by using the puller strap and rolling and maneuvering the cell as required.

(8) Rotate the cell toward upright position as much as possible; then untie the aft strap holding the lengthwise fold. Continue rotating the cell and removing the straps by working toward the front. The lengthwise crease in the cell must be rotated inboard past the hat section stiffener under the floor (working from aft end forward) before cell will unfold.

NOTE

A heat lamp or heat gun (blower) may be used to warm corners and creases of the cell that resist unfolding. Heat only until warm to touch, **NOT TO EXCEED 150 DEGREES F.**

(9) Align cell fittings with associated openings in aircraft structure.

NOTE

Hardwood dowels, 1.0 inch diameter by 24 inches long, and various lengths of hardwood blocks 1.0 inch by 3 inches may be used to assist in tank positioning.

CAUTION

Do not use sharp tools. Wooden dowels may be used but do not pry directly against thin skin of floor openings. Use wooden blocks to space out from some stronger structural member and to spread the load.

Table 10-1. Premaintenance Requirements for Pressure Check of CR Fuel Cells.

Conditions	Requirements
Support Equipment	(S13)

10-11. Pressure (Leak) Check — CR Fuel System. *a.* If fuel cell access doors are not installed, place strips of 3/8-inch plywood across fuel cells exposed in floor openings (under the floor hat sections) so that cells will not bulge when pressurized.

b. Close the cell overboard vent tube with a suitable pressure tight but removable plug. Such a plug can be made with a 4-inch length of 5/8-inch ID plastic tubing (Tygon) and an 8-inch length of 1/2-inch OD wooden or plastic dowel. Insert the Tygon tubing in the vent tube, and then insert the dowel to expand seal the Tygon tube in the vent tube.

c. Check that fuel shutoff valve is open (control knob down) and the filler caps are secure.

d. Disconnect the engine end of the fuel hose, tee-in a manometer or an extra sensitive 0 to 10-psi gage and a source of accurately regulated air pressure (S13).

CAUTION

DO NOT PRESSURIZE FUEL SYSTEM ABOVE 2.0 PSIG during the following operation. Serious structural damage or deformation will result from higher pressures.

e. Pressurize the system to 2.0 psig (4 in. Hg). **Air leakage from the system shall not allow a decrease in pressure below 1.88 psig (3.75 in. Hg) after 15 minutes.**

f. Locate leaks, if required, by brushing bubble fluid or a detergent and water solution over suspected fittings and connections. Repair as required and retest.

g. Remove plug from overboard vent tube. Remove wood strips across cells at the floor openings.

h. Remove gage and pressure source. Connect engine fuel hose to the engine fuel pump and **TIGHTEN HOSE FITTINGS TO 270 — 420 INCH-POUNDS TORQUE.**

SECTION II CRASH-RESISTANT FUEL SYSTEM

NOTE

This section of this manual is applicable only to aircraft which have been modified to incorporate the Crash-Resistant (CR) Fuel System per MWO 55-1520-214-50-6.

For information relative to aircraft which have not been modified per MWO 55-1520-214-50-6, refer to sections III and IV.

Major paragraphs, figures and tables of sections in this chapter are coded "(CR)" or "(NCR)" in order to clearly distinguish between the two alternate fuel systems.

10-12. CR FUEL SUPPLY SYSTEM.

10-13. Description — CR Fuel Supply System. The crash-resistant (CR) fuel supply consists of two interconnected self-sealing fuel cells located in separate compartments beneath the cargo compartment floor. (See fig. 10-1.) The fuel cells are serviced by using either the gravity filler or the closed circuit pressure receptacle located on the right side of the aircraft. Two drain valves are located on the fuselage underside. A submerged electric fuel pump is located in the left fuel cell. The pump provides a positive supply of fuel to the engine-driven fuel pump during starting and during high altitude/high temperature flight operations. Fuel is then drawn from the cell, through a submerged fuel shutoff valve and submerged fuel lines to a self-sealing hose in the engine compartment by the engine-driven fuel pump. A fuel quantity tank unit with associated electrical wiring is located in the the top of the left cell. Forward and aft vents are located on each fuel cell. The vent system vents overboard through an emergency shutoff vent valve arrangement located in the cargo compartment forward bulkhead. The fuel cell filling, venting and engine fuel supply systems are so constructed that fuel in the cells will not escape through fillers, vents or fuel hoses in the event of structural damage and contribute to a post-crash fire. This is accomplished with frangible (breakaway) fittings that are installed at all fuel systems-to-aircraft joints. At impact, the fitting will separate and a spring-loaded valve in each fitting will close to retain fuel upstream and down stream of the valve. The breakaway fittings are identified in figure 10-1 with an asterisk (*) symbol and are discussed in the description of their respective systems.

10-14. Inspection — CR Fuel Supply System (Installed). *a.* Check the exterior underside of the fuselage beneath the fuel cells for indications of fuel leakage.

b. Remove left and right inspection access covers (fig. 10-3) from fuel cell access doors to gain access to the fuel cell covers.

c. Check fuel cell drain valves for leaks and security.

d. Check visible fuel vent hoses, fittings, and fuel cell covers for leakage, damage, and secure attachment.

e. Check tank unit cover and electrical wiring for damage and secure attachment.

f. Check fuel shut off valve area for leaks, damage, and security.

g. Check fuel cell hanger assemblies for damage and security.

h. Replace left and right inspection access covers on cargo floor.

i. Open left engine access door. Check engine fuel pump inlet line, and fuel line connections at pump and bulkhead fittings for leaks, damage, and secure attachment. Check fire sleeve for cuts, tears and punctures. Close access door.

j. Check that overboard vent opening is not clogged by ice, mud, or other foreign matter that may have come in contact with the underside of fuselage.

10-15. Troubleshooting — CR Fuel Supply System. Refer to table 10-2.

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

10-16. CR FUEL CELL ACCESS DOOR.

CAUTION

These are stressed doors. The aircraft must never be flown, towed or jacked with either door removed.

10-17. Description — CR Fuel Cell Access Door. The two fuel cell access doors (fig. 10-3) form a structural part of the cargo compartment floor and provide three major attachment points for each fuel cell. Each attachment point consists of a slotted upper bracket attached by a pin to another mating bracket mounted on the fuel cell cover; this provides a floating swivel for the cell. Each upper bracket is secured under the door with two screws that are installed from the cargo compartment side and thread into nutplates on the upper bracket. Inspection of the fuel system covers and fuel shutoff valve-control is accomplished by removing a small square inspection access cover located in the center of the access door. If further inspection or adjustment is required, either access door is removed as follows.

10-18. Removal — CR Fuel Cell Access Door. *a.* Remove two screws and washers (fig. 10-3) and lift off square inspection access cover.

b. From the cargo compartment floor side, remove two screws and washers located three places directly aft, left, and right of access hole.

NOTE

Screws are threaded into nutplates on hanger assemblies.

c. Remove 62 retaining screws and washers and lift out access door.

10-19. Inspection and Repair — CR Fuel Cell Access Door. Refer to chapter 2 for fuel access door inspect and repair.

10-20. Installation — CR Fuel Cell Access Door.

NOTE

Before installation of fuel cell access doors, check that protective caps on access door nutplates under cargo floor that might contact fuel cell are in place. Install new caps, if required, and puncture caps to relieve air pressure. Protect top of fuel cell in areas of possible chafing with tape (C102).

a. Position access door over opening (fig. 10-3) and secure in place with 62 retaining screws and washers.

b. Reach through the small inspection access hole and align the holes in each upper hanger bracket with the corresponding attachment holes in the fuel cell access door. Install two screws and washers at each of the hanger locations.

NOTE

Do not disassemble pin from hanger assembly. If the cell has settled and screws will not reach into hanger nutplates, temporarily install two long bolts (AN3-10) and pry against head of bolt to align bracket holes with cargo floor; then remove one bolt at a time and install attachment screws and washers.

c. Reinstall square inspection access cover with two screws and washers.

10-21. CR FUEL QUANTITY INDICATING SYSTEM.

10-22. Description — CR Fuel Quantity Indicating System. The fuel quantity indicating system consists of a fuel quantity transmitter (tank unit) mounted in a

Table 10-2. Troubleshooting the CR Fuel System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **No fuel flow to engine driven fuel pump (engine fails to light off). If unable to open valve, inspect control cable and tank valve for correct operation.**

STEP 1. Check that fuel shutoff valve control is not closed (para 10-38).

If valve is closed, push in to open fuel shutoff control.
If unable to open valve, inspect control cable and tank valve for correct operation (para 10-45).

STEP 2. Check for fuel in fuel cells.

If cells are empty, service according to chapter 1.

STEP 3. Check for loose, damaged, or disconnected fuel line.

If discrepancy exists, tighten loose or disconnected fuel line(s); or replace damaged component (para 10-53).

STEP 4. Check for clogged filter at engine driven fuel pump.

If filter is clogged, drain filter and replace filter element (TM 55-2840-231-24).

STEP 5. Check for defective fuel outlet or in-line breakaway valve.

If fuel outlet or in-line breakaway valve is defective, replace it (para 10-53).

STEP 6. Check for disconnected submerged fuel hose connection.

If submerged hose is disconnected, reconnect and torque nuts (para 10-49).

2. **Restricted fuel flow to engine driven fuel pump.**

STEP 1. Perform all steps in Malfunction No. 1, as required.

STEP 2. Check for crimped or twisted fuel inlet hose (para 10-53).

If discrepancy exists, remove hose and reposition fittings of hose to eliminate twist or replace hose.

STEP 3. Check for clogged fuel line.

If line is clogged, remove, clean and/or replace defective fuel line. Drain and check for fuel cell contamination.

STEP 4. Check for loose submerged fuel hose connection (para 10-49).

If submerged hose is loose, tighten and retorque.

covered well located on the left fuel cell access cover (fig. 10-4) and a fuel quantity indicator and FUEL LOW caution light mounted on the instrument panel. When the fuel quantity level is 35 pounds or less, a ground circuit is completed from the tank unit to the caution light, causing the light to illuminate. On aircraft equipped with the CR fuel system, a 369A4245 tank unit and a 369A4519-3 quantity indicator must be used. There are no external adjustment provisions for either the fuel quantity indicating system of FUEL LOW caution light. Refer to Chapter 9 for FUEL LOW caution light maintenance information and appendix F for electrical wiring diagrams. Refer to table 10-3 for troubleshooting the indicating system.

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

10-23. CR FUEL QUANTITY TRANSMITTER (TANK UNIT).

10-24. Description — CR Fuel Quantity Transmitter (Tank Unit). The tank unit is a vertical mount, float type unit mounted in a covered well located in the left fuel cell access cover. (See fig. 10-4.) Mechanical stops

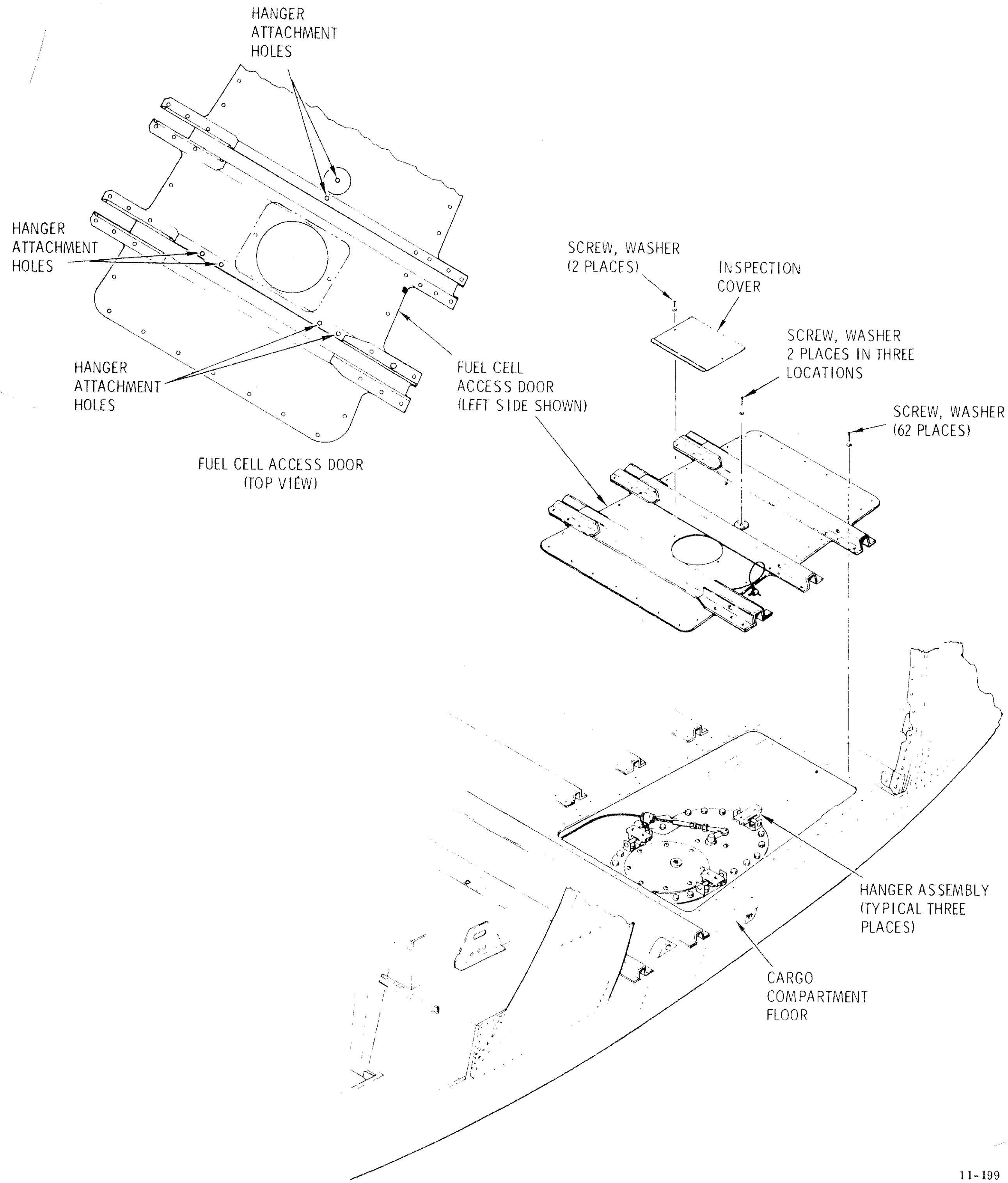


Figure 10-3. CR Fuel Cell Access Door.

Table 10-3. Troubleshooting the Fuel Quantity Indicating System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

1. **Fuel quantity indicator inaccurate, erratic, or inoperative.**

STEP 1. Check for loose, shorted, or broken electrical wires or connectors at the tank unit, fuel quantity indicator and associated wire harnesses (para 10-23).

If discrepancy exists, repair or replace electrical wires or connectors as necessary (chapter 9).

STEP 2. Check for water (moisture) accumulation in access cover or electrical connections.

If discrepancy exists, clean, air dry and seal as necessary.

STEP 3. Check for defective fuel quantity indicator.

If fuel quantity indicator is defective, replace it.

STEP 4. Check for defective tank unit (para 10-23).

If the tank unit is defective, replace it.

2. **FUEL LOW caution light erratic or inoperative.**

STEP 1. Check for defective caution light.

If discrepancy exists, replace defective caution light (chapter 9).

STEP 2. Check for loose, shorted, or broken electrical wires or connectors at the tank unit, caution light and associated wires harnesses (chapter 9).

If discrepancy exists, repair or replace electrical wires or connectors as necessary.

STEP 3. Check for broken or loose tank unit fuel low contact spring wire (fig. 10-4).

If discrepancy exists, replace tank unit (para 10-23).

3. **FUEL LOW caution light illuminates with fuel cell quantity above 35 pounds.**

STEP 1. Check for short in caution light circuit (chapter 9).

If discrepancy exists, repair or replace electrical wires or connectors as necessary.

STEP 2. Check for defective tank unit.

If tank unit is defective, replace it (para 10-23).

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in the tank unit housing limit float travel. Movement of the float-actuated rheostat within the housing produces a current flow imbalance in three indicator pointer electromagnets, causing the pointer magnet to move. (Refer to para 10-31.) Fuel quantity transmitter 369A4245 is used with the CR fuel system.

10-25. Inspection — CR Fuel Quantity Transmitter (Tank Unit)(Installed). *a.* Check visible electrical harness for obvious damage and security. Check tank unit cover for secure attachment.

b. Refer to fuel quantity indicator operational check (para 10-29) to verify correct operation of tank unit.

c. Check for evidence of water accumulation around fuel tank unit. If water is or has been present, the tank unit may be sealed against moisture (para 10-26).

10-26. Repair — CR Tank Unit Sealant. Electrical shorting and water damage of the tank unit may be prevented by sealing the tank unit transistor wiring against moisture as follows.

a. Set the power selector switch at OFF. Ensure that external power and battery are disconnected.

b. Thoroughly clean the left fuel cell cover area around the tank unit with naphtha (C70) and allow to air-dry.

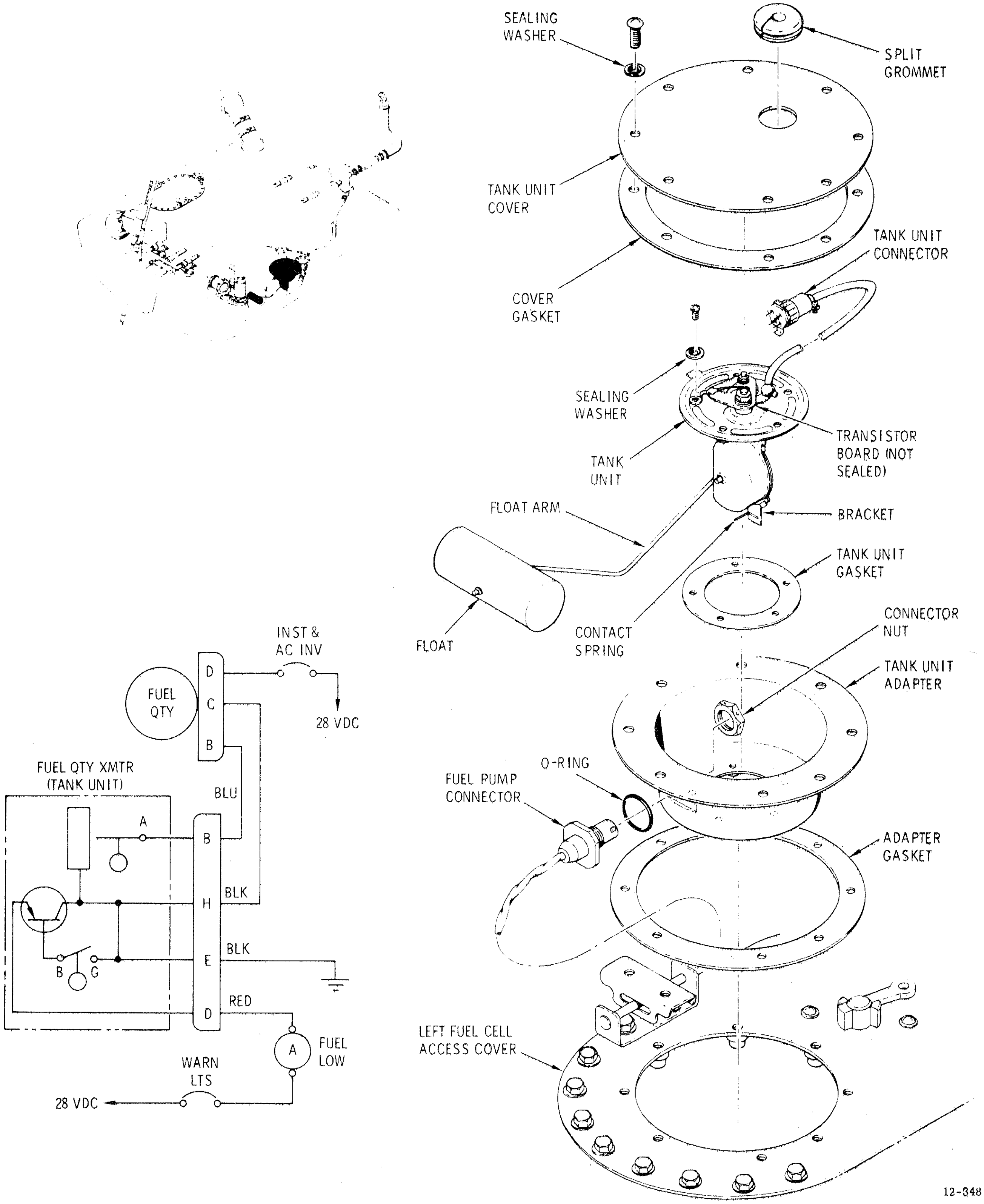


Figure 10-4. CR Fuel Quantity Transmitter (Tank Unit 369A4245).

c. Using a wooden spatula, carefully pack the tank unit exposed transistor board wiring and surrounding area with sealing compound (C89). Do not cover the five mounting screws.

d. Perform an operational check (para 10-29).

10-27. Removal — CR Fuel Quantity Transmitter (Tank Unit). (See fig. 10-4.) *a.* Prepare aircraft for maintenance according to paragraph 10-3.

b. Remove left fuel cell access door.

c. Disconnect tank unit electrical connector.

d. Remove tank unit cover screws and sealing washers; lift cover far enough to disconnect the fuel pump electrical connector located inside of the tank unit adapter.

e. Remove the split grommet in tank unit cover and separate cover and gasket from electrical wiring.

f. Raise the tank unit adapter to gain access to rear side of the fuel pump electrical connector; hold connector and remove connector nut, connector and preformed packing (O-ring).

g. Carefully remove tank unit adapter and adapter gasket with tank unit installed. Do not allow fuel low contact spring on tank unit to be bent during removal.

h. Remove five screws and sealing washers and carefully remove tank unit and tank unit gasket.

10-28. Inspection — CR Fuel Quantity Transmitter (Tank Unit). *a.* Inspect tank unit housing, float and arm, and exposed pivot mechanism for cracks, deformation or corrosion. Replace unit if damaged.

b. Check float and arm for free movement within travel limits.

c. Check that float offset is sufficient for the float to clear the edge of the cell sump and allow full travel of the arm.

d. If transistor tank unit board is not sealed (potted) against moisture, check wiring for condition.

Table 10-4. Premaintenance Requirements for Testing Tank Unit (369A4245).

Conditions	Requirements
Special Tools	(T4)

10-29. Operational Check — CR Tank Unit 369A4245 (Testing). *a.* Set up tank unit on a level platform approximately 9 inches above a second level surface such as a table top (fig. 10-5).

b. Set multimeter (T4) on low range resistance scale.

c. Allow float arm to rest on the float empty stop; then measure resistance across pins B and H of tank unit connector. **RESISTANCE MUST BE 0 to 3.0 OHMS.** Adjust (bend) the float empty stop until this resistance is obtained.

NOTE

Old type pins B and H can be used on the new transmitter in lieu of the new split pins.

d. Raise the float until arm is in contact with the full stop and observe that resistance across pins B and H increases to approximately 88 ohms.

e. Measure from the centerline of arm pivot in float when in full position, to the same point when in empty position. This measurement should be **8.99 TO 9.11 INCHES.** Bend the float full stop as required to obtain **88 to 95 OHMS** within these float travel limits.

f. Disconnect the multimeter. Make test lamp wiring connections as shown in figure 10-5 schematic wiring diagram.

g. Apply power and check that test lamp illuminates when centerline of arm to floor attachment is **1.25 INCHES ABOVE THE EMPTY POSITION.**

h. Bend the spring attached to tank unit bracket, if required, and recheck that lamp remains illuminated from the empty (rest) position to the **1.25 INCH RAISED POSITION.**

i. Disconnect power and all test equipment.

10-30. Installation — CR Fuel Quantity Transmitter (Tank Unit). *a.* Install tank unit with new gasket in tank unit adapter (float directed inboard as shown in figure 10-4) with 5 screws and sealing washers. Do not over-tighten screws.

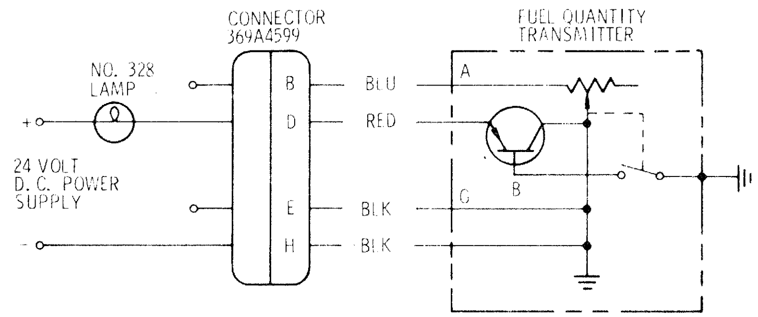
NOTE

When installing screws with NAS1598-3R sealing washers, be sure that correct length screws are used and that the screws are not overtightened. Excessive torque can destroy the washer O-ring sealing capability, overcompress the rubber gasket and bottom the screw in the attaching platenut.

b. Position new tank unit adapter gasket on fuel cell access cover.

NOTE

Attachment holes in access cover, gaskets, and associated parts are unevenly spaced so that parts may only be installed one way. If holes do not line up, part is inverted.



TEST SETUP AND COMPONENTS

NOTES:

1. NO. 328 LAMP.
2. 24 VDC POWER SUPPLY.
3. WIRING (SIZE AWG 20) AS REQUIRED.

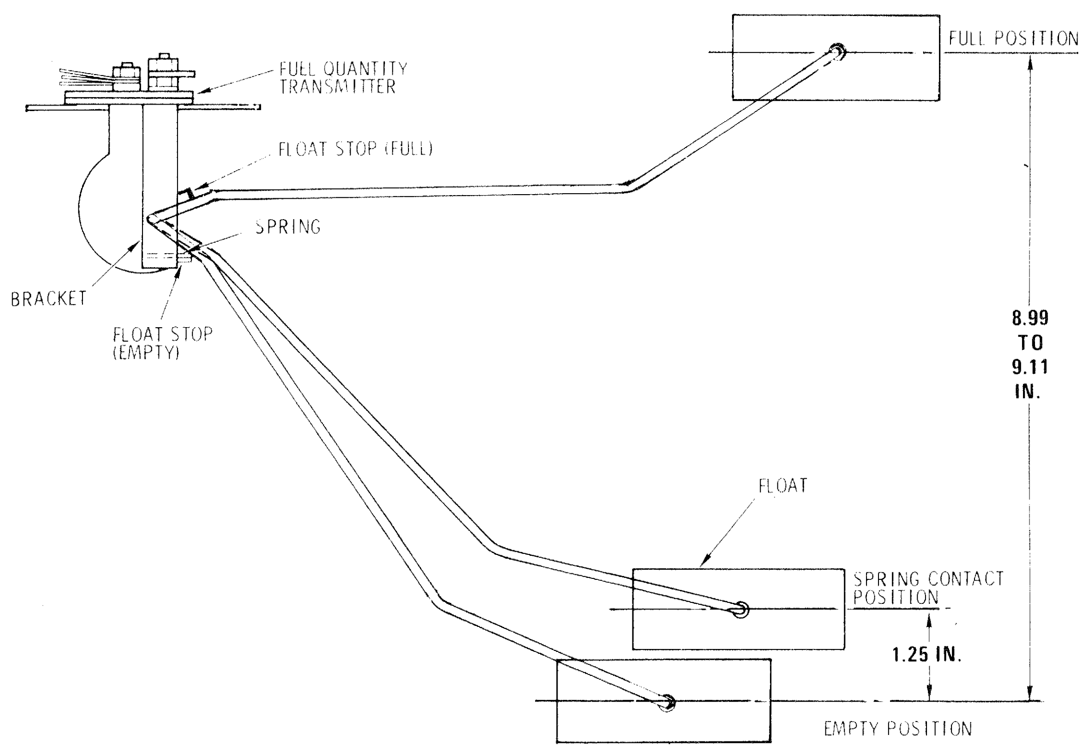


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Figure 10-5. CR Tank Unit Testing (Transistor Switch Unit).

c. Position fuel pump electrical connector with new O-ring seal in tank unit adapter and install nut.

d. Position the assembled tank unit and adapter on the fuel cell access cover. Check that lower gasket remains in place.

e. Place new tank unit cover gasket in position on tank unit adapter.

f. Assemble tank unit cover by inserting fuel pump connector electrical harness down through cover hole and tank unit connector electrical harness up through cover hole; then install split grommet.

g. Connect both electrical connectors and position cover on gasket. Secure cover with 8 screws and sealing washers. Do not overtighten screws.

h. Install left fuel cell access door.

i. Refill fuel cells and check tank unit cover area for leakage by removing the inspection access cover in the access door.

10-31. CR FUEL QUANTITY INDICATOR.

10-32. Description — CR Fuel Quantity Indicator. The fuel quantity indicator is a 28 vdc operated ratiometer that indicates, in pounds, the quantity of fuel present in the fuel cells. Current is applied to the fuel quantity indicator through the INST and AC INV or INST circuit breaker and is distributed to three internal electromagnets. A permanant magnet, with pointer attached, is positioned so that current flow through the three electro-magnets affects pointer position.

10-33. Operational Check — CR Fuel Quantity Indicator. Perform the following check to determine that fuel quantity indicator readings are within tolerance.

a. Defuel the aircraft (chapter 1).

b. Jack the aircraft and block under the skids so that the aircraft is in a 2 degree nose-down position. Nose-down angle may be measured with a propeller protractor located on the cargo compartment floor.

c. Connect 28 vdc electrical power and set power selector switch at BATT. Check indicator reading. Pointer must be at zero or below and the FUEL LOW caution light illuminated.

d. Prepare to service the fuel cells. Note that the gravity filler valve is opened by pulling outward on the filler cap and hooking the valve cable in a notched tab inside the filler opening.

e. Pour approximately 7 gallons (45.5 lb) of fuel into the gravity filler opening and observe that the FUEL LOW caution light goes out.

NOTE

Density of JP-4 fuel is 6.5 lb/gal on standard day 59°F (15°C), 29.92 IN. HG, dry air. Fuel density will vary under other ambient conditions and should always be taken into account when indicator accuracy is questionable.

f. Drain fuel from the fuel cell drain valves into a container of known weight until the FUEL LOW caution light illuminates. Weigh the fuel drained and subtract container weight. The fuel weight should be approximately 7 pounds, or within the range of 3 pounds minimum to 10 pounds maximum. This assures that FUEL LOW caution light will illuminate when fuel total in cells decreases to a range of 42 to 35 pounds.

g. Remove aircraft from jacks and allow it to rest on the skids in the normal ground attitude (4° nose-up).

h. Cut a circular template of cardboard or stiff paper to fit on the glass inside the fuel quantity indicator bezel (rim). Cut a 12 degree segment (wedge) out of the template.

i. Proceed with gravity fueling until total in tank is 13.6 gallons (89 lb) and check that indication is at the 1/4 mark ±6 deqrees by centering the 12 degree sector in the template over the 1/4 mark. Make additional checks at 1/2 mark (27.3 gallons (178 lb)), 3/4 mark (41.0 gallons (267 lb)), and full mark (54.7 gallons (356 lb)), using the tamplate to check for ±6 degree tolerance.

10-34. Removal — CR Fuel Quantity Indicator. *a.* Set the power selector switch at OFF.

b. At rear of fuel quantity indicator, loosen clamp on plastic electrical connector hood. Hand-loosen hood far enough to disengage plug lock spring from receptacle lock ring by rotating lock spring in either direction. Remove electrical plug from indicator.

c. Open instrument light fixture. First loosen the cinching screw (10-32 thread) in the lower right corner of the mounting clamp to release the clamp adjusting mechanism; then loosen the upper left holding screw (8-32 thread). Slide the fuel quantity indicator outward, away from instrument panel face and out of the clamp. (Clamp screws do not have to be removed.)

d. Install suitable dust covers on indicator receptacle and mating electrical plug.

10-35. Inspection — CR Fuel Quantity Indicator. *a.* Check fuel quantity indicator case for dents and cracks.

b. Check glass for cracks, scratches, cleanliness, and security in case.

c. Check electrical receptacle and plug for obvious damage and corrosion.

d. Check markings and numerals for legibility. Check pointer and dial for cracked or peeling paint.

10-36. Repair — CR Fuel Quantity Indicator. *a.* Replace an indicator that does not register within operational tolerance (para 10-33). Refer to paragraph 10-21 for replacement part number.

b. If an indicator pointer registers above zero with all fuel drained, replace the indicator.

NOTE

Should the replacement indicator produce similar error, the tank unit should be replaced and the original indicator reinstalled to determine which part is unserviceable.

c. Repeat the operational check (para 10-33) and parts replacement as necessary to obtain fuel level indications that are within accuracy requirements.

10-37. Installation — CR Fuel Quantity Indicator. *a.* Install fuel quantity indicator in mounting clamp. Secure by tightening upper left and then lower right mounting screw. Close instrument light fixture.

b. Remove dust covers from electrical plug and receptacle. Install electrical plug in receptacle, rotate plug lock spring to engage receptacle lock ring. Hand tighten hood to secure lock spring, and tighten hood clamp screws.

10-38. CR FUEL SHUTOFF CONTROL SYSTEM.

10-39. Description — CR Fuel Shutoff Control System. The fuel shutoff valve control (fig. 10-6) installation consists of a sheathed flexible control cable extending from the console, through the pilot's floor, back to the left fuel cell fuel shutoff valve. The red PULL TO CLOSE control knob is mounted on the electrical control console. A threaded terminal swaged to the cable on the shutoff valve end acts as an internal stop. A bracket provides support for the sheathed control cable at the fuel shutoff valve. An adjustable clevis threads onto the cable terminal and is secured to the valve lever by a clevis pin to provide a rotatable valve lever connection. The cable is dry film lubricated and cannot be disassembled. Two fixed and one sliding aluminum spool provide protection for the cable at the airframe attachment clamp locations.

10-40. Inspection — CR Fuel Shutoff Control System. *a.* Remove left fuel cell inspection cover.

b. Remove left foot fairing and open pilot's compartment floor access door (chapter 2).

c. Inspect control assembly (fig. 10-6) and attaching hardware for condition, security and smooth bend radius.

d. Check that forward attachment spool is FREE-FLOATING on cable and that two others are secure.

e. Inspect shutoff valve area for fuel leakage, corrosion and freedom of lever movement.

f. Actuate control and check for smooth operation (minimum bend radius at any point must not be less than 2.75 inches). With the shutoff valve control knob down and a **0.015 TO 0.090-INCH GAP** between bottom of operating control plunger and the surface of the fixed housing, check that the shutoff valve lever contacts the open (clockwise) stop as indicated in figure 10-6.

10-41. Adjustment — CR Fuel Shutoff Control System. *a.* Operate fuel shutoff control (fig. 10-6) to open fuel valve (plunger full down).

b. Observe fuel shutoff valve lever to ensure that lever has moved clockwise to the open stop.

c. Loosen checknut on cable clevis. Remove cotter pin and clevis pin from shutoff valve arm. With valve arm on the open (clockwise) stop, position the valve control knob so that there is a **0.015 TO 0.090-INCH GAP** between the bottom of the plunger and the surface of the fixed housing as indicated in figure 10-6; then adjust the cable clevis so that attachment holes align.

d. Reinstall clevis pin and cotter pin. Tighten checknut.

e. Operate control valve through several cycles and check that shutoff valve lever moves through the full throw of 90 degrees and that lever contacts both the open and closed stops.

10-42. CR FUEL SHUTOFF VALVE CONTROL.

10-43. Removal — CR Fuel Shutoff Valve Control. (See fig. 10-6.) *a.* Remove left fuel cell access door, left foot support fairing, and open pilot's compartment left floor access door.

b. Remove cotter pin and clevis pin at fuel shutoff valve lever.

c. Release checknut and remove cable terminal and checknut.

d. Remove hardware and clamps at three cable support (spool) locations shown in figure 10-6.

e. Under electrical console, loosen fuel shutoff control nut and washer until free from threaded portion of cable shield.

f. Remove fuel shutoff control cable through mounting hole in console.

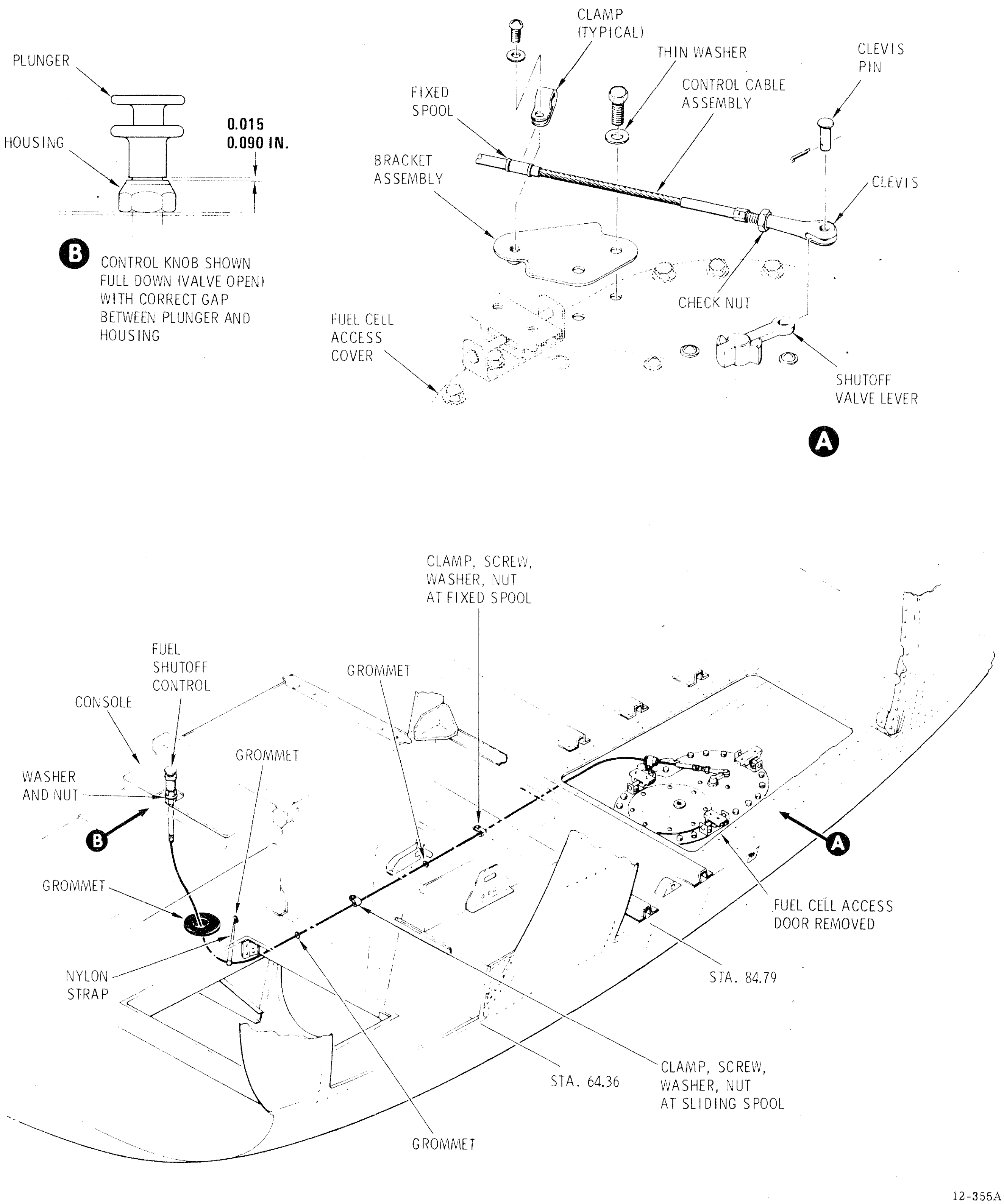


Figure 10-6. CR Fuel Shutoff Valve Control Installation.

g. Remove cable grommets and nylon strap, if unserviceable.

h. Remove two bolts, washers and cable support bracket assembly from fuel access cover, if required.

10-44. Installation — CR Fuel Shutoff Valve Control. (See fig. 10-6.) *a.* Position shutoff cable support bracket on fuel cell access cover and install two attaching bolts with thin washers.

b. Install new cable support grommets and nylon strap if damaged.

c. Place threaded terminal end of fuel shutoff control assembly through mounting hole in console and place control attachment washer and nut over end of control assembly.

d. Route control assembly cable through mounting hole in console, through grommets and strap, and back to fuel shutoff valve.

e. Tighten nut to secure forward end of control.

f. Slide forward spool on cable to forward clamp attachment position. Install cable support clamps and attaching hardware in locations shown in figure 10-6.

g. Install checknut and cable terminal; then adjust control according to paragraph 10-41.

10-45. CR FUEL SHUTOFF VALVE.

10-46. Removal — CR Fuel Shutoff Valve. (See fig. 10-7.) *a.* Remove tank unit according to paragraph 10-27, except do not remove tank unit from adapter.

b. Disconnect fuel shutoff control cable by removing cotter pin and clevis pin at cable terminal.

c. Separate the upper and lower halves of the three support brackets by removing each cotter pin, washer and clevis pin.

d. Remove bolts and washers securing fuel cell access cover and attaching brackets. Remove brackets and position fuel shutoff control out of the way. Retain washers for installation.

e. Remove two fuel shutoff valve mounting screws and sealing washers (fig. 10-7). Valve will remain supported by attaching hoses.

f. Remove fuel cell access cover and large O-ring.

g. Disconnect fuel hoses at fuel shutoff valve and remove valve and O-ring.

10-47. Inspection — CR Fuel Shutoff Valve. *a.* Inspect fuel shutoff valve for cracks or excess corrosion. Replace valve if cracked, damaged or corroded.

b. Check that control lever arm operates freely from the open to closed position with a **PULL OF NOT MORE THAN 3.5 POUNDS.**

c. Ensure that the control lever arm seats firmly in detents at the full open and closed position.

d. Inspect for crossed or stripped threads. Replace a valve exhibiting such damage.

10-48. Installation — CR Fuel Shutoff Valve. *a.* Check that aft fuel hose passes through support loop inside top of tank; then connect hose to the fitting on the lever arm side of the fuel shutoff valve (fig. 10-7).

b. Connect fuel pump hose to opposite side of valve, being careful to loop the hose outboard from the pump and then aft to avoid interference with the tank unit float.

c. Lubricate a new large O-ring with petrolatum (C73) and install in oval shaped groove around fuel cell opening.

d. Position fuel cell access cover over cell opening. Attach fuel shutoff valve with new O-ring lubricated with petrolatum (C73), using two screws and sealing washers.

e. Install access cover bolts and washers, fuel control bracket, and three lower bracket halves as shown in figure 10-7. Use thin washers at bracket attachment bolts.

f. Attach three upper bracket halves to lower bracket halves with clevis pins, washers and cotter pins.

g. Connect shutoff valve control cable terminal with clevis pin and cotter pin. Verify that fuel shutoff valve rigging adjustment is correct (para 10-41).

h. Install tank unit (para 10-30).

i. Refill fuel cells and check fuel shutoff valve area for leakage by removing inspection access cover in the access door.

10-49. CR ELECTRIC (SUBMERGED) FUEL PUMP.

10-50. Description — CR Electric (Submerged) Fuel Pump. The 28 vdc electric, submerged-type fuel pump is installed on the inlet mounting pad inside the left fuel cell (fig. 10-8). The pump operates automatically during engine starts to provide a positive supply of fuel to the engine-driven fuel pump. Pump operation at times other than during engine starting is controlled by the FUEL PUMP-OFF toggle switch located on the electric control console. This switch may be used to operate the fuel pump to check for leakage or to bleed or prime the fuel system after repairs have been made. The fuel pump circuit is protected by the FUEL PUMP 7.5 amp circuit breaker.

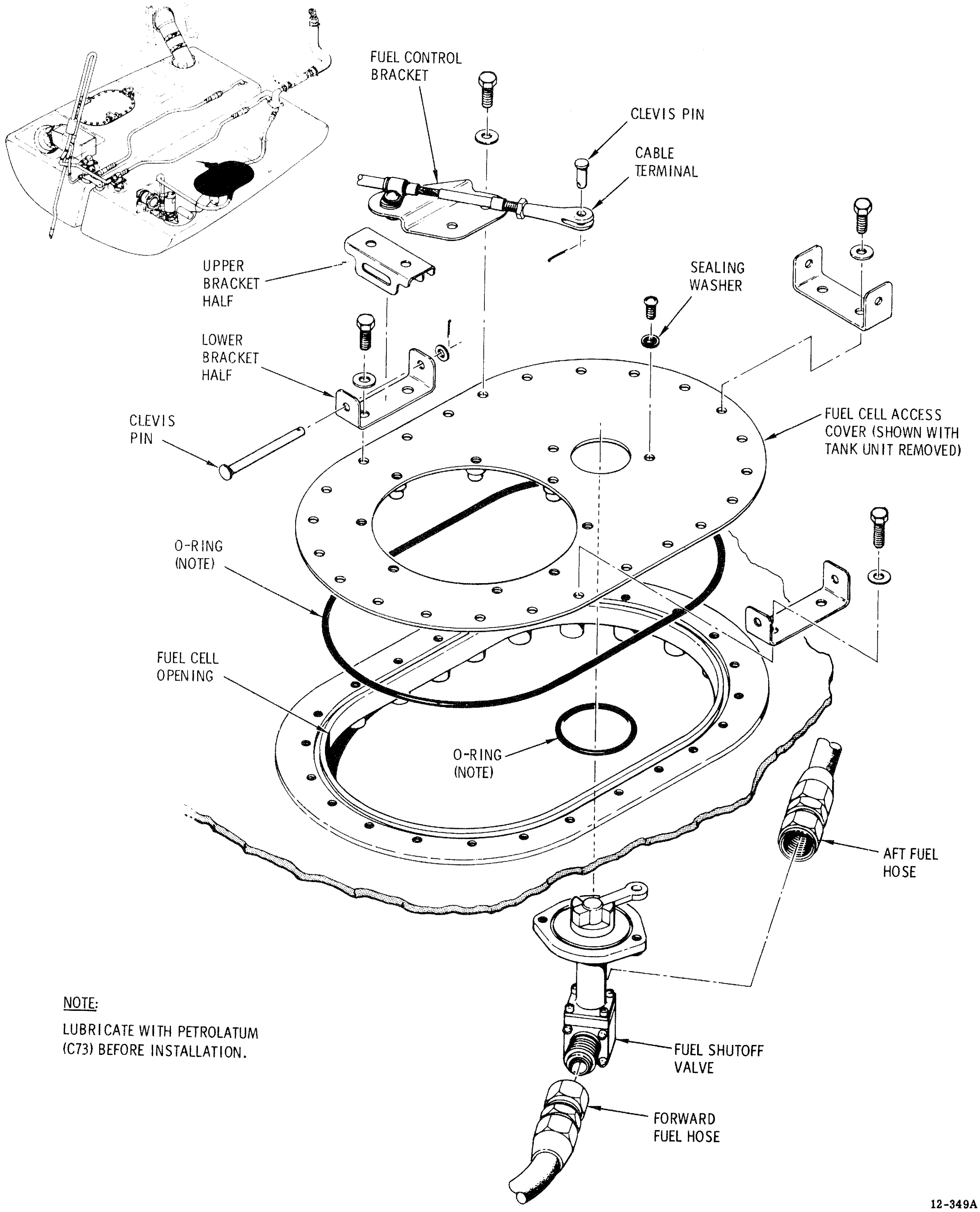


Figure 10-7. CR Fuel Shutoff Valve Installation.

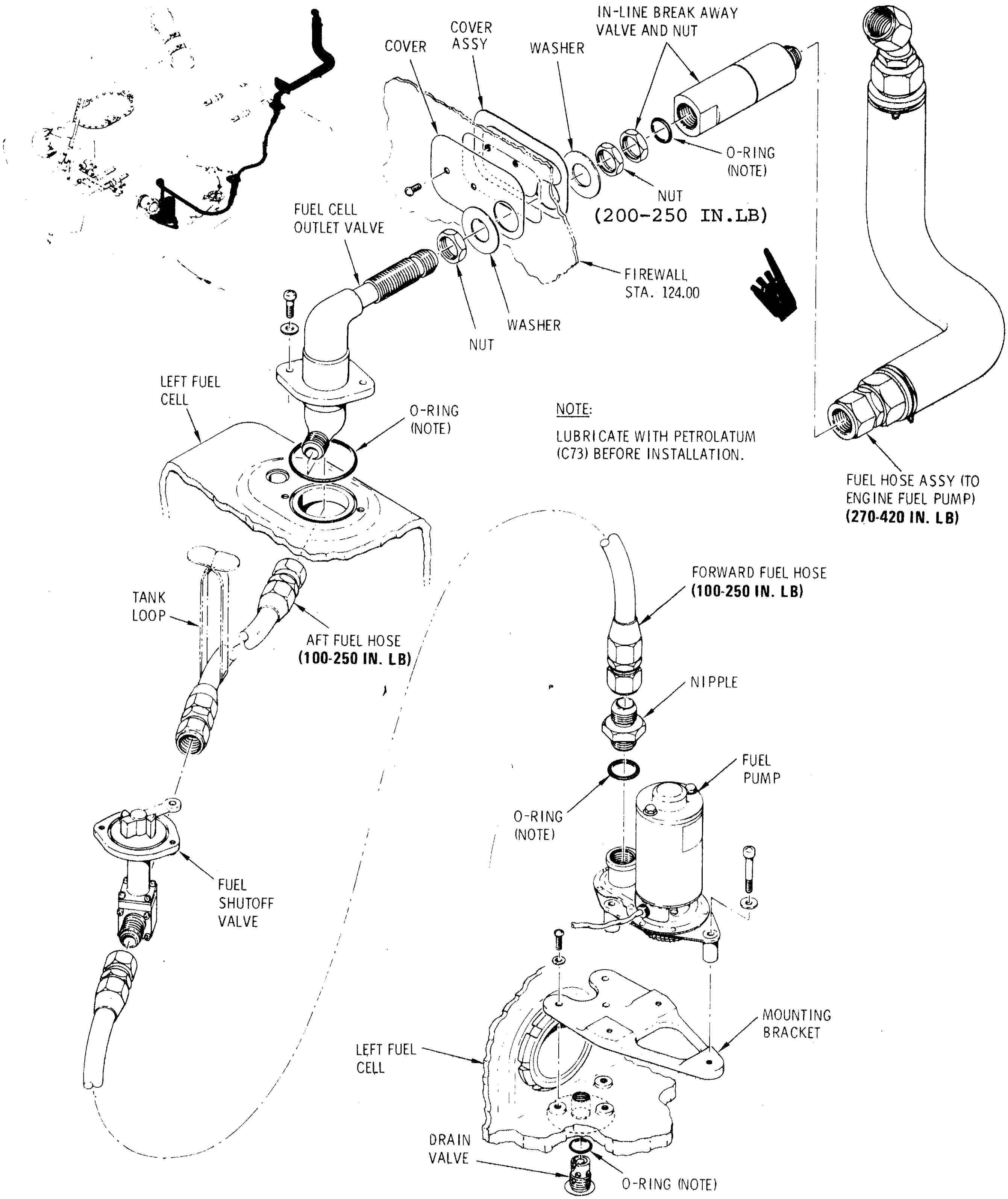


Figure 10-8. CR Fuel Supply System.

NOTE

The fuel pump operates when the engine starting circuit is energized. The FUEL PUMP-OFF switch will not prevent operation. Pull the FUEL PUMP circuit breaker when necessary to operate without fuel pump operation.

10-51. Removal — CR Electric (Submerged) Fuel Pump. (See fig. 10-8.) *a.* Remove tank unit according to paragraph 10-27, except do not remove tank unit from adapter.

b. Remove fuel cell access cover according to paragraph 10-46.

c. Reach through fuel cell access cover opening and disconnect forward fuel hose from pump.

d. Remove three screws and washers attaching pump to mounting bracket and remove pump.

e. Remove hose nipple and O-ring from pump. Nipple is used on replacement pump with new O-ring.

f. If required, remove three screws and washers and lift out mounting bracket.

10-52. Installation — CR Electric (Submerged) Fuel Pump. *a.* Position mounting bracket in left fuel cell as shown in figure 10-8 and install three attaching screws and washers.

b. Lubricate new O-ring with petrolatum (C73) and install with nipple on fuel pump boss.

c. Position pump on mounting bracket and install three attaching screws and washers.

d. Connect forward fuel hose to pump fitting being careful to loop the hose outboard from the pump and then aft to avoid later interference with the tank unit float. **TORQUE HOSE FITTING TO 100 - 250 INCH-POUNDS.**

e. Check that fuel cell is clean and free of foreign material.

f. Install access cover according to paragraph 10-48.

g. Install tank unit (para 10-30) and electrical harness receptacle.

h. Actuate fuel pump and check for proper operation.

i. Install access door and cover (para 10-20).

10-53. CR ENGINE FUEL SUPPLY HOSES AND VALVES.

10-54. Removal — CR Engine Fuel Supply Hoses and Valves. (See fig. 10-8.) *a.* Remove fuel shutoff valve according to paragraph 10-46.

b. In the engine compartment, disconnect fuel hose assembly at engine fuel pump and firewall in-line valve and remove hose.

c. On the engine compartment firewall, loosen in-line valve nut and remove in-line valve and O-ring; then remove nut.

d. In the cargo compartment, remove four screws and washers and fuel valve cover (chapter 2).

e. Remove two firewall cover assembly attachment screws.

f. In the engine compartment, remove nut, washer and cover assembly.

g. Inside the left fuel cell, disconnect aft fuel hose at outlet valve and remove hose.

h. Remove outlet valve attachment screws and washers.

i. Back-off nut and swivel the valve (with cover attached) back through the firewall and withdraw valve from the fuel cell and manipulate out through cargo floor opening. Remove O-ring.

j. Remove cover, washer and nut from the valve.

k. If required, disconnect and remove forward fuel hose from submerged fuel pump.

10-55. Installation — CR Engine Fuel Supply Hoses and Valves. *a.* Lubricate new O-ring with petrolatum (C73) and install in the left fuel cell outlet fitting groove.

b. Manipulate outlet valve and the cell as required to seat valve on the fuel cell fitting. Install two attaching screws and washers.

c. Install nut, washer and cover on firewall end of valve; then swivel the valve so that it extends through the firewall.

d. On the aft side of the firewall, install cover assembly, washer and nut. Do not tighten nuts.

e. Align the cover plates on the forward and aft sides of firewall and then install two screws to secure them in place.

f. At the forward side of the firewall, adjust nut on valve threads until it is finger-tight against washer and cover on the firewall.

g. At the aft side of the firewall, TORQUE NUT TO 200 - 250 **INCH-POUNDS** to clamp the assembly in the firewall. Check that valve body clears the cargo floor cutout.

h. Install in-line valve nut and run past O-ring groove on valve; then install lubricated O-ring.

i. Adjust in-line valve nut so that the O-ring will be centered in the outlet valve groove when the in-line valve is installed in the next step.

j. Install in-line valve; then tighten nut to seat O-ring.

k. Install hose assembly between engine fuel pump and in-line valve. Reposition and tighten hose connections as required and **TORQUE TO 270 - 420 INCH-POUNDS.**

l. Inside the left cell, pass aft fuel hose through loop in top of tank and connect to outlet valve. **TORQUE HOSE FITTING TO 100 - 250 INCH-POUNDS.**

m. Perform step *d* of paragraph 10-52 if forward hose assembly was removed.

n. Install fuel shutoff valve according to paragraph 10-48.

o. Install door and covers.

10-56. CR CROSSOVER FITTING.

10-57. Removal — CR Crossover Fitting. (See fig. 10-9.) *a.* Remove left and right fuel cell access doors (para 10-18).

b. With fuel pump and mounting bracket removed (para 10-49), hold right side crossover sleeve and remove left side nut and O-ring, using a suitable hook spanner wrench.

c. Withdraw crossover sleeve and remove O-ring.

10-58. Inspection — CR Crossover Fitting. *a.* Check crossover sleeve and nut for condition and corrosion.

b. Lubricate nut with petrolatum (C73) and run nut up to sleeve self-locking insert to check that nut runs freely on threads without binding.

10-59. Installation — CR Crossover Fitting. *a.* In the right cell, lubricate the crossover fitting threads and new O-ring with petrolatum (C73) and slip into position in the right cell.

b. In the left cell, lubricate new O-ring and nut threads with petrolatum (C73) and install O-ring on crossover sleeve.

c. Using suitable hook spanner wrenches, hold right side cross-over sleeve and **TORQUE NUT TO 150 — 250 INCH-POUNDS.**

d. Install fuel pump and mounting bracket (para 10-52).

e. Install fuel cell access doors and covers (para 10-20).

10-60. CR FUEL CELL VENT SYSTEM.

10-61. Description — CR Fuel Cell Vent System. Fuel cell vents (fig. 10-10) are located in the forward and aft top ends of each fuel cell. Each aft vent hose is routed forward and connected to a crossover manifold

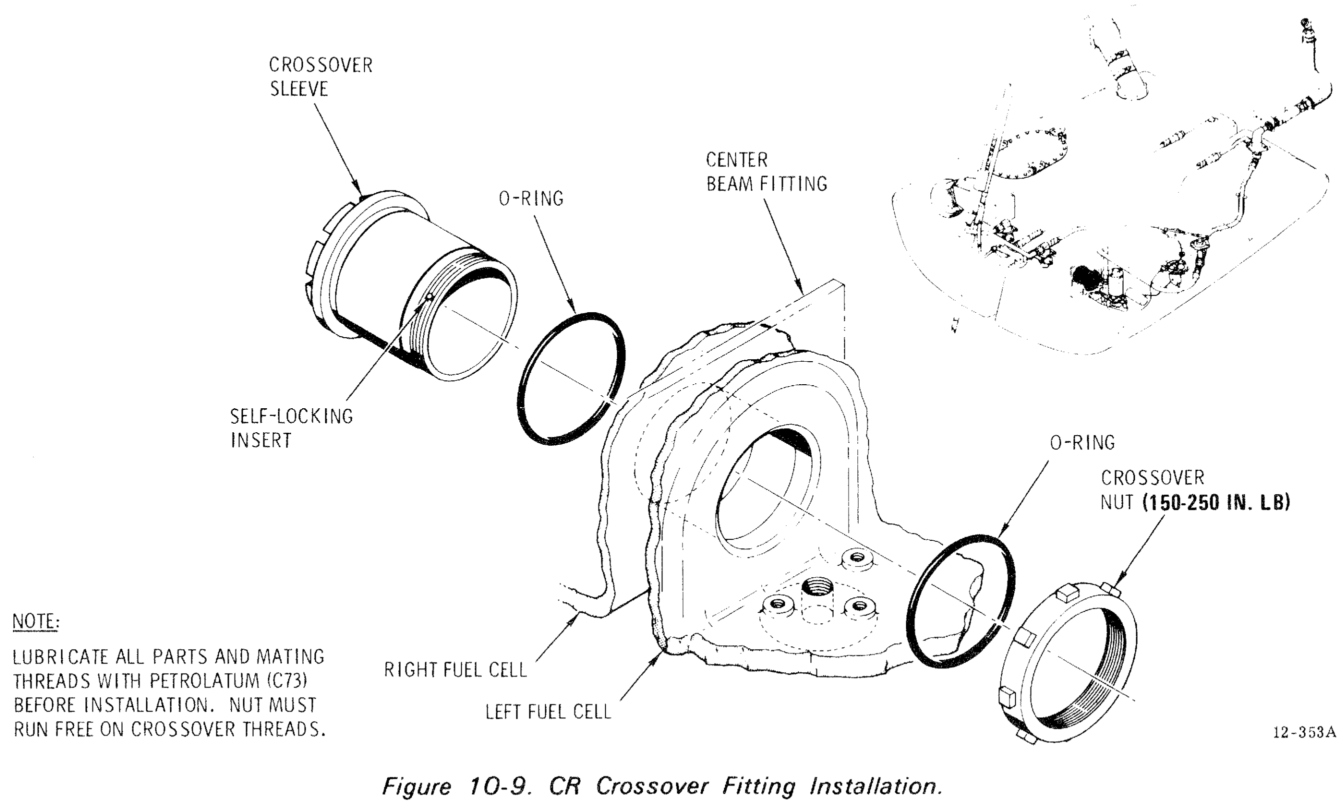


Figure 10-9. CR Crossover Fitting Installation.

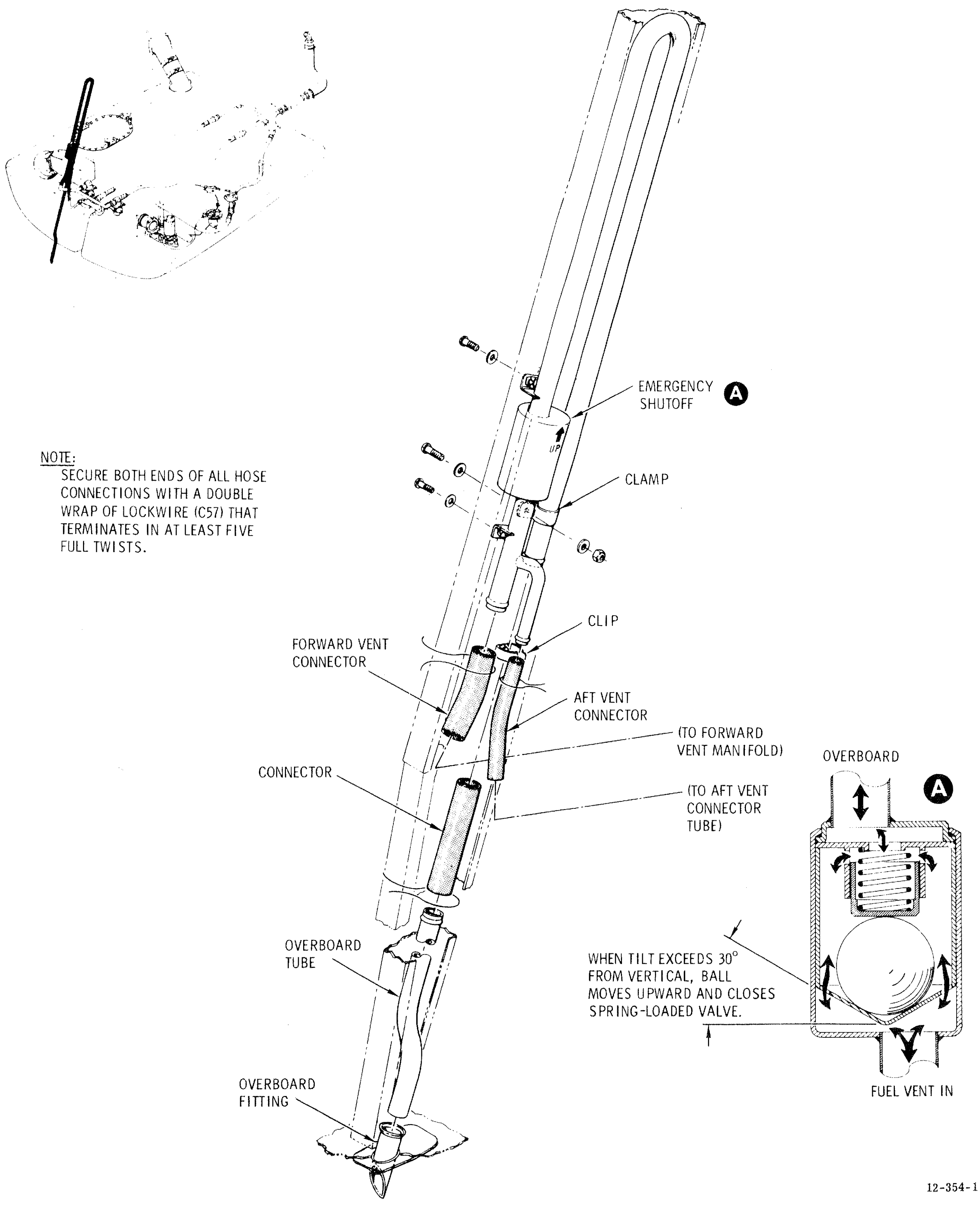


Figure 10-10. CR Vent System Installation (sheet 1 of 2).

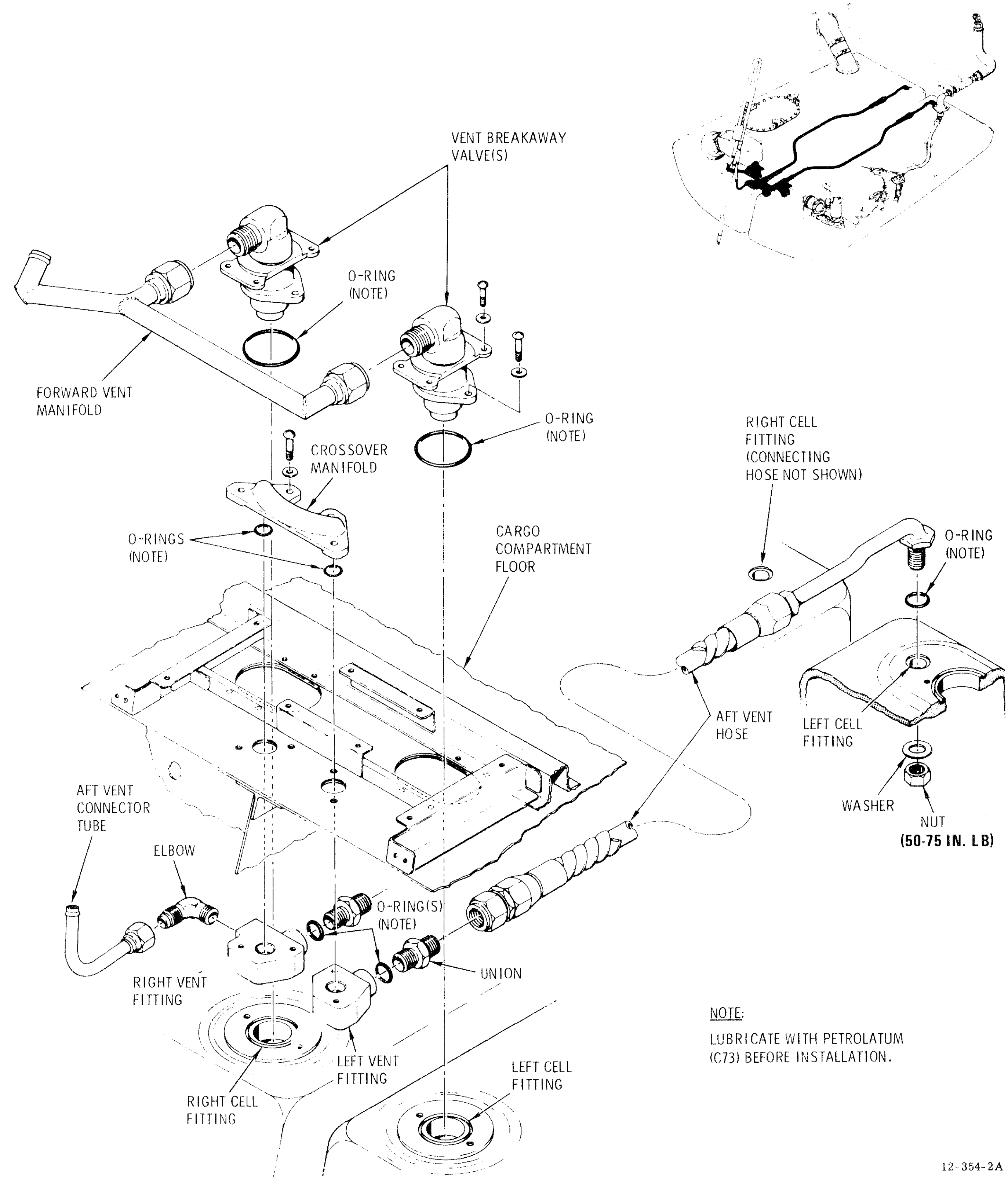


Figure 10-10. CR Vent System Installation (sheet 2 of 2).

that connects with the emergency shutoff valve system located in the cargo compartment forward bulkhead. The crossover manifold is attached to the cargo compartment floor structure. The aft vent hoses have a surplus length, allowing for any displacement or shifting of the fuel cells in the event of aircraft impact damage. The forward vents are connected directly to frangible (breakaway) fittings that are mounted on the cargo compartment floor structure. These fittings also connect to the emergency shutoff vent valve system. The vent system emergency shutoff valve remains open for all normal aircraft attitudes. During abnormal attitudes (such as a rollover accident) the vent valve automatically closes to prevent loss of fuel. From the vent shut off valve, the vent tube is routed downward, through the control tunnel, to the underside of the aircraft.

10-62. Inspection — CR Fuel Cell Vent System (Installed). *a.* Remove controls access door and cargo floor covers (chapter 2).

b. Check that overboard vent is open, not clogged, and secure.

c. Check emergency shutoff valve tubes for dents, cracks, or corrosion. Minor (or smooth) dents are permissible.

d. Check hoses for cracks or deterioration, and lockwire for security.

e. Check fuel vent line emergency shutoff valve for cracks, dents, or corrosion.

f. Check forward vent fittings and manifold for fuel leaks, and security.

g. Check visible portions of aft vent system for fuel leaks and security.

10-63. Removal — Forward CR Fuel Cell Vent System. (See fig. 10-10.) *a.* Prepare aircraft for maintenance according to paragraph 10-3.

b. Remove right foot support fairing, forward vent system cover, controls access door and pilot's seatback (chapter 2).

c. Remove lockwire at upper end of aft vent connector hose.

d. Remove lockwire at forward vent upper hose connection.

e. Remove attaching hardware at three places on station 78.50 bulkhead as shown in figure 10-10, sheet 1.

f. Remove lockwire and remove overboard tube.

g. Remove forward emergency shutoff valve system.

NOTE

After disconnecting the vent system tubing and attaching hardware, removal is best accomplished with the collective pitch stick in the full down position. Then rotate the vent valve assembly inboard and slide down over landing gear structure. This positions the assembly for removal through the controls access door opening.

10-64. Removal — Aft CR Fuel Cell Vent System. (See fig. 10-10, sh 2.)

NOTE

If only right or left side vent removal is required, disregard instructions for the opposite side.

a. Remove left and right fuel cell access doors (para 10-18) and forward vent access and fuel valve access covers (chapter 2).

b. Remove left and right fuel cell access covers (para 10-46).

NOTE

For access to work areas in the following steps, compress the cell by inserting wood blocks between the top of the cell and underside of floor structure.

c. In the fuel cell, remove nut and washer securing the aft vent hose end fitting. Depress the fuel cell, pull out end fitting and remove O-ring.

d. Disconnect and remove forward vent manifold.

e. Remove four screws attaching forward vent breakaway valve to cargo floor structure and two screws attaching valve to fuel cell fitting. Remove valve and O-ring.

f. Remove four screws and remove rear vent crossover manifold with two O-rings.

g. Above top left fuel cell, remove left vent fitting with hose attached; then disconnect hose.

h. Separate union from vent fitting and discard O-ring.

NOTE

Removal of the right vent fitting with attaching tube requires removal of the right fuel cell (para 10-1) to gain access to the aft vent connector tube nut.

i. Above top of right fuel cell (cell removed) slide assembled right vent fitting, elbow and aft vent connector tube out of cell cavity structure with hose attached; then disconnect hose. Do not disassemble unless parts replacement is required. Disassembly will require realignment of the elbow and tube with the forward vent system during installation.

10-65. Inspection — CR Fuel Cell Vent System. *a.* Blow gently into the overboard tube opening to determine that emergency shutoff valve is open when tilted in any direction within 30 degrees off vertical.

b. Check that shutoff valve closes when tilted in any direction more than 30 degrees from vertical.

c. Check nutplates on vent shutoff valve mounting brackets for security and damaged threads.

d. Check crossover manifolds for cracks, dents, or corrosion. Replace manifold if cracked or dented.

e. Check that breakaway valves are open and in serviceable condition.

10-66. Installation — Forward CR Fuel Cell Vent System. (See fig. 10-10, sh 1.) *a.* Position overboard tube in the scarfed fuselage opening and snap upper end into bulkhead clip.

b. Install vent connector hose, if removed.

c. Loosely install vent emergency shutoff valve assembly with attaching hardware shown in figure 10-10, sheet 1.

d. Secure each end of hoses with a double wrap of 0.032-inch lockwire (C57) that terminates in at least five full twists.

e. Install forward and aft vent connector hoses.

NOTE

If connector hoses do not align, the top and bottom vent system attachment screw holes in the bulkhead channel may be elongated to assist in alignment.

f. Tighten vent system attachment screws.

g. Check that complete installation does not interfere with free movement of flight controls.

h. Install access doors, covers and pilot's seatback.

10-67. Installation — Aft CR Fuel Cell Vent System. (See fig. 10-10, sh 2.) *a.* Lubricate O-rings with petrolatum (C73) and install with unions on left and right vent fittings.

b. Connect aft vent hoses to vent fittings and tighten in approximate position for installation. Hose fittings cannot be tightened with fuel cells installed.

NOTE

For installation of right vent fitting, the right fuel cell must be removed. The forward emergency vent must also be installed so that correct tubing angle may be established.

c. Install elbow on right fitting as shown in figure 10-10, sheet 2.

d. Secure connector tube loosely to the right vent fitting.

e. Check that the grommet is installed in corner bulkhead hole of upper right fuel cell compartment. Insert tube through hole and position fitting for attachment under the cargo floor.

NOTE

In the next step, some trial and error fitting will be required to align the connector tube at the proper angle with the connecting vent hose.

f. Align tube with the forward vent system hose connection and tighten the tube nut.

g. Position left vent fitting and install crossover manifold with two lubricated O-rings above cargo floor. Install four attaching screws and washers.

h. Route hoses rearward and install with lubricated O-rings through left and right cell fittings.

i. In the fuel cell, install washer and nut on each hose terminal and **TORQUE NUT TO 50 — 75 INCH-POUNDS.**

CAUTION

When installing and connecting vent valves, use care to prevent stressing the undercut breakaway area.

j. Install lubricated O-rings on vent breakaway valves and position through cargo floor into fuel cell fittings. Some manipulation of the fuel cell may be required to align holes.

k. Attach lower vent breakaway valve flange to fuel cell with two screws and washers and upper flange to cargo floor with four screws and washers.

l. Install forward vent manifold between the two valves and forward vent system hose connection. Use a second wrench on the valve flats to prevent movement of valves.

m. Install access doors and covers.

10-68. CR CLOSED CIRCUIT FUEL RECEIVER.

10-69. Description — CR Closed Circuit Fuel Receiver. The closed-circuit filler cap (fig. 10-11) is mounted flush just below the forward door sill of the right cargo compartment door. Closed-circuit fueling is used only with compatible refueling hose connections. The closed-circuit fuel receiver is bolted to the right fuel cell. A frangible (breakaway) retainer and collar assembly attaches the unit to the aircraft structure. The closed circuit receiver accepts the mating fuel nozzle and opens only when there is sufficient delivered fuel pressure to unseat a spring-loaded, self-closing valve. A series of internal diaphrahms regulates the rate of fuel passing through the receiver into the aircraft fuel system so that air pressure buildup does not exceed vent system venting capacity. When the cells are almost full, two receiver floats rise with the level of fuel to shut off the flow of fuel at the full point. One float raises slightly ahead of the other to allow for a two-stage shutoff. When cells are almost full, a normal on-off cycling of the delivery system will occur due to the action of surging fuel against the float system shutoff mechanism.

CAUTION

When reinstalling closed circuit fuel cap, be sure that the cap retention cable is coiled and positioned inside the receiver well so that no interference (with locking mechanism and sealing ring) occurs when the cap is installed.

10-70. Operational Check — CR Closed Circuit Fuel Receiver. *a.* Refuel the aircraft using the pressure refueling method.

b. Check that the fuel cells fill in approximately one minute (dependent upon delivery pressure) and that the receiver shuts off automatically when the cells are full.

NOTE

Fuel sloshing may cause the receiver floats to bounce, causing momentary on-off cycling of the delivery system.

10-71. Removal — CR Closed Circuit Fuel Receiver. (See fig. 10-11.) *a.* Prepare aircraft for maintenance according to paragraph 10-3.

b. Remove four bolts and washers at corners of collar assembly and remove assembled collar. Loosen collar bolts if difficult to remove.

c. Remove eight bolts and washers attaching retainer to aircraft support structure and remove retainer with refueling receiver attached. Some manipulation may be required to withdraw the receiver from the fuel cell.

d. Remove three screws and washers and retainer from the receiver.

e. Remove receiver O-ring.

f. Disassembly of the collar assembly is not required.

10-72. Inspection — CR Closed Circuit Fuel Receiver. *a.* Check receiver structural support assembly for damage, corrosion and galling of flanges and webs.

b. Check retainer for condition, cracks, or scoring of grooved breakaway tabs. Replace retainer if damaged.

c. Check receiver for condition and corrosion, floats for freedom, and fuel nozzle receptacle for scoring or damage.

10-73. Repair — CR Closed Circuit Fuel Receiver. Repair scoring and nicks in the cylindrical fuel nozzle opening by polishing out damage. Treat all repaired surfaces with chemical film (C20).

10-74. Installation — CR Closed Circuit Fuel Receiver. (See fig. 10-11.) *a.* Attach retainer to outboard side of receiver flange with three screws and washers. Grooves on retainer breakaway tabs are inboard.

b. Check that receiver fitting in fuel cell is aligned with the fitting in structure. Some manipulation of the cell may be required.

c. Lubricate the receiver flange and adjacent surfaces with petrolatum (C3) and install a new lubricated O-ring; then slide the receiver into position in the right cell from the aircraft exterior.

d. Start the eight receiver attaching bolts and washers and then tighten down evenly, using care to avoid cocking the receiver and damaging the O-ring. **TORQUE THE BOLTS TO 50 — 70 INCH-POUNDS.**

e. Check that all right collar bolts are slightly loose on collar assembly. If collars have been removed, reinstall as shown in figure 10-11. Note that chamfered

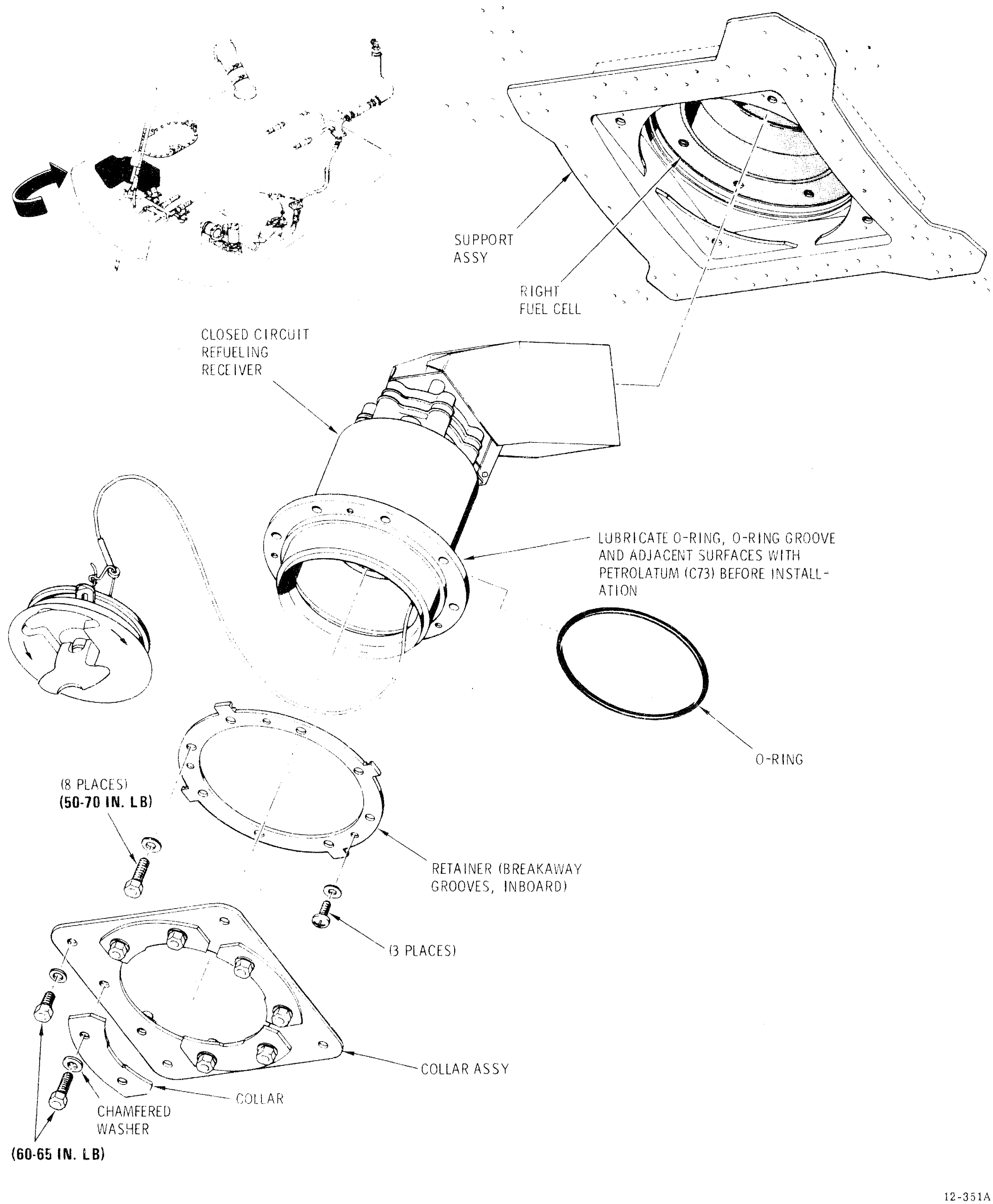


Figure 10-11. CR Closed Circuit Fuel Receiver Installation.

washers are used under collar bolts, with chamfered sides of washers against bolt heads.

f. Align the collar assembly over the receiver neck and install four corner bolts and washers. **TORQUE THE CORNER BOLTS TO 60 — 65 INCH-POUNDS.**

g. After each collar is tightly aligned around receiver neck, **TORQUE FOUR BOLT PAIRS TO 60 — 65 INCH-POUNDS.**

10-75. CR GRAVITY FILL VALVE.

10-76. Description — CR Gravity Fill Valve. The gravity filler cap (fig. 10-12) is mounted flush with the fuselage skin, aft of the right cargo door. The filler cap is attached to the filler neck by a safety chain. Pulling outward on the filler cap and hooking the cable in a notched tab inside the filler opening opens a normally closed spring loaded flapper valve in the top of the fuel cell. The flapper valve is part of a breakaway fitting attached to the fuel cell that will break away from the airframe in its normally closed position, thus retaining the fuel. Because the valve seals off the fuel cell opening below the external filler cap, the LONG-RANGE TANK CONN fitting originally provided for range extending torso fuel tanks is not useable.

10-77. Inspection — CR Gravity Fill Valve. *a.* With cap removed, pull chain and observe that flapper valve at bottom of fuel neck is open. Check that valve closes when chain is released.

b. Check for positive seal by pouring a small quantity of fuel into fuel neck with valve closed. Fuel should remain trapped at bottom of filler neck or drain very slowly into fuel cell.

10-78. Removal — CR Gravity Fill Valve. (See fig. 10-12.) *a.* Prepare aircraft for maintenance according to paragraph 10-3.

b. Remove right fuel cell access door (para 10-18).

c. Remove right fuel cell access cover (para 10-46).

d. Remove fuel filler shield (chapter 2).

e. Release clasp and remove filler cap from filler pull cable.

f. Remove hose clamps and slide hose upward to clear valve neck.

g. In the fuel cell, remove eight valve attachment cap screws and washers.

h. Remove filler valve and O-ring; then remove hose.

10-79. Installation — CR Gravity Fill Valve. (See fig. 10-12.) *a.* Position hose up over aircraft filler assembly neck. Roll lower end of hose back over upper end of hose to form a cuff and temporarily prevent hose from extending beyond lower end of filler assembly.

b. Lubricate O-ring with petrolatum (C73) and install in right fuel cell fitting groove.

c. Position gravity fill valve and install eight cap screws with reduced OD washers.

d. Reposition hose over both filler necks and secure with two clamps as shown in figure 10-12. Clamp nuts must be secured close to cargo floor opening to clear the filler shield when installed.

e. Connect filler cap with gravity fill valve pull cable.

f. Check that fill valve flapper valve is closed and that valve opens when cable is pulled.

g. Install access doors and covers.

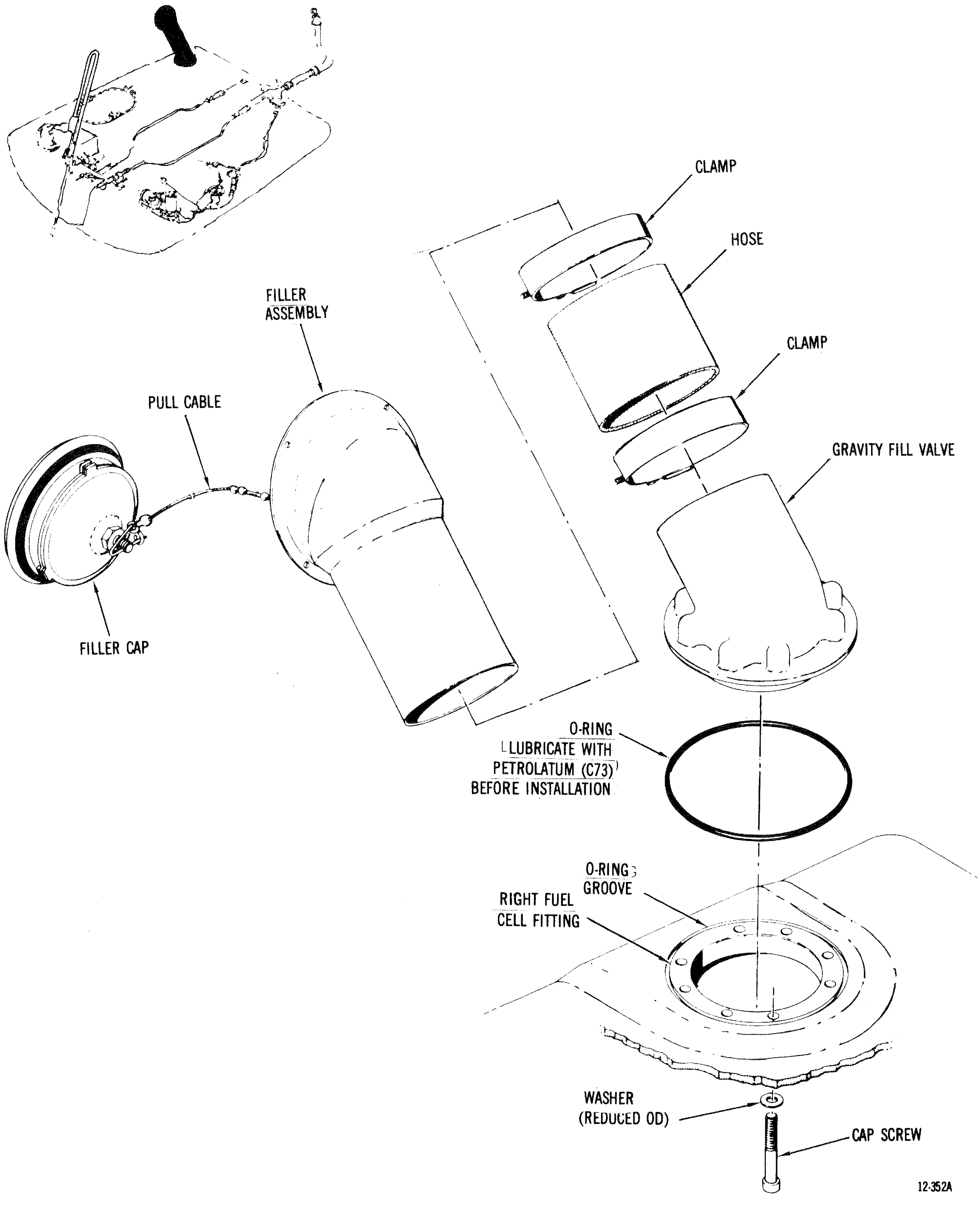


Figure 10-12. CR Gravity Filler Installation

Pages 10-30 through 10-47 are deleted in their entirety, including Figures 10-13 through 10-20 and Tables 10-5 and 10-6. This includes paragraphs 10-80 through 10-149.

CHAPTER 11

FLIGHT CONTROLS

SECTION I CONTROL SURFACES

(Not Applicable)

SECTION II FLIGHT CONTROLS

11-1. MAIN ROTOR CONTROL SYSTEM.

11-2. General — Main Rotor Control System. The main rotor control system (fig. 11-1) consists of pilot's collective controls, pilot's cyclic controls, and the push rod linkage that transfers control movements to the main rotor swashplate and mixer controls.

11-3. Troubleshooting — Main Rotor Control System. Troubleshooting information is divided into: (1) investigation of operational vibration problems originating with the main rotor assembly (table 5-1); (2) investigation of symptoms that can be recognized (table 11-2); and (3) isolation of an unusual controls malfunction (table 11-3). First determine which of the four major installations is defective. Isolate each linkage installation from the others until the area in which the malfunction is occurring has been located. Then investigate and locate the malfunction and repair or replace the defective component. Since the first indication of trouble will appear at the cyclic control stick or the collective pitch stick, the isolation procedures in tables 11-2 and 11-3 begin with symptoms detected during operation of these controls.

Table 11-1. Premaintenance Requirements for Rigging Main Rotor Control System.

Conditions	Requirements
Special Tools	(T19) (T20) (T21) (T22)

11-4. MAIN ROTOR CONTROLS RIGGING.

11-5. General — Main Rotor Controls Rigging. Rigging of the main rotor controls system must be accomplished immediately after replacement of linkage that cannot be accurately measured (by trammeling, etc) before it is installed in the main rotor control system, or if aircraft operation reveals a rigging deficiency.

NOTE

The cyclic and collective controls must be rigged, in sequence, starting with the collective. Control rod end bearing adjustments are to be made to the nearest half turn that will produce correct rigging. When tightening the jam nut at the adjustable end of control rods, always hold the rod end with a wrench to prevent jamming of the bearing.

11-6. Rigging — Main Rotor Collective Control. (See fig. 11-2.) *a.* Remove main rotor blades (chapter 5), or manually raise the blades off their droop stops by simultaneously lifting the blade tips. All four blades must be off the stops during rigging. Padded supports should be used, if available.

b. Release the friction and raise pilot's collective stick. Position collective rigging fixture (T19) in outboard lower end of slot in collective friction guide link. (See detail A, sh 1.)

c. Lower collective stick on rigging fixture and tighten friction grip (drive gear).

d. Position mixer rigging plate (T22) on mast base and secure firmly by use of the two toggle clamps.

e. Measure the distance from horizontal centerline of the mast support bracket hingline bolt (attaching the longitudinal idler bellcrank and collective pitch mixer bellcrank to the mixer support bracket) to the surface of mixer rigging plate. Record actual distance. (See detail B, sh 1.)

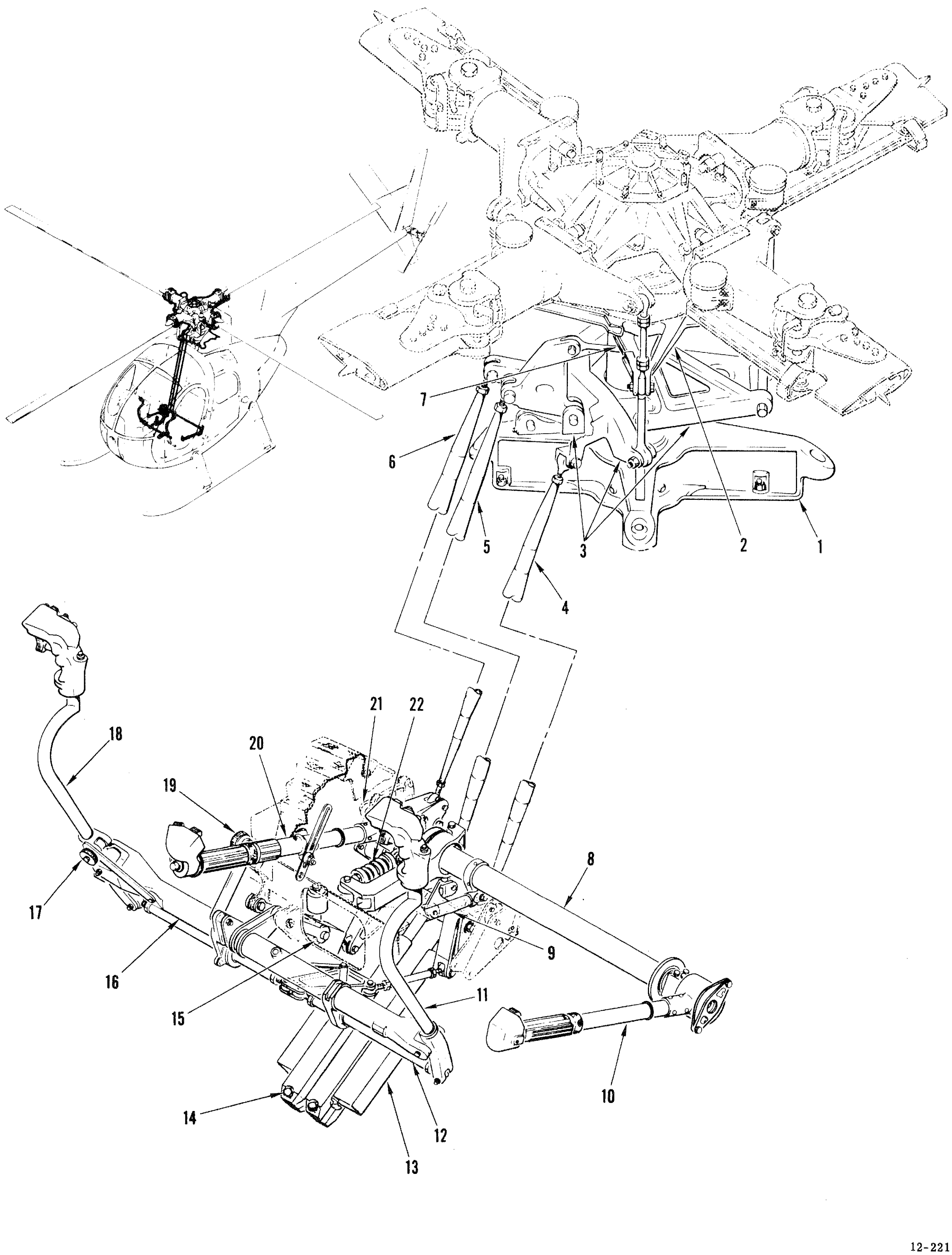


Figure 11-1. Main Rotor Control System.

Key to Figure 11-1.

1. Main rotor mast base
2. Rotating swashplate
3. Main rotor controls (mixer, idler bellcranks and links)
4. Lateral mixer control rod (cyclic)
5. Longitudinal mixer control rod (cyclic)
6. Collective pitch control rod
7. Stationary swashplate
8. Collective control torque tube
9. Droop control override link
10. Copilot's collective pitch stick
11. Copilot's cyclic stick
12. Cyclic lateral control rod
13. Lateral cyclic trim actuator
14. Longitudinal cyclic trim
15. One-way lock (longitudinal cyclic control)
16. Cyclic control torque tube
17. Lateral cyclic friction knob
18. Pilot's cyclic stick
19. Longitudinal cyclic friction knob
20. Pilot's collective pitch stick
21. Gas producer control rod
22. Collective bungee

f. Measure the distance from the horizontal centerline of the bolt that attaches the collective pitch mixer bellcrank (detail C, sh 1), to the surface of mixer rigging plate; it should be the same as the dimension recorded in *e* above. If not, loosen checknut, disconnect upper end of collective control rod, and adjust rod end until dimensions are the same.

g. When adjustment is complete, tighten checknut. Make certain that length of control rod is not changed.

CAUTION

When tightening rod end checknuts, ensure that rod end bearings at both ends of rod are aligned between the bellcrank ears.

h. Reconnect collective pitch control rod and secure with a new cotter pin.

i. Leave collective rigging fixture installed. All mixer rigging is done with collective in midposition.

NOTE

Loosen lateral and longitudinal friction devices.

11-7. Rigging — Main Rotor Cyclic Control. (See fig. 11-2.)

CAUTION

When timing the longitudinal trim motor for full travel in the next step, do not continue to hold the trim switch after the actuator has reached the travel limit.

NOTE

Timing trim travel should not be necessary if the blades are removed. The no-load trim point can be easily located by moving the cyclic stick.

a. Loosen longitudinal friction knob. Actuate longitudinal cyclic trim for no load by accurately timing trim motor from full aft to full forward trim. Return trim to full aft; then trim forward for one-third the travel time. Position cyclic stick so that longitudinal rigging fixture (T21) will fit in outboard side of slot in longitudinal friction guide link. Bushing end of fixture butts against upper end of the slot, and friction knob stud fits in fixture slot. (See detail D, sh 2.)

b. Secure fixture hook in lower end of slot, tighten knurled thumbnut to seat bushing at upper end, and tighten friction knob.

c. Measure the distance from the horizontal centerline of bolt at aft end of longitudinal pitch mixer bellcrank (detail E, sh 2) to the surface of mixer rigging plate; it should be the same as the dimension recorded in step *e* of paragraph 11-6. If not, loosen checknut, disconnect longitudinal pitch control rod, and adjust rod end until dimensions are the same.

d. When adjustment is complete, tighten checknut and make certain that length of control rod does not change.

e. Reconnect control rod to longitudinal pitch idler and secure with a new cotter pin.

f. Leave longitudinal rigging fixture installed and continue with lateral rigging.

g. Loosen friction knob and actuate lateral cyclic trim for no load. Position cyclic stick so that lateral rigging fixture (T20) will fit in forward side of slot in lateral friction guide link. Bushing end of fixture butts against outboard end of slot, and friction knob stud fits in fixture slot. (See detail F, sh 2.)

h. Secure fixture hook in inboard end of slot, tighten knurled thumbnut to seat bushing at outboard end, and tighten friction knob.

i. Measure the distance from the horizontal centerline of bolt connecting lateral bellcrank to stationary swashplate mixer link (detail G, sh 2) to the surface of mixer rigging plate; it should be the same as the dimension recorded in step *e* of paragraph 11-6. If not,

Table 11-2. Troubleshooting of Cyclic and Collective Controls.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Rapid continuous beat felt throughout aircraft (four per revolution). (Do not confuse with tail rotor vibrations felt through the tail rotor pedals.)**

STEP 1. Check for rotor blade out of track (para 5-43).

If the rotor blade(s) are found to be out of track, readjust as required (para 5-43).

2. **Cyclic control stick pressures cannot be reduced by operating trim switch.**

STEP 1. Check for an improperly set cyclic trim.

If the cyclic control stick is found to be out-of-trim, readjust as required (para 11-131).

STEP 2. Check for a defective or inoperative trim motor.

NOTE: Trim change is relatively slow.

If the trim motor is found to be defective or inoperative, troubleshoot the cyclic trim electrical circuit (para 11-131); replace defective component(s).

3. **Cyclic control sticks have a tendency to move to the aft position.**

STEP 1. Check for low fluid level in the one-way lock fluid reservoir (para 11-123).

If the fluid is low, service the reservoir (chapter 1).

4. **Inadequate cyclic control during flight operations.**

STEP 1. Check aircraft weight and balance per TM 55-1520-214-10.

If the center of gravity (CG) moments are found to be out of the limits specified in TM 55-1520-214-10, check the aircraft for improper loading.

5. **Excessive pressure required for lateral movement of cyclic control sticks on the ground and during flight.**

NOTE

Some droop stop friction will be felt when there is no rotor rpm.

STEP 1. Check for improperly adjusted lateral friction device (para 11-108).

If the lateral friction device is improperly adjusted, readjust as required (para 11-108).

STEP 2. Check for binding or frozen scissors crank or hub lower shoe bearings (para 11-17).

If the bearings are found to be defective, replace the defective bearing(s).

STEP 3. Check swashplate spherical bearing assembly surface for damage (para 11-24).

If the swashplate spherical bearing assembly fails to meet the torque requirements of paragraph 11-28, replace the bearing assembly.

STEP 4. Check for binding of the lateral interconnection rod end bearing due to incorrect alignment (para 11-113).

If the lateral interconnection rod end bearing is out-of-alignment, realign the rod end bearing (para 11-113).

Table 11-2. Troubleshooting of Cyclic and Collective Controls. (cont)

STEP 5. Check for damaged droop stop striker plate(s) or cam roller(s) (para 5-19).

If components are found to be damaged, replace the damaged component(s) (para 5-22).

6. **Lateral feedback in cyclic control stick (no longitudinal feedback detected).**

STEP 1. Check for out-of-track main rotor blades (para 5-43).

If the main rotor blades are found to be out-of-track, readjust blade track as required (para 5-43).

7. **Excessive pressure required for longitudinal movement of cyclic control sticks on the ground and during flight.**

NOTE

Some droop stop friction will be felt when there is no rotor rpm.

STEP 1. Check for improperly adjusted lateral friction device (para 11-108).

If the lateral friction is improperly adjusted, readjust as required (para 11-108).

STEP 2. Check for binding longitudinal push rod end bearings.

If the longitudinal push rod is out of alignment, realign the rod end bearings.

STEP 3. Check that the one-way lock check valve and push rod shaft components are within wear limits (para 11-123).

If wear exceeds the limits specified in paragraph 11-123, replace the one-way lock assembly.

STEP 4. Same as STEPS 2 and 3 of 5 above.

8. **Binding Collective.**

STEP 1. Swashplate spherical bearing teflon liner galled (para 11-24).

If the swashplate spherical bearing teflon liner is galled, remove mixer and pitch control linkage; check swashplate assembly sliding movement on mast for smoothness. (Maximum spherical bearing teflon liner-to-mast sliding force with swashplate disassembled is 20 lb.)

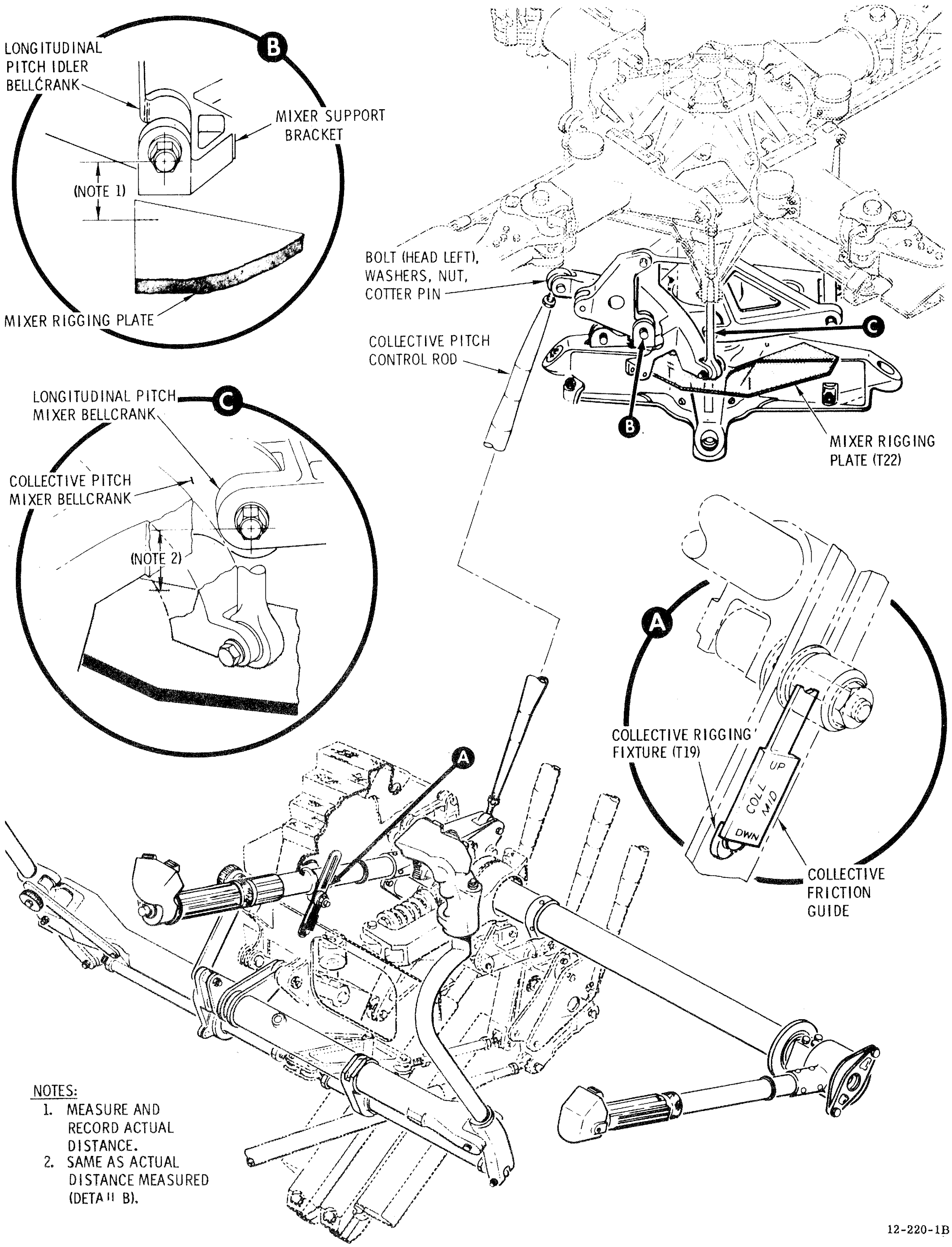


Figure 11-2. Main Rotor Control System Rigging. (sheet 1 of 2).

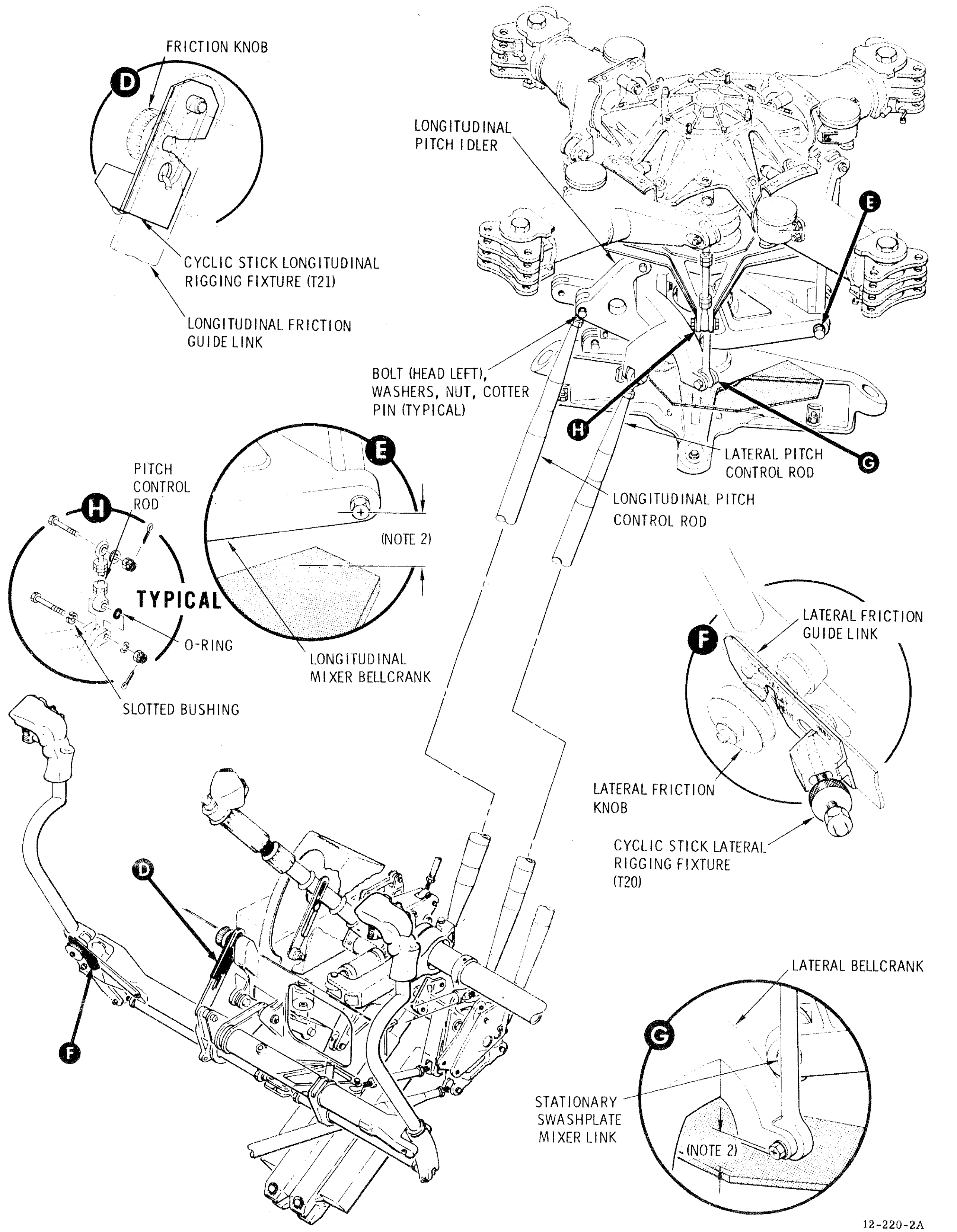


Figure 11-2. Main Rotor Control System Rigging. (sheet 2 of 2).

Table 11-3. Isolating Cyclic and Collective Control System Troubles.

Symptom	Isolating Step	Corrective Action
Binding, locking-up, and erratic action of cyclic control stick.	Disconnect one-way lock (15, fig. 11-1).	If symptom gone, replace one-way lock.
Symptom remains	Disconnect upper ends of longitudinal and lateral control rods (4 and 5, fig. 11-1).	If symptom gone and rod ends are good, disassemble and inspect main rotor mixer controls until defective part is located.
Symptom remains	Reconnect upper ends of longitudinal and lateral control rods, and disconnect lower ends.	If symptom gone and rod ends are good, disassemble and inspect pilot's compartment controls until defective part is located.
Trouble corrected (Reconnect all control linkage)		
Binding, locking-up, and èrratic action of collective control stick.	**CAUTION** Before disconnecting the collective pitch control rod, install collective bungee installation tool (T27) to prevent spring reaction on the control rod.	
	Disconnect upper end of collective pitch control rod (6, fig. 11-1) and operate stick.	If symptom gone, and rod end is good, disassemble and inspect main rotor mixer controls until defective part is located.
Symptom remains	Leave upper end of collective pitch control rod disconnected.	Trouble is in pilot's compartment collective control linkage.
Symptom remains	**WARNING** Do not disconnect any hardware from the pilot's collective stick unless collective bungee installation tool is installed. Strong bungee spring pressure can cause linkage reaction and severe personal injury, or part damage.	
	Disconnect gas producer control rod (21, fig. 11-1) from collective stick.	If symptom gone, trouble is in engine controls. (Refer to chapter 4.)
Symptom remains	Disconnect droop control override link (9, fig. 11-1) from collective controls torque tube.	

Table 11-3. Isolating Cyclic and Collective Control System Troubles. (cont)

Symptom	Isolating Step	Corrective Action
Symptom remains	Leave gas producer control rod and droop control override link disconnected.	**NOTE** *Override link starts to compress with maximum N2 trim and collective stick raised one-third of travel. Disassemble and inspect pilot compartment collective control linkage until defective part is located.*
Trouble corrected (Reconnect all control linkage)		

loosen checknut, disconnect lateral pitch control rod, and adjust rod end until dimensions are the same.

j. When adjustment is complete, tighten checknut, and make certain that length of control rod is not changed.

k. Reconnect control rod to lateral bellcrank and secure with a new cotter pin.

l. Loosen friction control, friction knobs, knurled thumbnut and hooks of rigging fixtures and remove all three fixtures.

NOTE

Steps m and n below establish initial adjustment of the main rotor pitch control rods (detail H, sh 2). Tower-tracked main rotor blades can be identified by a decal located outboard of the blade root fitting on the lower side of the blade.

m. On aircraft without tower-tracked blades, initial adjustment of the pitch control rods will be as follows:

(1) Set two pitch control rods at 6.35 inches (centerline to centerline of rod bearings) and install on two opposite blades attached to hub lower strap pack.

NOTE

To prevent damage to the main rotor hub pitch housing caused by jam nut contact, ensure that at least four threads are exposed at the upper pitch control rod end bearings.

(2) Set remaining two control rods at 6.60 inches (centerline to centerline of rod bearings) and install on two opposite blades attached to hub upper strap pack.

n. On aircraft with tower-tracked blades, initial adjustment of the pitch control rods will be as follows:

(1) On blades attached to hub lower strap pack, set individual pitch control rod (centerline to centerline of rod bearings) to the lower strap pack length specified on tower-tracking decal of mating blade.

(2) On blades attached to upper strap pack, set individual pitch control rod (centerline to centerline of rod bearings) to the upper strap pack length specified on tower-tracking decal of mating blade.

o. Install the pitch control rods.

p. Install main rotor blades (chapter 5) or remove supports and simultaneously lower all four blades to normal position.

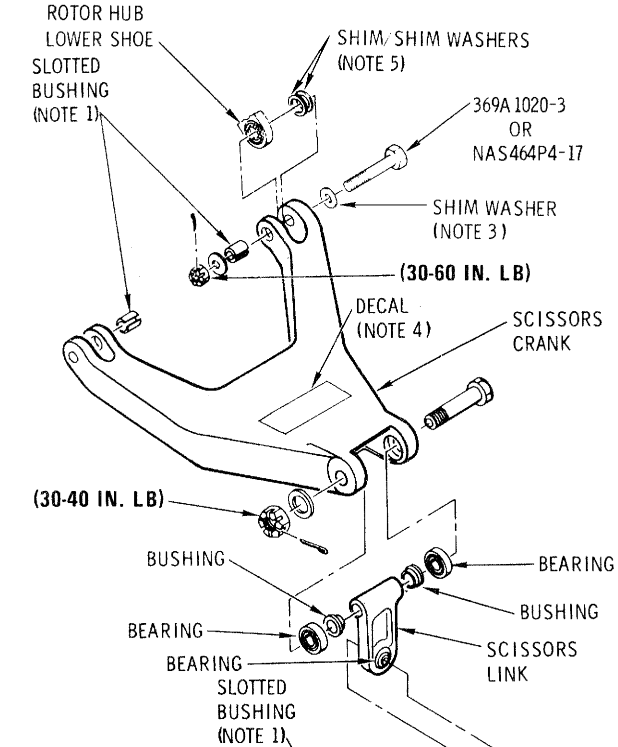
q. Track main rotor blades and adjust autorotation rpm (chapter 5).

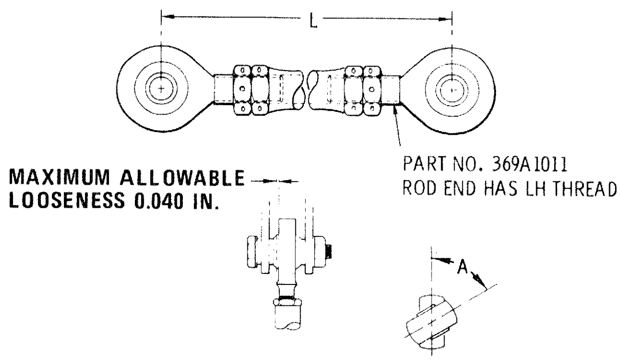
11-8. MAIN ROTOR CONTROLS.

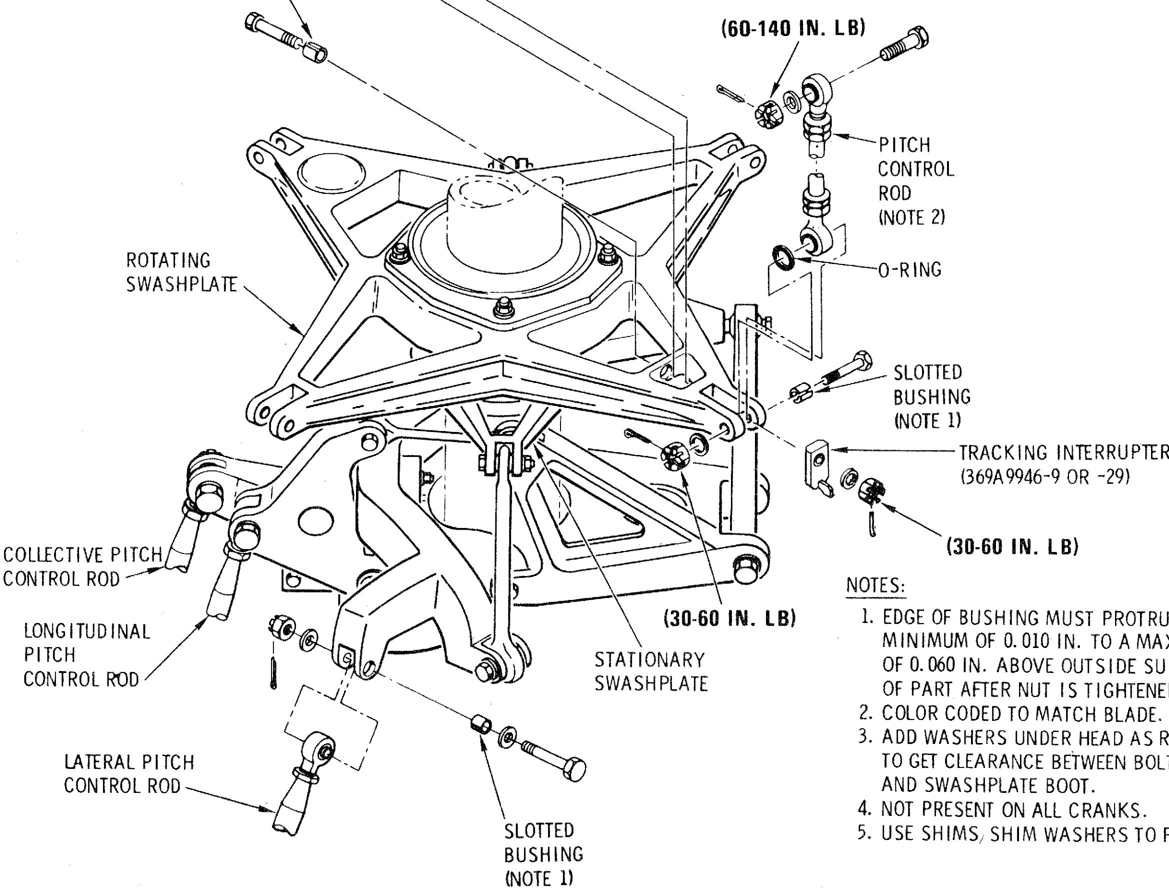
11-9. Description — Main Rotor Controls. The main rotor controls (fig. 11-3) consist of a scissors assembly, rotating swashplate, stationary swashplate, mixer controls (lateral bellcrank, longitudinal idler, collective pitch mixer bellcrank, longitudinal bellcrank, longitudinal control-mixer link, longitudinal link, and two mixer links), and four pitch control rods. Movement of the collective pitch stick and cyclic control stick is transferred to the mixer controls through control rods routed up through the controls tunnel in the center of station

INITIAL ROD ASSY LENGTH AND BEARING ANGULARITY

CONTROL ROD ASSY	DIM L (IN.)	ANGLE A	ASSY PART NUMBER
MAIN ROTOR PITCH (LOWER STRAP PACK)	REF PARA 11-3b(13) & b(14)	45°	369A1008-3
MAIN ROTOR PITCH (UPPER STRAP PACK)		45°	369A1008-3







NOTES:

1. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. COLOR CODED TO MATCH BLADE.
3. ADD WASHERS UNDER HEAD AS REQUIRED TO GET CLEARANCE BETWEEN BOLT END AND SWASHPLATE BOOT.
4. NOT PRESENT ON ALL CRANKS.
5. USE SHIMS/SHIM WASHERS TO FILL GAP.

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Figure 11-3. Pitch Control Rods and Swashplate Scissors Assembly.

78.50 bulkhead. The mixer controls transfer the required combination of collective, longitudinal, and lateral travel to the main rotor blades through the main rotor swashplate assembly.

11-10. Inspection — Main Rotor Controls. (See fig. 11-1.) *a.* Inspect all attaching hardware for secure attachment by trying to finger turn nut and/or bolt.

CAUTION

If any visible surface defects are discovered anywhere, the defective part must be replaced or repaired.

b. Inspect all accessible areas of rotating swashplate (2), stationary swashplate (7), main rotor controls (3) consisting of: lateral bellcrank, longitudinal pitch idler, collective pitch mixer bellcrank, longitudinal pitch mixer bellcrank, longitudinal control-mixer link, longitudinal link, and mixer links. Check for scratches, nicks, dents, cracks and corrosion. Refer to paragraph 11-39 for rework and repair limits.

c. Inspect the swashplate bearing assembly spherical ball for condition. **MAXIMUM RADIAL PLAY BETWEEN THE SLIDING SURFACE AND THE MAST IS 0.010 INCH. MAXIMUM PLAY (VERTICAL MOTION) BETWEEN ADJACENT ARMS OF THE STATIONARY AND ROTATING SWASHPLATE IS LIMITED TO 0.015 INCH. NICKS AND DENTS THAT DO NOT DEFORM THE ID CHAMFER OF THE HARD ANODIZED BALL AND THAT DO NOT EXTEND MORE THAN 0.040 INCH ALONG SPHERICAL BALL SURFACE FROM EDGE ARE PERMISSIBLE.**

d. Inspect the swashplate bearing assembly grease seal for serviceable condition. Breaks in the seal lip are permissible with the following limits: edges are to be smooth; break width 0.050 inch maximum; break length 0.50 inch maximum; not more than one break per inch of lip length; and maximum of five breaks per side of bearing.

NOTE

The swashplate bearing assembly will normally show some signs of grease leakage for the first 10 to 15 hours of operation after a new installation.

e. Check for clearance between the rotating swashplate and longitudinal pitch idler. With the cyclic stick simultaneously positioned full left and full forward, **CLEARANCE BETWEEN THE ROTATING SWASHPLATE AND LONGITUDINAL MIXER IDLER SHOULD NOT BE LESS THAT 0.15 INCH.** Refer to autorotation adjustment (chapter 5).

f. If present, check that the two stationary swashplate drain holes are clear and free of obstruction. Holes are located opposite the longitudinal link attachment ear (fig. 11-4).

11-11. PITCH CONTROL RODS.

11-12. General — Pitch Control Rods. *The four pitch* control rods transfer pitch control from the rotating swashplate to the main rotor blades (fig. 11-3).

11-13. Removal — Pitch Control Rods. (See fig. 11-3.) *a.* Remove cotter pin, nut, washer and bolt that secure upper end of control rod to main rotor pitch housing.

b. Remove cotter pin, nut, washer, and bolt from lower rod end and rotating swashplate.

c. Remove pitch control rod and O-ring.

NOTE

Rods are color-coded to identify location. Check for coding and location at time of removal. If coding no longer exists, reapply suitable color code to match blade.

d. Measure rod length before replacing.

11-14. Inspection — Pitch Control Rods. (See fig. 11-3.) Inspect four pitch control rods for scratches, nicks, dents, cracks, corrosion, excessive wear **(0.040-INCH MAXIMUM AXIAL LOOSENESS)**, and binding of bearings.

NOTE

There are two lengths of pitch control rod barrels in service; one is 0.27 inch longer than the other. Regardless of individual rod lengths, control rod assemblies must be set as stated in paragraph 11-15.

11-15. Installation — Pitch Control Rods. (See fig. 11-3.) Check rods for color code and for the same length as when removed. If correct length for removed rods is not known, check the tower-tracking decal on underside of mating main rotor blade for pitch control rod length. On blades with a decal, set rods for upper strap pack pitch housings to the UPPER dimension, and rods for lower strap pack to the LOWER dimension. On blades without a decal, set rods for upper strap pack pitch housings to 6.60 inches, and rods for lower strap pack to 6.35 inches. Blades must be tracked (chapter 5) if rod length is changed.

CAUTION

Be sure that pitch rod bolts are installed with the bolt head toward the direction of rotor rotation (counterclockwise as viewed from above). Rod end alignment must be centered; if not centered, fitting wear will result. Be sure that rod end threads are engaged beyond witness hole in rod body (hole blocked). After proper bearing alignment, tighten locknuts and install lockwire.

a. Install O-ring and pitch control rod.

b. Install bolt, washer, and nut that attach upper rod end to pitch housing. **TORQUE NUT TO 60 — 140 INCH-POUNDS**, and install new cotter pin.

c. Install bolt, washer, and nut that attach lower rod end to rotating swashplate. **TORQUE NUT TO 30 — 60 INCH-POUNDS** if castellation alignment is not possible at this torque, add another washer. **DO NOT EXCEED THE MAXIMUM TORQUE VALUE.** Install a new cotter pin.

11-16. Adjustment — Pitch Control Rods. Check track of main rotor blades following installation of any removed or replaced parts.

11-17 SCISSORS ASSEMBLY.

11-18. Description — Scissors Assembly. The scissors assembly (fig. 11-3) is a pivoting link that provides a moveable connection between the rotating swashplate and the main rotor hub. The scissors assembly causes the swashplate to rotate with the hub.

11-19. Removal — Scissors Assembly. (See fig. 11-3.) *a.* Disconnect lower end of pitch control rod that is in line with the scissors link in the rotating swashplate.

CAUTION

If rotor head is off the aircraft, or if all pitch control rods are disconnected at the time scissors is being removed, color code each leg of scissors crank to match rotor hub lower shoe lugs. It is possible to reinstall the crank on the wrong side of the hub and mismatch the pitch control rod attachments.

b. Remove cotter pin, nut, washer, and bolt from each leg of the scissors crank. Disconnect crank from hub lower shoe; use care to prevent loss of spacer shims.

c. Remove cotter pin, nut, washer, and bolt that connects scissors link to washplate; remove scissors assembly.

11-20. Disassembly — Scissors Assembly. (See fig. 11-3.) *a.* Remove cotter pin, nut, washer, and bolt that join the link and crank.

b. Remove two bushings from link only if replacement is necessary.

c. Do not remove bearing from link. Replace the complete link if bearing is defective.

d. Remove two slotted bushings and the two bearings from crank only if replacement is necessary.

11-21. Inspection — Scissors Assembly. (See fig. 11-3.)

WARNING

Any evidence of failure, damage, or deformation of the crank or link is justification for replacement of either the crank, the link, or the complete scissors assembly. Failure of any of these components in flight can result in failure of the moveable flight controls, loss of the aircraft, and possible injury or loss of life.

CAUTION

During cleaning, do not allow solvent to enter the races of scissors assembly bearings.

a. Inspect crank and link for evidence of impact damage, and deformation. If condition is questionable, perform a fluorescent penetrant inspection (TM 55-1500-204-25/1).

b. Inspect bearings for binding. looseness in mating bore, and wear. **BALL BEARING MAXIMUM WEAR LIMITS ARE 0.010 INCH RADIAL AND 0.020 INCH AXIAL. TEFLON BEARING MAXIMUM WEAR IS LIMITED TO 0.040 INCH RADIAL.**

c. Inspect bushings for condition.

11-22. Reassembly — Scissors Assembly. *a.* If bushings were removed from the crank (fig. 11-3) during disassembly, install new bushings. Use air-drying solid film lubricant (C64); do not use zinc chromate. Seat bushings until ends are flush with inside surface of crank inboard ears.

b. If bearings were removed from the crank during disassembly, press bearings until outer face is sealed firmly against shoulder of counterbore.

c. If bushings were removed from link during disassembly, install new bushings. After installation, dimension between outside faces of bushings should be 1.616 to 1.628 inches.

d. Align bore of link with bore of crank; install bolt, washer, and nut. **TORQUE NUT TO 30 — 40 INCH-POUNDS.** Install new cotter pin.

11-23. Installation — Scissors Assembly. *a.* Place scissors assembly link in position (fig. 11-3). Check that slotted bushing is in swashplate web and install link bolt, washer, and nut. **TORQUE NUT TO 30 — 60 INCH-POUNDS.** Install new cotter pin.

CAUTION

Scissors crank must be positioned as shown in figure 11-3 with hole in crank web down and decal, if present, up. Refer also to Caution in paragraph 11-19.

b. Position scissors crank on same side of hub from which removed and install required spacer shims on hub lower shoe lugs. Install bolt, washer, and nut. **TORQUE NUT TO 30 — 60 INCH-POUNDS.** Install new cotter pin.

c. Position O-ring and lower end of pitch control rod in swashplate lug and install bolt, washer, and nut. (Check that head of bolt is in direction of rotor rotation.) **TORQUE NUT TO 30 — 60 INCH-POUNDS.** Install new cotter pin.

11-24. MAIN ROTOR SWASHPLATE.

11-25. Description — Main Rotor Swashplate. The main rotor swashplate (fig. 11-3) consists primarily of a rotating swashplate, stationary swashplate, a bearing assembly and retainer, and a counterweight.

11-26. Removal — Main Rotor Swashplate. *a.* Tag or color-identify the four pitch control rods (fig. 11-3) so they will be reinstalled in the same location.

CAUTION

Failure to reinstall the four pitch control rods in the correct location will most likely result in the main rotor blades being badly out of track.

b. Remove scissors assembly (para 11-19).

c. Disconnect the three remaining pitch control rods from the rotating swashplate.

d. Remove main rotor hub (chapter 5).

e. Remove cotter pin, nut, bolt, and two washers that secure longitudinal link to stationary swashplate.

f. Remove cotter pin, nut, washers, and bolt that secure right and left side mixer links to stationary swashplate.

CAUTION

Do not strike rotating swashplate or bearing assembly (fig. 11-4) with tools or in any way deface these components. The rotating swashplate is a stressed part that has had all surfaces shot peened. Use particular care to protect teflon liner in bearing bore. A damaged swashplate bearing must not be retained in service.

g. Lift swashplate assembly carefully up and off of mast.

11-27. Disassembly — Main Rotor Swashplate (AVIM). (See fig. 11-4.) *a.* Remove the tracking interrupters. Mark the -3/-23 and -5/-25 (match the blade color code) so that original position can be maintained for tracking pattern consistency.

CAUTION

When bolts that attach the bearing inner race flange to the stationary swashplate are removed, the ball of the bearing assembly must be swiveled. Use care when swiveling bearing to prevent dirt and foreign matter from entering between the ball and teflon liner.

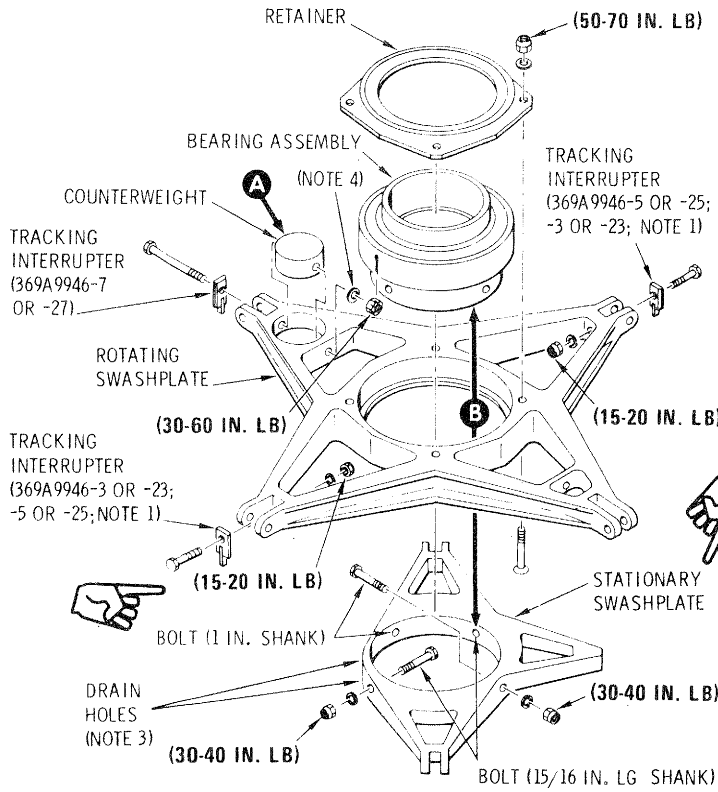
b. Remove four nuts, washers, and close-tolerance bolts.

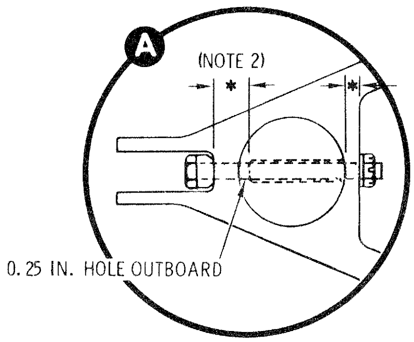
c. Using suitable removal tool (detail B) positioned against lower end of spherical bearing, press on stationary swashplate arms to separate bearing assembly and rotating swashplate from stationary swashplate. Little pressure is required except in corroded or damaged parts.

d. Remove four nuts, washers, and bolts that secure bearing retainer to rotating swashplate. Remove bearing retainer.

NOTES:

1. COLOR CODE TO MATCH BLADES AND MAINTAIN SEQUENCE.
2. COAT SURFACES MARKED BY ASTERISK WITH GREASE (C46).
3. NOT PRESENT ON ALL AIRCRAFT.
4. **RETAIN COUNTERWIEGHT FOR INSTALLATION ON THE NEW SWASHPLATE ASSEMBLY**





CAUTION:
USE CARE TO AVOID NICKING OR DENTING SPHERICAL BALL.

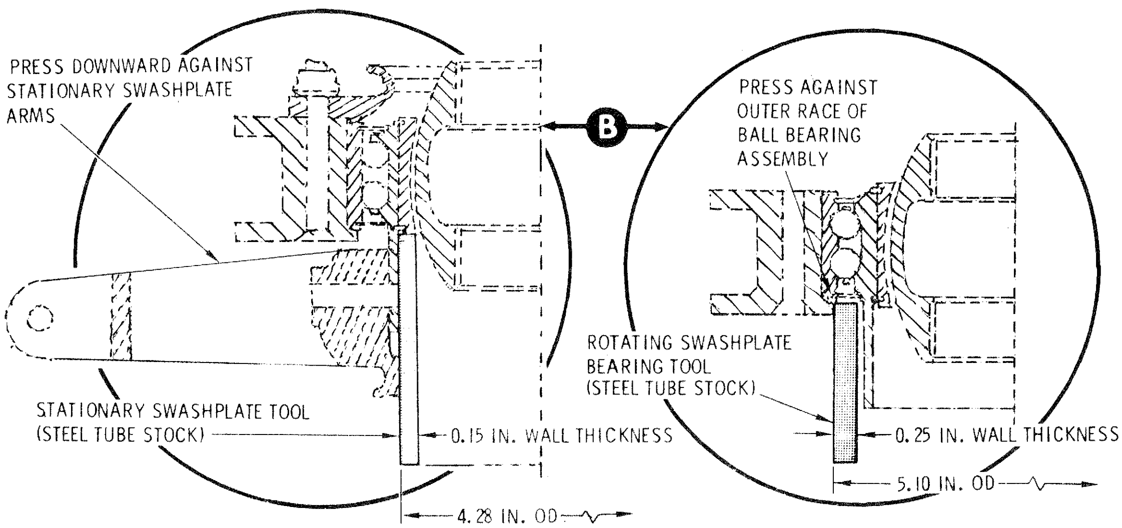


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Figure 11-4. Main Rotor Swashplate Assembly.

e. Using suitable removal tool (detail B), carefully press swashplate bearing assembly from sealed position in rotating swashplate. Do not apply any pressure on bearing inner race during pressing operation.

f. To remove counterweight, remove cotter pin, nut, washer, tracking interrupter, and bolt; then press counterweight from swashplate.

11-28. Inspection — Main Rotor Swashplate (AVIM).

NOTE

During cleaning, do not allow solvent to enter race area of bearing to prevent dirt from being washed into teflon liner. The liner and the bearing may be wiped clean with alcohol (C82) using a soft clean cloth.

a. Inspect swashplate bearing assembly (fig. 11-4) for evidence of binding in either radial or axial movement. Inspect teflon liners for condition. Ensure that flexible rubber seals enclosing ball bearing races are in place; inspect for deterioration and indication of grease leakage. Refer to paragraph 11-8 for swashplate and seal condition tolerances.

b. Visually inspect swashplate and spherical bearing for evidence of scoring, cracks, corrosion, and galling.

c. Check the existing preload on the spherical bearing. Secure the flats of the ball in a padded vise, hook a standard spring scale to the bearing assembly outer race, and measure the torque required to rotate the race. With the ball in motion, **TORQUE SHOULD BE FROM ZERO TO 120 INCH-POUNDS**. (Use the average of two readings taken 90 degrees apart.)

d. Perform a fluorescent dye-penetrant inspection of rotating swashplate, stationary swashplate, and bearing retainer if their condition is questionable. (Refer to TM 55-1500-204-25/1.)

e. Check that the counterweight is securely bonded in rotating swashplate.

11-29. Reassembly — Main Rotor Swashplate (AVIM). *a.* Using methyl ethyl ketone (C69) on a clean cloth, clean the bearing OD and swashplate ID.

b. Apply a coating of surface primer (C91) on bearing OD and swashplate ID and air-dry for 5 to 10 minutes.

c. Apply a thin coating of grade A locking compound (C90) on bearing OD and swashplate ID.

CAUTION

Be sure to protect teflon liner in spherical bearing bore during reassembly. Make sure that the bearing assembly is started straight and does not cock while being pressed in.

d. Using suitable installation tool (detail B), apply pressure to the outer race only and install bearing assembly in swashplate bore. Remove excess locking compound and ensure that compound does not enter bearing race.

e. After bearing assembly is seated, apply a fillet of compound to bondline of bearing and swashplate.

f. Allow compound to harden for 24 hours before using the assembly.

g. Position bearing retainer over swashplate bearing assembly and install four bolts, washers, and nuts. **TORQUE NUTS TO 50 — 70 INCH-POUNDS.**

h. Apply a coat of zinc chromate primer (C79) to bore of stationary swashplate and carefully align mating bolt holes. Using suitable installation tool while primer is still wet, push on end of spherical bearing race and install bearing assembly and rotating swashplate in stationary swashplate. Completely seal the interface with primer.

i. Install bolts, washers, and nuts to secure bearing assembly flange to stationary swashplate. Using wet zinc chromate primer (C79) install the two shorter bolts first; then install the two longer bolts. **TORQUE THE NUTS TO 30 — 40 INCH-POUNDS.**

j. If counterweight was removed: Apply a coating of mixed two-part adhesive (C7) to counterweight OD and reinstall weight; the smaller bolt hole goes outboard. Use a dummy bolt to align weight in swashplate; then remove bolt. Cure adhesive according to container instructions. Reinstall bolt, interrupter (beveled edges inboard), washer, and nut; **TORQUE TO 30 — 60 INCH-POUNDS** and install new cotter pin.

k. Check for correct location (color code marking, etc) and reinstall other two interrupters, but **TORQUE NUTS TO 15 — 20 INCH-POUNDS** (fig. 11-4).

11-30. Installation — Main Rotor Swashplate. *a.* Position swashplate over main rotor mast and carefully lower into place.

b. Align longitudinal link with stationary swashplate (fig. 11-5).

c. Install flanged bushing (if removed), bolt, two washers, nut, and new cotter pin to secure upper end of longitudinal link.

d. Align upper ends of mixer links with stationary swashplate. Install slotted bushing (if removed), bolt, two washers, nut, and new cotter pin to secure each link to swashplate.

e. Install scissors assembly (para 11-23).

f. Use color code and reinstall pitch control rods in the same locations from which removed.

11-31. Adjustment — Main Rotor Swashplate. Check track of main rotor blades following installation of any removed or replaced parts.

11-32. MIXER CONTROLS.

11-33. Description — Mixer Controls. The mixer controls (fig. 11-5) consist of various links, bellcranks, idlers, a support bracket and associated fasteners that secure these components in place. The mixer controls connect the control rods with the swashplate.

11-34. Removal — Mixer Controls. *a.* Remove air intake forward fairing for access (chapter 2).

b. Remove cotter pin, two washers, and bolt that secure upper end of the collective mixer, lateral mixer, and longitudinal mixer control rods (fig. 11-3).

NOTE

The control rods are attached by slightly different length bolts. Reinstall each bolt in its bellcrank following control rod removal to assure reinstallation of the correct size.

c. Remove cotter pins, nuts, washers, and bolts from both ends of mixer links (fig. 11-5).

d. Remove mixer links. (Mixer link bearings are not replaceable.)

e. Remove cotter pin, nut, two washers, bolt, and disconnect longitudinal link.

f. Remove cotter pin, nut, three washers, and bolt from each end of longitudinal control-mixer link and remove link.

g. Remove cotter pin, nut, two washers, and bolt from lateral bellcrank. Lift lateral bellcrank from collective pitch mixer bellcrank.

h. Remove cotter pin, nut, three washers, flanged bushing and bolt from each side of longitudinal pitch mixer bellcrank. Carefully remove bellcrank.

i. Remove cotter pin, nut, two washers, and bolt from hingeline of support bracket, longitudinal pitch idler and collective pitch mixer bellcrank. Separate the three parts.

j. Remove the three nuts, six washers, and three bolts that attach support bracket to mast base. Remove support bracket from mast base.

k. Remove the mixer support attaching bushing.

11-35. Inspection — Mixer Controls. (See fig. 11-5.)

CAUTION

Do not allow solvent to enter bearing races when cleaning mixer control components. This will prevent washing of dirt into bearing surfaces or grease out of the bearings.

a. Inspect all bushings for security of fit.

b. Inspect mixer links for evidence of impact damage, and spherical bearings in each link for binding or looseness in link bore. Any evidence of link damage or bearing malfunction requires replacement of complete mixer link.

c. Inspect longitudinal link, support bracket, longitudinal control-mixer link, longitudinal pitch idler, longitudinal pitch mixer bellcrank, lateral bellcrank, and collective pitch mixer bellcrank for damage. Perform a fluorescent dye-penetrant inspection on components that are questionable. (Refer to TM 55-1500-204-25/1.)

d. Inspect spherical bearing in longitudinal link for binding, looseness in link bore, and axial wear; **WEAR LIMIT IS 0.040 INCH MAXIMUM AXIAL OR 0.008 INCH MAXIMUM RADIAL.** Inspect bearings in other linkage for binding, and looseness in mating bores.

11-36. Repair — Mixer Controls. *a.* Remove defective bearings or bushings by pressing out the old part. Install a new bearing of the correct type with surface primer and grade A locking compound (C91 and C90), according to container instructions.

b. The left-hand bearing in the support bracket is not supplied as a part of the bracket. This bearing is installed at the time the support bracket and longitudinal pitch idler are assembled. Install a new bearing by seating it flush with inner face of support bracket flange.

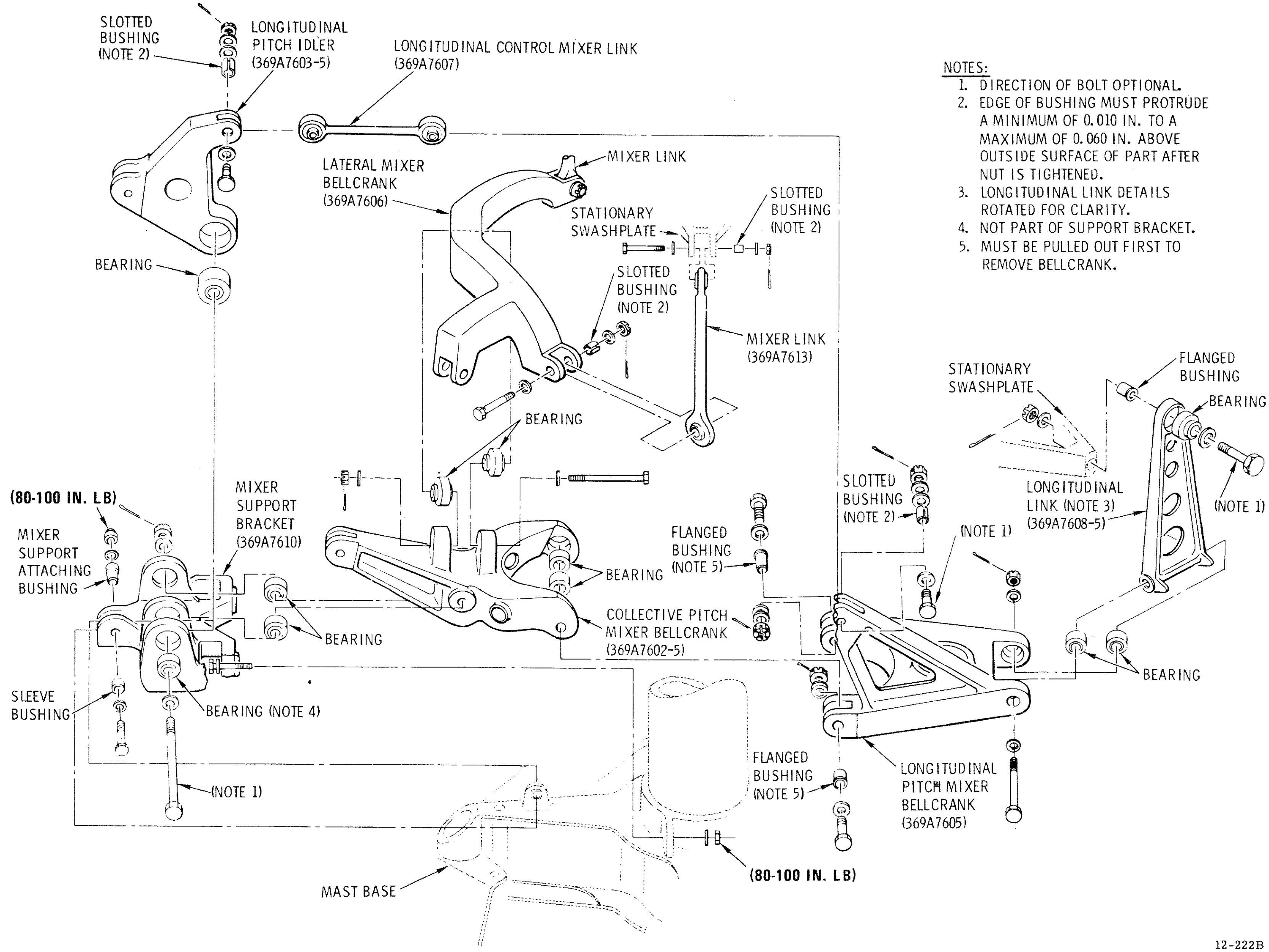


Figure 11-5. Mixer Controls Assembly.

c. Refer to paragraph 11-39 for rework of minor surface defects such as scratches, nicks, and corrosion.

11-37. Installation — Mixer Controls. (See fig. 11-5.) *a.* Position support bracket over forward lug on mast base.

b. Install the two aft bolts and washers in the support bracket and mast base. Install two thin washers and nuts; **TORQUE NUTS TO 80 — 100 INCH-POUNDS ABOVE DRAG TORQUE.**

c. Using a ball hole gage and standard micrometer, check that misalignment between the forward bolt hole bore in the support bracket and the bore in the mast base lug is **0.015 INCH OR LESS.**

NOTE

If necessary, loosen the aft two bracket-to-fitting nuts, position support bracket for minimum misalignment, and retighten nuts.

d. Install mixer support attaching bushing in bracket and mast base lug.

e. Check that the sleeve bushing is in place; then install bolt, two washers, and nut. **TORQUE NUT TO 80 — 100 INCH-POUNDS.**

f. Position lateral bellcrank between collective pitch mixer bellcrank bearings and install bolt, two washers, nut, and new cotter pin.

g. Position longitudinal pitch idler and collective pitch mixer bellcrank on support bracket. Install left-hand bearing, step *b* of paragraph 11-36, with bolt, two washers, nut, and new cotter pin.

h. Position longitudinal bellcrank with collective pitch mixer bellcrank; insert the two flanged bushings into place. Install two bolts, six washers (two under each nut), two nuts, and new cotter pins.

i. Position longitudinal control-mixer link between longitudinal pitch idler and longitudinal pitch mixer bellcrank; check that two slotted bushings are in place. Install bolt, three washers (two under each nut), and new cotter pin at each end of link.

j. Position longitudinal link between longitudinal pitch mixer bellcrank bearings. Install bolt, two washers, nut, and new cotter pin.

k. Position mixer links in stationary swashplate and lateral bellcrank; check that slotted bushings are in place. Install bolt, two washers, nut, and new cotter pin at each end of link.

l. Place the collective mixer, lateral mixer, and longitudinal mixer control rods (fig. 11-3) in position; check that slotted bushings are in place. Install bolt, two washers, nut, and new cotter pin to secure each rod end.

m. Install air intake forward fairing (chapter 2).

11-38. Adjustment — Mixer Controls. Check rigging of collective and cyclic controls following installation of any removed or replaced parts (para 11-4).

11-39. Rework and Wear Limits — Mixer Control Components. Refer to table 11-5 and rework minor surface defects such as scratches, nicks and corrosion as follows. **THE FOLLOWING DEPTH LIMITS REPRESENT TOTAL LIMITS, INCLUDING THE EFFECTS OF ALL PREVIOUS REPAIRS TO ANY GIVEN AREA.**

a. Cast and forged surfaces may be reworked to a depth of 0.020 inch.

b. Flat machined surfaces, except clevis inner ears, may be reworked to a depth of 0.015 inch; clevis inner ear surfaces may be reworked to a depth of 0.020 inch.

c. Machined holes may have 0.003 inch removed from the bore wall in an area no greater than 15 percent of the circumference and 50 percent of the depth.

d. All edges may have 0.030 inch removed except around machined holes which are limited to 0.010 inch chamfer.

e. All rework must be smoothly blended into the adjacent surfaces and the finish must be restored according to chapter 2.

f. Check spherical teflon lined bearings (fig. 11-4) for excessive wear as indicated by a maximum of 0.008-inch radial play or 0.040-inch axial play.

g. Machined areas within a radius of 0.125 inch from attached bolts holes, and including such holes, may be reworked up to 25 percent of total machined surface; maximum depth after rework must not exceed 0.020 inch.

11-40. COLLECTIVE MIXER, LATERAL MIXER AND LONGITUDINAL MIXER (TUNNEL-ROUTED) CONTROL RODS.

11-41. Description — Collective Mixer, Lateral Mixer and Longitudinal Mixer (Tunnel-Routed) Control Rods. Each tunnel-routed control rod (fig. 11-6) consists of two rod end bearings and an anodized aluminum alloy rod. The control rods mechanically transfer displacement of the flight controls to the associated bellcranks and idler at the front of the main rotor mast base. All the tunnel-routed control rods are removed, inspected, repaired and installed in the same manner.

Table 11-4. Premaintenance Requirements for Removal of Collective Pitch Control Rod.

Conditions	Requirements
Special Tools	(T27)

11-42. Removal — Collective Mixer, Lateral Mixer and Longitudinal Mixer (Tunnel-Routed) Control Rods. *a.* Gain access to lower rod ends by removing controls access door at base of canted bulkhead in cargo compartment.

CAUTION

Before disconnecting the collective pitch control rod, install collective bungee installation tool (T27) to prevent spring reaction into the collective stick.

b. (See fig. 11-6.) Remove cotter pin, nut, two washers, and bolt to disconnect lower rod end from bellcrank or fitting.

c. Hand-turn main rotor blades to obtain clear space for rod removal.

d. Remove cotter pin, nut two washers, and bolt that secure upper end of control rod.

e. Remove boot from control tunnel cover.

CAUTION

Use care during removal of control rods. Any surface damage caused by hasty removal may result in unserviceable control rods.

f. Grasp upper end of control rod and carefully withdraw the rod until the lower rod end clears the cover.

11-43. Inspection — Collective Mixer, Lateral Mixer and Longitudinal Mixer (Tunnel-Routed) Control Rods. *a.* Remove controls access door (chapter 2).

b. Inspect all attaching hardware for secure condition by trying to finger-turn nut and/or bolt. Check that there is clearance between hingeline hardware when controls are moved.

c. Check visible length of rods for bends or similar distortions, dents, cracks, scratches, corrosion deposits, and excessive wear in rod end bearings **(0.040-INCH AXIAL LOOSENESS MAXIMUM).** Inspect rod end bearings for correct alignment (no evidence of jamming).

(1) **MAXIMUM SCRATCH DEPTH BEFORE REWORK ON COLLECTIVE AND LATERAL MIXER CONTROL RODS (UPPER- AND LOWER-THIRD) IS 0.003 INCH. CENTER-THIRD MAXIMUM DEPTH IS 0.001 INCH.**

(2) **MAXIMUM SCRATCH DEPTH BEFORE REWORK ON LONGITUDINAL MIXER CONTROL (UPPER- AND LOWER-THIRD) IS 0.004 INCH. CENTER-THIRD MAXIMUM DEPTH IS 0.002 INCH.**

(3) **MAXIMUM SCRATCH DEPTH BEFORE REWORK ON COLLECTIVE MIXER CONTROL ROD (UPPER- AND LOWER-THIRD) IS 0.002 INCH. CENTER-THIRD MAXIMUM DEPTH IS 0.001 INCH.**

(4) **DENTS NOT GREATER IN DEPTH THAN 10 PERCENT OF TUBE DIAMETER AND WITH SMOOTH EDGE RADIUS NOT LESS THAN 10 PERCENT OF TUBE DIAMETER ARE ACCEPTABLE.**

(5) Replace control rods that are not within these limits.

d. Inspect for loose rivet at fixed rod end (lower end).

CAUTION

When tightening loosened jam nuts on rod ends as in *e* below, always hold the rod end with a wrench.

e. Determine if excessive rod end looseness exists in tunnel-routed control rod assemblies by checking the upper end as follows:

CAUTION

Do not disconnect rod ends from the mixer assembly to perform this inspection, except as instructed.

(1) Loosen jam nut a minimum of 2 threads.

(2) Apply light finger pressure to the top of the rod end to preload it against one side of attaching clevis. This will eliminate play other than where inspection is desired.

(3) Use a dial indicator or position a 6-inch rule

Table 11-5. Mixer Control Component Rework and Wear Limits.

Nomenclature	Part No. (fig. 11-5)	Material & Type	Rework Limits (para 11-39).
Collective pitch mixer bellcrank	369A7602-5	*Mag casting	*a, b, c, d, e*
Longitudinal mixer idler	369A7603-5	Mag casting	*a, b, c, d, e, f*
Longitudinal mixer bellcrank	369A7605-5	Mag casting	*a, b, c, d, e*
Lateral mixer bellcrank	369A7606	Mag casting	*a, b, c, d, e*
Longitudinal mixer link	369A7607	**Al aly bar	*b, c, d, e*
Longitudinal link	369A7608-5	Al aly forging	*a, c, d, e, f*
Mixer support bracket	369A7610	Al aly casting	*a, b, c, d, e*
Mixer links	369A7613	Al aly bar	*b, c, d, e, f*
Scissors crank	369A1002-5	Al aly casting	*a, b, c, d, e*
Scissors link	369A1003-5	Al aly forging	*a, b, c, d, e*
Stationary swashplate	369A7612	Mag casting	*a, b, c, d, e, g*
Rotating swashplate	369A7611	Al aly forging	*a, b, c, d, e, g*

*Mag = Magnesium **Al aly = Aluminum alloy

horizontally at the upper end of the tube (where the rod end enters). Lightly deflect the tube laterally from one extreme to the other.

NOTE

Maintain the preload of the rod end bearing to eliminate the effect of bearing looseness.

(4) Measure the total movement of the top of the tube where the rod end enters.

(5) IF THE TOTAL LATERAL MOVEMENT FROM ONE EXTREME TO THE OTHER IS IN EXCESS OF 0.06 INCH, THE ENTIRE ROD ASSEMBLY MUST BE REPLACED WITH A SERVICEABLE PART.

NOTE

Because of normal tolerance buildup some play or looseness is expected. Play or looseness less than 0.016 inch is no cause for concern.

(6) IF THE TOTAL LATERAL MOVEMENT IS LESS THAN 0.047 INCH, THE ROD ASSEMBLY IS SERVICEABLE; TIGHTEN THE JAM NUT.

(7) IF THE LATERAL MOVEMENT IS 0.047 TO 0.06 INCH THE ROD END SHOULD BE FURTHER INSPECTED FOR SERVICEABILITY, (8) through (10) below.

(8) Disconnect rod end from the control system. Screw rod end completely out of control rod. Use a 0.339-inch-diameter drill (size R) shank free of burrs, etc to check the rod threads.

(9) IF DRILL CAN BE INSERTED THE ROD ASSEMBLY IS UNSERVICEABLE AND MUST BE REPLACED.

(10) If drill cannot be inserted, the rod assembly is serviceable. Reinstall the rod end; then reconnect the rod to the control system and tighten the jam nut. Check the rigging of the reconnected control system (para 11-4).

f. Install controls access door.

11-44. Repair — Collective Mixer, Lateral Mixer, and Longitudinal Mixer (Tunnel-Routed) Control Rods. *a.* Perform a straightness check on a control rod that appears bent or bowed. The total length of any tunnel-routed rod (excluding rod ends) must be **STRAIGHT WITHIN 0.050 INCH, WITH STRAIGHTNESS VARIATION LIMITED TO A MAXIMUM OF 0.010 INCH IN EACH FOOT OF LENGTH.**

WARNING

A dye-check for cracking shall always be performed after cold-straightening Replace a cracked rod, or a cracked or bent rod end.

b. Cold-straighten a bent rod that is not within tolerance, *a* above, provided there are no nicks or sharp dents in the bend length. Do NOT use the rod ends to support the rod during the straightening process.

CAUTION

Use care when drilling to remove or install riveted rod end; the rod end is steel and the rod is aluminum.

c. **REPLACE A CONTROL ROD END IF BEARING AXIAL PLAY IS MORE THAN 0.040 INCH.** Set initial control rod length and bearing angularity as shown in figure 11-6.

d. Use abrasive paper (C3) or (C4), not coarser than grade 400, to remove scratches from control rods described in step *c* of paragraph 11-43. Use finer grades, as necessary, to restore surface to the original finish. Remove material in such a manner that no abrupt changes occur in surface contours. Apply chemical film treatment (C20) to the repaired surface.

11-45. Installation — Collective Mixer, Lateral Mixer and Longitudinal Mixer (Tunnel-Routed) Control Rods. *a.* Hand-turn main rotor blades to obtain clear space for rod installation.

CAUTION

Use care during installation of a control rod to avoid striking other installed control rods.

b. With the riveted rod end at the lower position, carefully lower rod through cover opening into tunnel.

c. Install control tunnel cover boot and secure with self-clinching nylon strap.

CAUTION

Do not tighten loosened jam nuts on rod end bearings without holding the rod end with a wrench.

d. Align lower rod end with mating bellcrank. Install bolt, two washers, nut, and new cotter pin (fig. 11-6).

e. Align upper rod end of control rod with mating idler or bellcrank. Install bolt, two washers, nut, and new cotter pin.

f. Remove bungee installation tool used for removal of collective control rod.

11-46. Adjustment — Collective Mixer, Lateral Mixer and Longitudinal Mixer (Tunnel-Routed) Control Rods. Perform controls rigging check (para 11-4) following installation of removed or replaced parts.

11-47. CONTROLS SUPPORT BRACKET AND BELLCRANKS.

11-48. Description — Controls Support Bracket and Bellcranks. The controls support bracket (fig. 11-7) is a cast magnesium bracket that mounts below the pilot's collective stick cover, between the right and left inboard sides of the seat structure. The bracket provides the hingeline for the longitudinal and lateral idler bellcranks, the tail rotor bellcrank, and the engine droop control bellcrank, which are also made of magnesium. The bracket also provides the mounting point for the collective torque tube support bearing.

11-49. Removal — Controls Support Bracket and Bellcranks. (See fig. 11-7.) *a.* Remove foot fairings and controls access door (station 78.50), and remove pilot's collective stick cover.

b. Remove collective bungee (para 11-80).

c. Disconnect tunnel-routed control rods (fig. 11-6).

d. Remove cotter pin, nut, two washers and bolt; disconnect upper end of each cyclic trim actuator.

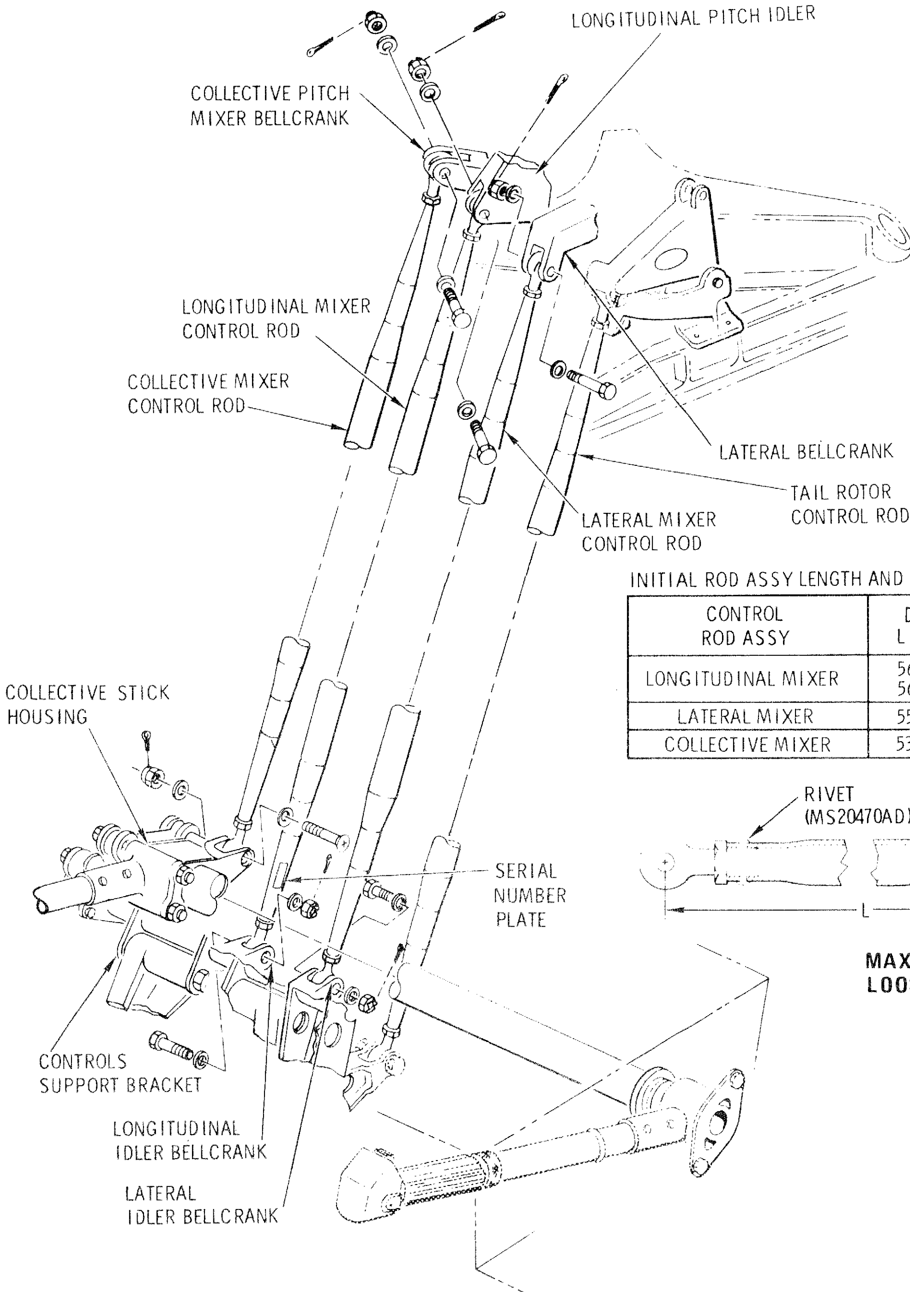
e. Disconnect floor-routed tail rotor control rod from tail rotor bellcrank.

f. Disconnect station 70 lateral control rod from lateral idler bellcrank.

g. Disconnect one-way lock from longitudinal idler bellcrank (fig. 11-7).

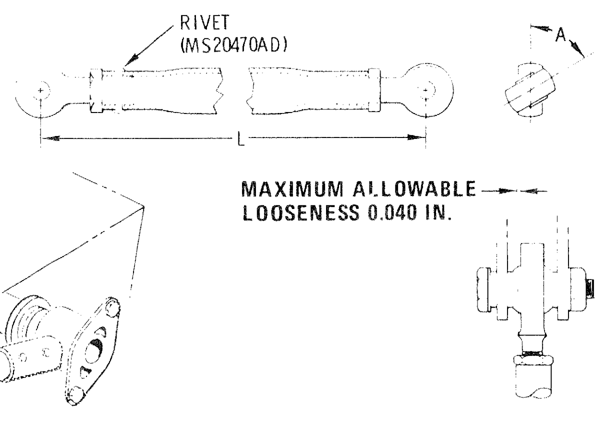
h. Remove pilot's collective pitch stick (para 11-62).

i. Provide support for collective torque tube. Remove two bolts, four washers, and two nuts that secure bungee support bracket and collective torque tube to controls support bracket.



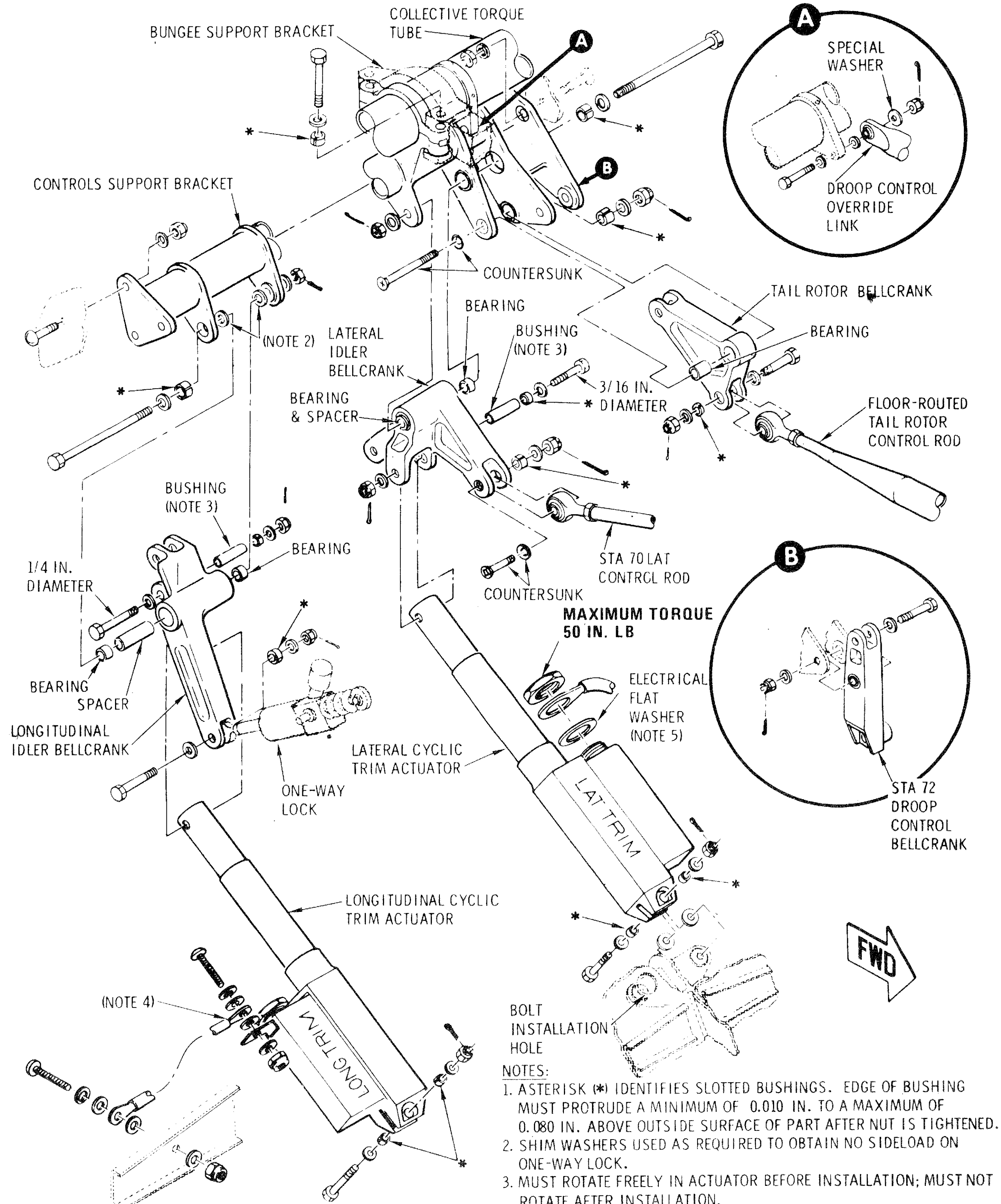
INITIAL ROD ASSY LENGTH AND BEARING ANGULARITY (BEFORE RIG)

CONTROL ROD ASSY	DIM L (IN.)	ANGLE A	ASSY PART NUMBER
LONGITUDINAL MIXER	56.67 56.51	INLINE	369A7011
LATERAL MIXER	55.67	INLINE	369A7012
COLLECTIVE MIXER	53.14	INLINE	369A7009



12-223A

Figure 11-6. Tunnel-Routed Control Rods.



NOTES:
1. ASTERISK (*) IDENTIFIES SLOTTED BUSHINGS. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.080 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. SHIM WASHERS USED AS REQUIRED TO OBTAIN NO SIDELOAD ON ONE-WAY LOCK.
3. MUST ROTATE FREELY IN ACTUATOR BEFORE INSTALLATION; MUST NOT ROTATE AFTER INSTALLATION.
4. ON AIRCRAFT WITH BOND JUMPER INSTALLED, ATTACHMENT AT ACTUATOR MAY BE EITHER CONFIGURATION SHOWN.
5. NOT INSTALLED ON ALL AIRCRAFT. INSTALL AT FIRST OPORTUNITY.

Figure 11-7. Controls Support Bracket, Idler Bellcranks, and Cyclic Trim Actuators *j.* Remove cotter pin, nut, three washers and bolt (detail A, fig. 11-7); disconnect aft end of droop control override link from collective torque tube bracket.

k. Remove cotter pin, nut, two washers, bolt, and station 72 droop control bellcrank from support bracket (detail B).

l. Remove six screws, six washers, and six nuts that secure support bracket to seat structure.

m. Remove support bracket; use care to avoid striking any of the bellcranks and pushrods.

11-50. Disassembly — Controls Support Bracket and Bellcranks. (See fig. 11-7.) *a.* Remove cotter pin, nut, two washers, bolt, and lateral idler bellcrank. Do not remove slotted bushing unless replacement is necessary.

b. Remove cotter pin, nut, two washers, bolt, and tail rotor bellcrank. Do not remove slotted bushing unless replacement is necessary.

c. Remove cotter pin, nut, two washers, bolt, and longitudinal idler bellcrank. Do not remove slotted bushing unless replacement is necessary.

NOTE

The shim washers between the longitudinal idler bellcrank and support bracket are used to align the one-way lock for no sideload. Keep the washer selection with the bracket for use during reassembly. Note also that the lateral trim actuator upper hingeline has 0.187-inch ID; longitudinal has 0.25-inch ID.

11-51. Inspection — Controls Support Bracket and Bellcranks. *a.* Remove controls access door (chapter 2).

b. Inspect all attaching hardware for secure attachment by trying to finger-turn nut and/or bolt.

c. Check bellcranks for dents, cracks, corrosion deposits, and evidence of excessive wear or binding in hingeline bearings. (Refer to area 2, step 6.1 page 1-31.

d. Perform a fluorescent penetrant inspection on any questionable part (TM 55-1500-204-25/1).

e. Closely inspect the critical area of the controls support bracket shown in figure 11-8.

f. Install controls access door.

11-52. Repair — Controls Support Bracket and Bellcranks. *a.* Replace unserviceable bearings. Install a new bearing of the correct type with surface primer and grade A locking compound (C91) and (C90), according to container instructions.

NOTE

Ensure that idler bellcrank spacers are reinstalled between bearings whenever idler bellcrank bearings are replaced.

b. Replace bellcranks or the controls support bracket for distortion, cracks, or elongated holes.

11-53. Reassembly — Controls Support Bracket and Bellcranks. (See fig. 11-7.) *a.* Install longitudinal idler bellcrank with bolt, two washers, nut, and new cotter pin. Check that the slotted bushing is in place in the support bracket lug, and reinstall the shim washers

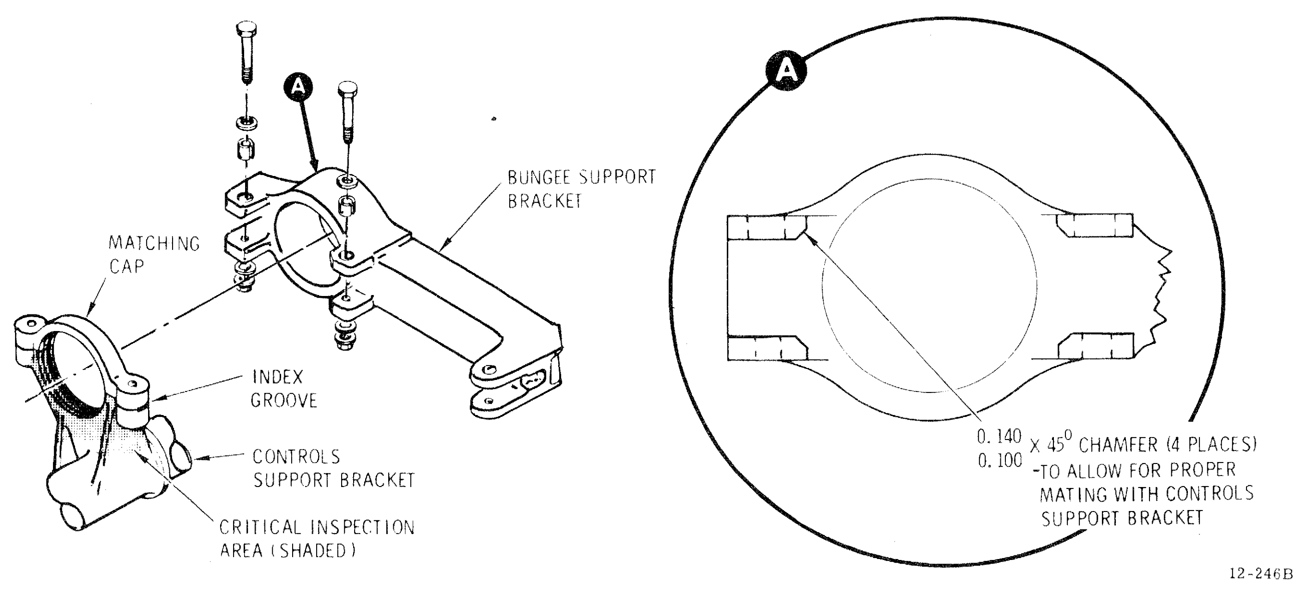


Figure 11-8. Controls Support Bracket Critical Inspection Area.

used between the bellcrank and bracket for no sideload of the one-way lock.

b. Install lateral idler bellcrank, bolt, two washers, nut, and new cotter pin. Check that slotted bushing is in place in the support bracket lug.

c. Install tail rotor bellcrank, countersunk head bolt, two washers (one countersunk), nut, and new cotter pin. Check that slotted bushing is in place in the support bracket lug.

11-54. Installation — Controls Support Bracket and Bellcranks. (See fig. 11-7.) *a.* Carefully position assembled support bracket and bellcranks between the seat structure bulkheads and secure with six screws, six washers, and six nuts.

b. Position station 72 droop control bellcrank in support bracket and install with bolt, two washers, nut, and new cotter pin (detail B).

c. Check that two slotted bushings are in place in the bungee support bracket. Position collective torque tube and bungee bracket on support bracket cradle and install cradle bolts, four washers, and two nuts. Check that index groove in edge of cap that clamps torque tube to support bracket cradle mates with matching index at bracket cradle parting surface. Check for possible binding or interference between bungee support bracket, controls support bracket and matching cap. If there is interference, rework the bungee support bracket as shown in figure 11-8.

d. Install pilot's collective pitch stick (para 11-67).

e. Connect aft end of droop control override link to collective torque tube droop control bracket.

f. Connect one-way lock to longitudinal idler bellcrank (fig. 11-7).

g. Connect station 70 lateral control rod to lateral idler bellcrank. Check that slotted bushing is in place in the bellcrank ear.

h. Connect floor-routed tail rotor control rod to tail rotor bellcrank. Check that slotted bushing is in place in bellcrank ear.

i. Connect upper end of each cyclic trim actuator with bolt, two washers, bushing, nut, and new cotter pin. Check that slotted bushing is in place in bellcrank ear.

NOTE

Lateral actuator upper hingeline has 0.187-inch ID; longitudinal has 0.25-inch ID.

j. Connect lower ends of tunnel-routed control rods (fig. 11-6).

k. Install collective bungee (para 11-84).

11-55. COLLECTIVE CONTROLS.

11-56. Description — Collective Controls. The collective controls (fig. 11-9) consist of the pilot's collective pitch stick, copilot's collective pitch stick, collective bungee, collective interconnecting torque tube, and tunnel-routed collective mixer control rod. All components, except the sticks and tunnel-routed control rod, are located within the pilot's seat structure.

11-57. COLLECTIVE PITCH STICKS.

11-58. Description — Collective Pitch Sticks. Each collective pitch stick consists primarily of a stick housing, throttle grip and friction mechanism, and gas producer control shafts having right angle (pinion and bevel gear) drive. (See fig. 11-9.) The pilot's collective pitch stick is attached to the collective interconnecting torque tube, between the pilot's and copilot's seats. The copilot's collective pitch stick is mounted on the outboard end of the torque tube at the left side of the seat structure. (Refer to Chapter 9 for maintenance of electrical switches and connectors.)

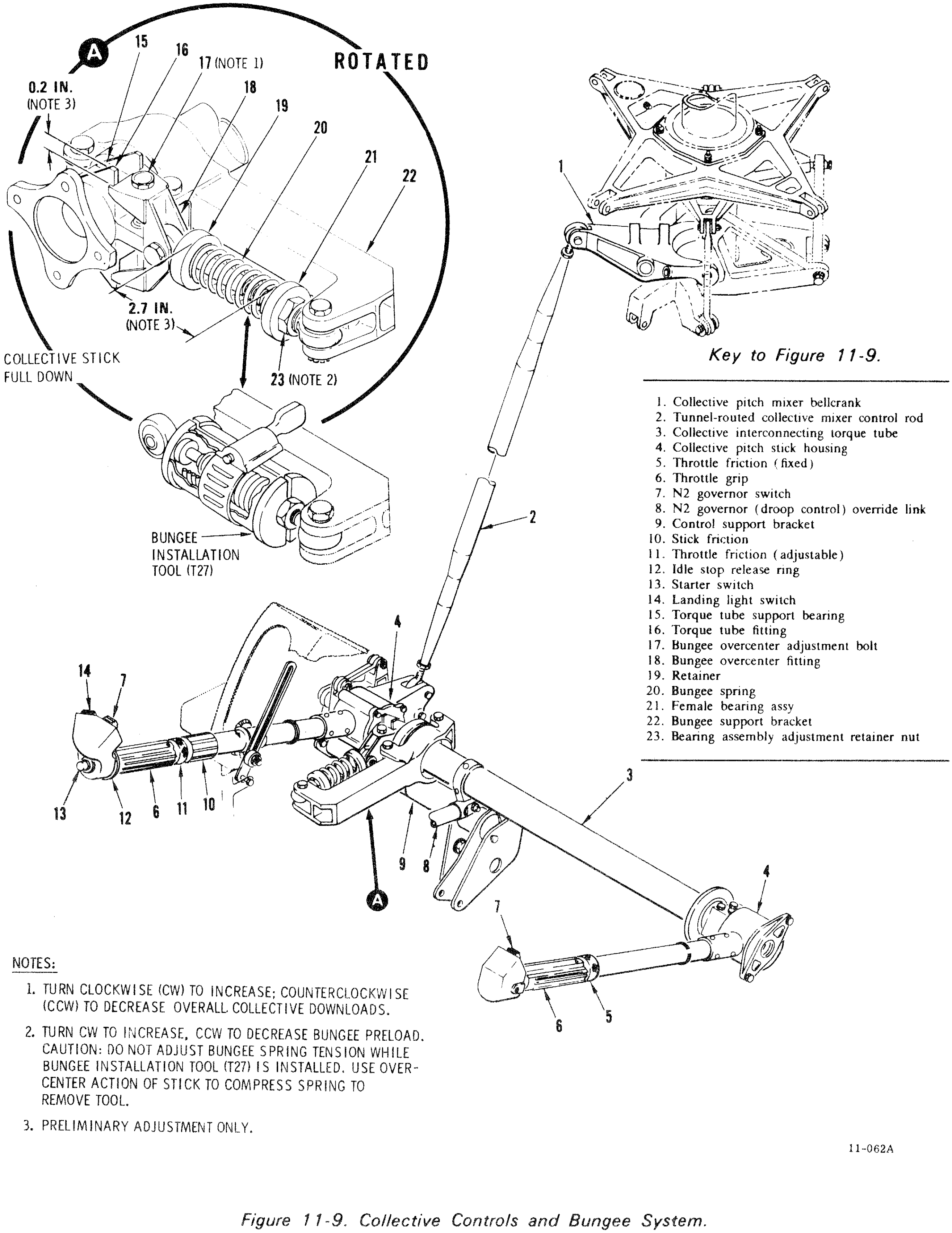
11-59. Operational Check — Pilot's Collective Pitch Stick Friction Mechanism. (See fig. 11-10.) *a.* Remove pilot's collective stick cover (chapter 2).

b. Check that teeth of friction drive gear, idler gear, and driven gear are not stripped or otherwise damaged.

c. Rotate the friction drive gear grip counterclockwise to the maximum friction stop; the gear train must rotate freely. Check the position of the friction gear assembly pin. The pin should be at the highest point of cam (detail B).

d. If the pin is not at the approximate peak of the cam or has overriden the peak (detail B), the friction mechanism must be readjusted (para 11-69). If the pin is in correct position, continue with *e* below.

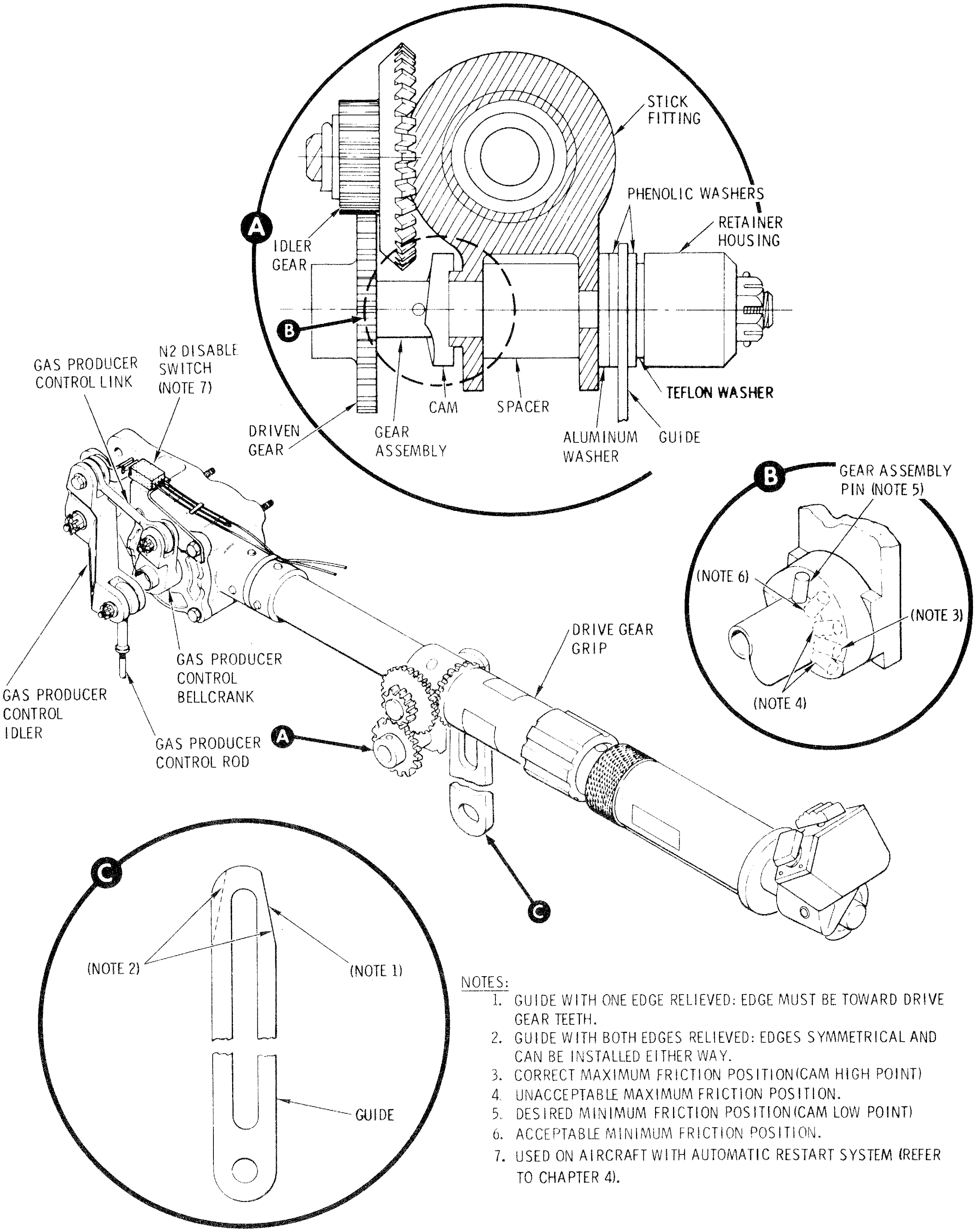
e. Rotate the drive gear grip clockwise to the minimum friction position and release the grip; the gear train must rotate freely. The pin should be at the approximate low point of the cam. See detail B for the



Key to Figure 11-9.

1. Collective pitch mixer bellcrank
2. Tunnel-routed collective mixer control rod
3. Collective interconnecting torque tube
4. Collective pitch stick housing
5. Throttle friction (fixed)
6. Throttle grip
7. N2 governor switch
8. N2 governor (droop control) override link
9. Control support bracket
10. Stick friction
11. Throttle friction (adjustable)
12. Idle stop release ring
13. Starter switch
14. Landing light switch
15. Torque tube support bearing
16. Torque tube fitting
17. Bungee overcenter adjustment bolt
18. Bungee overcenter fitting
19. Retainer
20. Bungee spring
21. Female bearing assy
22. Bungee support bracket
23. Bearing assembly adjustment retainer nut

Figure 11-9. Collective Controls and Bungee System.



NOTES:
1. GUIDE WITH ONE EDGE RELIEVED: EDGE MUST BE TOWARD DRIVE GEAR TEETH.
2. GUIDE WITH BOTH EDGES RELIEVED: EDGES SYMMETRICAL AND CAN BE INSTALLED EITHER WAY.
3. CORRECT MAXIMUM FRICTION POSITION(CAM HIGH POINT)
4. UNACCEPTABLE MAXIMUM FRICTION POSITION.
5. DESIRED MINIMUM FRICTION POSITION(CAM LOW POINT)
6. ACCEPTABLE MINIMUM FRICTION POSITION.
7. USED ON AIRCRAFT WITH AUTOMATIC RESTART SYSTEM (REFER TO CHAPTER 4).

11-068B

Figure 11-10. Pilot's Collective Friction and Throttle Linkage.

range of desired and acceptable minimum friction positions for the pin.

f. With the pin in the low point of the cam, hand-rotate the spring retainer housing. If there is drag on the retainer, there is friction on the guide and collective friction must be readjusted (para 11-70). If the retainer rotates freely, without obvious looseness, the low point friction setting is correct.

g. Check that the relieved edge of the friction guide is toward the geared end of the drive gear grip. If not, the guide must be repositioned.

11-60. Inspection — Collective Pitch Sticks. *a.* Remove pilot's collective stick cover (chapter 2).

b. Remove controls access door and left foot support fairing (chapter 2).

c. Inspect pilot's collective pitch stick attachment to collective interconnecting torque tube for looseness by trying to finger-turn bolts.

d. (See fig. 11-10.) Inspect gas producer control bellcrank, gas producer control link and gas producer control idler for scratches, nicks, dents, corrosion, excessive wear, and any other apparent surface defects.

e. Inspect pitch stick housings (4, fig. 11-9) and stick tubes for cracks, dents and loosened rivets. Use straightedge to inspect tubes for bends and similar distortions.

CAUTION

Set the power selector switch at OFF to prevent accidental ignition of the engine when inspecting throttle grips, *f* through *h* below.

f. Inspect throttle grips for cracks or evidence of binding during rotation. With collective sticks full down, rotate pilot's grip for MAXIMUM RPM and then to IDLE. Gas producer pointer should be at 30 to 31 degrees on the quadrant. If so, rotate copilot's grip for MAXIMUM RPM and then to IDLE. Gas producer pointer must not drop more than 0.08-inch below the 30 degree mark with a maximum of 10 inch-pounds torque on the grip.

g. Inspect idle stop release ring (12) for cracks, and check for binding by sliding ring toward switch housing.

h. Inspect landing light switch, N2 governor switches, and starter switch for secure attachment. Check N2 disable switch for secure mounting on the base of collective stick. (Refer to para 4-244 for switch adjustment.)

i Install controls access door and left foot support fairing.

j. Install pilot's collective stick cover (chapter 2).

Table 11-6. Premaintenance Requirements for Removal of Pilot's Collective Pitch Stick.

Conditions	Requirements
Special Tools	(T27)

11-61. PILOT'S COLLECTIVE PITCH STICK.

11-62. Removal — Pilot's Collective Pitch Stick.

WARNING

Install bungee installation tool (T27) before disconnecting any pilot's collective stick hardware. There is strong bungee spring pressure present in the stick linkage; if suddenly released, spring reaction in the linkage can cause personal injury, or parts damage. Raise the collective stick and use the torque tube over-center action to compress the bungee spring before installing the tool.

a. Remove both pilot's seat bottoms, pilot's collective stick cover and the controls access door (chapter 2).

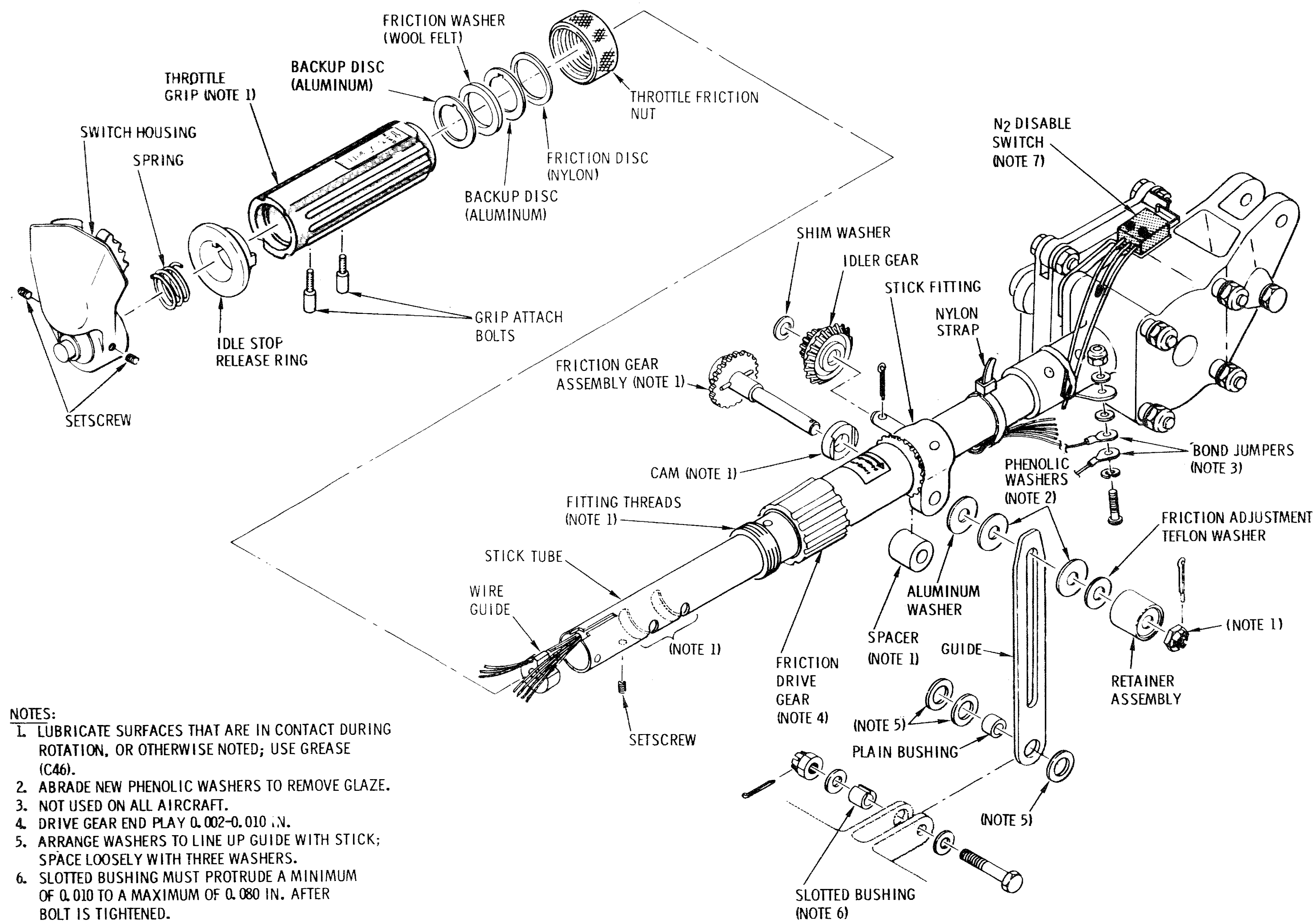
b. Disconnect pilot's collective pitch stick electrical plug. On a stick without the plug, disconnect the stick wiring at the wiring terminal block in the electrical control console. Disconnect wiring from ground terminal connection.

c. Remove attaching hardware that secures the guide to the seat structure bracket (fig. 11-11).

d. Disconnect collective mixer tunnel-routed control rod (fig. 11-6).

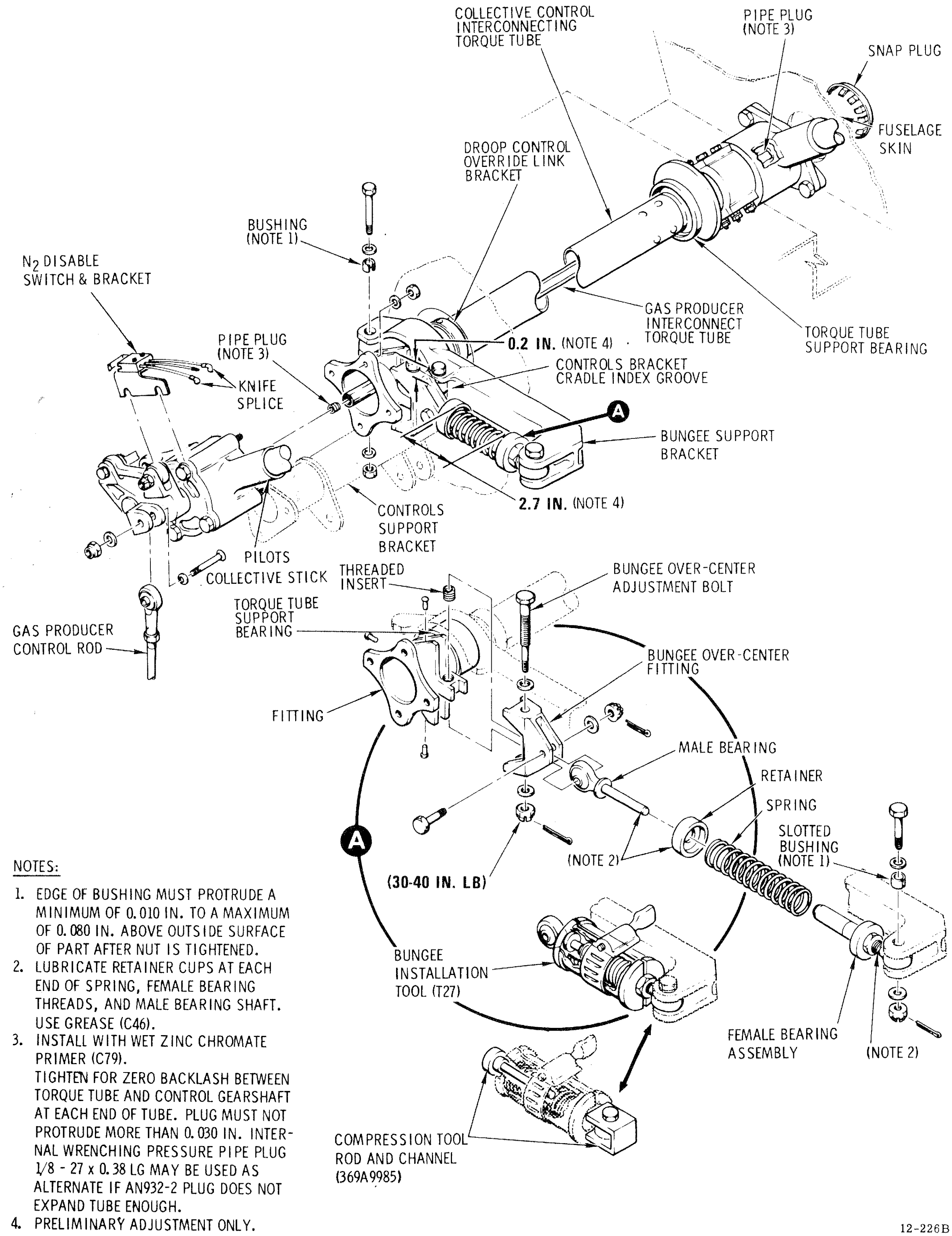
e. Disconnect gas producer control rod from idler (fig. 11-12). Disconnect N2 disable switch at wire splices (fig. 11-11).

f. Remove the four nuts and washers that attach the collective stick housing to the collective torque tube and remove the stick assembly (fig. 11-13).



NOTES:
1. LUBRICATE SURFACES THAT ARE IN CONTACT DURING ROTATION, OR OTHERWISE NOTED; USE GREASE (C46).
2. ABRADE NEW PHENOLIC WASHERS TO REMOVE GLAZE.
3. NOT USED ON ALL AIRCRAFT.
4. DRIVE GEAR END PLAY 0.002-0.010 IN.
5. ARRANGE WASHERS TO LINE UP GUIDE WITH STICK; SPACE LOOSELY WITH THREE WASHERS.
6. SLOTTED BUSHING MUST PROTRUDE A MINIMUM OF 0.010 TO A MAXIMUM OF 0.080 IN. AFTER BOLT IS TIGHTENED.
7. PART OF AUTOMATIC RESTART SYSTEM (CHAPTER 4).

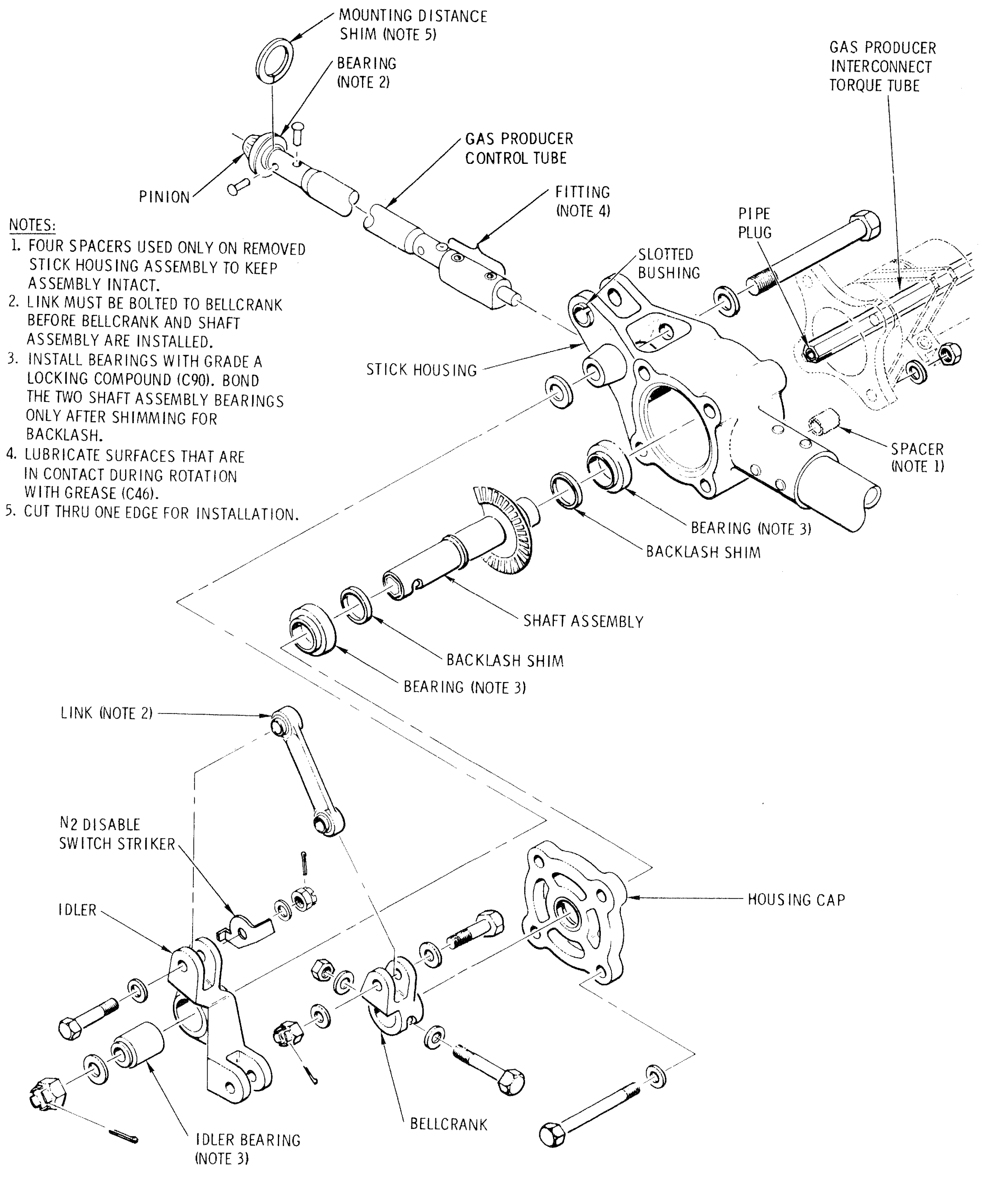
Figure 11-11. Pilot's Collective Stick, Switch, Throttle, and Friction Controls.



NOTES:

1. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.080 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. LUBRICATE RETAINER CUPS AT EACH END OF SPRING, FEMALE BEARING THREADS, AND MALE BEARING SHAFT. USE GREASE (C46).
3. INSTALL WITH WET ZINC CHROMATE PRIMER (C79). TIGHTEN FOR ZERO BACKLASH BETWEEN TORQUE TUBE AND CONTROL GEARSHAFT AT EACH END OF TUBE. PLUG MUST NOT PROTRUDE MORE THAN 0.030 IN. INTERNAL WRENCHING PRESSURE PIPE PLUG 1/8 - 27 x 0.38 LG MAY BE USED AS ALTERNATE IF AN932-2 PLUG DOES NOT EXPAND TUBE ENOUGH.
4. PRELIMINARY ADJUSTMENT ONLY.

Figure 11-12. Collective Torque Tube, Gas Producer Torque Tube, and Collective Bungee.



12-227B

Figure 11-13. Assembly of Pilot's Collective Stick, Gas Producer Linkage.

NOTE

If disassembly is not intended, install four spacers (fig. 11-13) and reinstall the washers and nuts on the protruding bolts to keep the assembly intact.

11-63. Disassembly — Pilot's Collective Pitch Stick. (See fig. 11-11.) *a.* Remove setscrews that secure switch housing.

b. Cut nylon strap or twine that secures electrical wiring to stick tube. Push wire slack into stick, carefully pull housing and wiring from end of stick tube and disconnect wiring from switches.

c. Remove setscrew and wire guide from forward end of stick. Tie twine (C112) on each wire bundle to aid reassembly and remove wiring.

d. Remove spring and idle stop release ring.

e. Remove grip attach bolts. Slide throttle grip, friction washer and discs, and friction nut from stick tube.

f. Remove friction mechanism and guide from stick fitting.

g. Remove idler gear from stick fitting.

h. (See fig 11-13.) Disconnect the link from the idler and remove N2 disable switch striker.

i. Remove the idler from the housing. Do not remove idler bearing unless replacement is necessary.

j. Remove the four bolts and washers that attach housing cap to socket housing. Remove N2 disable switch and bracket. Rotate the shaft assembly until gear tooth cutout clears pinion (bellcrank ears at approximately 195 degrees); then separate the bellcrank, housing cap and shaft assembly from the housing.

k. Remove bellcrank from shaft assembly to separate housing cap from shaft. Do not press bearings from cap or housing unless replacement is necessary. Keep backlash shims with shaft assembly.

l. Remove link from bellcrank.

m. Remove gas producer control tube through access hole at back of stick housing. Remove mounting distance shim only if necessary to adjust pinion gear mounting distance. (fig. 11-14). Remove pinion and bearing from control tube only if replacement is necessary.

11-64. Inspection — Pilot's Collective Pitch Stick. *a.* Inspect bearings for binding or looseness.

b. Inspect all gears for cracks, and chipped or broken teeth.

c. Inspect stick tube and gas producer control tube for corrosion, deformation, and loose rivets.

d. Inspect throttle friction nut and associated parts for wear, corrosion, cracks and operation.

11-65. Repair — Pilot's Collective Pitch Stick. *a.* Replace bearings if corroded, excessively worn, or if the outer or inner races of bearings rotate on mating surfaces and locking compound is inadequate to prevent rotation. Install new bearings with grade A locking compound (C90). Check that replacement bearing is properly seated in its bore.

b. Replace any part that is cracked.

c. Replace entire link (fig. 11-13) if bearings are defective; link bearings are not replaceable.

d. Replace a pinion gear or shaft assembly bevel gear when correct backlash cannot be maintained because of tooth irregularities caused by wear or deformmtion.

e. Replace loose rivets in the gas producer control tube.

f. Replace a friction drive gear or idler gear (fig. 11-15) having cracked, chipped or broken teeth.

g. Replace friction drive gear if it does not rotate freely on stick tube, or if the rotating stop (detail B, fig. 11-15) is damaged. Drill out rivets that attach throttle friction nut fitting to the stick to remove gear.

NOTE

Before replacing a drive gear for binding, clean and then relubricate the ID with grease (C46). Reinstall gear on stick tube and recheck for free rotation. If gear still binds it must be replaced.

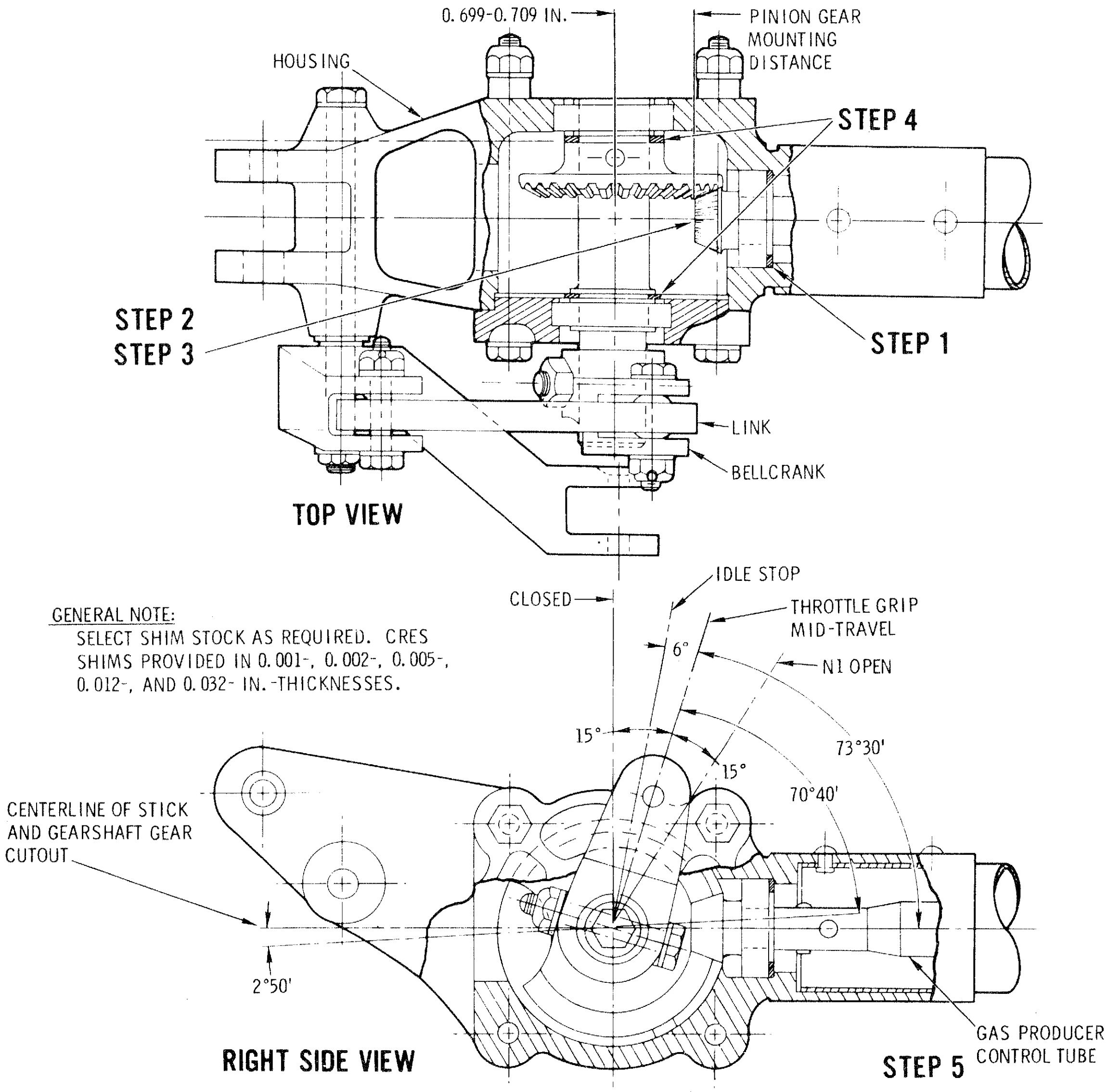
h. Replace the phenolic friction washers that slide against the guide if worn to less than 0.031-inch thickness.

i. Replace the friction gear assembly (fig. 11-15) if the driven gear has cracked, chipped or broken teeth, or if the cam pin is bent.

j. Replace a damaged idle stop release ring. The ring must slide freely on the tube and in the throttle grip.

11-66. Reassembly — Pilot's Collective Pitch Stick. (See fig. 11-13.) *a.* Check gearshaft bearings in housing cap and housing for security of outer races. Use grade A locking compound (C90) to install replacement bearings. Use care to prevent compound from entering bearings, and make sure that each bearing is seated against its bore shoulder.

b. Using grease (C46) lubricate the stick tube interior where the gas producer control tube fitting makes contact.



GENERAL NOTE:
SELECT SHIM STOCK AS REQUIRED. CRES SHIMS PROVIDED IN 0.001-, 0.002-, 0.005-, 0.012-, AND 0.032- IN.-THICKNESSES.

STEP 1: INSTALL MOUNTING DISTANCE SHIMS FOR 0.699-0.709 IN. PINION GEAR MOUNTING DISTANCE. AFTER SHIMMING IS CORRECT, INSTALL BEARING IN HOUSING WITH GRADE A LOCKING COMPOUND (C90).
NOTE: SHIMS ARE CUT THROUGH ONE EDGE FOR INSTALLATION.

STEP 2: APPLY 10 LB MINIMUM LOAD TO END OF PINION DURING CHECK OF MOUNTING DISTANCE DIMENSION.
NOTE: TIGHTENING THROTTLE FRICTION NUT APPLIES SUFFICIENT LOAD.

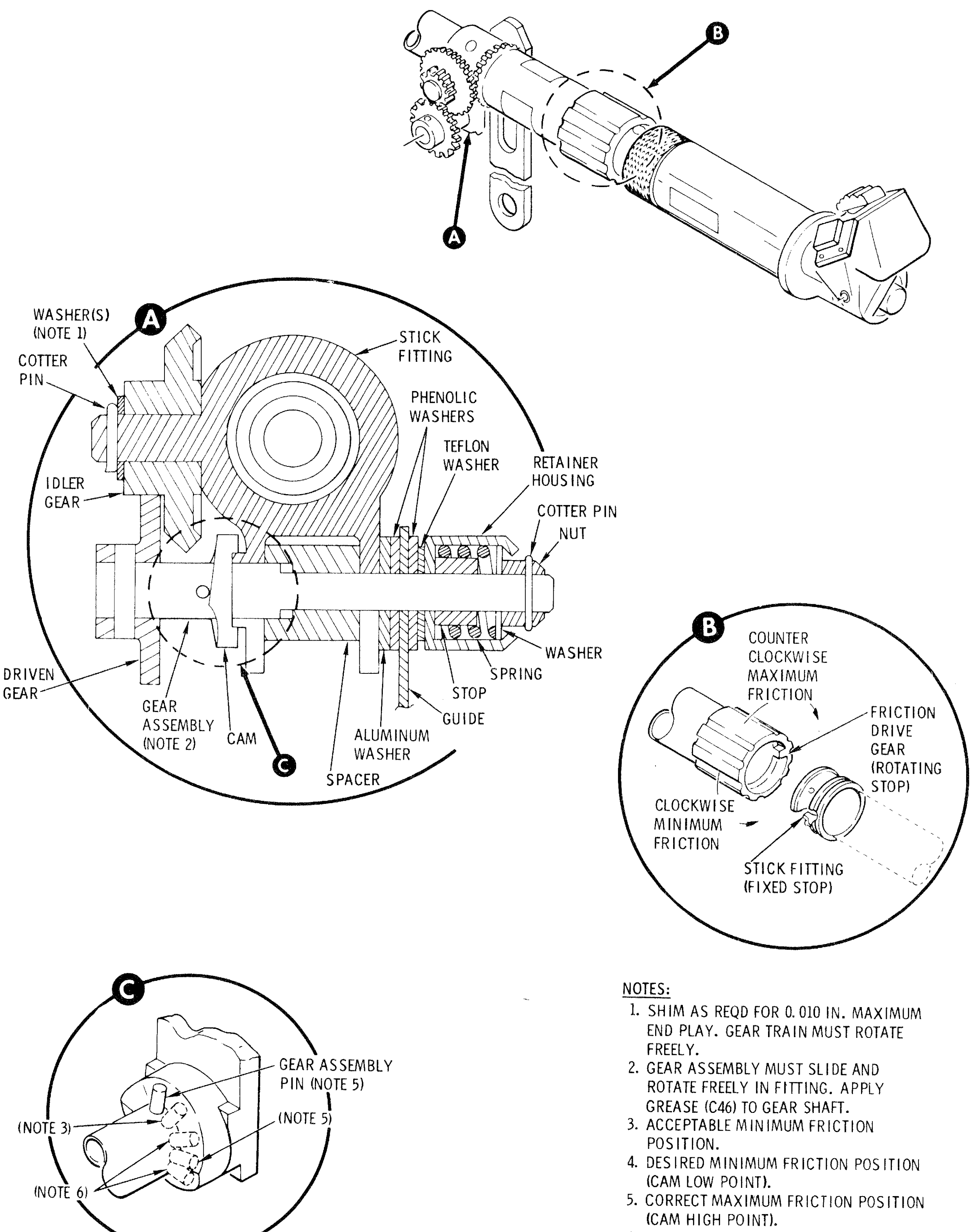
STEP 3: POSITION THROTTLE GRIP AT MID-TRAVEL.
NOTE: ONE PINION GEAR TOOTH EQUALS 5°38'.

STEP 4: INSTALL BACKLASH SHIMS TO OBTAIN 0.003 IN. MAXIMUM BACKLASH
CAUTION: HOUSING AND CAP BEARINGS MUST BE SEATED AGAINST BORE SHOULDER. GEARS MUST NOT BIND, AND PINION MOUNTING DISTANCE MUST BE CORRECT.

STEP 5: AFTER SHIMMING IS CORRECT, INSTALL GEARSHAFT BEARINGS WITH GRADE A LOCKING COMPOUND (C90). INSTALL GEARSHAFT (WITH LINK ATTACHED TO BELLCRANK) SO THAT POSITION OF BELLCRANK, AND CENTERLINE OF GEARSHAFT GEAR CUTOUT ARE AS SHOWN. CENTERLINE OF GEARSHAFT CUTOUT SHOULD MATCH CENTERLINE OF STICK.
NOTE: CENTERLINE OF GEARSHAFT GEAR CENTER TOOTH IS 2°50' OFF THE CUTOUT CENTERLINE.

12-228C

Figure 11-14. Adjustment of Gas Producer Drive Mounting Distance and Backlash (Pilot's Collective Stick).



NOTES:

1. SHIM AS REQD FOR 0.010 IN. MAXIMUM END PLAY. GEAR TRAIN MUST ROTATE FREELY.
2. GEAR ASSEMBLY MUST SLIDE AND ROTATE FREELY IN FITTING. APPLY GREASE (C46) TO GEAR SHAFT.
3. ACCEPTABLE MINIMUM FRICTION POSITION.
4. DESIRED MINIMUM FRICTION POSITION (CAM LOW POINT).
5. CORRECT MAXIMUM FRICTION POSITION (CAM HIGH POINT).
6. UNACCEPTABLE MAXIMUM FRICTION POSITION.

12-229C

Figure 11-15. Adjustment of Collective Pitch Stick Friction.

c. (See fig. 11-11.) Attach twine, routed through the stick tube during disassembly, and thread the electrical wire bundles from the plug through the exit hole, throttle friction nut, friction washers and discs, throttle grip, idle stop release ring, and the spring. Pull wiring out through front end of stock tube. Remove twine.

d. Install wire guide so that it divides the switch wiring, three wires in one cutout and four in the other. Align guide with matching hole in stick tube and install setscrews. **WHEN TIGHTENED, SETSCREW MUST BE AT LEAST FLUSH AND NOT RECESSED MORE THAN 0.010 INCH BELOW OUTER SURFACE OF TUBE.**

e. Install gas producer control tube through the access hole in back of stick housing, and through stick tube until control tube fitting engages wire guide bore. Reinstall mounting distance shims at forward side of pinion bearing if removed during disassembly. (One edge of shims must be cut for installation.)

f. Establish gas producer control tube pinion mounting distance as shown in figure 11-14.

g. After mounting distance shimming is correct, remove gas producer control tube and apply grade A locking compound (C90) to OD of pinion bearing outer race, and the stick housing bore. Reinstall control tube, and check that compound does not enter the bearing and that the bearing outer race and shims are firmly seated against housing bore.

h. Using grease (C46), lubricate ID of the throttle friction nut threaded surfaces. Install friction nut, discs and washers.

i. Using grease (C46), lubricate ID of throttle grip and install grip on stick tube. Align grip and has producer control tube fitting and install grip attach bolts. **WHEN TIGHTENED, BOLTS MUST BE AT LEAST FLUSH AND NOT MORE THAN 0.010 INCH BELOW OUTER SURFACE OF GRIP.**

j. Establish gas producer control tube pinion and shaft assembly bevel gear backlash as shown in figure 11-14.

k. After backlash shimming is correct, remove housing cap and gearshaft.

l. Lubricate teeth of pinion gear and shaft assembly bevel gear with grease (C46).

m. Apply grade A locking compound (C90) to gearshaft and ID of bearings. With throttle grip at mid-travel, install shaft assembly gear as shown in figure 11-14.

n. Install housing cap on shaft assembly and seat with hand pressure while slowly rotating the grip back and forth. Secure cap to side of housing and cover to back of housing.

o. Using solder (C93), connect wiring to stick switches. Pull electrical slack out through exit holes in stick tube, position release ring and spring, and install switch housing with setscrews. Using a nylon strap, or twine (C112), secure electrical wiring to stick tube approximately one inch aft of the exit holes.

p. Install friction cam and friction gear assembly on stick fitting (fig. 11-11). Temporarily mesh idler gear and friction gear assembly, and rotate friction drive gear counterclockwise to maximum stop (detail B, fig. 11-15).

q. Hold drive gear at stop and remesh idler gear and friction gear assembly so that gear assembly pin is at the high point of cam (detail C, fig. 11-15).

r. Use shim washers sufficient to **LIMIT GEAR END PLAY TO A MAXIMUM OF 0.010 INCH** (detail A, fig. 11-15), and secure idler gear with new cotter pin.

s. Assemble remainder of friction mechanism as shown in figures 11-11 and 11-15.

Table 11-7. Premaintenance Requirements for Installation of Pilot's Collective Pitch Stick.

Conditions	Requirements
Special Tools	(T27)

11-67. Installation — Pilot's Collective Pitch Stick.

CAUTION

Check that bungee installation tool (T27) is properly installed.

a. Remove the four nuts, washers and spacers from the housing assembly mounting bolts (fig. 11-13).

b. Ensure that pilot's and copilot's throttle grips are in the mid-travel position.

c. Rotate bellcrank (fig. 11-13) back and forth slightly to engage housing assembly gearshaft on hexagonal end of gas producer interconnect torque tube. (Bellcrank centerline, N1 pinion gear centerline tooth-space and pilot's throttle shall be as shown in figure 11-14.) Align housing with collective torque tube and secure stick to torque tube.

d. Rotate pilot's throttle and check for zero backlash between the gas producer interconnect torque tube and the hexagonal ID of the drive gearshaft in the stick housing. Eliminate any backlash by removing stick and tightening the pipe plug in the end of torque tube (fig. 11-13); zero backlash is required at both ends of the tube. After plug is tightened, reinstall stick as in *c* above. Recheck for backlash.

e. Install N2 disable switch and bracket on two upper bolts in housing cap.

f. Connect gas producer control rod to idler (fig. 11-12) and install N2 disable switch striker (fig. 11-13).

g. Connect collective mixer tunnel-routed control rod (fig. 11-6).

h. Position friction guide in the seat structure bracket with plain bushing, slotted bushing, and three washers loosely spaced between bracket ears so that guide is in line (parallel) with stick (fig. 11-11). Install bolt, two washers, nut and new cotter pin.

i. As applicable, connect electrical plug or connect stick wiring to the terminal block in the electrical control console. Connect wiring to ground terminal. Connect N2 disable switch wire splices and replace tie straps.

NOTE

If a replacement pilot's collective pitch stick assembly is being installed and the wire numbers do not correspond to those of the original stick assembly, refer to paragraph 9-214 instructions.

j. Remove bungee installation tool.

k. Adjust N2 disable switch according to chapter 4. Check engine controls for correct operation and check that wiring is not fouled when collective stick is raised or lowered.

l. Reinstall the pilot's collective stick cover, controls access door, and both pilot's seat bottoms (chapter 2).

11-68. Adjustment — Pilot's Collective Pitch Stick. Perform an operation check of collective stick friction and adjust friction mechanism as necessary (para 11-70).

11-69. COLLECTIVE PITCH STICK FRICTION MECHANISM ADJUSTMENT.

11-70. Adjustment — Collective Pitch Stick Friction Mechanism. The collective stick friction mechanism (fig. 11-15) is designed so that positive locking of the pitch stick cannot be obtained at the maximum friction point. Safety of flight considerations requires that the pilot be able to instantly overcome the established collective pitch stick position, without changing the friction adjustment, in the event of a power failure. There is no suitable check that the pilot can make, with the aircraft on the ground, to determine if maintenance adjustment of collective friction is correct. This is due to the large force application necessary to overcome the collective bungee and blades resting on the droop stops. If stick friction is inadequate during flight, a maintenance check should be performed. Once friction adjustment is determined to be correct at the low friction point, any further mechanical adjustment to alter (increase) friction at the low point can cause the mechanism gears to be stripped when the grip is fully rotated for maximum friction.

CAUTION

For the collective friction mechanism to be correctly adjusted for maximum friction, three actions must occur at the same time: The drive gear must contact the stick fitting fixed stop (detail B, fig. 11-15); the spring retainer housing washer must contact the retainer internal stop (detail A); and the gear assembly pin must be at the approximate peak of the cam (detail C). Do not overtighten the retainer nut so that the retainer washer contacts the retainer stop before the friction drive gear reaches the stick fitting stop. Adjustment that does not produce approximately simultaneous contact will allow excessive additional rotation of the drive gear grip and cause the composition nylon gears to be stripped.

a. Rotate the friction drive gear counterclockwise to the maximum friction position.

b. Remove cotter pin, nut, retainer assembly, teflon washer, and one phenolic washer.

c. Slide the gear assembly out far enough to disengage it from the idler gear. Use care to keep the other washers and spacer from dropping.

d. Remesh the gear assembly with the idler gear to position the pin on the highest part of the cam (detail C). Check that the drive gear is still at the maximum friction stop position.

e. If the gear assembly does not correctly position the pin, reposition the idler gear on the drive gear.

NOTE

The gear ratio of idler gear-to-gear assembly is 3:1 which permits fine adjustment. When repositioning the idler gear, be sure to reinstall the same number of washers that were removed from between the cotter pin and idler. These washers limit gear end play to a maximum of 0.010 inch. When reassembled, the gear train must rotate freely.

f. Reinstall phenolic washer, teflon washer, retainer assembly and nut.

g. With the drive gear still at the maximum friction position, tighten the retainer nut until it just bottoms on retainer stop.

h. Rotate the friction drive gear clockwise to the minimum friction stop position and release the grip. Check the following:

(1) If the pin has overridden the low point of the cam as shown in detail C (desired minimum friction position), and grip-to-stick friction prevents springback of the pin to the cam low point, manually turn the grip and move pin to low point.

NOTE

A drive gear that does not rotate freely (binds) on the collective stick tube should be replaced (para 11-65).

(2) With pin in cam low point, manually rotate the retainer assembly; it should turn freely on the shaft of the gear assembly with only light finger pressure. Adjust retainer nut to the nearest castellation that produces zero friction (no drag on the retainer assembly during rotation). Install new cotter pin.

NOTE

If undesirable collective forces (that is, "light," "heavy" or "creeping" collective) are reported to exist in the collective stick during flight, do not attempt to compensate by an increase in collective friction. The reported condition should be corrected elsewhere in the control system. (Refer to collective rigging and bungee adjustment procedures.)

11-71. COLLECTIVE CONTROL INTERCONNECTING TORQUE TUBE.

11-72. Description — Collective Control Interconnecting Torque Tube. The collective interconnecting torque tube (3, fig. 11-9) consists of the tube, a bungee bracket, bungee fitting, and a tube support bearing. The torque tube interconnects the pilot's and copilot's collective pitch sticks. The torque tube is supported at the left end by a bearing installed in the seat structure, and at the right end by the controls support bracket.

11-73. Inspection — Collective Control Interconnecting Torque Tube. *a.* Remove pilot's collective stick cover (chapter 2).

b. Remove controls access door and left foot support fairing (chapter 2).

c. Inspect collective interconnecting torque tube (3, fig. 11-9) for cracks, and bends or similar distortions. Use straightedge to make inspection.

d. Inspect bungee bracket (22), and fittings (16 and 18) for condition.

e. Inspect all accessible rivets for looseness, and support bearings for secure attachment and evidence of binding.

f. Install controls access door and left foot support fairing.

g. Install pilot's collective stick cover (chapter 2).

11-74. Removal — Collective Control Interconnecting Torque Tube. *a.* Remove collective bungee (para 11-80).

b. Remove pilot's collective pitch stick (para 11-62).

c. Remove copilot's collective pitch stick (para 11-89).

d. Remove gas producer interconnect torque tube by carefully sliding it out of collective torque tube (fig. 11-13).

e. Disconnect aft end of droop control override link from torque tube droop control bracket (detail A, fig. 11-7).

f. Remove two bolts, four washers, and two nuts that secure bungee support bracket and torque tube to controls support bracket (fig. 11-12). Slide bungee bracket aside and remove controls bracket cradle cap.

g. Remove collective control torque tube by sliding it approximately 3 inches toward the right side of the seat structure to disengage left end from support bearing. Provide support at seat structure lightening hole and carefully withdraw torque tube from structure.

11-75. Repair — Collective Control Interconnecting Torque Tube. *a.* Replace loose or binding torque tube support bearings. Drill out rivets that secure retainer and doubler to seat structure to replace left support bearing. Drill out three rivets that secure right end fitting to torque tube to replace right support bearing. Pick up existing hole patterns and install new bearings.

b. Replace torque tube if it is cracked or has elongated holes.

c. Replace a bent or stripped bungee adjusting bolt. **TORQUE NUT TO 30 — 40 INCH-POUNDS** and install new cotter pin.

11-76. Installation — Collective Control Interconnecting Torque Tube. *a.* Insert torque tube into lightening hole in seat structure. Use care when inserting tube into place to prevent scraping on edge of hole, and position left end of tube in fixed bearing bore. Rotate torque tube slowly back and forth until right end bearing rests in cradle of controls support bracket (fig.11-12).

b. Check that two slotted bushings are in place in upper lugs of bungee bracket. Position torque tube, controls bracket cradle cap and bungee bracket on controls support bracket. Check that cap-to-cradle index grooves are matched (fig. 11-12). Install two bolts, four washers, and two nuts. Apply a thin layer of grease (C46) to sliding surfaces of bungee overcenter fitting.

c. Install droop control override link to torque tube droop control bracket (detail B, fig. 11-7).

d. Install gas producer interconnect torque tube by sliding it carefully into place in collective torque tube.

e. Install copilot's collective pitch stick (para 11-94).

f. Install pilot's collective pitch stick (para 11-67).

g. Install collective bungee (para 11-84).

11-77. COLLECTIVE BUNGEE.

11-78. Description — Collective Bungee. The collective bungee installation (detail A, fig. 11-9) consists of a male bearing assembly, female bearing assembly, spring, and retainer. This unit attaches between the bungee overcenter fitting and bungee bracket of the collective interconnecting torque tube. The collective bungee is provided to help maintain selected collective pitch stick position in flight by counteracting forces that are fed back to the collective pitch sticks; blade pitching moments; rotor head strap pack torsion when collective pitch stick is raised or lowered from mid-position (strap pack neutral position); combined imbalance of forces in the controls system. The flight characteristics of the aircraft are such that collective forces are relatively low during most of stick travel from low pitch toward high pitch. At a point near full pitch the stick forces reverse and become heavy. The purpose of the adjustable bungee and overcenter bracket attachment is to counteract these forces so that collective stick loads are relatively constant throughout the full range of travel. There are two adjustments available to establish or correct collective flight loads. Adjustment of the collective bungee spring (20) will correct a variation in collective load from low pitch (level flight) to high pitch (climb). Setting of overcenter bolt (17) to raise or lower bungee fitting (18) will cause an overall reduction or increase of collective forces in both low pitch (level flight) and high pitch (climb).

11-79. Inspection — Collective Bungee (Installed). *a.* Remove pilot's collective stick cover (chapter 2).

b. Check attaching hardware for looseness by trying to fingerturn bolts.

c. Inspect collective bungee spring (20, fig. 11-9), bungee spring bracket (22), and the control support bracket (9) for corrosion, cracks and other apparent surface defects.

d. Install pilot's collective stick cover (chapter 2).

Table 11-8. Premaintenance Requirements for Removal of Collective Bungee.

Conditions	Requirements
Special Tools	(T27)

11-80. Removal — Collective Bungee. *a.* Remove pilot's collective stick cover from seat structure.

b. Raise the collective stick and use the torque tube overcenter action to compress the bungee spring until bungee installation tool (T27) will fit over the spring retainers (fig. 11-12). Secure tool halves in place with clamp.

c. With stick in overcenter position, remove cotter pin, nut, washer, and bolt that secure male bearing to bungee overcenter fitting.

d. Remove cotter pin, nut, two washers, and bolt that attach female bearing end of bungee to the bungee bracket.

e. Remove collective bungee and installation tool as a unit.

CAUTION

Remove installation tool only as directed in paragraph 11-81.

Table 11-9. Premaintenance Requirements for Disassembly of Collective Bungee.

Conditions	Requirements
Special Tools	(T34)

11-81. Disassembly — Collective Bungee. *a.* Remove male bearing from bungee clamped in compression by the installation tool.

b. Install bungee compression tool rod and channel (T34) on compressed bungee. Insert a 1/4-inch bolt through channel and female bearing rod end (detail A, fig. 11-12).

c. Line the jaws of a bench vise with masking tape or similar nonskid material.

d. Place compression tool and bungee in the vise so that the vise jaws make contact with tool rod and channel.

e. Compress spring by closing vise until the bungee clamp may be removed from bungee.

f. Slowly open vise until bungee spring is fully extended.

g. Remove retainer, spring, and female bearing assembly.

NOTE

Do not disassemble female bearing assembly. Replace as a unit if defective.

11-82. Inspection — Collective Bungee (Removed). Replace any bungee parts in questionable condition.

a. Inspect spherical bearings in male bearing and female assembly for evidence of binding, corrosion, and galling.

b. Check female bearing assembly threads for damage.

c. Inspect spring for evidence of deformation. **FREE LENGTH OF SPRING MUST BE 4.50 ±0.060 INCHES.**

d. Check male bearing rod for cracks, evidence of binding, corrosion, and deformation.

Table 11-10. Premaintenance Requirements for Reassembly and Installation of Collective Bungee.

Conditions	Requirements
Special Tools	(T27) (T34)
Consumable Materials	(C46)

11-83. Reassembly — Collective Bungee. *a.* Use a bench vise having jaws that are lined with a non-skid material.

b. Lubricate cups of spring retainers and threads and bore of female bearing assembly with grease (C46).

c. Assemble retainer, spring, and female bearing assembly. Install compression tool rod and channel (T34) in position. Insert a 1/4-inch bolt through channel and female bearing rod end (detail A, fig. 11-12).

d. Position assembly in jaws of bench vise. Close the jaws to compress bungee spring until bungee installation tool (T27) will fit over the retainers. Secure tool halves in place with clamp.

e. Open bench vise jaws slowly, and remove the bungee compression tool rod and channel.

f. Install male bearing into place in compressed bungee.

11-84. Installation — Collective Bungee. *a.* Position the collective bungee, compressed in installation tool (para 11-83) so that the female bearing assembly is forward and aligns with mating hole in the bungee bracket (detail A, fig. 11-12). Check that slotted bushing for bracket lug is in place, install bolt, two washers, nut, and new cotter pin.

b. Position male bearing to align with mating hole in bungee overcenter fitting. Install bolt, washer, nut, and new cotter pin.

CAUTION

Do not turn the female bearing assembly spring retainer while the bungee installation tool is under spring load.

c. Raise the collective stick and use the torque tube overcenter action to compress spring until bungee installation tool can be removed.

d. Make preliminary adjustment (para 11-86). Readjust bungee after flight evaluation, if required.

11-85. COLLECTIVE BUNGEE ADJUSTMENT.

11-86. Preliminary Adjustment — Collective Bungee. Preliminary collective bungee adjustment should only be necessary when there is no available flight history of the aircraft or if the bungee has been disassembled.

a. Remove pilot's collective stick cover (chapter 2).

NOTE

Rotation of the adjustment bolt head is the only mechanical operation required to adjust the overcenter position of the bungee fitting. The spring forces produced by adjustment of the bungee system can only be evaluated during flight.

b. (See fig. 11-9.) Turn the bungee overcenter adjustment bolt (17) to position the top of bungee fitting (18) 0.2 inch down from the top of torque tube fitting (16).

CAUTION

During the following step *c*, if the spring bottoms, the resulting load can break the bungee support bracket.

c. With collective full down, turn adjustment retainer nut (23) to get a dimension of 2.7 inches between the facing edges of the spring retainers (19 and 21). Check that spring will not bottom at overcenter position by slowly cycling collective stick through full travel.

d. Reinstall pilot's collective stick cover (chapter 2).

11-87. Adjustment — Collective Flight Loads. Adjustment of the bungee system is only permissible when the aircraft is on the ground.

a. Collective load forces are affected by any adjustment made in the main rotor system. Observe the following:

(1) Bungee adjustment should only be made when the main rotor blades are in track and autorotation rpm is established (para 5-51).

(2) Ensure that collective stick friction is correctly adjusted (para 11-69).

(3) Perform flight evaluation of collective forces with takeoff weight of 2200 pounds, **N2 AT 101% AND ZERO COLLECTIVE STICK FRICTION.**

b. Before test flight, establish the half-way point of the collective friction adjustment grip (drive gear) as follows:

(1) Using a soft, colored pencil, index the collective friction adjustment grip at zero friction.

(2) Roll on maximum friction (approximately 275 degrees rotation). Note travel and reset friction grip half-way. Hold in this position.

(3) Using a dot of white paint or similar method, temporarily mark the top of the friction grip and the adjacent exposed area of the collective stick. The friction half-way index mark should be visible to the pilot.

CAUTION

After each bungee adjustment is made, the collective stick shall be slowly cycled through its full travel to assure that there is no binding or restriction of stick motion. Check that the spring does not bottom at the overcenter position. If spring bottoms, the resulting load can break the bungee support bracket.

c. The bungee is to be adjusted, as necessary, to avoid excessive download or upload at 60 knots and 120 knots level flight, as outlined below. The aircraft MUST be landed after each flight check, before making corrective adjustment.

WARNING

Use all necessary precautions to prevent the possible entry of any foreign objects into the controls linkage exposed by removal of the pilot's collective stick cover to make adjustment of the bungee system.

(1) Fly the helicopter and feel collective force at 60 knots, level flight.

(2) Feel collective force in climb at 60 knots, 75 psi torque.

(3) If download in climb is greater (heavy collective) than download in level flight, turn bungee adjustment retainer nut clockwise to increase preload in the bungee system (fig. 11-9). Observe CAUTION preceding.

(4) If download in climb is less (light collective)

than download in level flight, turn bungee adjustment retainer nut counterclockwise to decrease preload in the bungee system.

(5) When both the level flight and climb loads are relatively constant, increase overall collective download by turning overcenter adjustment bolt clockwise (fig. 11-9). Decrease downloads by turning adjustment bolt counterclockwise.

NOTE

Rotation of the adjustment bolt head is the only mechanical operation required to adjust the overcenter position of the bungee fitting.

(6) After minimum and balanced collective force has been obtained, not more than one-half the available collective stick friction (friction grip at half-way index) must cancel collective forces during the flight test at 120 knots level flight.

(7) If friction applied in (6) above is insufficient to cancel collective forces, repeat the flight loads adjustment procedure.

11-88. COPILOT'S COLLECTIVE PITCH STICK.

11-89. Removal — Copilot's Collective Pitch Stick. (See fig. 11-16.) *a.* Remove outboard collective stick cover.

b. Disconnect electrical plug.

c. Remove nuts, washers and bolts that connect stick housing to torque tube.

d. Remove stick by sliding it outboard and off torque tube.

11-90. Disassembly — Copilot's Collective Pitch Stick. *a.* Cut the nylon strap or twine that secures wiring to stick tube (fig. 11-16).

b. Remove setscrews that secure the switch housing.

c. Push wire slack into stick and carefully pull switch housing from end of stick tube; disconnect wiring from N2 switch.

d. Remove grip attach bolts and grip.

e. Remove adjusting nut pin and throttle end play adjusting nut only if stick tube replacement is necessary, or if end play must be adjusted (para 11-93).

f. Remove setscrew and wire guide. Tie twine (C112) on end of wire bundle to aid reassembly, and then remove the wiring.

g. Remove cap from stick housing. Do not remove bearing from cap unless replacement is necessary.

h. Remove gearshaft assembly. Keep backlash shims with gearshaft and do not remove bearing from stick housing unless replacement is necessary.

i. Remove cover from back of stick housing.

j. Remove gas producer control tube through access hole in back of stick housing. Remove mounting distance shim only if necessary to adjust pinion gear mounting distance (fig. 11-17). Remove pinion and bearing from control tube only if replacement is necessary.

11-91. Inspection — Copilot's Collective Pitch Stick. *a.* Inspect bearings for binding or looseness.

b. Check all gears for cracks, and chipped or broken teeth.

c. Inspect stick tube and gas producer control tube for corrosion, deformation and loose rivets.

11-92. Repair — Copilot's Collective Pitch Stick. *a.* Replace bearings if locking compound does not prevent outer or inner races from rotating on mating surfaces, if corroded, or if excessively worn.

b. Replace loose rivets in the gas producer control tube, or a damaged pinion gear.

11-93. Reassembly — Copilot's Collective Pitch Stick. *a.* Check gearshaft bearings (fig. 11-16) in housing cap and housing for security of outer races. Use grade A locking compound (C90) to install a replacement bearing. Use care to prevent compound from entering bearings, and make sure that each bearing is seated against its bore shoulder.

b. Using grease (C46), lubricate the stick tube interior where the gas producer control tube fitting makes contact.

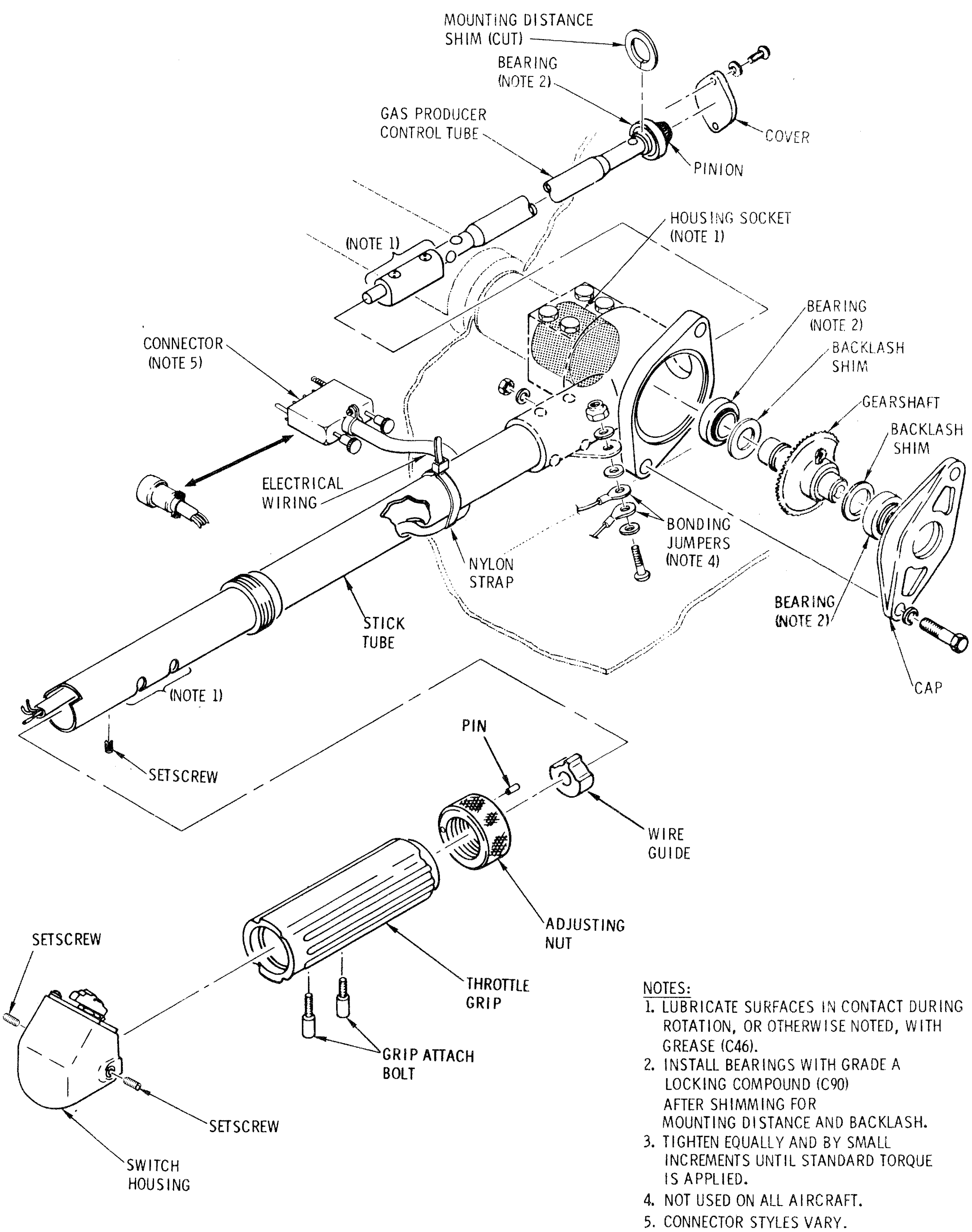
c. Attach the twine, routeo through the stick during disassembly, and thread the electrical wire bundle from the electrical plug through the exit hole in the stick tube and out the front end of the tube. Remove twine.

d. Install wire guide in stick tube so that it positions the wire bundle in the right cutout of the guide. Align guide with matching hole in stick tube and install setscrew. When tightened, **SETSCREW MUST BE AT LEAST FLUSH AND NOT RECESSED MORE THAN 0.010 INCH BELOW OUTER SURFACE OF TUBE.**

e. Install gas producer control tube through the access hole in back of stick housing, and through stick tube until control tube fitting engages the wire guide bore. Reinstall mounting distance shims at forward side of pinion bearing if removed during disassembly. (One edge of shims must be cut for installation.)

f. Establish gas producer control tube pinion mounting distance as shown in figure 11-17.

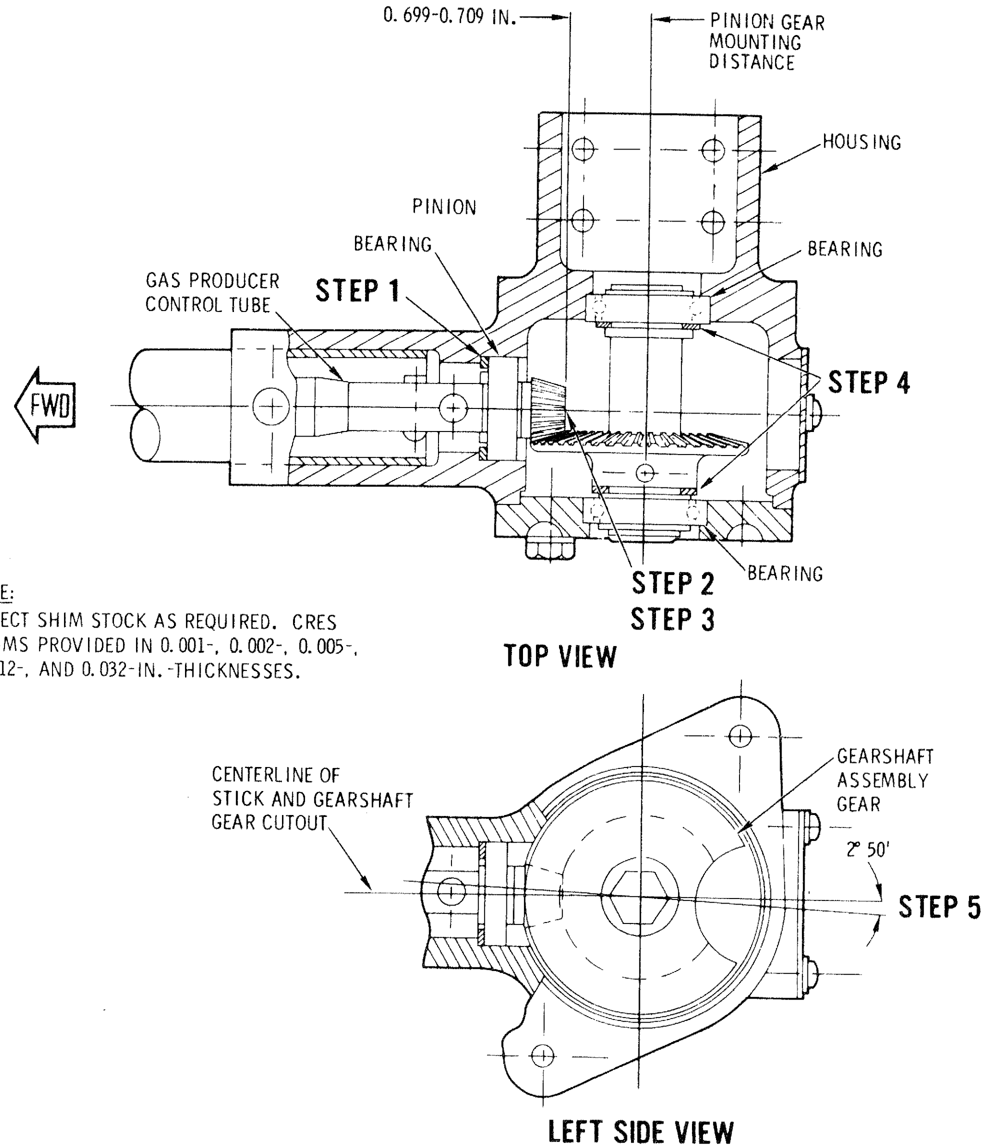
g. After mounting distance shimming is correct, remove gas producer control tube and apply grade A



NOTES:
1. LUBRICATE SURFACES IN CONTACT DURING ROTATION, OR OTHERWISE NOTED, WITH GREASE (C46).
2. INSTALL BEARINGS WITH GRADE A LOCKING COMPOUND (C90) AFTER SHIMMING FOR MOUNTING DISTANCE AND BACKLASH.
3. TIGHTEN EQUALLY AND BY SMALL INCREMENTS UNTIL STANDARD TORQUE IS APPLIED.
4. NOT USED ON ALL AIRCRAFT.
5. CONNECTOR STYLES VARY.

12-237B

Figure 11-16. Assembly of Copilot's Collective Pitch Stick.



GENERAL NOTE:

SELECT SHIM STOCK AS REQUIRED. CRES SHIMS PROVIDED IN 0.001-, 0.002-, 0.005-, 0.012-, AND 0.032-IN.-THICKNESSES.

STEP 1: INSTALL MOUNTING DISTANCE SHIMS FOR 0.699-0.709 IN. PINION GEAR MOUNTING DISTANCE. AFTER SHIMMING IS CORRECT, INSTALL BEARING IN HOUSING WITH GRADE A LOCKING COMPOUND (C90).

NOTE: SHIMS ARE CUT THROUGH ONE EDGE FOR INSTALLATION.

STEP 2: APPLY 10 LB MINIMUM LOAD TO END OF PINION DURING CHECK OF MOUNTING DISTANCE DIMENSION.

STEP 3: POSITION THROTTLE GRIP AT MID-TRAVEL.
NOTE: ONE PINION GEAR TOOTH EQUALS 5° 38'.

STEP 4: INSTALL BACKLASH SHIMS TO OBTAIN 0.003 IN. MAXIMUM BACKLASH. HOUSING AND CAP BEARINGS MUST BE SEATED AGAINST BORE SHOULDER, GEARS MUST NOT BIND, AND PINION MOUNTING DISTANCE MUST BE CORRECT.

STEP 5: AFTER SHIMMING IS CORRECT, INSTALL GEAR-SHAFT SO THAT CENTERLINE OF GEAR TOOTH CUTOUT MATCHES CENTERLINE OF STICK. USE GRADE A LOCKING COMPOUND (C90). TO BOND SHAFT TO ID OF BEARINGS.

NOTE: CENTERLINE OF GEARSHAFT GEAR CENTER TOOTH IS 2° 50' OFF THE CUTOUT CENTERLINE.

12-225C

Figure 11-17. Adjustment of Gas Producer Drive Mounting Distance and Backlash (Copilot's Collective Stick).

locking compound (C90) to OD of pinion bearing outer race, and the stick housing bore. Reinstall control tube and check that compound does not enter the bearing and that the bearing outer race and shims are firmly seated against housing bore.

h. Using grease (C46), lubricate ID of throttle grip and install grip on stick tube. Align grip and gas producer control tube fitting and install grip attach bolts. When tightened, **BOLTS MUST BE AT LEAST FLUSH AND NOT RECESSED MORE THAN 0.010 INCH BELOW OUTER SURFACE OF GRIP.**

i. Establish gas producer control tube pinion and gearshaft bevel gear backlash as shown in figure 11-17.

j. After backlash shimming is correct, remove housing cap and gearshaft.

k. Lubricate teeth of pinion gear and gearshaft bevel gear with grease (C46).

l. Apply grade A locking compound (C90) to gearshaft and ID of bearings. With throttle grip at midtravel, install shaft assembly gear as shown in figure 11-17.

m. Install housing cap on gearshaft and seat with hand pressure while slowly rotating the grip back and forth. Secure cap on side of housing and cover on back of housing.

n. Check throttle grip for zero end play on stick tube and that **NOT MORE THAN 5 POUNDS TORQUE IS REQUIRED TO ROTATE THE GRIP.** If these conditions exist, proceed with *r* below. If there is end play, or too much torque is required to rotate the grip, perform *o* through *r* below.

o. Remove grip attach bolts and grip.

p. Insert a 0.47 inch-diameter drift punch into access hole on forward face of adjusting nut and drive the grooved taper pin from adjusting nut and threaded fitting of stick tube (fig. 11-16).

q. Reinstall grip with grip attach bolts and establish zero end play and correct rotational friction **(5 POUNDS MAXIMUM)** between nut and grip. Then match-drill 0.47 inch maximum diameter the stick tube fitting threads to the existing pin groove in the nut and install pin.

r. Using solder (C93), connect wiring to N2 switch. Pull electrical wiring slack out through exit hole in stick tube and install switch housing with setscrews. Using a nylon strap, or twine (C112), secure electrical wiring to stick tube approximately 1.0 inch aft of the exit hole.

11-94. Installation — Copilot's Collective Pitch Stick. (See fig. 11-16.) *a.* Lubricate stick housing socket with grease (C46).

b. Slide stick housing on collective control interconnecting torque tube. Align matching holes and install bushings, bolts, washers and nuts. Tighten equally and by small increments until standard torque is applied.

c. Install outboard collective stick cover. Secure the electrical plug and check that wiring will not foul when the stick is raised and lowered.

d. Remove snap plug from exterior skin, rotate throttle and visually check for zero backlash between gas producer interconnect tube and the hexagonal ID of the drive gearshaft in the stick housing. Eliminate any backlash by tightening pipe plug in end of torque tube; **ZERO BACKLASH IS REQUIRED AT BOTH ENDS OF THE TUBE.**

11-95. CYCLIC CONTROLS.

11-96. Description — Cyclic Controls. The cyclic controls (fig. 11-18) include the pilot's and copilot's cyclic control sticks, two lateral interconnecting rods, lateral cyclic bellcrank, longitudinal cyclic control interconnecting torque tube, and lateral and longitudinal friction mechanisms. The one-way lock, longitudinal and lateral cyclic trim actuators, longitudinal idler bellcrank, and lateral idler bellcrank are located beneath the pilot's seat structure. The tunnelrouted longitudinal and lateral control rods link the underseat idler bellcranks to the main rotor lateral pitch mixer bellcrank and longitudinal idler.

Key to Figure 11-18.

1. Longitudinal mixer control rod
2. Longitudinal pitch idler
3. Longitudinal pitch mixer bellcrank
4. Lateral pitch mixer bellcrank
5. Lateral mixer control rod
6. Controls support bracket
7. Copilot's cyclic stick guard
8. Lateral idler bellcrank
9. Lateral control rod (sta 70)
10. Copilot's cyclic control stick
11. Quick-release pins
12. Lateral cyclic trim actuator
13. Longitudinal cyclic trim actuator
14. Lateral cyclic bellcrank (sta 67)
15. Longitudinal cyclic control pitch interconnecting torque tube
16. Lateral cyclic control interconnecting rod
17. Lateral friction device mechanism
18. Pilot's cyclic control stick
19. Pilot's cyclic stick guard
20. Longitudinal friction mechanism
21. One-way lock
22. Longitudinal idler bellcrank

NOTES:

1. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.080 IN. ABOVE OUTSIDE SURFACE OF PART AFTER BOLT IS TIGHTENED.
2. CONTROL ROD END BEARINGS MUST NOT JAM THROUGH FULL RANGE OF LONGITUDINAL TRAVEL (34°).
3. CONNECTOR STYLES VARY.

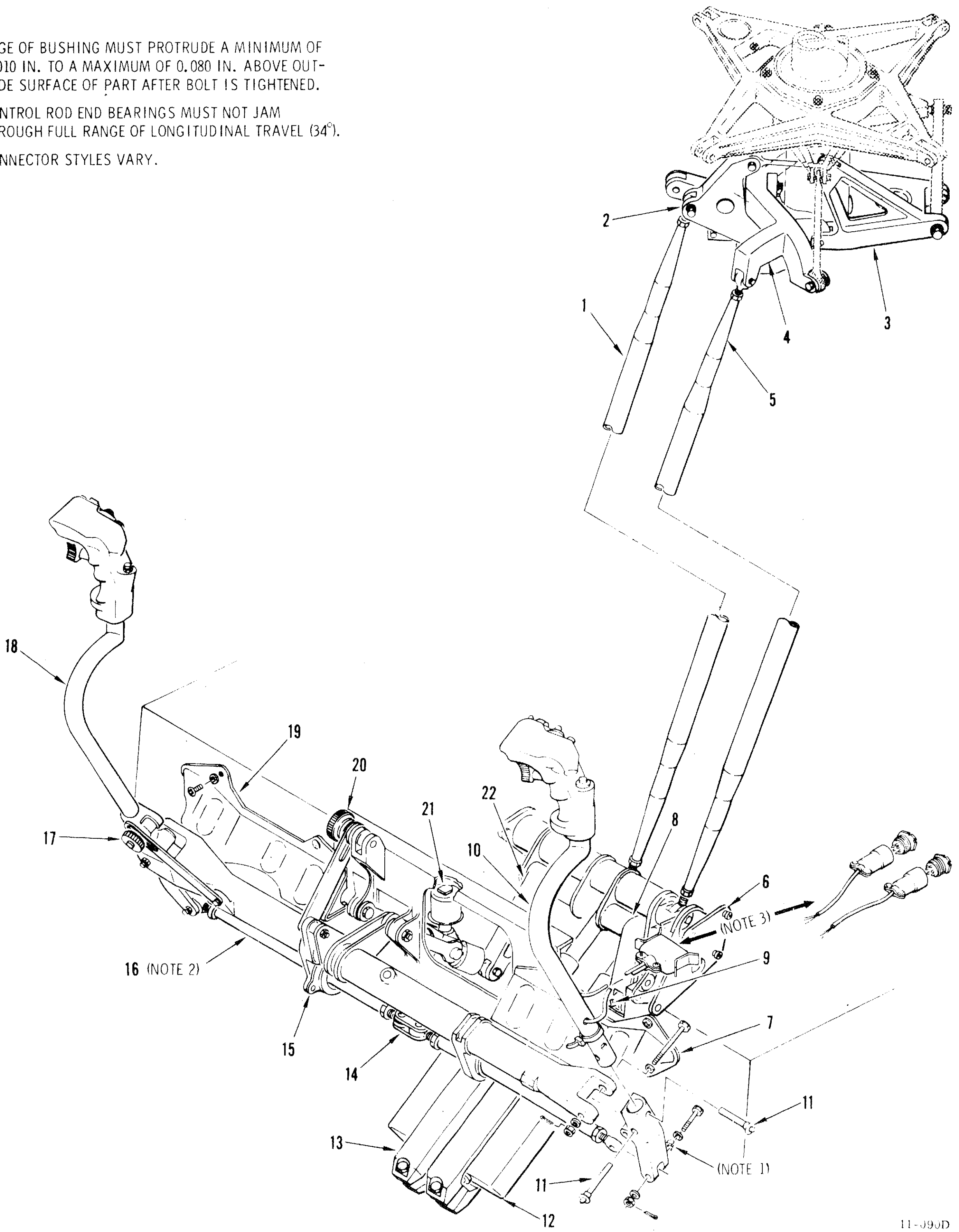


Figure 11-18. Cyclic Controls and One-Way Lock.

11-97. CYCLIC CONTROL STICKS.

11-98. Description — Cyclic Control Sticks. Each cyclic control stick (fig. 11-18) consists primarily of a grip, and an anodized aluminum alloy tube riveted (pilot's) or pinned (copilot's) in a magnesium socket. (Refer to chapter 9 for maintenance of electrical switches and connectors.)

11-99. Inspection — Cyclic Control Sticks (Installed). *a.* Check all attaching hardware for secure attachment by trying to finger-turn nut and/or screw.

NOTE

Inspect co-pilot's stick for quick-release pin for head facing forward.

b. Check copilot's stick quick-release pins for positive lock and serviceable condition. Check for a light film of grease in the stick tube socket and on the stick end. Reapply a thin coating of petrolatum or grease (C73) or (C46) to the contact surfaces when they are dry.

c. Inspect each cyclic control stick tube for dents, corrosion and cracks. Inspect welded area directly below the grip for cracks.

d. Check pilot's stick tube attachment to its socket for evidence of loose rivets, distortion and corrosion.

e. Check parts of friction mechanism for physical damage. Free length of friction spring should be approximately 0.58 inch. (See fig. 11-19.)

f. Loosen lateral and longitudinal friction knobs. Move cyclic control sticks and check for binding or unusual noises. Check that the rod end bearings of the lateral control rods (fig. 11-18) do not jam when stick is full forward and full aft.

11-100. Removal — Copilot's Cyclic Stick. Pull the two quick-release pins, and separate the electrical connector(s); then remove cyclic stick assembly.

11-101. Removal — Pilot's Cyclic Stick Assembly. See fig. 11-19.) *a.* As applicable, disconnect control stick electrical plug or wiring snap-in contacts from the terminal blocks in the electrical control console.

b. Remove the cotter pin, nut, washers and bolts that secure the pilot's lateral control rod to stick socket (detail A, fig. 11-19).

c. Remove the cotter pin, nut, washers and bolt that attach the lateral friction mechanism link to the cyclic torque tube.

d. Remove the cotter pin, nut, washers and bolt that attach the stick socket to the end of the cyclic torque tube. Remove stick with lateral friction mechanism attached.

11-102. Repair — Cyclic Control Stick. Disassemble cyclic control stick (fig.11-19) only as necessary to replace damaged or faulty parts. The pilot's stick tube should not be removed from its socket; the assembly should only be replaced as a unit. Refer to chapter 9 for maintenance and testing of the grip assembly.

NOTE

Maintenance on the copilot's cyclic control stick is performed as outlined in this paragraph, excepting the information on the friction mechanism.

11-103. Removal — Cyclic Stick Grip. (See fig. 11-19.) *a.* Disconnect removable contacts from electrical plug, if plug is installed (chapter 9).

b. Remove the clamp that secures wiring to stick socket.

c. Remove screw from grip and separate grip from tube.

d. Tie twine (C112) to each wire bundle to aid reassembly. Push wire slack into wiring exit holes in stick socket while pulling grip wiring from stick tube. Remove grip and wiring; leave strings in tube.

11-104. Installation — Cyclic Stick Grip. (See fig. 11-19.) *a.* Separate grip wiring into approximately equal color-coded bundles, according to the applicable wiring diagram (appendix F).

b. Install an 8-inch length of AWG Size No. 3 electrical insulation sleeving (C49) on each bundle and push sleeving up into grip.

c. Route wiring through stick tube and out wiring exit holes with the aid of twine. Push wires at grip end while pulling slack out of socket.

d. Install an additional 8-inch length of insulation sleeving to protect each wire bundle where routed through exit holes in socket.

e. Position grip on stick tube and install screw.

f. Push as much wire slack as possible back into the stick. Secure wiring to socket with clamp.

g. Cut wiring to correct length, reinstall replaceable contacts and connect electrical wiring (chapter 9).

11-105. Replacement — Pilot's Cyclic Friction Mechanism. Refer to paragraph 11-108 for removal and installation of the pilot's cyclic friction mechanism.

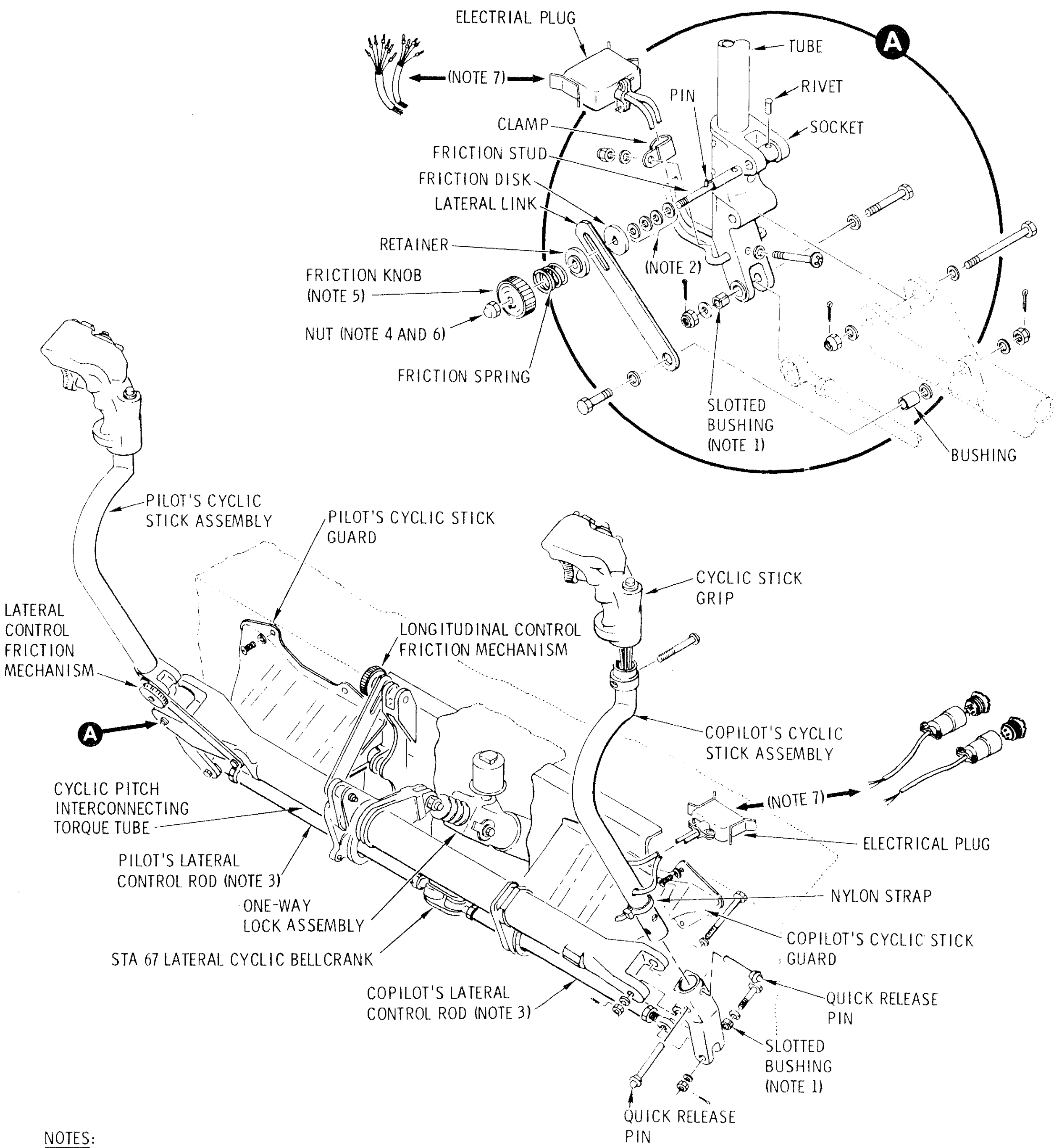
11-106. Installation — Copilot's Cyclic Stick. (See fig. 11-19.) *a.* Install cyclic stick in socket and install two quick-release pins.

b. Connect electrical plugs.

11-107. Installation — Pilot's Cyclic Stick Assembly. (See fig. 11-19.) *a.* Position pilot's cyclic control stick to align with mating holes in torque tube.

b. Install bolt, two washers, nuts and new cotter pin.

c. Check that slotted bushing is in place; then align lateral control rod with stick socket. Install bolt, two washers, nut and new cotter pin.



NOTES:

1. EDGE OF BUSHING MUST PROTRUDE A MAXIMUM OF 0.010 IN. TO A MAXIMUM OF 0.080 IN. ABOVE OUTSIDE SURFACE OF PART AFTER BOLT IS TIGHTENED.
2. ARRANGE WASHERS TO LINE UP LINK WITH TORQUE TUBE. SPACE LOOSELY WITH FOUR WASHERS.
3. CONTROL ROD END BEARING MUST NOT JAM THROUGH FULL RANGE OF LONGITUDINAL TRAVEL (34°).
4. ADJUST CAPNUT TO STOP THE KNOB WITH SPRING AT FREE LENGTH (± 0.03 IN.)
5. FRICTION MECHANISM ASSEMBLY IS THE SAME AT THE LONGITUDINAL POSITION.
6. IF NUT CAN BE TURNED WITH FINGER PRESSURE, THE NUT MUST BE REPLACED.
7. ELECTRICAL CONNECTORS VARY.

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Figure 11-19. Cyclic Grip and Friction Mechanism.

d. Position guide link to align with mating hole in torque tube.

e. Install bolt, sleeve bushing, three washers, nut and new cotter pin.

f. Connect electric plug or wiring snap-in-contacts, as appropriate.

11-108. CYCLIC FRICTION MECHANISMS.

11-109. Description — Cyclic Friction Mechanisms. The lateral and longitudinal cyclic friction mechanisms (fig. 11-19) each consist of a knurled knob, guide link, spring, and two phenolic resin friction discs that link the pilot's cyclic control stick to the longitudinal cyclic pitch interconnecting torque tube, and the torque tube to the seat structure. Turning the knurled knob clockwise tightens the spring retainer against the guide link; turning the knurled knob counterclockwise loosens the compression on the quide link. The amount of friction disc compression on the guide link sets the amount of force required to move the cyclic control sticks in either direction.

11-110. Removal — Cyclic Friction Mechanism. *a.* Remove cap nut from friction stud and disassemble friction mechanism from stick socket or seat structure fitting (detail A, fig. 11-19).

b. Replace a bent or stripped friction stud. Drill out rivet and press stud from socket; install new stud, pick up predrilled hole in stud and rivet in place. Press a replacement spring pin into friction stud until one end of pin is flush with surface of stud.

11-111. Inspection — Cyclic Friction Mechanism. *a.* Inspect all attaching hardware for secure attachment by trying to finger-turn nut and/or bolt.

b. Loosen knobs and move cyclic control sticks laterally and then longitudinally to check for free movement.

c. Tighten knobs and move cyclic control stick laterally and longitudinally to check for resistance to control movement.

d. Check for corrosion and excessive wear.

11-112. Installation — Cyclic Friction Mechanism. *a.* Assemble friction mechanism to stick socket or structure fitting (detail A, fig. 11-19).

b. Adjust friction knob cap nut to stop the knob when the spring reaches free length **(no compression ±0.03 inch).**

11-113. LATERAL AND LONGITUDINAL CYCLIC CONTROL LINKAGE.

11-114. Description — Lateral and Longitudinal Cyclic Control Linkage. The lateral and longitudinal cyclic control linkage consists of the three control rods and bellcrank that interconnect lateral control movement, and the cyclic torque tube and one-way lock that interconnect longitudinal control movement of the cyclic sticks to the tunnel-routed mixer control rods. (See fig. 11-18.) For underseat controls support bracket and idler bellcranks information, refer to paragraph 11-47; for one-way lock information to paragraph 11-123; and for longitudinal and lateral mixer (tunnel-routed) control rods to paragraph 11-40.

11-115. Inspection — Lateral and Longitudinal Cyclic Control Linkage (Installed). (See fig. 11-18.) *a.* Check all attaching hardware for secure attachment by trying to finger-turn nut and/or bolt.

b. Check station 67 lateral cyclic bellcrank (14, fig. 11-18) for corrosion and cracks. Check free play in bellcrank pivot bearing by applying light up and down pressure at forward end of bellcrank. **BELLCRANK TOTAL PLAY MEASURED AT CENTERLINE OF BOLT THAT ATTACHES PILOT'S LATERAL CONTROL ROD (16) MUST NOT BE MORE THAN ±0.024 INCH FOR THE BEARING TO BE ACCEPTABLE.**

c. Check lateral rods (16) for corrosion, rod ends for excessive wear **(0.040-INCH AXIAL LOOSENESS MAXIMUM)** and damage. Check rod end jam nuts for tightness.

d. Use a straightness to check lateral rods (16) that appear bent or bowed. **THE TOTAL LENGTH OF THE ROD (EXCLUDING ROD ENDS) MUST BE STRAIGHT WITHIN 0.010 INCH.** If rods are bowed they must be straightened (para 11-117).

e. Check rod end bearings or rods (16) with cyclic sticks full forward and full aft. Rod ends must not jam in maximum throw positions.

f. Check cyclic torque tube for dents, cracks, corrosion, or deformed bearing supports. Check bearings for binding or excessive wear.

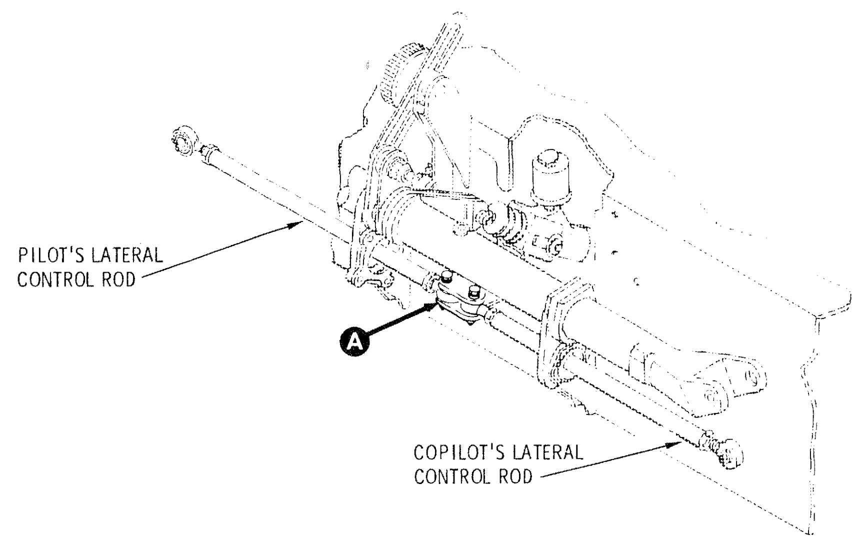
11-116. Removal — Lateral and Longitudinal Cyclic Control Linkage. (See fig. 11-20.) *a.* Remove the pilot's collective stick cover, and the controls access door and left foot support-fairing from station 78.50 canted bulkhead, as needed.

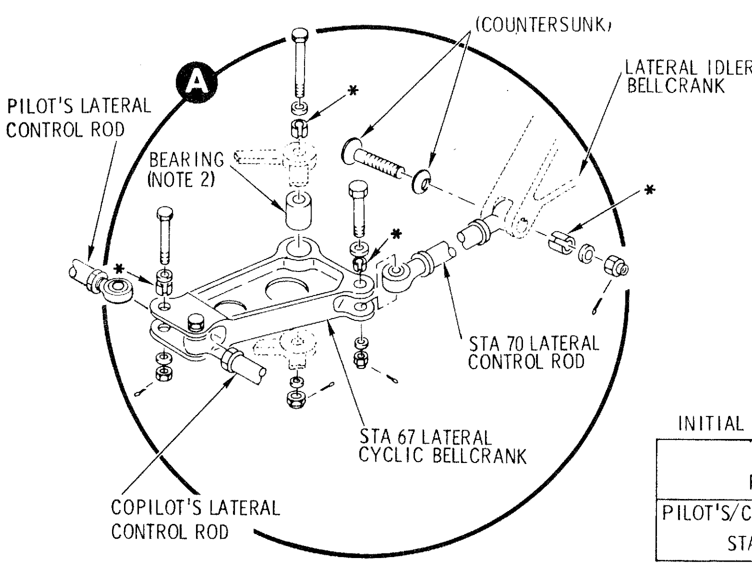
b. Remove the cotter pin, nut, two washers and bolt from each end of the pilot's/copilot's lateral control rod, or station 70 lateral control rod, as applicable; remove rod assembly.

c. Remove the cotter pin, two washers and bolt that secure station 67 lateral cyclic bellcrank to seat structure fitting; remove bellcrank.

d. If rod or rod ends require replacement, accurately measure and record distance between rod end bearing centers for future reference; use the trammel method or equivalent.

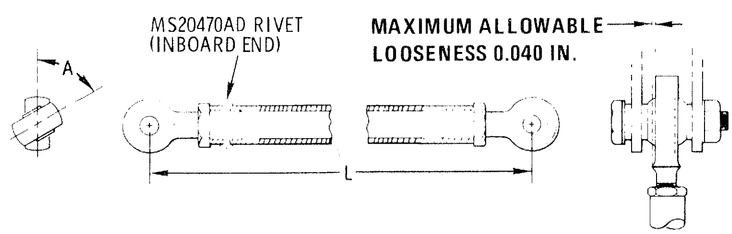
11-117. Repair — Lateral and Longitudinal Cyclic Control Linkage. (See fig. 11-20.) *a.* Replace the station 67 lateral bellcrank if it has elongated holes or is cracked.





INITIAL ROD ASSY LENGTH AND BEARING ANGULARITY (BEFORE RIG)

CONTROL ROD ASSY	DIM L (IN.)	ANGLE A	ASSY PART NUMBER
PILOT'S/COPILOT'S LATERAL	12.63	42°	369A7102
STA 70 LATERAL	6.65	90°	369A7005



NOTES:

1. ASTERISK (*) IDENTIFIES SLOTTED BUSHING. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 INCH TO A MAXIMUM OF 0.060 INCH ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. INSTALL WITH GRADE A LOCKING COMPOUND (C90).

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Figure 11-20. Cyclic Stick Lateral Control Linkage.

b. Replace the bellcrank pivot bearing if it binds or is excessively worn; install replacement with grade A locking compound (C90). Check free play of a bellcrank having a newly installed pivot bearing by applying light up and down pressure at forward end of bellcrank with bearing secured. **BELLCRANK TOTAL PLAY MEASURED AT CENTERLINE OF BOLT THAT ATTACHES PILOT'S LATERAL CONTROL ROD END MUST NOT BE MORE THAN ±0.024 INCH FOR THE BEARING TO BE ACCEPTABLE.**

c. Perform a straightness check on a control rod that appears bent or bowed. **THE TOTAL LENGTH OF THE ROD (EXCLUDING ROD ENDS) MUST BE STRAIGHT WITHIN 0.010 INCH.** Cold-straightening of the rod is permissible provided there are no nicks or sharp dents in the bend length, and the rod ends are NOT used to support the rod during the straightening process. A dye-check for cracking shall always be performed after cold-straightening. Replace a cracked rod, or cracked or bent rod end.

CAUTION

Use care when drilling to remove or install riveted rod end; the rod end is steel and the rod is aluminum. Do not tighten adjustable rod end jam nut on pilot's/copilot's lateral control rod until rod is installed and cyclic stick longitudinal travel is checked.

d. **REPLACE A CONTROL ROD END IF BEARING AXIAL PLAY IS MORE THAN 0.040 INCH.** Set initial control rod length and bearing angularity as shown in figure 11-20.

11-118. Installation — Lateral and Longitudinal Cyclic Control Linkage. (See fig. 11-20.) *a.* Check that slotted bushing is in upper web of seat structure fitting. Align station 67 lateral bellcrank and install bolt, two washers, nut and new cotter pin.

b. Check that slotted bushings are in place and install station 70 lateral control rod and pilot's/copilot's lateral control rod at each end with bolt, two washers, nut and new cotter pin.

c. Move cyclic stick full forward, then full aft and set pilot's lateral control rod end bearing angularity so that the bearings do not jam at the full throw positions; hold rod end and tighten jam nut. If a control rod or rod ends were replaced, perform a cyclic control rigging check (para 11-4).

CAUTION

Check that the controls area is free of loose objects or tools.

d. Reinstall the pilot's collective stick cover, controls access door, and left foot support fairing.

11-119. CYCLIC PITCH INTERCONNECTING TORQUE TUBE.

11-120. Removal — Cyclic Pitch Interconnecting Torque Tube. *a.* Remove pilot's collective stick cover (chapter 2).

b. Position pilot's cyclic stick to align holes in one-way lock support links with the bolt that attaches forward end of one-way lock to torque tube; tighten longitudinal friction. Disconnect rod end from torque tube arm (fig. 11-21).

c. Remove the pilot's and copilot's cyclic stick assemblies (para 11-97).

d. Remove the longitudinal control friction mechanism from the torque tube.

e. Remove pilot's and copilot's lateral control rods from station 67 lateral cyclic bellcrank (fig. 11-20).

f. Cut lockwire and remove the support nuts, support bolts and the two wide (left-hand) support spacers from the seat structure fitting lugs and torque tube (fig. 11-21). Use care to prevent the torque tube from dropping as the bolts are removed.

g. Remove the torque tube and the two narrow (right-hand) support spacers.

11-121. Repair — Cyclic Pitch Interconnecting Torque Tube. *a.* Replace the cyclic torque tube if it is cracked, has elongated holes or deformed bearing supports.

b. Replace the main support bearings or cyclic stick pivot bearings if they bind or excessively worn. Drill out the four rivets and remove bearing retainer to replace right support bearing (fig. 11-21). Install cyclic stick pivot bearings and the right support bearing with grade A locking compound (C57).

NOTE

The left support bearing is a slip-fit in the torque tube lug; coat OD of replacement bearing with grease (C46).

11-122. Installation — Cyclic Pitch Interconnecting Torque Tube. *a.* Align cyclic torque tube support bearings with holes in structure fitting lugs. Place a narrow support spacer on the right side of each support bearing, and install the two wide support spacers and support bolts (fig. 11-21).

NOTES:
1. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.080 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
2. MAY BE INSTALLED IN EITHER DIRECTION.
3. MAY BE INTERCHANGED TO ALIGN ONE-WAY LOCK.
4. INSTALL WITH GREASE (C46) ON OD.
5. INSTALL WITH GRADE A LOCKING COMPOUND (C90).
6. CAUTION: DO NOT BUCK RETAINER RIVETS; SQUEEZE ONLY.
7. SHIM WASHERS AS REQUIRED TO ALIGN ONE-WAY LOCK (REMOVE SIDELOAD).

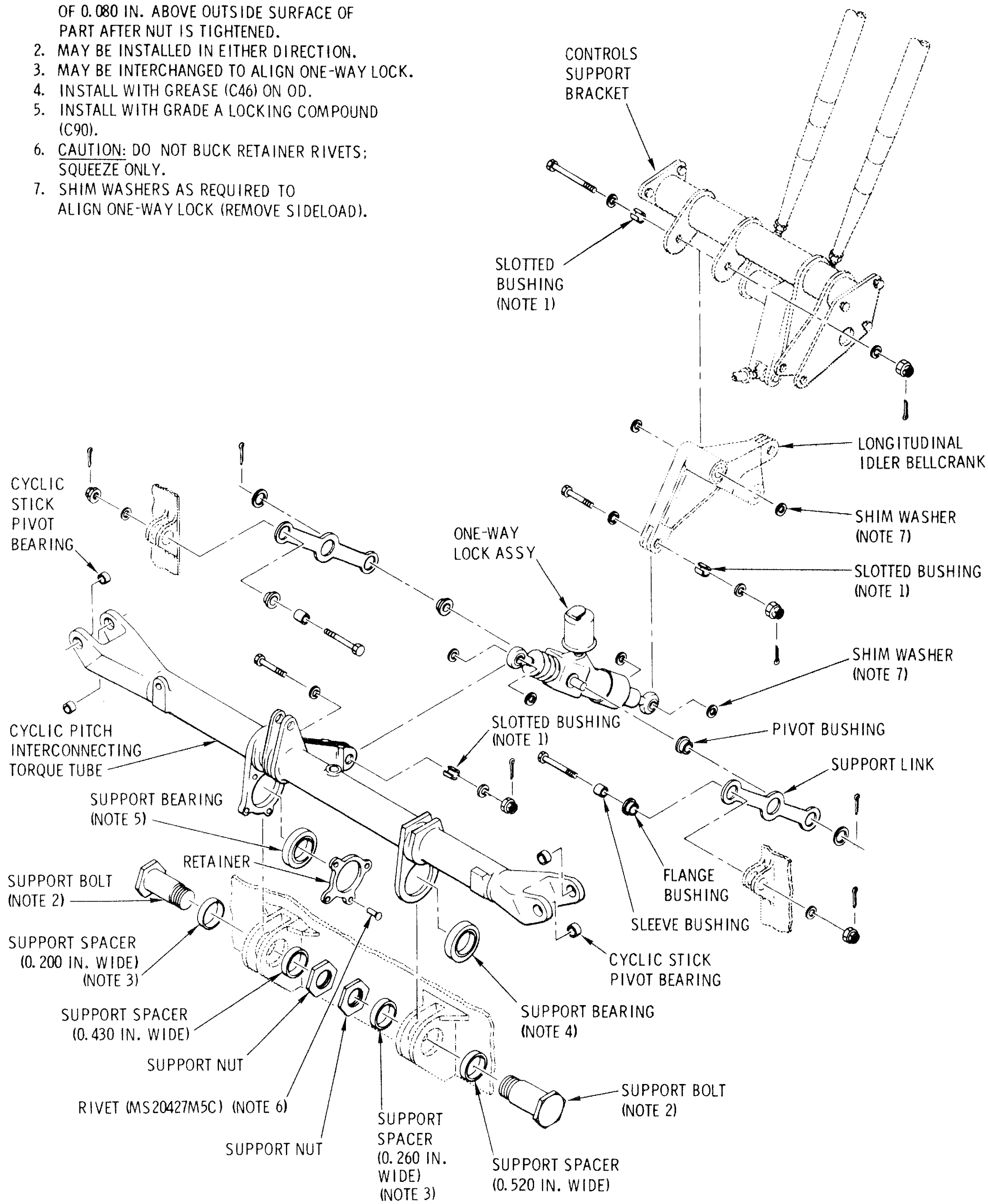


Figure 11-21. Cyclic Torque Tube and One-Way Lock.

NOTE

The support bolts may be installed from either direction, and the two narrowest (right-hand) support spacers may be interchanged to get best alignment of the torque tube with the one-way lock.

b. Check that slotted bushing is in torque tube arm, and fit one-way lock rod end against unbushed lug of arm. If tightening the attachment bolts will apply side-load on the rod end, interchange the two narrowest (right-hand) support spacers, and/or adjust the lateral position shimming of the longitudinal idler bellcrank or one-way lock rod ends until there is no sideload on the one-way lock (fig. 11-21). Check that no interference with structure results from the spacer interchange.

c. When there is no sideload on one-way lock, install and tighten the support nuts. Use 0.032 inch lockwire (C57) and safety each nut to the hole in the threaded end of the mating bolt. Use care when installing lockwire as both the nut and bolt are aluminum.

d. Install pilot's and copilot's lateral control rods (riveted rod end) to station 67 lateral bellcrank (fig. 11-20).

e. Attach the longitudinal control friction mechanism to the torque tube (detail A, fig. 11-19).

f. Install the pilot's and copilot's cyclic stick assemblies (para 11-97).

g. Position pilot's cyclic stick to align bolt holes in torque tube arm with holes in one-way lock support links. Install bolt, two washers, nut and new cotter pin (fig. 11-21).

h. Reinstall pilot's collective stick cover (chapter 2).

11-123. ONE-WAY LOCK.

11-124. Description — One-Way Lock. The cyclic control system one-way lock is located in the longitudinal control linkage within the pilot's seat structure. (See fig. 11-18 and 11-19.) The one-way lock is essentially a self-contained closed-loop hydraulic unit consisting of a check valve, relief valve, and pushrod mechanism. The check valve is seated when longitudinal control force (feedback) originated by the main rotor tends to move the one-way lock (and cyclic stick) in an aft direction. Seating the check valve prevents unwanted aft movement of the cyclic stick and shunts the feedback force into the airframe structure. Normally, only very slight aft movenent of the cyclic stick is required to unseat the check valve. Should the check valve or push rod shaft that unseats the valve gall and freeze in the valve-closed position, a force of approximately 30 pounds is necessary to open the relief valve and bypass the check valve. This force will then be required for each subsequent movement of the cyclic stick, either forward or aft. Conversely, should the check valve spring fail, the one-way lock will not function to shunt longitudinal feedback forces into the structure. The unit is mounted in two pivoting supports attached to the seat structure, and has an unbreakable, transparent, vented reservoir with a capacity of approximately 0.76 fluid ounce (20 cc). The one-way lock is serviced through the filler as necessary with hydraulic fluid (C48).

11-125. Inspection — One-Way Lock (Installed). *a.* Remove pilot's collective stick cover (chapter 2).

b. Check all attaching hardware for secure attachment by trying to finger-turn nut and/or screw. Inspect rod end bearings for binding and good alignment; there must be no side-load on bearings.

c. Check flexible boot for deterioration.

d. Check reservoir for oil leakage and full reservoir. Check vent in reservoir to ensure that it is clear. Check filler cap for proper closure.

NOTE

Hydraulic leakage from any part of the one-way lock is not permisible. When leakage is observed the assembly should be repaired or replaced with a serviceable unit.

e. Inspect housing and supports for dents, cracks, corrosion and excessive wear.

f. Check for excessive end play at input end of one-way lock mechanism. Measure between rod ends; play is limited to 0.010 inch maximum.

g. Reinstall pilot's collective stick cover (chapter 2).

11-126. Lubrication — One-Way Lock. Refer to paragraph 1-19 for servicing of the one-way lock.

11-127. Removal — One-Way Lock. (See fig. 11-21.) *a.* Remove pilot's collective stick cover, and controls access door from station 78.50 canted bulkhead.

b. Position pilot's cyclic stick to align holes in one-way lock support links with the bolt that attaches forward end of one-way lock to cyclic torque tube; tighten longitudinal friction. Disconnect rod end from torque tube arm.

c. Disconnect one-way lock lower rod end from longitudinal idler bellcrank.

d. Disconnect upper end of each support link from the seat structure and remove one-way lock with links attached.

e. Remove the two cotter pins, washers and links. Do not remove the pivot bushings from the links unless replacement is necessary.

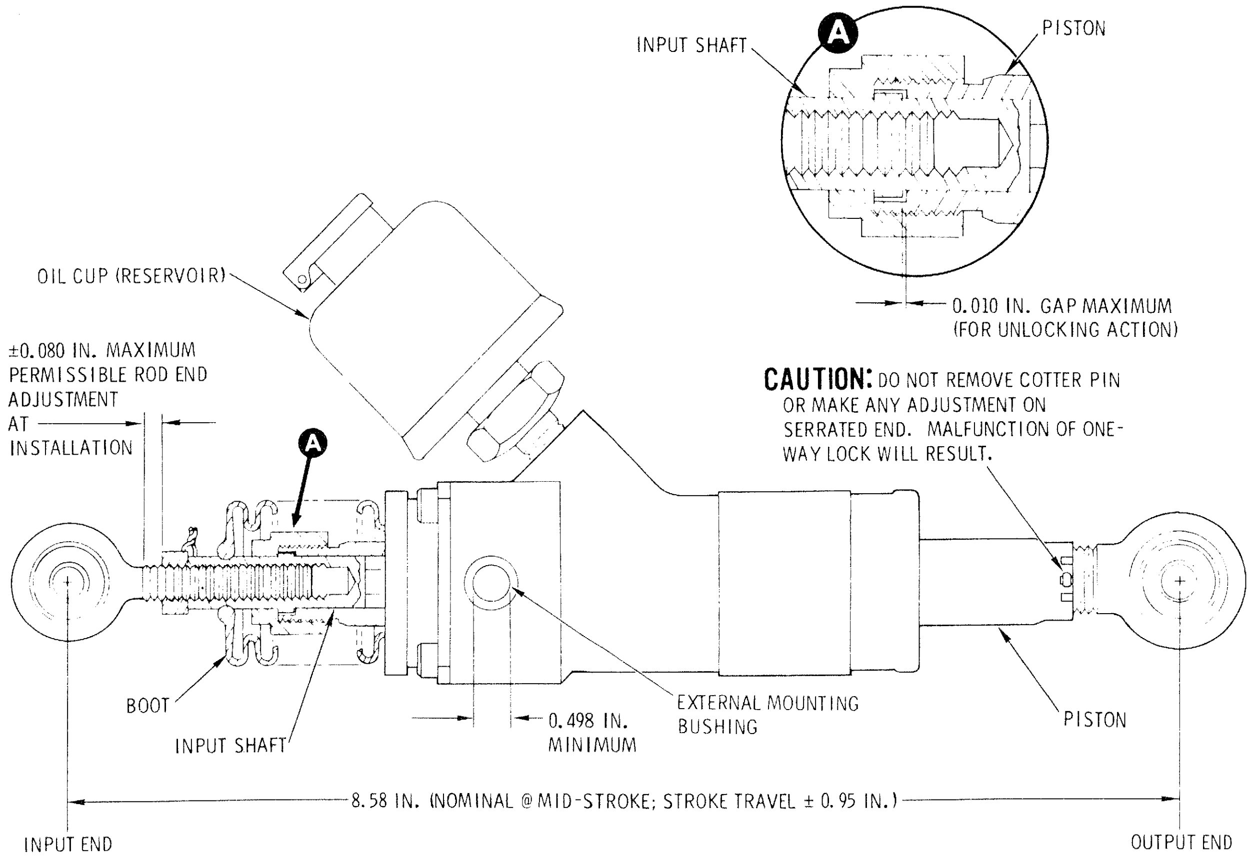


Figure 11-22. One-Way Lock Installation Adjustment.

11-128. Inspection — One-Way Lock (Removed). *a.* Replace reservoir if it is cracked, leaking or has a loose or damaged filler cap.

b. Replace input end rod end bearing if bearing is binding.

c. Replace one-way lock if output shaft end bearing is binding.

d. Replace protective boot if it is cracked, torn or deteriorated.

e. Replace the one-way lock if the exposed portion of the piston has nicks, scratches or wear which penetrate the chrome plating.

f. **REPLACE THE ONE-WAY LOCK IF THE OD OF THE EXTERNAL MOUNTING BUSHING IS LESS THAN 0.498 INCH.**

g. **ONE-WAY LOCK MUST BE REPLACED IF NICKS, DENTS, AND SCRATCHES TO THE BODY EXCEED 0.040 INCH AFTER REPAIR.**

h. **ONE-WAY LOCK MUST BE REPLACED IF END PLAY BETWEEN ROD ENDS EXCEEDS 0.010 INCH MEASURED AT THE INPUT END (fig. 11-22).**

i. One-way lock must be replaced if it is leaking, except as noted for the reservoir in *a* above.

11-129. Repair — One-Way Lock. *a.* Remove dents, nicks, and scratches by using grade 400 abrasive paper (C3) to remove rough edges. Finish by polishing with crocus grade abrasive cloth (C25). Maintain a smooth transition into the surrounding area. After repair treat the area with chemical film (C20).

b. When replacing input shaft rod end bearing ensure that with the piston in mid stroke the bearing center to center distance is 8.58 inches (fig. 11-22). After correct length has been obtained tighten and safety the lock nut with 0.032 inch lockwire (C57).

11-130. Installation — One-Way Lock. (See fig. 11-21.) *a.* Install new pivot bushings in support links if replacement is necessary.

b. Attach lower ends of links to one-way lock; use new cotter pins.

c. Align upper ends of links with mating holes in seat structure fitting and attach each link with bolt, sleeve bushing, washer, nut and new cotter pin.

d. Check that slotted bushing is in place in longitudinal idler bellcrank; then fit lower rod end bearing against unbushed lug of bellcrank to check if sideloading will occur when attaching bolt is tightened. There must be no sideloading applied to the one-way lock mechanism. Arrange shim washers, as necessary, to position longitudinal idler bellcrank and/or lower rod end for no-sideload alignment.

e. Position pilot's cyclic stick to align holes in support links with hingeline for upper end of one-way lock and cyclic torque tube arm; tighten longitudinal friction.

f. Repeat the check for no sideload on the one-way lock, *d* above, and arrange shim washers, as necessary, to position upper rod end for no-sideload alignment.

NOTE

The two narrowest (right-hand) cyclic torque tube support spacers may also be interchanged to align the one-way lock (para 11-122).

g. When there is no sideload on one-way lock, secure each rod end bearing with bolt, two washers, nut and new cotter pin.

h. Fill one-way lock reservoir with hydraulic fluid (C76) or (C48).

i. Install pilot's collective stick cover, and controls access door (chapter 2).

11-131. CYCLIC TRIM ACTUATORS.

11-132. Description — Cyclic Trim Actuators. Each cyclic trim actuator (12 and 13, fig. 11-18) consists of an actuator, housing support, trim tube, and spring assembly. When energized, the actuator motor-driven shaft extends or retracts the spring assembly which applies the necessary pressure in the cyclic linkage to counteract feedback forces from the main rotor or to compensate for imbalance conditions such as those imposed by crosswinds or unevenly distributed cargo.

11-133. Inspection — Cyclic Trim Actuators (Installed). *a.* Remove two foot support fairings and controls access door (chapter 2).

b. Check all attaching hardware for secure attachment by trying to finger-turn nut and/or bolt.

c. Inspect electrical connectors and bonding for secure attachment and corrosion; check condition of wiring harness.

d. Inspect actuator housing, housing support and spring assembly for dents, cracks, corrosion, and excessive wear at pivot points.

e. When actuator speed (response time) is questionable, a bench test should be performed (para 11-136).

f. Install foot support fairings and controls access door (chapter 2).

11-134. Removal — Cyclic Trim Actuators. (See fig. 11-7.) *a.* Jack up the aircraft until landing gear is fully extended. Remove foot fairings, control access door from station 78.50 canted bulkhead, and open or remove pilot's compartment floor access doors.

b. Disconnect bonding jumper and electrical connector from actuator housing.

c. Disconnect upper end of trim actuator. Keep pivot bushing with actuator unless actuator is being replaced.

NOTE

Lateral actuator upper hingeline has 0.187 inch ID; longitudinal has 0.25 inch ID.

d. Disconnect lower end of trim actuator. Access to the attaching nut is through a hole in the underfloor compartment aft bulkhead, near the centerline beam.

e. Remove trim actuator. Reinstall the lower end attaching hardware to go with actuator if the actuator is being replaced; a new actuator includes this hardware.

11-135. Inspection — Cyclic Trim Actuators (Removed). *a.* Check electrical connector for evidence of damage, and broken or missing contacts.

b. Check exterior of trim actuator for evidence of damage and deformation.

c. When actuator speed (response time) is questionable, perform the bench test outlined in paragraph 11-136.

11-136. Testing — Cyclic Trim Actuator (AVIM). *a.* Connect cyclic trim actuator to test harness shown in figure 11-23.

b. Mount the actuator in a suitable fixture that will allow vertical extension and retraction.

c. Turn on direct-current power and adjust output to 25.75 — 26.25 volts. Monitor voltage during test to maintain voltage within tolerance.

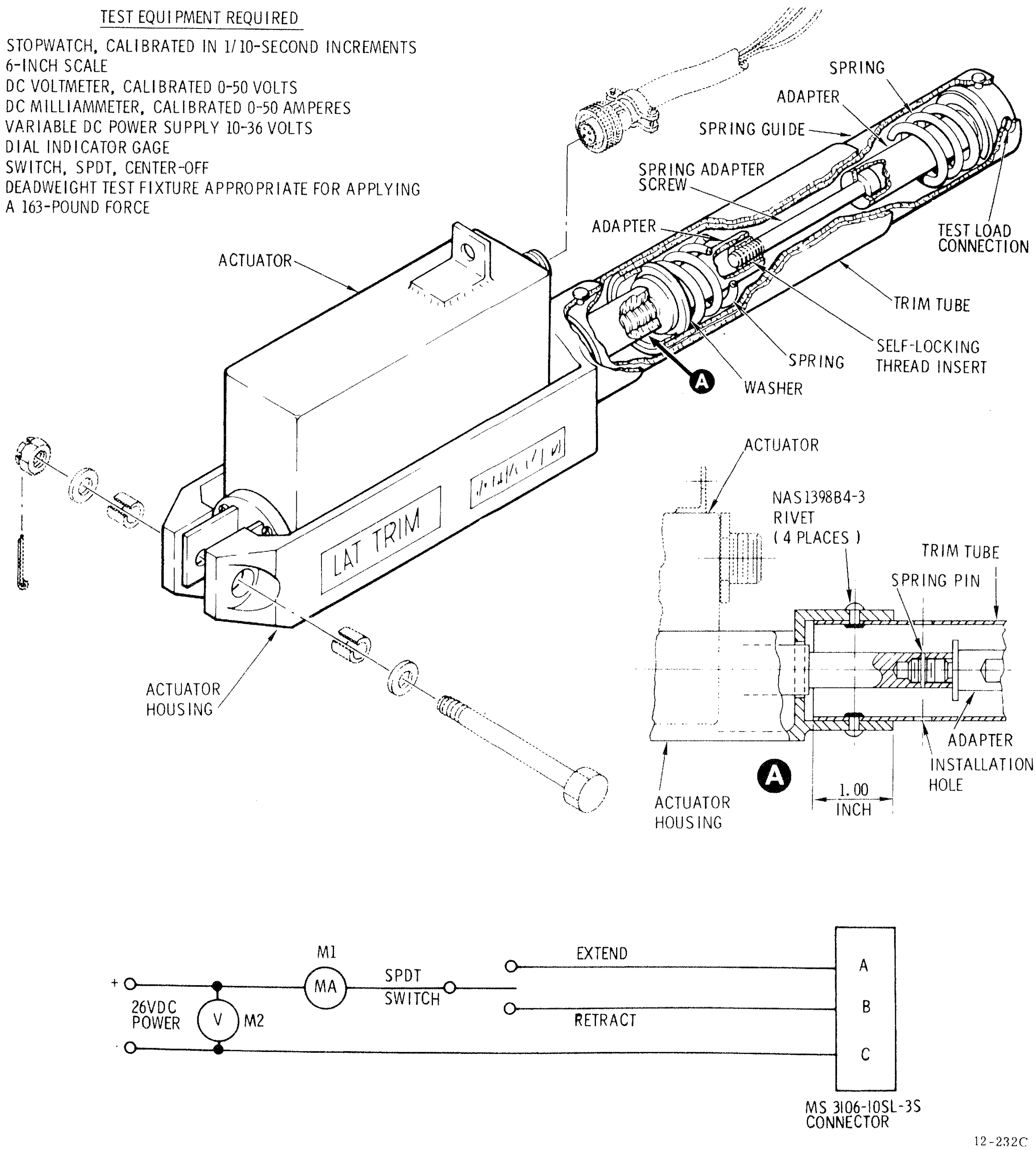


Figure 11-23. Cyclic Trim Actuator, Test Hookup.

CAUTION

During the following tests, do not retract the spring assembly (ram) so that actuator length is less than 13.50 inches between attach bolt centers.

d. With actuator spring assembly (ram) approximately halfway between travel limits (actuator length at 14.46 inches between attach bolt centers) set up test dial indicator and measure the ram end play. The end play must not exteed 0.007 TIR when measured under a 10 pound reversing load. Remove any spring free play in the spring assembly by tightening the spring adapter screw until no play can be felt. While tightening the screw, check that there is thread-locking friction from the self-locking threaded insert (fig. 11-23). Replace actuator assembly if end play is excessive.

e. Position test switch to RETRACT and allow the actuator ram to fully retract. Using the end of the trim tube as a reference point, measure the portion of the ram that remains out of the tube. Record this measurement for use in *g* below.

f. Set the stopwatch to zero seconds. Start the actuator and stopwatch at the same time. Observe the time it takes the actuator ram to travel from fully retracted to fully extended. The time interval should be less than 58.4 seconds. The operating current with actuator stalled should not exceed 2.0 amperes.

g. Measure the length of the extended ram. Subtract the measured result of *e* above from this value. The result must be 3.47 to 3.53 inches.

CAUTION

In the following test, do not run actuator against extend limit stop with the 163-pound weight aiding actuation. The actuator thrust bearing can be damaged.

h. Connect a 163-pound load to the actuator. Turn the power ON. Check the time it takes the ram to travel 1.72 to 1.78 inches. The retraction time should be from 30 to 50 seconds.

i. The operating current under load conditions must not exceed 1.2 amperes running.

j. Reduce the voltage input to 21 volts. The actuator ram must retract and extend. Overtravel is limited to 0.200 inch with an overriding (aiding) load; otherwise there must be no overtravel.

NOTE

Is is not necessary to extend or retract the actuator ram more than 0.25 inch for the test in j above or k below.

k. Increase the input voltage to 28 volts. The actuator ram must retract and extend. Overtravel is limited to 0.020 inch maximum with an overriding (aiding) load; otherwise there must be no overtravel.

l. Remove 163-pound load and repeat *d* above.

m. Run actuator until the ram is extended 1.75 inches. Turn power off and disconnect the cyclic trim actuator.

11-137. Replacement — Cyclic Trim Actuator Spring Assembly (AVIM). *a.* Insert a screwdriver into the open end of the spring guide and remove the screw that connects the two spring adapters (fig. 11-23).

NOTE

Identify the screw for lateral or longitudinal trim because the screws are not interchangeable. The lateral actuator screw is 2.88 inches long; the longitudinal screw is 2.63 inches long.

b. Slide spring assembly out of the trim tube.

c. Clean the inside of the trim tube with solvent (C94).

d. Coat the outside of the replacement spring assembly tube lightly with grease (C46).

e. Slide replacement spring assembly into the trim tube.

f. Install spring adapter screw in the open end of spring assembly and tighten screw until no free play can be felt while pushing and pulling on spring assembly. While tightening the screw, check that there is thread-locking friction from the self-locking threaded insert (fig. 11-23).

g. Wipe off all excess grease.

h. Perform a bench test to check actuator operation (para 11-136).

11-138. Replacement — Cyclic Trim Actuator Housing or Trim Tube (AVIM). *a.* Remove the actuator spring assembly (para 11-137).

b. Carefully drill out the four trim tube attaching rivets (fig. 11-23). Remove trim tube.

c. Remove actuator from actuator housing (para 11-139).

d. Use the dimensions on figure 11-23 to assemble the actuator housing and trim tube. Pick up and drill (size 40, 0.098-inch) the four attaching rivet holes, if required.

e. Coat the mating surfaces of the trim tube and actuator housing with primer (C79).

f. Install the four mechanically expanding rivets (NAS1398B4-3).

g. Reinstall the actuator (para 11-139).

h. Reinstall the actuator spring assembly (para 11-137).

i. Perform a bench test (para 11-136).

11-139. Replacement — Cyclic Trim Actuator (AVIM). *a.* Remove actuator spring assembly (para 11-137).

b. Remove the trim tube (para 11-138).

c. Support the actuator tube and press out the spring pin (fig. 11-23).

d. Unscrew the adapter from the actuator and remove the actuator from the housing.

e. Screw the adapter into the replacement actuator shaft until it bottoms. If the spring pin hole drilled in the adapter intersects the guide hole in the actuator the adapter shall be replaced; otherwise, the adapter may be reused.

f. Use a 0.063-inch drill to drill the spring pin hole through the adapter and actuator. Deburr the hole.

g. Reassemble the actuator housing and trim tube (para 11-138).

h. Install the actuator in the housing. Screw the adapter onto the actuator.

i. Support the actuator shaft. Press in the spring pin. Be sure that the pin is flush with the shaft.

j. Install the actuator spring assembly (para 11-137).

k. Bench test the actuator assembly (para 11-136).

11-140. Installation — Cyclic Trim Actuators. *a.* Check the trim actuator spring assembly for free play between the spring and spring adapters (fig. 11-23). Tighten spring adapter screw until no play can be felt while pushing and pulling on spring assembly. While tightening adapter screw, check that there is thread-locking friction from the self-locking threaded insert.

b. Remove the hardware supplied in housing end of a new trim actuator.

c. Check that slotted bushings are in place in actuator housing. Place one AN960-716L washer on each side of the pivot bearing in the center beam support lug and align the actuator housing with the bearing (fig. 11-7). Install the bolt, two washers, nut and new cotter pin (all new if actuator is a replacement).

d. Align actuator with idler bellcrank. Check that slotted bushing is in bellcrank lug; install actuator pivot bushing if actuator is a replacement. The pivot bushing must rotate freely in the actuator. Install bolt, two washers, nut and new cotter pin. After the bolt is tightened the pivot bushing must not rotate in the idler bellcrank.

NOTE

Lateral actuator upper hingeline has 0.187 inch-diameter bolt hole; longitudinal has 0.25 inch-diameter bolt hole.

e. Connect bonding jumper (fig. 11-7). Jumper connected over actuator electrical receptacle should have an electrical flat washer installed under jumper terminal to keep nut from bottoming on threads. **RETAINING NUT TORQUE MUST NOT EXCEED 50 INCH-POUNDS.** Connect electrical plug to actuator.

f. With main rotor blades lifted off the droop stops, perform a power-on operational check of the cyclic trim actuator. The actuator must not bind throughout the travel range in either direction, and must maintain at least 0.015-inch (minimum) clearance with the center beam structure.

g. Reinstall foot fairings, controls access door, floor access doors and remove jacks from aircraft (chapter 2).

11-141. CYCLIC STICK GUARDS.

11-142. General — Cyclic Stick Guards. The pilot's and copilot's cyclic stick guards (fig. 11-19) block the entry of any large foreign objects that might jam or foul the control linkage. Each guard is made of polycarbonate plastic.

11-143. Removal — Cyclic Stick Guards. (See fig. 11-19.) Remove attaching screws, washers, and cyclic stick guard.

11-144. Inspection — Cyclic Stick Guards. Check cyclic stick guards for cracks or other damage.

11-145. Installation — Cyclic Stick Guards. (See fig. 11-19.) Position cyclic stick guard and install attaching screws with washers.

11-146. TAIL ROTOR CONTROL SYSTEM.

11-147. General — Tail Rotor Control System. The tail rotor control system (fig. 11-24) consists of a pedal installation, floor and tunnel-routed control rods and bellcranks, and upper fuselage and boom tail rotor control linkage.

11-148. Troubleshooting — Tail Rotor Control System. Troubleshooting information is divided into: (1)

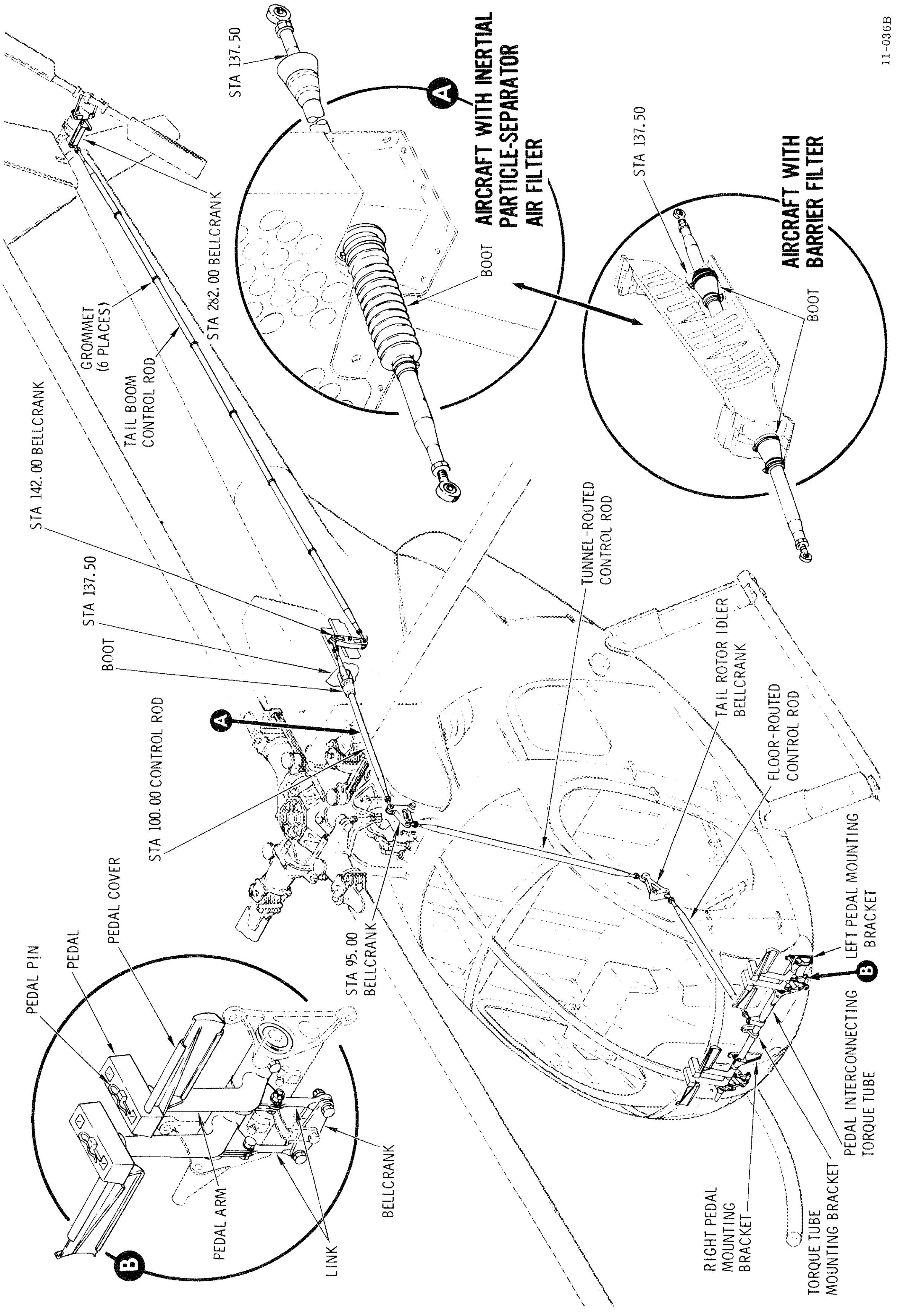


Figure 11-24. Tail Rotor Control System.

Table 11-11. Isolating Tail Rotor Control System Troubles.

Symptom	Isolating Step	Corrective Action
Binding, locking-up and erratic action of foot pedals	Disconnect pitch control links from pitch control assembly (fig. 5-21).	If symptom gone, replace tail rotor assembly.
Symptom remains	Disconnect station 282 aft-boom bellcrank from pitch control assembly (fig. 5-21).	If symptom gone, replace pitch control assembly.
Symptom remains	Disconnect floor-routed tail rotor control rod from foot pedal torque tube fitting (fig. 11-26).	If symptom gone, inspect tail rotor control rods and bellcranks until defective part is located.
Symptom remains	Loosen foot pedal torque tube brackets mounted on front of floor structure.	If symptom gone, pedal torque tube misaligned; shim for correct alignment (fig. 11-26).
		If symptom remains, disassemble and inspect tail rotor control foot pedal installation until defective part is located.
Trouble corrected		

investigation of operational vibration problems originating with the tail rotor assembly or symptoms that can be recognized (table 5-11); and (2) isolation of an unusual controls malfunction that will require disconnection or removal of control components (table 11-11).

Table 11-12. Premaintenance Requirements for Rigging Tail Rotor Control System.

Conditions	Requirements
Special Tool	(T23)

11-149. Rigging — Tail Rotor Control System. Rigging of the tail rotor control system must be accomplished immediately after replacement of linkage that cannot be accurately measured (by trammeling, etc) before it is installed in the tail rotor control system, or if aircraft operation reveals a rigging deficiency.

CAUTION

Remove all pedals to prevent possible contact with the lower windshield during the rigging sequence.

NOTE

Control rod end bearing adjustment is to be made to the nearest half turn that will produce correct rigging.

a. Using two pieces of wood and a C-clamp or rope, secure pilot's foot pedal arms so they are aligned within 0.25 inch (detail A, fig. 11-25).

b. With the pilot's foot pedals clamped in neutral and control rod lengths adjusted as shown in figures 11-26 and 11-27, fuselage bellcrank positions may be verified as follows:

(1) Tail rotor idler bellcrank (pilot's compartment) — Centerline of aft arm 90 ±2 degrees to station 78.50 bulkhead.

(2) Station 95 bellcrank — Centerline of forward control rod attach bolt 4.30 ± 0.090 inches above the mast base structure.

(3) Station 142 bellcrank — Centerline of aft

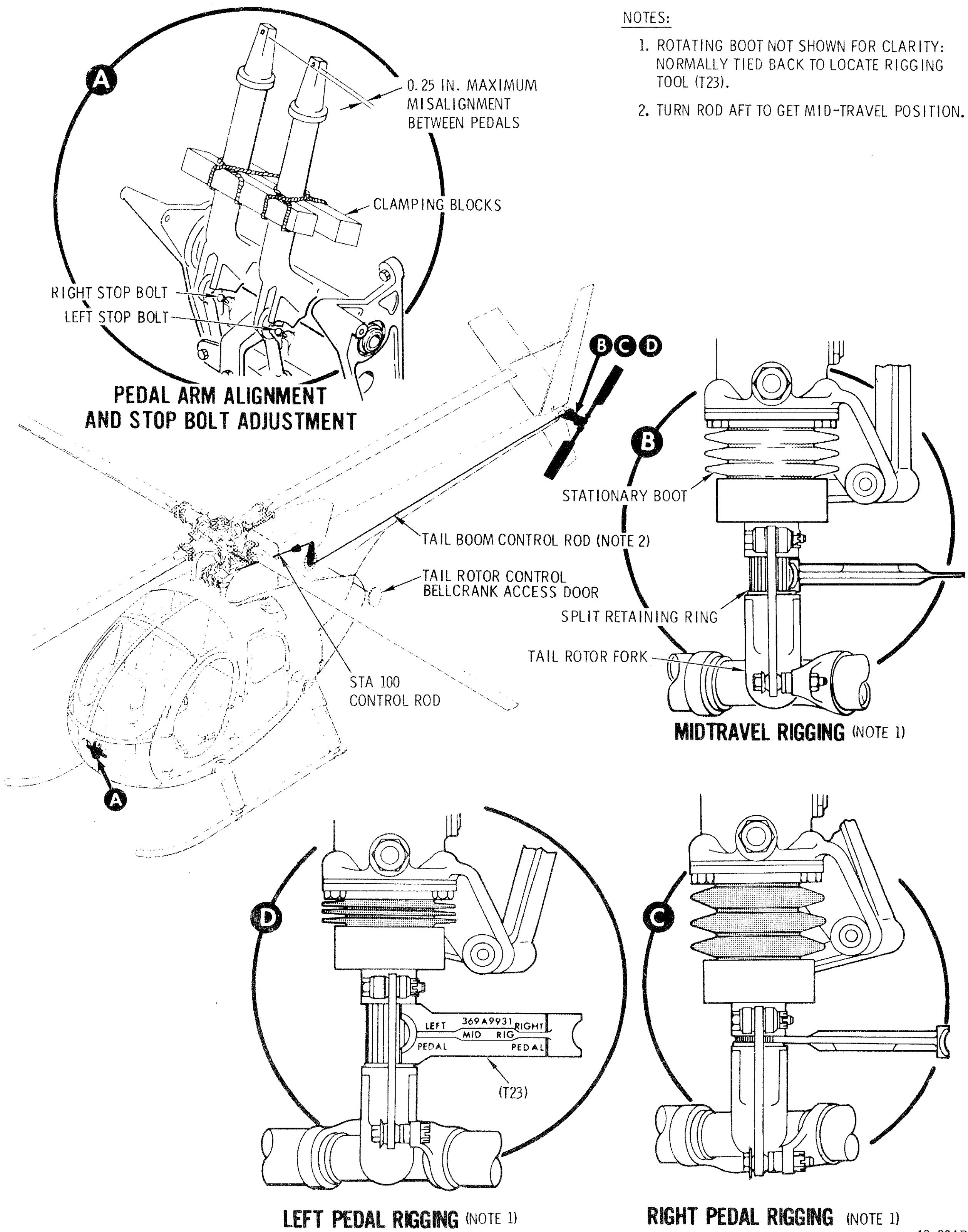


Figure 11-25. Tail Rotor Control System Rigging.

control rod attach bolt 5.0 ± 0.090 inches from the aft face of the station 137.50 bulkhead.

c. Gain access to station 142 bellcrank by removing the tail control bellcrank access door.

d. Loosen rod end checknut at forward end of tailboom control rod.

e. Remove hardware that attaches tailboom control rod to inboard end of station 282 bellcrank on tail rotor gearbox.

f. Tie back the tail rotor rotating boot. Place the midtravel portion of tail rotor swashplate rigging tool (T23) between the swashplate and the tail rotor fork split retaining ring (detail B, fig. 11-25).

NOTE

Check that the rigging tool is bearing against the split ring and not on the fork.

g. Turn rod at aft end. Adjust rod length to the nearest half turn of the rod end that will allow the tool to just slide between the split ring and swashplate.

h. Ensure that slotted bushing is in place in bottom ear of bellcrank. Connect tailboom control rod to station 282 bellcrank by using hardware removed and a new cotter pin.

i. Check witness hole for proper threat engagement. Align rod end to get approxmiately equal angular throw in both bellcranks, and tighten checknut at station 142 rod end. When tightening the jam nut at the adjustable end of a control rod, always hold the rod end with a wrench to prevent jamming of the bearing.

j. Remove clamping device from foot pedal arms.

k. Screw in both sets of pedal stop bolts approximately 0.50 inch.

CAUTION

Use care when actuating pedals to avoid the possibility of damaging the tool or the windshield.

l. Adjust right pedal stop bolt (detail A, fig. 11-25) so that when pedal arm is held against the stop (with 20 to 25 pounds pull or pressure) the right pedal portion of rigging tool will just slide between the split ring and swashplate (detail C). The fit should be tight enough to support the tool weight.

m. Adjust left pedal stop bolt (detail A) so that when pedal arm is held against the stop (with 20 to 25 pounds pull or pressure) the left pedal portion of rigging tool will just slide between the split ring and swashplate (detail D). The fit should be tight enough to support the tool weight. Remove rigging tool and restore stationary boot to normal position.

NOTE

Check tail rotor pedal stop bolts for minimum friction (bolts should not be finger loose). If bolts are finger loose, install Jam Nut, P/N AN 316-4R, on stop bolts and proceed to step "n".

n. Adjust the copilot's pedal stop bolts to match pilot's pedal travel with a 0.001- to 0.003-inch gap between stop bolts and pedals.

o. Reinstall all pedals. Slowly press the outboard pedal to its full travel position against the stop bolt. With not more than 20 pounds of pressure applied, **THE UPPER AND LOWER EDGES OF THE PEDAL MUST CLEAR THE CANOPY GLASS BY NOT LESS THAN 0.20 INCH.** Check both sets of pedals.

p. Operate pedals through full range of travel. While controls are being moved:

(1) Check that there is never less than 0.060 inch clearance around the station 100 control rod where it passes through the structure at station 137.50.

(2) Check that there is never less than 0.010 inch clearance around the tailboom control rod where it exits between the boom and tail rotor gearbox.

q. Reinstall station 142 bellcrank access door.

11-150. TAIL ROTOR PEDAL INSTALLATION.

11-151. Description — Tail Rotor Pedal Installation. The tail rotor pedal installation consists of the pilot's and copilot's pedals, pedal arms with interconnecting linkage, two pedal mounting brackets, a torque tube with control rod fitting, and a torque tube mounting bracket. (See fig. 11-26.) Forward pressure on either right pedal rotates the torque tube, moving control system linkage to increase tail rotor blade pitch. Pressure on either left pedal decreases tail rotor blade pitch.

11-152. Inspection — Tail Rotor Pedal Installation (Installed). (See fig. 11-26.) *a.* Remove controls access door and station 142 access door (chapter 2).

b. Check all control linkage attaching hardware for looseness by examining cotter pins, rod end rivets and jam nuts; try to finger-turn bolts. Check rods and bellcranks for corrosion.

c. Slowly press the outboard pedals to full travel position against the stop bolt. Check that corresponding pedals contact the stops simultaneously. With not more than 20 pounds of pressure applied, **THE UPPER AND LOWER EDGES OF THE PEDAL MUST CLEAR THE CANOPY GLASS BY NOT LESS THAN 0.20 INCH.**

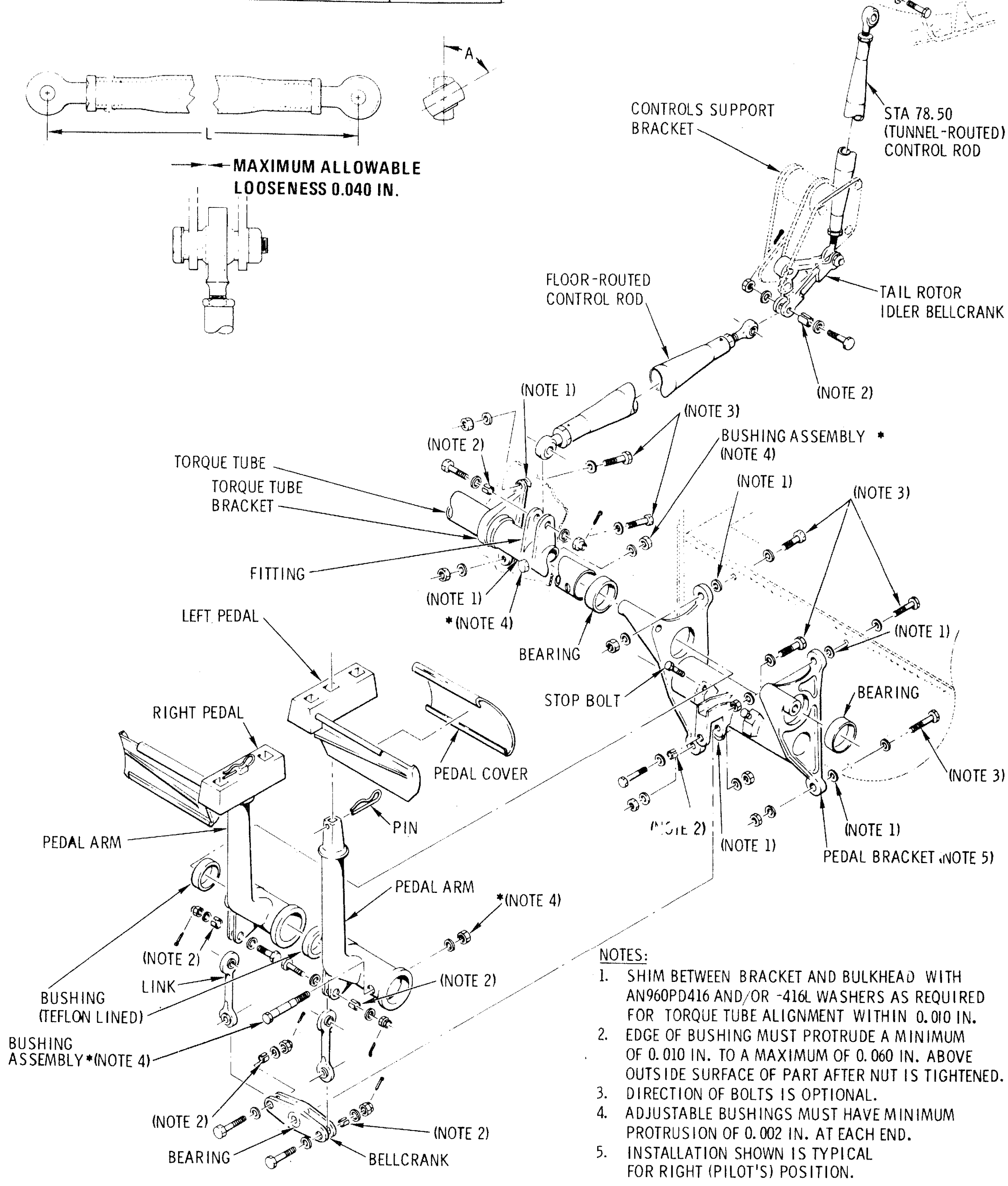
d. Operate pedals through full range of travel. While the controls are being moved:

(1) Check for binding.

(2) **CHECK THAT THERE IS NEVER LESS THAN 0.060 INCH CLEARANCE AROUND THE**

INITIAL ROD ASSY LENGTH AND BEARING ANGULARITY (BEFORE RIG)

CONTROL ROD ASSY	DIM L (IN.)	ANGLE A	ASSY PART NUMBER
STA 78.50	57.95	IN LINE	369A7007
FLOOR	30.06	IN LINE	369A7006-3



NOTES:

1. SHIM BETWEEN BRACKET AND BULKHEAD WITH AN960PD416 AND/OR -416L WASHERS AS REQUIRED FOR TORQUE TUBE ALIGNMENT WITHIN 0.010 IN.
2. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
3. DIRECTION OF BOLTS IS OPTIONAL.
4. ADJUSTABLE BUSHINGS MUST HAVE MINIMUM PROTRUSION OF 0.002 IN. AT EACH END.
5. INSTALLATION SHOWN IS TYPICAL FOR RIGHT (PILOT'S) POSITION.

Figure 11-26. Pilot's Compartment and Intermediate Tail Rotor Controls.

STATION 100 CONTROL ROD WHERE IT PASSES THROUGH THE STRUCTURE AT STATION 137.50.

(3) **CHECK THAT THERE IS NEVER LESS THAN 0.010 INCH CLEARANCE AROUND THE TAILBOOM CONTROL ROD WHERE IT EXISTS BETWEEN THE BOOM AND TAIL ROTOR TRANSMISSION.**

e. (See fig. 11-26.) Inspect pedal covers for loose bond and excessive wear.

f. Use straightedge to check pedal interconnecting torque tube for bends and similar distortions. Check for looseness of pedal teflon bushings and wear on torque tube.

g. Check four links and two bellcranks for cracks and bends (no bends or cracks allowed), and bearings for excessive looseness **(0.040 INCH AXIAL LOOSENESS MAXIMUM).**

h. Inspect control rod fitting, torque tube mounting bracket, and left and right pedal mounting brackets for cracks and corrosion. Using a 5-power magnifying glass, a mirror and a flashlight, closely inspect the pedal link bellcrank fitting of each pedal bracket, in the center forward area where the fitting (fork piece) joins the tubular section. If any cracks are detected, replace the bracket assembly. Inspect bracket bearings for excessive looseness.

i. Check rod end bearings for binding, looseness, and for jamming at maximum throw positions.

j. Install controls access door and station 142 bellcrank access door.

11-153. Removal — Tail Rotor Pedal Installation. (See fig. 11-26.) *a.* Pull the pivot pins from pilot's compartment floor access door hinges and remove doors.

b. Remove battery and/or avionics equipment as necessary for access.

c. Remove pins that secure pedal arms and remove pedals.

d. Remove cotter pin, nut, two washers, and bolt that connect floor-routed control rod to torque tube fitting.

e. Remove two nuts, four washers, any shim washers, and two bolts that secure torque tube bracket to bulkhead. (Keep shim washer selection with bracket to aid torque tube alignment during reinstallation.)

f. Remove four nuts, eight washers, any shim washers, and four bolts that secure each of the pedal brackets to the bulkhead. Remove the control pedal installation. (Keep shim washer selection with bracket.)

11-154. Disassembly — Tail Rotor Pedal Installation. (See fig. 11-26.)

NOTE

The following instructions are typical for both locations. Before disassembly, matchmark mating pedal arms, links and bellcranks with pedal brackets to avoid intermixing components between locations on reassembly.

a. Disconnect links from pedal arms and bellcrank; remove links.

b. Remove bellcrank from pedal bracket.

c. Remove nut and washer of the two bushing assemblies that secure the left pedal arm and control rod fitting to torque tube. Carefully remove bushing bolt and reinstall washer and nut on bolt.

d. Slide pedal bracket and pedal arms off end of torque tube.

e. Do not remove the two teflon-lined bushings from the right pedal arm unless replacement is required.

f. Do not remove the two stop bolts or bearings from the pedal bracket unless replacement is required.

g. Slide torque tube bracket and control rod fitting off torque tube.

11-155. Inspection — Tail Rotor Pedal Installation. *a.* Check pedals and pedal arms for cracks and elongated pedal attach holes. Inspect teflon-reinforced bushing liners for deterioration.

b. Check links and bellcrank for cracks and bends, and bearings for excessive looseness.

c. Check control rod fitting, torque tube mounting bracket, and pedal mounting bracket for cracks and corrosion. Check torque tube for cracks, scratches, nicks, dents, and similar surface defects.

11-156. Repair — Tail Rotor Pedal Installation. *a.* Replace parts that are cracked, or have elongated attachment holes. Do not attempt to straighten a bent torque tube or pedal link.

b. Replace unserviceable pedal covers. Apply compound (C89) to approximately 30 percent of the replacement cover upper and lower edge surfaces that contact the pedal; cure according to container instructions.

c. Replace complete pedal link if it is cracked or contains unserviceable bearings; bearings are not replaceable.

d. Replace unserviceable bearings in the mounting brackets or pedal link bellcrank. Install replacement bearing with grade A locking compound (C90).

CAUTION

The bushings at the right end of the pedal arm are glass-filled phenolic. Make certain that bushing does not cock during replacement, and that it is fully seated in arm bore. Keep tools from contact with bushing liners to avoid fraying of liner.

e. Replace bushings in right pedal arm when teflon-reinforced liner is galled or frayed. Pull bushing to remove; press to install.

11-157. Assembly — Tail Rotor Pedal Installation. (See fig. 11-26.)

NOTE

The following instructions are typical for both locations. When reassembling components, check for correct matchmark identification.

a. If previously removed, install two stop bolts into threaded inserts of pedal bracket.

b. Slide torque tube bracket and control rod fitting onto torque tube.

c. Locate pedal bracket and pedal arms for correct position on torque tube; slide bracket and arms onto torque tube.

d. Fasten left pedal arm and control rod fitting to torque tube with the adjustable bushing assemblies; install the bushing bolts so that the shank at the bolt head end protrudes at least 0.002 inch. Install nut finger-tight; then use a torque wrench to **CHECK DRAG TORQUE OF NUT ON BOLT. ADD 50 — 80 INCH-POUNDS TO THE DRAG TORQUE AND TIGHTEN THE NUT TO THIS VALUE.** Do not allow the bolt to turn.

e. Check that each end of the bushing assembly shank protrudes at least 0.002 inch. If not, loosen and repeat *d* above.

f. Attach bellcrank to pedal bracket with bolt, two washers, nut, and new cotter pin.

g. Check that slotted bushings are in place; then fasten each link to bellcrank with bolt, two washers, nut, and new cotter pin.

h. Check that slotted bushings are in place; then fasten each link in pedal arm lugs with bolt, two washers, nut, and new cotter pin.

11-158. Installation — Tail Rotor Pedal Installation. (See fig. 11-26.) *a.* Install each of the pedal brackets with four bolts, eight washers, and four nuts. Add shim washers as required at pedal bracket attachment points to keep centerline alignment of pedal bracket bearings within 0.010 inch. **CORRECT ALIGNMENT IS INDICATED BY FREE ROTATIONAL MOVEMENT OF TORQUE TUBE AFTER THE NUTS ARE TIGHTENED.**

b. Position torque tube bracket over mating holes in bulkhead and check alignment with bulkhead. Add shim washers as required at bracket attachment points to maintain the bearing alignment, established in *a* above. Install two bolts, four washers, and two nuts.

c. Check that slotted bushing is in place; then install floor-routed control rod fitting and install bolt, two washers, nut, and new cotter pin.

d. Check rigging of tail rotor controls and pedal-to-canopy clearance (para 11-149) following installation of any removed or replaced parts.

11-159. FLOOR- AND TUNNEL-ROUTED CONTROL RODS.

11-160. Description — Floor- and Tunnel-Routed Control Rods. The floor-and tunnel-routed control rods interconnect the tail rotor pedal installation with the upper fuselage and tailboom control linkage (fig. 11-24). The control rods, each an anodized, aluminum alloy tube with two rod end bearings, are linked by the tail rotor idler bellcrank mounted on the underseat controls support bracket.

11-161. Inspection — Station 78.50 Tail Rotor (Tunnel-Routed) Control Rod. Refer to paragraph 11-40. All the tunnel-routed control rods are inspected in the same manner.

11-162. General Repairs — Floor-and Tunnel-Routed Control Rods. The floor-routed control rod is not removable unless the lower canopy fiberglass panel is removed or an exit hole is cut in the panel. Normally, the rod will not require replacement unless the aircraft has been extensively damaged as a result of crash or similar impact. The rod end bearings, however, are replaceable without removing the control rod from its location under the compartment floor. (Refer to para 11-40 and 11-47 for tunnel-routed rod and idler bellcrank.)

NOTE

Check rigging of tail rotor controls following installation of any removed or replaced parts (para 11-146).

11-163. Replacement — Forward Rod End (Floor-Routed Rod). Replace the forward rod end if bearing axial play is more than 0.040 inch.

a. Disconnect control rod from pedal torque tube fitting (fig. 11-30).

b. Accurately measure and record distance between unchamfered edge of rod end jam nut and center of rod end bearing hole.

CAUTION

Use care when drilling to remove or install riveted rod end; the rod end is steel and the rod is aluminum.

c. Drill out rivet, loosen jam nut, and unscrew rod end.

d. Screw replacement rod end into rod, and adjust jam nut to obtain same measurement made in *b* above.

e. Install rivet to secure rod end.

f. Check that slotted bushing is in place; then secure control rod in torque tube fitting with bolt, two washers, nut, and new cotter pin.

11-164. Replacement — Aft Rod End (Floor-Routed Rod). Replace the aft rod end **IF BEARING AXIAL PLAY IS MORE THAN 0.040 INCH.**

a. Gain access to lower controls through controls and foot fairing access doors in canted bulkhead station 78.50.

b. Disconnect control rod from tail rotor idler bellcrank (fig. 11-26).

c. Replace (unriveted) rod end by the method described in step *b* of paragraph 11-163. Align control rod ends so that the bearings have equal space for angular throw; hold rod end and tighten jam nut.

d. Check that slotted bushing is in place; then secure control rod in tail rotor idler bellcrank with bolt, two washers, nut, and new cotter pin.

e. Reinstall access doors.

11-165. TAIL ROTOR IDLER BELLCRANK.

11-166. Description — Tail Rotor Idler Bellcrank. Refer to paragraph 11-47.

11-167. UPPER FUSELAGE AND BOOM TAIL ROTOR CONTROL LINKAGE.

11-168. Description — Upper Fuselage and Boom Tail Rotor Control Linkage. Upper fuselage tail rotor controls consist of the station 95 bellcrank and support at the front of the main rotor mast base, the station 100 control rod, and the station 142 bellcrank inside the boom fairing (fig. 11-27). The boom linkage consists of the tailboom control rod, supported by six teflon-lined grommets, and the station 282 bellcrank that is mounted on the tail rotor transmission.

11-169. Inspection — Upper Fuselage and Boom Tail Rotor Control Linkage (Installed). *a.* Remove station 142 bellcrank access door and boom bolts access doors (chapter 2).

b. Check rod end bearings for binding and excessive wear **(0.040-INCH MAXIMUM AXIAL LOOSENESS)**. Check for loose rivet at fixed rod ends.

c. Check control rods for surface damage and evidence of bending. Control rods that are bowed must be straightened (para 11-40).

d. Check boot and grommet at station 137.50 bulkhead for good condition. On aircraft with air filter installed, inspect boots for cuts, deterioration, and security of attachment. Clean any accumulations of dirt from the grommet to minimize abrasive wear on the control rod. Inspect station 100 control rod for chafing of the surface that feeds through the boot/grommet or air filter installation boots; **SERVICRABLE WEAR IS LIMITED TO 0.004 INCH (BELOW PAINT).**

e. Check for wear of the tailboom control rod surfaces that pass through the control rod grommets. Serviceable wear is limited to the thickness of the hard anodic coating. Check that all grommets are in place.

f. Check bellcranks and control support for scratches cracks, corrosion and similar surface defects. Refer to table 11-13 for rework and repair limits.

g. Check bellcrank bearings for looseness and binding.

h. Check that station 282 (tall rotor gearbox) bellcrank drain slot, if provided, is free of obstruction.

i. Reinstall access doors.

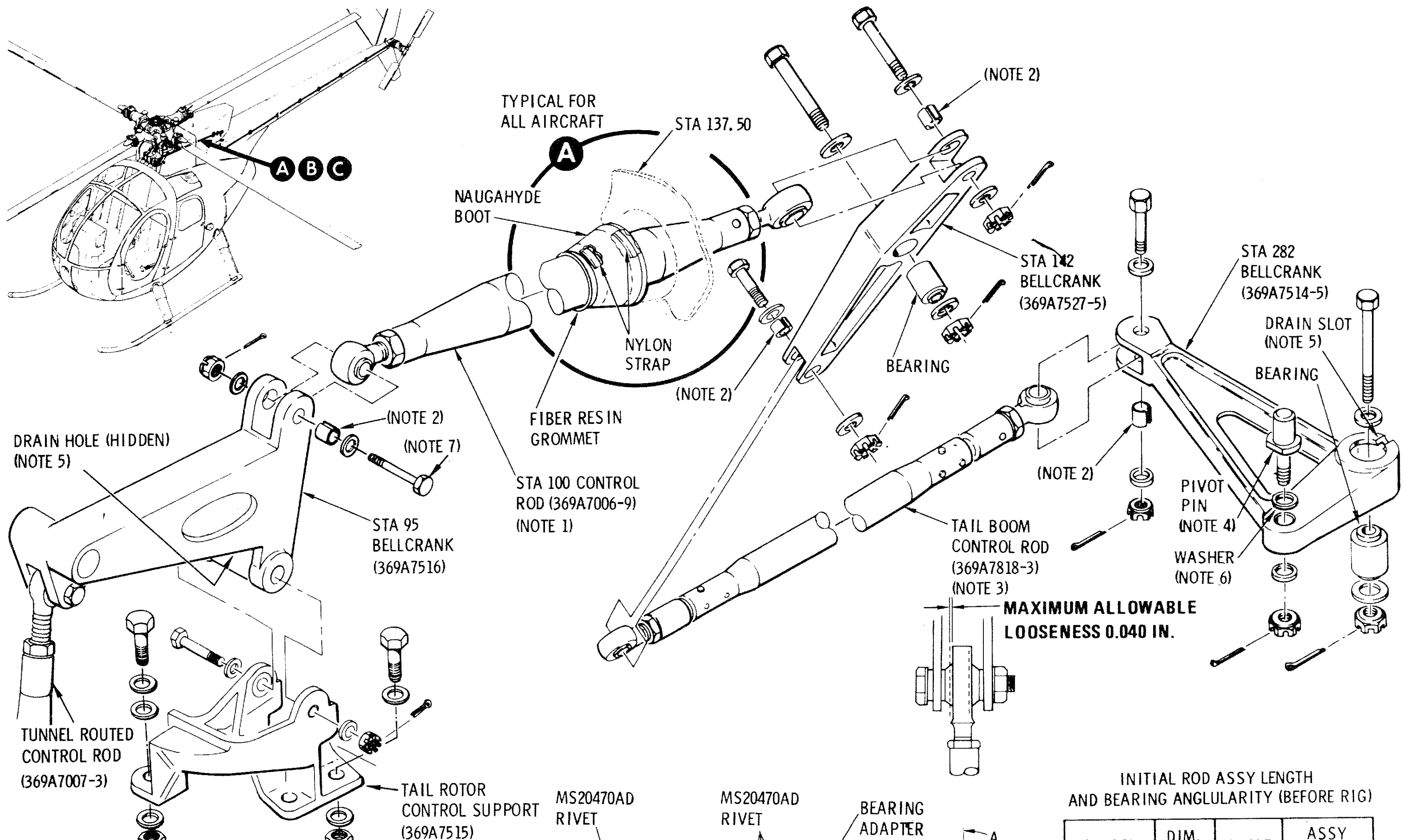
11-170. Removal — Station 100 Control Rod or Station 142 Bellcrank. (See fig. 11-27, sh 1 and 2.) The station 142 bellcrank and station 100 control rod must be removed as an assembled unit.

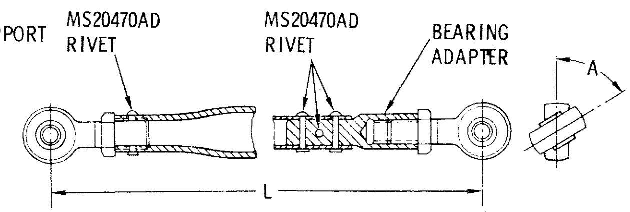
a. Remove tail rotor control bellcrank access door from left side of boom fairing.

b. Disconnect tailboom control rod from lower end of station 142 bellcrank.

c. Remove hardware that secures station 142 bellcrank to boom fairing brackets.

d. Remove plenum chamber access door or open filter bypass door, as applicable, at right side of air intake aft fairing. On aircraft with a barrier filter, remove filter element and loosen or remove straps on the two boots (detail B, sh 2). On aircraft with an inertial particle separator air filter, remove left half of air inlet front fairing; then loosen or remove straps and clamps on the two boots (details A and C, sh 1 and 2). If no filter is installed, loosen or remove straps on boot at station 137.50 (detail A, sh 1).





INITIAL ROD ASSY LENGTH AND BEARING ANGLULARITY (BEFORE RIG)

CONTROL ROD ASSY	DIM. L (IN.)	ANGLE A	ASSY PART NO.
STA 100	45.88	IN LINE	369A7006-5
TAILBOOM	141.70	63°	369A7518

NOTES:

1. MINIMUM OF 0.060 IN. CLEARANCE FOR FULL RANGE OF TRAVEL THRU STA 137.50 BULKHEAD.
2. EDGE OF BUSHING MUST PROTRUDE A MINIMUM OF 0.010 IN. TO A MAXIMUM OF 0.060 IN. ABOVE OUTSIDE SURFACE OF PART AFTER NUT IS TIGHTENED.
3. MINIMUM OF 0.010 IN. CLEARANCE FOR FULL RANGE OF TRAVEL BETWEEN BOOM AND TAIL ROTOR GEARBOX.
4. PIN MUST BE CHROME-PLATED OR NITRIDED.
5. NOT PRESENT ON ALL AIRCRAFT.
6. TO PREVENT CORROSION, INSTALL AN960PD516L WASHER AS SOON AS POSSIBLE.
7. LOCATE BOLT HEAD IN DIRECTION SHOWN.

12-231E

Figure 11-27. Upper Fuselage and Boom Tail Rotor Control Linkage. (Sheet 1 of 2)

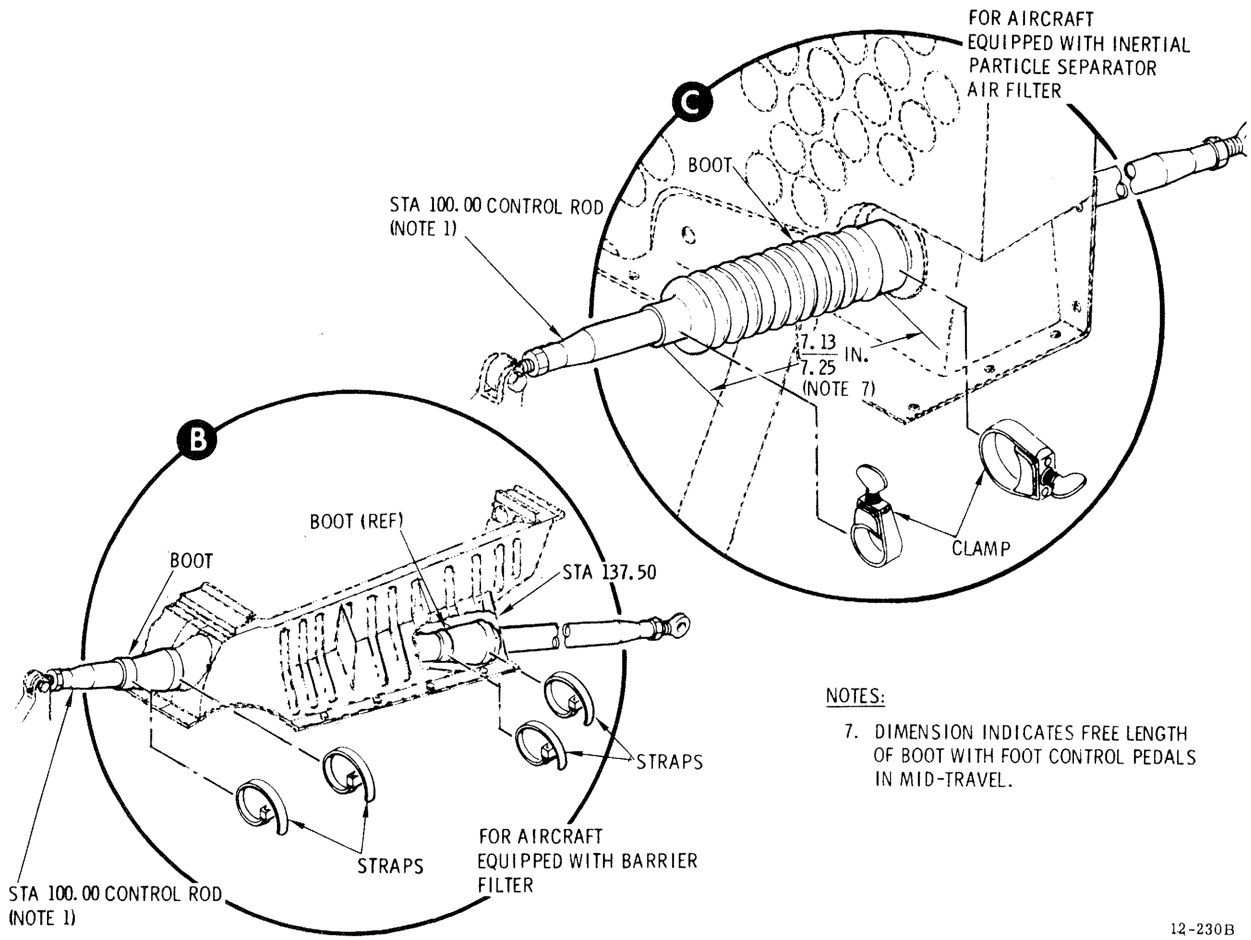


Figure 11-27. Upper Fuselage and Boom Tail Rotor Control Linkage. (Sheet 2 of 2)

Table 11-13. Tail Rotor Control Component Rework and Wear Limits.

Nomenclature	Part No. (fig. 11-27)	Material & Type	Rework Limits (para 11-39)
Sta 282 bellcrank	269A7514-5	*Mag casting	*a, b, c, d, e*
Tail rotor control support	369A7515	Mag casting	*a, b, c, d, e*
Sta 95 bellcrank	369A7516	Mag casting	*a, b, c, d, e*
Sta 142 bellcrank	369A7527-5	Mag casting	*a, b, c, d, e*

*Mag = Magnesium

e. Disconnect forward end of station 100 control rod from station 95 bellcrank.

NOTE

Depending on bolt direction, the station 95 bellcrank may also have to be disconnected from its support to get clearance from the air inlet fairing for bolt removal at the forward end.

f. Carefully pull at forward end of control rod and withdraw assembled rod and bellcrank through boot hole in plenum chamber, and the openings in the filter frame as applicable.

g. Disconnect control rod from upper end of station 142 bellcrank.

11-171. Removal — Station 282 Bellcrank. (See fig. 11-27.) *a.* Disconnect bellcrank from tailboom control rod.

b. Disconnect bellcrank from tail rotor transmission arm. Disengage bellcrank pitch control pin from tail rotor assembly and remove bellcrank.

11-172. Removal — Tailboom Control Rod. (See fig. 11-27.) *a.* Disconnect control rod from station 282 and station 142 bellcranks.

b. Accurately measure and record distance between forward rod end bearing center and end of control rod. Loosen jam nut and unthread control rod end from control rod.

c. Slowly pull at aft end of control rod and withdraw rod through opening in tail rotor transmission mounting frame.

11-173. Removal — Station 95 Bellcrank and Support. (See fig. 11-27.) *a.* Disconnect tunnel-routed control rod from bellcrank.

b. Disconnect bellcrank from support.

c. Disconnect bellcrank from station 100 control rod and remove bellcrank.

d. Remove hardware that secures support to mast structure; remove support.

11-174. Inspection (Removed) — Upper Fuselage and Boom Tail Rotor Control Linkage. *a.* Check rod end bearings for binding and excessive wear **(0.040-INCH MAXIMUM AXIAL LOOSENESS.)** Check for loose rivet at fixed rod end (fig. 11-27, sh 1).

b. Inspect control rods for surface damage and evidence of bending.

c. Check boot and grommet at station 137.50 bulkhead and boots used with air filter installations for condition. Clean any accumulation of dirt from the grommet to minimize abrasive wear on the control rod. Check station 100 control rod for chafing of the surface that feeds through the boot/grommet or air filter installation; **SERVICEABLE WEAR IS LIMITED TO 0.004 INCH.**

d. Check for wear of the tailboom control rod surfaces that pass through the control rod grommets (six places). Serviceable wear is limited to the thickness of the hard anodic coating. Check that all grommets are secure.

e. Inspect bellcranks for scratches, cracks, corrosion and similar surface defects.

f. Check bellcrank bearings for looseness and binding.

11-175. Repair — Upper Fuselage and Boom Tail Rotor Control Linkage. *a.* Perform a straightness check on a control rod that appears bent or bowed. The total length of the station 100 control rod (excluding rod ends) must be straight within 0.040 inch, with straightness variation limited to a maximum of 0.010 inch in each foot of length. The total length of the tailboom control rod (excluding rod ends) must be straight within 0.120 inch, with straightness variation limited to a maximum of 0.010 inch in each foot of length.

CAUTION

A fluorescent penetrant inspection (TM 55-1500-204-25/1) for cracking shall always be performed after cold-straightening. Replace a cracked rod, or a cracked or bent rod end.

b. Cold-straighten a bent rod that is not within tolerance, *a* above, provided there are no nicks or sharp dents in the bend length. Do NOT use the rod ends to support the rod during the straightening process.

CAUTION

Use care when drilling to remove or install riveted rod end; the rod end is steel and the rod and bearing adapter (fig. 11-27, sh 1) are aluminum.

c. Replace a control rod end if bearing axial play is more than 0.040 inch. Set initial control rod length and bearing angularity as shown in figure 11-27, sh 1.

d. Replace unserviceable control rod bearing adapter. The trammel method (or equivalent) may be used to establish rod length when replacing an adapter.

(1) Measure length and record position of rod end in adapter; then remove affected rod end.

(2) Observe the CAUTION note above and drill out rivets that secure adapter to rod, and aft rod end.

(3) Install rod end in replacement adapter and position at recorded measurement.

(4) Fit adapter into rod and position to fit trammel point spacing; pick up and drill through existing rivet holes in rod and install rivets to secure adapter to rod; at aft end, drill and rivet rod end to adapter.

e. Replace bellcranks that are damaged beyond repair limits specified in table 11-13.

f. Replace unserviceable bellcrank and support pivot bearings. Install replacement bearing with grade A locking compound (C90). Check that bearing seats against bore shoulder.

NOTE

The installation direction of station 142 bellcrank bearing may be from either side, depending on location of the bearing shoulder. There must be not less than 0.010 inch protrusion of the bearing inner race beyond the sides of the bellcrank after the bearing is installed.

g. Polish out minor scratches, nicks, dents, and other similar surface defects on bellcranks and restore protective finish. Refer to table 11-13 for repair limits.

h. Repair surface abrasion on station 100 control rod by smoothing the area with grade 400 abrasive cloth (C24) and restoring protective finish with primer (C79). **REPLACE ROD IF WEAR OR DEPTH OF REPAIR EXCEEDS 0.004 INCH.**

i. Replace tailboom control rod if hard anodic coating is worn through and aluminum base metal is visible.

j. Replace split or torn boots, or a cracked grommet. Remove station 100 control rod (para 11-170). Replace grommet or boot, as required, and reinstall station 100 control rod (para 11-176).

k. Replace station 282 bellcrank pivot pin if worn or corroded. Install washer (AN960PD516L) between pin and bellcrank.

11-176. Installation — Station 100 Control Rod or Station 142 Bellcrank. (See fig. 11-27.) The station 142 bellcrank and station 100 control rod must be installed as a unit by routing assembled end of bellcrank and control rod through boot holes in filter frame or panel, and/or in plenum chamber.

a. Gain access to bellcrank support bracket through access door in boom fairing.

b. Check that slotted bushings are in place at each end of bellcrank; then position riveted rod end in bellcrank and install bolt, two washers, nut, and new cotter pin. Check to make sure that the hump of the bellcrank will face forward.

c. Carefully feed assembled control rod and bellcrank through forward air intake at left side of control mixer installation while an assistant guides the assembly through the boot holes to support bracket.

d. Position station 142 bellcrank in support bracket and install bolt, two washers, nut, and new cotter pin.

e. Install and secure boots and clamps or straps, as applicable. (See details A, B and C, sh 1 and 2.)

f. Check that slotted bushing is in place in station 95 bellcrank; then install forward rod end in bellcrank with bolt (head OUTBOARD), two washers, nut, and new cotter pin.

g. Reinstall station 95 bellcrank in its support with bolt, two washers, nut, and new cotter pin.

h. Check that slotted bushing is in place in lower end of station 124 bellcrank; then attach tailboom control rod to bellcrank with bolt, two washers, nut and new cotter pin.

i. Check rigging of tail rotor controls after installation of any removed or replaced parts (para 11-149).

11-177. Installation — Station 282 Bellcrank. (See fig. 11-27). *a.* Position tail rotor pitch control assembly to align with pitch control pin in bellcrank and engage pin.

b. Position bellcrank to align with mating hole in tail rotor gearbox arm and install bolt, two washers, nut and new cotter pin.

c. Check that slotted bushing is in place in lower lug of bellcrank; then attach tailboom control rod to bellcrank with bolt, two washers, nut and new cotter pin.

d. Check rigging of tail rotor controls after installation of any removed or replaced parts (para 11-149).

11-178. Installation — Tailboom Control Rod. (See fig. 11-26, sh 1.)

NOTE

The tailboom control rod is routed through the boom with the forward rod end removed. For replacement of tail rotor control rod grommets, refer to chapter 2.

a. Guide control rod through boom aft frame opening and carefully route through the six control rod grommets. Rotate the rod slightly to start it through each grommet.

b. Check that slotted bushing is in place in lower lug of station 282 bellcrank; then attach aft rod end to bellcrank with bolt, two washers, nut and new cotter pin.

c. Using measurement recorded at time of rod removal, install forward rod end bearing. Align rod ends so that bearings have equal space for angular throw; hold rod end and tighten jam nut.

d. Check that slotted bushing is in place in lower end of station 142 bellcrank; then attach forward rod end to bellcrank with bolt, two washers, nut and new cotter pin.

e. On aircraft with a barrier filter, replace filter element. On aircraft with an inertial particle separator air filter, replace left half of air inlet front fairing. Reinstall plenum chamber access door or close filter bypass door, as applicable. (Refer to chapter 2.)

f. Check rigging of tail rotor controls after installation of any removed or replaced parts (para 11-149).

11-179. Installation — Station 95 Bellcrank and Support. (See fig. 11-27.) *a.* Install bellcrank support to mast structure with three bolts, seven washers, and three nuts. The forward bolt requires an extra washer (AN960PD416) under the bolthead; two thin washers are used at all three locations. **TORQUE NUTS TO 80 — 100 INCH-POUNDS.**

b. Check that slotted bushings are in place in bellcrank; then attach station 100 control rod to bellcrank with bolt (head to right), two washers, nut, and new cotter pin.

c. Install bellcrank in support with bolt, two washers, nut and new cotter pin.

d. Attach tunnel-routed control rod to bellcrank with bolt, two washers, nut, and new cotter pin (fig. 11-26).

e. Check rigging of tail rotor controls after installation of any removed or replaced parts (para 11-149).

CHAPTER 12
UTILITY SYSTEMS

(Not Applicable)

CHAPTER 13

ENVIRONMENTAL CONTROL SYSTEM

SECTION I HEATING SYSTEM

13-1. HEATING SYSTEM.

13-2. Description — Heating System. The heating system (fig. 13-1 and 13-2) consists of a heat mixing valve, a manual heat valve control and a series of hot and cold air ducts. The heat control valve mixes compressor bleed air (hot air) and ambient outside air (cold air). The heat level or mixture of air is adjusted through a cable by a pilot operated control lever. A system of ducts and tubes routes heated air to both the pilot's and cargo compartments. The heating duct system is associated with the engine cooling duct system (chapter 4). Both duct systems receive forced cold air input from a manifold duct attached to the oil cooler blower. The blower is driven by the main transmission drive shaft. Temperature sensing controls and indicating systems are not used in the heating system.

13-3. Inspection — Heating System. *a.* Inspect heat valve control lever and cable for secure attachment to canopy frame.

b. Remove left aft bulkhead sound insulation blanket and access cover in cargo compartment. Inspect cable for condition and secure attachments.

c. Inspect universal joint at valve end of cable for offset angle, from cable centerline at clamp, **OF NOT MORE THAN 5 DEGREES**. Reposition clamp, if necessary.

d. Inspect cable clevis for secure lockwire at spring pin in valve pulley arm.

e. Inspect for secure attachment of heat control valve on firewall. Check pulley belt for condition. Check for tight connection at heat control valve bleed air fitting (if used).

f. Inspect all ducts and coupling hardware for secure attachment.

g. Inspect flexible fiberglass hose for tears. Check for deformation due to failure of inner supporting wire or short radius bends.

h. Inspect formed fiberglass and polycarbonate sections for damage allowing air leaks.

i. Inspect rubber hose for tears and deterioration.

j. Check cargo compartment control valve for positive opening and closing of the valve vane when control rim is turned.

k. Open left engine access door. Inspect tube connecting engine air bleed fitting to control valve for cracks and security.

l. Close access doors and reinstall bulkhead access cover and blanket.

13-4. Troubleshooting — Heating System. Refer to table 13-1.

13-5. HEAT DISTRIBUTION DUCTS.

13-6. Description — Heat Distribution Ducts. The heat distribution ducting system (fig. 13-2) consists of various lengths of rigid and flexible tube shapes made of: rubber compound; laminated, 3-ply resin impregnated fiberglass; spring supported, 2-ply fiberglass fabric, aluminum alloy; or clear polycarbonate plastic. The ducting system begins at the firewall, behind the cargo compartment aft bulkhead access door. (See fig. 13-2.) The ducting is routed overhead through the cargo compartment and into the pilot's compartment. Six fan-shaped diffusors direct heated air against the canopy. An additional heated air outlet for the cargo compartment area is mounted on the aft bulkhead. The outlet is located between the troop seats. A valve allows adjustments of air flow. Duct coupling is by bolted flanges, closely fitted slip joints (some sealed by O-rings), clamped joints, or by bonding at the two lowest diffusors.

13-7. Removal — Heat Distribution Ducts. Use figure 13-2 as a guide for removal of heat distribution ducts. Remove only those ducts requiring repair.

13-8. Repair — Heat Distribution Ducts. Repair of ducting will depend upon the type of material used in the section that requires repair. Replace any rubber hose or flexible fiberglass hose that is torn, flattened, or deteriorated.

a. Repair rigid fiberglass and plastic sections by patching according to TM 55-1500-204-25/1.

b. Repair small areas of rib or seam separation in

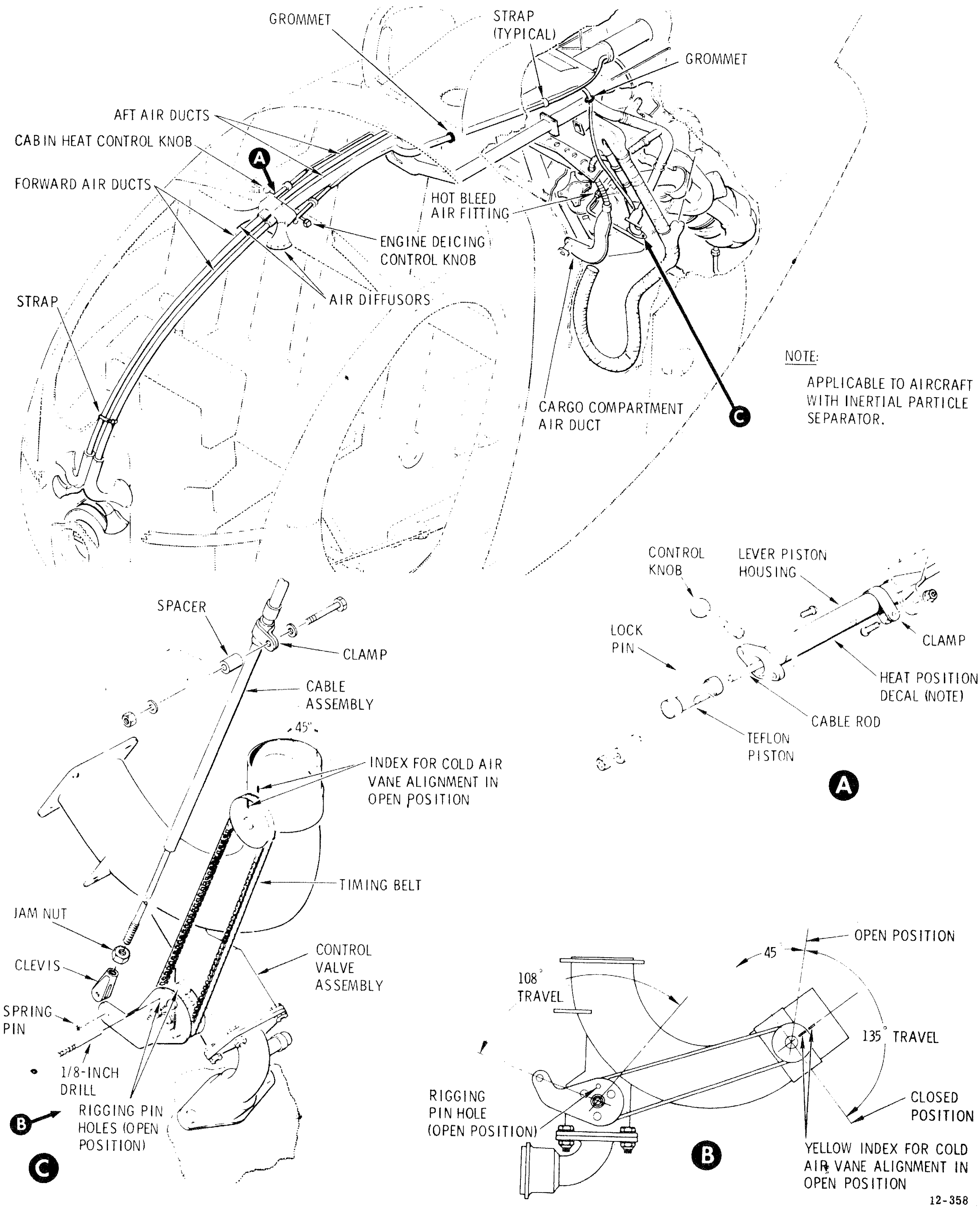
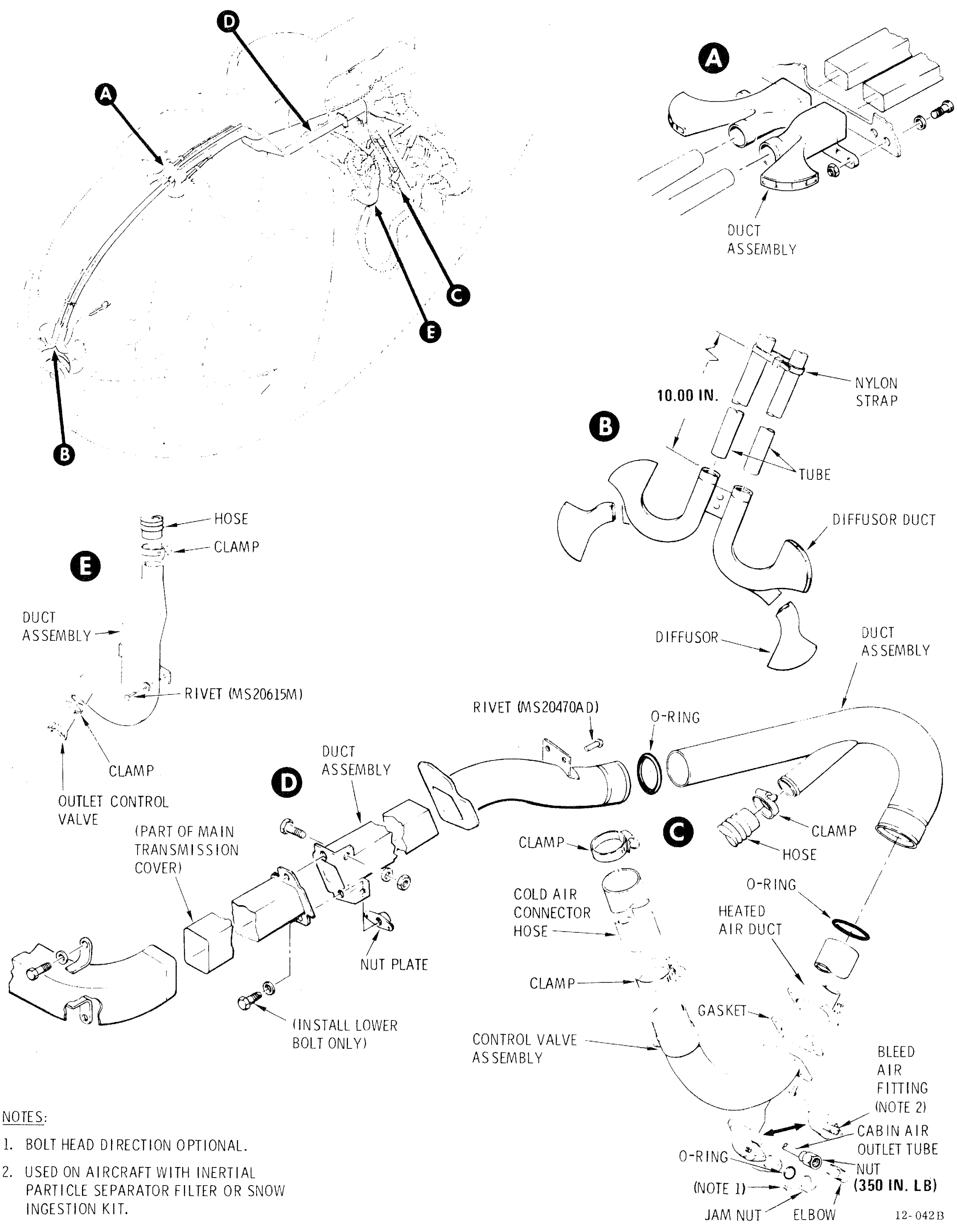


Figure 13-1. Heating System.



NOTES:

1. BOLT HEAD DIRECTION OPTIONAL.
2. USED ON AIRCRAFT WITH INERTIAL PARTICLE SEPARATOR FILTER OR SNOW INGESTION KIT.

Figure 13-2. Heat Control Valve and Distribution Ducts.

Table 13-1. Troubleshooting the Heating System.

MALFUNCTION

TEST OR INSPECTION

CORRECTIVE ACTION

NOTE

Before you use this table, be sure you have performed all normal operational checks. If you have a malfunction which is not listed in this table, notify the next higher level of maintenance.

1. **Actuation of heat control lever produces no heated air or insufficient air at outlets.**

STEP 1. Inspect ducting system for cracked, broken or disconnected ducts or hoses.

If a defect in the ducting system is located, repair or replace as required (para 13-5).

STEP 2. Check for correct control cable rigging (para 13-10). Position heat control knob fully aft. Insert a 1/8-inch drill shank through the heat control valve lever into the valve boss.

If lever and valve boss rigging hole does not align, readjust cable end clevis (para 13-10).

STEP 3. Check that the control valve pulley drive belt is not slipping or broken.

If belt is slipping or broken, replace belt and adjust valve (para 13-10).

STEP 4. Check for sheared pulley shaft retaining pin or control cable spring pin.

If a pin is sheared, replace the pin (para 13-20).

STEP 5. Check that heat control valve internal ball is free and rotating to correct open position viewed through hot side of valve.

If valve operation is defective, remove and disassemble valve for repair (para 13-10).

2. **Movement of heat valve control lever does not shut off hot air distribution.**

STEP 1. Operate control knob and check for motion at the heat control valve cable end.

If cable does not move when knob is operated, the piston lock pin is sheared within the lever housing and must be replaced (para 13-20).

STEP 2. Perform steps 2 through 5 in Malfunction No. 1 as required.

3. **Heat control knob difficult to operate.**

STEP 1. Disconnect control cable at heat control valve and operate control knob.

If control knob actuation is still difficult, inspect for crushing of the cable sheath and bend radii less than the 3-inch minimum. Reposition or replace the cable as applicable (para 13-20).

STEP 2. Check for binding at cable universal joint. With the cable disconnected, operate the control knob while moving cable at the clamp nearest the heat control valve.

If cable does not free-up, install a serviceable cable assembly (para 13-20).

4. **Heat valve control lever does not remain in selected position.**

STEP 1. Inspect the cable sheathing for crushed, damaged or kinked areas causing preload or spring back.

If damage cannot be repaired, install a serviceable cable (para 13-20).

CAUTION

Do not operate engine until the following condition is corrected. Canopy plastic or ducting may be badly damaged if operation is continued.

5. **Air excessively hot at air diffusers.**

STEP 1. Check for correct heat control valve rigging (para 13-10). Position the valve lever and install a 1/8th-inch drill shank through the lever into the valve boss.

If index marks on the cold air vane do not align, rerig the valve timing belt (para 13-3).

plastic sections with dichloromethane (C38). Inject into the void area and clamp together under light pressure.

c. Repair cracks in polycarbonate material (para 2-15h).

13-9. Installation — Heat Distribution Ducts. Use figure 13-2 as a guide for installation of heat distribution ducts.

13-10. HEAT CONTROL VALVE.

13-11. Description — Heat Control Valve. The heat control valve (fig. 13-1) consists of a valve housing that contains the air control ports for mixing hot and cold air flow. A 90-degree inlet elbow attaches and ports the control valve through the firewall to the engine air bleed tube. The control valve is mounted on the forward left side of the station 124.00 canted bulkhead. The inlet elbow may have a bleed fitting that is used for inertia particle separator filter air or snow ingestion kit heat. The housing inlet body contains a centerbored stainless steel valve check ball. The ball is rotated by actuation of the drive pulley to control the inlet of engine bleed air. A nozzle on the inlet body injects the hot air downstream of the cold air inlet. Hot air is mixed with blower (cold) air at a variable ratio, depending on the relative positions of the hot air valve check ball and the cold air vane. Because engine bleed air is approximately 260°C (500°F), cold air is never completely shut off.

13-12. Operational Check and Adjustment — Heat Control Valve. *a.* Move control lever knob to full aft (heat valve open) position. Lever should remain at the slot stop with no creep away from the stop.

b. Remove left aft bulkhead sound insulation blanket and access cover in cargo compartment.

c. Check that the valve is in the full open position by inserting a 1/8-inch drill shank (or rod) into the drive pulley rigging pin hole (fig. 13-1). The drill shank should also enter the mating hole in the valve housing inlet body.

d. Check the cold air vane driven pulley for a yellow alignment mark. If present, the mark should align with a similar mark on the cold air vane housing.

NOTE

If holes or marks do not align, refer to l below. If holes and marks do align, complete the check according to g, j, and m below. If there are no yellow alignment marks, continue check with the following steps.

e. Remove the clamp that secures lower end of rubber air duct connector to valve housing tube fitting (just above driven pulley); then remove the duct connector.

f. Check position on the cold air vane in housing tube fitting throat. The cold air vane should be in a diagonal position **(AT 45 DEGREES)** across the throat opening. In this position, there is a gap between edge of vane and inner surface of tube fitting. The vane-to-pulley shaft screw heads should not be visible.

g. Remove drill shank from pulley alignment holes. Move control lever knob to full forward (heat valve closed) position. The lever should remain at slot stop with no creep away from the stop. Refer to *l* below.

h. Check position of control valve pulley. Vertical centerline alignment of the clevis hole in pulley arm and pulley pivot shaft should be nearly the same.

i. Check position of cold air vane in housing tube fitting throat. The cold air vane should be positioned horizontally across the throat opening and the vane attachment screw heads should be visible.

j. When the control cable operation is stiff, isolate the trouble as follows. Remove the lockwire and spring pin from the cable clevis; then check pull at the lever end with a suitable spring scale. **PULL REQUIRED TO ACTUATE THE CABLE SHOULD NOT EXCEED 1 POUND.** Reinstall spring pin and secure with lockwire after pull test.

k. If valves are correctly positioned in both open and closed positions of the control lever, reinstall rubber connector and secure with clamp.

l. If valves do not position correctly, or if valve lever does not have enough travel, either the control cable and/or drive belt need adjustment.

m. Reinstall left aft bulkhead cover and insulation blanket.

13-13. Removal — Heat Control Valve. (See fig. 13-2.) *a.* Remove left aft bulkhead access cover in cargo compartment.

b. Remove spring pin attaching cable clevis to control valve lever

c. In the left side of the engine compartment, disconnect the cabin air outlet tube at the control valve elbow fitting.

d. Loosen the elbow fitting jamnut and remove elbow and preformed packing (O-ring).

e. Remove hardware that attaches the lower end of control valve assembly to the canted bulkhead.

f. On valve assemblies with a bleed air fitting, disconnect air tube from valve elbow fitting.

g. Remove the clamp that secures cold air connector hose to valve.

h. Remove valve-to-heated air duct attaching hardware; then remove control valve assembly and gasket.

13-14. Disassembly — Heat Control Valve. (See fig. 13-3.) *a.* Rotate the ball valve lever fully counterclockwise to position the ball valve open.

b. Remove hardware that attaches elbow to valve housing. Remove the elbow.

c. The O-ring, wave washer, retainer, ball valve, and coupling are now free for removal.

d. Cut lockwire and remove three screws from valve lever and pulley. Remove valve lever timing belt.

e. Remove pin from ball valve pulley. Remove pulley from shaft. Remove shaft from valve housing; inner and outer washers will fall free.

f. Remove screws that attach cold air valve vane to shaft. Remove plate spring and bushings, if installed. Remove valve vane.

g. Remove pulley, shaft, and washer from valve housing.

h. Remove pin from pulley to complete the disassembly.

13-15. Inspection — Heat Control Valve. *a.* Inspect valve housing for deep dents and cracks; mating flanges and connection area for bends or dents which would prevent tight connections; bearing bosses for galling and excessive wear; interior seat and ball valve for galling.

b. Inspect pulley shafts for galling and evidence of wear.

c. Inspect ball valve and retainer for cracks, galling, and evidence of wear; cold air valve vane for flatness and evidence of excessive edge wear.

13-16. Repair — Heat Control Valve. *a.* Repair small dents in valve housing by bumping out and burnishing. Repair warped flanges by straightening on a surface plate. No weld repairs are permitted.

b. No repairs are permitted on moving parts, ball valve, retainer, coupling, shafts, or wave washer if cracked, bent, or badly worn.

c. Restore solid film lubricant (C66) on ball valve, retainer, coupling, or shafts (para 13-17).

13-17. Restoration — Heat Control Valve Solid Film Lubricant (For Temperature Operation Up to 600°F). *a.* Immerse the parts to be treated in a trichloroethylene (C111) vapor for 1-minute periods. Continue immersions until all traces of grease and oil are removed.

b. Parts to be lubricated with solid film lubricant (C66) should have rust or corrosion removed as follows:

(1) Immerse parts for 15 to 60 minutes in a 2- to 3-pounds-per-gallon solution of surface cleaner (C36) heated to 190°-200°F (88°-92°C).

(2) Rinse in cold water for 3 to 5 minutes.

(3) Rinse with warm deionized or distilled water for 2 to 5 minutes.

c. Treat (passivate) parts for 30 minutes in a room temperature bath of 35-45 percent by weight nitric acid (C71).

d. Rinse parts by immersion in room-temperature water.

e. Dilute lubricant (C66) with solvent (C93 or C97) to obtain a suitable mixture for spraying. A proper mixture will spray on thin and wet.

f. Allow to air-dry for 30 minutes; then oven-cure for 60 to 80 minutes at 300°-320°F (149°-163°C).

13-18. Reassembly — Heat Control Valve. (See fig. 13-3.) *a.* Place valve housing upside down on a clean work bench.

b. Install washer on ball valve shaft; then insert this assembly into valve housing boss.

c. Install coupling on shaft; then position ball valve in housing to engage coupling (ball valve in open position).

d. Install new O-ring in elbow flange groove.

e. Install retainer and wave washer in housing on ball valve.

NOTE

A 1/2-inch wood dowel may be inserted through the top of the assembly to temporarily hold these parts in position.

f. Install elbow on valve housing flange with four bolts and nuts. Use washers under both the bolthead and the nut. Avoid nicking or cutting the O-ring during mating.

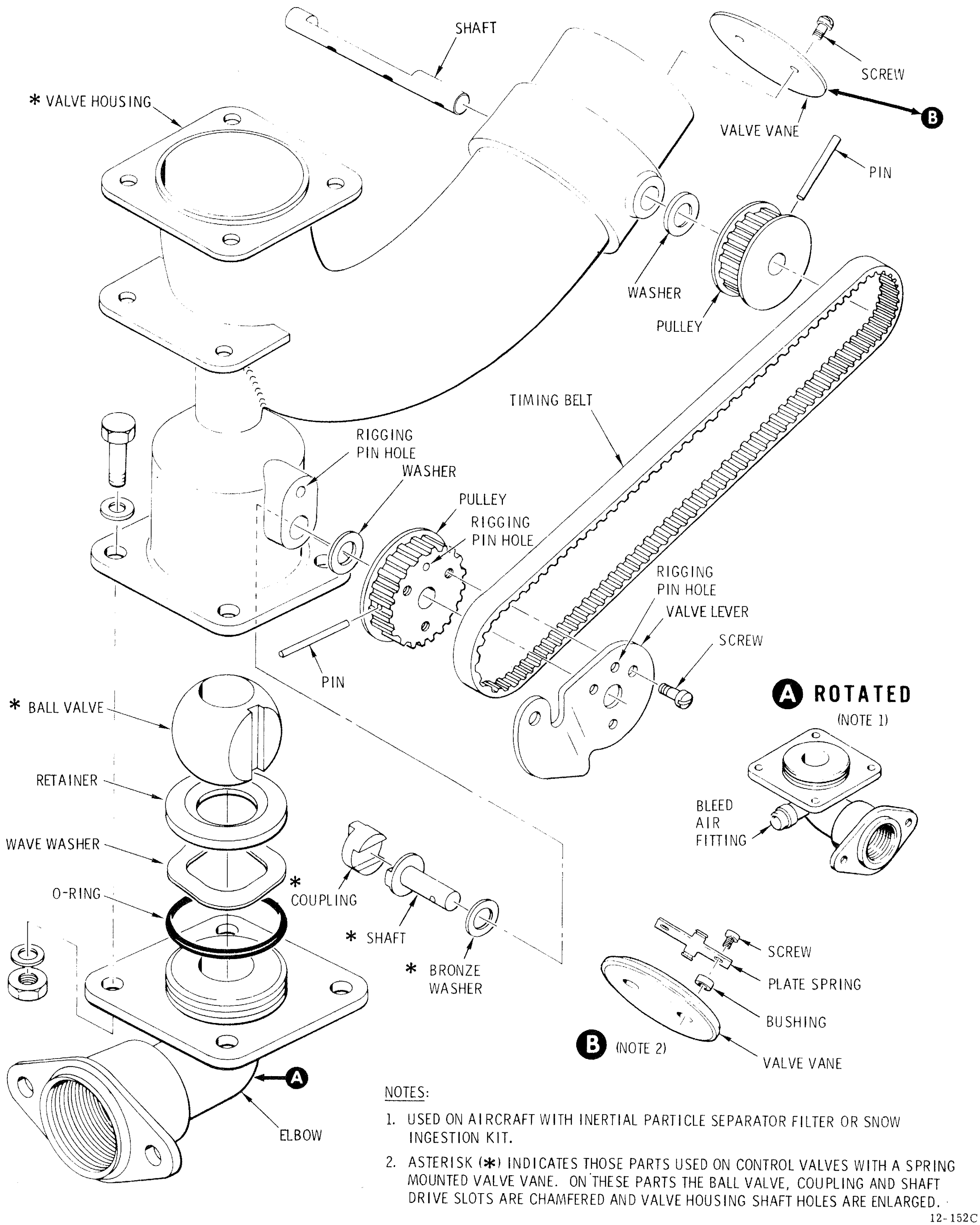
CAUTION

The flanges of the valve housing and elbow should mate without use of undue force. If flanges do not mate easily, the internal parts of valve may be misaligned. Investigate before tightening bolts.

g. Install washer and ball valve pulley on shaft. Rigging pin holes in pulley and in valve housing boss must be aligned (ball valve in open position).

h. Install pin through ball valve pulley and shaft.

i. Remove 1/2-inch dowel if used to align internal parts; then rotate pulley through full travel several times



NOTES:

1. USED ON AIRCRAFT WITH INERTIAL PARTICLE SEPARATOR FILTER OR SNOW INGESTION KIT.
2. ASTERISK (*) INDICATES THOSE PARTS USED ON CONTROL VALVES WITH A SPRING MOUNTED VALVE VANE. ON THESE PARTS THE BALL VALVE, COUPLING AND SHAFT DRIVE SLOTS ARE CHAMFERED AND VALVE HOUSING SHAFT HOLES ARE ENLARGED.

12-152C

Figure 13-3. Heat Control Valve Assembly.

to assure ease of operation. Check for positive closing of ball valve.

j. Install cold air pulley and washer on shaft. Install pin through pulley and shaft.

k. Insert cold air pulley and shaft assembly into valve housing.

l. Attach cold air vane to shaft by using two screws. If the vane is spring mounted, assemble with spring, plate and bushings as shown in detail B, figure 13-3.

m. Rotate ball valve pulley so that rigging pin hole aligns with hole in valve body boss. Insert a 1/8-inch pin to hold the assembly in position (ball valve open).

n. Rotate cold air pulley so that the vane is 45 degrees from fully closed position. If present, yellow marks will be aligned. (The heads of vane attachment screws should not be visible.)

o. Install a serviceable timing belt between pulleys while holding valve position located in *m* and *n* above.

p. If rigging marks have not been applied, paint a yellow index mark (0.06-inch width) on pulley and housing (fig. 13-1).

q. Install valve lever on ball valve pulley with three screws. Remove rigging pin installed in *m* above. Lockwire screws to each other.

13-19. Installation — Heat Control Valve. (See figure 13-2.) *a.* Position control valve assembly on forward left side of canted bulkhead. Install two bolts, washers, and nuts.

b. In the left side of the engine compartment, install new O-ring, nut, and elbow. Connect cabin air-outlet tube to elbow. **TORQUE NUT TO 350 INCH-POUNDS.**

c. Connect control valve flange to warm air duct using new gasket and four bolts, washers, and nuts.

d. Install cool air connector hose on control valve inlet and secure with clamps.

e. On valve assemblies with a bleed air fitting, connect air tube to valve elbow fitting.

f. Position heat control knob (fig. 13-1) fully aft. Position lever on control valve to open and insert a 1/8-inch drill or pin through lever and into valve boss (detail B).

g. Loosen jamnut on control cable. Adjust clevis until clevis attach holes align with hole in valve lever. Install spring pin through clevis and valve lever; then secure with lockwire.

h. Tighten clevis jamnut. Remove pin from valve lever.

i. Move the heat control knob full back and forward several times to check for ease of heat control operation.

j. Reinstall bulkhead access cover and sound insulation blanket.

13-20. HEAT VALVE CONTROL LEVER AND CABLE.

13-21. Description — Heat Valve Control Lever and Cable. The heat valve control lever and cable (fig. 13-1) consists of a control lever, lever piston housing, and a push-pull cable ending in a clevis fitting. Control lever piston travel in the housing is approximately 2.75 inches, from open to closed positions. The last 4-inch section of the cable has a universal joint that provides up to 10 degrees total movement for heat control valve arm throw.

13-22. Operational Check and Adjustment — Heat Valve Control Lever and Cable. (Refer to paragraph 13-12.)

13-23. Removal — Heat Valve Control Lever and Cable. (See fig. 13-1.) *a.* Remove the left half of engine air inlet forward fairing and left aft bulkhead access cover.

b. Remove two screws, washers, and nuts that attach cable lever piston housing to canopy structure.

c. Remove clamps and straps that attach cable to structure along its full length.

d. Cut lockwire and remove spring pin from clevis at heater control valve arm.

e. Remove control cable.

13-24. Disassembly — Heat Valve Control Lever and Cable. (See fig. 13-1.) The control assembly is normally replaced as a unit. However, disassembly to the extent shown on figure 13-1 can be accomplished for inspection purposes or lever mechanism parts replacement.

a. Move control knob to full extent of forward travel.

b. Pull control knob shaft out of piston.

c. Pull piston out of housing.

d. Remove lock pin from piston to separate cable and piston.

13-25. Inspection — Heat Valve Control Lever and Cable. *a.* Inspect cable for kinks, crushed sleeve, and corrosion.

b. Inspect piston for wear or elongation of knob shaft or lockpin holes.

13-26. Repair — Heat Valve Control Lever and Cable. Replace unserviceable parts.

13-27. Reassembly — Heat Valve Control Lever and Cable. (See fig. 13-1.) *a.* Extend cable rod through housing so parts can be assembled.

b. Fit serviceable piston on cable rod and secure with lockpin.

c. Push the piston into housing until large hole in piston is aligned with the large hole in end of housing slot.

d. Insert control rod knob through housing slot and into piston.

e. Use masking tape or lockwire to hold control knob in position until installed.

13-28. Installation — Heat Valve Control Lever and Cable. (See fig. 13-1.) *a.* Route the cable assembly into position. Be sure the two grommets are in place on the cable.

b. Attach the cable lever piston housing to the canopy structure bracket with two screws, washers, and nuts.

c. Install clamps and straps that attach the cable to structure.

NOTE

The clamp nearest to the valve should not be tightened until after travel is checked.

d. Position heat control knob fully aft. Position the lever on heat control valve to open; then insert a 1/8-inch drill or pin through the lever into valve boss (fig. 13-1).

e. Adjust clevis at heat control valve; then tighten the jamnut.

f. When assured that control will operate the valve through full range of travel, tighten the clamp nearest the valve.

g. Reinstall bulkhead access-cover and insulation blanket. Reinstall the left half of engine air inlet forward fairing.

13-29. CARGO COMPARTMENT HEAT CONTROL VALVE.

13-30. Description — Cargo Compartment Heat Control Valve. The cargo compartment heat control valve (fig. 13-2) is located low on the aft bulkhead between the troop seats. The valve allows control of heated air flow into the compartment. Turning the valve outer rim counterclockwise opens a small internal vane for air flow. Clockwise turning closes the valve.

13-31. Removal — Cargo Compartment Heat Control Valve. Loosen clamp (detail E, fig. 13-2); then pull valve out of duct.

13-32. Repair — Cargo Compartment Heat Control Valve. Disassembly or repair of the valve is not practical. Install a serviceable valve if defective.

13-33. Installation Cargo Compartment Heat Control Valve. Reinstall loosened clamp on duct. Insert the replacement valve and tighten the clamp.

SECTION II AIR COOLING SYSTEMS

13-34. VENTILATING SYSTEM.

13-35. Description — Ventilating System. The main fresh air source for the fuselage interior is a manually operated vent control valve (fig. 13-4). The valve is recessed in the canopy panel forward of the instrument panel. The vent control valve has a round 4-inch inlet that supports the valve shaft and contour matching valve door. The valve door is actuated by the control cable knot marked PULL FOR AIR. The knob is a friction catch type that locks the cable in any selected position throughout the full stroke when the knob plunger is released. The vent control valve is basically polycarbonate plastic. The valve is fan-shaped to deflect inlet air for circulation. Each pilot's and cargo compartment door window contains an adjustable plastic ventilator (snapvent). Each snapvent can be opened, closed, and positioned by rotation of the vent cutout. The snapvents add to fresh air intake or provide for air flow-through, as desired.

13-36. Inspection — Ventilating System. *a.* Inspect vent control cable for security of knob to instrument panel, and cable sleeve to support bracket on vent valve.

b. Inspect control cable wire for secure attachment to valve arm swivel. Check swivel for free rotation by actuating the control knob through a full stroke.

c. Inspect valve for complete closure when control knob is pushed full in. Check valve door for looseness on shaft and distortion. The valve must be flat and a firm, snug fit in tube opening.

d. Inspect plastic walls, tube and rib separator between the walls of valve for cracks. Check for secure rivet attachment to the canopy panel, and parted joints.

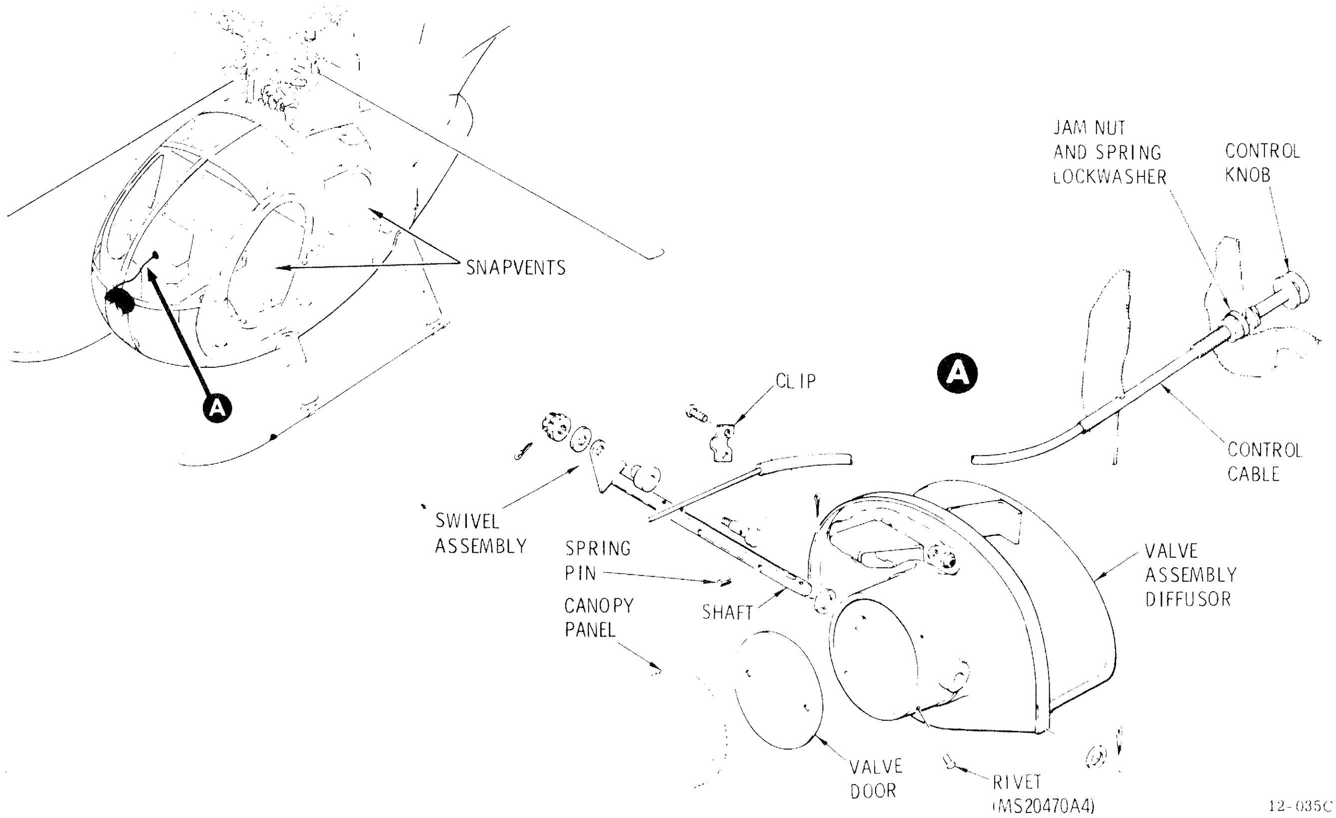


Figure 13-4. Ventilating System.

13-37. VENTILATING VALVE CONTROL.

13-38. Description — Ventilating Valve Control. The ventilating valve cable is a flexible, coaxial-wire type that incorporates a pushbutton locking device in the control housing. This feature allows locking of the ventilating valve in the open, closed, or any intermediate position.

13-39. Removal — Ventilating Valve Control. *a.* Set the power selector switch at OFF.

b. Disconnect cable wire from valve arm swivel.

c. Remove screws, nuts, washers, and clip that secure cable to support bracket.

d. Loosen and remove hex nut and lockwasher that secure cable knob to instrument panel and withdraw cable from panel.

13-40. Repair — Ventilating Valve Control. *a.* The control assembly is not repairable. Failure of the internal locking device is cause for replacement.

b. If the plunger of the ventilating valve control is pulled out too far and the detent ball drops out of the plunger, reassemble control according to instructions for the fuel shutoff valve control in chapter 10.

13-41. Installation — Ventilating Valve Control. *a.* Route new cable through instrument panel and secure knob in place with lockwasher and nut.

b. Fasten cable sleeve to support bracket on valve and secure with clip, screws, washers, and nuts.

c. Actuate control knob to align cable wire with hole in valve arm swivel; push knob to cable panel fitting stop.

d. Position valve to full closed and tighten swivel on cable wire. Install cotter pin.

e. Check swivel for free rotation by actuating control knob through full stroke. Inspect valve assembly for complete closure of inlet opening when control knob is pushed forward against cable fitting stop. Check valve door for looseness on shaft and deformation; valve door must be flat, and a firm, snug fit in the inlet.

13-42. CABIN VENTILATING VALVE.

13-43. Description — Cabin Ventilating Valve. The cabin ventilating valve is made of polycarbonate plastic and incorporates a valve vane assembly to control ventilating air. The valve assembly is attached by four rivets to a duct molded into the fiberglass canopy panel.

13-44. Removal — Cabin Ventilating Valve. (See fig. 13-4.) *a.* Remove cable clip.

b. Pull cable control to open valve door.

c. Loosen swivel assembly and disconnect control cable.

d. Drill out the four rivets that attach valve to canopy duct.

e. Remove valve assembly.

13-45. Disassembly — Cabin Ventilating Valve. *a.* Remove two spring pins from shaft and door. Slide door from shaft.

b. Remove cotter pin and slide shaft from valve body.

13-46. Inspection — Cabin Ventilating Valve. Inspect for bent shaft or valve door and excessive wear in the pivot areas.

13-47. Repair — Cabin Ventilating Valve. Repair polycarbonate valve body as follows:

a. Repair small areas of rib or seam separation by injecting dichloromethane (C38) into the void area and clamping together under light pressure.

b. Other than straightening of minor bends, no repairs should be made to the shaft or valve door.

c. Repair cracks (para 2-15h).

13-48. Reassembly — Cabin Ventilating Valve. (See fig. 13-4). *a.* Install cotter pin in hole at control arm end of shaft. Install washer on shaft to bear on cotter pin.

b. Insert shaft in body of valve.

c. Insert valve door into slot in shaft.

d. Install two spring pins through shaft and door.

e. Install washer on end of shaft and insert cotter pin.

f. Operate valve several times to assure freedom of operation and positive closing.

13-49. Installation — Cabin Ventilating Valve. (See fig. 13-4.) *a.* Position valve on canopy panel duct and align rivet holes.

b. Install four MS20470A4 rivets with 90° spacing to attach valve to duct. If rivet holes are elongated or oversize install next larger size rivets (5/32-inch diameter).

c. Fasten cable to valve bracket with clip.

d. Install swivel assembly loosely in arm of shaft.

e. Insert control wire into swivel assembly.

f. Position shaft arm so that valve door is fully closed.

g. Push control knob all the way in; then tighten swivel assembly on wire. Install cotter pin.

h. Operate valve through several open-close cycles to assure ease of operation and to determine that valve locks in any position when control knob is released.

13-50. DOOR WINDOW (SNAPVENTS).

13-51. Description — Door Window (Snapvents). The plastic door window snapvents should be replaced whenever damaged or distorted. Repair is not practical.

NOTE

If an anti-removal tube is installed to prevent removal of the snapvent, refer to chapter 2 Cargo Door Installation.

13-52. Removal — Door Window (Snapvents). Push snapvent outward in window (open position). Grasp the snapvent open side and gently squeeze. When vent is out-of-round, slip out of window at a 30-degree angle.

13-53. Installation — Door Window (Snapvents). Gently squeeze the open end of the snapvent. When vent is out-of-round, slip into window hole at a 30-degree angle. Be sure that ends of wire stiffener are inside window hole.

CHAPTER 14
HOISTS AND WINCHES

(Not Applicable)

CHAPTER 15
AUXILIARY POWER PLANTS

(Not Applicable)

CHAPTER 16
MISSION EQUIPMENT

(Refer to appendix A for appropriate maintenance manuals discussing armament Systems for Army OH-6A aircraft)

CHAPTER 17

EMERGENCY EQUIPMENT

17-1. EMERGENCY EQUIPMENT.

17-2. Description — Emergency Equipment. The emergency equipment contained in the aircraft (fig. 17-1) consists of a fire extinguisher, a first-aid kit, and emergency release jettison mechanism for the pilot's and cargo doors.

17-3. FIRE EXTINGUISHER.

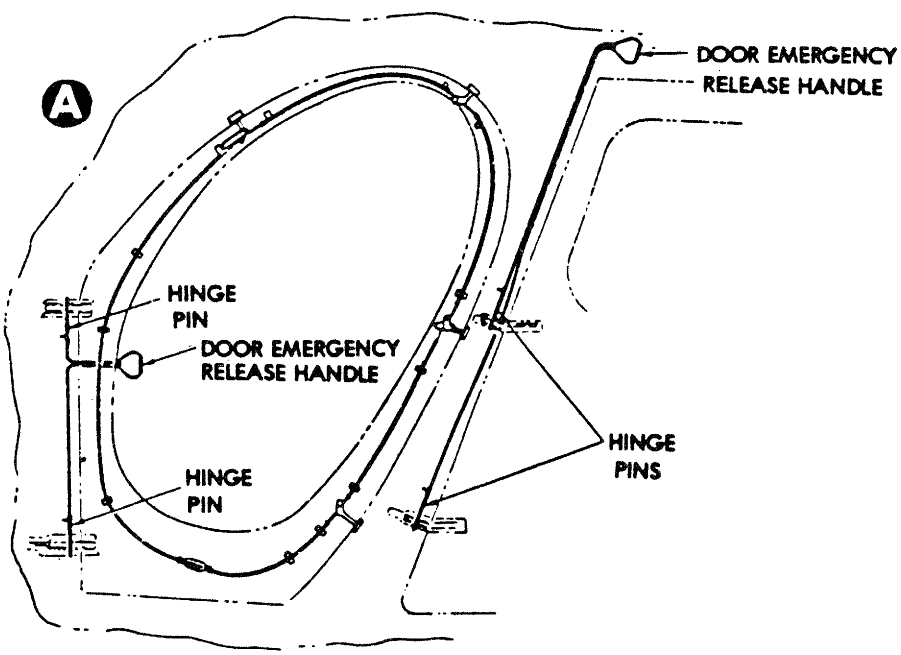
17-4. Description — Fire Extinguisher. A pressurized monobromofluoromethane (CF3Br) fire extinguisher (fig. 17-1) is vertically mounted in a quick-release support clamp at the lower right side of the pilot's seat support structure. The extinguisher is supported by a polycarbonate cup riveted to the floor. When removed, the fire extinguishing compound is released by a hand-operated lever on top of the extinguisher. Serviceability of the extinguisher is indicated by the presence of a light-gage safety wire across the actuating lever. Operating instructions are printed on the extinguisher.

17-5. FIRST-AID KIT.

17-6. Description — First-Aid Kit. A first-aid kit (fig. 17-1) is located on the control bulkhead between the pilot's and copilot's seats. The kit is attached to quick-release fasteners located on mounting brackets riveted to the bulkhead.

17-7. DOOR JETTISON MECHANISM.

17-8. Description — Door Jettison Mechanism. A door jettison handle for each door (fig. 17-1) is located inside the aircraft between the door hinges. When the door is unlatched, pulling the handle removes hinge pins that are attached by cables and releases the door at the two hinges.



NOTE: DOOR LATCHING MECHANISM IS TYPICAL

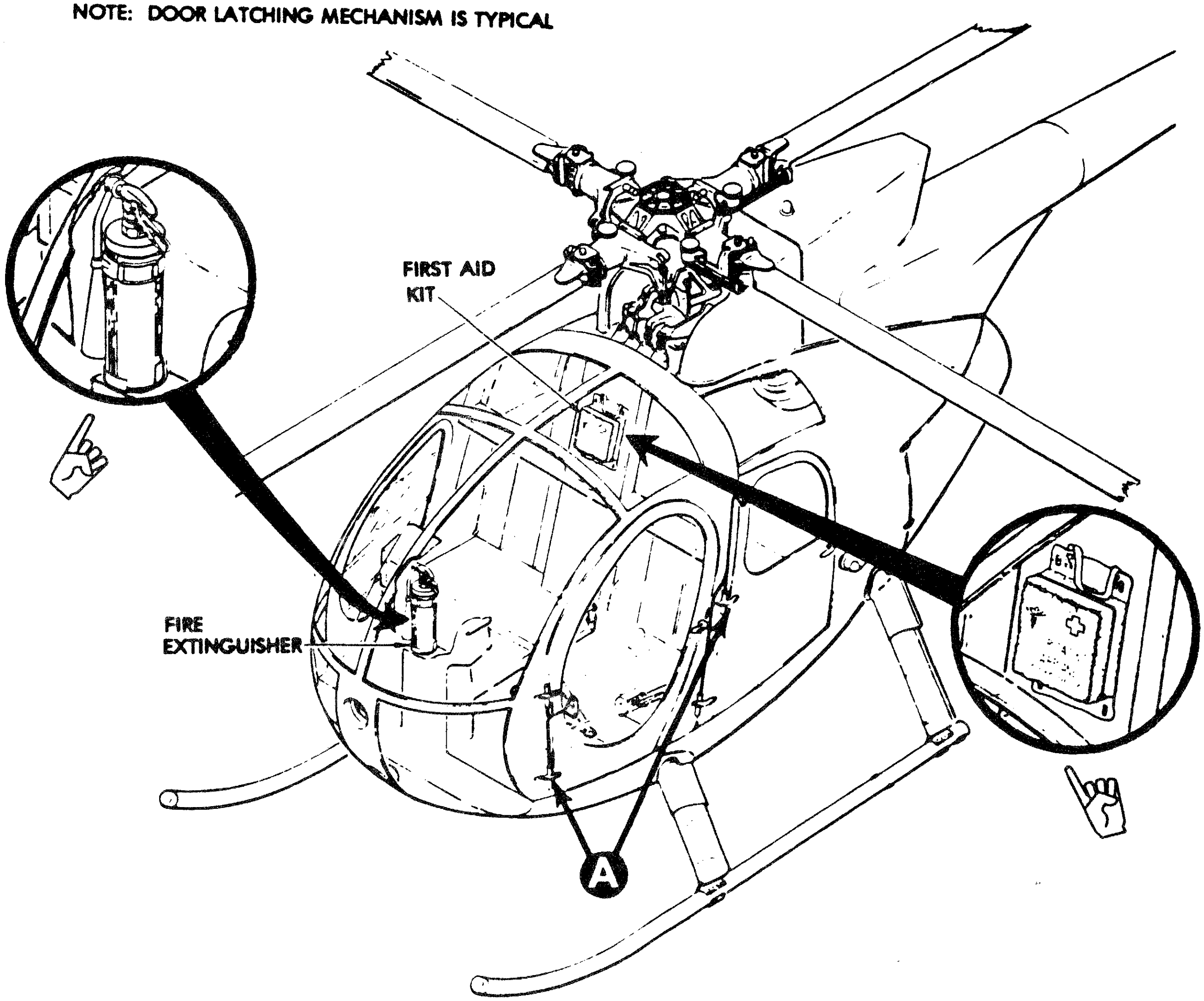


Figure 17-1. Emergency Eqipment Locations

APPENDIX A

REFERENCES

A-1. REFERENCES.

A-2. General — References. This appendix contains a list of official publications applicable and available to Unit and Intermediate Maintenance activities. The references list includes official publications directly related to unit and intermediate maintenance and care of the Army Model OH-6A Helicopter.

AR 40-14	Control and Recording Procedures, Occupational Exposure to Ionizing Radiation
AR 735-11	Accounting for Lost, Damaged, and Destroyed Property
AR 750-1	Army Materiel Maintenance Concepts and Policies
AR 750-55	Inspection and Preparation of Army Aircraft for Transfer to Foreign Governments as Grant Air or Foreign Military Sales
ASA B40.1-1962	Surface Texture
FM 55-42	Army Aviation Intermediate Maintenance
FM 55-63	Fundamentals of Airframe Maintenance
FM 55-411	Army Aircraft Control and Technical Inspection
MIL-A-8625	Anodic Coatings for Aluminum and Aluminum Alloys
MIL-H-6088	Heat Treatment of Aluminum Alloys
MIL-I-6866	Penetrant Method of Inspection
MIL-M-3171	Processes for Treatment and Prevention of Corrosion on Magnesium Alloy
QQ-P-416	Cadmium Plating (Electrodeposited)
TB 5-4200-200-10	Hand Portable Fire Extinguisher Approved for Army Users
TB 43-0001-2-1, -2, -3, -4	Equipment Improvement Report and Maintenance Digest (Rotary Wing Aircraft)
TB 43-0106	Aeronautical Equipment Army Oil Analysis Program (AOAP)
TB 43-0108	Handling, Storage, and Disposal of Army Aircraft Components Containing Radioactive Materials
TB 43-180	Calibration
TB 55-1500-307-24	Aircraft Components Requiring Maintenance Management and Historical Data

TB 55-1500-334-25	Conversion of Aircraft to Fire Resistant Hydraulic Fluid
TB 55-1560-276-24/1	Polish Kit, Glass, P/N RS-69, NSN 1560-00-450-3622
TB 55-8100-200-24	Maintenance of Specialized Reusable Containers for Aircraft Equipment
TB 55-9150-200-24	Engine and Transmission Oils, Fuels, and Additives for Army Aircraft
TB 746-93-2	Painting and Marking of Army Aircraft
TB 750-25	Maintenance of Supplies and Equipment: Army Metrology and Calibration System
TB 750-125	Assembly and Inspection of Medium-Pressure Hose and Hose Assemblies Tetrafluoroethylene (Teflon)
TM 3-261	Handling and Disposal of Unwanted Radioactive Materials
TM 5-200	Camouflage Materials
TM 9-1005-298-12	Equipment Serviceability Criteria for Armament Subsystem, Helicopter, 7.62-MM Machine Gun, High Rate, M27 Operator and Organizational Maintenance Manual, Armament Subsystem, Helicopter, 7.62 Millimemter Machine Gun: High Rate, M27
TM 9-1005-298-20P	Organizatinal Maintenance Repair Parts and Special Tools List, Armament Subsystem, Helicopter, 7.62 Millimeter Machine Gun: High Rate, M27 (1005-933-6242) (Used on OH-6A Helicopter)
TM 9-1005-298-34	Direct and General Support and Depot Maintenance Manual, Armament Subsystem, Helicopter, 7.62 Millimeter Machine Gun: High Rate, M27 (Used on OH-6A Helicopter)
TM 9-1005-298-35P	Direct Support, General Support, and Depot Maintenance Repair Parts and Special Tools List, Armament Subsystem, Helicopter, 7.62 Millimeter Machine Gun: High Rate, M27 (1005-933-6242) (Used on OH-6A Helicopter)
TM 11-1520-214-20	Organizational Maintenance Manual Electronic Equipment Configuration, Army Model OH-6A Helicopter
TM 11-1520-214-20-1	Organizational Maintenance Manual, Electronics Equipment Configuration, Army Model OH-6A Helicopter (with Standard Lightweight Avionics Equipment)
TM 11-1520-214-20P	Organizational, DS, GS, and Depot Maintenance Repair Parts and Special Tools List: Electronics Equipment Configuration for Army Helicopter OH-6A
TM 11-1520-214-20P-1	Organizational Maintenance Repair Parts and Special Tools Lists, Electronics Equipment Configuration, Army OH-6A Helicopter (with Standard Lightweight Avionics Equipment)
TM 11-1520-214-34 TM 11-1520-214-34P-1 TM 11-1520-214-34P	DS, GS, and Depot Maintenance Manual, Electronics Equipment and Configuration, Army Model OH-6A Helicopter DS, GS, and Depot Maintenance Repair Parts and Special Tools Lists, Electronic Configuration, Army Model OH-6A Helicopter (with Standard Lightweight Avionics Equipment)

TM 11-1520-214-34-1	DS, GS, and Depot Maintenance Manual, Electronics Equipment Configuration, Army Model OH-6A Helicopter (with Standard Lightweight Avionics Equipment)
TM 11-2019	Test Sets I-49, I-49A, and I-49B and Resistance Bridges ZM-4A/U and ZM-4B/U
TM 11-6140-203-14-1	Operator and Organizational Manual, Aircraft and Non-Aircraft Nickel-Cadmium Storage Batteries
TM 11-6140-203-14-2	Operators Organizational, DS, GS, and Depot Maintenance Manual (including Repair Parts and Special Tools Lists) Aircraft Nickel- Cadmium Batteries
TM 11-6625-277-14	Operators Organizational, Direct Support, and General Support Maintenance Manual; Meter Test Set TS-682A/GSM-1
TM 38-230-1	Preservation, Packaging, and Packing of Military Supplies and Equipment, Preservation and Packaging (Volume I)
TM 38-230-2	Preservation, Packaging, and Packing of Military Supplies and Equipment, Packing (Volume II)
TM 38-750	The Army Maintenance Management System (TAMMS)
TM 38-750-1	The Army Maintenance Management System (TAMMS) Field Command Procedures
TM 43-0103	Nondestructive Inspection Methods
TM 55-405-9	Army Aviation Maintenance Engineering Manual: Weight and Balance
TM 55-406	Fundamentals of Aircraft Power Plant Maintenance
TM 55-407	Fundamentals of Electricity, Army Aircraft
TM 55-410	Aircraft Maintenance, Servicing, and Ground Handling Under Extreme Environmental Conditions
TM 55-412	Fundamentals of Aircraft Instruments
TM 55-1500-204-25/1	General Aircraft Maintenance Manual (for nondestructive inspection methods refer to TM 43-0103)
TM 55-1500-322-24	Organizational, DS, GS, and Depot Maintenance Manual: Maintenance of Aeronautical Antifriction Bearings
TM 55-1500-323-25	Organizational, DS, GS, and Depot Maintenance Manual: Installation Practices for Aircraft Electric and Electronic Wiring
TM 55-1500-326-24	Standards of Serviceability for Transfer of Aircraft
TM 55-1500-328-25	Aeronautical Equipment Maintenance Management Policies and Procedures
TM 55-1500-333-24	Organizational, Direct and General Support Maintenance Manual for Cleaning procedures for Army Aircraft

TM 55-1520-214-S	Preparation for Shipment of OH-6A Helicopter
TM 55-1520-214-PMS	OH-6A Helicopter, Preventive Maintenance Services
TM 55-1520-214-10	Operator Manual, Helicopter, Observation, OH-6A
TM 55-1520-214-CL	Checklist
TM 55-1520-214-23P	Aviation Unit and Intermediate Maintenance Repair Partss and Special Tools Lists (Including Depot Maintenance Repair Parts and Special Tools List)
TM 55-1520-214-MTF	Maintenance Test Flight
TM 55-2840-231-23	Organizational, Direct Support, and General Support Maintenance Manual for Engine Assemblies Model T63-A-5A, Model T63-A-700
TM 55-2840-231-23-P	DS, and GS, Maintenance and Repair Parts and Special Tools List — Engine, Aircraft, Gas Turbine (Allison) Army Aviation Maintenance Engineering Manual: Ground Handling and Service Equipment
TM 55-4920-231-14	Operator, Organizational, Field and Depot Manual and Repair Parts List: Pitot and Static Systems Tester (NSN 4920-00-475-7161)
TM 55-4920-244-14	Organizational, DS, GS and Depot Maintenance Manual: Tester, Exahust Gas Temperature, Model BH112JA36, NSN 4920-00-673-5514
TM 55-4920-402-13&P	Operating Instructions, Aviation Unit and Intermediate Maintenance Manual, Vibrex Balancing Kit
TM 55-6600-200-20	Organizational Maintenance Manual: Marking of Instruments and Interpretation of Markings
TM 55-6670-200-14&P	Operator, Organizational, DS, GS, and Depot Maintenance Manual: Aircraft Electronic Weighing Kit, Part No. C-7500
TM 750-244-1-5	Procedures for the Destruction of Aircraft and Associated Equipment to Prevent Enemy Use
TM 740-90-1	Administrative Storage of Equipment
TM 750-134	Procedures for Rapid Deployment, Redeployment, and Retrograde: United States Army Rotary Wing Aircraft
TM 750-199	Procedures for Rapid Deployment, Redeployment, and Retrograde: United States Army Aircraft Components, Spare Parts, and Support Equipment (Class II(A) and Class IV(A) Supplies)
TM 750-244-1-2	Procedures for the Destruction of Life Support Equipment to Prevent Enemy Use
TM 750-244-1-5	Procedures for the Destruction of Aircraft and Associated Equipment to Prevent Enemy Use

APPENDIX B

MAINTENANCE ALLOCATION CHART

MAINTENANCE ALLOCATION CHART THREE LEVELS OF MAINTENANCE CONCEPT

B-1. MAINTENANCE ALLOCATION CHART.

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B-2. Description — Maintenance Allocation Chart.

a. This Maintenance Allocation Chart (MAC) assigns maintenance functions in accordance with the Three Levels of Maintenance concept for army aircraft. These maintenance levels; Aviation Unit Maintenance (AVUM), Aviation Intermediate Maintenance (AVIM) and Depot Maintenance are depicted on the MAC as:

O — AVUM

F — AVIM

D — DEPOT

b. The maintenance to be performed below depot and in the field is described as follows:

(1) Aviation Unit Maintenance (AVUM) — AVUM activities will be staffed and equipped to perform high frequency 'On-Aircraft' maintenance tasks required to retain or return aircraft to a serviceable condition. The maintenance capability of the AVUM will be governed by the Maintenance Allocation Chart (MAC) and limited by the amount and complexity of ground support equipment (GSE), facilities required, and number of spaces and critical skills available. The range and quantity of authorized spare modules/components will be consistent with the mobility requirements dictated air mobility concept. (Assignment of maintenance tasks to divisional company size aviation units will consider the overall maintenance capability of the division, the requirement to conserve personnel and equipment resources and air mobility requirements.)

(a) Company Size Aviation Units: Perform those tasks which consist primarily of preventive maintenance and maintenance repair and replacement functions associated with sustaining a high level of aircraft operational readiness. Perform maintenance inspections and servicing to include preflight, daily, intermediate, periodic and special inspections as authorized by the MAC or higher headquarters. Identify the cause of equipment/system malfunctions using applicable technical manual troubleshooting instructions, built-in-test equipment (BITE), installed aircraft instruments, or easy to use/interpret diagnostic/fault isolation devices (TMDE). Replace worn or damaged modules/components which do not require complex adjustments or system alignment and which can be removed/installed with available skills, tools and equipment. Perform operational and continuity checks and make minor repairs to the electrical system. Inspect, service and make operational, capacity and pressure checks to hydraulic systems. Perform servicing, functional adjustments, and minor repair/replacement to the flight control, propulsion, power train and fuel systems. Accomplish air frame repair which does not require extensive disassembly, jigging, or alignment. The manufacture of air frame parts will be limited to those items which can be fabricated with tools and equipment found in current air mobile tool and shop sets. Evacuate unserviceable modules/components and end items beyond the repair capability of AVUM to the supporting AVIM.

(b) Less than Company Size Aviation Units, Aviation elements organic to brigade, group, battalion headquarters and detachment size units are normally small and have less than ten aircraft assigned. Maintenance tasks performed by these units will be those which can be accomplished by the aircraft crew chief or assigned aircraft repairman and will normally be limited to preventive maintenance, inspections, servicing, spot painting, stop drilling, application of nonstress patches, minor adjustments, module/component fault diagnosis and replacement of selected modules/components. Repair functions will normally be accomplished by the supporting AVIM unit.

(2) Aviation Intermediate Maintenance (AVIM) — AVIM provides mobile, responsive "One Stop" maintenance support. (Maintenance functions which are not conducive to sustaining air mobility will be assigned to depot maintenance.) Performs all maintenance functions authorized to be done at AVUM. Repair of equipment for return to user will emphasize support or operational readiness requirements. Authorized maintenance includes replacement and repair of modules/components and end items which can be accomplished efficiently with available skills, tools, and equipment. Establishes the Direct Exchange (DX) program for AVUM units by repairing selected items for return to stock when such repairs cannot be accomplished at the AVUM level. Inspects, troubleshoots, tests, diagnoses, repairs, adjusts, calibrates, and aligns aircraft modules/components. AVIM units will have the capability to determine the serviceability of specified modules/components removed prior to the expiration of the Time Between Overhaul (TBO) or finite life. Module/component disassembly and repair will support the DX program and will normally be limited to tasks requiring cleaning and the replacement of seats, fittings and items of common hardware. Air frame repair and fabrication of parts will be limited to those maintenance tasks which can be performed with available tools and test equipment. Unserviceable repairable modules/components and end items which are beyond the capability of AVIM to repair will be evacuated to Depot Maintenance. This level will perform aircraft weight and balance inspections and other special inspections which exceed AVUM capability. Provides quick response maintenance support, including aircraft recovery and air evacuation, on-the-job training, and technical assistance through the use of mobile maintenance contact teams. Maintains authorized operational readiness float aircraft. Provides collection and classification services for serviceable/unserviceable material. Operates a cannibalization activity in accordance with AR 750-50. (The aircraft maintenance company within the maintenance battalion of a division will perform AVIM functions consistent with air mobility requirements and conservation of personnel and equipment resources. Additional intermediate maintenance support will be provided by the supporting non-divisional AVIM unit.)

B-3. Use of the Maintenance Allocation Chart. *a.* The Maintenance Allocation Chart assigns maintenance functions to the lowest level of maintenance based on past experience and the following consideration:

(1) Skills available.

(2) Time required.

(3) Tools and test equipment required and/or available.

b. Only the lowest level of maintenance authorized to perform a maintenance function is indicated.

c. A maintenance function assigned to a maintenance level will automatically be authorized to be performed at any higher maintenance level.

d. A maintenance function that cannot be performed at the assigned level of maintenance for any reason may be evacuated to the next higher maintenance organization. Higher maintenance levels will perform the maintenance functions of lower maintenance levels when required or directed by the appropriate commander.

e. The assignment of a maintenance function will not be construed as authority to carry the associated repair parts in stock. Authority to requisition, stock, or otherwise secure necessary repair parts will be as specified in the repair parts appendix.

f. Normally there will be no deviation from the assigned level of maintenance. In cases of operational necessity, maintenance functions assigned to a maintenance level may, on a one-time basis and at the request of the lower maintenance level, be specifically authorized by the maintenance officer of the level of maintenance to which the function is assigned. The special tools, equipment, etc, required by the lower level of maintenance to perform this function will be furnished by the maintenance level to which the function is assigned. This transfer of a maintenance function to a lower maintenance level does not relieve the higher maintenance level of the responsibility of the function. The higher level of maintenance will provide technical supervision and inspection of the function being performed at the lower level.

g. Organizational through depot maintenance of the U. S. Army Electronics Command equipment will be performed by designated U. S. Army Electronics Command personnel.

h. Changes to the Maintenance Allocation Chart will be based on continuing evaluation and analysis by responsible technical personnel and on reports received from field activities.

B-4. Definitions. *a. Inspect.* To determine serviceability of an item by comparing its physical, mechanical and electrical characteristics with established standards.

b. Test. To verify serviceability and to detect electrical or mechanical failure by the use of test equipment.

c. Service. To clean, to preserve, to charge, and to add fuel, lubricants, cooling agents and air.

d. Adjust. To rectify to the extent necessary to bring into proper operating range.

e. Align. To adjust specified variable elements of an item to bring to optimum performance.

f. Calibrate. To determine the corrections to be made in the readings of instruments or test equipment used in precise measurement. Consists of the comparison of two instruments, one of which is a certified standard of known accuracy, to detect and adjust any discrepancy in the accuracy of the instrument or test equipment being compared with the certified standard.

g. Install. To set up for use in an operational environment such as an emplacement, site or vehicle.

h. Replace. To replace unserviceable items with serviceable assemblies, subassemblies or parts.

i. Repair. To restore an item to serviceable condition through correction of a specific failure or unserviceable condition. This includes, but is not limited to, inspection, cleaning, preserving, adjusting, replacing, welding, riveting, and strengthening.

j. Overhaul. To restore an item to a completely serviceable condition as prescribed by maintenance serviceability standards prepared and published for the specific item to be overhauled.

k. Rebuild. To restore an item to a standard as nearly as possible to the original or new condition in appearance, performance, and life expectancy. This is accomplished through the maintenance technique of complete disassembly of the item, inspection of all parts or components, repair or replacement of worn or unserviceable elements (items) using original manufacturing tolerances and specifications, and subsequent reassembly of the item.

B-5. Functional Groups. The standard functional groupings shown below are used throughout this MAC. Maintenance manuals and RPSTLs will be revised eventually utilizing these standard groupings as individual chapters with sections in each chapter relative to the individual complete systems, subsystems, modules, components, assemblies, or specific parts noted.

B-6. Symbols. The letters "O-(AVUM), F-(AVIM), and D-(Depot)", when placed on the Maintenance Allocation Chart indicate the lowest level of maintenance responsible for performing the particular maintenance function. Maintenance level higher than the level of maintenance marked by the symbol are authorized to perform the indicated function.

GROUP NUMBER	DESCRIPTION
00	Aircraft System
01	Aircraft General
	Servicing, handling, inspection requirements, lubrication charts, overhaul and retirement schedules
02	Airframe
	Fuselage, empennage, pylons, wings, mounts (engine and transmission), armor, seat·, ramps, decks, and tiedowns
03	Alighting Gear
	Landing gear, skids, floats, skis, struts, wheel brakes, and mechanical mechanisms
04	Power Plant Installation Removal, installation, cooling systems, air induction, exhaust, oil systems, components, ignition system, power control, harnesses, carburetors, fuel controls, pumps (engine driven), filters, particle separators, and Quick Change Assemblies (QCA)
05	Propellers/Rotor Systems
	Propellers, governors, rotors (main and tail), hubs, blades, dampeners, stabilizer bars, and swashplates
06	Drive Train Systems
	Transmissions, gearboxes, clutches, shafting, oil system, bearings, hangers, oil tanks, and free-wheeling units
07	Hydraulic and Pneumatic Systems
	Pumps, filters, reservoirs, cylinders, valves, servos, motors, starters, and accumulators
08	Instrument Systems
09	Flight instruments, navigation instruments, engine instruments, miscellaneous instruments (i.e., clocks), sending units, panels, and flow meters Electrical Systems
	Motors actuators, regulators, generators, starters, batteries, lighting, caution and warning lights, inverters, fault isolation systems, rotor brakes, and avionics provisions
10	Fuel Systems
	Tanks, cells, filters, pumps, valves, auxiliary fuel systems, and refueling systems
11	Flight Controls System
	Control sticks, pedals, cables, pulleys, push-pull rods, torque tubes, quadrants, force gradients, control surfaces, bellcranks, and trim actuators (mechanical)
12	Utility Systems
	Fire detecting/extinguishing systems, oxygen systems, windshield wiper systems, mirrors, and de-ice/anti-ice systems
13	Environmental Control Systems (ECS) Heaters, air conditioners, defrosters, control mixing valves, and ducts
14	Hoists and Winches
	Cargo/rescue hoists, winches, hooks, slings, loading systems, and emergency release systems
15	Auxiliary Power Plants (APP)
	Fuel, exhaust, and ducting
16	Mission Equipment
	Spraying equipment, stores racks, armament, reconnaissance, photography, pods, and litters
17	Emergency Equipment
	Ejection seats, canopies, jettison systems, portable fire extinguishers, axes, and first aid kits
18	Installed Avionics Components Communications and navigational black boxes

MAINTENANCE ALLOCATION CHART
FOR
LIGHT OBSERVATION HELICOPTER OH-6A "CAYUSE"
(AVSCOM Reg 310-10)

(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
01	AIRCRAFT GENERAL													
	Cleaning													O
	Moor													O
	Tow													O
	Jack													O
	Preservation & Depreservation													O
	Weight & Balance													F
	Complete Painting													F
	Spot Painting													O
	Hoist													O
	Sub-Assembly Painting													F
02	AIRFRAME													
	Sheet metal, structural members (not requiring jigs & fixtures)	O							O*	O*			*Limited to airframe repairman tool kit portable hand tools	
	Sheet metal & structural members requiring jigs & fixtures	O							D*	D*			*Normally a Depot function. However, where facilities, skills, equipment, and test capabilities are available function can be performed at AVIM.	
	Fuselage assembly windshield	O							O*	O*			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.	

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MAINTENANCE ALLOCATION CHART
FOR
LIGHT OBSERVATION HELICOPTER OH-6A "CAYUSE"
(AVSCOM Reg 310-10)

(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Fairings, access doors & panels	O							O	O				
	Windows, plexiglas panels	O							O	O				
	Doors, entrance	O			O				O	O				
	Doors, engine access	O			O				O	O				
	Cockpit & cabin floors	O							O	O				
	Fuselage furnishings crew & passenger seats & cushions	O							O	O*			*Patching by hand methods only.	
	Armor, crew seats	O							O	O				
	Seat belts	O							O					
	Inertia reels & shoulder harness	O							O					
	Upholstery & sound proofing	O							O	O*			*Patching by hand methods only	
	Mast support	O							F	F				
	Engine mounts	O							O	F				
	Tailboom (not requiring jigs & fixtures)	O							O	O				
	Tailboom (requiring jigs & fixtures)	O							O	F				
	Vertical & horizontal stabilizers	O							O	O				

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(AVSCOM Reg 310-10)

(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Horizontal stabilizer (requiring jigs & fixtures)								O	F				
03	ALIGHTING GEAR													
	Skids, struts, braces & fairings	O							O	O*				*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Skid tube abrasion strips, plugs & caps	O							O					
	Shock struts	O							O					
04	POWER PLANT INSTALLATION													
	Engine, complete assembly	O		O	*				O	*	D			*Ref to TM 55-2840-231-24 for other maintenance functions
	Tailpipe assembly	O							O	F				
	Torquemeter system	O							O					
	Armor, Fuel Systems	O							O	O*				*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.

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(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Armor, Compressor	O							O	O*				*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Oil Cooler	O	F						O	F				
	Oil Cooler Blower	O							O					
	Lubrication System:													
	Oil Tank	O		O					O	O*				*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Check Valve	O							O	F				
	Valve relief, temp & press by-pass, drain	O	O*						O	O*				*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Engine Controls:													
	Control linkage	O			O				O	O				
	Droop compensator	O			O				O	O				
	Actuator, power turbine	O							O	F				
	Filter (Engine Air)	O							O	O				
	Filter by-pass switch	O							O					

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(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
05	PROPELLORS/ROTOR SYSTEMS													
	Main rotor blades	O			O*				O	O*	*		*Tracking	**When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Main rotor hub assembly	O							O	O*	D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Swashplate assembly	O							O	F				
	Stationary and/or rotating links	O			O				O	O				
	Main rotor dampers	O	O	O	O				O	F	D			
	Tail rotor assembly	O			O*				O	O**	D		*Balance	**When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Control, tail rotor	O							O					

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(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Tail rotor hub	O							D	D				
	Tail rotor blades	O							D	F				
06	DRIVE TRAIN SYSTEMS													
	Tail rotor transmission	O		O					O	O*	D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Main transmission	O		O					O	O*	D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Mast assembly	O							F					
	Main rotor drive shaft	O							O					
	Tail rotor drive shaft	O							O	O				
	Tail rotor drive shaft damper	O			O				O					
	Shaft assy main transmission drive	O							O	F				
	Transmission Lubrication System Main Transmission Oil Pump (External)	O							O		D			
	Filters	O							O					

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(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Magnetic plugs & chip detectors	O							O					
	Sight gage	O							O					
	Clutch assy	O		O					O	O*	D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
07	HYDRAULICS AND PNEUMATIC SYSTEMS													
	(NOT APPLICABLE)													
08	INSTRUMENT SYSTEMS													
	Airspeed Indicator	O	O						O					
	Altimeter	O	O*				F		O		D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Compass Stand-by	O			O		O		O					
	Attitude Indicator	O							O		D			
	Heading, Bearing, Homing Indicator	O	O		O				O	*				*Ref TM11 Series ECOM
	Pitot Static System	O	O							O				
	Pitot Tube	O	O		O				O					

MAINTENANCE ALLOCATION CHART FOR LIGHT OBSERVATION HELICOPTER OH-6A "CAYUSE"

(AVSCOM Reg 310-10)

(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Engine Rotor & Transmission Instruments:													
	Engine Oil Temp, Press & ammeter	O							O					
	Turbine outlet temperature indicator	O	O		O				O					
	Engine & rotor tachometer (Dual)	O	O*				O*		O		D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Engine Tachometer (gas producer)	O	O*				O*		O		D			*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.
	Torque Meter	O	O						O					
	Miscellaneous Instruments:													
	Clock	O							O		D			
	Free Air Temperature Indicator	O	O						O					
	Tachometer Generators	O	O						O					
	Fuel Quantity Measuring System:													
	Fuel Quantity Indicator	O							O					
	Fuel Quantity Transmitter	O	O		O				O					

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(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
09	ELECTRICAL SYSTEMS													
	Switches, Circuit breakers shockmounts	O							O					
	Receptacles, conduits, leads connectors plugs	O							O	O				
	Relays & Flasher Units	O							O					
	Wiring	O	O						O	O				
	Starter Generator	O	F*						O	F*			*Dependent on disposition of gen. test stand	
	Battery	O	*	*					O	F*			*Ref to TM 11-6140-203-14-2	
	Trim Actuators	O	F						F	F				
	Voltage Regulators	O	O*		O				O				*When the maintenance function cannot be accomplished at AVUM because of skills or equipment, evacuate to AVIM.	
	Exterior Lighting System: Position & Anti-Collision	O							O	O				
	Landing/Hover light	O							O	O				
	Interior Lighting System: Instrument, Utility Warning, Caution and Panel Lights	O							O	* O			*Replace Bulbs & Lens only.	

MAINTENANCE ALLOCATION CHART FOR LIGHT OBSERVATION HELICOPTER OH-6A "CAYUSE" (AVSCOM Reg 310-10)														
(1)	(2)	(3)											(4)	(5)
		MAINTENANCE FUNCTION												
GROUP NO	FUNCTIONAL GROUP	INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	TOOLS AND EQUIPMENT	REMARKS
	Inverters	O							O	*			*Refer to TM11 series for repairs	
	Annunciator Assembly	O							O	F				
	Warning & Caution Assembly	O							O					
	Communications & Navigation Equipment	O							O	*			*Refer to TM 11-1520-214-20/20/1 Maint Functions	
10	FUEL SYSTEMS													
	Fuel Tanks	O		O					O	O				
	Valves, Fittings, lines, hose, filters & filter housing	O							O					
11	FLIGHT CONTROLS SYSTEM													
	Lock Control, One-way	O		O					O	O				
	Bellcranks, tubes & Quadrants	O			O				O	O				
	Push-pull rods, links	O			O				O	O				
	Cyclic Control Stick (Pilot)	O			O				O	O				
	Flight Control System Not Requiring Rigging													
	Bellcranks, tubes & Quadrants	O							O	O				

MAINTENANCE ALLOCATION CHART
FOR
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(AVSCOM Reg 310-10)

(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
	Push-pull rods, links & rodend bearings	O							O	O				
	Cyclic Control Stick (Co-pilot)	O							O	O				
12	UTILITY SYSTEMS													
	(NOT APPLICABLE)													
13	ENVIRONMENTAL CONTROLS SYSTEMS (ECS)													
	Heater Valves	O			O				O	O*			*V Belt Replacement	
	Heater Ducts	O							O	O				
	Heater Controls	O							O					
	Vent Valve	O							O					
	Vent Control	O							O					
14	HOISTS AND WINCHES													
	(NOT APPLICABLE)													
15	AUXILIARY POWER PLANTS (APP)													
	(NOT APPLICABLE)													
16	MISSION EQUIPMENT													
	Armament Control	O							O					
	Panels	O							O	O				

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(1) GROUP NO	(2) FUNCTIONAL GROUP	(3) MAINTENANCE FUNCTION: INSPECT	TEST	SERVICE	ADJUST	ALIGN	CALIBRATE	INSTALL	REPLACE	REPAIR	OVERHAUL	REBUILD	(4) TOOLS AND EQUIPMENT	(5) REMARKS
17	EMERGENCY EQUIPMENT													
	Fire Extinguisher	O	O*						O				*Weight test only.	
	First Aid Kit	O							O					
	Door jettison mechanism	O			O				O					
18	INSTALLED AVIONICS COMPONENTS													

NOTE

TM 11-1520-214-20 and TM 11-1520-214-20-1 contain maintenance instructions for avionics.

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All data on pages B-17 through B-27/B-28 deleted.

APPENDIX C

AIRCRAFT INVENTORY MASTER GUIDE

C-1. AIRCRAFT INVENTORY MASTER GUIDE.

C-2. Introduction — Aircraft Inventory Master Guide. *a.* Appendix C lists those items of installed or loose equipment required by and authorized for using organizations to accomplish their primary or alternate mission. This list will serve to standardize present inventory procedures, using the inventory master guide to determine the inventoriable items of installed and/or loose equipment. Insofar as possible, items of equipment are listed in the sequence of their physical location within the airuraft area.

b. Aircraft inventory is subject to changes as a result of authorized changes (MWO's) and additions or deletions of property for special mission requirements; therefore, the selection of items of inventory from the inventory master guide may or may not provide a complete inventory list. When it is known that the master guide does not provide a complete inventory list, it will be necessary to research authorized changes (MWO's) and local command directives in order to compile an accurate and exact inventory list.

c. Refer to TM 38-750 for applicable forms and records.

C-3. SECURITY.

C-4. General — Security. It is desired that aircraft inventory records be unclassified. Therefore, when equipment bearing a security classification or the installation of unclassified equipment is of a confidential or secret nature, accomplishment of the classification will be in accordance with existing security regulations.

C-5. INVENTORIABLE ITEMS.

C-6. General — Inventoriable Items. The selection of inventoriable items is without regard to the agency, governmental or contractual, furnishing the items.

a. Items to be listed are:

(1) Items essential to the execution of the designated mission of the aircraft, such as electronic, photographic, armament, special mission instruments, and safety and comfort equipment.

(2) Loose equipment delivered with the aircraft and items subject to pilferage or readily converted to personal use.

(3) Modification kits which are issued or distributed to using organizations for installation and which are not immediately placed in work will be recorded on the affected aircraft's DA Form 2408-17 (Aircraft Inventory Record) and identified as loose equipment until modification is completed.

(4) Equipment required for operation in special environment.

b. Items to be excluded are:

(1) Nonaccountable items coded as expendable in the applicable stock lists.

(2) Personal issue or furnished on unit allowance or other authority.

(3) Items or components considered as basic or integral parts of the airframe or basic aircraft such as engines, rotors, skids, and standard instruments.

(4) Equipment publications, checklists, and aircraft forms.

C-7. PERIODS OF INVENTORY.

C-8. General — Periods of Inventory. Inventoriable items will be checked against the Aircraft Inventory Record (DA Form 2408-17) at the following periods:

a. Upon receipt of the aircraft.

b. Prior to transfer of the aircraft to another organization.

c. Upon placing aircraft in storage and upon removing from storage. Aircraft need not be inventoried while in storage.

d. Twelve months elapsed time since last inventory.

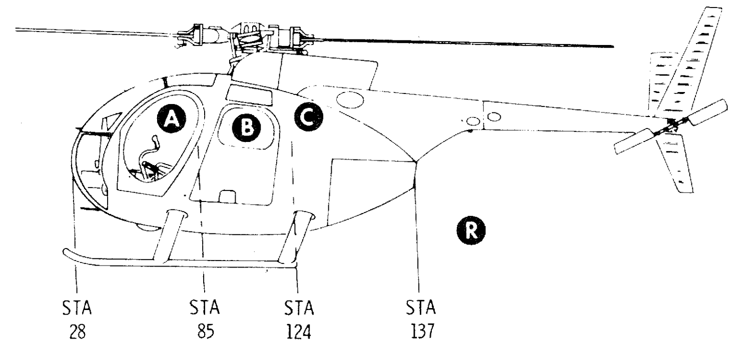
e. Loose equipment shipped under separate cover are inventoried upon transfer by the sending activity and immediately upon receipt by the receiving activity.

(A) PILOTS COMPARTMENT
STA 28 TO 85
Armor Seat Back Assy L-H P/N 1560-OH6-007
Armor Seat Back Assy R-H P/N 1560-OH6-008
Armor Assy Under Seat R-H P/N 1560-OH6-030
Armor Assy Under Seat L-H P/N 1560-OH6-009

(B) PASSENGER-CARGO COMPARTMENT
STA 85 TO 124

(C) POWER PLANT COMPARTMENT
STA 124 TO 137
Armor Assy, Engine Fuel System, P/N 1560-OH6-028
Armor Assy, Engine Compressor, P/N 369A700-9

(R) ACCESSIBLE FROM OUTSIDE FUSELAGE



NOTE: ONLY THOSE ITEMS LISTED WHICH ARE INSTALLED OR ASSIGNED TO A PARTICULAR AIRCRAFT ARE TO BE LISTED ON FORM DA 2408-17, AIRCRAFT INVENTORY RECORD, FOR THE AIRCRAFT.

NOMENCLATURE	AIRCRAFT SERIES AND NUMBER OF ITEMS NORMALLY INSTALLED: OH-6A SERIES 1 & 2	OH-6A SERIES 3	LOCATION OR REMARKS
SECTION A - PILOTS COMPARTMENT			
Aircraft Manufacturer's Data Plate	1	1	
Pilots Seats 369A2005	2	2	
Pilots Seat Belt Shoulder Harness-Inertia Reel Assemblies 369A4536	2	2	
Copilots Cyclic Pitch Stick 369A7801 or 369A7812	1	1	Note 4
Outside Air Temperature Thermometer NSN 6685-00-557-5316	1	1	
Receiver-Transmitter RT-348( )/ARC-54	1	-	
Receiver-Transmitter RT-742/ARC-51BX	1	1	Notes 1, 5
Directional Gyro CN-998( )/ASN-43	1	1	Under copilots seat
Receiver R-1391( )/ARN-83	1	-	
Receiver-Transmitter RT-802/ARC-111	1	-	
Static Inverter 369A4554	2	-	Under pilots seat (Note 3)
Interphone C-1611/AIC	2	-	
IFF Transponder Set AN/APX-72	1	-	
FM Auxiliary Receiver AN/ARC-114	1		

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NOTE: ONLY THOSE ITEMS LISTED WHICH ARE INSTALLED OR ASSIGNED TO A PARTICULAR AIRCRAFT ARE TO BE LISTED ON FORM DA 2408-17, AIRCRAFT INVENTORY RECORD, FOR THE AIRCRAFT.

NOMENCLATURE	AIRCRAFT SERIES AND NUMBER OF ITEMS NORMALLY INSTALLED		LOCATION OR REMARKS
	OH-6A SERIES 1 & 2	OH-6A SERIES 3	
SECTION A - PILOTS COMPARTMENT (CONT)			
HF Exciter AN/ARC-98	1		
HF Antennafier	1		
Static Inverter 369A4654	-	1	
Intercom C-1611D/AIC	2	-	
Magnetic Compass NSN 6605-00-526-7256 (Alternate NSN 6605-00-551-8187)	1	1	
Ni-Cad Battery 369A4530	1	-	
Ni-Cad Battery 369A4360	-	1	
Communication Control System C-6533/AIC	-	2	
Radio Set, VHF (FM No. 1) AN/ARC-114	-	1	
Radio Set, VHF (AM) AN/ARC-115	-	1	
Radio Set, UHF (AM) AN/ARC-116	-	1	Note 5
Radio Receiver R-1496( )/ARN-89	-	1	
ADF Sense Preamplifier AM-4859/ARN-89	-	1	
Radio Receiver Transmitter RT-859/APX-72	-	1	
Antenna Transponder AT-884/APX-44	-	1	
8-Day Clock NSN 6645-00-950-8599	1	1	
First Aid Kit NSN 6545-00-919-6650	1	1	
Fire Extinguisher NSN 4210-00-555-8837	1	1	
Utility Light Type C4-A	1	1	
Computer Kit KIT-1A T/SEC	-	1	
Transponder Test Set TS-1843APX	-	1	
Comm Security Set TSEC/KY-28	-	1	
Control Ind Assembly C-8157	-	3	
SECTION B - PASSENGER-CARGO COMPARTMENT			
Troop Seats 369A2011	2	2	
Seat Belt Extension 369A2575	4	4	
Lap Belt 56E589	2	2	
Shoulder Harness 57D677	2	2	

11-184-2

NOTE: ONLY THOSE ITEMS LISTED WHICH ARE INSTALLED OR ASSIGNED TO A PARTICULAR AIRCRAFT ARE TO BE LISTED ON FORM DA 2408-17, AIRCRAFT INVENTORY RECORD, FOR THE AIRCRAFT.	AIRCRAFT SERIES AND NUMBER OF ITEMS NORMALLY INSTALLED		LOCATION OR REMARKS
NOMENCLATURE	OH-6A SERIES 1 & 2	OH-6A SERIES 3	
SECTION B - PASSENGER-CARGO COMPARTMENT (CONT)			
Jacking Fittings 369A2010	2	2	
Cargo Tiedown Fittings 369A2009	8	8	
Exhaust Covers 369ASK1970	2	2	Note 6
Pitot Tube Cover 369A4026	1	1	
Pitot Tube Cover 369H4009	1	1	Note 7
Blade Tiedown Sock 369A4027	4	4	
Passenger Footswitch Cable Assembly 369A4463	-	1	
Communication Control System C-6533/ARC	-	1	
SECTION C - POWER PLANT COMPARTMENT			
SECTION R - ACCESSIBLE FROM OUTSIDE OF FUSELAGE			
Barrier Air Filter Element 102640	1	-	Inside air inlet aft fairing on modified aircraft
Inertial Particle-Separator Air Filter 369A8413	-	1	Inside air inlet aft fairing
Induction Compass Transmitter T-611/ASN	1	1	Inside air inlet aft fairing
Magnetic Flux Compensator CN-405( )/ASN	1	1	Mounted on T-611/ASN

NOTES:

1. CONUS, USARPAC, USARAL.
2. USAREUR, USARSOUTHCOM.
3. Serial no. 65-12945 and subsequent contain only one inverter. Prior aircraft may be converted to one-inverter system.
4. Series 1 and 2 aircraft (armored) must use cyclic stick part no. 369A7812. Part no. 369A7812 is standard on series 3 aircraft.
5. RT-742/ARC-51BX used for UHF comm until radio set AN/ARC-116 available.
6. For aircraft with upward exhaust system.
7. For aircraft with heated pitot tube.

11-184-3

APPENDIX D

ILLUSTRATED FIELD MANUFACTURE ITEMS LIST

D-1. ILLUSTRATED FIELD MANUFACTURE ITEMS LIST.

D-1. Introduction — Illustrated Field Manufacture Items List. This appendix includes complete instructions, including bills of material, for field manufacture of all items listed in TM 55-1520-214-23P, Aviation Unit, Intermediate, and Depot Maintenance Repair Parts and Special Tools List, bearing MO, MF, or MD Source Code.

The part number index lists all items in part number order with a cross-reference to the figure in which the item appears. All materials necessary for manufacture of an item are listed, by National stock number, in the bill of material for the item. Unless otherwise specified all dimensions are in inches. Dimensional tolerances for 3-place decimal ±0.030, and for angular measurements ±1/2 degree.

PART NUMBER INDEX

PART NUMBER	FIGURE NUMBER
HS208-1400	D1
HS208-423	D1
HS208A1400	D1
HS208A324	D1
HS208A412	D1
HS208A424	D1
HS208A710	D1
HS4029-200	D1
MS20253-2-1275	D9
MS20257HP2-1300	D18
MS27212-1-2	D20
369A1133	D15
369A2000-13	D28
369A2000-14	D1
369A2000-15	D1
369A2000-16	D1
369A2000-3	D1
369A2000-7	D1
369A2001-11	D17
369A2002-19	D1
369A2002-31	D1
369A2002-32	D1
369A2012-41	D1
369A2012-43	D1
369A2012-45	D1
369A2012-49	D1
369A2012-52	D1
369A2012-65	D1
369A2017-17	D1
369A2017-19	D1
369A2017-21	D1
369A2017-23	D1
369A2017-25	D1
369A2019-7	D1
369A2024-59	D1
369A2024-83	D26
369A2025-49	D1
369A2025-69	D1
369A2025-71	D1
369A2025-85	D1
369A2044-23	D1
369A2044-27	D1
369A2044-28	D1
369A2044-34	D1
369A2089	D1
369A2090	D27
369A2401-35	D1
369A2401-36	D1
369A2545-97	D1
369A2639-10	D23
369A2639-11	D23
369A2639-12	D23
369A2639-13	D23
369A2639-14	D23
369A2639-15	D23
369A2639-16	D23
369A2639-17	D23
369A2639-18	D23
369A2639-3	D23
369A2639-4	D23
369A2639-5	D23
369A2639-6	D23
369A2639-7	D23
369A2639-8	D23
369A2639-9	D23
369A2677-3	D1
369A2677-4	D1
369A2677-5	D1
369A3023-35	D26
369A3055-11	D1
369A3055-12	D1
369A3055-13	D1
369A3055-15	D1
369A3055-17	D1
369A3055-19	D1
369A3055-21	D1
369A3060	D6
369A3625-27	D1
369A4007-7	D5

PART NUMBER	FIGURE NUMBER
369A4007-11	D5
369A4007-13	D5
369A4007-15	D5
369A4007-19	D5
369A4007-21	D5
369A4007-27	D5
369A4007-39	D5
369A4007-43	D5
369A4007-45	D5
369A4007-47	D5
369A4007-51	D5
369A4007-53	D5
369A4007-55	D5
369A4120-3	D7
369A4122	D3
369A4133-3	D21
369A4143-15	D8
369A4156	D4
369A4182-103	D19
369A4182-105	D19
369A4182-107	D19
369A4203-19	D19
369A4204-37	D22
369A4217-3	D21
369A4217-7	D21
369A4232	D1
369A4250-3	D14
369A4250-7	D21
369A4743-7	D1
369A5012-5	D1
369A5012-7	D1
369A5012-9	D1
369A5020-11	D1
369A5020-13	D1
369A5020-15	D1
369A5020-17	D1
369A5020-19	D1
369A5020-7	D1
369A5020-9	D1
369A5140	D11
369A5306-21	D1
369A5369	D13
369A5370	D10
369A5424	D12
369A5430	D16
369A6100-3	D1
369A6200-15	D1
369A6436-3	D24
369A6436-5	D24
369A6439-11	D25
369A6501-3	D21
369A6501-5	D21
369A7121-11	D2
369A7121-13	D2
369A7121-15	D2
369A7121-7	D2
369A7121-9	D2
369A7124-19	D21
369A7124-5	D21
369A7307-11	D2
369A7307-13	D2
369A7307-15	D2
369A7307-17	D2
369A7307-19	D2
369A7307-21	D2
369A7307-23	D2
369A7307-25	D2
369A7307-27	D2
369A7307-29	D2
369A7307-3	D21
369A7307-31	D2
369A7307-33	D2
369A7307-5	D21
369A7307-7	D2
369A7307-9	D2
369A7349-25	D2
369A7719-19	D19
369A7719-5	D2
369A7719-9	D2
369A7801-11	D2
369A7801-13	D2
369A7801-15	D2
369A7801-7	D2
369A7801-9	D2
369A7807-3	D2
369A7807-5	D2
369A7807-7	D2
369A7807-9	D21
369A7812-19	D21
369A7812-5	D21
369A8050-13	D1
369A8131-17	D21
369A8131-19	D21
369A8131-21	D21

PART NUMBER	ITEM NAME	FABRICATE FROM NSN	TRIM WIDTH IN INCHES	TRIM LENGTH IN INCHES
HS208-1400	EXTRUSION	9390-00-106-2478		14.00
HS208-423	EXTRUSION	9390-00-106-2478		4.71
HS208A1400	EXTRUSION	9390-00-106-2478		14.00
HS208A324	SEAL	9390-00-106-2478		3.75
HS208A412	SEAL	9390-00-106-2478		4.37
HS208A424	SEAL	9390-00-106-2478		4.75
HS208A710	SEAL	9390-00-106-2478		7.31
HS4029-200	SEAL	9390-00-807-3982		2.00
369A2000-3	TAPE	8135-00-634-2946		AR
369A2000-7	TAPE	8135-00-634-2946		AR
369A2000-13	**STRIP**	9390-00-142-6389		
369A2000-14	STRIP	9320-00-142-6389		AR
369A2000-15	STRIP	**9390-00-142-6389**		AR
369A2000-16	STRIP	**9390-00-142-6389**		AR
369A2002-19	STRIP	9320-00-420-7896		AR
369A2002-31	SEAL	9390-00-817-5855		AR
369A2002-32	SEAL	9390-00-817-5855		AR
369A2012-41	STRIP	5330-00-817-5855		60.00
369A2012-43	STRIP	5330-00-817-5855		47.00
369A2012-45	STRIP	9320-00-420-7896		12.00
369A2012-49	STRIP	9320-00-420-7896		16.00
369A2012-52	STRIP	9320-00-420-7896		36.00
369A2012-65	SEAL	9390-00-420-7896		9.50
369A2017-17	SEAL	9320-00-880-2998		2.20
369A2017-19	SEAL	9320-00-880-2998		2.87
369A2017-21	SEAL	9320-00-880-2998		3.31
369A2017-23	SEAL	9320-00-880-2998		1.87
369A2017-25	SEAL	9320-00-880-2998		1.18
369A2019-7	GASKET	9390-00-983-7613		24.00
369A2024-59	STRIP	7510-00-888-4756	0.50*	40.00
369A2025-49	SEAL	7510-00-948-9543	0.75	4.00
369A2025-69	SEAL	7510-00-878-2052	1.00	23.00
369A2025-71	SEAL	7510-00-878-2052	1.00	23.00
369A2025-85	SEAL	7510-00-948-9543	0.38	12.00
369A2044-23	SEAL	9330-00-984-5834	0.38	12.00

PART NUMBER	ITEM NAME	FABRICATE FROM NSN	TRIM WIDTH IN INCHES	TRIM LENGTH IN INCHES
369A2044-27	SEAL	9330-00-133-6558	1.00	48.00
369A2044-28	SEAL	9330-00-133-6558	1.00	48.00
369A2044-34	SEAL	8135-00-052-9070	1.00	30.00
369A2089	TAPE	8135-00-880-7351		AR
369A2401-35	SEAL	9390-00-641-3368		AR
369A2401-36	SEAL	9390-00-641-3368		AR
369A2545-97	PAD	5940-00-066-2325	1.00	2.00
369A2677-3	LINER	9330-00-522-2855	23.30	28.50
369A2677-4	LINER	9330-00-522-2855	23.30	28.50
369A2677-5	LINER	9330-00-522-2855	27.50	28.60
369A3055-11	SEAL	8040-00-989-0026	0.62	7.00
369A3055-12	SEAL	8040-00-989-0026	0.62	7.00
369A3055-13	SEAL	8040-00-989-0026	0.62	4.00
369A3055-15	SEAL	8040-00-989-0026	0.62	10.00
369A3055-17	SEAL	8040-00-989-0026	0.62	3.00
369A3055-19	SEAL	9320-00-880-2998		3.00
369A3055-21	SEAL	9320-00-880-2998		2.00
369A3625-27	SEAL	9390-00-807-3312		4.12
369A4232	COVER	9535-00-232-0543	0.75	1.50
369A4743-7	TAPE	8315-00-926-1612		8.50
369A5012-5	SEAL	8040-00-989-0026	0.38	14.25
369A5012-7	SEAL	8040-00-989-0026	0.63	1.66
369A5012-9	SEAL	8040-00-989-0026	0.38	5.75
369A5020-7	GASKET	9330-00-133-6558	0.75	2.25
369A5020-9	GASKET	9330-00-133-6558	0.75	1.38
369A5020-11	GASKET	9330-00-133-6558	0.75	1.50
369A5020-13	GASKET	9330-00-133-6558	0.75	23.00
369A5020-15	GASKET	9330-00-133-6558	0.75	16.50
369A5020-17	GASKET	9330-00-133-6558	0.75	2.50
369A5020-19	GASKET	9330-00-133-6558	0.75	2.50
369A5306-21	SEAL	8040-00-989-0026	0.25	AR
369A6100-3	TAPE	7220-00-823-7419	2.40	12.00
369A6200-15	SEAL	9320-00-880-2998		9.00
369A8050-13	GASKET	8040-00-989-0026	0.25	0.25

NOTES:

1. TRIM TO FIT AT TIME OF INSTALLATION.
2. ASTERISK (*) INDICATES THAT ALIPHATIC NAPTHA (C70) MUST BE USED ON SURFACE PER TB 746-93-2.

11-201

Figure D-1.

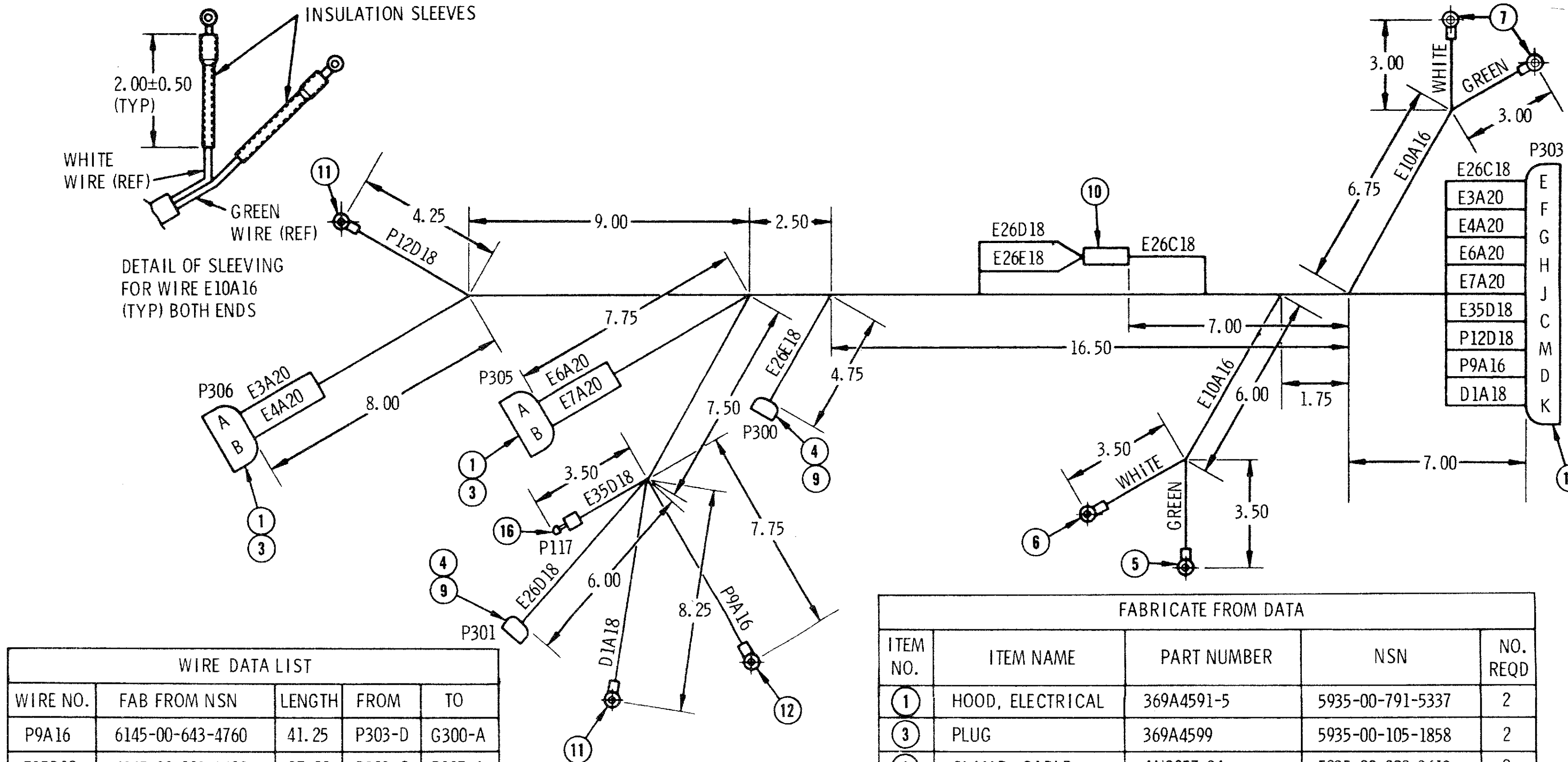
PART NUMBER	ITEM NAME	FABRICATE FROM NSN	LENGTH	WIRE NUMBER (NOTE)
369A7121-7	WIRE	6145-00-851-8505	14.00	C5C20
369A7121-9	WIRE	6145-00-851-8505	14.00	C4C20
369A7121-11	WIRE	6145-00-851-8505	14.00	C1C20
369A7121-13	WIRE	6145-00-851-8505	14.00	C3C20
369A7121-15	WIRE	6145-00-851-8505	14.00	C2C20
369A7307-7	WIRE	6145-00-851-8505	62.00	C9C20
369A7307-9	WIRE	6145-00-851-8505	62.00	C8C20
369A7307-11	WIRE	6145-00-851-8505	62.00	C10C20
369A7307-13	WIRE	6145-00-851-8505	62.00	L23A20N
369A7307-15	WIRE	6145-00-851-8505	62.00	L4B20
369A7307-17	WIRE	6145-00-851-8505	62.00	P11B20
369A7307-19	WIRE	6145-00-851-8505	62.00	P12A20
369A7307-21	WIRE	6145-00-851-8505	38.00	C109F20
369A7307-23	WIRE	6145-00-851-8505	38.00	C108G20
369A7307-25	WIRE	6145-00-851-8505	38.00	C110F20
369A7307-27	WIRE	6145-00-851-8505	38.00	L123A20N
369A7307-29	WIRE	6145-00-851-8505	38.00	L104C20
369A7307-31	WIRE	6145-00-851-8505	38.00	P111D20
369A7307-33	WIRE	6145-00-851-8505	38.00	P112A20
369A7349-25	LACING	4020-00-240-2154	4.00	--
369A7719-5	CABLE	6145-00-998-7609	8.50	--
369A7719-9	CABLE	6145-00-998-7609	5.75	--
369A7801-7	WIRE	6145-00-851-8505	6.00	C5D20
369A7801-9	WIRE	6145-00-851-8505	6.00	C4D20
369A7801-11	WIRE	6145-00-851-8505	8.00	C1D20
369A7801-13	WIRE	6145-00-851-8505	10.00	C3D20
369A7801-15	WIRE	6145-00-851-8505	12.00	C2D20
369A7807-3	WIRE	6145-00-851-8505	18.00	C8D20
369A7807-5	WIRE	6145-00-851-8505	18.00	C9D20
369A7807-7	WIRE	6145-00-851-8505	18.00	C10D20

NOTE:

WIRE NUMBER TO BE APPLIED ON WIRE WITH MARKING MACHINE NSN 3611-00-204-2809.

11-202

Figure D-2.

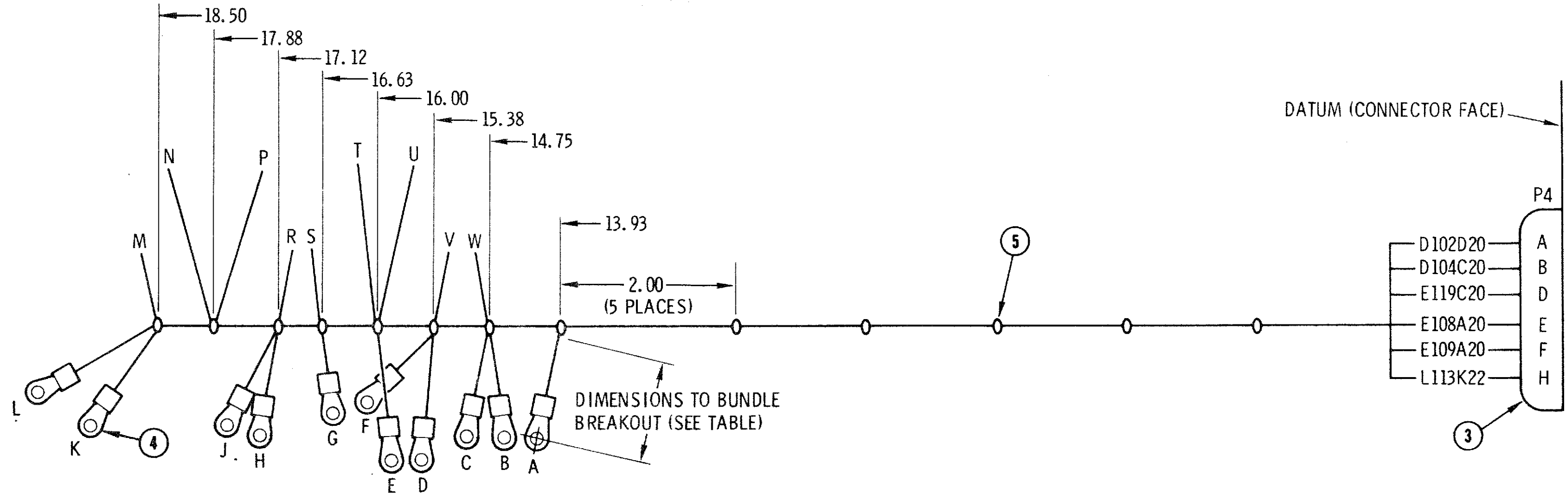


WIRE DATA LIST				
WIRE NO.	FAB FROM NSN	LENGTH	FROM	TO
P9A16	6145-00-643-4760	41.25	P303-D	G300-A
E35D18	6145-00-993-5490	37.00	P303-C	P117-A
E26E18	6145-00-993-5490	14.25	SPLICE	P300
E26D18	6145-00-993-5490	25.50	SPLICE	P301
E26C18	6145-00-993-5490	14.00	P303-E	SPLICE
WHITE			TB4-2	TC300
E10A16	6145-00-643-0252	21.00		
GREEN			TB4-1	TC300
E7A20	6145-00-851-8505	33.75	P303-J	P305-B
E6A20	6145-00-851-8505	33.75	P303-H	P305-A
E4A20	6145-00-851-8505	43.00	P303-G	P306-B
E3A20	6145-00-851-8505	43.00	P303-F	P306-A
D1A18	6145-00-993-5490	41.75	P303-K	G300-D
P12D18	6145-00-993-5490	39.25	P303-M	G301

FABRICATE FROM DATA				
ITEM NO.	ITEM NAME	PART NUMBER	NSN	NO. REQD
1	HOOD, ELECTRICAL	369A4591-5	5935-00-791-5337	2
3	PLUG	369A4599	5935-00-105-1858	2
4	CLAMP, CABLE	AN3057-3A	5935-00-028-3613	2
5	TERMINAL LUG	AN5538-1	5940-00-502-7570	1
6	TERMINAL LUG	AN5538-2	5940-00-229-7564	1
7	TERMINAL LUG	AN5539-1	5940-00-197-9756	2
9	PLUG	MS106R8S1S	5935-00-849-4268	2
10	SPLICE CONDUCTOR	MS25181-2	5940-00-502-8806	1
11	TERMINAL LUG	MS25036-3	5940-00-143-4771	2
12	TERMINAL LUG	MS25036-108	5940-00-143-4780	1
15	PLUG	CA3106E22-19SF80	5935-00-992-5309	1
16	KNIFE SPLICE	32445	5940-00-236-1572	1
20	LACING CORD	TYPE 3333	4020-00-807-4491	AR
22	SOLDER	QQS571	3439-00-224-3567	AR
	INSULATION SLEEVE	MIL1631SZ5	5970-00-284-8640	AR

11-209

Figure D-3. Part Number 369A4122 Wiring Harness.



WIRE NO.	LENGTH	(REF DESIGNATOR TERMINATION) FROM	TO	DIM TO BREAKOUT	SYM
D102D20	15.05	P4-A	M4-1	1.12	A
D104B20	2.75	M6-2	M4-3	1.12	B
D104C20	15.88	M4-3	P4-B	1.12	C
E119C20	16.75	P4-D	M6-1	1.38	D
E119D20	3.62	M6-1	M7-1	1.38	E
D104B20				1.00	F
D104A20	4.00	M7-2	M6-2	0.88	G
E108A20	18.25	P4-E	M6-3	1.12	H
E119D20				1.12	J
D104A20				1.25	K
E109A20	20.00	P4-F	M7-3	1.50	L
L113M22	2.88	XDS112-1	XDS111-1	0.75	M
L115A22	3.50			1.75	N
L115A22				1.75	P
L113M22				0.75	R
L113L22				0.75	S
L115B22				1.75	T
L115B22	3.50	XDS112-2	XDS113-2	1.75	U
L113L22	2.75	XDS113-1	XDS112-1	0.75	V
L113K22	15.50	P4-H	XDS113-1	0.75	W

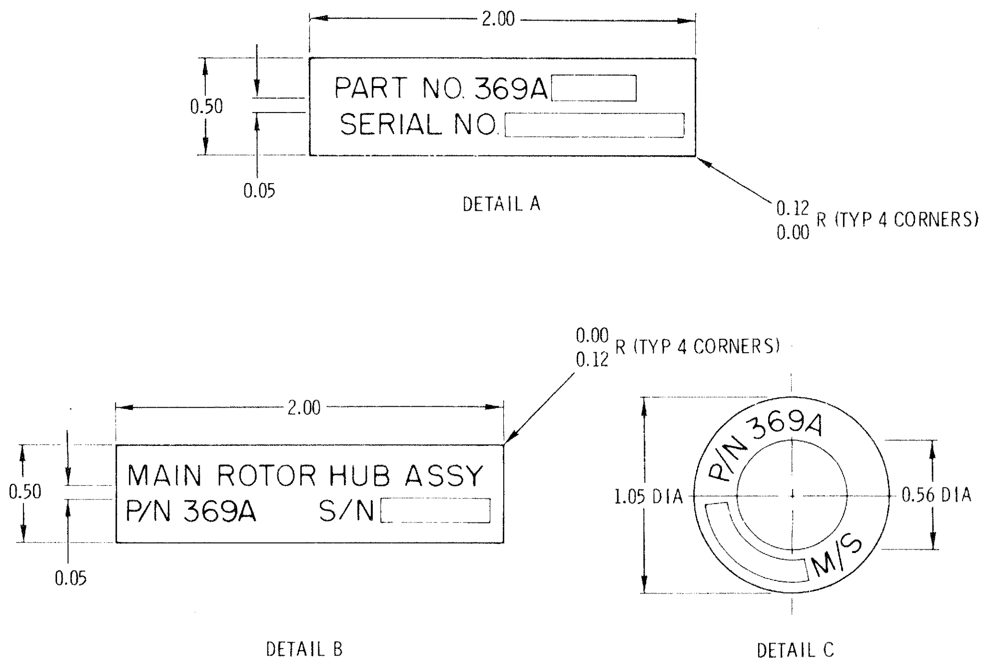
FABRICATE FROM DATA				
ITEM NO.	ITEM NAME	PART NO.	NSN	NO. REQD
1	WIRE	MS25190A20	6145-00-578-7519	AR
2	WIRE	MS25190A22	6145-00-578-7520	AR
3	PLUG	M7P95LSH19CS	5935-00-486-9099	1
4	TERMINAL LUG	MS25036-102	5940-00-204-8966	11
5	STRAP TIE DOWN	MS17821-1-9	5975-00-074-2072	13
6	INSULATION SLEEVE		5970-00-954-1624	AR
7	SOLDER	QQ-S-571	3439-00-224-3567	AR

NOTES:

1. WIRE ENDS "A" THRU "J" ARE SOLDER TERMINATION.
2. REFER TO APPENDIX F, ELECTRONIC SYSTEM WIRING DIAGRAM (SERIES 3 AIRCRAFT) FOR INTERCONNECTION OF HARNESS ON INSTRUMENT ASSEMBLY.

11-216

Figure D-4. Part Number 369A4156 Wiring Harness, Instrument.



PART NUMBER	FABRICATE FROM NSN	DETAIL	PART NUMBER	FABRICATE FROM NSN	DETAIL
369A4007-7	9535-00-554-1003	A	369A4007-39	9535-00-554-1003	A
369A4007-11	9535-00-554-1003	B	369A4007-43	9535-00-554-1003	A
369A4007-13	9535-00-554-1003	A	369A4007-45	9535-00-554-1003	A
369A4007-15	9535-00-554-1003	A	369A4007-47	9535-00-554-1003	C
369A4007-19	9535-00-554-1003	A	369A4007-51	9535-00-554-1003	A
369A4007-21	9535-00-554-1003	A	369A4007-53	9535-00-554-1003	A
369A4007-27	9535-00-554-1003	A	369A4007-55	9535-00-554-1003	A

NOTES:

1. LETTERS AND NUMBERS ARE ALUMINUM; BACKGROUND TO BE FLAT BLACK.
2. IMPRESSION STAMP SERIAL NUMBER.
3. PART NUMBERS ETCHED WITH RESPECTIVE DASH NUMBER.
4. ADHESIVE BACKING TO BE A PERMANENT TYPE.

11-213

Figure D-5.

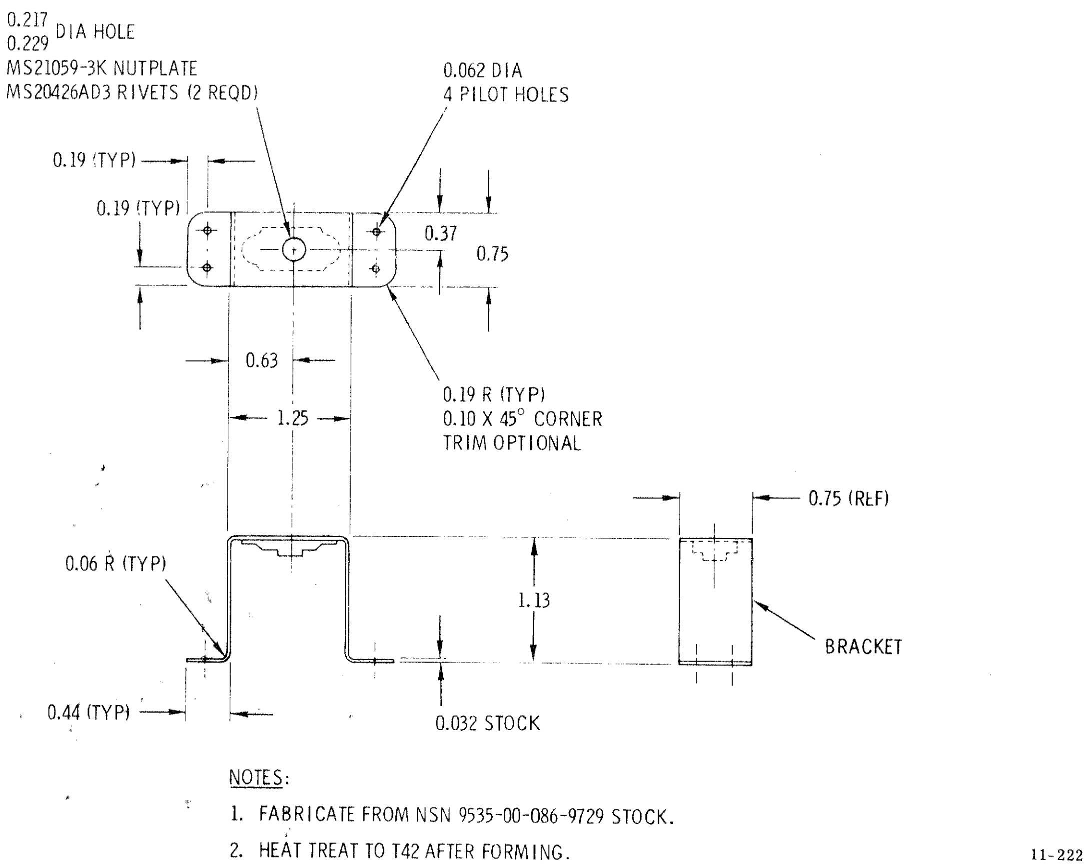


Figure D-6. Part Number 369A3060 Bracket Assembly.

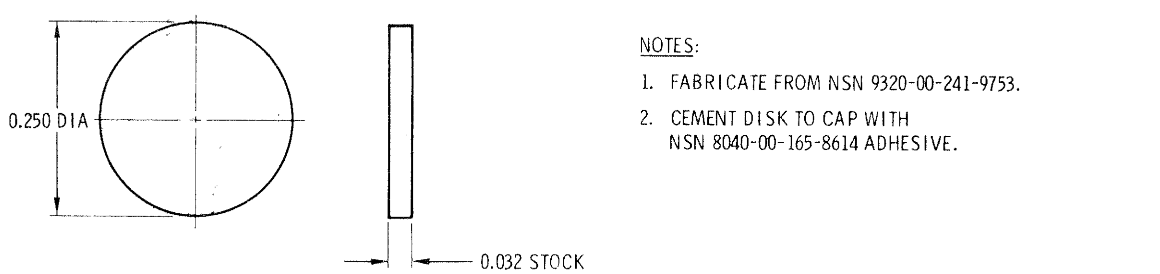
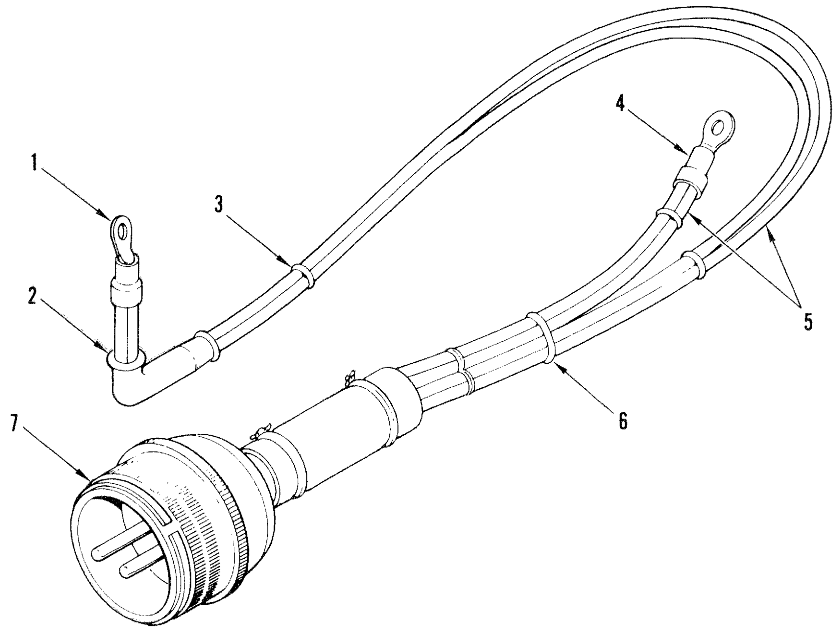


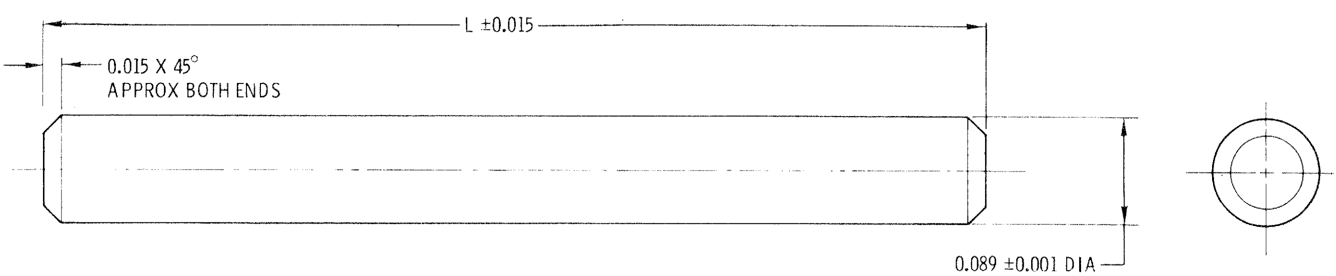
Figure D-7. Part Number 369A4120-3 Receptacle Cap Disk.



ITEM NO.	PART NUMBER	ITEM NAME	FABRICATE FROM NSN	NO. REQD
1	MS25036-123	TERMINAL LUG	5940-00-557-4341	1
2	MS25171-2S	CABLE NIPPLE	5975-00-553-7151	1
3	MS17821-1-9	STRAP TIEDOWN	5975-00-074-2072	1
4	MS25036-125	TERMINAL LUG	5940-00-557-4338	1
5	MS25190B10	WIRE	6145-00-578-7513	AR
6	MS17821-4-9	STRAP TIEDOWN	5975-00-727-5153	1
7	SC06P24-9P	CONNECTOR PLUG	5935-00-933-7398	1

11-215

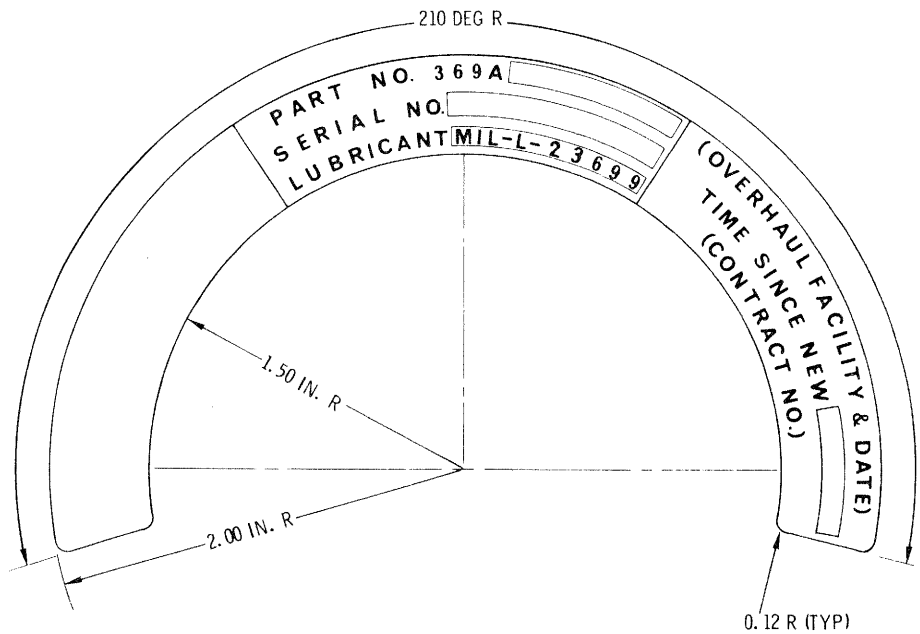
Figure D-8. Part Number 369A4143-15 Harness.



PART NUMBER	FABRICATE FROM NSN	DIM L
MS20253-2-1275	5340-00-043-3723	12.75

11-210

Figure D-9. Pin, Headless.

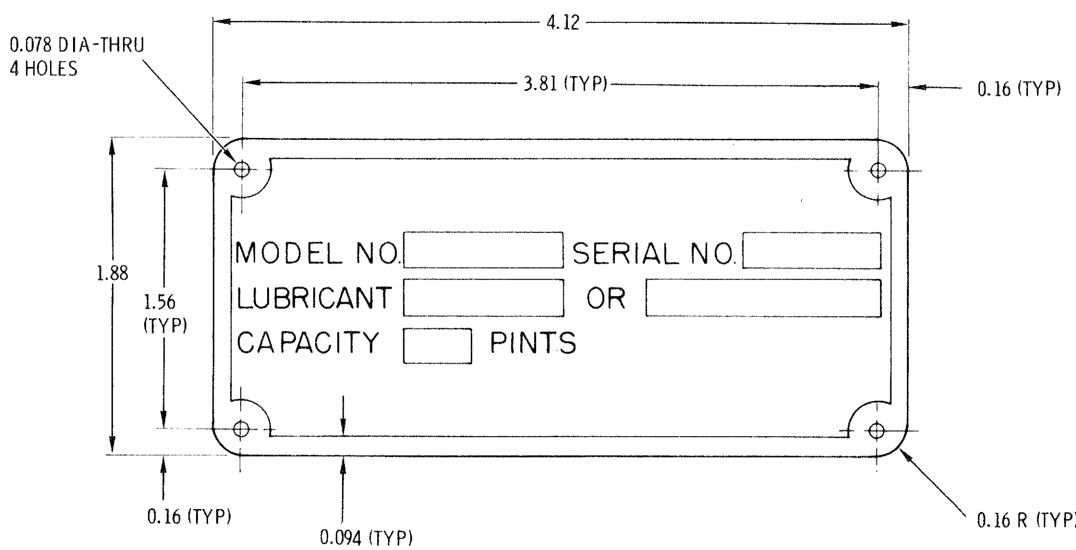


NOTES:

1. FABRICATE FROM NSN 9535-00-554-1003.
2. LETTERS, NUMBERS AND BLOCKS TO BE BARE ALUMINUM. BACKGROUND TO BE FLAT BLACK.

11-221

Figure D-10. Part Number 369A5370 Plate, ID.

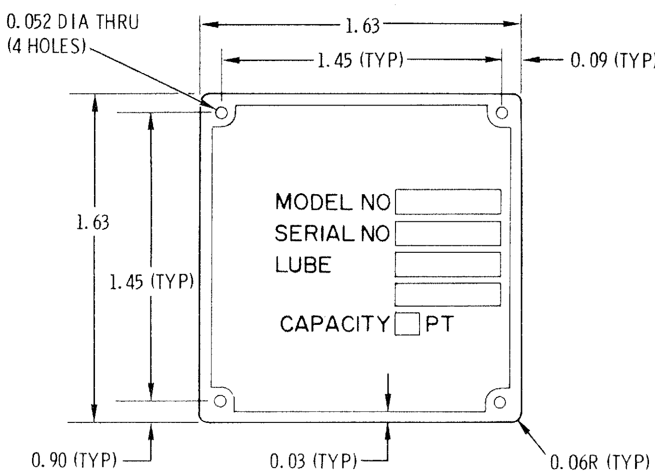


NOTES:

1. FABRICATE FROM NSN 9535-00-554-1003.
2. FINISH: BLACK ANODIZE BACKGROUND PER MIL-A-8625, TYPE II WITH SATIN FINISH ETCHED IMAGE.

11-219

Figure D-11. Part Number 369A5140 Plate, ID.

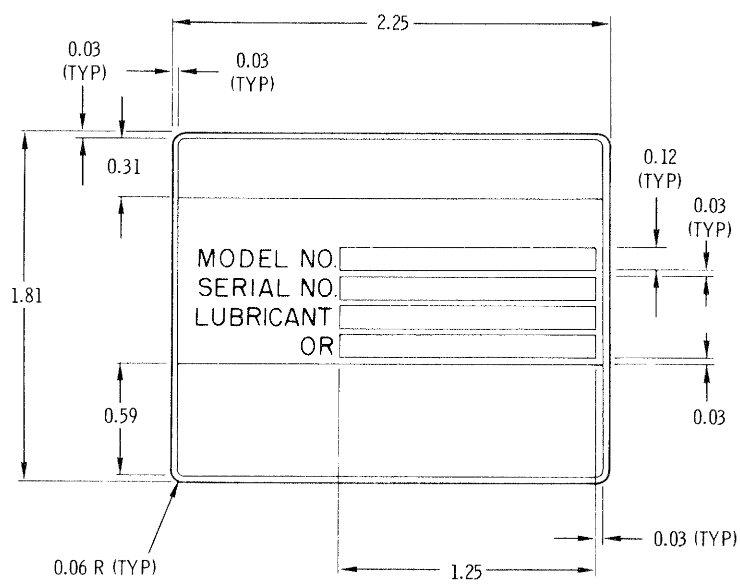


NOTES:

1. FABRICATE FROM NSN 9535-00-554-1003.
2. MIL-L-23699 IS PREFERRED LUBRICANT. DO NOT MARK FOR USE OF MIL-L-7808.

11-218

Figure D-12. Part Number 369A5424 Plate, ID.

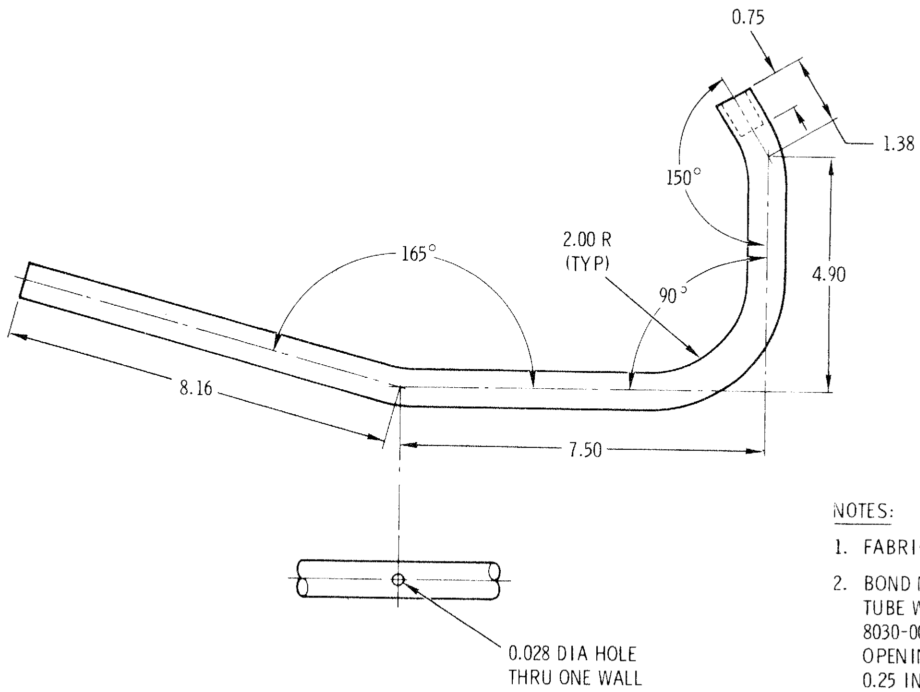


NOTES:

1. FABRICATE FROM NSN 9535-00-242-5665.
2. MIL-L-23699 IS PREFERRED LUBRICANT. DO NOT MARK FOR USE OF MIL-L-7808.

11-217

Figure D-13. Part Number 369A5369 Plate, ID.

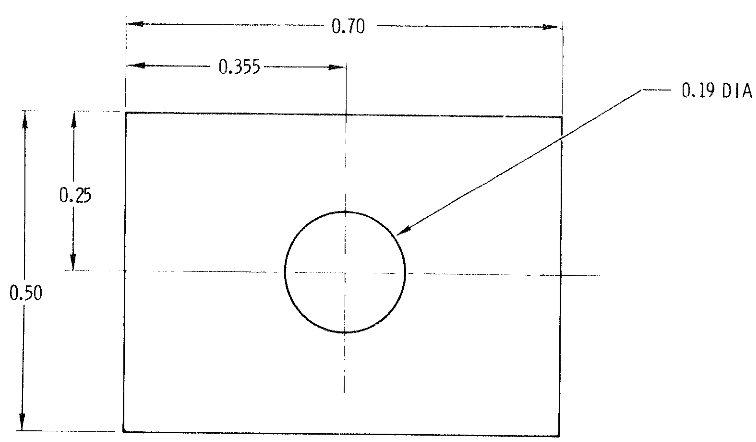


NOTES:

1. FABRICATE FROM NSN 4710-00-279-0969.
2. BOND NYLON PRESSURE LINE TO PITOT TUBE WITH SEALING COMPOUND (NSN-8030-00-281-2339). DO NOT OBSTRUCT OPENING. REAM ID OF PITOT TUBE TO 0.25 IN. IF REQUIRED.

11-214

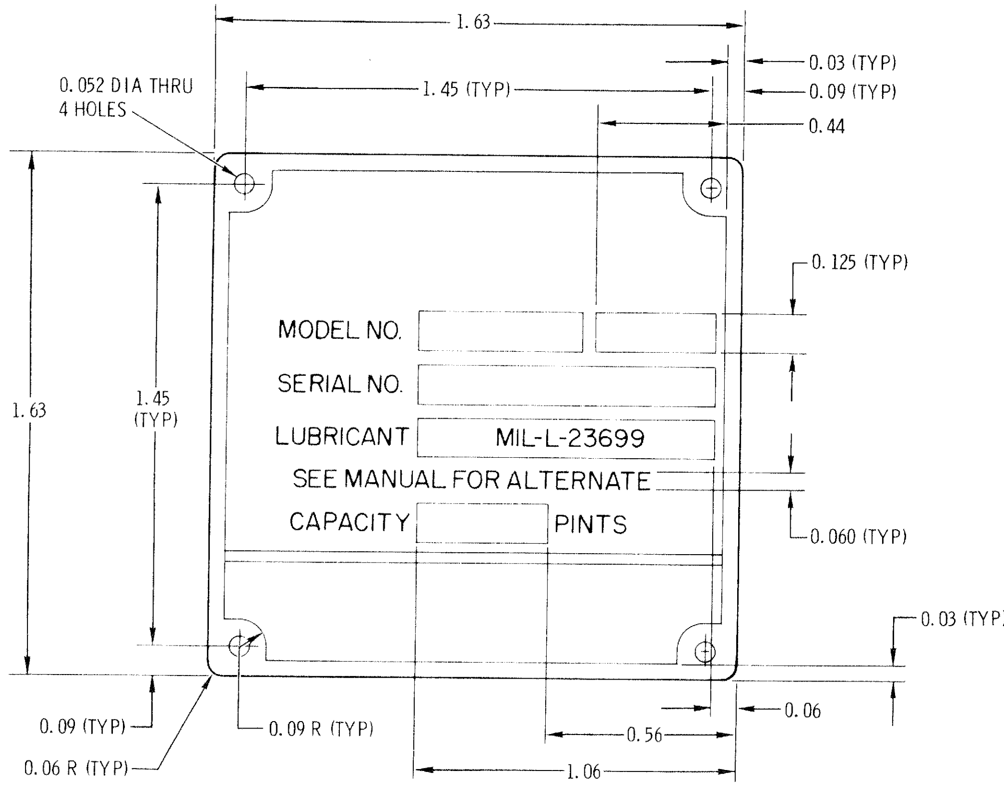
Figure D-14. Part Number 369A4250-3.



PART NUMBER	FABRICATE FROM NSN
369A1133	7510-00-948-9543

11-211

Figure D-15. Part Number 369A1133 Pad.



NOTES:

1. FABRICATE FROM NSN 9535-00-554-1003.
2. FINISH: BLACK ANODIZED BACKGROUND PER MIL-A-8625 TYPE II, WITH SATIN FINISH ETCHED IMAGE.
3. LETTERING SHALL BE VERTICAL, CONDENSED, NEWS GOTHIC STYLE, PER MIL-STD-1, OR IN ACCORDANCE WITH MS33558. SIZE & LOCATION OF CHARACTERS APPROX. AS SHOWN, EXCEPT, LETTERS TO BE NO LESS THAN 0.060 HIGH.

11-212

Figure D-16. Part Number 369A5430 Plate, ID.

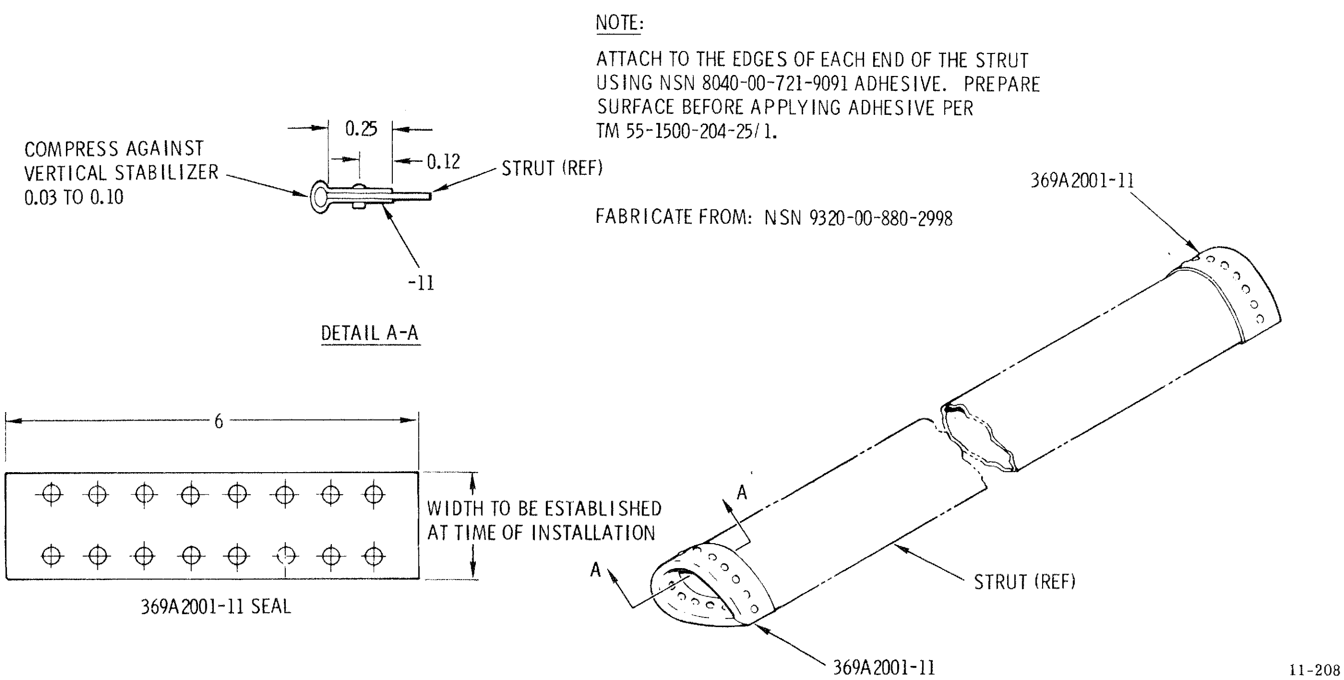
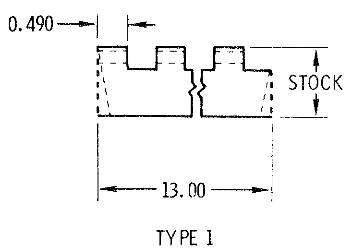


Figure D-17. Part Number 369A2001-11 Seal, Strut Assembly.



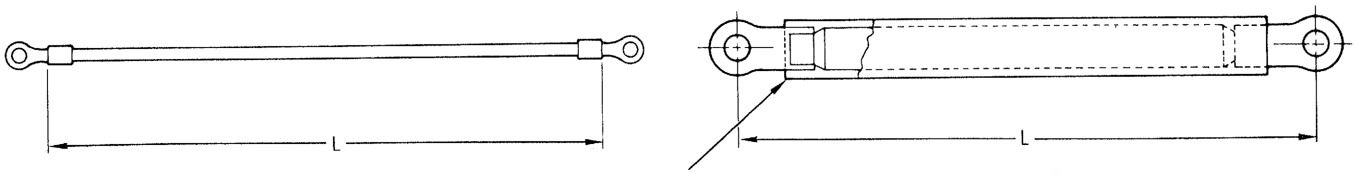
PART NUMBER	FABRICATE FROM NSN	TYPE
MS20257HP2-1300	5340-00-993-1461	1

NOTES:

1. CRIMP BOTH ENDS OF HINGE TO RETAIN HINGE PIN.
2. HOLES TO BE DRILLED AT TIME OF INSTALLATION AND TO BE EQUALLY SPACED ALONG HINGE HALF.

11-207

Figure D-18. Leaf, Butt Hinge.



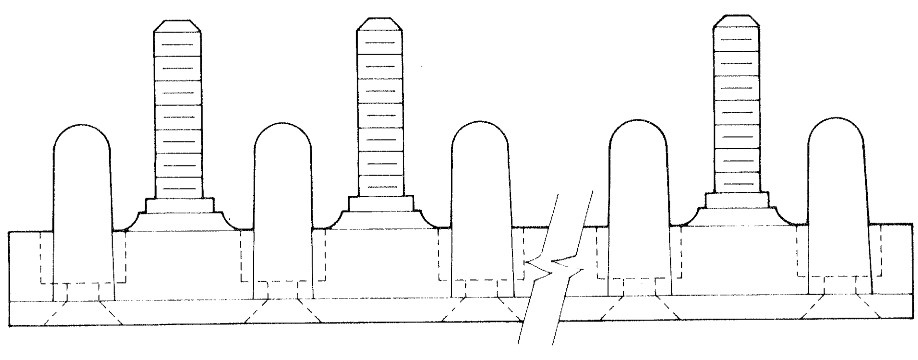
PART NUMBER	ITEM NAME	"L"	WIRE	TERMINAL 1	TERMINAL 2	INSULATION SLEEVE	STYLE
369A4182-103	JUMPER ASSY	7.50	MS25190B10	MS25036-114	MS25036-112	5970-00-815-1295	2
369A4182-105	JUMPER ASSY	7.50	6145-00-578-6602	5940-00-143-4771	5940-00-143-4771	5970-00-812-2969	2
369A4182-107	JUMPER ASSY	4.50	6145-00-578-7519	5940-00-143-4771	5940-00-204-8966	5970-00-812-2967	2
369A4203-19	JUMPER ASSY	6.00	6145-00-578-7519	5940-00-557-1629	5940-00-813-0698		1
369A7719-19	JUMPER ASSY	4.60	6145-00-998-7609	5940-00-143-4777	5940-00-143-4777	5970-00-235-2728	2

NOTE:

MANUFACTURE WIRE ASSEMBLIES PER TM 55-1500-204-25/1.

11-206

Figure D-19.



PART NUMBER	FABRICATE FROM NSN
MS27212-1-2	5940-00-950-1610

NOTE:

LAST DASH NUMBER INDICATES
NUMBER OF STUDS.

11-204

Figure D-20. Terminal Board.

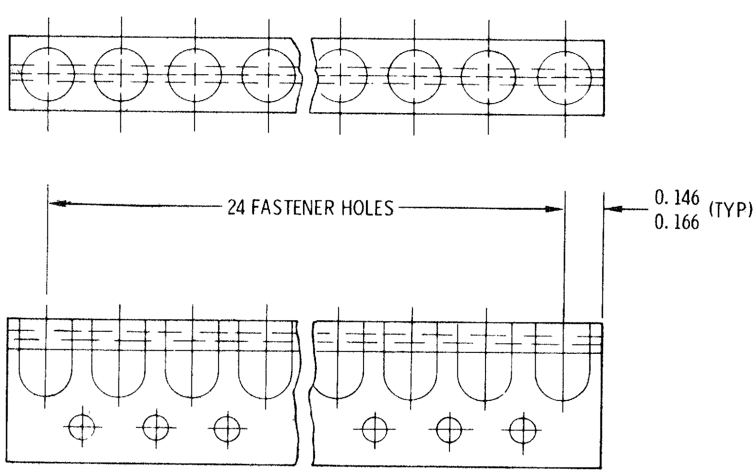
PART NUMBER	ITEM NAME	FABRICATE FROM NSN	LENGTH
369A4133-3	SLEEVE, INSULATION	5970-00-617-5798	23.50
369A4217-3	TUBE, PRESS	9330-00-931-7097	10.00
369A4217-7	TUBE, PRESS	9330-00-931-7097	15.50
369A4250-7	TUBE, PRESS	9330-00-931-7097	34.00
369A6501-3	TUBE (NOTE 2)	9330-00-931-7097	12.00
369A6501-5	TUBE (NOTE 2)	9330-00-931-7097	186.00
369A7124-5	SLEEVE, INSULATION	5970-00-824-0167	AR
369A7124-19	SLEEVE, INSULATION	5970-00-636-1771	AR
369A7307-3	SLEEVE, INSULATION	5970-00-543-1098	18.00
369A7307-5	SLEEVE, INSULATION	5970-00-235-2725	18.00
369A7807-9	SLEEVE, INSULATION	5970-00-543-1098	18.00
369A7812-5	SLEEVE, INSULATION	5970-00-543-1647	AR
369A7812-19	SLEEVE, INSULATION	5970-00-636-1771	AR
369A8131-17	HOSE, RUBBER	4720-00-720-1096	3.50
369A8131-19	HOSE, RUBBER (NOTE 1)	4720-00-540-3644	4.50
369A8131-21	HOSE, (NOTE 1)	4720-00-540-3644	3.30

NOTES:

1. INSTALL HOSE WITH LOCKWIRE NSN 9525-00-803-3044. LENGTH AS REQUIRED.
2. DIMENSIONAL TOLERANCE: 1 PLACE DECIMAL ±0.1; MINIMUM BEND RADIUS 1.0; BURST STRENGTH 1000 PSI MINIMUM.

11-205

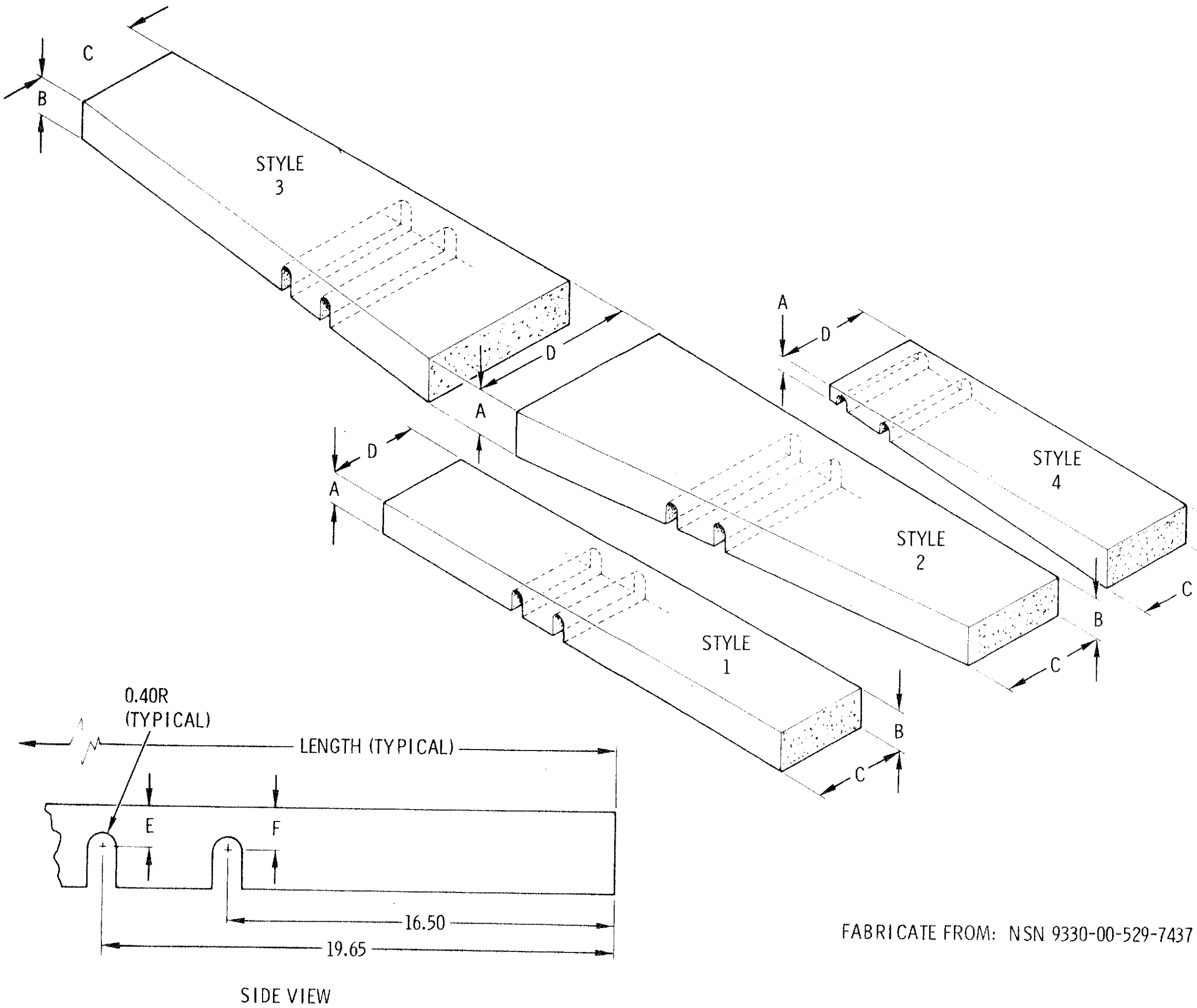
Figure D-21.



PART NUMBER	FABRICATE FROM NSN
369A4204-37	5325-00-036-9305

11-203

Figure D-22. Strip, Receptacle.

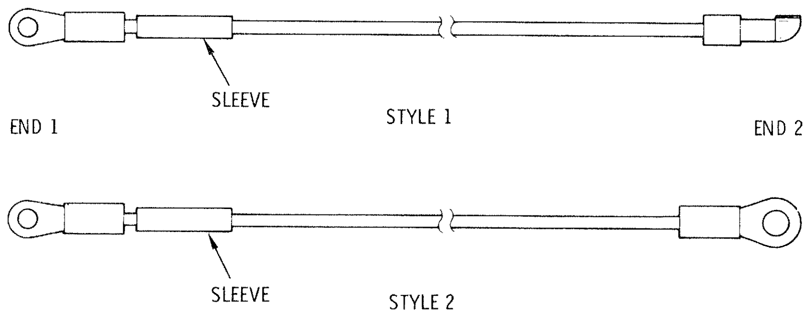


FABRICATE FROM: NSN 9330-00-529-7437

369A2639	STYLE	LENGTH	THICKNESS A	THICKNESS B	WIDTH C	WIDTH D	SLOT E	SLOT F
-3	2	29.75	2.90	2.50	6.50	10.40	1.52	1.45
-4	3	29.75	2.90	2.50	6.50	10.40	1.03	1.07
-5	1	29.75	2.00	2.50	5.82	5.82	1.03	1.07
-6	1	29.75	2.00	2.50	5.82	5.82	0.70	0.78
-7	4	20.62	1.20	2.50	5.82	5.82	0.70	0.78
-8	4	20.62	1.20	2.50	5.82	5.82	0.72	0.82
-9	1	29.75	1.80	2.50	5.82	5.82	0.72	0.82
-10	1	29.75	1.80	2.50	5.82	5.82	1.00	1.21
-11	1	29.75	2.30	2.50	5.82	5.82	1.00	1.21
-12	1	29.75	2.30	2.50	5.82	5.82	1.20	1.53
-13	1	29.75	2.60	2.50	5.82	5.82	1.20	1.53
-14	1	29.75	2.60	2.50	5.82	5.82	1.34	1.71
-15	1	29.75	2.60	2.50	5.82	5.82	1.34	1.71
-16	1	29.75	2.60	2.50	5.82	5.82	1.34	1.75
-17	1	29.75	2.60	2.50	4.33	4.33	1.34	1.75
-18	1	29.75	2.60	2.50	4.33	4.33	1.23	1.37

11-224

Figure D-23.



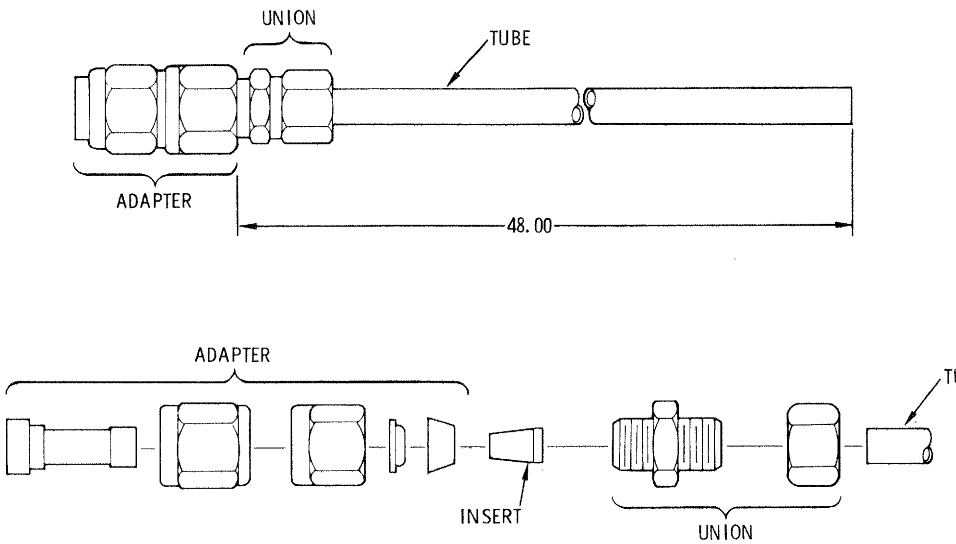
PART NUMBER	STYLE	LENGTH	WIRE	END 1	END 2	SLEEVE	WIRE NO.
369A6436-3	2	58.00	6145-00-177-4583	5940-00-143-4771	5940-00-113-8184	5970-00-088-2975	F98A20
369A6436-5	1	16.00	6145-00-177-4585	5940-00-143-4771	5940-00-236-1572	5970-00-088-2975	F99A20

NOTE:

MANUFACTURE WIRE ASSEMBLIES PER TM 55-1500-204-25/1.
USE HEAT GUN TO SHRINK IDENTIFICATION SLEEVE ON WIRE.
ANY HEAT DISCOLORATION OF WIRE OR SLEEVE IS NOT
ACCEPTABLE.

11-226

Figure D-24.



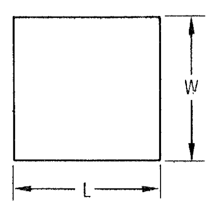
NOTES:

1. DISCARD ONE NUT ATTACHED TO UNION.
2. TO ASSEMBLE UNION, PUSH TUBING INTO FITTING AND TIGHTEN NUT 2 TO 2 1/2 FULL TURNS AFTER FINGER-TIGHT POSITION.
3. TIGHTEN ADAPTER NUTS 1/4 TURN AFTER FINGER-TIGHT POSITION.

ITEM NAME	FABRICATE FROM NSN
TUBE	4720-00-045-4714
UNION	4730-00-572-2346
INSERT	4730-00-733-3884
ADAPTER	1560-00-963-6226

11-227

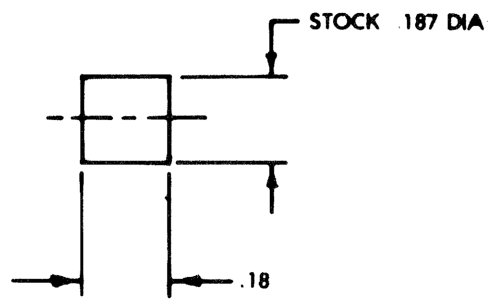
Figure D-25.



PART NUMBER	L	W	FABRICATE FROM NSN
369A3023-35	0.50	0.50	9935-00-142-6891
369A2024-83	1.50	2.00	9535-00-086-9729

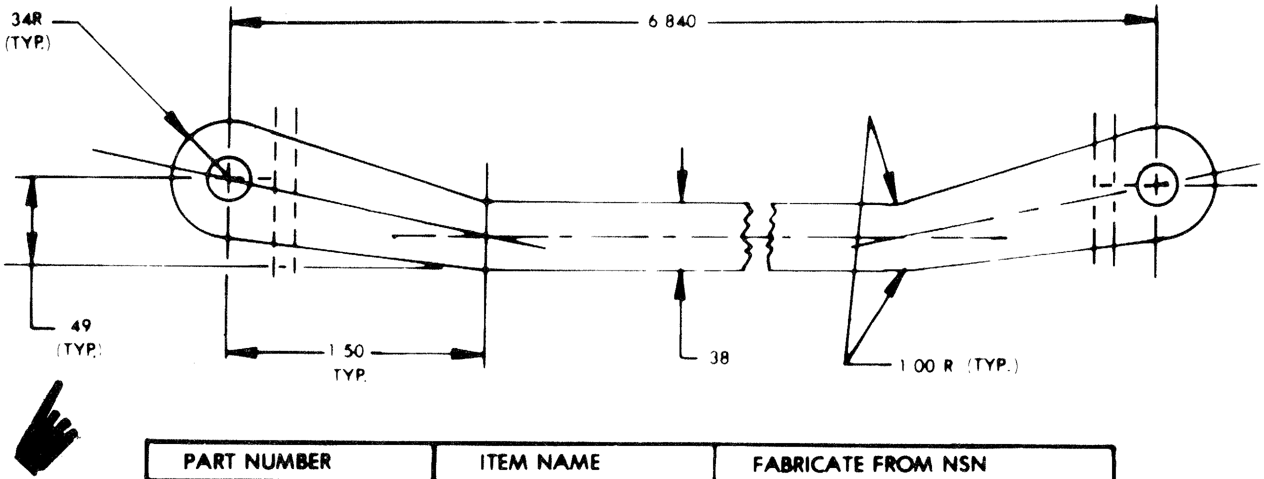
11-229

Figure D-26.



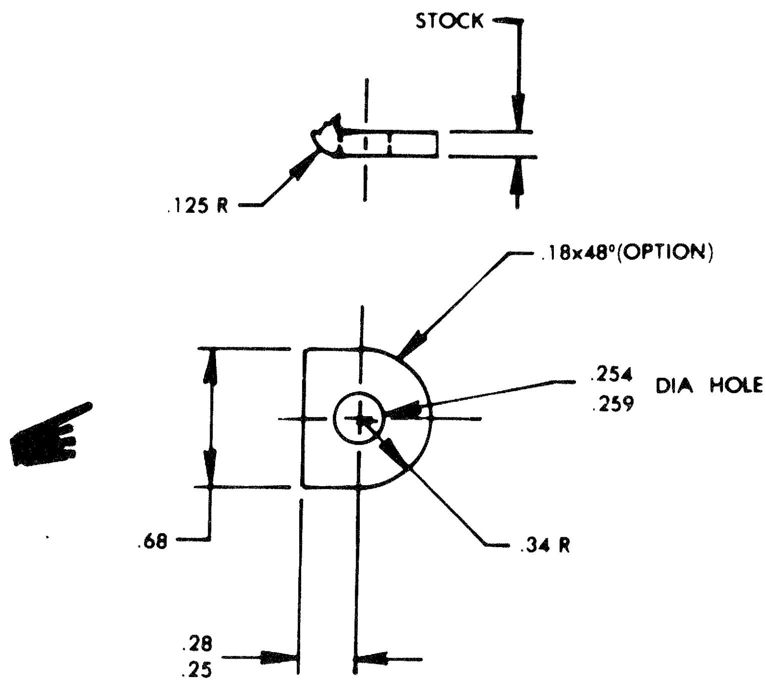
PART NUMBER	FABRICATE FROM
369A2090	5340-00-131-2047

Figure D-27



PART NUMBER	ITEM NAME	FABRICATE FROM NSN
369ASK806-5	STRAP	9515-00-184-8815

Figure D-28



PART NUMBER	ITEM NAME	FABRICATE FROM NSN
369ASK806-9	SPACER	9535-00-554-1417

Figure D-29

APPENDIX E

STORAGE OF AIRCRAFT

SECTION I GENERAL INFORMATION

E-1. GENERAL — AIRCRAFT STORAGE.

E-2. Components Involved in an Accident. Any component removed for reason of accident shall not be preserved, but shall be shipped in the same condition as it was in after the accident.

E-3. Corrosion Control. Storage of aircraft includes corrosion control which consists primarily of preventing moisture from contacting exposed metal surfaces by the use of preservatives. There are two main types of corrosion: direct chemical and electrochemical. Direct chemical corrosion occurs when adjacent surfaces of dissimilar metal become electrolytic cells in the presence of moisture. Preservation procedures are based on the principle that corrosion will not occur on clean moisture-free surfaces. Preservation of the helicopter consists of cleaning the exposed metal surfaces and providing a barrier between the metal surfaces and the moisture-laden atmosphere.

E-4. Environmental Conditions. The existing environmental conditions and available facilities must be taken into account when a helicopter is to be placed in storage. A choice of storage procedures is permissible for short periods of storage. For example, a choice must be made between flyable storage and short term storage for any period of time up to 45 days. The decision will be based on such on-site conditions as availability of men, materials and equipment necessary to perform ground runups, motoring of engines, defueling and purging of fuel tanks, and other elements of the various procedures. Wet weather conditions create corrosion, rot, mildew and mold. To prevent these deteriorating effects, perform inspections regularly and take proper preventive maintenance action. The following practices should be used as a guide during exceptionally wet weather conditions.

a. Prevent rot, mildew, and mold from forming on nonmetallic materials by keeping them clean and as dry as possible. Keep fabric material in the helicopter clean according to procedures in chapter 1.

b. Treat for visible corrosion according to TM 55-1500-204-25/1.

c. Check that drain holes are open, covered with screening and free of blockage.

d. Keep fuel cells full to prevent condensation in the cells for flyable and short term storage.

e. Store helicopters in a hangar or shed if space permits.

E-5. Categories of Storage.

NOTE

Aircraft that are inactive for seven (7) or more consecutive days must be placed in storage.

a. Flyable Storage (No Time Limit). Flyable storage is the procedure prescribed to maintain a stored helicopter in an operable condition. Next to daily use this keeps the aircraft in the best possible condition. It does, however, require attention periodically.

b. Short Term Storage (From 1 to 45 Days). This type of storage is used to store helicopter up to 45 days with very little attention during the storage period.

c. Intermediate Storage (From 46 to 180 Days). A helicopter that will be inactive for more than 45 days, but not exceeding 180 days, shall be maintained in intermediate storage.

E-6. Procedures Common to all Categories of Storage.

a. Preservation. Preservation should be accomplished in an uninterrupted series of operations. When periods of interruption are necessary, temporary protection shall be provided for partially processed items as required to avoid contamination.

b. Corrosion Prevention for Short Term and Intermediate Storage. Before engine runup, add corrosion preventive concentrate (C32) to engine, main transmission, and tail rotor transmission in the quantities specified in TB 55-9150-200-24. Flushing of the engine, main transmission and tail rotor transmission oil systems is not required during depreservation and activation of the aircraft.

NOTE

Refer to TM 38-230-1 for preservation and packaging of component assemblies removed from the aircraft.

c. Ventilation. The prevention of corrosion depends to a large extent on the control of moisture. One very satisfactory method of doing so is by means of ventilation. On days when the relative humidity is 55 percent or below, the doors and other openings can be temporarily opened to allow a circulation of the dry air through the helicopter. Fans or blowers, when available, are very helpful.

d. Drainage. Ensure that water drain holes are free from obstruction and are kept open for the duration of the storage period.

e. Towing, Parking and Mooring. Accomplish all ground handling according to procedures in chapter 1.

E-7. Inspection of Stored Aircraft. Perform inspections periodically to ensure that the aircraft remains properly preserved.

Aircraft in flyable storage will have a daily inspection performed once every 7 days. (Refer to TM 55-1520-214-PMS.) Aircraft in short term and intermediate storage, where the local humidity exceeds 40 percent, shall be inspected for corrosion at least once every 15 days. Aircraft in short term or intermediate storage, where the local humidity is 40 percent or less shall be inspected for corrosion at least once every 30 days. Aircraft in storage shall be inspected for the condition of protective covers whenever they have been subjected to high winds (exceeding 40 mph). In addition, the following general inspection procedures apply to all types of storage:

a. When the aircraft protective covers are not available, the areas concerned will be protected with barrier material (C16) and the wrap secured with tape (C107). The barrier material should be installed in such a manner as to prevent the accumulation of water on the surface of the cover. Provide drains if necessary. Replace aircraft protective covers or barrier material closures that are damaged or deteriorated.

b. Determine peak interior helicopter temperatures during hot weather conditions. Obtain temperature information from standard thermometers temporarily installed in the helicopter. Record interior temperatures at intervals during the hottest part of the day. Ventilate the helicopter if interior temperatures exceed 135°F (57°C). Provide forced ventilation if normal ventilation procedures are not adequate to prevent condensation and possible mildew and corrosion.

c. Inspect and treat the helicopter against corrosion. Inspection for corrosion includes close observation of areas where moisture does not evaporate rapidly. Evidence of corrosion will not be as prevalent on painted surfaces as it is on unpainted surfaces. Corrosion can attack metal through paint and will be evidenced by blisters or scaly appearance.

d. Inspect static ground wires, rotor block tiedown tethers, and mooring devices (ropes, cables, rods or eye) at regular intervals. Inspect tiedown devices immediately after the helicopter has been subjected to winds exceeding 40 mph. Replace ground wires, mooring devices or tiedown tethers which are deformed or deteriorated.

e. Inspect communication equipment for fungus or corrosion. Remove, clean, repair, package and identify communication equipment which is deteriorating. Stow the packaged equipment in the helicopter from which it was removed. Do not remove or package antennas.

f. If possible the helicopter will be stored in a hangar or under a shed roof; otherwise, it will be parked and moored according to procedures in chapter 1.

g. Enter the type of storage and the date aircraft was placed in storage in the aircraft log book.

SECTION II FLYABLE STORAGE

Table E-1. Premaintenance Requirements for Flyable Storage.

Condition	Requirements
Support Equipment	(S2) (S4) (S6) (S12)
Minimum Personnel Required	Two
Consumable Materials	(C15) (C16) (C51) (C67) (C107)

E-8. FLYABLE STORAGE.

E-9. Inspection — Prior to Flyable Storage. Helicopters in this storage category will be maintained in a serviceable condition. Perform daily inspection prior to placing the aircraft in storage. (Refer to TM 55-1520-214-PMS.) Be sure the date and type of storage is recorded in the aircraft log book.

E-10. Flyable Preservation — Power Train System. *a.* Check the power train system lubrication including sight plugs.

b. Service the power train system according to procedures in chapter 1. The engine preservation run-up completes this preservation of the main and tail rotor transmissions.

E-11. Flyable Preservation — Engine. Preserve the engine as follows:

a. Take every precaution to keep the engine and accessories clean. Keep the air intake duct, plenum chamber, and compressor inlet screens clean and free of any foreign materials. When external cleaning is necessary, refer to TM 55-1500-333-24.

CAUTION

Do not use contact preservatives of any kind either internally or externally on the compressor section.

b. Fill the engine oil tank as necessary to normal operating level with standard operating oil (C67). Start the engine, refer to TM 55-1520-214-10. If available, use external auxiliary power unit.

CAUTION

Do not exceed maximum temperatures and pressure in engine runup and ground run.

NOTE

This runup may be omitted if the helicopter was operated within the preceding 24 hour period.

c. Run at idle to ensure that the engine is operating satisfactorily. Accelerate to 100% N2 with collective full down. Operate engine until oil temperature reaches normal operating range and then idle engine for 2 minutes prior to shutting down.

NOTE

Ground runup, when preparing the engine, will complete the necessary preservation of the main and tail rotor transmissions.

d. Shut down engine (TM 55-1520-214-10).

e. Install engine inlet and exhaust protective covers. If engine covers are not available, seal the air inlet and exhaust openings with barrier material (C16) and secure material with tape (C107).

f. Record the date the engine was placed in flyable storage in the aircraft log book. Also mark records to indicate that a corrosion preventive concentrate has been added to the engine, main transmission, and tail rotor oil systems according to TB 55-9150-200-24. Indicate that flushing is not required during depreservation.

E-12. Flyable Preservation — Fuel System. Service fuel cells to normal capacity after each engine preservation run. Drain water from fuel cell before adding fuel.

NOTE

If fuel cells are filled to normal capacity it reduces fuel contamination by condensation. Maintenance officer will determine the interval of periodic water drainage.

E-13. Flyable Preparation — Airframe. *a.* Install pitot tube cover or, if the cover is not available, wrap the pitot tube with barrier material (C15) and secure wrap with tape (C107).

b. Install aircraft covers if available.

c. Open (pull out) all circuit breakers.

d. Close all doors and windows.

E-14. Maintenance — Preservation During Flyable Storage. *a.* Helicopters in flyable storage will be inspected according to local directives and those requirements outlined in paragraphs E-7 and E-9.

b. Perform a preventive maintenance daily inspection at least once every 7 days according to TM 55-1520-214-PMS.

c. Operate the engine at least once every 7 days according to steps *b* and *c* of paragraph E-11.

d. Install engine and exhaust covers according to step *e* of paragraph E-11.

Table E-2. Premaintenance Requirements for Depreservation after Flyable Storage.

Condition	Requirements
Support Equipment	(S2) (S4) (S6) (S12)
Minimum Personnel Required	Two
Consumable Materials	(C82) (C94)

E-15. Depreservation — After Flyable Storage. *a.* Remove protective covers and stow them in designated location in aircraft.

CAUTION

Tape residue on plastic surfaces must be removed with isopropyl alcohol (C8?).

b. Remove all barrier material and tape. Remove tape residue with solvent (C94).

c. Clean aircraft as necessary according to procedures in chapter 1.

d. Open all doors and ventilate helicopter.

e. Remove tiedowns and restraints, if applicable.

f. Perform preventive maintenance daily inspection according to TM 55-1520-214-PMS.

g. Record in aircraft log book the date the aircraft was prepared for service.

h. Remove static ground wire installed for storage.

E-16. Inspection — Flyable Post Storage. *a.* Clean aircraft as necessary according to procedures in chapter 1.

b. Check that all removed components have been reinstalled in the aircraft. Check the aircraft log book for a record of components that have been removed or disconnected. Check for subsequent installations or connections.

c. Check that related systems have been properly depreserved and serviced before any system or component operational check is performed.

d. Perform preventive maintenance daily inspection according to TM 55-1520-214-PMS and appropriate special inspections listed in chapter 1.

e. Record in aircraft log book the date the aircraft was prepared for service.

SECTION III SHORT TERM STORAGE

Table E-3. Premaintenance Requirements for Short Term Storage.

Condition	Requirements
Support Equipment: Main rotor blades extended	(S1) (S2) (S3) (S4) (S6) (S7) (S9) (S10) (S12)
Support Equipment: Main rotor blades folded	(S1) (S2) (S3) (S5) (S6) (S7) (S9) (S10) (S11) (S12)
Minimum Personnel Required	Two
Consumable Materials	(C15) (C16) (C22) (C23) (C30) (C51) (C61) (C67) (C68) (C92) (C107)

E-17. SHORT TERM STORAGE.

E-18. Inspection — Prior to Short Term Storage. The following inspection is required.

a. Ensure that all removed components along with attaching hardware are preserved and stowed within the helicopter in suitable containers.

b. Be sure that a record of all removed or disconnected parts is entered in the aircraft log book.

c. Check fuel and oil lines and hoses for leakage.

d. Record the date and type of storage in the log book.

E-19. Short Term Preservation — Power Train System.

CAUTION

The main and tail rotor hub assemblies contain teflon bearings which are sensitive to many types of petroleum compounds. Do not use any type of solvent to clean any components in these areas.

a. Clean the exposed metal surfaces of the power train system with a clean cloth dampened with a solution of mild soap (C92) and water. Throroughly dry the cleaned area with a dry, lint-free cloth or with dry, compressed air.

b. Check the power train lubrication sight plugs and perform any required service according to chapter 1.

c. Remove and clean the main rotor drive shaft. Preserve the main rotor drive shaft and the interior of the main rotor mast with corrosion inhibiting lubricant (C61). Reinstall main rotor driveshaft.

E-20. Short Term Preservation — Operable Engine.

NOTE

If the engine cannot be motored, no effort will be made to preserve the engine fuel system; however, comply with the provisions of a, b, j, l, and n below.

a. Exercise every precaution to keep the engine and its accessories clean. Keep the air intake ducts, plenum chambers, and compressor inlet screens clean and free of any foreign materials. Electrically (static) ground the helicopter. When external cleaning is required, refer to TM 55-1500-333-24.

CAUTION

Do not use contact preservatives of any kind either internally or externally on the compressor section.

b. Fill the engine oil tank to normal operating level with standard operating oil according to procedures in chapter 1.

CAUTION

Do not exceed maximum temperatures and pressures in engine runup and ground run.

NOTE

This runup may be omitted if the helicopter was operated within the past 24 hour period.

c. Start and operate engine (Refer to TM 55-1520-214-10). After idle has stabilized, accelerate engine to 100% N2 with collective full down. Operate engine until oil temperature reaches normal operating range. Idle engine for 2 minutes and then shut down engine (Refer to TM 55-1520-214-10).

NOTE

Ground runup, when preparing the engine, will complete the necessary preservation of the main and tail rotor transmissions.

WARNING

Be sure ignition system has been off for at least 5 minutes before removing igniter lead. To dissipate all energy stored in condenser, ground igniter lead to engine, using an insulated screwdriver.

d. Disconnect power input lead from ignition exciter, and igniter lead at igniter.

e. Allow the engine to cool sufficiently to prevent autoignition.

f. Close the fuel shutoff valve. Disconnect the fuel line at the inlet port of the engine fuel pump. Cap disconnected fuel line. Pull fuel pump circuit breaker.

g. Connect a source of lubricating oil (C68) to the inlet port of the engine fuel pump.

h. Move the throttle to the IDLE detent. Observing starter time limits, motor engine with the starter (use APU if available). Continue motoring until fuel-free oil flows from combustor burner drain valve hose.

i. Disconnect source of lubricating oil from the engine fuel pump and reconnect disconnected fuel line.

j. Seal vents in the main and tail rotor transmissions with barrier material (C16) and secure with tape (C107).

k. Reconnect igniter and exciter input leads. **TORQUE EXCITER LEAD TO 8 — 12 INCH-POUNDS, AND IGNITER LEAD TO 70 — 90 INCH-POUNDS.**

l. Cover air intake, exhaust, and all engine openings with barrier material (C16) and secure with tape (C107).

NOTE

Do not cover fuel and oil openings with tape only. The tape adhesive is soluble in petroleum compounds and can cause contamination.

m. Tag engine and cyclic stick with the following information: **ENGINE FUEL SYSTEM HAS BEEN PRESERVED WITH LUBRICATING OIL MIL-L-6081, GRADE 1010. NO FLUSHING REQUIRED PRIOR TO OPERATION.**

n. Record date and extent of engine preservation in the engine historical records. Also mark records to indicate that a corrosion preventive concentrate has been added to the engine, main transmission, and tail rotor oil systems according to TB 55-9150-200-24 and that flushing is not required during depreservation.

E-21. Short Term Preservation — Fuel System. Maintain fuel cells at the full level for the duration of the storage period. A full fuel cell prevents fuel contamination. Drain any water from fuel cells before adding fuel. Fill fuel cells according to procedures in chapter 1. Refer to paragraph E-36 for damaged fuel cell preservation procedure.

E-22. Short Term Preparation — Main Rotor Blades. The main rotor blades may be folded for convenience in storing the aircraft. Use main rotor blade covers (S5) and folding rack (S11) for folded storage. Use covers (S5) and blade tiedown socks (S7) for storage of blades in normal position. Refer to procedures in chapter 1.

E-23. Short Term Preservation — Electrical System.

WARNING

Extreme care should be taken to keep battery electrolyte from coming in contact with clothing, the skin or the eyes.

Set the battery switch on the control panel at OFF before working on the battery.

a. Remove battery and turn it in to battery shop for storage.

b. Thoroughly clean the battery supports and the exterior of the disconnect plug with a citric acid solution mixed 14 parts water to one part citric acid (C23). Follow by flushing with chromium triox (C22); then thoroughly flush with clean water and allow to dry.

c. Cover the disconnect plug with barrier material (C15) and secure to the airframe with tape (C107).

E-24. Short Term Preservation — Instruments. *a.* Install pitot tube cover (S6) or (S12). If cover is not available, protect pitot tube with barrier material (C16) and tape (C107).

b. Cover air static port with tape (C107).

c. Remove clock, apply condition tag and turn into supply.

E-25. Short Term Preservation — Avionics. *a.* Remove, attach condition tags, and return all installed headsets and microphones to supply.

b. Leave all other unclassified avionics equipment installed in the aircraft.

E-26. Short Term Preservation — Landing Gear. *a.* Place blocks or shoring under skid tubes at strut attach points to provide free air passage.

b. Clean the skid tubes and treat for corrosion according to TM 55-1500-204-25/1.

c. Areas of the tubes needing paint will be protected. If painting is not possible, use corrosion preventive compound (C30).

E-27. Short Term Preservation — Fuselage. *a.* Park and moor the aircraft according to procedures in chapter 1.

b. Close all doors.

c. Close and secure all inspection panels and covers.

d. Close all openings not yet covered with barrier material (C16) and secure with tape (C107).

e. Open (pull out) all circuit breakers.

E-28. Maintenance — Preservation During Short Term Storage. *a.* Helicopters in short term storage will be inspected at least once every 30 days according to local directives, and those requirements outlined in paragraphs E-7 and E-18.

b. If conditions change so that a helicopter prepared for short term storage must remain in storage for a longer period of time, represerve the helicopter according to section IV. **DO NOT RENEW SHORT TERM STORAGE.**

Table E-4. Premaintenance Requirements for Depreservation after Short Term Storage.

Condition	Requirements
Minimum Personnel Reqired	Two
Consumable Materials	(C51) (C67) (C82) (C92) (C94)

E-29. Depreservation — After Short Term Storage.

a. Airframe.

(1) Remove protective covers and stow them in designated location in the aircraft.

CAUTION

Tape residue on plastic surfaces must be removed with isopropyl alcohol (C82).

(2) Remove all barrier material and tape. Remove tape residue with solvent (C94).

(3) Open all doors and ventilate helicopter.

(4) Remove tiedowns and restraints, if applicable.

b. Landing Gear. Remove blocks from under skid gear.

c. Avionics. Reinstall all removed equipment.

d. Instruments. Obtain clock from supply and install in aircraft.

CAUTION

Be sure battery switch is set at OFF before installing battery.

e. Battery. Obtain battery from battery shop, install in aircraft; then secure the disconnect plug.

f. Fuel System. Check fuel cells for presence of water and drain if necessary. Fill fuel cells as necessary with operating fuel according to procedures in chapter 1. If fuel cell has been purged, proceed in the same manner.

g. Power Train System.

(1) Fill main and tail rotor transmissions as necessary with operating oil according to procedures in chapter 1.

(2) Clean the exposed metal surfaces of the power train system according to paragraph E-19.

E-30. Inspection — Short Term Post-Storage. *a.* Clean aircraft as necessary according to procedures in chapter 1.

b. Check that all removed components have been reinstalled in the aircraft. Check the aircraft log book for a record of components that have been removed or disconnected. Check for subsequent installation or connection.

c. Check that related systems have been properly depreserved and serviced before any system or component operational check is performed.

d. Perform preventive maintenance daily inspection according to TM 55-1520-214-PMS.

e. Remove static ground wire installed for storage.

f. Record in aircraft log book the date the aircraft was prepared for service.

SECTION IV INTERMEDIATE STORAGE

Table E-5. Premaintenance Requirements for Intermediate Storage.

Condition	Requirements
Support Equipment	(S1) (S2) (S3) (S4) (S6) (S12)
Minimum Personnel Required	Two
Consumable Materials	(C15) (C16) (C26) (C29) (C61) (C67) (C68) (C92) (C107)

E-31. INTERMEDIATE STORAGE.

E-32. Inspection — Prior to Intermediate Storage. Inspect the aircraft according to paragraph E-7 and as follows.

a. Be sure that all removed components and attaching hardware are preserved and either stowed in the helicopter or at a designated location as prescribed in the respective paragraph.

b. Main rotor blades will be placed in storage container and stored under cover.

c. Make certain that a record of all removed or disconnected components is entered in the aircraft log book.

d. Check fuel and oil lines and hoses for leakage.

e. Record the type and date of storage in the aircraft log book.

E-33. Intermediate Preservation — Power Train System. Preserve the power train system according to paragraph E-19.

E-34. Intermediate Preservation — Operable Engine.

NOTE

If the engine cannot be motored, no effort will be made to preserve the engine fuel system; however, comply with the provisions of steps l and n of paragraph E-20.

a. Exercise every precaution to keep the engine and its accessories clean. Keep the air intake ducts, plenum chambers, and compressor inlet screens clean and free of any foreign materials. Electrically (static) ground the helicopter. When external cleaning is required refer to TM 55-1500-333-24.

CAUTION

Do not use contact preservatives of any kind either internally or externally on the compressor system.

b. Fill the engine tank to normal operating level with standard operating oil according to procedures in chapter 1.

NOTE

This runup may be omitted if the helicopter was operated within the past 24 hour period.

c. Start and operate engine. (Refer to TM 55-1520-214-10.) After idle has stabilized, accelerate engine to 100% N2 with collective full down. Operate engine until oil temperature reaches normal operating range. Idle engine for an additional 2 minutes prior to shutting down. (Refer to TM 55-1520-214-10.)

NOTE

Ground runup, when preparing the engine, will complete the necessary preservation of the main and tail rotor transmissions.

d. Tag engine and cyclic stick with the following information: **LUBRICATION SYSTEM PRESERVED WITH OPERATING LUBRICANT; NO FLUSHING REQUIRED.**

e. Record extent of engine preservation in log book. Also mark records to indicate that a corrosion preventive concentrate has been added to the engine, main transmission, and tail rotor oil systems according to TB 55-9150-200-24 and that flushing is not required during depreservation.

E-35. Intermediate Preservation — Fuel System.

NOTE

Combat damaged fuel tanks will be preserved according to paragraph E-36.

The fuel system may be preserved by one of two methods. The method to be used will be determined by the availability of lubricating oil, equipment or inert gases.

WARNING

In the interest of safety for personnel and equipment, the following precautions must be observed while preparing aircraft fuel cells for storage: The aircraft and all equipment used in performing the operation must be properly grounded. This includes defueling equipment, work stands, purging equipment, and any powered or pneumatic devices. Work stands shall be equipped with a personnel static discharge plate of copper or zinc plate, this plate shall be affixed in such a position that personnel can contact it before coming in contact with the aircraft. High static electrical charges are created by the contact and separation of unlike substances, or by any sort of motion of persons or material. Such charges are a constant source of danger when generated in the presence of fuels or flammable vapors. Use only explosive proof work lights. Fuel cells should not be drained near the end of the working day and then allowed to stand empty over night. Such action could make a perfect set of conditions for producing explosive vapors. The fact is that the system is not completely empty. Residual fuel drains down the sides of the cell and forms puddles. During the night, fuel from these puddles evaporates into the air in the tank and if the critical fuel-air ratio develops, an explosion could be set off by a spark. The remedy is to avoid such a lapse of time between draining and purging.

Table E-6. Premaintenance Requirements for Primary Method of Fuel System Preservation.

Condition	Requirements
Minimum Personnel Reqired	Two
Consumable Materials	(C16) (C68) (C107)

a. Primary Method. If an adequate supply of lubricating oil, (C68) is available, preserve as follows.

(1) Disconnect igniter lead at igniter and place plastic cap over ceramic connector. Prevent lead from twisting while removing nut. Separate lead from igniter by pulling straight out without any rotational motion. Disconnect exciter input lead.

WARNING

Be sure ignition system has been off for at least 5 minutes before removing igniter lead. To dissipate all energy stored in condenser, ground igniter lead to engine, using an insulated screwdriver.

(2) Drain all fuel from the fuel system and close drains. (Refer to FM 10-68.)

NOTE

Reduction of the flashpoint in purging operations (3) below, will extend the useful life of the lubrication oil when an assembly line operation is set up. Do not use JP-4 as a substitute for JP-5.

(3) The flashpoint of the empty fuel cells may be reduced by pouring 5 gallons of JP-5 fuel into the fuel cells. Allow the JP-5 fuel to remain in the cell 10 to 15 minutes and then drain. When completely drained, close the drain valve and fill the cells with lubricating oil (C68). Allow oil to remain in fuel cells for at least an hour.

(4) Move the throttle to the IDLE STOP. Observing starter time limits, motor engine with the starter (use APU if available).

(5) Continue motoring until fuel-free oil flows from combustor drain line.

(6) Reconnect igniter and exciter input leads. **TORQUE EXCITER LEAD TO 8-12 INCH-POUNDS, and IGNITER LEAD TO 70-90 INCH-POUNDS.**

(7) Remove oil from fuel cells and save to flush other tanks.

(8) After 2 or 3 hours test the fuel cells with an explosion meter for the presence of fuel vapors. If an unsafe condition exists, discard the drained lubrication oil and flush with fresh oil until a safe reading is obtained.

(9) Cover air intake, exhaust and all other engine openings with barrier material (C16) and secure with tape (C107).

(10) Attach a tag to the cyclic stick and the fuel filler cap stating: THIS FUEL SYSTEM HAS BEEN PRESERVED WITH LUBRICATING OIL, MIL-L-6081, GRADE 1010. NO FLUSHING REQUIRED DURING DEPRESERVATION. FILL TANK WITH STANDARD OPERATING FUEL.

Table E-7. Premaintenance Requirements for Alternate Method of Fuel System Preservation.

Condition	Requirements
Minimum Personnel Required	Two
Consumable Materials	(C15) (C16) (C29) (C31) (C68) (C102) (C107)

b. Alternate Method. Use the following procedure to preserve the fuel system.

WARNING

Ensure that all applicable previous fuel system preservation precautionary data is adhered to before implementing the alternate method of system preservation.

(1) Drain all fuel from the fuel system and close drains. (Refer to FM 10-68.)

(2) Pour approximately 5 gallons of lubricating oil (C68) into the fuel tank.

(3) Preserve the fuel control system according to steps (1) through (6) of paragraph E-35*a*.

(4) Drain oil from fuel tanks.

(5) Purge fuel tanks with CO2 or nitrogen as follows:

WARNING

When using a fire extinguisher bottle as a source of CO2 for purging fuel tanks, regardless of the size of the bottle used, the fiber horn shall be removed, not only because it is too large for insertion into the tank filler neck, but also to avoid generating static electrical charges which can build up by gas moving rapidly through the horn. The nozzle as well as the bottle itself must be grounded

to the aircraft. The CO2 must be released slowly because the rapid passage of a gas through a hose can generate static electricity. In addition to this, a very rapid rate of discharge allows rapid expansion of the CO2 when it flows into a fuel cell. The expanding gas can lower the temperature to the point that will cause damage to the cell.

It is permissible to use nitrogen in place of the CO2 gas called out in any of the purging procedures. The same precautionary measures stated above will be observed.

NOTE

The size of the CO2 bottle to be used can be varied to meet existing conditions. The 15-pound size is handy to use. The total amount recommended is based on the quantity usually needed to purge a tank or tanks of the size under discussion: however, more may be needed to obtain a safe reading on the explosion meter.

CAUTION

During the following step use only dehydrated air.

(a) With the drains opened or removed and vents open, and the filler cap off, introduce into the filler neck a reduced pressure air hose supplying air through a 1/4-inch orifice at approximately 50 psi. Purge fuel tank for approximately one-half hour. Close or install all drains.

(b) Purge fuel cells with CO2 or nitrogen gas.

(c) Introduce into the fuel tank filler neck CO2 or nitrogen from a tank set to discharge at a rate of not more than 1.0 pound of purging gas per minute. Use not less than 3 pounds of CO2 or 5 pounds of nitrogen to purge fuel cells.

(d) After purging of the fuel tank has been completed, wait approximately 2 to 3 hours and then test main fuel tanks for the presence of dangerous fuel vapors with an explosion meter. If an unsafe condition exists use additional purging gas until a satisfactory test is made.

(e) Fog cells with a suitable spray gun and 1/2 pint of lubricating oil. Replace filler cap.

(6) Cover air intake, exhaust and all other engine openings with barrier material (C16) and secure with tape (C107).

(7) Attach a tag to the cyclic stick and fuel filler cap stating: THIS FUEL SYSTEM HAS BEEN PRESERVED WITH LUBRICATING OIL, MIL-L-6081, GRADE 1010. NO FLUSHING REQUIRED DURING DEPRESERVATION. FILL TANK WITH STANDARD OPERATING FUEL.

E-36. Intermediate Preservation — Damaged Fuel System. Fuel tanks that cannot be filled with lubricating oil because of leaks will be purged and preserved as follows:

a. Open fuel cells according to procedures in chapter 10 and ventilate tanks with rapidly moving dry air until fumes are below the danger level as shown by test with an explosion meter.

b. Preserve the tanks by spraying with lubricating oil (C68) through the access openings; coat the entire interior surface.

c. Preserve the engine fuel pump according to paragraph E-20.

E-37. Intermediate Preservation — Electrical System. Prepare the electrical system for storage according to paragraph E-23.

E-38. Intermediate Preservation — Main Rotor Blades. *a.* Remove the main rotor blades.

b. Color code blade attach pins, rotor blades, and main rotor hub upon removal of main rotor blades.

c. Coat the blade attach pins with a light coat of preservative compound (C29) and reinstall in the main rotor hub exactly as removed.

d. Coat blade attach pin holes in rotor blades and exposed metal surfaces with a light coat of corrosion preventive compound (C31).

e. Wrap butt end of blade and those areas of the blade which are mounted in the rotor container saddles with grade A barrier material (C15) and secure with tape (C107).

f. Pack main rotor blades in a plywood rotor blade crate (fig. 1), TM 1-OH6-S) or place on locally fabricated racks. Tag blades with serial number of aircraft.

g. Secure rotor head to prevent turning.

h. Stencil the serial number of the helicopter on the top, sides and ends of the main rotor blade crate (if used).

E-39. Intermediate Preservation — Instruments. Process instruments according to paragraph E-24.

E-40. Intermediate Preservation — Avionics. Process avionic equipment according to paragraph E-25.

E-41. Intermediate Preservation — Landing Gear. Process landing gear according to paragraph E-26.

E-42. Intermediate Preservation — Fuselage. *a.* Clean entire aircraft according to procedures in chapter 1.

b. Close all doors.

c. Cover all transparent surfaces with barrier material (C16). Pre-cut material to fit and secure around outside edges with tape (C102). Whenever possible, do not tape to transparent surface.

d. Close and secure all inspection panels and covers.

e. Close all openings not yet covered with barrier material (C16) and secure with tape (C107).

f. Park and moor the helicopter according to procedures in chapter 1.

E-43. Intermediate Preservation — Utility Equipment. *a.* Remove fire extinguisher, apply condition tag, and return to local supply.

b. Remove, apply condition tag, and return to supply such items as the first aid kit and other equipment subject to mildew and deterioration.

E-44. Maintenance — Preservation During Intermediate Storage. *a.* Helicopters in intermediate storage will be inspected every 30 days according to local directives and those requirements outlined in paragraph E-32. Record intermediate inspection in aircraft log book.

b. If conditions change so that a helicopter that was prepared for intermediate storage must remain in storage for a longer period of time, represerve the helicopter as required (Refer to paragraph E-5).

Table E-8. Premaintenance Requirements for Depreservation after Intermediate Storage.

Condition	Requirements
Minimum Personnel Required	Two
Consumable Materials	(C51) (C67) (C82) (C94)

E-45. Depreservation — After Intermediate Storage.

a. Airframe.

(1) Remove protective covers and stow them in designated location in the aircraft.

CAUTION

Tape residue on plastic surfaces must be removed with isopropyl alcohol (C82).

(2) Remove all barrier material and tape. Remove tape residue with solvent (C94).

(3) Open all doors and ventilate helicopter.

(4) Remove tiedowns and restraints, if applicable.

b. Landing Gear. Remove blocks from under skid gear.

c. Avionics. Reinstall all removed equipment.

d. Instruments.

(1) Remove pitot cover and store with other protective covers in aircraft.

(2) Remove tape from static vent in airspeed system. Remove tape residue with solvent (C94).

(3) Obtain clock from supply and install.

CAUTION

Be sure battery switch is set at OFF before installing battery.

e. Battery. Obtain battery from battery shop, install in helicopter and secure the disconnect plug.

f. Utility Equipment. Obtain from supply and install the fire extinguisher, first aid kit, and other equipment that was removed because it was subject to mildew and deterioration.

g. Main Rotor Blades.

CAUTION

Do not allow solvent to contact teflon bearings.

(1) Remove corrosion preventive compound from the main rotor blades and the blade attach pins with a clean cloth dampened with solvent (C94).

(2) Install main rotor blades.

h. Power Train System.

(1) Fill the main transmission and tail rotor transmission as necessary with operating oil according to procedures in chapter 1.

(2) Clean the exposed metal surfaces of the power train system according to chapter 1.

i. Fuel System. Check fuel cells for the presence of water and lubricating oil and drain if necessary. Service fuel cells according to procedures in chapter 1.

NOTE

No flushing of fuel cells or the engine fuel system is necessary.

E-46. Inspection — Intermediate Post-Storage. *a.* Clean aircraft as necessary according to procedures in chapter 1.

b. Check that all removed components have been reinstalled in the aircraft. Check the aircraft log book for a record of components that have been removed or disconnected. Check for subsequent installation or connection.

c. Check that related systems have been properly depreserved and serviced before any system or component operational check is performed.

d. Perform preventive maintenance daily inspection according to TM 55-1520-214-PMS and appropriate special inspections listed in chapter 1.

e. Remove static ground wire installed for storage.

f. Record in aircraft log book the date the aircraft was prepared for service.

APPENDIX F

WIRING DIAGRAMS AND LOAD CHARTS

F-1. WIRING DIAGRAMS AND LOAD CHARTS.

F-2. Contents — Wiring Diagrams and Load Charts. This appendix provides essential wiring information for maintenance of the OH-6A electrical systems and circuits. The information is presented as wire identification, equipment lists, an electrical load chart, and wiring diagrams.

F-3. Data — Wiring. *a. Wire Identification.* Wiring used in the aircraft is identified by a code of numbers and letters (fig. F-1) stamped with regular spacing along the wire length. The wire code is read as follows:

(1) In rare instances, a wire code may be duplicated. When this occurs, a prefix unit number 1 or 2 identifies the differences between circuits.

(2) The first letter (or letters) is the circuit designator indicating circuit function.

(3) The next number (or numbers) separates one leg of the circuit wiring from all others.

(4) The next letter indicates a segment of the wiring leg that ends, passes through, or branches at a connector or terminal block. Any continuation of the same leg then has the next higher (or lower) letter in sequence.

(5) The number (or numbers) after the segment letter is the standard AWG wire size.

(6) If the wire is the ground return segment of a circuit, the suffix N is added to indicate a ground connection.

b. Abbreviations. Where used on the wiring diagrams, abbreviations conform to Military Stand MIL-STD-12 and Army Regulation AR 310-50 except when the abbreviation depicts a marking actually found in the aircraft.

c. Symbols. The electrical symbols used in the wiring diagrams conform to ANSI 14.15 Part 1. (See fig. F-2.)

d. Equipment Lists. Tables F-1 and F-2 contain lists of the electrical equipment items used in the aircraft and shown in the wiring diagrams. Each major item of equipment listed is assigned an index number for use as a locator on figures F-3 and F-4.

NOTE

Each equipment list contains both the part reference designator and index item number. The wiring diagrams give item identification by reference designator only. The locator illustrations show only the item index number assigned on the equipment list.

e. Electrical Power Loading Chart. Table F-3 presents a listing of power input requirements and the electrical load imposed by each item of electrical equipment in the aircraft.

CAUTION

External auxiliary power units connected to the aircraft for engine starting purposes must have a power output capacity of not less than 375 amperes.

F-4. Diagrams — Wiring. *a. Arrangement.* An individual group of wiring diagrams is provided for each series of aircraft (i.e., series 1, 2, and 3). Within each aircraft series, separate wiring diagrams are provided for each major grouping of electrical equipment.

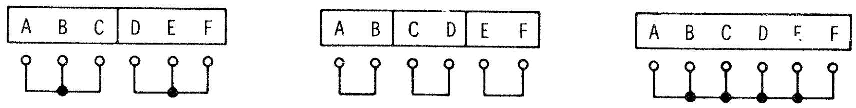
b. Wiring Diagram Index. The following wiring diagram index provides a listing that will enable you to locate any desired circuit diagram by means of the aircraft series number and electrical equipment or subsystem.

NOTE

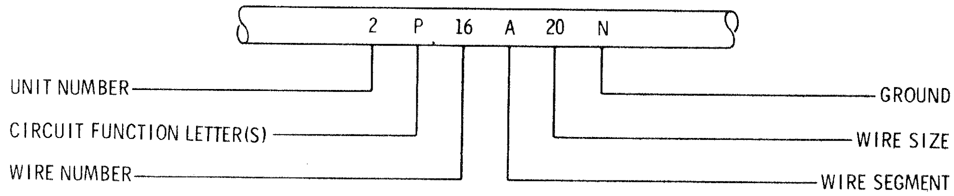
Single page illustrations, including wiring diagrams, are numbered F-1, F-2, etc. Larger fold-out type illustrations are numbered FO-1, FO-2, etc. and will be found at the rear of this manual. Foldout illustrations are arranged in figure number sequence, without page numbers.

Wiring Diagram Index

EXAMPLE: TERMINAL BLOCK BUSSING ARRANGEMENT



EXAMPLE: WIRE NUMBER 2P16A20N



CIRCUIT FUNCTION LETTER(S)	CIRCUIT
A	ARMAMENT
ARC-111	VHF RADIO
ARC-114	VHJ/FM RADIO (SLAE)
ARC-115	VHF RADIO (SLAE)
ARC-116	UHF RADIO (SLAE)
ARN-89	ADF RADIO (SLAE)
C	FLIGHT CONTROL
C6533(A, B, C)	INTERCOM AND COMMUNICATION SYSTEM CONTROL (A-PILOT; B-COPILOT; C-PASSENGER)
E	ENGINE INSTRUMENTS
EQ	AUDIO COMMON
F	FLIGHT INSTRUMENTS
KY	COMMUNICATIONS SECURITY SYSTEM
L	LIGHTING
P	DC POWER OR POWER CONTROL
Q	FUEL AND OIL
RF	FM RADIO
RN	ADF RADIO
RU	UHF RADIO
RV	VHF RADIO
RZ	INTERCOM
SX	TRANSPONDER SET
TN	DIRECTIONAL GYRO
U	ELECTRONIC WIRING COMMON LEADS

11-074C

Figure F-1. Wire Identification and Circuit Designators.

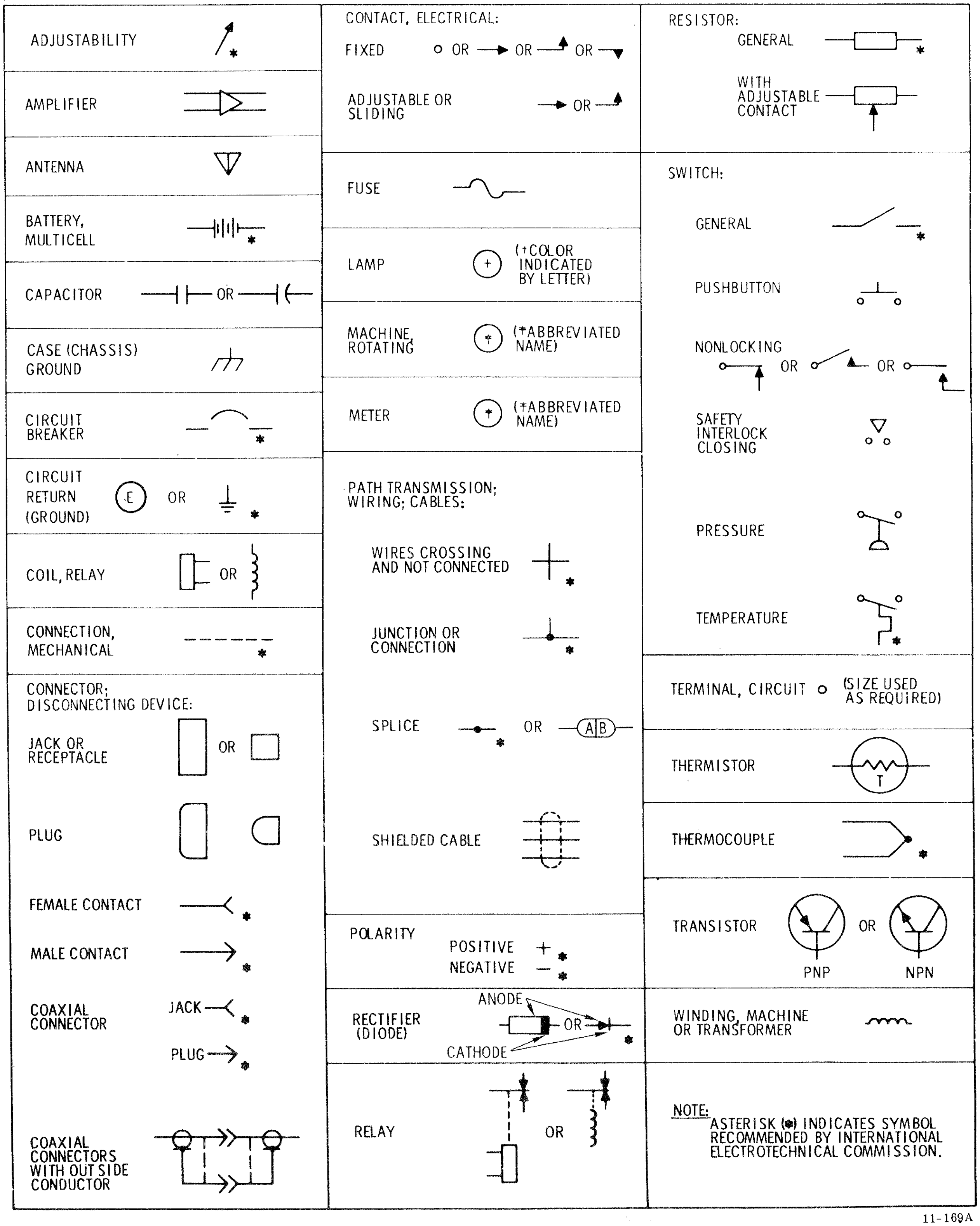


Figure F-2. Electrical Symbols.

Table F-1. Equipment List (Series 1 and 2 Aircraft).

Diagram reference designator	Item description	Figure F-3 item number	Part number
A2	Warning Unit, Engine Power Out (W/re-ign)	1	369A4614
AR1	Amplifier, audio	2	369A4563
B100	Actuator, lateral trim	3	369A7001
B101	Actuator, longitudinal trim	3	369A7001
B102	Actuator, N2 governor	4	369A7002
B250	Pump assembly, auxiliary fuel	5	369A8143
BT1	Battery	6	369A4556 369A4530
C110	Capacitor (Re-ign indicator)		GE CAT. No. 69F2361, 270 mfd, 50 vdc working volts
CB 1	Circuit breaker, anticollision lights 15 amps	7	MS24509-15
CB 3	Circuit breaker, auxiliary fuel, 7.5 amps	8	MS24509-7
CB 4	Circuit breaker, armament power, 50 amps	9	369A4590-13
CB 5	Circuit breaker, arm, 10 amps	10	369A4590-7
CB 6	Circuit breaker, hover light, 20 amps	11	369A4590-9
CB 7	Circuit breaker, warning lights, 7.5 amps	12	369A4590-5
CB 8	Circuit breaker, cyclic trim, 7.5 amps	12	369A4590-5
CB 9	Circuit breaker, instruments and inverter, 7.5 amps	12	369A4590-5
CB 10	Circuit breaker, instrument lights, 7.5 amps	12	369A4590-5
CB 11	Circuit breaker, N2 governor, 7.5 amps	12	369A4590-5
CB 12	Circuit breaker, UHF/VHF, 15 amps	13	369A4590-3
CB 13	Circuit breaker, position lights, 7.5 amps	12	369A4590-5
CB 14	Circuit breaker, AIC, radio, 7.5 amps	12	369A4590-5
CB 15	Circuit breaker, FM, radio 7.5 amps	12	369A4590-5
CB 16	Circuit breaker, ADF, radio	12	369A4590-5
CB 17	Circuit breaker, fuel pump, 7.5 amps	12	369A4590-5

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
CB 26	Circuit breaker, auto re-ign, 5 amps	14	MS25244-5
CB 27	Circuit breaker/switch, 7.5 amps		MS24509-7-1/2
CR1 thru CR21	Diode	15	IN2070
CR2	Diode, re-ign	16	IN3210
CR25	Diode	15	IN2070
DS 1	Warning unit, engine out	17	369A4540-369A4614(W/re-ign)
DS 3	Lamp, transmission oil temperature	18	MS25237-327
DS 4	Lamp, transmission oil pressure	18	MS25237-327
DS 5	Lamp, engine out	18	MS25237-327
DS 6	Lamp, fuel low	18	MS25237-327
DS 8	Lamp, fuel filter	18	MS25237-327
DS 9	Lamp, inverter off	18	MS25237-327
DS 10	Lamp, generator off	18	MS25237-327
DS 11	Lamp, oil chips	18	MS25237-327
DS 12	Lamp, arm	18	MS25237-327
DS 13	Lamp, safe	18	MS25237-327
DS 14	Lamp, armament power	18	MS25237-327
DS 16	Lamp, compass	18	MS25237-327
DS 20	Lamp, airspeed indicator	18	MS25237-327
DS 21	Lamp, clock	18	MS25237-327
DS 22	Lamp, gyro	18	MS25237-327
DS 23	Lamp, homing, heading , bearing indicator	18	MS25237-327
DS 24	Lamp, altimeter	18	MS25237-327
DS 25	Lamp, dual tachometer indicator	18	MS25237-327
DS 26	Lamp, torque indicator	18	MS25237-327
DS 27	Lamp, TOT indicator	18	MS25237-327
DS 28	Lamp, N1 tachometer indicator	18	MS25237-327
DS 29	Lamp, fuel quantity indicator	18	MS25237-327
DS 30	Lamp, instrument cluster	18	MS25237-327
DS 32 thru DS 34	Lamp, switch panel lighting	18	MS25237-327
DS 35 thru DS 38	Lamp, light control panel	18	MS25237-327

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
DS 39 thru DS 42	Lamp, circuit breaker panel	18	MS25237-327
DS 45, DS 46	Lamp, homing, heading, bearing indicator	18	MS25237-327
DS 47	Lamp, oil cooler bypass	18	MS25237-327
DS 100	Light, hover	19	369A4591
DS 101	Light, anticollision, lower	20	369A4562-5
DS 102	Flasher, anticollision light	21	369A4561-3
DS 103	Flasher, anticollision light	22	369A4561-5
DS 117	Indicator re-ign	23	369A4513
DS 200	Light, anticollision, upper	20	369A4562-5
DS 201	Lamp, utility light	24	M25231-313
DS 250	Light, position, left (red)	25	369A4589-3
DS 251	Light, position, right (green)	26	369A4589-5
DS 400	Light, position, tail (part of 369A4583)	27	
FL100	Filter, waveshape	28	AMI2049
G300	Starter generator	29	369A4550
G301	Ignition, exciter (furnished with engine)	30	GFP
HR1	Pitot tube, heated	75	369A6439
HS1	Knob, power rheostat		369A4546
J2	Receptacle, audio amplifier		M9S-LR-H19C
J4	Receptacle, instrument cluster		369A4598
J12	Receptacle, armament control panel assembly		XAC18SG0616X-V
J21	Receptacle, engine power out warning control unit (NAS J2)		MS112E14-18P(W/re-ign)
J100	Receptacle, external power		AN2552-A3A
J103	Receptacle, auxiliary fuel		VR2/4AB27-GC(alt: VR 2/ 4CP27CG)
J108	Receptacle, battery		10-87324-9S
J109	Receptacle, cyclic, copilot's		369A4598
J110	Receptacle, cyclic, copilot's		369A4598
J111	Receptacle, collective, copilot's		369A4596
J130	Receptacle, electrical		369A4598
J131	Receptacle, electrical		M95-95LRH19C
J132	Receptacle, electrical		M95-95LRH19C

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
J133	Receptacle, electrical		369A4596
J134	Receptacle, electrical		369A4596
J150	Receptacle, cyclic, pilot's		XAC20SG0516X-V
J151	Receptacle, cyclic, copilot's		XAC20SG0516X-V
J152	Receptacle, underfloor, RH		XAC18SG0516X-V
J153	Receptacle, underfloor, RH		XAC14SG0516X-V
J154	Receptacle, underfloor, LH		XAC26SG0516X-V
J155	Receptacle, light panel		XAC18SG0016-V
J156	Receptacle, switch panel		XAC18PG0516X-V
J157	Receptacle, circuit breaker panel		XAC34PG0616X-V
J158	Receptacle, instrument panel		XAC34SF0616X
J200	Receptacle, gun pod		SG3102E28-66S
J250	Receptacle, electrical fuel pump		PTO7-8-3P
J251	Receptacle, fuel quantity transmitter		369A4599-3
J303	Receptacle, engine		CA3100E22-19PF80
K1	Relay, hover light	31	369A4558
K101	Relay, fuel pump	32	MS25323-D2
K103	Relay, overvoltage	33	369A4557
K104	Relay, dimming	34	801-12B
K105	Relay, oil cooler bypass	35	369A4557
K300	Relay, reverse current	36	A700XA
K301	Relay, start	37	AN3371-2
L200	Valve, oil cooler bypass	38	369A4730
M1	Indicator, dual tachometer	39	369A4515
M2	Indicator, TOT	40	369A4521
M3	Indicator, N1 tachometer	41	369A4516
M4	Indicator, ammeter (part of 369A4230)	42	
M5	Indicator, fuel quantity	43	369A4525 (NCR fuel system)
M6	Indicator, oil temperature (part of 369A4230)	42	
M7	Indicator, oil pressure (part of 369A4230)	42	
M8	Indicator, fuel quantity	44	369A4519 (NCR fuel system) 369A4519-3 (CR fuel system)
MP1	Gyro, attitude	45	369A4502
MT200	Generator, tachometer, rotor	46	369A4517

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
MT250	Transmitter, fuel quantity	47	369A4526
MT251	Transmitter, fuel quantity	47	369A4245
MT300	Generator, tachometer, N1	48	369A4517
MT301	Generator, tachometer, N2	49	369A4517
MT302	Sender, oil pressure	50	369A4534
P1	Plug, dual tachometer indicator		369A4593
P2	Plug, audio amplifier		369A4595
P3	Plug, tachometer indicator, N1		369A4593
P4	Plug, instrument cluster		369A4594
P5	Plug, fuel quantity indicator		369A4592
P6	Plug, voltage regulator		MS3106A14S-6S
P7	Plug, attitude gyro		369A4592
P9	Plug, compass		AN3116-2
P10	Plug, caution light relay		8601
P11	Plug, caution light relay		8601
P12	Plug, armament control panel assembly		CA3106E20-27PF80 (Series 1) XAC18PG0506X-VL (Series 2)
P13	Plug, audio amplifier		M9P-LSH-19C
P21	Plug, engine power out unit (was P2)		PT06E-14-18S(SR)
P100	Plug, inverter, static		369A4554
P101	Plug, inverter, static		369A4554
P105	Plug, trim acuator, lateral		MS3106A10SL-3S
P106	Plug, trim actuator, longitudinal		MS3108A10SL-3S
P107	Plug, governor actuator, N2		MS3106A10SL-3S
P108	Plug, battery		SC06P24-9P
P109	Plug, cyclic, copilot's		369A4594
P110	Plug, cyclic, copilot's		369A4594
P111	Plug, collective, copilot's		369A4592
P114	Knife disconnect		32447
P115 thru P121	Knife disconnect		32445
P130	Plug, electrical		369A4594
P131	Plug, electrical		369A4595
P132	Plug, electrical		369A4595
P150	Plug, cyclic, copilot's		XAC20PG506X-VL
P151	Plug, cyclic, copilot's		XAC20PG0506X-VL

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
P152	Plug, under floor, RH		XAC18PG0506X-VL
P153	Plug, under floor, RH		XAC14PG0506X-VL
P154	Plug, under floor, LH		XAC26PG0506X-VL
P155	Plug, light panel		MRC18PGH-VL
P156	Plug, switch panel		XAC18SG0506X-VL
P157	Plug, circuit breaker panel		XAC34SG0506X-VL
P158	Plug, instrument panel		XAC34PC0406X
P200	Plug, tachometer, rotor		369A4599
P202	Plug, tachometer, rotor		369A4599
P203	Plug, oil cooler bypass		MS3106R10SL-4S
P204	Knife disconnect		32445
P205	Knife disconnect		32445
P250	Plug, electrical, fuel pump		PTOGEB-3S(SR)
P251	Plug, fuel quantity transmitter		369A4593
P300	Plug, engine chip detector, upper		MS3106A8S-15
P301	Plug, engine chip detector, lower		MS3106A8S-15
P303	Plug, engine		CA3106E22-19SF80
P305	Plug, tachometer indicator, N1		369A4699
P306	Plug, tachometer indicator, N2		369A4699
P400	Knife, disconnect		32445
P401	Knife, disconnect		32445
PS100	Inverter, static	51	369A4554
R1	Resistor		
R2	Resistor	52	0200C
R3	Resistor		0400B
R4 thru R15	Resistor, warning lights	53	RC32GF511J
R16	Resistor		3730
R17	Resistor		RC32GF470K
R18	Rheostat, switch panel	54	369A4541
R19	Rheostat, radio lights	54	369A4541
R20	Rheostat, flight instrument lights	55	369A4545
R21	Rheostat, engine instrument lights	54	369A4541
R22	Resistor (series 1)		AN5534-2
R200	Resistor (series 2)		AN5534-2
S1	Switch, push-to-test		369A4548

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
S2	Switch, chip detector selector		13ATI
S2	Switch, armament safe (used when removable armament panel not installed)		MS25125-C3
S3	Switch, armament power		369A4573
S4	Switch, position lights		MS35058-21
S5	Switch, fuel pump		MS35058-22
S7	Switch, armament selector		31303A
S10	Switch, N2 disable		MS27994-1
S11	Switch, generator off		MS35059-22
S13	Switch, inverter		MS35058-22
S14	Switch, external power		AN3230-1
S101	Switch, trigger		GFP
S102	Switch, pod elevation		GFP
S103	Switch, cyclic trim		GFP
S104	Switch, rpm trim		369A4580
S105	Switch, hover light		369A4581
S106	Switch, start		369A5482
S107	Switch, cyclic trim		GFP
S108	Switch, rpm trim		369A4580
S109	Switch, trigger		GFP
S110	Switch, pod elevation		GFP
S111	Switch, radio		GFP
S112	Switch, radio		GFP
S200	Switch, chip detector	56	369A5160
S201	Switch, chip detector	56	369A5160
S202	Switch, transmission oil temp	57	369A5186
S203	Switch, transmission oil pressure	58	369A5187
S204	Switch, oil tank float	59	369A4749
S205	Switch, filter pressure sensor		NSN5930-00-782-8634
S300	Switch, engine, upper (chip detector)	60	MS3106A8S-1S
S301	Switch, engine, lower (chip detector)		MS3106A8S-1S
S303	Switch, fuel filter (part of fuel filter)	61	
S350	Switch, fuel filter (part of fuel filter)	61	
S400	Switch, chip detector	56	369A5160
S500(KL)	Switch, keylock		44HY26254

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
SP5 thru SP10	Splice, knife		32445
SP101	Splice		MS25181-2
SP102	Splice		MS25181-3
SP103	Splice		35653
SP104	Splice		MS25181-2
SP105	Splice		35653
SP106 thru SP118	Splice		WM-22-2
SP119	Splice		MS25181-2
SP120	Splice		MS25181-1
SP121	Splice		WM-22-2
SP122	Splice		MS25181-2
SP123	Splice		MS25181-1
SP124	Splice		WM-22-2
TB1	Terminal block assembly	62	369A4219-3
TB2	Terminal block assembly	63	369A4539
TB5	Terminal block assembly	64	369A4551
TB200	Terminal stud		
TC300	Thermocouple, TOT (furnished with engine)	65	
TC301	Sender, oil temperature		369A4533
VR1	Voltage regulator	66	369A4652
W1	Bus bar		369A4571
W2	Bus bar		369A4255-3
XDS3	Light, signal legend type		369A4572-11
XDS4	Light, signal legend type		369A4572-9
XDS5	Light, signal legend type		369A4572-7
XDS6	Light, signal legend type		369A4572-15
XDS8	Light, signal legend type		369A4572-25
XDS9	Light, signal legend type		369A4572-21
XDS10	Light, signal legend type		369A4572-23
XDS11	Light, signal legend type		369A4572-13
XDS12	Light, signal legend type		369A4572-3
XDS13	Light, signal legend type		369A4572-5
XDS14	Light, armament power (part of arm power switch)		

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
XDS16	Light, compass (part of compass)		NSN6605-00-526-7256
XDS20	Fixture, instrument lighting		369A4577
XDS21	Fixture, instrument lighting		A89701R-4
XDS22	Fixture, instrument lighting		369A4577
XDS23	Post light		5952-8-ND-R-28
XDS24	Fixture, instrument lighting		369A4577
XDS25	Fixture, instrument lighting		369A4577
XDS26	Fixture, instrument lighting		369A4578
XDS27	Fixture, instrument lighting		369A4574
XDS28	Fixture, instrument lighting		369A4574
XDS29	Fixture, instrument lighting		369A4574
XDS30	Socket, cluster light installation		369A4230-31
XDS32	Panel lighting, circuit breaker		369A4209
XDS33	Panel lighting, circuit breaker		369A4209
XDS34	Panel lighting, circuit breaker		369A4209
XDS35	Panel lighting, switch	67	369A4565
XDS36	Panel lighting, switch	67	369A4565
XDS37	Panel lighting, switch	67	369A4565
XDS38	Panel lighting, light control	67	369A4565
XDS39	Panel lighting, light control	68	369A4564
XDS40	Panel lighting, light control	68	369A4564
XDS41	Panel lighting, light control	68	369A4564
XDS42	Panel lighting, light control	68	369A4564
XDS43	Socket, instrument cluster lights		369A4230-31
XDS44	Socket, instrument cluster lights		369A4230-31
XDS45	Post light		20115
XDS46	Post light		20115
XDS47	Light, signal legend type		369A4572-35
XDS48	Light, signal legend type		369A4572-37
XDS101	Bracket, anticollision lamp, lower	69	369A4283-5
XDS200	Bracket, anticollision lamp, upper	69	369A4283-3
XDS201	Light, utility	70	MS17245-5
XDS250	Light, position, LH	71	369A4589-3
XDS251	Light, position, RH	72	369A4589-5
XDS400	Light, position, tail	73	369A4588

Table F-1. Equipment List (Series 1 and 2 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-3 item number	Part number
XF1	Holder, fuse, 15 amp		HHJ-B
XK101	Socket, fuel pump relay		MS25324-4
Z1	Control panel assembly, armament	74	369A4511 (Series 1) 369A4511-11 (Series 2)

Table F-2. Equipment List (Series 3 Aircraft).

Diagram reference designator	Item description	Figure F-4 item number	Part number
A1	Sender, oil pressure	1	369A4534
A2	Control unit, warning, engine out	2	369A4640-3 369A4614 (W/re-ign)
A3	Panel assembly, switch	3	369A4665
A4	Panel assembly, circuit breaker	4	369A4667
A5	Control panel, armament	5	369A4611
A6	Instrument assembly	6	369A4330
B2	Pump assembly, fuel	7	369A8103
B100	Actuator, lateral trim	8	369A7001
B101	Actuator, longitudinal trim	8	369A7001
B102	Actuator, N2 governor	9	369A7002-3
BT1	Battery (Sonotone) Battery (Gulton)	10 10	369A4630-1 369A4630-2
C110	Capacitor (Re-ign indicator)		GE CAT. No. 69F2361, 270 mfd, 50 vdc working volts
CB1	Circuit breaker, collision lights	11	MS24509-10
CB4	Circuit breaker, armament power	12	MS25244-50
CB5	Circuit breaker, armament control	13	MS25244-10
CB6	Circuit breaker, arming/hover light	14	MS25244-15
CB7	Circuit breaker, annunciator panel	15	MS25244-5
CB8	Circuit breaker, cyclic trim	16	MS25244-7
CB9	Circuit breaker, instruments	15	MS25244-5
CB10	Circuit breaker, instrument lights	15	MS25244-5
CB11	Circuit breaker, N2 governor trim	15	MS25244-5
CB12	Circuit breaker, utility lights	15	MS25244-5
CB13	Circuit breaker, position lights	15	MS25244-5
CB14	Circuit breaker, oil fuel pump	16	MS25244-7
CB15	Circuit breaker, oil cooler and scav-enge air	16	MS25244-7
CB20	Circuit breaker, radio C-6533/ARC	15	MS25244-5
CB21	Circuit breaker, APX-72	13	MS25244-10
CB22	Circuit breaker, transponder test set	15	MS25244-5
CB23	Circuit breaker, radio C-6533/ARC	15	MS25244-5
CB24	Circuit breaker, ADF, radio	15	MS25244-5
CB25	Circuit breaker, UHF-AM, radio	15	MS25244-5
CB26	Circuit breaker, re-ign	15	MS25244-5

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
CB27	Circuit breaker/switch		MS24509-7-1/2
CB30	Circuit breaker, utility power	16	MS24509-7
CB31	Circuit breaker, generator field	13	MS25244-10
CB32	Circuit breaker, VHF-FM, radio No. 1	15	MS25244-5
CB33	Circuit breaker, VHF-FM, radio No. 2	15	MS2544-5
CB34	Circuit breaker, VHF-AM, radio	15	MS25244-5
CB36	Circuit breaker, TSEC/KY-28, control	15	MS25244-5
CB37	Circuit breaker, inverter	15	MS24509-5
CR2	Diode, re-ign	17	IN3210
DS1	Lamp, landing/hover light	18	MS25241-5481
DS2	Lamp, position, left	19	A7512-24
DS3	Lamp, position, right	19	A7512-24
DS4	Lamp, anticollision, upper	20	369A4562
DS5	Lamp, anticollision, lower	20	369A4562
DS6	Lamp, position, tail (part of 369A4588)	21	1683-32CP-28V
DS12	Lamp, gun not cleared	22	MS25237-327
DS13	Lamp, ammunition low	22	MS25237-327
DS14	Lamp, armed	22	MS25237-327
DS100	Lamp, compass	22	MS25237-327
DS101	Light assembly, master caution and warning (contains lamps P/N 7341)	23	369A4673
DS102	Lamp, torque indicator	22	MS25237-327
DS103	Lamp, airspeed indicator	22	MS25237-327
DS104	Lamp, attitude indicator	22	MS25237-327
DS105	Lamp, altimeter	22	MS25237-327
DS106	Lamp, TOT indicator	22	MS25237-327
DS107	Lamp, N2 tachometer indicator	22	MS25237-327
DS108	Lamp, homing, heading, bearing indicator	22	MS25237-327
DS109	Lamp, clock	22	MS25237-327
DS110	Lamp, N1 tachometer indicator	22	MS25237-327
DS111	Lamp, instrument cluster	22	MS25237-327
DS112	Lamp, instrument cluster	22	MS25237-327
DS113	Lamp, instrument cluster	22	MS25237-327

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
DS114	Lamp, fuel quantity indicator	22	MS25237-327
DS115	Lamp, homing, heading, bearing indicator	22	MS25237-327
DS116	Lamp, homing, heading, bearing indicator	22	MS25237-327
DS117	Indicator, re-ign	24	369A4513
DS118	Caution light assembly, annunciator (contains lamps P/N 7341)	25	369A4672
DS119	Control unit, dimmer	26	369A4655
DS120	Panel assembly, light control	27	369A4664
DS200	Lamp, light control panel	22	MS25237-327
DS201	Lamp, light control panel	22	MS25237-327
DS202	Lamp, light control panel	22	MS25237-327
DS203	Lamp, light control panel	22	MS25237-327
DS216	Lamp, utility light	28	MS25231-313
DS217	Flasher, anticollision light	29	369A4661
G300	Starter generator	30	369A4550
G301	Ignition exciter (furnished with engine)	31	GFP
HR1	Pitot tube, heated	88	369A6439
J4	Receptacle, instrument assembly		M7S-95LR-H19CS
J5	Receptacle, fuel quantity indicator		M4S-LR
J6	Receptacle, voltage regulator		MS3102E14S-6P
J7	Receptacle, attitude gyro		M4S-LRN
J8	Static jack	32	AN3117-1
J9	Receptacle, compass light (part of compass; mates with AN3116-2)		
J11	Receptacle, dual tachometer		126-216
J12	Receptacle, armament panel		XAC20SG-0616X-V
J13	Receptacle, circuit breaker panel		XAC42PG-0616XF-V
J14	Receptacle, master caution and warning assembly (contains contacts P/N 100-2620P)		MRAC14P-G7V
J15	Receptacle, annunciator panel (contains contacts P/N 100-2620P)		MRAC34P-J6
J16	Receptacle, edge lighted circuit breaker panel		MS90335-4
J18	Receptacle, edge lighted switch panel		MS90335-4
J19	Receptacle, fuel pump		PT07P8-3P

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
J21	Receptacle, engine power out warning control unit		MS112E14-18P (W/re-ign) MS3112E12-10P
J31	Receptacle, N1 tachometer indicator		M5S-LRN
J100	Receptacle, external power		AN2552A3A
J103	Receptacle, utility power (uses P/N VS4/16C6 hood)		VR2/4CP27-CC or VR2/ 4AB27-CC
J104	Receptacle, dimmer unit		MS3120E14-19P
J105	Receptacle, lateral trim actuator		97-3102A10SL-3P(115)
J106	Receptacle, longitudinal trim actuator		97-3102A10SL-3P(115)
J107	Receptacle, N2 governor actuator		97-3102A10SL-3P(115)
J108	Receptacle, battery		CA3102E24-9SB
J109	Receptacle, pilot's collective stick (contains MS17804-16-20 and -16-16 socket contacts)		XAC9SF-0016
J110	Receptacle, copilot's cyclic stick		XAC20SG-0516X-V
J111	Receptacle, copilot's collective stick (contains MS17804-16-20 and -16-16 socket contacts)		XAC9SF-0016
J115	Receptacle, light control panel		XAC20SG-0516X-V
J118	Receptacle, switch panel		XAC26PG-0516X-V
J122	Receptacle, armament panel lamps		MS90335-4
J130	Receptacle, pilot's cyclic stick		XAC20SG-0516X-V
J136	Receptacle, inverter		MS3110P8-3P
J152	Receptacle, inline disconnect to junction box		XAC14PG-0516X-V
J153	Receptacle, inline disconnect to engine compartment		XAC18PG-0516X-V
J154	Receptacle, inline disconnect to armament		XAC20PG-0516X-V
J200	Receptacle, armament (uses dust cover and chain MS25043-28C)		SG3102E28-66S
J202	Receptacle, rotor tachometer generator (part of MT200)		
J203	Receptacle, oil cooler bypass valve		MS3102E10SL-4P
J206	Receptacle, air bleed shutoff valve solenoid		MS3102R10SL-4P
J251	Receptacle, fuel quantity transmitter		369A4599-3
J300	Receptacle, engine chip detector (furnished with engine)		

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
J301	Receptacle, engine chip detector (furnished with engine)		
J303	Receptacle, engine harness		CA3100E22-19PF80
J305	Receptacle, N1 tachometer generator (part of MT300)		
J306	Receptacle, N2 tachometer generator (part of MT301)		
K2	Relay, fuel pump	33	MS25323-D2
K3	Relay, landing/hover light	34	369A4658
K103	Relay, overvoltage	35	BR26-S28
K105	Relay, oil cooler bypass	36	BR26-S28
K300	Relay, reverse current	37	A700-XA
K301	Relay, start	38	AN3371-2
K500	Relay, radio control		BR16S1200C5-26V
K501	Relay, radio control		BR16S1200C5-26V
K502	Relay, radio control		BR16S1200C5-26V
K503	Relay, radio control		BR16S1200C5-26V
K504	Relay, radio control		BR16S1200C5-26V
L101	Solenoid, scavenge air valve	39	369A8462
L200	Solenoid, oil cooler bypass valve	40	369A4730
M1	Indicator, dual tachometer	41	369A4515
M2	Indicator, TOT	42	369A4521
M3	Indicator, N1 tachometer	43	369A4516
M4	Indicator, ammeter	44	6473108
M6	Indicator, oil temperature	45	648876
M7	Indicator, oil pressure	46	6461245
M8	Indicator, fuel quantity	47	369A4519 (NCR fuel system) 369A4519-3 (CR fuel system)
MP1	Gyro, attitude	48	369A4503 or 369A4509
MT200	Generator, tachometer, rotor	49	369A4517
MT251	Transmitter, fuel quantity	50	369A4245-501
MT300	Generator, tachometer, N1	49	369A4517
MT301	Generator, tachometer, N2	49	369A4517
P4	Plug, instrument assembly		M7P-95LS-H19CS
P5	Plug, fuel quantity indicator		369A4592
P6	Plug, voltage regulator		MS3106E14S-6S
P7	Plug, attitude gyro		M4P-95LS-H10C

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
P9	Plug, compass light (part of FSN6605-526-7256)		AN3116-2
P11	Plug, dual tachometer		369A4593
P12	Plug, armament panel		XAC20PG-0506X-VL
P13	Plug, circuit breaker panel		XAC42SG-0506X-VL
P14	Plug, master caution and warning assembly (contains contacts P/N 100-2020S95)		MRAC-14S-G7
P15	Plug, annunciator panel (contains contacts P/N 100-2020S95)		MRAC-34S-JTC6H-8
P16	Plug, edgelighted circuit breaker panel		320-HG
P18	Plug, edgelighted switch panel		320-HG
P19	Plug, fuel pump		MS3116FB8-3S (alt: PT06E8-3S (SR) or PT06CE8-3S(SR)
P21	Plug, engine power out warning control unit		MS3126F12-10S PT06E-14-18S(SR)(W/re-ign)
P31	Plug, N1 tachometer indicator		369A4593
P104	Plug, dimmer unit		MS3126F14-19S
P105	Plug, lateral trim actuator		MS3106E10SL-3S
P106	Plug, longitudinal trim actuator		MS3106E10SL-3S
P107	Plug, N2 governor actuator		MS3108B10SL-3S
P108	Plug, battery		369A4647 (CA3108E24-9PB)
P109	Plug, pilot's collective stick (contains MS17803-16-20 and -16-16 pin contacts)		XAC9PC-0306X
P110	Plug, copilot's cyclic stick		XAC20PG-0506X-VL
P111	Plug, copilot's collective stick (contains MS17803-16-20 and -16-16 pin contacts		XAC9PC-0306X
P115	Plug, light control panel		XAC20PG-0506X-VL
P118	Plug, switch panel		XAC26SG-0506X-VL
P122	Plug, armament panel lamps		320-HG
P130	Plug, pilot's cyclic stick		XAC20PG-0506XF-VL
P136	Plug, inverter		MS3116F8-3S (alt: PT06E8-3S(SR) or PT06CE8-3S(SR)
P152	Plug, inline disconnect to junction box		XAC14SG-0506X-VL
P153	Plug, inline disconnect to engine compartment		XAC18SG-0506X-VL

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
P154	Plug, inline disconnect to armament		XAC20SG-0506X-VL
P200	Plug, armament		11697795 (CS3106A28-66P)
P202	Plug, rotor tachometer generator		369A4599
P203	Plug, oil cooler bypass valve		MS3106R10SL-4S
P206	Plug, air bleed shutoff valve solenoid		MS3106R10SL-4S
P251	Plug, fuel quantity transmitter (part of MT251)		369A4593
P300	Plug, engine chip detector		MS3106E8S-1S
P301	Plug, engine chip detector		MS3106E8S-1S
P303	Plug, engine harness		CA3106E22-19SF80
P305	Plug, N1 tachometer generator		369A4699
P306	Plug, N2 tachometer generator		369A4699
PS100	Inverter, static	51	369A4654
R1	Resistor, 120 ohm, 5W ± 5%	52	HL-6-06Z, 120Ω ± 5%
R2	Resistor, 25 ohm, 20W	53	HL-24-08Z, 25Ω ± 5%
R3	Resistor, 20 ohm, 50W	54	HL-55-08Z, 20Ω ± 5%
R16	Resistor, ammeter, 1Ω, 12W	55	HL-12-06Z, 1Ω ± 5%
R17	Resistor, 47Ω, 1W	56	RC32GF470K
R18	Potentiometer, switch panel lights, 10 KΩ	57	W4191-10K
R19	Potentiometer, radio lights, 5 KΩ	58	W0123-5K
R20	Potentiometer, flight instrument lights, 10 KΩ	59	57243
R21	Potentiometer engine instrument lights, 10 KΩ	57	W4191-10K
R22	Resistor, thermocouple	60	AN5534-2
RT1	Sender, engine oil temperature	61	369A4533
S1	Switch, scavenge air		MS35058-22
S2	Switch, arm/safe		MS25125-C3
S3	Switch, fuel pump		MS35058-22
S4	Switch, position lights		MS35058-21
S6	Switch, bright-dim bus (supplied with R20)		
S7	Switch, master selector		31303A
S10	Switch, N2 disable	62	MS27994-1
S11	Switch, generator		MS35059-22
S14	Switch, master		MS25106-21

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
S101	Switch, trigger, pilot's		GFP
S102	Switch, pod elevation, pilot's		GFP
S103	Switch, pilot's cyclic trim		GFP
S104	Switch, N2 governor control		369A4580
S105	Switch, landing/hover light		369A4581
S106	Switch, starter		369A4582
S107	Switch, copilot's cyclic trim		GFP
S108	Switch, N2 governor control		369A4580
S109	Switch, trigger, copilot's		GFP
S110	Switch, pod elevation, copilot's		GFP
S111	Switch, xmit/radio (A218-101076-00)		GFP
S112	Switch, xmit/radio (A218-101076-00)		GFP
S200	Switch, main xmsn chip detector, fwd	63	369A5160
S201	Switch, main xmsn chip detector, aft	63	369A5160
S202	Switch, transmission oil temperature	64	369A5186
S203	Switch, transmission oil pressure	65	369A5185
S204	Switch, oil tank float	66	369A4749-3
S205	Switch, bypass air	67	369A8402
S300	Switch, engine chip detector (upper)	68	GFP
S301	Switch, engine chip detector (lower)	68	GFP
S303	Switch, fuel filter	69	GFP
S400	Switch, T/R xmsn chip detector	63	369A5160
S500(KL)	Switch, keylock		44HY26254
SPA1	Splice, closed end		MS25181-1
SP1	Splice, closed end		MS25181-2
SP2	Splice, closed end		35115
SP5 thru SP10	Splice, knife		32445
SP112-A	Splice, knife		32445
SP112-B	Splice, knife		32445
SP113-A	Splice, knife		32445
SP113-B	Splice, knife		32445
SP114-A	Splice, knife		32447
SP114-B	Splice, knife		32447

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
SP116-A thru SP401-B	Splice, knife		32445
SQ1	Igniter plug (1322)		GFP
TB100	Terminal block assembly	70	369A4319
TB200	Terminal block assembly	71	369A4319-601
TB300	Terminal block assembly	72	369A4319-603
TB350	Terminal block assembly	73	369A4319-613
TB400	Terminal block assembly	74	369A4319-605
TB500	Terminal block assembly	75	369A4604
TB700	Terminal block assembly	76	369A4319-607
TB800	Terminal block assembly	77	369A4319-611
TC300	Thermocouple (furnished with engine)	78	GFP
VR1	Voltage regulator static	79	369A4652
W1	Bus, circuit breaker	80	369A4329-5
W2	Bus, circuit breaker	80	369A4329-5
W3	Bus, circuit breaker	80	369A4329-5
W4	Bus, circuit breaker	81	349A4329-3
XDS2	Light assembly, position, left	82	369A4589-3
XDS3	Light assembly, position, right	83	369A4589-5
XDS4	Light assembly, anticollision	84	369A4285-5
XDS5	Light assembly, anticollision	85	369A4285-3
XDS6	Light assembly, tail	86	369A4588
XDS12	Light assembly, gun not cleared		369A4572-31
XDS13	Light assembly, ammunition low		369A4572-33
XDS14	Light assembly, armed		369A4572-29
XDS102	Light fixture, torque indicator		369A4578
XDS103	Light fixture, airspeed indicator		369A4577
XDS104	Light fixture, attitude indicator		369A4577
XDS105	Light fixture, altimeter		369A4577
XDS106	Light fixture, TOT		369A4574
XDS107	Light fixture, N2/rotor tach		369A4577
XDS108	Light fixture, heading/bearing indicator		5952-8ND-R28
XDS109	Light fixture, clock		5952-8ND-R28
XDS110	Light fixture, N1 tach		369A4574

Table F-2. Equipment List (Series 3 Aircraft). (cont)

Diagram reference designator	Item description	Figure F-4 item number	Part number
XDS111	Light fixture, instrument subassy		T-1
XDS112	Light fixture, instrument subassy		T-1
XDS113	Light fixture, instrument subassy		T-1
XDS114	Light fixture, fuel quantity indicator		369A4574
XDS115	Light fixture, heading/bearing indicator		5952-8ND-R28
XDS116	Light fixture, heading/bearing indicator		5952-8ND-R28
XDS200	Socket, edge lighting, light control panel		MS25010C11B327
XDS201	Socket, edge lighting, light control panel		MS25010C11B327
XDS202	Socket, edge lighting, light control panel		MS25010C11B327
XDS203	Socket, edge lighting, light control panel		MS25010C11B327
XDS216	Light, utility	87	MS17245-5
XK2	Socket, fuel pump relay		MS25324-4
XK103	Socket, overvoltage relay		75VB8/1KA3-14
XK105	Socket, oil cooler bypass relay		75VB8/1KA3-14
XK500	Socket, radio control relay		75VB8/1KA3-14
XK501	Socket, radio control relay		75VB8/1KA3-14
XK502	Socket, radio control relay		75VB8/1KA3-14
XK503	Socket, radio control relay		75VB8/1KA3-14
XK504	Socket, radio control relay		75VB8/1KA3-14

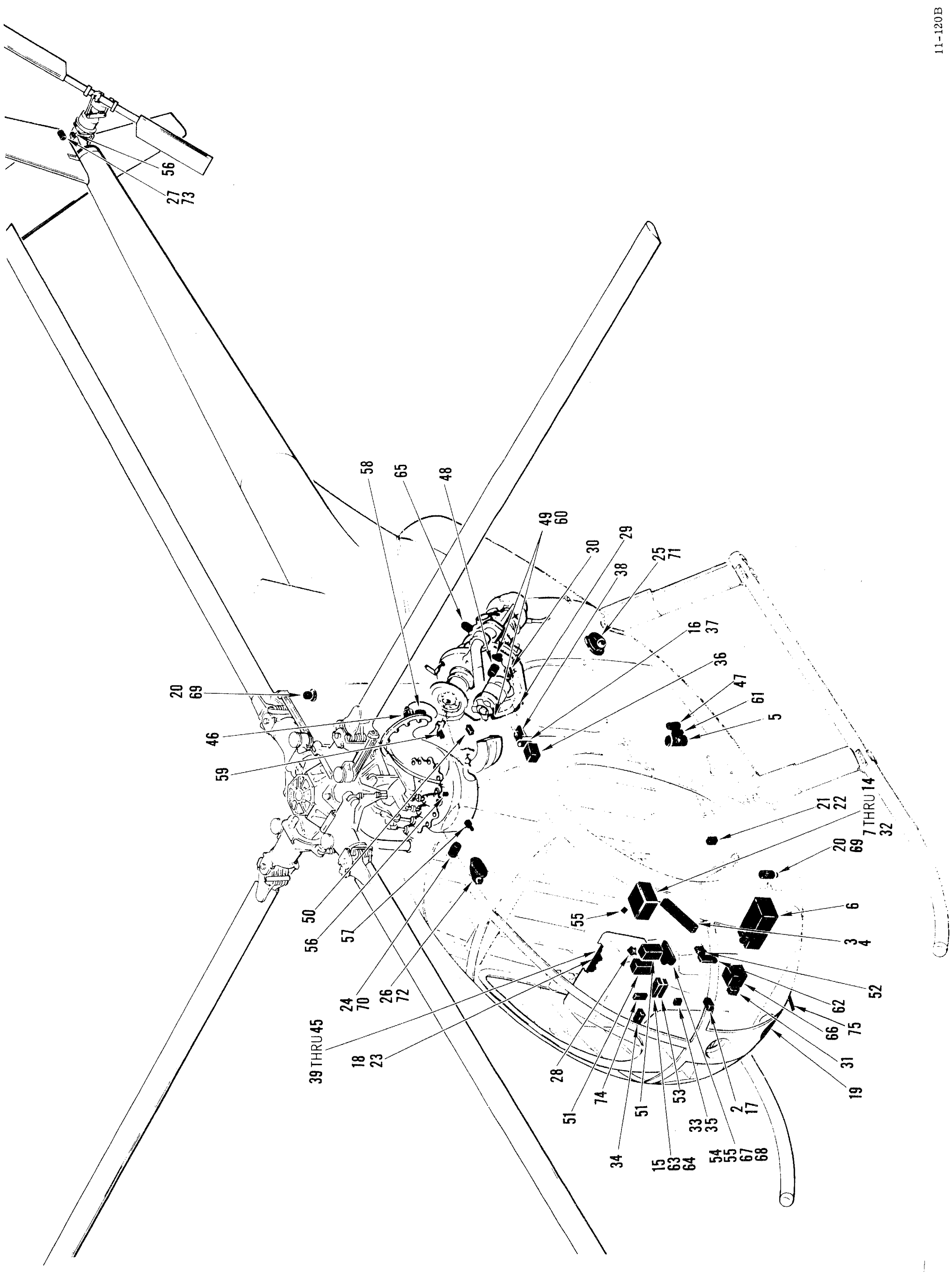


Figure F-3. Electrical Component Locator (Series 1 and 2 Aircraft).

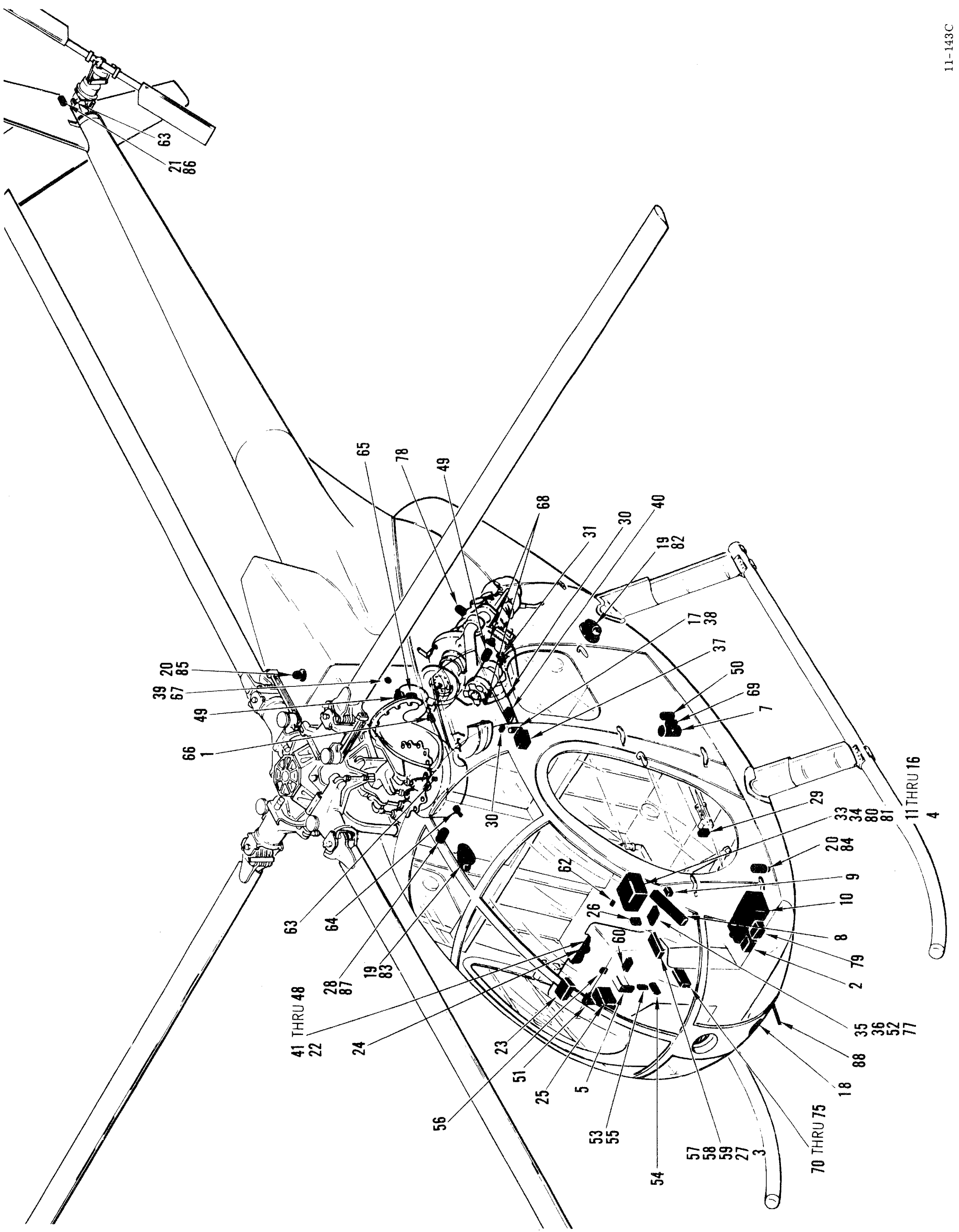


Figure F-4. Electrical Component Locator (Series 3 Aircraft).

Table F-3. Electrical Power Loading Chart.

Equipment	Part Number Series: 1 & 2	Part Number 3	Units per aircraft 1 & 2	Units per aircraft 3	Amperes per unit 1 & 2	Amperes per unit 3
Engine instruments						
Instrument pack	369A4531	369A4532	1	1	1.10	1.10
Ammeter						
Oil pressure indicator						
Oil temperature indicator						
Indicator, fuel quantity	369A4525	369A4519	1	1	1.10	1.10
Control unit, engine power out warning	369A4540	369A4540	1	1	0.5	0.5
Control unit, engine power out warning (w/re-ign)	369A4614	369A4614	1	1	0.65	0.65
Solenoid valve, oil cooler	369A4730	369A4730	1	1	2.0	2.0
Flight instruments						
Indicator, attitude gyro	369A4502	369A4502	1	1	See inverter	
Indicator, heading, bearing, homing	369A4504	369A4504	1	1	See inverter	
Compass set, directional gyro	AN/ASN-43	AN/ASN-43	1	1	See inverter	
Pitot tube, electrically heated	AN5813-1	AN5813-1	1	1	4.0	4.0
Ignition circuit						
Igniter	6843984 or 6877518	6843984 or 6877518	1	1		
						24.0 peak
Exciter	6870885 or 6877517	6870885 or 6877517	1	1		
Engine controls						
Actuator, RPM governor trim (N2)	369A7002	369A7002	1	1	1.2	1.2
Relay, starter	AN3371-2	AN3371-2	1	1	4.5	4.5
Engine starter-generator	369A4550	369A4550	1	1	350.0 peak	
Lighting						
Lamp, anticollision	369A4562	369A4562	2	2	7.5	7.5
Light, utility	MS25231-313	MS25231-313	1	1	0.2	0.2
Light, instrument panel	MS25237-327	MS25237-327	29	38	0.04	0.04
Light, hover	MS254241-4581	MS254241-4581	1	1	16.0	16.0
Relay, landing light	369A4658	369A4558	1	1	0.1	0.35
Light position, LH, red	369A4589-3	369A4589-3	1	1	0.8	0.8
Light, position, RH, green	369A4589-5	369A4589-5	1	1	0.8	0.8
Light, tail	369A4588	369A4588	1	1	1.02	1.02

Table F-3. Electrical Power Loading Chart. (cont)

Equipment	Part Number Series: 1 & 2	Part Number 3	Units per aircraft 1 & 2	Units per aircraft 3	Amperes per unit 1 & 2	Amperes per unit 3
Light, main transmission oil temperature	MS25237-327	MS25237-327	2	2	0.04	0.04
Light, main transmission oil pressure	MS25237-327	MS25237-327	2	2	0.04	0.04
Light, chip detector warning	MS25237-327	MS25237-327	2	2	0.04	0.04
Light, fuel pressure warning	MS25237-327	MS25237-327	2	2	0.04	0.04
Light, fuel quantity warning	MS25237-327	MS25237-327	2	2	0.04	0.04
Light, engine power out warning	MS35237-327		2		0.04	
Lights, caution and warning		369A4672		1		0.96
Lights, master caution and warning		369A4673		1		0.32
Indicator, auto re-ignition and reset	369A4513	369A4513	1	1	0.08	0.08
Power						
Inverter, instrument (No. 1)	369A4554		1		1.9	
Inverter, instrument (No. 2) Standby	369A4554		1		1.9	
Inverter, instrument		369A4654		1		0.76
Relay, overvoltage	369A4557	369A4557	1	1	0.2	0.2
Relay, reverse current	A700XA	A700XA	1	1	0.5	0.5
Auxiliary power unit requirements			1	1	375 min	
Fuel and oil						
Starting fuel pump	369A8143	369A8143	1	1	4.0	4.0
Armament						
Light, indicator	MS25237-327	MS25237-327	4	4	0.04	0.04
Drive motor			1	1	40.0 low rate max 75.0 high rate max 78.0 intermittent	
Gun	XM-27	XM-27	1	1	40.0	40.0
Flight control						
Motor, cyclic trim	369A7001	369A7001	2	2	1.0	1.0
Radio communication and navigation						
Interphone	C-1611/AIC	C-6533/AIC	2	3	0.2	0.18
VHF FM Liaison radio set	AN/ARC-54		1		3.8 rcv 7.4 xmt	

Table F-3. Electrical Power Loading Chart. (cont)

Equipment	Part Number Series: 1 & 2	Part Number 3	Units per aircraft 1 & 2	Units per aircraft 3	Amperes per unit 1 & 2	Amperes per unit 3
VHF AM radio set		AN/ARC-115		1		0.72 rcv
						3.35 xmt
VHF AM Liaison radio set alternate	AN/ARC-111		1		3.2 rcv	
					5.0 xmt	
VHF Communications set 1		AN/ARC-114		1		0.75 rcv
						3.35 xmt
UHF Command radio set	AN/ARC-51BX	AN/ARC-51BX	1	1	5.9 receive	
					11.1 transmit	
					10.6 switching	
UFH AM radio set		AN/ARC-116		1		0.75 rcv
						3.35 xmt
ADF	AN/ARN-83	AN/ARN-89	1	1	2.0	1.43
IFF Transponder set	AN/APX-72	AN/APX-72	1	1	3.61	3.61
Computer kit	1A/TSEC	1A/TSEC	1	1	1.07	1.07
Transponder test set		TS-1843APX		1		0.72
Communications security set		TSEC/KY 28		1		0.80
Control, indicator assembly		C-8157		3		0.04

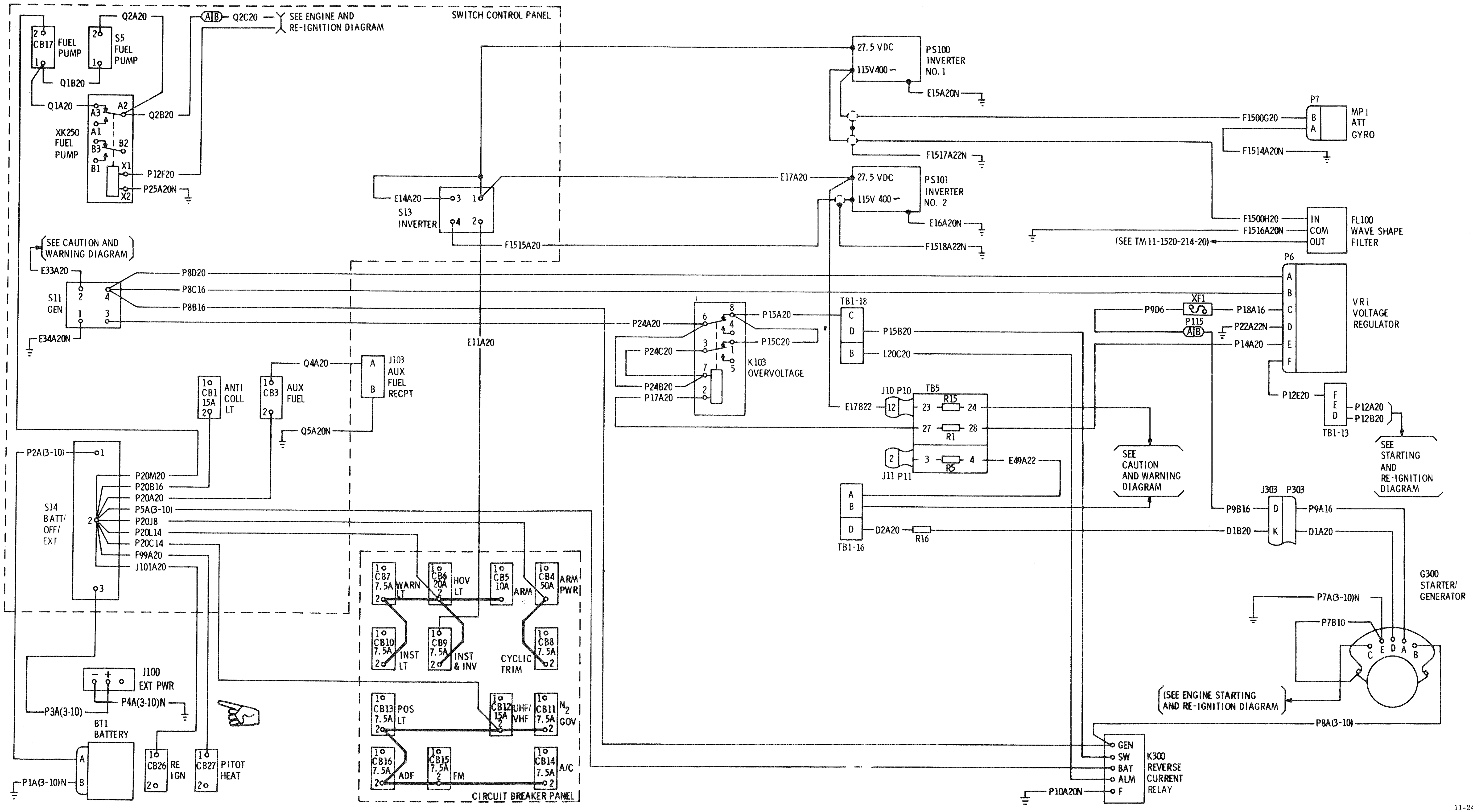


Figure FO-1. Series 1 Aircraft Power Supply Wiring Diagram.

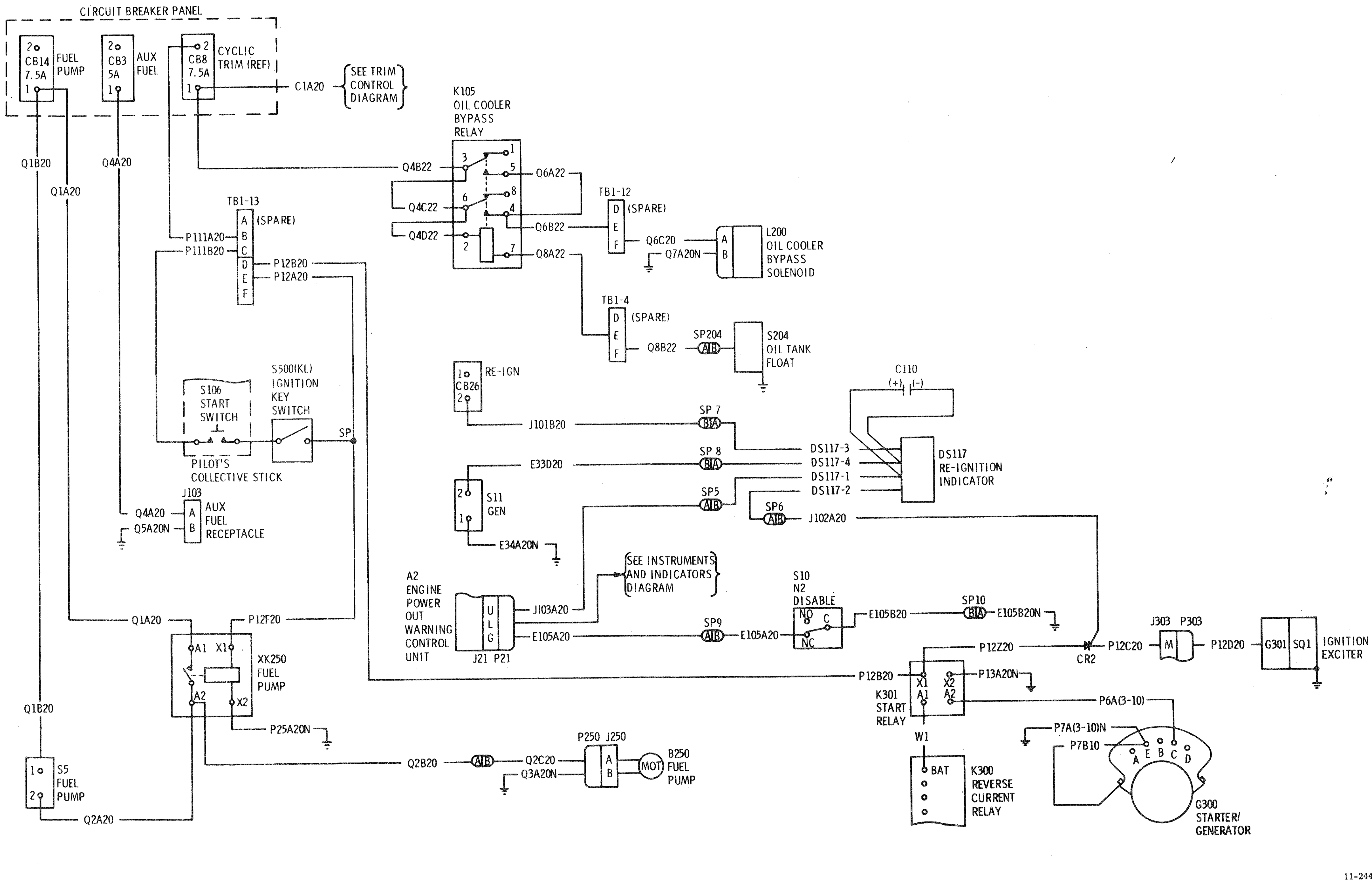


Figure FO-2. Series 1 Aircraft Engine Starting and Re-ignition Wiring Diagram.

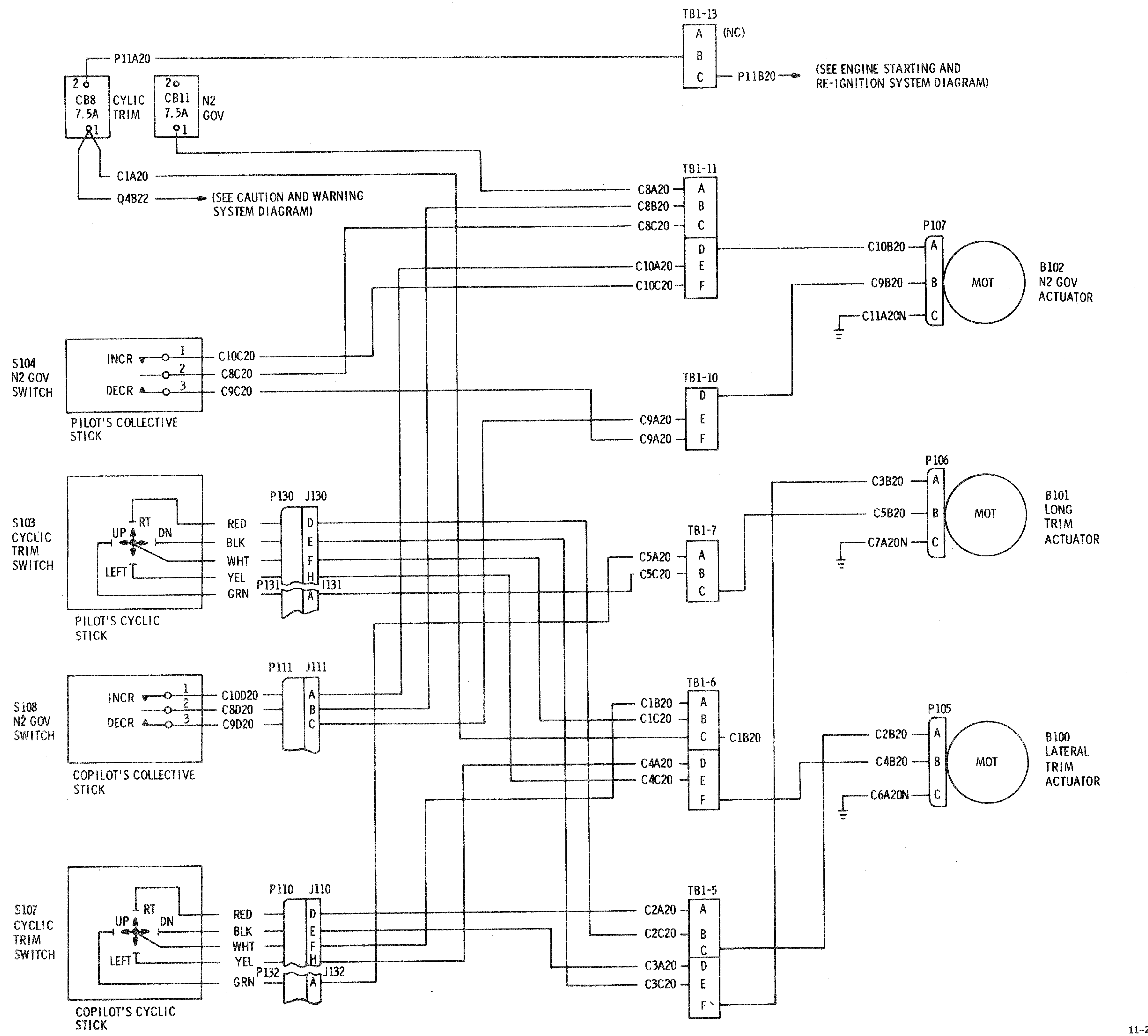


Figure FO-3. Series 1 Aircraft Trim Controls Wiring Diagram.

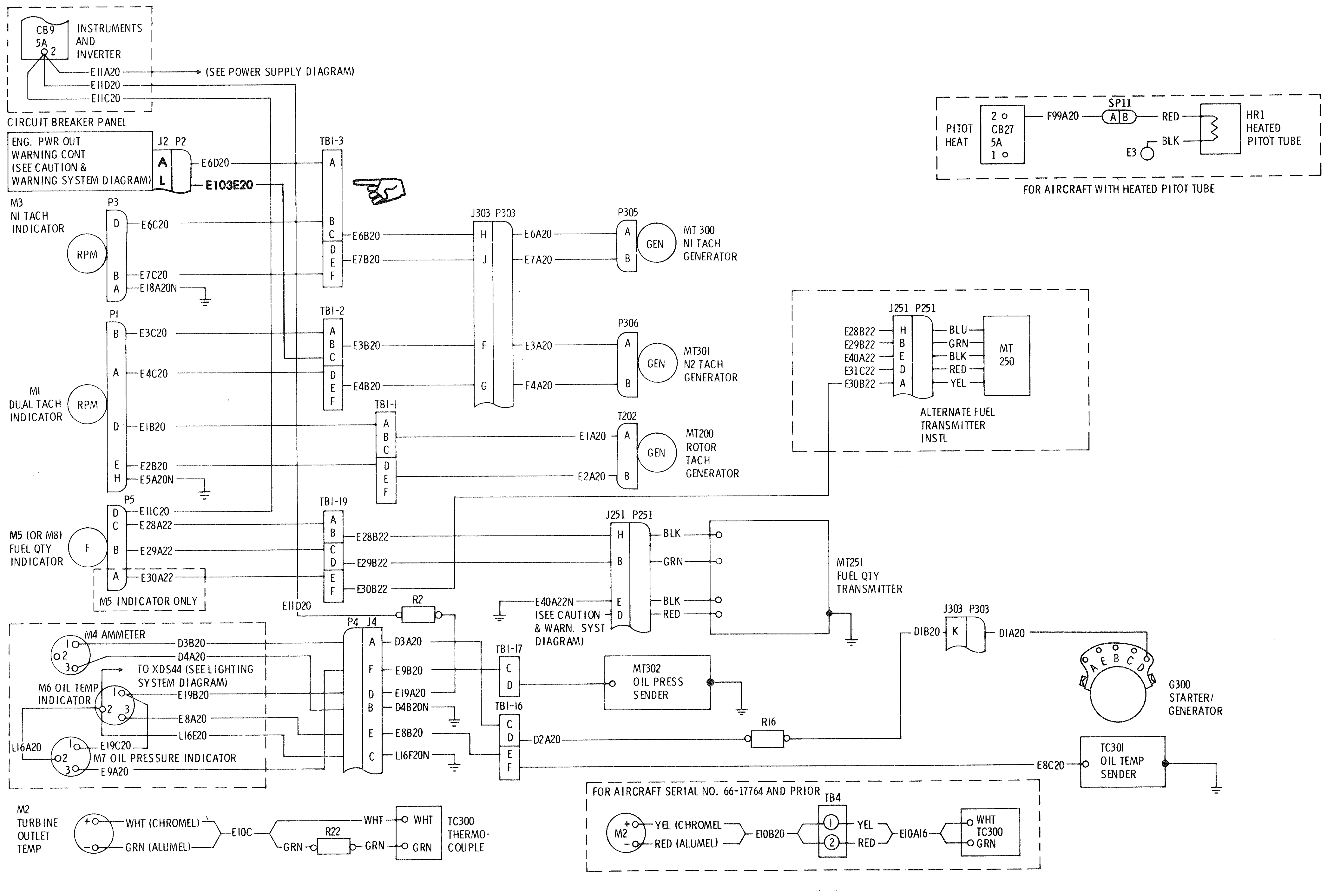


Figure FO-4. Series 1 Aircraft Instruments and Indicators Wiring Diagram.

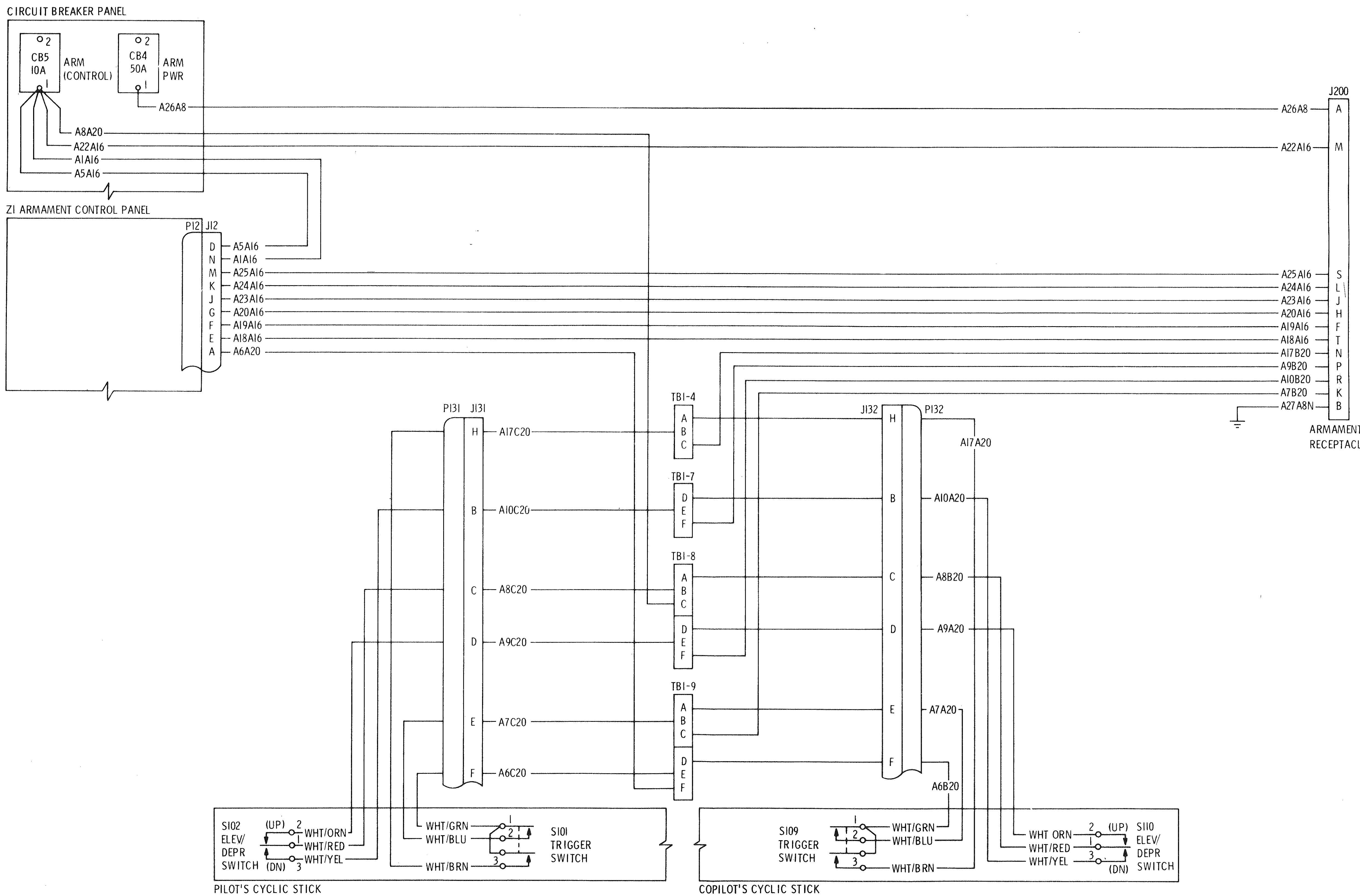


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Figure FO-5. Series 1 Aircraft Armament Control System Wiring Diagram.

NOTES:
1. USED ONLY ON AIRCRAFT WITH ENGINE BARRIER AIR FILTER INSTALLED.
2. USED ONLY ON AIRCRAFT WITH DUAL-INVERTER SYSTEM.

11-255

Figure FO-6. Series 1 Aircraft Caution and Warning System Wiring Diagram. Change 21

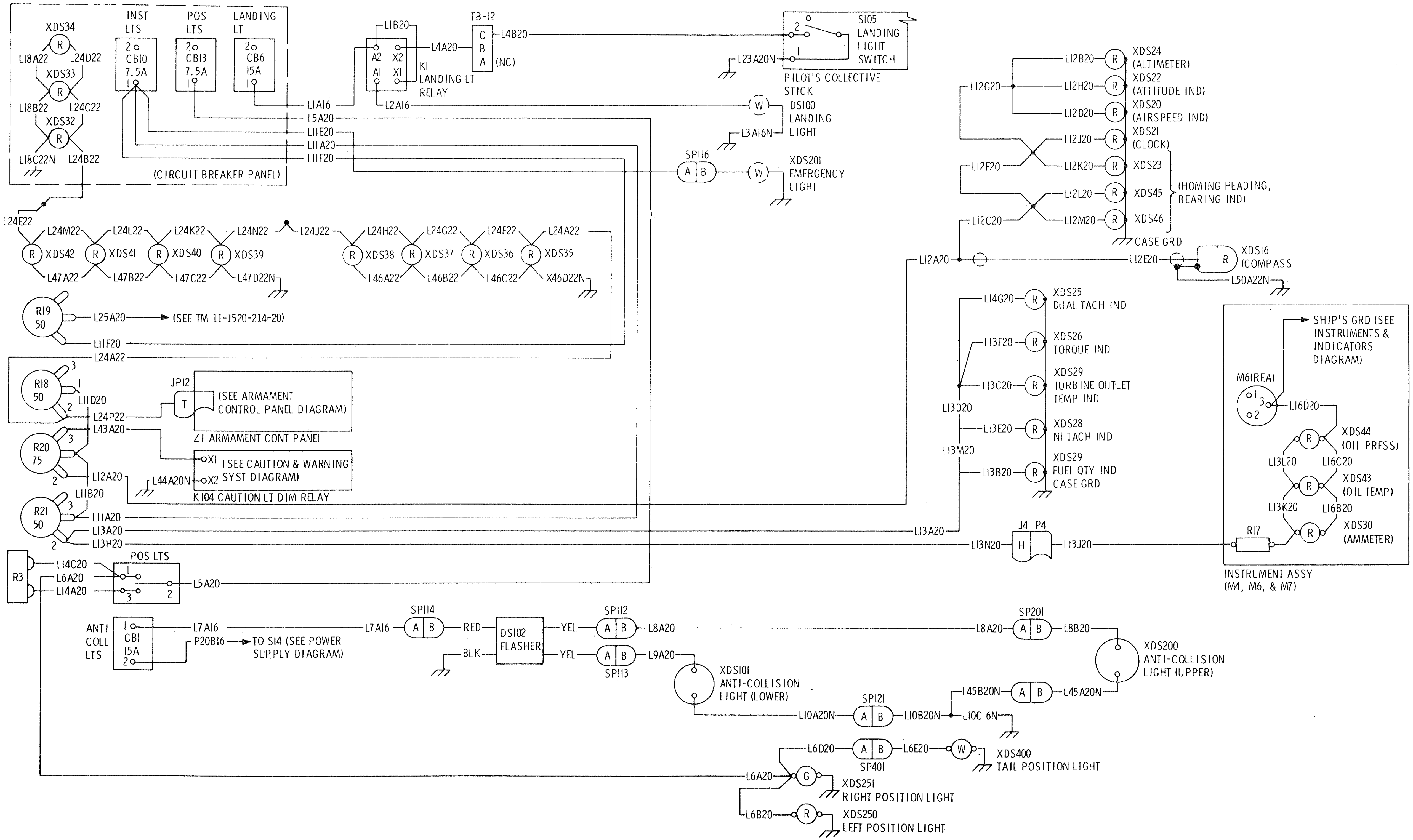

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Figure FO-7. Series 1 Aircraft Lighting Systems Wiring Diagram.

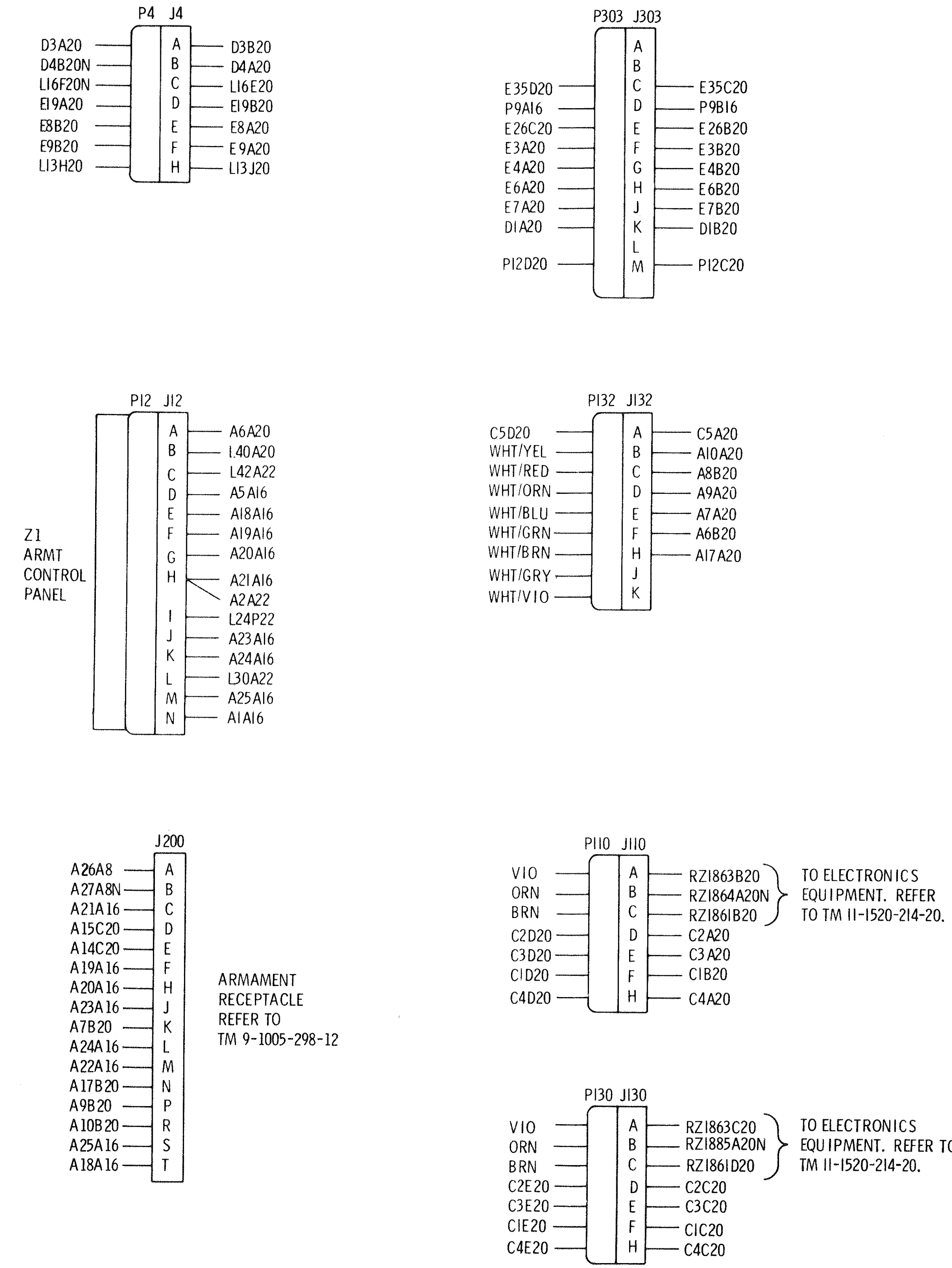

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Figure FO-8. Series 1 Aircraft Major Connectors Wiring Diagram.

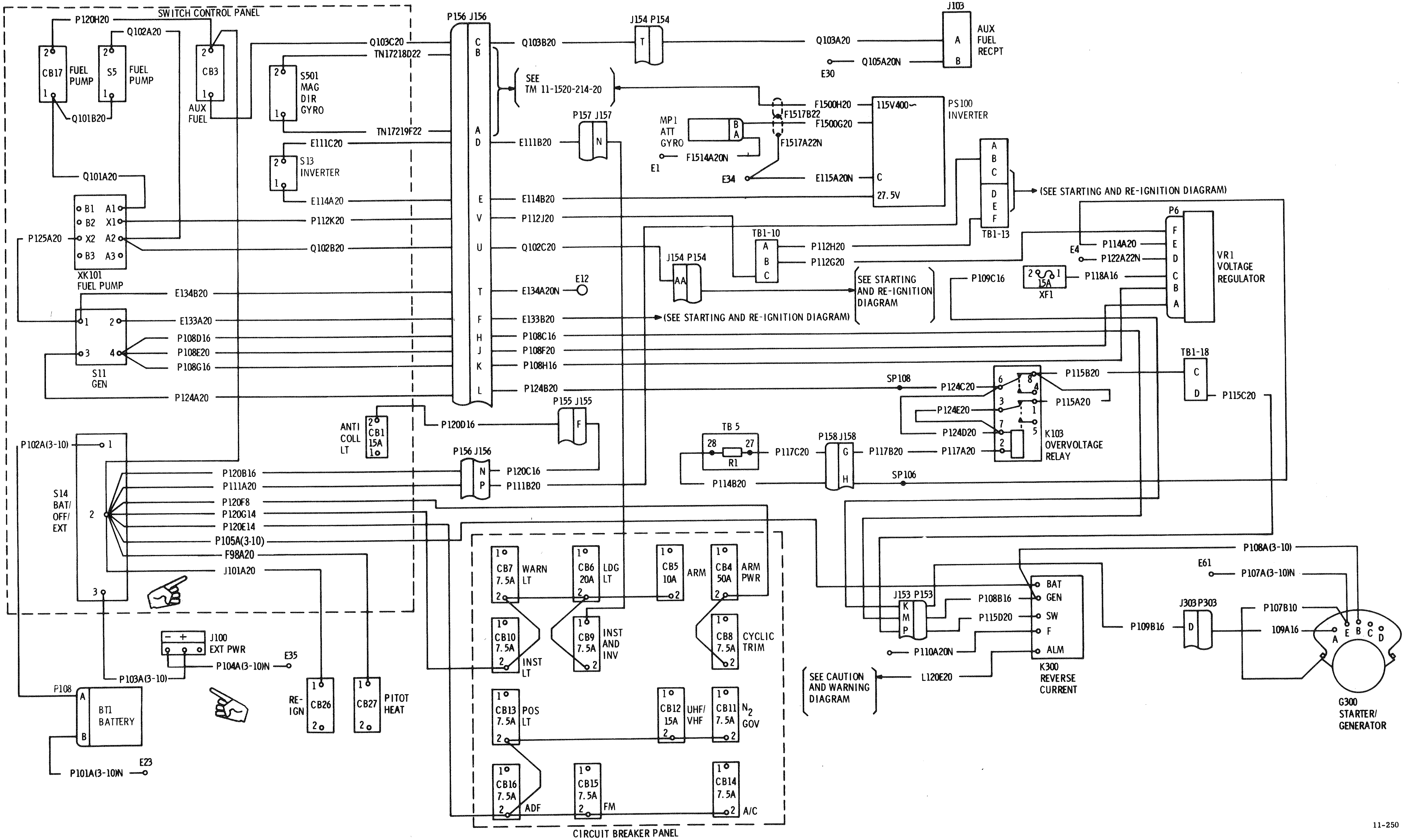


Figure FO-9. Series 2 Aircraft Power Supply Wiring Diagram.

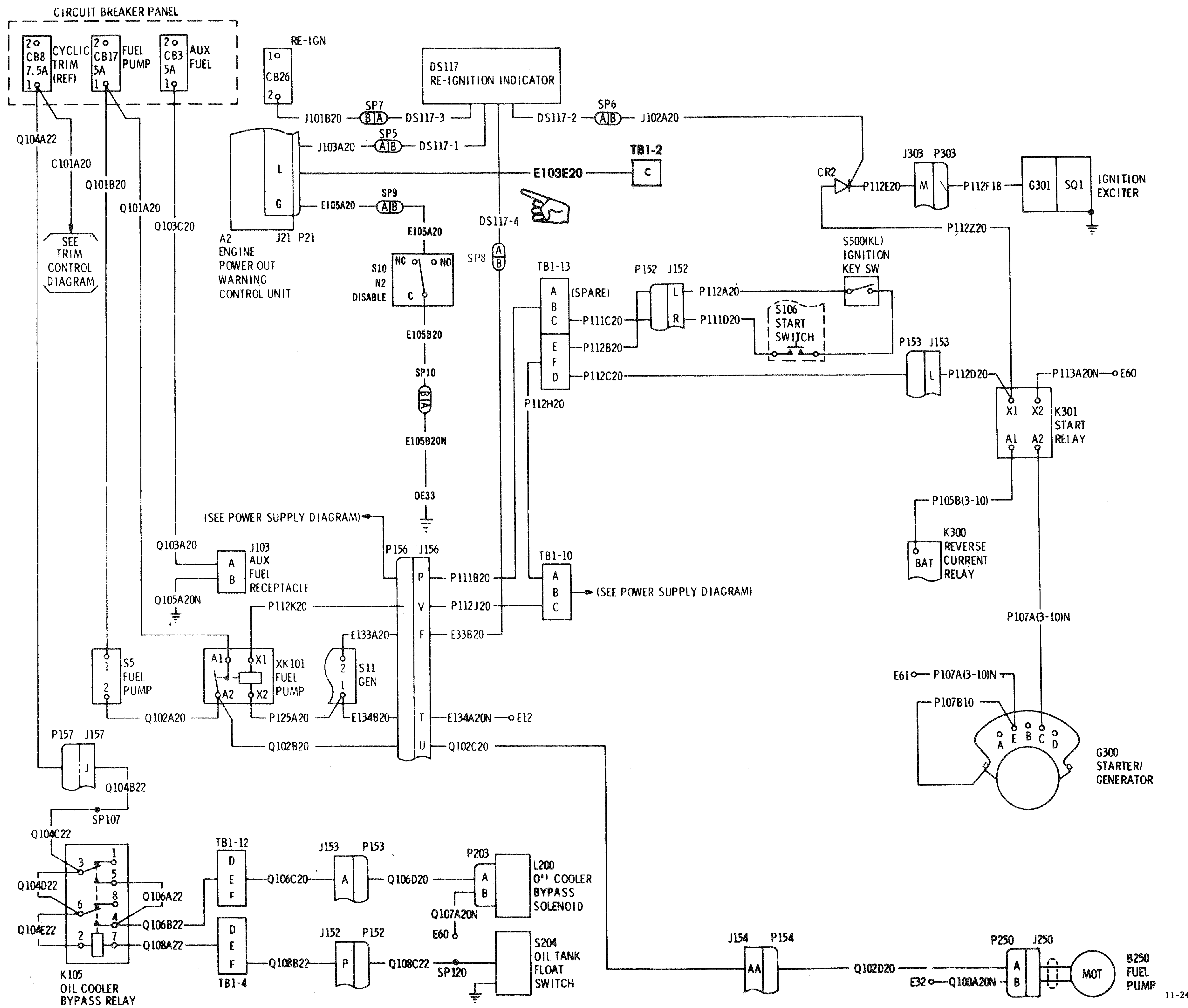


Figure FO-10. Series 2 Aircraft Engine Starting and Re-ignition Wiring Diagram.

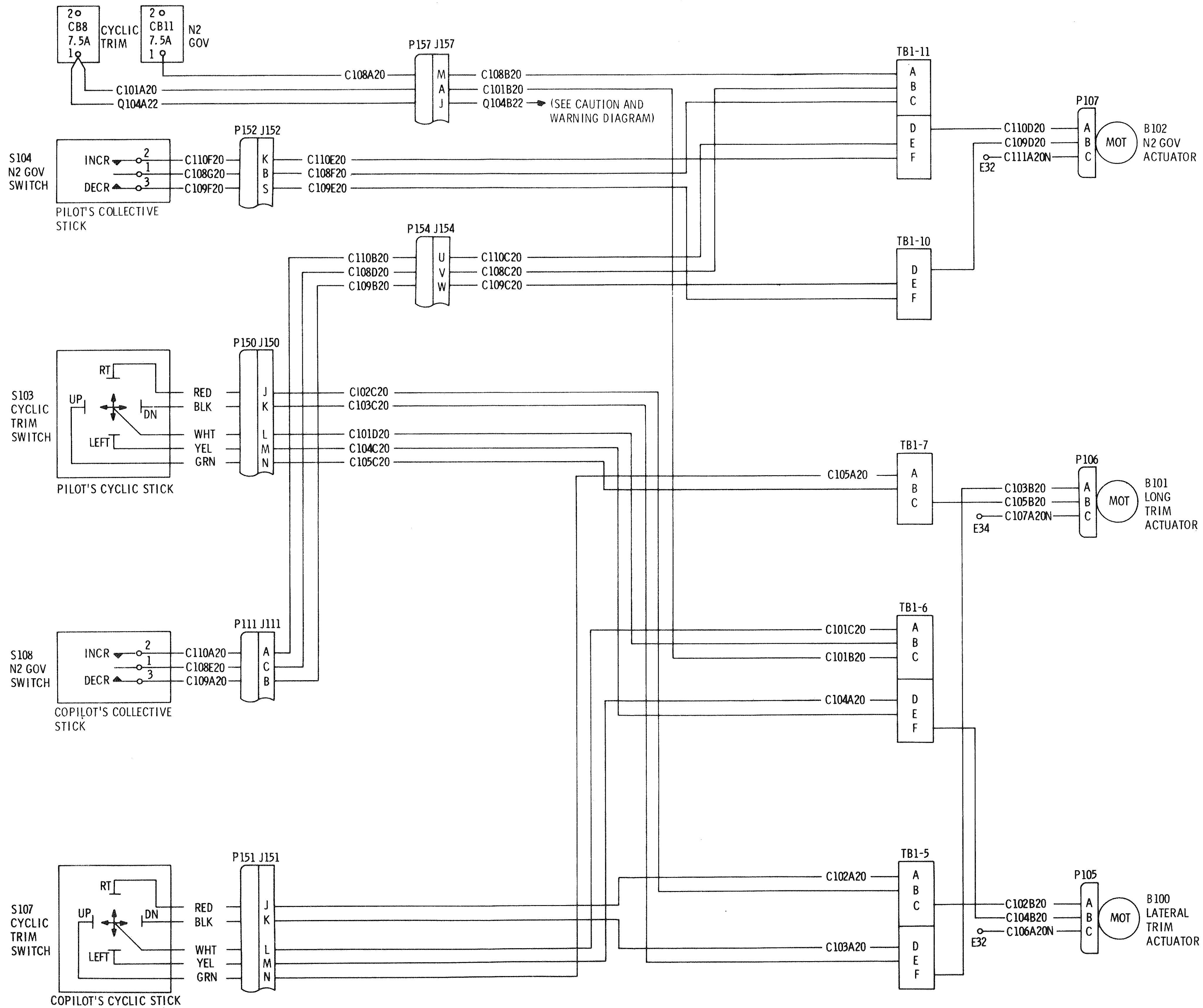


Figure FO-11. Series 2 Aircraft Trim Controls Wiring Diagram.

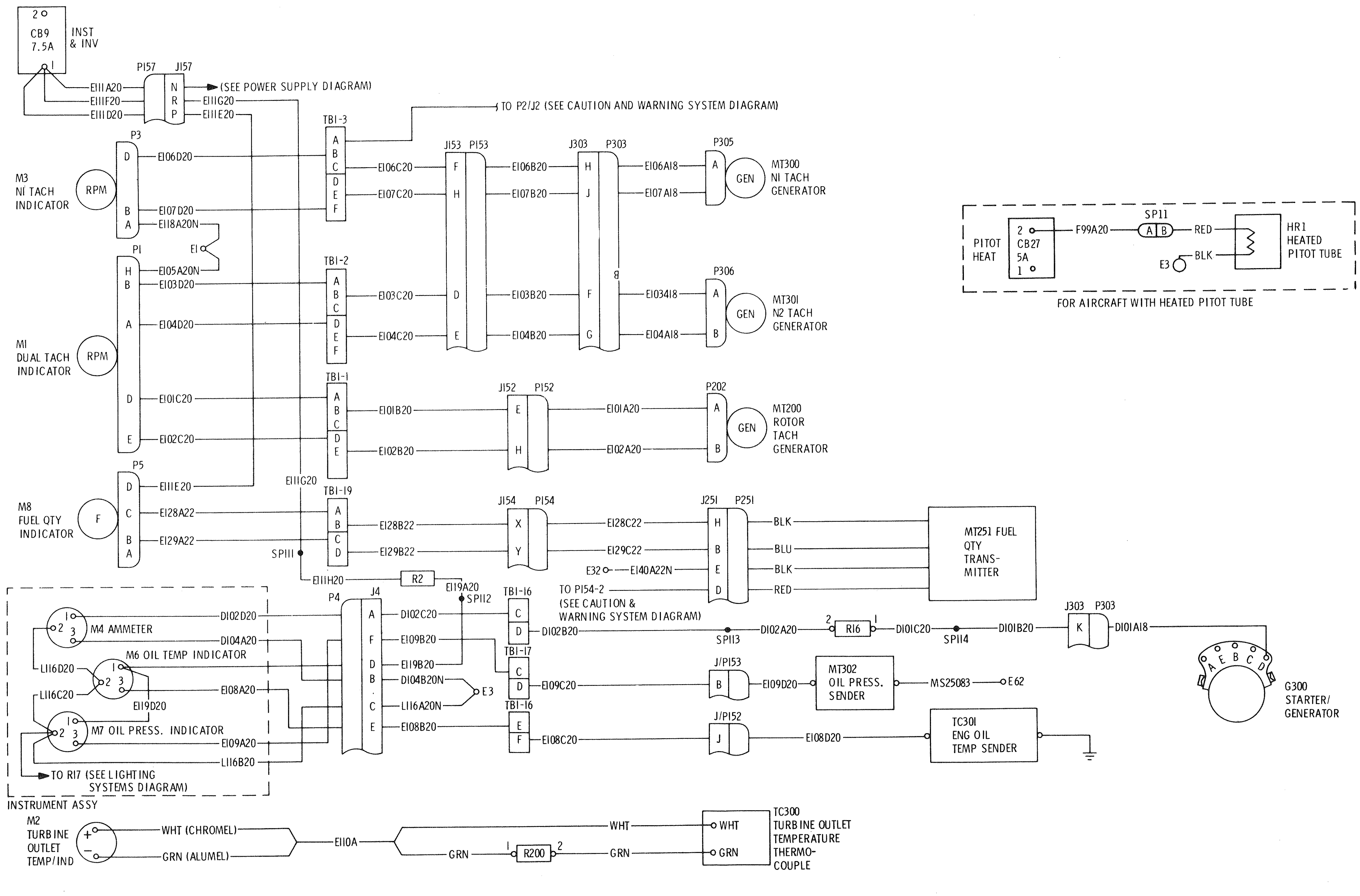


Figure FO-12. Series 2 Aircraft Instruments and Indicators Wiring Diagram.

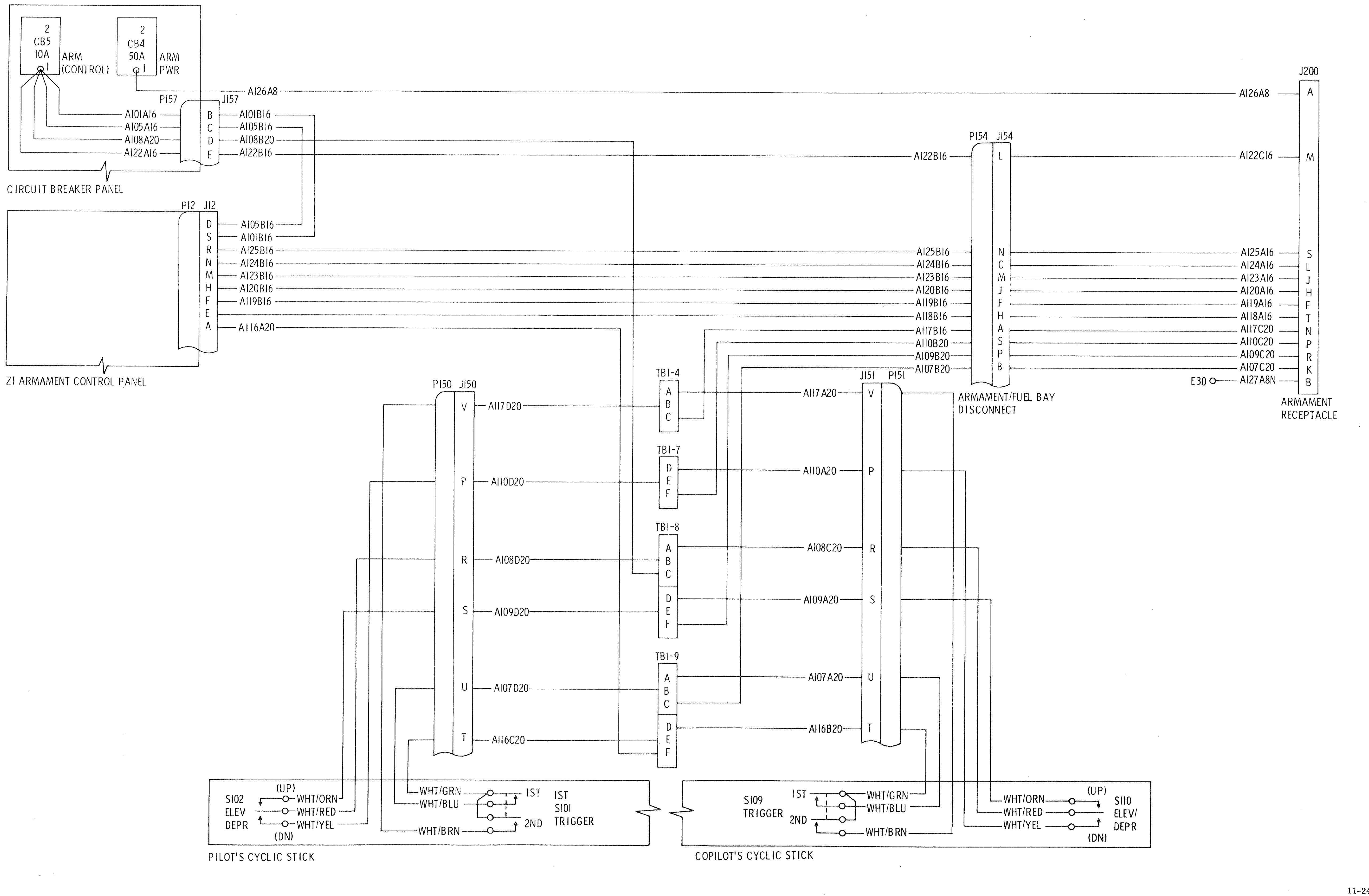
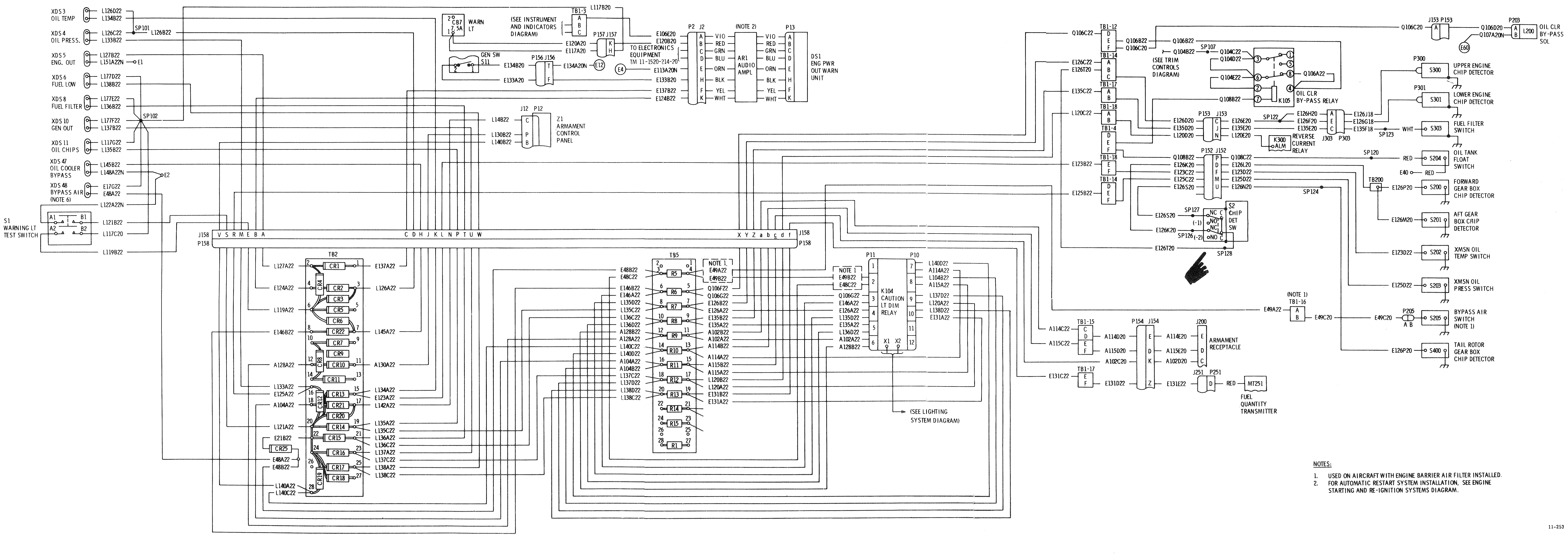


Figure FO-13. Series 2 Aircraft Armament Control System Wiring Diagram.



NOTES:

1. USED ON AIRCRAFT WITH ENGINE BARRIER AIR FILTER INSTALLED.
2. FOR AUTOMATIC RESTART SYSTEM INSTALLATION, SEE ENGINE STARTING AND RE-IGNITION SYSTEMS DIAGRAM.

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Figure FO-14. Series 2 Aircraft Caution and Warning System Wiring Diagram.

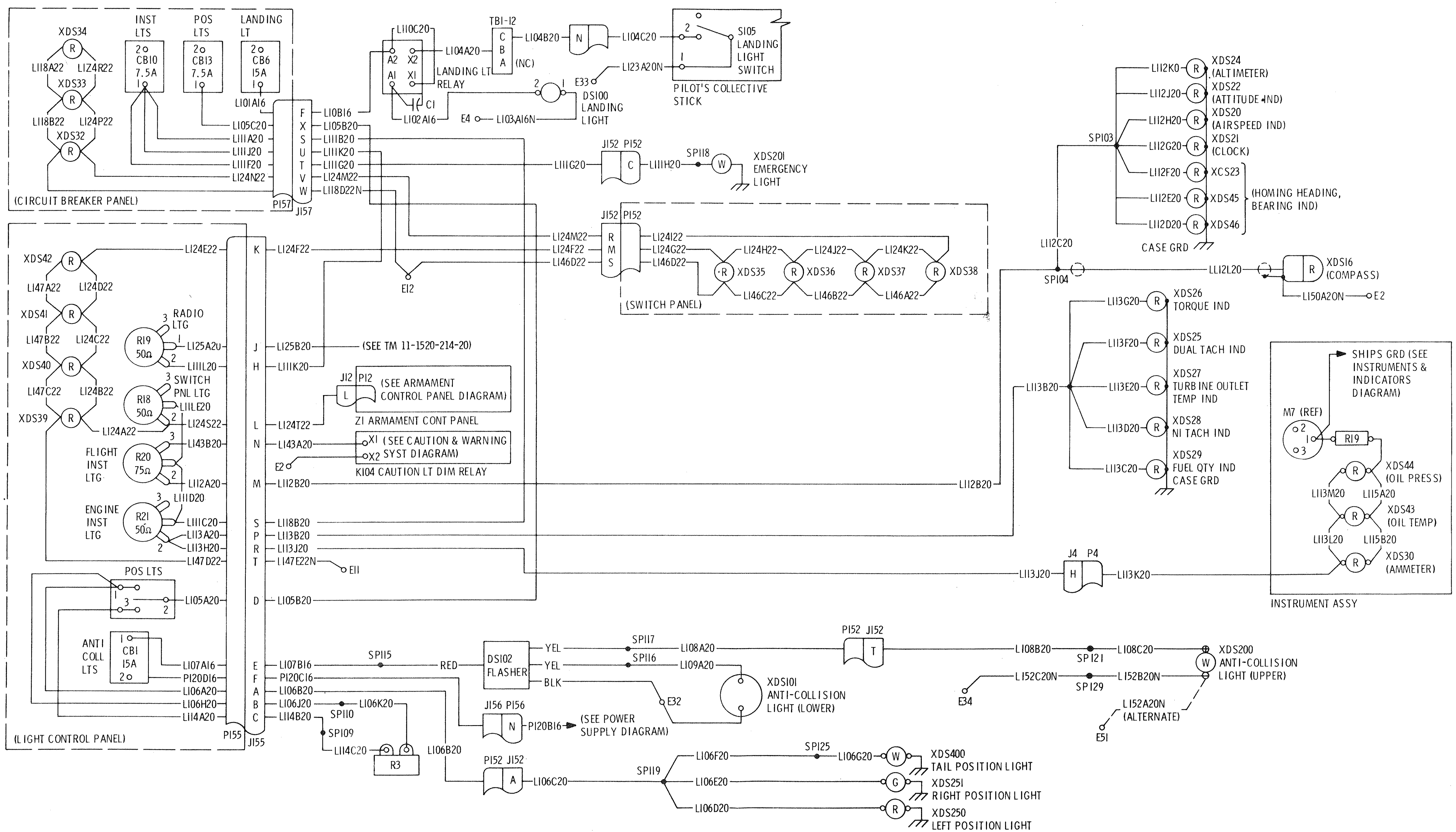

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Figure FO-15. Series 2 Aircraft Lighting Systems Wiring Diagram.

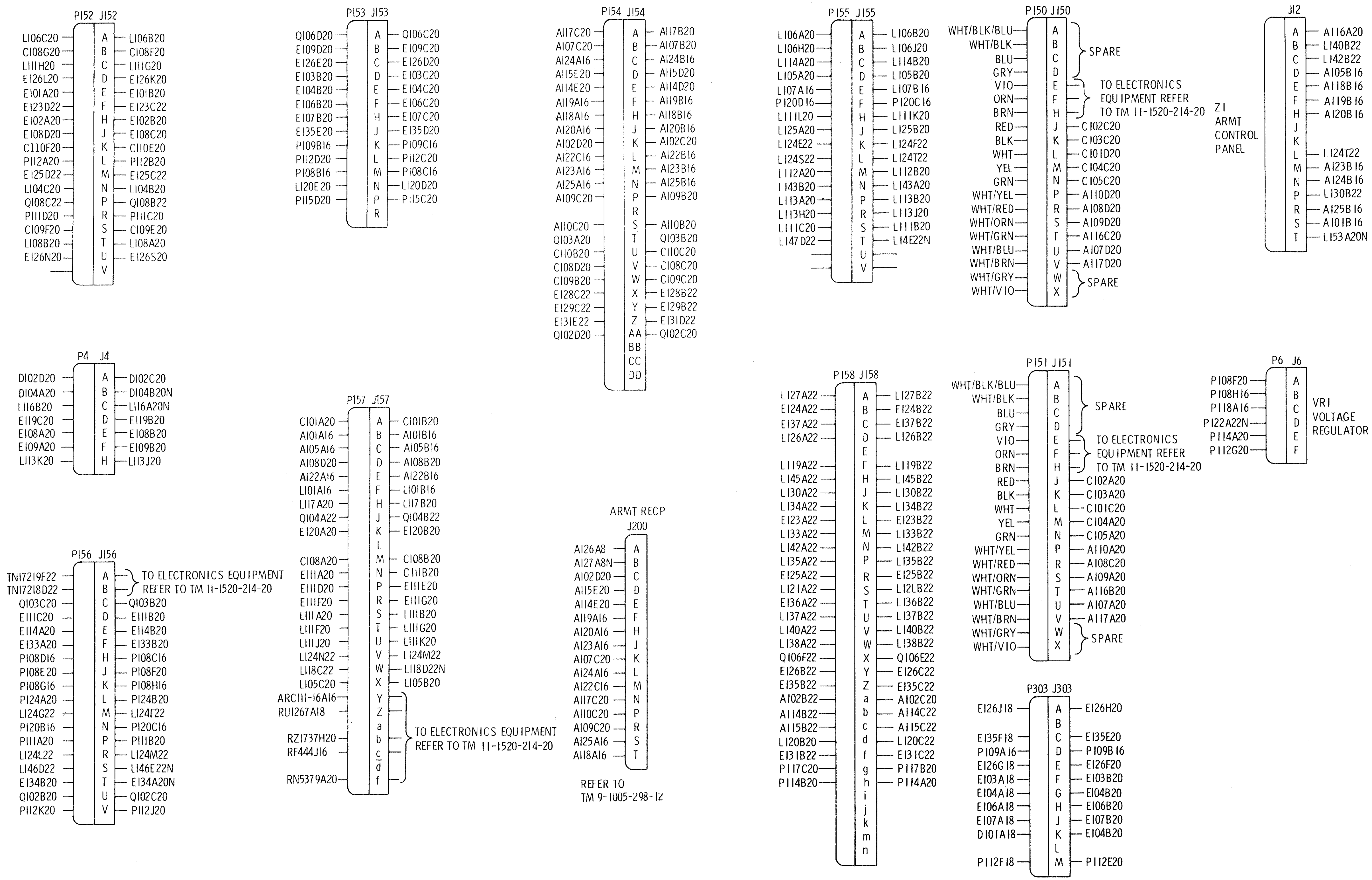

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Figure FO-16. Series 2 Aircraft Major Connectors Wiring Diagram.

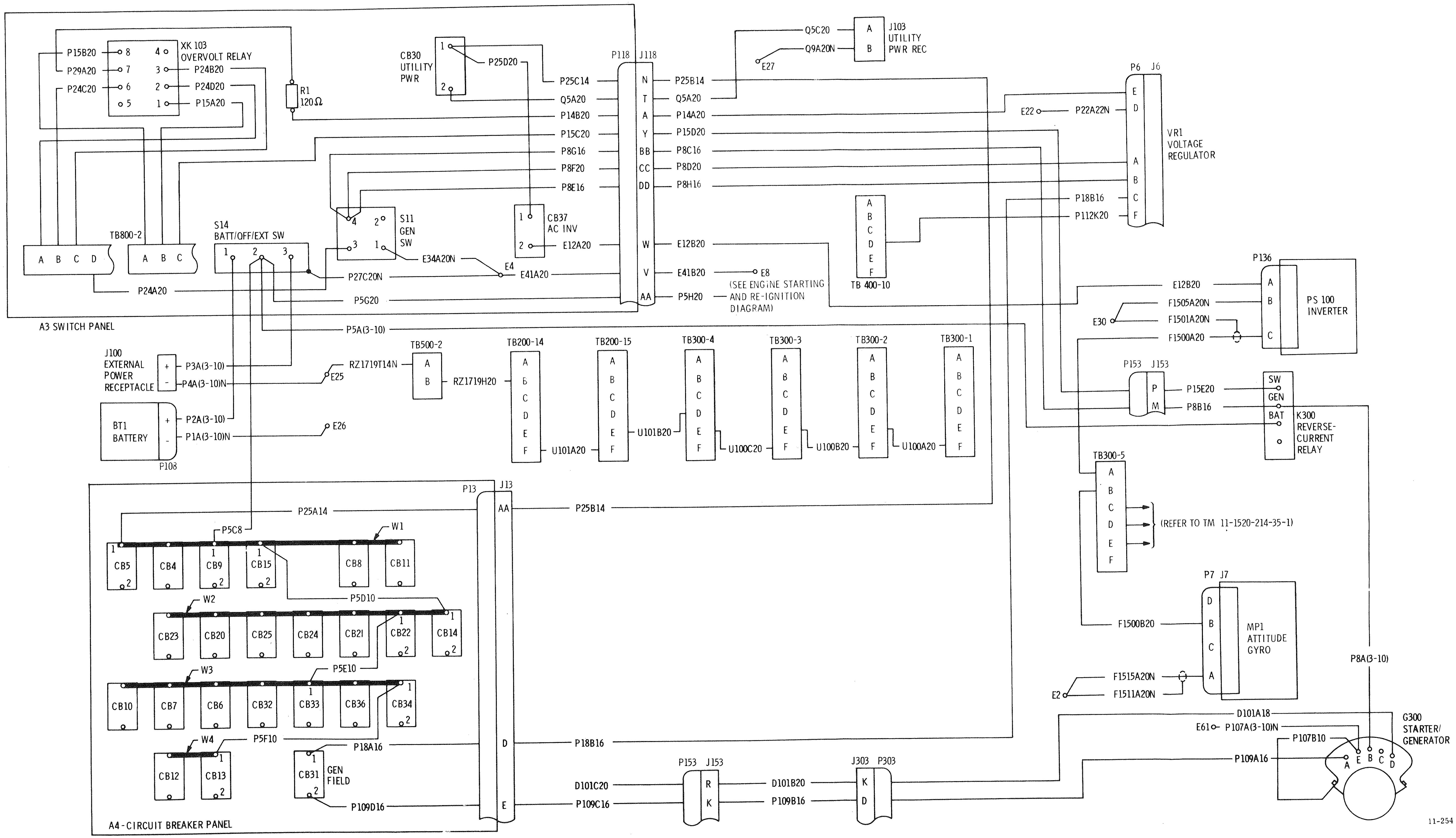

Figure FO-17. Series 3 Aircraft Power Supply Wiring Diagram.

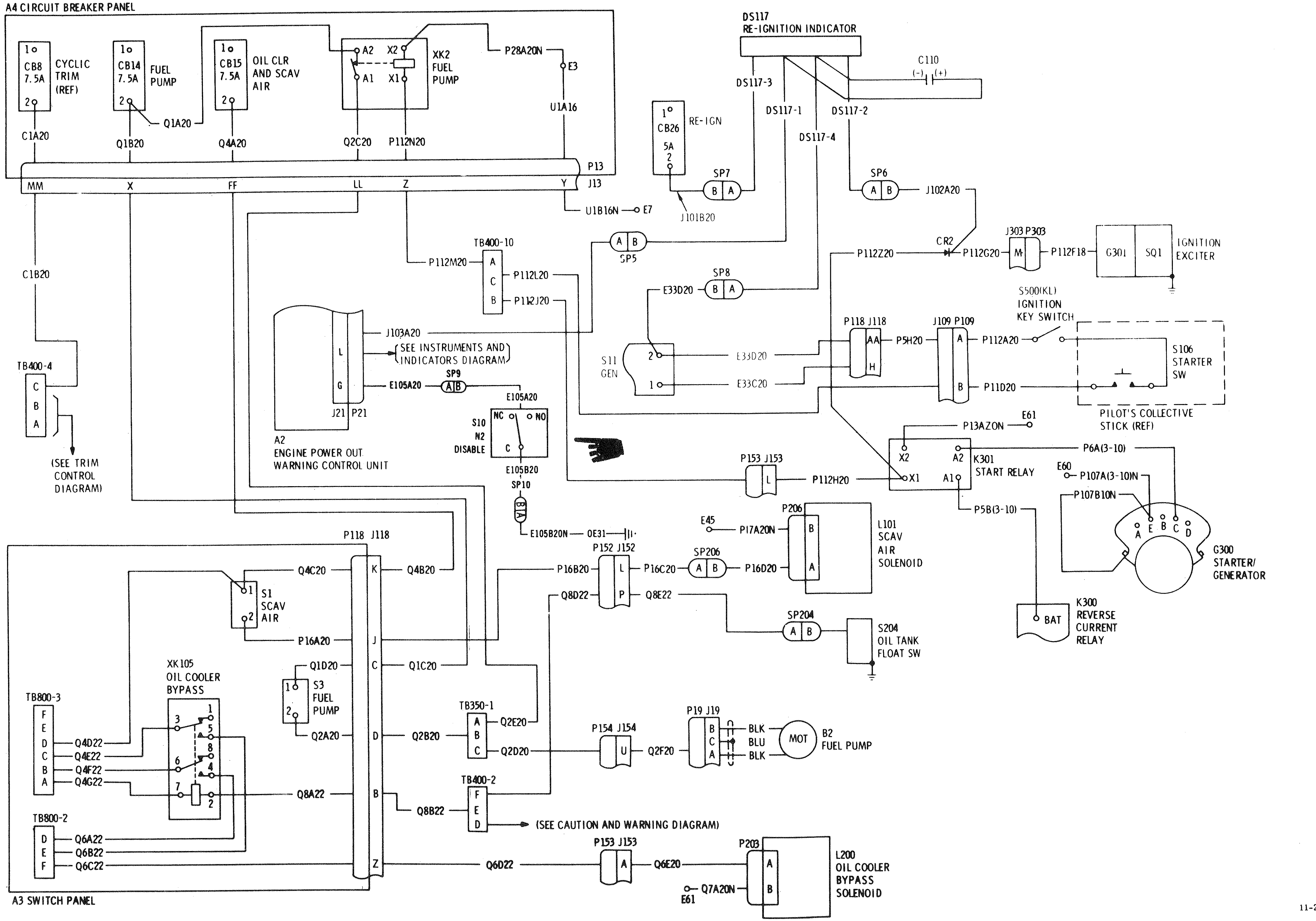

Figure FO-18. Series 3 Aircraft Engine Starting and Re-ignition Wiring Diagram.

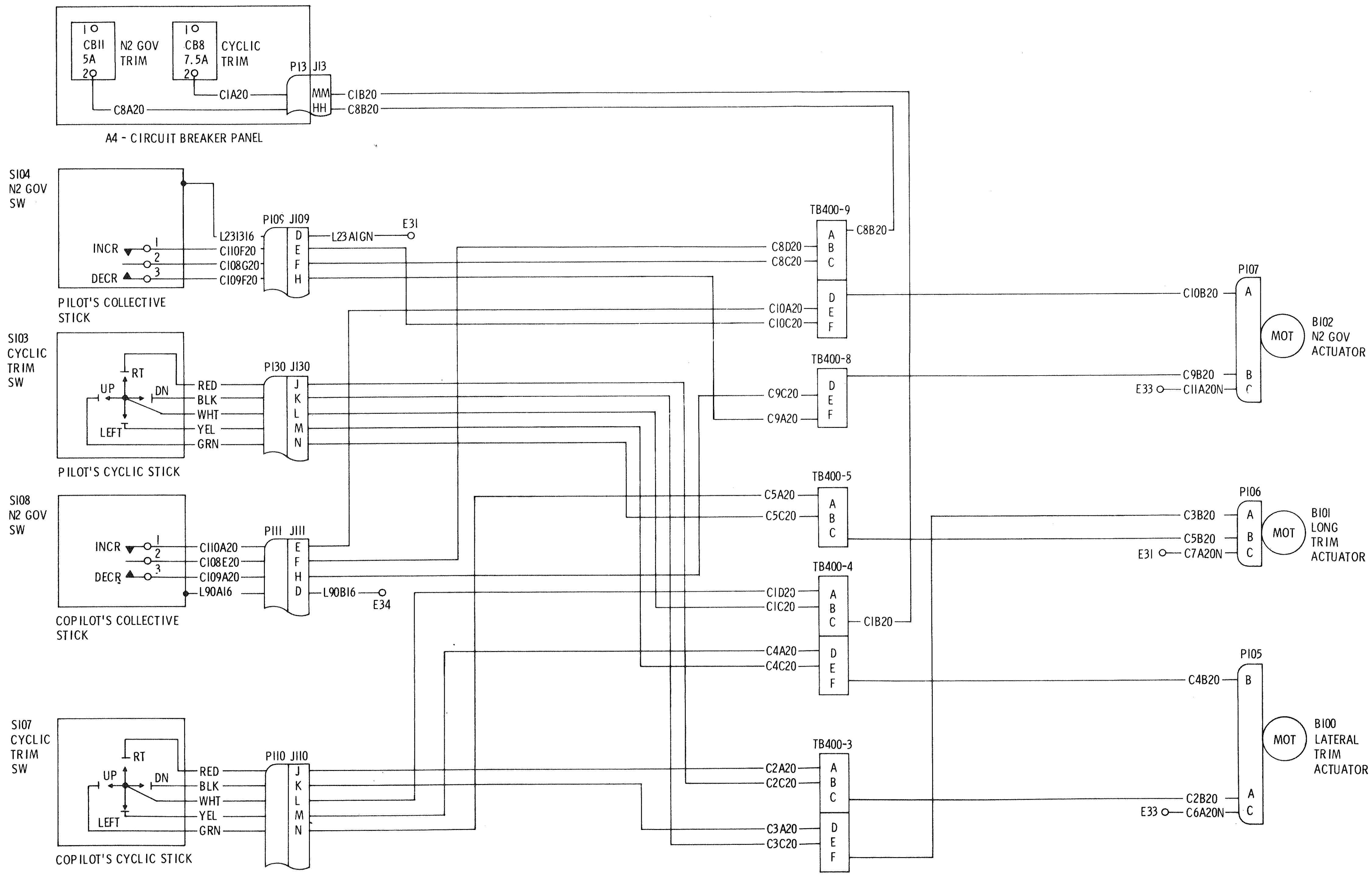

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Figure FO-19. Series 3 Aircraft Trim Controls Wiring Diagram.

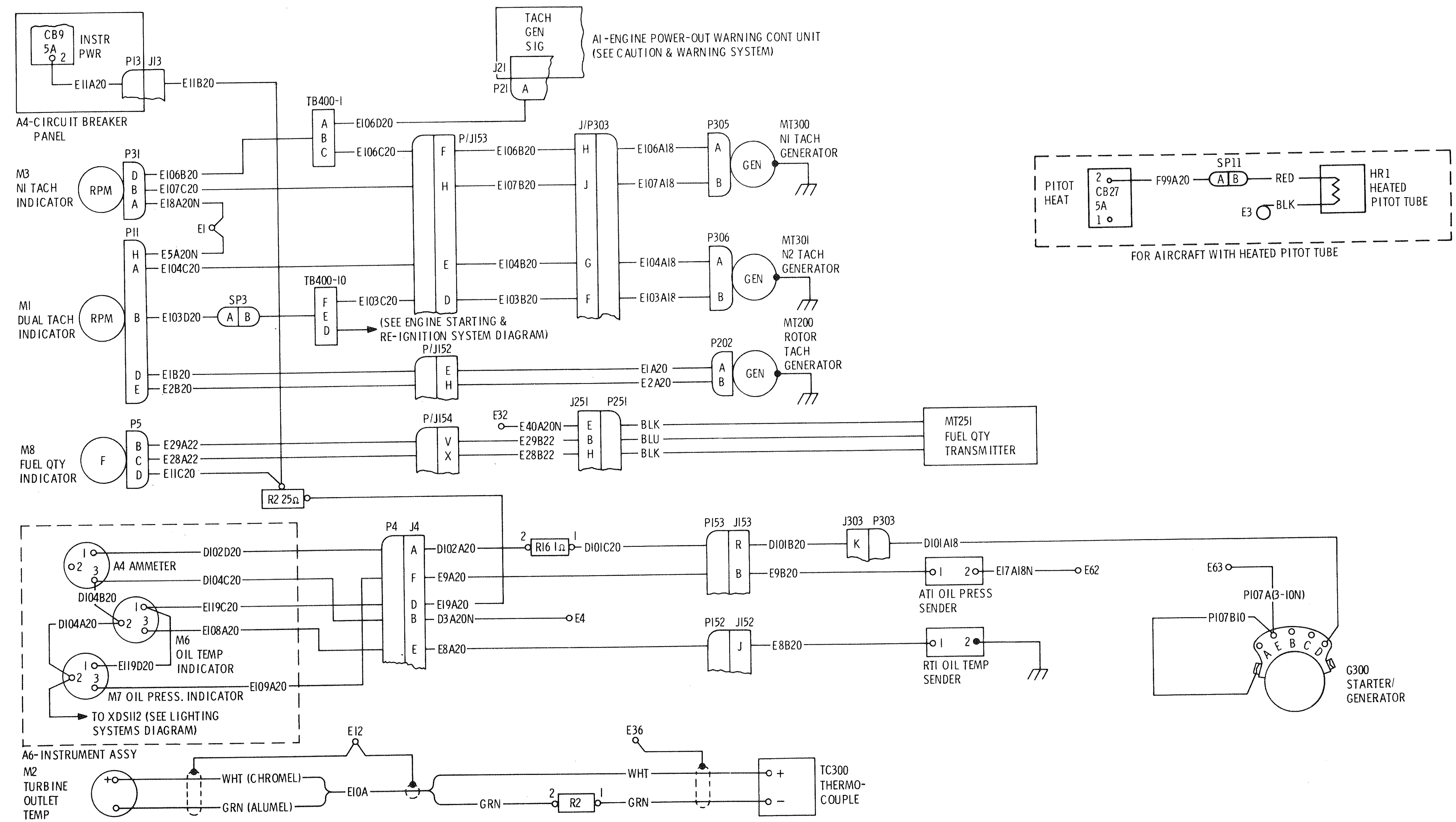

Figure FO-20. Series 3 Aircraft Instruments and Indicators Wiring Diagram.

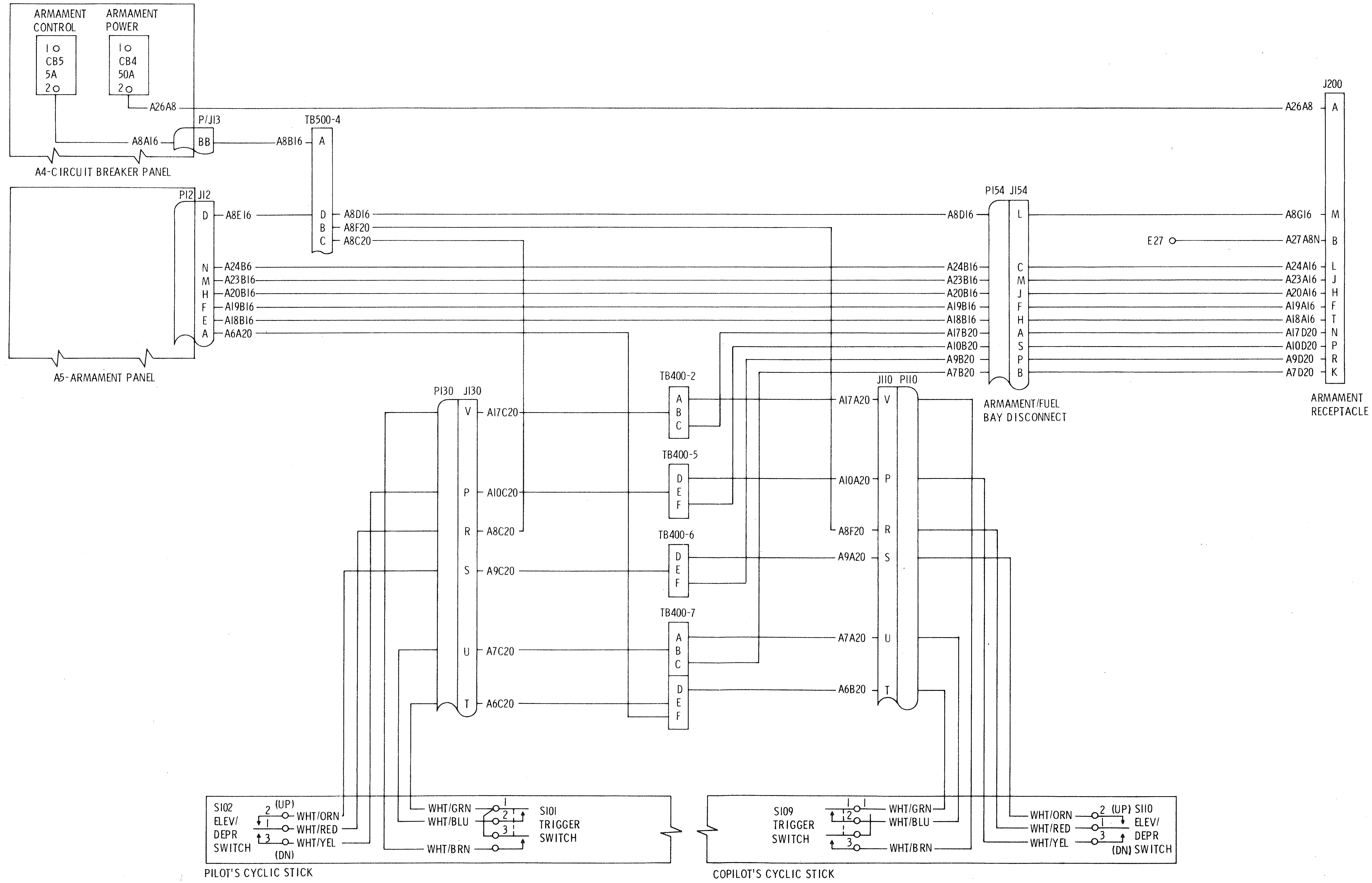

Figure FO-21. Series 3 Aircraft Armament Control System Wiring Diagram.

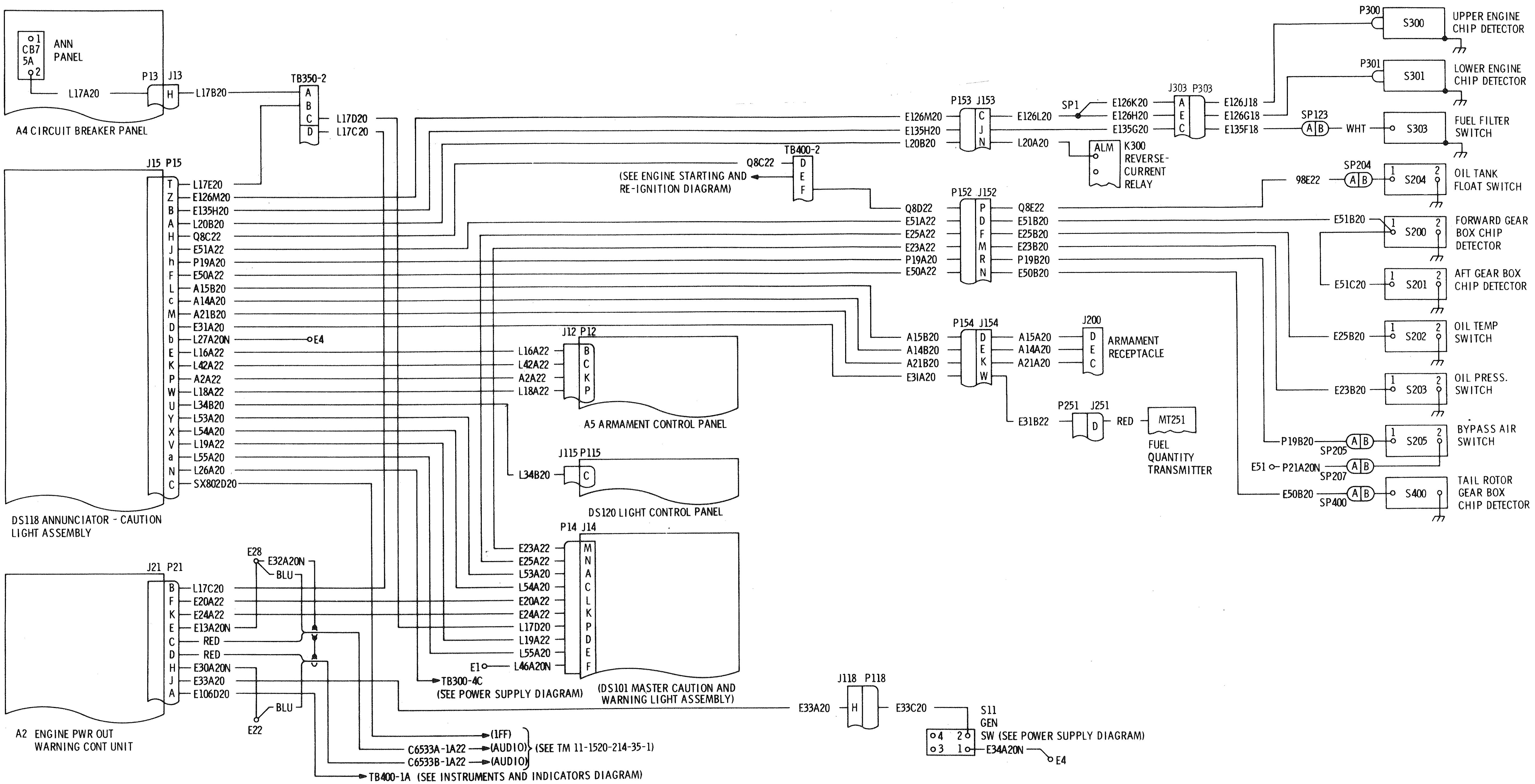

Figure FO-22. Series 3 Aircraft Caution and Warning System Wiring Diagram.

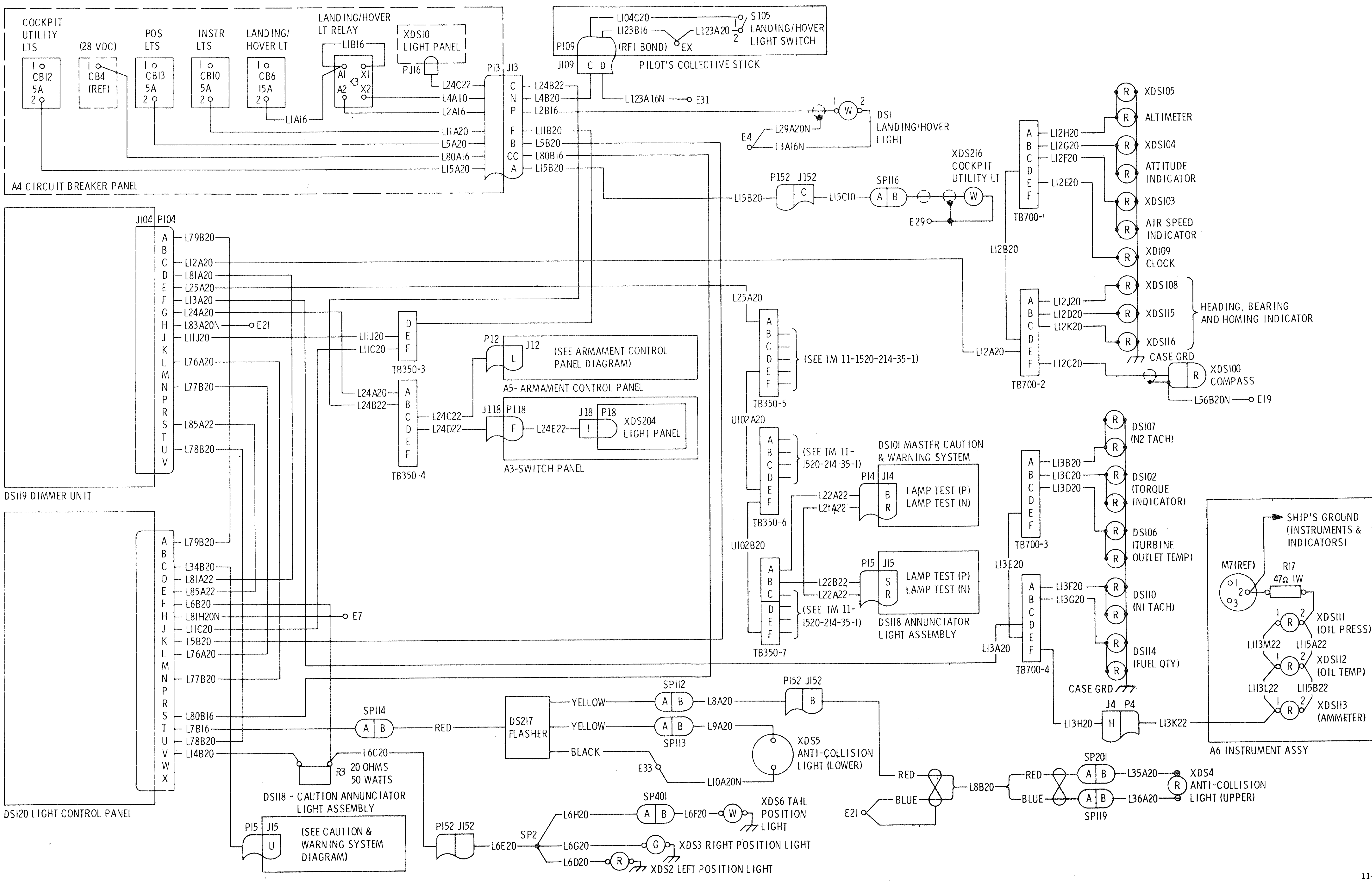

Figure FO-23. Series 3 Aircraft Lighting Systems Wiring Diagram.

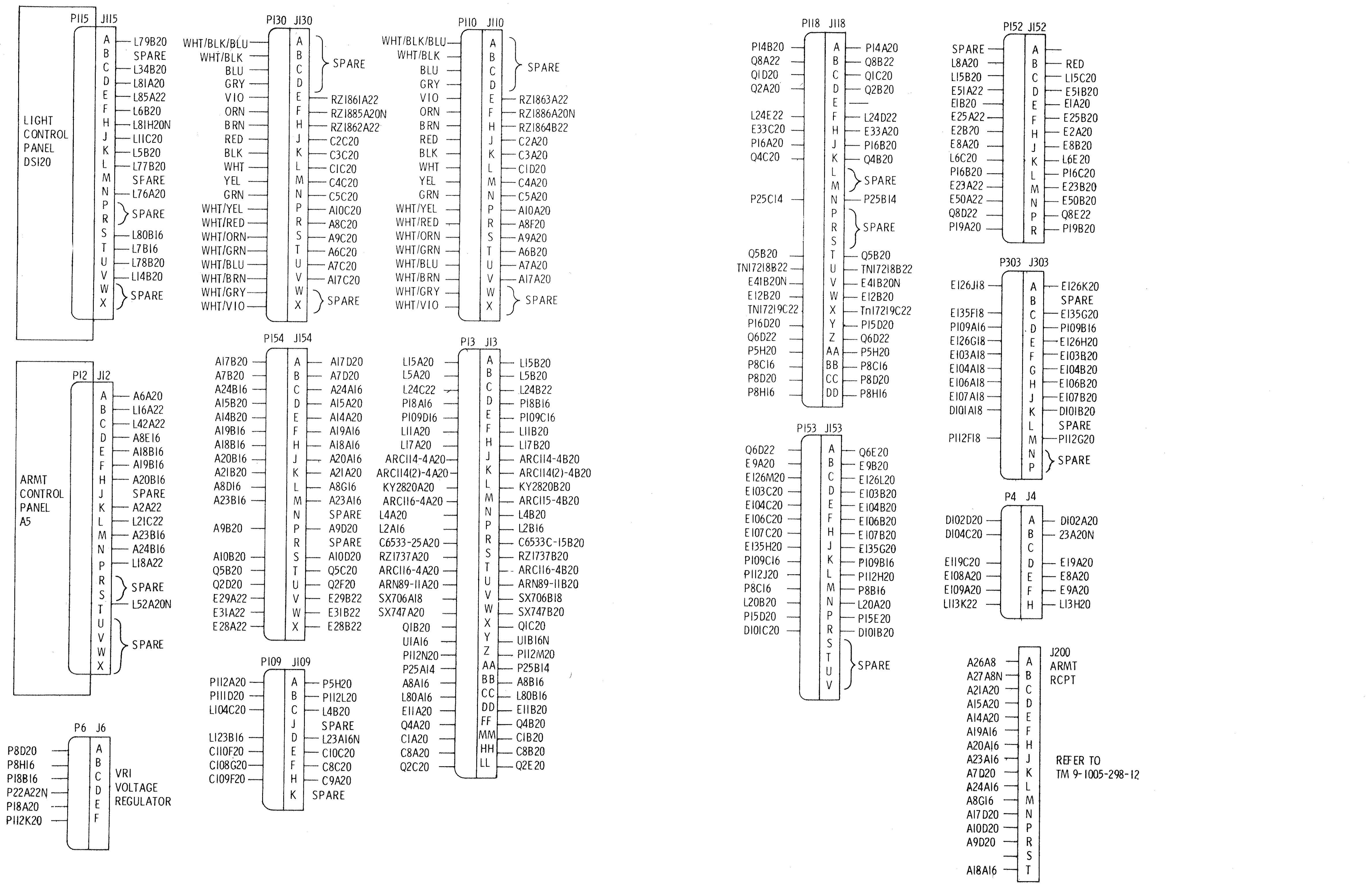

Figure FO-24. Series 3 Aircraft Major Connectors Wiring Diagram.

ALPHABETICAL INDEX

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13				Item No. 5.7 Recommend that X's be added to Sequence Number 5.7 under D (daily) and I (intermediate). The teardown analysis result from Corpus Christi Army Depot for engine number LE 21632 from aircraft 69-15324 revealed packed hard dirt had formed over the selfpurging port. This prevented sand and dust from being ejected overboard and eventually recirculated until ingested into the engine. This caused an excessive amount of erosion on internal components preventing the engine from producing maximum power. This inability contributed to a major accident involving fatalities. Increasing the inspection frequency would preclude a reoccurance of this malfunction.

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