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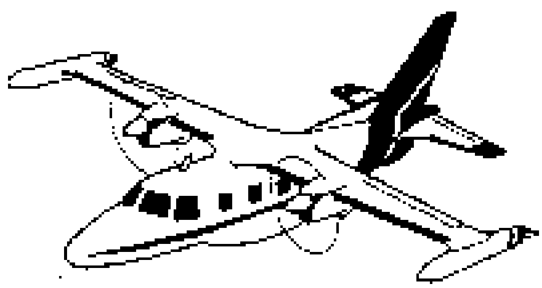

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Fax: (972) 248-3321

4951 AIRPORT PARKWAY,
SUITE 800
ADDISON, TEXAS 75001

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13-34	6-1-79	13-73	Original		
13-35	Original	13-74	4-1-78		
13-36	3-31-90	13-75	4-1-78		
13-36-1	3-31-90	13-76	4-1-78		
13-36-2		13-77	Original		
13-37	Original	13-78	Original		
13-38	Original	13-79(80)	Original		
13-39	Original	13-81	4-1-78		
13-40	Original	13-82	4-1-78		
13-41(42)	Original	13-83	Original		
13-43	Original	13-84	4-1-78		

Supplement

Sup 1-i	1-10-98
Sup.1-1	1-10-98
Sup.1-2	1-10-98
Sup.1-3/1-4	1-10-98
Sup.2-i	1-10-98
Sup.2-1	1-10-98
Sup.2-2	1-10-98
Sup.2-3/2-4	1-10-98
Sup.2-5/2-6	1-10-98
Sup 3-i	1-10-98
Sup.3-1	1-10-98
Sup 3-2	1-10-98
Sup 3-3/3-4	1-10-98

*Revised
**Added
( )Deleted

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CHAPTER

CHAPTER

1

GENERAL INFORMATION

CHAPTER I

GENERAL INFORMATION

CONTENTS

1. GENERAL

The MU-2 is a high wing, multi-engine turboprop type aircraft powered by two Garrett TPE 331 6 251M 715 SHP (*1), or TPE 331-5 252M 715 SHP (*2). The fuselage is of semi-monocoque construction with additional rigidity gained through the use of two lower parallel keels.

The aircraft is standard with an 8 place seating configuration and an 11 place high density seating arrangement is available.

The main wing center section LH W STA 2590 through RH W STA 2590 is of integral tank design (154 U.S. gallons) utilizing a built up front and rear spar, ribs, stringers, and stressed, chem-milled skin containing tank access doors. The outboard wings are of the same basic construction utilizing lighter skins and stringers. Each outboard wing contains a removable aluminum fuel tank of 15 U.S. gallon capacity. The outboard wing is attached to the inboard section by four bolts through hard points at the upper and lower spar caps. The main wing is attached to the fuselage by four bolts through clevis type wing to fuselage attach fittings.

The tip tanks (90 U.S. gallons each) are attached to the forward and rear spar by three bolts; fuel from these tanks is transferred by regulated air pressure.

Lateral control is accomplished through a spoiler system. Enabling the use of full span, doubled slotted flaps are actuated by a single DC motor through a reduction gearbox and drive shafts.

The tricycle type landing gear is retracted by a single DC motor through a reduction gear drive train. The main gear retracts forward into the fuselage bulge, while the nose gear retracts forward into the nose gear wheel bay. Emergency gear extension is accomplished through a manual ratchet system. Nose gear steering is manually linked to the rudder pedals.

All flight controls are manual systems except lateral trim which is through electrical actuators installed in the trailing edge of the flaps. The MU-2 hydraulic system consists of two disc type brakes, two mixer valves and four master cylinders. The pressurization system for the cabin compartment is provided by engine compressor bleed air which is circulated across a two stage cooling turbine to provide conditioned air. The system is capable of 5 psi (*1), 6 psi (*2) differential pressure and will provide a cabin altitude of 6,500 feet (*1), 9,000 feet (*2) at a flight altitude of 25,000.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

WARNING

Use only genuine MITSUBISHI or MITSUBISHI approved parts obtained from MITSUBISHI approved sources and Beech parts support network, in connection with the maintenance and repair of MITSUBISHI airplanes.
Genuine MITSUBISHI parts are produced and inspected under rigorous procedures to ensure airworthiness and suitability for use in MITSUBISHI airplane applications. Parts purchased from sources other than MITSUBISHI and Beech parts support network, even though outwardly identical in appearance may not have had the required tests and inspections performed, may be different in fabrication techniques and materials, and may be dangerous when installed in an airplane. Salvaged airplane parts, reworked parts obtained from non-MITSUBISHI/ Beech approved sources, or parts, components, or structural assemblies, the service history of which is unknown or cannot be authenticated, may have been subjected to unacceptable stresses or temperatures or have other hidden damage, not discernible through routine visual or usual nondestructive testing techniques.
This may render the parts, components or structural assembly, even though originally manufactured by MITSUBISHI, unsuitable and unsafe for airplane use. MITSUBISHI expressly disclaims any responsibility for malfunctions, failures, damage or injury caused by use of non-MITSUBISHI/Beech approved parts.
Contact Beech Aircraft Corporation for MITSUBISHI approved parts support network.

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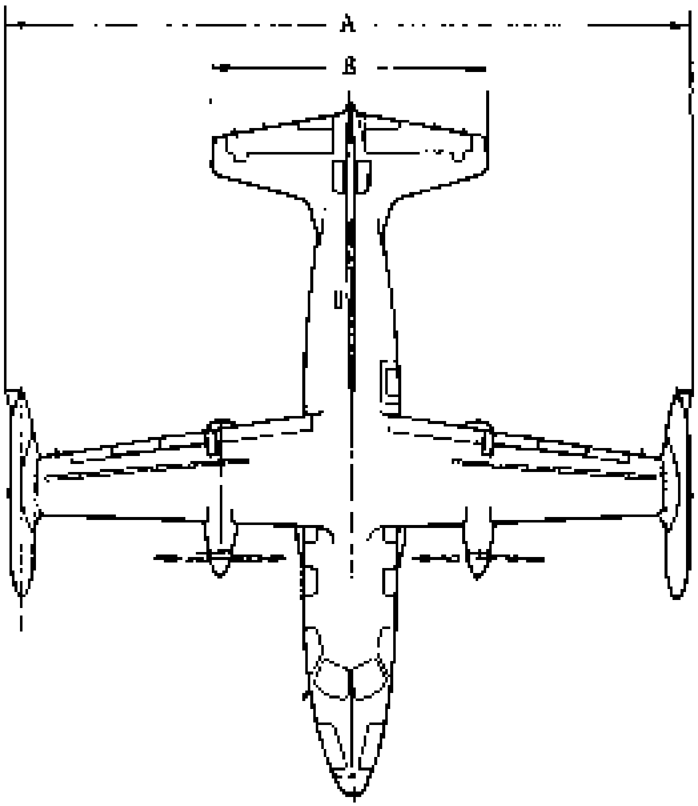
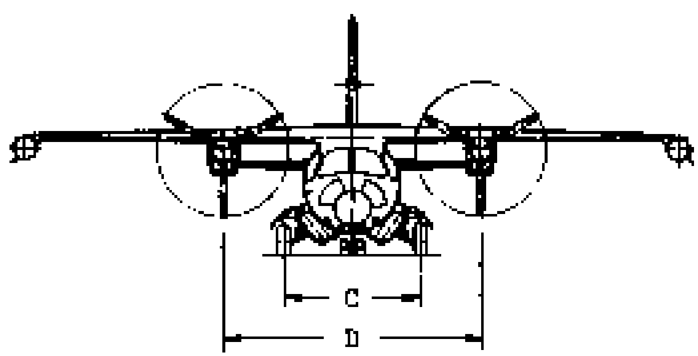




A = 39 Ft 2 In (11.94 M)
B = 15 Ft 9 In (4.80 M)
C = 7 Ft 11 In (2.40 M)
D = 14 Ft 9 In (4.50 M)
E = 14 Ft 5 In (4.40 M)
F = 38 Ft 10 In (11.84 M)
G = 39 Ft 5 In (12.02 M)
H = 13 Ft 8 In (4.17 M)

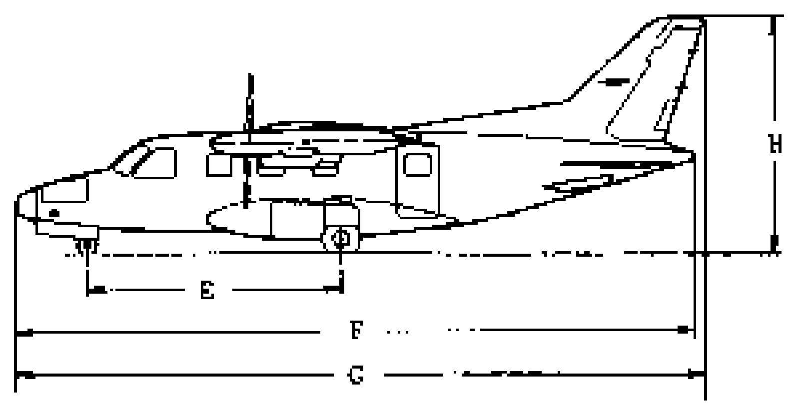


Fig 1-1 General dimensions (1/2)
(Aircraft S/N 652SA)

A = 39 Ft 2 In (11.94 M)

B = 15 Ft 9 In (4.80 M)

C = 7 Ft 11 In (2.40 M)

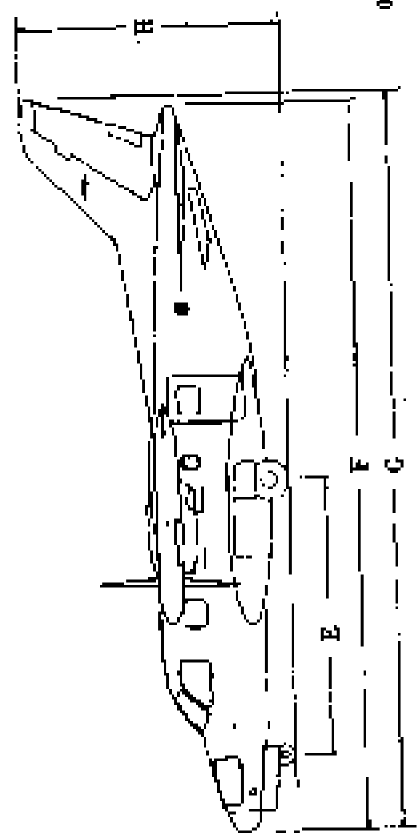
D = 14 Ft 9 In (4.50 M)

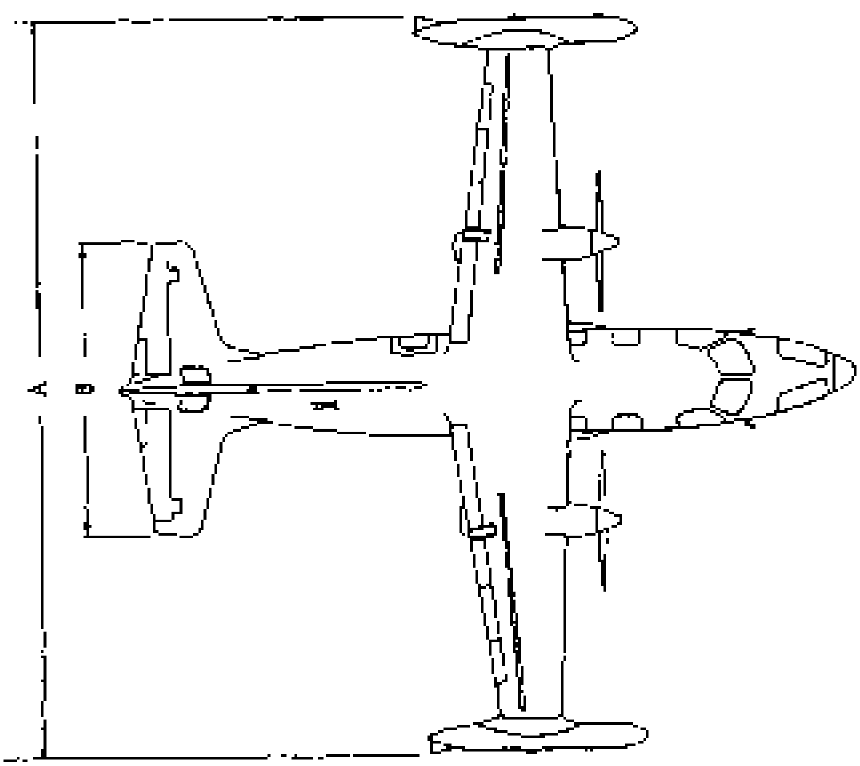
E = 14 Ft 5 In (4.40 M)

F = 38 Ft 10 In (11.84 M)

G = 39 Ft 5 In (12.02 M)

H = 13 Ft 8 In (4.17 M)





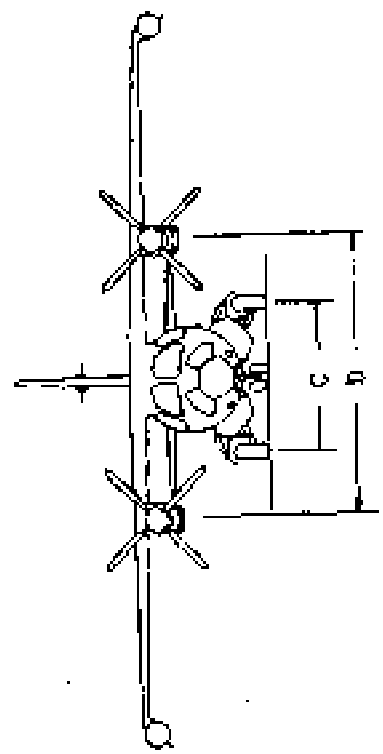


Fig 1-1 General dimensions (2/2)
(Aircraft S/N 661SA, 697SA and subsequent)

2. STATION DIAGRAM (See Fig. 1-2)

2.1 GENERAL DESCRIPTION

(1) Fuselage

Fuselage station (F. STA) is a plane perpendicular to the center line of fuselage circular section. Station 0 is located 170 mm (6.693 in.) aft of the fuselage nose.
Buttock plane (B. P.) is a plane parallel to fuselage center plane, (B. P. 0).
Water plane (W. P.) is a plane parallel to fuselage horizontal center plane.

NOTE

Station locations are given in metric measurement (millimeters). To convert station locations to inches divide the number given by 25.4.
(Example: $\frac{4660}{25.4} = 183.46$)

(2) Wing

Wing station (W. STA) is a plane perpendicular to wing reference plane and W. STA 0 coincide with B. P. 0.
Wing reference plane is a plane canted 2° to horizontal plane at B. P. 0 and passing through the chord line at W. STA 0.

(3) Nacelle

Nacelle station (N. STA) is a plane perpendicular to the direction of propeller thrust. N.STA 0 coincides with F. STA 3945.

(4) Flap

Flap station (FLAP STA) is a plane perpendicular to flap hinge line.
FLAP STA 0 is a plane passing through F. STA 4958.74 and B.P. 39.10.

(5) Horizontal Stabilizer

Horizontal stabilizer station (H. STA) is a plane parallel to fuselage center plane. H. STA 0 coincides with fuselage center plane and perpendicular to hinge center line of elevator.

(6) Vertical Stabilizer

Vertical stabilizer station (V.STA) is a plane perpendicular to rudder spar reference plane. V.STA 0 is a plane passing through F.STA 9758.85 and W.P. 13. Rudder spar reference plane is a plane canted 58° 18' to horizontal plane and passing through F.STA 9758.85 and W.P. 13.

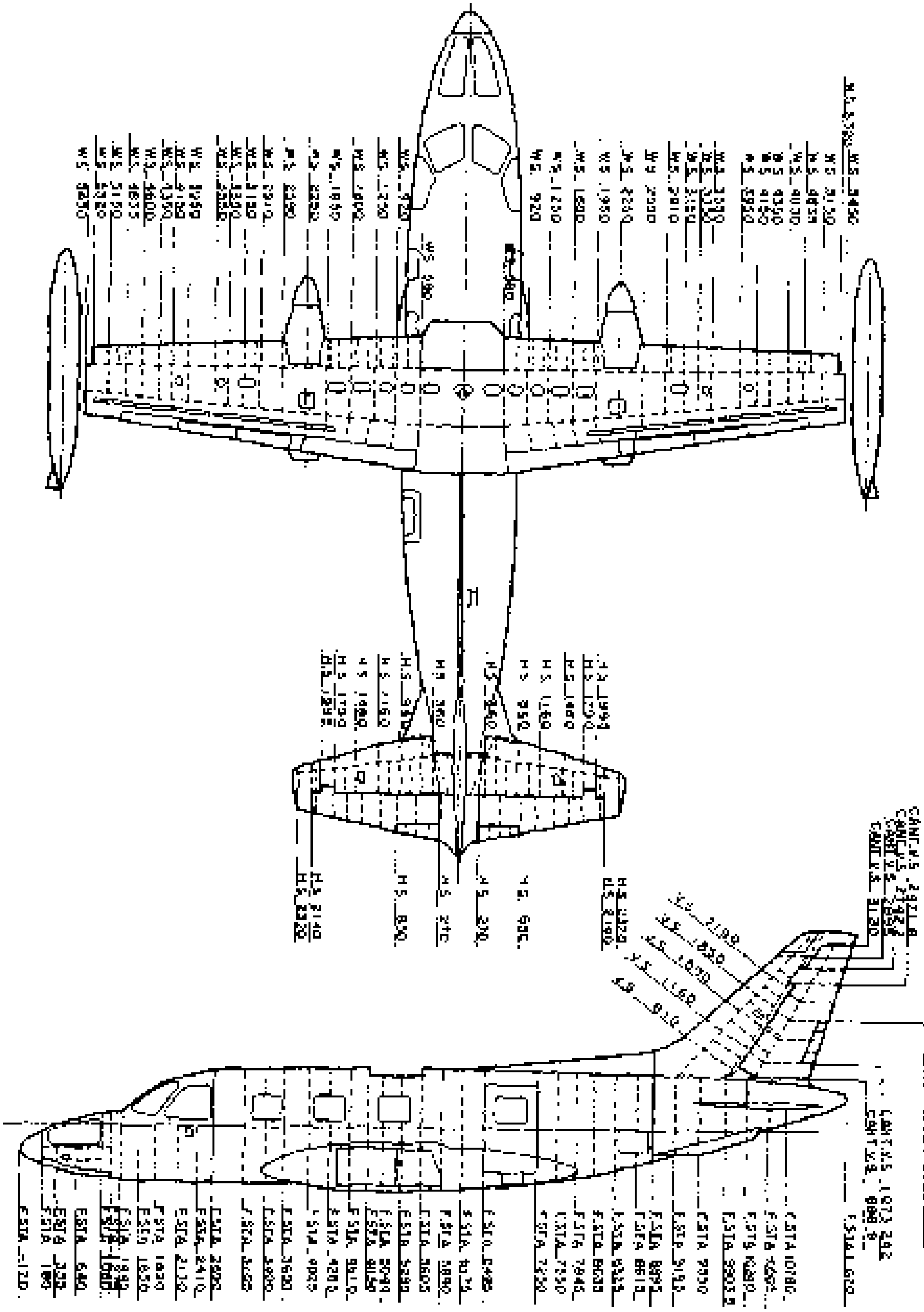


Figure 1-2 Station diagram

3. GENERAL SPECIFICATIONS

3.1 ENGINE

(*1) GARRETT TPE 331-6-251M (715 SHP)
(*2) GARRETT TPE 331-5-252M (715 SHP)

Compressor :	Two-stage centrifugal type
Combustion chamber :	Annular type
Turbine :	Three-stage axial type
Anti-icing :	Bleed-air type
RPM :	41,730 RPM - takeoff power, maximum continuous power and cruise power 40,100 RPM (Min.) - cruise power
Power :	715 SHP plus jet thrust for takeoff and maximum continuous power operation

3.2 PROPELLER

Hub :	(*1) Hartzell Model HC-B3TN-5 (*2) Hartzell Model HC-B4TN-5DL, HC-B4TN-5GL or HC-B4TN-5JL
Blade :	(*1) Hartzell Model T10178HE-11, T10178HB-11 or T10178HB-11R (*2) Hartzell Model LT10282B-5.3R, LT10282HB-5.3R, LT10282K-5.3R, LT10282NHB-5.3R, LT10282NB-5.3R or LT10282NK-5.3R
Control :	Hydraulic, constant-speed, full-feathering system with reverse pitch
Feather :	Spring type
Diameter :	(*1) 7 ft. 6 in. (2286 mm) (*2) 8 ft. 2 in. (2489 mm)
Blade :	(*1) 3 (*2) 4
RPM :	(*1) 2,000 rpm ... Takeoff, maximum continuous and cruise power 1,922 rpm (Min.) ... cruise power (*2) 1,591 rpm ... Takeoff, maximum continuous and cruise power 1,527 rpm (Min.) ... cruise power

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

3.3 AIRFRAME

3.3.1 DIMENSIONS

(1) AIRFRAME

Overall span	39 ft. 2 in. (11.94 m)
Overall length	39 ft. 5 in. (12.02 m)
Overall height	13 ft. 8 in. (4.17 m)
Propeller ground clearance	(*1) 2 ft. 10 in. (863.6 mm) (*2) 2 ft. 6 in. (762.0 mm) (Dimension given for normal tire inflation and strut extension)
Propeller fuselage clearance	(*1) 1 ft. 3 in. (381.0 mm) (*2) 11 in. (279.4 mm)

(2) Wing

Area	178 sq. ft. (16.55 m^2)
Span	39 ft. 2 in. (11.94 m)
M.A.C.	5 ft. 1 in. (1.538 m)
Aspect ratio	7.71
Wing section Root	NACA 64A415
Tip	NACA 63A212

Outboard leading edge is modified with droop.

Incidence	2°
Wash out	3°
Dihedral	0°
Sweep back (25% chord)	-0° 21'

(3) Flap

Type	Double slotted
Area	21.0 sq. ft. x 2 (1.95 m^2 x 2)
Maximum deflection	40°

(4) Spoiler

Area	2.91 sq. ft. x 2 (0.270 m^2 x 2)
Maximum deflection	Up 60° Down 14°

(5) Fuselage

Total length	38 ft. 10.0 in. (11.84 m)
Maximum diameter	5 ft. 5.4 in. (1.660 m)
Cabin length	19 ft. 8 in. (5.995 m)
Cabin width	4 ft. 11 in. (1.500 m)
Cabin height	4 ft. 3.2 in. (1.300 m)

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

(6) Horizontal stabilizer

Area	58.3 sq. ft. (5.412 m^2)
Span	15 ft. 9 in. (4.800 m)
M.A.C.	4 ft. (1.219 m)
Aspect ratio	4.26
Wing section	NACA64A010

The leading edge is modified with droop.

Incidence	0°
Wash out	0°
Dihedral	0°
Elevator area	7.52 sq. ft. x 2 (0.699 m^2 x 2)
Maximum deflection	Up 28°
	Down 12°

(7) Vertical stabilizer

Area	43.3 sq. ft. (4.02 m^2)
Total length	8 ft. 1.4 in. (2.473 m)
M.A.C.	5 ft. 10.7 in. (1.796 m)
Wing section	NACA64A008

(8) Rudder

Area	12.6 sq. ft. (1.171 m^2)
Maximum deflection	Left 22° (*1) 24° (*2)
	Right 24° (*1) 22° (*2)

(9) Trim tab deflection

Elevator tab	Nose up	30°
	Nose down	10°
Rudder Tab	Left	25°
	Right	25°
Aileron trim	Up	20°
	Down	20°

(10) Landing gear system

Main wheel tire	T.R.A. Type III 8.50-10 10 ply tubeless or regular tire
Nose wheel tire	T.R.A. Type III 5.00-5 6 ply tire
Wheel base	14 ft. 5.0 in. (4.40 m)
Tread	7 ft. 11 in. (2.4 m)

(11) Fuel tank

Main tank capacity	159 U.S. gallons (602 L)

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697 and subsequent

Outer tank capacity	15 U.S. gal. x 2 (56.8 L x 2)
Wing tip tank capacity	93 U.S. gal. x 2 (352 L x 2)
Total tank capacity	375 U.S. gal. (1419.4 L)
Unusable fuel	11 U.S. gal. (41.7 L)

(12) Engine oil

Tank capacity	6.25 qt. x 2 (5.91 L x 2)

3.4 WEIGHT

(1) Weight

*1 Maximum ramp weight	10,850 lbs (4,921 kgs)
Maximum takoff weight	10,800 lbs (4,899 kgs)
Maximum landing weight	10,260 lbs (4,654 kgs)
Maximum zero fuel weight	9,950 lbs (4,513 kgs)
*2 Maximum ramp weight	11,625 lbs (5,273 kgs)
Maximum takeoff weight	11,575 lbs (5,250 kgs)
Maximum landing weight	11,025 lbs (5,000 kgs)
Maximum zero fuel weight	9,950 lbs (4,513 kgs)

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

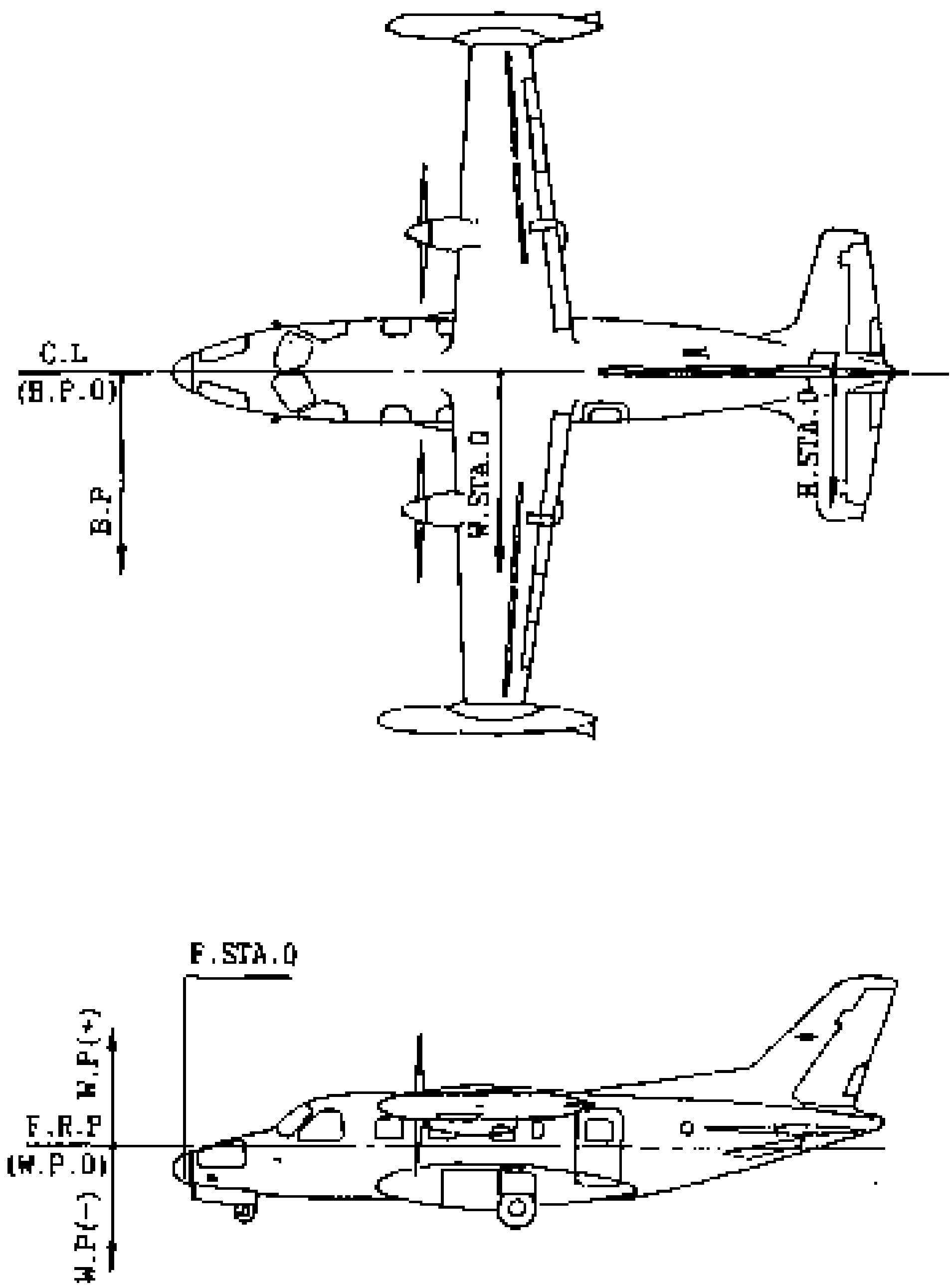


Fig 1-3 Abbreviation

CHAPTER

1A

AIRWORTHINESS LIMITATIONS

CHAPTER 1A

AIRWORTHINESS LIMITATIONS

CONTENTS

1 GENEAL

The requirements in the Airworthiness Limitations Section ARE MANDATORY REQUIREMENTS approved by the FAA.

NOTE

Refer to the FAA Approved Airplane Flight Manual for a detailed delineation of the flight limitations of the airplane.

Airworthiness limitation "Time-Limited Inspections" assigned to airplane components and assemblies are based upon experience, testing and engineering judgement and are subject to change only with FAA approval.

Listed inspection intervals shall be considered maximum. Inspection frequency may be decreased and the number of items increased as deemed necessary to fit operator's application of airplane.

2. TIME-LIMITED INSPECTIONS (see Table 1A-1)

maintenance manual

Table 1A-1 Time limited inspections

Item	Component	Condition	Inspection Interval	Reference for Inspection Criteria
1)	Windows			
	a. Windshield and windows (Acrylic):	CRAZING and CRACKS	Every 100 hours	Page 3-27, Paragraph 5.4
	b. Heated windshield (If installed):	DELAMINATION and CRACKS	Every 100 hours	Page 9-11, Paragraph 3.2.2.1
2)	Surface Anti/De-Ice Control System			
	a. De-ice boot system:	PROPER OPERATION and CONDITION	Every 100 hours	Page 9-8-1/9-8-2, Paragraph 2.0
	b. Windshield anti-ice system (Heated windshield):	PROPER OPERATION	Every 100 hours	Page 9-14, Paragraph 3.2.2.4
	Windshield anti-ice system (Fluid):	PROPER OPERATION	Every 100 hours	Page 9-20, Paragraph 3.3.6
	c. Tip tank taxi lights (if installed):	PROPER OPERATION	Every 100 hours	Page 13-66, Paragraph 4.2.2
	d. Engine air intake anti-ice system:	PROPER OPERATION	Every 100 hours	Page 9-38, Paragraph 6.2.2
	e. Propeller de-ice system:	PROPER OPERATION and CONDITION	Every 100 hours	Page 9-41, Paragraph 7.3 - 7.6
	f. Pitot/static anti-ice system:	PROPER OPERATION	Every 100 hours	Page 9-47, 8.2.1 and Page 9-49 Paragraph 8.2.2
	g. Stall warning anti-ice system:	PROPER OPERATION	Every 100 hours	Page 9-51, Paragraph 9.3
	h. Oil cooler air inlet anti-icing system:	PROPER OPERATION	Every 100 hours	Page 9-53, Paragraph 10.4

CHAPTER

2

GROUND HANDLING, SERVICING AND REPAIR

CHAPTER II

GROUND HANDLING, SERVICING AND REPAIR

CONTENTS

1. GENERAL

This chapter describes procedures for servicing and maintenance which are performed most frequently during routine operation, and is intended as a guide for mechanics engaged in routine and periodic inspections, repairs and servicing. For servicing and maintenance of systems not described in this chapter, see the respective chapter.

2. GROUND HANDLING

When handling the airplane on the ground, observe the following:

(1) Do not lock control surfaces while towing or taxiing the airplane.
(2) Do not set parking brake if brakes are overheated.
(3) Before starting engine.

(a) Remove all towing equipment.
(b) Head airplane into wind and chock wheels.
(c) All personnel, work stands and equipments shall be clear of dangerous area.
(d) Remove control surface locks.
(e) Set parking brake.
(f) Remove engine intake and exhaust covers.

2.1 JACKING (See Figure 2-1)

2.1.1 REQUIRED GROUND SUPPORT EQUIPMENT

(1)	Jack pad	FWD fuselage	GSE. 016A - 99050
		Wing	GSE. 016A - 99069-1, -2 (*1)
			GSE. 016A - 99069-31, -32 (*2)
(2)	Jack	Fuselage	Capacity more than 1,300 lbs. (600 kg)
		Wing	Capacity more than 8,800 lbs. (4,000 kg)

2.1.2 PROCEDURE

(1) Remove screws on the under surface of wings, outboard of each nacelle (W.S. 2560) and fuselage STA1275, and attach jack pads.

(2) Position jacks under the jack pads and, keeping airplane level, raise airplane until the landing gears leave the ground.

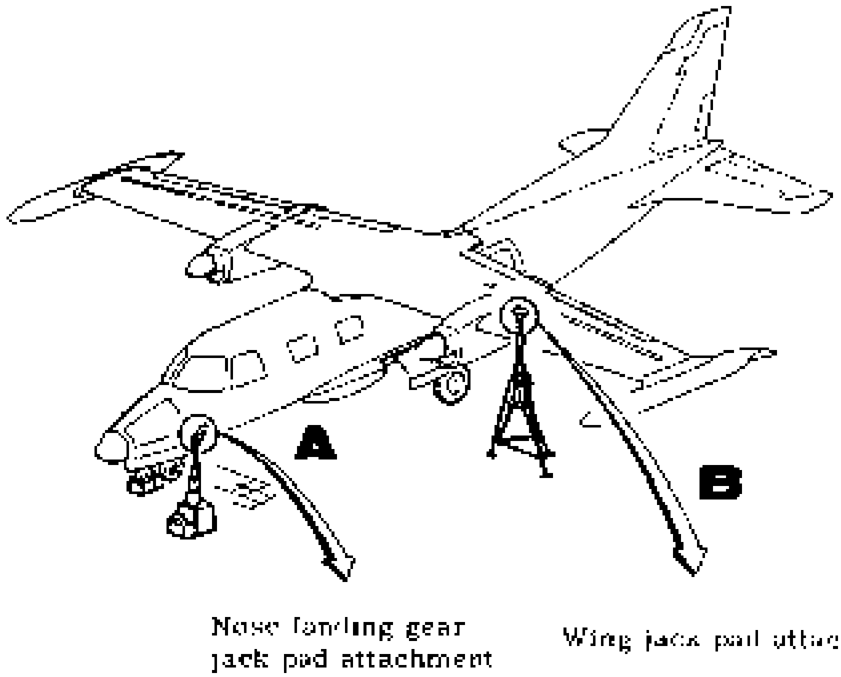
CAUTION

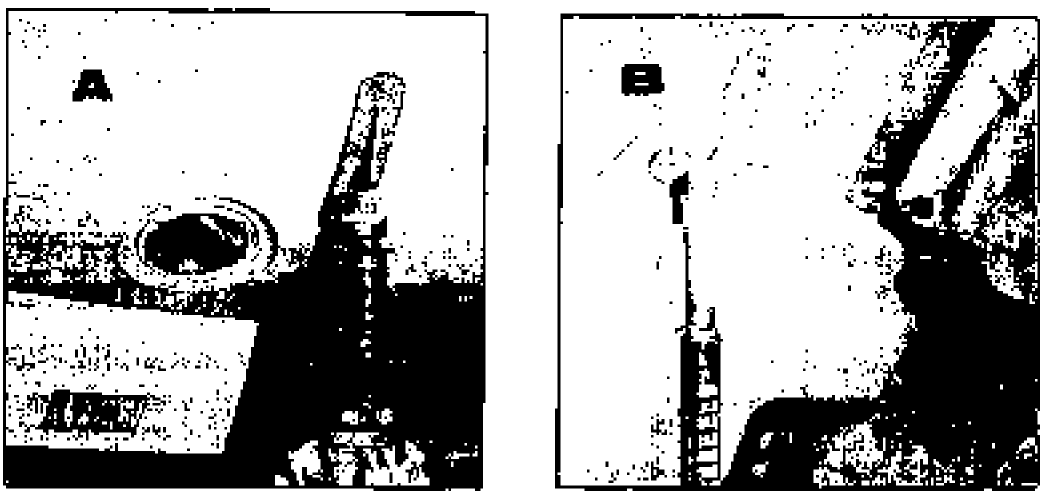
(i) Avoid jacking airplane outside hangar during gusty wind conditions.

(ii) Avoid jacking on the insecure ground where there is the possibility of airplane slipping off the jack.

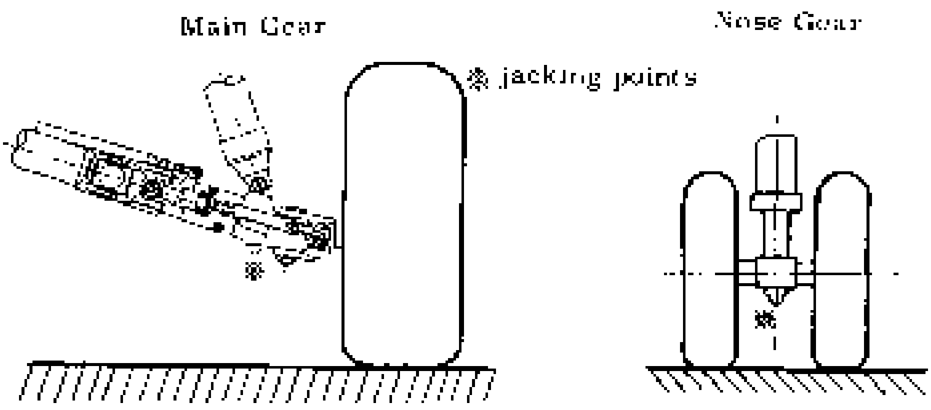
(iii) It is recommended to drain fuel tanks before jacking airplane. Especially it is desired to drain tip tanks less than half.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent





For installing or removing nose and main wheels, and replacing tires, jacking is performed by means of small box jacks placed in the respective jacking points.



ORIGINAL
As Received By
ATP

Fig 2 - 1 Jacking

2.2 MOORING (See Figure 2-2)

Mooring procedures are variable for anticipated wind speed ranges of less than 20 kt (23 mph), 20 kt to 60 kt (23 to 69 mph) and above 60 kt (69 mph).

2.2.1 REQUIRED GROUND SUPPORT EQUIPMENT

(1)	Gust lock plate	Rudder	GSE. 016A - 99006
		Elevator	GSE. 016A - 99007
(2)	Cover	Engine air intake	GSE. 016A - 99010
		Engine tail pipe	GSE. 016A - 99011
		Pitot tube	GSE. 016A - 99015
		Windshield	GSE. 030A - 99016
(3)	Wheel chock		GSE. 016A - 99021

2.2.2 PROCEDURES

(1) When wind speed is less than 20 kt (23 mph).

(a) Raise flaps, lock rudder and elevator with gust locks, set trim tabs in neutral position and set parking brake.

(b) Chock the main wheels fore and aft.

(c) Cover windshield, pitot tube, engine air intakes and tail pipes.

(d) Tie the nose wheel and tail skid.

(2) When wind speed is 20 to 60 kt (23 to 69 mph).

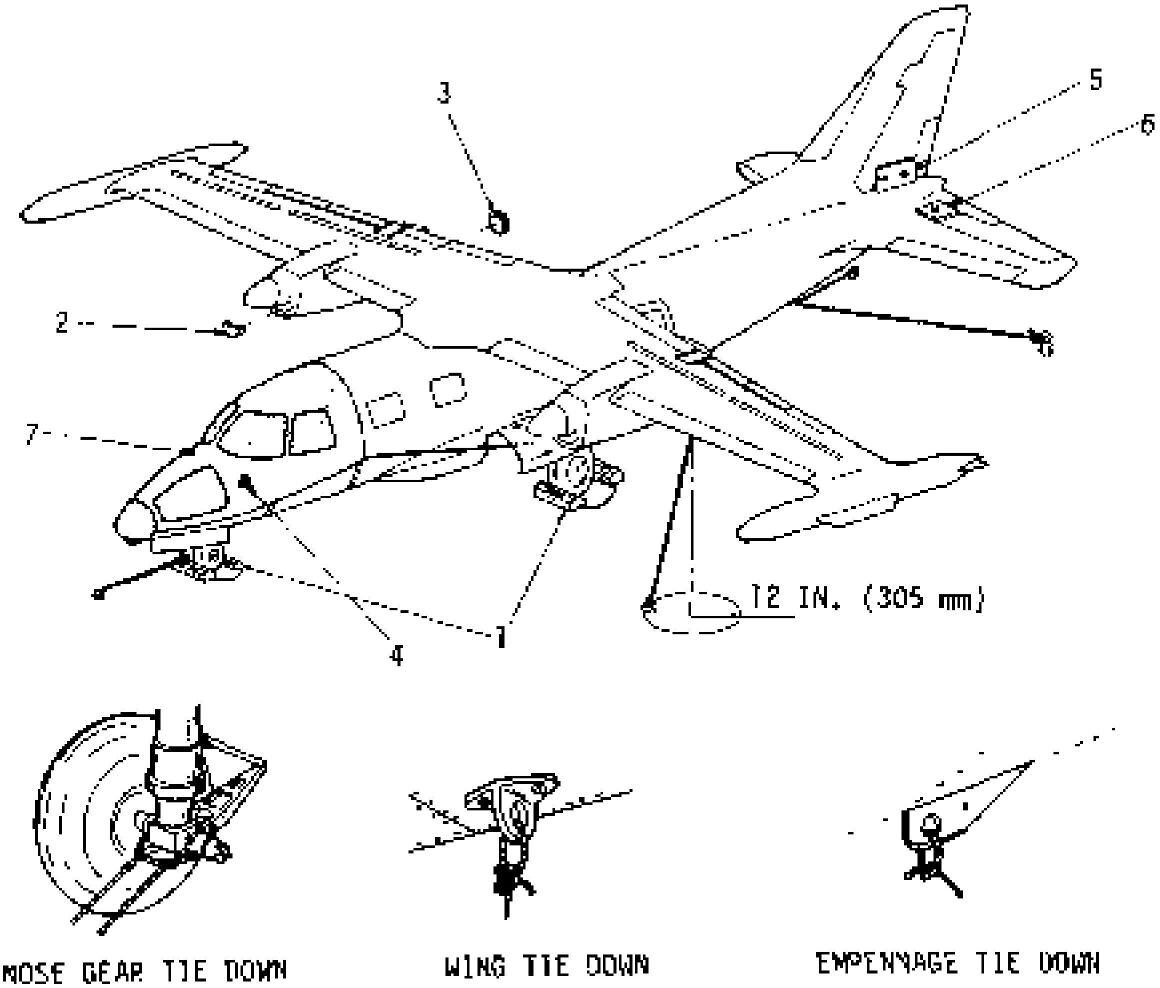
Tie down wings at W STA 4738.5.

(3) When wind speed is more than 60 kt (69 mph), hangar airplane.

CAUTION

(i) For tie down, use over 1/2 in. (13 mm) manila rope or rope having over 2,200 lbs. (1,000 kg) breaking strength. For mooring procedures, see Fig. 2-2.

(ii) When strong wind is anticipated, head airplane into wind and maintain sufficient distance from neighboring airplanes.



1. WHEEL CHOCKS
2. ENGINE INTAKE PLUG
3. ENGINE EXHAUST PLUG
4. PITOT TUBE COVER
5. RUDDER GUST LOCK PLATE
6. ELEVATOR GUST LOCK PLATE
7. WINDSHIELD COVER (IF EQUIPPED)

Fig 2-2 Mooring

2.3 HOISTING

2.3.1 REQUIRED GROUND SUPPORT EQUIPMENT

(1)	Sling	Aft fuselage	GSE. 016A - 99046
		Wing	GSE. 016A - 99025 -11
		Fuselage	GSE. 016A - 99026 -11
		Complete airplane	GSE. 016A - 99027 -11
(2)	Crane	Capacity more than 22,000 lbs. (11 tons)	

2.3.2 PROCEDURE

(1) Empty fuel tanks.

(2) Hoist complete airplane : As illustrated in Fig 2-3
Hoist fuselage : As illustrated in Fig 2-5
Hoist wing : As illustrated in Fig 2-4
Hoist aft fuselage : As illustrated in Fig 2-6

(3) Hook the shackle of sling and hoist slowly by crane.

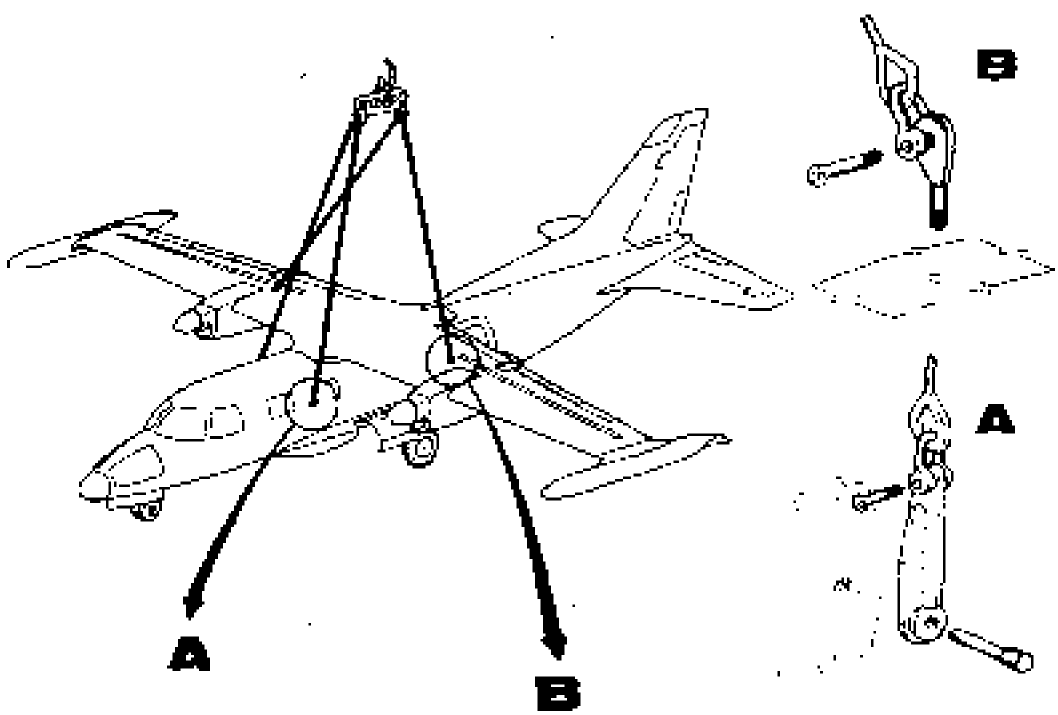


Fig 2 - 3 Hoisting

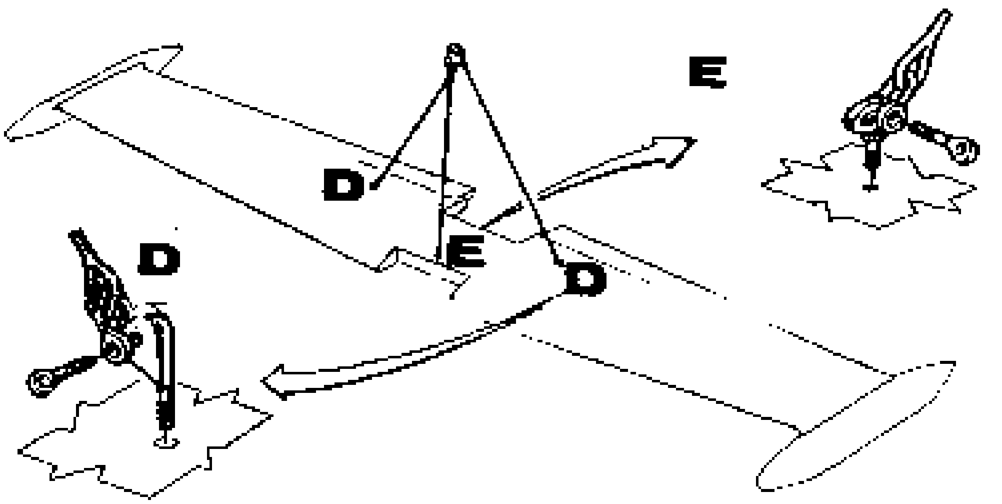


Fig 2 - 4 Hoisting wing

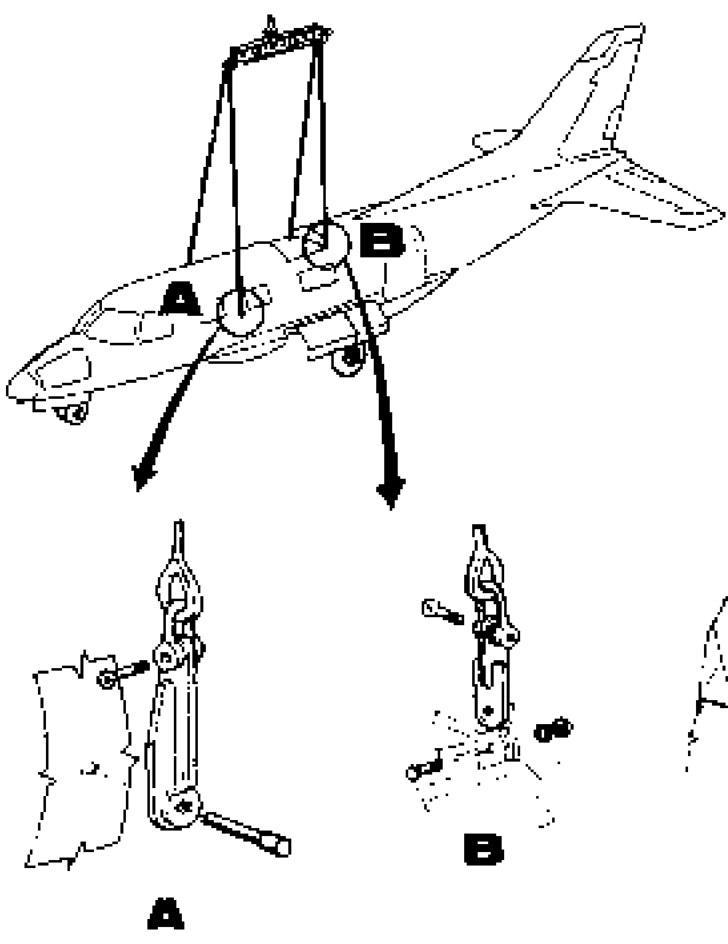


Fig 2 - 5 Hoisting fuselage

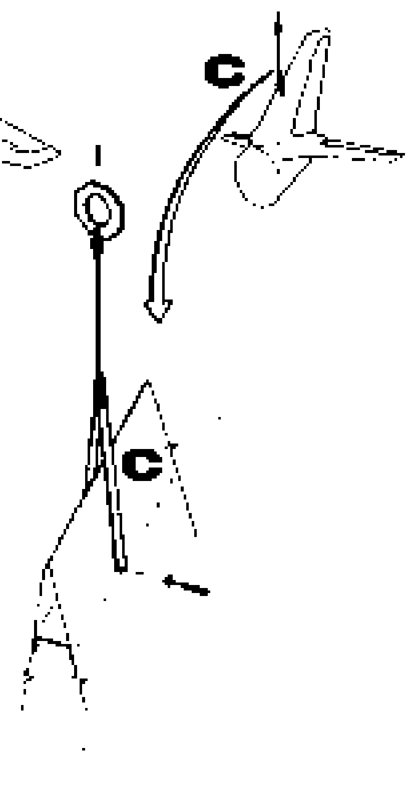


Fig 2 - 6 Hoisting aft fuselage

2.4 LEVELING (See Fig 2-7)

It may be necessary to level the airplane for various operations, i.e. weight and balance or replacement of major airframe components.
For leveling, hang a weight from the leveling clip attached to the pressure bulkhead channel (F. STA 8035) in the electric compartment and adjust the airframe so that the tip of the weight may come to the set point as shown in Fig. 2-7. Adjustment can be accomplished by either reducing tire air pressure or jacking at specified locations until the aircraft becomes level.

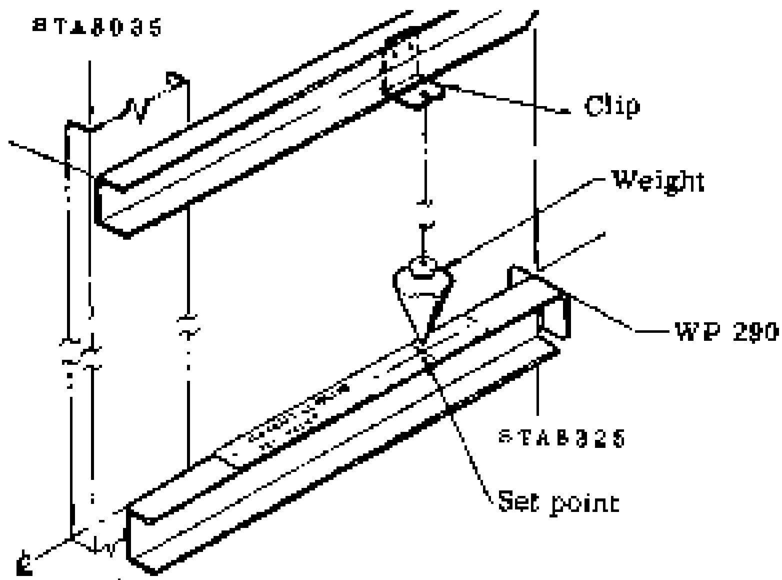


Fig. 2-7 Leveling procedure

2.5 TOWING

Towing is used to move airplane on the ground without running the engines. Minimum crew required for towing airplane is as follows:

One operator - tractor
One operator in cockpit, to handle brake as necessary
Two guides near wing tips, watching right and left side of the wing tips respectively and rearward of tail section of the airplane, when towing near obstacles.

2.5.1 REQUIRED GROUND SUPPORT EQUIPMENT

(1) Tow bar : GSE 016A-99078-21 or equivalent

(2) Tow tractor

2.5.2 PROCEDURE

(1) Make sure that airplane is clear of obstacles, ground support equipment, etc.

(2) Remove wheel chocks and check for normal tire pressure.

CAUTION

Before towing, disconnect nose landing gear torque arm connection. Re-connect after towing.

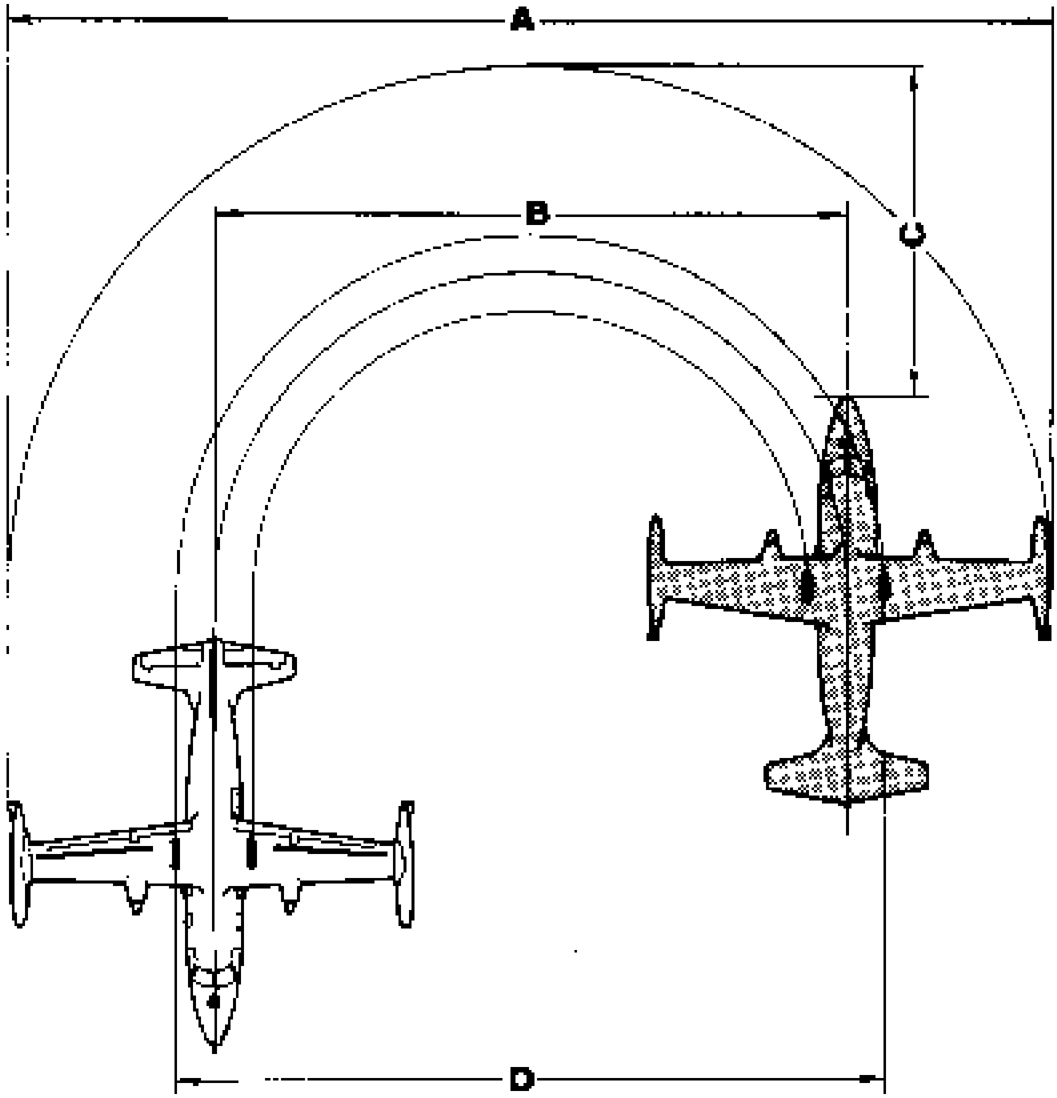
(3) Attach the tow bar in the holes in the nose wheel axle.

(4) Avoid jerky motion with tow tractor. Tow tractor speed should not exceed 5 MPH.

(5) If airplane is turned at a sharp angle, tires of main landing gear will drag. Therefore, avoid sharp turns. Tow airplane with turning radius as great as possible.

CAUTION

Tail skid should not be used for towing. Never push, pull or lift airplane by control surfaces, tail cone, nose of fuselage, nacelles or pitot tube.



DISTANCE	LEFT ft.(m)	RIGHT ft.(m)	EFFECTIVITY
A	114.5(34.9) 107.6(32.8)	104.0(31.7) 110.9(33.8)	*1 *2
B	75.5(23.0) 68.2(20.8)	65.0(19.8) 71.5(21.8)	*1 *2
C	38.7(11.8) 35.1(10.7)	33.5(10.2) 36.8(11.2)	*1 *2
D	83.0(25.3) 76.1(23.2)	72.8(22.2) 79.4(24.2)	*1 *2

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

Figure 2-8 Minimum Turning Distances

2.6 TAXIING

The nose steering system assists in turning the airplane. The steering angle is 24° right and 21° left (*1), 22° right and 23° left (*2). A more effective turn is accomplished by applying the main wheel brake.

CAUTION

Do not operate the engines at high power, when taxiing on sandy or rough ground.
Damage to the propeller blades may result.

2.7 PARKING

Parking is defined as leaving the aircraft for a short period of time when there is no possibility of immediate change in weather. Select the area as level as possible and located near the fire-fighting equipment and the ground crew when parking.

2.7.1 REQUIRED GROUND SUPPORT EQUIPMENT

(1) Gust lock pin	GSE. 016A-99005-11
(2) Wheel chocks	GSE. 016A-99021
(3) Pitot cover	GSE. 016A-99015
(4) Ground wire	One

2.7.2 PROCEDURE

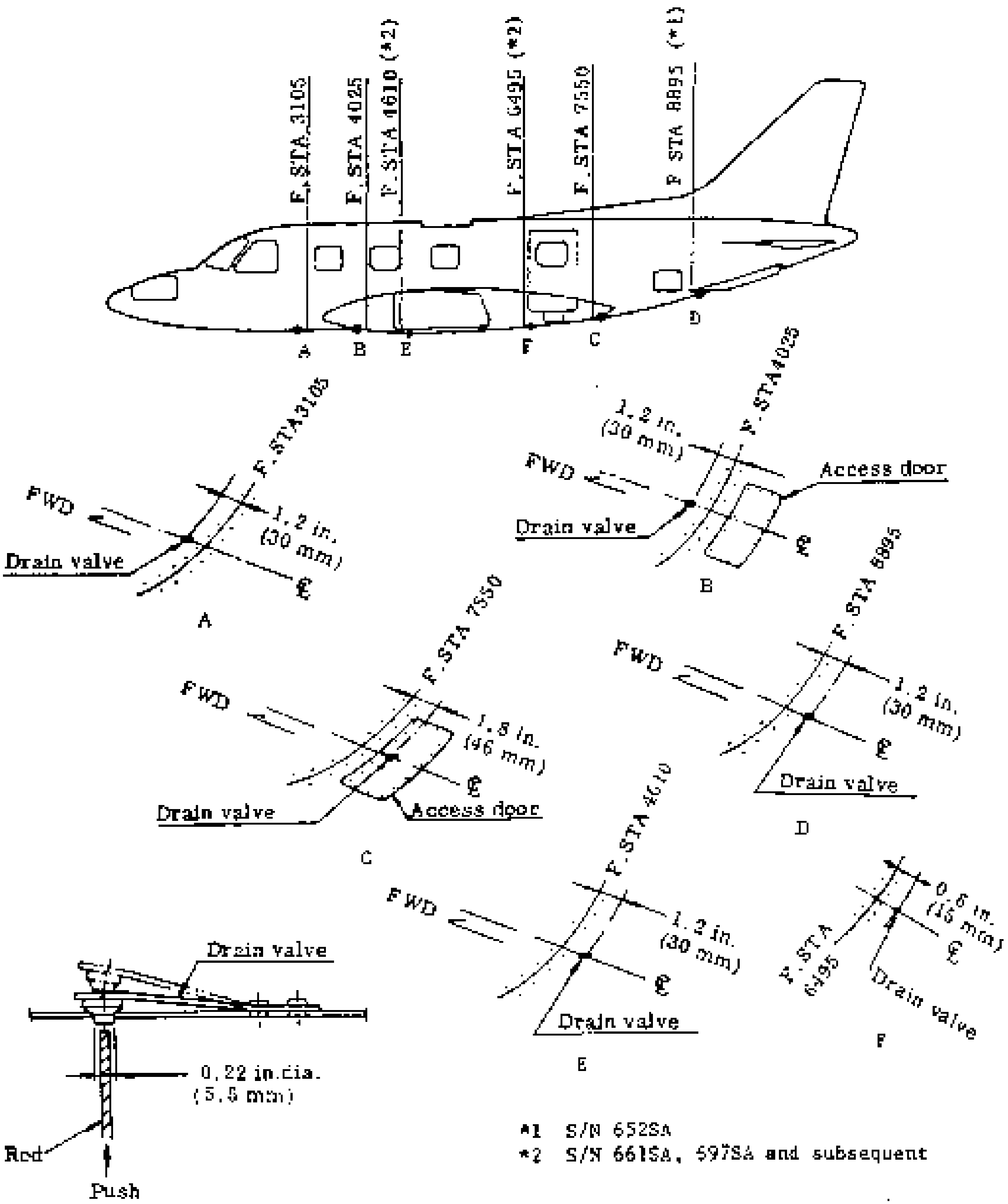
(1) Place the aircraft in the parking area at a sufficient distance from other airplanes and with ample space left for fire-fighting and servicing activities. On a windy day, park the aircraft nose to windward.
(2) Install gust lock pin.
(3) Set parking brake.
(4) Ground the aircraft from the under surface of wing at W.S. 3165.
(5) Chock the wheels.
(6) Install pitot tube cover as necessary.
(7) Close doors.

CAUTION

Do not set parking brake in freezing weather. In case of overheated brake, set the brake after cooling.

2.8 FUSELAGE DRAIN (See Fig 2-9)

The water drains are operated by pushing the valve up slightly with a rod, etc.



*1 S/N 652SA
*2 S/N 661SA, 697SA and subsequent

Fig 2-9 Drain valve

3. ENGINE GROUND RUN

Occasionally it becomes necessary to pull (turn) the engine through by hand. Turn in the direction of normal rotation only, due to possible damage to engine component parts.

Perform engine ground run in the following manner at the 100 hour periodic inspection, except Para 3.2 and 3.3 which shall be performed at time intervals specified in the applicable flight manual.

(1) Required ground support equipment

Wheel chock	GSE. 016A-99020 or Equivalent

(2) Preparation

(a) Set wheel chocks.
(b) Apply parking brake.
(c) Make sure that there are no safety hazards around airplane.

3.1 FEATHER VALVE TEST

Perform Feather Valve test in accordance with paragraph FEATHER VALVE CHECK in Section 5 "NORMAL PROCEDURES" of the applicable FAA Approved Airplane Flight Manual.

3.1A STARTING TEST

Start engine and confirm items per applicable FAA Approved Airplane Flight Manual Section "NORMAL PROCEDURES" paragraph "Before Starting Engines" and "Starting Engines."

3.2 NEGATIVE TORQUE SENSOR (NTS) TEST

Perform NTS test in accordance with paragraph NEGATIVE TORQUE SENSOR CHECK, "NORMAL PROCEDURES" Section of the applicable flight manual

3.3 OVERSPEED GOVERNOR (OSG) TEST

Perform OSG test in accordance with paragraph OVERSPEED GOVERNOR CHECK, "NORMAL PROCEDURES" Section of the applicable airplane flight manual

3.4 PROPELLER BLADE LOCK TEST

Perform propeller blade lock test in accordance with paragraph STARTING ENGINES step 14, "NORMAL PROCEDURES" Section, of the applicable airplane flight manual.

3.5 OPERATION TEST
Check the following items.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

Operating Item	Power Lever	Condition Lever	Item to be checked
1. LOW SPEED TAXI	GROUND IDLE	TAXI	1. Ensure engine speed is 64 to 66% RPM (*1), 76.5 to 78.5% RPM (*2). 2. Beta Light normally illuminated. 3. Oil pressure min. 40 psi. 4. Fuel pressure min. 15 psi. 5. No Caution Lights illuminated.
2. HIGH SPEED TAXI	GROUND IDLE	TAKEOFF-LAND	1. Engine speed must be 96 to 97% RPM. 2. Beta Light must be illuminated.
3. TAKEOFF	TAKEOFF POWER POSITION	TAKEOFF-LAND	1. Engine speed must be 99.5 to 100.5% RPM. 2. Oil pressure must be 70 to 120 psi. 3. Beta Light must be off.
4. REVERSE	REVERSE	TAKEOFF-LAND	1. Engine speed must be 94.5 RPM. 2. Beta Light must be illuminated.
5. ENGINE POWER LIMITING	TAKEOFF	TAKEOFF-LAND	1. Engine power must be limited at Torque 100% or ITT 923 ℃.

Perform this test with setting "CABIN AIR SELECT" switch to "OFF" or "RAM" position.

3.6 ANTI-ICING AND DE-ICING

(1) The loadmeter should deflect to operating range when the prop De-Ice switch is placed to ON position and the Loadmeter selector placed to the corresponding prop de-ice switch.

(2) The Oil Cooler Anti-Ice Indicator light should illumination when the anti-ice switch is placed to the ON position.

CAUTION

Oil Cooler Inlet Anti Icing system must not be operated on ground for more than 10 seconds.

(3) The Engine Intake Anti-Ice indicator should illuminate when the anti-ice switch is placed to the ON position.

CAUTION

Engine Intake Heat Switch must not be placed ON at an ambient temperature condition of 39°F (4°C) or above for more than 10 seconds.

3.7 CONTINUOUS IGNITION EQUIPMENT TEST (If installed)

(1) Set "CONTINUOUS IGNITION" switch of LH and RH engine to "ON", one at a time.

(2) Check to ensure igniter operates and "IGNITION" indicator light illuminates for the engine set to "ON".

3.8 NORMAL STOP TEST

(1) Shut down engine in accordance with applicable flight manual.

(2) Check the following items.
(a) COAST DOWN TIME - Approximately 50 to 60 seconds for each engine coast down time should be about the same (time from setting the RUN-CRANK-STOP switch to the STOP position until complete engine stops).
(b) Propeller stops in flat position by the action of propeller latch.

NOTE

Coast down time is apt to be affected by power lever handling time, environment winds, etc.
Therefore, when abnormal incidence is found in coast down time, determine whether the change is an incidental one, or is steady abnormality based on data of several operations. In the case of steady abnormality, troubleshoot propellers, and engines including engine accessories.

1. When abnormal incidence in coast down time occurs, perform the below confirmation/check items as follows.
   (1) Confirm whether abnormal sounds exist and/or whether the engines rotate smoothly, rotate propellers by hand after engines stop.
   (2) Check damage for first stage compressor, or contacts.
   (3) Check FOD, overheating or cracks in third stage turbine wheel.
   (4) Check damage or cracks in turbine rear support struts.
   (5) Check any evidence of metal spray inside tailpipe or at front surface of third stage turbine wheel blades.
   (6) Check any abnormal metal particles in oil filter.

2. If an abnormality is not detected in the confirmation/check, retry 3 to 5 more times to determine if the original abnormality is an incidental or steady condition.

3. In steady abnormality case of coast down time, troubleshoot propellers and engines including accessories.

3.9 UNFEATHER TEST

Shut down engine with propeller in feather condition and check the unfeather function as follows:

(1) Set condition lever at TAXI and power lever at REVERSE.

(2) Push Prop UNFEATHER button until propeller latches actuate.

(3) Propeller can be set to flat pitch.

NOTE

It is recommended that operating time of propeller unfeathering pump be limited as follows: (both type I and II Oil)
(1) Oil in cool condition Max 1 min.
(2) Oil in operating temperature Max 30 sec.
When the above limit time is over, turn propeller with hands or crank by starter to drain oil from oil sump in gear case until dip stick in oil tank shows FULL.

3.10 EMERGENCY STOP TEST

CAUTION

This test should be performed only when emergency stop is necessary for inspection. Take care of the following items.
(i) Perform this test after operation at ground idle condition for at least 3 minutes.
(ii) Prepare fire extinguisher and connect external power.

(1) Slowly move the CONDITION LEVER to EMERGENCY STOP while watching the fuel flow. Fuel flow should shut off prior to prop starting to feather. When fuel flow shuts off, depress the STOP BUTTON to purge the E P A . After the E P A is purged continue moving the CONDITION LEVER to feather the Prop.

CAUTION

If turbine temperature rises abnormally after operation of emergency stop, take the following action.
(i) Immediately place RUN-CRANK-STOP switch to "STOP" position, then allow it to return to "CRANK" position.
(ii) Return condition lever to TAXI.
(iii) Push UNFEATHER button and motorize engine by depressing start button.
(iv) After engine reaches normal shutdown temperature, shut electrical system OFF.
(v) Inquire into the cause for abnormal rise of turbine temperature.

(2) Place the following switches and circuit breaker OFF after engine stops completely.

FUEL MAIN switch

FUEL TRANSFER switch

FUEL BOOST PUMP circuit breaker

(3) Check the following items.
(a) Propeller stops in feather position.
(b) Turbine temperature does not rise rapidly after engine stops.

3.11 TURNAROUND

If engine restarts are anticipated in 10 to 45 minutes.

1. Park airplane into wind if possible.

2. Manually turn engine rotating group in direction of normal rotation occasionally to minimize thermal distortion.

NOTE

One blade width movement turns rotating group about 180°.

3. Continue these procedures until engine restart required.

CAUTION

Do not attempt to start an engine with thermal distortion. Stagnate acceleration may occur between the 18% to 28% rpm range accompanied by a rapid increase in ITT. Engine rotating group damage may occur.

4. STORAGE OF ENGINE

In regard to preservation and depreservation of engine, see Maintenance Manual for engine.

5. SERVICING

5.1 ENGINE OIL CHANGE AND REPLACEMENT OF FILTER ELEMENT

In regard to engine oil change and replacements of oil and fuel filter elements, see Maintenance Manual for engine.

5.2 LUBRICATION

Materials, handling procedures, cautions, applicable locations and servicing procedures and intervals are shown in the following diagrams and illustrations. Also use the "Maintenance Inspection and Maintenance Requirements" manual in conjuction with lubrication information data.

(1) Materials

Symbol	MIL Spec.	Nomenclature
GP	MIL-G-23827	ASG 7 (SHELL) Grease 27 (MOBIL)
GP-A	MIL-G-81322	ASG 22 (SHELL) Grease 28 (MOBIL)
GH	MIL-G-3545	ASG 5 (SHELL) ANDOK 260 (ESSO)
LGB	MIL-L-6086 GRADE L	Aero Shell Fluid (ASF) 5L (SHELL) Aerocraft Gear Oil EP Light (CALTEX) Aviation Gear Oil Light (ESSO) Royal Products ROYCO586V
OGP	MIL-L-7870	ASF 3 (SHELL) Aviation Instrument Oil (MOBIL) Braycote #363
OHA	MIL-H-5606	ASF 4 (SHELL) UNIVIS J-43 (ESSO) Aero Hydraulic Oil HFA (MOBIL)
RX	MIL-G-21164	Roex 65 (MOLYCOTE) ASG 17 (SHELL) Grease Special (MOBIL)
OPS	VV-L-800	VV-L-800 (SHELL) Petrotect 800 (Pennsylvania)
DC4	MIL-S-8660B	Dowcorning 4 compound or Toresilicone SH 4 compound KS-64 (Shinetsu Chemical)
FGB	MIL-L-7808G	Lubrication Oil, Aircraft Turbine Engine Synthetic Base
TES	MIL-L-23699	Exxon 2380, 2392 (Exxon) Royco 899 (Royal Lubri), Brayco 899 (Bray oil) Shell Aircraft turbine oil 551 (Shell)

(2) Handling procedures

Keep lubricant in a closed container. Be sure that quality of lubricant in small container, for frequent use, can be easily identified.

(3) Cautions
a. Clean lubricated surface with clean solvent before lubricating.
b. Keep quantity of lubricant applied to a minimum.
c. Wipe off excessive lubricant around press in type fitting.
d. Mixing of different greases is to be avoided.
If previously used grease is no Longer available, completely clean and relube unit with another authorized grease.

(4) Symbols used in figures.


With hand


With oil can

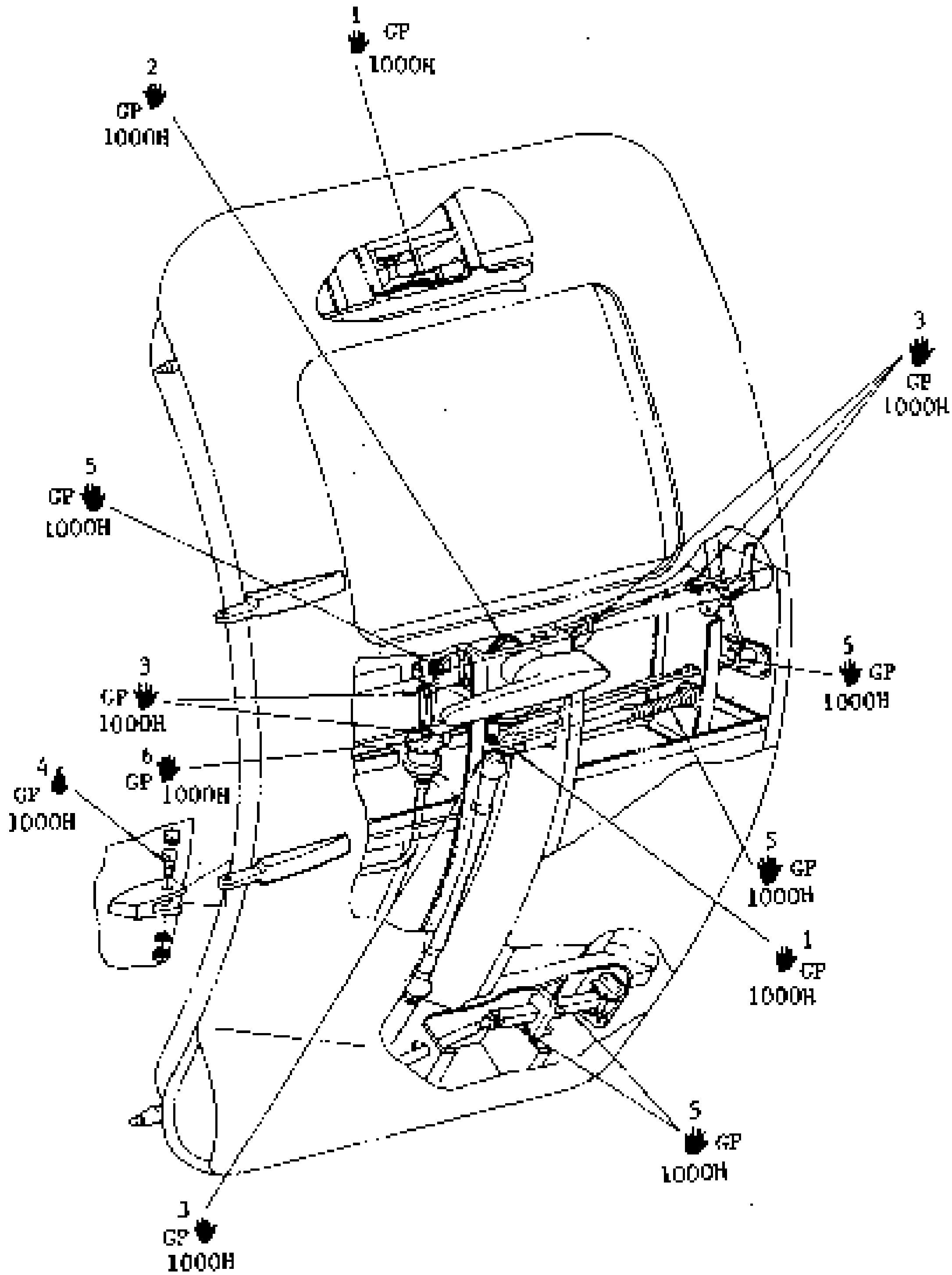

With grease gun

NOTE

Should the time intervals not be defined clearly at the lubrication points, refer to the applicable MAINTENANCE INSPECTION AND MAINTENANCE REQUIREMENTS Manual.

LUBRICATION CHART

FIGURE	ITEM
2-10	Passenger entrance door
2-11	Emergency exit door
2-12	Nose landing gear door
2-13	Nose landing gear, steering and door mechanism
2-14	Main landing gear
2-15	Landing gear retracting mechanism
2-16	MLG door mechanism
2-17	MLG door emergency release mechanism
2-18	Control column
2-19	Cable seal
2-20	Rudder pedal
2-21	Elevator aft quadrant
2-22	Rudder trim tab actuator and hinge
2-23	Elevator trim tab actuator and hinge
2-24	Flap stop assembly
2-25	Flap main gear box
2-26	Flap inboard auxiliary actuator
2-27	Flap main actuator
2-28	Flap outboard auxiliary actuator
2-29	Outboard flap guide link fitting
2-30	Outboard flap guide link mechanism
2-31	Trim aileron push rod
2-32	Spoiler feel spring
2-33	Center pedestal assembly
2-34	Engine accessories
2-35	Engine control mechanism
2-35A	Engine control torque tube/pulley
2-36	Propeller
2-37	Air conditioning and refrigeration unit
2-38	Flight chair



1. Rod end (10 places)
2. Gear (1 place)
3. Rod joint (9 places)
4. Door hinge (2 places)
5. Bracket (9 places)
6. Pressure rod (1 place)

Fig. 2-10 Passenger entrance door

1. Yoke (2 places)
2. Latch pin (1 place)
3. Lock pin (1 place)
4. Joint (2 places)
5. Bolt (2 places)

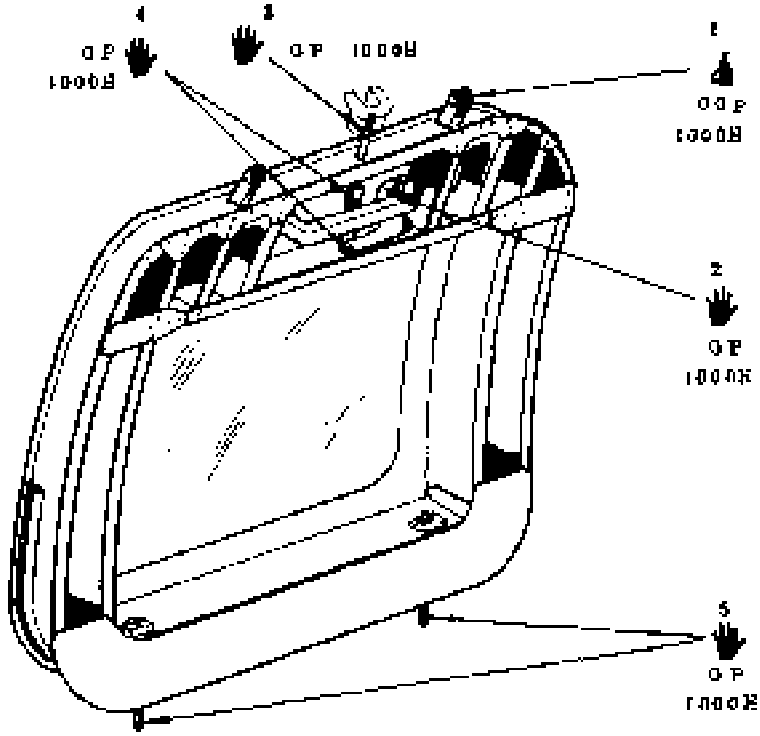


Fig 2-11 Emergency exit door

1. Hinge (6 places)

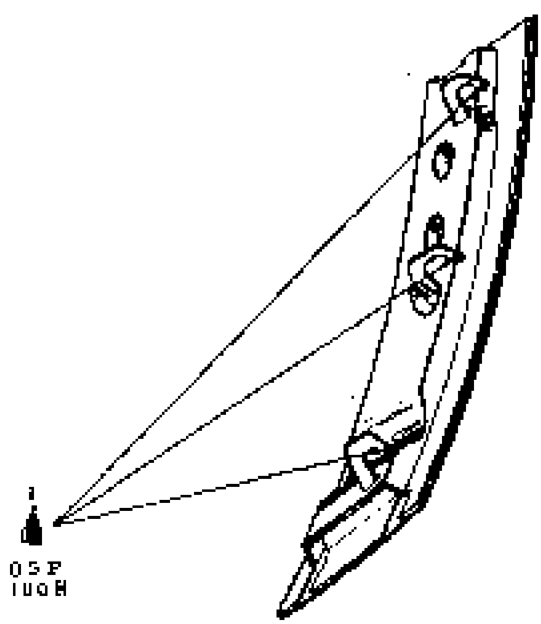
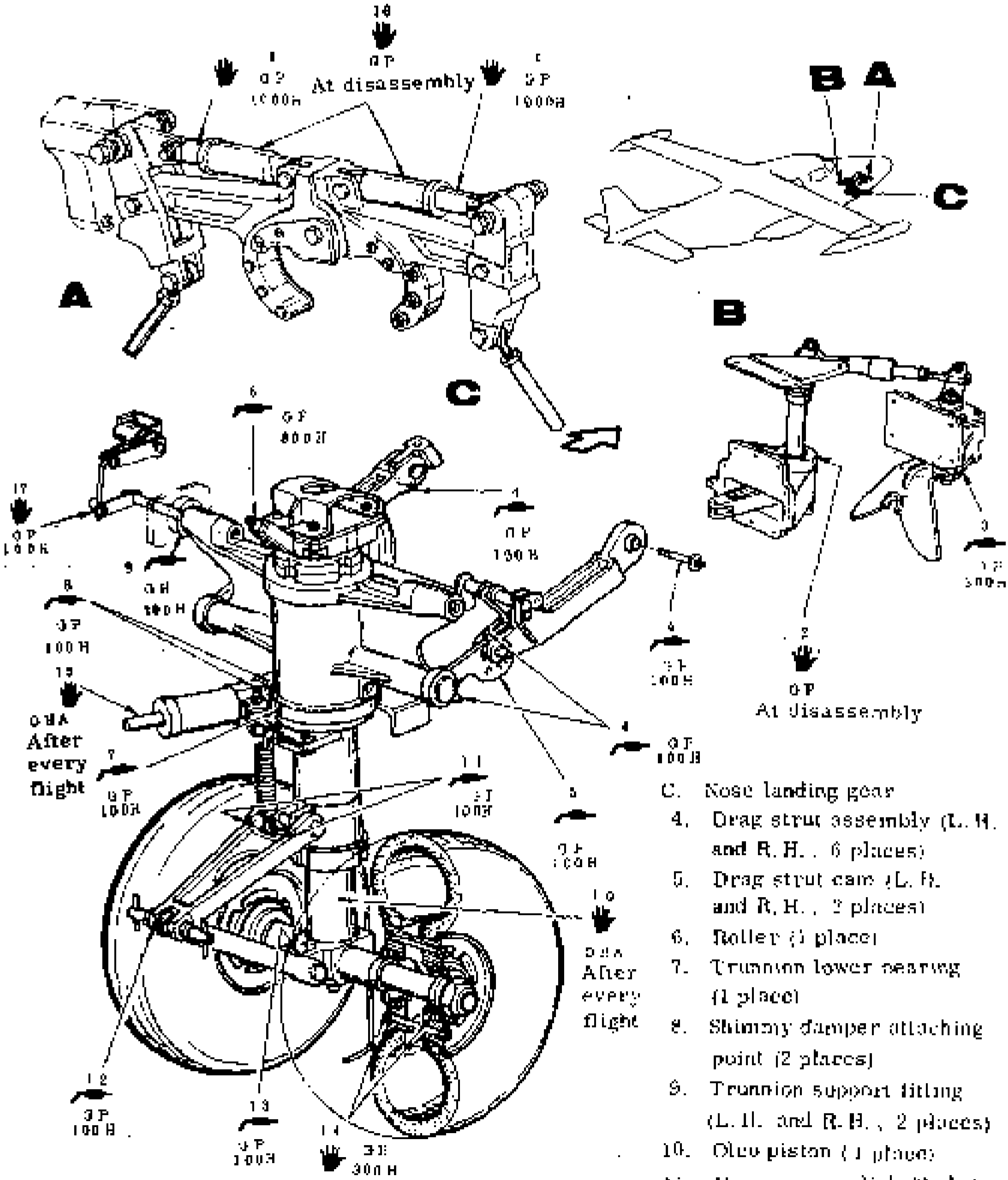


Fig 2-12 Nose landing gear door

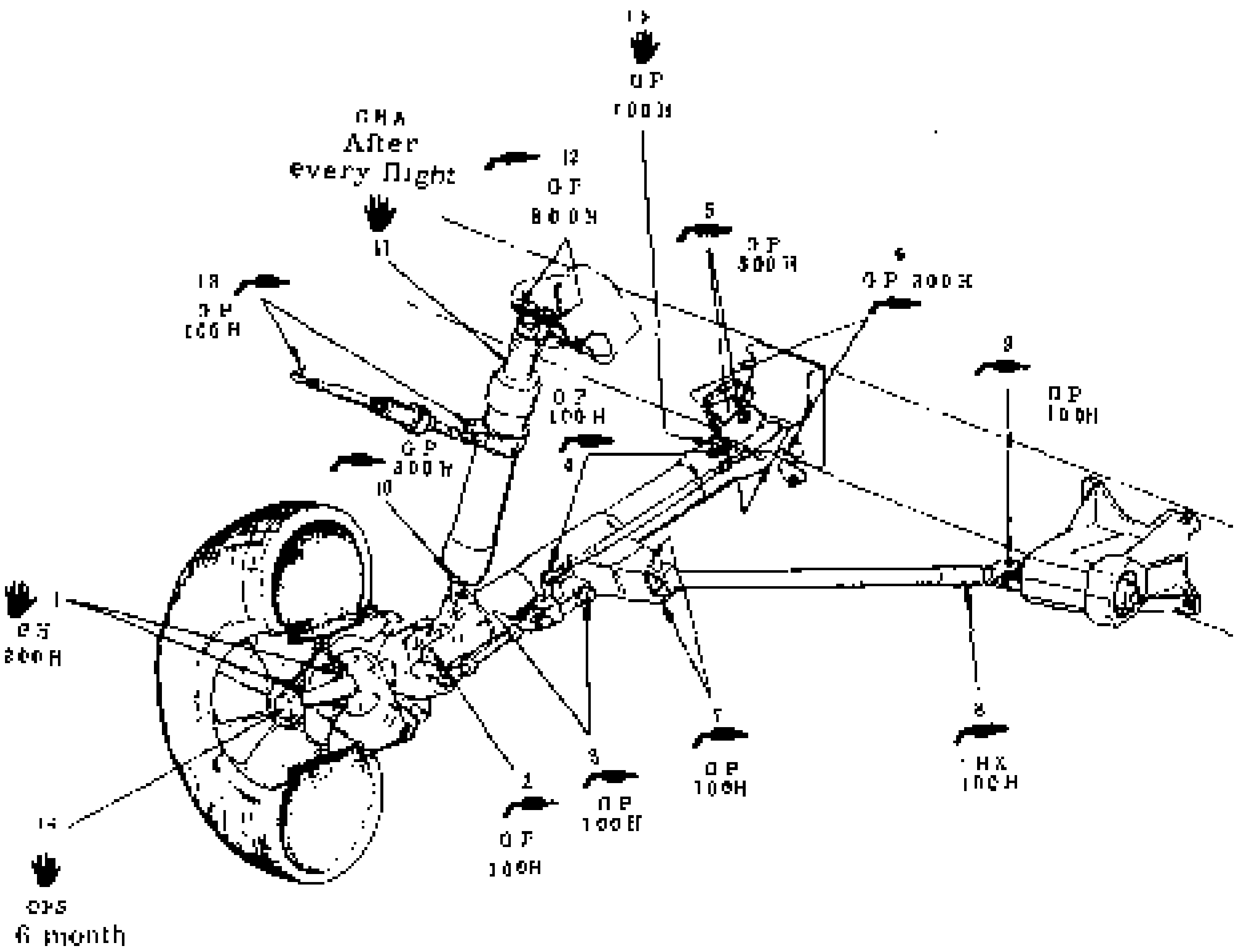


C. Nose landing gear
4. Drag strut assembly (L.H. and R.H., 6 places)
5. Drag strut cam (L.H. and R.H., 2 places)
6. Roller (1 place)
7. Trunnion lower bearing (1 place)
8. Shimmy damper attaching point (2 places)
9. Trunnion support fitting (L.H. and R.H., 2 places)
10. Oleo piston (1 place)
11. Upper torque link (2 places)
12. Upper and lower torque link joint (1 place)
13. Lower torque link (2 places)
14. Wheel bearing

A. Nose landing gear door mechanism
1. Spring cartridge (sliding surface, 2 places)

B. Steering mechanism
2. Crank shaft (1 place)
3. Cam roller (1 place)
15. Shimmy damper piston outer surface (2 places)
16. Inner spring (2 places)
17. Pendulum (1 place)

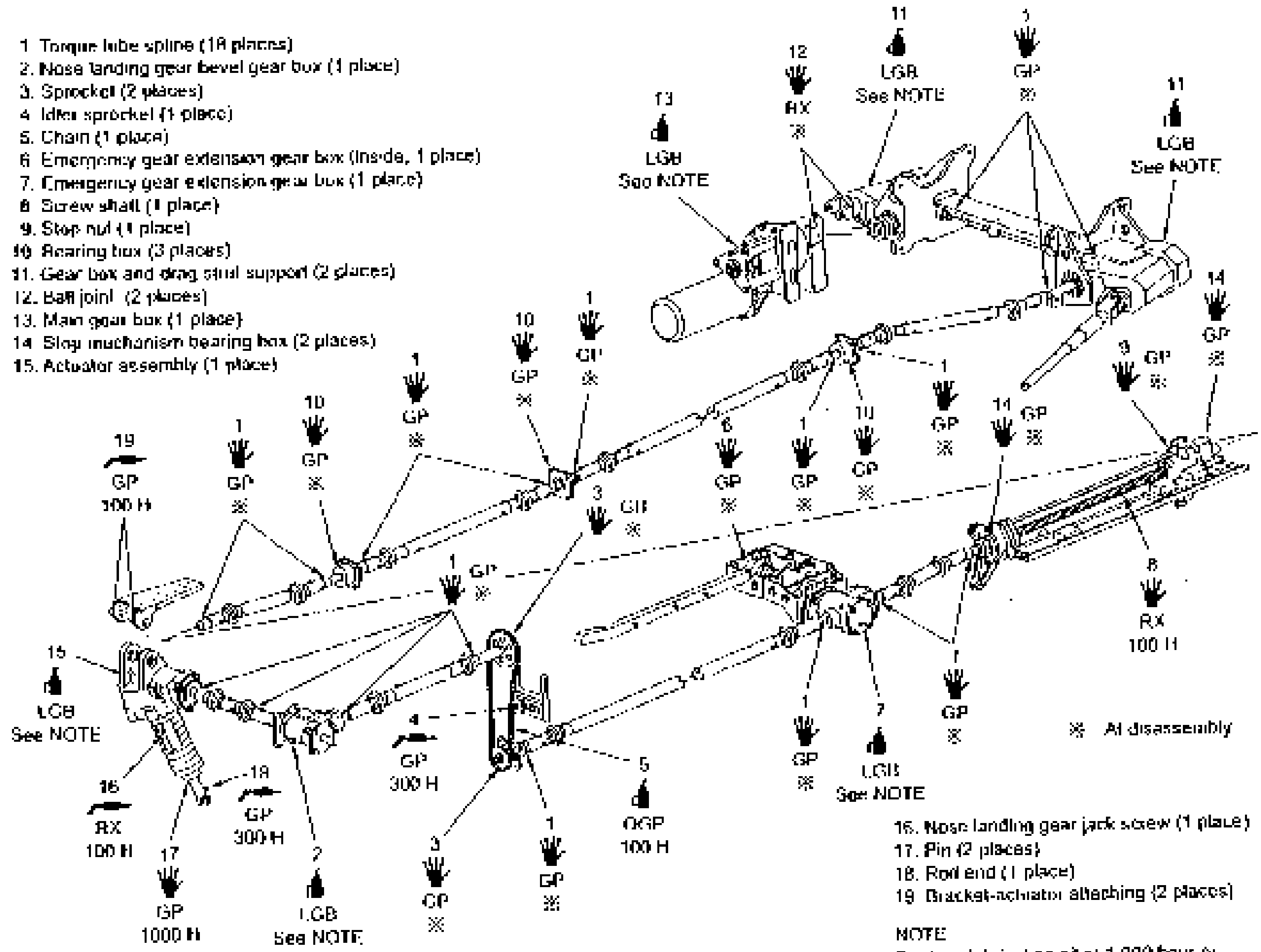
Fig 2-13 Nose landing gear, steering and door mechanism



1. Wheel bearing (4 places)
2. Leg end (4 places)
3. Position rod (lower, 6 places)
4. Position rod (upper, 4 places)
5. Leg pin (4 places)
6. Leg support fitting and pin (4 places)
7. Drag strut joint (4 places)
8. Drag strut (2 places)
9. Drag strut joint (4 places)
10. Oleo lower support fitting (2 places)
11. Oleo piston (clean with oil, 2 places)
12. Oleo attaching pin (4 places)
13. Door mechanism rod (4 places)
14. Wheel axle (cap inside, 2 places)
15. Pendulum (1 place)

Fig 2-14 Main landing gear

1. Torque tube spline (18 places)
2. Nose landing gear bevel gear box (1 place)
3. Sprocket (2 places)
4. Idler sprocket (1 place)
5. Chain (1 place)
6. Emergency gear extension gear box (inside, 1 place)
7. Emergency gear extension gear box (1 place)
8. Screw shaft (1 place)
9. Stop nut (1 place)
10. Bearing box (3 places)
11. Gear box and drag strut support (2 places)
12. Ball joint (2 places)
13. Main gear box (1 place)
14. Stop mechanism bearing box (2 places)
15. Actuator assembly (1 place)



※ At disassembly

16. Nose landing gear jack screw (1 place)
17. Pin (2 places)
18. Rod end (1 place)
19. Bracket-actuator attaching (2 places)

NOTE
Replace lubricating oil at 1,000 hour or 3 year intervals whichever comes first.

Fig. 2-15 Landing gear retracting mechanism

maintenance manual

TEMPORARY REVISION NO.2-1

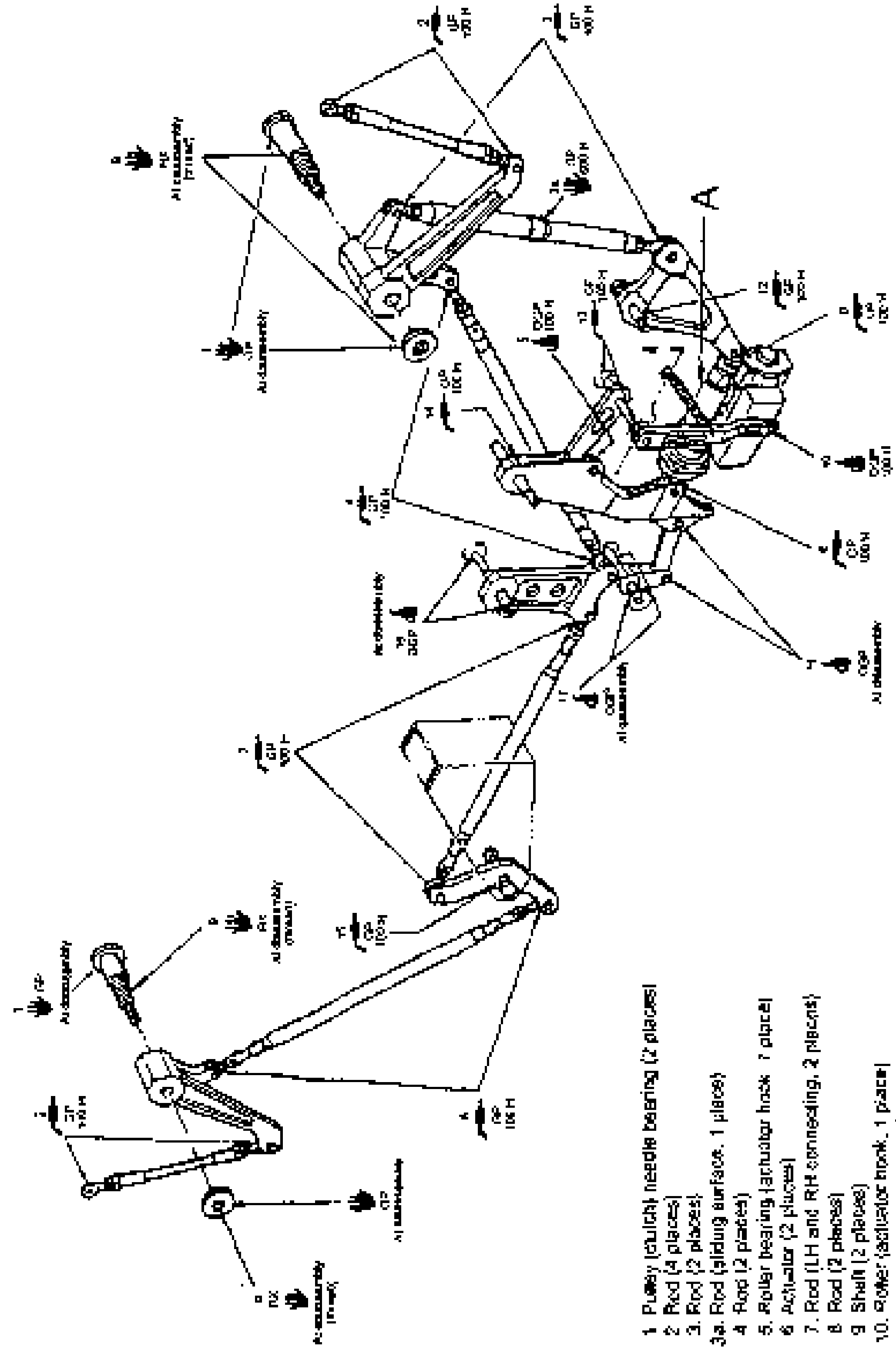
This Temporary Revision No. 2-1 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	2-24
MU-2B-60	MR-0336	2-24

Insert facing the page Indicated above for the applicable Maintenance Manual.
Retain this Temporary Revision until such time as a permanent revision on this subject is issued.

REASON : To add lubrication of safety pin rod.

CHANGE : Fig 2-16 (1/2) as follows.



1. Pulley (clutch) needle bearing (2 places)
2. Rod (4 places)
3. Rod (2 places)
3a. Rod (sliding surface, 1 place)
4. Rod (2 places)
5. Roller bearing (actuator hook, 1 place)
6. Actuator (2 places)
7. Rod (LH and RH connecting, 2 places)
8. Rod (2 places)
9. Shaft (2 places)
10. Roller (actuator hook, 1 place)
11. Bolt (link pin, 2 places)
12. Link (1 place)
13. Bearing (actuator hook, 2 places)
14. Link (1 place)
15. Link (1 place)
16. Bearing (shaft, 2 places)
17. Bearing (shaft, 2 places)

Fig. 2-16 MLG door mechanism(1/2)

1. Pulley (clutch) needle bearing (2 places)
2. Rod (4 places)
3. Rod (2 places)
4. Rod (2 places)
5. Roller bearing (actuator hook, 1 place)
6. Actuator (2 places)
7. Rod (L.H. and R.H. connecting, 2 places)
8. Rod (2 places)
9. Shaft (2 places)
10. Roller (actuator hook, 1 place)
11. Bolt (link joint, 2 places)
12. Link (1 place)
13. Bearing (actuator hook, 2 places)
14. Link (1 place)
15. Link (1 place)
16. Bearing (shaft, 2 places)
17. Bearing (shaft, 3 places)

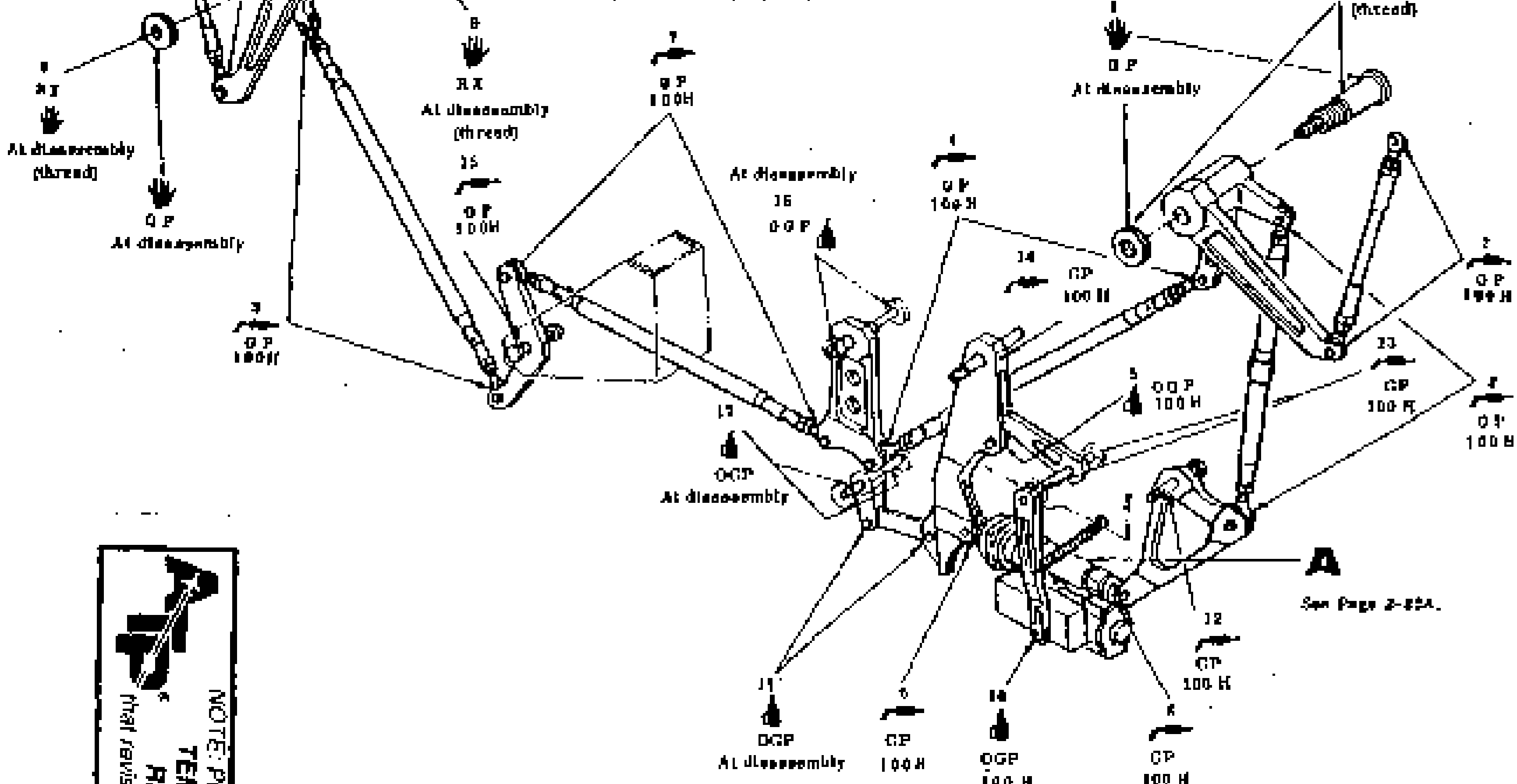
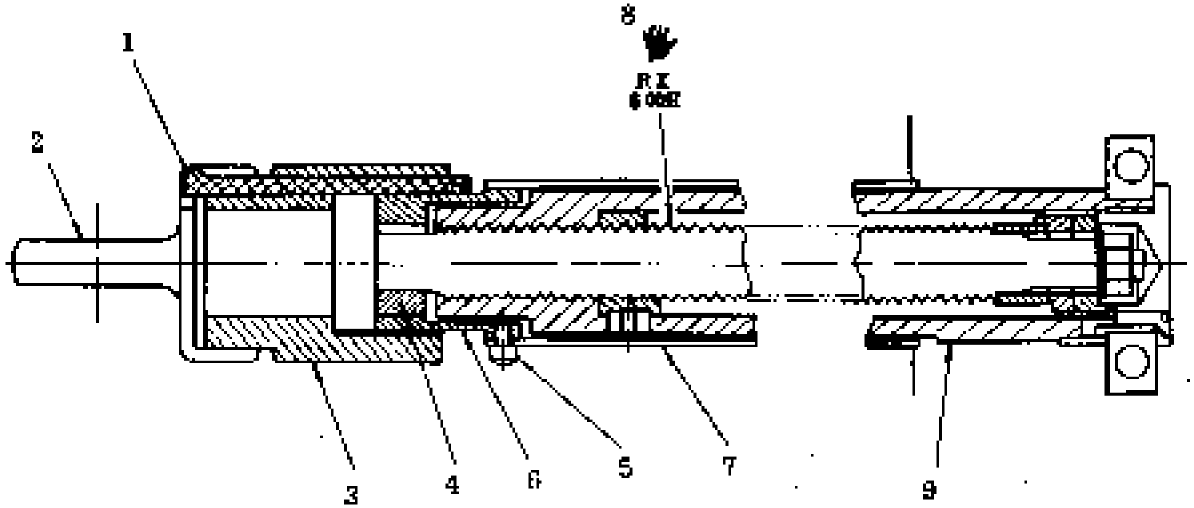


Fig. 2-18 MLG door mechanism (1/2)

NOTE: Please see the **TEMPORARY REVISION** that revises this page.



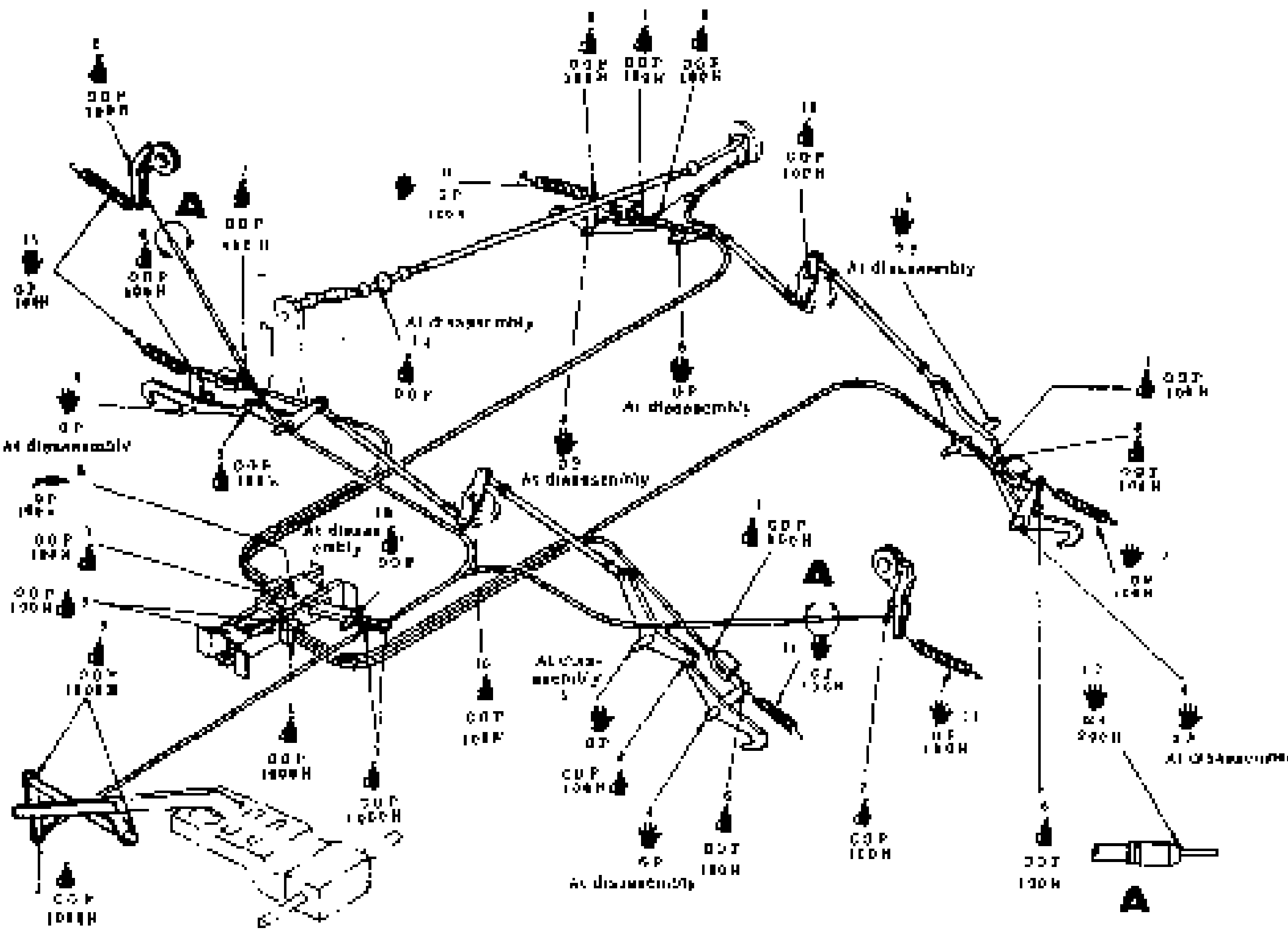
1. Key
2. End fitting
3. Nut
4. Key
5. Screw
6. Adaptor
7. Shield
8. Jack screw
9. Nut-jack screw (Ref)

Lubricating procedures :

1. Extend jack screw ⑧ fully with end fitting ②.
2. Remove screw ⑤.
3. Slide shield ⑦ to the direction of the arrow up to the end.
4. Lubricate jack screw ⑧.
5. Reassemble in reverse sequence of disassembly.

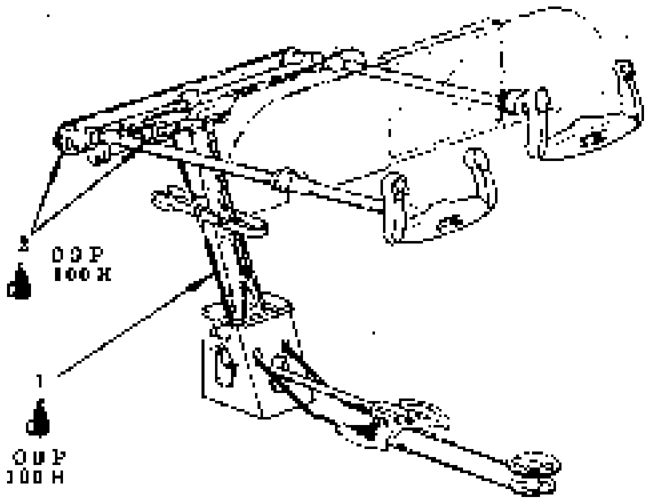
DETAIL **A**

Fig 2-16 MLG door mechanism (2/2)



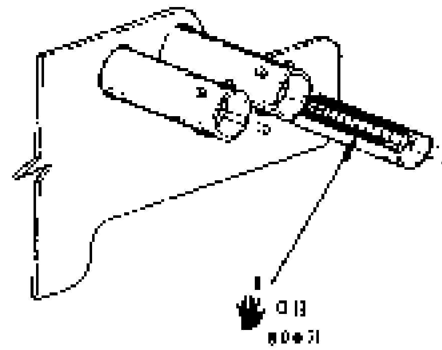
1. Cable terminal (15 places)
2. Pul[illegible] attaching point ([illegible] places)
3. Roller attaching point (6 places)
4. Door hook attaching shaft (4 places)
5. Lever attaching shaft (4 places)
6. Hook and rod attaching point ([illegible] 4 places)
7. Emergency [illegible] on lever (2 places)
8. L[illegible] attaching shaft ([illegible])
9. Spring pin attaching point (2 places)
10. Crank attaching shaft (2 places)
11. Spring (2 places)
12. Lever [illegible] (2 places)
13. Bolt (2 places)
14. [illegible]

Fig 2-17 MLG door emergency release mechanism



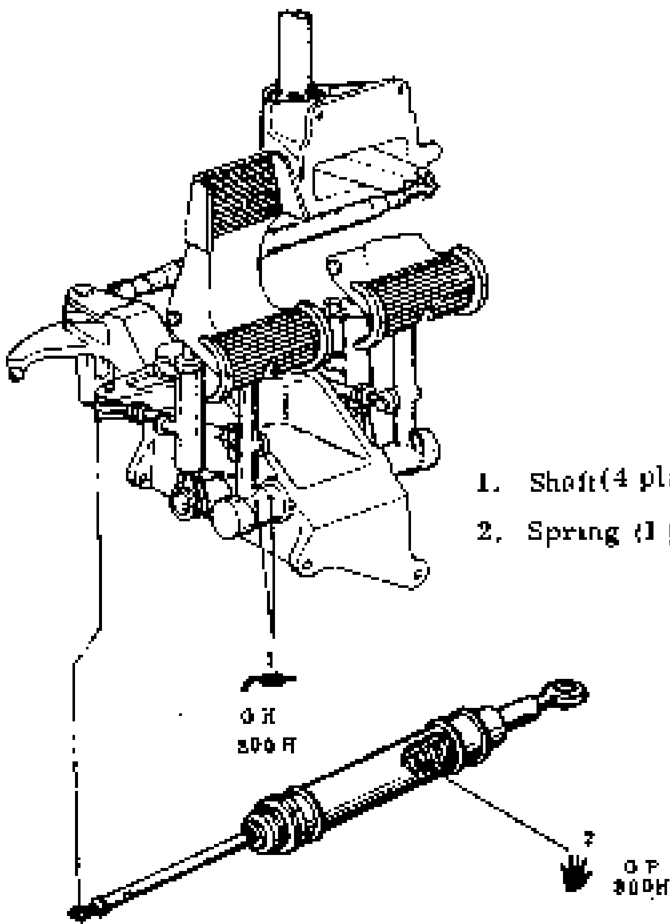
1. Chain (1 place)
2. Sprocket (4 places)

Fig 2-18 Control column



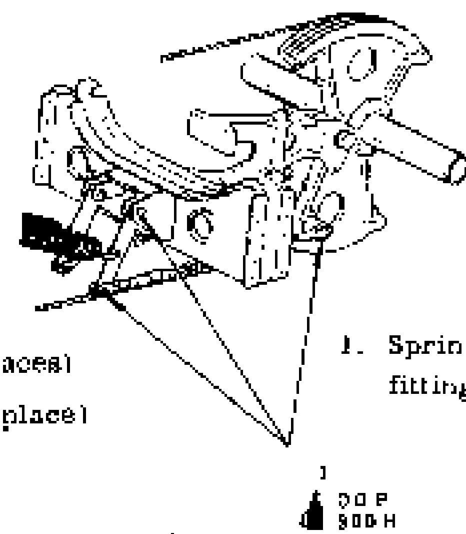
1. Cable seal (18 places)

Fig 2-19 Cable seal



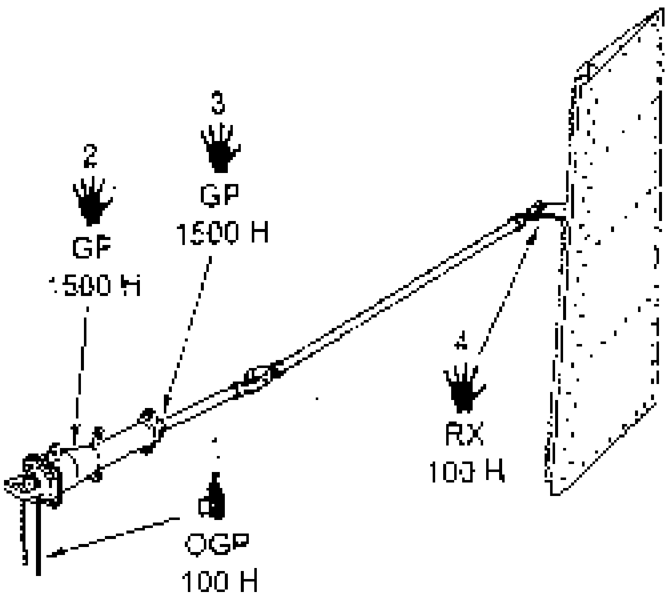
1. Shaft (4 places)
2. Spring (1 place)

Fig 2-20 Rudder pedal



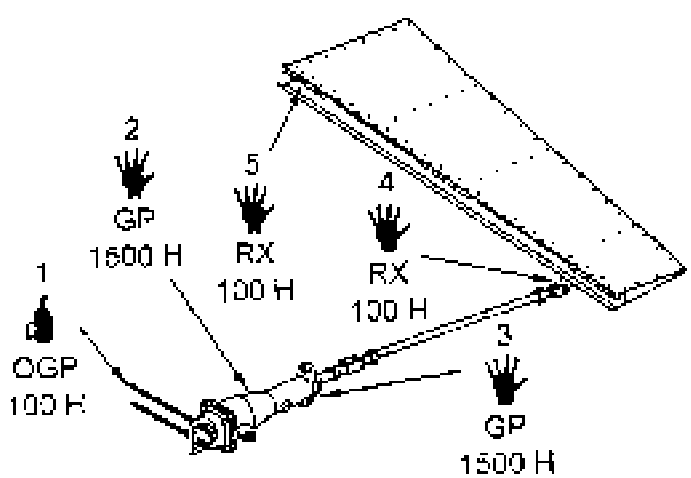
1. Spring end fitting (6 places)

Fig 2-21 Elevator aft quadrant



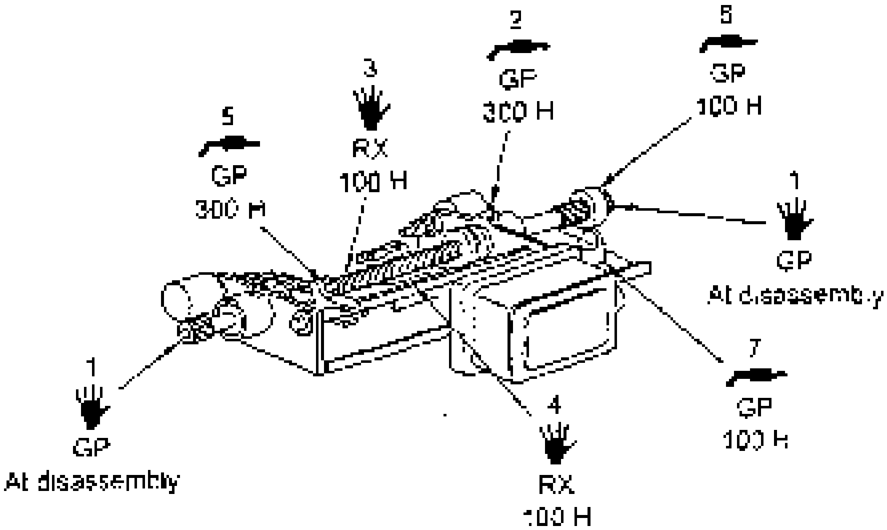
1. Chain (1 place)
2. Actuator (1 place)
3. Rod (sliding surface, 1 place)
4. Bush (1 place)

Fig 2-22 Rudder trim tab actuator and hinge



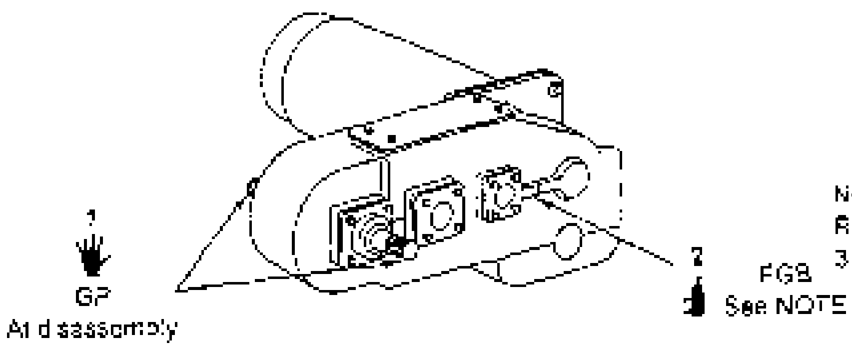
1. Chain (1 place)
2. Actuator (2 places)
3. Rod (sliding surface, 2 places)
4. Bush (2 places)
5. Hinge (8 places)

Fig 2-23 Elevator trim tab actuator and hinge



1. Serration shaft (2 places)
2. Bearing (2 places)
3. Screw shaft (1 place)
4. Guide (1 place)
5. Nut (1 place)
6. Cam (1 place)
7. Pin (1 place)

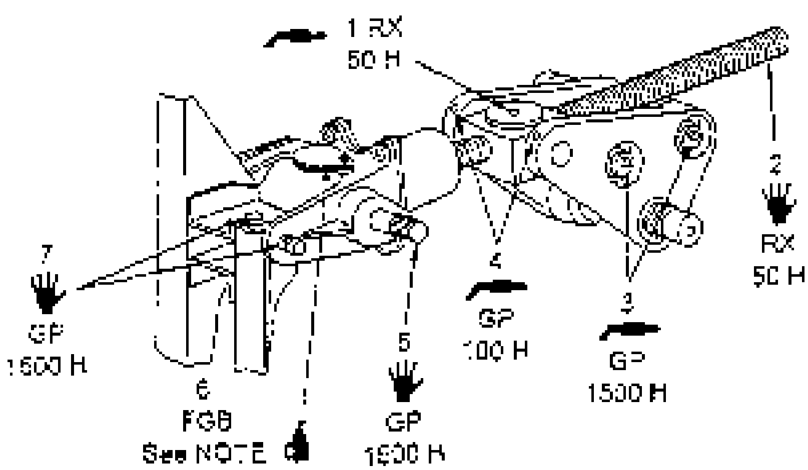
Fig 2-24 Flap stop assembly



1. Serration shaft (2 places)
2. Gear box (1 place)

NOTE
Replace lubricating oil at 1,000 hour or 3 year intervals whichever comes first.

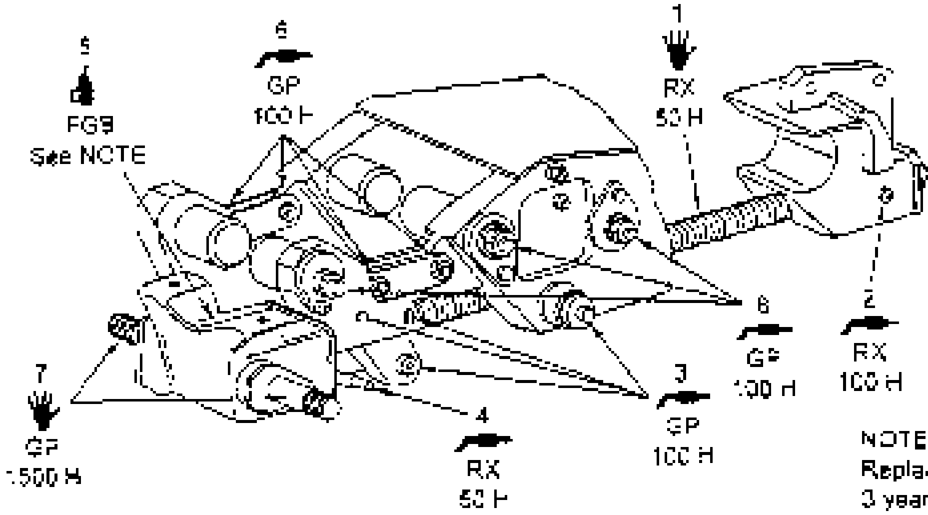
Fig 2-25 Flap main gear box



1. Jack nut (2 places)
2. Jack screw (2 places)
3. Pin (4 places)
4. Bolt (4 places)
5. Serration shaft (4 places)
6. Gear box (2 places)
7. Bolt and pin (4 places)

NOTE
Replace lubricating oil at 1,000 hour or 3 year intervals whichever comes first.

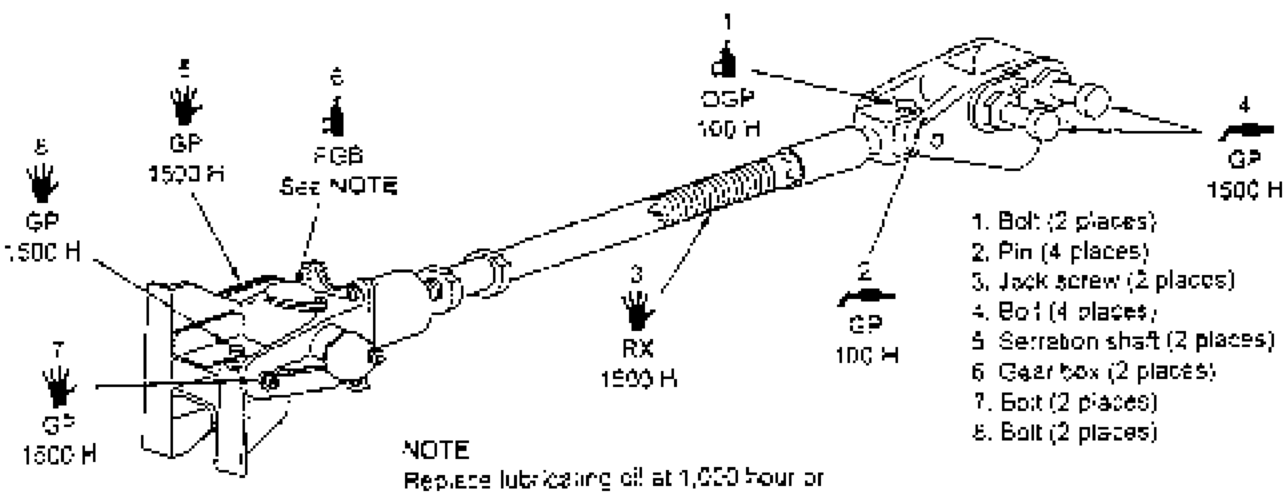
Fig 2-26 Flap inboard auxiliary actuator



1. Jack screw (2 places)
2. Jack support (2 places)
3. Pin and nut (16 places)
4. Jack nut (2 places)
5. Gear box (2 places)
6. Pin (5 places)
7. Serration shaft (4 places)
8. Roller support (12 places)

NOTE
Replace lubricating oil at 1,000 hour or 3 year intervals whichever comes first.

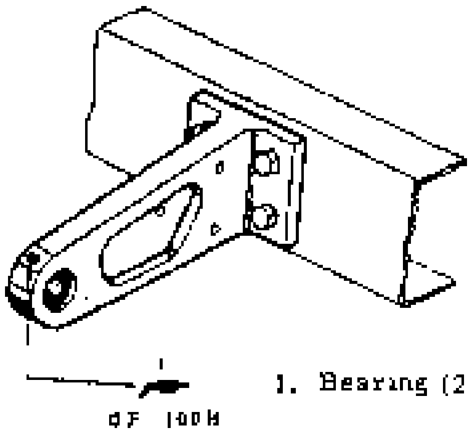
Fig 2-27 Flap main actuator



1. Bolt (2 places)
2. Pin (4 places)
3. Jack screw (2 places)
4. Bolt (4 places)
5. Serration shaft (2 places)
6. Gear box (2 places)
7. Bolt (2 places)
8. Bolt (2 places)

NOTE
Replace lubricating oil at 1,000 hour or 3 year intervals whichever comes first.

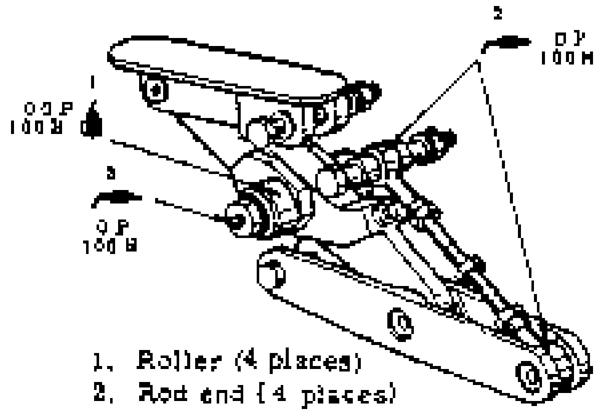
Fig 2-28 Flap outboard auxiliary actuator



1. Bearing (2 places)

Outboard flap guide link fitting

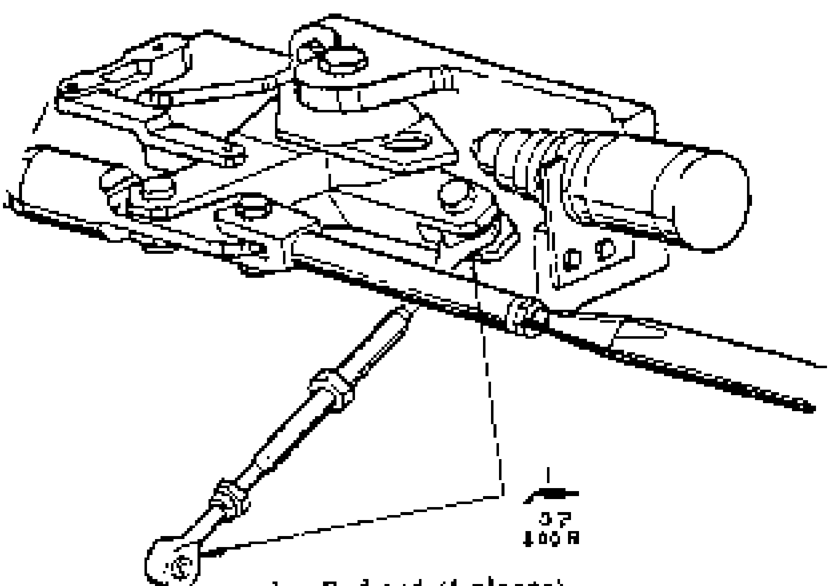
Fig 2-29



1. Roller (4 places)
2. Rod end (4 places)
3. Bolt (2 places)

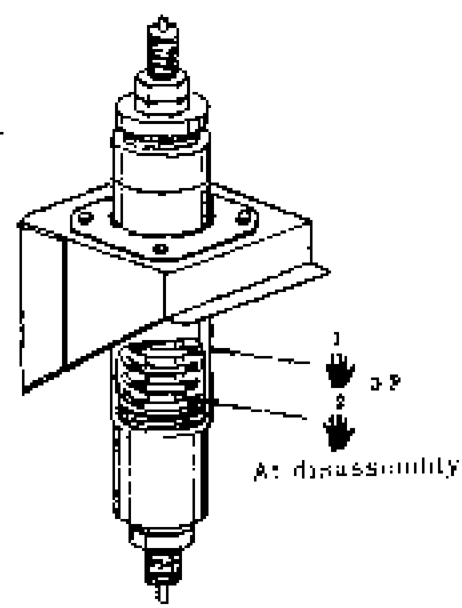
Outboard flap guide link mechanism

Fig 2-30



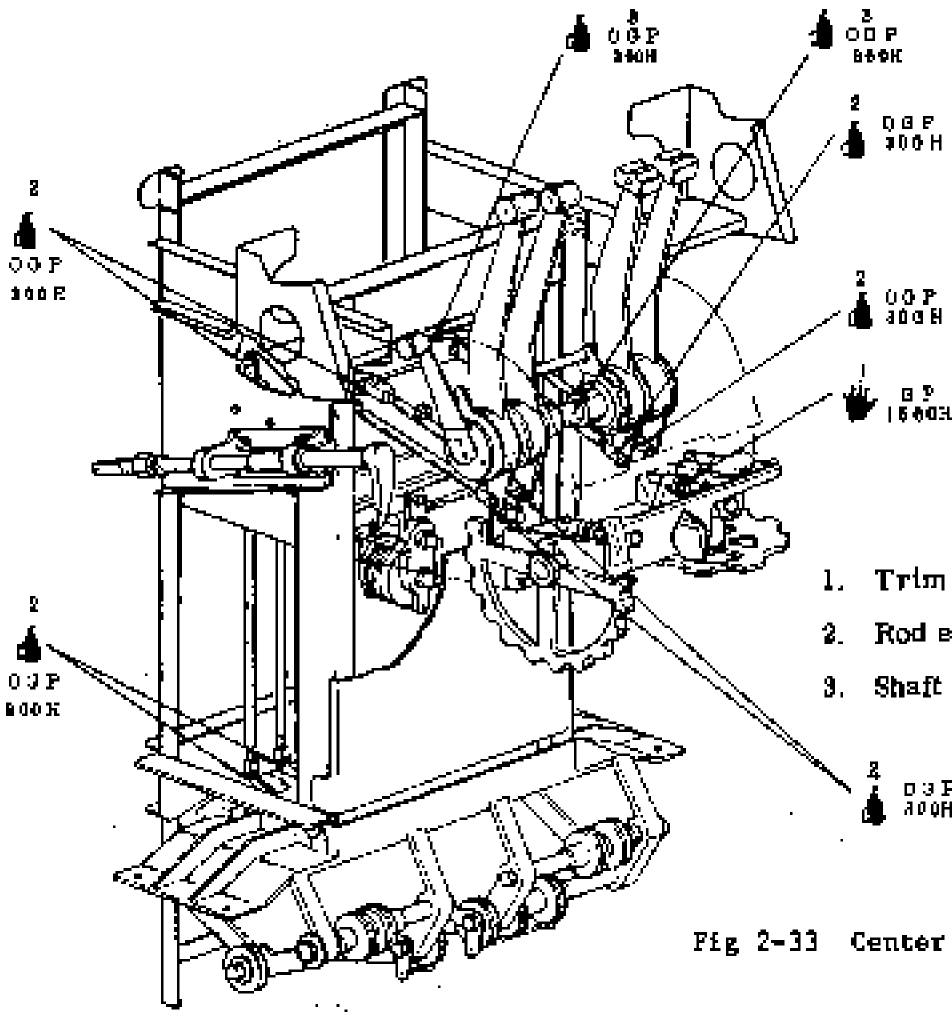
1. Rod end (4 places)

Fig 2-31 Trim aileron push rod



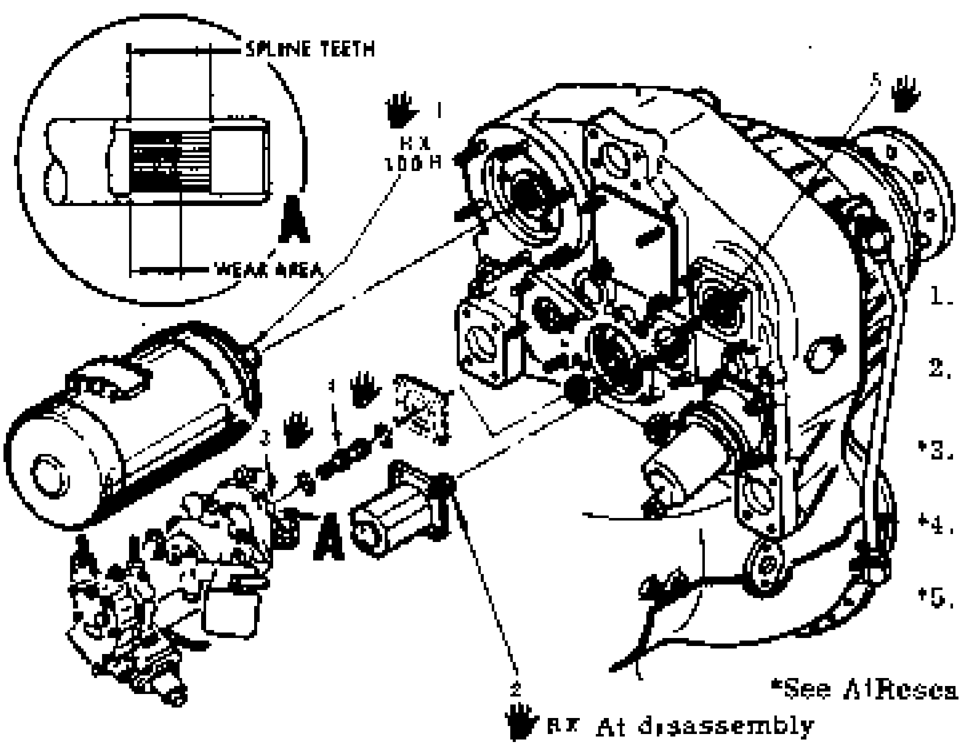
1. Cylinder inner and outer surface (1 place)
2. Piston (1 place)

Fig 2-32 Spoiler feel spring



1. Trim control gear box (2 places)
2. Rod end (18 places)
3. Shaft (3 places)

Fig 2-33 Center pedestal assembly

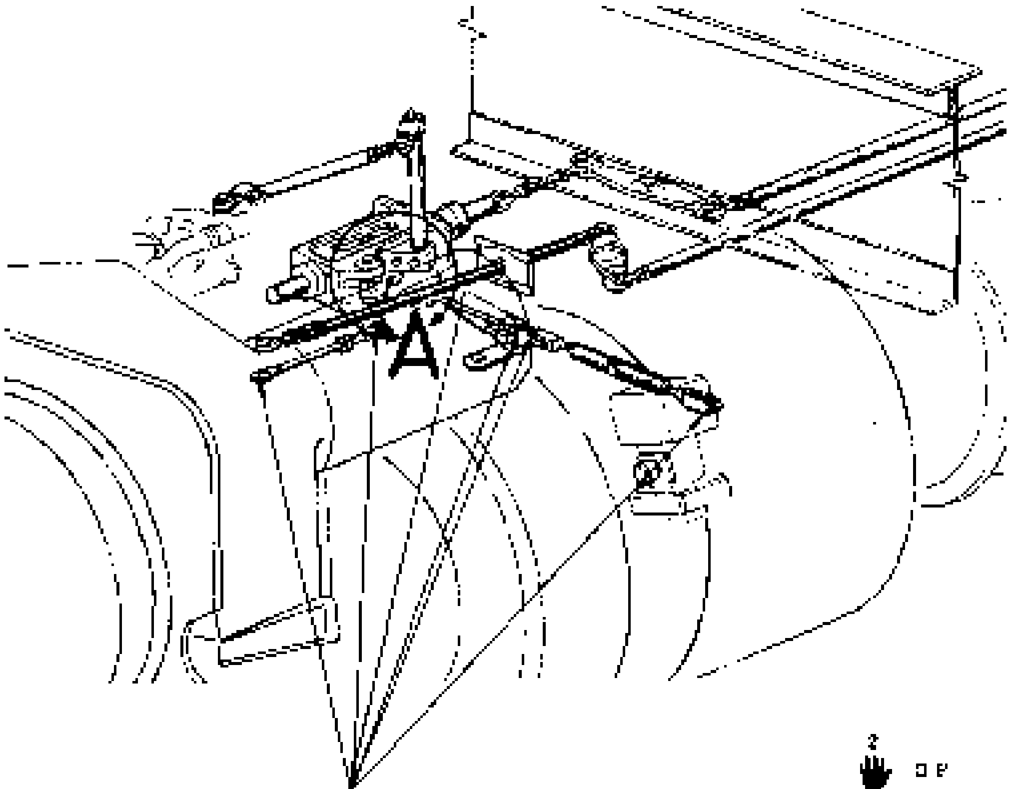


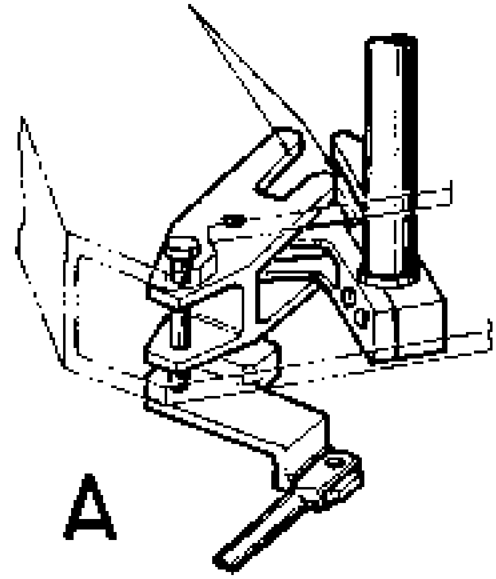
1. Starter-generator spline (2 places)
2. Tachometer-generator spline (2 places)
*3. Fuel pump assembly (internal spline)(2 places)
*4. Drive coupling (external spline)(2 places)
*5. Oil pump assembly (internal spline)(2 places)

*See AiResearch S/B TPE 331-72-027.

RX At disassembly

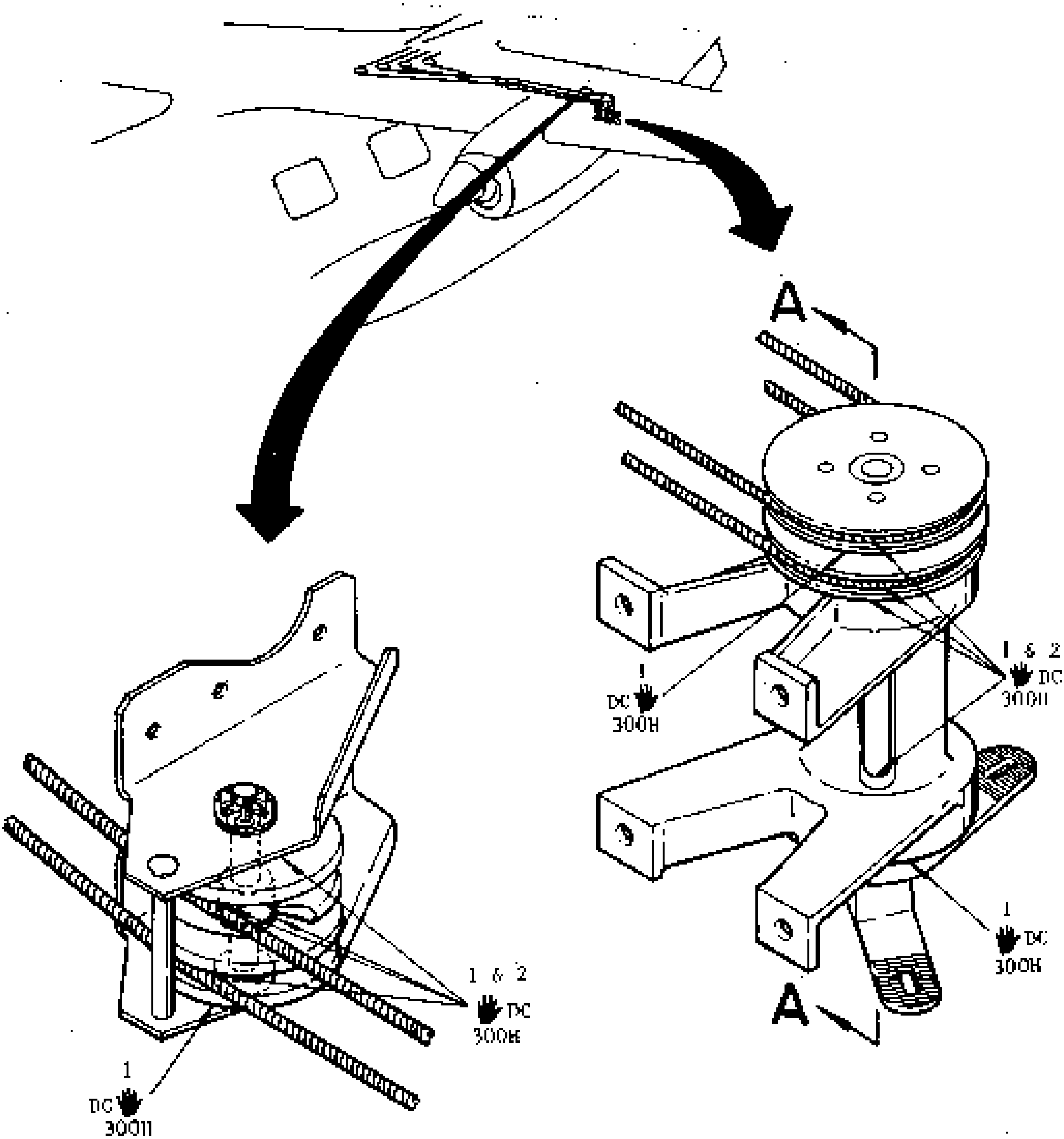
Fig 2-34 Engine accessories





1. All pin connections (as required)
2. Cam sliding surface (as required)

Fig 2-35 Engine control mechanism



1. Pulley (7 places)
2. Control Cable (4 places)

Fig. 2-35A Engine control torque tube/pulley (1/2)

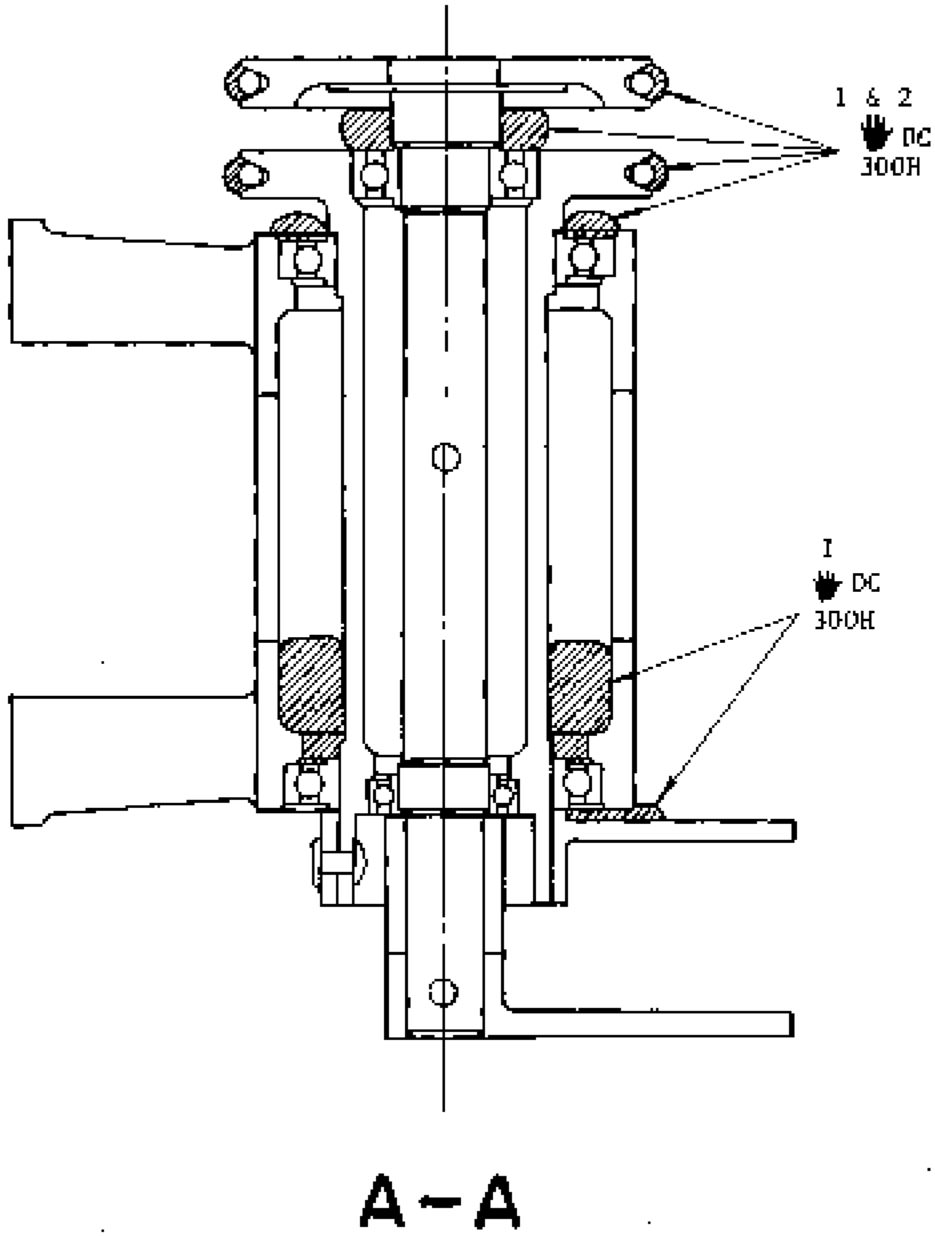
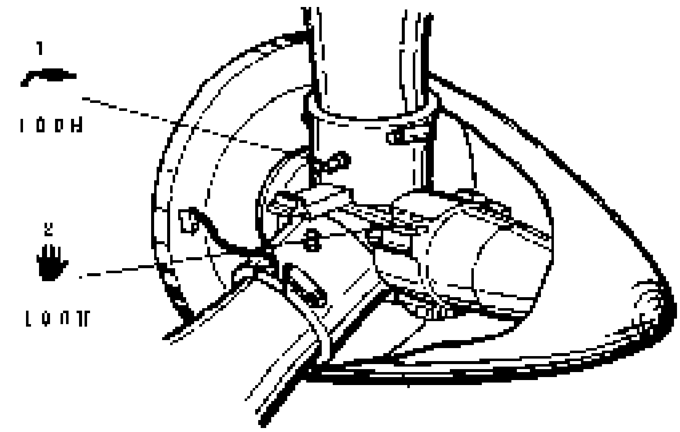
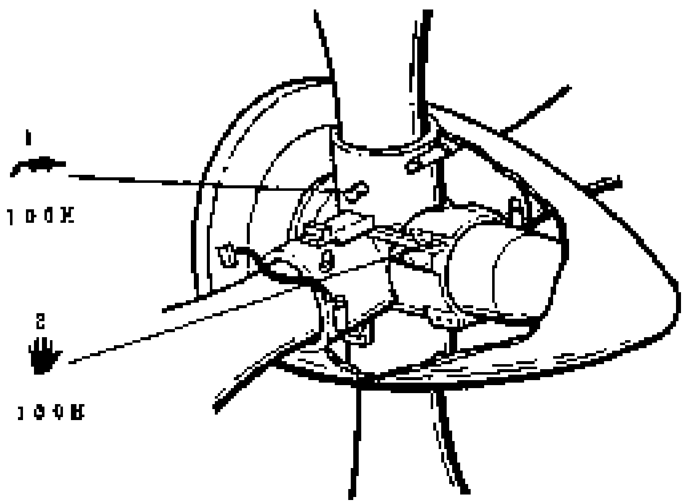


Fig. 2-35A Engine control torque tube/pulley (2/2)



1. Propeller hub (6 places)
2. Link connections (6 places)

Aircraft S/N 652SA



1. Propeller hub (8 places)
2. Link connections (8 places)

Aircraft S/N 661SA, 697SA and subsequent

The following greases are considered acceptable for use in Hartzell propellers.

When lubricating the propeller, refer to SERVICE ADVISORY No.178 by Hartzell.

Approved Greases

- ASG 6 (SHELL)
- ASG 7 (SHELL)
- 5114EP (EXXON)
- ASG 22 (SHELL)
- Royco 22C (ROYAL LUBRI)
- ASG 5 (SHELL)

CAUTION

ASG 5 IS PROHIBITED FOR USE BELOW -40°F.

Fig 2-36 Propeller

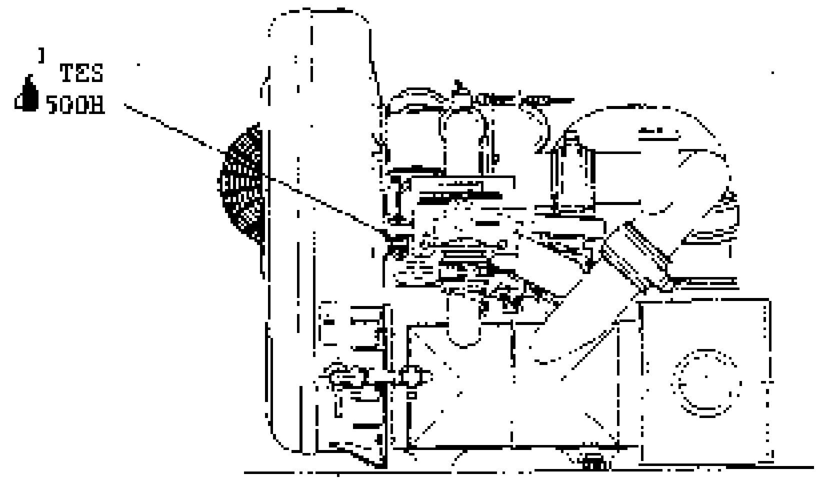


Fig 2-37 Air conditioning and refrigeration unit

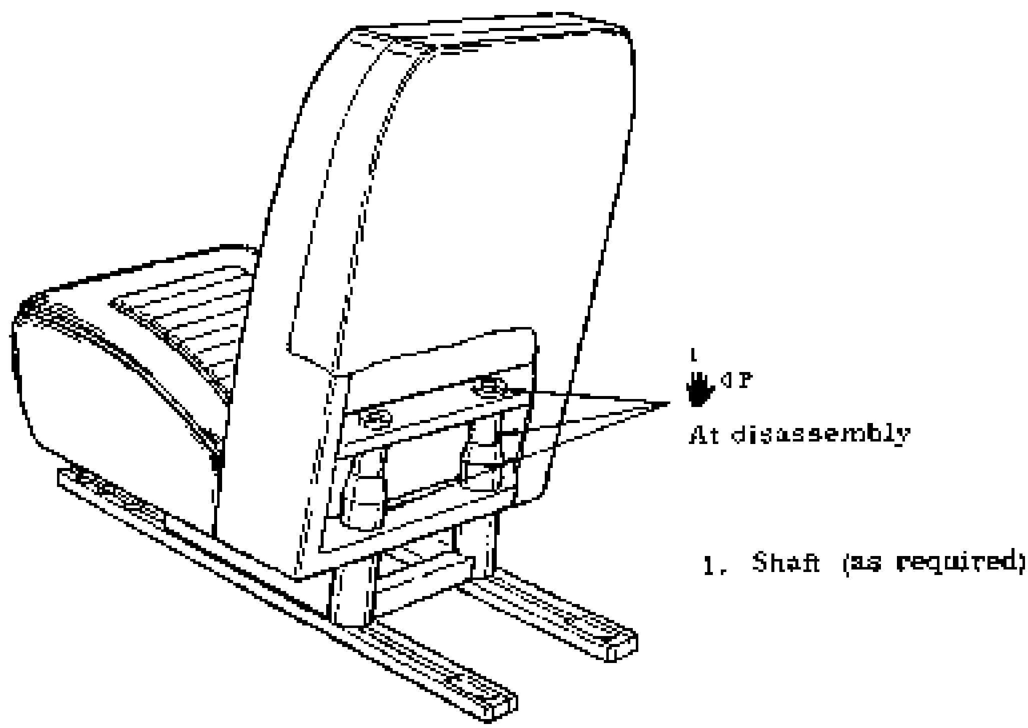


Fig 2-38 Flight chair

6 REPAIRS

In regard to repair of general structural parts, see Civil Aeronautics Manual 18, Maintenance, Repair, and Alternation of Airframes, Power Plants, Propeller, and Appliances

This paragraph provides only the repair instructions for specific structures.

6.1 SEALING

Seals consist of integral tank leakage protection seals, pressure tight seals, water tight seals, and self contouring seals

6.1.1 SEALANT MATERIALS

(Sealant)	(Manufacturer)	(Application)
Pro-Seal 890 A-2	Coast Pro-Seal A Teledyne Co	Precoat for integral tank and brush sealing
Pro-Seal 890 B-2	Coast Pro-Seal A Teledyne Co	Faying Surface Sealant for integral tank, Fillet Sealant, Void Filler Sealant.
Pro-Seal 890 B-1/2	Coast Pro-Seal A Teledyne Co.	Same as B-2 quick hardening type
454-4-1 Fungus Resistant Paint	Finch Paint & Chemical Co	Top coating inside of the bottom of tank
PR1221	Products Research Corp. or Yokohama Rubber Co.	Pressure-tight Sealant
PR1222 or PR1425B	Products Research Corp. or Yokohama Rubber Co.	Pressure-tight and water-tight sealant
PR340 or PR341 or PS895	Products Research Corp. or Yokohama Rubber Co.	Water-tight and self contouring sealant
Petrolatum (VV-P-236)		Self Contouring Sealant
Tape (JAN-P-127 Type II Class C)		Pressure-tight sealing

Naphtha (MIL-N-15178)		Cleaning
Naphtha (TT-N-95 Type II)		Cleaning Acrylic Plastics for pressure tight sealing
Methylethylketone (TT-M-261)		Cleaning
Toluene (TT-T-548)		Removing sealant
Leak Detector	Turco Products Co.	Detecting leakage
Cheese Cloth or Gauze	Commercial	Wiping for cleaning
Alodine solution Alodine 1200 20-25g Nitric acid 2cc Distilled water 1 l	Am. Chem. Product Corp	Treatment for metal surface
Benzine	Commercial	Cleaning
Test tape Scotch No.250	Minnesota Mining Mfg Co.	Test for adhesion of topcoating

6.1.2 CLEANING

(1) Protect the area around repair from foreign material, contamination and scratches using thick paper or cloth.

(2) Using a plastic scraper (preferably Nylon), remove sealant for approximately 1 inch width around the area of repair.

Care should be taken not to remove sound sealant in the outer area.

(3) Cleaning of surfaces: After removing old sealant from area of repair, wipe the area carefully with clean Cheese cloth or gauze saturated with Methylethylketone or Naphtha.

Immediately wipe the area with clean dry cheese cloth or gauze and allow the area to dry.

For cleaning acrylic glass, use Naphtha (TT-N-95 Type II)

NOTE

(i) After cleaning, do not touch the area with bare hands.

Use polyethylene gloves preferably.

(ii) Do not use a metal scraper, because it may scratch the aluminum clad surface.

6.1.3 PREPARATION OF SEALANTS

(1) Using a scale or balance, accurately weigh base compound and accelerator and mix them in the following ratio.

		(Mixing Ratio by weight)
PR1221 PR1222 PR1425 B PR341 Pro-Seal 890 A-2, B-2, B-1/2 PS835	Base compound / Accelerator	100 / 10

(2) Mix for 5~6 minutes or 7~10 minutes (PR1425) without including air, using clean spatula and container. If mixing is not complete, brown streaks will remain.

(3) Application time limit and tack free time (at 23°C (73.5°F), 50% RH)

Sealants	Application Time Limit (Hour)	Tack Free Time (Hour)
PR1221	2	10
PR1222	2	10
PR1425 B-2	2	24
PR1425 B-1/2	0.5	8
PR340, PR341	2	10
Pro-Seal 890 A-2	2	40
Pro-Seal 890 B-2	2	40
Pro-Seal 890 B-1/2	0.5	16
PS835 B-2	2	24
PS835 B-1/2	0.5	10

NOTE

Application time limit and tack free time of sealants are reduced with the increase of temperature and humidity.

(4) Curing Standard time for curing is shown below. As it is affected by temperature and humidity, hot curing with infrared lamp is preferable.

Standard time for curing at 23°C (73.5°F) 50% RH 14 days (336Hrs)
48 hours (PR1425)

Hot hardening. One hour at room temperature, and 6 hours at 50~55°C (122~131.5°F)

(5) Storage Sealants (unmixed) packed in a closed container, if stored below 27°C(80.6°F), will be effective for 6 months

PR1425 (unmixed) in original unopened container, if stored at temperatures between 40 ~80°F (4.5~27°C) will be effective at least 9 months

Mixed sealants may be stored for 72 hours at -18~-20°C (-1~-4°F).

6.1.4 APPLICATION OF SEALANTS

Sealing methods include faying surface sealing, fillet sealing, brush sealing and void filling. The typical examples of applications are shown in Fig 2-39 below.

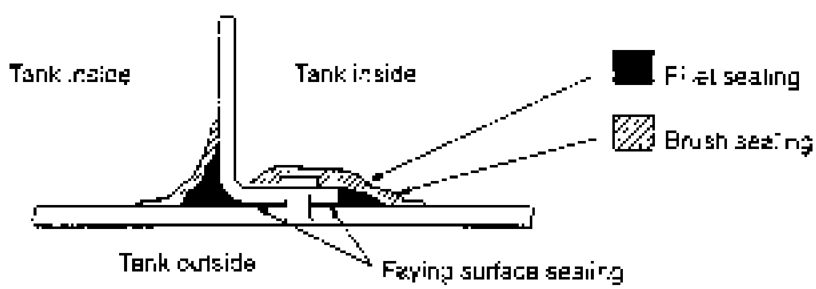
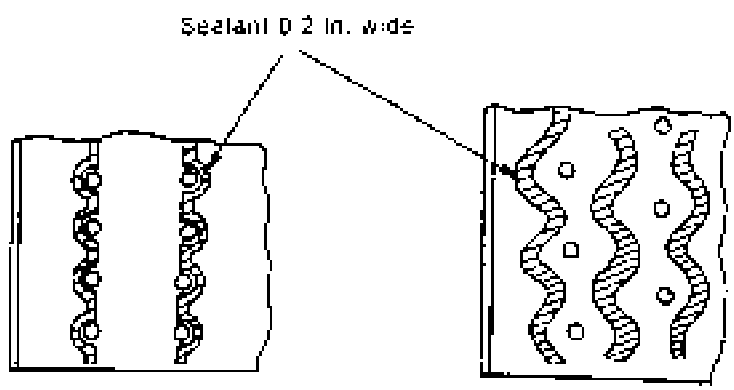


Fig 2 – 39

6.1.5 FAYING SURFACE SEALING

(1) Fill the cartridge with sealant without including air. Using sealant gun with 57~100ps (4~7kg/cm²) pressure and 0.2in.DIA (5mmø) nozzle, apply sealant continuously to one side of the fastening holes laying beads.



Fastening holes on a straight line Fastening holes situated at random

Fig 2 – 40

(2) Have the faying surfaces fastened with rivets and screws within time limit of sealants, and remove any excessive sealants with clean cloth saturated with toluene (or Naphtha TT-N-95 Type v for acrylic glass)

(3) Sealants filled in fastening holes may be removed within the time limits as needed

3.1.6 FILLET SEALING

(1) Apply sealant with spatura or gun avoiding inclusion of air, and form with spatula.

(2) Apply sealant overlapping sound sealant in surrounding area by approximately 1/2 in. (13mm).

6.1.7 BRUSH SEALING

Apply two coats on rivet, bolt, nut and fillet seals with a brush free from loose bristles.

(1) Apply brushcoats after fillet sealant cures tack free.

(2) Apply uniform thickness without including air, overlapping brush sealant on the surrounding area by approximately 1/2 in. (13mm).

(3) Apply the second brush coat after the first coat cures tack free.

6.1.8 TOP COATING

(1) Inspection

(a) Visually inspect inner surfaces of the wing integral tanks for evidence of discoloration, blistering or folds of topcoating materials. (When there exists no such irregularity, no further action is required.)

(b) If such an evidence is found to indicate faulty adhesion, determine the defected area by pressing a topcoat with fingers or a plastic scraper to locate the spot where the topcoat peels off tank bottom.

(2) Clean up

(a) Defuel the tanks and wipe off residual fuel on the tank bottom with cheesecloth or gauze.

(b) Clean the area of faulty adhesion and adjacent area with benzine immersed clean cheesecloth or gauze to remove any surface contamination together with any minor amounts of paint material remaining.

Following this cleaning, the area should again be thoroughly cleaned using clean cheesecloth or gauze and allow the area dry.

(3) Removal of defective topcoat

(a) Remove defective topcoat, using plastic or nonmetallic scrapers or test tape with the exception of sealant applied area where scrapers should not be used, but finger tip, brush or test tape should be used taking extreme care not to damage sealant.

(b) Apply a strip of one-inch wide test tape perpendicular to boundary between bared metal area and adjacent topcoat. Press the tape down firmly by hand.

(c) The tape shall be removed in one abrupt motion perpendicular to the faying surface, by grasping the tape by one end (the bared area side).

Whenever adhesion failures are observed, continue tape testing all edges of the coating perpendicular to the coating edge, pulling away from the bared area, until steady adhesion of the coating be ascertained.

(4) Preparation of primer 454-4-1

Mix the following parts by volume, agitating the base material while slowly adding accelerator: three parts of base material 454-4-1; one part of accelerator. Allow material to stand for 30 minutes, then reagitate thoroughly. Allow material to catalyze for one hour prior to application. Maximum pot life of material is 8 hours under normal temperature conditions. After maximum pot life, discard any unused catalyzed material.

(5) Recoating

NOTE

Sealant applied area does not required recoating of primer 454-4-1.

(a) Clean bared area and adjacent topcoated area one-inch wide from edge with clean cloth moistened with methylethylketone (MEK). Exercise care in cleaning DV1180/PR1005L applied area to keep MEK out of the area. DV1180 is subject to attacks of MEK solution.

(b) The cleaned area should be lightly scotch brited in conjunction with water. Wipe the repair area with clean cheesecloth or gauze and plain tap water. Change the cloths and wash water frequently.

(c) Make sure that the affected areas are free of a water resistant nature. Repeat the above steps (a) and (b) until water resistance is eliminated.

(d) Lay clean cotton cloths dampened with Alodine No. 1200 solution over the bared metal area.
Care must be exercised not to allow Alodine solution in contact with the surrounding intact coating or sealant.

(e) Allow Alodine solution to remain contacted with the metal surface for approximately 5 minutes

(f) Lay clean cheesecloth or gauze dampened with plain tap water over the affected area. Never use scrubbing motion.
Change the cloths and water as frequently as required to remove all residue.

(g) Lay and press down by hand clean, dry, lint-free cotton cloths over the repair area to dry it up.

(h) Let tank dry for 3 hours with air circulation after cleaning.
Use special care while drying to prevent the repair area from penetration of dust and other contaminants.

(i) Apply one uniform wet coat of primer 454-4-1 using good quality natural bristle brushes.

NOTE

(i) Dry film thickness should be approximately 0.016 to 0.04 in. (0.41 to 1.02 mm). Do not apply excessive amounts.

(ii) Ensure a minimum of 0.5 in. (13 mm) overlap of new primer over existing topcoating. Do not apply the primer beyond the scotch brited area. Immediately wipe off excessive primer with clean cloth dampened with MEK.

(j) Allow to dry at least 24 hours under normal temperature conditions before placing the tank in service.

The coating can be force dried for 1-1/2 hours at 50°C (122°F), if hot air source is available.

CAUTION

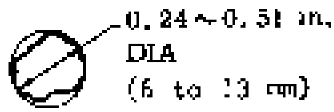
Observe all warnings, cautions and notes on material containers.

6.2 INTEGRAL TANK SEALING

6.2.1 TYPES OF LEAKS

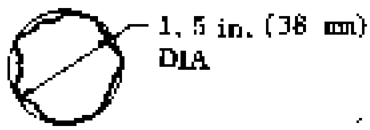
Stain: Stain caused by fuel seeping out of tank.

Action: Stains which do not affect flight safety; need not be repaired immediately, but their development should be checked periodically.



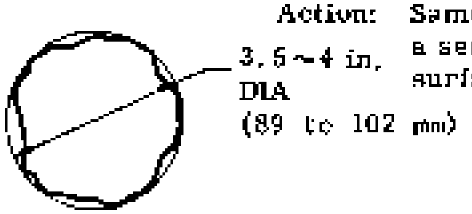
Seep: Advanced state of stain, a leak which reappears in a short period of time after being wiped clean.

Action: Same as that for stains.



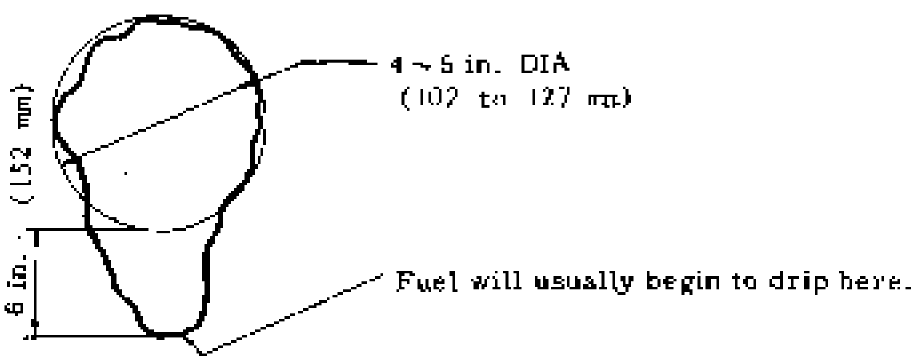
Heavy Seep: A fuel leak which reappears immediately after it is wiped clean with cloth.

Action: Same as that for stains. When a leak occurs in a sealed portion on the area other than outer surface of wing, repair immediately.



Running Leak: Fuel flows continuously out of tank and drips off wing.

Action: Repair immediately before next flight.



6.2.2 LOCATING LEAKING AREA (See Fig 2-41)

Even though a leakage is found on the outer surface of tank, the defective area in the interior of tank can not always be located immediately. The source of leakage should be located in accordance with the following procedures before attempting repair.

(1) Brush leak detector on the leaking area on the outer surface of tank. Do not apply to the inner surface of tank.

(2) Blow compressed air (max. 90 psi) that has been filtered by air cleaner to the edges of fillet, and brush seals in the tank interior corresponding to the leaking outer area as illustrated in Fig 2-41.

(3) Continue applying air until leak detector on the outer surface forms bubbles.

NOTE

Sealant may be peeled off by the compressed air. Loosened sealant must be repaired whether or not there is leakage.

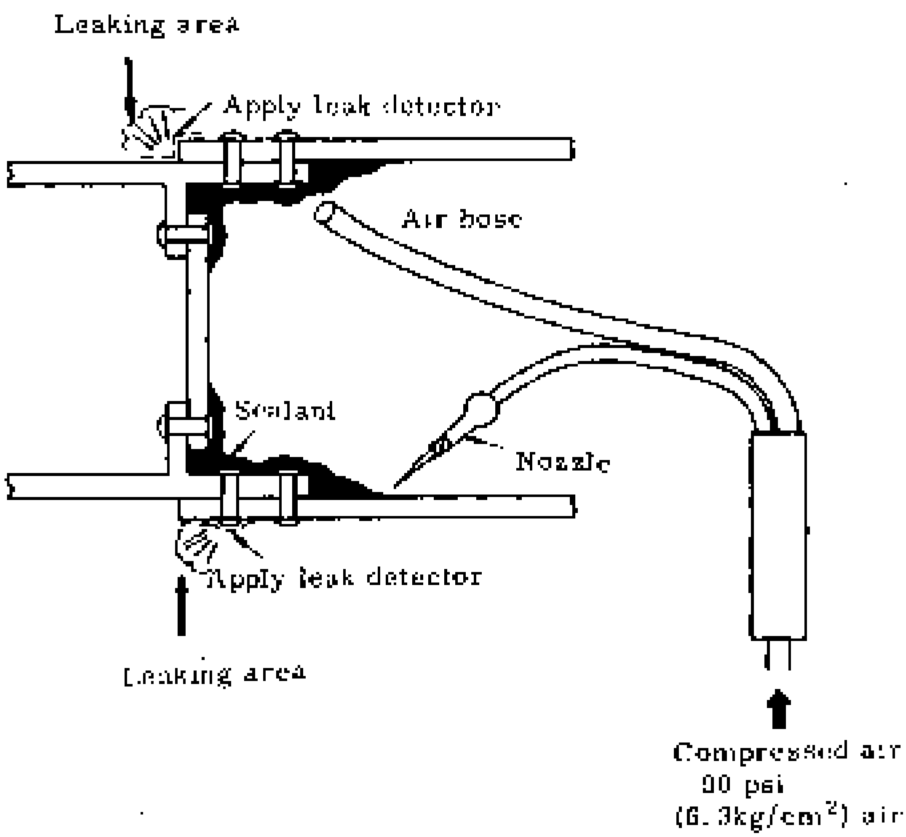


Fig 2 - 41 Locating leaking area

6.2.3 SEALING

6.2.3.1 GENERAL STRUCTURE (See Fig 2-42)

Perform faying surface sealing, fillet sealing, brush sealing and top coating in accordance with Para. 5.1

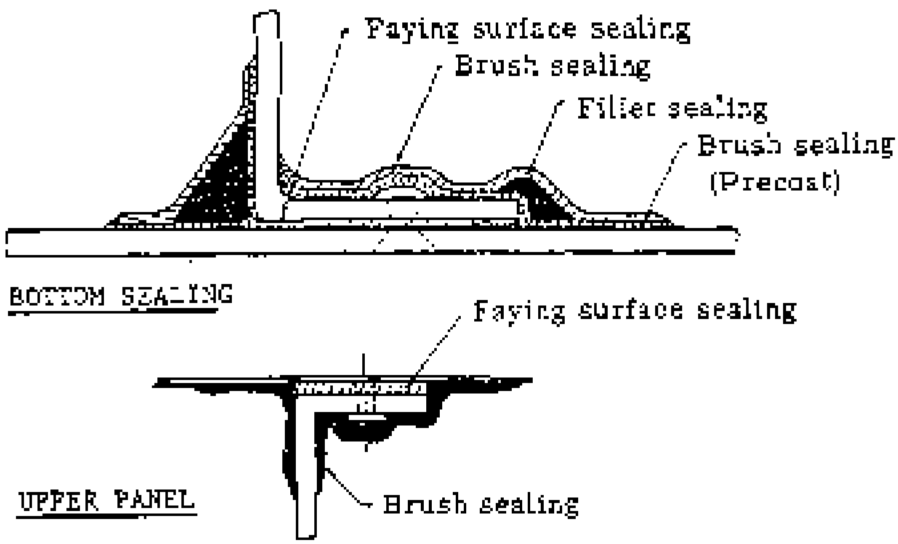
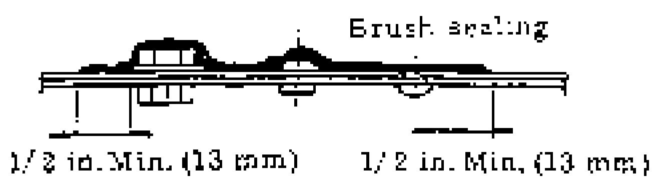


Fig 2 – 42 Integral tank sealing

6.2.3.2 BOLTS, SCREWS, RIVETS SEALS

(1) Apply brush sealant Pro-Seal 890 A-2 in accordance with instructions given in Para. 5.1.7



Sealant thickness 1/32 ~ 1/16 in (0.8 ~ 1.6mm)

(2) When replacing bolts, screws, or rivets, fill the holes with Pro-Seal 890 B-2 or B-1/2 using gun or spatula and install clean bolts, screws or rivets in the holes after Pro-Seal 890 A-2 has been applied to their shanks and beneath the heads with brush.

NOTE

Do not apply sealant to shanks, heads and holes for bolts and screws that are installed with "O" ring washer or self sealing nut.

6.2.3.3 VOID FILLING (See Fig 2-43)

All voids and holes in the tank that may cause leakage are sealed with Pro-Seal 890 B-2 or B-1/2.

(1) Apply Pro-Seal 890 B-2 or B-1/2 in a slightly excessive quantity with spatula or gun.

(2) Apply sealant to voids in the corners, in sufficient quantity to extrude when brackets are installed.

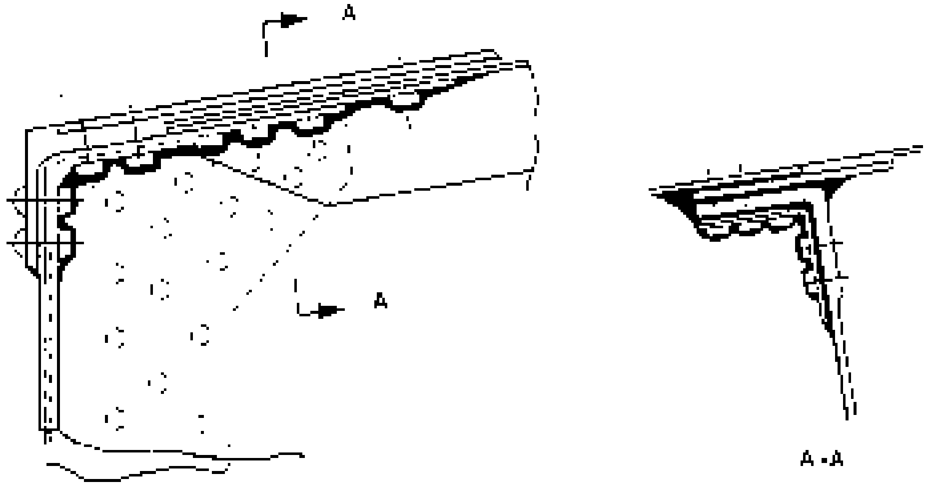


Fig 2 – 43

6.2.4 LEAKAGE TEST

Ensure that curing of repair sealants is complete, seal all doors and holes except compressed air inlet, and adjust internal pressure to 2.7 psi (0.19kg/cm^2) gage pressure using clean air that has been filtered by air cleaner. Apply leak detector to the outer surface of the tank and inspect visually for 15 minutes.

NOTE

(i) Never allow internal pressure to exceed 3 psi (0.21kg/cm^2).

(ii) Do not remove any seal before internal pressure falls to zero.

6.3 PRESSURE-TIGHT SEALING

6.3.1 SEALING OF GENERAL STRUCTURE (See Fig 2-44)

Perform laying surface sealing and fillet sealing with PR1222, PR1425B or PR1221 in accordance with Para 6.1.

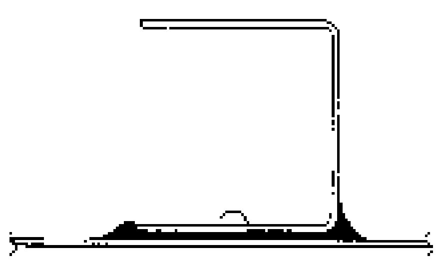


Fig 2 – 44

6.3.2 SEALING OF HOLES AND SLOTS (See Fig 2-45)

Apply sealant directly to holes with widths less than 1/8 in.(3mm).

Attach pressure sensitive tape (JAN-P-127 Type I Class C) over the holes with width 1/8 ~ 1/4 in.(3 ~ 6mm) and apply sealant.

For holes with a width of more than 1/4 in.(6mm), fill with a rivet, balsa wood plug or cap so as to reduce the opening below the 1/8 in.(3mm) max. and then seal with a layer of sealing compound, extending a min. of 5/16 in.(8mm) from the edge of the hole.

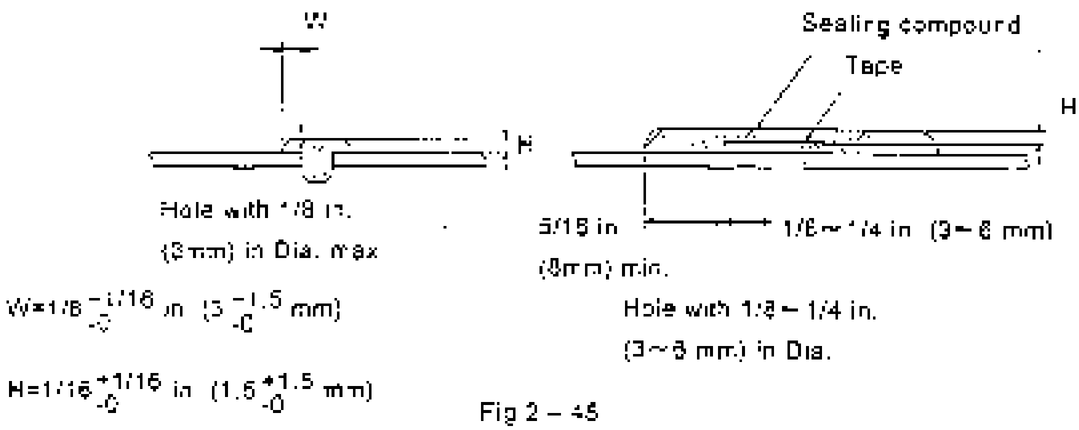


Fig 2 – 45

6.3.3 CABIN DOOR AND EMERGENCY DOOR (See Fig 2-46)

Perform faying surface sealing and fillet sealing with PR1222, PR1425B or PR1221 in accordance with Para.6.1.

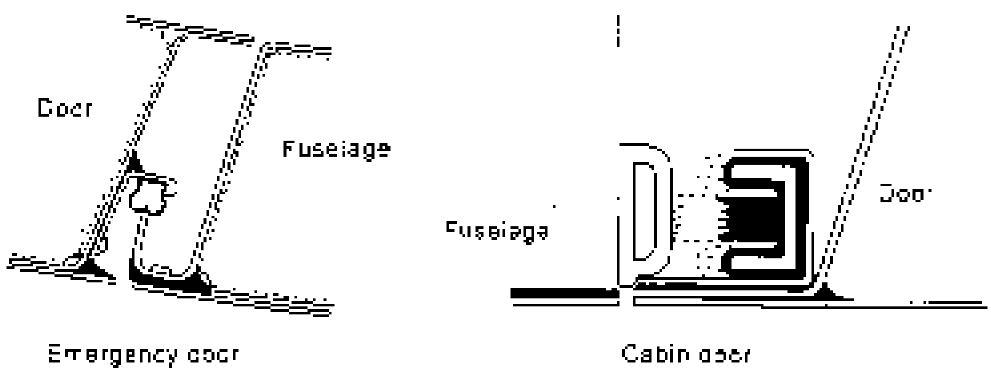


Fig 2 – 46

6.3.4 BOLT, SCREW AND RIVET SEALING

This type of sealing is performed where faying surface sealing is not practical.

Bolts, screws and rivets are sealed in accordance with the following procedures after all other sealing has been completed.

(1) Fill the hole with PR1222 or PR1425B using gun or spatula, and put a clean bolt, screw or rivet in the hole after applying sealant* to its shank and beneath the head with brush. Apply sealant* to both sides of washer before installing.

* Use a mixture of two parts of PR1222 or PR1425B with one part toluene.

(2) Apply brush sealant* in accordance with Para.4.1.7.

* Use a mixture of two parts of PR1222 or PR1425B with one part toluene

6.3.5 WINDSHIELDS AND WINDOWS (See Fig 2-47)

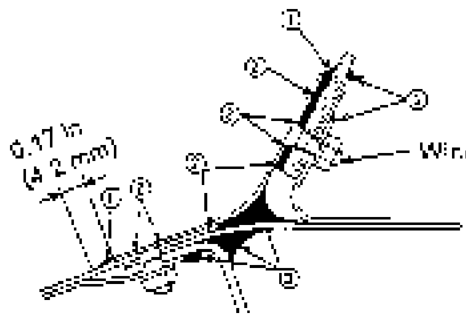
(1) Do not use any solvent other than Naphtha (TT-N-95, Type I) for cleaning.

(2) Do not use any sealant other than PR1222 or PR1425B for pressure-tight and self contouring sealing.

(3) Apply a thin coat of petrolatum to retainer before installing.

(4) Apply sealant PR1222 or PR1425B to counter sink portion of screw head before installing.

(5) Use PR341 or PS895 or PR340 for fillet sealing on the outer surface of airplane. Have the foot of fillet aligned with the existing sound seal.



1 Water-tight seal
2 Self contouring seal
3 Pressure-tight seal

Fig 2 – 47

NOTE

When part of pressure-tight seal is peeled off, noise increases. Locate and reseal the area.

6.4 WATER-TIGHT SEALING (See Fig 2-48)

Water-tight sealing is performed for protection of fuselage mating ends and quick fastener receptacles from water.

(1) Attach masking tape of approx. 3/4 in.(19mm) width on both sides of gap so that exposed surface with 1/32—3/32 in.(0.8—2.4 mm) width remains.

(2) Apply sealant, using sealant gun or wooden spatula. When a gun is used, place the nozzle of sealant gun on the gap. Tilt the gun about 45° to the direction towards which the gun is running and fill excess compound into the gap.

Squeeze the compound into the gap with a wooden spatula. Smooth the surface of applied compound and remove excess compound.

(3) When sealing has been completed, slowly remove masking tape on both sides, smooth out the edges with fingers immediately after removal of masking tape.

(4) Apply PR341 or PS895 or PR340 to outer surface of airplane, and PR1222 or PR1425B to inner surface

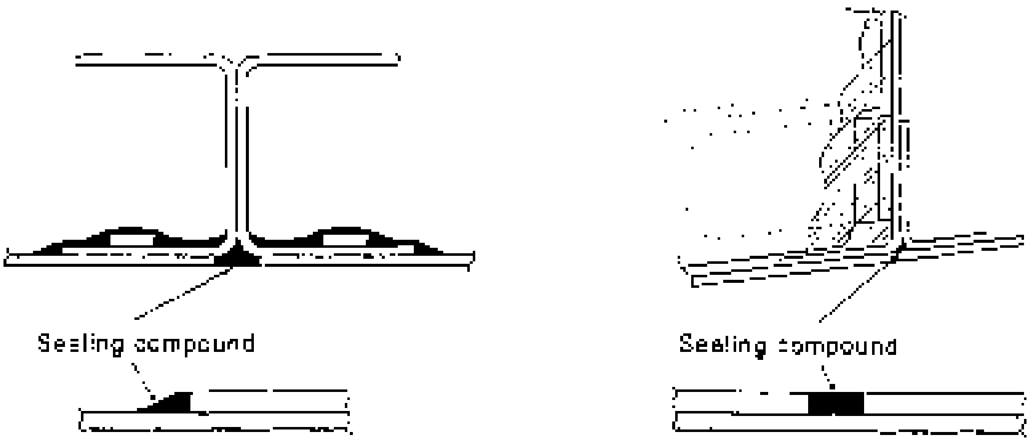


Fig 2 - 48

6.5 SELF CONTOURING SEALING

This type of sealing is performed around the lid of access door to protect the area from water.

(1) Apply PR341 or PS305 in ample quantity to airframe with gun or spatula.

(2) Apply petrolatum around lid of access door and to screws.

(3) Leave the assembly until sealant has completely cured.

(4) When a component, on which sealant has hardened, is disassembled, cut off protruding sealant with a sharp knife and wipe off the applied petrolatum with a naphtha moistened clean cloth.

(5) Wipe off excess sealing compound with a methylethylketone or toluene moistened cloth before curing (within approx. 4 hours after mixing).

6.6 REPAIR OF FIBER-GLASS REINFORCED POLYESTER-RESIN SKIN

The fuselage nose cone, tail cone, vertical stabilizer dorsal fin, rudder upper edge, and vertical horizontal stabilizer tips and forward bulge of leading gear well are made of polyester-resin reinforced with glass cloth.

6.6.1 CRACK REPAIR

(1) When the depth of crack is less than 0.008 in (0.2mm)

(a) Smooth out cracked portion with sand paper (#300).

(b) Apply resin sealing.

Pour 1～2 parts of methylethylperoxide liquid into 100 parts of resin approved in accordance with MIL-R-7575 such as; Polylite 8010, Laminac 4128, or Vibrin 152 and 6% Naphtensan Cobalt liquid, at 0.05%～0.2% weight ratio, agitate the mixture and apply with brush.

(c) Allow the patch to cold-cure for a minimum of 24 hours at room temperature, or for a minimum of 1～2 hours under a temperature of 210°F(98.8°C), using an infrared lamp.

WARNING

Do not mix methylethylperoxide liquid and 6% Naphtensan Cobalt liquid alone.

NOTE

Total amount mixed at one time should be less than 3.5 oz.(100g). Agitation should be completed within 30 min.

(2) When the depth is more than 0.008in., leave repair to special shop.

6.5.2 PEELING REPAIR

(1) Polish the surface with #100～#300 sandpaper.

(2) Clean the surface with acetone or methyl-ethyl-ketone

(3) Mixing of resin (weight ratio)

Epon 828	100
Diethylene-Tri-Amine (DTA)	8

(4) When the trailing edge of forward flap has peeled (See Fig 2-49), soak a sheet of glass cloth (#181) in resin and attach it to the trailing edge.

Fig 2 – 49

(5) When peeling has taken place between the foamed-plastics core material and cloth (See Fig 2-50).

(a) Remove 1 in. area around the peeled portion of the outer skin.

(b) Soak glass cloth in resin and attach.

(6) When peeling has taken place between an aluminium member and cloth, glue with resin.

NOTE

Peeling within 1/2 x 1/2 in.(13 x 13mm) is serviceable

Fig 2-50

6.7 REPAIR OF CABIN FLOOR

When the sandwich panel has a crack or big dip, repair in accordance with the following procedure:

6.7.1 REQUIRED TOOL AND MATERIALS

(1)	Hole saw	
(2)	Solvent	MEK, toluene or naphtha
(3)	Gluing material	Bond master M688
(4)	Re-inforcing plate	2024-T3 Clad sheet 0.04 in. (1 mm) thickness

6.7.2 PROCEDURE

(1) Cut off the damaged portion, using hole saw.

(2) Wipe off zinc chromate primer on gluing surface with MEK and clean, re-inforcing plate and gluing surface, with toluene or naphtha.

(3) Apply Bond Master M688 on both sides to be glued and place a weight on them. It takes about 3 days before the gluing is completed at room temperature.

NOTE

Do not apply solvent to foamed plastics.

Do not heat at a temperature in excess of 140°F (60°C).

6.8 REPAIR OF ACRYLIC WINDSHIELD

When depth of scratch on front, side or cabin windshield is less than 0.010 in. (0.25 mm), repair as follows.

6.8.1 MATERIAL

Solvent	TT-N-95 Type II
Castile Soap Solution	Hock Wald Chemical Co.
Fine Abrasive Type A5175	Linde Air Products Co.
Glasticoate #18	R.Killon Industrial Co.
Waterproof Sand Paper #400, #600, #800	
Flannel Cloth	

6.8.2 REMOVAL OF SCRATCH

(1) Clean the surface of acrylic.
(2) Rub the surface around the scratch with a circular motion within the limits of 0.6 in.(15 mm) dia. by waterproof sand paper #400 wrapped around wood or rubber block, keeping the sand paper wet.
(3) Perform sanding with 600 sand paper same as para. (2)
(4) Perform sanding with 800 sand paper same as para. (2)

NOTE

Partial sanding on small area may warp the acrylic and give bad visibility. Sanding should be made on a large area to prevent warping. Reduction of thickness of the acrylic panel should not exceed 0.01 in.(0.25 mm). If the panel is sanded beyond above limits the panel should be replaced.

6.8.3 REMOVAL OF SCRATCH BY ABRASIVE

(1) After sanding of acrylic, polish the sanded surface slightly with wet flannel cloth with proper amount of fine abrasive.
(2) Polishing with fine abrasive should be continued until no scratch or warp is on the acrylic.
(3) Wipe the acrylic two or three times slightly with wet flannel cloth with small quantity of Glasticoate #18. Glasticoate #18 is effective for shining and keeping off dust.

NOTE

Do not rub the acrylic with hard cotton cloth or dried flannel cloth. Wet flannel cloth should be used.

6.8.4 CLEANING INSTRUCTION

(1) Blow dust off with shop air.

(2) Flush the pane with grit-free water and a mild soap solution to remove all dust, mud, solvent, grease or other oily material, lightly rubbing with a clean soft cloth, chamois or sponge. Rinse with clean water.

CAUTION

(a) The use of gasoline, benzene, acetone, carbon tetrachloride, fire extinguisher fluid, lacquer, thinner, toluol, isopropyl alcohol, ethyl alcohol, neutral cleaner for automobile, etc., is strictly prohibited, because the effects of those fluids, when introduced to the acrylic panes, will cause them to craze or soften.

(b) Cleaning of the acrylic panes should never be attempted when dry to avoid making scratches on the surface and/or accumulating static electricity which would attract dust.
(c) Care must be exercised not to damage the surface of the acrylic panes with a ring, jewel, belt buckle or other sharp objects worn by worker while cleaning.

(3) For stubborne grease or oil deposits, use Soap Solution Castile or Cleaner MIL-C-18767.
(4) When anti-icing fluid is used, the windshield glasses should be flushed with clean water.

(5) Wipe off residual water and static electricity from the surface with a soft clean cloth, chamois or sponge and allow it dry.

6.9 REPAIR OF BOOTS

6.9.1 SCUFF DAMAGE

This type of damage will be most commonly encountered, and it is not necessary in most cases to make a repair. Repair is necessary if the scuff is severe and has caused the rubber underneath to be exposed.
To repair the damage, proceed as follows:

Material required for repairs

EC-1403 or EC-1300L	Cement	3M Co
A-56 B	Cement	Yokohama Rubber Co.
Solvent	Toluol or Methylethylketone	
Metal stitcher roller		0.24 in. (6 mm) in width

(1) Thoroughly clean the area around the damage with a cloth dampened slightly with solvent.

(2) Buff the area with steel wool.

(3) Wipe the buffed area with a cloth, slightly dampened with solvent, to remove all loose particles.

(4) Prepare a patch large enough to cover the damaged area.

(5) Apply one coat of cement on the patch and the corresponding area of boots and leave it for 2 ~ 3 min.

(6) Apply the patch to the boot with the edge, or the center adhering first. Work down the remainder of the patch carefully to avoid trapping air in pockets. Thoroughly roll the patch with stitcher-roller and allow to set for 10 ~ 15 min.

(7) Wipe the patch and surrounding area from the center outward with a cloth slightly dampened with solvent.

(8) Apply one thin coat of **A-56B** conductive cement and leave it for 4 hours.

(9) Inflate boots for at least 20 minutes for inspection.

6.9.2 CUTS, TEARS, RUPTURES OF TUBE AREA

Cuts, tears or ruptures of tube area can be repaired with fabric reinforced patches.

(1) Prepare a fabric reinforced patch of ample size to cover the damage and to extend to at least 5/8 in. (16 mm) beyond the ends and edges of the cut or tear.

(2) Repair should be done in accordance with para. (1) except that emery buffing stick is used for buffing.

(3) Apply the patch to the boot with the stretch in the widthwise direction of the inflatable tubes, sticking edge of patch in place, working remainder down with a slight pulling action so the injury is closed. Do not trap air between patch and boot surface.

6.9.3 LOOSE SURFACE PLY IN DEAD AREA (NON-INFLATABLE AREA)

Peel and trim the loose surface ply to the point where the adhesion of surface ply to the boot is good.

(1) Scrub (roughen) area in which surface ply is removed with steel wool. Scrubbing motion must be parallel to cut edge of surface ply to prevent loosening it.

(2) Scrub with steel wool and toluol directly over all edges, but parallel to the edges of surface ply in order to taper them down to the tan rubber ply.

(3) Cut a piece of surface ply material large enough to cover the damaged area and extend at least 0.1 in. (2.5 mm) in all directions.

(4) Mask off the damaged boot area 1/2 in. (13 mm) larger in length and width than the size of surface ply.

(5) Apply one coat of EC-1403 to damaged boot and surface ply.

(6) Attach the surface ply material to the boot, roll the surface ply with rubber roller, and roll edges with stitcher-roller.

(7) Apply just enough tension on the surface ply when rolling to prevent wrinkling and roll carefully to prevent trapping air. If air blisters appear, remove them with a hypodermic needle.

(8) Wipe off excessive EC-1403 with solvent.

6.9.4 LOOSE SURFACE PLY IN TUBE AREA

This type of failure is more easily detected in the form of a blister under the surface ply when the boot is pressurized. If this type of damage is detected while still a small blister [about 1/4 or 3/8 in. (6 or 9.5 mm) in dia.] and if patched immediately, the service life of the boot can be appreciably extended.

6.9.5 DAMAGE TO FABRIC BACK PLY OF BOOT DURING REMOVAL

If cement has pulled loose from wing skin and adhered to the back surface of boots, remove it with steel wool and MEK. In those spots where the coating has pulled off the fabric, leaving bare fabric exposed, apply at least two additional coats of cement EC-1403. Allow each coat to dry thoroughly.

6.10 REMOVAL OF BOOTS

When boots are removed from airplane for repair in shop, the following procedure should be followed;
Removal of boots should be done in a well ventilated place. Use toluol, and in order to remove cement remaining on airplane, use cement remover (Turco 2822).

WARNING

Use rubber gloves and overalls to prevent any harm to skin and clothes from Turco 2822.
If solvents come in contact with skin, wash it well with water. Be especially carefully not to get solvents in eyes or mouth. If solvents enter eyes or mouth, wash with plenty of water and consult a doctor immediately.

(1) Peel off one corner of boot on upper surface of wing using a minimum amount of solvent.

(2) Separate upper surface edge of boots at approximately every 4 in. (100 mm) along the wing.

(3) Apply solvent and separate boot, pulling down towards the leading edge with uniform tension.

(4) Separate boot from the leading edge and the lower surface.

(5) Wipe off cement remaining on the back surface of separated boot with toluol or MEK.

(6) Apply Turco 2822 to cement remaining on wing and leave it for approximately 15 to 30 min.; remove with a bamboo or aluminum spacula. Cement remaining on wing can be removed with toluol or MEK.

NOTE

Use as little solvent as possible. Pull boot lightly as the solvent dissolves cement on the back of boot.

NOTE

Do not peel off the rubber coating on the back side of the boot.

NOTE

Do not apply cement remover to any area which does not need it. Painted areas should be protected with masking tape.

6.11 INSTALLATION OF RUBBER BOOTS

6.11.1 PREPARATION OF AIRPLANE

(1) If leading edge has been painted, remove all paint with solvent. Alodine, iridite or anodized surfaces are satisfactory as is.

(2) Place (but do not install) boot in desired position on wing, and apply masking tape, allowing approximately 1/2 in.(13 mm) margin from the edges of boot.

(3) Wipe the metal surface with M.E.K to remove oil and dirt completly. Before drying solvent, wipe with dried cloth and make sure of no rust. If MEK dries too fast to work, toluol may be used.

(4) Fill in the clearance between skins with EC-801B. Round head rivets or misalignment less than 0.03 in.(0.8 mm) may be left as they are.

6.11.2 RUBBER BOOTS INSTALLATION PROCEDURE

Wipe the back of boot (the surface to which metal surface is attached) twice with M.E.K.

6.11.3 PROCEDURE

Materials required

Cement	EC1300L (FSN 8040-628-4199)
	or EC1403 (FSN 8040-514-1880)
Solvent	Toluol or Methylethylketone (M.E.K)
Cement	A-56 B Yokohama Rubber Co.
Filling Cement	EC-801B
Oil-proof Cement	EC-801A
Metal stitcher roller 1/4 in. (6.4 mm) wide	
Hard rubber roller 2 in. wide (50 mm) wide	
Cement remover	Turco 2822 (Turco Product Co.)

(1) Apply one brush coat of EC-1403 uniformly to the cleaned back surface of the boot and to the wing surface, and dry for one hour.

(2) Apply a second coat to both surfaces and dry at least one hour

NOTE

Thoroughly mix cement before using. Allow longer time to dry if relative humidity is 80 ~ 90%. If cemented area is kept free from dust, it can be left as is for 8 hours.

(3) Draw a center line accurately on leading edge of wing.

(4) Tighten hose attaching clamp to air connector of boot.

(5) Squeeze hose and air connector into hole in leading edge and extend boot to check cemented area equal to leading edge side and boot side. Wind boot in a roll approx. 6 in. (152 mm) in dia. from outside toward air connector to make the work easy.

(6) Align center lines of leading edge and boot.

(7) Wipe the surfaces of boot and wing with a cloth slightly moistened with MEK or toluol. Area to be covered by one wiping should be about 2 ~ 3 in. (50 to 75 mm) wide and 12 in. (305 mm) long.

(8) When adhesiveness is satisfactory, extend boot, roll boot with roller to cement boot on wing. Roll in such a manner that air is not trapped. Areas around edge, manifold and cutout should be rolled with 1/4 in. (6 mm) stitcher roller and other areas should be rolled with 2 in. (50 mm) rubber roller, always rolling parallel with the direction of the tube. Install boots in the following sequence: area along center line, area around air connector, upper side, and lower side.

NOTE

Always handle boots with clean hands and work quickly when MEK is used.

(9) If air pockets are left under the non-inflated area as a result of rolling, pierce boot with hypodermic needle, let air out, and use roller at the same time. Air pockets under tube inflated area are removed through the small area (approx. 1/16 in. (1.5 mm) wide) between tubes. Air pockets with diameters less than approx. 1/4 in. (6 mm) may be left as they are.

(10) After installation, remove all masking tape. Apply new masking tape again and apply static conductive coating on tapered edge and on cement EC-1403.

(11) After drying the coating, apply masking tape again and apply oil-proof cement coating.

8.11.4 TEST AFTER BOOTS INSTALLATION

(1) Sampling test
Prepare a sample, same material as boot, 1 in. x 8 in. and 0.12 in. in thickness (25 x 203 mm and 3 mm in thickness).
Stick the sample near the boots with the same procedure as the boot, leaving about 1 in. (25 mm) from the edge loose (See Fig 2-51).
At 4 hours after sticking, place a clamp to the loose edge and pull with a spring balance. Indication of balance should be 5 lbs. (2.3 kg) or more.

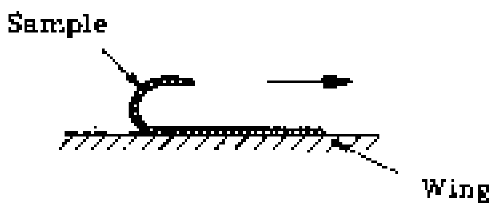


Fig 2 - 51

(2) When the procedures in Para. (1) have failed, perform the following test. (See Fig 2-52)

a. Peel the edge of the boots with toluol as shown in Fig. 2-52, pull the edge with fingers and check adhesion.

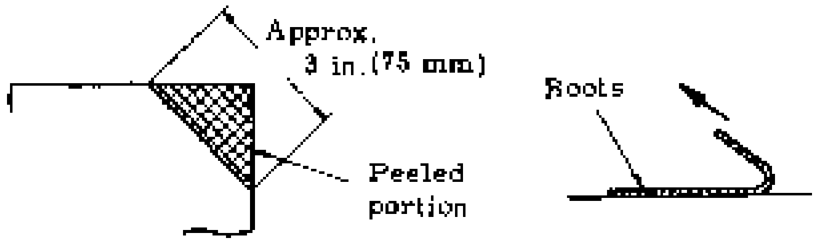


Fig 2-52

If the check is OK, flight may be performed but inflation of boots should not be done until 12 hours after repair.
When the sampling test has an indication of 10 lbs. (4.5 kg), boots may be inflated immediately.

6.12 REPAIR OF STATIC CONDUCTIVE COATING (See Fig 2-53)

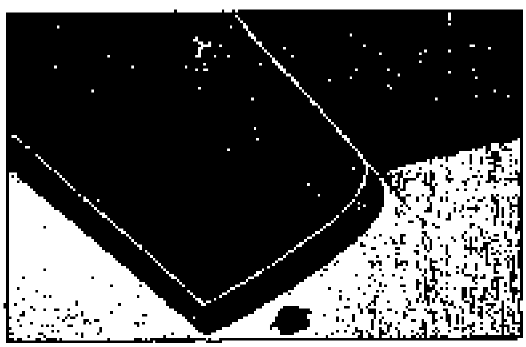
6.12.1 PROCEDURE (See Fig 2-54)

1. Clean the boot with cleaning solvent.
2. Polish the surface of boot with sand paper.
3. Wipe the surface with clean cloth dampened in solvent.
4. Apply masking tape.
5. Apply one brush coat of cement (A-56B) and dry for one hour.
6. Apply second coat of cement (A-56B), remove masking tape and dry for four hours at room temperature.

NOTE

(i) Dilute the cement (A-56B) with isopropyl acetate to obtain proper brushing consistency. Mix thoroughly. 5 parts cement to one part isopropyl acetate.

(ii) It is desirable that cement be stored in a place with a low temperature. Effective storage period is approx. 6 months.



After applying cement A-56B, seal filling cement EC-801B and apply oil-proof cement EC-801A.

Fig 2-53 Static conductive coating on rubber boot

ORIGINAL
As Received By
ATP

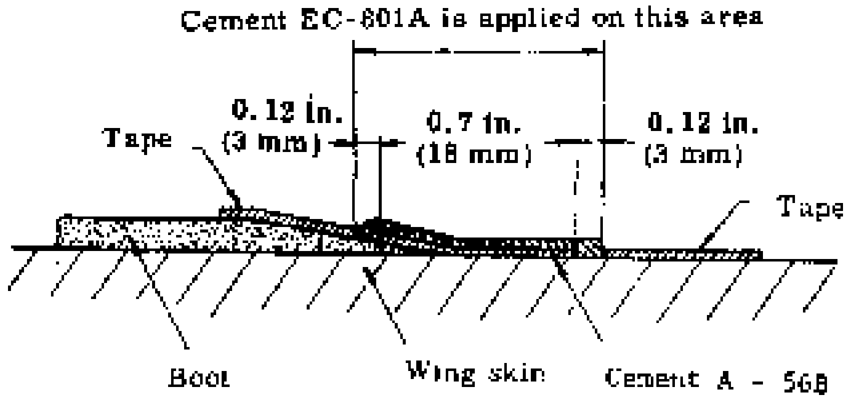


Fig 2-54

6.13 REPAIR AND REPLACEMENT OF DOOR RUBBER SEAL

If rubber seal is damaged or deteriorated, replace as follows:

6.13.1 REQUIRED MATERIAL

(1)	Adhesion (for repair)	MIL-A-5092 Type II Alon Alpha 201	TOA GOSEI Co.
(2)	Solvent	Methylethylketon	TT-M-261 or equivalent
(3)	Diluent	Toluol	
(4)	Sand paper	#200 ~ #400	

6.13.2 REMOVAL OF RUBBER SEAL

(1) Peel the edge of rubber seal off, using a small amount of Methylethylketone and applying M.E.K., pull the seal off slowly. Apply cement remover (Turco 2822) to cement remaining on airplane, leave it for about 15 ~ 30 minutes, and scrape with bamboo or aluminum spatula. Small amount of cement remaining should be removed with M.E.K.

NOTE

(i) Use rubber gloves and overalls to prevent any harm to skin and clothes from Turco 2822. Special care should be taken to prevent from getting them in eyes or mouth.

(ii) Do not apply cement remover (Turco 2822) to any area which does not need it. Painted areas should be protected with masking tape.

6.13.3 CEMENTING OF RUBBER SEAL

(1) Perform sanding on cementing surface with #200 ~ #400 sand paper and wipe with cloth dampened in Methylethylketone.

(2) Apply one thin coat of cement to the cementing surfaces of rubber seal side and airplane side.

(3) A few minutes later, attach rubber seal to airplane, taking care to keep dust free.
Necessary time to obtain proper consistency may differ due to temperature and humidity. It is 2 ~ 8 minutes at 74°F (23°C) and 60% humidity.

(4) After setting proper position, press the seal down to the cementing surface to remove air. Roll the seal if possible.

(5) After cementing, leave it at least 8 hours at 74°F (23°C). Heat or water should not be applied on the cementing surface for 72 hours.

(6) Wipe off excess cement with a Methylethylketon or toluene moistened cloth before drying.

6.13.4 REPAIR OF RUBBER SEAL (See Fig 2-55)

(1) Preparation
Wipe off the cement remaining on the damaged area with a Methylethylketon or toluene moistened cloth. Take care that Methylethylketon or toluene does not sink into the cemented area.

(2) Cementing
After cleaning the damaged area, apply Alon Alpha 201 and immediately press down the seal. A small amount of Alon Alpha should be used. Much use of Alon Alpha reduces cementing strength in addition to wasting cement. Since Alon Alpha has a short cementing time, quick action should be taken.

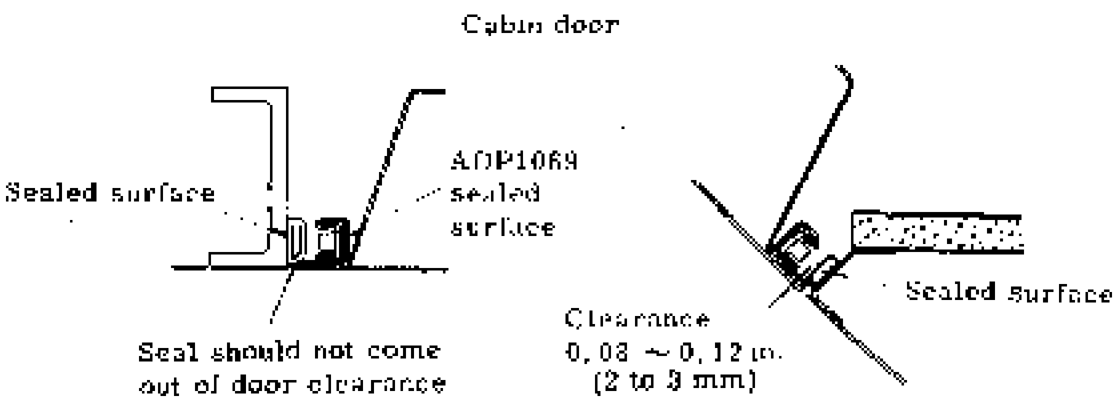


Fig 2-55 Repair of door rubber seal

7. AIRFRAME AND SKIN

Material thickness and type of skin used in this airplane are shown in Fig 2-56.

No	Gauge in.(mm)	Material	No	Gauge in.(mm)	Material
1	.016(0.41 mm)	2024-T3C	17	.050 → .0197 (1.27 → 0.5 mm)	2024-T3C *
2	.020(0.51 mm)	2024-T3C	18	.0788 → .059 (2.0 → 1.5 mm)	7075-T6C *
3	.025(0.64 mm)	2024-T3C	19	.0788 → .059 (2.0 → 1.5 mm)	7075-T6C *
4	.032(0.81 mm)	2024-T3C	20	.032(0.81 mm)	6061-T6C
5	.040(1.02 mm)	2024-T3C	21	.040(1.02 mm)	6061-T6
6	.050(1.27 mm)	2024-T3C	22	.0788 → .059 (2.0 → 1.5mm)	2024-T3C *
7	.0394 → .0237 (1.0 → 0.6 mm)	2024-T3C *	23	.025(0.64 mm)	2024-T42C
8	.063 → .032 (1.6 → 0.81mm)	2024-T3C *	24	.090 → .032 (2.3 → 0.81mm)	2024-T3C *
9	.016 (0.41 mm)	301(MIL-S-5059)	25	.032(0.81mm)	2024-T42C
10	.016(0.41 mm)	347(MIL-S-6721)	26	.063 → .051 (1.6 → 1.3mm)	2024-T3C *
11	.0197(0.5 mm)	FRP	27	.020 (0.51 mm)	2024-T42C
12	.0296(0.75 mm)	FRP	28	.025 (0.64 mm)	6061-T6
13	.0394(1.0 mm)	FRP	29	.040 (1.02 mm)	2024-T42C
14	.0492 (1.25 mm)	FRP	30	.040 (1.02 mm)	6061-T6
15	.050(1.27 mm)	6061-T5	31.	.032 (0.81 mm)	2024-T42C
16	.059(1.50 mm)	2024-T3C			

*denote chemical milling processed parts.

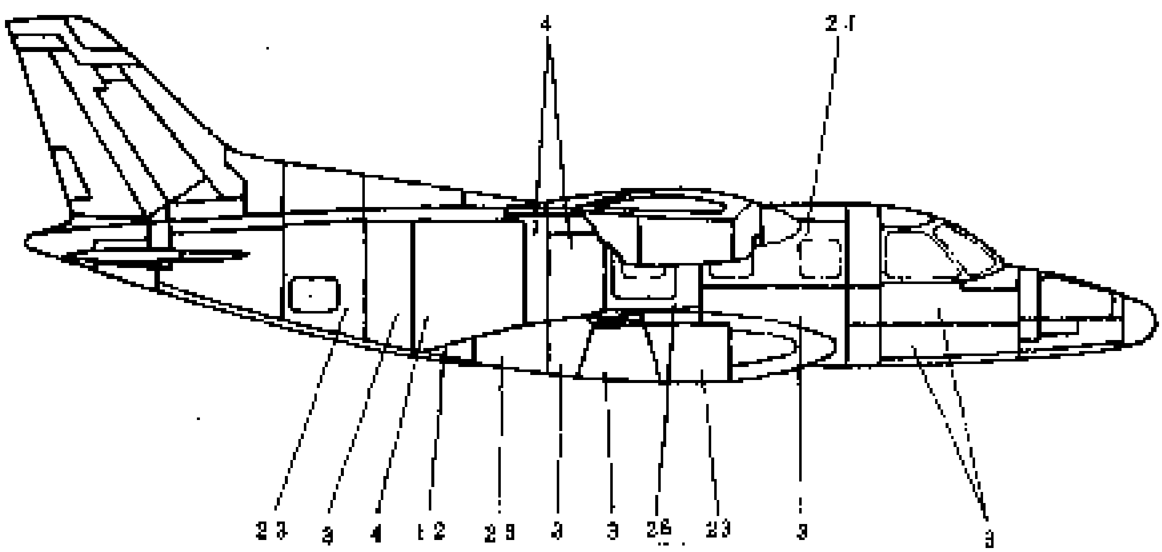
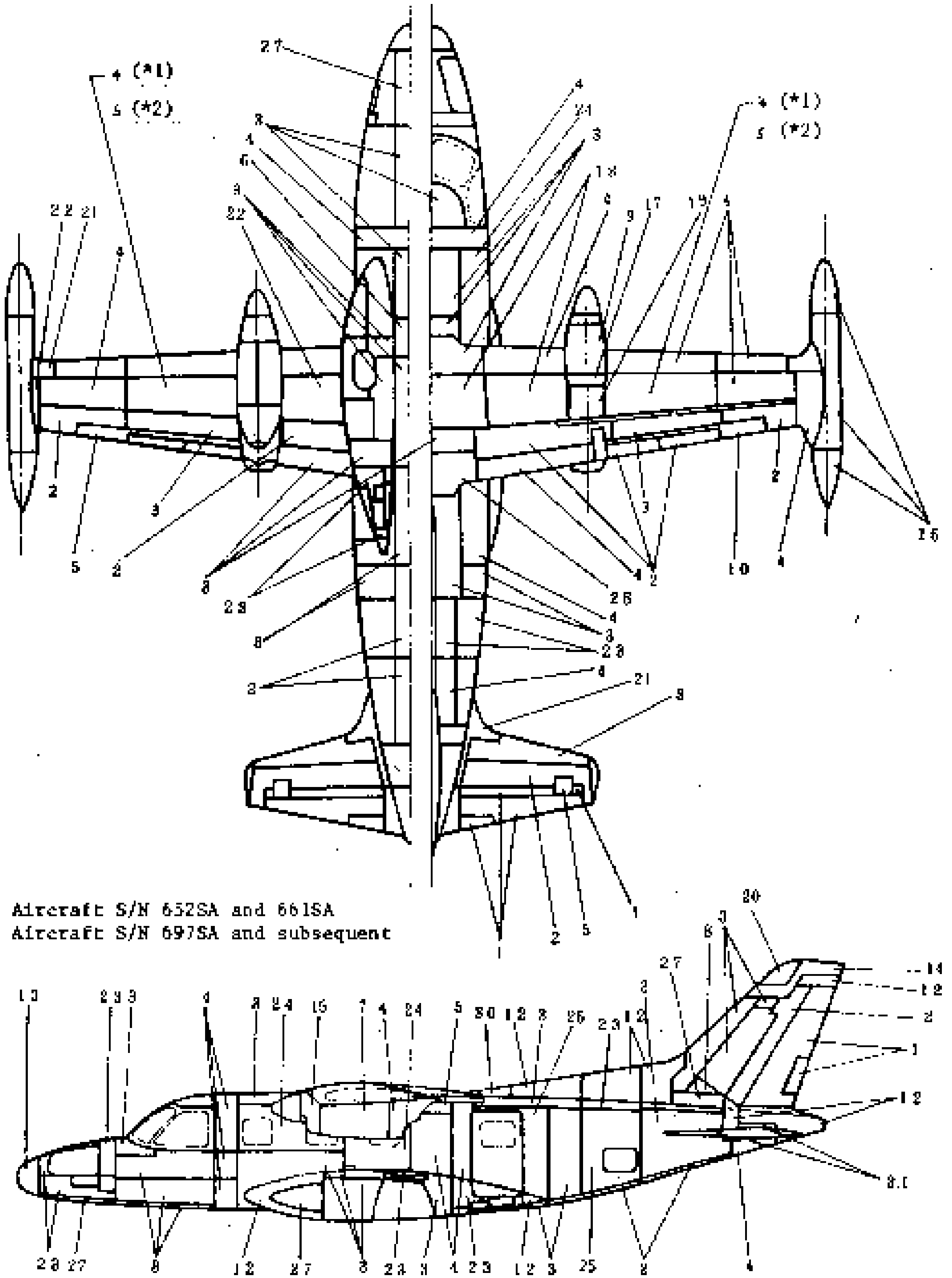


Fig 2-56 Skin chart (1/2)



*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

Fig 2-56 Skin chart (2/2)

8. ACCESS DOOR

Inspection of airframe structures and equipment should be done with necessary access door opened. Main access doors of wing and empennage are shown in Fig 2-58. Main access doors of fuselage are shown in Fig 2-59.

8.1 MAINTENANCE PRACTICE

8.1.1 When removing access doors (panels), care should be taken not to damage panels, adjacent structure and fasteners. Apply appropriate sealant when required and tighten fasteners to proper torque value.

8.1.2 Engine Nacelle Door Latches

(1) Close the side doors and lock the latches, closing force should be 5 to 15 lbs., adjust as required.
(2) With the side doors closed, close the upper nacelle door and lock the latches, closing force should be 5 to 15 lbs., adjust as required.

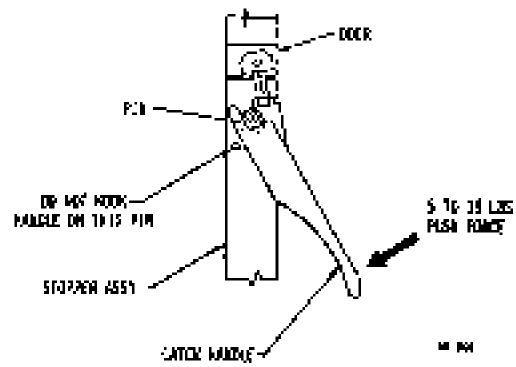
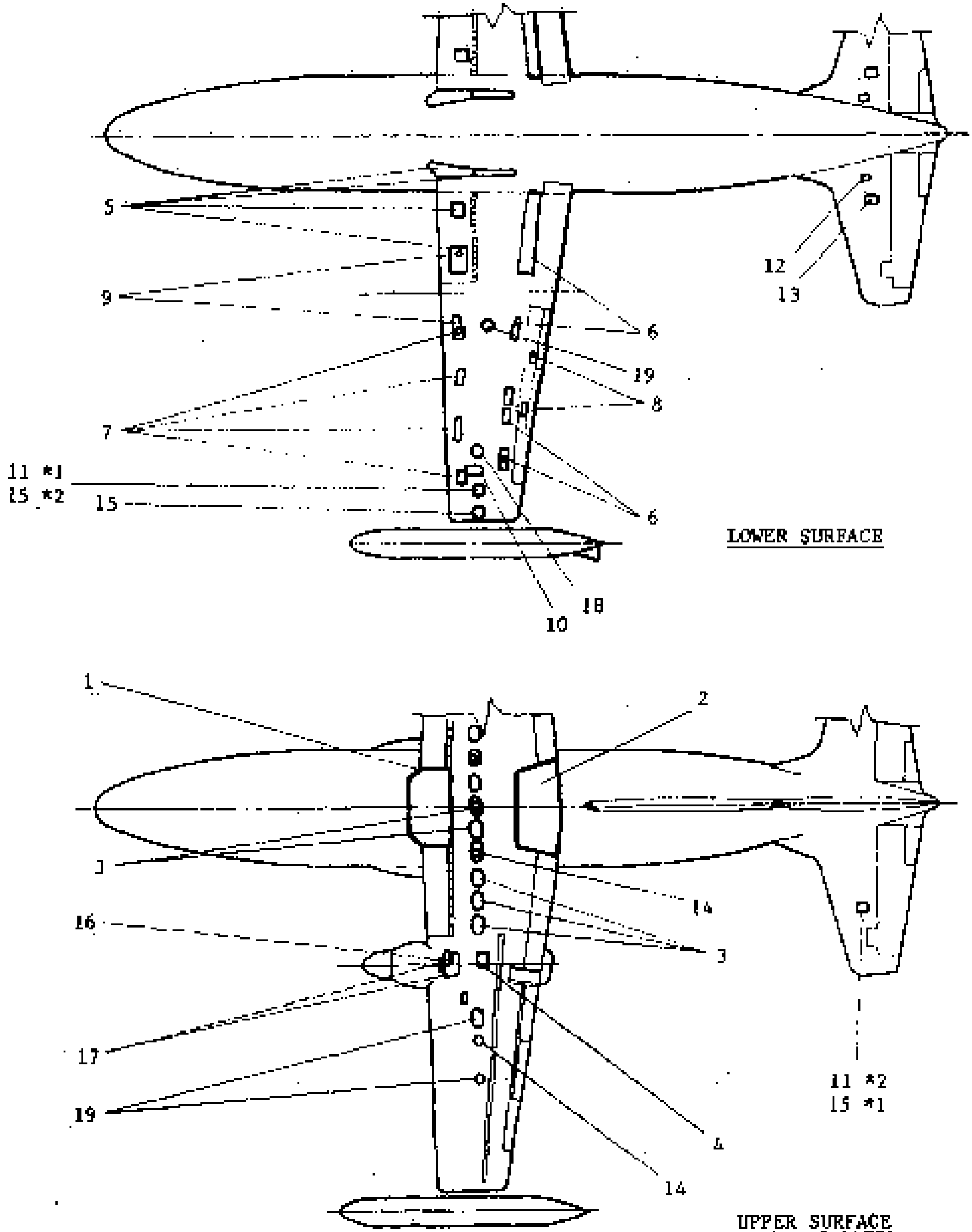


Fig 2-57 Nacelle door latch

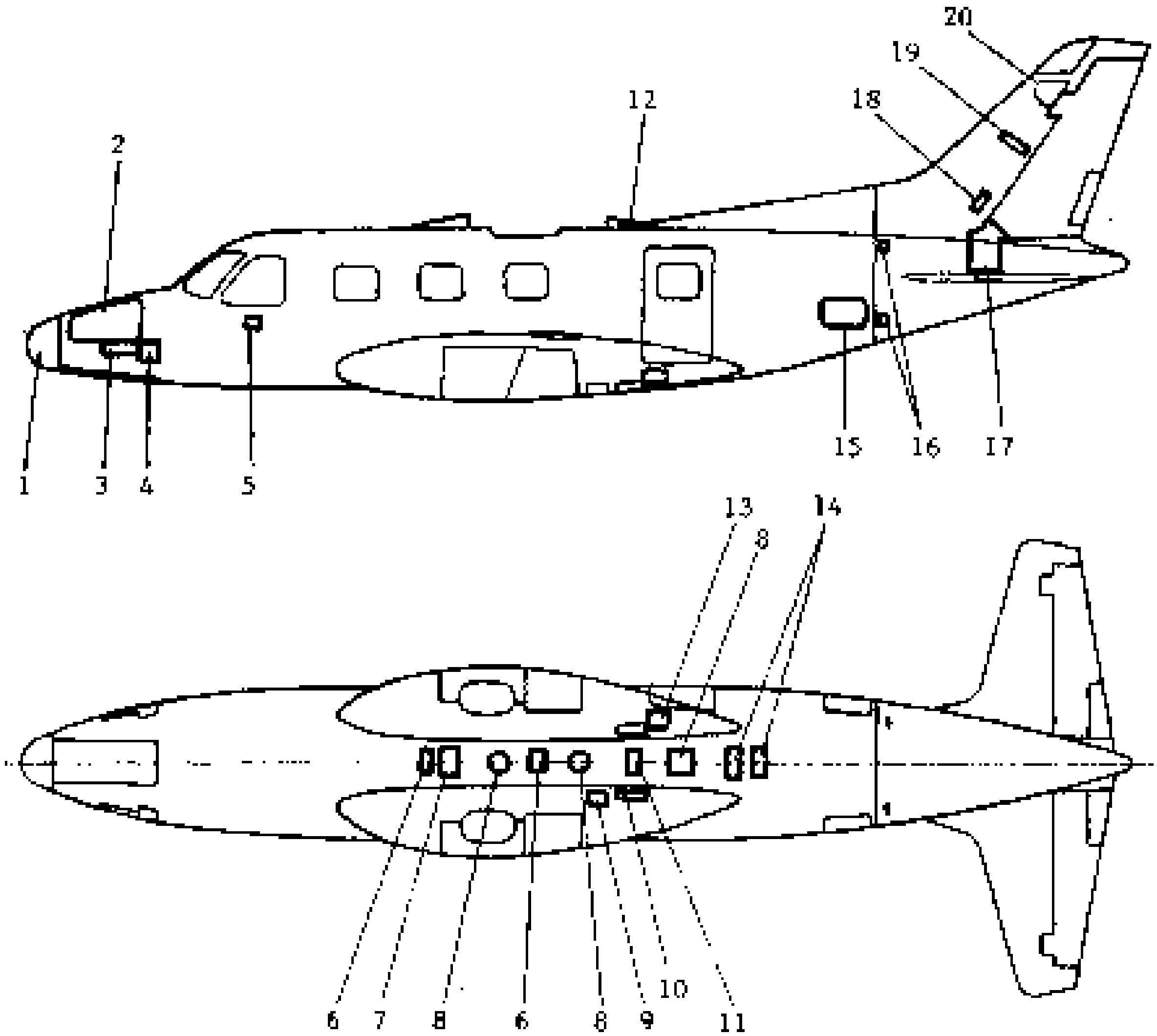
Key to Fig 2-58

No.	Name
1	Center wing leading edge upper door
2	Center wing trailing edge upper door
3	Fuel tank access door
4	Fire extinguisher access door
5	Electric and fuel system access door
6	Flap and spoiler mechanism access door
7	Fuel tip tank line access door
8	Trim aileron actuator access door
9	Nacelle door hook
10	Flap gear box access door
11	Gyro compass flux valve access door
12	Actuator chain access door
13	Elevator trim tab actuator access door
14	Fuel filler
15	Access door
16	Fuel system access door
17	Fuel leakage inspection door
18	Trim aileron actuator electric plug access door
19	Outer tank access door



*1 Acft modified by S/R 001/34-001, S/N 699SA and subsequent
*2 Acft S/N 652SA, 661SA, 697SA & 698SA except Acft modified by S/R 001/34-001

Fig 2-58 Wing and empennage access panels



NO.	NAME
1	Nose cone glide slope antenna
2	Electronic compartment door
3	Landing lights (LH & RH), anti-icer fluid tank (RH)
4	Landing gear chain (LH), wiper motor (RH), anti-icer control assy (LH)
5	Communication door (LH)
6	Landing gear door control mechanism
7	Cable fairlead
8	Cable
9	Landing gear motor
10	Gearbox & drag strut support
11	Landing gear power train
12	Spoiler servo actuator
13	Step mechanism
14	Pulley bracket, control cable
15	Electric & electronic equipments, air conditioning system
16	Main fuselage-aft fuselage connecting fitting longeron (upper & lower)
17	Rudder quadrant
18	Actuator chain
19	Rudder trim tab actuator
20	Antenna system

Fig 2-59 Fuselage main access panels

9. INSTALLATION TORQUE

(1) Installation torque of flare tube and flexible hose fittings

Flare tube and flexible hose fittings with steel coupling nut or steel insert (Oxygen tube excluded) in-lbs(kg-cm)

Tube O.D	5052-O Al alloy tube	6061-T6 Al alloy tube	Flexible hose	CRES tube	NOTE
1/8	20 - 30 (23 - 35)	--	25 - 45 (29 - 52)	45 - 55 (52 - 63)	in.(mm) Tube Thickness *1 : 0.049 (1.24) *2 : 0.042 (1.07) *3 : 0.028 (0.71) *4 : 0.035 (0.89)
3/16	30 - 40 (35 - 46)	45 - 55 (52 - 63)	70 - 100 (81 - 115)	90 - 100 (104 - 115)	
1/4	40 - 65 (46 - 75)	80 - 105 (92 - 121) 90 - 115 (104 - 132)*1	70 - 120 (81 - 138)	135 - 150 (156 - 173)	
5/16	60 - 80 (69 - 92)	80 - 105 (92 - 121) 125 - 175 (144-202) *2	85 - 180 (98 - 207)	180 - 200 (207 - 230)	
3/8	75 - 125 (86 - 144)	125 - 175 (144-202)	100 - 250 (115 - 288)	270 - 300 (311 - 346)	
1/2	150 - 250 (173 - 288)	135 - 180 (156-207) *3 200 - 300 (230-346) *4 400 - 500 (461-576)	210 - 420 (242 - 484)	450 - 500 (518 - 576)	
5/8	200 - 350 (230 - 403)	500 - 600 (576-691)	300 - 480 (346 - 553)	700 - 800 (806 - 922)	
3/4	300 - 500 (346 - 576)	600 - 700 (691-806)	500 - 850 (576 - 979)	1100 - 1150 (1267 -1325)	
1	500 - 700 (576 - 806)	1000 - 1300 (1152-1498)	700 - 1150 (806 - 1325)	1200 - 1400 (1382 -1613)	
1 1/4	600 - 900 (691 - 1037)	1300 - 1500 (1498-1728)		1300 - 1450 (1498-1670)	
1 1/2	600 - 900 (691 - 1037)	1400 - 1700 (1613-1958)		1350 - 1500 (1555-1728)	

(2) Installation torque of union to boss

in-lbs(kg-cm)

Tube O.D.	1/8	3/16	1/4	5/16	3/8	1/2	5/8	3/4	1 and up
Al or Steel Boss	50~75 (58~69)	75~90 (86~104)	80~100 (92~115)	100~130 (115~150)	100~130 (115~150)	200~240 (230~276)	360~400 (415~461)	390~430 (449~495)	600~900 (691~1037)

(3) Installation torque of oxygen tube

in-lbs(kg-cm)

Tube O.D	Al alloy tube	Copper tube (solder type)
3/16	-	40-50(46-58)
5/16	100-125(115-144)	-
3/8	200-250(230-288)	-
1/2	300-400(346-461)	-

(4) Installation torque of major joints and standard installation torque of nuts.

Item	Torque Value in-lbs(kg-cm)
Nut - Control wheel	750-1000(840-1152)
Nut - Flap stop assembly	250-350(278-403)
Bolt - Nose wheel assembly	60(69)
Bolt - Main wheel assembly	220(253)
Nut - Engine mount (To wing)	2 - 8(2.3 - 9.2)
Nut - Wing to fuselage	6 - 25(7 - 29)
Bolt - Propeller outer blade clamp	780(899)
Screw - Propeller inner blade clamp socket	480(553)
Screw - Propeller counter weight socket	780(899)
Nut - Propeller piston edge	860-1440(991-1659)
Valve - Tip tank sniffle	300-400(346-461)
Bolt - Fuel level control valve	10 - 14(11.5 - 16)
Bolt - Scavenge pump-rear bearing	Tighten to 75 to 80 in-lbs(86 to 92 kg-cm), then loosen and retighten to 50 to 55 in-lbs, (58 to 63 kg-cm).
Plug - Magnetic oil drain	60 - 70(69 - 81)
Coupling - Bleed air shutoff valve (24502-150)	90 - 110(104 - 127)
Coupling - Bleed air tubing in dorsal fin(NAS592-32, NAS596-32)	900-1200(1037-1382)
Coupling - Bleed air tubing (24502-200)	120-140(138-161)
Coupling - Bleed air tubing in wing trailing edge(NAS592-20 and NAS596-20)	600-900(691-1037)
Coupling - Bleed air tubing in nacelle fillet (24502-125)	65 - 75(75 - 86)
Coupling - Precooler inlet (24540-200)	30 - 40(35 - 46)
Coupling - Refrigeration unit bypass line (24540-150)	30 - 40(35 - 46)
Coupling - Refrigeration unit bypass valve (69738-150)	45 - 50(52 - 58)

CLASSIFICATION	1			
BOLT STUD SCREW NOMINAL DIA	° Bolt, stud and screw having a tensile strength of 125,000 psi ~145,000 psi (Ex. AN3, AN173, MS20073 series bolts, AN502, AN503, NAS 220 series screws) ° Shear bolt and screw having a tensile strength of 160,000 psi ~180,000 psi (Ex. NAS1103 NAS1143, NAS1153, NAS1203 series bolt, NAS517 series screw) ° Corrosion Resistant steel bolt and screw having a tensile strength of 140,000 psi or more (Ex. NAS1003 series bolt, NAS560H series screw)			
	° When the above bolts, studs and screws are used with shear nut (AN320, NAS1022 or equivalent) ° When the above bolts, studs and screws are used as hinge pins together with nuts (NAS679A or MS21042)			

Coarse	Fine	kg — cm	in — lb	kg — m	ft — lb
#8-32	#8-36	9~11	7~9		
#10-24	#10-32	14~18	12~16		
1/4-20		29~35	25~30		
	1/4-28	35~46	30~40		
5/16-18		56~64	48~55		
	5/16-24	70~78	60~68		
3/8-16		110~130	95~110		
	3/8-24	114~130	95~110		
7/16-14		165~180	140~155		
	7/16-20	310~560	275~300		
1/2-13		290~345	240~290	2.8~3.3	20~24
	1/2-20	340~470	290~410	3.3~4.7	24~34
9/16-12		350~490	300~420	3.6~4.5	26~36
	9/16-18	560~630	440~600	3.5~5.5	40~50
5/8-11		490~620	424~540	4.8~6.2	35~45
	5/8-18	700~900	640~780	7.6~9.0	56~65
3/4-10		810~1100	700~950	8.0~10.9	58~79
	3/4-16	1500~1700	1300~1600	14.5~18.3	104~136
7/8-9		1340~2100	1300~1800	14.6~20.7	108~150
	7/8-14	1800~2100	1600~1800	17.4~20.7	125~150
1-8		2400~3500	2200~3000	25.4~34.6	180~250
	1-14	2600~3300	2200~3000	25.3~33.2	183~275
1 1/8-8		3800~4400	3300~4400	33.0~46.0	275~333
	1 1/8-12	5640~6400	5000~6250	44.4~63.4	364~464
1 1/4-8		4600~5800	4000~5000	44.0~51.6	313~416
	1 1/4-12	6300~7500	6400~8400	63.2~76.0	457~554

CLASSIFICATION				
BOLT STUD SCREW NOMINAL DIA	° Bolt, stud and screw having a tensile strength of 125,000 psi~145,000 psi, (Ex. AN3, AN173, MS20073 series bolt, AN502, AN503, NAS220 series screw) ° Shear bolt and screw having a tensile strength of 160,000 psi~180,000 psi (Ex. NAS1103, NAS1143, NAS1153, NAS1203 series bolt, NAS517 series screw) ° Corrosion Resistant Steel bolt and screw having a tensile strength of 140,000 psi or more (Ex. NAS1003 series bolt, NAS560H series screw) ° When the above bolts, studs and screws are used with tension nut (AN310, MS21042, MS21043, NAS679A, NAS679C NAS1021 or equivalent), nut plate or equivalent threaded machine part			

Coarse	Fine	[illegible]	[illegible]	[illegible]	[illegible]
#8-32	#8-36	14~ 18	12~ 16		
#10-24	#10-32	20~ 23	20~ 26		
1/4-20		45~ 63	40~ 50		
	1/4-28	58~ 81	60~ 70		
5/16-18		95~ 106	80~ 90		
	5/16-24	115~ 164	100~ 140		
3/8-16		145~ 210	165~ 185		
	3/8-24	188~ 270	160~ 190		
7/16-14		270~ 340	235~ 255	28 — 29	22~ 21
	7/16-20	524~ 660	450— 500	51~ 56	37~ 42
1/2-13		440~ 560	400~ 480	44~ 66	66~ 44
	1/2-20	540— 800	480~ 630	55~ 85	40~ 58
9/16-12		650~ 810	500~ 700	58~ 80	42~ 63
	9/16-18	830~ 1204	800— 1400	58~ 11.8	57— 93
5/8-11		810~ 1000	300~ 900	80~ 104	58~ 76
	5/8-18	1300~ 1500	1104~ 1400	127~ 142	92—108
3/4-10		1400~ 1600	1350— 1600	133~ 184	96~135
	3/4-16	2700~ 2900	2500~ 2504	254~ 26.6	192—208
7/8-9		2500~ 3500	2200~ 3000	26.5~ 34.5	184~256
	7/8-14	3300~ 3500	2500~ 3404	33.5~ 54.1	208~230
1-8		4300~ 5800	3700— 5000	42.6~ 57.5	306~415
	1-14	4300~ 6300	3700~ 6600	42.6~ 48.8	308~450
1 1/8-8		4300~ 7600	5600— 6600	63.0~ 74.7	466~514
	1 1/8-12	5300~ 8104	5090~ 7000	61.6~ 84.2	415~680
1 1/4-8		7600~ 8200	6600~ 8400	74.7~ 92.6	540~670
	1 1/4-12	10400—12700	3400~11000	100.7~127.1	750~924

CLASSIFICATION		II			
BOLT STUD SCREW		* Bolt, stud and screw having a tensile strength of 140,000 psi ~ 160,000 psi			
NOMINAL DIA		Tension nut MS21042 or equivalent			
Coarse	Fine	[illegible]	[illegible]	[illegible]	[illegible]
#8-32	#8-36	18~ 24	14~ 17		
#10-24	#10-32	27~ 35	21~ 40		
1/4 -20		62~ 85	45~ 59		
	1/4 -28	49~ 92	40~ 80		
5/16 -18		98~ 135	85~ 117		
	5/16 -24	140~ 200	120~ 172		
3/8 -16		240~ 250	173~ 217		
	3/8 -24	240~ 310	176~ 271		
7/16 -14		280~ 540	245~ 342	2.8~ 4.0	20~ 29
	7/16 -20	550~ 724	476~ 624	5.5~ 7.2	40~ 52
1/2 -13		610~ 780	440~ 456	5.1~ 7.3	37~ 53
	1/2 -20	660~ 870	480~ 644	6.8~ 9.7	49~ 70
9/16 -12		690~ 964	610~ 845	6.7~ 9.7	50~ 70
	9/16 -18	1100~ 1490	960~ 1224	10.4~ 14.1	75~ 103
5/8 -11		920~ 1300	800~ 1126	9.3~ 15.9	67~ 94
	5/8 -18	1400~ 2400	1200~ 1789	13.8~ 19.9	100~ 144
3/4 -10		1600~ 2200	1380~ 1926	15.9~ 22.1	115~ 160
	3/4 -16	2800~ 4400	2400~ 3640	27.7~ 40.1	240~ 292
7/8 - 9		3000~ 4100	2600~ 3570	29.9~ 41.2	216~ 298
	7/8 -14	3200~ 5400	2750~ 4650	31.7~ 53.5	229~ 388
1 - 8		5000~ 6300	4350~ 5324	50.0~ 65.4	362~ 496
	1 -14	5000~ 8100	4600~ 7260	50.1~ 54.6	361~ 605
1 1/8 - 7		6800~10900	6040~ 8660	60.1~ 99.5	540~ 726
	1 1/8 -12	6900~11340	6040~10250	69.1~118.2	540~ 855
1 1/4 - 7		8400~12700	7250~11000	83.6~126.6	605~ 915
	1 1/4 -12	11500~14860	14000~16764	115.2~192.9	838~1390

CLASSIFICATION		III			
BOLT STUD SCREW NOMINAL DIA		° Bolt, stud and screw having a tensile strength of 160,000 psi and more. [Ex. MS20004~MS20024, MS21250 series bolt] ° When the above bolts; studs and screws are used with tension nut; MS21042 or equivalent, EB. H20, 42FW etc. [Commercial Type Nut]			
Coarse	Fine	lb — in	in — lb	kg — m	[illegible] — lb
# 8-32	# 8-36	18~21	16~18		
#10-24	#10-32	29~40	26~35		
1/4 - 20		62~78	50~65		
	1/4 -28	81~105	70~90		
5/16 -18		105~170	90~144		
	5/16 -24	166~245	140~208		
3/8 -16		224~290	185~248		
	3/8 -24	220~400	194~351		
7/16 -14		300~500	255~425	2.3~6.0	21~36
	7/16 -20	440~870	505~756	5.6~8.7	42~60
1/2 -13		650~910	480~792	6.5~9.1	40~64
	1/2 -20	800~1200	690~890	8.0~11.5	62~85
9/16 -12		910~1200	760~990	8.0~11.5	58~88
	9/16 -18	1290~1700	1040~1440	11.5~16.6	68~124
5/8 -11		1100~1600	900~1850	10.4~16.6	75~112
	5/8 -18	1600~2600	1300~2140	14.9~21.9	108~180
3/4 -10		1800~2640	1600~2250	15.4~28.0	130~188
	3/4 -16	2940~5200	2600~4500	20.8~51.8	208~478
7/8 - 9		3500~4304	[illegible]	34.8~47.7	250~845
	7/8 -14	3500~7400	[illegible]~6800	34.6~72.4	250~625
1 - 8		6500~7000	6000~6810	57.6~78.3	416~570
	1 -14	6400~10400	6600~9000	43.5~107.7	480~760
1 1/8 - 8		7600~12500	6500~10000	74.7~121.4	540~900
	1 1/8 -12	[illegible]~16800	7600~18500	50.2~164.8	580~1190
1 1/4 - 8		8200~14100	6000~14000	92.1~160.4	646~1160
	1 1/4 -12	12700~25800	11040~22600	127.1~269.1	920~1830

(5) Installation torque of non-structural nut

BOLT STUD SCREW NOMINAL DIA	* Standard bolt, stud and screw having a tensile strength of 40,000～60,000 psi			
	Shear nut		Tension nut and threaded machine part	
	kg - cm	in - ozs	kg - cm	in - ozs
#0-80	0.36～0.54	5.0 ～ 7.5	0.61～0.94	8.5～13.0
#1-64	0.65～0.97	9.0～13.5	1.1～1.6	15.0～22.5
#1-72	0.72～1.1	10.0～15.0	1.2～1.8	17.0～25.5
#2-56	1.1～1.6	15.0～22.0	1.8～2.7	24.5～37.0
#2-64	1.2～1.8	16.0～25.0	2.0～3.0	28.0～41.5
#3-48	1.5～2.3	21.0～31.5	2.6～3.8	35.5～53.0
#3-56	1.7～2.6	24.0～35.5	2.8～4.2	39.5～59.0
#4-40	2.2～3.3	31.0～46.0	3.7～5.5	51.5～77.0
#4-48	2.6～3.9	36.0～53.5	4.3～6.4	60.0～89.5
#6-32	4.2～6.3	58.0～87.5	7.0～10.5	97.5～145.5
#6-40	5.1～7.7	71.5～106.5	8.6～12.8	119.0～177.5
#8-32	6.0～8.0	83.3～111.1	10.0～13.0	138.9～180.6

BOLT STUD SCREW NOMINAL DIA	* Bolt, stud and screw having a tensile strength of 90,000 psi～125,000 psi			
	Shear nut		Tension nut and threaded machine part	
	kg - cm	in - ozs	kg - cm	in - ozs
#0-80	0.5～0.83	7.5～11.5	0.94～1.4	13.0～19.0
#1-64	0.9～1.4	13.5～20.0	1.6～2.4	22.5～33.5
#1-72	1.1～1.7	15.5～23.0	1.8～2.8	25.5～38.5
#2-56	1.6～2.4	22.0～33.0	2.7～4.0	37.0～55.0
#2-64	1.8～2.7	25.0～37.5	3.0～4.5	42.0～62.5
#3-48	2.3～3.4	32.0～47.5	3.8～5.7	53.0～79.0
#3-56	2.6～3.9	35.5～53.5	4.3～6.4	59.5～89.0
#4-40	3.3～5.0	46.5～69.0	5.5～8.3	77.0～115.0
#4-48	3.9～5.8	54.0～80.5	6.5～9.7	90.0～134.5
#6-32	6.3～9.4	88.0～131.0	10.5～15.7	146.5～218.5
#6-40	7.7～11.5	107.0～160.0	12.9～19.2	178.5～266.5

10. WEIGHT AND BALANCE

10.1 GENERAL DESCRIPTION

At time of delivery, the weight and balance is necessary data for the pilot or owner to compute individual loading schedules. All changes in weight and balance after delivery are the responsibility of the airplane owner and must be recorded in the aircraft log book. To facilitate computation of loading schedules, maintain a record of all weight and balance in the "Empty Weight and Balance Record" chart in section "WEIGHT AND BALANCE" of the PILOT OPERATING MANUAL.

10.2 MEASUREMENT, CALCULATION AND CHECK OF WEIGHT AND BALANCE

All necessary instruction information for measurement, calculation of the weight and balance of the airplane is in the "WEIGHT AND BALANCE" section of the PILOT OPERATING MANUAL.

10.3 EQUIPMENT NAMES, WEIGHTS AND CENTER OF GRAVITY

Basic equipment, their weights and center of gravity is located in the "WEIGHT AND BALANCE" section of the PILOT OPERATING MANUAL.

11. CHAIN WEAR LIMITS

11.1 VISUAL INSPECTION

(1) Gain access to chain by removing access panels, covers, etc.
(2) Carefully inspect the chain for dirt, rust and corrosion.
(3) Should the chain require cleaning, remove from aircraft and use a petroleum solvent for cleaning.

NOTE

Under no circumstances should rust or corrosion be removed with an acid because it may embrittle and crack the links.

(4) If after cleaning there is evidence of remaining rust or corrosion, also should there be pitted areas, especially on the rollers, the chain should be replaced.
(5) Visually inspect all links for uniform wear of the pin, roller and link.
(6) Re-install chain if removed.

11.2 MEASUREMENT OF CHAIN WEAR

11.2.1 Required tools

(1) Pull scale - 0 to 20 lbs (0 to 10 kgs)
(2) Linear scale

11.2.2 Procedures

(1) Remove chain from aircraft.
(2) Attach one end of the chain to a pull scale while securing the other chain end (or approximately 12 to 18 in. (305 to 457 mm) from the pull scale end) to a stationary fixture (such as hook through a link attached to a bench vise).
(3) Apply 18 $\pm$ 0.5 lbs (8.20 $\pm$ 0.24 kgs) pull on chain.
(4) Measure length of any 10 links of chain. Should the length "L" exceed the wear limit for chain listed, replace it with a new one.

CHAIN LOCATION	"L" MAX WEAR LIMIT
Landing gear Spoiler control	3.806 in (96.67 mm)
Rudder trim Elevator trim	2.538 in (64.47 mm)

(5) Re-Install chain.

12. NONDESTRUCTIVE INSPECTION

12.1 X-RAY INSPECTION

12.1.1 GENERAL

(1) Radiographic inspection is a nondestructive test method used for the inspection for airframe structure inaccessible and internal details of the parts. The inspection is accomplished by passing the X-ray beam through the part or assembly to expose a radiographic film in accordance with MIL-STD-453.

(2) This section shows a couple examples of radiographic inspection procedures as guidelines for the technically trained persons who are unfamiliar with MU-2 airplanes. Therefore, it is necessary to establish the appropriate exposure conditions of the equipment and film used.

12.1.2 APPLICABLE DOCUMENTS

The following documents will be referred to when conducting X-ray inspection.

MIL-STD-410 : "Nondestructive Testing Personnel Qualification and Certification"

MIL-STD-453 : "Inspection, Radiographic"

NBS Handbook 89 : "Methods of Evaluating Radiological Equipment and Material".

12.1.3 SAFETY

The use of X-rays in nondestructive inspection presents a potential hazard to operating and adjacent personnel, unless all safety precautions and protective requirements are observed. An excellent source of information on radiation protection can be found in National Bureau of Standards Handbook 93, "Safety Standard for Non-Medical X-Ray and Sealed Gamma - Ray Source".

12.1.4 ABBREVIATION

(1) KV=kilovoltage

(2) MA=Milliampere

(3) SFD=Source to Film Distance

12.1.5 PROCEDURE

(1) The surface of the part to be radiographed shall be clean. If cleaning is required, refer to the Maintenance Manual for proper cleaning materials.

(2) The direction of the central beam of radiation shall be, whenever possible, perpendicular to the surface of the film.

(3) Place the film as close to the surface of the material under test as practicable.

(4) Keep the distance from the X-ray source to the film (focal distance) great enough to produce a sharp image on the film.

NOTE

Settings specified in individual radiograph procedures in this section were established to provide quality radiographs. It may be necessary to vary the MA, time and KV settings due to differeces in equipment, film and method of processing in order to achieve the contrast, sesitivity and density specified. Therefore, the MA, KV and time settings should be construed as guidelines. X-ray equipment is considered acceptable provided it produces the quality radiographs specified for procedures contained in this manual.

(5) Use as low a potential (Kilo Voltage), as high a current (milliamperage) as is practicable to obtain proper sensitivity, contrast and density.

(6) Optimum densities are given for each inspection technique contained in this section; densities 1.0 to 2.0 are appropriate for the radiographic examination of this airplane.

(7) Exercise care in handling and storing undeveloped radiographic film to prevent the creation of blemishes which might interfere with interpretation of radiographs.

12.1.6 PERSONNEL QUALIFICATION

Personnel preparing parts, setting up or conducting the test shall be qualified MIL-STD-410, level I or higher. Personnel evaluating radiographs shall be qualified per MIL-STD-410, Level II or III.

12.1.7 EXAMPLES, TEST EQUIPMENT

The listed X-ray unit was employed (unless otherwise indicated in the specific inspection technique). When substitute equipment is used, it may be necessary to make appropriate adjustments to the established techniques.

NOTE

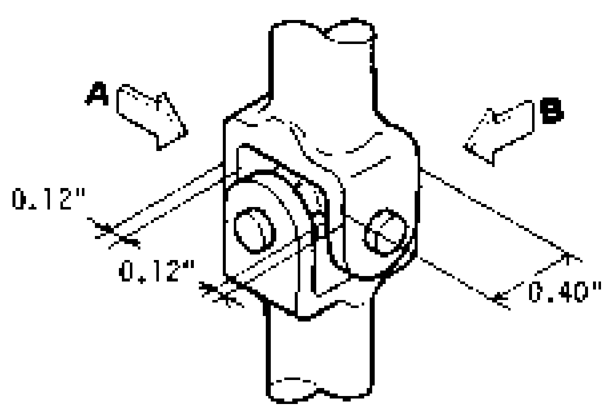
The use of RADIOATIVE ISOTOPES (Gamma Ray) for radiographic inspections contained in this manual is Prohibited.

(1) COMPONENT TO BE INSPECTED:

The following components are to be inspected by X-ray inspection as required in Maintenance Requirement.

Component	Material	Total Part thickness
Horizontal Stab. Front Spar Horizontal Stab. Rear Spar	Aluminum Aluminum	0.50"(root) to 0.17"(tip) 0.30"(root) to 0.13"(tip)
Flap control Sys. Universal Joint	Steel	0.24" for direction A 0.40" for direction B (NOTE)

NOTE : Thickness by direction



(2) RADIOGRAPHIC DATA

The following table represents the sample data which were used during actual radiographical inspection practice on several experiences of MU-2.
Therefore, specific figures or conditions herein are not to be considered as requirements for this radiographical inspection but to be utilized as a guide for a person who is engaging this inspection.

Data herein will vary from equipment to equipment and from condition to condition at the site where inspection is performed. All necessary requirements should be established by the qualified person and conformed to the requirements of MIL-STD-453.

Equipment	Andrex		Rigaku, Radioflex 100 GSB
Inspection Company	NDE-AID, Inc.(Fortworth, Tx)		MHI (Japan)
Component "ON" or "OFF" of aircraft	H.Stab. OFF	Universal Joint OFF	H.Stab. OFF
KV	100	150	65 - 70
Ma	4	4	5
Time	2 min 30 sec	6 min	1 min
SF Distance	46"	24"	20"
SP Distance	contact	contact	contact
Angle	0°	0°	0°
Filter	NO	NO	NO
Film Type	NDT55	NDT55	KODAK M
Film Size	7" X 17"	7" X 17" or smaller	7"X 17"
Lead Screens	Front - No Rear - .010"	Front - .005" Rear - .010"	Front - No Rear - .010"
Density	1.0 to 3.0	1.0 to 3.0	1.0 to 3.0
Penetrameter	As Required per MIL-STD-453		

TEMPORARY REVISION NO.2-2

This Temporary Revision No. 2-2 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-25/-26/-26A	MR-0215	2-80 thru 2-84
MU-2B-40	MR-0333	2-80 thru 2-84
MU-2B-35/-36A	MR-0218	2-83 thru 2-87
MU-2B-60	MR-0336	2-83 thru 2-87

Insert facing the page indicated above for the applicable Maintenance Manual. Retain this Temporary Revision until such time as a permanent revision on this subject is issued.

REASON : To change Fluorescent Liquid – Penetrant and Magnetic – Particle Inspection.

CHANGE : Paragraph 12.2 and 12.3 as follows.

12.2 FLUORESCENT LIQUID-PENETRANT INSPECTION

12.2.1 GENERAL

(1) Fluorescent Liquid-Penetrant Inspection is a nondestructive method for finding small cracks or other discontinuities that occur on the surface of a part or assembly which may not be evident by normal visual inspection.

This inspection is conducted by applying a liquid which penetrates into surface defects. The excess penetrant liquid is wiped off and a developer is applied to draw the penetrant from the surface crack or discontinuities. Visual indications are observed by the fluorescence of the penetrant under ultra violet (black) light. The penetrant method of inspection requires that the surface in the inspection area be thoroughly clean and free of paint.

(2) This section provides guidelines for Fluorescent Liquid-Penetrant inspections by technically trained persons.

12.2.2 APPLICABLE DOCUMENTS

The following documents will be referred to when conducting penetrant inspection.

(1) NAS 410 : "NAS Certification & Qualification of Nondestructive Test Personnel"

(2) ASTM E 1417 : "Standard Practice for Liquid Penetrant Examination"

(3) SAE AMS 2644 : "Inspection Material, Penetrant"

12.2.3 PROCEDURE

The fluorecent liquid-penetrant inspection should be conducted in accordance with ASTM E 1417 Latest revision.

12.2.4 PERSONNEL QUALIFICATION

Personnel preparing parts, setting up or conducting the inspection and evaluating its results shall be qualified per NAS 410.

12.3 MAGNETIC-PARTICLE INSPECTION

12.3.1 GENERAL

(1) Magnetic-Particle inspection is a method for locating surface and subsurface discontinuities in ferromagnetic materials (i.e. material capable of being magnetized), consequently, nonferromagnetic materials (such as aluminum alloys, magnesium alloys, copper alloys lead, nickel base alloys and many stainless steel alloys) can not be inspected by this method. Magnetic discontinuities not lying parallel to the magnetic field will cause a "leakage field" to be formed at and above the surface of the part. The presence of the leakage field is detected by pouring small ferromagnetic particles over the surface of the part. Some of these particles are magnetically gathered and held by the leakage field to form an outline indicating the location size shape and extent of the discontinuity.

WARNING

Improper operation of magnetic particle inspection equipment because of faulty equipment or by untrained persons can jeopardize the airworthiness of parts being inspected. Minute electrical burns caused during inspection by improper operation of the test equipment can result in eventual failure of the part.

(2) This section shows some examples of magnetic particle inspection procedures as guidelines for technically trained persons.

12.3.2 APPLICABLE DOCUMENTS

The following documents will be referred to when conducting Magnetic Particle Inspection
NAS 410 : "NAS Certification & Qualification of Nondestructive Test Personnel"
ASTM E 1444 : "Standard Practice for Magnetic Particle Examination"

12.3.3 PROCEDURE

The magnetic – particle inspection shall be conducted in accordance with ASTM E 1444 latest revision.

12.3.4 PERSONNEL QUALIFICATION

Personnel preparing parts, setting up or conducting the inspection and evaluating its results shall be qualified per NAS 410.

12.2 FLUORESCENT LIQUID - PENETRANT INSPECTION

12.2.1 GENERAL

(1) Fluorescent Liquid-Penetrant Inspection is a nondestructive method for finding small cracks or other discontinuities that are open to the surface which may not be evident by normal visual inspection. Fluorescent Liquid-Penetrant Inspection can be used on airframe parts and assemblies accessible for its application. This inspection is conducted by applying a liquid which penetrates into surface defects. Excess penetrant liquid is wiped off and optimum developer applied to draw the penetrant from the surface defects so that visual indications are obtained brilliantly by fluorescence of the penetrant under ultra violet (black) light. The penetrant method of inspection requires that the surface in the inspection area be thoroughly clean and free of paint.

(2) This section shows an example of Fluorescent Liquid-Penetrant inspection procedures as guidelines for the technically trained persons.

12.2.2 APPLICABLE DOCUMENTS

The following documents will be referred to when conducting penetrant inspection.

(1) MIL-STD-410 : "Nondestructive Testing Personnel Qualification and Certification"
(2) MIL-STD-6866 : "Inspection, Liquid Penetrant"
(3) MIL-I-25135 : "Inspection Materials, Penetrant"

12.2.3 PROCEDURE

(1) SURFACE PREPARATION

a. All surfaces of a work piece must be thoroughly cleaned by suitable chemical method free from any dirt, grease, oil, paint or any contaminants which would mask or close a defect open to surface, and must be water rinsed. No mechanical removing method should be applied since it would tend to mask defects. The surface should be completely dried before applying the liquid-penetrant.

b. The liquid-penetrant and area to be inspected shall have a temperature of 60 °F (15 °C) minimum to 130 °F (54 °C) maximum.

(2) PENETRATION

After the work piece has been cleaned, liquid penetrant is applied by aerosol spray or brush to form a film of penetrant over the surface being inspected. This film should remain on the area for a minimum of 30 minutes to allow maximum penetration of the penetrant into any surface openings that are present. The entire surface of the part inspected shall be maintained in the fluid wet condition until the penetrant has sufficiently penetrated into the defect interior.

NOTE: Please see the TEMPORARY REVISION that revises this page.

(3) REMOVAL OF EXCESS PENETRANT
Optimum removal of the excess penetrant is accomplished by wiping off as much of the penetrant as possible with a paper towel or a lint-free cloth, then wipe off the remaining penetrant with a clean cloth slightly dampened with the penetrant system cleaner ; and finally, with a dry paper towel or clean cloth so that the liquid-penetrant into the defect interior is left remaining as much as possible.

NOTE

The washing method such as flushing over the inspected surface shall be avoided since this will flush the penetrant from shallow defects.

(4) DEVELOPMENT
The developing agent is applied by aerosol spray to form a film over the surface being inspected. The developer acts as a blotter to assist the natural seepages of the penetrant out of any surface openings and to spread it at the edges to greatly magnify the apparent width of the crack. The developer also provides a uniform background to assist interpretation. Caution should be used in the application of the developer to provide the optimum coating thickness. If the coating is too thinly applied the penetrant will not be spread and a crack or other discontinuity will not be as easily detected. if the developer coating is thickly applied the penetrant might not bleed through the coating.

(5) INSPECTION
After being sufficiently developed, the surface is visually examined for indications of penetrant bleeding from surface openings. This examination must be performed under suitably darkened conditions for the penetrant to fluoresce during exposure to ultraviolet (black) light which meets the following minimum requirements : (1) 3200 to 4000 angstrom wave length (2) over 1000 $\mu W/cm^2$ at 15 inches (38 cm) from the surface to be inspected (3) darkness of under 1.5 foot-candle in the inspection area.

NOTE

Care should be taken not to leave it for too long as the penetrant may spread excessively resulting in misjudgement.

NOTE. Please see the
TEMPORARY REVISION
that revises this page.

(6) POST INSPECTION CLEANING

After completion of the fluorescent liquid penetrant inspection, the inspection areas are to be thoroughly cleaned by suitable solvent or water rinsed to remove the developer coating and any remaining traces of penetrant. Then, the exposed area shall be painted per the appropriate Maintenance Manual or Structural Repair Manual.

NOTE

All components of the penetrant inspection system must be from the same manufacturer and be designed to be used together. For instance, it is not permissible to use a penetrant from one manufacturer and a cleaner/remover from another manufacturer to inspect the same work piece.

12.2.4 PERSONNEL QUALIFICATION

Personnel preparing parts, setting up or conducting the inspection and evaluating of its results shall be qualified per MIL-STD-410.

12.3 MAGNETIC-PARTICLE INSPECTION

12.3.1 GENERAL

(1) Magnetic-Particle inspection is a method for locating surface and subsurface discontinuities in ferromagnetic materials (i.e. material capable of being magnetized) ; consequently, nonferromagnetic materials (such as aluminum alloys, magnesium alloys, copper alloys, lead, nickel base alloys and many stainless steel alloys) can not be inspected by this method. Magnetic discontinuities lying in a direction generally transverse to the direction of the magnetic field of the part magnetized for the test will cause a leakage field to be formed at and above the surface of the part. The presence of the leakage field denoting the discontinuity is detected by the use of finely divided ferromagnetic particles over the surface of the part. Some of the particles are magnetically gathered and held by the leakage field to form an outline indicating the location, size, shape and extent of the discontinuity.

NOTE: Please see the TEMPORARY REVISION that revises this page

WARNING

Improper operation of magnetic particle inspection equipment because of faulty equipment or by untrained persons can jeopardize the airworthiness of parts being inspected. Minute electrical burns caused during inspection by improper operation of the test equipment can result in eventual failure of the part.

(2) This section shows same examples of magnetic particle inspection procedures as guidelines for the technically trained persons.

12.3.2 APPLICABLE DOCUMENTS

The following documents will be referred to when conducting Magnetic Particle Inspection.

MIL-STD-410 : "Nondestructive Testing Personnel Qualification and Certification"
MIL-STD-1949 : "Inspection, Magnetic Particle"

12.3.3 PROCEDURE

(1) SURFACE PREPARATION
All foreign matters such as grease, paint, scale, and other matter that will impair the normal distribution of the magnetic particle, electro magnetic property, and identification, shall be removed from the surface of the item to be inspected.
(2) MAGNETIZATION
The inspection of both the circular magnetization and the longitudinal magnetization must be performed.

NOTE

a. In the magnetization of items to be inspected, it must be performed with sufficient care not to have any current burning.
b. The electric current flow time will be 1/2 to 1 second X more than 2 times.

NOTE: Please see the TEMPORARY REVISION that revises this page.

(3) INSPECTION

a. Unless otherwise specified, fundamentally, the inspection is performed by the direct current wet type continuous method and the fluorescent property magnetic particle inspection.

b. In the fluorescent magnetic particle inspection, the ultraviolet (black) light (which is 3200 to 4000 angstrom wave length) intensity illumination on the surface of the item inspected shall be more than 1000 $\mu w/cm^2$. Also, this examination must be performed under suitably darkened conditions that are under 2.0 foot-candle (20 lux) in the inspection area.

(4) DEMAGNETIZATION

The inspected items shall all go through the demagnetizer and have a complete demagnetization performed. If necessary, the confirmation of the result of the demagnetization shall be made with an adequate demagnetization meter. (field indicator etc.)

(5) POST INSPECTION CLEANING

After the completion of the inspection and demagnetization, all parts shall be sufficiently cleaned with proper cleaning solvent.

NOTE

Furthermore, the plug, mask, etc., used for protection purposes shall be removed after the cleaning.

12.3.4 PERSONNEL QUALIFICATION

Personnel preparing parts, setting up or conducting the inspection and evaluating its results shall be qualified per MIL-STD-410.

NOTE: Please see the TEMPORARY REVISION that revises this page.

CHAPTER

3

AIRFRAME

CHAPTER III

AIRFRAME

CONTENTS

1. GENERAL

The aircraft's structure is divided into three major components: Wing, fuselage, and empennage. The wing, being of an all-metal, cantilever, integral construction, is attached to the upper fuselage by 7075-T6 extrusion frame fittings with four bolts at F. STA4850 and F. STA5605. The wing's major components are: wing proper, flap, spoiler, and aileron trim tab. The engines are installed on the leading edges of both wings at W. STA2250, and the tip tanks are installed at the wing tips.

The fuselage consists of F. STA -170 to 8895 and the aft fuselage F. STA8895 to 10670, connected at F. STA8895 by four bolts. The cockpit and cabin, between F. STA1080 and F. STA8035, can be pressurized to 5.0(*1), 6.0 psi (*2) of the maximum differential pressure. The detachable nose cone under which the radar antenna is installed is forward of F. STA 180, is a honeycomb sandwich constructed radome. The forward electronics compartment is located in the upper section between F. STA 180 ~ 1080 and the nose landing gear, with its related mechanical components and landing light is located directly beneath it. The cockpit is situated between F. STA 2130 and F. STA 2690, the cabin between F. STA 2690 and F. STA 7845.

On the left hand side of the fuselage, between F. STA6495 and F. STA7250, is the entrance door, and on the right hand side, between F. STA4850 and F. STA5605, is the emergency door, respectively.

The main landing gear forward and aft doors are located at the side of bulge between F. STA4265 and F. STA5830. Air conditioning system, radio, and batteries are installed between F. STA8035 and F. STA8895.

2. WING

2.1 GENERAL DESCRIPTION

2.1.1 WING

The wing is divided into outboard section and inboard section outside of engine nacelle at W. STA2590. These two sections are joined together with four bolts and four shear pins. Inboard section is stressed skin construction utilizing two spars and longerons which run spanwise inside upper and lower skins. The front and rear Wagner spars are located at 22% and 60% of the wing chord, respectively. Ribs are placed every 11.8 to 13.8 in. (300 to 400 mm) intervals. Wing section inboard of engine nacelle (W. STA1950) forms an integral tank. The leading edge is hinged to wing section by a piano hinge and is detachable for ease of access. Outboard section is stressed skin construction utilizing two Wagner spars extended from those of the inboard section. Stiffeners are placed inside upper and lower skins to provide additional stiffness. Ribs are placed every 7.9 to 11.8 in. (200 to 300 mm) intervals. Between W. STA2590 and W. STA4350, provisions are made for metal auxiliary fuel tanks.

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*2 Aircraft S/N 661SA, 697SA and subsequent

2.1.2 FLAP (See Fig 3-1)

The double slot flap is installed along most of the wing trailing edge. The L.H. & R.H. flaps are divided into inner and outer sections. Each main flap is a stressed skin construction, and the front inner flap is of stressed skin construction and the outer flap is a metal spar construction with a foamed plastics core and an FRP outer skin. Both of them are attached to the main flap. The metal aileron trim tab is attached to the trailing edge section of the outer flap.

2.1.3 SPOILER (See Fig 3-1)

The spoiler is made of an aluminum extruded material, consisting of two sections (2ea for L.H. and R.H.) which are located behind the rear spar. It spans 31.5%~96% of the half wing and is attached to the rear wing spar.

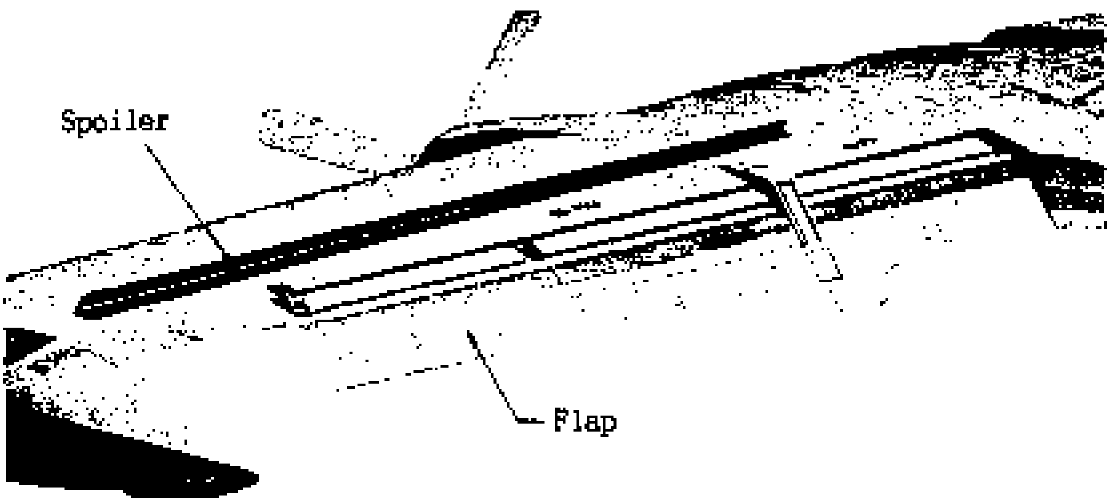

Fig 3-1 Flap & spoiler

2.2 REMOVAL AND INSTALLATION OF WING

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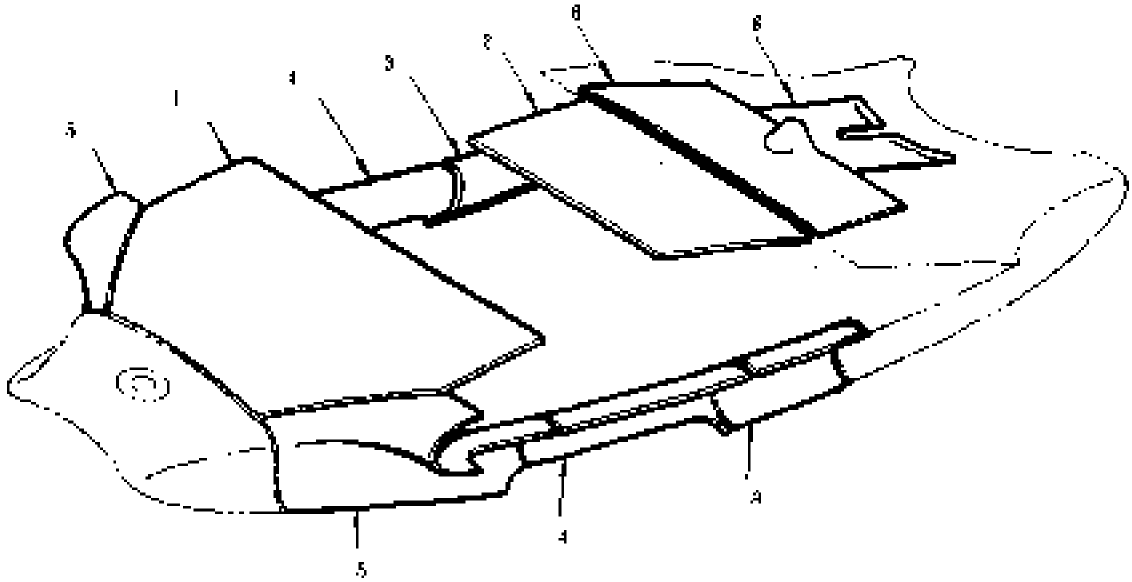
2.2.1 GROUND SUPPORT EQUIPMENT REQUIRED

(1) Sling-wing GSE 016A-99025
(2) Remover-wing-fuselage mating fitting GSE 016A-99029

2.2.2 PREPARATION

Before removing the wing, remove the following items. (See Fig 3-2)
(1) Center wing front upper access panel.
(2) Center wing rear upper access panel.
(3) Wing rear lower fillet (L.H., R.H.)
(4) Wing center lower fillet (L.H., R.H.)
(5) Front side fillet (L.H, R.H.)
(6) Rear fillet



1. Center wing front upper access panel
2. Center wing rear upper access panel
3. Wing rear lower fillet
4. Wing center lower fillet
5. Front side fillet
6. Rear fillet panel

Fig. 3 - 2 Removal of fillets and panels

2.2.3 PROCEDURES

(1) Fuel system

(a) Drain fuel.
(b) Remove pressurizing air line (W.STA 455) for tip tank. (See Fig. 3-3)
(c) Remove electric plug of pressurizing air line shutoff valve.
(d) Remove drain lines of L.H. and R.H. boost pump.
(e) Remove drain line of center fuel tank.

(2) Engine

(a) Insert rig pin into pulley of engine control cable at F.STA 6035.
(b) Remove turnbuckles (8 ea.) of engine control cable located behind the rear center wing spar.

(3) Air conditioning and wing de-icer system (See Fig. 3-4)

(a) Disconnect air pressure line at L.H. side of center wing trailing edge. (F.STA 6215)
(b) Remove ducts for engine bleed air at both side of center wing trailing edge. (See Fig. 3-5)
(c) Disconnect wing de-icer line at R.H. side of center wing leading edge. (F.STA 4745)

(4) Electrical and instrument system

(a) Remove electrical plugs P255, P256, P257, P258, P275 (*2), P276 (*2), P483 (*1), P484 (*1), P486 (*2) and P490 at both sides of center wing leading edge.

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(b) Remove electrical panel assembly located at center wing leading edge. (See Fig. 3-6)

(c) Disconnect electric wire (fuselage side) from terminal blocks TB405, TB406 and TB407 (*2) at center wing trailing edge and disconnect electric wire for generator from J277 and J278.

(d) Remove bonding jumper at center wing trailing edge. (F. STA 6125) (See Fig. 3-7)

(e) Remove electric wire of left and right boost pumps from boost pump line filters.

(f) Remove two clamps binding electric wire and located at center wing trailing edge. (W. STA 6125)

(g) Disconnect tubing for vacuum system for the gyro drive from elbow at center wing leading edge.

(h) Disconnect auto-pilot cable (if installed). (See Fig. 3-8)

*2 Aircraft S/N 661SA, 697SA and subsequent

Fig. 3-3 Removal of pressurizing air line for tip tank

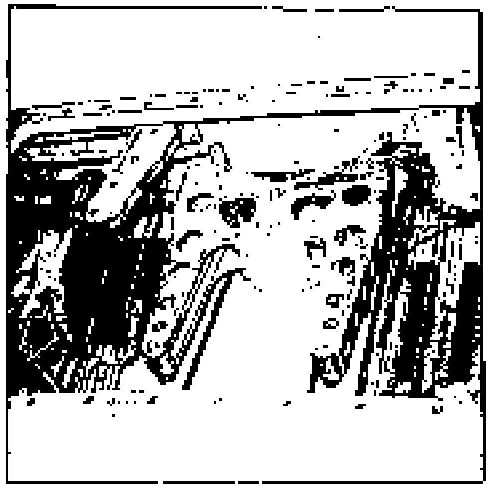

Fig. 3-4 Removal of turnbuckle for engine control cable

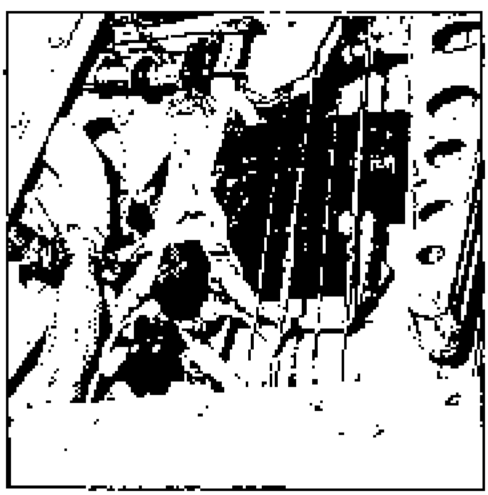

Fig. 3-5 Removal of bleed air duct

Fig. 3-6 Removal of electric panel assembly

(5) Flight control system

(a) Remove control cable of spoiler control system at center wing trailing edge. (F. STA 5772) (See Fig. 3-8)

(i) Loosen turnbuckle of cable, under the floor, F. STA 6725.

(ii) Remove retainer from cable fitting of quadrant in differential linkage.

Fig. 3-7 Removal of bonding jumper

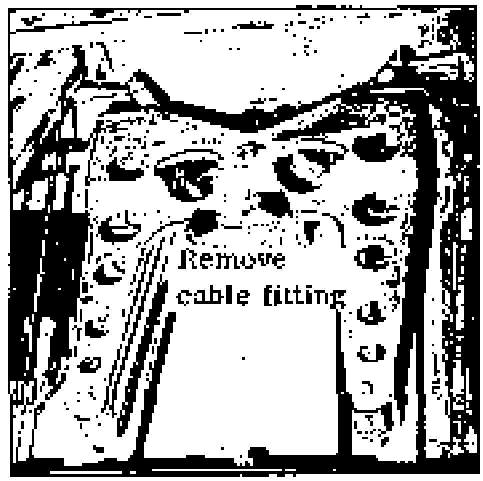

Fig. 3-8 Removal of spoiler control cable

(6) Wing

(a) Lift wing with sling fixtures attached at 3 points on W. STA 0 of the upper front spar surface and W. STA 1150 of the upper rear spar surface (both sides). Remove rear spar wing-fuselage connection bolts first and then front spar wing-fuselage connection bolts. (See Fig. 3-9)

(b) Lift wing, in such a manner that no impact load is applied to the wing. Rest it on a dolly with supports at W. STA 3950 (both sides).

CAUTION

Walk only on spar flanges or ribs. Do not step on the forward or rear wing fillets.

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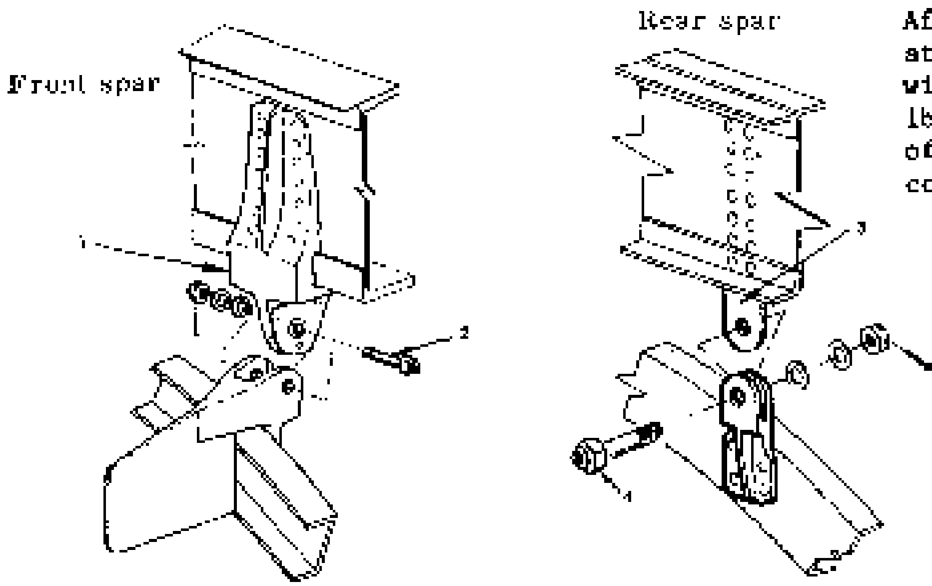

After installing attaching bolt with 24~89 in-lbs (28~103 kg-cm) of torque, install cotter pin.

1. Front spar fitting
2. Attaching bolt
3. Rear spar fitting
4. Attaching bolt

Fig. 3 - 9 Removal of wing

2.3 REMOVAL AND INSTALLATION OF OUTER WING (See Fig. 3-10)

(1) Remove inboard and outboard flap assemblies in accordance with Para. 9.2, CHAPTER V, Flight Control System.

(2) Remove spoiler assembly in accordance with Para. 5.2, CHAPTER V, Flight Control System.

(3) Remove outer wing side access panel of inner and outer wing connecting section.

(4) Remove access panel at outer wing leading edge lower surface. (W. STA 2800)

(5) Disconnect fuel vent line at W. STA 2810.

(6) Disconnect hose for wing leading edge de-icer boots at W. STA 2870.

(7) Disconnect pressurizing air line for tip tank at W. STA 2770.

(8) Disconnect fuel transfer line for tip tank at W. STA 2650.

(9) (*1) Remove electrical plug from receptacle J415 located at inner wing leading edge W STA2590, and J441 located at outer wing leading edge W STA2770.

(*2) Remove electrical plug from receptable (J441, J415 or J401 for L.H. and J442, J416 or J402 for R.H.) located at outer wing leading edge. (W. STA2785)

(10) Remove nacelle upper access panel at W. STA 2300.

(11) Disconnect fuel transfer line for outer wing at W. STA 1950.

(12) Remove inner and outer wing attaching bolts (4 ea.).

(13) Reinstall in reverse sequence of removal.
Take care of the following items when installing outer wing.

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*2 Aircraft S/N 661SA, 697SA and subsequent

(a) Apply rust preventive oil (MIL-C-16173 grade 1, MIL-C-11796 class 3 or MIL-C-16173 grade 2) on bolt shank when installing bolt.

(b) Installation torque values for bolts are as follows;

Rear spar upper	690 ~ 990 in-lbs
Front spar upper	1000 ~ 1440 in-lbs
Rear spar lower	1300 ~ 2160 in-lbs
Front spar lower	2500 ~ 4500 in-lbs

When attachment is used on top of torque wrench, torque values are corrected as follows;

$$\text{Corrected torque value} = \frac{\text{Wrench length} \times \text{torque value specified above}}{\text{Wrench length} + \text{attachment length}}$$

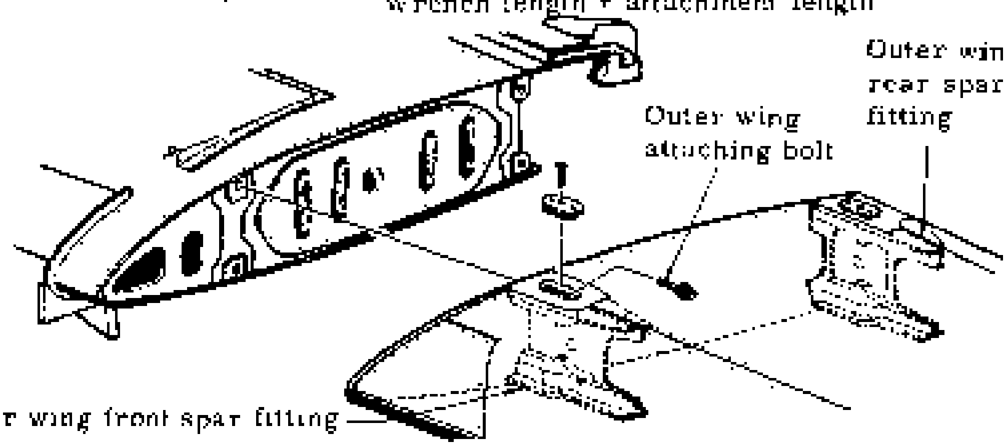


Fig. 3-10 Removal of outer wing

2.4 REMOVAL AND INSTALLATION OF WING LEADING EDGE (See Fig. 3-11)

(1) Remove upper and lower covers of wing front spar, W.STA 1100 ~ 1300.

(2) Pull out hinge pins (1 ea. upper and lower) running from W.STA 580 through W.STA 1950.

(3) Disconnect hose of boot.

(4) Remove wing leading edge.

(5) Reinstall in reverse sequence of removal.

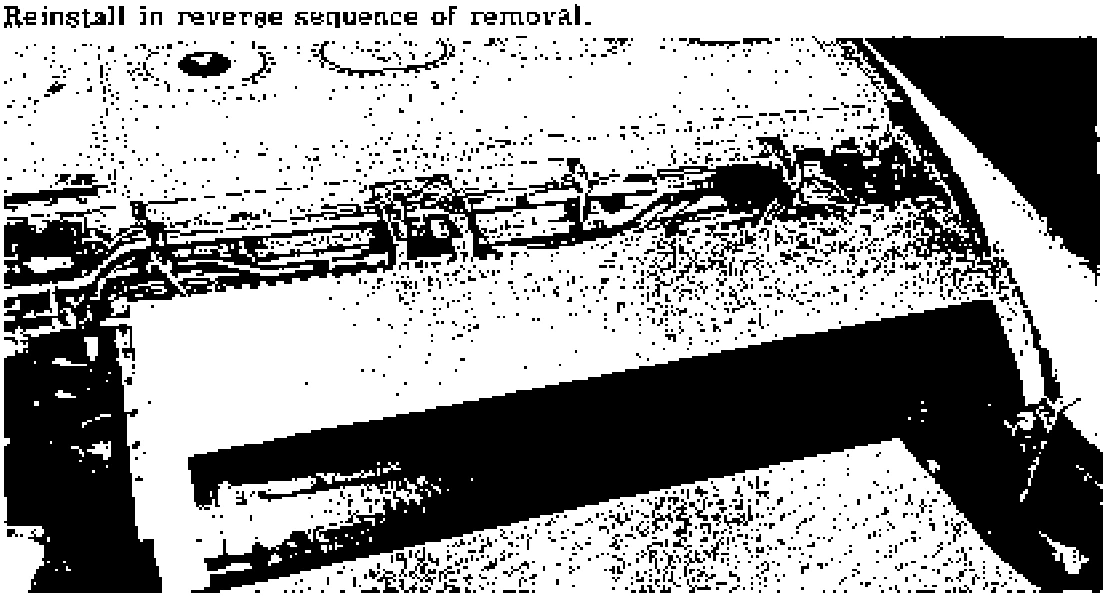


Fig 3 - 11 Removal of wing leading edge

3 EMPENNAGE

3.1 GENERAL DESCRIPTION

The empennage consists of a horizontal stabilizer, elevator, vertical stabilizer, and rudder. Trim tabs are attached to the trailing edges of the elevator and rudder.
Empennage also includes the detachable dorsal fin made of FRP and horizontal stabilizer leading edge fillet made of metal.
The horizontal stabilizer is all metal, integral (L.H. and R.H.), 2 spar, stressed-skin construction and is attached to fuselage with 4 bolts.
The vertical stabilizer is all metal (except the tip), cantilever, 2 spar, stressed skin construction and is attached to the aft fuselage.
The 18.5 in. (470mm) long tip section is made of FRP for VHF antenna installation.
The elevator is all metal, mono-spar, stressed skin construction with both L.H. and R.H. sections made in one piece connected by a torque tube. The elevator is attached to the horizontal stabilizer at a central hinge and 2 hinges on each side.
Rudder is all metal (except the tip which is made of FRP), mono-spar, stressed skin construction and is attached to the vertical stabilizer with 2 hinges.

3.2 REMOVAL AND INSTALLATION OF EMPENNAGE

3.2.1 HORIZONTAL STABILIZER

(1) Remove tail cone, horizontal stabilizer leading edge fillet, and inspection panel between F.STA 9903.5 ~ 10240.

(2) Remove elevator control cable at turnbuckle, F.STA 8605 (Main-Aft fuselage mating point).

(3) Remove trim tab control cable at turnbuckles, F.STA 8370 and 9625.

(4) Remove elevator control cable terminal at elevator quadrant. (See Fig. 3-12)

NOTE

After removing control cable on the main fuselage, keep the cable in tension by means of rubber cord.
It is not necessary to remove down spring.
Rigging pin is to be inserted at quadrant section.

(5) Remove clamp at the fuselage frame section, F. STA 9903.5, and remove hose for de-icing system. (See Fig. 3-13)

(6) Remove bonding jumper.

(7) Remove rear spar stabilizer-fuselage connecting bolts, and then remove front spar stabilizer-fuselage connecting bolts. (See Fig. 3-14 and Fig. 3-15)

(8) Pull horizontal stabilizer rearward.

(9) Install in reverse sequence of the removal.

NOTE

To install horizontal stabilizer, install stabilizer-fuselage connecting bolts in forward spar before installing stabilizer-fuselage connecting bolts in rear spar, and then install the bolts in rear spar.

3.2.2 VERTICAL STABILIZER

(1) Remove tail cone dorsal cover, fillet and access cover at FUS. STA 9903.5 thru 10240.

(2) Disconnect tail light electric wire at terminal block (TB319).

(3) Disconnect rudder control cable at turnbuckle located on FUS. STA 8605 (main fuselage to rear fuselage connection).

(4) Disconnect rudder trim tab control cable at turnbuckle locate on FUS. STA 9625.

NOTE

Maintain keep the tension, after disconnecting control cables in the forward fuselage side with rubber cords or other suitable means.

(5) Remove anti-ice equipment hose after removing clamps at forward top part of FUS. STA 9195 fuselage frame.

(6) Remove rudder from vertical stabilizer in accordance with Chapter 5, paragraph 4-2.

(7) Remove bolts connecting front spar fittings to fuselage while hanging vertical stabilizer in a suitable manner. Next, remove bolts connecting rear spar fittings to fuselage.

NOTE

Secure the shims (etc) used on surface contacting with fuselage side fittings, to the fuselage or to the vertical stabilizer in a suitable manner in order to keep them from becoming lost.

NOTE

Record the number and location of shims.

(8) Remove the vertical stabilizer lifting slowly upward taking care that undue pressure is not exerted between the spar fitting and the fuselage side fitting.

(9) Install in reverse sequence of removal.

3.2.3 ELEVATOR AND RUDDER

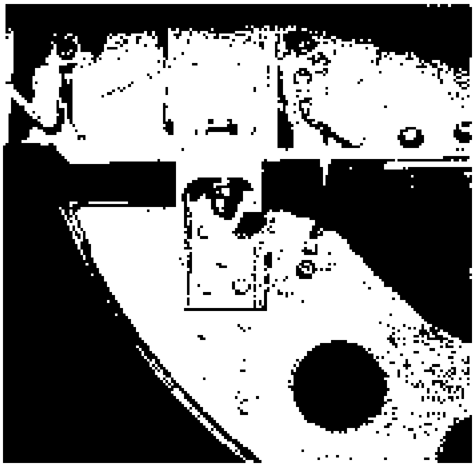
For removal and installation of elevator, see Chapter V section 3.2, and for removal and installation of rudder, see Chapter V section 4.2 of this manual.

Remove elevator control cable fitting
Fig. 3-12

Remove hose for de-icing system
Fig. 3-13

Tighten stabilizer-fuselage connecting bolt in forward spar with a torque of 60 to 85 in-lbs (69 to 98 kg-cm) and apply alignment mark. If it is hard to install cotter pin, nut may be tightened up to 140 in-lbs (161 kg-cm) torque maximum.

Fig. 3-14



Tighten stabilizer-fuselage connecting bolt in rear spar with a torque of 140 to 203 in-lbs (161 to 234 kg-cm) and apply alignment mark.

Fig. 3-15

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3.3 BALANCING OF ELEVATOR AND RUDDER

(1) For elevator and rudder, balance surfaces to the following required conditions in static condition.

Surface	Required Conditions
Elevator	0 to 3.17 in-lbs Over Balance (0 to 3.65 Kg-cm)(for both sides) Nose Heavy, Painted LH & RH
Rudder	0 to 9.6 in lbs(0 to 11.06 Kg-cm) Over Balance, Nose Heavy, Painted

(2) Adjustment

For both elevator and rudder, adjust mass balance in the following manner.

(a) To correct under balance, add the adjustment washer (010A-22012 or JIS-H-4301), each weighing 0.044 lbs(20 g), and for over balance, remove the adjustment washer or scrape the inside of mass balance surfaces to the above limits specified in Para. (1).

4. FUSELAGE

4.1 GENERAL DESCRIPTION

The fuselage is all metal, semimonocoque construction and consists primarily of longerons in ˥ or ⊓ section and a strong box beam keel located in the lower section of fuselage between F.STA 1275 and F.STA 7845.
The frames in] section are arranged at 11.8 in.(30 cm) intervals. The pressure bulkheads are made from stiffened panel sheets, while the floor panels are of sandwich construction with foamed plastic core and 2024-T3 clad sheet surfaces.
The wing is attached to 7075-T6 extrusion frame at F.STA 4850 and F.STA 5605.
The bulges are provided between F.STA3130 and F.STA7530 to retract main landing gears. The antenna is installed in the forward section of the bulge, main landing gear doors in the center section and step in the rear section.
The windshield, and windows of cabin, entrance door and emergency exit door are formed by one piece of Plexiglas respectively and consist of double Plexiglas panes which can be replaced.

4.2 NOSE CONE

4.2.1 REMOVAL AND INSTALLATION (See Fig. 3-16)

(1) Remove electronics compartment door and inspection doors (F. STA180 1080).

(2) Supporting nose cone, remove attaching bolts at F. STA180 frame.

(3) Install in reverse sequence of removal.

(4) After installation, seal the gap between nose cone and F. STA180 frame with sealant PR341 so that mold line is flush.

CAUTION

Nose cone should be handled with care.

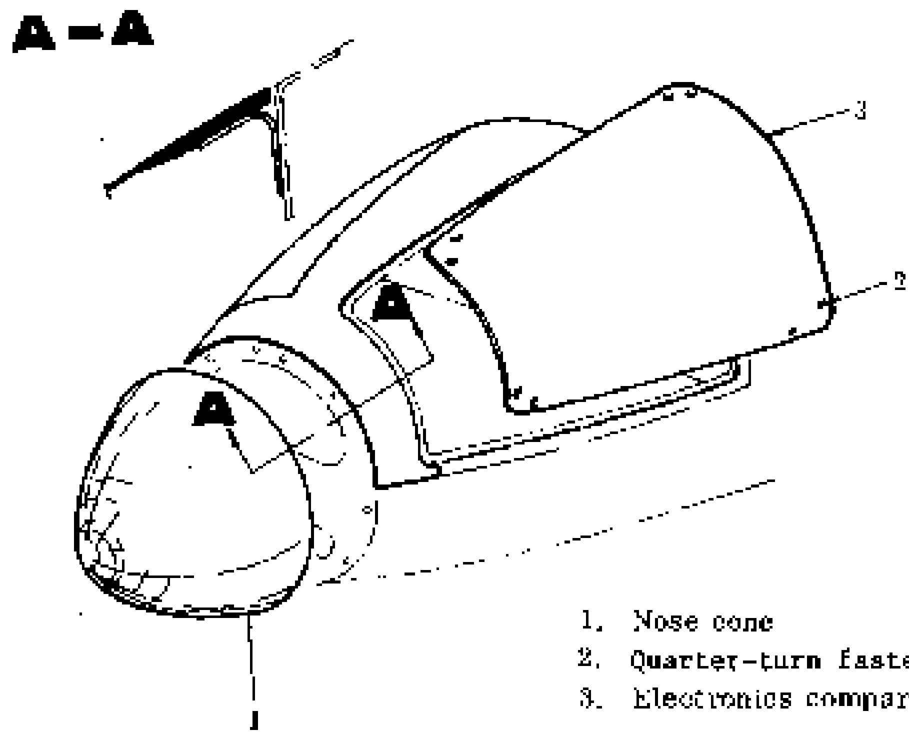


Fig. 3-16 Nose cone and electronic compartment door

4.3 ELECTRONICS COMPARTMENT DOOR (See Fig. 3-16)

4.3.1 REMOVAL AND INSTALLATION

(1) Turn quarter-turn fasteners counterclockwise to disconnect.

(2) Remove the door from the airframe.

(3) Install in reverse sequence of removal.

4.4 ENTRANCE DOOR

4.4.1 CONSTRUCTION (See Fig. 3-17)

The entrance door is opened outward, attached to the airframe with hinges ③ at the front end of the door. It opens forward to about 112 degrees. The door latching rods engage five steel latch plates on each side of the fuselage door frame, four at the corners and the one at rear side serves as an open lock.
The door has inside and outside handles ⑤ at about the center of door and the handles are connected mechanically to latch rods ③ . It is possible to lock or unlock the door by rotating handle. When latch rod is in unlocked position, a warning light notifies pilots of danger. Pressure lock mechanism ⑩ and open

lock mechanism ⑫ are provided to prevent handle from being turned while cabin pressurization is on or the door is opened. As the cabin is pressurized, an inflatable seal ⑬ is provided on the entrance door for pressure sealing.
The key lock is located on the outside door handle and is operated by pushing and turning the key clockwise approximately 180 degrees. A chloroprene rubber seal is also provided as a weather seal when the inflatable seal is not inflated.
In the center of the door at hinge side, a gust lock device ⑭ is provided to keep the door open even in the wind below 15 Kts.

4.4.2 REMOVAL AND INSTALLATION

(1) Remove a spring for gust lock device.

(2) Remove dot buttons.

(3) Remove retainer rings and remove pins.

(4) Install in reverse sequence of removal.

(5) Remove and reinstall window according to Para. 5.3.1 and 5.3.2.

4.4.3 INSPECTION

(1) Door handle can be operated by not more than 30 lbs (13.6 kgs) when pulled down to a point 1 in. (25 mm) inside of the handle edge.

(2) When the door is closed, a tight fit is obtained by pushing down the latch rod inside the cabin.

(3) Latch rod is correctly adjusted.

(4) Handle stopper is correctly adjusted.

(5) Rubber seal shows no indication of damage.

(6) Cable tension is correctly adjusted.

4.4.4 ADJUSTMENT

(1) Adjustment of handle operating force.
Adjust the position of bracket at the door frame.
Adjustments should be made in the following sequence.

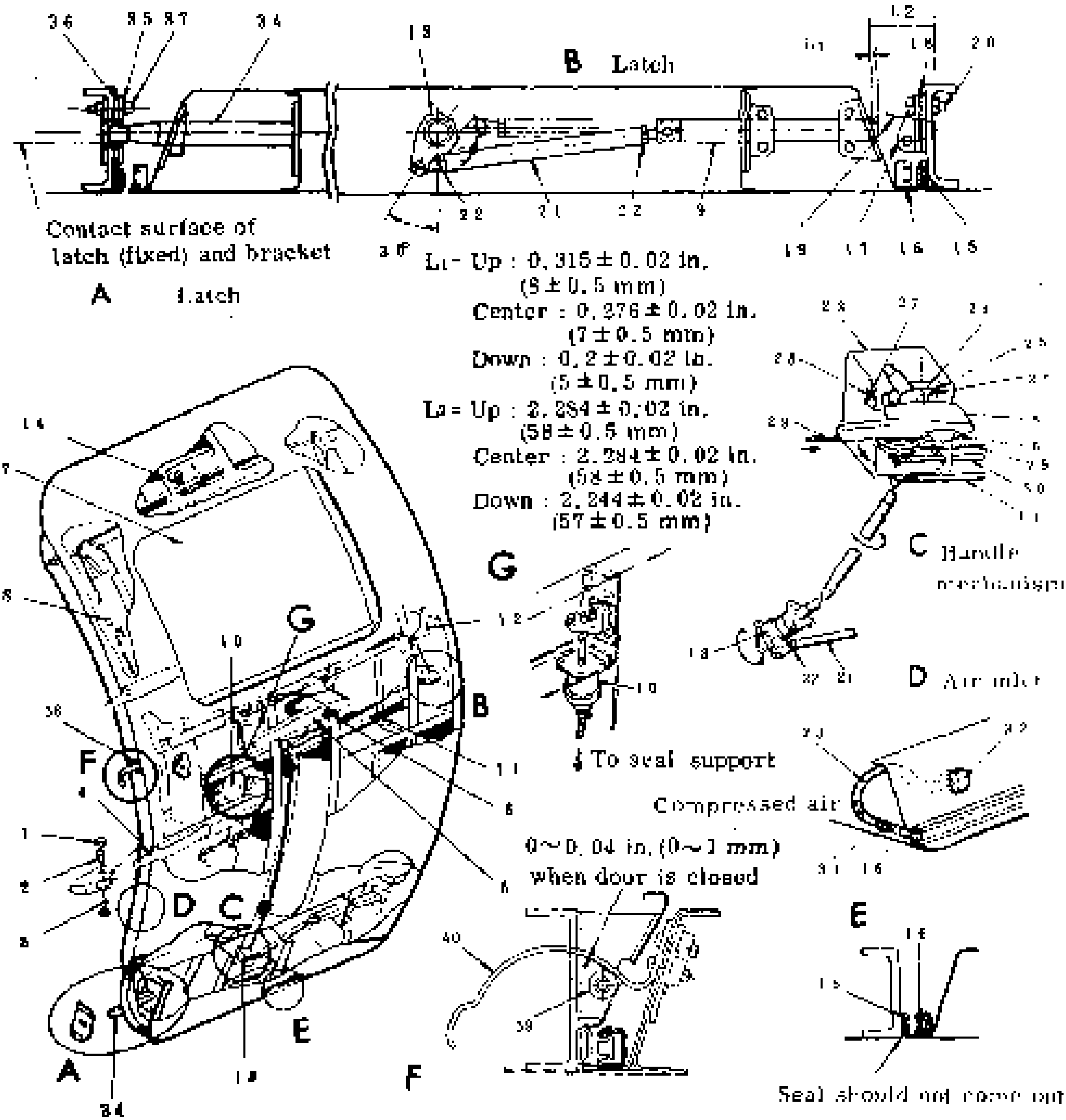
(a) center bracket of rear side
(b) upper bracket of rear side
(c) lower bracket of rear side
(d) upper bracket of front side
(e) lower bracket of front side

Airplane should not be on jacks and fuel tanks should be empty if possible.

NOTE

(i) Missmatch between door and skin is not more than 0.06 in. (1.5 mm).
(ii) Shim ⑫ and ⑬ used once should not be reused.
(iii) Installation torque for bolt ⑬ and ⑰ is 75 ~ 100 in-lbs. (86 to 115 kg-cm).

(2) Adjustment of latch rod clearance (See Fig 3-17 Detail A)
Loosen bolt ⑤ and adjust spring ⑥ in the installed position. Installation torque for bolt is 75 ~ 100 in-lbs. (86 to 115 kg-cm).



1. Dot button
2. Pin
3. Retainer ring
4. Hinge
5. Outside handle
6. Key
7. Window glass
8. Turnbuckle
9. Latch rod
10. Pressure lock mechanism
11. Handle feeling mechanism
12. Open lock mechanism
13. Crank
14. Quadrant crank
15. Weather seal
16. Inflatable seal
17. Bracket
18. Shim
19. Spring
20. Bolt
21. Rod assembly
22. Adjusting nut
23. Gear Box
24. Stop
25. Bolt
26. Nut
27. Bolt
28. Nut
29. Cable
30. Center latch mechanism
31. Clamp-air inlet
32. Seal support
33. Hose
34. Latch (fixed)
35. Bracket
36. Shim
37. Bolt
38. Gust lock device
39. Roller
40. Spring

Fig. 3-17 Entrance door

(3) Adjustment of latch rod (See Fig. 3-17 Detail "B")
Loosen two adjusting nuts and turn rod assembly so that L_1 is 0.315 ± 0.02 in. (8 ± 0.5 mm) for rear upper clutch, 0.276 ± 0.02 in. (7 ± 0.5 mm) for center clutch, 0.2 ± 0.02 in. (5 ± 0.5 mm) for rear down clutch.

(4) Handle stopper adjustment (Fig. 3-17 Detail "B" & "C")
With the door in the open position, adjust the stop bolt , located in the gear box, up or down to set the lower crank angle to 30°. Adjust the cable tension at turnbuckle to set the quadrant crank angle to 30°. With the door open depress the open lock mechanism and turn the handle to the "close" position and release the door open lock mechanism. Adjust the stop bolt , located in the gear box, up or down such that L_2 is 2.283 ± 0.02 in. (58 ± 0.5 mm) for the top and center latchse; and 2.244 ± 0.02 in. (57 ± 0.5 mm) for the bottom latch. After the above adjustments have been made, insert the key and check for key locking operation. Restore door to original open condition.

(5) Adjustment of cable tension (See Fig. 3-18)
Adjust cable tension with turnbuckle .

(6) Adjustment of latch (fixed) and bracket (See Fig. 3-17 Detail "A")
Remove bolt and replace shim . Install bracket with bolt so that latch (fixed) slightly contacts with bracket hole at the position shown in the figure when the door is closed.

(7) Adjustment of gust lock device (See Fig. 3-17 Detail "F")
Adjust spring installation so that the clearance between roller and spring may be 0 to 0.04 in. (0 to 1 mm) when the door is closed.

4.4.5 REPLACEMENT OF SEAL RUBBER (See Fig. 3-17 Detail "D" & "E")

4.4.5.1 When the inflatable seal rubber is damaged, replace it with a new seal in accordance with the following procedures.

(1) Loosen clamp at air inlet section; remove rubber hose.
(2) Peel off rubber seal from bonded surface in such a manner that channel is not damaged.
(3) Before attaching new rubber seal, clean channel, rubber seal sides and bottom with MEK or white gasoline.
(4) Apply one coat of cement EC-880 or EC-870 on channel and bottom of seal.
(5) Wipe the cement lightly with toluol.
    (a) Wipe about 24 inches (610 mm) and place seal on the surface immediately after wiping.

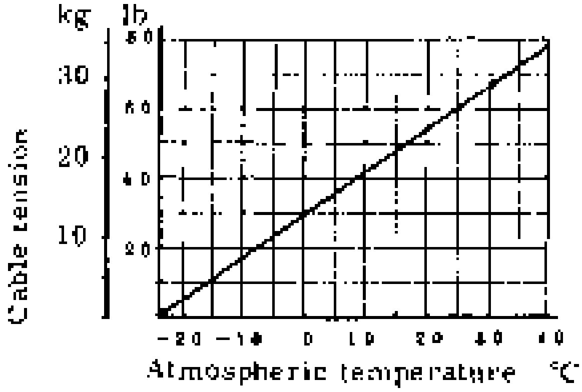
NOTE

Installation is to be started from inlet section and seal should not be over-stretched.

    (b) After determining attaching location exactly, press it firmly to channel so that it is bonded.
    (c) Leave seal for at least 4 hours before use, and 8 hours in case of no pressing.

4.4.5.2 When the weather seal is damaged, replace it with a new seal in accordance with the following procedures.

(1) Weather seal should be removed from its edge.



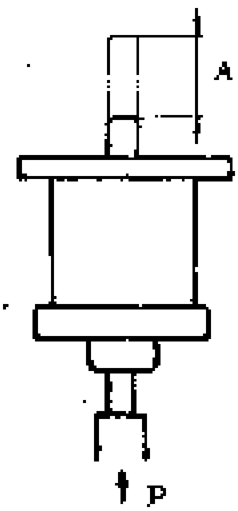
tolerance: +7 -0 lbs
(+3.2, -0 kg)

Fig 3-18 Tension of entrance door cable

(2) Before cementing, clean the surface with MEK or white gasoline.

(3) Apply one coat of cement EC-880 or EC-870 (Sumitomo, 3M) in accordance with the same procedures as inflatable seal. When the door is closed, weather seal should not come out from clearance.

4.4.6 OPERATIONAL CHECK OF PRESSURE LOCK MECHANISM



Operate the pressure lock mechanism by pressurized air P (15 psi) and measure the length of stroke A.
A shall be 0.63 ± 0.04 in (16 ± 1 mm).
In accordance with the procedure described in Para. 3.9.2, Chapter VIII, the pressure lock mechanism can be operated by pressurized air of 15 psi.(1 kg/cm²).

4.5 EMERGENCY EXIT DOOR

4.5.1 CONSTRUCTION (See Fig 3-19)

The emergency exit door, located on the R.H. side beneath the wing, is opened inward. Pressure load is carried by the 4 upper and lower hooks ⑨ of the door on the airframe side, and the air load, from outside of airframe, is carried by 2 latch pins ⑦ in the lower section of the door and the latch pin ③ in upper section of door. The rubber seal ⑫ around the door comes in close contact with the striker on the airframe side for a pressure-tight seal.

4.5.2 TO OPEN THE DOOR

(1) Pull the handle ① until it reaches the stop and the bell crank ⑤ rotates and the latch pin ③ disengages.

(2) Open the door, supporting it with the hands since it falls inward with 2 latch pins in the lower serration plate as hinges.

4.5.3 ADJUSTMENT

(1) Adjustment of door

(a) Circumference of door (See Fig. 3-19 Detail "A")

Mating of the door pin ⑦ and the hook ⑨ on the airframe side can be adjusted by increasing or decreasing the number of washers ⑧.

(b) Radial direction of door (See Fig. 3-19 Detail "B", "C")

Adjust the hook ⑨ and the serration plate ⑩ on the airframe by moving them in and out so there is no play in the in-and-out direction with the door closed.

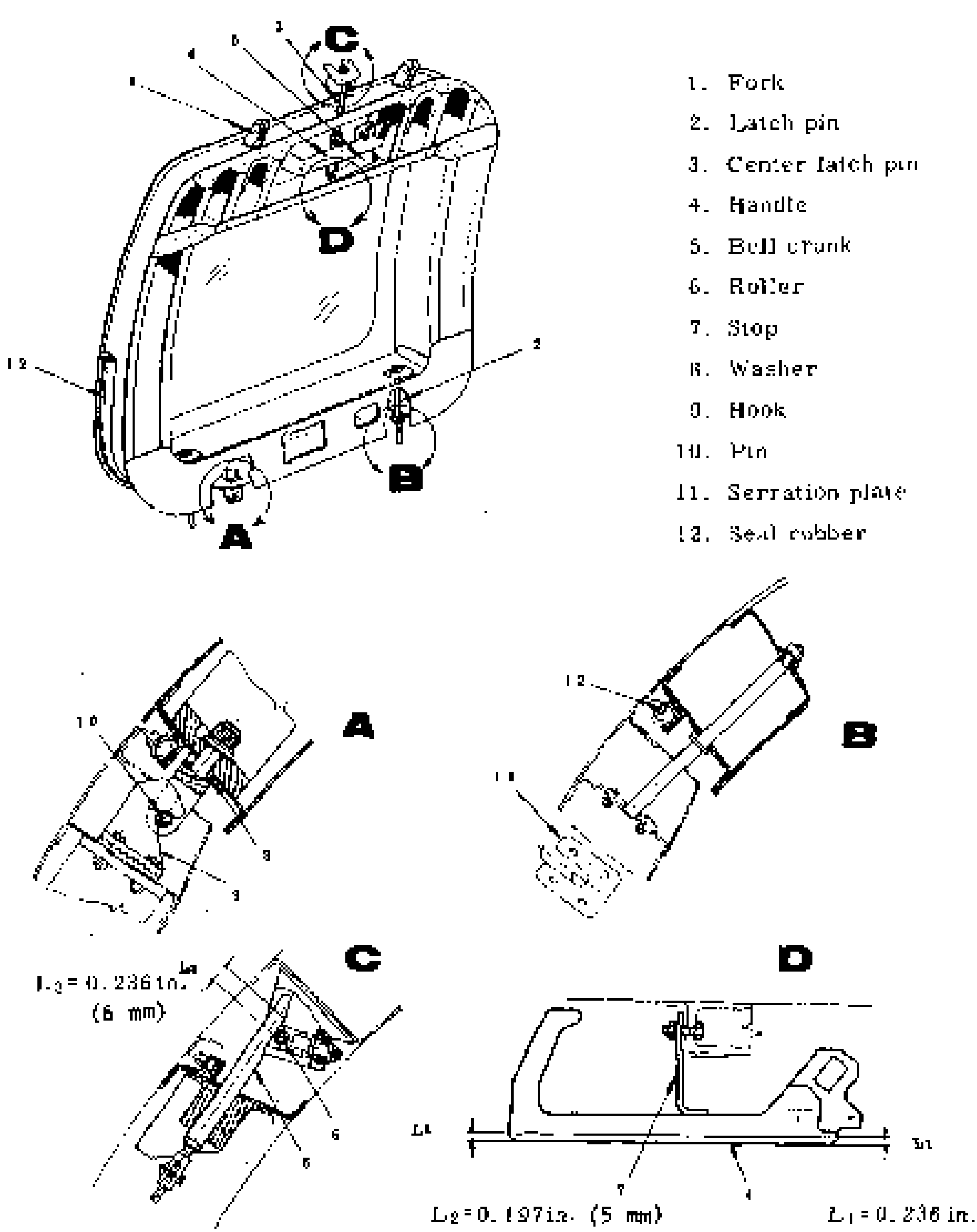


Fig 3 - 19 Emergency exit door

(2) Mechanical adjustment

(a) Mating of the latch pin ③ and the fork on the airframe side is adjusted by means of the fork thread so that the protrusion of the latch pin L_3 is 0.236 in. (6 mm). (See Fig. 3-19 Detail "C")

(b) Adjust fork thread so that roller ⑥ and latch pin are in contact. (See Fig. 3-19 Detail "C")

(c) Adjust stop ⑥ by moving it up or down so that distance between the handle ④ and the inner panel line L_2 is 0.197 in. (5 mm) with the door closed. (See Fig. 3-19 Detail "D")

4.5.4 REMOVING AND REINSTALLATION OF WINDOW

Remove and reinstall according to Paragraph 5.3 of this manual.

4.6 NOSE LANDING GEAR DOOR

4.6.1 REMOVAL AND INSTALLATION (See Fig. 3-20)

(1) Remove access doors at F.STA 640 ~ 1080 and F.STA 1080 ~ 1275.

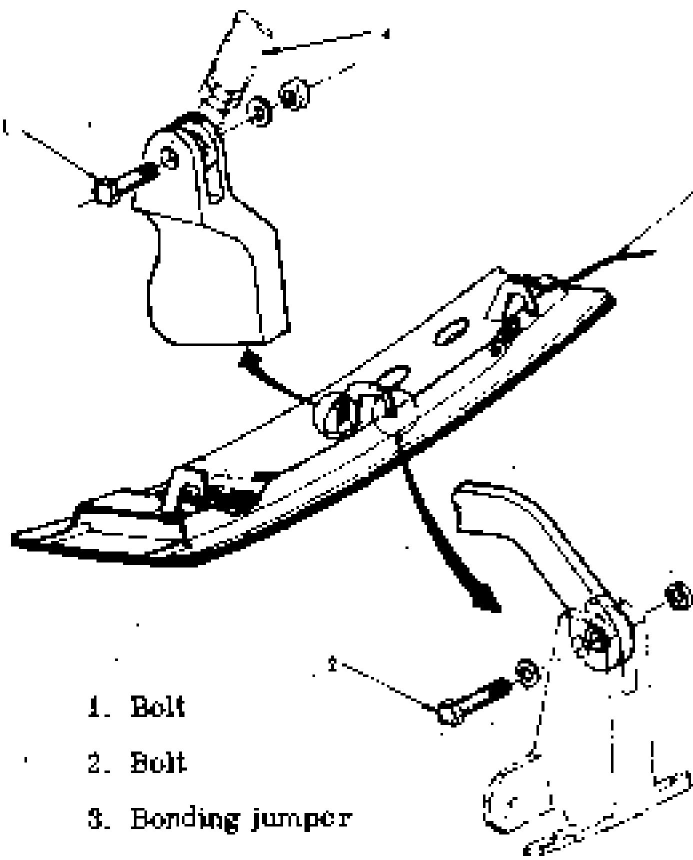
(2) Remove nose cone (or radome) and landing lights.

(3) Open the door and disconnect bonding jumper.

(4) Remove attaching bolts of actuator rod for door operating mechanism located at center section of door, with door open. Take care of the actuator rod so as not to change its length.

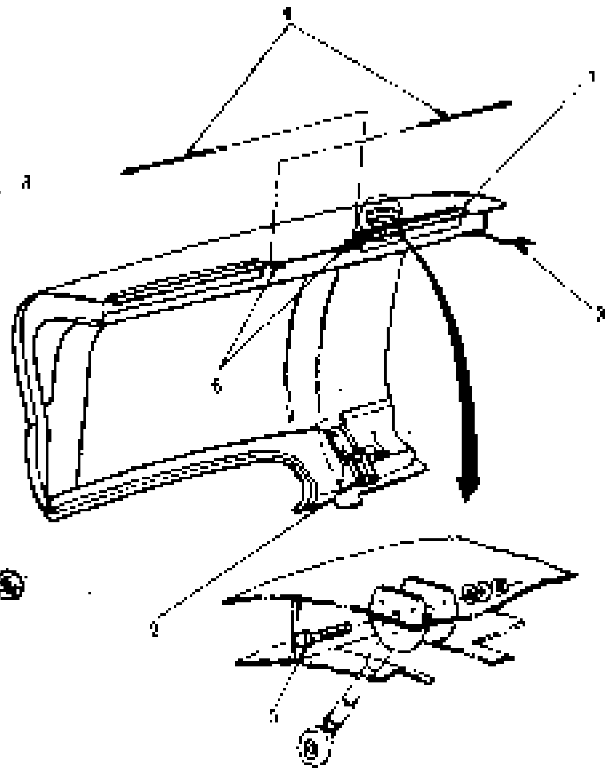
(5) Holding door, remove door hinge connecting bolt.

(6) Install in reverse sequence of removal. Take care of the actuator rod so as not to change its length.



1. Bolt
2. Bolt
3. Bonding jumper
4. Actuator rod

Fig. 3-20 Nose landing gear door



1. Continuous hinge
2. Lock pin
3. Bonding jumper
4. Pin
5. Bolt
6. Stop

Fig. 3-21 Main landing gear forward door

4.7 MAIN LANDING GEAR FORWARD DOOR

4.7.1 CONSTRUCTION AND FUNCTION

The main landing gear forward door is located at the side of bulge between F.STA 4265 and F.STA 5215. Two continuous hinges are provided in the upper section of the door. The link mechanism with independent actuator opens or closes the door before or after gear retraction and extension.

Link mechanism, operated by a motor, locks or unlocks the door in opening or closing.

4.7.2 REMOVAL AND INSTALLATION (See Fig. 3-21)

(1) Open the door and hold it open with rope or chain,etc.

Disconnect electrical system.

(2) Remove nuts connecting door and link mechanism and disconnect bonding jumper.

(3) Remove stop.

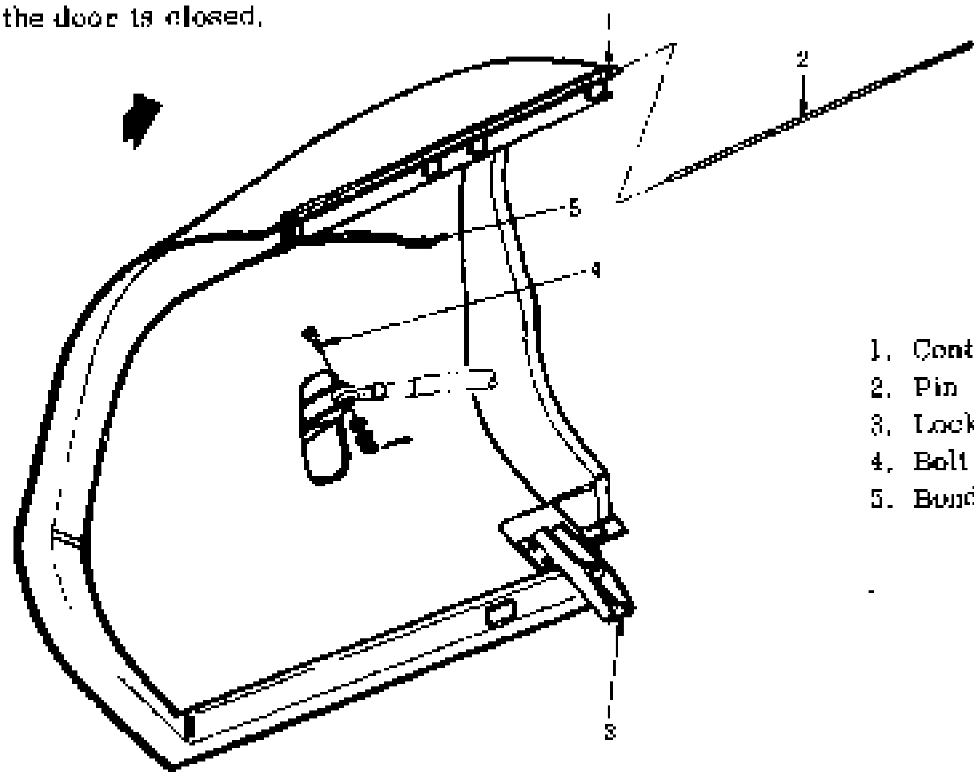
(4) Remove hinge pin in upper section.

(5) Install in reverse sequence of removal.

4.8 MAIN LANDING GEAR AFT DOOR

4.8.1 CONSTRUCTION AND FUNCTION

This door is located at the side of bulge between F.STA 5120 and F.STA 5830, and connected by a continuous hinge provided in upper section of door. The connecting rod of the main landing gear is attached to the door, so the door closes or opens together with main landing gear. The door is locked by a lock pin when the door is closed.



1. Continuous hinge
2. Pin
3. Lockpin
4. Bolt
5. Bonding jumper

Fig. 3-22 Main landing gear aft door

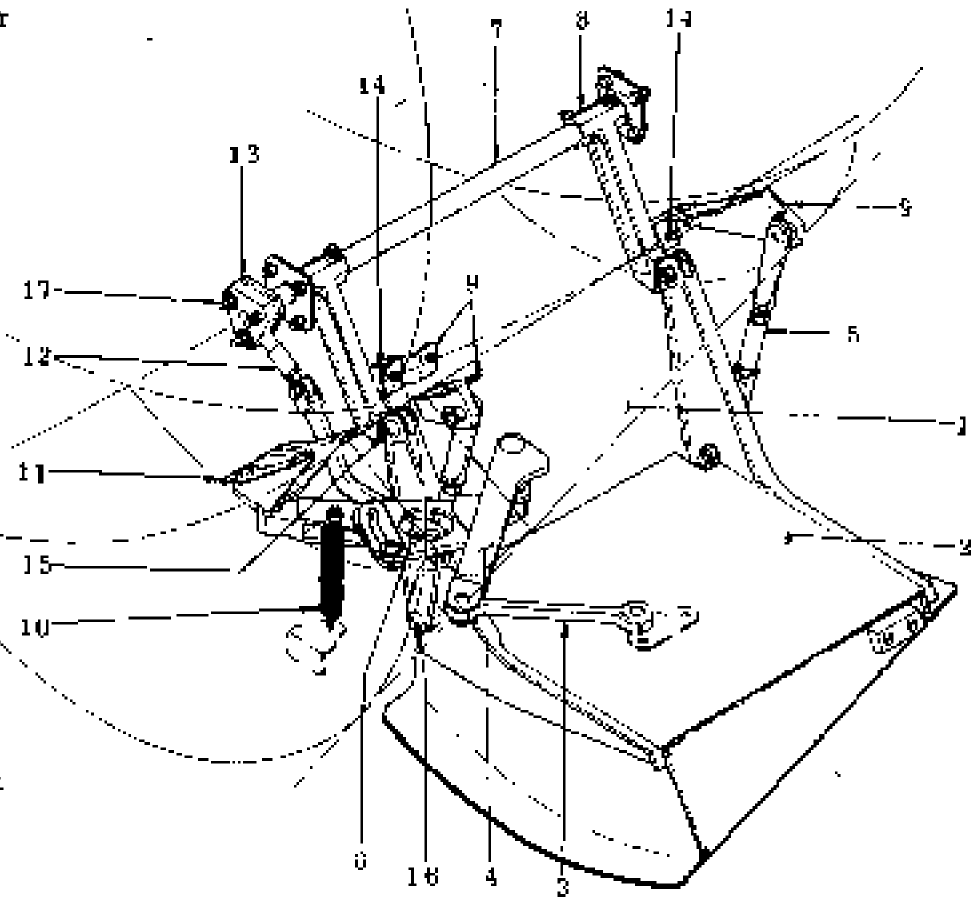
4.8.2 REMOVAL AND INSTALLATION (See Fig. 3-22)

(1) Remove pin connecting front door to connecting rod and disconnect bonding jumper.

(2) Remove cotter pin and hinge pin from upper hinge section and remove the door.

(3) Install in reverse sequence of removal.

1. Entrance door
2. Step body
3. Lever
4. Lever
5. Adjusting rod
6. Lever
7. Torque tube
8. Bracket
9. Bracket
10. Spring
11. Bracket



12. Adjusting rod
13. Shim
14. Adjusting bolt
15. Screw
16. Bolt
17. Bolt

Fig. 3-23 Step assembly and mechanism

4.9 STEP ASSEMBLY

4.9.1 CONSTRUCTION

The step assembly consists of the step body and operating mechanism, and is provided in the space, rear of the bulge, under the entrance door and connected mechanically to the entrance door.
When the entrance door is opened by half (about 77°), the step ② is fully opened, then the door moves individually for further opening (See Fig. 3-23).

4.9.2 REMOVAL AND INSTALLATION

(1) Open the step.

(2) Remove screw ⑬ and bolt ⑭.

(3) To install, tighten screw ⑬ and bolt ⑭.

4.9.3 ADJUSTMENT

(1) To level the step, adjust by the rod ⑤ and bolt ⑯.

(2) In case of mismatch between step and bulge skin when the step is closed, adjust by the rod ⑤ and bracket ⑫.

(3) Installation torque for bolt ⑰ is 75 ~ 100 in-lbs. (86 to 115 kg-cm).

NOTE

i. Mismatch between step and bulge skin shall be less than 0.04 in. (1 mm).

ii. Removed shim ⑱ should not be reused.

4.10 CABIN FLOOR (See Fig. 3-24)

4.10.1 CONSTRUCTION

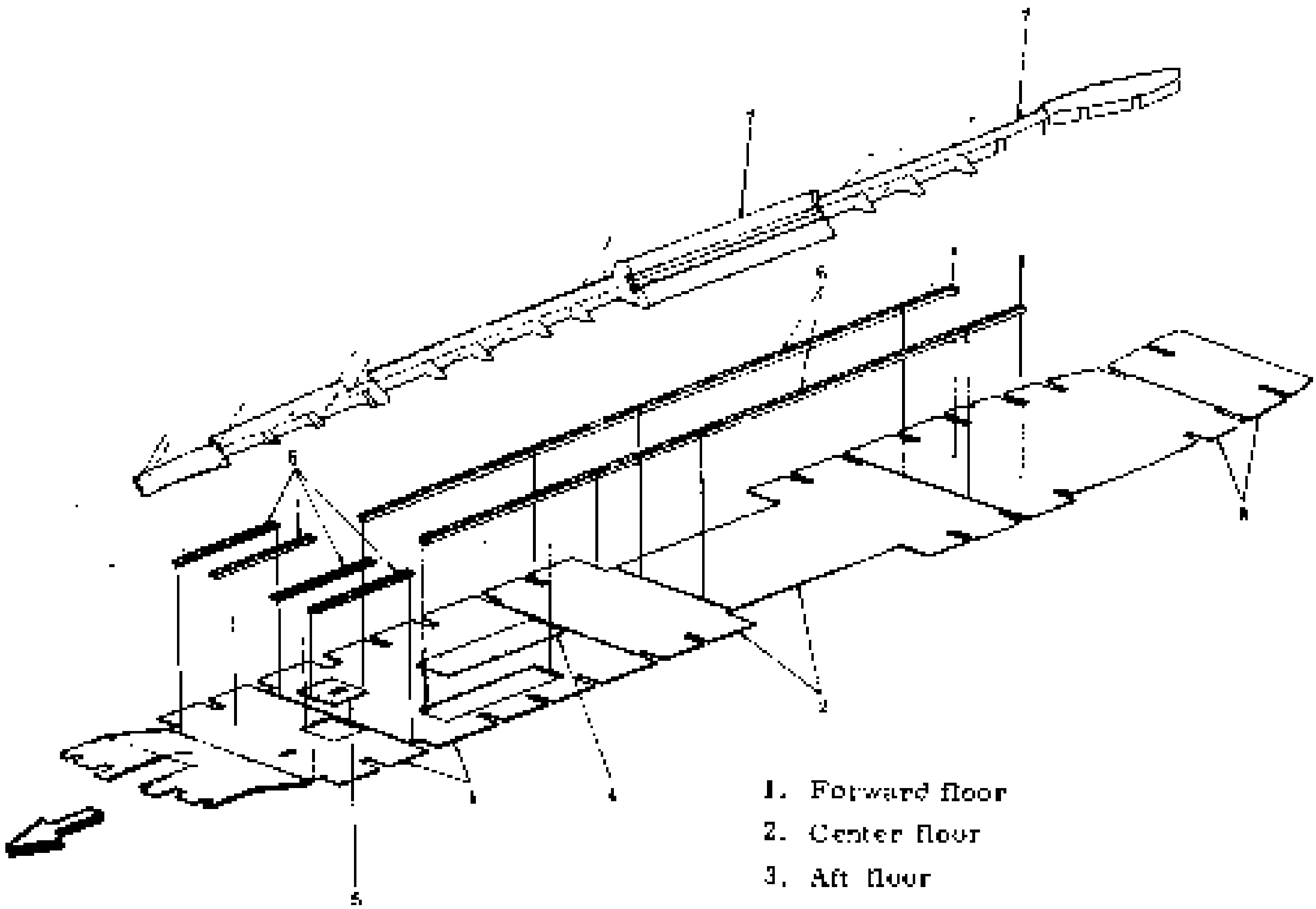
The cabin floors consist of six sandwich panels with a foamed plastic core, and are attached to the floor support members with screws. The two forward floors are detachable but the four rear floors can not be detached, since they constitute a pressure bulkhead.
The rails for pilot seats and passenger seats are screwed on the floors.
Inspection panels for the landing gear operating mechanism and the emergency landing gear extension mechanism are located in the two forward floors.

4.10.2 REMOVAL AND INSTALLATION

(1) Remove screws attaching rails for pilot seats and passenger seats.

(2) Remove screws from the floor panels and floor panel support member, and remove the floors.

(3) Install in reverse sequence of removal.



1. Forward floor
2. Center floor
3. Aft floor
4. Inspection panel-landing gear mechanical stop
5. Inspection panel-emergency landing gear extension mechanism
6. Track
7. Floor side cover

Fig 3-24 Reinstallation of cabin floor

4.11 AFT FUSELAGE

4.11.1 GROUND SUPPORT EQUIPMENT REQUIRED

Sling - aft fuselage (GSE 016A-99046)

4.11.2 REMOVAL AND INSTALLATION

(1) Remove tail cone. (See Fig. 3-25)

(2) Remove the dorsal fin in front of F.STA 5595. (See Fig. 3-26)

(3) Disconnect elevator and rudder trim tab control cables at the turnbuckles in F. STA 8370 and 9625. (See Fig. 3-27 and 28)

(4) Disconnect the lines for electric system and de-icing system in leading edges of vertical and horizontal stabilizer. (See Fig. 3-29 and 30)

(5) Remove rubber hose from adapter of ram air discharge port of the refrigeration unit.

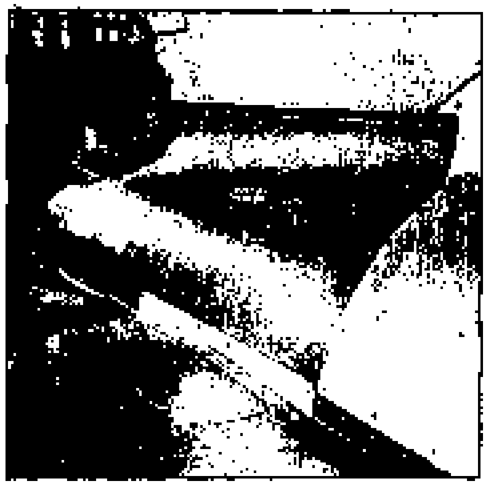


Fig. 3-25 Removal of tail cone

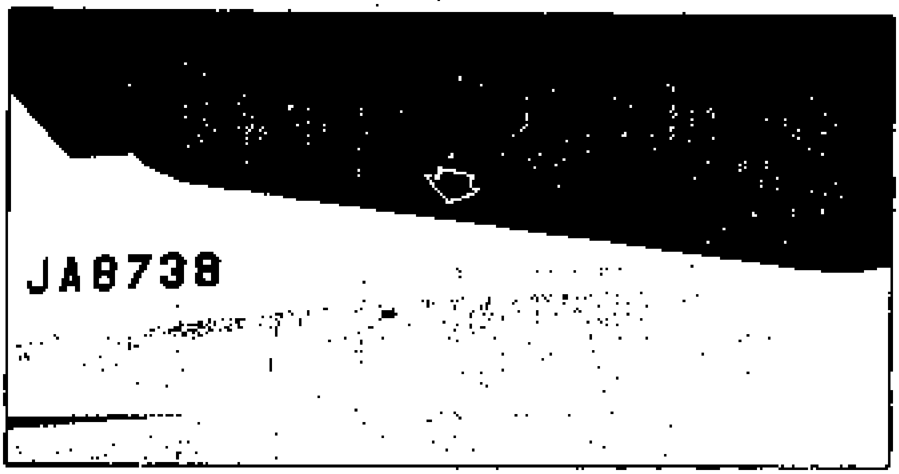


Fig. 3-26 Removal of dorsal fin

Fig. 3-27 Removal of elevator trim tab control cable

Fig. 3-28 Removal of rudder trim tab control cable

(6) Remove screws for hoisting from the front spar of vertical stabilizer to V.STA 1500. Attach sling to fuselage and lift up so that no load is applied to fuselage.

(7) Remove the four outboard access covers and the four fuselage connecting bolts. Move the aft fuselage rearward to the direction of the airplane axis to disconnect it. (See Fig. 3-31)

(8) Install in reverse sequence of removal. Installation torque for rear fuselage connecting bolts is 20 ~ 25 in-lbs. (23 ~ 29 kg-cm). Install lockwire.

CAUTION

Since tail cone and dorsal fine are made of FRP, do not damage or let them fall.

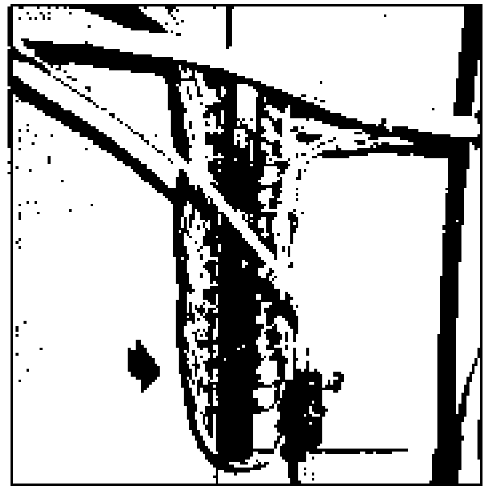


Fig. 3-29 Removal of electric connection

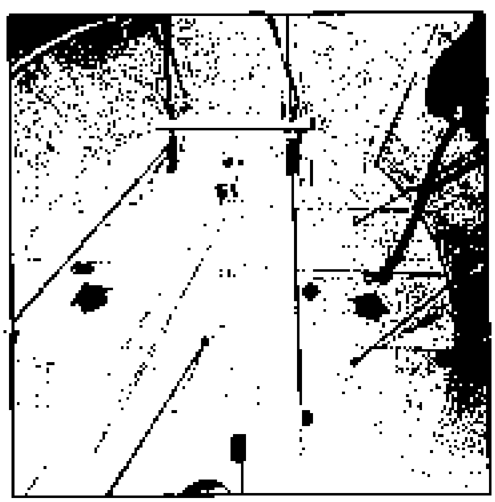


Fig. 3-30 Disconnect line for de-icing system

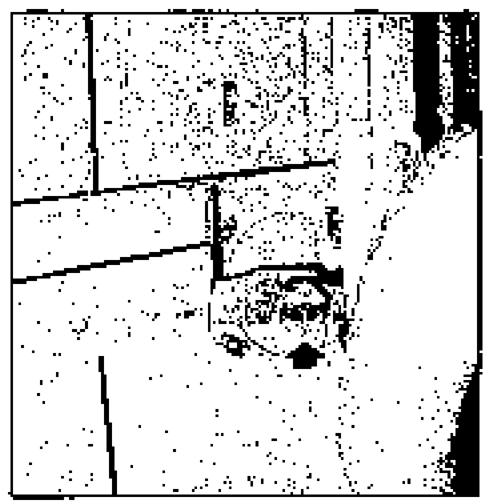


Fig. 3-31 Remove aft fuselage connecting bolts. Installation torque value 20 ~ 25 in-lbs.

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5. WINDSHIELD AND CABIN WINDOW

5.1 GENERAL DESCRIPTION

Applicable to Aircraft S/N 652SA

Windshield consists of two panes of front and side glass. Each pane is contoured and formed from one piece of acrylic plastic. Front outer pane is 0.312 in. (7.925 mm) and outer pane 0.25 in. (6.35 mm) thick. The parts of front pane and side outer pane which are attached to airframe are reinforced with glass fabric (FRP). Screw holes are provided on the reinforced part to install. Front and side inner panes are installed to airframe through very soft sponge rubber.

Applicable to Aircraft S/N 661SA, 697SA and subsequent

The windshield consists of two panes, inner and outer. The inner panes are contoured and formed from one piece acrylic plastic 0.187 in. (4.75 mm) thick. The outer pane consists of two, an inner and outer tempered glass ply with a vinyl sandwiched between them. The outer pane glass plies are designed to bear the cabin internal pressure and flight loads. The outer pane windshield is of a fail safe construction and vinyl ply. The outer pane vinyl ply contains a heating element for flight during icing conditions. Refer to Section IX "Anti-Atmospheric System" for details, operation and maintenance.

Each side window consists of two panes, inner and outer. Each pane is contoured and formed from one piece acrylic plastic; inner pane thickness is 0.187 in. (4.75 mm), whereas the outer pane thickness is 0.25 in. (6.35 mm). Each pane is constructed with a fiber reinforced glass and is installed to the airframe in this area. The inner panes are installed through a soft sponge rubber.

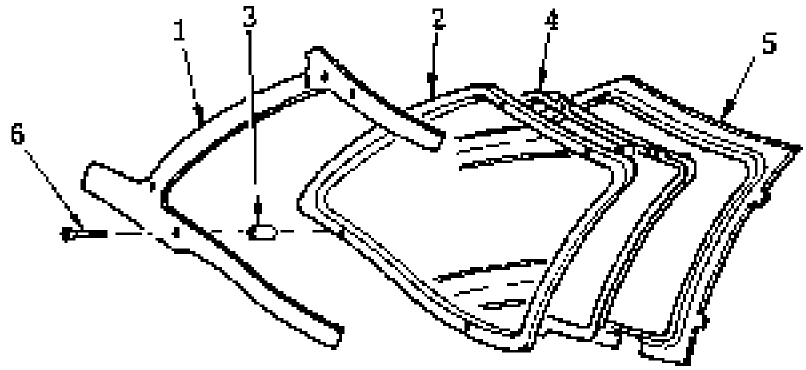
5.2 REMOVAL AND INSTALLATION OF WINDSHIELD

CAUTION

Since the aircraft is pressurized, handle front and outer side panes with care. Special attention should be paid not to damage with knife, etc., when installing or removal of glass or masking work in painting.

5.2.1 REMOVAL OF FRONT PANE (See Fig 3-32)

(1) Remove shroud and instrument panel (See Chapter X, Para. 1.4).
(2) Remove screws and inner glass assembly (S/N 652SA only).
(3) Remove screws and upper inside panel.
(4) Disconnect wires at terminal block and identify wires (except S/N 652SA).
(5) Remove screws and glass with retainer outward.



1. Retainer
2. Side pane
3. Rubber grommet
4. Trim frame
5. Inner glass assembly (S/N 652SA only)
6. Screw

Fig. 3-32 Removal of front pane

5.2.2 REMOVAL OF SIDE PANE (See Fig 3 33)

(1) Remove shroud and instrument panel.
(2) Remove screws and inner glass assembly.
(3) Remove screws and glass in frame, together with retainer outward.

NOTE

Note the length of each bolt.

5.2.3 INSTALLATION OF WINDSHIELD AND SIDE WINDOW PANE

(1) Place the outer pane on windshield frame, and position the rubber grommet in each corner.
(2) Apply self-contouring seal (Sealant PR1222 or PR1425B) over the outer pane.
(3) Place the retainer over the outer pane.
(4) Apply pressure-tight seal to the screw head and then install it.
(5) Tighten the outer pane and trim frame together with screws. Tightening torque is 25～30 in-lbs (29～35 Kg-cm).

NOTE

Tighten screws on the corners first. Then diagonal torquing method is recommended to obtain proper fitness of the windshield.

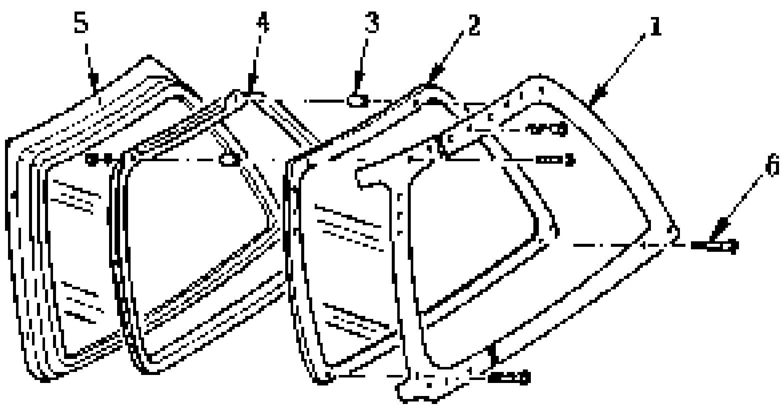
(6) Apply water-tight seal to the outer pane.
(7) Remove wire identification or tags and connect to terminal block. (not applicable to S/N 652SA).
(8) Tighten screws on front and side inner panes.
(9) Tighten screws on upper inside panel.
(10) Install shroud and instrument panel.

NOTE

Wind shield and window glass should be installed after cleaning the surface between outer and inner panes.

NOTE

Note the length of each bolt.

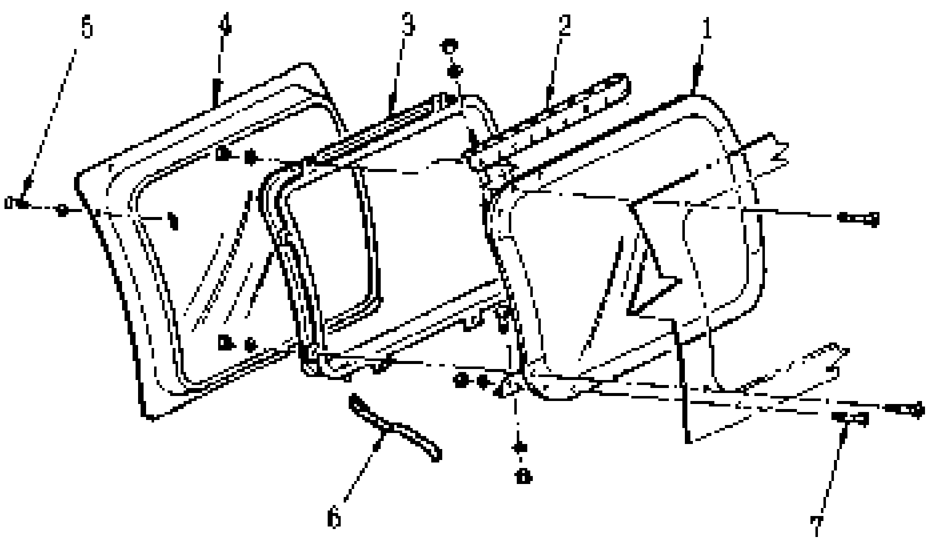


1. Retainer
2. Side pane
3. Rubber grommet
4. Trim frame
5. Inner glass assembly
6. Screw

Fig.3-33 Removal of side pane

5.3.1 REMOVAL

(1) Remove inner glass assembly.
(2) Remove rubber duct for defog.
(3) Remove screws and outer pane with frame and angle inward.



1. Outer pane
2. Glass support angle
3. Trim frame
4. Inner glass assembly
5. Screw
6. Rubber duct
7. Screw

Fig 3-34 Removal of cabin window

5.3.2 INSTALLATION OF CABIN WINDOW

(1) Apply self-contouring seal (sealant PR1222 or PR1425B) over the outer pane.
(2) Place the outer pane on the window frame, and position the glass support angle.
(3) Apply pressure-tight seal to the screw head and then install it.
(4) Tighten the window frame, the outer pane, trim frame and glass support angle together with screws.
Tightening torque is 25～30 in-lbs. (29～35 Kg-cm)

NOTE

Tighten screws on the corners first. Then diagonal torquing method is reccommended to obtain proper fitness of the cabin window.

(5) Apply water-tight seal to the outer pane.
(6) Install the inner glass assembly.

NOTE

Window glass should be installed after cleaning the surface between outer and inner panes.

NOTE

Note the length of each bolt.

5.4 WINDSHIELD AND CABIN WINDOW GLASS

5.4.1 INSPECTION

Inspection criteria for acrylic plastic glass is as follows:

(1) Replace with new glass if following conditions are found.

(a) Total crazed area exceeds one square inch.
(b) Depth of crazing exceeds 5% of panel thickness.
(c) Visibility is impaired.

(2) Replace when scratch exceeds 0.01 in. (0.25 mm) in depth.

NOTE

When scratch is less than the value specified above, glass should be reused after smoothing the scratch.

5.4.2 CLEANING

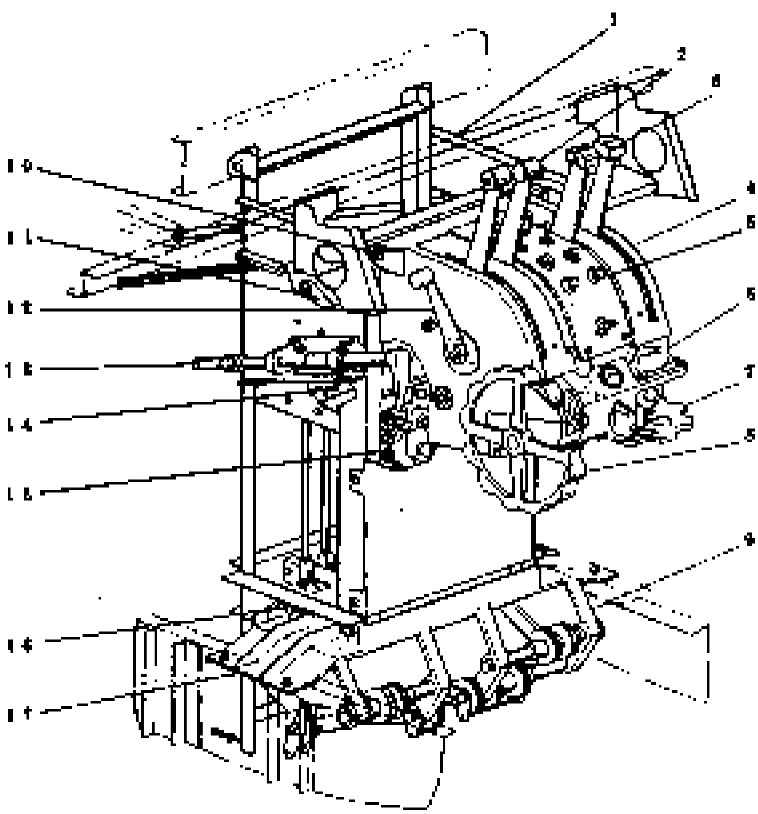
See Para. 6.8.4 Chapter II, CLEANING INSTRUCTIONS

6. CENTER PEDESTAL

6.1 GENERAL DESCRIPTION

The center pedestal is installed in forward center section of the cockpit at F.STA 1650 ~ F.STA 1820 and contains the mechanism to control the engines, elevator and rudder trim tab and parking brakes.
The center pedestal consists of the followings. (See Fig. 3-35)

(1) Engine control box
(2) Trim control box
(3) Bell crank assembly
(4) Pulley assembly
(5) Parking brake
(6) Drum assembly
(7) Switch panel
(8) Frame assembly



1. Frame assembly
2. Engine power lever
3. Engine condition lever
4. Engine control box
5. Engine switch panel
6. Trim control box
7. Rudder trim wheel
8. Elevator trim wheel
9. Pulley assembly
10. Bolt-engine control box
11. Upper bell crank
12. Friction lever
13. Parking brake lever
14. Lower bell crank
15. Drum assembly
16. Bolt-frame assembly
17. Support-keel

Fig. 3-35 Center pedestal assembly

6.2 REMOVAL AND INSTALLATION

6.2.1 ENGINE CONTROL BOX

(1) Remove four rods which are connected to bell crank, from power lever and condition lever. (See Fig. 3-36)

(2) Remove wiring of switch panel from connector J208 of main junction box. (See Fig. 3-37)

(3) Remove four bolts for box attachment.

(4) Install in reverse sequence of removal.

6.2.2 TRIM CONTROL BOX (See Fig. 3-38)

(1) Remove six screws for box attachment.

(2) Install in reverse sequence of removal.

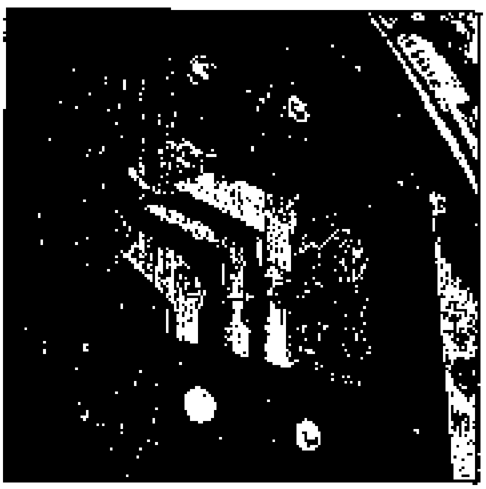


Fig. 3-36 Removal of rod from bell crank

Fig. 3-37 Disconnect switch panel wiring

6.2.3 BELL CRANK ASSEMBLY

(1) Remove attaching bolts (8 for upper bell crank and 2 for lower bell crank) for rods from bell crank.

(2) Remove bearing attaching bolts (4 for upper and 4 for lower) from frame assembly.

(3) Install in reverse sequence of removal.

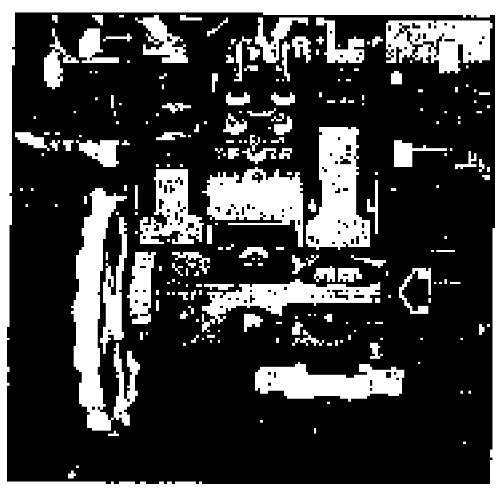


Fig. 3-38 Removal of screws

6.2.4 PULLEY ASSEMBLY

(1) Remove four bolts of control rods from bell crank.

(2) Remove four L.H. and R.H. bolts attaching frame assembly.

(3) Remove two attaching bolts at rear side from frame assembly.

(4) Remove four bolts attaching keel.

(5) Install in reverse sequence of removal.

NOTE

First, remove control cable and chain for engine and control system.

6.2.5 PARKING BRAKE (See Fig. 3-39)

(1) Disconnect parking brake lever from rod.

(2) Remove four screws and brake lever assembly.

(3) Install in reverse sequence of removal.

6.2.6 DRUM ASSEMBLY (See Fig. 3-40)

(1) Remove drum attaching bolt, and drum assembly.

(2) Install in reverse sequence of removal.

Fig. 3-39 Disconnect parking brake lever from rod

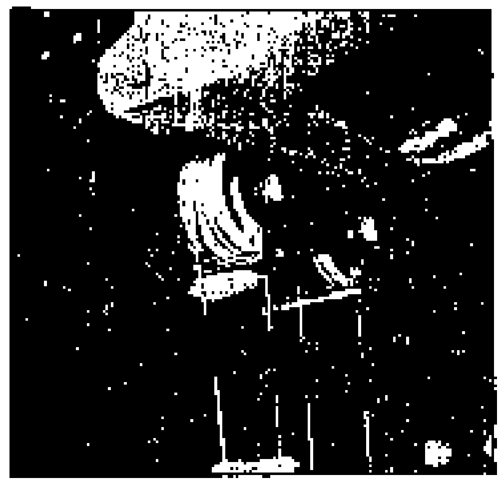


Fig. 3-40 Removal of drum assembly

7. STATIC DISCHARGER

Static electricity causes disturbances in aircraft radio equipment, etc. The discharger is installed to discharge static electricity collected in the airframe, so that airframe electric potential is almost the same as that of the atmosphere around the aircraft.
The sharper the tip of the discharger, the less equipment noise and the voltage of the discharging section drops.
The discharger controls rapid discharging and prevents noise.
The dischargers are installed on wing, horizontal stabilizer and tip tank. Care should be taken so that oil or dust does not flow into the discharging section with rain and that the discharger is not damaged.

CHAPTER

4

LANDING GEAR AND BRAKES

CHAPTER IV

LANDING GEAR AND BRAKES

CONTENTS

1. GENERAL

1.1 GENERAL DESCRIPTION

The landing gear is a retractable tricycle type and consists of two main gears and steerable nose gear. Landing gear retracting and landing gear door operating mechanisms are electric-mechanical type.

1.1.1 LANDING GEAR RETRACTING MECHANISM

Main and nose gear are connected through torque shafts and are retracted or extended at the same time by operation of the landing gear motor. In this system, mechanical stop and clutch are provided and prevent airframe and retracting mechanism from damage in case of limit switch failure. Main gear actuator is also used as drag struts and up-lock or down-lock because of internal screw jack being unreversible. Nose landing gear has down lock mechanism mechanically connected with nose gear actuator but no up lock mechanism is necessary because of the actuator being unreversible.

1.1.2 LANDING GEAR DOOR OPERATING MECHANISM

(1) Nose gear door operating mechanism
This mechanism is operaced together with nose gear strut and closes the doors in gear up operation, opens the doors during gear down operation.

(2) Main gear forward door operating mechanism
This mechanism is operated by main gear forward door actuator which is operated electrically together with landing gear retracting mechanism. If ground door open switch is operated on the ground, the main gear forward door can be opened or closed.

(3) Main gear aft door operating mechanism
Main gear aft door is connected mechanically with main gear oleo strut and operated together with the strut.

1.1.3 MAIN GEAR DOOR LOCK MECHANISM

This mechanism is connected mechanically with main gear forward door operating mechanism and locks main gear forward and aft doors respectively when the doors are fully closed.

1.1.4 EMERGENCY GEAR EXTENSION MECHANISM

When electric system fails, this system is to perform gear extension manually by operating emergency gear extension handle on the floor near the pilot seat.

1.1.5 NOSE GEAR STEERING MECHANISM

This system connects nose gear and rudder pedals mechanically when gear is extended for ground steering. Nose gear and rudder pedals are disconnected in gear up position allowing only rudder control.

1.1.6 BRAKE SYSTEM

This system can be operated by either the pilot or co-pilot rudder pedals, the right brake by the right pedal and the left brake by the left pedal. By applying toe pressure on rudder pedals and pulling the parking handle, brake pressure can be maintained as parking brake.

1.2 LANDING GEAR OPERATION

1.2.1 LANDING GEAR RETRACTION

When landing gear control switch is in the UP position, gear retraction is performed in the following sequence.

(1) Electric power is supplied to main gear forward door actuator, unlocking the forward doors and opening them (Red lamp indicating "unsafe" comes on).
(2) When main gear forward doors are fully opened, door open limit switch actuates and stops main gear forward door actuator. Simultaneously the landing gear motor begins to retract the gear (Green lamp indicating "down lock" goes out). The rotation of gear motor is transmitted to LH and RH gearbox and drag struts supports, LH and RH screw jacks in main gear drag strut through main reduction gearbox and torque limiter to retract LH and RH main wheels at the same time. Furthermore, this rotation is transmitted by torque shaft to nose gear actuator through mechanical stop to retract nose gear together with LH and RH main wheels.
(3) When the landing gear comes to the appointed position, up limit switch actuates and stops gear motor, so landing gear stops at up position. Nose gear door and main aft doors are operated and closed mechanically together with landing gear.
(4) Just before the landing gear is fully retracted, up zone switch actuates and operates main gear forward door actuator to close the door and stops after closing. When main gear forward door is closed, the door hook is locked together with main gear aft door (Red lamp goes out).

1.2.2 LANDING GEAR EXTENSION

When landing gear control switch is placed to DOWN position, gear extension is performed in the following sequence.

(1) Main gear forward door actuator actuates, unlocks and opens the forward door (Red lamp comes on). At the same time, main gear aft door lock, which is connected mechanically with main gear aft door operating mechanism, is unlocked.
(2) When the main gear forward door is fully opened, door open limit switch actuates and stops main gear door actuator. At the same time, landing motor begins to extend the gear. When the NLG reaches the down position, the NLG continues downward until the nose gear down lock limit switch is actuated, which stops the gear motor, then the Green lamps are illuminated.
(3) When the landing gear is fully extended, nose gear down lock limit switch actuates and operates main gear forward door actuator to close and lock the door and stops (Red lamp goes out).

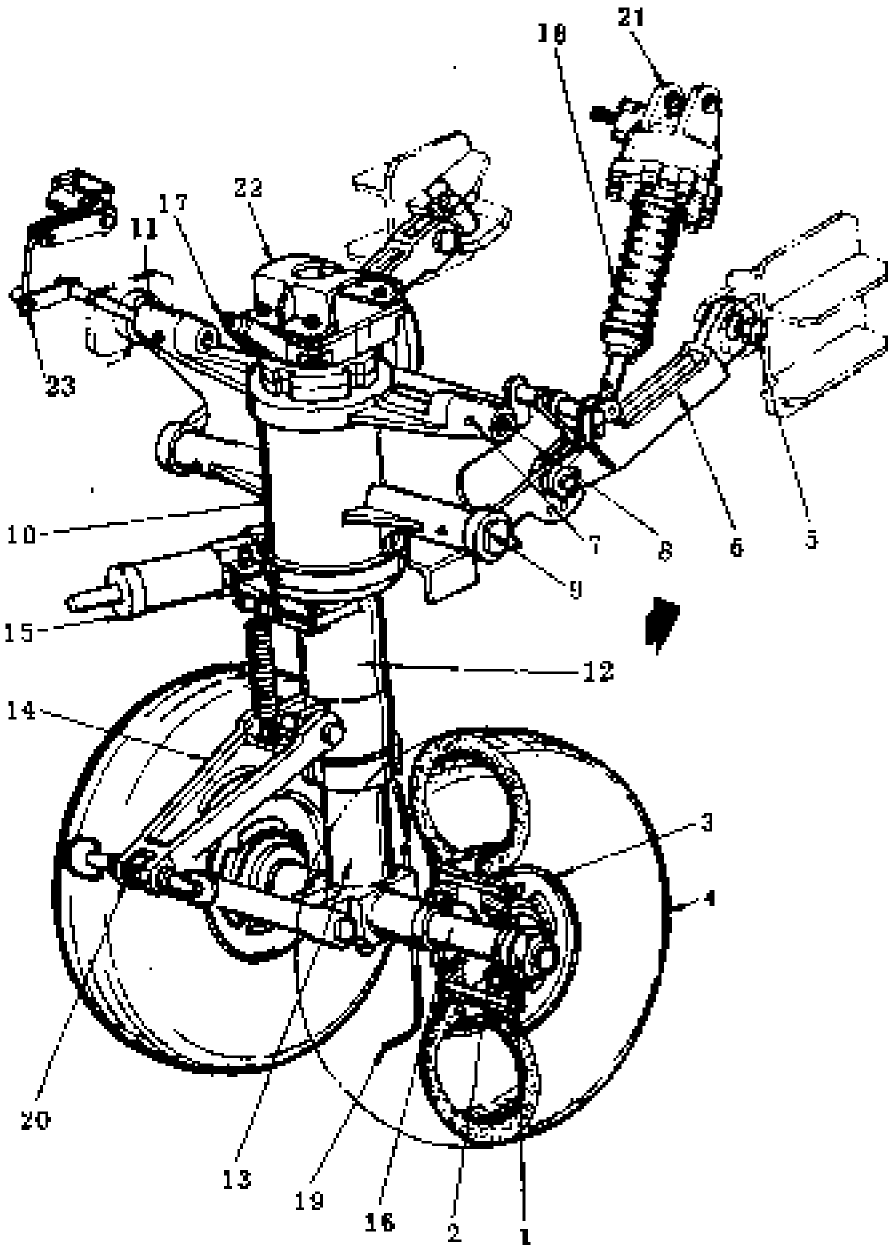
2. NOSE LANDING GEAR

2.1 GENERAL DESCRIPTION (See Fig. 4-1)

The nose landing gear consists of strut, drag strut, wheels, tires and tubes. The strut is composed of trunnion, cylinder, piston, torque link connecting cylinder and piston, shimmy damper, axle, steering roller and straightener. The duty of this strut is to absorb landing shock loads. At the top end of the strut is the steering roller, which turns the wheels with the cam of the steering mechanism at the gear extension and is disconnected automatically from the cam at the gear retraction. Each of the drag struts, one at each side, is provided with a down lock. The screw jack for gear retraction is connected to the drag strut on the right-hand side. The shimmy damper connects the trunnion with the cylinder to prevent the nose gear from shimmying during taxiing. The nose gear is equipped with the nose gear door actuating mechanism mechanically linked with the strut. At the lower end of the strut, a static ground wire is located to discharge static electricity of the aircraft. A disconnecting mechanism is located between torque links to provide a greater steering angle in towing.

2.2 REMOVAL AND INSTALLATION OF NOSE LANDING GEAR

(1) Jack up aircraft.
(2) Remove wheels and tires from Axle. (See Para. 2.3.3)
(3) Retract the nose gear slightly from the down lock position. Where lock is displaced from lock groove of nose drag strut, remove bolts connecting screw jack and drag strut.
(4) Remove attaching bolt at the forward end of drag strut.
(5) Fold drag strut upward. Draw drag strut out of forward end fitting.
(6) Remove bolts from trunnion, pull out trunnion pin while holding strut, and then remove strut from aircraft. Note number of shims between trunnion and fitting to facilitate reinstallation.
(7) Remove bolts from trunnion, pull out drag strut attaching pin, and then remove drag strut. Remove nose gear from aircraft.
(8) Reinstall in reverse sequence of removal. Inspection items to be covered at installation are as follows: (See Fig. 4-2)
   (a) Place nose strut so that clearance in lateral direction between strut and bearing of airplane may be 0.002～0.016 in. (0.05～0.4 mm). Add shims as necessary.
   (b) Place nose strut so that clearance between LH and RH drag struts and drag strut pin may be 0.002 to 0.012 in. (0.05 to 0.3 mm).
   (c) After reinstallation move nose strut by hand back and forth to ensure smooth operation with no interference.
   (d) Reinstall trunnion pin so that it may be 0.728 in. (18.5 mm) long from trunnion end to the end of the pin.
   (e) For reinstalling nose gear screw jack, see Para 4.2.1.
   (f) For reinstalling wheel on axle, see Para 2.3.3.



1. Nut
2. Washer
3. Wheel
4. Tire and tube
5. Bolt
6. Drag strut
7. Trunnion bolt
8. Trunnion pin
9. Drag strut pin
10. Nose gear strut
11. Trunnion
12. Cylinder
13. Piston
14. Torque link
15. Shimmy damper
16. Axle
17. Steering roller
18. Screw jack
19. Static ground wire
20. Torque link disconnecting mechanism
21. Nose gear actuator
22. Straightener
23. Pendulum

Fig. 4 - 1 Nose landing gear

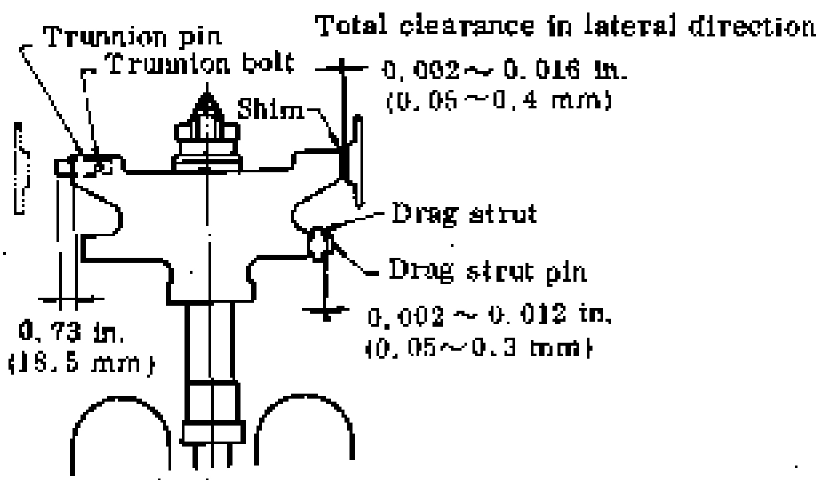


Fig. 4-2 Installation of nose landing gear

2.3 NOSE WHEEL

Nose wheel assembly is made of magnesium alloy casting to fit in 5.00 x 5 type III tire and tube. The wheel can be split into halves for installation of the tire. The wheel halves are held together by 3 tie bolts with self-locking nuts. The wheel has 2 tapered roller bearings for rotation. The bearing rotates in the bearing cup which is shrink fitted into hub of each wheel half and is protected by felt or rubber seals held by a metal retainer ring from foreign material or leakage of lubricating grease. The wheel requires an inner tube type tire. For disassembly, reassembly and check of nose wheel, see Para. 8.3.

2.3.1 PREPARATION OF AIRCRAFT

Jack nose gear or entire aircraft.

2.3.2 EQUIPMENT REQUIRED

Wrench for removing wheel GSE 016A-99018 or equivalent

2.3.3 Removal and reinstallation are made in accordance with the following procedures (See Fig 4-4).

(1) Remove wheel from axle by removing lock wire, screws, nuts and spacer.
(2) Reinstall in reverse sequence of removal. Tighten screws (AN500A6-6) finger tight. While rotating the wheel, tighten the nut to 240 in-lbs (276 kg-cm) until the wheel turns with noticeably greater resistance. Loosen the nut, retighten to 120 in-lbs (138 kg-cm) and install lockwire. The wheel must rotate more than a quater revolution by inertia when it is accelerated by hand without any side swing or free play.

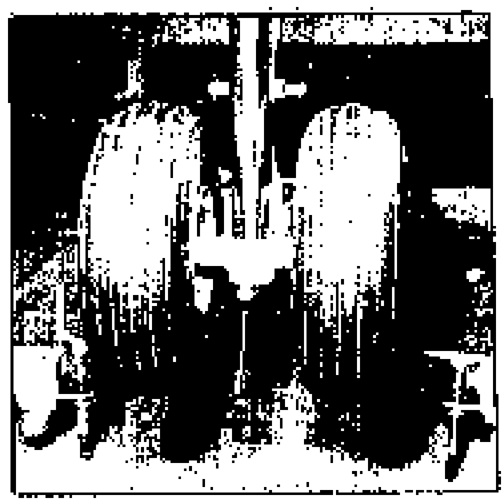


Fig. 4-3 Static ground wire

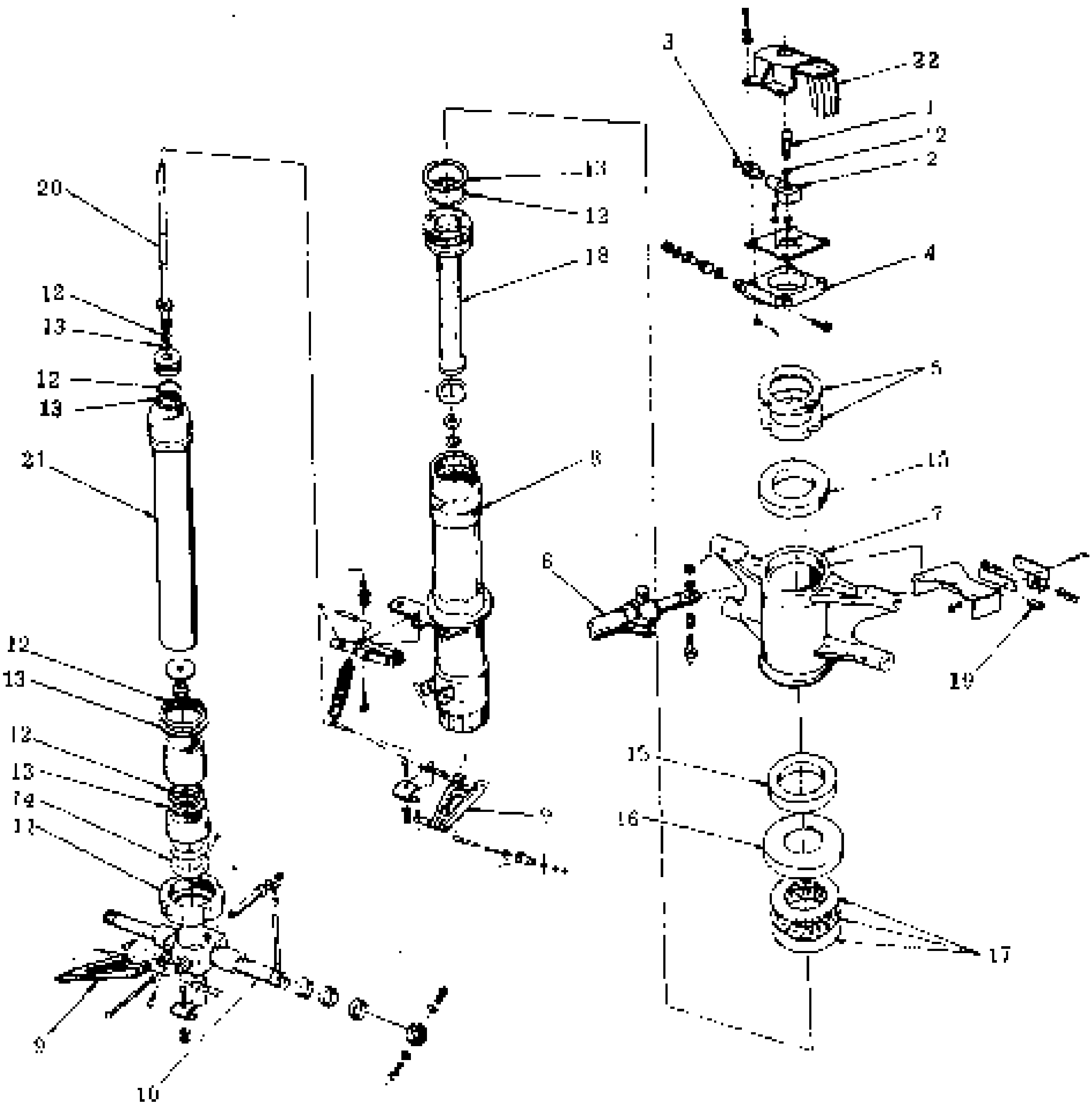
2.4 STATIC GROUND WIRE

2.4.1 REMOVAL AND INSTALLATION (See Fig. 4-3)

The static ground wire, located at the lower end of nose gear strut, is for the discharge of accumulated static electricity in the aircraft.

When the ground wire becomes worn, reset the wire to proper length, as required to give positive contact with the runway.

2.5 NOSE LANDING GEAR STRUT (See Fig. 4-4)



1. Bolt
2. Body
3. Air valve
4. Centering roller support
5. Nut
6. Shimmy damper
7. Trunnion
8. Cylinder
9. Torque link
10. Axle
11. Nut
12. "O" ring
13. Back up ring
14. Scraper
15. Radial bearing
16. Bearing cover
17. Thrust bearing
18. Support
19. Up-lock roller
20. Metering pin
21. Piston
22. Straightener

Fig. 4-4 Nose landing gear strut

2.5.1 DISASSEMBLY AND REASSEMBLY

See Overhaul Manual (YET69148)

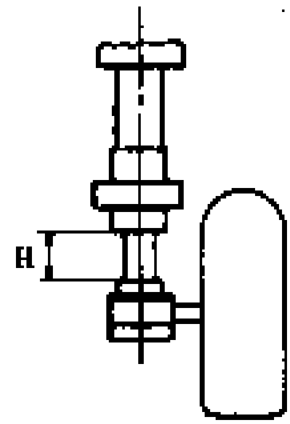
2.5.2 MAINTENANCE

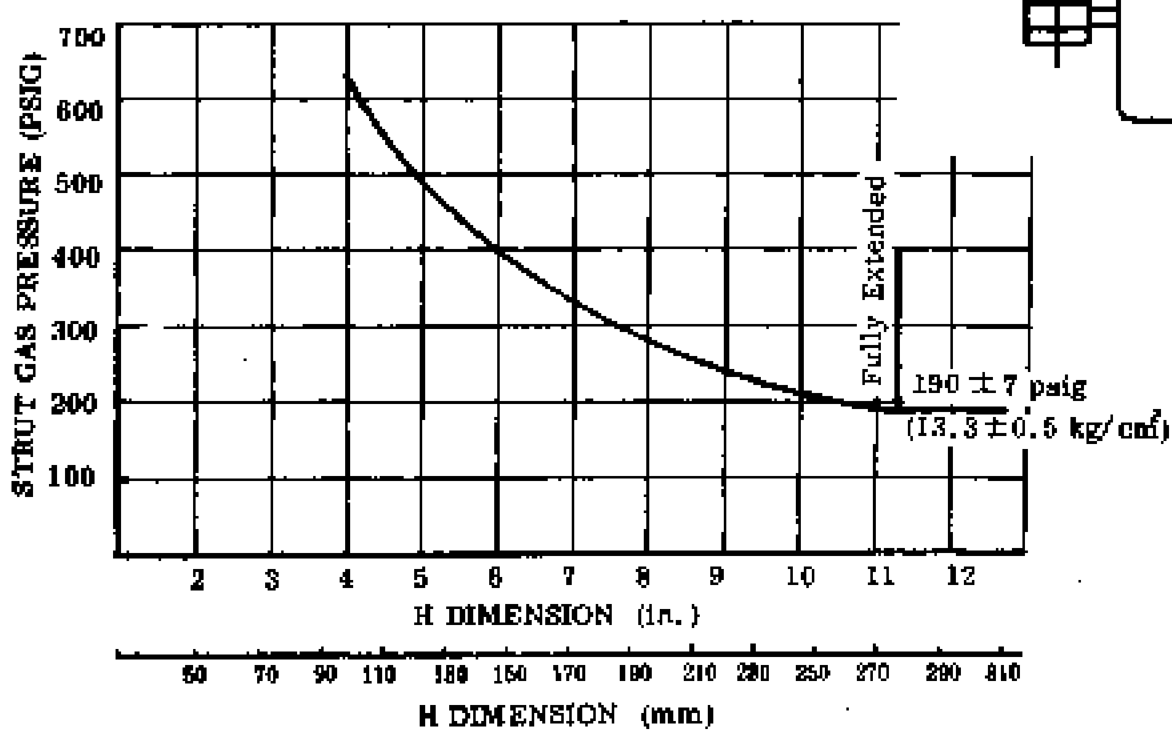
(1) Keep exposed strut piston clean by wiping with soft cloth moistened with hydraulic fluid MIL-H-5606.

(2) Inflate strut with N_2 gas to 190 ± 7 psig(13.3 ± 0.5 kg/cm²) with strut fully extended or until H dimension and pressure conform with the following chart within ± 15 psi. (1 kg/cm²).

CAUTION

When the temperature varies greatly from the temperature at the time of last inflation, pressure and dimension "H" will not conform with the chart. Reinflate strut to conform with the chart.





maintenance manual

TEMPORARY REVISION NO.4-1

This Temporary Revision No. 4-1 is applicable to the following Maintenance Manuals :

MODEL NO	REPORT NO	PAGE
MU-2B-35/-36A	MR-0218	4-9
MU-2B-60	MR-0336	4-9

Insert facing the page indicated above for the applicable Maintenance Manual

Retain this Temporary Revision until such time as a permanent revision on this subject is issued

REASON : To add a non destructive inspection for NLG trunnion.

ADD : Paragraph 2.5.3 as follows.

2.5.3 Non Destructive Inspection for NLG Trunnion

NOTE

This Inspection is required for Trunnion until replaced with improved parts P/N 010A-39190-19.

(1) Preparation

(a) Place aircraft on jacks.

(b) Remove the nose gear wheels.

(c) Remove the NLG strut assembly and the following parts from the trunnion.

- Shimmy Damper
- Trunnion Arm
- Spring holder and Uplock Roller

NOTE

Record lengths or mark adjustable parts, such as Uplock Roller, so they can be reinstalled in the same position to facilitate adjustment and rigging procedures.

(2) Inspection

Inspect by Magnetic Flux Inspection or Fluorescent Penetrant Inspection method as follows.

(a) Remove paint and clean the entire surface from effected area of the part to be inspected with Methyl Ethyl Ketone or equivalent. (Refer to Fig. 4-6A for inspection area.)

NOTE

It may be necessary to remove trunnion for inspection

CAUTION

Do not allow Methyl Ethyl Ketone to contact piston seal area.

(b) Inspect the specific areas on the trunnion to insure there are no indications of cracks using with appropriate inspection procedure

NOTE

Magnetic flux inspection must be made in accordance with ASTM E 1444. Fluorescent penetrant inspection must be made in accordance with ASTM E 1417 Type I, Method C.

PART NAME	PART NUMBER	RESULTS OF INSPECTION	
		IF NO CRACK IS FOUND	IF CRACK IS FOUND
TRUNNION	010A-39190-11/-13	PART MAY BE REUSED BUT MUST BE INSPECTED AT 200 HR INTERVALS.	SCRAP, REPLACE WITH NEW PART IN ACCORDANCE WITH PARAGRAPH 2.5.3.(3).

NOTE

If cracks are found in the trunnion, notify Field Service Operations Manager of Mitsubishi Heavy Industries America, Inc., of the following items:

a. Aircraft Serial Number
b. NLG Strut Assembly Serial Number
c. Clarify NLG Strut Assembly Hours In Use
d. Part Number of Trunnion

(c) If no cracks are found and part is to be reused, apply one coat of zinc chromate primer and one coat of aluminized lacquer.

(d) Repetitive inspection must be conducted at intervals not to exceed 200 flight hours in accordance with para. (2).

(3) Replacement of Defective Part

If cracks are found, replace the trunnion with 010A-39190-19 trunnion.

NOTE

1. When the trunnion is replaced, new trunnion pins and drag strut pins should be used.
2. If the trunnion is replaced with P/N 010A-39190-19, bearing cover should be replaced with new type P/N 010A-39191-11 simultaneously
3. The 010A-39190-19 trunnion does not require further repetitive inspections.

(4) Reassembly

Reassemble NLG strut in the reverse order of removal procedures (Rec 2.5.3(1)(c)).

maintenance manual

TEMPORARY REVISION NO.4-1

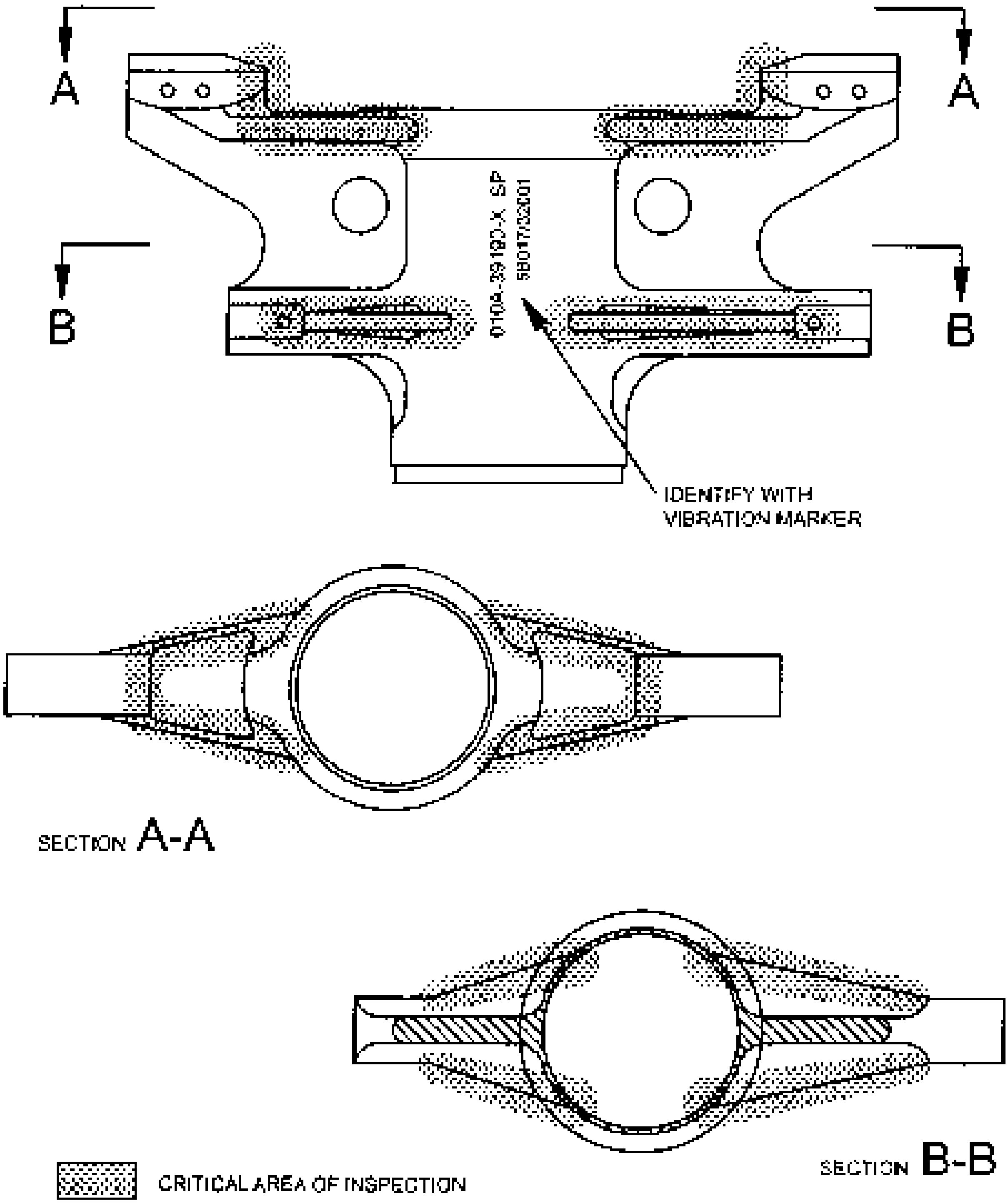


Fig. 4-6A 010A-39190-11/-13 Trunnion inspection area

(3) To check for fluid quantity in oleo strut or refill with fluid, perform as follows: (See Fig. 4-4 and 4-5)

(a) Remove oleo strut from aircraft.

(b) Deflate strut through air valve, and remove valve, body and bolts.

NOTE

Do not loosen air valve body unless gas is completely bled through air valve.

(c) Set strut in the most compressed position and hold vertically. Fill strut with hydraulic fluid (MIL-H-5606) up to the filler port.

(d) Extend strut to full length and compress fully again and fill with fluid up to the lower end of the filler port.

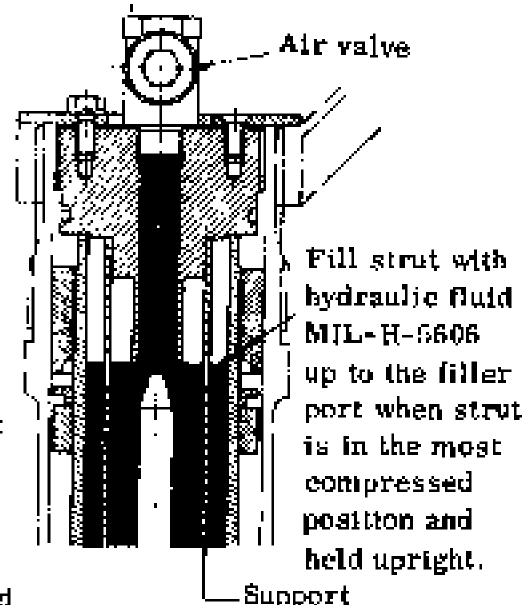


Fig. 4-5 Oleo strut

(e) Repeat these steps two or three times.

(f) Install valve body with air valve and bolts into the oleo strut, then put N_2 gas through the valve up to a pressure of 190 psi (13.3 kg-cm^2).

(g) Leave the strut for one hour, and check for gas and fluid leakage.

2.6 NOSE LANDING GEAR STEERING MECHANISM

The nose gear steering mechanism is linked to the rudder pedals.
When the nose gear is in the UP position, the rudder pedals are disengaged from the steering mechanism, and only the rudder is controlled.

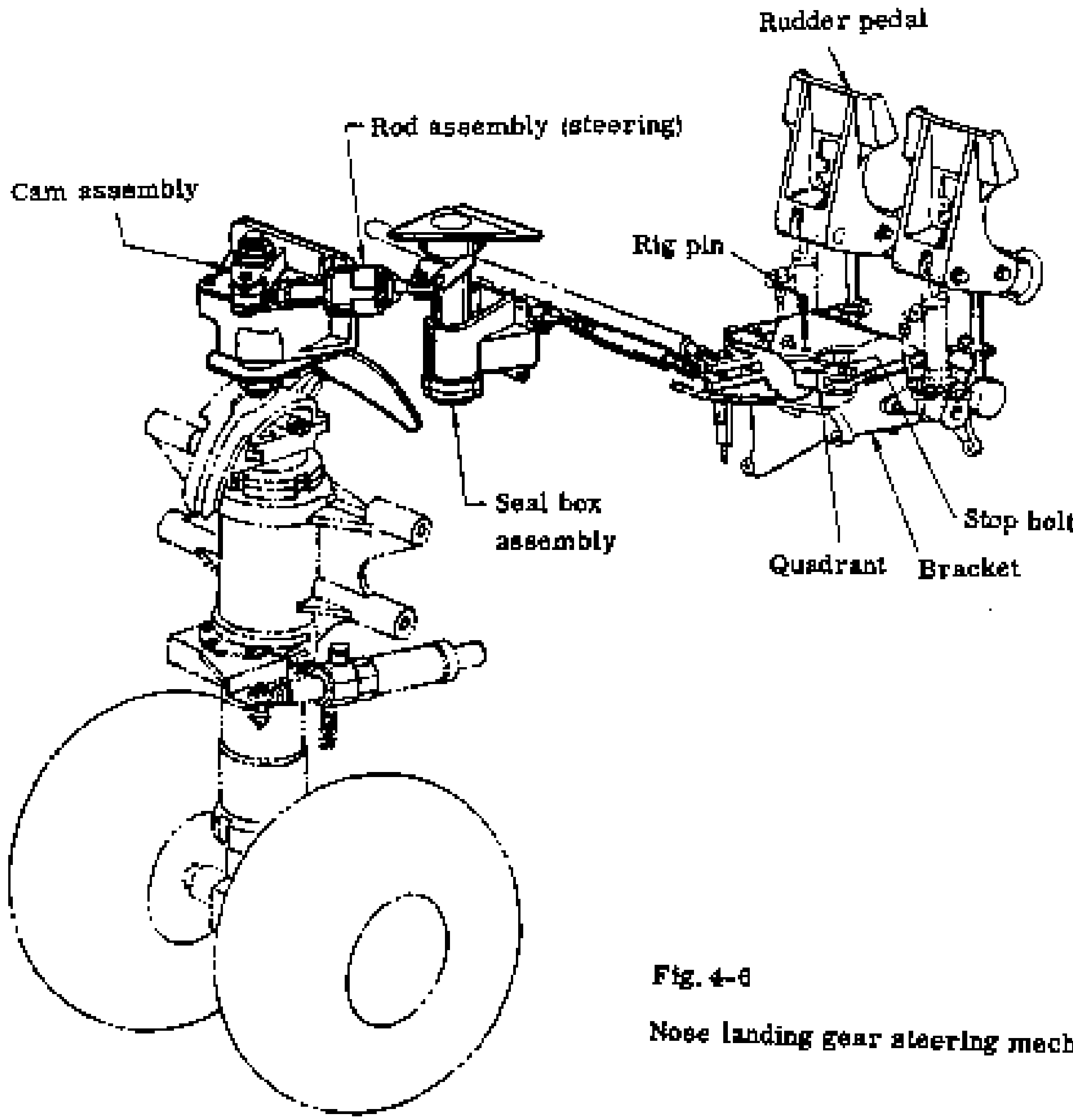


Fig. 4-6

Nose landing gear steering mechanism

2.6.1 REMOVAL AND INSTALLATION OF STEERING MECHANISM

Remove and install in accordance with the following procedures.

2.6.1.1 CAM (See Fig. 4-7)

(1) Jack aircraft.

(2) Remove nose strut.

(3) Remove steering rod. Separate cam from arm on top of cam assembly. To remove steering rod, loosen lock nut at both ends of steering rod. Rotate rod and remove the rod only.
(4) Remove 4 bolts and remove cam assembly.
(5) Remove nuts and roller assembly and draw out cam.
(6) Reinstall in reverse sequence of removal.

2.6.1.2 STRAIGHTENER AND ROLLER (See Fig 4-7)
(1) Remove straightener from strut by removing 4 bolts from straightener.
(2) Remove roller by removing bolt attaching roller to the top of the nose strut.
(3) Reinstall in reverse sequence of removal.

2.6.2 ADJUSTMENT OF NOSE LANDING GEAR STEERING MECHANISM

Jack aircraft and adjust nose landing gear steering mechanism in accordance with the following procedures. see Fig 4-6:

(1) Attach steering angle protractor (GSE 016A-99058-11 or equivalent) to nose landing gear.
(2) Insert rig pin in the quadrant under the pedals and obtain neutral point of the steering range.
(3) Adjust the length of the rod assembly (steering), so that the red mark on the trunnion may coincide with the red mark on the cylinder, and set the protractor graduation at "0".
(4) Remove the rig pin from quadrant.
(5) Disconnect rudder cable at turnbuckle F STA 8605, push rudder pedal to full travel with nose gear down.
<u>Applicable to Aircraft S/N 652SA</u>
Check steering angles 29° ± 1° left and 26° ± 1° right when the quadrant is against the stop, and no interference within the range of movement.
<u>Applicable to Aircraft S/N 661SA, 697SA and subsequent</u>
Check steering angles 29° ± 1° left and 26° ± 1° right when the quadrant is against the stop, and no interference within the range of movement.
(6) After connecting rudder cable with specified tension, push down pedal to full travel and check nose wheel for the following steering angles.

STEERING DIRECTION	STEERING ANGLE	
	S/N 652SA	S/N 661SA, 697SA & UP
Left	21° ± 2°	23° ± 2°
Right	24° ± 2°	22° ± 2°

(7) Remove steering angle protractor.
(8) Perform nose gear up and down operation while the left or right rudder pedal is pushed down and check the roller (top of the nose strut) for smooth movement in the cam and straightener.

NOTE

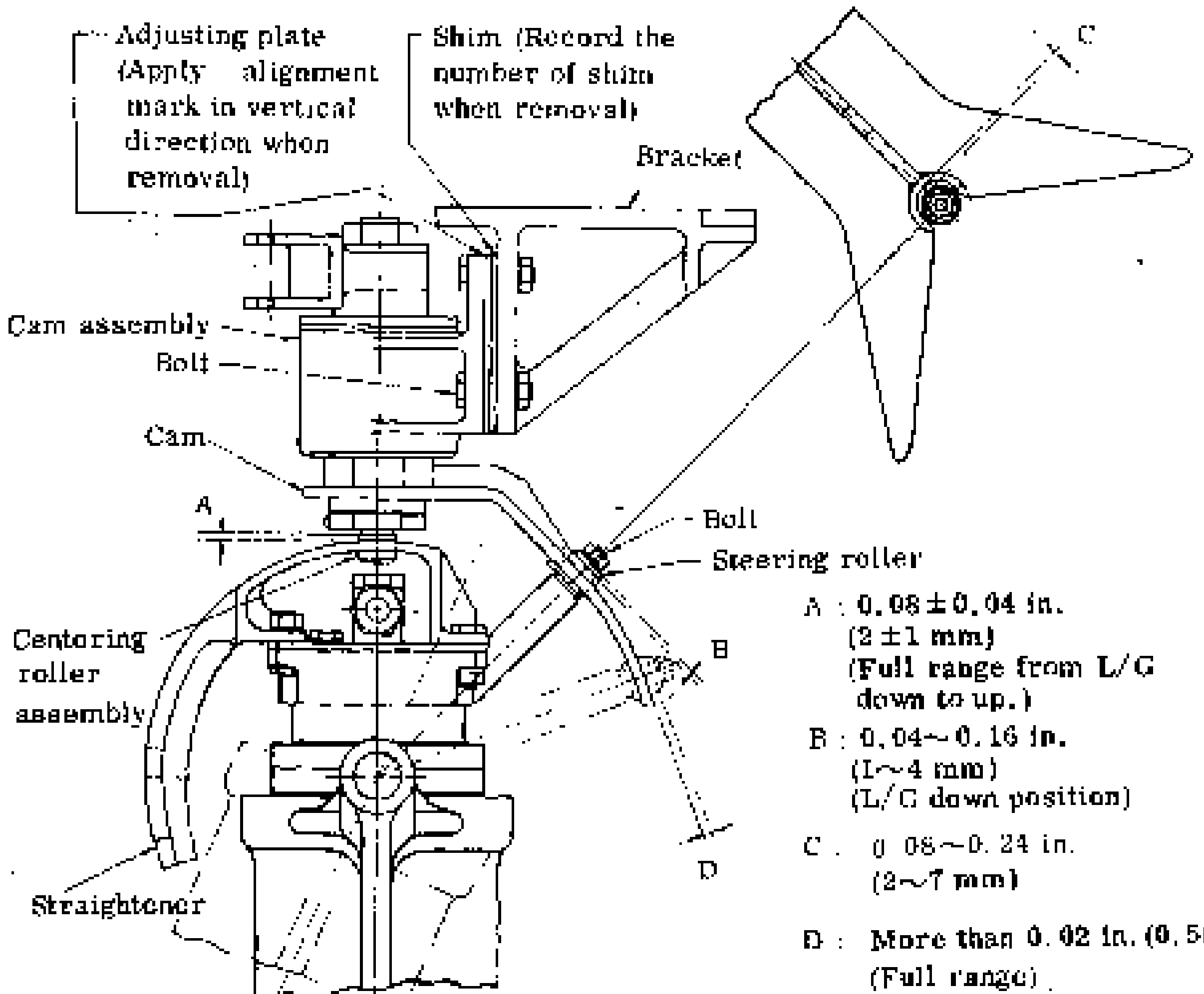
Ascertain that nose gear is not twisted in retracted position and that clearance difference between LH and RH tires and keels is less than 0.2 in. (5.0 mm).

2.6.3 ADJUSTMENT OF CAM ASSEMBLY (See Fig 4 - 7)

(1) Adjust cam assembly so that bite between centering roller and straightener may be dimension A in Fig. 4-7 and also bite between steering roller and cam may be dimension B.

(2) The steering roller should move smoothly and meet dimension C in Fig. 4-7 while the nose gear is steered to the right and left in down locked position.

(3) Extend nose gear with the wheel steered to full extent. The dimension D in Fig. 4-7 should be met when the steering roller comes in contact with the cam.



A : 0.08±0.04 in. (2±1 mm) (Full range from L/G down to up.)

B : 0.04~0.16 in. (1~4 mm) (L/G down position)

C : 0.08~0.24 in. (2~7 mm)

D : More than 0.02 in. (0.5mm) (Full range)

Fig. 4-7

2.7 SHIMMY DAMPER

2.7.1 REMOVAL AND INSTALLATION OF SHIMMY DAMPER (See Figs 4-8 and 4-9)

(1) Remove bolt attaching shimmy damper rod end.

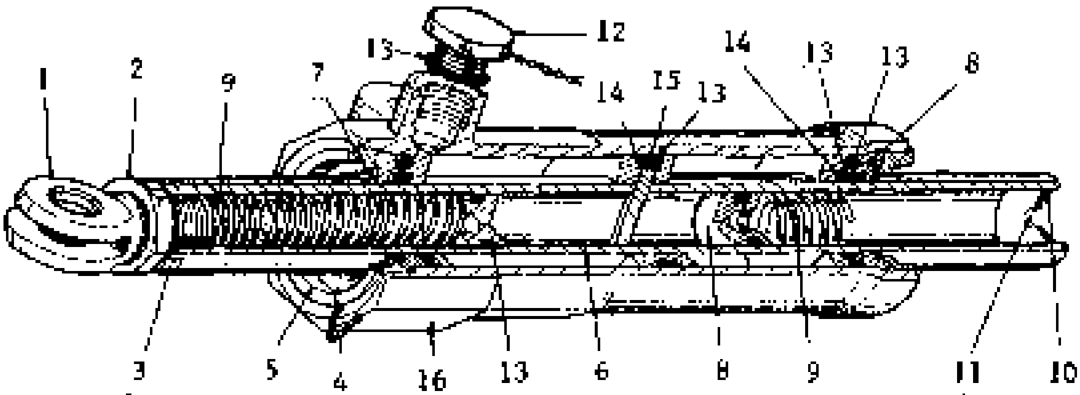
(2) Remove bolt connecting shimmy damper housing and nose strut.

(3) Reinstall in reverse sequence of removal.

The following are check items of installation:

(a) Check bolts, nuts and bushings for deformation and cracks.

(b) Tighten attaching bolt finger tight. Install cotter pin.
Insert adjustment shim of suitable thickness into the forward and aft attaching points of shimmy damper to eliminate free play.



1. Rod end
2. Nut
3. Washer
4. Lockwire
5. Retainer
6. Piston
7. Wiper
8. Plug
9. Spring
10. Retainer ring
11. Spring retainer
12. Filler cap
13. "O" ring
14. Back up ring
15. Cap seal
16. Housing

Fig. 4-8 Shimmy damper assembly

2.7.2 DISASSEMBLY, AND REASSEMBLY OF SHIMMY DAMPER

For disassembly and reassembly of shimmy damper, see Overhaul Manual YET70058.

2.7.3 ADJUSTMENT OF SHIMMY DAMPER (See Fig. 4-10)

(1) Check shimmy damper for fluid level by inserting 0.04 in. (1 mm) diameter probe in the hole opposite to rod end. If the probe is not allowed to enter 3.23 to 3.35 in. (82 to 85 mm) deep, fill shimmy damper with fluid in accordance with the following procedures.

NOTE

The depth to which the probe is inserted should be within 3.23 to 3.35 in. (82 to 85 mm) just after reassembly of shimmy damper. While in servicing, the maximum allowable depth is 3.94 in. (100 mm). Make adjustment if 3.94 in. (100 mm) is exceeded.

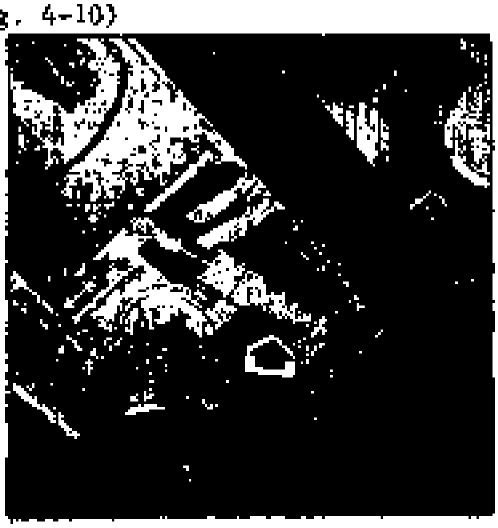


Fig. 4-9 Shimmy damper

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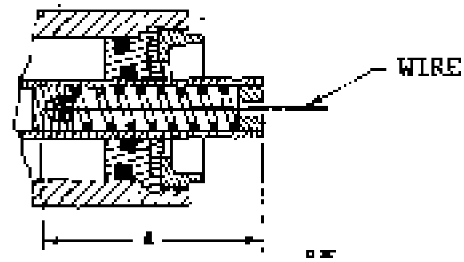

Dimension "A" = 3.29 to 3.94 in.
(83.6 to 100.1 mm)

Fig. 4-10 Adjustment of shimmy damper

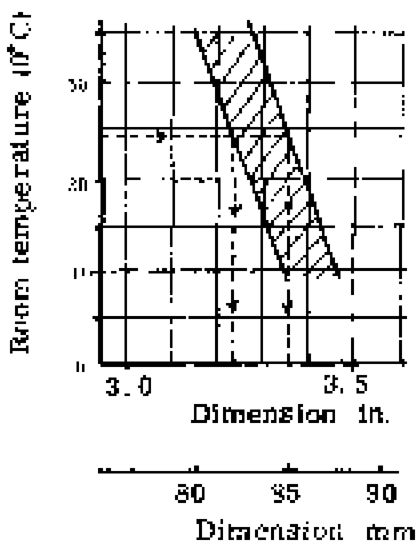

Fig. 4-11 Temperature-dimension diagram

(a) Remove shimmy damper from nose gear.
(b) Insert a screw assembly (Fig 4-12) into end opposite rod end. Insert through the hole and tighten into plug (Fig 4-8 Item 1). Slide washer of screw assembly to shimmy damper housing and tighten the nut against the washer until resistance is felt, which indicates spring compression.
(c) Remove filler cap.

NOTE

Slowly and gently compress spring in piston rod. Care must be taken not to damage O-ring when removing filler cap.

(d) Fill piston rod with actuating fluid MIL-H-5606.
(e) Install filler cap and release spring compressed per Para. (b) above.
(f) Operate shimmy damper by hand as quickly as possible. Check for unusual resistance and no leakage. Fluid level check probe must be freely inserted to a depth of 3.23 to 3.35 in. (82 to 85 mm).
(g) Reinstall shimmy damper to nose gear.

(2) Check shimmy damper attaching point for any abnormal looseness. Adjustment should be made per Para. 2.7.1 (b) to eliminate free play.
If looseness is detected at rod end side, replace the spacer (010A-39129-5) and adjust.

(3) Check shimmy damper attaching parts for proper lubrication.

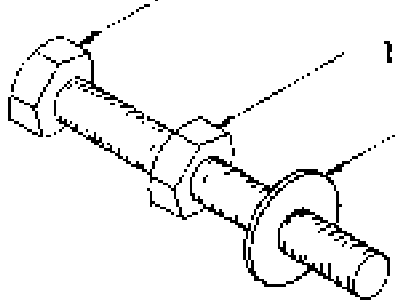

Fig. 4-12 Screw assembly

2.8 NOSE LANDING GEAR UP INDICATOR SWITCH (See Fig 4-13)

2.8.1 REMOVAL AND INSTALLATION OF NOSE LANDING GEAR UP INDICATOR SWITCH

The nose landing gear up indicator switch is installed on the LH sidewall of nose gear well and is connected with "UNSAFE" indicating circuit to indicate nose gear in retracted position.

(1) Remove lower forward LH nose access panel.
(2) Remove the pendulum which connects switch link and lever assembly and detach switch link.
(3) Loosen the lock nut attaching the switch lever and remove the lever.
(4) Remove the nuts attaching the switch and remove the switch.
(5) Install in reverse sequence of removal.

2.8.2 ADJUSTMENT OF NOSE LANDING GEAR UP INDICATOR SWITCH

After installation of nose landing gear up indicator switch, adjustment should be made in accordance with the procedures described in Para. 10.2.

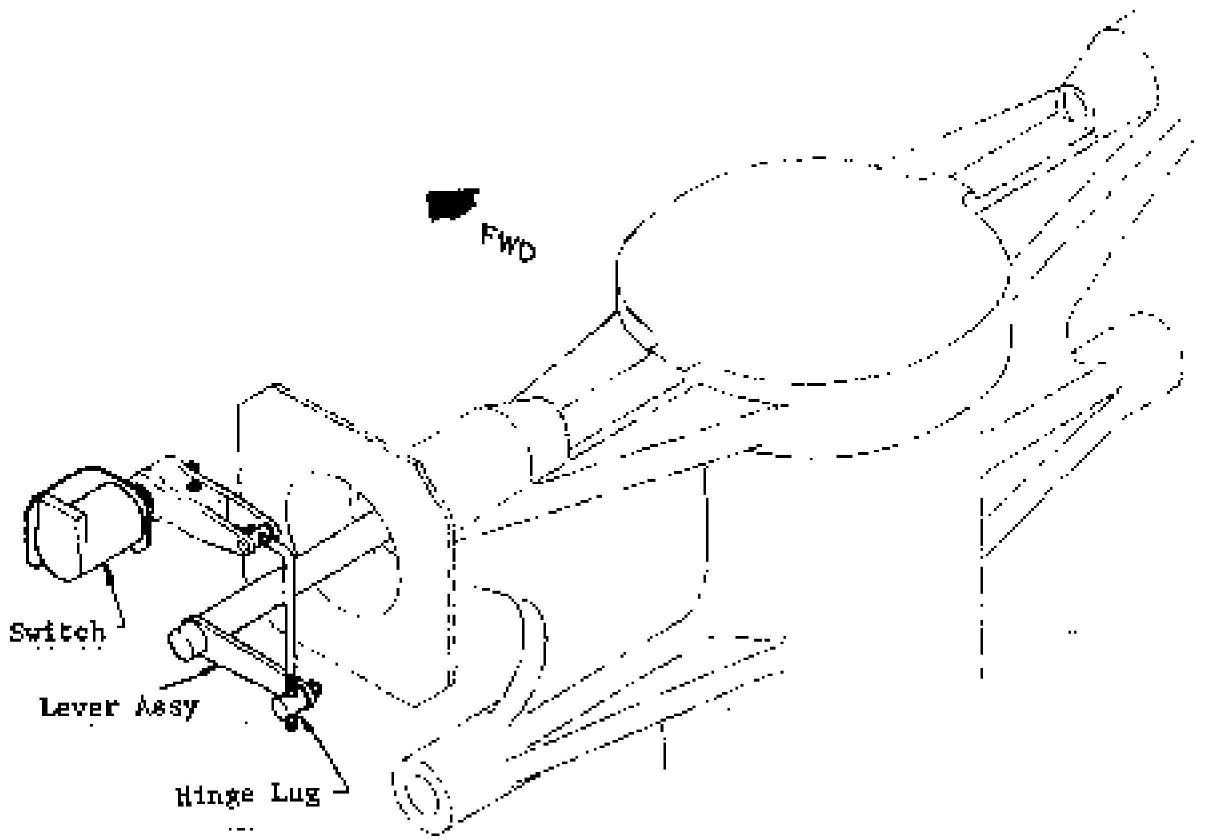

Fig. 4-13 Nose gear up indicator switch

2.9 NOSE LANDING GEAR DOWN LOCK

2.9.1 ADJUSTMENT (See Fig 4-14 and 4-15)

(1) Adjust location of nose gear down lock limit switch (located on RH keel, nose wheel) in accordance with procedures described in Para. 10.1, so that the switch may be actuated when down lock (LH side) of nose gear drag strut is placed in the groove more than 0.24 in. (6 mm) deep with gear in down position.

(2) Check for a clearance of 0.002 to 0.012 in. (0.05 to 0.3 mm) between nose gear drag strut and drag strut pin (See Fig 4-2).

(3) With nose gear in down locked position, check downlock for more than 0.08 in. (2 mm) clearance from the lock groove end at RH side, more clearance at LH side.

(4) Perform nose gear retraction and extension under a load of 11 lbs (5 kgs) on the axle to the airplane nose direction. Adjust the eccentric bolt on the LH drag strut so that the down lock may be operated smoothly under the following conditions.

(a) No abnormal noise on down lock (LH) in gear retraction.

(b) The difference between LH and RH down lock engagements is less than *0.12 in. (3 mm) when the down locks are about to engage in gear extension.

(5) Check for the following items in gear extension.

(a) RH downlock engages *0 to 0.06 in. (0 to 1.5 mm) deeper than LH downlock.

(b) The down lock has no interference with other parts up to to the end of its travel.

* When the difference between LH and RH exceeds the specified values, replace lock assembly with a new one.

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Fig. 4-14 Nose gear down lock

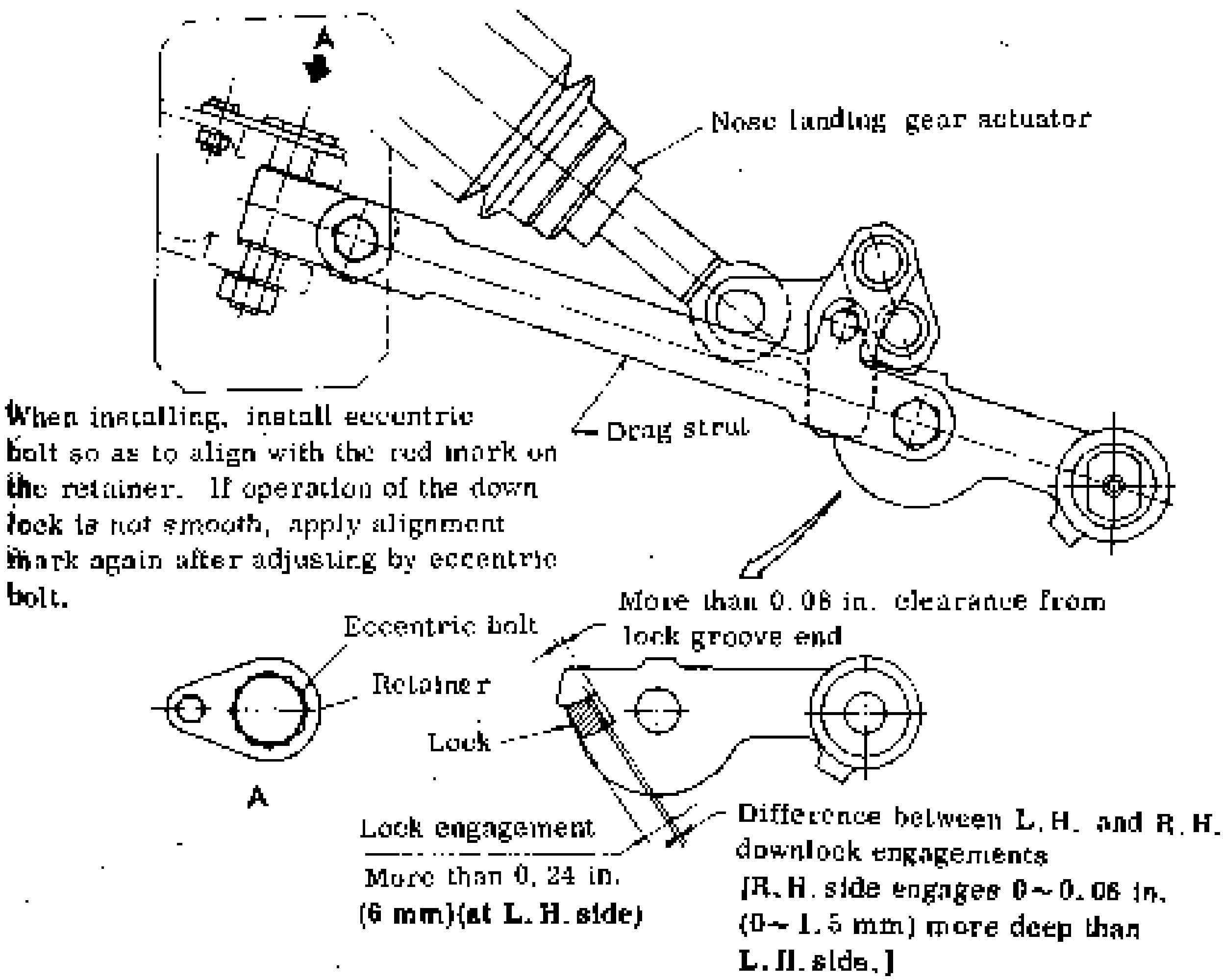


Fig. 4-15 Adjustment of nose gear down lock

2.10 NOSE LANDING GEAR DOOR MECHANISM (See Figs. 4-16 and 4-17)

The nose landing gear door mechanism is operated by up or down movements of nose gear strut. The sliding surface of the bungee spring and the contact surface of the strut should be kept free of dust and foreign objects at all times.

2.10.1 REMOVAL AND INSTALLATION OF NOSE LANDING GEAR DOOR MECHANISM

(1) Remove rods (L.H. and R.H.) connecting to the nose landing gear doors.

(2) Remove bolts (L.H. and R.H.) connecting the airframe fitting and arm. Remove door mechanism.

(3) Install in reverse sequence of removal.

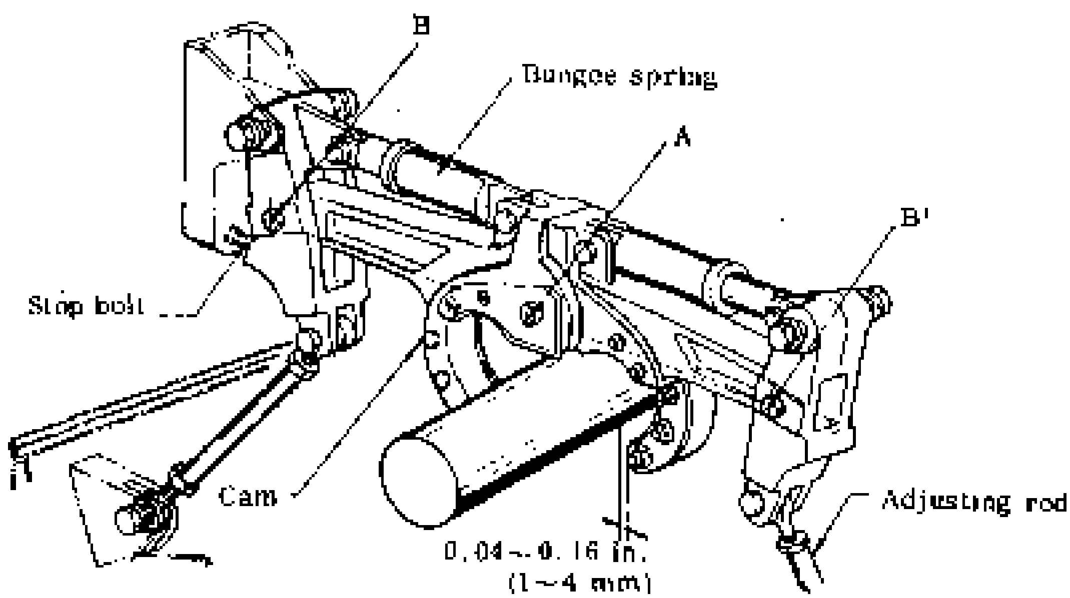


Fig. 4-16 Adjustment of nose gear door mechanism

2.10.2 ADJUSTMENT OF NOSE LANDING GEAR DOOR MECHANISM

(1) With the nose gear down and locked, adjust the stop bolt so that the arm touches the stop bolt when point A comes to point A' as illustrated in Fig. 4-16 and 4-17 and make sure that the point A may overcenter the line B-B'. After tightening the nut on the stop bolt, apply alignment mark.

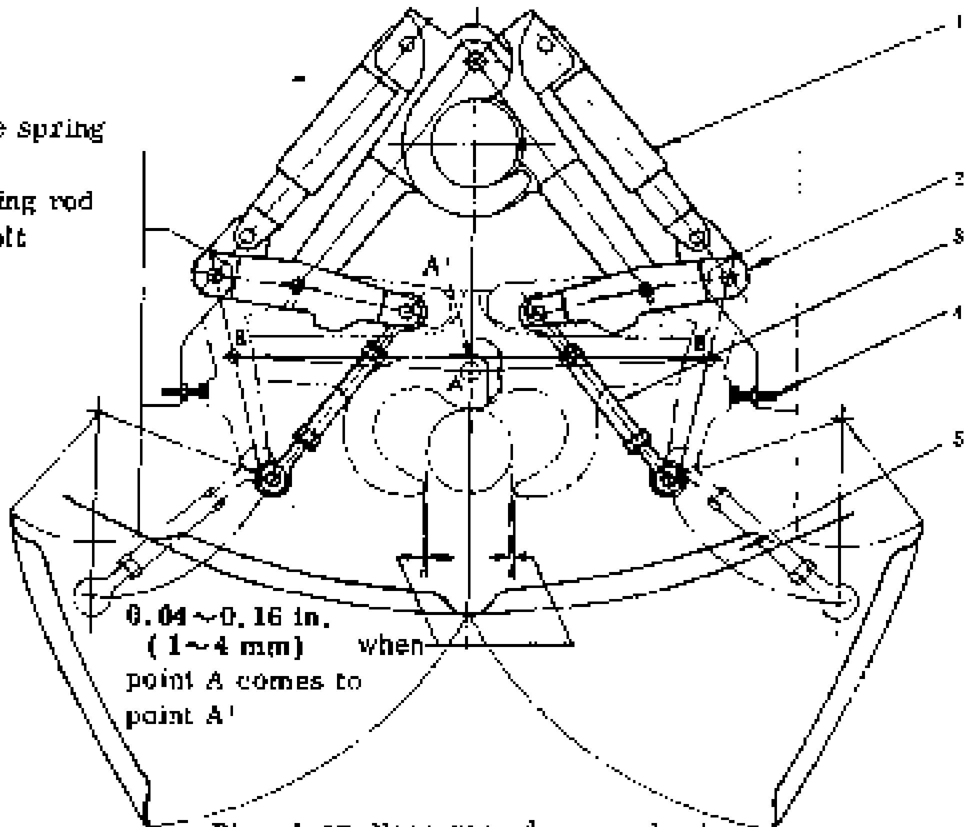


Fig. 4-17 Nose gear door mechanism

(2) When the point A falls on the line B – B', adjust the cam so that the end of the jaw may be located 0.04～ 0.16 in.(1～4 mm) away from the nose strut.

(3) Adjust the rod so that clearance between the doors and tire may be not less than 1.2 in.(30.5 mm) when the doors are open and not less than 0.8 in. (20.3 mm) when the doors are closed.

(4) Retract the nose gear till the clearance of the mechanical stop becomes zero. Hang a weight of 20 lbs (9 kg) on the center edge of each forward door and make sure that the doors come off the door jambs 0.39 to 0.59 in. (10 to 15 mm). Adjust the rod length if necessary.

(5) The distance between LH and RH nose landing gear door at the rear-lower end, should be more than 18in at the gear down condition. (See Fig 4-17A) (S/N 697 and up)

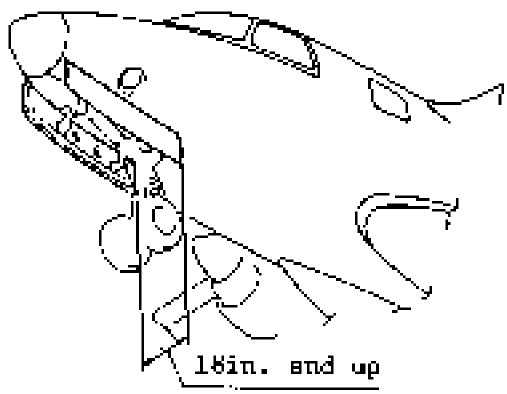


Fig. 4-17A. Adjustment of Nose Landing Gear Door

2.11 TORQUE LINK ASSEMBLY (See Fig 4-18)

The handle assemblies which are able to disconnect upper and lower torque links are installed on torque link assembly.
In towing, the upper and lower links shall be disconnected.

2.11.1 REMOVAL AND INSTALLATION

(1) Pull out L.H. and R.H. handles and disconnect upper and lower torque links.

(2) Remove attaching bolt of upper link, pull out pin and remove upper link from airplane.

(3) Remove attaching bolt of lower link, pull out pin and remove lower link from airplane.

(4) Cut off safety wire and remove handle assembly from the removed upper link.

(5) Reinstall in reverse sequence of removal.

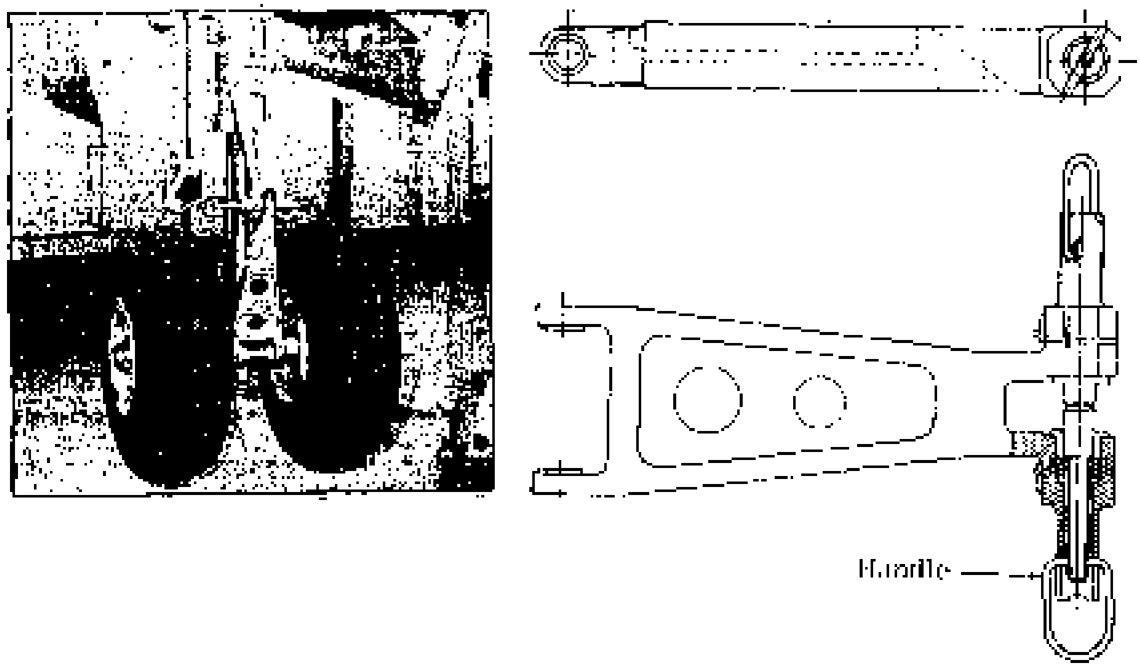


Fig. 4-18 Nose gear torque link assembly

2.12 MAINTENANCE OF NOSE LANDING GEAR

(1) For maintenance of oleo strut, perform per Para. 2.5.2.

(2) Keep shimmy damper piston rod clean by wiping with soft cloth moistened with hydraulic fluid MIL-H-5606.

(3) Check quantity of hydraulic fluid in shimmy damper.
When fluid is insufficient, refill fluid in accordance with procedures given in Para. 2.7.3.

When there is any evidence of air mixed in fluid, disassemble and refill with hydraulic fluid.

3. MAIN LANDING GEAR

3.1 GENERAL DESCRIPTION

The main landing gear consists of the wheel, tire, brake, axle, leg assembly, oleo strut, drag strut, and position rod. (See Fig. 4-19)
The main gear aft door is connected mechanically to the oleo strut. The main gear forward door is operated separately during landing gear retractions by means of an independent electric actuator.
The oleo strut is the standard air and oil type, consisting of cylinder and piston. The strut absorbs landing and taxiing shock loads. The drag strut supports load on the ground and also serves as a screw jack for gear retraction. The position rod, connected to the leg assembly and axle, receives horizontal loads from the axle and allows retraction of the wheels into the fuselage by swiveling the axle during gear retraction. The ground safety switch is mounted on the left gear leg assembly and prevents the gear from retracting while aircraft is on the ground.

3.2 REMOVAL AND INSTALLATION OF MAIN LANDING GEAR (See Fig 4-19)

Procedures for removing, installing, disassembling, inspecting and reassembling major components of main landing gear are given below.

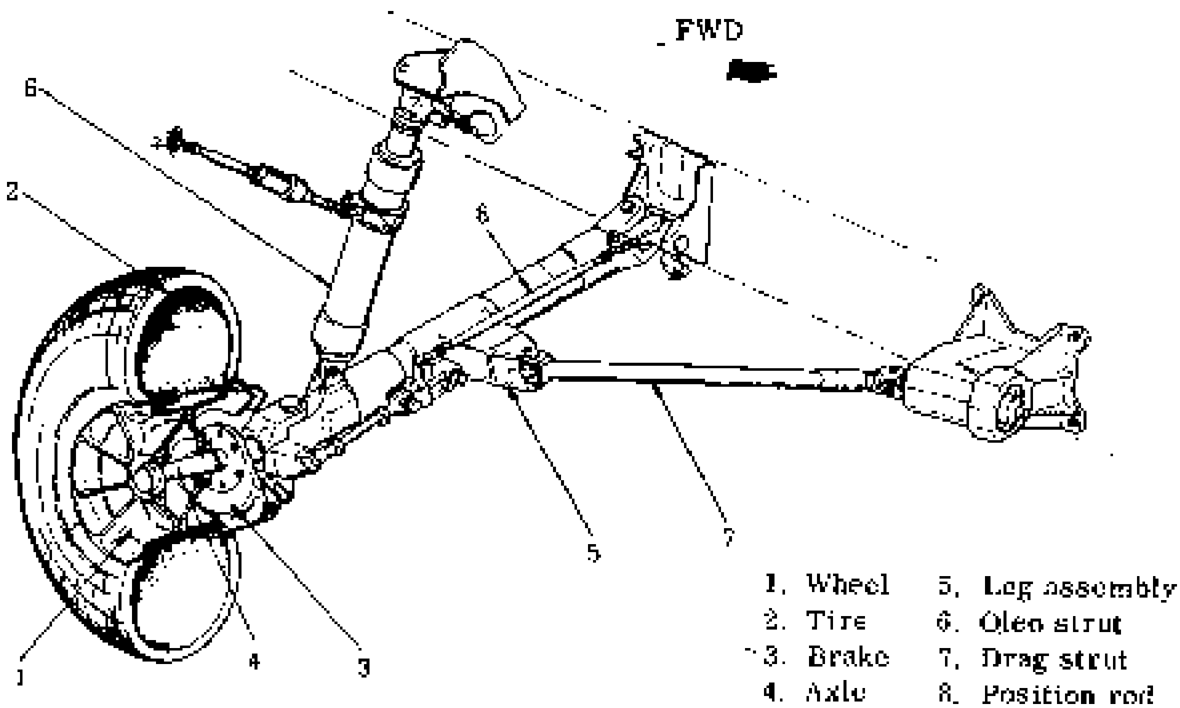


Fig. 4-19 Main landing gear

3.3 WHEELS

3.3.1 REMOVAL AND INSTALLATION

(1) Set parking brakes.

(2) Remove ring and cap at outside hub.

(3) Remove nut and washer from axle.

When reinstalling, tighten the nut to a torque of about 600 in-lbs (686 kg-cm) first. Loosen and retroque to 300 in-lbs (343 kg-cm). Loosen the nut again to align the castle nut groove to pin hole, then install cotter pin.

(4) Remove outer bearing cone.

(5) Remove wheel from axle.

(6) Install in reverse sequence of removal.

3.3.2 DISASSEMBLY, REASSEMBLY AND INSPECTION

(1) For disassembly and reassembly of wheels, see Para. 5.3 of this manual.

NOTE

(i) Care should be taken not to damage brake disc and bearing cup, when removing wheel from axle.

(ii) When installing wheel, install inner bearing cone to wheel first, then install wheel assembly on the axle.

3.4 BRAKES

3.4.1 REMOVAL AND INSTALLATION (See Fig 4-20)

(1) Remove wheels, in accordance with Para. 3.3.1 of this section.

(2) Disconnect brake tubing.

(3) Remove bolts, washers and nuts attaching brake housing to axle.

When reinstalling, tighten bolt to a torque of 90 to 110 in-lbs (104 to 127 kg-cm). Apply anti-seize compound MIL-T-5544 on bolt thread, surface under bolt head, and contact surfaces of nut and washer before installation.

(4) Remove brake from axle.

(5) Install in reverse sequence of removal.

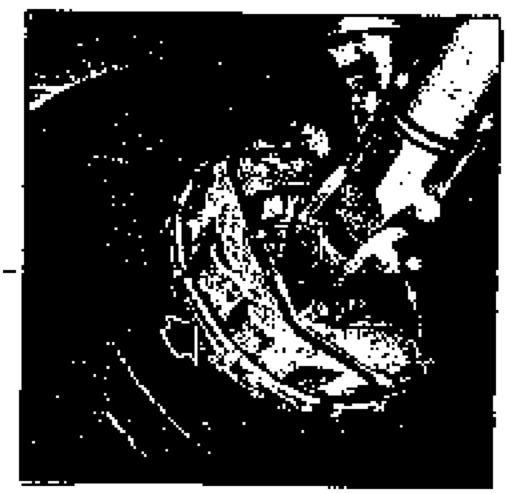


Fig. 4-20 Removal of brake

3.4.2 DISASSEMBLY, REASSEMBLY AND INSPECTION

(1) For disassembly and reassembly of brakes, see Para. 5.7 of this manual.

3.5 DRAG STRUT

3.5.1 REMOVAL AND INSPECTION

(1) Remove universal joint pin and slide connecting to leg assembly.

(2) Remove universal joint pin and slide connecting to gear box and drag strut support.

(3) Remove drag strut from aircraft.

(4) Install in reverse sequence of removal. When installing, adjust universal joint washer and nut to be concentric and tighten.

NOTE

Before removing drag strut, measuring and recording length of screw protruding from barrel will facilitate reinstallation.

3.5.2 DISASSEMBLY, INSPECTION AND REASSEMBLY OF MAIN LANDING GEAR DRAG STRUT (See Fig 4-21)

3.5.2.1 DISASSEMBLY

(1) After removing drag strut from aircraft, rotate rod by hand and take rod out of barrel.

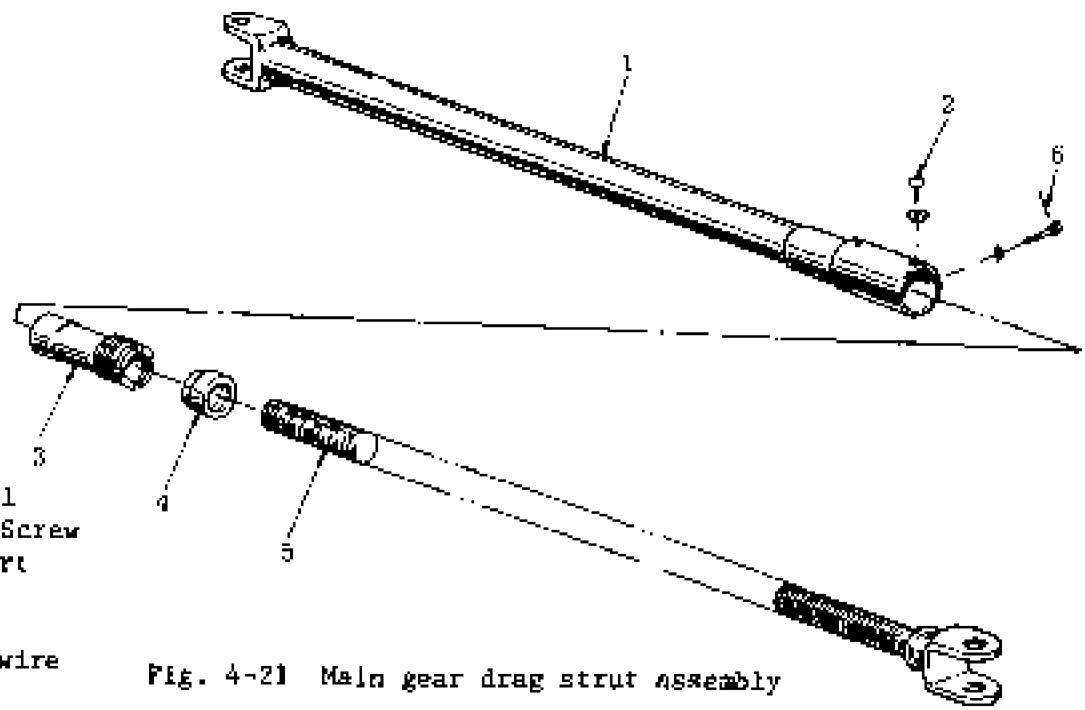
(2) Remove lock screw from barrel.

Rotate and remove support and ring.

3.5.2.2 INSPECTION

(1) Wash all parts in cleaning solution (Federal Specification P-D-680).

(2) Check acme thread carefully for wear.



1. Barrel
2. Lock Screw
3. Support
4. Ring
5. Rod
6. Lock wire

Fig. 4-21 Main gear drag strut assembly

(3) Fit support in rod. Check for play in axial direction. Play shall not exceed 0.018 in. (0.46 mm) for overall thread section.

(4) Inspect rod thread surface for condition of lubricant. Apply Molycote if necessary.

NOTE

It is not necessary to replace rod, even if lubricating film is broken on the surface of thread section of rod. MIL-G-21164 Molycote grease is usually applied to the thread for lubrication at 100 hour intervals. Apply Molycote grease by hand as necessary depending upon conditions on the surface of thread section.

3.5.2.3 REASSEMBLY

(1) Apply Molycote grease (MIL-G-21164) on the inner surface of support.
(2) Apply MIL-G-21164 grease to threaded portion of rod and screw rod into barrel.
(3) Screw in support and ring and install two lock screws. Apply lockwire.
(4) Assemble universal joint and connect to gear box and drag strut support, and leg assembly. Tighten nut and install new cotter pin.

NOTE

(i) Attaching bolt on the leg assembly side should be installed with its head on the inner side of airplane.

(ii) When ring is replaced with new one, make a drill hole of #21(4.04 mm dia.) from lock screw hole on the barrel to ring side after installing support and ring onto the barrel, and cut a thread of #10-32NF-3B. Reassemble in accordance with the above procedures.

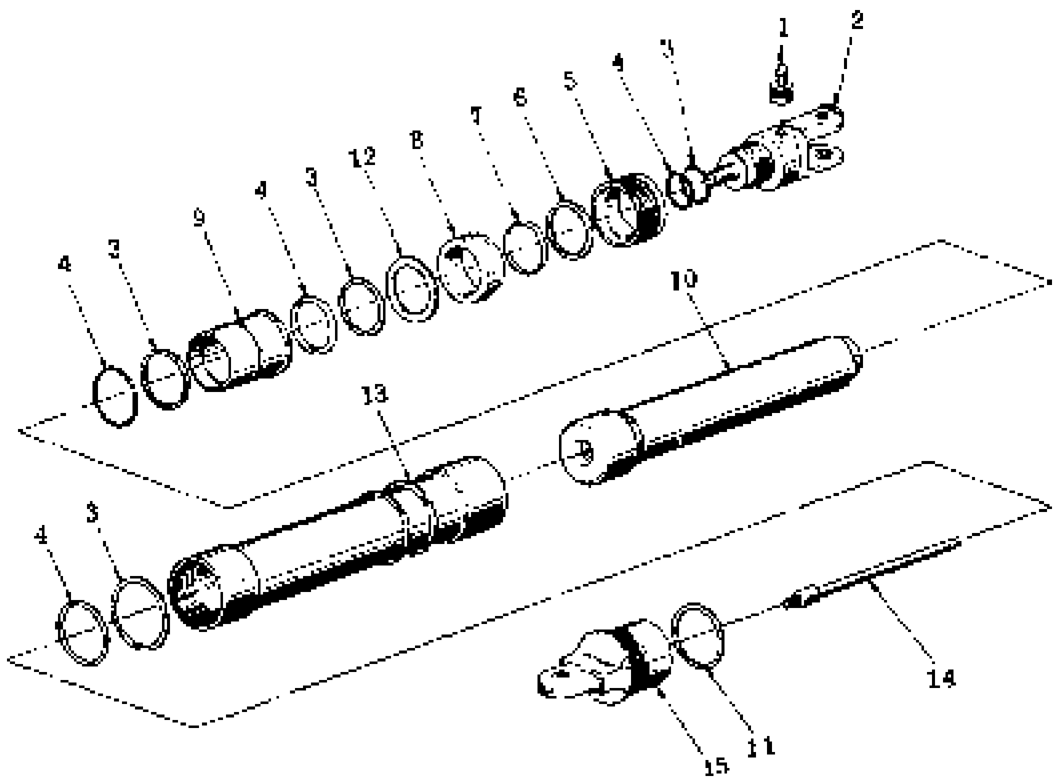
3.6 OLEO STRUT (Fig 4-22)

3.6.1 REMOVAL AND INSTALLATION

(1) Disconnect rod that links main gear forward door and oleo strut.
(2) Retract main gear half-way and remove cotter pin from upper connecting bolt of oleo strut.
(3) Extend main gear and remove upper universal joint of oleo strut and lower connecting bolts of leg assembly. For installtion, tighten upper and lower connecting bolts hand tight, and install cotter pin.
(4) Remove oleo strut from aircraft.
(5) Install in reverse sequence of removal.

NOTE

Before removing oleo strut, place axle of leg assembly on a stand. Care should be taken not to damage leg assembly.



1. Air valve
2. Upper fitting
3. Backup ring
4. O ring
5. Nut
6. Scraper
7. Felt
8. Bearing
9. Ring
10. Piston ass'y
11. Shim
12. Retainer ring
13. Cylinder
14. Metering pin
15. Lower fitting

At assembly, adjust relation of cylinder and lower fitting by shims. Standard thickness of shims is 0.014 in. (0.36 mm).

Fig.4-22 Main gear oleo strut

3.6.2 DISASSEMBLY, INSPECTION AND REASSEMBLY

This work should be done at the authorized repair facility. Confer with Beech Aircraft Corporation for more detail information.

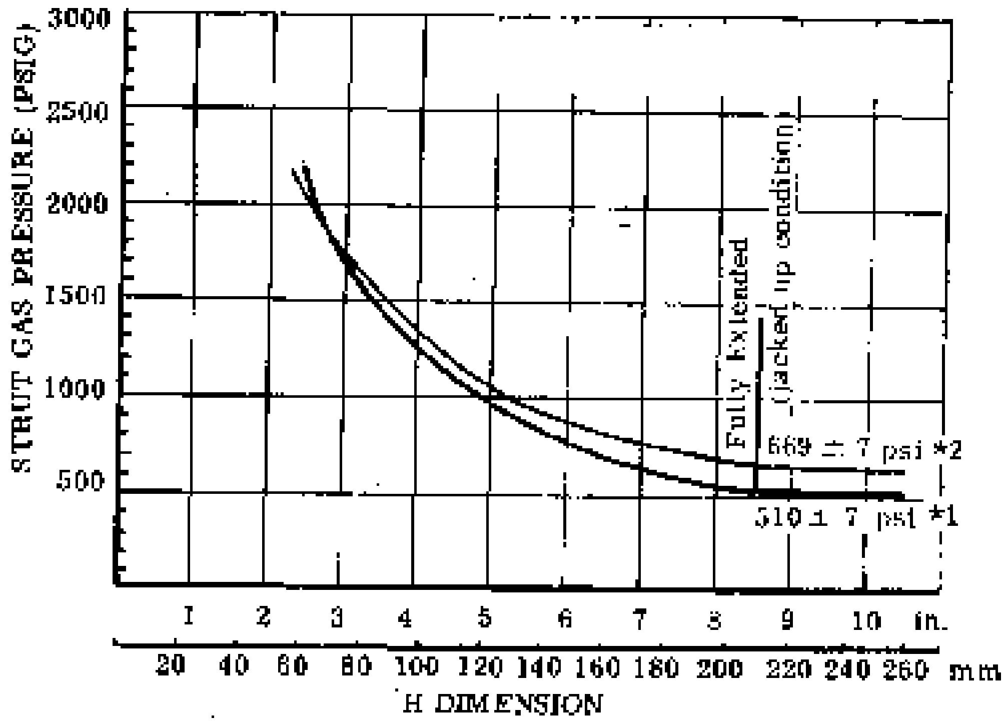
3.6.3 MAINTENANCE

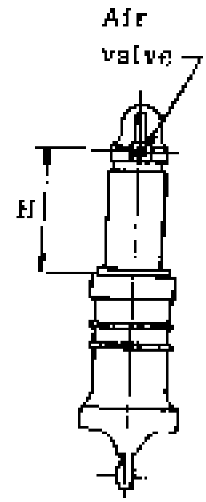
(1) Keep exposed piston rod clean by wiping it with a soft cloth moistened with hydraulic fluid. (MIL-H-5606)
(2) Inflate strut with N_2 gas to 510 ± 7 psi (35.7 ± 0.5 kg-cm^2)(*1), 669 ± 7 psi (46.8 ± 0.5 kg-cm^2) with strut fully extended or until dimension "H" and pressure conforms with the following chart within + 20 psi (1.4 kg-cm^2).

CAUTION

When the temperature varies greatly from the temperature at the time of last inflation, pressure and dimension "H" will not conform with the chart. Reinflate strut to conform with the chart.

*1 For oleo strut 030A-38151-11
*2 For oleo strut 030A-38151-21





(3) If necessary, check strut for fluid level in accordance with the following procedures.

(a) Remove oleo strut.

(b) Bleed gas through air valve. Remove valve.

NOTE

Do not loosen air valve body until gas is completely bled from air valve.

(c) Place oleo strut in fully compressed position. Fill strut held upright with fluid MIL-H-5606 through filler port.

(d) Extend strut to full length and return to fully compressed position. Fill strut with fluid.

(e) Repeat these steps two or three times.

(f) Reinstall air valve. Install oleo strut on aircraft. Apply N_2 gas pressure of 510 psi (*1), 669 psi (*2) through air valve.

(g) Leave the strut for one hour, and check for gas and fluid leakage.

*1 For oleo strut 030A-38151-11

*2 For oleo strut 030A-38151-21

(4) Installation of air valve shall be performed as follows:

(a) Make sure that the valve and O-ring are clean.
(b) Lubricate O-ring and install into the valve, being careful not to damage it.
(c) Screw the valve into boss until the hexagonal portion of the valve body contacts with the boss. Tighten to a torque of 100 to 110 in-lbs (115 to 127 kg-cm).
(d) Tighten swivel nut to a torque of 50 to 70 in-lbs (58 to 81 kg-cm).
(e) Install valve cap finger-tight.
(f) Apply lockwire.

(5) N_2 gas shall be filled into the oleo strut as follows:

(a) Remove valve cap.
(b) Connect gas supply source to the valve stem.
(c) Loosen swivel nut.

NOTE

Do not turn the hexagonal body when loosening swivel nut.

(d) Tighten swivel nut to a torque of 50 to 70 in-lbs (58 to 81 kg-cm) after filling oleo strut to a proper pressure.
(e) Disconnect gas supply source and install valve cap finger-tight.

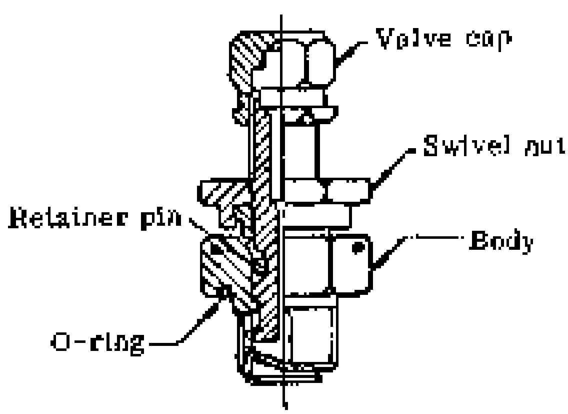


Fig. 4-23 Removal of bolt

3.7 LEG ASSEMBLY

3.7.1 REMOVAL AND INSTALLATION

(1) Remove axle, brakes, drag strut and oleo strut.
(2) Remove connecting bolt (see Fig 4-25) of slide and pin, and remove attaching pin of airframe by tapping (see Fig 4-23). Tighten bolt to a torque of 160 to 190 in-lbs (184 to 219 kg-cm) when installing.

(3) Remove leg assembly with position rod slide and axle attached.

(4) Install in reverse sequence of removal.

3.8 INSTALLATION AND INSPECTION OF MAIN LANDING GEAR (See Fig. 4-24, 4-25)

(1) Check bolts, washers, nuts and pins for distortion and cracks.

(2) When installing pins and bolts, lubricate all parts which require lubrication.

(3) Adjust drag strut length to the same dimension as that measured at the time of removal.

(4) With main gear in extended position (drag strut thread length 0.67 to 0.98 in. (17 to 25 mm), adjust clearance "E" in Fig.4-25 to 0.012 ±0.004 in. (0.3±0.1 mm) when oleo strut is fully extended.

NOTE

Do not retract main gear before completion of the adjustment described in Para.(4) above.

(5) Adjust position rod length as follows.

(a) Lift the aircraft on jack and level it horizontally. (See MU-2 STRUCTURAL REPAIR MANUAL YET-72C35, Section 1, Para.5.3)

(b) Suspend weight from the specified point of the aircraft and draw the center line of the aircraft at the longitudinal on the ground.

(c) Set moving nut of mechanical stop so that is comes in contact with the mechanical stop at gear down position. By hand, adjust the barrel of the main landing gear drag strut assembly to rotate downward. Confirm that the drag strut length is 0.69± 0.04 in. (17.4 ±1 mm).

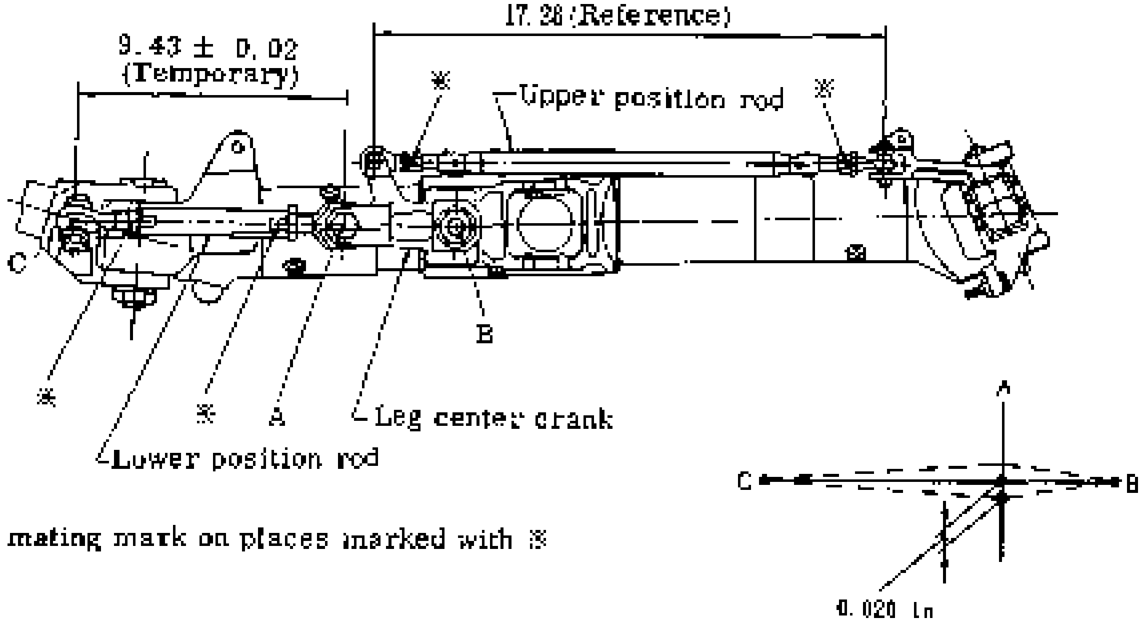
(d) Temporarily set the position rod length to the dimension shown in Fig.4-24

(e) Confirm that center line of lower position rod and leg center crank is within 0.02 in.(0.5 mm). If it is more than 0.02 in.(0.5 mm), adjust length of the upper position rod.

(f) Position a one (1) meter ruler inside the wheel and suspend weights at both ends of the ruler(or set the GSE 030A-99011 to the wheel boss and suspend weights one (1) meter apart). Project a shadow of the weight on the ground and adjust the length of lower position rod as the slope of wheel is 0.67in.(17 mm) to 0.87 in.(22 mm) tow-out for 39.37 in.(1000 mm) length on judgement in the different distance between the shadow and the center line of aircraft axis.

(g) Confirm that center line of lower position rod and leg center crank is within 0.02 in.(0.5 mm) as shown in Fig.4-24. If it is more than 0.02 in.(0.5 mm), adjust the length of upper position rod and accomplish adjustment from step (f).

(h) Adjust lower and upper position rod end bearings so they do not hang up on main gear retraction and extension lugs.



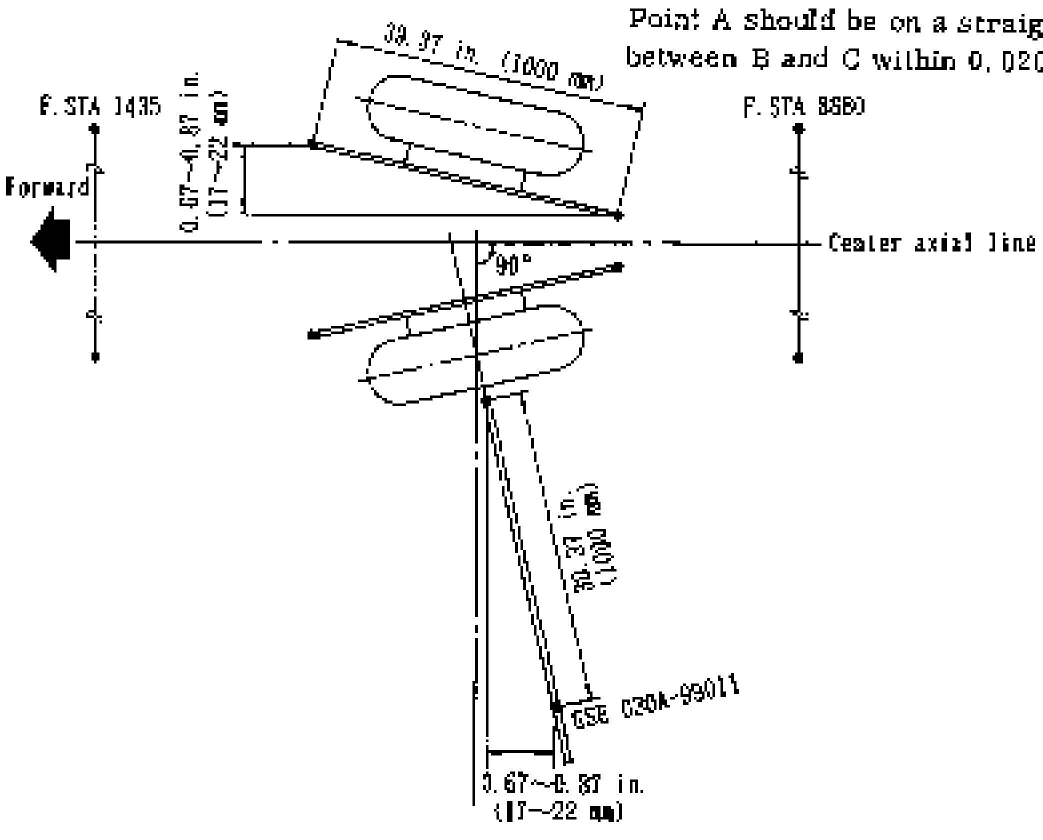


Fig. 4-24 Adjustment of position rod length

NOTE

If adjustment is accomplished on this configuration, aircraft is 0 through 5/1000 tow-in condition with max take-off weight with aircraft on ground.

After completion of the above adjustments, make sure that upper and lower rod lengths are within the following limits.

Lower rod length : dimension in Fig.4-24 ±0.04 in. (±1 mm)

Upper rod length : dimension in Fig.4-24 ±0.12 in. (±3 mm)

(6) Make sure that clearance "C" in Fig.4-25 is 0.02 to 0.04 in. (0.5 to 1 mm) with main gear in retracted position (drag strut thread length 17.64±0.04 in. (448±1 mm)). If necessary, adjust by scraping phenolic pad.

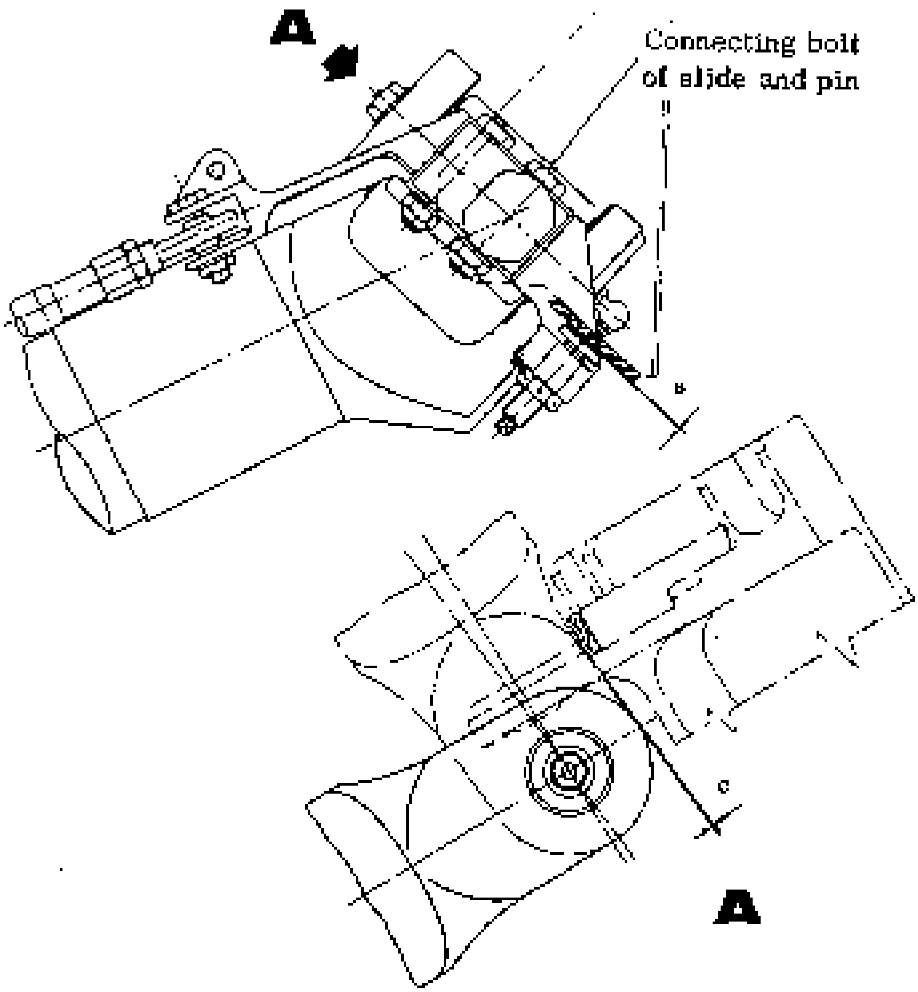


Fig. 4-25

3.9 MAIN LANDING GEAR DOOR MECHANISM

The L. H. and R. H. main landing gear doors are composed of forward and aft doors. The aft doors are connected mechanically with the main gear oleo strut and operated with the landing gear. The forward doors are opened before and closed after gear operation by an electric actuator. The forward door mechanism is connected mechanically to the main gear aft door lock mechanism which is unlocked when the forward doors open.

3.9.1 MAIN LANDING GEAR AFT DOOR MECHANISM (See Fig 4-26)

The main landing gear aft door mechanism consists of a rod having rod ends on both ends and a spring in center. One end of the rod is connected to main gear oleo strut and the other end to main gear aft door. Its length can be adjusted by means of rod ends.
The lock mechanism is operated together with the forward door mechanism, and also operated by emergency gear extension handle.

3.9.1.1 ADJUSTMENT OF MAIN LANDING GEAR AFT DOOR MECHANISM

Adjustment should be made by moving gear up and down slowly with no load and no inertia.

(1) Jack airplane.

(2) Adjust rod length so that the dimension "A" in Fig. 4-26 may be 0.40 to 0.44 in. (10 to 11 mm) when up side nut of power train mechanical stop reaches the end of its travel.

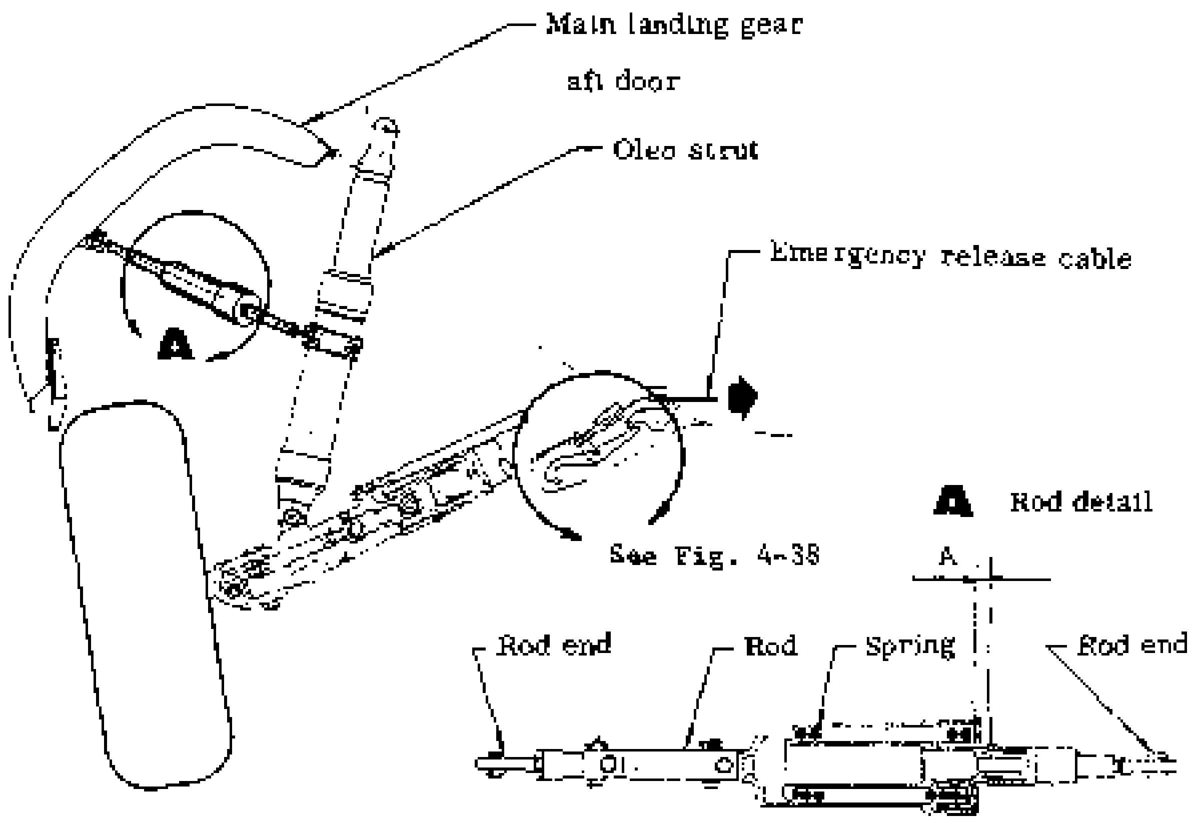


Fig. 4-26 Main landing gear door mechanism

(3) Under the conditions of Para. (2) above, check for the clearance between main landing gear forward and aft doors. The clearance shall be 0.08 ± 0.02 in. (2 ± 0.5 mm) at upper edge and 0.16 ± 0.02 in. (4 ± 0.5 mm) at lower edge. (The clearance between upper and lower edges changes almost linearly.) If necessary, trim again (See Fig 4-27).

(4) Check main gear up indicator switch for the following items:

(a) The switch actuates at the position where mechanical stop nut is less than 0.32 in. away from the end of its travel.

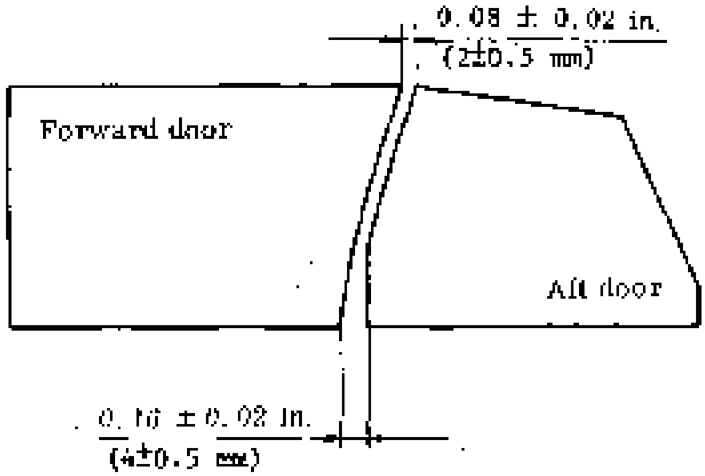
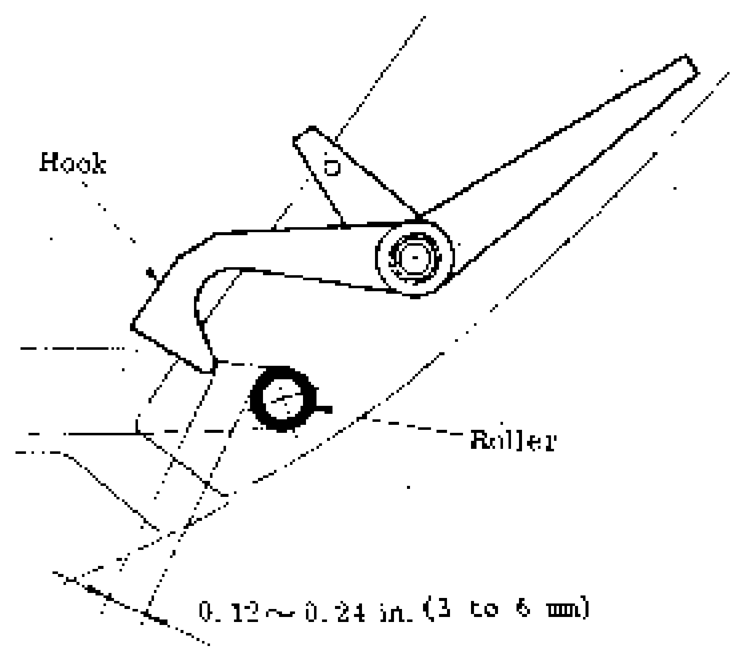


Fig. 4-27 Clearance between forward and aft doors

(b) With landing gear retracted, the forward door closed and the aft door loaded, the "UNSAFE" light must not come on until the door hook strikes the roller.

(5) Check by moving the aft door for the clearance between the aft door roller and hook tip as shown in the following figure.



3.9.2 MAIN LANDING GEAR FORWARD DOOR MECHANISM (See Fig 4-28)

The main gear forward door mechanism consists of an electric actuator (located in the RH forward main wheel well), clutch, rods and levers and a hook mechanism. This door mechanism is designed so that the hook is released by the first motion of the electric actuator, then the doors are opened.

The clutch disconnects the doors from the actuator in the event of trouble in the electric system. The disconnection is made by operation of the emergency gear down handle in cockpit together with door lock release.

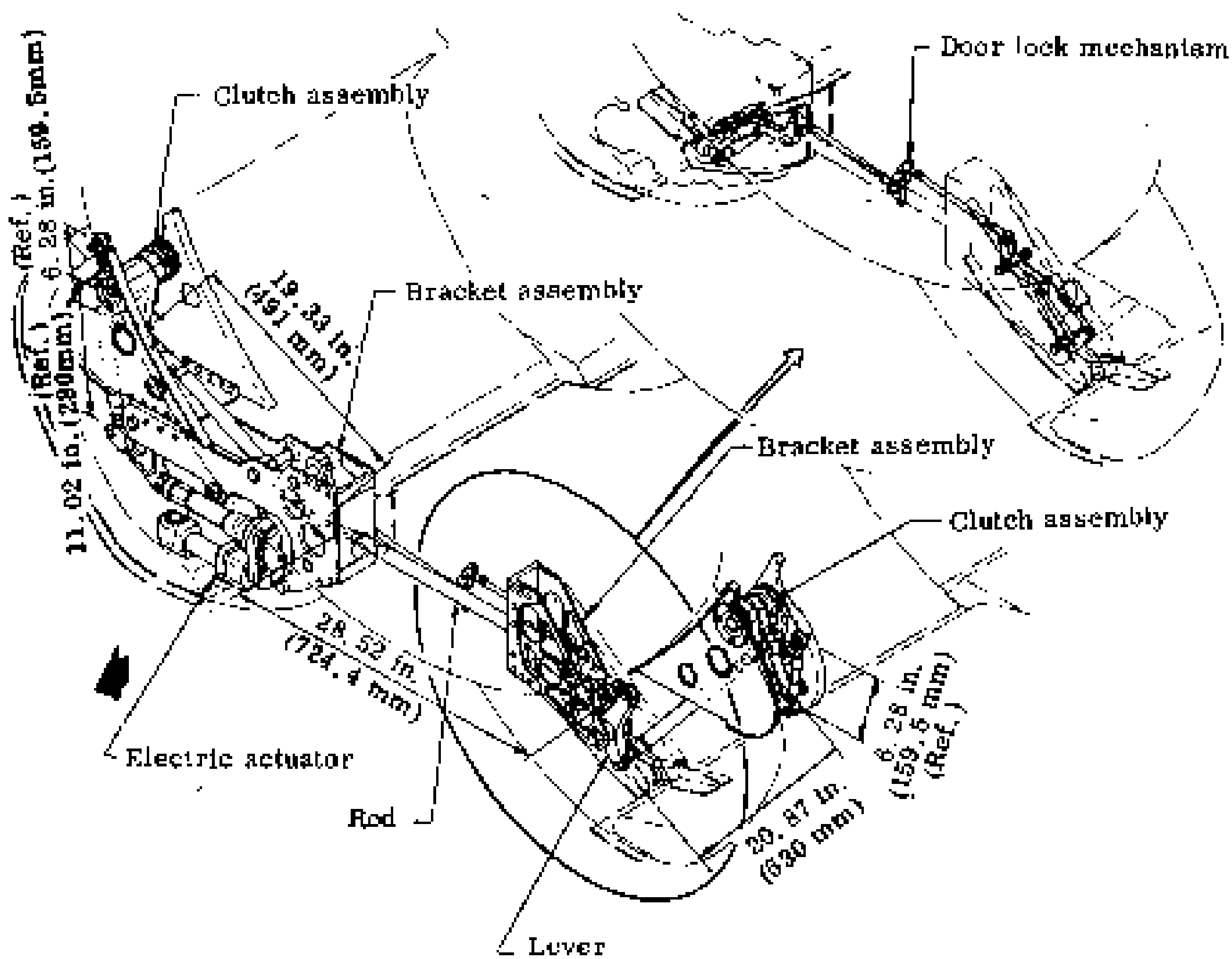


Fig. 4-28 Main landing gear forward door mechanism

3.9.2.1 OPERATION OF MAIN LANDING GEAR FORWARD DOOR

(1) Door closed and locked condition (See Fig 4-29).

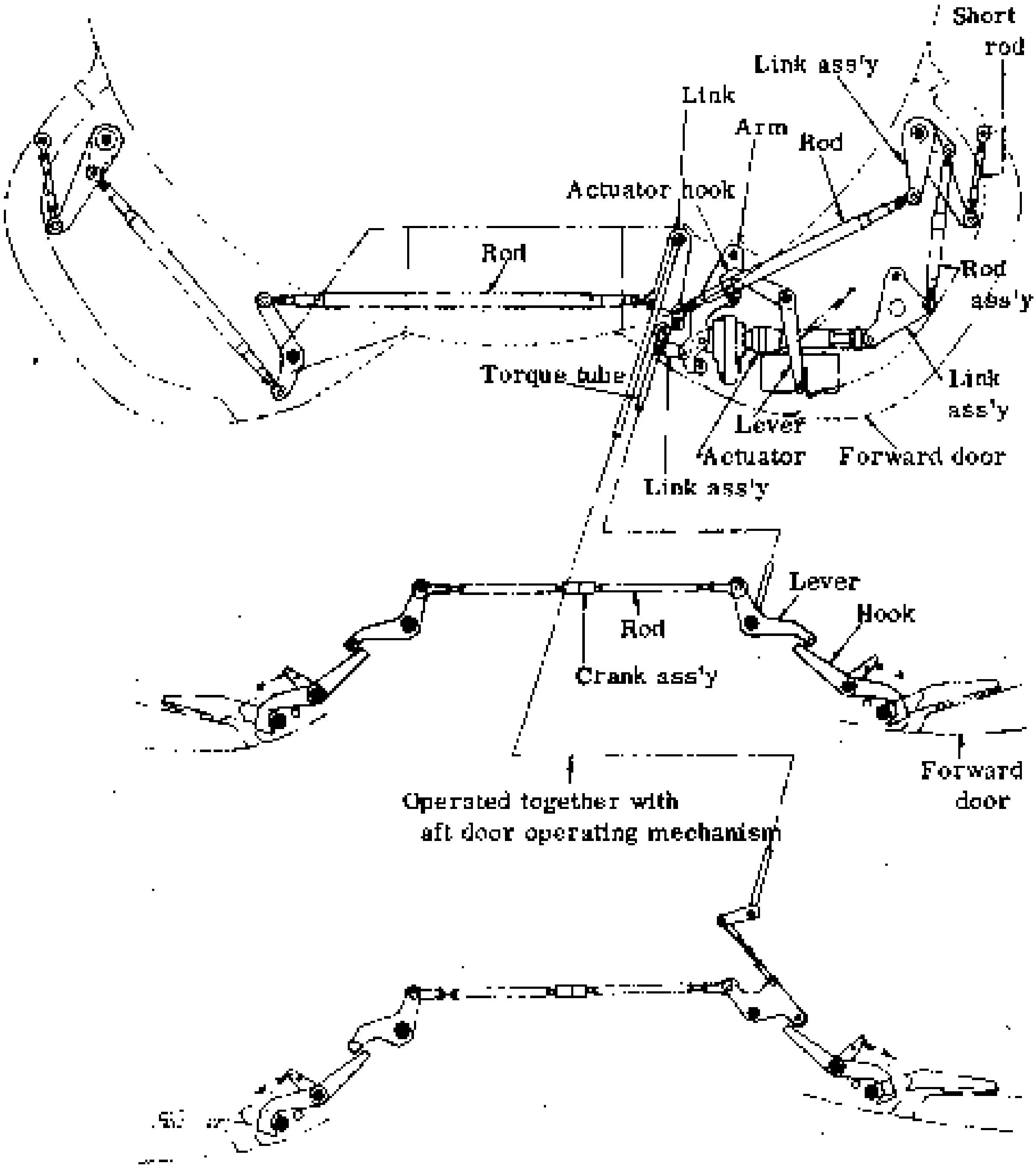


Fig. 4-29 Main landing gear forward door closed and locked condition

(2) Door opening (See Figs 4-30 and 4-31).

(a) When electric actuator actuates, the actuator shaft extends to the arm side, the direction of the arrow. The other link assembly side is fixed, because the door is still locked.

(b) Arm, link, torque tube, lever, rod and lock hook move or rotate to the direction of arrow respectively and lock is released. Then, actuator lock hook becomes free, because the door is opened slightly.

(c) The door is pulled to the open side furthermore by air pressure, so actuator which is connected with the door, moves outboard slightly and actuator lock hook is locked. (actuator locked condition.)

NOTE

When R.H. door lock is released earlier than L.H, actuator may be locked before L.H. door lock is released. In this case, actuator shaft continues to extend with L.H. door in locked condition, so shear pin will be sheared off.
Therefore, adjustment must be made so that L.H. door lock may be released earlier than R.H. side. (See Para. 3.9.2.3)

(d) Due to actuator lock, arm side of actuator shaft is fixed and actuator shaft extends to the link assembly side only in the opposite direction of Para. (a).

(e) Link assemblies, rod assembly and short rod, etc., move or rotate to the direction of arrow and door is opened.

(f) When door is about fully opened, operation of electric actuator is stopped by limit switch and door opening is stopped.

(3) Door closing

Door closing is made in reverse sequence of door opening.

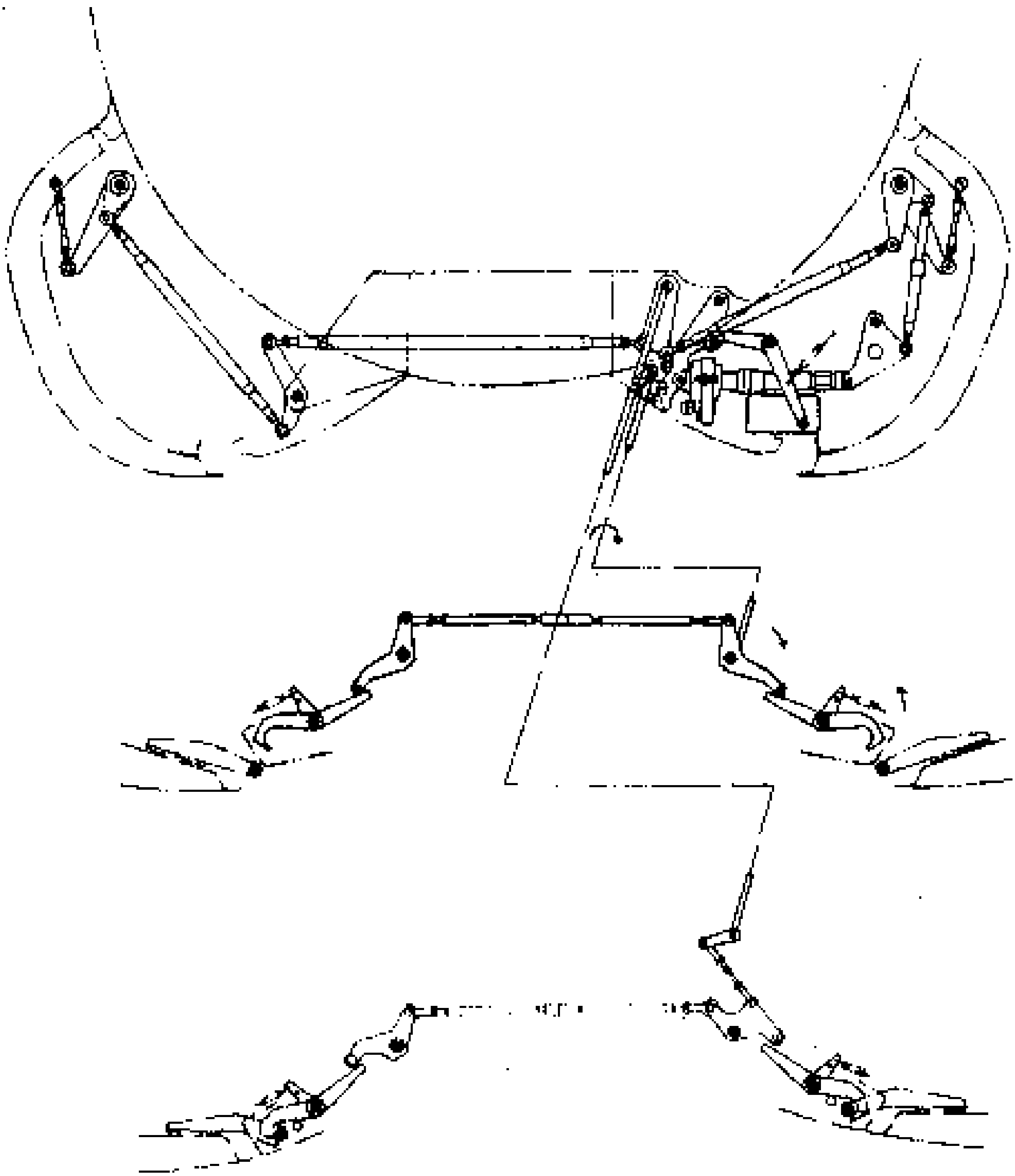


Fig. 4-30 Main landing gear forward door lock released condition

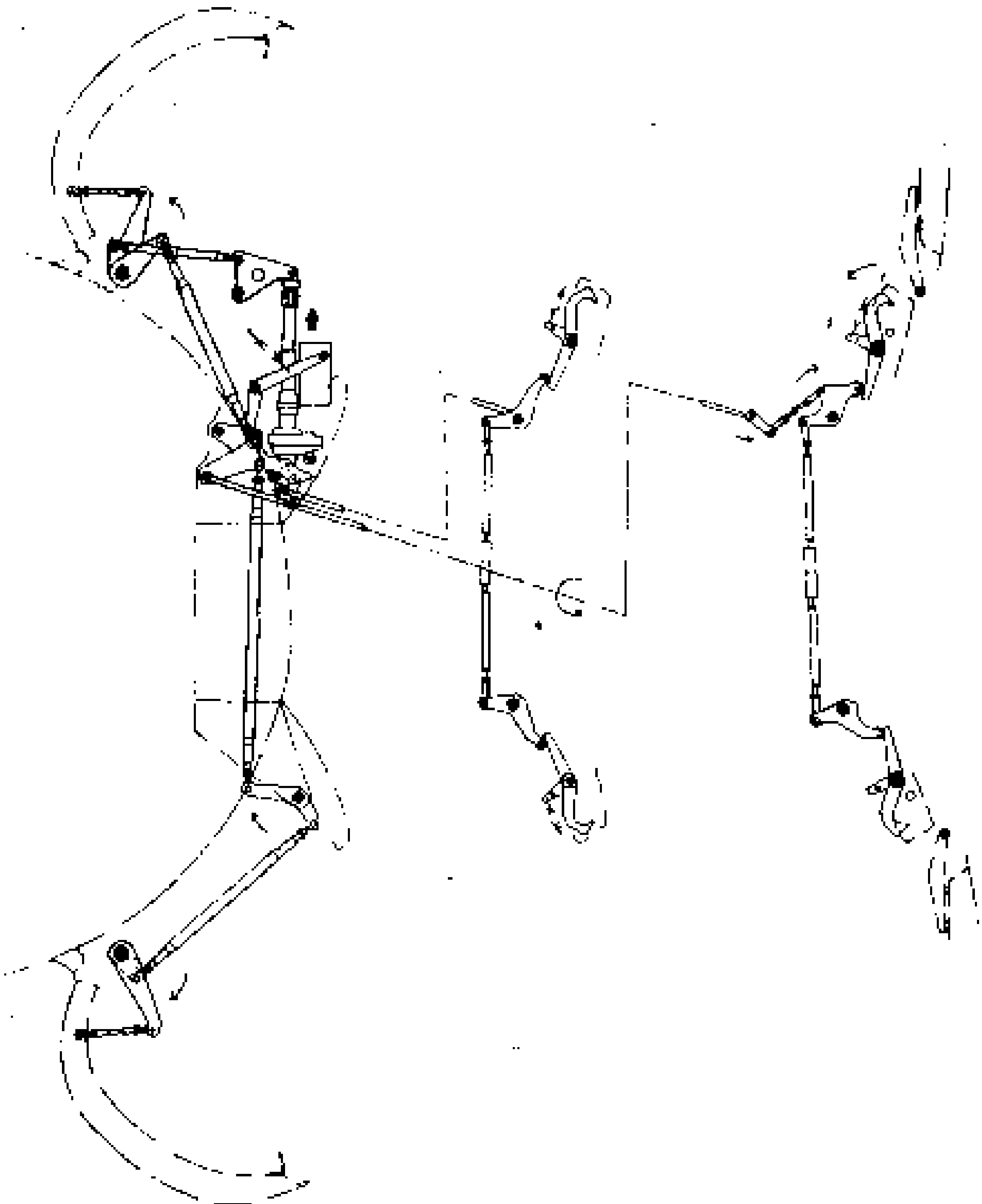


Fig. 4-31 Main landing gear forward door opened condition

3.9.2.2 ADJUSTMENT OF FORWARD DOOR MECHANISM

(1) Disconnect the forward doors from door mechanism by removing the short rod. (See Fig 4-32)

(2) Operate the actuator independently. Check the actuator shaft for extended and retracted lengths at the position where limit switch actuates.

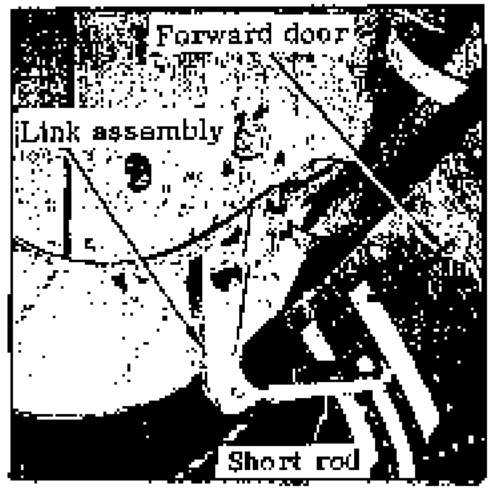


Fig. 4-32 Removal of forward door

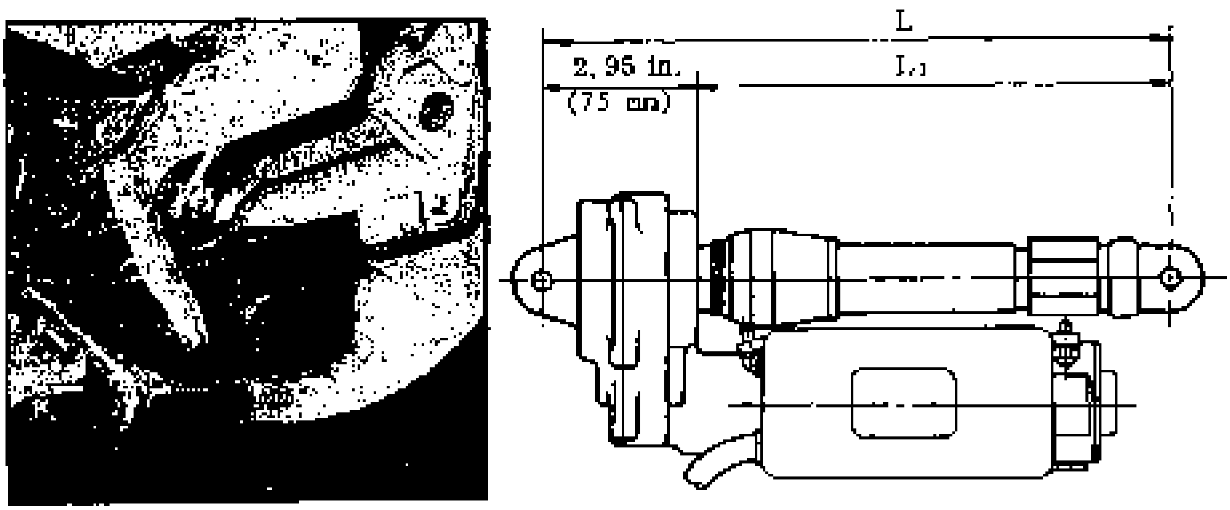


Fig. 4-33 Check actuator

Type	Extended length in. (mm)	Retracted length in. (mm)
AA-5C	$17.3\ ^{+0.3}_{-0}$ in. ($439.4\ ^{+7.6}_{-0}$ mm)	$12.03\ ^{+0}_{-0.3}$ in. ($305.6\ ^{+0}_{-7.6}$ mm)

Note: The shaft length measured should be the span (L) between attaching holes of actuator shaft, but L_1 + 2.95 in. (75 mm) may be used.

(3) Adjustment of R.H. stop (See Fig 4-34).

(a) With actuator in locked position, adjust stop "A" so that the clearance between stop "A" and arm may be 0.028 ~ 0.04 in.(0.7 to 1 mm).

(b) Release actuator hook and adjust stop "B" so that arm may strike stop "B" when actuator moves 0.55 ~ 0.59 in. (14 ~ 15 mm).

(4) **Adjust stop "E" so that the clearance "D" may be 0.12 ~ 0.2 in. (3 ~ 5 mm) by visual inspection when door (with no load) strikes stop "C".**

(5) Adjust each rod to the dimensions shown in Fig. 4-28 and connect to the forward door.

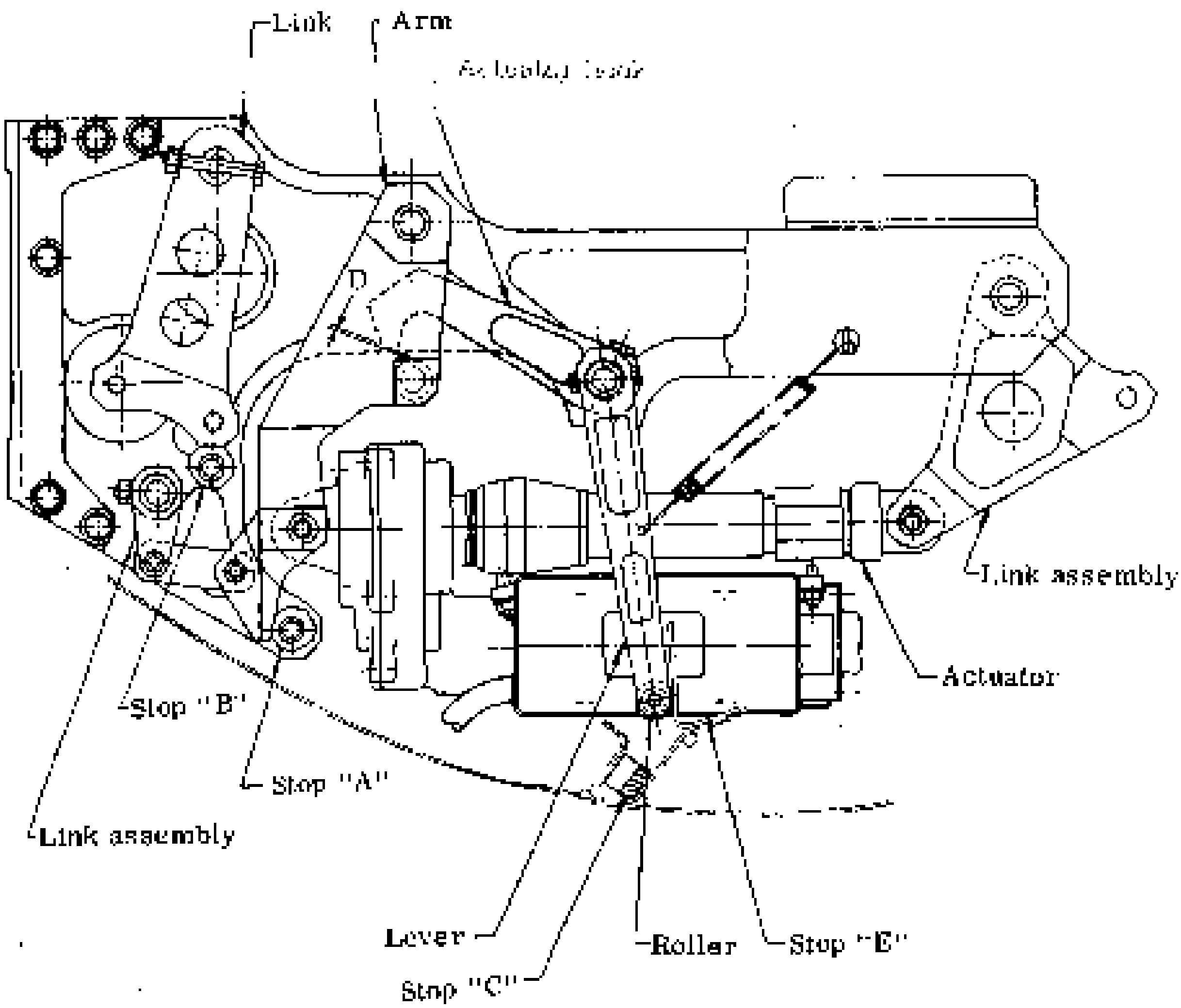


Fig. 4-34 Adjustment of R.H. stop

(6) Adjust stop "F" so that link may strike stop "F" when actuator comes to a length of 16.63 ± 0.08 in. (422.4 ± 2 mm). (See Fig 4-35).

(7) Adjustment of door open stop in emergency gear extension (See Fig 4-35).

(a) **When the actuator comes to a length of 16.34 in. (415 mm), adjust bumper "G" by means of bolt thread so that the clearance between attaching fitting and bumper may be 0 ~ 0.04 in. (0 ~ 1 mm).**

(8) Check main landing gear for interference.

(a) Check for interference between door mechanism and airframe structure when adjusting in Para. (6).

(b) Check for interference between door attaching point and door when adjusting in Para. (6).

(9) Adjust forward door open limit switch by means of nut, so that the stroke of L.H. limit switch may be 0.08 to 0.10 in. (2 to 2.5 mm), when actuator length is 16.34 in. (415 mm) (See Fig 4-68).

(10) Adjust limit switch so that the stroke of the switch may be 0.16 ~ 0.20 in. (4 ~ 5 mm) when the door strikes the stop.

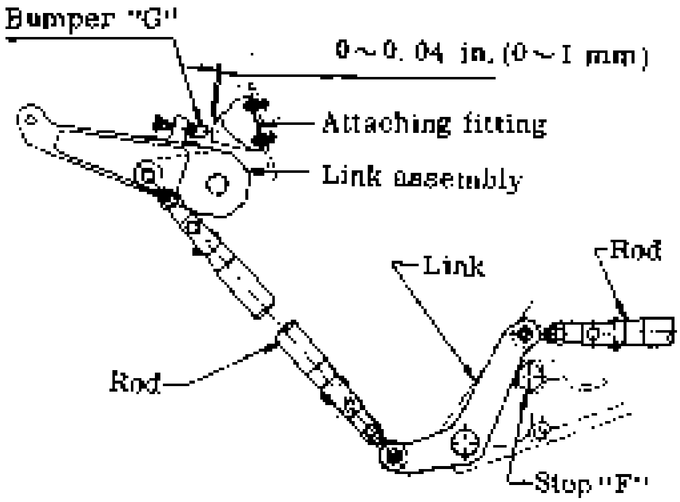


Fig. 4-35 Adjustment of stop

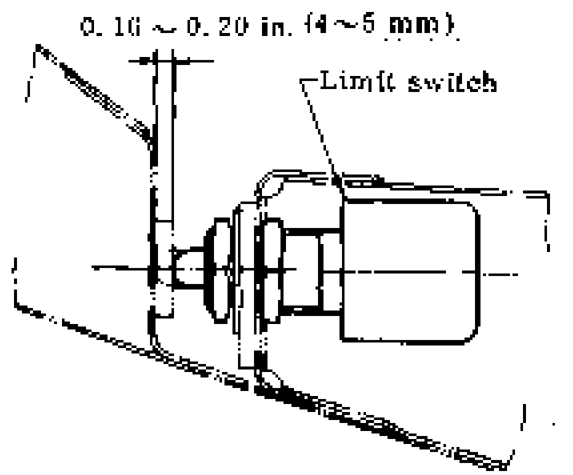


Fig. 4-36 Adjustment of limit switch

3.9.2.3 ADJUSTMENT OF MAIN LANDING GEAR DOOR LOCK MECHANISM (See Figs 4-37 and 4-38)

The door lock mechanism consists of forward door lock mechanism and aft door lock mechanism. Each lock is released before door operation.

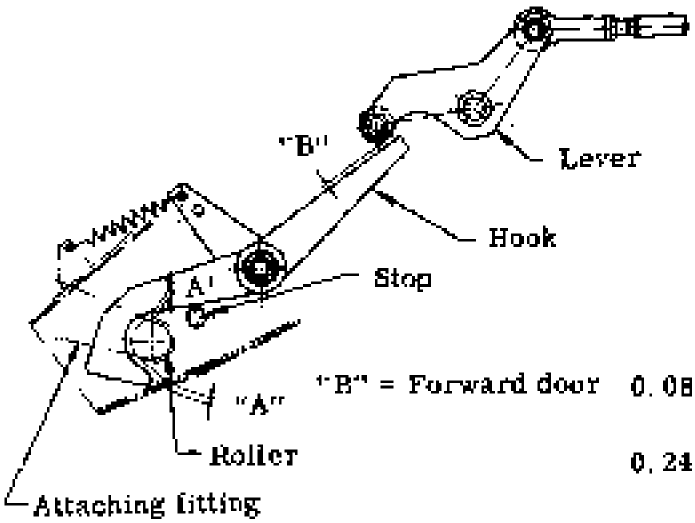
(1) Adjustment of door side roller
Under the conditions that the doors strike the door stops and the hook comes in contact with stop, adjust the attaching fitting by means of shim and serration under the fitting so that the clearance "A" between hook and roller may be 0.08 ~ 0.12 in. (2 ~ 3 mm) (both forward & aft doors) and "A'" may be 0.08 ~ 0.12 in. (2 ~ 3 mm) (forward door) and 0.12 ~ 0.20 in. (3 ~ 5 mm) (aft door).

(2) Adjustment of rod

(a) Forward door

(i) Adjust the clearance between L.H. hook and lever to 0.08 ± 0.02 in. (2 ± 0.5 mm) and the length of L.H. rod (L) to 11.02 $^{+0}_{-0.04}$ in. (280 $^{+0}_{-1}$ mm).

(ii) Adjust the clearance "B" between R.H. hook and lever to 0.24 ± 0.02 in. (6 ± 0.5 mm) and connect R.H. rod.

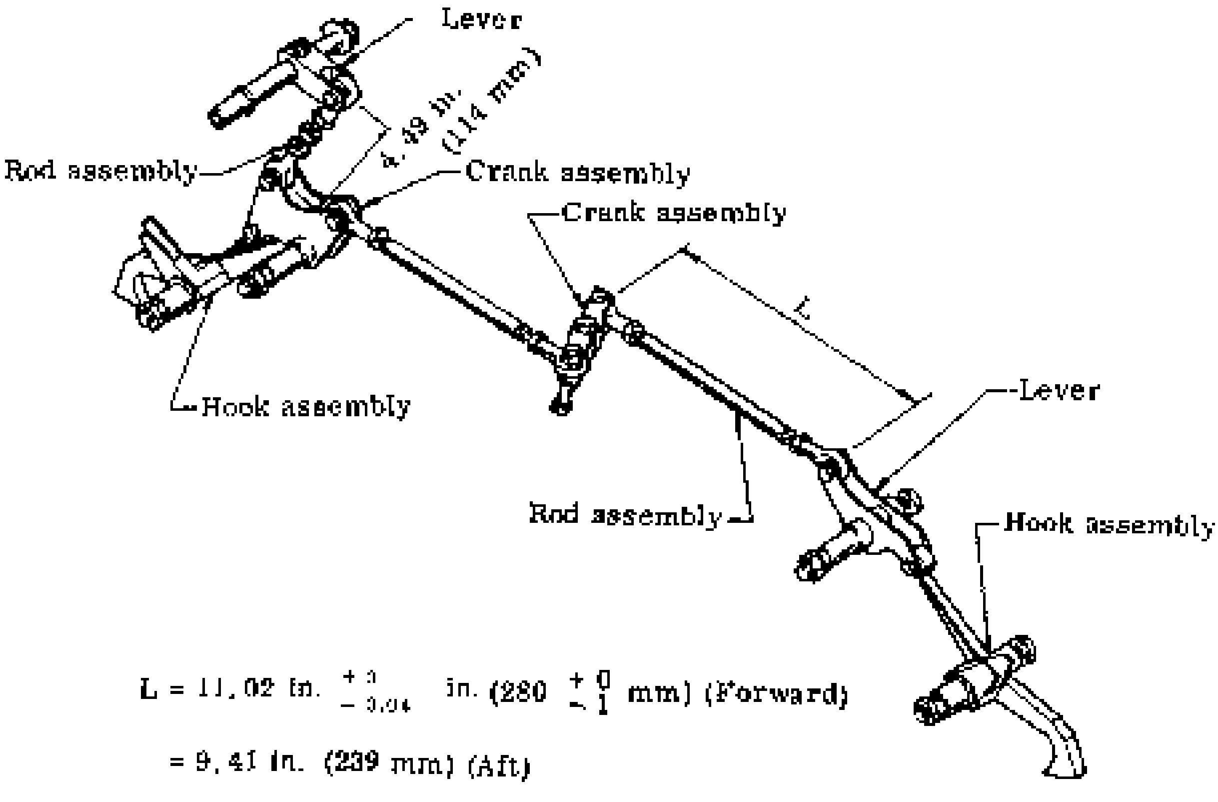


"B" = Forward door 0.08 ± 0.02 in. (2 ± 0.5 mm) (L.H.)

0.24 ± 0.02 in. (6 ± 0.5 mm) (R.H.)

Aft door 0

Fig. 4-37 Adjustment of lock mechanism



$L = 11.02$ in. $^{+0}_{-0.04}$ in. (280 $^{+0}_{-1}$ mm) (Forward)

= 9.41 in. (239 mm) (Aft)

Fig. 4-38 Main landing gear door and lever

(b) Aft door

(i) Set the clearance "B" between hook and lever to zero (L. H. and R. H.).

(ii) Set L. H. rod length to 9.41 in. (239 mm) and adjust R. H. rod.

(iii) Set short rod in R. H. to 4.49 in. (114 mm) and connect.

(iv) Connect torque tube to lever shaft with a bolt.

(3) Connection of the forward and aft door lock mechanisms and the forward door mechanism shall be made by bolts under the conditions that the forward doors are closed and the hook strikes the stop.

3.9.2.4 OPERATIONAL CHECK OF MAIN LANDING GEAR FORWARD DOOR MECHANISM

(1) In accordance with the following steps, ascertain that left and right doors close together.

(a) Close circuit breaker "LG DOOR MOTOR" and "LG CONT".

(b) Open or close the forward doors by means of ground door open switch installed on the bulge rear wall, R. H. side.

(c) Adjustment is made by adjusting the length of rod in Fig. 4-32.

NOTE

With the aircraft in jacked position, the doors open when the circuit breaker "LG CONT" is closed after the circuit breaker "LG DOOR MOTOR" is closed and the landing gear control switch is turned to "UP".
The doors close when the circuit breaker is closed after the control switch is turned to "DOWN".

(2) In accordance with the following steps, ascertain that left door opens first.

(a) Open forward doors little by little. Close circuit breaker "LG DOOR MOTOR", set ground door open switch to "OPEN" and then, turn circuit breaker "LG CONT" intermittently to close.

(b) Make sure that left door opens a moment ahead of right door, releasing hook.

(3) Check for actuator shaft length. Close the forward doors little by little without inertia. When the door pad strikes the close limit switch plunger, the length of the actuator shaft shall be 12.32 to 12.44 in. (313 to 316 mm) (See Fig 4-36). When the length is not within this limit, adjust per Para. (1) (c) above after changing the rod length in Fig 4-39.

maintenance manual

TEMPORARY REVISION NO.4-2

This Temporary Revision No. 4-2 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	4-46
MU-2B-60	MR-0336	4-46

Insert facing the page indicated above for the applicable Maintenance Manual

Retain this Temporary Revision until such time as a permanent revision on this subject is issued.

REASON : To change the installation procedure for the rod assembly, and add a check of the safety pin rod.

CHANGE : Change Fig 4-39 and add the paragraph 3.9.3A and Fig 4-40A as follows.

3.9.3A Check of SAFETY PIN ROD

(1) Apply electrical power to the airplane and open main landing gear forward door using the ground door open switch.

(2) Pull the circuit breakers "LG CONT" and "LG DOOR MOTOR".

(3) Remove the bolts, nuts and washers from the both ends of the safety pin rod assembly in the main landing gear door mechanism and disconnect the safety pin rod assembly.

NOTE

After removing the safety pin rod assembly the doors are free to move. Prop the doors open for clearance purposes

(4) Remove the cotter pin from the safety pin and remove the safety pin and washer from the rod assembly. Discard the cotter pin. Check the safety pin and the rods for anomalies as follows:

(a) The safety pin should be constructed of aluminum.

(b) Upper and lower rods should slide freely and have no sign of corrosion at their mating (sliding) surfaces.

(c) Inspect the safety pin for corrosion, wear, or other surface damage

NOTE

Ensure that the grease applied to the joint area fully covers the mating surfaces.

(5) Clean and lubricate the mating surfaces of jointed area between upper and lower rods with MIL-PRF-23827 grease.

(6) Reassemble the safety pin rod assembly, install the safety pin and washer and secure with new cotter pin per Figure 4-40A.

CAUTION

Positioning the large diameter rod in the lower position (upside down) may result in corrosion in the joint area of the rods and may result in the failure of the normal operation of the safety pin. This could bind and/or damage the landing gear doors and the door mechanism.

(7) Reconnect the safety pin rod assembly making sure that the large diameter rod (Outer Rod) is placed in the upper position as shown in Figure 4-40A.

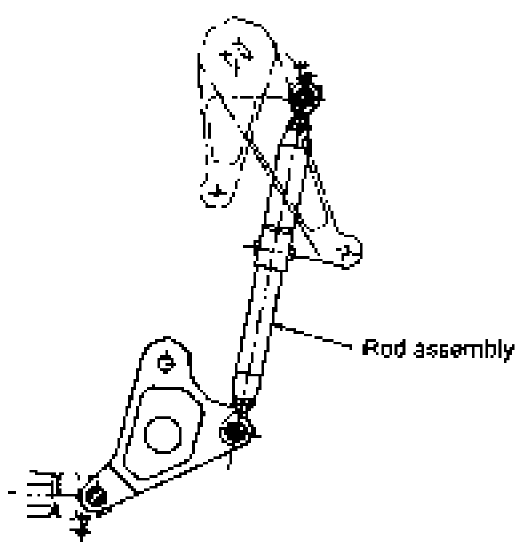


Fig. 4-39

TEMPORARY REVISION NO.4-2

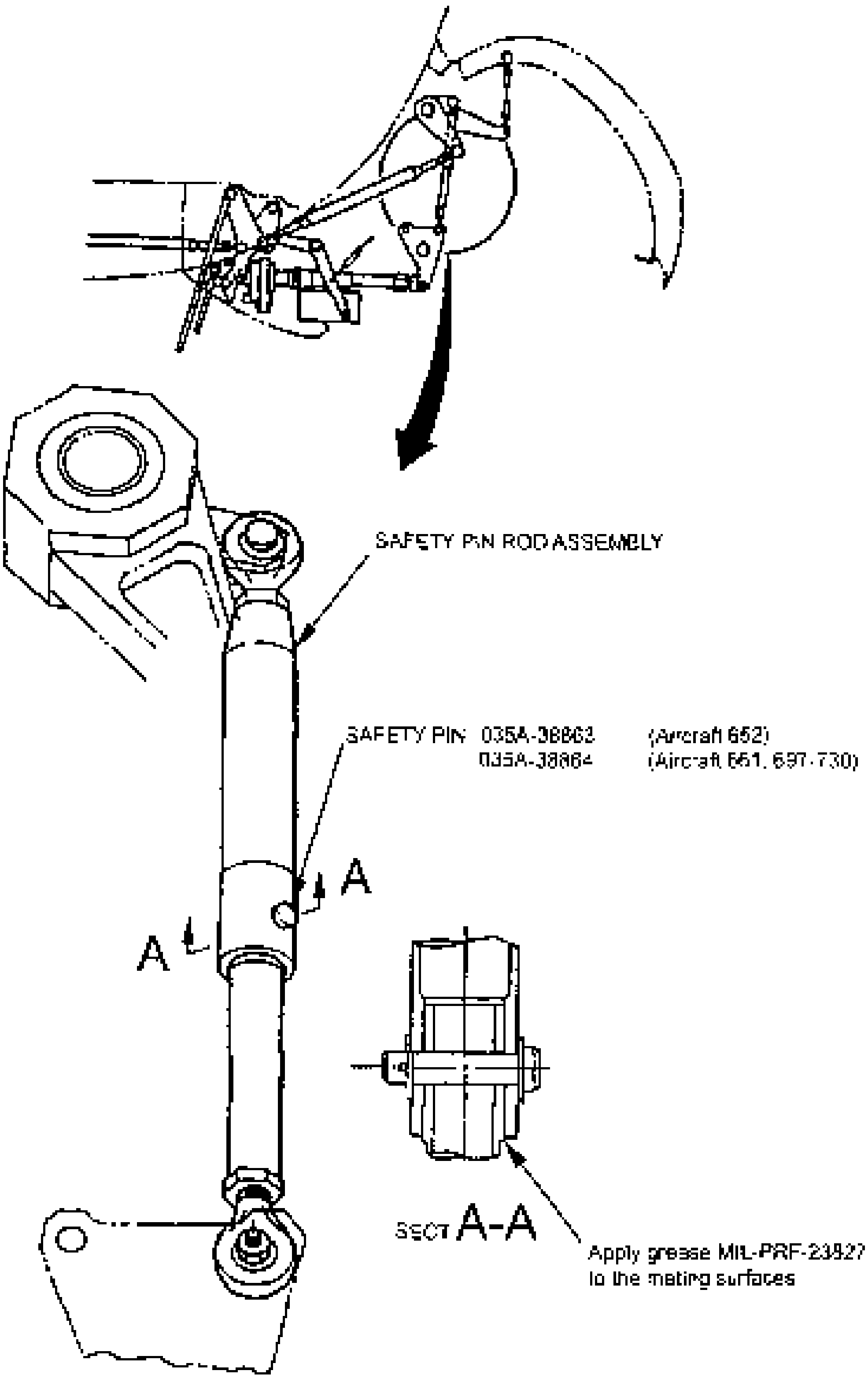


Fig 4-40A Installation of safety pin rod assembly

3.9.3 MAIN LANDING GEAR FORWARD DOOR CLUTCH ASSEMBLY
(See Fig 4-40)

The clutch assembly is provided one set each in L.H. and R.H. The clutch assembly is a coupling having 30 triangle teeth on the side of link and transmits torque by engagements of teeth. When the emergency gear down handle is pulled, the hook is released from the pulley, which then turns on the triple thread screw shaft to escape, and the clutch is disengaged by air force on the doors or pushing force of main tires.

In this case, the forward doors are disconnected from the door mechanism and in free conditions which can not be operated by electric actuator.

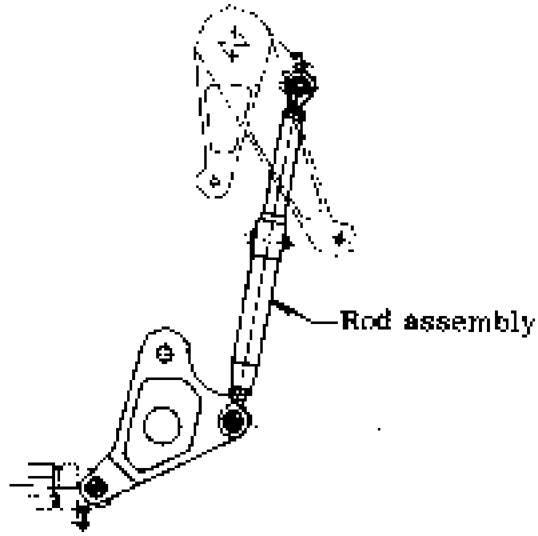
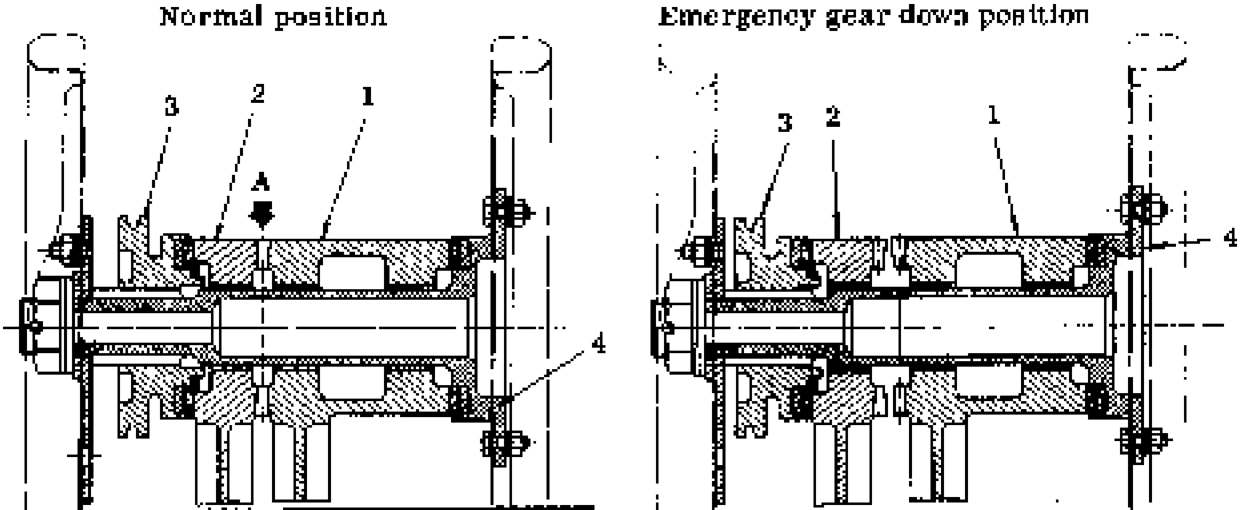


Fig. 4-39

Normal position

Emergency gear down position



1. Link
2. Link
3. Pulley
4. Shaft

VIEW - A

Fig. 4-40 Clutch assembly

3.9.4 ADJUSTMENT OF EMERGENCY LANDING GEAR DOOR RELEASE MECHANISM

Emergency landing gear door release mechanism is adjusted by cable tension.

3.9.4.1 ADJUSTMENT OF CABLE IN KEEL

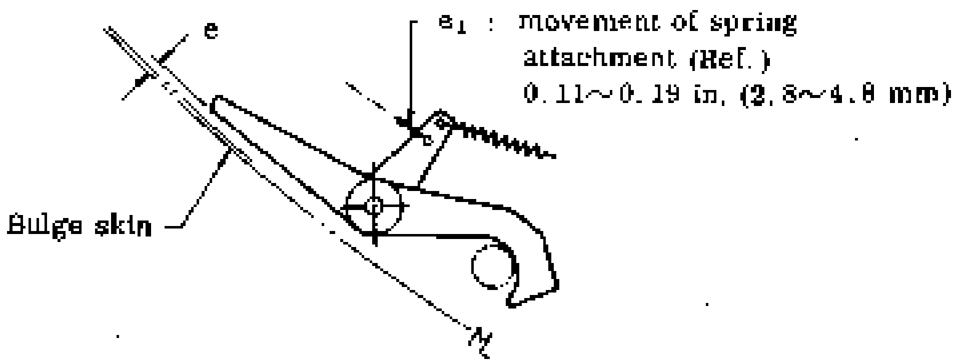
Under the condition that the small lever is set in emergency gear down handle and the center lever is fixed with a rig pin adjust cable tension by turning the turnbuckle. The maximum cable tension is 12 lbs.(5.4 kgs).

3.9.4.2 ADJUSTMENT OF CLUTCH RELEASE CABLE

Adjust the cable with the clutch retaining hook engaged.

3.9.4.3 ADJUSTMENT OF HOOK RELEASE CABLE

Adjust cable detaining fitting so that the clearance between each hook and outer skin (e) may be 0.20 to 0.32 in. (5 to 8 mm) when the small lever is about to come off the emergency gear extension handle after the handle is pulled up.



4. LANDING GEAR RETRACTING MECHANISM

4.1 GENERAL DESCRIPTION (See Fig 4-41)

The landing gear retracting mechanism consists of motor, main reduction gearbox, torque limiter, mechanical stop, emergency gear extension mechanism, gearbox and torque shafts.
When the wheel shaped switch on the switch panel is operated, the landing gear retracting mechanism actuates electrically together with main gear forward doors. One motor turns screw jacks of nose and main gear and retracts or extends the landing gears. The motor is provided with an electromagnetic clutch in which 10 centrifugal balls and brake mechanism are installed in order to improve disengagement characteristic and prompt stop function.
Motor speeds are reduced to approximately one-sixteenth through the two-stage spur gears in the main reduction gearbox. The torque limiter is provided for the prevention of overload upon the mechanism and slips when torque more than 380 to 560 in-lbs (449 to 560 kg/cm) has been transmitted.
The torque is transmitted by ball joint, gearbox and drag strut support, 9 torque shafts, bearing box, chain and sprocket, nose gear bevel gearbox and nose gear actuator. The mechanical stop and emergency gear down mechanism are provided between torque shafts. The gearbox and drag strut support have 2 sets of bevel gears and reduction ratio to jack screw is 35:43.

The reduction ratio of chain assembly is 15:30 and nose gear actuator 15:42. The bulkhead where the torque shaft goes through is sealed to prevent air leakage. The mechanical stop opens or closes limit switches by means of moving nut and adjusts operational times of landing gear motor and forward door motor, and furthermore, stops the retracting mechanism to prevent overtravel in the event of limit switch trouble.

The emergency gear extension mechanism is to extend landing gear manually in the event of failure in the motor system. The emergency gear down handle is linked mechanically to the main gear emergency unlock mechanism.

The main gear jack screw also serves as drag strut, but nose gear is provided with jack screw, down lock and drag strut.

4.2 NOSE GEAR ACTUATOR

4.2.1 REMOVAL AND INSTALLATION

(1) In advance of removing nose gear actuator, extend landing gear slowly with no load and no inertia. (hand operating shaft at the back end of gearbox and drag strut support may be used, see Fig 4-41).

When the nose gear down lock indicating light (green) illuminates, measure the following exposed thread lengths by means of slide calipers and record the dimension.

(a) Exposed thread length of main gear jack screw (see Fig 4-42).

(b) Exposed thread length of nose gear jack screw (see Fig 4-44).

(2) Remove attaching bolt of jack screw on rod end side per Para 2.2.

1. Landing gear motor
2. Torque limiter
3. Main reduction gearbox
4. Ball joint shaft
5. Gearbox & drag strut support
6. Main gear jack screw
7. Torque shaft
8. Hand operating shaft-adjusting nose & main gear phase difference
9. Bearing box
10. Bearing box with seal
11. Stop assembly
12. Emergency gear down mechanism
13. Emergency gear down handle
14. Lower sprocket assembly
15. Chain
16. Idler sprocket assembly
17. Upper sprocket assembly
18. Bevel gearbox assembly
19. Nose gear actuator
20. Nose gear jack screw

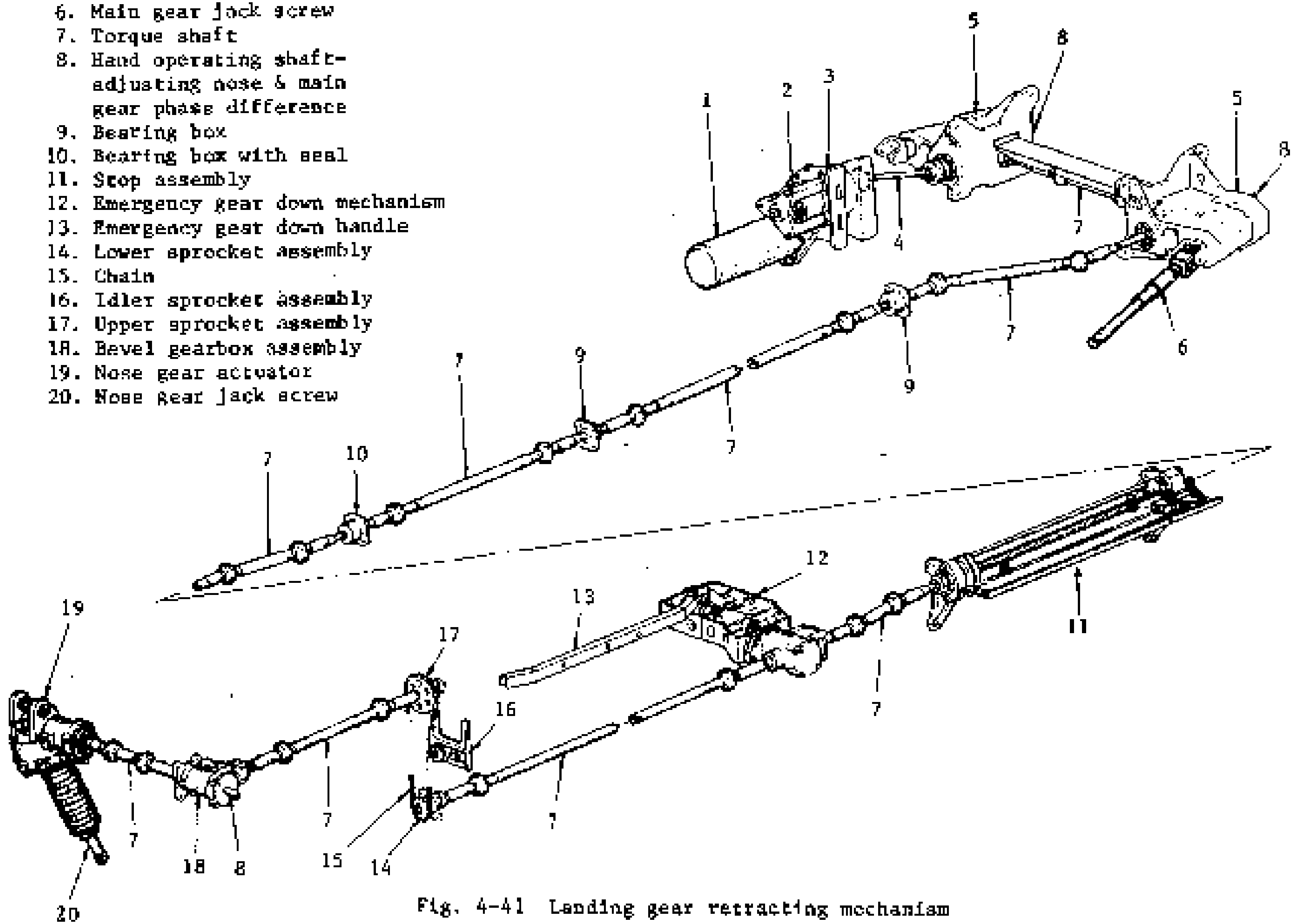


Fig. 4-41 Landing gear retracting mechanism

NOTE

When nose gear actuator or torque shaft is installed, reinstall on the same conditions (length and location) as former installation. After reinstallation, check for the specified value by stop mechanism.

(3) Reinstall in reverse sequence of removal.

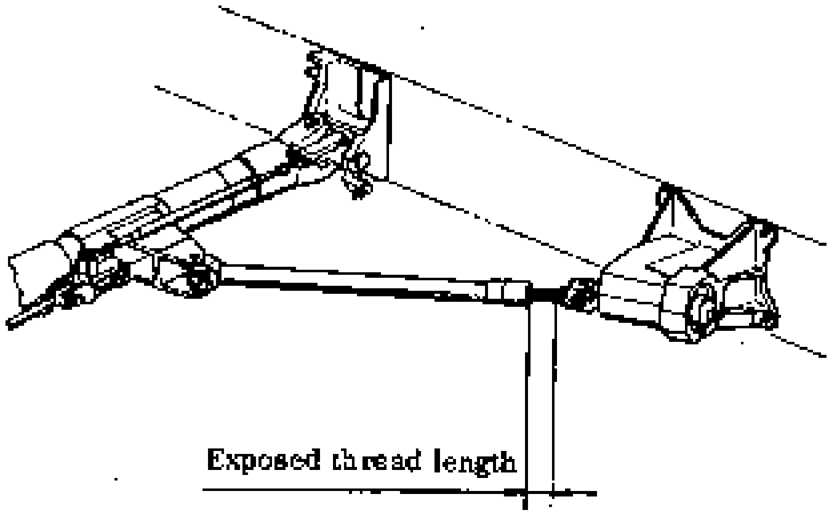


Fig. 4-42 Main gear jack screw

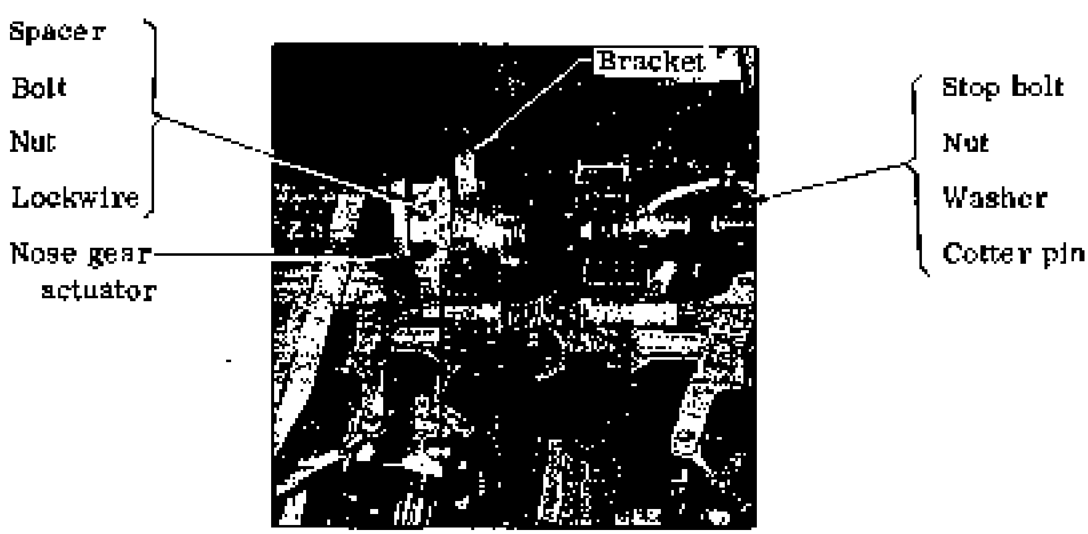


Fig. 4-43 Nose gear actuator

ORIGINAL
As Received By
ATP

4.2.2 ADJUSTMENT AFTER REINSTALLATION OF NOSE GEAR ACTUATOR

(1) Install nose gear actuator after setting the exposed thread lengths of nose and main gear jack screws to the lengths recorded in accordance with Para. 4.2.1 (1), (a) and (b).

(2) After installation, extend gear with no load and no inertia and check nose and main gear jack screws for exposed thread lengths.

(3) Perform gear retraction and extension. Make sure that "DOWN" side and "UP" side clearances of mechanical stop (clearance between moving nut and edge of thrust bearing) are the specified values, and that relative position of nose and main gear is correct, in accordance with Para. 4.7 "ADJUSTMENT OF STOP MECHANISM".

4.2.3 CHECK OF NOSE GEAR ACTUATOR

(1) Remove the dust cover and check jack screw for deformation, damage and foreign materials.

(2) Check for axial play over the entire stroke. If the play due to thread wear exceeds 0.014 in. (0.36 mm), replace screw jack.

(3) Remove rod and clean screw. Check screw surface for lubricant film (Molykote). Replace screw jack if the lubricant film is peeled off the bottom of screw thread over the entire length. Apply Molykote grease MIL-G-21164 when the lubricant film is not peeled off.

4.3 DUST COVER

4.3.1 REMOVAL AND INSTALLATION (See Fig 4-44)

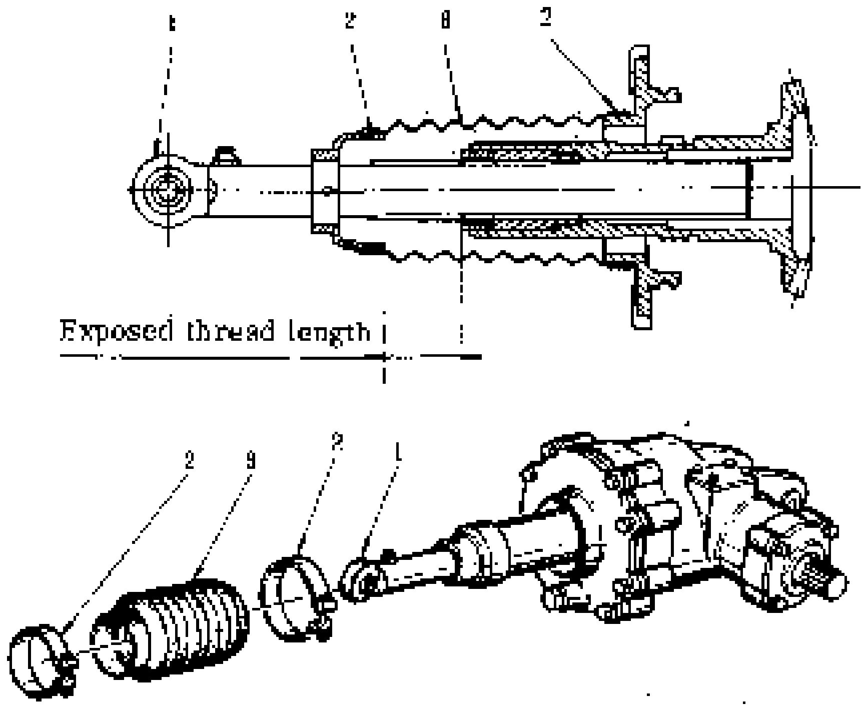
(1) Remove jack screw attaching bolt (rod end side) in accordance with Para. 2.2.

(2) Remove clamps and dust cover.

(3) Reinstall in reverse sequence of removal.

NOTE

Check jack screw and dust cover for cleanliness and no foreign materials such as metal chips, etc.



1. Rod end
2. Clamp
3. Dust cover

Fig. 4-44 Dust cover

4.4 MOTOR AND TORQUE LIMITER

4.4.1 REMOVAL AND INSTALLATION (See Fig 4-45)

(1) Remove motor cover.
(2) Drain oil through 2 drain ports on the reduction gear box.
(3) Remove motor attaching bolts (4 ea.) and remove motor.
(4) Remove torque limiter case attaching bolts (3 ea.), tab washer and nut on the top of the torque limiter shaft and remove torque limiter from shaft.

NOTE

Check the friction surface of the torque limiter for cleanliness, especially for any oil penetration which may exist.

(5) Reinstall in reverse sequence of removal.
(6) Fill with oil MIL-L-6086 Grade L until oil overflows the upper port.

4.5 BALL JOINT ASSEMBLY AND TORQUE SHAFT ASSEMBLY

4.5.1 REMOVAL AND INSTALLATION (Fig 4-41)

(1) Extend landing gear and set the forward doors in open position.

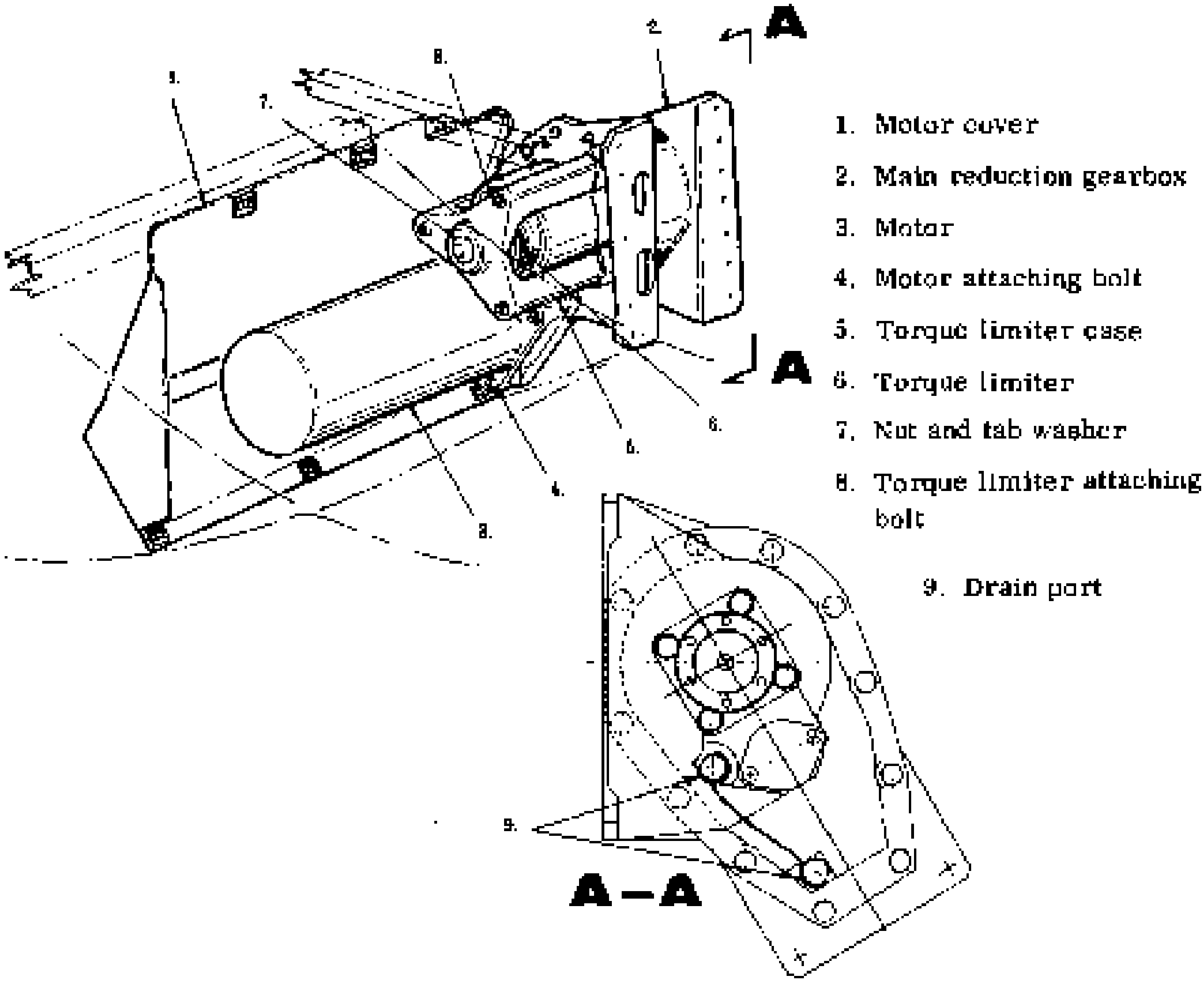


Fig. 4-45 Motor and torque limiter

(2) Remove access panels and cabin floor necessary to remove and install torque shafts.

(3) Ball joint assembly.

(a) Remove connecting bolts and nuts between ball joint and main reduction gearbox and gearbox & drag strut support after cutting off lock wire.

(b) Pull out ball joint from spline shaft on the reduction gearbox.

(c) Remove ball joint together with spline shaft on the gearbox & drag strut support.

(d) Peel off packing between ball joint and gearbox with care not to damage.

(e) Reinstall in reverse sequence of removal.

(4) Torque shaft assembly

(a) Remove stop bolts, cotter pins, nuts and washers in Fig 4-41.
(b) Slide the shafts, and remove the other ends from the spline.
(c) Slide the shafts to the reverse direction and remove.
(d) The torque shaft assembly of main gear forward door mechanism, in Fig 4-41 can not be slid, so remove connecting bolts, then pull out the shaft.
(e) As the rubber boots of universal joints are easily broken, remove with care.
(f) Reinstall in reverse sequence of removal.

NOTE

Make sure that stop bolt, nut, washer and cotter pin are properly installed.

(5) Inspection

(a) Wipe off grease and check spline for cracks.
(b) Pull out connecting bolts and check bolt holes for looseness. When the looseness exceeds 0.008 in. (0.2 mm), replace with oversize bolt or new shaft assembly.
(c) Check universal jount for excessive play. If the play exceeds 1.5° replace with new shaft assembly.

(6) Apply grease MIL-G-23827 on splines when installing.

(7) After installation, perform operational test in accordance with Para. 5.1 and adjust relative positions to nose gear, mechanical stop and main gear.

4.5.2 CHECK OF TORQUE SHAFT SYSTEM

To check for play over the entire system, turn by hand the hand operating shaft on the rear end of the RH gearbox and drag strut support or RH barrel slowly and measure revolution angle of the shaft or barrel until nose gear jack screw barrel begins to turn. When the angle exceeds 49°, check per Para 4.5.1.

4.6 CHAIN ASSEMBLY (See Fig 4-46)

Keep chain clean and free of dirt, rust and corrosion; for inspection and wear limitations see Chapter II, Para. 11. Adjust chain tension by moving the idler sprocket shaft to the left or right. Allow approximately 0.188 to 0.313 in. (5 to 8 mm) slack as measured with the thumb and the forefinger compressing the middle of the chain span. The proper amount of slack is 2% to 3% of the span.

4.6.1 REMOVAL AND INSTALLATION

(1) Remove access panel.

(2) Move idler sprocket and loosen chain.

(3) Remove chain clip and remove chain from upper and lower sprockets.

(4) Reinstall in reverse sequence of removal.

NOTE

When new chain is installed, check chain tension after operation and adjust chain tension again if necessary.

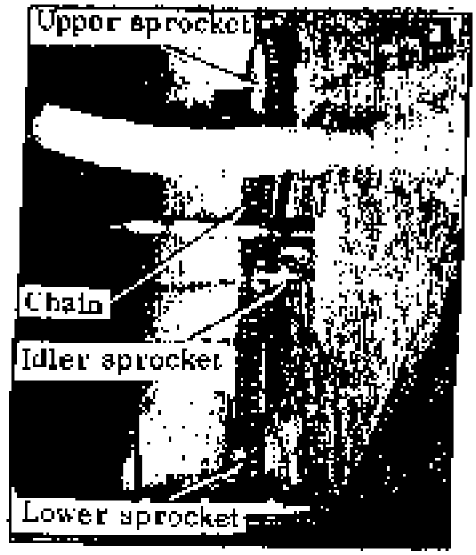


Fig. 4-46 Chain assembly

4.7 *ADJUSTMENT OF MECHANICAL STOP* (See *Fig 4-47*)

4.7.1 CONNECTION OF TORQUE SHAFT

(1) Connect main gear, nose gear and mechanical stop under the following conditions.

(a) Main gear : Exposed thread length of jack screw (drag strut) 17.62 in ± 0.04 in. (448±1 mm)

(b) Nose gear : Center of the nose wheel is at the position 18.07 ± 0.12 in. (459±3 mm) from the ceiling of nose wheel.

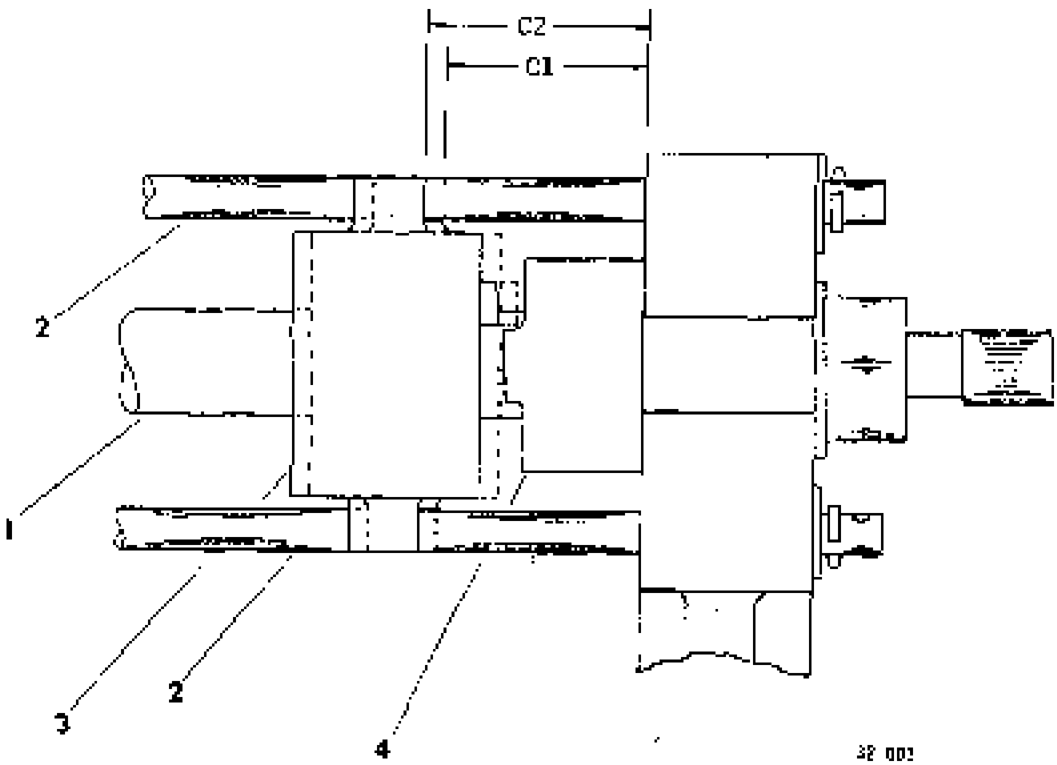
(c) Mechanical stop : Clearance B = 0

To connect torque shafts between mechanical stop and main gear, the torque shaft just before L.H. gearbox & drag strut support is convenient for adjustment.
To connect torque shafts between mechanical stop and nose gear, the torque shaft between bevel gearbox and chain may be used.
To operate main gear or nose gear separately, disconnect torque shafts as mentioned above and turn the hand operating shafts on the rear end of gearbox & drag strut support for main gear and on the left end of bevel gearbox for nose gear.

(2) Extend landing gear and set under the following conditions.

(a) Main gear : Exposed thread length of jack screw 0.69 in ± 0.04 in. (17.5±1 mm)

(b) Nose gear : R.H. - More than 0.08 in. (2 mm) clearance to reach downlock groove end.
L.H. - More clearance than R.H. side



1. MAIN SHAFT
2. GUIDE SHAFT
3. TRAVELING NUT ASSEMBLY
4. STOP NUT

Fig. 4-47 Mechanical stop (1/2)

(c) Adjust down side double nuts and spacer of mechanical stop so that the clearance "A" may be zero. (decision of maximum stroke)

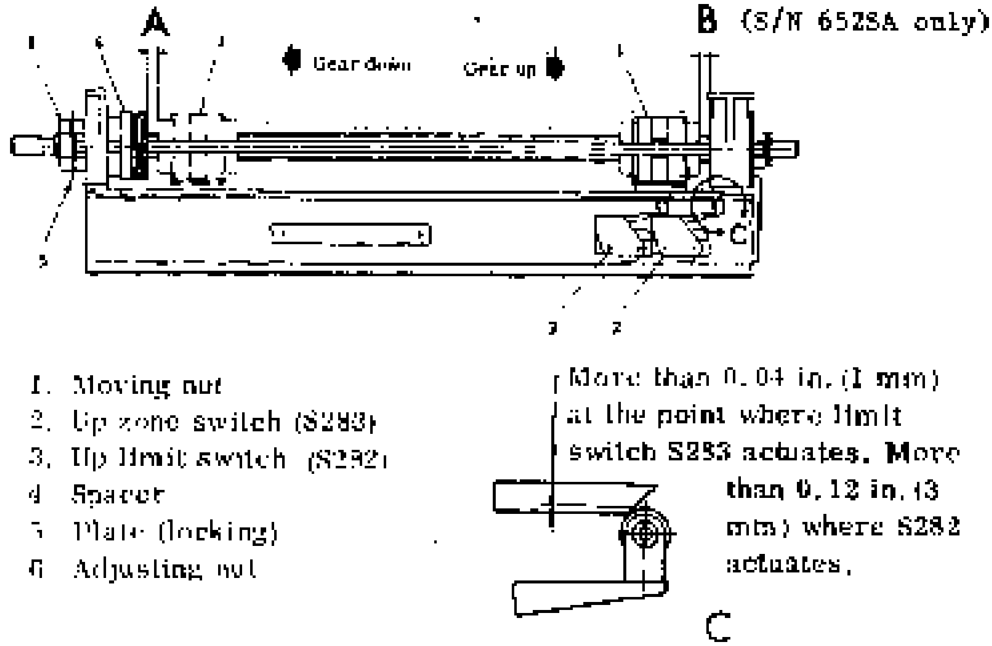


Fig. 4-47 Mechanical stop (2/2)

4.7.2 ADJUSTMENT OF MECHANICAL STOP LIMIT SWITCH

(1) Applicable to aircraft S/N 652SA

Retract landing gear slowly and continuously under no load and no inertia condition to UP position. Check for the following clearances:

(a) Adjust the "UP LIMIT" switch (S282) to actuate when clearance "B" is 0.16 to 0.20 in. (4.0 to 5.0 mm).
(b) Adjust the "UP ZONE" switch (S283) to actuate when clearance "B" is 0.14 to 0.16 in. (3.5 to 4.0 mm).

Applicable to aircraft S/N 661SA, 697SA and subsequent

Retract landing gear slowly and continuously under no load and no inertia condition to UP position. Check for the following clearances and adjust as required.

(a) Continue retraction slowly until mechanical stop reaches the end of its travel (zero clearance). Measure and record for clearance of "C1".
(b) Extend the landing gear by hand past the point where the "UP LIMIT" switch is in the OFF position.
(c) Retract the landing gear by hand until the "UP LIMIT" switch is in the ON position, measure and record clearance "C2".
(d) Adjust the "UP LIMIT" siwtch so that C2 - C1 = 0.16 to 0.20 in. (4.0 to 5.0 mm).
(e) Continue to retract the landing gear by hand until the "UP ZONE" switch is in the ON position, measure and record clearance "C3".
(f) Adjust the "UP ZONE" switch so that C3 - C1 = 0.14 to 0.16 in. (3.5 to 4.0 mm).

4.7.3 OPERATIONAL TEST OF MECHANICAL STOP (See Table 4-1)

Perform landing gear retraction and extension, and make sure of the specified values in Table 4-1. For operation with no load and inertia, check after ten operational cycles.

maintenance manual

TEMPORARY REVISION NO.4-3

This Temporary Revision No. 4-3 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	4-60
MU-2B-60	MR-0336	4-46

Insert facing the page indicated above for the applicable Maintenance Manual.
Retain this Temporary Revision until such time as a permanent revision on this subject is issued

REASON : To add a torque check of MLG thrust gearbox attachment bolt.

ADD : Paragraph 4.8 and Fig 4-47A as follows.

4.8 TORQUE CHECK OF MLG THRUST GEARBOX ATTACHMENT BOLT

Inspect to ensure the torque is correct for all four (4) MLG Thrust Gearbox attachment bolts per the Figure 4-47A to preclude a loose bolt condition. If a bolt(s) is loose, replace the nut(s) and retorque to proper value as required.

TEMPORARY REVISION NO.4-3

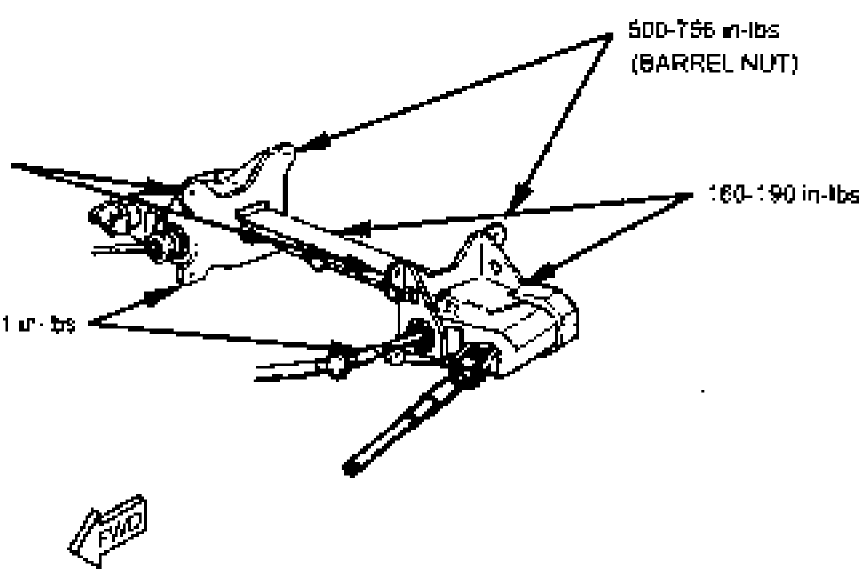


Fig 4-47A Torque value for main landing gear gearbox attachement bolt

Operation and switch			Clearance of Mechanical stop: L/G down (A)	Clearance of Mechanical stop: L/G up (B)	Checking and adjusting point
L/G Up	No inertia	Connection of torque shaft		0	Exposed thread length of main gear jack screw 17.64 ± 0.04 in. (448±1 mm) Center of nose wheel WP-18.07±0.12 in(459±3mm)
L/G Up	No inertia	Up zone switch		014~016in. (3.5~4mm)	Switch attaching screw
L/G Up	No inertia	Up limit switch		016~020in. (4~5 mm)	Switch attaching screw
L/G Up	No load	Main gear up indicating switch		Less than 032in/(8mm)	See Para. 10
L/G Up	No load	Nose gear up indicating switch		024~1.2in. (6~30mm)	See Para. 10
L/G Up		No load with inertia		002~008in (0.5~2mm) No shock	Actuating point of up limit switch. Disconnecting characteristic of motor.
L/G Down	No inertia		0		Exposed thread length of main gear jack screw 0.67 ± 0.04 in. (17 ± 1 mm) More than 0.08 in. (2 mm) clearance to reach nose gear downlock groove end (R.H.) More clearance than R.H. (L.H) Spacer thickness, Adjusting nut
L/G Down	No load	Down limit switch	$0.16^{+0.08}_{-0.04}$ in (4^{+2}_{-1} mm)		See Para. 10
L/G Down		No load with inertia	002~008in. (0.5~2mm) No shock		Actuating point of down limit switch. Disconnecting characteristic of motor

Table 4-1 Specified values for mechanical stop

NOTE: Please see the TEMPORARY REVISION that revises this page.

5. OPERATION AND CHECK FOR LANDING GEAR SYSTEM

5.1 OPERATIONAL TEST

Operational test (operational condition with no load and with inertia) is performed as described in the following procedures. If major components of landing gear system are replaced or adjustment is made by disconnecting power train of landing gear retracting mechanism, operational test should be performed as follows:

NOTE

(i) When landing gear is performed three times of up and down operations continuously, it is recommended to stop operation for about 20 minutes to cool gear motor and door motor.

(ii) When high current runs or any abnormal condition is observed during gear operation, stop operation immediately and inquire into the cause.

(iii) When airplane installed battery is used for gear operation, care should be taken not to drain battery capacity completely.

(1) Jack airplane.

(2) Close the following circuit breakers and turn battery key switch to ON.

"LG MOTOR"
"LG CONT"
"DOOR MOTOR"
"LG POS IND"

(3) Ensure that nose and main gear down lock indicating lights (green) are illuminated and the unsafe light is OFF.

(4) Place gear control switch to "UP" position and retract gears.

(5) Ensure that unsafe light (red) illuminates, and that as soon as it goes off, nose and main gear down lock indicating lights go off.

(6) Ensure that nose gear up limit switch actuates, gear retraction ceases and main gear aft doors are completely closed.

(7) With gears retracted, open main gear forward door and check for the following items.

(a) A minimum clearance of 0.79 in.(20 mm) between left and right main gear leg end and main gear well wall. A minimum clearance of 0.2 in.(5 mm) between the other points of the leg and main gear well wall.

(b) Clearance of 0.02 ~ 0.04 in. (0.5 ~ 1 mm) between main gear and leg attaching fitting stop.

(c) More than 0.4 in. (10 mm) clearance between tires and doors, and tire and main gear forward door, and more than 0.2 in. (5 mm) of clearance between tire and forward door cutout.

(d) More than 0.79 in. (20 mm) of clearance between nose tires and ceiling of nose gear wheel well.

(e) Exposed thread length of main gear drag strut jack screw is about 17.64 $^{+0.04}_{-0.14}$ in. (448 $^{+1}_{-3}$ mm).

(f) Up side clearance of mechanical stop is 0.02 ~ 0.08 in. (0.5 ~ 2 mm) (No shock against stop).

(g) Up limit switch, up zone switch and up zone safety switch should be at position operated by moving nut.

(h) Nose and main gear position indicating lights (green) should be off. Unsafe light (red) should be off when main gear aft doors are closed.

(8) Extend landing gear once and retract again. Stop retraction at approx. one fifth stroke of retraction and continue retraction manually. Check mechanical stop for clearance at the point where each limit switch actuates.

Switch	Clearance
Nose gear up indicating mechanism	0.24 ~ 0.32 in. (6.0 ~ 8.0 mm)
Up zone switch	0.14 ~ 0.16 in. (3.5 ~ 4.0 mm)
Up limit switch	0.16 ~ 0.20 in. (4.0 ~ 5.0 mm)

(9) Place gear control switch in DOWN position to extend gears. (If main gear aft door is open for inspection, close door before extending gears.)

(10) Ensure that unsafe light (red) illuminates and goes off, and that gear down lock indicating light (green) illuminates.

(11) With gears in "DOWN" position, check for the following.

(a) Interference in nose and main gear wells.

(b) Exposed thread length of main gear drag strut jack screw should be 0.69 $^{+0.08}_{-0.04}$ in. (17.4 $^{+2}_{-1}$ mm).

(c) Clearance at DOWN side of mechanical stop should be 0 to 0.04 in. (0 to 1 mm). (No shock against stop)

(d) Nose gear downlock (R. H.) has a clearance more than 0.08 in. (2 mm) to reach lock groove end and L. H. downlock has more clearance than R. H. side.

(e) Plunger of nose gear down limit switch does not reach the end of its travel and attaching bracket is not distorted.

(f) Nose and main gear position indicating lights (green) are illuminated.

(12) Retract landing gear once and extend again. Stop extension at approximately one fifth stroke of extension and continue extension manually. Check mechanical stop for clearance at the point where each limit switch actuates.

Switch	Clearance
Down limit switch	0.12 ~ 0.24 in. (3.0 ~ 6.0 mm)

(13) Time from start of movement of gear to end of movement both for retraction and extension shall be within 11 seconds.

(14) Landing gear position indicating light indicates correctly.

Gear down position	While gear retracting or extending (Door opening or closing)	Gear up position
Unsafe light(red) OFF	Unsafe light(red) ON	Unsafe light(Red) OFF
Nose light(green) ON	-	Nose light(green) OFF
L. H. light(green) ON	-	L. H. light(green) OFF
R. H. light(green) ON	-	R. H. light(green) OFF

5.2 OPERATION AND CHECK OF MAIN GEAR FORWARD DOOR

In regard to operation and check of main gear forward door, see Para. 3.9.2.3.

5.3 OPERATION AND CHECK OF NOSE GEAR DOOR

(1) Clearance between nose gear door and tires is more than 1.18 in. (30 mm) in door opening condition and 0.79 in. (20 mm) in door closing condition.

(2) When the moving nut of mechanical stop reaches the end of its stroke (gear up) and a weight of 20 lbs (9 kg) is hung on the door center edge, the door should come off mold line about 0.40 ~ 0.59 in. (10 ~ 15 mm).

5.4 MEASUREMENT OF CURRENT FOR MAIN GEAR DOOR MOTOR

Electric current of main gear door motor should be less than 14A during door operation.
When major components of main gear door mechanism are replaced, measure electric current as necessary.

5.4.1 PROCEDURE

(1) Disconnect wire G146B12 connected to door motor relay on the rear bulkhead of R.H. main gear wheel well, and connect it to the + terminal of ammeter (30A max.). Connect a wire (MIL-W-5086 Type II, AN-12) to - terminal of ammeter and connect the other end of the wire to the terminal of relay. *(See Fig 4-48)*

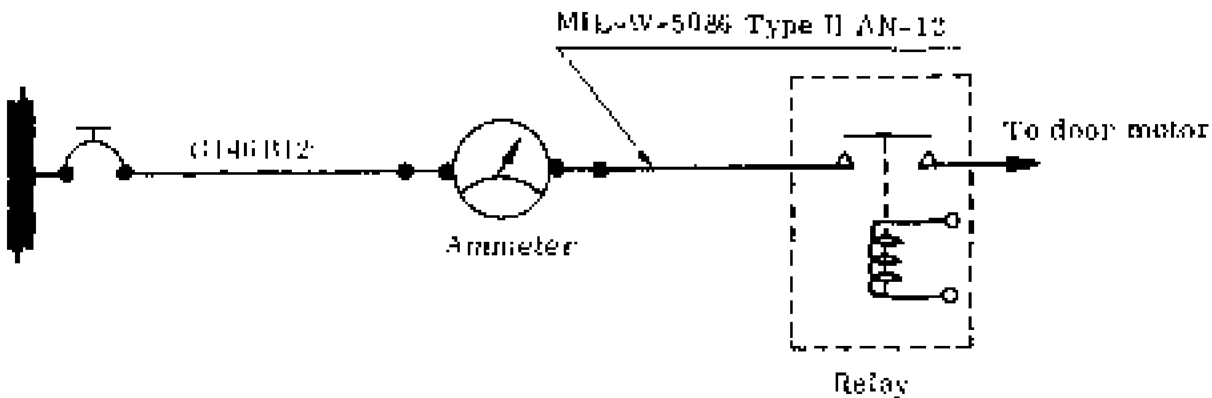


Fig. 4-48 Measurement of door motor current

CAUTION

Turn all power switches to "OFF" before connecting wire. Wire connecting to ammeter should not contact with main gear forward door.

(2) Operate main gear forward door and measure electric current of motor and record. It should be 7 ~ 9A (No load). This does not include current value at the moment of starting.

(3) After measurement, remove ammeter and wire and connect wire to its original position.

5.5 MEASUREMENT OF CURRENT FOR LANDING GEAR MOTOR

Electric current required for landing gear retracting mechanism should be less than 90A in both gear down or up operation.

When major components such as landing gear motor, gearbox & drag strut support and/or drag strut are replaced, measure electric current as necessary in accordance with the following procedures.

5.5.1 PROCEDURE

(1) Open access door at attaching point of landing gear motor relay, rear bulkhead in R.H. bulge.

(2) Disconnect wire G140A6 connected to relay terminal A_1 (or A_2) which is short-circuited at gear motor bus bar.

(3) Connect a wire (MIL-W-5086 Type II AN-8) 6 in. (152 mm) to the terminal A_1 (or A_2) and connect wire G140A6 as shown in Fig 4-49. (Shunt side . +, relay side: -)

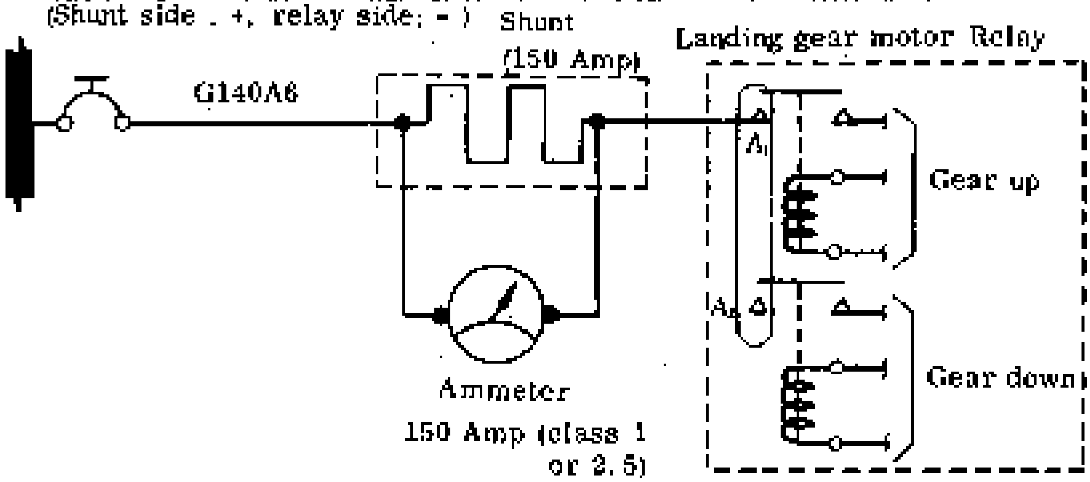


Fig. 4-49 Measurement of landing gear motor current

(4) Connect attached wire of regular length from shunt to ammeter terminal.

CAUTION

Turn all power source switches to "OFF" before connecting wire. Wire connecting to shunt and ammeter should not come in contact with airframe structure.

(5) Operate landing gear and measure electric current of motor. It should be less than 90A in both gear up or down operation. This does not include current value at the moment of starting.

(6) After measurement, remove ammeter, shunt and wire and connect wire to its original position.

6. EMERGENCY GEAR EXTENSION MECHANISM

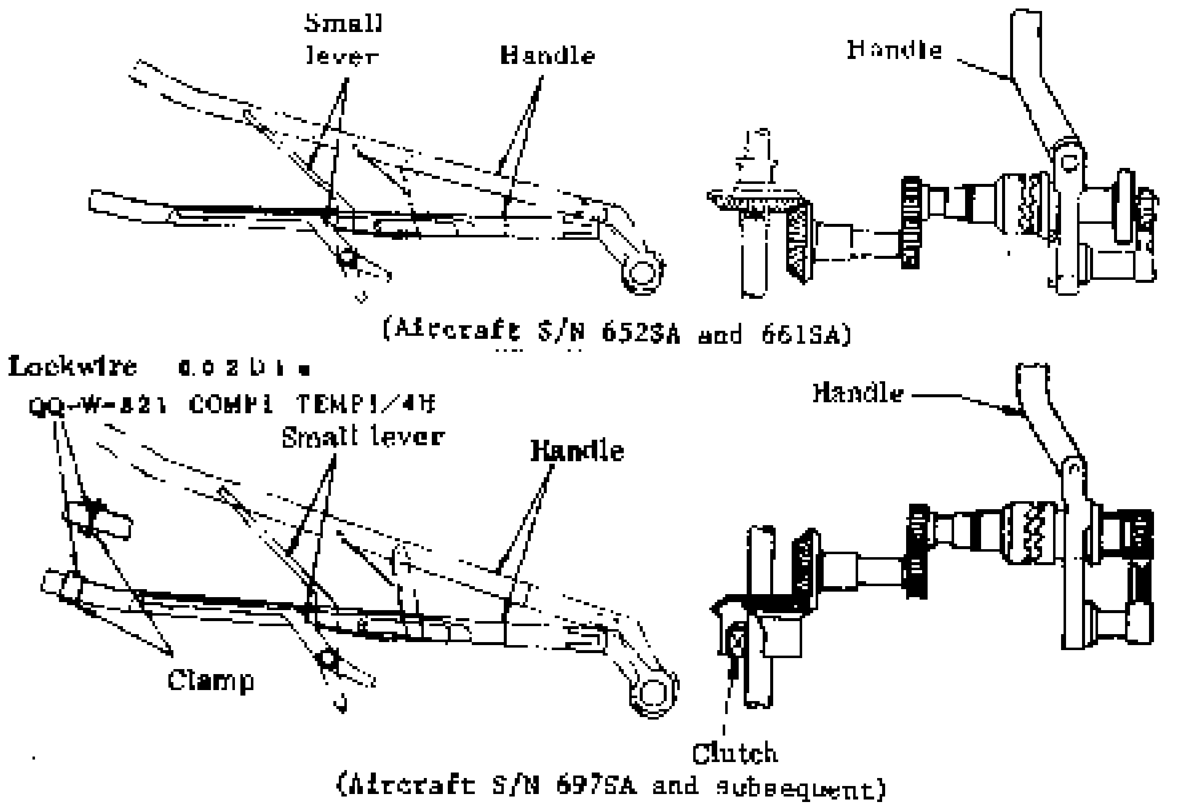
Fig. 4-50 Emergency gear down handle (A) and gears (B)

The emergency gear extension system extends the landing gear manually in the event of trouble in the electrical system. This mechanism, however, can not be used for retraction. To operate, pull up the handle on the floor of the pilot's compartment. A small lever in the handle will then actuate to disengage main gear forward and aft door locks and clutch of the main gear forward door mechanism. Doors are now free. Pump the lever approximately 215 times (*1) or approximately 130 times (*2), and the gear will extend until it reaches down-lock position. The completion of gear extension is indicated by green lamps illuminating and handle pumping becoming heavy.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

CAUTION

To make emergency gear extension, pull out circuit breakers LG CONT, LG MOTOR and DOOR MOTOR and place landing gear control switch to DOWN. Do not perform this operation with the above circuit breakers in CLOSE positions.



A Emergency gear down handle B Gear mechanism

A Emergency gear down handle B Gear mechanism

G.1 ADJUSTMENT AND OPERATIONAL TEST

(1) Jack up the airplane.

(2) Retract landing gear.

(3) Pull emergency handle to the extent of small lever coming off.

(a) Hook of main gear forward door is pulled and doors are unlocked.

(b) Clutches of main gear forward door mechanism are disengaged and the doors become free.

(c) Hook of main gear aft door is pulled and doors are unlocked.

(4) Remove access door (F.STA. 1025 ~ 4285) between keels and check the following items.

(a) Claw on the center lever engages with ratchet teeth. (Fig 4-51).

(b) No abnormal condition in cable system and center lever mechanism.

(5) L. H. and R. H. clutches which release main gear forward doors, do not reach the ends of their travels.

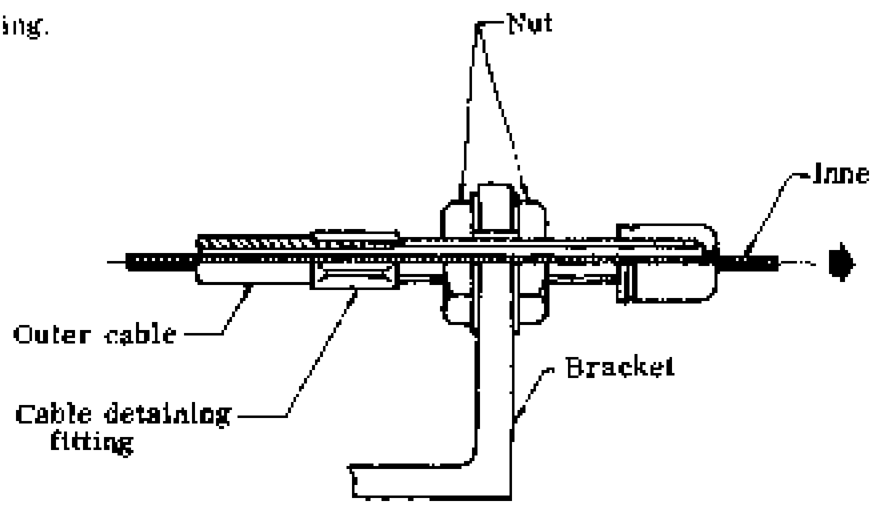
(6) Clearances between hooks and rollers in forward and aft doors should be more than 0.08 in.(2 mm).

(7) Perform emergency gear extension operation and check the following items.

(a) Ascertain that the operation is performed smoothly.

(b) Landing gear position indicating lights (green) which indicate main and nose gear being in down position illuminate.

(8) Adjustments of Para. (5) and (6) above are made by means of cable detaining fitting.



6.2 RESET AND CHECK AFTER EMERGENCY GEAR EXTENSION

After emergency gear extension, reset and adjustment can be done on the ground without jacking as follows.

(1) Reset of cable assembly.

(a) Set small handle into the emergency handle and return the handle to the normal position. Install safety wire (QQ-W-321 .02 Dia) (S/N 697SA and subsequent).

(b) Remove access door (F. STA4025~4285) between keels and return the center lever up to the position where claw disengages (See Fig 4-51).

(c) Pull back hook release cable.

(2) Reset of clutch assembly.

(a) Extend actuator of main gear forward door mechanism to the position where the forward door open limit switch actuates, by using the ground door open switch installed on the aft bulkhead of R. H. bulge main gear well.

(b) Align clutch mating mark and engage each link.

(c) Push up hook to stop, and return pulley while winding back clutch cable (See Fig. 4-52).

(d) Set hook to pulley groove.

1. Bracket
2. Center lever
3. Ratchet teeth
4. Claw
5. Spring
6. Cable (to handle in cockpit)
7. Clutch release cable
8. Unlock cable

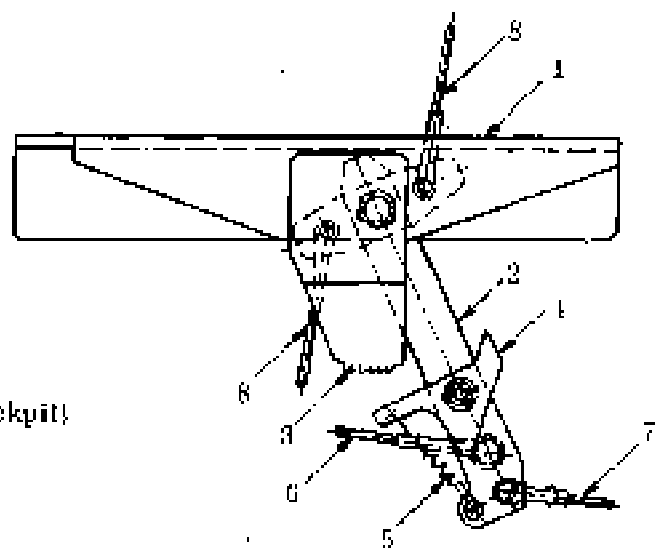


Fig. 4-51 Reset after emergency gear extension

(3) Operate forward door mechanism and make sure that opening and closing operations are performed normally. The forward door can be opened or closed by using ground door open switch on the bulge rear bulkhead of R. H. main gear wheel well.

(4) Operation of forward door shall be as follows.

(a) Close the circuit breakers DOOR MOTOR and LG CONT.

(b) Place landing gear control switch to "DOWN."

(c) Turn the forward door operating switch in R.H. main landing gear well up or down. The door fully opens or closes.

(d) After operation, close the door fully by the forward door operating switch turned down and return the switch guard and apply safety wire.

(5) Pull out all circuit breakers.

NOTE

When the forward door is checked for operation, pull out circuit breaker "LG MOTOR" to avert danger.

CAUTION

To change the sleeve (end of outer cable) location of Teflon cable (lock release cable) will cause a change in sag and travel. Sufficient care should be taken when changing the sleeve location.

Lockwire in Para. (1) should be cut in first handle operation of emergency gear down. Specified lockwire must be used and one lockwire must pass through holes on the both ends.

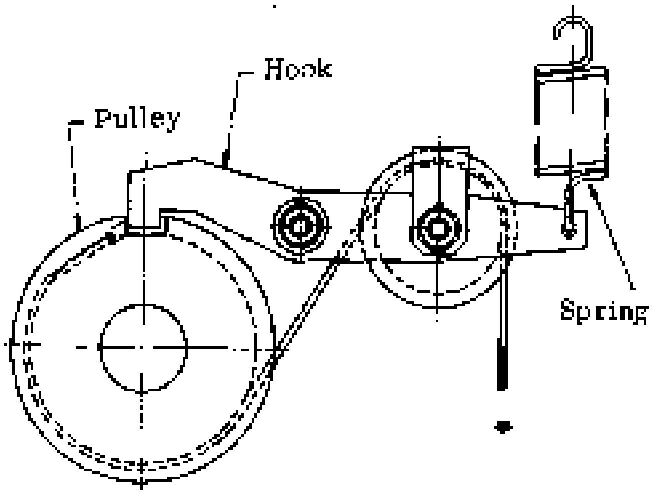


Fig. 4-52 Put hook in pulley groove

6.3 EMERGENCY GEAR EXTENSION PUSH BUTTON (See Fig 4-53, Aircraft S/N 652SA and 661SA)

The push button is located on the floor near the emergency gear extension handle. The push button is used when handle operation of emergency gear extension mechanism can not be operated due to improper engagement of the claw clutch of the mechanism. When the button is pushed, one side shaft of the claw clutch turns a little and corrects the engagement.

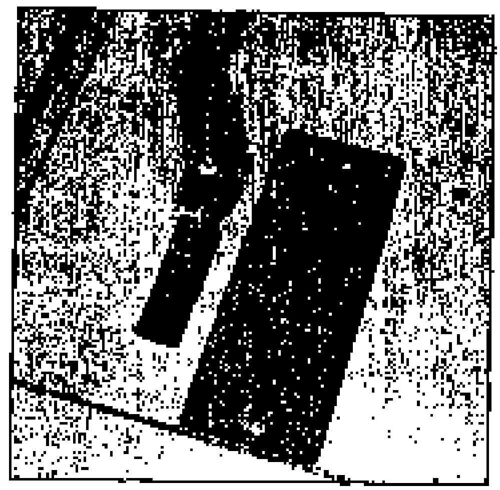


Fig. 4-53 Emergency push button

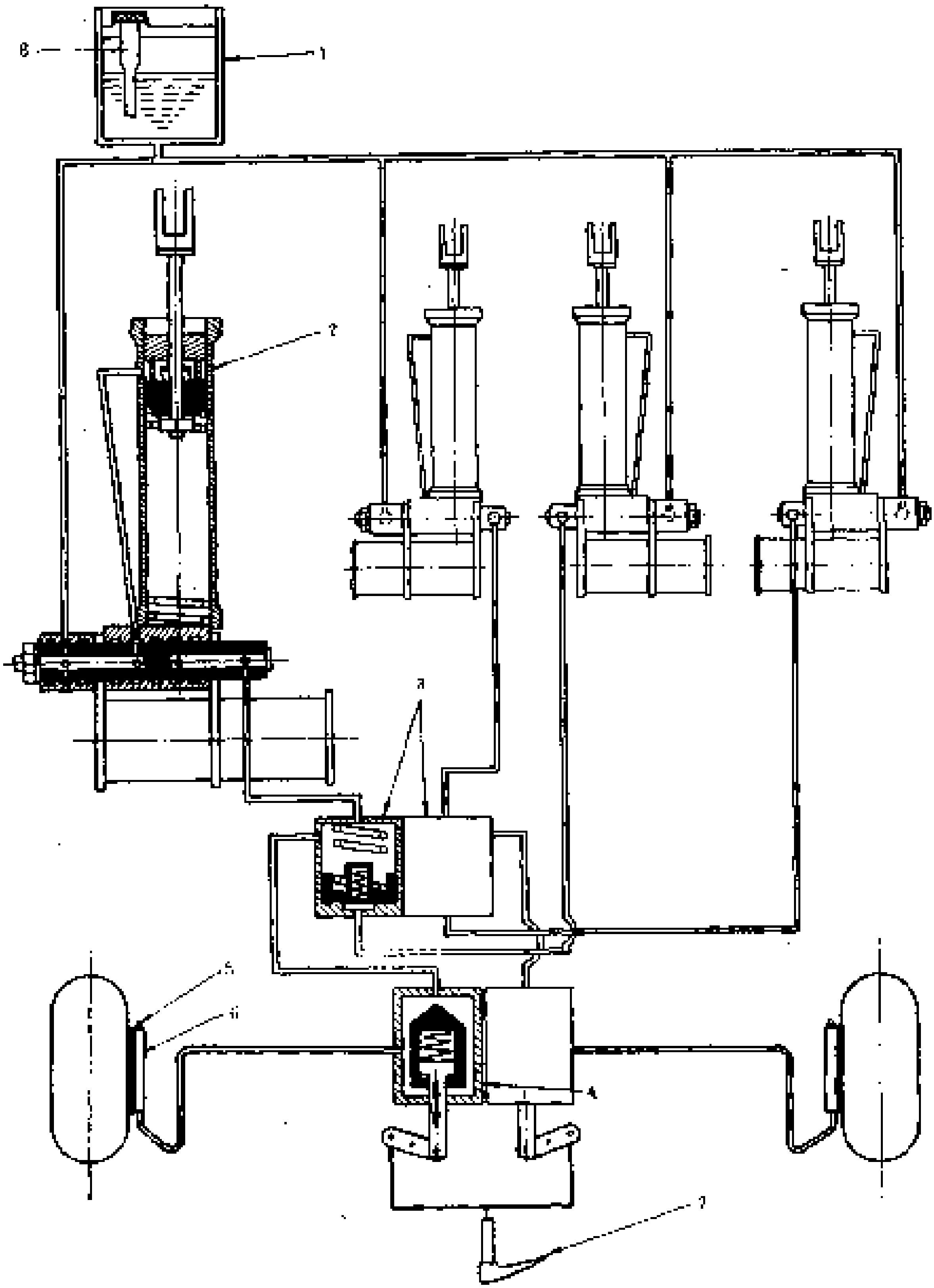
ORIGINAL
As Received By
ATP

Removal and installation of the push button is easily performed by 2 screws attached to the floor.
To install the push button, fix it at the location suited for the shaft turn.

7. BRAKE SYSTEM

7.1 GENERAL DESCRIPTION

The brake system is a master-cylinder-type hydraulic system consisting of reservoir, master cylinder, mixer valve, parking valve, tubing and wheel brakes. (See Fig 4-54).
The master cylinders, located beneath the rudder pedals, can be operated by applying toe pressure on either the pilot's or co-pilot's rudder pedals. The right pedal operates the right brake, the left pedal the left brake. By applying toe pressure on both rudder pedals and pulling the parking handle, brake pressure can be maintained.



1. Reservoir
2. Master cylinder
3. Mixer valve
4. Parking valve
5. Bleeder valve
6. Wheel brake
7. Parking handle
8. Level gage

Fig. 4-54 Brake system schematic diagram

7.2 RESERVOIR (See Fig. 4-55)

The reservoir, located on L.H. side of the rear wall of the electronics compartment, supplies hydraulic fluid (MIL-H-5606) to the master cylinders. There is a level gauge beneath the cap of the reservoir, which makes it easy to check the oil level. The cap is vented. Wipe dust off the cap surface during servicing operation.

7.3 MASTER CYLINDER (See 4-56 and 4-57)

The master cylinders are designed to convert the force applied to the toe pedals into hydraulic pressure, and are directly connected to the lower structure of the respective pedals.

7.3.1 REMOVAL AND INSTALLATION

(1) Remove the check valve from the wheel brake, and drain the hydraulic fluid from the brake system. Disconnect the lines at the master cylinders.

(2) Remove the rudder and steering (pilot's master cylinders only) rods, remove the connecting bolts to the frame and the pedal base, and remove the pedal and base together from the frame.

(3) Remove the connecting bolt from the pedal and master cylinder rod end.

(4) Remove the two bolts located at the bottom end of the master cylinder and remove the master cylinder from the base.

(5) Reinstall in reverse sequence of removal.

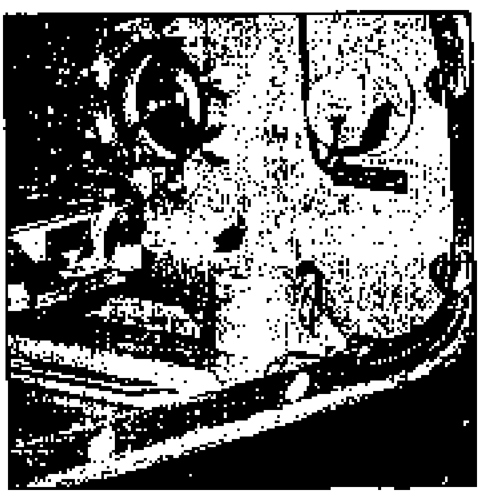


Fig. 4-55 Reservoir

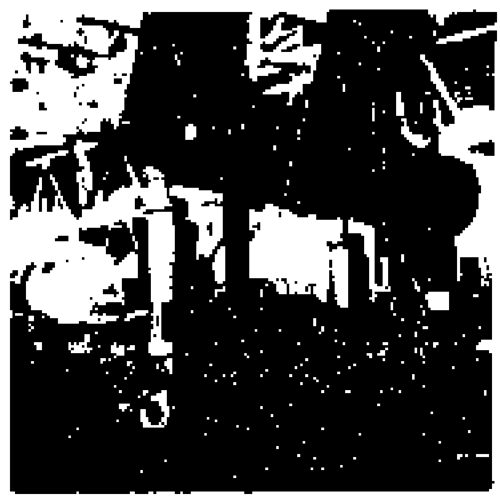


Fig. 4-56 Installation of master cylinder

7.3.2 DISASSEMBLY AND INSPECTION

In regard to disassembly and inspection of brake system, see Overhaul Manual YET 74146

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1. Rod end
2. Piston rod
3. Cap
4. Valve
5. Piston
6. Spring retainer
7. Spring
8. Cylinder
9. Elbow
10. Reservoir port
11. Brake port
12. Shaft
13. Support

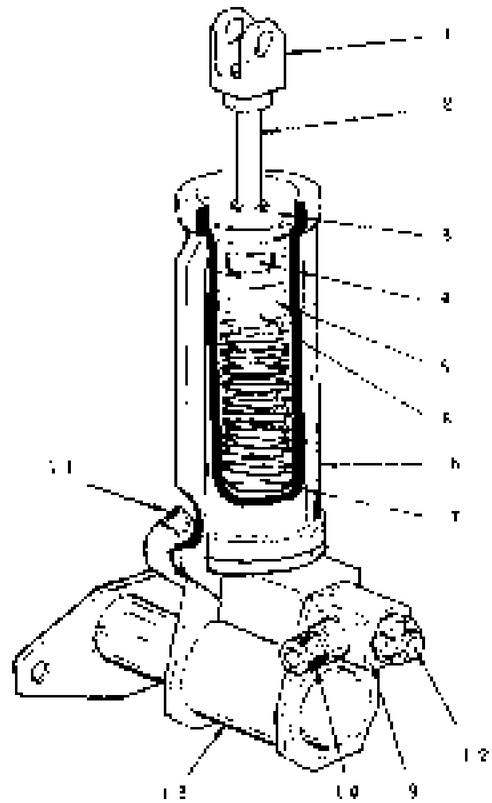


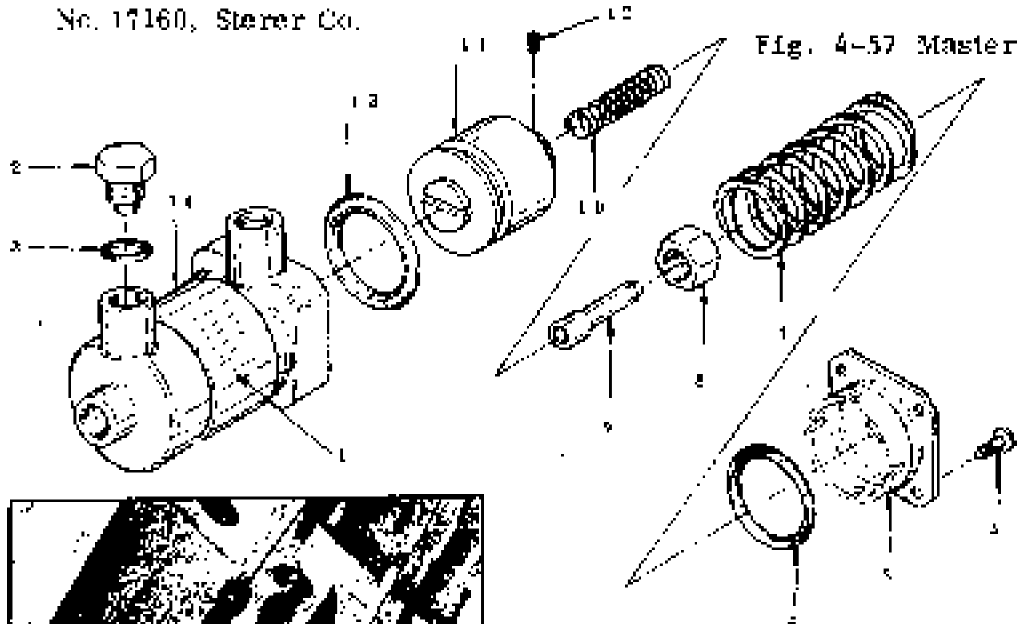
Fig. 4-57 Master cylinder

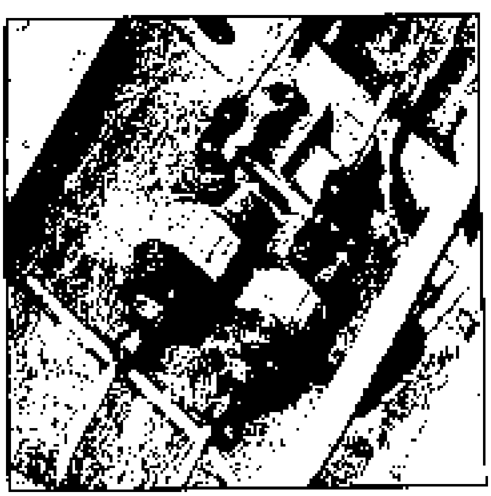
7.4 MIXER VALVE (See Fig 4-58)

The mixer valves transfer the hydraulic pressure of the pilot's or co-pilot's master cylinders to the brakes and are located under the floor of the pilot's seat.
The valves can be removed with ease by disconnecting the lines and removing the fitting nut.

7.4.1 DISASSEMBLY AND INSPECTION

In regard to disassembly and inspection of mixer valve, see Overhaul Manual No. 17160, Sterer Co.





1. Identification plate
2. Bleeder
3. Gasket
4. Screw
5. Cap
6. O-ring
7. Spring
8. Cap
9. Poppet
10. Spring
11. Piston
12. Nylok
13. O-ring
14. Body

Fig. 4-58 Mixer valve

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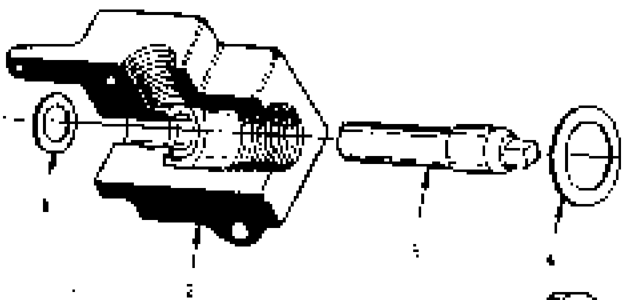
7.5 PARKING VALVE (See Fig 4-59)

The parking valve is located behind the center pedestal and can be easily removed by disconnecting the parking handle and lines and removing the mounting bolt.

7.5.1 DISASSEMBLY AND CHECK

Check the valve seat and stem for damage. Replace all O-rings with new ones and assemble.



1. O-ring
2. Body
3. Stem assembly
4. O-ring
5. Valve seat

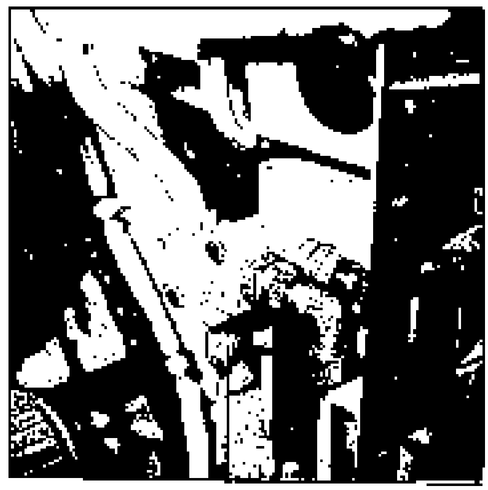


Fig. 4-59 Parking valve

7.5.2 OPERATIONAL TEST

(1) Connect a pressure gage to the outport and apply 1000 psi (70 kg/cm^2) of pressure at the pressure port.

(2) Push the plunger (stem) fully in and remove the pressure line.

(3) The valve shall maintain 1,000 psi (70 kg/cm^2) pressure and there should be no leakage.

7.6 SWIVEL JOINT

Two swivel joints are installed around L.H. and R.H. main gear respectively. (See Fig 4-60).

7.6.1 DISASSEMBLY AND INSPECTION

(1) Disassemble the joint in numerical order of the figure.

(2) Clean all metal parts with dry cleaning solvent and dry.

(3) Check for scratches, cracks or other damages.

7.6.2 ASSEMBLY

(1) Replace all O-rings with new ones. Apply thin coat of vaseline (VV-P-236) on O-rings and O-ring grooves.

(2) Reinstall in reverse sequence of removal.

NOTE

The washer (4 in Fig 4-60, Detail B) should be installed, facing the marked side to O-ring.

7.6.3 OPERATIONAL TEST

(1) Conditions

Actuating fluid : MIL-H-5606

Pressure : 1500 psi (105 kg/cm^2)

(2) After applying pressure, turn swivel 5 times. No leakage should be detected during succeeding five minutes.

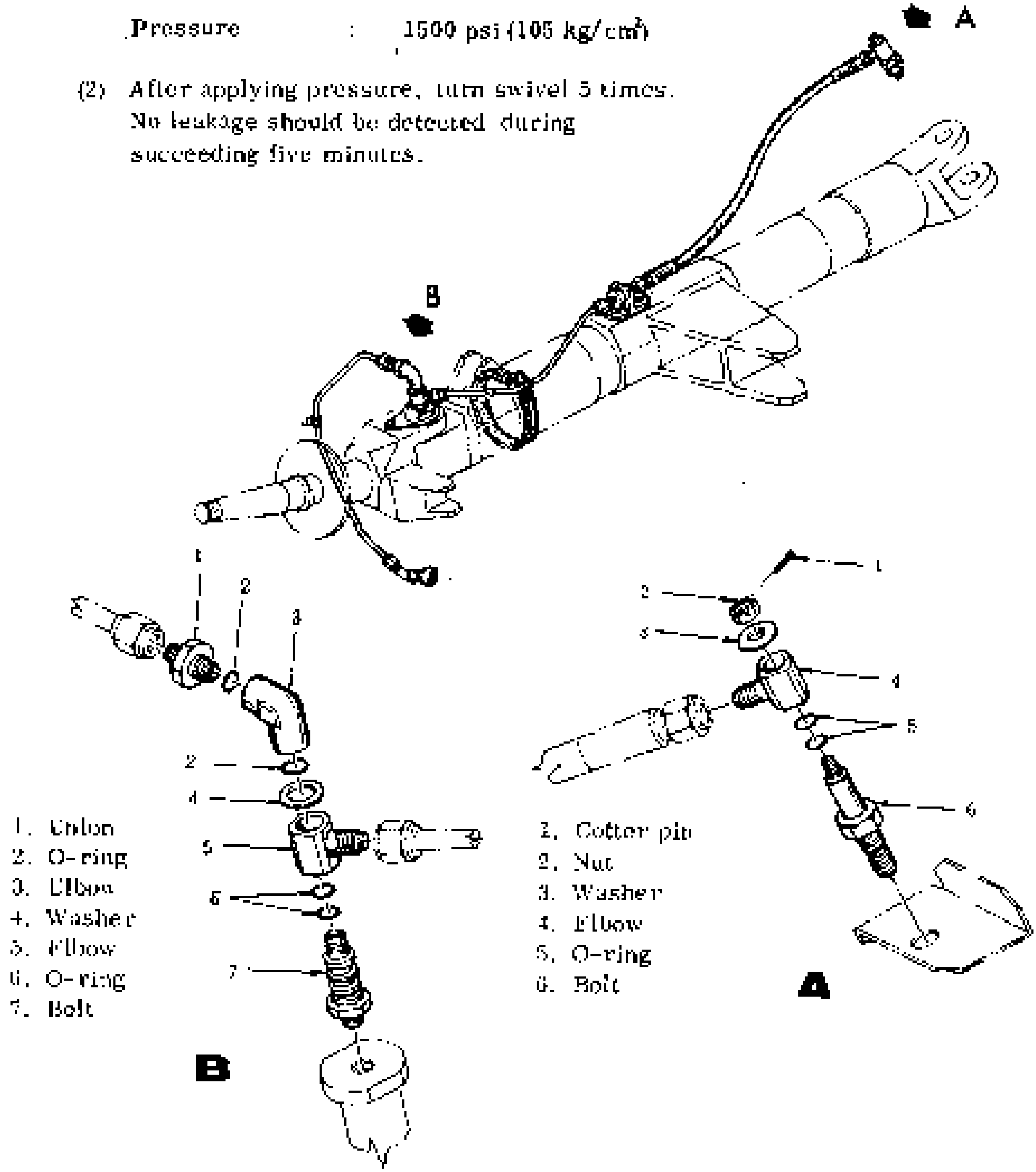


Fig. 4-60 Swivel Joint

7.7 BRAKE ASSEMBLY

The brake assembly is single disc type and brake housing is made from magnesium alloy casting. Braking is performed by friction produced when oil pressure is applied at the piston and the brake lining pushes discs. When no pressure is applied, the lining is pushed slightly against the disc with a spring installed on the back of the piston.

7.7.1 DISC LINING LIMIT AND REPLACEMENT (See Fig 4-61)

When the thickness of carrier lining assembly is less than 0.188 in. (4.8 mm), overhaul brake assembly per B. F. Goodrich Co. Overhaul Manual No. 168.

7.7.2 REMOVAL AND INSTALLATION

See Paragraph 3.4.1.

7.7.3 DISASSEMBLY AND CHECK

See B. F. Goodrich Co. Overhaul Manual No. 168.

NOTE

Immediately after the entire brake, disc or lining is replaced, brake effectiveness may refuse to work due to insufficient fitting of disc or lining. In this case, work the brake several times during taxiing to obtain better effectiveness of the brake.

1. Piston housing
2. Piston
3. Spring
4. Piston insulator
5. Torque plate
6. Insulator
7. Wheel
8. Disc
9. Brake lining
10. Carrier lining
11. Brake pressure

A = More than 0.34 in. (8.5 mm) with park brake set

B = Min. thickness 0.30 in. (7.6 mm)

C = Min. thickness 0.19 in. (4.8 mm) brake & carrier lining

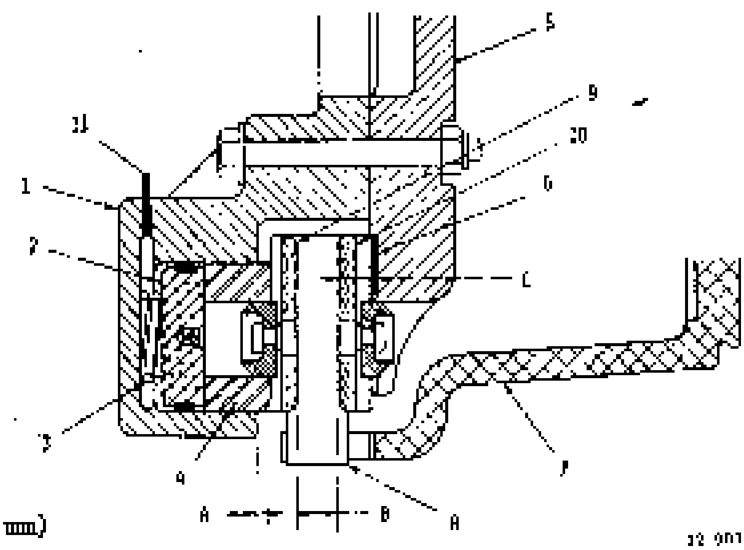


Fig 4-61 Brake lining wear

7.8 BRAKE LINE

The brake line consists of aluminum tube line (3/8 in. Dia. from reservoir to branch leading to master cylinder, other line : 1/4 in. Dia.) installed on airframe, and flexible tube and corrosion resistant tube installed around main gear leg. These lines are all standard parts.

(1) Do not disconnect lines unless parts replacement becomes necessary due to leakage or other damage.

(2) After disconnecting, flush line with MIL-H-5606 hydraulic fluid or denatured alcohol. Check the threaded section of fittings for damage, the flares for deformation or flaws, and the tubes for dents and cracks. Replace all defective parts.

(3) When removing the fittings located on the pressure bulkhead, replace the washers with new ones.

7.9 MAINTENANCE OF BRAKE SYSTEM

7.9.1 HYDRAULIC FLUID

Use specification MIL-H-5606 hydraulic fluid. The total fluid quantity for the system is approximately 0.18 gal. (0.7 ℓ), of which approximately 0.04 gal. (0.15 ℓ) is the fluid quantity in the reservoir.

7.9.2 FLUID CHARGE

(1) Remove reservoir line, connect a suitable line or hose to the line and lead the line into a container outside the aircraft. Remove bleeder valve and reducer located at the upper part of the brake. Connect a hand pump or test stand to brake body through a union and supply hydraulic fluid slowly. Continue until fluid appears in the container.

(2) Connect the hand pump or test stand to the line leading to the container and install bleeder valve and reducer on the brake body as original conditions.

(3) Unfasten the bleeder valve and supply fluid slowly under a pressure of less than 14 psi (1 kg/cm^2) from the hand pump.

(4) After making sure that there are no bubbles in fluid coming out of the bleeder valve, add approx. 0.13 gal. (0.5 ℓ) of fluid, close the bleeder valve, apply toe pressure on the pedal (co-pilot side) and make sure that is there is no air trapped.
Remove the hand pump or test stand and connect line to reservoir. Take care not to spill fluid.

(5) Keep fluid in the reservoir to the regular level. (middle of the level gage).

7.9.3 REFILL

If the oil level of the reservoir drops below the end of the level gage, refill hydraulic fluid (MIL-H-5606) up to the middle stage of the level gage.

(1) When it is necessary to bleed air during service, unfasten the bleeder valve of the brake and apply toe pressure on the pedal to let fluid flow. Refill fluid so that the reservoir does not empty.

7.9.4 DRAIN

(1) To drain fluid for the brake system, remove the elbow located in the lower part of the brake.

7.9.5 INSPECTION

Daily inspection can be made by feeling toe pressure on the pedals. After replacement of components or when sponginess and ineffective brake action is detected, proceed as follows;

(1) Remove the bleeder valve and connect a pressure gage.

(2) Apply toe pressure on the pedal and make sure that a pressure of more than [illegible] psi (42 kg/cm²) is available.

(3) With toe pressure applied on the pedal, pull the parking handle and release the pedal to see if the pressure is maintained.
Check the system for external leaks or deformation.

(4) Release the parking handle, rotate the wheel by hand, and check for excessive drag.

(5) Disconnect the pressure gage and bleed air from brake.

8. WHEELS AND TIRES

8.1 GENERAL DESCRIPTION

The wheels and tires used on the aircraft are as follows:

Main wheel	:	3-1304-2 (B. F. GOODRICH)		
Nose wheel	:	3-897 (B. F. GOODRICH), 9532926 (GOODYEAR), A9532926 (GOODYEAR) or 40-87 (CLEVELAND)		
Main wheel tire	:	8.50 x 10	10 ply	Type III tubeless
Nose wheel tire	:	5.00 x 5	6 ply	Type III regular tube

8.2 LUBRICATION OF BEARING

(1) Fill bearing cone and both edges of roller with grease (MIL-G-3545) every tire replacement.

NOTE

(i) Clean bearing with dry cleaning solvent (P-D-680) to remove old grease.

(ii) Do not over-lubricate. Wipe off excess grease on rollers.

(iii) Handle bearing with care to keep dust off.

(2) Apply thin coat of grease on bearing cone.

8.3 DISASSEMBLY, REASSEMBLY AND INSPECTION OF NOSE WHEEL

8.3.1 DISASSEMBLY (See Fig 4-62)

Disassemble nose wheel in accordance with numerical order of the figure. See Para. 8.4.1.

8.3.2 INSPECTION

See Para. 8.4.2.

8.3.3 REASSEMBLY

(1) Reassemble in reverse sequence of disassembly.

(2) Clean and lubricate bearing in accordance with Para. 8.2.
If felt seals are used, lubricate with light machine oil SAE10 (MIL-L-7178) or equivalent.

(3) Wipe wheel flange, bead seat and mating surfaces with a cloth soaked in denatured alcohol to remove foreign matters.

(4) Place wheel half on a clean flat surface and carefully position the tire, aligning the tube valve with the valve hole section in the wheel.

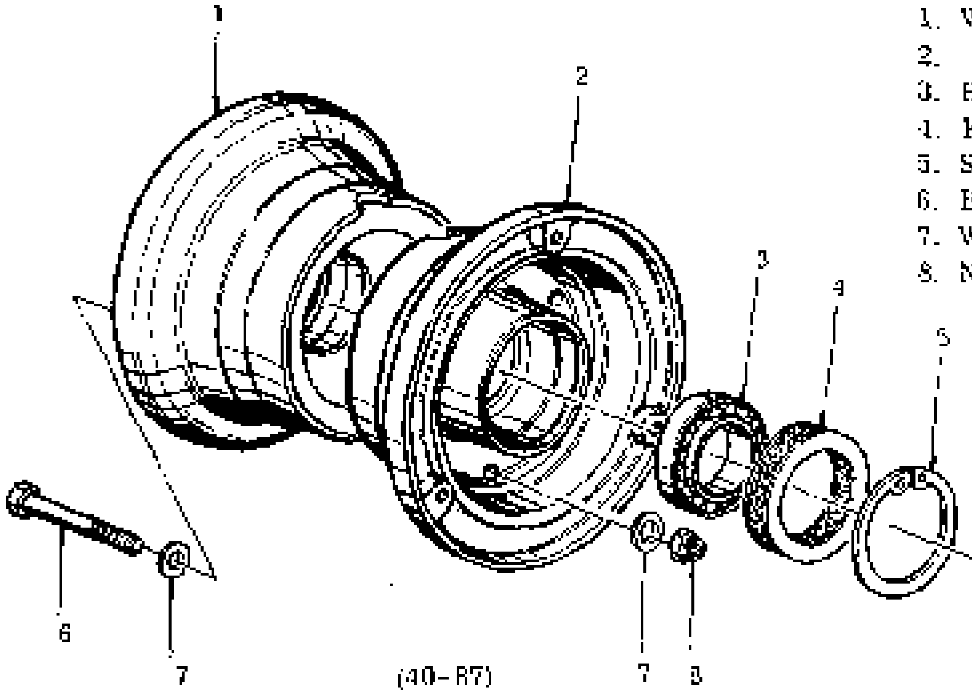
(5) Insert the other wheel half in the tire aligning the wheel with the tube valve.

(6) Insert wheel bolts as follows.

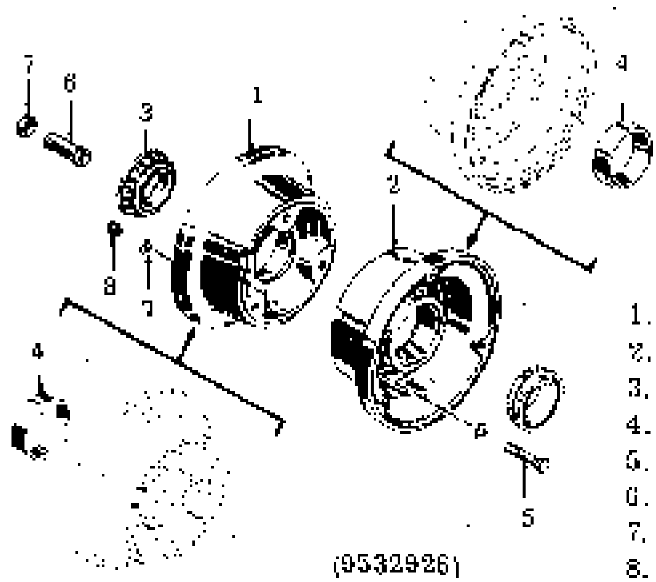
(a) Lubricate bolt threads and bearing surface of bolt heads, nuts and washer with MIL-T-5544 anti-seize compound before reassembly.

(b) Compress wheel section enough to allow installation of three bolts, washer and nuts.

(7) Tighten bolts evenly in increments of 15 in-lbs until a final value of 50 ~ 70 in-lbs is obtained.



1. Wheel half (Inboard)
2. " " (Outboard)
3. Bearing cone
4. Felt ring
5. Snap ring
6. Bolt
7. Washer
8. Nut



1. Wheel half (Inboard)
2. " " (Outboard)
3. Bearing cone
4. Bearing cup
5. Bolt
6. Bush
7. Washer
8. Nut

Fig. 4-62 Nose wheel assembly

8.4 DISASSEMBLY, REASSEMBLY AND INSPECTION OF MAIN WHEELS

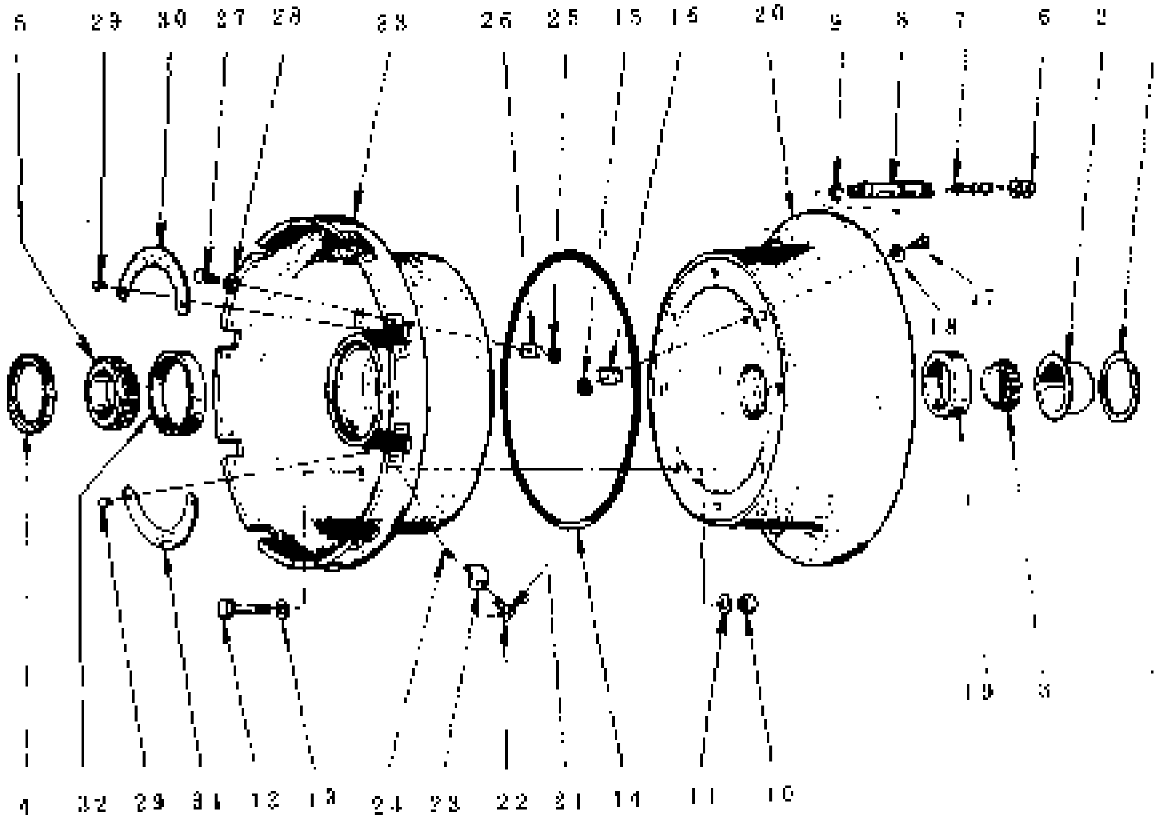
8.4.1 DISASSEMBLY (See Fig 4-63)

(1) Remove valve core and deflate tire.

(2) Break head from both flanges by applying pressure in even increments around the entire sidewall close to the tire beads.

(3) Remove bolts, nuts and washers which secure the wheel halves.

(4) Separate the wheel halves, remove tire.



1. Retainer ring	12. Bolt	23. Torque key
2. Hub cap	13. Washer	24. Helical insert
3. Bearing cone	14. O-ring	25. Nut
4. Seal	15. Nut	26. Balance weight
5. Bearing cone	16. Balance weight	27. Screw
6. Cap	17. Screw	28. Washer
7. Valve	18. Washer	29. Rivet
8. Valve stem	19. Bearing cup	30. Identification plate
9. Grommet	20. Wheel half (Outboard)	31. Instruction plate
10. Nut	21. Lockwire	32. Bearing cup
11. Washer	22. Screw	33. Wheel half (Inboard)

Fig. 4-63 Main wheel assembly

(5) Disassemble wheel in numerical order of the figure.

NOTE

Bearing cups should not be removed unless replacement is necessary. To remove the cups, it is necessary to heat wheel halves in an oven at a temperature not to exceed 300° F for 30 minutes.

(6) Clean all metal components in cleaning solvent (Federal Specification P-S-661).

(7) Clean O-rings in denatured alcohol.

8.4.2 INSPECTION

(1) Inspect for cracks, scratches or other damage. Replace cracked parts.
(2) Check O-ring for damage or excessive deformation. Replace if necessary.
(3) Check the rollers of bearing cone for wear, overheat or corrosion. Check the roller retainer for damage. Replace if necessary.
(4) Check the hub cab for local wear. Replace if necessary.
(5) Check the bolts which secure the wheel halves for cracks, deformations and corrosion. Replace if necessary.
(6) Replace the self-locking nuts for wheel half attaching bolts after ten times of disassembly and reassembly or before, should there be apparent thread deformation.

NOTE

To repair the wheels minimize the damage area repaired by polishing.

(7) Polished portions should be finished per specification MIL-M-3171 . Type I and apply two coats of zinc chromate primer per specification MIL-P-8585A. Bearing cup should not be finished with lacquer.
(8) For repair portions of which aluminized lacquer has been applied, apply two coats of aluminized lacquer per specification MIL-L-7178.
(9) Spray one coat of zinc chromate primer on holes for the attaching bolts per specification MIL-P-8585A.
(10) Make sure that bead and flange portions of the wheels are finished with zinc chromate primer and aluminized lacquer. This is for corrosion preventive purpose as well as to seal air pressure of tubeless tires.

8.4.3 REASSEMBLY

(1) Reassemble in reverse sequence of disassembly.
(2) Clean and lubricate bearing in accordance with Para. 8.2.
(3) If bearing cups have been removed, the wheel half should be heated at a temperature not to exceed 300°F and the cup should be cooled at 0°F, then reassembled.
(4) Before mounting tire, make certain it is free of foreign material, the bead areas are clean and there is no damage.
(5) Wipe wheel flange bead seat and wheel mating surfaces with a cloth dampened with denatured alcohol to remove foreign matters.
(6) Lubricate O-ring with a light coat of grease MIL-G-3545.
(7) Place the brake side wheel half on a clean flat surface and install O-ring.
(8) Place tire on the wheel half.
(9) Position the outboard wheel half in tire with the red dot at the valve stem and install the bolts.
    (a) Lubricate bolt threads and bearing surfaces of nuts, bolt heads and washers with anti-seize compound MIL-T-5544 before assembly.
    (b) Compress wheel section enough to allow installation of four bolts, washers and nuts 90° apart and install the bolts and tighten until wheel halves seat. Then, install remaining bolts, washers and nuts. The bolts should be inserted from the brake side of the wheel.
    (c) Tighten bolts in equal increment to a torque value of 190 to 210 in-lbs. (219 to 242 kg-cm).

8.5 TIRE

8.5.1 INFLATION PRESSURE

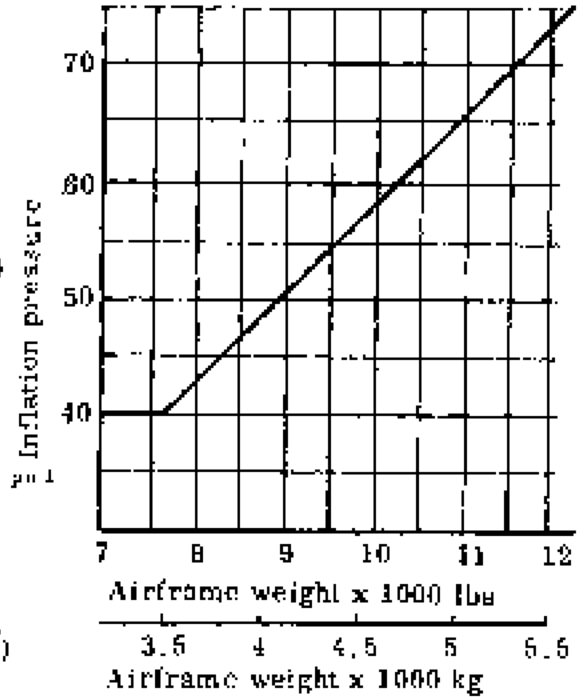
(1) Main wheel tire

(a) Reassembly : 60 ± 5 psi (4.2 ± 0.4 kg/cm²)

Long period of storage : 48 ± 5 psi (3.4 ± 0.4 kg/cm²)

(b) Mounting (no load): See Fig 4-64

When the tires are on the ground, the tires shall be inflated 4% higher than the value in the figure.



Main tire inflation pressure
Fig. 4-64

(2) Nose tire

(a) Reassembly : 55 ± 5 psi (3.9 ± 0.4 kg/cm²)

Long period of storage : 44 ± 5 psi (3.1 ± 0.4 kg/cm²)

(b) Mounting : 55 ± 5 psi (3.9 ± 0.4 kg/cm²)
On ground : 57 ± 5 psi (4.0 ± 0.4 kg/cm²)

8.5.2 MAINTENANCE AND CHECK OF TIRES

Tires which are worn to such an extent that the treads are barely recognizable and those considered unserviceable because of damage should be replaced.

8.5.3 BALANCE MARK

(1) Tube tire: Align the balance mark of the tube to that of the tire.
The balance mark of tube is approximately 1/2 in. (13 mm) wide and 2 in. (51 mm) long. The balance mark of tire is a red dot located above the bead of tire.

(2) Tubeless tire: Align the balance mark (red dot) of tire to the valve stem in the wheel.

8.5.4 TIRE TREAD

The main and nose wheel tires should be of the same tread pattern.

8.5.5 DUAL WHEELS

In order to ensure the equal distribution of load, tires of the same brand should be used for the dual wheels (nose gear).
When this is impossible, the diameters of tires (inflated), should be measured to make sure that they are the same, within the difference of 1/4 in. (6.4 mm).

8.5.6 SLIP MARK (Applicable to tube tire)

To detect tire slip and to reduce damage to the tire and tube caused by wheel slip, a marking shall be provided to indicate that the tire and rim are in the same position. The mark should be 1 in. (25 mm) wide and 2 in. (51 mm) long.

It shall be painted across the tire sidewall and wheel rim so that it will extend 1 in. (25 mm) over the tire and 1 in. (25 mm) over the rim.
Use insignia red, color number 11136 (Fed Std. 595), 3-M red tire marking paint No. EC1626.
When tire slip is detected after marking, deflate tire and place the valve in the correct position. Then, reinflate tire, remove the previous slip mark, and paint a new one.

CAUTION

Check tire slip mark daily. If tire slip is detected, deflate tire and correct.

9. LANDING GEAR POSITION INDICATING LIGHT AND WARNING SYSTEM

9.1 LANDING GEAR POSITION INDICATING LIGHT

The landing gear position indicating lights consist of three green lamps and one red lamp installed adjacent to the control switch. Each indicator lamp is push-to-test type.

(1) Gear down operation	:	Down limit switches are installed on nose gear and main gears. (left and right) Green lamp illuminates.
(2) Gear up operation	:	When the operation is normal, all lamps go off.
(3) Red lamp	:	Red lamp indicates unsafe condition and illuminates in the following cases.
(a) Gear down operation	:	Main gear forward doors are not closed or nose or main gear is not in down locked position.
(b) Gear up operation	:	Nose gear up indicator switch is not set. or main gear forward doors are not closed, or main gear up indicator switch is not set (main gear aft doors are not closed).

9.2 WARNING SYSTEM

The warning horn will sound in the following cases.
(1) When the control switch is placed to the UP position on the ground.
(2) If a Power Lever is pulled back to near FLIGHT IDLE and the landing gear is not down-locked.

NOTE

S/N 652SA, 661SA, 697SA thru 717SA
When airborne, this Warning Horn may be silenced by use of the Horn Cutout Switch.

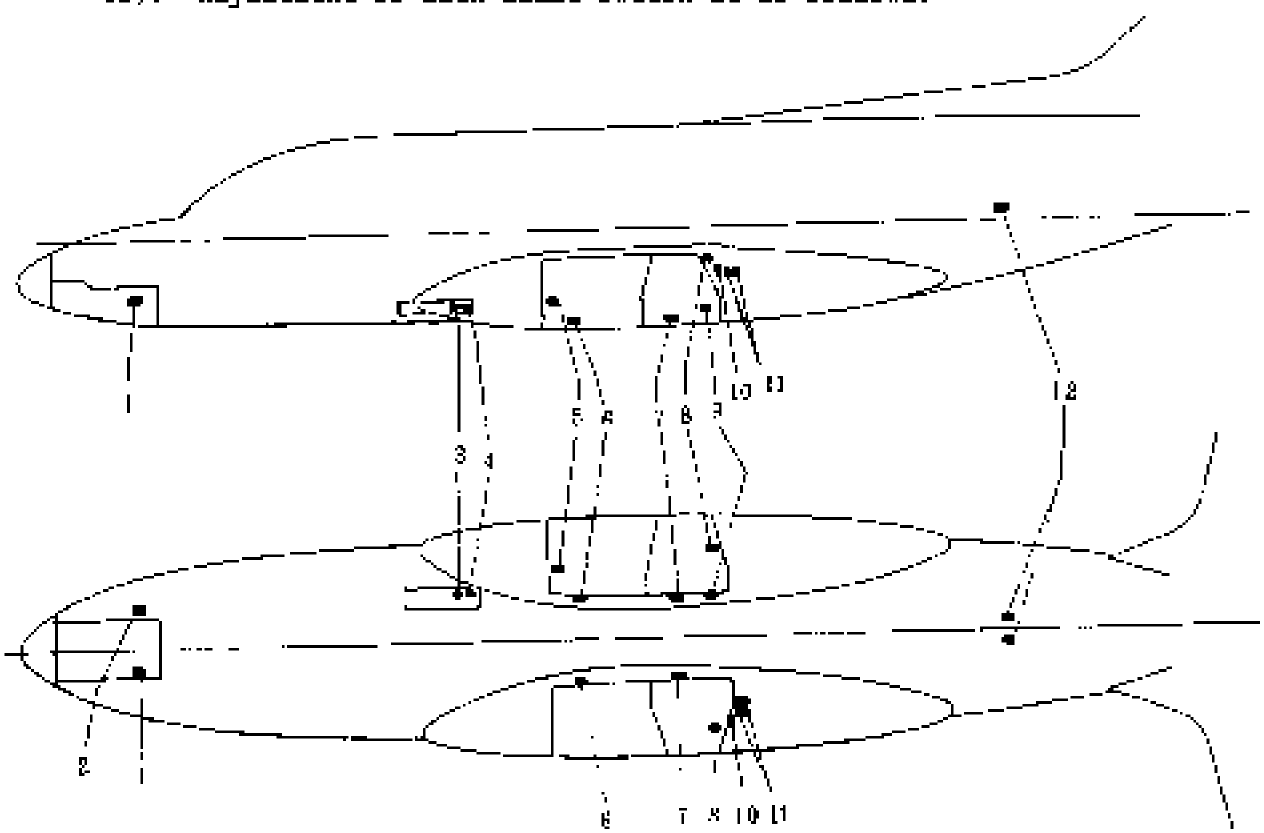
S/N 718SA and Subsequent
When airborne the Warning Horn may be silenced by use of the Horn Cutout Switch, except when the Flaps are extended to 20° or 40° down.

9.3 ADJUSTMENT AND CHECK

As to adjustment of switch, see Para 10.9 LAND GEAR SAFETY SWITCH and Para. 10.11 ADJUSTMENT OF PROPELLER LOW PITCH LIMIT SWITCH. In daily check, check for proper installation of switches and that no foreign material is on the switches.

10. ADJUSTMENT OF LIMIT SWITCH

In order to perform landing gear operation safely and exactly, 13 limit switches are used for linkages of landing gear operation mechanism (see Fig 4-65). Adjustment of each limit switch is as follows.



1. Nose gear down limit switch (S105)
2. Nose gear up indicator switch (S106)
3. Up limit switch (S282)
4. Up zone switch (S283)
5. Main gear fwd. door open limit switch (S330)
6. Main gear fwd. door close limit switch (S328, S329)
7. Main gear up indicator switch (S326, S327)
8. Main gear down indicator switch (S324, S325)
9. Landing gear safety switch (S322)
10. Ground door open switch (S321)
11. Zone control relay (K305)
    Landing gear up relay (K306)
    Landing gear down relay (K307)
    Door open relay (K308)
    Door close relay (K309)
12. Propeller low pitch limit switch (S317, S318)

Fig. 4-65 Installed position of limit switches

10.1 NOSE GEAR DOWN LIMIT SWITCH (See Fig 4-66)

(1) This limit switch is installed on the RH keel, nosegear wheel and stops revolution of landing gear motor through landing gear down relay after gear extension. This switch also actuates main gear forward door actuator and closes the door.

(2) Adjust the limit switch temporarily so that the switch actuates at the position where the LH downlock enters more than 0.24 in. (6 mm) in the groove and RH enters more deeply. Final adjustment is made together with the landing gear power train (see Para 4.7).

Fig. 4-66 Nose gear down limit switch

10.2 NOSE GEAR UP INDICATOR SWITCH (S106) ADJUSTMENT (See Fig 4-13)

After installation of nose landing gear up indicator switch, adjustment should be made in accordance with the following procedures. Retract landing gear slowly under no load and no inertia condition. Locate the hinge lug connecting the lever and link so that the switch actuates at the position where the gear travel stop nut comes within a point approximately 0.28 ± 0.04 in. (7 ± 1 mm) to the end of the gear travel. Fine adjustment of lever position is made by the worm gear.

10.3 UP ZONE SWITCH (See Fig 4-67)

This switch is installed on the mechanical stop and makes a circuit to close the main gear forward door at a moment before landing gear motor stops. Adjust installed position of the switch in accordance with Para. 4.7.2.

10.4 UP LIMIT SWITCH (See Fig 4-67)

This switch is installed on the mechanical stop and stops the revolution of the landing gear motor through landing gear up relay. Adjust installed position of the switch in accordance with Para. 4.7.2.

1. Up limit switch
2. Up zone switch

Fig. 4-67 Switches in mechanical stop

10.5 MAIN GEAR FORWARD DOOR OPEN LIMIT SWITCH (See Fig 4-68)

This switch is installed on the main gear forward door mechanism and stops the door movement when the forward door is fully opened, and also revolves the landing gear motor through a relay. Adjust in accordance with Paragraph 3.9.2.1 (9).

Fig. 4-68 Main gear fwd. door open limit switch

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10.6 MAIN GEAR FORWARD DOOR CLOSE LIMIT SWITCH (See Fig 4-69)

(1) These switches are installed on the door jambs (L.H. and R.H.), main gear forward doors, and actuate to stop door actuator through door actuator relay. These switches are connected to "UNSAFE" indicating circuit to indicate main gear forward door closed.

(2) Adjust in accordance with Para. 3.9.2.1.(10)

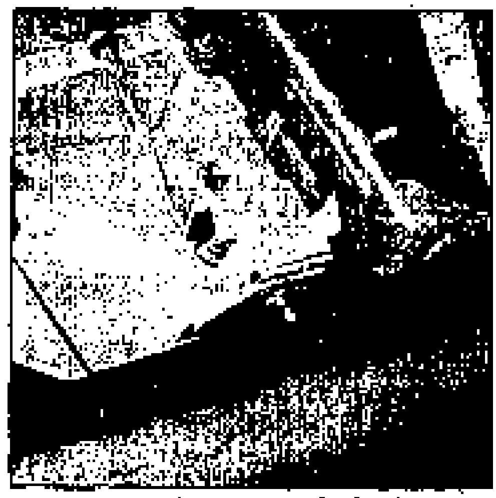


Fig. 4-69 Main gear fwd. door close limit switch

10.7 MAIN GEAR UP INDICATOR SWITCH (See Fig 4-70)

(1) This switch is installed on the LH and RH door jambs, main gear aft door, and actuated by main gear aft doors which are connected mechanically to main gear.

This switch is connected to "UNSAFE" indicating circuit to indicate main gear up position.

(2) Adjust the switch so that the limit switch plunger may move 0.16 ~ 0.20 in. (4.1 ~ 5.1 mm) from the most projected position when main gear aft door is closed.

Fig. 4-70 Main gear up indicator switch

10.8 MAIN GEAR DOWN INDICATOR SWITCH (See Fig 4-71)

(1) This switch is installed on the frame (L.H. and R.H.), attaching point of main gear oleo strut, F.STA5605. This switch lights indicator light to indicate main gear down position.

(2) Adjust the switch so that the switch may actuate when the exposed thread length of main gear drag strut becomes 1.18 ~ 1.58 in. (30 to 40 mm)

10.9 LANDING GEAR SAFETY SWITCH (See Fig 4-72)

(1) This switch is installed on the frame, attaching point of L. H. leg assembly, F. STA5605, and actuates when the aircraft is on the ground, sensing a deflection of main gear oleo. This switch is used to prevent gear up operation by mistake while the airplane is on the ground. The stall warning system and the entrance door seal system are also controlled by this switch through landing gear safety relay.

Fig. 4-71 Main gear down indicator switch

(2) **Adjust the switch so that the switch may be turned to "ON GROUND" while oleo strut is retracted to a point 1.97 in. (50 mm) away from the most retracted position, and that the switch may be turned to "IN AIR" while oleo strut is extended to a point 0.4 in. (10.2 mm) away from the most extended position.**

(3) Check the switch for operation as follows.

(a) Open (pull) circuit breaker "LG MOTOR".

(b) Place landing gear control switch to "ON".

(c) Landing gear warning horn sounds when main gear oleo is retracted.

(d) The horn stops its sound when the oleo is extended.

Fig. 4-72 Landing gear safety switch

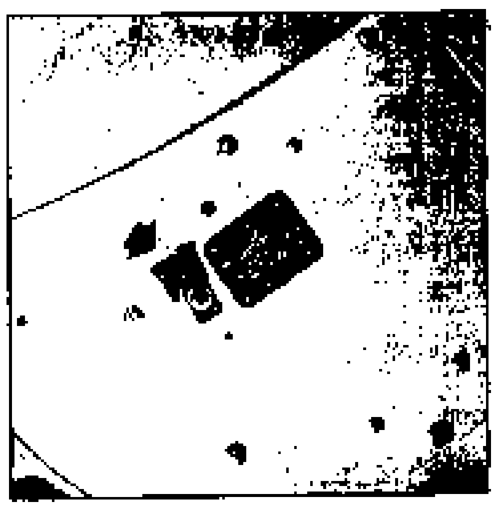


Fig. 4-73 Ground door open switch

10.10 GROUND DOOR OPEN SWITCH (See Fig 4-73)

(1) This switch is installed on the aft bulkhead (F. STA5800) of R. H. bulge main gear well, and used when necessary to open main gear forward door for maintenance or inspection.
When circuit breakers "LG CONT" and "DOOR MOTOR" in circuit breaker panel are closed using ground power source or airplane battery, this switch can open or close the main gear forward door.

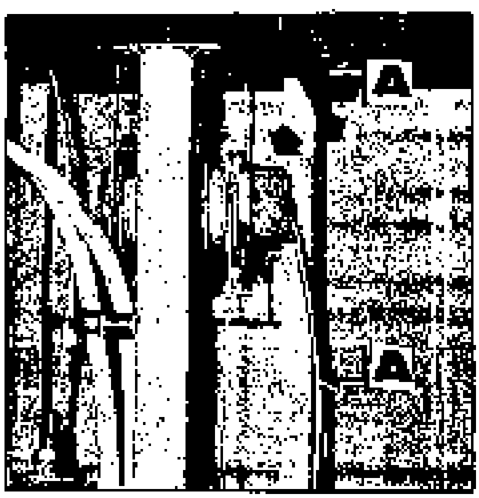
NOTE

Turn the circuit breakers to "OFF" so that the door may not be opened or closed inadvertently while working in main gear well.

(2) Adjustment is not necessary.

10.11 PROPELLER LOW PITCH LIMIT SWITCH (See Fig 4-74)

(1) This switch is installed on the engine control cable pulley bracket near F STA 8035, and energizes a warning horn when the Power Lever is moved near the FLIGHT IDLE position before the landing gear is in the down-locked position.



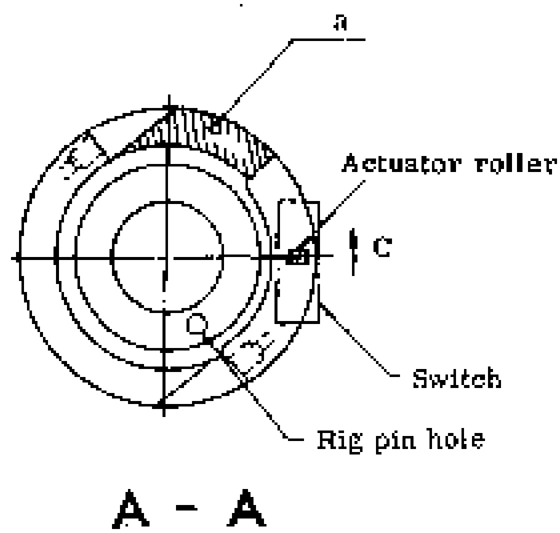


Fig. 4-74 Propeller low pitch limit switch

(2) Adjustment

(a) Install the switch so that the switch may be turned to "OFF" at the condition of A-A shown in Fig. 4-74 and that the switch may be turned to "ON" when the highest point (a) of pulley side cam comes to actuator roller position.

(b) Install the switch at about center of long hole. When the switch is moved to the direction of arrow "C" after loosening screws, the switch comes "ON" at the condition which power lever is near TAKEOFF position. Adjustable amount by long hole is approx. $\pm 9^{\circ}$ on power lever in the center pedestal.

(c) Make sure that the landing gear warning switch comes "ON" at about 0.2 in. (5.1 mm) before FLIGHT IDLE position when power lever is moved to FLIGHT IDLE position from TAKEOFF position.

11. ELECTRIC CIRCUIT

Control circuit of the landing gear motor and main landing gear forward door actuator is through the circuit breaker LG CONT. Power lines to the landing gear motor and the main landing gear forward door actuator are protected by independent circuit breakers LG MOTOR and DOOR MOTOR. The landing gear position indicating light circuit is protected by an independent circuit breaker LG POS. The circuit diagram is shown in Fig 4-75 thru 4-83 by each phase, i.e., the landing gear in downlock position on the ground, in retracted position, and extended position.

11.1 GROUND STATIC CONDITION

All power is cut off by switches and relays and system is in inoperative condition.

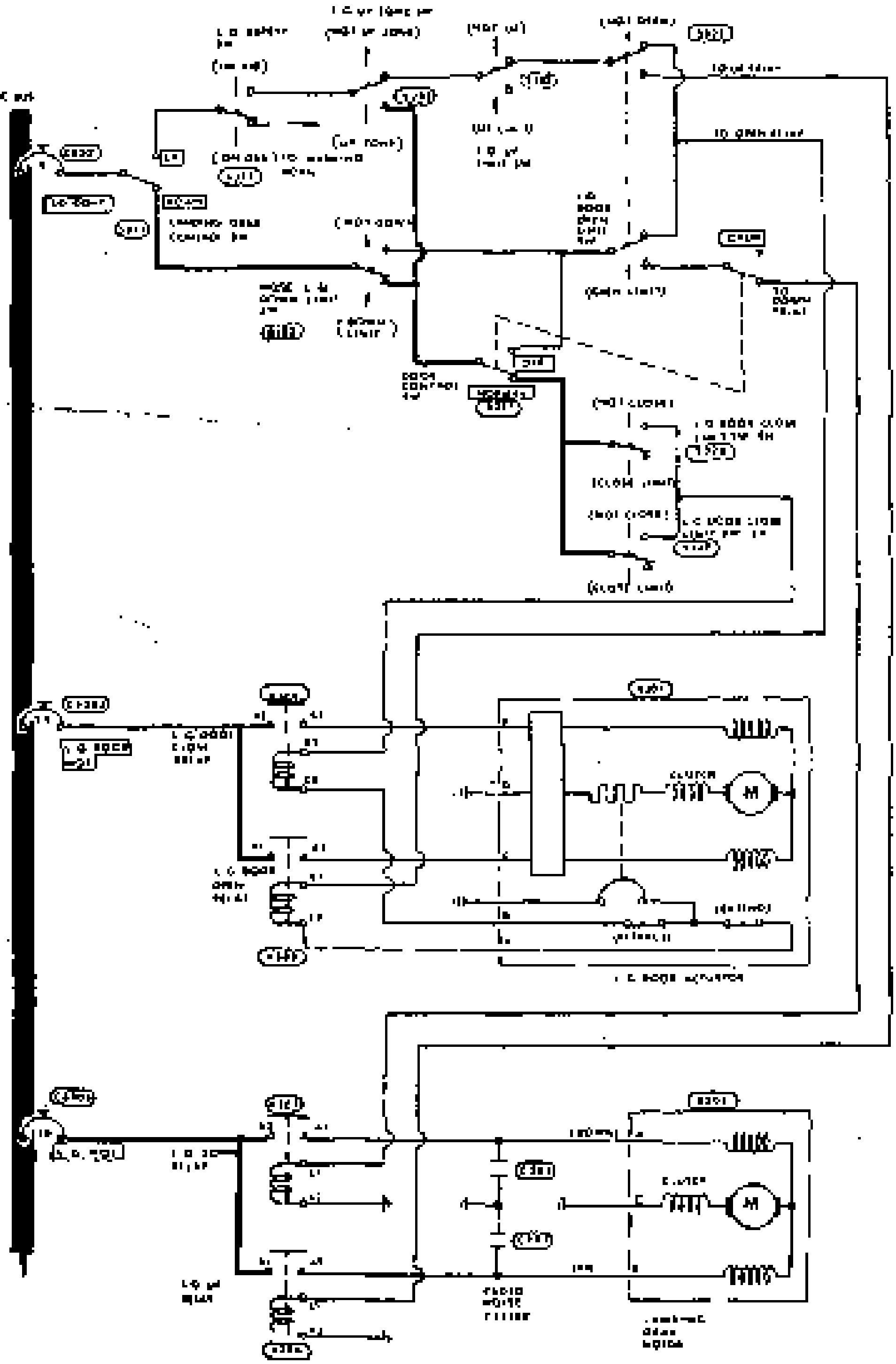
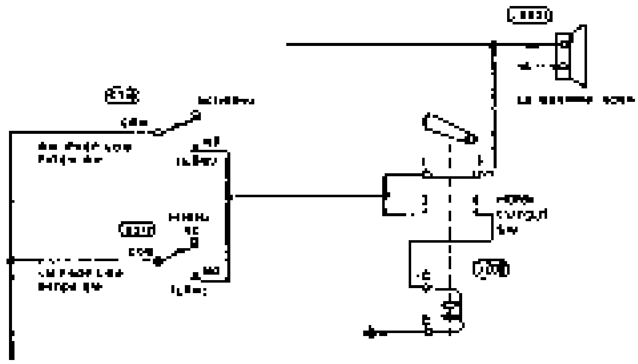
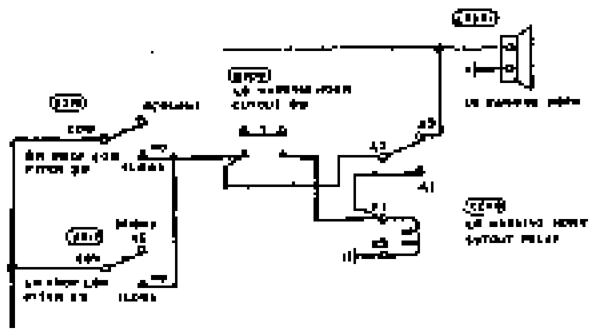


Fig 4-75 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-75 Landing gear system circuit (2/3) - (Position indicating system)

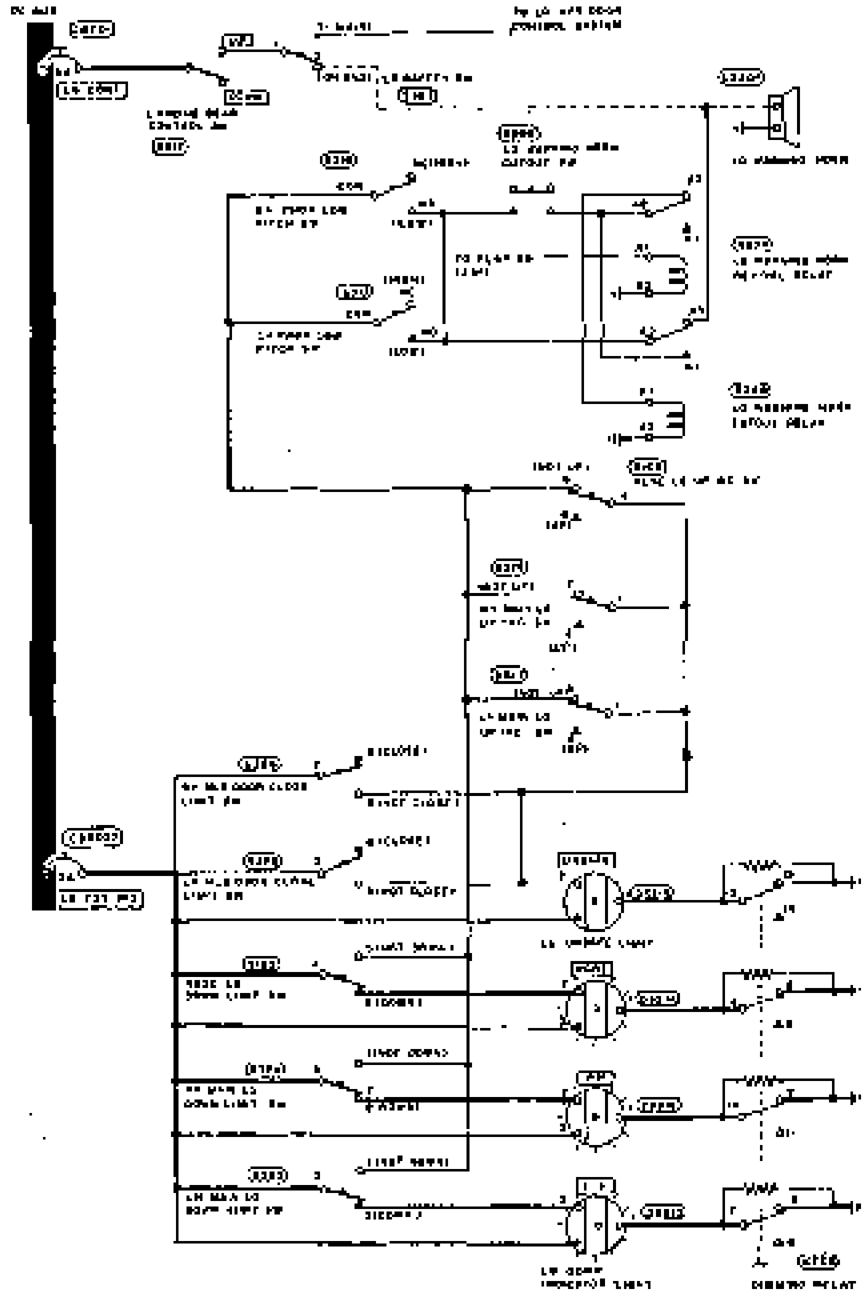


Fig 4-75 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.2 GEAR RETRACTION IN FLIGHT - DOOR OPEN

Landing gear doors are opened in the following sequence.

a. Switch ① is automatically turned when aircraft is in flight.

b. Switch ② is turned by pilot.

c. As a result of step b, power is supplied through thick line and relay ③ actuates.

d. As a result of step c, power is supplied through thick line and motor ④ actuates and opens doors.

e. At the same time when door is opened, switches ⑤ are turned, but power is not supplied, so they have no connection with the system operation.

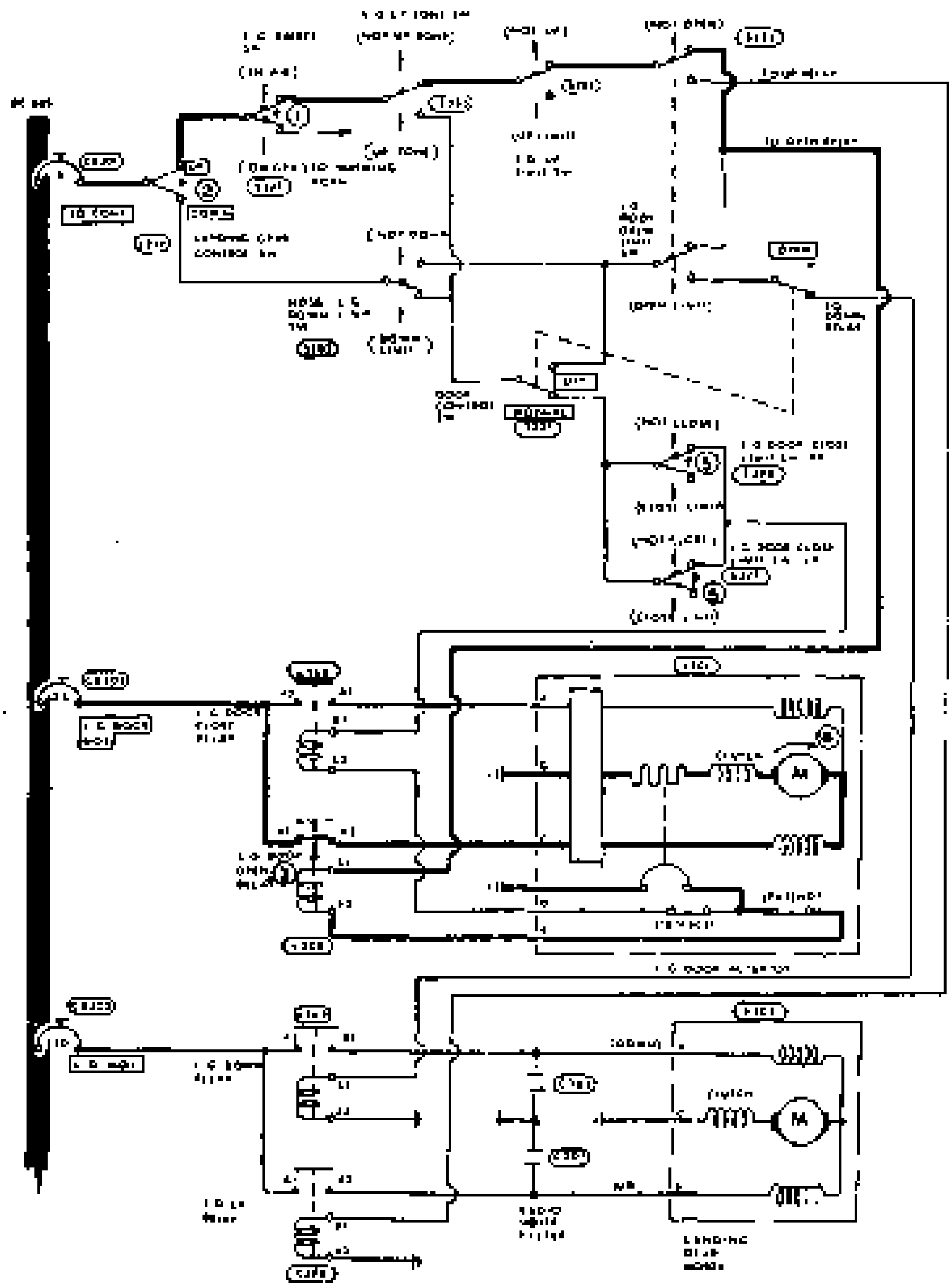
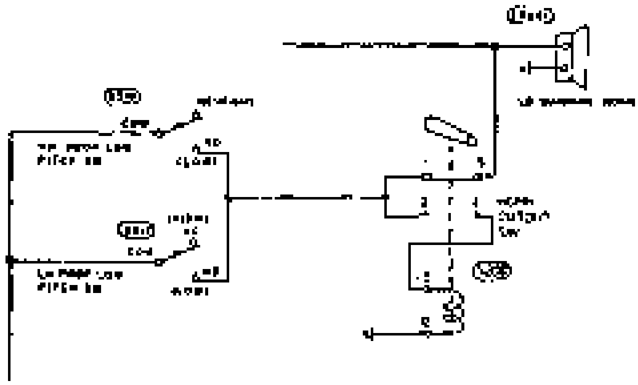
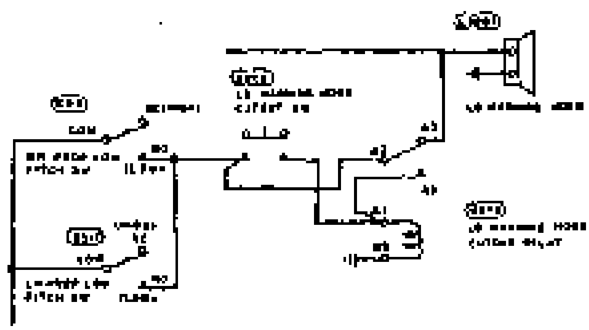


Fig 4-76 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-76 Landing gear system circuit (2/3) - (Position indicating system)

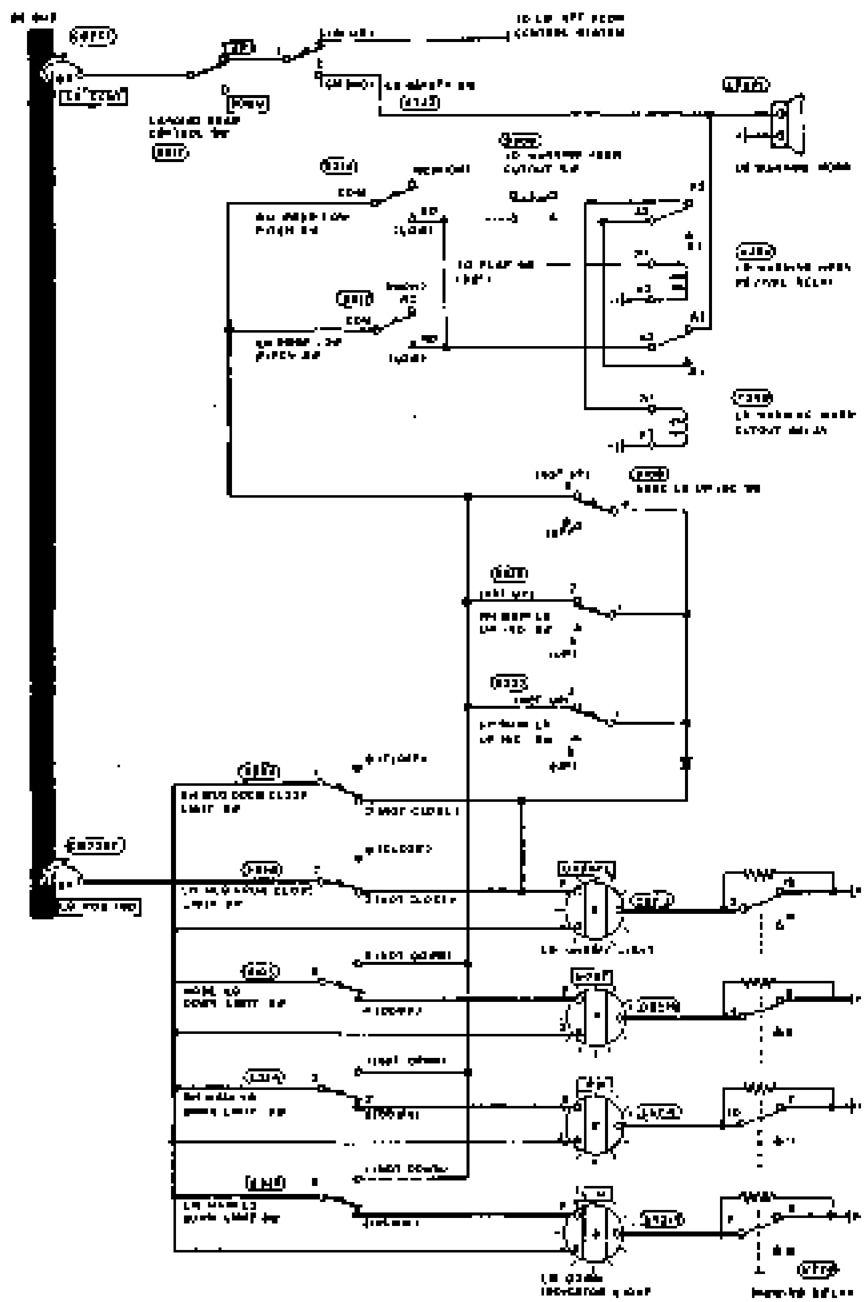


Fig 4-76 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.3 GEAR RETRACTION IN FLIGHT - GEAR UP

Landing gears are retracted in the following sequence.

a. When doors are fully opened, switches ① are turned automatically.

b. As a result of step a, power is cut off and relays ② become inoperative and door opening operation is stopped.

c. As a result of step a, power is supplied through thick line and relay ③ actuates.

d. As a result of step c, power is supplied through thick line and motor ④ actuates and retracts gears.

e. At the same time when gears begin to retract, switch ⑤ is turned, but power is not supplied, so it has no connection with the system operation.

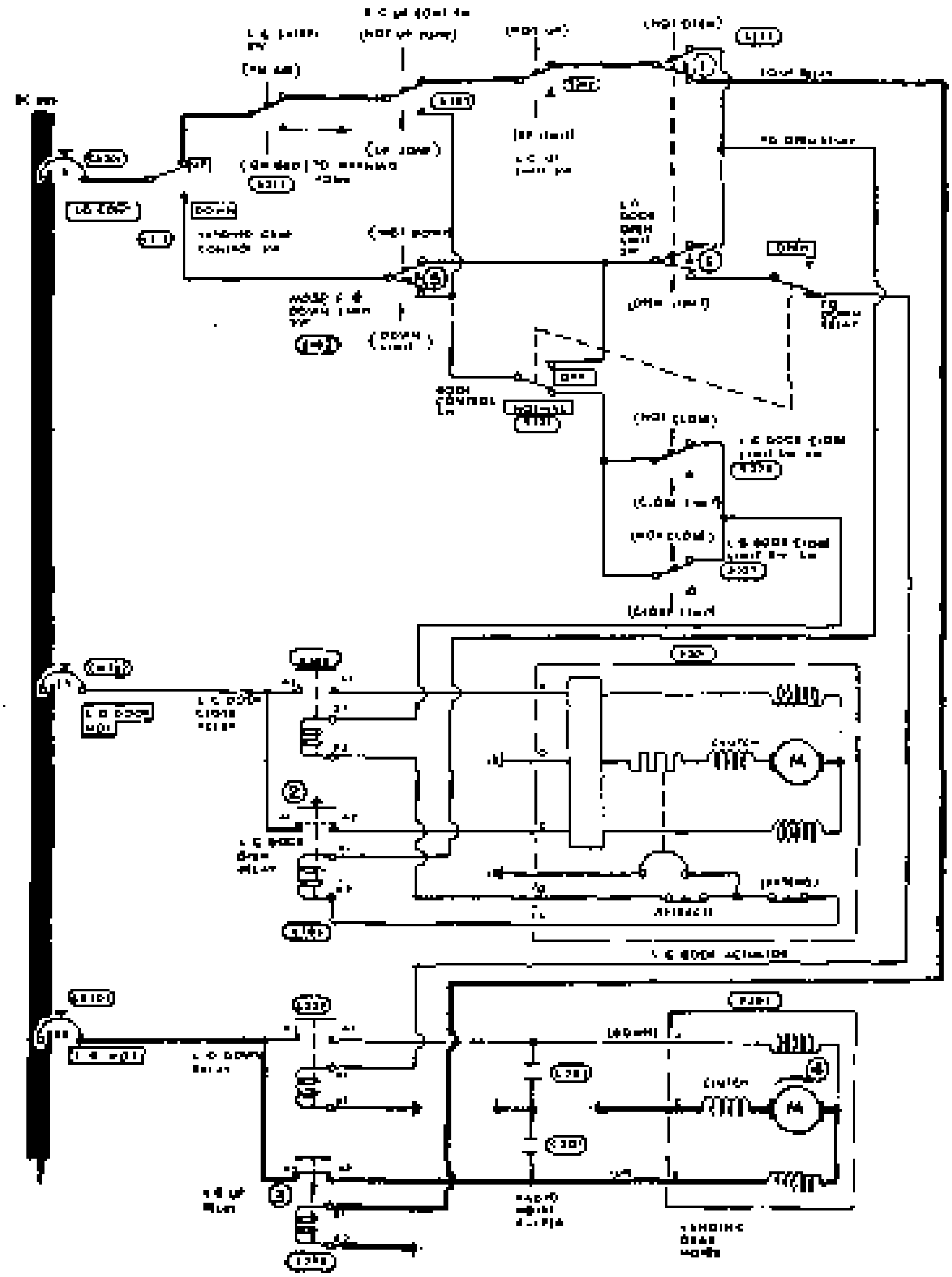
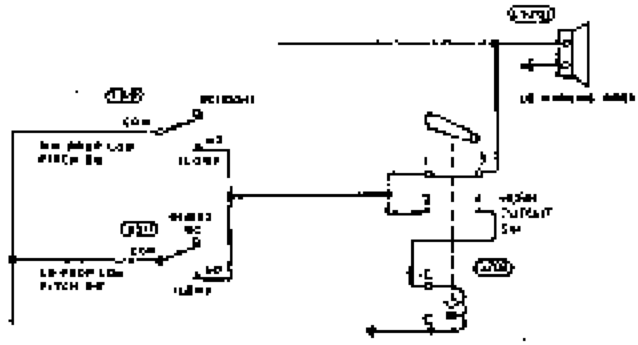
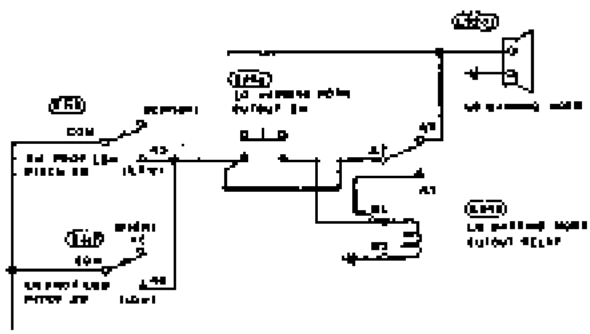


Fig 4-77 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 6525A & 6615A Unless Modified By SR015/32-002



S/N 6978A Thru 7178A Unless Modified By SR015/32-002

Fig 4-77 Landing gear system circuit (2/3) - (Position indicating system)

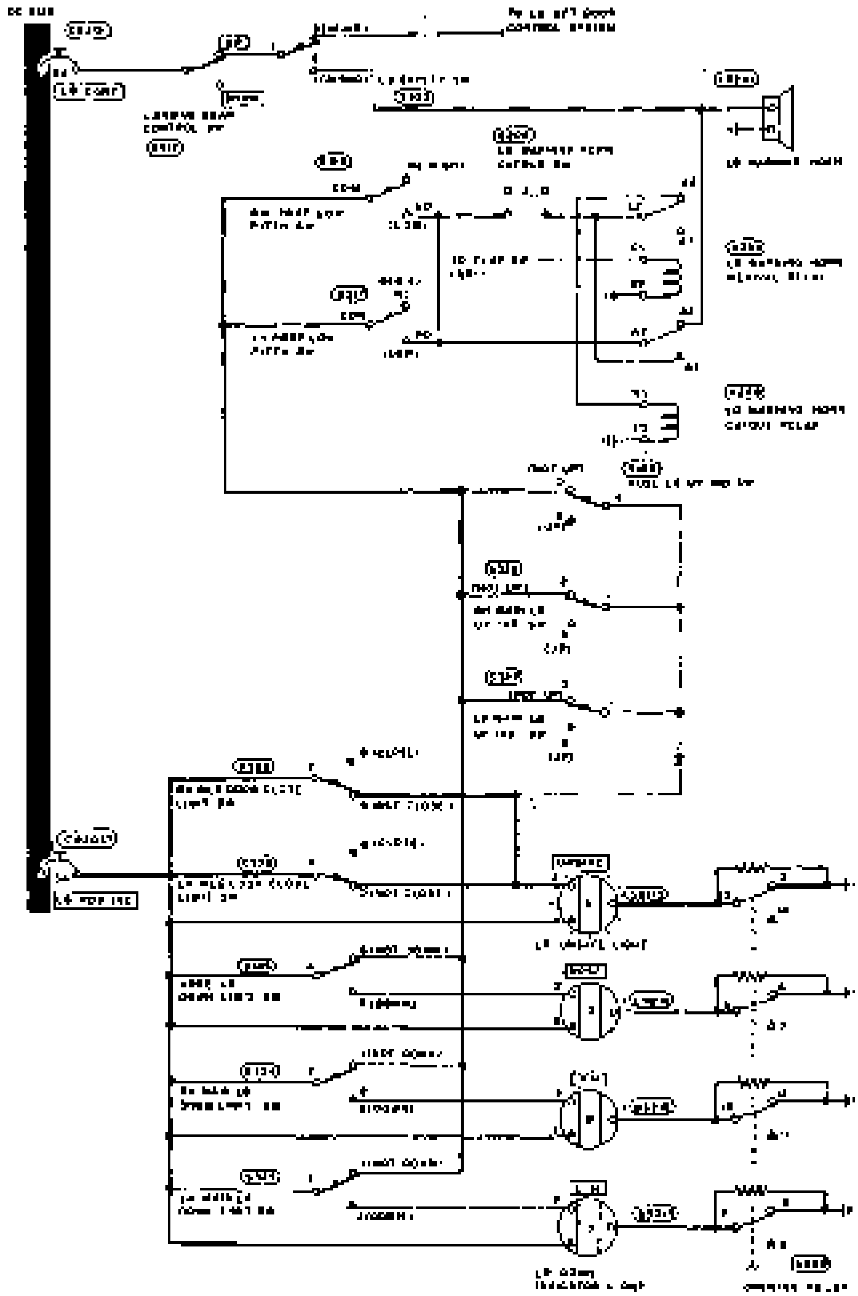


Fig 4-77 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.4 GEAR RETRACTION IN FLIGHT - DOOR CLOSE

Doors are closed in the following sequence.

a. When gear come to up position, switch ① is automatically turned.

b. As a result of step a, power is cut off and relay ② becomes inoperative and gear retraction stops.

c. Switches ③ is turned just before gears are operated by system inertia and stop.

d. As a result of step c, power is supplied through thick line and relay ④ actuates.

e. As a result of step d, power is supplied through thick line and motor ⑤ actuates.

f. Switches ⑥ are turned when doors begin to be closed, but power is not supplied, so they have no connection with the system operation.

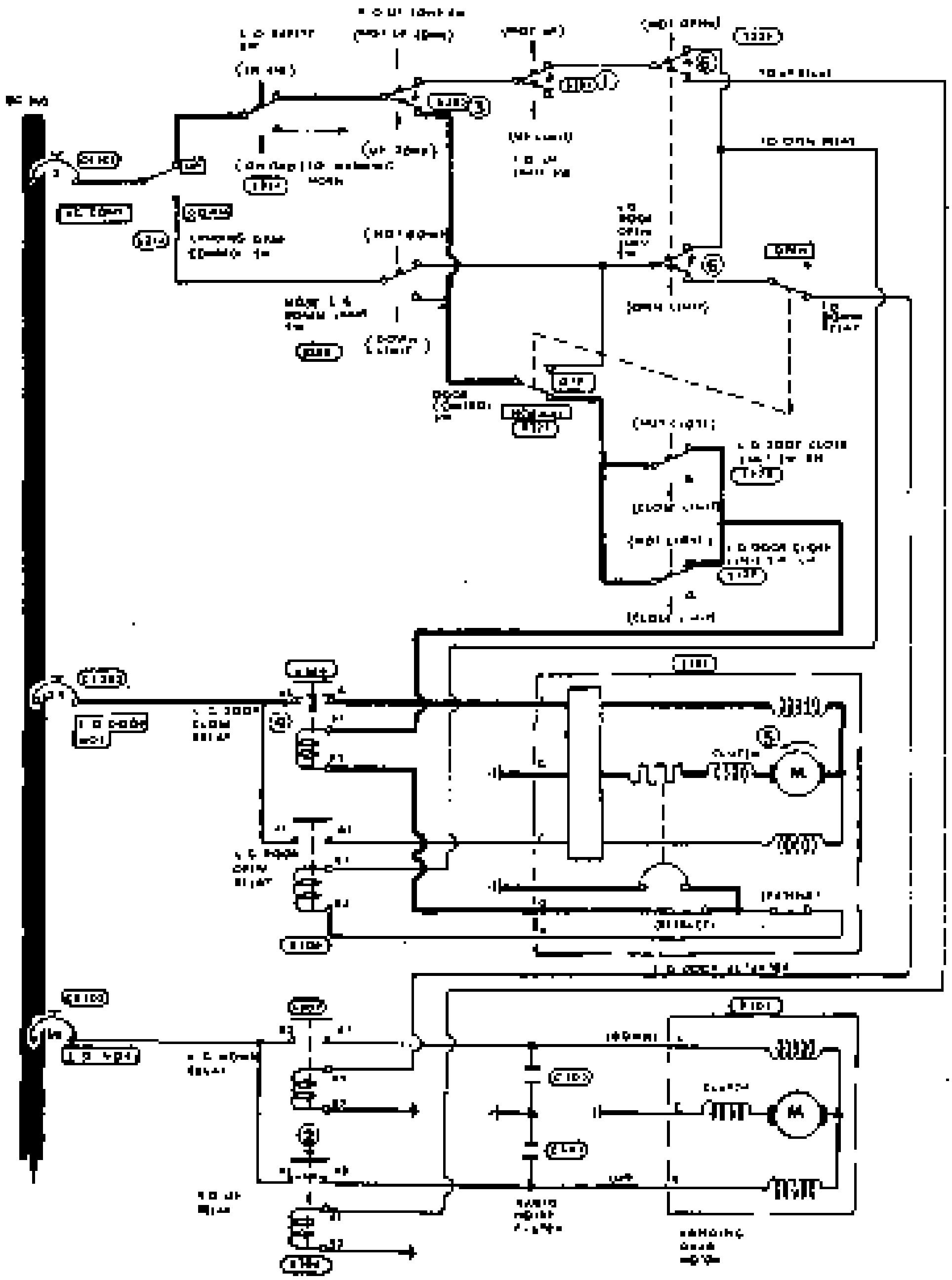
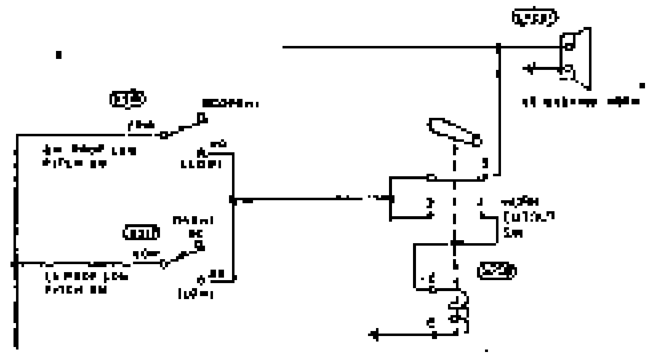
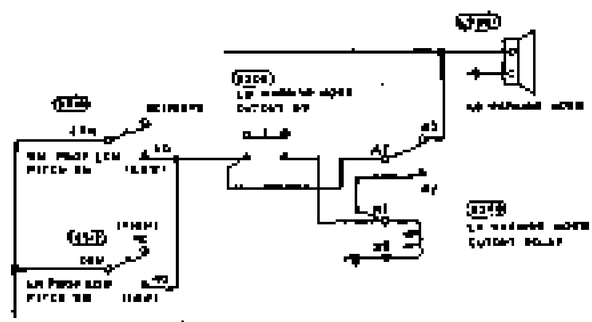


Fig 4-78 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-78 Landing gear system circuit (2/3) - (Position indicating system)

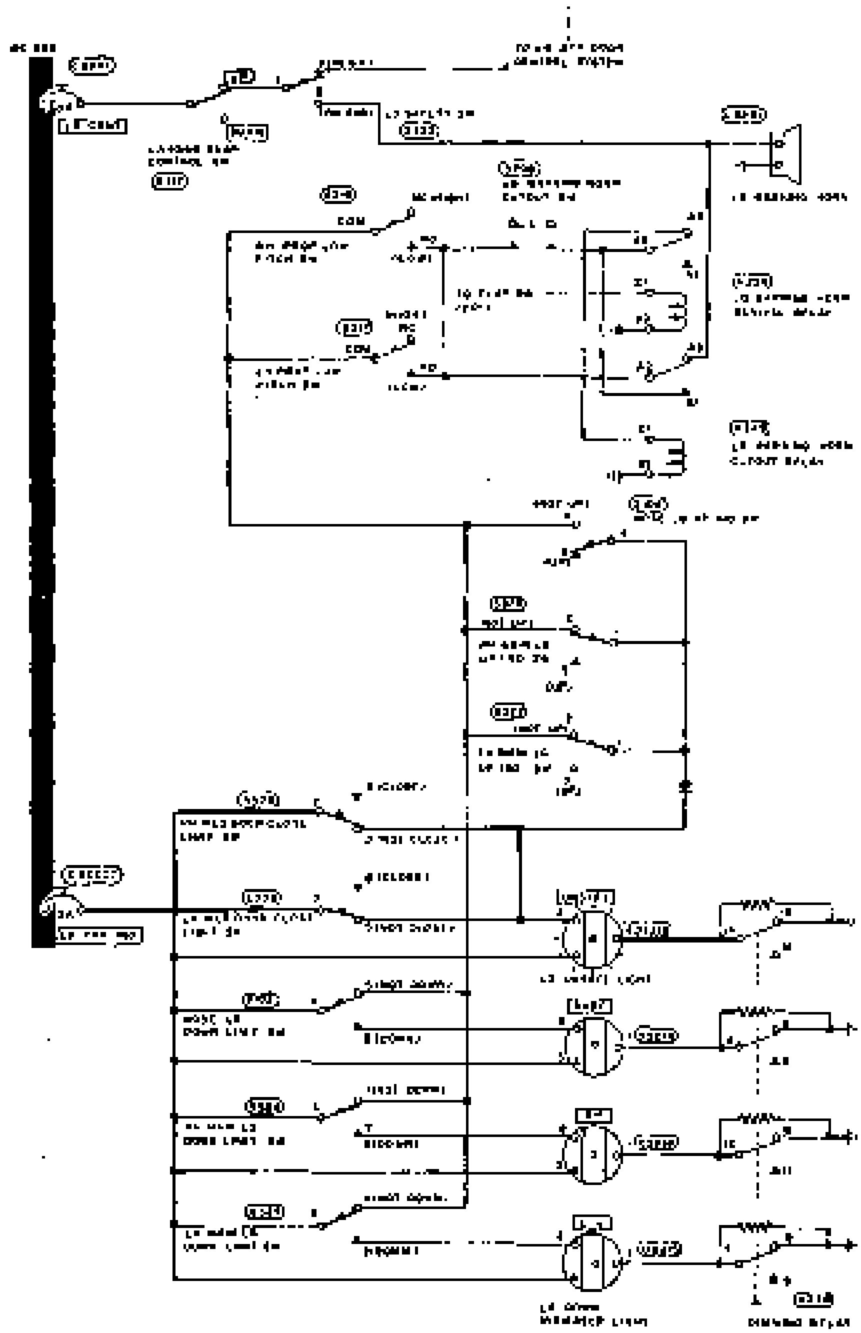


Fig 4-78 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.5 GEAR RETRACTION STOP IN FLIGHT

Gear up operation is completed in the following sequence and all power is cut off by relays and switches, so the system becomes inoperative.

a. Switches ① are turned automatically when doors are closed completely.

b. As a result of step a, power is cut off and relay ② becomes inoperative and door closing operation is stopped.

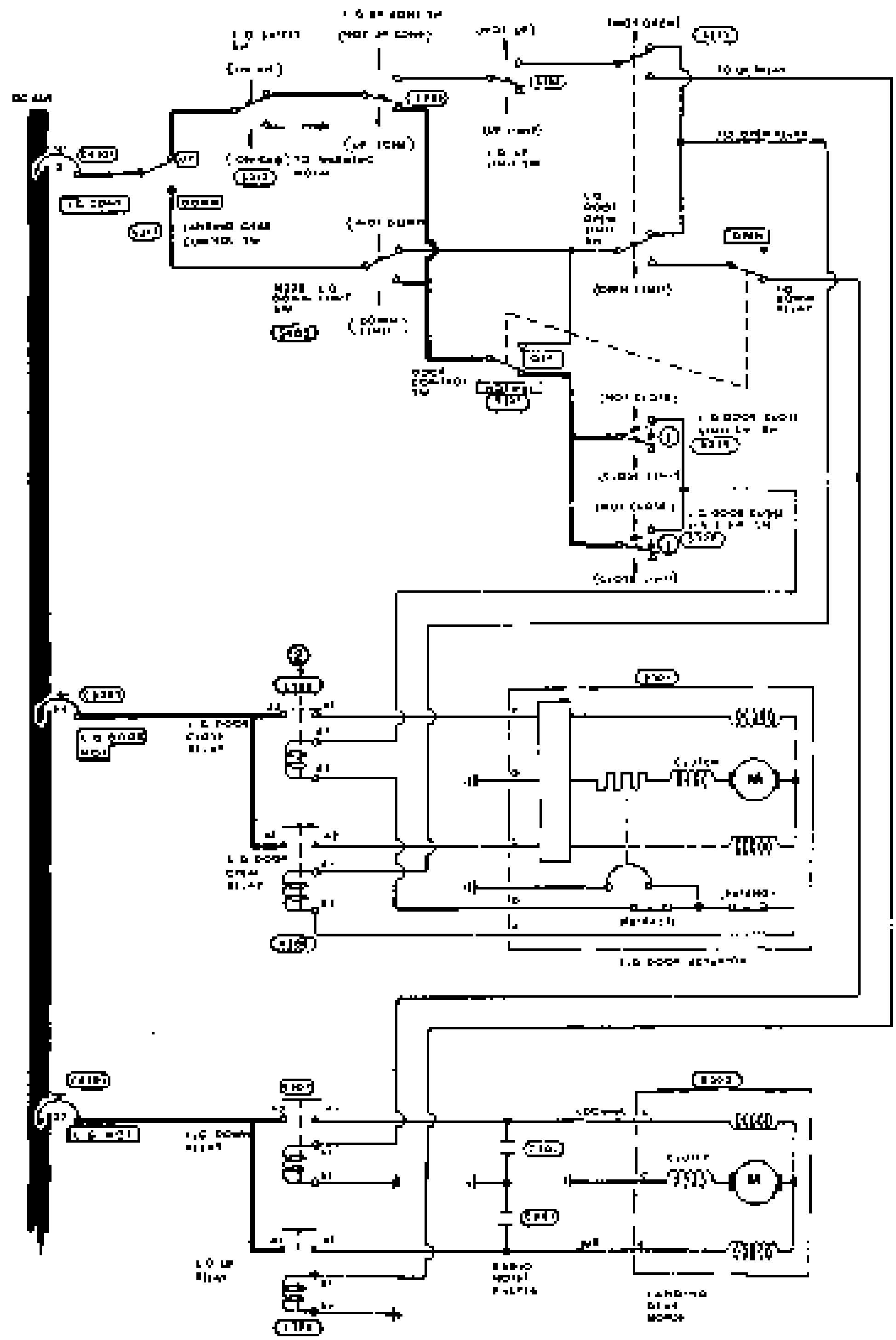
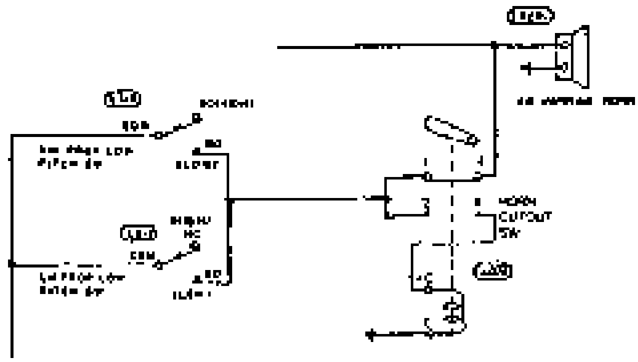
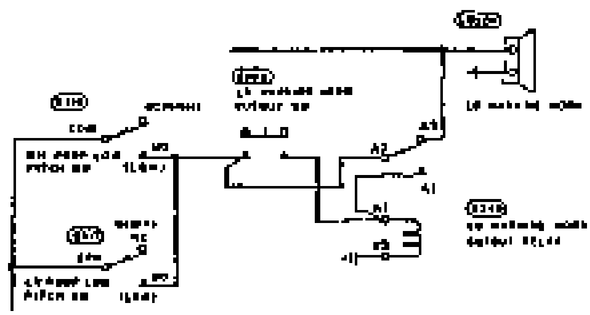


Fig 4-79 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-79 Landing gear system circuit (2/3) - (Position indicating system)

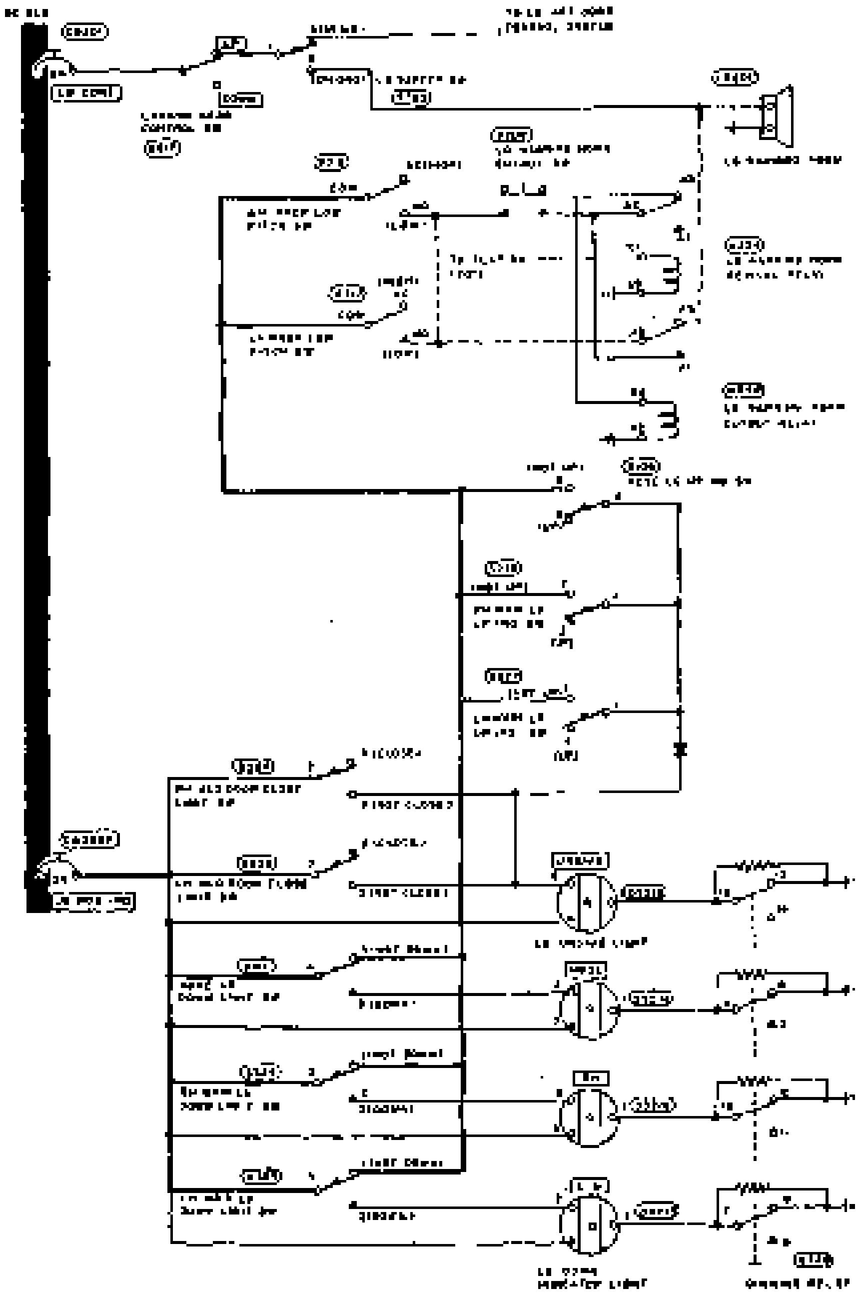


Fig 4-79 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.6 GEAR EXTENSION IN FLIGHT - DOOR OPEN

Doors are opened in the following sequence.

a. Switch ① is turned by pilot.

b. As a result of step a, power is supplied through thick line and relay ② actuates and doors are opened.

c. As a result of step b, power is supplied through thick line and motor ③ actuates and opens doors.

d. Switches ④ are turned when doors begin to be opened, but power is not supplied, so they have no connection with the system operation.

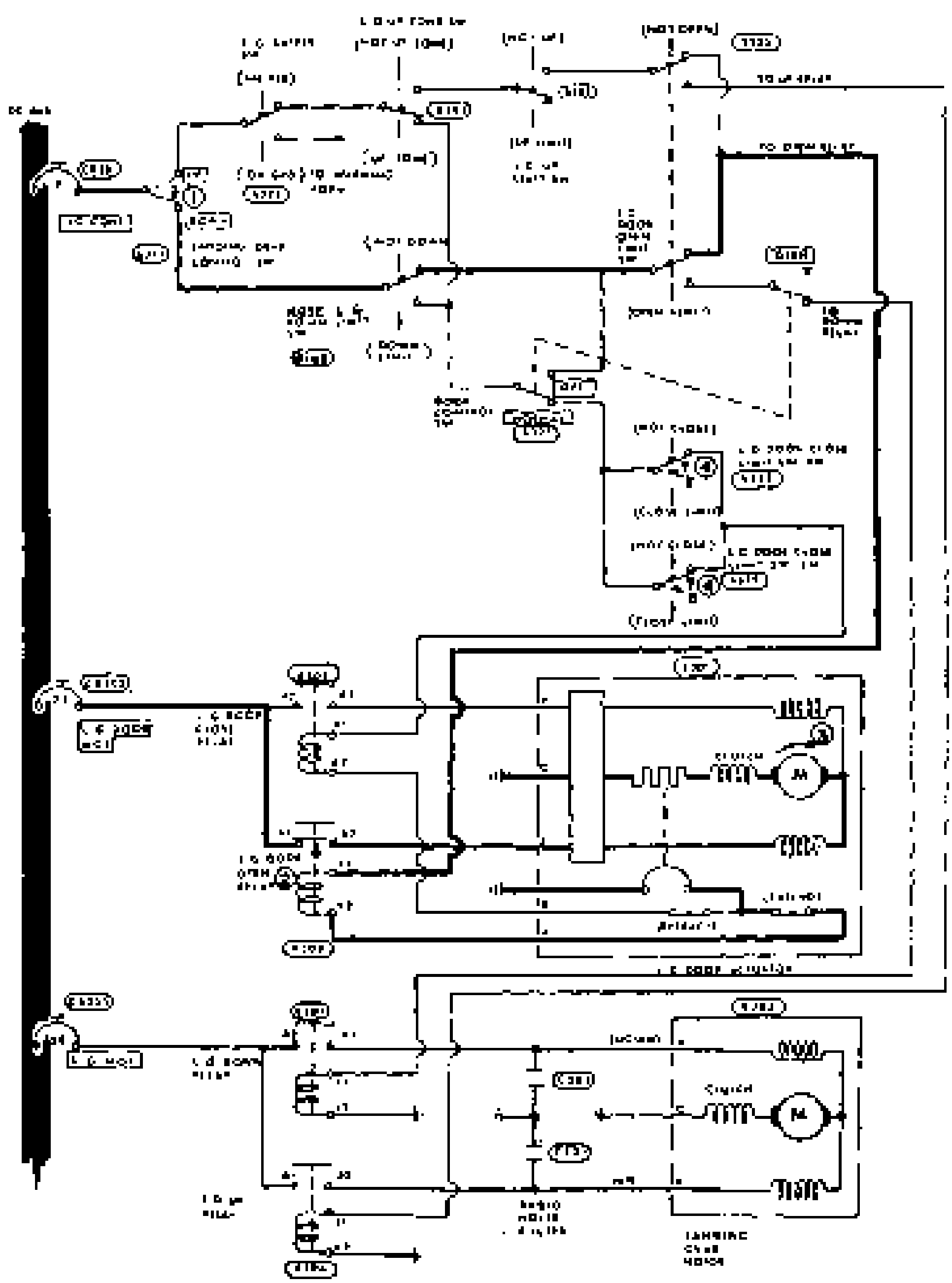
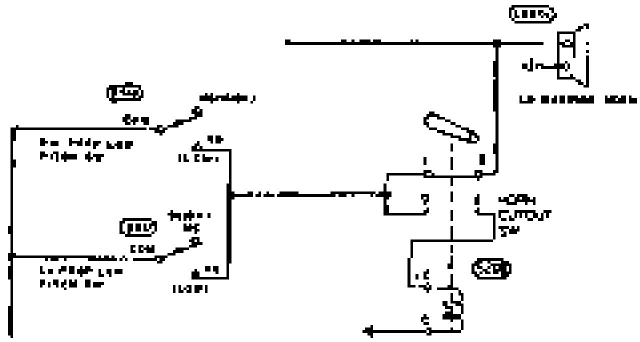
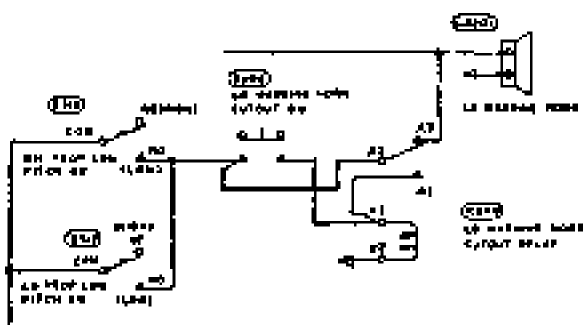


Fig 4-80 Landing gear system circuit (1/1)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-80 Landing gear system circuit (2/3) - (Position indicating system)

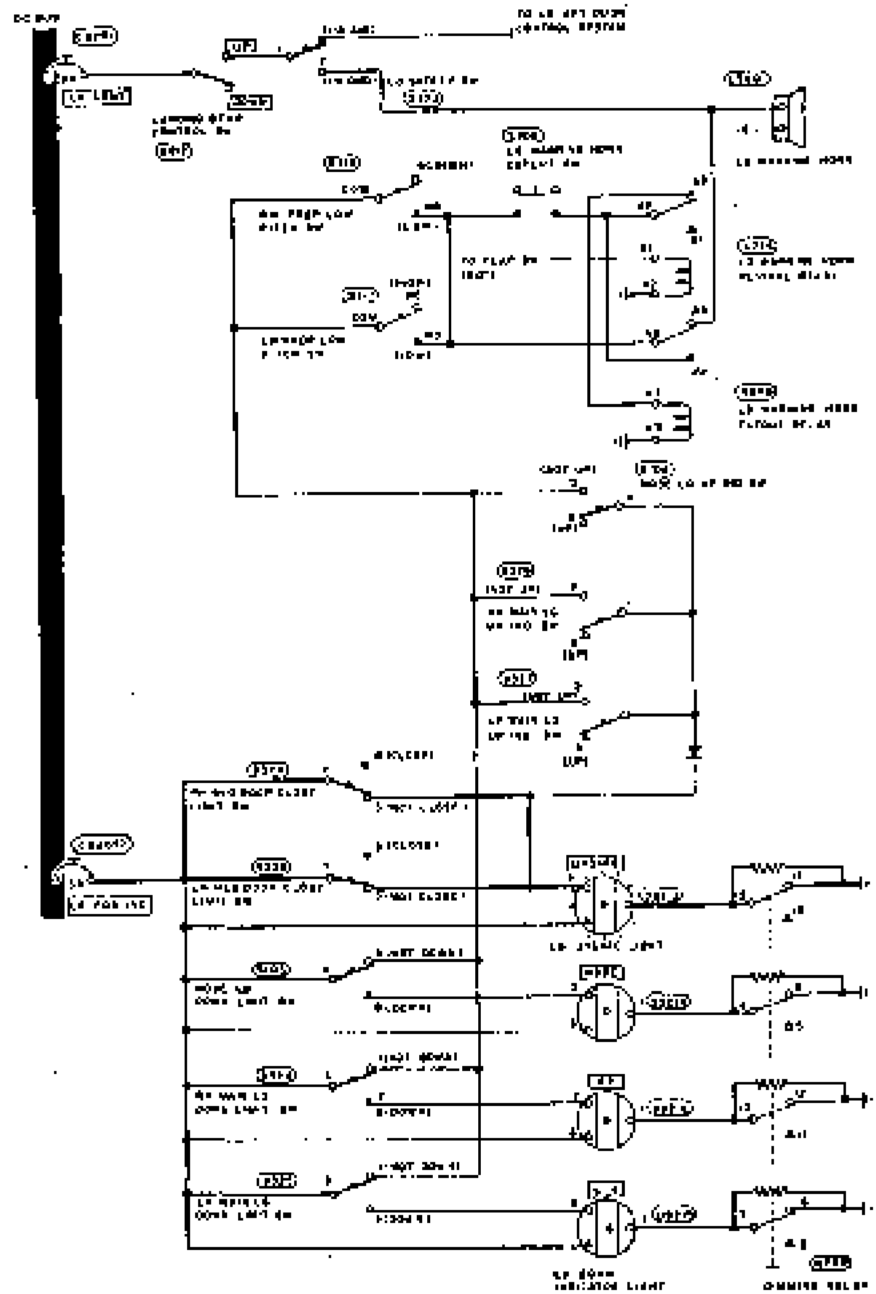


Fig 4-80 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.7 GEAR EXTENSION IN FLIGHT - GEAR DOWN

Landing gears are extended in the following sequence.

a. Switches ① are automatically turned when doors are completely opened.

b. As a result of step a, power is cut off and relay ② becomes inoperative and door opening operation stops.

c. As a result of step a, power is supplied through thick line and relay ③ actuates.

d. As a result of step c, power is supplied through thick line and motor ④ actuates and extends the gears.

e. Switch ⑤ is turned when the gears begin to be extended, but power is not supplied, so it has no connection with the system operation.

f. Switch ⑥ is turned following switch ⑤, but power is not supplied, so it has no connection with the system operation.

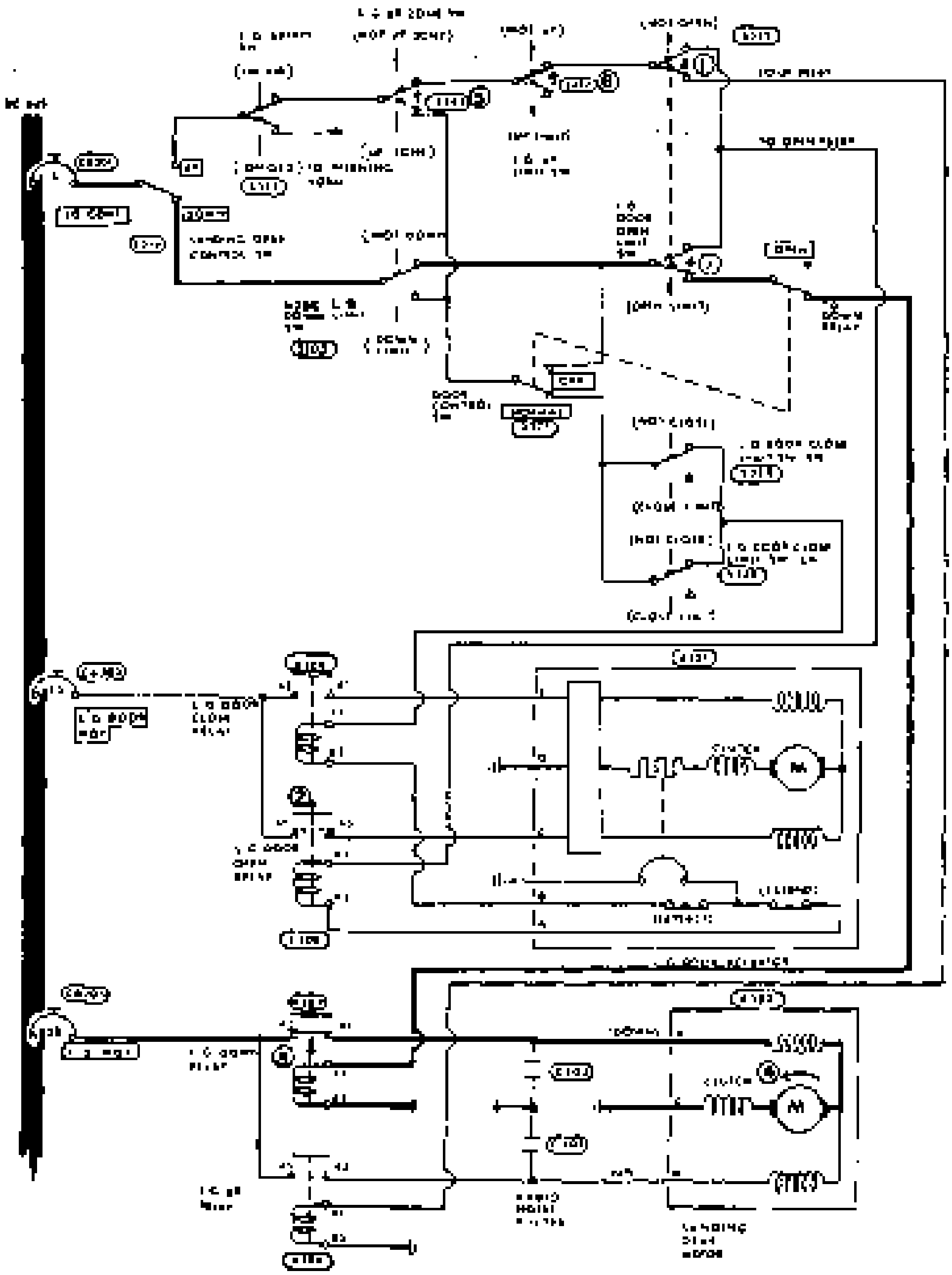
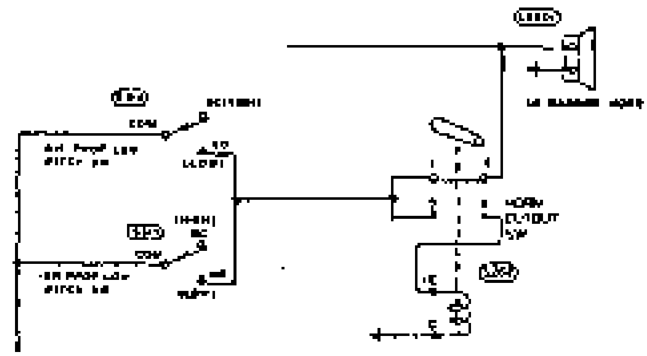
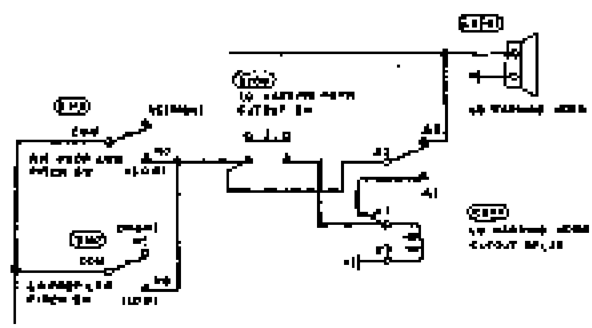


Fig 4-81 Landing gear system circuit (1/3)

____(Landing gear control circuit)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-81 Landing gear system circuit (2/3) - (Position indicating system)

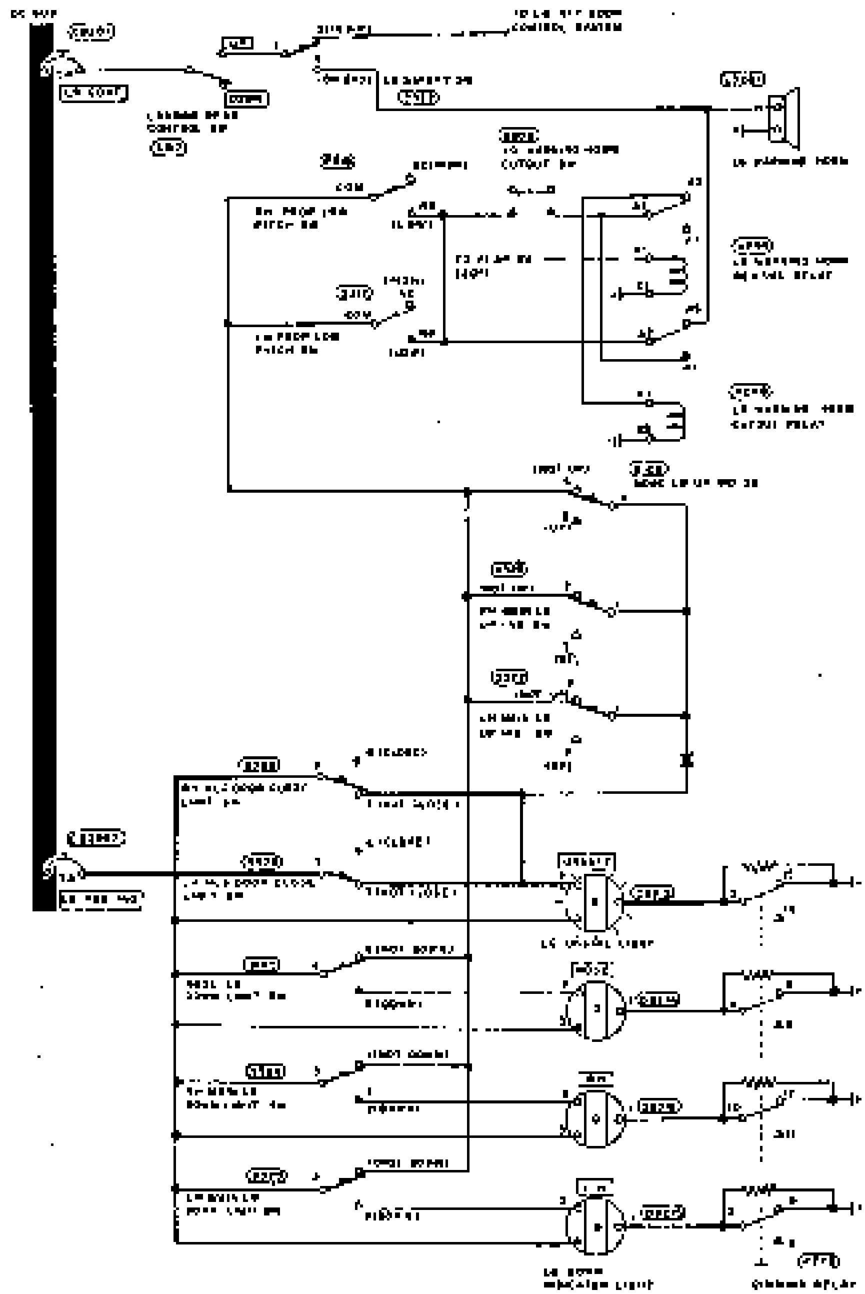


Fig 4-81 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.8 GEAR EXTENSION IN FLIGHT – DOOR CLOSE

Doors are closed in the following sequence.

a. Switch ① is turned automatically when the gears are completely extended.

b. As a result of step a, power is cut off and relay ② becomes inoperative and gear extending operation stops.

c. As a result of step a, power is supplied through thick line and relay ③ actuates.

d. As a result of step c, power is supplied through thick line and motor ④ actuates and closes doors.

e. Switches ⑤ are turned when doors begin to be closed, but power is not supplied, so they have no connection with the system operation.

f. Switches ⑥ are turned when doors are closed completely, so relay ③ and motor ④ become inoperative, and door closing operation stops.

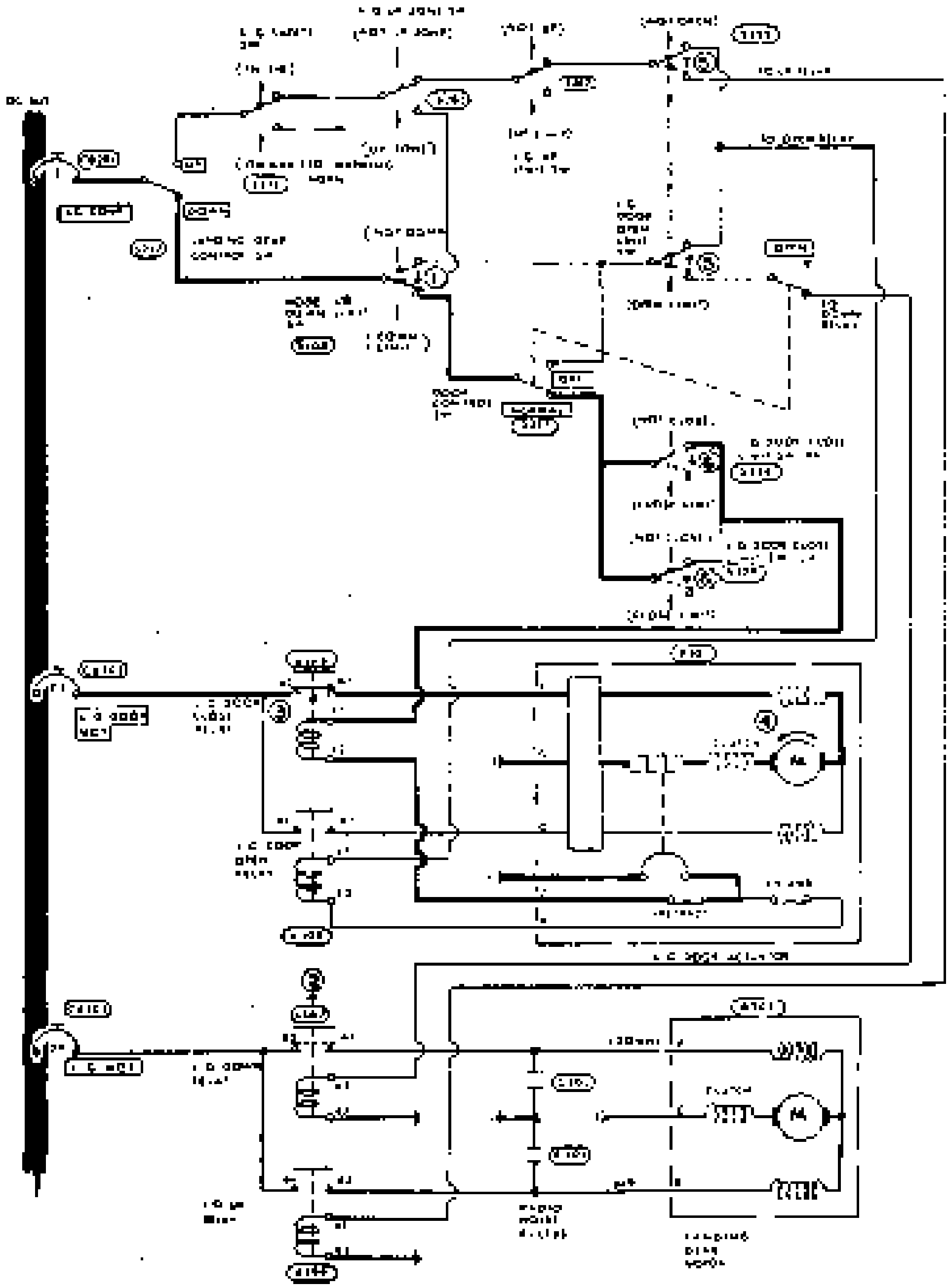
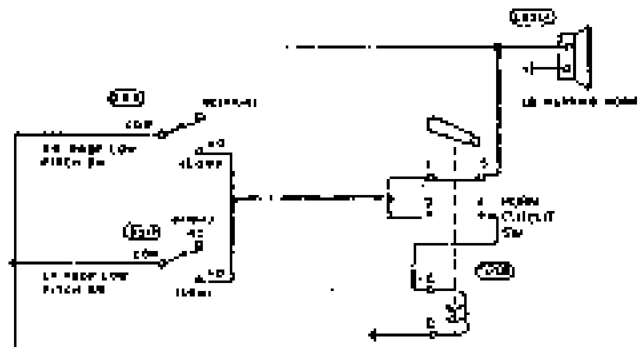
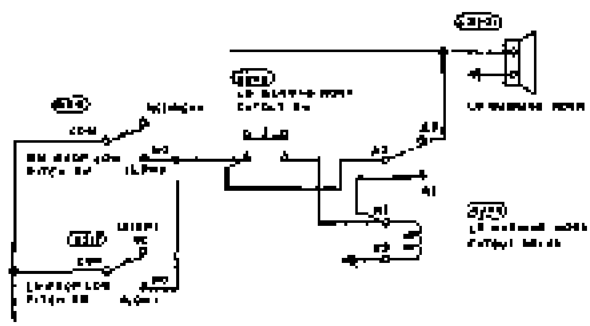


Fig 4-82 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-82 Landing gear system circuit (2/3) - (Position indicating system)

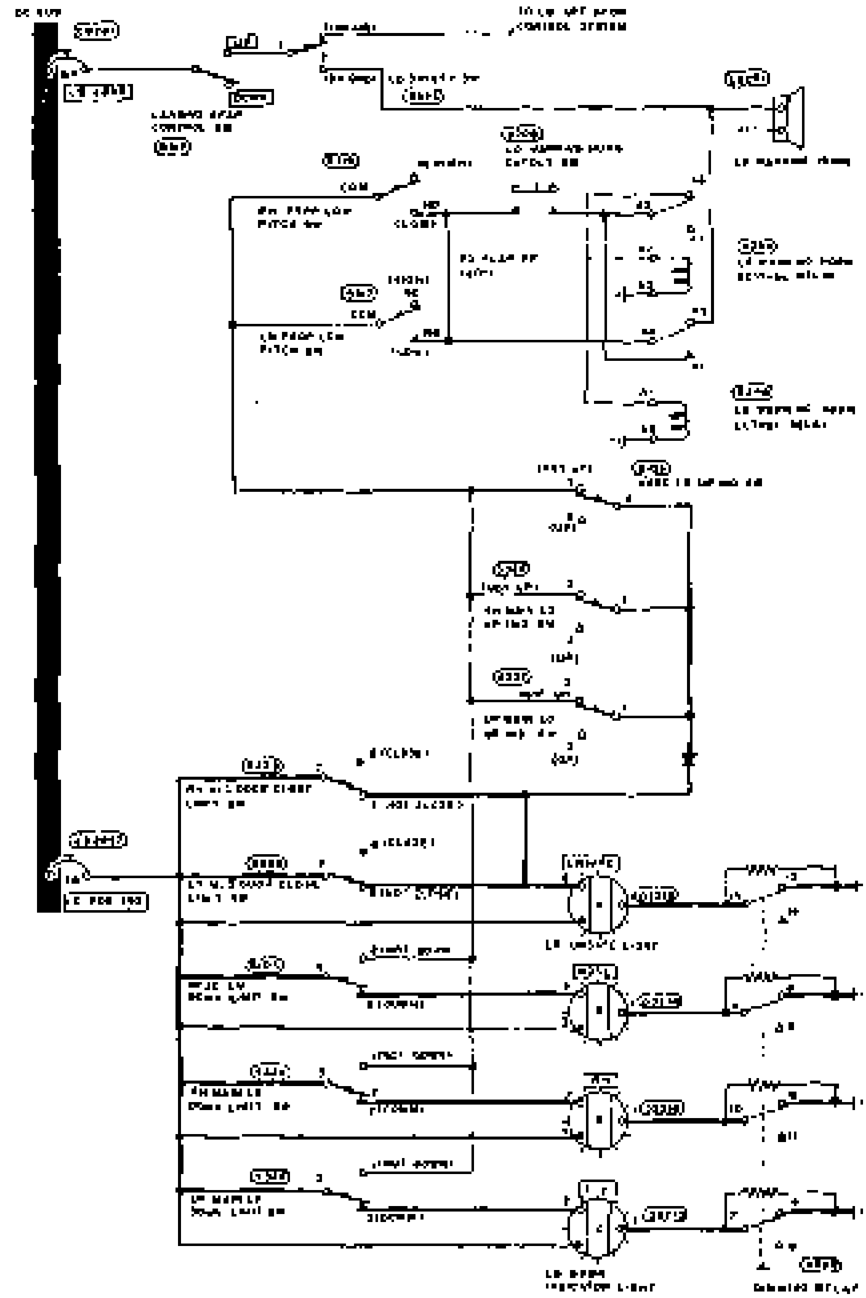


Fig 4-82 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

11.9 DOOR OPERATION ON GROUND

Doors are opened in the following sequence.

a. Switch ① is turned only when mechanic is operating. When releasing his hand, subsequent operation stops and remains as it is.

b. As a result of step a, power is supplied through thick line and relay ② actuates.

c. As a result of step b, power is supplied through thick line and motor ⑤ actuates and opens doors.

d. Switches ④ are turned when doors begin to be opened, but power is not connected, so they have no connection with the system operation.

e. Switch ⑤ is automatically turned when doors are fully opened, and door opening operation stops. Door closing operation is same as Para 11.8A.

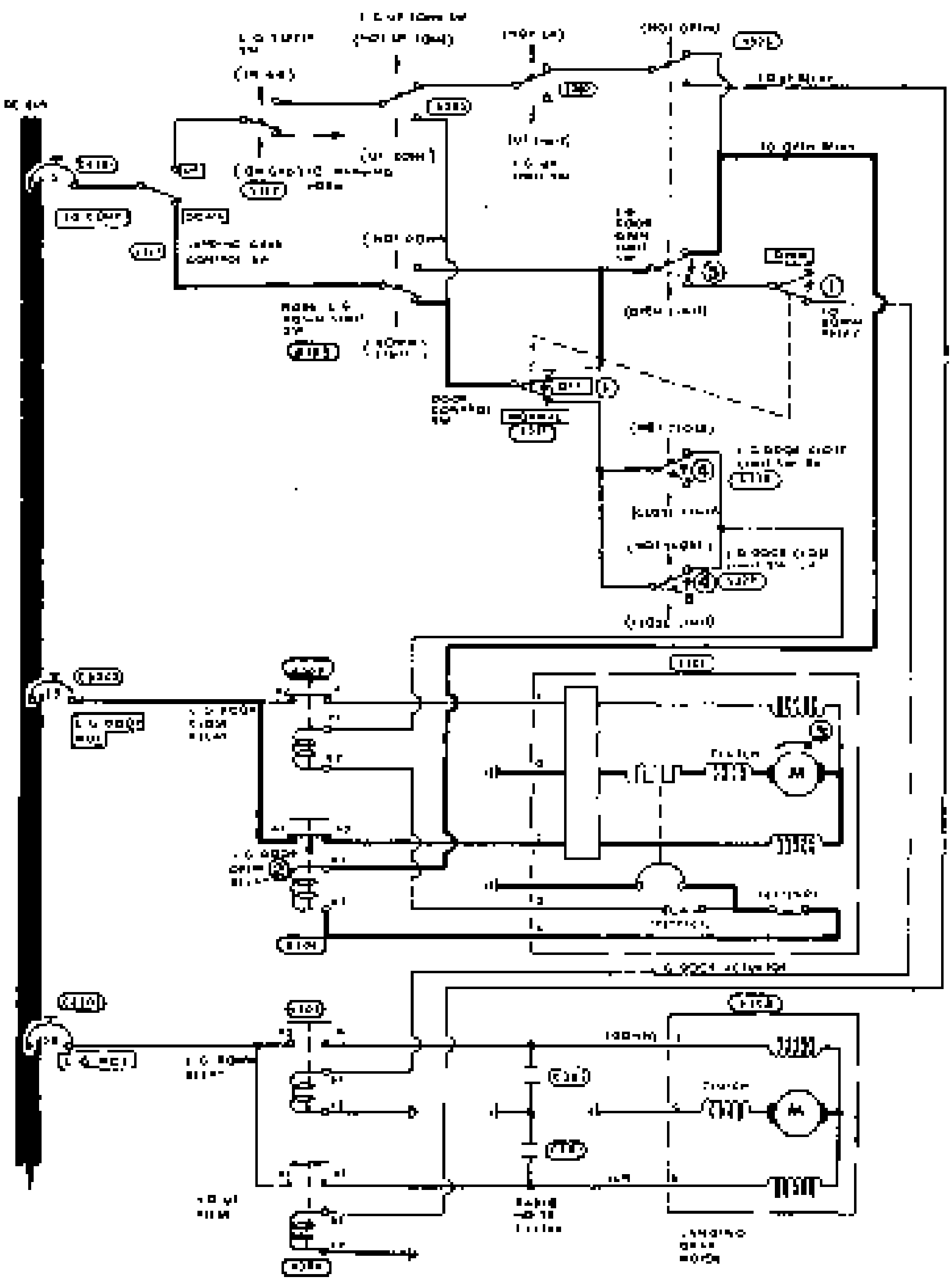
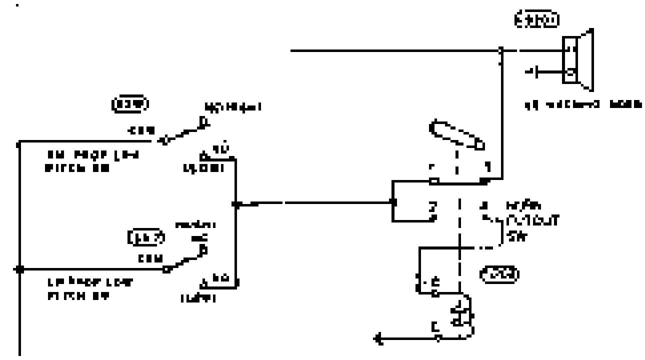
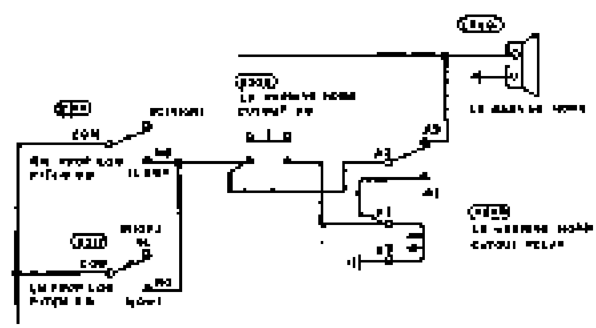


Fig 4-83 Landing gear system circuit (1/3)
(Landing gear control system)



S/N 652SA & 661SA Unless Modified By SR015/32-002



S/N 697SA Thru 717SA Unless Modified By SR015/32-002

Fig 4-83 Landing gear system circuit (2/3) - (Position indicating system)

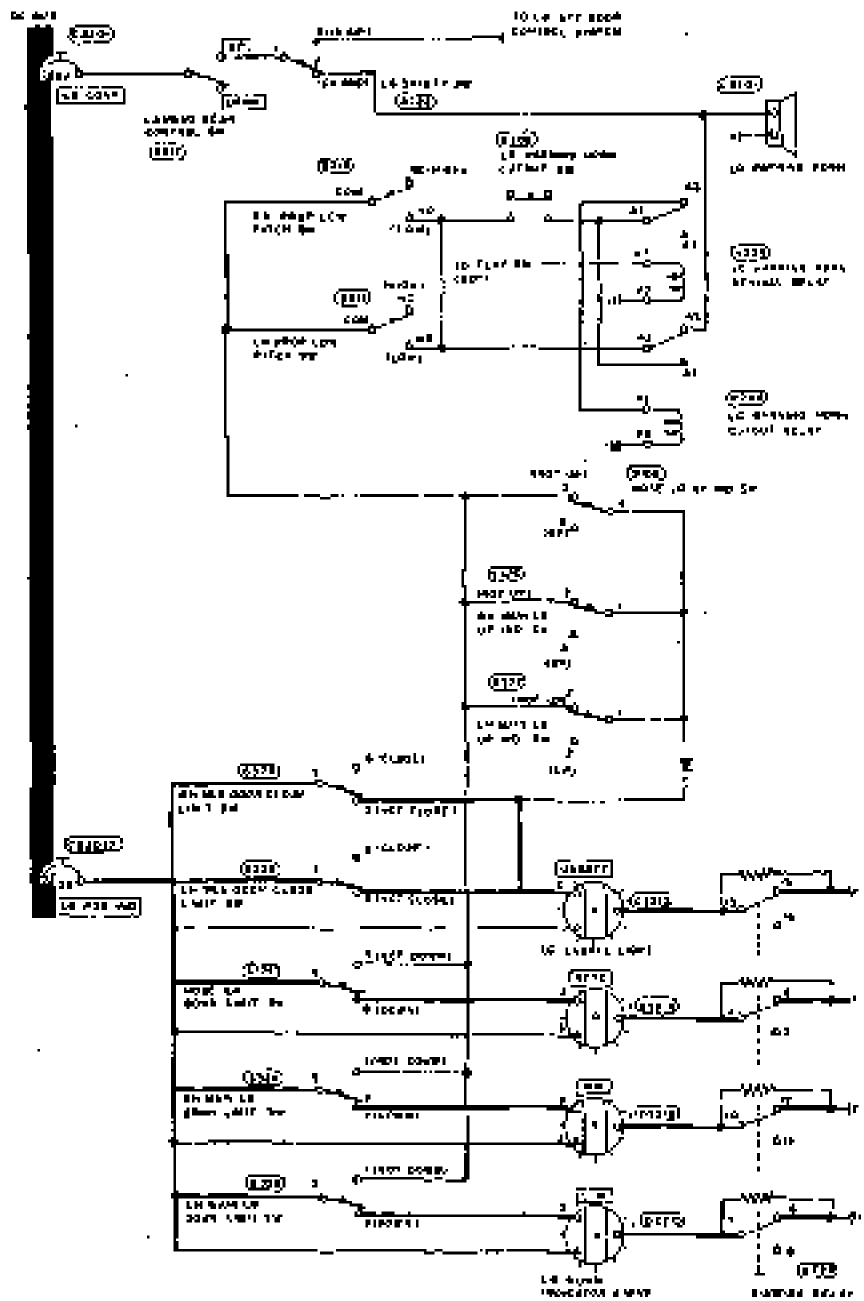


Fig 4-83 Landing gear system circuit (3/3) - (Position indicating system)
S/N 718SA thru 730SA, and S/N's Modified By SR015/32-002

12. TROUBLE SHOOTING LANDING GEAR SYSTEM AND BRAKE SYSTEM

Trouble	Probable Cause	Remedy
Main gear forward door does not open in gear up operation	a. Faulty circuit breaker "LG CONT" or "DOOR MOTOR".	Replace.
	b. Functional defect of landing gear control switch.	Replace.
	c. Failure of main gear forward door actuator.	Replace.
	d. Door open relay does not operate.	Check or replace.
	e. Open wire or loose terminal of door open circuit.	
	f. Functional defect or poor adjustment of switch in door open circuit.	Replace or readjustment.
	g. Main gear forward door hook is not released.	Check and readjust.
Main gear forward door opens, but landing gear is not retracted	a. Faulty circuit breaker LG MOTOR.	Replace.
	b. Landing gear up relay does not operate.	Check or replace.
	c. Failure of landing gear motor.	Replace.
	d. Faulty torque limiter.	Replace
	e. Open wire or loose terminal of door open circuit.	Check and repair.
	f. Functional defect or poor adjustment of switch in gear up circuit.	Replace or readjustment.
	h. Nose gear down lock sticks.	Readjustment or replace.
	i. Mechanical stop moving nut sticks on "DOWN" side.	Readjustment.
	j. Failure of nose gear actuator.	Replace.
Landing gear is retracted, but main gear forward door does not close	a. Door close relay does not operate.	Check or replace.
	b. Failure of main gear forward door actuator.	Replace.
	c. Functional defect or poor adjustment of up zone switch.	Replace.
	d. Open wire or loose terminal of door close circuit.	Check and repair.
	e. Functional defect or poor adjustment of switch in door close circuit.	Replace or readjustment.
	f. Ground door open switch is not set to "CLOSE" side.	Set the switch to "CLOSE" side.

Trouble	Probable Cause	Remedy
	g. Shear of shear pin in main gear forward door mechanism.	Check and replace.
	h. Functional defect of up zone door safety limit switch.	Replace or readjustment.
Main gear forward door does not open in gear down operation.	a. Functional defect or poor adjustment.	Replace or readjustment.
	b. Door open relay does not operate.	Replace.
	c. Failure of main gear forward door actuator.	Replace.
	d. Open wire or loose terminal of door open circuit.	Check and repair.
	e. Main gear forward door hook is not released.	Check and readjust.
Main gear forward door opens, but gear does not extend	a. Landing gear down relay does not operate.	Check or replace.
	b. Internal defect of landing gear motor.	Replace.
	c. Open wire or loose terminal or gear down circuit.	Check and repair.
	d. Functional defect or poor adjustment of gear down circuit.	Replace or adjust.
	e. Mechanical stop moving nut sticks on "UP" side.	Readjust.
	f. Faulty torque limiter.	Replace.
	g. Main gear aft door hook is not released.	Check and readjust.
Landing gear extends, but main gear forward door does not close	a. Functional defect or poor adjustment of down zone limit switch.	Replace or readjust.
	b. Door close relay does not operate.	Replace.
	c. Failure of main gear forward door actuator.	Replace.
	d. Open wire or loose terminal door close circuit.	Check and repair.
	e. Ground door open switch is set to OPEN side.	Set the switch to "CLOSE" side.
	f. Shear of shear pin in main gear forward door mechanism.	Check and replace.
Unsafe light comes on in gear up condition in flight.	a. Faulty circuit breaker POS IND.	Check or replace.
	b. Nose gear up indicator switch does not operate.	Check or replace.
	c. LH and RH main gear up indicator switches do not operate.	Check or replace.
	d. LH and RH main gear forward door close limit switches do not operate.	Check or replace.
	e. Faulty unsafe light test switch	Check or replace.

Trouble	Probable Cause	Remedy
Unsafe light comes on in gear down condition in flight.	a. Faulty circuit breaker "POS IND"	Check or replace.
	b. LH and RH main gear forward door close limit switches do not operate.	Check or replace.
	c. Nose gear up indicator switch does not operate.	Check or replace.
	d. LH and RH main gear up indicator switches do not operate	Check or replace.
	e. Nose gear down limit switch does not operate.	Check or replace
	f. LH and RH main gear down indicator switches do not operate.	Check or replace.
	g. Poor engagement of nose gear down lock	Check or replace.
Shear pin is sheared off	a. Main gear forward door open limit switch does not operate	Check or replace.
	b. Main gear forward door close limit switch does not operate	Check or replace
	c. Door hook is not released when main gear forward door is opened.	Check or replace
Shimmy or vibration in taxiing	a. Air is mixed in shimmy damper.	Bleed air.
	b. Unbalance assembly of nose wheel	Readjustment.
	c. Weariness or looseness of torque link pin or shimmy damper attaching pin.	Readjustment or replace
Abnormal noise in wheel in taxiing	a. Damaged bearing	Check or replace.
	b. Poor lubrication.	Relubrication and check.
	c. Excessive torque for bearing.	Readjustment.
Equal depression on brake pedals does not produce uniform braking. Insufficient braking effect.	a. Oily lining	Clean lining
	b. Excessive wearness of lining.	Replace lining.
	c. Air is mixed in the system.	Bleed air.
	d. Operational defect of master cylinder.	Check or replace.
	e. Blockage in the system	Drain, charge fluid and clean the system.
	f. Deterioration of hose	Replace
	g. Insufficient fitting of disc or lining immediately after replacement	Work brake several times in taxiing to get good fitting

13 CHECK OF ELECTRICAL COMPONENT

13.1 LANDING GEAR UP/DOWN RELAY AND DOOR OPEN/CLOSE RELAY

This check will intend to preclude a significant closed-failure (fail to open) condition of the relays.

The following equipment is required to conduct this check.

- Insulation resistance tester (500 volts r.m.s.).
- DC power supply (0 to 50 volts, 1 ampere or above)
- Adequate Continuity checker

(1) Preparation

(a) Remove all electrical power from the aircraft.

(b) Disconnect all wires from the terminals of each relay and tag for identification.

(2) Insulation resistance check

(a) Measure insulation resistance between power terminals (A1 and A2) and power terminal (A1 or A2) to case (grounded)

(b) Insulation resistance values shall be greater than 50 megohms.

(c) Any relay which does not meet above requirement shall be replaced with a new relay.

(3) Contact operation check

(a) Connect a continuity checker between the power terminals (A1 and A2).

(b) Connect a power supply to the coil terminals X1(pos.) and X2(neg.) and set voltage at 28 volts.

(c) Check for proper continuity / discontinuity in the energized and de-energized positions.

(d) Replace the relay with a new relay if this check indicates a failure.

(4) Restoration

(a) Remove the above test equipment and reconnect the wires.

(b) Restore electrical power to the aircraft.

CHAPTER

5

FLIGHT CONTROL SYSTEM

CHAPTER V

FLIGHT CONTROL SYSTEM

CONTENTS

1. GENERAL

1.1 GENERAL DESCRIPTION

The control system consists of dual control wheels and rudder pedals for pilot and co-pilot. Elevators and spoilers are controlled conventionally by push-pull and by rotating the control wheel left and right.

The rudder is controlled by the rudder pedals located above the floor in front of the seats. These controls are mechanically connected to torque tubes, push-pull rods, bell cranks, etc. and connected to control surfaces on empennage and wing by 5/32 in. 7 x 19 cables.

In the spoiler system, the L.H. and R.H. spoilers are moved by means of differential linkage located on wing center section.

Cables, except for the spoiler, have turnbuckles near F.STA8490 and 9625 so that tension adjustment and disconnection can be accomplished through the access door. Cables should be inspected periodically for corrosion, scratches and broken strands, etc.

Elevator trim tabs and rudder trim tabs are controlled by rotating the wheels located on the center pedestal.

Both systems are linked to an empennage by 1/16 in. 7 x 7 cables, which operate actuators, installed in the horizontal stabilizer and the vertical stabilizer, and move the tabs. Tab motion is shown mechanically on the indicator installed on the center pedestal.

Lateral trim is accomplished by actuating trim ailerons, of unique construction, which are attached to the outboard flap trailing edge.

Trim ailerons are controlled by an electrical actuator energized when the control switch is rotated to the left or to the right. Trim tab movement is shown electrically on the indicator.

Double slotted full span flaps consist of inboard and outboard flaps on both sides of the nacelles. Flaps are moved as one unit through 3 actuators (each wing half). The actuators are driven by an electric motor (down: 32A @ 4000 rpm, up: 16A @ 4500 rpm) which is operated by a control switch in the cockpit. Gears and torque tubes connect the flaps to the actuators. It is possible to stop the flaps at the UP, 5°, 20° and 40° down positions which are shown by indicator lights located adjacent to the flap control switch position. Sealed bearings are used in rotating and oscillating mechanism such as pulleys, bellcranks, quadrants, etc., so that routine inspection and maintenance (except special) is minimized.

2. CONTROL COLUMN

2.1 GENERAL DESCRIPTION

The T shaped control column, located in front of instrument panel, is of welded construction and moves fore and aft around pivot bearings on the airframe attachment section, this motion being transmitted to the elevator control system. A U shaped stop is provided to avoid interference with instrument panel, etc., due to excessive column travel.

Control shafts attached on both arm ends of the T-column extend through the instrument panels for control wheel mounting. The control wheel and shaft for the co-pilot can be removed as required.

Gust lockpins are provided on both control shafts. Pilot side lockpin is used to lock control column usually and co-pilot side lockpin is used in case of removal and installation of instrument panel.

Fore-and-aft motion of the wheel is transmitted directly to the T-column, which is connected to the elevator system.

Wheel rotation drives the chain attached to the T-column, which is connected to the spoiler system. The chain, with follow-through and underfloor center quadrant, is made up in two loops; tension of each loop can be adjusted independently.

On the pilot side, the lateral bar of the T-column has a stall warning shaker installed which gives stall warning in response to a signal from the vane installed at the wing tip.

2.2 REMOVAL AND INSTALLATION (See Fig. 5-1)

(1) To gain access to column, remove pilot and co-pilot rudder pedals.

(2) Disconnect wiring at terminal block, located on top of T-column. (See Fig. 5-2)

(3) Remove connecting bolt at universal joint section forward of instrument panel and remove control shaft and wheel. (See Fig. 5-3)

(4) Disconnect chain of spoiler control. (See Fig. 5-4)

(5) Remove center floor.

(6) Loosen turnbuckles inserted on both sides of the column and disconnect chains.

(7) Remove stop. (See Fig. 5-5)

(8) Remove 3 tapered pins below the T-column and remove the T-column towards pilot side.

(9) Install in reverse sequence of removal.
Make sure that the wheel falls forward of its own weight when the wheel is pulled full aft and released.
If wheel does not fall forward, check column for interference and check pivot bearing at column base for defects. Lubricate spherical bearing located in instrument panel. (MIL-L-7870)

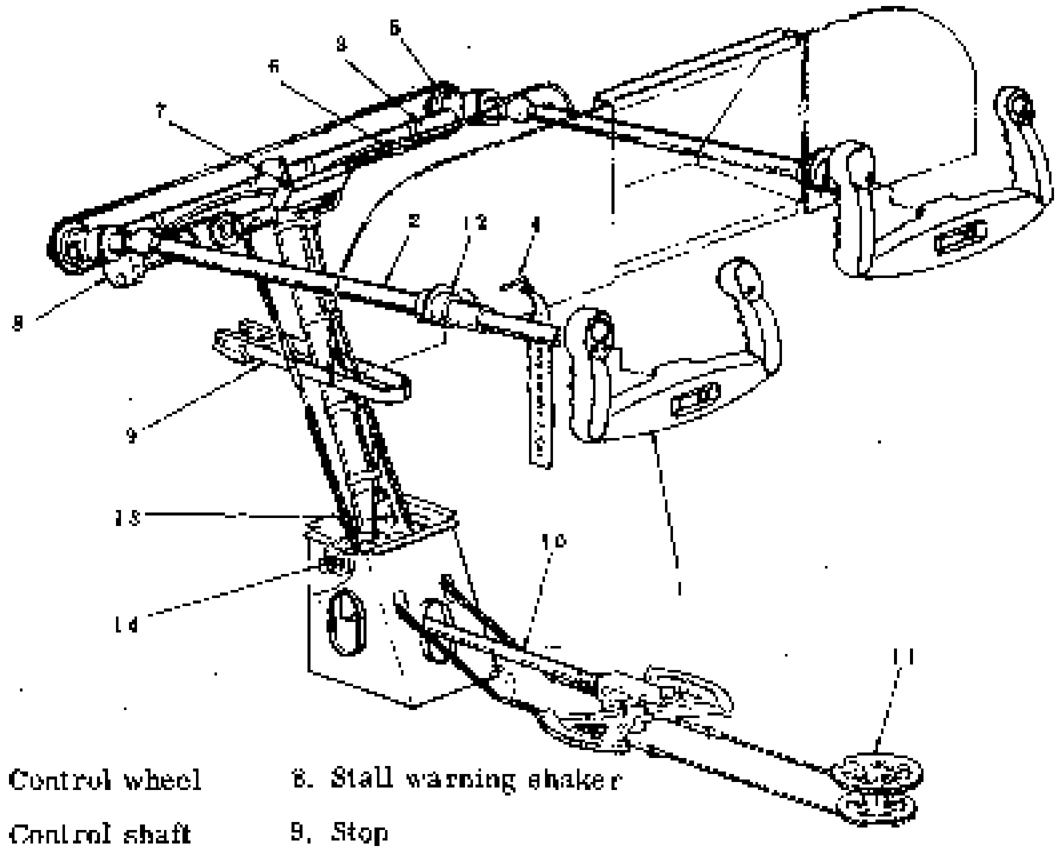
(10) To remove control wheel from shaft, remove wheel bottom plate, disconnect wiring at knife disconnect section and loosen attaching nuts. Installation torque value of this nut is 750 ~ 1000 in-lb.(864 ~ 1152 kg-cm).

2.3 RIGGING

(1) Install chain on sprocket so that wheel rotating angles both to the right and left hit the stops equally.

(2) Adjust chain tension with turnbuckle and follow-through cable in the upper section of T-column. It should be as loose as possible, with no free play in wheel, in rotational direction. (See Fig 5-6) When one wheel is inclined, although the other is in neutral position, it is because tension in the upper side and the lower side is not equal or wheel installation is not correct. In the latter case, reinstall wheel moving one pitch of serration.

(3) Wheel operating force shall be approximately 2 lbs. (0.454 kgs.). Indications of LH and RH wheels to horizontal plane shall be less than $\pm 1^{\circ}$.

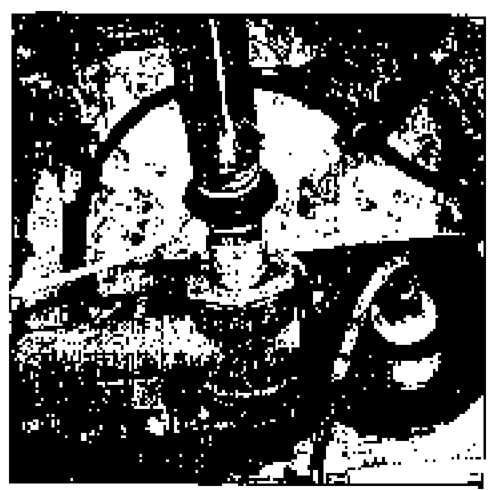


1. Control wheel
2. Control shaft
3. T-column
4. Gust-lock pin
5. Sprocket
6. Follow through
7. Terminal block
8. Stall warning shaker
9. Stop
10. Elevator push pull rod
11. Center quadrant
12. Spherical bearing
13. Taper pin
14. Pivot bearing

Fig. 5-1 Installation of control column

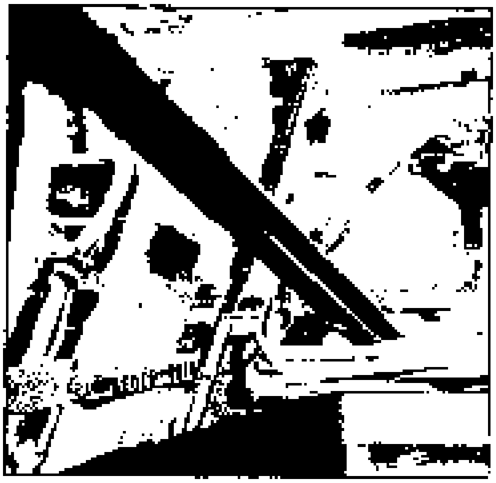
Disconnect wiring at terminal block

Fig. 5-2



Remove connecting bolt at universal joint

Fig. 5-3



Disconnect chain of spoiler control
Fig. 5-4

Remove stop
Fig. 5-5

Adjust follow-through cable
Fig. 5-6

3. ELEVATOR CONTROL SYSTEM

3.1 GENERAL DESCRIPTION

The elevator is controlled by pushing forward and pulling backward on the control wheel. Fore-and-aft motion of the T-column rotates the forward quadrant under the floor, by push-pull rods located in the lower end, and moves the 5/32 in.cables running backward under the floor between keels (second from outboard end, in both L. H. and R. H. middle sections.). The cables pass through 7 fairleads and pressure bulkhead seals on the way, then go upward from F. STA 7550, change their lateral and vertical directions at pulley brackets at F. STA 8805 and reach the aft quadrant on empennage. The aft quadrant is connected to the L. H. and R. H. elevator by a torque tube.

Two down-springs, attached to the horizontal stabilizer front spar, are connected to the aft quadrant through levers and cables. The elevator is normally down with the aircraft on the ground. The down-spring affords good control feel to pilots and facilitates longitudinal trim at low air speeds.

3.2 REMOVAL AND INSTALLATION OF ELEVATOR

3.2.1 REMOVAL AND INSTALLATION OF L. H. AND R. H. ELEVATORS IN ONE UNIT

(1) Remove tail cone.

(2) Disconnect tail light wires at terminal block (TB319), and identify.

(3) Disconnect control cable at turnbuckle at F. STA 8605.

(4) Remove control cables and down-spring cables at aft quadrant. (See Fig. 5-7)

(5) Remove bonding jumpers.

(6) Remove bolts, nuts and cotter pins connecting trim tab rods. (See Fig. 5-8)

(7) Remove 5 bolts which connect horizontal stabilizer and elevator. (H. STA 840 and 1790 intermediate fittings and aft quadrant). (See Fig. 5-9)

(8) Install in reverse sequence of removal.

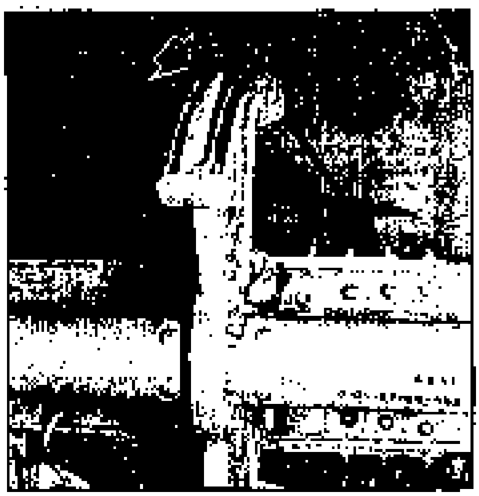


Fig. 5-7 Remove cables at aft quadrant

NOTE

As elevators can be removed in one unit, make sure of adequate support so that torque tube is not bent or twisted excessively.

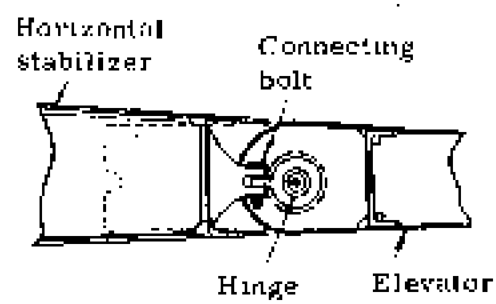


Fig. 5-9 Remove bolts for intermediate fitting

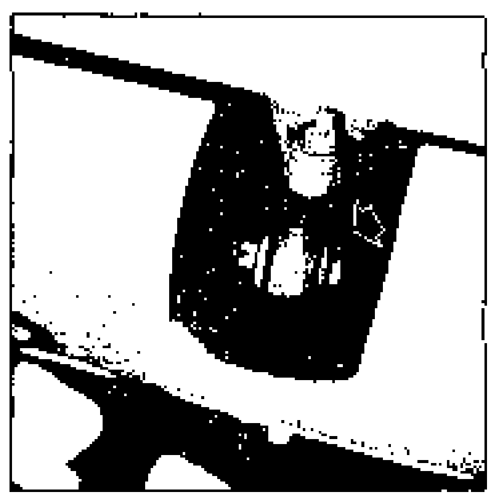


Fig. 5-8 Remove connecting bolt

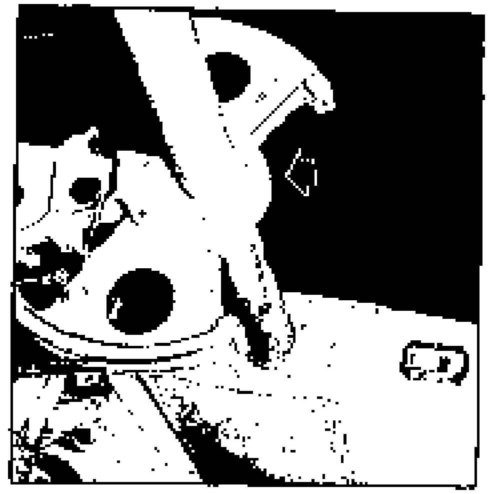
3.2.2 REMOVAL AND INSTALLATION OF L.H. OR R.H. ELEVATOR

(1) Insert rig pin into aft quadrant.

(2) Remove bolt, nut and cotter pin from elevator trim tab rod.

(3) Remove 6 bolts connecting elevator torque tube to aft quadrant. (See Fig. 5-10)
For installation, tighten nut to 20 to 25 in-lbs (23 to 28.8 kg-cm) torque.

(4) Remove bolt, nut and cotter pin from intermediate fitting, between H.STA 840 and H.STA 1790 of horizontal stabilizer, connecting horizontal stabilizer and elevator.
Remove elevator from aircraft.



Remove bolts connecting elevator torque tube to aft quadrant
Fig. 5-10

3.3 REMOVAL AND INSTALLATION OF CONTROL CABLE (See Fig. 5-11)

(1) Insert rig pin into forward quadrant. (See Fig. 5-12)

(2) Insert rig pin into aft quadrant. (See Fig. 5-13)

(3) Remove turnbuckle from control cable at F.STA 8605.

(4) Remove cable terminal fitting from forward and rear quadrant.
Remove cable from aircraft.

(5) Install in reverse sequence of removal.

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NOTE

When removing cable, a piece of string tied to the end of cable and routed in place of cable which is drawn out will facilitate reinstallation.

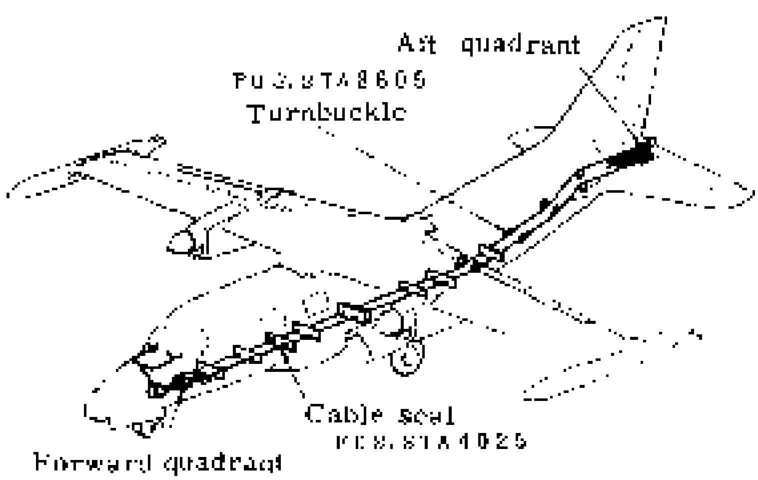

Fig. 5-11 Removal and installation of control cable

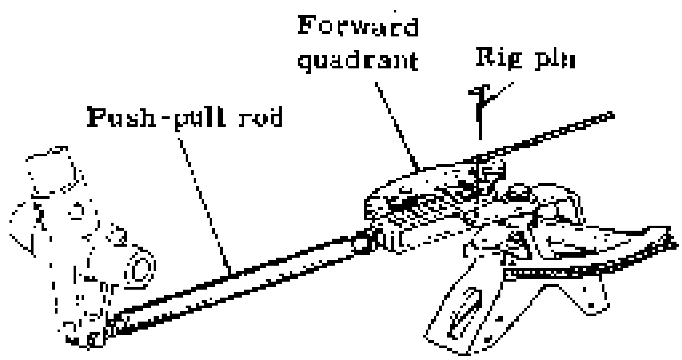

Fig. 5-12 Forward quadrant

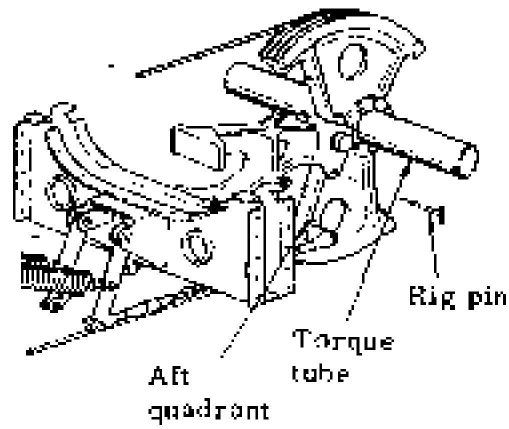

Fig. 5-13 Insert rig pin in aft quadrant

3.4 REMOVAL AND INSTALLATION OF CABLE SEAL

Cable seal is provided at F.STA 4025 to prevent leakage of cabin air pressure through the hole where cable is routed. Remove cable seal in accordance with the following procedures.

(1) Remove cabin floor.

(2) Cable seals at F.STA 4025 can be removed by removing cotter pins from casing and pulling seal forward. (See Fig. 5-14)

(3) Install in reverse sequence of removal.

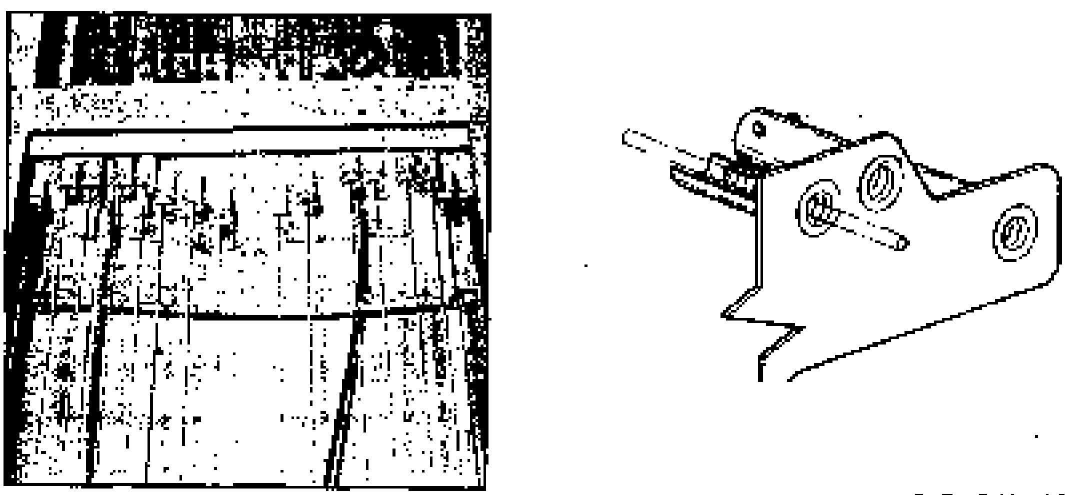

Fig. 5-14 Removal of cable seal

3.5 RIGGING

3.5.1 RIGGING OF CABLE

(1) Insert rig pin or gust lock pin into pilot's control wheel shaft.

(2) Insert rig pin into forward quadrant.

(3) Adjust length of push-pull rod.

(4) After confirming that control cable terminal is secured to quadrant, cables are placed on pulleys properly and guard pins are installed securely, insert rig pin into aft quadrant.

(5) Miss-matching between elevator trailing edge and tail-cone inner edge and horizontal stabilizer trailing edge should be less than 0.12 in.

(6) Adjust cable tension by turning turnbuckles of F. STA 8605 to the specified value of 60 lbs $^{+3}_{-0}$ (70° F).

Compensation of cable tension for temperatures shall be made in accordance with Fig. 5-62.

(7) After rigging the cable, install safety wire on turnbuckle and apply alignment mark to lock nut of push-pull rod.

3.5.2 ADJUSTMENT OF DEFLECTION ANGLES

Adjustment of deflection angles is accomplished in accordance with the following procedures after rigging of cable is completed.

(1) Attach protractor to elevator at H. STA 855 (See Fig. 5-15).

(2) Remove gust lock pin from control wheel and rig pins from forward and aft quadrant.

(3) Ensure elevator deflection is 25° ± 30' in up direction and 12° ± 30' in down direction when wheel is pulled and pushed fully.

For adjustment of angle, adjust rear stop bolts. Tighten lock nut on stop bolt. Apply alignment mark. Install safety wire. (See Fig. 5-16)

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(4) Detach protractor.

Fig. 5-15 Attaching of protractor

Fig. 5-16 Apply alignment mark and safety wire

3.5.3 ADJUSTMENT OF DOWN-SPRING AND MEASUREMENT OF CONTROL FORCE

Adjust down-spring in accordance with the following procedures after rigging of cable and adjustment of deflection angle are completed.

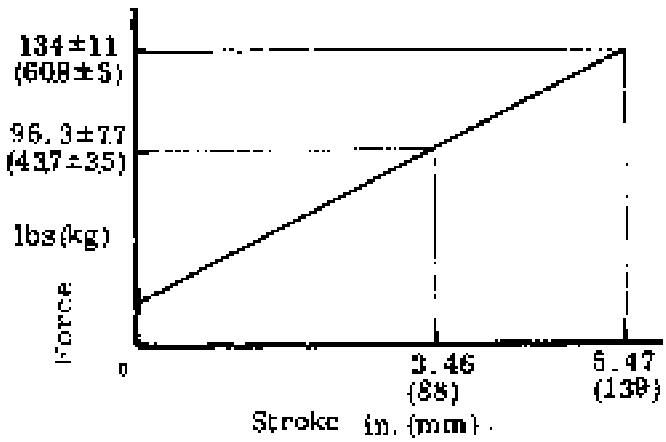
(1) Adjust length of cables connecting down-springs and quadrant to 12.40 ± .12 in. (315 ± 3 mm) (Ref.) by turnbuckle.

(2) Attach push-pull scale to control wheel and measure control force with autopilot servo control cables disconnected. (See Fig. 5-17)

(a)

Control wheel	Control force
Pushed forward	-20 ± 2 lbs. (-9.0 ± 0.9 kg)
Pulled back	(*1) 26 ± 2 lbs. (11.8 ± 0.9 kg) (*2) 28 ± 2 lbs. (12.7 ± 0.9 kg)

Before control force is measured, place gust lock pin in neutral position and make push-pull scale read 0.



Characteristics of down spring

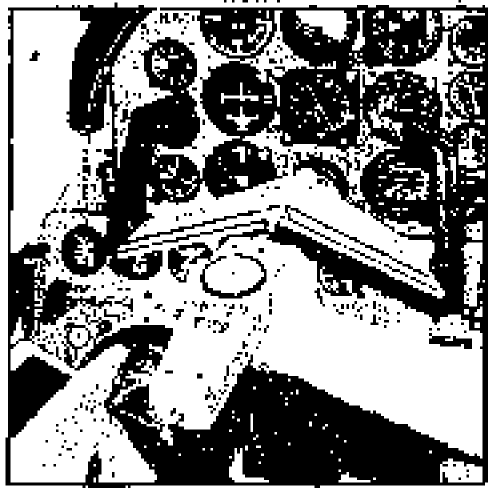


Fig. 5-17 Measure control force

(b) Control force required for pushing and pulling control wheel.

(i) Pull from fully down position (control surface down position)
25 ± 2 lbs (11.3 ± 0.9 kg)

(ii) Push further from fully down position (control surface down position)
-19 ± 2 lbs (-8.6 ± 0.9 kg)
(Measure just before control wheel contacts with stop, when control force is abruptly increased.)

(iii) Push from fully up position (control surface up position)

-21 ± 2 lbs (-9.5 ± 0.9 kg)

(iv) Pull further from fully up position (control surface down position)
28 ± 2 lbs (12.7 ± 0.9 kg)

(Measure just before control wheel contacts with stop, when control force is abruptly increased.)

unit: lbs(kg)

Position \ Control	Pull (up)	Push (down)
Control surface full down	25 ± 2 (11.3±0.9)	-19 ± 2 (-8.6±0.9)
Neutral	26 ± 2 (11.8±0.9)	-20 ± 2 (-9 ± 0.9)
Control surface full up	28 ± 2 (12.7±0.9)	-21 ± 2 (-9.5±0.9)

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

(3) Re-rig cable connecting down-spring with aft quadrant so that measured control force may be within values tabulated above. When control force found abnormal, check the following:

(a) Interference in system.

(b) Adequate cable tension.

(c) Proper tension of down-spring. (See Fig. 5-18)

(d) Control column for binding or interference.

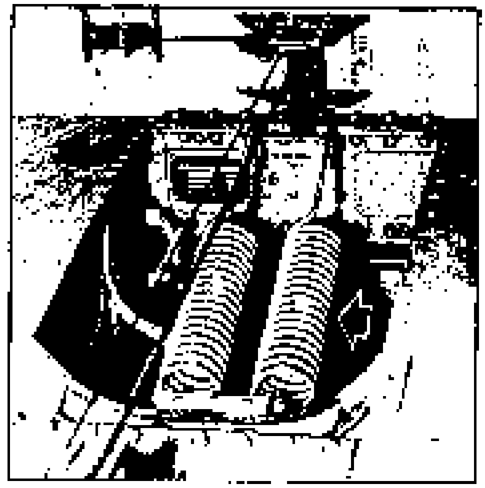


Fig. 5-18 Installation of down-spring

4. RUDDER CONTROL SYSTEM

4.1 GENERAL DESCRIPTION

The rudder is controlled by pedals which are located on the floor under the instrument panel. Pedals for pilot and co-pilot are attached independently to cast brackets installed on the airframe. Pedals can be removed alone or with lever and attaching shaft as required.
When a pedal is pushed, a lever swings, rotating forward quadrant through push-pull rod and cables attached to the quadrant.
L.H. and R.H. quadrants are connected together by the forward push-pull rod. Pedals for pilot and co-pilot move in the same direction and L.H. and R.H. pedals move in opposite directions to each other.
A bungee spring is installed in the lower section of the quadrant on the pilot's side to give the pilot good control feel.
By adjusting the length of the push-pull rods within ± 0.375 in. (9.5 mm), the fore-and-aft travel of the pedal can be adjusted within ± 1 in. (25.4 mm).
Cables from quadrants run along the pressure bulkhead and at F.STA 1275, turn aft at right angles, and run between keels. (L.H. and R.H. cables are the outward cables in the middle section.)
Cables pass through 7 fairleads, seals in the pressure bulkhead, and the cable bracket at F.STA 7550, and like the elevator system change their vertical and lateral direction at brackets at F.STA 8895, to reach the aft quadrant in the empennage.
The aft quadrant is attached to the rudder by a torque tube.

4.2 REMOVAL AND INSTALLATION OF RUDDER

(1) Remove tail cone.
(2) Disconnect tail light wires from terminal block (TB319) and identify.
(3) Disconnect control cable from turnbuckle at F.STA 8605.
(4) Remove cables from aft quadrant.
(5) Remove bolt connecting trim tab rod.
(6) Remove bonding jumpers (2 places).

(7) Remove bolts connecting aft quadrant and rudder torque tube (See Fig. 5-19). For installation, tighten to a torque of 20 to 25 in-lbs (23 to 29 kg-cm).

(8) Remove bolt fastening rudder fitting to two hinges at V. STA 1500 and V. STA 2720. Remove rudder from aircraft.

(9) Install in reverse sequence of removal.

Fig. 5-19 Remove bolts connecting quadrant and torque tube

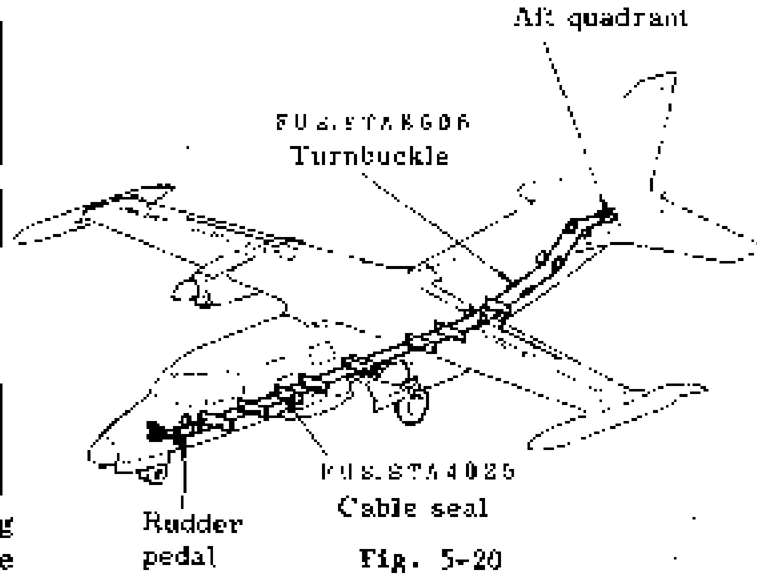


Fig. 5-20

4.3 REMOVAL AND INSTALLATION OF CONTROL CABLE (See Fig. 5-20)

(1) Insert rig pin into forward quadrant. (See Fig. 5-22)

(2) Insert rig pin into aft quadrant. (See Fig. 5-21)

(3) Remove turnbuckle from control cable at F. STA 8605.

(4) Remove cable terminal fitting from forward and aft quadrant. Remove cable from aircraft.

(5) Install in reverse sequence of removal.

NOTE

When removing cable, a piece of string tied to the cable end will facilitate reinstallation.

4.4 REMOVAL AND INSTALLATION OF CABLE SEAL

Cable seals at F STA 4025 can be removed by removing cotter pin from casing and pulling seal forward after removing cabin floor (see Paragraph 3.4).

Fig. 5-21 Insert rig pin in aft quadrant

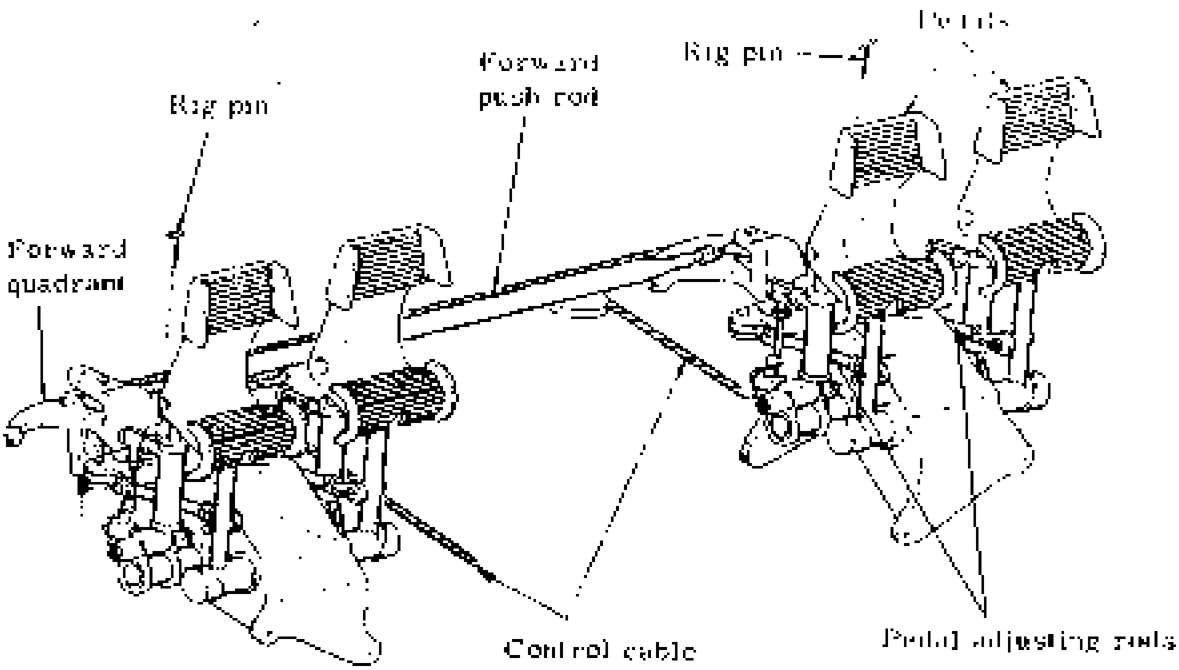


Fig. 5-22 Insert rig pin in forward quadrant

4.5 RIGGING

4.5.1 RIGGING OF CABLE

When aircraft is on the ground, movement of rudder is linked to that of nose gear steering system. Rigging of cable should be performed after disconnecting the linkage at steering rod in nose wheel well, or disconnecting nose gear torque link or jacking aircraft up.

(1) Insert rig pin into pilot quadrant.

(2) Adjust length of forward push rod and insert rig pin into co-pilot quadrant. Line up rudder pedals by adjusting length of push rods so that they can be reached easily.

Ensure that rig pin can be inserted and removed properly.

(3) After confirming that control cable fitting is secured to quadrant, cables are placed on pulleys properly and guard pins are installed securely, insert rig pin into aft quadrant.

(4) Miss-matching of rudder trailing edge and vertical stabilizer trailing edge (upper) and tail cone trailing edge should be within 1/8 in.(3.2 mm).

(5) Adjust cable tension by turning turnbuckles at F.STA 8605 to the specified value of 60 lbs +5/-0 (27.2 +2.3/-0 kg) 70°F (21°C). Compensation of cable tension for temperature shall be made in accordance with Fig 5-62.

(6) After rigging the cable, install safety wire on turnbuckle. Apply alignment mark to lock nut of push-pull rod.
If pedal is adjusted, apply alignment mark to lock nut of the adjusting rod.

NOTE

If steering rod in nose wheel well is disconnected before rigging, restore rod to the operational status. Restore nose gear torque link to the operational status.

4.5.2 ADJUSTMENT OF DEFLECTION ANGLE

After rigging of cable is completed, adjust deflection angle in accordance with the following procedures.

(1) Attach protractor to rudder. (See Fig. 5-23)

(2) Remove rig pins.

(3) Adjust length of stop bolts so that maximum rudder deflection angles are within the following values when the pedal is fully pushed down.
After adjustment is complcted, install lock nut. Apply alignment mark.
Install safety wire.

Pedal Position	Deflection Angle	
	*1	*2
Left pedal is fully pushed down	To left $22^{\circ} \pm 30'$	To left $24^{\circ} \pm 30'$
Right pedal is fully pushed down	To right $24^{\circ} \pm 30'$	To right $22^{\circ} \pm 30'$

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

(4) Detach protractor.

4.5.3 INSTALLATION AND ADJUSTMENT OF BUNGEE SPRING (See Fig. 5-24)

(1) Measure bungee spring characteristics. Ensure that relation between bungee stroke and load is as tabulated below.
When spring characteristics are not normal, tighten or loosen adjusting nut. (See Fig. 5-25)

(2) Install bungee spring after checking adjustment.
Adjust length at rod end so that spring is not deflected by force.

Fig. 5-23 Protractor attached to rudder.

Fig. 5-24 Installation of bungee spring

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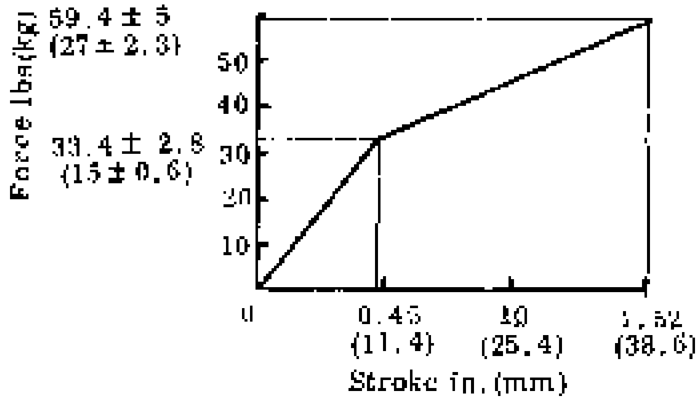

Fig. 5-25 Characteristics of bungee spring

4.5.4 MEASUREMENT OF CONTROL FORCE (See Fig. 5-26)

(1) Attach push-pull scale to pedals for pilot. Measure control force for L.H. and R.H. pedals. When L.H. and R.H. pedals are moved back and forth from the neutral position (position where rig pin is inserted) the control forces for both pedals must be equal.
To make control forces for both pedals equal, length of rod end of bungee spring may be adjusted. Apply alignment mark to rod end.

LBS (KGS)

DIRECTION / POSITION		PUSH (P1)	RETURN (P2)	$\frac{P1 + P2}{2}$
NEUTRAL		Less than 7 (3.2)	Less than 7 (3.2)	Less than 7 (3.2)
Rudder deflection	*1	30 ± 4 (13.6 ± 1.8)	-18 ± 4 (-8.2 ± 1.8)	Less than 7 (3.2)
Approximately 24°	*2	38 ± 4 (17.2 ± 1.8)	-24 ± 4 (-10.9 ± 1.8)	Less than 8 (3.6)

NOTE

i Control force is usually measured with the aircraft jacked up and the nose gear down.

ii Measure control force with auto-pilot servo control cables disconnected.

(2) If control force is found unusual, check for the following.

(a) Interference in system.

(b) Characteristics of bungee spring.

(c) Brake cylinder for distortion, when pedal is depressed.

(d) Cable for proper tension.

(e) Rotating parts of each mechanism for lubrication.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

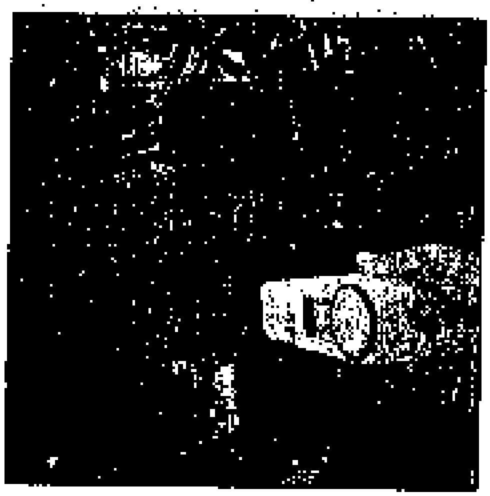

Fig. 5-26 Measurement of control force

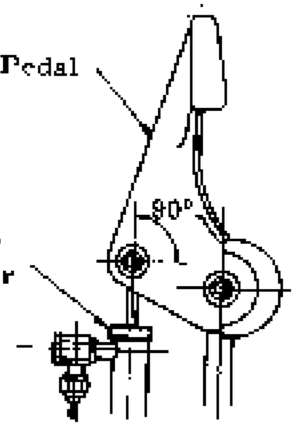

Fig. 5-27 Rigging of pedal

4.5.5 RIGGING OF PEDAL (See Fig. 5-27)

(1) Insert rig pins into forward quadrants for both pilot and co-pilot.

(2) Adjust the pedal adjusting rods (Fig 5-22) to the most comfortable position for the pilot (co-pilot). The adjusting length of the rod is between 5.82 to 6.61 in. (148 to 168 mm) which allows the pedal to be adjusted a total of 2.0 in. (50.8 mm), full forward to full aft.

(3) Adjust length of rod of master cylinder so that pedal may be placed at correct angle.

(4) Rigging of pedal for co-pilot is accomplished in the same way.

(5) Remove rigging pin.

5. SPOILER CONTROL SYSTEM

5.1 GENERAL DESCRIPTION

The spoilers are controlled by rotating the control wheel to the left and right in the same manner as a conventional aileron control. Motion of the chain on control column is transmitted to cables through the center quadrant and under the floor. The cables run aft between keels, and turn upward at F.STA 8035 along the bulkhead. Then they turn at right angles (through 2 pulley brackets) and reach the differential linkage at the wing center section. Turnbuckles are installed at F.STA 6869. A feel spring in parallel with the cable (linearizing control force) is installed on the bulkhead. Travel ratio of differential linkage, for spoiler up and down, is approximately 4:1 against equal input cable travel on LH and RH sides. Differential linkage consists of: bracket made of cast magnesium, lever, link, etc., made of cast magnesium or aluminum.
From this point, conventional push-pull rods are used in the LH and RH wings to the bellcranks. The spoiler is divided into inboard and outboard sections at W.STA 3800. These sections can be adjusted independently but their movements are synchronized.

5.2 REMOVAL AND INSTALLATION OF SPOILER

(1) Extend flaps fully and open panel door on the under surface of trailing edge.

(2) Hold control surface of spoiler to be removed in retracted position.

(3) Remove bolts connecting spoiler push rods at W.STA 2851. (See Fig. 5-28)

(4) Remove bolt connecting spoiler push rod at W.STA 4545 (See Fig. 5-29)

(5) Remove bolts connecting spoilers and rear spar, and lateral connecting bolts, at spoiler attachment fittings at W.STA 1980, W.STA 2980, W.STA 3580, W.STA 3980, W.STA 4630, and W.STA 5180. (See Fig. 5-30)

(6) Install in reverse sequence of removal.

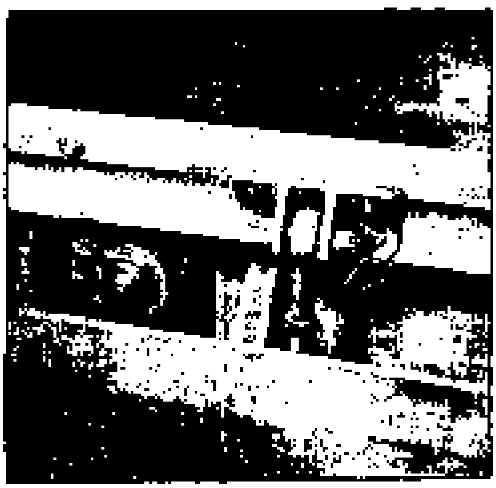

Fig. 5-28 Remove bolt at W.STA 2851

Fig. 5-29 Remove bolt at W.STA 4545

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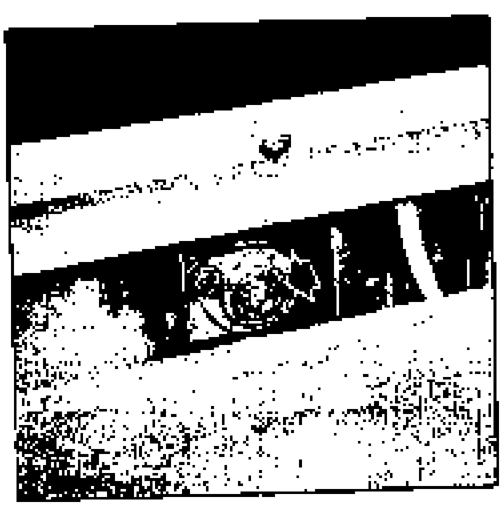

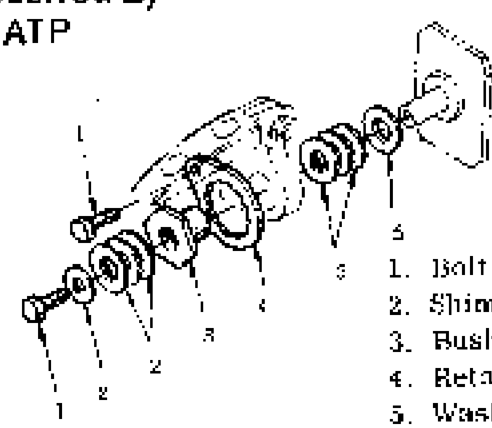

Fig. 5-30 Remove bolt connecting spoilers and rear spar

NOTE

i When spoiler is removed, record number of shims used for attachment fittings to facilitate installation.
ii Handle spoiler with care so that its sharp edges are not deformed.
iii Sufficient care must be taken not to have fingers hurt by sharp edge of spoiler or compressed between spoiler control surface and airframe.

5.3 REMOVAL AND INSTALLATION OF CONTROL CABLE

(1) Insert rig pin in forward quadrant under floor at F STA 2400.
(2) Insert rig pin in differential linkage in wing center section (see Fig 5-32).
(3) Remove turnbuckle from control cable at F STA 6869.
(4) Disconnect feel spring assembly from bracket on bulkhead at F STA 8035.
(5) Remove cable terminal fitting from center quadrant, and differential linkage. Remove cable from aircraft.
(6) Install in reverse sequence of removal.

NOTE

When removing cable, a piece of string tied to cable end will facilitate re-installation.

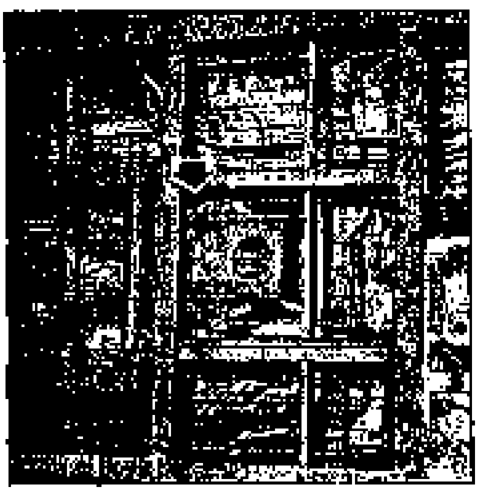

Fig. 5-31 Insert rig pin into center quadrant

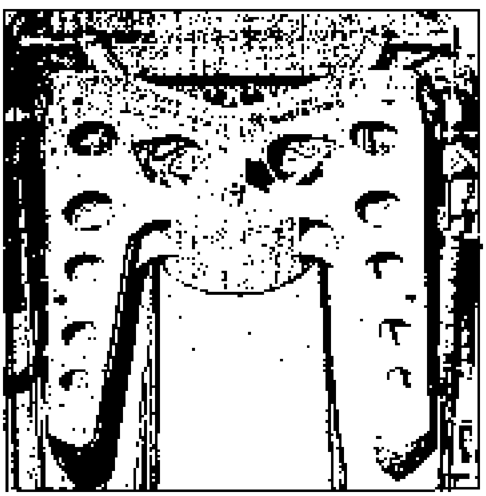

Fig. 5-32 Insert rig pin into differential linkage

5.4 REMOVAL AND INSTALLATION OF CABLE SEAL

Cable seal at STA 4025 can be removed by removing cotter pin from casing and pulling seal forward, after removing cabin floor. (See Para. 3.4 of this section.)

5.5 RIGGING OF SPOILER SYSTEM

5.5.1 RIGGING OF CABLE TENSION

(1) Insert rig pins into center quadrant. (See Fig. 5-31)

(2) Loosen nut of feel spring on bulkhead.
Release control cable. (See Para. 5.5.3)

(3) Insert rig pin into wing center section differential linkage. (See Fig. 5-32)

(4) Adjust cable tension by turning turnbuckle at F.STA 6360 to specified value of 50 $^{+5}_{-0}$ lbs (70° F) (22.7 $^{+2.3}_{-0}$ kg) (21°C). (See Fig. 5-33)

Compensation of cable tension for temperatures shall be performed in accordance with Fig. 5-62.

(5) Retighten nut of feel spring on bulkhead. (See Para. 5.5.3)

(6) After rigging of cable is accomplished, remove rig pins.

(7) Attach a protractor to control wheel for pilot. (See Fig. 5-34)

(8) Place elevator in neutral position. Rig tension of control column chain by means of follow through assembly of control column and turnbuckle(s) under floor or on both sides of column, so that play of control wheel in the direction of rotation may be less than 1.5°.
Play means range of travel of control wheel before spoiler control surface begins to move.

(9) After rigging is completed, install safety wire in all turnbuckles.

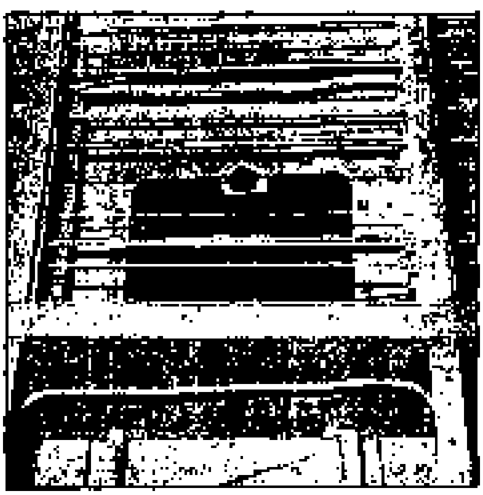

Fig. 5-33 Adjusting cable tension

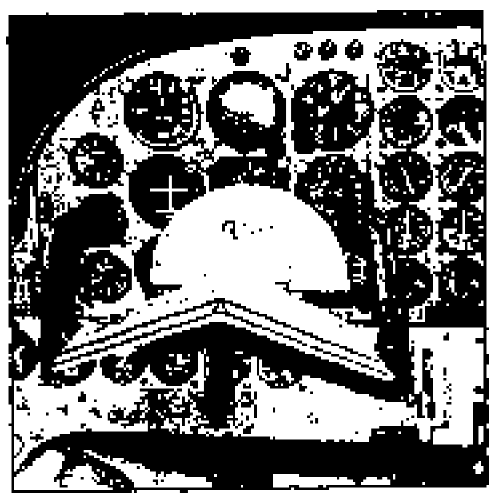

Fig. 5-34 Attach a protractor to control wheel

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5.5.2 RIGGING OF DEFLECTION ANGLE

After rigging of cable and chain is completed and feel spring is installed, rig deflection angle of spoilers in accordance with the following procedures.

(1) Insert gust lock pin into control wheel.

(2) Insert rig pin into bell cranks, 2 in R.H. wing and 2 in L.H. wing. (See Figs. 5-35 and 5-36)

(3) Ensure link rod between differential link and bellcrank has been adjusted to specified length. Rig if necessary. (See Fig. 5-37)
Check other rods for extreme difference in range of extension at both ends by looking into inspection hole.

(4) Adjust length of push-pull rod and position of eccentric bushing and shims. so that gap and mismatch between trailing edge of spoiler and trailing edge of main wing may be within tolerances. (See Fig. 5-38)

Fig. 5-35 Insert rig pin into bell crank (W.STA 2800)

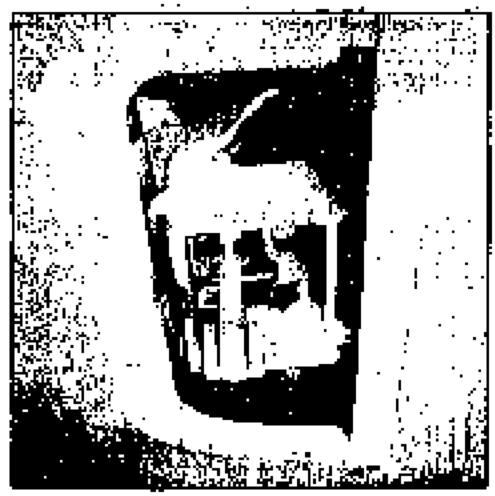


Fig. 5-36 Insert rig pin into bell crank (W.STA 4494)

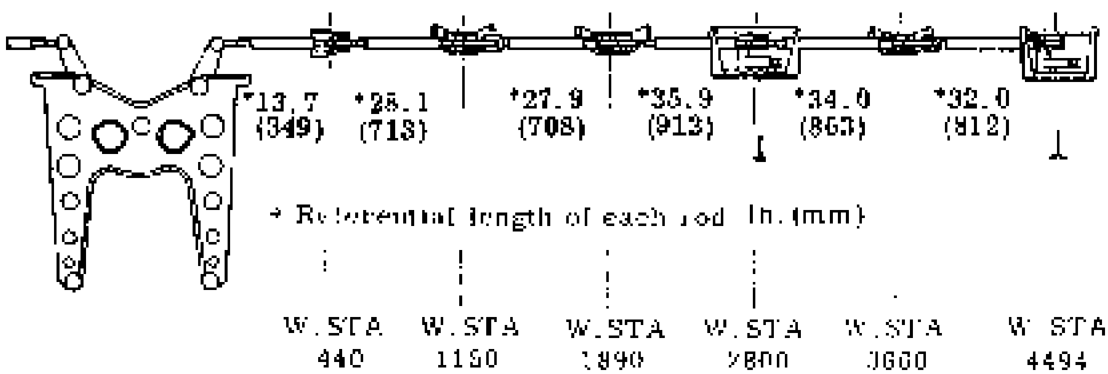
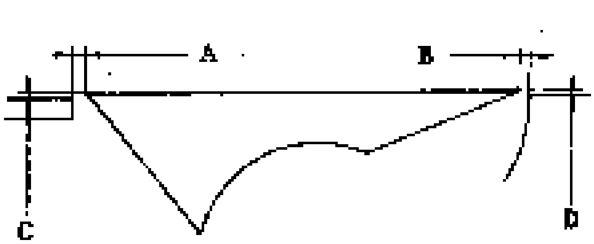


Fig. 5-37 Rigging of differential link



	DIM	MAX ALLOW
A	1.5 to 2.5 (*1) 1.0 to 2.0 (*2)	3.0 (*1) 2.5 (*2)
B	1.0 to 2.5 (*3) 2.0 to 3.5 (*2) 2.0 to 4.0 (*4)	3.5 (*3) 4.5 (*2) 5.0 (*4)
C	UP 1.2 DOWN 0.0	1.2 to 2.2 -
D	UP 2.5 DOWN 0.5	2.5 to 3.0 0.0 to 0.5

Fig. 5-38

*1 Aircraft S/N 652SA, 733SA and subsequent
*2 Aircraft S/N 661SA, 697SA thru 732SA
*3 Aircraft S/N 652SA
*4 Aircraft S/N 733SA and subsequent

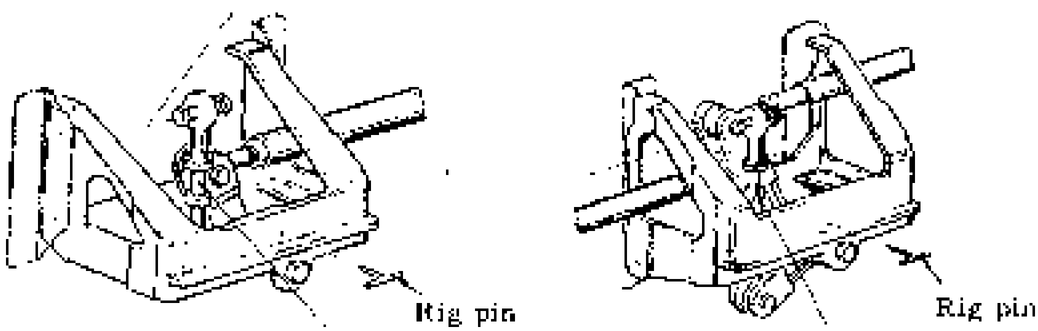
(a) Adjustment of push-pull rod. (See Fig. 5-39)
Push-pull rod is adjustable for changing the mismatch between trailing edge of spoiler and that of wing which is given in dimension "D" of Fig. 5-38.

(b) Adjustment of eccentric bushing. (See Fig. 5-40)
Remove retainer. Rotate eccentric bushing clockwise or counterclockwise. Change in mismatch between leading edge of spoiler and that of wing given in dimension "C", Fig. 5-38, may be changed for vertical adjustment.

(c) The fwd and aft horizontal gap adjustment between the spoiler and wing is given in Fig. 5-38, dimensions "A" and "B". This adjustment may be made by changing the quantity of shims between the hinge fitting and the rear spar, refer to Figs. 5-30 and 5-41. All remaining shims should be installed under the bolt head.

NOTE

After link rod and push-pull rod are adjusted, check all rods for range of extension. Tighten lock nut. Apply alignment mark.



Internal push-pull rod (W.STA 4494)

External push-pull rod (W.STA 2800)

Fig. 5-39 Adjustment of push-pull rod

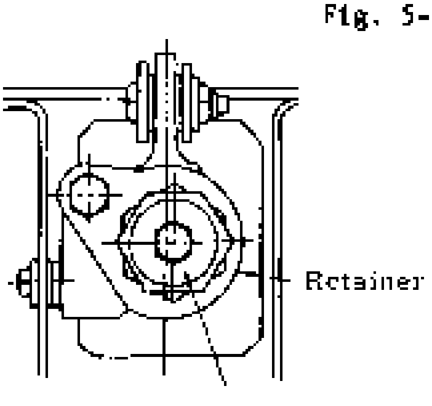


Fig. 5-40 Adjustment of eccentric bushing

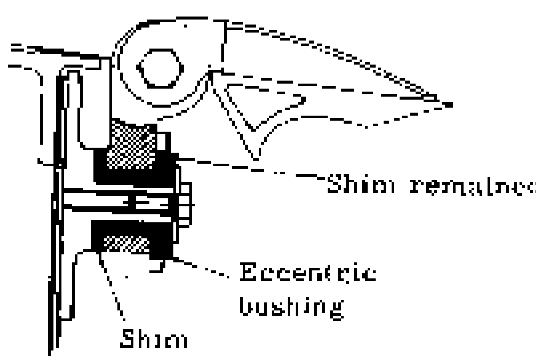


Fig. 5-41 Adjustment of shim

(5) Attach a protractor to spoiler. (See Fig. 5-42)
Before attaching, remove screws from inboard and outboard holes.

(6) Remove all rig pins.

(7) Operate spoilers and ensure deflection angle being within specified range.

Right spoiler	Left Spoiler
Down $14°^{+0.5°}_{-1°}$	Up $60°^{+2°}_{-1°}$
Up $60°^{+2°}_{-1°}$	Down $14°^{+0.5°}_{-1°}$

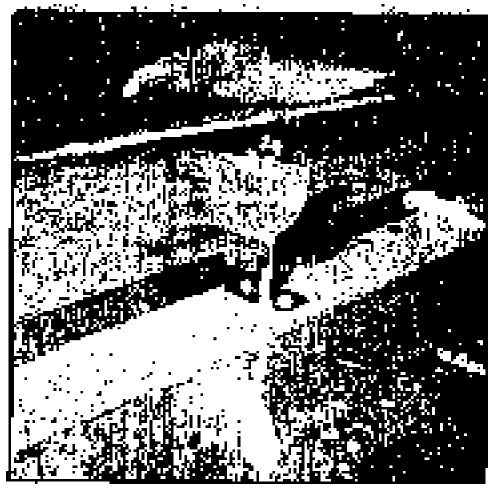


Fig. 5-42 Attach protractor to Spoiler

Angle differences between L. H. and R. H. spoilers are within 1° for both inboard and outboard spoilers.
Adjustment is made by adjusting length of stop bolt in differential linkage. After adjustment is completed, tighten lock nut on stop bolt. Apply alignment mark.
When spoiler deflection does not satisfy the specified values, complete adjustment in accordance with the following procedures:

(a) When spoiler angles of both L. H. and R. H. are larger or smaller, adjust stop bolt in differential linkage.

(b) Spoiler deflections of L. H. or R. H. wing are large or small.

(i) If rotating angles of control wheel to the left and right are equal, accomplish re-adjustment so that inclinations of bell cranks in both wings are equal.

(ii) If wheel angles for L. H. and R. H. are not equal, adjust stop bolt for differential linkage.

(c) When deflections of inboard and outboard spoilers are not equal, adjust lengths of push rods for inboard and outboard spoilers independently. (When deflection is small, lengthen rod. When it is large, shorten it. For this adjustment, mis-alignment of rigging pin hole position may be necessary.)

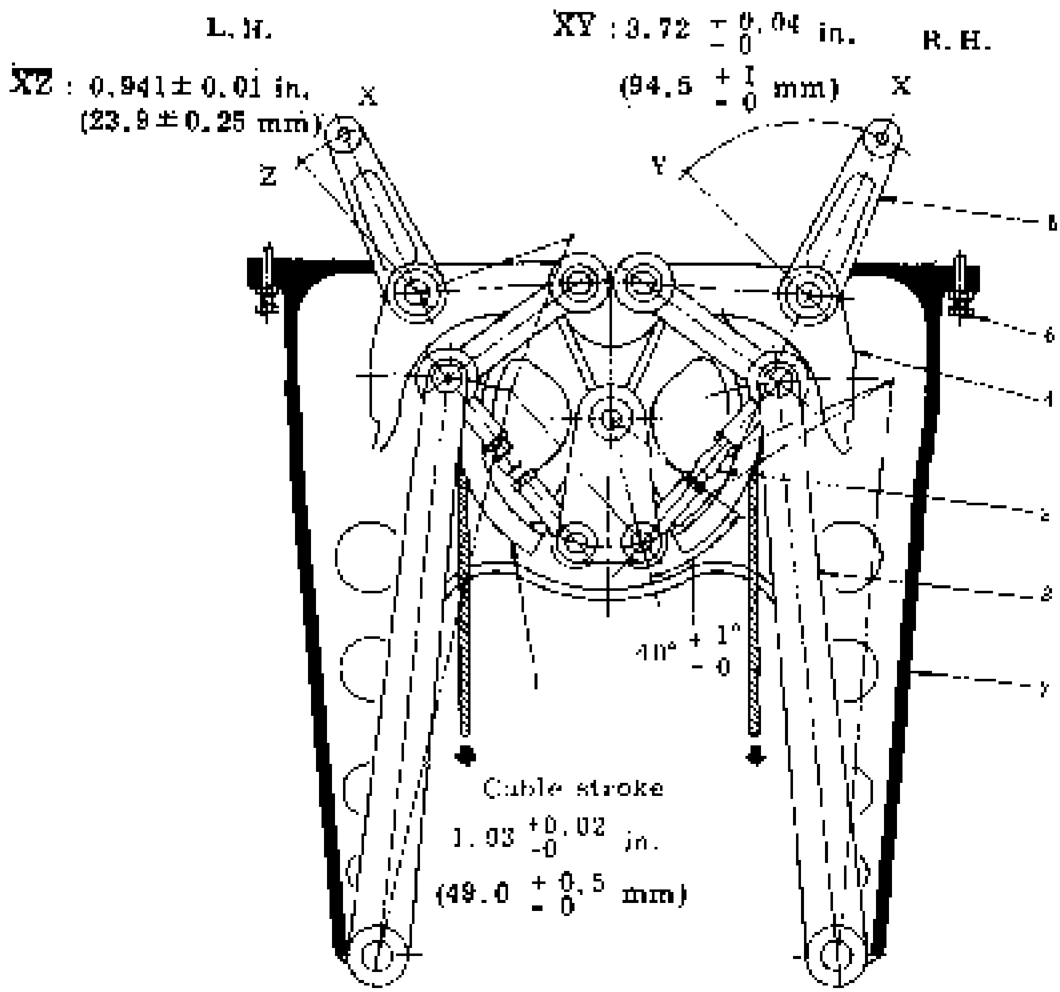
(d) When deflection is out of limits for any reason, re-adjust differential linkage in accordance with the following procedures. (See Fig. 5-43)

(i) Confirm that cable moves $1.93^{+0.02}_{-0}$ in. ($49.0^{+0.5}_{-0}$ mm) and quadrant turns $40°^{+1}_{-0}$ counterclockwise, when wheel is turned $80°^{+1}_{-0}$ (except free play) to the right.

(ii) Adjust length of R. H. rod assembly so that motion of R. H. bell crank along X-Y chord is $3.72^{+0.04}_{-0}$ in. (94.5^{+1}_{-0} mm) and adjust length of stop bolt so that bell crank makes contact.

(iii) Adjust L. H. side in the same manner as (ii).
Turn wheel or quadrant in the direction opposite to (i).
Confirm that motion of R. H. bell crank is 0.941 ± 0.01 in (23.9 ± 0.25 mm) along X-Z chord.

(iv) Motion of L. H. bell crank is 0.941 ± 0.01 in. (23.9 ± 0.25 mm) when R. H. bell crank is turned until it hits the stop.



1. Pulley
2. Rod assembly
3. Lever
4. Arm
5. Bellcrank
6. Stop bolt
7. Bracket

Fig. 5-43 Differential linkage

5.5.3 REMOVAL, INSTALLATION AND ADJUSTMENT OF FEEL SPRING (See Fig. 5-44)

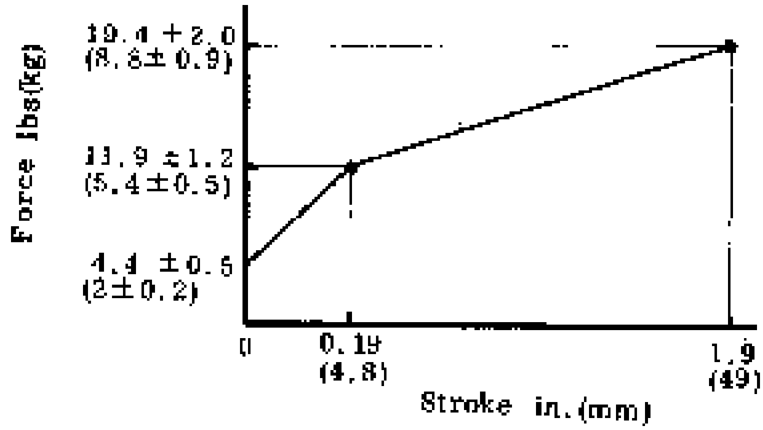
(1) Measure characteristics of feel spring.
Ensure relationship between stroke and load of spring conforms to figures given in the following cable.
If it is found to be abnormal, loosen the nuts in upper cylinder section and move cap for adjustment. If adjustment in this manner is not successful, replace spring.

NOTE

Check and confirm spring characteristics before installation of feel spring.

(2) Since the feel spring is assembled with the cable into a unit, it can only be removed after disconnecting the cable at the quadrant in the differential

linkage and at the turnbuckle.
When installing the assembly to the serration bracket, loosen the nuts at both ends sufficiently so that the spring is not deflected initially.



Characteristics of feel spring

NOTE

After feel spring is mounted on serration bracket, and tension of control cable is rigged, ensure no interference between inner cylinder of feel spring (in which cable is put through) and the cable. In case of interference, move serration bracket to left and right or adjust the bracket backward and forward by inserting adjusting shim in order to avoid interference.

(3) Finger-tighten the adjusting nuts on both ends of the feel spring. Do not twist the terminal in tightening or loosening the adjusting nuts and lock nut.
(4) After installation, tighten and lock the nuts at both ends, hold tab washers in position and apply alignment marks.

NOTE

Care must be taken not to over torque the nuts on both ends.

1. Feel spring body
2. Lock nut
3. Adjusting cap
4. Adjusting nut
5. Tab washer
6. Serration bracket
7. Serration plate
8. Adjusting shim
9. Terminal

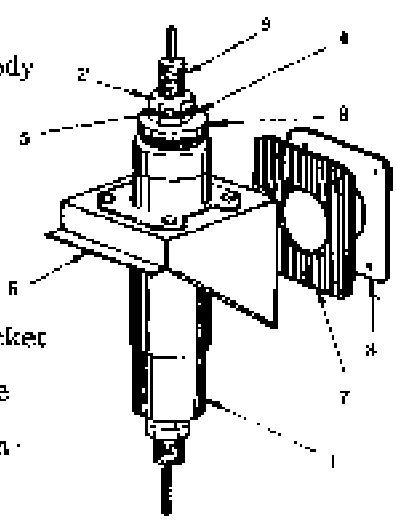


Fig. 5-44 Feel spring assembly

5.5.4 MEASUREMENT OF CONTROL FORCE

(1) Place push-pull scale on pilot's control wheel. Measure control force. (See Fig. 5-45)
Specified values are tabulated below.

lbs(kg)

Direction / Position	CONTROL FORCE Push (P_1)	Return (P_2)	FRICTION $P_1 + P_2$
Neutral	5(2.3) Max.	3(1.4) Max.	8(3.6) Max.
20° position	8±1(3.6±0.5)	-3±1(-1.4±0.5)	8(3.6) Max.
80° position	9±1.5(4.1±0.7)	-4±1(-1.8±0.5)	8(3.6) Max.

P_1 = Force increasing deflection angle.

P_2 = Force which returns control wheel to neutral position.

NOTE

Measure control force with autopilot servo control cables disconnected.

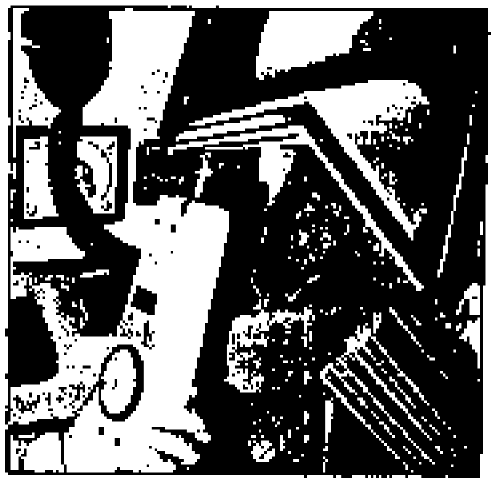


Fig. 5-45 Push-pull scale placed on control wheel

5.5.5 OPERATIONAL CHECK

(1) Check relation between travel of control wheel and that of control surface. When control wheel contacts with stop of differential linkage, maximum range of travel of control wheel shall conform to the value given below.

Rotation of control wheel	Maximum range of travel	Movement of control surface
Clockwise (CW)	80° +5° / -3°	LH down / RH up
Counter-clockwise(CCW)	80° -5° / -3°	LH up / RH down

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Difference between L.H. and R.H. control wheel range of travel should be less than 5°.

(2) Rotate control wheel clockwise and counter-clockwise.
Check for noise, interference and smooth movement of control surface.
When control forces are not normal, check the following.

(a) Interference in system.
(b) Chain and cable tension.
(c) Characteristics of feel spring.
(d) Lubrication.

6. ELEVATOR TRIM TAB CONTROL SYSTEM

6.1 GENERAL DESCRIPTION

Elevator trim tabs, installed in trailing edge of LH and RH elevators, are controlled by rotating trim wheel, located on the LH side of center pedestal. As trim wheel is rotated, cables wound on drum move, operating two trim actuators located in LH and RH horizontal stabilizers, deflecting tabs up and down. An indicator, which is driven in conjunction with wheel motion, is located above the wheel and indicates tab position. Cable runs aft between keels, under floor (2 LH lower ones) like primary control cables. Turnbuckles are located at STA 8370. (See Fig. 5-46)

6.2 REMOVAL AND INSTALLATION OF ELEVATOR TRIM TAB

(1) Remove rod assembly connecting bolt from bracket on bottom surface of tab. (See Fig. 5-47)

(2) Removing cotter pin at continuous hinge end, pull hinge pin. Remove tab from aircraft. (See Fig. 5-48)

(3) Install in reverse sequence of removal.

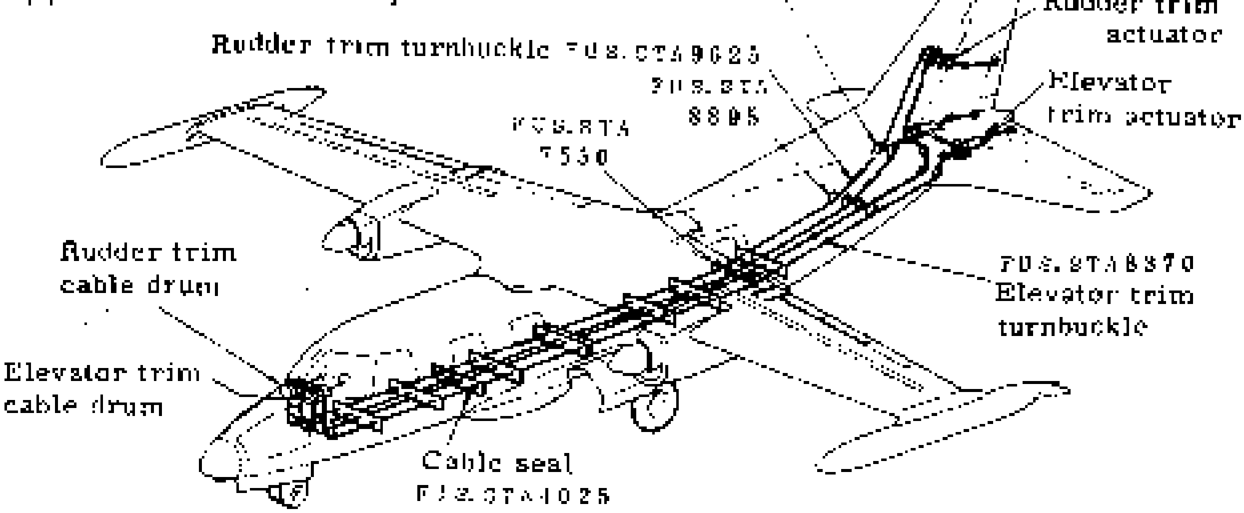

Fig. 5-46 Cables for elevator and rudder trim tab controls

Fig. 5-47 Remove bolt from bracket on bottom surface of tab

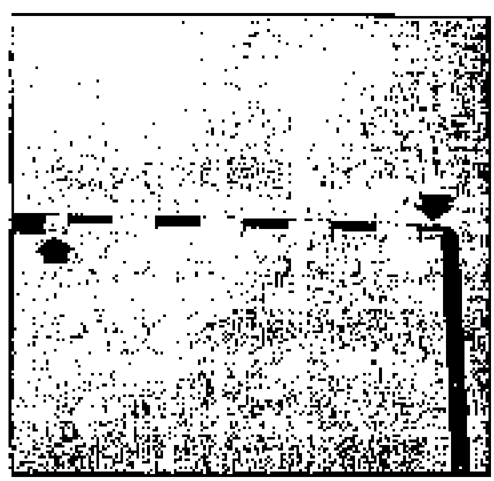

Fig. 5-48 Remove cotter pin at hinge end

6.3 REMOVAL AND INSTALLATION OF CONTROL CABLE AND CHAIN

(1) Insert gust lock pin into control wheel.

(2) Insert rig pin into drum of elevator trim forward of center pedestal in the cockpit. (See Fig. 5-49)

(3) Insert rig pin into elevator trim actuators (both L.H. and R.H.) (See Fig. 5-50).

(4) Disconnect cable from turnbuckle at F.STA 8370.

(5) Remove rig pin from drum.

(6) Rotating drum, remove cable terminal without impairing cable on drum of elevator trim. Remove forward cable from aircraft.

(7) Remove rig pin from trim actuator. Open access panel on bottom surface of inboard horizontal stabilizer. Remove pin from chain. Remove aft cable from aircraft. (See Fig. 5-51)

(8) Remove tail cone. Remove trim actuator chain cover. Install chain.

(9) Install in reverse sequence of removal. Note the following when installing chain.

(a) Adjust length of turnbuckle of chain to 6 ± 0.04 in. (152.4 ± 1 mm).

(b) Ensure no interference of pin on cable terminal with chain sprocket and pulley of trim actuator when trim is moved to maximum extent.

NOTE

i If drum is rotated inadvertently in removing control cable, cable may be entangled or compressed between guide pins and impaired.

ii When removing cable, a piece of string tied to cable end will facilitate re-installation.

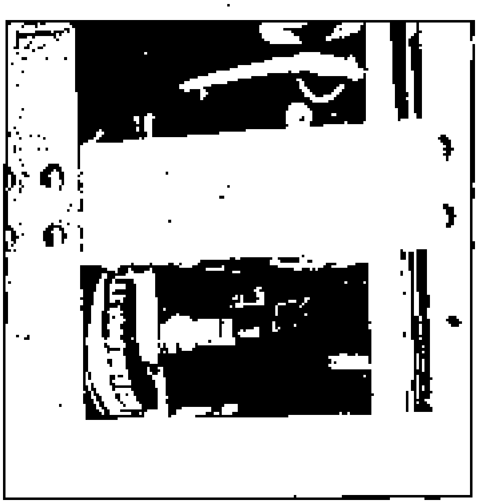

Fig. 5-49 Insert rig pin into drum

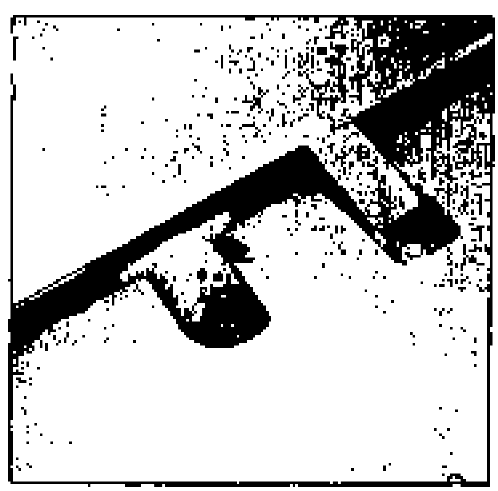

Fig. 5-50 Insert rig pin into trim actuator

Fig. 5-51 Disconnect cable and chain

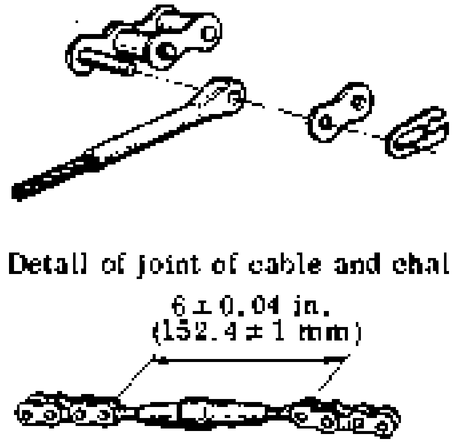

Fig. 5-52 Adjustment of turnbuckle

6.4 ELEVATOR TRIM ACTUATOR

6.4.1 REMOVAL AND INSTALLATION

(1) Insert gust lock pin into control wheel.

(2) Insert rig pin into drum of elevator trim in center pedestal.

(3) Remove bolt connecting trim tab to rod assembly of actuator.

(4) Remove 2 bolts attaching trim actuator from the gap between horizontal stabilizer and elevator.

(5) Open access panel of trim actuator. Remove 2 bolts attaching actuator.

(6) Remove chain guide. Remove chain from sprocket.
Remove trim actuator from aircraft. (See Fig. 5-53)
Pin may be removed from connecting part of cable and chain to facilitate removal of chain.

(7) Install in reverse sequence of removal.
Precautions for installation of trim actuator are shown below.

(a) Before installation, place trim actuator and rod assembly in neutral position and ensure dimension as indicated in Fig. 5-54.

(b) Tighten bolt connecting rod assembly with tab finger tight. Insert cotter pin securely.

NOTE

Al alloy washer (NAS 1197-10) should be placed under attaching bolt head when installing.

Fig. 5-53 Remove chain from sprocket

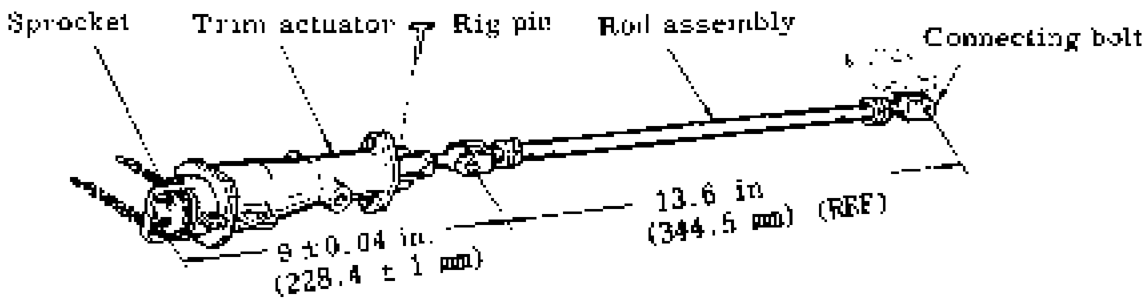

Fig. 5-54 Elevator trim actuator and rod assembly

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6.4.2 DISASSEMBLY OF ELEVATOR TRIM ACTUATOR (See Fig. 5-55)

(1) Remove elevator trim actuator.

(2) Loosen screw (1), and remove chain cover (2).

(3) Cut lockwire, remove bolts (7) and pull out sprocket shaft (5) from actuator box (13) together with end cover (8) and bearing (10).

(4) Rotate sprocket with hand. Pull out screw shaft (14) from sprocket shaft (5).

(5) Set up tabs of tab washer (11) and remove nut (12).

(6) Pull end cover (8) and bearing (10) out of sprocket shaft (5).

(7) Remove lock ring (3) and remove cap (4) from sprocket shaft (5).

(8) Remove bolts and nuts (16) and pull spline shaft (15) out of actuator box (13).

(9) Disassemble further, if necessary.

6.4.3 INSPECTION AND LUBRICATION OF ELEVATOR TRIM ACTUATOR

(1) Wash all metallic parts with dry cleaning solvent (P-D-680) and dry.

(2) Visually check all parts for abnormal wear, damage, crack or corrosion and replace if necessary.

(3) Apply grease (MIL-G-23827) on the following points during reassembly.

a. Ball and needles of bearing (10)

b. Thread and spline of screw shaft (14)

c. Internal surface of spline shaft (15), especially internal spline and internal surface of bushing (17)

6.4.4 REASSEMBLY OF ELEVATOR TRIM ACTUATOR

(1) Reassemble in reverse sequence of disassembly.

(2) Make sure that the actuator actuates smoothly for full stroke after reassembly.

(3) Adjust the length of rod end (19) so that distance between outer side of sprocket and center of rod end hole may be 9 ± 0.04 in (228.4 ± 1 mm) when actuator is set to neutral position and rig pin is inserted, and tighten lock nut (18). (See Fig.5-54)

(4) Install elevator trim actuator on the airplane.

6.5 RIGGING

6.5.1 RIGGING OF CABLE TENSION

(1) Insert rig pin into elevator aft quardrant. (See Fig.5-13)

(2) Insert rig pins into L.H. and R.H. trim actuators.

(3) Insert rig pin into cable drum in center pedestal.

(4) Rotating turnbuckle at F.STA 8370, adjust cable tension to the specified value of 10^{+5}_{-0} lbs (70°F) ($4.5^{+2.3}_{-0}$ kg) (21°C).
Compensation of cable tension for temperatures shall be performed in accordance with Fig 5-62.

(5) After rigging is completed, install safety wire on turnbuckle.

(6) Remove rigging pins.

NOTE

After rig pins are removed, chain must not be looser than it has been before rigging.

1. Screw and nut
2. Chain cover
3. Lock ring
4. Cap
5. Sprocket shaft
6. Oil seal
7. Bolt
8. End cover
9. Packing
10. Bearing
11. Tab washer
12. Nut
13. Actuator box
14. Screw shaft
15. Spline shaft
16. Bolt and nut
17. Bush
18. Lock nut
19. Rod end

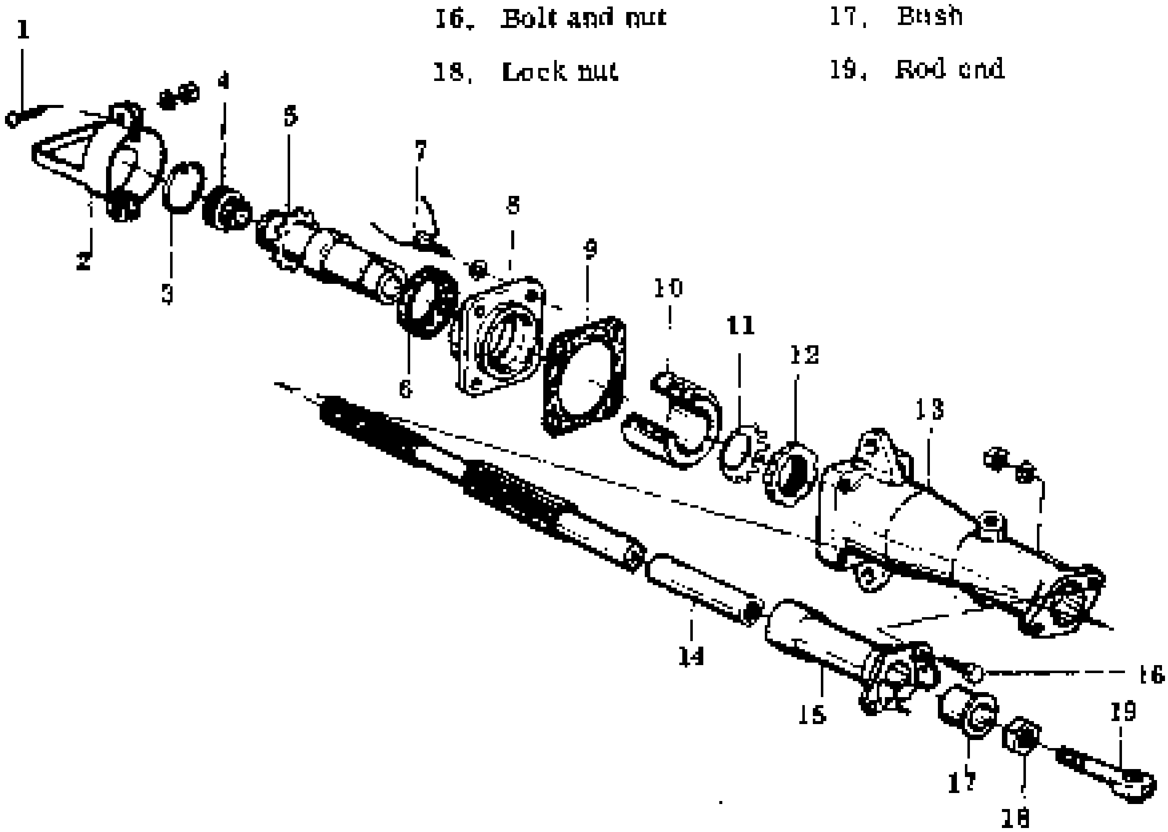


Fig.5-55 Elevator trim actuator

6.5.2 RIGGING OF DEFLECTION

After cable rigging is accomplished, rig tab deflection in accordance with the following procedures.

(1) Insert rig pin into elevator aft quadrant.
(2) Insert rig pin into trim actuator.
(3) Adjust length of rods so that mis-match of trim tab trailing edges and elevator trailing edges is less than ±0.04 in. (±1 mm).

NOTE

(i) Check to see if a threaded portion of clevis is visible through the inspection hole in the operating rod.
(ii) If threads are not visible, loosen the other end of the operating rod and adjust both ends so that threaded portions are visible through inspection holes, and alignment is achieved at trailing edge.

(4) Attach protractor to trim tab at H.STA 400. Set pointer to "0".
(5) Remove rig pin from trim actuator.
(6) Operate elevator trim wheel and check the relation between the indication and deflection.

Trim wheel	Indication	Tab deflection
"UP" Turn to "NOSE UP" direction Reach to stop after 4 4/5 turns	30° ± 2°	DOWN 30° $^{+1}_{-0}$
NEUTRAL	0°	0° ± 1°
"DOWN" Turn to "NOSE DOWN" direction STOP will be reached after 1 3/5 turns (*1) STOP will be reached after 1/6 turn (*2)	10° ± 2° (*1) 1° (*2)	UP 10° $^{+1}_{-0}$ (*1) 1° $^{+1}_{-0}$ (*2)

(*1) Not modified by S/B 079/27-010
(*2) Modified by S/B 079/27-010

(7) Remove rig pin from elevator aft quadrant.
(8) Tab should not move more than ±2° (or 0.20 in. (5.1 mm) at tab inboard trailing edge) at 33° up and 10° down positions of elevator when control wheel is fully pulled or pushed with the condition of trim indicator setting to the neutral position "0". Adjustment is made by adjusting the length of actuator [9 ±0.04 in. (228.6±1 mm) ref.] or by adjusting location of actuator.
(9) After rigging is completed, tighten lock nut on rod assembly, and apply alignment mark.

6.5.3 RIGGING OF TRIM WHEEL

Trim wheel control force is 1.3 to 2.2 lbs (0.6 to 1.0 kg).
If the control force is beyond this limit, rerig trim wheel in accordance with the following procedures. (See Fig.5-56)

(1) Remove cap from center of trim wheel.
(2) Remove screw from nut holding wheel.
(3) Loosen or retorque nut holding wheel so that trim wheel may be rotated with specified control force.
(4) Tighten screw on nut to hold wheel in position. Reinstall cap.

6.5.4 OPERATIONAL CHECK

(1) Operate elevator trim wheel through full range of travel.
Check for interference with structure or other equipment, binding, noise and smooth movement of control surface.

(2) Check cable rods and turnbuckles of entire system for damage and satisfactory locking.

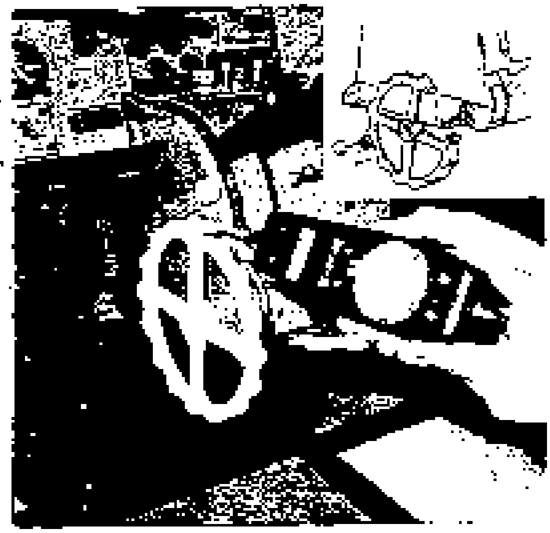


Fig. 5-56 Installation and adjustment of trim wheel

7. RUDDER TRIM TAB CONTROL SYSTEM

7.1 GENERAL DESCRIPTION

The rudder trim tab, installed on the lower section of the rudder trailing edge, is controlled by rotating the trim wheel, located on R.H. lower section of the center pedestal. As trim wheel is turned, cables wound on drum move, operating trim actuator, located in vertical stabilizer, deflecting tab. Indicator, which is actuated in conjunction with wheel motion, is located in the upper section of wheel and indicates tab position. Cables run rearward between keels under floor (2 R.H. lower ones), as do the cables for the primary control systems. Turnbuckles are located near F.STA 9625. (See Fig. 5-46)

7.2 REMOVAL AND INSTALLATION OF RUDDER TRIM TAB

(1) Remove rod assembly connecting bolt from bracket. (See Fig. 5-57)

(2) Removing cotter pin at continuous hinge end, pull hinge pin.

(3) Install in reverse sequence of removal.

7.3 REMOVAL AND INSTALLATION OF CONTROL CABLE AND CHAIN

(1) Insert rig pin into drum of rudder trim forward of center pedestal in the cockpit.

(2) Insert rig pin into rudder trim actuator. (See Fig. 5-58)

(3) Insert rig pin into quadrant of rudder pedal for pilot.

(4) Disconnect cable from turnbuckle at F.STA 9625.

(5) Remove rig pin from drum. Rotating drum, remove cable terminal without impairing cable on drum of rudder trim. Remove forward cable from aircraft.

(6) Remove rig pin from trim actuator. Open access panel of vertical stabilizer. Remove pin from chain.
Remove aft cable from aircraft. (See Fig. 5-59)

(7) Install in reverse sequence of removal.
Note the following when installing chain.

(a) Install chain so that both ends may meet at neutral position when rig pin is inserted in actuator.

(b) Ensure no interference of pin on cable terminal with chain sprocket and pulley of trim actuator when trim is moved to maximum extent.

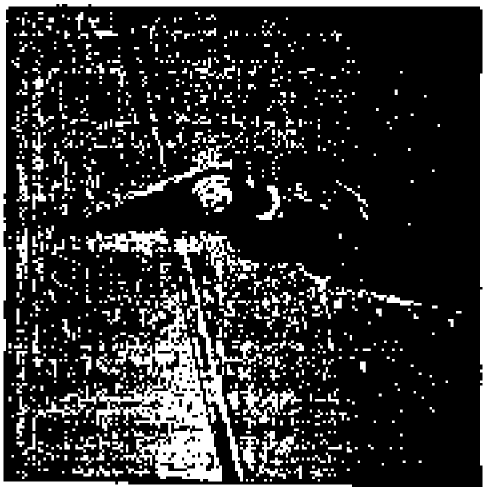


Fig. 5-57 Remove connecting bolt from bracket

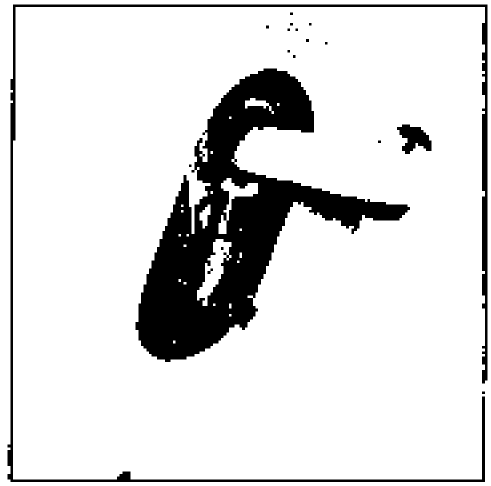


Fig. 5-58 Insert rig pin into trim actuator

NOTE

i. If drum is rotated inadvertently when removing control cable, cable may be entangled or compressed between guard pin and impaired.

ii. When removing cable, a piece of string tied to cable end will facilitate re-installation.

Fig. 5-59 Remove connecting pin and cable

7.4 RUDDER TRIM ACTUATOR

7.4.1 REMOVAL AND INSTALLATION OF RUDDER TRIM ACTUATOR

(1) Insert rig pin into quadrant of rudder pedal, pilot side (see Fig 5-22).
(2) Insert rig pin into drum of rudder trim in center pedestal.
(3) Disconnect cable from turnbuckle at F STA 9625.
(4) Remove chain guide. Remove chain from sprocket. Pin may be removed from connecting part of cable and chain to facilitate removal of chain.
(5) Remove 4 bolts attaching trim actuator. Remove trim actuator from aircraft.
(6) Install in reverse sequence of removal. Precautions for installation of trim actuator are shown below.
   (a) Before installation, place trim actuator and rod assembly in neutral position and ensure dimension as indicated in Fig 5-60.
   (b) Tighten bolt connecting rod assembly with tab finger tight. Insert cotter pin securely.

NOTE

Al alloy washer (NAS1197) should be placed under attaching bolt head when installing.

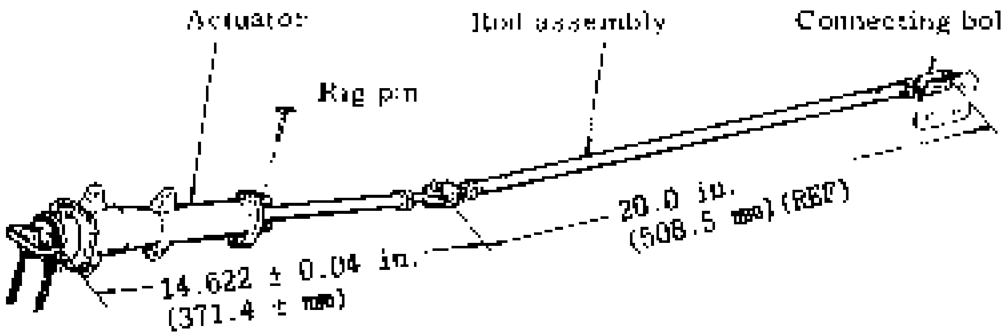


Fig.5-60 Rudder trim actuator and rod assembly

7.4.2 DISASSEMBLY OF RUDDER TRIM ACTUATOR (See Fig.5-61)

(1) Remove rudder trim actuator.

(2) Loosen screw (1), remove chain cover (2).

(3) Cut lockwire, remove bolts (7) and pull out sprocket shaft (5) from actuator box (13) together with end cover (8) and bearing (10).

(4) Rotate sprocket with hand. Pull out screw shaft (14) from sprocket shaft (5).

(5) Set up tabs of tab washer (11) and remove nut (12).

(6) Pull end cover (8) and bearing (10) out of sprocket shaft (5).

(7) Remove lock ring (3) and remove cap (4) from sprocket shaft (5).

(8) Remove bolts and nuts (16) and pull spline shaft (15) out of actuator box (13).

(9) Disassemble further, if necessary.

7.4.3 INSPECTION AND LUBRICATION OF RUDDER TRIM ACTUATOR

(1) Wash all metalic parts with dry cleaning solvent (P-D-680) and dry.

(2) Check all parts for abnormal wear, damage, crack or corrosion visually and replace if necessary.

(3) Apply grease (MIL-G-23827) on the following points during reassembly .
   a. Balls and needles of bearing (10)
   b. Thread and spline screw shaft (14)
   c. Internal surface of spline shaft (15), especially internal spline and internal surface of bushing (17)

1. Screw and nut	2. Chain cover	3. Lock ring
4. Cap	5. Sprocket shaft	6. Oil seal
7. Bolt	8. End cover	9. Packing
10. Bearing	11. Tab washer	12. Nut
13. Actuator box	14. Screw shaft	15. Spline shaft
	16. Bolt and nut	17. Bushing
	18. Lock nut	19. Rod end

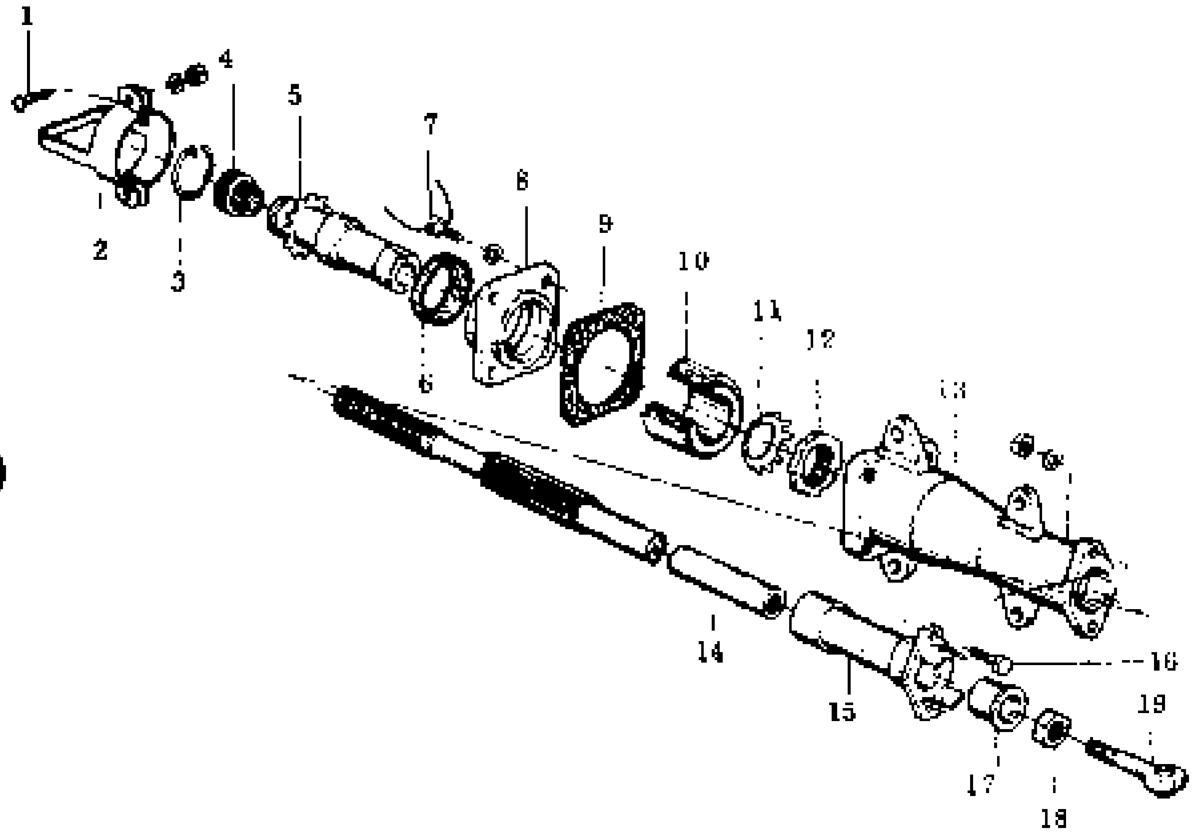


Fig.5-61 Rudder trim actuator

7.4.4 REASSEMBLY OF RUDDER TRIM ACTUATOR

(1) Reassemble in reverse sequence of disassembly.

(2) Make sure that the actuator actuates smoothly for full stroke after reassembly.

(3) Adjust the length of rod end (19) so that distance between outer side of sprocket and center of rod end hole may be 14.622 ± 0.04 in. (371.4 ± 1 mm) when actuator is set to neutral position and rig pin is inserted, and tighten lock nut (18). (See Fig. 5-61)
Ensure that the bolt hole in the rod end (19), swivel bearing, when centered, is parallel with the rig pin.

(4) Install rudder trim actuator on the airplane.

7.5 RIGGING

7.5.1 RIGGING OF CABLE

(1) Insert rig pin into trim actuator.

(2) Insert rig pin into cable drum in center pedestal.

(3) Remove access panel forward of tail cone. Insert rig pin into rudder aft quardrant.

(4) Rotating turnbuckle at F.STA 9625, adjust cable tension to specified value of 10^{+5}_{0} lbs (70°F) ($4.5^{+2.3}_{-0}$ kg) (21°C).
Compensation of cable tension for temperatures shall be performed in accordance with Fig.5-62.
After rigging is accomplished, install safety wire on turnbuckle.

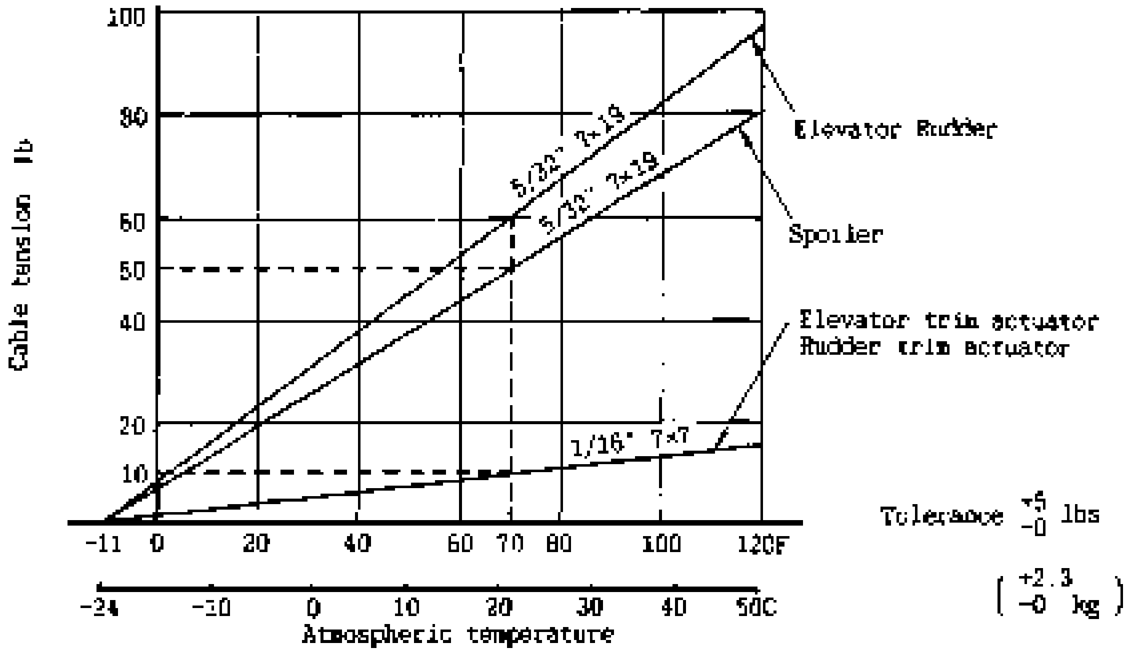


Fig.5-62 Temperature compensation table for cable tension

(5) Remove rig pins.

7.5.2 RIGGING OF DEFLECTION

After cable rigging is accomplished, rig tab deflection in accordance with the following procedures.

(1) Insert rig pin into trim actuator.

(2) Insert rig pin into rudder aft quadrant.

(3) Adjust length of rods so that mis-matching of trim tab trailing edges and rudder trailing edges is less than ± 0.06 in.

(4) Attach protractor to vertical stabilizer. (V.STA 1470)

(5) Remove rig pin from aft quadrant trim actuator.

(6) Operate rudder trim wheel and check the relation between the indication and deflection.

Trim wheel		Indication	Deflection
Right	Turn to "NOSE RIGHT" direction. Reach to stop after 4 turns	RIGHT 25° ± 2°	To left $25^{+1°}_{-0}$
Neutral		0°	0 ± 1°
Left	Turn to "NOSE LEFT" direction. Reach to stop after 4 turns	LEFT 25° ± 2°	To right $25^{\circ}\,{}^{+3°}_{-0}$

(7) Remove protractor.

(8) Tab should not move more than ± 2° (or 0.20 in. at tab inboard trailing edge) when rudder pedal is fully pushed right or left with the condition of trim indicator setting to neutral position.
Adjustment is made by changing actuator length or changing actuator attaching location. (Reference actuator length 14.622 ± 0.04 in.)

(9) After rigging is completed, tighten lock nut on rod assembly, and apply alignment mark.

7.5.3 OPERATIONAL CHECK

(1) Operate rudder trim wheel through full range of travel. Check for interference with structure or other equipment, binding, noise and smooth movement of control surface.

(2) Check cable rods and turnbuckles of entire system for damage and satisfactory locking.

8. AILERON TRIM CONTROL SYSTEM

8.1 GENERAL DESCRIPTION

The aileron type trim consists of inboard and outboard surfaces located in the inside section of L.H. and R.H. outboard flaps, and is operated by an electric actuator.

Actuators, bell cranks, push rods, etc., are located in flaps and are electrically connected with wings. If the control switch located in the center of the center pedestal is held in the left or right position, the actuators in the flaps are energized to move the control surfaces. The momentarily ON switch springs back to its original neutral position, when it is released, stopping actuators at once. If the switch is held in the operating position, the actuators are stopped at the maximum deflected position automatically by a built-in limit switch. Actuator motion is picked up by potentiometers and shown on an indicator located above the control switch. Inboard and outboard aileron trim tabs connected by linkage are moved in the same direction. The L. H. and R. H. electrically connected aileron type trim tabs move in opposite directions to each other.
Although operating speeds of the L. H. and R. H. actuators are usually different, accumulation of differences is avoided because the actuator is stopped momentarily by means of neutral limit switch.
When the aileron trim selector switch, located on the switch panel, is selected in the right or left position, interconnection between L. H. and R. H. tabs is disconnected and the control surface which has been selected can be operated independently. Therefore, flight safety is assured even if the control surface on one side is in-operative.

E. 2 REMOVAL AND INSTALLATION OF AILERON TYPE TRIM

(1) Remove bolts connecting inboard and outboard push rods.
(2) Remove hinge pin holder of trim aileron, draw the pin out and remove trim aileron from aircraft. (See Fig. 5-63)
(3) Install in reverse sequence of removal.

Fig. 5-63 Remove connecting bolt and hinge pin holder

8.3 REMOVAL AND INSTALLATION OF TRIM AILERON ACTUATOR (See Fig 5-64)

(1) Place trim aileron in neutral position.

(2) Remove actuator access panel.

(3) Disconnect wiring to actuator and switches and identify.

(4) Remove cotter pin, nut, washer and bolt connecting the bellcrank with the rod end of the actuator.

(5) Remove cotter pin, nut, washer and bolt attaching actuator.

(6) Remove actuator from the aircraft.

(7) Install in reverse sequence of removal, noting the following.

(a) Before installing, adjust stroke "L" (Fig 5-64) of trim actuator to 8.62 +0.04/-0.00 in. (219 +1/-0 mm) at retracted position. The rigging is made by means of the actuator rod end and EXT and RET screws of the trim actuator limit switch. After rigging, install safety wire on the screws.

(b) Install actuator and connect bellcrank.

NOTE

Place actuator in RET position to facilitate installation.

(c) Tighten attaching hardware to proper torque, install cotter pin and apply alignment mark.

(d) Remove wire identification and connect wiring.

(e) Perform operational check in accordance with Para 8.5.

8.4 REMOVAL AND INSTALLATION OF TRIM AILERON BELLCRANK BRACKET (See Fig 5-64)

(1) Place trim aileron in neutral position.

(2) Remove actuator access panel.

(3) Disconnect wiring to actuator and switches and identify.

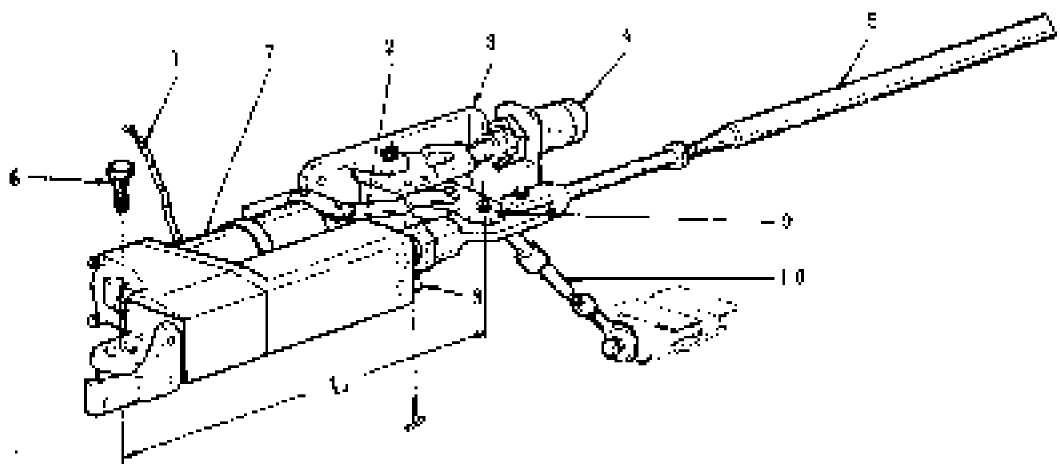
(4) Remove push-pull rod from bracket on outboard trim aileron control surface.

(5) Remove cotter pin, nut washer and bolt connecting the bellcrank with the rod end of the actuator.

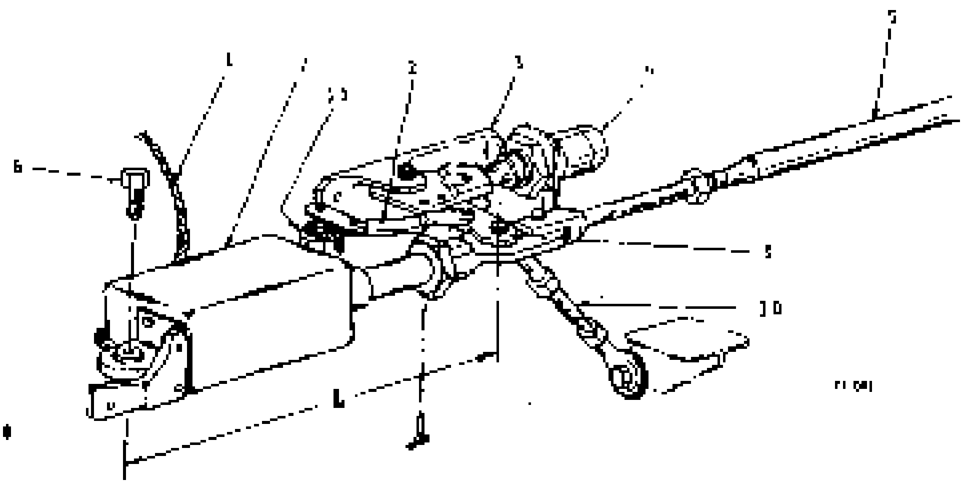
(6) Remove 2 bolts attaching neutral switch to bracket and remove switch.

(7) Remove 4 bolts attaching bellcrank bracket assembly to the aircraft and remove bracket assembly.

(8) Install in reverse sequence of removal; after torquing attaching hardware, apply alignment mark. Remove wire identification and connect wiring. Perform operational check in accordance with Para 8.5.



Aircraft S/N 652SA, 661SA, 697SA thru 713SA and 715SA



Aircraft S/N 714SA, 716SA and subsequent

1. Wire
2. Potentiometer arm
3. Bracket
4. Neutral limit switch
5. Push-pull rod
6. Attaching bolt
7. Actuator
8. Limit switch
9. Bellcrank
10. Push rod
11. Poisiton Indicator Potentiometer

Fig. 5-64 Installation of trim aileron actuator

8.5 RIGGING OF DEFLECTION AND OPERATIONAL CHECK

NOTE

Do not operate trim actuator continuously for checking after installation.
Stop for 3 to 5 minutes after 2 consecutive operations.

8.5.1 LOCATING NEUTRAL POSITION

(1) Insert rig pin into inboard and outboard bellcranks.
(2) Attach protractor to control surface at upper portion of push rod (see Fig 5-65).
(3) Adjust the lengths of outboard and inboard push rods so that mismatching of trim tab trailing edge and flap trailing edge is less than 0.04 in. (1 mm) at trim tab outboard edge. (For mismatching of inboard edge and flap actuator box, see Para. 9.2.7). After rigging, tighten lock nut on the rod. Check through inspection hole for proper length of rod screwed in.
(4) Remove rig pin.

8.5.2 RIGGING OF DEFLECTION

(1) Operate trim aileron as follows:
Close circuit breakers TRIM AIL and TRIM POS.
Place control switch in LOWER LH WING or LOWER RH WING position.
(2) Adjust limit switches located in the actuators so that deflection is $20^\circ +2^\circ/-0^\circ$ in down direction. After rigging, install safety wire.
(3) Adjust neutral limit switch of trim aileron so that switch may operate satisfactorily at neutral position ($0 \pm 1^\circ$) (see Fig 5-66). Loosen switch lock nut so that plunger of neutral limit switch may be ON at 0° of deflection. After rigging, install safety wire. Confirm that plunger of switch is not bottoming and has sufficient margin of stroke at ON position.

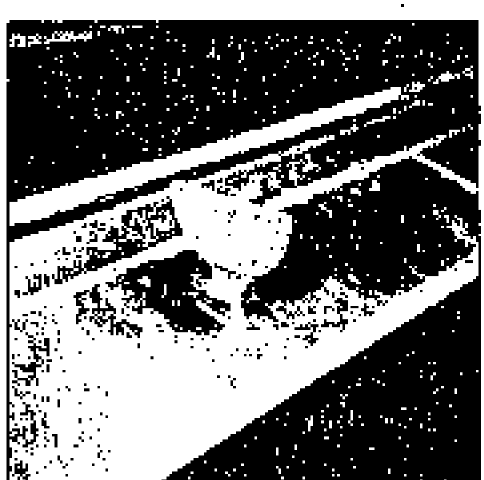


Fig. 5-65 Attach protractor to control surface

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8.5.3 ADJUSTMENT OF POTENTIOMETER

(1) Remove actuator access panel to gain access to the trim aileron indicator potentiometer.

NOTE

It may be necessary to remove the potentiometer-bellcrank assembly from the aircraft.

(2) Insert a rig pin through the bellcrank for neutral position and connect a multitester to terminals per below. Loosen lever arm set screw and turn potentiometer shaft to indicate 5 ± 0.5 K ohms resistance. Tighten lever arm set screw against potentiometer shaft.

LH potentiometer - Pins B (2) and CCW (1)

RH potentiometer - Pins B (2) and CW (3)

(3) Remove rig pin and install potentiometer-bellcrank assembly in position, tighten attaching bolts and nuts to proper troque value and apply alignment mark.

(4) Connect electrical wires.

(5) Close circuit breakers TRIM AIL and TRIM AIL POS. Adjust resistor at the back of center pedestal, so that trim aileron indicator may read 0 with control surface held in neutral position. After the adjustment, lock resistor knob securely. (See Fig. 5-68)

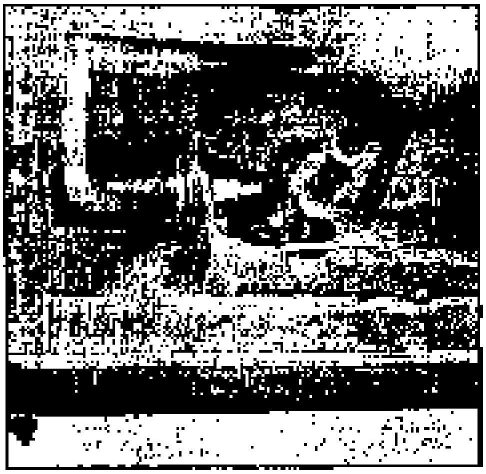


Fig. 5-66 Adjustment of neutral limit switch

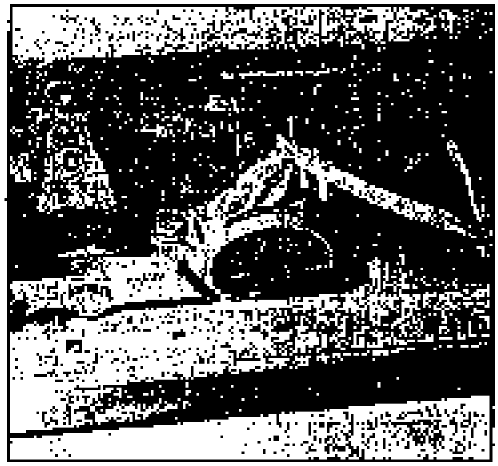


Fig. 5-67 Adjustment of potentiometer

8.5.4 OPERATION OF TRIM AILERON

(1) Close circuit breakers TRIM AIL and TRIM POS. Place selector switch in BOTH position.

(2) Place trim aileron control switch on center pedestal in LOWER LH WING or LOWER RH WING position, and the control surface will move as follows:

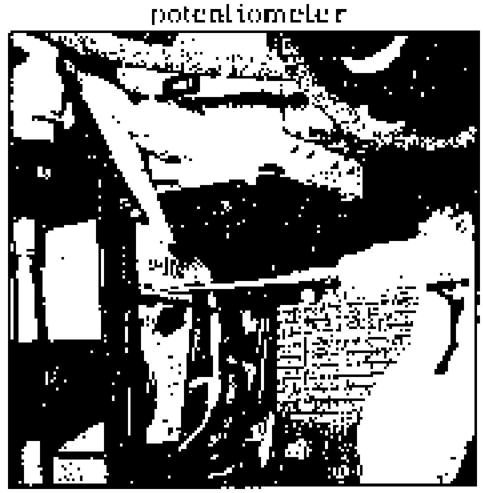


Fig. 5-68 Adjustment of "0" position

Position Control switch	Movement of control surface		Swing of Indicator
	LH	RH	
LOWER L WING	Up	Down	Left
LOWER R WING	Down	Up	Right

If trim aileron is operated in opposite direction, check wiring and correct.

(3) Place selector switch in LH EMERG or RH EMERG positions.

(4) Place trim aileron control switch in LOWER LH WING or LOWER RH WING positions, and the control surface will move as follows:

Selector switch	Control switch	Movement of control surface
LH EMERG	LOWER LH WING	LH alone up
	LOWER RH WING	LH alone down
RH EMERG	LOWER LH WING	RH alone down
	LOWER RH WING	RH alone up

Indicator reading at maximum deflection of trim aileron is half as much as that of normal operation where selector switch is in BOTH position.

(5) Operate trim aileron in normal way with selector switch in BOTH position, observe indicator and measure deflection angle and time required.

Trim aileron control switch	Indicator	Deflection angle		Time required		Operating voltage
		Left	Right	Left	Right	
LOWER LH WING	Swings to left	Up $20^{\circ}{}^{-2^{\circ}}_{-0^{\circ}}$	Down $20^{\circ}{}^{+2^{\circ}}_{-0^{\circ}}$	9 ± 3 sec.	9 ± 3 sec.	28V～29V
LOWER RH WING	Swings to right	Down $20^{\circ}{}^{+2^{\circ}}_{-0^{\circ}}$	Up $20^{\circ}{}^{-2^{\circ}}_{-0^{\circ}}$			

8.5.3 CHECK

(1) Check actuating mechanism for interference, actuator rod and bellcrank for damage, lock nut for security of installation, and alignment mark for proper location.

(2) Control surface should stop automatically at $0 \pm 1^{\circ}$ and $20^{\circ}{}^{+2^{\circ}}_{-0^{\circ}}$.
If control surface does not stop, limit switch is defective.
When control surface deflection can not be kept within specified limits, adjust limit switch.

(3) Also confirm that direction of motion shown on indicator is the same as the direction which was turned by control switch, and the indication also corresponds to neutral position or maximum deflected positions.

(4) Confirm that surface moves from neutral position to maximum deflection in 9 ± 3 seconds. If this requirement is not met, check actuator and replace it, if necessary.

(5) Confirm that R. H. aileron trim tab moves independently while L. H. aileron trim tab keeps stopping, when control switch is set with selector switch in the right position, and in the same way, when selector switch is turned to left, L. H. aileron trim tab moves alone.

9. FLAP CONTROL SYSTEM

9.1 GENERAL DESCRIPTION

The double slotted, full span flap, installed on the wing trailing edge, consists of vanes and a main flap, which are further divided into nacelle, inboard, and outboard flaps.

The driving mechanism consists of the following components. (See Fig. 5-69)

(1) Electric motor (See Fig. 5-70)

(2) Reduction gear box (See Fig. 5-70)

(3) Stop assembly (See Fig. 5-70)

(4) Main actuator (See Fig. 5-71)

(5) Inboard auxiliary actuator (See Fig. 5-72)

(6) Outboard auxiliary actuator (See Fig. 5-73)

(7) Guide mechanism (See Fig. 5-74)

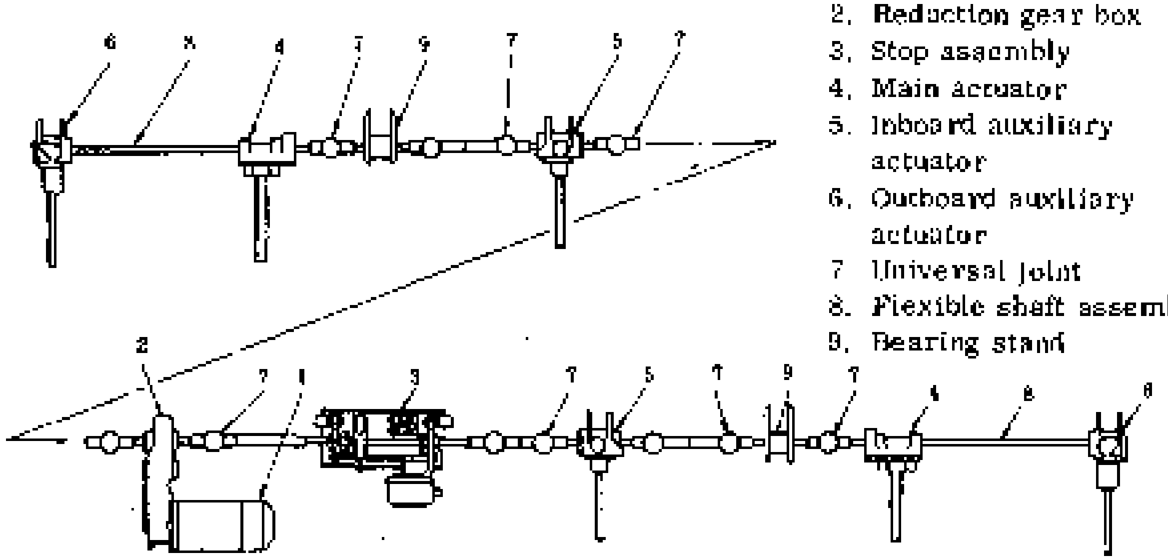
(8) Torque tube assembly

(9) Flexible shaft assembly

Each actuator consists of a bevel gear box, jack screw, jack nut, guide rail, rollers and links and transforming torque tube rotation to flap deflection.
Flap moves backward along guide rail and then turns with its center of rotation at a rear end guide rail when the flap touches it, resulting in a specific deflection. Limit switch is energized and stops the flap at the same time.
Flap deflection is determined by main actuators, and auxiliary actuators only correct fore-and-aft mis-alignment of flaps.
The jack screw in the L. H. wing is a L. H. threaded screw and in the R. H. wing is a R. H. threaded screw.
The stop assembly protects the system, overcoming stall torque of motor, when limit switch is inoperative. Control switch in cockpit has 3 or 4 positions, i.e., UP, 5°, 20°, and 40°. When switch is placed in any one of them, the flap stops automatically in the selected position, and the indicator lamp illuminates.
Between 1° and 20°, the control surface is moved intermittently.

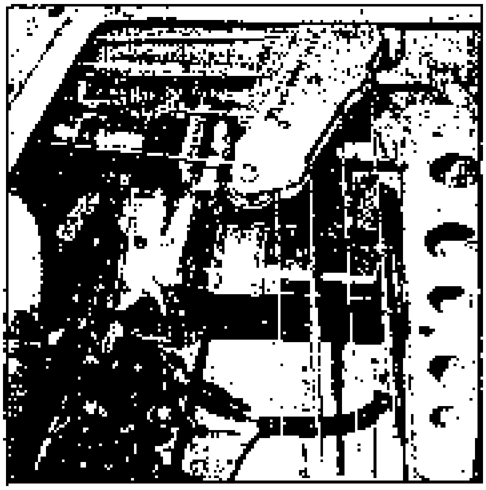
In normal operation, flap does not stop at intermediate positions.

When greasing nuts of inboard auxiliary actuator and main actuator, inject grease (MIL-G-21164) until the grease overflows from top of nuts or threads.



1. Flap motor
2. Reduction gear box
3. Stop assembly
4. Main actuator
5. Inboard auxiliary actuator
6. Outboard auxiliary actuator
7. Universal joint
8. Flexible shaft assembly
9. Bearing stand

Fig. 5-69 Flap driving mechanism



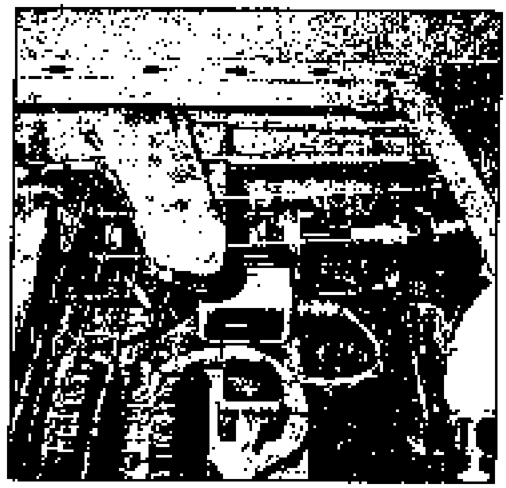


Fig. 5-70. Flap motor, reduction gear box and stop assembly

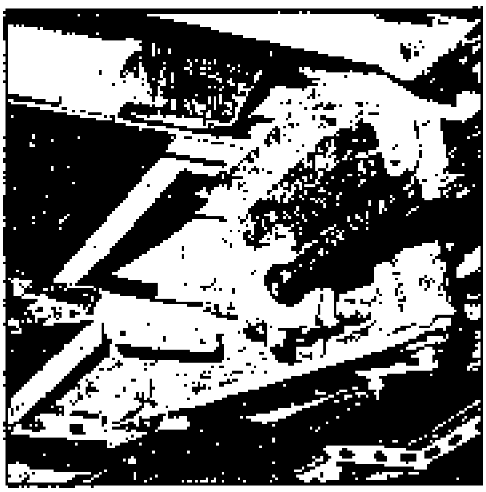


Fig. 5-71 Main actuator

Fig. 5-72 Inboard auxiliary actuator

Fig. 5-73 Outboard auxiliary actuator

Fig. 5-74 Control guide (Outboard flap)

9.2 REMOVAL AND INSTALLATION OF FLAP

Removal and installation of flap is accomplished in accordance with the following procedures. Detailed procedures are described on the following pages.

(1) Remove nacelle fairing (See Para. 9.2.1)

(2) Disconnect central main actuator jack screw from gear box. (See Para. 9.2.2)

(3) Disconnect inboard actuator jack screw from gear box. (See Para. 9.2.3)

(4) Disconnect outboard actuator jack screw from roller assembly fitting. (See Para. 9.2.4)

(5) Remove guide link mechanism from outer flap. (See Para. 9.2.5)

(6) Disconnect trim aileron wiring routed from outer flap to wing. (See Para. 9.2.6)

(7) Disconnect inboard flap and outboard flap from central main actuator box assembly. (See Para. 9.2.7)

After above processes are completed, remove inner flap, outer flap and central main actuator box assembly from aircraft.
Install in reverse sequence of removal.

9.2.1 REMOVAL AND INSTALLATION OF NACELLE FAIRING

(1) Place the flap in the extended position.

(2) Remove 5 screws on the upper side of nacelle fairing trailing edge.

(3) Remove 6 screws on the upper and inner side of nacelle fairing. (See Fig. 5-75)

(4) Remove access panel. Remove 4 bolts fastening nacelle fairing and flap. (See Fig. 5-76)

(5) Remove nacelle fairing from aircraft.

(6) Install in reverse sequence of removal.

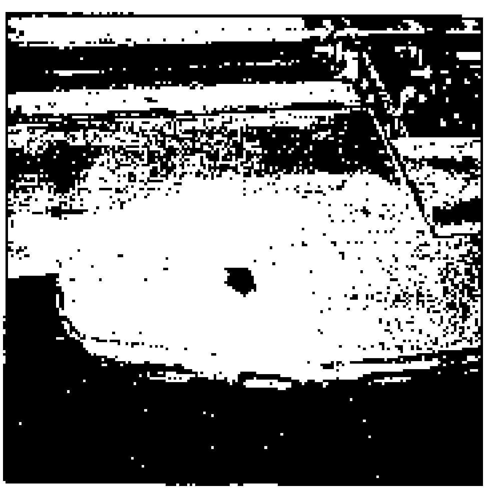


Fig. 5-75 Remove screws from nacelle fairing

Fig. 5-76 Remove bolts fastening nacelle fairing and flap

9.2.2 REMOVAL AND INSTALLATION OF CENTRAL MAIN ACTUATOR JACK SCREW (See Fig. 5-77)

Removal should be done as follows.

(1) Place flap at 20° down position.

(2) Remove cap from jack support.

(3) Disengage jack screw from serration of gear box with flap held by hand.

(4) Take jack screw accompanied by bearing out of nut support by rotating by hand.

NOTE

Do not allow flap to be lowered, when removing jack screw from nut support.

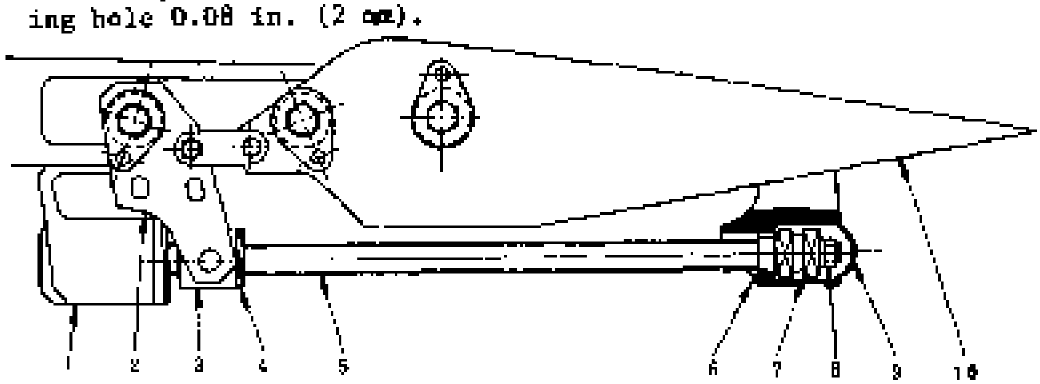
Installation should be done as follows.

(1) Place jack nut on gimbal.

(2) Screw in jack screw from back of jack support.

(3) Screw in jack screw until serration at the tip of jack screw reaches gear box.

(4) Connect jack screw to gearbox with flap held at 20° down position. After inserting jack screw into gearbox serrated hole, turn jack screw to make sure of smooth operation. If jack screw does not move smoothly, loosen gearbox attaching screws once with jack screw connected to gearbox, and then retighten the screws. Make sure that gearbox does not interfere with wing rear spar web. If it interferes, elongate the gearbox attaching hole 0.08 in. (2 mm).



1. Gear box
2. Jack nut block
3. Gimbal
4. Jack nut
5. Screw jack
6. Jack support
7. Bearing
8. Nut
9. Cap
10. Actuator box

Fig. 5-77 Installation of main actuator screw jack

(5) To hold jack screw in position, tighten nut in jack support, install cotter pin and place cap in position.

9.2.3 REMOVAL AND INSTALLATION OF INBOARD ACTUATOR JACK SCREW (See Fig. 5-78)

Removal should be done with flap in 20° down position as follows:

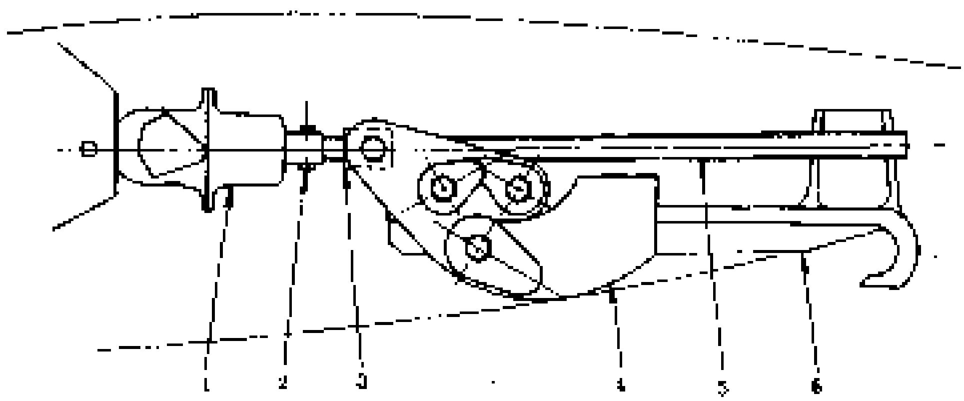
(1) Remove bolt and nut attaching jack screw. Disconnect jack screw from gear box.

(2) Rotate jack screw by hand and remove from jack nut.

Installation:

(1) Screw jack screw into jack nut.

(2) Place jack screw into gear box. Hold it in position with bolt and nut. Install cotter pin.



1. Gear box
2. Screw jack attaching bolt
3. Jack nut
4. Plate
5. Screw jack
6. Guide rail

Fig. 5-78 Installation of inboard actuator screw jack

9.2.4 REMOVAL AND INSTALLATION OF OUTBOARD ACTUATOR JACK SCREW (See Fig. 5-79)

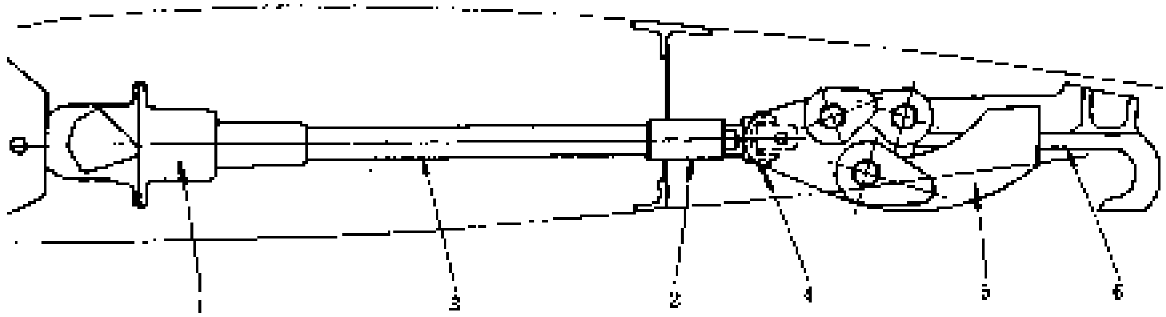
Removal:

(1) Place flap in 20° down position. Remove bolt and nut fastening tip of jack screw and gimbal.

Installation:

(1) Extend jack screw from wing by screwing to allow installation of flap.

(2) Connect tip of jack screw from gimbal. Hold it in position with bolt and nut. Install cotter pin.



1. Gear box
2. Jack nut
3. Screw jack
4. Gimbal attaching screw
5. Plate
6. Guide rail

Fig. 5-79 Installation of outboard actuator screw jack

9.2.5 REMOVAL AND INSTALLATION OF OUTBOARD FLAP GUIDE LINK MECHANISM (See Fig. 5-80)

(1) Remove bolt from outboard flap, which is held in 20° DOWN position.

(2) Install in reverse sequence of removal.

9.2.6 REMOVAL AND INSTALLATION OF TRIM AILERON (See Fig. 5-81)

With flap held in 20° down position.

(1) Remove access panel and disconnect electrical connector.

(2) Remove guide. Take electrical wires and connector out.

(3) Install in reverse sequence of removal.

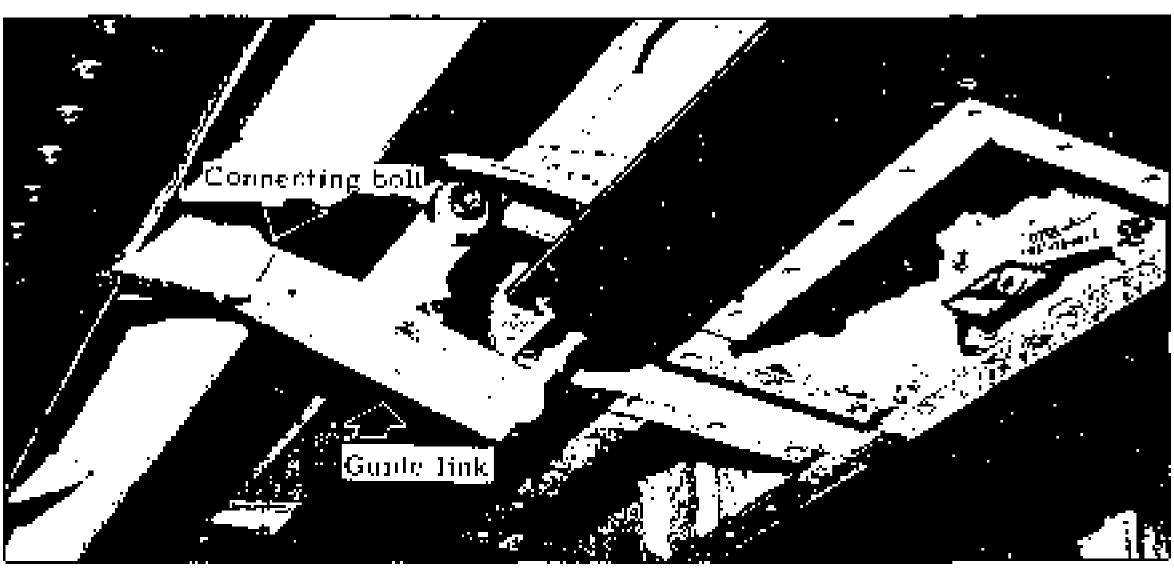


Fig. 5-80 Outboard flap guide link mechanism

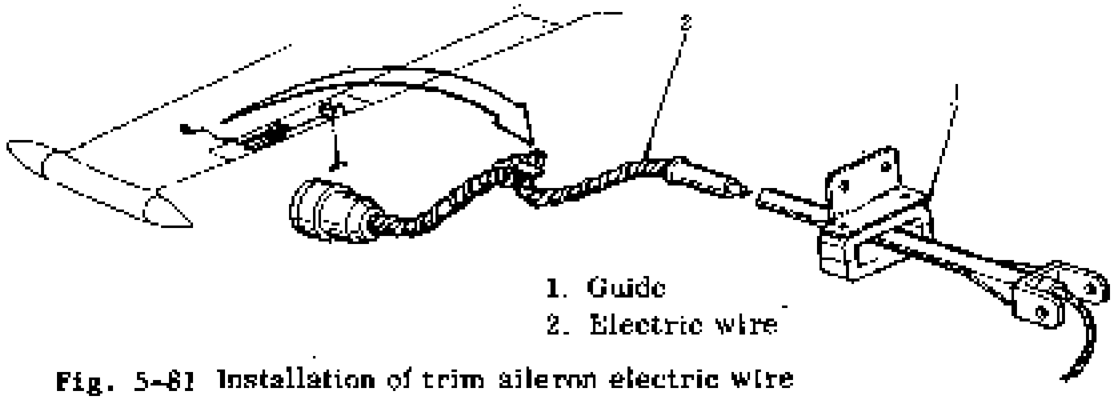


Fig. 5-81 Installation of trim aileron electric wire

9.2.7 REMOVAL AND INSTALLATION OF INBOARD AND OUTBOARD FLAPS

Removal should be done after jack screw link mechanism and electrical wiring are removed, as follows:

(1) Place flap in 20° down position.

(2) Remove bolt fastening inboard and outboard flaps in central main actuator box assembly. (See Fig. 5-82)

(3) Remove roller assembly at inner end of flap from guide rail with inboard flap held. Remove inboard flap from aircraft.

(4) Remove roller assembly at outer end of flap from guide rail with outboard flap held. Remove outboard flap from aircraft.

Installation should be done as follows.

(1) Set flap actuator box as configured for 20° down on guide rail of central main actuator. (See Fig. 5-83)

(2) Set roller assembly at inner end of inboard flap on guide rail of inboard actuator at W.STA 664.

(3) Set roller assembly at outer end of outboard flap on guide rail of outboard actuator at W.STA 4755.

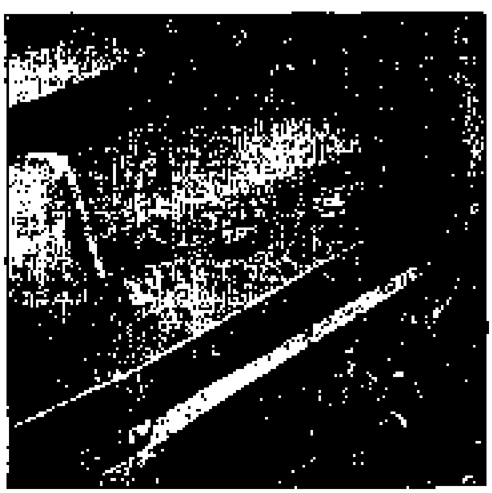


Fig. 5-82 Remove flap connecting bolt from actuator box

Fig. 5-83 Actuator box on guide rail

(4) Install inboard and outboard flaps and actuator box as a unit into a space where flap is retracted.
Tentatively attach inboard and outboard flaps to actuator box with bolts. (See Fig. 5-84)

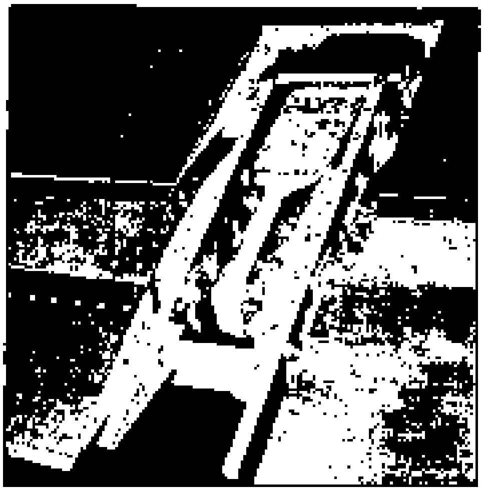


Fig. 5-84 Attach tentatively inboard and outboard flaps to actuator box

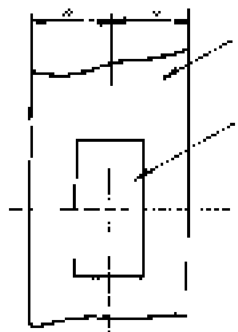


Fig. 5-85 Position of roller

Bolts need not be tightened to regular torque.

Use 25/32 in. wrench.

(5) Place assembled inboard and outboard flaps in 20° down position and adjust as follows:

(a) Inboard and outboard actuator rollers should contact the surface of guide rails, when roller of flap actuator box contacts the rotating surface of jack support

(b) Rollers of inboard and outboard actuators should be placed at the center of guide rail. (See Fig. 5-85) The above mentioned adjustments are possible by increasing or decreasing number of washers or by relocating eccentric bushing (See Fig. 5-86)

(6) With flap retracted, adjust gap and mismatch between flap and actuator box, and between flap and wing to the specified limit. (See Fig. 5-87)

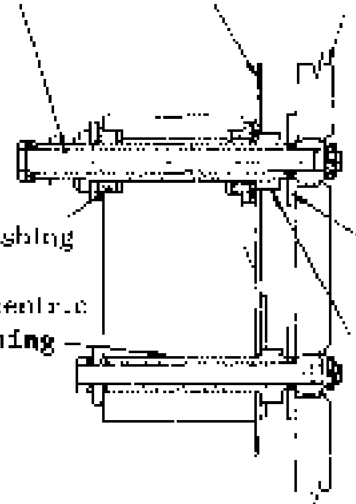


Fig. 5-86 Adjustment of roller position

MU-2 maintenance manual

TEMPORARY REVISION NO.5-1A

This Temporary Revision No. 5-1A is applicable to the following Maintenance Manuals ;

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	5-53
MU-2B-60	MR-0336	5-53

Replace existing Temporary Revision No.5-1 with this Temporary Revision No.5-1A and insert facing the page indicated above for the applicable Maintenance Manual. The revised points are indicated by the revision bar on the right margin.
Retain this Temporary Revision until such time as a permanent revision on this subject is issued.

REASON : To add the removal/installation of flap torque tube assembly.

CHANGE : Page 2/11 and Page 5/11 to revise the installation of cotter pins for torque tubes.

9.2A REMOVAL/INSTALLATION OF FLAP TORQUE TUBE ASSEMBLY (See (A) in Fig. 5-87A, and Fig 5-87B)

NOTE

Do not rotate the flap drive mechanism while the torque tube assembly is removed and until reinstallation is complete.

a. Remove the center wing rear upper access panel.
b. Apply a mating mark on the reduction gear box shaft and the torque tube assembly.
c. Remove the flap torque tube assembly in accordance with the following procedures.
 (1) Remove the cotter pins.
 (2) Disconnect the torque tube assembly from the reduction gear box shaft by sliding the tube assembly toward the stop assembly side.
 (3) Remove the torque tube assembly from the aircraft.
d. Connect the torque tube assembly by aligning the mating marks on the reduction gear box shaft, and install the torque tube assembly in the reverse order of removal procedures.

9.2B REPLACEMENT OF FLAP TORQUE TUBE ASSEMBLY (See A in Fig. 5-87A, and Fig. 5-87B)

NOTE

Do not rotate the flap drive mechanism while the torque tube assembly is removed and until reinstallation is complete

a. Remove the torque tube assembly in accordance with Para. 9.2A a. and c.
b. Position the new torque tube assembly so as to align the inspection hole of the shaft spline with the existing cotter pin hole of the reduction gear box shaft, and install it in the reverse order of removal procedures. (See Fig. 5-87B)
c. Slide the torque tube assembly toward the reduction gear box until the torque tube assembly bottoms out.
d. Drill one cotter pin hole through the torque tube assembly shaft spline and the reduction gear box shaft, using a #36 (.107 in. [2.71 mm]) drill bit, at about a right angle position (90°) to the existing cotter pin hole. The new cotter pin hole should be .276 in. (7 mm) from the edge of shaft spline. (See Fig. 5-87B, Section A-A)
e. Apply a mating mark on the reduction gear box shaft and the torque tube assembly. (See Fig. 5-87B)
f. Remove the torque tube assembly and clean all debris and then apply grease (MIL-G-23827) to spline.
g. Align the mating marks and install the torque tube assembly in the reverse order of removal procedures (Para. 9.2A c. (3), (2))
h. Install new cotter pins (P/N MS24665-285) in the torque tube assembly.
i. With the cotter pins installed, slide the torque tube in an inboard and outboard direction by hand. If the cotter pins are properly installed, there should be no movement. If movement does exist, reinstall and/or replace parts as required.
j. Set flaps in the retracted position and ensure that the drilled alignment holes on the flap upper and lower skin are in line with specified marks on wing respectively.
k. Perform the flap operational checks.

9.2C REPLACEMENT OF SHAFT SPLINES IN CENTER FLAP TORQUE TUBE ASSEMBLY (See A in Fig. 5-87A, and Fig. 5-87B)

NOTE

1. The following shaft spline replacement procedure is applicable to A torque tube assembly.
2. Do not rotate the flap drive mechanism while the torque tube assembly is removed and until reinstallation is complete.

a. Remove the torque tube assembly in accordance with Para. 9.2A a. and c.

b. Replacement of the shaft spline connected to the stop assembly shaft.

(1) Remove the torque tube assembly bolts, washers, nuts and cotter pins.

(2) Remove the existing shaft spline from the stop assembly side of the torque tube assembly.

(3) Insert the new shaft spline into the stop assembly side of the torque tube assembly.

(4) Connect torque tube assembly by aligning the cotter pin holes of the reduction gear box shaft and the torque tube assembly, and install the torque tube assembly temporarily in the reverse order of removal procedures. (Para. 9.2A c. (3), (2))

(5) Install the cotter pin in the torque tube assembly temporarily.

(6) Set the shaft spline protrusion to the specified dimension (L2) and retain the shaft spline insertion position using masking tape or other suitable materials. (See Fig. 5-87D)

(7) Remove the torque tube assembly in accordance with Para. 9.2A c. (1), (2) and (3).

(8) Drill two #3 (.213 in. [5.41 mm]) drill bit holes in the shaft spline to match the existing tube assembly holes and ream $.2213^{+.0015}_{-0}$ in.($5.62^{+0.038}_{-0}$ mm) and clean all debris.

(9) Install the shaft spline with new bolts (NAS3003-12D), washers (AN960-10L) and nuts (AN320-3), and secure the nuts with new cotter pins (MS24665-132).

c. Replacement of the shaft spline connected to the reduction gear box shaft.

(1) Remove the torque tube assembly bolts, washers, nuts and cotter pins.

(2) Remove the existing shaft spline from the reduction gear box side of the torque tube assembly.

(3) Insert the new shaft spline into the reduction gear box side of the torque tube assembly.

(4) Connect the torque tube assembly by aligning the inspection hole of the shaft spline with the existing cotter pin hole of the reduction gear box shaft, and install the torque tube assembly temporarily in the reverse order of removal procedures. (Para. 9.2A c. (3), (2))

(5) Install the cotter pin in the torque tube assembly temporarily.

(6) Set the shaft spline protrusion to the specified dimension (L1) and retain the shaft spline insertion position using masking tape or other suitable materials. (See Fig. 5-87D)

(7) Remove the torque tube assembly in accordance with Para. 9.2A c. (1), (2) and (3).

(8) Drill two #1 (.272 in. [6.91 mm]) drill bit holes in the shaft spline to match the existing tube assembly holes and ream $.2813^{+.0015}_{-0}$ in.($7.145^{+0.038}_{-0}$ mm) and clean all debris.

(9) Install the shaft spline with new bolts (NAS3004-14D), washers (AN960-416) and nuts (AN320-4), and secure the nuts with new cotter pins (MS24665-134).

d. After completion of shaft spline replacement, install the torque tube assembly in accordance with Para. 9.2B, b through j.

TEMPORARY REVISION NO.5-1A

9.2D REMOVAL/INSTALLATION OF OUTBOARD FLAP TORQUE TUBE ASSEMBLY (See B, C and D in Fig. 5-87A, and Fig. 5-87C)

NOTE

Do not rotate the flap drive mechanism while the torque tube assembly is removed and until reinstallation is complete.

a. Open the LH (RH) inboard wing trailing edge lower access panel(s).
b. Apply a mating mark on the shaft of cotter pin side and the torque tube assembly.
c. Remove the flap torque tube assembly in accordance with the following procedures.
   (1) Remove the cotter pin.
   (2) Remove the retainer ring.
   (3) Slide the torque tube assembly toward the shaft (retainer ring side) and disconnect the torque tube assembly from the shaft (cotter pin side).
   (4) Remove the torque tube assembly from the aircraft.
d. Install the torque tube assembly by aligning the mating marks on the shaft, and install the torque tube assembly in the reverse order of removal procedures.

9.2E REPLACEMENT OF OUTBOARD FLAP TORQUE TUBE ASSEMBLY (See B, C and D in Fig. 5-87A, and Fig. 5-87C)

NOTE

1. The following procedure is applicable to all B, C and D torque tube assembly replacements.
2. Do not rotate the flap drive mechanism while the torque tube assembly is removed and until reinstallation is complete.

a. Remove the torque tube assembly in accordance with Para. 9.2D a. and c.
b. Position the new torque tube assembly so as to align the inspection hole of the shaft spline with the existing cotter pin hole of the shaft (cotter pin side), and install it in the reverse order of removal procedures described in Para. 9.2D c. (3) and (4). (See Fig. 5-87C)
c. Insert the torque tube assembly to the position where the end of the shaft (cotter pin side) appears in the inspection hole. Then slide the torque tube assembly .20 in. (5 mm) farther into the torque tube.
d. Ensure that the clearance between end of the torque tube assembly and the retainer ring groove is .080 to .20 in. (2 to 5 mm) (See Fig. 5-87C)

e. Drill one cotter pin hole through the shaft spline and the shaft using a #36 (.107 in. [2.71 mm]) drill bit at about right angle position (90°) to the existing cotter pin hole. The new cotter pin hole should be .276 in (7 mm) from edge of shaft spline. (See Fig. 5-87C, Section A-A)

f. Apply a mating mark on the shaft of cotter pin side and the torque tube assembly. (See Fig. 5-87C)

g. Remove the torque tube assembly and clean all debris and then apply grease (MIL-G-23827) to spline.

h. Align the mating marks and install the torque tube assembly in the reverse order of removal procedures.

i. Install new cotter pin in the torque tube assembly.

j. With the cotter pin installed, slide the torque tube in an inboard and outboard direction by hand. If the cotter pin is properly installed, there should be no movement. If movement does exist, reinstall and/or replace parts as required.

k. Install the retainer ring in the groove and make sure that the clearance between end of the torque tube assembly and the retainer ring is .080 to .20 in. (2 to 5 mm).

l. Set flaps in the retracted position and ensure that the drilled alignment holes on the flap upper and lower skin are in line with specified marks on wing respectively.

m. Perform the flap operational checks.

9.2F REPLACEMENT OF JOINTS IN OUTBOARD FLAP TORQUE TUBE ASSEMBLY (See B, C and D in Fig. 5-87A, and Fig. 5-87C)

NOTE

1. The following shaft spline replacement procedure is applicable to all B, C and D torque tube assemblies.
2. Do not rotate the flap drive mechanism while the torque tube assembly is removed and until reinstallation is complete.

a. Remove the torque tube assembly in accordance with Para. 9.2D a. and c.

b. Replacement of the shaft spline connected to the shaft at retainer ring side.

(1) Drill out two rivets and remove the existing shaft spline.

(2) Insert the new shaft spline into the torque tube assembly.

(3) Connect torque tube assembly by aligning the cotter pin holes of the shaft (cotter pin side) and the torque tube assembly, and install the torque tube assembly temporarily in the reverse order of removal procedures.

(4) Temporarily install the cotter pin in the torque tube assembly.

(5) Set the shaft spline protrusion to the specified dimension (L1) and retain the shaft spline insertion position using masking tape or other suitable materials. (See Fig. 5-87D)

(6) Remove the torque tube assembly in accordance with Para. 9.2D c. (3) and (4).

(7) While holding the torque tube with the shaft spline inserted, drill two rivets holes in the shaft spline to match the existing rivets holes on the torque tube assembly using a #21 (.159 in. [4.04 mm]) drill bit.

(8) Install the shaft spline with rivets (NASM20615-5M18 for 010A-61260-( ) and 017A-61805-( ) torque tube assemblies or NASM20615-5M20 for 010A-61250-( ) and 010A-61251-( ) torque tube assemblies).

c. Replacement of the shaft spline connected to the shaft at cotter pin side.

(1) Drill out two rivets and remove the existing shaft spline.

(2) Insert the new shaft spline into the torque tube assembly

(3) Connect the torque tube assembly by aligning the inspection hole of the shaft spline with the existing cotter pin hole of the shaft (cotter pin side), and install the torque tube assembly temporarily in the reverse order of removal procedures.

(4) Set the shaft spline protrusion to the specified dimension (L2) and retain the shaft spline insertion position using masking tape or other suitable materials, and apply a mating mark on the shaft (cotter pin side) and the torque tube assembly as shown in Fig. 5-87C. (See Fig. 5-87D)

(5) Remove the torque tube assembly in accordance with Para. 9.2D c. (3) and (4).

(6) While holding the torque tube with the shaft spline inserted, drill two rivet holes in the shaft spline to match the existing rivets holes on the torque tube assembly using a #21 (.159 in. [4.04 mm]) drill bit.

(7) Install the shaft spline with rivets (NASM20615-5M18 for 010A-61260-( ) and 017A-61805-( ) torque tube assemblies or NASM20615-5M20 for 010A-61250-( ) and 010A-61251-( ) torque tube assemblies).

d. After completion of shaft spline replacement, install the torque tube assembly in accordance with Para. 9.2E, b through i.

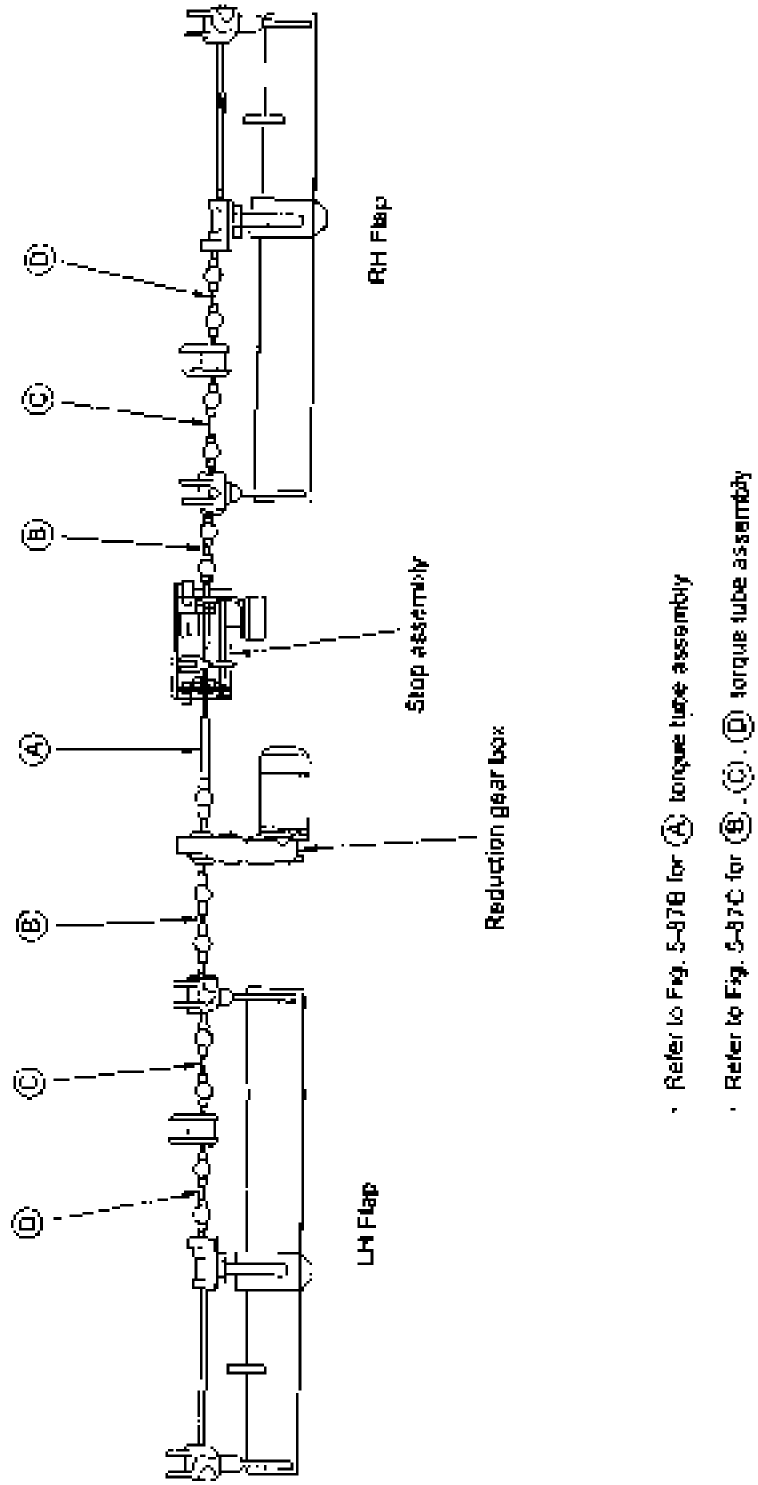
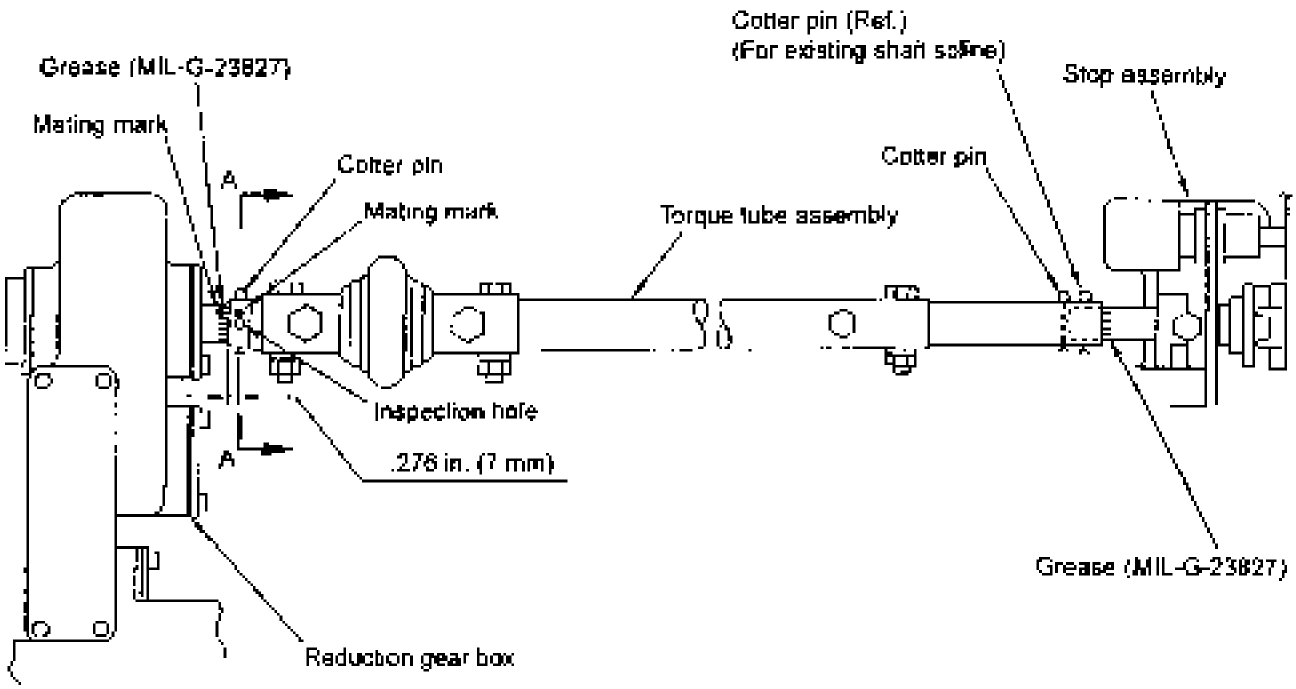
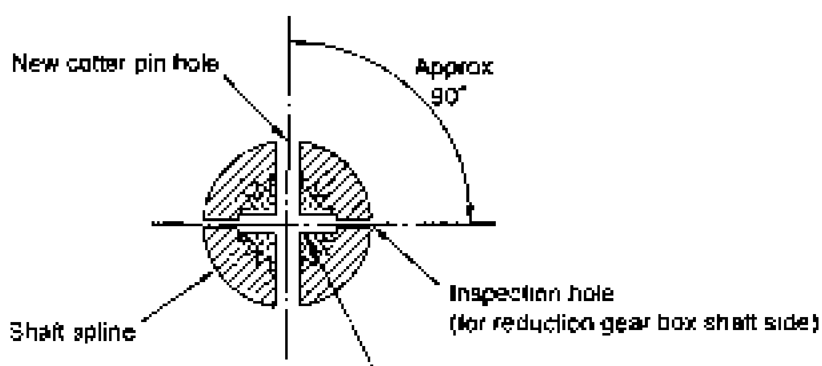


Fig. 5-87A Assembly of flap torque tube assembly

TEMPORARY REVISION NO.5-1A





Section A - A

Fig. 5-87B Installation/Removal of center flap torque tube assembly

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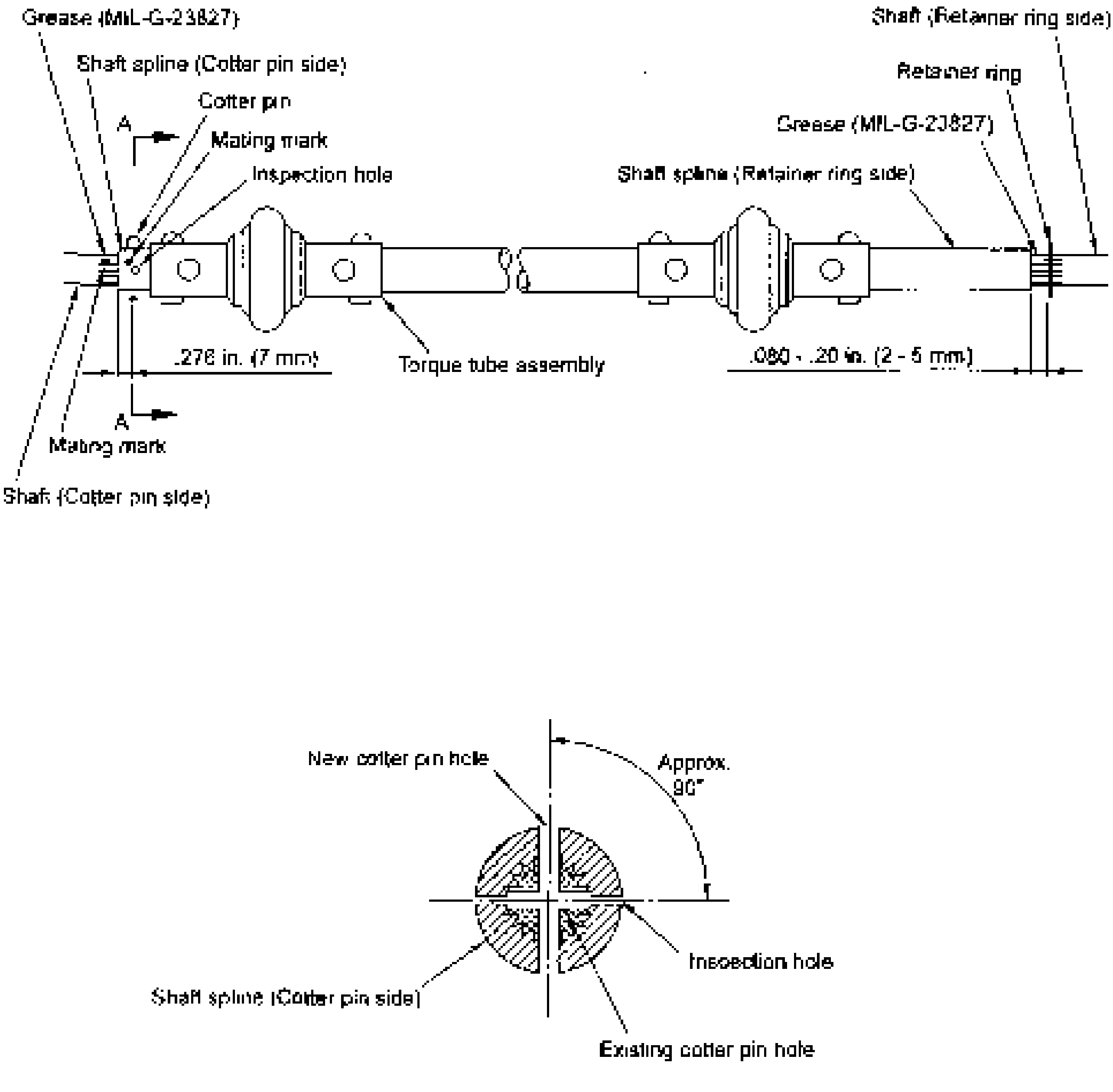
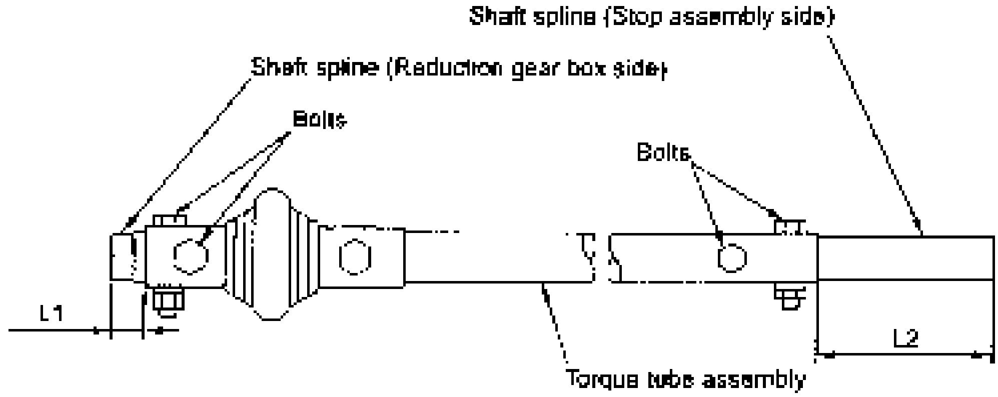


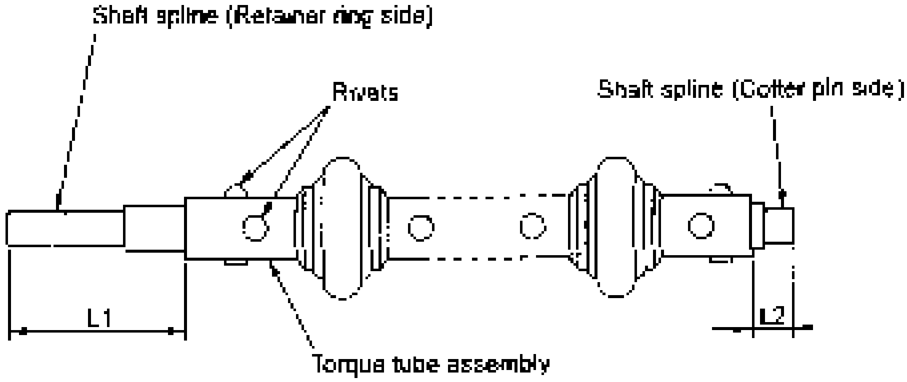
Section A - A

Fig. 5-87C Installation/Removal of outboard flap torque tube assembly



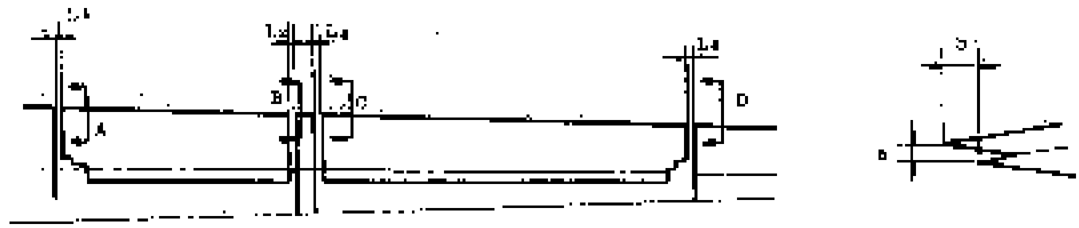
Torque tube assembly (Refer to Fig. 5-87A) \ Shaft spline length	L1	L2
Ⓐ	.45 in. (11.5 mm)	2.56 in. (65.0 mm)

Fig. 5-87D Required value of shaft spline insertion (1/2)



Torque tube assembly (Refer to Fig. 5-87A) \ Shaft spline length	L1	L2
Ⓑ	.81 in. (20.5 mm)	.45 in. (11.5 mm)
Ⓒ, Ⓓ	1.81 in. (46.0 mm)	.63 in. (16.0 mm)

Fig. 5-67D Required value of shaft spline insertion (2/2)



Gap limitation

$L_1 = 0.59 \pm 0.08$ in. (15 ± 2 mm)
$L_2 = 0.20 \pm 0.08$ in. (5 ± 2 mm)
$L_3 = 0.20 \pm 0.08$ in. (5 ± 2 mm)
$L_4 = 0.59 \pm 0.08$ in. (15 ± 2 mm)

in. (mm) Max.

Mismatch \ Section	A	B	C	D
a	0.16 (4)	0.20 (5)	0.20 (5)	0.16 (4)
b	0.32 (8)	0.20 (5)	0.20 (5)	0.24 (6)

Fig. 5-87 Gap and mismatch of flaps and wing

9.3 SYSTEM RIGGING AND CHECK

Rig flap system in accordance with the following sequence.
Detailed procedures are described in subsequent paragraphs.

(1) Rigging of flap actuator box nut support (Para. 9.3.2).

(2) Adjustment of extended flap position at 0° of deflection (Para. 9.3.3).

(3) Adjustment of flap central main actuator roller (Para. 9.3.4).

(4) Adjustment of inboard actuator roller (Para. 9.3.5).

(5) Adjustment of outboard actuator roller (Para. 9.3.6).

(6) Adjustment of retracted flap position (Para. 9.3.7).

(7) Adjustment of outboard flap guide link (Para. 9.3.8).

9.3.1 EQUIPMENT REQUIRED FOR ADJUSTMENT

(1) Rigging fixture

GSE 010A-99054-1, -2 1 ea
GSE 010A-99055-1, -2 1 ea
GSE 010A-99056-1, -2 1 ea
GSE 010A-99057-1, -2 1 ea

(2) Protractor

GSE 010A-99036 2 ea

9.3.2 ADJUSTMENT OF NUT SUPPORT OF FLAP ACTUATOR BOX (See Fig. 5-88)

(1) Attach protractor to upper surface of flap at W.STA 1630 and W.STA 4450.
Set pointer to 0 with flap in retracted position.

(2) Adjust gaps as follows with flap in extended to 20° down positions.

NOTE: Please see the **TEMPORARY REVISION** that revises this page.

(a) Gap when nut support set to one side: 0 in.

(b) Gap between guide rail and side shoe: 0.008 ~ 0.016 in.

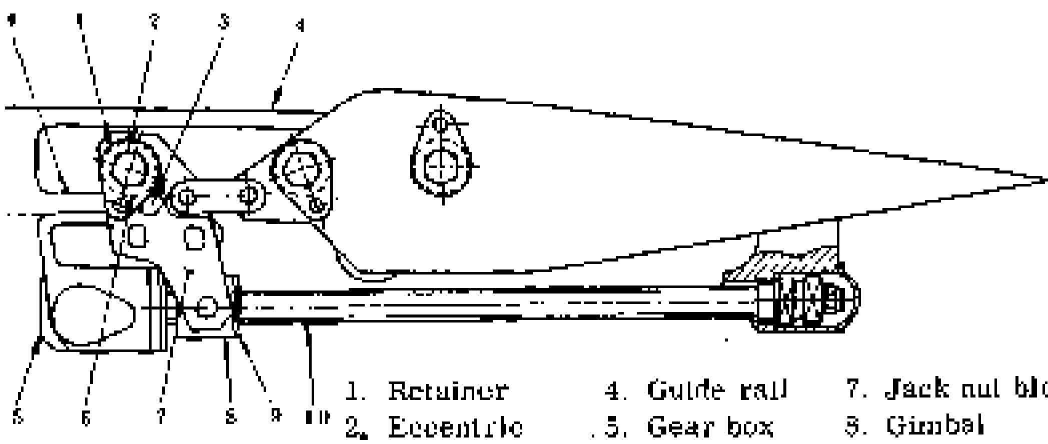
(c) Measure gap of 0.008 +0.002/-0.004 in. (0.2 +0.05/-0.1 mm) between the guide rail and roller with the flap at 20° and 40° down position. The gap may be adjusted by removing the retainer and adjusting the eccentric bushing.

(3) After adjustment is completed, apply alignment mark to eccentric bushing. Install safety wire on retainer attaching bolt.

NOTE

(i) When measuring flap deflection, lift flap by hand to eliminate play in flap system.

(ii) Flap may be operated by hand by disconnecting torque shaft between flap motor and stop assembly, or disconnecting torque shaft between stop assembly and inboard actuator. Flap may be operated by motor after all adjustments are completed.



1. Retainer
2. Eccentric bushing
3. Side shoes
4. Guide rail
5. Gear box
6. Roller
7. Jack nut block
8. Gimbal
9. Jack nut
10. Jack screw

Fig. 5-88 Adjustment of jack nut block

9.3.3 ADJUSTMENT OF EXTENDED FLAP POSITION

(1) Disconnect torque shafts connecting stop assembly with actuator on R.H. and flap main reduction gear with inboard actuator on L.H. Place flap in fully extended position by rotating torque shaft by hand.

(2) Check rollers of central main actuator and inboard and outboard actuators for contact with guide rail.

(3) Disconnect torque tube from inboard actuator gear box at W.STA 664. Disconnect flexible shaft from central main actuator gear box. Rotate actuators individually by hand so that rollers contact with guide rail. (At this time, adjust the number of washers in Fig 5-86 so as not to cause twisting between inboard and outboard flaps. Reassemble torque tubes and flexible shaft.) (See Figs 5-89 and 5-90)

NOTE

Tighten flexible shaft finger tight. Install safety wire. If flexible shaft is tightened excessively, rotation of flexible shaft may not be conveyed smoothly to outboard actuator. Position water seal over shaft nut and secure with tie strap (MS3367-5-X).

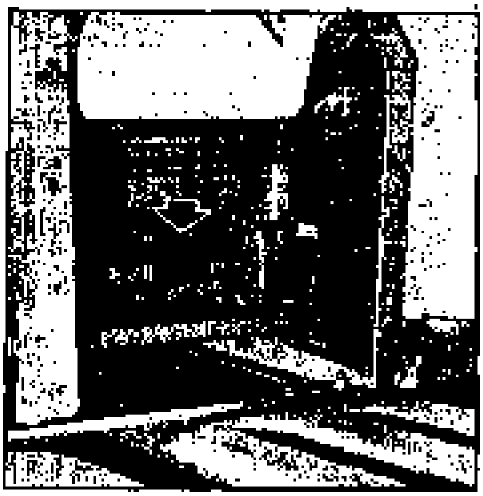


Fig. 5-89 Disconnect torque tube

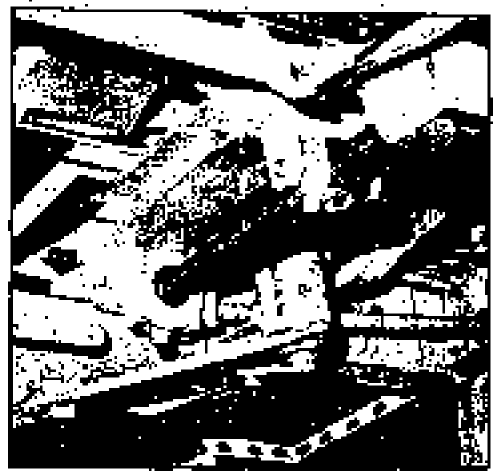


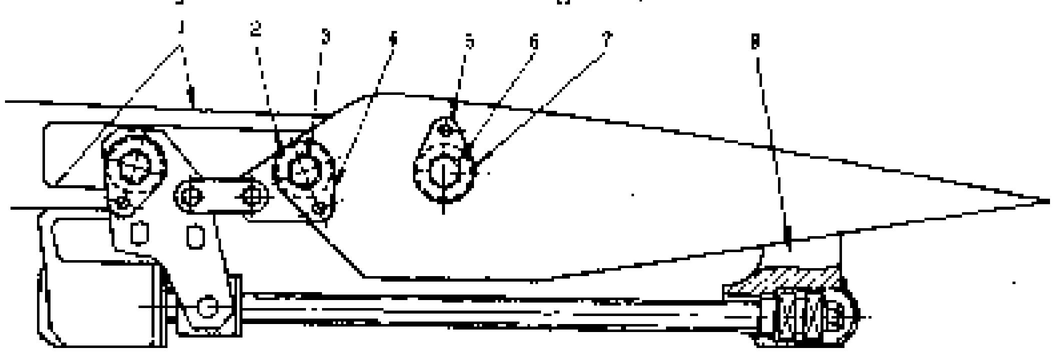
Fig. 5-90 Remove flexible shaft

9.3.4 ADJUSTMENT OF CENTRAL MAIN ACTUATOR ROLLER (See Fig. 5-91)

(1) Place flap in 20° and 40° down position by rotating torque shaft by hand at the inboard actuator gearbox. Adjust the aft roller and jack support to 0.008 +0.002/-0.004 in. (0.2 +0.05/-0.1 mm) by removing the retainer and adjusting the eccentric bushing.

(2) Extend the flap by hand to 2/3 position on the guide rail from 0° and adjust the gap between the forward roller and guide rail to 0.008 +0.002/-0.004 in. (0.2 +0.05/-0.1 mm) by removing the retainer and adjusting the eccentric bushing.

(3) After adjustment is completed, apply alignment mark to eccentric bushing. Install safety wire on retainer attaching bolt.



1. Guide rail
2. Front roller
3. Eccentric bush
4. Retainer
5. Retainer
6. Eccentric bushing
7. Rear roller
8. Jack support

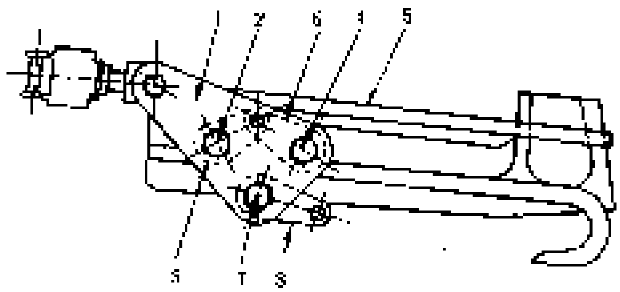
Fig. 5-91 Adjustment of main actuator roller

9.3.5 ADJUSTMENT OF INBOARD ACTUATOR ROLLER (See Fig. 5-92)

(1) Place flap in 20° and 40° down position by rotating the torque shaft at the inboard actuator gearbox. Adjust the gap between the aft roller and guide rail to 0.01 +0.004/-0.006 in. (0.3 +0.1/-0.15 mm) by adjusting the eccentric bushing.

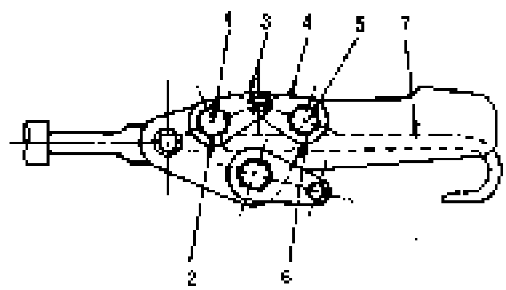
(2) Extend the flap by hand to 2/3 position on the guide rail from 0° and adjust the gap between the forward roller and guide rail to 0.01 +0.004/-0.006 in. (0.3 +0.1/-0.15 mm) by adjusting the eccentric bushing.

(3) After adjustment is completed, apply alignment mark to eccentric bushing.



1. Plate
2. Front roller & eccentric bushing
3. Retainer
4. Rear roller & eccentric bushing
5. Guide rail
6. Retainer
7. Roller & eccentric bushing
8. Retainer

Fig. 5-92 Adjustment of inboard actuator roller



1. Front roller
2. Eccentric bushing
3. Retainer
4. Retainer
5. Rear roller
6. Eccentric bushing
7. Guide rail

Fig. 5-93 Adjustment of outboard actuator roller

9.3.6 ADJUSTMENT OF OUTBOARD ACTUATOR ROLLER (See Fig. 5-93)

(1) Place flap in 20° and 40° down position by rotating torque shaft by hand at the outboard actuator gearbox. Adjust the gap between the aft roller and guide rail to 0.01 +0.004/-0.006 in. (0.3 +0.1/-0.15 mm) by adjusting the eccentric bushing.

(2) Extend the flap by hand to 2/3 position on the guide rail from 0° and adjust the gap between the forward roller and the guide rail to 0.01 +0.004/-0.006 in. (0.3 +0.1/-0.15 mm) by adjusting the eccentric bushing.

(3) After adjustment is completed, apply alignment mark to eccentric bushing.

9.3.7 ADJUSTMENT OF RETRACTED FLAP POSITION (See Fig. 5-94)

(1) Place flap in retracted position.

(2) Attach flap rigging fixture on upper surface of wing. (See Figs. 5-94 and 5-95)

Attaching position

W.STA 950	GSE 016A-99054-1, -2	1 ea.
W.STA1750	GSE 016A-99055-1, -2	1 ea.
W.STA3150	GSE 016A-99056-1, -2	1 ea.
W.STA4750	GSE 016A-99057-1, -2	1 ea.

(3) Disconnect flexible shaft at central main actuator gear box. Disconnect torque shaft at inboard actuator gear box.

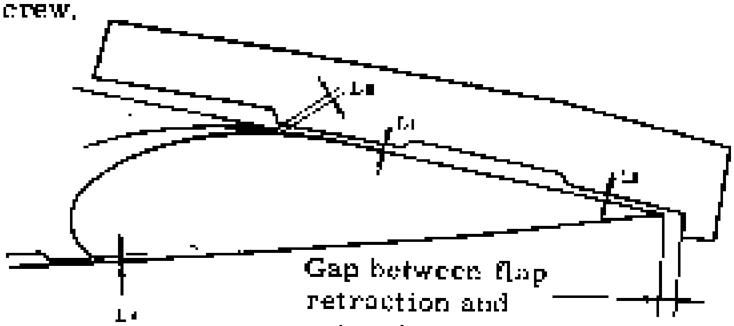
(4) Rotate inboard and outboard jack screws individually, and place them in retracted position. Trim flap trailing edge.

W. STA 950 0.04 in. (1 mm) Min. W. STA 1750 0.04 in. (1 mm) Min.

W. STA 3150 0.08 in. (2 mm) Min. W. STA 4750 0.12 in. (3 mm) Min.

Measure flap trailing edge against rigging fixture, and verify the gap between flap extension and retraction as specified above. The gap is adjusted by rotating jack screw.

ORIGINAL
As Received By
ATP



$L_1 = 0.2 \pm 0.12$ in. (5 ± 3 mm) [Difference between L. H. and R. H. 0.12 in. (3 mm) Max.]
$L_2 = 0.2 \pm 0.16$ in. (5 ± 4 mm) [" " "]
$L_3 = 0.06\,^{+0.08}_{-0.04}$ in. ($1.5\,^{+2}_{-1}$ mm) [" " "]
L_4 = Uniformly contact with flap surface.

Fig. 5-94

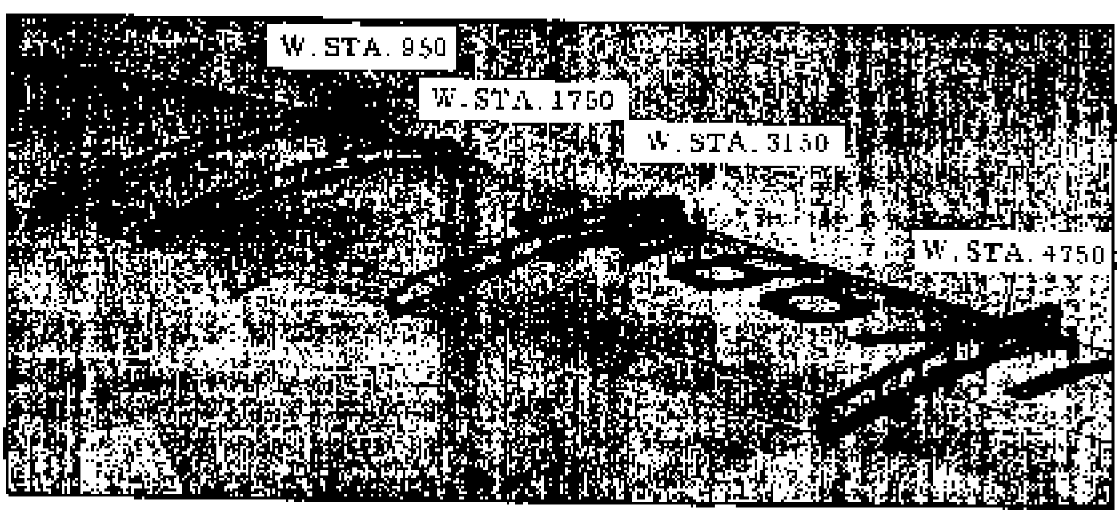


Fig. 5-95 Installation of flap rigging fixture

(5) Adjust gap between flap and rigging fixture L_1 and L_2.

(6) Adjust gap between flap and upper surface of wing trailing edge.

(7) Adjust gap between flap and seal retainer on lower surface of wing trailing edge.

TEMPORARY REVISION NO.5-2

This Temporary Revision No. 5-2 is applicable to the following Maintenance Manuals .

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-35A	MR-0218	5-58
MU-2B-60	MR-0336	5-58

Insert facing the page indicated for the applicable Maintenance Manual above. Retain this Temporary Revision until such time as page is revised per the change below.

REASON : To change and add the removal/installation of flap torque tube assembly.

CHANGE : Revision of NOTE next to Paragraph 9.3.7.(15) and addition of Fig.5-96A and 5-96B as follows.

(15) Connect torque shaft which leads to flap stop assembly to inboard actuator gear box. (REF)

NOTE

i When replacing or reinstalling the flap flexible shaft, install the short end ferrule to main flap gear box side (See Fig. 5-96A) Aircraft that have the flap flexible shaft installed with the longer end ferrules attached to the main flap gear box side are acceptable provided that 8.7 inches minimum radius is maintained.

ii Tighten flexible shaft finger tight. Overtightening of flexible shaft may hinder smooth rotation of the shaft. Make sure that the bending radius of flexible shaft is more than 8.7 in (221 mm).

iii When the flexible shaft is installed and safety wired, apply sealant around the connections as follows. (See Fig. 5-97B)

iv When connecting torque shaft, do not change alignment of flap with wing. Install retainer ring and cotter pin securely. Ensure engagement of over 0.28 in. (7.1 mm) in depth. (See Fig. 5-95)

v The flap cable should be degaussed prior to installation.

TEMPORARY REVISION NO.5-2

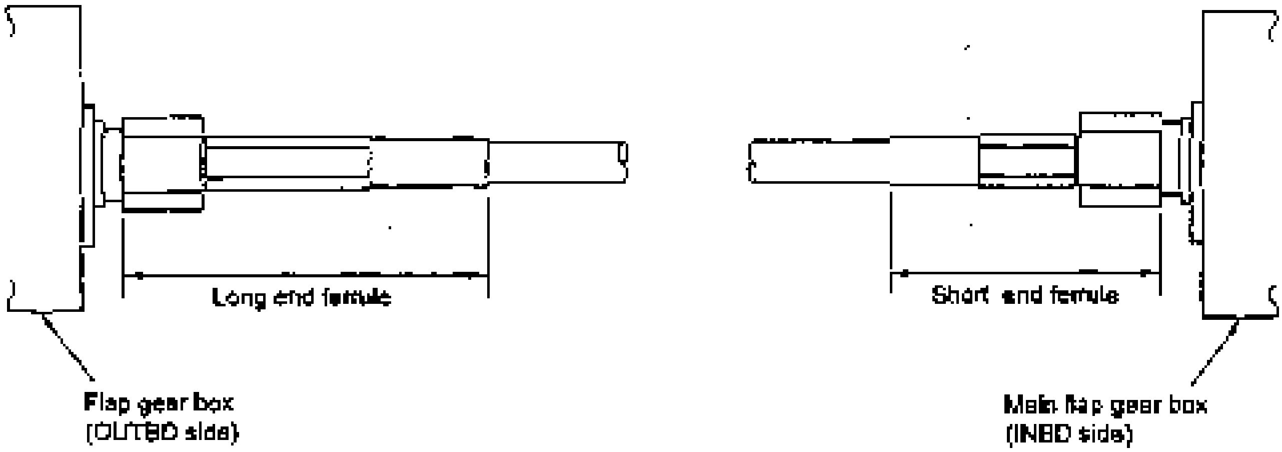


Fig. 5-96A

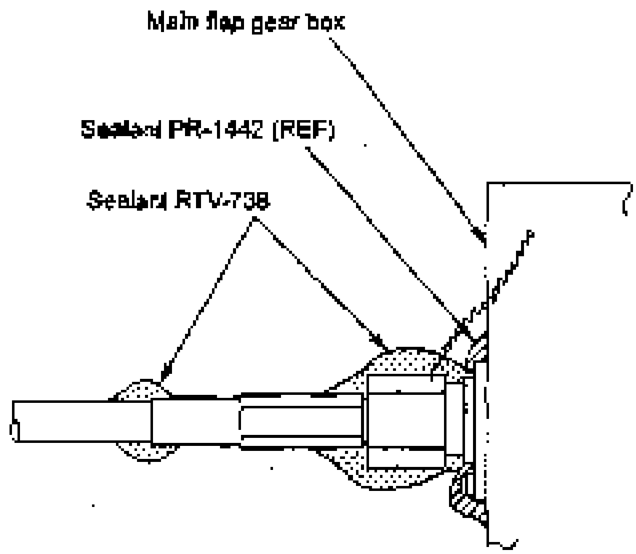


Fig. 5-96B

(8) Adjust mismatches between flap and central main actuator box assembly, and between outer end of flap and wing in accordance with Fig. 5-87.

(9) Adjustments described in (5), (6), (7) and (8) above are possible by means of eccentric bushing on bolt incorporated in central main actuator box assembly for attaching inboard and outboard flaps.

(10) Tentatively reassemble torque shaft and flexible shaft.

(11) Retract inboard and outboard flaps as a unit until flaps contact with structure by rotating torque shaft. Rotate torque shaft in opposite direction by one and a half turn. Ensure flap trailing edge being trimmed as specified in paragraph (5) above. Ensure gaps conforming to limits specified in paragraphs (6), (7) and (8) above. If those limits are not complied with, readjust positions of jack screw and eccentric bushings.

(12) After position for retraction of flap is adjusted, tighten inboard and outboard flaps that were tentatively assembled. Install flap attaching bolt and retainer of eccentric bushing. Install safety wire, cotter pin and apply alignment mark. The tightening torque of flap attaching bolt is 52 ~ 61 in-lbs(60 ~ 70 kg-cm), and that of the hollow bolt is 1300 ~ 1390 in-lbs(1498 ~ 1601 kg-cm).

(13) Ensure alignment of flaps with wing at inner end of inboard flap, outer end of outboard flap and central actuator box.

(14) Remove rigging fixture from upper surface of wing. Replace screw in the attaching hole for rigging fixture.

(15) Connect torque shaft which leads to flap stop assembly, to inboard actuator gear box.

NOTE

i Tighten flexible shaft finger tight. Overtightening of flexible shaft may hinder smooth rotation of the shaft. Make sure that the bending radius of flexible shaft is more than 8.7 in. (221 mm).

ii Position water seal over shaft nut and secure with tie strap (MS 3367-5-X).

iii When connecting torque shaft, do not change alignment of flap with wing. Install retainer ring cotter pin securely. Ensure engagement of over 0.28 (7.1 mm) in depth. (See Fig. 5-96)

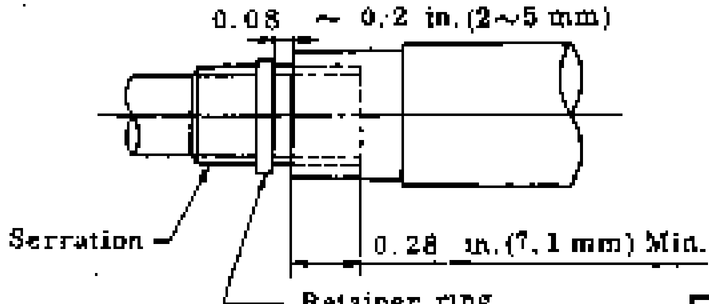


Fig. 5-96 Detail of serration fit

NOTE: Please see the **TEMPORARY REVISION** that revises this page.

9.3.8 RIGGING OF OUTBOARD FLAP GUIDE LINK (See Fig. 5-97)

(1) Attach protractor on upper surface of flap at W.STA 1630 and W.STA 4450.

(2) Tentatively fasten outboard flap to flap guide link with bolt.

(3) Place flap in retracted position.

(4) With flap in retracted position, adjust mismatch and gap as follows by means of adjusting rod and eccentric bolt.

(a) Mismatch between upper link and upper surface of wing = 0.12 in. (3 mm) Max. (D)

(b) Mismatch between lower link and lower surface of wing = 0.15 in. (4 mm) Max. (A) = 0.1 in. (2.5 mm) Max. (B)

(c) Gap between roller and rail should be 0.010 + 0.004/-0.008 in. (0.25 + 0.11/-0.19 mm). (C)

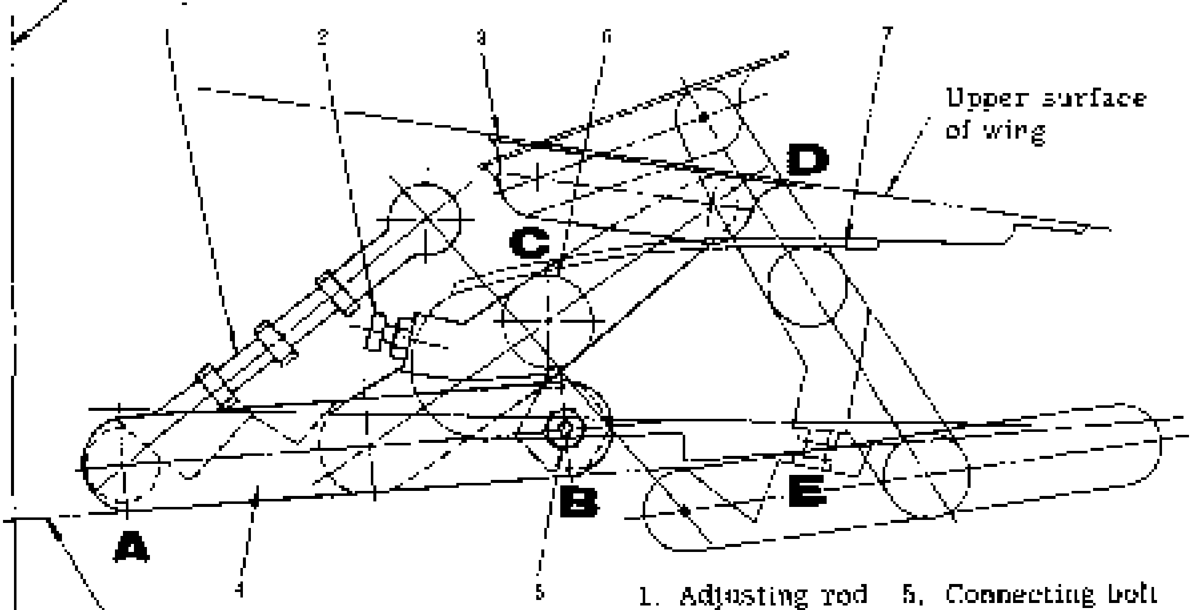


Fig. 5-97 Rigging of outboard flap guide link

(5) Place flap in 40° down position. Adjust gap between roller and rail to less than 0.04 in. (1 mm) by means of Adjusting rod and eccentric bushing with the stop bolt fully in. After the mismatch and gap are adjusted, install lock nut on bolt and apply alignment mark. Gap between stop bolt and stop : 0.02 to 0.06 in.(0.05 to 1.5 mm). (E)

(6) After adjustment of outboard flap guide link is accomplished, tighten outer flap attaching bolt. Install cotter pin.

9.3A SIMPLIFIED FLAP RIGGING PROCEDURES

NOTE

If the rigging fixture are not available, these procedures are acceptable when the flap assembly is original and has never been replaced. But, all other adjustment and rigging requirements specified in Para.9.3 have been complied with.

(1) Retract the flaps.

(2) Ensure that the holes on the underside of the flap are in line with the punch mark on the guide rail at both ends. Lay a straight edge across the holes to see if they line up.

(3) Ensure that holes drilled on the top of flap on each side of the engine line up with the trailing edge of the wing.

(4) Mount a protractor on each flap.

(5) Advance the flap to the first position of 5 (or 20) degrees with air load (pull up) applied.

(6) Check the deflection.

(7) Continue to check at other positions.

CAUTION

ONLY CHECK THE DEFLECTION GOING DOWN. DO NOT CHECK DEFLECTION GOING UP.

(8) Ensure that all measurements meet requirements in Para. 9.3 of this maintenance manual.

(9) Readjust flap as required.

(10) Remove the protractor.

9.4 ADJUSTMENT OF FLAP ACTUATING SWITCH

9.4.1 PREPARATION

Prior to adjustment, check flap for the following.

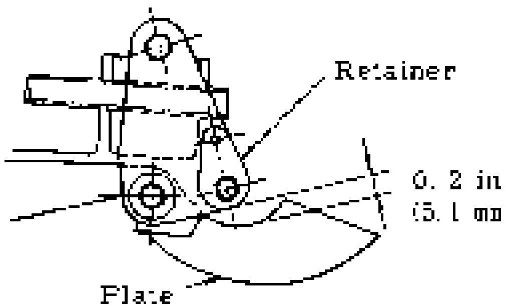
(1) L.H. and R.H. flaps for smooth movement by hand from retracted position to 40° down position.

(2) Rollers at inner end, outer end and middle of flap for contact with guide rail during the movement of flap from extended position to down position. (See Para.9.3.3 of this manual)

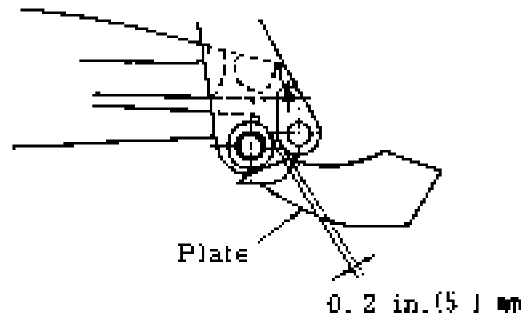
(3) Extra deflection of flap from 40° to 42° without interference with other mechanism.

(4) Gap of over 0.2 in.(5.1 mm) between retainer and plate connecting inboard and outboard actuators with flap. (See Fig.5-98)

(5) Attach protractor to flap at W.STA 1630 and W.STA 4450.



Gap between retainer and inboard flap connecting plate



Gap between retainer and outboard flap connecting plate

Fig. 5-98

(6) Adjustment is made on R.H. flap in accordance with following procedures:
It is recommended that flaps be operated by hand.
If this is not possible, connect R.H. torque shaft to flap motor and reduction gear. Leave L.H. shaft free. Operate flap intermittently to prevent inertial operation.
When adjustment of R.H. flap is completed, disconnect R.H. torque shaft from inboard actuator gear box. Connect L.H. torque shaft to flap.
Adjust L.H. flap.

(7) When adjustment of L.H. and R.H. flaps is completed, connect both torque shafts, and ensure normal operation.

9.4.2 ADJUSTMENT OF S419, S420, S421, S422, S431, S433 and S434 (See Fig.5-99)

The above limit switches are installed in flap stop assembly and are operated as follows.

9.4.2.1 OPERATION OF SWITCH

S419 20° down limit switch

This limit switch stops flap at 20° ± 1° of flap angle, when flap moves from retracted (UP) or 5° down position to 20° down position.

S420 22° down limit switch
This limit switch stops flap at 22° ±1° of flap angle, when flap moves from 40° down position to 20° down position.

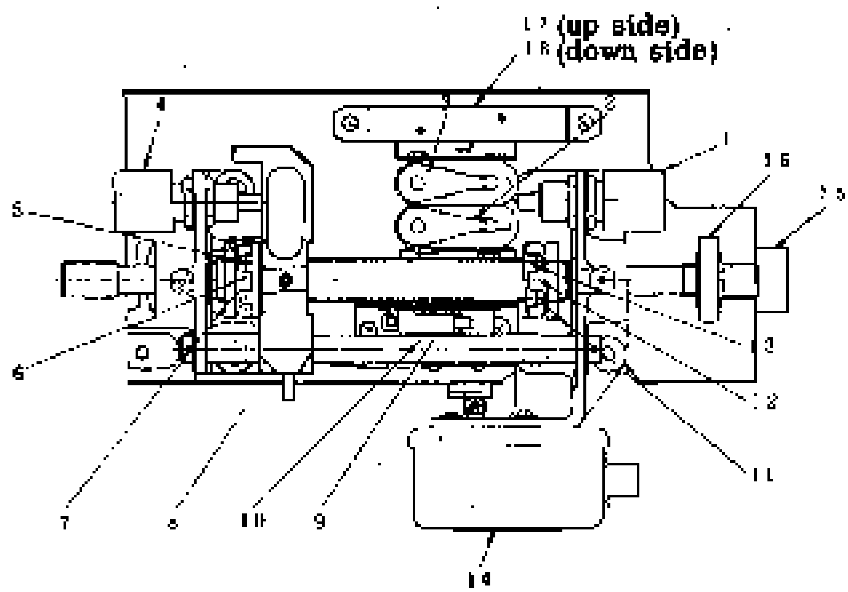
S421 5° down limit switch
This limit switch stops flap at 5° ± 1° of flap angle, when flap moves from UP position to 5° down position.

S422 6° down limit switch
This limit switch stops flap at 6° ±1.5° of flap angle, when flap moves from 40° or 20° down positions to 5° down position.

S431 pulse operating limit switch
This limit switch makes pulse operation such as 1.5 seconds stop per one turn of pulse operating cam when flap moves from 0° DOWN to 20° DOWN or from 20° DOWN to 2° DOWN.

S433 0° down limit switch
This limit switch stops flap automatically for 1.5 seconds at 0° of deflection when flap moves from UP to 20° DOWN.

S434 2° down limit switch
This limit switch releases pulse operation automatically at 2° +1°/-0° of deflection when flap moves from 40° DOWN (or 20° DOWN) to UP.



1. 40° down limit switch
2. 20° down limit switch
3. 22° down limit switch
4. Up limit switch
5. Double lock nut
6. Lock washer
7. Mechanical stop nut (UP side)
8. Moving arm
9. 6° down limit switch
10. 5° down limit switch
11. Mechanical stop nut (DOWN side)
12. Lock washer
13. Double lock nut
14. Flap transmitter
15. Pulse operating limit switch
16. Pulse operating cam
17. 2° down limit switch
18. 0° down limit switch

Fig. 5-99 Flap stop assembly

9.4.2.2 PROCEDURE FOR ADJUSTMENT

(1) Connect the rotating shaft between flap motor main reduction gearbox and stop assembly so that the shaft may slide axially and smoothly.

(2) Disconnect rotating shaft of L.H. flap from flap motor main reduction gear and inboard actuator. Disconnect rotating shaft of R.H. flap from stop assembly and inboard actuator.

(3) Rotate shaft of flap stop assembly by motor. Stop shaft at the position where 20° down limit switch (S419) actuates.

(4) Set L.H. and R.H. flaps at 20° ± 1° of deflection by protractor.

(5) Connect L.H. flap with L.H. rotating shaft.

(6) Place flap in extended position.

NOTE

Do not place flap in UP position, while UP limit switch has not been adjusted.

(7) Return flap from extended position to 20° down position.
Check flaps for deflection of 20° ± 1°, and difference in deflection of less than ± 0.5° between L.H. and R.H. flaps.
If deflection is not within specified range, adjust flap by relocating 20° down limit switch (S419) or by changing engagement of rotating shaft serration.

(8) Gently extend flap by motor from 20° down position to 40° down position. Stop at 40° of deflection. Tentatively locate 40° down limit switch.

(9) Gently retract flap from 40° down position to UP. Stop at the position where flap is aligned with wing at the alignment marks. Tentatively locate UP limit switch.

(10) Gently retract flap from 40° down position to 20° down position. Stop at 22° ± 1° of deflection. Locate 22° down limit switch (S420).

(11) Gently, retract flap from 20° down position to UP. Stop at 6° ± 1.5° of deflection. Locate 6° down limit switch (S422).

(12) Gently, extend flap from UP to 5° down position. Adjust the pulse operating limit switch and pulse operating cam so that the switch may be turned ON in the point "a" of cam at 1° (+30', -0) of deflection. Then adjust 2° down limit switch so that the switch may be turned ON at the position where the torque tube is returned back 0.5 turn to the UP direction.

(13) Gently extend flap from UP to 5° down position. Stop at 5° ± 1° of deflection. Locate 5° down limit switch (S421).

(14) After adjustment of all limit switches is completed, actuate flap by motor. Check for proper deflection at UP, 5°, 20° and 40° down positions. If deflection is within specified range, fix limit switches securely.

(15) Adjust mechanical stop at UP and 40° down positions. (See Para. 9.4.3 and 9.4.4 of this manual.)

NOTE

It must be noted that S419≠S420, which S419 denotes deflection of flap with S419 actuated, and S420 denotes deflection of flap with S420 actuated.
It must be also noted that S421≠S422, which S421 denotes deflection of flap with S421 actuated, and S422 denotes deflection of flap with S422 actuated.

9.4.3 ADJUSTMENT OF UP LIMIT SWITCH AND UP POSITION MECHANICAL STOP

This limit switch automatically stops flap when flap moves to retracted position (0° ± 1° of deflection).

(1) Place flap in retracted position, where flap comes in contact with wing structures. Return torque shaft one and a half turns. Apply alignment mark to clarify the flap position against wing trailing edge.

(2) Return torque shaft an additional one turn by hand. Tentatively locate UP limit switch.

(3) After adjustment of UP limit switch is completed, normally operate flap by motor, and allow flap to stop automatically at UP position.

(4) At this position, set UP position mechanical stop nut to touch the moving arm. Return stop nut one turn. Stop nut must be away from moving arm by 0.04 ± 0.005 in. (1 ± 0.13 mm). Tighten stop nut which is double nut to 250~350 in-lbs (288~403 kg-cm) torque using lock washer. Make sure that the lock washer tabs securely catch the stop nut.

9.4.4 ADJUSTMENT OF 40° DOWN LIMIT SWITCH AND DOWN POSITION MECHANICAL STOP

This limit switch automatically stops flap when flap moves to 40° down position. (40° ± 1° of deflection).

(1) Extend flap by hand to the position, where flap comes in contact with wing structure. Return torque shaft one and a half turns. Record the deflection.

(2) Return torque shaft an additional one turn. Locate 40° down limit switch.

(3) After adjustment of down limit switch is completed, normally operate flap by motor, and allow flap to stop automatically at 40° down position.

(4) At this position, set 40° down side mechanical stop nut to touch the moving arm. Return stop nut one turn. Stop nut must be away from moving arm by 0.04 ± 0.005 in. (1 ± 0.13 mm). Tighten stop nut which is double nut to 250~350 in-lbs (288~403 kg-cm) torque using lock washer. Make sure that the lock washer tabs securely catch the stop nut.

(5) Normally operate flap by motor. Allow flap to stop automatically at 40° down position. Check for clearance from the structure. (See Fig. 5-100) Gap between jack support and jack nut should be over 0.2 in. (5 mm). Gap between nut support and roller under guide rail should be over 0.2 in. (5 mm).

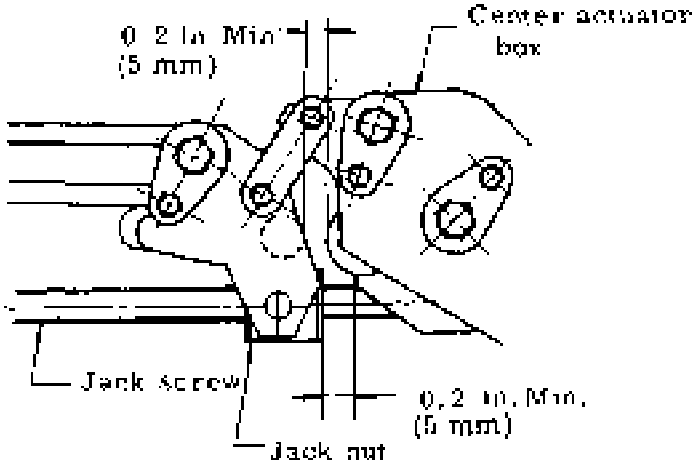


Fig. 5-100 Gap in 40° down position

9.5 OPERATIONAL CHECK OF FLAP SYSTEM

After rigging of system and operation of flap system is completed, perform operational check in accordance with the following procedures.

(1) Turn circuit breakers FLAP CONT and FLAP MOTOR on.

(2) Place flap control switch in UP, 5°, 20° and 40° positions in sequence. Measure deflections as follows:

Control SW	Flap movement	Deflection	Difference in deflections of LH and RH flaps
UP ➡ 5°	Stop at 5° position	5° ± 1°	± 30'
5° ➡ 20°	Stop at 20° position	20° ± 1°	± 30'
20° ➡ 40°	Stop at 40° position	40° ± 1°	± 30'
40° ➡ 20°	Stop at 20° position	22° ± 1°	± 30'
20° ➡ 5°	Stop at 5° position	6° ± 1.5°	± 30'
5° ➡ UP	Stop at UP position	0° ± 1°	± 30'
UP ➡ 40°	Move to 40° position without stopping	40° ± 1°	± 30'
40° ➡ UP	Move to UP position without stopping	0° ± 1°	± 30'

(3) Count travel time for movement of flap with 28 ~ 29V of bus voltage.

	(*1)	(*2)	
UP ➡ 40° DOWN	**23.5 +4/-0**	**25.0 +4/-0**	**(including pulse operation on the way)**
40° DOWN ➡ UP	**29.5 +4/-0**	**31.0 +4/-0**	**(including pulse operation on the way)**

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*2 Aircraft S/N 661SA, 697SA and subsequent

(4) Operate flap control switch. Check control surface and indicator light for proper operation.

Flap control SW	Control Surface	Indicator light
UP	Stop at retracted position	Green
5°	Stop at 5°	Yellow
20°	Stop at 20°	Yellow
40°	Stop at 40°	Yellow

(5) Check flap system for deformation, interference, strange noise and smooth movement over the entire range of travel.

(6) Check torque shafts and actuator guide rail for damage.
Check lock nut, cotter pin and retainer ring for proper installation.

(7) Check alignment marks for proper indication at mid point and at inner and outer ends of flap.

(8) Check flap jack screw and jack nut for lubrication.

9.6 MEASUREMENT OF FLAP MOTOR CURRENT

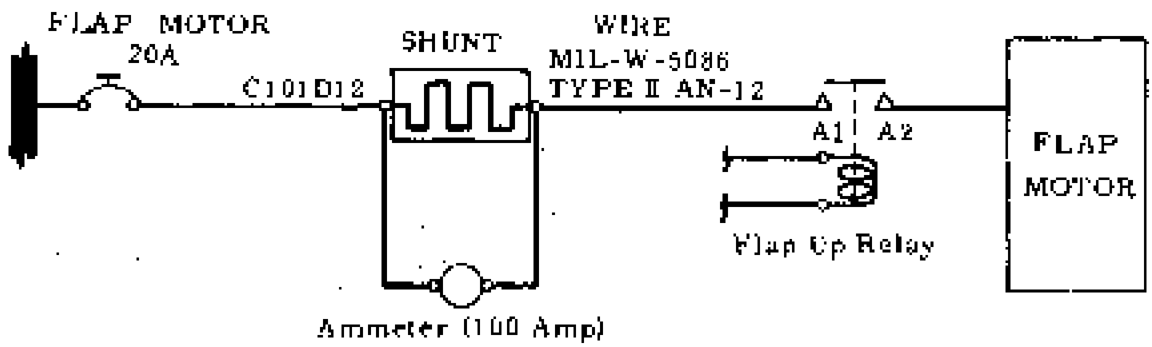
Measurement of flap motor current should be made as necessary when abnormal load is acted on flap control system or major components of this system are replaced.

9.6.1 PROCEDURE:

(1) Remove access panels on wing and center wing trailing edge.

(2) Disconnect electric wire (C101D12) connected to terminal of flap motor relay located at L.H. side, center wing trailing edge.
Connect the above wire to positive side terminal of shunt.

(3) Connect an electric wire (MIL-W-5086 TYPE II AN-12) prepared for measurement to negative side terminal of shunt and flap motor relay.



maintenance manual

TEMPORARY REVISION NO.5-3

This Temporary Revision No. 5-3 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	5-60
MU-2B-60	MR-0336	5-56

Insert facing the page indicated above for the applicable Maintenance Manual.
Retain this Temporary Revision until such time as a permanent revision on this subject is issued

REASON : To add wear limits for flap actuator jack screws.

ADD : Paragraph 10.1. (1A) and table as follows after paragraph 10.1.

(1A) Flap actuator jack screws

Name of parts inspected	Inspection method	Maximum Wear limit
Flap main actuator jack screw	Measure width of top of thread	0.06 in. (1.5 mm)
Flap inboard actuator jack screw	Measure width of top of thread	0.032 in (0.8 mm)
Flap outboard actuator jack screw	Measure width of top of thread	0.032 in (0.8mm)

CAUTION

Turn power switch to "OFF" before connecting wire. Connect a wire to shunt and ammeter so as not to interfere with flap mechanism.

(4) Connect wire from shunt to ammeter terminal.

(5) Actuate flaps and measure current of flap motor carefully.

(6) Make sure that operating currents are as follows, except for a moment of starting.

Full cycle of flap up operation	12 A or less
Full cycle of flap down operation	14 A or less

(7) After measurement is completed, remove ammeter, shunt and electric wire and restore the system.

10. WEAR LIMIT AND FREE PLAY ALLOWANCE

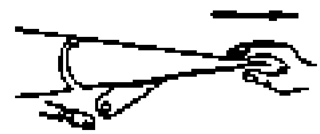
10.1 WEAR LIMIT

(1) Flap actuator jack nut

Name of parts inspected	Inspection method	Abrasion limit
Flap main actuator jack nut	Measure width of top of thread	0.04 in. (1 mm)
Flap inboard actuator jack nut	" "	0.032 in. (0.8mm)
Flap outboard actuator jack nut	" "	0.032 in. (0.8mm)

NOTE: Please see the TEMPORARY REVISION that revises this page.

(2) Attaching points and operating mechanisms of elevator trim tab, rudder trim tab and trim aileron.

Element	Total Value	Measuring Method	Remarks
Hinge	Hinge hole dia – Pin dia = 0.016 in. (0.4 mm)	Disconnect actuating rod and control surface, push and pull control surface and measure movement.	
Connecting point and control surface	Bracket hole dia – Bolt dia. = 0.016 in. (0.4 mm)	Disconnect actuating rod and control surface and measure inner dia. of hole and outer dia. of bolt	
Clevis Hole	Bracket hole dia. – Bolt dia. = 0.016 in. (0.4 mm)	Measure outer dia. of bolt and inner dia. of hole in actuating rod clevis	
Actuator	Free play in longitudinal direction = 0.016 in. (0.4 mm)	Push and pull actuating shaft in longitudinal direction and measure movement	

(3) Flap guide rail

Name of Parts Inspected	Inspection method	Abrasion Limit
Flap main guide rail	Measure the depth of wear resulted from roller contact on lower track	0.04 in. (1 mm), (Max) Otherwire ; 0.35 in. (9 mm) Min. (Residual thickness)

10.2 CONTROL SURFACE FREE PLAY

(1) Flap, spoiler, elevator, rudder

Control Surface	Measuring Conditions	Total Travel Value
Flap	Deflection of trailing edge at retracted position, W.STA 2370. Deflection of trailing edge at 40° down position, W.STA 2370	0.2 in. (5 mm) 0.28 in. (7 mm)
Spoiler	Free play of trailing edge	0.06 in. (1.5 mm)
Elevator	Up and down deflection, inboard section of trailing edge	0.08 in. (2 mm)
Rudder	Left and right deflection, lower trailing edge.	0.12 in. (3 mm)

(2) Elevator trim, rudder trim, trim aileron

Check free play of trailing edge as free play of the whole system.

Control Surface	Total Travel Value	Measuring Method
Elevator trim tab	Trailing edge free play δ inboard edge 0.16 in. (4 mm) outboard edge 0.04 in. (1 mm)	Deflection angle 0° 
Rudder trim tab	Trailing edge free play lower edge 0.12 in. (3 mm) upper edge 0.1 in. (2.5 mm)	Deflection angle 0° Same figure as above.
Trim aileron (inboard)	Trailing edge free play 0.12 in (3 mm)	Deflection angle 0°
Trim aileron (outboard)	Trailing edge free play 0.08 in (2 mm)	Same figure as above.

11. CHECK OF ELECTRICAL COMPONENT

11.1 FLAP UP AND DOWN RELAY

This check will intend to preclude a significant closed failure (fail to open) condition of the relays.

The following equipment is required to conduct this check.

- Insulation resistance tester (500 volts r.m.s.)
- DC power supply (0 to 50 volts, 1 ampere or above)
- Adequate Continuity checker

(1) Preparation

(a) Remove all electrical power from the aircraft.

(b) Disconnect all wires from the terminals of each relay and tag for identification.

(2) Insulation resistance check

(a) Measure insulation resistance between power terminals (A1 and A2), and power terminal (A1 or A2) to case (grounded).

(b) Insulation resistance values shall be greater than 50 megohms.

(c) Any relay which does not meet above requirement shall be replaced with a new relay.

(3) Contact operation check

(a) Connect a continuity checker between the power terminals (A1 and A2).

(b) Connect a power supply to the coil terminals X1(pos.) and X2(neg.) and set voltage at 28 volts.

(c) Check for proper continuity / discontinuity in the energized and de-energized positions.

(d) Replace the relay with a new relay if this check indicates a failure

(4) Restoration

(a) Remove the above test equipment and reconnect the wires

(b) Restore electrical power to the aircraft.

CHAPTER

6

POWER PLANT

CHAPTER VI

POWER PLANT

CONTENTS

1. GENERAL DESCRIPTION

The power plant of the aircraft consists of engines, propellers, engine control system, propeller synchrophaser system and cooling system. Engine mounts are of soft type. Openings are provided on left and right sides and top of engine nacelles for ease of maintenance and check. The aircraft is equipped with two each GARRETT (*1) TPE 331-6-251M, (*2) TPE 331-5-252M single shaft, turbo-prop engines.

The engine consists of three main sections and five main systems: a two stage centrifugal flow compressor section, a three stage axial flow turbine section, a reduction gear section, a propeller governing system, a fuel system, a lubrication system, a torque sensing system and an electrical system.

Occasionally it becomes necessary to pull (turn) the engine through by hand. Turn in the direction of normal rotation only, due to possible damage to engine component parts.

2. CONSTRUCTION AND OPERATION OF ENGINE

2.1 FUEL CONTROL SYSTEM

2.2 LUBRICATING SYSTEM

2.3 PROPELLER CONTROL SYSTEM

2.4 ELECTRIC SYSTEM

2.5 TORQUE SENSING SYSTEM

2.6 ENGINE ANTI-ICING SYSTEM

2.7 TORQUE AND INTERSTAGE TURBINE TEMPERATURE CONTROL SYSTEM

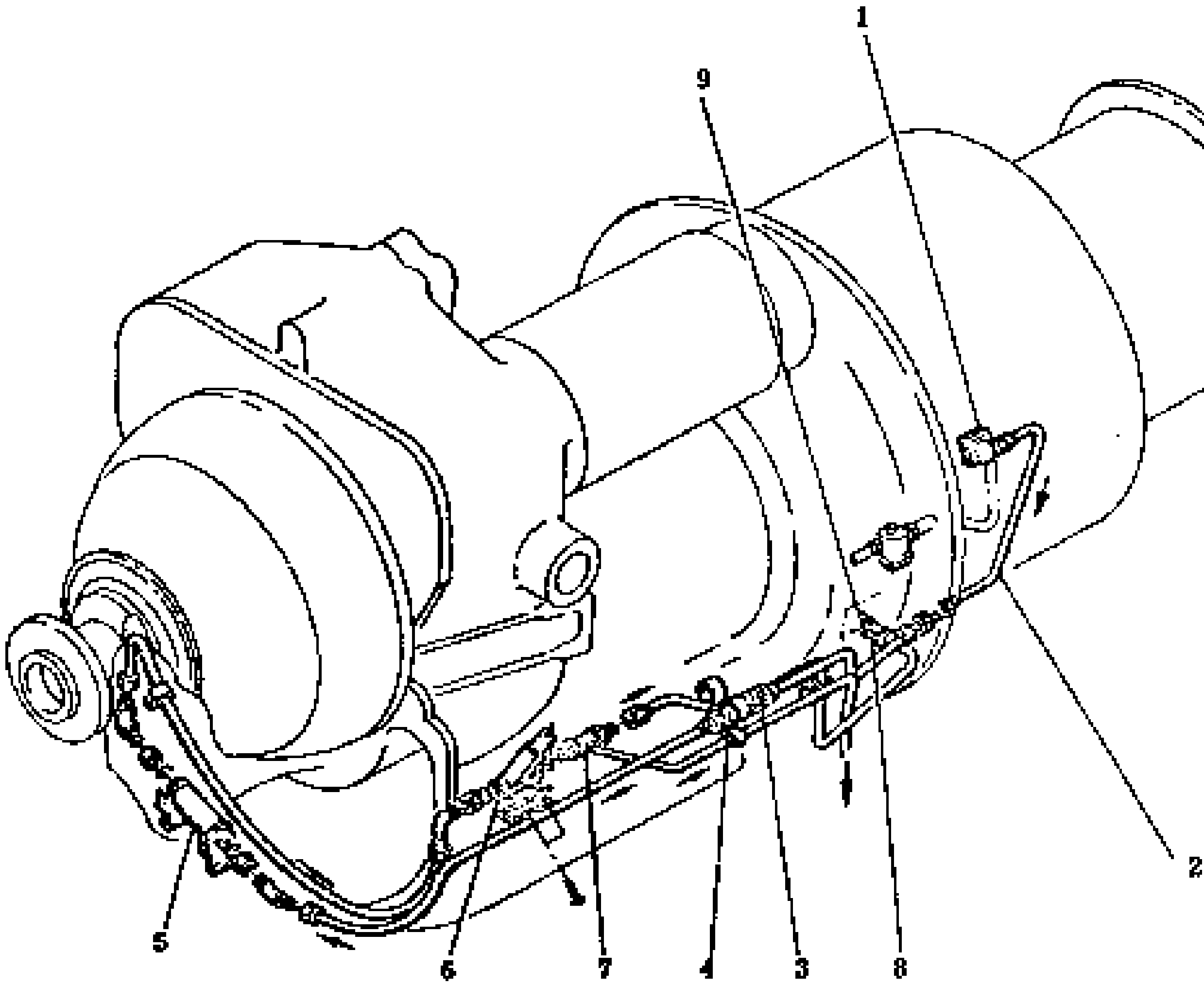
For Para.2.1 thru Para.2.6, see GARRETT Engine Maintenance Manual.
For Para.2.7, see GARRETT Engine Maintenance Manual and Para.14.

2.8 FUEL PURGE SYSTEM

(1) General Description
This system is to burn fuel remaining in manifold and flow divider, etc., by force prior to complete shutdown of engine in order to prohibit discharging fuel from engine manifold and flow divider into the atmosphere during engine shutdown. This system consists of engine plenum air (P3) tubing, filter, accumulator, solenoid valve, elbow and check valve, see Fig.6-1.

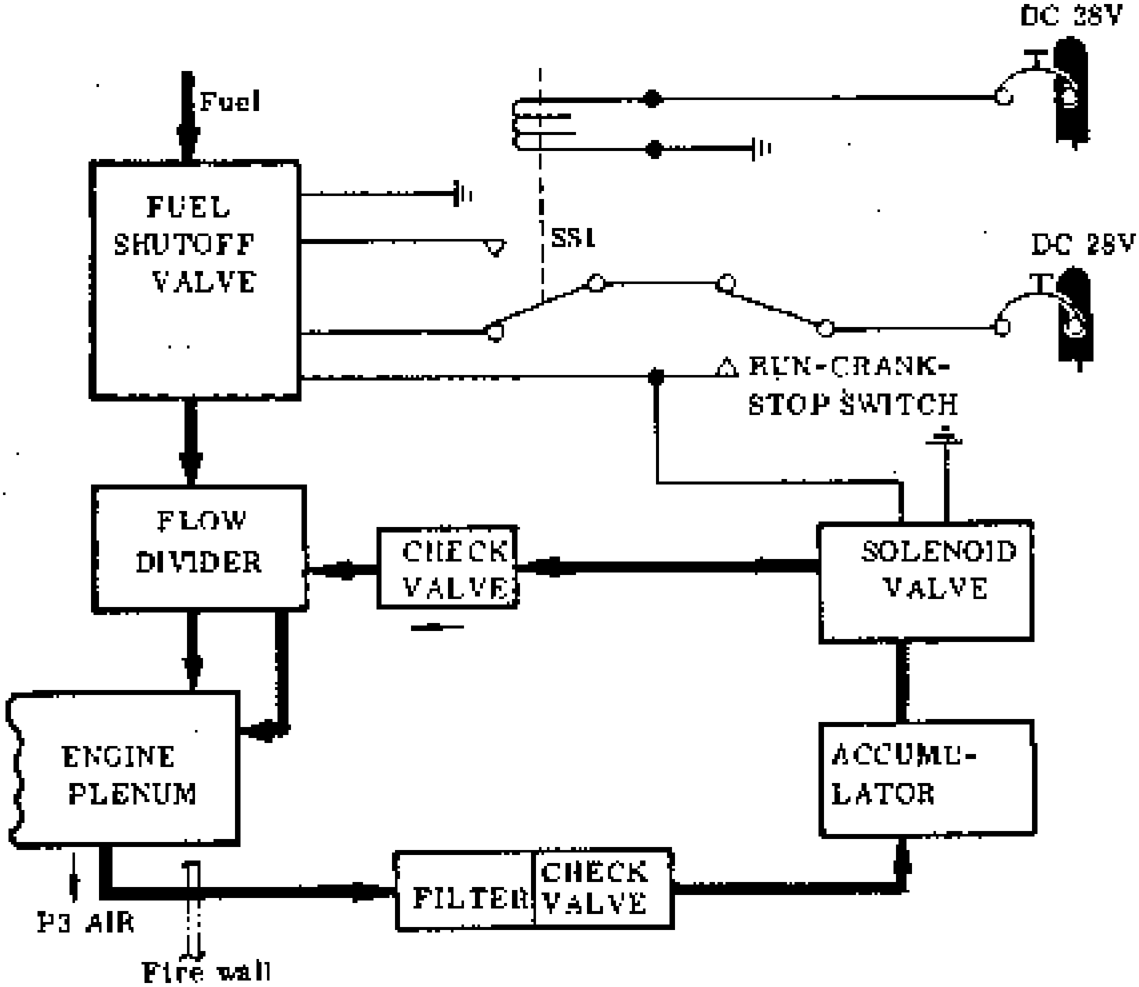
(2) Operation
This system is to accumulate engine plenum air (P3) in accumulator during engine operation. When RUN-CRANK-STOP switch is turned to STOP position, solenoid valve is opened, engine plenum air (P3) is fed to flow divider, and remaining fuel is blown out of fuel nozzle and burned.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent



1. Engine plenum chamber P3 port
2. Engine P3 line (engine side)
3. Air filter
4. Check valve
5. Accumulator
6. Solenoid valve
7. Elbow
8. Check valve
9. Flow divider

Fig. 6-1 Fuel purge system



3. REMOVAL, INSTALLATION AND OPERATIONAL CHECK OF ENGINE ACCESSORIES

For this paragraph, see AiResearch Engine Maintenance Manual.

NOTE

i To close the nacelle upper door after maintenance and inspection of engine, the stay must be fixed surely with lockpin to the nacelle upper door or removed and stored.

ii When work is done near plenum chamber in engine nacelle, air intake leading to front of oil cooler from nacelle should be covered with a tape to keep foreign objects out.

3.1 REMOVAL AND INSTALLATION OF SPEED SWITCH (See Fig. 6-2)

The speed switch is installed on the airframe side, but it is handled as one of engine accessories and must be replaced together with engine in engine replacement.
The speed switch is removed as follows ;

(1) Remove center wing leading edge access door.

(2) Remove plugs. (P455 for L.H and P456 for R.H)

(3) Remove washers and nuts. (4 ea. for L.H and R.H)

(4) Remove speed switch. (1 ea. for L.H and R.H)

(5) Install in reverse sequence of removal.

3.2 REMOVAL AND INSTALLATION OF TORQUE TRANSDUCER (See Fig. 6-3)

The torque transducer is installed on the airframe side, but it is handled as one of engine accessories and must be replaced together with engine in engine replacement.
The torque transducer is removed as follows ;

(1) Remove wing upper fire extinguisher access door.

(2) Remove plugs. (P424, P425).

(3) Remove tubings. (2 ea. for L.H and R.H)

(4) Remove washers and nuts. (2 ea. for L.H and R.H)

(5) Remove torque transducer. (1 ea. for L.H and R.H)

(6) Install in reverse sequence of removal.

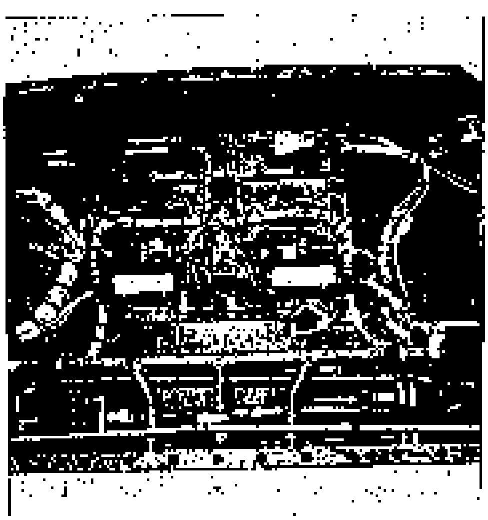


Fig. 6-2 Installation of speed switch

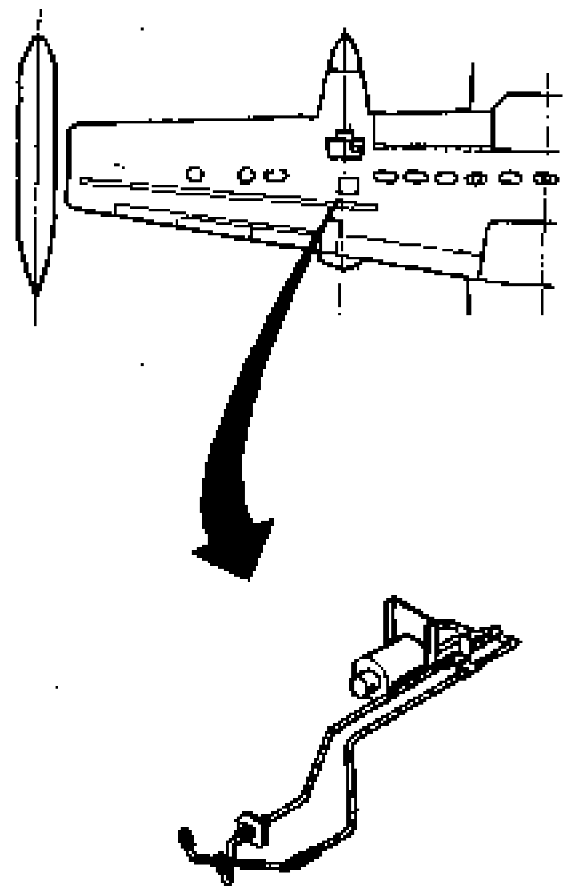


Fig. 6-3 Installation of torque transducer

4. LUBRICATING OIL COOLING SYSTEM

4.1 GENERAL DESCRIPTION

The lubricating oil cooling system consists of fuel-oil heat exchanger, air-oil heat exchanger and bypass valve. The fuel-oil heat exchanger located on the side wall of the output housing transfers heat from lubricating oil to fuel. Air-oil heat exchanger located beneath the exhaust pipe in the rear of nacelle introduces air from L.H. and R.H. air intakes and transfers heat from lubricating oil to air.
The bypass valve adjusts oil temperature and regulates pressure.
When oil temperature rises, the valve closes. When oil temperature falls, the valve opens to keep the oil temperature more than 131°F. When the pressure between oil pump and air-oil heat exchanger rises due to blocking of the exchanger, the valve opens and returns oil to oil tank. (See Fig. 6-4)

4.2 BYPASS VALVE (See Fig. 6-5)

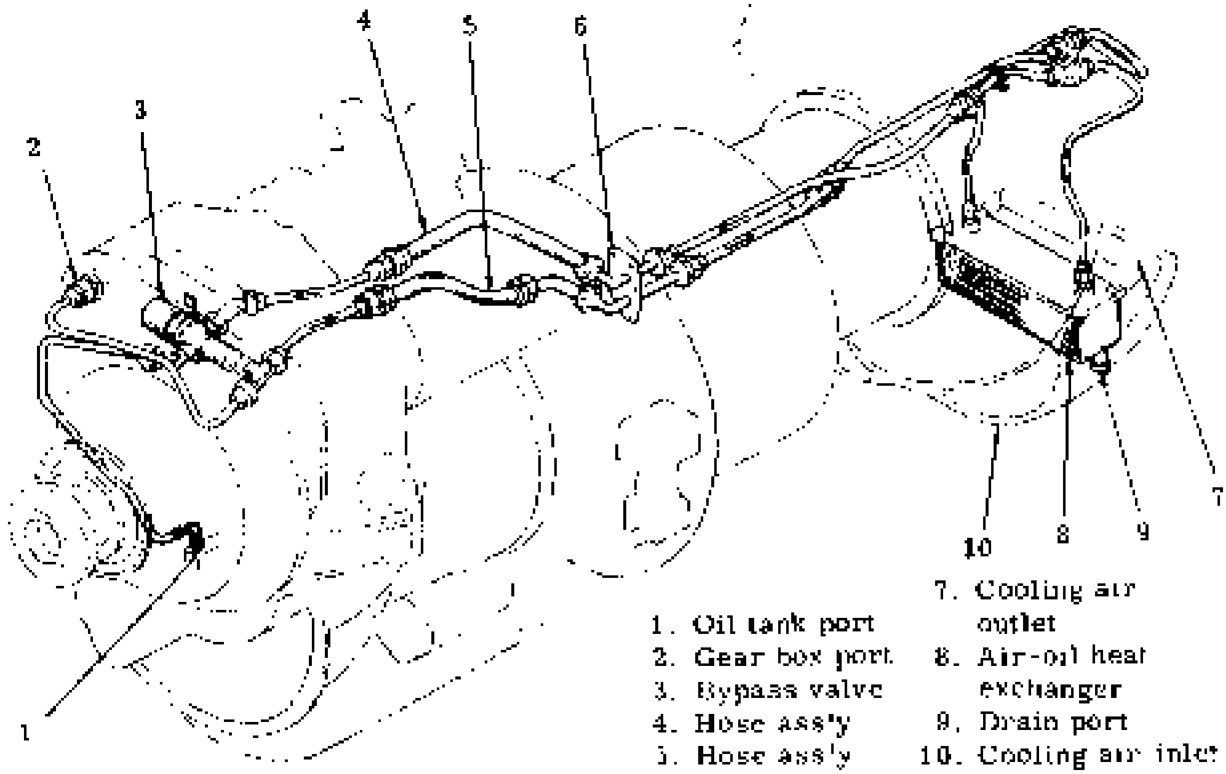
Temperature and pressure settings of bypass valve are as follows:

Temperature settings	150°F............Valve is open.
	165°F............Valve is fully closed.
Pressure settings	(Oil temperature 90°C)
	10 psi ....Valve is closed.
	40 psi ....Valve is open.

4.3 COOLING IN NACELLE

Air introduced from air intake prevents temperatures in the following locations from rising.

(1) Zone I
(2) Zone II
(3) Area between nacelle side door outer skin and firewall
(4) Area between upper firewall and wing lower surface
(5) Fire extinguishing container compartment
(6) Starter generator



1. Oil tank port
2. Gear box port
3. Bypass valve
4. Hose ass'y
5. Hose ass'y
6. Tube ass'y-fire wall
7. Cooling air outlet
8. Air-oil heat exchanger
9. Drain port
10. Cooling air inlet

Fig. 6-4 Lubrication oil cooling system

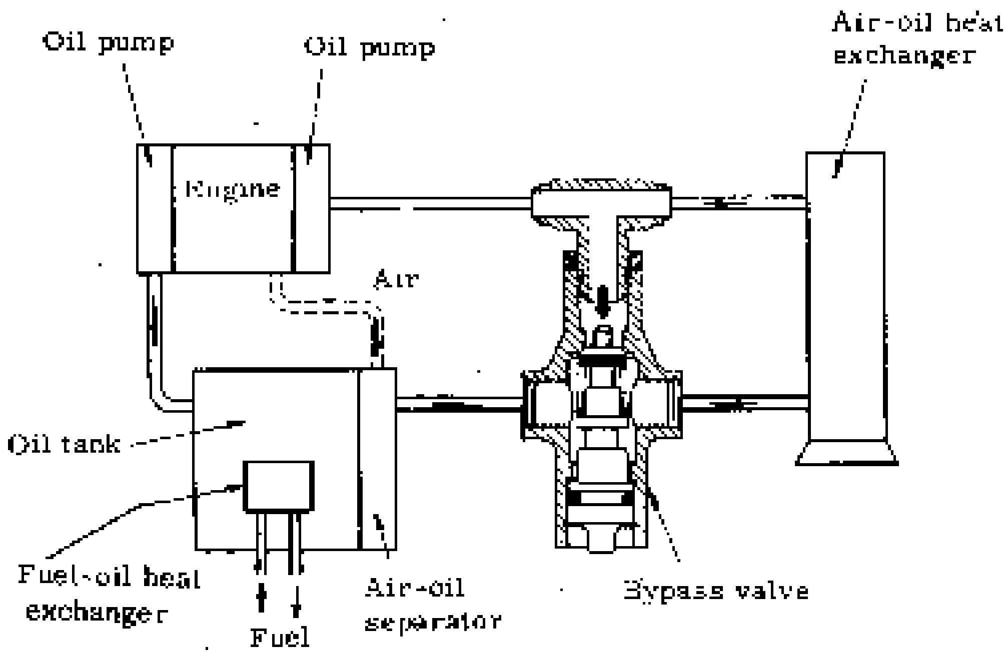


Fig. 6-5 Bypass valve

5. ASSEMBLY OF ENGINE

5.1 ASSEMBLY

NOTE

For installation torque of bolts, nuts, tubing and fittings, etc., in case of direct installation of airframe parts onto engine and engine accessories, see Engine Maintenance Manual unless otherwise specified.

(1) Preparation

(a) Make sure that engine is in the following condition and adjust per Engine Maintenance Manual if necessary.

. Power control system

Fuel Control Unit	PROP PITCH CONT.
F/I	Rig pin can be inserted.
Max. Stop	$100^{\circ} \pm 1^{\circ}$
Min. Stop	Rig pin can be inserted.

. Condition control system

Fuel Control Unit	PROP GOVERNOR
Max. Stop ($42^{\circ} \pm 1^{\circ}$)	Max. Stop
$30^{\circ} \pm 3^{\circ}$	Min. Stop, micro adjusting screw and arm G hit at the same time.

(b) Specific gravity adjustment

Perform specific gravity adjustment of fuel control unit depending upon type of fuel used, in accordance with Engine Maintenance Manual.

(c) Make sure that lever of fuel shutoff valve is in the position as shown in Fig 6-22. If not, remove the lever and reinstall so that the mark on the shaft is as shown in the figure.

(d) Remove fuel line between fuel control unit and fuel shutoff valve and install fuel flow transmitter. (See Fig. 6-7)

(2) Install β-pressure switch, NTS check switch, and fuel pressure switch with adapter.

(3) Install tachometer generator. (See Fig 6-6) Tighten bolt(NAS1021C4) to 65 70 in-lbs torque. Apply a light coating of grease(MIL-G-21164) on fit of generator shaft.

(4) Install bracket for starter generator on engine mounting pads. Tighten nuts to 120~130 in-lbs. (See Chapter XIII Para. 2.2.2)

(5) Connect wiring harness assembly to connector of each component.

(6) Install emergency stop mechanism. For installation and adjustment of link mechanism, see Para. 5.4.

(7) Install levers to propeller pitch control governor and underspeed governor of the fuel control unit. Install engine shock mount and engine mount assembly, see Para. 5.3.

(8) Install bracket to engine mount assembly. Install unfeathering pump. Apply safety wire on four attaching bolts, see Fig. 6-7.

Fig. 6-6 Installation of tachometer generator

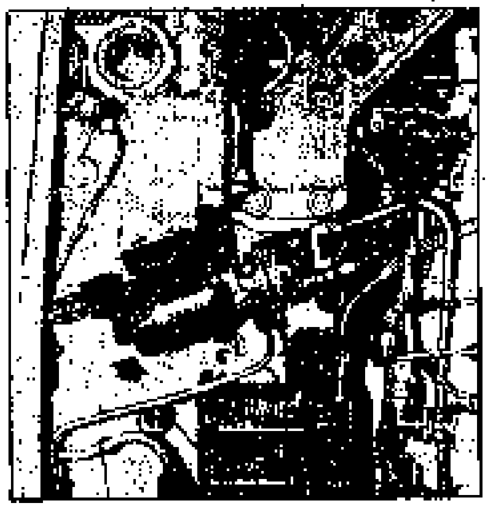


Fig. 6-7 Installation of unfeathering pump and fuel flow transmitter

(9) Install tubes between unfeathering pump and engine, and unfeathering pump and oil tank.

(10) Install engine bleed air duct, see Fig. 6-8.

(a) Tighten adapter of bleed air duct on engine plenum chamber, and gasket with bolts. Installation torque is 40~50 in-lbs. (46~58 kg-cm).

(b) Install gasket on flange in front of air bleed duct between N. STA 1300 to 1635 and connect duct with coupling.
Installation torque for coupling is 65 to 75 in-lbs. (75 to 85 kg-cm).

(11) Connect a test wire to magnetic drain plug at the bottom of engine output housing, see Fig. 6-9.

(a) Remove magnetic drain plug and install wire.

(b) Reinstall the plug and apply safety wire. Tighten the plug and nut to 90~95 in-lbs. (104 ~ 109 kg-cm). Check magnetic plug for installation resistance.

(12) Install engine front cowling, see Fig. 6-10.

(a) Install an electric terminal block with two screws on cowling before installation of cowling, see Fig. 6-11.

(b) Install front cowling on engine. Tighten attaching bolts to 15~20 in-lbs. (17~23 kg-cm).

Fig. 6-8 Installation of engine bleed air duct

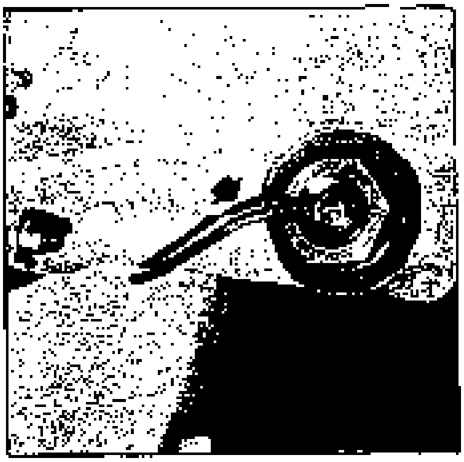


Fig. 6-9 Installation of wire for magnetic drain plug

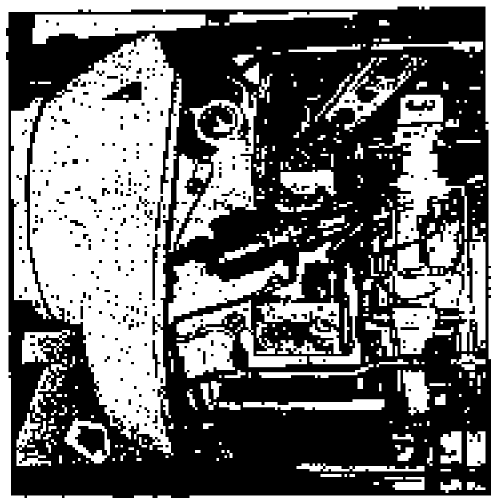


Fig. 6-10 Installation of engine cowling

(13) Install oil pressure transmitter.

(a) Install attaching bracket for oil pressure transmitter to cowling and install oil pressure transmitter to bracket.

(b) Install tube between oil pressure output and oil pressure transmitter.

(14) Install engine oil bypass valve, see Fig. 6-12.

(a) Place union, elbow and tee into bypass valve and install the valve with clamp and bracket on output gear housing.
Apply a light coating of grease (VV-P-236) on O-rings used in union, elbow and tee.

Fig.6-11 Installation of terminal block

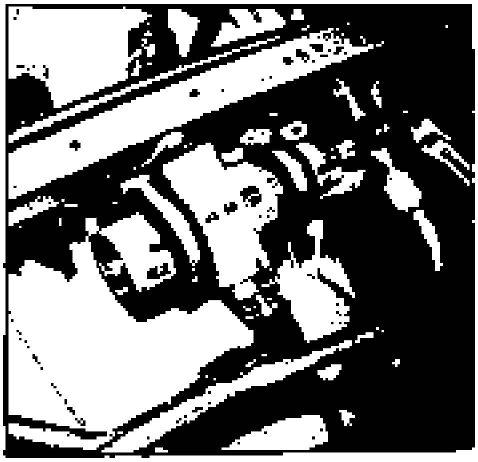


Fig.6-12 Installation of engine oil bypass valve

(15) Install cowling anti-ice line. (See Fig.6-13)
(a) Tighten coupling nuts. And paint slipmarks on nut and sleeve plumbing.
(b) Apply safety wire to coupling nuts.

(16) Install bracket for brush block of propeller anti-icing. (See Fig.6-14)

(17) Install drain and vent lines. (See Figs.6-15 and 6-16)
(a) Install drain line for start pressure regulator.
(b) Install drain line for unfeathering pump.
(c) Install drain lines for fuel control unit.
(d) Install vent line for oil tank.
(e) Install front and rear drain lines on turbine plenum and fasten with clamps.
(f) Install drain line for input gear housing.

(18) Connect tubing and hose to torque sensor adapter.

Fig.6-14 Installation of bracket for prop. anti-icing brush block

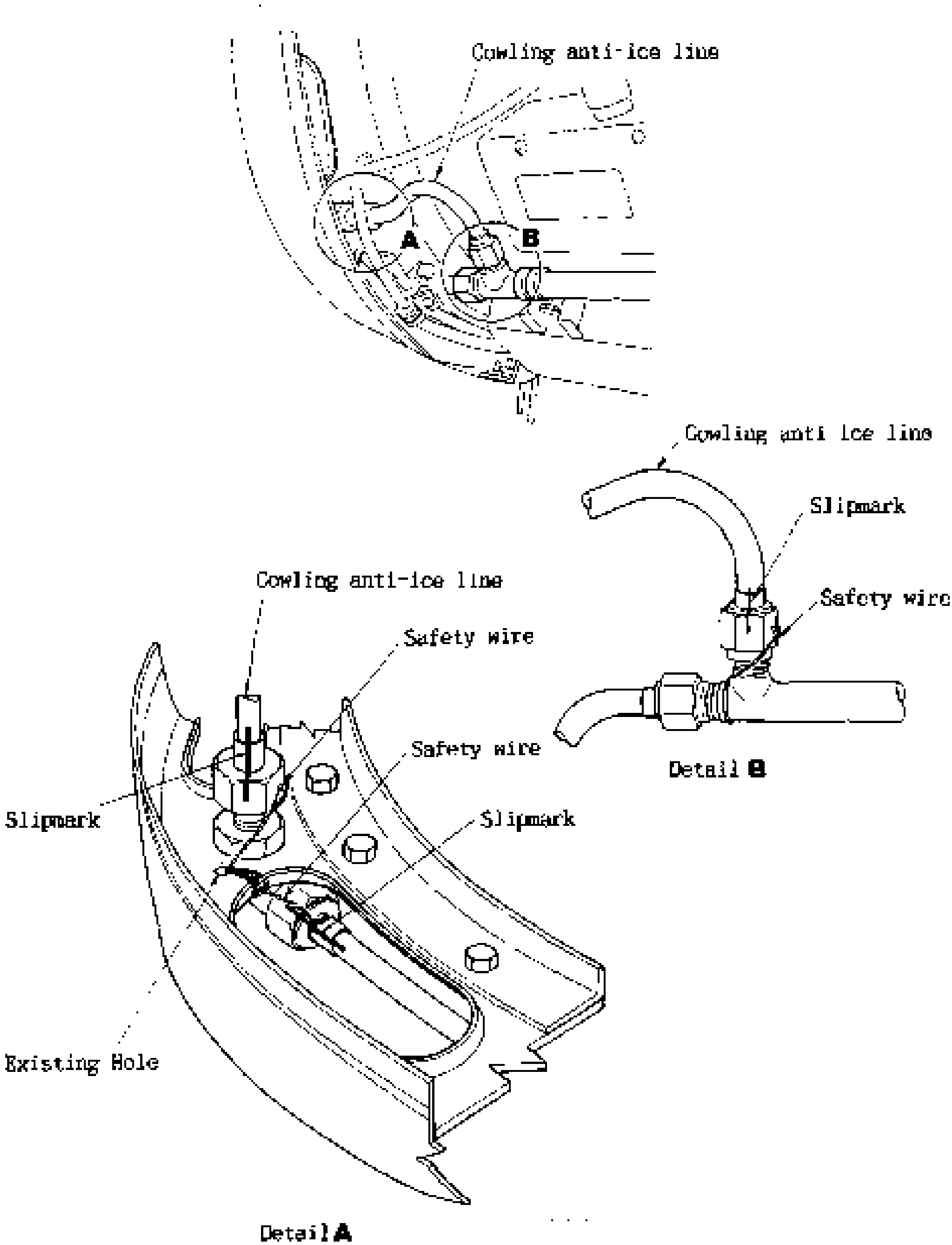


Fig.6-13 Installation of cowling anti-ice line

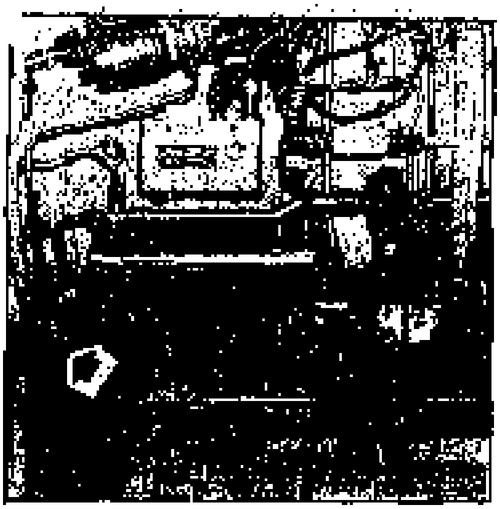


Fig. 6-15 Installation of forward drain & vent line

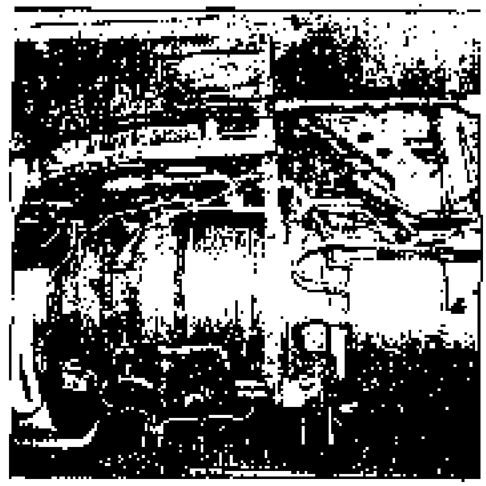


Fig. 6-16 Installation of aft drain line

(19) Install engine fire extinguisher line (if installed). (See Figs. 6-17, 6-18)

(a) Install clamps and brackets on circumference of "ZONE I" together with engine output gear housing.

(b) Install clamps and brackets on circumference of "ZONE II" together with turbine plenum attaching bolts. Tighten bolt to 50~55 in-lbs(58~63 kg-cm).

(c) Install engine fire extinguisher line and fix with clamps.

(20) Install fire detector.

(a) Fix sensing element attaching clips at brackets around engine. When the bracket is tightened together with turbine plenum, tighten bolt to 50~55 in-lbs.(58~63 kg-cm).

NOTE

Sensing element should be installed with grommet.

(b) Connect sensing element and flexible cable at three connector plugs.

Fig. 6-17 Installation of fire extinguisher line in ZONE I

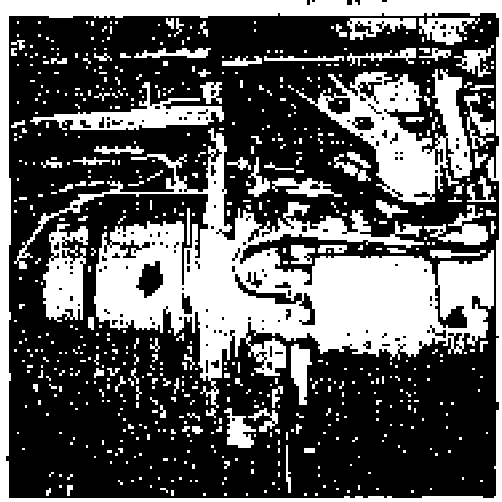


Fig. 6-18 Installation of fire extinguisher line in ZONE II

(21) Install fuel purge system.

(22) Install fuel pressure transmitter. (See Fig. 10-24)

(a) Install bracket for fuel pressure transmitter.

(b) Install fuel pressure transmitter on the bracket.
(c) Install filter on the fuel pressure line of fuel pressure transmitter.
(d) Install hose between filter and fuel pressure transmitter and fix it on horse collar with clamp.

5.2 INSTALLATION OF EXHAUST PIPE

(1) Install duct on engine exhaust section with bolts, nuts, and washers. Apply safety wire.

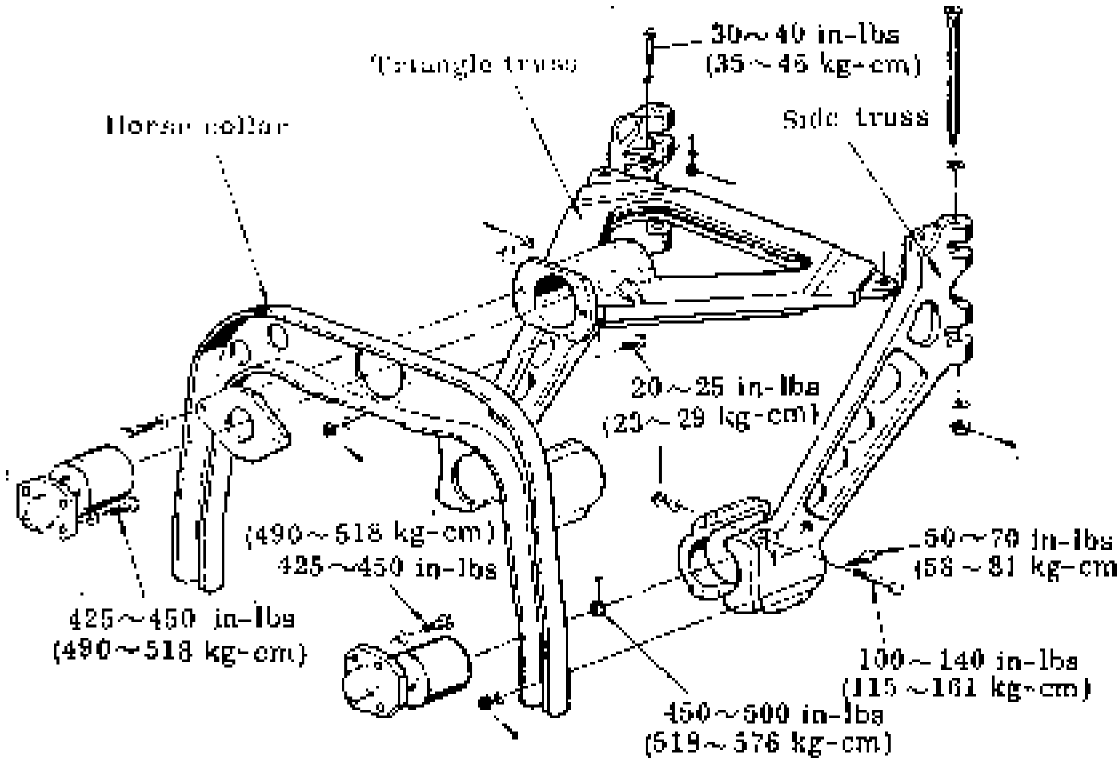


Fig. 6-19 Engine mount installation

5.3 ASSEMBLY OF ENGINE MOUNT (See Fig 6-19).

(1) Install shock isolator retaining fittings (3 ea) on shock isolator mount. Tighten bolts to 425 to 450 in-lbs (490 to 518 kg-cm) torque. Apply safety wire, MS20995C-32.

(2) Insert shock isolators into left and right side trusses and triangle truss. Tighten bolts to 100 to 140 in-lbs (115 to 161 kg-cm) torque and install cotter pins.

(3) Tighten attaching nuts of shock isolator to 450 to 500 in-lbs (518 to 576 kb-cm) torque. Apply alignment marks and install cotter pins.

(4) After installation of shock isolator, install side truss and triangle truss on the horse collar. Tighten bolts to 50 to 70 in-lbs (58 to 81 kg-cm) torque (side truss) and 20 to 25 in-lbs (23 to 29 kg-cm) torque (triangle truss). Install cotter pins.

(5) Connect engine mount and triangle truss with bolts. Tighten bolts to 30 to 40 in-lbs (35 to 46 kg-cm) torque. Install cotter pin.

5.3.1 ASSEMBLY OF REAR ENGINE MOUNT (See Fig. 6-20)

(1) Place the isolator into the attaching fitting; install the cover with bolts, washers and nuts and apply cotter pins.

(2) Tighten the bracket with bolts, washers and nuts and apply cotter pins.

NOTE

When installing the rear engine mount on the airframe, place the attaching fitting cover and bracket with the cutout facing forward.

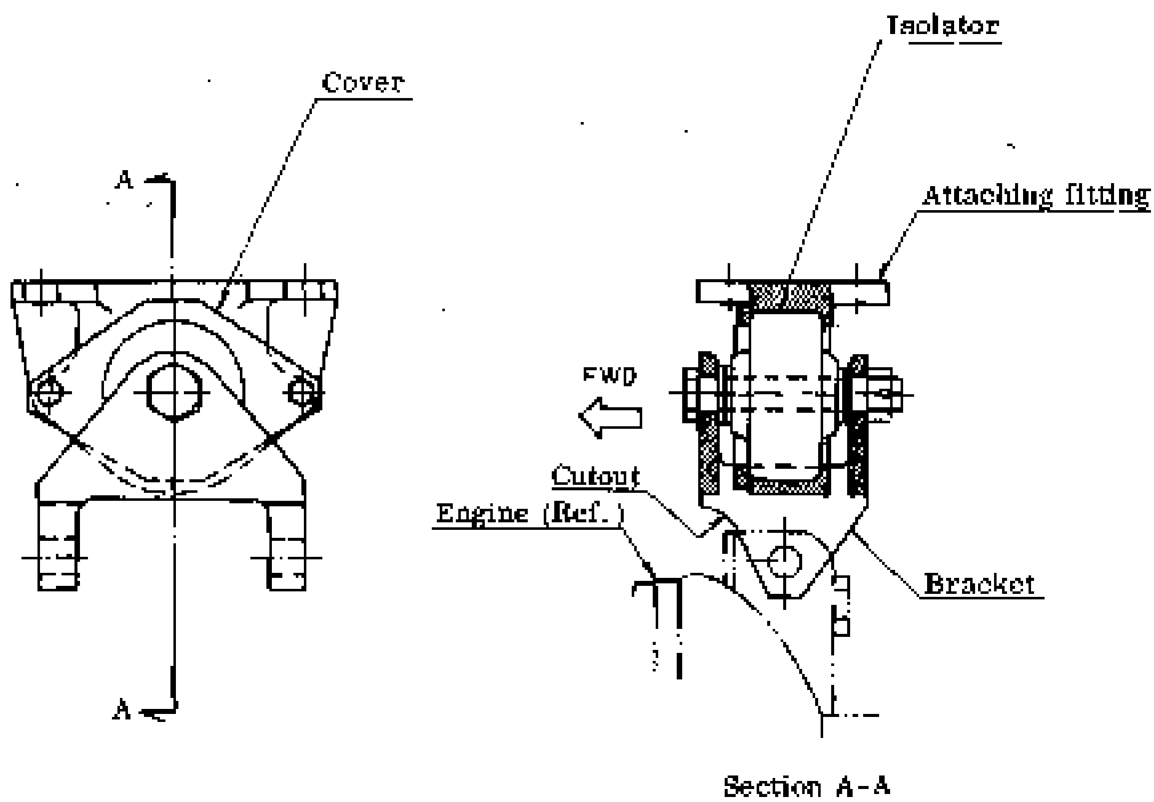
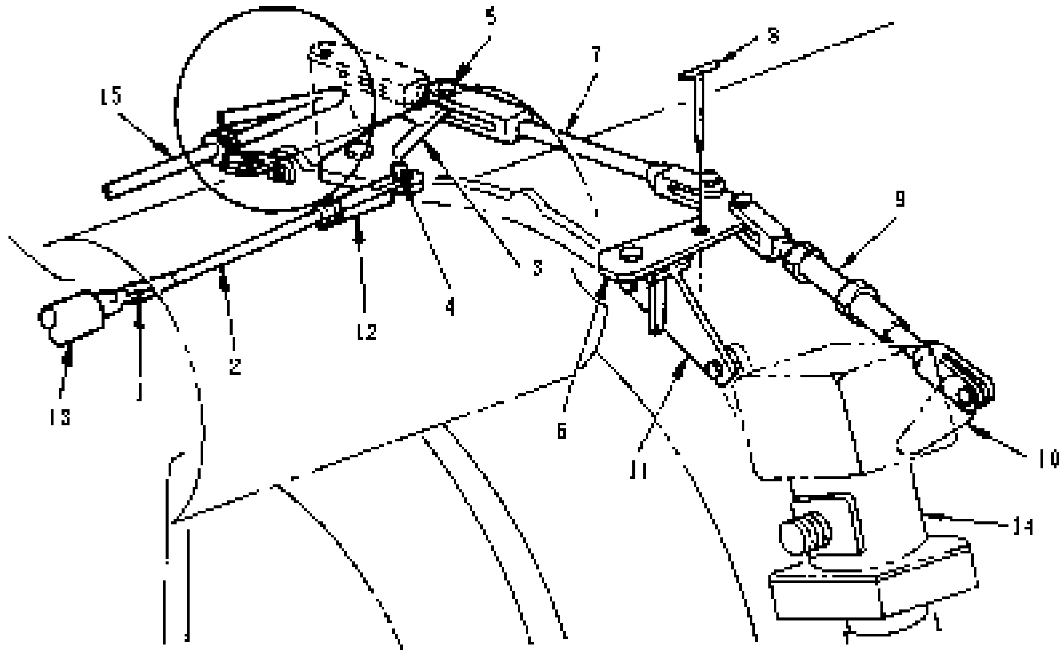


Fig. 6-20 Assembly of rear engine mount

5.4 INSTALLATION OF ENGINE EMERGENCY STOP MECHANISM (See Fig. 6-21)

(1) Install support 11 of linkage together with forward attaching bolts of compressor housing and connect bellcrank 6 and rod 9.
Tighten the bolts to 50 to 60 in-lbs (58 to 69 kg-cm) torque.
Connection of bellcrank and rod is made by pin (MS20392-2C15) and cotter pin.

(2) Install support 12 with forward attaching bolts of compressor housing.



1. Pin	5. Pin	9. Rod	13. Feather valve
2. Rod	6. Bellcrank	10. Lever	14. Fuel shutoff valve
3. Bellcrank	7. Rod	11. Support	15. P3 Line
4. Pin	8. Rig pin	12. Support	

Fig. 6-21 Installation of emergency engine stop mechanism

(3) Connect bellcrank 3 and rod 7 to support 12 of linkage.
Connection of bellcrank and rod is made by pin (MS20392-2C15) and cotter pin.

(4) Insert a rig pin 8 into bellcrank 6.

(5) Install rod 2 on feather valve 13, adjust the rod length and connect to bellcrank 3. Connection of feather valve 13 and rod 2 is made by pin (MS20392-2C15) and cotter pin. Make sure that connecting pin 4 of bellcrank 3 and rod 2 is in the long hole edge near the shaft.

(6) Adjust lever 10 of fuel shutoff valve to the position as shown in Fig. 6-22.

(7) Adjust length of rod 9 so that fuel shutoff valve is in "OPEN" position.

(8) Attach protractor to lever ⑩ of fuel shutoff valve. (See Fig.6-23)

(9) Pull out rig pin.

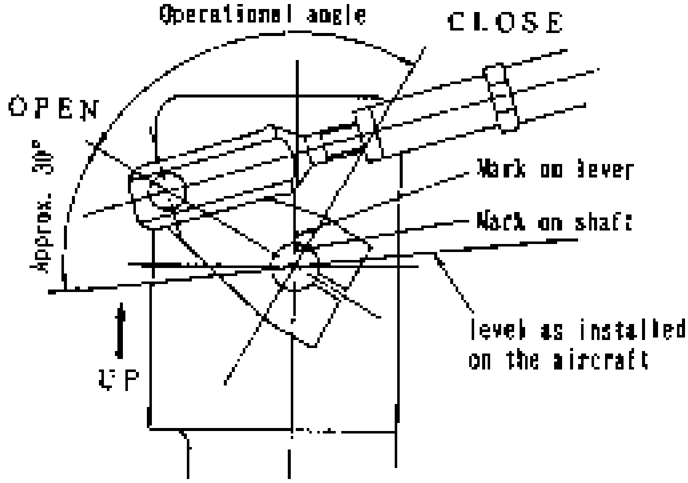


Fig. 6-22 Installation of lever

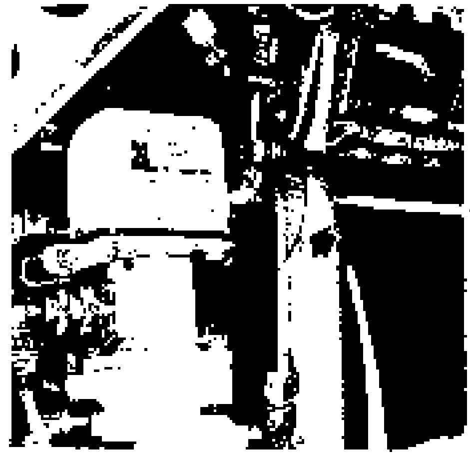


Fig. 6-23 Measurement of shutoff valve lever operational angle

(10) Move lever of fuel shutoff valve by hand to "CLOSE" position and measure the following items:

Operational angle of fuel shutoff valve	87° to 93°
Travel of feather valve	0.24 to 0.405 in. (4.5 to 10.3 mm)

When the above specified values are not met.

(a) Readjust length of rod ② connecting to feather valve ⑬.
(b) Move attaching place of pin ④ connecting bellcrank ③ and rod ② to change lever ratio and readjust.

(11) Set the lever ⑩ of fuel shutoff valve to "OPEN" position and ascertain that the rod ⑦ does not pull bellcrank ⑤.

(12) After adjustment, move lever of fuel shutoff valve by hand to "CLOSE" and "OPEN" positions and ascertain that the lever moves smoothly.

(13) Make sure that clearance between P3 Line ⑬ (P/N 31011415-1) and rod ⑦/pin ⑤ is more than 0.125 in. (3mm). If not satisfied, the P3 line and its attaching clamp/bracket should be installed at the position as shown and adjusted to make more than 0.125 in.(3 mm) clearance.

5.5 CONNECTION AND ADJUSTMENT OF ENGINE EMERGENCY STOP MECHANISM

(1) Assembly and adjustment of the linkage of emergency stop mechanism should be completed per Para. 5.4 before installation of engine on aircraft.

(2) Move lever of fuel shutoff valve by hand to "OPEN" position.

(3) Place condition lever of center pedestal to "TAXI" position, connect engine and aircraft structure with rod ⑰. (See Fig. 6-41)

(4) Move condition lever of center pedestal to the direction of "EMERG STOP" so that a rig pin is inserted into "EMERG STOP" position of cylinder ass'y.

(5) Place condition lever to "EMERG STOP" position and lock it. Fuel shutoff valve should be in "CLOSE" position and engine feather valve in "OPEN" position. Adjustment is made by changing the length of rod ⑰.

(6) Unlock condition lever and move to "TAXI" position. Fuel shutoff valve is in "OPEN" position and engine feather valve in "CLOSE" position. Make sure that clearance between pin and cam of cylinder ass'y is 0.04 to 0.08 in. (1 to 2 mm). Adjustment is made by moving connecting position of rod. (See Fig. 6-44)

(7) Move condition lever between "EMERG STOP" and "TAXI" positions and check for smooth operation without any interference, contact, deformation and abnormal noise.

NOTE

When condition lever is moved to "EMERG STOP" from "TAXI", adjust the lever so that there is no abnormal heaviness or obstruction in movement.

6. INSTALLATION AND REMOVAL OF ENGINE

Installation of engine should be performed as follows.

NOTE

When work is done near plenum chamber in engine nacelle, air intake leading to front of oil cooler from nacelle should be coverd with a tape to keep foreign object out.

6.1 CONNECTION OF ENGINE MOUNT

(1) Hang up engine with a sling (GSE 016A-99024).

(2) Install bolts (016A-13264) by setting engine front mount to mount fitting of witn leading edge. (See Fig. 6-24)
Tighten nuts handtight and apply cotter pin on bolts.

NOTE

Attaching bolts of engine front and rear mount should be inspected by magnetic particle inspection at every engine removal if the previous inspection has not been conducted within 1,500 hours in service.

(3) Install bolts by setting engine rear mount to attaching fitting on airplane. (See Fig. 6-25)
Tighten nuts handtight and apply cotter pin on bolts.

NOTE

Apply a light coat of anti-seize compound (MIL-G-6711) on attaching bolts of engine rear mount to prevent seizing.

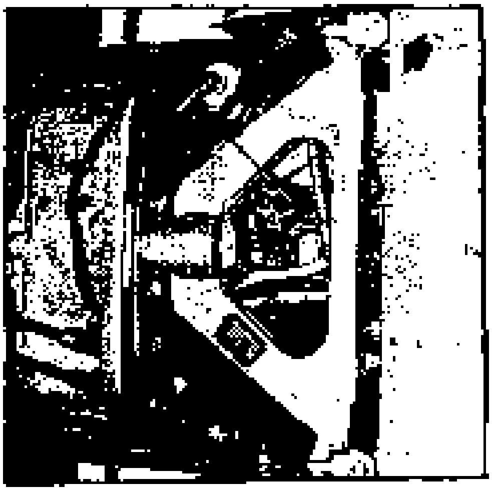


Fig. 6-24 Connection of engine front mount

Fig. 6-25 Connection of engine rear mount

6.2 CONNECTION OF ENGINE EMERGENCY STOP MECHANISM

See Para. 5.5 (1) ~ (7).

6.3 CONNECTION OF POWER CONTROL MECHANISM

See Para. 8.

6.4 CONNECTION OF CONDITION CONTROL MECHANISM

See Para. 9.2

6.5 WIRING CONNECTION OF STARTER GENERATOR

6.6 WIRING CONNECTION OF FIRE DETECTING SYSTEM (See Figs. 6-26 and 6-27)

(1) Connect plug of flexible wire for fire detecting in "ZONE I" to connector in wing leading edge. Install safety wire on plug.

(2) Connection of flexible wire plug for fire detecting in "ZONE II", located at fire wall of rear nacelle (N. STA1300), is performed after connection of engine air bleed line.

6.7 WIRING CONNECTION OF ELECTRICAL SYSTEM (See Figs. 6-28 and 6-29)

Connect electric plug J422 and J428 (R. H. engine) and J421 and J427 (L. H. engine) to connectors in wing leading edge. Install safety wire on each plug.

6.8 TUBING CONNECTION OF LUBRICATING SYSTEM

Install two hoses leading to oil cooler, in front of nacelle fire wall (N. STA790) (See Fig 6-30).

Fig. 6-26 Connection of fire detecting wire (forward)

Fig. 6-27 Connection of fire detecting wire (rear)

Fig. 6-28 Connection of J421 (J422) plug

Fig. 6-29 Connection of J427 (J428) plug

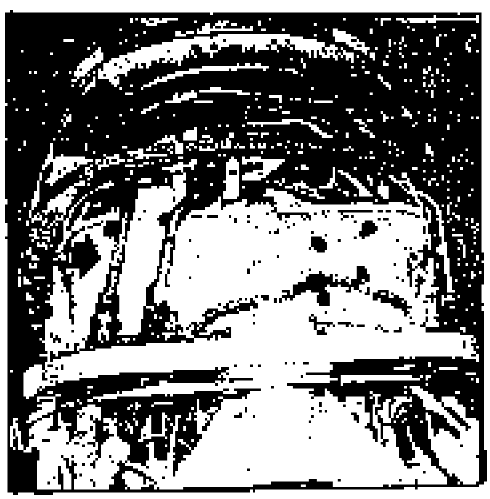


Fig. 6-30 Connection of oil hose

maintenance manual

TEMPORARY REVISION NO.6-1

This Temporary Revision No. 6-1 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	6-22

Insert facing the page indicated above for the applicable Maintenance Manual.
Retain this Temporary Revision until such time as a permanent revision on this subject is issued.

REASON : To change the inspection of the engine bleed air tube

<u>ADD</u> : Paragraph 6.11A as follows after paragraph 6.11 and the note.

6.11A Inspection of the engine bleed air tube 022A-964220-7

(1) Remove nut and screw from clip securing the engine bleed air tube.

(2) Disconnect the tube from the adapter assy and the tube assy.

(3) Perform a pressure check to the tube using typical shop method with shop air (if possible, use the air pressurized of 100 psig (7.1 kg/cm^2)).

(4) If no leaks are found in the line, it may be reinstalled. If any leaks are detected, the tube should be replaced with a new one.

6.9 TUBING CONNECTION OF FIRE EXTINGUISHING SYSTEM (If applicable)

Install a union connecting tubings from engine and fire extinguisher container located between nacelle fire wall (N. STA.790) and N. STA 1300. (See Fig. 6-31)

6.10 TUBING CONNECTION OF FUEL SYSTEM

Connect the fuel hose coming from fuel inlet union of fuel control unit to a union (or elbow) of fuel supply line in wing leading edge. (See Fig. 6-32)

Fig. 6-31 Connection of fire extinguisher line

Fig. 6-32 Connection of fuel hose

6.11 TUBING CONNECTION OF ENGINE BLEED AIR SYSTEM AND VACUUM SYSTEM (See Fig. 6-34)

Install two couplings for connection of air bleed duct in front of nacelle fire wall N. STA 1300. Tighten couplings to 90 to 110 in-lbs (105 to 125 kg-cm) torque for 24502-150 and 55 to 75 in-lbs (75 to 85 kg-cm) torque for 24502-125. Connect vacuum system tubing to a union at nacelle fire wall N. STA 1300.

NOTE

Gasket installed into coupling should be replaced during reinstallation.

6.12 TUBING CONNECTION OF ENGINE TORQUE SENSING SYSTEM (See Fig 6-33)

Connect oil hoses (2 ea.) of engine torque sensor to elbows in the wing leading edge.

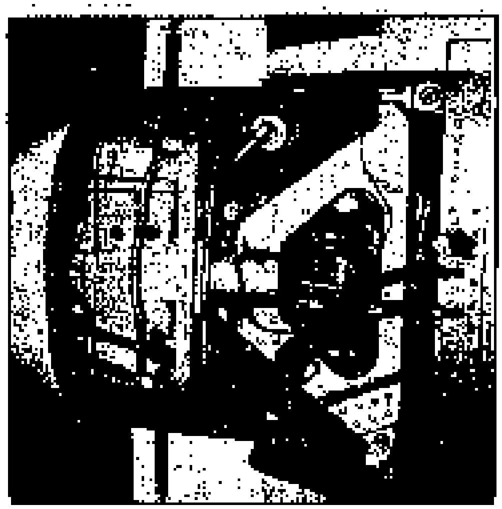


Fig. 6-33 Installation of oil hose

6.13 REMOVAL OF ENGINE

Remove in reverse sequence of installation.

7. RIGGING OF ENGINE CONTROL CABLE

(1) Insert a rig pin into pulley arm of pulley assembly under center pedestal and fix pulley and arm to bracket. (See Fig. 6-35)

(2) Insert rig pin into pulley of pulley assembly located at the back of pressure bulkhead. (F. STA8035) (See Fig. 6-36)

(3) Insert rig pin into bracket of pulley assembly located at upper part of rear nacelle (wing rear spar), and fix pulley. (See Fig. 6-37)

Fig. 6-34 Connection of air bleed line

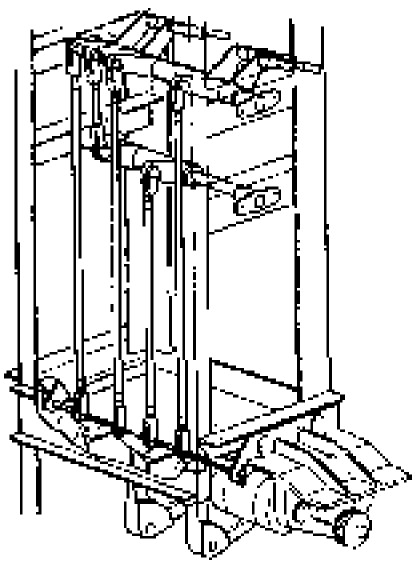


Fig. 6-35

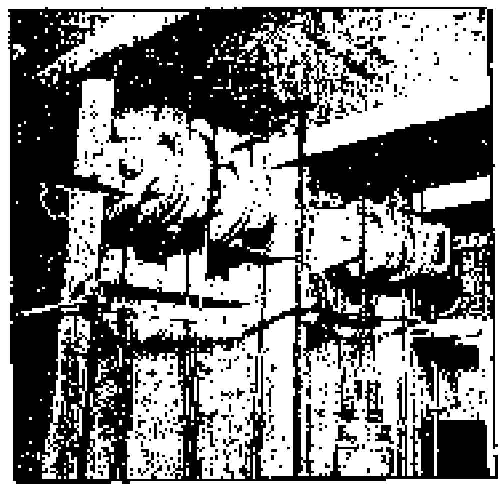


Fig. 6-36 Insert rig pin into pulley ass'y at STA 8035

(4) Adjust power control cable and condition control cable by turnbuckles so that the cable tensions become the specified values. For specified value, see Fig 6-38. Turnbuckles are located at the back of pressure bulkhead (F. STA8035) and rear access panel of center wing.

(5) After rigging cables, install safety wire on turnbuckles and pull out rig pins.

(6) Check control cable for specified clearance from neighboring structure and for specified alignment with pulley installation.

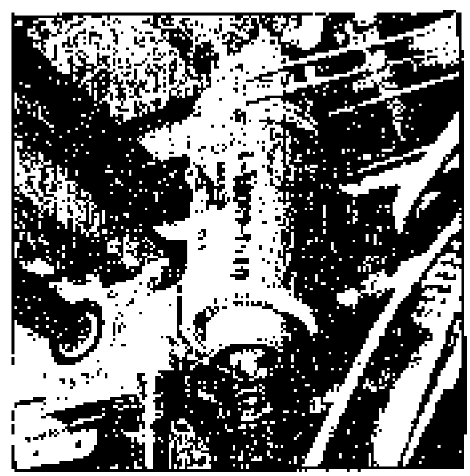


Fig. 6-37 Insert rig pin into pulley ass'y at aft nacelle

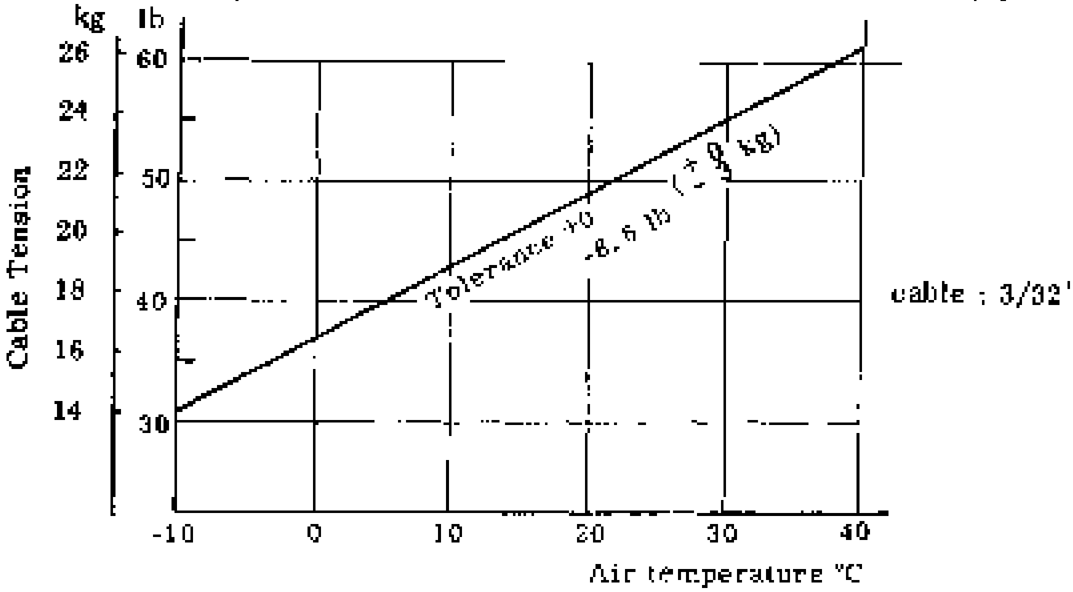


Fig. 6-38 Control cable tensions

Cable deflection to pulley	Max 2°
Clearance between cable and other parts	Cable near pulley must be more than 0.24 in. (6.1 mm) away from structures or installed equipment.
	Cable between pulley or fairleads should not contact with structures or equipments.
	Clearance between cable and electrical line must be more than 0.47 in(11.9 mm).
	Clearance between turnbuckle or cable fitting and structures or equipments must be more than 0.75 in.(19.1 mm).

8. CONNECTION AND ADJUSTMENT OF POWER CONTROL MECHANISM
(See Fig. 6-41)

8.1 CONNECTION

(1) Set power lever of center pedestal to "FLIGHT IDLE" position.
(2) Set lower lever of propeller pitch control to "FLIGHT IDLE" position on the rigging plate.
(3) Adjust rod ㉘ by turning turnbuckle ㉕ to adjust rod length, and connect the rod to upper lever of propeller pitch control. (See Fig. 6-40)

8.2 ADJUSTMENT

(1) Set power lever to "REVERSE" position.
(2) Insert rig pin into "REVERSE" position of propeller pitch control. When the rig pin can not be inserted, move adjusting serration retainer ③ to change lever ratio and readjust. To shorten one pitch of serration makes movement of power lever larger approx. 0.16 in. (4.1 mm) on the dial of center pedestal.
(3) When power levers of L.H. and R.H. engines are moved together, the difference of each stroke should not be more than 0.12 in. (3 mm) on the dial of center pedestal.

(4) When power lever of center pedestal is set to "TAKE-OFF" and "REVERSE" positions, adjust stops so that clearance of 0.08 to 0.16 in. (2 to 4.1 mm) is between lever and stops. (See Fig. 6-39)

When clearance of power lever is extremely different between "TAKEOFF" side and "REVERSE" side, readjust by moving lever attaching serration of propeller pitch control to divide the clearance equally.

Power lever	Check item
"FLIGHT IDLE" → "TAKEOFF"	0.08 to 0.16 in. (2 to 4.1 mm) clearance between lever and stop.
"FLIGHT IDLE" → "REVERSE"	0.08 to 0.16 in. (2 to 4.1 mm) clearance between lever and stop.

ORIGINAL
As Received By
ATP

Fig. 6-40 Power lever

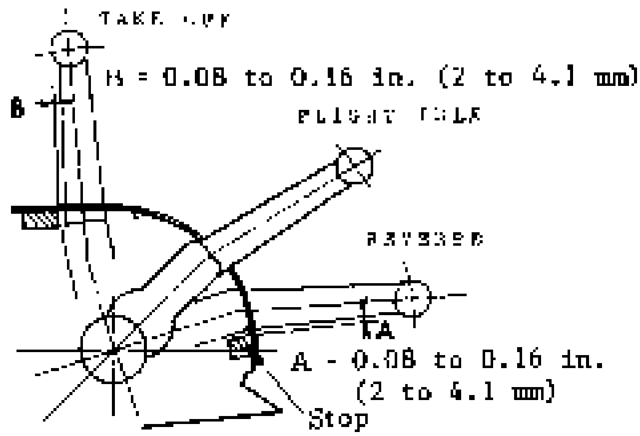


Fig. 6-39 Connect rod to pitch control arm (power control)

1. Rod
2. Lever
3. Cylinder ass'y
4. Rod (Ref. dimension 3.39 ± 0.02 in)
5. Bell crank
6. Rod
7. Cable drum
8. Retainer
9. Lever
10. Rig pin hole
11. Rod ass'y
12. Rod
13. Turnbuckle
14. Bell crank
15. Rod ass'y
16. Lever
17. Rod

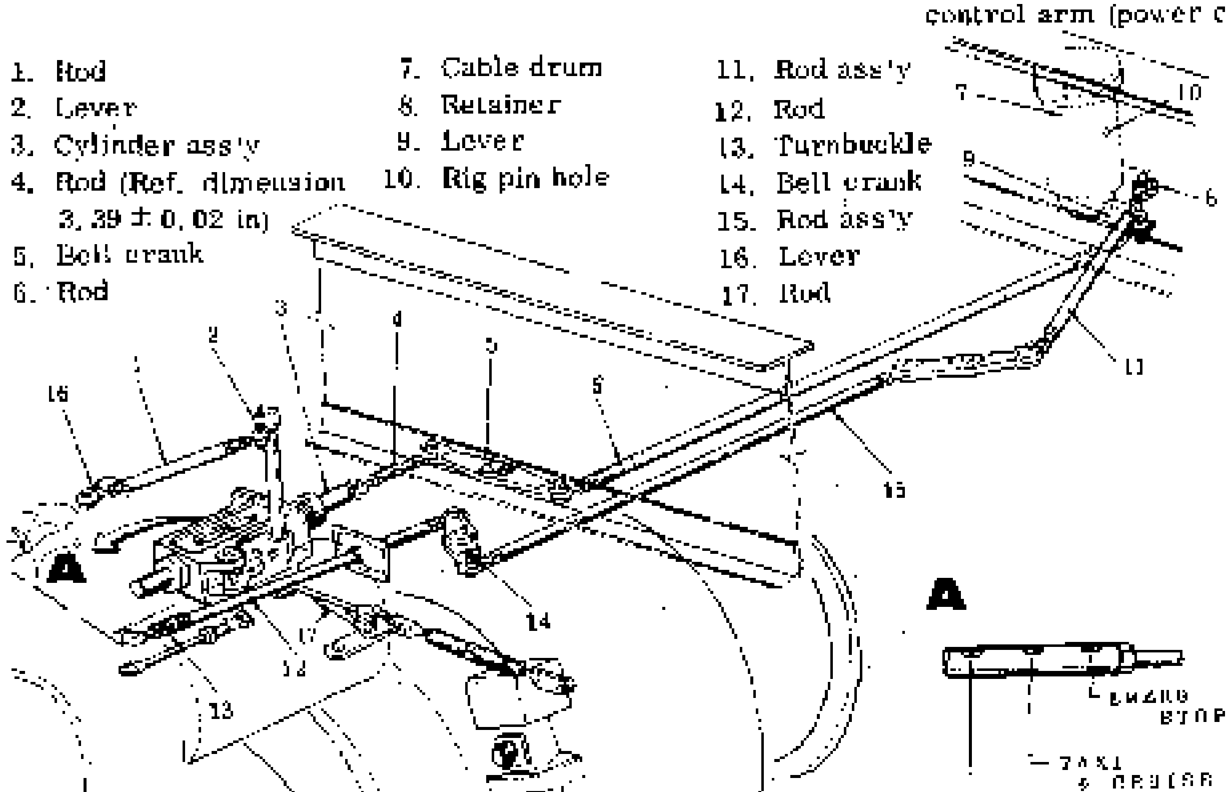


Fig. 6-41 Adjustment of engine control system

NOTE

Adjustment of stops should be performed after making sure that no difference of stroke is in L. H. and R. H. engine power levers.

8.3 CHECK AFTER ADJUSTMENT

(1) Move power levers of L. H. and R. H. engines from "TAKEOFF" to "FLIGHT IDLE", and make sure that landing gear warning switch becomes "ON" at position approx. 0.2 in. (5 mm) before reaching "FLIGHT IDLE".

(2) Make sure that rig pin can be inserted into "FLIGHT IDLE" position on the rigging plate of propeller pitch control when power lever is set to "FLIGHT IDLE" position.

(3) Make sure that a rig pin can be inserted into "REVERSE" position on the rigging plate of propeller pitch control when power lever is set to "REVERSE" position.

(4) Make sure that manual fuel valve lever is in contact with the stop properly when power lever is set to "TAKEOFF" position. If the lever is not in contact with the stop, repeat rigging of fuel control unit and propeller pitch control.

9. ADJUSTMENT OF CONDITION CONTROL MECHANISM (See Fig. 6-41)

9.1 ADJUSTMENT AND CHECK BEFORE ENGINE INSTALLATION

(1) Set condition lever of center pedestal to "TAXI" position and fix lever with friction lock.

(2) Set cylinder ass'y (3) to "TAXI" position and insert rig pin.

(3) Connect rod (4) between bellcrank (5) and cylinder ass'y (3). Do not change length of rod (4). Connect by adjusting length of rod (6). Connection of rod (6) and lever (9) should be done at the tip of long hole of lever.

(4) Pull out rig pin and loosen lock of condition lever.

(5) Set condition lever to "TAKEOFF-LAND" and "EMERG STOP" positions and make sure that rig pins are inserted into each rig pin hole on cylinder ass'y (3). If rig pins can not be inserted, move adjusting serration retainer at connection of rod (6) and lever (9) to change lever ratio and readjust. To shorten one pitch of serration makes stroke of cylinder ass'y (3) smaller approx. 0.05 in. (1.3 mm).

Fig. 6-42 Connect airframe and engine with rod (condition control)

9.2 CONNECTION AND ADJUSTMENT AFTER ENGINE INSTALLATION

9.2.1 CONNECTION (See Fig. 6-41)

(1) Connect rod (J) at engine end to one of two holes (inside one near to the shaft) on lever attached on underspeed governor shaft of fuel control unit.

(2) Set condition lever of center pedestal to TAKEOFF-LAND position and fix lever with friction lock.

(3) Fix one end of the rod (J) and put the lever of underspeed governor shaft to "MAX RPM" stop.

(4) Connect the other end of the rod (J) to long hole on lever together with the serration plate, while the lever of underspeed governor shaft is in "MAX RPM" stop. (See Fig. 6-42)

9.2.2 ADJUSTMENT

(1) Set condition lever of center pedestal to TAXI position and insert a rig pin into "TAXI" position of cylinder ass'y while the lever of underspeed governor shaft is in "MIN RPM" stop.
If rig pins can not be inserted, readjust by moving the serration plate at connection of rod (J) and lever. To lengthen one pitch of serration makes angle of lever smaller approx. 1°.

(2) Relation between cylinder ass'y (P) and cam must be in the condition shown in Fig 6-44, mechanism of pin and bellcrank.

(3) Pull out rig pin.

(4) Adjust the stop so that clearance of lever is 0.16 to 0.24 in. (4.1 to 6.1 mm) when condition lever is moved to TAKEOFF-LAND position. (See Fig. 6-43)

NOTE

Adjustment of stops should be performed after making sure that no difference of stroke is in L.H. and R.H. engine condition levers.

A = 0.16 to 0.24 in.
(4.1 to 6.1 mm)

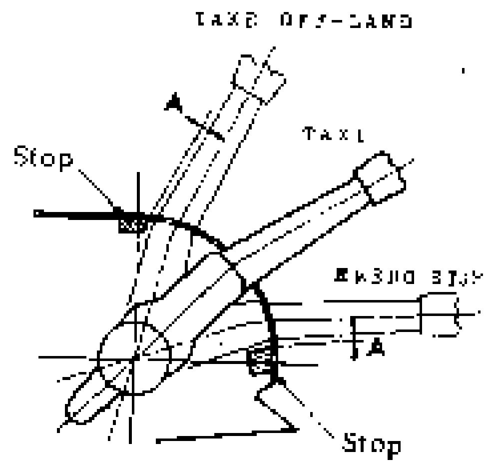


Fig. 6-43 Condition lever

9.2.3 CHECK

(1) Move condition lever of center pedestal to "TAXI" and "TAKEOFF-LAND" and check for smooth operation without interference, contact, deformation and abnormal noise.

(2) When condition lever of center pedestal is set to "TAKEOFF-LAND" position, lever of underspeed governor shaft should be in contact with "MAX RPM" stop.

(3) When condition lever of center pedestal is set to "TAXI" position, lever of underspeed governor shaft should be in contact with "MIN RPM" stop. At the same time, check that the lever of propeller governor is in contact with "LOW SPEED" stop. If the levers are not in contact with the stops, repeat rigging of fuel control unit and propeller governor.

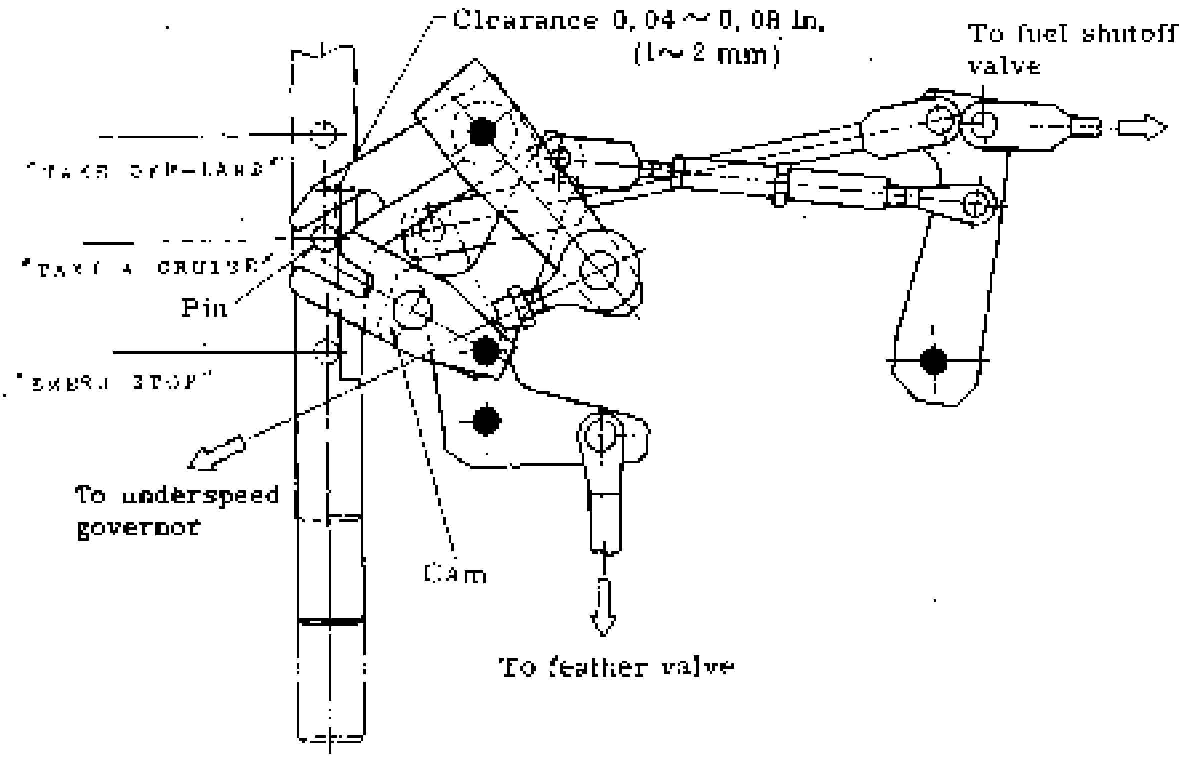


Fig. 6-44 Mechanism of pin and bellcrank

(4) Set condition lever to "EMERG STOP" position. Check lever operation for abnormal heaviness or any interference.

10. ADJUSTMENT OF FRICTION LOCK OF LEVER

(1) Set power lever to "TAKE OFF" position and condition lever to "TAKEOFF-LAND" position and lock the levers with friction lock.

(2) Adjust the position of friction bolt (016A-91529) hole by three plate attaching screws (AN507-832R5) and tighten the bolt so that the power and condition levers move when about 10 lbs (4.5 kg) of tension forces are applied on the top of levers in tangential directions.

(3) Release friction lock of each lever.

(4) Set power lever to the position other than "TAKEOFF" and condition lever other than "TAKE-OFF LAND". When about 0.5 lbs (0.2 kg) of tension force is applied on the top of each lever, the lever must move slightly.

(5) The force necessary to pull up each lever from "FLT IDLE" and "MIN CRUISE" stops should be 4 to 8 lbs (1.8 to 3.6 kg).

11. ENGINE OPERATING LIMITATION

Operating limitations of engine are as shown in the following table. Remedy shown in the table must be taken if the engine is operated exceeding the limitations.

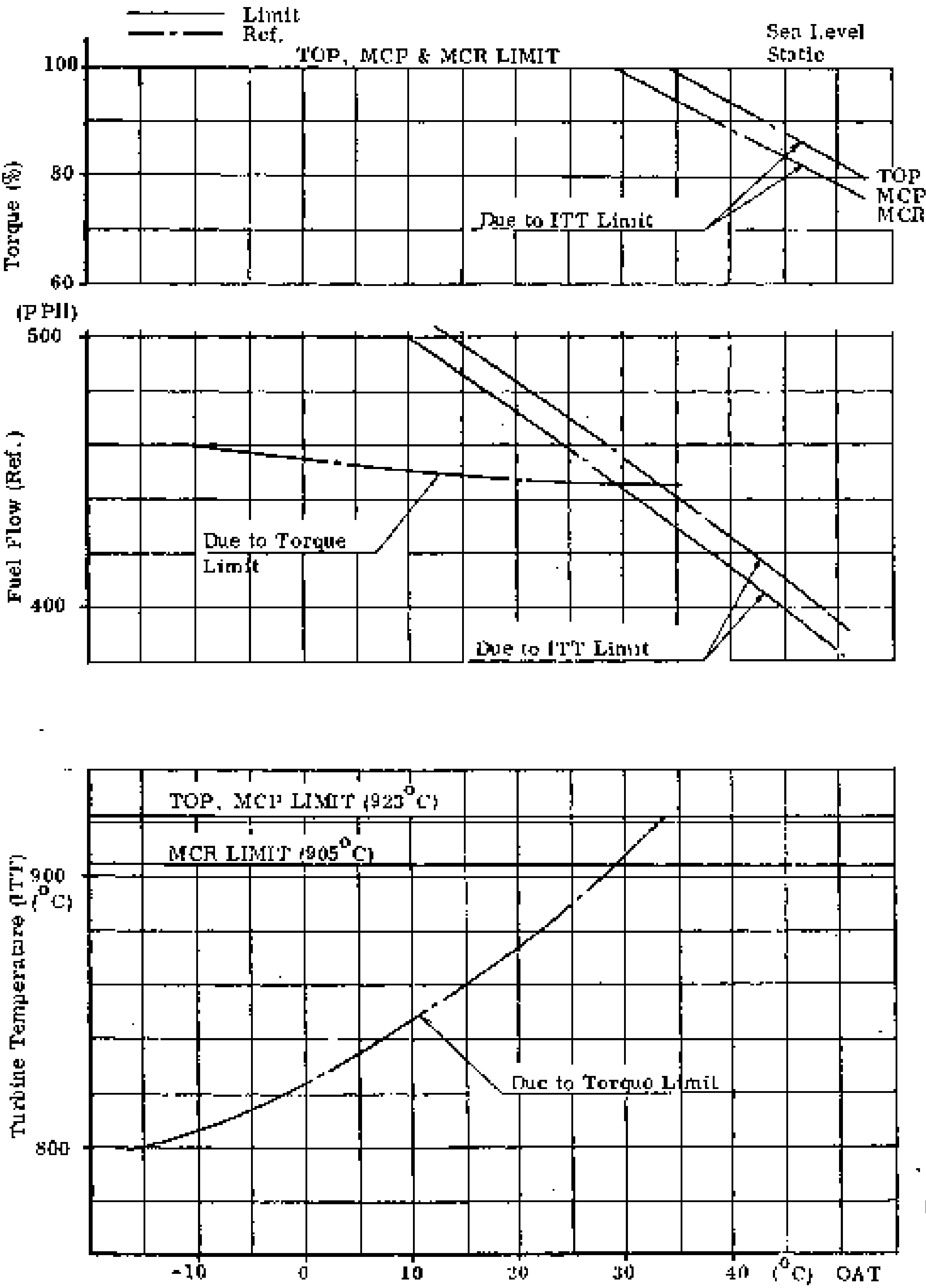
ITEM	LIMITATION	REMEDY
Light-off time (Time from 10% rpm to light off)		
(1) Ground start	10 seconds (max.)	Stop starting. Clear up the cause and correct. Crank engine 10 seconds before restarting.
(2) Air start	15 seconds (max.)	Stop starting. Clear up the cause and correct. Perform windmill 1 minute before restarting.
Oil pressure		
(1) 96 to 100% rpm	70 to 120 psi	If pressure is less than 70 psi or more than 120 psi, shut down engine immediately and correct.
(2) (*1) 64% to 66% rpm (*2) 76.5% to 78.5% rpm	More than 40 psi	If pressure is less than 40 psi, shut down engine immediately and correct.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

ITEM	LIMITATION	REMEDY
Ambient air temperature		
(1) Start	-40℃ to ISA+30℃	Do not start.
(2) Operating	-54℃ to ISA+30℃	If in flight, search for the place where the limitation is met.
Turbine temperature		
(1) During starts (1 sec. max.)	1149℃	Stop starting. Overhaul engine.
(2) Takeoff power	923℃	Reduce power. When temp. 10° to 20℃ is exceeded for more than 30 sec. or more than 20℃ is exceeded, record the max. temp. and exceeding time on engine log book and consult with AiResearch Field Service.
(3) Maximum continuous power	923℃	See Takeoff power limitation.
(4) Maximum cruise power	905℃	See Takeoff power limitation.
Engine speed		
(1) Min. RPM	64% (*1) 76.5% (*2)	Correct with condition lever.
(2) Continuous maximum	99.5 to 100.5%rpm	Correct with condition lever.
(3) 5 min. maximum	101% rpm	Correct with condition lever.
(4) Absolute max. (5 sec.)	106% rpm (*1) 105% rpm (*2)	If exceeding 5 seconds, overhaul engine.
Oil temperature		
Type II oil	Min. Start -40℃ Operating 55℃ Max. Ground run 127℃ Takeoff & Climbing 127℃ (5 min. max.) Other operation 110℃	Clear up the cause and correct.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

ITEM	LIMITATION	REMEDY
Torque		
(1) Takeoff power (5 min.)	Static condition 100%	Reduce power with power lever.
(2) Max. continuous power	100%	Same as above.
(3) Max. cruise power	100%	Same as above.
Fuel Pressure	15 to 90 psi	If pressure is less than 15 psi or more than 90 psi, shut down engine immediately and correct.


Limit
Ref.
Sea Level Static
TOP, MCP & MCR LIMIT
Torque (%)
100
80
60
TOP
MCP
MCR
Due to ITT Limit
(PPH)
500
400
Fuel Flow (Ref.)
Due to Torque Limit
Due to ITT Limit
TOP, MCP LIMIT (923°C)
MCR LIMIT (905°C)
Turbine Temperature (ITT) (°C)
900
800
Due to Torque Limit
-10
0
10
20
30
40
(°C) OAT

12. TROUBLE SHOOTING OF ENGINE

For trouble shooting of engine, see GARRETT Engine Maintenance Manual. This is provided as an aid in locating the most probable cause of a malfunction based on the symptoms it produces. Though primarily intended for use by maintenance personnel, it may aid a pilot or other operator of the engine in writing-up more informative and meaningful "flight squawks", thereby reducing the time required by maintenance personnel in locating and correcting the problem.

13. PROPELLER

13.1 GENERAL

Applicable to aircraft S/N 652SA

The propellers are Hartzell three-bladed propellers of the following series.

HC B 3TN 5 T1078 11 Series

(1) Type of hub includes HC-B3TN-5B, -5C and -5E.

(2) Type of blades for 90 inch (2286 mm) propeller is T1078HB-11.

Applicable to aircraft S/N 661SA, 697SA and subsequent

The propellers are Hartzell four-bladed propellers of the following series.

HC B 4TN 5DL/LT10282 B (or HB) 5.3R Series

(1) Type of hub includes HC-B4TN-5DL.
(2) Type of blades for 98 inch (2489 mm) propeller is LT 10282 B-53R (or HB-5.3R).

Model designation used above is explained in example below.

(a) Symbols for propeller hub (Shown for 4-blade propeller)

```
HC - B  4  T  N - 5  D  L
|    |  |  |  |   |  |  └ Propeller rotational direction (CCW)
|    |  |  |  |   |  └─── Minor modification
|    |  |  |  |   └────── Specific design features
|    |  |  |  └────────── Flange with 8-9/16 in. bolts on circumference
|    |  |  |                of 4 1/4 in. diameter
|    |  |  └───────────── Blade shank size
|    |  └──────────────── Number of blades
|    └─────────────────── Basic designation
└──────────────────────── Hartzell controllable pitch
```

(b) Symbols for propeller blade (Shown for 4-blade propeller)

T 102 82 H B - 5.3 R

- R — Blade tip is elliptic
- 5.3 — Diameter reduced 11 in.
- B — Anti-icing boots equipped
- H — Blade material is 7076-T61 (SYMBOL H)
  2025-T6 (SYMBOL None)
- 82 — Design basic number
- 102 — Basic diameter before reduction
- T — Needle bearing is installed in blade shank

13.2 GENERAL DESCRIPTION

The propeller is constant speed, feathering and reversible; designed for the GARRETT TPE-331 series engine. The total pitch range possible is from full feather to reverse, but actual pitch range allowed to the aircraft is about 92° as described later.

These propellers are designed to be controlled by a governor installed on the engine supplying engine oil through the propeller shaft. Governor oil pressure (0 to 400 psi) decreases pitch, including the reversing range; while counterweights attached to the blade clamps plus springs, located inside the cylinder, increase pitch to the feathered position.

The propeller has spring-loaded centrifugal responsive latches (low pitch stop) which prevent feathering when the propeller is stationary. This is a type that centrifugal force balances with spring force. When the propeller reaches high speed with condition lever in "TAXI" and power lever moving from "FLIGHT IDLE" to "REVERSE" side, the propeller latch is disengaged.

An electric unfeathering pump is also provided to set feathered blades to low-pitch.

13.3 REMOVAL OF PROPELLER

NOTE

Cover engine air intake so that screws, etc., may not enter into engine through air intake.

(1) Open nacelle upper door.

(2) Remove cowling upper section.

(3) Remove spinner dome.

(4) Remove electric connector for propeller de-icing brush block.

(5) Remove brush block from attaching bracket (see Fig 6-47). When removing, take care not to lose nuts and washers installed on the lower section of the bracket.
Place a piece of tape or rubber band around the brush block, thus retaining the brushes and avoiding their loss from the block.

(6) Remove brush block attaching bracket from engine.

(7) Set propeller in feathered position.

(a) Set power lever of center pedestal to "REVERSE" position.
(b) Push "PROP UNFEATHER" button, operate unfeather pump and move blades slightly to reverse side.
(c) Hold pin of high pitch stop. (See Fig. 6-45)
(d) Set power lever to "TAKEOFF" position and make "PROP UNFEATHER" button off.

(8) Remove oil transfer tube stop bolt on top of propeller and take out oil transfer tube, turning slowly to prevent damage. (See Fig. 6-46)

NOTE

Handle oil transfer tube with care to prevent damage or bend.

Fig. 6-45 Remove prop. high pitch stop and set prop. in feather position.

Fig. 6-46 Remove screw and nut from top of propeller

(9) Loosen propeller attaching bolts from the back of engine flange. (See Fig. 6-48)

(10) After hanging propeller with a sling (016A-99074 or equivalent), pull out 8 bolts and remove propeller from engine.

(11) Remove O-ring from engine flange.

Fig.6-47 Remove prop. de-icing brush assembly

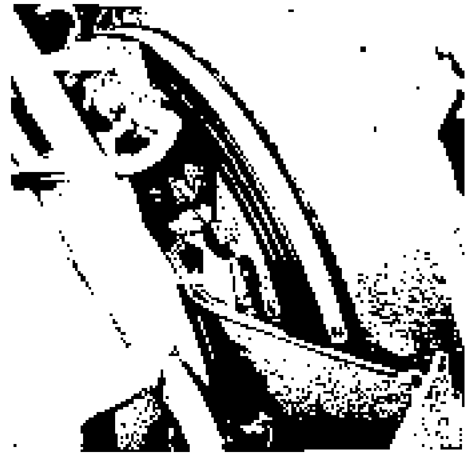


Fig.6-48 Remove eight bolts from prop. flange

13.4 INSTALLATION OF PROPELLER

(1) Open nacell upper door and make sure that propeller de-icing brush block is removed.

(2) Remove cover from propeller attaching flange, engine side and clean up the flange.

(3) Lubricate a new O-ring with engine oil and put it in the flange.

(4) Make sure that propeller assembly has passed balance test and functional test.

(5) Remove spinner dome.

(6) Make sure propeller is in the feathered position. If no feathered position, push up high pitch stop unit pin to set to feathered position.

(7) Hang propeller with a sling (016A-99074 or equivalent).

(8) Clean up propeller flange and check for damage.

(9) Install propeller so that dowel pin on engine flange may align with hole on propeller flange.

(10) Install eight each of propeller attaching bolts (A-2047 or B-3339) and new washers (A-2048-2) from the rear of the propeller mounting flange of engine. Install washers under boltheads as shown in Figure 6-48A.

NOTE

Replace with new propeller attaching bolts (A 2047 or B-3339) and new washers (A-2048-2) when replacing propellers.

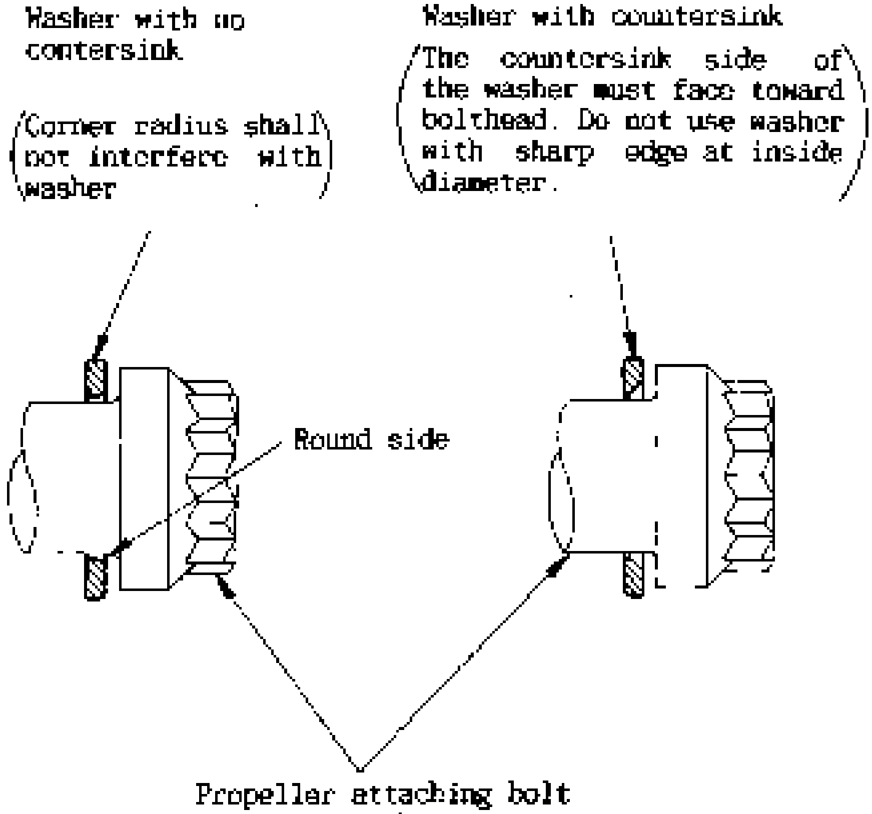


Fig.6-48A

a. When attaching bolt A-2047 is used. Torque to 125 ft-lbs.

NOTE

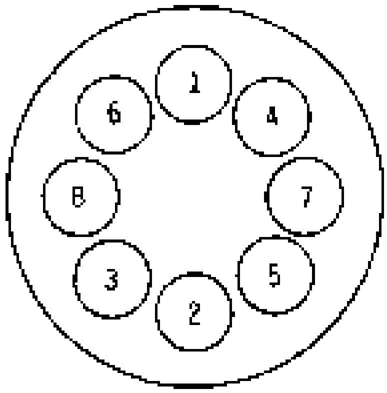
Apply torque in sequence as shown in Figures 6-48B and 6-48C.

b. When attaching bolt B-3339 is used.

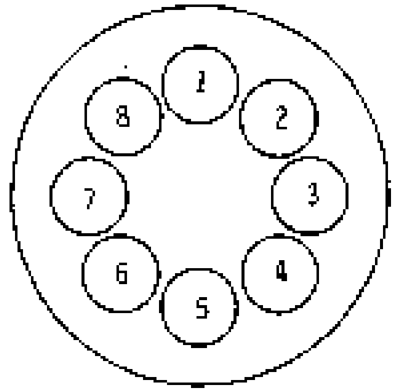
(a) Apply lubricant (petrolated graphite : MIL-T-5544 or TRW Hartzell lubricating oil : A-3338-1) on bolt thread and washer surface before installation.

(b) Ensure the mounting flange surfaces of propeller and engine are in close contact before tightening attaching bolts.

(c) Tighten bolts as follows.
   (i) Temporarily tighten eight bolts.
   (ii) Torque in the order as shown in Figure 6-48B to 40 ft-lbs, and then in the same order to 80 ft-lbs.
   (iii) Torque in the order as shown in Figure 6-48C to 100 thru 105 ft-lbs.



(Propeller mounting flange
looking from rear)
Fig.6-48B



(Propeller mounting flange
looking from rear)
Fig.6-48C

(11) Install safety wire on bolts.

(12) Set power lever of center pedestal to "FLIGHT IDLE" position.

(13) Lubricate a new O-ring (MS29561-112) with engine oil and put in oil transfer tube.

(14) Insert oil transfer tube, turning it slowly from the top of propeller. (See Fig.6-49)
Handle with care not to damage propeller servo with the tip of the tube.

(15) Screw in the oil transfer tube until tube forward edge coincides with the hub tip.

(16) Insert oil transfer tube stop bolt and install nut.

(17) Install propeller de-icing brush block attaching bracket on stud at the front of engine output housing. Install washers under the bracket so that brush may be parallel with slip ring.

(18) Install brush block on the bracket. If necessary, insert shims between bracket and brush block to adjust contact between slip ring and brush.

For relative location of slip ring and brush, see Chapter IX Para. 6. 3.

When installing, take care not to lose nuts and washers installed on the lower section of the bracket.

Fig. 6-49 Insert oil transfer tube into prop. dome

(19) Connect electric connector to brush block.

(20) After completion of adjustment of propeller pitch angle in Para. 13.5, install spinner dome with 15 screws, and lock spinner.

(21) Install upper section of cowling with 13 screws.

13.5 PITCH ADJUSTMENT

13.5.1 ADJUSTMENT OF FLIGHT IDLE PITCH

After installing propeller on engine, adjust in accordance with the following procedures.

(1) Set power lever to "FLIGHT IDLE" position. Make sure that fuel control underspeed governor and propeller pitch control are in "FLIGHT IDLE" positions. When power lever is moved to "FLIGHT IDLE", it should be done by pulling lever from "TAKEOFF" side to "FLIGHT IDLE".

(2) Operating unfeathering pump, decrease propeller blade pitch from feather position.

(3) With unfeathering pump in operation, measure propeller pitch angles at which propeller blade has stopped and confirm that angles are within the specified limits.

Limits: Flight idle pitch angle: $12^\circ \pm 0.1^\circ$

Angle is measured at 30 in. (762 mm) of propeller radius.

NOTE

Before operating unfeathering pump, check oil tank for proper oil level. When oil is insufficient, unfeathering pump races and propeller blade proceeds to feathering.

(4) When propeller pitch angle is not within limits, turn adjusting screw of oil transfer tube on the top of propeller dome with a driver. (See Fig 6-50)

Clockwise (C. W.)	Pitch reduce
Counterclockwise (C. C. W.)	Pitch increase

1/6 turn corresponds to approx. 0.4° of pitch angle

(5) After adjustment is made, attach bolt and nut to adjusting thread of oil transfer tube. Check power lever for smooth movement through its entire range.

13.5.2 ADJUSTMENT OF HIGH PITCH STOP (See Fig. 6-52)

(1) With propeller blade locked by high pitch stop, measure propeller pitch.
$2.5^\circ \pm 0.2^\circ$
The angle is measured at 30 inch (762 mm) of propeller radius.

(2) When the high pitch stop angle is not within the limits, adjust adjusting screw on high pitch stop of propeller blade to change the angle. (See Fig. 6-51)

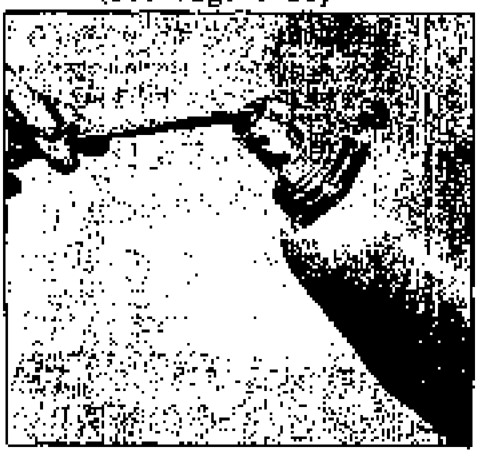


Fig. 6-50 Adjustment of flight idle angle

Fig. 6-51 Adjustment of high pitch stop

13.6 MEASUREMENT OF PROPELLER PITCH ANGLE (See Fig. 52)

13.6.1 GENERAL PROCEDURE

(1) Remove spinner dome from propeller.

(2) Place a protractor on straight level section of piston and set to zero on the basis of horizontal and vertical lines of the protractor level.

(3) Set cockpit power lever to "FLIGHT IDLE".

(4) Place blade in level position.

(5) Operate unfeathering pump. When propeller blade stops in "FLIGHT IDLE" position, set protractor at 30 in. (762 mm) propeller radius, perpendicular to blade surface, with unfeathering pump in operation.

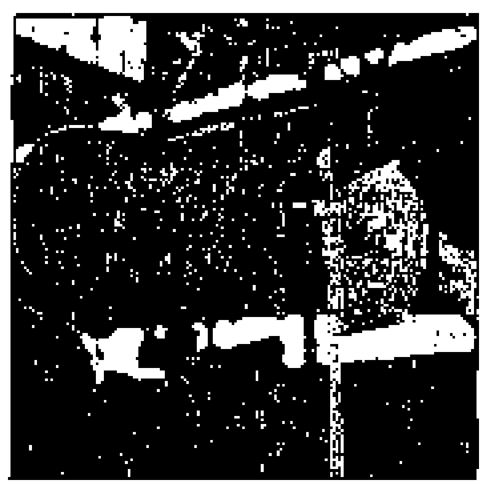


Fig. 6-52 Measurement of prop. pitch angle

The protractor reading indicates "FLIGHT IDLE" pitch angle. "REVERSE", "FEATHER" and "START" pitch angles can be measured in the same way.

13.6.2 MEASUREMENT OF PITCH WITH REFERENCE TO FEATHERING PITCH ANGLE

(1) Set propeller in "FEATHER" position. Place protractor at 30 in. (762mm) propeller radius, perpendicular to blade surface and set to zero on the basis of horizontal and vertical lines of the protractor level.

(2) When blade stops in "FLIGHT IDLE" position, note the protractor reading "FLIGHT IDLE" pitch angle is obtained by deducting the protractor reading from feather pitch angle.

13.7 REPAIRING CRITERIA OF BLADES

13.7.1 SCOPE OF REPAIR

Nicks located within a quarter of length of blades from the center shall not be repaired, but the blade shall be replaced.

13.7.2 REPAIRING PROCEDURES

(1) Nicks on leading edge or trailing edge of blades (See Fig. 6-53)

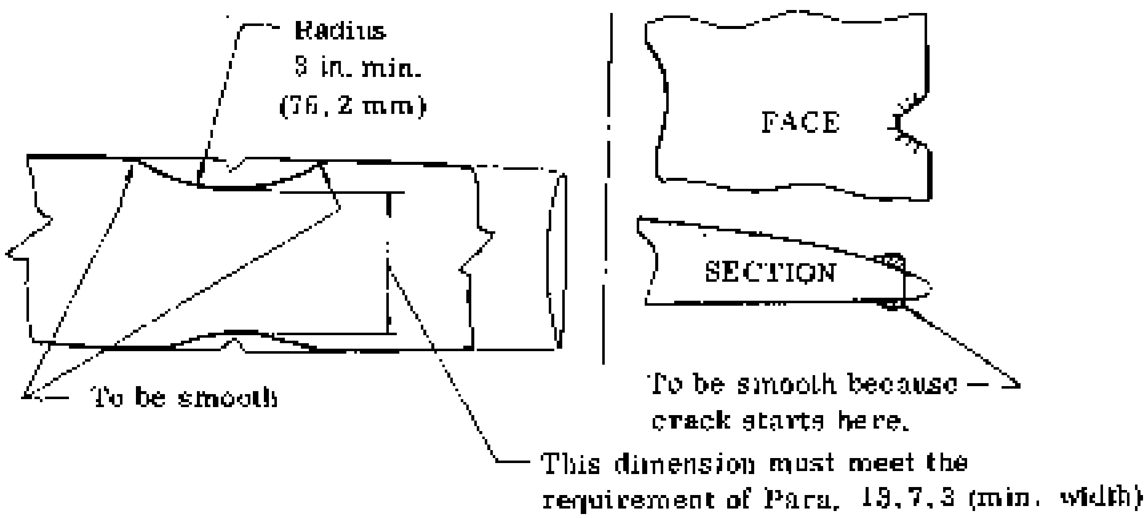
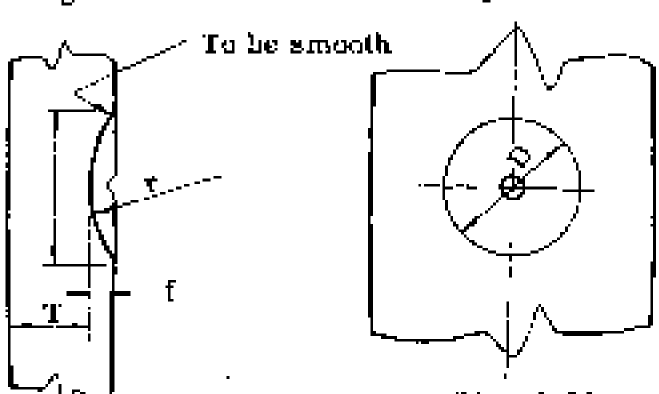


Fig. 6-53

(2) Damages on blade surface (See Fig. 6-54)



r >= 5 f
D >= 10 f
T : This dimension must meet the requirement of Para. 13.7.3. (min. thickness)

Fig. 6-54

13.7.3 REPAIR TOLERANCE

Repaired blade shall satisfy the following tolerances:

RADIUS	PITCH ANGLE (degree) MIN	MAX	* FACE ALIGNMENT MIN	MAX	MIN WIDTH	MIN THICKNESS
8					3.830	2.913
12					5.165	1.541
18	43.1	44.1	0.319	0.381	*1 6.430 *4 6.435	0.942
24	35.3	35.8	0.089	0.151	*1 7.080 *4 7.090	*1 0.655 *4 0.657
30	30.0	Set Up	-0.081	-0.019	*1 7.315 *4 7.345	*1 0.505 *4 0.508
36	*2 25.6 *3 27.2 *4 26.4	*2 25.6 *3 27.2 *4 24.4	*1 -0.231 *4 -0.261	*1 0.169 *4 0.199	*2 7.250 *3 7.000 *4 7.400	*1 0.364 *4 0.367
42	*2 20.9 *3 24.7 *4 23.1	*2 21.3 *3 25.1 *4 23.5	*1 -0.351 *4 -0.431	*1 -0.289 *4 -0.369	*2 6.800 *3 5.200 *4 7.130	*2 0.241 *3 0.153 *4 0.253
48 *4	*4 20.9	*4 21.3	*4 -0.571	*4 -0.509	*4 6.250	*4 0.153

* Face Alignment : Distance, measured perpendicular to the flat face (rear face), from the flat face to the center line of the shank axis.

*1 Applicable to T10178 11 & -11R Blades

*2 Applicable to T10178 11 Blades

*3 Applicable to T10178 11R Blades

*4 Applicable to T10282 Blades

ORIGINAL AS RECEIVED BY ATP

13.7.4 PROCESSING AFTER REPAIR

If a small area of blade surface is repaired, the area shall be touched up with chemical film and refinished with a coating of the following paint.

Neogosei #200 (Shinto Toryo KK)
or equivalent epoxy resin paint

13.8 TROUBLE SHOOTING FOR PROPELLER

Trouble	Probable Cause	Remedy
RPM fails to increase normally at starting or will reduce during running	Start lock blade angle too high.	Adjust start lock.
RPM change in both directions is sluggish	Excessive friction in hub mechanism.	See "Excessive friction in hub mechanism".
Excessive friction in hub mechanism	a. Pilot tube has slipped out slightly and is rubbing hard against end of hole in blade.	Push down pilot tube and measure height from top of hub spider to top end of pilot tube. Should be 3 3/4 in. (95.3 mm).
	b. Damage to split bearing.	Replace bearing.
	c. Excessive friction of piston assembly.	Check parts individually. If tightness is encountered in sliding parts, increase clearances slightly or apply oil externally to moving parts.

Trouble	Probable Cause	Remedy
RPM is not adequate	a. Inadequate adjustment of propeller governor.	See "Excessive friction in hub mechanism" in this chart.
	b. Inadequate adjustment of underspeed governor.	See "Excessive friction in hub mechanism."
Failure to feather	a. Governor control does not provide enough travel to allow governor to move from feather to high pitch stop.	Provide sufficient travel in governor control. Governor must drain oil out of propeller during feathering.
	b. Excessive friction.	See "Excessive friction in hub mechanism" in this chart.
	c. High pitch stop pin fails to slide out at feathering rpm.	Check for burrs on pins. Check for stiffness of spring. If propeller feathers at high rpm but fails to feather at low rpm, too-stiff spring is indicated. Reduce length of spring by one or two coils.
	d. Weak or broken feathering spring.	Replace spring.
Surging (When governor control is changed rapidly or rough air is encountered.)	a. Air trapped in propeller actuating piston or in engine shaft.	Provision should be incorporated in engine to allow trapped air to escape from system during one half of the pitch change cycle. Exercise propeller by changing pitch or feathering before each flight.
	b. Governor pressure too low.	Adjust governor relief pressure so that rate of pitch change is the same in both directions.
	c. Excessive friction.	See "Excessive friction in hub mechanism".
	d. Governor lacks sufficient dampening.	Change or adjust speeder spring. This gives governor more stability, but makes pitch control more sensitive, which may in turn require re-rigging or control to offset higher spring rate. Changing to stiffer speeder spring should be done only after all other factors have been checked and corrected.

Trouble	Probable Cause	Remedy
Oil leakage	Faulty O-ring seals as follows: a. Propeller shaft b. Cylinder c. Piston d. Pilot tube	Disassemble and inspect O-rings and the surfaces they seal. Replace O-rings if defective. Replace cylinder if surface is scratched or nicked in area where O-ring slides. Use gasket compound on cylinder threads when replacing. Peen down raised places caused by tightening of cylinder with bar.
Grease leakage (Only source of grease leakage is blade bearings.)	a. Grease leaks past clamp seal gaskets.	Loosen clamp bolts and replace gaskets. Standard thickness of gasket is 0.05 in., but 0.06 in. gaskets are available in case 0.05 in. gaskets are not compressed sufficiently to hold.
	b. Grease leaks from between blade and clamp.	Inscribe blade with ink or pencil to assure correct blade angle on reassembly, and remove blade and clamp. Add gasket compound in radius of blade butt. Replace blade and clamp.

13.9 DISASSEMBLY OF PROPELLER

(1) Remove the spinner dome. Mount the propeller on a table (016A-99034 or equivalent) with 4 bolts.

(2) Identify hub spider and propeller blades by number with grease pencil for ease of pitch angle setting.

(3) Remove lower safety screw of link arm from piston unit.

(4) Withdraw link pin unit and disconnect link arm from piston unit.

(5) Remove hub clamps that hold the de-icer boots lead strap.

(6) Remove outer blade clamp bolt and nut, and inner blade clamp socket screw.

(7) Remove blade clamp and gasket.

(8) Remove blades from hub spider.

(9) Remove bearing retention ring and split bearing ball and ball spacer.

(10) Remove "O" ring.

(11) Remove flexlock nut.

(12) Remove "O" ring and dust seal.

The following procedures shall be performed only if necessary.

(13) Remove mushroom together with spring assemblies from cylinder.

(14) Remove cylinder from hub spider.

(15) Remove "O" ring.

13.10 CLEANING

(1) Clean parts disassembled in accordance with Para. 13.9 with solvent P-D-680 Type I or equivalent.

NOTE

Do not clean dust seal B-1843 and de-icer boots and lead strap.

(2) Clean grease on hub spider with solvent.

(3) Clean grease in pilot tube of blade with solvent.

NOTE

Do not clean de-icer boots and lead strap.

13.11 INSPECTION

(1) Inspect blade clamps and hub spiders for cracks and corrosion.

(2) Visually inspect the blades for cracks, particularly around the shank retention area and along the leading edge near the tip.

(3) Electrical resistance in de-icer boots line shall conform to the provision of Para. 5.4.1, Chapter IX.

(4) Inspect cylinder surface for damage that may cause leakage.

(5) Check parts removed in accordance with Para. 13.9 for dimensions as set forth in Para. 13.12.

(6) Check ball spacer, split bearing and the balls for wear.

(7) Inspect hub pilot tube for slippage. The tube should extend out from hub 3 3/4 ± 1/16 inch only. If this distance is greater, the indication is that the tube has slipped out of the hub spider. An oversize tube should replace the one which slipped.

(8) Check dimension "C", Fig. 6-57.

(9) Temporarily attach spinner dome to bulkhead. Check dimension "A" in Fig. 6-57.

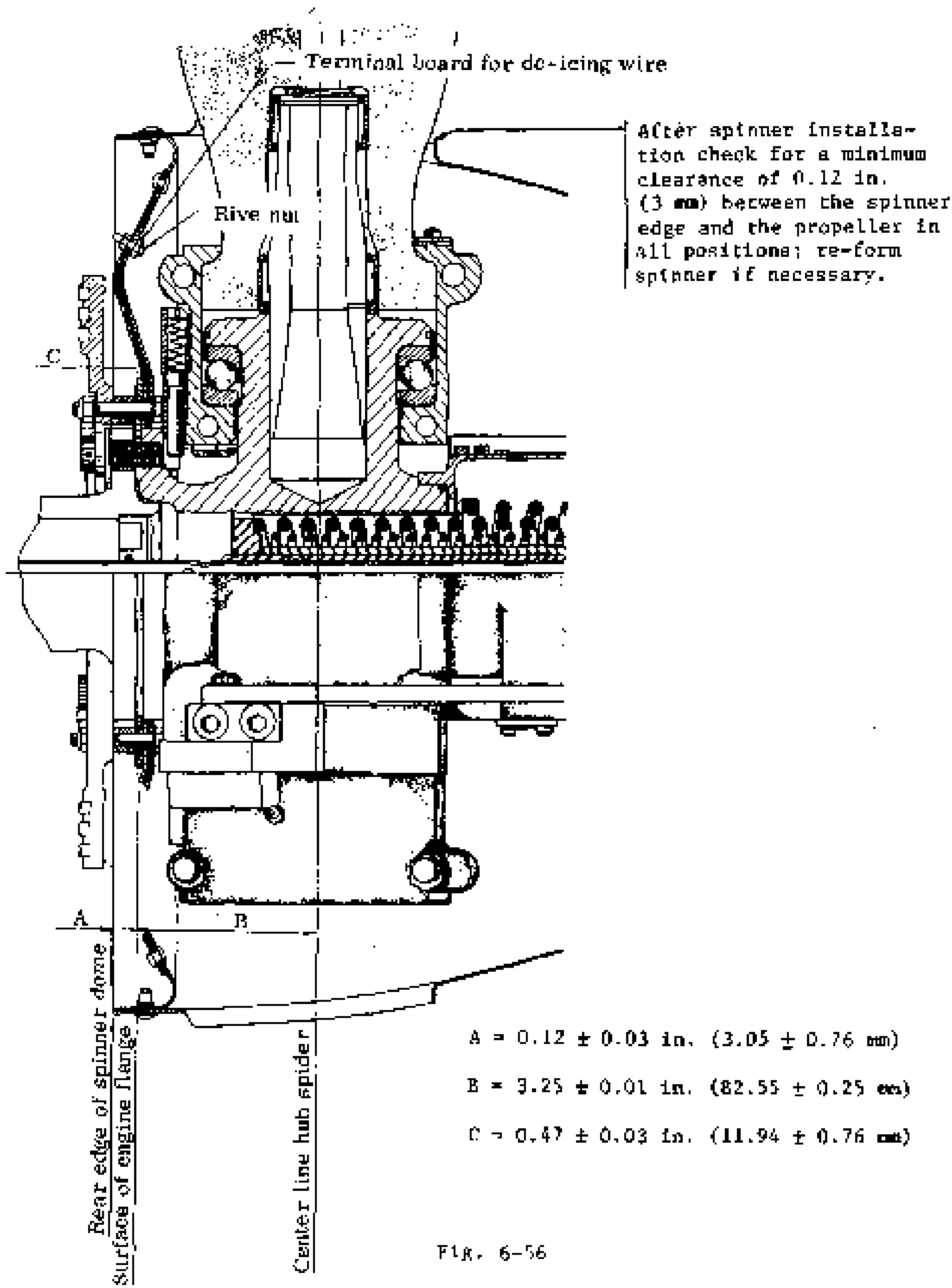


Fig. 6-56

13.12 TOLERANCE

For overhaul tolerances, see Hartzell Manual No. 118.

13.13 ASSEMBLY OF PROPELLER

13.13.1 PREPARATIONS

(1) Mount "O" ring on table (016A-99034 or equivalent).

(2) Install high pitch stop and slip ring to the bulkhead and place assembled bulkhead on the table.
High pitch stop shall be temporarily attached for adjustment after installation.

(3) Mount hub spider unit onto table with 4 bolts.

13.13.2 CYLINDER INSTALLATION

(1) Clean the threads of the hub and cylinder.
Inspect the inside of the cylinder for foreign materials.

(2) Apply oil (MIL-L-23699) to the cylinder "O" ring. Install "O" ring in the cylinder (B-1803-2) behind the threads.

(3) Apply sealing compound to the threads of the hub only.

(4) Screw the cylinder onto the hub. Tighten the cylinder hard against the hub (about 150 to 200 ft-lbs. (20.7 to 27.6 kg-m) torque).

(5) Inspect the inside of the cylinder to be sure the O-ring has not been forced out of place, and that the sealing compound is not present to contaminate the engine oil.

(6) Inspect around the outside of cylinder to be sure that there are not any sharp edges which might cut the piston "O" ring.

13.13.3 FEATHERING SPRING SUB-ASSEMBLY

(1) Spring assembly as assembled shall consist of mushroom, 3 feathering springs and 4 feathering stop screws, etc. Feathering stop screw shall be applied with safety wire.

(2) Clean threads of cylinder.

(3) Coat the threads of mushroom with grease (MIL-G-23827).

(4) Check inside of mushroom and clearance in spring for foreign materials.

(5) Slide the spring sub-assembly into cylinder and screw it in place.
place.

(6) Apply safety wire on mushroom and cylinder, inserting the wire through a drilled hole in the flange of cup.

13.13.4 INSTALLATION OF BLADE

(1) Fill the blade pilot tube hole and pilot tube with grease (MIL-G-23827).
Be sure air is not trapped under the grease.

(2) Slip "O" ring over flange of hub spider arms.

(3) Install ball spacer and ball split bearing in hub spider with grease. Mount bearing retention rings. Put the bearing parting line parallel with the hub shaft axis.

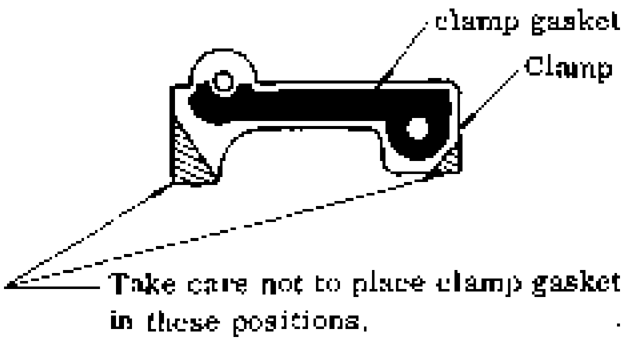
(4) Fill split bearing with grease.

(5) Install blade on hub pilot tube. Spread sealing compound around the blade shank in the shoulder radius.

Do not change original combination of blades and hub spider.

(6) Install the correct matching clamp. Slip the clamp gaskets between the clamps at the parting line.
This procedure will locate the inner bearing race parting line at right angles with the blade clamp parting line.
The installation shall meet the requirement in Para. 13.14 (6).

NOTE



(7) Tighten the outer blade clamp bolts and nuts, and inner blade clamp socket screw.

Outer clamp bolts : 60～65 ft-lbs (8.3～9.0 kg-m)
Inner clamp socket screw : 40 ft-lbs, (5.5 kg-m)

(8) *1 Hold de-icer boots lead strap in position with hub clamp. Fasten with screw.

*2 Secure wire leads with tie straps (MS18034-6).

(9) Connect de-icer boots wire harness to terminal strip. Support with lead clip.

(10) Inspect blade for smooth rotation by turning blades by hand. Check bearing for friction.

13.13.5 PISTON ASSEMBLY

(1) Install O-ring in the second groove and dust seal in the first groove of the piston unit. Soak seal and O-ring with engine oil (MIL-L-23699).

(2) Carefully inspect inside of cylinder for foreign materials. Slide the piston onto the cylinder.

(3) Install the link arms in the piston. Insert all link pin units.

(4) Install safety screw and safety wire.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

(5) Soak "O" ring in engine oil (MIL-L-23699). Install "O" ring in the cylinder.

(6) Install flexlock nut on end of pilot tube. Torque to 120 ft-lbs. (16.6 kg-m).

13.14 BLADE SETTING

(1) Blade end play in peripheral direction shall not exceed 1/16 in. (1.6 mm).

(2) Fore and aft blade movement at tip shall not exceed 0.1 in. (2.5 mm).

(3) Blade track of blades shall not exceed ± 1/16 in. (± 1.6 mm) when measured at 4 in. (101.6 mm) point from tip with blades in flat pitch position.

(4) No blade slip shall occur between clamp and blade when blade is torqued to 167 ft-lbs. (23 kg-m) with respect to pitch control axis.

(5) Counterweight angle should be located within 2° beyond the propeller axis when in reverse pitch. (See Fig. 6-57)

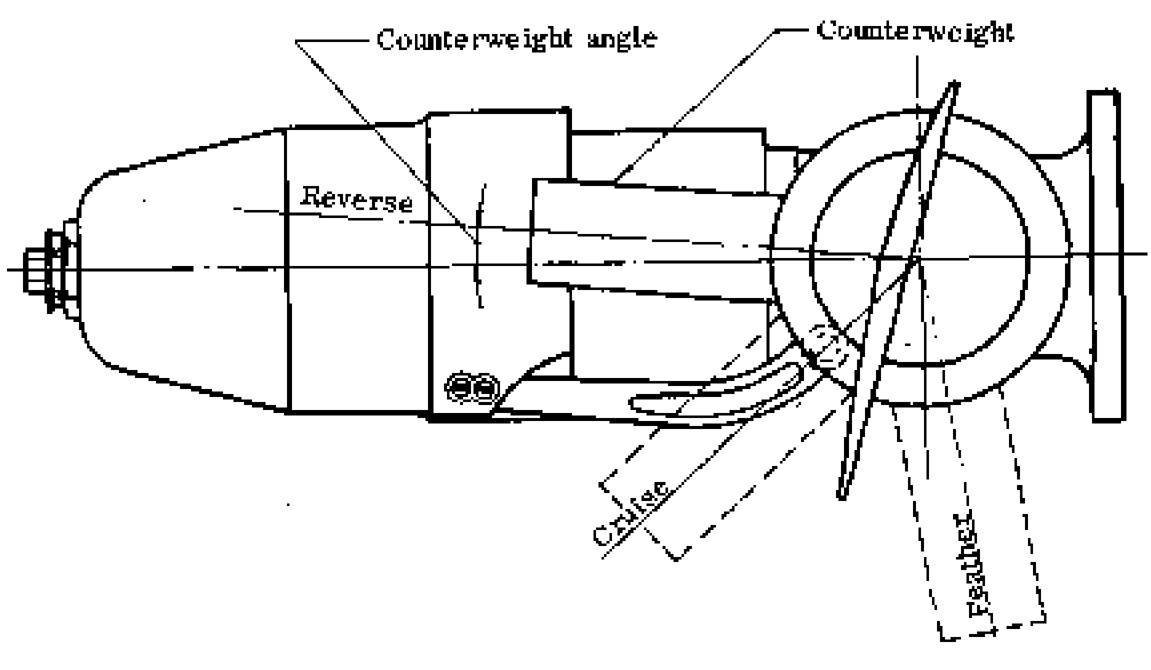


Fig. 6-57

13.15 PROPELLER BALANCING PROCEDURE

13.15.1 GENERAL DESCRIPTION

Dynamic balancing of the propeller should be performed to prevent possible engine and prop hub damage due to a heavy blade. An unbalanced propeller may cause a harmonic vibration resulting in engine component damage and flight and passenger compartment noise and discomfort. Propeller balancing should be performed when deemed necessary; after routine prop maintenance, overhaul, propeller replacement of flight squawk by the pilot.

13.15.2 SPECIAL TOOLS AND/OR EQUIPMENT

Items 1 through 2 are manufactured by Spectral Dynamics Corp. of San Diego; items 4 and 5 are manufactured by IRD Mechanalysis, Inc., Columbus, Ohio; and items 3 and 6 are manufactured by Mitsubishi Aircraft Int'l.

	MODEL NO.	ITEM NOMENCLATURE
1.	DS119B	Trim Balance Analyzer
2.	M 80	Velocity transducer
3.	MTS 700A or equivalent*	Mounting Bracket
OR		
4.	330	Vibration Analysis/Dynamic Balancing
5.	544 T-321-1	Vibration Pickup
6.	MTS 700 or equivalent*	Mount Bracket

*MFG by MAI

13.15.3 PRELIMINARY

Check to be sure the following have been complied with before balancing propeller.

1. Prop has been properly greased.
2. Aircraft has flown or run at least thirty minutes since lubrication.
3. Start locks are set at $2.5 \pm 0.1^\circ$.
4. Flight idle blade angles are $12.0 \pm 0.1^\circ$
5. Propeller spinner properly installed and clocked.

Any discrepancy in the above shall be corrected prior to balancing the propeller(s).

13.15.4 SET UP

A. General

(1) Gain access to top engine gear case by raising upper engine nacelle door and securing support rod in open position.

NOTE

Secure or tighten door latches to avoid loss during engine run.

(2) Remove clamp securing prop de-ice and synchrophaser wiring harness from bossed area on engine gear case and move to one side.

(3) Install mounting bracket to bossed area of gear case, Fig. 6-58, and install velocity transducer (M80) or vibration pickup (544 T-321-1) on mounting bracket.

(a) Move and/or remove the following as required:

(1) Disconnect fuel pressure transmitter electrical connector.
(2) Disconnect pressure line between fuel control assembly and fuel pressure transmitter at transmitter.
(3) Remove safety wire and 4 bolts attaching fuel pressure transmitter assembly; remove transmitter from aircraft.
(4) Plug pressure line and port.

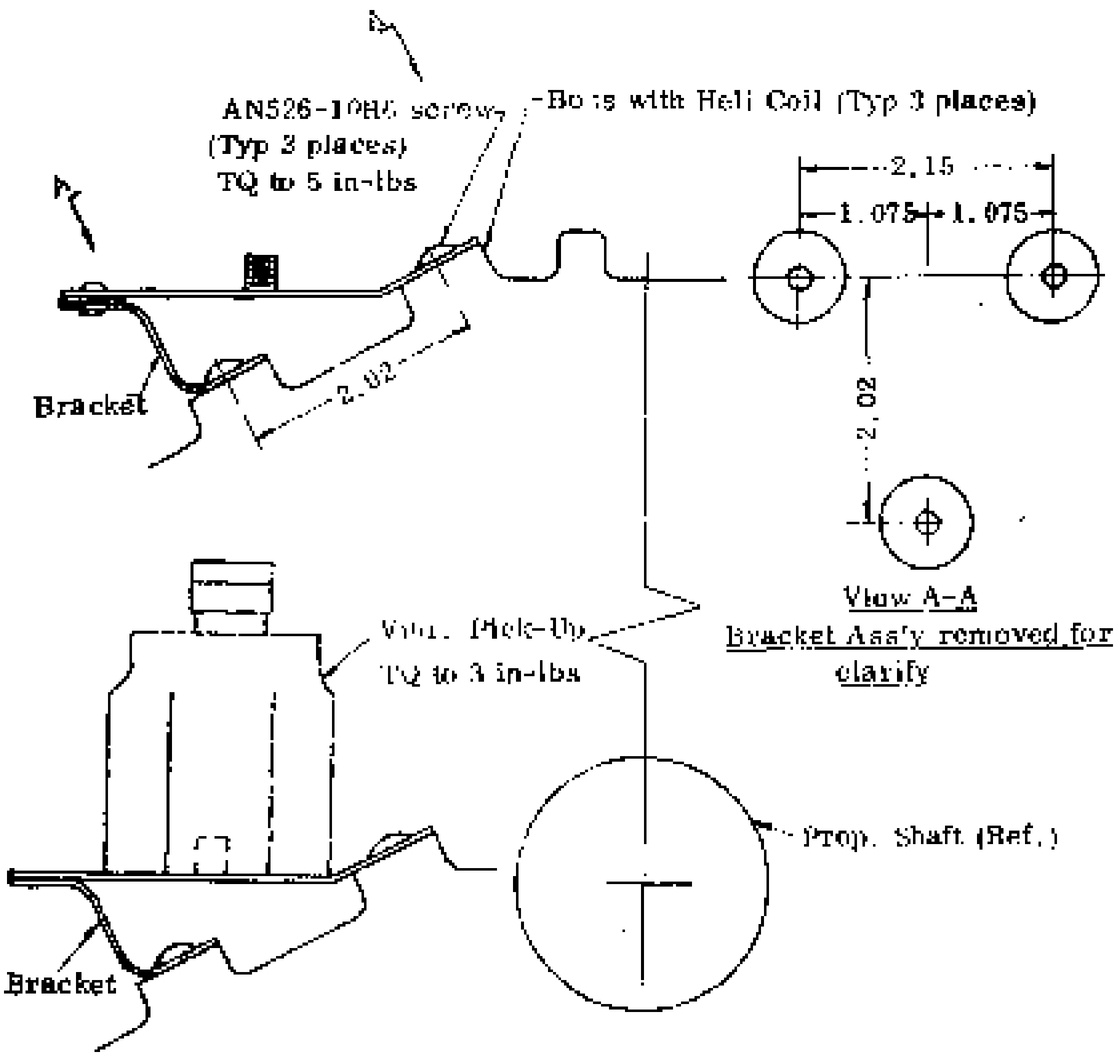
(4) Connect one coax cable end to velocity transducer or vibration pickup and the other end to the analyzer. It is suggested to route coax as shown in Fig. 6-59.

B. Set-up Using Spectral Dynamics Equipment

(1) Connect power cord (115 VAC) to trim balance analyzer and to power source. Turn analyzer to test and allow stabilization time. Check display on trim balance analyzer to ensure proper calibration. The following readings should be observed: 5.0 mils displacement, 6,000 rpm, 180° phase (if applicable), a tolerance of ± 1% of the values is acceptable for prop balance.

(2) The following settings and/or adjustments should be made before beginning the prop balance check.

SWITCH	POSITION/SETTING
Display Range	10
Pickup-Ampl Display	VEL D
Mode	0° Automatic
Filter	IN
Input Selector	As Required



(View looking aft from front of engine)

Fig. 6-58 Installation - Bracket-Vib Pick-Up

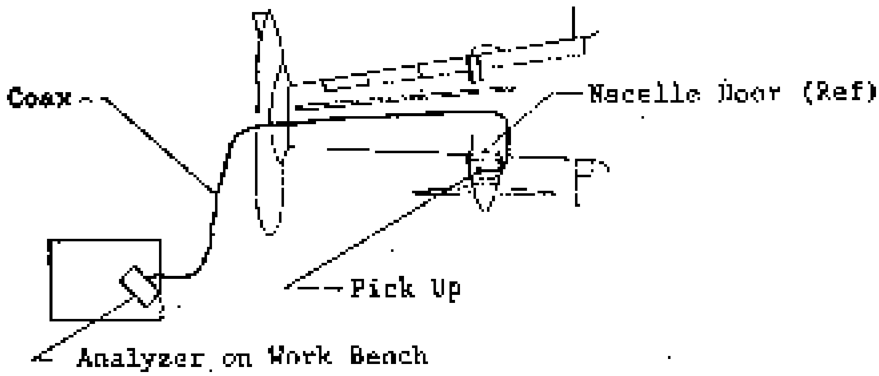


Fig. 6-59

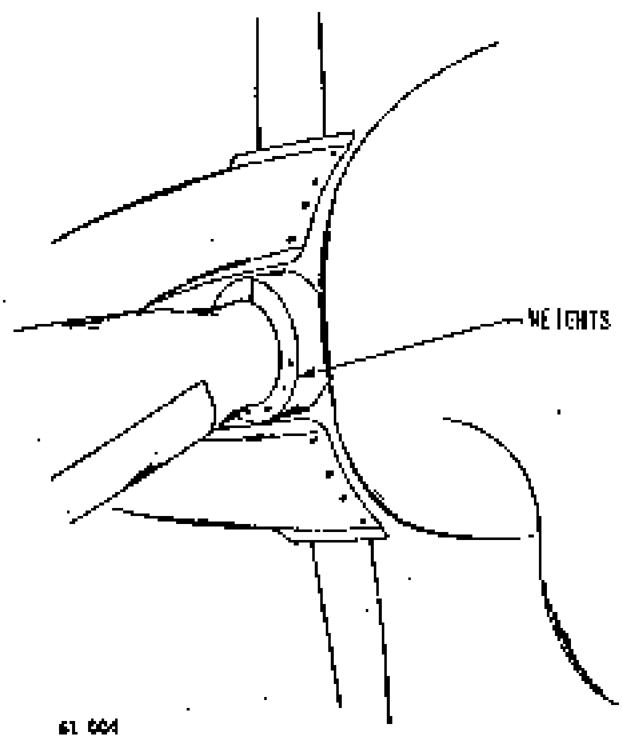


Fig. 6-60 Balance Weight Location

(3) During Engine Run

Adjust the rpm speed manually (smaller mode knob) to indicate 1,600 ± 1 rpm.

NOTE

It may be necessary to adjust the rpm setting during each engine run.

Allow amplitude to settle, depress "Display Hold/Update" switch to lock the "mil displacement" indication.

C. Set-up Using IRD Equipment

(1) Turn power switch to AC or BATT position as required; if AC, connect to power source prior to turning AC ON. Turn "Amplitude Range" select switch to TEST, allow needle to settle, check amplitude "test" indication.

(2) The following settings and/or adjustments should be made before beginning prop balance check.

SWITCH	POSITION/SETTING
Amplitude Range	10
DISP VEL	DISP MILS
Frequency Select	500 - 5 K CPM
Filter Tuning Dial	1,590
Input Select	L or R as required

13.15.5 BALANCING PROCEDURE

A. If the propeller has been balanced before or has balance weights installed, a preliminary run may be required; refer to Fig. 6-60.

(1) Start engine and allow time for warm up.
(2) Accelerate engine to 100% rpm with propeller on start locks.
(3) Allow engine to stabilize, then observe and record displacement reading.
(4) Should the reading be 0.5 mils or less, no further action is required; proceed to Paragraph "C".

B. Perform the following displacement checks for each engine run. All runs to be at 100% rpm and propeller blades on start locks. For placement of weights (slugs), see Fig. 6-60. Number prop blades, 1, 2, etc.

(1) Run No. 1

Remove all dynamic balance weights (slugs) from the propeller. Run engine and record the value "mil" displacement.

(2) Run No.2

Install 3 weights to blade 1, run the engine and record the value "mil" displacement.

(3) Run No.3

Remove weights from blade 1 and install on blade 2, run engine and record the value "mil" displacement.

(4) Run No.4

Remove weights from blade 2 and install on blade 3, run engine and record the value "mil" displacement.

(5) Run No.5 (4 blade propeller only)

Remove weights from blade 3 and install on blade 4, run engine and record the value "mil" displacement.

(6) Subtract the "Disp Mil" value of R1 from each "Disp Mil" value of the additional runs to find the unbalance of each blade and record in the "Balance Data" column.

(7) Determine the heaviest blade by noting the highest displacement value as found in (6) above.

(8) Subtract each light blade from the heaviest blade and record in the "Disp Value" column.

(9) Find the "Disp Value," determined in (8) above, in column A of the Displacement Balance Chart under the heading "DISP MILS" to determine the quantity of weights required for each blade.

(10) Install the required number of weights on the appropriate blade(s). Run the engine and record the mil displacement in the "DISP MILS" column for the next run; also record the weight quantity per blade in the space provided.

(11) A maximum mil displacement of 1.0 mil is acceptable, but 0.5 mil or less is preferred. If the recorded value is greater than 1.0 mil, find the "DISP VALUE" in column B of Displacement Balance Chart to find the adjusted quantity of weights required per blade, and record in the space provided for the next run. Perform next run and record DISP MILS.

NOTE

i. No more than four stacks of eleven weights per blade should be used.

ii. A maximum of three weights should be installed on the de-icer wiring clamp (on the back side of the blade).

iii. A minimum of three threads engaging with clamp thread is necessary for screw length to install balance weight to the clamp .

iv. If dynamic balance cannot be adjusted by balance weight installed on clamp, balance weight may be installed on bulkhead by HARTZELL S/I No.148A.

(12) All propellers that have been dynamically balanced should be identified with special Decal No. A-2803* installed on the propeller cylinder. The presence of this decal will alert propeller repair station personnel that the existing balance weight configuration may not be correct for static balance purposes.

Decal No. A-2803

C. Restore aircraft to original configuration.

(1) Disconnect coax cable from velocity transducer (M80) or vibration pickup (544 T-321-1) and test unit. Remove velocity transducer or vibration pickup and mounting bracket.

(2) Install fuel pressure transmitter.if removed.

(a) Remove pressure line and port plugs.
(b) Position fuel pressure transmitter in place, attach with bolts, tighten to proper torque value and safety wire.
(c) Connect pressure line to transmitter and tighten to proper torque value.
(d) Connect electrical connector and safety wire.
(e) Move wire harness into position, secure with clamp and safety wire.
(f) Secure and/or remove door support rod, close upper engine nacelle door and latch.

(3) Reference the spinner to a blade and remove spinner.

(4) Tighten weight attaching screws and safety wire.

(5) Install spinner per blade reference, clock (center) spinner and torque attaching screws to correct valu.

Sample Propeller Balance

RUN NO	DISP MILS	BALANCE DATA	DISP VALUE (HEAVY - LIGHT = ___)	WEIGHT QUANTITY BLADE NO. 1	2	3	4
R1	7.5			0	0	0	0
R2	6.6	R2 - R1 = - 7.5 = -0.9	1.9 - (-0.9) = 2.8	3	0	0	0
R3	6.2	R3 - R1 = - 7.5 = -1.3	1.9 - (-1.3) = 3.2	0	3	0	0
R4	9.4	R4 - R1 = - 7.5 = 1.9	1.9 - 1.9 = 0.0	0	0	3	0
R5	9.1	R5 - R1 = - 7.5 = 1.6	1.9 - 1.6 = 0.3	0	0	0	3
R6	0.5			4½ / -	5 / -	- / -	- / -
R7							

Propeller Balance Form

NOTE

Column "Blade No. 4" is not required for 3 blade propeller balancing.

RUN NO	DISP MILS	BALANCE DATA	DISP VALUE (HEAVY - LIGHT = ___)	WEIGHT QUANTITY BLADE NO. 1	2	3	4
R1				0	0	0	0
R2		R2 - R1 = ___ - ___ = ___	___ - ___ = ___	3	0	0	0
R3		R3 - R1 = ___ - ___ = ___	___ - ___ = ___	0	3	0	0
R4		R4 - R1 = ___ - ___ = ___	___ - ___ = ___	0	0	3	0
R5		R5 - R1 = ___ - ___ = ___	___ - ___ = ___	0	0	0	3
R6							
R7							

RUN NO	DISP MILS	BALANCE DATA	DISP VALUE (HEAVY - LIGHT = ___)	WEIGHT QUANTITY BLADE NO. 1	2	3	4
R1				0	0	0	0
R2		R2 - R1 = ___ - ___ = ___	___ - ___ = ___	3	0	0	0
R3		R3 - R1 = ___ - ___ = ___	___ - ___ = ___	0	3	0	0
R4		R4 - R1 = ___ - ___ = ___	___ - ___ = ___	0	0	3	0
R5		R5 - R1 = ___ - ___ = ___	___ - ___ = ___	0	0	0	3
R6							
R7							

DISPLACEMENT BALANCE CHART

COLUMN A		COLUMN B
DISP MILS	WT QTY	DISP MILS
0 to 0.30	0.0	0 to 0.30
0.35 to 0.60	0.5	0.35 to 0.65
0.65 to 0.90	1.0	0.70 to 1.00
0.95 to 1.20	1.5	1.05 to 1.35
1.25 to 1.50	2.0	1.40 to 1.70
1.55 to 1.80	2.5	1.75 to 2.00
1.85 to 2.10	3.0	2.05 to 2.40
2.15 to 2.45	3.5	2.45 to 2.70
2.50 to 2.75	4.0	2.75 to 3.05
2.80 to 3.05	4.5	3.10 to 3.40
3.10 to 3.35	5.0	3.45 to 3.75
3.40 to 3.70	5.5	3.80 to 4.10
3.75 to 3.95	6.0	4.15 to 4.50
4.00 to 4.30	6.5	4.55 to 4.80
4.35 to 4.65	7.0	4.85 to 5.20
4.70 to 4.95	7.5	5.25 to 5.55
5.00 to 5.30	8.0	5.60 to 5.90
5.35 to 5.60	8.5	5.95 to 6.25
5.65 to 5.90	9.0	6.30 to 6.60
5.95 to 6.20	9.5	6.65 to 6.95
6.25 to 6.55	10.0	7.00 to 7.35
6.60 to 6.85	10.5	7.40 to 7.70
6.90 to 7.20	11.0	7.75 to 8.05
7.25 to 7.50	11.5	8.10 to 8.40
7.55	12.0	8.45

13.16 FUNCTIONAL CHECK

13.16.1 FEATHERING

(1) Measure pitch angles of blades, with propeller set in feathered position. The angle should be 87.0 ± 0.5° (3-blade prop), 87.5 ± 0.5° (4-blade prop).

(2) If feathering pitch angles should not conform to the specified value, remove piston and adjust feathering stop screw.

13.16.2 DIFFERENCE IN PITCH ANGLES

(1) With propeller set in 30 ± 1° position, measure pitch angles of blades. Difference of pitch angles of blades shall be less than 0.2°.

(2) If difference of pitch angles is beyond the specified value, loosen clamp nut and clamp socket screw, and rotate the blade for adjustment.

13.16.3 REVERSE

Apply oil or air pressure of 200 to 250 psi (14 to 17.5 kg/cm^2) to the piston forcing it out to the full reverse pitch position. Measure pitch angles shall be -6.5 ± 0.5°.

13.16.4 BLADE CLAMP SLIP CHECK

(1) Inspection shall be performed in accordance with the method shown in Fig. 6-56 and shall meet the specified value in Para. 13.14 (4).

(2) If the blade has turned in the clamp, the clamp must be removed and the cause be determined for correction.

(3) Care must be taken not to damage surface of blades.

13.16.5 HIGH PITCH STOP

(1) Apply oil or air pressure of 100 to 125 psi (7 to 8.8 kg/cm^2) to the piston, forcing it out to the reverse pitch position. Bleed pressure.

(2) With high pitch stop on, measure pitch angles of blades. Pitch angle shall be 2.5 ± 0.2°.

(3) If pitch angle should not conform to the specified value, loosen socket screw and adjust. If further adjustment is required, trim stop plate.

13.16.6 BLADE END PLAY

(1) Play between blades and hub spider is measured in peripheral direction.

(2) Blade end play shall meet the requirement of Para. 13.14 (1).

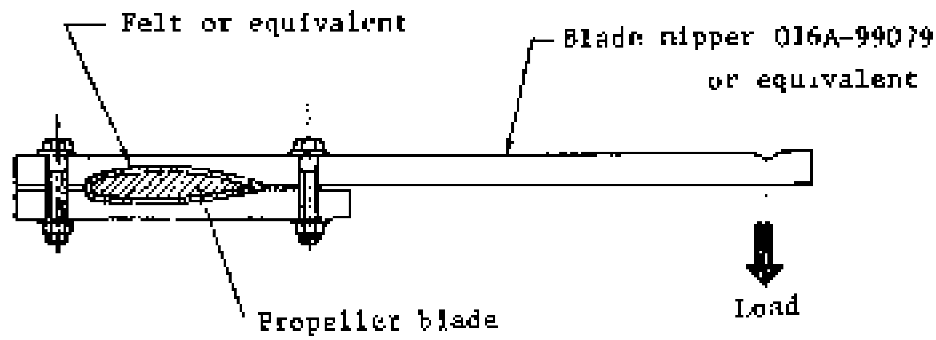


Fig. 6-61

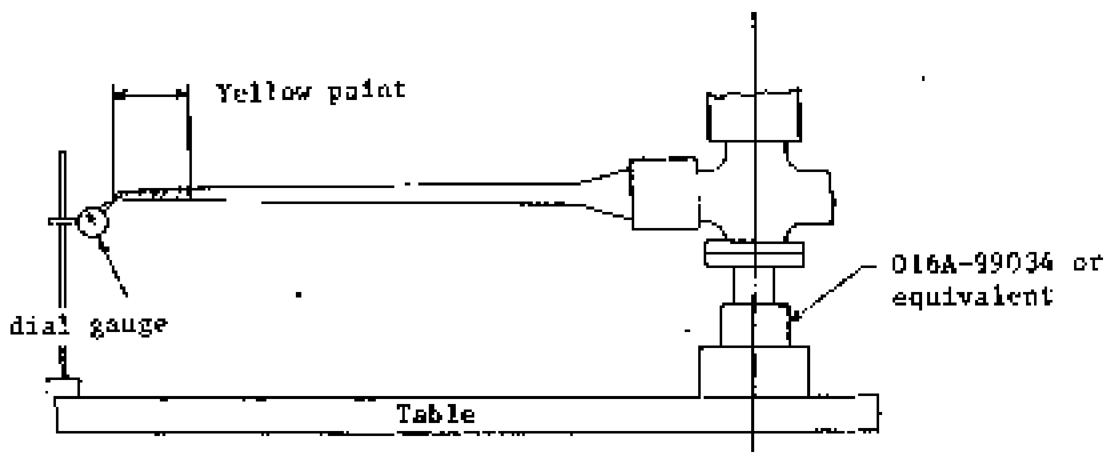


Fig. 6-62

13.16.7 BLADE FIT

(1) Play between blades and hub spider is measured at a right angle to propeller revolution surface.

(2) Blade fit shall meet the requirement of Para. 13.14 (2).

13.16.8 BLADE TRACK

(1) Insert pitch adjusting screw (016A-99190 or equivalent) from the tip of piston and set blade pitch angle to approximately 30° position. (See Fig. 6-62)

(2) Measure difference in amount of blade tracks as shown in Fig. 6-62. Measurement may be made at inboard end of yellow painted section of leading edge.

(3) Blade track shall meet the requirement of Para. 13.14 (3).

13.16.9 LEAKAGE

(1) Apply air or oil pressure of 100 to 125 psi (7 to 8.8 kg/cm^2) to the piston.

(2) Check the following for leakage.

(a) Connection of hub spider with cylinder.
(b) Sliding surface of piston and cylinder.
(c) Clearance between flexlock nut and piston.

13.16.10 MEASUREMENT OF RESISTANCE IN DE-ICER BOOTS LINER

See Para. 5.4.1, Chapter IX.

13.16.11 OPERATIONAL TEST

Rotate blades several times between feathered and full reverse positions by applying or bleeding air or oil pressure of 100 to 125 psi (7 to 8.8 kg/cm^2) to the piston. Check for the following.

(1) For smooth movement over the full stroke.

(2) For interference of link arm with counterweight.

(3) For adequate engagement of high pitch stop plate at unfeathered position.

(4) For smooth return of blades to feathered position, when pressure is bled.

14. TORQUE AND INTERSTAGE TURBINE TEMPERATURE CONTROL SYSTEM

14.1 GENERAL DESCRIPTION

This system is to limit maximum torque and maximum interstage turbine temperature (ITT) automatically to the presetting values (maximum torque 100%, maximum ITT 923°).
This paragraph describes the electric system only. For the system description pertaining to the engine, see AiResearch Maintenance Manual.

14.2 OPERATION (See Fig. 6-63)

This system actuates when circuit breaker ENGINE POWER AUTO LIMIT on the circuit breaker panel is closed and ENG PWR LIMIT switch on the LH switch panel is placed to the AUTO position.

This system operation is that a torque signal from torque transducer, and an interstage turbine temperature signal from engine thermocouple are sent to the limiter controller installed on the RH side of main junction box, and if torque exceeds 100% or interstage turbine temperature exceeds 923°C, the torque motor bypass valve is driven and engine torque or interstage turbine temperature is automatically reduced to the presetting values by bypassing fuel to boost pump inlet and decreasing fuel to engine. Even if either torque system or interstage turbine temperature system fails, the other system can be operated normally.

When ENGINE PWR LIMIT switch is placed to MAN, this system is inoperative.

14.3 SYSTEM CHECK

(1) After engine starting, set condition lever to TAKEOFF-LAND and set engine rpm to 100% by power lever.

(2) Ascertain that ENG PWR LIMIT switch is in the AUTO position and circuit breaker ENGINE POWER AUTO LIMIT is closed.

(3) Set PWR AUTO LIMIT TEST switch, on the center instrument panel, to TORQUE and hold for about 2 seconds and ascertain that fuel decreases and engine rpm reduces about 3%.

(4) Set PWR AUTO LIMIT TEST switch to ITT and hold for about 2 seconds and ascertain that fuel flow decreases and engine rpm reduces about 3%.

(5) Advance power lever and ascertain that engine power is limited to 100% torque or interstage turbine temperature 923°C.

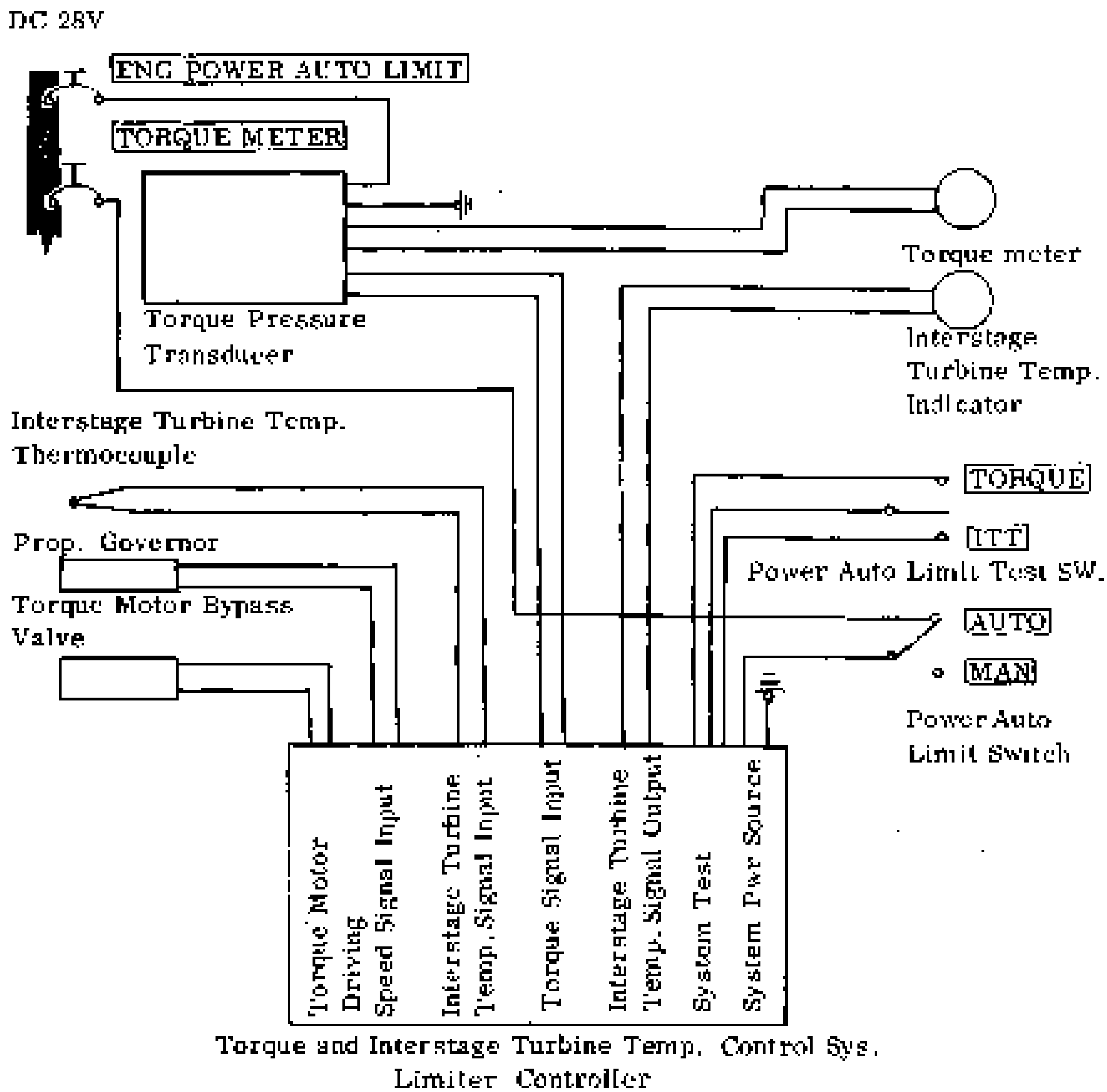

DC 28V
ENG POWER AUTO LIMIT
TORQUE METER
Torque Pressure Transducer
Torque meter
Interstage Turbine Temp. Indicator
Interstage Turbine Temp. Thermocouple
Prop. Governor
Torque Motor Bypass Valve
TORQUE
ITT
Power Auto Limit Test SW.
AUTO
MAN
Power Auto Limit Switch
Torque Motor Driving
Speed Signal Input
Interstage Turbine Temp. Signal Input
Torque Signal Input
Interstage Turbine Temp. Signal Output
System Test
System Pwr Source
Torque and Interstage Turbine Temp. Control Sys. Limiter Controller

Fig. 6-63

15. SYNCHROPHASER SYSTEM

15.1 GENERAL DESCRIPTION

The synchrophaser is a system which is installed as an aid to the pilot to maintain identical engine rpm of both engines. It automatically matches the engine rpm and the phase relationship between the propellers. The system operates within a predetermined range limit, approximately 2%.

There are three major components: magnetic pickups, control box and actuator. The magnetic pickups are located in the propeller governor for each engine. The control box is located in the cockpit and contains all the necessary transistorized circuitry to maintain both engine rpm and propeller blade angle phase. The actuator is a stepping type motor that operates on command from the control box and is mounted on the slave (right) engine nacelle.

15.2 OPERATION

Electrical pulses from each magnetic pickup are fed into the control box where they are compared. Any difference in these pulse rates or phasing pulse position will cause the control box to run the actuator motor and, through the flexible shaft, trim the slave (right) engine propeller governor speed setting to match the master (left) engine rpm and preset phase relationship. Normal propeller governor operation is unchanged by the synchrophaser and will continuously monitor engine rpm and propeller phase angle and reset the slave engine propeller governor as required.

NOTE

The synchrophaser should not be operating during engine ground run operation, taxi, takeoff and landing.

15.3 FUNCTIONAL TEST

To functionally test the synchrophaser on the ground, start both engines and manually synchronize the rpm. Place the synchrophaser switch to ON, allow the engines to become synchronized. Slowly adjust, in small increments, the master (left) engine propeller governor to increase and decrease the rpm. The rpm range over which the slave (right) engine remains synchronized with the master engine is limited to approximately 2%. With the synchrophaser near its travel end, turn the system OFF, allow the actuator time to return to its mid-position, then turn the system ON again; synchronization should result.

15.4 Removal/Installation Synchrophaser Magnetic Speed Pick-up

A. Removal

NOTE

Removal and installation procedures for left and right engine synchrophaser pick-ups are identical.

NOTE

Should the synchrophaser control unit require maintenance, refer to the nearest authorized Woodward Service Center.

(1) Open synchrophaser circuit breaker.

(2) Remove forward upper engine cowling by removing countersunk screws.

(3) Cut safetywire and remove electrical connector from pick-up.

(4) Remove two screws, washers and nuts from pick-up bracket and remove magnetic speed pick-up.

NOTE

Removal of the brush block may be necessary to remove the speed pick-up.

B. Installation

(1) Position speed pick-up in place and secure with screws, washers and nuts. Torque screws to 14.5 ± 1.5 in-lb.

(2) Connect electrical connector to speed pick-up and secure with safety wire.

(3) Position forward upper engine cowling in place and secure with countersunk screws.

(4) Close the synchrophaser circuit breaker.

15.5 Removal/Installation - Synchrophaser Control/Switch

A. Removal

(1) Open synchrophaser circuit breaker.

(2) Remove radio control panel, radar indicator and radar mounting rack.

(3) Remove nut and washer from synchrophaser toggle switch.

(4) Remove electrical connector from toggle switch and remove switch.

(5) Remove electrical connector from synchrophaser control box.

NOTE

Removal of additional electrical connectors to other equipment may be necessary for access and removal of the control box.

(6) Remove screws from control box and remove box.

B. Installation

(1) Position control box under-dash and secure with screws.

(2) Connect electrical connector to control box and connect electrical connectors removed for access.

(3) Connect electrical connector to toggle switch and position switch in panel.

(4) Secure switch to panel with washer and nut.

(5) Position radio control panel in place and secure.

(6) Close synchrophaser circuit breaker.

NOTE

With the propeller loaded aft, the minimum gap between the synchrophaser support bracket and spinner backing plate must be .050 in. measured around the diameter.

16. CONTINUOUS IGNITION SYSTEM (if installed)

16.1 GENERAL DESCRIPTION

The continuous ignition system provides independent ignition switch in each ignition circuit, and allows engine ignition as required. This system is used to prevent engine flame out during takeoff, flight or landing under adverse weather conditions, especially when icing weather condition exists or may exist.

16.2 OPERATION TEST (See Fig.6-64)

(1) Remove access door from center wing leading edge.

(2) Disconnect the left and right engine switch connectors and connect a 20 AWG jumper wire between pins E and F of the respective airframe plug.

(3) Restore electrical power, battery or APU.

(4) Turn the BATTERY key switch ON.

(5) Place the RUN-CRANK-STOP switches to RUN.

(6) Place the continuous ignition switch ON, left then right. The respective ignition light should illuminate to confirm ignition unit function.

CAUTION

Do not exceed 10 seconds operating time during this test

NOTE

Operation may also be verified by an audible click from the respective ignition unit.

(7) Remove the jumper wire and restore the speed switch plugs.

(8) Start the engines in accordance with the applicable Airplane Flight Manual. Ensure that simultaneous illumination of the ignition light and engine start switch light does not occur during engine start.

NOTE

The continuous ignition switches must be OFF during this check.

(9) Shut down the engines and turn the BATTERY key switch OFF. Disconnect the APU if connected.

(10) Reinstall center wing leading edge access door.

16.3 RECOMMENDED DUTY CYCLES OF IGNITION UNIT

Applicable to 868962-1/-2 Ignition Unit
(Engine not modified by GTEC S/B TPE/TSE 331-74-0003)

1 Minute Cycles
First Cycle – 1 Minute ON – 1 Minute OFF
Repetitive Cycles – 1 Minute ON – 1 Minute OFF

2 Minute Cycles
First Cycle – 2 Minutes ON – 2 Minutes OFF
Repetitive Cycles – 2 Minutes ON – 23 Minutes OFF

5 Minute Cycles
First Cycle – 5 Minutes ON – 55 Minutes OFF
Repetitive Cycles – 5 Minutes ON – 55 Minutes OFF

Applicable to 868962-3 Ignition Unit
(Engine modified by GTEC S/B TPE/TSE 331-74-0003 and not modified by GTEC S/B TPE/TSE 331-75-0004)
Up to one hour continuous duty. The total "ON" cannot exceed one hour without one hour "OFF". The one hour "ON" can be either continuous or intermittent.

(Engine modified by GTEC S/B TPE/TSE 331-74-0003 and GTEC S/B TPE/TSE 331-75-0004)

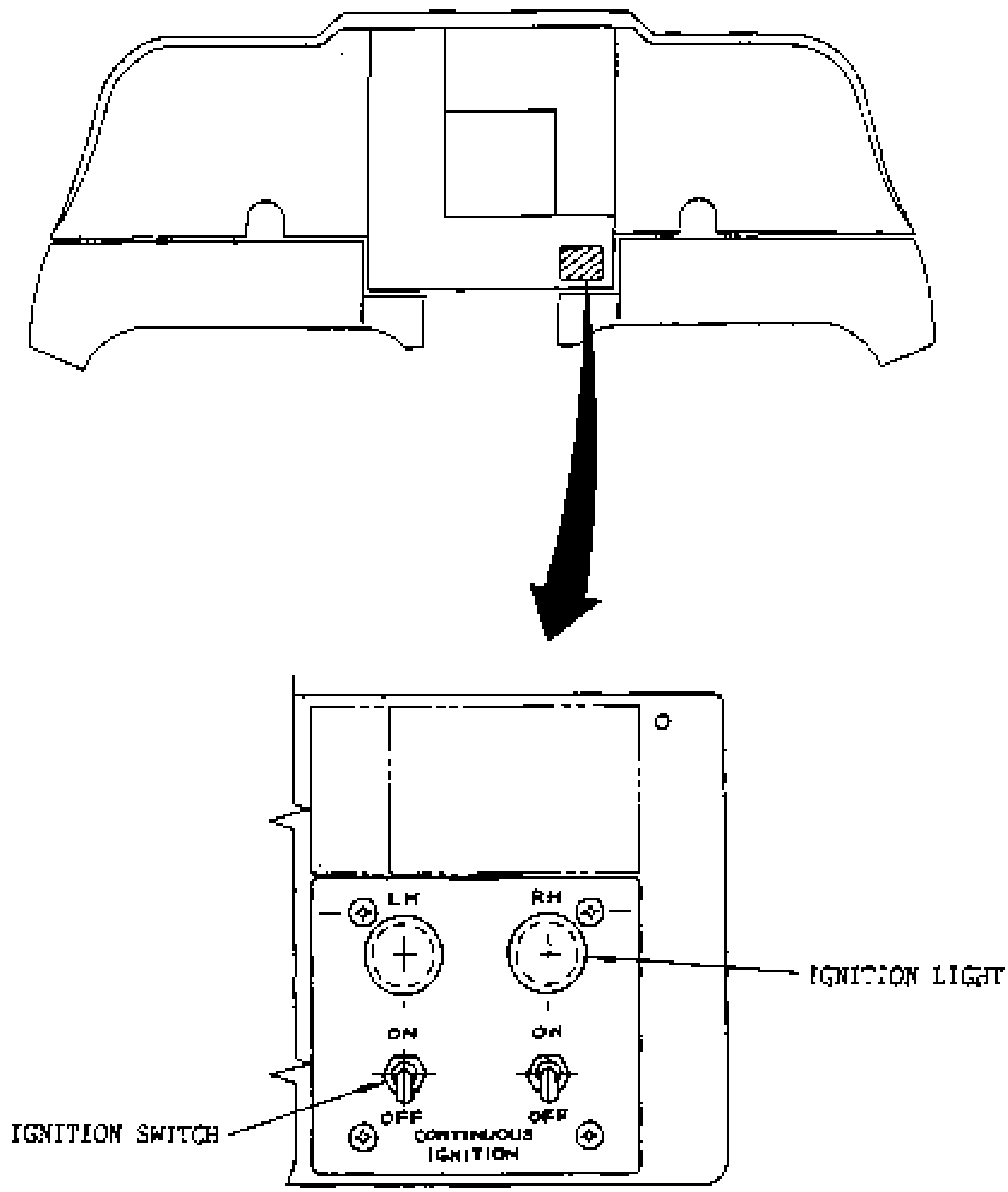
Above +50 degrees F(+10°C) ambient temperature.

Up to one hour continuous duty. The total "ON" cannot exceed one hour without one hour "OFF". The one hour "ON" can be either continuous or intermittent.

Below +50 degrees F(+10) ambient temperature with modified engine and continuous duty ignition box. There is no duty cycle limitation.

CAUTION

Operational times in excess of the duty cycles will decrease the life of igniters and ignition unit.



Ignition Switch Detail

Fig 6-64 Ignition Switch Installation

17. AUTO IGNITION SYSTEM (If installed)

17.1 GENERAL DESCRIPTION

The Auto Ignition System is activated by a torque pressure switch that senses the pressure output of the hydraulic torque sensor . If the engine flames out, the torque pressure drops rapidly below the torque switch set point, thus actuating the ignition. Following relight, the ignition is deactivated as the torque pressure goes above the torque switch set point. The system is deactivated unless the "CRANK-RUN-STOP" switch is in the "RUN" position.
During ignition operation, the yellow "LH IGNITION" or "RH IGNITION" annunciator is illuminated.
The Auto-Ignition System shall be placed in "AUTO" for all normal flight conditions.
The Auto-Ignition System shall be placed in "CONT" for certain additional conditions that are described in the Airplane Flight Manual.

17.2 OPERATION TEST

(1) Perform AUTO-IGNITION SYSTEM CHECK in accordance with paragraph AFTER STARTING ENGINES in Section 5 "NORMAL PROCEDURES" of the aplicable FAA Approved Airplane Flight Manual.

CHAPTER

7

FUEL SYSTEM

CHAPTER VII

FUEL SYSTEM

CONTENTS

1. GENERAL DESCRIPTION

The fuel system consists of the following systems (See Fig. 7-1).

Main fuel system	:	System which feeds fuel from wing tank to engines.
Tip tank fuel system	:	System which transfers fuel from tip tanks on the wing tip to the wing fuel tank.
Outer tank fuel system	:	System which transfers fuel from outer tanks in the outer wing to the main fuel tank.

Fuel Specifications : The following lists the approved fuels.

(1) Aviation Turbine fuels ASTM D 1655-68T Types Jet A, Jet A-1 and Jet B
(2) MIL-T-5624-1 Turbine Fuel : Grades JP-4 and JP-5
(3) MIL-F-5616-1 Fuel : Grade JP-1
(4) MIL-F-46005 (MR)-1 Types I and II
(5) British Ministry of Supply Specifications
 (a) D. Eng. R.D. 2482 Issue No. 2
 (b) D. Eng. R.D. 2486 Issue No. 2
 (c) D. Eng. R.D. 2494 Issue No. 4
(6) MIL-G-5572 Aviation gasoline (emergency fuel use only)
 (a) Grade 80/87 octane
 (b) Grade 100/130 (low lead) octane

CAUTION

Avoid operations above 10,000 ft. msl when using aviation gasoline.

Anytime the mixture of aviation gasoline to turbine fuel is 25% or more, add one quart of aviation grade oil (MIL-L-6082, Grade 1065 or 1100) per 100 gallons of aviation gasoline to the fuel mixture.

(a) Grade 80/87 octane aviation gasoline may be used as emergency fuel only. The amount must not exceed 1,000 gallons per engine for each 100 hours of engine operation.

(b) Grade 100/130 (low lead) aviation gasoline may be used as emergency fuel only. The amount must not exceed 250 gallons per engine for each 100 hours of operation. Maximum total usage is 7,000 gallons per engine during any 3,000 hour period.

(c) If combinations of Grades 80/87 octane and 100/130 LL are used, the following formula is used to establish allowable proportions of each for any 3,000 hour period.

$$\frac{\text{gal. 100/130 LL}}{7{,}000} + \frac{\text{gal. 80/87 octane}}{30{,}000} = \text{Less than } 1.0$$

CAUTION

Water may freeze within fuel system. Use MIL-I-27686 jet fuel icing inhibitor in fuel of all tanks. Do not mix in excess of 0.15% in volume of MIL-I-27686 jet fuel icing inhibitor to fuel of all tanks. Be careful because some commercial fuels already contain icing inhibitor. Do not add to such fuels.

NOTE

Shell ASA-3 anti-static additive, or equivalent, may be added in amount to bring the fuel up to 300 conductivity units, but in no event shall the additive exceed 1 ppm (parts per million).

Fuel Tank		Qty	Capacity (U.S. gallon)	Unusable fuel (U.S. gallon)
Main tank		1	159	5
Outer tank	Left tank	1	15	–
	Right tank	1	15	–
Tip Tank	Left tip tank	1	93	3
	Right tip tank	1	93	3
Total fuel capacity		375		
Unusable capacity		11		
Total usable capacity		364		

1.1 MAIN FUEL SYSTEM

The main fuel system consists of three integral tanks in wing center section, right and left wing outboard sections (W STA 580 to 1950), two boost pumps for supplying fuel to both engines, two fuel shutoff valves, two fuel filters, two boost pump warning switches, manifold check valve, tubing, hoses, fuel quantity transmitter, fuel flow transmitters, etc.

The three integral tanks are interconnected to act as one fuel tank, having a total capacity of 159 U.S. gallons including unusable fuel of 5 gallons. The fuel filler port is located on top of outboard tank. The fuel in the tanks flows into the center tank by gravity. Between the center tank and outboard tanks, a check valve is provided to prevent the reverse flow of fuel.

Boost pumps are located at the left rear and right center bottom inside the center tank. Through these pumps, fuel is injected under pressure into the manifold located on the front wall in the center tank. The boost pumps supply fuel to both engines through the manifold. The fuel flows out of the manifold, departs right and left, and is delivered under pressure through the electric-motor operated fuel shutoff valves and the fuel filters to both engines.

The boost pump fail warning switch is installed on the front, right-hand side of the center tank, and automatically illuminates the lights of BOOST PUMP FAIL on the annunciator panel of cockpit and of CAUTION on the upper part of the instrument panel, when fuel pressure falls below 3.2 psi, indicating the presence of trouble in the pump. In the center tank and outboard tanks, capacitor type fuel quantity transmitters are mounted to show the total quantity of fuel in the tanks.

1.2 AUXILIARY TANK FUEL SYSTEM

Auxiliary tank fuel system consists of tip tank fuel system and outer tank fuel system.

1.2.1 TIP TANK FUEL SYSTEM

This system consists of tip tanks (93 U.S. gallons each) mounted at the wing tips, air pressure shutoff valve, air pressure regulator, fuel shutoff valves, fuel level control valve, and tubing. The fuel in the tip tanks is transferred into the center fuel tank by air pressure. Fuel transfer from the tip tank to the center tank is controlled by the fuel level control valve installed in the center tank.

1.2.2 OUTER TANK FUEL SYSTEM

This system consists of outer tanks (15 U.S. gallons each) mounted at the LH and RH outer wings, transfer pumps, fuel low pressure warning switch, check valve, vent valve, dip stick, and tubing. The fuel in the outer tanks is transferred into the center tank by transfer pumps, flowing together with the tip tank fuel transfer line.

1.2.3 CONTROL OF AUXILIARY TANK FUEL SYSTEM

Tip tank and outer tank fuel is transferred in AUTO or TIP MANUAL mode. In AUTO mode, tip tank fuel and outer tank fuel is transferred in turn, selecting automatically. TIP MANUAL mode is used when AUTO mode is defective and only tip tank fuel is transferred. Mode is selected by placing FUEL TRANSFER switch to AUTO or TIP MANUAL.

1.3 FUEL ADDITIVES

1.3.1 ANTI-ICING ADDITIVES

Fuel system icing inhibitors which conform to MIL-I-27686 are acceptable for use with fuel which is not premixed with anti-ice additives. The amount of additive to be used in the fuel should not exceed 0.15% (maximum) by volume.

For blending procedures, see Paragraph 8.7.

1.3.2 ANTI-STATIC ADDITIVE

SHELL ASA-3 anti-static additive (or equivalent) is approved for use in amounts to raise the fuel up to 300 conductivity units, except that in no case shall the additive concentration exceed 1.0 ppm.

1.3.3 MICROBICIDE ADDITIVE

Microbicides should not be used in fuels unless there is presence of foreign organisms or conditions are such that their presence may be anticipated. In such instances Sohio Biobor JF Biocide (mfg. by U.S. Borax) or equivalent may be used in the fuel for pesticide and microbicide purposes in amounts not to exceed 270 ppm maximum for the initial sterilization, and 135 ppm for continuous usage or intermittent use.

Should there be an indicated presence of micro-organisms in the fuel system, it is recommended to use the biocide additive for a minimum of three months or 600 hours of engine operation. The initial sterilization level (270 ppm) should be maintained at least 72 hours before adding untreated fuel.

Should conditions require continued use, stored fuel may be pre-mixed with the biocide additive in amounts not to exceed 135 ppm.

The biocide may be meter blended with the fuel during the fueling process or by batch blending. When batch blending, the biocide should be added while the tank is approximately 1/2 full.

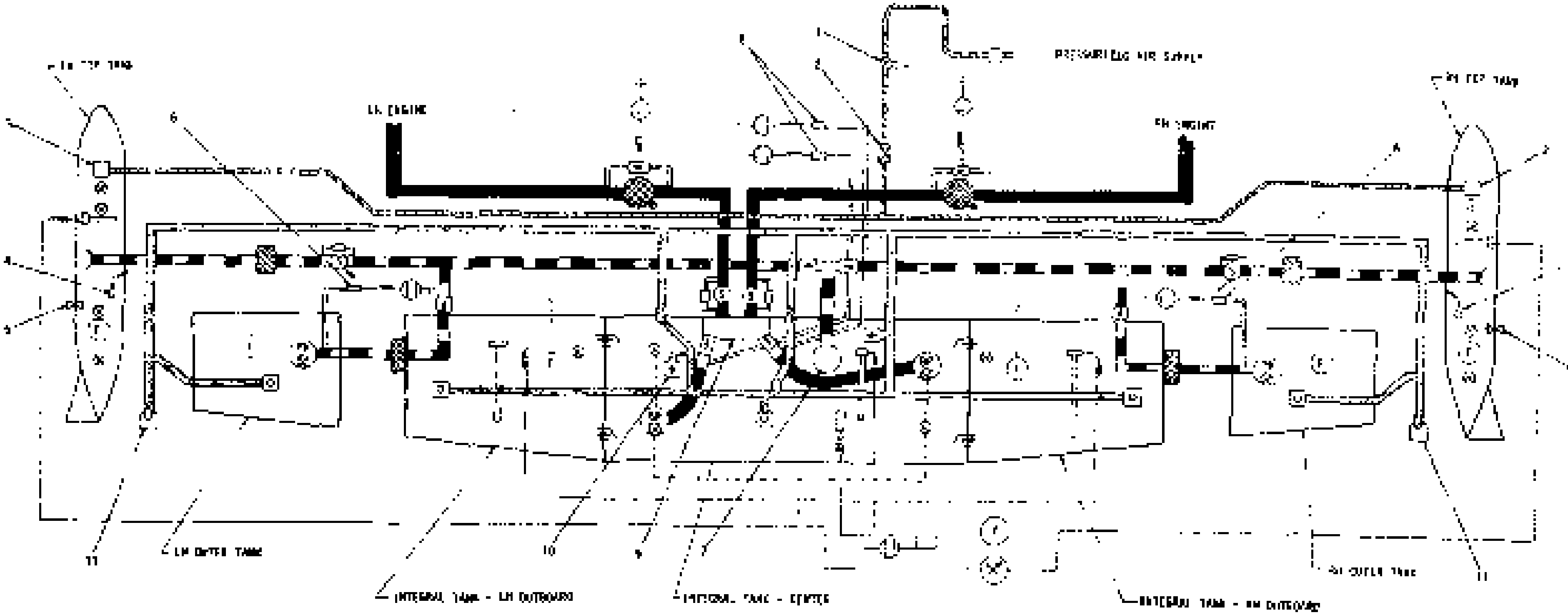
NOTE

Complete mixing is necessary for fungicidal activity.

The following chart lists the required amount of biocide to be blended with various fuel tank quantities. The biocide requirement is based on a fuel weight of 6.7 lbs/gal for this chart. To blend with other fuels, multiply the amount of fuel in pounds by 0.004; this will give the amount of biocide in fluid ounces required to meet the 270 ppm requirement (use a factor of 0.002 for 135 ppm).

AMOUNT BIOCIDE REQ'D
(BASED ON FUEL DENSITY 6.7 LBS/GAL)

AMT OF FUEL	BIOCIDE REQ'MT	
	270 ppm gal (fl. oz.)	135 ppm gal (fl. oz.)
MAIN TANK		
159 gallons	0.032 (4.13)	0.016 (2.065)
80 "	0.016 (2.08)	0.008 (1.040)
40 "	0.008 (1.04)	0.004 (0.520)
OUTER AUX TANK		
15 gallons	0.0030 (0.390)	0.00150 (0.195)
8 "	0.0015 (0.208)	0.00075 (0.104)
TIP TANK		
93 gallons	0.0186 (2.418)	0.0143 (1.209)
47 "	0.0094 (1.222)	0.0047 (0.611)
23 "	0.0046 (0.598)	0.0023 (0.299)
AMT OF STORED FUEL		
10	0.002 (0.26)	0.001 (0.13)
500	0.100 (13.00)	0.050 (6.50)
1,000	0.200 (26.00)	0.100 (13.00)
2,500	0.500 (65.00)	0.250 (32.50)
5,000	1.000 (130.00)	0.500 (65.00)
10,000	2.000 (260.00)	1.000 (130.00)



1. AIR SHUTOFF VALVE
2. AIR PRESSURE REGULATOR
3. [illegible]
4. LOW LEVEL SWITCH
5. SHUTOFF VALVE
6. FUEL LOW PRESSURE WARNING SWITCH
7. FUEL LEVEL CONTROL VALVE
8. BOOST PUMP WARNING SWITCH
9. MANIFOLD
10. PRESSURE RELIEF VALVE
11. VENT PORT

FUEL FILLER PORT
FUEL QUANTITY TRANSMITTER UNIT
FUEL QUANTITY INDICATOR - [illegible]
FUEL QUANTITY INDICATOR - [illegible]
ELECTRICAL WIRE
FUEL TRANSFER LINE
FUEL SCREEN
CHECK VALVE
PRESSURIZED AIR LINE
TRANSFER PORT
FUEL SHUTOFF VALVE

ELECTROMAGNETIC [illegible] VALVE
BOOST PUMP
FUEL FILTER
[illegible]
VENT LINE
WARNING LIGHT
DRAIN VALVE
DRAIN LINE

25205

Fig 7-1 Fuel system

2. FUEL FEED SYSTEM

Fuel in the wing tanks flows into the center tank by gravity. A check valve is located between the center tank and outboard tanks to prevent reverse flow of fuel.

Fuel is fed into the manifold mounted in front of the tank by two boost pumps in the center tank. A boost pump warning switch is mounted in front of the center tank, and turns on the light of BOOST PUMP FAIL on the annunciator panel of the cockpit and of CAUTION on the instrument panel, to indicate the presence of trouble when fuel pressure has dropped below 3.2 psi.

Fuel flow branches off into two lines at the manifold and passes through the fuel shutoff valve and flows into the fuel filter. Filtered fuel is fed through the fuel line to the fuel inlet of the engine. In case the fuel filter becomes blocked with dirt, the fuel flows through the bypass valve to each engine.

2.1 BOOST PUMP

The boost pumps are constant-speed, centrifugal, submerged type pumps located one each at the left rear and right center bottom of the center fuel tank. These pumps are operated by closing the circuit breaker BOOST PUMP. The boost pump is designed to operate on 28 volt 10 amp DC power, has a capacity of 0 to 2,000 PPH of flow rate at 14.1 to 50 psi (0.9 to 3.5 kg/cm^2) of delivery pressure, which is sufficient to operate both engines at the same time with one pump operational). The electrical wiring to the pumps is designed to operate both pumps when either engine has malfunctioned.

2.1.1 REMOVAL AND INSTALLATION (See Fig. 7-2 and 7-3)

Turn main tank switch and aircraft power source OFF by pulling out circuit breaker "BOOST PUMP".

(1) Make sure fuel shutoff valve is closed.
(2) Drain integral tanks.
(3) Remove center tank access panel on top of wing.

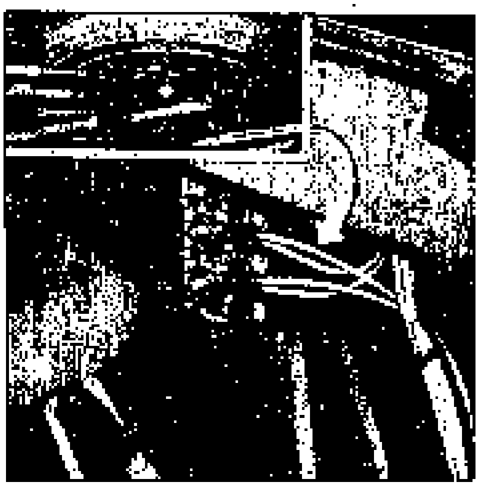


Fig 7 - 2 Disconnect fuel drain line and wire

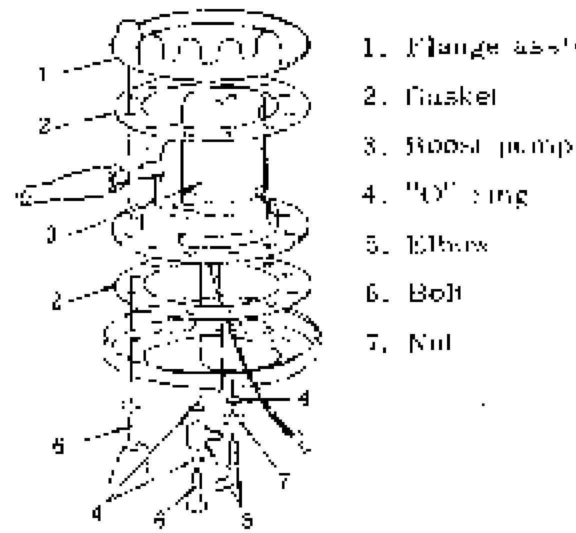


Fig 7 - 3 Installation of boost pump

(4) Disconnect hose from boost bump to manifold.

(5) Remove wing fillet door, and disconnect fuel drain line and seal drain line from the bottom of pump.

(6) Disconnect electrical wiring at knife disconnect.

(7) Cut lock wire and remove bolts attaching boost pump.

(8) Take out pump together with gasket and flange assembly through center tank access door on top of wing.

(9) Install in reverse sequence of removal.
Take care of the following items when installing.

(a) Installation torque of boost pump attaching bolts is 25 to 30 in-lbs (28.8 to 34.6 kg/cm). After installation, apply lock wire to bolts.

(b) Installation torque for universal elbow attaching bolts of boost pump drain is 40 to 65 in-lbs (46 to 74.9 kg/cm).

(c) Installation torque for fitting attaching bolt at fuel delivery port is 75 to 85 in-lbs (86.4 to 97.9 kg/cm).

(d) Installation torque for access panel attaching screws on center tank upper surface is 30 to 35 in-lbs (34.6 to 40.3 kg/cm).

NOTE

Access panel of center tank on top of wing should be installed in correct position.
Incorrect installation may cause fuel leakage.

NOTE

Place pump with its fuel delivery port faced in proper direction when installing.
Proper direction of delivery port is about 20 degrees to the right from front (flight direction) on left-hand side pump and about 95 degrees to the left from front (flight direction) on right-hand side pump.

2.1.2 OPERATION CHECK

Operation check of boost pump is accomplished as follows:

(1) Close MASTER CAUTION circuit breaker and make sure that left and right boost pump warning lights BOOST PUMP FAIL are illuminated.

(2) Close RH BOOST PUMP and LH BOOST PUMP circuit breakers and make sure that left and right boost pump warning lights BOOST PUMP FAIL go out.

(3) Make sure that operation noise of pump is normal, no leakage from pump section and drain line connecting sections, and that leakage from the seal drain is less than 2 cc per hour.

(4) Pull out circuit breakers RH BOOST PUMP and LH BOOST PUMP. Left and right boost pump warning lights BOOST PUMP FAIL should be illuminated.

(5) Pull out circuit breaker MASTER CAUTION.

NOTE

Do not operate boost pump in dry fuel condition.

2.2 FUEL SHUTOFF VALVE

The fuel shutoff valve is a motor-operated gate valve, located on the manifold at the center tank front outlet. To open the shutoff valve, close "SHUTOFF VALVE" circuit breaker and turn "MAIN TANK" switch to on. To close the valve, turn the switch to off. In case of emergency such as engine fire, trouble, etc., when the fire handle in the cockpit is pulled the valve closes immediately. The valve is designed to close normally in one second. The valve-driving motor operates on 18 to 30V DC power.

2.2.1 REMOVAL AND INSTALLATION (See Fig 7-4)

(1) Drain integral tanks and fuel lines.

(2) Remove center wing front access panel.

(3) Disconnect fuel line at the valve.

(4) Disconnect electric connector and bonding jumper.

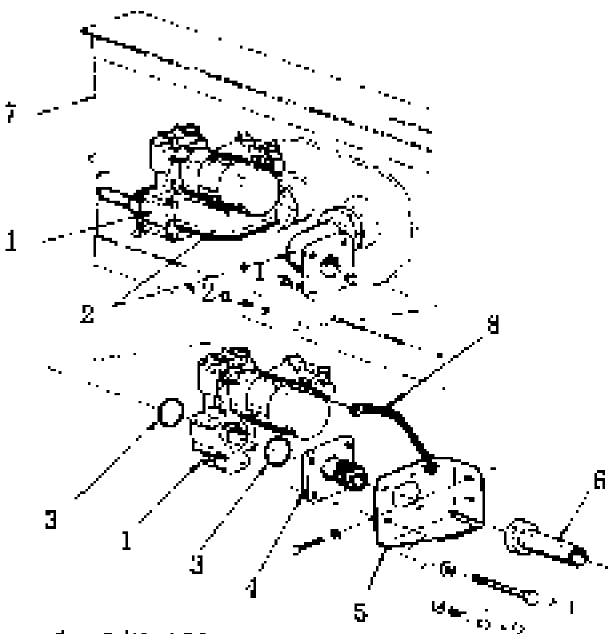
(5) Remove two bolts to separate bracket from front spar of wing.

(6) Remove bolts which connect shutoff valve and remove shutoff valve.

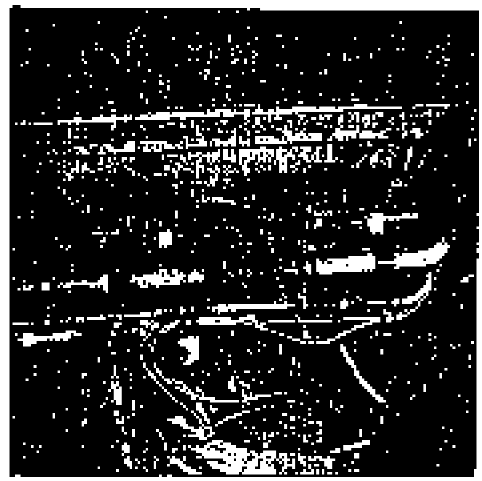
(7) Install in reverse sequence of removal.
Take care of the following items when installing.

(a) Install "O" ring, lubricating with petrolatum (VV-P-236).

(b) Installation torque of bolt connecting the valve and connector to elbow is 50 to 70 in-lbs. (57.6 to 80.6 kg-cm) *1, 20 to 25 in-lbs. (23 to 28.7 kg-cm) *2.



1. Shutoff valve
2. Elbow
3. "O" ring
4. Connector
5. Bracket
6. Line
7. Front spar
8. Bonding jumper



*1 Aircraft S/N 652SA, 661SA
*2 Aircraft S/N 697SA and subsequent

Fig 7 - 4 Installation of fuel shutoff valve

2.2.2 OPERATION CHECK

(1) Remove center wing front access panel.

(2) Close "BOOST PUMP" and "SHUTOFF VALVE" circuit breakers.

(3) Turn "MAIN TANK" switch to on.

(4) Make sure that valve override lever (Red) moves positively from FULL CLOSE position to FULL OPEN position and that pressured fuel flows out when fuel hose to engine is removed.

(5) Turn "MAIN TANK" switch to off.

(6) Make sure that valve override lever moves positively from FUEL OPEN position to FULL CLOSE position and that fuel flow from hose connecting to engine stops.

NOTE

Check that override lever of valve does not interfer with tubing or wiring around the valve.

(7) Pull out circuit breaker.

(8) Install center wing front access panel.

2.3 FUEL FILTER (See Fig. 7-5)

The fuel filter is attached to the bracket located at about the middle of the wing leading edge between the fuselage and the nacelle with four bolts. The filter element is accessible from the access door located on the lower surface of the wing leading edge. The filter element is produced from 60 to 70 mesh metal screen. The filter is provided with a pressure switch, which automatically operates to turn on the fuel pressure warning light in the cockpit, indicating obstruction of filter element, when a difference between inlet pressure and outlet pressure has reached 1.9 ± 0.3 psi (0.13 ± 0.02 kg/cm^2). The filter element consists of a 1st and 2nd element. When differential pressure has reached 2.0 ± 0.25 psi (0.14 ± 0.02 kg/cm^2) due to obstruction of the 1st element, the 1st valve opens to allow the flow through the 2nd element. In case the 2nd element also has been blocked and the differential pressure has reached 4.0 ± 0.5 psi (0.28 ± 0.03 kg/cm^2), the 2nd valve opens and fuel does not flow through the element but flows through the bypass line. The drain valve is located on the bottom of the filter bowls, for draining fuel and water.

2.3.1 REMOVAL AND INSTALLATION OF FILTER (See Fig. 7-6)


Fig. 7 - 5 Fuel filter

(1) Make sure that fuel shutoff valve is closed.

(2) Remove leading edge lower access panel.

(3) Drain fuel from drain valve on filter bowl.

(4) Cut off lock wire on the bowl nut, loosen the clamp or bowl nut that holds filter bowl and gently lower bowl and filter element straight down.

(5) Remove filter element from the bowl.

(6) Install in reverse sequence of removal. Take care of the following items when installing.

(a) Install O-ring lubricated with petrolatum (VV-P-236).

ORIGINAL
As Received By
ATP

(b) Check filter element and bowl for scratches, abrasions or cracks. Damaged part should be replaced.

(c) Insert element and baffle with bypass valve into the bowl, set O-ring in groove on head, and install the bowl at correct position. Tighten nuts hand-tight and secure lock wire. For clamp type filter tighten clamp to 25 ~ 30 in-lbs torque.

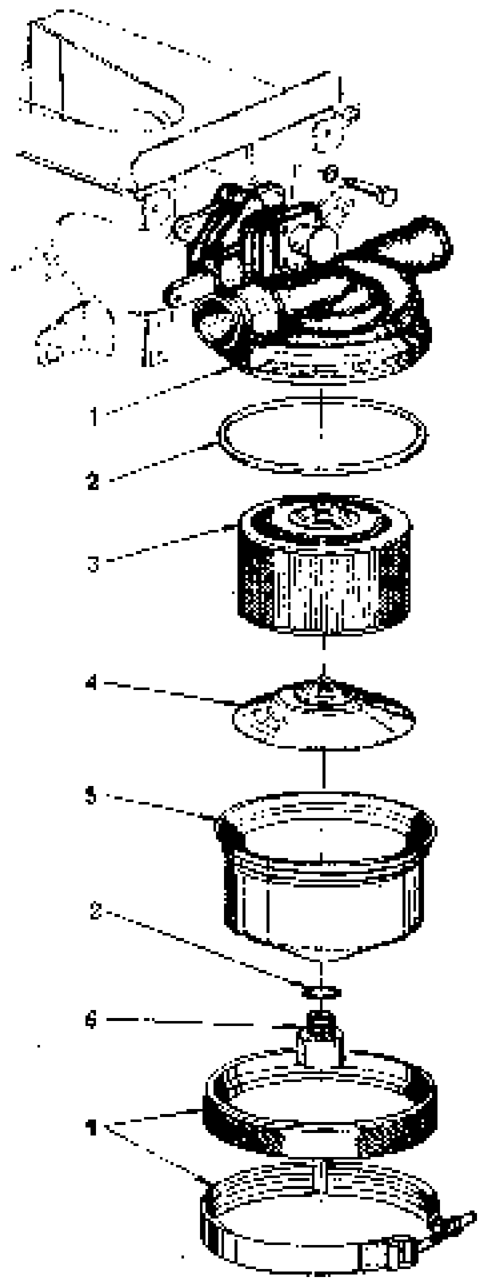
(d) If nut is tightened with "O" ring out of groove, damage or cuts to "O" ring will occur, so installation should be accomplished with great care.

NOTE

When removing bowl from filter assembly, lower the bowl down gently to avoid spilling of fuel.

CAUTION

After removing filter element, do not leave the head open. Reinstall the bowl temporarily so as not to let dust in.



1. Filter head
2. "O" ring
3. Filter element
4. Baffle
5. Bowl
6. Drain valve
7. Nut or clamp

Fig 7 - 6 Fuel filter element

2.3.2 CLEANING OF FILTER ELEMENT

The life and performance of the filter element depends on the method for handling it; therefore, handling should be done carefully.
To clean the fuel filter element, proceed as follows.

(1) Soak the element in trichloroethylene for 10 minutes and clean off bulk of dirt by vapor degreasing.

(2) Sonicly clean element in mild detergent solution (MIL-D-16791, type I) for 20 ~ 30 minutes.

(3) Rinse in hot water, 140° ~ 200°F for 20 minutes.

(4) Allow screen to air dry, or dry in oven at a temperature between 100° ~ 300°F.

CAUTION

i Do not blow across element screen with damp air.

ii Never allow removed filter elements to dry in a dirty condition. When it is impossible to immediately clean the removed element, immerse it in fuel or solvent until able to clean.

iii If ultrasonic cleaning is not available, clean it in solvent P-D-680 or equivalent and dry with air.

2.4 BOOST PUMP WARNING SWITCH

The boost pump warning switch is located at the center of the forward side of the front spar. It is designed to sense fuel pressure from the fuel line extending from the boost pump and, when fuel pressure decreases to 3.2 psi (0.224 kg/cm^2) or lower, it will operate automatically to turn on CAUTION warning light in the cockpit and BOOST PUMP FAIL on the annunciator panel, indicating the presence of trouble in the pump. The switch is accessible from the center wing front access door. When the quantity of fuel in the tank is 120 gal. or less, the switch can be removed without draining fuel.

2.4.1 REMOVAL AND INSTALLATION (See Fig 7-7)

(1) Open fuel filler port and make sure fuel level is more than 2.7 in. below full fuel level. When more fuel remains in tank, drain fuel to lower level. (Drain so that fuel quantity indicator of main tank indicates 120 gal. or lower.)

(2) Open center wing front access panel.

(3) Disconnect fuel line and switch.

(4) Disconnect electric connector.

(5) Pull out switch from bracket.

(6) Install in reverse sequence of removal.

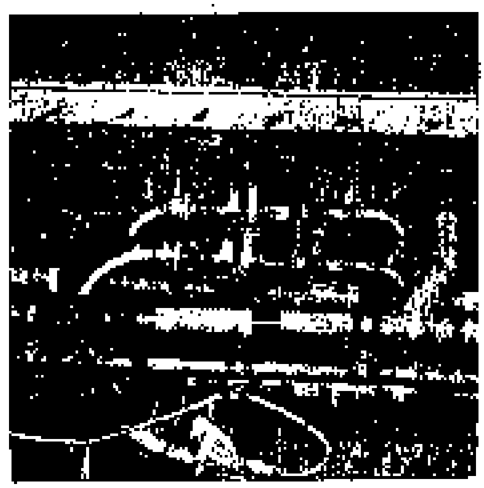


Fig 7-7 Installation of fuel low level warning switch

2.5 FUEL LOW LEVEL WARNING SWITCH

When the fuel indicator of wing tank indicates 30 ± 5 U.S. gallons, a micro switch installed inside the fuel quantity indicator operates automatically and turns on the warning light FUEL LOW LEVEL in the cockpit. In addition to this switch, a float switch is installed in the center tank and turns on the warning light FUEL LOW LEVEL when the fuel in the center tank becomes 10 ± 1.5 U.S. gallons. This float switch actuates independently of micro switch in the fuel quantity indicator.

2.5.1 OPERATION AND CHECK

(1) Turn LH and RH MAIN TANK switches to ON.
(2) Drain fuel in wing tank until the warning light FUEL LOW LEVEL illuminates.
(3) When the warning light illuminates, fuel in the tank should be 30±5 U.S. gallons, excluding unusable fuel.
(4) Inspection of float switch in the center tank is performed without operating the fuel quantity indicator. Discharging fuel (center tank and LH and RH outboard tanks) after illumination of warning light should be 33±5 U.S. gallons.

2.6 MANIFOLD

The manifold consisting of adapter and body is located slightly to the left from the center of the front side of the center fuel tank. Fuel coming from two boost pumps in the center tank collects in this manifold, and is then fed to both engines. The adapter is attached to the front spar of the center fuel tank. The elbow on which the fuel shutoff valve is installed is mounted on the adapter.

2.6.1 REMOVAL AND INSTALLATION (See Fig 7-8)

(1) Drain integral tank.

(2) Remove center tank upper LH access panel.

(3) Disconnect fuel hose which connects boost pump to manifold, at the the tee joint.

(4) Remove pressure sensing hose connecting to boost pump warning switch.

(5) Remove center wing front access panel.

(6) Remove fuel shutoff valve and elbow.

(7) Cut lock wire and remove ten attaching bolts for manifold.

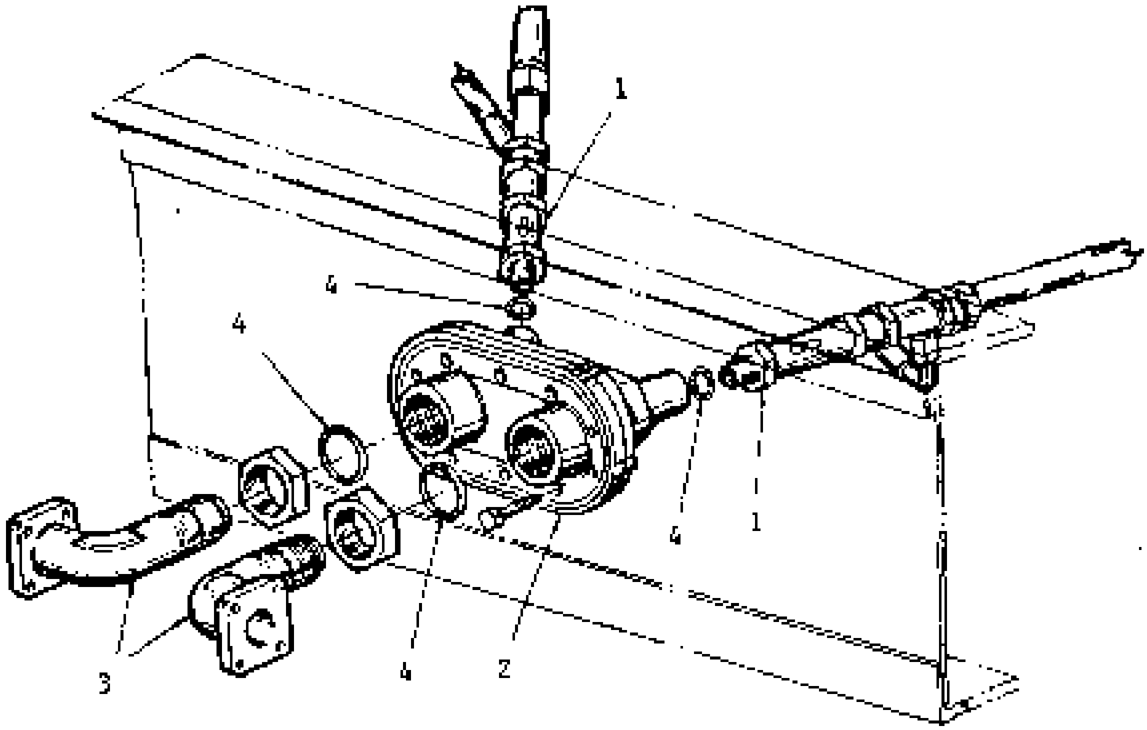
(8) Take manifold out through center tank upper access door.

(9) Install in reverse sequence of removal.

3. FUEL VENT SYSTEM

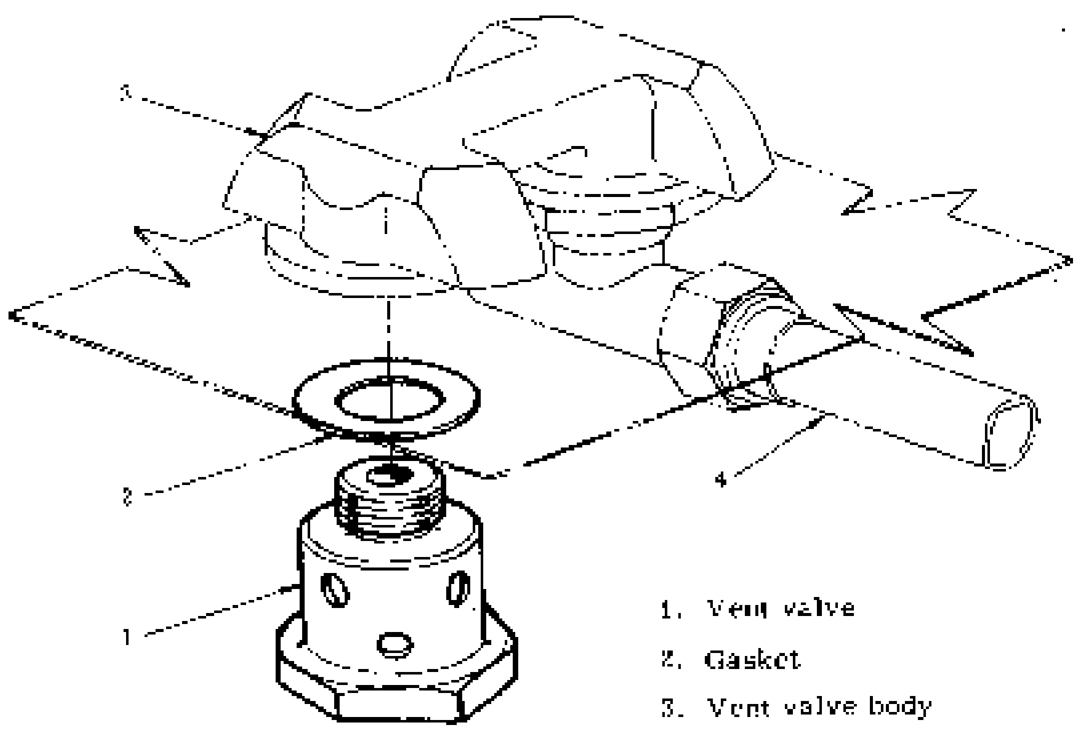
The fuel vent system is provided for the proper adjustment of fuel tank inner pressure under every condition. Integral tanks are interconnected by vent holes and vent lines and air in the tanks are freely open to each other. Vent holes are provided between the center tank and outboard tanks. The vent valves are located on the top of the center tank and outboard tanks. Vent valves connect the outboard tank valves to the center tank vent valve and are led to the flush type vent outlet on the lower surface of wing between spars, near the wing tip. This vent outlet is designed to provide slight pressure to fuel tanks, preventing obstruction by icing.

The vent valve is designed to operate during climb or descent of the aircraft by corresponding float movement to the change of the fuel level. Two relief valves are provided on the center fuel tank which are connected to the vent line. These valves are designed to open when the pressure reaches and/or exceeds 2.0 psi (0.14 kg/cm^2) in the tank to allow the air and/or fuel into the vent line to prevent pressure buildup in the tanks.



1. Check valve
2. Adapter
3. Elbow
4. "O" ring

Fig 7 - 8 Installation of manifold



1. Vent valve
2. Gasket
3. Vent valve body
4. Vent line

Fig 7 - 9 Center tank vent valve

3.1 CENTER TANK VENT VALVE

The center tank vent valve is float-type and located at the access door on the top of the middle part of the center fuel tank. It is designed to close the vent by a float when the fuel level rises.

3.1.1 REMOVAL AND INSTALLATION (See Fig 7-9)

(1) Drain the integral tank.

(2) Remove the L.H. and R.H. access panel on the top of the center wing.

(3) Remove four screws which attach the vent valve cover and vent valve body, and disconnect the vent lines leading to the L.H. and R.H. outboard tanks and also leading to the main vent line.

(4) Cut the lock wire from the valve body and remove the valve with a gasket from the valve body.

(5) Install in reverse sequence of removal.
Take care of the following items when installing.
(a) Installation torque of the vent valve to the body is 150 to 250 in-lbs. (171 to 286 kg-cm).
(b) Installation torque for the screws of the access door is 30 to 35 in-lbs. (34 to 40 kg-cm).

3.2 OUTBOARD TANK VENT VALVE

The outboard tank vent valve is installed on the top of the outboard tank and the same one as the center tank vent valve. It is easily accessible through the access door on the top of the tank.

3.2.1 REMOVAL AND INSTALLATION (See Fig 7-10)

To remove and install the vent valve, proceed as follows.

(1) Open the access panel (W.STA1250~W.STA1600) on the top of the outboard tank.

(2) Disconnect the vent lines.

(3) Remove four bolts which secure the valve and fitting.

(4) Remove the valve and fitting from the bracket and remove the valve with "O" ring from the fitting.

(5) Install in reverse sequence of removal. Installation torque of the valve to screw into the fitting is 390 to 430 in-lbs. (445 to 491 kg-cm).

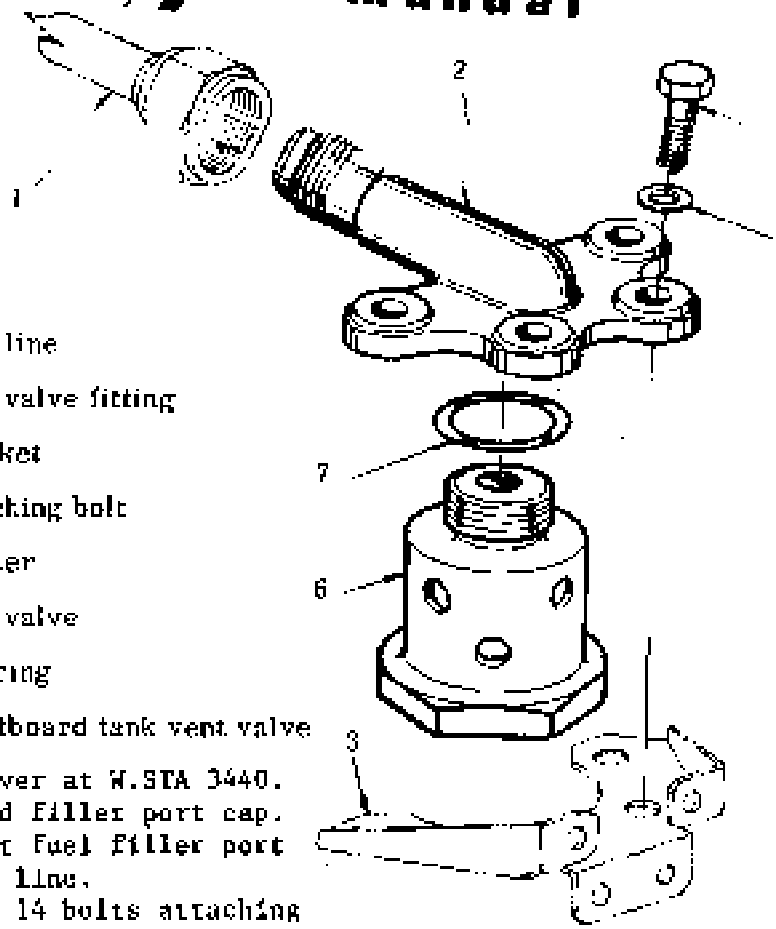
3.3 OUTER TANK VENT VALVE (See Fig 7-11)

The outer tank vent valve is a ball-type valve installed on the top of the outer
It is designed to close the vent by a ball-type float when the fuel level rises.
It is easily accessible through the access door on the top of the tank.

3.3.1 REMOVAL AND INSTALLATION

To remove or install the vent valve, perform the following procedures with aircraft power off.

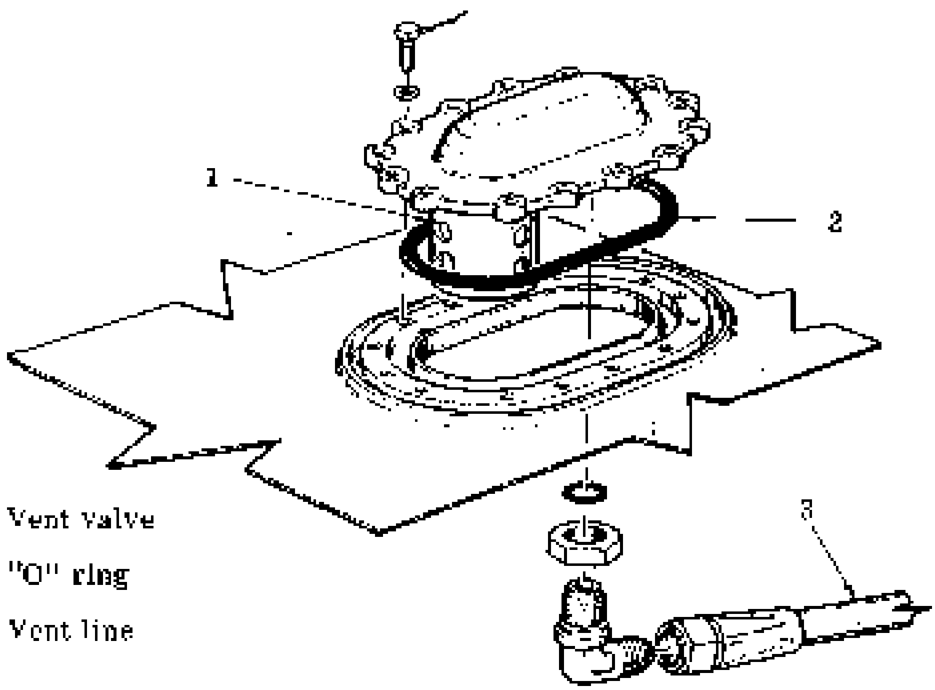
(1) Drain the outer tank.



1. Vent line
2. Vent valve fitting
3. Bracket
4. Attaching bolt
5. Washer
6. Vent valve
7. "O" ring

Fig 7-10 Outboard tank vent valve

(2) Open filler port cover at W.STA 3440.
(3) Remove dip stick and filler port cap.
(4) Loosen hose clamp at fuel filler port and disconnect vent line.
(5) Cut safety wires of 14 bolts attaching vent valve, remove bolts, and remove vent valve with vent line and connector hose.
(6) Install in reverse sequence of removal.



1. Vent valve
2. "O" ring
3. Vent line

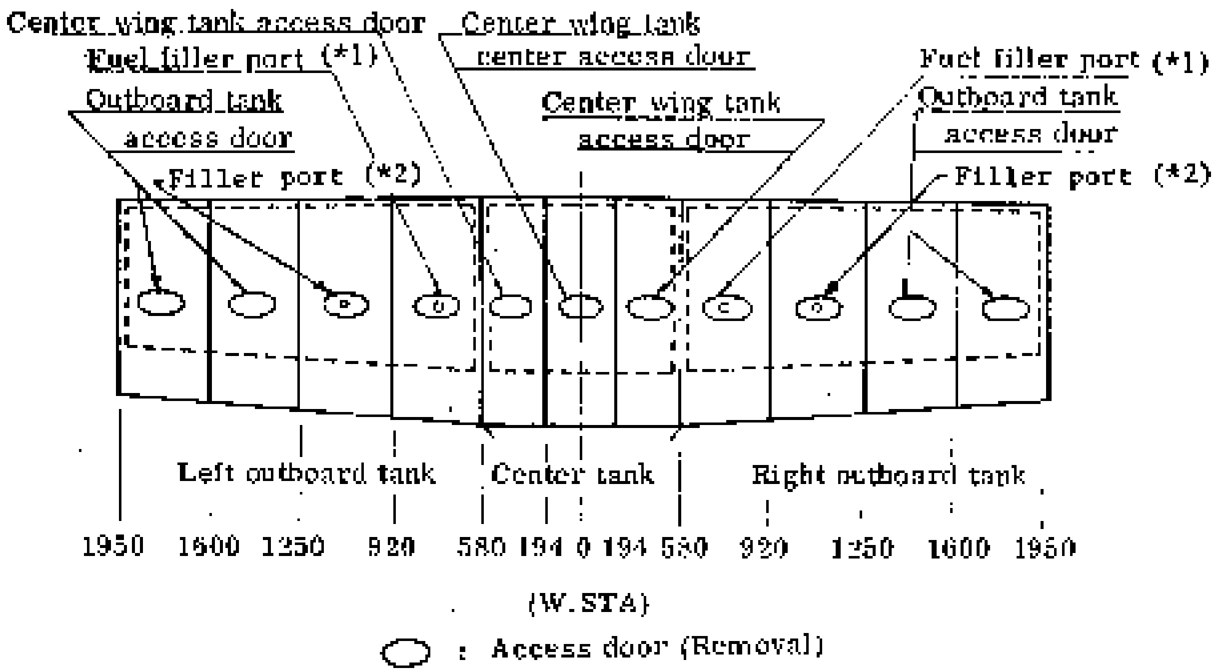
Fig 7-11 Outer tank vent valve assembly

4. INTEGRAL TANK (See Fig 7-12)

Space between the two spars of wing inboard section inboard of nacelles from W.STA 1950 L.H to W.STA 1950 R.H constitutes an integral fuel tank. The tank is partitioned into three sections, center tank, L.H outboard tank and R.H outboard tank by bulkheads at W.STA 580. Each section is interconnected by vent system. For repair of integral tank, see Chapter 2 Section 6 of this manual.

NOTE

Access panel should be installed in correct position. Incorrect installation may cause fuel leakage.



*1 Aircraft S/N 652SA, 661SA, 697SA thru 724SA
*2 Aircraft S/N 725SA and subsequent

Fig 7-12 Location of integral tank access door

5. TIP TANK FUEL SYSTEM

The tip tank fuel system consists of two 93 U.S. gallon tip tanks located on the wing tips, pressure air line for pressure supply of fuel, air shutoff valve, air pressure regulator, check valve, vent disc, fuel line for transferring fuel to center tanks, fuel shutoff valve, and fuel level control valve. The fuel in tip tanks is transferred to the center tank automatically by air pressure in accordance with the fuel level of the center fuel tank. The screen and orifice are installed in the line for air conditioning system.

5.1 FUEL TRANSFER SYSTEM

Close the TRANS CONT circuit breaker and place the left or right TRANSFER switch to the TIP MANUAL position. When the fuel quantity remaining in each tank is over 3 gallons, the pressured air taken from the air conditioning system flows through the air shutoff valve and the air pressure regulator where it is regulated to 4 to 5.5 psi (0.28 to 0.39 kg/cm^2), thus adding air pressure to the tip tanks. The pressurized fuel from each tip tank passes through a fuel line, strainer, fuel shutoff valve and joins at the center main tank inflow port. The fuel level control valve acts when the fuel level in the center tank drops below the operating level of the fuel level control valve and allows fuel to flow into the tank. The valve closes when the fuel level rises over the operating level. The fuel low level switch, in each tip tank, operates when the tip tank fuel level is 3 gallons or less. This low level switch operates the TIP LOW LEVEL annunciator light on the lower engine instrument panel and also energizes the air pressure shutoff valve to the closed position.

5.2 AIR SHUTOFF VALVE (See Fig 7-13)

The air shutoff valve, which is solenoid-operated, is designed to operate in 0.05 second. It is located in the wing-fuselage fillet on the tip of the top of the right-hand side of the fuselage. When both the right and left low level switches of the tip tanks operate, the valve automatically closes.

5.2.1 REMOVAL AND INSTALLATION

(1) Remove center wing front access panel.

(2) Disconnect electric connector.

(3) Disconnect air lines on both sides of shutoff valve and remove the valve.

(4) Install in reverse sequence of removal.

5.3 AIR PRESSURE REGULATOR

The pressure regulator is located on the same bracket as the air shutoff valve.

This valve is the same as the fuel shutoff valve used in the main fuel feed system. The valve is opened when circuit breaker TRANS CONT is closed and TRANSFER switch is placed to the TIP MANUAL position. The valve is closed when the switch is placed to the OFF position. This valve is accessible through the access door on the wing leading edge lower surface inboard of the nacelle.

5.3.1 REMOVAL AND INSTALLATION (See Fig 7-13)

To remove and install the air pressure regulator, proceed as follows.

(1) Remove center wing front access panel.

(2) Disconnect lines at both ends of the regulator and disconnect drain line.

(3) Remove three screws and spacers, and regulator.

(4) Install in reverse sequence of removal.

5.4 FUEL SHUTOFF VALVE

This valve is the same as the fuel shutoff valve used in the main fuel feed system. The valve is opened when circuit breaker FUEL TRANSFER CONT is closed and FUEL TRANSFER switch is placed to AUTO (when fuel remains in tip tank) or TIP MANUAL. The valve is closed when the switch is placed to the OFF position.

This valve is accessible through the access door on the wing leading edge lower surface inboard of the nacelle.

5.4.1 REMOVAL AND INSTALLATION (See Fig 7-14)

(1) Remove access door on wing leading edge lower surface inboard of the engine nacelle.

(2) Disconnect electric connector and bonding jumper.

(3) Disconnect fuel lines at both sides of shutoff valve.

(4) Remove attaching bolts for the valve and take the valve out of bracket.

(5) Install in reverse sequence of removal.

Fig 7-13 Installation of air shutoff valve and air pressure regulator

Fig 7-14 Installation of fuel shutoff valve

5.5 FUEL LEVEL CONTROL VALVE

The fuel level control valve, located in the center tank, consists of a float, pilot valve, and diaphragm.
The pilot valve is interlocked with the float. The diaphragm chamber, the top of which constitutes the wall, is connected to the pilot valve chamber. The bottom, formed by the diaphragm, is provided with a small hole in the center. The fuel outlet is closed by the seal of the diaphragm.

5.5.1 REMOVAL AND INSTALLATION (See Fig. 7-15)

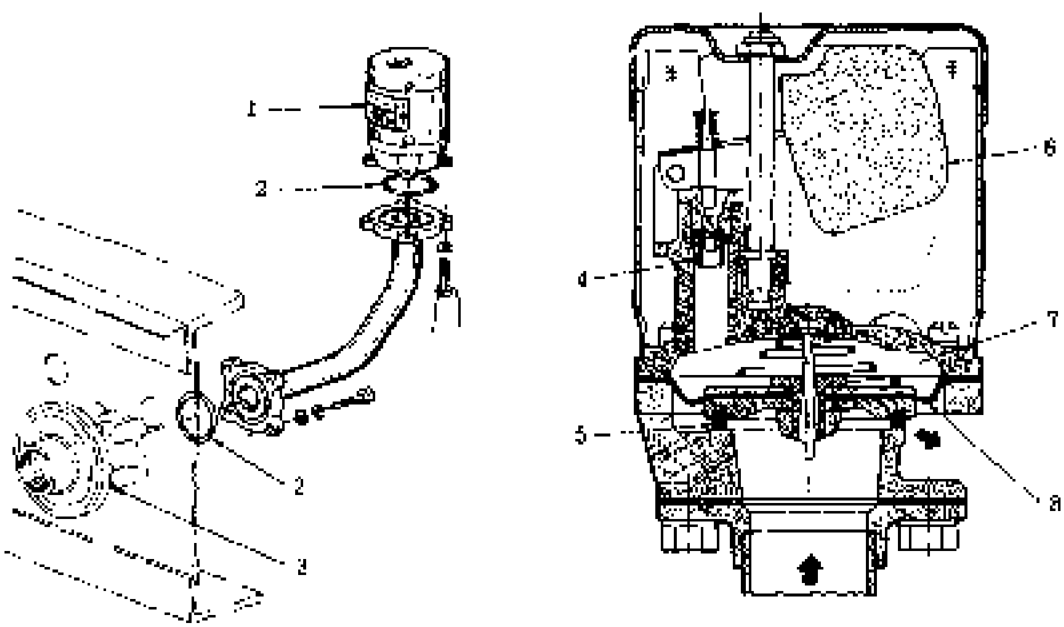
To remove the fuel level control valve, proceed as follows:

(1) Drain fuel system.

(2) Open access door on top of right-hand center tank.

(3) Disconnect fuel line by removing four bolts and take out the valve.

(4) Install in reverse sequence of removal.



1. Fuel control valve	3. Adapter	5. Seal	7. Diaphragm chamber
2. O-ring	4. Pilot	6. Float	8. Diaphragm

Fig. 7 - 15 Installation of fuel control valve

Take care of the following items when installing.

(a) Installation torque of four bolts connecting the fuel level control valve is 10 ~ 14 in-lbs. (11.4 to 16.0 kg-cm).

(b) Install O-ring, lubricating with petrolatum (VV-P-236).

5.6 SNIFFLE VALVE (See Fig 7-16)

The relief valve operates at above 7 psi(0.49 kg/cm^2) of tank inner pressure to let pressure out into the atmosphere, and at -0.7~-1.2 psi(-0.05~-0.08 kg/cm^2), to supply outside air into the tank. The valve is screwed in on the tank wall and permits removal and installation directly from outside. Installation torque is 300~400 in-lbs(346~461 kg-cm).

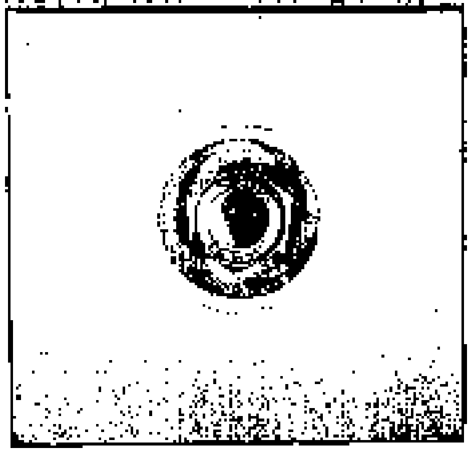


Fig. 7 - 16 Sniffle valve

5.7 TIP TANK

The tip tanks are made of alluminum alloy and each tank is installed to the wing tip with two brackets. Each tank is divided into three parts: front, center and rear. These three parts are connected to one another with 20 bolts at each seam. There is a vacant space in the tip of the front and rear to prevent tank damage from lightning strike.

Two fins are provided on the tank for improved flying characteristics. An access hole is provided on the top of the front section. At the center of the tip tank, the fuel quantity gauge tank unit is installed, by which fuel quantity is transmitted to the dual pointer indicator in the cockpit, indicating fuel quantity of the tank.

Tip light and sniffle valve are located on outer side of each tip tank.

The pressure air line and fuel feed line are connected with hoses to the counterpart lines on the airframe. The area between the tip tank and the wing is faired with fillets. The drain valve is located on the bottom of the tank. On the bottom inside of the tank is a low-level switch, which automatically operates at 3 gallons of remaining fuel to close the air shutoff valve.

On the top at the rear of the tank is a fuel filler port.

Connection and separation of the three sections of the tip tank can be accomplished through access doors on the top of each section.

NOTE

When work is done in the tank, special attention must be paid not to apply strong force on tubing and conduit, etc.

5.7.1 REMOVAL AND INSTALLATION

To remove the tip tank, proceed as follows.

(1) Push sniffle valve to relieve pressure inside the system.

WARNING

i High pressure may be present in the system and, therefore, be sure to relieve pressure prior to tank removal.

ii Do not stand directly in front of the sniffle valve when releasing pressure, due to possible fuel ejection through the valve.

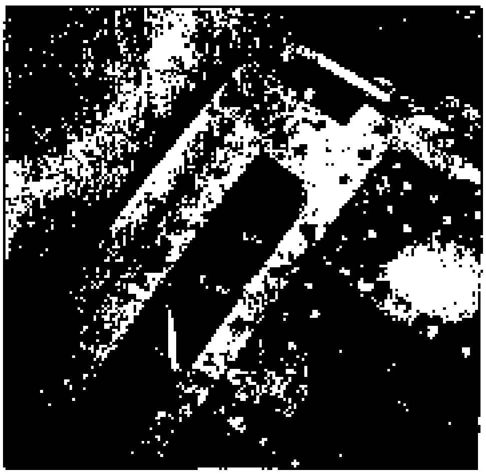


Fig. 7 - 17 Remove fillet and wire at terminal block

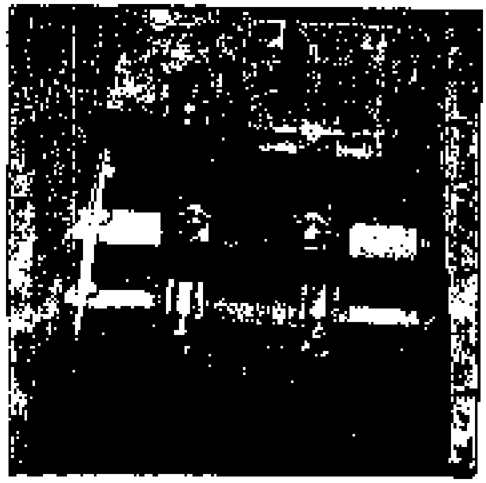


Fig. 7 - 18 Pull out air pressure line and fuel transfer line

(2) Drain fuel.

(3) Remove fillet.

(4) Disconnect electric wires at terminal block, see Fig 7-17.

(5) Loosen hose clamps of pressure air line and fuel transfer line, and then pull hoses off, see Fig 7-18.

(6) Remove three attaching bolts and then remove tip tank carefully.

(7) Install in reverse sequence of removal.

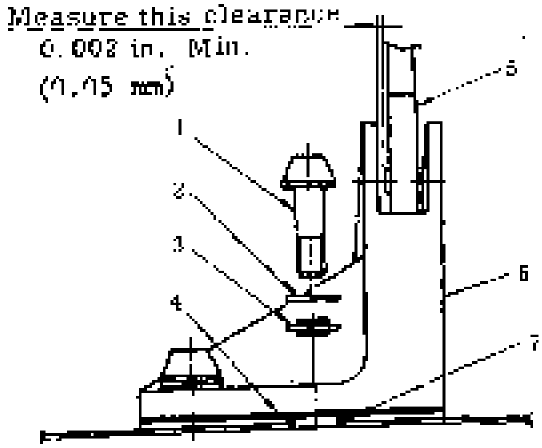
5.7.2 NOTICE TO INSTALL TIP TANK

(1) Tighten hand-tight the bolts which connect wing side rear fitting to tip tank rear bracket after applying MIL-G-21164 grease on the bolt shank. Apply lock wire to prevent bolt from rotation.

(2) Apply MIL-G-21164 grease on the bolt shank. Make sure that the clearance between wing side fitting and tip tank bracket is less than 0.01 in. (0.25 mm), and then apply 270 to 300 in-lbs (309 to 343 kg-cm) torque.

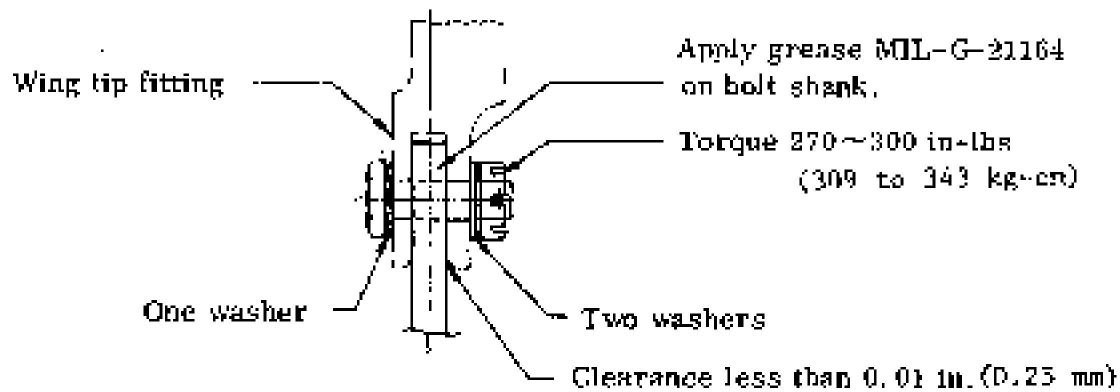
(3) Make sure that the clearance between wing side rear fitting and tip tank rear bracket is more than 0.002 in. (0.05 mm) after installation, see Fig 7-19.

(4) Installation torque of hose clamps used in pressure air line and fuel transfer line is 25 in-lbs (28.8 kg-cm). Check the bolts for torque after installation or flight, and retighten if necessary.

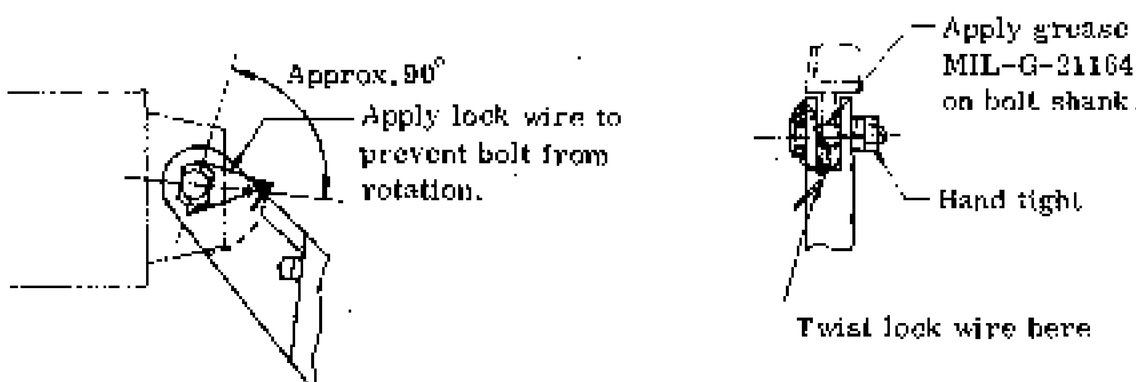


1. Bolt (MS20006-12)
   Installation torque value
   160 ~ 190 in-lbs (183 to 217 kg-cm)
2. Washer (MS20002 C6)
3. Sealing washer (2230-6)
4. Spacer (022A-48454)
5. Fitting-wing side (010A-12137)
6. Bracket-tank side (016A-48204)
7. O-ring

Fig. 7 - 19 Details of tip tank rear joint



FRONT SPAR SIDE



REAR SPAR SIDE

5.7.3 TIP TANK LEAK TEST

When a part in the tip tank is replaced, work is done in the tip tank or strobe light is replaced, perform the following leak test and make sure of no leak at the relative point.

5.7.3.1 When tip tank is not installed on the airplane:

(1) Plug fuel transfer line.
(2) Connect pressure source with pressure air line.
(3) Clog sniffle valve with tape.
(4) Apply air pressure of 7.5 psi +0.05/-0.0 on the tank.
(5) Check for leaks.

5.7.3.2 When tip tank is installed on the airplane:

(1) Remove the access panel of the center wing leading edge, disconnect tip tank pressure air line at tee and connect pressure source with it.

5.8 OPERATIONAL CHECK AFTER TIP TANK INSTALLATION

Supply fuel and accomplish operational check of the system in accordance with the following procedures after removal and installation of tip tank.

(1) Close circuit breaker TIP FUEL QTY and make sure that the tip tank fuel quantity indicator points correctly.

(2) Close circuit breaker NAV LIGHT and turn the navigation light switch NAV in the overhead switch panel to ON. Make sure that the wing tip light becomes illuminated.

(3) Close circuit breaker STROBE LIGHT and turn strobe light switch STROBE to ON. See that the strobe light becomes illuminated and goes out.

(4) Start the engine and close circuit breaker FUEL TRANSFER CONT and place FUEL TRANSFER switch to TIP MANUAL. Make sure that fuel is transferred normally from the tip tank, in accordance with indication on fuel quantity indicator.

(5) Inspect that pressure air line and joint of fuel line between tank and airframe have no leaks.

5.9 OPERATION CHECK OF TIP TANK FUEL TRANSFER

When major parts of this system (fuel control valve and low level switch) are replaced, check the fuel transfer system as follows:

(1) Feed 30 gallons of fuel to each LH and RH tip tank.

(2) Open center tank RH access door and feed about 47 gallons of fuel. Fuel level at this time is approximately 2.7 inches (69 mm) from the surface of the access door.

(3) Disconnect tip tank air line at the air source side of the air shut-off valve, connect to ground air source and apply 30 psi (2.1 kg/cm^2) of air to the tip tank air pressure system.

(4) Close TRANS CONT circuit breaker and place the TRANSFER switch to the TIP MANUAL POSITION.

(5) Make sure fuel level of center tank rises due to transfer from tip tank and when the level reaches approximately 1.85 inch (47 mm) from the surface of the center tank RH access door, fuel transfer stops.

(6) Operate boost pump and take out fuel from the center tank.

(7) Make sure that when the fuel level of center tank becomes approximately 2.0 inches (51 mm) fuel transfer starts again.

(8) Place TRANSFER CONT switch to AUTO position and repeat steps (5) thru (7) above.

(9) To operate LH and RH tip tank low level switches, continue to transfer fuel until the air shutoff valve for tip tank pressure air line (fuel quantity indicator points at ZERO gallons) closes. While transferring, check difference between LH and RH fuel quantity indicators for excessive fuel transfer imbalance.

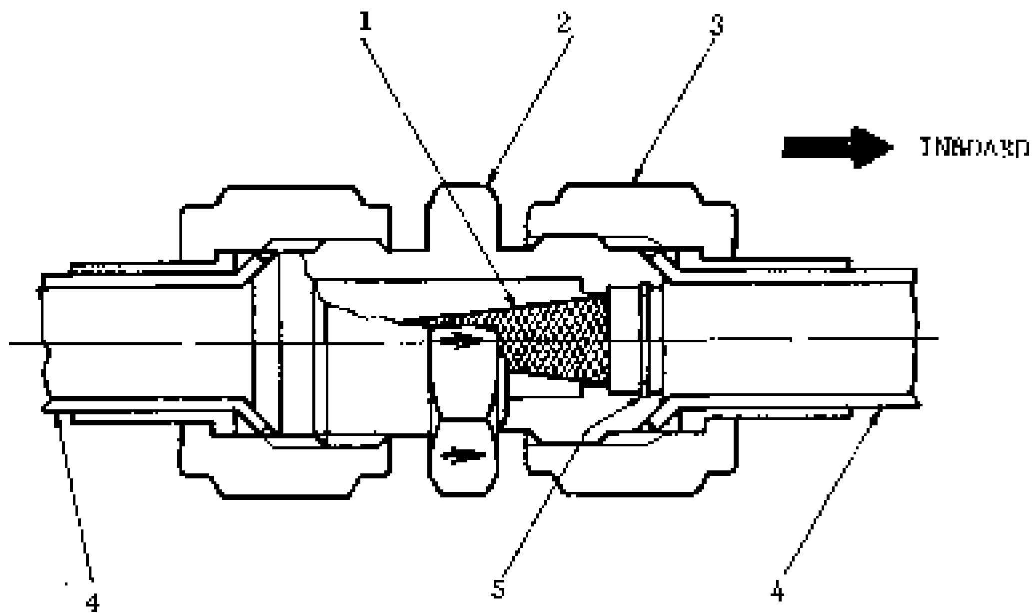
(10) Place the TRANSFER switch to the OFF position and open the TRANS CONT and BOOST PUMP circuit breakers.

(11) Check the system for fuel leaks.

(12) Install access panel after checking that no foreign substance is mixed in center tank fuel. Disconnect ground air source and connect pressure air line for tip tank.

5.10 STRAINER (See Figure 7-20)

The strainer is a screen of #65 mesh made from corrosion resistant steel and installed inside the union on the way of fuel transfer line. Removal and installation of strainer is performed through fuel line access door on outside nacelle under wing.
To install strainer assembly, pay attention to the arrows on the union. Proper installation is shown by the arrows pointing inboard.



1. Strainer 2. Union 3. Nut
4. Tube 5. Strainer Assembly 6. Tube

Strainer Assembly
Fig. 7-20

6. OUTER TANK FUEL SYSTEM

6.1 GENERAL DESCRIPTION

The outer tank fuel system consists of two 15 gallon metal tanks, transfer pump, fuel low pressure warning switch, check valve, vent valve all of which are installed in the outer wing, and the tubing, hose, dip stick, and fuel level control valve.

Fuel in the outer tank is forced through a strainer and the tip tank fuel transfer line to the center tank by means of a transfer pump. The fuel is controlled by a fuel level control valve installed in the center tank. Fuel in the outer tank is not used until the fuel in the tip tank is exhausted.

6.2 OUTER TANK (See Fig 7-21)

The outer tank is made of aluminum alloy and has a capacity of 15 U.S. gallons. The tanks are located between the forward and rear spars in the LH and RH wings between W STA 2590 and 4350, and are fastened to the wing with two bands. The tank is provided with a pump compartment enclosed by bulkhead to prevent fuel from flowing in reverse direction allowing the fuel in the outer tank to be used effectively. A vent valve is provided on top of the tank in the center for ease of access through access door that is located on top of the outer wing as illustrated in Fig 7-22.

6.2.1 REMOVAL AND INSTALLATION

(1) Drain outer tank.
(2) Remove access doors between W STA 1950 and 2550. Remove the hose from the fuel transfer line out-going from the outer tank.
(3) Remove outer wing in accordance with Chapter III, Para 2.3 (1)-(13).
(4) Remove access doors from top of outer wing at W STA 2910 to 3350 and at W STA 3950 to 4150.
(5) Remove turnbuckle and remove attaching band.
(6) Open filler port cover on top of outer wing at W STA 3350 to 3550. Remove dip stick through filler port adapter.
(7) Remove cap assembly and rubber seal from filler port.
(8) Remove cap from filler port.
(9) Remove vent valve in accordance with Para. 3.3.1.
(10) Remove access door from bottom of outer wing leading edge at W STA 2590 to 2910. Remove hose union from fuel low pressure warning switch sensing line and electric connector.
(11) Remove electrical wiring of transfer pump from terminal block.
(12) Remove transfer pump in accordance with instructions given in Para 6.3.1.
(13) Remove access door from bottom surface of outer wing at W STA 4450. Remove hose from vent line.
(14) Remove tank from outer wing.
(15) Install in reverse sequence of removal.

1. Fuel transfer line
2. Outer tank
3. Bonding wire
4. Attaching band
5. Dip stick
6. Filler port
7. Drain line
8. Vent line
9. Transfer pump
10. Low level warning switch
11. Vent valve
12. Strainer

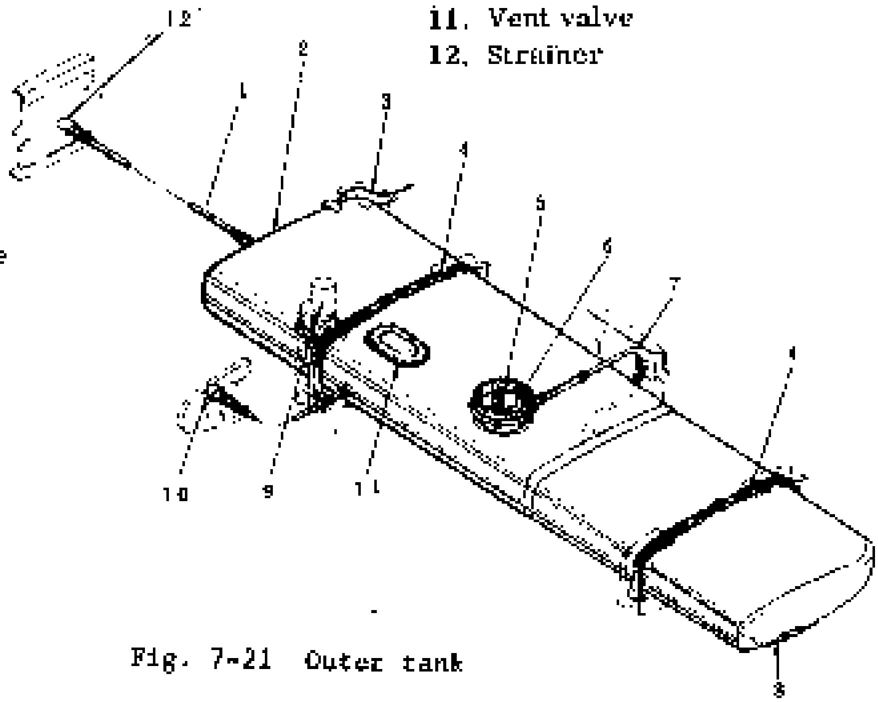
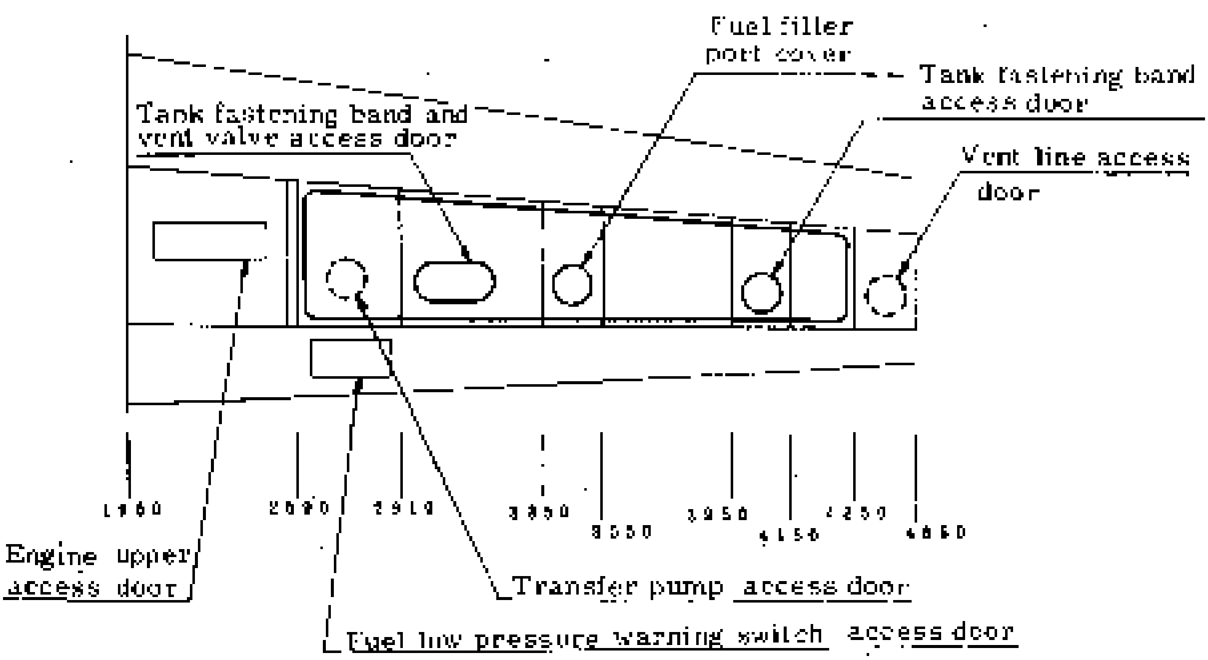


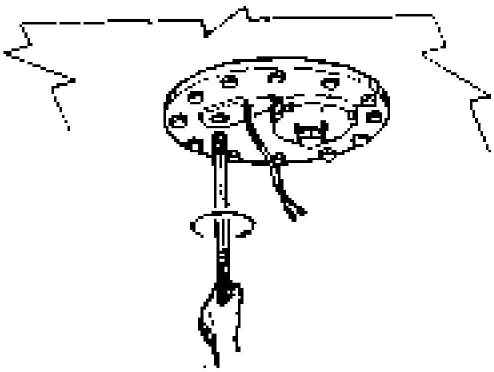
Fig. 7-21 Outer tank



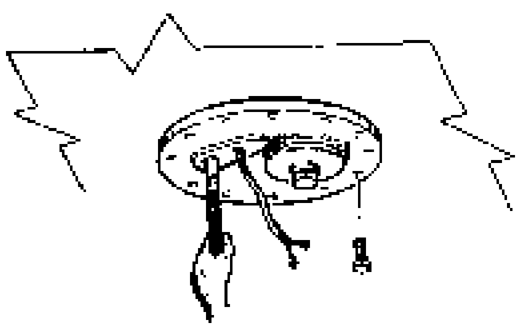
Note 1. The figure shows wing upper surface
2. ◯:Access door on wing upper surface
3. ◌:Access door on wing lower surface

Fig. 7-22 Outer tank access door

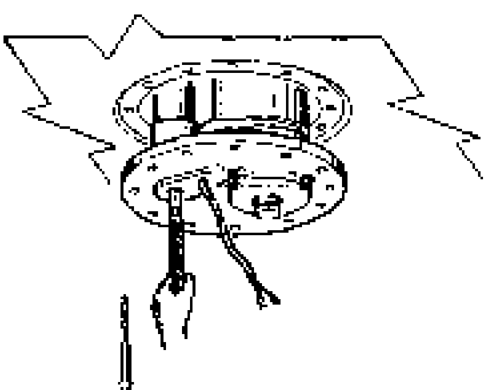
1. Remove a bolt (AN 4H34A) that attaches fuel discharge port fitting to the transfer pump. Screw a removing bar (GSE 016A-99072) in the bolt hole.

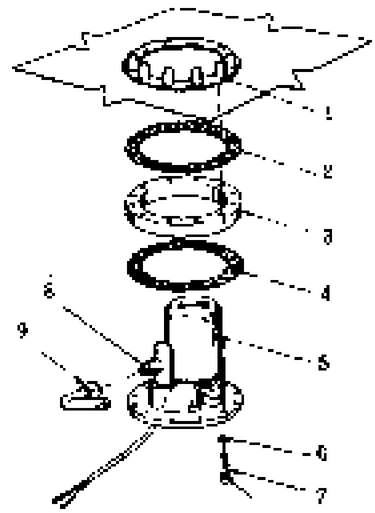


2. Remove 12 bolts (AN4H12A) that hold transfer pump in position.



3. Draw transfer pump out by the removing bar as shown in the figure.





1.	Flange	6.	Washer
2.	Gasket	7.	Bolt
3.	Spacer	8.	Fitting
4.	Gasket	9.	Tube ass'y
5.	Transfer pump	10.	Removing bar (GSE016A-99072)

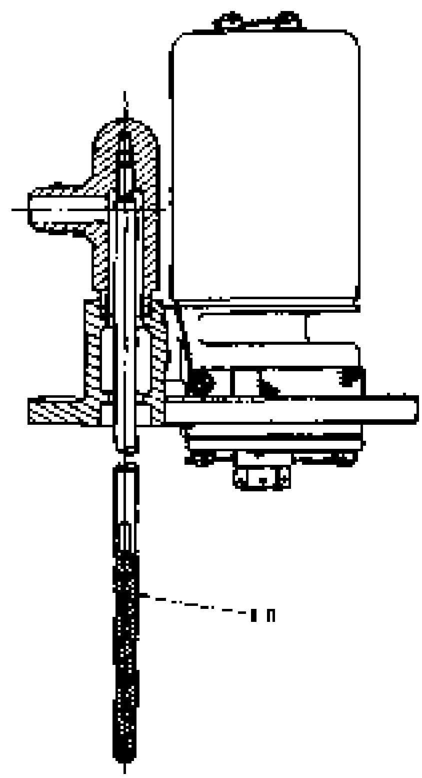


Fig. 7-23 Removal of transfer pump

6.3 TRANSFER PUMP

Applicable to aircraft S/N 652SA, 661SA, 697SA thru 720SA

A constant speed, centrifugal, submerged type transfer pump (Mitsubishi Electric model AP15B) is installed in the outer tank. The pump is rated at 0 to 608 PPH of flow, 2 to 7.5 psi (0.14 to 0.5 kg/cm^2) of discharge pressure, 28 VDC and 3.75 Amps maximum.

Applicable to aircraft S/N 721SA and subsequent

A constant speed, centrifugal, submerged type transfer pump (Airborne model 1G12-3) is installed in the outer tank. The pump is rated at 0 to 625 PPH of flow, 6 to 30 psi (0.42 to 2.1 kg/cm^2) of discharge pressure, 28 VDC and 3.75 Amps maximum.

6.3.1 REMOVAL AND INSTALLATION (See Fig 7-23)

(1) Place FUEL TRANSFER switch on LH instrument panel to OFF, open TRANS CONT circuit breaker, and turn aircraft power OFF.
(2) Remove access door from bottom of outer tank at W STA 2590 to 2910.
(3) Drain outer tank.
(4) Remove access door from bottom of outer tank leading edge at W STA 2590 to 2910.
(5) Remove electric terminal from transfer pump, Fig 7-24.
(6) Remove a bolt that attaches fuel discharge port fitting to the transfer pump. Screw a removing bar (GSE 016A-99072) in the bolt hole. Installation torque of the bolt is 75 to 85 in-lbs (86.4 to 97.9 kg-cm).
(7) Cut safety wires of bolts that hold transfer pump in position. Remove bolts. Installation torque of the bolts is 25 to 35 in-lbs (28.8 to 40.3 kg-cm).
(8) Draw transfer pump out by the removing bar, see Fig 7-25.
(9) Remove transfer pump drain valve Installation torque of the valve is 25 to 35 in-lbs (28.8 to 40.3 kg-cm).
(10) Remove hose fitting from fuel discharge port. Remove transfer pump.
(11) Install in reverse sequence of removal.

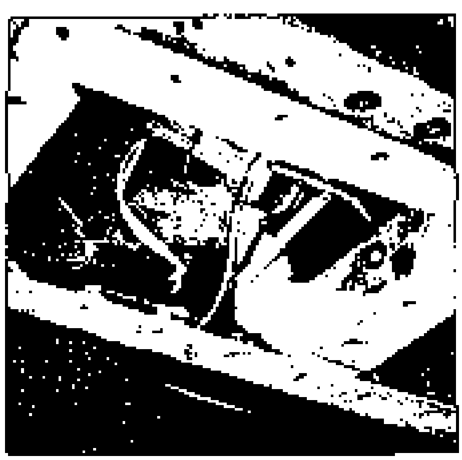


Fig. 7-24 Remove electric terminal from transfer pump

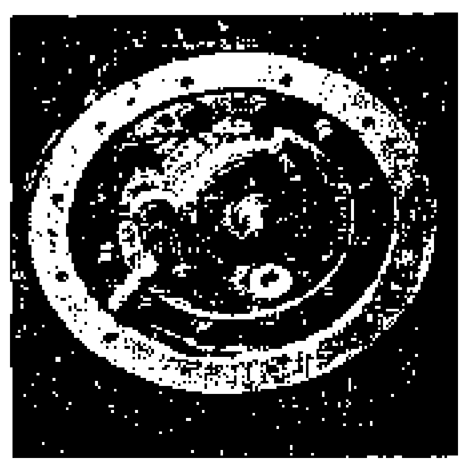


Fig. 7-25 Remove transfer pump

ORIGINAL
As Received By
ATP

NOTE

When reinstalling transfer pump, head the neck of fuel discharge fitting forward.

6.3.2 OPERATIONAL CHECK OF FUEL TRANSFER PUMP

(1) Close left and right circuit breakers FUEL TRANSFER CONT and OUTER FUEL EMP WARN.

(2) Place left and right FUEL TRANSFER switches to AUTO position.

NOTE

Operational check should be performed in condition that both LH and RH tip tanks are empty. When some fuel is in tip tank and low level float switch is operating, transfer pump cannot be operated.

(3) Make sure that about 2 minutes after the switch is placed to AUTO, OUTER FUEL EMP light is illuminated in the moment transfer pump starts to turn and goes out instantly.

(4) Make sure that transfer pump has no abnormal noise.

(5) Place left and right FUEL TRANSFER switches to OFF.

(6) Open left and right circuit breakers FUEL TRANSFER.

NOTE

i Do not operate transfer pump when outer tanks are empty.

ii Replace transfer pump, if fuel leakage from seal drain exceeds 2 cc/hr. (S/N 652SA, 661SA, 697SA thru 720SA)

6.4 FUEL LOW PRESSURE WARNING SWITCH

A fuel low pressure warning switch is provided in the outer wing leading edge at W STA 2950 to sense the pressure of fuel discharge from the transfer pump. The switch operates when fuel pressure falls below 1.7 psi (0.12 kg/cm^2) (*1) or 2.5 psi (0.17 kg/cm^2) (*2) and illuminates OUTER FUEL EMP light on center instrument panel when the pump fails or the tank is empty. The switch is the same type as boost pump warning switch in the center tank, but pressure setting is different (Fig 7-26).

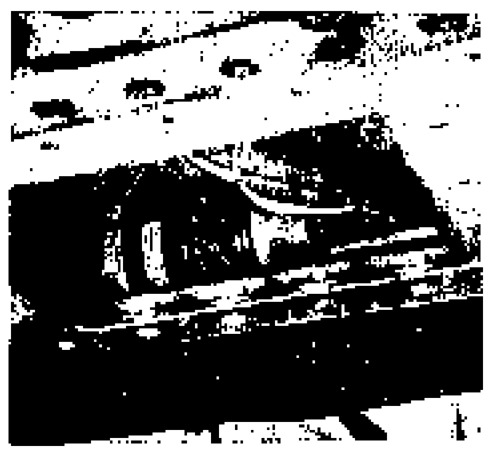


Fig. 7-26 Installation of fuel low pressure warning switch

6.4.1 REMOVAL AND INSTALLATION

To remove or install the fuel low pressure warning switch, perform the following procedures with aircraft electric power off.

(1) Drain outer tank. If fuel in outer tank is below the level of 7 U.S gallons, tank need not be drained.

(2) Remove access door from bottom of outer wing leading edge at W.STA 2590 ~2910.

(3) Remove electric connector of fuel low pressure warning switch and hose from pressure sensing line. Fit a plug AN806-4D or AN806-4 into the hose to prevent fuel leakage.

(4) Remove switch from bracket.

(5) Install in reverse sequence of removal.

*1 Aircraft S/N 652SA, 661SA, 697SA thru 720SA

*2 Aircraft S/N 721SA and subsequent

6.5 DIP STICK (See Fig. 7-27)

Outer tank is not provided with fuel quantity indicator tank unit. Instead, a dip stick is used for checking fuel level.

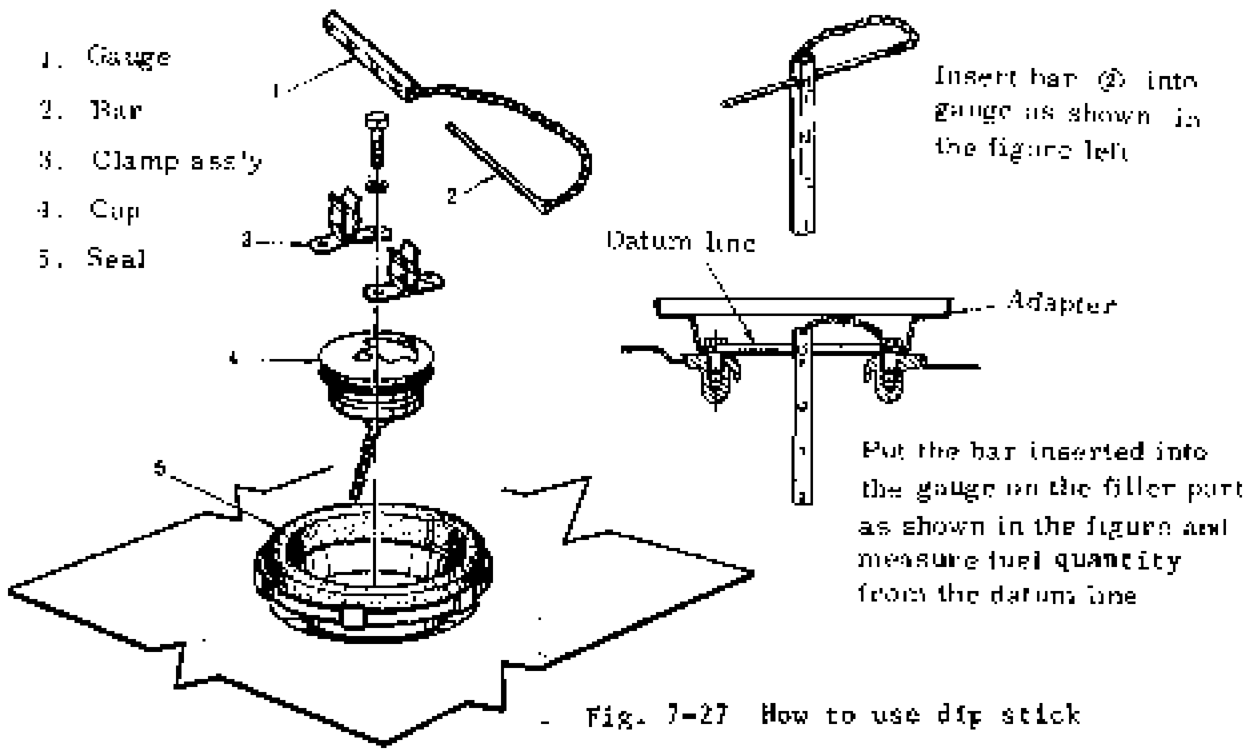


Fig. 7-27 How to use dip stick

6.6 OPERATIONAL CHECK OF OUTER TANK FUEL TRANSFER SYSTEM

Operational check of outer tank fuel transfer system is accomplished as follows.

(1) Fill LH and RH outer tank with 10 U.S. gallons of fuel each.
(2) Open RH access door of center tank. Refuel the tank to a level of approximately 2 in. (50 mm) below access door.
(3) Turn aircraft electric power ON. Close circuit breaker FUEL TRANS CONT and OUTER FUEL EMP WARN.
(4) Place LH and RH FUEL TRANSFER switch to AUTO.

NOTE

Operational check should be performed in condition that both LH and RH tip tanks are empty. When some fuel is in tip tank and low level float switch is operating, transfer pump can not be operated.

Make sure that about 2 minutes after the switch is placed to AUTO, OUTER FUEL EMP light is illuminated at the moment transfer pump starts to turn and goes out instantly.

(5) Bleed fuel from center tank at a rate of approximately 2.4 gal/min.
(6) LH and RH OUTER FUEL EMP lights are illuminated, then place FUEL TRANSFER switches to OFF. Stop draining center tank.
(7) Open circuit breaker FUEL TRANSFER CONT and OUTER FUEL EMP WARN and turn aircraft power OFF.
(8) Ensure that LH and RH outer tanks are empty, seeing through filler ports.

NOTE

i If EMPTY light is not momentarily illuminated with FUEL TRANSFER switch in AUTO, it is usually due to poor contact or failure of lamp.

ii If empty light does not go off and remains on, it is usually due to fuel low pressure warning switch being preset at a higher operating point. Replace the switch.

6.7 STRAINER

This is the same strainer as the one for tip tank installed on the way of tip tank fuel transfer line (See Para. 5.10). The strainer is installed on the way of outer tank fuel transfer line and can be removed or installed through the engine upper access door on wing. The strainer assembly should be installed so that the arrows on the union point to the airplaine center line. Installation torque to the boss is 390 to 430 in-lbs. (446 to 491 kg/cm) (Fig. 7-20).

7. FUEL QUANTITY INDICATING SYSTEM

The fuel quantity indicators in the cockpit, indicating in U.S. gallons, indicate the quantity of fuel in the fuel tanks in the wing and in the tip tanks. For this purpose, capacitor-type tank units are employed. For the quantity of fuel in the wing tanks, one fuel quantity indicator is employed, which indicates the total quantity of fuel in the tanks. The quantity of fuel in the tip tanks, however, is indicated by a dual indicator. When the quantity of fuel remaining in the fuel tank in the wing has decreased to 30 ± 5 gallons, the warning light FUEL LOW LEVEL illuminates. For adjusting fuel quantity indicator, see Chapter X, section 5.

7.1 FUEL QUANTITY INDICATOR

The fuel quantity indicators are arranged on the front instrument panel in the cockpit.

7.2 FUEL QUANTITY TANK UNITS

In this aircraft, five capacitor-type fuel quantity tank units are employed: one in the center tank, one each in outboard tanks, and one in each tip tank. Each tank unit is installed with bolts to a bracket attached to the tank rib.

7.2.1 REMOVAL AND INSTALLATION OF CENTER TANK UNIT (Fig. 7-28)

(1) Drain integral tank.

(2) Remove center tank upper R.H and center access panels.

(3) Through RH access panel opening, disconnect vent valve lines from the vent valve/panel assembly, and disconnect electric connector.
(4) Remove bolts attaching tank unit to bracket through center tank upper access panel, and remove tank unit.
(5) Install in reverse sequence of removal.

NOTE

When removing unit from tank, caution should be exercised not to damage inner wall of tank.

7.2.2 REMOVAL AND INSTALLATION OF OUTBOARD TANK UNIT

(1) Drain integral tank.
(2) Open upper access door W STA 1250 to 1600 of outboard tank.
(3) Remove electric connector from tank unit.
(4) Remove bolts which attach tank unit, and take out tank unit through access door.
(5) Install in reverse sequence of removal. Installation torque of access door attaching screws is 30 to 35 in-lbs (35 to 40 kg-cm).

7.2.3 REMOVAL AND INSTALLATION OF TIP TANK UNIT (See Fig 7-29)

(1) Drain tip tank.
(2) Remove tip tank upper center access door.
(3) Disconnect the electric connections from tank unit and remove bolts which attach tank unit and remove the unit through access door.
(4) Install in reverse sequence of removal.

NOTE

When work is done in the tank, special attention must be paid not to apply strong force on tubing, conduit, etc.

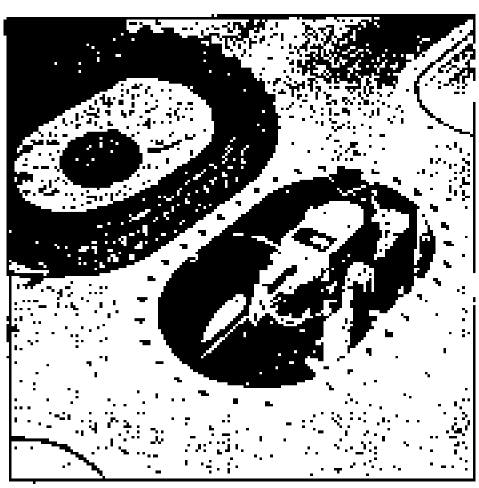


Fig. 7-28 Center tank fuel quantity indicator tank unit

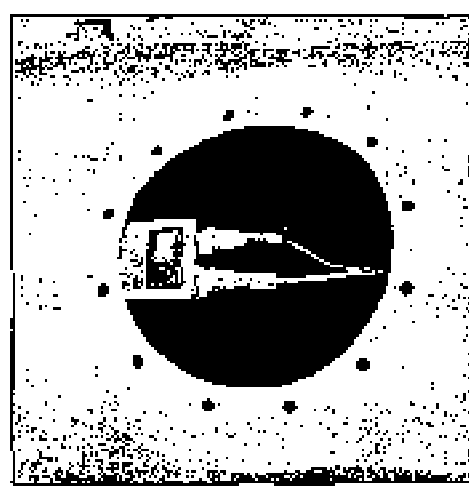


Fig. 7-29 Tip tank fuel quantity indicator tank unit

7.2.4 INSPECTION AND CLEANING OF TANK UNIT FOR FUEL QUANTITY INDICATING SYSTEM

(1) Defuel the wing integral tank and the LH and RH tip tanks.

(2) Remove access panels from the upper wing and upper tank surface to gain access to the tank units and connectors.

(3) Check the connector for dirt and other contamination which might affect continuity. Clean the connectors in accordance with Fig 7-29A and/or Fig 7 29B as required. Perform electrical resistance check in accordance with Fig 7-29C after inspection and cleaning in the location noted below;

a. Between tank unit of main integral outboard tank (or connector on tank wall) and electrical ground.

b. Between tank unit of LH tip tank (or connector on tank wall) and electrical ground; and,

c. Between tank unit of RH tip tank (or connector on tank wall) and electrical ground.

NOTE

Installation screw holes of fuel tank inspection panels may be used for the grounding point.

d. Reinstall the access panels that were removed by step (2) above after completion of inspection and cleaning procedure.

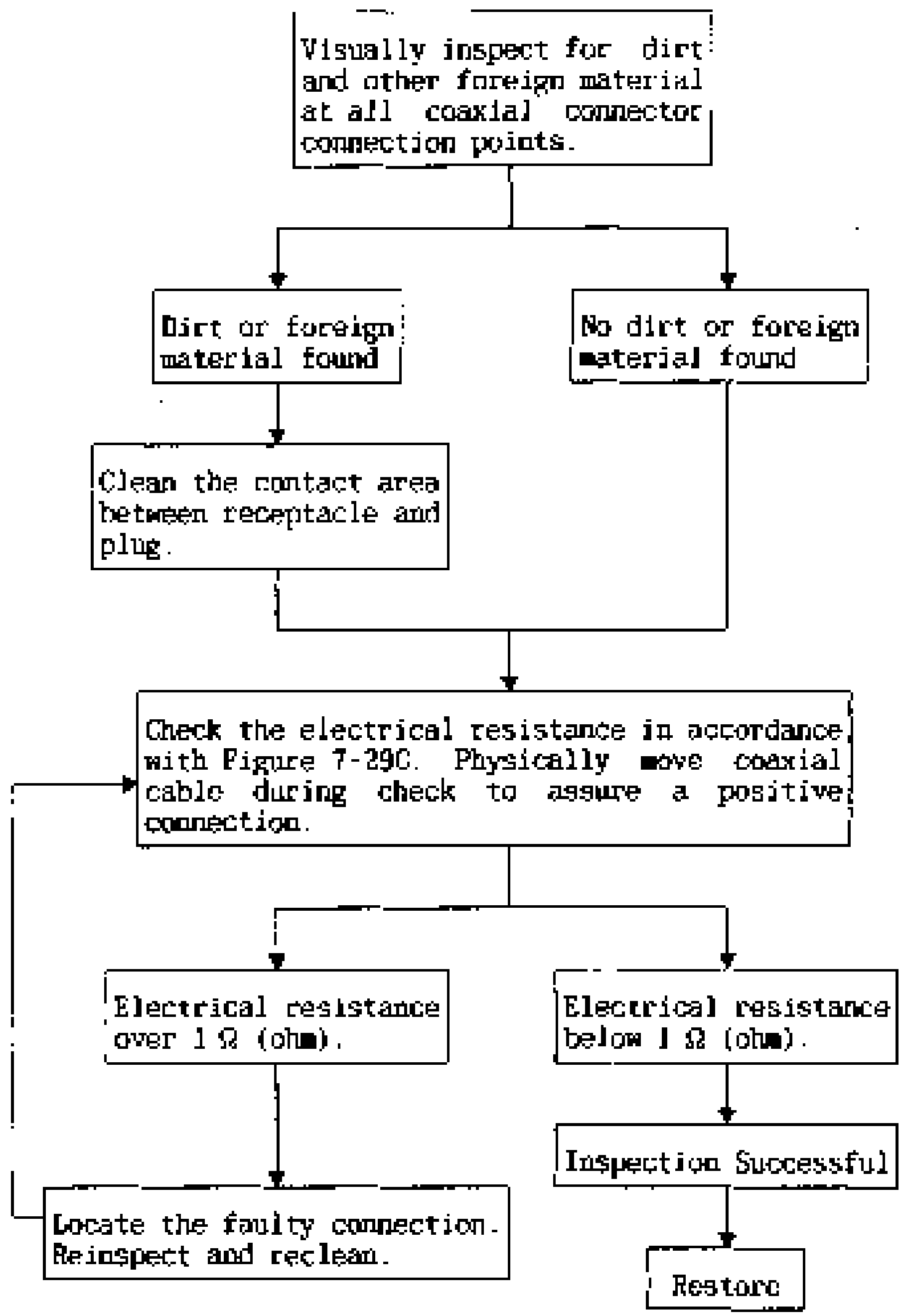
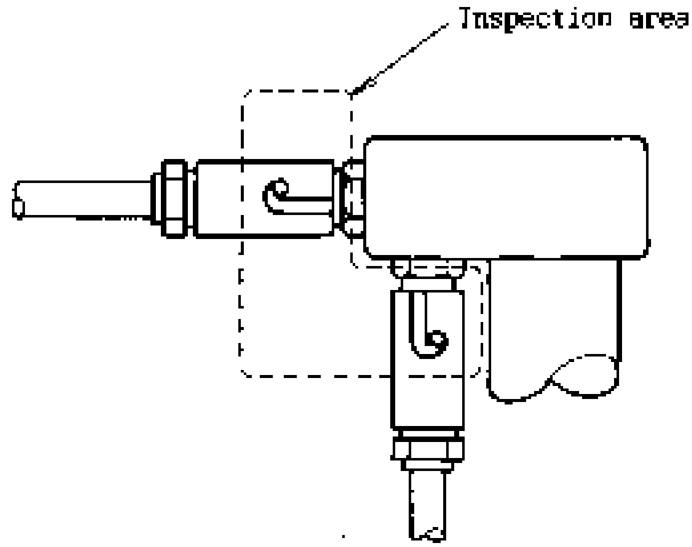
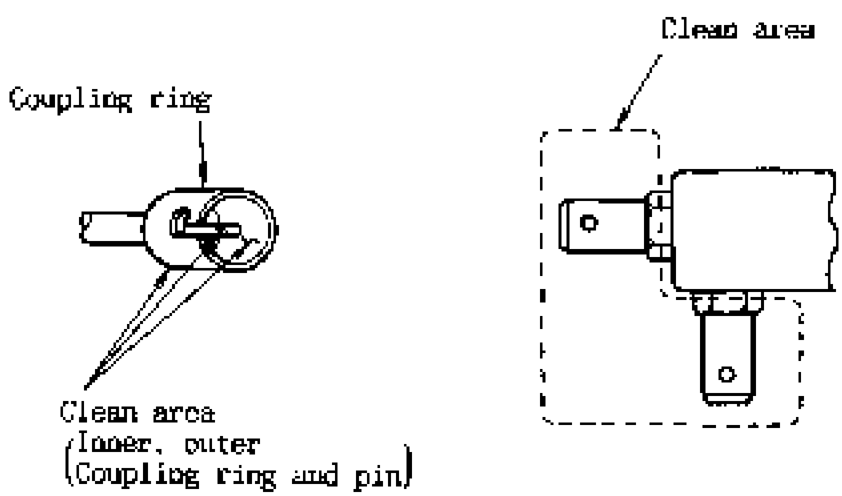


Figure 7-29A Instruction for Inspection and Maintenance

(1) Inspection area of connector (Typical)



(2) Clean area of connector (Typical)



NOTE

Corrosion preventive compounds (MIL-C-81309 Type III) may be used to clean the dirt on the connector. Care must be exercised not to scratch plating using corrosion preventive compounds while cleaning the connector.

Fig 7-29B Inspection and cleaning of connectors

(3) Plastic cover type (Typical)

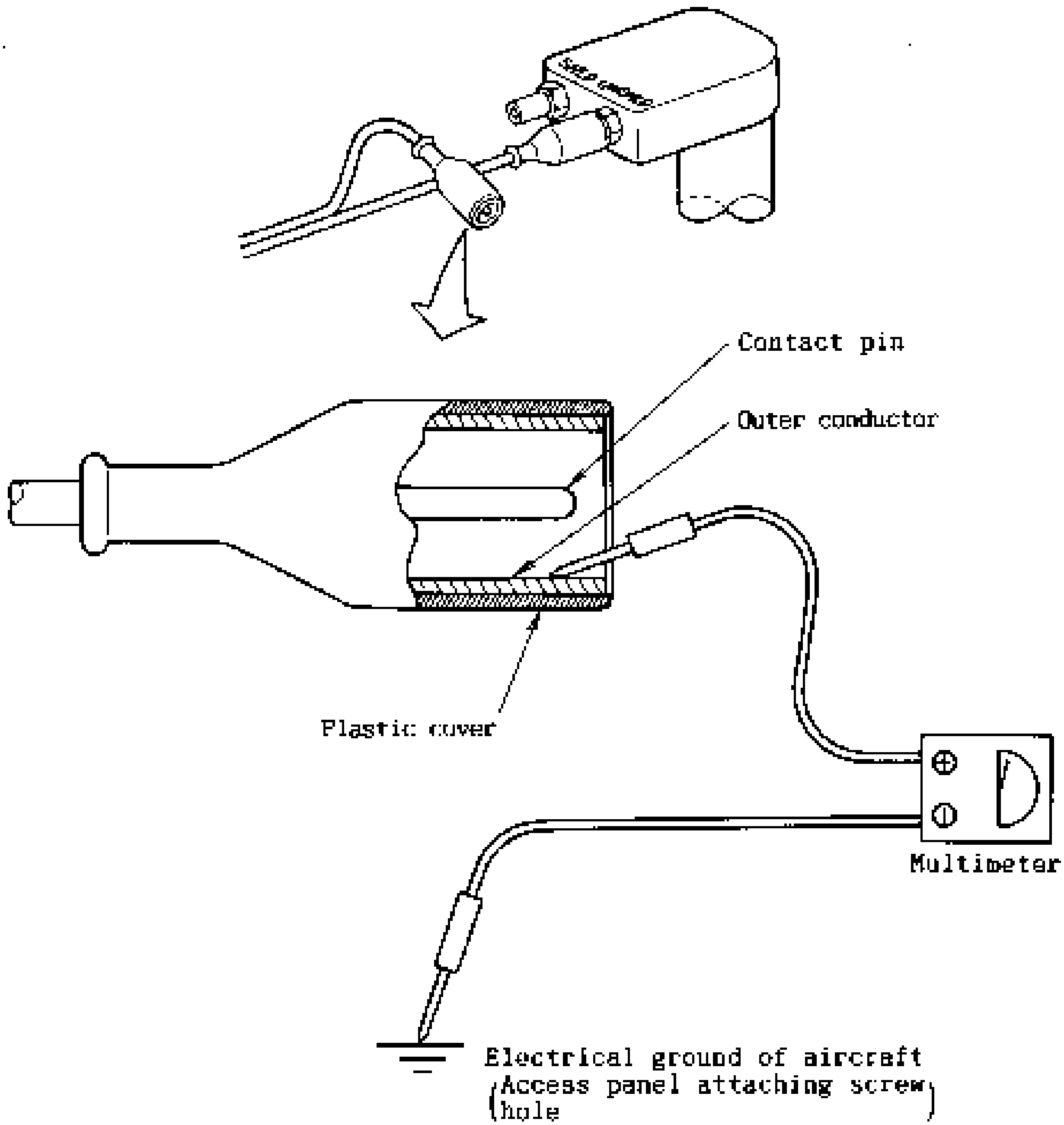


Fig 7-29C Electrical resistance checks (1/2)

(4) Metal bayonet type (Typical)

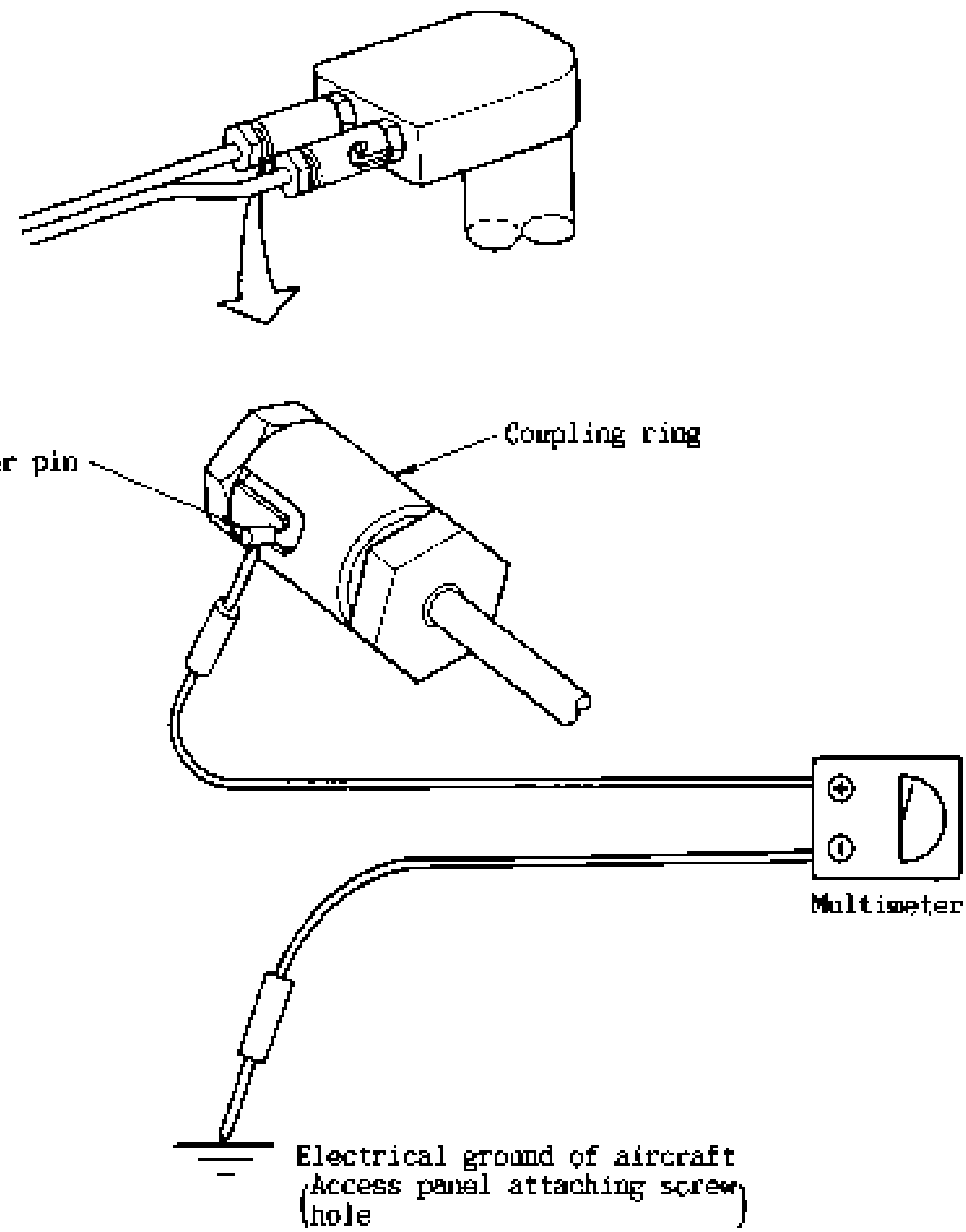


Fig 7-29C Electrical resistance checks (2/2)

8. DRAINING AND REFUELING

8.1 FUEL TANK DRAIN (See Fig. 7-30)

Drain valves are installed on the bottom of each fuel tank: one on either side of the center tank, one each on the outboard tanks and outer tanks, and four near the center of the bottom of each tip fuel tank.

8.1.1 CENTER FUEL TANK DRAIN

Drain holes with drain valves are located on the right rear and left front bottom of the center fuel tank. Drain valves are designed to be operated through the access holes at the wing fillet.

(1) Center fuel tank drain valve.

The valve, line-mount type, is designed to drain fuel by turning handle clockwise with fingers, and provides a lock detent so that draining continues. To remove and install the drain valve, proceed as follows.

(a) Remove wing bottom fillet door.
(b) Disconnect drain lines from valve.
(c) Loosen nuts and remove valve from fitting.
(d) Install in reverse sequence of removal.

8.1.2 OUTER MAIN TANK DRAIN

A drain valve through which the fuel is discharged outside of the aircraft is located on the bottom of each tank. The drain valve can be operated directly from outside of the aircraft.

(1) This valve is a flush type drain valve. When draining fuel, push and turn valve stem 90° with a screw driver fitted in the slot of the valve stem. The drain valve is kept open for continuous draining by a lock detent. Removal is accomplished easily by a spanner (*1), removing bolts and nuts through tank skin (*2).

8.1.3 TIP TANK DRAIN

Four flush-type drain valves are installed on the tank bottom. Its operation is the same as the valve of the outer tank. The valve can be removed by removing four screws (*1), bolts and nuts through tank skin (*2).

8.2 BOOST PUMP DRAIN

The boost pump has a seal and pump drains. The seal drainage is led downward by tubing and discharged from the wing fillet. The pump drainage flows through the drain valve, being led downward by tubing to join the center tank drain. Fuel is discharged through the bulge bottom. Drain valve operation can be performed through the access hole at the wing fillet. To drain the left-hand boost pump, open the far side valve of the two drain valves. To drain the right-hand boost pump, open the near side valve of the two drain valves. Operation, installation and removal of these drain valves are accomplished by the same procedure as the center tank drain valve.

*1 Aircraft S/N 652SA and 661SA

 *2 Aircraft S/N 697SA and subsequent

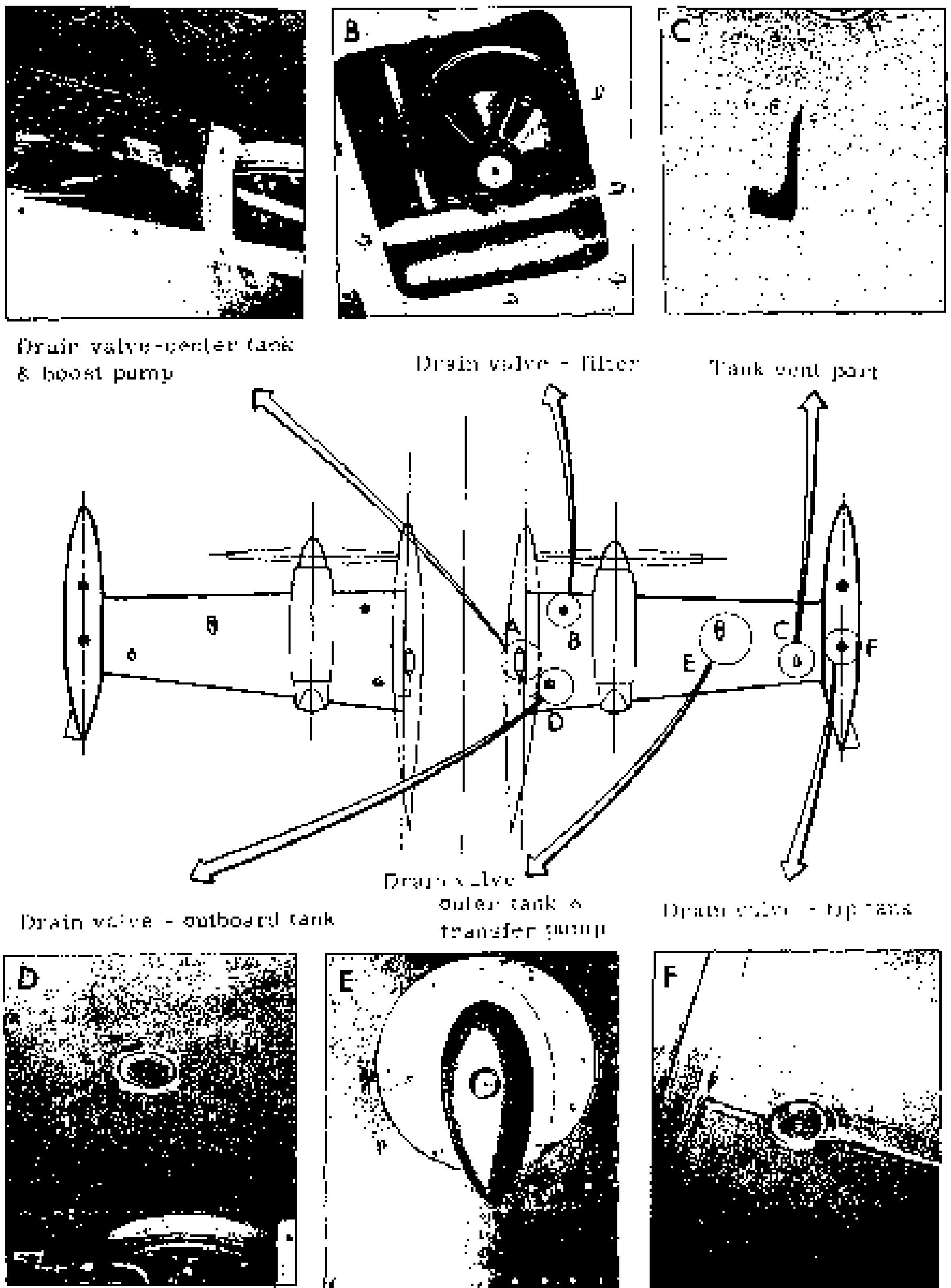


Fig. 7-30 Location of drain valve in fuel system

8.3 TRANSFER PUMP DRAIN

The transfer pump has a seal drain and a pump drain. The seal is open at the bottom of the pump. A hollow stem drain valve is provided in the pump drain port. The valve is opened by inserting a screw driver in the slot of the valve through a small hole in the lower access panel of the tank and by turning it clockwise. The valve locks in OPEN position for continuous draining. The transfer pump drain valves are also used as the outboard tank drain valves. Removal and installation of the valve can be accomplished by using a 3/8 in. square bar of a torque wrench, breaker bar or ratchet drive.

8.4 FUEL FILTER DRAIN

The hollow stem drain valve, the same as the transfer pump, is located in the bottom of the filter, and is opened by inserting a screw driver in the slot of the valve through a small hole in the access panel on the lower side of the inboard leading edge, and by turning it clockwise. The valve locks in OPEN position for continuous draining.
Removal and installation of the valve can be accomplished by the same procedure as the transfer pump. For access, remove the inboard leading edge lower access panel.

8.5 DEFUELING

Rapid draining of a considerable volume of fuel remaining in fuel tanks shall be accomplished in accordance with the following procedures.

(1) Ground aircraft.

(2) Make sure that fuel shutoff valve is closed.

(3) Make sure that all drain valves are closed.

(4) Open upper nacelle door.

(5) Disconnect hose at fuel inlet of engine, and connect draining hose leading to a vessel.

(6) Close circuit breaker MAIN VALVE and BOOST PUMP.

(7) Place MAIN TANK switch to ON.

(8) Wing tank fuel is discharged through hose.

(9) When boost pump begins no-load running and no more fuel comes out, place MAIN TANK switch to OFF and open circuit breaker.

(10) Drain remaining fuel through drain valve of each tank.

(11) Open fuel filter drain valve to drain.

(12) Close fuel filter drain valve.

8.6 REFUELING

Refueling is accomplished through the fuel filler ports located on the top of the right and left outer tanks.

CAUTION

(i) Caution should be exercised during refueling operation. Aircraft, fuel truck and fuel nozzle must be grounded.

(ii) Before refueling, make sure that there is no pressure in tip tanks by drawing air out through sniffle valve, and then remove fueling cap.

8.6.1 FULL REFUELING

(1) Reach L.H. filler port using ladder or appropriate work stand.

(2) Remove cap by lifting up handle and turning it counter-clockwise.

(3) Ground refueling nozzle to aircraft.

(4) Start fuel flow.

(5) When fuel level has reached filler port, stop filling; and after few minutes for stabilization, fill up the shortage.

(6) Install cap.

(7) Reach R.H. filler port and refuel in accordance with procedure specified in Para. (1) thru (6) above.

8.6.2 PARTIAL REFUELING

(1) Open LH and RH filler ports and check fuel levels.
(2) If levels are different, refuel the low level side first.
(3) Service lower level tanks first with 60% of fuel to be added. Service remainder 40% of fuel to opposite tank. If the lower level tank becomes full before filling 60% of fuel, fill up opposite tank.

8.6.3 TIP TANK REFUELING

CAUTION

Release tip tank pressure by pressing the sniffle valve with a non-metallic rod or probe. Do not stand directly in front of the valve when releasing pressure due to possible fuel ejection through the valve.

(1) Reach filler port.
(2) Remove cap by turning it counter-clockwise 90 degrees.
(3) Ground refueling nozzle to aircraft.
(4) Refuel.
(5) When fuel level has reached filler port, stop filling; and after a few minutes for stabilization, fill up the shortage.
(6) Install filler cap.
(7) Refuel the opposite tank in accordance with procedures specified in Para.(1) through (6) above.

CAUTION

After refueling, close the filler port immediately to keep out dust and water.

NOTE

In the following cases, drain water from each drain valve.

(i) Before and after flight.
(ii) 30 minutes after aircraft is moved from warm place to cold place.
(iii) 5 to 10 minutes after refueling.

8.6.4 OUTER TANK REFUELING

(1) Open the filler port cover by pushing the latch with a finger tip. Remove the dip stick.
(2) Remove the filler port cap by pulling up the cap handle and turning it counter-clockwise.
(3) Ground refueling nozzle to aircraft.
(4) Refuel.
(5) When the fuel level has reached filler port, stop filling.
(6) Install cap.
(7) Refuel the opposite tank in accordance with procedures specified in Para.(1) through (6) above.

8.7 BLENDING ANTI ICING ADDITIVE TO FUEL

8.7.1 REFUELING FUEL CONTAINING ANTI-ICING ADDITIVE

Refuel in normal refueling procedures. This mixture can be used in the main tank.

8.7.2 REFUELING WITH ANTI-ICING ADDITIVE

Blending procedures are as follows.

WARNING

"HI-FLO PRIST" may be harmful if inhaled or swallowed. Use adequate ventilation. Avoid contact with skin and eyes.

(1) Using "HI-FLO PRIST" blender manufactured by PPG INDUSTRIES, INC., remove actuator cap.

(2) Press valve button (attached to tube and clip assembly) into valve on top of can.

(3) Reattach actuator cap by positioning onto can.

(4) Place clip with tubing onto fuel nozzle.

(5) To start flow, press actuator down fully. To stop flow, press tilt to side and return to normal position.

(6) Use can upright and start flow of PRIST after refueling begins (refueling should be at a minimum rate of 30 gal/min. to a maximum of 60 gal/min.). A rate of less than 30 gal/min. may be used when topping off tanks.

(7) Stop flow of PRIST a moment before refueling stops.

CAUTION

Assure that the additive is directed into and blends with flowing fuel from fueling nozzle. Do not allow concentrated additive to contact interior of fuel tanks or aircraft painted surfaces. Use not less than 20 fl. oz. of additive per 260 gallons of fuel or more than 20 fl. oz. of additive per 104 gallons of fuel.

ALTERNATE BLENDERS

If alternate blenders must be used such as PRIST proportioner Model PRB-101 or AP-2, use instructions furnished with blender.

8.7.3 MEASUREMENT OF ANTI-ICE ADDITIVE DENSITY

To measure density of anti ice additive in fuel, use the following equipment.

Differential Refractometer AC-500
Seiscor Corp., Tulsa, Oklahoma

See Manufacture's Manual.

CHAPTER

8

CABIN AIR CONDITIONING AND PRESSURIZATION SYSTEM

CHAPTER VIII

CABIN AIR CONDITIONING AND PRESSURIZATION SYSTEM

CONTENTS

1. GENERAL

The cabin air conditioning and pressurization system provides cabin temperature control, ventilation and pressurization by air with controlled temperature and pressure. For this purpose, left and right engine compressor bleed air from the final stage is used and sent to the cabin after temperature is controlled by the air conditioning system. This system consists of the refrigeration unit, water separator, temperature control system, and cabin pressure control system, etc., and will maintain cabin pressurization, and air conditioning on the ground or during flight. When the system is switched to ram air condition during flight, cabin ventilation is accomplished with ram air.

Cabin pressurization capability is 5.25 (*1), 6.10 (*2) psi maximum, that is, at an altitude of 25,000 ft., cabin pressure altitude of 8,500 (*1), 6,700 (*2) feet can be maintained. Cabin pressure control and temperature control systems are provided with automatic and manual controls.

2. CABIN AIR CONDITIONING SYSTEM

2.1 GENERAL DESCRIPTION

This system controls temperature and pressure of engine compressor bleed air and sends it to the cabin as air source for pressurization and air conditioning. The system consists of refrigeration unit, water separator, refrigeration unit bypass valve, cooling turbine bypass valve, ram air shutoff valve, engine bleed air pressure regulator, temperature control system and air outlets.

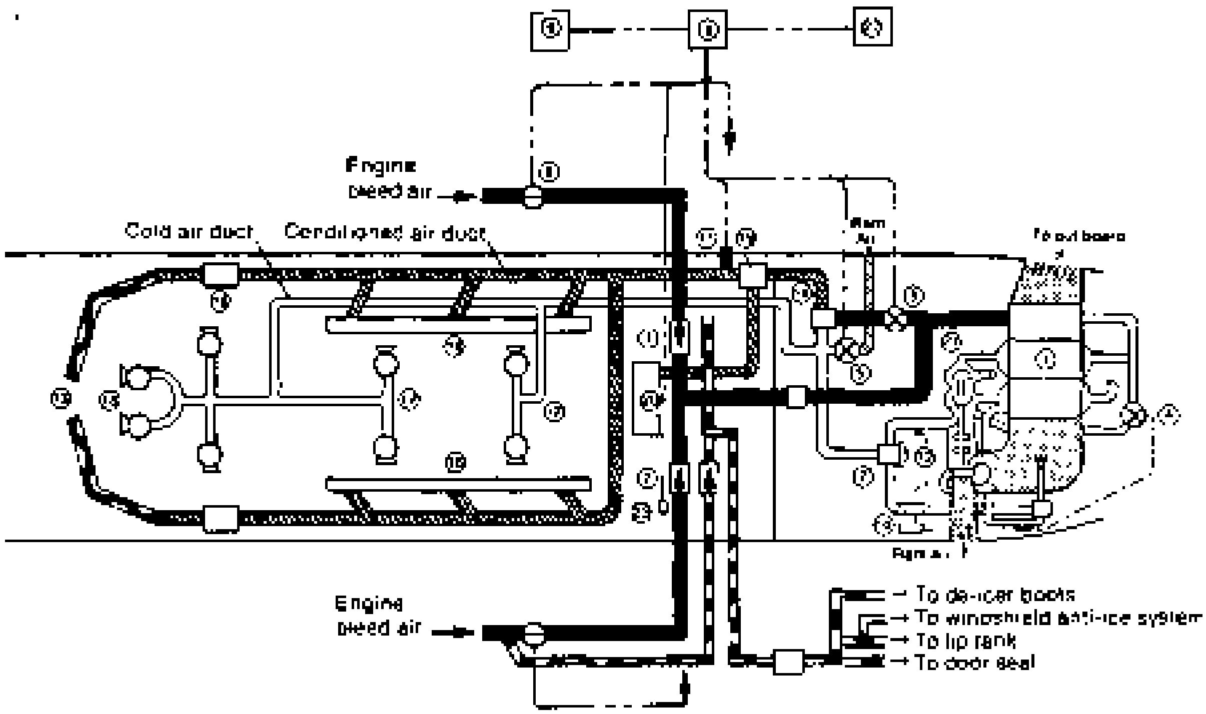
High temperature and pressure air from the engine compressor is regulated by pressure regulator and is precooled by precooler and primary heat exchanger and then compressed into high temperature and high pressure air by the compressor. The air is again cooled by the secondary heat exchanger and further cooled by the cooling turbine. (See Fig. 8-1)

The power absorbed by the cooling turbine is used to turn the compressor and fan connected directly to the turbine. The fan draws ram cooling air.

The cold turbine discharge air passes into the cabin for cooling through the water separator where the condensed water is removed.

To provide cabin heating, high temperature air is branched off from the upstream side of the precooler, regulated by refrigeration unit bypass valve and mixed with cold water separator discharge air to provide air temperature selected by cabin temperature control.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent



1. Refrigeration unit
2. Water separator
3. Ref. unit by-pass valve
4. Cooling turbine by-pass valve
5. Ram air shutoff valve
6. Bleed air shut-off valve
7. Bleed air check valve
8. Air conditioning control unit
   - Cabin temperature control
   - Cabin temperature selector
   - Auto-manual selector switch
   - Cabin air outlet selector switch
   - Cabin air selector switch
   - Transfer switch
9. Cabin temp. control selector and transfer switch
10. Water separator inlet air temp. control
11. Cabin supply air temp. sensor
12. Water separator inlet air temp. sensor
13. Windshield defog air outlets
14. Mixing chamber
15. Forward conditioned air outlets
16. Main conditioned air outlets
17. Cold air outlets (For passengers)
18. Cold air outlets (For pilots)
19. Distributor valve
20. Ceiling outlet
21. Cooling turbine
22. Warning light
23. Cabin temperature sensor

- Engine bleed air
- Engine bleed air
- Cooling air
- Conditioned air
- Cold air
- Wiring

Fig. 8-1 Air conditioning system

In order to prevent water separator inlet icing, the water separator inlet air temperature is maintained at approximately 38°F (3°C) by mixing turbine discharge air with precooled air branched from primary heat exchanger inlet through cooling turbine bypass valve. Cabin air temperature control is accomplished by controlling cabin supply air temperature, and the air temperature is sensed to control the refrigeration unit bypass valve automatically or manually. Water separator inlet air temperature is controlled automatically.

Cabin air outlets consist of the forward, center and aft cabin air outlets (LH and RH), private room conditioned air outlets, ceiling outlet and cold air outlets for pilots and passengers.

2.2 REFRIGERATION UNIT

The refrigeration unit consists of precooler, heat exchanger (primary and secondary), air cycle machine, water aspirator, turbine bypass valve and temperature limit switch. The precooler is a tube with fin and heat exchanger or the plate fin type. The air cycle machine is the one in which compressor and fan are installed on a shaft supported with ball bearings. Lubricating oil for the bearing is drawn up by wick and distributed throughout the bearing cartridge by the use of oil slingers. The water aspirator is an ejector which uses some of the engine bleed air from the downstream side of the secondary heat exchanger as power source and sprays drained water of water separator to cooled air side of heat exchanger in order to raise cooling capacity of heat exchanger.

The turbine bypass valve is an electric motor type butterfly valve which controls bypassed air flow.

The temperature limit switch actuates when the compressor discharge temperature exceeds preset point of 435 ± 15°F (224 ± 8°C) and the signal closes LH engine bleed air shutoff valve and also illuminates the ASC FAIL warning light.

2.2.1 REMOVAL AND INSTALLATION (See Fig. 8-2)

2.2.1.1 Refrigeration Unit (Removal of entire unit)

(1) Remove wiring connectors for turbine bypass valve and temperature switch.

(2) Remove screws at fan inlet, loosen clamp and remove rubber hose.

(3) Loosen clamp at turbine outlet and remove rubber hose.

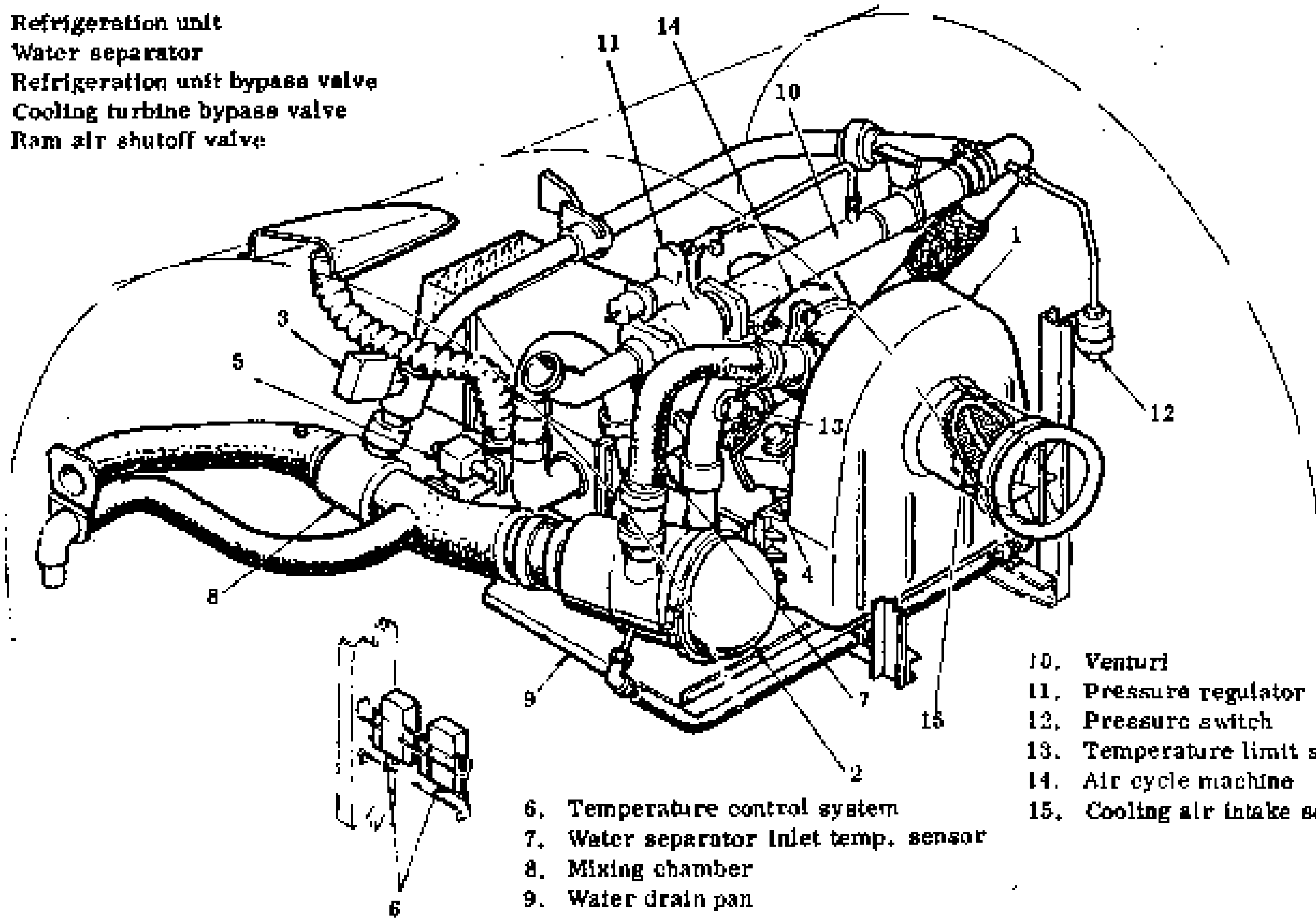
(4) Loosen couplings at compressor inlet and primary heat exchanger outlet, and remove tube.

(5) Loosen coupling at primary heat exchanger inlet and separate tube assembly at upstream side.

(6) Separate drain line for water separator and water aspirator.

(7) Remove sensing line for pressure switch.

1. Refrigeration unit
2. Water separator
3. Refrigeration unit bypass valve
4. Cooling turbine bypass valve
5. Ram air shutoff valve



6. Temperature control system
7. Water separator inlet temp. sensor
8. Mixing chamber
9. Water drain pan

10. Venturi
11. Pressure regulator
12. Pressure switch
13. Temperature limit switch
14. Air cycle machine
15. Cooling air intake screen

Fig.8-2 Air conditioning and pressurization equipment

(8) Remove engine bleed air tubing at precooler inlet.

(9) Remove bolts attaching unit to the frame.

(10) Remove cooling air discharge port adapter.

(11) Remove bolts connecting flanges of precooler and heat exchanger and separate precooler and heat exchanger.

(12) Cover all openings.

(13) Remove the unit from the frame and take outboard from fan housing through the LH access door.

(14) When other units interfere with the refrigeration unit, remove them if necessary.

(15) Install in reverse sequence of removal.

(a) Spray MS122 Flourocarbon Parting Agent (Miller Stephens Chemical Co., Canbury, Conn.) on the connecting flange surface of precooler and heat exchanger and apply RTV60 and Thermolite 12 (General Electric Co., Pittsfield, Mass.).

(b) Installation torque of unit attaching bolts and connecting bolts of precooler and heat exchanger is 25 to 30 in-lbs. (29 to 34 kg/cm).

(c) Installation torque of couplings at primary heat exchanger inlet and outlet and compressor inlet is 45 to 50 in-lbs. (51 to 57 kg/cm).

(d) The nylon coupling nut of water separator drain line should be re-tightened 2 or 2.5 turns more by tool after hand-tightening.

2.2.1.2 Air Cycle Machine (Removal of single unit only)

(1) Remove attaching bolts at the downstream tube of air cycle machine and turbine bypass valve and separate tube assembly.

(2) Loosen couplings at compressor inlet and outlet and turbine inlet and separate each tube assembly from air cycle machine.

(3) Loosen hose clamp at turbine outlet and separate downstream tube assembly from air cycle machine.

(4) Cover all openings.

(5) Remove bolts attaching air cycle machine to heat exchanger.

(6) Remove bolts attaching air cycle machine to fan housing.

(7) Remove air cycle machine.

(8) Install in reverse sequence of removal.
   (a) Installation torque of connecting bolts at air cycle machine and fan housing, heat exchanger and downstream tube assembly of turbine bypass valve is 25 to 30 in-lbs. (29 to 35 kg/cm).
   (b) Istallation torque of couplings at compressor outlet and turbine inlet is 45 to 50 in-lbs. (52 to 58 kg/cm).
   (c) Use as many washers as required to maintain gap between turbine housing flange and heat exchanger bracket within ± 0.016 in. (0.4 mm).

2.2.1.3 Turbine Bypass Valve (Removal of single unit only)

(1) Remove wiring connector from the valve and identify.

(2) Slide spring clamp and rubber hose back onto the downstream ducts.

(3) Remove bolts attaching the valve to the heat exchanger.

(4) Remove connecting bolts at valve inlet and upstream duct and remove the valve.

(5) Cover all openings.

(6) Install in reverse sequence of removal.
   (a) Installation torque of connecting bolts at valve inlet and up-stream duct is 25 to 30 in-lbs. (29 to 35 kg/cm).
   (b) Use as many washers as required to maintain gap between valve attaching flange and heat exchanger bracket within ± 0.016 in. (0.4 mm).

NOTE

Do not incline air cycle machine with lubricating oil more than 45° wherever practicable. In case of reinstalling turbine bypass line, replace packing (6949J129) at air cycle machine inlet and gasket (69149A150) at bypass valve inlet with new ones.

2.2.2 CHECK AND REPLACEMENT OF LUBRICATING OIL

Check lubricating oil quantity every 100 hours and refill per below, as necessary.

(1) Remove filler plug (AN814-2DL) and packing (6949K902).

(2) Fill lubricating oil (MIL-L-23699) up to the level of filler port.

(3) Reinstall packing and filler plug. Apply 15 to 20 in-lbs. (17 to 23 kg/cm) of torque.

Change lubricating oil every 500 hours as follows.

(1) Remove screw (NAS600308) and washer (NAS620C10L) attaching sump to air cycle machine.

(2) Remove and clean sump and refill with new oil (MIL-L-23699) up to the level of filler port.

(3) Install sump. Replace packing (69494K33) with new one if necessary. Tighten screw with a torque 12 to 18 in-lbs. (14 to 21 kg/cm) more than running torque.

NOTE

Running torque is a torque necessary to turn screw (torque before tightening).

CAUTION

Do not allow solvent to enter any rotating components. The air cycle machine must not be operated for 3 hours after filling with lubricating oil. If the machine is to be operated at 1 hour after lubricating, the wick should be soaked with oil prior to installing to ensure immediate bearing lubrication.

2.2.3 TEST

Carefully determine that the rotating section of the air cycle machine is free turning.

(1) Before installation

Carefully insert a non-metallic probe through the fan inlet and gently turn the fan blades.

(2) After installation

Gain access to intake air duct, open spring door and carefully insert a non-metallic probe through the door and fan inlet and gently turn the fan blades.

2.3 WATER SEPARATOR

The water seperator is located downstream of the turbine and is designed to remove condensed water from incoming air. It is divided into 4 parts functionally: coalescer, swirl generator, discharger and bypass relief valve. Small water drops in the incoming air become larger drops in the coalescer and are thrown outward by the centrifugal force in the swirl generator and are collected in the discharger. The water, then, is sprayed at the upstream side (cold air) of heat exchanger by means of water aspirator through drain line. When the coalescer is blocked and difference pressure between upper and downstream reaches 3 ± 0.5 psi (0.21 ± 0.04 kg/cm^2), the bypass relief valve is opened and incoming air flows directly downstream.

2.3.1 REMOVAL AND INSTALLATION

(1) Loosen clamps at inlet and outlet and remove rubber hose.

(2) Separate drain line.

(3) Remove 2 screws attaching water separator to frame bracket.

(4) Remove water separator.

(5) Cover all openings.

(6) Install in reverse sequence of removal.

NOTE

Make sure that drain line (aspirator line) is installed firmly to water separator. When connections between water separator, nylon elbow and line are faulty, water may spout out.

2.3.2 CHECK, CLEANING OR REPLACEMENT OF COALESCER

(1) Loosen V-band coupling at the center of water separator and remove end housing.

(2) Carefully remove coalescer from the inner housing and remove rubber end from the flange.

(3) Check coalescer for damage. If not damaged, clean by washing in a mild soap solution, thoroughly rinsing and drying.

(4) Reinstall coalescer. Stretch rubber end with care.

(5) Make sure that all edges of coalescer are in the fixed position.

(6) Install end housing, tighten nut for V-band coupling to 6 to 8 in-lbs. (7 to 9 kg/cm) torque.

2.4 ENGINE BLEED AIR SHUTOFF VALVE

The valve is installed in engine bleed air tubing in the nacelle rear section and controlled by the cabin air selector switch located on RH switch panel. It is a butterfly valve operated by 28VDC actuator and operating time required from "full close" to "full open" is 7.5 to 12 seconds.

2.4.1 REMOVAL AND INSTALLATION (Fig. 8-3)

(1) Remove access panel located in RH upper rear section of the nacelle.

(2) Remove wiring connectors.

(3) Remove joint clamps upstream and downstream of the valve.

(4) Remove valves and two gaskets.

(5) Cap valve ports.

(6) Install in reverse sequence of removal.

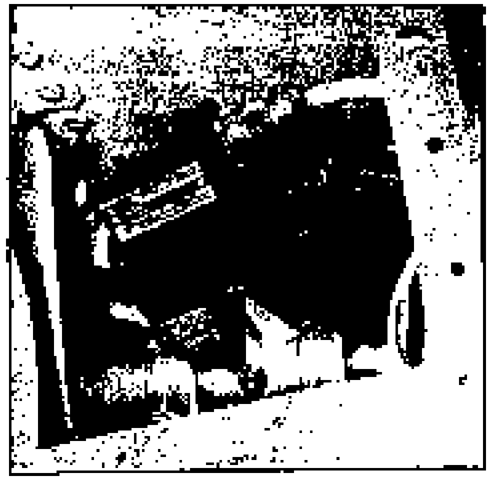


Fig. 8-3. Bleed air shutoff valve

NOTE

When bleed air shutoff valve is re-installed, gasket (Marman 24096-150-C) should be replaced with a new one.

2.4.2 OPERATIONAL CHECK

(1) Remove access panel located in RH upper rear sections of LH and RH engines.

(2) Connect DC power source (battery or APU).

(3) Set circuit breaker for AIR COND of the group AIR COND & DE-ICE.

(4) Turn AIR COND (cabin air selector switch located on RH switch panel) to OFF. Make sure position indicator attached to each valve in LH and RH nacelles shows CLOSE.

(5) Select LH, check the actuation of the LH valve only, and make sure OPEN is shown.

(6) Select BOTH, check the actuation of the RH valve, and make sure OPEN is shown.

(7) Select RH, check the actuation of the LH valve, and make sure CLOSE is shown.

(8) Turn switch to OFF.

2.5 REFRIGERATION UNIT BYPASS VALVE

This valve is located on the bypass line connecting bleed line, upstream of the precooler and mixing chamber, downstream of the water separator. This valve controls bleed bypass flow by means of signal from cabin temperature regulator to maintain cabin temperature at the selected value. This switch can also be directly operated by the pilot. Required time from full close to full open is 7.5 to 12 seconds.

2.5.1 REMOVAL AND INSTALLATION (See Fig. 8-2)

(1) Remove RH electric compartment door at F.STA 8615.

(2) Remove wiring connector from the valve.

(3) Loosen coupling at valve outlet.

(4) Remove connecting bolts at valve inlet and upstream duct and remove the valve.

(5) Cover all openings.

(6) Install in reverse sequence of removal.
Installation torque of connecting bolt is 25 to 30 in-lbs. (29 to 35 kg/cm).

NOTE

In reinstallation, replace gasket (51134-150A) at valve inlet with new one.

2.6 RAM AIR SHUTOFF VALVE

This unit is located between ram air intake and upstream of the mixing chamber and is opened by setting cabin air selector switch to RAM to ventilate cabin with ram air. If necessary, turn manual pressure control valve in the direction from INC to DEC positions. Ram air shutoff valve requires 15 to 21 seconds (*1), approximately 1 second (*2) to full open from full close.

2.6.1 REMOVAL AND INSTALLATION (See Fig. 8-4)

(1) Remove LH access panel of electrical compartment, F.STA 8615.

(2) Remove wiring connector and bonding wire.

(3) Remove valve attaching bolts.

(4) Remove valve and two gaskets.

(5) Cap valve ports.

(6) Install in reverse sequence of removal.

Fig. 8-4 Ram air shutoff valve

*1 Aircraft S/N 652SA, 661SA, 697SA thru 732SA
*2 Aircraft S/N 733SA and subsequent

NOTE

Replace with new gasket (AN6230-9) when ram air shutoff valve is reinstalled. *1

CAUTION

Do not attempt to adjust limit switches. *1

2.6.2 OPERATIONAL CHECK

(1) Remove LH and RH electric compartment door, F.STA 8615, as required.

(2) Connect DC current (battery or APU).

(3) Close circuit breaker AIR COND.

(4) Turn OFF cabin air selector switch on RH switch panel and make sure that the valve position indicator shows CLOSE.

(5) Turn to RAM and make sure this indicator shows OPEN.

(6) Turn it again to OFF and make sure CLOSE is shown.

(7) Restore aircraft to original configuration.

2.7 BLEED AIR PRESSURE REGULATOR

This regulator is located upstream of venturi in engine bleed air line and regulates the regulator output pressure at the specified value. The regulating pressure is 33 ± 2 psi (2.3 ± 0.14 kg/cm^2) when the upstream pressure is 40 psi (2.8 kg/cm^2) and 30 ± 3 psi (2.1 ± 0.21 kg/cm^2) for 100 psi (7 kg/cm^2) of upstream pressure.

This regulator also works to control flow in combination with venturi.

2.7.1 REMOVAL AND INSTALLATION

(1) Remove LH electric compartment door, F.STA 8615.

(2) Remove pressure sensing line.

(3) Remove 8 connecting bolts at regulator inlet flange and remove valve.

(4) Cover all openings.

(5) Install in reverse sequence of removal.

NOTE

In reinstallation, replace gaskets (51134-200S) at valve inlet and outlet with new ones.

*1 Aircraft S/N 652SA, 661SA, 697SA through 732SA

2.8 PRESSURE SWITCH

This switch is located on engine bleed air line, upstream of precooler and actuates when the bleed air pressure exceeds 37 psi (2.6 kg/cm^2) of the specified value. The signal closes LH engine bleed air shutoff valve and illuminates the "ACS FAIL" warning light.

2.8.1 REMOVAL AND INSTALLATION

(1) Remove wiring connector of pressure switch.

(2) Disconnect pressure sensing line.

(3) Remove 2 screws and remove pressure switch.

(4) Cover all openings.

(5) Install in reverse sequence of removal.

2.9 TEMPERATURE CONTROL SYSTEM

The temperature control system in air conditioning system consists of the following two systems.

1. Cabin temperature control system.
2. Water separator inlet temperature control system.

2.9.1 CABIN TEMPERATURE CONTROL

The cabin temperature control system maintains cabin temperature at the preset temperature. The major components are as follows:

(1) Cabin temperature control.

(2) Cabin temperature selector (cockpit, cabin).

(3) Auto-manual selector switch.

(4) Cabin temperature sensor.

(5) Cabin temperature sensor fan.

(6) Air supply temperature sensor.

(7) Air supply temperature limit switch.

(8) Transfer switch.

The cabin temperature control is made by regulating opening of refrigeration unit bypass valve and adjusting mixing ratio of engine compressor bleed air and cold water separator discharge air. In automatic control, the signal controlling valve opening is originated by the difference between cabin temperature settled by the selector.

The signal passes temperature control circuit before reaching valve and standard value of supply air temperature is settled. This standard value is compared with the temperature sensed by duct sensor and the difference actuates bypass valve.

Temperature range settled by temperature selector is 60° to 90°F (15.6° to 32.2°C) and supply air temperature is controlled within the limit of 36° to 185° ± 10°F (2.2° to 85° ± 5.5°C). In manual control, 36°F (2.2°C) of cold air is obtained in "COLD" position of auto-manual selector switch and 200° ± 10°F (93.3° ± 5.5°C) of hot air which is the actuating point of cabin supply air temperature limit switch is obtained in "HOT" position.

2.9.2 WATER SEPARATOR INLET TEMPERATURE CONTROL SYSTEM

The water separator inlet temperature control system maintains the water separator inlet air temperature at 38°F (3.3°C) and its major components are as follows:

1. Water separator inlet air temperature control.
2. Water seperator inlet air temperature sensor.

The temperature control is made by adjusting opening of turbine bypass valve.

2.9.3 REMOVAL AND INSTALLATION

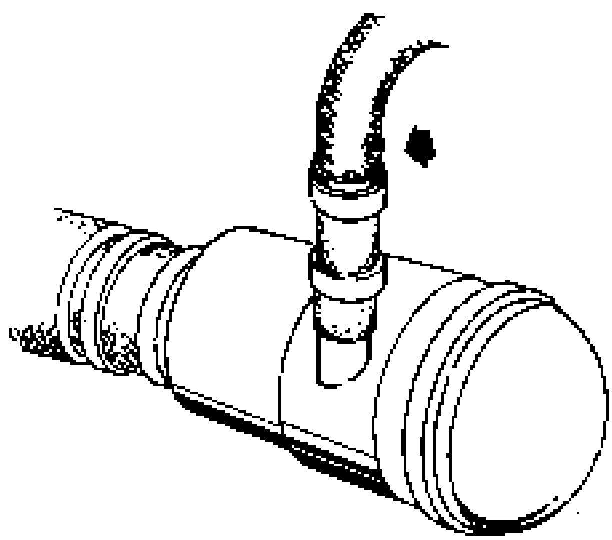


Fig. 8-5 Water separator inlet temp. sensor

2.9.3.1 Cabin Temperature Control (See Fig. 8-2)

(1) Remove LH electronic compartment access door at F.STA 8615.
(2) Remove wiring connector from the control.
(3) Remove 4 screws attaching the control.
(4) Remove the control.
(5) Install in reverse sequence of removal.

2.9.3.2 Water Separator Temperature Control (See Fig. 8-5)

(1) Remove LH access panel of electrical compartment F.STA 8615.
(2) Remove wiring connector.
(3) Remove screws attaching the control.
(4) Remove the control.
(5) Install in reverse sequence of removal.

2.9.3.3 Water Separator Inlet Temperature Sensor

(1) Remove LH electric compartment door at F.STA 8615.
(2) Remove wiring connector from the sensor.
(3) Remove sensor and gasket.
(4) Install in reverse sequence of removal.

NOTE

If gasket (MS28778-8) is damaged, replace.

2.9.3.4 Cabin Supply Air Temperature Sensor

(1) Remove RH junction box cover.
(2) Remove wiring connector.
(3) Remove sensor and gasket.
(4) Install in reverse sequence of removal.

NOTE

If gasket (MS28778-8) is damaged, replace.

2.9.3.5 Cabin Supply Air Temperature Limit Switch

(1) Remove RH junction box cover.
(2) Remove wiring connector.
(3) Remove screws attaching the control.
(4) Remove the control.
(5) Install in reverse sequence of removal.

NOTE

If gasket (MS28778-5, S/N 652SA and 661SA; MS2778-8, S/N 697SA and up) is damaged, replace it.

2.9.3.6 Cabin Temperature Sensor and Fan

(1) Remove access panel on the LH side at F.STA 5890.
(2) Remove screws attaching bracket to side panel.
(3) Remove wiring connector.
(4) Remove sensor and fan.
(5) Install in reverse sequence of removal.

2.10 DISTRIBUTOR VALVE

The distributor valve is installed on the conditioned air tubing between aft pressurizing bulkhead and side panel, and distributes air among conditioned air outlets and ceiling outlet. This valve is actuated by turning the switch "CABIN AIR OUTLET" to "CEIL", and thereby airflow from cabin conditioned air outlets decreases and comes out of the ceiling outlet.

This valve produces satisfactory cooling results if operated in cooling condition.

2.10.1 REMOVAL AND INSTALLATION (See Fig. 8-6)

(1) Remove main junction box and RH side panel.
(2) Remove wiring from rotary solenoid.
(3) Remove bracket on which rotary solenoid is installed, from valve body.
(4) Disconnect rubber duct downstream of the valve.
(5) Remove clamp.
(6) Remove screws attaching the valve to bulkhead and remove the valve.
(7) Install in reverse sequence of removal.

1. Rotary solenoid
2. Bracket
3. Rubber duct
4. Clamp
5. Screw

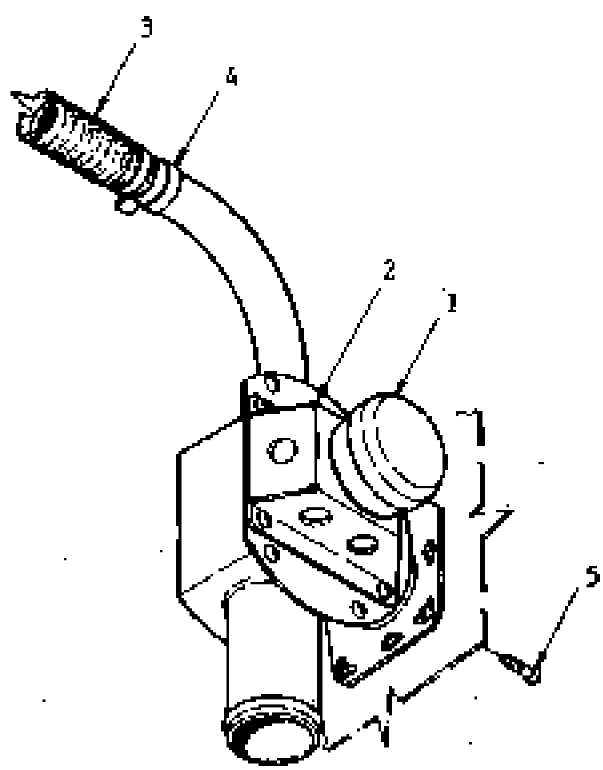


Fig. 8-6 Distributor valve

2.10.2 OPERATIONAL CHECK

(1) Connect DC power (battery or ground power source).

(2) Close circuit breaker AIR COND.

(3) Turn the switch CABIN AIR OUTLET on RH switch panel to CEIL and ensure valve operation.

2.11 COOLING AIR INTAKE SCREEN (See Fig. 8-7)

A screen is installed in the cooling air intake to keep sand and foreign objects from entering the air cycle machine. Two relief doors are provided to draw air from air conditiong electrical compartment should the screen become blocked.

2.11.1 REMOVAL AND INSTALLATION

(1) Remove LH door of electric compartment, F.STA 8615.

(2) Remove rubber seal at joint of cooling air intake and fan housing of refrigeration unit.

(3) Remove screws and cooling air intake from aircraft.

(4) Remove screws and screen from cooling air intake.

(5) Install in reverse sequence of removal.

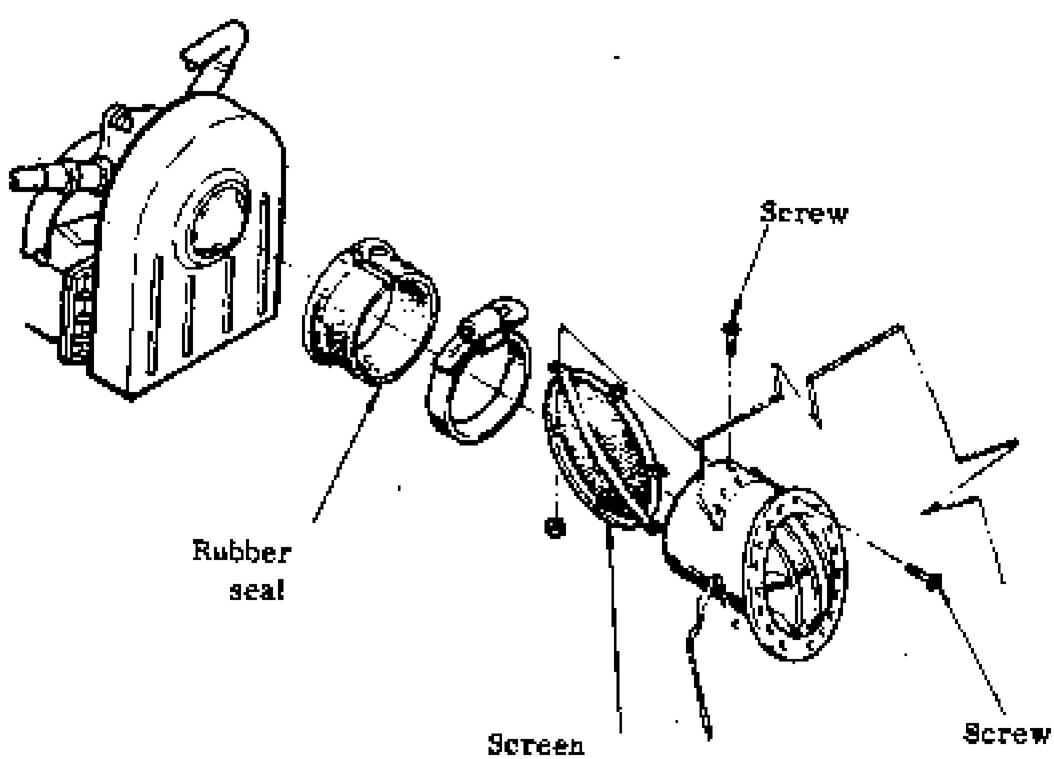


Fig. 8-7 Cooling air intake screen

3. CABIN PRESSURE CONTROL SYSTEM

3.1 GENERAL DESCRIPTION

The cabin pressure control system controls cabin air pressure at a specified value. Cabin pressure control is accomplished by throttling discharge air flow into the cabin through air conditioning system.

The major components of this system are as follows (See Fig. 8-8):

(1) Cabin air outflow valve control.

(2) Outflow safety valve.

(3) Manual pressure control valve.

(4) Pneumatic relays.

(5) Air flow venturi (Aircraft S/N 652SA and 661SA).

(6) Jet pump venturi (Aircraft S/N 652SA and 661SA).
Ejector (Aircraft S/N 697SA and subsequent).

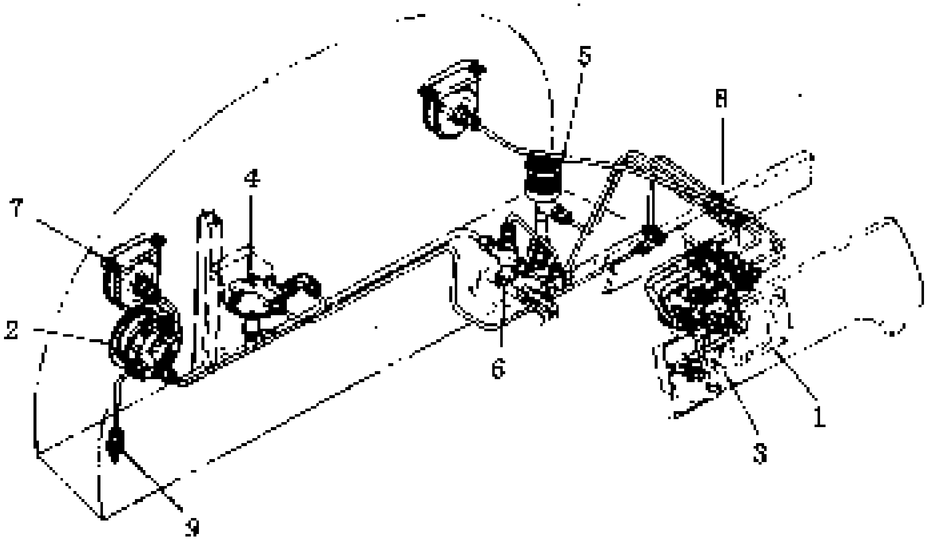
(7) Filter.

(8) Ground test valve.

Cabin pressure control system controls automatically or manually the following operations.

(1) Cabin altitude control.
(2) Cabin rate of climb control.
(3) Maximum (positive) differential pressure relief control.
(4) Negative pressure relief control.
(5) Cabin pressure dumping control.

Discharged air from the cabin enters the electronic compartment in the airplane nose through outflow safety valve(s) installed on the forward pressurized bulkhead and performs heating and/or cooling for the electronic compartment and then goes outboard.



Components of cabin pressure control system at F.STA. 1080~1820

1. Cabin air outflow valve control
2. Outflow/safety valve
3. Manual pressure control valve
4. Pneumatic relay
5. Air flow venturi
6. Jet pump venturi
7. Filter
8. Ground test valve
9. Tubing for atmospheric pressure

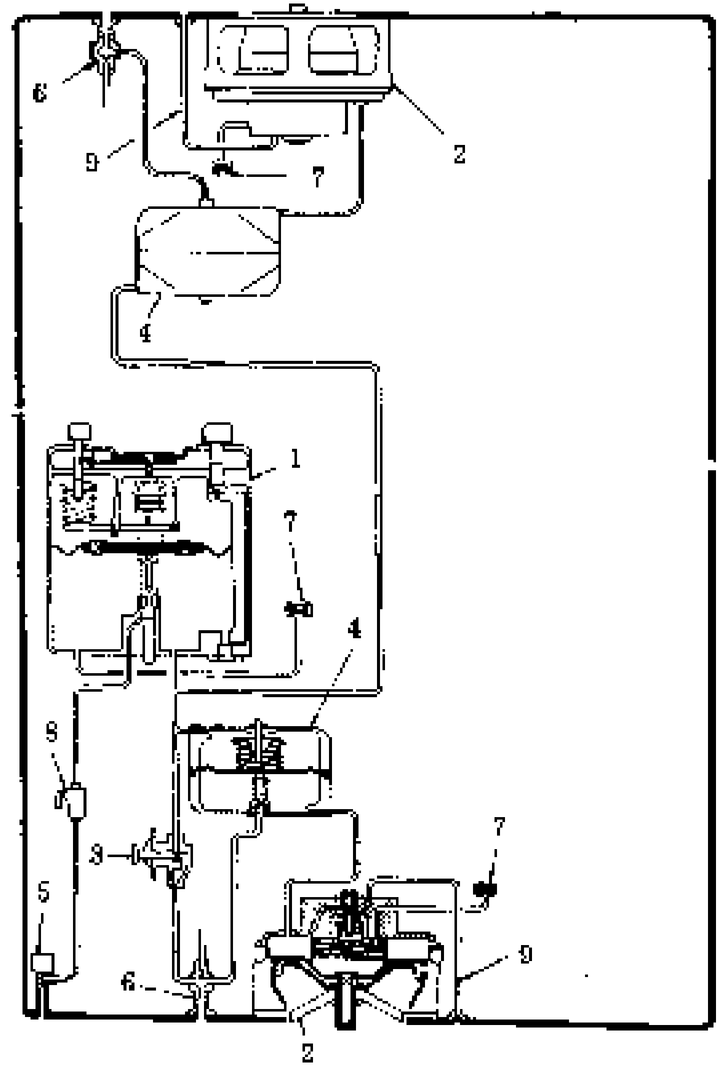
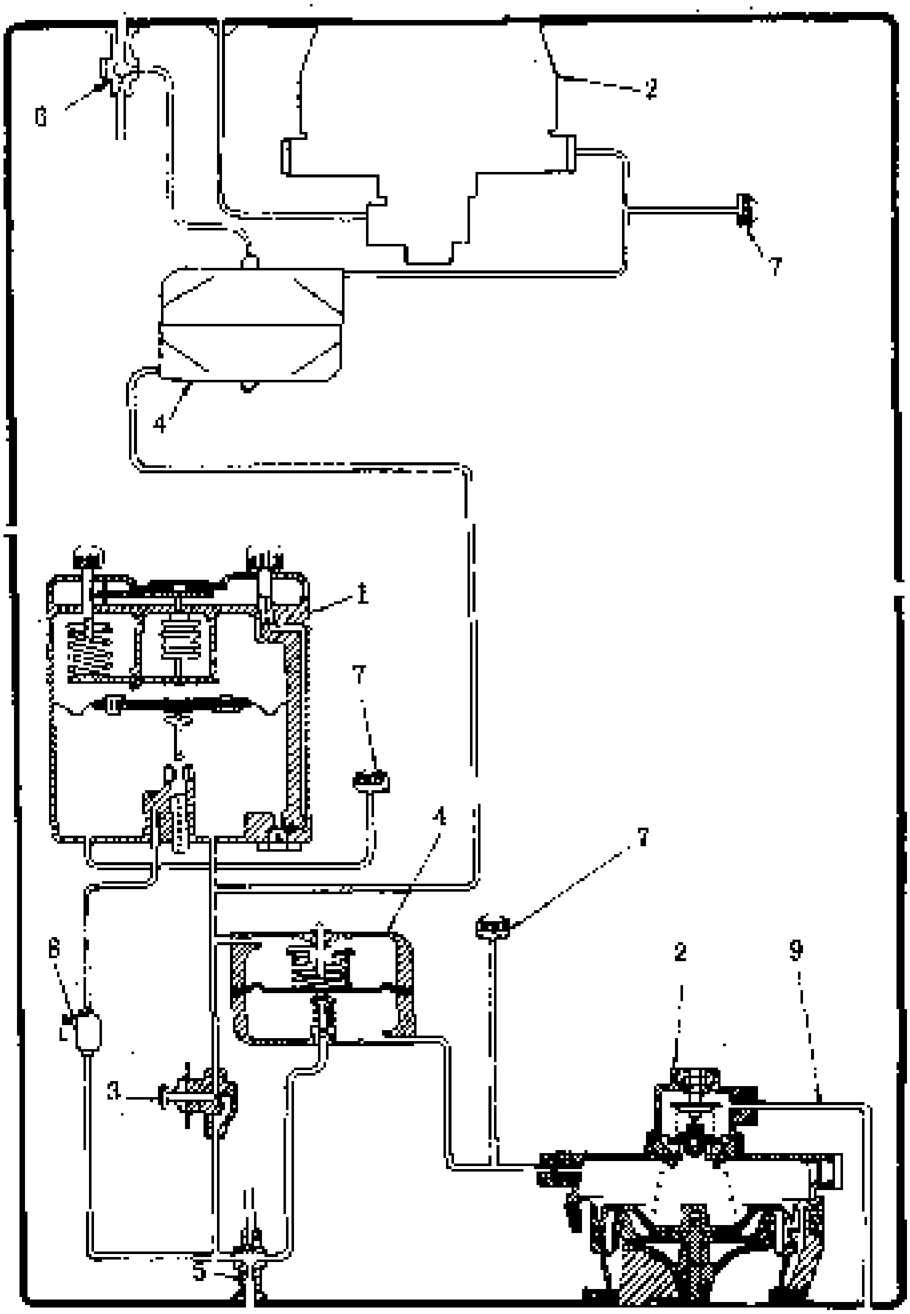


Fig. 8-8 Cabin pressure control system (1/2)
(Aircraft S/N 652SA and 661SA)



1. Cabin air outflow valve control
2. Outflow/safety valve
3. Manual pressure control valve
4. Pneumatic relay
5. Ejector
6. Jet pump venturi
7. Filter
8. Ground test valve
9. Tubing for atmospheric pressure

Fig. 8-8 Cabin pressure control system (2/2)
(Aircraft S/N 697SA and subsequent)

3.2 CABIN AIR OUTFLOW VALVE CONTROL

This control unit is installed on the RH switch panel of the pilot seat and controls cabin altitude and cabin rate of climb. To this unit are connected tubes sensing cabin pressure and atmospheric pressure and a tube connected to the outflow safety valve through a relay. On the control unit there is provided a cabin altitude selector knob and a cabin rate of climb control knob.

The former is used to select desired cabin altitude during cruising by turning the knob and setting the pointer to desired cabin altitude.

Selectable range of altitude is from -1,000 ft. to 10,000 ft. If the pointer is set to desired altitude, the maximum aircraft altitude at which aircraft can fly with the desired cabin altitude is indicated in a small window below the dial.

At altitudes above this, cabin pressure is constantly maintained at a value of 5.25 (*1), 6.0 ± 0.1 (*2) psi higher than atmospheric pressure. A cabin rate of climb control knob is provided to control cabin pressure variation during climb and descent and is capable of controlling cabin rate of climb between 50 ft/min. and 2,000 ft/min. Marking in the center between "MIN" and "MAX" corresponds to rate of climb of 500 ft/min.

3.2.1 REMOVAL AND INSTALLATION (See Fig. 8-9)

(1) Remove three tubes connected to the control unit.

(2) Remove four screws attaching the control.

(3) Remove electric wiring for instrument light.

(4) Remove the control and cap ports.

(5) Install in reverse sequence of removal.

CAUTION

Do not attempt to adjust or repair the control unit. Replace it if trouble occurs.

*1 Aircraft S/N 652SA

*2 Aircraft S/N 661SA, 697SA and subsequent

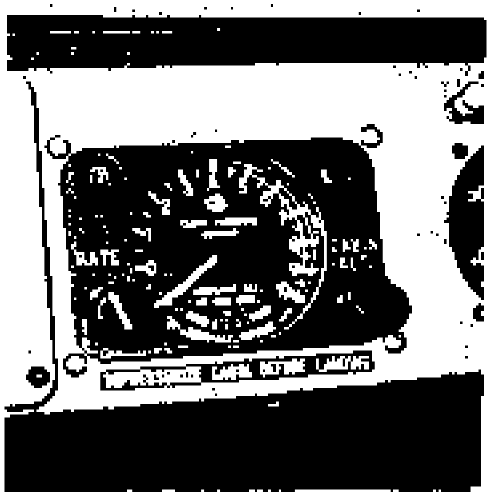


Fig. 8-9 Cabin air outflow valve

Fig. 8-10 Outflow safety valve (*1)

3.3 OUTFLOW SAFETY VALVE

This valve controls discharged airflow from the cabin by a signal (air pressure) from the cabin air outflow valve control in order to control cabin air pressure and, at the same time, performs maximum (positive) differential pressure relief and negative pressure relief. Controlled pressure from the cabin air outflow valve control through a relay is applied to one side of the diaphragm in the valve and cabin pressure on the other side. The differential pressure across the diaphragm moves a poppet valve attached to a diaphragm controlling the area of the outlet section for cabin pressure of 5.25 (*1), 6.0 ± 0.1 (*2) psi during climb or in case of failure of automatic control of cabin air outflow valve control. The maximum pressure relief mechanism, which is actuated by differential pressure between cabin pressure and atmospheric pressure, performs pressure relief, and, therefore, differential pressure does not exceed the specified value. In case of rapid descent, if cabin pressure becomes less than atmospheric pressure, differential pressure between cabin and atmosphere moves the poppet valve to prevent negative differential pressure from increasing.

3.3.1 REMOVAL AND INSTALLATION (See Fig. 8-10)

(1) Remove four tubes (*1) or two rubber hoses (*2) connected to valve.

(2) Remove eight screws (*1) or 16 nuts and washers (*2) attaching valve.

(3) Remove valve and seal.

(4) Remove two unions and one tee (*1) for tube connection and cap port.

(5) Install in reverse sequence of removal.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

WARNING

After removal, check for tobacco tar and other foreign matter accumulated on poppet valve and poppet seat and clean with dry cleaning solvent (P-S-661). When solvent is being used, adequate ventilation should be provided and open flame should not be permitted near the work site.

CAUTION

Do not attempt to adjust or repair the outflow safety valve. Replace it, if trouble occurs.

3.3.2 CHECK

(1) Wipe valve attaching seat and its vicinity on airframe side with clean soft cloth moistened with naptha (MIL-N-15178) to remove oil and dirt.

(2) When seal is damaged or has deteriorated, replace it.

(3) Apply sealant (MIL-S-8802 Class B) to circumference of valve attaching flange and screw heads which may cause a leakage.

CAUTION

Be careful not to apply sealant to poppet valve and/or poppet seat.

(4) When the aircraft has not been flown for one consecutive month or longer, apply a negative pressure of 5.0 to 5.3 psi (*3), 5.9 to 6.1 psi (*4) to the atmospheric port of the valve before flight, and ensure poppet valve is actuated.

3.4 MANUAL PRESSURE CONTROL VALVE

One end of this valve is located between the cabin air outflow valve control and pneumatic relay and the other end is connected to a jet pump venturi (*1), an ejector (*2). In case of failure of the cabin air outflow valve control, cabin pressure is controlled manually by this valve and ram air ventilation can be made by this needle type valve according to RAM position of cabin air selector and fine adjustment is possible.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent
*3 Aircraft S/N 652SA
*4 Aircraft S/N 661SA, 697SA and subsequent

3.4.1 REMOVAL AND INSTALLATION

(1) Remove tubes (2 ea.) connected to valve.

(2) Remove screws (4 ea.) attached to switch panel.

(3) Remove valve and cap ports.

(4) Install in reverse sequence of removal.

NOTE

Should the manual pressure control valve develop trouble, replace it. Do not disassemble the valve or replace component parts except to clean the valve parts and passages with compressed air.

3.5 PNEUMATIC RELAY (Fig. 8-12)

This relay is divided into two parts by a diaphragm. Control pressure from cabin air outflow valve control is applied to one side and air pressure chamber of outflow safety valve is applied to the other side. According to diaphragm motion, a metering valve is provided in outflow safety valve to control airflow to ejector. Small fluctuation of control pressure from cabin air outflow valve control is transmitted to metering valve, which, in turn, controls passage between reference pressure chamber to vacuum source of jet pump venturi (*1), ejector (*2), and operates outflow safety valve rapidly and positively.

3.5.1 REMOVAL AND INSTALLATION

(1) Remove tubes (3 ea.) attached to relays.

(2) Remove bolts (2 ea.), nuts (2 ea.), washers (2 ea.) and cushion (2 ea.) attaching relay.

(3) Remove relay.

(4) Remove unions and elbow or tee from relay and cap ports.

(5) Install in reverse sequence of removal.

NOTE

(i) In case of trouble of pneumatic relay, replace it with new one without performing repair or adjustment.

(ii) If attaching cushion is found deteriorated, replace cushion.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

3.6 AIR FLOW VENTURI (Aircraft S/N 652SA and 661SA)

This venturi is utilized to furnish the outflow valve control with a source of vacuum during take-off and landing when differential pressure between inside and outside of cabin is too low to obtain full open valve operation.

3.6.1 REMOVAL AND INSTALLATION (See Fig. 8-13)

(1) Remove tube (1 ea.) from venturi.

(2) Remove access panel of RH side of STA 1080 and remove attaching nuts through this opening.

(3) Remove venturi and gasket and then remove union from venturi.

(4) Cap each port of venturi.

(5) Install in reverse sequence of removal.

NOTE

Replace gasket (AN901-12A) during reinstallation. Apply sealant (MIL-S-8802 Class B) around attaching flange of venturi.

3.6.2 CLEANING AND INSPECTION FOR BLOCKAGE

(1) Wipe off dirt or dust on screen of venturi.

(2) Check atmospheric pressure port for obstruction. This can be checked through access hole of RH side at STA 1080.

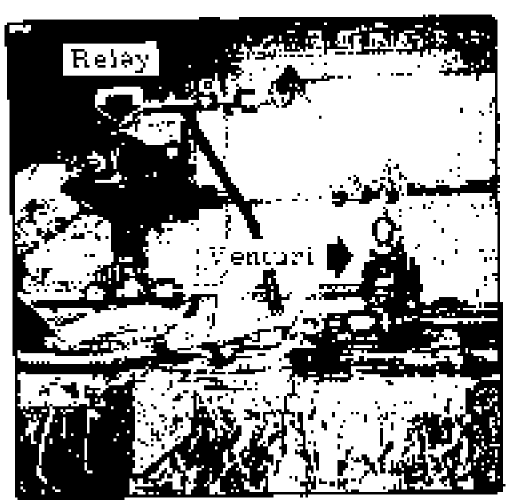


Fig. 8-12 Pneumatic relay and jet pump venturi (S/N 652SA & 661SA)

Fig. 8-13 Air flow venturi (S/N 652SA & 661SA)

3.7 JET PUMP VENTURI

(1) Forward Jet Pump Venturi (Aircraft S/N 652SA and 661SA)

This venturi produces vacuum pressure with high pressure air and controls reference pressure of outflow safety valve through a pneumatic relay. It is also connected to a manual pressure control valve to control the valve.

Pressure regulated engine bleed air is used for high pressure air source.

(2) Aft Jet Pump Venturi

This venturi produces vacuum pressure with high pressure air and controls reference pressure of outflow safety valve through a pneumatic relay. It is also connected to a manual pressure control valve to control the valve.

Pressure regulated engine bleed air is used for high pressure air source.

3.7.1. REMOVAL AND INSTALLATION (See Fig. 8-12)

(1) Remove connecting tubes (3 ea.).

(2) Remove screws (2 ea.) and nuts (2 ea.) attaching venturi.

(3) Remove venturi and seal. Remove union (3 ea.) and cap each port.

(4) Install in reverse sequence of removal.

NOTE

If seal (010A-63113) is damaged or deteriorated, replace it. Apply sealant (MIL-S-8802 Class B) around venturi attaching flange.

3.8 EJECTOR (Aircraft S/N 697SA and subsequent)

This ejector produces vacuum pressure with high pressure air and controls reference chamber of outflow safety valve through reference pressure of cabin pressure regulator and pneumatic relay. It is also used as vacuum source in manual pressure regulating. Pressure regulated engine compressor bleed air is used for high pressure air source.

3.8.1 REMOVAL AND INSTALLATION

(1) Disconnect tubings and remove fitting.

(2) Remove attaching bolts of ejector plate and remove ejector from airframe.

(3) Remove plate and seal by removing screws.

(4) Cap each port.

(5) Install in reverse sequence of removal.

NOTE

When seals (035A-63113 and 035A-63115) become damaged or deteriorated, replace them. Apply sealant (MIL-S-8802 Class B) around ejector attaching plate to prevent leakage.

3.8.2 INSPECTION FOR BLOCKAGE

Check ejector discharge port in nose landing gear well for blockage.

3.9 AIR FILTER

The filter prevents dirt or tobacco tar from entering cabin pressure sensing tubes of cabin air outflow valve controls and outflow safety valve.

This filter consists of a nylon screen for large matter and a filter element for small particles.

3.9.1 REMOVAL AND INSTALLATION (See Fig. 8-14)

(1) Remove connecting tube (1 ea.).

(2) Remove cushion clamp attaching filter.

(3) Remove filter. Cover inlet and outlet air ports.

(4) Install in reverse sequence of removal.

3.9.2 REPLACEMENT OF FILTER ELEMENT CARTRIDGE

(1) Remove filter from airplane.

(2) Remove boots (rubber cover) from filter and replace filter element cartridge (AiResearch 137847-1).

NOTE

Cartridge should be replaced every 500 hours.

Clean dirt from boot interiors and tube joint fittings, etc., before boots are reinstalled.

Union installed between outflow safety valve and pneumatic relay is an orifice, so do not mix with other unions.

(Aircraft S/N 697SA and subsequent)

3.10 ENTRANCE DOOR INFLATABLE SEAL SYSTEM

This entrance door seal is an inflatable seal to maintain cabin pressure and is pressurized by engine compressor bleed air, regulated by surface de-icer pressure regulator and relief valve, through selector valve. Selector valve is actuated by main landing gear safety switch. When main landing gear is retracted, safety switch operates to open pressure line of selector valve.

When the gear is extended, exhaust port of the valve is opened and sealing air is deflated.

Fig.8-14 Installation of filter

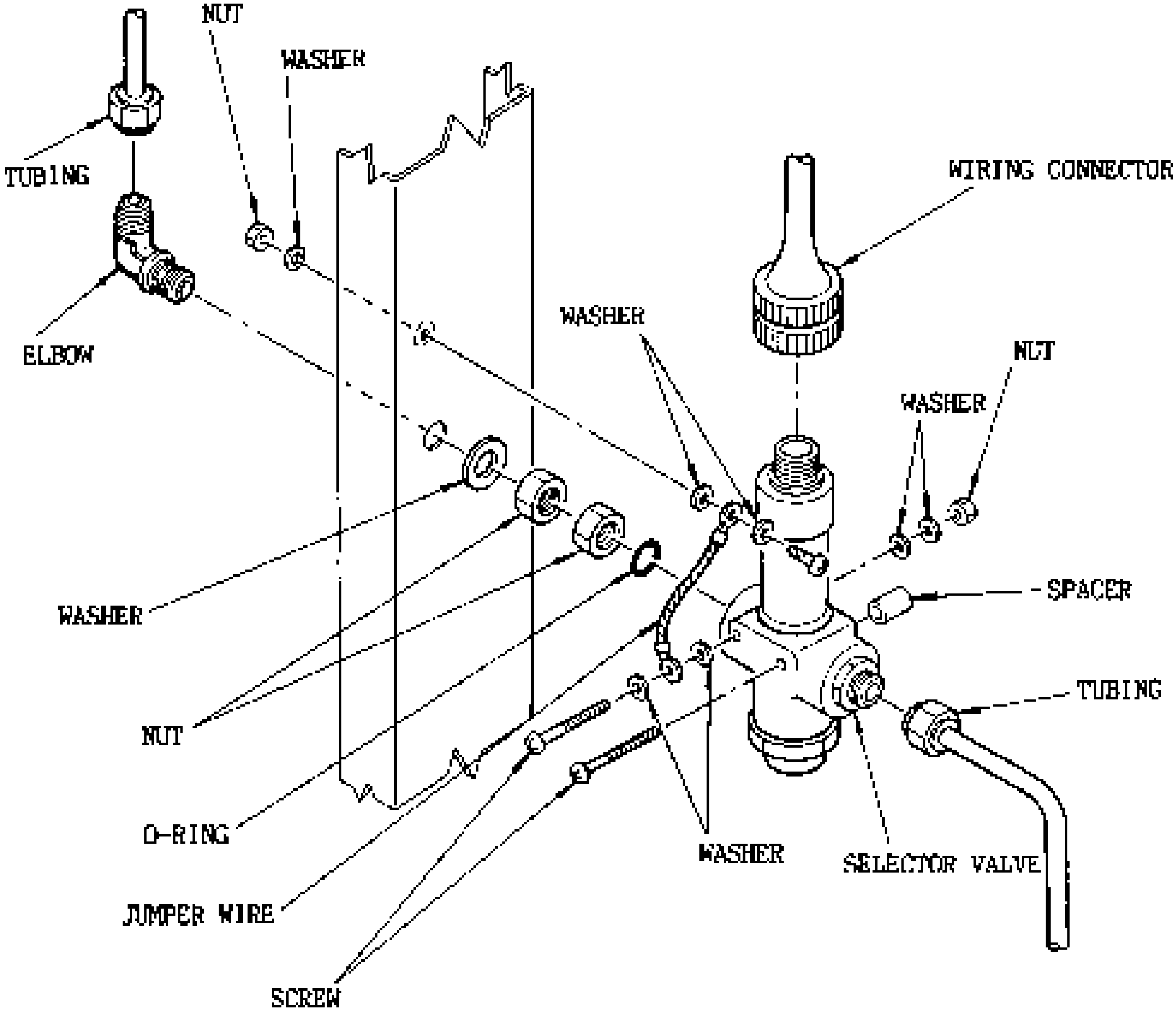


Fig 8-15 Installation of selector valve

3.10.1 REMOVAL AND INSTALLATION OF SELECTOR VALVE (See Fig.8-15)

(1) Remove LH electric compartment door at F.STA 8615.
(2) Disconnect wiring connector from the valve.
(3) Disconnect jumper wire.
(4) Disconnect tubing.
(5) Remove the valve attaching screw.
(6) Remove valve and cap each port.
(7) Install in reverse sequence of removal.

3.10.2 PROCEDURE FOR GROUND INSPECTION OF ENTRANCE DOOR PRESSURE SEAL SYSTEM

(1) When ground air source is used:
  (a) Connect DC power (ground power or battery).
  (b) Connect high pressure air source to ground test port of de-icing system and close entrance door.
  (c) Supply pressurized air gradually and stabilize pressure at 30 psi (2.1 kg/cm^2).
  (d) Close circuit breaker DOOR SEAL, open LG POS IND circuit breaker and make sure that seal is pressurized and inflated. Check for air leaks at seal.
  (e) After confirming (d), open circuit breaker and make sure seal pressure is deflated.
  (f) After confirming (e), restore airplane to original condition.

(2) (a) Make preparation in accordance with (1)(a) above and close entrance door.
  (b) Run up engine, per AIRPLANE FLIGHT MANUAL.
  (c) Open circuit breaker LG POS IND and close DOOR SEAL circuit breaker.
  (d) Make sure seal is pressurized and inflated and check for air leaks at seal.
  (e) Open circuit breaker DOOR SEAL, close LG POS IND circuit breaker and make sure that the seal pressure is deflated.
  (f) Shut down engine and restore airplane to original condition.

CAUTION

Do not operate engine unless entrance door is fully closed. Entrance door must not be opened during test.

3.10.3 PRESSURE LINE DRAIN

(1) When ground air source is used:
  (a) With the entrance door open, perform steps (a), (b) and (c) of paragraph 3.10.2 (1).
  (b) Close circuit breaker DOOR SEAL and drain water from entrance door pressure line connector with compressed air.
  (c) After completing step (b), open the circuit breaker and wipe off water drops around the connector.
  (d) Restore airplane to original condition.

(2) When engine compressor bleed air is used:
   (a) Open entrance door.
   (b) Run up engine per AIRPLANE FLIGHT MANUAL.
   (c) Close circuit breaker DOOR SEAL and drain water from entrance door pressure line connector with compressed air.
   (d) Open the circuit breaker.
   (e) Stop engine.
   (f) Wipe off water drops around the connector.
   (g) Restore airplane to original condition.

3.11 INSTRUMENT AND WARNING LIGHT RELATED TO CABIN PRESSURIZATION

3.11.1 CABIN ALTITUDE AND DIFFERENTIAL PRESSURE GAUGE

This gauge indicates cabin altitude and differential pressure, and altitude being shown is based on standard atmosphere (sea level 29.92 in-Hg abs). Differential pressure is indicated by small hand of inner scale in psi and altitude is indicated by large hand of outer scale. The range from 0 to 5.25 psi (*1), 0 to 6.10 psi (*2) is marked green, and a red line is at 5.25 psi (*1), 6.10 psi (*2). When differential pressure comes to the red line, cabin pressure is decreased by the manual pressure control valve so that cabin pressure is kept within the green range.

NOTE

This instrument shall not be used as an altimeter.

3.11.2 CABIN ABSOLUTE PRESSURE WARNING LIGHT AND SWITCH

The cabin absolute pressure warning light "CABIN PRESS LOW" is installed on the annunciator panel and illuminates when cabin absolute pressure altitude exceeds 10,000 ft. (3,048 m). The warning light is operated by a pressure switch, and contact closes at 10,000 ft. (3,048 m) and opens at 7,800 ft. (2,377 m).

NOTE

When cabin absolute pressure warning light comes on, pilot should put on oxygen mask immediately and descend until cabin altitude drops below 10,000 ft. (3,048 m).

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

4. CONTROL AND TROUBLE SHOOTING FOR CABIN AIR CONDITIONING AND PRESSURIZATION SYSTEM

4.1 AIR CONDITIONING AND PRESSURIZATION SYSTEM

Switch panel for air conditioning and pressurization control is shown in Fig. 8-16.

4.1.1 CABIN AIR SELECTOR SWITCH

This switch selects supply air for air conditioning and pressurization.

Position "RAM" - Ram air shutoff valve fully opens and LH and RH engine bleed air shutoff valves are fully closed. Only cabin ventilation is possible with ram air. Pressurization and temperature control are possible.

Position "OFF" - Both ram air shutoff valve and engine bleed air shutoff valves are fully closed. No air supply to the cabin is available.

Position "LH" ("RH") - Ram air shutoff valve is closed, "LH" ("RH") engine bleed air shutoff valves open, "RH" ("LH") engine bleed air shutoff valve is closed. Pressurization and air conditioning are possible on one engine bleed air source.

Position "BOTH"- Ram air shutoff valve is closed, LH and RH engine shutoff valves open. Pressurization and air conditioning with both engines bleed air are possible.

4.1.2 AUTO-MANUAL SELECTOR SWITCH

This switch is used to select automatic or manual cabin temperature control, and is a toggle switch with four positions. This switch is operative only for "LH", "BOTH" and "RH" positions of cabin supply air selector switch.

Position "AUTO"- Automatically maintains cabin air temperature at value set by cabin air temperature selector. In other words, cabin air temperature control is actuated and the value sensed by cabin air temperature sensor is compared with the value set by cabin air temperature selector and analyzed, and regulates refrigeration unit bypass valve as required, controlling air supply temperature.

Position "MAN COLD" - At manual cooling position, refrigeration unit bypass valve is fully closed and air supply temperature reaches its minimum. Air supply temperature is approximately 38°F (3.3°C) except on especially hot days.

Position "MAN HOT" - At manual heating position, refrigeration unit by-pass valve tends to open. However, supply air temperature is kept below 200°F (93.3°C) by high temperature limit switch. Therefore, signal from limit switch restricts valve opening.

Position "OFF" - Temperature is not controlled. Refrigeration unit bypass valve remains in the position set before it is turned to "OFF".

4.1.3 CABIN TEMPERATURE SELECTOR

Setting range of the temperature selector is 60°F thru 90°F (15.6° thru 32.2°C) and supply air temperature is controlled between 38°F (3.3°C) and 185° ± 10°F (85 ± 5.5°C). Selection is available only when the auto-manual selector is in the AUTO position.

4.1.4 CIRCUIT BREAKER

Circuit breaker "AIR COND" shall be closed to actuate cabin air select switch, cabin temperature selector, auto-manual select switch, cabin temperature control, and water separator inlet air temperature control.

4.1.5 CABIN ALTITUDE SELECTOR KNOB

This knob is attached to cabin air outflow valve control and is marked "CABIN ALT". Rotating the knob, set the pointer on center dial of the control at desired altitude. It is advisable to set cabin altitude at altitude in excess of pressure altitude of airport + 1,000 ft.(305 m) in normal flight so that airplane is not pressurized at takeoff or landing.

4.1.6 CABIN RATE OF CLIMB CONTROL KNOB

This knob is also attached to cabin air outflow valve control and is marked "RATE". Control range of cabin rate of climb is between 50 ft/min. (15. m/min.) "MIN" and 2,000 ± 500 ft/min. (610 ± 152 m/min.) "MAX" and " " mark in the center corresponds to 500 ± 75 ft/min. (152 ± 23 m/min.). Control of cabin rate of climb is available, within climb condition from airport elevation to selected cabin altitude.

4.1.7 MANUAL PRESSURE CONTROL VALVE

This valve permits manual pressure control and this is turned from INC position to DEC position (counter-clockwise), as required. Knob is rotated approximately 6.5 times to reach full open position (DEC position). However, valve begins to open when knob is rotated about one-half turn.

CAUTION

Manual pressure control valve controls pressure with jet pump venturi or ejector and response is so sensitive that the valve should not be turned rapidly.

4.1.8 TRANSFER SWITCH

This switch is to select where operation of cabin air temperature selector is controlled, at cockpit or cabin, and it is changed over alternately every one push.

4.1.9 GROUND TEST VALVE

This valve is not used in flight, but is used only during pressurization test on the ground. This valve is located between the cabin air outflow valve control and the airflow venturi or ejector. Valve is kept fully open and safety wired except when being used for tests.

4.1.10 CABIN AIR OUTLET SELECT SWITCH

This switch actuates distributor valve to distribute air among cabin conditioned air outlet and ceiling outlet.

FLOOR - Valve is not actuated and most air comes out of cabin outlet.

CEIL - Valve is actuated, air flow from cabin outlet decreases and increases from the ceiling outlet.

4.2 OPERATION AND CHECK OF AIR CONDITIONING SYSTEM

4.2.1 Operation and functional test of air conditioning system to check for electric circuit should be accomplished in the following procedures:

(1) Inspection equipment and materials

Air pressure source	40 psi (2.8 kg/cm^2)
Electric power source	28 VDC
Sensor dummy (variable resistor)	0 to 100 KΩ
Electric wire	As required
Cold water	Less than 63°F (17°C)
Warm water	More than 86°F (30°C)

(2) Inspection and requirements

a. Switches used for this inspection are as follows:

Switch No.	Name	Selecting Position	Installing Location
1	Cabin air selector switch	RAM-OFF-LH-BOTH-RH	R.H. switch panel
2	Auto-manual selector switch	OFF-MAN COLD-AUTO-MAN HOT	" " "
3	Cabin Temp. selector	COLD HOT	" " "

b. Make sure that the valves are operated properly as indicated on the following table at the selecting position of the above table. Operational check should be performed three times respectively for each valve.

c. Operational check of engine bleed air shutoff valve and ram shutoff valve.

Switch No. 1 Selecting Position	Valve Position		
	Engine bleed air shutoff valve		Ram air shutoff valve
	LH	RH	
OFF	CLOSE	CLOSE	CLOSE
RAM	CLOSE	CLOSE	OPEN
LH	OPEN	CLOSE	CLOSE
BOTH	OPEN	OPEN	CLOSE
RH	CLOSE	OPEN	CLOSE

Cycle: OFF-RAM-OFF-LH-BOTH-RH-BOTH-LH-OFF

d. Operational check of refrigeration unit bypass valve

(i) Operational check at MANUAL position

Switch No. 2 Selecting Position	Valve Position
MAN COLD	FULL CLOSE
MAN HOT	FULL OPEN

Cycle: OFF-MAN COLD-MAN HOT-MAN COLD-OFF

(ii) Operational check at AUTO position

Check in accordance with Para. (ii)-1 or (ii)-2

(ii)-1 Method with sensor dummy

Remove plug of cabin sensor and connect sensor dummy between terminals and make sure that valve is operated normally.

Switch No. 2 Selecting Position	Switch No. 3 Selecting Position	Variable Resistor Position	Valve Position
AUTO	Neutral	HIGH (more than 39 K Ω)	Operated to the direction of "OPEN"
AUTO	Neutral	LOW (less than 28 K Ω)	Operated to the direction of "OPEN"

Cycle: LOW-HIGH-LOW

(ii)-2 Method with cold water or warm water

Make sure that valve is operated normally in the following conditions. Put cabin sensor into cold water (less than 17°C) and warm water (more than 30°C) alternately and make sure of proper operation.

Switch No. 2 Selecting Position	Switch No. 3 Selecting Position	Sensor Temp. Condition	Valve Position
AUTO	Neutral	Cold Water	Operated to the direction of "OPEN"
AUTO	Neutral	Warm Water	Operated to the direction of "CLOSE"

Cycle: Cold Water-Warm Water-Cold Water

e. Operational check of cooling turbine bypass valve

(i) Remove plug of anti-ice sensor and connect sensor dummy (variable resistor and fixed resistor connected in series) between plug terminals.

(ii) Make sure that valve is operated normally in the following conditions.

Variable Resistor Position	Valve Position
HIGH (more than about 90 K Ω)	Operated to the direction of "OPEN"
LOW (more than about 74 K Ω)	Operated to the direction of "CLOSE"

Cycle: LOW-HIGH-LOW

f. Operational check of fan

Put circuit breaker "AIR COND" in and make sure that fan is operated normally.

g. Operational check of thermal switch

(i) Set switch No. 1 to "BOTH" and make sure that LH and RH engine bleed air solenoid valves open.

(ii) Make sure that "ACS FAIL" lamp illuminates and LH engine bleed air solenoid valve closes when over-temp switch circuit is shorted.

(iii) Release short-circuit and make sure that the above conditions are kept.

(iv) Turn circuit breaker "AIR COND" off and set on again. Make sure that LH engine bleed air solenoid valve opens and "ACS FAIL" lamp goes out.

h. Operational check of pressure switch

(i) Make sure that LH and RH engine bleed air solenoid valves are open when switch No. 1 is in "BOTH" position.

(ii) Connect air pressure source to over pressure switch and make sure that "ACS FAIL" lamp illuminates at 37 psi (2.6 kg/cm^2) and LH engine bleed air solenoid valve closes.

(iii) Reduce pressure and make sure that the above conditions are kept at 0 psi.

(iv) Turn circuit breaker "AIR COND" off and on again and make sure that LH engine bleed air solenoid valve opens and "ACS FAIL" lamp goes out.

(v) Turn switch No. 1 off.

(vi) Disconnect air pressure source and restore the system.

4.2.2 Operation and functional test of air conditioning system during engine run should be accomplished in the following procedures: (See Fig. 8-16)

(1) Set switches and knobs as follows:

(a)	Cabin air selector switch	"OFF"
(b)	Auto/manual selector switch	"OFF"
(c)	Cabin temperature selector	Optional position HOT-COLD
(d)	Cabin altitude selector knob	Aiport elevation + 1,000 ft. (305 m)
(e)	Cabin rate of climb control knob	Optional position MIN-MAX
(f)	Manual pressure control valve	"INC"
(g)	Ground test valve	"OPEN"
(h)	Circuit breaker "AIR COND"	"ON"
(i)	Cabin air outlet select switch	"FLOOR"

(2) Start LH and RH engines and set switches as follows when engines have stabilized: (See Fig. 8-16)

(a)	Cabin air selector switch	"BOTH ENG"
(b)	Auto/manual selector switch	"AUTO"
(c)	Transfer switch	"LIGHT ON" (switch panel)

(3) Turn cabin temperature selector to full COLD and make sure that cold air comes out of the front and aft conditioned air outlets and cold air outlets. However, if cabin temperature is below approximately 41°F (5°C) the conditioned air does not become cold.

(4) Then, turn the selector to full HOT and make sure that hot air comes out of the conditioned air outlets and cold air comes out of cold air outlets. However, if cabin temperature is over approximately 77°F (25°C) the conditioned air does not become hot.

(5) Push transfer switch and operate cabin supply air temperature selector for full COLD and full HOT temperature conditions.

(6) Set auto/manual selector switch in MAN COLD and make sure that cold air comes out of each outlet (see Fig. 8-17).

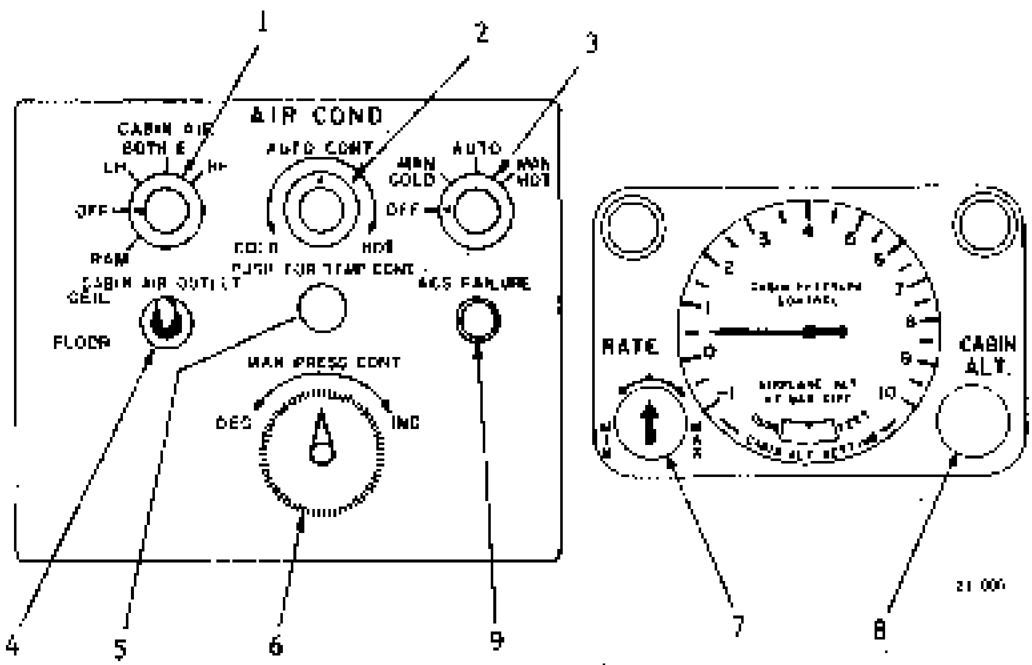
(7) Set the selector switch in MAN HOT and make sure that hot air comes out of conditioned air outlet and cold air out of air outlet (see Fig. 8-18).

(8) Return auto/manual selector switch to AUTO. After setting cabin temperature selector in optional position between HOT and COLD, set DEFOG COND selector knob of the forward conditioned air outlet to DEFOG and make sure that air comes out of front and side defog air outlets.

(9) Turn the cabin air outlet select switch to CEIL and make sure that airflow from cabin conditioned air outlets decreases and comes out of ceiling outlet.

(10) Set cabin air selector switch in RAM position and make sure that ram air comes out of conditioned air outlets (see Fig. 8-19). However, the amount of ram air during ground operation is not sufficient; therefore, the opening and closing operation of ram air shutoff valve and other valves of this system should be checked thoroughly.

(11) Return switches and knobs to OFF or their original positions, and stop engines.



1. Cabin air selector switch
2. Cabin supply air temperature switch
3. Auto-manual selector switch
4. Cabin air outlet select switch
5. Transfer switch
6. Manual pressure control valve
7. Cabin rate of climb knob
8. Cabin altitude select switch
9. Air Conditioning System Fail Warning Light (S/N 652SA and 661SA)

Fig. 8-16 Air Conditioning and Pressurization Control Panel

Key to Figures 8-17, 8-18, 8-19 and 8-20

1. Engine bleed air shutoff vlave
2. Ram air shutoff valve
3. Refrigeration unit bypass valve
4. Cooling turbine bypass valve
5. Cold air intake & screen
6. Precooler
7. Primary heat exchanger
8. Secondary heat exchanger
9. Air discharge port
10. Coalescer
11. Water aspirator
12. Ram air intake port

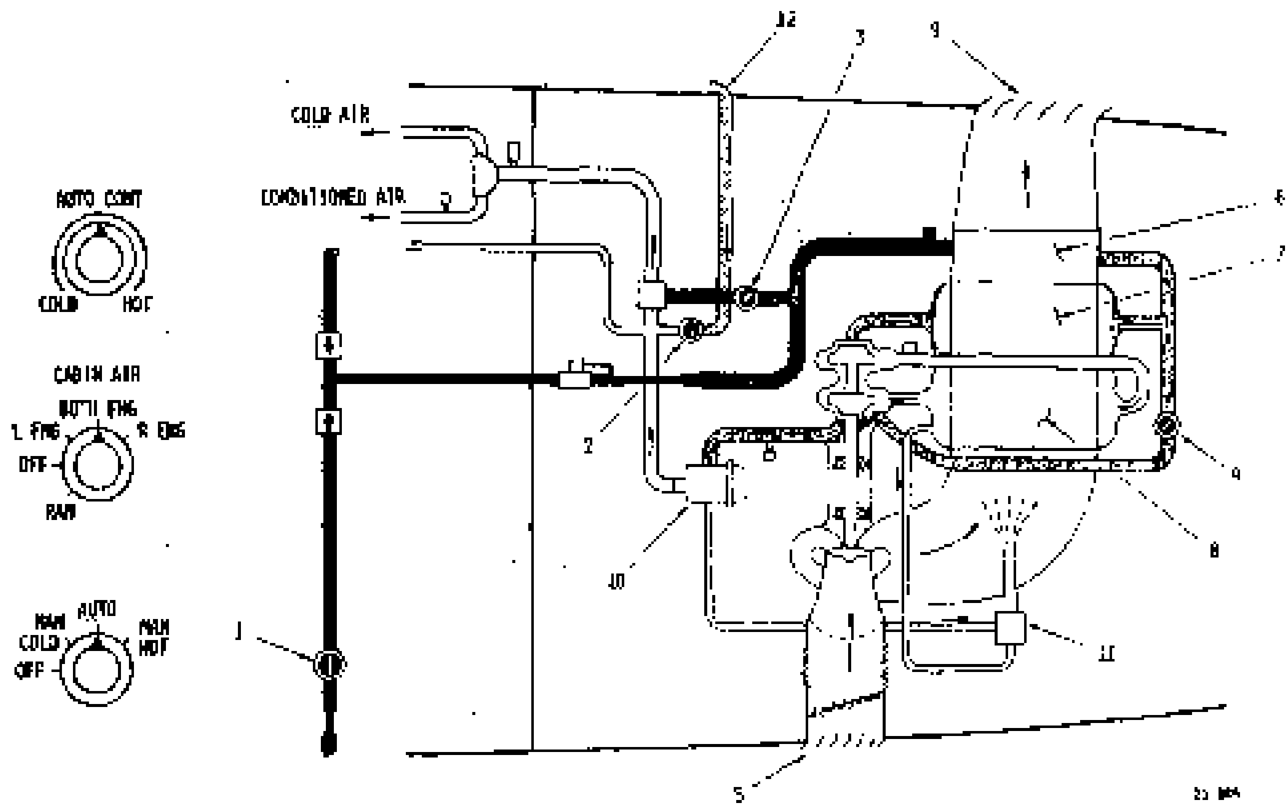


Fig. 8-17 Automatic Cabin Air Supply Temperature Control Condition

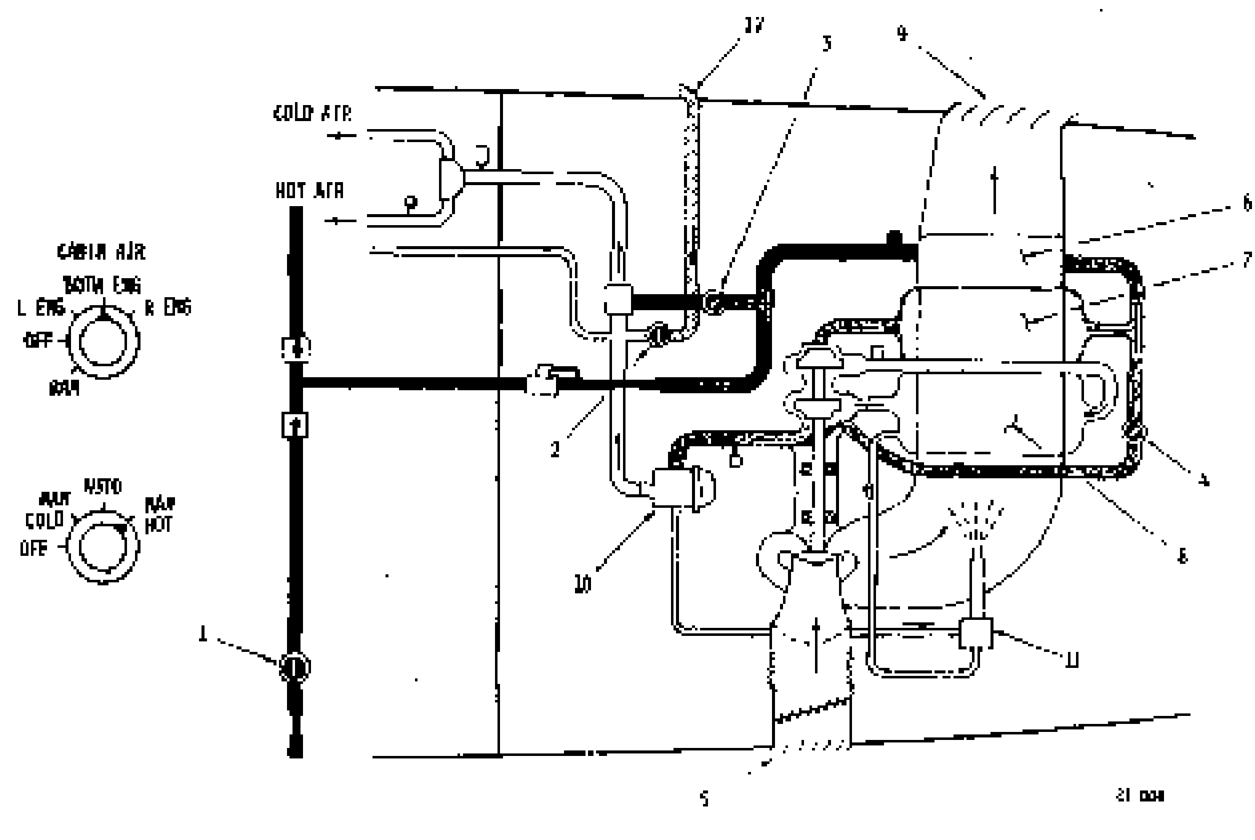


Fig. 8-18 Manual Heating Condition

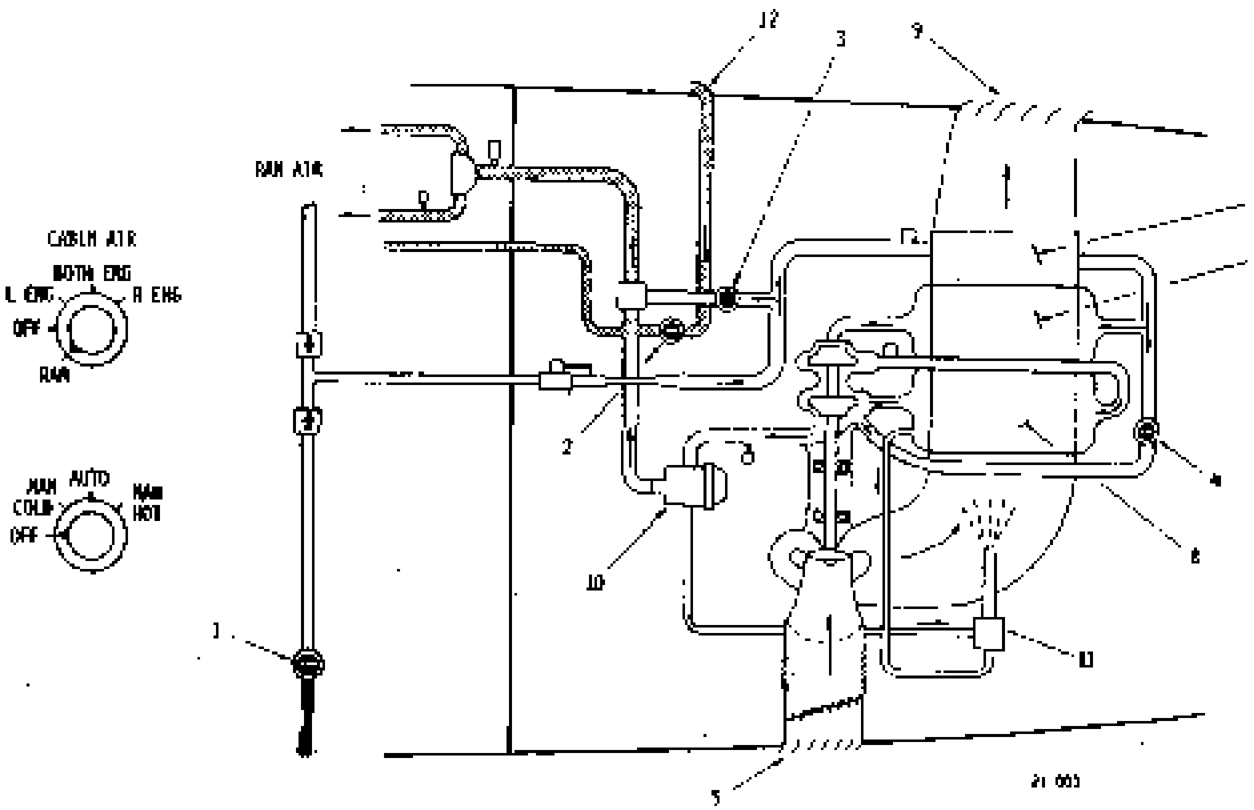


Fig. 8-19 Ram Air Ventilating Condition

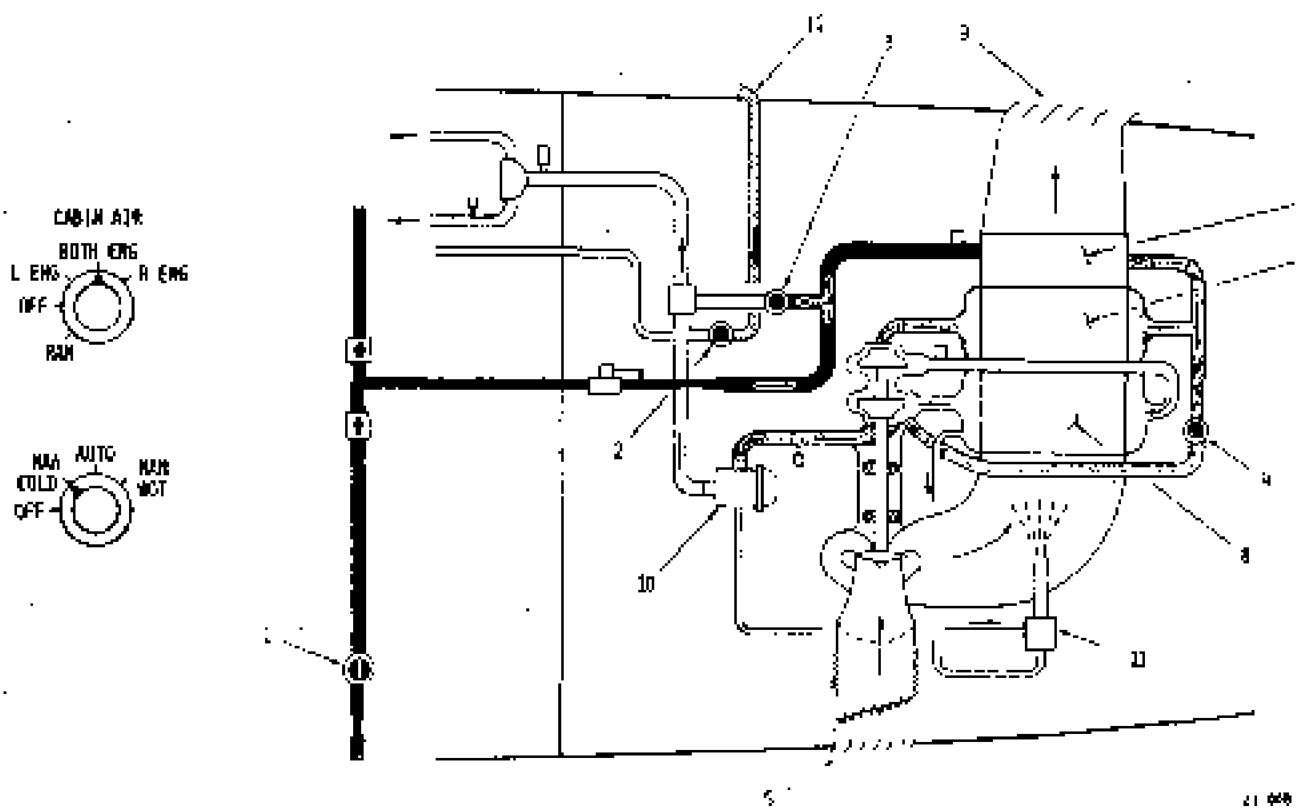


Fig. 8-20 Manual Cooling Condition

4.3 TROUBLE SHOOTING CABIN AIR CONDITIONING SYSTEM

Trouble	Probable cause	Remedy
Engine compressor bleed air is not available (or quantity of bleed air is too low)	a. Engine bleed air shut-off valve cannot be opened.	i. Check electrical circuit. ii. Replace valve (when valve is faulty.
	b. Engine bleed air check valve cannot be opened.	Replace valve.
	c. Bleed air tubing defective.	i. Check bleed air tubing. ii. Replace faulty components.
	d. Ram air shutoff valve not fully closed.	i. Check electrical circuit. ii. Replace valve (when valve is faulty).
Engine compressor bleed air cannot be stopped	a. Engine bleed air shut-off valve cannot be closed.	i. Check electrical circuit. ii. Replace valve (when valve is faulty).
Heating air cannot be obtained	a. Temperature control defective.	i. Check electrical circuit. ii. Check each component and replace faulty units.
	b. Refrigeration unit by-pass valve cannot be opened.	i. Check electrical circuit. ii. Replace valve (when valve is faulty).
Cold air is not obtained	a. Temperature control defective.	i. Check electrical circuit. ii. Check each component and replace faulty units.
	b. Refrigeration unit by-pass valve cannot be closed.	i. Check electrical circuit. ii. Replace valve (when valve is faulty).
	c. Cooling turbine bypass valve cannot be closed.	i. Check electrical circuit. ii. Replace valve (when valve is faulty).
	d. Cooling air intake or outlet is blocked.	Remove obstacle.
	e. Operation of refrige-ration unit is faulty.	i. Replace unit. ii. Try to turn the fan blades by a non-metallic probe.
Ram air ventilation can not be accom-plished	a. Ram air shutoff valve cannot be opened.	i. Check electrical circuit. ii. Replace valve (when valve is faulty).
	b. Ram air intake is blocked.	Remove obstacle.

Trouble	Probable cause	Remedy
Cabin air temperature fluctuates	a. Cabin supply air temperature sensor does not work normally.	Replace sensor.
	b. Cabin air temp. control does not work normally.	Replace control.
	c. Refrigeration unit does not work normally.	Replace the unit.
Temperature cannot be controlled when auto-manual selector is placed in "AUTO"	a. Cabin air temp. control does not work.	Replace the control.
	b. Cabin supply air temperature sensor does not work normally.	Replace sensor.
Very hot air flows out when auto/manual selector is placed in MAN HOT	a. High temperature limit switch does not work normally.	Replace limit switch.
	b. Operation of relay is faulty.	Replace relay.
	c. Refrigeration unit by-pass valve does not work normally.	Replace valve.
Cabin supply air contains too much moisture, or carries ice flakes	a. Operation of water separator inlet air temp. sensor is faulty.	Replace sensor.
	b. Operation of water separator inlet air temp. control is faulty.	Replace regulator.
	c. Condenser in water separator is blocked with dirt, oil, etc., so bypass valve opens.	Replace water separator.
Air does not come out of ceiling outlet when cabin air outlet select switch is turned to CEIL	a. Rotary section of the valve interferes with fixed section.	i. Disassemble the valve and adjust not to interfere.
	b. Rotary solenoid does not work.	i. Check electric circuit. ii. Replace rotary solenoid.
	c. Ceiling outlet or tubing is blocked.	i. Remove obstacle.
ACS FAIL light illuminates	a. Operation of engine bleed pressure regulator is blocked.	Replace pressure regulator.
	b. Operation of cooling fan is faulty.	Replace air cycle machine.
	c. Cooling air system is blocked.	Remove obstacle.

4.4 OPERATIONAL CHECK OF CABIN PRESSURE CONTROL SYSTEM

(1) Start engines per AIRPLANE FLIGHT MANUAL and stabilize at 100% rpm and 50% torque (condition lever; takeoff-land; power lever - torque 50%).

(2) Set air conditioning and pressurization controls as follows:

Cabin supply air selector	BOTH
Auto/manual selector	AUTO
Cabin altitude selector	Field elevation -1,000 ft.
Cabin rate-of-climb control	Adjust the arrow mark on knob to calibrated mark
Ground test valve	OPEN
Cabin Temp control	As desired
Manual pressure control valve	FULL INC
DOOR SEAL circuit breaker	Closed

(3) When the airflow volume into the cabin has stabilized, check the differential pressure indicator; the value should be approximately 0.5 psi.

(4) After confirming the differential pressure, set the controls as follows:

Manual pressure control valve	FULL DEC
Ground test valve	CLOSE

(5) Turn manual pressure control valve to FULL INC position. Note the following:

Differential pressure	4.80 to 5.25 (*1) 5.80 to 6.10 (*2)
Cabin rate-of-climb	Up approximately 300 ft/min.
Outflow safety valves	Functioning normally

WARNING

Should the differential pressure exceed the maximum limit, decrease the cabin pressure by turning the manual pressure control toward decrease. Shut down all systems and locate cause for cabin over-pressurization.

The differential pressure should not fluctuate.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

NOTE

The manual pressure control valve requires approximately 6.0 to 6.5 turns from FULL INC to FULL DEC. The cabin pressure will increase rapidly the last 1 to ½ turn before INC position, so valve operation should be carefully performed.

(6) Turn the manual pressure control valve from FULL INC to FULL DEC gradually. The cabin rate-of-climb indicator should indicate Down approximately 4,000 ft/min. and the differential pressure indicator should indicate "0" (zero).

(7) Set the cabin altitude selector to "0" (zero) feet and the manual pressure control valve to FULL INC.

(8) Adjust each engine rpm to ground idle.

(9) Set all air conditioning and pressurization controls as described in paragraph (2) except the cabin altitude selector, which should be set to "0" (zero) feet.

(10) Shut down engines per AIRPLANE FLIGHT MANUAL.

(11) Open the ground test valve and install safety wire.

4.5 CABIN PRESSURE LEAKAGE TEST AND OPERATIONAL CHECK FOR OUTFLOW SAFETY VALVE

(1) Disconnect conditioned air tube and cold air tube on the side of the rear bulkhead (F.STA 8035), and plug tube ends on bulkhead side. Plug vacuum line for gyro. Pull out circuit breaker LDG 'POS IND'.

(2) Disconnect atmospheric pressure sensing tubes of outflow safety valves and plug the two unions.

(3) Remove RH lower access panel at STA 1080 and plug atmospheric port of venturi (*1). Optionally select positions for cabin altitude selector knob, and cabin rate of climb control knob.

(4) Set ground test valve lever to CLOSE located at right side front of co-pilot pedal.

(5) Close circuit breaker DOOR SEAL and actuate selector valve for entrance door seal.

(6) Connect pressurizer to the port (STA 1080 bulkhead, LH) for pressure test and pressure sensing line to sensing port (STA 1080 bulkhead, RH) respectively.

*1 Aircraft S/N 652SA

(7) (*1) Increase airpressure in pressurizer gradually and stabilize at 5.0 ± 0.1 psi. Leakage should be less than 70 cu. ft/min.; should this limit be exceeded, seal the leaking points.

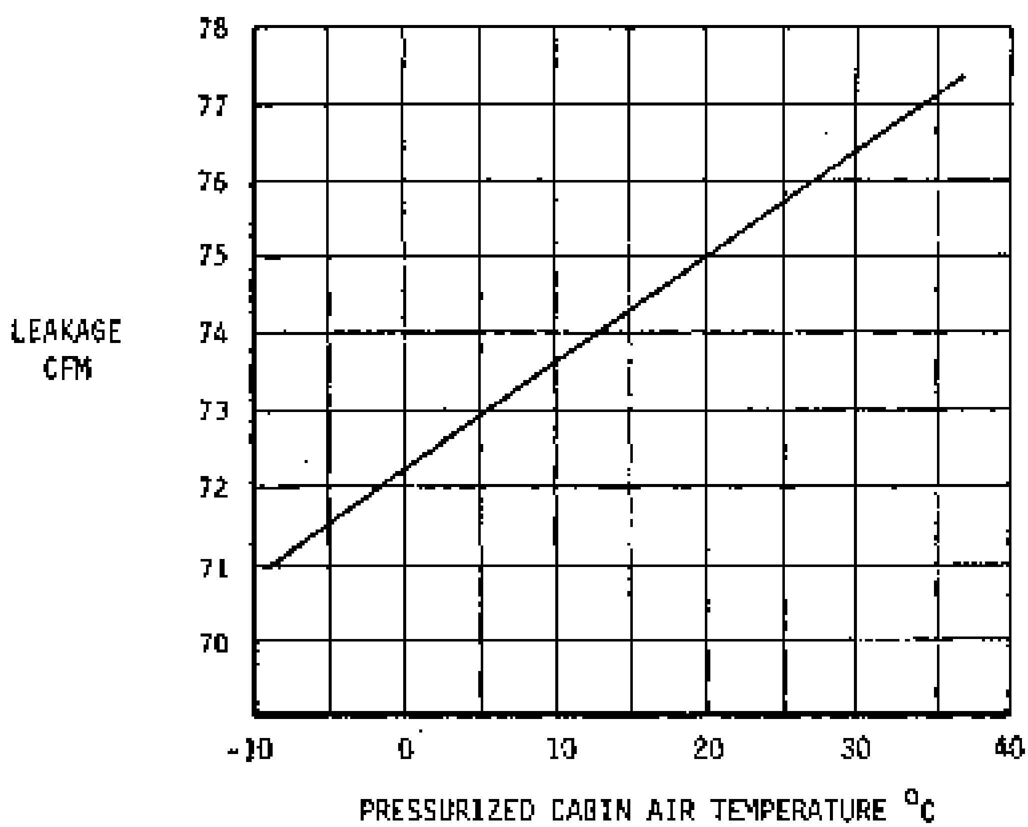
(*2) Increase air pressure in pressurizer gradually and stabilize at 6.0 ± 0.1 psi. Leakage should be less than the value shown in Fig. 8-24. Should the limit be exceeded, seal the leaking points.

(8) After confirming Para. (7), reduce pressure and reconnect atmospheric pressure sensing tubes of outflow safety valve to normal tubing. Then, pressurize cabin again and check for pressure relief at 5.0 +0.25/-0.10 psi (*1), 6.0 ± 0.1 (*2) psi. If relief does not work when air pressure is increased up to 5.35 psi (*1), 6.1 psi (*2), outflow safety valve is defective. Replace it.

(9) After completion of test, restore airplane to original condition.

CAUTION

After test, set ground test valve to full open position (valve handle is parallel to tube), and apply lock wire.



Allowable leakage of pressurized cabin

*1 Aircraft S/N 652SA

*2 Aircraft S/N 661SA, 697SA and subsequent

4.6 TROUBLE SHOOTING CABIN PRESSURE CONTROL SYSTEM

Trouble	Probable cause	Remedy
Pressurization cannot be obtained	a. Leakage from airplane is excessive.	Repair defective seal.
	b. Foreign matter accumulated on valve poppet seat.	Remove foreign matter.
	c. Cabin altitude selector knob is not set properly.	Reset at desired altitude.
	d. Cabin pressure air outflow valve control is faulty.	Replace control.
	e. Outflow safety valve is defective.	Replace valve.
	f. Air supply to cabin is low.	Check supply air system for tube damage, defective connections, etc.
	g. Manual pressure control valve is open.	Close valve (to INC direction).
	h. Pneumatic relay is faulty.	Replace relay.
Cabin pressure is higher than selected	a. Tube connections between cabin air outflow valve control and outflow safety valve are loose.	Tighten loose connections.
	b. Outflow safety valve is defective.	Replace valve.
	c. Internal defect of cabin air outflow valve control.	Replace control.
	d. Pneumatic relay is defective.	Replace control.
Selected cabin altitude cannot be maintained	a. Tube connection between cabin air outflow valve control and outflow safety valve is faulty.	Check tubing and tighten loose connections.

Trouble	Probable cause	Remedy
	b. Cabin air outflow valve control is faulty.	Replace valve.
	c. Outflow safety valve is faulty.	Replace valve.
Maximum differential pressure exceeds limit	a. Outflow safety valve is faulty.	Replace valve.
	b. Connection of static tube of outflow safety valve is faulty.	Tighten loose connection.
	c. Cabin altitude and differential pressure gauge is faulty.	Replace gauge.
Manual pressure control cannot be performed properly	a. Manual pressure control valve is faulty.	Replace valve.
	b. Tube between manual pressure control valve and jet pump venturi or ejector is loose or blocked.	Tighten loose connection. If blocked, remove obstacle.
Cabin pressure varies	a. Water separator temperature controller or its temp. sensor is at fault.	Replace controller or sensor.
	b. Water separator is faulty.	Check the bag for contamination and clean or replace if necessary.
	c. Cabin temp. control system is faulty.	Replace controller.

5. AIR SUPPLY SYSTEM

5.1 GENERAL DESCRIPTION

This system consists of tubing, flexible tube assembly and pressure control valve, and supplies air to the surface de-icer system, windshield anti-icing system (if installed), cabin pressure control system, entrance door sealing system, tip tank fuel transfer system, vacuum system and auto-pilot system.

The compressed hot air from the last stage of the L. H. and R. H. engine compressors is led to the fuselage through the lines in the wing trailing edge, regulated by the pressure regulator to a constant pressure and divided to each system.
The check valves installed in the lines in the wing are to prevent the reverse flow of hot air from the operating engine into the compressor of the dead engine, should one engine become inoperative.
This system is sufficient to supply air to each system even if one engine is inoperative. (See Fig. 8-21)

5.2 PRESSURE CONTROL VALVE (18 PSI) (See Fig. 8-22)

This valve regulates engine bleed air, air source for the ejector which produces gyro driving vacuum, to 18 psi (1.3 kg/cm²). This valve also has a pressure relief function, and it works when the valve outlet pressure exceeds 20 psi.(1.4 kg/cm²).

5.2.1 REMOVAL AND INSTALLATION

(a) Remove center wing rear access door.
(b) Remove tube assembly, upstream and downstream of the valve.
(c) Remove four screws from the valve attaching bracket and remove valve.
(d) Remove reducer and union upstream and downstream of valve and cap valve ports.
(e) Install in reverse sequence of removal.

Fig. 8-21 Pressure control valve

5.3 PRESSURE CONTROL VALVE (15 PSI)

This valve regulates engine bleed air, air sources for the surface de-icer system, windshield anti-icing system (if installed), cabin pressure control system, entrance door sealing system and tip tank fuel transfer system, to 15 psi (1 kg/cm^2).

5.3.1 REMOVAL AND INSTALLATION

See Chapter IX, Para. 2.2.1.

maintenance manual

TEMPORARY REVISION NO.8-1

This Temporary Revision No. 8-1 is applicable to the following Maintenance Manuals :

MODEL NO.	REPORT NO.	PAGE
MU-2B-35/-36A	MR-0218	8-47
MU-2B-60	MR-0336	8-47

Insert facing the page indicated above for the applicable Maintenance Manual.

Retain this Temporary Revision until such time as a permanent revision on this subject is issued.

REASON : To add Tubing Inspection and Repair.

ADD : Paragraph 5.4 as follows.

5.4 TUBING INSPECTION AND REPAIR

5.4.1 PREPARATION

(1) Remove access panels and equipment as required to gain access to the pneumatic line to be inspected as shown in Figure 8-22A.

(2) Remove all lines supporting clamps identified on the figure and retain these clamps and attachment hardware for reinstallation.

5.4.2 INSPECTION

(1) Visually inspect the clamped surface of the pneumatic lines for any signs of corrosion. Use a mirror and flashlight to check the entire periphery surface of clamp attachment area

a. If inspection does not reveal corrosion damage (slight oxidized material without pitting or discoloration is acceptable), it is acceptable to reuse. Proceed to section 5.4.3 for surface refinish.

b. If inspection does reveal corrosion damage, remove and replace the tube assembly. Proceed to section 5.4.4 for local fabrication of replacement tube assembly

(2) Visually inspect the clamp for signs of corrosion, burning, hardening, or absence of the rubber gasket. Remove and replace the clamp with a new one as necessary.

NOTE

Unauthorized clamps, especially steel clamps, are susceptible to galvanic corrosion and could cause a loss of pressurization, pneumatic air, and/or surface deice systems.

(3) Visually inspect each clamp removed from the aircraft and verify the usage of the correct part number that is specified in figure as applicable.

5.4.3 SURFACE REFINISH OF TUBE ASSEMBLY

The following steps are applicable to the all tube assemblies that are confirmed serviceable after the inspection in Paragraph 5.4.2.

Before restoring plumbing of those pneumatic lines, treat the clamp contact area on the tube assembly as follow

NOTE
Use only fine grade sandpaper #400 with aluminum oxide or silicon carbide sandpaper.
No other material is permitted.

5.4.3.1 For Electrical Bonding Surface (See Note-b in the figure 8-22A)

(1) Remove light corrosion and deposits from the contact area of the bonding clamp by sanding with fine #400 sandpaper as shown in Figure 8-22B.

(2) Clean all sanded surfaces with Methyl Ethyl Ketone, TT-M-261.

NOTE
Do not disturb the treatment surface.
Avoid contact with body oils or acids which may cause corrosion or prevent adhesion of alodine film.

(3) Apply alodine #1200, chemical film.

(4) Keep the treated area wet for approximately 3 to 5 minutes until a yellow color develops.

(5) After the alodine has changed color, rinse the area with a wet cloth and blow dry with shop air.

5.4.3.2 For Non-Bonding Surface

The clamp attachment positions where the top coating exhibits light corrosion and/or peeled paint appearence, treat the surface per following steps.

(1) Thoroughly sand the area of the damaged coating with fine #400 sandpaper.

(2) Clean all sanded surface with Methyl Ethyl Ketone, TT-M-261.

NOTE
Do not disturb the treatment surface.
Avoid contact with body oils or acids which may cause corrosion or prevent adhesion of alodine film.

(3) Apply alodine #1200, chemical film, to area of exposed metal.

(4) Keep the treated area wet for approximately 3 to 5 minutes until a yellow color develops.

(5) After the alodine has changed color, rinse the area with a wet cloth and blow dry with shop air.

(6) Touch-up with one coat of zinc-chromate primer TT-P-1757.

5.4.4 FABRICATION OF TUBE ASSEMBLY

Replacement tube assembly may be locally fabricated per the following steps.

(1) Refer to the Table 8-1 in order to prepare parts and material for tube assembly.

(2) Cut and form the tube to fit with actual mold made from the existing tube assembly.

(3) Install the coupling nuts (2 ea) and sleeves (2 ea) before flaring the tube ends.

(4) Make single flare at both ends of tube per MS33584.

(5) Degrease and flush tube using Methyl Ethyl Ketone, TT-M-261.

(6) Cap or mask both ends of tube to prevent contamination

NOTE
Do not disturb the treatment surface.
Avoid contact with body oils or acids which may cause corrosion or prevent adhesion of alodine film.

(7) Apply alodine #1200 solution with a brush to the entire external surface of the tube assembly. Keep treated area wet for approximately 3 to 5 minutes until a yellow color develops.

(8) After the alodine has changed color, rinse the surface with water and blow dry with shop air.

(9) Mask off all contact surfaces of the bonding clamp contact area 1-1/2 times the clamp width as shown in Figure 8-22B preparation of electrical bonding surface. (Refer to mark ⟨A⟩ in Figure 8-22A for the location of ground bonding)

(10) Mask off external surface of both ends of tube approximately 1 inch from the end of tube.

(11) Apply one coat of zinc chromate primer TT-P-1757 with brush to the exterior surface of tube assembly.

(12) Remove masking tape.

TABLE 8-1 TUBING MATERIAL AND END FITTINGS

TUBE ASSEMBLY	MATERIAL (SPEC.)	OUTER DIAMETER	THICKNESS	COUPLING NUT	SLEEVE
030A-64304-17	5052-0 tube (WW-T-700/4)	5/8 inch dia.	0.035 inch	AN818-10D (2ea)	MS20819-10D (2ea)
030A-64304-67	5052-0 tube (WW-T-700/4)	5/8 inch dia.	0.035 inch	AN818-10D (2ea)	MS20819-10D (2ea)

5.4.5 RESTORATION

(1) Restore the serviceable or fabricated tube assembly.

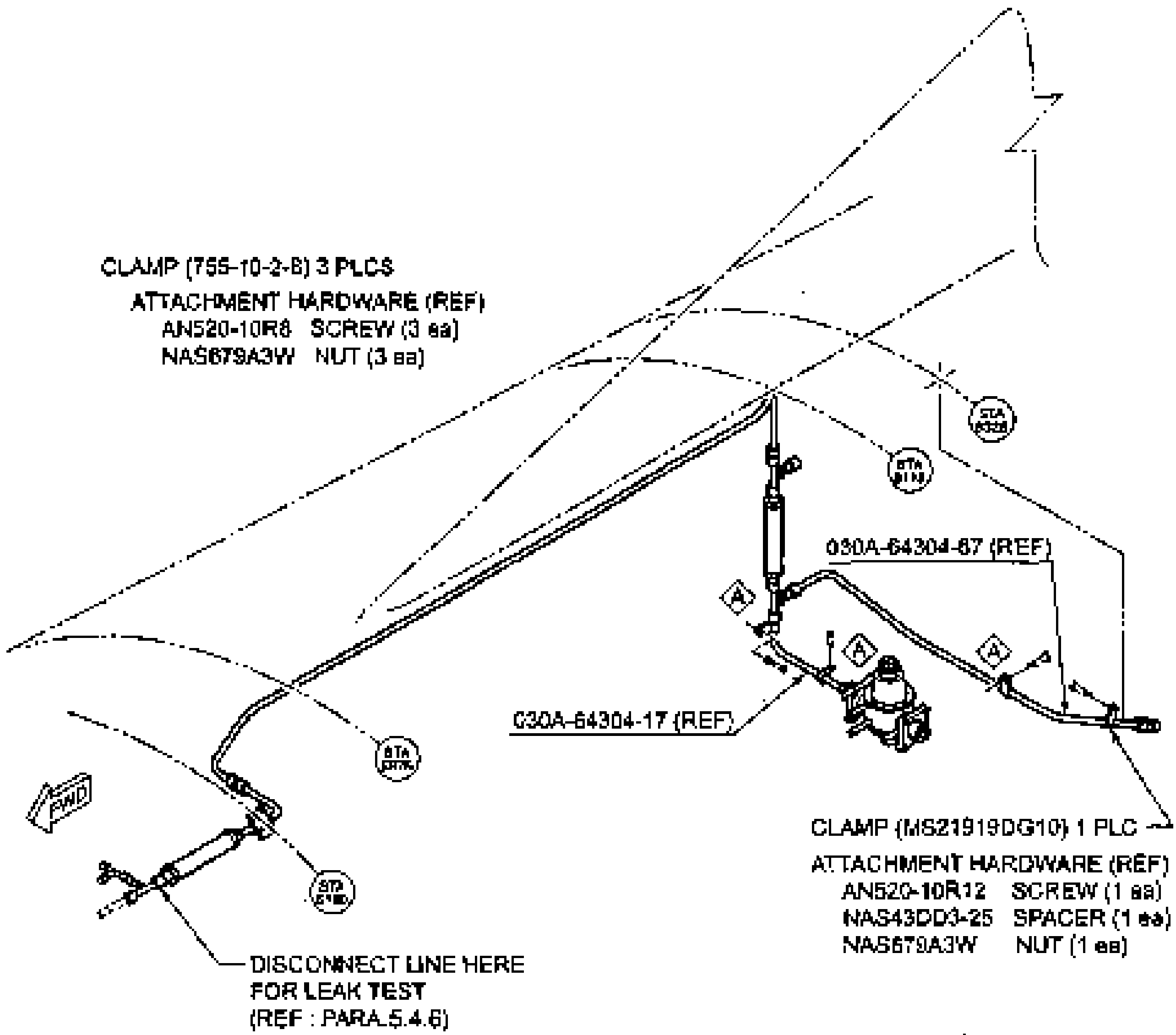
(2) Reinstall all line supporting clamps that were removed in section 5.4.1 for inspection.
- Sand and clean all contact surfaces on the clamp with a suitable solvent before installation.

5.4.6 LEAK TEST

Perform the pneumatic system leakage test to ensure proper restoration of the reinstalled tube assembly as follow.

(1) Disconnect pneumatic line at the location as shown in the applicable figure.
(2) Connect air pressure source to the system and supply air pressure gradually and set at 100 psi.
(3) Check for leaks from the line connections while air pressure is applied.
(4) Disconnect pressure source and restore the system.

TEMPORARY REVISION NO.8-1



NOTE

a Remove all the clamps deploied in this figure and inspect the clamp contact surfaces on the tube assembly and the clamps.
(Ref. Paragraph 5.4.1)

b Clamp position identified with A indicates the special clamp position that function as an electrical bond and requires bonding surface treatment at the clamp attachment position on the tube assembly.
(Ref. Paragraph 5.4.3.1, 5.4.4 (9) and Figure 8-22B)

Fig. 8-22A Inspection Areas

TEMPORARY REVISION NO.8-1

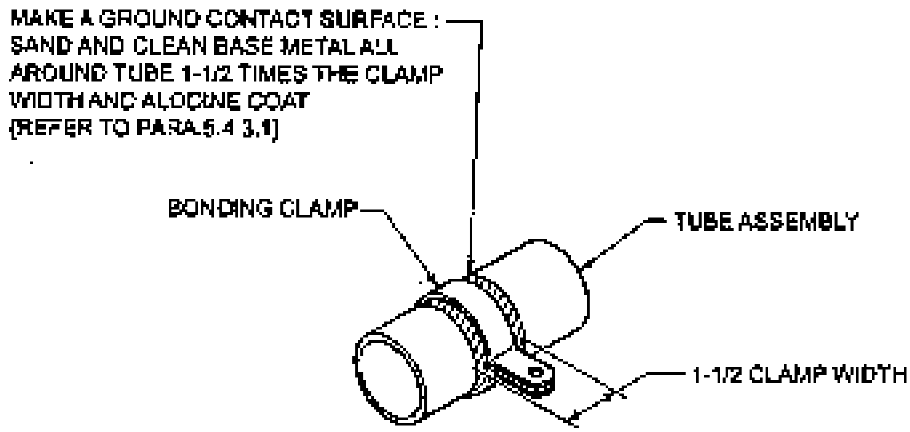


Fig. 6-22B Electrical Bonding Clamp Installation

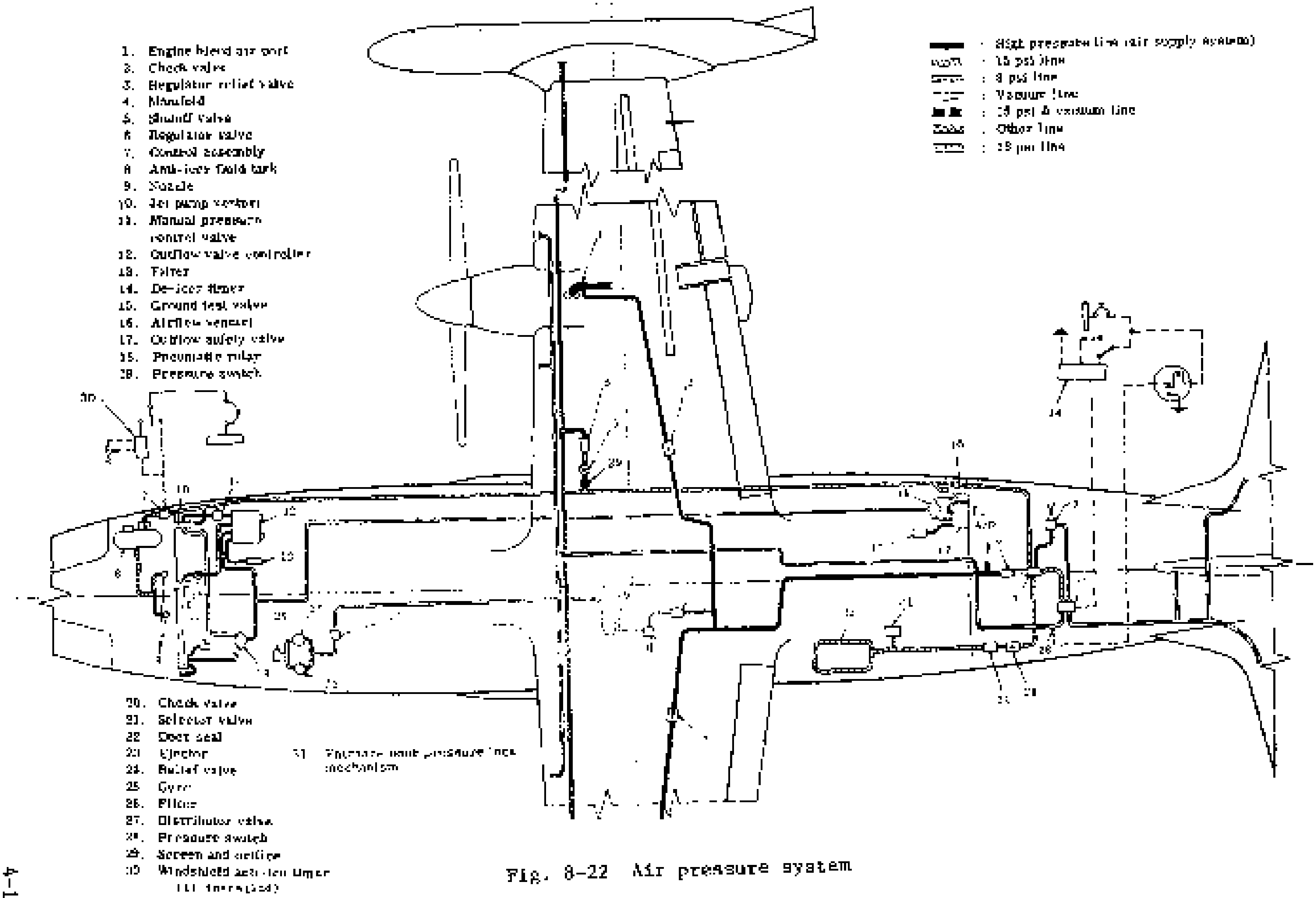


Fig. 8-22 Air pressure system

CHAPTER

9

ANTI-ATMOSPHERIC SYSTEM

CHAPTER IX

ANTI-ATMOSPHERIC SYSTEMS

CONTENTS

1. GENERAL (Fig. 9-1)

A pneumatic rubber boot de-icing system is installed on the wing and empennage leading edges. Electrically heated anti-ice and de-ice systems are utilized for various components which are pitot tube, static port, stall warning lift transducer, propellers, windshields and oil cooler air inlets. The engine air intake is anti-iced by means of engine compressor bleed air. Cabin conditioned air is supplied to all windows for defogging between the double window panes. A wiper is provided for removal of rain and slush ice from the windshields.

2. SURFACE DE-ICING SYSTEM (Fig. 9-2)

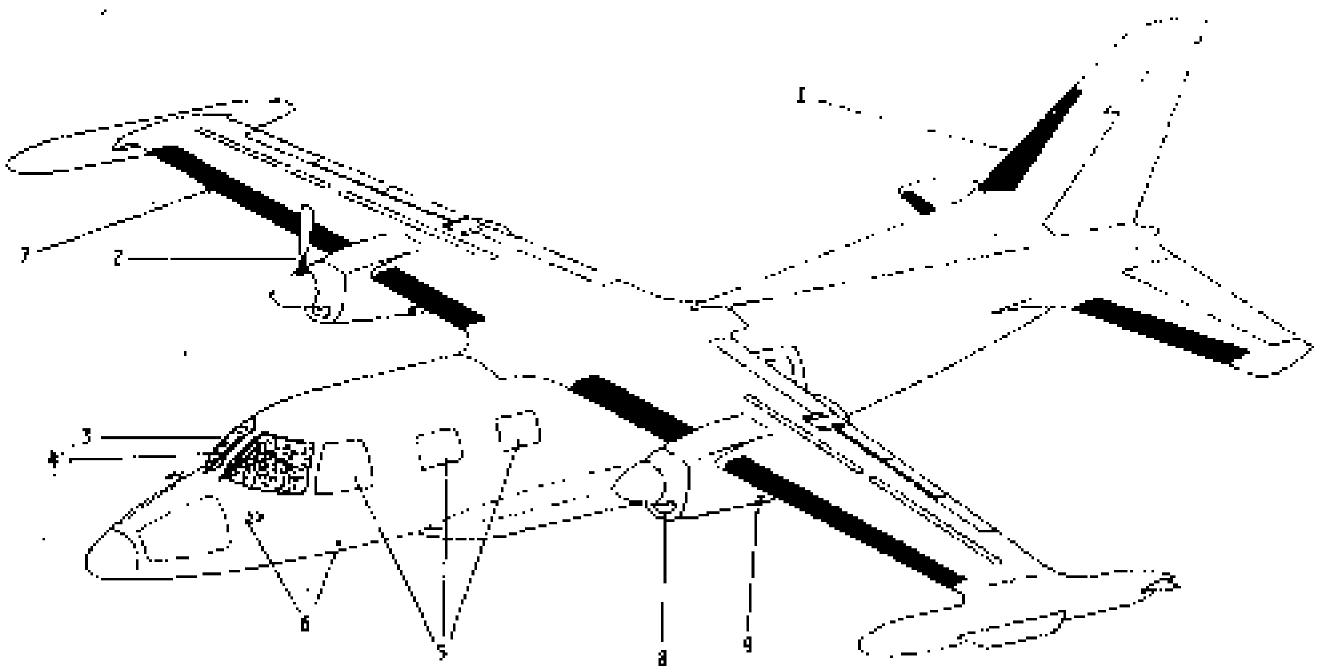
2.1 GENERAL DESCRIPTION

De-icing of the wing and empennage leading edges is accomplished by inflating and deflating rubber boots attached to the leading edges. The system consists of a pressure regulator and relief valve, manifold, distributor valve, ejector, timer and required tubing.

When the WING DE-ICE switch in the overhead console is placed to the ON position, a timer is energized and the pressurization port of the distributor valve is opened in response to the timer signal, thus allowing air pressure to inflate the boots causing ice removal from the leading edge surfaces. A pressure switch is installed in the tube to the wing boots causing an indicator light in the overhead console to illuminate. Deflation occurs when the timer signals the distributor valve to close the pressure port and open the negative pressure (suction) and exhaust ports. During automatic operation, cycle time of inflation and deflation is approximately three minutes, but manual control can shorten the cycle to 16 to 18 seconds. When the switch is in the OFF position, the pressurization and exhaust ports are in the closed position and the negative pressure (suction) port is open. The negative pressure (suction) is connected to the ejector, causing a negative pressure in the de-ice boots, so that the boots do not inflate during flight.

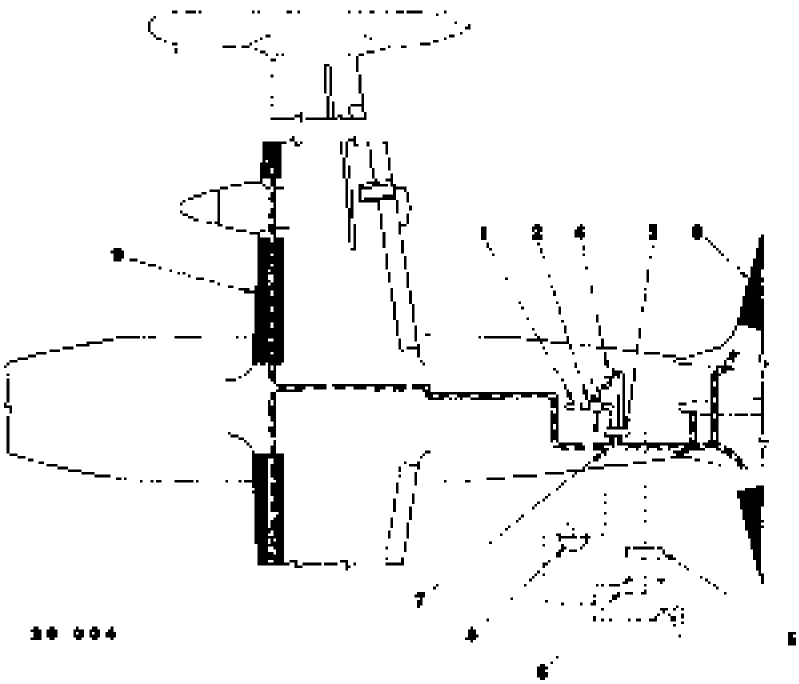
2.2 PRESSURE REGULATOR AND RELIEF VALVE (Fig. 9-3)

Engine compressor bleed air is the air source for the surface de-icing system, and is regulated at 15 psi (1.1 kg/cm^2) by this valve. Regulated air flows to the distributor valve for pressurization and ejector for high pressure air source. The valve also has a pressure relief function, should the outlet pressure exceed 17.5 psi (1.2 kg/cm^2).



1. Pneumatic de-icing
2. Propeller anti-icing
3. Windshield anti-icing
4. Wiper
5. Windshield and cabin window defogging
6. Pitot tube and static port anti-icing
7. Stall warning transducer anti-icing
8. Engine air intake anti-icing
9. Oil cooler inlet anti-icing

Fig. 9-1 Anti-atmospheric system (Typical)



1. Regulator & relief valve
2. Manifold
3. Distributor valve
4. Ejector
5. Timer
6. De-Icer switch
7. Pressure switch
8. Indicator light
9. De-Icer boots

Fig. 9-2 Surface de-icer equipment and system

2.2.1 REMOVAL AND INSTALLATION

(1) Remove LH electrical compartment door (F.STA 8615).

(2) Disconnect tube at the upstream side of the pressure regulator and relief valve.

(3) Disconnect manifold from the downstream side of the valve.

(4) Remove valve attaching screws (3 ea.) and remove the valve.

(5) Remove reducer from the upstream and downstream ports; cap the ports.

(6) Install in reverse sequence of removal, installing safety wire to the attaching screws, and apply alignment mark to the coupling nuts.

2.3 MANIFOLD

The manifold divides the regulated air pressure for the boot de-icing system, fluid anti-icing system (if installed) and also to the entrance door seal and pressure lock. The manifold is installed at the downstream side of the pressure regulator and relief valve.

2.3.1 REMOVAL AND INSTALLATION

(1) Remove LH electrical compartment door (F.STA 8615).

(2) Disconnect coupling nut from the upstream side of the manifold.

(3) Disconnect coupling nuts (3 ea.) from the downstream side of the manifold.

(4) Loosen and/or remove the hose clamp from the downstream side of the manifold, upstream side of the ejector.

(5) Remove the manifold.

(6) Install in reverse sequence of removal. After tightening nuts and hose clamp to proper torque value, apply alignment mark.

Fig. 9-3 Pressure regulator and relief valve

Fig. 9-4 Distributor valve

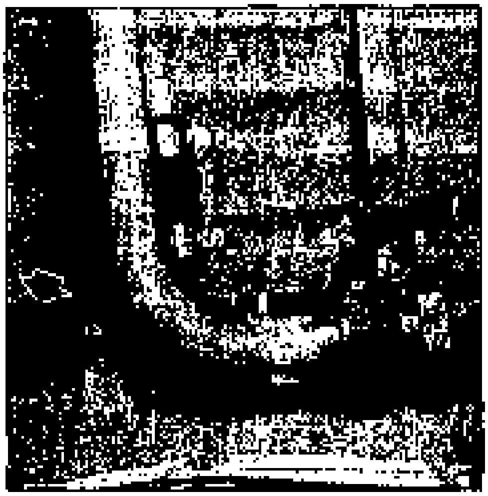


Fig. 9-5 Ejector

Fig. 9-6 Timer

Fig. 9-7 Pressure switch

2.4 DISTRIBUTOR VALVE (Fig. 9-4)

The distributor valve is connected to the manifold on the upstream side and to each boot on the downstream side. The valve incorporates a solenoid to operate the valve's three functions: boot inflation, boot deflation and a valve for the ejector to maintain a negative air pressure in the boots. These functions are controlled by signals from the timer.

2.4.1 REMOVAL AND INSTALLATION

(1) Remove LH electrical compartment access door (F.STA 8615).

(2) Disconnect wiring connector from the distributor valve.

(3) Disconnect tubing joints (4 ea.).

(4) Remove safety wire and 2 bolts installing the valve.

(5) Remove the valve and cap each port.

(6) Install in reverse sequence of removal. After tightening attaching bolts, install safety wire. After tightening hose clamps apply alignment marks to each.

2.5 EJECTOR (Fig. 9-5)

The ejector is connected to the negative pressure (suction) port of the distributor valve and to the manifold. The air pressure from the manifold causes a negative pressure (suction) on the distributor valve, thus causing the boots to remain deflated with the system OFF and aids in deflation of the boots with the system ON.

2.5.1 REMOVAL AND INSTALLATION

(1) Remove the LH electrical compartment access door (F.STA 8615).

(2) Loosen and/or remove hose clamps (2 ea.) from the upstream side (manifold and distributor valve) of the ejector.

(3) Remove nut, screw and clamp installing the ejector to the airframe.

(4) Remove the ejector.

(5) Install in reverse sequence of removal. Apply alignment mark to hose clamps after tightening to proper torque value.

2.6 TIMER (Fig. 9-6)

The timer sends a signal to actuate the distributor valve periodically so that the boot inflation and deflation cycle is completed. During automatic operation, the timer sends a signal of 6 seconds ON and 174 seconds OFF, for a complete cycle (3 minutes). When the WING DE-ICE switch is placed to OFF, approximately 2 seconds is required for the timer to return to the starting position.

2.6.1 REMOVAL AND INSTALLATION

(1) Remove LH electrical compartment access door (F.STA 8615).

(2) Disconnect wiring connector.

(3) Remove attaching screws (4 ea.).

(4) Remove timer.

(5) Install in reverse sequence of removal.

2.7 PRESSURE SWITCH (Fig. 9-7)

The pressure switch is located in the pressure line between the distributor valve and the wing boots. The switch is closed when air pressure increases to 10 ± 2 psi (0.7 ± 0.1 kg/cm^2), illuminating an indicator light in the overhead console. The indicator light is illuminated when the surface de-icer system is in operation and the system is functioning normally.

2.7.1 REMOVAL AND INSTALLATION

(1) Remove LH electrical compartment access door (F.STA 8615).

(2) Disconnect wiring from switch and identify.

(3) Remove switch from tubing.

(4) Install in reverse sequence of removal, noting the following:
   (a) Tighten to proper torque and apply alignment mark.
   (b) Remove wiring identification, connect to switch and install wire shielding as required.

2.8 RUBBER BOOTS

The de-icer boots are long thin plates made of rubber and are inflated and deflated by means of compressed air to break away the ice mechanically. Service life depends on handling and maintenance practices to a great extent.

2.8.1 MAINTENANCE

(1) General

(a) Direct contact of ladders and worktables to leading edge boots should not be made during airplane maintenance, etc. Apply pads of cushion materials, etc. when contact with boots is unavoidable.
(b) Do not drag hoses over boots when servicing fuel or oil to airplane. Use protective pad in such cases.
(c) Do not step on boots when doing airplane maintenance, etc. Do not put tools and equipment on boots.

(2) Cleaning of boots

After flights, checks for oil, fuel and other harmful objects to boots. Give special attention to horizontal stabilizer boots, because fuel and oil coming out of engine tend to adhere to them. These materials should be removed as soon as possible.

(a) Clean de-icer boots with hot neutral detergent or hot water of no higher than 180°F (82°C).
(b) During cold seasons, clean boots at the inside of warm hanger if possible. Preheat neutral detergent or water if cleaned at the outdoors.
(c) If freezing may occur, warm with portable heater being careful not to heat boots surface to 180°F or over.
If it is inevitable to use petroleum type solvents in cleaning, wipe lightly with clean cloth moistened with the solvent, and then quickly wipe dry with clean dry cloth before the solvent penetrates into rubber. Because these petroleum type solvents give adverse effects to rubber, they must be used sparingly.

(3) Surface coating (Application of ICEX)

(a) ICEX is a silicone based material specially compounded to reduce adhesive force between ice and rubber surface, harmless to rubber, and gives protection against ozone.
(b) When ICEX is applied, smooth glossy film is formed on the surface, and the boot surface with tiny microscopic irregularity becomes flat. A chance of icing is reduced, and ice is quickly and completely removed when boots are operated.
(c) Give attention to the fact ICEX is not the greatest in de-icing ability. The effects of application of ICEX is not in prevention of icing or removal of ice, but in prevention of strong adherence of ice to make removal easier.
(d) Boots tend to be worn away by dust in air during flight. It is recommended to reapply ICEX in every 150 hours of flight.
(e) ICEX can be purchased from the following maker.

Nomenclature	Maker's Name	Address
ICEX	YOKOHAMA RUBBER CO.	5-chome, Shinbashi, Minato-ku, Tokyo-to

(4) Environmental conditions

Because de-icer boot is a rubber product it ages when placed outdoors for a long time. This phenomenon is expedited in very hot or high ozone density areas. Therefore, it should be avoided to place airplanes equipped with boots in the following areas for a long time as much as possible.

(a) Near motors or other machines which generate ozone.
(b) High temperature areas over 160°F (70°C).
(c) Areas with sunshine, harmful vapors, or unreasonably dense dust.

2.9 OPERATIONAL CHECK

Operational check of the surface de-icing system on the ground may be accomplished as follows:

Required equipment:

Air supply source capable of 30 to 50 psi (2.1 to 3.5 kg/cm^2) with a regulator and gauge.

(1) Remove LH electrical compartment access door (F.STA 8615).

(2) Locate and remove plug cap from the ground test port.

(3) Connect ground air supply to the test port and apply air pressure gradually until 30 psi (2.1 kg/cm^2) is attained.

(4) Connect APU and turn the Battery Key switch to ON.

(5) Check for WING DE-ICE circuit breaker "closed" condition and press the WING DE-ICE indicator light for illumination check.

(6) Place the WING DE-ICE switch in the overhead console to the ON position.
(a) Check for uniform inflation and deflation of each boot; approximate inflation height should be 0.25 in. (6 mm).
(b) Check for the indicator light illumination during the inflation and going out during deflation.
(c) The inflation and deflation time shall be within the acceptable area in Charts 9-1 and 9-2.

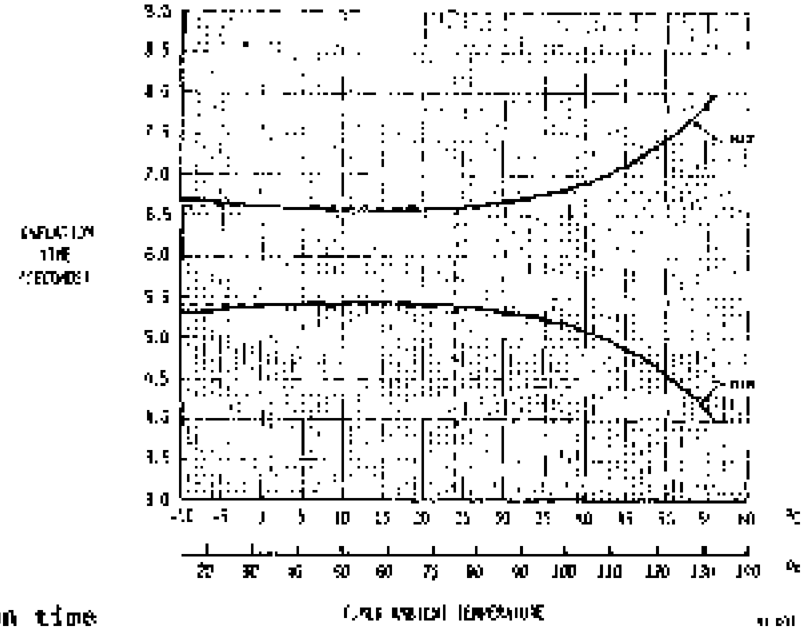


Chart 9-1 De-Ice boot inflation time

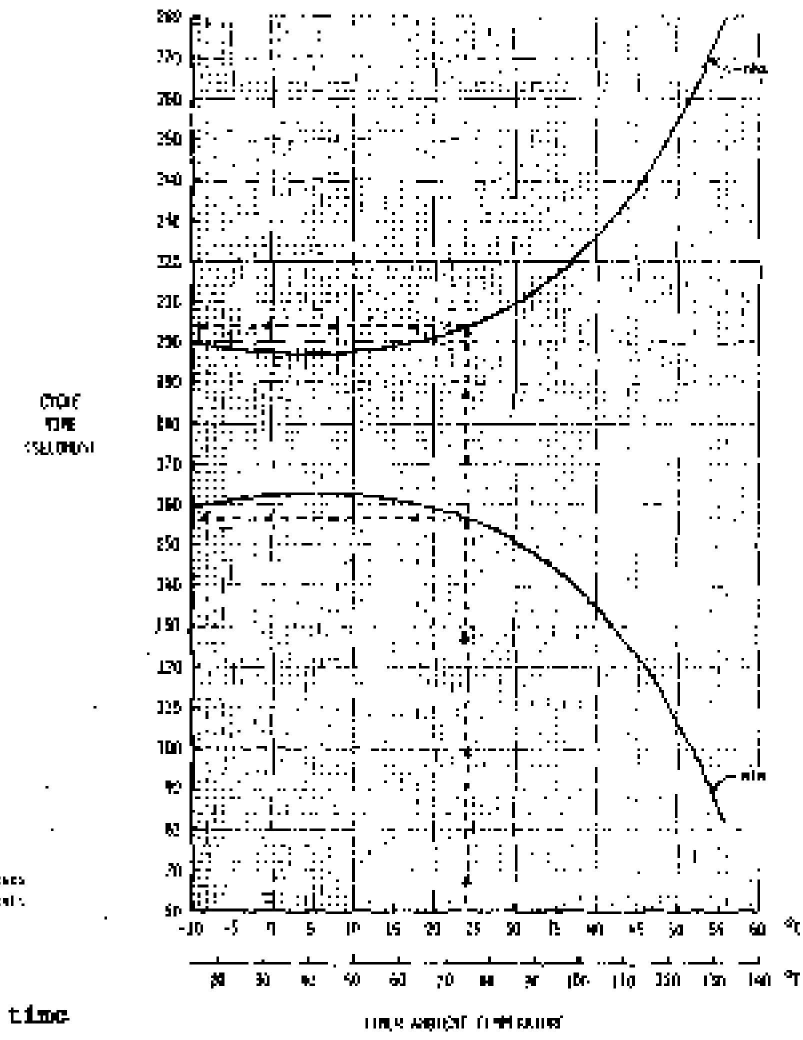


Chart 9-2 De-Ice boot cycle time

(7) Place WING DE-ICE switch to OFF and ON again 3 to 4 seconds later to ensure the timer has reset to the inflation (start) position of the cycle.

(8) Close the air supply source and release the air pressure in the surface de-ice system.

(9) Place WING DE-ICE switch and turn the Battery Key switch to the OFF position.

(10) Remove the air supply source from the test port and install the plug cap to the test port.

(11) Install the LH electrical compartment access door (F.STA 8615).

(12) When the operational check of the system is performed on the ground during engine ground run, steps (1) through (4) and (8) through (11) are not required.

2.10 TROUBLE SHOOTING SURFACE DE-ICING SYSTEM

Trouble	Probable cause	Remedy
Boots do not inflate	a. Distributor valve does not operate.	i. Make sure that light illuminates when indicator light is pushed. (If not, electrical system is faulty.) ii. Confirm timer operation. (Replace if necessary.) iii. Inspect valve. (Replace if found faulty.)
	b. Boot line is broken and air is leaking.	Repair.
	c. There is leakage or obstructed tubing between distributor valve and boots.	Correct leakage or remove foreign material.
Boots inflate but indicator light does not illuminate	a. Faulty wiring.	Check.
	b. Faulty indicator light.	Replace light.
	c. Faulty pressure switch.	Replace switch.
	d. Air pressure is insufficient.	Check air lines for leakage.
	e. Faulty distributor.	Check air lines for leakage.
	f. Faulty regulator and relief valve.	Replace regulator and relief valve.
Boots inflate but do not deflate	a. Faulty distributor valve.	Replace valve.
	b. Faulty timer.	Replace timer.

3. WINDSHIELD ANTI-ICING SYSTEM

3.1 GENERAL DESCRIPTION

3.1.1 There are two anti-icing systems which may be applicable, electrically heated windshield and/or fluid type anti-icing systems.

3.2 HEATED WINDSHIELD ANTI-ICING SYSTEM (Aircraft S/N 661SA, 697SA and subsequent)

3.2.1 DESCRIPTION (See Fig. 9-7A)

The windshield anti-icing system employed on this airplane is an electrically heated windshield. This system is designed to perform anti-icing and de-icing functions on the forward windshield. The major components of the system consist of a three-ply windshield with an electrical heating matte and a temperature controller which controls the operating temperature of the windshield. The system is operated by a control switch located in the overhead switch console. This switch, when placed in the ON position, provides a continuous LO heat to the windshield. A switch is provided in the pilot and co-pilot control wheels for the respective windshield HI heat mode operation. This switch must be depressed and held for HI heat operation.

The heated windshield consists of an inner and outer tempered glass ply with a vinyl ply sandwiched between them. The glass plies are designed to withstand the internal cabin pressure and flight loads. The windshield is of a fail safe construction.

The electric heating matte is located between the outer glass ply and vinyl. Power is supplied to the high and low heat areas through bus bars located on the LH and RH extremes of the windshield. A sensing element controls the temperature of the heated area to approximately 104°F (40°C). Five electrical terminals are located on the lower surface for power, sensing element and static drain.

There are two temperature controllers installed below the LH floor of the forward electronic compartment. They send signals to operate relays in accordance with the windshield temperature, ON at 95 ± 5°F (35 ± 2.8°C) and OFF at 104 ± 2°F (40 ± 1.1°C). Should the windshield temperature reach 129 ± 5°F (53.9 ± 2.8°C), a signal is generated to the H/W OVER TEMP warning light in the annunciator panel which will illuminate.

3.2.2 MAINTENANCE PRACTICES

3.2.2.1 Delamination

(1) The strength of an aircraft windshield in bending or in tension is not affected by a moderate amount of delamination. Generally, the first safety consideration in cases of delamination is reduced vision and/or electrical failure. A panel would be removed for these causes before an unsafe condition due to loss of strength is reached. Delamination is generally permissible provided the vision or panel heating are not adversely affected.

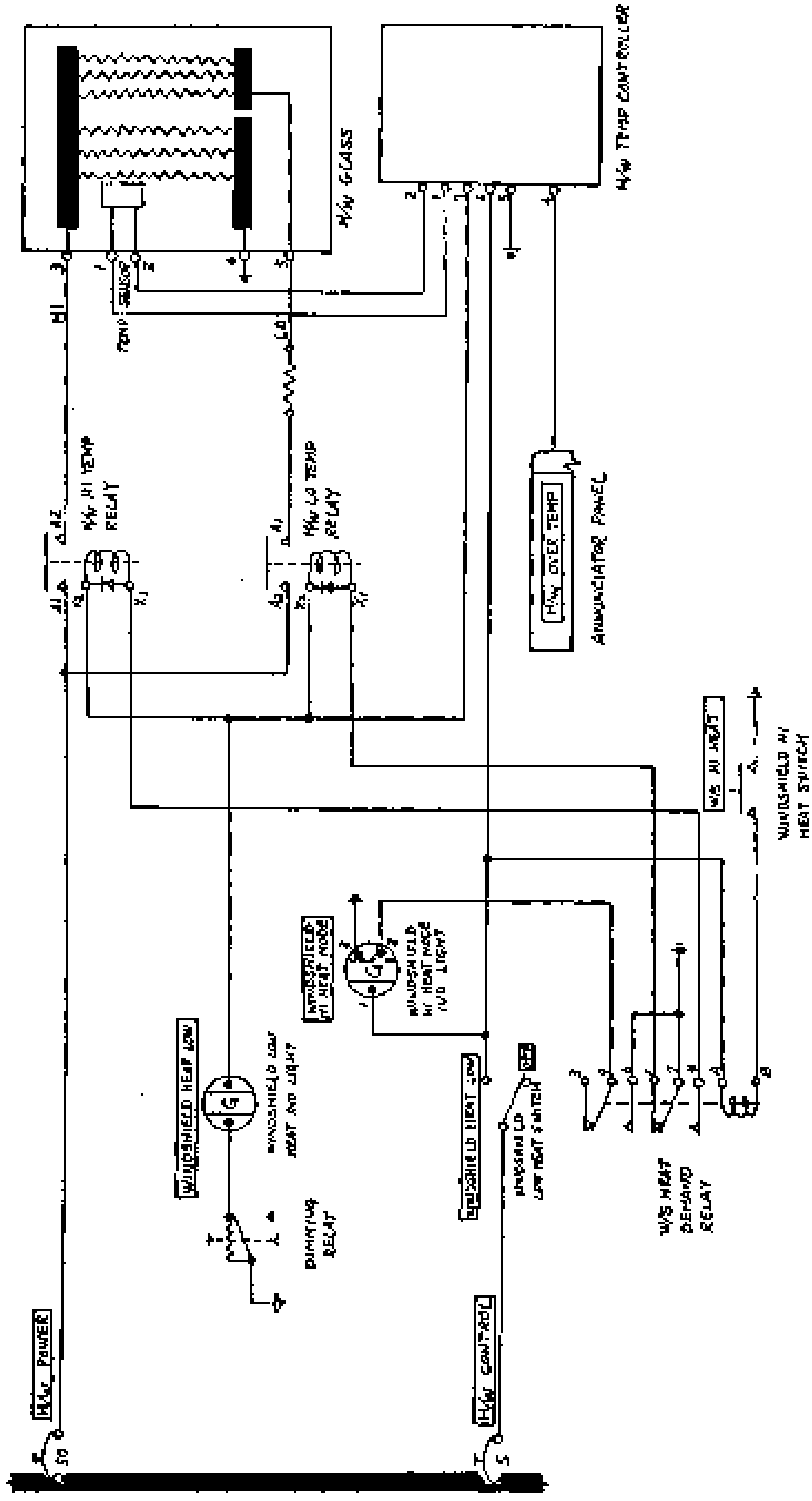


Fig 9-7A
Heated windshield
system schematic

(2) Delamination can be caused by moisture penetration into the interlayer. A large percentage of delaminated windshields result from the absence or lack of maintenance of the weather sealant around the periphery of the windshield. A cloudy or milky appearance in the delamination indicates usual progression, and, generally, the panel should be replaced. A delamination area is characterized by an irregular or jagged boundary. This separation indicates that the vinyl and glass are not uniform. Such condition can cause the vinyl to pull chips from the inner glass surface, which could lead to failure of the glass ply. In case of such delamination, a periodic inspection is recommended to determine if the damage is progressive, or if chipping of the inner glass surface is present. Replace the windshield if either condition exists.

(3) Generally, a delamination is characterized by a smooth-edge boundary and is clear (not cloudy); therefore non-progression exists as the stresses causing such a delamination are relieved when the delamination occurs.

3.2.2.2 Anti-Static Tabs

Windshields that have a conductive coating on the surface of the outboard ply of the glass also have an anti-static tab, ground contact. There are generally two types of tabs. One is a metal finger that grounds to the airframe, and the other is a fiber-glass shielded tab that is connected to the electrical grounding system of the aircraft. All fiber-glass shielded tabs should be painted with a urethane base paint to protect against wind and rain erosion and ultra-violet light exposure.

3.2.2.3 Removal/Installation

A. Removal - Windshield

(1) Disconnect wires from terminal block and identify.

(2) Remove windshield in accordance with Chapter III, para. 5.2.

(3) Install in reverse sequence of removal.

B. The temperature controller may be removed and installed as follows:

(1) Remove the forward LH access panel below the electronic compartment floor.

(2) Disconnect the wires and identify.

(3) Remove the attaching screws and the controller.

(4) Install in reverse sequence of removal.

3.2.2.4 Operational Check

The heated windshield anti-icing system may be operated when the aircraft power system is in operation. The system is operational by placing the control switch in the overhead console to ON. Normally, when the control switch is ON, the LO HEAT portion of the glass is heated. In severe icing conditions the HI HEAT portion of the glass is heated by operation of the HI HEAT switch in the control wheels. During HI HEAT operation, the LO HEAT portion becomes inoperative.

The inspection procedures are as follows:

(1) Make sure that both control switches in the overhead console are in the OFF position.

(2) Connect DC power.

(3) Place the WINDSHIELD HEAT-LH switch in the overhead console to ON and make sure that the indicator light ON illuminates and the LH windshield becomes warm gradually.

(4) Place the WINDSHIELD HEAT-RH switch in the overhead console to ON and make sure that the indicator light ON illuminates and the RH windshield becomes warm gradually.

(5) Insure the indication lights repeat the ON and OFF conditions with the switches ON.

(6) With the WINDSHIELD HEAT-LH switch ON, depress the W/S HI HEAT switch in the pilot's control wheel. Make sure the LH W/S HI HEAT MODE ON indicator light illuminates and the HI HEAT operation of the LH windshield becomes warm gradually.

(7) With the WINDSHIELD HEAT-RH switch ON, depress the W/S HI HEAT switch in the co-pilot's control wheel. Make sure the RH W/S HI HEAT MODE ON indicator light illuminates and the HI HEAT portion of the RH windshield becomes warm gradually.

(8) Insure that the W/S HI HEAT MODE ON indicator light repeats ON and OFF conditions with the W/S HI HEAT switch in the depressed condition.

NOTE

The operational check should be performed at a temperature less than 100°F (37.8°C). When the windshield temperature is more than 100°F (37.8°C) the relay is OFF at all times and operational check can not be performed.

3.2.2.5 Cleaning

A. The heated windshields have a surface coating for anti-static protection. The following procedure is recommended for windshields with the anti-static surface coatings.

(1) Remove all excessive amounts of dirt and other substances from the surface with clean water.

(2) Wash with mild soap and water of a solution of 50% isopropanol and 50% water by volume. When wiping the surface, use a clean soft cloth or sponge with a straight rubbing motion.

CAUTION

Do not use any abrasive materials, strong acids or bases.

(3) Rinse thoroughly and dry.

NOTE

Do not apply wax to the windshields.

(4) It is recommended to clean the wiper blades as often as possible to help prevent scratches on the glass.

CAUTION

Do not attempt to polish out nicks or scratches in the glass surface. If scratch removal is done improperly, the results may be distortion from irregular glass removal.

3.2.2.6 Repair

A. Terminal Block

(1) Terminal blocks usually do not need any maintenance performed on them.

The following procedure should be followed for repair.
(a) Clean the base of the terminal and the glass surface with methyl ethyl keytone (MEK).
(b) Apply a thin coat of PR-1221-B1/2 to the base of the terminal.
(c) Place the terminal on the glass in the proper place and secure it to prevent movement and to maintain good contact with the glass.

NOTE

Use masking tape to hold block in place.

(d) Remove excess PR-1221-B1/2 from around edges of terminal block.

NOTE

Allow the PR-1221-B1/2 sealant to cure for 24 hours before removing the tape.

E. Anti-Static Tabs

(1) Mask all exposed glass surfaces.

(2) Remove tab from over bonding position.
 (a) For metal tab, remove the screw that fastens it to the airframe and lay the tab to one side.
 (b) For fiber-glass tab, the tab cannot be completely removed since it is tied into the unit by a metal braid.

NOTE

Be very careful not to tear the braid connecting the tab to the grounding system; just place the braid away from the bonding area.

(3) Bonding Surface Preparation
 (a) Silicone under tab finger.
  (1) Carefully and thoroughly remove and clean all the conductive silicone that remains on the glass.

NOTE

Be extremely careful not to scratch the glass surface.

 (b) Epoxy based under tab finger.
  (1) Carefully roughen the surface of the electrically conductive epoxy which remains on the glass surface.
  (2) Clean the tab and rough the surface.

(4) Apply conductive silicone (Technik RTV silicone, P/N 72-00002) to the tab and also to the glass surface.

(5) Apply the tab to the windshield(s) and if the tab is of the metal finger type, install the screws.

(6) Remove windshield maskings.

3.3 FLUID ANTI-ICING SYSTEM (Aircraft S/N 652SA and Aircraft equipped with system)

3.3.1 DESCRIPTION

The fluid anti-icing system consists of a pressure control valve which controls air pressure from the engine bleed air system which forces the fluid from the tank to the spray nozzles. The anti-icing fluid is a mixture of 60% ethylene glycol (MIL-A-8243, Amend 1, MIL-E-9500, MIL-E-5559 or O-A-00548) and 40% water. The fluid flow rate is 1.7 U.S. pints (0.804 liters)/minute to both the left and right windshield surfaces. The tank capacity is approximately 2.5 U.S. gallons (9.46 liters).

When the windshield anti-icer switch is placed ON in icing conditions, the solenoid valve in the pressure control valve opens, allowing engine compressor bleed air to flow. The air, controlled to specified pressure, pressurizes the fluid tank. Pressurized anti-icer fluid is sprayed from the nozzle to windshield surface and then dispersed by free air stream flow. The wiper is used as required. The fluid tank is located on the fuselage on the right side of the nose landing gear bay. The anti-icer fluid filler port is located in the electronic compartment above the tank (Figure 9-8).

NOTE

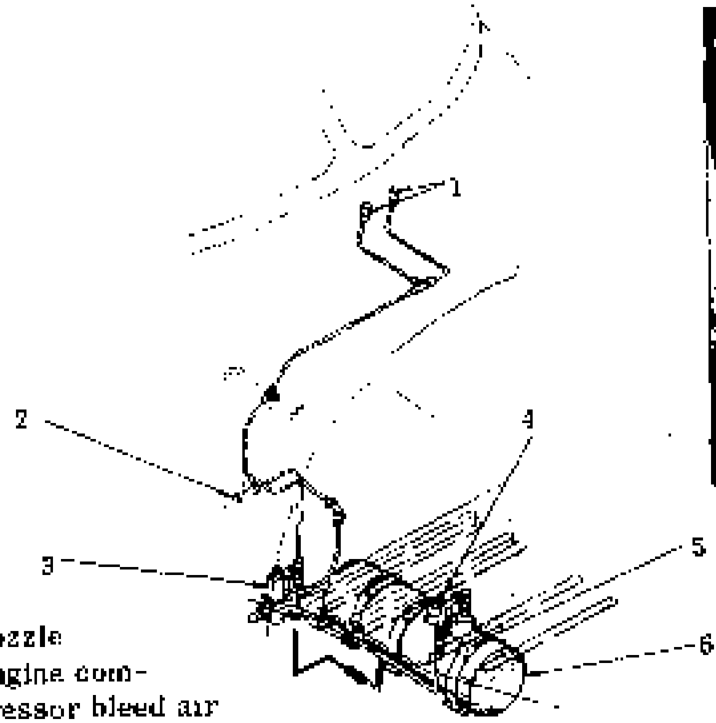
i. Do not use the anti-icer fluids (such as isopropyl alcohol, ethyl alcohol, etc.) other than ethylene glycol.

ii. Flush the windshield with clean water after using anti-icer fluid.

3.3.2 PRESSURE CONTROL VALVE

The pressure control valve, installed on the RH forward section of the pressure bulkhead in the electronic compartment, is installed between pressurizing air source and anti-icer fluid tank. This valve not only controls pressure of engine compressor bleed air to 8 psi (0.6 kg/cm^2), but also actuates the pressure relief mechanism when pressure on the downstream side of the valve reaches 12 psi (0.8 kg/cm^2), preventing pressure from rising above the limit.

When the anti-icer system is not in use (i.e., solenoid valve in pressure control valve is closed and pressurized air is not flowing), the discharge port opens to the atmosphere, preventing differential pressure between inside and outside of the tank. A check valve is installed in the pressure control valve to avoid reverse flow of anti-icer fluid.



Control assembly

1. Nozzle
2. Engine compressor bleed air
3. Control assembly
4. Filler cap
5. Clamp
6. Anti-icer fluid tank

Fig. 9-8 Windshield anti-icer equipment

3.3.2.1 REMOVAL AND INSTALLATION

(1) Remove RH electronic compartment access door.

(2) Remove access panel on RH STA. 1080.

(3) Disconnect wiring connectors.

(4) Disconnect tubing on upstream and downstream sides of valve.

(5) Remove bolts and washers attaching valve, and remove the valve.

(6) Remove union (1 ea.) and elbow (1 ea.) from valve and plug each port.

(7) Install in reverse sequence of removal.

3.3.3 ANTI-ICER FLUID TANK

The anti-icer fluid tank is installed in the fuselage, RH section of nose landing gear wheel well, and connected with pressure line to supply air from control assembly to tank, and anti-icer fluid line to lead fluid to nozzles.

A float type check valve is provided at the pressure line connecting port and prevents reverse flow of anti-icer fluid in pressure line and leakage to relief port of control assembly while system is not in operation.

3.3.3.1 REMOVAL AND INSTALLATION

(1) Remove RH electronic compartment access door.

(2) Open access door at F.STA 350-930.

(3) Remove pressure line and anti-icer fluid line from tank.

(4) Remove tank attaching bolts and remove tank.

(5) Plug each port.

(6) Remove clamps from tank.

(7) Install in reverse sequence of removal.

3.3.4 TIMER

The timer is installed on the overhead console and is operated by placing the windshield anti-ice switch to "WINDSHIELD ANTI-ICE". The timer actuates the control assembly periodically by sending electric current. "ON" time is about 4 seconds (constant) and "OFF" time can be set to optional value between 0.5 thru 8 seconds by means of anti-ice cycle selector.

3.3.5 ANTI-ICE CYCLE SELECTOR

The anti-ice cycle selector changes operating cycle of the timer. "OFF" time of the timer is about 0.5 seconds in "FAST" position and about 8 seconds in "SLOW" position. Setting should be made according to icing condition.

3.3.6 OPERATIONAL CHECK (Fig. 9-9)

(1) Confirm that the WINDSHIELD switch in the overhead console is in the OFF position and circuit breaker AIR COND is open.

(2) Connect DC power (28 VDC battery or external power).

(3) Pour 0.5 to 0.8 gal. (2 to 3 L) of water into the fluid tank.

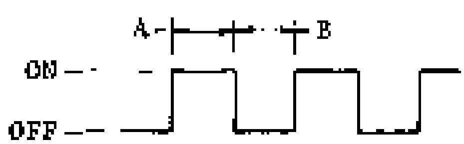
(4) Remove the LH electrical compartment access door (F.STA 8615).

(5) Locate and remove cap plug from ground test port; connect external air supply source (capacity: 30 to 50 psi (2.1 to 3.5 kg/cm^2) with regulator and gauge).

(6) Gradually pressurize and stabilize the system to 30 psi (2.1 kg/cm^2).

(7) Turn Battery Key switch to ON, close circuit breaker, place the switch to the AUTO position and confirm the fluid is sprayed from the nozzles to the windshield surfaces to the specified area as shown in Fig. 9-9.

(8) Confirm cycle operation of spraying for 4 seconds and stopped for 0.5 seconds in the FAST position, and spraying for 4 seconds and stopped for 8 seconds in the SLOW position.



A - 4 seconds
B - 0.5 seconds } FAST

A - 4 seconds
B - 8 seconds } SLOW

(9) Place switch to OFF position and confirm spraying is stopped.

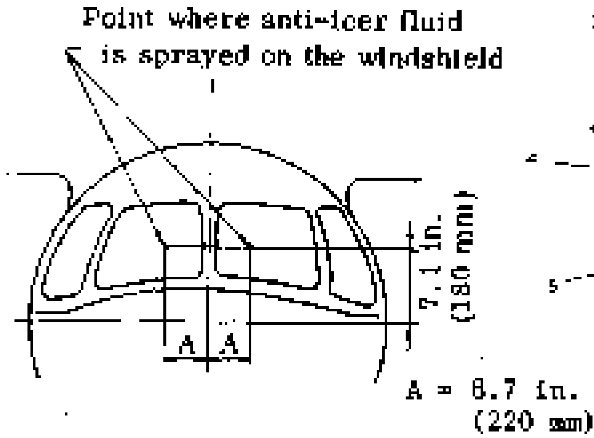
(10) Place the switch to MANUAL position and confirm a continuous spray from the nozzles; place to OFF when the fluid tank is empty.

(11) Turn the Battery Key switch to OFF.

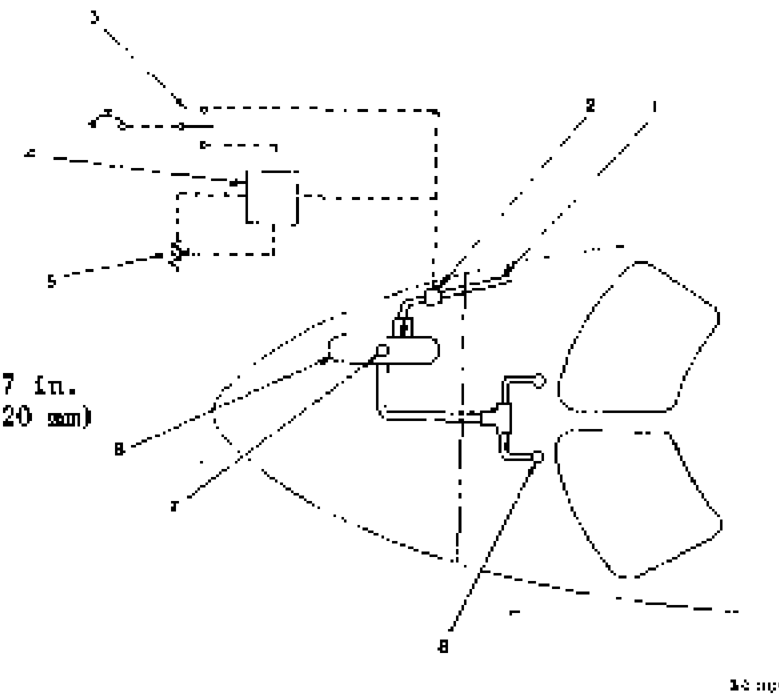
(12) Close air supply source, release the air pressure from the system and disconnect the air supply source from the ground test port. Install the cap plug.

(13) Install the LH electrical compartment access door.

(14) When the operational check of this sytem is performed on the ground during engine ground run, steps (2), (4) through (6) and (11) through (13) are not required.



All dimensions are measured on the glass surface.



1. Air supply line
2. Control assembly
3. Anti-ice control switch
4. Timer
5. Cycle selector
6. Anti-icer fluid tank
7. Filler cap
8. Spray nozzle

Fig. 9-9 Windshield anti-icer system

3.3.7 TROUBLE SHOOTING WINDSHIELD ANTI-ICING SYSTEM

Trouble	Probable cause	Remedy
Anti-icer fluid does not spray	a. Anti-icer fluid tank is empty.	Fill tank with anti-icer.
	b. Control assembly does not operate.	i. Check electrical circuit. ii. Replace valve if it is defective.
	c. There is leak between control assembly and nozzles.	Repair component which leaks.
	e. Nozzle is blocked.	Tighten cap.
	d. Tank filler cap is not closed tightly.	Remove foreign material in nozzle hole with wire.
	f. Timer does not work.	i. Check electrical circuit. ii. Replace timer if it is defective.
Anti-icer fluid spray can not be stopped	a. Control assembly faulty.	Replace valve.
	b. Timer faulty.	Replace.
Anti-icer fluid spraying cycle varies and anti-ice cycle selector does not change cycle	Timer faulty.	Replace.

4. WIPER

4.1 GENERAL DESCRIPTION

Two wipers are installed on the front windshields and operated by an electrical motor.

Wiper blades can be easily removed by depressing the pin attaching blades and driving arms. Wiper blades should be replaced as necessary. Take sufficient care that use of wiper blades for a long period may not cause damage to windshield due to blade deterioration.

CAUTION

Do not operate wipers above 130 Kts (*1), 175 Kts (*2).

NOTE

Operating the wipers on a windshield covered with dust and sand will cause damage to the glass surface. During preflight or operation on the ground, lift the wiper blades off the glass surface and remove dust, sand or dirt from blade section. Apply dust repellent (CW-100 Marquette Division, Curtiss-Wright Corp.) on the glass surface with a soft cloth.

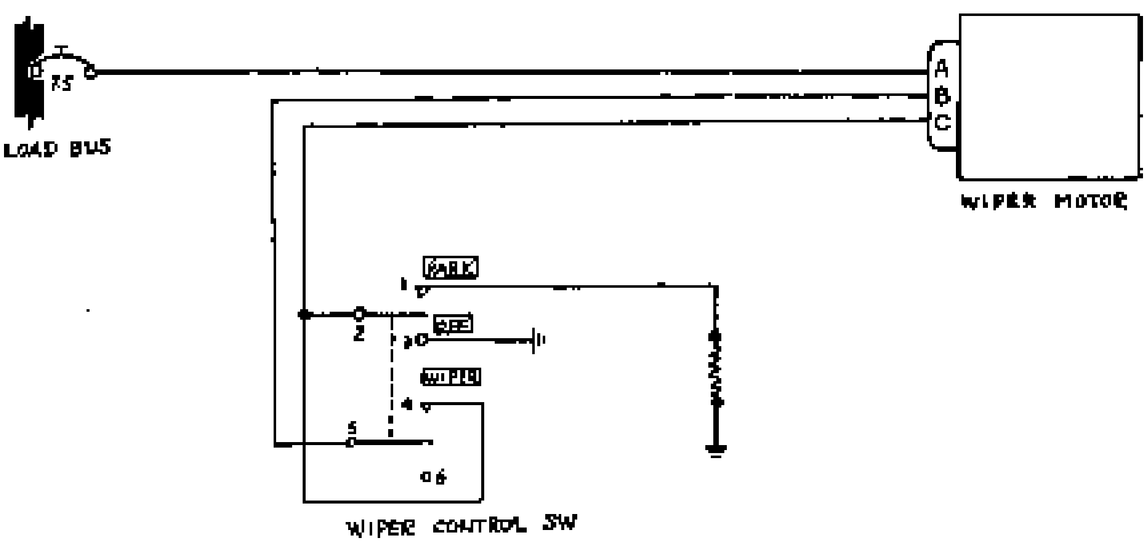


Fig 9-10A Wiper control system schematic

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

4.2 REMOVAL AND INSTALLATION OF WIPER MOTOR (See Fig. 9-10)

(1) Remove access panel above RH landing light.

(2) Remove electrical wiring leading to motor.

(3) Detach flexible shaft leading to converter from coupling.

(4) Remove motor and fitting from aircraft by removing 2 bolts attaching motor.

(5) Install in reverse sequence of removal.

Fig. 9-10 Installation of Wiper motor

4.3 REMOVAL AND INSTALLATION OF CONVERTER (See Fig. 9-11)

(1) Disconnect flexible shafts from each converter located at upper back (F.STA 1275) of instrument panel.

(2) Remove wiper blade and drive arm from converter shaft.

(3) Remove converter from aircraft by removing bolts attaching each converter.

(4) Install in reverse sequence of removal.

1. Motor
2. Converter
3. Flexible shaft
4. Wiper

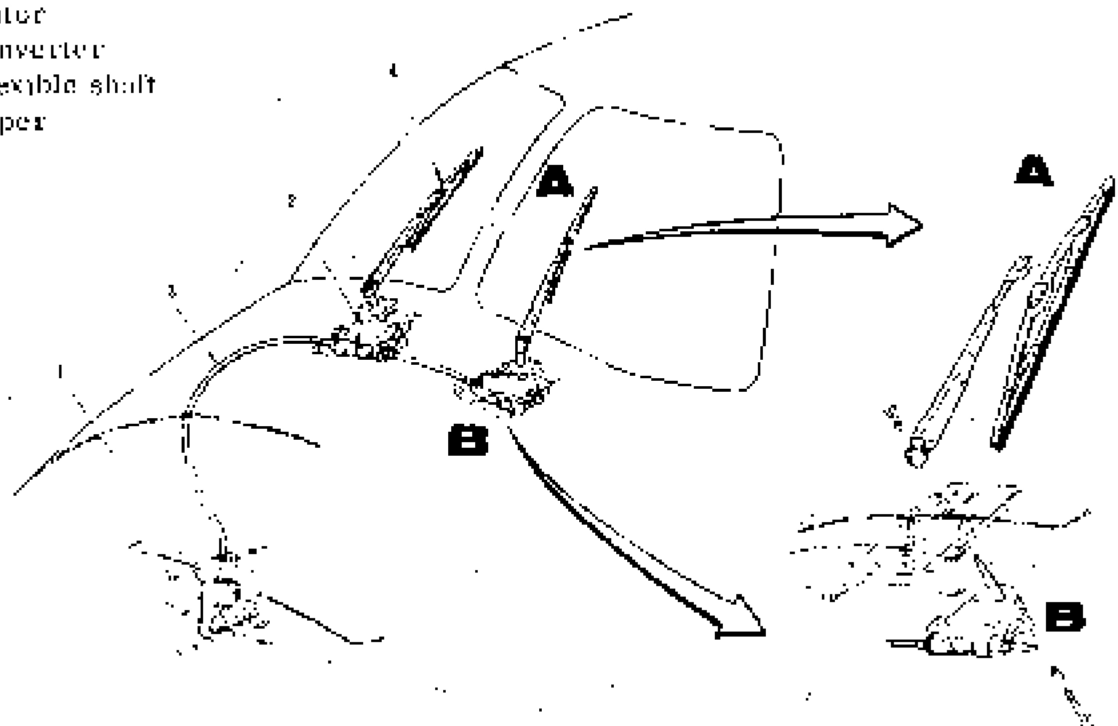


Fig. 9-11 Installation of motor and converter (1/2)
(Typical S/N 652SA and 661SA)

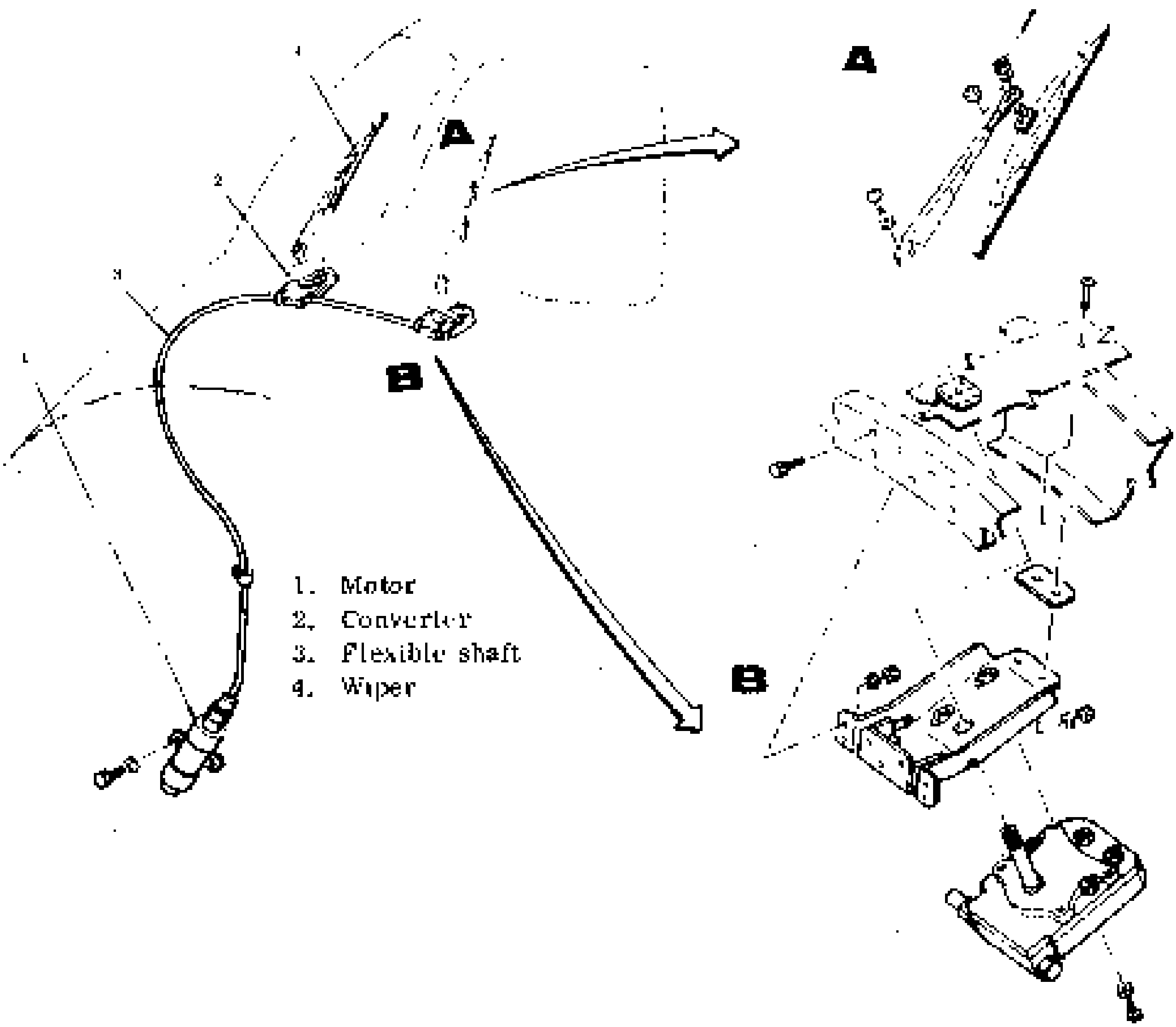


Fig. 9-11 Installation of motor and converter (2/2)
(Typical S/N 697SA and subsequent)

4.4 REMOVAL AND INSTALLATION OF WIPER BLADE

(1) Aircraft S/N 652SA and 661SA

Holding up drive arm by pin that connects the wiper blade with the drive arm, remove the drive arm.

Aircraft S/N 697SA and subsequent

Lift the drive arm perpendicular to the converter shaft, insert MS24665-372 cotter pin into the arm fixing pin hole at the root of the arm, loosen the blade fixing nut and remove blade.

(2) Remove the drive arm by removing the nut from the converter shaft and moving the drive arm upward.

(3) Install in reverse sequence of removal.

NOTE

Tighten nut (safety wire, if required) after confirming proper engagement of serration. If the drive arm is connected to the converter shaft by tightening the nut with excessive force, serration on drive arm may be damaged.

4.5 ASSEMBLY AND ADJUSTMENT OF WIPER SYSTEM

4.5.1 ASSEMBLY

(1) Aircraft S/N 652SA and 661SA

Place the flexible shaft between LH and RH converters. Before placing the shaft in position, ensure that the gears in the LH and RH converters, which are normally covered by the converter gear case, are positioned symmetrically (Fig. 9-11). The position of the gears is easily changed by rotating the converter shaft. It may be possible to recognize symmetrical arrangement of the LH and RH gears by recognizing the angle of rotation of the converter shaft.

Aircraft S/N 697SA and subsequent

Connect the flexible shaft between LH and RH converters. Before connecting the shaft, ensure the gears are readily changed by rotation of the converter shaft.

(2) Connect the flexible shaft between the converters and the motor.

4.5.2 ADJUSTMENT OF SYMMETRICAL MOTION OF LH AND RH BLADES

(1) Aircraft S/N 697SA and subsequent

Lift the blade off the glass surface and insert a 1/8 inch dia. cotter pin into the side hole of the arm.

(2) Turn wiper switch ON. When LH wiper blade travels to innermost position, turn switch to OFF. If RH blade is not symmetrical with LH blade, adjust as follows:

(a) Remove flexible shaft from LH converter.

(b) Turn wiper switch ON, actuate RH blade. When blade travels to the innermost position, turn switch OFF. The position of the RH blade must coincide with that of the LH blade, where both blades cease to travel at the innermost position.

(c) Connect flexible shaft. Turn wiper switch ON and ensure symmetrical movement of LH and RH wiper blades.

(3) Aircraft S/N 652SA and 661SA

Position of converter gears can be set symmetrically by Para. (2) above. Adjustment to set arm symmetry is made by removing and turning adjusting sleeve (adjusting sleeve has 50 serrations inside and 51 outside; minute adjustment of 0.03 in. (0.8 mm) at the arm tip is available). In this case, remove the bolt at the converter shaft end and loosen allen screw. Adjust and return hardware to original configuration.

(4) Aircraft S/N 652SA and 661SA

Remove cotter pin and allow blade to return to position on the glass surface.

4.5.3 ADJUSTMENT OF PARKING POSITION

Applicable to aircraft S/N 652SA and 661SA

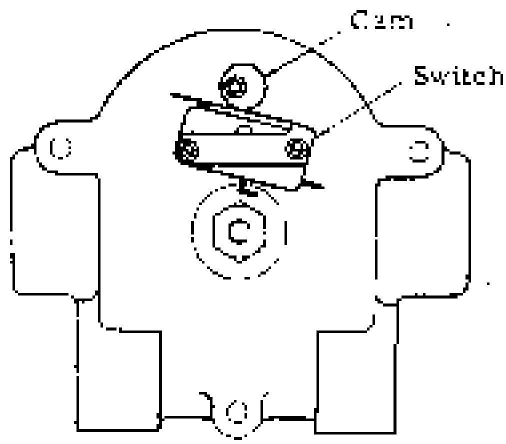
(1) Adjust the parking limit switch in accordance with the following procedures.

(a) Turn wiper switch ON; when the blade travels to its outer position, place the switch to PARK.

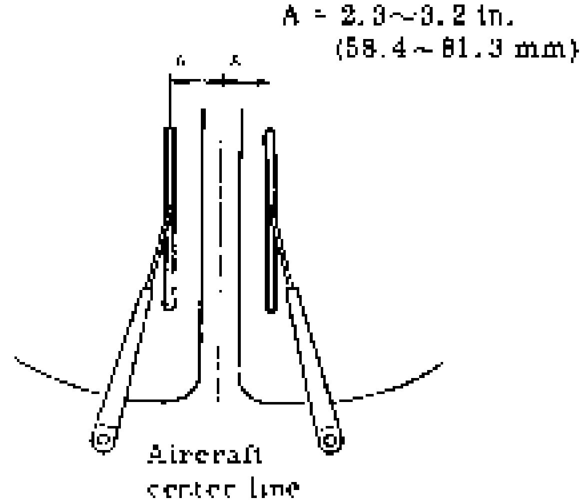
NOTE

Wiper should not be operated on a dry windshield.

(b) Adjust the cam position in the converter as shown below, such that the blade may travel from outer position to inner position and return to the parking position automatically.



Parking position is adjusted by cam position of converter



Wiper blade parking position

Applicable to aircraft S/N 697SA and subsequent

(1) Turn wiper switch ON; when the blade travels to its outer position, place the switch to PARK.

NOTE

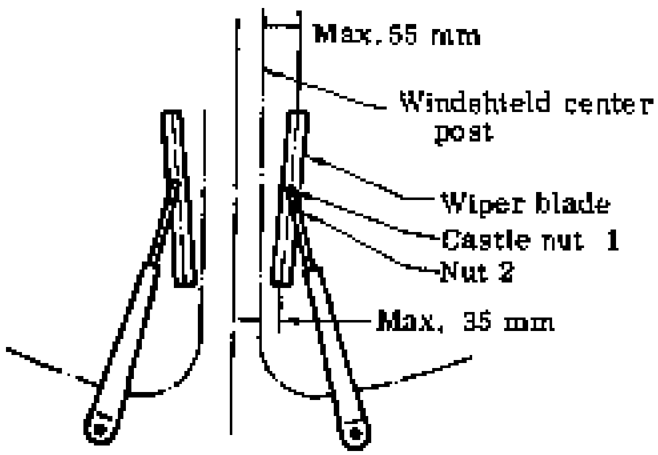
Wiper should not be operated on a dry windshield.

(2) If the wiper stops at a point other than the parking position, remove the flexible shaft at the motor and operate the converters by hand until the wipers are in the parking position (refer to Para. 4.5.2).

(3) Reinstall the flexible shaft to the motor and check for the parking position.

(4) Apply lock wire on flexible shaft nut and bolt to which converter and arm are attached.

(5) Adjustment of blade angle against arm is made by loosening castle nut 1 and nut 2. After adjustment, tighten the nuts and install cotter pin.



NOTE

Blade should not be on retainer and/or retainer seal of center post.

4.5.4 ADJUSTMENT OF WIPER BLADE CONTACT PRESSURE

Check the wiper blade contact pressure in accordance with the following procedures:

(1) Set the wiper blades to the position shown in Fig. 9-12.

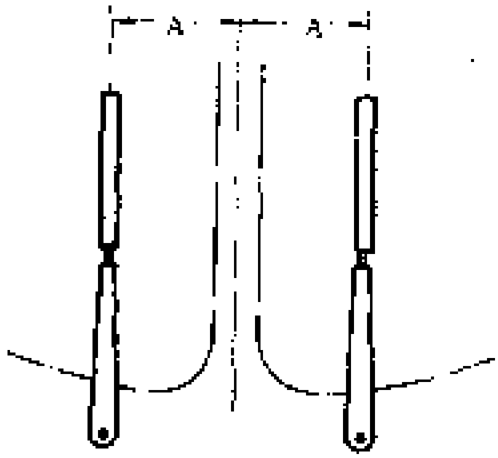
(2) Pull the blades perpendicular to the glass surface at the position where the driving arms are most extended.

(3) Measure the tension force when the center of the blades comes off the glass surface.

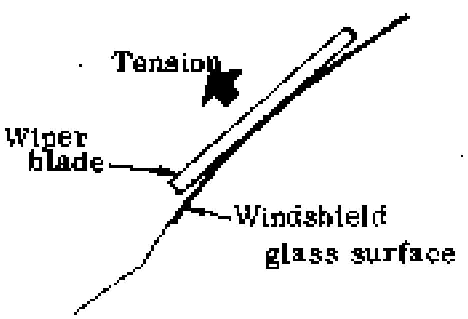
If the wiper blade contact pressure is not within the specific value, adjust in accordance with the following procedures:

(1) *1 Adjustment of blade contact pressure is made by adjusting the length of the spacer.

*1 Aircraft S/N 652SA and 661SA



A = 8.47 ± 0.4 in. (215 ± 10 mm) *1
A = 8.86 ± 0.4 in. (225 ± 10 mm) *2



Specified Value

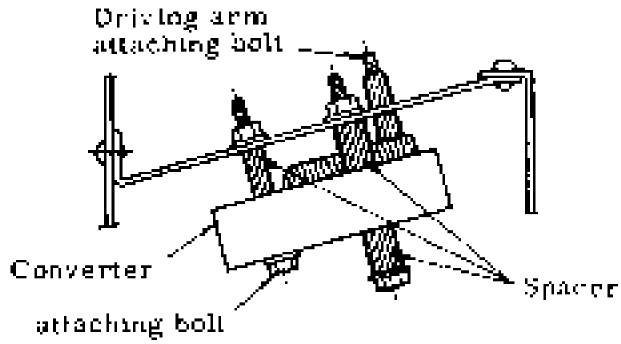
1.7 to 2.7 lbs. (0.8 to 1.2 kgs) *1
3.5 to 7.0 lbs. (1.6 to 3.2 kgs) *2

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

Fig. 9-12 Blade Contact Pressure

NOTE

The converter shaft angle should be adjusted such that it remains parallel with the airplane axis. Do not lean the shaft to the left or right.

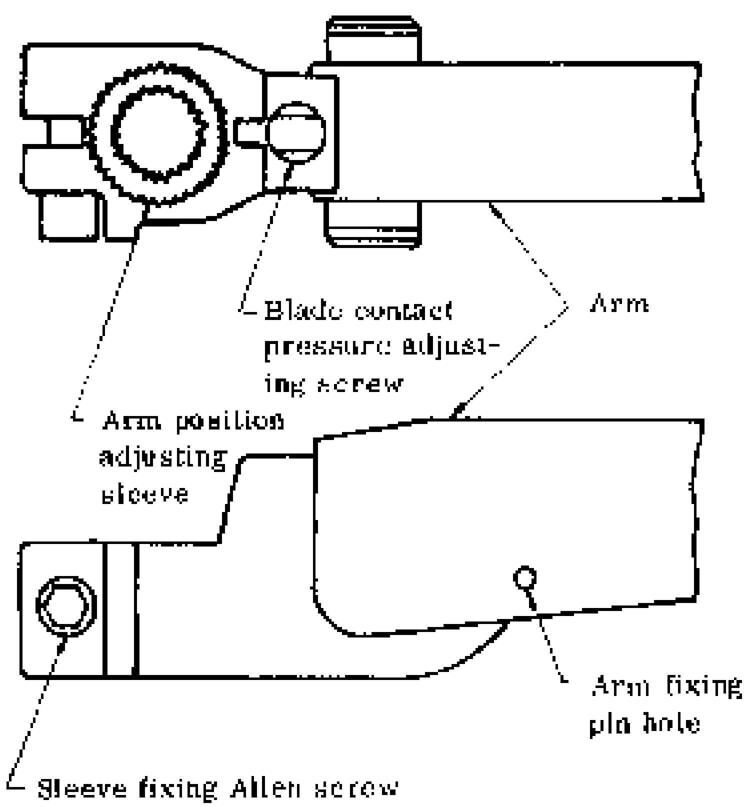


*2 Adjustment of blade contact pressure is made by turning adjusting screw of the drive arm with WX20509 wrench.

NOTE

To increase the contact pressure, turn clockwise, and to decrease contact pressure, turn counterclockwise.

(2) *2 After adjustment, operate for 30 seconds and recheck the contact pressure.



*2 Aircraft S/N 697SA and subsequent

4.6 TROUBLE SHOOTING

Trouble	Probable cause	Remedy
Striped or spot patterns remain on wiper area	a. Blade is broken. b. Loose connection of driving arm with blade.	Replace. *1 Check spring clip attachment and tighten. *2 Check connection and tighten.
Wipers do not move or move irregularly	a. *1 Flexible drive shaft fittings do not fit firmly into groove of motor or converter.	*1 Disconnect the flexible drive shaft and adjust the position of flexible drive shaft.
	*2 Coupling does not engage securely with end fitting of flexible drive shaft or square shafts of motor and converter.	*2 Disconnect and check flexible shaft, coupling motor and converter shaft. Replace if weariness is found. When drive shaft is pushed one side and engagement of drive shaft and coupling is insufficient, add a steel ball of 1/8 in. in dia. each to both sides of drive shaft.
	b. Flexible shaft is broken.	Replace.
	c. Defective motor.	Replace.
	d. Defective converter.	Insert drive shaft in the groove of converter. If irregular turning is detected, replace the converter.
*1 Trace of wiper blade is irregular	*1 Tie rod is loose.	*1 Fasten tightly.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

5. DEFOGGING SYSTEM

5.1 GENERAL DESCRIPTION

Windows consist of front windshields, side windshields and cabin windows. Windshields and cabin windows are defogged by conditioned air from the defogging air outlet in the lower section of windshield and cabin windows.

A thermostat switch is installed on the front windshield defogging air outlet. When defogging air temperature reaches 200 ± 5°F (93.3 ± 2.8°C) due to air conditioning system failure, the switch actuates and illuminates DEFOG OVER TEMP warning light to indicate abnormal condition of defogging air.

5.2 WINDSHIELD DEFOGGING

The windshield defogging system consists of air conditioning-defogging selector valve incorporated in the forward conditioned air outlet, defogging air outlets, tubing and main outlet flow control valve for defogging capacity augmentation in extremely cold weather.

5.2.1 DEFOGGING SELECTOR VALVE (See Fig. 9-13)

This selector valve is a butterfly type and can be set for COND to supply conditioned air to cabin and DEFOG for windshield defogging. For windshield defogging, turn the knob in the outlet towards the arrow direction (clockwise). Normally, the selector valve is set to COND position.

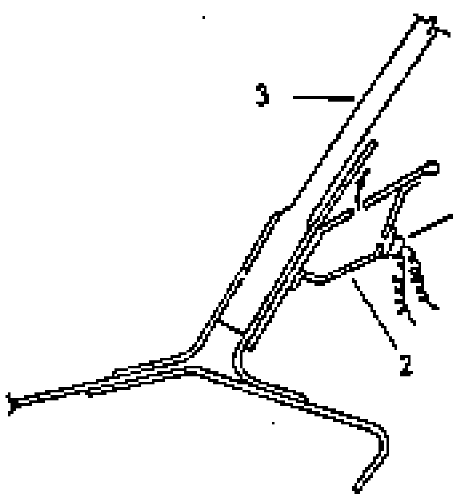
5.2.2 WINDSHIELD DEFOGGING AIR TEMPERATURE WARNING SYSTEM (See Fig. 9-14)

The windshield defogging air temperature warning system consists of thermostat switch and warning light. The thermostat switches are installed on LH and RH front windshield defogging air outlets. When defogging air temperature reaches 200 ± 5°F (93.3 ± 2.8°C), the switch panel actuates and illuminates DEFOG OVER TEMP warning light. When defogging air temperature drops below 180 ± 10°F (82.2 ± 5.6°C) the switch opens and warning light goes out.

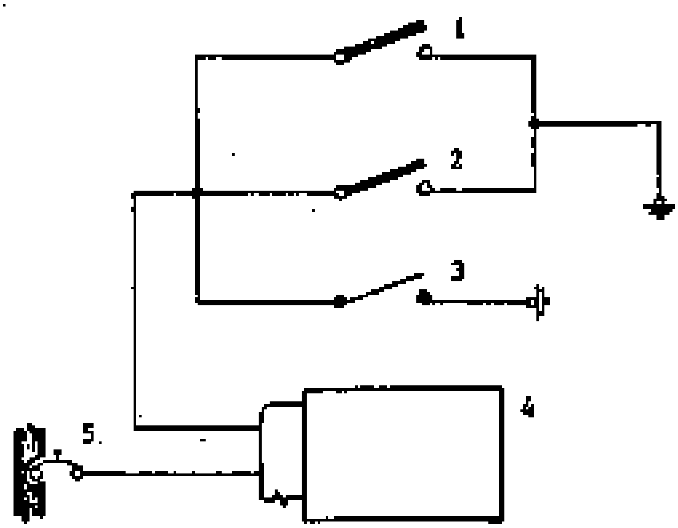
5.3 CABIN WINDOW DEFOGGING

Cabin window defogging is performed by feeding conditioned air between double window glasses. The conditioned air blows off constantly while the air conditioning system is in operation.

Fig. 9-13 Defogging selector valve



1. Thermostat Switch
2. Defogging Air Outlet
3. Windshield Glass



1. LH Thermostat Switch
2. RH Thermostat Switch
3. Test Switch
4. Warning Light
5. Master Caution

Fig. 9-14 Defogging air temp. warning system

5.4 TROUBLE SHOOTING OF DEFOGGING SYSTEM

Trouble	Probable cause	Remedy
Air does not come from defogging air outlet or air flow is insufficient	a. Front outlet knob is not set to DEFOG.	Set to DEFOG.
	b. Valve in outlet is faulty.	Repair valve.
	c. Defogging air outlet is blocked.	Remove foreign material.
Cabin window becomes foggy	a. Tubing leaks.	Repair.
	b. Tubing is blocked.	Remove foreign material.
Defog OVER TEMP light illuminates	a. Defogging air exceeds 200 ± 5°F duct air conditioning system failure.	Check air conditioning system and replace faulty components.
	b. Operation of thermostat switch.	Replace thermostat switch.

6. ENGINE AIR INTAKE ANTI-ICING SYSTEM (See Fig. 9-15 and 9-15A)

6.1 GENERAL DESCRIPTION

Engine air intake, cowling lip, air intake duct, pressure and temperature sensors are provided with anti-icer devices. The cowling lip has a double outer skin through which bleed air from the anti-icing bleed air valve passes. The forward lower section of the engine air intake duct has a double skin, and in the area of the first stage compressor, the back side of this surface skin forms a large cavity. The lower outer skin of this double construction is glass fiber. Bleed air through the anti-icing bleed air valve is fed into the forward double skin section and discharged outboard. Bleed air passed through the anti-icing bleed air valve is also fed to the pressure and temperature sensors located on the RH wall in the air intake duct.

To test the system, place the EH (RH) ENG INTAKE HEAT switch, located in the overhead console, to the ON position; the indicator light illuminates. A yellow (amber) indicator light, located adjacent to the respective switches in the overhead console, is employed to indicate the "open" position of the anti-icing bleed air valve. When the engine is not running, the switch may be placed to ON and the indicator light will illuminate, but no anti-icing function will be performed.

6.2 ANTI-ICING BLEED AIR VALVE (See Fig. 9-16)

The engine anti-icing bleed air valve is installed at ZONE I of the engine which bleeds air from the second stage compressor for engine anti-icing. It is a solenoid type valve spring loaded to close when electric power is cut off. The micro switch for the anti-icing indicating light is installed in the valve head section. Current drain of the valve is approximately 1A/28V.

6.2.1 REMOVAL AND INSTALLATION

(1) Disconnect the wiring connector at valve head.

(2) Disconnect the inlet and outlet tubing at valve, and cover the tube openings.

(3) Remove bolts and washers attaching valve, and remove the valve.

(4) Install in reverse sequence of removal.

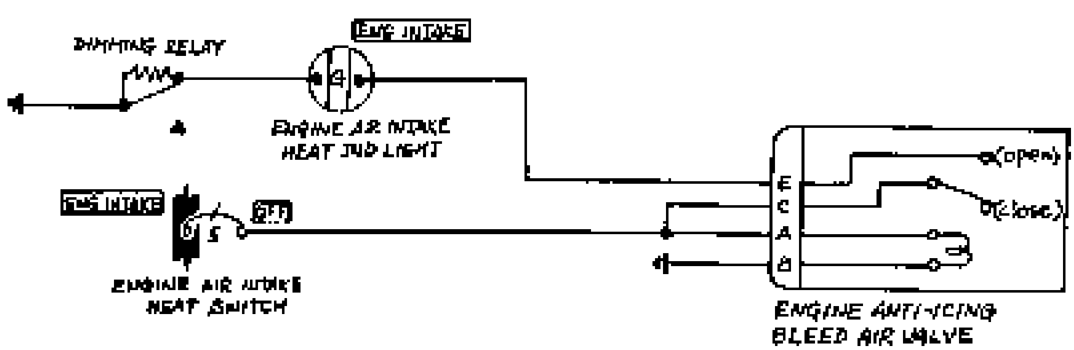
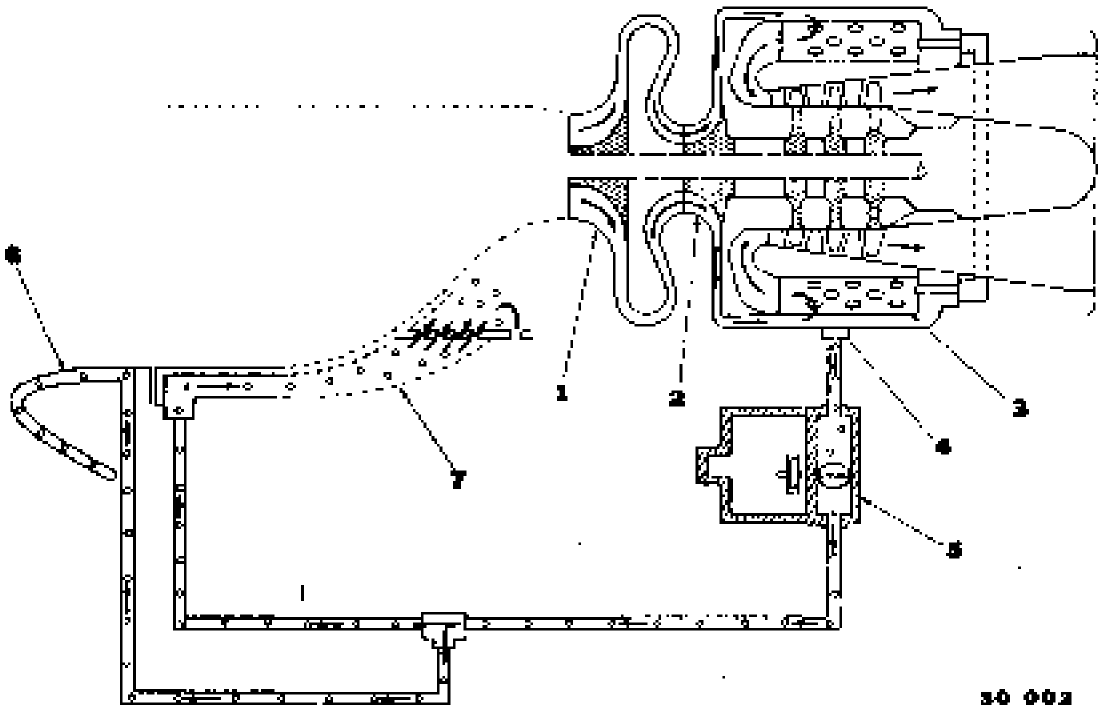


Fig 9-15A Engine anti-icing system schematic



1. First State Compressor/Impeller
2. Second Stage Compressor/Impeller
3. Combustion Chamber
4. Bleed Air Port
5. Anti-Icing Bleed Air Valve
6. Cowling Lip
7. Air Intake Duct

Fig. 9-15 Engine anti-icing

ORIGINAL
As Received By
ATP

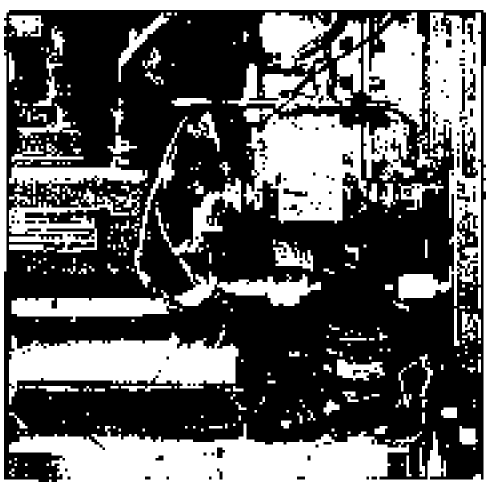


Fig. 9-16 Anti-icing bleed solenoid valve

6.2.2 OPERATIONAL CHECK

Perform operational check of the left and right engine anti-ice in accordance with the following procedures.

(1) Connect DC power (battery or external power), turn Battery Key switch ON.

(2) Place L (R) ENG INTAKE switch to the ON position.

(3) When the anti-ice bleed air valve is "open", operating sound is heard; the indicator light will illuminate.

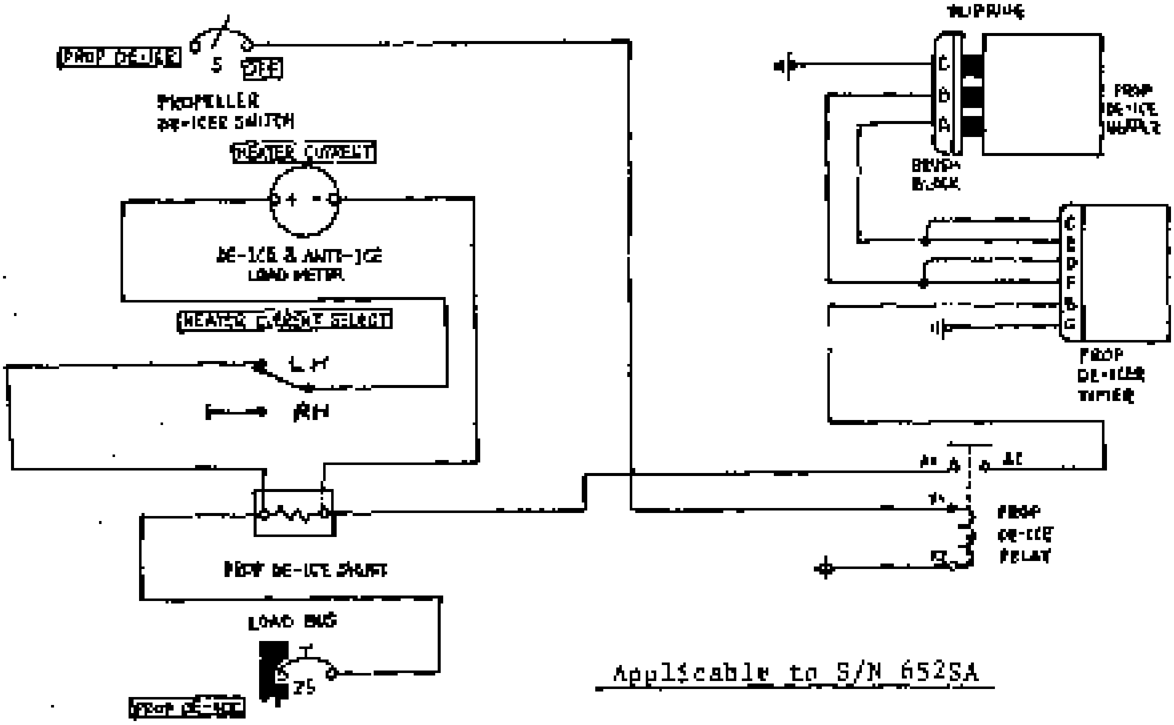
(4) Place the switch to OFF.

(5) When the anti-ice bleed air valve is "closed", operating sound is heard; the indicator light will go out.

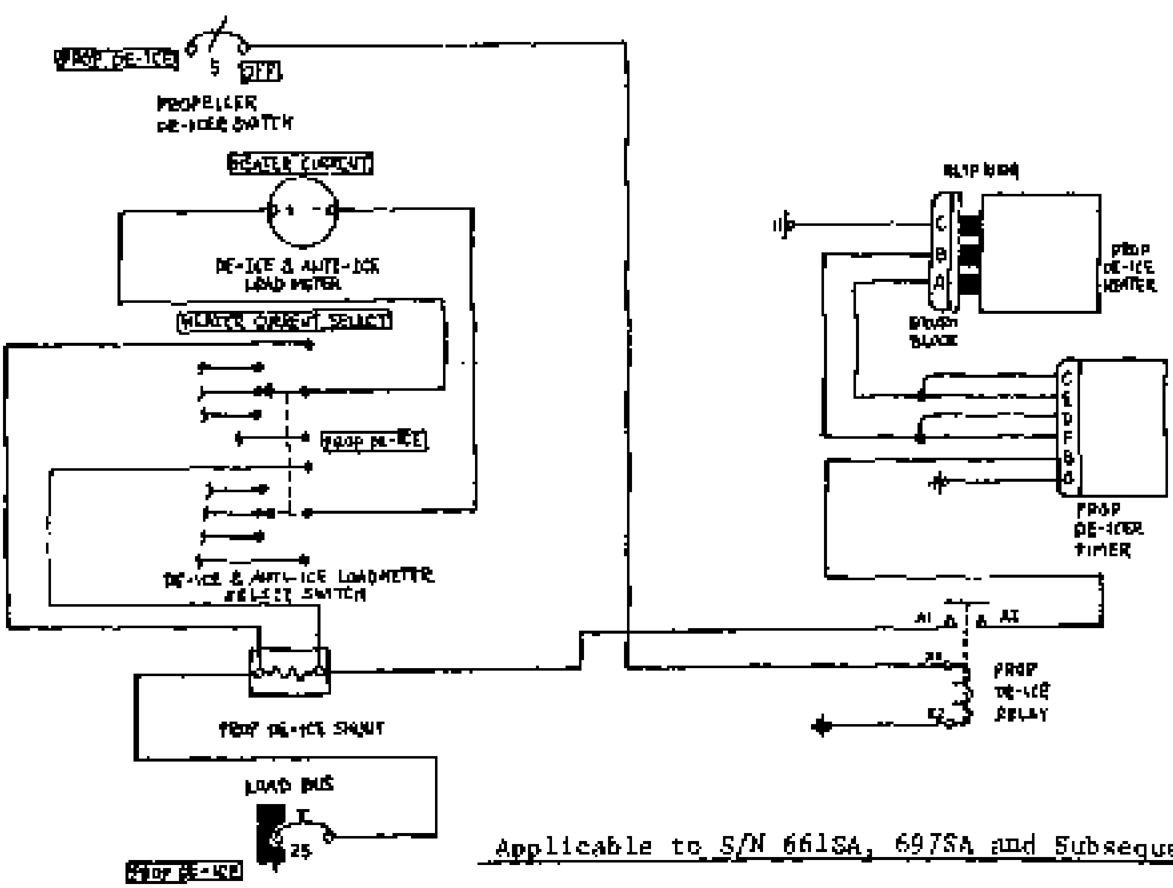
(6) Turn Battery Key to OFF; disconnect external power (if used).

NOTE

If this operational check is performed during engine ground run, a decrease in engine rpm (side being checked) and an increase in fuel flow occur during indicator illumination.



Applicable to S/N 652SA



Applicable to S/N 661SA, 697SA and Subsequent

Fig 9-16A Propeller de-icing system schematic

7. PROPELLER DE-ICING SYSTEM

7.1 GENERAL DESCRIPTION

Applicable to aircraft S/N 652SA

Electrically heated de-icer strips containing two heating elements, inner and outer, are bonded to the propeller blades for de-icing. Electrical power is supplied through brushes and slip rings. When the PROP DE-ICE switch, located in the overhead console, is ON, current through the timer is sequentially delivered to the inner and outer heating element of the LH or RH propeller. The heating sequence, RH outer, RH inner, LH outer, LH inner, completes the cycle in approximately 136 seconds (34 seconds per each heating element). A loadmeter in the overhead console deflects between 0.93 to 1.20 to indicate system operation.

A manual back-up supplements the main system, should a malfunction occur. The STAND BY PROP DE-ICE switch (three position, center position OFF) turned to the OUTER position for 30 to 35 seconds, then to the INNER for 30 to 35 seconds will properly operate the prop de-ice system manually.

NOTE

During STANDBY (manual) operation, the loadmeter will not function.

Applicable to aircraft S/N 661SA, 697SA and subsequent

Electrically heated de-icer strips containing two heating elements, inner and outer, are bonded to the propeller blades for de-icing. Electrical power is supplied through brushes and slip rings. When the LH and RH PROP DE-ICE switch, located in the overhead console, is ON, current is delivered through each timer to their inner and outer heating elements sequentially. The cycle duration is 60 to 68 seconds or 30 to 34 seconds when the de-ice selector switch, adjacent to the loadmeter, is positioned to the LH or RH PROP DE-ICE position.

7.2 TIMER

Applicable to aircraft S/N 652SA (See Fig. 9-17)

The timer is installed in the trailing edge of the center wing and distributes current in the sequence described in Para. 7.1 above to the de-ice strips. The sequence of strip activation must be: right outboard, right inboard, left outboard, left inboard heaters.

Applicable to aircraft S/N 661SA, 697SA and subsequent

The timers are attached to the inboard rib of the wing leading edge adjacent to the fuselage fillet. Each of the dual timers works its propeller de-ice strips alternating between the outer and inner elements at 30 to 34 second intervals.

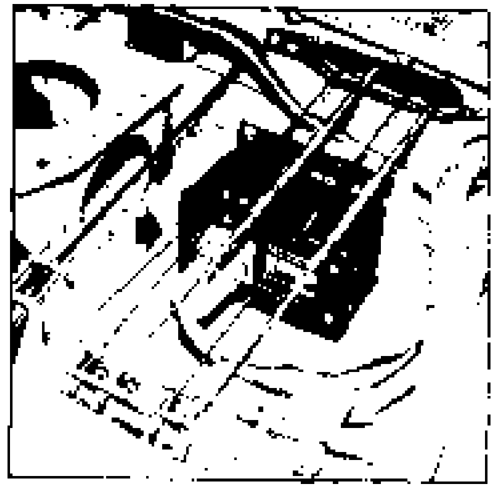


Fig. 9-17 Propeller anti-icing timer
(S/N 652SA only)

Fig. 9-18 Propeller slip ring
and brush block

7.3 BRUSH BLOCK AND SLIP RING (See Fig. 9-18)

The slip ring is a copper ring installed in the rear section of the propeller spinner bulkhead and consists of 3 coaxial rings for propeller outer and inner circuits and a common return line. Brushes are installed in contact with these slip rings and supply electric current to the slip ring.

7.3.1 BRUSH BLOCK (Fig. 9-19)

(1) Removal and installation
(a) Remove engine cowling.
(b) Remove brush block mounting screws and brush block at rear of propeller spinner bulkhead.

CAUTION

If brush block is over-tightened, the brush may not contact the ring freely. Torque to 14.5 ± 1.5 in-lb.

(c) Install in reverse sequence of removal.

CAUTION

Install brush and slip ring with care. The improper installation may hinder proper operation of de-icing system and cause radio noise.

(2) Inspection

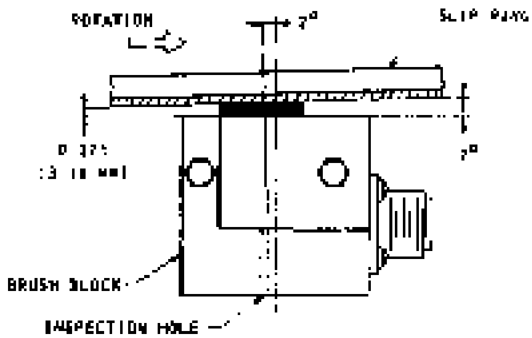
(a) Inspect the contact condition of the brush and slip ring after installation, rotate the propeller 360° (in the direction of normal rotation only) and check for local wear and eccentricity.

S/N 652SA, 661SA, 697SA thru 730SA unless modified by SL003/30-001
(b) Check the brush wear by inserting a piece of wire (use as a depth gauge) into the inspection hole of the brush block. Replace the brush block when the wire can be inserted more than 0.40 inch (10.2 mm).

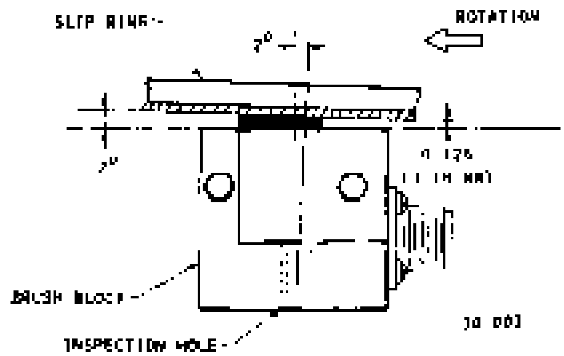
Aircraft modified by SL003/30-001
(b) Check the brush wear by inserting a piece of wire, 0.10 inch (2.5 mm) dia., (use as a depth gauge) into the inspection hole of the module. The module should be replaced when the wire can be inserted to a depth of 1.08 inch (27.4 mm) and must be replaced when the depth is 1.14 inch (29.0 mm).

NOTE

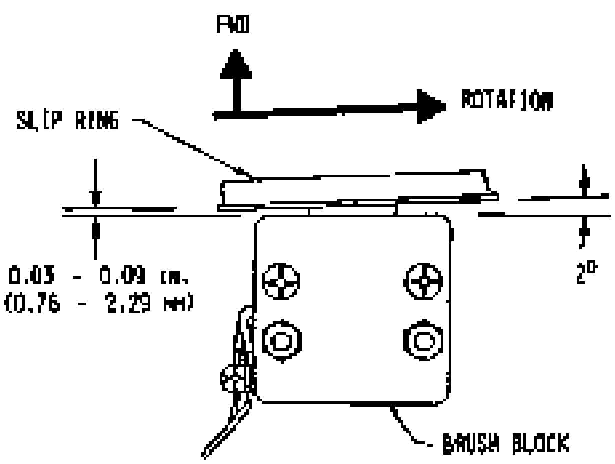
Some modules have a rod which protrudes through the inspection hole. As the brush wears, the rod moves into the hole. With a piece of wire, check the depth of the rod. When the depth becomes 0.23 inch (6.0 mm), the module should be replaced. When the depth becomes 0.30 inch (7.5 mm), the module must be replaced.



Aircraft S/N 652SA Unless
Modified By SL003/30-001



Aircraft S/N 661SA, 697SA thru 730SA
Unless Modified By SL003/30-001



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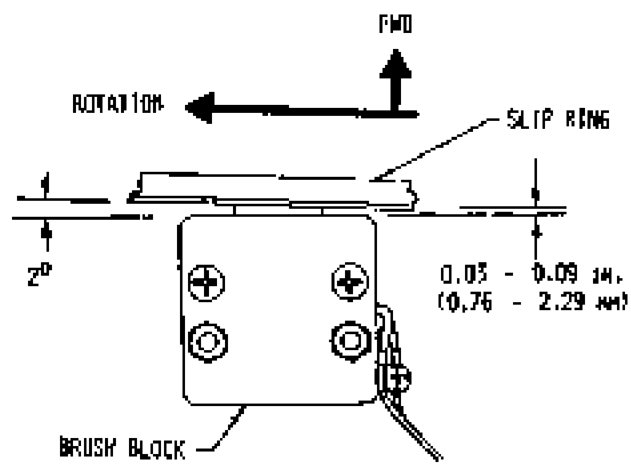
Modified By SL003/30-001



Aircraft S/N 661SA, 697SA thru 730SA
If Modified By SL003/30-001

Fig 9-19 Installation of brush block

7.3.2 INSPECTION AND ADJUSTMENT OF SLIP RING

(1) Inspection

With a dial gauge installed, check for uneven wear or wobble by rotating the propeller. If run-out over 360° rotation is over 0.005 inch (0.13 mm) total or exceeds 0.002 inch (0.05 mm) arc, adjust in the following manner.

NOTE

Push aft on propeller to eliminate play in propeller thrust bearing while turning.

(2) Adjustment

(a) Use washer AN960C416L between slip ring and spinner bulkhead to shim for true running. If necessary, fabricate thinner shim to the proper thickness.

NOTE

Place AN963B416 lock washer between two AN960C416L washers and apply shim above them. If the relative position of brush and slip ring has changed due to the above operation, readjust in accordance with Para. 7.3.I(2).

(b) In mounting the slip ring assembly to the spinner bulkhead, snug mounting bolts to approximately 25 in-lbs (29 kg/cm) of torque. Using the dial indicator to allow the points of maximum deviation, adjust slip ring assembly to prescribed run-out by gradually tightening the mounting bolts until all are within 40 to 75 in-lbs (46 to 86 kg/cm) torque.

(c) The wiring connection to the stud soldered on the slip ring requires 10 to 12 in-lbs (12 to 14 kg/cm) torque.

NOTE

When slip ring wiring connection torque is excessive, stud may fail.

(d) If brush loses contact with the slip ring during rotation and run-out can not be corrected by adjustment, replace slip ring.

7.3.3 MEASUREMENT OF INSULATION RESISTANCE OF BRUSH BLOCK

(1) To determine when open or short circuit or high resistance is present in the brush block, use a low range multi-tester to measure resistance from the face of the brush to its receptacle pin or terminal stud. The probe contacting the brush should have an area of 1/16 sq. in. minimum. If the resistance is over 0.013 ohms, locate and repair the cause of excessive resistance or replace the brush.

(2) Check the insulation resistance between the connector pins (and pins to connector shell) or terminal studs. The resistance should not be less than 0.5 megohms after one minute. Should the resistance be less than 0.5 megohms, check for a short between the brush leads and repair in accordance with B.F. Goodrich Overhaul Manual Report No. 68-04-714.

7.4 DE-ICER (See Fig. 9-20)

The de-icers are rubber strips glued on the leading edge of the propeller blades and containing resistance heating wire (hereinafter referred to as heater) inside. The heater consists of 2 circuits, one for de-icing of the outer half of the blade and one for the inner half. The power supply wire to and return wire from the heater are connected to slip rings with lead strips and screws.

7.4.1 MEASUREMENT OF HEATER ELEMENT RESISTANCE (See Fig. 9-21)

There are two methods for checking the resistance of the heating elements. First, check the resistance of inner and outer elements of the propeller. The parallel resistance of three elements should be 1.55 to 1.78 ohms and of four elements should be 1.15 to 1.33 ohms between the power supply and return (ground) wires. If not, then check the resistance of each element to be within 4.58 to 5.26 ohms between the power supply and return (ground) wires.

Fig. 9-20 De-icer boots

Fig. 9-21 Electrical connection of de-icer boots

7.5 OPERATIONAL CHECK OF TIMER

Operational check of the timer should be performed in accordance with B.F. Goodrich Maintenance Manual (Report No. 68-04-712).

7.6 OPERATIONAL TEST OF PROPELLER DE-ICING SYSTEM

7.6.1 TEST FOR ENTIRE SYSTEM

Applicable to aircraft S/N 652SA

(1) Turn aircraft power ON, use of APU is preferred, and close the necessary circuit breakers.

(2) Place the PROP DE-ICER switch in the overhead console ON. Observe and confirm the loadmeter needle movement from the timer automatic cycling of 30 to 34 seconds. Touch each propeller blade de-icer with bare hand to confirm all blades are heated uniformly and in proper sequence.

(3) Place the switch to OFF and restore the system to the original condition.

ORIGINAL
As Received By
ATP

Applicable to aircraft S/N 661SA, 697SA and subsequent

(1) Turn aircraft power ON, use of APU is preferred, and close required circuit breakers for LH and RH PROP DE-ICE.

(2) Turn the HEATER CURRENT SELECT switch to LH PROP position and place the LH PROP DE-ICE switch ON.

(3) Observe and confirm deflection of load meter needle between 0.85 and 1.02 at time intervals of 30 to 34 seconds. Touch each propeller blade de-icer with bare hand and confirm uniform heating and proper sequence per Para. 7.1 and 7.2.

(4) Place LH PROP DE-ICE switch OFF.

(5) Turn HEATER CURRENT SELECT switch to RH PROP position and place RH PROP DE-ICE switch ON.

(6) Perform steps (2) and (3) above for RH prop blade de-icer.

(7) Place RH PROP DE-ICE switch OFF.

(8) Turn aircraft power OFF, disconnect APU (if used).

7.6.2 ELECTRICAL LOAD TEST

Applicable to aircraft S/N 652SA

(1) Remove the overhead switch console and connect an ammeter (25 to 30 Amps) to the load side of PROP DE-ICER switch.

(2) Place the propeller de-icing switch PROP DE-ICER in the overhead console ON. When power is supplied from the aircraft battery only, the ammeter must read 12 to 15.5 Amps, and the loadmeter must read 0.7 to 0.9. When power is supplied from the generator, with the engine running at Ground Idle, the ammeter must read 14 to 18 Amps and the loadmeter must read 0.8 to 1.0.

Applicable to aircraft S/N 661SA, 697SA and subsequent

(1) Connect ammeter (25 to 30 Amps) in line to terminal 8 of the prop timer. When power is supplied to the de-icer boots, the ammeter should read 17 to 20 Amps while the loadmeter deflects between 0.85 to 1.02 for both LH and RH prop de-ice timers.

7.7 TROUBLE SHOOTING PROPELLER DE-ICING SYSTEM

Trouble	Probable cause	Remedy
Load meter does not deflect when switch is turned on	a. Faulty switch.	Replace switch.
	b. Power source is disconnected	Check electrical power system.
	c. Faulty loadmeter, fuse or shunt wire.	Check and/or replace.
	d. Faulty timer.	Replace timer.
	e. Wire broken.	Repair wire.
Heating is out of sequence	a. Faulty wiring.	Check wiring.
Load meter indication is too low	a. Corresponding de-icer or wiring is broken.	Replace or repair wire or de-icer.
	b. Contact of corresponding slip ring and brush is poor.	Correct attachment conditions.
Load meter indication is too high	a. Short circuit of de-icer or circuit.	Replace or repair.
	b. Short circuit of diode.	Replace diode.
Brush wears	a. Brush is attached improperly.	Correct brush attachment.
	b. Slip ring is eccentric.	Adjust slip ring attachment or replace slip ring.
Radio noise is produced when de-icer is operated	a. Brush arc.	Polish or clean slip ring and brush and adjust attachment. Replace brush or slip ring if necessary.
	b. Faulty contact of wiring connection.	Check circuit and overhaul connector and/or terminal section.
	c. Faulty switch.	Apply jumper to switch and replace switch if noise disappears.

8. PITOT AND STATIC ANTI-ICING SYSTEM

8.1 GENERAL DESCRIPTION

The anti-icing of the pitot tubes and static ports is carried on continuously by electrical heaters installed in the units. These circuits are protected by the OVERHEAD circuit breakers. The system is operated by means of LH and RH PITOT & STATIC HEAT switch located in the overhead switch console.

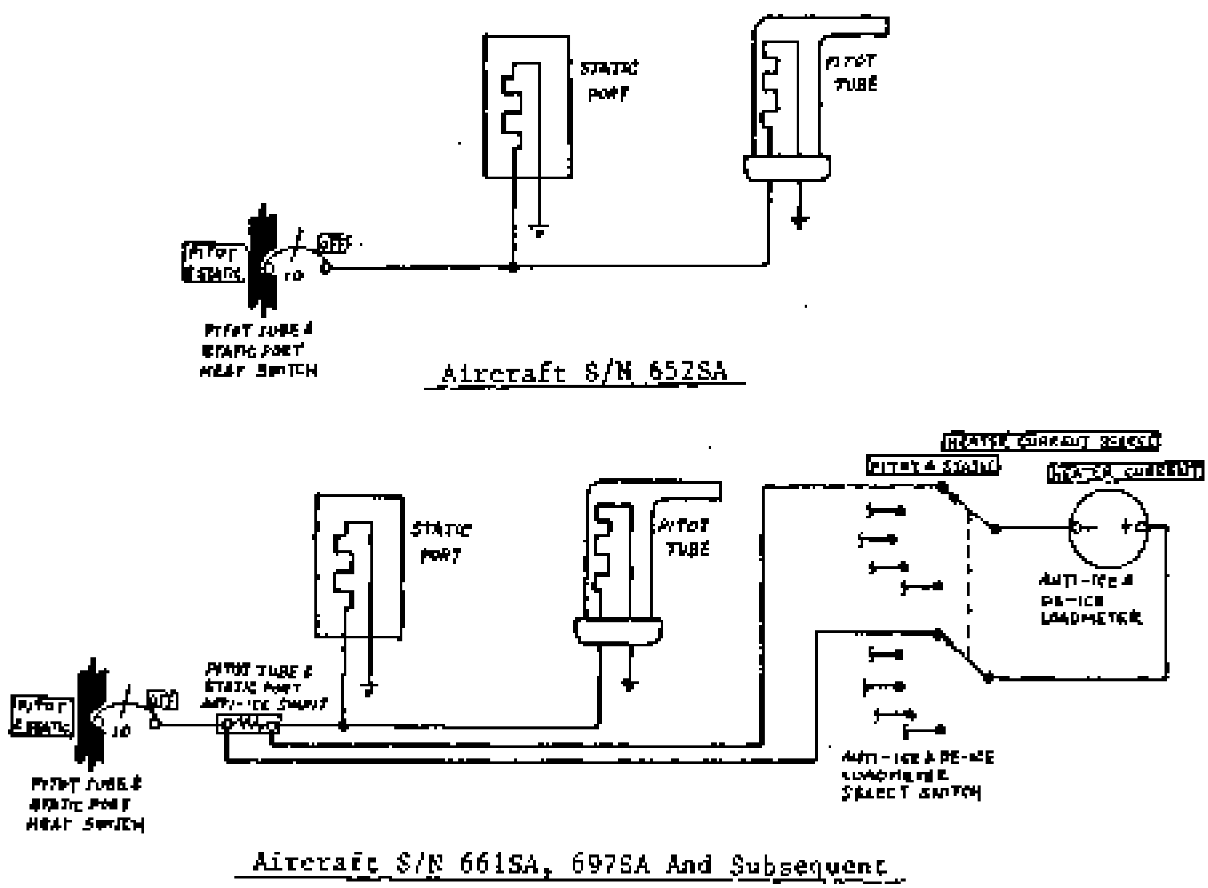


Fig 9-21A Pitot and static anti-icing system schematic

8.2 MAINTENANCE PRACTICES

8.2.1 STATIC PORT ANTI-ICE HEATER

A. Removal and Installation

(1) Disconnect the electrical wiring at the static port.

(2) Disconnect the plumbing.

(3) Remove 6 screws.

(4) Remove the static port from inside the aircraft.

(5) Install in reverse sequence sealing with PR1222 sealant.

B. Adjustment and Test

(1) Operational check of the static port heater should be accomplished as follows:
   (a) Turn the Battery Key Switch to the ON position.
   (b) Ensure the left and right overhead console circuit breakers are closed.
   (c) Place the left and right Pitot and Static Anti-Ice switches to the ON position.
   (d) Applicable to Aircraft S/N 652SA

   Ensure the heating of the static ports by touching them with the hand and feeling for warmth.

CAUTION

Max 10 sec ON during ground operation.

Applicable to Aircraft S/N 661SA, 697SA and subsequent

Place HEATER CURRENT SELECT switch to LH PITOT AND STATIC position and check loadmeter for operating range of 0.50 to 0.85. Place selector switch to RH PITOT AND STATIC position and check loadmeter for operating range of 0.50 to 0.85.

CAUTION

Max 10 sec ON during ground operation.

   (e) Place switches to OFF.
   (f) Turn Battery Key switch to OFF position.

8.3 TROUBLE SHOOTING

Trouble	Probable cause	Remedy
Loadmeter does not indicate.	a. Loadmeter is inoperative. b. Switch is defect. c. Wiring is broken.	Replace loadmeter. Replace switch. Repair wiring.
Loadmeter indication is too low or high.	a. Loadmeter is defect. b. Heater of pitot tube or static port is defect.	Replace loadmeter. Replace defect part

8.2.2 PITOT TUBE ANTI-ICE HEATER

A. Removal and Installation

(1) Remove pilot tube from pitot tube fitting inside the aircraft.

(2) Remove connector for heater from inside the aircraft.

(3) Remove pitot tube. Remove heater by removing 4 screws attaching heater.

(4) Install in reverse sequence of removal, sealing as required.

B. Adjustment and Test

(1) Operational check of the pitot tube heater should be accomplished as follows:
(a) Turn the Battery Key switch to the ON position.
(b) Ensure the left and right overhead console circuit breakers are closed.
(c) Place the left and right Static and Pitot Anti-Ice switches to the ON position.
(d) Applicable to Aircraft S/N 652SA

Ensure the heating of the pitot tubes by touching them with the hand and feeling for warmth.

CAUTION

Max 10 sec ON during ground operation.

Applicable to Aircraft S/N 661SA, 697SA and subsequent

Place HEATER CURRENT SELECT switch to LH PITOT AND STATIC position and check loadmeter for operating range of 0.50 to 0.85. Place selector switch to RH PITOT AND STATIC position and check loadmeter for operating range of 0.50 to 0.85.

CAUTION

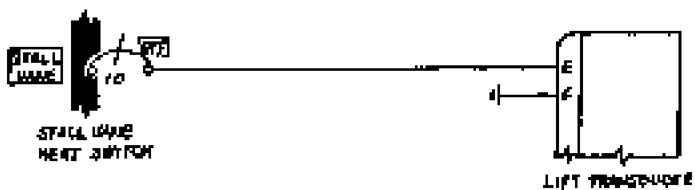
Max 10 sec ON during ground operation.

(e) Place switches to OFF.
(f) Turn Battery Key switch to OFF position.

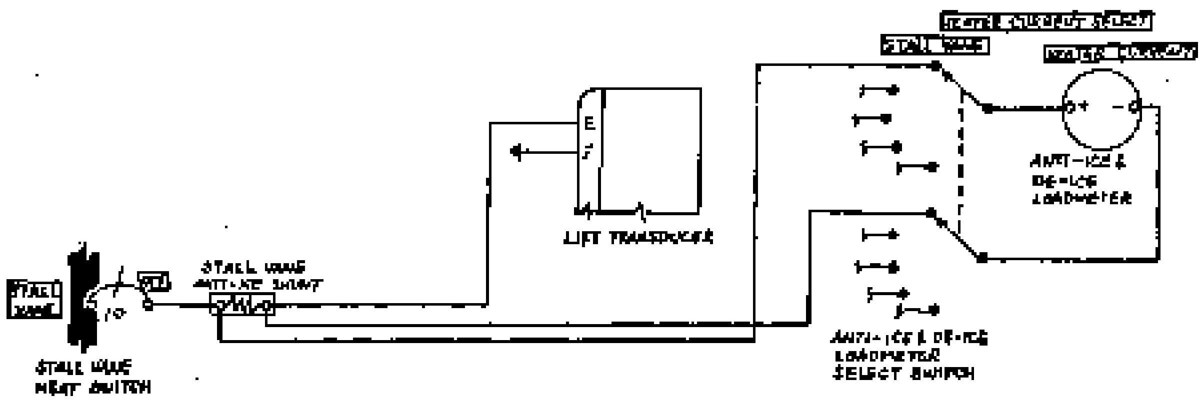
9. STALL WARNING ANTI-ICING SYSTEM

9.1 GENERAL DESCRIPTION

The anti-icing of the lift transducer of the stall warning system is accomplished by electrical heaters installed in the unit. This anti-icing circuit is operated by placing the STALL VANE HEAT switch in the overhead console to ON position.



Aircraft S/N 652SA



Aircraft S/N 661SA, 697SA thru 730SA

Fig 9-21B Stall warning anti-icing system schematic

9.2 REMOVAL AND INSTALLATION OF LIFT TRANSDUCER

(1) Remove access panel at RH W.STA 4300.

(2) Remove electric connector.

(3) Remove sealant from lift transducer.

(4) Remove lift transducer by removing 4 screws attaching lift transducer.

(5) Install in reverse sequence of removal, sealing as required.

9.3 OPERATIONAL CHECK OF LIFT TRANSDUCER

Operational check of the Stall Warning Anti-Ice System should be accomplished as follows:

(1) Turn the Battery Key switch to the ON position.

(2) Ensure the overhead console circuit breakers are closed in.

(3) Place the STALL VANE HEAT switch to ON.

(4) Applicable to Aircraft S/N 652SA

Ensure the heating of the lift transducer by touching it with the hand and feeling for warmth.

CAUTION

Max 10 sec ON during ground operation.

Applicable to Aircraft S/N 661SA, 697SA and subsequent

Turn HEATER CURRENT SELECT switch to STALL VANE position and check loadmeter for operating range of 0.30 to 0.70.

CAUTION

Max 10 sec ON during ground operation.

(5) Place switch to OFF.

(6) Turn Battery Key switch to OFF position.

9.4 TROUBLE SHOOTING

Trouble	Probable cause	Remedy
Lift transducer does not heat	a. Broken wiring b. Defective switch c. Defective lift transducer heating element	Repair wiring Replace switch Replace lift transducer
Load meter does not indicate (*1)	a. Broken wiring b. Defective shunt c. Defective load meter	Repair wiring Replace shunt Replace load meter
Load meter indication is too low or high (*1)	a. Defective shunt b. Static port or pitot tube heating element defective	Replace shunt Replace defective part.

*1 Aircraft S/N 661SA, 697SA thru 730SA

10. OIL COOLER INLET ANTI-ICE HEAT

10.1 GENERAL DESCRIPTION

The oil cooler air inlet electrical heater protects the oil cooler air inlet from ice build-up during icing conditions. The rubber boots with embedded electrical heating elements are installed on the air inlet of both engine oil coolers. The maximum operating temperature of the heating elements is limited by the thermostat installed under the rubber boots. When the oil cooler inlet anti-ice switch in the overhead console is placed in the ON position, the electrical power is supplied to the heating elements and the indicator light illuminates.

CAUTION

Oil cooler inlet anti-icing system must not be operated on ground for more than 10 seconds.

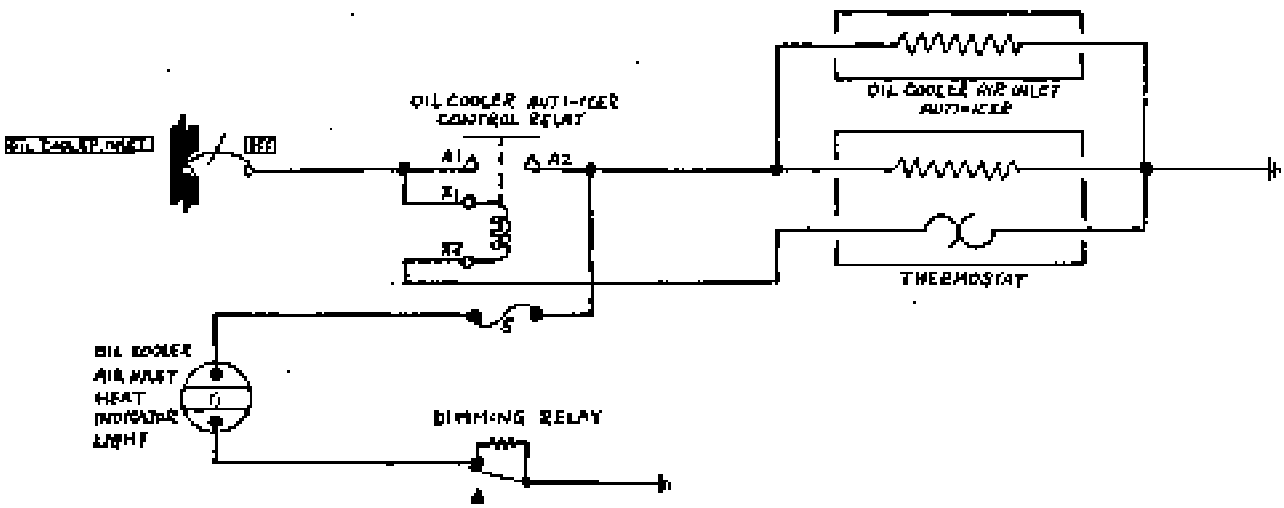


Fig 9-21C Oil cooler anti-icing system schematic

10.2 ANTI-ICER

The anti-icer heater boots are attached to oil cooler air intake with adhesives (EC 1300L from 3M Co.). To clean anti-icer heater boots, use neutral soap water. In cold weather, cleaning should be accomplished in a warm hangar, or with warm soap water.

10.3 THERMOSTAT

The thermostat is attached to oil cooler air intake with epoxide resin (EC 2216 A/B from 3M Co.) and control temperature. Operating temperature is 123°F (50.6°C) for ON and 145°F (62.8°C) for OFF.

10.4 OPERATIONAL CHECK

(1) Turn the Battery Key switch to the ON position.

(2) Check for closed condition of the LH and RH overhead circuit breakers.

(3) Place LH and RH OIL COOLER HEAT switch to the ON position.

(4) Check LH and RH OIL COOLER HEAT indicator lights for illumination.

(5) Confirm system is operational by touching the left and right engine oil cooler inlet heat boots with the hand.

(6) Place the LH and RH OIL COOLER switches to the OFF position.

(7) Turn the Battery Key switch to OFF.

10.5 TROUBLE SHOOTING

Trouble	Probable cause	Remedy
Indicator light does not illuminate with system ON	a. Faulty light.	Replace.
	b. Poor wire connection.	Check wire connection with ohmmeter.
Boots will not heat	a. Poor wire connection.	Check wire connection.
	b. Faulty relay.	Replace relay.
	c. Faulty heating element in boot.	Replace boot.
	d. Malfunction in thermostat.	Replace.

CHAPTER 10

INSTRUMENTS

CHAPTER X

INSTRUMENTS

CONTENTS

1. GENERAL

1.1 GENERAL DESCRIPTION

The instrument panel is installed in front of the pilots, and consists of LH instrument panel, center instrument panel, and RH instrument panel. LH and RH instrument panels are mounted on the frames with quick fasteners to facilitate removal. The center instrument panel forms integral construction with frames and is supported by 6 shock mounts on the lower sections and 4 on the upper sections. Instruments are lighted by pillar lights above each instrument. Brightness of lighting is controlled by rheostats (PILOT FLT INST, COPILOT FLT INST and ENG INST) installed in the overhead switch console. Switch panels consist of LH and RH switch panels installed below LH and RH instrument panels at a 20° angle, and the overhead switch console is located in the ceiling above the pilot and co-pilot. The switch panels are edge-lighted by bulbs mounted in the panel faces. Brightness of the lighting is controlled by rheostats (COPILOT FLT INST, PANEL and ENG INST) located in the overhead switch console. Aircraft power source 28 VDC is supplied to the lights through circuit breaker INST PANEL.

1.2 RANGE MARKING

See Figure 10-1 for range marking of major instruments.

1.3 INSTRUMENT PANEL

Various kinds of instruments are installed on the LH, RH and center instrument panels as shown in Figure 10-2. Instrument panels are made of aluminum alloy sheet 1/8 inch (3.2 mm) in thickness and are painted with matte acrylic lacquer and lighted by pillar lights. For vibration prevention, rubber type shock mounts are used. Rubber type shock mounts should be inspected for mounting condition and vibration prevention effects. When excessive stretching is observed or vibration prevention effect is abnormal, shock mounts should be replaced.

1.3.1 REMOVAL AND INSTALLATION OF LH INSTRUMENT PANEL (See Fig. 10-3)

(1) Disconnect hose of pitot, static pressure and vacuum system at union, F.STA 1440, back of instrument panel.

(2) Disconnect plug at connection box, back of center pedestal.

(3) Remove fasteners attaching LH instrument panel.

(4) Pull control wheel and turn to the right, and remove LH instrument panel.

(5) Install in reverse sequence of removal.

1.3.2 REMOVAL AND INSTALLATION OF RH INSTRUMENT PANEL (See Fig. 10-3)

(1) Disconnect hose of pitot, static pressure and vacuum system at union, F.STA 1440, back of instrument panel.

(2) Disconnect plug at connection box, back of center pedestal.

(3) Remove fasteners attaching RH instrument panel.

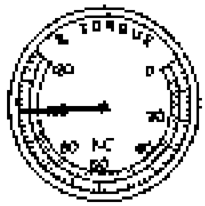
(4) Pull control wheel and turn to the left, and remove RH instrument panel.

(5) Install in reverse sequence of removal.

TORQUEMETER (% Q)

G 0 to 100

R 100



OIL TEMP IND (°C)

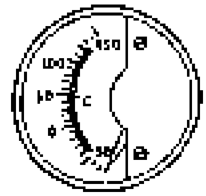
R -40

Y -40 to 55

G 55 to 110

Y 110 to 127

R 127



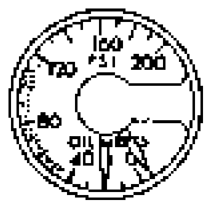
OIL PRESS IND (PSI)

R 40

Y 40 to 70

G 70 to 120

R 120



FUEL PRESS IND (PSI)

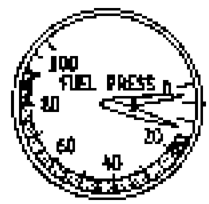
R 15

Y 15 to 20

G 20 to 80

Y 80 to 90

R 90

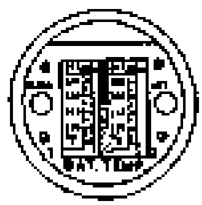


BATTERY TEMP IND (°F)

G 0 to 120

Y 120 to 150

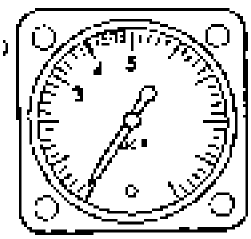
R 150 to 190



VACUUM GAUGE (IN-HG)

A 4.0

G 4.2 to 5.0



Aircraft S/N 313SA & 349SA

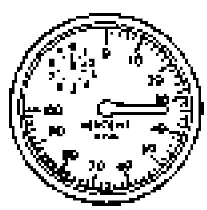
TACHOMETER (% RPM)

Y 65 to 96%

G 96 to 100%

Y 100 to 101%

R 101%



Aircraft S/N 321SA, 348SA, 350SA thru 394SA

TACHOMETER (% RPM)

R 50 to 76.5

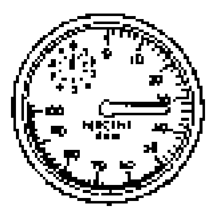
Y 76.5 to 96

G 96 to 100

Y 72 to 101

Y 100 to 101

R 101



G GREEN

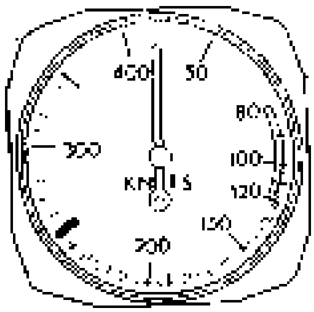
Y YELLOW

R RED

W WHITE

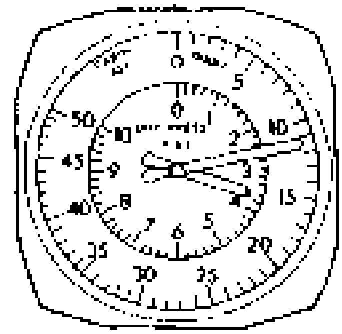
Fig. 10-1 Instrument range markings (1/2)

Aircraft S/N 652SA



AIRSPEED IND.

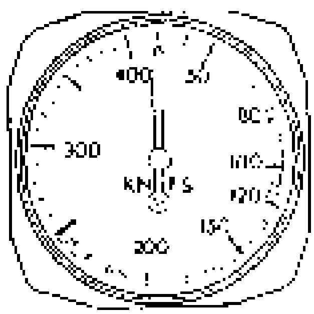
W 80 to 120 KT
G 104 TO 250 KT
R 250 KT



CABIN ALTITUDE DIFF. PRESS. IND

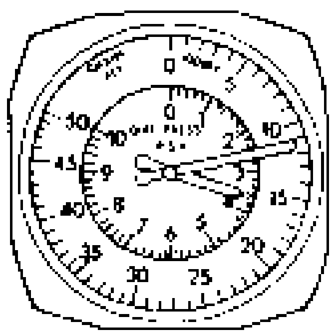
G 0 TO 5.25 PSI
R 5.25

Aircraft S/N 661SA, 697SA and subsequent



AIRSPEED IND (KTS)

W 80 to 120
G 106 to 250
R 250



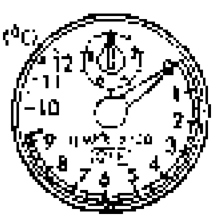
CABIN ALTITUDE DIFF PRESS IND (PSI)

G 0 to 6.00
R 6.10

Aircraft S/N 652SA, unless modified by Kit Drawing K926A-8202

INTERSTAGE TURBINE TEMP IND (°C)

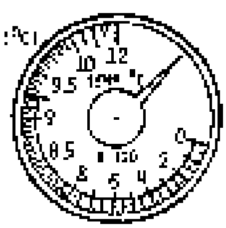
G 0 to 923
R 923
W 1149

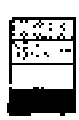


Aircraft S/N 661SA, 697SA and subsequent, and aircraft modified by Kit Drawing K926A-8202

INTERSTAGE TURBINE TEMP IND (°C)

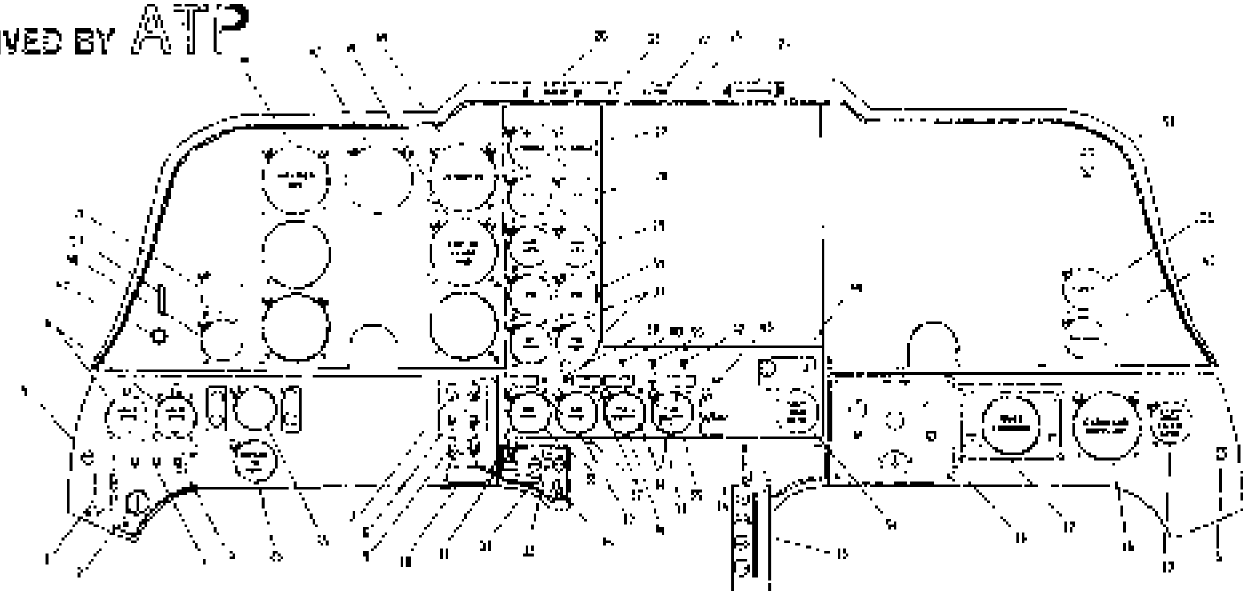
G 0 to 923
R 923
W 1149





G GREEN
Y YELLOW
W WHITE
R RED

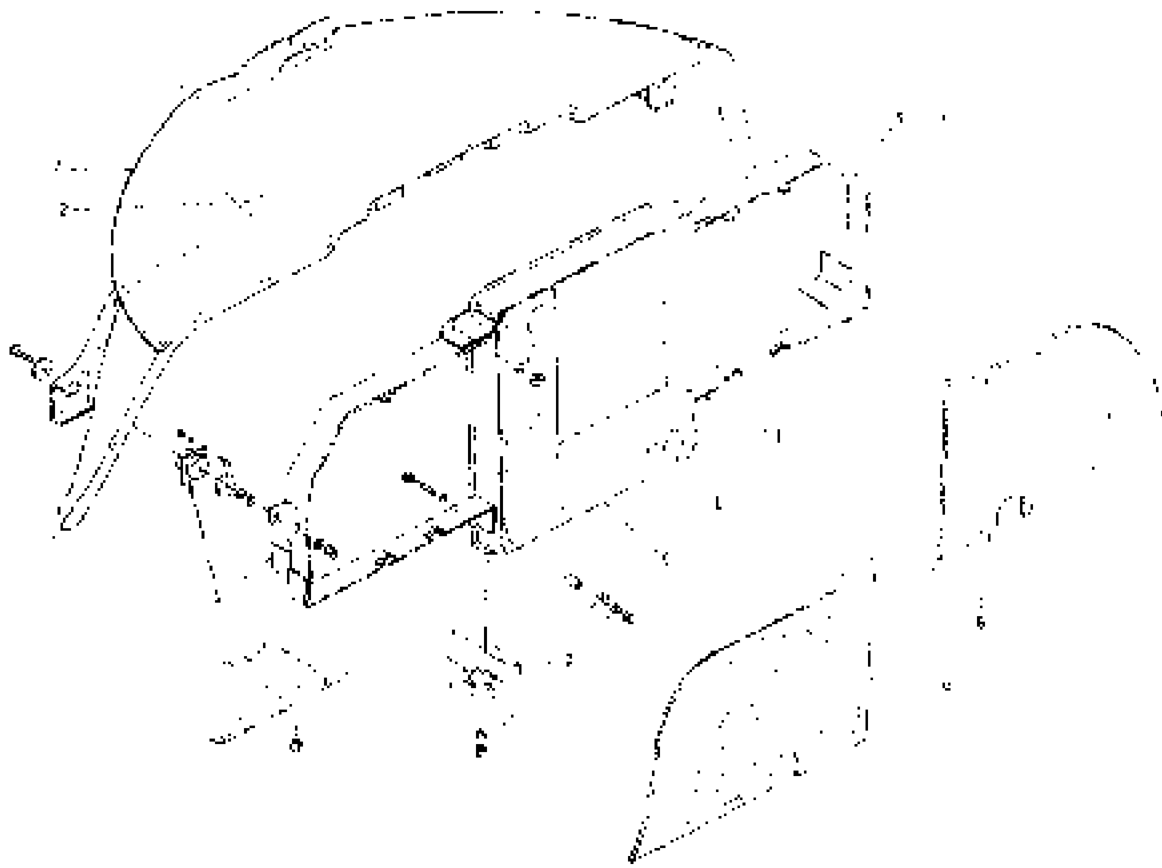
Fig. 10-1 Instrument range markings (2/2)



1. Master switch
2. Battery switch
3. DC generator control switches
4. Inverter switch
5. Microphone jack
6. Volt-ammeters
7. Main fuel valve switches
8. Transfer switches
9. Power auto limit switches
10. Trim aileron select switch
11. Landing gear control switch
12. Landing gear position indicating lights
13. Landing gear unsafe light
14. Landing gear horn cutout switch
15. Flap switch
16. Air conditioning control panel
17. Cabin controller
18. Cabin altitude differential pressure indicator
19. Cabin rate-of-climb indicator
20. LH engine fire extinguisher handle
21. Engine fire detector test button
22. Master caution light
23. Master caution system test switch
24. RH engine fire extinguisher handle
25. Outside air temperature indicator
26. Clock
27. Torque meters
28. ITT indicators
29. Fuel flow indicators
30. Tachometers
31. Oil pressure indicator
32. Fuel pressure indicator
33. Oil temperature indicator
34. Main tank fuel quantity indicator
35. Tip tank fuel quantity indicator
36. Fuel consumption totalizer
37. Beta range indicator lights
38. Outer fuel empty warning lights
39. Power auto limit test switches
40. Panel indicator light test switch
41. Fuel quantity indicator test switch
42. Stall warning test switch
43. Fuel low level test switch
44. Prop. synchrophaser switch
45. Vacuum gauge
46. Airspeed indicator
47. Attitude gyro
48. Altimeter
49. Rate-of-climb indicator
50. Turn and bank indicator
51. Turn and bank indicator power fail lights(*1)
52. Defog air over temperature warning light(*1)
53. Battery temperature indicator and isolate switches
54. Hour meter
55. Cabin sign light switch(*2)
56. Alternate static select valve

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 652SA only

Fig. 10-2 Instrument and switch panel (Typical)



1. Lower Shock Mount (6 ea.)
2. Upper Shock Mount (4 ea.)
3. Mounting Frame
4. LH Instrument Panel
5. Center Instrument Panel
6. RH Instrument Panel
7. Shroud Panel

Fig. 10-3 Instrument panel

1.3.3 REMOVAL AND INSTALLATION OF RADIO PANEL

(1) Remove quick fasteners attaching radio panel.

(2) Disconnect electric connector for radio equipment.

(3) Remove radio panel.

(4) Reinstall in reverse sequence of removal.

1.3.4 REMOVAL AND INSTALLATION OF SHROUD PANEL

(1) After removing quick fasteners, pull out LH and RH instrument panel.

(2) Remove 4 shock mounts attached on upper section of shroud panel.

(3) Remove screws and remove panel.

(4) Remove 4 screws and 4 bolts attaching shroud panel to frame and remove the panel.

(5) Reinstall in reverse sequence of removal.

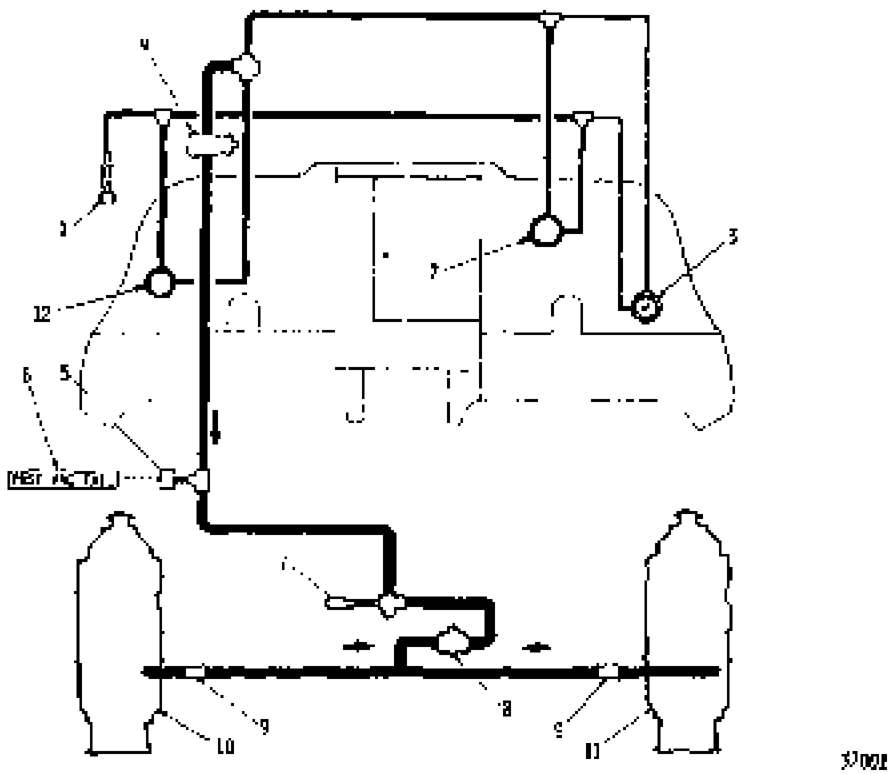
2. VACUUM SYSTEM

2.1 GENERAL DESCRIPTION

The vacuum system operates air driven type gyros by suction pressure. This system consists of one ejector, one relief valve, one filter and vacuum gauge. A warning switch is installed for low vacuum pressure warning.

This system regulates vacuum pressure produced by ejector operation, by a relief valve, and rotates the gyros in the instruments (See Fig. 10-4).

Vacuum pressure is adjusted to 4.6 in-Hg at the instrument end and is shown on a vacuum gauge. A filter through which clean air is supplied is installed on the suction side of the instruments. The ejector is operated while engines are operative. If either one of the engines is inoperative, vacuum pressure required for instrument operation is available from the other engine. A vacuum pressure warning switch is installed in the vacuum line. The warning switch is actuated when vacuum pressure falls below 4 in-Hg, lighting the warning light on the annunciator panel. When the engine is inoperative or an ejector is faulty, the trouble can be warned.



1. Air filter
2. Attitude gyro
3. Vacuum gauge
4. Relief valve
5. Vacuum pressure warning switch
6. Annunciator panel
7. Ejector
8. Regulator valve
9. Check valve
10. LH engine
11. RH engine
12. Turn and bank indicator

Fig. 10-4 Vacuum system

2.2 EJECTOR

The ejector is operated by engine bleed air pressure and is installed on the LH wing-fuselage fillet and produces vacuum pressure through ejector operation.

2.1.1 REMOVAL AND INSTALLATION (See Fig. 10-5)

(1) Disconnect tube from ejector.

(2) Remove screws and nuts attaching the ejector.

(3) Reinstall in reverse sequence of removal.

2.3 RELIEF VALVE

The relief valve regulates vacuum pressure, produced by the ejector, to the pressure required by the instruments.

When suction pressure rises, outside air flows into the valve through a screen and, when suction pressure falls, air flow is stopped by spring tension. The spring tension may be adjusted by the upper screw which regulates the suction pressure. The valve is installed on the LH section in the back of the instrument panel.

2.3.1 REMOVAL AND INSTALLATION (See Fig. 10-6 and 10-7)

(1) Remove hose attaching clamps at valve ends.

(2) Remove screws and nuts attaching the valve (S/N 652SA and 661SA). Remove nuts attaching the valve (S/N 697SA and subsequent).

(3) Reinstall in reverse sequence of removal.

NOTE

Check for dust sticking to air intake. Clean with a non-petroleum base solvent and dry with compressed air.

2.4 VACUUM GAUGE

The vacuum gauge is a flange type instrument and is installed in the RH instrument panel with 4 screws. Range of indication is 0 to 10 in-Hg.

2.5 AIR FILTER (See Fig. 10-8)

The air filter is installed at the back of the LH instrument panel to avoid abnormal functioning of instruments due to foreign materials in the air.

Fig. 10-5 Installation of ejector

Fig. 10-6 Installation of relief valve (S/N 652SA, 661SA)

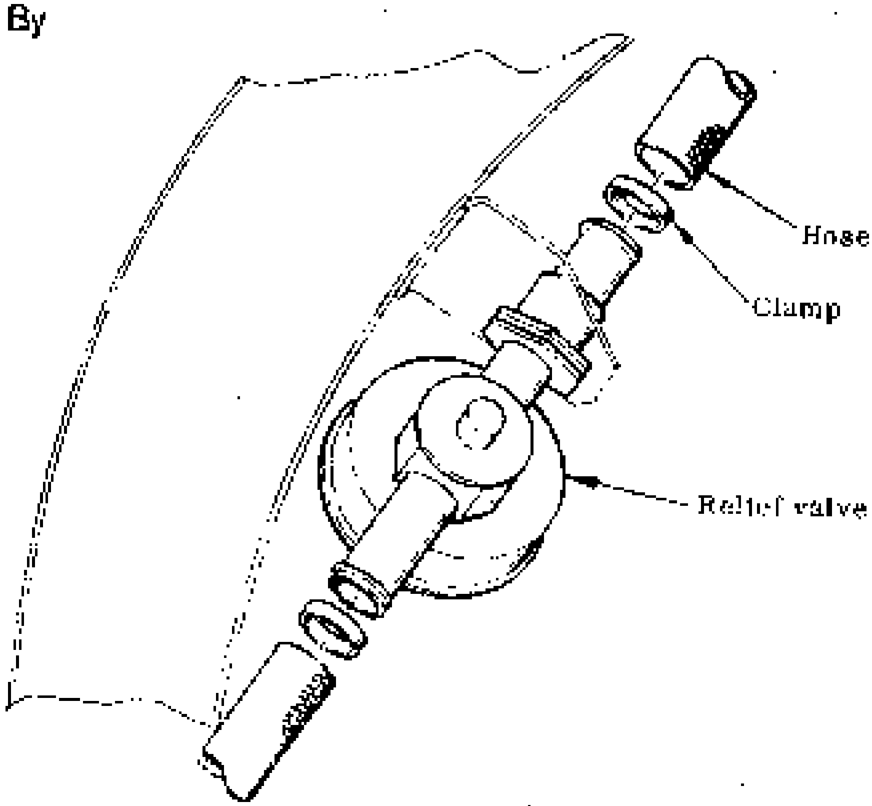


Fig. 10-7 Installation of relief valve (S/N 697SA and subsequent)

2.5.1 REMOVAL AND INSTALLATION

(1) Remove the filter front cap and remove the element.

(2) Reinstall in reverse sequence of removal.

2.6 WARNING SWITCH (See Fig. 10-9)

The warning switch is installed at F.STA 1275 and is actuated when vacuum falls below 4 in-Hg due to vacuum system failure. The warning switch illuminates INST VAC FAIL warning light.

2.7 OPERATION AND ADJUSTMENT OF VACUUM SYSTEM

(1) Start LH and RH engines and set power lever, condition lever and cabin air selector to the position shown in the table below.

(2) Adjust relief valve by turning a button and a nut on the center of the valve, so that vacuum gauge indicates 4.6 ± 0.4 in-Hg in the position 1 shown in the table below.

(3) Make sure that warning light INST VAC FAIL is off.

(4) Make sure that attitude indicator is operating normally.

(5) Set levers to position 2 shown in the table and make sure that the vacuum gauge indicates 4.6 ± 0.4 in-Hg and attitude indicator operates normally.

POSITION	CONDITION LEVER	POWER LEVER	CABIN AIR SELECTOR
1	Takeoff-land	Takeoff	Off
2	Taxi	Ground Idle	Both

Fig. 10-8 Installation of air filter

Fig. 10-9 Installation of vacuum pressure warning switch

2.8 LEAK TEST

(1) Disconnect hose (outlet side of each indicator) of instrument panel and plug it.

(2) Disconnect hoses, both ends of relief valve and connect pipe 5/8 in. (16 mm) in diameter, instead of valve.

(3) Disconnect ejector line, and apply vacuum pressure 5 in-Hg to the system.

(4) No leak should be found while vacuum pressure is applied.

(5) Restore aircraft to original configuration.

2.9 TROUBLE SHOOTING

Trouble	Probable cause	Remedy
Vacuum pressure insufficient	a. Relief valve faulty or adjustment erroneous.	Adjust relief valve or replace, if defective.
	b. Vacuum line is leaking or broken.	Perform line leak test and repair defective component.
	c. Ejector is broken or bleed air piping is leaking.	Repair or replace ejector. Perform leak test of bleed air line and repair or replace it.
	d. Vacuum gauge indication is faulty while relief valve is adjusted.	Repair or replace vacuum gauge.
Vacuum pressure is excessively high	a. Relief valve faulty or adjustment erroneous.	Adjust relief valve or replace it if defective.
	b. Blocked screen of relief valve.	Clean screen.
	c. Vacuum gauge indication is faulty while relief valve is adjusted.	Repair or replace vacuum gauge.
Vacuum gauge does not work	a. Line is blocked.	Clean line and replace or repair defective component.
	b. Instrument is defective.	Repair or replace vacuum gauge.
	c. Ejector is defective.	Replace ejector and conduct operational test.
Vacuum pressure warning light is on	a. Vacuum pressure is too low or is not available.	See "vacuum pressure insufficient".
	b. Electrical wiring is short circuited or warning switch is defective.	Perform continuity check and repair defective component or replace warning system.

3. PITOT AND STATIC PRESSURE SYSTEM

3.1 GENERAL DESCRIPTION

The pitot-static system consists of two pitot tubes, one each installed on the LH and RH forward section of the fuselage and static pressure port fittings on LH and RH lower sections at F.STA 2550. The pitot tube head is electrically heated by 28 VDC for anti-icing. (See Chapter IX Para. 6, Pitot Tube Anti-Icing System.) Quarter inch (6 mm) aluminum tubes are used between pitot and instruments and back of instrument and back of instrument panel, and 3/8 inch aluminum tubes are used from static pressure port to side panel. Flexible hoses are used between the back side of the instrument panel and instruments. The pitot is connected to the airspeed indicator and static pressure is connected to the airspeed, altimeter, rate-of-climb indicator and cabin altitude differential pressure indicator (see Fig. 10-13). An alternate static port is installed outside of the pressure bulkhead on the LH lower section at the back of the instrument panel. A selector valve (Fig. 10-12) is installed on the LH instrument panel for the purpose of an alternate static source, should the normal source become inoperative due to ice, rain, etc.

Instruments for pitot and static pressure system are very sensitive to air pressure and inspection should be performed very carefully. After the aircraft is washed or flight in rain, drain pitot and static lines. The drain is located near the LH and RH rudder pedals on the lower section at the back of the instrument panel (See Figs. 10-10 & 10-11). If the indication of instruments (airspeed indicator, altimeter, rate-of-climb indicator, etc.) is erratic after draining, apply low pressure air, 2.5 psig (0.18 kg/cm^2), to the lines to remove foreign materials.

NOTE

When pitot and static pressure lines are disconnected from instruments, cover connection ports of instruments and cap lines.

CAUTION

Do not connect pitot pressure to static lines.

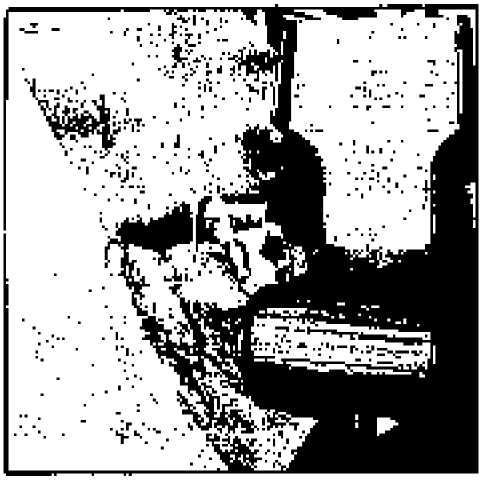
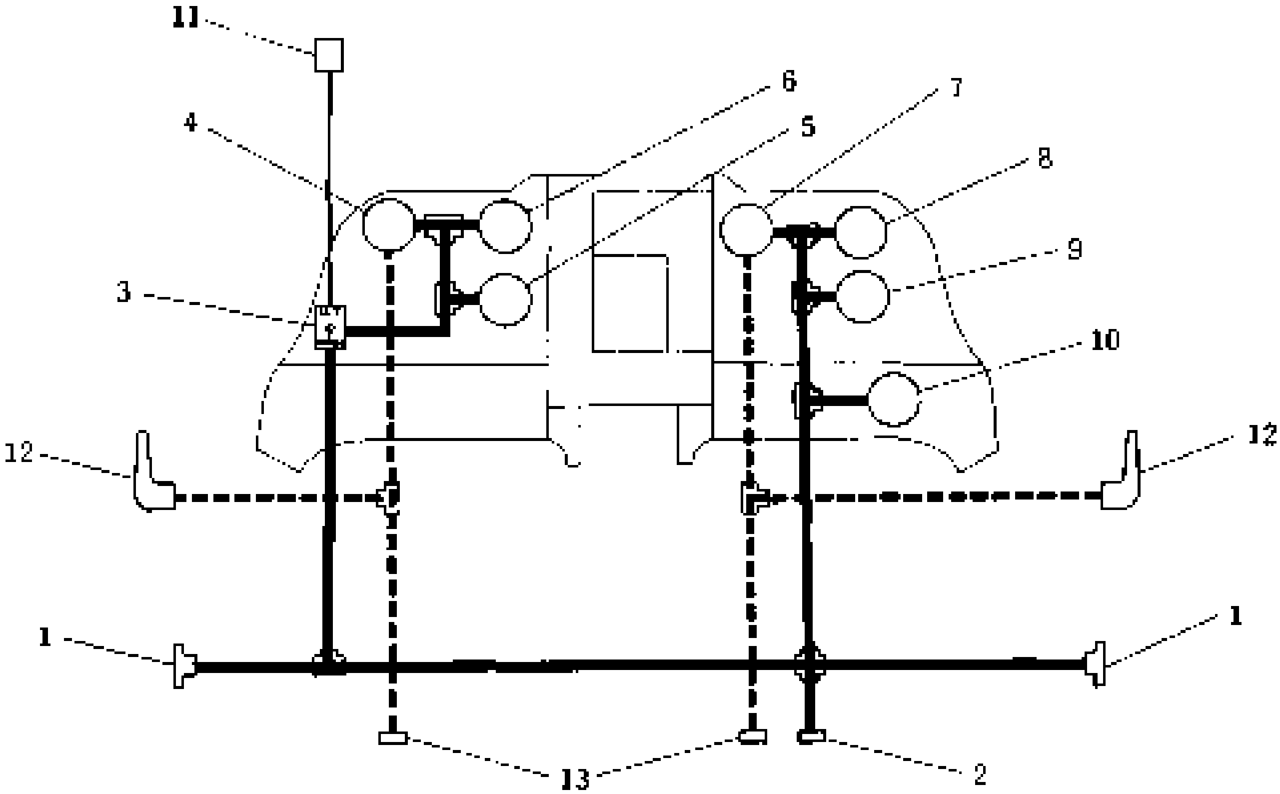


Fig.10-10 Drain for pitot tube

Fig.10-11 Drain for pitot static tube

ORIGINAL
As Received By
ATP

Fig.10-12 Alternate static source
valve



1. HEATED STATIC PORT
2. STATIC LINE DRAIN
3. ALTERNATE STATIC SELECT VALVE
4. PILOT AIRSPEED INDICATOR
5. PILOT RATE OF CLIMB INDICATOR
6. PILOT ALTIMETER INDICATOR
7. COPILOT AIRSPEED INDICATOR
8. COPILOT ALTIMETER INDICATOR
9. COPILOT RATE OF CLIMB INDICATOR
10. CABIN DIFFERENTIAL PRESSURE INDICATOR
11. ALTERNATE STATIC SOURCE
12. HEATED PITOT TUBE
13. PITOT LINE DRAIN

Fig.10-13 Pitot-Static system

3.2 LEAK TEST OF STATIC PRESSURE LINE

(1) Disconnect static lines to cabin altitude differential pressure indicator and airspeed indicator, and cap.

(2) Connect air source to static pressure air port and supply air regulated to about 2.5 psi. Check air blowing off LH and RH static pressure air ports and no blocking in the lines. After check, cover the ports with tape.

(3) Remove caps for cabin altitude differential pressure indicator and airspeed indicator, one after the other, and check air flowing out. After check, cap again.

(4) Cap static pressure line of tester MB-1, apply 12,000 ft. (*1), 13,800 ft. (*2) of negative pressure to tester and maintain for one minute, then record indication (record leak for tester only).

(5) Connect tester to static pressure air port.

(6) One minute after applied pressure, the differential pressure between indication after one minute and indication recorded per Para. (4) should be less than 20 ft.

(7) After completion of test, restore airplane to original condition.

3.3 OPERATIONAL CHECK OF STATIC PRESSURE LINE

(1) Check airspeed indicator, altimeter, rate-of-climb indicator and cabin differential pressure indicator connecting to static pressure lines.

(2) Cover static pressure air port with tape.

(3) Cap static pressure line of tester MB-1, apply 12,000 ft. (*1), 13,800 ft. (*2) of negative pressure to tester and maintain for one minute, then record indication.

(4) Connect tester to drain port.

(5) Apply 12,000 ft. (*1), 13,800 ft. (*2) of negative pressure with tester MB-1. At the same time, to the pitot connecting port of airspeed indicator apply negative pressure lest the pointer should deflect over the full scale. Check altitude indicator and cabin differential pressure indicator indicating about 12,000 ft. (*1), 13,800 ft. (*2).

(6) One minute after applied pressure, the differential pressure between indication after one minute and indication recorded per Para. (3) should be less than 240 ft. (*1), 276 ft. (*2).

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

3.4 LEAK TEST OF PITOT LINE

(1) Disconnect pitot line to airspeed indicator and cap the line so that pitot pressure will not be supplied to airspeed indicator.

(2) Connect air source to pitot tube and supply air regulated to about 2.5 psi.

(3) Remove cap and check that air flows out and there is no blockage in line. After check, cap again.

(4) Cap pitot line of tester MB-1, apply pressure of 300 kt to pitot line and maintain for one minute, then record indication (record leak for tester only).

(5) Disconnect air source from pitot tube and connect tester MB-1.

(6) Apply pressure of 300 kt with tester and maintain for one minute. One minute later, indication should be the value recorded per Para. (4) plus or minus 1 kt.

(7) After completion of test, restore airplane to original condition.

3.5 OPERATIONAL CHECK OF PITOT LINE

(1) Make sure airspeed indicator is connected to pitot tube.

(2) Cap pitot line of tester MB-1 and apply pressure of 300 kt to pitot line, maintain for one minute, then record indication.

(3) Connect tester MB-1 to pitot tube.

(4) Apply pressure of 300 kt with tester and make sure airspeed indicator indicates about 300 kt. One minute later, indication should be the value recorded per Para. (2) +0/-5 kts.

4. FLIGHT INSTRUMENTS

4.1 ALTIMETER

The altimeter is a 3 inch dial, flange-mount type instrument, connected to static pressure line, and indicates pressure in terms of altitude corresponding to atmospheric pressure change due to flight altitude change.

The instrument has 3 needles; the long needle indicates hundreds of feet, middle needle thousands of feet and short needle ten thousands of feet. Range of indication is 0 to 50,000 feet and atmospheric pressure setting range is 28.1 to 31.0 in-Hg (951 to 1,050 mb). The atmospheric pressure setting is adjustable by turning a knob located in one corner of the indicator.

When atmospheric pressure is set at sea level atmospheric pressure, height from sea level is indicated. In order to know height above a certain airfield, set altimeter at the airfield atmospheric pressure. When the needle is set to 0, the pressure indication indicates atmospheric pressure of that place.

4.1.1 TEST FOR SCALE ERROR AT ROOM TEMPERATURE

Test for scale error at room temperature is to be made at atmospheric pressure approximately 29.92 in-Hg and at room temperature approximately 77°F (25°C). Vibration condition of the test is at a frequency of 1500 to 2000 cpm and at amplitude of 0.002 to 0.005 in. The barometric pressure scale shall be set at 29.92 in-Hg and the reading of the pointers shall be taken. The error of this indication shall be determined by comparison with a barometer. This error will be used later in the after effect test. The reduction in pressure shall be made at a rate not in excess of 20,000 ft. per minute to less than about 2,000 ft. of the test point. The test point shall be approached at a rate compatible with the test equipment. The altimeter shall be kept at the pressure corresponding to each test point for at least 1 minute, but not more than 10 minutes before a reading is taken. The error at all test points must not exceed the tolerances specified in the table.

ALTITUDE (feet)	PRESSURE (in-Hg)	TOLERANCE ± (feet)
-1,000	31.018	20
0	29.921	20
500	29.385	20
1,000	28.856	20
1,500	28.335	25
2,000	27.821	30
3,000	26.817	30
4,000	25.842	35
6,000	23.978	40
8,000	22.225	60
10,000	20.577	80
12,000	19.029	90
14,000	17.577	100
16,000	16.216	110
18,000	14.942	120
20,000	13.750	130
22,000	12.636	140
25,000	11.104	155
30,000	8.885	180

4.1.2 HYSTERESIS TEST

Not more than 15 minutes after the altimeter's initial exposure to the pressure corresponding to 30,000 feet in the last step of the scale error test, the pressure shall be increased until the pressure corresponding to 16,000 feet (first test point) is reached. The altimeter shall be kept at this pressure for at least 5 minutes, but not more than 15 minutes, before the test reading is taken. After the reading has been taken, the pressure shall be increased further until the pressure corresponding to 12,000 ft. (second test point) is reached. The altimeter shall be kept at this pressure for at least 1 minute, but not more than 10 minutes, before the test reading is taken. After the reading has been taken, the pressure shall be increased further until atmospheric pressure is reached. In above cases, pressure shall be increased at a rate simulating a descent in altitude at the rate of 5,000 to 20,000 ft. per minute until within 3,000 ft. of the test point and then the test point shall be approached at a rate of approximately 3,000 ft. per minute. The reading of the altimeter readings for 16,000 ft. and 12,000 ft. recorded during the scale error test prescribed in Para. 4.1.1.

4.1.3 AFTER EFFECT TEST

Not more than 5 minutes after the completion of the hysteresis test, the reading of the altimeter (corrected for any change in atmospheric pressure) shall not differ by more than 30 ft. from the original atmospheric pressure reading recorded in the first step of the scale error test.

4.1.4 FRICTION TEST

The friction test shall be made at each altitude listed in the following table. The altimeter shall be subjected to a steady rate of decrease of pressure approximately 750 ft. per minute, then kept at the pressure corresponding to each test point while two readings are taken: the first before the altimeter is tapped, the second after tapping. The difference of any two such readings shall be recorded as friction and shall not exceed the tolerances listed in the table.

ALTITUDE (feet)	TOLERANCE ± (feet)
1,000	70
2,000	70
3,000	70
5,000	70
10,000	80
15,000	90
20,000	100
25,000	120
30,000	140

4.1.5 LEAK TEST

When the altimeter is kept to the pressure corresponding to an altitude of 18,000 ft., the reading shall not change by more than 100 ft. within one minute.

4.1.6 TEST FOR POSITION ERROR

With atmospheric pressure applied to the altimeter, the difference between the pointer indications in normal operating position and in any other positions shall not exceed 20 ft.

4.1.7 TEST FOR BAROMETRIC SCALE ERROR

At constant atmospheric pressure, the barometric pressure scale is set at each of the pressures listed in the following table. The altimeter shall indicate the equivalent altitude difference shown in the table with a tolerance of 25 ft.

PRESSURE (in-Hg)	ALTITUDE DIFFERENCE (feet)
28.10	1,728
28.50	1,340
29.00	863
29.50	392
29.92	0
30.50	531
30.90	893
30.99	974

4.2 AIRSPEED INDICATOR

The airspeed indicator is a 3 inch dial, flange-mount type instrument. Full pressure from the pitot tube is led to the inside of a diaphragm capsule and static pressure from the static tube is led to the outside of the diaphragm capsule. The indicator indicates airspeed in knots by means of transmitting different pressure between inside and outside of the diaphragm capsule to the pointer. Range of indication is 40 to 400 kts.

4.2.1 TEST FOR SCALE ERROR AT NORMAL ROOM TEMPERATURE

Test for scale error at normal room temperature is accomplished by applying pressure corresponding to airspeed specified in the following table, to pitot connecting port of indicator, first with pressures increasing, then with pressures decreasing. The errors should not exceed those specified in the table. The test is to be made at atmospheric pressure approximately 29.92 in-Hg and at room temperature approximately 77°F (25°C).

SPEED (Knots)	PRESSURE (in-H_2O)	ALLOWABLE ERRORS (Knots)
50	1.634	± 4
60	2.354	± 4
100	6.563	± 4
140	12.94	± 10
160	16.95	± 10
200	26.71	± 10
250	42.27	± 10
300	61.82	± 10
350	85.70	± 10
400	114.33	± 10

4.2.2 INSPECTION FOR FRICTION

Inspection for friction is performed by applying pressure specified in the following table, to pitot connecting port of the indicator. The pressure should be increased to bring the pointer approximately to the desired reading, and then held constant while two readings are taken, the first before the indicator is tapped, the second after. The difference between any two such readings should not exceed the values specified in the table. The pointer should move smoothly while the pressure is varied uniformly without tapping.

SPEED (Knots)	PRESSURE (in-H_2O)	ALLOWABLE ERROR (Knots)
50	1.634	4.8
100	6.563	4.8
160	16.95	4.8
250	42.27	4.8
350	85.70	4.8

4.2.3 INSPECTION FOR POINTER ZERO POSITION ERROR

When pitot connection of indicator is opened to atmosphere, zero position of the pointer should be within 1/16 in. (1.6 mm).

4.2.4 LEAK TEST

Pitot and static pressure connections of the indicator are jointed and connected together to sources of suction and pressure. A suction of 15 in-Hg is applied to pitot and static pressure connections. With the source disconnected for 10 seconds, the pressure should not change by more than 0.4 in-Hg. Then, the static connection is opened to the atmosphere and a pressure sufficient to produce approximately a full scale deflection of the pointer is applied to the pitot connection. During a period of 1 minute after disconnecting the pressure source, the pointer should not change its position by more than 1 kt.

4.3 RATE-OF-CLIMB INDICATOR

The rate-of-climb indicator is a 3 inch dial, flange-mount type instrument. It senses change of atmospheric pressure during climbing or descending through static pressure lines and indicates change in terms of speed (ft/min) in vertical direction. Range of indication is 0 to 6,000 ft/min.

4.3.1 TEST FOR SCALE ERROR AT ROOM TEMPERATURE

Test for scale error at room temperature is performed by giving a rate of climb at each altitude interval specified in the following table. Scale errors should not exceed the allowable value in the table. The test should be conducted at atmospheric pressure approximately 29.92 in-Hg and at room temperature approximately 77°F (25°C).

When the test is made with atmospheric pressure or room temperature differing materially from the above values, proper allowance should be made for the difference from the specified condition. Vibration condition of the test is at a frequency of 1500 to 2000 cpm and at amplitude of 0.002 to 0.005 inch.

STD ALTITUDE (ft)	TEST RATE (ft/min)	ALLOWABLE ERROR (ft/min)
2,000 ~ 2,500	500	100
2,000 ~ 3,000	1000	200
2,000 ~ 4,000	2000	300
2,000 ~ 5,000	3000	300
2,000 ~ 6,000	4000	400
2,000 ~ 7,000	5000	500
15,000 ~ 17,000	2000	300
15,000 ~ 17,000	4000	400
28,000 ~ 30,000	2000	300
28,000 ~ 30,000	4000	400

4.3.2 ZERO SETTING TEST

When the adjusting screw is moved throughout its range in both UP and DOWN directions, adjustable range of the pointer movement should be not less than 400 ft/min. for each direction.

4.3.3 POINTER LAG TEST

Check that the static pressure connection of the indicator is connected to the vacuum source. Vacuum should be carefully applied to the static pressure connection sufficient to make the pointer indicate a descent of 2,000 ft/min, after the pointer has passed through zero position when this vacuum is instantaneously released by opening the static pressure tubing to the atmosphere. Then, the static pressure connection is opened to the atmosphere, and the time in which the pointer moves from 2,000 ft/min. graduation to 200 ft/min. graduation should be 3 to 15 seconds.

4.3.4 LEAK TEST

Check that the static pressure connection of the indicator is connected to the vacuum source, and vacuum of 15 in-Hg is applied to the static pressure connection. With the source closed off, pressure depression during a period of 1 minute should not exceed 0.05 in-Hg.

NOTE

When vacuum or pressure is given to the static connection of the indicator, it should be done at a rate not exceeding 20,000 ft/min.

4.3.5 TEST FOR POSITION ERROR

With atmospheric pressure applied, readings of the indicator should be taken in each of several different positions, inclining 90° forward or backward and to left or right. The change in indication should not exceed 50 ft/min.

4.4 ATTITUDE GYRO

The attitude gyro is a 3 inch dial, flange-mount type, and air driven instrument or electric instrument. The air driven instrument is operated by vacuum pressure (See Para. 2) and the electric instrument by 400 Hz 115 VAC from the inverter in the airplane. The attitude gyro indicates bank and pitch attitude of the airplane. The horizontal bar which can be seen through the indicator front glass is attached to a gyro mechanism and indicates the horizon. The horizontal bar indicates longitudinal attitude and it moves down when the airplane nose goes up, and up when the nose goes down. A setting knob is provided at the lower side of the case and the pilot may adjust the pointer up and down which is the base of longitudinal attitude. The electric instrument is provided with a manual quick erect knob and a power-warning indicator flag.

4.5 TURN & BANK INDICATOR

The turn and bank indicator is a 3 inch dial flange-mount type instrument which incorporates a turn indicating pointer actuated by electrical gyro powered by 28 VDC and a bank indicator which contains damping fluid and black ball in a glass tube. The turn indicating pointer is calibrated to a two minute turn. The indicator is installed so that pointer may be placed in vertical position with aircraft in level attitude.

4.6 CLOCK

The clock is a 2 inch dial, flange-mount type instrument which permits reading seconds. This is an 8-day spring-wound clock.

4.7 MAGNETIC COMPASS

The magnetic compass is installed in the center of the windshield center post and indicates magnetic heading of the airplane. Compensation of indication is possible by means of a compensator contained in the compass. Indication errors are recorded on the compass card located immediately above the instrument. An illuminating light on 28 VDC power source is installed inside. The magnetic compass must be calibrated usually every three months.

4.7.1 CALIBRATION OF MAGNETIC COMPASS

Compensate the magnetic compass on the ground in accordance with the following procedures.

(1) Place aircraft on compass rose, ready to be oriented to magnetic heading.

NOTE

Do not place any magnetic masses near aircraft.

(2) Check compensator of magnetic compass for emplacement in neutral position. If compensator is not placed in neutral position, replace it in proper position.

(3) Head the aircraft to magnetic S, note compass reading and determine deviation.

(4) Turn aircraft to W, N, and E in succession and repeat procedures in Para. (3) above.

(5) Calculate coefficients C, B and A from deviation of slaving meter obtained at every heading of aircraft.

$$C = \frac{\alpha n - \alpha s}{2}$$

$$B = \frac{\alpha e - \alpha w}{2}$$

$$A = \frac{\alpha n + \alpha e + \alpha s + \alpha W}{4}$$

Where αn means deviation of slaving meter at heading to N of aircraft.
αe means deviation of slaving meter at heading to E of aircraft.
αs means deviation of slaving meter at heading to S of aircraft.
αw means deviation of slaving meter at heading to W of aircraft.

Examples

Magnetic Heading	Gyro Indications	Deviations
N	006° 1/2	- 6° 1/2
E	090°	0°
S	175° 1/2	+ 4° 1/2
W	276°	- 6°

(6) With aircraft headed to N, compensate for reading of compass by coefficient C.

$$C = \frac{\alpha n\ (-6^{\circ}1/2) - \alpha s\ (+4^{\circ}1/2)}{2} = -5^{\circ}\ 1/2$$

By means of compensator N - S, compensate for coefficient C (-5° 1/2) so that compass may read 001°.

(7) With aircraft headed to E, compensate for reading of compass by coefficient B.

$$B = \frac{\alpha e\ (0^{\circ}) - \alpha w\ (-6^{\circ})}{2} = + 3^{\circ}$$

By means of compensator E - W, compensate for coefficient B (- 3°) so that compass may read 093°.

(8) With aircraft headed to W, turn magnetic compass by (-2°), coefficient A. Reinstall magnetic compass.

(9) Starting from E, change heading of aircraft by 30° in succession. Record every compass reading on compass card. The difference between maximum plus deviation and maximum minus deviation in final compass swing on ground must not exceed 6°.

(10) Ensure no change in indication of compass when instruments and radio are operated, or engine started with power source connected to aircraft on the ground.

(11) After calibration of compass is completed, make compass calibration card, with notation stating radio ON or OFF.

4.8 ELECTRIC COMPASS CALIBRATION

4.8.1 COMPASS SWING CALIBRATION

Perform the compass swing calibration for the compass system as described in the following procedures.

(1) Energize the compass system and allow several minutes for the gyro to reach operating speed and for the system to slave to the magnetic heading.

NOTE

The aircraft should be in its normal flight position with the electrical system and radio equipment operating.

(2) Position the aircraft on a compass rose and turn it to each of the four cardinal headings, ±°.

(3) Allowing sufficient time for the heading indicator to settle, record the differences in readings between the heading indicator and the compass rose at each cardinal heading as plus or minus, depending on whether the dial readings are greater or less than the compass rose readings.

NOTE

Instead of the compass rose, a magnetic sighting compass may be used. To take a reading, the compass is located at a considerable distance fwd and aft of the aircraft and is moved back and forth from the line of sight coinciding with the plane. When a sight is taken facing aft, 180° must be added or subtracted from the sighting compass reading. When facing fwd, the compass reads directly.

(4) Add the errors algebraically and divide by four. The result is the index error.

(5) Loosen screws holding flux detector to its mounting surface and rotate. The amount of rotation should equal the index error.
   (a) If error is positive, rotate counterclockwise, indicating a minus reading on index as observed from top of unit.
   (b) If error is negative, rotate clockwise, indicating a plus reading on index as observed from top of unit.

(6) Tighten the mounting screws and recheck the readings at the four cardinal headings. Recalculate the index error to make sure it is zero.

(7) If the index error is not zero, readjust the flux detector flange until this error is cancelled.

(8) Any remaining errors in excess of $\pm 1^{\circ}$ which are caused by extraneous magnetic fields should be counteracted by using the compensator (if installed).

4.8.2 ADJUST DRC-1 TO COMPENSATE FOR SINGLE-CYCLE ERRORS (HARD IRON EFFECTS) AS DESCRIBED IN THE FOLLOWING PROCEDURES.

(1) Remove the DRC-1 cover.

(2) Make sure compensation potentiometers are in their center position.

(3) Using a compass rose, place the aircraft on a North heading $\pm 1^{\circ}$ and allow compass dial or Radio Deviation Indicator (RDI) to settle.

(4) Compensate for any difference between actual heading and RDI indication by loosening the locking nut and adjusting N – S (North-South) potentiometer on DRC-1 (tighten locking nut).

(5) Place the aircraft on an East heading $\pm 1^{\circ}$ and allow the compass dial to settle.

(6) Compensate for any difference between actual heading and RDI indication by loosening lock nut and adjusting E - W (East-West) potentiometer on DRC-1 (tighten locking nut).

(7) Place the aircraft on a South heading $\pm 1^{\circ}$ and allow compass dial to settle.

(8) Compensate for half of any difference between actual heading and RDI indication by loosening locking nut and adjusting N – S potentiometer on DRC-1 (tighten locking nut).

(9) Place the aircraft on a West heading $\pm 1^{\circ}$ and allow compass dial to settle.

(10) Compensate for half of any difference between actual heading and RDI indication by loosening locking nut and adjusting E – W potentiometer on DRC-1 (tighten locking nut).

(11) The DRC-1 should now be fully adjusted for proper compensation. As a check, swing the aircraft on 30° increments and note readings on compass dial. All readings should be within $\pm 1^{\circ}$ of the actual heading. If errors are greater than $\pm 1^{\circ}$, repeat index error adjustment of paragraph 15.1 and the above adjustments for greater accuracy.

4.8.3 CONFIRMATION

Stop the aircraft at taxiways parallel to and at a right angle to the runway and confirm the compass indications. The compasses should not be split by more than 2° at any point.

4.9 AILERON-TRIM POSITION INDICATOR (Fig. 10-14)

The aileron-trim position indicator is installed on the lower center section of the center pedestal and is an ammeter which works in a bridge circuit consisting of a transmitter and adjusting resistance installed on the forward back side of the center pedestal. The electric power source is 28 VDC. When indicator or transmitter has been replaced, adjustment should be accomplished in accordance with Para. 8.4.3, Chapter V.

Fig. 10-14 Aileron-trim position ind.

4.10 TROUBLE SHOOTING

4.10.1 ALTIMETER, AIRSPEED INDICATOR AND RATE-OF-CLIMB INDICATOR

Trouble	Probable cause	Remedy
Instrument indication is inaccurate, instrument does not work or needle fluctuates	a. Static pressure line is blocked.	Drain static pressure line and clean line. **CAUTION** Instruments must be disconnected before applying air to clean lines.
	b. Static pressure line is leaking.	Perform leak test and repair defective component.
	c. Instrument defective.	Replace instrument.
Rate of climb indicator does not indicate zero in level flight	a. Calibration of instrument is faulty.	Turn adjusting screw located on L. H. lower section of instrument. Give a little vibration to instrument during adjustment.
Air speed indicator indication is inaccurate, instrument does not work or needle fluctuates	a. Pitot line is blocked.	Drain static pressure line and clean. **CAUTION** Never use air to clean lines without disconnecting lines from instruments.
	b. Pitot line is leaking.	Perform leak test and repair defective component.
	c. Instrument faulty.	Replace instrument.

4.10.2 ATTITUDE GYRO

Trouble	Probable cause	Remedy
Indication of attitude gyro is inaccurate or horizontal bar drifts	a. Blocked filter for vacuum system. (air driven type)	Replace filter.
	b. Instrument faulty.	Replace instrument.
	c. Operating range of gyro is exceeded.	Place airplane in level position and check.

4.10.3 TURN AND BANK INDICATOR AND MAGNETIC COMPASS

Trouble	Probable cause	Remedy
Needle of turn indicator does not return	a. Balance of gyro faulty. Zero adjustment erroneous.	Replace instrument.
Needle of turn indicator fluctuates	a. Adjustment of instrument damper faulty.	Replace instrument.
Needle of turn indicator inoperative	a. Poor contact of wiring connector or wire broken.	Perform conductivity check and repair.
Indication of magnetic compass is too erratic	a. Compass out of adjustment.	Perform compass swing and adjust.
	b. Defective compass.	Replace compass.
Fluid in magnetic compass is discolored or leaking	a. Deterioration of fluid and crack in instrument.	Replace compass.

5. FUEL QUANTITY INDICATOR

5.1 DESCRIPTION

Capacitor type fuel quantity indicators are provided. With several cylindrical tank units dipped in fuel, a change in fuel level and specific weight of fuel are indicated as compensated by electrical capacity on the basis of different rates of induction of fuel and air. Fuel quantity indicators include wing tank fuel quantity indicator and tip tank fuel quantity indicator.

5.2 WING TANK FUEL QUANTITY INDICATOR

The wing tank fuel quantity indicator system consists of a 2 inch dial clamp mount type instrument installed on the center switch panel, capacitor type tank units, each one in the center tank, LH and RH outer tanks and a test switch for operational check on the center switch panel. Capacitor indication is given in terms of GAL. When residual fuel is decreased until it is sufficient only for 30 minutes flight (30 ± 5 gal.), FUEL LOW LEVEL warning light illuminates. The amplifier contained in the fuel quantity indicator converts an input signal from the tank unit into U.S. gallons. The fuel quantity indicator system receives power supply of 115 VAC 400 Hz from the inverter. The electrical capacity of the tank unit is compared with that of a reference condenser in re-balancing the type bridge circuit through the medium of change in fuel level or quantity in the tank. An unbalancing signal, amplified by a transistor voltage amplifier, energizes the transistor of the phase discriminator output. The output voltage is transmitted to one of the 2 phase AC motors, and mechanically actuates the rebalancing potentiometer and needle of the fuel quantity indicator. For the other phase of the AC motor, power of 115 VAC 400 Hz is transformed to 15 V by a transformer contained in the indicator and a rectified convertible two-phase induction motor-driven sliding part of the balance potentiometer, through the medium of a gear mechanism of a large reduction ratio, and keeps the bridge circuit balanced. A pointer attached on the same axis as that of the sliding part indicates fuel quantity in U.S. gallons. Components of the fuel quantity indicator are contained in a sealed case. Electrical connections are provided at the back of the instrument to set the indicator of the electrical capacity of the tank unit with fuel EMPTY and FULL.

5.3 TIP TANK FUEL QUANTITY INDICATOR

The tip tank fuel quantity indicator system consists of: a 2-inch dual clamp mount; dual pointer type instrument installed on the center switch panel; one each capacitor type tank units in the LH and RH tip tanks; an amplifier at the back of left instrument panel; and a test switch for operational check on the RH switch panel. Fuel quantities in both tip tanks are indicated by pointers marked L and R, respectively. Indication is given in terms of GAL. The amplifier contained in the fuel quantity indicator converts an input signal from the tank unit into U.S. gallons. Electrical capacity of the tank unit is compared with that of reference condenser in a re-balancing type bridge circuit through the medium of change in fuel level or quantity in the tank. The unbalancing signal, amplified by a transistor voltage amplifier, energizes the transistor of the phase discriminator output. The output voltage is transmitted to one of the 2-phase AC motors, and mechanically actuates the re-balancing potentiometer and needle of the fuel quantity indicator. For the other phase the AC motor, power of 115 VAC 400 Hz, is transformed to 15 V by a transformer contained in the indicator and rectified. Two pointers (L over R) are attached on the same axis of the dial. Each pointer is connected to a balancing potentiometer and convertible two-phase induction motor through the medium of gear mechanism of a large reduction ratio. Electrical connections are provided at a receptacle at the back of the instrument.

5.4 AMPLIFIER

The amplifier is provided at F.STA 1250 at the back of the LH switch panel. The amplifier consists of individual amplifiers for LH and RH tip tanks, bridge circuits and regulating screws. Incorporated in each assembly are amplifying circuits with phase discriminating function, transformer and reference condenser for the bridge circuit, EMPTY and FULL regulator, potentiometer and phase power source for the indicator motor. An electrical connection is provided at the receptacle located on one side of the container. On the other side of the container, a regulating screw is provided.

5.5 CENTER TANK UNIT

The tank unit consists of 2 concentric electric poles. The poles are insulated by a plastic spacer interfaced for the entire length of the poles. The induction ratio between 2 poles of the tank unit changes, depending upon fuel level in the tank. The tank unit is clamped to the inner wall by 4 bolts. On top of the unit, 4 connectors are provided.

5.6 OUTER TANK UNITS

The outer tank units are the same as the center tank unit in construction, function and installation, but smaller in length.

5.7 TIP TANK UNITS

Tip tank units are the same as the center tank unit in construction, function and installation. On top of the units, 2 connectors are provided.

5.8 TEST OF INDICATOR

5.8.1 WING TANK FUEL QUANTITY INDICATOR

(1) Connect the indicator as shown in Fig. 10-15, select the condenser capacity per the following table, and read the indicator.

(2) When indicator reads 30 ± 5 gallons, check pointer deflection.

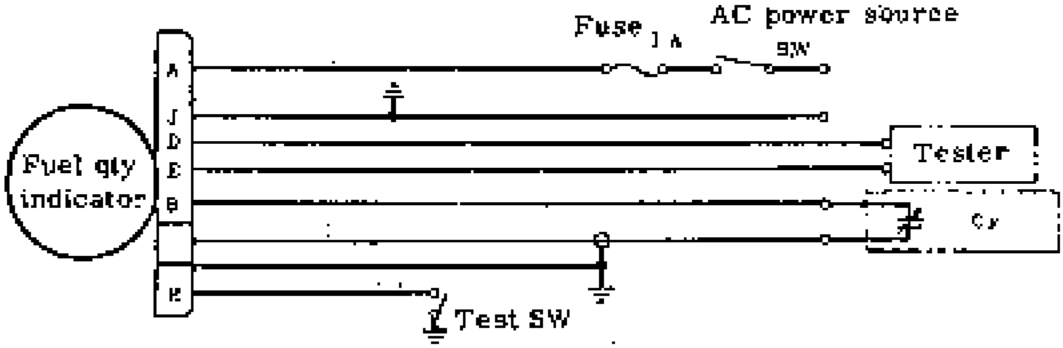


Fig. 10-15 Wing tank fuel qty. ind.

Indicator Reading (GAL)	Static Electric Capacity of CX (PF)
E (0)	129.3
20	143.0
60	178.3
90	204.7
120	231.1
150	257.5
F (154)	260.5

5.8.2 TIP TANK FUEL QUANTITY INDICATOR

Connect the indicator as shown in Fig. 10-16, select the condenser capacity per the following table, and read the indicator.

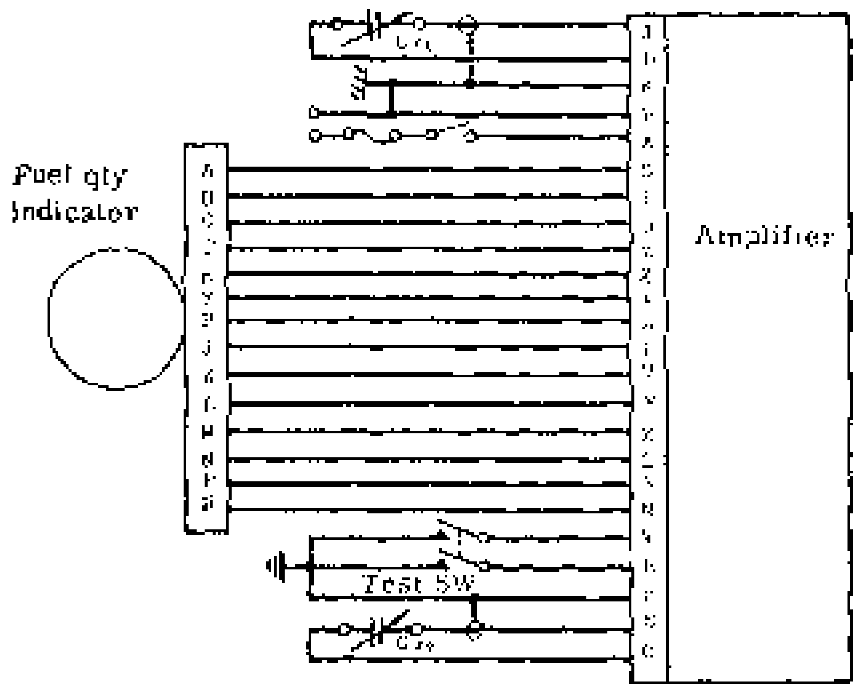


Fig. 10-16 Tip tank fuel qty. ind.

Indicator Reading (GAL)	Static Electric Capacity (PF)	
	Cx1 (Right)	Cx2 (Left)
E (0)	55.0	55.0
15	64.4	64.4
30	73.8	73.8
45	83.3	83.3
60	92.7	92.7
75	102.1	102.1
F (90)	111.5	111.5

5.9 ADJUSTMENT OF MAIN TANK FUEL QUANTITY INDICATING SYSTEM

5.9.1 ADJUSTMENT AT "EMPTY" POINT

(1) Level the airplane and drain fuel through drain valve to keep the tank unit dry. (Undrainable fuel 1.0 U.S. gallons remains in the tank.)

(2) Supply fuel 4.0 U.S. gallons into the fuel tank.

(3) Turn Battery Key switch, circuit breaker (FUEL QTY IND) and inverter switch to ON.

(4) Ascertain that fuel quantity indicator indicates "E" point under the above condition. If the indicator does not show "E" point, adjust by the "E" adjusting screw on the back of the indicator so that the indicator may read "E" point.

5.9.2 ADJUSTMENT AT "FULL" POINT

(1) Fill up the tank after completing adjustment at "E" point.

(2) Ascertain that fuel quantity indicator indicates "F" point when the tank is filled up. If the indicator does not show "F" point, adjust by the "F" adjusting screw on the back of the indicator so that the indicator may read "F" point.

5.10 ADJUSTMENT OF TIP TANK FUEL QUANTITY INDICATING SYSTEM

5.10.1 ADJUSTMENT AT "EMPTY" POINT

(1) Level the airplane and drain fuel through drain valve to keep the tank unit dry.

(2) Turn Battery Key switch, circuit breaker (TIP FUEL QTY IND) and inverter switch to ON.

(3) Ascertain that fuel quantity indicator indicates "E" point. If the indicator does not show "E" point, adjust by the "E" adjusting screw on the side of the amplifier so that the indicator may read "E" point.

5.10.2 ADJUSTMENT AT "FULL" POINT

(1) Fill up the tank after completing adjustment at "E" point.

(2) Ascertain that fuel quantity indicator indicates "F" point when the tank is filled up. If the indicator does not show "F" point, adjust by the "F" adjusting screw on the side of the amplifier so that the indicator may read "F" point.

5.11 OPERATIONAL CHECK OF FUEL QUANTITY INDICATOR

Operational check of the fuel quantity indicators should be performed following the adjustment of the "F" point indication. To test the fuel quantity indicators, place the FUEL QTY TEST switch to the MAIN position and confirm the main fuel quantity indicator needle moves smoothly toward EMPTY (zero) direction. Place the test switch to the TIP position and confirm the tip fuel quantity indicator needles move smoothly toward EMPTY (zero) direction.

5.12 CHECK OF LOW LEVEL WARNING LIGHT

Ascertain that the warning light "FUEL LOW LEVEL" on the annunciator panel illuminates when the fuel quantity indicator indicates 30 ± 5 gallons while in check per Para. 5.11.

5.13 TROUBLE SHOOTING

Trouble	Probable cause	Remedy
Indication of fuel quantity indicator is inaccurate	a. System adjustment faulty.	Readjust system.
	b. Tank unit defective.	Check tank unit.
	c. Indicator faulty.	Replace indicator.
Needle of indicator is stuck	a. Friction error of indicator excessive.	Replace indicator.
Needle of indicator fluctuates during flight	a. Faulty Amplifier.	Check and repair amplifier.

6. ENGINE INSTRUMENTS

The engine instruments are 2-inch dial clamp mount type instruments and two sets are installed on the center instrument panel for LH and RH engines.

6.1 TORQUEMETER

The torquemeter indicates output torque of the engine and is actuated by a signal from the torque transducer. The torque transducer senses the difference between pressure of the torque sensor of the engine and of engine reduction gearbox. The torquemeter is a potentiometer type and is powered by 28VDC from airplane power. Scale is 0 to 120% torque.

Check torquemeter for indication of 100% in power off condition. If the torquemeter does not point to 100%, replace.

6.1.1 TORQUEMETER INSPECTION

(1) Check for errors in torque indicator.

(a) Connect the indicator as shown in Fig 10-16A. Note that torquemeters (PN 935A-5001) are connected directly a variable DC power source.

(b) Adjust the variable DC voltage source so that the torquemeter indicates the check point reading shown in Table 10-1. Read the indication on the volt meter. (See Table 10-1)

(c) Insure that the indication shown in Table 10-1 on the volt meter does not exceed the allowable error.

(d) The torquemeter should be repaired or replaced if the values are not within the allowable error per Table 10-1.

Table 10-1

Torquemeter check point (% torque)	Volt meter nominal voltage (VDC)	Allowable error (VDC)
0	3.96	±0.095
50	1.98	±0.095
100	0	±0.047
110	-0.396	±0.095

(2) Check torque transducer

(a) Connect the torque transducer as shown in Fig 10-16B.

(b) The engine DSC (Data Sheet Customer)curve is a straight line with the slope determined by the three points (0 in-lb, 3,000 in-lb and 21,000 in-lb (for MU-2B-35) or 28,020 in-lb (for MU-2B-36A))

From this curve determine the ΔP corresponding to the torque values listed in Table 10-2 for each engine and record the pressure values on a data sheet.

(c) Confirm that the output voltage is within the allowable error parameters.

(d) The torque transducer should be recalibrated if the torque transducer output-voltage is not within the allowable error per Table 10-2.

Table 10-2

For MU-2B-35

Torque load inch/pounds		ΔP (psid)	Torque transducer output-voltage (VDC)	Allowable error (VDC)
in-lb	%			
0	0.0		3.960	±0.1
3,000	14.3		3.394	±0.1
21,000	100.0		0.00	±0.03

For MU-2B-36A

Torque load inch/pounds		ΔP (psid)	Torque transducer output-voltage (VDC)	Allowable error (VDC)
In-lb	%			
0	0.0		3.960	±0.1
3,000	10.6		3.541	±0.1
28,320	100.0		0.00	±0.03

(3) Torque transducer calibration

- For P/N 895369-1, -2, and 3101299-1

(a) Connect the torque transducer as shown in Fig 10-16B.

(b) Set ΔP corresponding to 21,000 in-lb (for MU-2B-35) or 28,320 in-lb (for MU-2B-36A) by adjusting the pressure source.

(c) Loosen the zero adjust lock screw on the front of the transducer. Using a spanner wrench in the two holes on the face, rotate the inner body of the torque transducer until 0.00± 0.03VDC output is observed. (Refer to Fig 10-16C)

NOTE

Clockwise rotation increases positive voltage.

(d) Set ΔP corresponding to 3,000 in-lb by adjusting the pressure source.

(e) Remove dust cover from span adjustment screw. Rotate span adjustment screw until an output voltage versus torque % is observed as 3.394±0.1volts (for MU-2B-35), 3.541±0.1volts (for MU-2B-36A) on the volt meter. (Refer to Fig 10-16C)

NOTE

Clockwise rotation increases positive voltage.

(f) Increase pressure of the pressure source to 100% torque and verify that a 0.00± 0.03VDC output is observed. A recheck of steps (b) to (e) may be required.

(g) Obtain ΔP output voltage corresponding to the torque listed in Table 10-2. By varying pressure of the pressure source, confirm that the output voltage is within the values described in Table 10-2.

(h) Upon completion of the adjustment, install the span adjustment dust cover finger tight. Tighten zero adjustment lock screw to 10 ~ 15 in-lbs and lockwire the cover to the zero adjustment lock screw. Apply sealant.

(i) Recheck the setting of torque transducer.

(j) Reinstall torquemeters and torque transducers in the airplane making sure that the proper transducer is mated to its matched engine.

- <u>For P/N 897311-46-5</u>

(a) Connect the torque transducer as shown in Figure 10-16B.

(b) Set ΔP corresponding to 21,000 in-lb (for MU-2B-35) or 25,020 in-lb (for MU-2B-36A) by adjusting the pressure source

(c) Remove the adjustment plug with a 3/16 inch Allen wrench to provide access to the span and zero adjustment screws. Rotate the zero screw (marked "Z") until 0.00± 0.00VDC output observed. (Refer to Figure 10-16C)

NOTE

Counterclockwise rotation increases positive voltage.

(d) Set ΔP corresponding to 3,000 in-lb by adjusting the pressure source.

(e) Rotate range adjust screw until an output voltage versus torque % is observed as 3.394 ±0.1volts (for MU-2B-35) or 3.541±0.1volts (for MU-2B-36A) on the volt meter. (Refer to Figure 10-16C)

NOTE

Counterclockwise rotation increases positive voltage

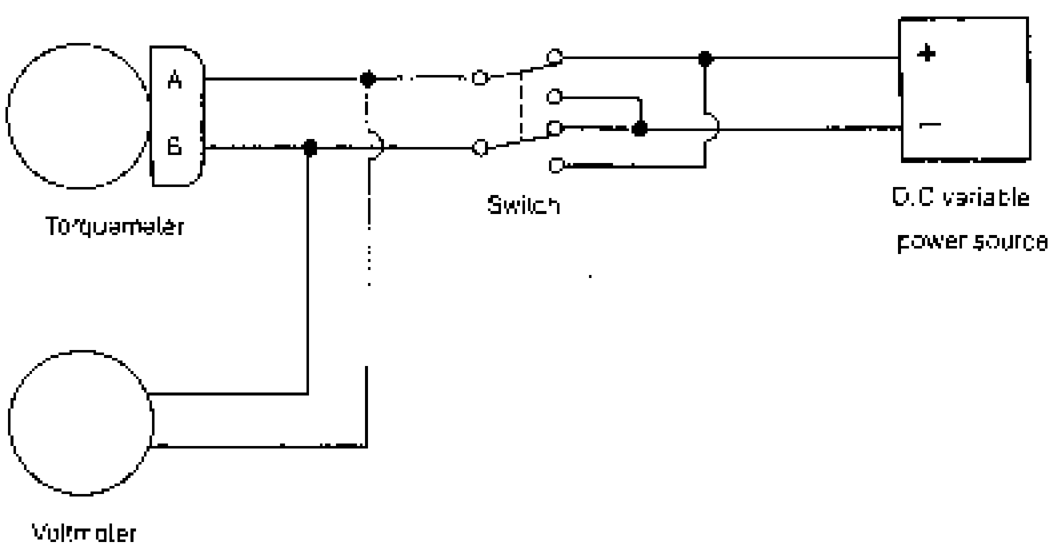


Fig 10-16A Test setup for torquemeter inspection

(f) Increase pressure of the pressure source to 100% torque and verify that a 0.00± 0.03VDC output is observed. A recheck of steps (b) to (e) may be required.

(g) Obtain ΔP output voltage corresponding to the torque listed in Table 10-2. By varying pressure on the pressure source, confirm that the output voltage with the values described in Table 10-2.

(h) Replace adjustment plug. Safety wire, as appropriate.

(i) Recheck the setting of torque transducer.

(j) Reinstall torquemeters and torque transducers in the airplane making sure that the proper transducer is mated to its matched engine.

6.1.2 Ground Check

With the torque transducers and torquemeters installed in the airplane conduct a ground run to verify that full power can be set and the engines are essentially symmetric.

(a) Start engines and verify no leakage from the torque transducers.

(b) Check engine output torque for the ambient pressure altitude and temperature with the Power Assurance Chart in the Performance Section of the Airplane Flight Manual.

(c) Record torque, RPM, EGT, and Fuel Flow for both engines at the takeoff power obtained from the AFM power assurance chart for the ambient conditions.

(d) Record the engine parameters at the ground idle setting.

(e) If the power assurance torque check is successful and the other engine parameters are reasonable, proceed to the next step. If the torque cannot be attained, and the reason for the low torque indication cannot be determined, the engine may have internal damage and require additional maintenance inspections before takeoff is attempted.

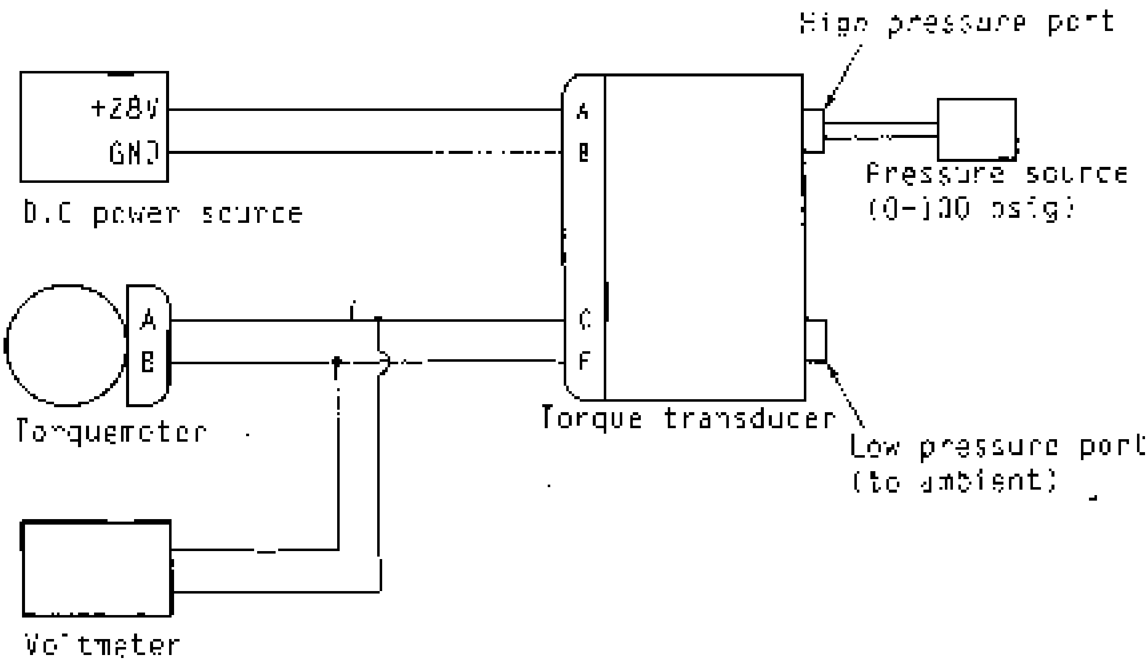
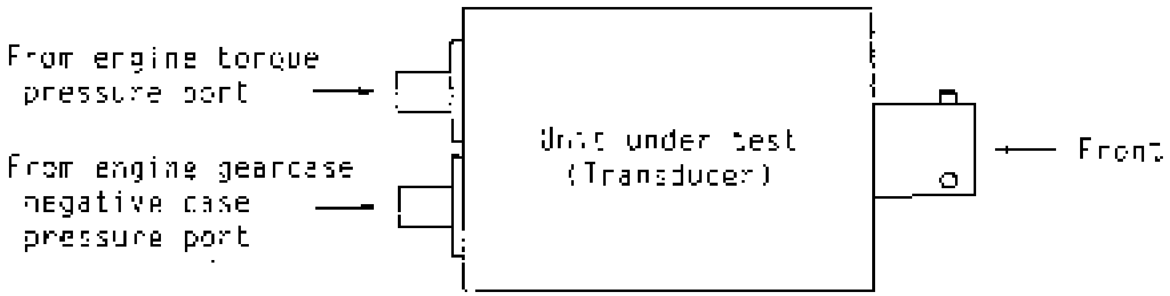


Fig 10-16B Test setup for torque transducer



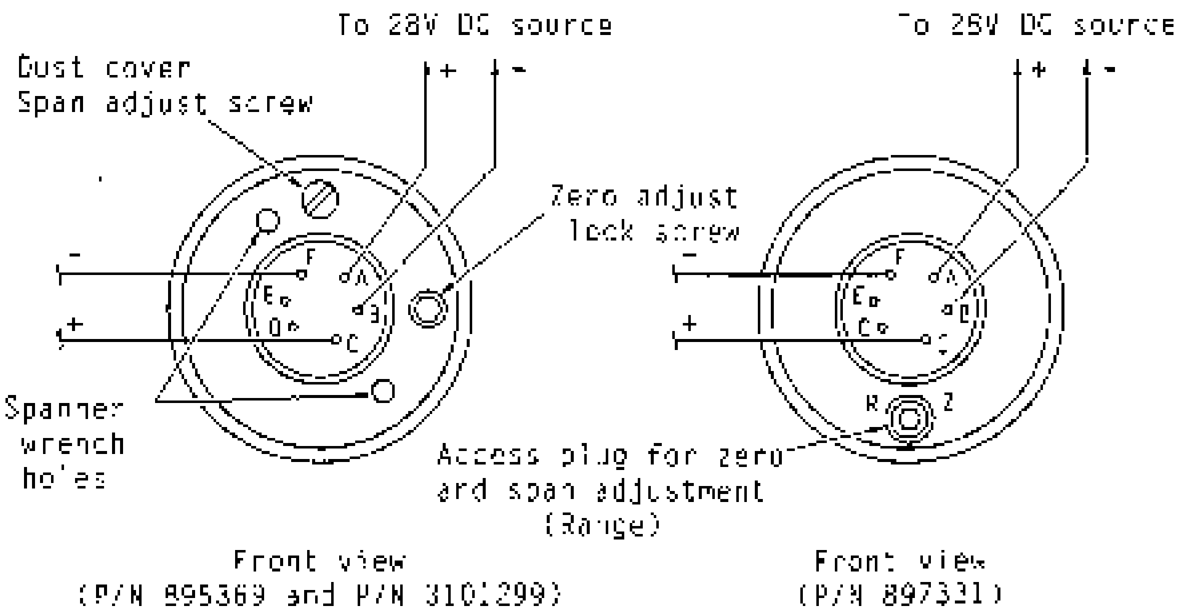


Fig 10-16C Location of adjustment on torque transducer

6.2.3 CHECKOUT OF POSITION ERROR

When the indicator is pitched or rolled 45°, the indication must not deviate any more than 0.3% RPM from that of the indication driven in normal attitude at 100% RPM.

6.2.4 CHECKOUT OF POINTER ALIGNMENT

For checking synchronized operation of main and auxiliary pointers, place the main pointer in 90% and 96% RPM positions, and the auxiliary pointer is supposed to read 0 and 6% RPM within tolerance of 0.2% RPM.

6.3 INTERSTAGE TURBINE TEMPERATURE INDICATOR

The interstage turbine temperature indicator is to measure temperature at engine interstage turbine and is actuated by thermoelectromotive force in connecting 12 thermocouples at 2nd stage stator of turbine and the indicator with chromelalumel wire.

6.3.1 CHECKOUT OF SCALE ERROR AT ROOM TEMPERATURE

Remove the indicator from the instrument panel and check as follows.

(1) Connect the indicator as shown in Fig. 10-17. Apply the voltage tabulated below and read the indication. Scale error must not exceed allowable error below.

Temperature (°C)	Input voltage corresponding to temp. (mV)	Allowable error (±°C)
100	4.10	10
200	8.13	5
300	12.21	5
400	16.40	5
500	20.65	5
600	24.91	5
700	29.14	5
800	33.30	5
900	37.36	5
1000	41.31	5
1100	45.16	5
1200	46.89	10

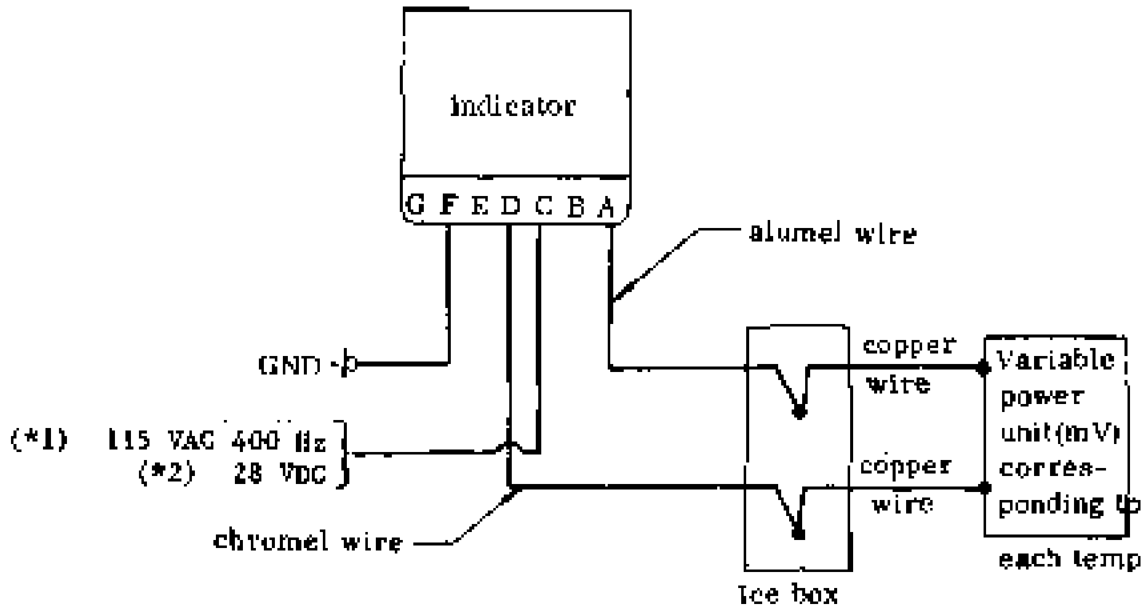


Fig. 10-17

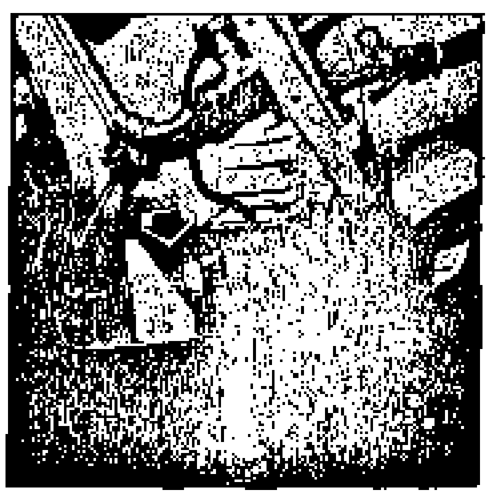
6.4 OIL PRESSURE INDICATOR

This system indicates pressure of lubricating oil. This instrument, a dual needle read-out, is a voltage ratio type instrument and is powered by 26 VAC 400 Hz power from the aircraft inverter. The indication range is 0 to 200 psi.

6.5 OIL TEMPERATURE INDICATOR (See Fig. 10-18)

This system indicates temperature of the lubricating oil and its sensor is attached to the reduction gear housing of the engine to a torque of 60 to 65 in-lbs. (69 to 74.3 kg/cm). The indicator is operated by the change in resistance in response to temperature change and is powered by 28 VDC. Range of indication is -70 to +150°C (-94 to +302°F).

Fig. 10-18 Installation of transmitter (sensor)



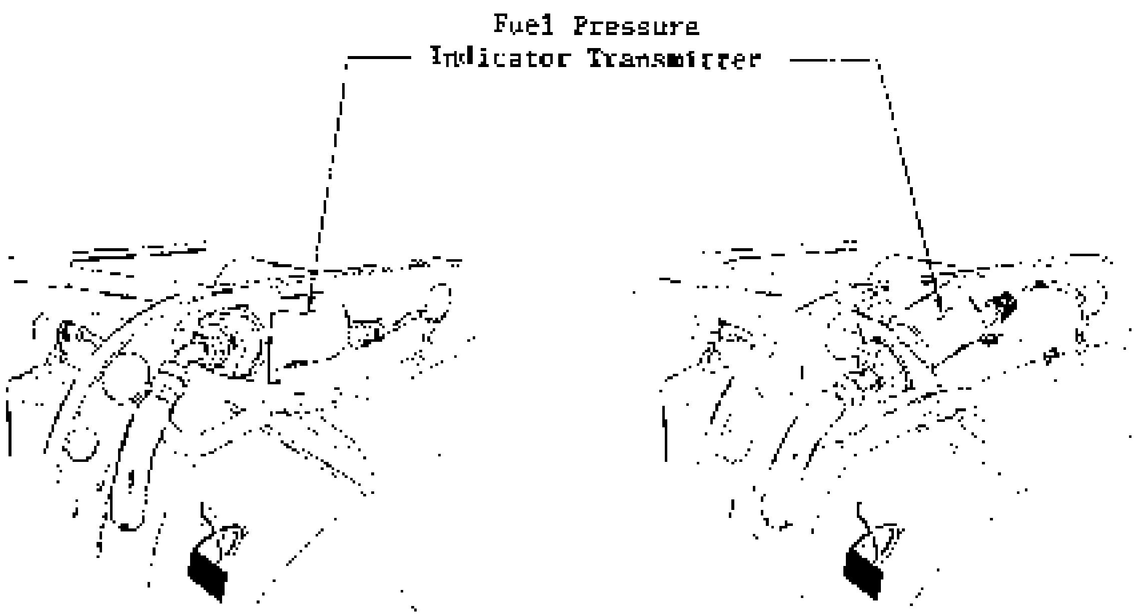
*1 Aircraft S/N 652SA unless modified by Kit Drawing K926A-8202
*2 Aircraft S/N 661SA, 697SA and subsequent, and aircraft modified by Kit Drawing K926A-8202

6.6 FUEL FLOW SYSTEM

The fuel flow system senses fuel flow supplied to each engine by a transmitter installed in the middle of fuel supply line within engine and through signal conditioning unit makes each fuel flow indicator and fuel consumption totalizer indicate fuel flow. The range of indication is a double type or 0 to 80 gal/hr. and 0 to 500 lb/hr. The system is powered by 28 VDC.

6.7 FUEL PRESSURE INDICATOR

The fuel pressure indicator indicates fuel pressure and the sensor is installed on the RH upper section of the engine reduction gearbox (Fig. 10-19). This indicator is voltage ratio type and is powered by 26 VAC 400 Hz from the inverter in the airplane. The range of indication is 0 to 100 psi.



Aircraft S/N 661SA, 697SA, 699SA, 701SA thru 704SA, unless modified by S/R 003/73-001

Aircraft S/N 652SA and 698SA unless modified by S/R 003/73-001

Fig. 10-19 Installation of fuel pressure indicator transmitter (1/2)

6.2 TACHOMETER

The tachometer indicates engine speed in percentage in response to output of the tachometer generator. The tachometer generator is installed on the rear section of the engine reduction gear box and rotates approximately 1/10 of engine speed. Scale of the instrument is 0 to110% and consists of a principal needle which indicates 0 to 100% and a small needle located in the LH upper section which indicates 0 to 10%. This system is not connected to the airplane electrical power source.

6.2.1 CHECK OUT OF SCALE ERROR AT ROOM TEMPERATURE

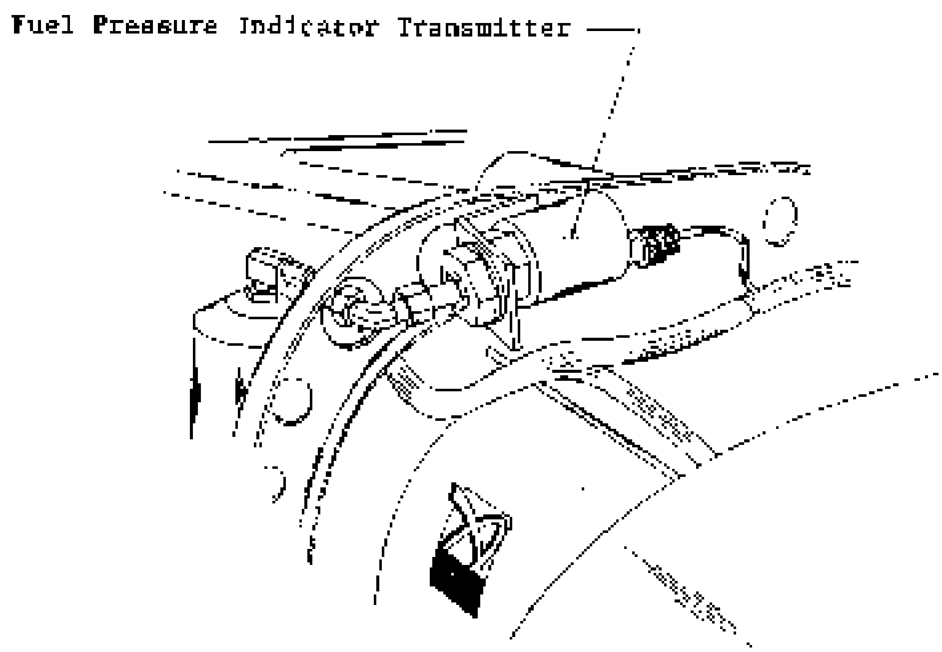
Scale error at room temperature determined under increased or reduced RPM must not exceed the tolerance tabulated below. The pointer oscillation shall not exceed 0.5% from zero to 15% RPM. and 0.3% from 15 to 110% RPM. Perform checkout under atmospheric pressure of about 29.92 in-Hg and ambient temperature of about 25°C.

Drive shaft (r.p.m.)	Indicator (% r.p.m.)	Allowable error (% r.p.m.)
0	0	± 0.5
210	5	± 0.5
420	10	± 0.5
630	15	± 0.5
840	20	± 0.5
1260	30	± 0.8
1680	40	± 0.8
2100	50	± 0.8
2520	60	± 0.8
2940	70	± 0.8
3150	75	± 0.5
3360	80	± 0.5
3570	85	± 0.5
3780	90	± 0.5
3990	95	± 0.5
4200	100	± 0.5
4410	105	± 0.5
4620	110	± 0.5

6.2.2 CHECKOUT OF FRICTION

When the indicator is driven in increased or reduced RPM, the tolerance due to friction must not exceed tolerance for each RPM tabulated below as check points. Allow the pointer to stabilize at a checking RPM without tapping the indicator, and read the indication. Then tap the indicator and record the indication. Note the difference between the two readings.

Indicator (% rpm)	Allowable Error (% rpm)
5	1.5
20	0.6
40	0.5
70	0.5
85	0.5
100	0.5



Aircraft S/N 705SA thru 730SA
and aircraft modified by S/R 003/73-001

Fig 10-19 Installation of fuel pressure indicator transmitter (2/2)

6.8 TROUBLE SHOOTING

Trouble shooting torquemeter, tachometer, oil temperature indicator, interstage turbine temperature indicator, oil pressure indicator, fuel flow indicator and fuel pressure indicator.

Trouble	Probable cause	Remedy
Instrument indication faulty or inoperative	a. Contact of wiring connector defective, short circuit of electrical wiring or broken wire.	Perform conductivity test and check connector and circuit breaker.
	b. Instrument defective.	Replace instrument.
Indication of torquemeter, oil pressure indicator or fuel pressure indicator is faulty	a. Sensor defective.	Adjust sensor or replace it.
Indication of tachometer is faulty	a. Tachometer generator faulty.	Check output and repair or replace it.
Indication of oil temperature indicator is faulty	a. Temperature sensor faulty.	Replace sensor.
Indication of interstage turbine temp. indicator is faulty	a. Defective engine thermocouple.	Replace thermocouple.
Indication of fuel flow ind. or totalizer is faulty	a. Transmitter defective.	Replace transmitter.
	b. Signal conditioning unit defective.	Replace signal conditioning unit.
	c. Instrument defective.	Replace instrument.

7. OTHER INSTRUMENTS

7.1 VOLTMETER-AMMETER

There are two volmeter-ammeter instruments installed on the LH switch panel for LH and RH power source. They measure voltage and current at the electric power source bus.

Range of indication is 0 to 300A and 0 to 40V.

Pointer indicates current normally and, if the knob on the left lower corner of the dial is pushed, indicates voltage.

7.2 CABIN ALTITUDE DIFFERENTIAL PRESSURE INDICATOR

This indicator is a 3-inch dial, flange-mount type instrument installed on the RH switch panel operated directly by cabin pressure and atmospheric pressure. Cabin altitude is shown in feet, and differential pressure is shown in psi. Range of indication is 0 to 50,000 ft. and scale of differential pressure is 0 to 10.0 psi. The maximum differential pressure for this airplane is set at 5.25 ± 0.1 psi (Aircraft S/N 652SA), 6.0 ± 0.1 psi (Aircraft S/N 661SA, 697SA and subsequent).

7.3 CABIN RATE-OF-CLIMB INDICATOR

This indicator is a 2-inch dial, flange-mount type instrument installed on the RH switch panel and indicates cabin pressure increase rate and decrease rate in ft/min. Range of scale is 0 to ± 6,000 ft/min.

7.4 OUTSIDE AIR TEMPERATURE INDICATOR

This indicator is a 2-inch dial, flange-mount type instrument installed on the LH instrument panel and is powered by 28 VDC. Its sensor is a resistance bulb installed on the lower outer skin of the rear fuselage. Range of scale is -50° to +50°C.

7.5 TROUBLE SHOOTING

Trouble shooting for the cabin altitude differential pressure indicator, cabin rate-of-climb indicator, outside air temperature indicator, voltmeter-ammeter.

Trouble	Probable cause	Remedy
Indication of cabin altitude and differential pressure is erratic	a. Instrument faulty. b. Static pressure line is leaking.	Replace instrument. Perform leak check. (See static pressure system.)
Indication of cabin rate-of-climb indicator is erratic	Instrument faulty.	Replace instrument.
Cabin rate-of-climb indicator does not indicate zero in constant cabin altitude	Zero adjustment erroneous.	Perform zero adjustment.
Indication of outside air temperature indicator is erratic	a. Instrument faulty. b. Broken wire or poor contact.	Replace instrument. Perform conductivity test and repair defective component.
Indication of voltmeter-ammeter is erratic or it is inoperative	a. Instrument faulty. b. Broken wire or poor contact.	Replace instrument. Perform conductivity test and repair defective component.

CHAPTER

11

CABIN EQUIPMENT

CHAPTER XI

CABIN EQUIPMENT

CONTENTS

1. GENERAL

Cabin equipment consists of the cockpit section, passenger seat section and baggage compartment section. Above each seat a reading light and cold air outlet are provided.
Also provided are room lights.

2. PILOT AND CO-PILOT SEAT

The pilot and co-pilot seats are attached to tracks attached to the cockpit floor. The pilot and co-pilot seats are adjustable fore/aft and up/down. A handle is provided for fore/aft adjustment at the inboard side of each seat. To move the seat forward or backward, pull the handle up and slide the seat to the desired position, then release the handle and slide the seat to the nearest locking position. To move the seat up or down, pull the lever up and adjust by increasing or decreasing the body weight to actuate the seat to the desired height; release the handle to the nearest lock position. The inboard arm rests are folding and are attached to the seat back. The outboard arm rests are fixed to the side walls. See Figure 11-1.

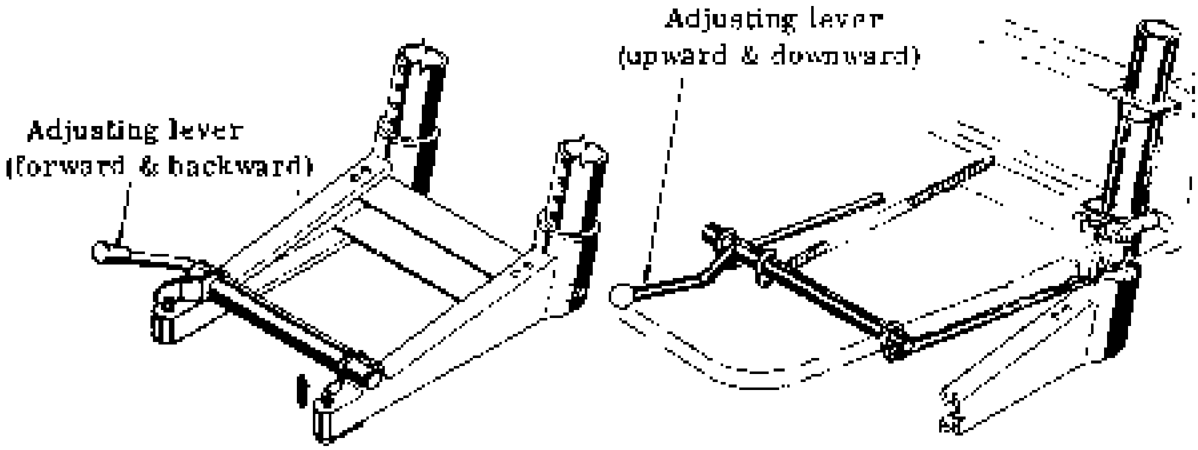


Fig. 11 - 1 Pilot seat adjusting mechanism

2.1 REMOVAL AND INSTALLATION (See Fig. 11-2)

(1) Remove stops fixed from tracks.

(2) Pulling up adjusting lever, slide pilot seat backward and engage guide in notch for installation and removal, then remove it.

(3) Install in reverse sequence of removal.

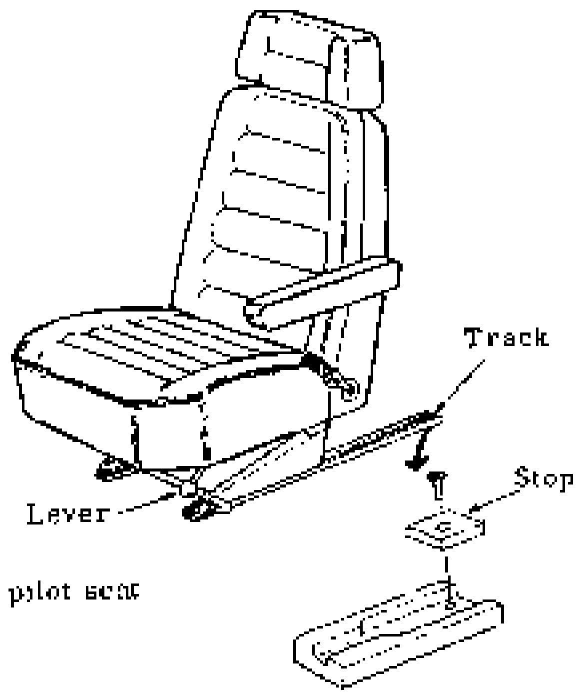


Fig. 11 - 2 Installation of pilot seat

3. INDIVIDUAL RECLINING SEATS

The individual passenger seats are attached to seat tracks attached to the floor and fuselage side wall of the passenger compartment. The reclining seats can be adjusted to suit the comfort of the occupant. A button on the inboard side of each seat adjusts the reclining position. To adjust the reclining angle, push the button and lean forward, then release the button. The side panel of the cabin may be used as outboard arm rests for the seats. Lift the inboard arm rest into position. Lift arm rest and then lower for stowing.

3.1 REMOVAL AND INSTALLATION (See Fig. 11-3)

1. Remove seat stops from floor track.

2. Lift the adjusting lever and slide the seat in its facing direction until the retaining guide is in the appropriate track notch and remove. Continue to slide the seat until all retaining guides are removed from the track. Remove the seat.

3. Install in reverse sequence of removal.

NOTE

When installing the seat stops, place a small amount of adhesive on one side before positioning in the seat track.

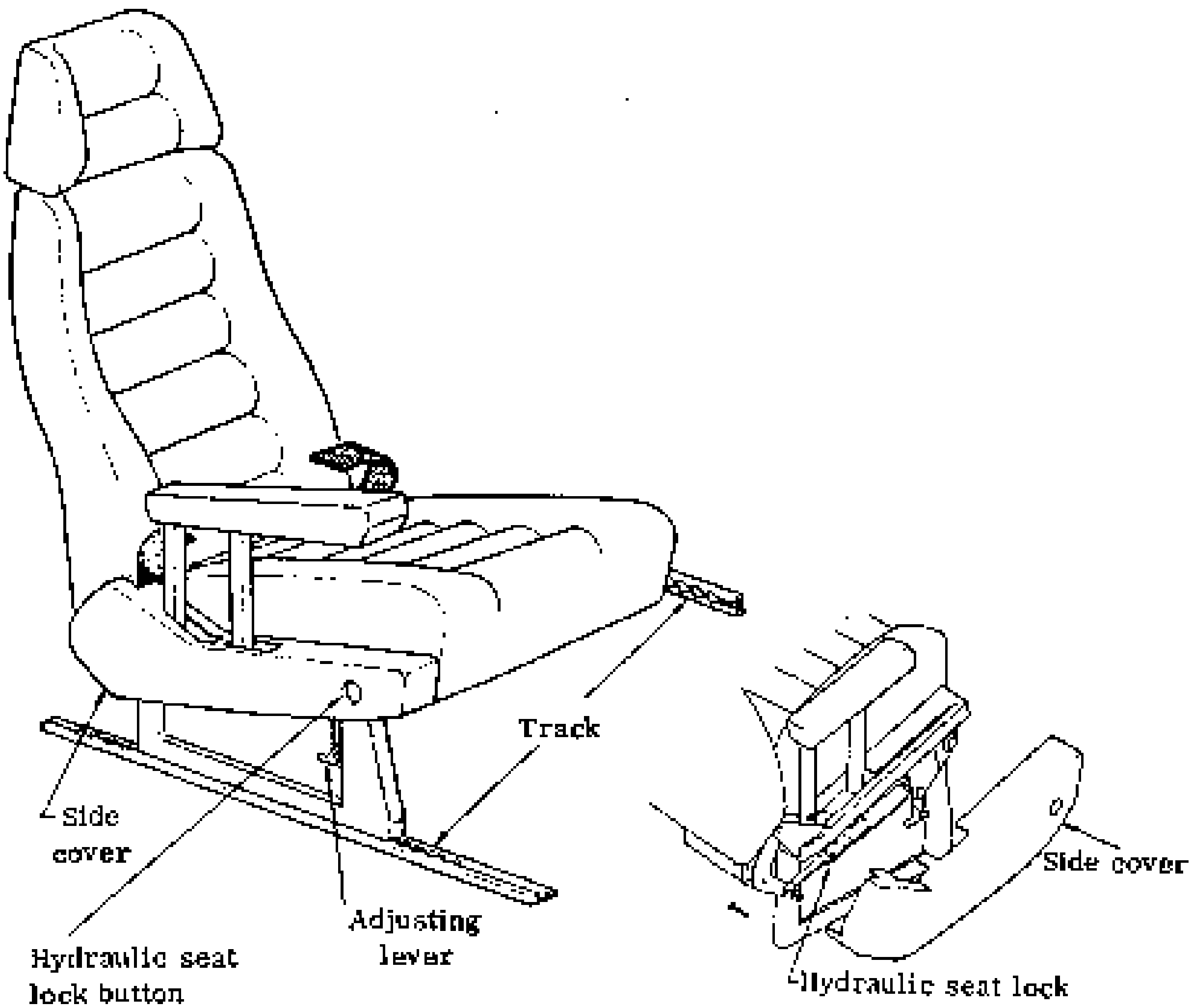


Fig 11-3 One passenger seat

3.2 HYDRAULIC SEAT LOCK

The hydraulic seat lock should be sent to the manufacturer (P.L. Porter Co.) for repair, disassembly and inspection as necessary when an oil leak is found or an abnormal load has been applied to the seat back due to emergency landing.

3.2.1 REMOVAL AND INSTALLATION (See Fig. 11-3)

(1) Remove side cover.

(2) Turn hydraulic seat lock counterclockwise and remove it.

(3) Install in reverse sequence of removal.

4. REFRESHMENT CENTERS

Hot and cold refreshment centers are available to be installed in the passenger compartment. The refreshment centers contain space available for liquor dispensers and bottles, ice, hot liquid container, stereo tape player and available storage space.

CHAPTER 12

SAFETY EQUIPMENT

CHAPTER XII

SAFETY EQUIPMENT

CONTENTS

1. GENERAL

An oxygen system, portable fire extinguisher, and first aid kit are provided as safety equipment of the aircraft.

2. OXYGEN SYSTEM

2.1 GENERAL DESCRIPTION (See Fig. 12-1)

The standard oxygen system supplies the pilot, co-pilot and one passenger with oxygen in case of an emergency, such as cabin pressurization failure at a high altitude. This system consists of an 1800 psi (126.6 kg/cm^2), 11 cu. ft. (0.3 cu.m) oxygen cylinder, regulator, 3 outlets and masks.

An optional oxygen system may be installed to supply oxygen to the remaining passenger seat location. This optional system is completely independent of the standard system, and employs an 1800 psi (126.6 kg/cm^2), 22 cu. ft. (0.6 cu.m) oxygen cylinder, regulator, outlets and masks (for the remaining passenger locations).

2.2 REMOVAL AND INSTALLATION

WARNING

Before removing any component of the oxygen system, ensure that no oxygen pressure remains in the system.

CAUTION

Care should be exercised when handling tubing so that it does not come in contact with any type of lubricant.

2.2.1 OXYGEN CYLINDER

(1) Close cylinder shutoff valve.

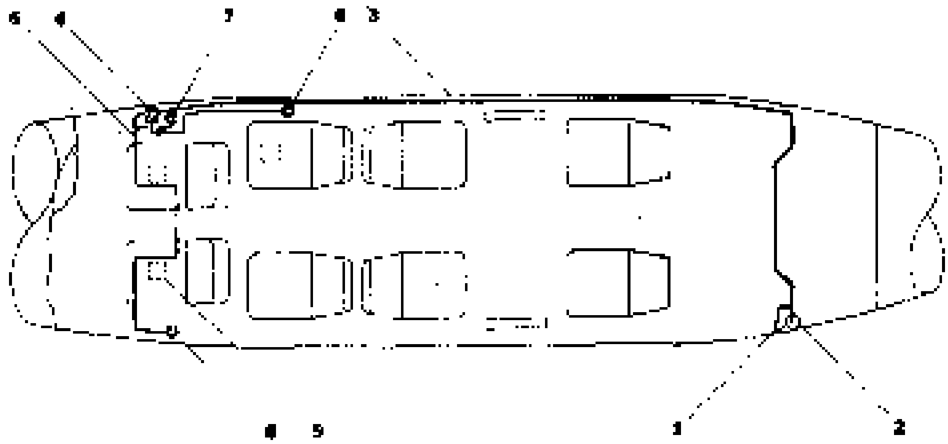
(2) Disconnect high pressure tube from cylinder.

(3) Loosen and/or disconnect securing clamps and remove cylinder assembly.

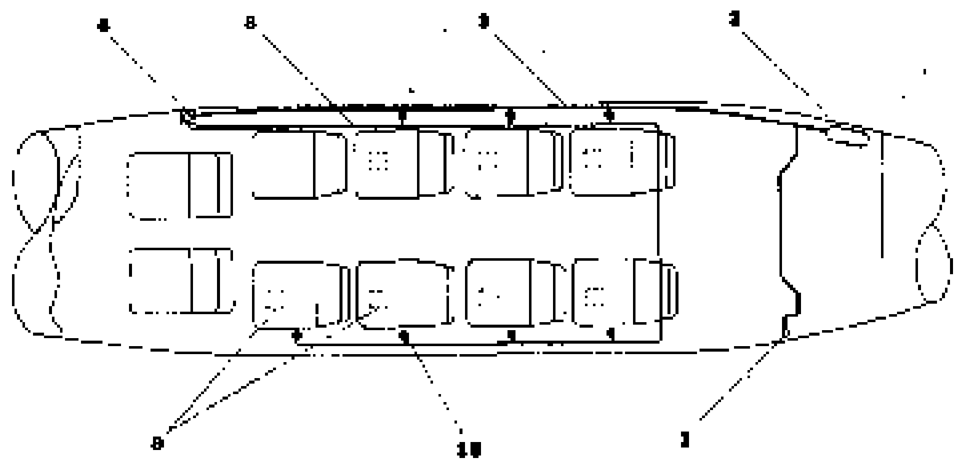
NOTE

It may be necessary to remove cabinet assembly enclosing cylinder (standard system only).

(4) Install in reverse sequence of removal. Apply alignment mark after tightening tubing connections to proper torque value.

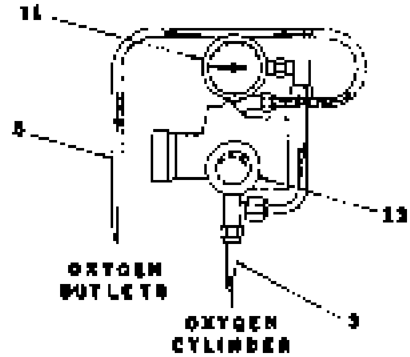


Standard Oxygen System



Optional Oxygen System

1. Oxygen filler valve
2. Oxygen cylinder/pressure gauge
3. High pressure tubing
4. Oxygen regulator
5. Low pressure tubing
6. Pilot oxygen outlet
7. Co-pilot oxygen outlet
8. Passenger oxygen outlet
9. Oxygen mask under seat
10. Passenger oxygen outlet
 (Located as req'd max 7)
11. Oxygen regulator pressure gauge
12. Oxygen regulator shutoff valve



REGULATOR

Fig. 12-1 Oxygen system

2.2.2 OXYGEN REGULATOR

(1) Close cylinder shutoff valve.

(2) Remove co-pilot side panel.

(3) Disconnect high and low pressure tubing from the regulator.

(4) Remove screws attaching regulator valve assembly and remove regulator valve assembly.

(5) Install in reverse sequence of removal. Apply alignment mark after tightening tubing connections to proper torque value.

NOTE

Installation torque for oxygen tubing should be applied the following values. In case of taper fittings, use Permacel Tape P412 (Permacel Tape Corp.) for sealing before installation.

120 to 140 in-lbs. (138 to 161 kg/cm) . . .stainless steel or copper tube.

90 to 110 in-lbs. (104 to 127 kg/cm) . . .aluminium tube.

2.3 INSPECTION

(1) Check oxygen cylinder for security of attachment, deformation, damage and oiliness.

(2) Check oxygen cylinder for pressure of 1500 to 1800 psi (105.5 to 126.6 kg/cm^2).

(3) Check pressure gauge for "0" (zero) indication with shutoff valve of oxygen regulator mounted on side panel in OFF position.

(4) Check oxygen outlets for cleanliness, rust and damage.

(5) Check oxygen mask for cleanliness and damage. Check mask bag for proper position.

CAUTION

High pressure oxygen is liable to cause an explosion upon contact with oil, grease, or solvent. The handling should be done with extreme care.

2.4 OPERATIONAL TEST

(1) Open cylinder shutoff valve gradually to supply oxygen in the high pressure line.

(2) Make sure indication of regulator pressure gauge is equal to that of the cylinder gauge.

WARNING

If oxygen cylinder valve is not open, the oxygen pressure gage will indicate the lower pressure in the system. If the indicated values for the oxygen cylinder and the regulator pressure gage are equal, the cylinder valve is open.

(3) Open regulator shutoff valve.

(4) Plug mask assembly into outlet, put mask on, and breathe normally. Make sure the red mark is not visible in the flow indicator attached to the mask assembly.

(5) Close regulator shutoff valve.

(6) Close cylinder shutoff valve.

CAUTION

High pressure oxygen is liable to cause an explosion upon contact with oil, grease, or solvent. Handling should be made with extreme care.

NOTE

When bleeding oxygen from the system after operation, close the shutoff valve of the cylinder, check the pressure gage of the regulator for 0 reading which indicates system pressure being equal to atomospheric pressure, and close valve of the regulator.

2.5 TUBING LEAK TEST

2.5.1 HIGH PRESSURE TUBING

(1) Close shutoff valves of regulator and cylinder.

(2) Remove filler port cap.

(3) Connect oxygen (Spec. BB-O-925, Grade A, Type 1) or nitrogen charger (N_2: MIL-N-6011, Grade A, Type 1) to filler port.

(4) Pressurize system gradually to 1800 ± 50 psi (126.6 ± 3.5 kg/cm^2). Maintain the pressure 5 to 10 minutes, to allow the tubing and gas to adjust to room temperature, then close the valve.

(5) Record the regulator pressure gauge indication and the room temperature.

(6) After 5 minutes perform step (5).

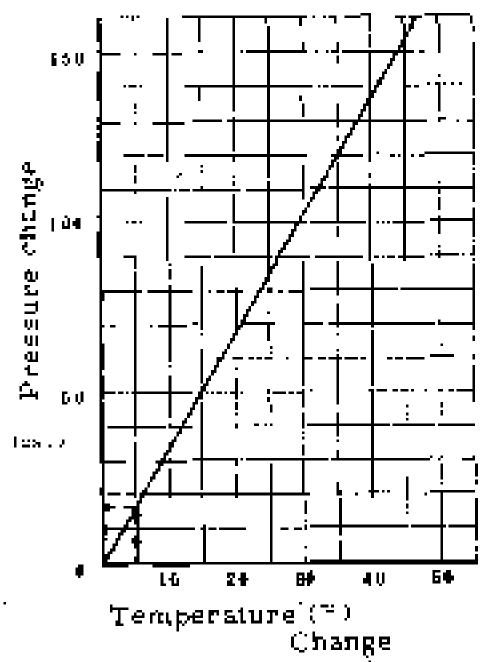
(7) Compensate the pressure for the room temperature in accordance with Figure 12-2.

(8) Ensure the pressure drop is below 50 psi (3.5 kg/cm^2). To calculate the pressure drop, use the following example.

Example:

(a)	Initial gauge pressure	1800 psi (126.6 kg/cm^2)
(b)	Initial temperature	75°F (24°C)
(c)	Final gauge pressure	1780 psi (125 kg/cm^2)
(d)	Final temperature	70°F (21°C)
(e)	Temperature change (b)-(d) =	5°F (3°C)
(f)	Pressure change corresponding to temperature change (Fig. 12-2)	17 psi (1.2 kg/cm^2)
(g)	Initial pressure compensated for 75°F (24°C) Temperature rise (g) = (a)+(f) Temperature fall (g) = (a)-(f)	1817 psi (127.8 kg/cm^2)
(h)	Pressure drop (g)-(c)	37 psi (2.6 kg/cm^2)

Fig. 12-2 Initial pressure compensation for temperature change



(9) Should the pressure drop exceed 50 psi (3.5 kg/cm^2), proceed as follows:
   (a) Remove interior as necessary.
   (b) With the system pressurized, apply soap water (MIL-S-4282 or equivalent) to all fittings and connections of the high pressure section from the filler port, including cylinder and regulator, to detect leaks. Repair the leak(s) and recheck the system.

(10) Restore the aircraft to its original configuration.

2.5.2 LOW PRESSURE TUBING

(1) Gradually open cylinder valve until the system is pressurized to 1800 ± 50 psi (126.6 ± 3.5 kg/cm^2).

(2) Open the regulator shutoff valve and pressurize the low pressure tubing lines.

(3) Apply soap suds (MIL-S-4282 or equivalent) to all fittings and joints. Leakage causing a bubble of 0.24 in. (6.1 mm) in diameter to be formed 10 seconds or longer, or a bubble of 0.47 (12 mm) in diameter 60 seconds or longer may be permissible. Leakage must be repaired if found excessive.

(4) Close regulator and cylinder shutoff valve.

2.5.3 CHECK AFTER LEAK TEST

(1) Wash the system with clean water and wipe with a dry cloth after leak test is performed.

(2) If nitrogen is used for testing, bleed from the system and reduce the system pressure to that equal to atmospheric. Blow oxygen into the system for cleaning for at least 2 minutes prior to recharging.

2.6 OXYGEN CYLINDER RECHARGING PROCEDURE

(1) Connect oxygen supply source to filler port.

(2) Open cylinder shutoff valve and recharge the cylinder at a charging rate not to exceed 500 psi/min. (35.2 kg/cm^2) until cylinder pressure gauge indicates 1800 +50 psi (126.6 ± 3.5 kg/cm^2).

(3) Allow time for the cylinder and oxygen to equalize that of room temperature, then recheck the cylinder gauge for 1800 ± 50 psi (126.6 ± 3.5 kg/cm^2). Recharge if necessary.

(4) Close the cylinder valve and detach supply source from the filler port and install port cap.

(5) Watch gauge indication for a while to ensure no leakage occurs.

3. PORTABLE FIRE EXTINGUISHER (See Fig. 12-3)

A portable fire extinguisher is installed on the right hand armrest by co-pilot's seat. The extinguisher is for Class A, B and C fires. To operate, remove it from the mounting bracket and break the seal. Aim the nozzle at the base of the fire and depress the handle (or trigger) to extinguish fire.

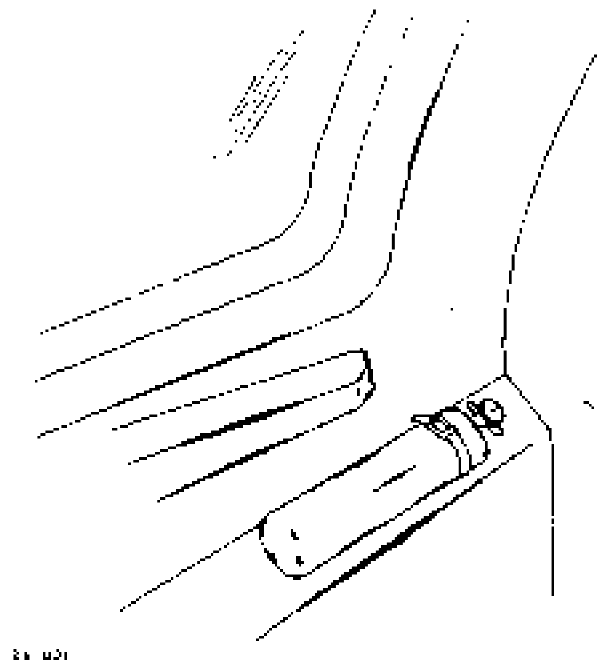


Fig. 12-3 Portable fire extinguisher

4. FIRST AID KIT (See Fig. 12-4)

The first aid kit is installed on the side of the personal convenience center. To remove, pull the bottom up slightly and lift from the two screw heads.

The first aid kit and supplies are from Scott Aviation, Lancaster, New York, 14086. The kit contains sterile adhesive bandages; triangle bandages and bandage compresses; iodine swabs; carbolated petrolatum, and ammonia inhalant. The kit lid may contain general instructions for administering immediate first aid for common injuries.

The equipment and supplies in the kit should be checked periodically and items replenished as required after each usage.

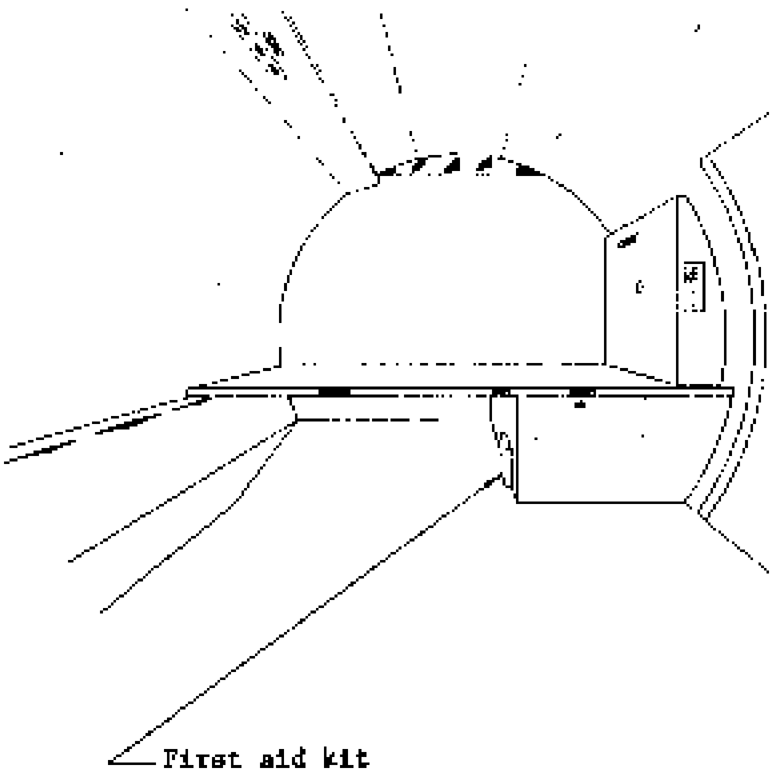


Fig. 12-4 First aid kit

5. EMERGENCY LOCATOR TRANSMITTER (ELT) (If installed)

5.1 DESCRIPTION

Emergency locator transmitter (ELT. DORNE and MARGOLIN Inc.) is a transmitter to assist in locating downed aircraft and is operable under a wide range of environmental conditions.
ELT consists of transmitter (including battery pack), antenna and control switch. (See Figure 12-5)

(1) Transmitter and battery

(a) Transmitter is equipped with internal battery pack, composed of 6 long life alkaline D type cells operable at low temperatures.
Battery pack is installed in transmitter properly separated from transmitter electronic circuit.

(b) G switch incorporated in transmitter detects abnormal shocks in airplane, and automatically turns on transmitter.

(2) Antenna

(a) DM ELT 6 unit utilizes flexible antenna installed inside dorsal fin.

(3) Control switch

(a) Control switch is installed on switch panel to allow remote control from cockpit.

Transmitter is installed at upper part of aft fuselage. (See Fig 12-6 (1/3))

Flexible antenna is installed protruding into interior of dorsal fin. (See Fig 12-6 (2/3))

Toggle type remote control switch is installed on auxiliary switch panel in left switch panel. (See Fig 12-6 (3/3))

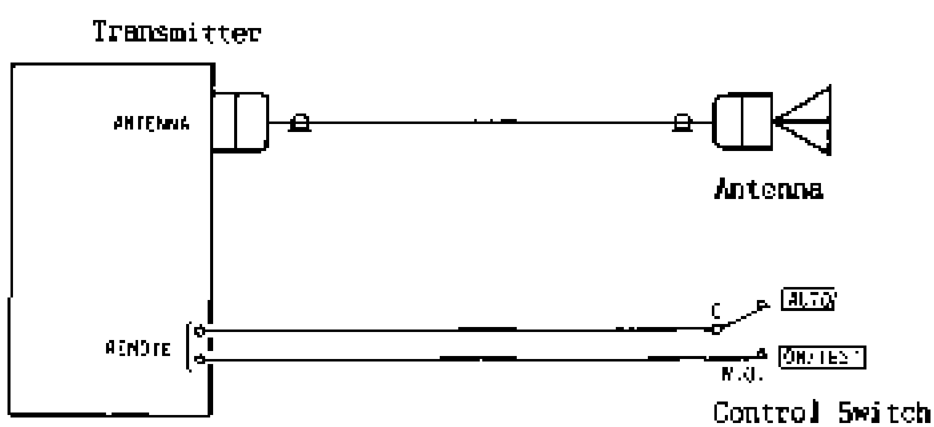


Fig.12-5 ELT schematic diagram

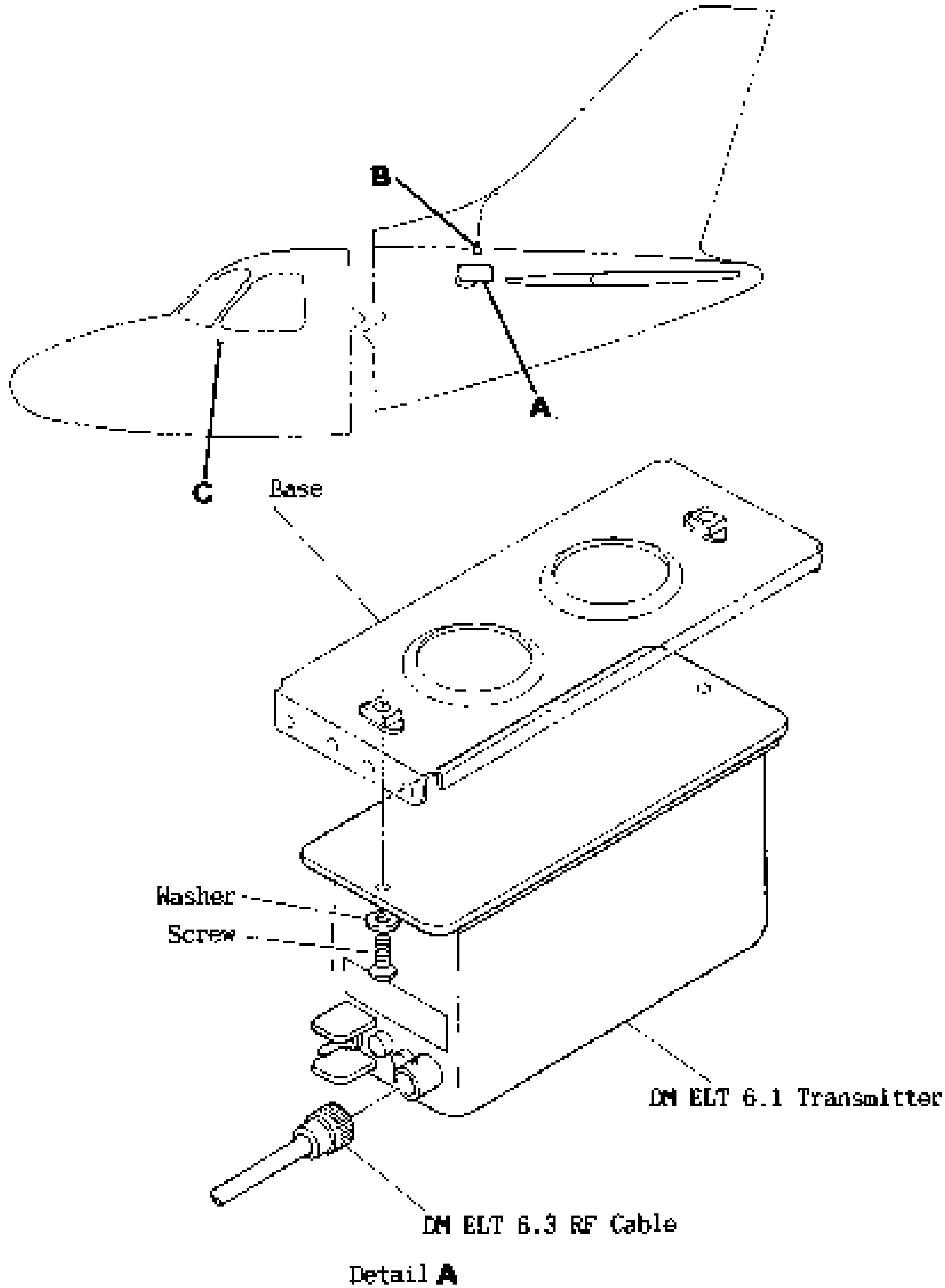
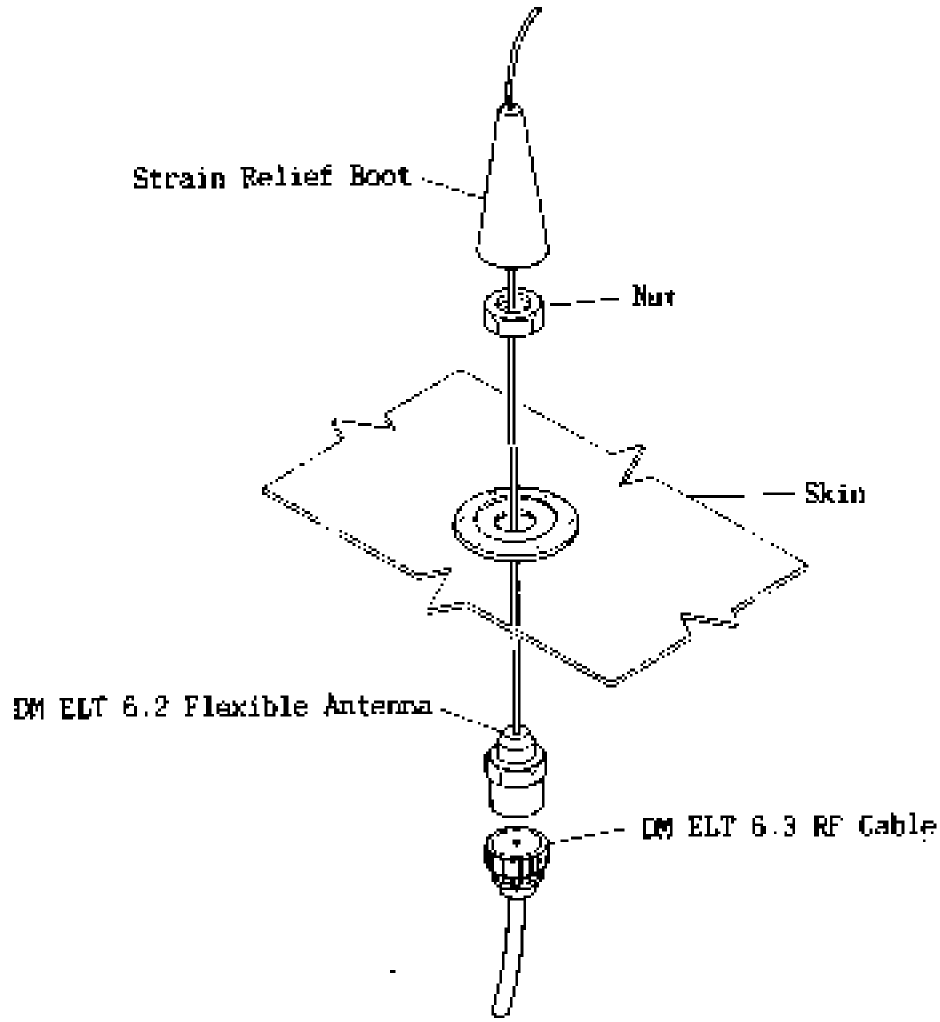
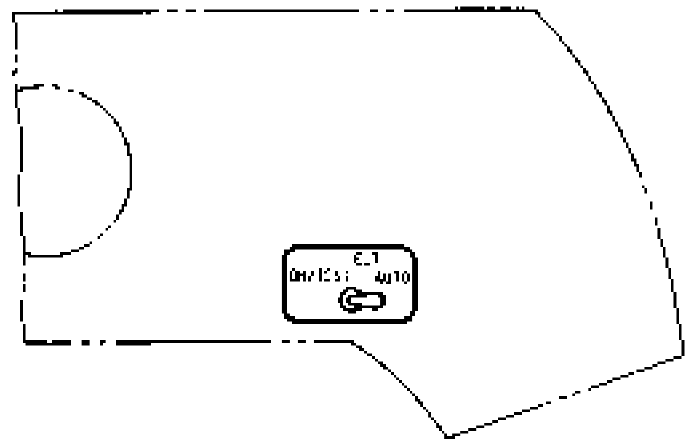


Figure 12-6 Location of DM ELT 6 Components (1/3)



Detail B

Figure 12-6 Location of DM ELT 6 Components (2/3)



Detail C

Fig. 12-6 Control switch Installation (3/3)

5.2 REMOVAL AND INSTALLATION

5.2.1 REMOVAL OF ELT 6.1 TRANSMITTER

(1) Open aft fuselage door, and make access to transmitter.

(2) Disconnect DM ELT 6.3 RF matching cable assembly from transmitter.

(3) Remove screws and washers securing transmitter to base.

(4) Remove transmitter from airplane.

5.2.2 INSTALLATION OF DM ELT 6.1 TRANSMITTER

(1) Position transmitter on base, and secure with screws and washers.

(2) Connect RF cable assembly to transmitter.

(3) Do operation test.

5.2.3 REMOVAL OF DM ELT 6.2 FLEXIBLE ANTENNA.

(1) Remove dorsal fin.

(2) Open aft fuselage door.

(3) Disconnect DM ELT 6.3 RF matching cable assembly from antenna.

(4) Remove nuts, and then remove antenna.

5.2.4 INSTALLATION OF DM ELT FLEXIBLE ANTENNA

(1) Position antenna in the interior of dorsal fin.

(2) Secure antenna with nuts.

(3) Connect RF cable to antenna.

(4) Do operation test.

(5) Install dorsal fin.

5.3 CHECK

5.3.1 VISUALLY CHECK TRANSMITTER COMPONENTS

(1) Check for evidence of damage in antenna assembly, and ensure it is correctly installed.

(2) Check for evidence of corrosion in transmitter, battery pack assembly and surrounding structures. Check condition of installation and base for evidence of damage and function switch.

(3) To check transmitter, remove antenna of unit.

5.3.2 BATTERY PACK

(1) Replace battery pack when emergency locator transmitter (ELT) was used one hour or over in total, and/or the battery reached its usable period.

(a) Date showing usable period is marked on battery by maker.

(b) Whenever battery is installed in ELT, note the battery usable period on the space provided on Name and Data Plate of ELT unit.

(2) Operation time of battery pack should be recorded when doing operation test on ELT unit with battery pack power. Replace battery when battery pack has been operated one hour in total.

5.4 OPERATION

5.4.1 OPERATION OF SWITCHES

ELT unit is automatically turned on by a shock given to impulse ("G") switch, or manually turned on with remote control switch.

(1) Impulse switch functions and turns on transmitter when it is given a force of 5+2/-0 g for 11+5/-0 milliseconds to the direction of longitudinal axis of airplane.

(2) If ELT does not operate at crashed landing, it can be manually operated with remote control switch. Impulse ("G") switch functions only when ELT is set to automatic mode.
That is, it functions only when "ON/OFF/AUTO" switch on transmitter front panel is set to "AUTO". When resetting transmitter to automatic mode after it was turned off, set remote control switch to "ON/TEST" and then reset to "AUTO". When transmitter functions, modulating signals are transmitted with emergency frequencies of 121.50 and 243.00 Hz simultaneously. Modulating signal is a 700 Hz sweep signal. (Max. 1600 Hz, Min. 300 Hz)
In normal operations, remote control switch is set to "AUTO" position. Transmitter automatically operates by shock and sends emergency distress signal. Impulse switch can be reset as follows when transmitter is accidentally turned on by impulse switch for some reason.

(3) Set remote control switch to "ON/TEST" position, and then reset to "AUTO".

5.4.2 OPERATION TEST

(1) Obtain authorization from control tower and flight service station for monitoring of ELT.

(2) Operate ELT unit.

(a) Ensure transmitter "ON/OFF/AUTO" switch is at "AUTO".

(b) Communicate with control tower, and set remote control switch to "ON/TEST" to test ELT unit. After testing, reset remote control switch to "AUTO", or communicate with control tower, and set transmitter "ON/OFF/AUTO" switch to "ON" to test ELT unit. After testing, reset "ON/OFF/AUTO" switch to "AUTO".

CAUTION

Authorization by control tower and flight service station should always be obtained before checking operation of ELT.

(c) Record the length of time powered transmitter with battery pack. Total time should conform to replacement and inspection schedules.

5.5 REPLACEMENT OF BATTERY PACK

NOTE

Replace battery pack when total operation time exceeds one hour, or usable period of battery is exceeded.

(1) Set transmitter "ON/OFF/AUTO" switch to "OFF".

(2) Disconnect transmitter RF cable.

(3) Remove transmitter from airplane.

(4) Loosen screw securing base to bottom of transmitter, and remove base.

(5) Disconnect and remove battery pack from unit.

(6) Replace with new battery pack, DM ELT 6.13 or DM ELT 8.B.

(7) Reassemble and reinstall transmitter.

(8) Connect transmitter to antenna.

(9) Do operation test.

(10) Stamp on transmitter the date replaced with new battery. Date on battery pack and transmitter should be the same.

CHAPTER

13

ELECTRICAL SYSTEM

CHAPTER XIII

ELECTRICAL SYSTEM

CONTENTS

1. GENERAL

1.1 GENERAL DESCRIPTION

The electrical system consists of a negative grounded 28 VDC power supply, and one line grounded, single phase, 400 Hz, constant frequency, AC power supply with associated wiring and protective devices. The DC electric power supply is the primary electrical power source for the airplane and is supplied by two generators; two batteries; or by an external power supply through an external power receptacle. The batteries are used for engine starting and as electrical power in case of no power being supplied from the generator. Two nickel cadmium type batteries are installed in order to facilitate engine starting during cold weather. When the battery switch is turned to ON, the two batteries are connected to the main bus, in parallel. While starting an engine, the batteries can be connected automatically, in parallel or series, to the starter by a battery selector switch. A DC generator, with a nominal output of 30V, 200A, is installed on each engine, supplying power to the airplane electrical systems for operation and battery charging. Each generator circuit is provided with a reverse current relay which prevents current from the battery flowing back to the generator, and a shunt for the ammeter. The output voltage of each generator is kept constant by a transistorized voltage regulator. External power can be connected directly to the airplane by connection of an external power source to the receptacle located forward of the LH electrical compartment door. When the MASTER switch, located on the LH switch panel, is placed in EMER position, all electric power sources in the airplane are cut off. The left and right engine fire detecting circuits remain operative, even when the master switch is placed in EMER position, because the circuits are directly operated by battery. The MASTER switch is used only in case of emergencies such as an electrical fire, forced landing, etc., and is normally protected by a switch guard in the NORMAL position. The amperage output of each generator and voltage of the monitored bus is indicated by a volt-ammeter on the LH switch panel. Unusually high voltage from the generators is detected by an overvoltage protecting relay. If a generator fails, it is automatically disconnected by reverse current cutout relay from the aircraft electrical bus. When the generator is cut off from the bus, the L(R) DC GEN OUT warning light on the annunciator panel illuminates. Alternating current is provided by means of a static inverter that is installed in the LH side of the electrical compartment.

While starting an engine, the inverter AC voltage output drops, following any DC power drops. In order to prevent this drop, a DC-DC converter, operated only during engine start, is installed and normal AC power is generated and supplied to the interstage turbine temperature indicator (ITT) (Aircraft S/N 652SA only). After an engine start, the DC-DC converter is automatically cut off. A single phase, 250V, 400 Hz inverter supplies the air conditioning system, fuel quantity indicating system and engine instrument system with the required AC power. Inverter failure is detected by an inverter fail relay and indicated by an INVERTER FAIL warning light on the annunciator panel. The electrical circuit of each system is protected by trip-free circuit breakers or fuses. The majority of circuit breakers are installed on the circuit breaker panel to the left side of the pilot seat. Some circuit breakers for anti-icing are in the overhead switch console.

The circuit breakers for the cabin lighting engine start circuits, utility and fuses for the DC power source (ammeters, generator voltage, etc.) are installed in the upper LH side of the main junction box.

CAUTION

Circuit breakers once tripped due to excessive current cannot be reset as long as an overload current exists. Circuit breakers which trip twice should not be reset except in an emergency. When necessary, extreme care should be exercised to prevent electrical fire when resetting the circuit breaker. The number of manual circuit breaker operations on the ground should be kept to a minimum.

Warning lights which are important to flight safety are grouped on the annunciator panel and a master caution light is used for general caution. Two landing lights are installed on each side of the lower surface of the forward fuselage and one in each tip tank. Anti-collision lights are installed on the lower surface of the fuselage and on top the vertical stabilizer. Navigation lights and strobe lights are installed on LH and RH wing tips and aft fuselage and also an ice inspection light on LH center wing leading edge fillet. Cabin room lights and reading lights are installed in the cabin. Room lights and map reading lights are installed in the cabin. Room lights and map reading lights are provided in the cockpit. Illumination of flight instruments, engine instruments, and switch panels can be adjusted independently by the rheostats located in the overhead switch console. Location of the electrical equipment is shown in Figs. 13-1 through 13-12.

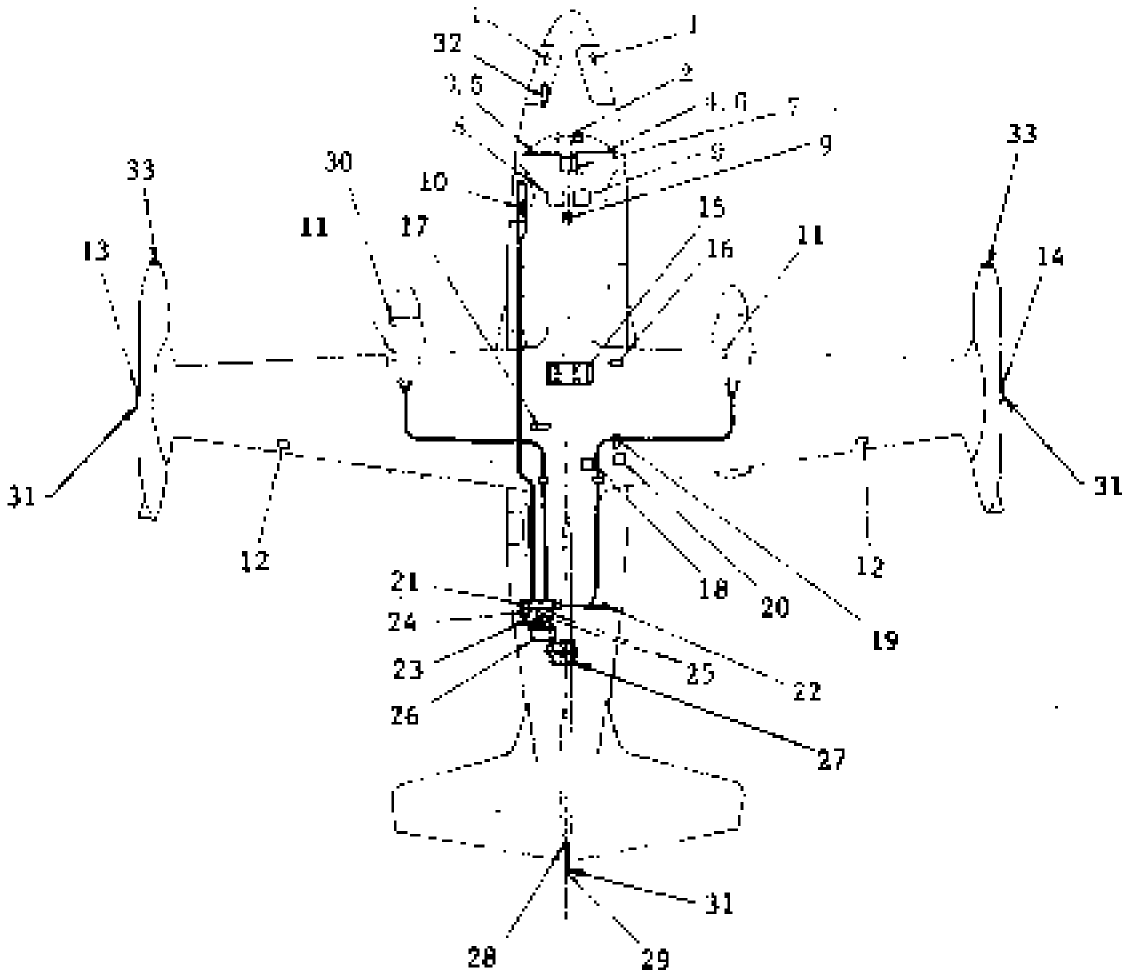
1.2 POWER DISTRIBUTION SYSTEM

In the DC power system, LH and RH generators are connected to the LH and RH main bus through individual reverse current cutout relays installed on the main junction box. The No. 1 battery is connected to the main bus and the No. 2 battery and the external power source are connected to the main bus through the starter bus and the series parallel relay.

The LH and RH main buses are connected with power circuit breaker. In the cockpit circuit breaker panel, there are the LH and RH buses which are supplied power from the LH and RH main buses through feeder-protected circuit breaker, feeder and reverse current cutout diode. When the feeder-protected breaker is tripped due to failure, etc., the L(R) FEEDER OUT warning light(s) in the annunciator panel illuminates for warning.

The bus in the overhead switch console is supplied by power through the circuit breaker OVERHEAD PANEL located on the circuit breaker panel, left side of pilot seat. LH and RH power for radio equipment is supplied from the load bus through a toggle type circuit breaker. The AC power distribution system from the inverter supplies power from the electrical compartment, LH to the circuit breaker panel, left of the pilot seat.

AC power is supplied to the fuel quantity indicator system and engine instrument system through circuit breaker. The AC power 26V, 400 Hz necessary for the engine instrumentation system is provided by the main or standby inverter in the electric compartment and is supplied to the circuit breaker.



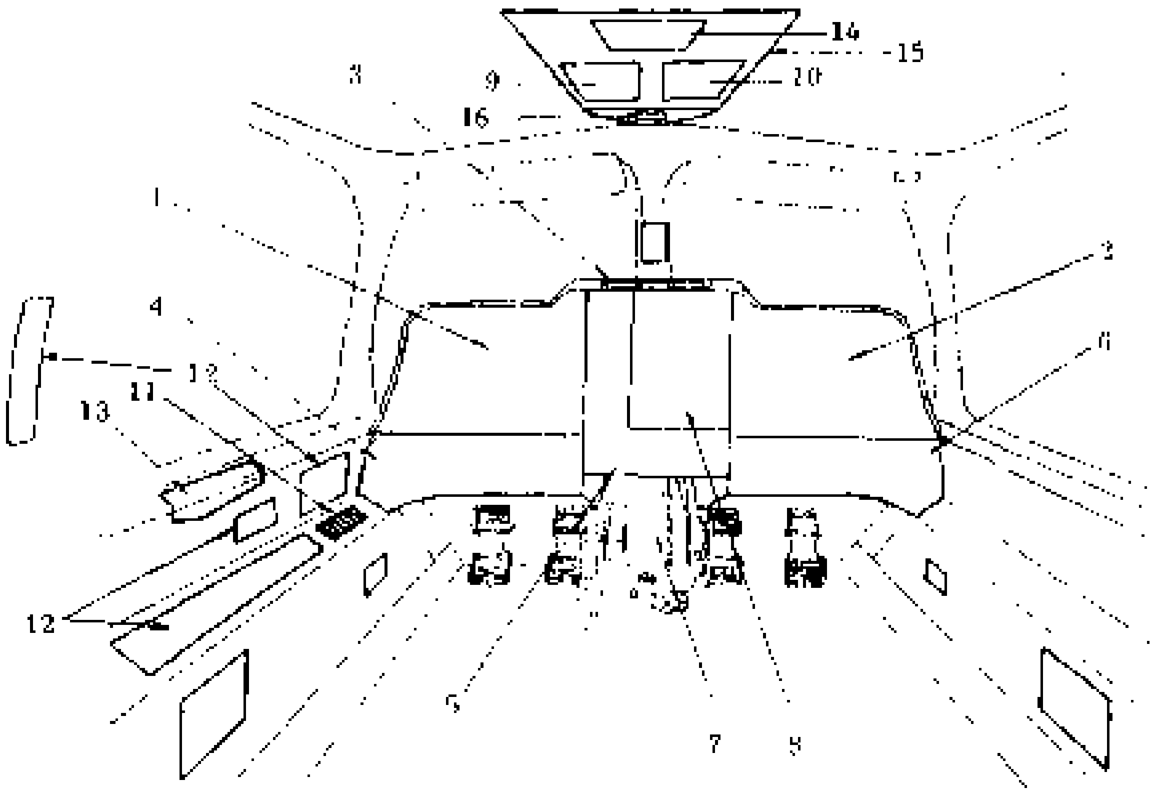
1. Landing and taxi light
2. Connection box
3. LH instrument panel
4. RH instrument panel
5. LH switch panel
6. RH switch panel
7. Center pedestal switch panel
8. Overhead switch console
9. Anticollision light (Fuselage lower surface)
10. Circuit breaker panel
11. Starter generator
12. Trim aileron actuator
13. LH wing tip light
14. RH wing tip light
15. Relay panel (Center wing leading edge)
16. Landing gear door actuator (in bulge)
17. Flap motor (Center wing trailing edge)
18. Propeller de-icer timer
19. Landing gear motor
20. Landing gear control relay (in bulge)
21. Main junction box
22. Inverter
23. External power receptacle
24. Wing de-icer timer
25. Air conditioning controller
26. No. 1 battery
27. No. 2 battery
28. Anti-collision light (Vertical stabilizer)
29. Tail light
30. Ice inspection light
31. Strobe light

(*1) 32. Heated windshield anti-ice controller & relay

33. Tip tank taxi light

*1 Aircraft S/N 661SA, 697SA and subsequent

Fig. 13-1 Main electrical equipments

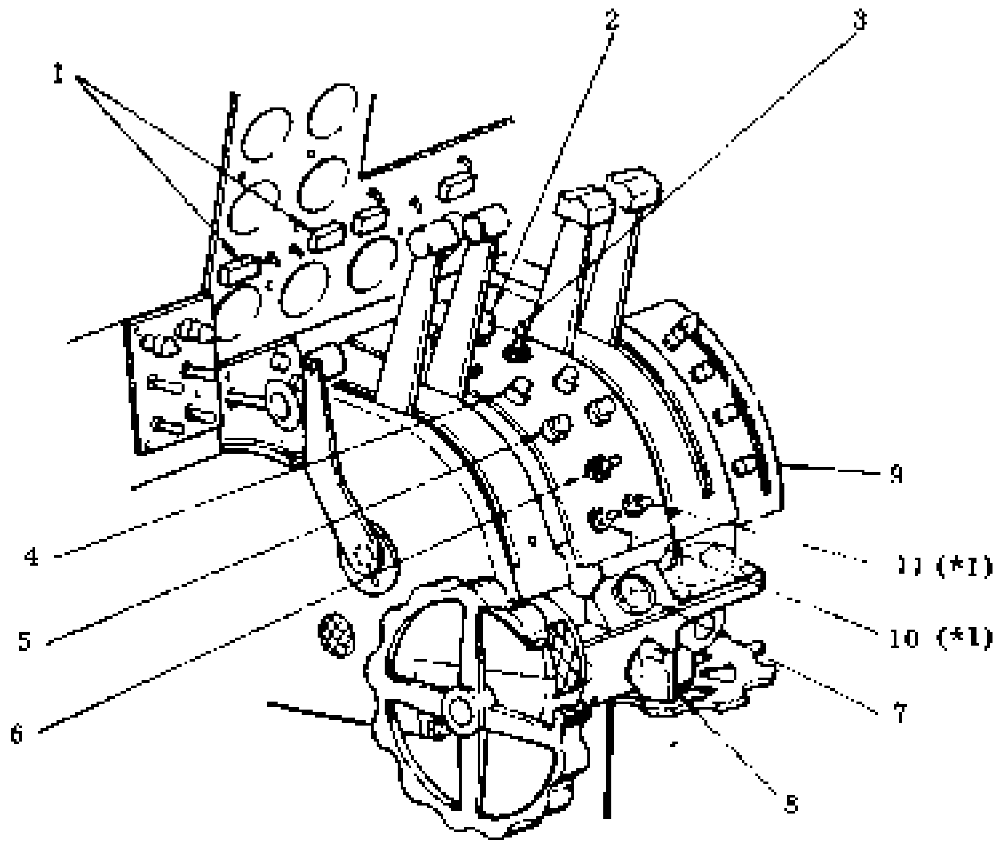


1. LH instrument panel
2. RH instrument panel
3. Shroud switch panel
4. LH switch panel
5. Engine & instr. panel
6. RH switch panel (See Fig. 13-4)
7. Center pedestal switch panel (See Fig. 13-3)
8. Radio panel
9. LH upper switch panel (*1)
10. RH upper switch panel (*1)
11. Annunciator panel
12. Circuit breaker panel
13. Circuit breaker panel light
14. Center pedestal lighting panel (*1)
15. Overhead switch console (*2)
16. Loadmeter & selector switch (*2)

*1 Aircraft S/N 652SA

*2 Aircraft S/N 661SA, 697SA and subsequent

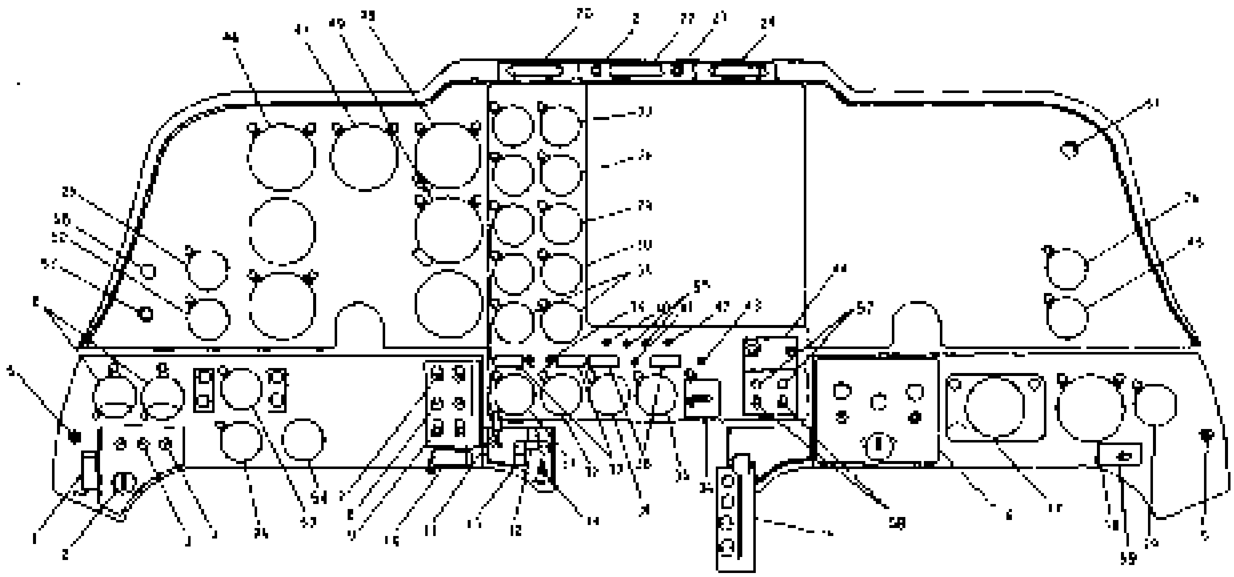
Fig. 13-2 Cockpit electrical installation



1. Beta range light
2. Engine start switch
3. Engine crank run stop switch
4. Unfeathering switch
5. Start fuel enrich switch
6. Engine start select switch
7. Aileron trim indicator
8. Aileron trim control switch
9. Flap controller
10. Manual propeller de-ice switch (*1)
11. Manual propeller de-ice indicator light (*1)

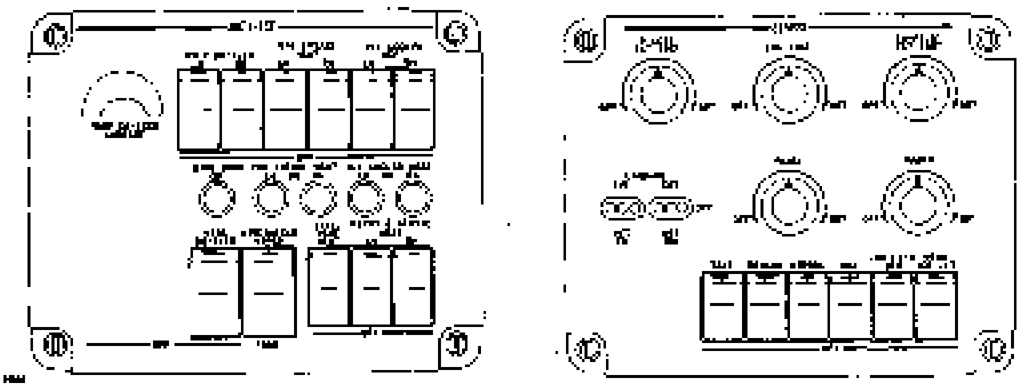
Fig. 13-3 Center pedestal switch panel

*1 Aircraft S/N 652SA only

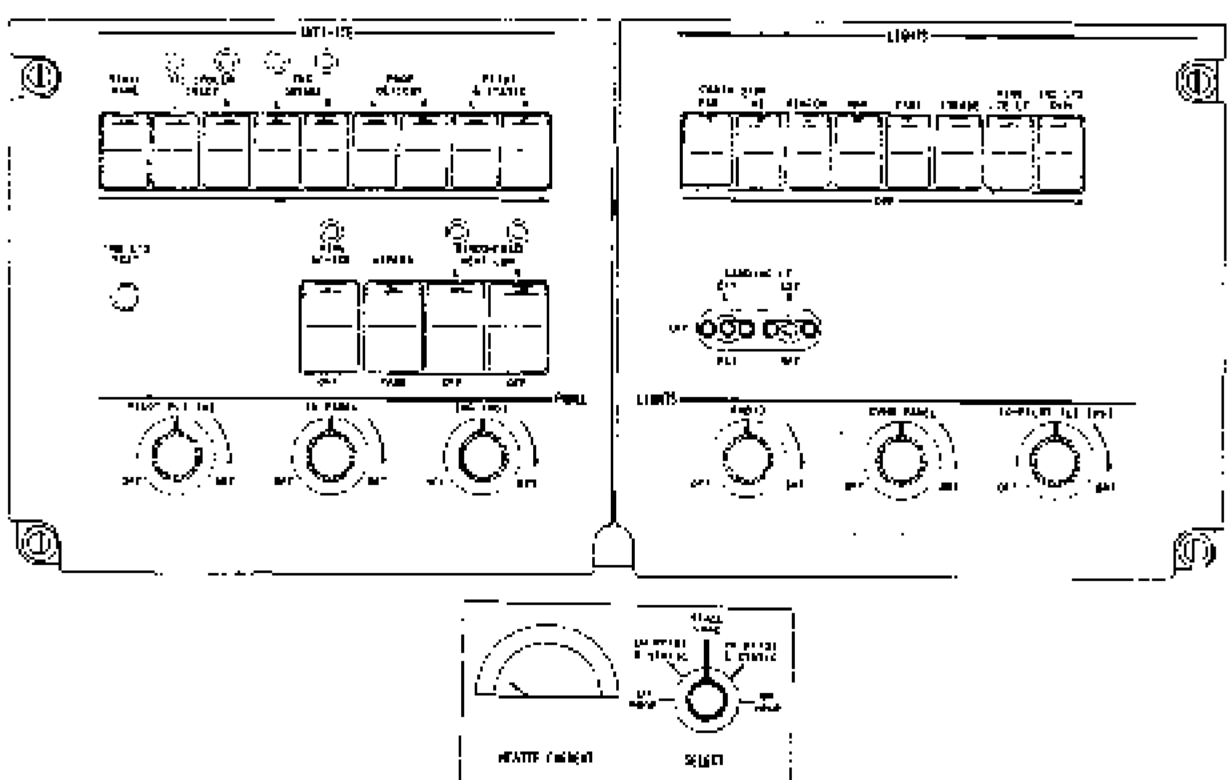


1. MASTER SWITCH
2. BATTERY SWITCH
3. DC GENERATOR CONTROL SWITCHES
4. INVERTER SWITCH
5. MICROPHONE JACK
6. VOLTAMMETERS
7. MAIN FUEL VALVE SWITCHES
8. TRANSFER SWITCHES
9. POWER AUTO LIMIT SWITCHES
10. TRIM AILERON SELECT SWITCH
11. LANDING GEAR CONTROL SWITCH
12. LANDING GEAR POSITION INDICATING LIGHTS
13. LANDING GEAR UNSAFE LIGHT
14. LANDING GEAR HORN CUTOUT SWITCH
15. FLAP SWITCH
16. AIR CONDITIONING CONTROL PANEL
17. CABIN CONTROLLER
18. CABIN ALTITUDE DIFFERENTIAL PRESSURE INDICATOR
19. CABIN RATE OF CLIMB INDICATOR
20. LH ENGINE FIRE WARNING LIGHT AND FIRE EXTINGUISHER HANDLE
21. ENGINE FIRE DETECTOR TEST BUTTON
22. MASTER CAUTION LIGHT
23. MASTER CAUTION SYSTEM TEST SWITCH
24. RH ENGINE FIRE WARNING LIGHT AND FIRE EXTINGUISHER HANDLE
25. OUTSIDE AIR TEMPERATURE INDICATOR
26. CLOCK
27. TORQUE METERS
28. ITT INDICATORS
29. FUEL FLOW INDICATORS
30. TACHOMETERS
31. OIL PRESSURE INDICATOR
32. FUEL PRESSURE INDICATOR
33. OIL TEMPERATURE INDICATOR
34. MAIN TANK FUEL QUANTITY INDICATOR
35. TIP TANK FUEL QUANTITY INDICATOR
36. FUEL CONSUMPTION TOTALIZER
37. BETA RANGE INDICATOR LIGHTS
38. OUTER WING TANK FUEL EMPTY WARNING LIGHTS
39. POWER AUTO LIMIT TEST SWITCHES
40. PANEL INDICATOR LIGHT TEST SWITCH
41. FUEL QUANTITY INDICATOR TEST SWITCH
42. STALL WARNING TEST SWITCH
43. FUEL LOW LEVEL TEST SWITCH
44. PROPELLER SYNCHROPHASER SWITCH
45. VACUUM GUAGE
46. AIRSPEED INDICATOR
47. ATTITUDE GYRO
48. ALTIMETER
49. RATE OF CLIMB INDICATOR
50. TURN AND BANK INDICATOR
51. TURN AND BANK INDICATOR POWER FAIL LIGHT
52. DEFOG AIR OVER TEMPERATURE WARNING LIGHT
53. BATTERY TEMPERATURE INDICATOR AND ISOLATE SWITCHES
54. HOUR METER
55. CABIN SIGN LIGHT SWITCH (S/N 652 only)
56. ALTERNATE STATIC SELECT VALVE
57. CONTINUOUS IGNITION INDICATOR LIGHT (If installed)
58. CONTINUOUS IGNITION SWITCH (If installed)
59. ELT CONTROL SWITCH (If installed)

Fig.13-4 Instrument and Switch Panel (Typical)



Aircraft S/N 652SA



Aircraft S/N 661SA, 697SA and subsequent

Fig. 13-5 Overhead switch console

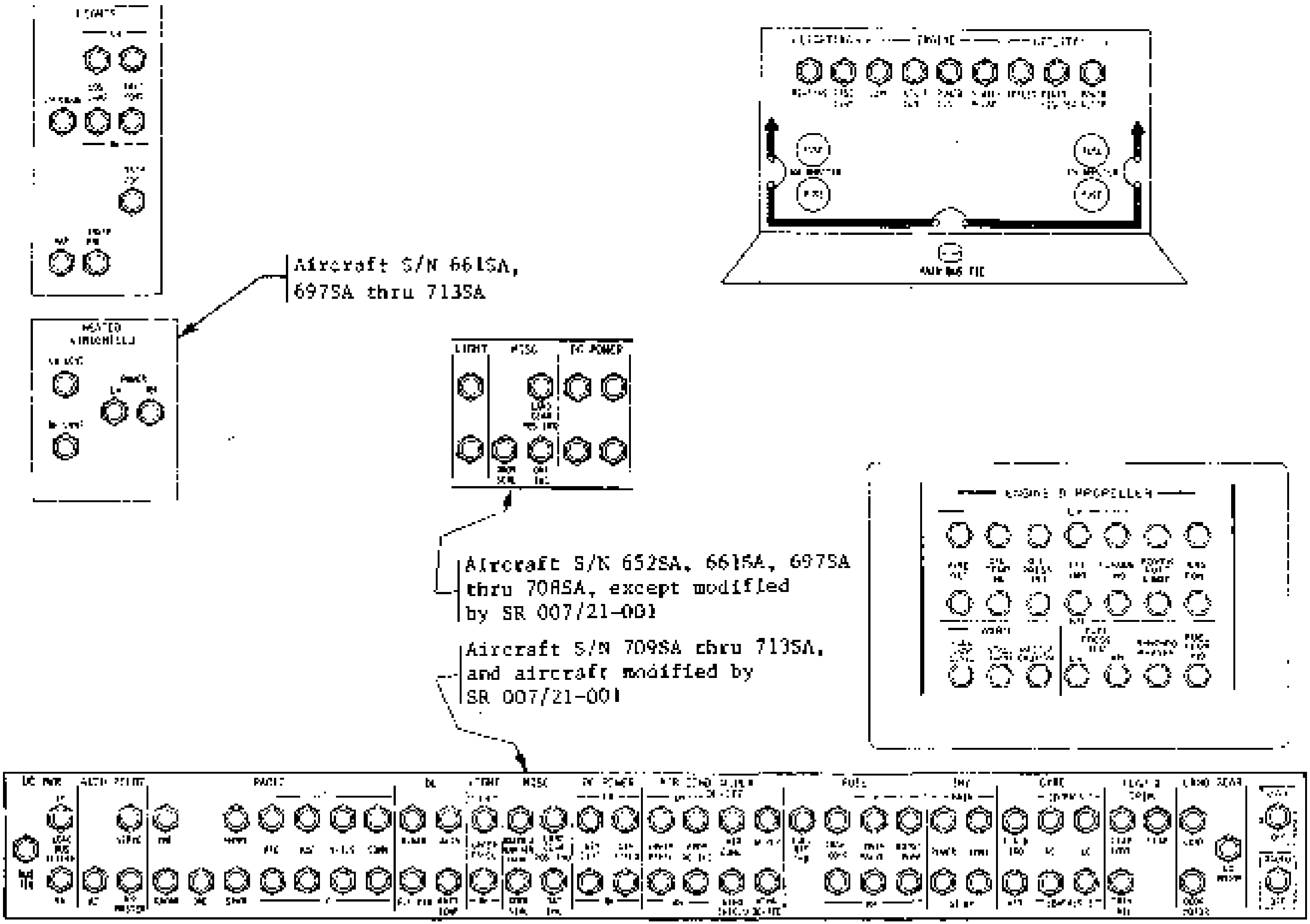


Fig. 13-6 Circuit breaker panel (1/2)
Aircraft S/N 652SA, 661SA, 697SA thru 713SA

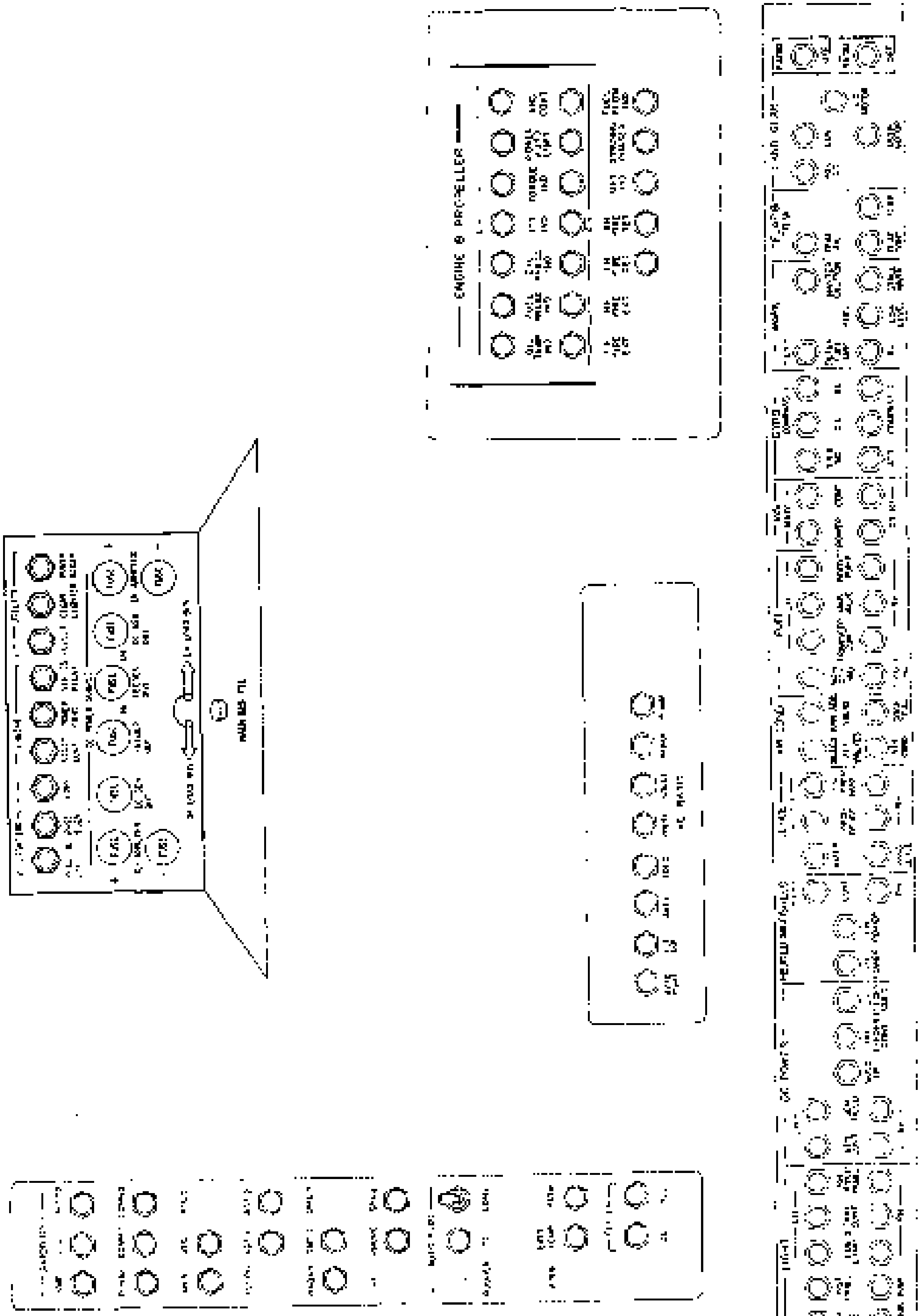
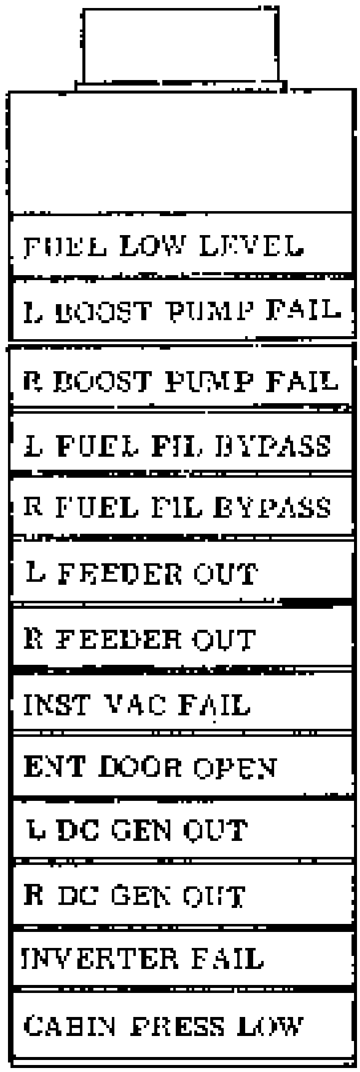


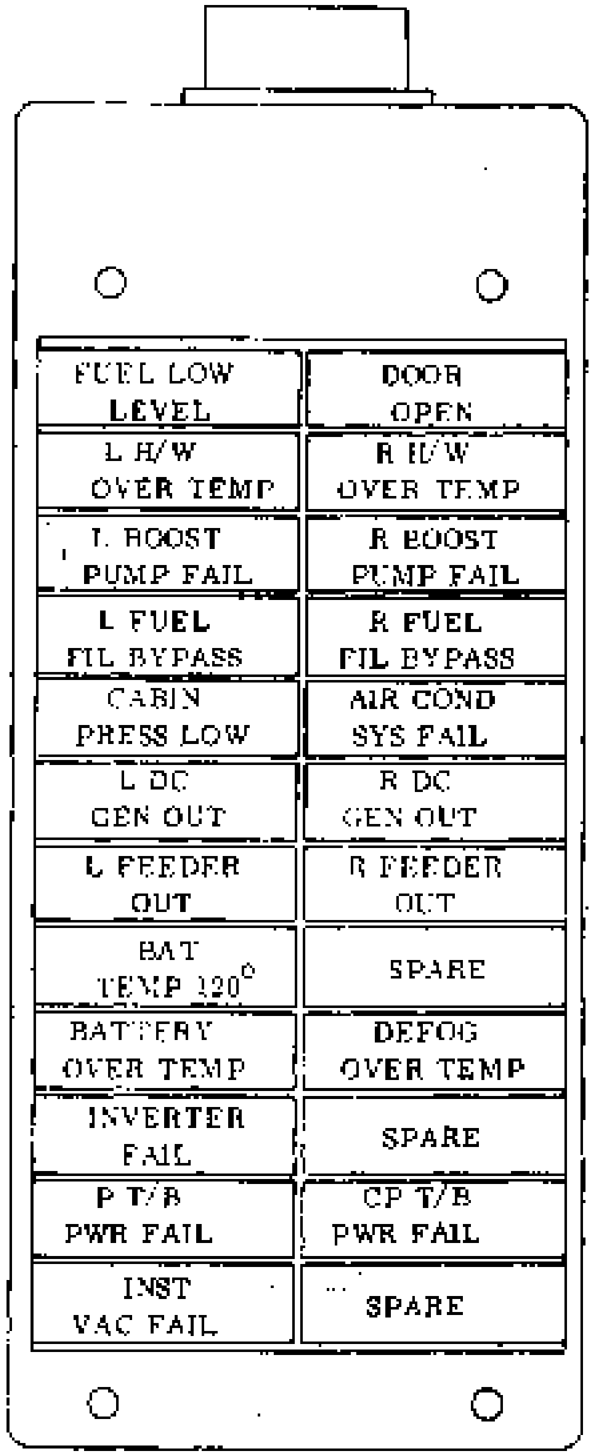
Fig. 13-6 Circuit breaker panel (2/2)

Aircraft S/N 714SA and subsequent



Aircraft S/N 652SA and 661SA

Fig. 13-7 Annunciator panel (1/2)



Aircraft S/N 697SA and subsequent

Fig 13-7 Annunciator panel (2/2)

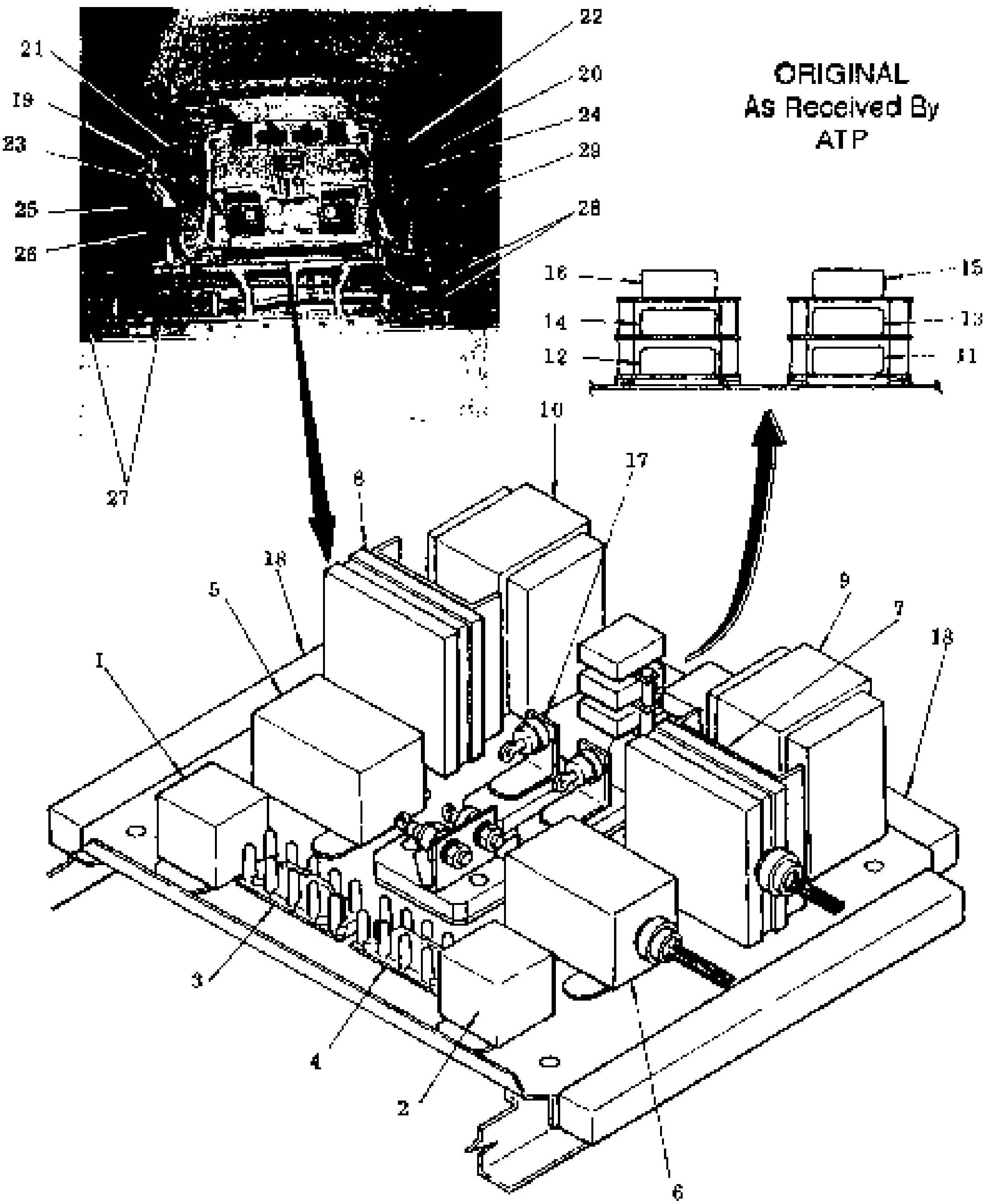
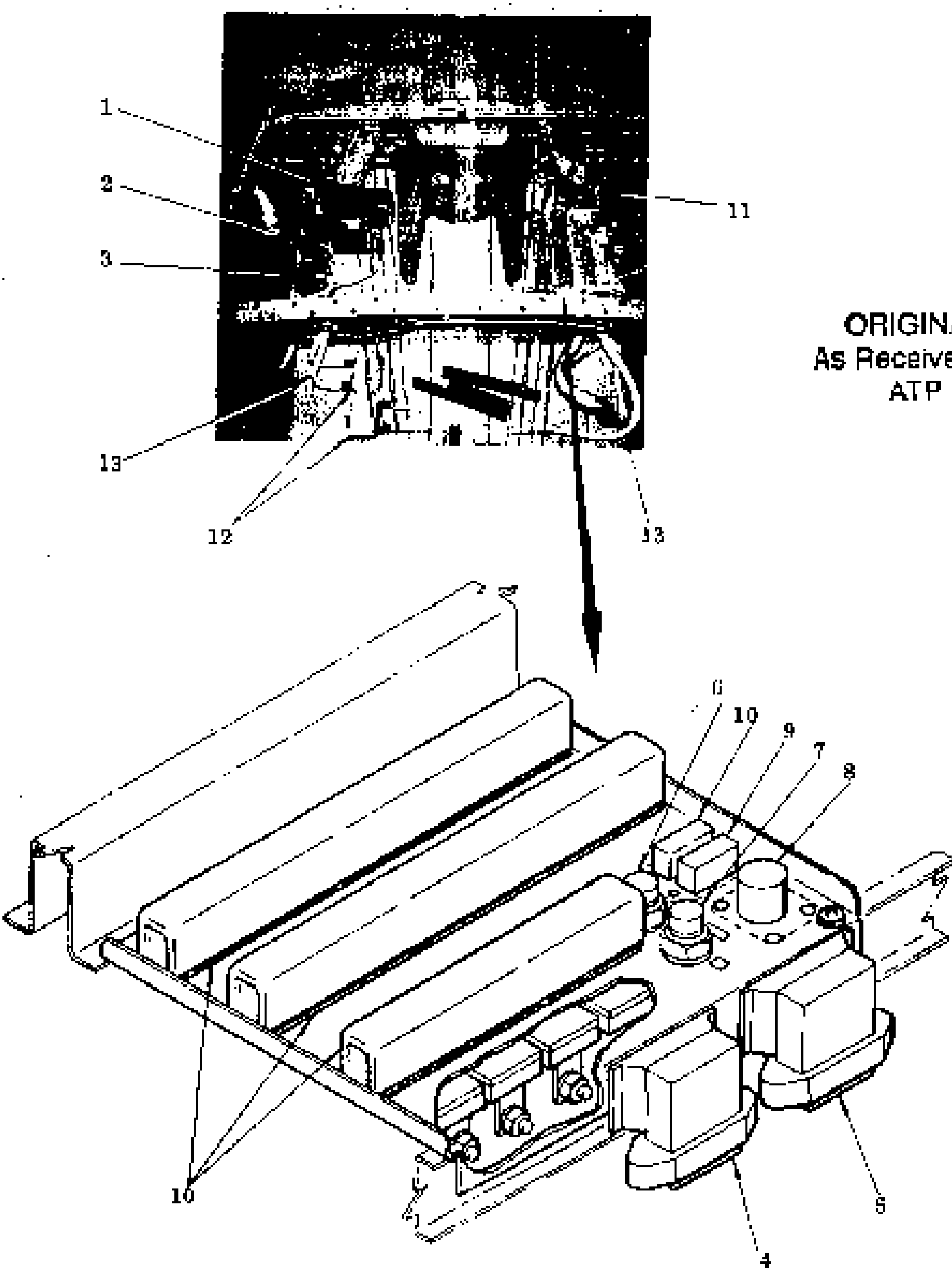


Fig. 13-8 Electrical [illegible] center wing leading edge

1. LH trim aileron emergency relay
2. RH trim aileron emergency relay
3. LH rectifier for NTS check
4. RH rectifier for NST check
5. LH engine fire detecting control unit
6. RH engine fire detecting control unit
7. LH engine speed switch
8. RH engine speed switch
9. LH tip tank fuel transfer control relay
10. RH tip tank fuel transfer control relay
11. LH tip tank fuel level relay
12. RH tip tank fuel level relay
13. LH tip tank fuel transfer relay
14. RH tip tank fuel transfer relay
15. LH outer tank fuel transfer relay
16. RH outer tank fuel transfer relay
17. Rectifier for aux. tank fuel transfer control
18. Terminal board
19. LH wing-fuselage joint wire connector
20. RH wing-fuselage joint wire connector
21. LH chromel-alumel wire connector
22. RH chromel-alumel wire connector
23. LH tip tank fuel qty. indicator coaxial wire connector
24. RH tip tank fuel qty. indicator coaxial wire connector
25. Wing tank fuel qty. indicator coaxial wire connector
26. Signal condition unit
27. Fuel shutoff valve
28. Boost pump fuel pressure warning switch
29. Tip tank pressure air shutoff valve

Key to Fig. 13-8



ORIGINAL
As Received By
ATP

Fig. 13-9 Electrical system in center wing trailing edge

1. Flap motor
2. Flap up relay
3. Flap down relay
4. LH oil cooler air intake heater control relay
5. RH oil cooler air intake heater control relay
6. LH oil cooler air intake heater indication light fuse
7. RH oil cooler air intake heater indication light fuse
8. Flap pulse drive delay relay
9. Flap pulse drive control relay
10. Flap 20° control relay
11. Stall warning system flap position transmitter
12. Terminal-starter generator wiring, wing-fuselage joint
13. Starter generator wiring

Key to Fig. 13-9

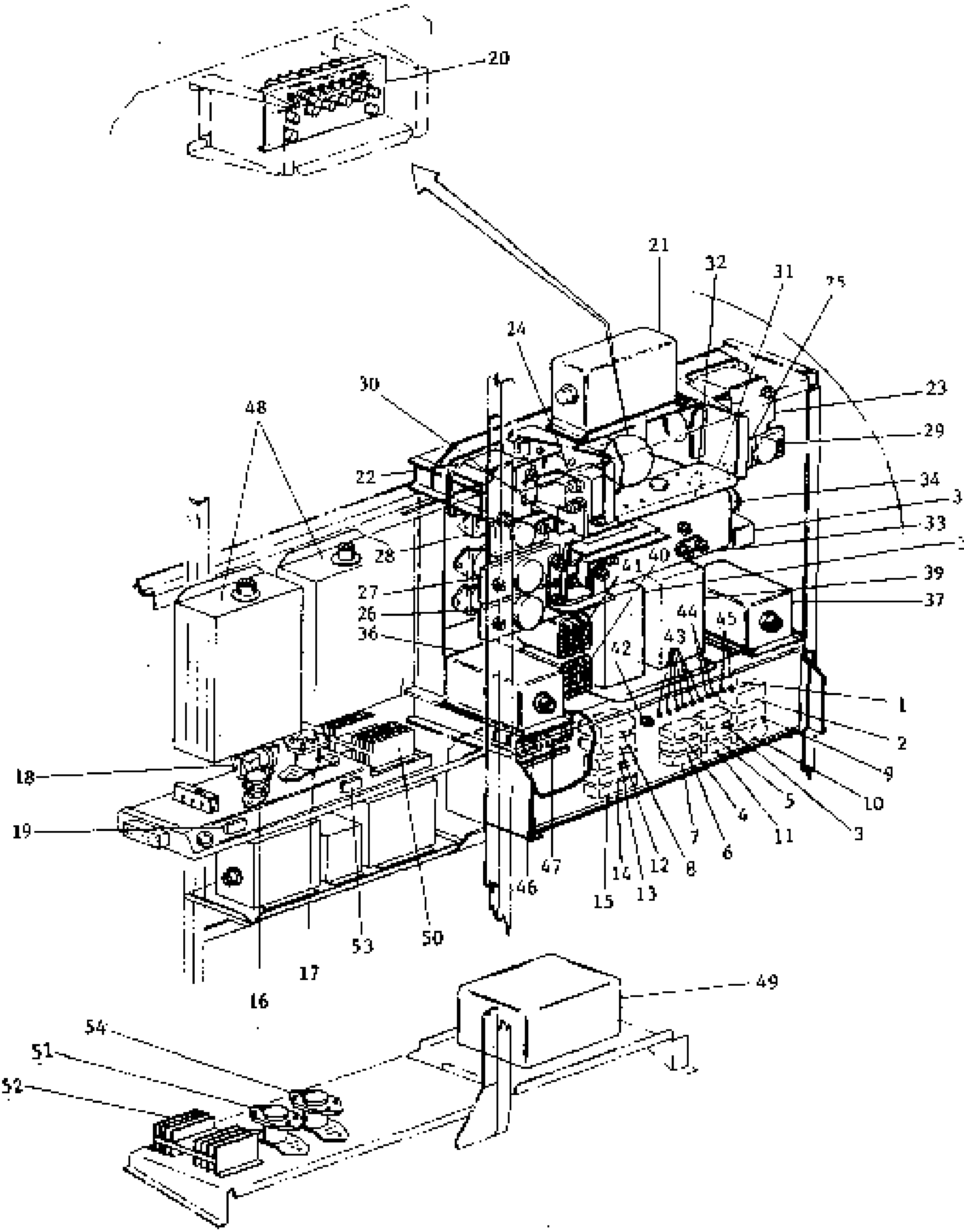


Fig. 13-10 Main junction box

1. LH starter auxiliary relay
2. RH starter auxiliary relay
3. LH ignition relay
4. RH ignition relay
5. LH start select relay
6. RH start select relay
7. External power control relay
8. Engine start control relay
9. Landing gear safety relay
10. Air conditioning room temp. auto-control relay
11. Air conditioning room temp. manual-control relay
12. Flap delay relay
13. Flap takeoff position control relay
14. Flap pulse drive control relay
15. Fuel low level relay
16. Main inverter relay
17. Converter relay
18. Inverter failure detecting relay
19. Instrument power exchange relay
20. Circuit breaker panel
21. Signal summing unit
22. RH reverse current cutout relay
23. LH reverse current cutout relay
24. RH shunt
25. LH shunt
26. RH starter relay
27. LH starter relay
28. RH feeder relay
29. LH feeder relay
30. RH feeder overload sensor
31. LH feeder overload sensor
32. L & R main bus tie circuit breaker
33. Series paralleling relay
34. Series parallel control relay
35. Monitored bus control relay
36. RH voltage regulator
37. LH voltage regulator
38. RH overvoltage relay
39. LH overvoltage relay
40. RH voltage regulator control relay
41. LH voltage regulator control relay
42. Voltage detector adjusting resistor
43. Voltage detector adjusting test terminal
44. RH generator voltage adjusting test terminal
45. LH generator voltage adjusting test terminal
46. Voltage detector
47. Voltege detector condenser
48. Inverters
49. Strobe light power supply
50. Main inverter diodes
51. Standby inverter relay
52. Standby inverter diodes
53. Inverter select relay
54. Cabin lighting relay

Key to Fig. 13-10

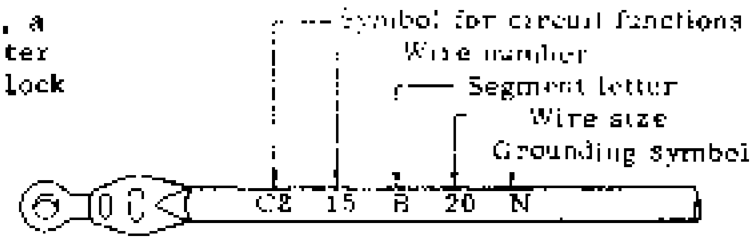
1.3 AIRPLANE WIRING

1.3.1 GENERAL DESCRIPTION

Wire bundles in the airplane are routed as follows: Wiring from each system in the forward fuselage passes through the forward pressure bulkhead, and is connected to the connection box on the lower front of the instrument panel. Wire bundles from the instrument panels and switch panels and wiring around the cockpit are concentraced and connected together in the connection box. Wiring between the connection box, annunciator panel, circuit breaker panel, bulge, left upper switch panel and LH wing leading edge disconnect plug is made through LH side panel. Wiring between the connection box, right upper switch panel, RH wing leading edge, main junction box and aft fuselage is made through RH side panel. Wiring for engine and propeller control, wing tip light and fuel control is connected to LH and RH disconnect plug in front of the center wing through wing leading edge. Wiring between the starter generator and main junction box passes through the wing trailing edge, center wing rear section and aft passenger compartment ceiling. Feeder is distributed from the main junction box to the circuit breaker panel through the aft passenger compartment ceiling and LH side panel.

1.3.2 WIRING IDENTIFICATION MARKING

Wire identification markings are stamped on all wires throughout the airplane. These identification markings consist of a circuit function letter which shows system to which it belongs, a wire number, a segment letter provided for each wiring block and a wire size number.



CIRCUIT FUNCTION CODES

CIRCUIT FUNCTION CODE	CIRCUIT FUNCTION/SYSTEM
C	Flight Controls
C1	Flap Control
C2	Trim Aileron Control
CW	Control Wheel
D	Instruments (Except Flight and Engine)
D1	O.A.T., Outside Air Temperature
D3	Trim Aileron Position Indication
E	Engine Instruments
E1	Engine Instruments
E2	Fuel Quantity
F	Flight Instruments
F1	Turn and Bank Indication
F3	Pitot and Static Port Anti-Ice
G	Landing Gear
G1	Landing Gear Control
G2	Landing Gear Position Indication
H	Heating, Cooling and Anti-Ice
H1	Engine Anti-Ice, Propeller De-Ice
H2	Wing De-Ice
H3	Air Conditioning
H4	Door Seal
H5	Heated Windshield - Anti-Ice
H6	Oil Cooler Inlet - Anti-Ice
K	Engine and Propeller Control
K1	Engine and Propeller Control
K2	Engine and Propeller Control
K7	Propeller Synchrophaser
K9	Propeller Synchrophaser
L	Lighting
L1	Navigation Lights
L2	Fuselage Landing Lights
L3	Room Lights
L4	Instrument and Switch Panel Lights
L5	Trim Indication Lights
L6	Instrument Lights
L7	Strobe Lights
L8	Tip Tank Taxi Lights

CIRCUIT FUNCTION CODE	CIRCUIT FUNCTION/SYSTEM
M	Miscellaneous Electric
M1	Windshield Wiper Control
M2	Toilet
M3	Cigar Lighter
P	DC Power
P1	DC Power
P2	DC Power
P5	DC Power
Q	Fuel and Oil
Q1	Main Tank Fuel Control
Q2	Fuel Transfer Control
Q3	Fuel Transfer Control
Y	DC Power and DC Control for AC
Y1	AC Power Source
Y2	AC Power Source
X	AC Power
X1	AC Power Source
X2	AC Power Source
W	Warning and Emergency (Master Caution)
W1	Vacuum Pressure Warn
W2	Door Lock Warn, Master Caution
W3	Fuel Pressure Indication/Warn
W4	Stall Warn
W5	Engine Fire Detect
W6	Engine Fire Detect
W8	Fuel Filter Bypass/Boost Pump Failure
W9	Cabin Pressure Warn
W20	Battery Overtemp
W21	Defog Overtemp
BT	Battery Overtemp
FPM	Fuel Pressure Meter
OC	Oil Cooler
HWS	Heated Windshield
TTL	Tip Tank Light (Taxi)
P	Power

1.3.3 WIRE SPECIFICATIONS

Wires in the airplane have been selected according to the current and nature of the circuit and installation in the airplane. When the wire is replaced, wire of the same type and size should be used. General policies covering the wire selection are as follows.

Specification	Use
MIL-W-5086 Type II	General airplane wiring
MIL-W-8777 (MS27110)	Wiring within engine nacelle
MIL-W-5846 Type II	Wiring for engine exhaust gas temperature thermocouple
MIL-W-7078 Type II	Wiring required for static shielding
MIL-C-17/78 Type RG-187/U	Fuel quantity indicating circuits
Mitsubishi Specification M9013	Wiring in flap sliding section of aileron trim system and cockpit connection box
Mitsubishi Specification M9007 & M9014	Wiring required for static shielding in engine nacelle

1.4. INSPECTION AND MAINTENANCE OF ELECTRICAL SYSTEM

Inspection and preventive maintenance of the electrical system should be accomplished, thereby minimizing probability of failure.

(1) Check the wiring for fuel or oil contamination.

(2) Check all terminal blocks for loose connections. Check adjacent terminals for contact and for short circuits due to foreign material.

(3) Check all units for security of installation.
Check wiring for security and for evidence of overheating or damage.

(4) Check all bonding for positive installation electrically and freedom from corrosion.

(5) Check operation of each system.

NOTE

Electrical bonding is required for safe and steady flow of load current, preventing radio noise from occuring, and for protecting the airplane from lightning.

1.5 TROUBLE SHOOTING FOR ELECTRICAL SYSTEM

1.5.1 PROCEDURES FOR TROUBLE SHOOTING

Trouble shooting the electrical system can be accomplished in many ways. The following is an example of a practical and rapid trouble shooting method. Trouble after 50 hours of service is mostly due to defects of individual parts or units rather than defective wiring. Therefore such items should be checked before checking wiring.

(1) Referring to wiring diagram, determine that there is specified voltage between terminals.

(2) If a unit involved does not operate though there is voltage between terminals, check grounding and bonding. If normal, replace the unit.

(3) If there is no voltage between terminals, check circuit breaker, fuses and switches.

(4) If voltage is still absent between terminals, finally check the wiring.
For continuity check, disconnect all power sources and disconnect plug for system which is being checked.
Check continuity between plugs, wiring for contact with other wiring or airframe structure and for leakage through insulation.

2. DC POWER SUPPLY SYSTEM

2.1 GENERAL DESCRIPTION (See Fig. 13-11)

In the DC power supply system, two batteries are provided in order to facilitate engine starting. This system is operated as follows. When the battery key switch and battery switch on the LH sub-panel are turned to ON and the series parallel switch is in parallel, the two batteries are connected in parallel to the DC power system. If the engine start switch on the center pedestal switch panel is pushed down with the battery selector switch in parallel position, the batteries are connected in parallel to the starter and engine start.

If the engine start switch is pushed down with the battery selector switch in series position, the batteries are automatically connected in series to the starter and engine start. When an engine reaches about 50% of RPM and starting completes, the batteries return to a parallel connection with the DC power system. The ignitors, fuel shutoff valve, speed switch and inverter get power from the starter bus during engine starting. Since the voltage of the starter bus exceeds 30V sometimes in battery series starting, the voltage detector is installed in the main junction box. When the voltage exceeds 29V, the power source is automatically turned to the main bus to protect ignitor and inverter from overvoltage.

When the generator control switch DC GEN on the LH switch panel is turned to ON after-starting engines, LH and RH generators are connected to the main bus respectively through the reverse current cutout relay. The MASTER switch is used to cut off all power sources in case of emergency, such as an electrical fire.

Generator voltage is controlled at 28.5V by the voltage regulator. In order to balance load current to two generators, voltage drop of generator ground winding is utilized. This voltage controls the generator through a voltage regulator so that the difference in current load of both generators is kept to a minimum. To protect electrical equipment from unusually high voltage of the generators, an overvoltage relay is provided in the main junction box. When a generator involves high voltage due to failure, the relay automatically disconnects the generator from the aircraft electrical bus. To restore the generator to normal operation, momentarily turn the generator control switch from OFF to RESET position and return to ON position. Ammeter shunt of 50mv/300A rating shows the current load on the LH switch panel ammeter. When the generator voltage becomes lower than the main bus voltage in the main junction box due to failure of voltage regulator or generator circuit and a reverse current of 9 to 25A is generated, the reverse current relay opens, disconnecting a generator from the main bus.

When the generator voltage increases, the relay closes, connecting the generator to the bus. The nickel cadmium battery will withstand the high charging current at the initial stage of charging, depending on discharged condition. The charging rate usually exceeds 300A after an engine is started by battery. The charging rate decreases gradually as time elapses and usually becomes less than 50A in a few minutes.

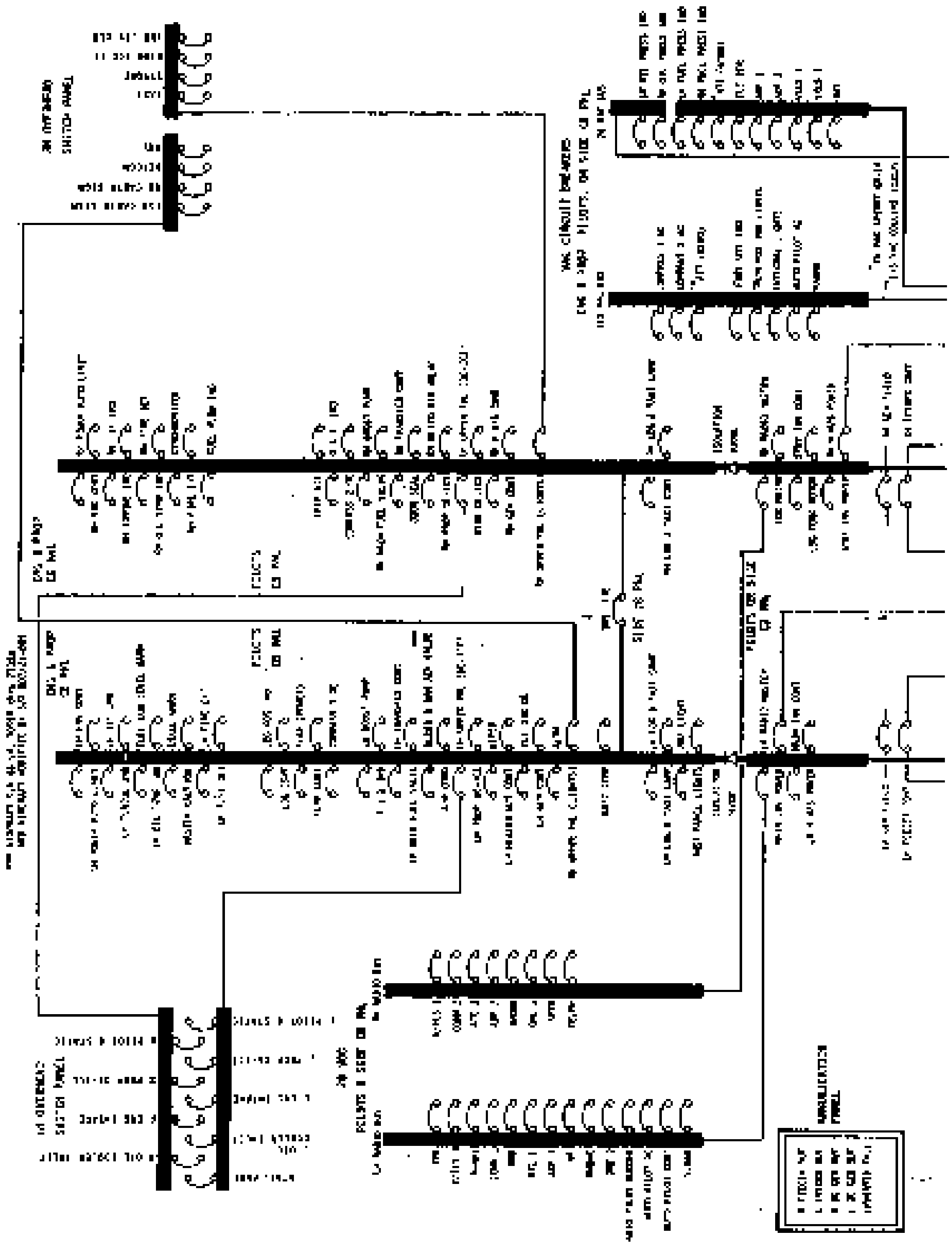


Fig. 13-11 Power distribution (1/4)
(Typical S/N 652SA, 661SA, 697SA thru 713SA)

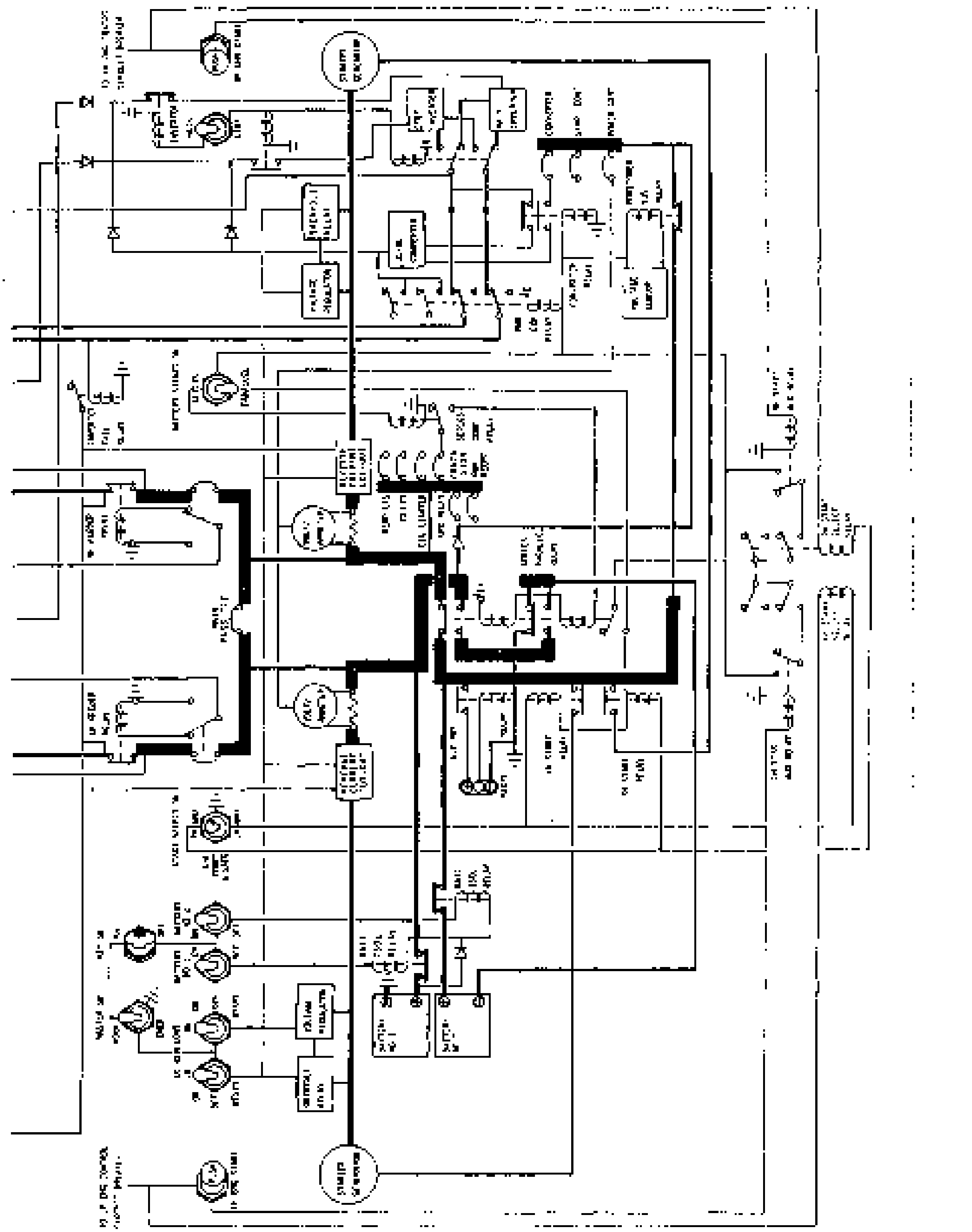


Fig. 13-11 Power distribution (2/4)
(Typical S/N 652SA, 661SA, 697SA thru 713SA)

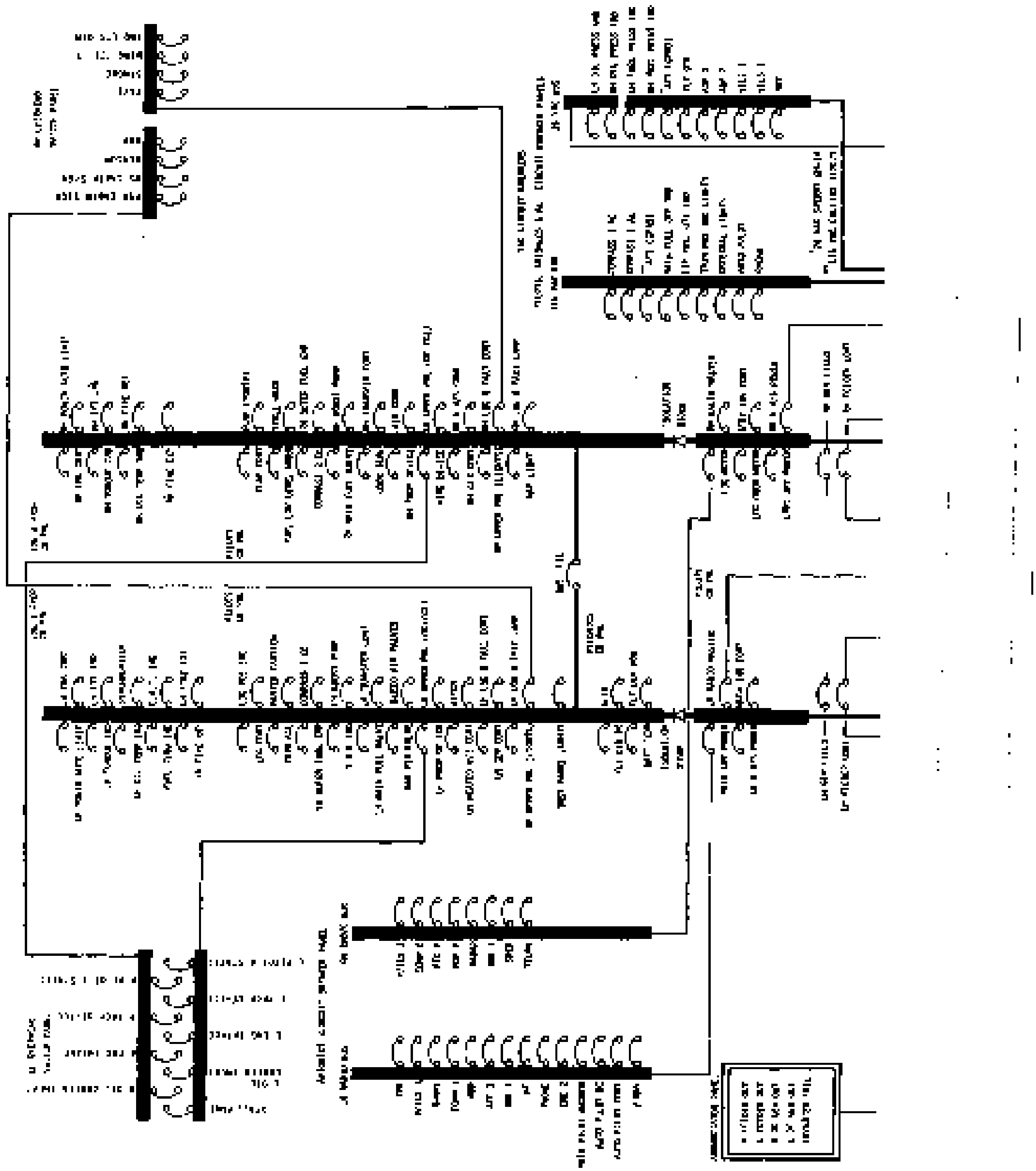


Fig. 13-11 Power distribution (3/4)
(Typical S/N 714SA and subsequent)

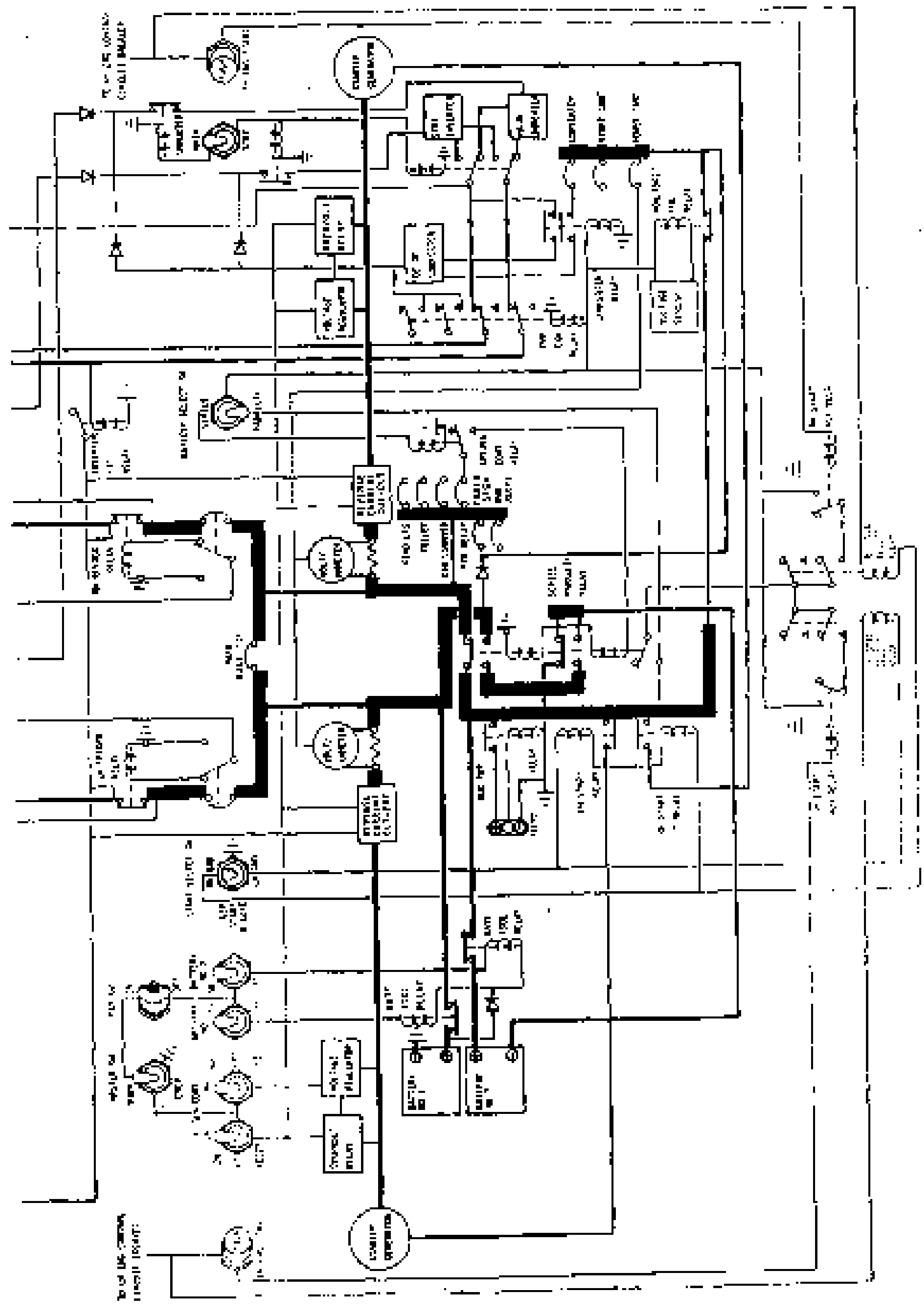


Fig. 13-11 Power distribution (4/4)
(Typical S/N 714SA and subsequent)

2.2 DC GENERATOR

The DC generator is installed on the LH upper side of the engine gear box. The generator is coupled to the engine rotor with a gear ratio of 1:0.261. Excitation of the field of the generator is by shunt and voltage is controlled at 28.5V by a voltage regulator in engine flight idle condition. The generator is cooled by a built-in cooling fan and by cooling air from a scoop on the nacelle outer skin.

2.2.1 REMOVAL

(1) Remove 2 hoses from oil line leading to oil cooler.

(2) Remove electric connector (P4001) from wing leading edge.

(3) Remove hoses of fuel line from wing leading edge (RH only).

(4) Remove 2 oil hoses of engine torque pressure line from engine.

(5) Remove wiring from terminal block of generator. Attach identification marks to each wire to facilitate reinstallation.

(6) Unclamp Marman clamp. Remove generator from engine mounting pad.

(7) Remove 6 nuts from engine mounting pad, if necessary.

NOTE

Care should be taken not to have protecting cage of cooling fan deformed when removing generator.

2.2.2 INSTALLATION (See Fig. 13-12)

(1) Mount generator in accordance with 3 guide pins on bracket of engine mounting pad. Clamp with Marman clamp. Tighten 6 nuts to 120 to 130 in-lbs. (137 to 149 kg/cm) for attaching bracket to engine mount. Tighten nut on one of the bolts attaching bracket together with ground wire from terminal block E of generator. Tighten Marman clamp to a torque value of approximately 50 in-lbs. (58 kg/cm).

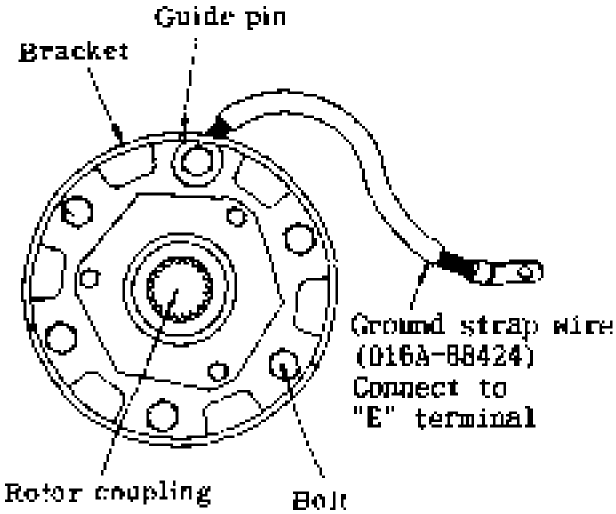
(2) Connect wires securely to each terminal of terminal block of generator.

(3) Connect 2 hoses of engine torque pressure line.

(4) Connect 2 hoses of oil line leading to oil cooler.

(5) Connect hose of fuel line (RH only).

(6) Connect electric connector (P4001).



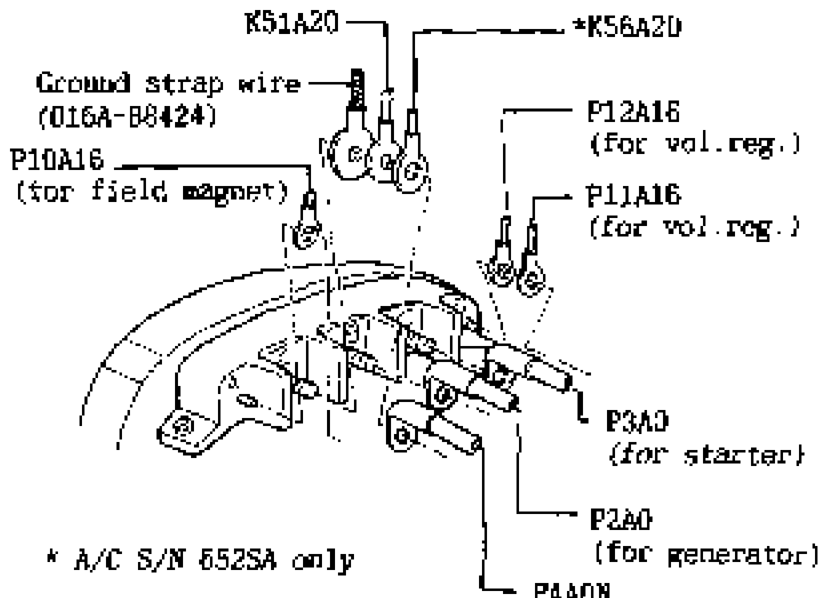


Fig.13-12 Wiring of DC Generator

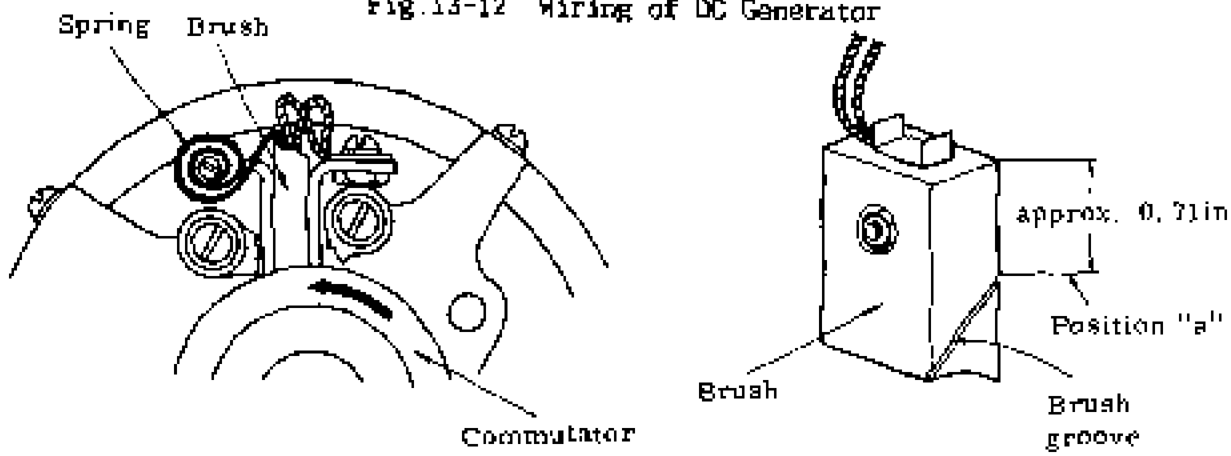


Fig.13-13 Generator brush

2.2.3 INSPECTION (See Fig. 13-13)

(1) Open nacelle cover and check wiring for security of attachment to corresponding terminals.

(2) Check brushes for position and smooth operation in guide fixtures. Check for excessive play, cracks or damage. If brush measures less than point "a" of Fig. 13-13, it should be replaced with a new one.

(3) Check surface of commutator for cleanliness and freedom from grease or other fluids. Blow off carbon powder by dry pressure air.

2.3 VOLTAGE REGULATOR

The voltage regulator is installed in the main junction box and maintains power source voltage 28.5V regardless of loaded current. This voltage regulation is accomplished automatically,changing exciting current in the generator field coil. In addition to the voltage regulating circuit, the load current equalizing circuit is provided. This circuit equalizes the load current of the two generators by changing the generator output voltage.

2.4 REVERSE CURRENT CUTOUT RELAY

Two differential type reverse current cutout relays are installed in the main junction box. These relays connect the generators to the main bus when the generator switch is turned to ON. Under this condition, load current should be flowing in the circuit and generator voltage should be higher than main bus voltage by 0.35 to 0.65V. If generator voltage should decrease and 9 to 25A of the reverse current flows through the relay, the generator is disconnected from the main bus. At the same time, an auxiliary contact of the relay closes, illuminating L(R) DC GEN OUT on the annunciator panel.

2.5 OVERVOLTAGE PROTECTING RELAY

The overvoltage protecting relay is installed in the main junction box. This relay protects equipment from high voltage of the generator by automatically cutting off generator output from the main bus, if the generator happens to generate unusually high voltage. The circuit can be restored to the original condition by the generator switch turned to RESET from OFF, then returned to ON. Trip voltage of this relay is 32 to 34V.

2.6 VOLTAGE SENSING UNIT

The voltage sensing unit is installed on the inside of the relay box in the main junction box. When the voltage of starter bus exceeds 29V while starting engine, the sensing unit actuates the monitored bus control relay to disconnect the monitored bus and the inverter from the starter bus.

2.7 CHECKOUT AND ADJUSTMENT OF DC POWER SYSTEM

Voltage adjustment and parallel operation adjustment should be performed when generator or voltage regulator is replaced.

2.7.1 VOLTAGE ADJUSTMENT

(1) Close the circuit breakers of engine control, DC power, AC power, engine instrument and fuel control systems on the circuit breaker panel in cockpit and the main junction box.

(2) Perform engine warm up more than 2 min. with 96% RPM operation.

(3) Hold battery switch and battery key switch in ON positions.

(4) Turn the left and right generator switches in LH switch panel to OFF.

(5) Connect the terminal GEN of LH reverse current cutout relay in the main junction box to "+" terminal of an accurate voltmeter (30VDC Class 0.5) and "–" terminal to the airframe for ground.

(6) Turn the left generator switch to ON.

(7) Turn the adjusting screw of voltage regulator slowly with a screw driver (clockwise to increase voltage, counterclockwise to reduce voltage) and adjust the voltmeter to read 28.5V.

(8) Turn the left generator switch to OFF.

(9) Connect the terminal GEN of RH reverse current cutout relay in the main junction box to "+" terminal of the voltmeter and "–" terminal to the airframe for ground.

(10) Turn the right generator switch to ON.

(11) Turn the adjusting screw of voltage regulator slowly with a screw driver and adjust the voltmeter to read 28.5V.

(12) Turn the right generator switch to OFF.

NOTE

Ensure no fluctuation in voltage in five minutes operation of engine after adjustment.

2.7.2 PARALLEL OPERATION ADJUSTMENT

Equalize load current of LH and RH generators in the following manner after voltage adjustment is performed in accordance with Para. 2.7.1.

(1) Close circuit breaker MASTER CAUTION.

(2) To conduct tie in test of LH generator, turn right generator switch to ON. R DC GEN OUT on annunciator panel goes off.

(3) Turn left generator switch to ON. L DC GEN OUT on annunciator panel goes off.

(4) Turn left generator switch to OFF. L DC GEN OUT on annunciator panel illuminates.

(5) Turn left generator switch on and off repeatedly, and verify L DC GEN OUT going off and illuminating.

(6) To conduct tie in test of RH generator, turn left generator switch to ON. L DC GEN OUT on annunciator panel goes off.

(7) Turn right generator switch to ON. R DC GEN OUT on annunciator panel goes off.

(8) Turn right generator switch to OFF. R DC GEN OUT on annunciator panel illuminates.

(9) Turn right generator switch on and off repeatedly, and verify R DC GEN OUT going off and illuminating.

(10) To conduct tie in tests of LH and RH generators simultaneously, turn left generator and right generator switches ON or OFF simultaneously. L and R DC GEN OUT on annunciator panel simultaneously go off or illuminate, respectively.

NOTE

(i) Apply 20 to 40A load to generator for performing parallel operation adjustments of generator. It is recommended to operate inverter, landing lights, anti-collision lights and cabin lights.

(ii) If the test under load does not result in proper tie in, that is, L DC GEN OUT on annunciator panel does not go off with left generator switch ON or R DC GEN OUT illuminates with the switch OFF, the output voltage must be regulated by means of voltage regulator. Turn voltage regulator adjusting knob for generator with greater current slightly counterclockwise; for decreasing output voltage, turn the opposite regulator knob slightly clockwise for balancing. Confirm that voltage measured at main bus in the main junction box is 28 to 28.5V.

2.7.3 OPERATIONAL CHECK OF VOLT-AMMETER

After conducting parallel operation adjustment of LH and RH generators under engine operation at constant speed, perform operational check of volt-ammeters in accordance with the following procedures with left generator and right generator switches concurrently in tie in condition.

(1) LH and RH volt-ammeters must read about 28V.

(2) To perform "0" load check, turn all circuit breakers OFF except MAIN INV POWER, MAIN INV CONT, LH ITT IND, RH ITT IND, LH FIRE DET, RH FIRE DET, LH GEN CONT and RH GEN CONT in cockpit circuit breaker panel and LH GEN FIELD and RH GEN FIELD in the main junction box circuit breaker panel, and turn battery switch OFF to make electrical load to 0. (ref. value). LH and RH volt-ammeters should read zero (ref. value).

(3) To perform load check, operate inverter, landing light, anti-collision light and cabin lights to an aggregate load of about 50A. The differential amperage between LH and RH volt-ammeters should be less than 5A (half graduation).

2.7.4 CHECKOUT OF VOLTAGE VARIATION

(1) When "0" load check is performed (see Para. 2.7.3), set engine RPM to 100% and operate inverter, illuminate landing lights, anti-collision light and cabin lights to give a load of about 50A.

(2) Set engine RPM to 65% (*1), 75% (*2) and operate inverter, illuminate landing lights, anti-collision light and cabin lights to give a load of about 50A.

(3) Make sure that voltage variation in the above tests should be -1.0 to +0.5V.

*1 Aircraft S/N 652SA
*2 Aircraft S/N 661SA, 697SA and subsequent

2.7.5 OPERATIONAL CHECK OF OVERVOLTAGE RELAY

When the generator, voltage regulator and overvoltage relay are replaced, voltage adjustment, parallel operation adjustment and operational check of the overvoltage relay must be checked.

2.7.5.1 PREPARATION

(1) Connect overvoltage tester to LH and RH voltage regulators. (See Fig. 13-14)
 (a) Remove wires (P141D20 and P101D20) connecting to terminal "G" of reverse current circuit breaker and connect the overvoltage tester between the above wires and terminal "G" as shown below.
 (b) Turn tester field rheostat RF1 and RF2 counterclockwise fully and set to "0" ohm.

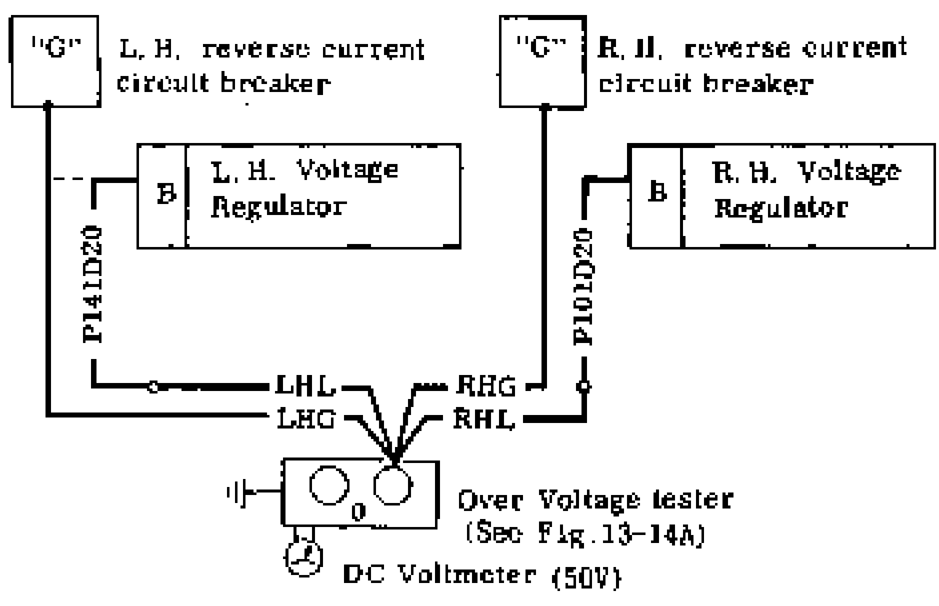
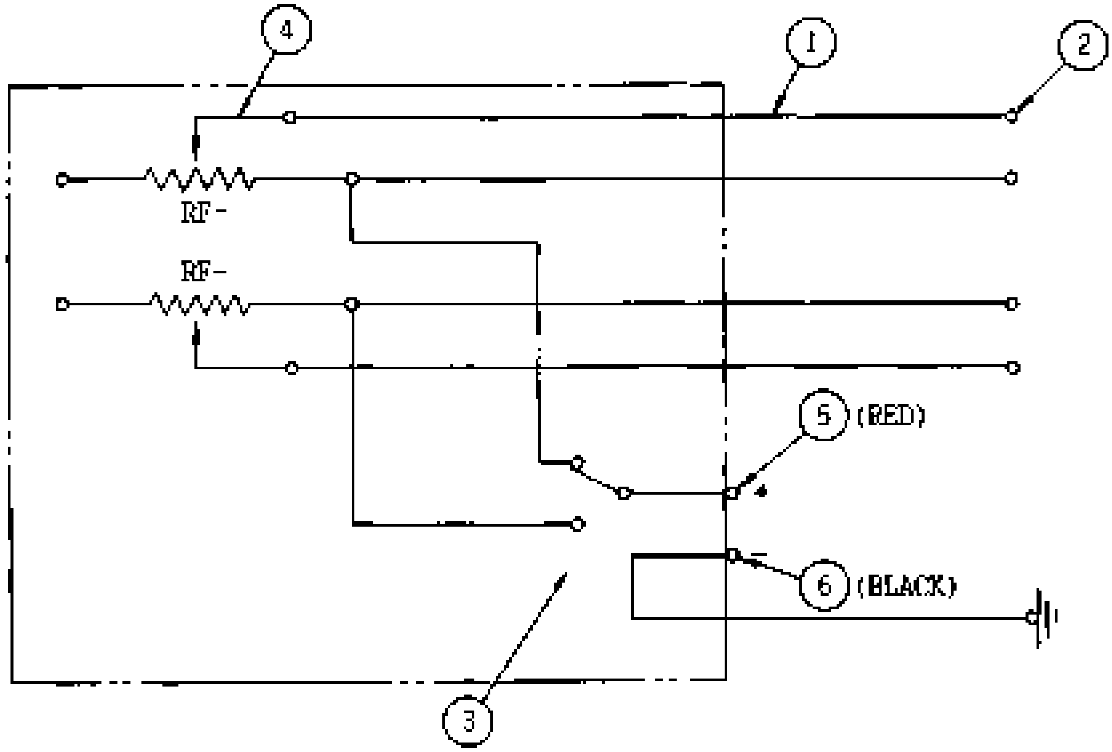


Fig. 13-14 Connection of overvoltage tester



1. Wire, MIL-W-5086/2 AWG #18 or equivalent
2. Terminal lug, MS25036-6 or equivalent
3. Toggle switch(1A), SPDT type commercially available
4. Variable resister, 0 - 350 ohms 3 watts rating commercially available
5. Insulated red banana jack, commercially available
6. Insulated black banana jack, commercially available

Fig 13-14A Schematic wiring diagram for overvoltage tester

(2) Perform engine warm up more than 2 min. with 96% RPM.

(3) Turn battery switch and battery key switch ON.

(4) Close circuit breakers LH ENG FIRE DET, RH ENG FIRE DET, LH GEN FIELD, RH GEN FIELD, LH GEN CONT and RH GEN CONT. Open other circuit breakers.

(5) Adjust output voltage of LH and RH generators to 28.5 ± 0.5V of tester by means of resistors RF_1 and RF_2.

NOTE

When the circuit breakers are turned off in accordance with paragraph (4) above with the engine in operation at a constant speed, only the tachometer will remain in operation. Use caution when operating the engine in this condition.

2.7.5.2 OPERATIONAL CHECK OF LH OVERVOLTAGE RELAY

(1) To check operation of LH overvoltage relay, place selector switch of volt-ammeter of tester in LH position, and volt-ammeter indicates voltage of LH generator.

(2) Turn left DC GEN and right DC GEN switches simultaneously ON.

(3) Watching volt-ammeter reading, gently turn RF_1 clockwise to reduce excitation resistance and gradually raise voltage. The rising voltage drops from 32V to 34V to 3V or less in about 2 seconds.

NOTE

(i) Never raise voltage to 35V or higher.

(ii) Raise voltage to 30V at a rate of about 1V/sec., and in excess of 30V at a rate of about 0.2V/sec.

(iii) Sometimes the voltage indicator of the volt-ammeter in the aircraft may scale over. The volt-ammeter is well capable of withstanding overvoltage of 32 to 34V.

(4) Reset left DC GEN switch with LH overvoltage relay tripped. Verify that overvoltage relay is re-energized, and generator is not reset.

NOTE

When the overvoltage relay is reset, the volt-ammeter indicates approximately 28V. When the overvoltage relay is not reset, the volt-ammeter indicates below 3V.

(5) Place left DC GEN and right DC GEN switches in OFF position.

(6) Return RF_1 fully counterclockwise to original position.

(7) Place left DC GEN switch alone in RESET position and turn off.

(8) Adjust output voltage of LH generator to 28.5V by means of RF_1.

2.7.5.3 OPERATIONAL CHECK OF RH OVERVOLTAGE RELAY

(1) To check operation of RH overvoltage relay, place selector switch of volt-ammeter of tester in RH position, and volt-ammeter indicates voltage of RH generator.

(2) Turn left DC GEN and right DC GEN switches simultaneously ON.

(3) Watching volt-ammeter reading, gently turn RF_2 clockwise gradually to raise voltage. The rising voltage drops from 32V to 34V to 3V or less in about 2 seconds.

NOTE

Never raise voltage to 35V or higher.

(4) Reset right DC GEN switch with RH overvoltage relay tripped.

(5) Place right DC GEN and left DC GEN switches in OFF position.

(6) Return RF_2 fully counterclockwise to original position.

(7) Place right DC GEN switch alone in RESET position and turn OFF.

(8) Adjust output voltage of RH generator to 28.5V by means of RF_2.

2.7.5.4 OPERATIONAL CHECK OF MASTER SWITCH

(1) Turn master switch on LH switch panel in cockpit upward. Battery relay is de-energized. Overvoltage relay trips. Volt-ammeter in the aircraft indicates zero volts.

(2) Turn the master switch downward.

(3) Place left generator and right generator switches in RESET position, and turn on. Volt-ammeter in the aircraft indicates approximately 28V.

(4) After all operational checks are completed, close circuit breakers required for operating engine at constant speed.

(5) If operation of overvoltage relay is found normal after engine is stopped, disconnect testers from LH and RH voltage regulators. Restore wiring to original condition.

2.7.6 ADJUSTMENT AND CHECK OF VOLTAGE SENSING UNIT

2.7.6.1 REQUIRED EQUIPMENT

Variable power source : DC 20V to 30V	1 set
M : DC Voltmeter 0 to 50V Class 0.5	1 ea.
DS-1 : Light MS25041-6-327 or equivalent	1 ea.
S-1, S-2 : Switch MS35058-23 or equivalent	1 ea.

2.7.6.2 ADJUSTMENT

(1) Set all circuit breakers and switches to OFF.

(2) Connect wires as shown in Fig. 13-15.

(3) Turn the adjusting knob of sensing voltage adjusting resistor (R225) counterclockwise until it stops.

(4) Turn battery key switch and battery switch ON (or connect external power source).

(5) Set the output voltage of the variable power source to 29V.

(6) Turn the switch S-1 ON and turn the knob (R225) clockwise slowly until the light DS-1 comes on.

(7) When the light DS-1 comes on, lock the adjusting knob.

(8) Turn the switch S-1 OFF and set the voltage of the variable power source to 20V.

(9) Turn the switch S-1 ON and raise the voltage of the variable power source at a rate of about 1V/sec. and make sure that the light comes on when the voltage comes to 29 ± 0.5V.

(10) Repeat steps (8) and (9) two or three times.

2.7.6.3 OPERATIONAL CHECK

To check operation of the voltage detector only, take the following steps.

(1) Turn battery key switch and battery switch ON (or connect external power source).

(2) Make sure that the engine start select switch is in the position of AIR START & SAFE.

(3) Set all circuit breakers in the cockpit to OFF.

(4) Connect variable power source, voltmeter M-1, switches S-1 and S-2 and lights DS-1 as shown in Fig. 13-15.

(5) Set the voltage of the variable power source to 24V.

(6) Ascertain operation of voltage detector by illumination of light DS-1 or operating noise of monitored bus control relay when switches S-1 and S-2 are turned to ON.

(7) Disconnect variable power source, voltmeter M-1, switches S-1 and S-2 and light DS-1.

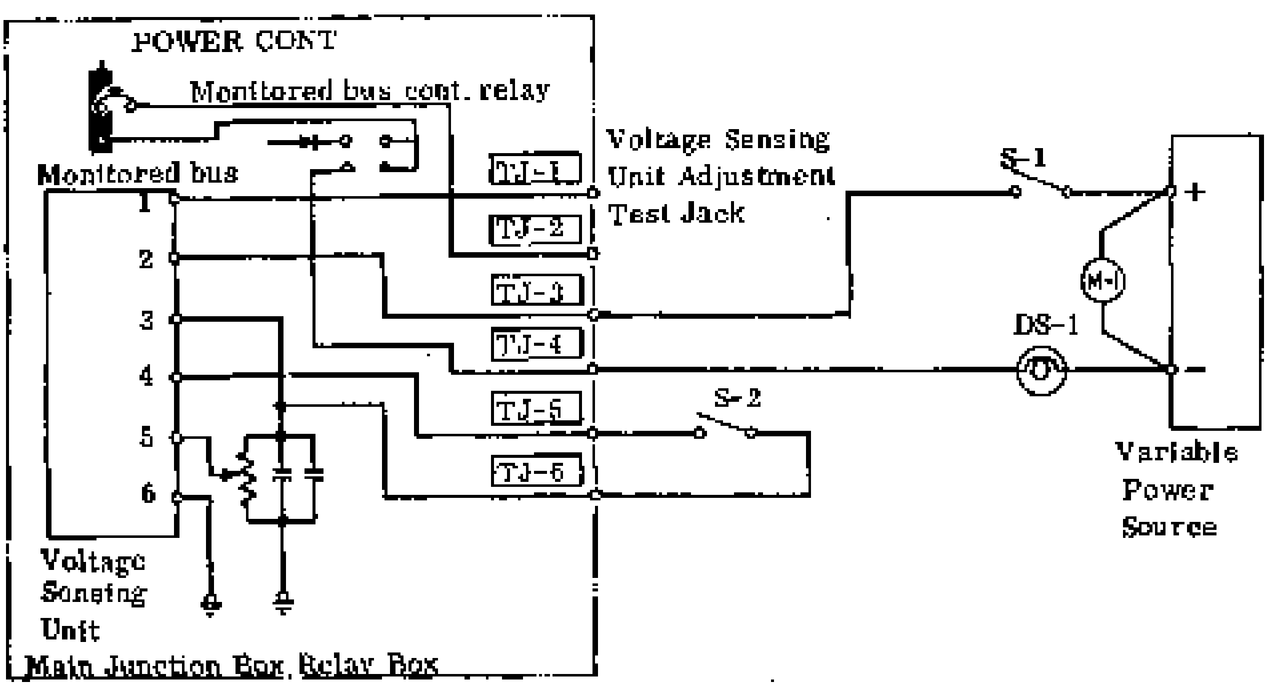


Fig. 13-15 Adjustment of voltage sensing unit (1/2)
(Aircraft S/N 652SA)

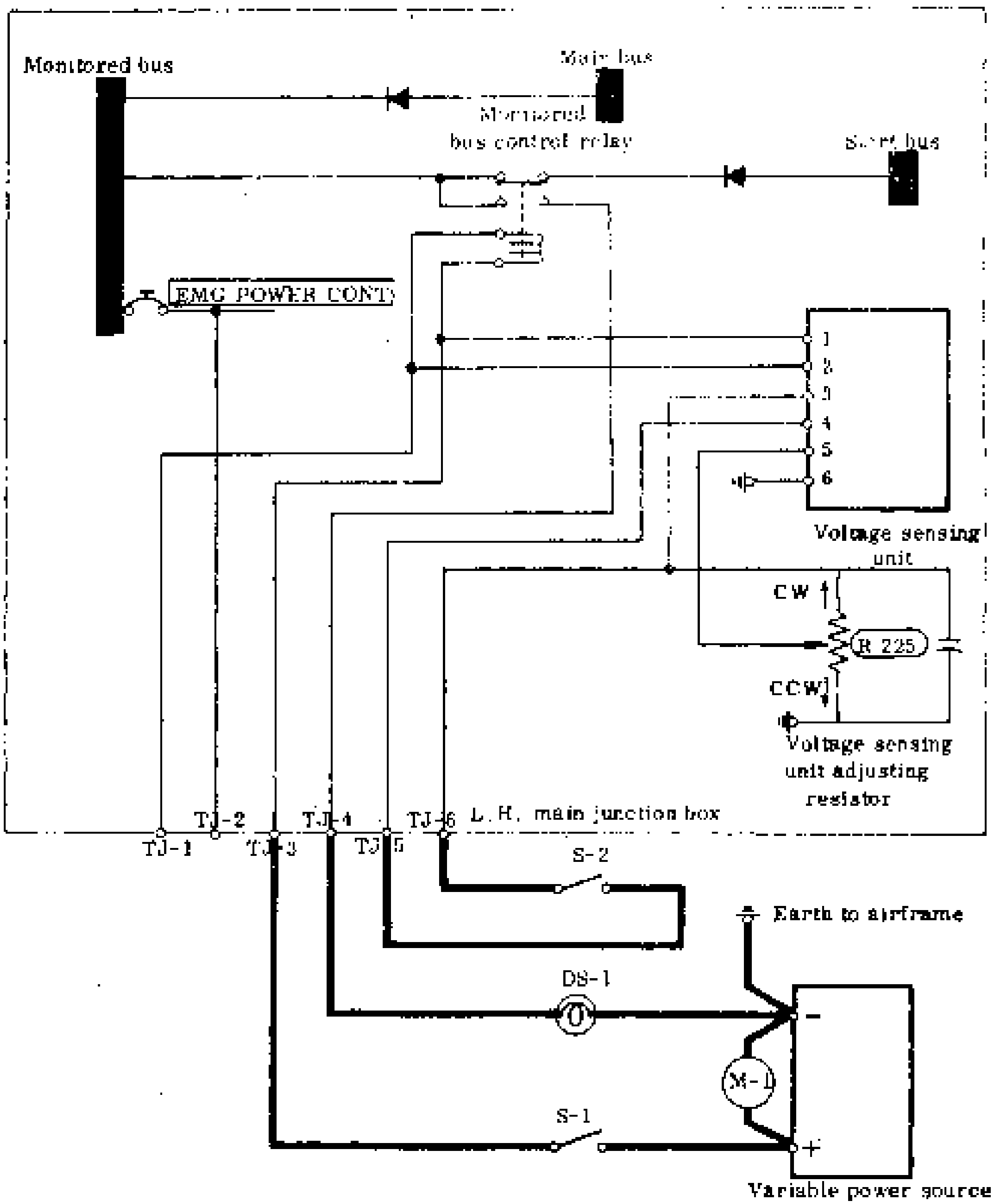


Fig. 13-15 Adjustment of voltage sensing unit (2/2)
(Aircraft S/N 661SA, 697SA and subsequent)

2.8 TROUBLE SHOOTING DC POWER SYSTEM

Trouble	Probable cause	Remedy
(1) No. 1 battery is not connected to airplane when battery key switch and No. 1 battery switch are turned ON	(a) MASTER switch on LH switch panel is in up position.	Set MASTER switch downward.
	(b) No. 1 battery has been discharged or is defective.	Check battery.
	(c) Faulty No. 1 battery relay.	Replace battery relay.
	(d) Faulty contact of No. 1 battery plug.	Clean or replace plug.
	(e) Defective battery key switch, No. 1 battery switch or master switch.	Replace switch.
	(f) Open diode CR303.	Replace diode.
	(g) Faulty wiring.	Check continuity of wire and repair it if necessary.
(2) No. 2 battery is not connected to airplane when battery key switch and No. 2 battery switch are placed in ON	(a) MASTER switch is in up position.	Set MASTER switch downward.
	(b) No. 1 battery has been discharged or defective.	Check No. 1 battery.
	(c) Faulty No. 2 battery relay.	Replace No. 2 battery relay.
	(d) Faulty contact of No. 1 battery plug.	Clean or replace plug.
	(e) Defective battery key switch, No. 1 battery switch or master switch.	Replace switch.
	(f) Open diode CR303.	Replace diode.
	(g) Fusing of fuse F103.	Check wiring for grounding and replace fuse.
	(h) Faulty airplane wiring.	Check wiring for continuity and repair.

Trouble	Probable cause	Remedy
(3) Battery is not disconnected from airplane when battery key switch or battery switch is placed in OFF	(a) Faulty battery key switch or battery switch.	Replace switch.
	(b) Battery relay is stuck.	Replace relay.
	(c) Grounded wire between battery relay and battery key switch.	Check continuity and repair wire.
(4) External power can not be connected	(a) Poor contact of external power receptacle.	Clean receptacle and insert power plug securely.
	(b) Defective external power relay.	Replace relay.
	(c) Defective diode CR301.	Replace diode.
(5) Starter does not start	(a) Faulty starter generator.	Replace starter generator.
	(b) Faulty starter relay, start auxiliary relay, start select relay, L/G safety relay or engine start control relay.	Replace relay.
	(c) Faulty engine start switch, start select switch, battery select switch or crank-run-stop switch.	Replace switch.
	(d) Malfunction to L/G safety switch.	Readjust timing and replace if necessary.
	(e) Circuit breaker LH(RH) ENG CONT, ENG POWER CONT, ENG START CONT or LG POS IND have tripped.	Check wiring for grounding and reset the circuit breaker.
	(f) Faulty engine speed switch.	Replace engine speed switch.
	(g) Open diode CR205, CR206.	Replace diode.

Trouble	Probable cause	Remedy
(5) continued:	(h) Faulty external power control relay, series control relay or series paralleling relay. (Battery series start only.)	Replace relay.
	(i) Circuit breaker SERIES RELAY have tripped. (Battery series start only.)	Check wiring for grounding and reset circuit breaker.
	(j) No. 2 battery has not been connected to airplane. (Battery series start only.)	See Para. (2).
	(k) External power can not be connected.	See Para. (4).
	(l) Faulty airplane wiring.	Check continuity and repair.
(6) Starter does not accelerate fully	(a) No. 1 and No. 2 batteries have been discharged or defective.	Check batteries.
	(b) Faulty starter.	Check starter.
	(c) No. 1 or No. 2 battery is not connected to airplane.	See Para. (1) and (2).
	(d) Main bus tie circuit breaker has tripped. (Battery parallel start only.)	Check circuit and reset.
(7) Starter does not accelerate while starting	(a) Faulty engine speed switch.	Replace engine speed switch.
	(b) Faulty ignition relay.	Replace relay.
	(c) Faulty ignitor.	Replace ignitor.
	(d) Faulty tachometer generator.	Replace tachometer generator.

Trouble	Probable cause	Remedy
(8) Voltmeter indicates more than 29V in engine starting	(a) Faulty monitored bus control relay.	Replace relay.
	(b) Malfunction of voltage sensor.	Adjust or replace.
	(c) Faulty wiring.	Check wiring for continuity.
(9) L(R) DC GEN OUT on annunciator panel does not go out when generator switch is turned to ON	(a) Faulty generator switch or master switch.	Replace switch.
	(b) Faulty voltage regulator control relay.	Replace relay.
	(c) Faulty overvoltage protecting relay.	Replace relay.
	(d) Faulty reverse current cutout relay.	Replace relay.
	(e) Faulty generator failure defective relay (S/N 652SA & 661SA).	Replace relay.
	(f) Fusing of generator failure warning fuse (S/N 697SA & up).	Check wiring for grounding and replace fuse.
	(g) Faulty starter generator.	Check generator and replace if necessary.
	(h) Faulty voltage adjustment.	Adjust voltage regulator and replace if necessary.
	(i) Disconnection of rheostat.	Replace rheostat.
	(j) Overvoltage protective relay has tripped.	Reset by generator switch and turn to ON again.
	(k) Circuit breaker LH(RH) GEN FIELD of LH(RH) GEN CONT has tripped.	Check wiring for grounding and reset circuit breaker.
	(l) Poor contact or disconnecting of airplane wiring.	Check continuity and repair if necessary.

Trouble	Probable cause	Remedy
(10) L(R) DC GEN OUT on annunciator panel is not illuminated when generator switch is turned to OFF	(a) Faulty generator switch.	Replace switch.
	(b) Faulty reverse current cutout relay.	Replace relay.
	(c) Defective warning lamp.	Replace lamp.
	(d) Faulty annunciator panel.	Replace annunciator panel.
	(e) Faulty airplane wiring.	Check wiring for continuity and repair.
(11) Load connecting to left and right load bus does not actuate even if L(R) FEEDER OUT on annunciator panel goes out	(a) Faulty feeder.	Check wiring for continuity and repair.
	(b) Open diode CR215 or CR216.	Replace lamp.
	(c) Defective warning lamp.	Replace lamp.
(12) Load connecting to left and right load bus actuates even if L(R) FEEDER OUT on annunciator panel is illuminated	(a) Fusing of feeder failure warning fuse.	Check wiring for grounding and replace fuse.
	(b) Faulty annunciator panel.	Replace annunciator panel.
	(c) Faulty airplane wiring.	Check wiring for continuity and repair.
(13) L(R) FEEDER OUT on annunciator panel is illuminated even if circuit breaker L(R) FEEDER CONT is closed	(a) Faulty circuit breaker.	Replace circuit breaker.
	(b) Faulty feeder overload sensor.	Replace feeder overload sensor.
	(c) Faulty feeder relay.	Replace relay.
	(d) Fusing of feeder failure warning fuse.	Check wiring for grounding and replace fuse.
	(e) Faulty airplane wiring.	Check wiring for continuity and repair.
(14) Load connecting to cockpit circuit breaker panel bus does not actuate	(a) Faulty wiring of feeder.	Check wiring for continuity.

Trouble	Probable cause	Remedy
(15) Voltmeter does not swing	(a) Faulty volt-ammeter.	Replace volt-ammeter.
	(b) Fusing of volt-ammeter fuse (F201 or F203) (only when ammeter also does not swing).	Check wiring for grounding and replace fuse.
	(c) Circuit breaker ENG POWER CONT has tripped (only when engine starting).	Check wiring for grounding and reset.
	(d) Faulty starter auxiliary relay.	Replace relay.
	(e) Faulty airplane wiring.	Check wiring for continuity and repair.
(16) Ammeter does not swing	(a) Faulty volt-ammeter.	Replace volt-ammeter.
	(b) Fusing of volt-ammeter fuse (F201, F202, F203 or F204) (voltmeter also does not swing when F201 or F203 is fused).	Check wiring for grounding and replace fuse.
	(c) Faulty shunt.	Replace shunt.
	(d) Faulty airplane wiring.	Check wiring for continuity and repair.
(17) Indication difference of left and right ammeters is large in parallel operation of left and right generators	(a) Unbalance of voltage adjustment of left and right generators.	Check wiring for continuity and repair.

Trouble	Probable cause	Remedy
(18) Load connecting to the left overhead switch panel bus does not actuate (pitot tube, stall warning, air in-take and prop. heater in anti-icing system)	(a) Faulty wiring of feeder. (b) Circuit breaker OVHD PANEL in circuit breaker panel has tripped.	Check wiring for continuity. Check feeder for fault and reset.
(19) Loads of anti-icing system and lighting system connecting to overhead switch panel do not actuate	(a) Faulty feeder. (b) Circuit breaker UPPER PANEL has tripped.	Check for con-tinuity and repair wiring. Check wiring for grounding and re-set.

2.9 BATTERY

2.9.1 GENERAL DESCRIPTION

The batteries are two nickel cadmium. They are installed in the aft electrical compartment, No. 1 battery on the LH side and the No. 2 battery in the center.

Since high discharge current is necessary for engine starting, the nickel-cadmium battery which has superior rapid discharge characteristics is used. This battery has such advantage, but, on the other hand, it is too sensitive for poor maintenance. Insufficient maintenance not only causes the battery not to exhibit full performance, but also damages it, so the following special care should be taken to obtain the peak performance and life.

Operational practice to prevent battery overheating:

a. Reduce the number of consecutive engine starts by programming the use of well regulated external power supply when a series of short duration flights or consecutive engine starts are planned. Also, avoid prolonged engine cranking.

b. When ambient temperature or engine oil temperature is high, avoid series-batteries engine start and perform parallel-batteries engine start.

c. When battery is charged after engine-starting by battery, it is advisable that engine RPM is set to idling condition until charging current drops.

Maintenance practice to prevent battery overheating:

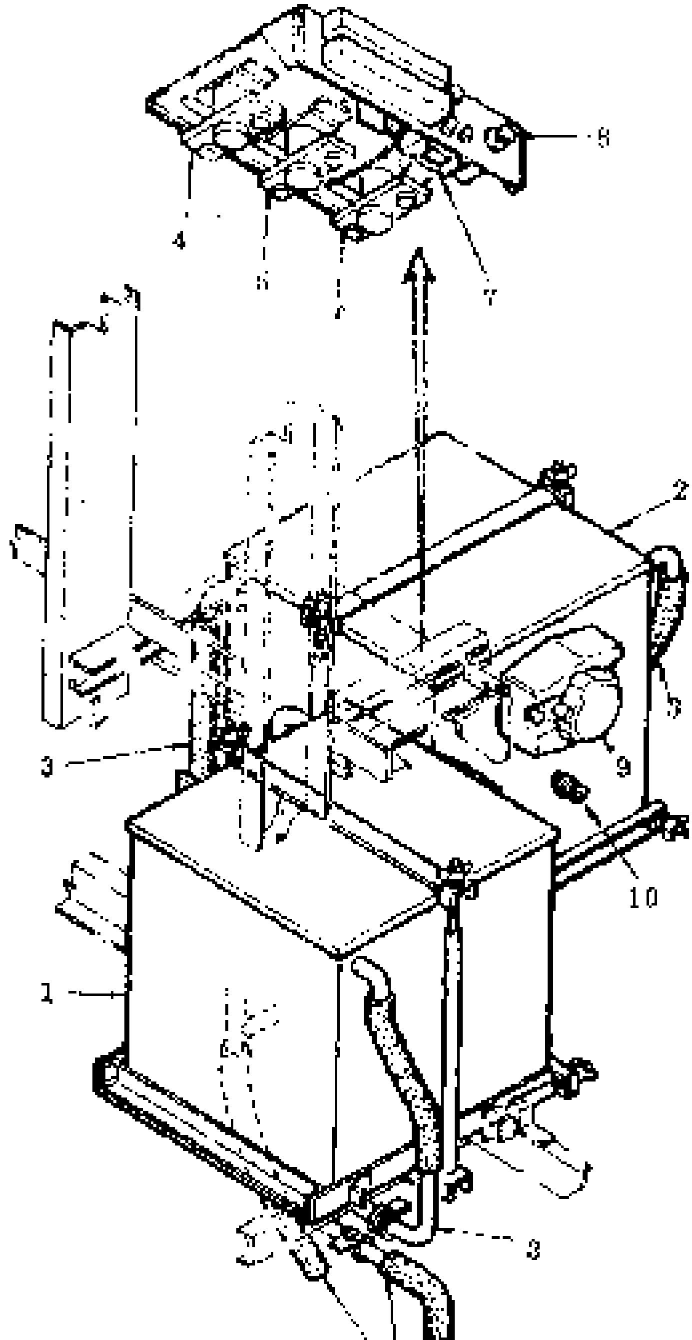
a. Perform visual inspection and maintenance check periodically.

b. The use of a service log about the following items provides an accurate service record of battery inspections and malfunctions. It can also be a useful tool in determining the optimum period between reconditionings.

(i) Voltage of each cell before maintenance
(ii) Voltage of each cell after maintenance
(iii) Water quantity supplied to each cell

c. Frequent inflight monitoring of the aircraft bus voltage and load current will provide an indication of any abnormal condition.

d. During extended ground operation, under high outside ambient temperatures, keep the battery loads to a minimum and ensure there is adequate battery compartment ventilation. Additional ventilation may be provided by opening the battery compartment access door or using forced air ventilation.



1. No. 1 battery
2. No. 2 battery
3. Battery vent line
4. No. 1 battery relay
5. External power source relay
6. No. 2 battery relay
7. Emergency relay
8. Room light circuit breaker
9. Battery plug
10. Battery temp. sensor plug

Fig. 13-16 Battery

2.9.2 REMOVAL AND INSTALLATION

(1) Make sure that the battery switch is in OFF position and remove battery plug.

(2) Remove plug for battery temperature sensor.

(3) Loosen clamps and remove vent tubes.

(4) Cut off safety wire, loosen wing nuts, lay attaching rods on each side and take out battery.

(5) Install in reverse sequence of removal after cleaning battery stand.

2.9.3 VISUAL INSPECTION

Visually inspect the battery for the following items and conduct a detailed investigation when any of the abnormal conditions are noted. Perform reconditioning service on reference to the Paragraph of trouble shooting.

(1) Cell case distortion.

(2) Burn marks or signs of overheating on battery terminals or cell links.

(3) Overflow of electrolyte or white deposit on the tops of the cells.

(4) Check electrolyte level and adjust if necessary (See Para. 2.9.4.(4)).

NOTE

The electrolyte level rises considerably during charge, and then drops fairly rapidly for an hour or so after charge. Adding water to a discharged battery is to take the risk of liquid overflow during charge, causing damage to cell, so careless adding of water should be avoided.

Frequent inspection of electrolyte level is desirable and supply of water periodically or at bench check is recommended.

(5) Measure electrical leakage between battery connector pin and battery case.

(6) Any clogging in battery vent hole, check for clear (unclogged) battery vent hole.

2.9.4 MAINTENANCE

Reconditioning services are determined mainly by the airplane flight hours depending on:

(1) Number of engine startings by battery.

(2) Ambient temperature.

It is recommended to perform it every 100 flight hours and, if the battery is subjected to severe starting and high ambient temperature, at 50 flight hours.

NOTE

(i) Anything associated with the lead acid battery (acid fumes included) should never come in contact with the nickel-cadmium battery or its electrolyte. Even traces of sulfuric acid from a lead acid battery entering the electrolyte of a nickel-cadmium battery will result in damage.

(ii) The electrolyte used in nickel-cadmium batteries is a caustic solution of Potassium Hydroxide. Use rubber glves, rubber apron and protective goggles when handling this solution.

If electrolyte gets on the skin, wash the affected areas with large quantities of water, neutralize with 3 percent acetic acid, vinegar or lemon juice. If electrolyte gets into the eyes, flush with water and get immediate medical attention.

(1) Inspection

Check battery for installed condition, then remove it and perform visual inspection per Para. 2.9.3 (1) thru (3).

(2) Cleaning

(a) Remove white deposits on the top of the cell and cell connector using non-metallic brush.

(b) Make sure that vent caps are properly tightened.

(c) Tip battery on side.

(d) Flush with tap water.

(e) Remove excess water using compressed air.

NOTE

(i) Do not use metal brush.

(ii) Do not attempt to clean the battery with solvents, acids or any chemical solution.

(iii) Remove battery cover to avoid gas accumulation while reconditioning.

(iv) Keep rings, metal watch bands and tools away from the exposed parts of the battery to prevent from a short circuit.

(3) Leakage current flow check

(a) Using a tester, such as Simpson 261 or 263 which has an amperage scale of the minimum of 500 mA, measure the leakage current flow connecting "+" terminal of tester with "+" terminal of battery, and "−" terminal of tester with battery case. Also measure the leakage current flow connecting "−" terminal of tester with "−" terminal of battery and "+" terminal of tester with battery case.

(b) If the readings exceed 50 mA even though cleaning is accomplished per Para. (2), disassemble the battery, flush and check coating of battery case and cell case for abnormal conditions.

NOTE

Estimate the quality of battery not by between battery terminal and case but by electric current.

(4) Charging and electrolyte level adjustment

(a) Charge the battery at room temperature by applying 28.5 constant voltage method and continue until the current tapers off and stabilizes at below 10 amperes.

(b) Cut-off the charging and loosen the all filler caps from each cell, then allow to stand for three hours.

(c) Remove filler caps and check level and adjust, if necessary, to the proper level above the baffle inside the cells by adding distilled, deionized or demineralized water.

(d) Tighten the filler caps to original condition.

Manufacturer	Battery Type	Electrolyte Level above Baffle
Marathon or Sonotone	CA-5	* 1/8 in. (3.2 mm)
" "	CA-20H	* 1/8 in. (3.2 mm)
" "	CA-21H	* 1/8 in. (3.2 mm)
Gulton	GB-40A	** 3/8~1/2 in. (9.5~13 mm)
"	GB-122	** 1/8~3/8 in. (3.2~9.5 mm)

Note : * 2 thru 4 hours after charging.

** 2 thru 24 hours after charging.

NOTE

(i) As the liquid level is different due to charging condition, the level should be measured after charge.

(ii) Do not use any equipment associated with lead acid battery or its electrolyte in liquid level adjustment. Electrolyte is solution of potassium hydroxide and alkaline, and even traces of sulfuric acid from a lead acid battery entering the electrolyte of a nickel-cadmium battery will result in damage.

(iii) To adjust liquid level, use only distilled water. The battery is easily contaminated through the use of tap water which contains minerals, chlorines, softening agents and other foreign materials.

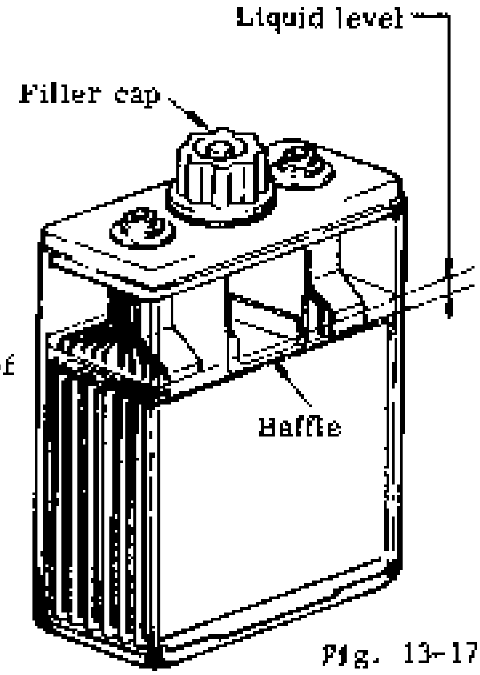


Fig. 13-17

(iv) For liquid level adjustment, distilled water is generally used, but cells which have lost electrolyte by accidental overfilling may have a lower specific gravity than 1.24, so they should be sent to the Battery Manufacturer for adjustment. The battery which has specific gravities ranging between 1.24 and 1.32 is normal.

(v) After pouring distilled water, recharging is recommended to prevent water freezing.

(vi) Liquid level measurement

Remove each cell filler cap and look down into the vent well to determine liquid level. If it is impossible in this manner, use a glass tube, open on both ends. Insert the tube into filler opening deep enough to touch the top of the plates. Place the index finger over the top open end of the tube and remove the tube from the filler well. The level in the lower end of the tube shows the liquid level above the tops of the plates. (See Fig. 13-17)

(5) Capacity check

(a) After recharging in accordance with Para. (4) above, check for charging capacity as follows :
Discharge the battery at the discharging rate of one hour or two hours by connecting the resistance load bank or some equipment suitable to obtain the constant discharge current by monitoring the voltage until cut-off voltage is reached and then measure the time required.

Manufacturer	Battery Type	Discharge Current (Ampere)		Cut-off Voltage (V)
		1 H Disch. Rate	2 H Disch. Rate	
Marathon or Sonotone	CA-5	34	17	19.0
"	CA-20H	20	11	19.0
"	CA-21H	20	11	19.0
Gulton	GB-40A	35	17	19.0
"	GB-122	20	11	19.0

NOTE

When ambient temperature is high, two hour discharging rate which has a little temperature rise, is recommended.

(b) The time to reach cut-off voltage must be longer than 42 minutes at a one-hour rate discharge and 84 minutes at a two-hour rate discharge.

NOTE

(i) If discharging time is less than 42 minutes or 84 minutes, respectively, the battery should be deep-cycled and a capacity check made.

In this case, measure individual cell voltages after 42 minutes or 84 minutes, depending on one- or two-hour rate. If any cell is below one volt, replace it or again perform a deep cycling and capacity check.

(ii) The charging capacity of a nickel-cadmium battery, different from lead acid battery, can not be measured by electrolyte specific gravity.

(6) Deep-cycling

(a) Continue discharging in Para. (5).

(b) Short out each cell as it drops below 0.6 volts by using copper shorting clips, one by one, while the load is still applied.

(c) When about 75% of total cells have been shorted, a 1.0 ohm resistor of 1 thru 2 watts should be placed across remaining cells. The battery should remain shorted as above for a period of 3 or more hours.

NOTE

After stopping discharge, cell capacities recover in process of time. In this condition, if cells are short-circuited, high short-circuit current may burn cells. Therefore, short-circuit of cells should be made during discharging.

(7) Final charge

(a) Charge the battery by the constant current method as follows.

Manufacturer	Battery Type	Charge Current (A)	Charge Time (H)
Marathon or Sonotone	CA-5	8.0	7
" "	CA-20H	5.0	7
" "	CA-21H	5.0	7
Gulton	GB-40A	8.0	7
"	GB-122	5.0	7

NOTE

For perfect charging of battery, charging capacity, 140% of the ampere-hour is necessary. Therefore, if the charging current drops during charge, adjust manually to maintain proper charging current.

(b) During the final 5 minutes of charge, the voltage of each cell should be checked. Each cell should be between 1.55 V thru 1.75 V.

NOTE

(i) Charge should not be started if the battery temperature is above $100^{\circ}F$.
If the temperature is raised up more than $120^{\circ}F$ while charging, stop the charging and continue after cooling.

(ii) When any cell in which temperature is raised remarkably (more than $68^{\circ}F$ above starting temperature as an aim) or in which the voltage reaches high value (more than 1.75 V), there is the possibility of low level of electrolyte or damaged separator.
If it seems that it is caused by poor adjustment of electrolyte level, add about 5 thru 10 cc water judging from the conditions at that time and recheck the level after charging.
The cell which does not recover after adding water should be replaced.

(iii) If any cell fails to rise to at least 1.55 volts, the constant current charge should be continued for an additional hour. Any cell that fails to rise above 1.55 volts should be replaced.

(iv) When replacing cell, do not use cell other than the one of the same manufacturer. Also the cell of the nearest manufacturing date is desirable, if possible.

(8) Inspection after reconditioning

Check for the following items -

(a) Electrolyte level [See Para. (4)]

(b) Leakage current flow [See Para. (3)]

(c) Installation torque of connector nut between cells

Manufacturer	Battery Type	Nut Installation Torque
Marathon or Sonotone	CA-5	35 ~ 50 in-lbs(40 ~ 58 kg-cm)
"	CA-20H	35 ~ 50 in-lbs(40 ~ 58 kg-cm)
"	CA-21H	35 ~ 50 in-lbs(40 ~ 58 kg-cm)
Gulton	GB-40A	34 ~ 38 in-lbs(39 ~ 44 kg-cm)
"	GB-122	34 ~ 38 in-lbs(39 ~ 44 kg-cm)

2.9.5 STORAGE

Nickel-cadmium battery should be stored with completely discharged condition. Storage can be done in an ambient temperature between $-50^{\circ}C$ and $+70^{\circ}C$. For long period of storage, thin coat of grease should be applied on connectors between cells. A battery being placed into service after long period of storage should be given a freshening charge.

2.9.6 TROUBLE SHOOTING BATTERY

Trouble	Probable Cause	Remedy
Decrease of capacity	Unbalanced cells.	Perform deep cycling.
Voltage is not generated	a. Defective connection of connecting bars between cells. b. Poor contact of battery connector.	Find out defective terminal connector by a voltmeter. Clean and retighten using a specified torque. Check contact surface of connector for dirt. Clean or replace if necessary.
White crystals are found on the top of cell or electrolyte is found on bottom of battery case	a. Excessive charging current or surrounding temp. is too high. b. Electrolyte quantity is not proper. c. Loose vent cap or defective O-ring.	Clean battery and charge. Clean battery, charge and check for electrolyte level. Clean battery and tighten vent cap or replace O-ring.
Cell cases are distorted	a. Overcharging. b. Operation under improper electrolyte level.	Replace defective cells.
Failure of one or more cells to balance with others	a. Different temp., charging efficiency and self-discharging of each cell.	Charge by constant current charging for more time than specified. If these cells still fail to rise to the required voltage, replace them.
One or more cells rise excessively high in temperature or voltage than other cells while charging	a. Insufficient electrolyte. b. Damage of separator.	Adjust electrolyte level. Replace cells.
Foreign material is deposited	a. Foreign substances are most commonly introduced into the cell thru the addition of impure water or water contaminated with acid.	Replace if capacity is insufficient.

Trouble	Probable Cause	Remedy
	b. Charge by excessive high rate charging. c. Charging without sufficient electrolyte. d. Too high a concentration of electrolyte.	
One or more cells require more water than the others	Unbalanced cells.	Perform deep cycling.
Electrolyte consumption is relatively high	Charging voltage is too high or an ambient temperature is too high.	Set charging voltage properly.
Corrosion of top hardware	Presence of acid fumes.	Clean battery and battery compartment.
Cell connectors become hot in charging or discharging or connector shows burn marks	Connector has not been properly tightened.	Tighten with proper torque.
Voltage is detected between connector pin and battery case	Electrolyte spews from cell and wets the outside of cell. Electrolyte is conductive, so voltage can be read.	If leakage current exceeds 50 mA, flush the battery with tap water per Para.9.4.(2). If the current still exceeds 50 mA after cleaning and drying, repeat disassembly, cleaning and inspection.

3. AC POWER SUPPLY SYSTEM

3.1 GENERAL DESCRIPTION

Constant frequency AC power is generated by a 250VA static inverter installed on the RH side of main junction box. The inverter can be operated by DC generator at normal operation or external power or battery on the ground check when the inverter switch on the LH switch panel is turned to MAIN from INV. AC input power is supplied generally to the inverter through the circuit breakers in cockpit, but while starting engine, the power is supplied through DC-DC converter (constant voltage equipment) operated automatically in engine start only and installed on RH main junction box in order to prevent AC power voltage drop, together with DC power voltage drop, and regular AC power is originated.

The AC power system supplies 115V, 400 Hz AC power to the fuel quantity indicating system, ITT indicating system (Aircraft S/N 652SA) and trim indicator lighting system. It also supplies 26V, 400 Hz of AC power to the oil pressure and fuel pressure indicating systems. The inverter fail detecting relay is connected to the 115 VAC output circuit and if this output falls below a specific value, the contact closes and illuminates the "INVERTER FAIL" light on the annunciator panel.

Should failure of the main inverter occur, a standby inverter is installed for a backup (located adjacent to the main inverter in the RH main junction box). The standby inverter is switched ON by placing the inverter switch, LH switch panel, from the MAIN position to the STBY position, while the STBY INV circuit breaker is in the closed position.

3.2 INVERTER

The inverter converts 28V DC power of the aircraft to 115V and 26V, 400 Hz AC power. This inverter is transistor type and the output voltage is constantly regulated by the level adjusting circuit regardless of variation of load and circumferential temperature. The AC power of stabilized frequency is supplied by the L/C tuned oscillator.

The rating capacity is 250VA, but 150% (375VA) of overload current can be supplied for at least five minutes. In case of load more than 160% or grounding of the output terminal, the overload detecting circuit works, to shut down the inverter output, thus protecting the inverter.

3.2.1 REMOVAL AND INSTALLATION (See Fig. 13-10)

(1) Remove the RH access panel of main junction box.

(2) Disconnect an electrical connector of inverter.

(3) Remove four attaching bolts and remove inverter.

(4) Install in reverse sequence of removal.

3.2.2 INVERTER PROTECTING CIRCUIT

The inverter protecting circuit protects the inverter against short-circuit of the output circuit. If the output circuit is short-circuited, this circuit sends a short-circuit current to trip the short-circuit breaker immediately within the allowable overload time in order to prevent a shutdown of the inverter. When the circuit breaker trips, AC power is restored to its original condition. If a short-circuit occurs upstream side of the circuit breaker, this circuit installed into the inverter actuates to protect the inverter from damage.

3.3 INSPECTION OF AC POWER SOURCE SYSTEM

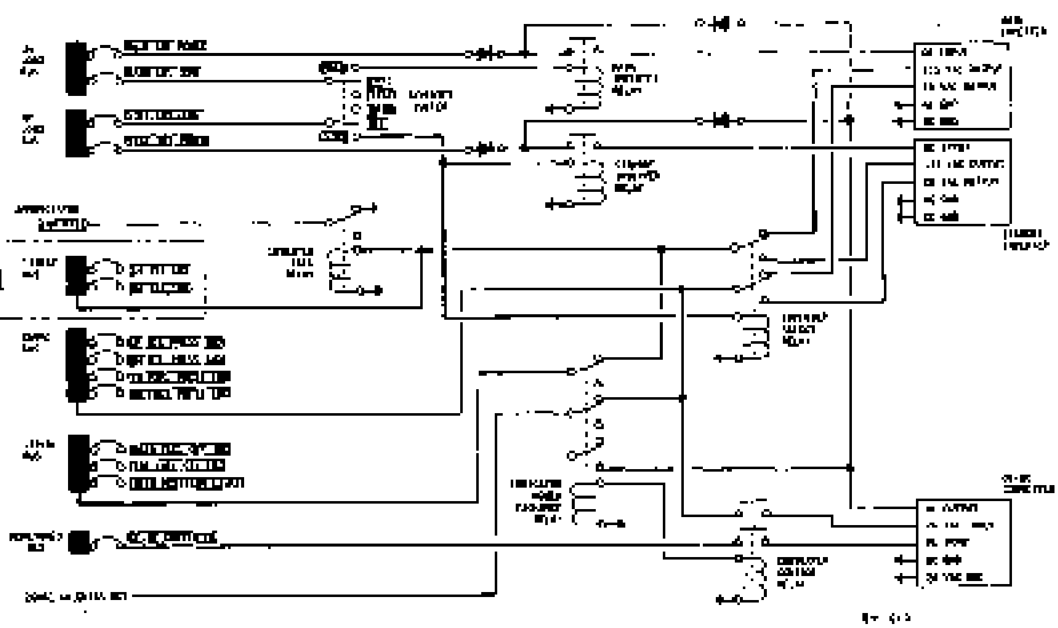
(1) Close circuit breakers INV POWER, INV CONT and MASTER CAUTION in the cockpit circuit breaker panel.

(2) Turn the switch INV on the LH switch panel to MAIN. Ascertain that the inverter actuates and the light INVERTER FAIL goes out.

(3) Measure the voltage and frequency of the terminal TB248 "1" on the terminal board installed on the relay box of the main junction box and make sure that they are 115 +6/-8V, 400 Hz ± 4 Hz. Then, measure voltage of the terminal TB248 "2" and make sure it is 26 +1.3/-1.8V.

(4) Set the switch INV to OFF and make sure that the light INVERTER FAIL on the annunciator panel illuminates.

(5) Pull out all circuit breakers.



*1 Aircraft S/N 652SA unless modified by MAI Kit Drawing K926A-8202

Fig. 13-18 AC power system

3.4 TROUBLE SHOOTING AC POWER SYSTEM

Trouble	Probable cause	Remedy
INVERTER FAIL light is not illuminated even if INV switch is turned to OFF and MASTER CAUTION is closed	(a) Defective master caution system.	Check master caution system.
	(b) Defective inverter failure detecting relay installed on the relay box.	Replace relay.
	(c) Faulty inverter relay.	Replace relay.
	(d) Faulty inverter switch.	Replace switch.
	(e) Faulty airplane wiring.	Check wiring for continuity and repair.
INVERTER FAIL light does not go out even if INV switch is turned to MAIN position	(a) Faulty inverter.	Replace inverter.
	(b) Faulty inverter relay.	Replace relay.
	(c) Faulty inverter failure detecting relay.	Replace relay.
	(d) Grounded AC power feeder.	Check wiring for continuity.
	(e) Faulty wiring.	Check wiring for continuity.
	(f) Faulty inverter switch.	Replace switch.
	(g) Faulty circuit breaker.	Replace circuit breaker.
INVERTER FAIL light is illuminated while starting engine	(a) Faulty DC-DC converter.	Replace DC-DC converter.
	(b) Faulty converter relay.	Replace relay.
	(c) Open diode CR207.	Replace diode.
	(d) Faulty wiring.	Check for continuity.
	(e) Circuit breaker CONVERTER of main junction box has tripped.	Check for faulty wiring and reset.
Frequency and voltage deviate from the specified values	(a) Faulty inverter.	Adjust or replace inverter per vendor overhaul manual.
High level of radio noise	(a) Poor bonding to airframe structure.	Improve bonding.

ORIGINAL
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ATP

4. LIGHTING

4.1 LANDING AND TAXI LIGHT

Landing and taxi lights are installed on both sides symmetrically on the lower sides of the nose. These are used as landing lights at landing and as taxi lights while taxiing.

These lights are controlled by a switch LDG & TAXI located in the overhead switch console and they are extended or retracted separately as the switch is placed in EXT or RET.

If the switch is placed to OFF, the light will turn off at the intermediate position.

The landing light circuit is protected by circuit breakers, LDG & TAXI LAMP and LDG & TAXI LIGHT CONT on the circuit breaker panel. Lamps, 26V 250W sealed beam type, are so installed that the center of the light beam is directed to the forward of the attaching position.

4.1.1 REMOVAL AND INSTALLATION (See Fig. 13-19)

(1) Remove rear inspection panel of landing and taxi lights.

(2) Disconnect wiring from lights.

(3) Remove outer frame from landing and taxi lights by removing 6 screws.

(4) Remove landing and taxi lights by removing 11 screws.

(5) Install in reverse sequence of removal.

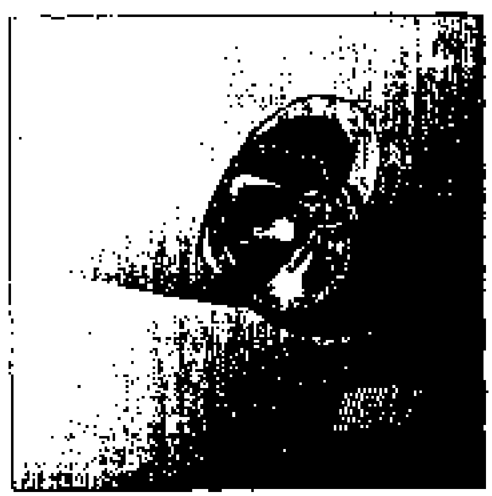


Fig. 13-19 Landing and taxi light

4.1.2 ADJUSTMENT OF LANDING LIGHTS

(1) Level the airplane.

(2) Place target board at 18 ft. 8 in. (5.7 m) forward of the nose landing gear wheel axle, perpendicular to axis of airplane. Using plumbs suspended from four alignment points at STA 1435 and 8880, project the distance between two points on each side of fuselage on target board to locate axis of aircraft.

(3) Using 32.7 in. (830 mm) long plumbs suspended from four alignment points at FUS STA 1435 and 8880, project the distance between two points on each side of fuselage on target board.

(4) Place RH landing light in EXT position. Adjust position of light so that center of beam may fall on a circle of 3.16 in. (80 mm) radius drawn on target board with center located 27.55 in. (700 mm) to the right of axis of aircraft and projection that is obtained in accordance with paragraph (3) above.

(5) Adjust LH landing light to the symmetrical position with RH light in the same way.

(6) Return landing lights to RET position.

4.1.3 OPERATIONAL CHECK

(1) Close circuit breakers LH LDG & TAXI LAMP, RH LDG & TAXI LAMP, LH LDG & TAXI LIGHT CONT and RH LDG & TAXI LIGHT CONT.

(2) Place landing and taxi switch to EXT position, and landing and taxi lights are actuated and illuminate in extended position.

(3) Place switch to RET position, and landing and taxi lights are actuated and go out and retract.

NOTE

Check for noise and interference during operation of landing lights.

4.1.4 MAINTENANCE PRACTICE

Refer to Grimes Manufacturing Co. "OVERHAUL INSTRUCTIONS WITH ILLUSTRATED PARTS LIST" Manual Number 85-50069.

4.2 TIP TANK TAXI LIGHT (If equipped)

A taxi light is installed in the leading end of each tip tank and is controlled by the TAXI switch in the overhead switch console. The lights illuminate when the TAXI switch is placed to the ON position and go off when the switch is placed to the OFF position.

4.2.1 REMOVAL AND INSTALLATION (See Fig. 13-20)

(1) Make sure all aircraft power is OFF.

(2) Remove 6 screws attaching the shroud.

(3) Remove shroud and shield.

(4) Remove 3 screws from the retaining clips; remove the clips.

(5) Bring lamp forward from socket and disconnect electrical wiring; identify the wires.

(6) Install in reverse sequence of removal.

NOTE

(i) Apply RTV732 adhesive sealant (Dow Corning Co.) or equivalent to inside shroud rim before installing as required.
(ii) When installing, make sure that the shield is on the inboard side of the shroud.

4.2.2 OPERATIONAL CHECK

(1) Close circuit breaker "NO.2 OVHD PANEL (LIGHT)".
(2) Turn taxi light switch to "TAXI" position. Make sure that taxi lights illuminate.
(3) Turn switch to "OFF" position, and taxi lights go out.
(4) Pull out circuit breaker.

1 Tip tank
2 Spacer
3 Ring assembly
4 Screw
5 Lamp
6 Clip
7. Screw
8 Shell
9 Shroud
10. Screw

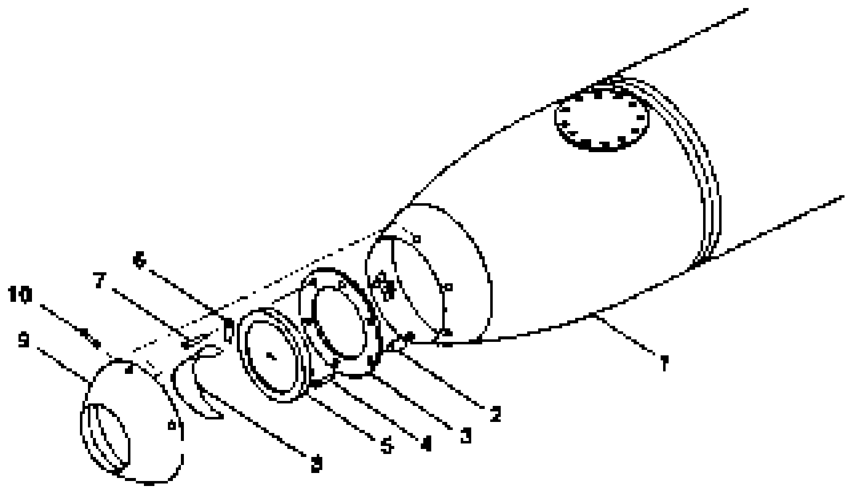


Fig. 13-20 Tip tank taxi light (LH shown, RH opposite)

4.3 NAVIGATION LIGHT

Wing tip lights and tail light are installed on the outboard sides of the LH and RH tip tanks and rearmost section of the fuselage, respectively, and they can be controlled by switch NAV located in the overhead console. When they are on, their color is red (LH tip light), green (RH tip right), and white (tail light), and they are stationary.

4.3.1 OPERATIONAL CHECK

(1) Close circuit breaker RH UPPER PANEL
(2) Place NAV switch ON Verify that wing tip lights and tail light illuminate
(3) Place switch OFF, verify that lights go out, and return circuit breaker to open position.

4.4 ANTI-COLLISION LIGHT

Anti-collision lights required by regulation are installed one each on the upper surface of the vertical stabilizer and the lower fuselage (if installed). They are controlled by switch BEACON in the overhead console and radiate a red flashing light to all directions

4.4.1 OPERATIONAL CHECK

(1) Close circuit breaker UPPER PANEL (Aircraft S/N 661SA).

(2) Place anti-collision light switch BEACON to ON. Verify that anti-collision lights on top of vertical stabilizer and on bottom surface of fuselage rotate and illuminate.

(3) Place switch OFF, verify that anti-collision lights cease to rotate and go out, and return circuit breaker to open position.

4.5 STROBE LIGHT

The strobe lights are installed on the outsides of LH and RH tip tanks and tail cone, and are operated by STROBE switch in the overhead console. The strobe light flashes out at 50 cycles per minute.

NOTE

(i) When cover-attaching screws are unscrewed to remove tip tank strobe light, navigation light cover (colored glass) also comes off, so take care of handling.

(ii) When strobe light is installed or the tip tank is replaced, pay attention not to cause fuel leak due to rough work on the small conduit tube in the tank.

(iii) When strobe light installed on tip tank is replaced, perform leak test and make sure of no leaks at conduit tube. See Chapter VII for leak test procedure.

4.5.1 OPERATIONAL CHECK

(1) Close circuit breaker RH UPPER PANEL.

(2) Place strobe light switch STROBE to ON and ascertain that the strobe lights on LH and RH wing tips and tail cone flash out.

(3) Place the switch OFF and, upon seeing the lights stop flashing, pull out circuit breaker.

4.6 ICE INSPECTION LIGHT

The ice inspection light is installed in the LH outboard engine nacelle and illuminates the outer wing half. This light is illuminated by the switch WING ICE LIGHT on the overhead console.

4.6.1 OPERATIONAL CHECK

(1) Close circuit breaker RH UPPER PANEL.

(2) Set the ice inspection light switch ICE LIT to ON. Ascertain that the ice inspection light comes on, illuminating the LH outboard wing leading edge.

(3) Place the switch OFF, make sure that the light goes off, and then pull out the circuit breaker.

4.7 INSTRUMENT LIGHT

Each instrument is illuminated by post type lights located above the instrument. The lighting system consists of three groups: pilot flight instrument group, co-pilot flight instrument group and engine instrument group. The brightness of each group can be adjusted independently. The instrument light circuit is protected by circuit breaker INST PANEL LIGHT. The brightness is adjusted by rheostats PILOT FLT INST, COPILOT FLT INST and ENG INST in the overhead console.

PILOT FLT INST:	Standby compass, airspeed indicator, altimeter, turn and bank indicator, rate of climb indicator, outside air temperature indicator, attitude gyro, magnetic compass navigation instruments and volt-ammeter.
CO-PILOT FLT INST:	Cabin rate of climb indicator, cabin altitude differential pressure indicator, cabin pressure regulator, co-pilot flight instruments, fuel quantity indicator, fuel totalizer and vacuum gauge.
ENG INST:	Torquemeter, ITT indicator, fuel flow indicator, tachometer, oil pressure indicator, oil temperature indicator, trim aileron position indicator and fuel pressure indicator.

4.7.1 OPERATIONAL CHECK

(1) Close circuit breaker INST PANEL LIGHT.

(2) Turn rheostats in the overhead console (for pilot flight instruments, co-pilot flight instruments and engine instruments) clockwise, and the following instrument lights gain brightness.
(a) Pilot flight instrument light (Pillar light)
(b) Co-pilot flight instrument light (Pillar light)
(c) Engine instrument light (Pillar light)

(3) Turn rheostat counterclockwise to original position. Pull circuit breaker to open.

4.8 PANEL LIGHT

The LH and RH switch panels and overhead console are lighted internally by means of an edge-lighted panel and brightness is controlled by the rheostat PANEL in the overhead console. The panel illumination circuit is protected by circuit breaker INST PANEL LIGHT. The forward circuit breaker panel is illuminated by a post type light installed on the panel and center circuit breaker panel is illuminated by a light under the arm rest. The light is illuminated by a switch installed under the arm rest. This circuit is protected by circuit breaker MAP LIGHT.

4.9 RADIO CONTROLLER PANEL LIGHT

A panel type illumination system is built into each radio controller and brightness is adjusted simultaneously by rheostat in the overhead console. The circuit breaker for this system is the same as the instrument light circuit.

4.10 TRIM INDICATOR LIGHT

The rudder and elevator trim indicators on the center pedestal are illuminated by electro luminescent lights in the dials. This system is operated by 115V 400 Hz AC power and the lights come on when the switch IND LITS on the center instrument panel is turned to DIM. Brightness of the lights is previously adjusted. If necessary, readjusting is possible by means of the variable resistor installed behind center pedestal. This circuit is protected by circuit breaker TRIM POS LIGHT.

4.10.1 OPERATIONAL CHECK

(1) Close circuit breakers INV POWER, INV CONT, INST PANEL LIGHT and TRIM POS LIGHT.

(2) Place switch INV on the LH switch panel to MAIN.

(3) Turn switch IND LITS in the overhead console to DIM. Make sure that elevator and rudder trim dials grow light.

(4) Place the switch OFF and pull out the circuit breaker.

4.11 CENTER PEDESTAL LIGHT

The center pedestal light is installed on the rear side of overhead console. When map light and pedestal light switch is in MAP position, it illuminates map, and in NORMAL position illuminates center pedestal. The center pedestal light is supplied power through circuit breaker INST PANEL LIGHT and brightness can be adjusted by the rheostat ENG INST in the overhead console.

4.12 MAP LIGHT

The map lights are installed one each on the lower section of the LH and RH control wheel and one combination light of center pedestal light in the cockpit ceiling. The map lights in the control wheels are controlled by rheostats in the center of the control wheel. The combination light for map and center pedestal light is illuminated when the switch is placed to MAP position, and brightness is controlled by a rheostat installed adjacent to the switch. These map lights are powered through the circuit breaker MAP LIGHT.

4.13 ROOM LIGHT

The two room lights are installed in cockpit ceiling and continuous cabin ceiling lighting. The cockpit room lights are supplied power through circuit breaker MAP LIGHT and lighted by the switch COCKPIT ROOM LIT on the LH side of cockpit. The cabin room lights are supplied power through circuit breaker ROOM LIGHT directly connected to No. 1 battery in the electric compartment, so they can be illuminated by cabin room light switches, having no connection with switch operation in power source.

4.14 READING LIGHT

The reading lights are installed in the cabin ceiling above each seat location and supplied power through circuit breaker READING LIGHT in the LH main junction box.

4.15 PRIVATE ROOM LIGHT

The private room light is an extension of the cabin ceiling lighting. The light is supplied power through circuit breaker ROOM LIGHT connected to No. 1 battery in the electric compartment, so it can be illuminated by private room light switch in the private room, having no connection with switch operation in power source.

4.16 CABIN SIGN LIGHT

Aircraft S/N 652SA

The cabin sign light is installed on the cabin ceiling and can be operated by the switch CABIN SIGN on the center instrument panel. When the switch is set to FASTEN BELT, the light FASTEN SEAT BELT comes on and for CABIN SIGN, the light NO SMOKING FASTEN SEAT BELT comes on. This circuit is protected by the circuit breaker PASS SIGN LIGHT in the circuit breaker panel of the main junction box. Five lamps are installed in the light assembly and are replaced by removing the light body from the front side.

Aircraft S/N 661SA, 697SA and subsequent

The cabin sign light is installed in the forward ceiling of the cabin. The FS3 CABIN SIGN switch (in the overhead console), when placed to the ON position, illuminates the FASTEN SEAT BELT sign while the NS CABIN SIGN switch (in the overhead console), when placed to ON, illuminates the NO SMOKING sign. These circuits are protected by the RH UPPER PNL (LIGHTS) circuit breaker.

4.17 CIRCUIT BREAKER PANEL LIGHT

The forward circuit breaker panel (side wall) on the LH side of cockpit is lighted by pillar lights and the rear circuit breaker panel is lighted by light box installed under the armrest of pilot seat. Each circuit breaker panel light is controlled by the ON-OFF operation of button type switch in light box under the arm rest of pilot seat. The circuit breaker panel light circuits are protected by circuit breaker MAP LIGHT on circuit breaker panel, together with map light circuit.

4.18 CIGARETTE LIGHTER

The cigarette lighters are installed in the cabin arm rest. This circuit is protected by the circuit breaker CIGAR LIGHTER in the main junction box. Power is supplied to each cigarette lighter through the resistor on the RH upper side of the main junction box.

4.19 LAMPS AND FUSES

Lamp Number	Locations
#327	Master caution light, annunciator panel light, instrument lights, standby compass light, fire extinguisher handle light, warning light and indicator light, cabin sign light
#334	Engine instrument light
#7512	Wing tip light (Navigation light)
#1683	Tail light (Navigation light)
*1 A-7079B-24	Anti-collision light, ice inspection light
#1495	Map light, reading light
#307	Private room light
*2 916-2020/961-Ta00	Cabin light
*3 4553	Landing light

*1: Grimes Mfg. Co., Urbana, Ohio, U.S.A.

*2: Ichiko Industries Co., Ltd.

*3: General Electric Co.

Spare Parts (lamps, fuses, etc.)	Part Number	Quantity
Lamp	MS25237-327	2
"	ASA #334	3
"	MS25232-1683	1
"	MS25232-307	1
"	ASA #7512	1
"	MS25069-1495	2
"	*1 961-2020/961-Ta00	2
	*2 A-7079B-24	2
Fuse	*3 AGC-3	*4 1, *5 2
"	*3 AGC-5	2
Safety Pin	035A-38864	2

NOTE *1 : Ikko Ind. Co.

*2 : Grimes Mfg Co.

*3 : Bussmann Mfg. Co.

*4 : Aircraft S/N 652SA, 661SA, 697SA thru 699SA

*5 : Aircraft S/N 700SA and subsequent

5. WARNING SYSTEM

5.1 STALL WARNING SYSTEM

This system consists of a lift transducer, a signal summing unit, a control shaker installed on the cross bar of the column assembly and a flap position potentiometer. The lift transducer actuates the control shaker to alert the pilot(s) that the airspeed has decreased to 4 to 9 kts. above the stall speed of each flap position.

5.1.1 LIFT TRANSDUCER (See Fig. 13-21)

The lift transducer is installed on the RH wing leading edge W.STA 4220. Displacement of movable piece exposed on the bottom surface of the wing changes the magnetic flux, changing voltage. An anti-icing system is provided on the exposed section.

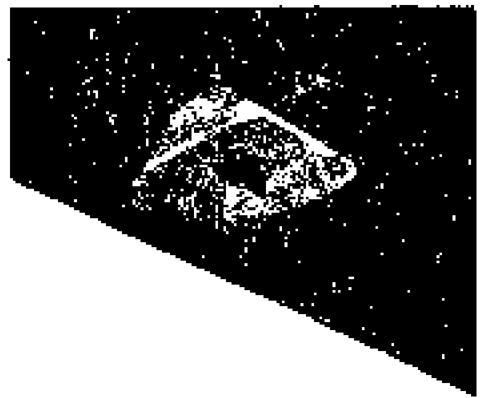


Fig. 13-21 Lift transducer

5.1.1.1 REMOVAL AND INSTALLATION

(1) Remove 4 attaching screws.

(2) Disconnect wiring connector through access door.

(3) Pull out the assembly slowly in outward direction of wing.

(4) Install in reverse sequence of removal.

CAUTION

Do not deform exposed movable piece. If it is bent, do not try to correct it.

5.1.2 FLAP POSITION TRANSMITTER (See Fig. 13-22)

This is a potentiometer installed on the center wing trailing edge on RH side, resistance of which varies as flap travels.

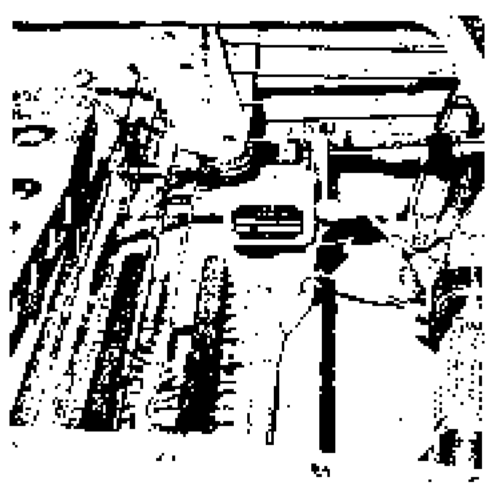


Fig. 13-22 Flap position transmitter

5.1.3 SIGNAL SUMMING UNIT

This unit is installed on the relay box of the main junction box and, upon receiving a signal from the flap position transmitter and the lift transducer, sends out a signal which actuates the shaker.

5.1.4 OPERATION AND ADJUSTMENT OF STALL WARNING SYSTEM

5.1.4.1 CHECKOUT BEFORE ADJUSTMENT

(1) Check vanes of lift transducer for deformation and damage.

(2) Replace flap from UP to 20° position.

(3) Verify that resistance between output terminals B and C of flap potentiometer is 180 Ω ± 10 Ω.

For adjustment, change position of lever relative to shaft by loosening set screw on shaft of flap potentiometer.

(4) With flap in UP and 40° positions, verify that resistance between output terminals B and C of potentiometer is as follows:

UP position	3000 Ω ± 100 Ω (Reference value)
40° position	0 Ω

5.1.4.2 OPERATIONAL CHECK

(1) Turn battery key switch ON, make sure that the landing gear safety switch is operative, and place flap in 40° position.

(2) Close circuit breakers STALL WARN and L/G CONT.

(3) Place switch STALL WARN TEST on RH switch panel in cockpit in GND position, and shaker operates and control wheel shakes.

(4) Jack aircraft. (Landing gear safety switch inoperative)

(5) Hold lift transducer vane full aft.

(6) Place STALL WARN TEST switch in FLT position, and shaker operates, and control wheel shakes. Release lift transducer vane to neutral position; shaker may continue to operate.

5.1.4.3 MEASUREMENT OF SHAKER OPERATING ELECTRIC CURRENT (See Fig. 13-23)

Measure the electric current with the aircraft jacked, landing gear safety switch inoperative, and flap in 40° position.

(1) Remove LH access panel in the main junction box.

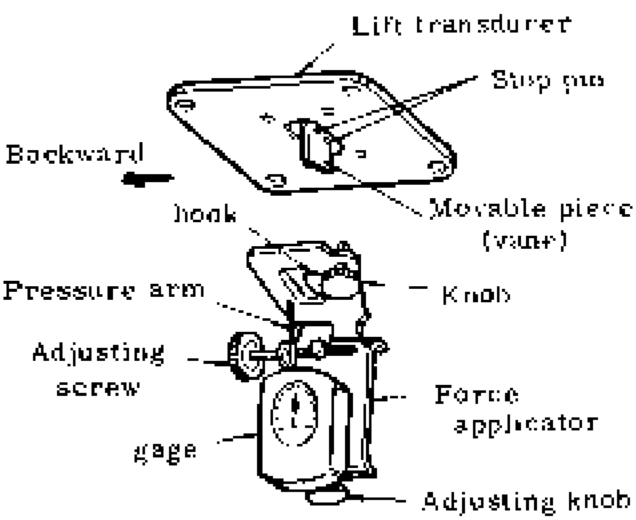
(2) Connect ± 100 μA ammeter to lead wires from E and F terminals of signal summing unit. (A lead wire from E terminal should be connected to (+) terminal and lead wire from F terminal to (-) terminal of ammeter.)

(3) Move vanes of lift transducer forward slowly by hand. Repeat the process a few times. The indicator of ammeter travels smoothly from positive (+) to negative (-). Record average value of amperage when shaker operates.

Fig. 13-23 Connect ammeter to terminal board TB745

5.1.4.4 ADJUSTMENT OF STALL WARNING SYSTEM

(1) Attach force applicator (Safe Flight Instr. Corp. P/N 990) to the lift transducer in accordance with the following procedure, see Fig 13-24.
   (a) Attach gauge to force applicator mount.
   (b) Place the hook of applicator on stop pin of vane. Fasten applicator with fastening knob.
   (c) By means of adjusting screw, adjust engagement of tip of pressure arm of gauge with vanes to 0.02 to 0.03 in. (0.5 to 0.8 mm).



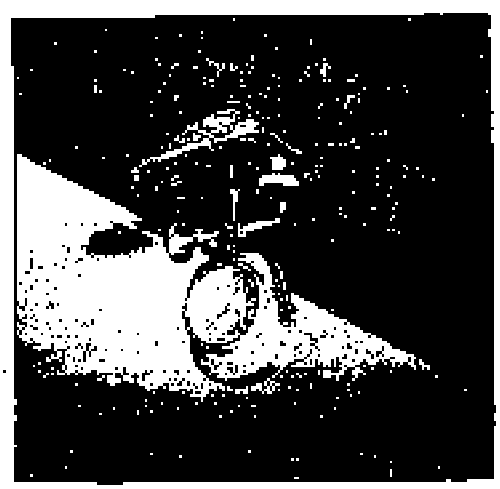


Fig. 13-24 Attach force applicator to lift transducer

(2) Place flap in 40° position.

(3) Apply force of 0.6 G of gauge indication to vanes of lift transducer by rotating adjusting screw of force applicator. By rotating NULL adjusting screw of signal summing unit, adjust shaker actuating amperage to read within $\pm$ 5μA that is noted in accordance with 5.1.4.3 (3) of this Paragraph.

(4) Place flap in 20° position.

(5) Apply force of 2 G of gauge indication to vanes of lift transducer by rotating adjusting screw of force application. By rotating FLAP adjusting screw of signal summing unit, adjust shaker actuating amperage to read within $\pm$ 5μA that is noted in accordance with 5.1.4.3 (3) of this Paragraph.

(6) Place flap in UP position.

(7) Apply force of 4.5 G of gauge indication to vanes of lift transducer by rotating adjusting screw of force applicator. If amperage does not read within shaker actuating amperage of $\pm$ 5μA that is noted in accordance with 5.1.4.3 (3) of this section, adjust amperage to read within specified value by rotating F adjusting screw of signal summing unit.

(8) Detach applicator and ammeter. Turn circuit breaker off. Restore aircraft to original condition.

NOTE

If the shaker actuating amperage is adjusted with the flap in UP position, the amperage adjusted with the flap in 20° position becomes out of order. For readjustment, steps (4) to (7) must be repeated with flap in 20° position.

CAUTION

Do not adjust sensitivity adjusrment screw of signal summing unit.

5.2 ENGINE FIRE DETECTION SYSTEM

The fire detection system is installed independently on LH and RH engines and consists of thermistor type continuous temperature sensor, control unit and warning light.

The circuit for the fire detection system is protected by circuit breaker LH(RH) FIRE DET.

The system continues to operate even if the MASTER switch is placed to the EMER position, because the system is operated by battery through the emergency relay.

5.2.1 TEMPERATURE SENSOR

Two temperature sensors with different lengths are installed in Zone I and Zone II of the engines.

CAUTION

(i) Temperature sensors should not be bent frequently.

(ii) Bending radius should be greater than 1.19 in. (30 mm).

5.2.2 CONTROL UNIT

When resistance of temperature sensor has decreased below 350 Ω due to fire, the relay in the control unit is actuated to light warning lamp. The control unit is installed on the LH and RH sides in the relay panel in the center wing leading edge.

A short discriminator is provided in each control unit to prevent the fire warning circuit from operating when sensor is shorted due to failure. Therefore, if the temperature sensor is shorted, the fire warning light will not illuminate even if the test switch for the fire detector is pushed.

5.2.3 WARNING LIGHT

The warning lights are built in the fire extinguisher handles, which are in the instrument shroud, one for the left and one for the right engine.

CAUTION

Do not pull handle until fire is confirmed.

5.2.4 INSPECTION

When the test switch is pushed, the LH and RH fire extinguishing handles are illuminated in red, indicating that the entire circuit of the fire detector is operating normally. If they do not illuminate, perform an inspection in the following manner.

(1) Faulty lamp
Test lamps built in the fire extinguisher handle.

(2) Faulty control unit
Check operation. If operation is faulty, replace or repair.

(3) Open wire in circuit or faulty contact
Check continuity of circuit in accordance with wiring diagram.

(4) Short circuit of temperature sensor
(a) Disconnect the connector of each sensor and flexible cable.
(b) Measure the resistance at the plug between inner conductor and outer sleeve. Resistance more than 200 Ω is normal. Shorted wire should be replaced.

CAUTION

In measuring the resistance, use a circuit tester only. When a voltage in excess of 10V is applied, disconnect connector so that voltage is not applied to the control unit.

(5) Open circuit of wire of temperature sensor
(a) Open the nacelle door.
(b) Disconnect the plug at joint section of temperature sensor and flexible cable.
(c) Check continuity of inner conductor of temperature sensor. If wire is open circuited in temperature sensor, replace.

5.3 OUTER TANK FUEL LOW PRESSURE WARNING SYSTEM

When outer tank fuel is empty or outer tank fuel transfer pump is defective, a warning light LH(RH) OUTER TANK EMP in the center instrument panel is illuminated (See Chapter VII). Test of warning light is performed by closing circuit breaker LH(RH) OUTER FUEL EMP WARN and turning panel indicator test switch to TEST. When indicator light dimmer switch is turned to IND LITS DIM, warning light can be dimmed.

5.4 LANDING GEAR WARNING SYSTEM

The warning is given by a horn installed on the forward section of LH cockpit, and an UNSAFE light mounted on the LH switch panel.

5.4.1 HORN WARNING

The horn sounds if the landing gear control switch is placed to UP position by mistake when the aircraft is on the ground, or if the nose landing gear uplimit mechanism is misoperated during inspection, and/or if the power lever is moved close to FLT IDLE during flight with the landing gear retracted. The horn ceases to operate with the shift of the power lever from FLIGHT IDLE to TAKEOFF, or the landing gear control switch is placed from UP to DOWN.

If the horn sounds in flight, operate the horn cutout switch adjacent to the landing gear control switch and indicator lights, and the horn will cease to sound. When the power lever is moved to TAKEOFF position, the horn cutout switch returns automatically to the original position.

5.4.2 UNSAFE LIGHT

The unsafe light is a push-to-test type warning light mounted on the LH switch panel and can be tested by depressing the lamp. The light can be dimmed by turning the switch IND LITS on the overhead switch console to DIM.

When the following troubles or misoperations occur in the landing gear system, UNSAFE light illuminates for warning.

(1) Landing gear control switch is placed UP on the ground.

(2) Main gear forward door is not closed.

(3) One of landing gear is not perfectly retracted.

(4) One of landing gear is not downlocked.

NOTE

The UNSAFE light illuminating during retraction and extension of the landing gear is normal. The UNSAFE light illuminating after the landing gear extending or retracting operation has been completed, and landing gear has been downlocked or retracted perfectly is not normal.

5.5 MASTER CAUTION SYSTEM AND ANNUNCIATOR PANEL

The master caution system consists of master caution light which indicates trouble of a system grouped in the annunciator panel. The annunciator panel indicates trouble of a system. It is protected by circuit breaker MASTER CAUTION.

5.5.1 MASTER CAUTION LIGHT

The master caution light is a push-to-test type light installed in the center upper shroud and is illuminated and indicates CAUTION if any one of the systems shown on the annunciator panel is abnormal.

After confirming the nature of failure from the annunciator panel, push the master caution light, then indication of CAUTION goes off and it is reset for the next warning.

5.5.2 TEST SWITCH

The test switch is installed adjacent to the MASTER CAUTION light. When the test switch is pushed, master caution light and annunciator panel are illuminated and lamp test is done.

5.5.3 ANNUNCIATOR PANEL

The annunciator panel is installed on the side panel of cockpit. When a system included in the panel has a trouble, master caution light and panel are illuminated, indicating a troubled system, and remain on until failure or abnormal condition of the system has been corrected.

Aircraft S/N 652SA and 661SA

Nomenclature	Lighting	See Para.
FUEL LOW LEVEL	Lighted when indication of main tank fuel quantity indicator or main tank fuel drops below 30 ± 5 gals.	5.6
L(R) BOOST PUMP FAIL	Lighted when boost pump pressure drops below 3.2 ± 0.4 psi.	5.8
L(R) FUEL FIL BYPASS	Lighted when differential pressure between inlet and outlet of fuel filter reaches more than 1.9 ± 0.3 psi.	5.9
INST VAC FAIL	Lighted when differential pressure between vacuum pressure of vacuum system for gyro instruments and cabin pressure drops below 4 ± 0.25 in-Hg.	5.16
L(R) DC GEN OUT	Lighted when generated voltage drops below main bus voltage or abnormal high voltage is generated.	5.17
INVERTER FAIL	Lighted when inverter output voltage drops below 68V.	5.19
CABIN PRESS LOW	Lighted when cabin pressure is lower than atmospheric pressure corresponding to 10,000 (+0, -1000) ft. of altitude.	5.10
DOOR OPEN	Lighted when entrance door or baggage compartment door is open or door lock is imperfect.	5.13

Aircraft S/N 697SA and subsequent

Nomenclature	Lighting	See Para.
FUEL LOW LEVEL	Lighted when indication of main tank fuel quantity indicator or main tank usable fuel drops below 30 ± 5 gals.	5.6
DOOR OPEN	Lighted when entrance door is open or door lock is imperfect.	5.13
L(R) BOOST PUMP FAIL	Lighted when boost pump pressure drops below 3.2 ± 0.4 psi.	5.8
L(R) FUEL FIL BYPASS	Lighted when differential pressure between inlet and outlet of fuel filter reaches more than 1.9 ± 0.3 psi.	5.9
CABIN PRESS LOW	Lighted when cabin pressure is lower than atmospheric pressure corresponding to 10,000 (+0, -1000) ft. of altitude.	5.10
AIR COND SYS FAIL	Lighted when air conditioning inlet bleed air pressure reaches more than 37 psi or air conditioning compressor outlet temperature reaches more than 224 ± 8°C (435 ± 46°F).	5.11
L(R) DC GEN OUT	Lighted when generated voltage drops below main bus voltage or abnormal high voltage is generated.	5.17
L(R) FEEDER OUT	Lighted when load bus is not supplied power due to failure of feeder or overload.	5.18
BAT TEMP 120°	Lighted when internal temperature of battery reaches more than 120°F.	5.21
BATTERY OVER TEMP	Lighted when internal temperature of battery reaches more than 150°F.	5.21
DEFOG OVER TEMP	Lighted when defogging air temperature reaches more than 200°F (93°C).	5.12

Nomenclature	Lighting	See Para.
INVERTER FAIL	Lighted when inverter output voltage drops below 68V.	5.19
PI T/BANK PWR FAIL	Lighted when power supplied to turn & bank indicator for pilot is lost.	5.15
COPI T/BANK PWR FAIL	Lighted when power supplied to turn & bank indicator for co-pilot is lost.	5.15
INST VAC FAIL	Lighted when differential pressure between vacuum pressure of vacuum system for gyro instruments and cabin pressure drops below 4 $\pm$ 0.25 in-Hg.	5.16

5.5.4 OPERATIONAL CHECK

(1) Close circuit breaker MASTER CAUTION. Make sure that master caution light and relative warning light on the annunciator panel, corresponding to the airplane condition, are illuminated.

(2) Push test switch and make sure that all warning lights are illuminated.

(3) Pushing test switch, turn indicator light dimmer switch to DIM and OFF.

DIM Make sure that master caution light and all warning lights dim.

OFF Make sure that master caution light and all warning lights become bright.

(4) Push master caution light and make sure that CAUTION light goes off.

5.6 FUEL LOW LEVEL WARNING SYSTEM

The fuel low level warning system gives a warning by a switch in the fuel quantity indicator, which actuates when main tank fuel quantity indicator indicates less than 30 $\pm$ 5 gal., and by a float switch adjusted to actuate when usable fuel in main tank is 30 $\pm$ 5 gallons.

This warning system illuminates FUEL LOW LEVEL light on the annunciator panel when the switch in the main tank fuel quantity indicator or the float switch in the main tank actuates.

5.6.1 OPERATIONAL CHECK

(1) Close circuit breakers MAIN INV POWER, MAIN INV CONT, MASTER CAUTION, FUEL LOW LEVEL WARN and FUEL QTY IND.

(2) Place the inverter switch to MAIN.

(3) Turn fuel quantity indicator test switch to MAIN and make sure that FUEL LOW LEVEL light on the annunciator panel is illuminated when main tank fuel quantity indicator indicates 30 ± 5 gallons.

(4) Place fuel low level test switch to TEST and make sure that FUEL LOW LEVEL light on the annunciator panel is illuminated.

(5) Place inverter switch to OFF and open circuit breakers except FUEL LOW LEVEL WARN.

(6) Set the airplane in level position and after confirming main tank fuel being in the same level, drain fuel until FUEL LOW LEVEL light on the annunciator panel is illuminated. Make sure that the remaining usable fuel is 30 ± 5 gallons. If not, adjust float switch installing position.

5.7

5.8 BOOST PUMP FAILURE WARNING SYSTEM

The boost pump failure warning system gives a warning of boost pump failure by actuating a fuel pressure switch of boost pump installed in the leading edge of the center wing. Boost pump fuel pressure switch actuates when boost pump discharge pressure drops below 3.2 ± 0.4 psi and illuminates L(R) BOOST PUMP FAIL light. This warning system illuminates L(R) BOOST PUMP FAIL light with circuit breaker LH(RH) BOOST PUMP opened and puts out the light with the circuit breaker closed.

5.9 FUEL FILTER BYPASS WARNING SYSTEM

The fuel filter bypass warning system gives a warning of fuel filter clogging when differential pressure between inlet and outlet of fuel filter installed in the leading edge of inner wing is 1.9 ± 0.3 psi or more due to foreign material, and illuminates L(R) FUEL FIL BYPASS light.

5.10 CABIN PRESSURE WARNING SYSTEM

The cabin pressure warning system gives a warning of low cabin pressure by actuating the cabin pressure warning switch installed on the forward section of LH cockpit. The cabin pressure warning switch actuates when cabin pressure drops below atmospheric pressure corresponding to 10,000 (+0, -1000) ft. of altitude and illuminates CABIN PRESS LOW light on the annunciator panel.

5.11 AIR CONDITIONING FAILURE WARNING SYSTEM

The air conditioning failure warning system gives a warning of air conditioning failure by actuating air conditioning failure warning pressure switch and temperature switch installed on the air conditioning compartment. The pressure switch actuates at more than 37 psi of bleed air pressure, inleft of air conditioning system. The temperature switch actuates at 224 ± 8°C (435 ± 46°F) of outlet air temperature, air conditioning compressor. When any of pressure switch or temperature switch actuates, a press-to-test type air conditioning warning light ACS FAILURE (*1) on RH switch panel, or AIR COND SYS FAIL light (*2) on the annunciator panel is illuminated.

5.12 DEFOG AIR TEMPERATURE WARNING SYSTEM

The defog air temperature warning system gives a warning of defog system failure by actuating defog air-temperature warning switch installed on the LH and RH windshield defog air outlets.

Defog air temperature warning switch actuates when windshield defog air outlet temperature reaches more than 200°F (93°C) and illuminates DEFOG OVER TEMP warning light on the pilot instrument panel (*1), the annunciator panel (*2).

5.13 ENTRANCE DOOR OPEN WARNING SYSTEM

The door open warning system gives a warning of door open or imperfect lock by actuating door lock switches installed on the entrance door.

This warning system illuminates the ENT DOOR OPEN light on the annunciator panel when the entrance door is open or lock is imperfect.

5.14 HEATED WINDSHIELD ANTI-ICE TEMPERATURE WARNING SYSTEM (Aircraft S/N 661SA, 697SA and subsequent)

The heated windshield anti-ice temperature warning system gives a warning of windshield anti-ice system failure by actuating windshield anti-ice temperature controller installed on the lower section of electronic compartment when temperature sensor installed on the LH and RH windshield glass reaches 129 ± 5°F (54 ± 3°C), and by illuminating L(R) H/W OVER TEMP annunciator light.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

5.15 TURN/BANK INDICATOR POWER WARNING SYSTEM

The turn/bank indicator power warning system illuminates push-to-test type warning light T/B IND POWER FAIL (*1) on the instrument panel or PI(COPI) T/BANK PWR FAIL (*2) on the annunciator panel, when turn/bank indicator is not supplied power.

5.15.1 OPERATIONAL CHECK (*1)

(1) Close circuit breaker MASTER CAUTION.

(2) Close circuit breaker T & B IND. Make sure that T/B IND POWER FAIL light goes out.

(3) Open circuit breaker T & B IND. Make sure that T/B IND POWER FAIL light illuminates.

5.16 VACUUM PRESSURE WARNING SYSTEM

The vacuum pressure warning system gives a warning of vacuum pressure source failure for gyro instruments by actuating vacuum pressure warning switch installed on the forward section of LH cockpit.

The vacuum pressure warning switch actuates when differential pressure between vacuum pressure of vacuum system and cabin pressure drops below 4 ± 0.25 in-Hg and illuminates INST VAC FAIL light on the annunciator panel.

5.17 GENERATOR FAILURE WARNING SYSTEM

The generator failure warning system gives a warning of generator output failure by actuating reverse current cutout relay installed in the LH main junction box or overvoltage protecting relay installed in the RH main junction box and by illuminating L(R) DC GEN OUT light on the annunciator panel.

The reverse current cutout relay actuates when generated voltage drops below main bus voltage and 9 to 25A of reverse current is generated in the generator, and cuts generator out of main bus. The overvoltage protecting relay actuates when generated voltage reaches more than 32 to 34V and cuts generator out of main bus and protects the equipment from high voltage.

5.18 FEEDER FAILURE WARNING SYSTEM

The feeder failure warning system gives a warning of feeder failure by tripping feeder-protecting circuit breaker installed on the main junction box and illuminating L(R) FEEDER OUT light in the annunciator panel when overcurrent passes through feeder.

*1 Aircraft S/N 652SA and 661SA
*2 Aircraft S/N 697SA and subsequent

5.19 INVERTER FAILURE WARNING SYSTEM

The inverter failure warning system gives a warning of inverter failure by actuating inverter failure warning relay installed on the AC power relay box of electric compartment when inverter output voltage drops and by illuminating INVERTER FAIL light on the annunciator panel. The inverter failure warning relay actuates when inverter output voltage drops below 68V.

5.20 INDICATING LIGHT

5.20.1 BETA RANGE INDICATING LIGHT

The beta range indicating light is installed on the center instrument panel and illuminates LH(RH) BETA RANGE when engine is in beta range mode or propeller low pitch condition. To test lamp, close circuit breakers LH(RH) ENG CONT and INST PANEL LIGHT and turn panel indicator test switch to TEST. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.2 LANDING GEAR POSITION INDICATING LIGHT

The landing gear position indicating light is a push-to-test type light installed on the LH switch panel. It has three lamps: NOSE, LH and RH, and is illuminated when landing gear is in down position. To test lamp, close circuit breaker LG POS IND and push indicating light. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.3 ENGINE START INDICATING LIGHT

The engine start indicating light is in the head of engine start switch installed on the center pedestal switch panel and is illuminated in engine starting.

5.20.4 MANUAL PROPELLER DE-ICE INDICATING LIGHT (Aircraft S/N 652SA)

This is a push-to-test type indicating light installed on the center pedestal switch panel and is illuminated during manual operation of propeller de-ice system. To test this lamp, close circuit breaker RH MAN PROP DE-ICE and push the light.

5.20.5 FLAP POSITION INDICATING LIGHT

The flap position indicating light is a push-to-test light installed on the flap controller, RH side of center pedestal. It has the four positions of UP, 5^{0}, 20^{0} and 40^{0} and indicating light corresponding to flap position is illuminated. To test lamp, close circuit breaker FLAP CONT and push the light. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.6 HEATED WINDSHIELD HI HEAT MODE INDICATING LIGHT (Aircraft S/N 661SA, 697SA and subsequent)

The heated windshield HI HEAT MODE indicating light is a push-to-test type light installed on the instrument panel and illuminated when heated windshield switch to ON and push the indicating light. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.7 HEATED WINDSHIELD INDICATING LIGHT (Aircraft S/N 661SA, 697SA and subsequent)

The heated windshield indicating light is a push-to-test type light installed on the LH overhead switch panel and LH(RH) WINDSHIELD HEAT ON light is illuminated during operation of heated windshield anti-icing. To test the lamp, close circuit breaker LH(RH) HEATED WINDSHIELD CONT and push the indicating light. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.8 WING DE-ICE INDICATING LIGHT

The wing de-ice indicating light is a push-to-test type light installed in the overhead switch console and WING BOOT ON light is illuminated during operation of wing de-ice. To test the lamp, close circuit breaker WING DE-ICE and push the indicating light. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.9 ENGINE AIR INTAKE ANTI-ICE INDICATING LIGHT

The engine air intake anti-ice indicating light is a push-to-test type light installed on the overhead switch console and LH(RH) ENG INTAKE HEAT ON light is illuminated during operation of engine air intake anti-ice. To test the lamp, push the indicating light. When indicator light dimmer switch is turned to DIM, indicating light can be dimmed.

5.20.10 OIL COOLER AIR INTAKE ANTI-ICE INDICATING LIGHT

The oil cooler air intake anti-ice indicating light is a push-to-test type light installed on the overhead console and LH(RH) OIL COOLER HEAT ON light is illuminated during operation of the oil cooler air intake anti-ice. To test the lamp, push the indicating light. When the indicator light dimmer switch is turned to DIM, the indicating light can be dimmed.

5.21 BATTERY TEMPERATURE WARNING SYSTEM

The battery temperature probe consists of a temperature sensing element contained in the connecting strap which replaces a strap of the battery in the hottest area. The sensing element is connected to a quick disconnect connector mounted on the battery case near the existing battery connector. The indicator is a dual battery temperature read-out type which is designed to read the temperature of two nickel cadmium batteries in conjunction with the probe. Two warning lights are provided: AMBER light labeled 120°F which illuminates if either battery temperature exceeds 120°F, and one RED light labeled HOT which illuminates if either battery temperature exceeds 150°F. It is a 2" indicator with test switches and battery isolate switches.

5.20.21 REMOVAL & INSTALLATION OF BATTERY PROBE

CAUTION

Make sure aircraft power is OFF by turning the master key switch to the OFF position.

(1) Gain access to the battery (or batteries).
(2) Disconnect battery power connector.
(3) Disconnect temperature warning connector (located near battery connector).
(4) Remove battery cell cover.
(5) Remove 2 each nuts attaching the probe to the battery cells.
(6) Remove nuts, washers and screws attaching connector to battery case.

NOTE

Do not pull teflon tubing at connector or probe strap.

(7) Install in reverse sequence of removal.

NOTE

When installing connector seal both sides of flange with Coast Pro Seal #707 or DOW Corning 732 RTV sealant.

5.21.2 REMOVAL & INSTALLATION OF TEMPERATURE INDICATOR

(1) Aircraft power OFF.
(2) Loosen (1) clamp screw.
(3) Pull indicator from panel.
(4) Disconnect connector from back of indicator.
(5) Install in reverse sequence of removal.

5.21.3 MAINTENANCE

Inspect the temperature probe any time the battery is removed from the aircraft for inspection or service.

(1) PROBE

Perform leakage check as shown in Fig. 13-25.

NOTE

Be careful not to let the test pins touch the metal case of plug.

Perform resistance check between pins A to B and between B to C of the temperature plug as shown in Fig. 13-25. Check the resistance between pin B and the plug metal case; this resistance should be 5 megohms or higher.

NOTE

Be careful not to let the test pins touch the metal case of plug.

If either leakage or resistance check does not pass tests, replace the probe.

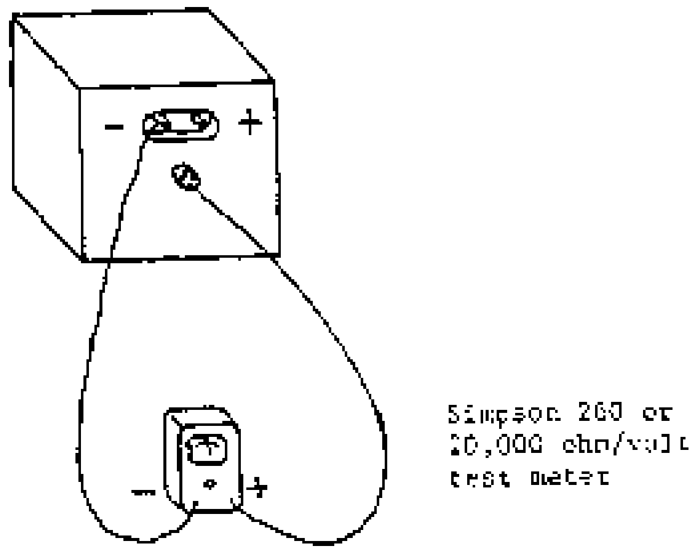
(2) INDICATOR

Calibration check per below

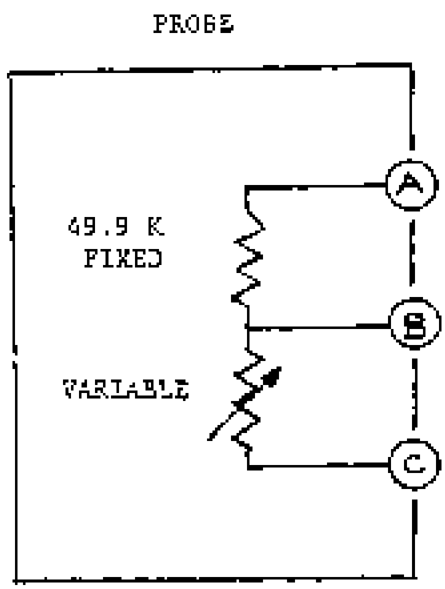
(a) Where possible use P601-V or P602 Tester (available from Foxtronics/Tramm).
(b) Refer to manufacturer's operational handbook for use of the P601-V or P602 test boxes.

NOTE

If indicator calibration is required, return to the indicator manufacturer.



Probe check



'B' to 'C' resistance depending on temperature

-20°F	5.5 Meg Ω
0°F	3.05 Meg Ω
+32°F	1.08 Meg Ω
+75°F	300 K Ω
+120°F	102 K Ω
+150°F	51.1 K Ω
+170°F	33.2 K Ω
+190°F	22.1 K Ω

Resistance check

Fig 13 - 25

5.22 ICE DETECTOR SYSTEM (Ships Modified by S/B No.080/30-003A)

5.22.1 GENERAL

(a) The Rosemount ICE DETECTION SYSTEM provides an advisory light and aural warning to the flight crew when the airplane has entered icing conditions so that ANTI-ICE and deicing procedures may be initiated.

(b) In some light icing conditions at or near 0°C, the ice detection system may not indicate icing before it becomes visible on the leading edges or the windshield surfaces. Nevertheless the deicing systems should be turned on at the first sign of icing, either visually, or by activation "ICING" light.

5.22.2 FUNCTIONAL TEST (See Fig 13-26 thru 13-28)

(1) Obtain a stopwatch.

(2) Confirm the following:

(a) Ensure "ICE DET" circuit breaker is disengaged.

(b) Connect external power supply or battery to aircraft.

(c) Engage the "MASTER CAUTION" circuit breaker.

(d) Engage the "INST" or "INST PANEL" circuit breaker

(e) Ensure the "IND LTS DIM" switch or "IND LT" switch is in "OFF" position, or is in "BRT" position.

(3) Sensing Mode Test

Engage the "ICE DET" circuit breaker.

(a) Verify "ICING" indicator light remains "OFF"

(b) Verify "FAIL" indicator light extinguishes.

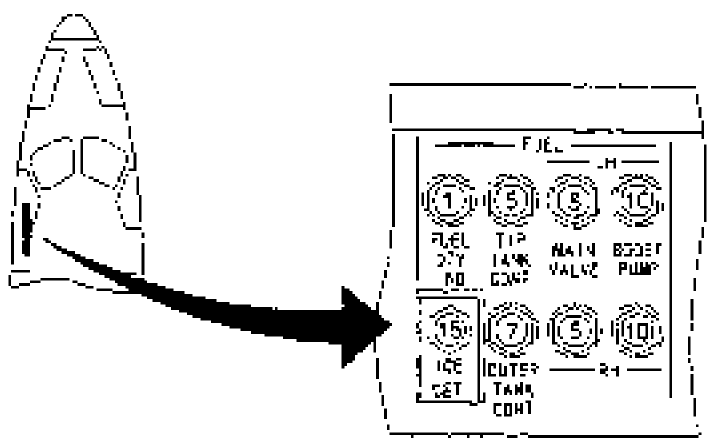


Fig 13 – 26

(4) Lamp Test

Press the "ICING" and the "FAIL" indicator light caps respectively.

(a) Verify respective indicator light illuminates brightly.

Turn the "IND LTS DIM" switch to "DIM" position, and press the "ICING" and the "FAIL" indicator light caps respectively.

(b) Verify respective indicator light illuminates dimly.

(5) Detecting Mode Test

NOTE

The ice detector incorporates a probe that vibrates continuously at a specific frequency. If ice is encountered in flight, ice adhering to the probe will result in a change in this frequency. The frequency change will be detected and the advisory light and aural warning will be activated. The probe is then heated automatically to remove the ice so that a new cycle of ice accumulation and detection can start. To simulate ice on the probe for ground tests, the vibrating probe can be grasped between the thumb and forefinger.

WARNING

The sensing probe and strut become very hot in a few seconds when an ice signal is turned on. Be careful to prevent burns.

Initiate the detecting mode by slowly squeezing the sensing probe between the thumb and forefinger until the "ICING" indicator light turns on, then start the stopwatch. Release the probe when the temperature of the probe and the strut begins to rise. Stop the stopwatch when the "ICING" indicator light turns off.

(a) Verify that the probe and the strut heaters activate.

(b) Verify the "ICING" indicator light illuminates for 60±10 seconds, and the horn sounds for 0.5 seconds when the "ICING" light first illuminates.

(c) Verify the "FAIL" indicator light remains "OFF".

(6) Press to Test

Press and hold the "TEST" switch for 5～10 seconds.

(a) Verify the "ICING" indicator light illuminates while the switch is pressed, and the horn sounds for 0.5 seconds when the "ICING" light first illuminates.

(b) Verify the "FAIL" indicator light remains "OFF".

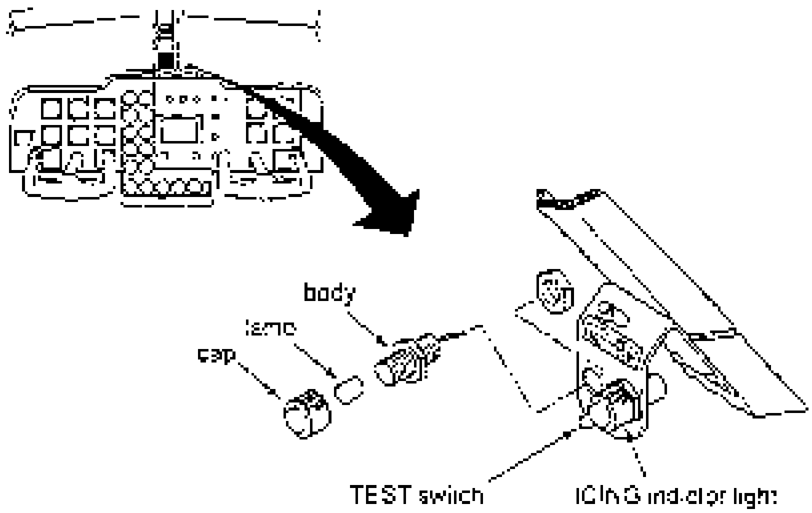


Fig 13 – 27

(7) Fail Mode Test

Pull the "ICE DET" circuit breaker. Verify the "FAIL" indicator light illuminates.

(8) Disengage the following circuit breakers, as applicable:

(a) "MASTER CAUTION"

(b) "INST" or "INST PANEL"

(c) "RH OVHD PANEL No 2"

(9) Remove electrical power from the aircraft.

5.22.3 Replacement of Indicator Light Lamp

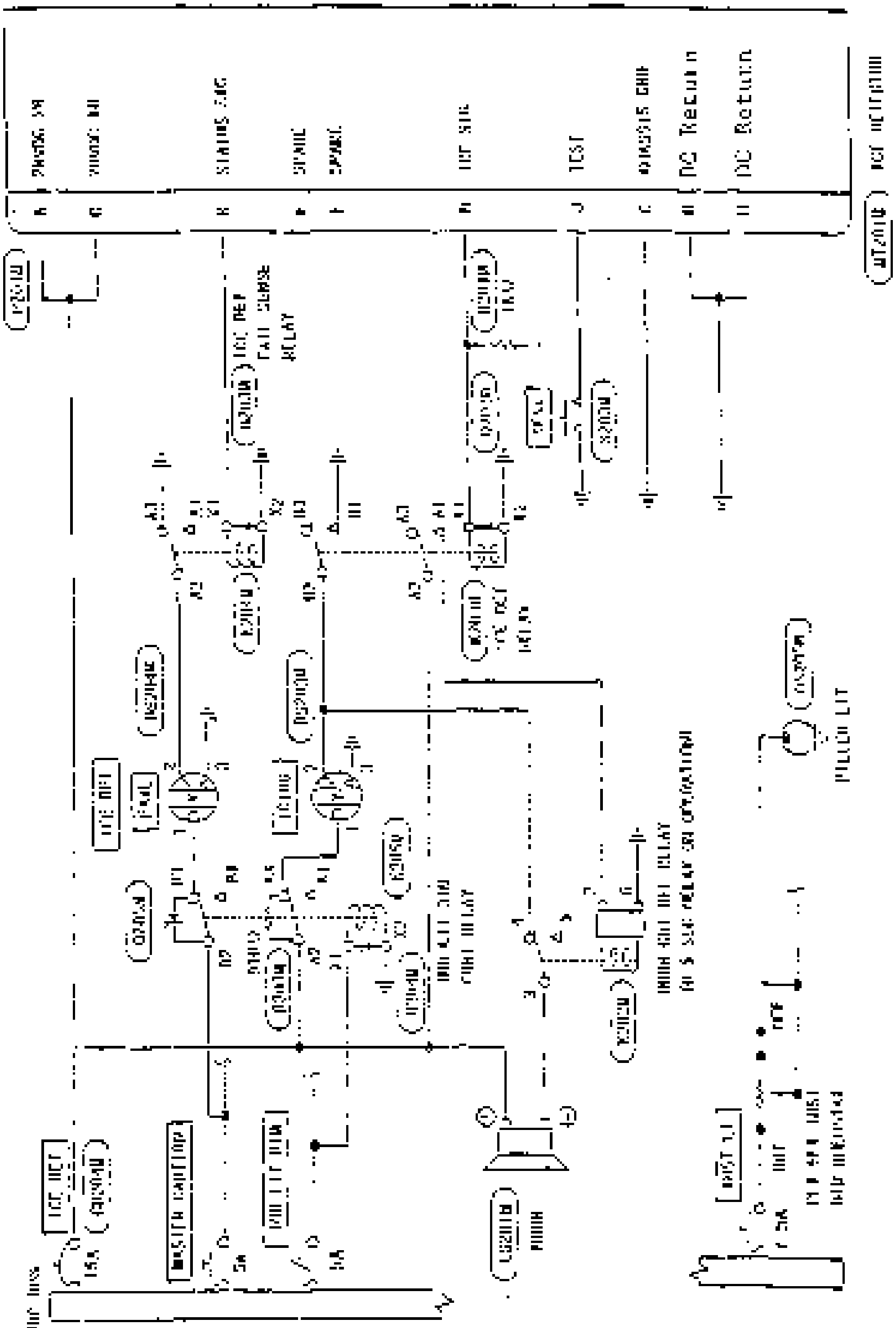
(1) Turn off electrical power to the indicator light

(2) Remove the cap from the indicator light body

(3) Remove the defective lamp and insert a new lamp. (MS25237-327)

(4) Reinstall the cap on the indicator light body. (finger tight)

(5) Reset the electrical power.



S/N 651, 657 THRU 699, 701 THRU 799

Fig 13 – 28 Ice detector system (1/2) (amps modified by S/B No.050/30-003A.)

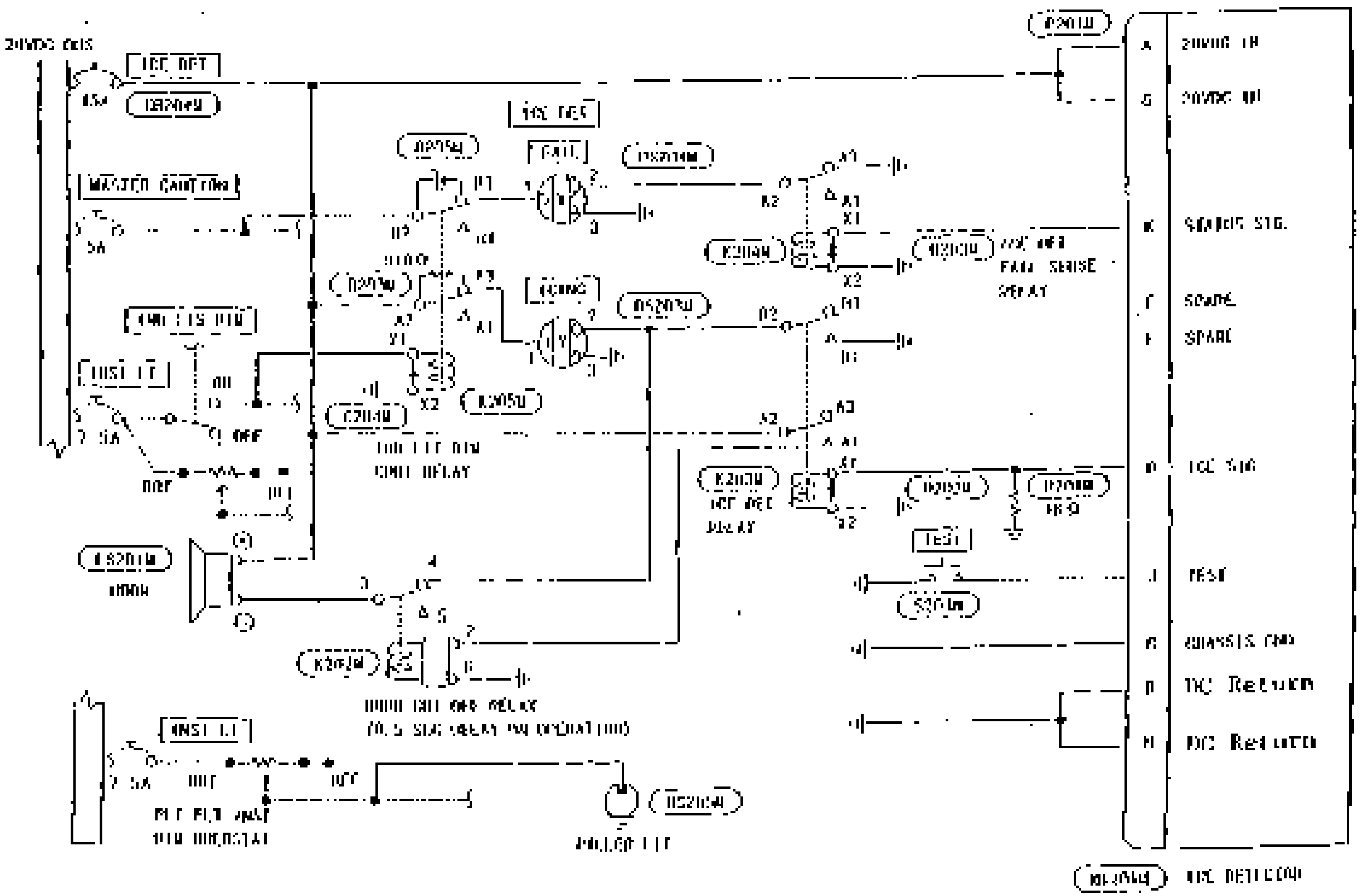


Fig 15-28 Ice detector system (212 series) (Replaced by SB No 080/99-003A)

S/N 682 ONLY

CHAPTER 14

SUPPLEMENTS

CHAPTER XIV

SUPPLEMENTS

SUP.1 TRIM-IN-MOTION ALERT SYSTEM

NOTE: This system is installed in accordance with MHI Service Bulletin 093/22-009 (current revision)

1. GENERAL

The Trim-in-Motion Alert System is installed to provide an aural alert that the elevator trim is operating in the airplane nose up direction with the flaps up. This system is designed specifically to warn of an inadvertent slow down while in cruise flight with the autopilot engaged. The alerts will occur with increasing frequency as the airspeed is reduced. The Trim-in-Motion aural tone sounds only when the trim is operating in the nose up direction, and only when the flaps are UP.

2. SYSTEM DESCRIPTION

The Trim-in-Motion Alert System consists of a one way cam directly connected to the elevator trim cable drum assembly (PN:016A-91620) located in the center pedestal of the cockpit, a micro switch actuated by the one way cam, an electronic/relay equipment assembly [PN:MU2-1203-( )] located in the forward cockpit area and mounted on the connection box (PN:016A-88141), and an alert horn mounted in the forward ceiling panel. 28VDC discrete signals from the micro switch are converted to an impulse by a timing circuit and intermittent voltage is sent to the alert horn.

3. MAINTENANCE

This system and its components require no periodic servicing.

4. FUNCTIONAL CHECK

Proper operation should be verified as part of the 100 hour check by performing following.

(1) Aircraft power ON

(2) With flaps UP, run elevator trim in the airplane nose up direction. Verify tone sounds.

(3) Run elevator trim in the nose down direction. Verify no tone.

(4) Select flaps 5, run elevator trim in the nose up direction. Verify no tone.

5. TROUBLE SHOOTING

If the system fails to function properly in the functional check, conduct the following:

(1) If the system fails to sound the alert horn.

(a) Cycle the "FLAP CONT" circuit breaker and verify it is IN. Verify the flap handle is in the "UP" position.

(b) Cycle the "TRIM IN MTN" circuit breaker (CB206M) and assure it is full IN.

(c) Listen for the micro-switch on the cable drum assembly to alternately make and break as the trim is run in the airplane nose up direction.

(d) Verify intermittent 28V electrical power is provided to pin 5 of the equipment assembly [PN:MU2-1203-( )] when the elevator trim is run in the airplane nose up direction. (See Fig. 14-1)

(e) Verify operation of the Trim-in-Motion horn by removing the connector to the equipment assembly and applying 28VDC power to socket 4 and ground to socket 3 of the connector (Do not apply power and ground to the pins of the equipment assembly). The horn should sound. If not, the horn has failed, and needs to be replaced.

(f) Verify operation of the flap interlock relay (K209M) by applying 28VDC power to pin 5 and 6 and ground to pin 7 of the equipment assembly. Place a voltmeter between pin 4 and ground. Relay operation should be audible and 28VDC* should be measured on the voltmeter. If voltage* is not measured at pin 4, either the flap interlock relay or the Timer Module (K208M) has failed and needs to be replaced.

(2) If the alert horn sounds when the elevator trim moves in both the airplane nose up and nose down direction.
The clutch in the cam assembly (PN:035A-991514-105) has locked up and should be replaced.

(3) If the alert horn sounds continuously:
The Horn Timer Module (PN.9514-2-0) has failed and must be replaced.

6. WIRING DIAGRAM

Fig 14-1 shows the wiring connections of the Trim-in-Motion Alert System.

*Voltage Spike is of 0.2 seconds duration and should be measured with an analog voltmeter.

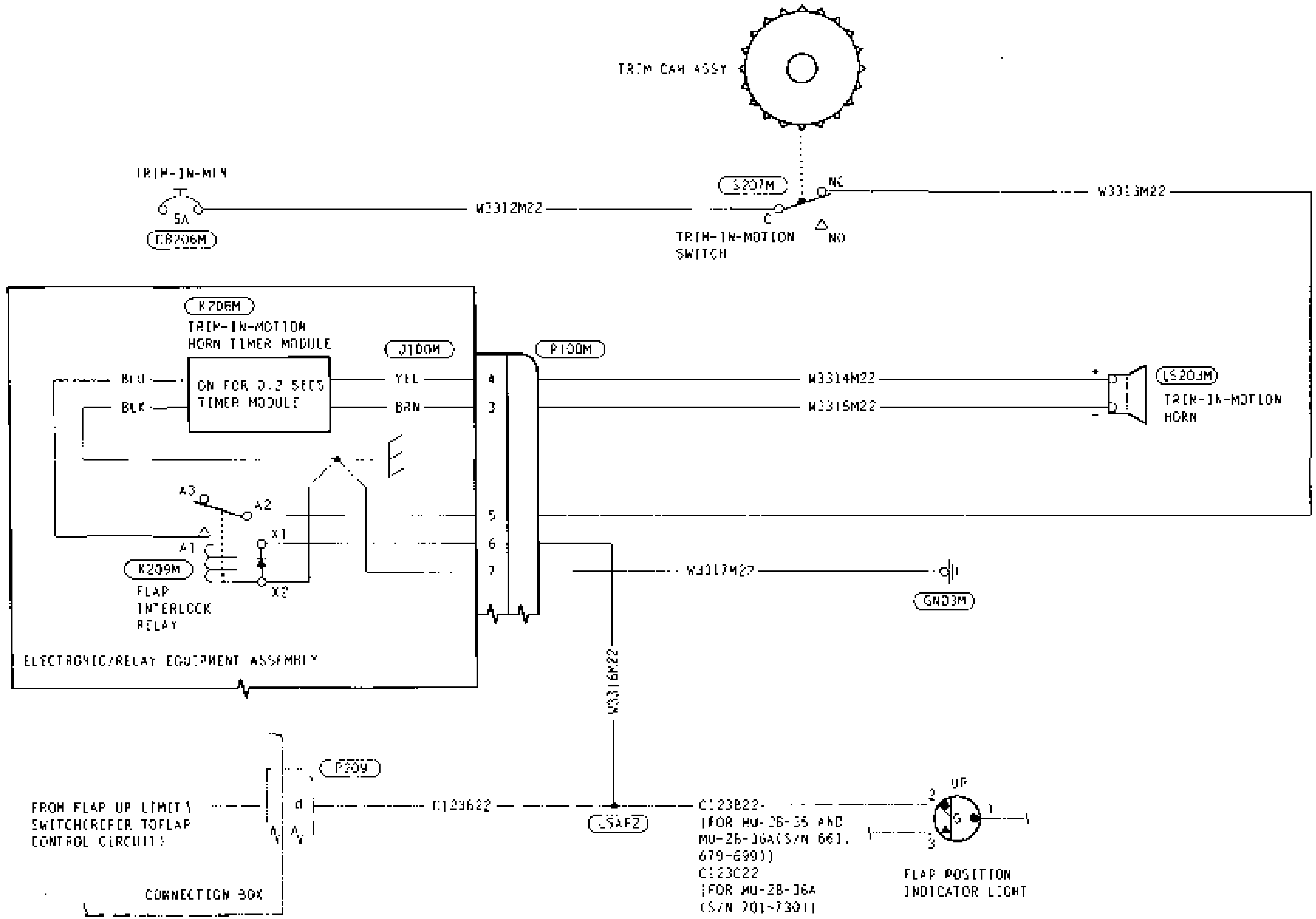


Fig 14-1 Trim-in-motion alert system

CHAPTER XIV

SUPPLEMENTS

SUP.2 AUTOMATIC AUTOPILOT DISCONNECT SYSTEM

NOTE: This system is installed in accordance with MHI Service Bulletin 093/22-009 (current revision)

1. GENERAL

The Automatic Autopilot disconnect System is designed to disconnect the autopilot when the airplane slows to 135 kias for normal operation with flaps UP. It is designed specifically to prevent the airplane from being inadvertently stalled by the autopilot. It is operable only with the flaps UP. A three second aural (horn) and visual light alert prior to the disconnect is provided by this system. The disconnect airspeed has been selected to provide a large margin above stall. To engage the autopilot, the airspeed must be approximate 10 knots above the disconnect airspeed.

2. SYSTEM DESCRIPTION

The Automatic Autopilot disconnect System consists of an electronic/relay equipment assembly [PN:MU2-1203-( )] located in the forward cockpit area and mounted on the connection box (PN:016A-88141), a pressure switch placed between the copilot's pitot and ship's alternate static pressure source and mounted on the equipment assembly, an alert horn mounted in the forward ceiling panel, an "A/P OFF" indicator light mounted in the windshield center post and integrated with the disconnect circuitry of the existing autopilot. The alert horn and "A/P OFF" indicator light provide an aural and visual alert for 3 seconds before disconnected. Additionally, the light remains illuminated until the pilot presses the "AP DISC" switch on the control wheel.

3. MAINTENANCE

This system and its components require no periodic servicing.

4. FUNCTIONAL CHECK

Proper operation should be verified as part of the 100 hour check by performing following.

NOTE

"A/P OFF" light should illuminate for approximately 1 second and alert horn sounds 0.5 second after turning on Radio Master Switches.

(1) Connect the pitot-static tester to the pilot's and copilot's pitot tube. Vent the static pressure to ambient.
(2) Aircraft power ON.
(3) For airplane equipped with an air driven pilot attitude gyro, disconnect engine bleed air tube from LH or RH engine and connect an auxiliary air pressure source to the tube. Apply air pressure (18 psi or above) and ensure the "INST VAC FAIL" annunciator light goes out and the attitude gyro is self-sustaining.
(4) For airplane equipped with an electrical driven pilot attitude gyro, ensure indicator flag disappears and the attitude gyro is self-sustaining.
(5) Turn on inverters, and radio master switches. Autopilot master switch to ON (if equipped).
(6) Set the flaps to UP. Adjust the pitot pressure to an airspeed of 150 kias or greater. Engage the autopilot. It should engage without any warnings.

(7) Slowly bleed off pitot pressure and note the pilot's indicated airspeed at which the disconnect warning horn sounds. With a slow rate of speed deceleration, this should be approximately 128±2 kias (for MU-2B-35), 133±2 kias (for MU-2B-36A) (due to position error of alternate static source, this number on the ground will result in 135 kias (for MU-2B-35), 140 kias (for MU-2B-36A) in flight). An average of several runs may be necessary to verify the airspeed. Pilot's airspeed indicator error must be taken into account for this check to be valid.

(8) Verify that the warning horn sounds for approximately 3 seconds prior to disconnect and for an additional 1 second after disconnect. Ensure "A/P OFF" light stays illuminated after aural warnings stop. Press "AP DISC" switch on control wheel to first detent, and ensure "A/P OFF" light goes off.

(9) Increase airspeed to approximately 10 knots greater than the noted disconnect airspeed, and verify that the autopilot can be engaged.

(10) With the autopilot engaged and the airspeed above 150 kias, select flaps 5° and repeat the test by slowly bleeding off pitot pressure. The autopilot should not automatically disengage regardless of the in dicated airspeed. Repeat at flaps 20° and 40°.

5. TROUBLE SHOOTING

Refer to Wiring Diagram.

If the system fails to disconnect the autopilot, conduct the following:

(1) Verify autopilot disconnect horn is functioning by cycling the "AUTOPILOT DC" circuit breaker.

(2) Set the flap handle to the "UP" position and cycle the "FLAP CONT" circuit breaker.

(3) Verify that connector P100M is properly inserted and locked into the equipment assembly [PN:MU2-1203-( )] located on the forward side of connection box.

(4) Engage the autopilot and verify that 28VDC is available to the white lead of pressure switch. If 28VDC is not available on this circuit (check at available point), either K213 or K209M relay has failed and must be replaced.

(5) Increase pitot pressure on the pilot's and copilot's systems above 150 kias and verify 28VDC is available downstream of the red lead of pressure switch. If no 28VDC is available on this circuit (check at available point), the pressure switch has failed and must be replaced.

(6) If the system still fails to function properly, remove the equipment assembly [PN: MU2-1203-( )] and return it to vender for repair. Replace with a new or overhauled unit.

6. WIRING DIAGRAM

Fig 14-2 and Fig 14-3 shows the wiring connections of the Automatic Autopilot Disconnect System.

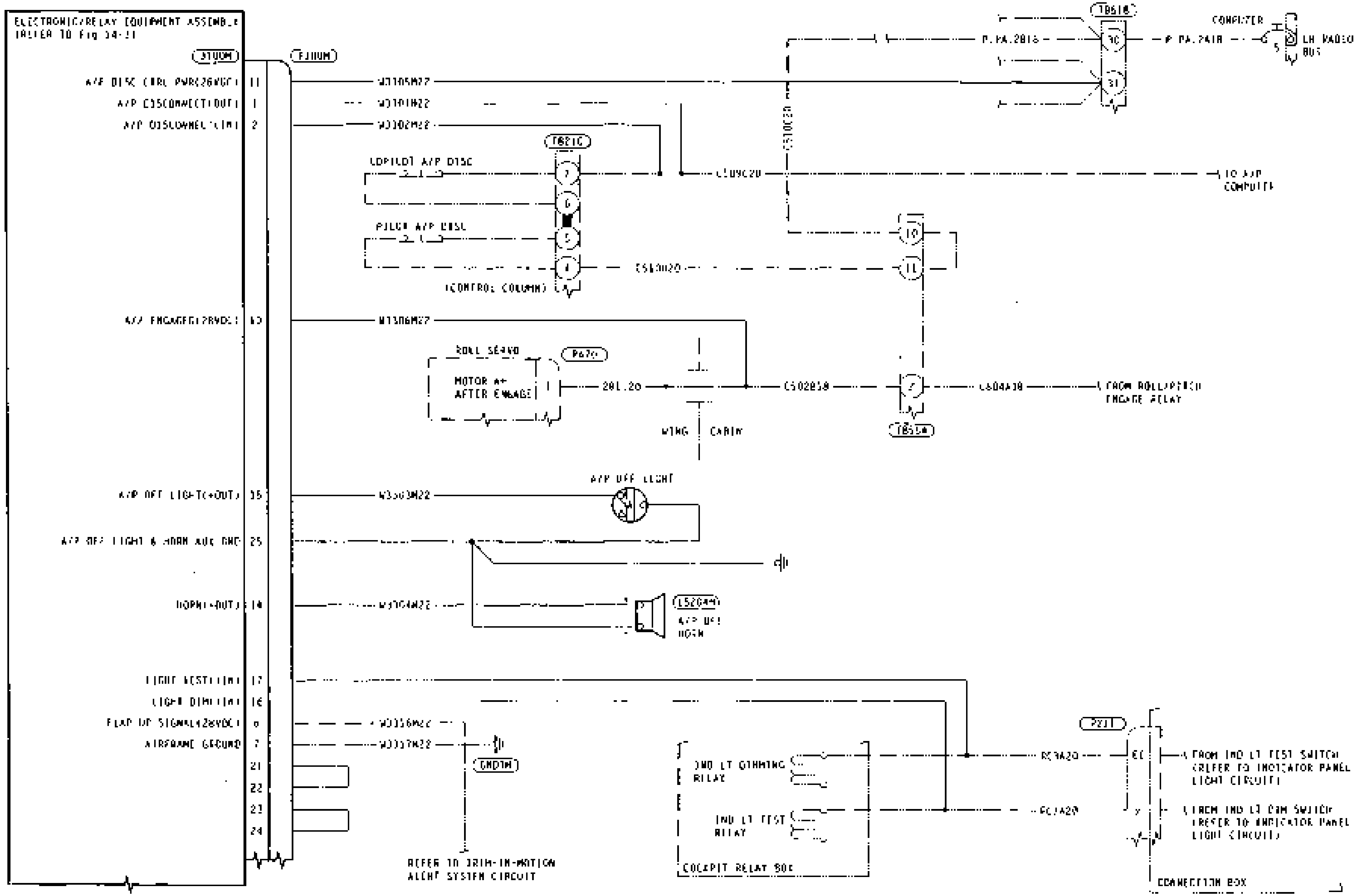


Fig 14-2 Automatic autopilot disconnect system
(for A/C equipped with Bendix M-4D or sperry SPZ-200 Autopilot)

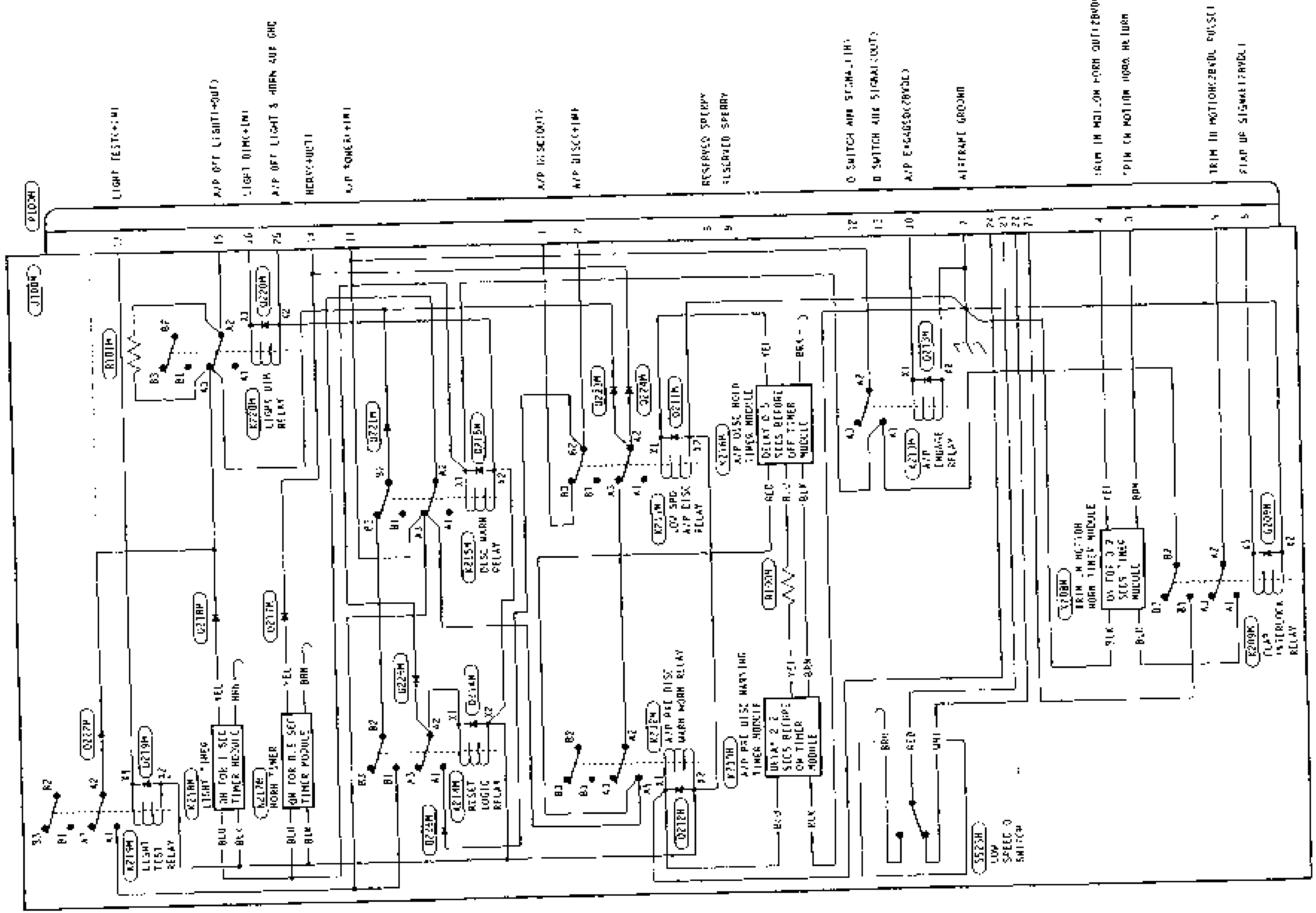


Fig 14-3 Electronic/relay equipment assembly
(for A/C equipped with Bendix M-4D or sperry SPX-200 Autopilot)

CHAPTER XIV

SUPPLEMENTS

SUP.3 PNEUMATIC DE-ICE MONITORING SYSTEM

NOTE: This system is installed in accordance with MHI Service Bulletin 096/30-004 (current revision)

1. GENERAL

This Pneumatic De-Ice Monitoring System is designed to inform the pilot that both wing and tail pneumatic de-ice boot systems are receiving pressure when the "WING DE-ICE" switch is selected "ON". The wing de-ice light will illuminate only when pressure is applied to both the wing and tail boot pneumatic lines. In automatic operation, the light will cycle approximately 6 seconds on and 3 minutes off. If the wing de-ice system is selected "ON" and the light does not illuminate, it indicates one or both systems are not receiving pressure and icing conditions must be avoided.

2. SYSTEM DESCRIPTION

The Pneumatic De-Ice Monitoring System consists of two pressure switches located in the pneumatic lines feeding the wing and tail boots. Since these lines are separate, a switch in each line is needed to assure operation of both systems. In normal operation, the "WING DE-ICE" light will illuminate only when both systems are operating properly. If the wing de-ice system is selected "ON" and the light does not illuminate, it indicates that one or both systems are not receiving pressure and icing conditions must be avoided. Each pressure switch is checked for failure in the closed position during regularly scheduled maintenance (100 hr. inspection). Note that operation of the wing boots may be visually checked from the cockpit but that ground personnel are required to check the tail boots.

3. MAINTENANCE

As part of the normal 100 hour inspection open the drain line provided in the cross-ship de-ice pressure line to the tail at F. STA 9903 and drain any water collected. Reseal the drain plug. Water in this line is a sign of deteriorated tail de-ice boots or a leak in the pressure lines. Repair before returning the aircraft to service

4. FUNCTIONAL CHECK (See Fig 14-4)

(1) Verify the pressure switches have not failed in the closed position by checking for continuity across terminals on terminal board with referring Table 14-1. These should be open circuits.

Table 14-1 Continuity check of pressure switch

Model	TB	Location	Continuity Check	
MU-2B-35	305	Just forward the hell-hole access door	1 and 3	2 and 3
MU-2B-36A	308	Just above the hell-hole access door	10 and 11	10 and 12

(2) Perform normal preflight checks from the Airplane Flight Manual. Verify operation of all de-ice boots using ground personnel to observe.

5. TROUBLE SHOOTING (See Fig 14-4)

(1) If the "WING DE-ICE" light fails to illuminate when the de-ice switch is selected:

(a) Verify at least one engine is running at 96% RPM or greater. Increase the engine power and select wing de-ice to see if the operation is normal with increased engine power.

(b) Cycle the "WING DE-ICE" (5A) circuit breaker.

(c) Have ground personnel visually check the wing and tail de-ice boots for inflation and deflation when the cockpit switch is selected.

(d) If the de-ice boots are cycling normally:

a) Check wiring to the pressure switches.

b) Check operation of the pressure switches by pressurizing to 15 psi and verifying continuity across terminals 1 and 3 (or 10 and 11), for the wing pressure switch and terminals 2 and 3 (or 10 and 12), for the tail pressure switch. Replace any defective pressure switch.

(e) If the de-ice boots are not cycling normally:

a) Check the regulated pressure from the de-ice distributor valve with a pressure gage. Replace valve if the pressure is less than 15 ± 1 psi. Conduct any other tests described in the maintenance manual.

(2) If the "WING DE-ICE" light illuminates without corresponding inflation of one of the boots:

(a) Repair the de-ice boot that is inoperative.

(b) Replace the pressure switch monitoring that portion of the de-ice system.

6. WIRING DIAGRAM

Fig 14-4 shows the wiring connections of the wing de-ice circuits.

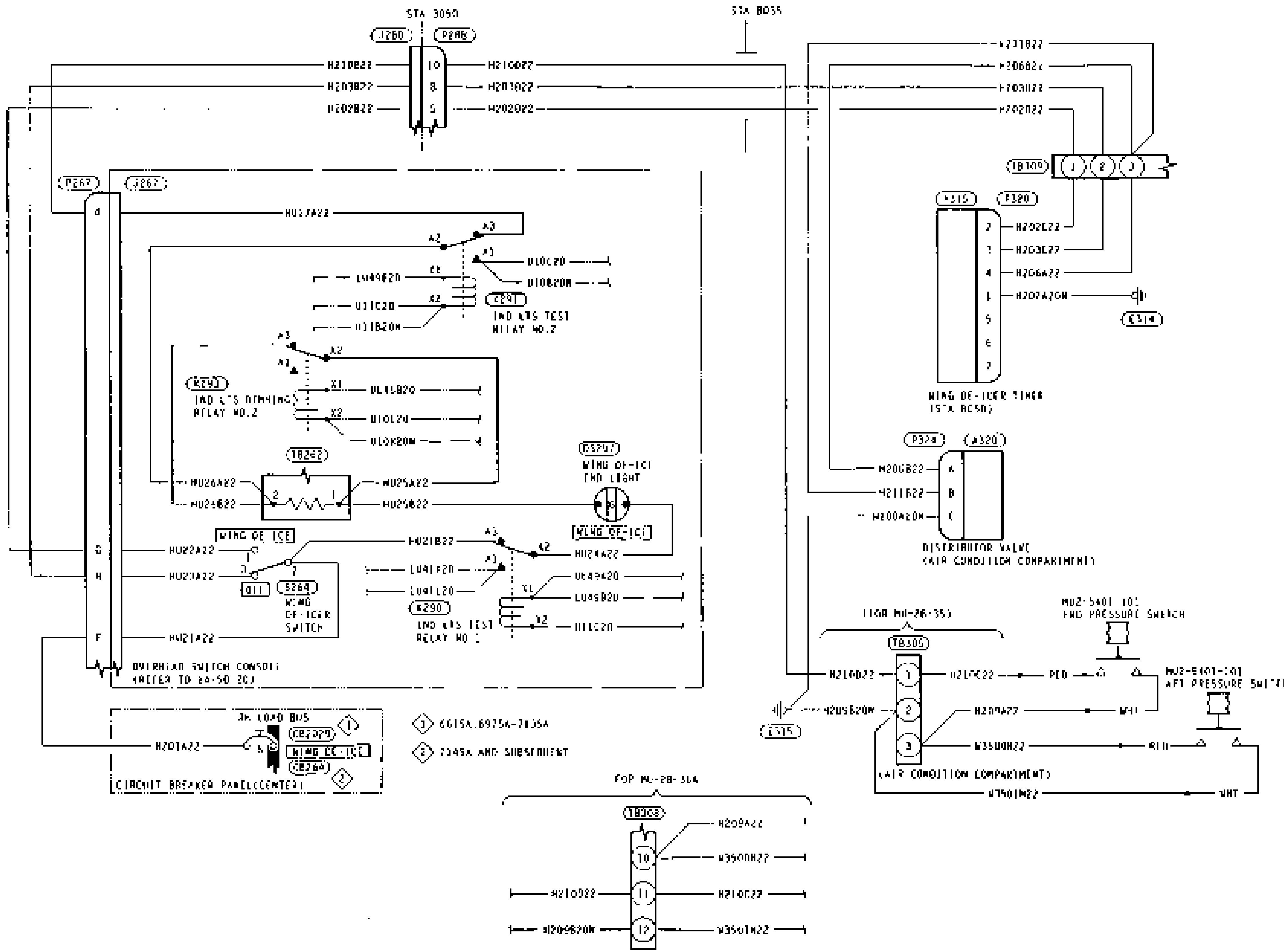
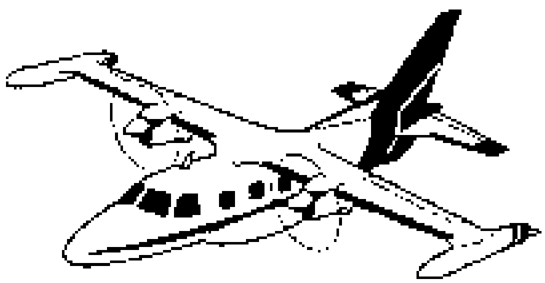


Fig 14-4 Wing de-ice circuits

MFG.

INTRO

MU-2B-36A WIRING DIAGRAM MANUAL



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MITSUBISHI
HEAVY INDUSTRIES, LTD.

(LOGO CHANGE 27 Aug. 1999)

APPLICABLE TO AIRPLANE SERIAL NUMBER 661SA
697SA THROUGH 699SA AND 701 THROUGH 730.

WIRING DIAGRAM MANUAL

RECORD OF REVISIONS

Retain this record in the front of the manual. Upon receipt of a revision, remove the pages being revised and insert the new pages. Enter the revision number (and/or revision date), date inserted and initials of the person incorporating the revision.

REV. NO. OR DATE	INSERTION DATE	BY	REV. NO. OR DATE	INSERTION DATE	BY	REV. NO. OR DATE	INSERTION DATE	BY
No. 1	8/27/80	RW						
No. 2	9/1/80	MAI						
8/27/99	11-13-00	JB						

RECORD OF REVISIONS

MFG REV NO	DESCRIPTION	ISSUE DATE	ATP REV DATE	INSERTED BY
8/27/99	SEE LIST OF EFFECTIVE PAGES	8/27/99	11/13/00	ATP/JB

RECORD OF TEMPORARY REVISIONS

TEMP REV NO	DESCRIPTION	ISSUE DATE	ATP REV DATE	INSERT BY	DATE REMOVED	REV INCOR	REMOVE BY

CH/SEC/SUB	PAGE	SHEET	DATE
Title	* 1		Aug. 27/99
Highlights Rev 1	1		Aug. 1/79
Record of Revisions	1		Aug. 1/79
Effective Pages	* 1		Aug. 27/99
	** 2		Aug. 27/99
Introduction	1		Original
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	3		Original
	4		Original
	5		Original
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21-00-00	1	1/5, 2/5	Original
	1	3/5	Aug. 1/79
	1	4/5, 5/5	Original
21-00-10	1	1/1	Original
	2	1/1	Original
24-20-00	1	1/3 thru 3/3	Aug. 1/79
	2	1/3 thru 3/3	Aug. 1/79
	3	1/3 thru 3/3	Aug. 1/79
24-30-00	1	1/8	Original
	1	2/8	Aug. 1/80
	1	3/8 thru 8/8	Original
	2	1/8 thru 8/8	Original
	3	1/10 thru 6/10	Original
	3	7/10	Aug. 1/80
	3	8/10 thru 10/10	Original
24-30-10	1	1/2, 2/2	Original
	2	1/2, 2/2	Original
24-50-10	1	1/2	Original
	1	2/2	Aug. 1/79
	2	1/2	Original
	2	2/2	Aug. 1/79
	3	1/2, 2/2	Original
24-60-20	1	1/4, 2/4	Original
	1	3/4, 4/4	Aug. 1/79
24-60-30	1	1/1	Aug. 1/79
26-10-00	1	1/1	Original
	2	1/1	Original
27-00-00	1	1/1	Original

CH/SEC/SUB	PAGE	SHEET	DATE
27-10-00	1	1/2, 2/2	Original
	2	1/2, 2/2	Original
27-10-10	1	1/1	Original
27-30-10	1	1/2, 2/2	Original
27-50-00	1	1/2	Aug. 1/79
	1	2/2	Original
	2	1/2	Aug. 1/79
	2	2/2	Original
	3	1/2	Aug. 1/79
	3	2/2	Original
	4	1/2	Aug. 1/79
	4	2/2	Original
28-20-00	1	1/1	Original
28-20-10	1	1/4, 2/4	Aug. 1/80
	1	3/4	Aug. 1/79
	1	4/4	Original
	2	1/4, 2/4	Aug. 1/80
	2	3/4	Aug. 1/79
	2	4/4	Original
28-40-00	1	1/2, 2/2	Original
	2	1/2, 2/2	Original
28-40-10	1	1/1	Original
30-10-00	1	1/1	Original
30-20-00	1	1/2, 2/2	Original
30-20-10	1	1/2, 2/2	Original
	2	1/2, 2/2	Original
30-30-00	1	1/2, 2/2	Aug. 1/79
30-40-00	1	1/1	Original
	2	1/1	Original
30-40-10	1	1/4	Aug. 1/79
	1	2/4 thru 4/4	Original
	2	1/3 thru 3/3	Original
	3	1/3 thru 3/3	Original
	4	1/3 thru 3/3	Original
30-60-00	1	1/2	Original
	1	2/2	Aug. 1/79
	2	1/2	Original
	2	2/2	Aug. 1/79
30-80-00	1	1/1	Original
31-20-00	1	1/1	Original
32-30-00	1	1/4 thru 4/4	Original
32-60-00	1	1/4	Original
	1	2/4, 3/4	Aug. 1/79

* Changed
** Added

CH/SEC/SUB	PAGE	SHEET	DATE	CH/SEC/SUB	PAGE	SHEET	DATE
32-60-00	1	4/4	Original				
	2	1/4	Original				
	2	2/4	Aug. 1/79				
	2	3/4, 4/4	Original				
33-10-50	1	1/5 thru 5/5	Original				
33-10-10	1	1/1	Original				
33-10-20	1	1/1	Aug. 1/79				
	2	1/1	Aug. 1/79				
33-20-00	1	1/3 thru 3/3	Original				
33-40-00	1	1/2, 2/2	Original				
33-40-10	1	1/1	Original				
33-40-20	1	1/1	Original				
33-40-30	1	1/1	Original				
34-20-00	1	1/1	Original				
	2	1/1	Original				
36-20-00	1	1/1	Original				
37-25-00	1	1/1	Original				
52-15-00	1	1/1	Original				
52-70-00	1	1/1	Original				
61-25-00	1	1/1	Original				
73-30-10	1	1/1	Original				
	2	1/1	Original				
	3	1/1	Original				
76-00-00	1	1/5	Original				
	* 1	2/5	Aug. 27/90				
	1	3/5 thru 5/5	Original				
77-00-00	1	1/5, 2/5	Original				
	1	3/5	Aug. 1/79				
	1	4/5, 5/5	Original				

* Changed
** Added

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TABLE OF CONTENTS

TABLE OF CONTENTS

ALPHABETICAL INDEX & WIRE CODES

ALPHABETICAL INDEX & WIRE CODES

MU-2B-36A
WIRING DIAGRAM MANUAL
INTRODUCTION

ATA CHAPTERS

The wiring diagrams in this manual are basically organized according to the ATA Specification 100. The numbering consists of three sets of two numbers. Example for 24-30-10 would be

```
┌──Electrical Power (ATA Chapter)
│         ┌───────DC Generation (ATA Sub-System)
24  -   30   -  10 ──────── Battery Overtemp (Mitsubishi) Sub-System of
                            DC Generation
```

Sub-Systems are not always broken out per ATA Code, for instance 24-30-00 includes -30 (DC Generation), -40 (External Power) and -50 (Load Distribution). When there is no specific Sub-System listing such as External Power, refer to other diagrams within the chapter.

PAGE NUMBERS - SHEET NUMBERS

Page numbers start at 1 for each ATA Chapter number, Sub-System number and Sub-Sub-System number.

The page number is used to separate changes in effectivity with (Page 1) representing the original configuration and (Page 2) and on showing changes for subsequent aircraft. The page number will represent the block of serial numbers listed immediately under the title of the page. Additional serializations may appear on the page but will have no effect on the page number.

The majority of the diagrams are broken down into segments which are identified by sheet numbers such as (Sheet 2/5) which means sheet 2 of 5.

INDIVIDUAL COMPONENT OR WIRE SERIALIZATION

Minor changes such as an added wire or reidentification of a component will normally be shown as a part of the basic diagram but will be flagged by a diamond-shaped code (◇1). The serialized effectivity will be shown on each sheet whenever a code is used and the code will remain constant for all sheets applicable to a particular page.

When tracing a coded wire (◇1) on a diagram, make sure that the terminal point or end of traced portion is coded the same.

WIRE IDENTIFICATION

All wires are normally identified with a letter and number code unless the wire is less than six inches in length. The code is either printed, stenciled or banded to each wire segment between two points in a circuit. The letter number code will identify the wire to a system and will give the wire gage.

The following wire identification and explanation is a typical example.

```
P1   20   A   20   N   (SA)-------San Angelo Modification
|    |    |   |    L---Wire Function
|    |    |   L---Wire Gage
|    |    L---Wire Segment
|    L--- Wire Number
L---Circuit Function
```

CIRCUIT FUNCTION

Circuit Function codes are listed (next page) and will serve as an aid when trying to determine the purpose of a particular wire in the aircraft. These codes are also listed beside the Alphabetical Index for association with each diagram.

The letter code is a general designation and the number will refer to a particular circuit.

WIRE SEGMENT

The wire segment letter identifies each segment (portion of a wire between two connecting points) and will normally start with "A" on the segment connecting to the power source, signal source or ground. Each connecting segment will be assigned the next letter alphabetically. Letters "I" and "O" are not used.

WIRE SIZE

Numerical digit(s) stating the wire gage for each segment. Coaxial cables and thermocouple wires will not be identified as to wire size.

WIRE FUNCTION

The last alphabetical letter further identifies the wire segment function. The ground or conductor material letter indicates the following:

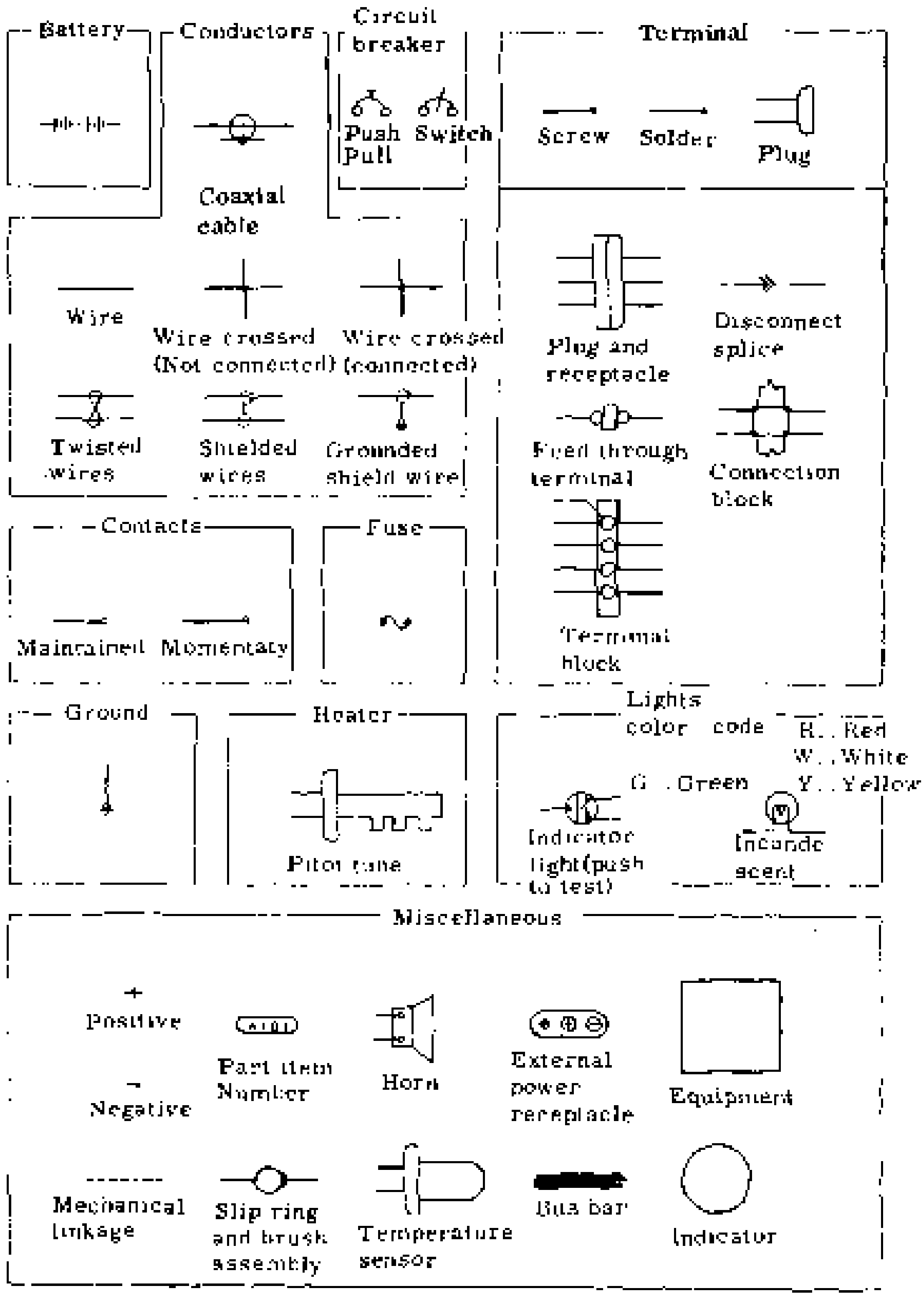
Letter	Indicates
(SA)	San Angelo Modification or addition.
N	That the wire completes a ground circuit to ground.
T	That the wire is in a thermocouple circuit.
(AL)	Alumel wire in the segment
(CR)	Chromel wire in the segment.

CIRCUIT FUNCTION CODES

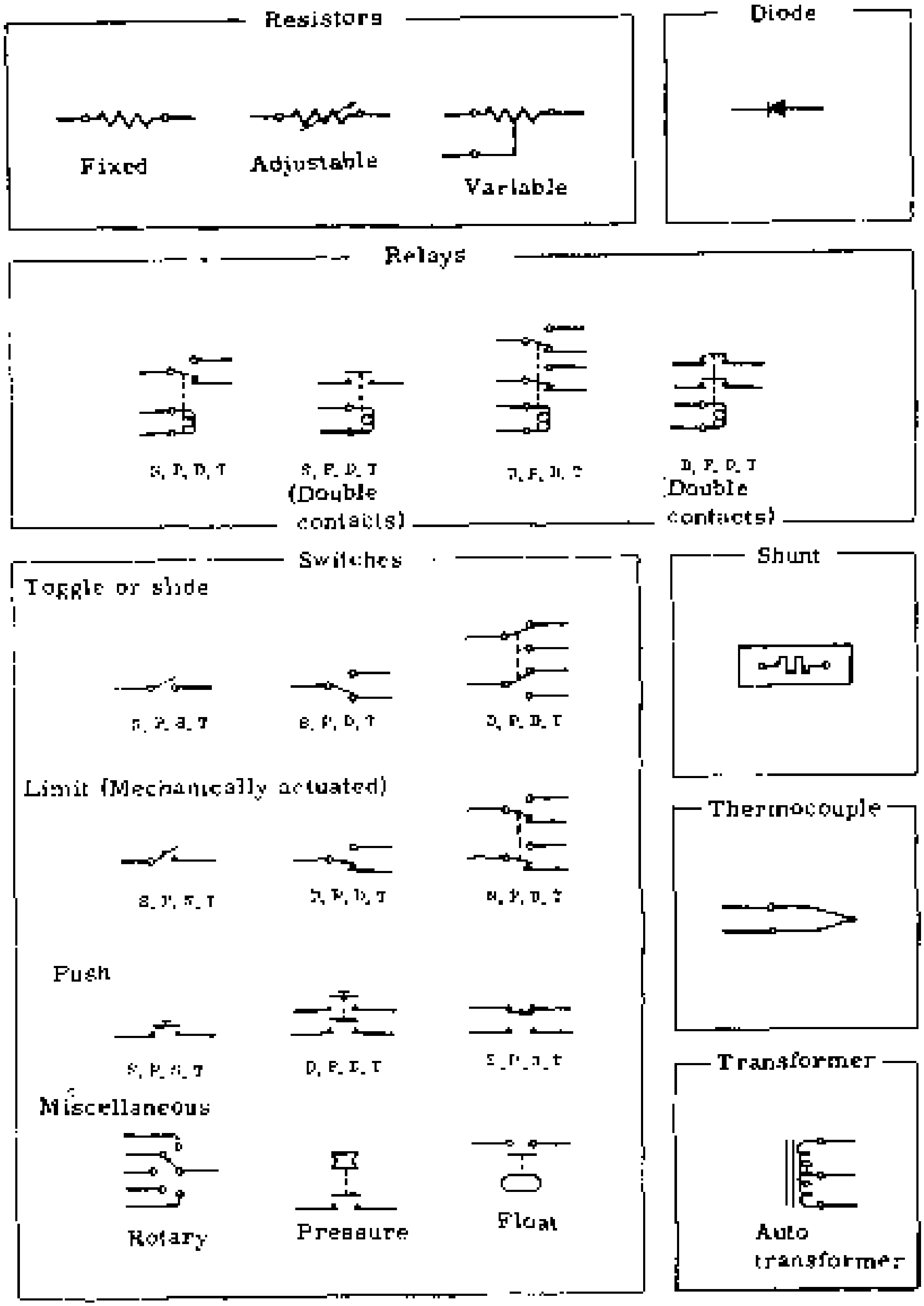
CIRCUIT FUNCTION CODE	CIRCUIT FUNCTION/SYSTEM
C	Flight Controls
C1	Flap Control
C2	Trim Aileron Control
CW	Control Wheel
D	Instruments (Except Flight and Engine)
D1	O.A.T., Outside Air Temperature
D3	Trim Aileron Position Indication
E	Engine Instruments
E1	Engine Instruments
E2	Fuel Quantity
F	Flight Instruments
F1	Turn and Bank Indication
F3	Pitot and Static Port Anti-Ice
G	Landing Gear
G1	Landing Gear Control
G1	Landing Gear Position Indication
H	Heating, Cooling and Anti-Ice
H1	Engine Anti-Ice, Propeller De-Ice
H2	Wing De-Ice
H3	Air Conditioning
H4	Door Seal
H5	Heated Windshield - Anti-Ice
H6	Oil Cooler Inlet - Anti-Ice
K	Engine and Propeller Control
K1	Engine and Propeller Control
K2	Engine and Propeller Control
K7	Propeller Synchrophaser
K9	Propeller Synchrophaser
L	Lighting
L1	Navigation Lights
L2	Fuselage Landing Lights
L3	Room Lights
L4	Instrument and Switch Panel Lights
L5	Trim Indication Lights
L6	Instrument Lights
L7	Strobe Lights
L8	Tip Tank Taxi Lights

CIRCUIT FUNCTION CODE	CIRCUIT FUNCTION/SYSTEM
M	Miscellaneous Electric
M1	Windshield Wiper Control
M2	Toilet
M3	Cigar Lighter
P	DC Power
P1	DC Power
P2	DC Power
P5	DC Power
Q	Fuel and Oil
Q1	Main Tank Fuel Control
Q2	Fuel Transfer Control
Q3	Fuel Transfer Control
V	DC Power and DC Control for AC
V1	AC Power Source
V2	AC Power Source
X	AC Power
X1	AC Power Source
X2	AC Power Source
W	Warning and Emergency (Master Caution)
W1	Vacuum Pressure Warn
W2	Door Lock Warn, Master Caution
W3	Fuel Pressure Indication/Warn
W4	Stall Warn
W5	Engine Fire Detect
W6	Engine Fire Detect
W8	Fuel Filter Bypass/Boost Pump Failure
W9	Cabin Pressure Warn
W20	Battery Overtemp
W21	Defog Overtemp
BT	Battery Overtemp
FPM	Fuel Pressure Meter
OC	Oil Cooler
HWS	Heated Windshield
TTL	Tip Tank Light (Taxi)
P	Power

ELECTRICAL SYMBOLS

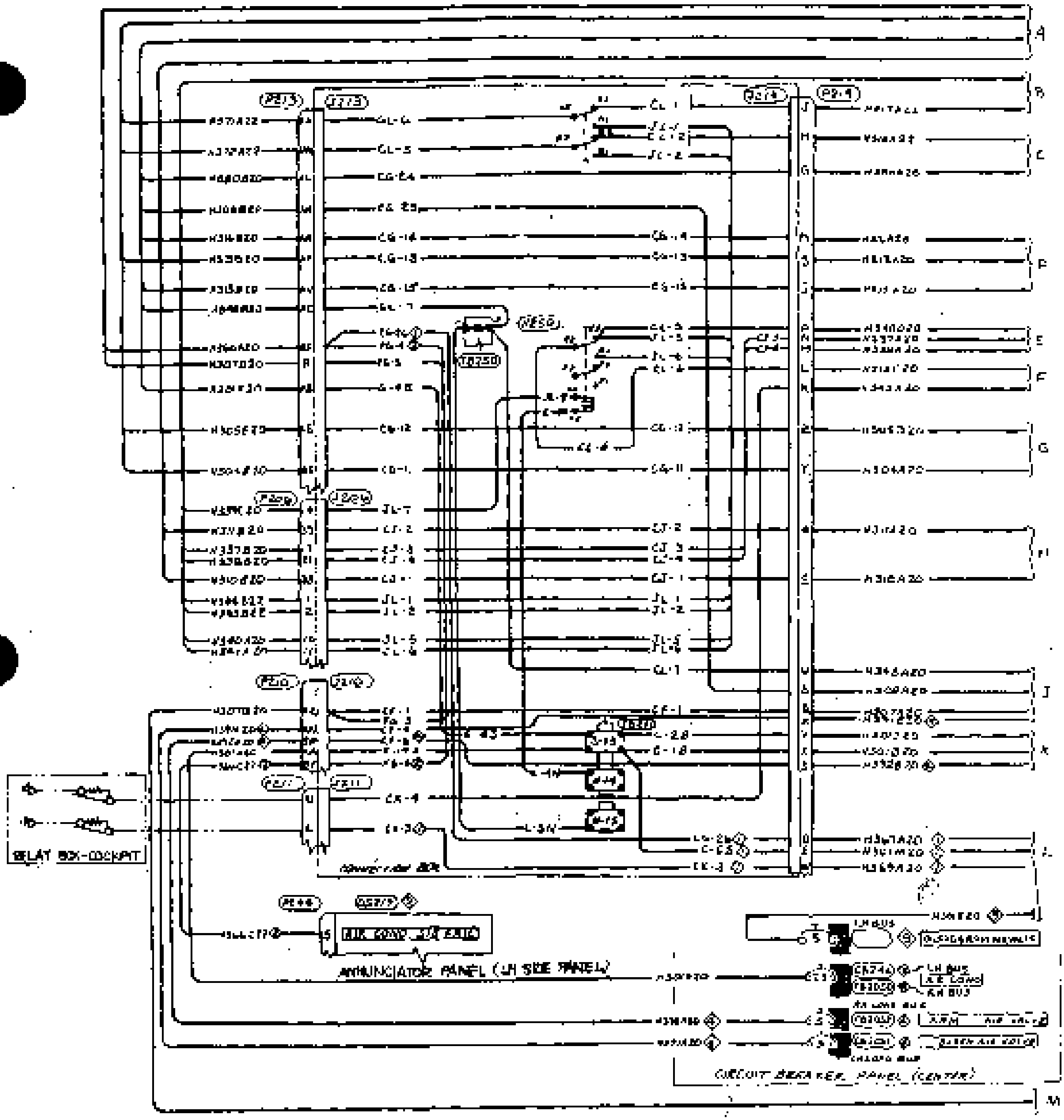


ELECTRICAL SYMBOLS



CHAPTER 21

AIR CONDITIONING



⟨1⟩ S/N 662SA

⟨2⟩ S/N 697SA and Subsequent

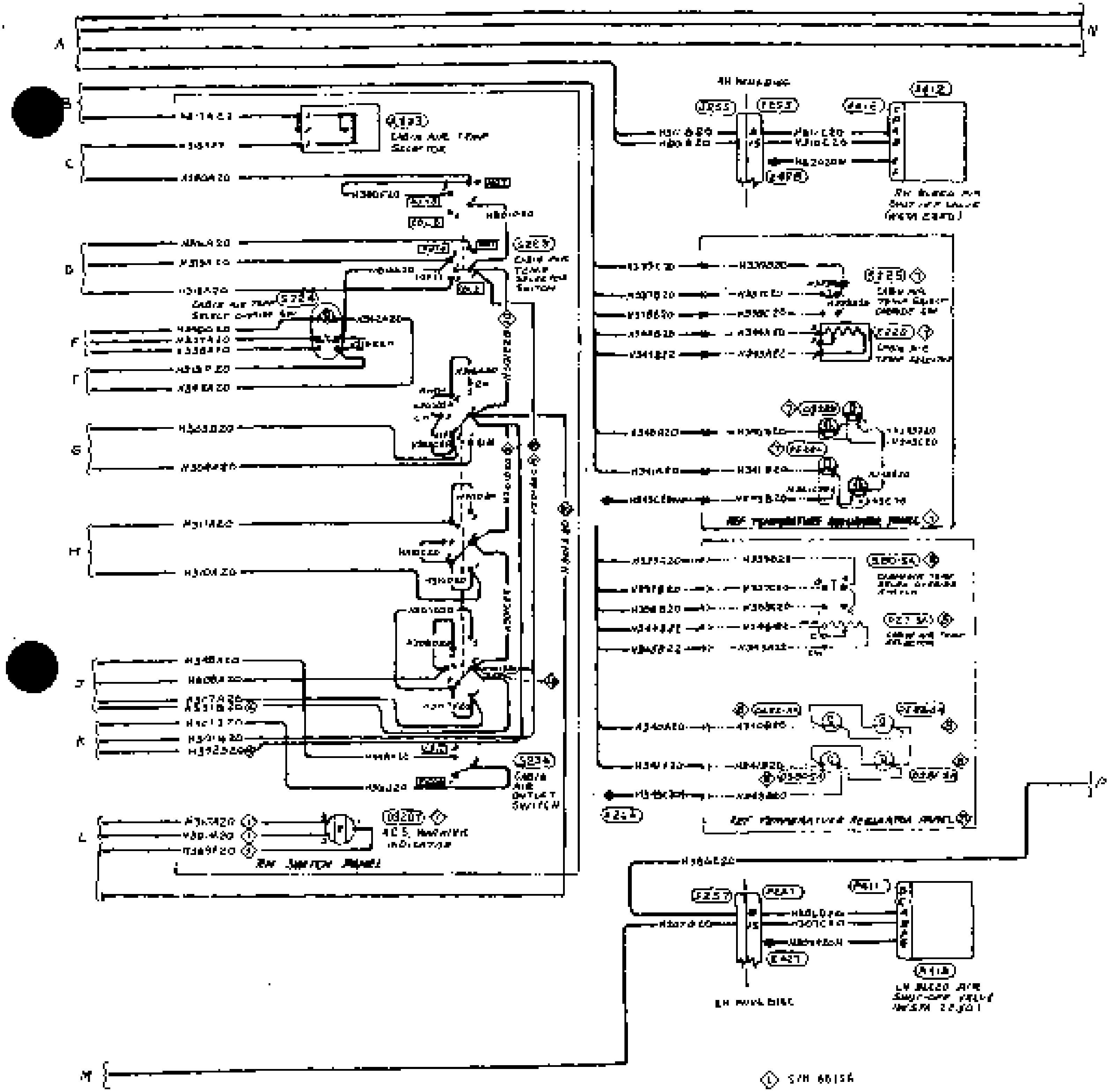
⟨3⟩ S/N 661SA, 697SA thru 713SA

⟨4⟩ S/N 714SA and Subsequent

⟨9⟩ S/N 700SA thru 713SA plus S/N per ⟨10⟩ If modified per SB207/26-051

⟨13⟩ S/N 661SA, 697SA thru 709SA not modified per [illegible]

AIR CONDITIONING CONTROL
Aircraft S/N 661SA, 697SA - 730SA



◇ S/N 7035A thru 7135A also S/N per ◇ if modified per SB007/21-005

◇ S/N 661SA, 697SA thru 7035A not modified per SB007/21-001

◇ S/N 661SA

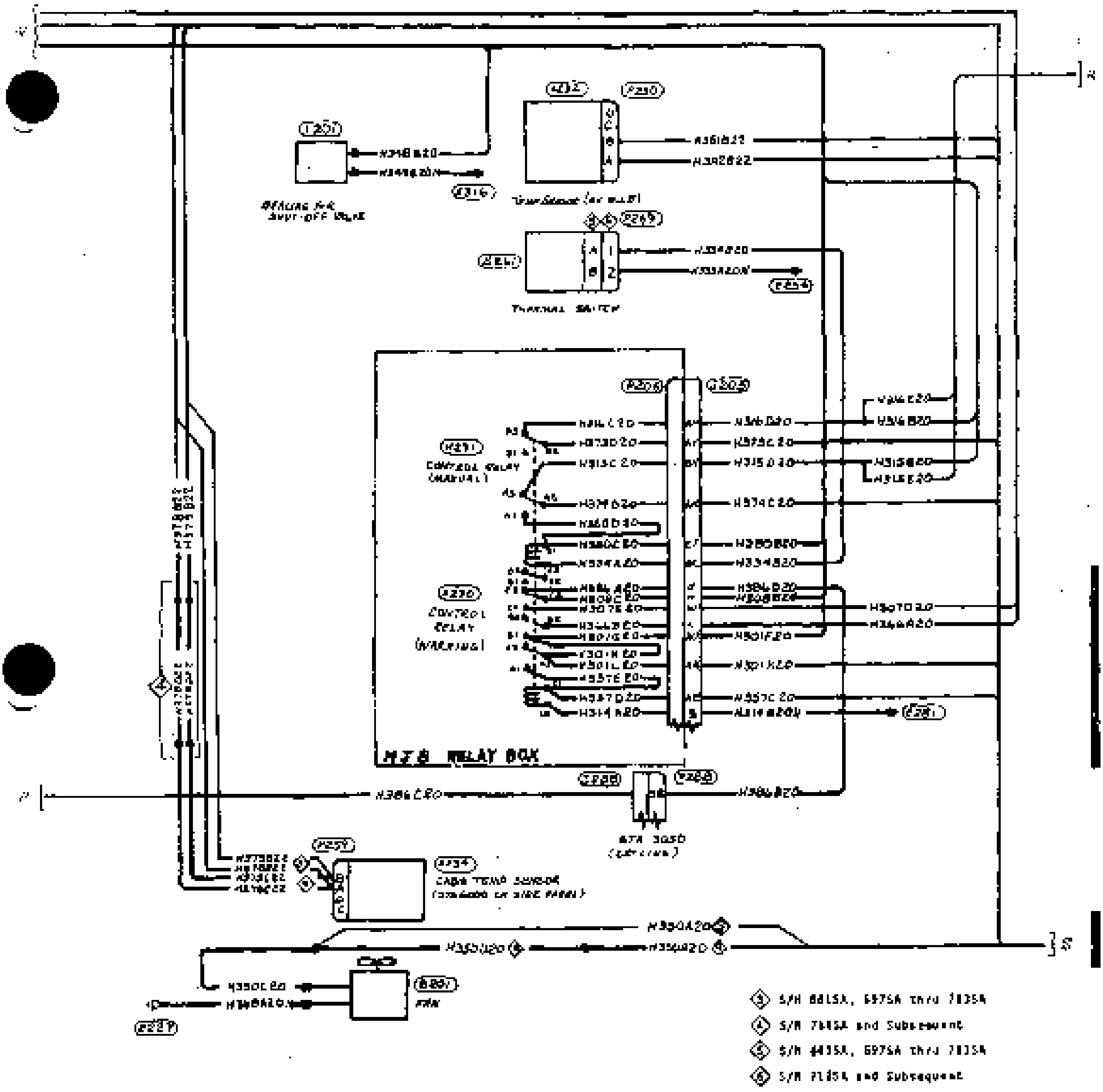
◇ S/N 661SA, 697SA thru 713SA

◇ S/N 714SA and Subsequent

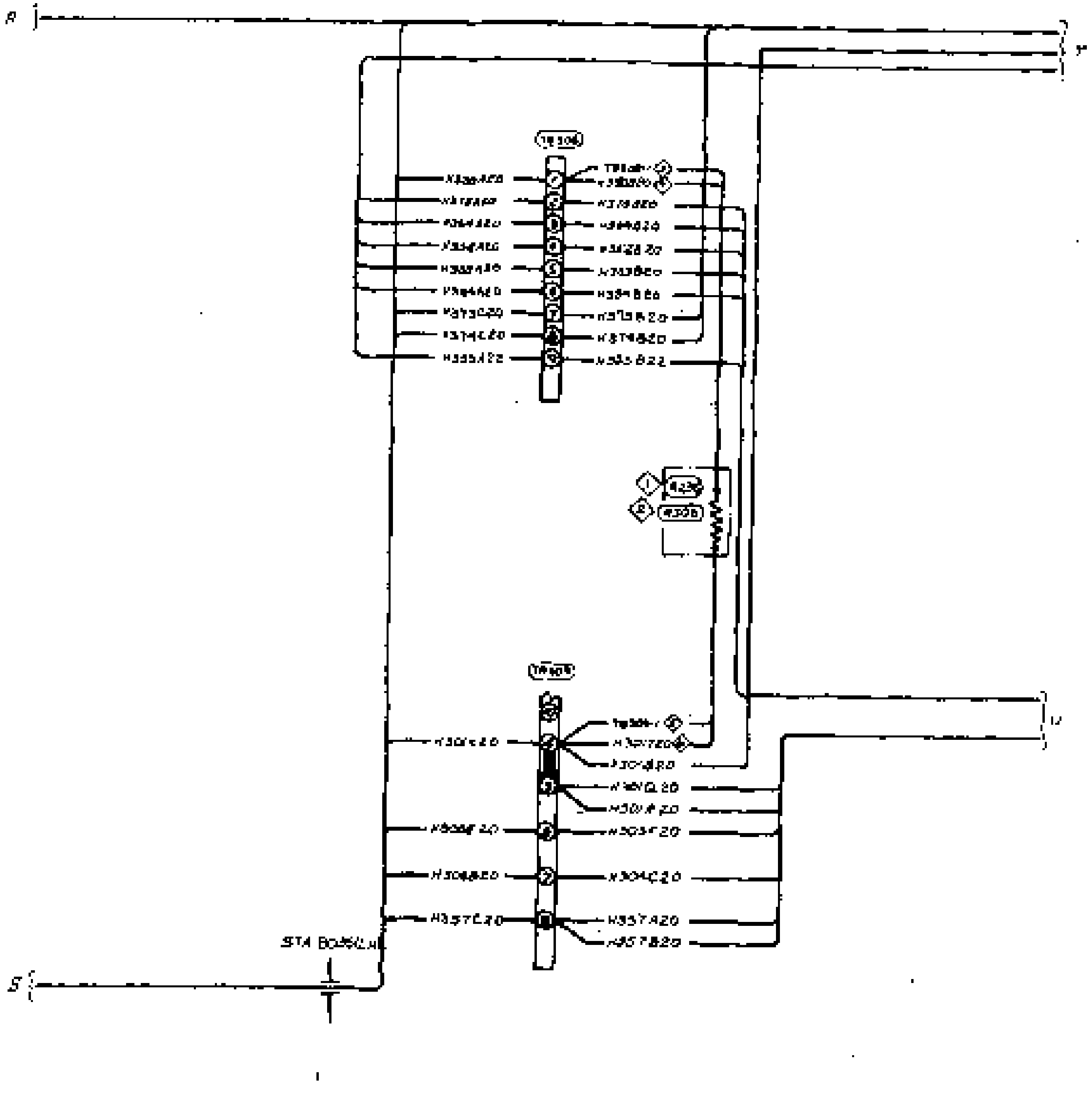
◇ S/N 661SA, 697SA thru 706SA

◇ S/N 707SA and Subsequent

AIR CONDITIONING CONTROL
Aircraft S/N 661SA, 697SA - 730SP

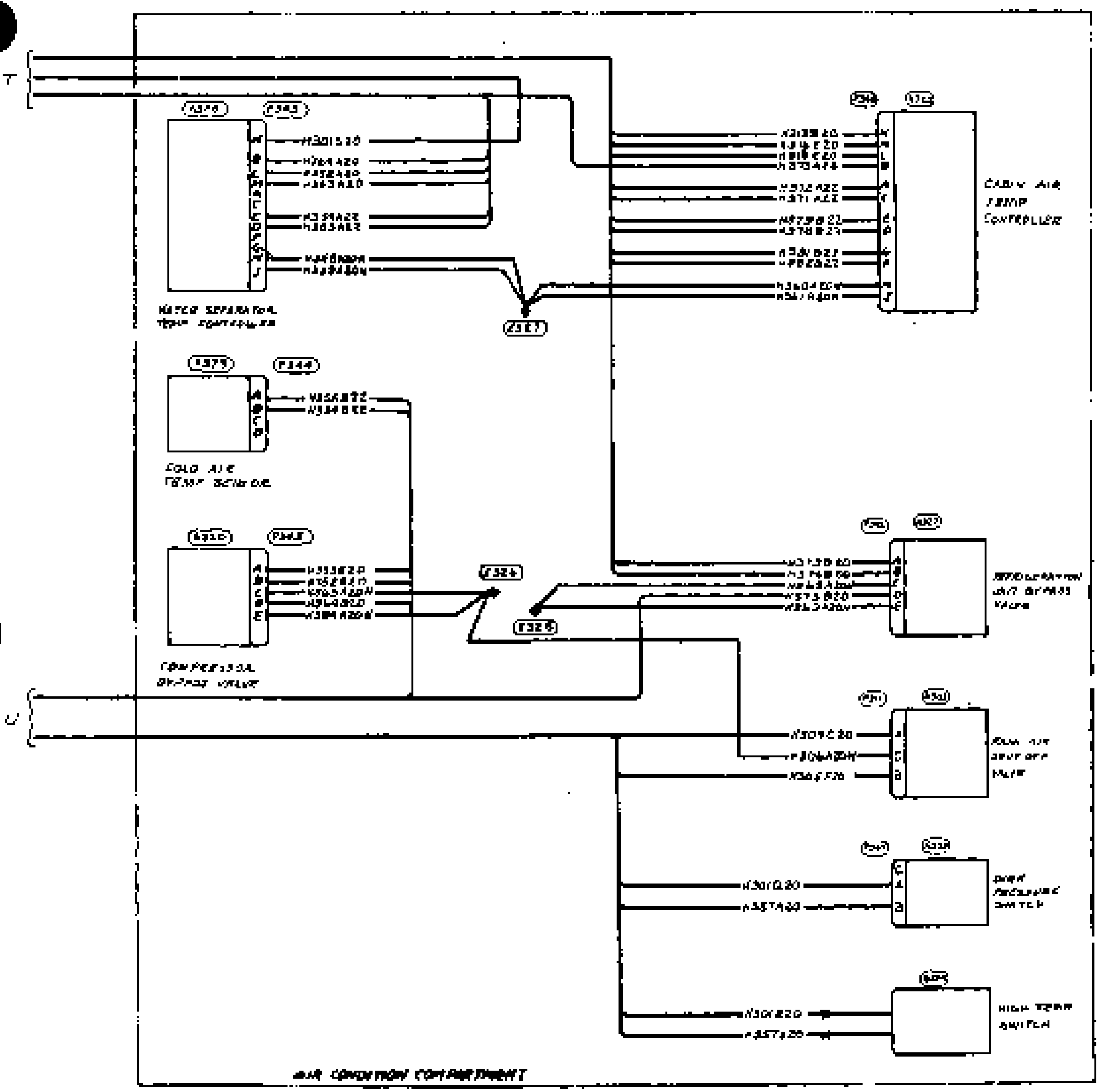


SEE BLOW-UP FICHE NO. AMUG352 ITEM A

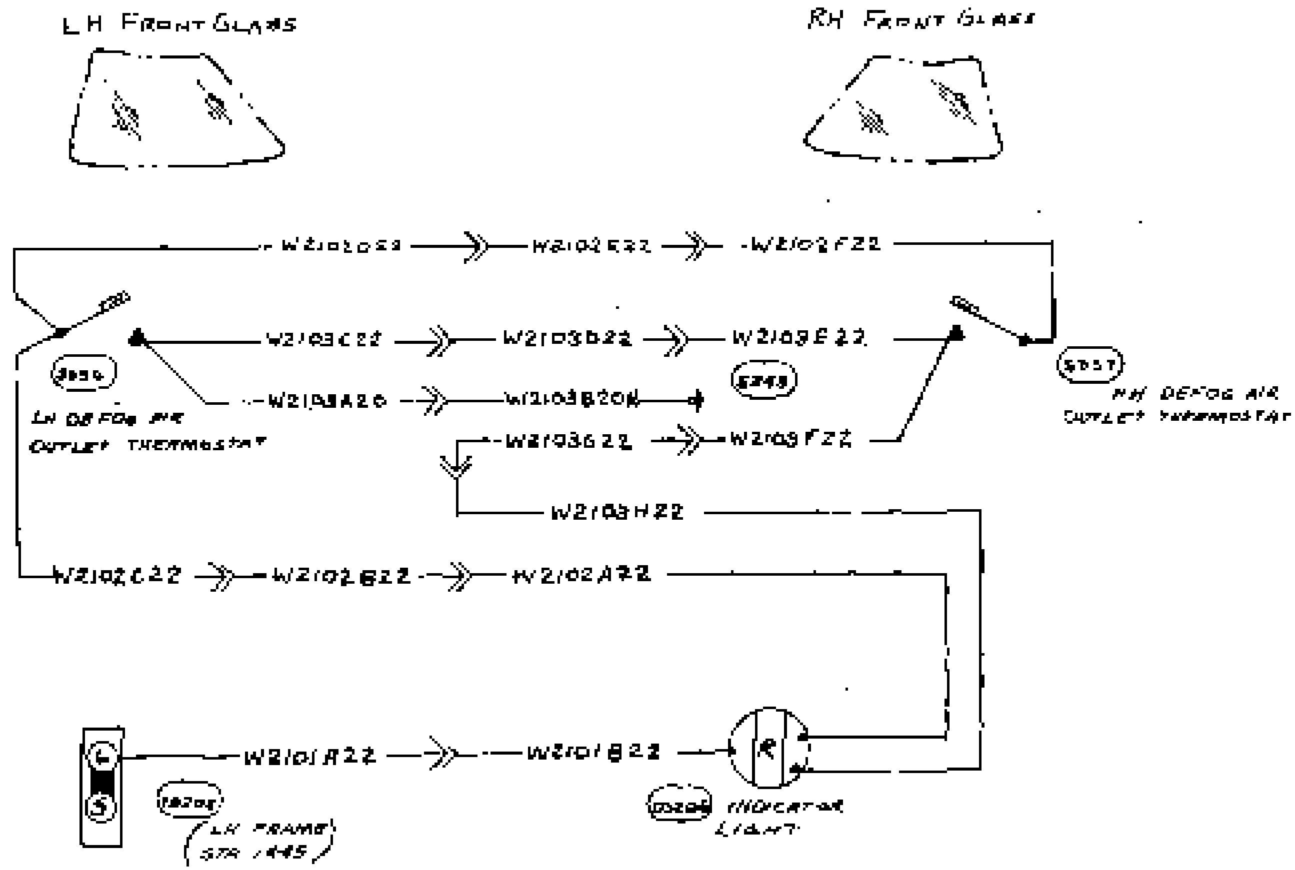


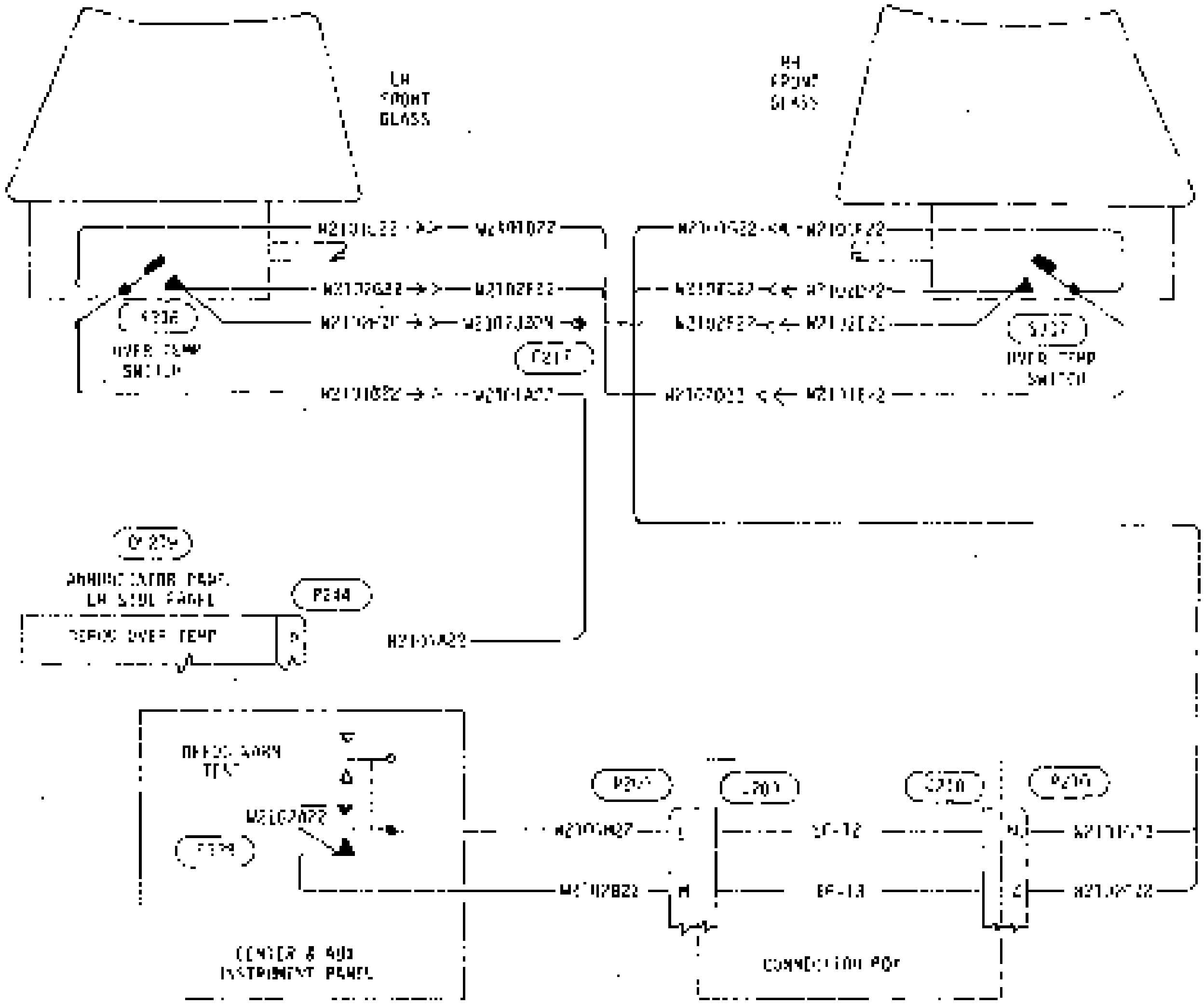
◇5 S/N 661SA, 697SA thru 711SA
◇6 S/N 712SA and Subsequent
◇3 S/N 661SA
◇4 S/N 697SA and Subsequent

AIR CONDITIONING CONTROL
Aircraft S/N 661SA, 697SA - 730SA



DEFOG OVERTEMP
Aircraft S/N 661SA





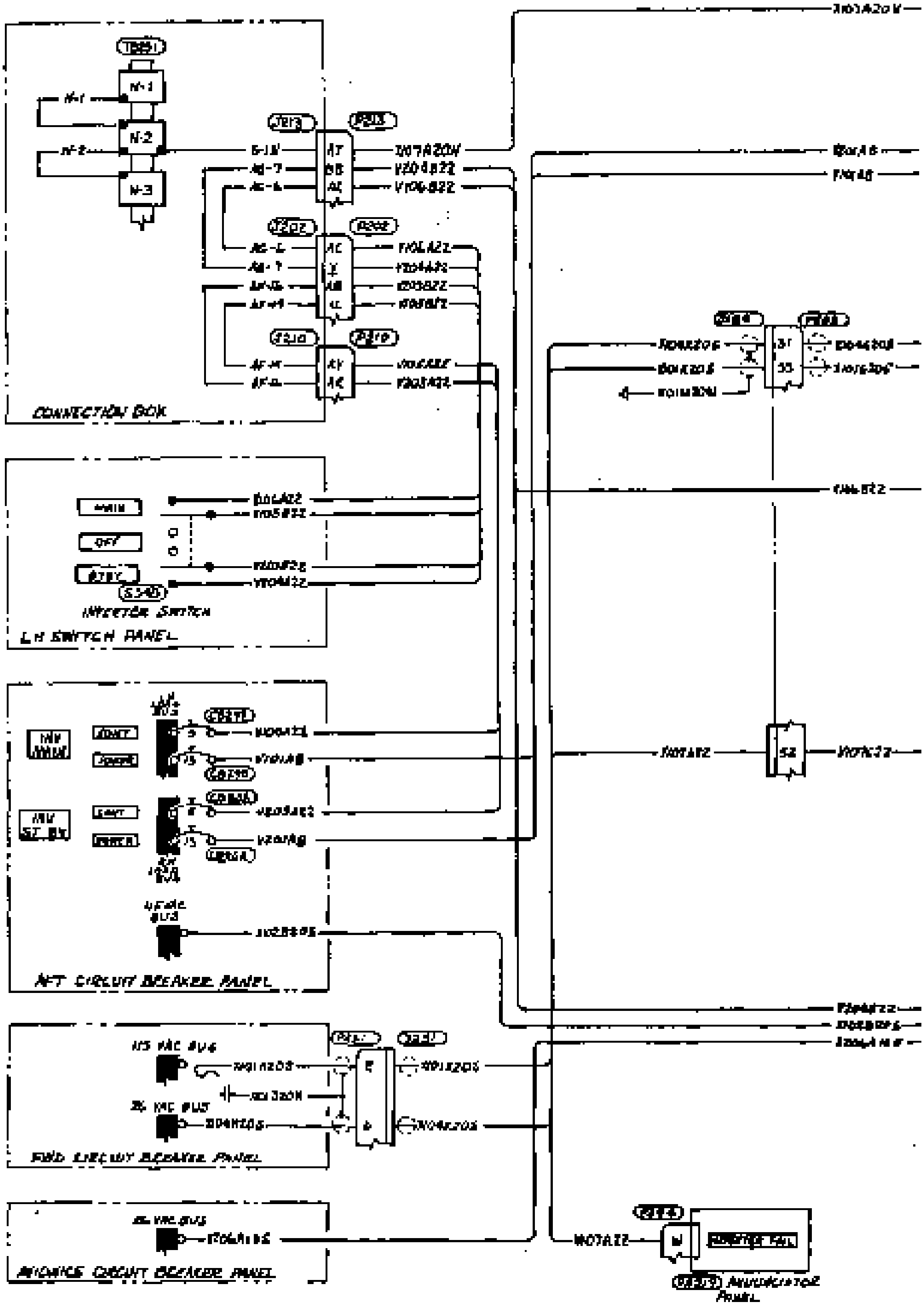
DEFOG OVERTEMP

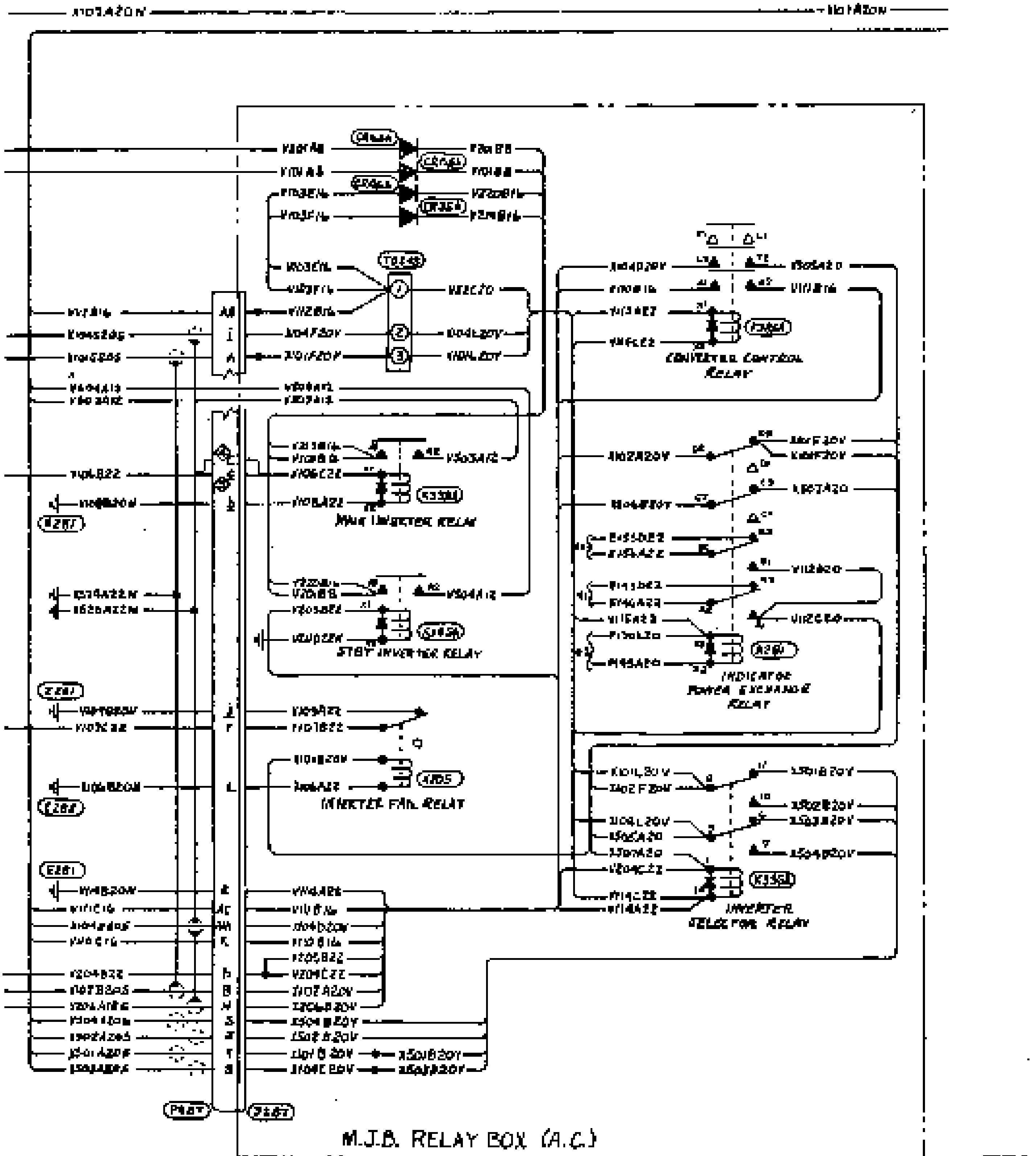
Aircraft S/N 697SA - 730SA

CHAPTER

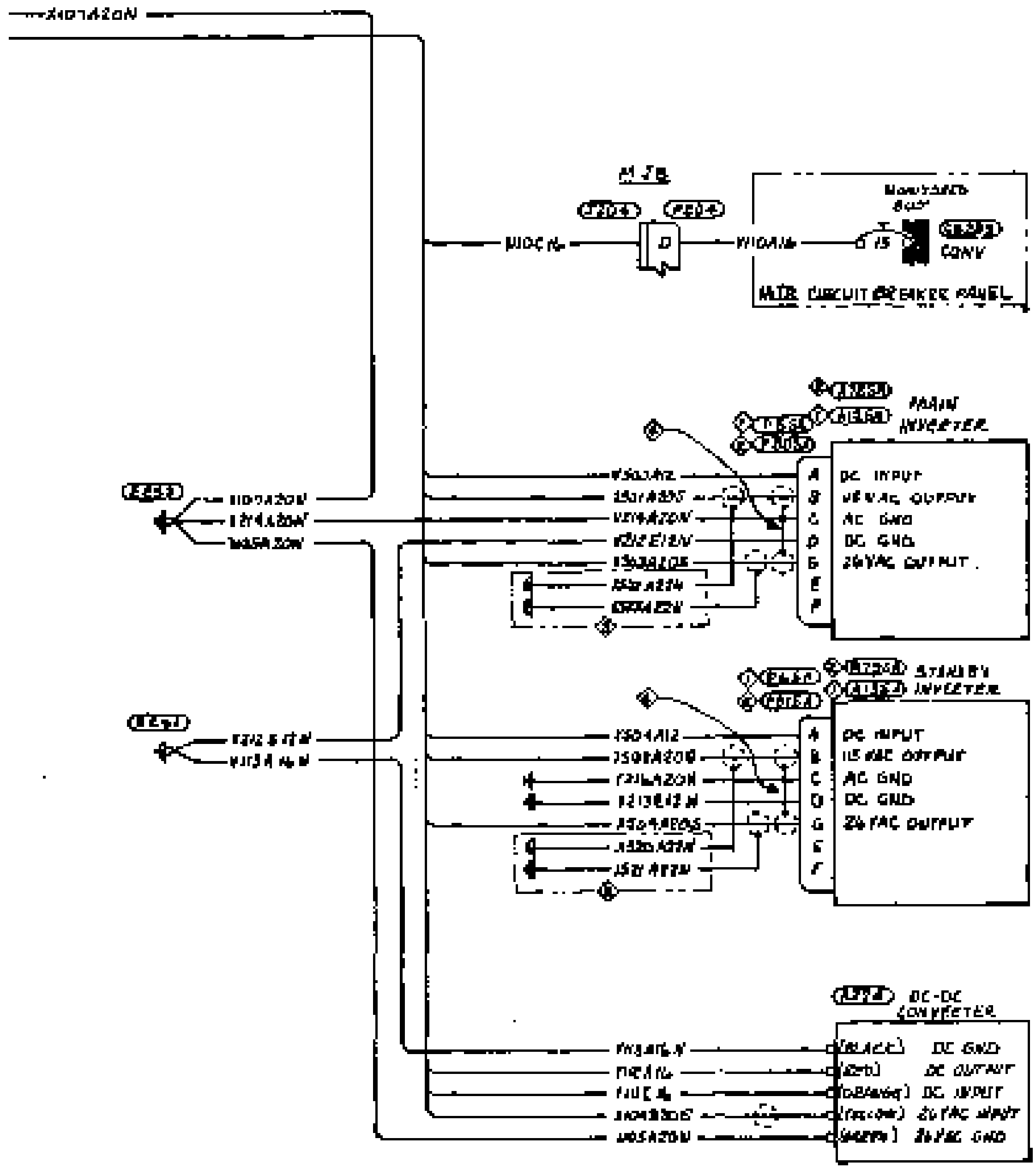
24

ELECTRICAL POWER

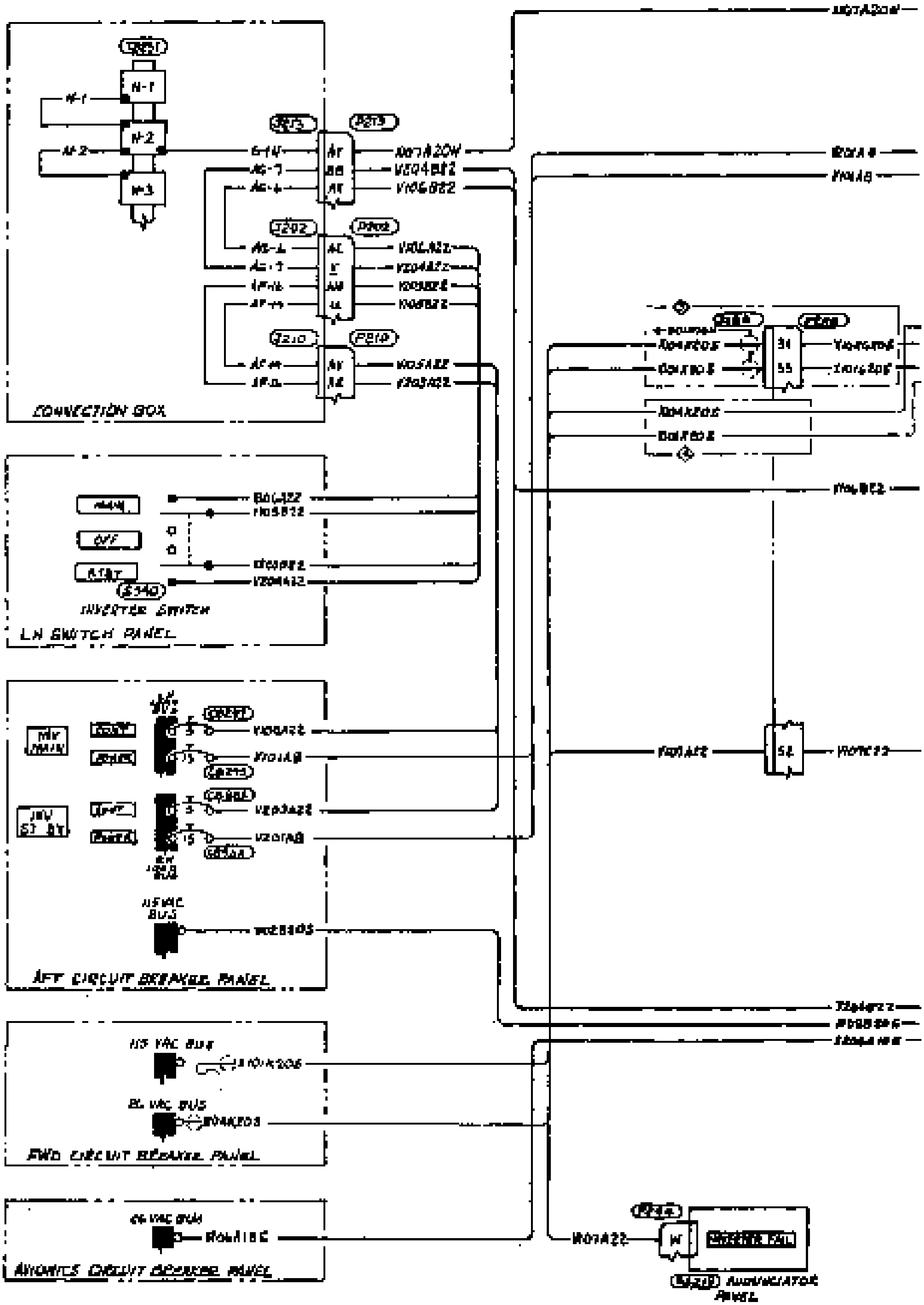

CONNECTION BOX
INVERTER SWITCH
LH SWITCH PANEL
AFT CIRCUIT BREAKER PANEL
FWD CIRCUIT BREAKER PANEL
AVIONICS CIRCUIT BREAKER PANEL
ANNUNCIATOR PANEL



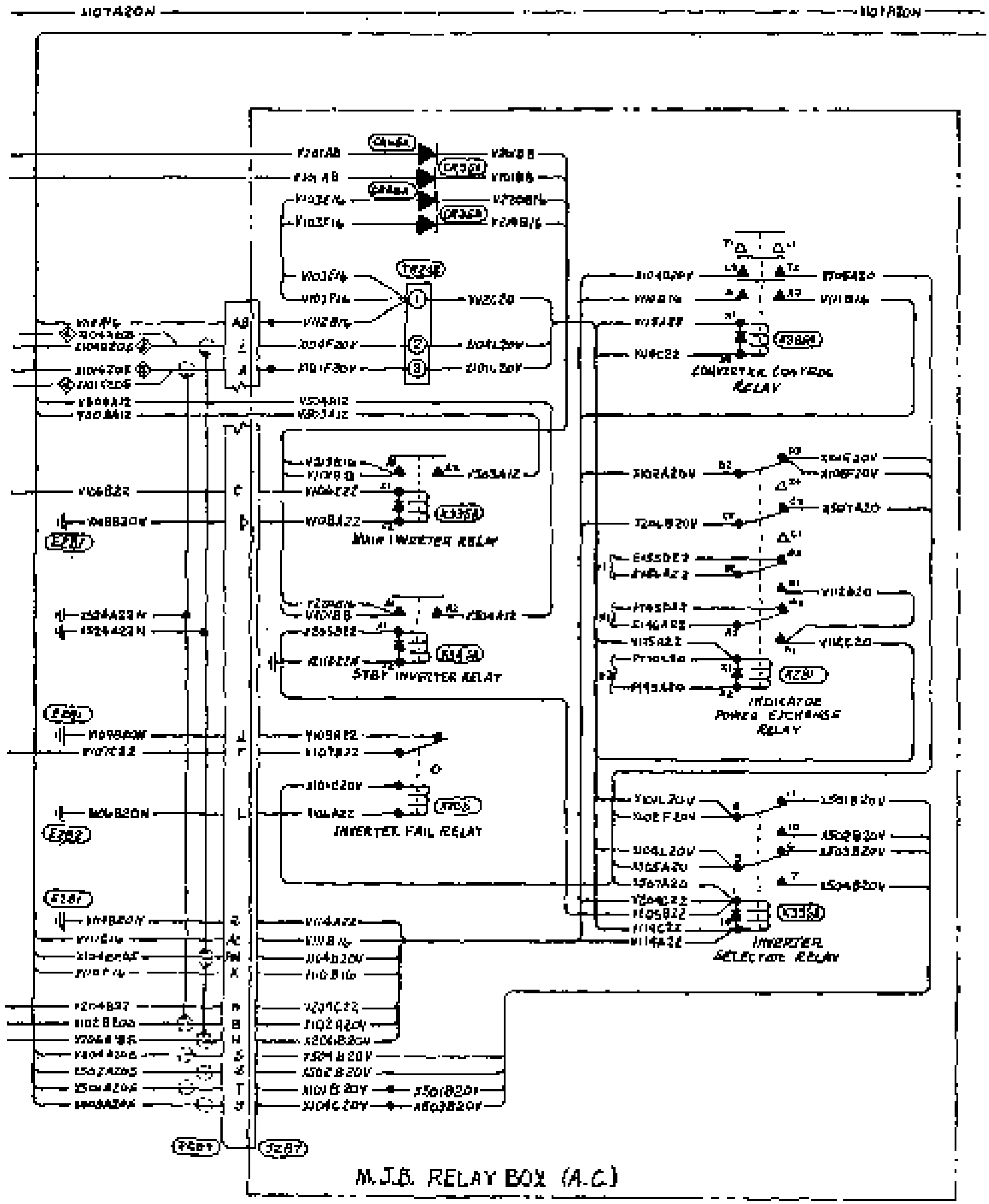
*1 Refer To 77-10-00
*2 Refer To 24-50-00
◇5 Aircraft S/N 661SA, 697SA, 698SA
◇4 Aircraft S/N 699SA Thru 710SA



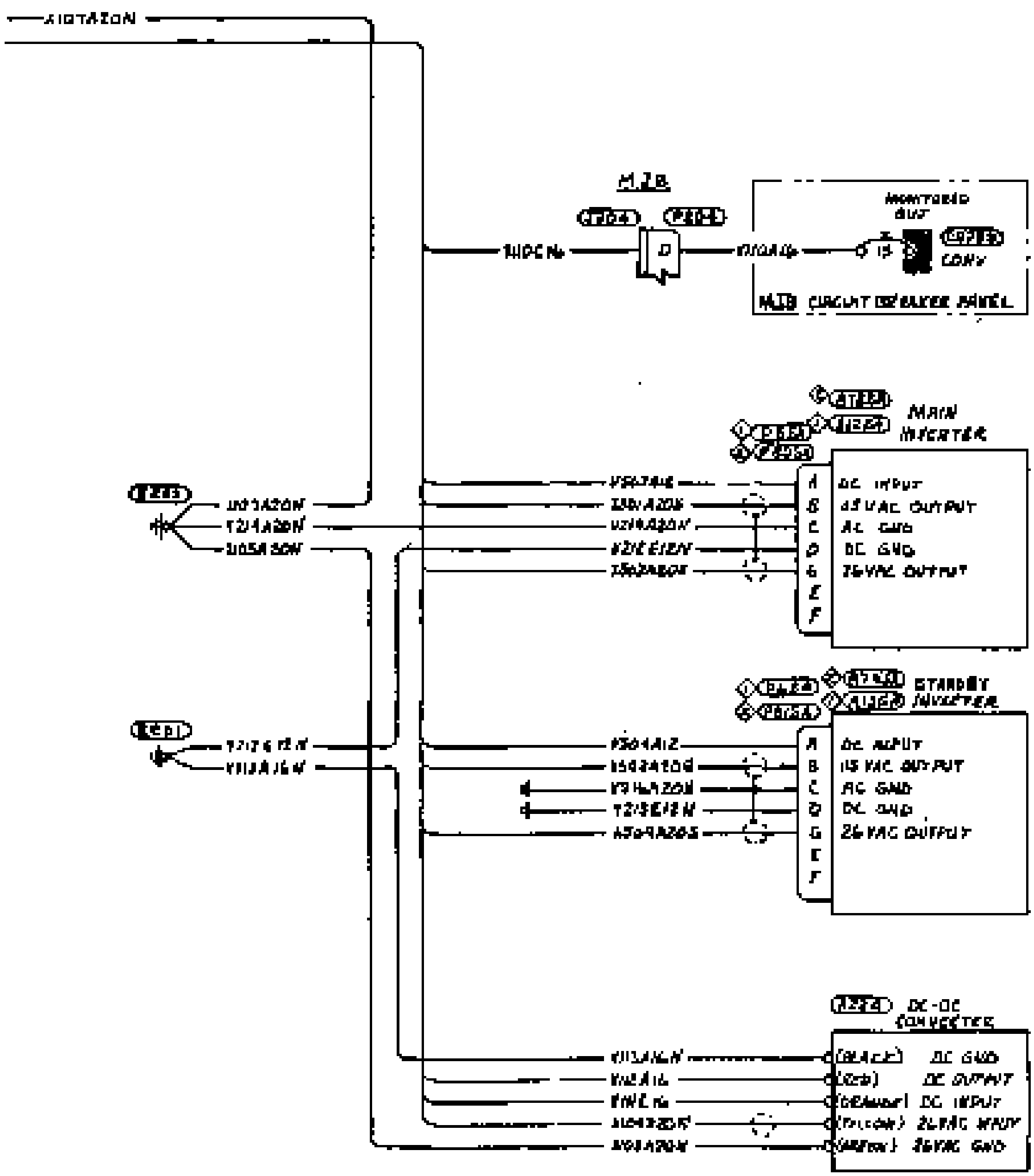
1. Applicable To Aircraft Equipped With PC-15B/PC-15C Static Inverters
2. Applicable To Aircraft Equipped With 1A250 (1B) Static Inverters
3. Aircraft S/N 661SA, 697SA, 698SA
4. Aircraft S/N 699SA Thru 710SA



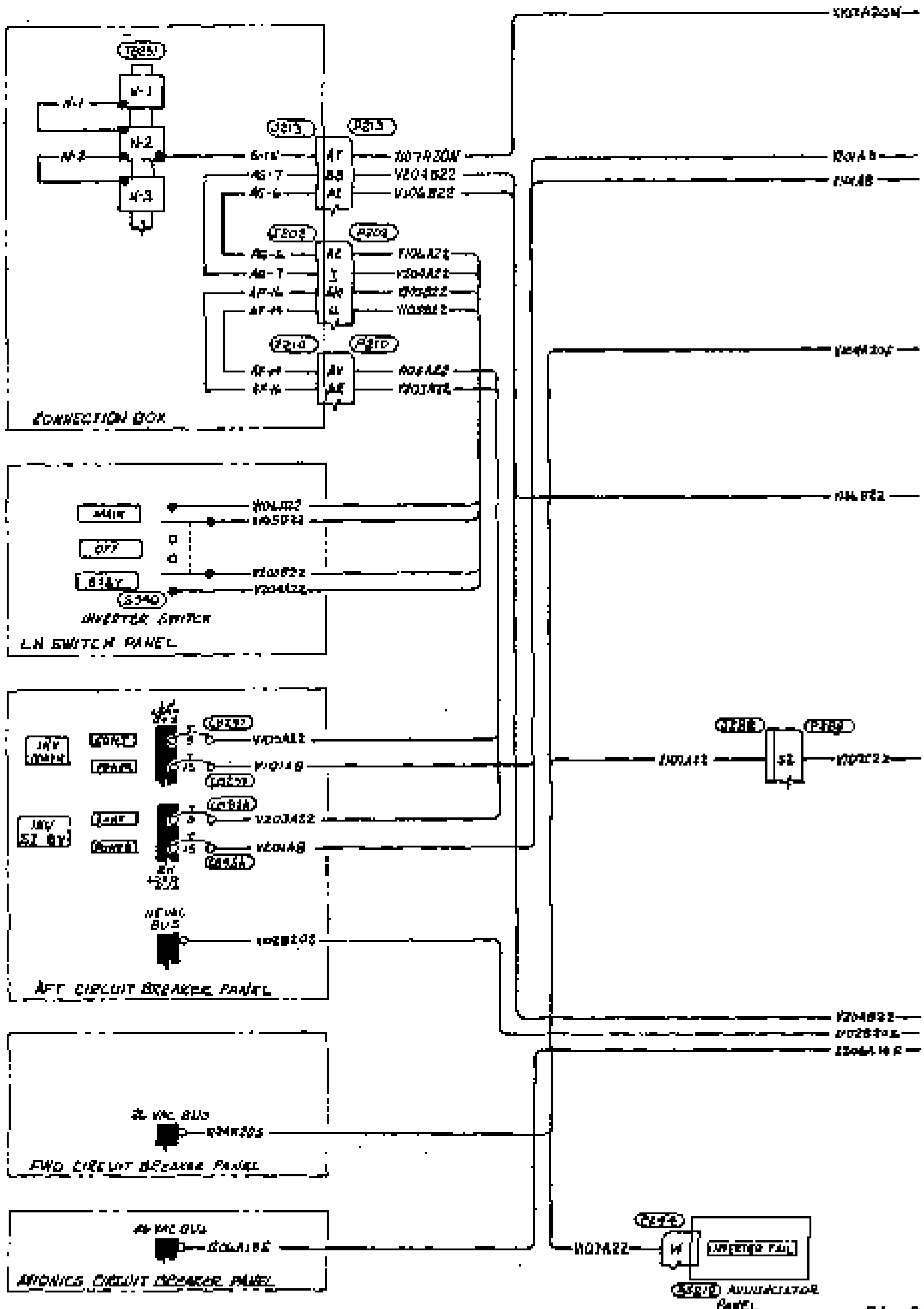
◈ AIRCRAFT S/N 711SA THRU 720SA
◈ AIRCRAFT S/N 721SA, 722SA

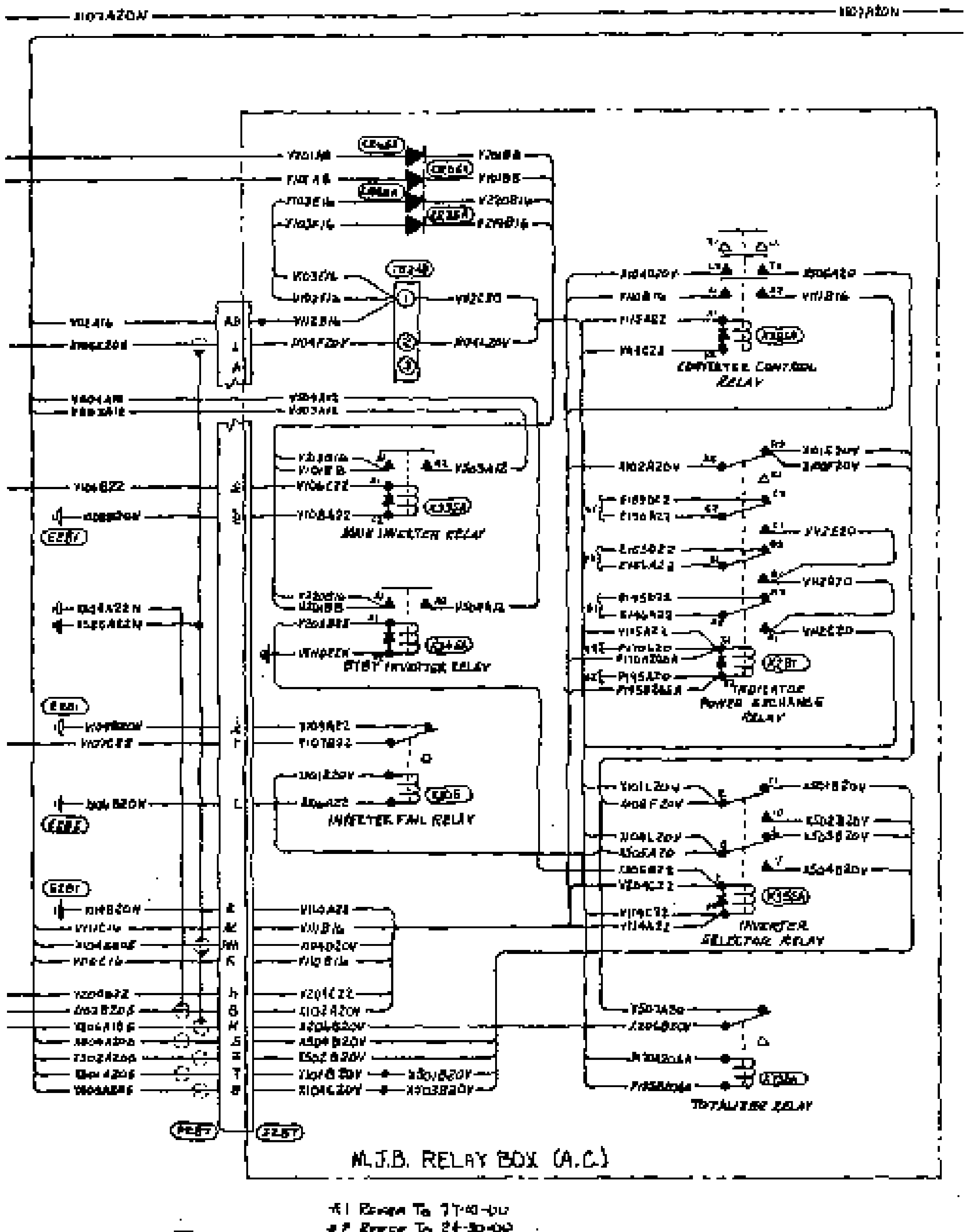


*1 Refer To 77-10-00
*2 Refer To 24-30-00
◈ Aircraft S/N 711SA Thru 720SA
◈ Aircraft S/N 721SA, 722SA



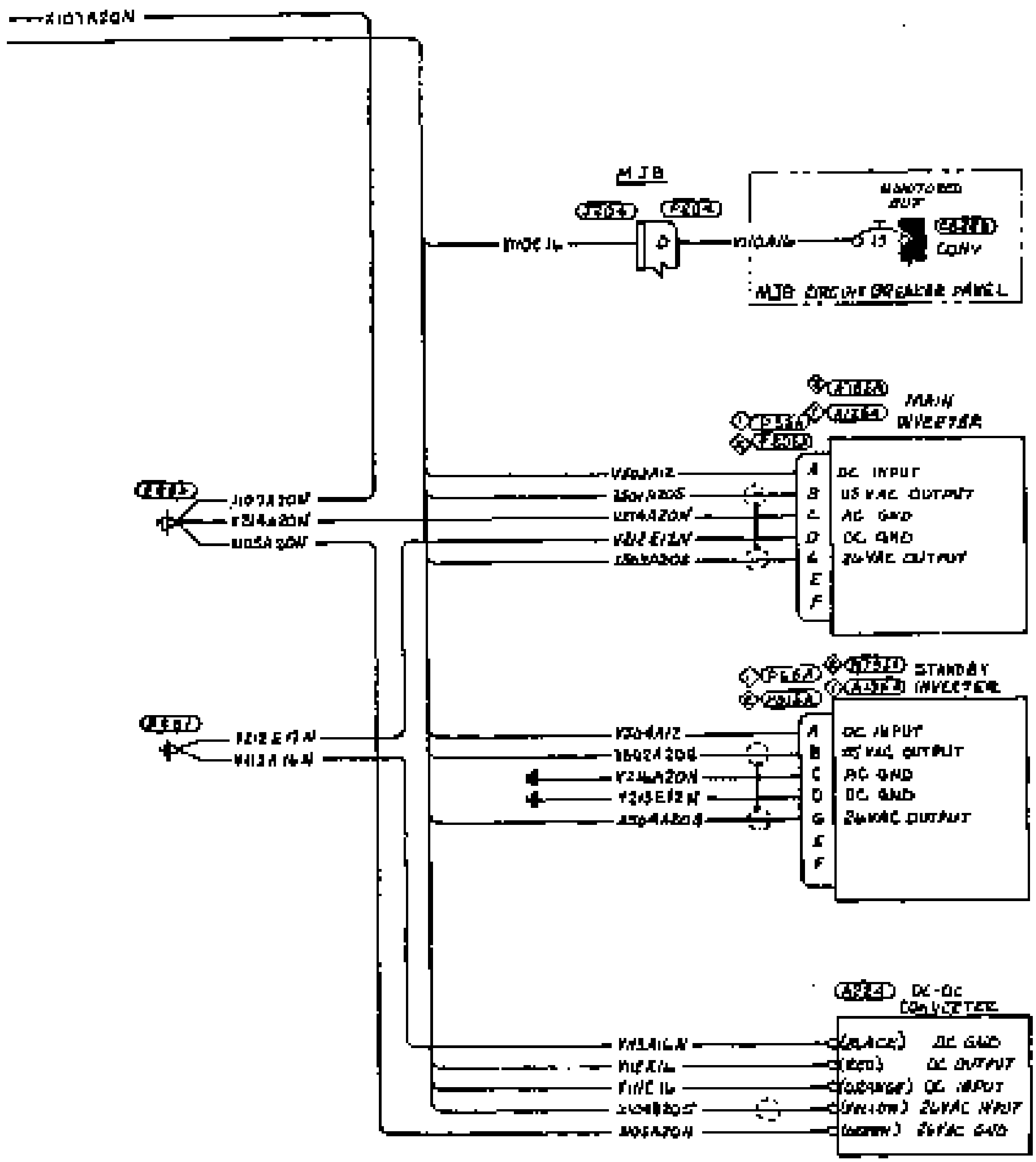
① Applicable To Aircraft Equipped With PC-15B/PC-15C Static Inverters
② Applicable To Aircraft Equipped With 1A250(1B) Static Inverters


CONNECTION BOX
INVERTER SWITCH
LH SWITCH PANEL
MAIN
OFF
STBY
AFT CIRCUIT BREAKER PANEL
FWD CIRCUIT BREAKER PANEL
AVIONICS CIRCUIT BREAKER PANEL
INVERTER FAIL
ANNUNCIATOR PANEL

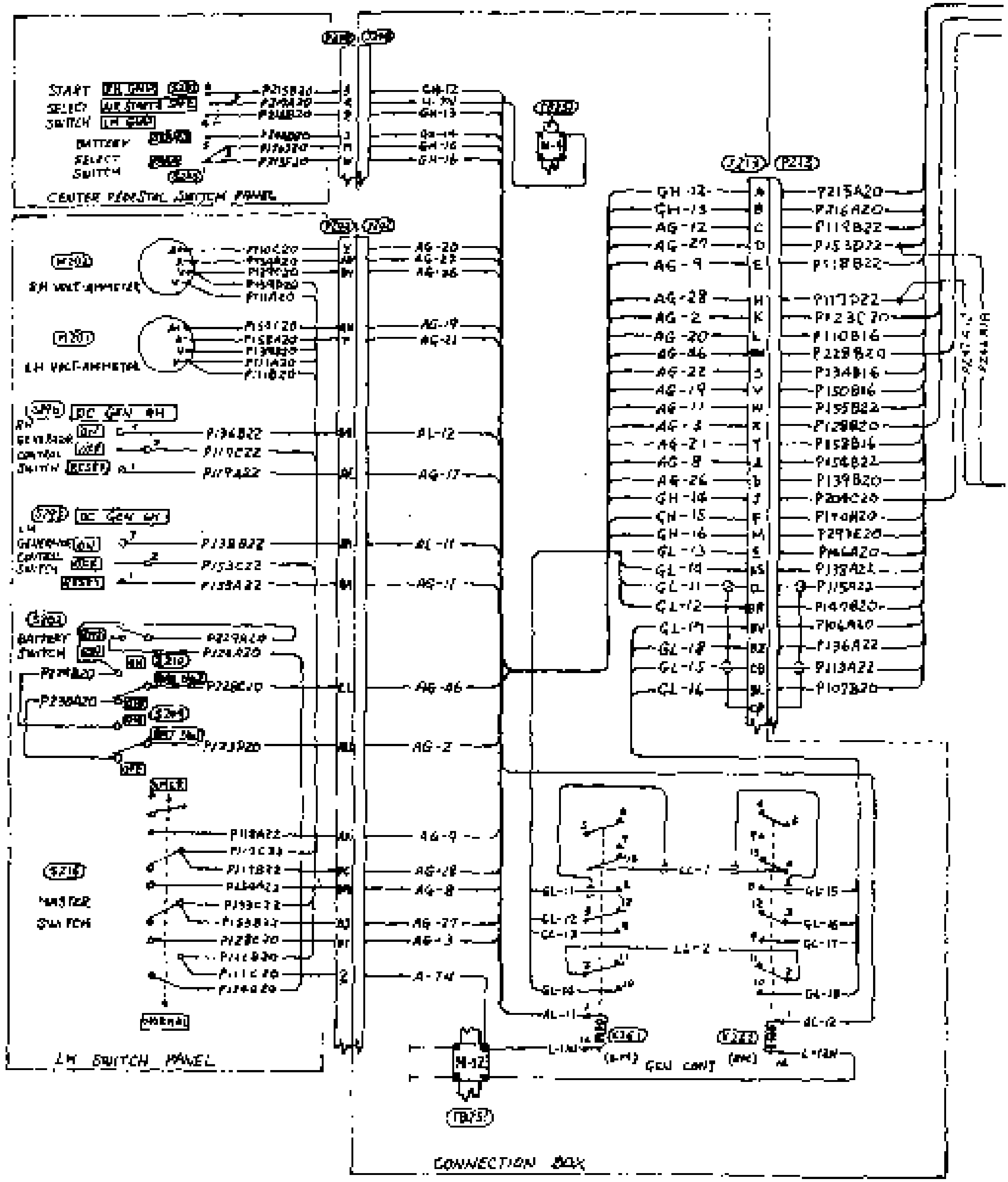


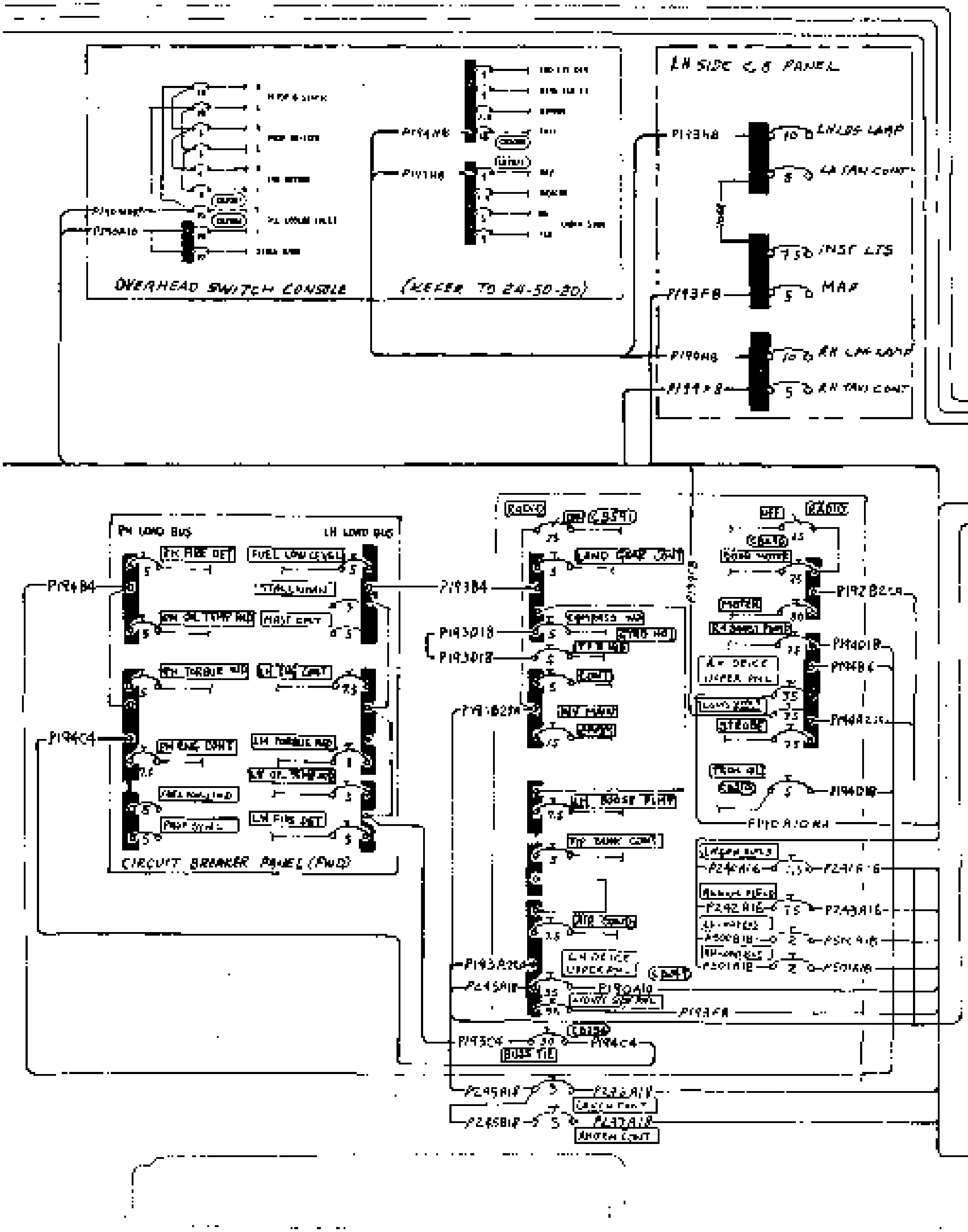
AC POWER SOURCE

Aircraft S/N 723SA - 730SA



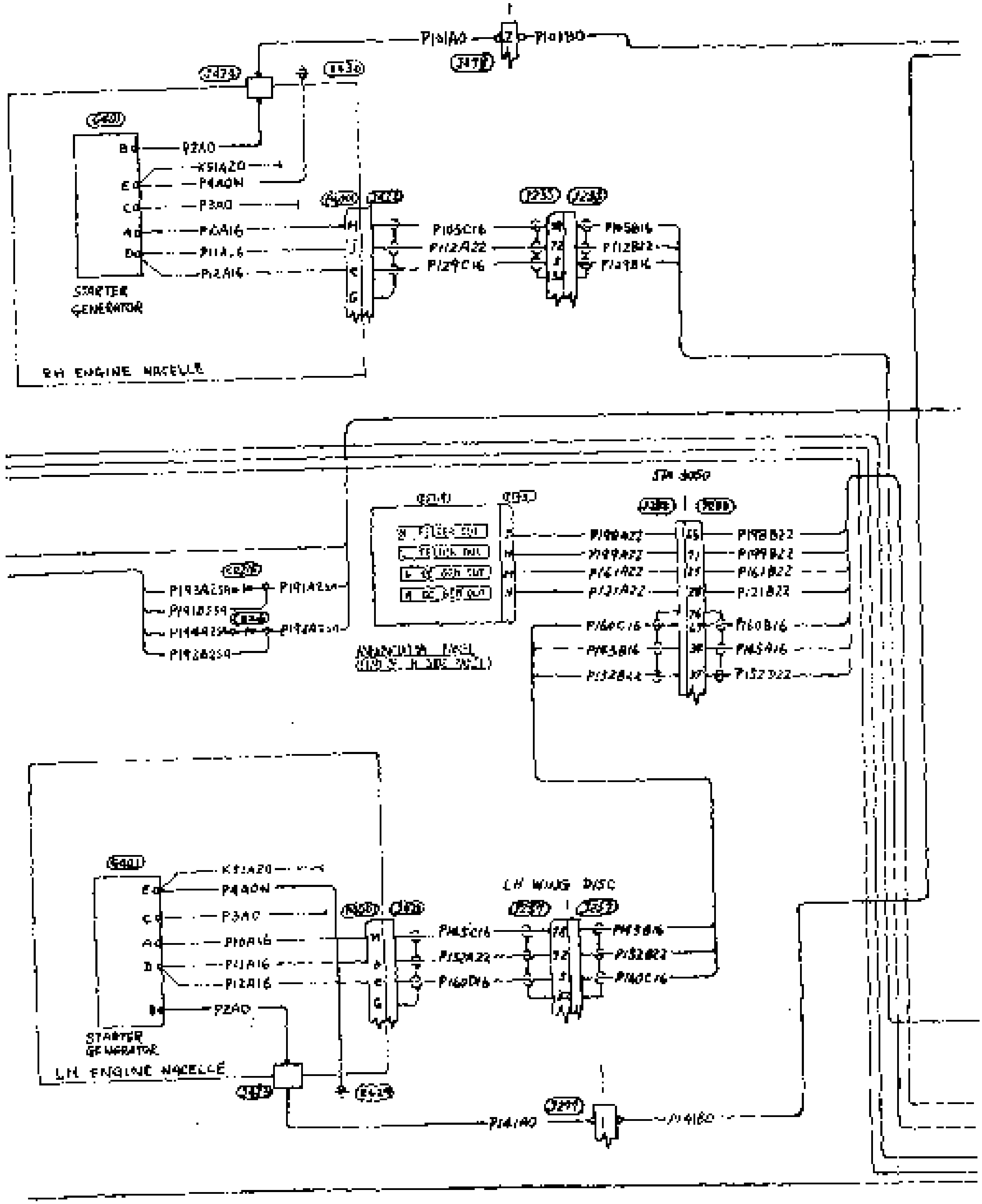
◇1 Applicable To Aircraft Equipped With PC-15B/PC-15C Static Inverters
◇2 Applicable To Aircraft Equipped With 1A250(1B) Static Inverters

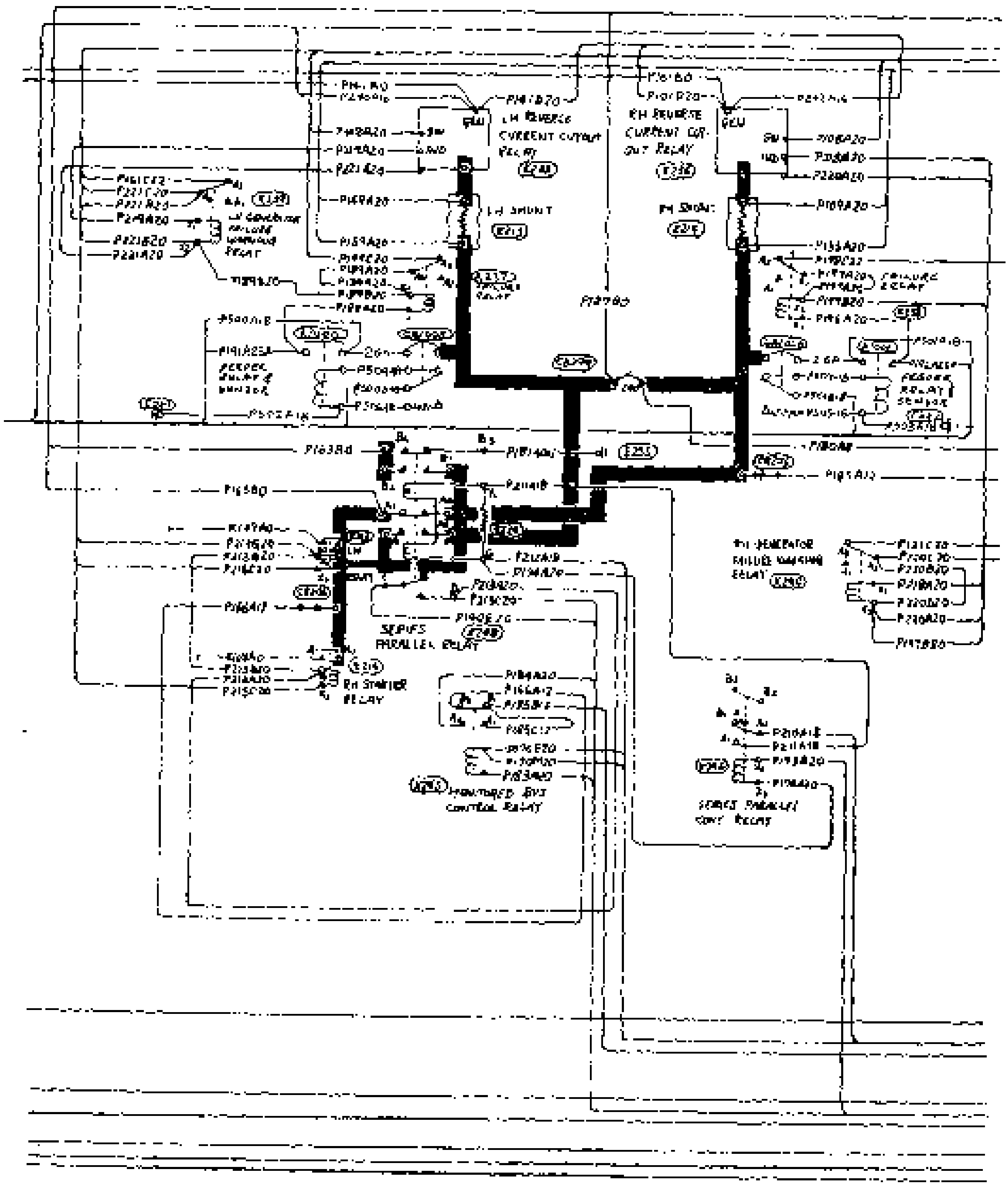

START SELECT SWITCH
BATTERY SELECT SWITCH
CENTER PEDESTAL SWITCH PANEL
RH VOLT-AMMETER
LH VOLT-AMMETER
DC GEN RH
DC GEN LH
BATTERY SWITCH
MASTER SWITCH
LH SWITCH PANEL
GEN CONT
CONNECTION BOX



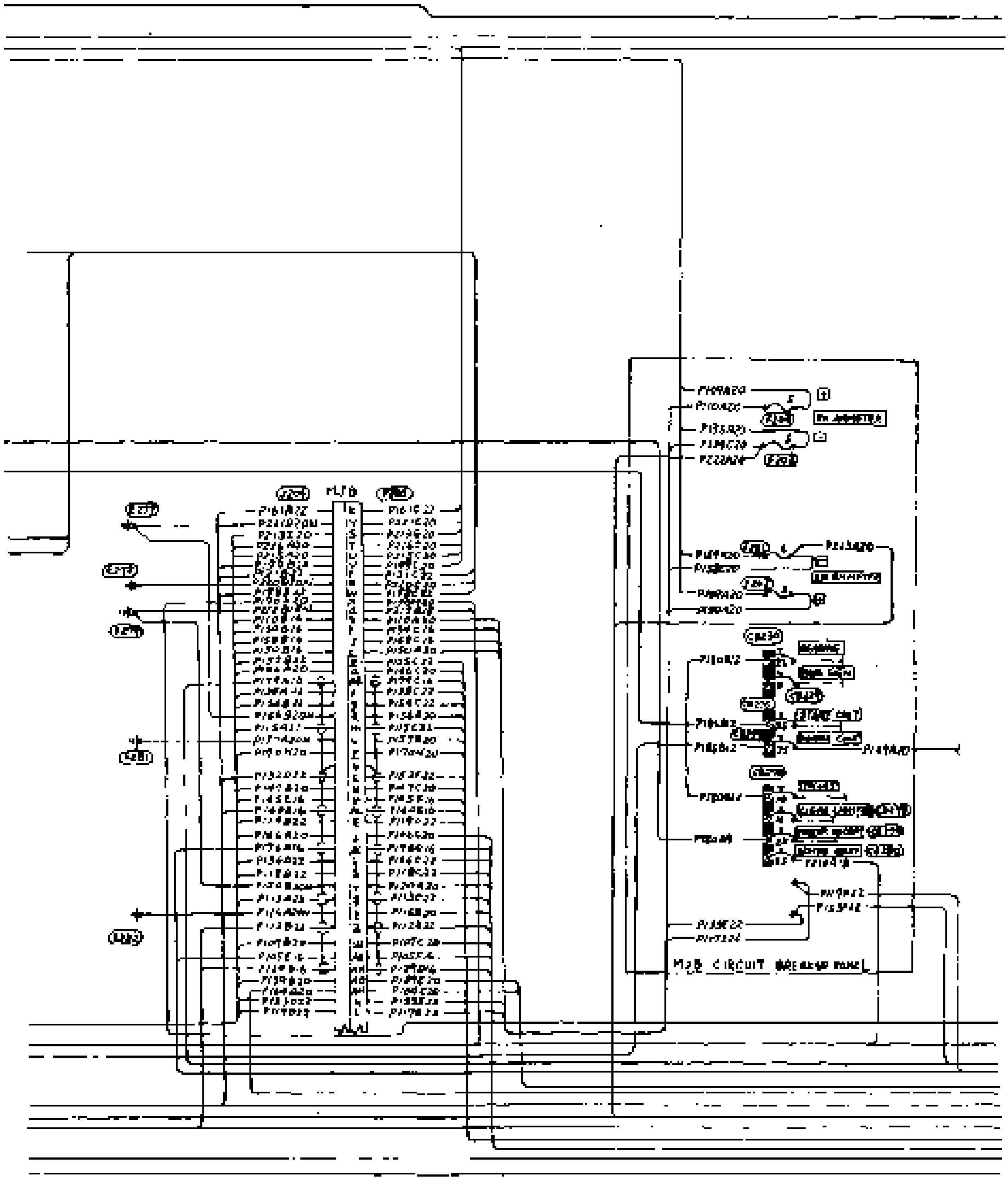
DC POWER SOURCE

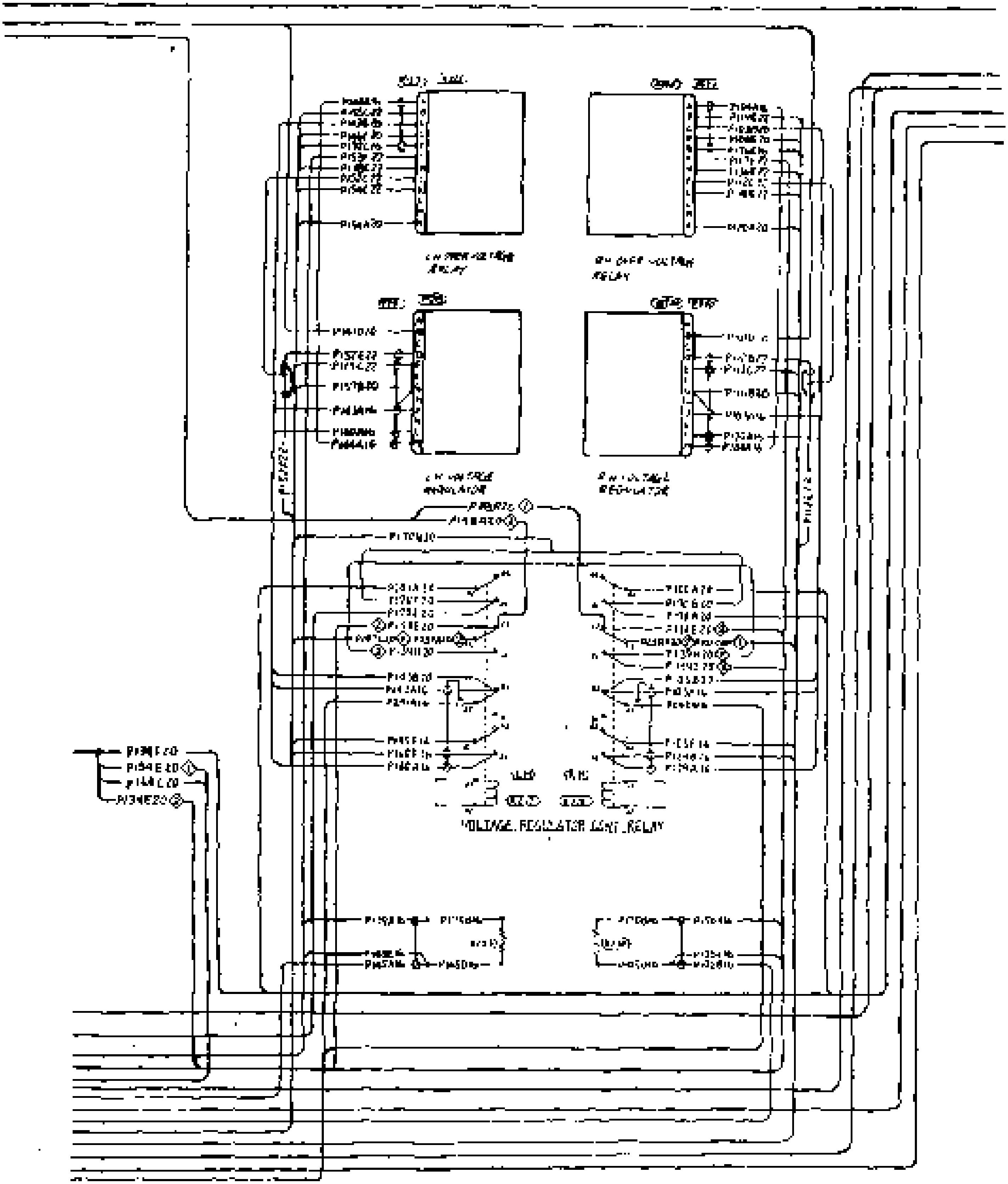
Aircraft S/N 6615A


STARTER GENERATOR
RH ENGINE NACELLE
STA 3050
LH WING DISC
STARTER GENERATOR
LH ENGINE NACELLE

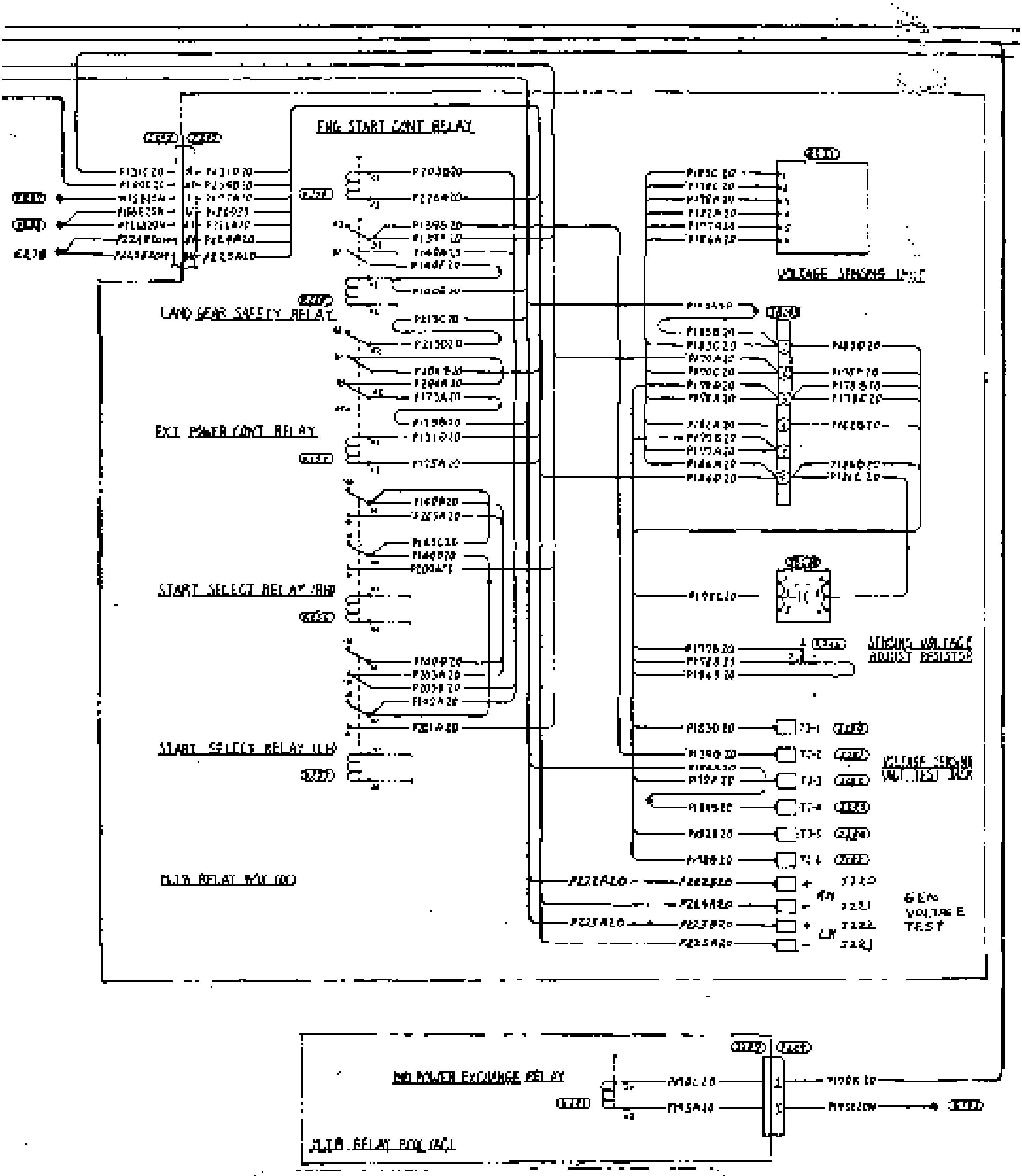


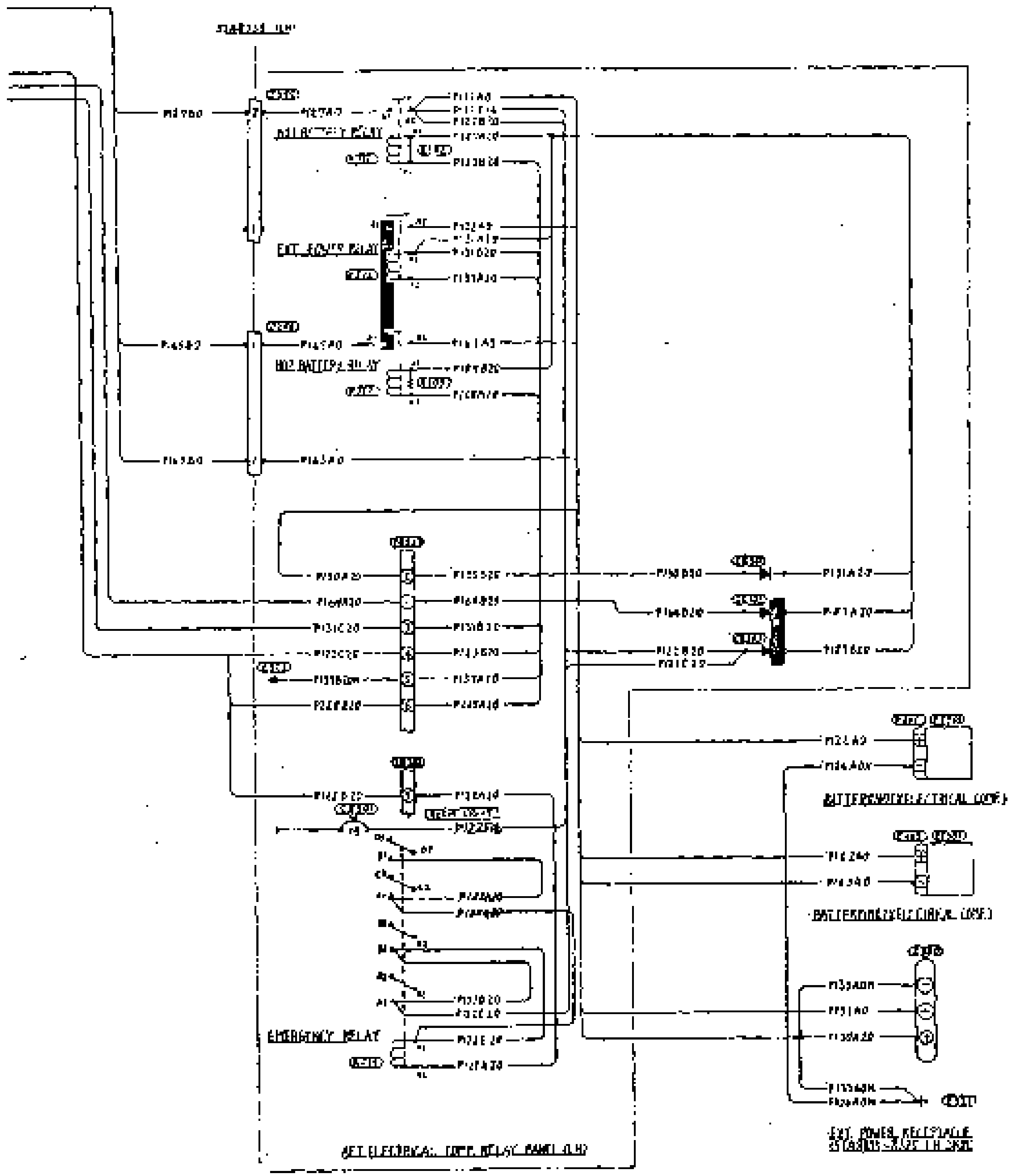
DC POWER SOURCE
Aircraft S/N 661SA



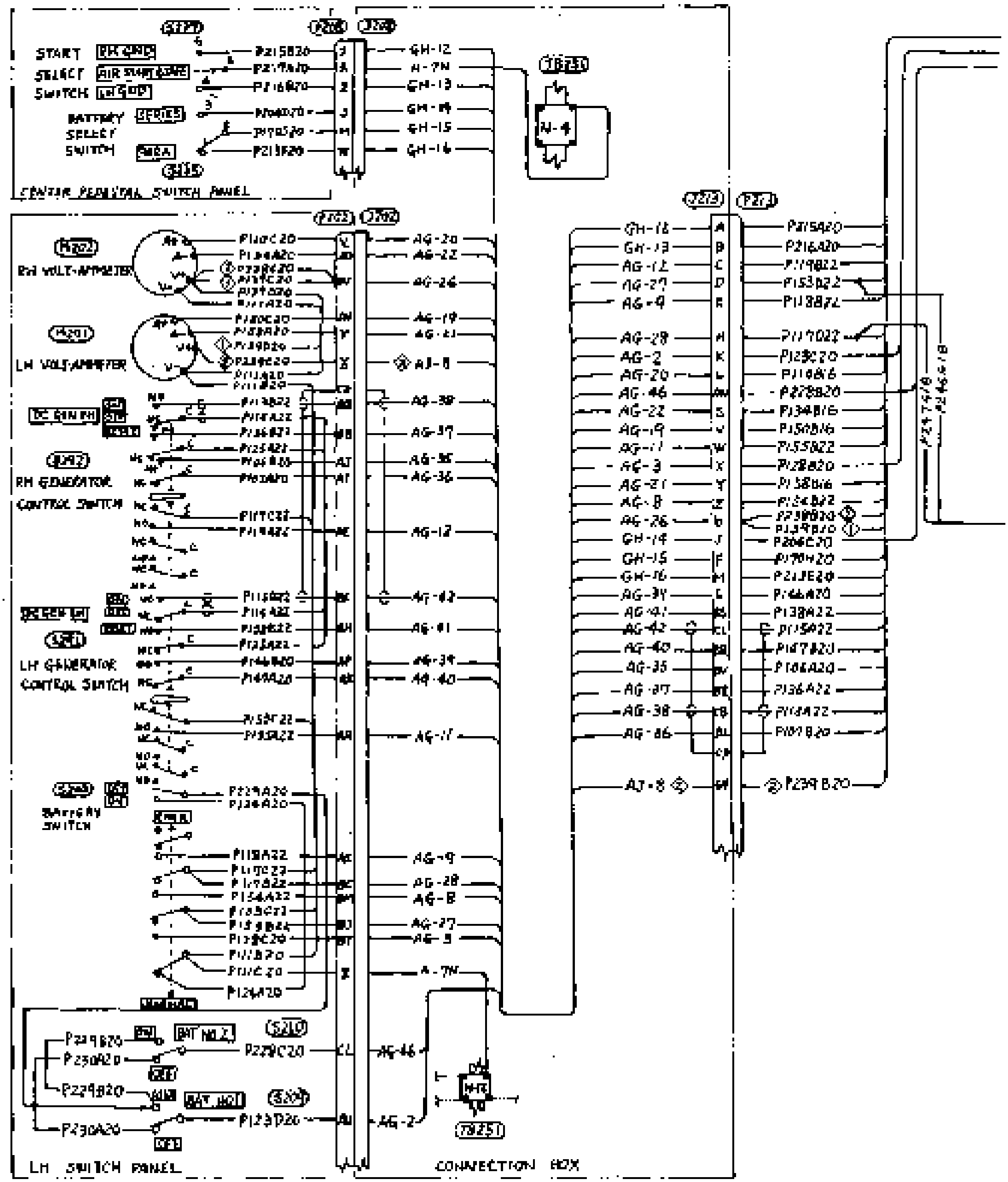


◇1 661SA, 697SA - 700SA
◇2 705SA - 713SA


ENG START CONT RELAY
LAND GEAR SAFETY RELAY
EXT POWER CONT RELAY
START SELECT RELAY (RH)
START SELECT RELAY (LH)
VOLTAGE SENSING UNIT
SENSING VOLTAGE ADJUST RESISTOR
VOLTAGE SENSING UNIT TEST JACK
GEN VOLTAGE TEST
RH
LH
MAIN RELAY BOX (DC)
RH POWER EXCHANGE RELAY
MAIN RELAY BOX (AC)

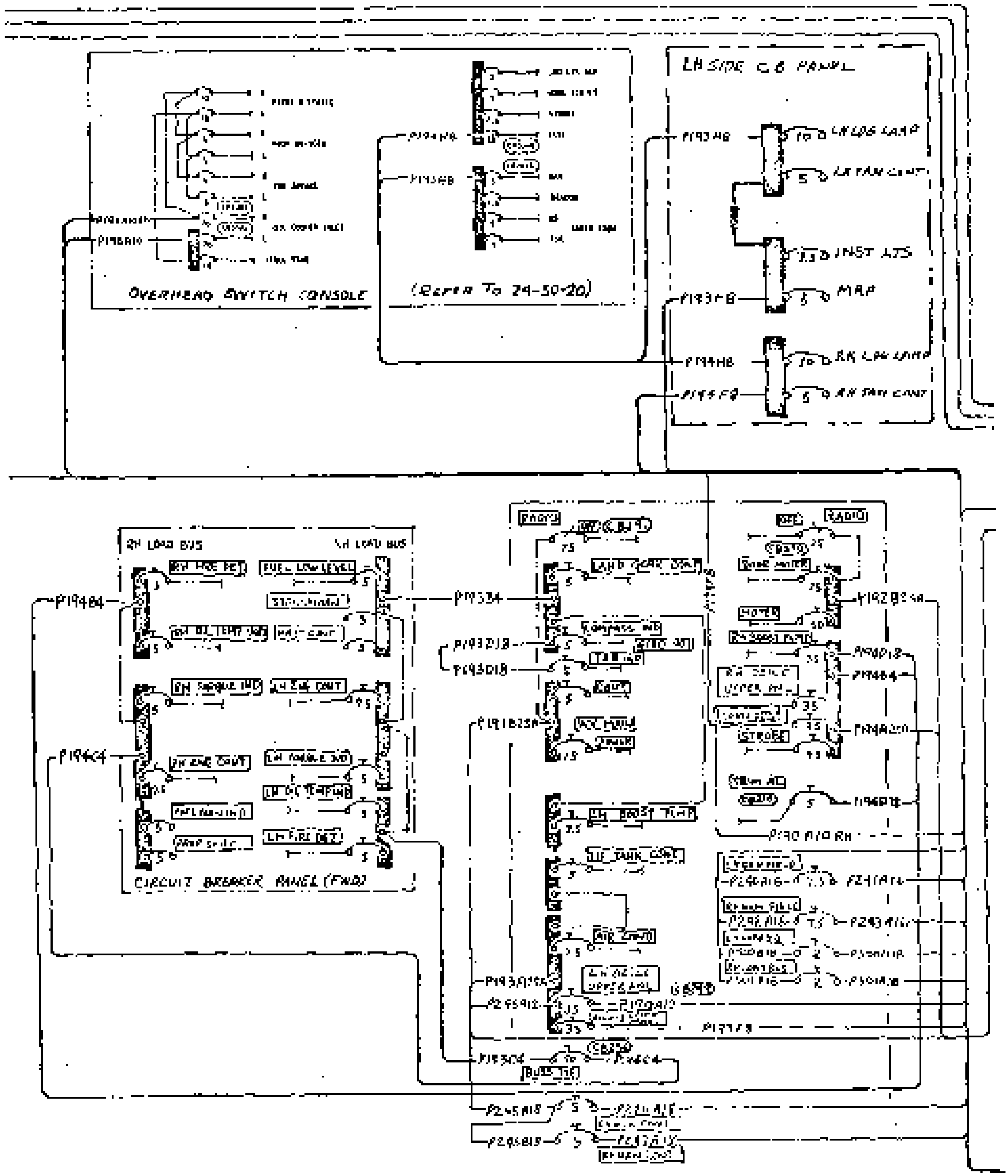


DC POWER SOURCE
Aircraft S/N 661SA



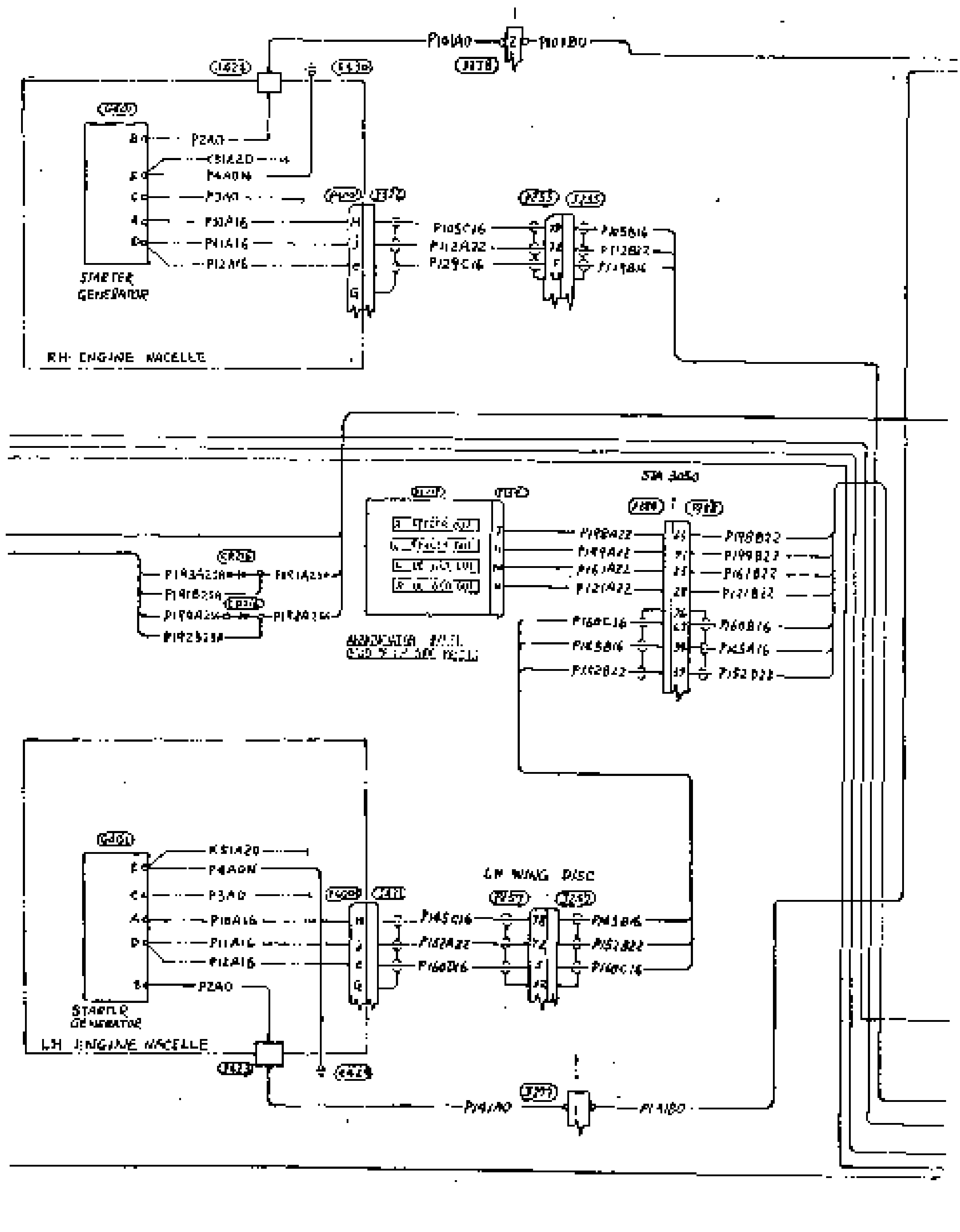
① 661SA, 697SA - 700SA
② 701SA - 713SA

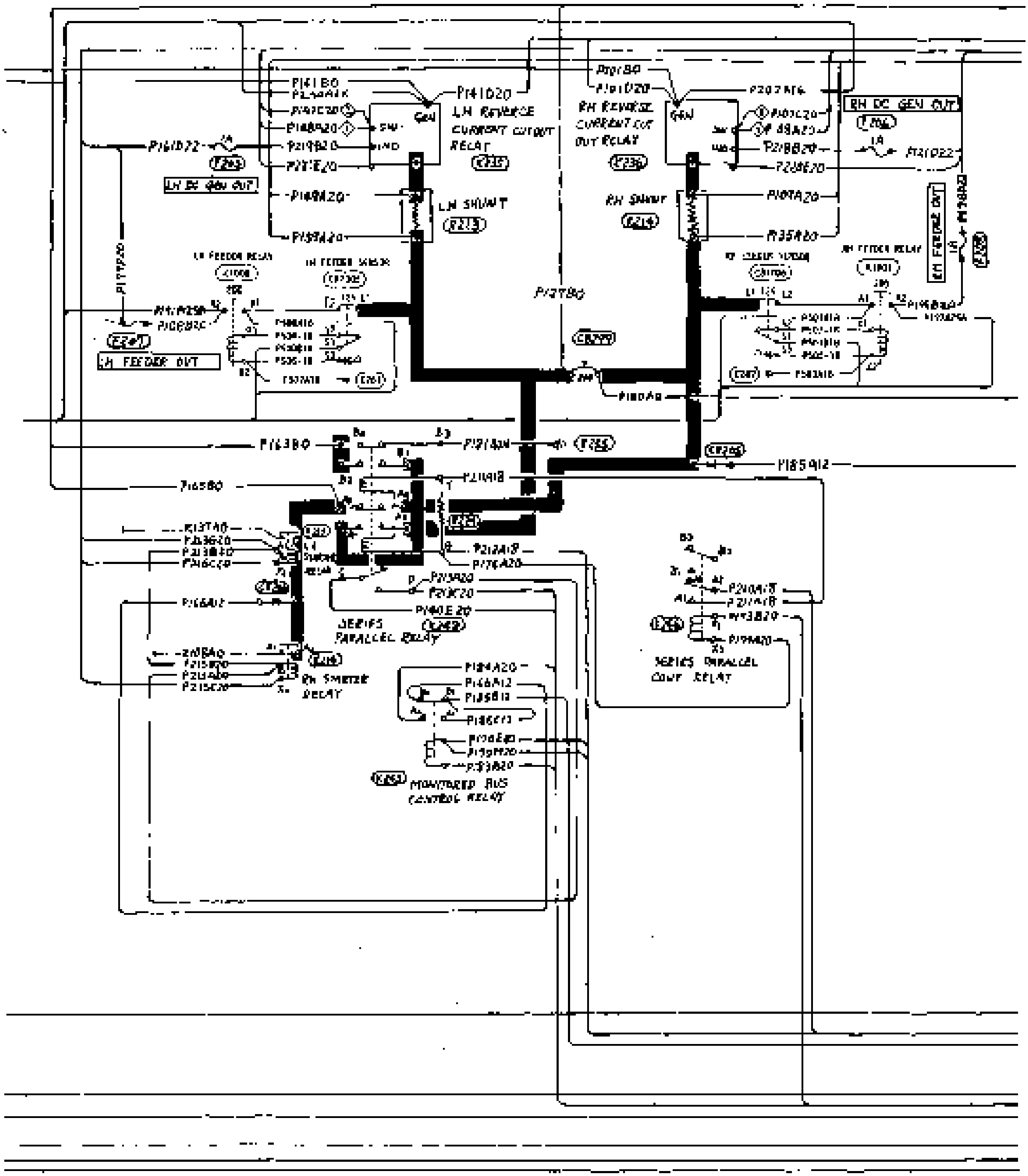
DC POWER SOURCE
Aircraft S/N 697SA - 713SA



DC POWER SOURCE

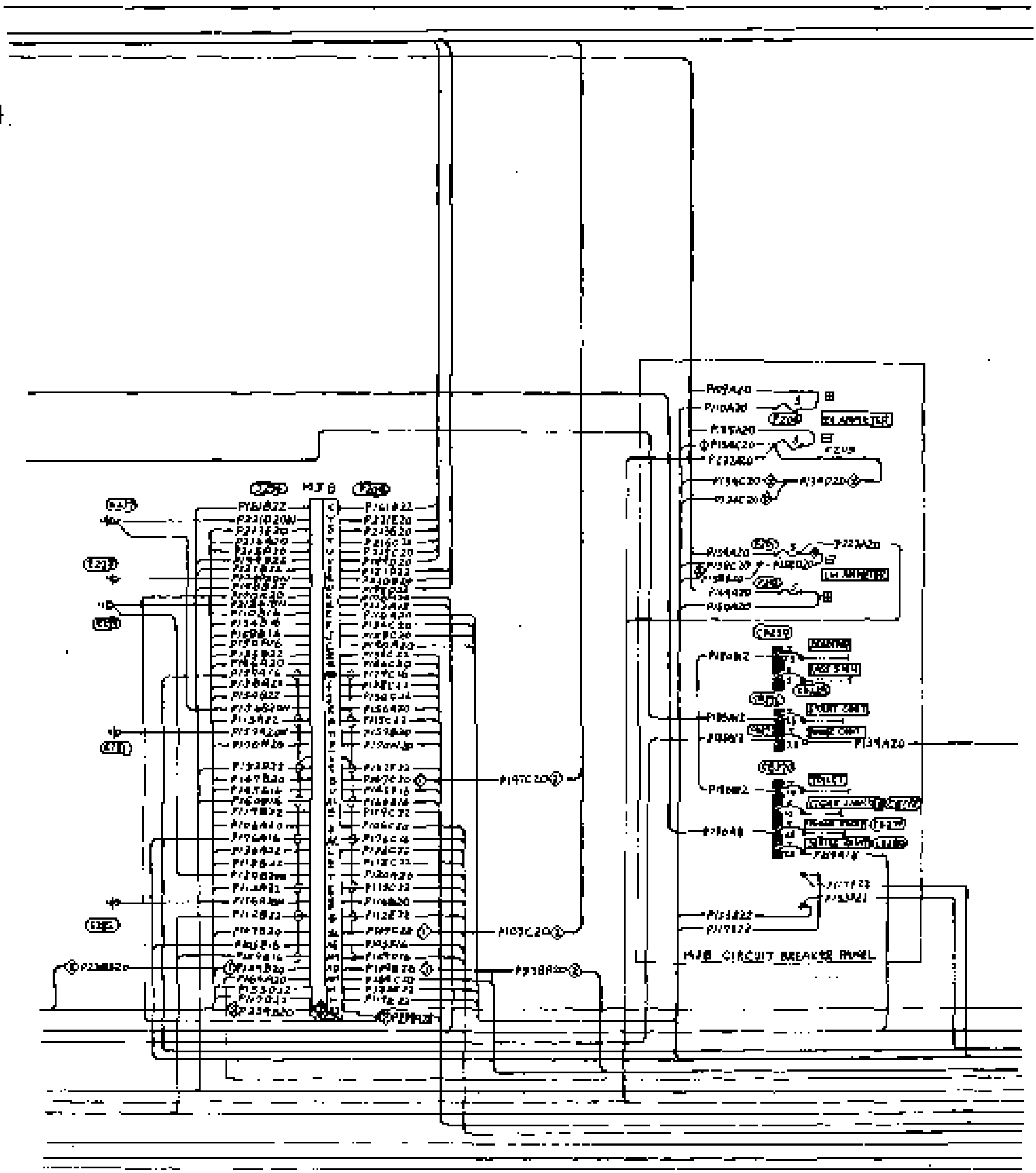
Aircraft S/N 697SA - 713SA





1 661SA, 697SA - 703SA
2 701SA - 713SA

DC POWER SOURCE
Aircraft S/N 697SA - 713SA

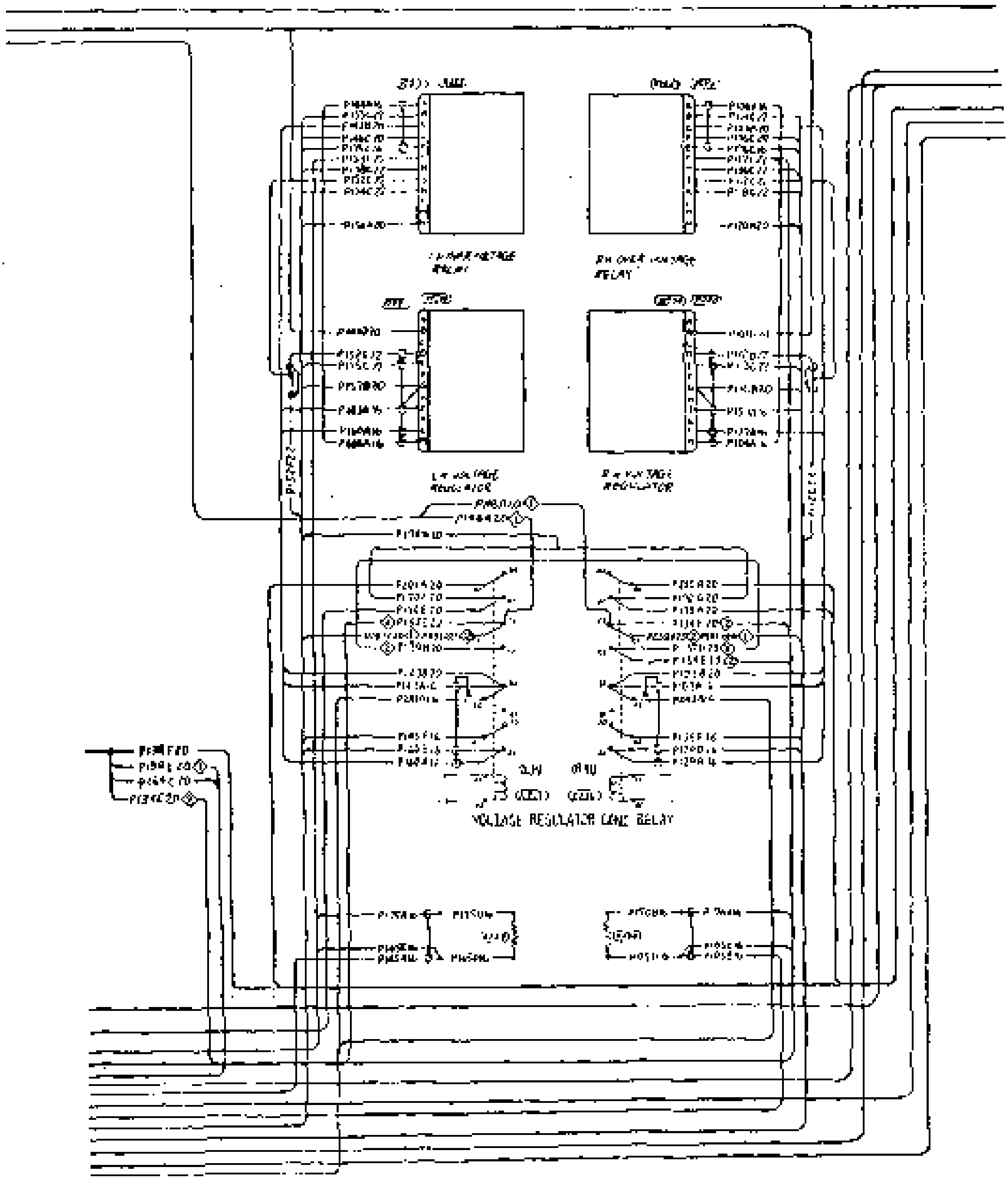


◇1 6815A, 6971A - 7005A

◇2 7015A - 7135A

DC POWER SOURCE

Aircraft S/N 6975A - 7135A

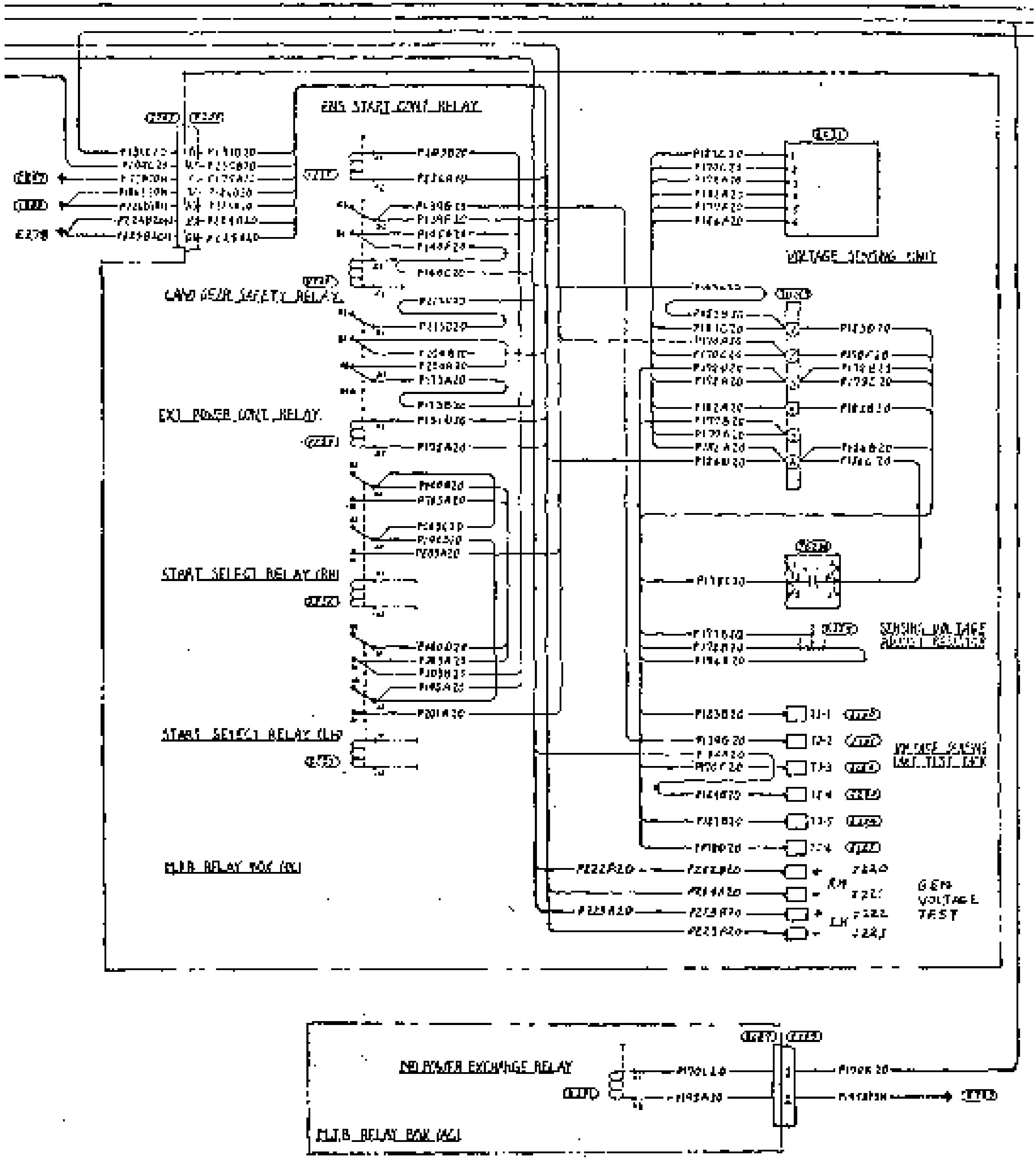


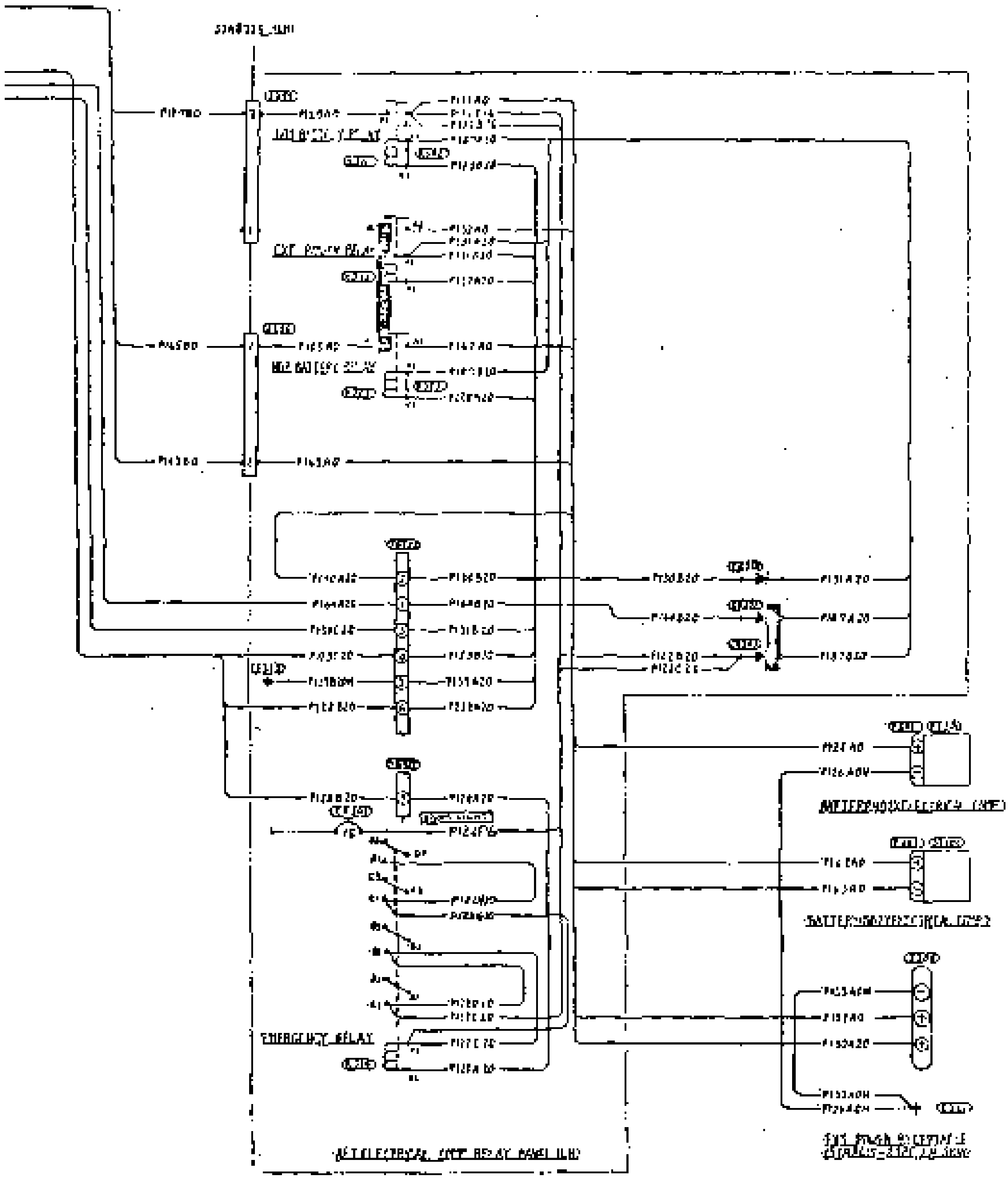
① 661SA, 697SA - 700SA

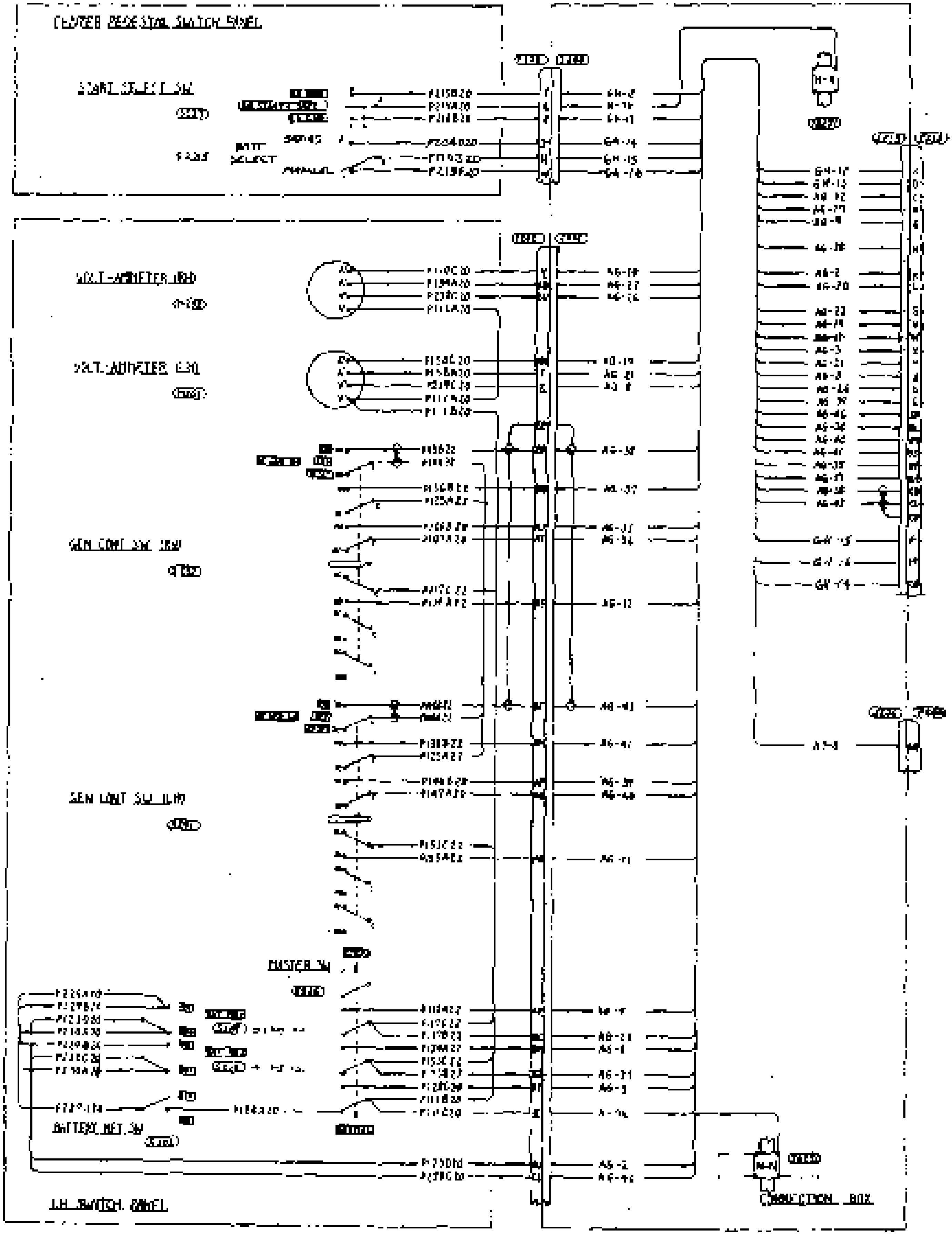
② 701SA - 713SA

DC POWER SOURCE

Aircraft S/N 697SA - 713SA

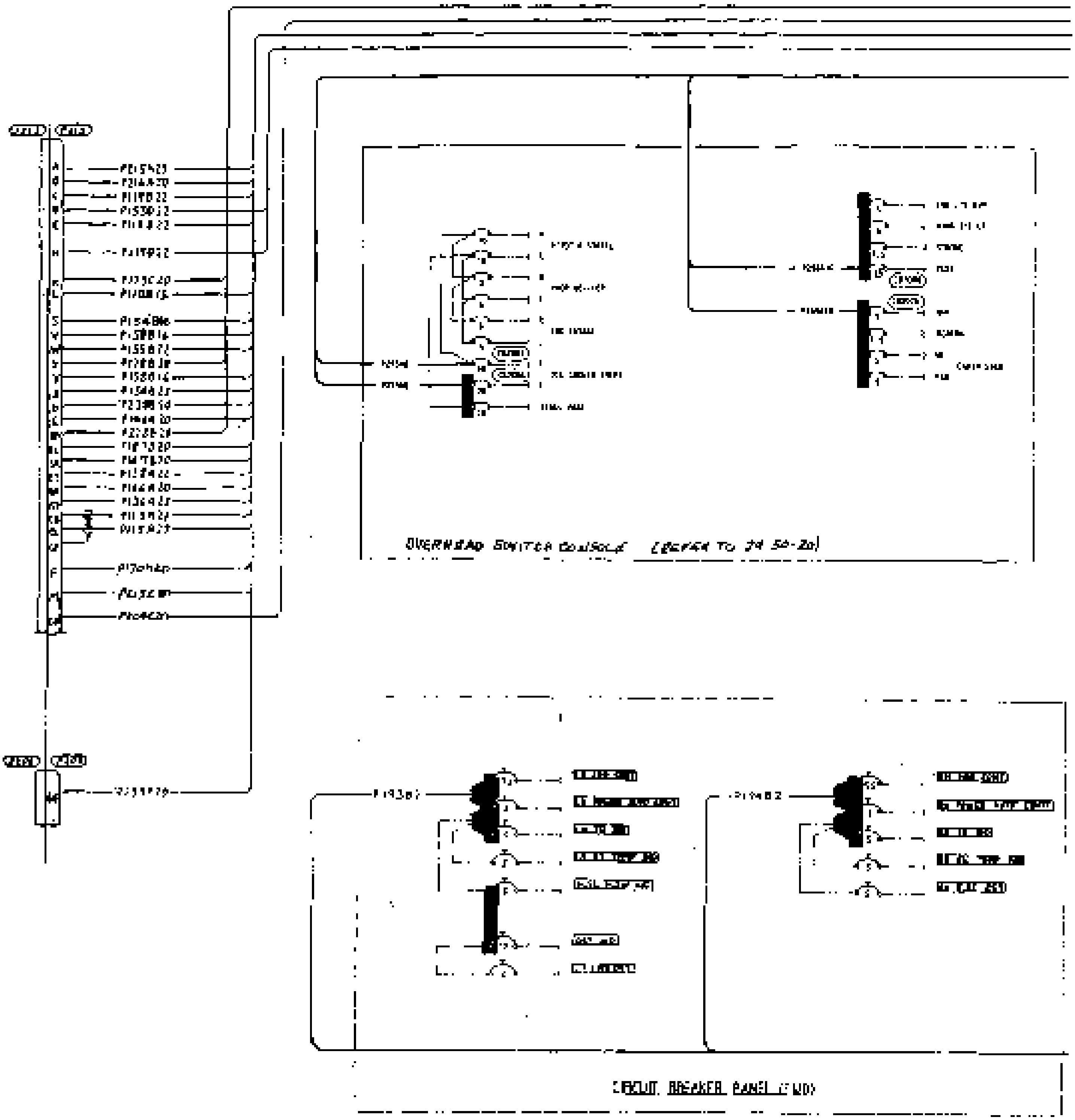

ENG START CONT RELAY
VOLTAGE SENSING UNIT
LAND GEAR SAFETY RELAY
EXT POWER CONT RELAY
START SELECT RELAY (RH)
START SELECT RELAY (LH)
SENSING VOLTAGE ADJUST RESISTOR
VOLTAGE SENSING UNIT TEST JACK
GEN VOLTAGE TEST
FLTB RELAY BOX (K)
NO POWER EXCHANGE RELAY
FLTB RELAY BOX (AC)


EMERGENCY RELAY



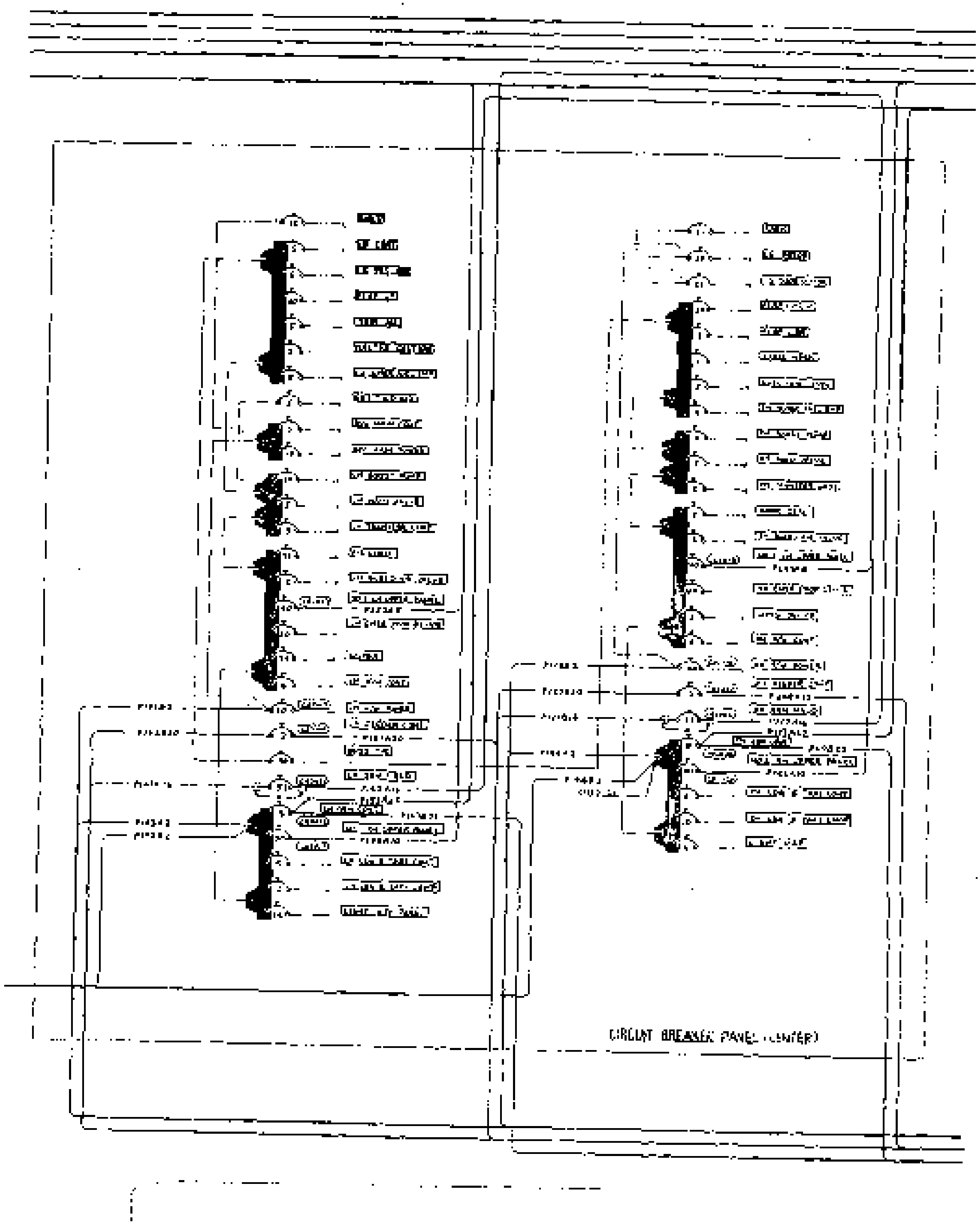
DC POWER SOURCE

Aircraft S/N 714SA - 730SA

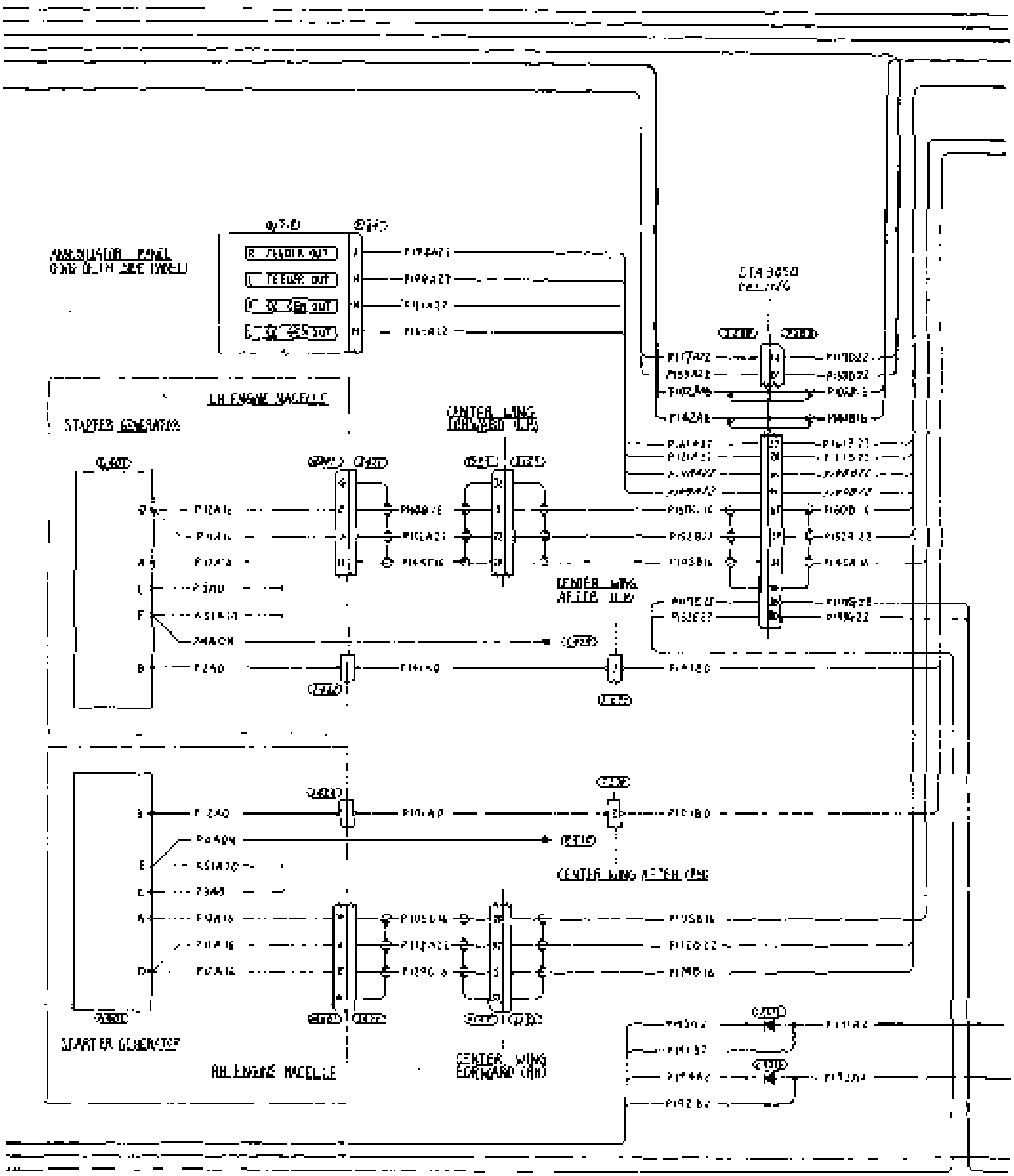


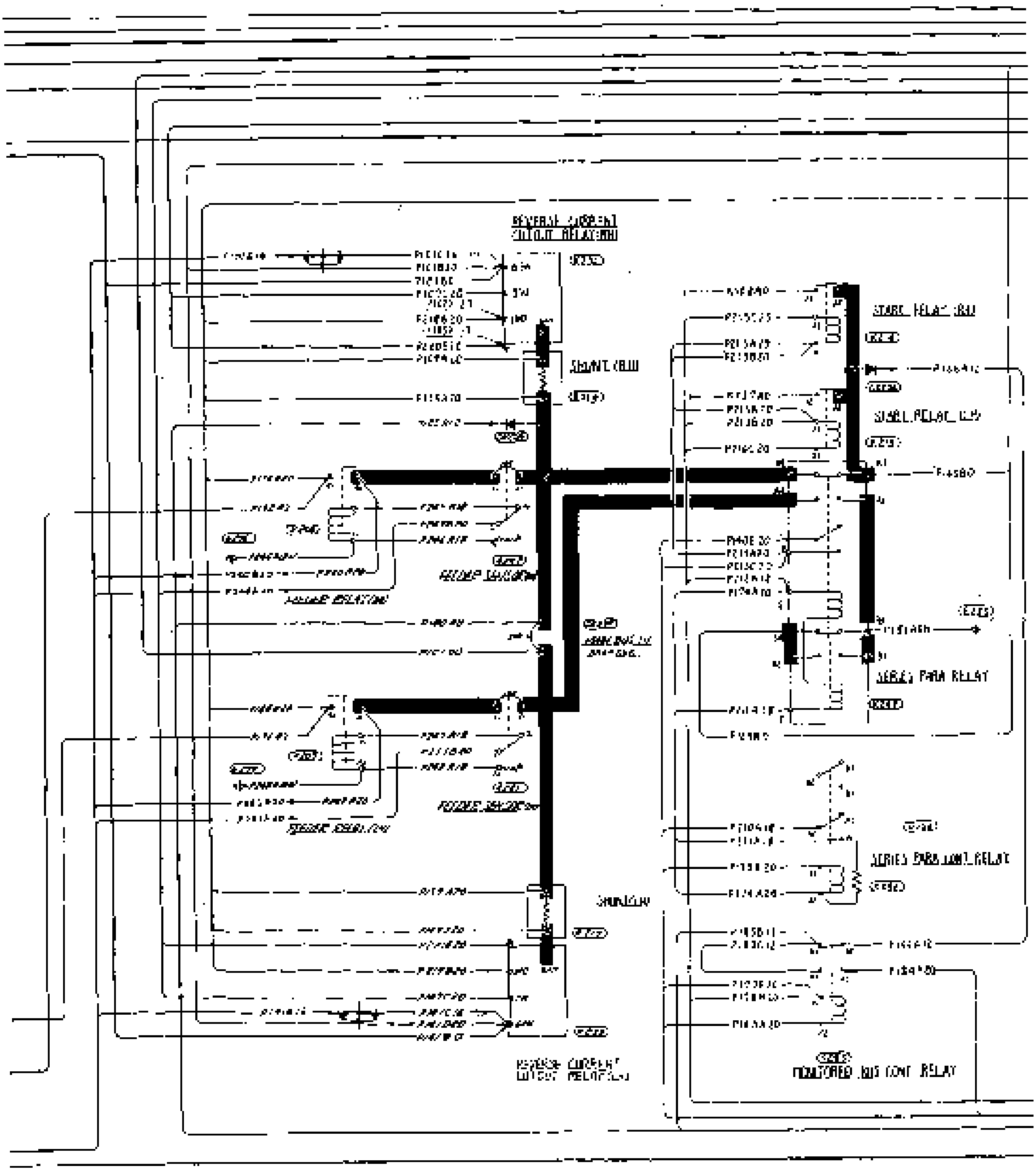
DC POWER SOURCE

Aircraft S/N 714SA - 730SA

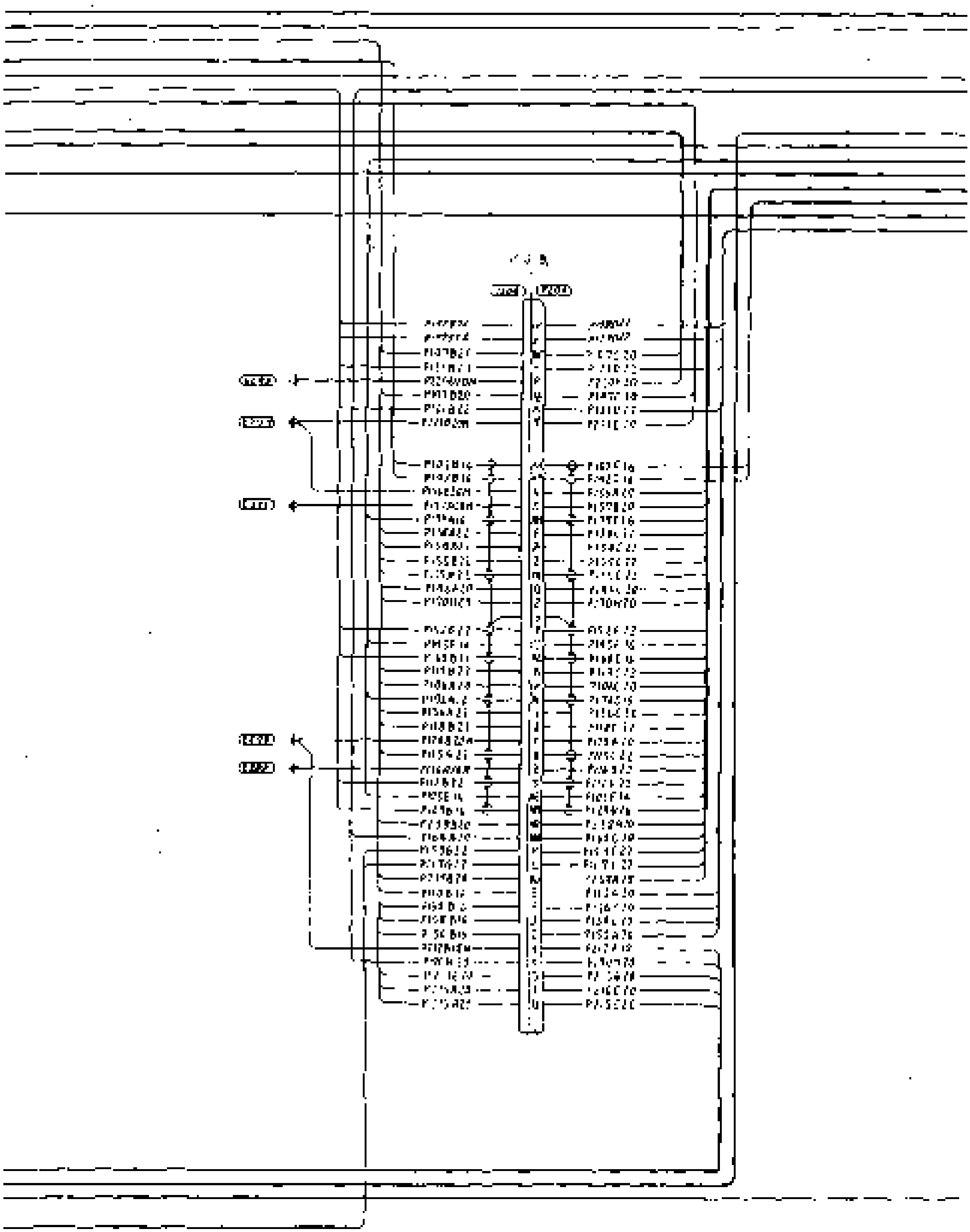


DC POWER SOURCE
Aircraft S/N 714SA - 730SA

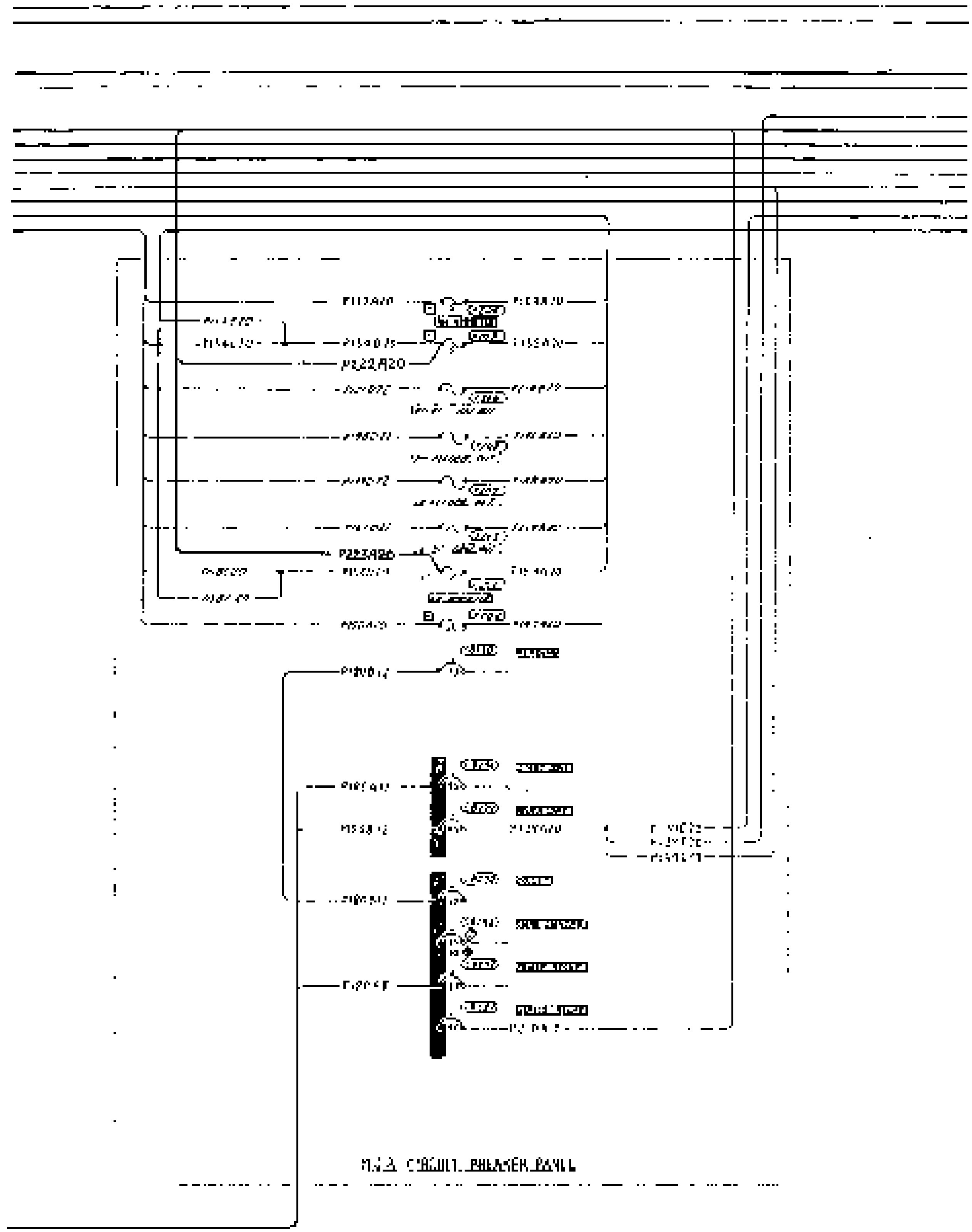

LH ENGINE NACELLE
STARTER GENERATOR
RH ENGINE NACELLE
STARTER GENERATOR



DC POWER SOURCE
Aircraft S/N 714SA - 730SA



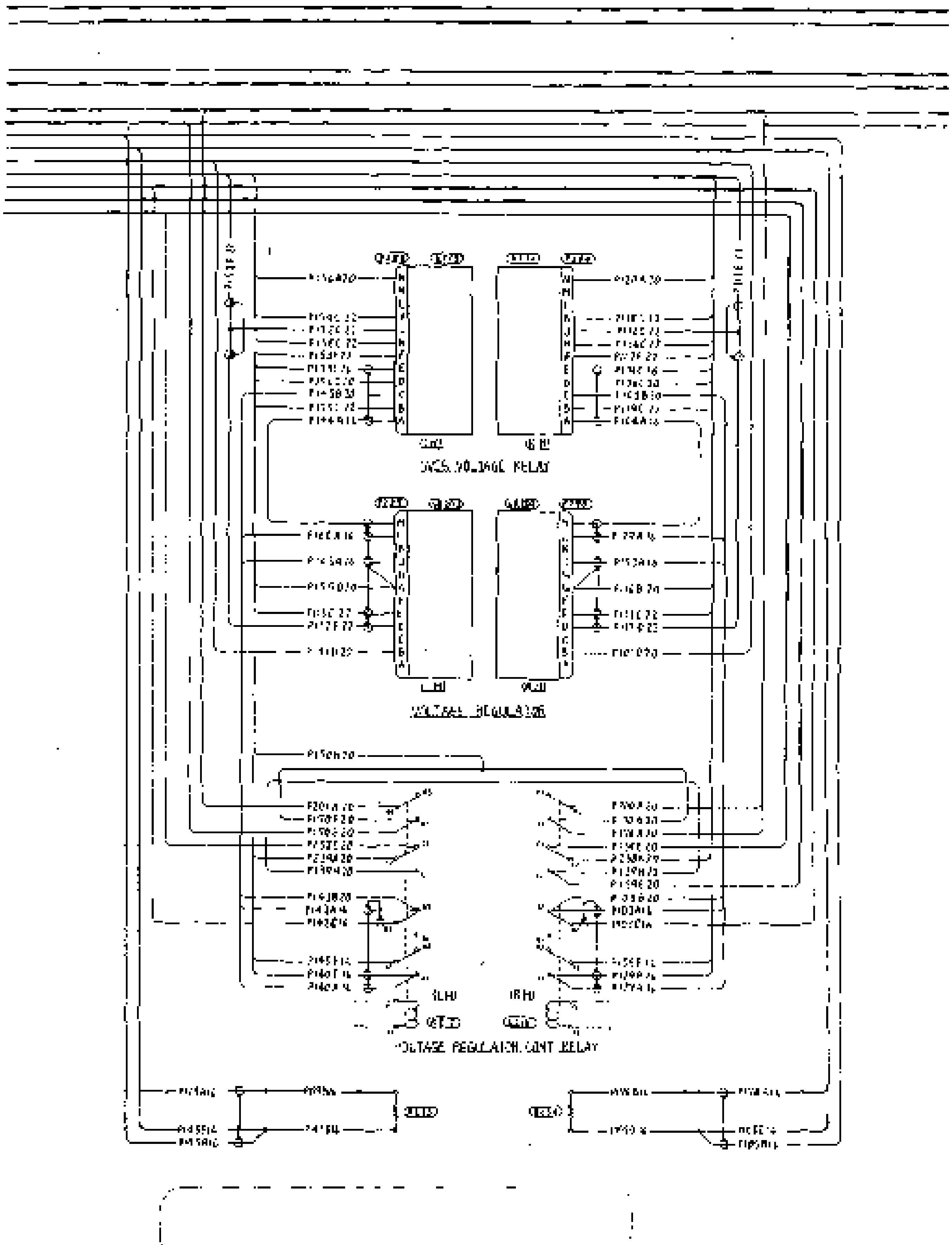
DC POWER SOURCE
Aircraft S/N 714SA - 730SA

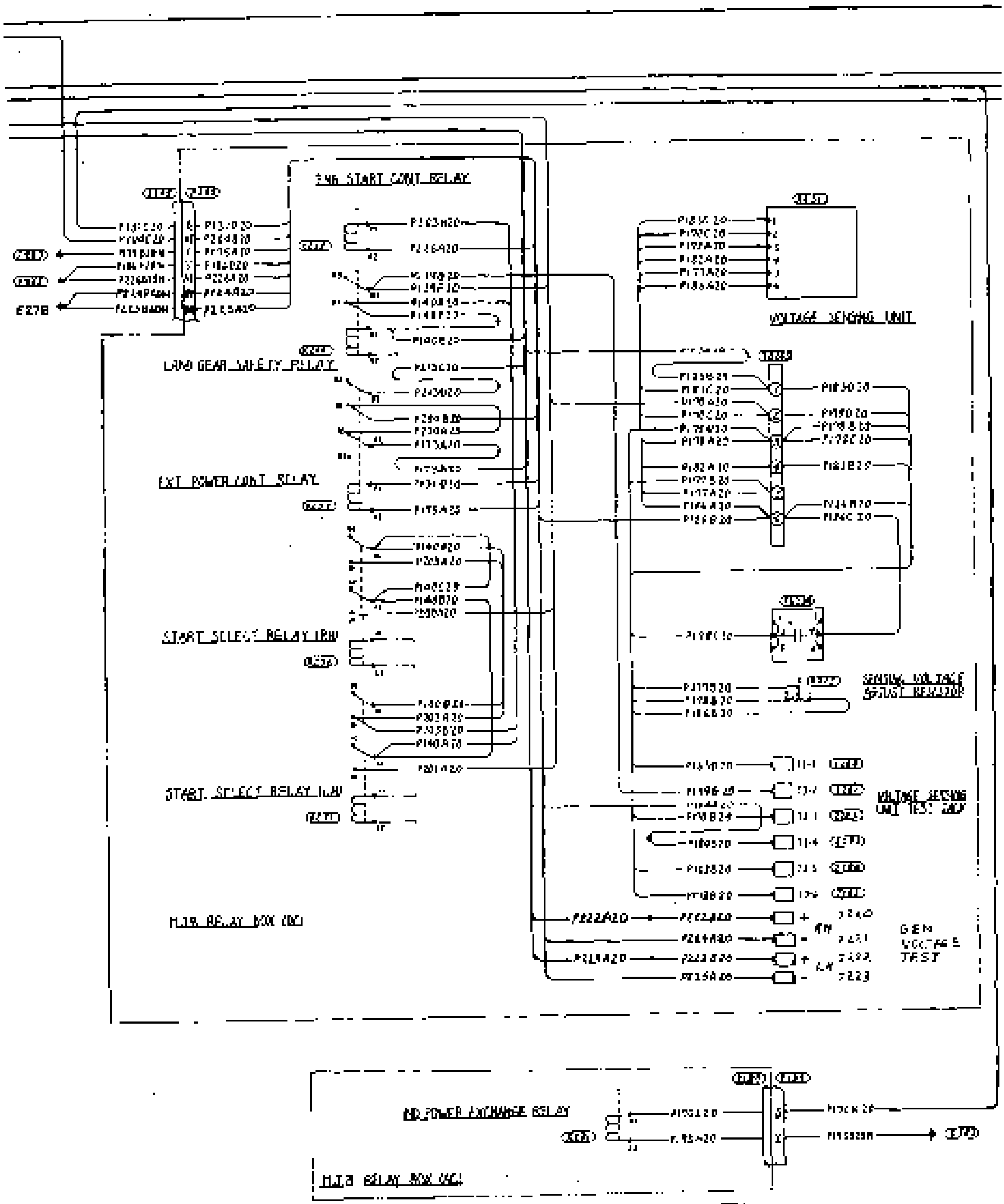


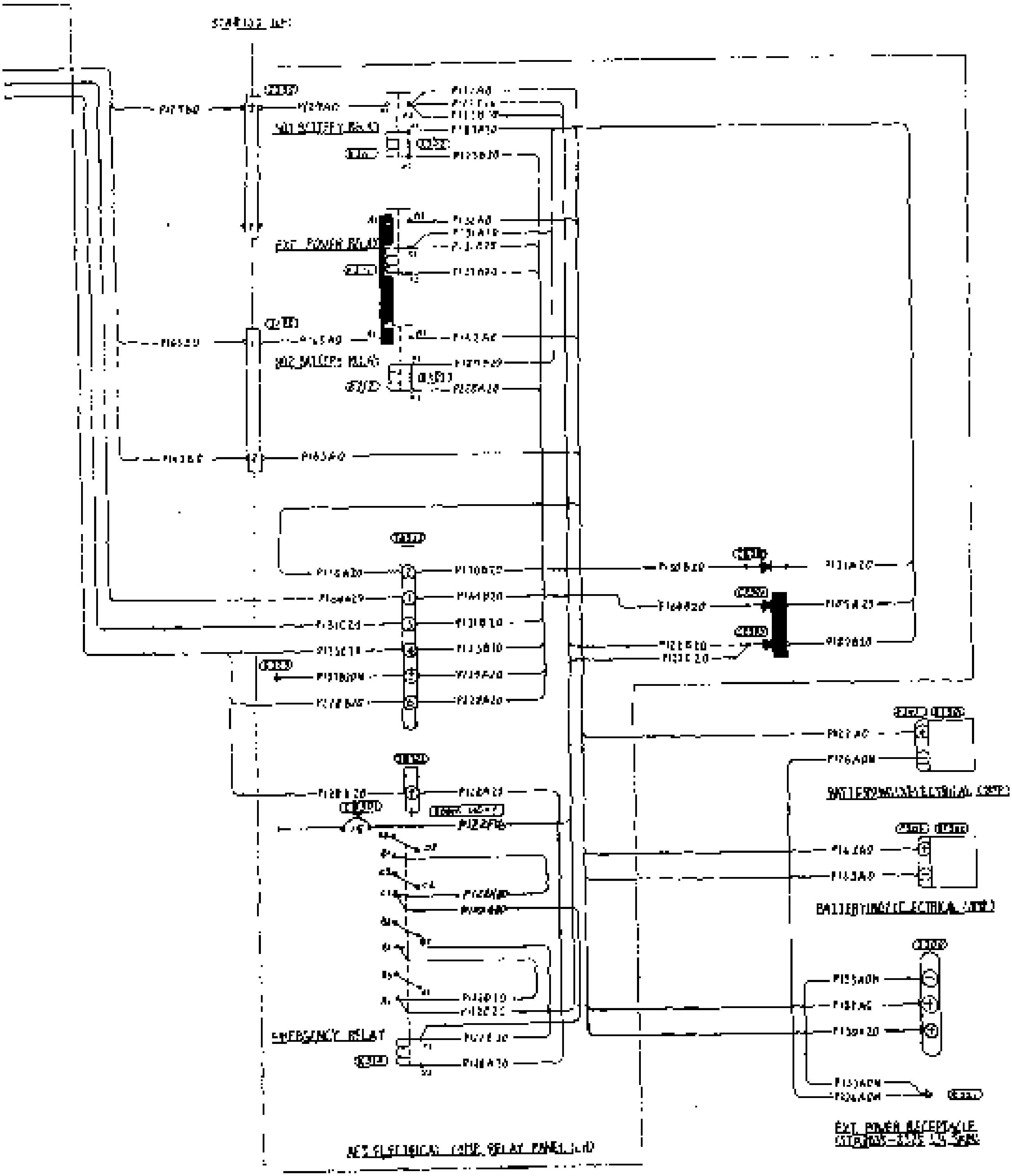
◇1 Aircraft S/N 714SA Thru 723SA
◇2 Aircraft S/N 724SA And Up

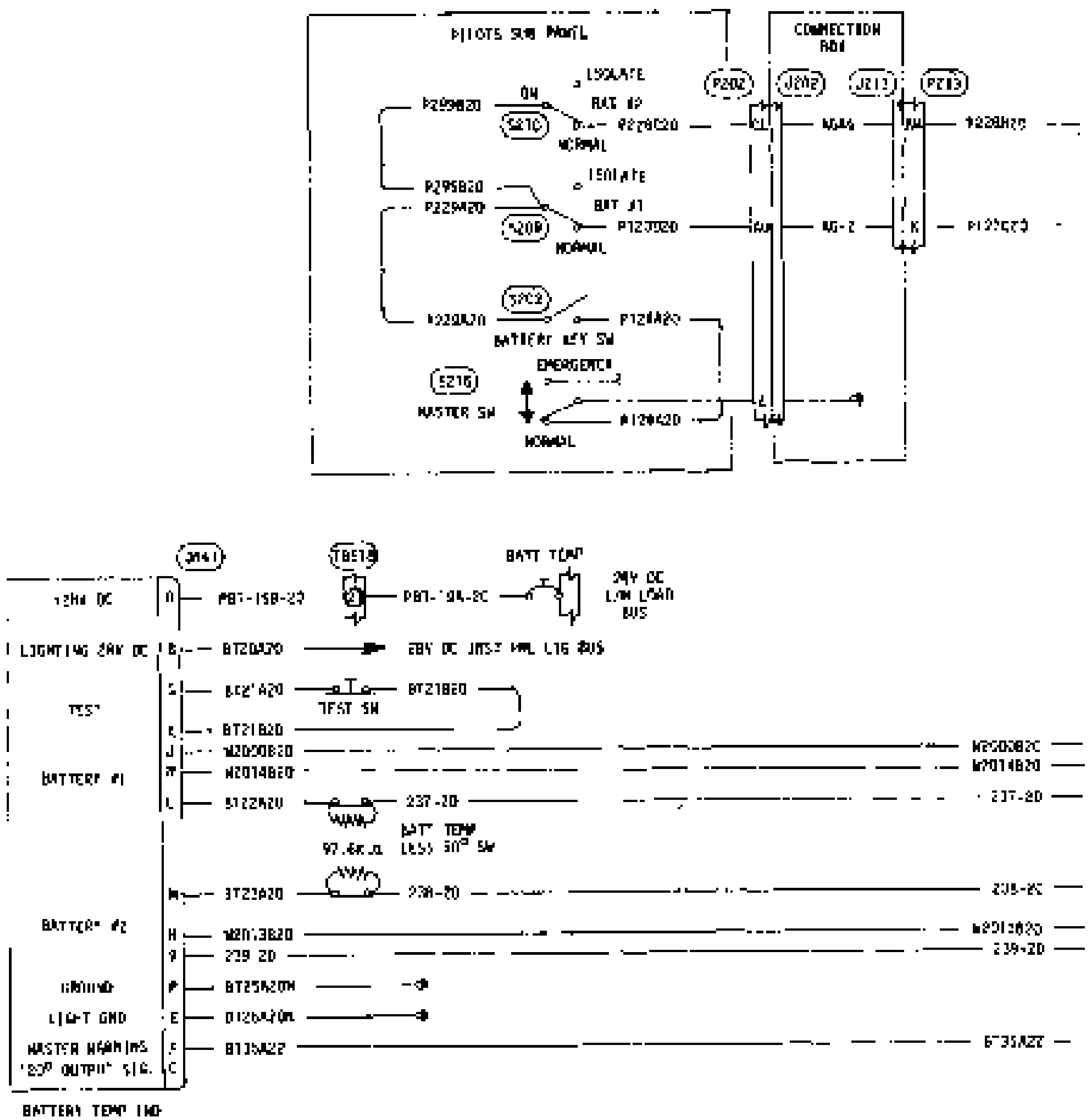
DC POWER SOURCE

Aircraft S/N 714SA - 730SA


OVER VOLTAGE RELAY
VOLTAGE REGULATORS
VOLTAGE REGULATION CONT RELAY

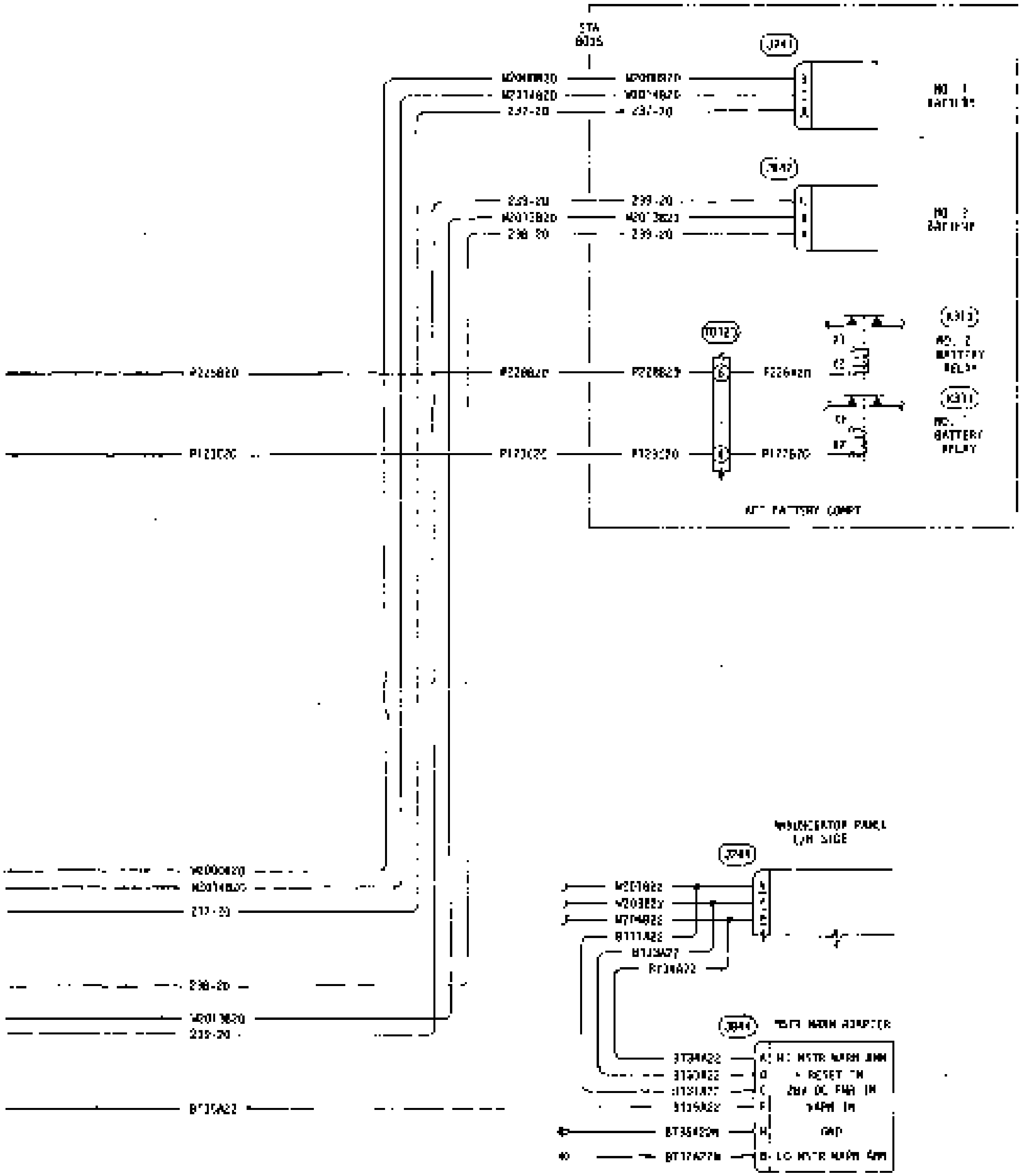

ENG START CONT RELAY
LAND GEAR SAFETY RELAY
EXT POWER CONT RELAY
START SELECT RELAY (RH)
START SELECT RELAY (LH)
M18 RELAY BOX (DC)
VOLTAGE SENSING UNIT
SENSING VOLTAGE ADJUST RESISTOR
VOLTAGE SENSING UNIT TEST JACK
GEN VOLTAGE TEST
E27B
AC POWER EXCHANGE RELAY
M18 RELAY BOX (AC)


STARTER GEN
NO1 BATTERY RELAY
EXT. POWER RELAY
NO2 BATTERY RELAY
EMERGENCY RELAY
BATTERY NO.1 ELECTRICAL COMP.
BATTERY NO.2 ELECTRICAL COMP.
EXT. POWER RECEPTACLE
AFT ELECTRICAL COMP. RELAY PANEL



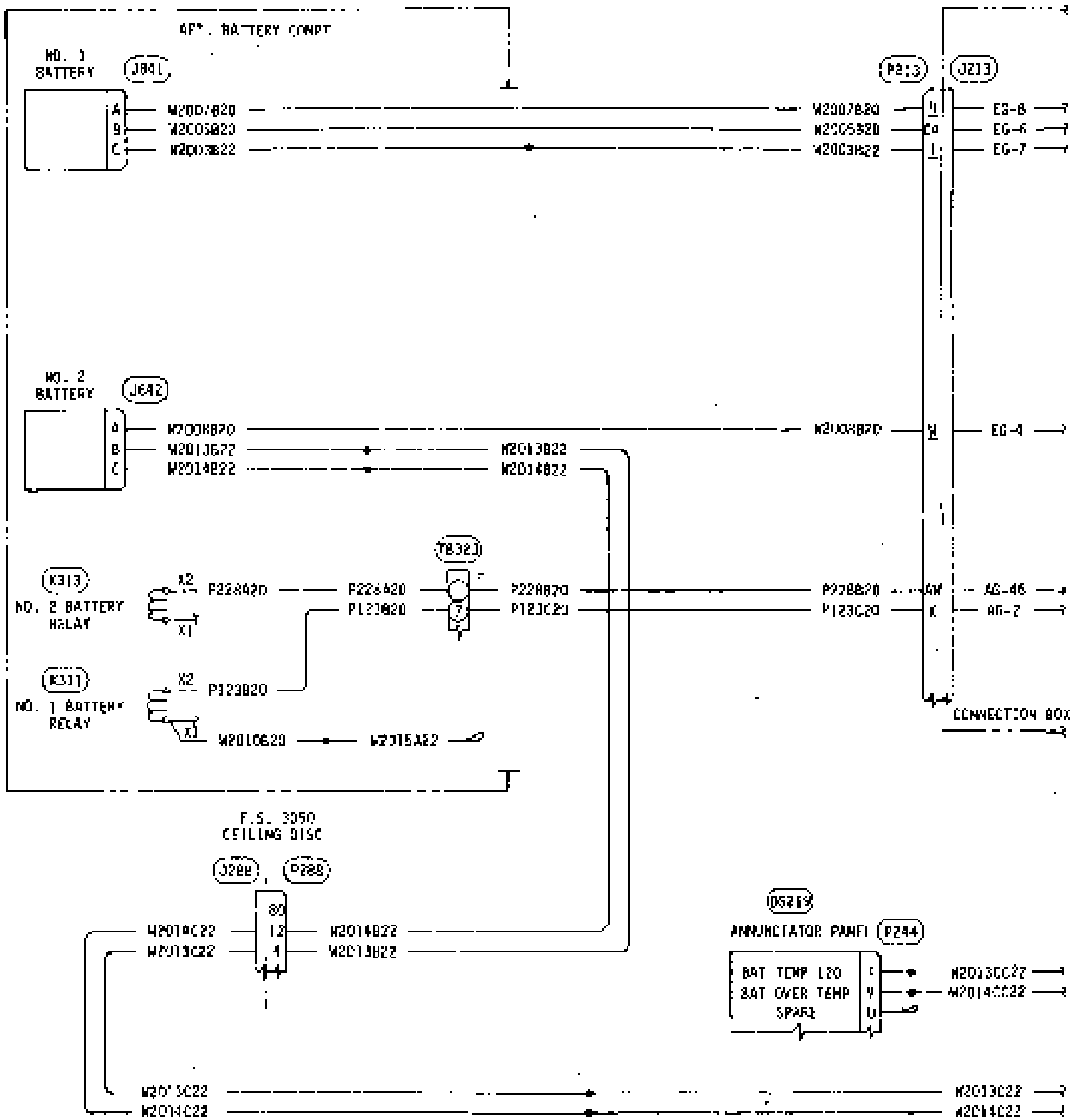
BATTERY OVERTEMP WARNING

Aircraft S/N 661SA



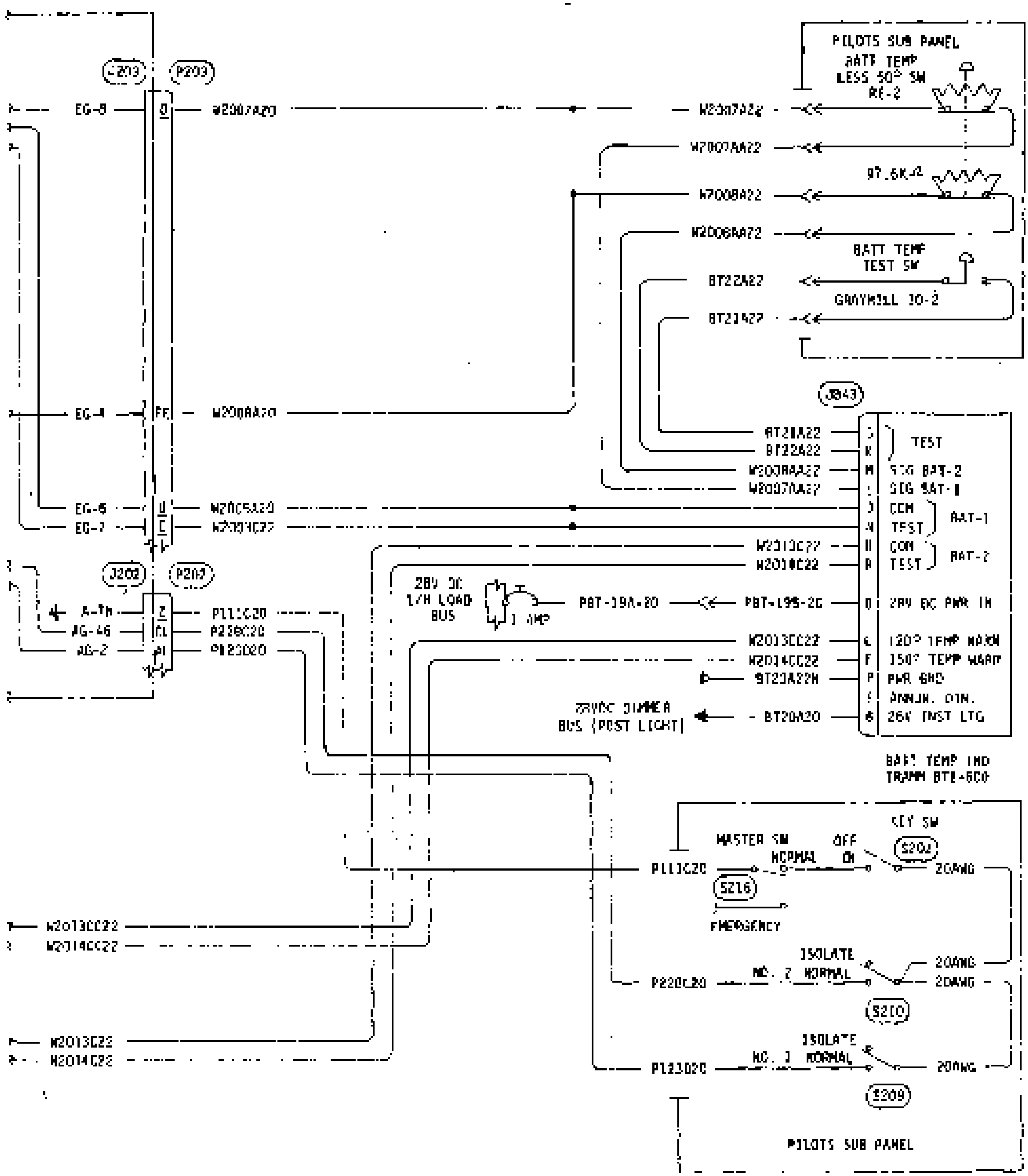
BATTERY OVERTEMP WARNING

Aircraft S/N 66ISA



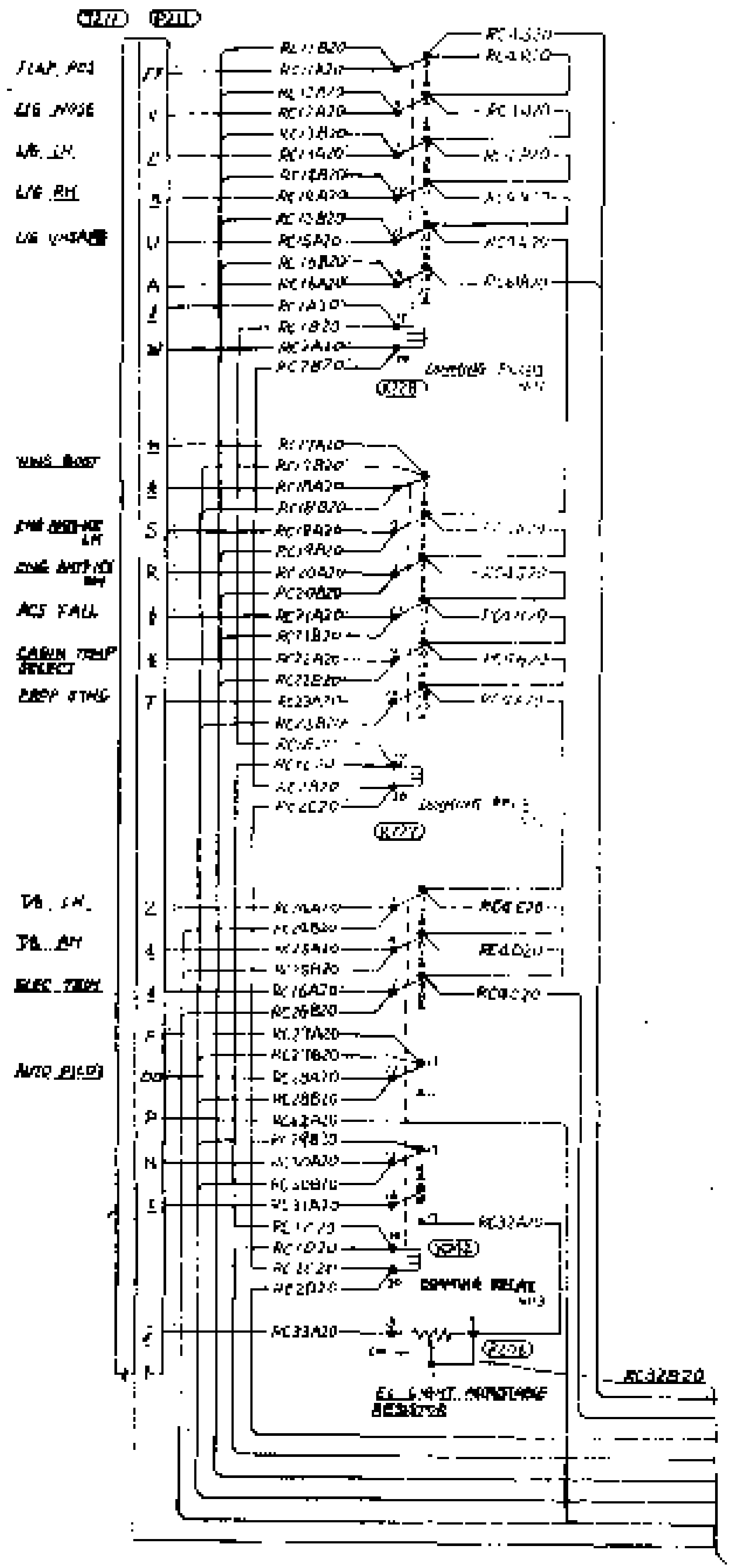
BATTERY OVERTEMP WARNING

Aircraft S/N 697SA - 730SA



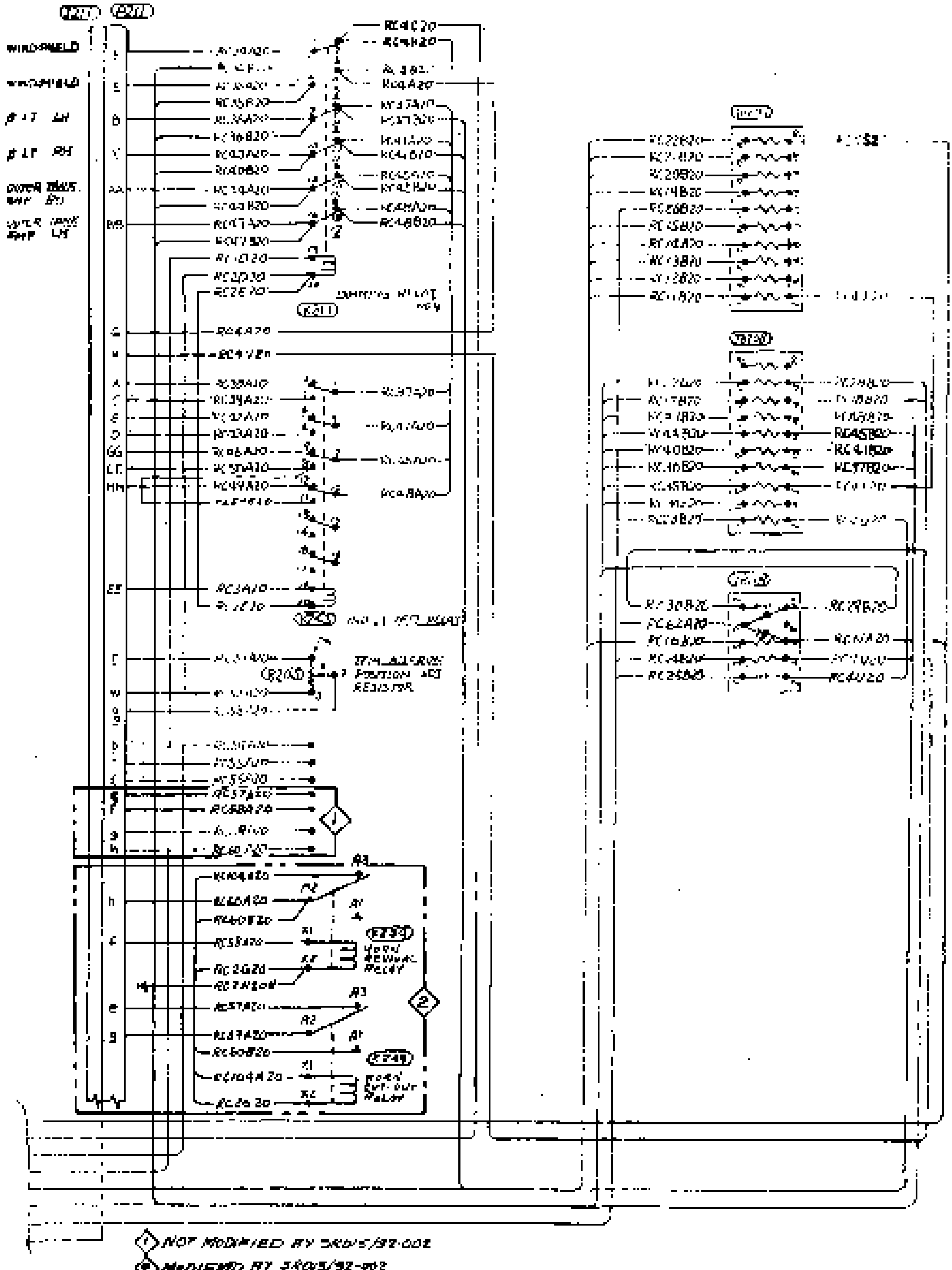
BATTERY OVERTEMP WARNING

Aircraft S/N 697SA - 730SA



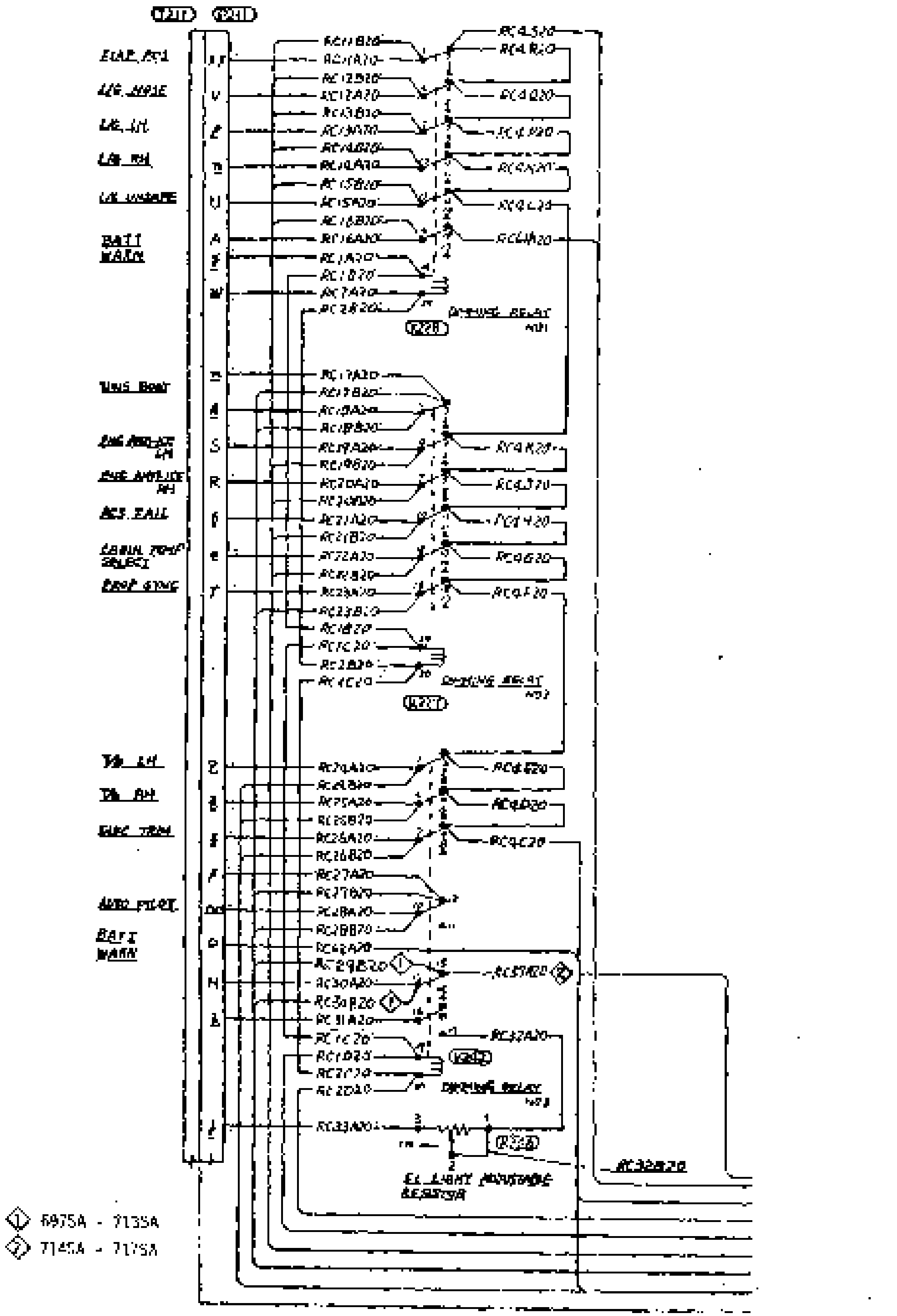
COCKPIT RELAY BOX

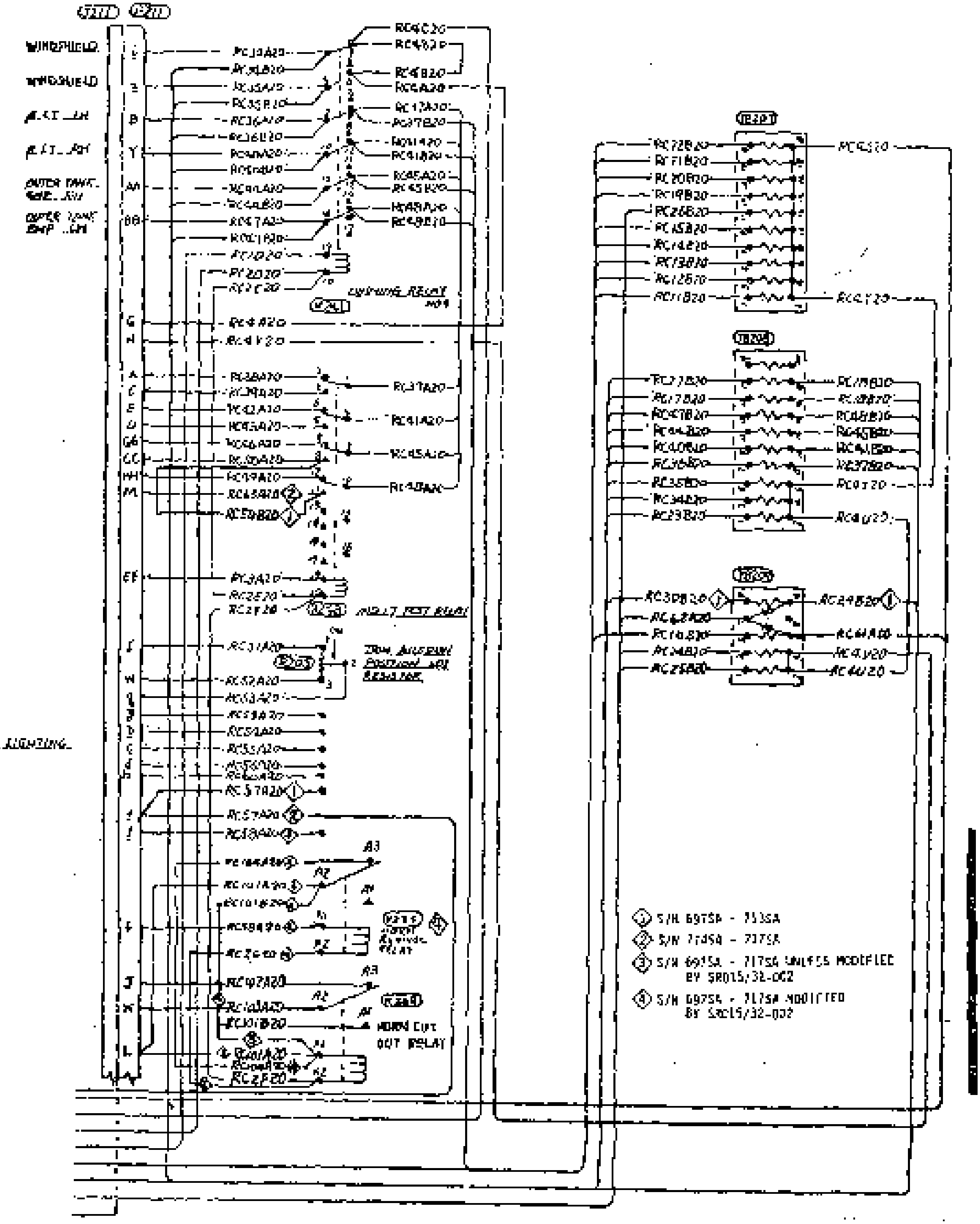
Aircraft S/N 661SA



COCKPIT RELAY BOX

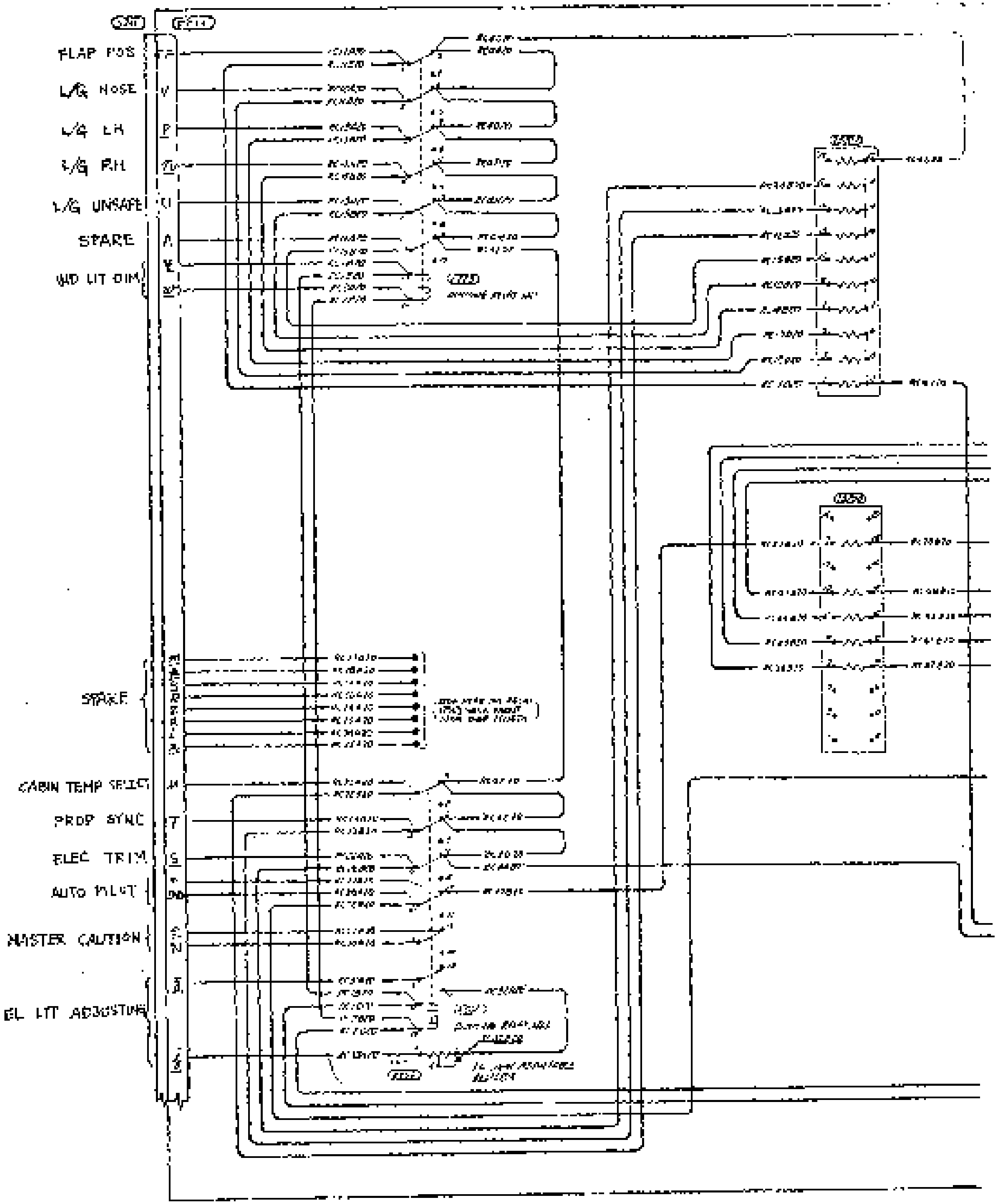
Aircraft S/N 6615A





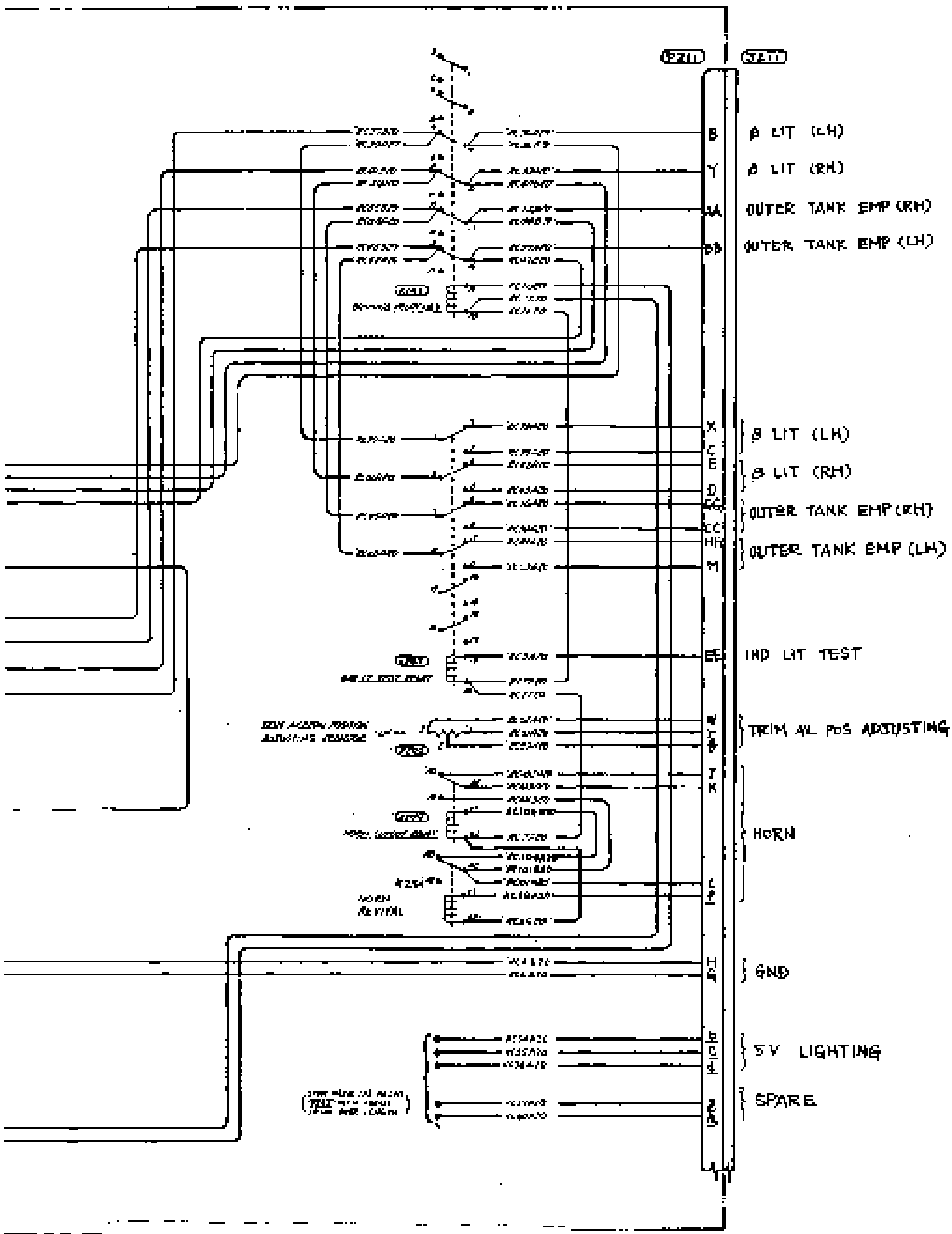
COCKPIT RELAY BOX

Aircraft S/N 697SA - 717SA

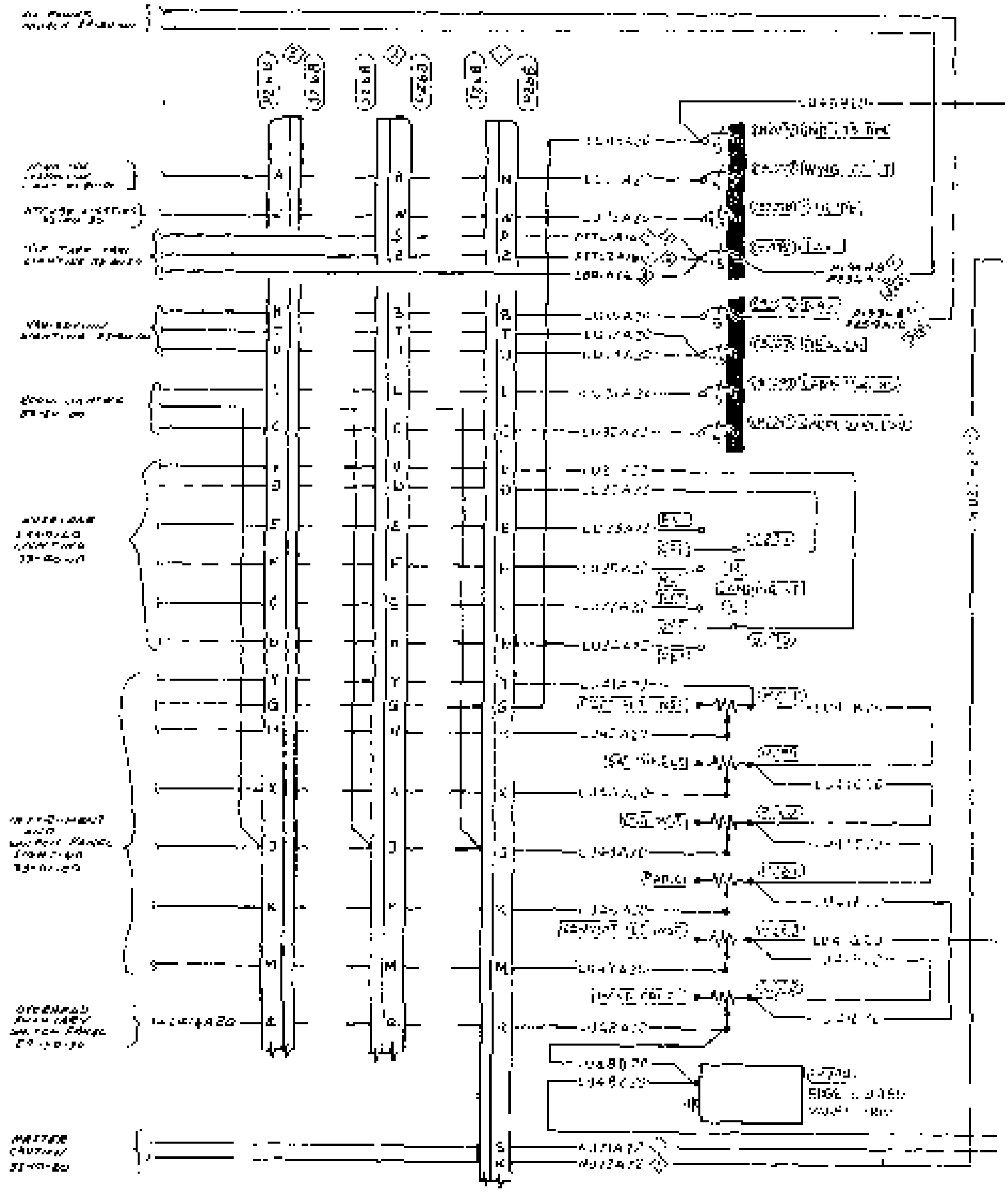


COCKPIT RELAY BOX

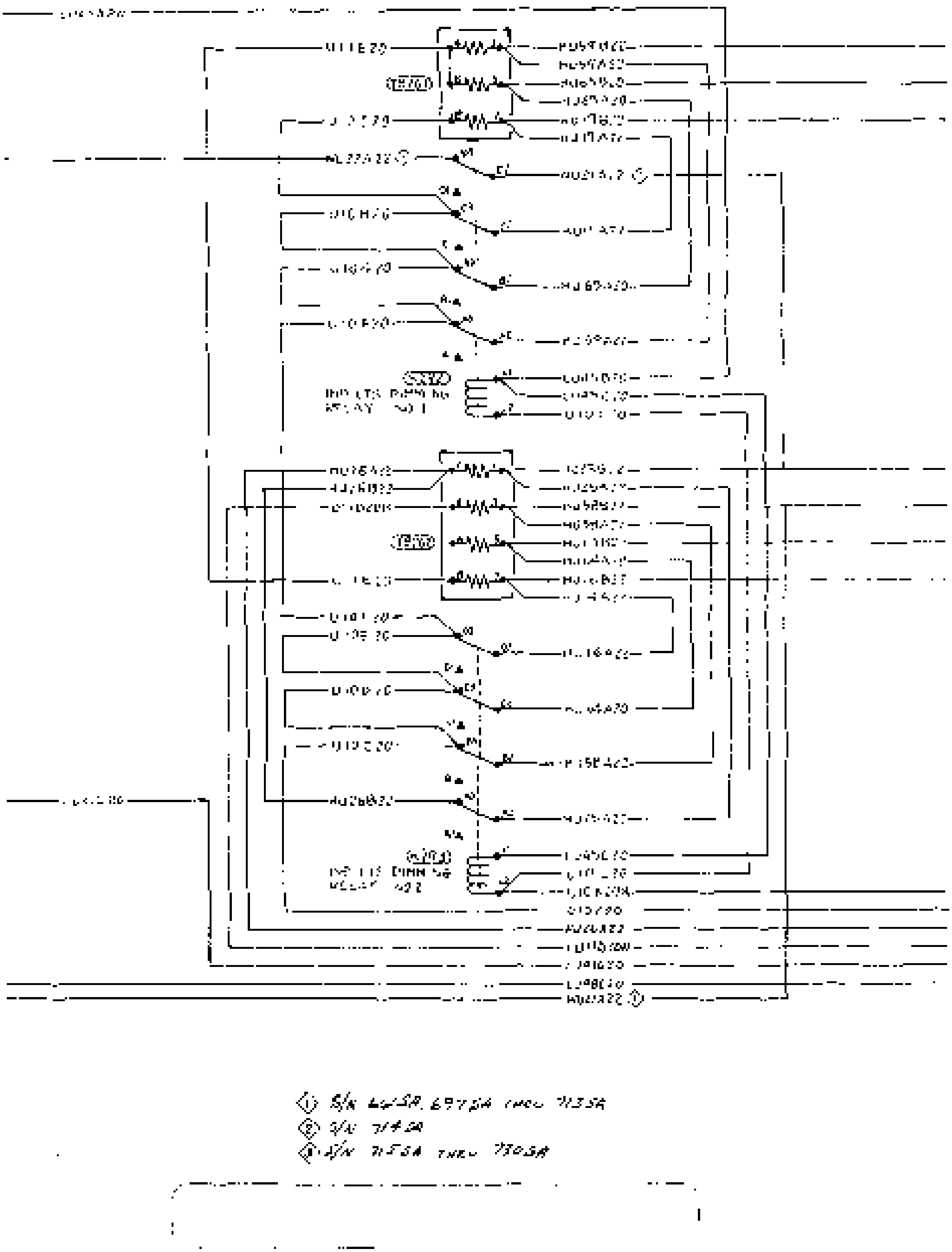
Aircraft S/N 718SA - 730SA

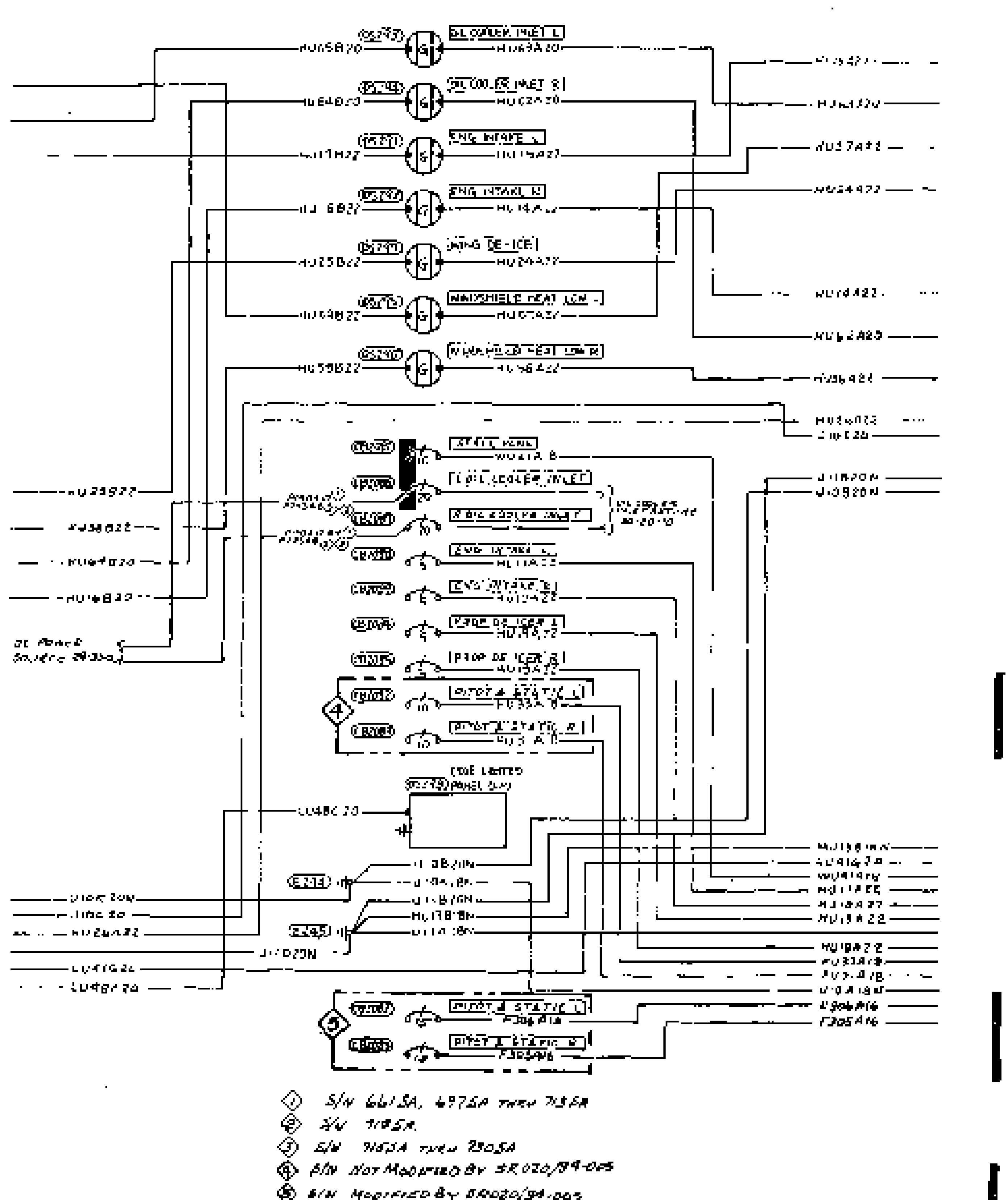


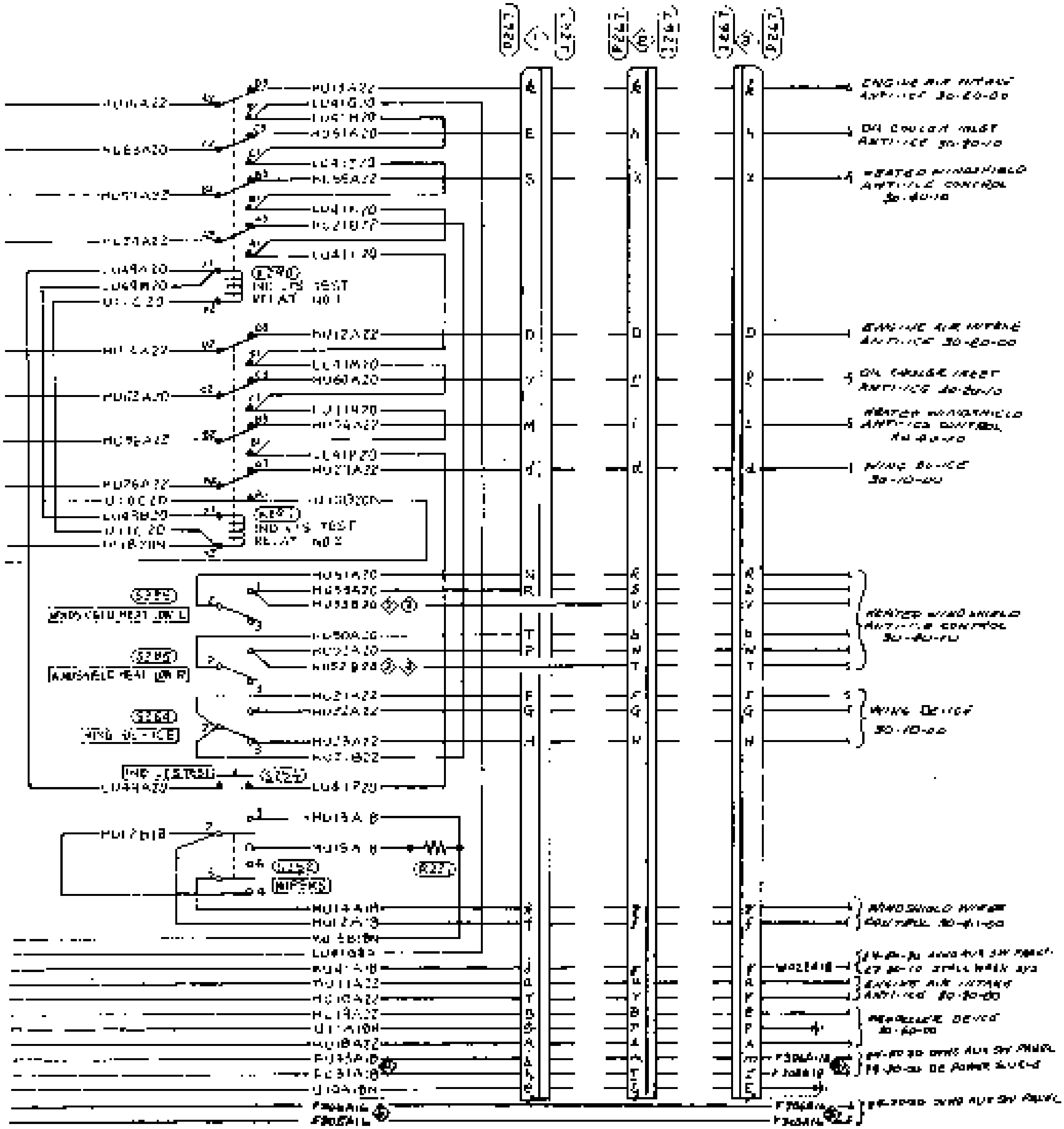
ø LIT (LH)
ø LIT (RH)
OUTER TANK EMP (RH)
OUTER TANK EMP (LH)
ø LIT (LH)
ø LIT (RH)
OUTER TANK EMP (RH)
OUTER TANK EMP (LH)
IND LIT TEST
TRIM AL POS ADJUSTING
HORN
GND
5V LIGHTING
SPARE



1. S/N 661SA, 697SA THRU 713SA
2. S/N 714SA
3. S/N 715SA THRU 730SA





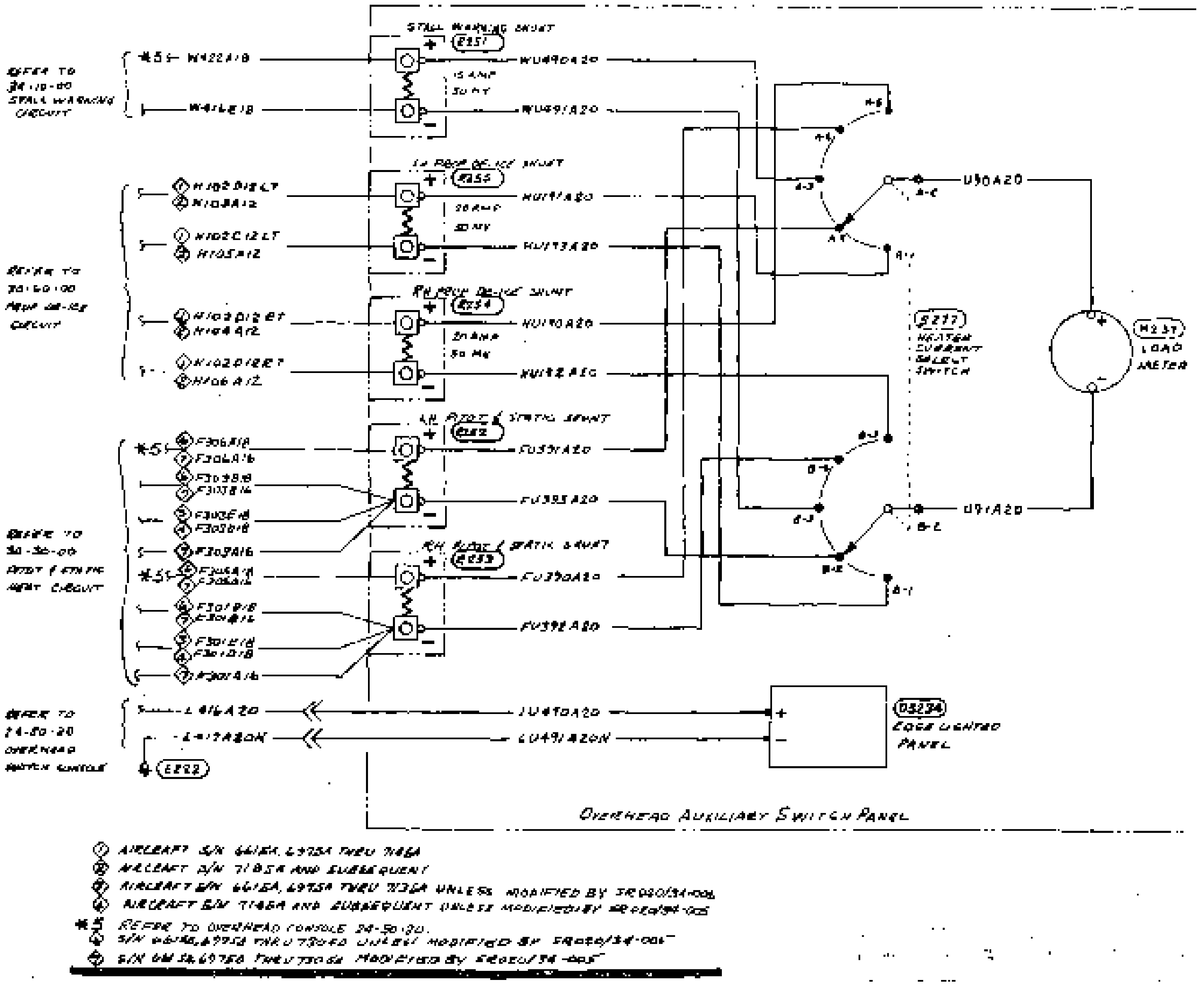


1. S/N 661SA, 697SA THRU 713SA
2. S/N 714SA
3. S/N 714SA THRU 730SA
4. S/N NOT MODIFIED BY SB020/34-005
5. S/N MODIFIED BY SB020/34-005

OVERHEAD SWITCH CONSOLE

Aircraft S/N 661SA, 697SA-730SA

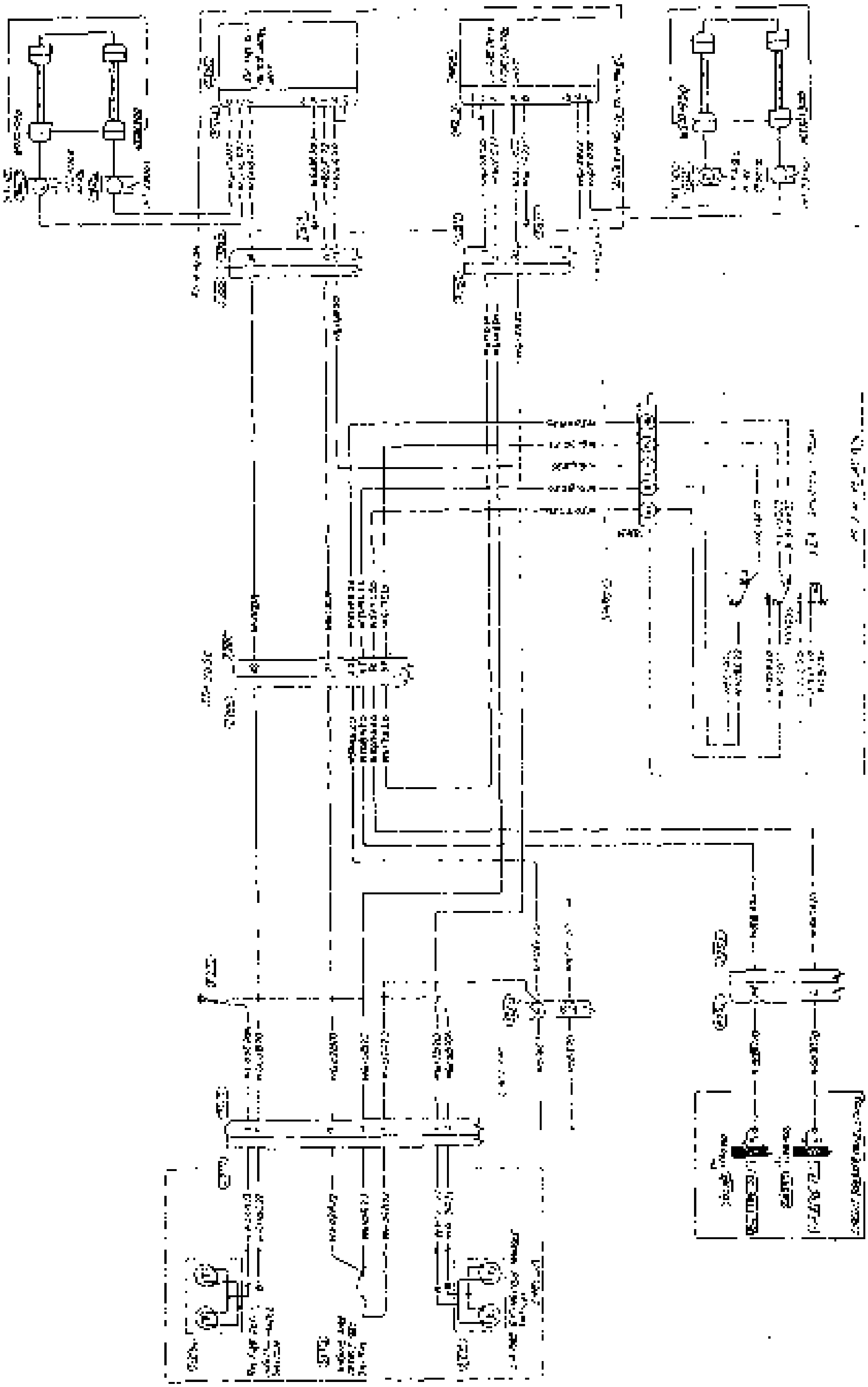
OVERHEAD AUXILIARY SWITCH PANEL
Aircraft S/N 661SA, 697SA - 730SA



CHAPTER

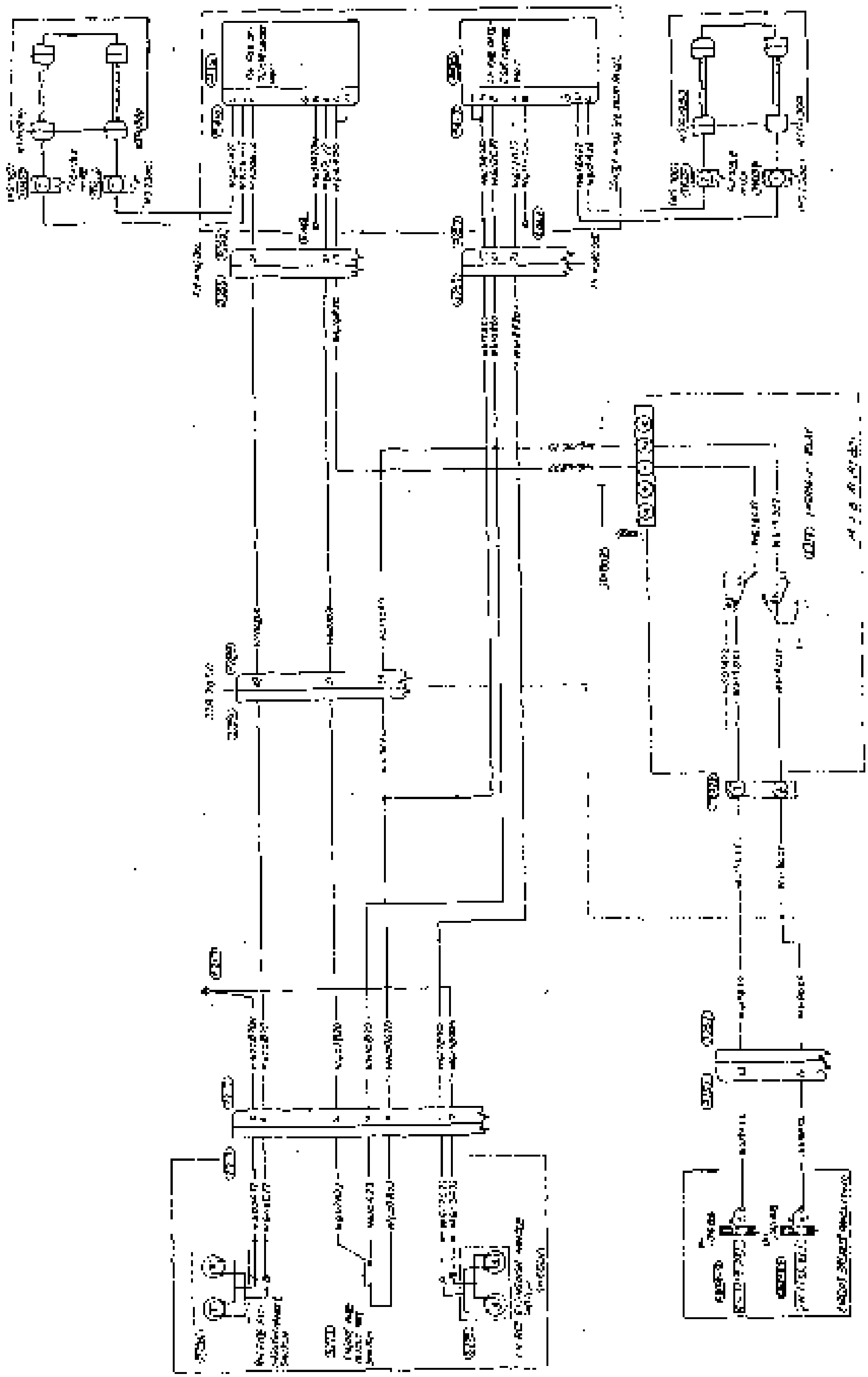
26

FIRE PROTECTION



ENGINE FIRE DETECTING CIRCUIT

Aircraft S/N 661SA, 697SA - 713SA

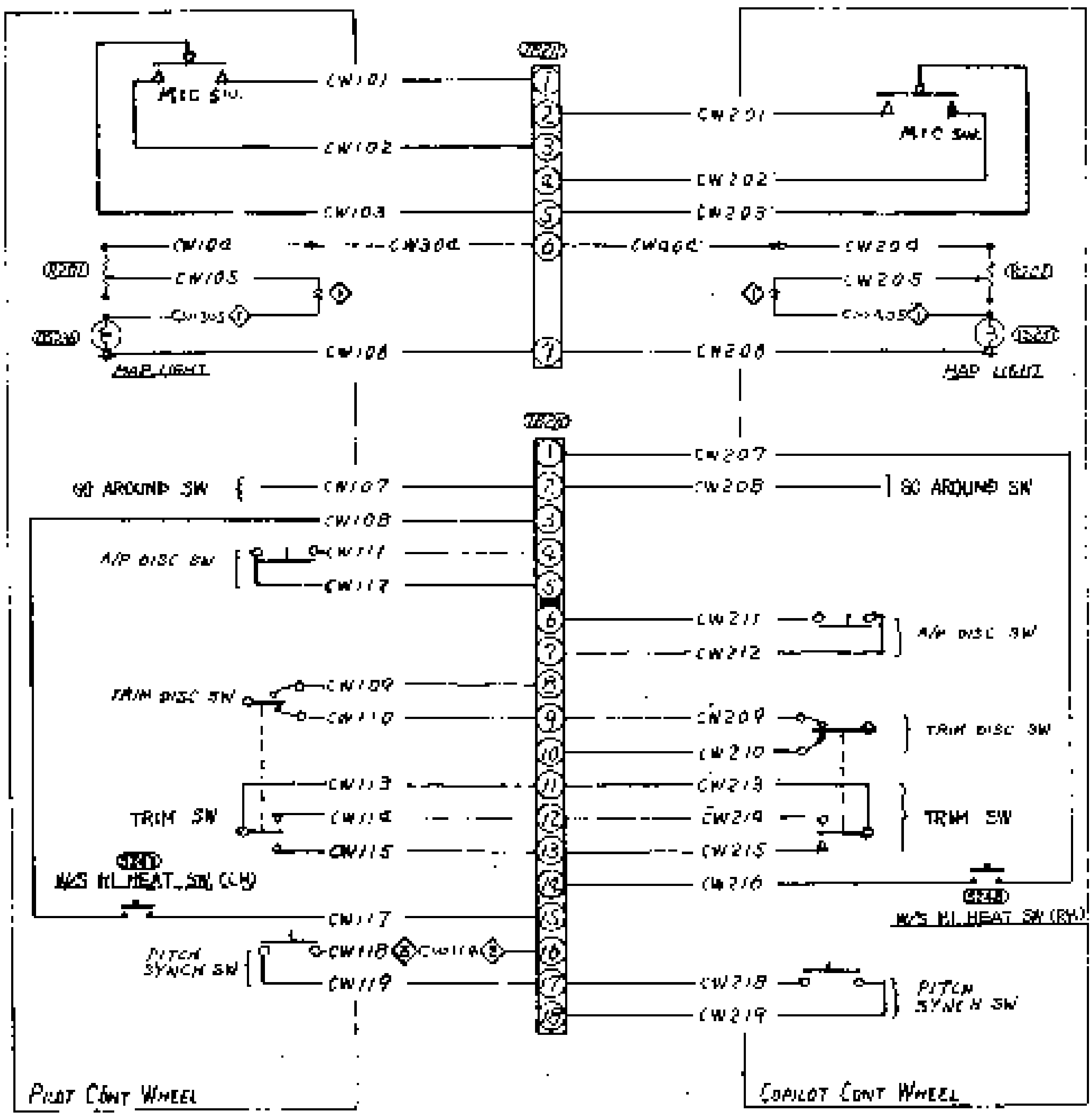


ENGINE FIRE DETECTING CIRCUIT

Aircraft S/N 714SA - 730SA

CHAPTER 27

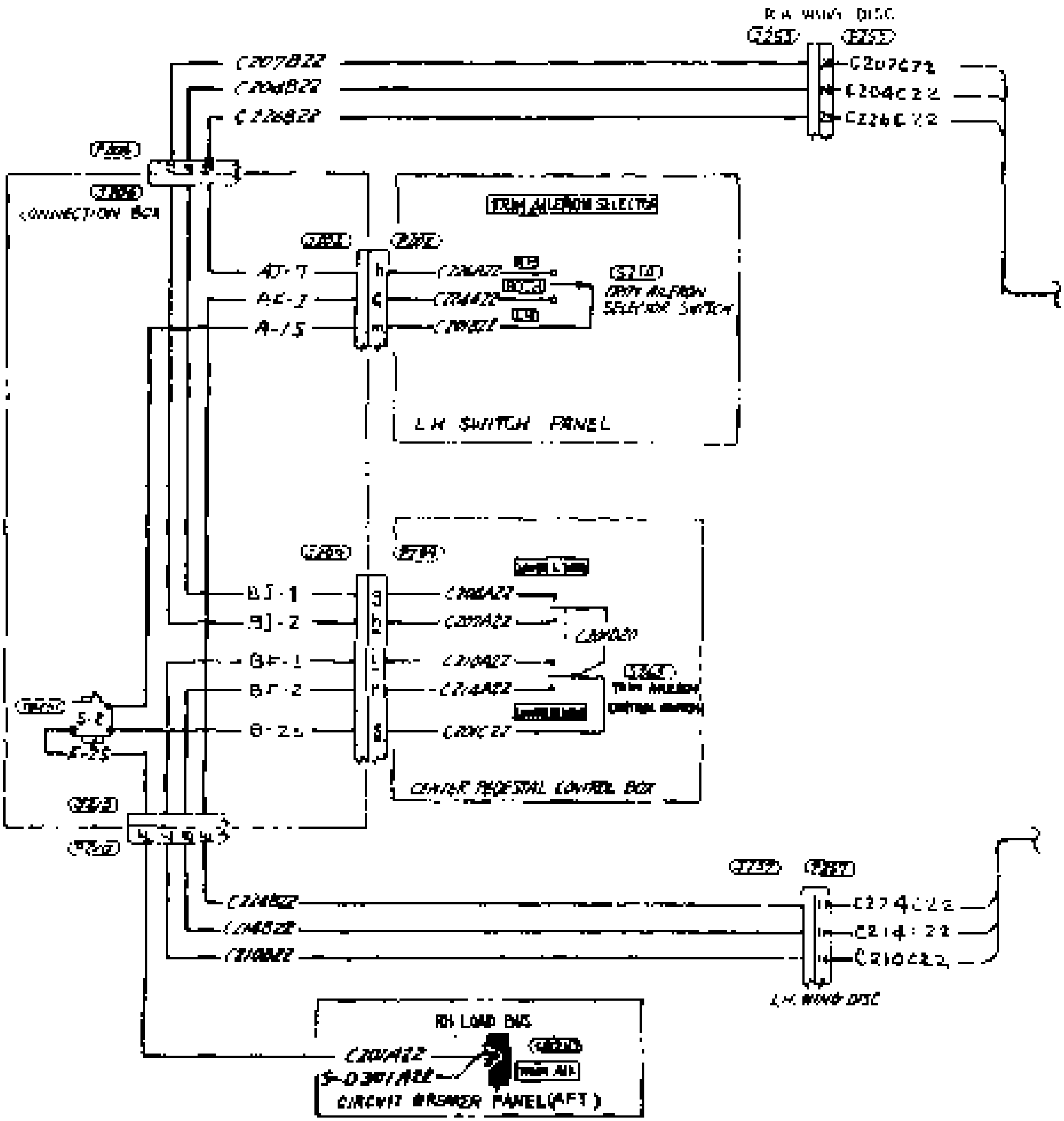
FLIGHT CONTROLS



1. 661SA, 697SA - 713SA
2. 661SA, 697SA - 704SA
3. 705SA AND SUBSEQUENT

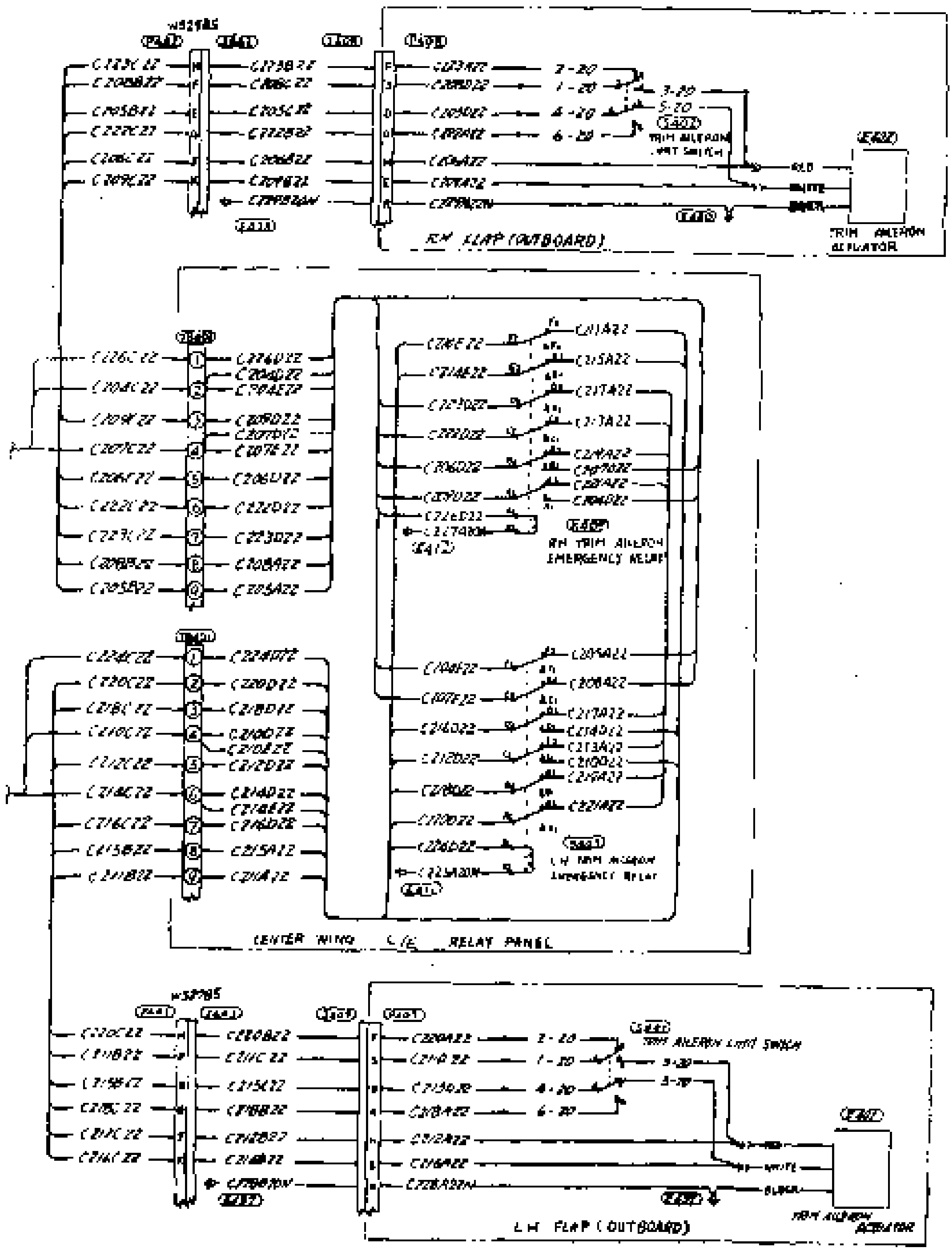
CONTROL COLUMN

Aircraft S/N 661SA, 697SA - 730SA



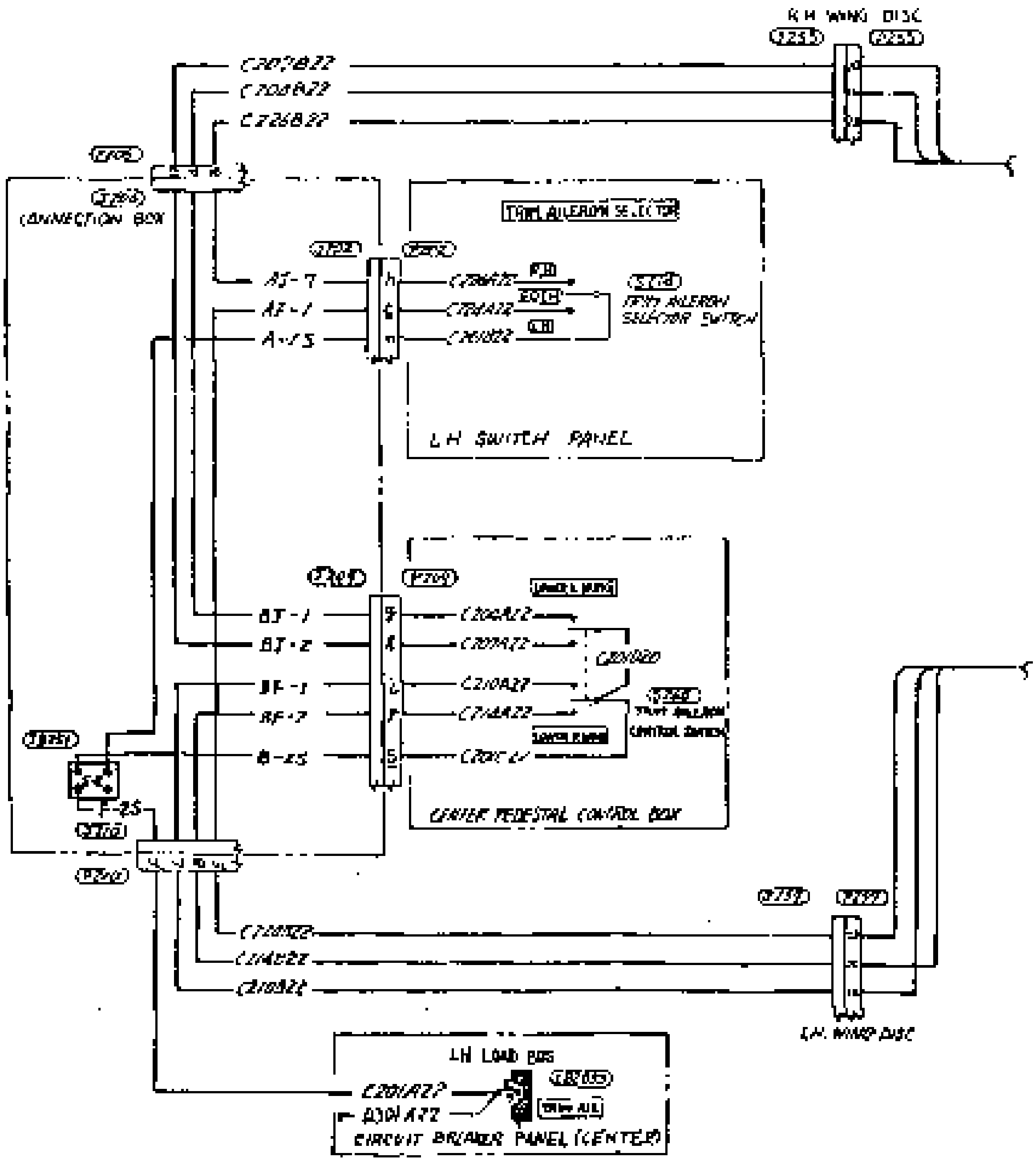
TRIM AILERON CONTROL

Aircraft S/N 661SA, 697SA - 713SA



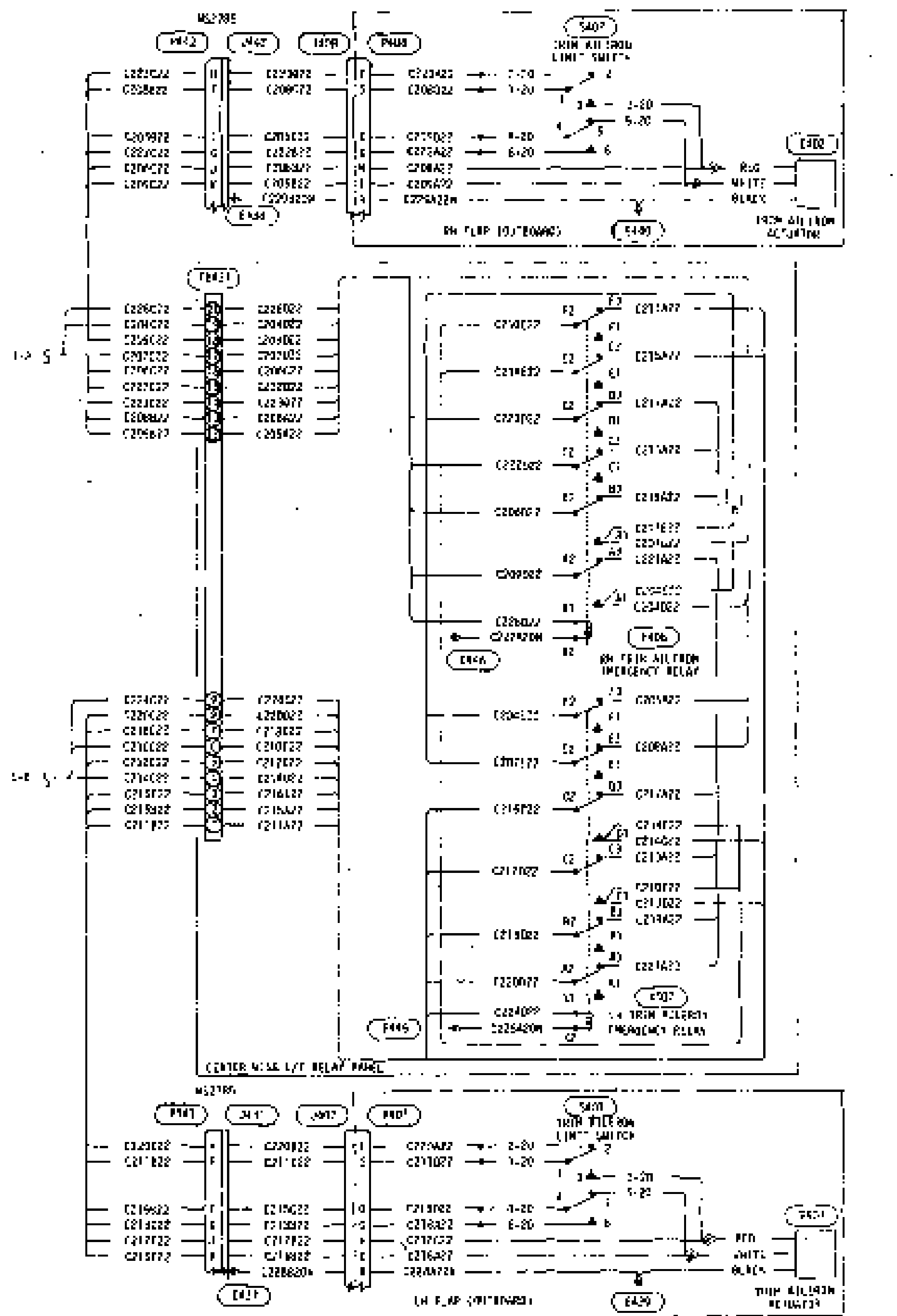
TRIM AILERON CONTROL

Aircraft S/N 661SA, 697SA - 713SA

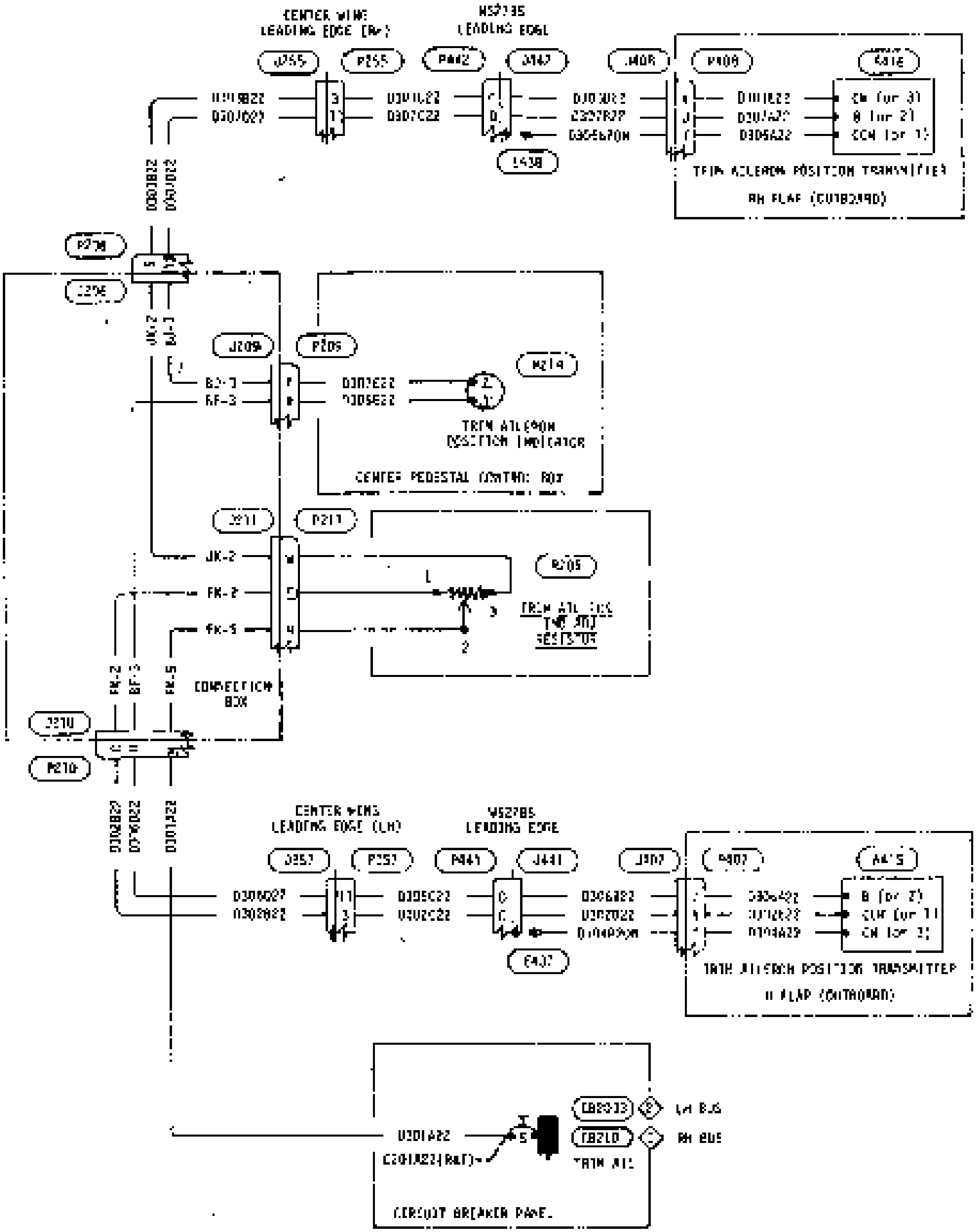


TRIM AILERON CONTROL

Aircraft S/N 714SA - 730SA



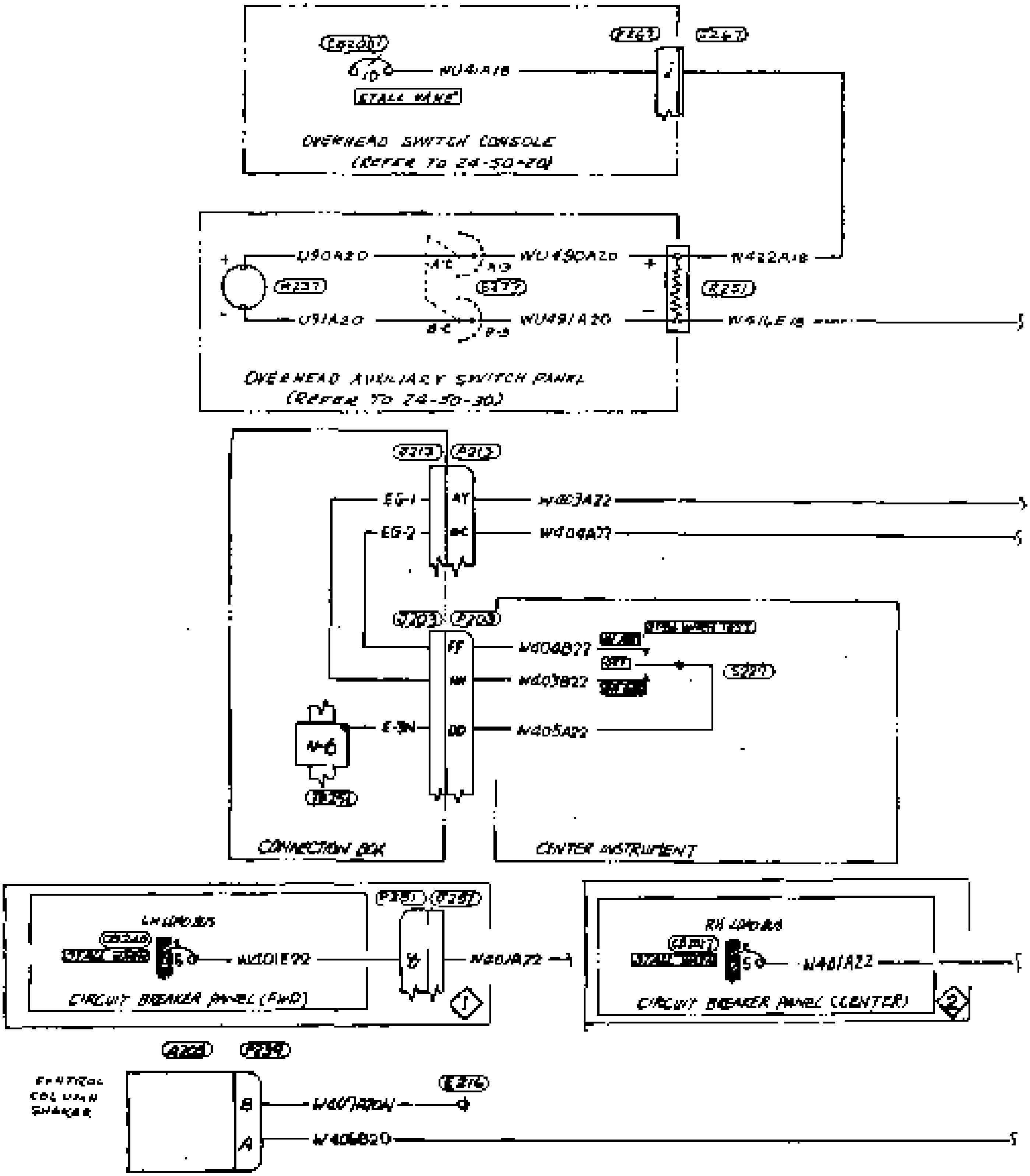
TRIM AILERON CONTROL
Aircraft S/N 714SA - 730SA



1 Aircraft S/N 661SA, 697SA thru 713SA
2 Aircraft S/N 714SA and Subsequent

TRIM AILERON POSITION INDICATION

Aircraft S/N 661SA, 697SA - 730SA

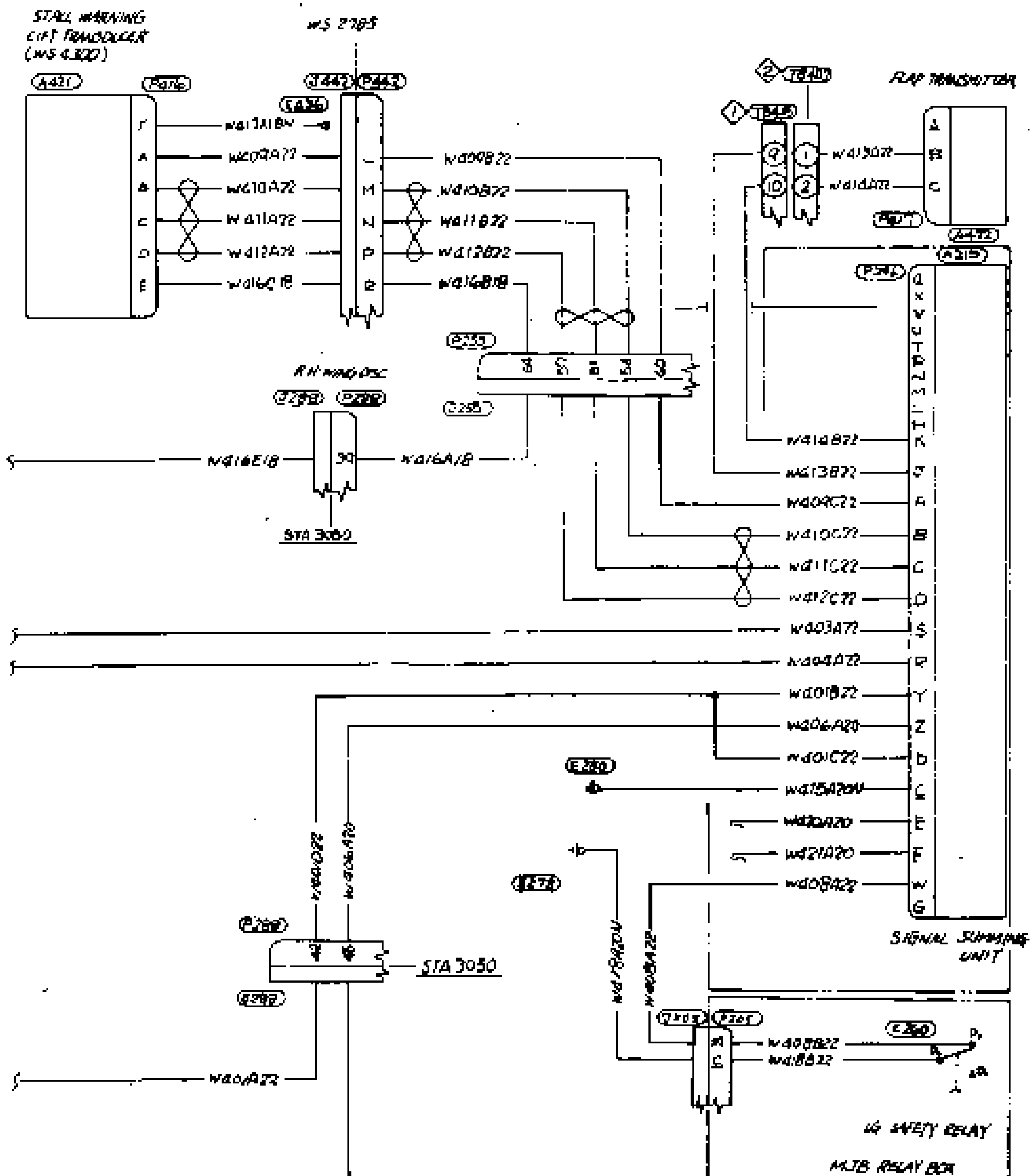


① AIRCRAFT S/N 661SA, 697SA THRU 713SA

② AIRCRAFT S/N 714SA THRU 730SA

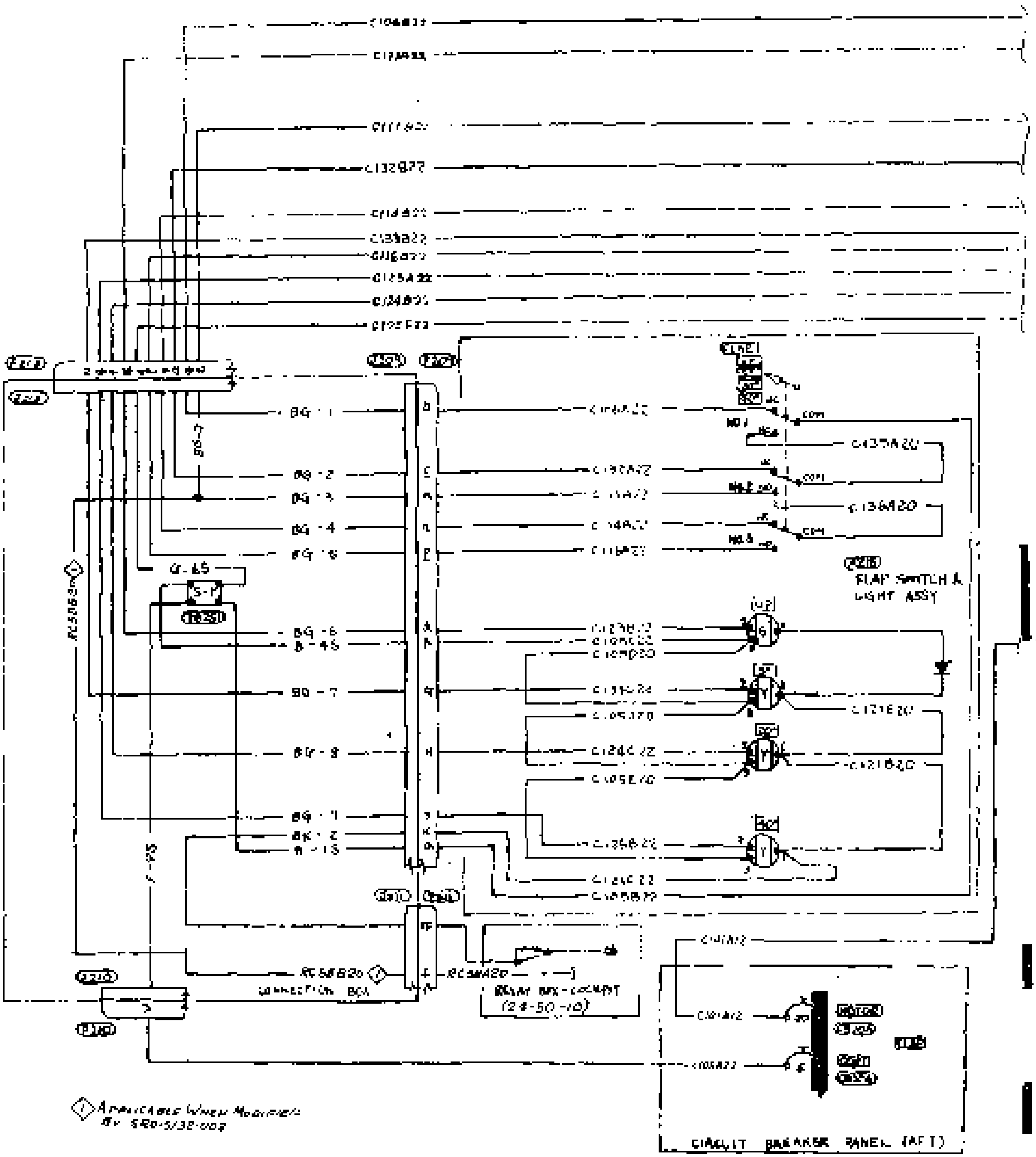
STALL WARNING SYSTEM

Aircraft S/N 661SA, 697SA - 730SA



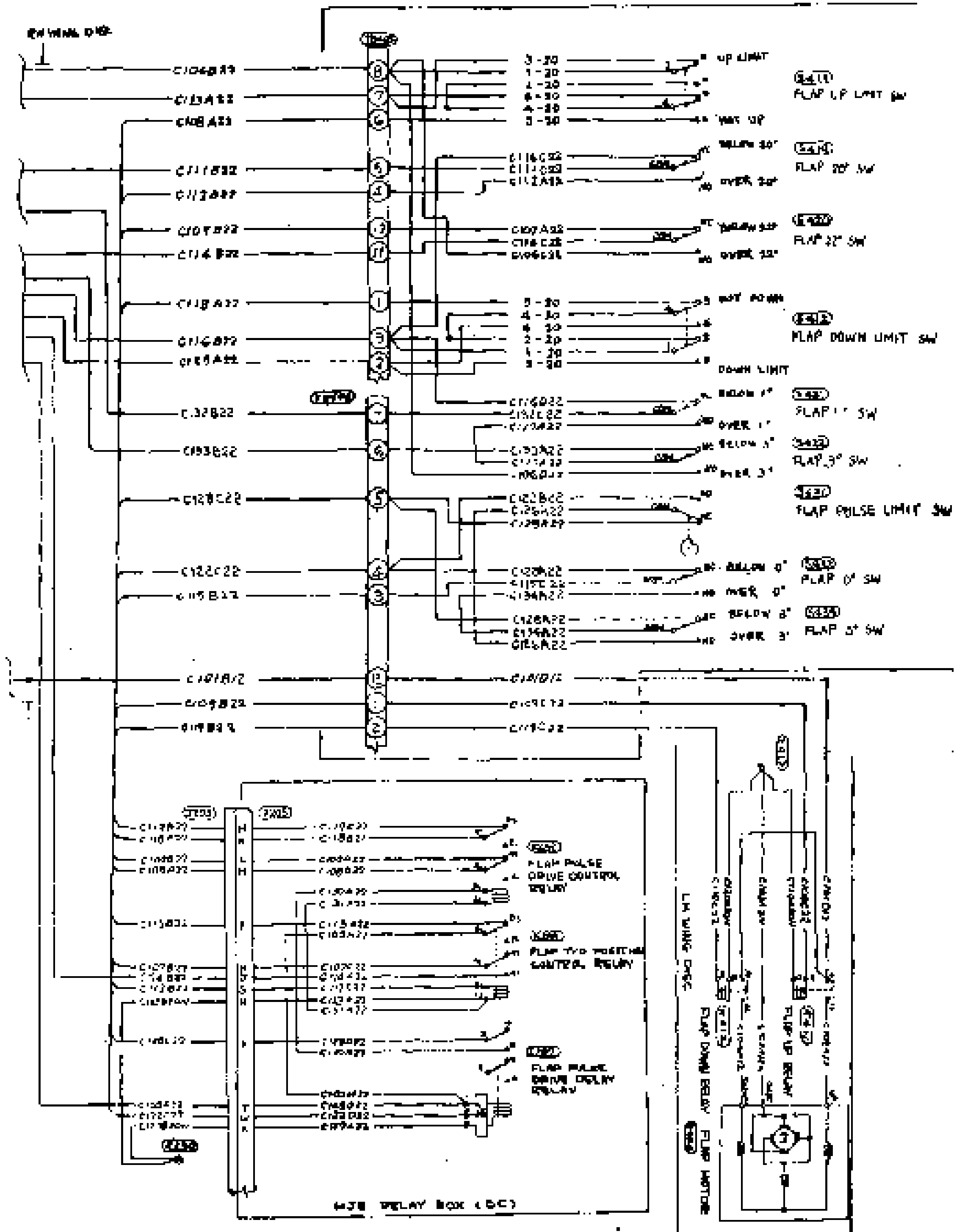
① AIRCRAFT S/N 661SA, 697SA THRU 713SA

② AIRCRAFT S/N 714SA THRU 730SA



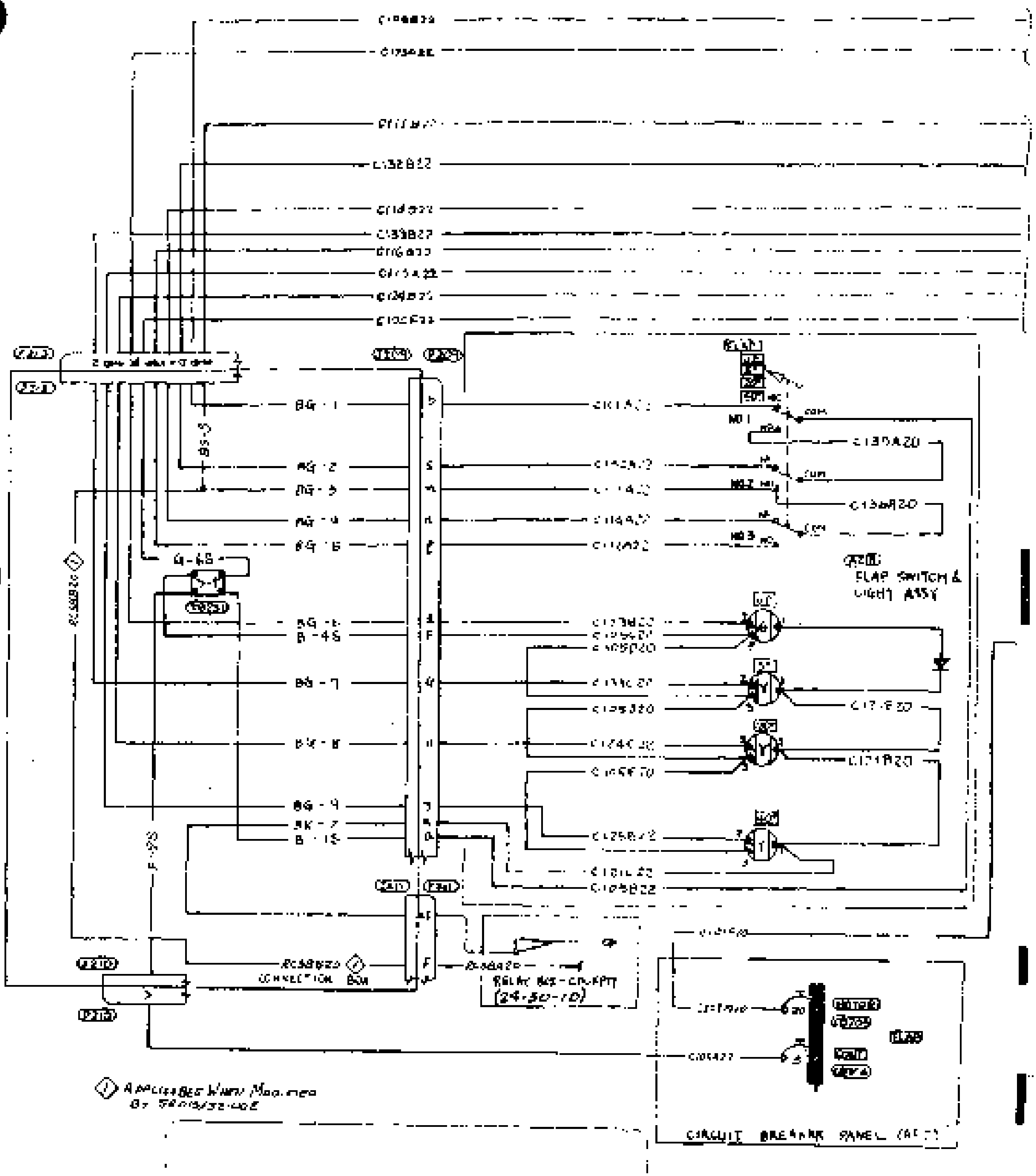
FLAP CONTROL

Aircraft S/N 661SA



FLAP CONTROL

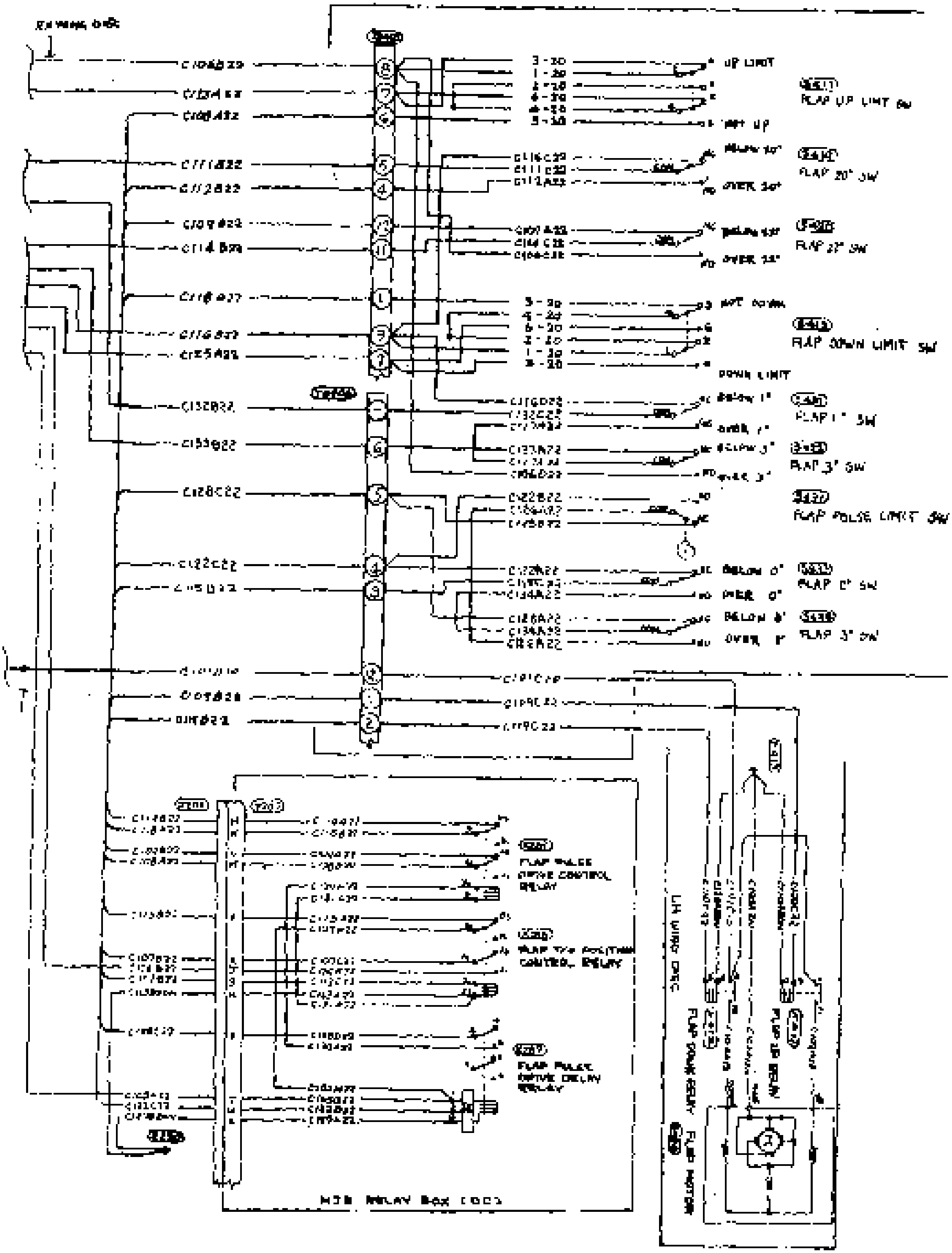
Aircraft S/N 661SA



FLAP CONTROL

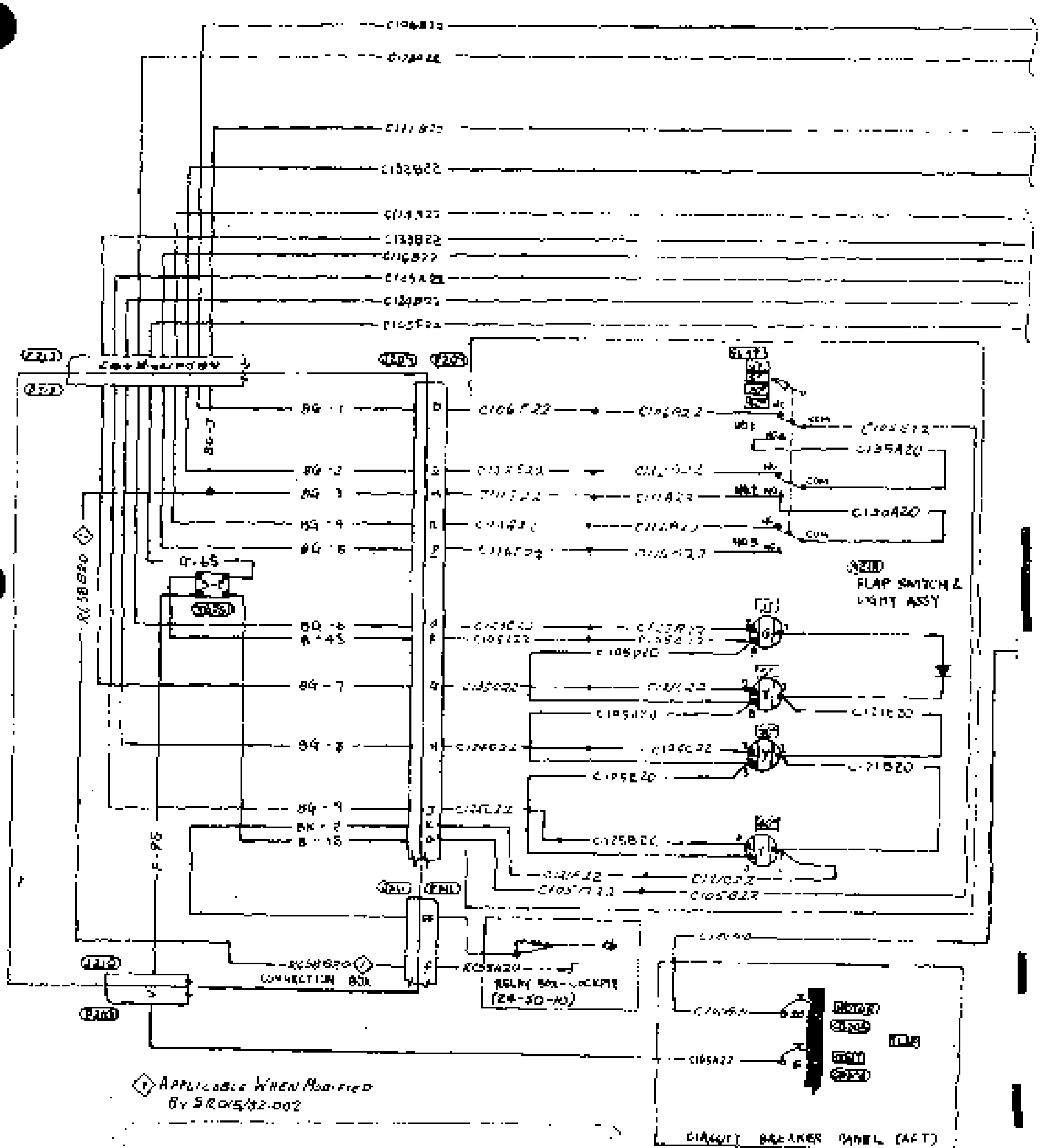
Aircraft S/N 697SA - 700SA

Rev No 1
Aug 1/79



FLAP CONTROL

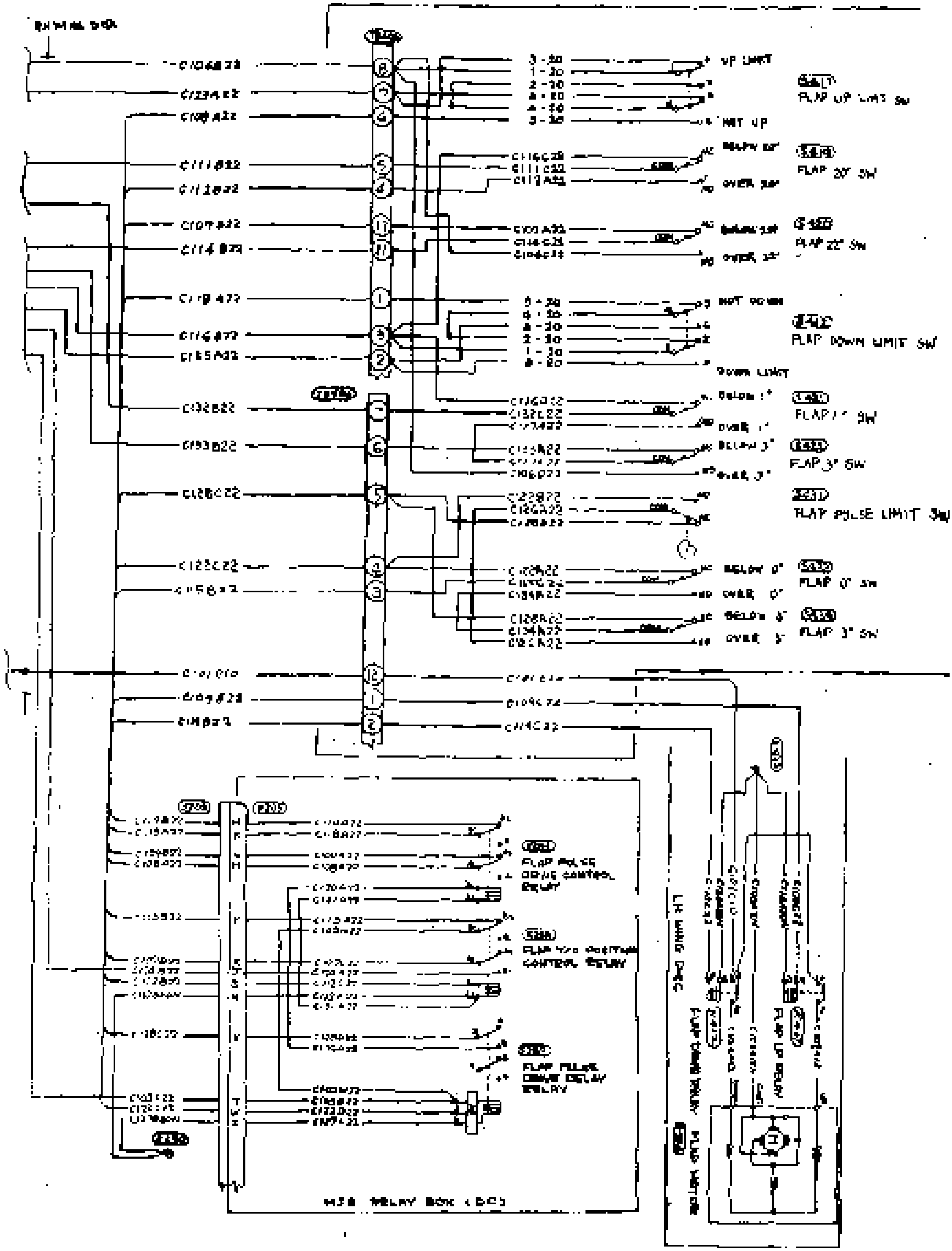
Aircraft S/N 697SA - 700SA



1 APPLICABLE WHEN MODIFIED BY SB 05/82-002

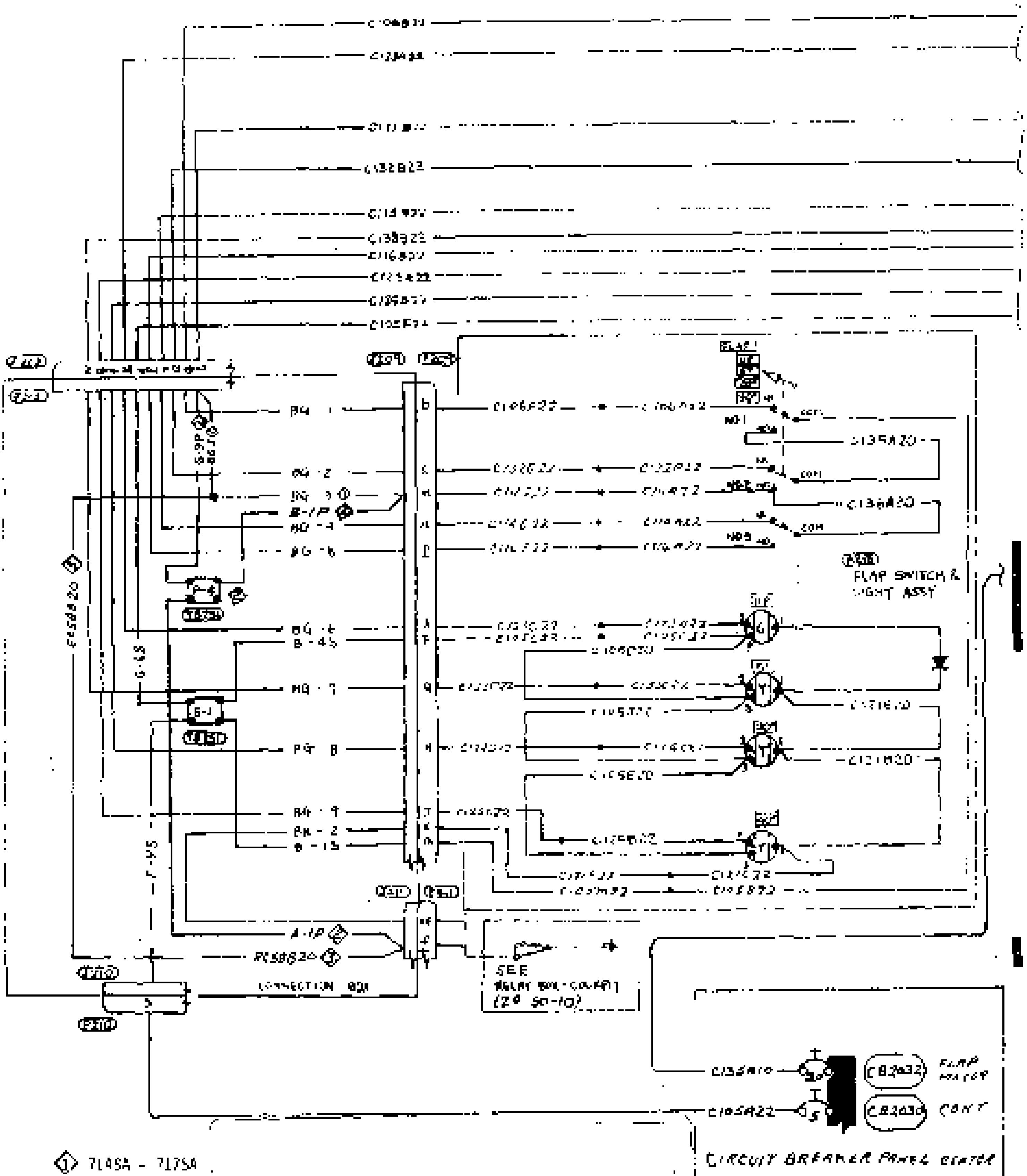
FLAP CONTROL

Aircraft S/N 7015A - 7135A



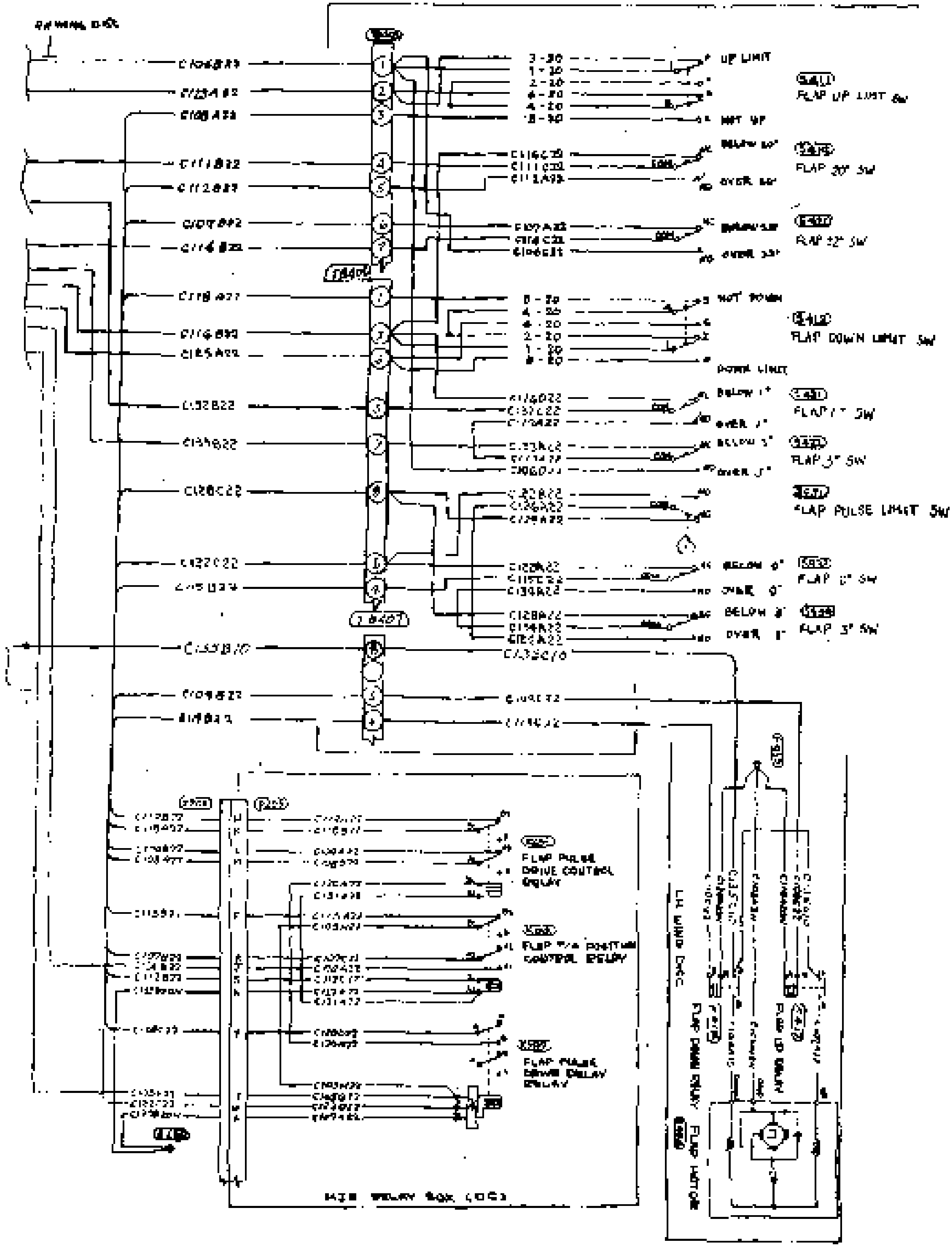
FLAP CONTROL

Aircraft S/N 701SA - 713SA



<1> 714SA - 717SA
<2> 718SA AND SUBSEQUENT
<3> APPLICABLE WHEN MODIFIED BY SB 2015/27-06

FLAP CONTROL
Aircraft S/N 714SA - 730SA



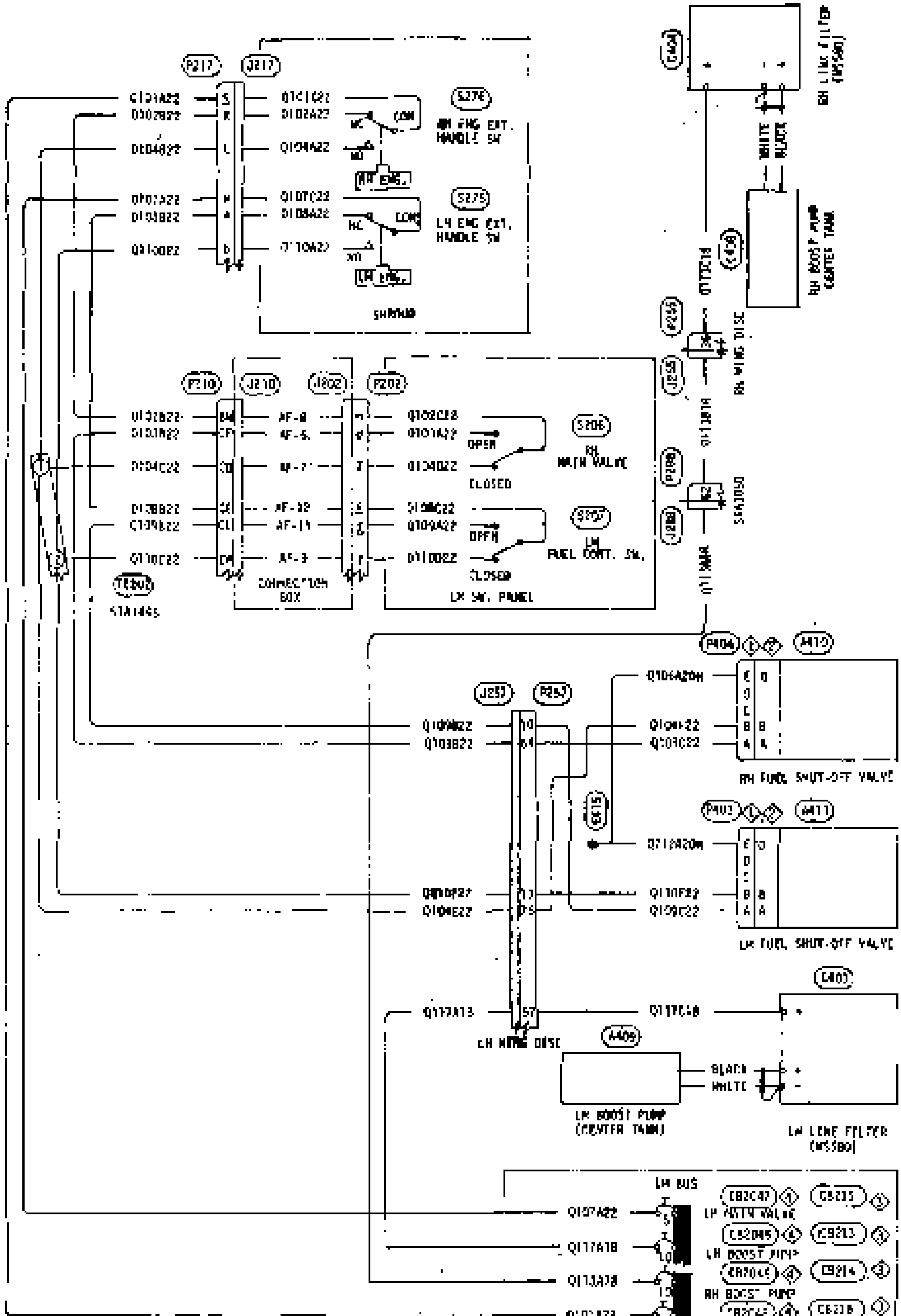
FLAP CONTROL

Aircraft S/N 714SA - 730SA

CHAPTER

28

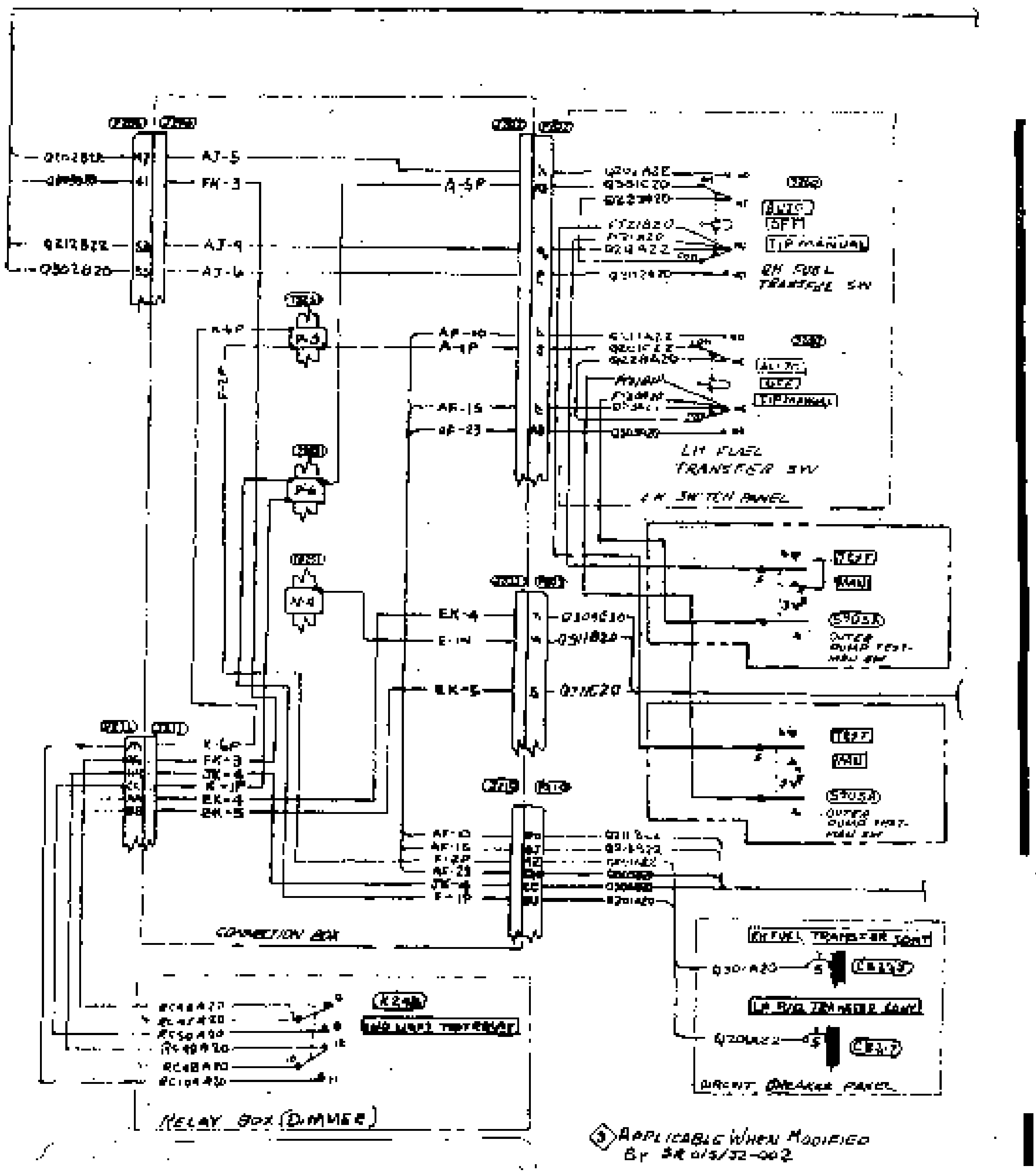
FUEL



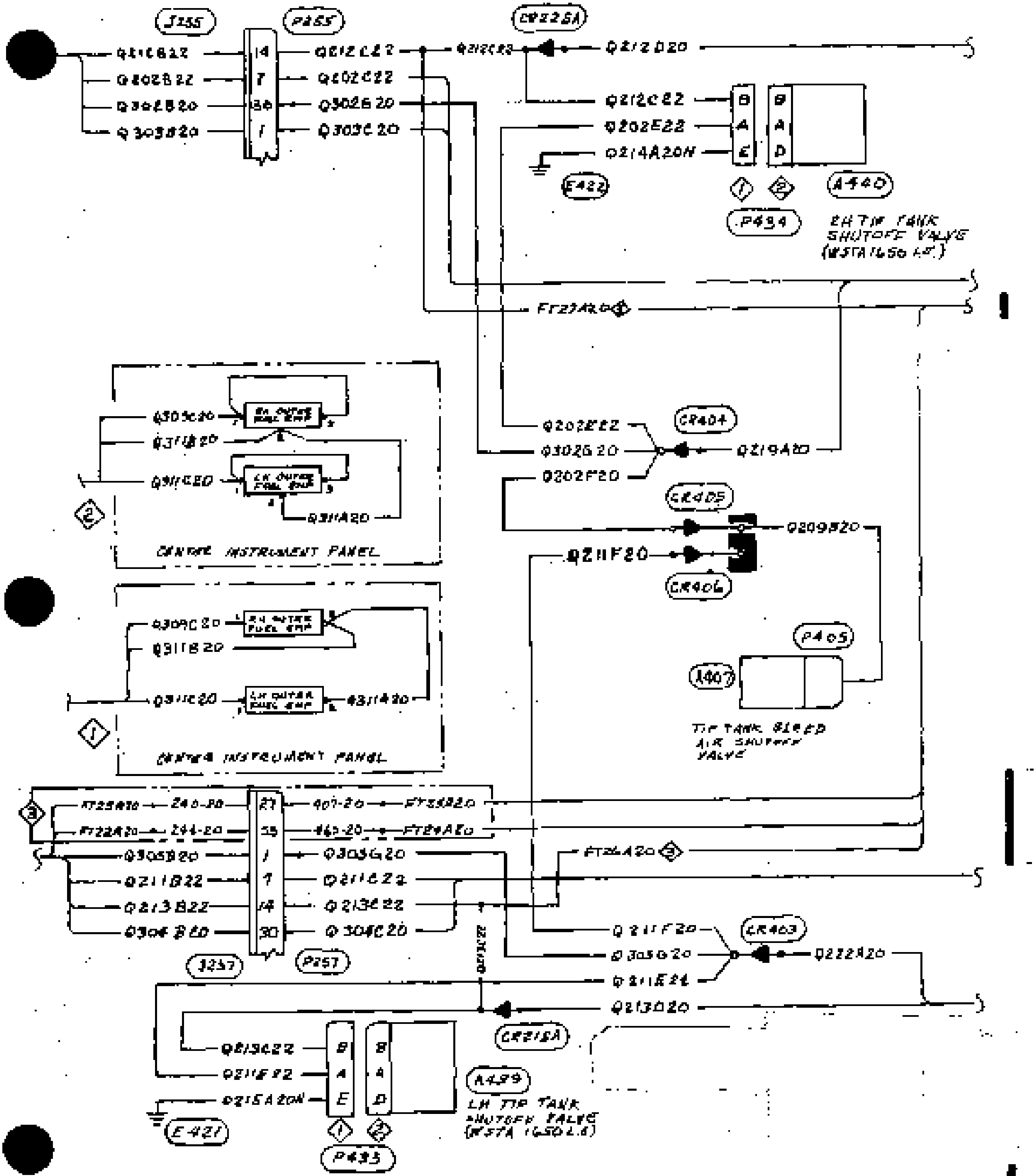
1 Aircraft S/N 661SA
2 Aircraft S/N 697SA and Subsequent
3 Aircraft S/N 661SA, 697SA thru 713SA
4 Aircraft S/N 714SA and Subsequent

MAIN TANK FUEL CONTROL

Aircraft S/N 661SA, 697SA - 730SA



* See Cockpit Relay Box Circuits (24-50-00)

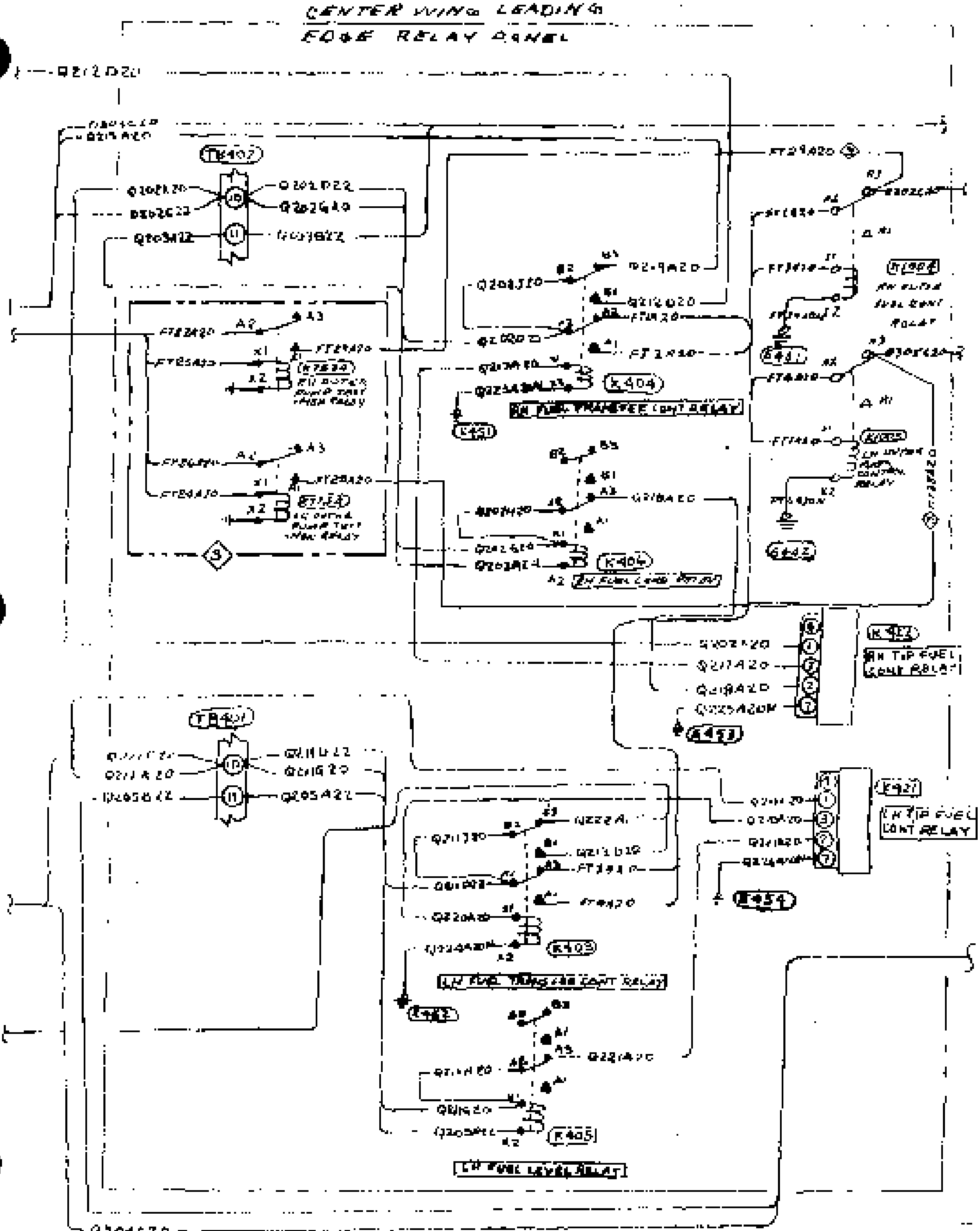


1. Aircraft S/N 661SA
2. Aircraft S/N 697SA thru 713SA
3. Applicable When Modified by SB

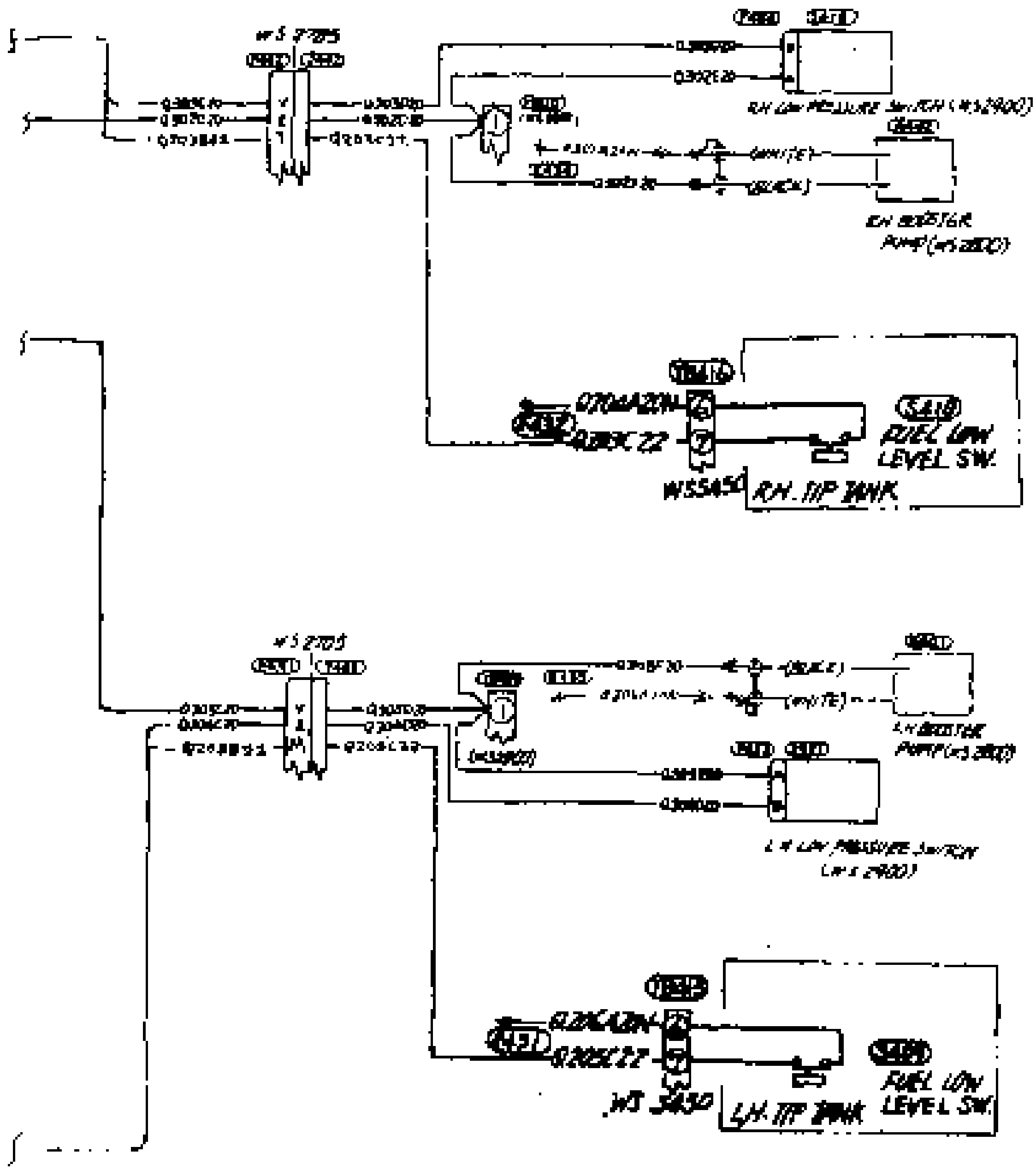
Rev No 2
Aug 1/80

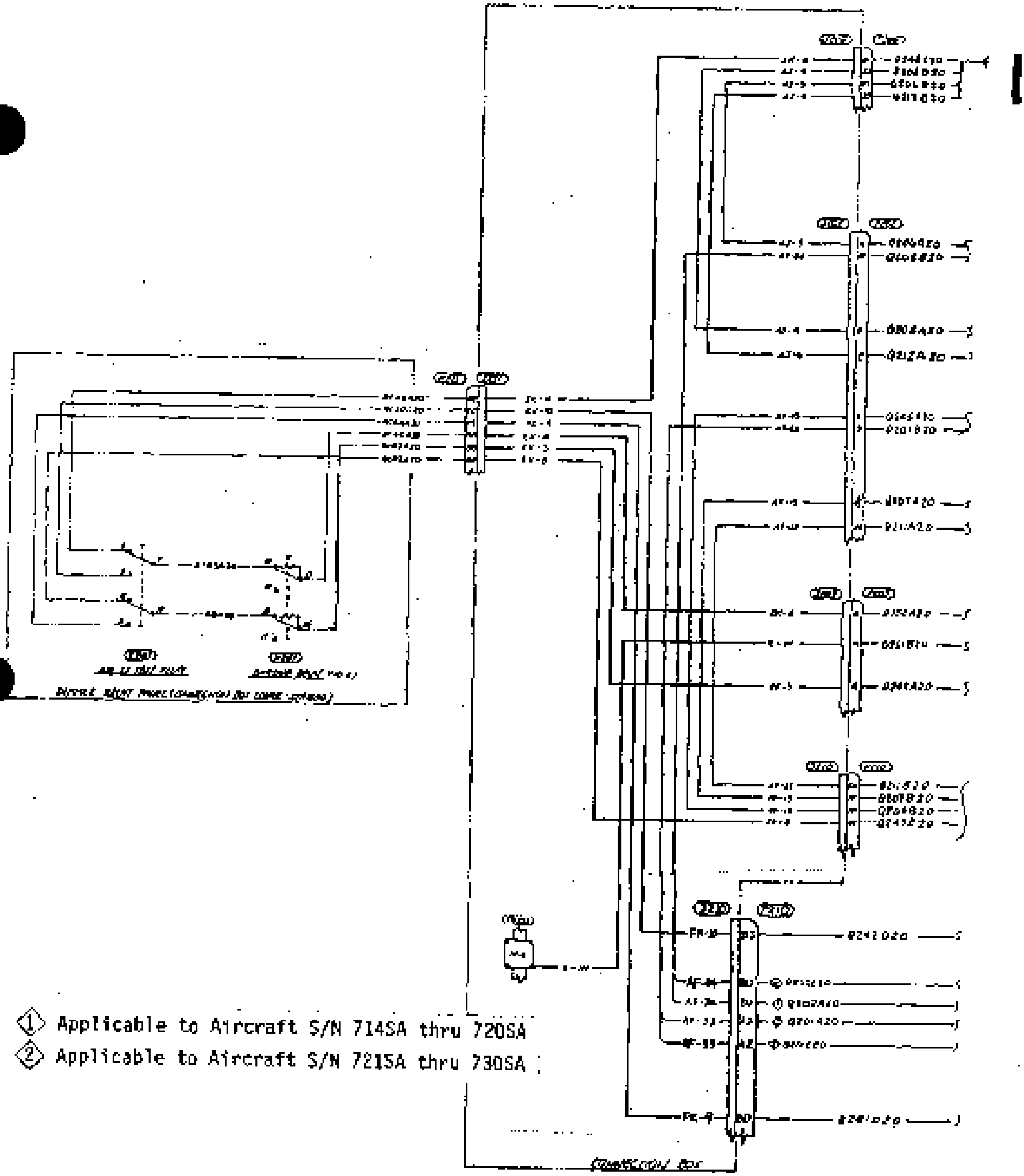
FUEL TRANSFER CONTROL
Aircraft S/N 661SA, 697SA - 713SA

SEE BLOW-UP FICHE NO. AMUG353 ITEM N

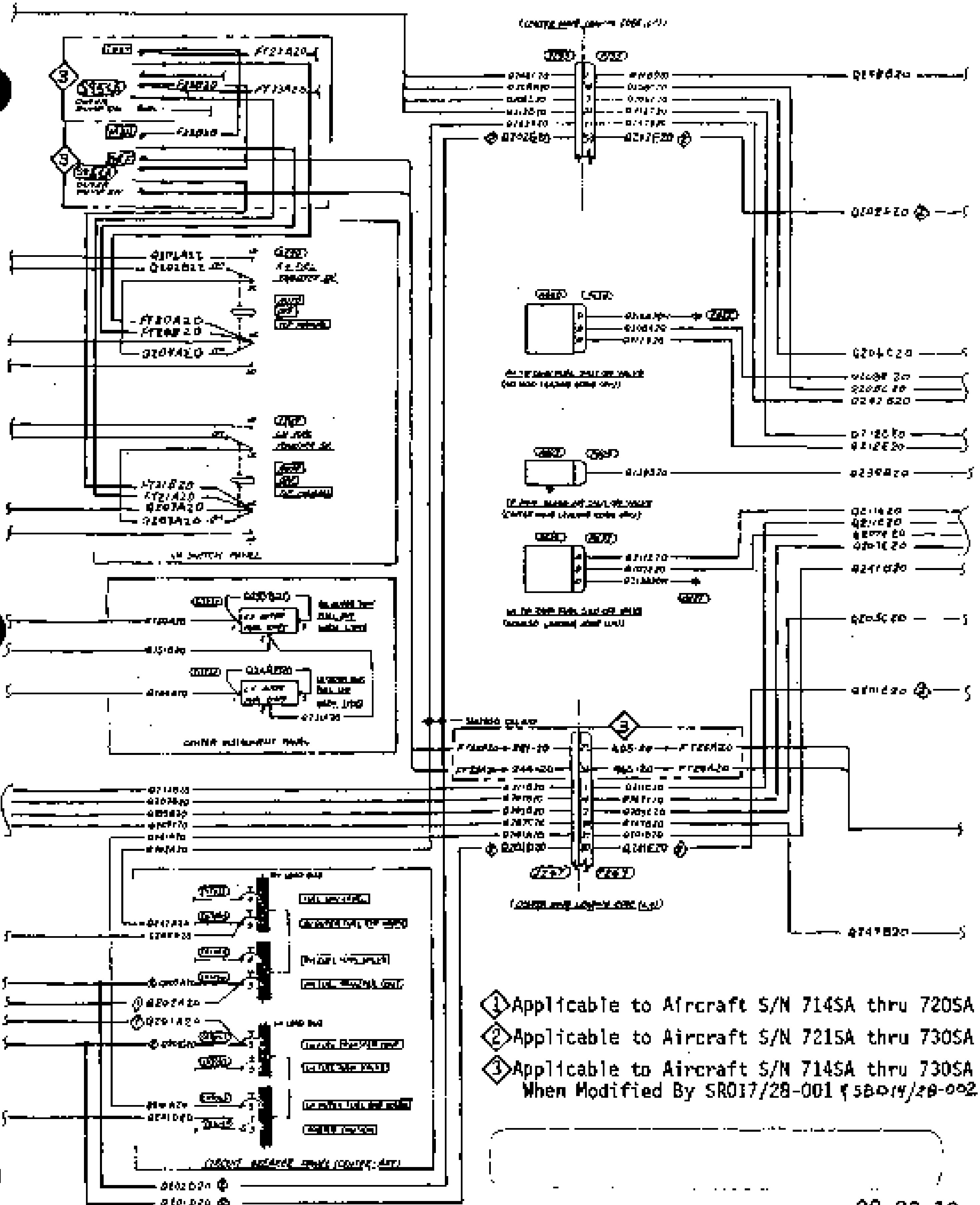


⑤ APPLICABLE WHEN MODIFIED BY SK015/32-002


RH LOW PRESSURE SWITCH
RH BOOSTER PUMP
FUEL LOW LEVEL SW.
R.H. TIP TANK
LH BOOSTER PUMP
LH LOW PRESSURE SWITCH
FUEL LOW LEVEL SW.
L.H. TIP TANK



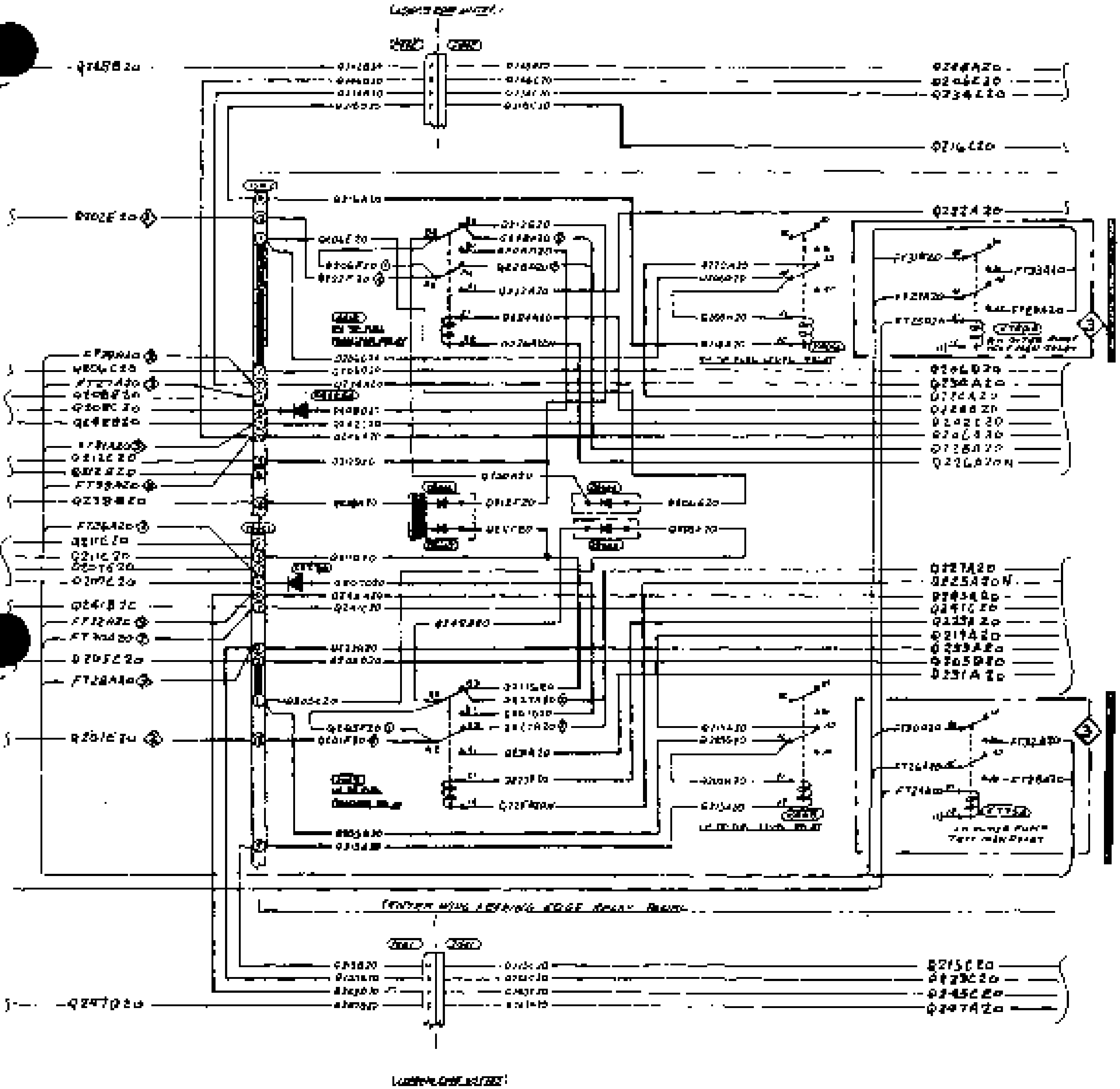
<1> Applicable to Aircraft S/N 714SA thru 720SA
<2> Applicable to Aircraft S/N 721SA thru 730SA



①Applicable to Aircraft S/N 714SA thru 720SA

②Applicable to Aircraft S/N 721SA thru 730SA

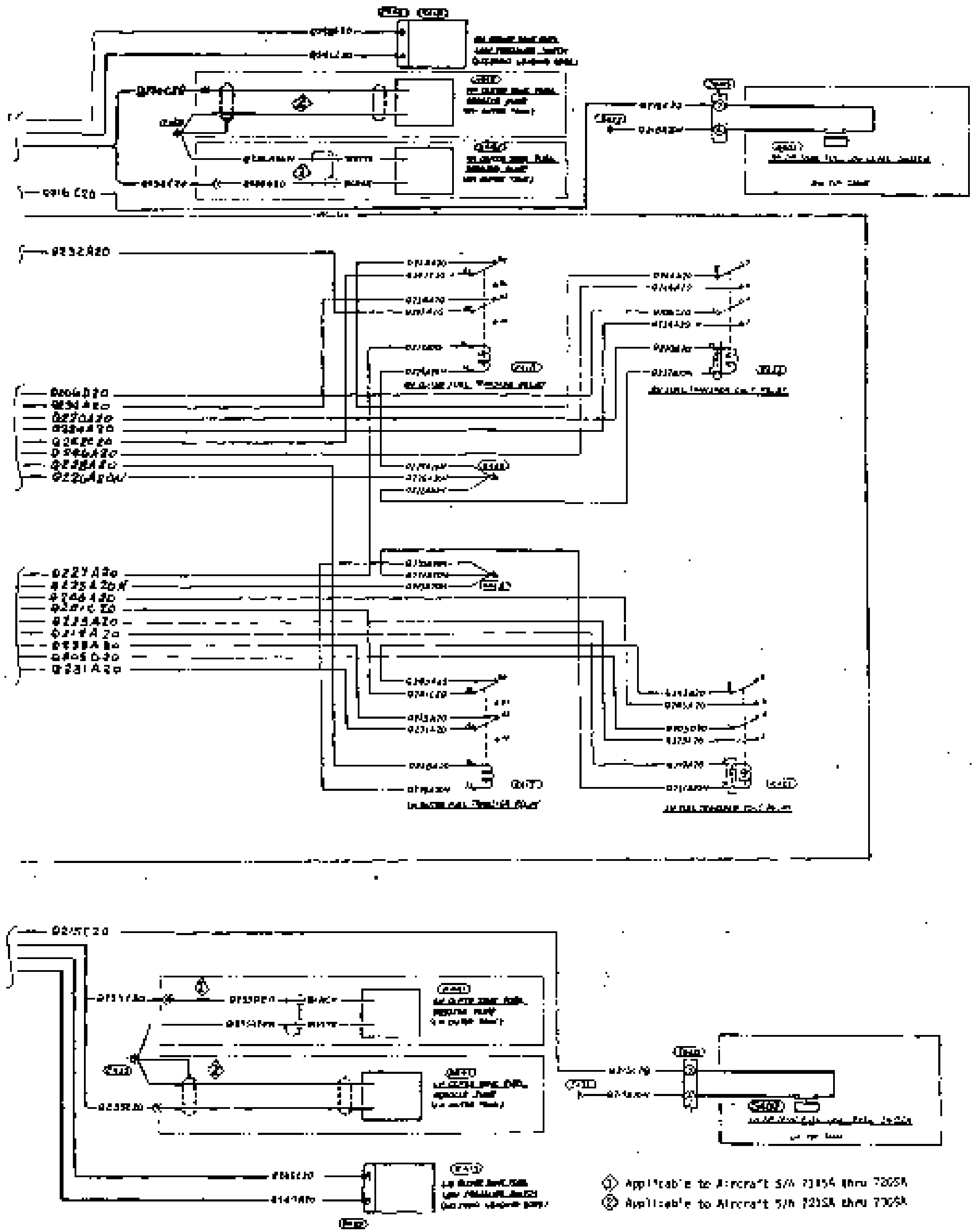
③Applicable to Aircraft S/N 714SA thru 730SA When Modified By SR017/28-001 & SB014/28-002



①Applicable to Aircraft S/N 714SA thru 720SA

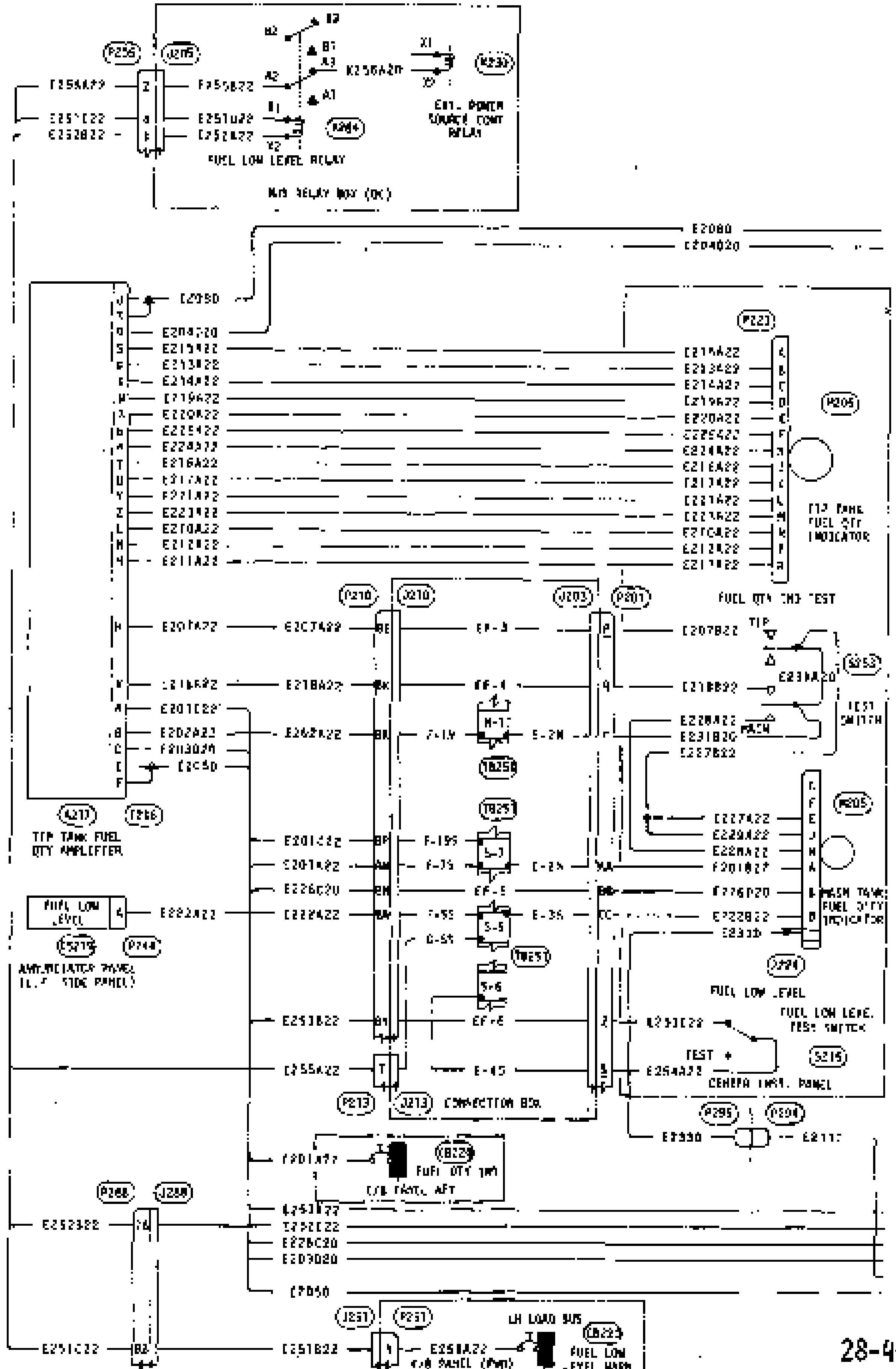
②Applicable to Aircraft S/N 721SA thru 730SA

③Applicable to Aircraft S/N 714SA thru 730SA When Modified By SR017/28-001



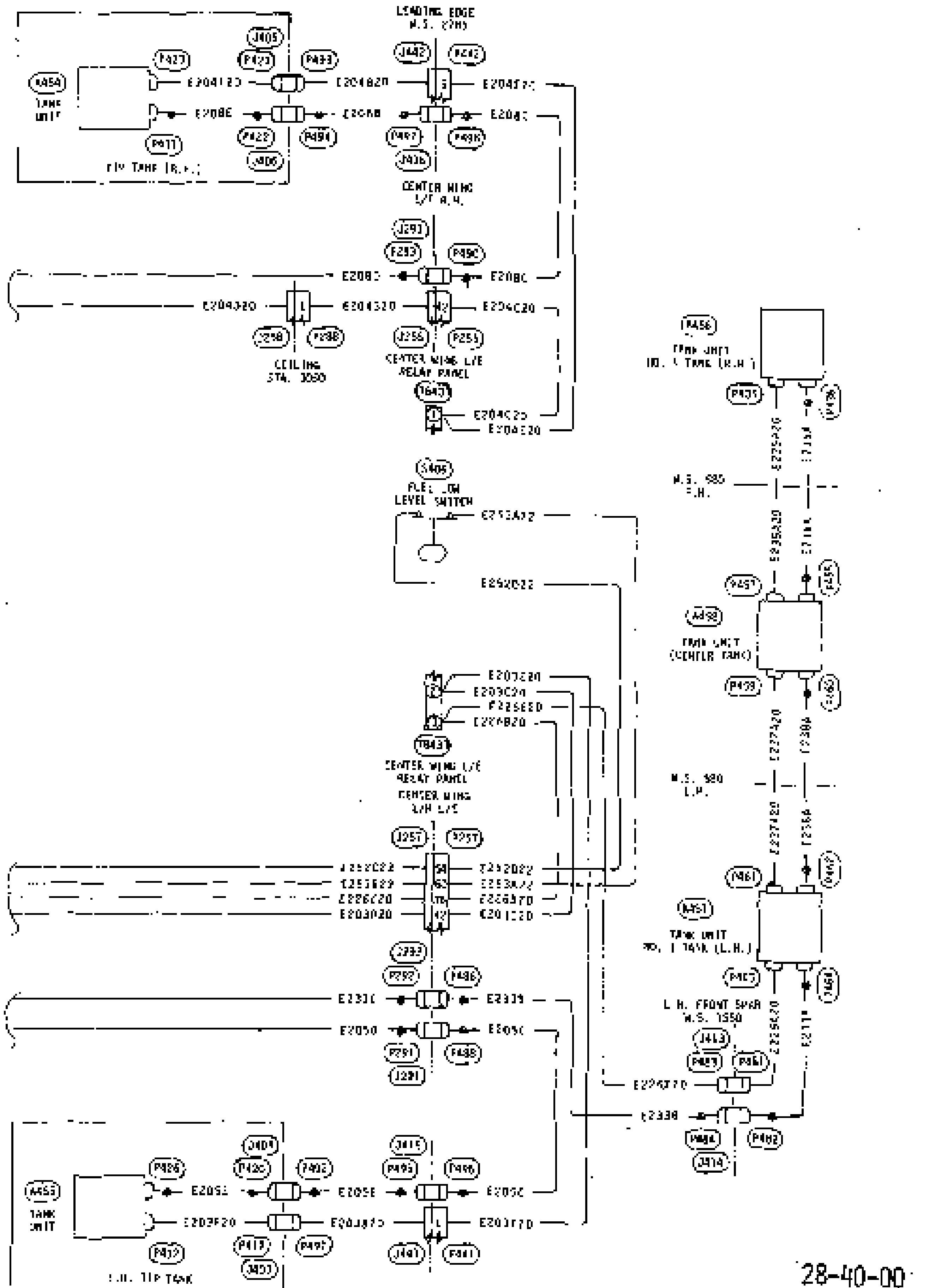
① Applicable to Aircraft S/N 714SA thru 720SA
② Applicable to Aircraft S/N 721SA thru 730SA

FUEL TRANSFER CONTROL
Aircraft S/N 714SA - 730SA

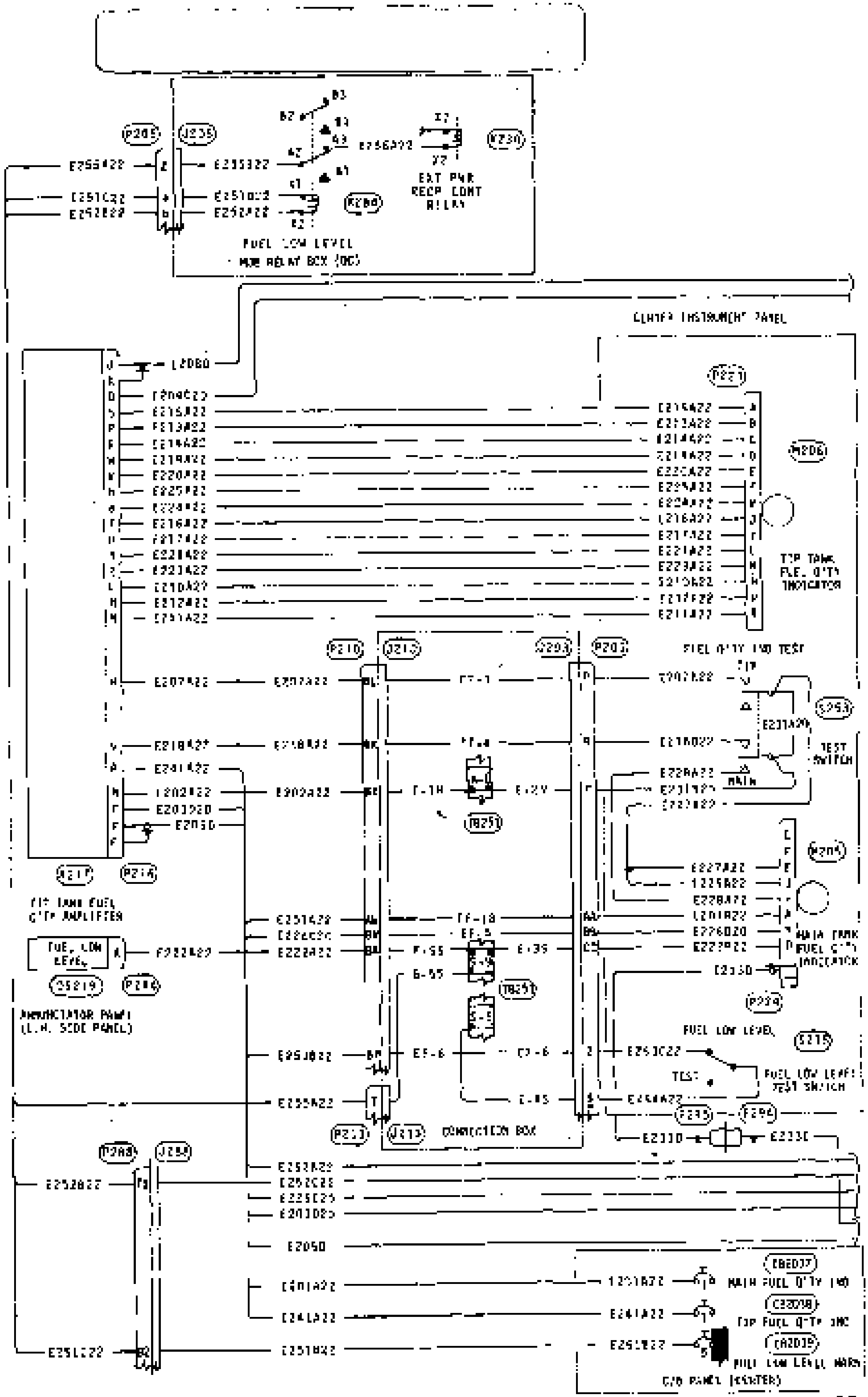


FUEL QUANTITY INDICATION

Aircraft S/N 661SA, 697SA - 713SA

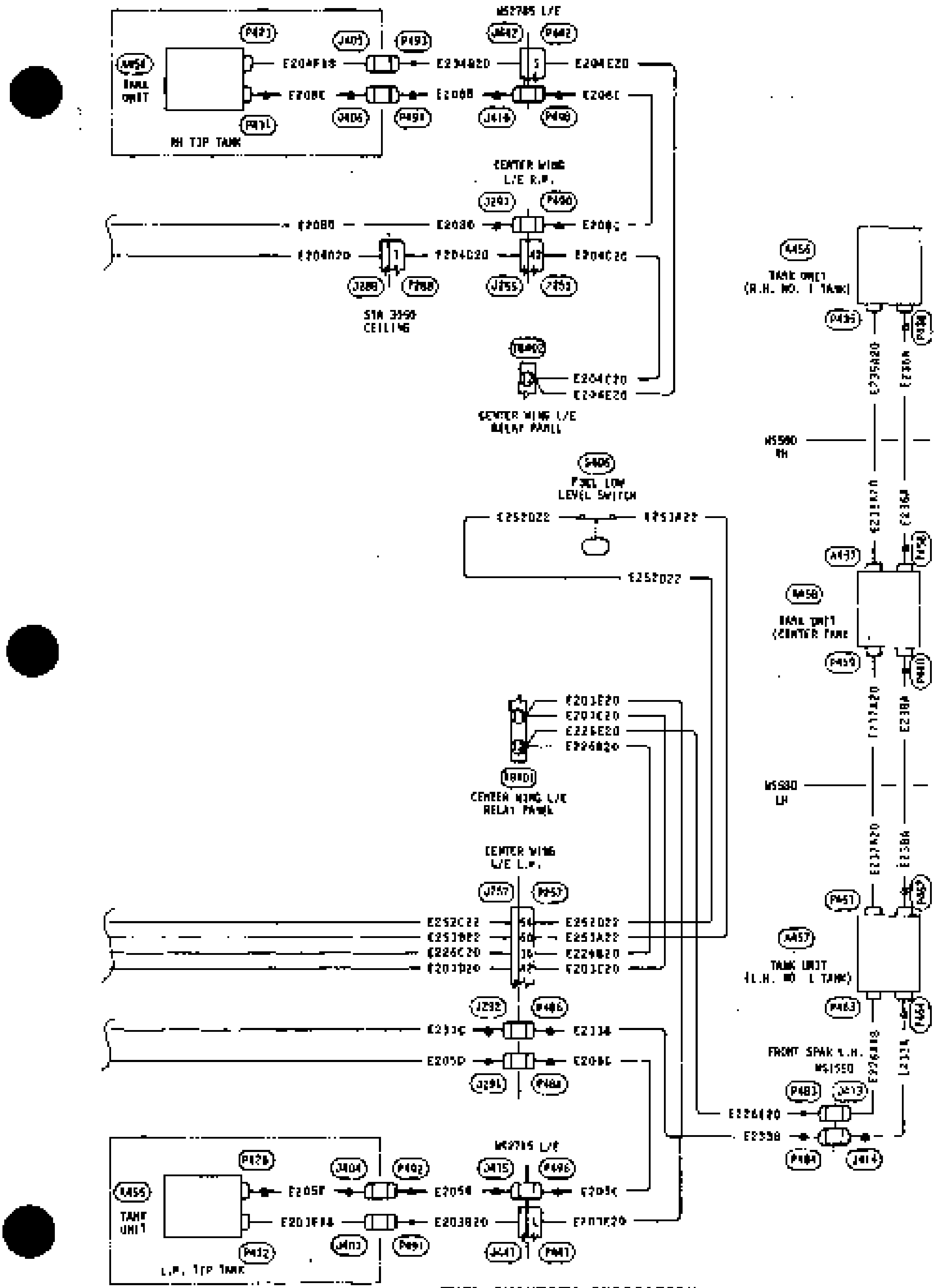


FUEL QUANTITY INDICATION
Aircraft S/N 661SA, 697SA - 713SA

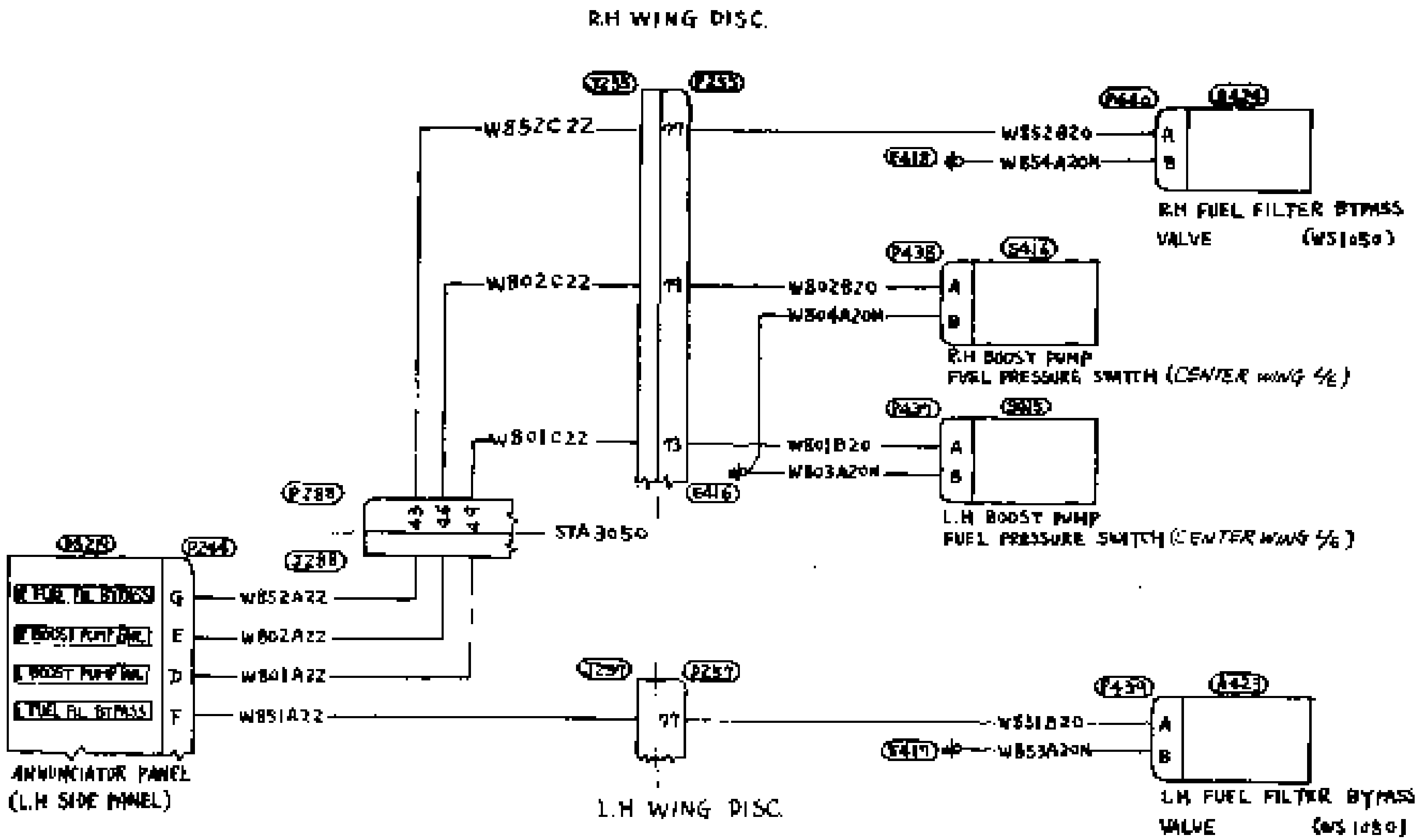


FUEL QUANTITY INDICATION

Aircraft S/N 714SA - 730SA



RH TIP TANK
TANK UNIT
CENTER WING L/E R.P.
STA 3950 CEILING
CENTER WING L/E RELAY PANEL
FUEL LOW LEVEL SWITCH
TANK UNIT (R.H. NO. 1 TANK)
TANK UNIT (CENTER TANK
TANK UNIT (L.H. NO. 1 TANK)
FRONT SPAR L.H.
CENTER WING L/E L.P.
L.H. TIP TANK



FUEL FILTER BYPASS/BOOST PUMP FAILURE
Aircraft S/N 661SA, 697SA - 730SA

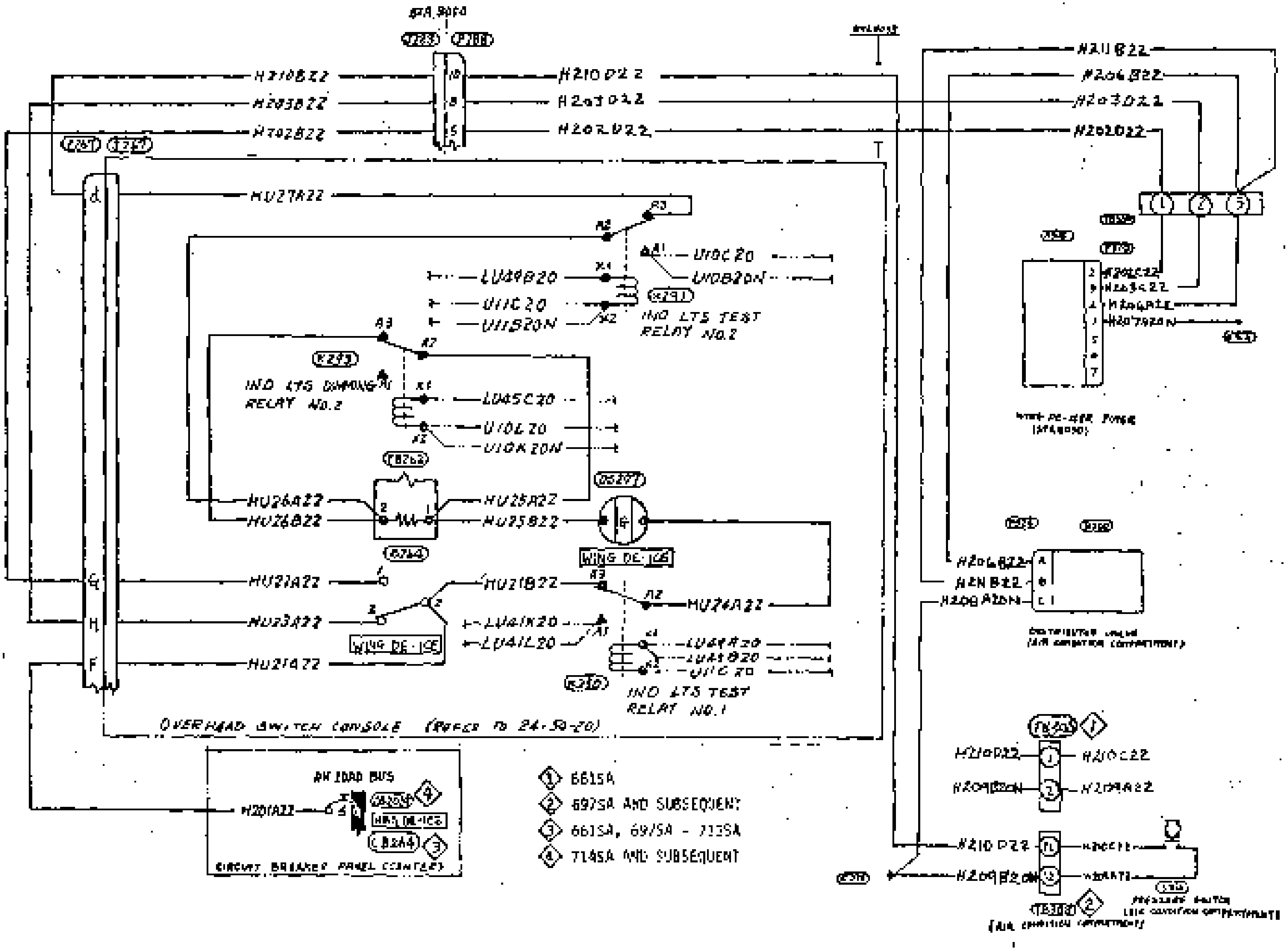
CHAPTER

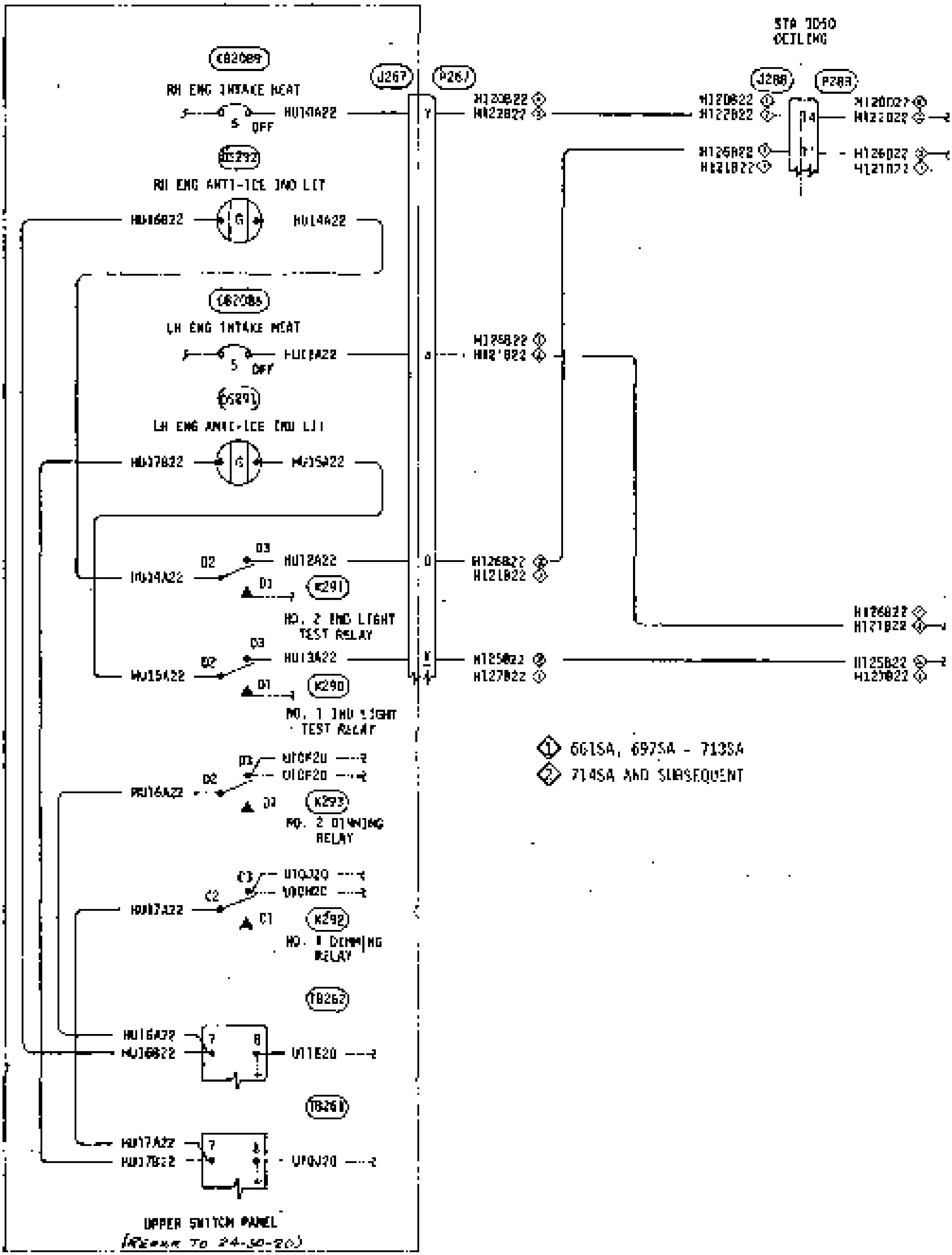
30

ICE AND RAIN PROTECTION

WING DE-ICE

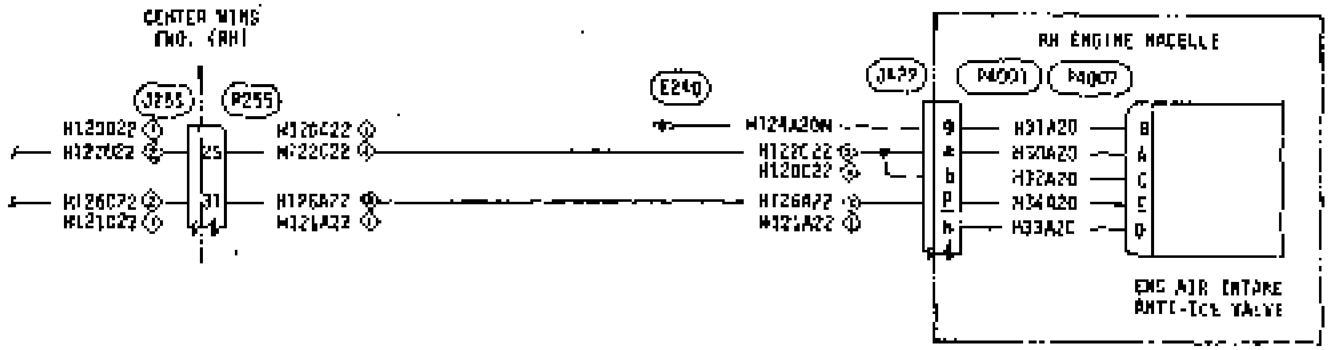
Aircraft S/N 661SA, 697SA - 730SA

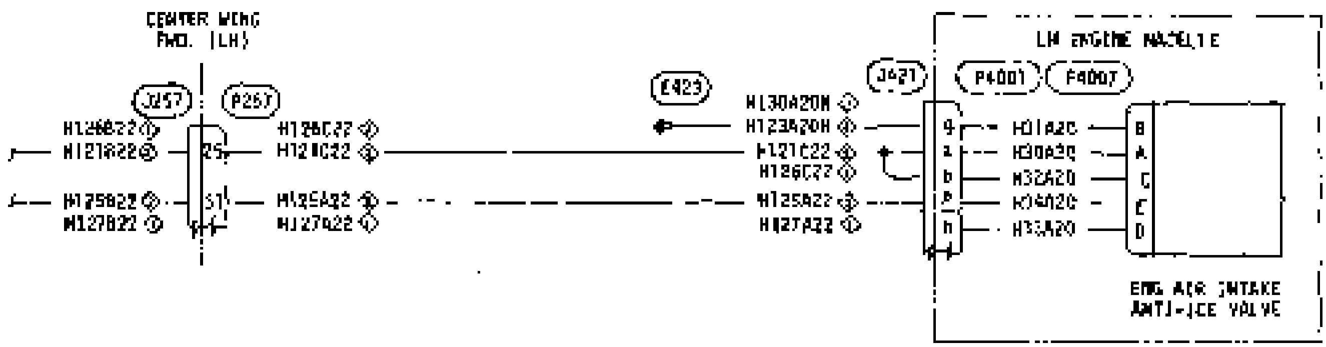




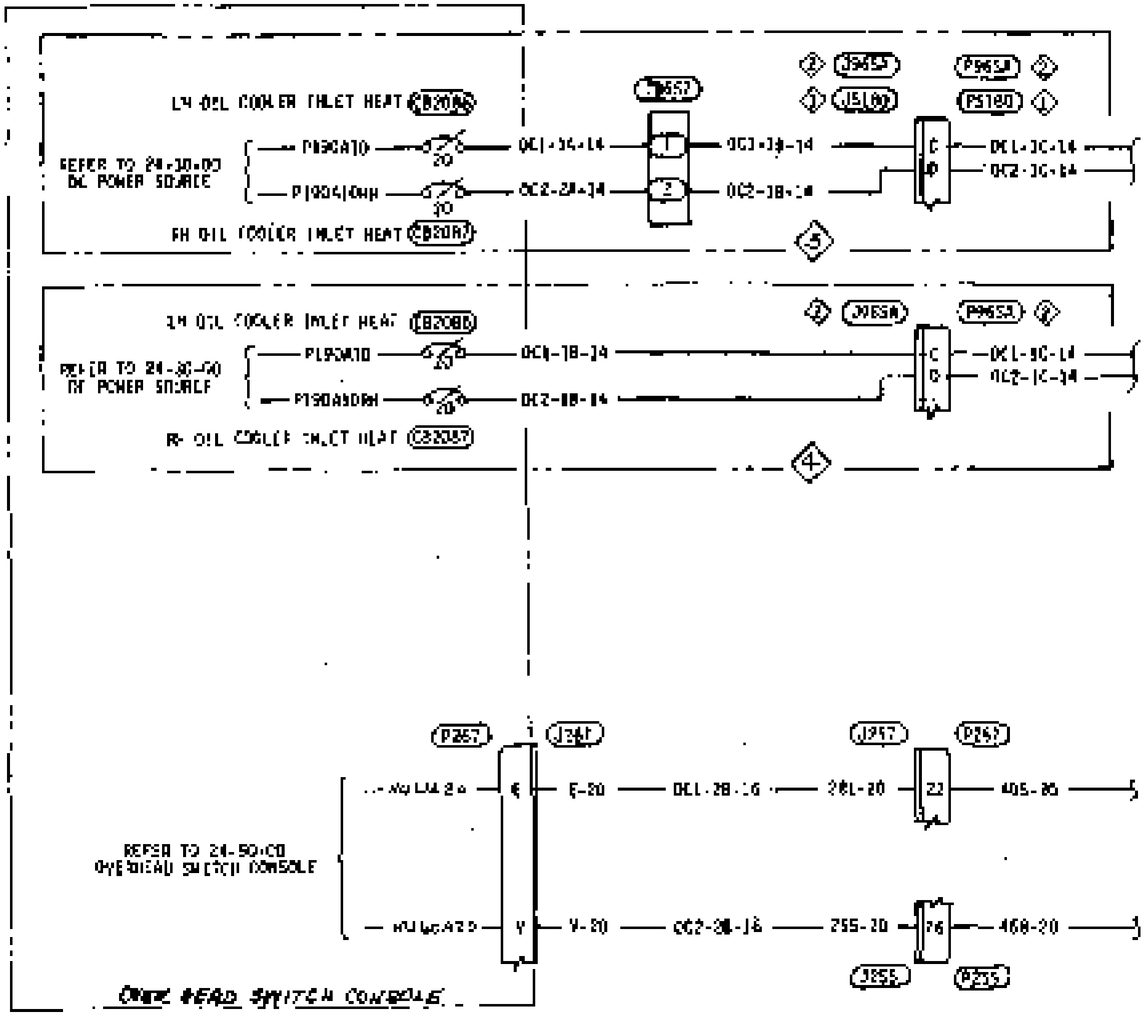
ENGINE AIR INLET ANTI-ICE

Aircraft S/N 661SA, 697SA - 730SA

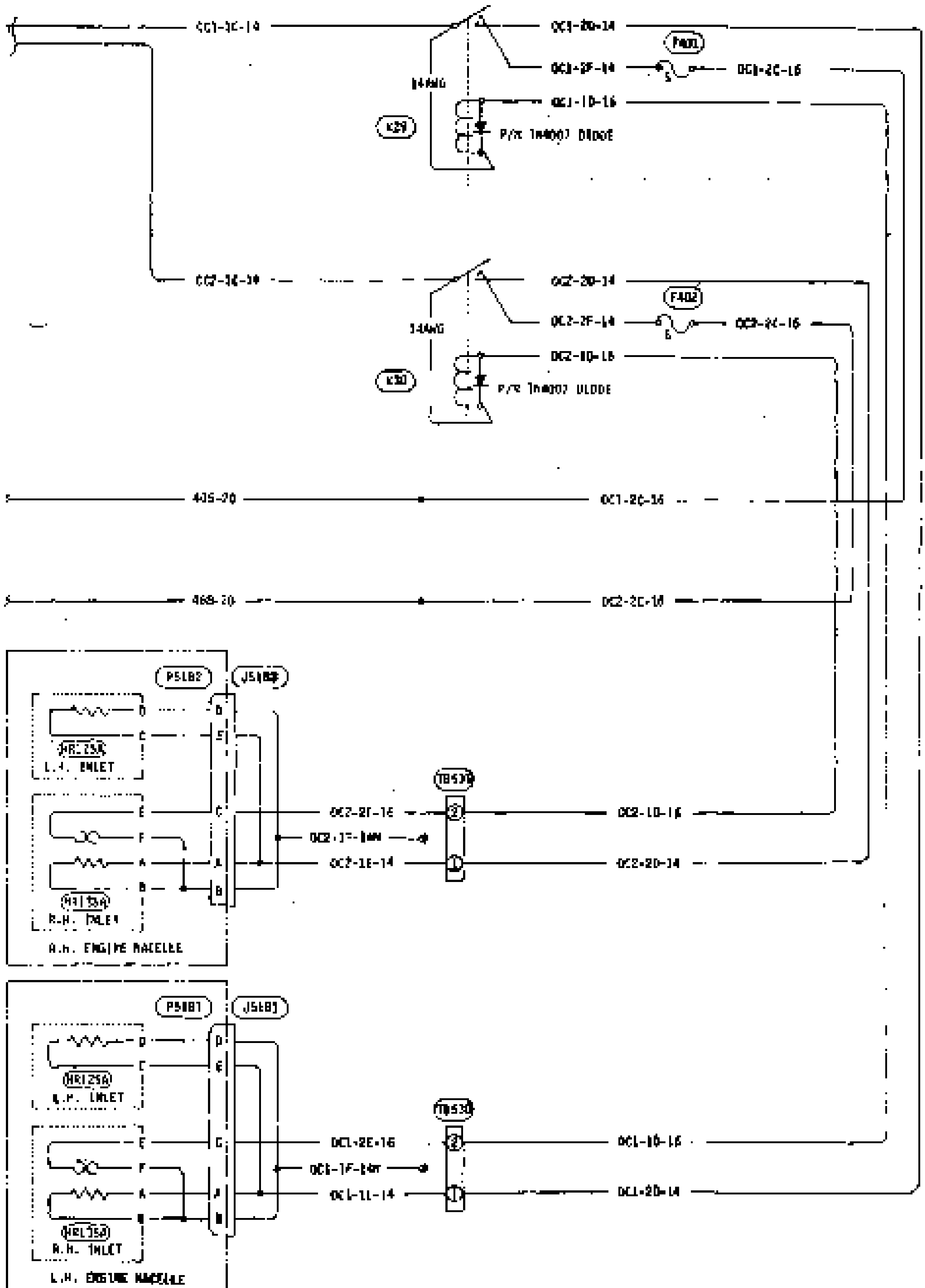




1 661SA, 697SA - 713SA
2 714SA AND SUBSEQUENT

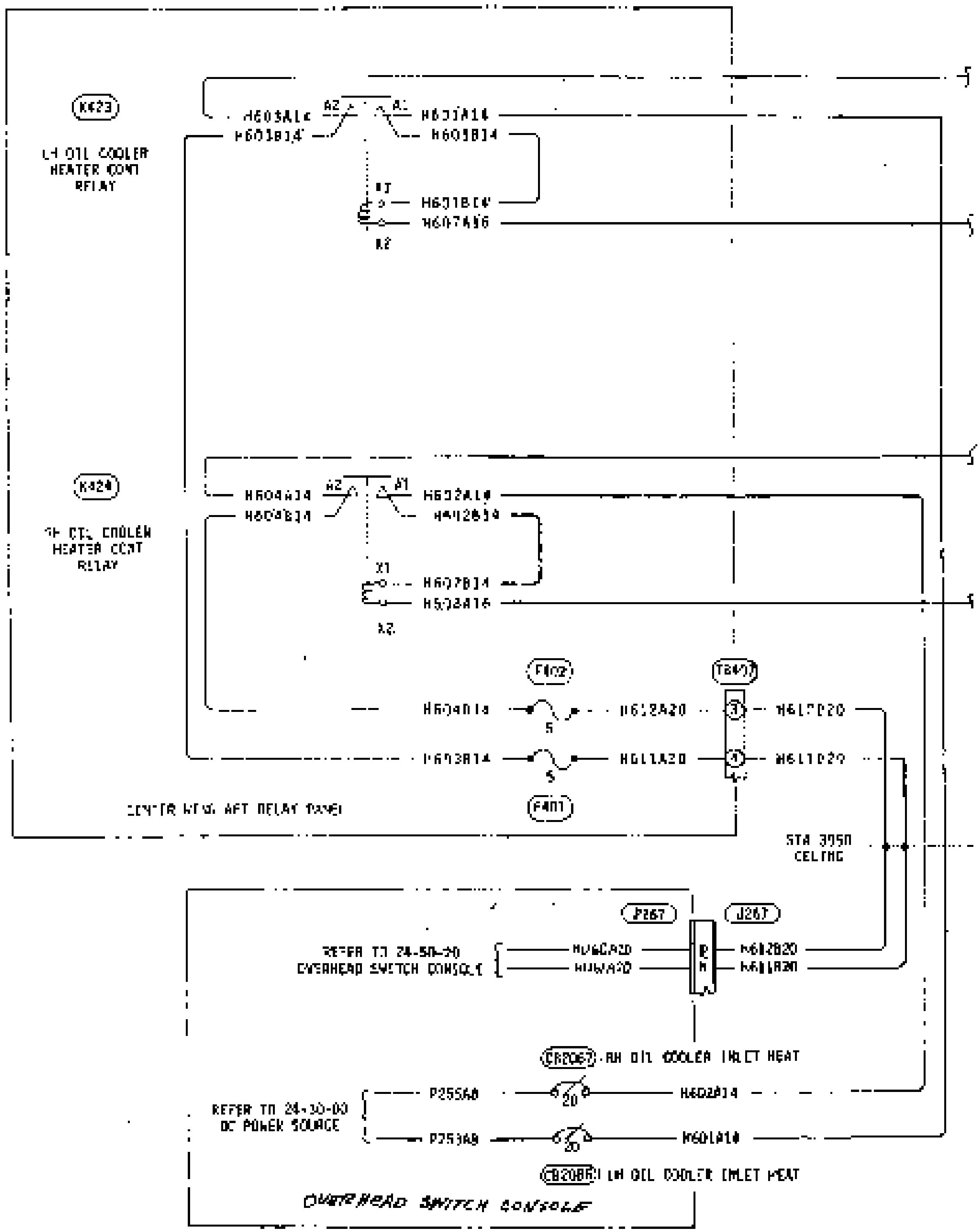


1. AIRCRAFT S/N 661SA, 697SA AND 698SA
2. AIRCRAFT S/N 699SA THRU 713SA
3. AIRCRAFT S/N 661SA, 697SA THRU 704SA
4. AIRCRAFT S/N 705SA THRU 713SA



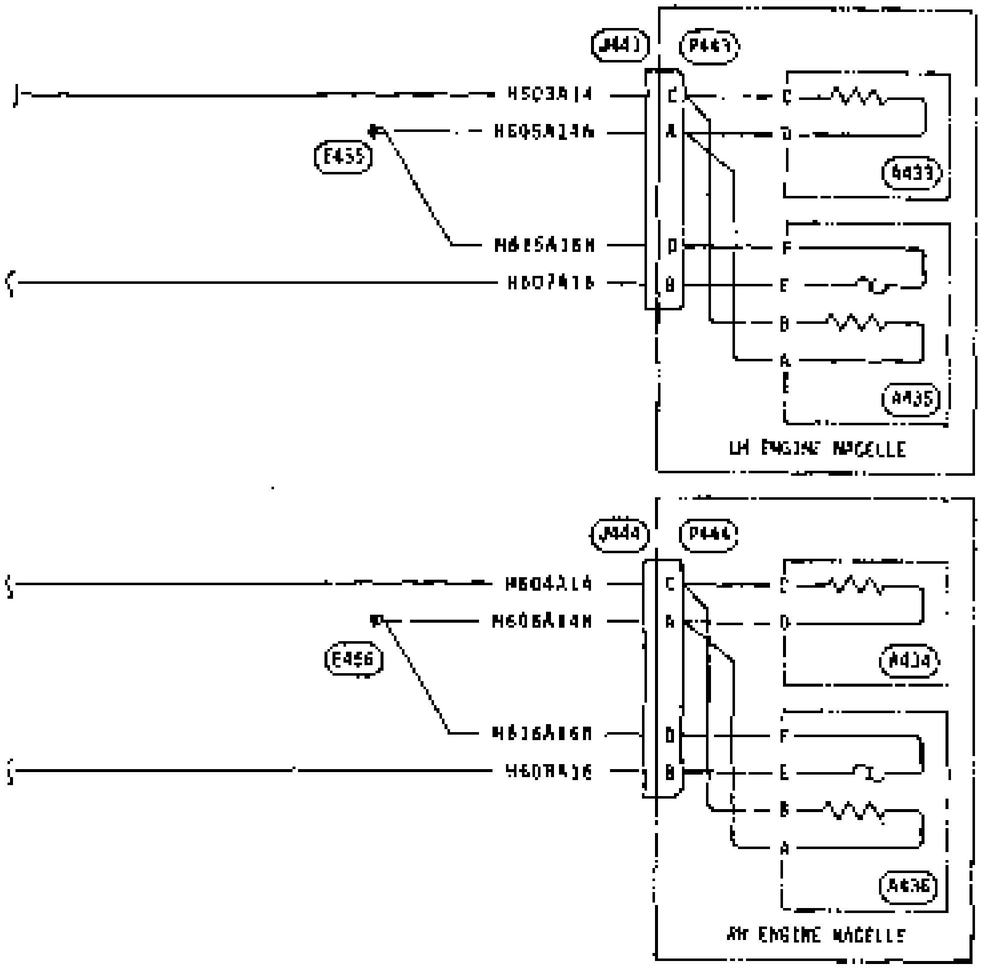
OIL COOLER INLET ANTI-ICE

Aircraft S/N 661SA, 697SA - 713SA

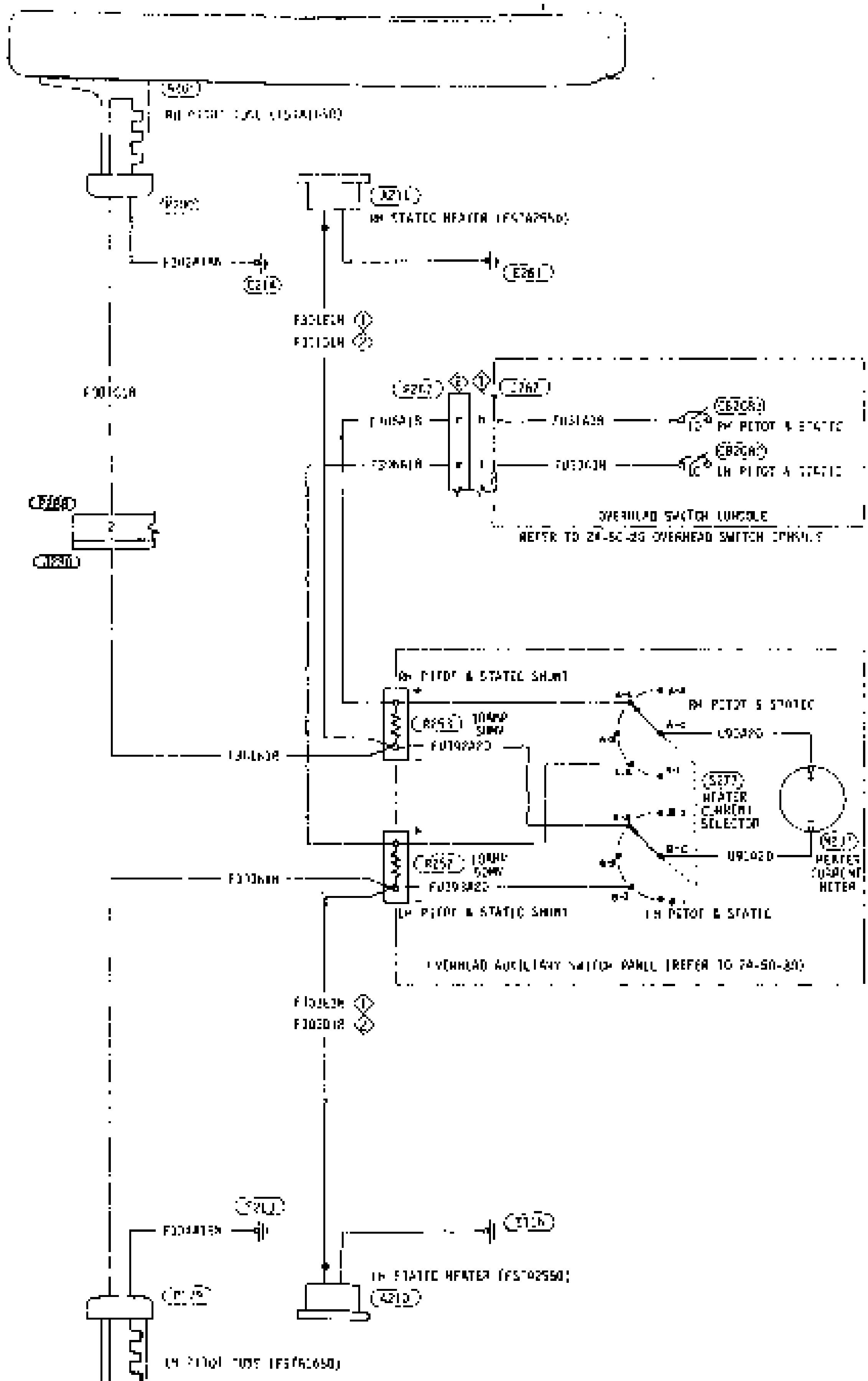


OIL COOLER INLET ANTI-ICE

Aircraft S/N 714SA - 730SA



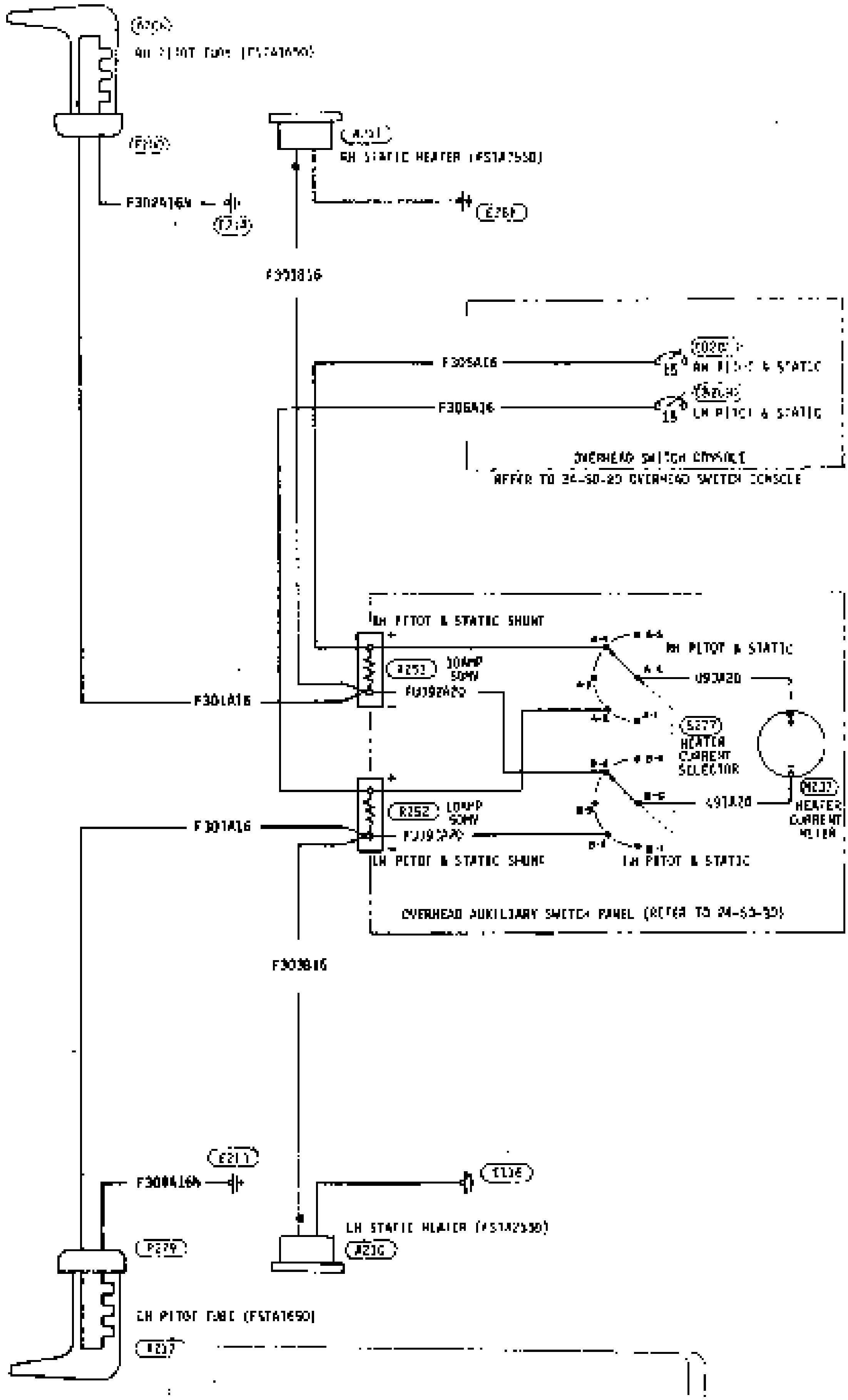
J443
P443
H503A14
E455
A433
A435
LH ENGINE NACELLE
J444
P444
H604A14
E456
A434
A436
RH ENGINE NACELLE



(1) Aircraft S/N 661SA, 697SA thru 730SA
(2) Aircraft S/N [illegible] and subsequent

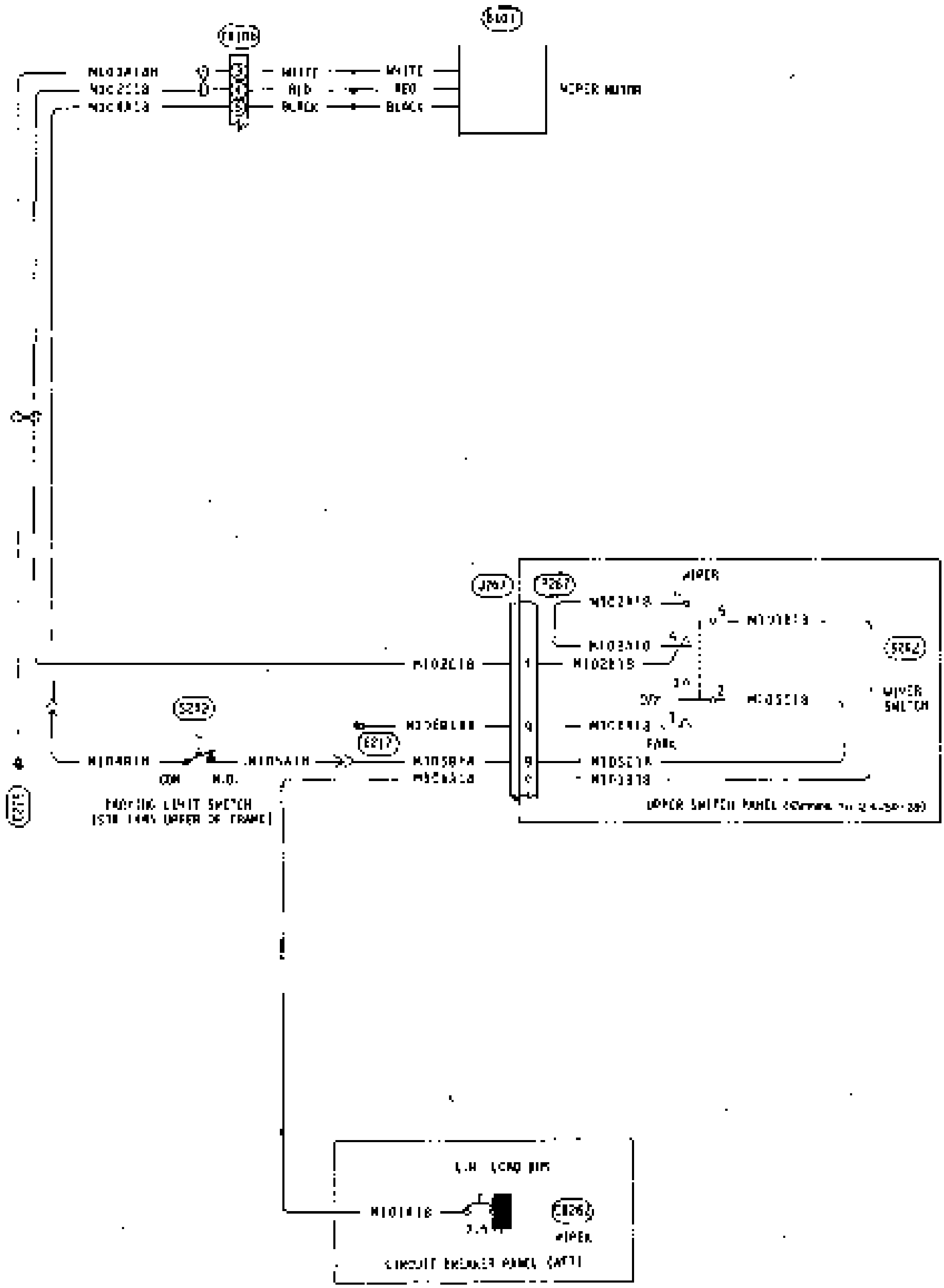
PITOT TUBE & STATIC PORT ANTI-ICE

Aircraft S/N 661SA, 697SA - 730SA
Not Modified By SRO20/34-005

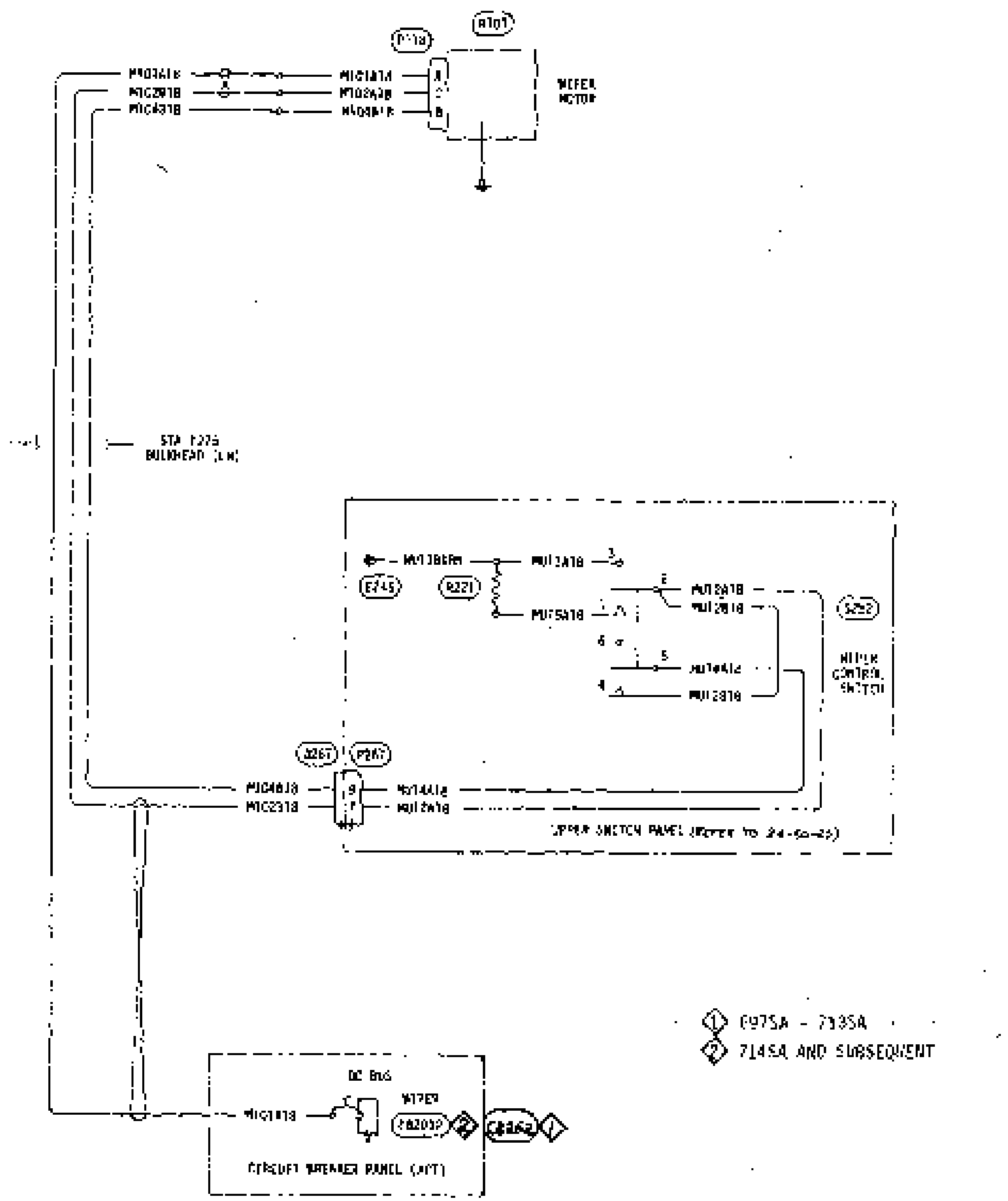


PITOT TUBE & STATIC PORT ANTI-ICE

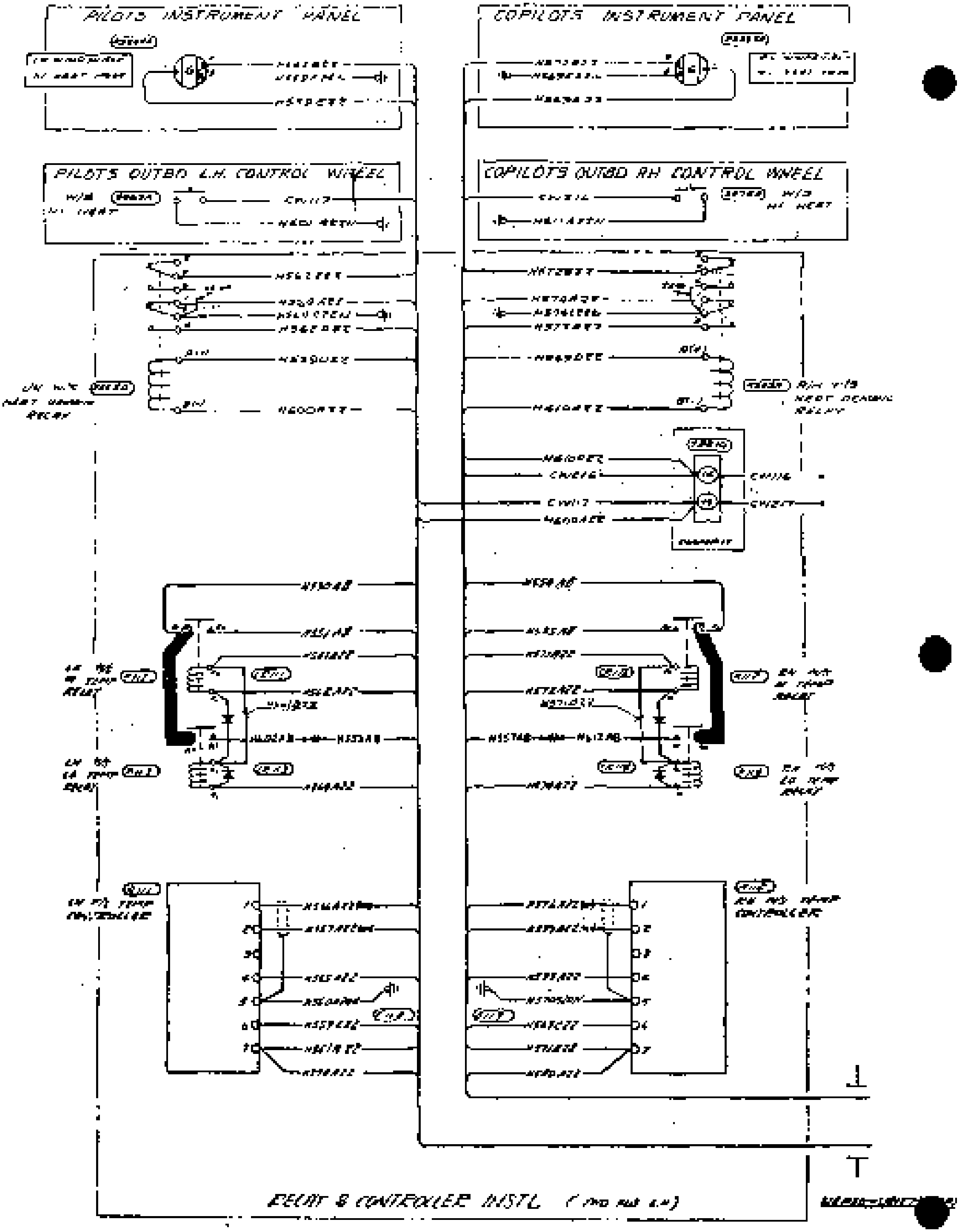
Aircraft S/N 661SA, 697SA - 730SA
Modified By SR020/34-005

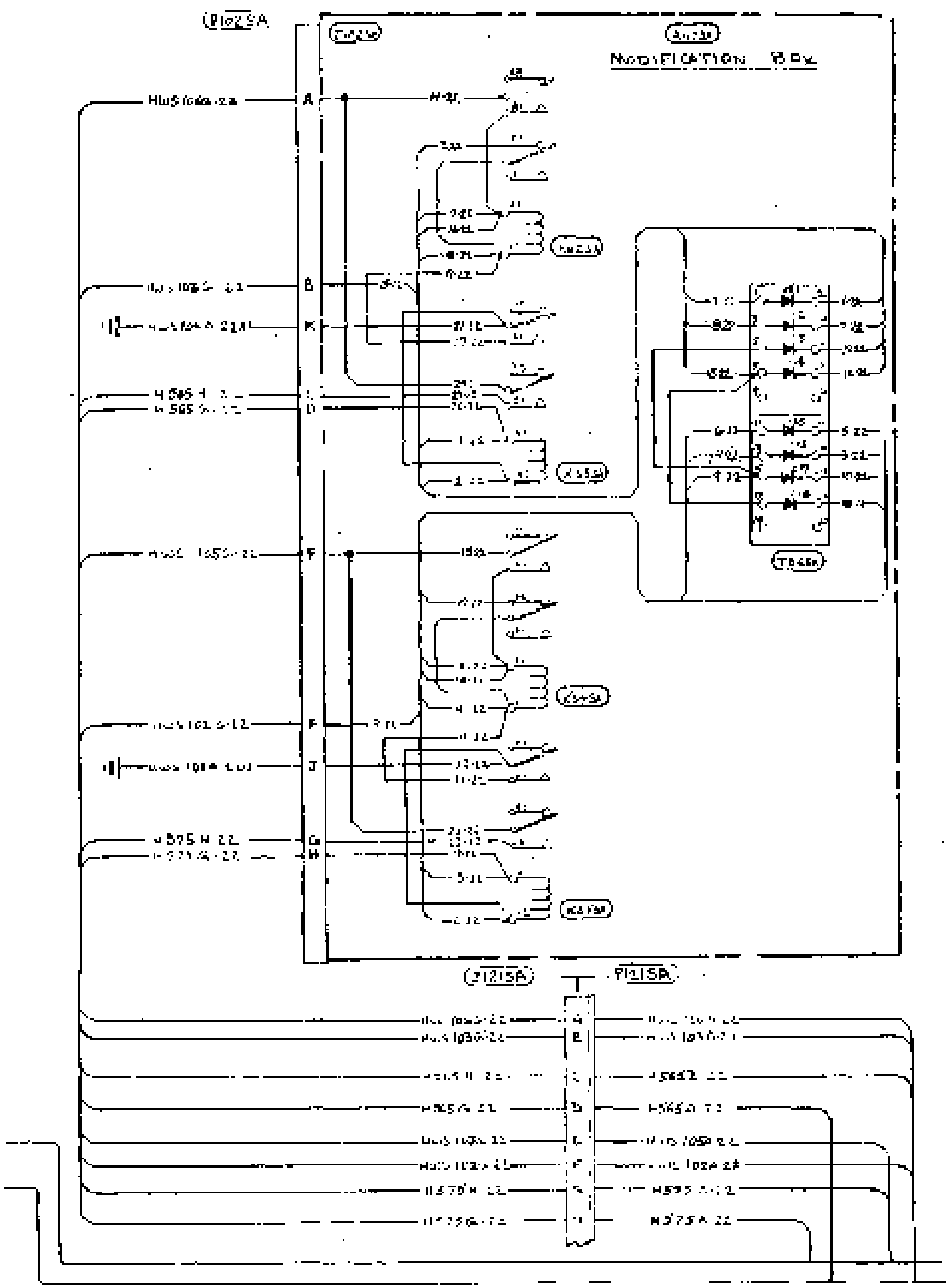


WINDSHIELD WIPER CONTROL
Aircraft S/N 661SA

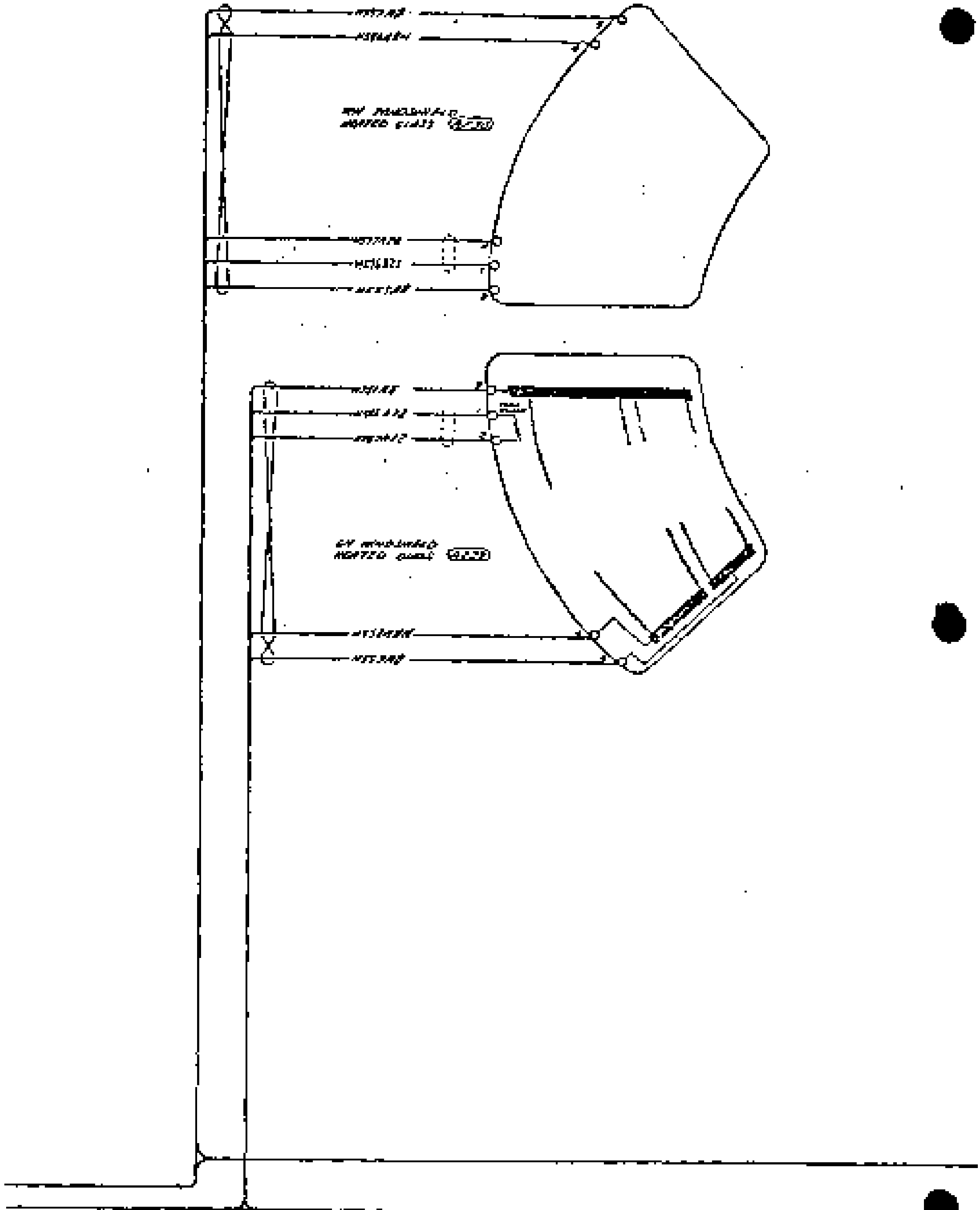


WINDSHIELD WIPER CONTROL
Aircraft S/N 697SA - 730SA

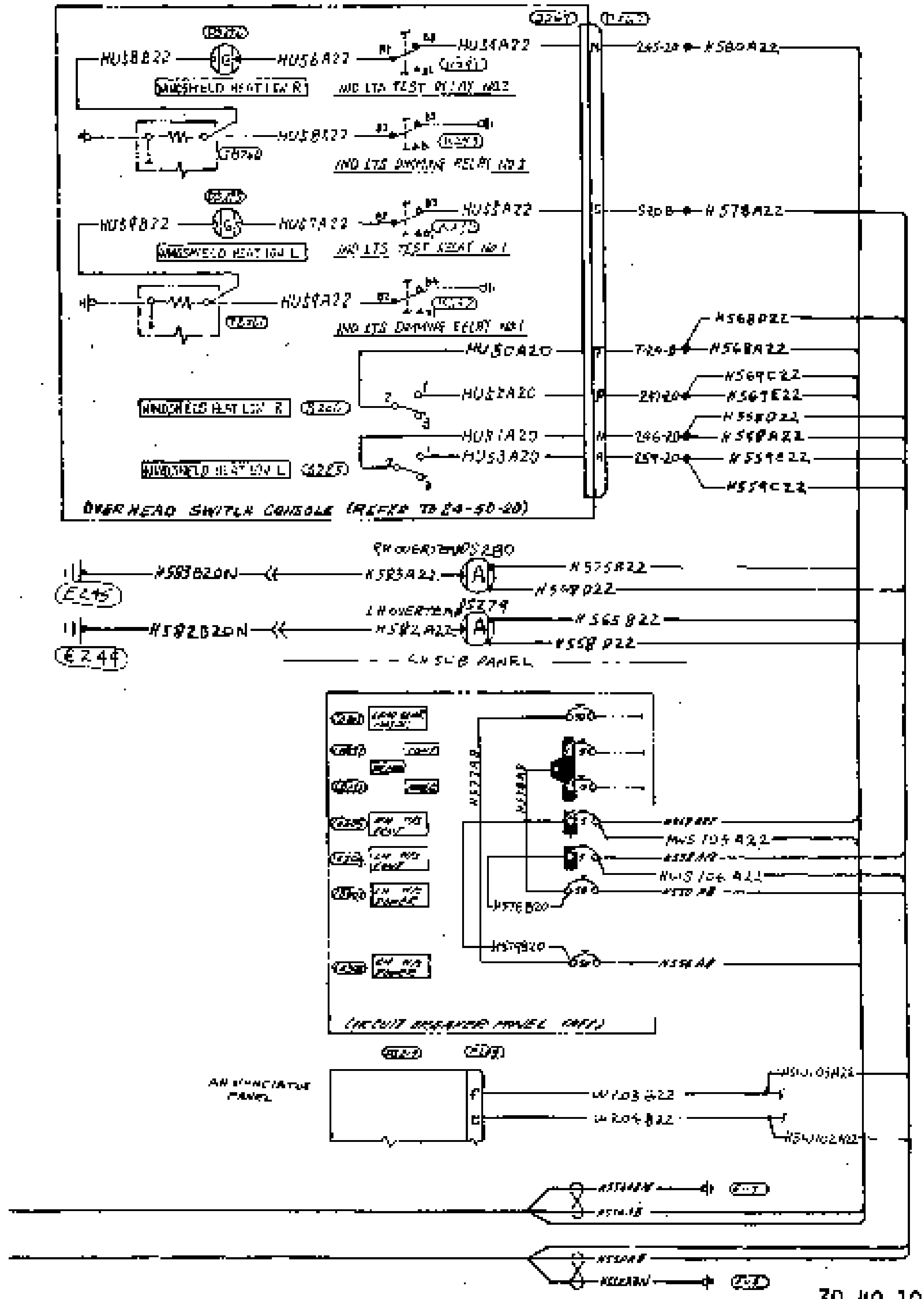

PILOTS INSTRUMENT PANEL
COPILOTS INSTRUMENT PANEL
PILOTS OUTBD LH CONTROL WHEEL
COPILOTS OUTBD RH CONTROL WHEEL
RELAY & CONTROLLER INSTL



HEATED WINDSHIELD ANTI-ICE CONTROL
Aircraft S/N 6615A

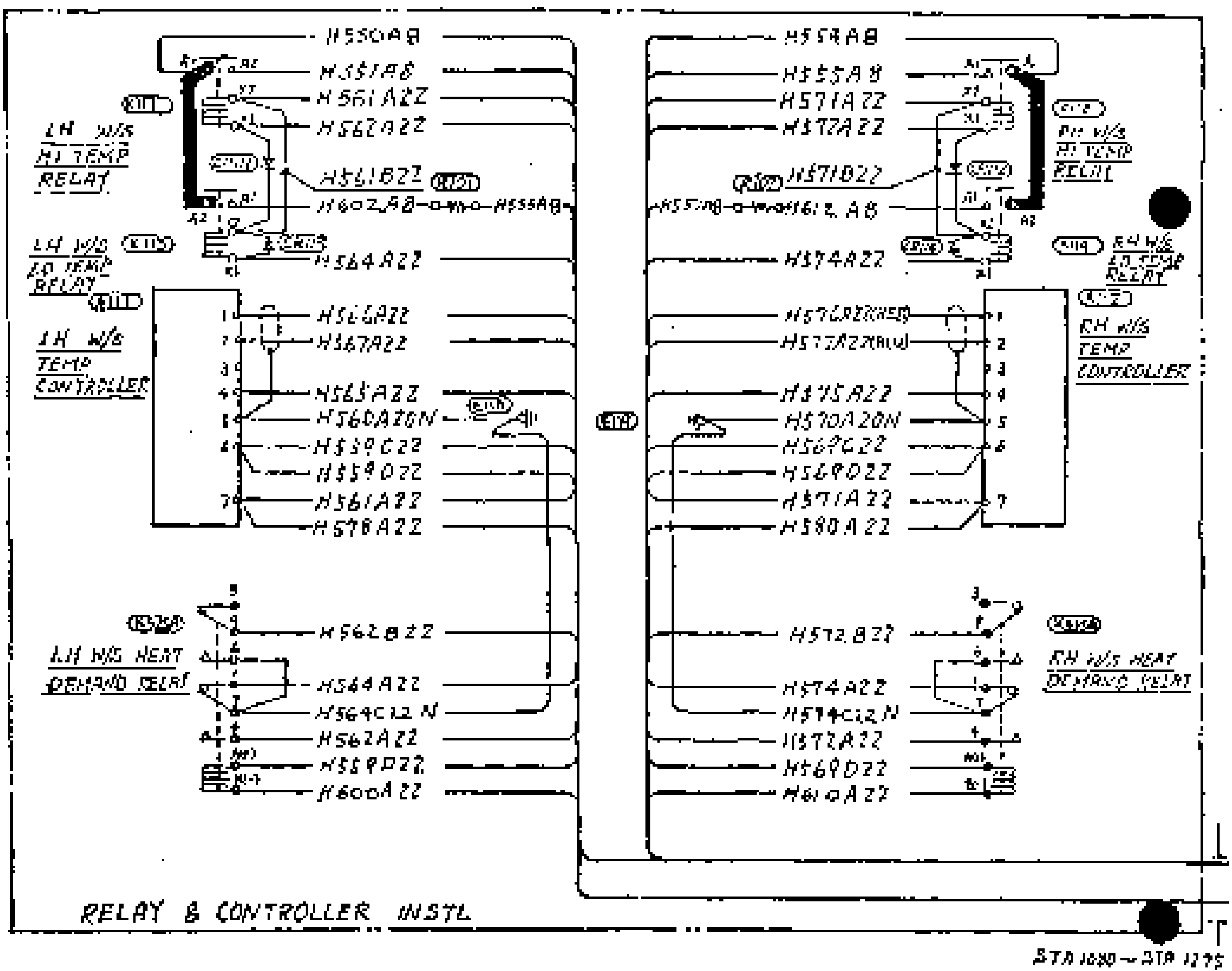


HEATED WINDSHIELD ANTI-ICE CONTROL
Aircraft S/N 661SA



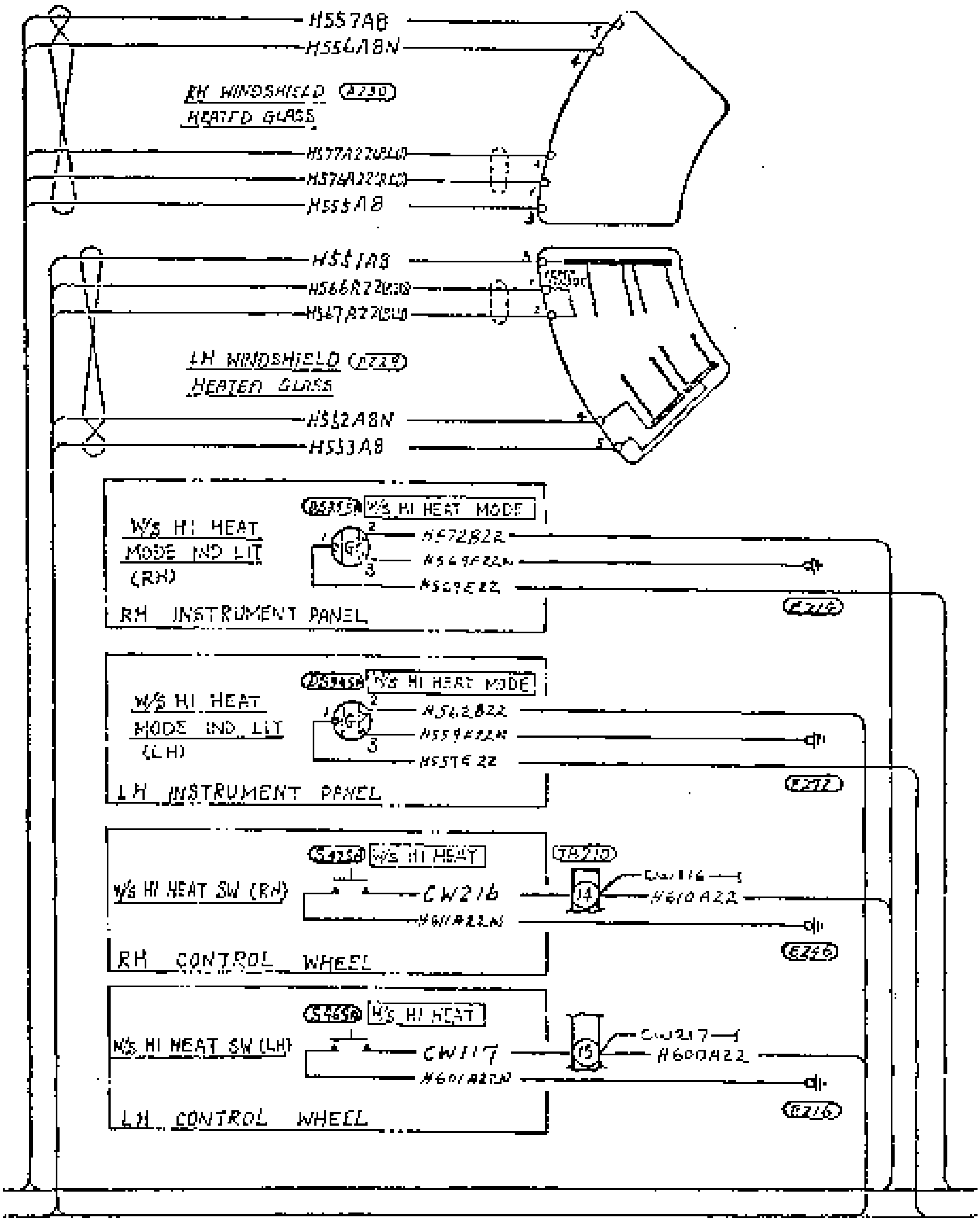
HEATED WINDSHIELD ANTI-ICE CONTROL

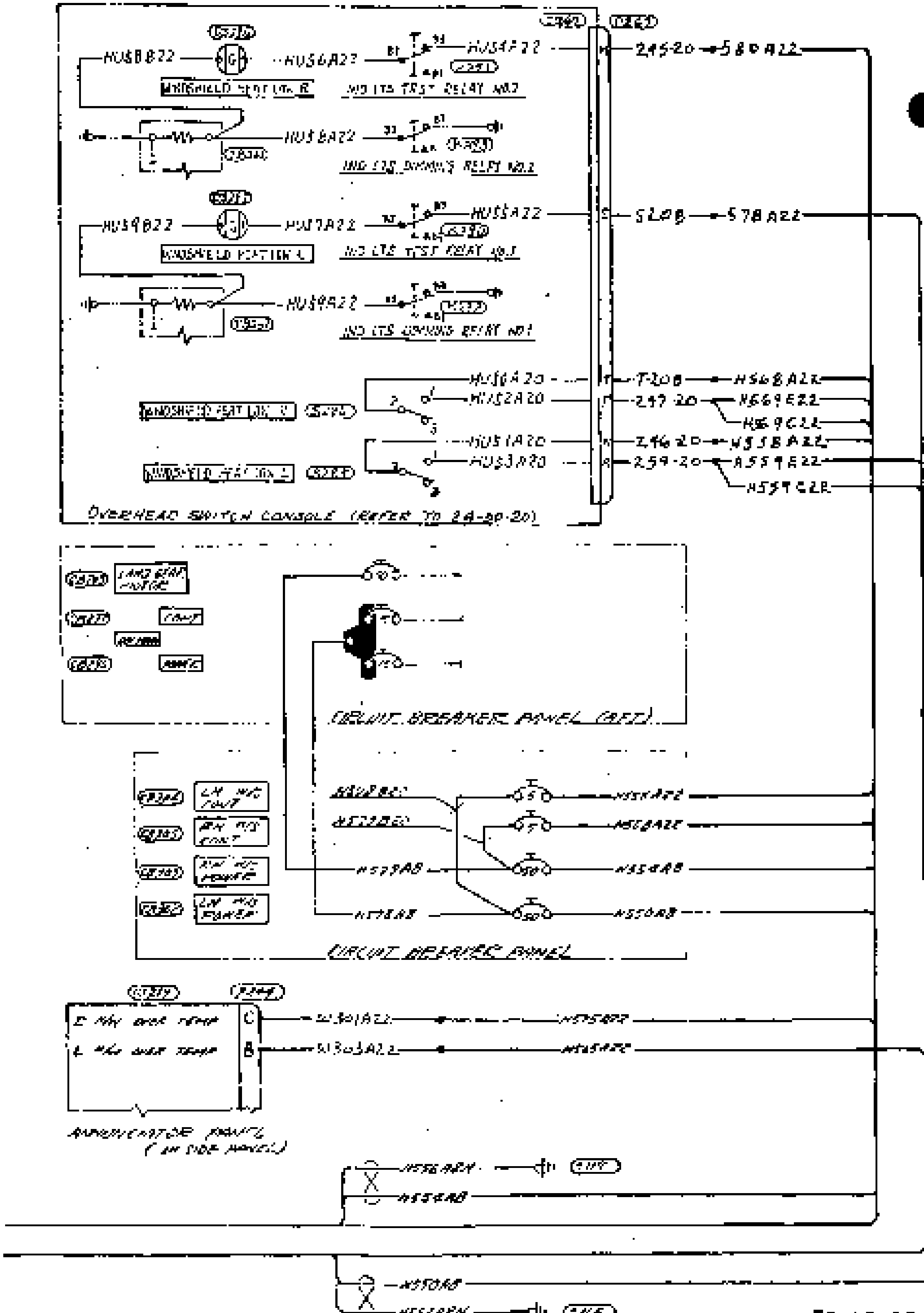
Aircraft S/N 661SA



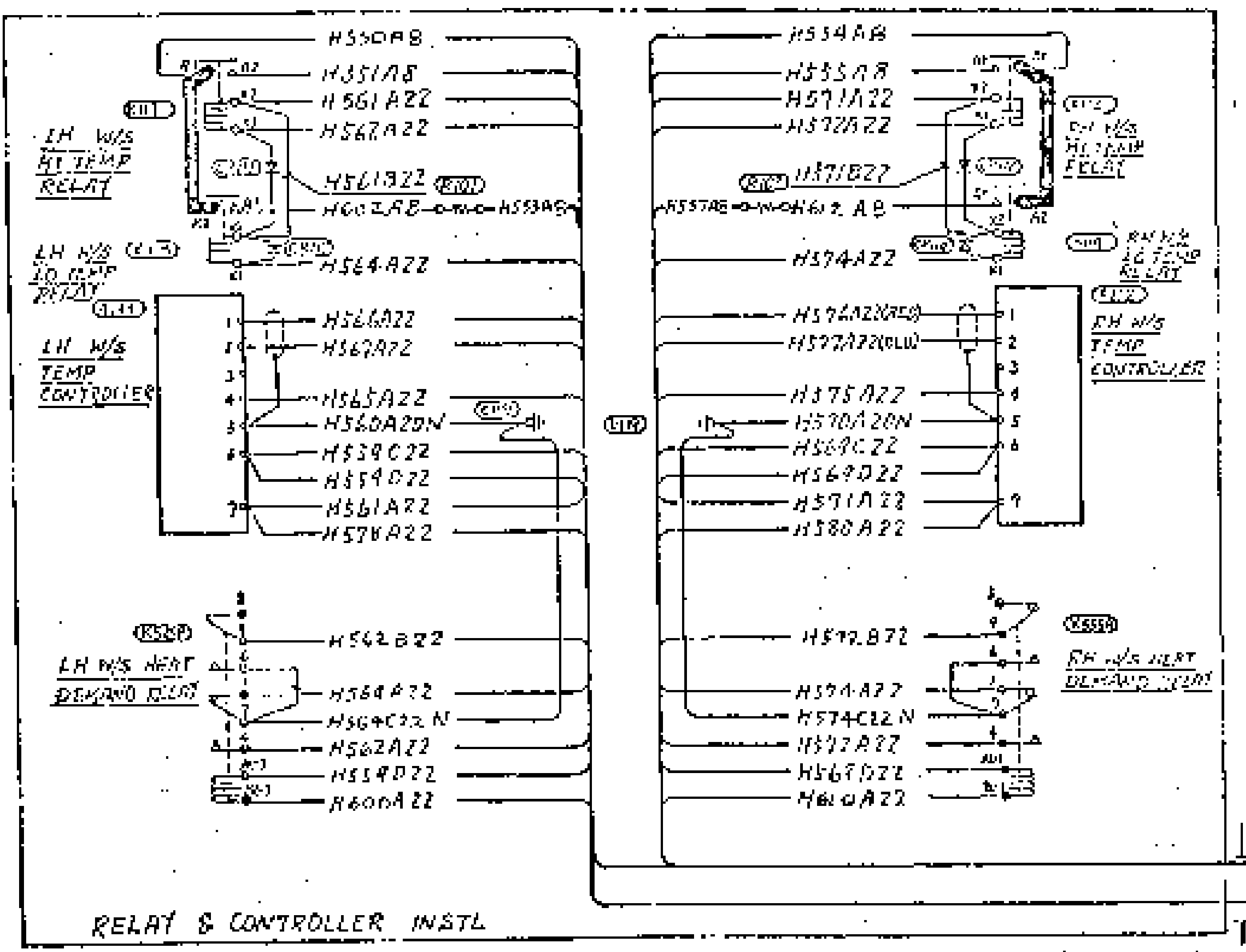
HEATED WINDSHIELD ANTI-ICE CONTROL

Aircraft S/N 697SA - 713SA

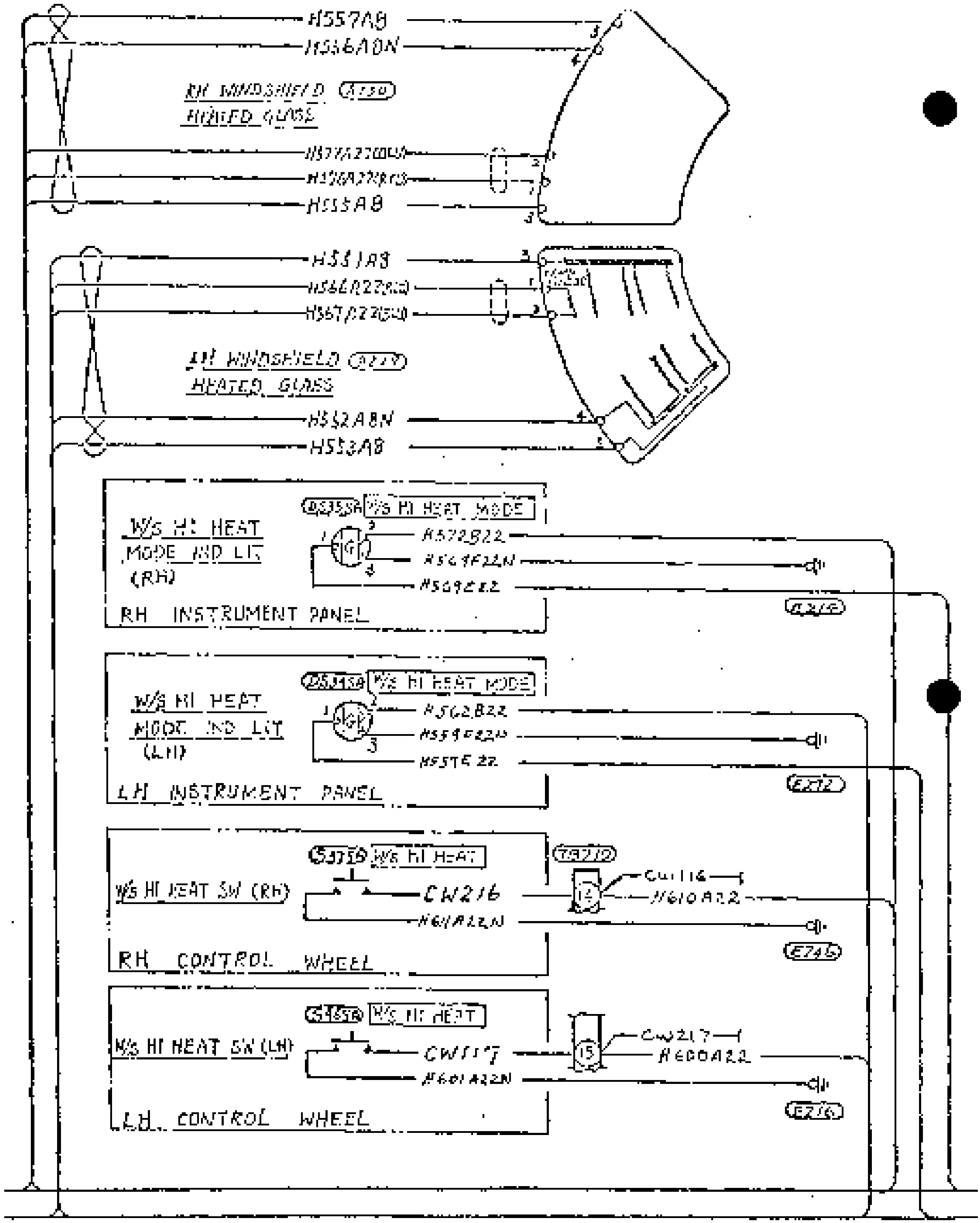

H557A8
H556A8N
RH WINDSHIELD
HEATED GLASS
H555A8
H551A8
LH WINDSHIELD
HEATED GLASS
H552A8N
H553A8
W/S HI HEAT MODE
W/S HI HEAT
MODE IND LIT
(RH)
RH INSTRUMENT PANEL
W/S HI HEAT
MODE IND LIT
(LH)
LH INSTRUMENT PANEL
W/S HI HEAT
W/S HI HEAT SW (RH)
CW216
RH CONTROL WHEEL
W/S HI HEAT SW (LH)
CW117
H600A22
LH CONTROL WHEEL

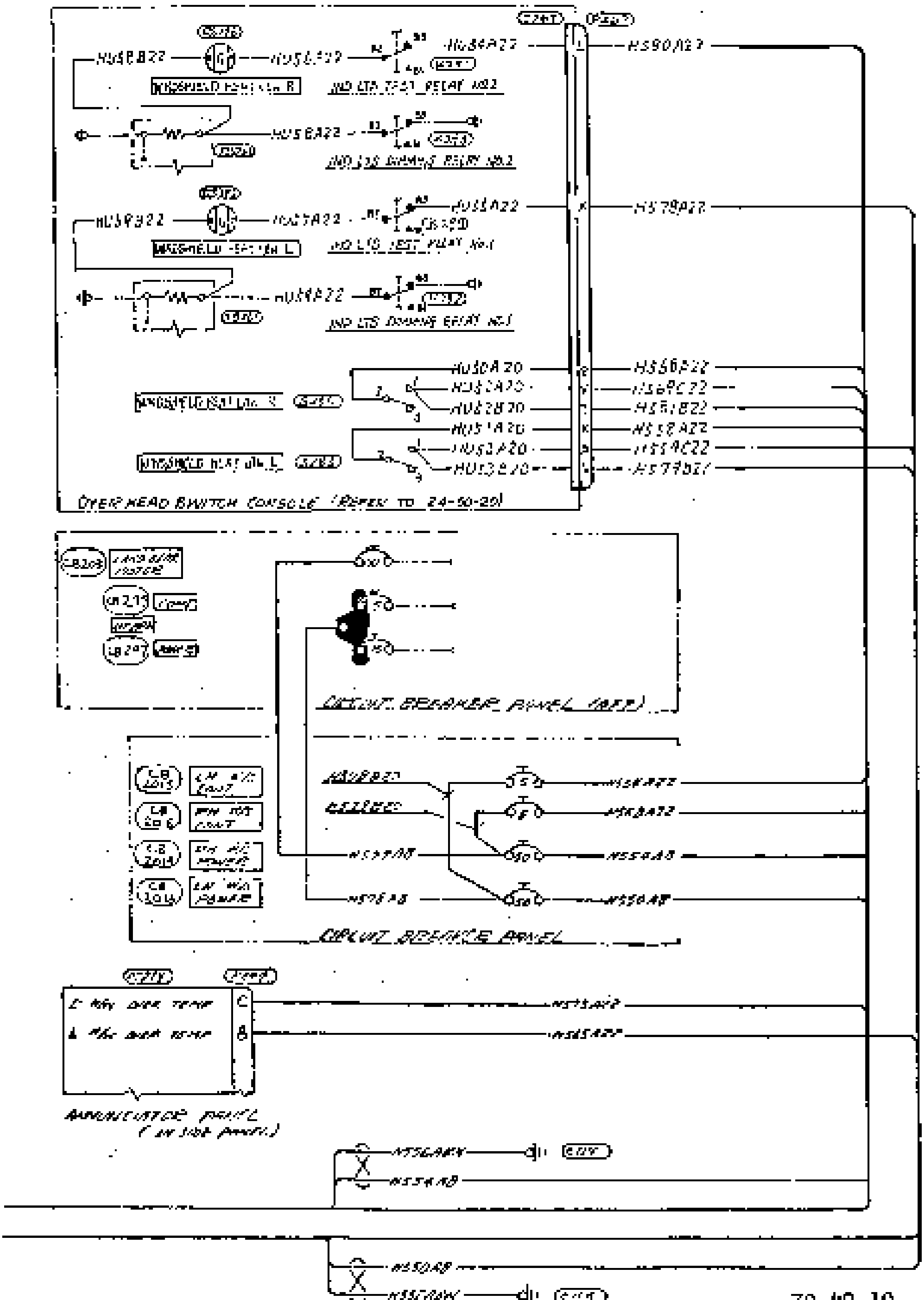


HEATED WINDSHIELD ANTI-ICE CONTROL
Aircraft S/N 697SA - 713SA



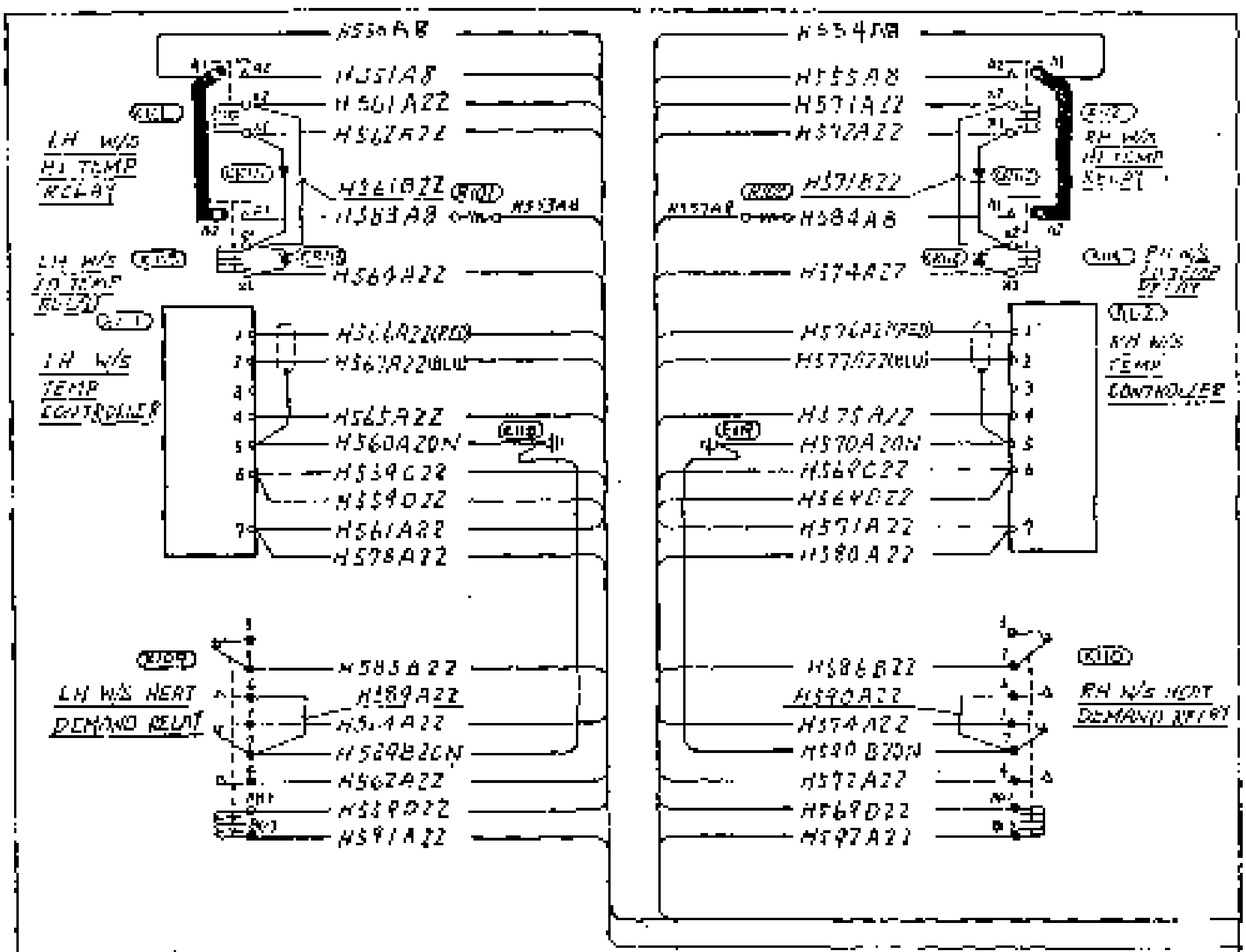
STA 1080 → STA 1275


H557A8
H556A8N
RH WINDSHIELD
HEATED GLASS
H555A8
H551A8
LH WINDSHIELD
HEATED GLASS
H552A8N
H553A8
W/S HI HEAT MODE
W/S HI HEAT
MODE IND LIT
(RH)
RH INSTRUMENT PANEL
W/S HI HEAT MODE
W/S HI HEAT
MODE IND LIT
(LH)
LH INSTRUMENT PANEL
W/S HI HEAT
W/S HI HEAT SW (RH)
CW216
RH CONTROL WHEEL
W/S HI HEAT
W/S HI HEAT SW (LH)
LH CONTROL WHEEL

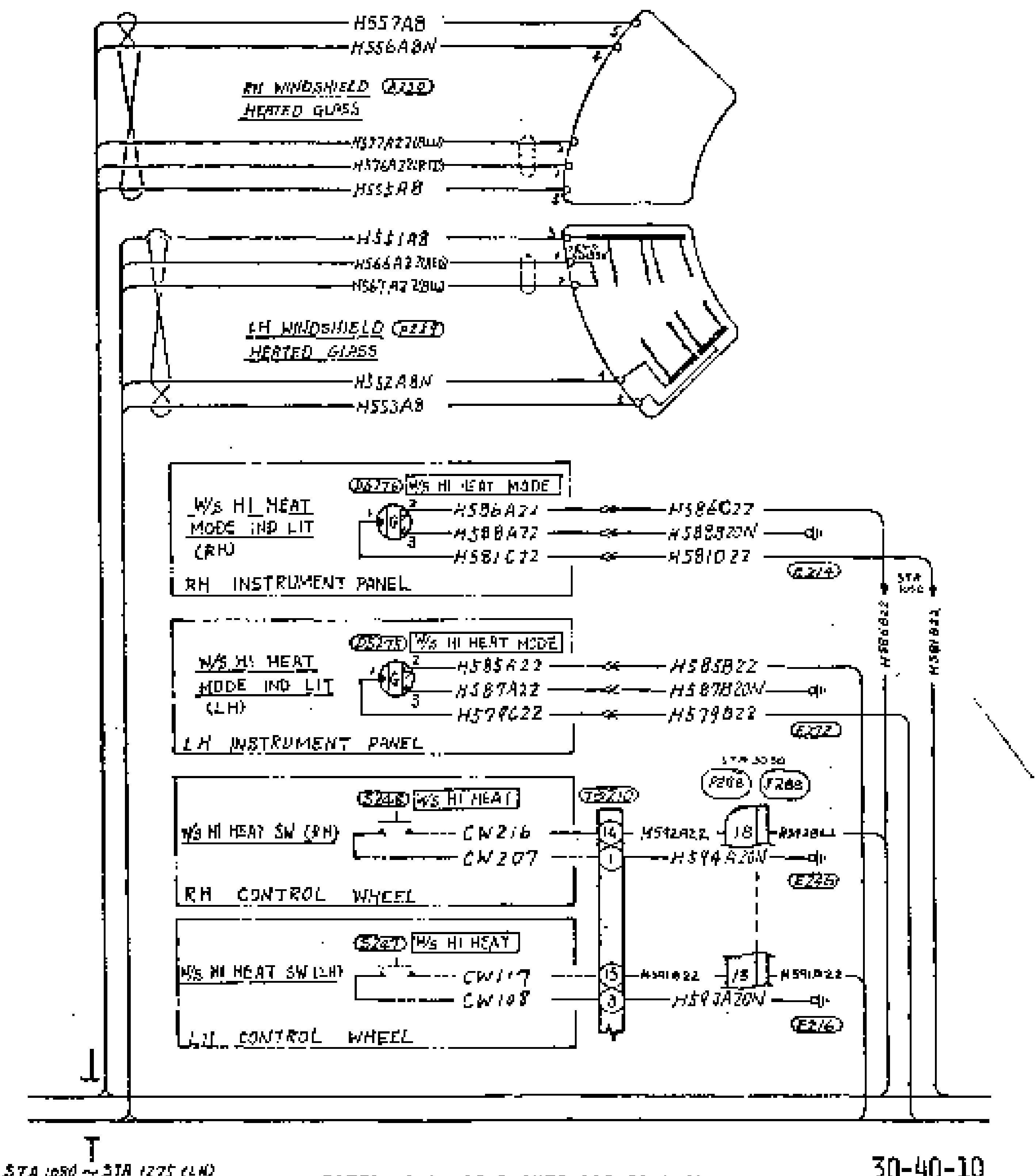


HEATED WINDSHIELD ANTI-ICE CONTROL

Aircraft S/N 714SA

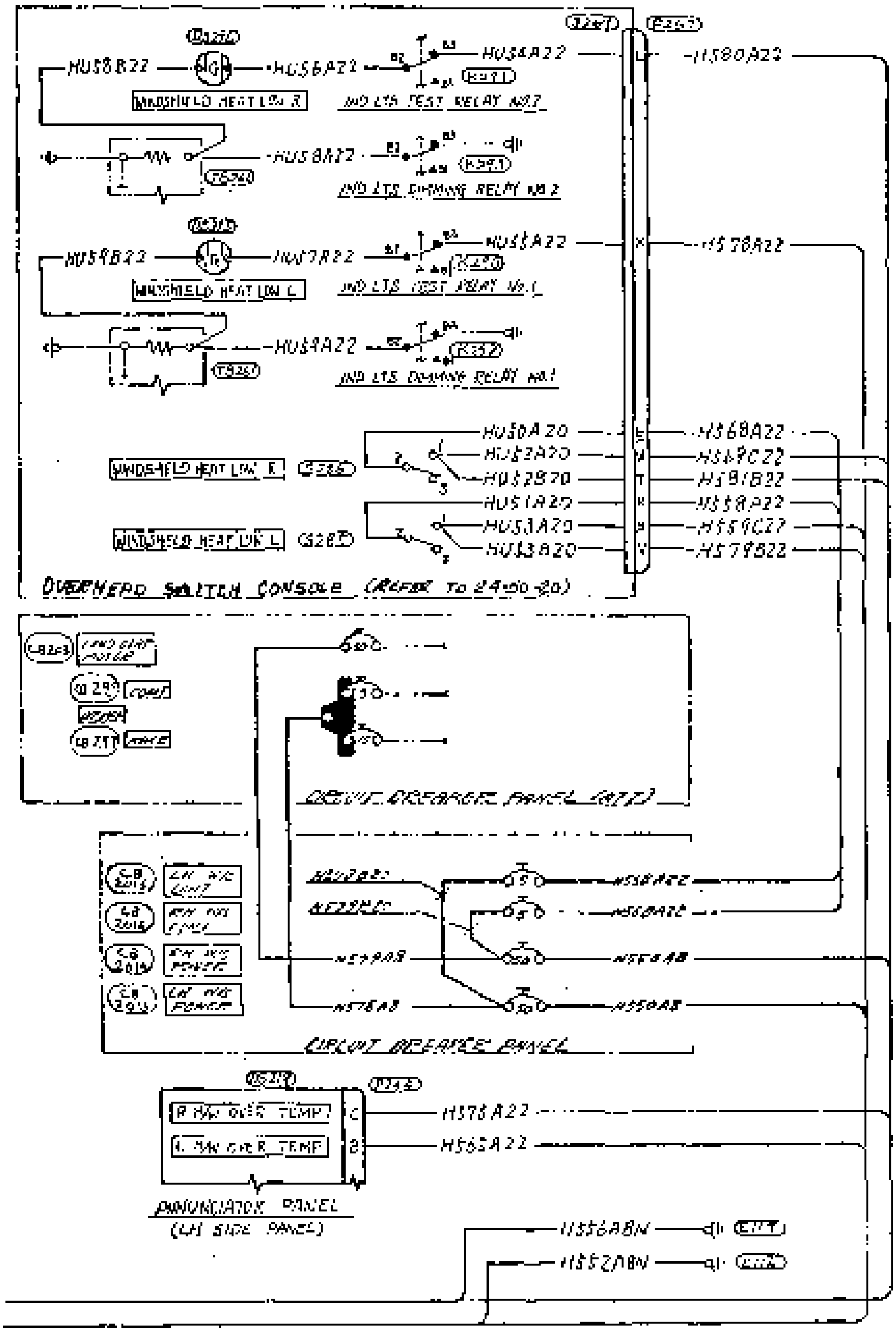


RELAY & CONTROLLER INSTL

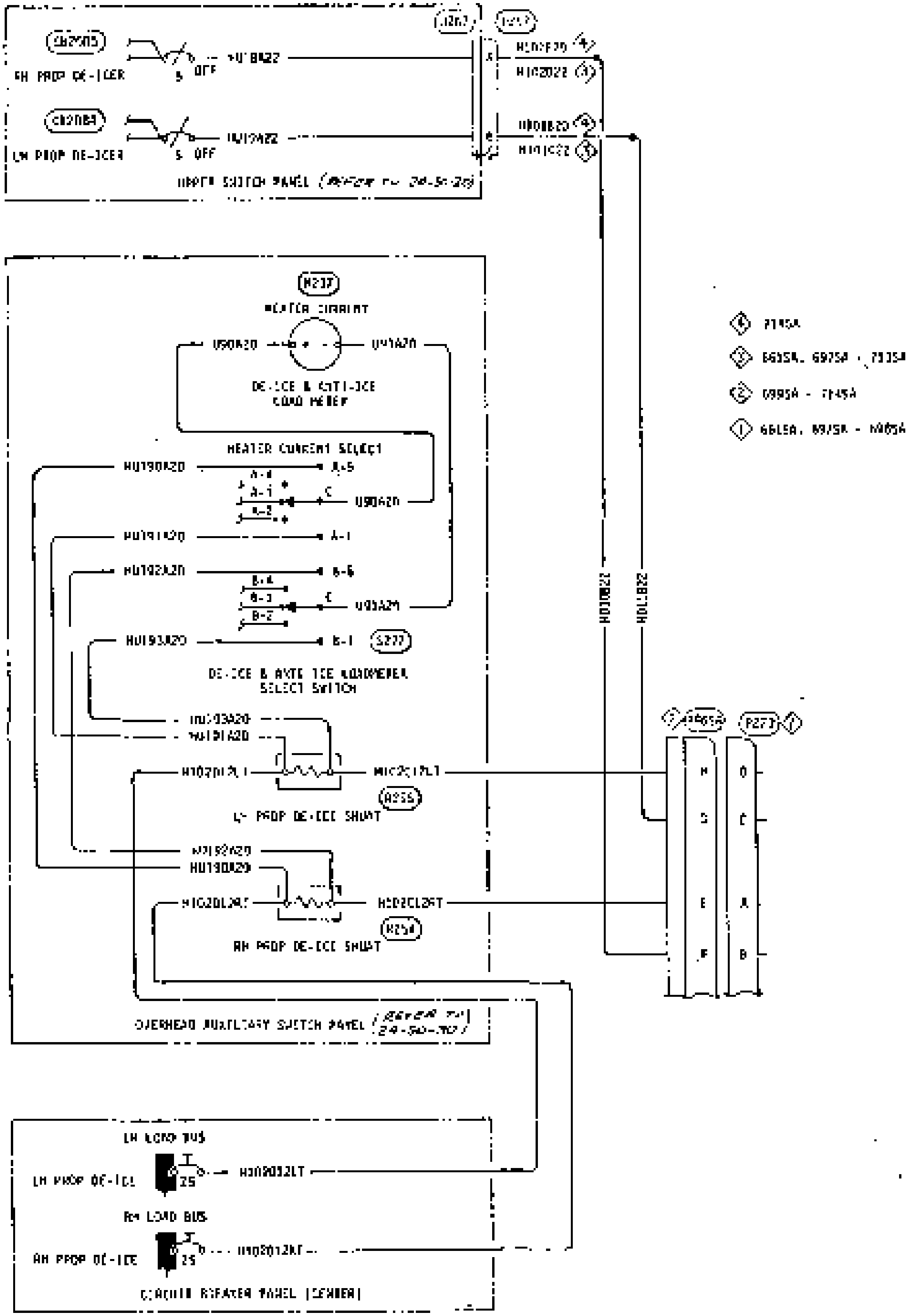


HEATED WINDSHIELD ANTI-ICE CONTROL

Aircraft S/N 718SA - 730SA

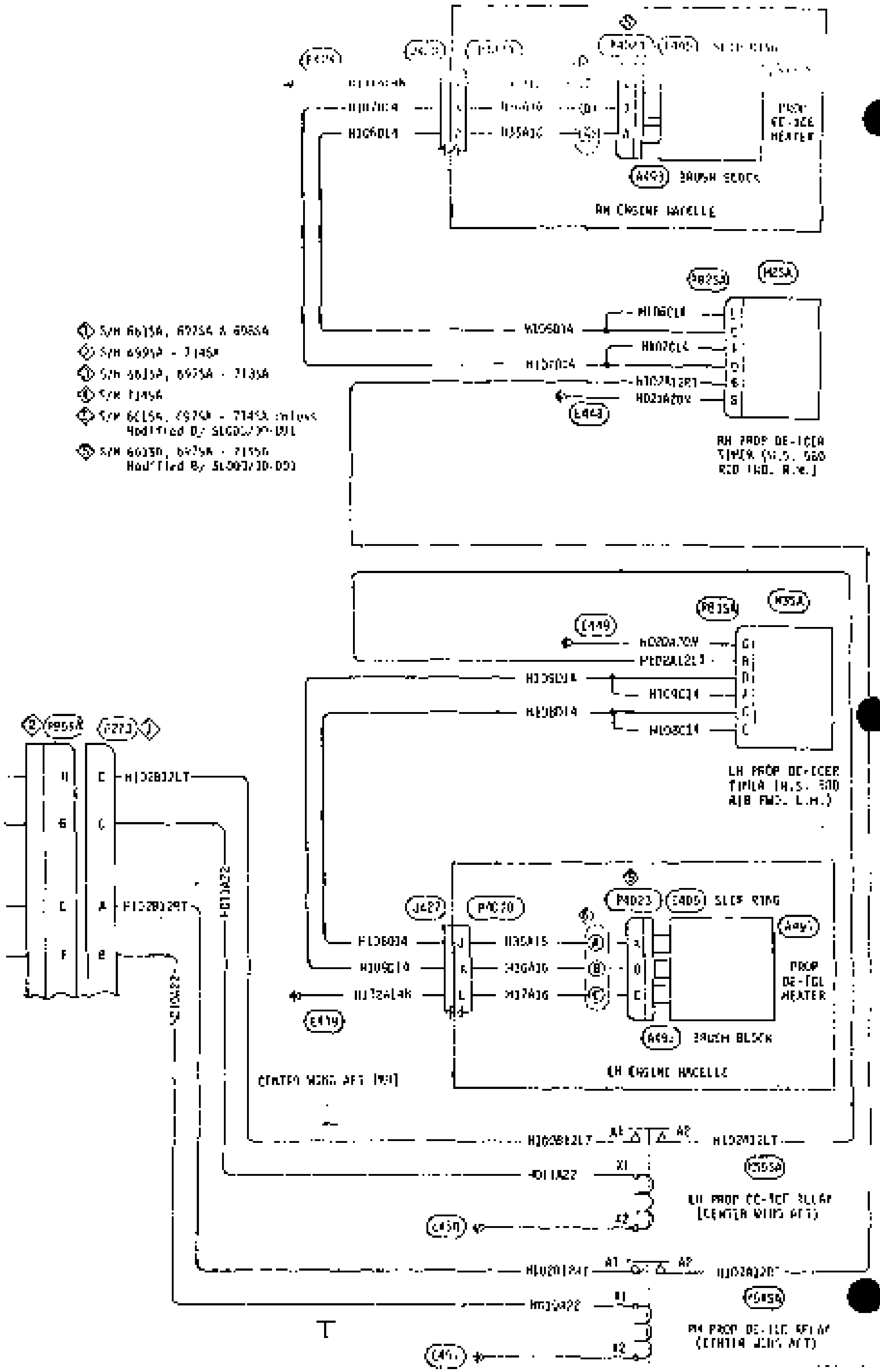


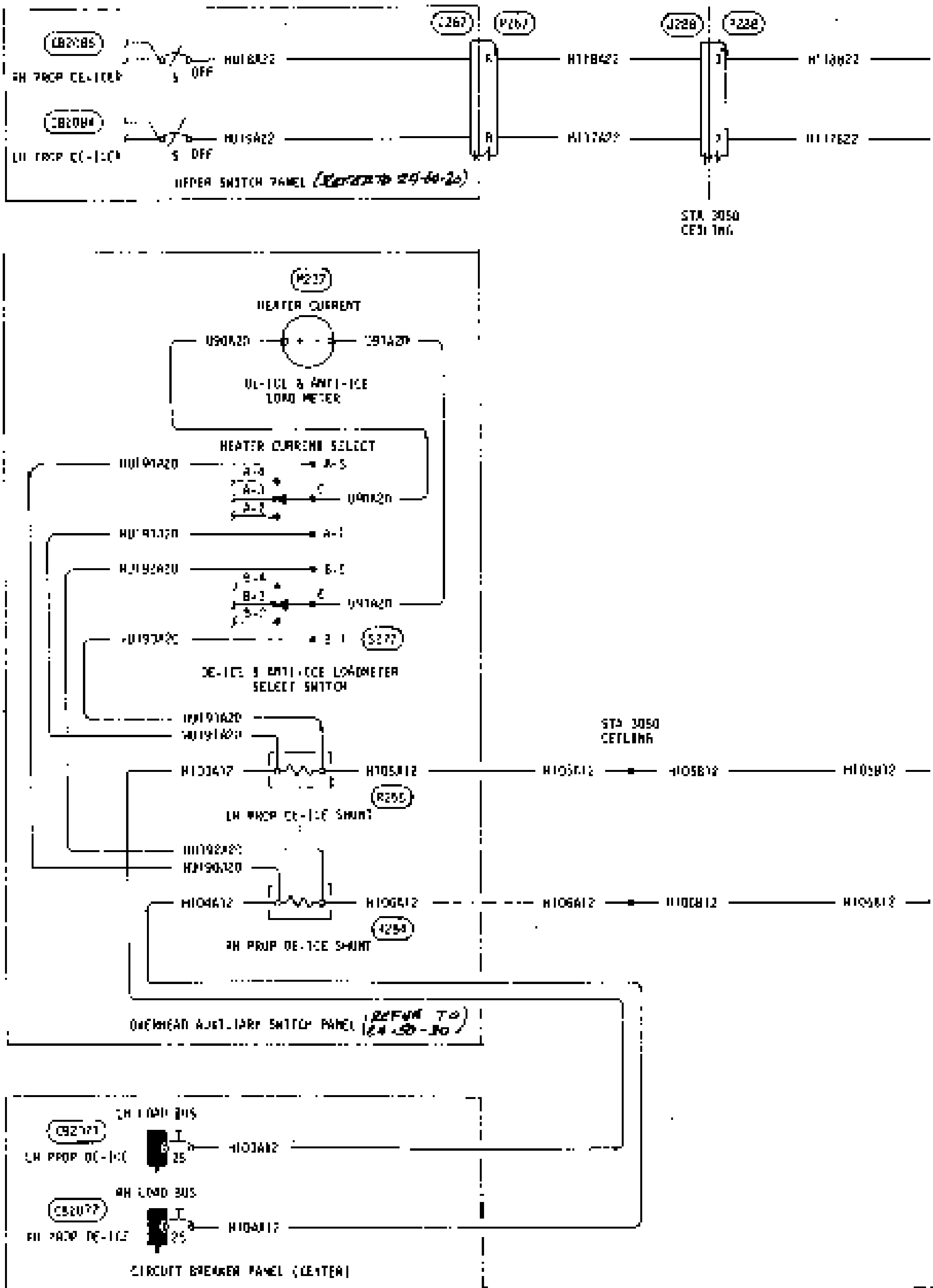
HEATED WINDSHIELD ANTI-ICE CONTROL
Aircraft S/N 718SA - 730SA



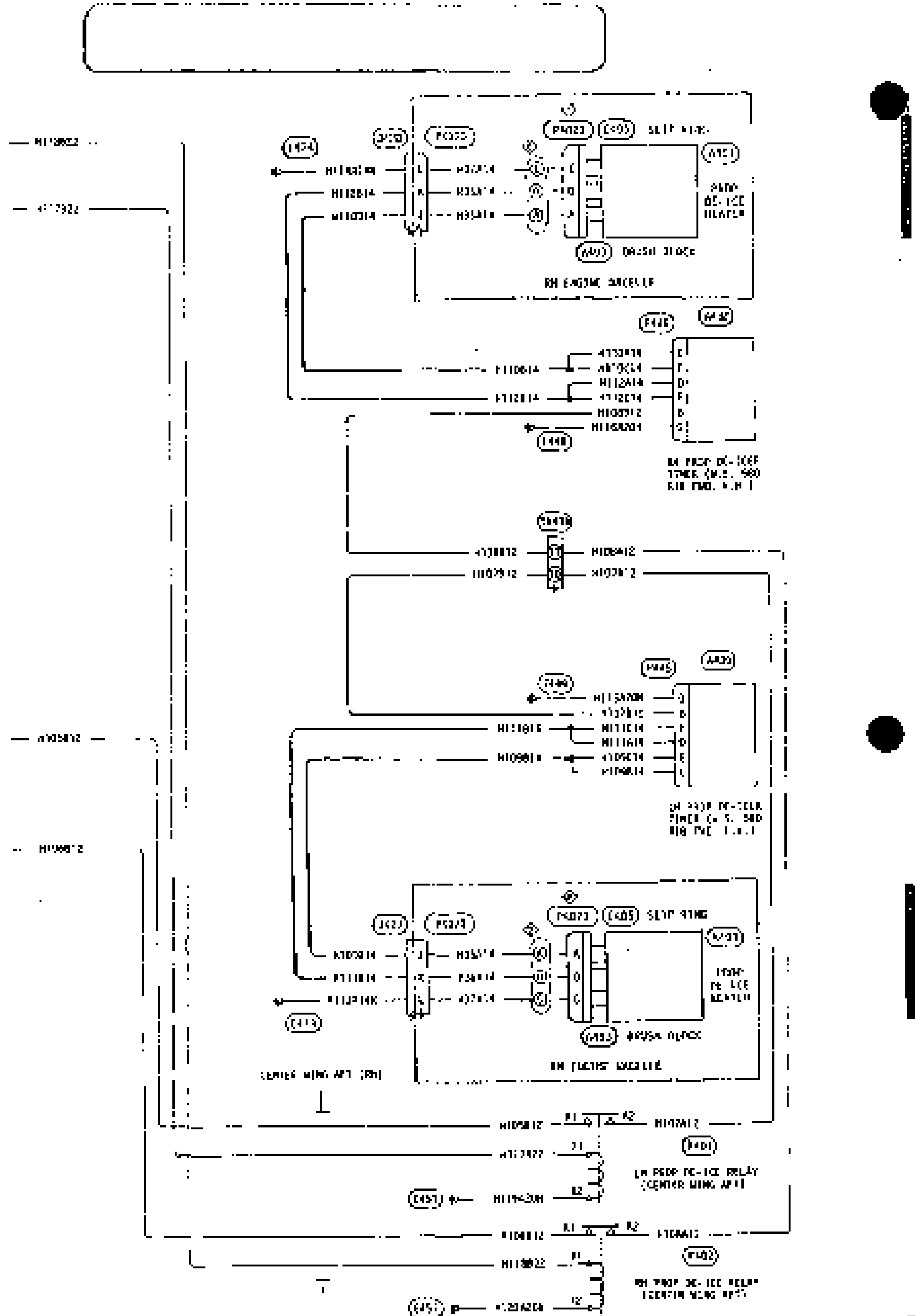
PROPELLER DE-ICE CONTROL

Aircraft S/N 661SA, 697SA - 714SA


SLIP RING
PROP DE-ICE HEATER
BRUSH BLOCK
RH ENGINE NACELLE
LH ENGINE NACELLE
LH PROP DE-ICE RELAY (CENTER WING AFT)
RH PROP DE-ICE RELAY (CENTER WING AFT)

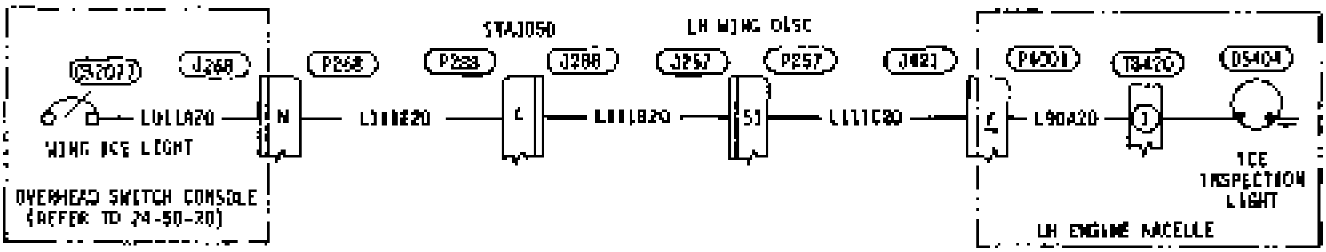


PROPELLER DE-ICE CONTROL
Aircraft S/N 718SA - 730SA

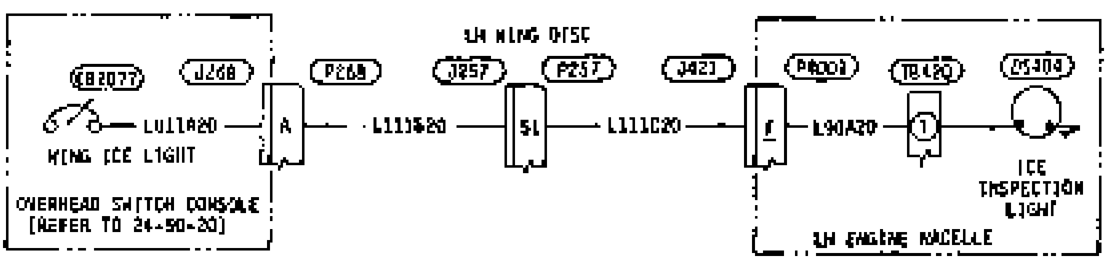


1 Aircraft Not Modified By SL003/30-001

2 Aircraft Modified By SL003/30-001



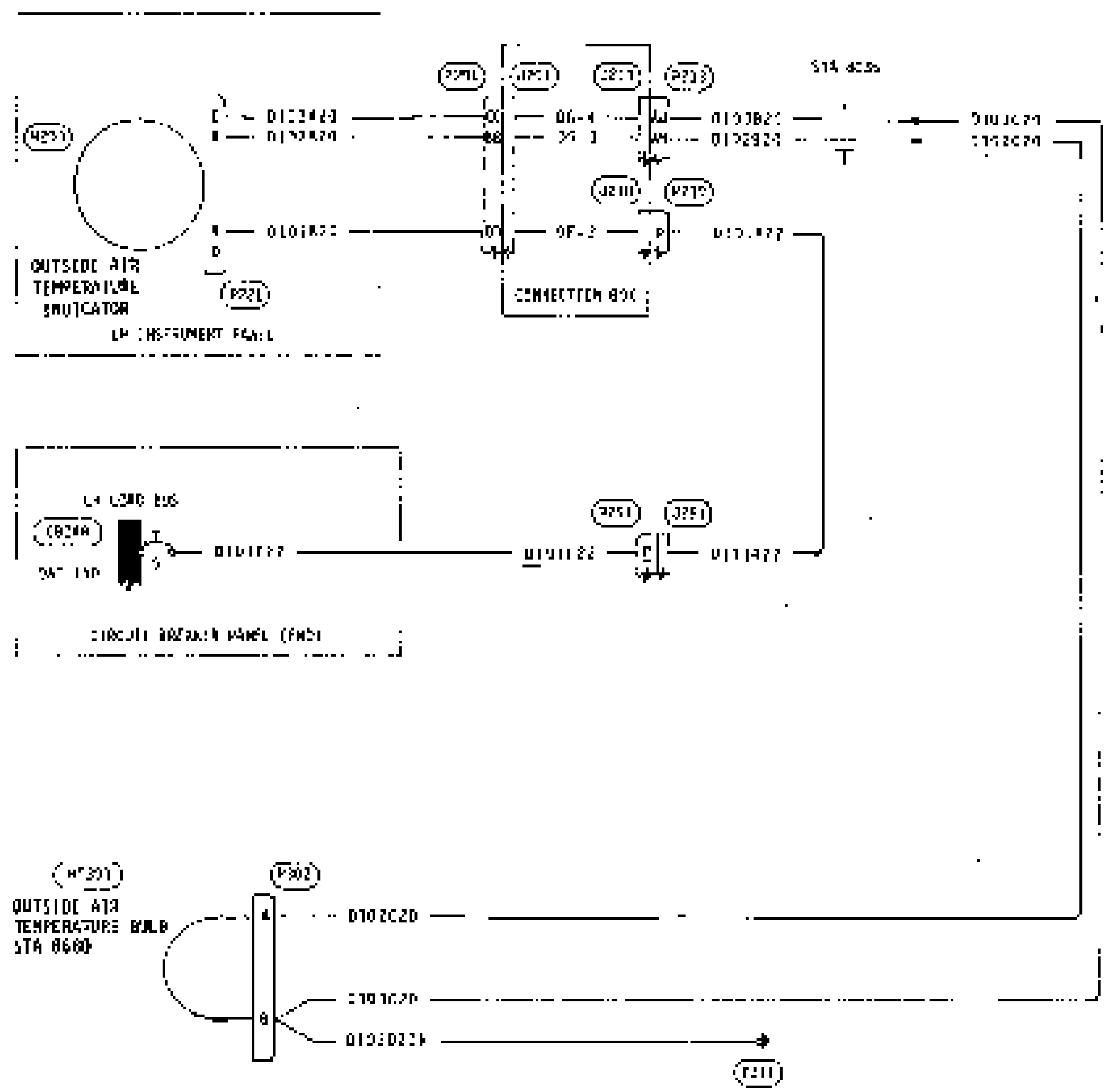
Aircraft S/N 661SA, 697SA thru 713SA



Aircraft S/N 714SA thru 730SA

CHAPTER 31

INDICATING/ RECORDING SYSTEMS



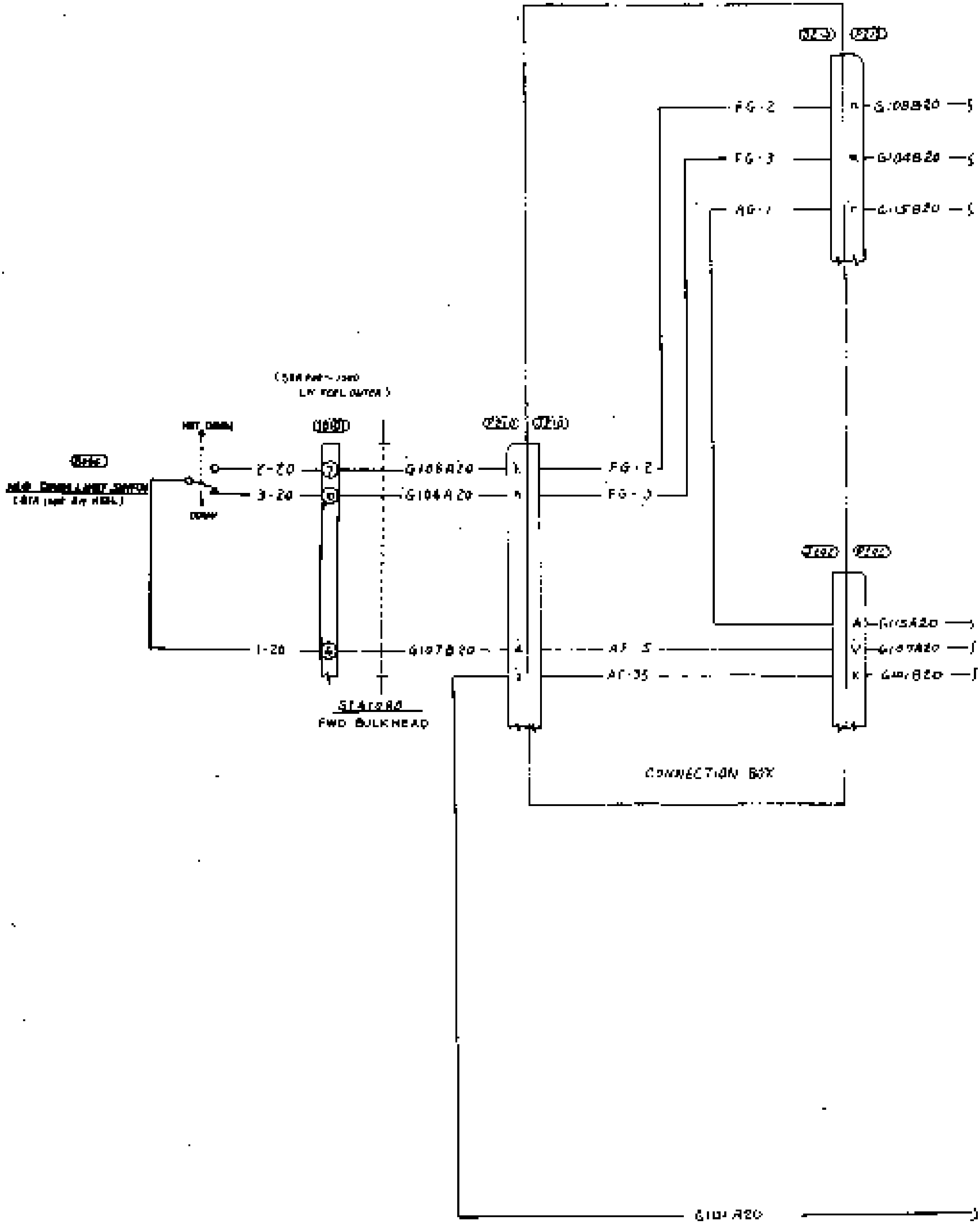
OUTSIDE AIR TEMPERATURE INDICATION

Aircraft S/N 661SA, 697SA - 730SA

CHAPTER

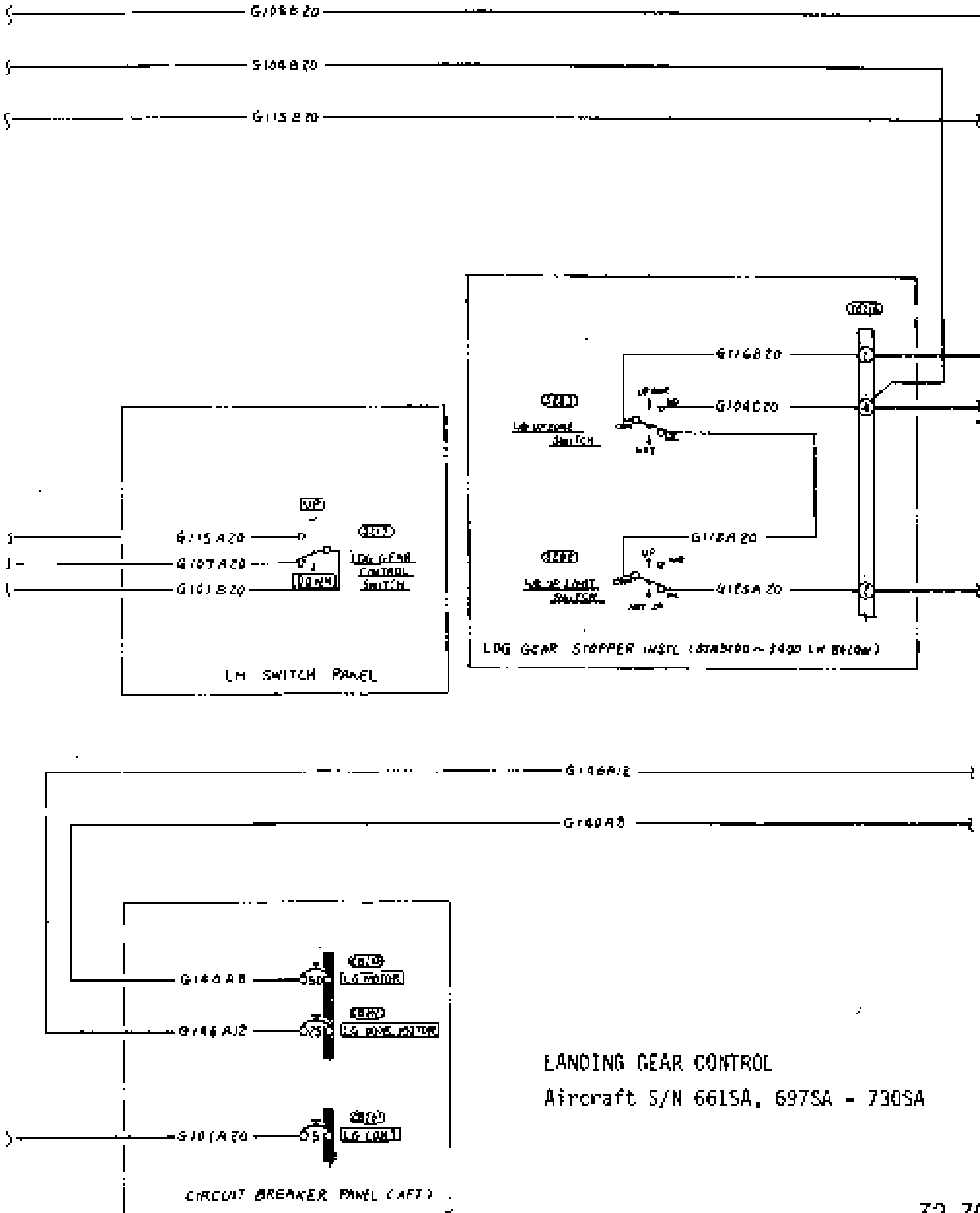
32

LANDING GEAR



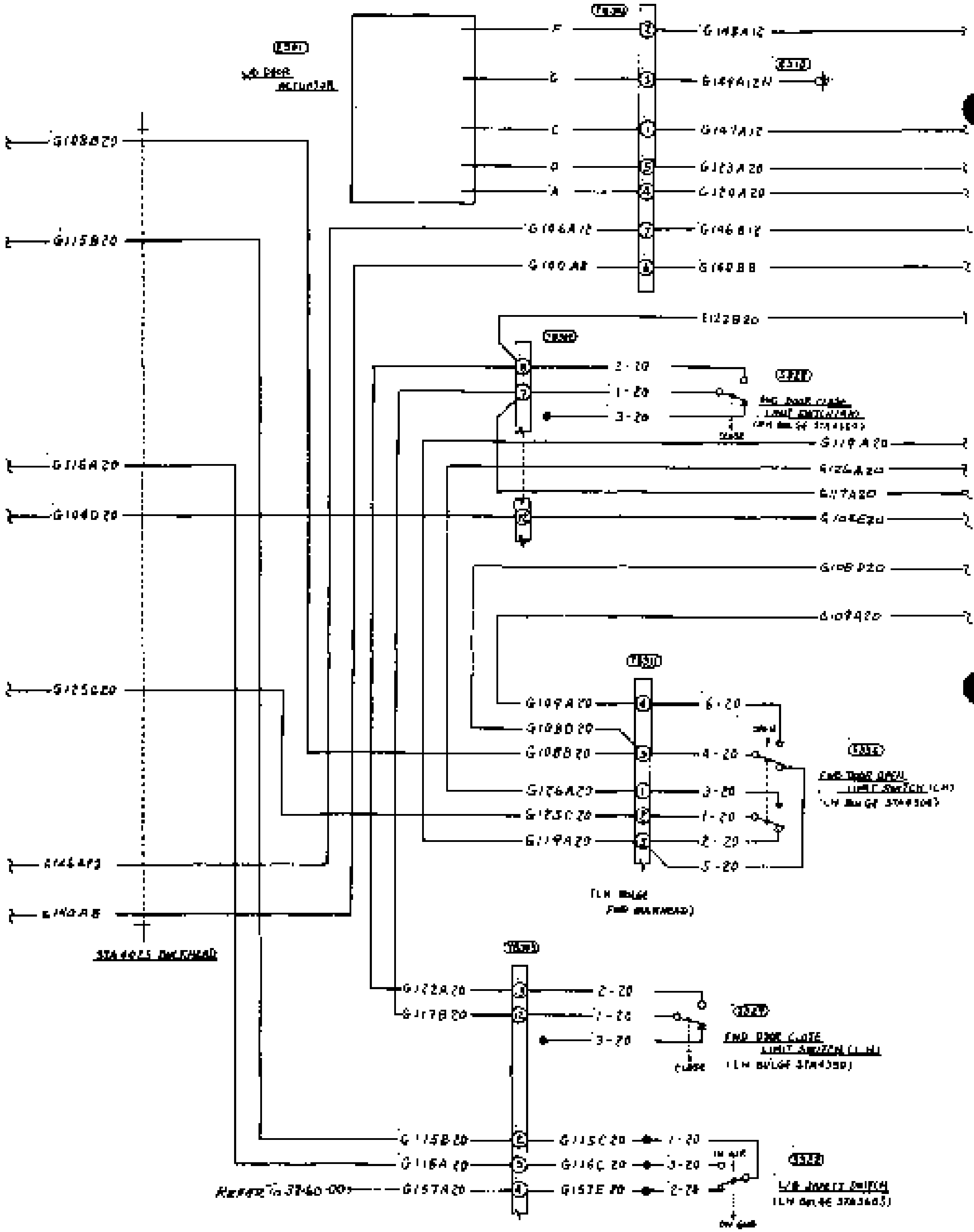
LANDING GEAR CONTROL

Aircraft S/N 661SA, 697SA - 730SA



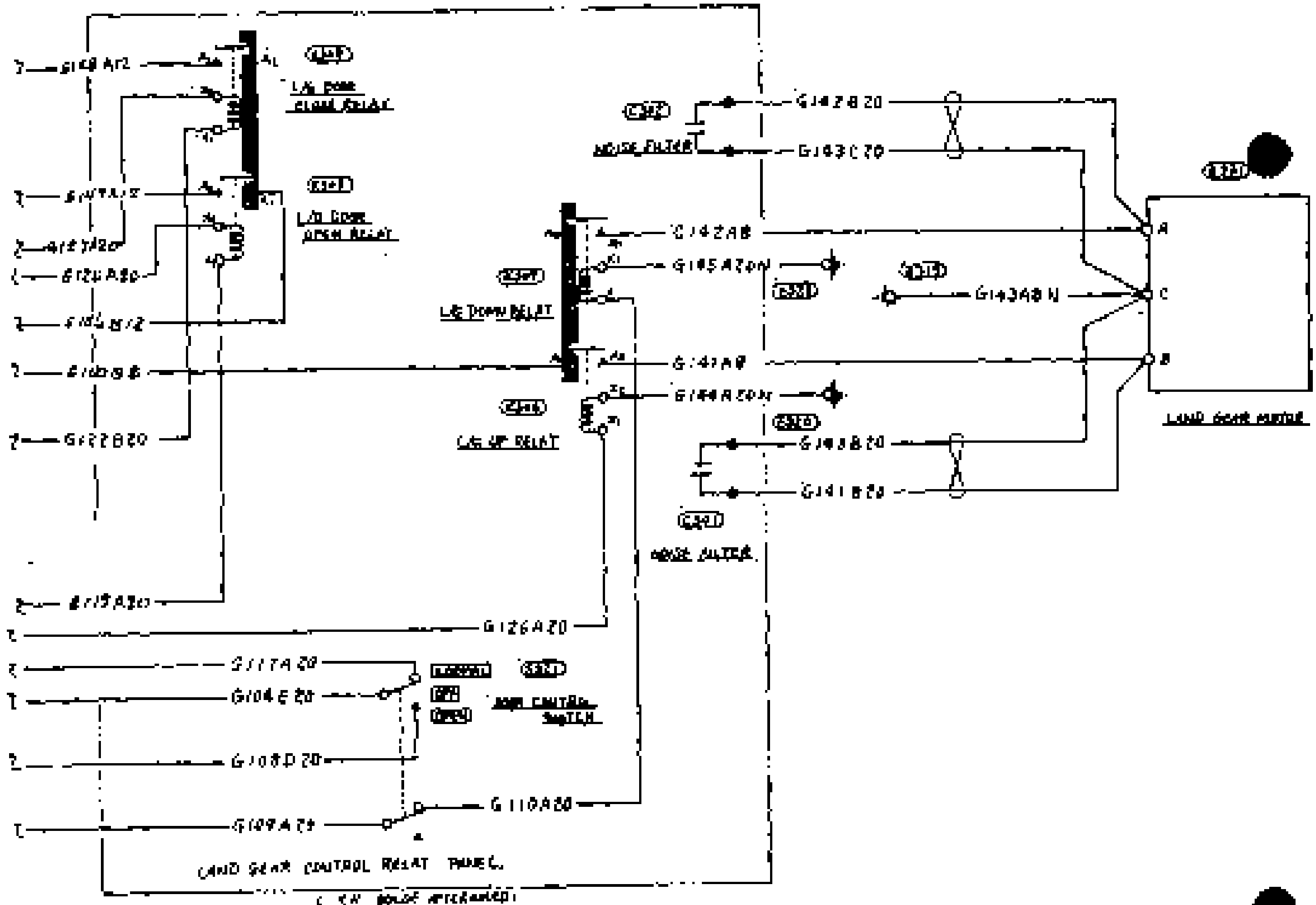
LANDING GEAR CONTROL

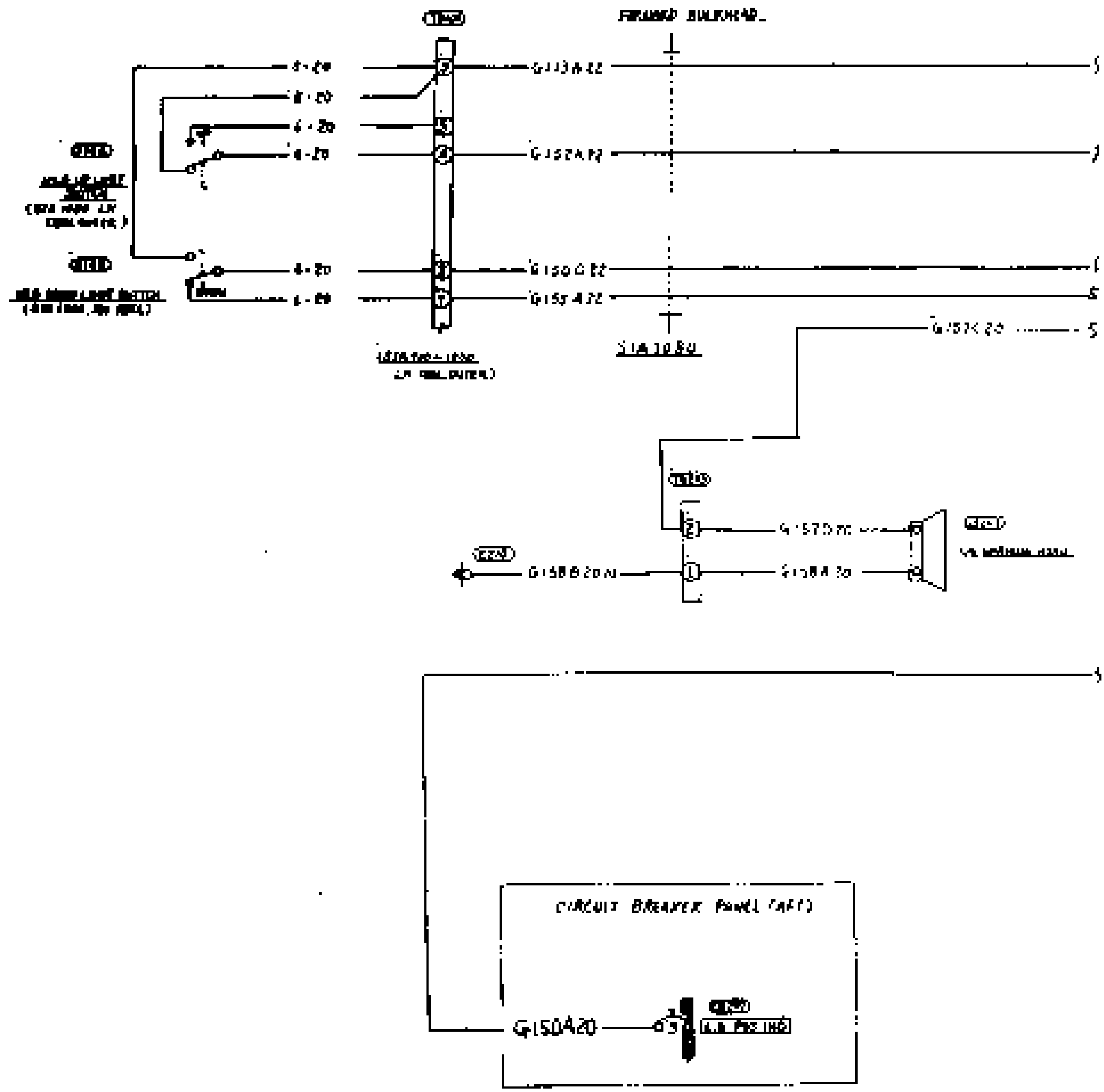
Aircraft S/N 661SA, 697SA - 730SA

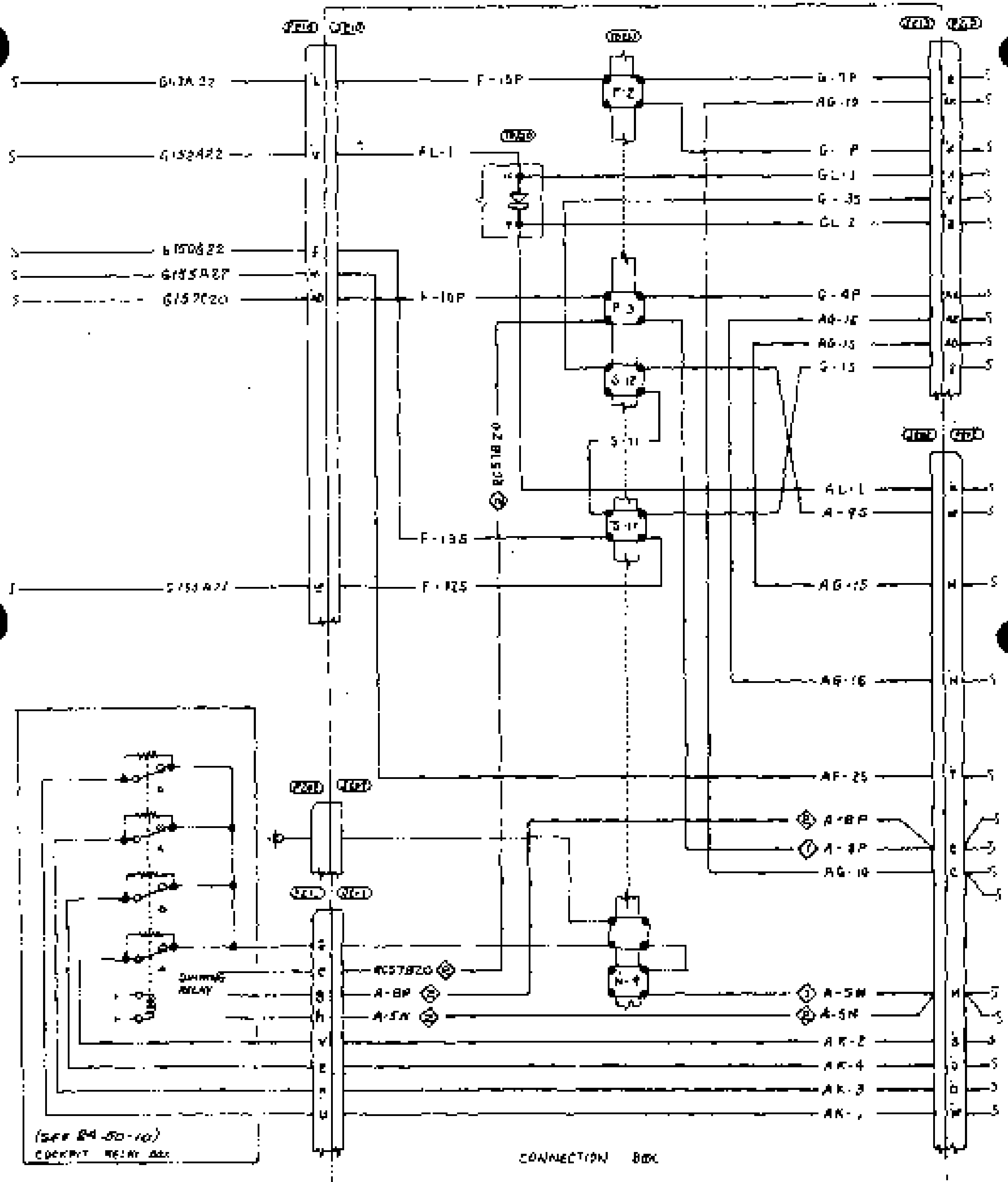


LANDING GEAR CONTROL

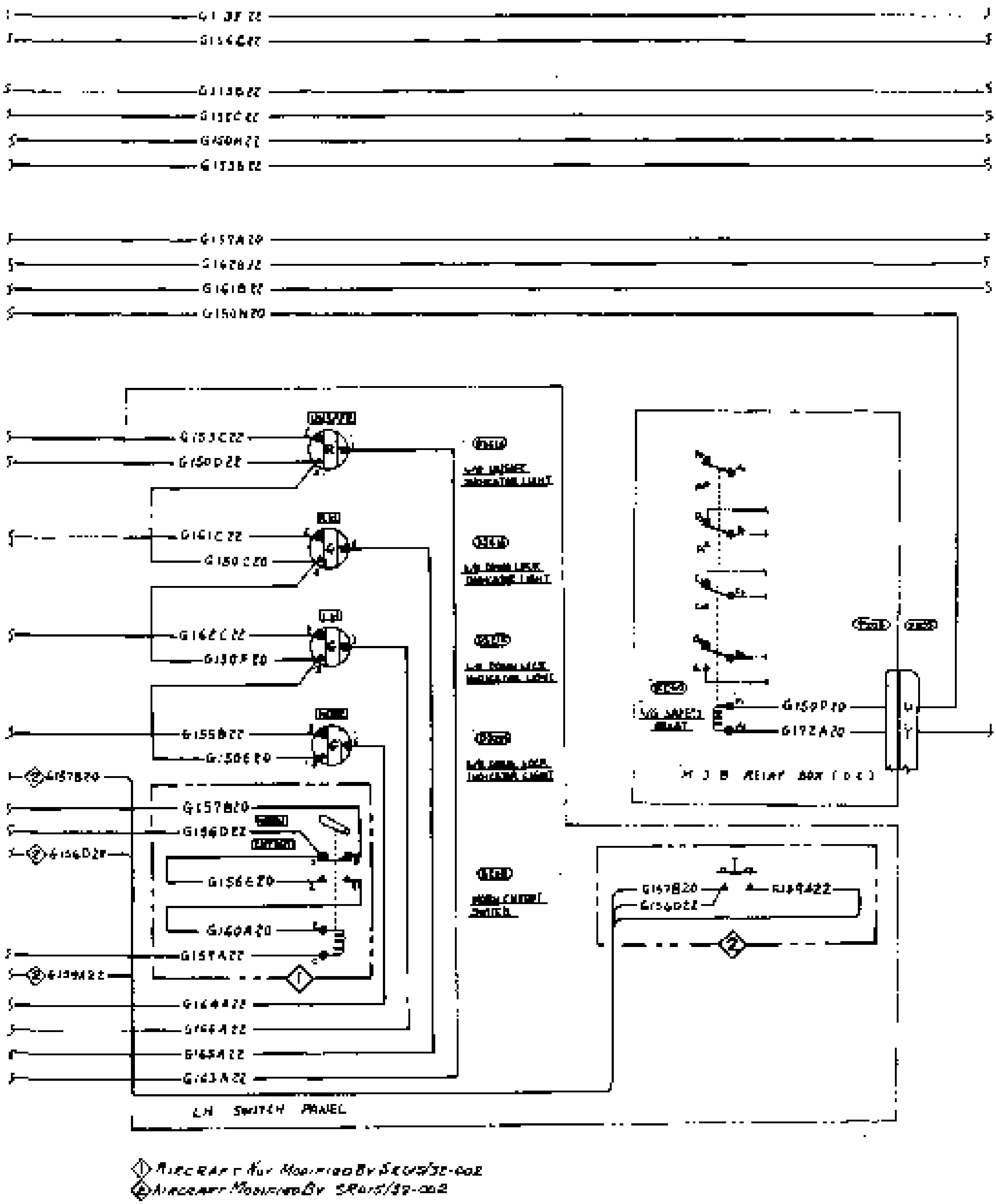
Aircraft S/N 661SA, 697SA - 730SA

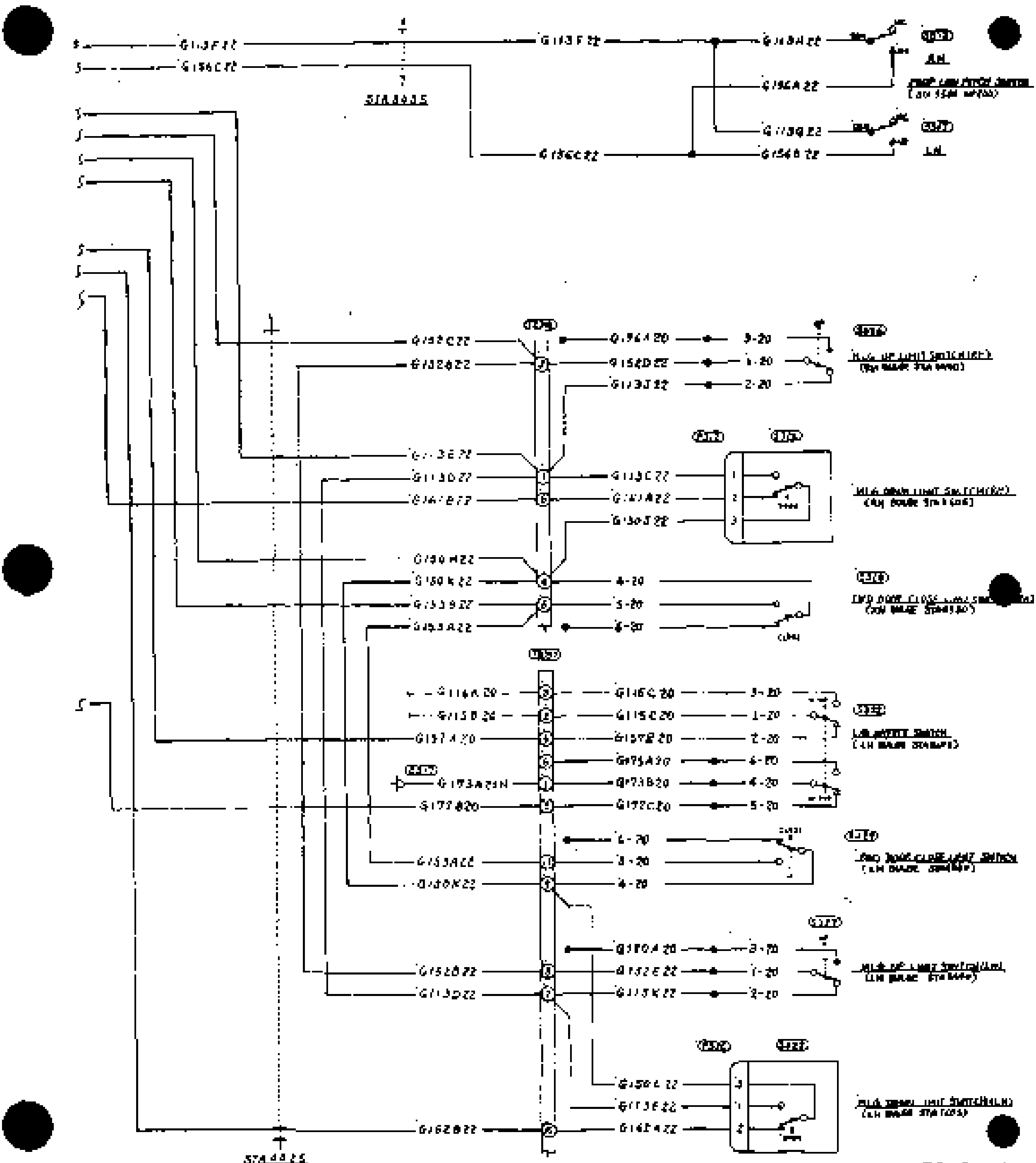



CIRCUIT BREAKER PANEL (AFT)
G150A20



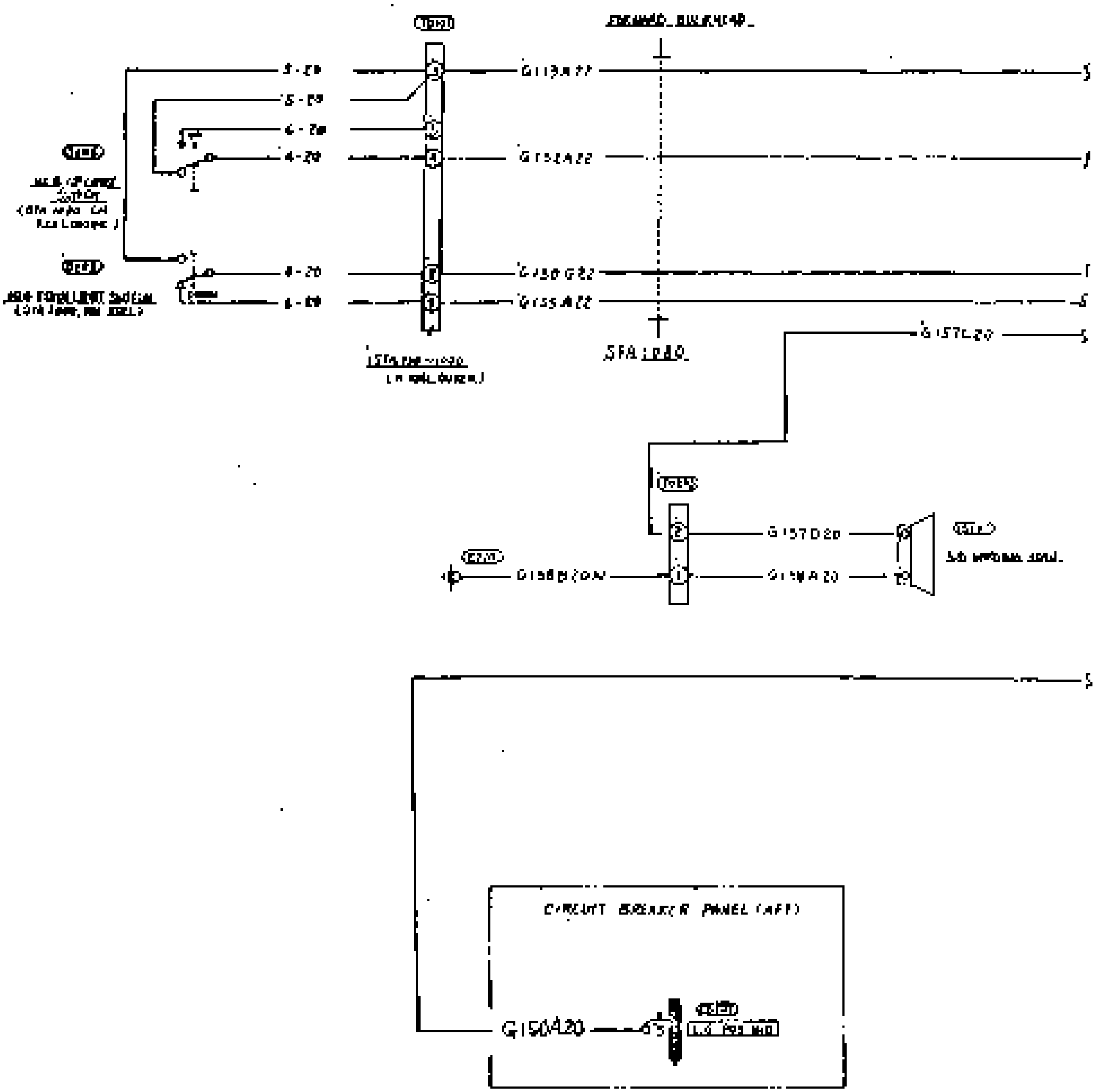
① Aircraft Not Modified By SB018/32-002
② Aircraft Modified By SB018/32-002


LH SWITCH PANEL
M J B RELAY BOX



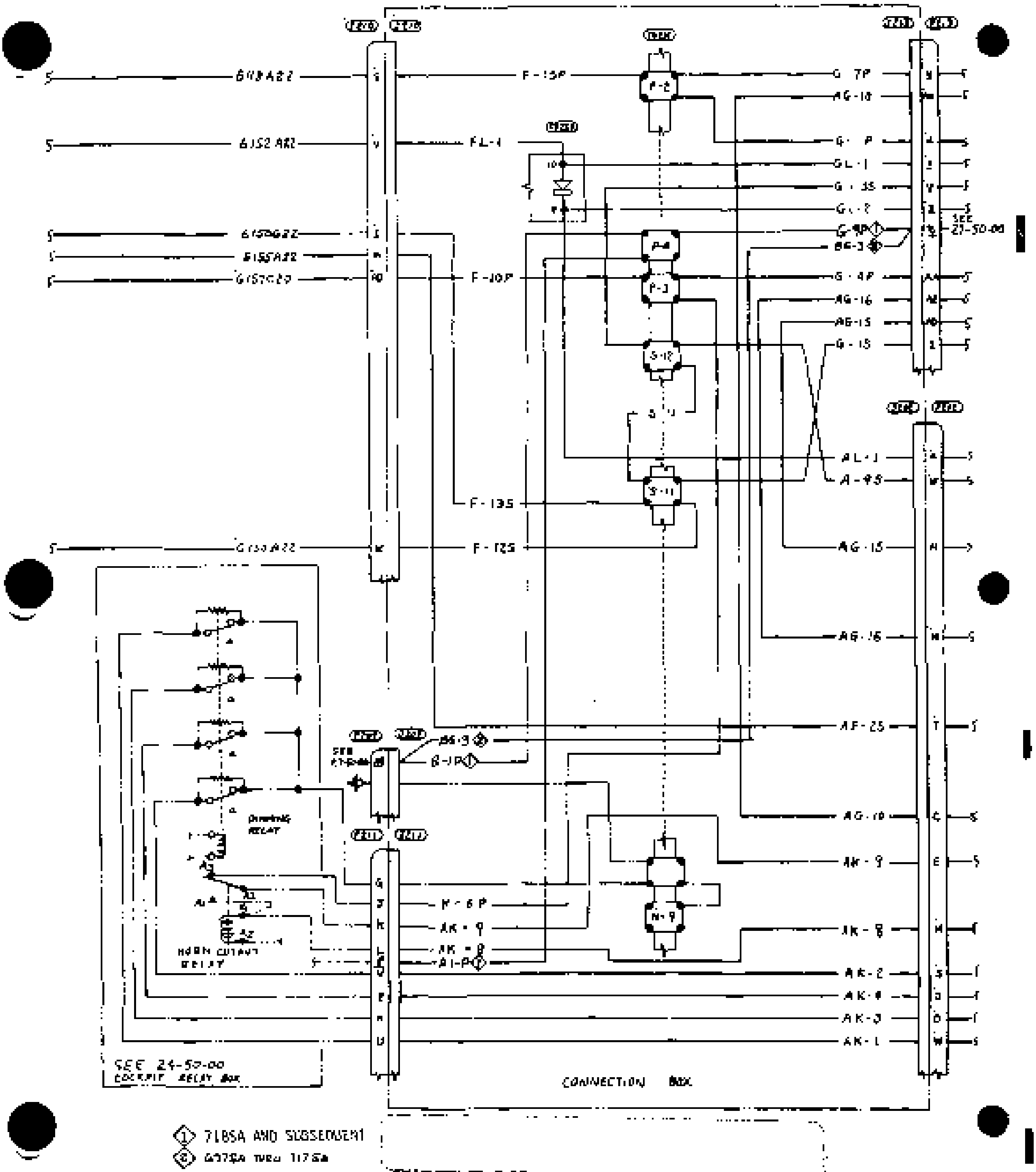
LANDING GEAR POSITION INDICATION

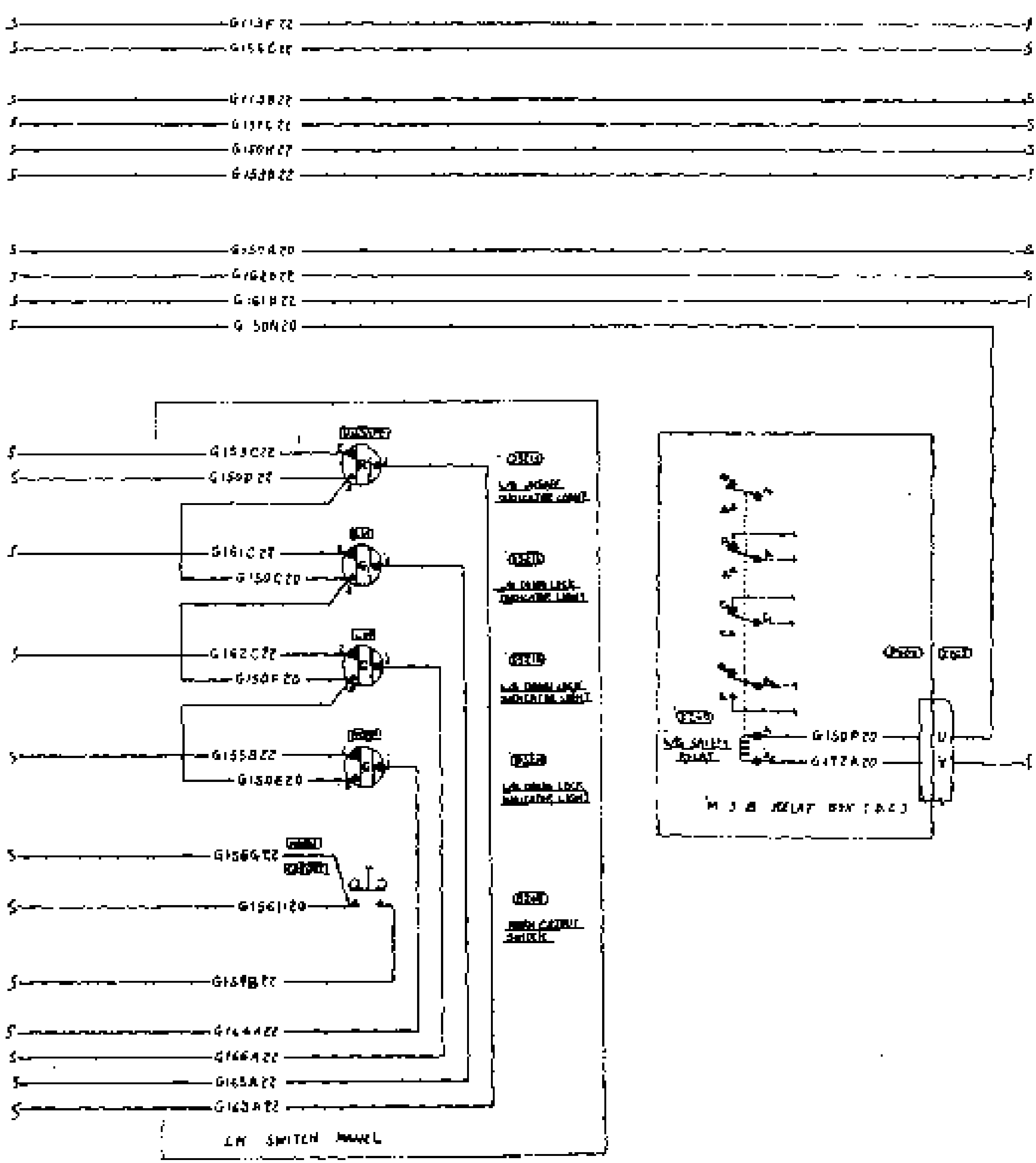
Aircraft S/N 661SA



LANDING GEAR POSITION INDICATION

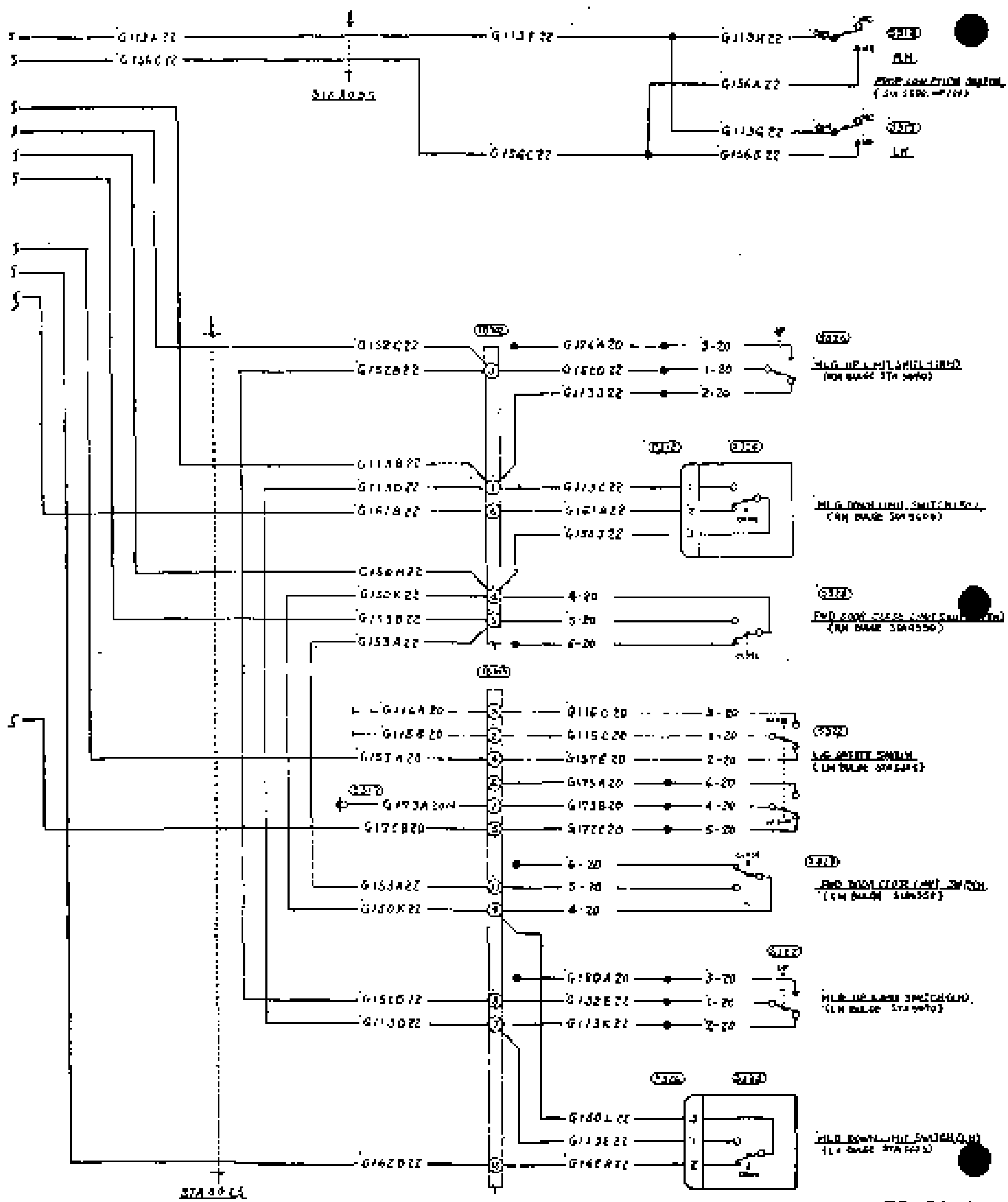
Aircraft S/N 697SA - 730SA





LANDING GEAR POSITION INDICATION

Aircraft S/N 697SA - 730SA

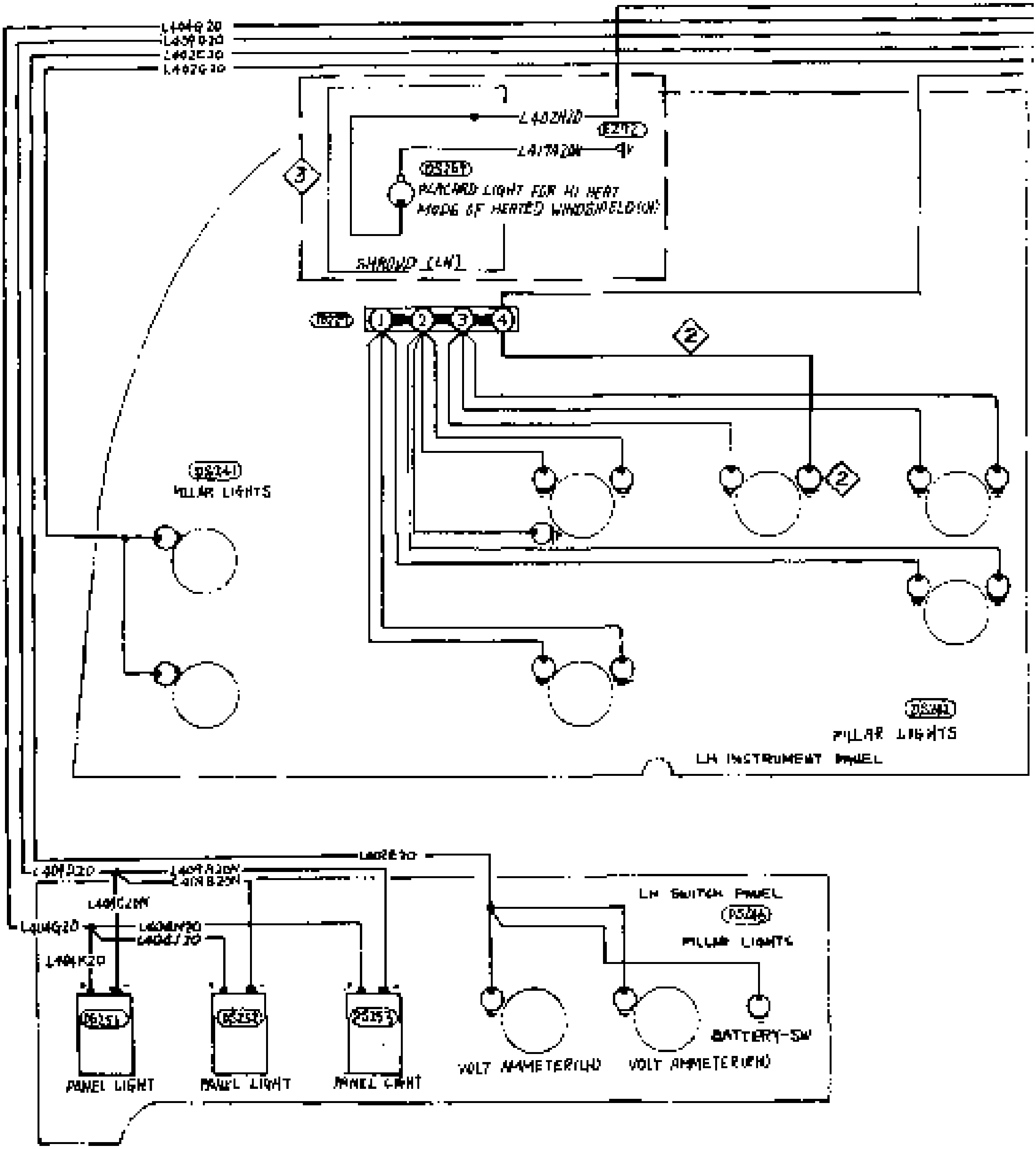


LANDING GEAR POSITION INDICATION

Aircraft S/N 697SA - 730SA

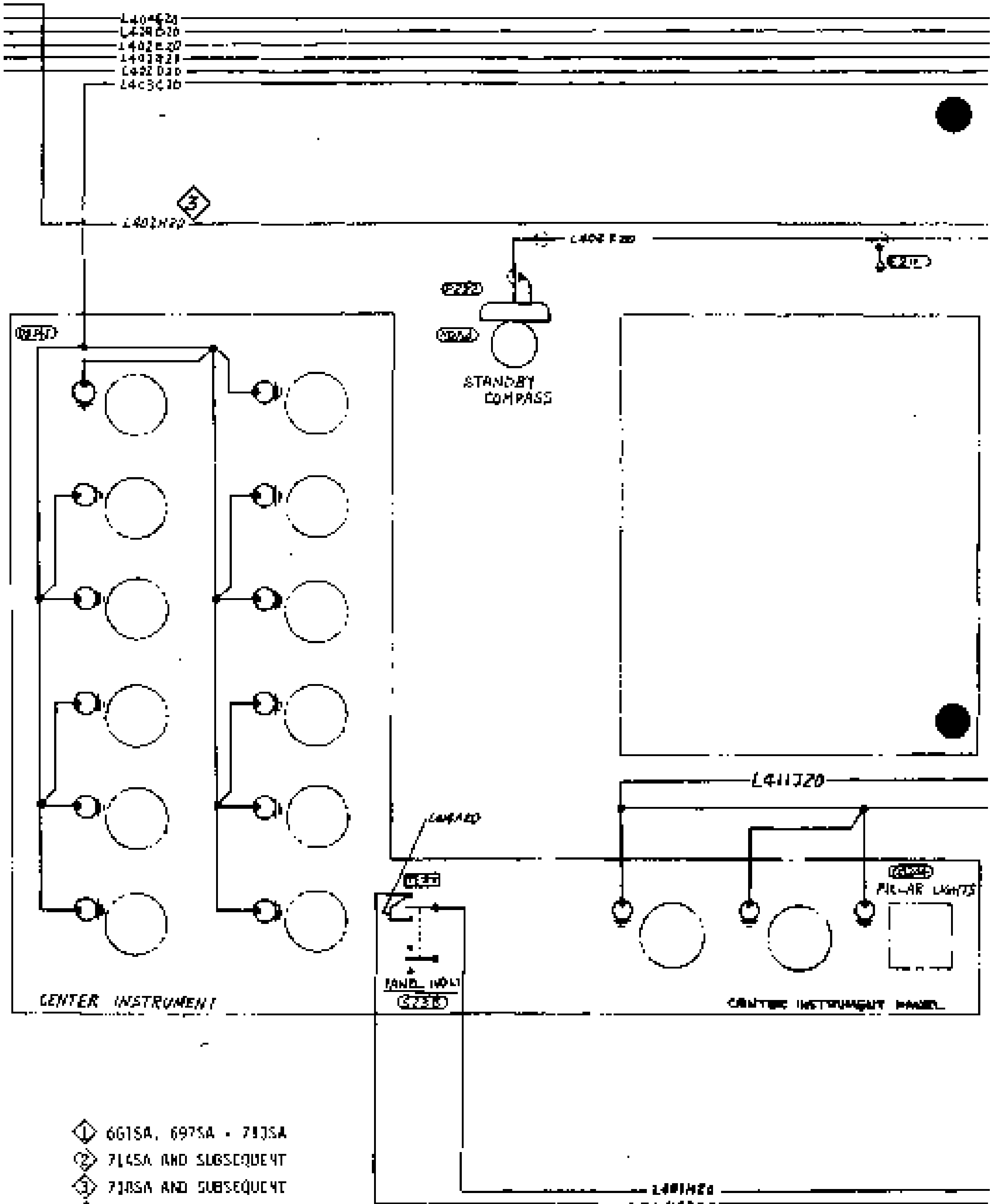
CHAPTER 33

LIGHTS

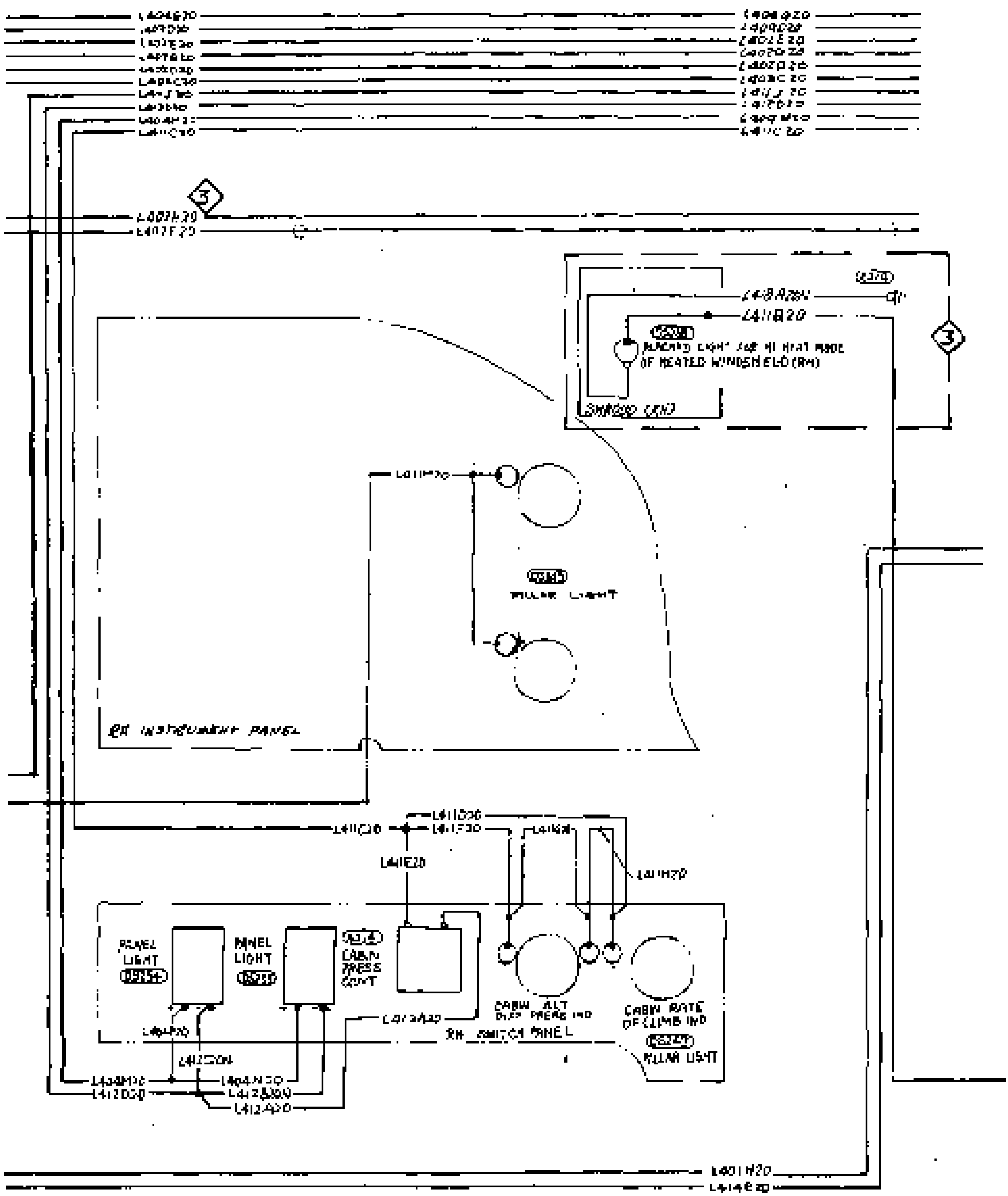


3 661SA, 697SA - 713SA
2 714SA AND SUBSEQUENT
1 710SA AND SUBSEQUENT
4 661SA, 697SA - 714SA

INSTRUMENT & SWITCH PANEL LIGHTING
Aircraft S/N 661SA, 697SA - 730SA



◇1 661SA, 697SA - 713SA
◇2 714SA AND SUBSEQUENT
◇3 718SA AND SUBSEQUENT
◇4 661SA, 697SA - 714SA



1 661SA, 697SA - 713SA

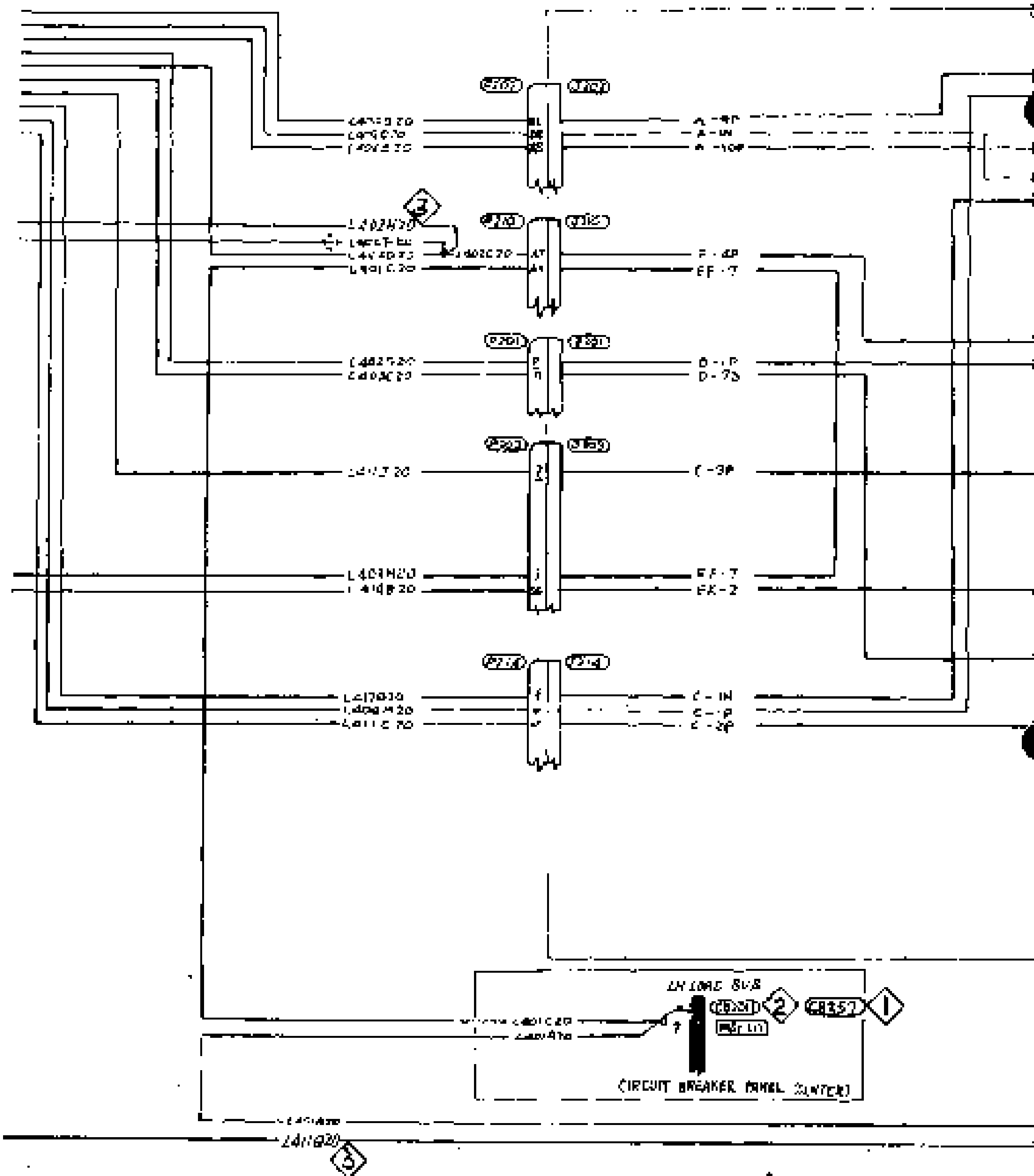
2 714SA AND SUBSEQUENT

3 719SA AND SUBSEQUENT

4 661SA, 697SA - 734SA

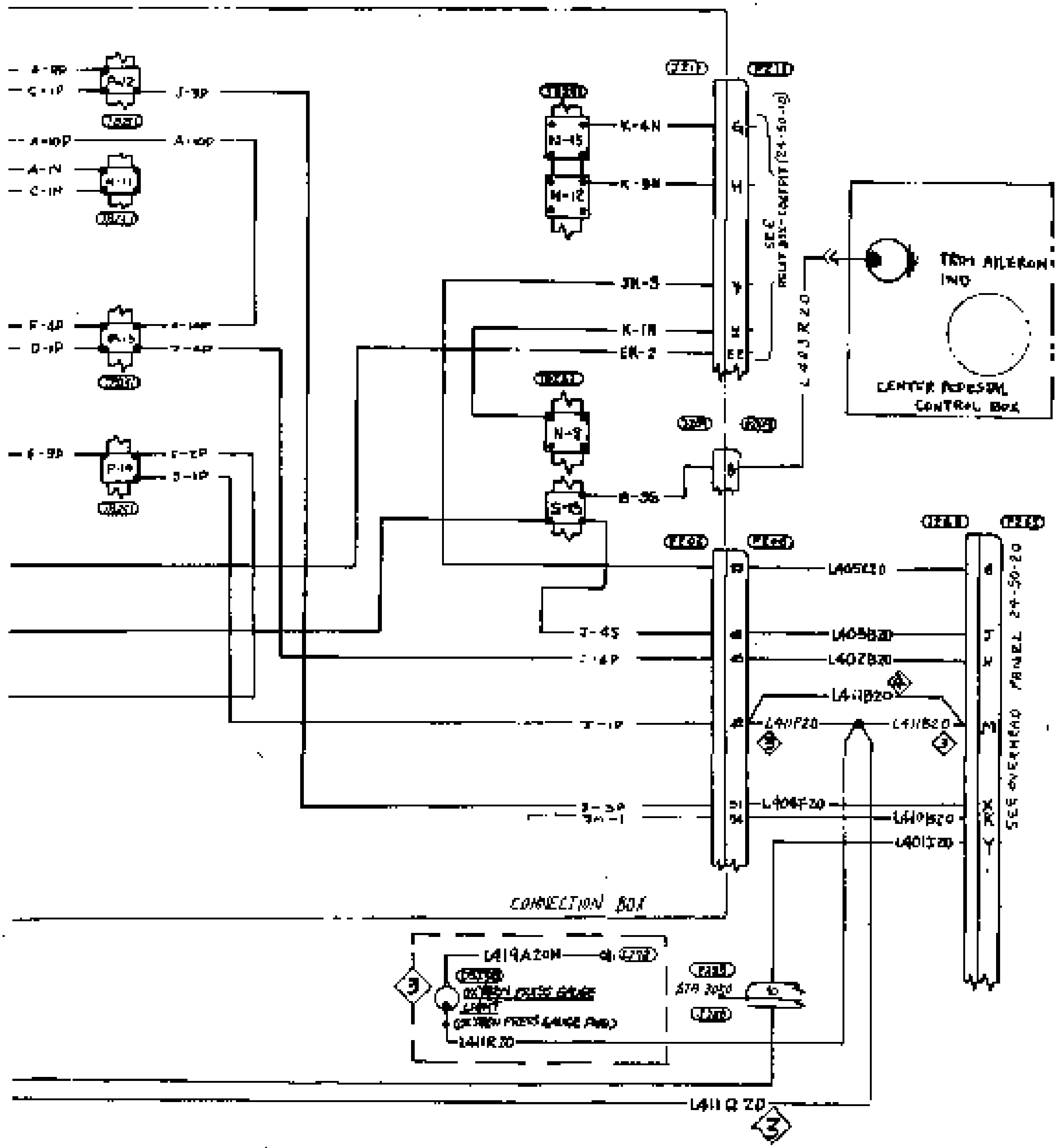
INSTRUMENT & SWITCH PANEL LIGHTING

Aircraft S/N 661SA, 697SA - 730SA



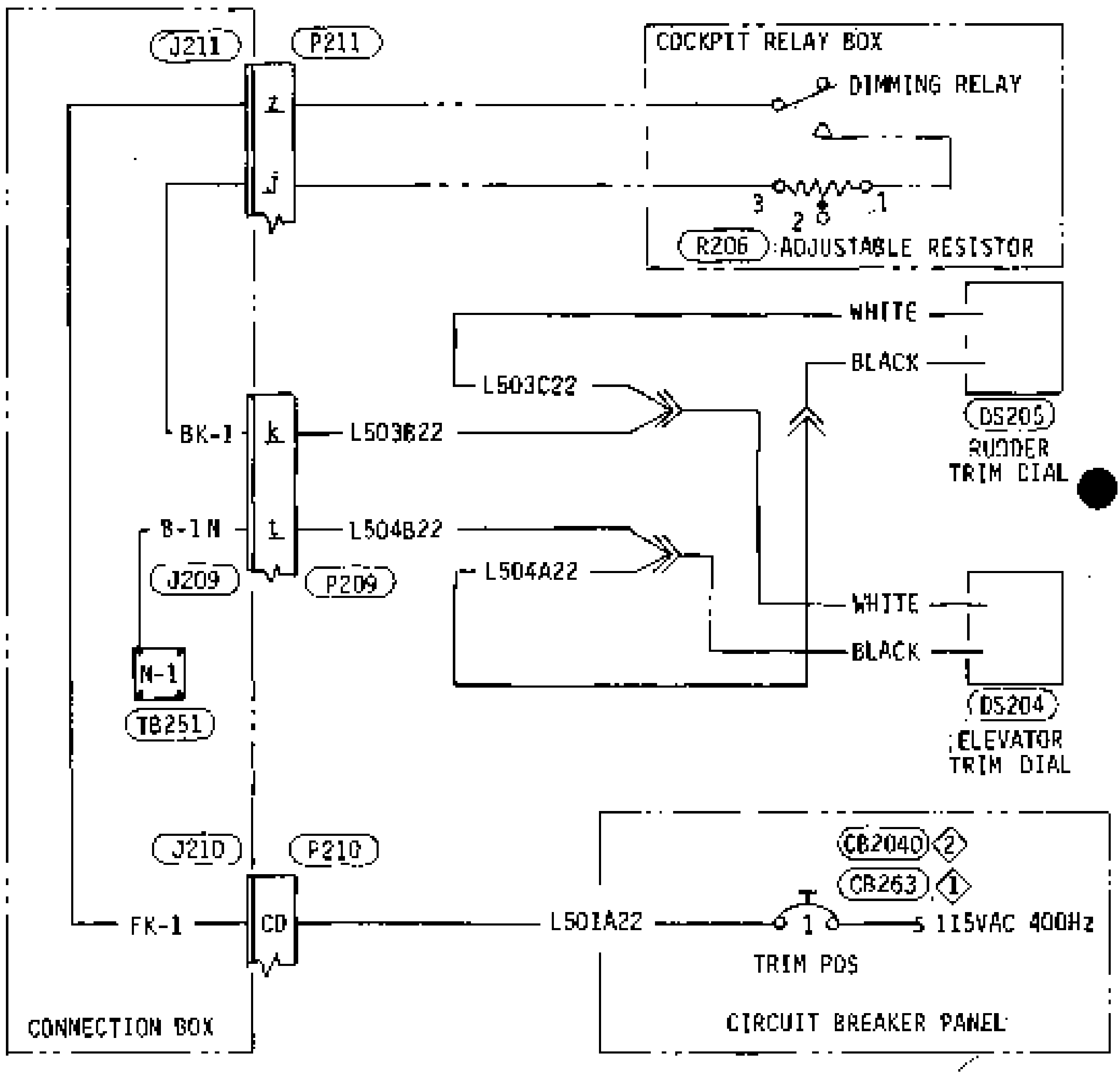
1 661SA, 697SA - 713SA
2 714SA AND SUBSEQUENT
3 718SA AND SUBSEQUENT
4 661SA, 697SA - 714SA

INSTRUMENT & SWITCH PANEL LIGHTING
Aircraft S/N 661SA, 697SA - 730SA



1. 661SA, 697SA - 713SA
2. 714SA AND SUBSEQUENT
3. 718SA AND SUBSEQUENT
4. 661SA, 697SA - 714SA

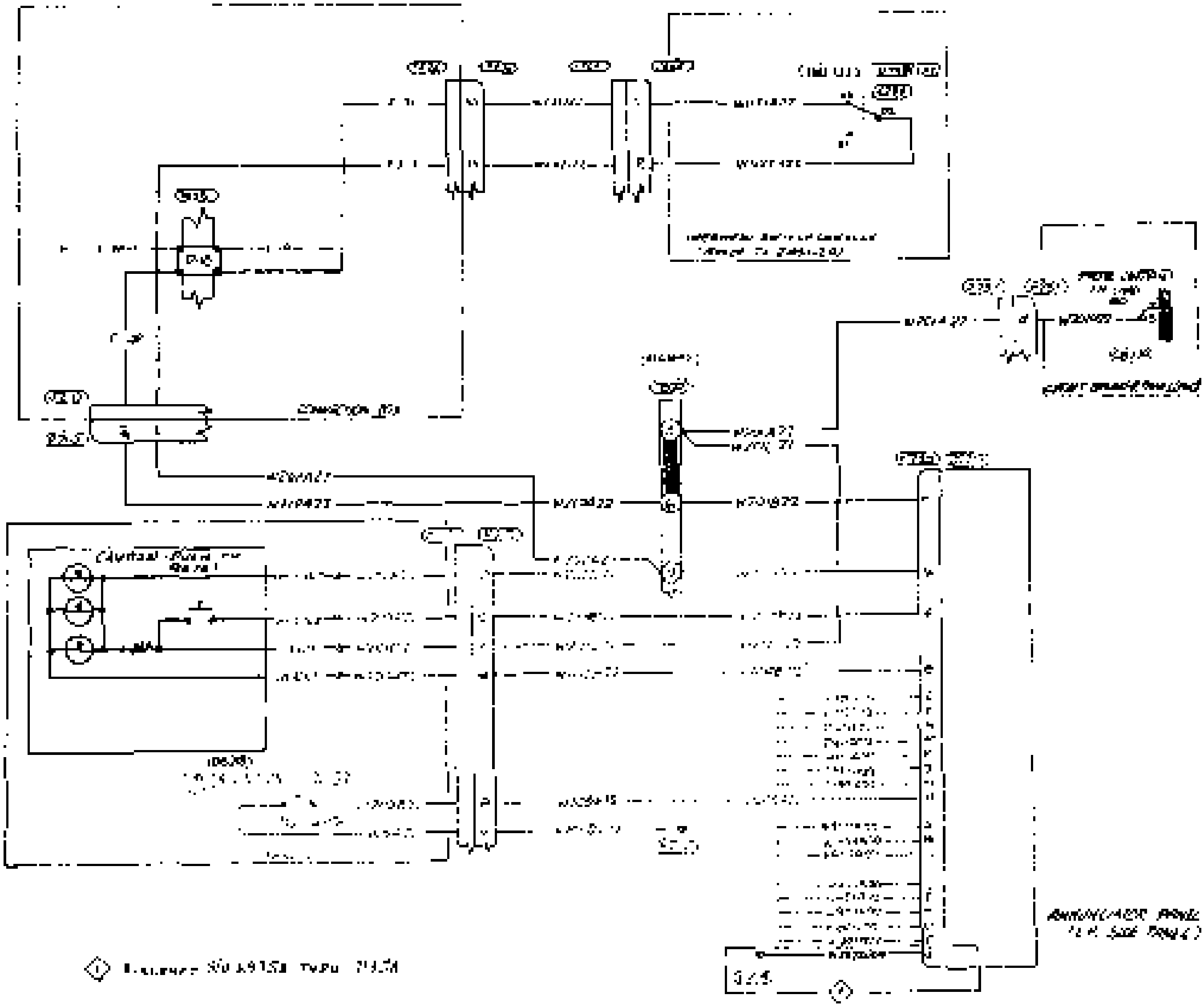
INSTRUMENT & SWITCH PANEL LIGHTING
Aircraft S/N 661SA, 697SA - 730SA



1 Aircraft S/N 661SA, 697SA thru 713SA

2 Aircraft S/N 714SA thru 730SA

TRIM INDICATOR LIGHTING
Aircraft S/N 661SA, 697SA - 730SA

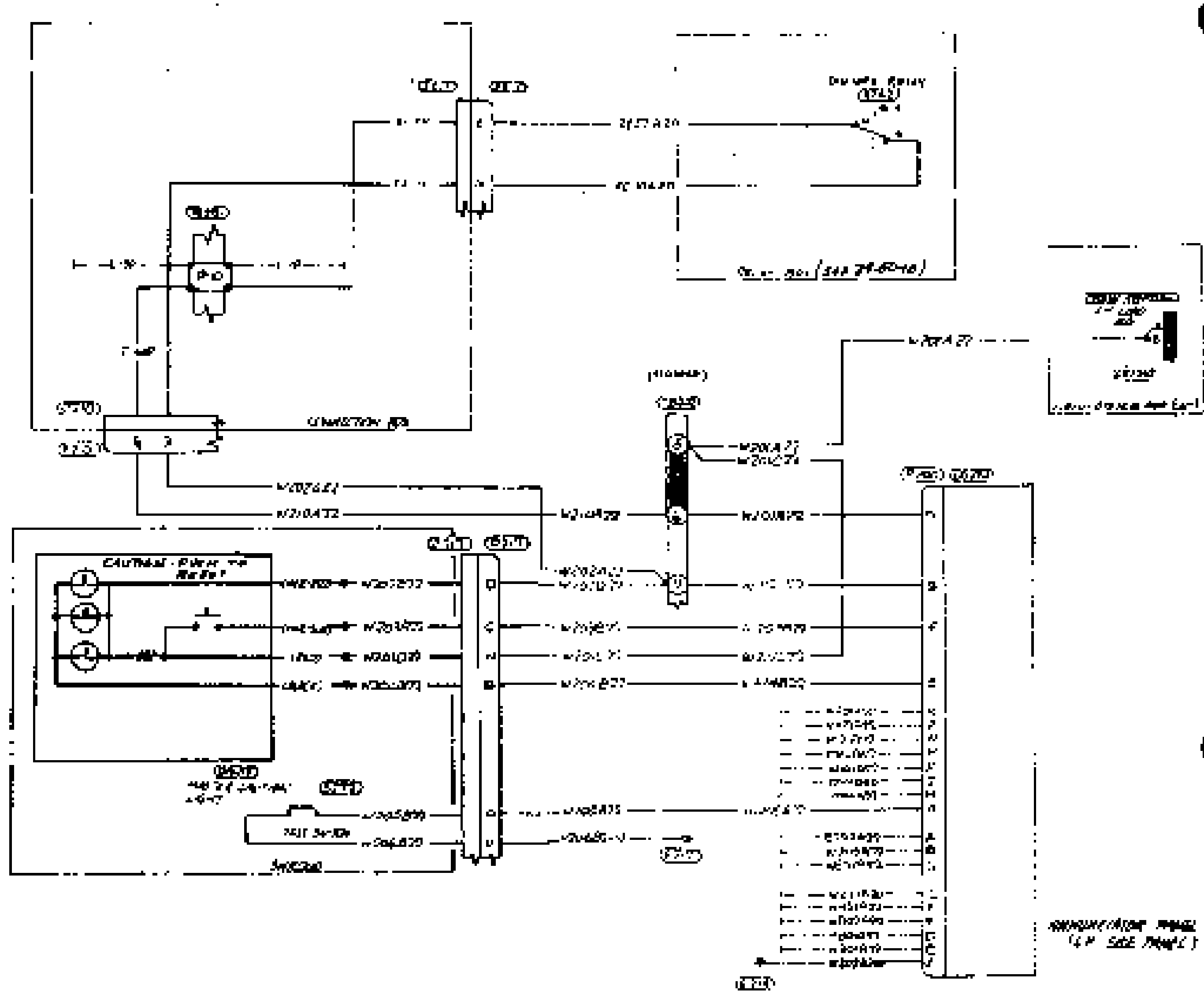


ORIGINAL AS RECEIVED BY ATP

MASTER CAUTION WARNING LIGHTING

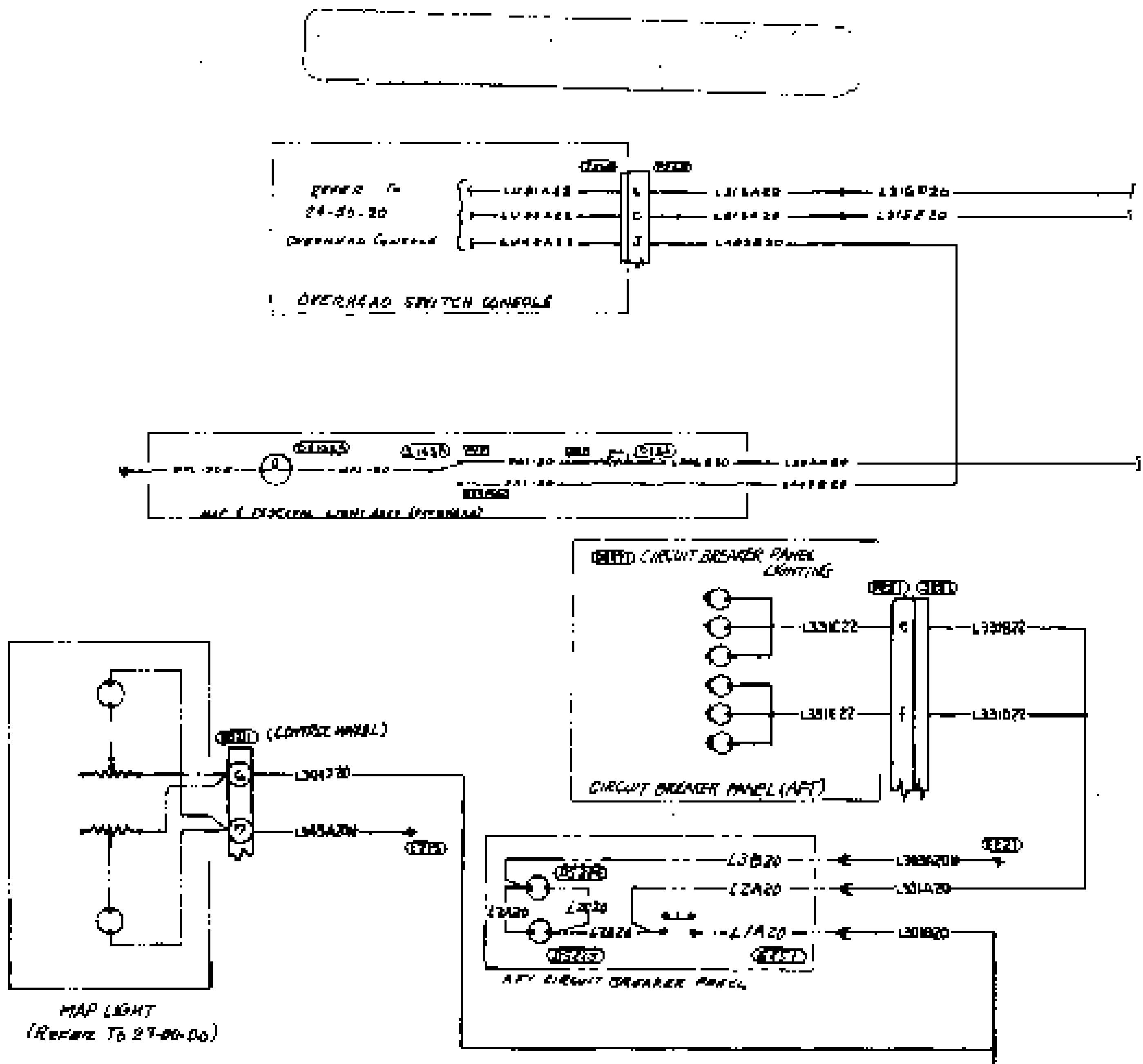
Aircraft S/N 661SA, 697SA - 713SA

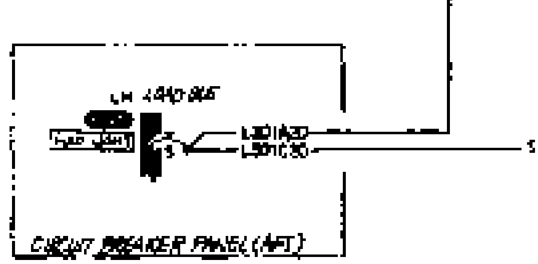
Rev No 1
Aug 1/79



MASTER CAUTION WARNING LIGHTING

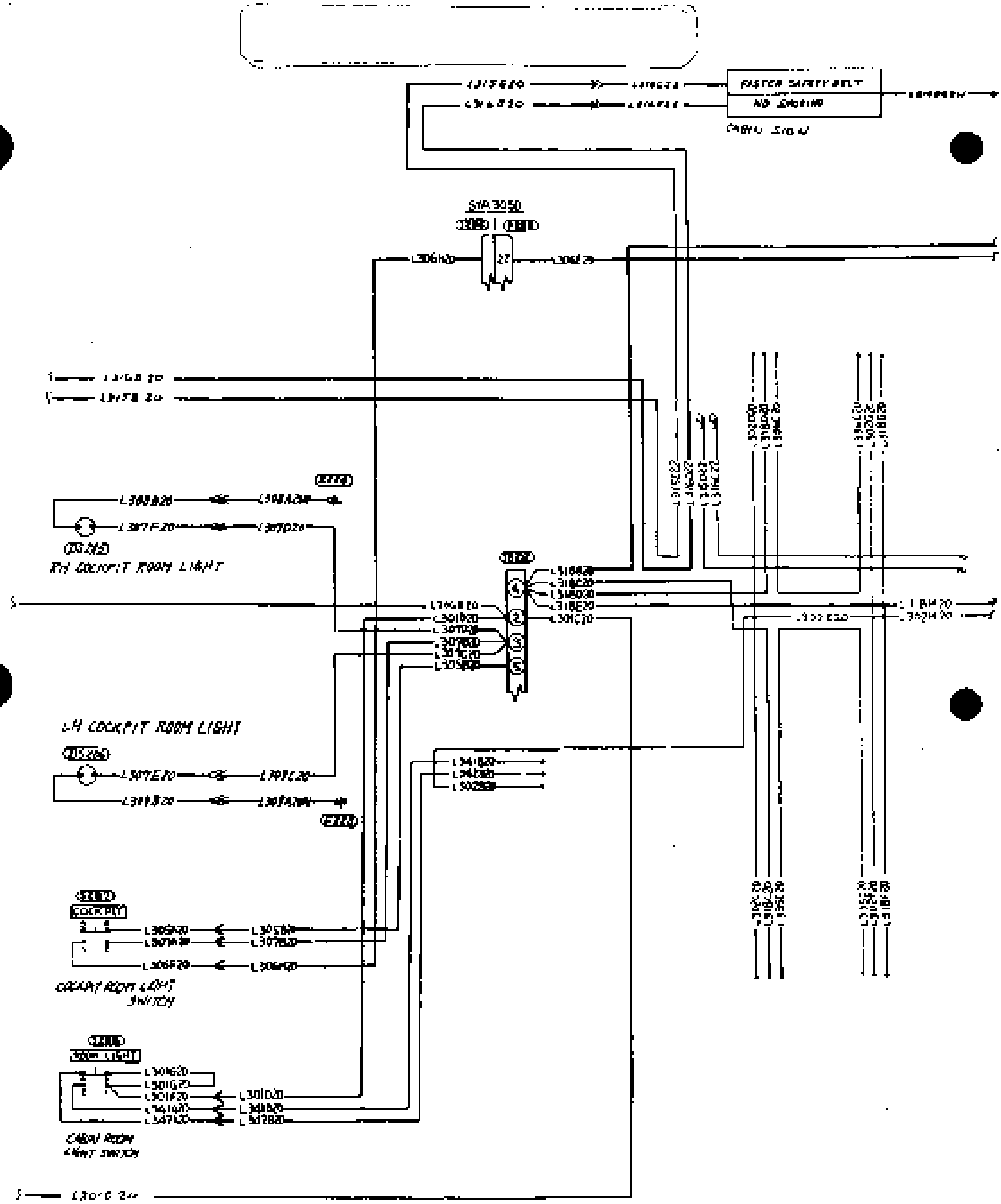
Aircraft S/N 714SA - 730SA





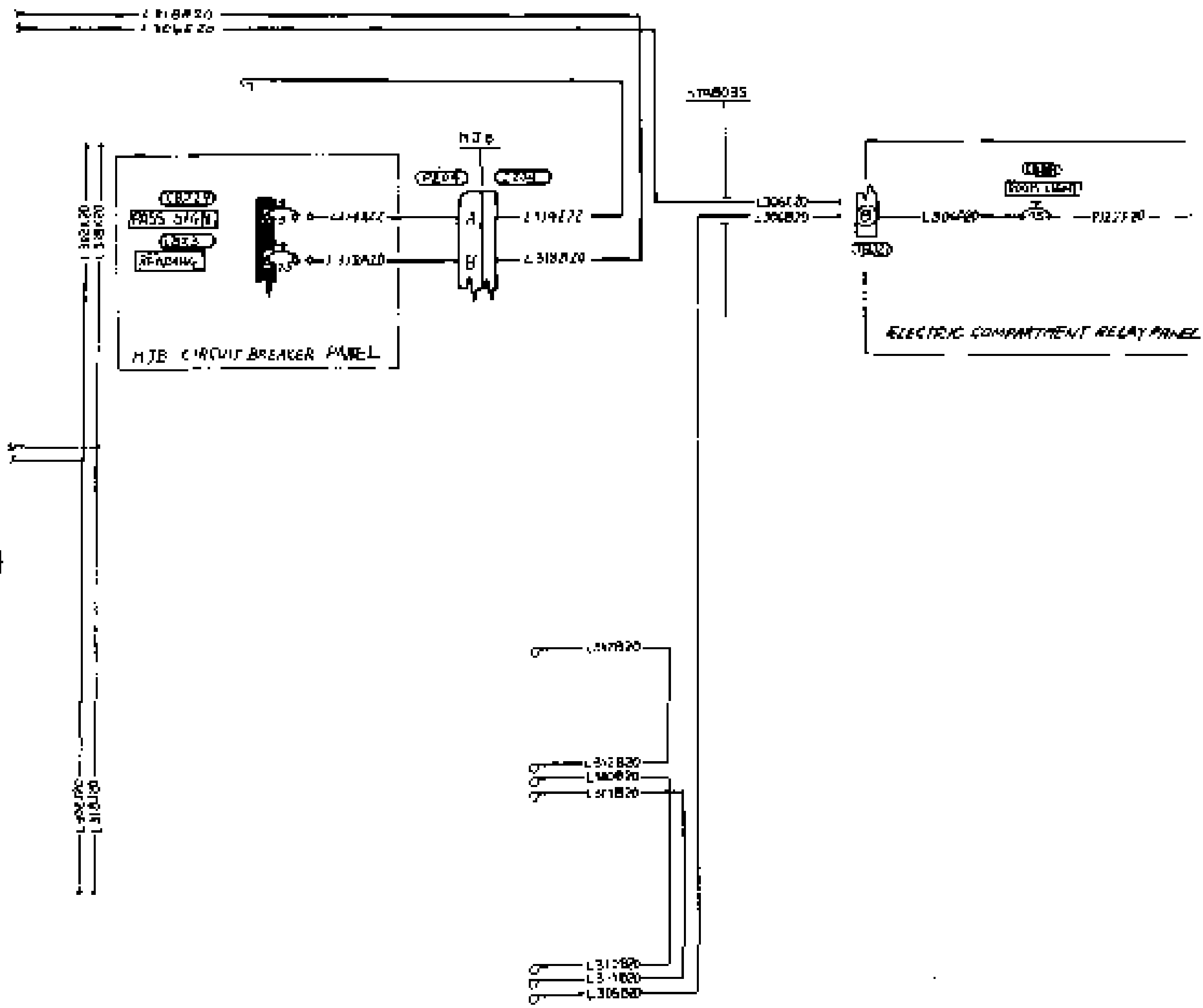
Applicable to aircraft S/N 661SA when used in conjunction with MAI Drawing 935A-4001.

NOTE: See MAI Drawing 936A-4028 for aircraft S/N 697SA and subsequent.



Applicable to aircraft S/N 661SA when used in conjunction with MAI Drawing 935A-4001.

NOTE: See MAI Drawing 936A-4028 for aircraft S/N 697SA and subsequent.

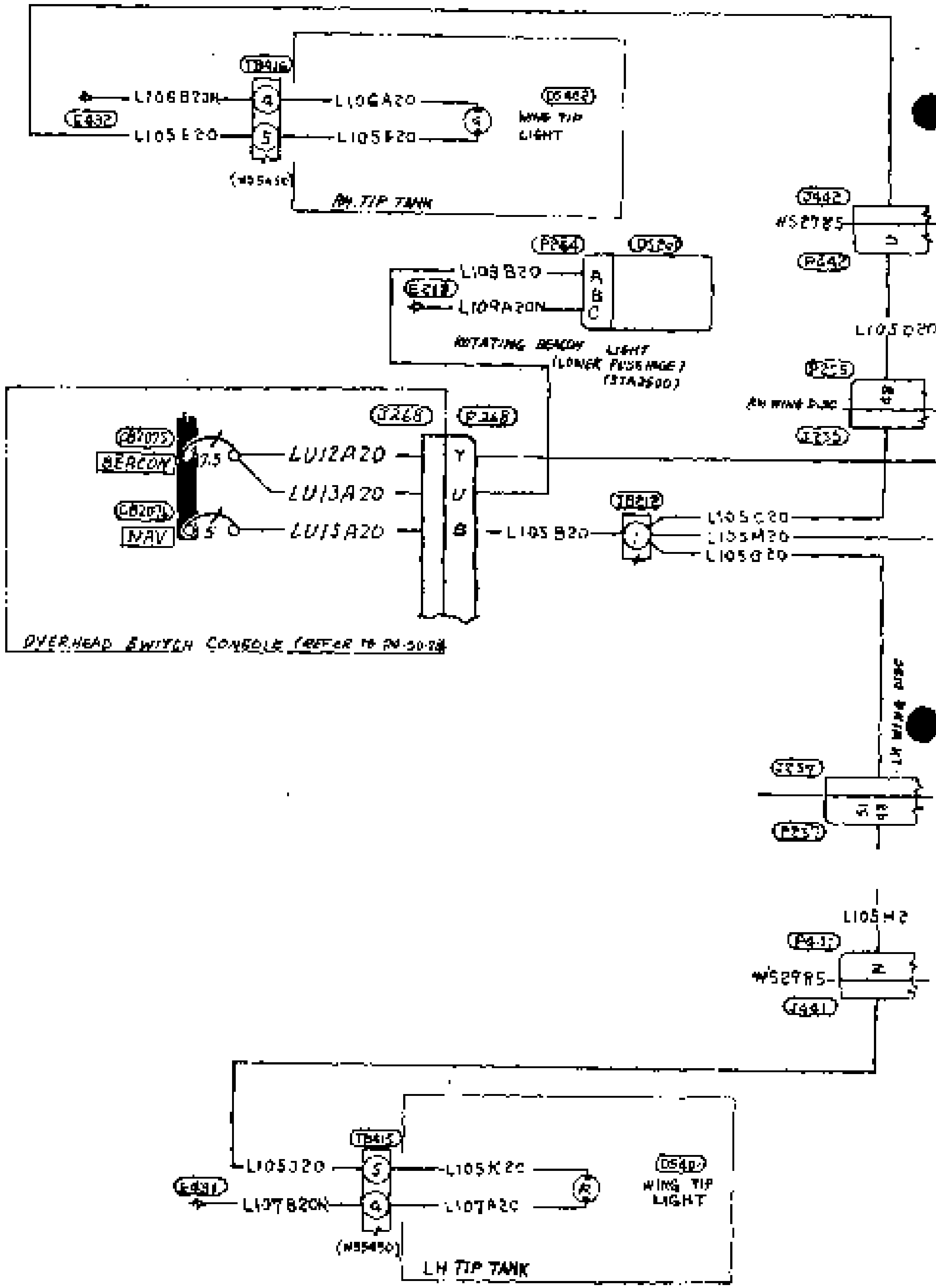


Applicable to aircraft S/N 661SA when used in conjunction with MAI Drawing 935A-4001.

NOTE: See MAI Drawing 936A-4028 for aircraft S/N 697SA and subsequent.

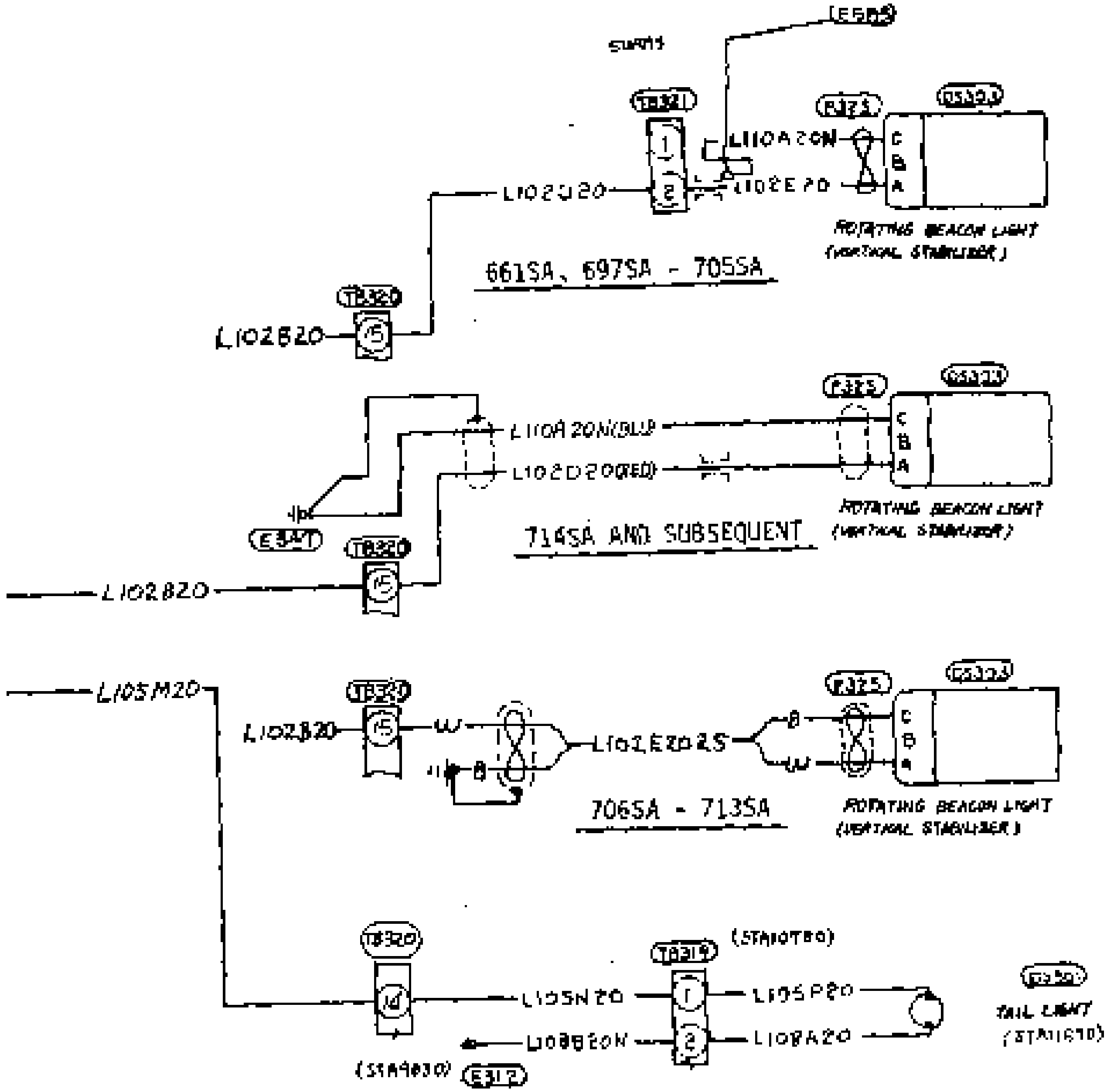
ROOM LIGHTING

Aircraft S/N 661SA, 697SA - 730SA



NAVIGATION LIGHTING

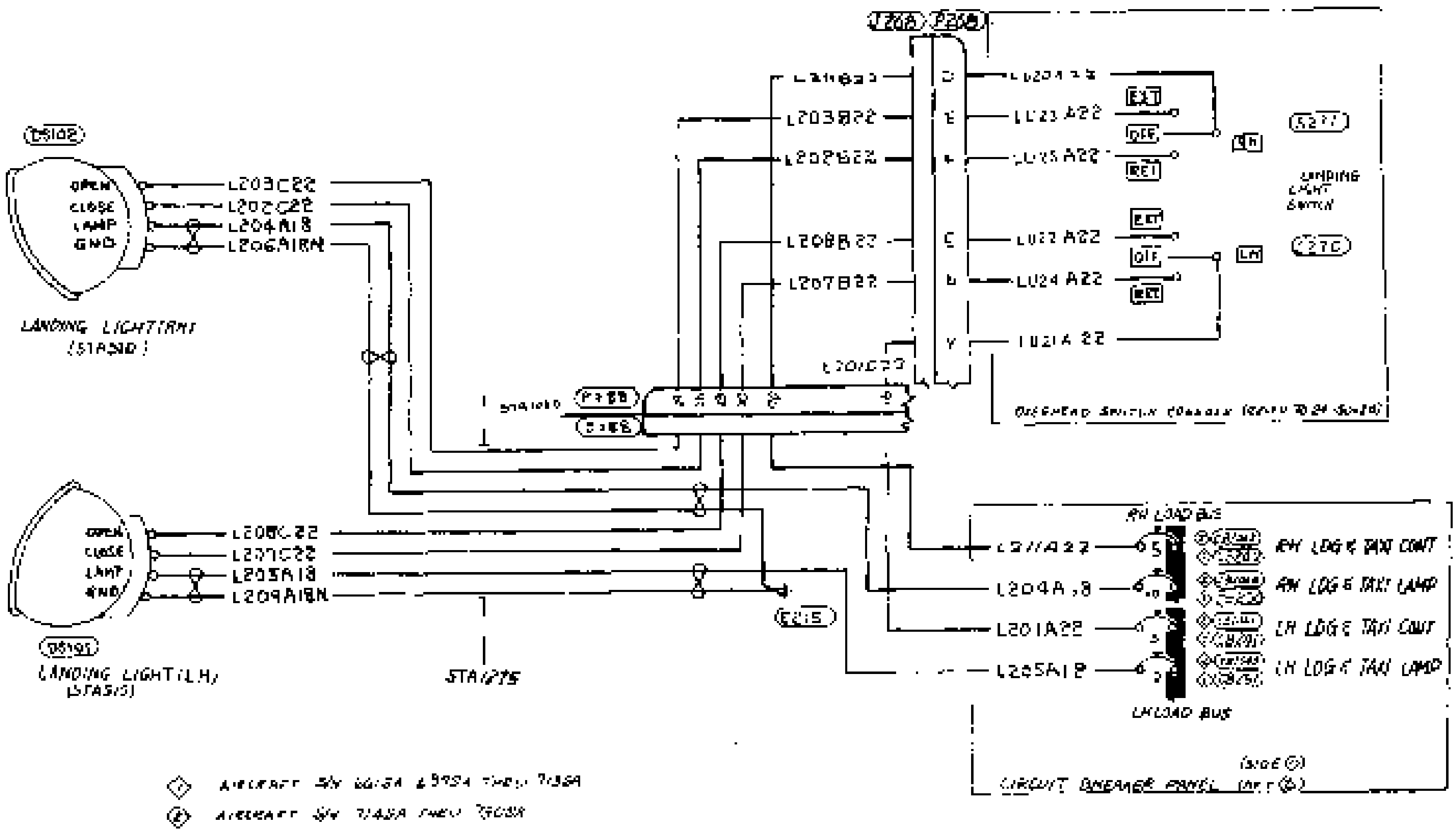
Aircraft S/N 661SA, 697SA - 730SA

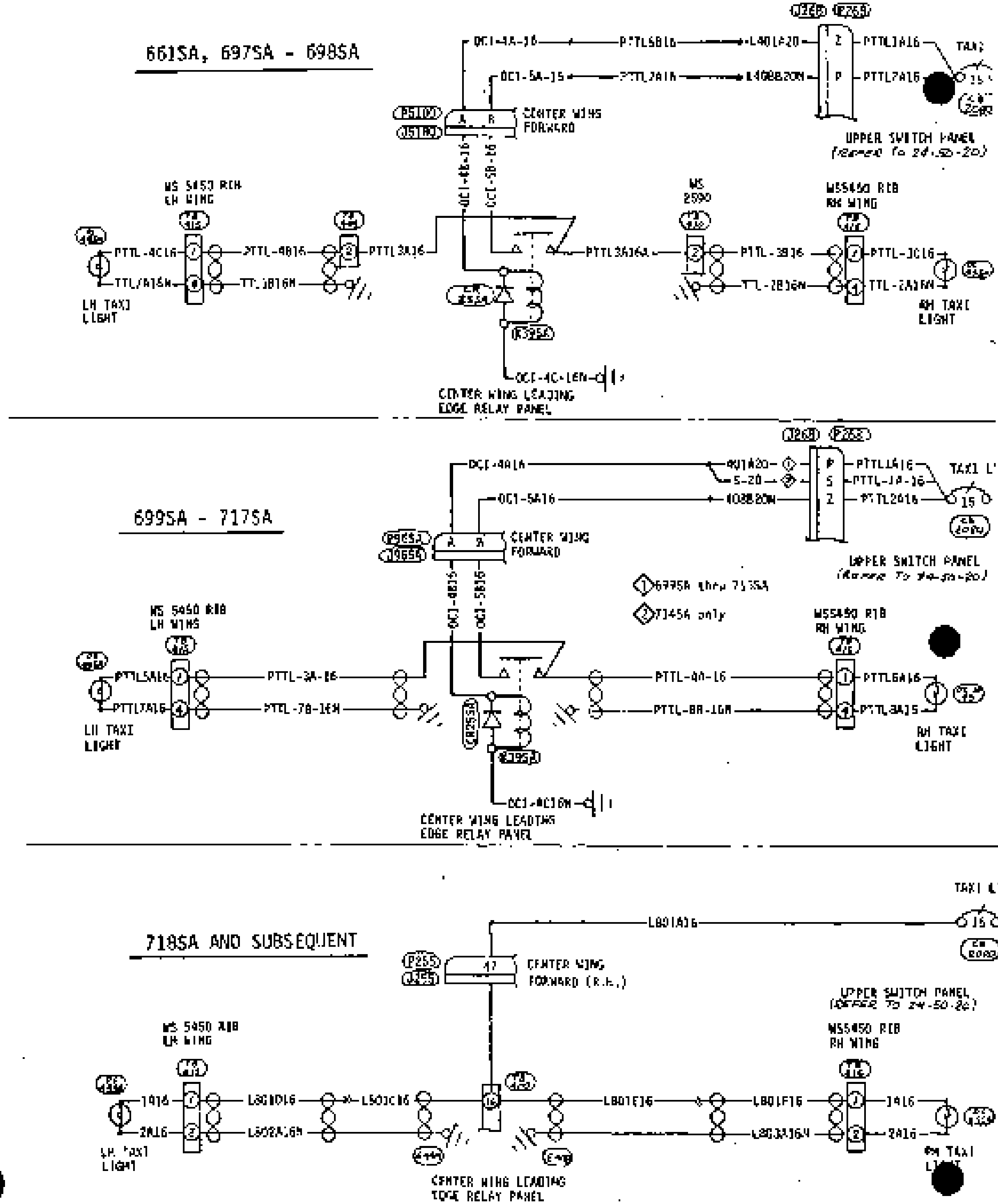


L102B20
L102D20
L110A20N
661SA, 697SA - 705SA
ROTATING BEACON LIGHT
(VERTICAL STABILIZER)
L110A20N(BLU)
L102D20(RED)
L102B20
714SA AND SUBSEQUENT
L105M20
L102B20
L102E20
706SA - 713SA
L105N20
L105P20
L108B20N
L108A20
TAIL LIGHT

FUSELAGE LANDING LIGHTING

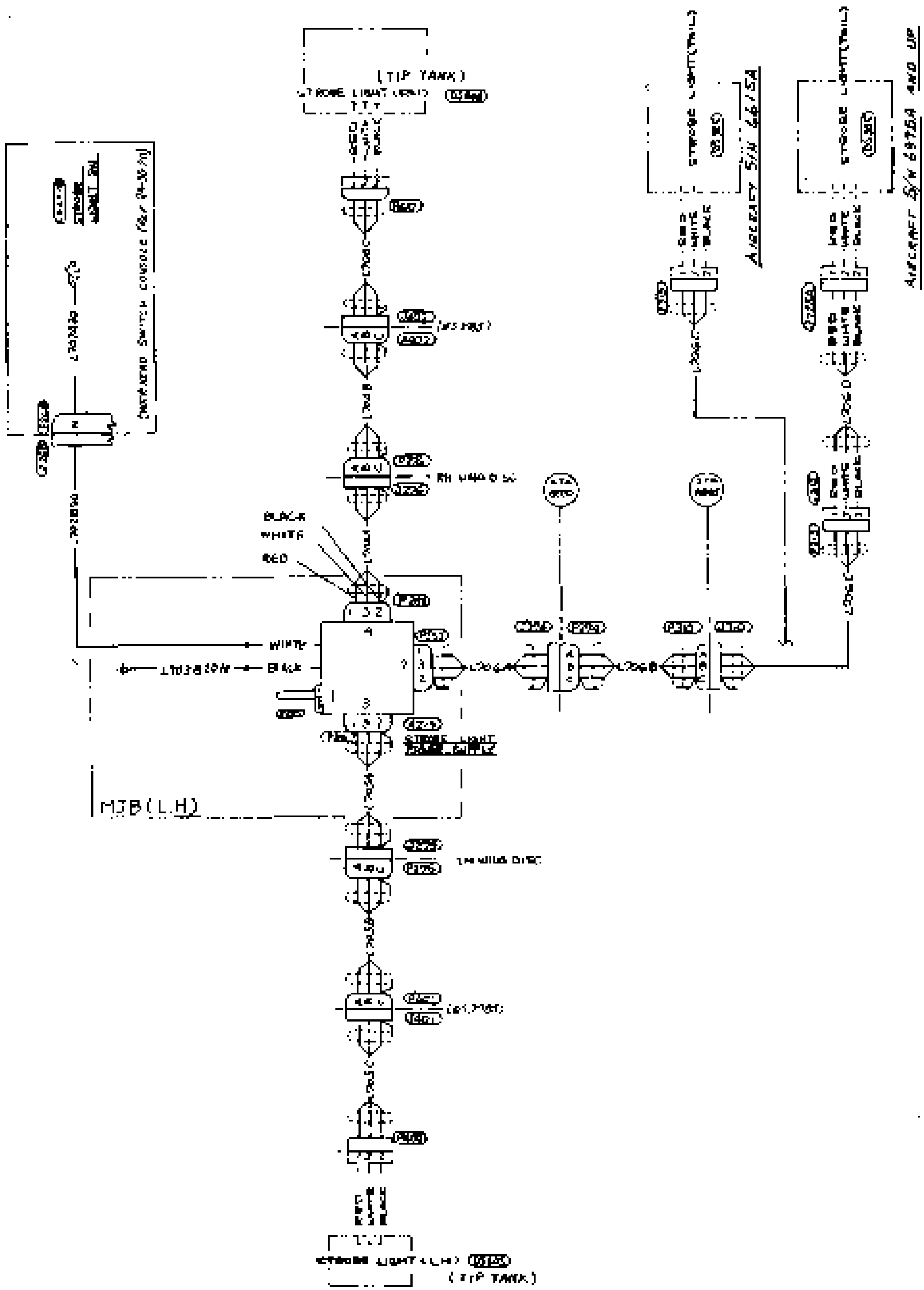
Aircraft S/N 661SA, 697SA - 730SA





TIP TANK TAXI LIGHTING

Aircraft S/N 661SA, 697SA - 730SA



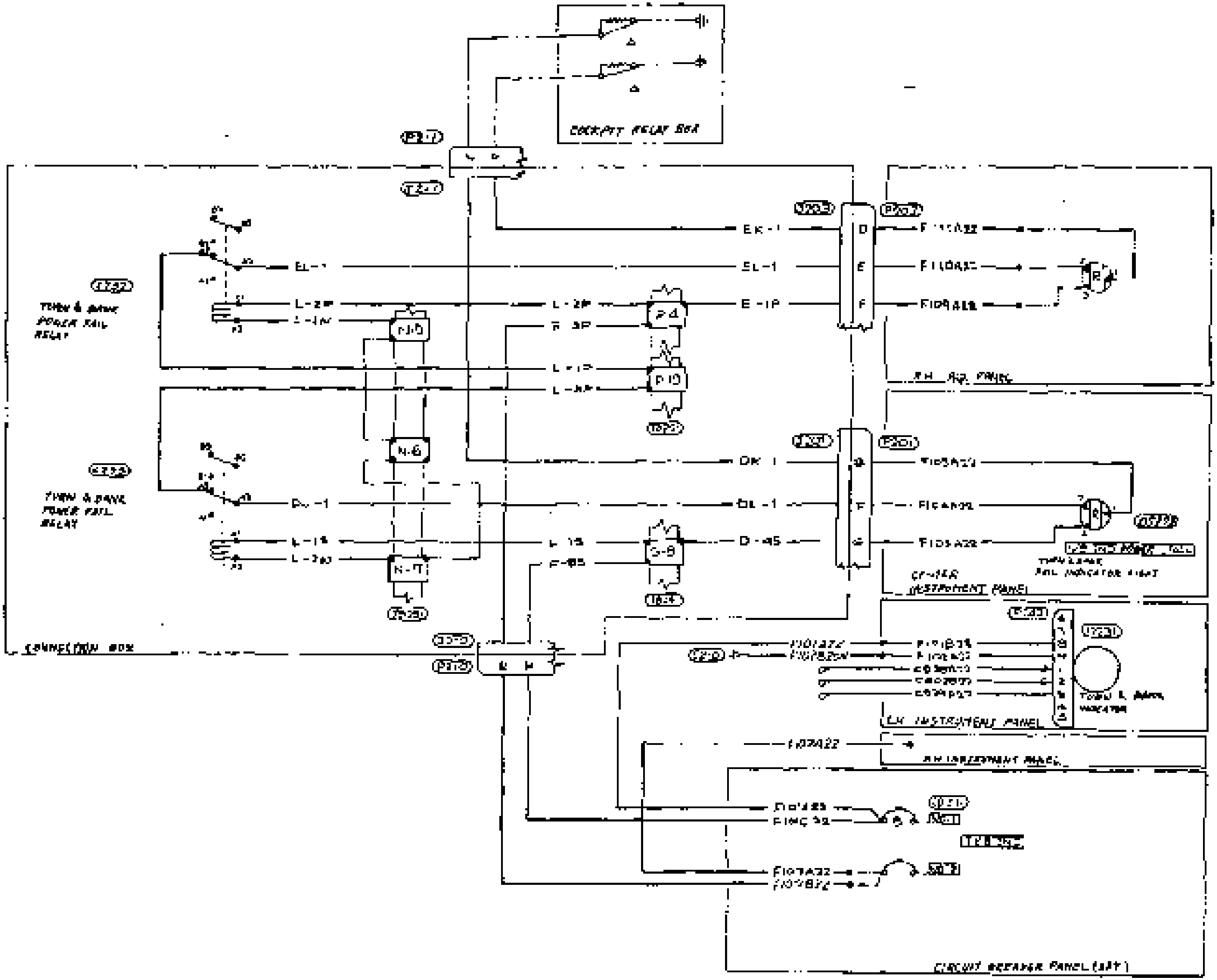
STROBE LIGHTING

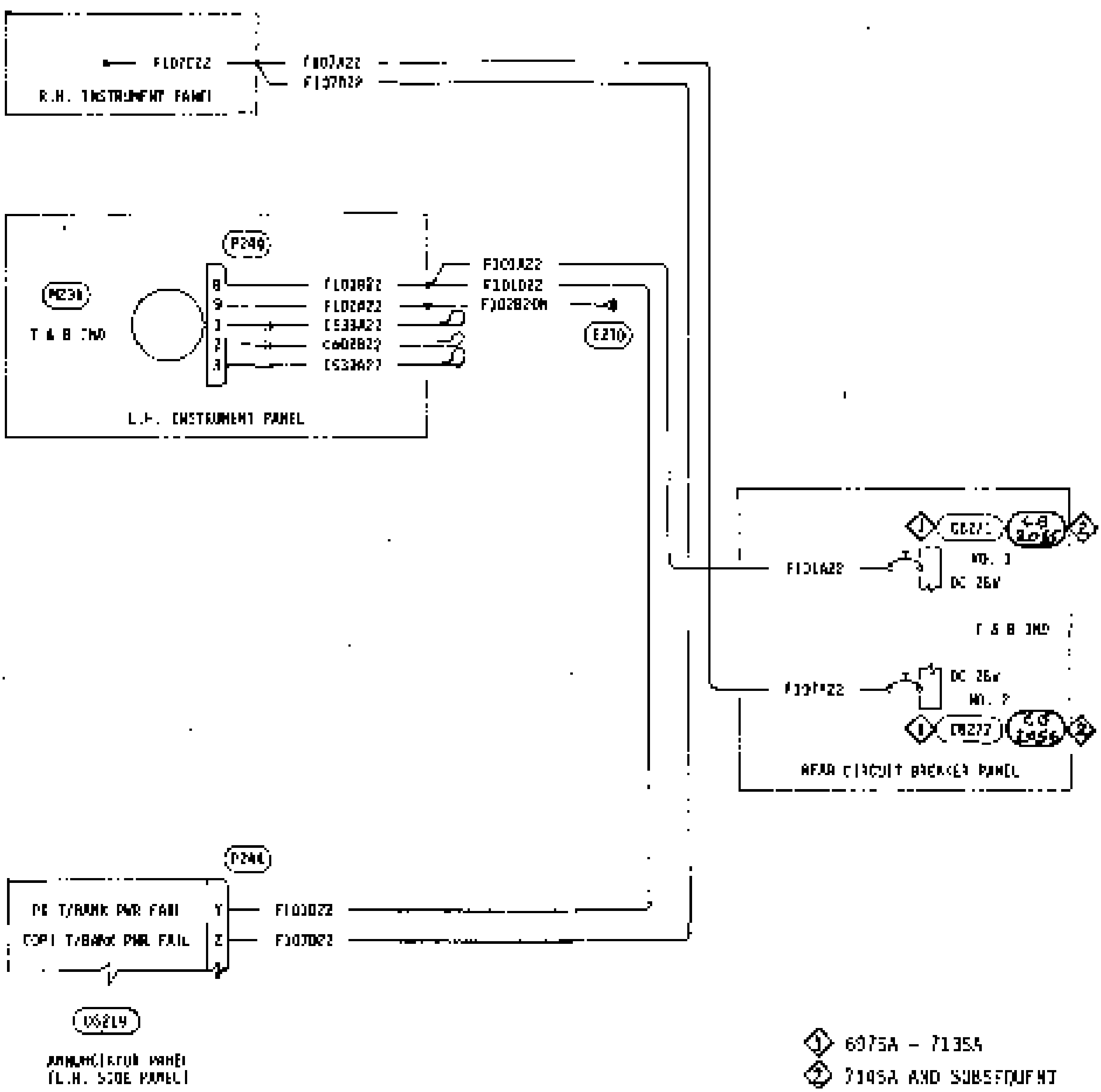
Aircraft S/N 661SA, 697SA - 730SA

CHAPTER 34

NAVIGATION

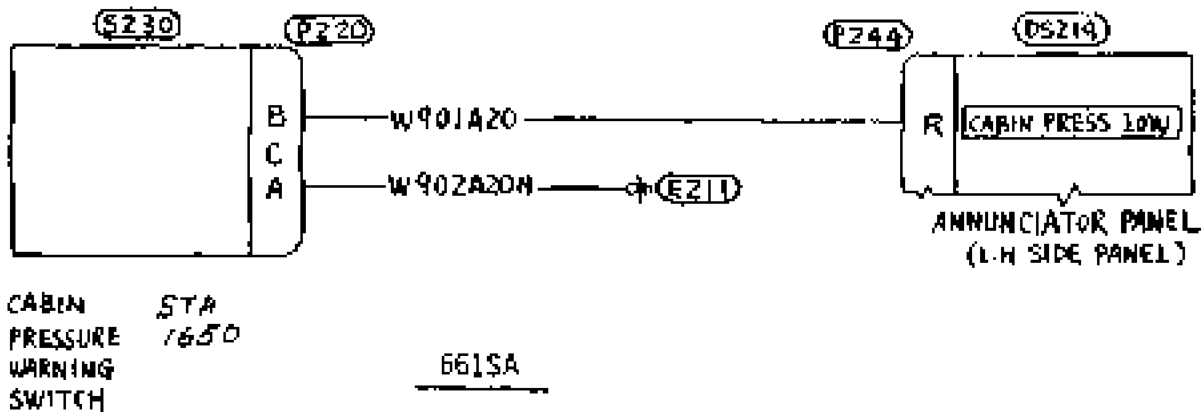
TURN & BANK INDICATOR
Aircraft S/N 661SA



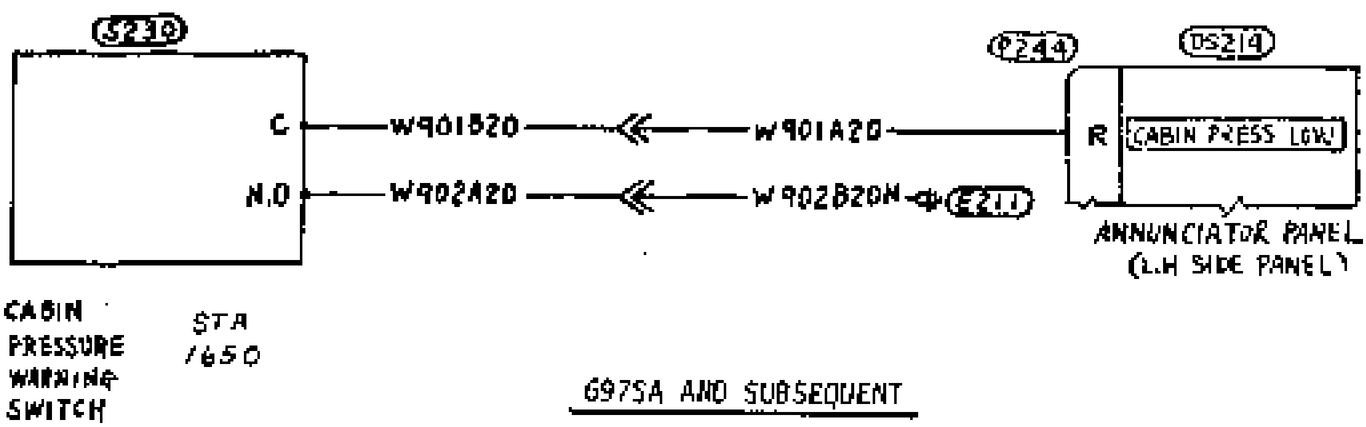

R.H. INSTRUMENT PANEL
P249
M238
T & B IND
L.H. INSTRUMENT PANEL
E210
T & B IND
NO. 1
DC 26V
DC 26V
NO. 2
REAR CIRCUIT BREAKER PANEL
P244
PE T/BANK PWR FAIL
COPI T/BANK PWR FAIL
ANNUNCIATOR PANEL
(L.H. SIDE PANEL)
6975A - 7135A
7145A AND SUBSEQUENT

CHAPTER 36

PNEUMATIC



661SA

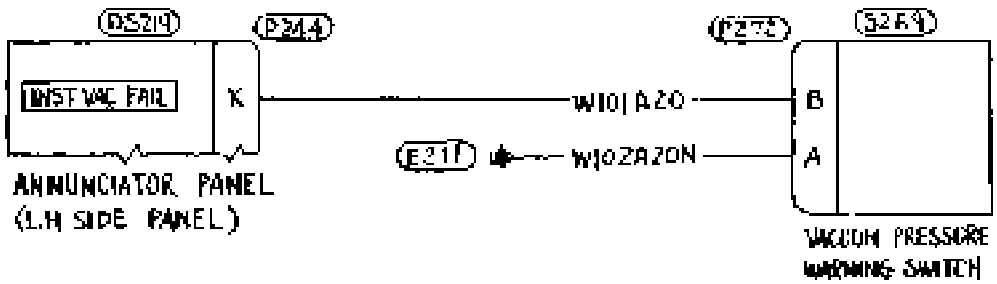


697SA AND SUBSEQUENT

CHAPTER

37

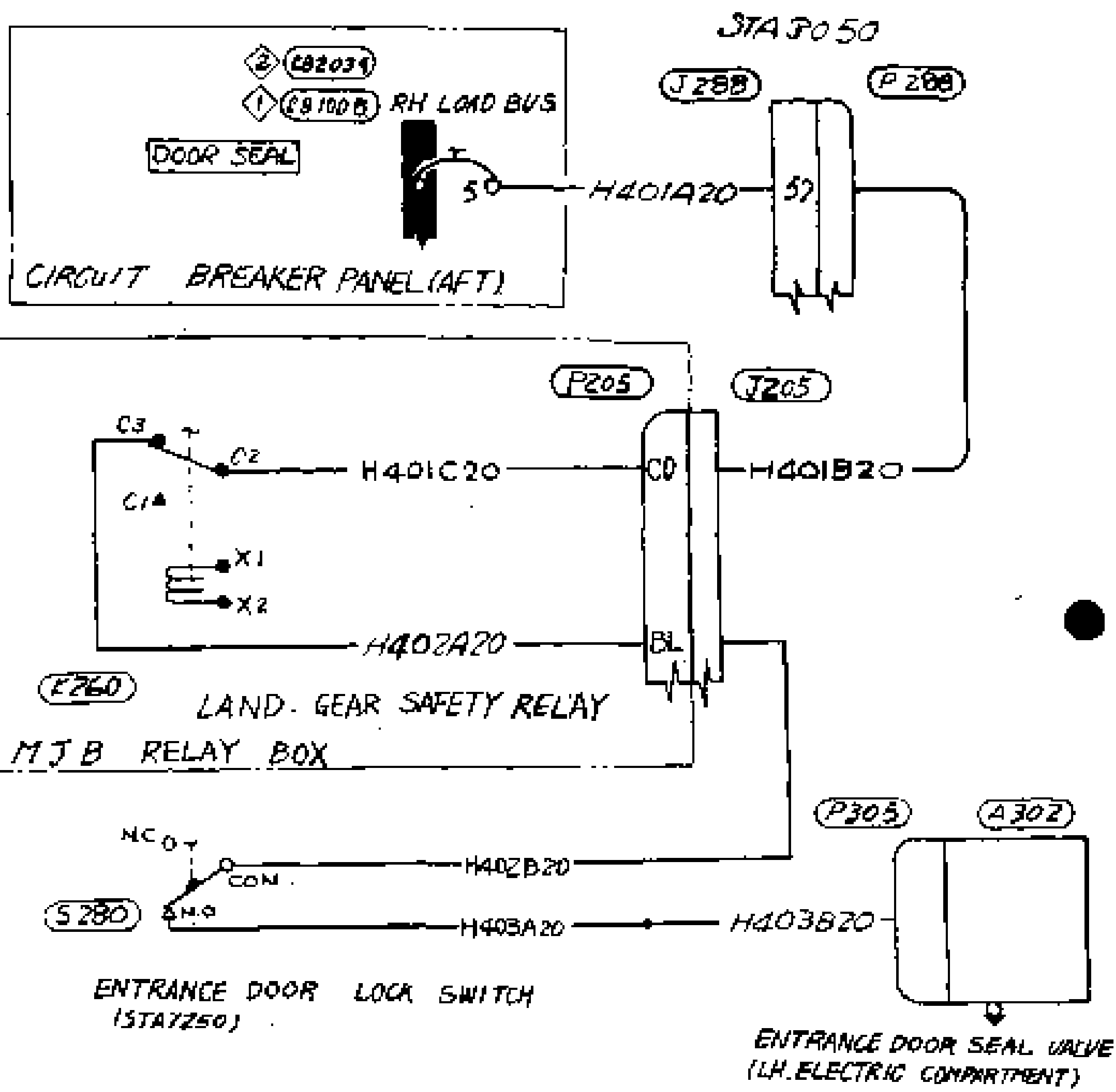
VACUUM


DS219
P244
P272
S269
INST VAC FAIL
K
W101A20
B
E211
W102A20N
A
ANNUNCIATOR PANEL
(L.H SIDE PANEL)
VACUUM PRESSURE
WARNING SWITCH

CHAPTER

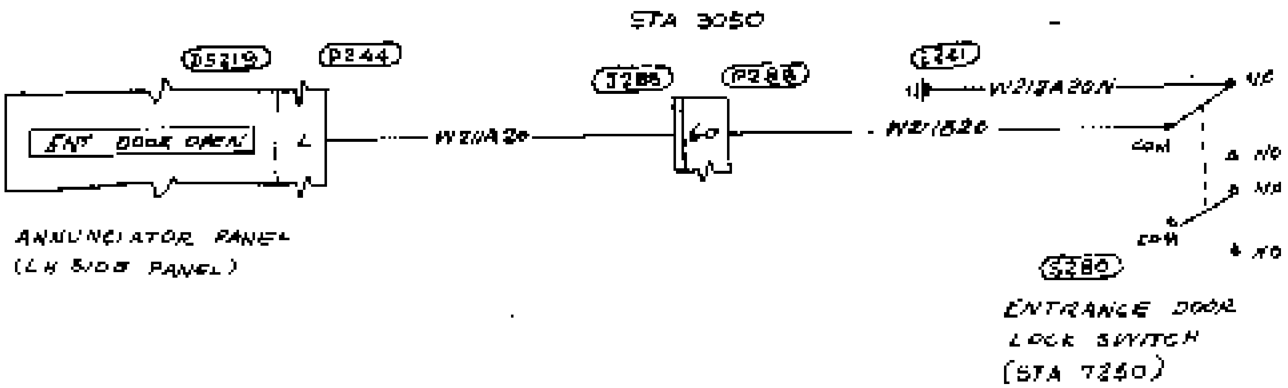
52

DOORS



◇1 AIRCRAFT S/N 661SA, 697SA THRU 713SA

◇2 AIRCRAFT S/N 714SA THRU 730SA

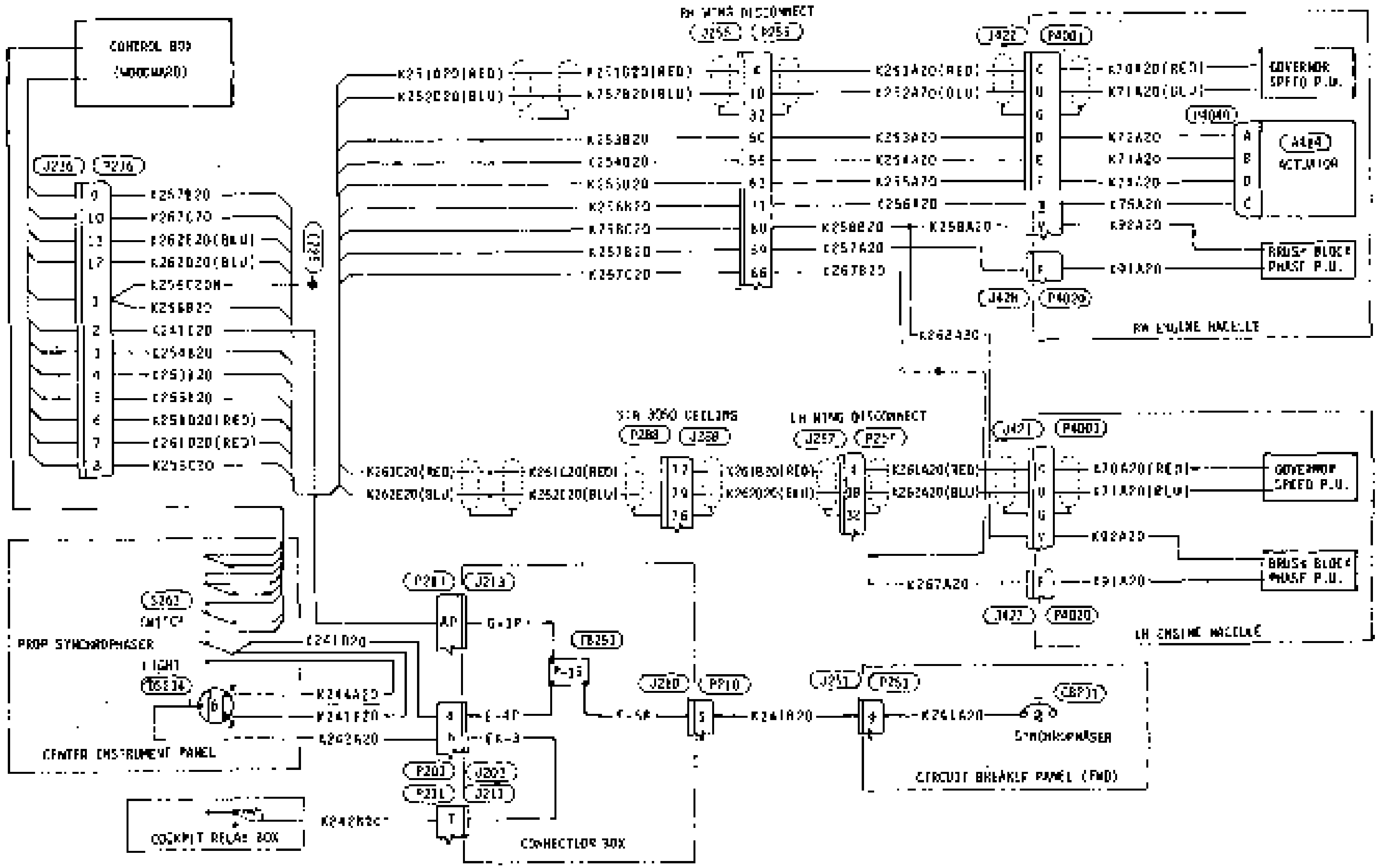

STA 3050
DS219
P244
J288
P288
ENT DOOR OPEN
L
W211A20
60
W211B20
W212A20N
NC
COM
NO
NO
NO
COM
S280
ANNUNCIATOR PANEL
(LH SIDE PANEL)
ENTRANCE DOOR
LOCK SWITCH
(STA 7250)

CHAPTER

61

PROPELLERS/
PROPULSORS

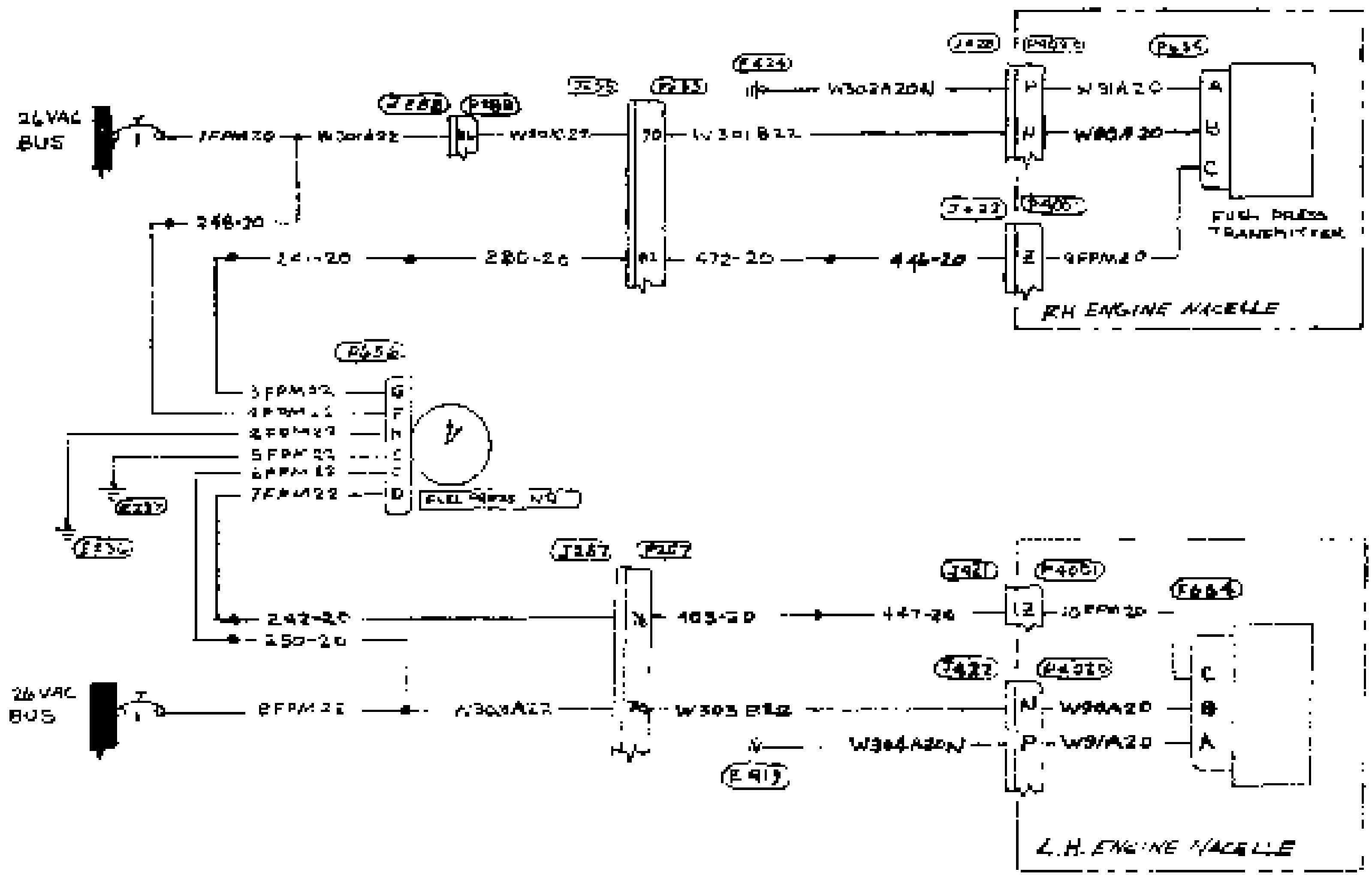
PROPELLER SYNCHROPHASER CONTROL
Aircraft S/N 661SA, 697SA - 730SA



CHAPTER

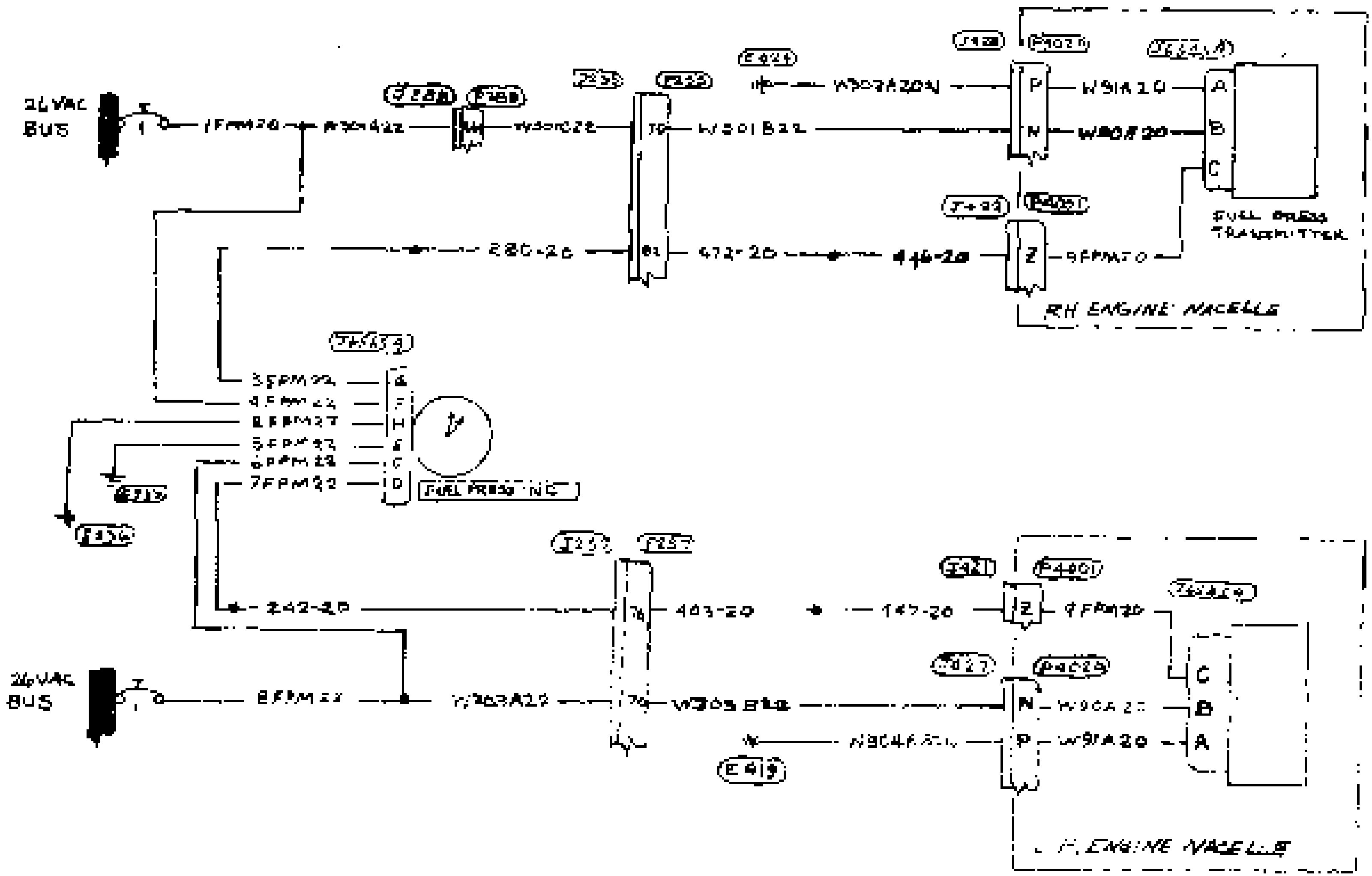
73

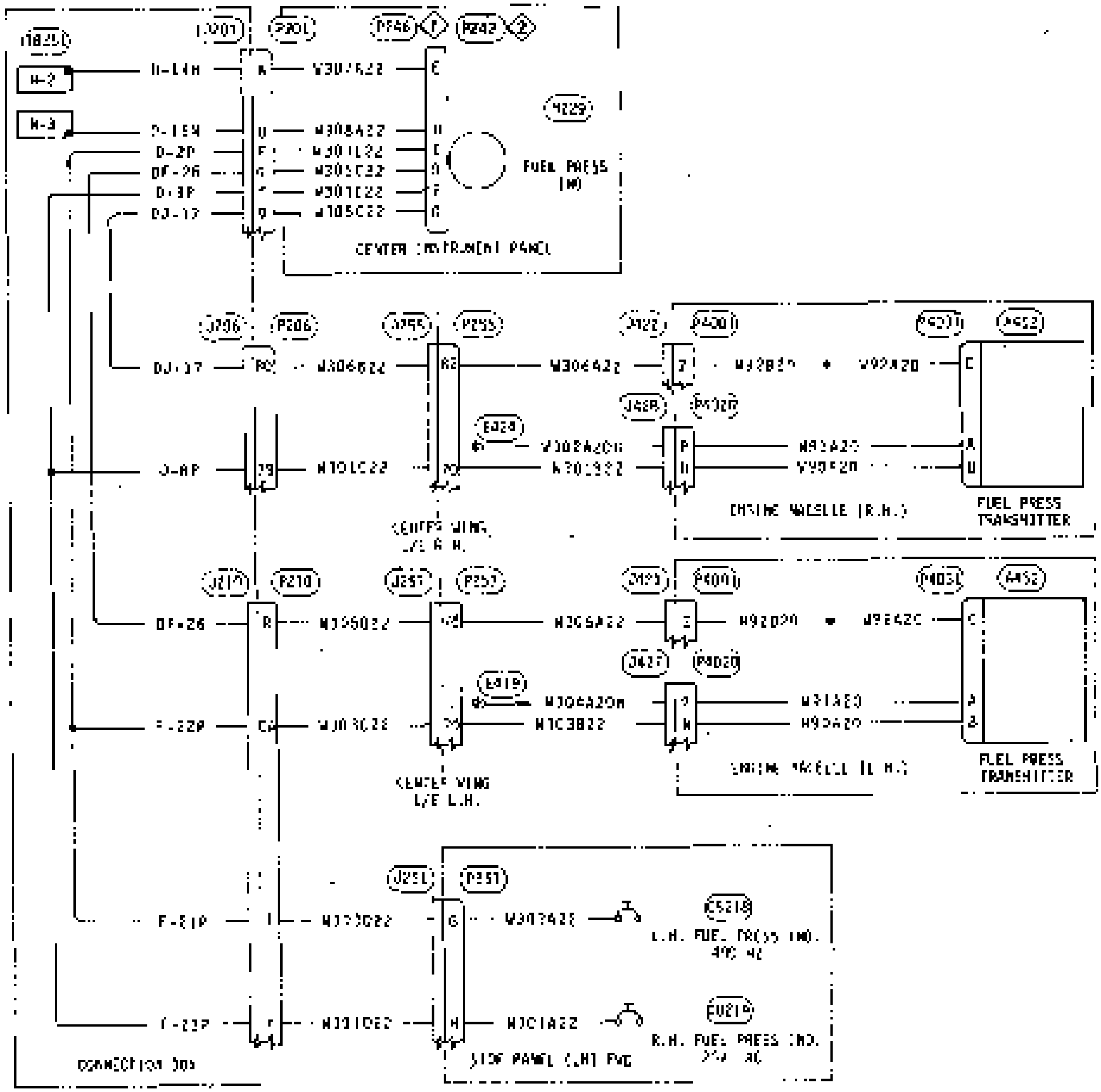
ENGINE FUEL AND CONTROL



FUEL PRESSURE INDICATION
Aircraft S/N 661SA

FUEL PRESSURE INDICATION
Aircraft S/N 697SA - 713SA



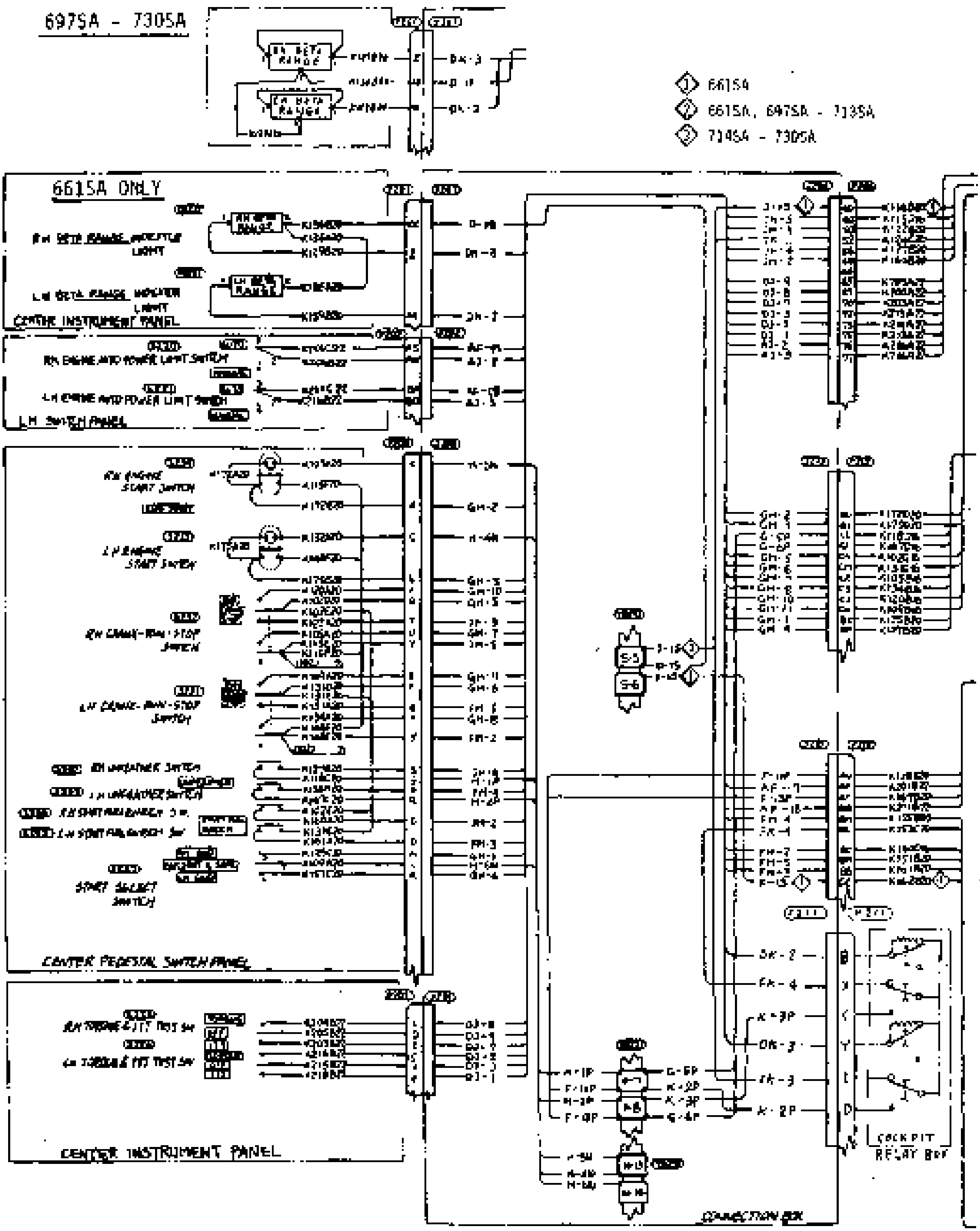


◇1 714SA - 717SA
◇2 718SA AND SUBSEQUENT

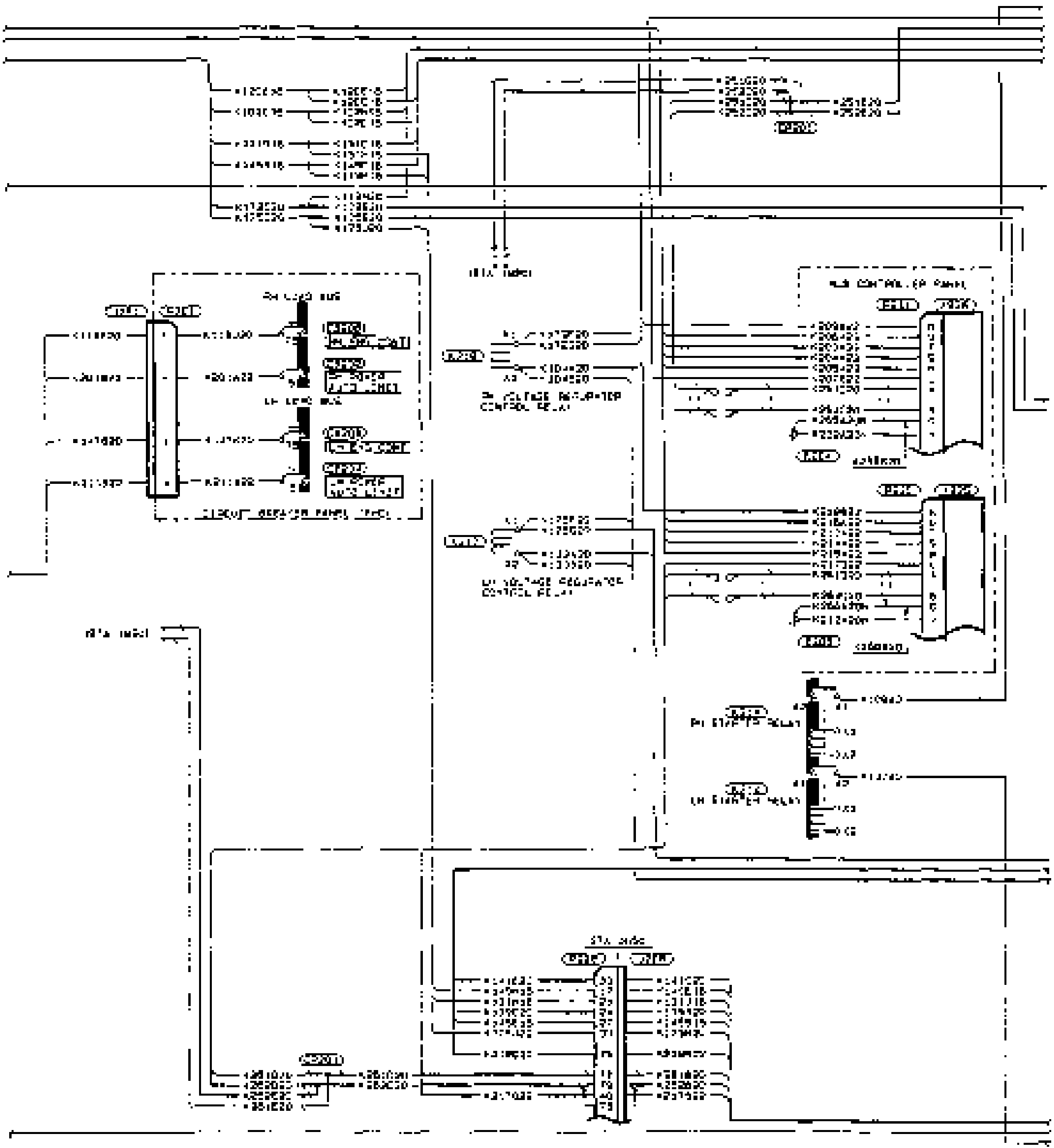
CHAPTER

76

ENGINE CONTROLS

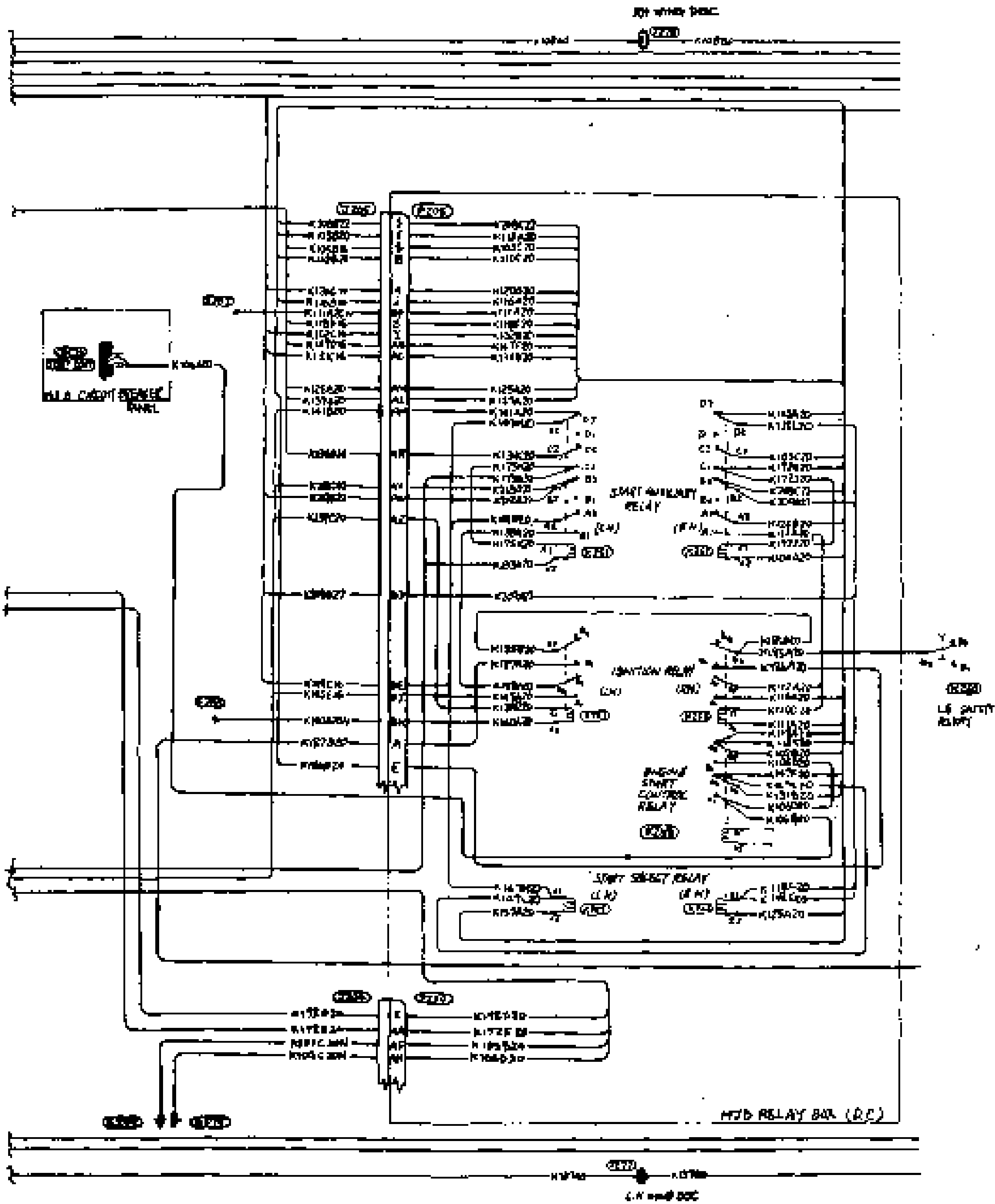


ENGINE & PROPELLER CONTROL
Aircraft S/N 661SA, 697SA - 730SA

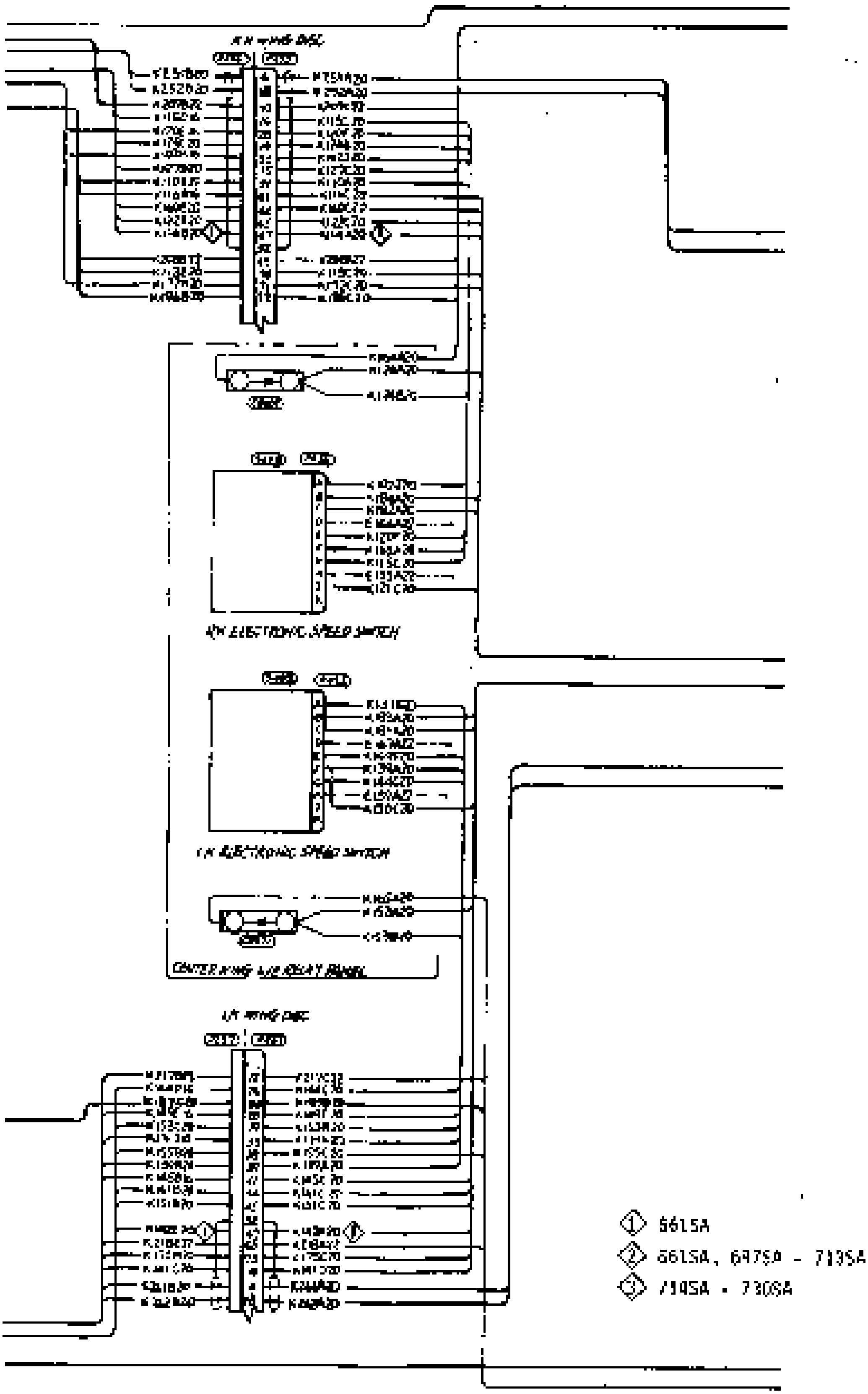


ENGINE & PROPELLER CONTROL AIRCRAFT S/N 661SA, 697SA - 730SA

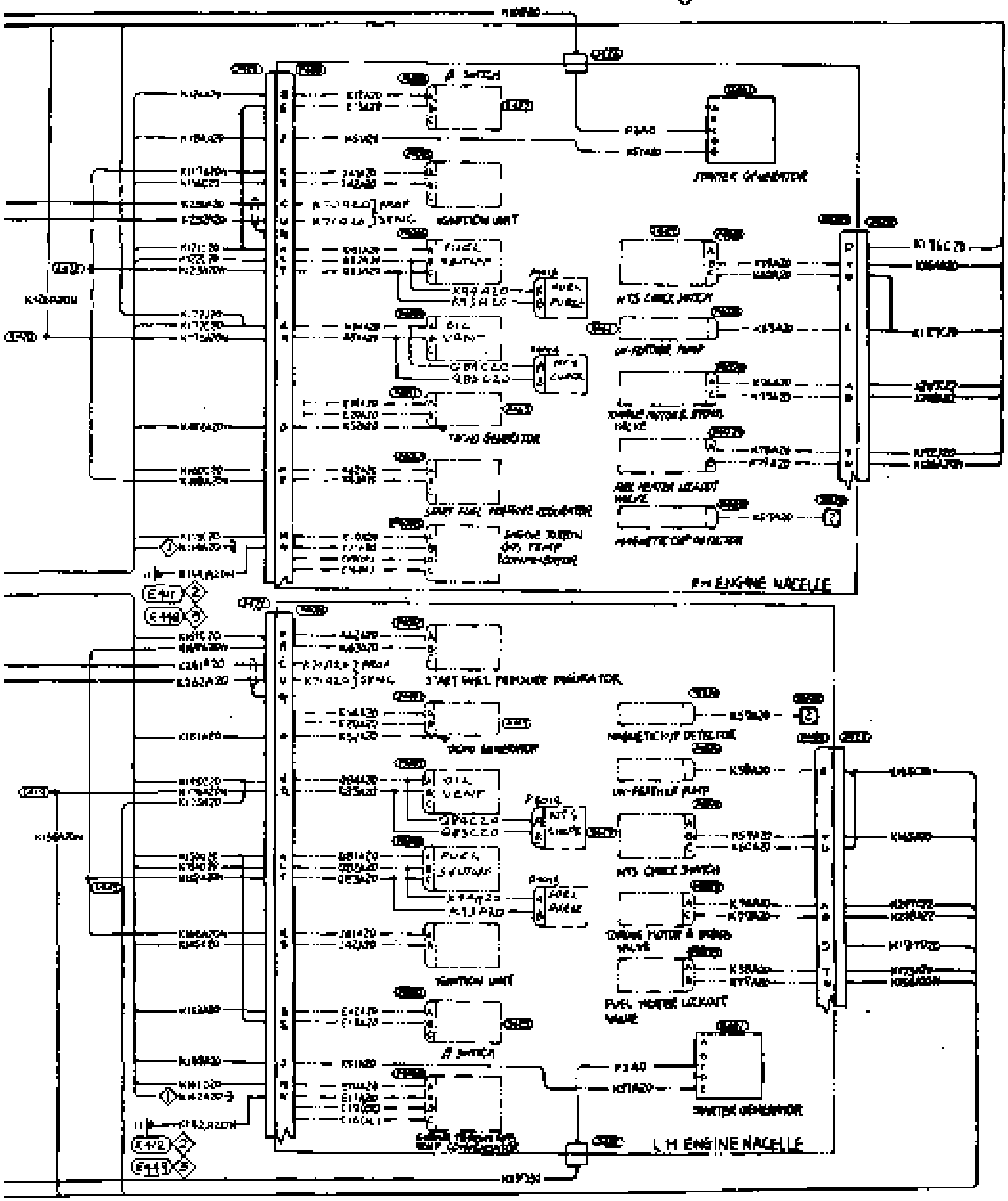
Changed
27 AUG. 1999



IGNITION RELAY
ENGINE START CONTROL RELAY
START SELECT RELAY


R/H ELECTRONIC SPEED SWITCH
L/H ELECTRONIC SPEED SWITCH
1 661SA
2 661SA, 697SA - 713SA
3 714SA - 730SA

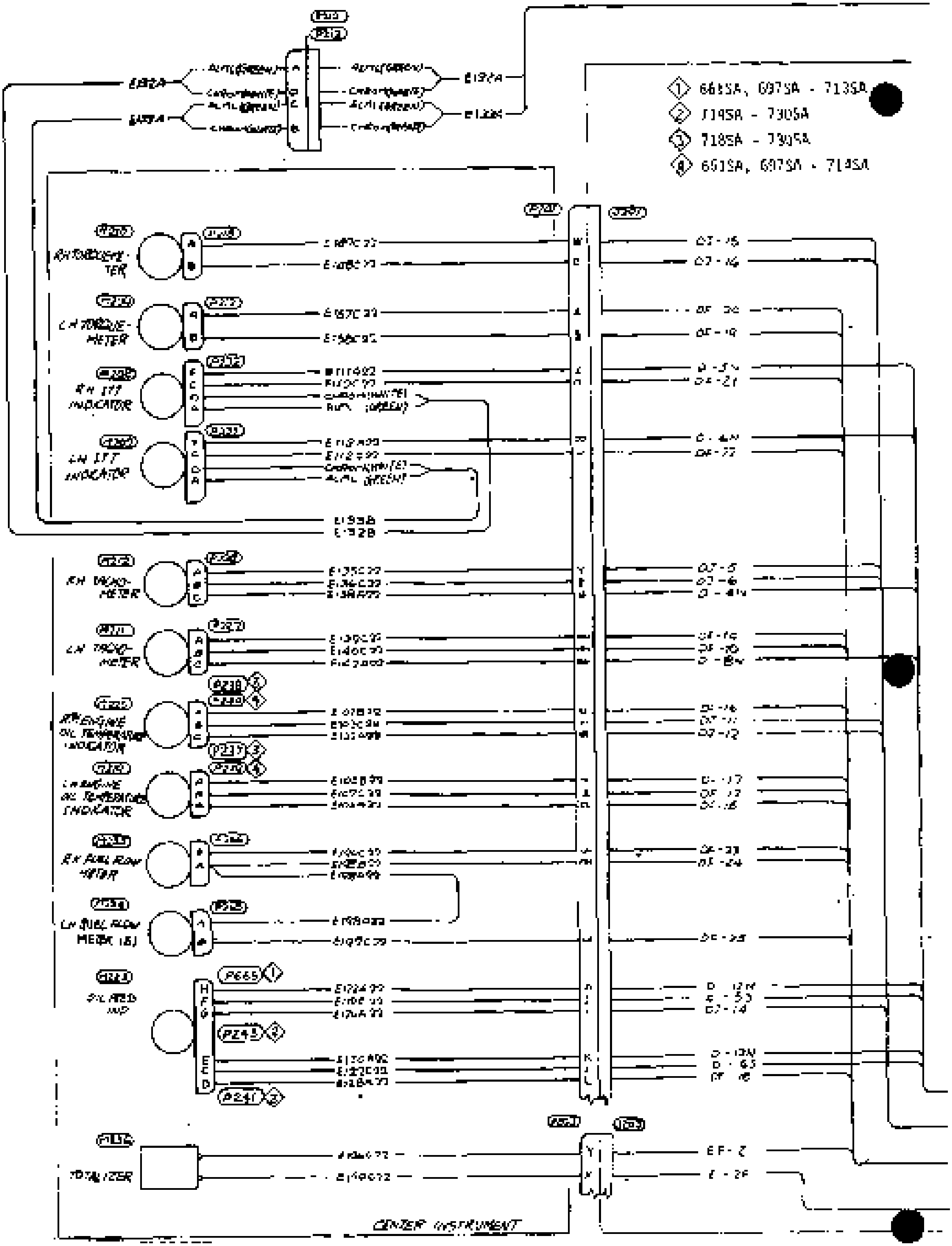
1 661SA
2 661SA, 697SA - 713SA
3 714SA - 730SA



CHAPTER

77

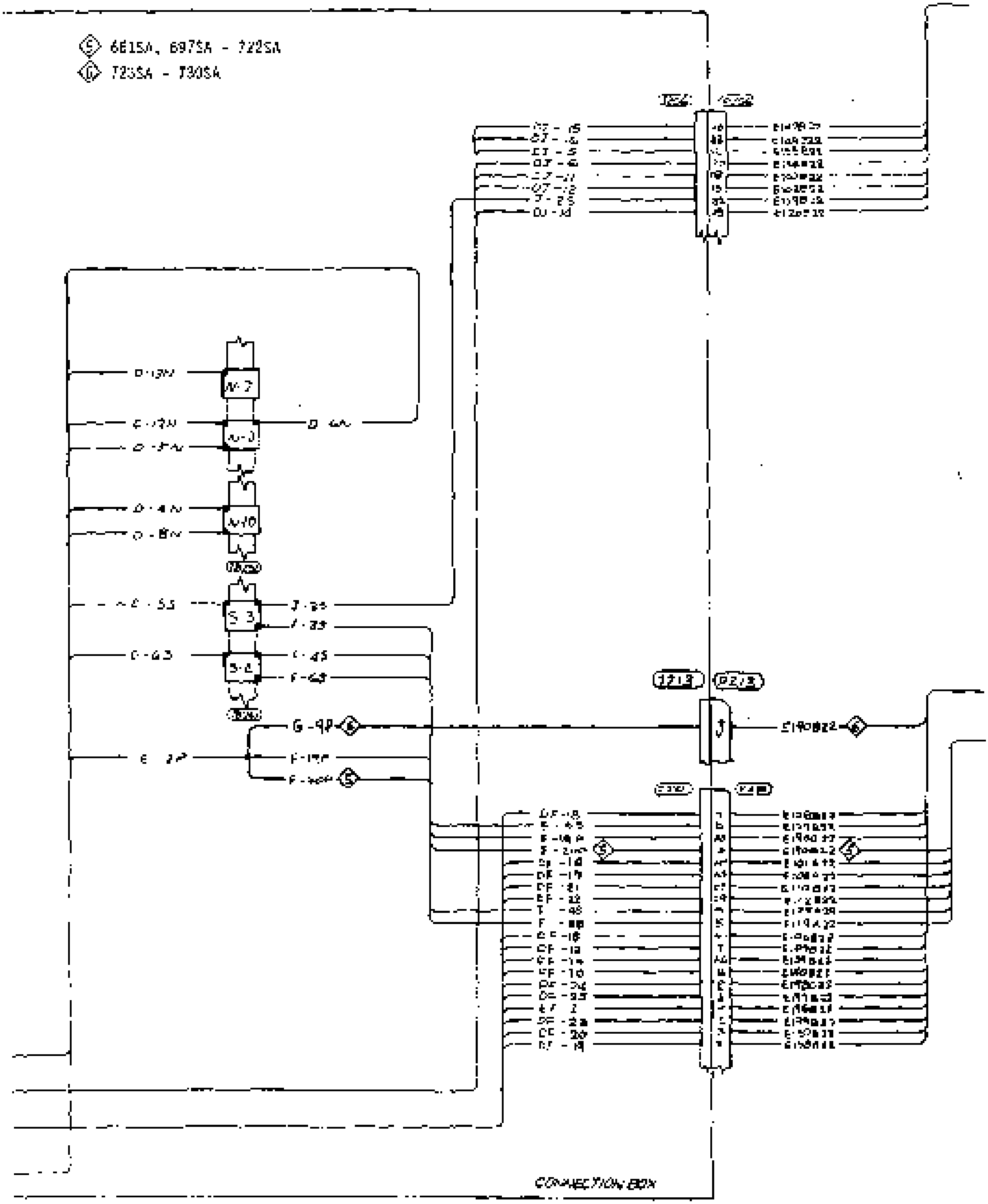
ENGINE INDICATING

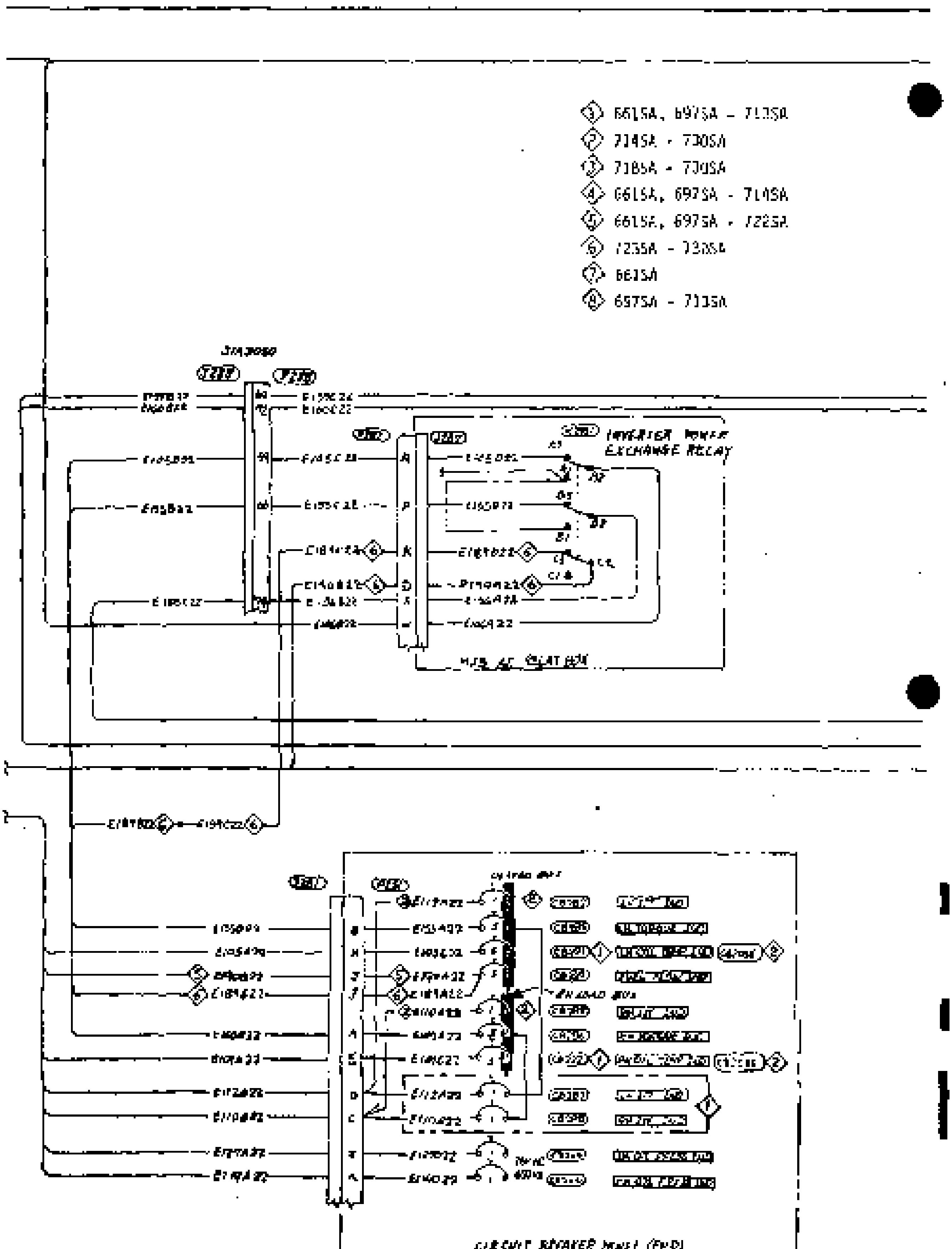


ENGINE INSTRUMENT INDICATION
Aircraft S/N 661SA, 697SA - 730SA

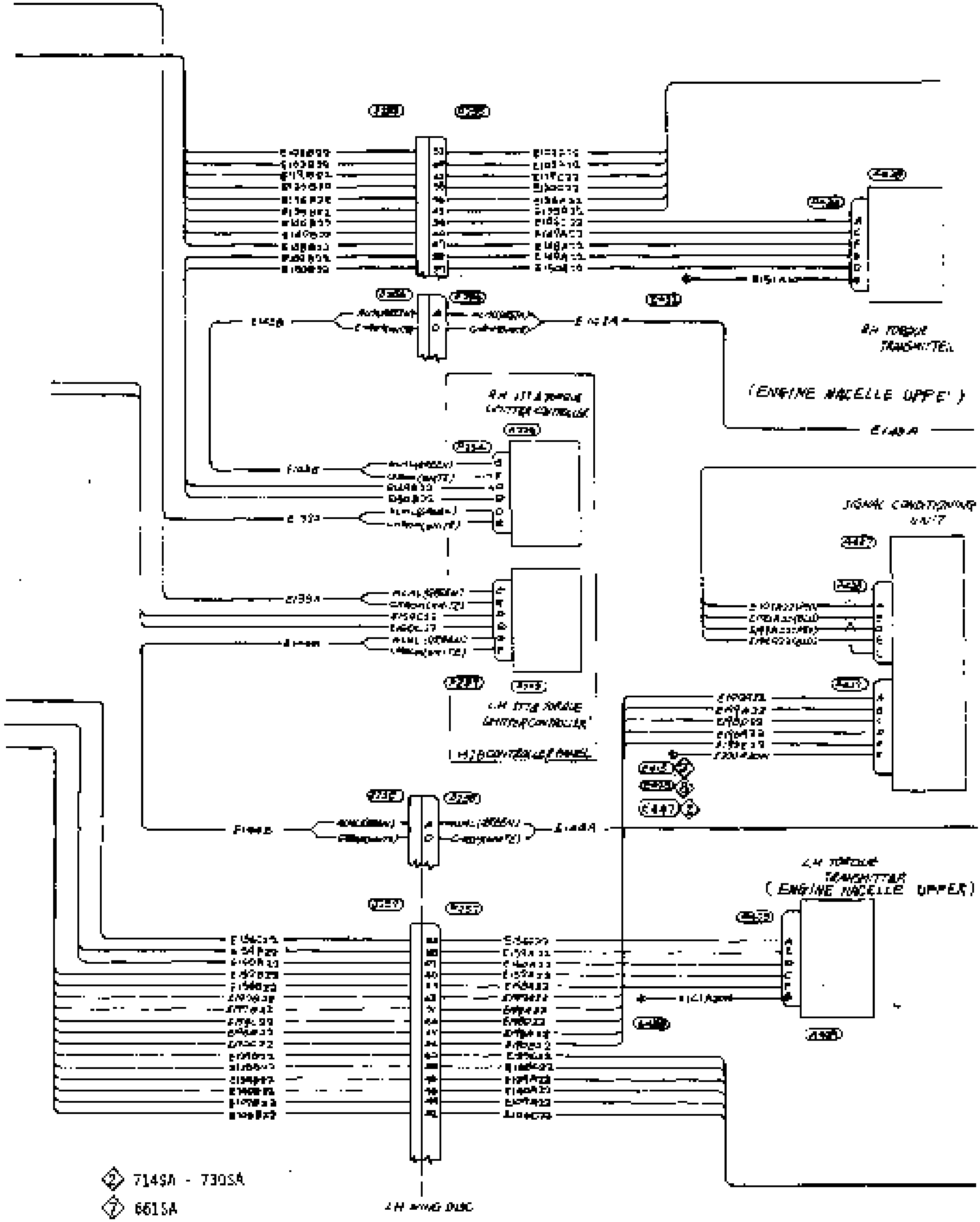
5 661SA, 697SA - 722SA

6 723SA - 730SA





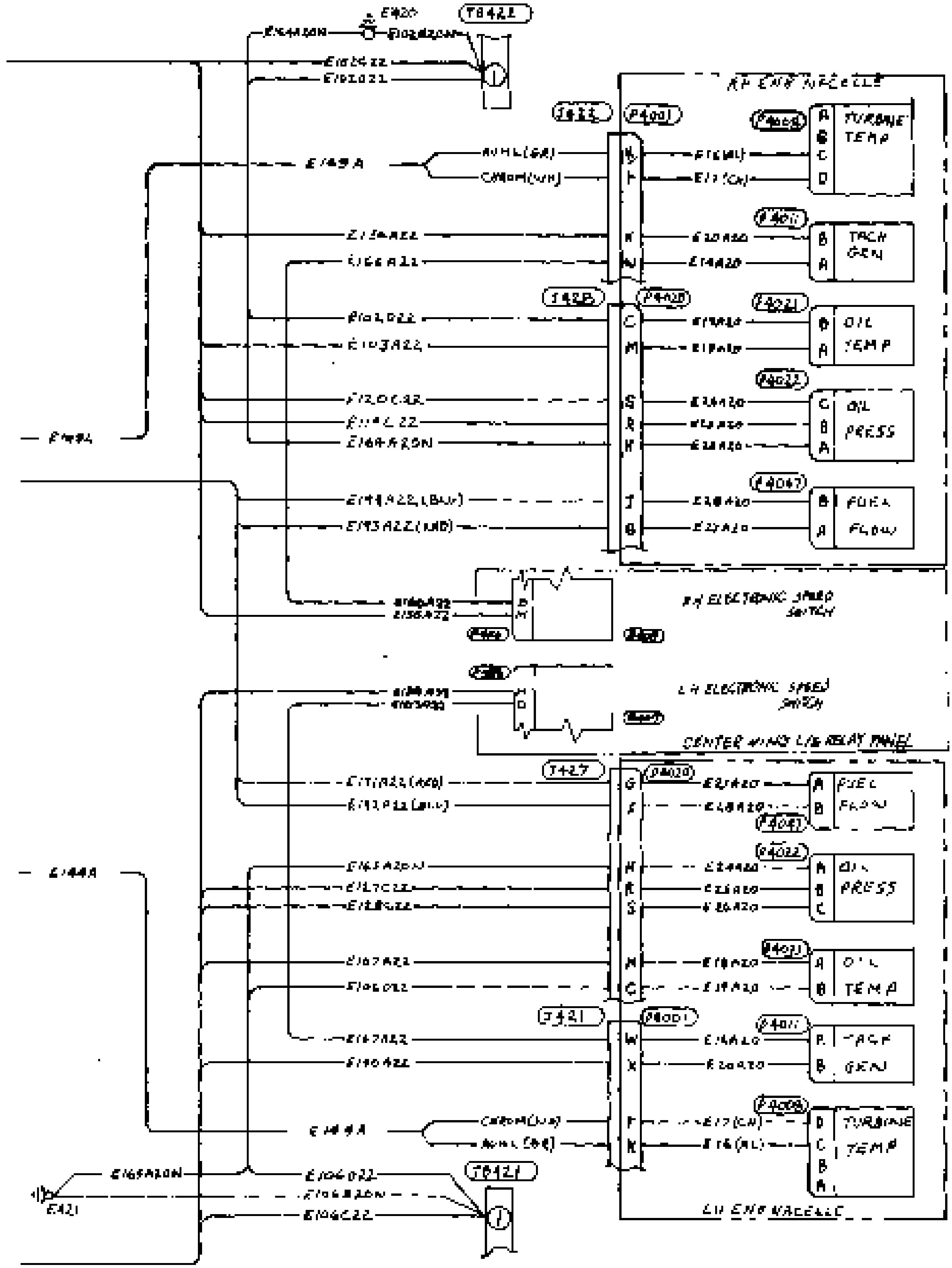
3 661SA, 697SA - 713SA
2 714SA - 730SA
1 718SA - 730SA
4 661SA, 697SA - 714SA
5 661SA, 697SA - 722SA
6 723SA - 730SA
7 661SA
8 697SA - 713SA
INVERTER POWER EXCHANGE RELAY



◇2 714SA - 730SA

◇7 661SA

◇8 697SA - 713SA


TB421
P4001
TURBINE TEMP
TACH GEN
OIL TEMP
OIL PRESS
FUEL FLOW
LH ELECTRONIC SPEED SWITCH
CENTER WING L/G RELAY PANEL
J427
J421
E421