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MERLIN IVC/METRO III PILOT TRAINING MANUAL

Record of Revision No. 5

Enclosed is Revision No. 5, dated May 1992. This updates the Revision print of the Merlin IVC/Metro III Pilot Training Manual in September 1991. The revision is revised pages only and consists of approximately six pages.

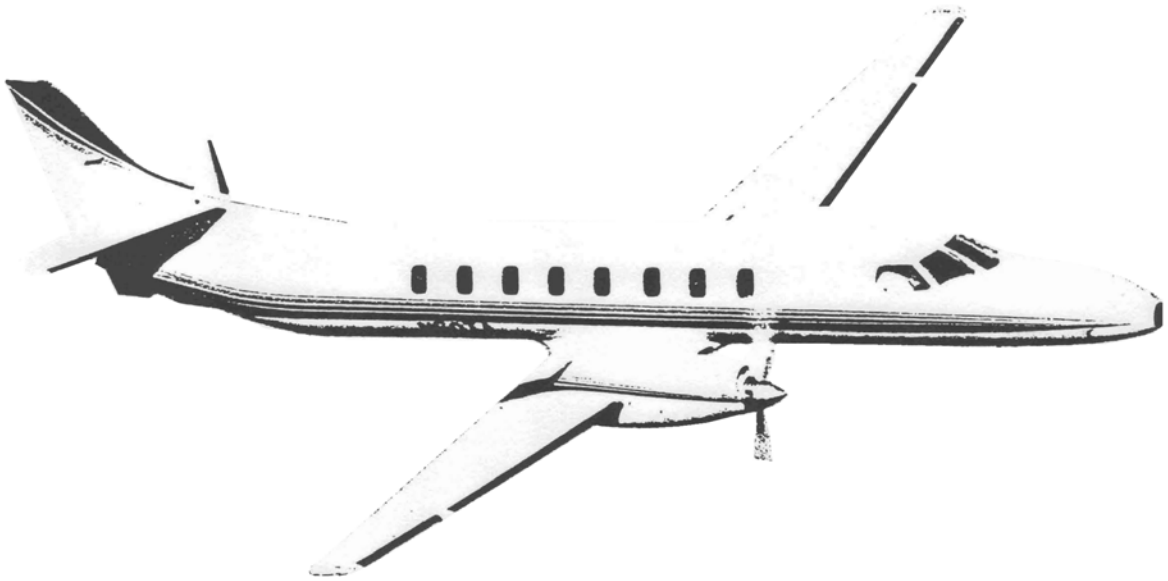
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The changes made in this revision will be further explained at the appropriate time in the training course.

... the best safety device in any aircraft is a well-trained pilot ...

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MERLIN IVC
METRO III
**PILOT
TRAINING
MANUAL**

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NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Pilot Manuals and Maintenance Manuals. It is to be used for familiarization and training purposes only.

At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the that of the manufacturer or the FAA shall take precedence.

We at want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

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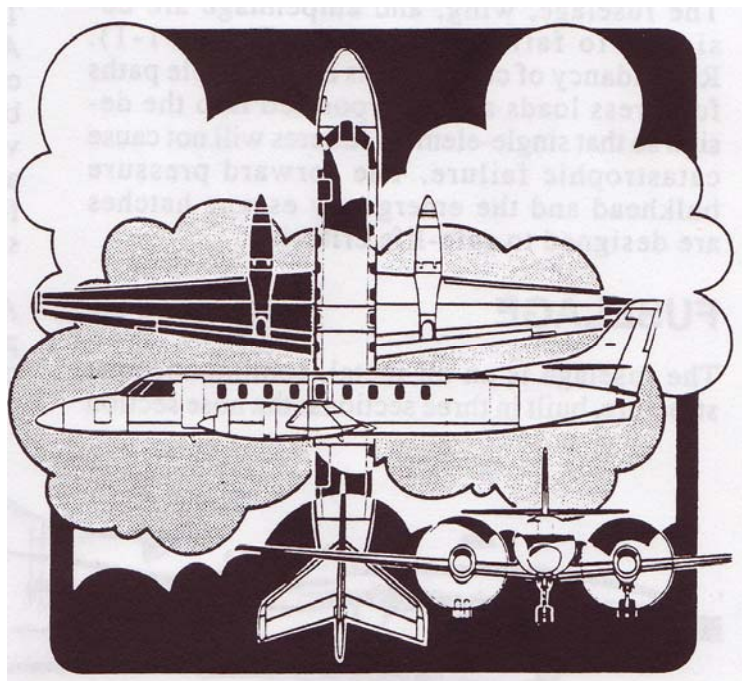


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CHAPTER 1

AIRCRAFT GENERAL



INTRODUCTION

This training manual provides a description of the major systems installed in the Fairchild SA227-AT (Merlin IVC) and SA227-AC (Metro III). The Merlin IVC and Metro III are similar in appearance, and both come in 14,500- and 16,000-pound models; distinctions between the aircraft will be made when necessary. A military version of the 16,000-pound Metro III has been designated the C-26, and all references to the Metro III are applicable to the C-26 unless otherwise noted.

This chapter covers the structural makeup of the different models and discusses the major airplane systems. No material is meant to supersede or substitute for any of the manufacturer's system or operating manuals, and you will frequently be instructed to refer to the appropriate Airplane Flight Manual (AFM) for information pertaining to your specific aircraft.

GENERAL

The Merlin IVC and the Metro III are pressurized twin turboprop airplanes. The Merlin IVC is designed for use as an executive transport, while the Metro III is designed to be a com-

muter airplane. The Merlin IVC may be ordered in a high-density seating configuration with 20 passenger seats or as an all-cargo aircraft. The Metro III can be equipped to be easily converted to accommodate passengers, cargo, or mixed loads. All airplanes are certified for operation to an altitude of 31,000 feet.



STRUCTURES

GENERAL

The fuselage, wing, and empennage are designed to fail-safe criteria (Figure 1-1). Redundancy of components and multiple paths for stress loads are incorporated into the design so that single-element failures will not cause catastrophic failure. The forward pressure bulkhead and the emergency escape hatches are designed to safe-life criteria.

FUSELAGE

The fuselage is an all-metal, semimonocoque structure, built in three sections: the nose section (including the cockpit), the tail section,

and the constant-diameter cabin section.

Nose Section

The nose section is constructed of aluminum. A heavy-duty beam extends from the bottom center of the forward pressure bulkhead to the bottom center of the radome bulkhead. It provides drag support for the nose gear and door assemblies. The nose gear is attached to the forward pressure bulkhead, which transmits gear stress loads.

A baggage compartment door measuring approximately 23 inches by 18 inches is located on each side of the nose section. The door has two hinge points at the forward side and two

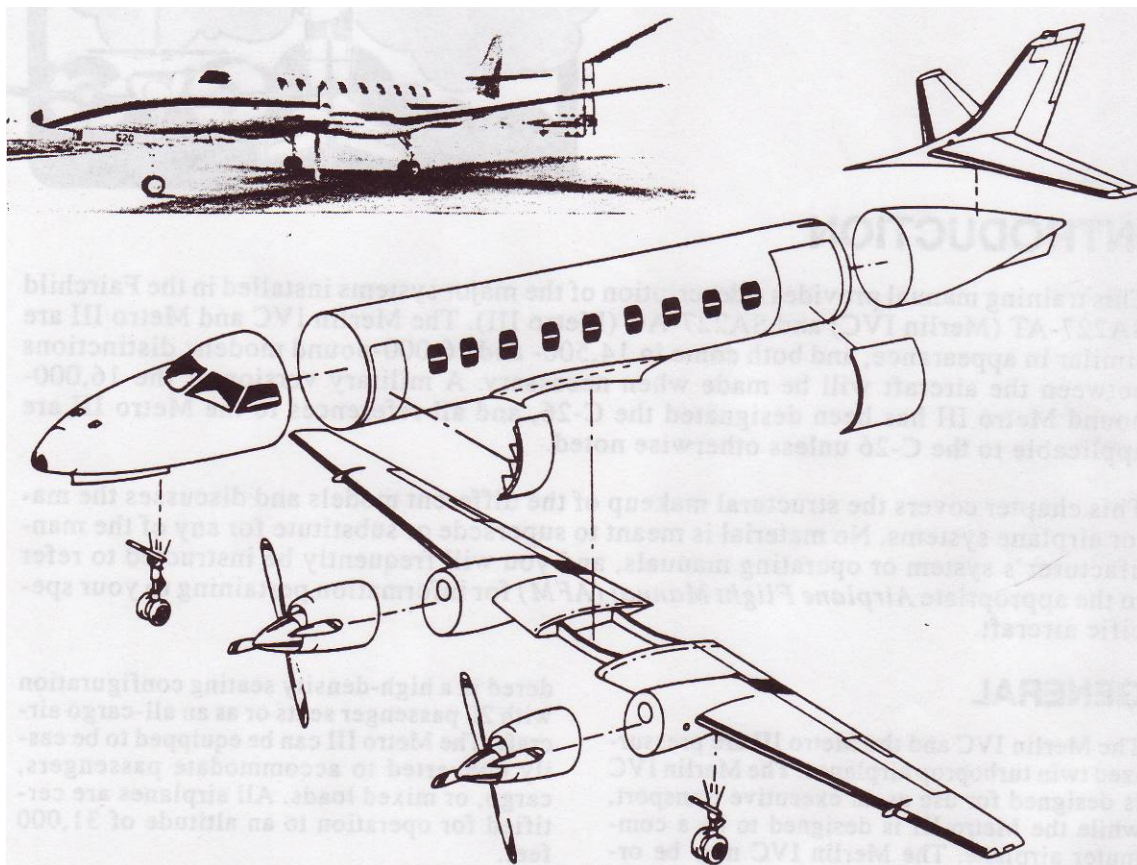


Figure 1-1. Merlin IVC and Metro III Major Assemblies



Hartwell trigger lock latches at the aft side, plus a key lock. The lock latches are designed to prevent baggage or equipment from accidentally opening the doors during flight.

The nose baggage compartment is located between the forward pressure bulkhead and the radome bulkhead.

The CAWI tank is located inside the nose baggage compartment.

Cockpit

A typical cockpit configuration is seen in Figure 1-2.

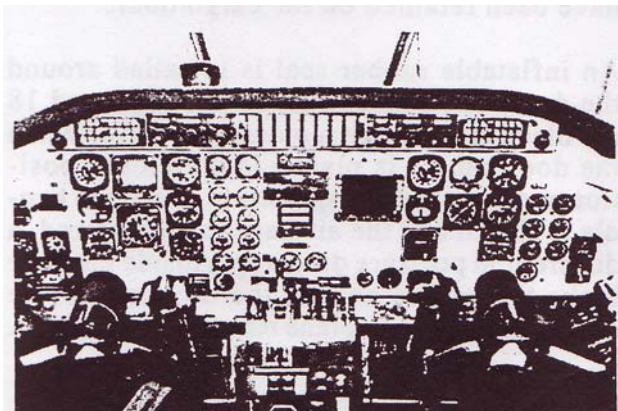


Figure 1-2. Typical Cockpit Configuration

Cabin Section

The aluminum cabin section has a constant diameter. All windows, doors, and joints are reinforced for stress equalization.

Tail Section

The aluminum tail section has three heavy-duty frames for attachment of the vertical stabilizer. The complete tail section is pressurized with the outflow valve mounted in the center of the aft pressure bulkhead.

The tail section contains the aft equipment rack, which houses inverters, SRL computers, oxygen cylinder, and other airplane system components.

DOORS AND EXITS

Cabin Doors

The airstair passenger door is located on the left side of the fuselage, forward of the wing, just behind the cockpit. A cargo door is also on the left side, aft of the wing. A stowable ladder is provided for access.

Passenger Entrance Door

The airplane main entrance is through the 25-by-53-inch airstair door (Figure 1-3). The latch mechanism, which can be key-locked for security, can be operated from either inside or outside the airplane. A snubber assembly is built in to ensure that the door opens slowly and smoothly. When the door is being opened, the operator should support the door until sure that the snubber assembly is operating properly. Covered cables or chains at both sides act as handrails as well as door-opening limiters.

The passenger entrance door is equipped with seven click-clack latches and two alignment pins. The latches extend from the door into the

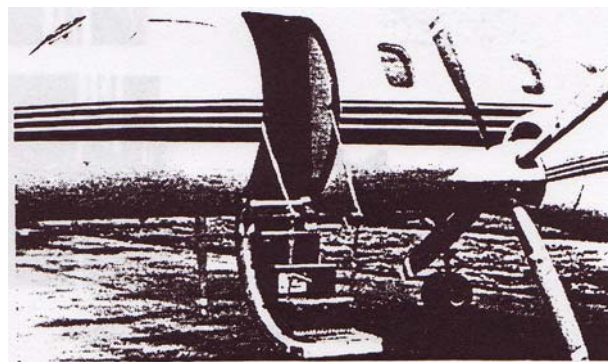


Figure 1-3. Passenger Door



doorframe receptacles during the initial closing movement of the door handle. Continued movement of the door handle further extends the latches, expands the split barrel around the plunger, and secures each latch firmly into its receptacle in the fuselage doorframe. Figure 14 shows a cabin door click-clack in three configurations, from retracted to fully extended.

CAUTION

Ensure that the click-clack latches are completely retracted into the door before attempting to close the door. Attempting to close the door with a latch extended can cause serious damage to the latch. This damage,

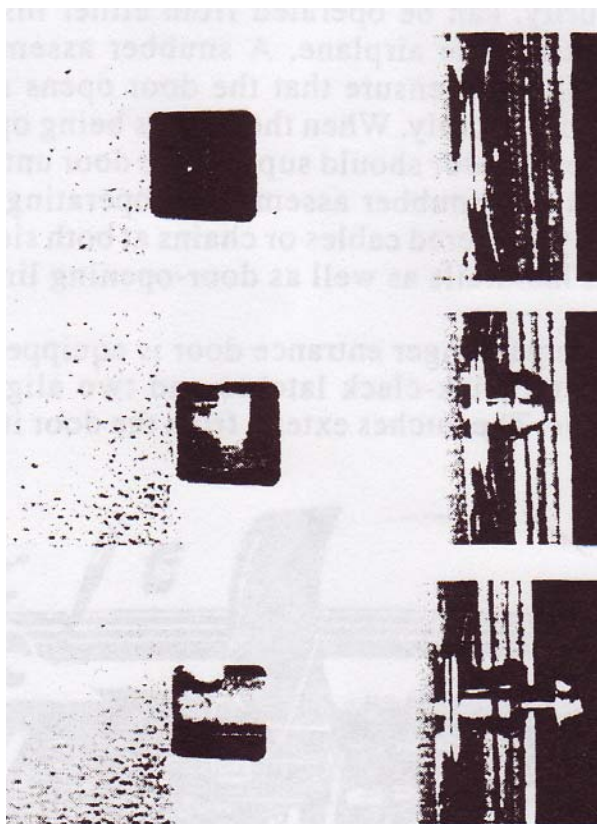


Figure 1-4. Cabin Door Click-Clack

in turn, may make it impossible to latch the door, or it may cause the door to be impossible to open by the normal procedure.

A door warning light microswitch is installed in each click-clack receptacle. All of the microswitches are wired in parallel, so the red CABIN DOOR warning light on the annunciator panel illuminates if any one of the latches is unlocked.

On airplanes SNs 579 and subsequent, the main cabin door click-clacks have been replaced with bayonet-type latches and the doorframe reinforced with doubler skins. Click-clacks have been retained on the cargo door.

An inflatable rubber seal is installed around the door. The seal is inflated by regulated 18 psi bleed air through a pneumatic valve when the door handle is placed in the closed position and an engine is operating. When the handle is closed and the airplane is pressurized, a differential pressure diaphragm inside the door locks the handle so that the door cannot be opened while the airplane remains pressurized.

Cargo Door

A cargo door, 53 by 51 1/4 inches, is located on the aft left side of the fuselage (Figure 1-5). It is hinged at the top. Early airplanes have an assist spring and an overcenter mechanism to hold the door open. Later airplanes are modified with gas springs replacing the mechanical door opening devices. The handles, locking devices, click-clack latches, and a pneumatic pressure seal are similar to those for the passenger door; however, the differential pressure lock for the operating handle is not included.

A red CARGO DOOR annunciator light is provided to indicate an unlocked cargo door

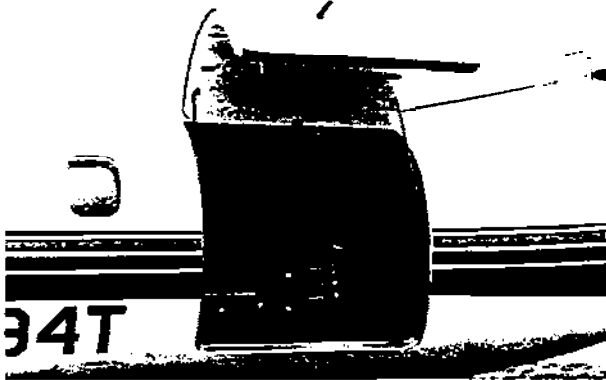


Figure 1-5. Cargo Door

The CARGO DOOR WARNING AND TEST on the copilot's side console (Figure 1-6) provides a means to confirm proper operation of the cargo door latching system. When the door handle is unlatched, the red DOOR UNSAFE light is illuminated. If all microswitches in the warning system are in correct (open) position, the green SWITCHES NORMAL light illuminates when the test switch is held to SWITCH TEST. In the LAMP TEST position both lights illuminate to verify bulb integrity

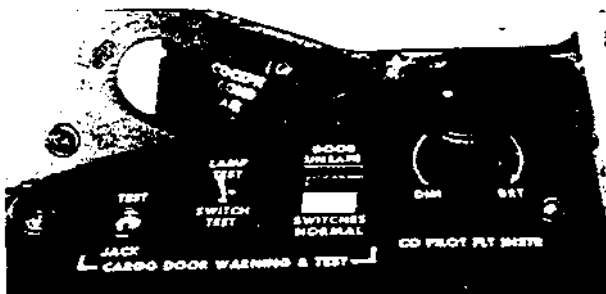


Figure 1-6. Cargo Door Warning and Test

Emergency Exits

All emergency exits (Figure 1-7) are plug-type and open into the cabin. Three emergency exits are located over the wings — two on the right side and one on the left. Each

emergency exit measures approximately 20 by 28 inches. These exits lock at the top in two places.

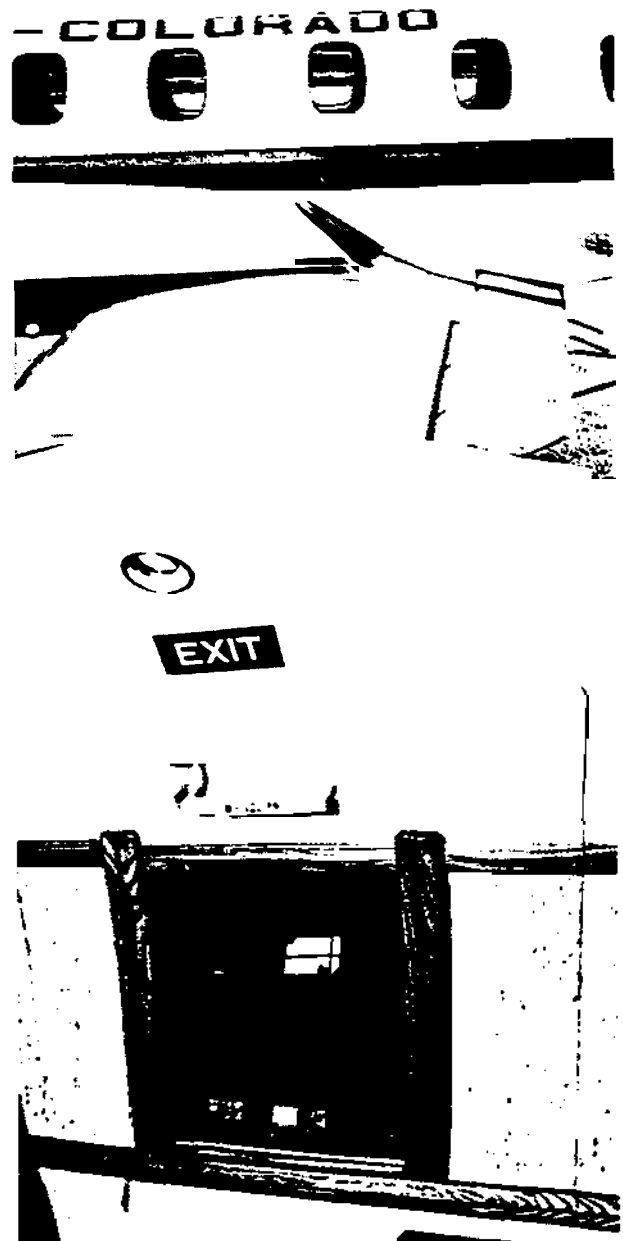


Figure 1-7. Emergency Exit



WINDOWS

Windshields

Two direct-vision glass windshields are located in front of the pilot and copilot. These are the only glass windshields, and they are electrically heated. The center windshield and the side windows are stretched acrylic sheets (Figure 1-8).

Merlin IVC—The center windshield and each of the cockpit side windows have dual acrylic panes. Bleed air flows between the panes for defogging.

Metro 111—The center windshield and each of the cockpit side windows have single acrylic panes, although dual acrylic panes are available. Bleed-air flow is also available for windshield defogging.

Passenger Compartment Windows

All passenger compartment windows are of acrylic construction.

Merlin IVC—Fifteen dual-pane windows are installed in a dry air sandwich configuration. A desiccant bag is attached at the bottom of each window.



Figure 1-8. Windshield

Metro 111—Nineteen single-pane windows are installed in the cabin. Optional dual-pane dry air sandwich-type windows are available for installation. Window shades are also optional.

WING

The cantilevered wing contains integral fuel tanks, battery wells, and air-conditioning components. Bleed-air lines, electrical cables, engine control cables, and hydraulic lines are enclosed in conduits along the leading edge. The wing trailing edge houses the flight control surfaces and the actuating mechanism for the flaps, ailerons, and aileron trim tabs. The wing span is 57 feet.

EMPENNAGE

The horizontal stabilizer is attached to the vertical stabilizer (Figure 1-9). Pitch trim is provided by a DC-powered dual actuator trim motor that adjusts the angle of attack of the stabilizer leading edge.



Figure 1-9. Empennage

AIRPLANE SYSTEMS

ELECTRICAL POWER

General

Electrical power is supplied by two batteries, two generators, and two inverters. Provisions are also made for use of a ground power unit. The location of these components is shown in Figure 1-10.

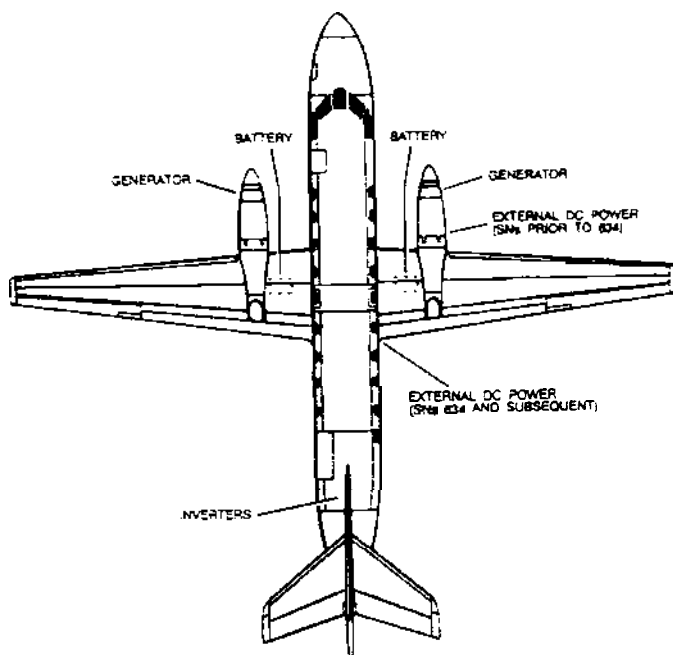


Figure 1-10. Electrical Power Components

DC Power

DC electrical power can be provided by two nickel-cadmium batteries, two dual-function starter-generators, or through a ground power unit. When either battery, either generator, or the GPU is operating and the applicable battery or generator switch is on, DC power is available to the battery bus. Battery bus power is then available to any or all three DC distribution buses, as selected by bus-tie switches. The three buses are found on the left and right consoles. Nine circuits are normally powered by the left essential bus but have specific bus transfer switches to allow them to be powered by the right essential bus. Figure 1-11 shows a simplified DC distribution schematic.

AC Power

AC power is supplied by either of the two inverters. The operating inverter is selected with the inverter selector switch. The left essential bus powers the No. 1 inverter which, in turn, supplies AC to the left 115-VAC bus and to the left 26-VAC bus. The right essential bus powers the No. 2 inverter for comparable right 115- and 26-VAC buses. The left and right 115-VAC buses have a bus tie, and so do the two 26-VAC buses. As long as one inverter is operational, all four AC buses can be powered, as seen in Figure 1-17.



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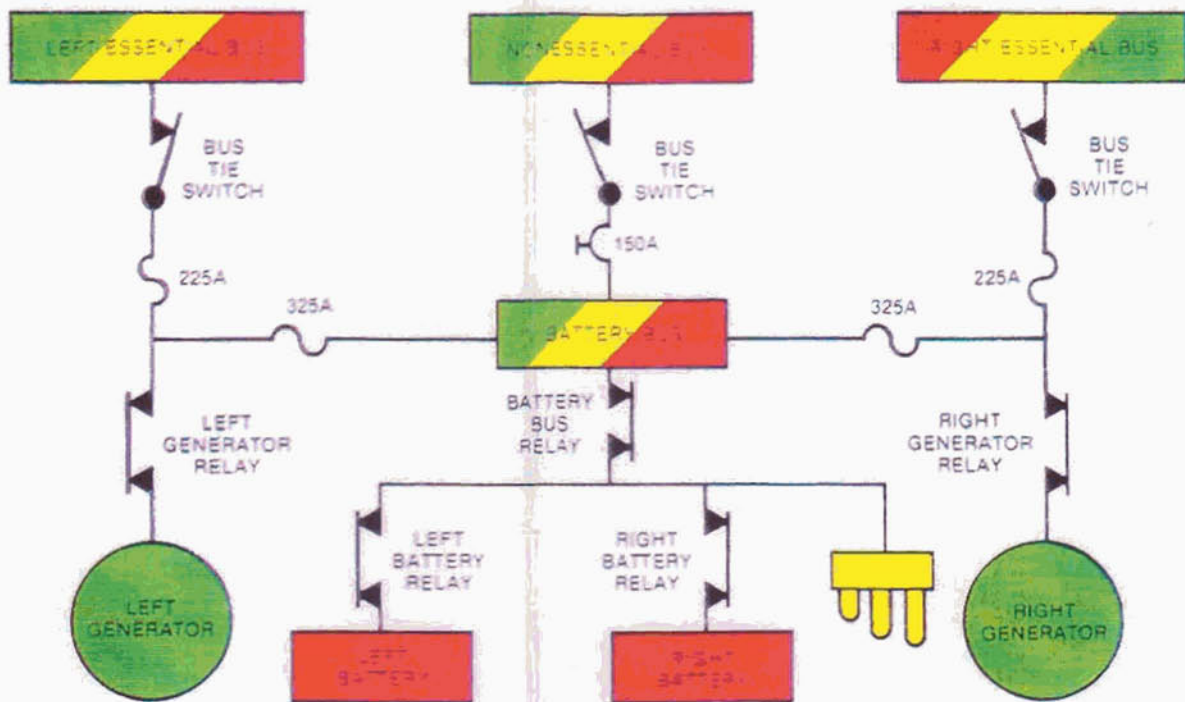


Figure 1-11. DC Distribution

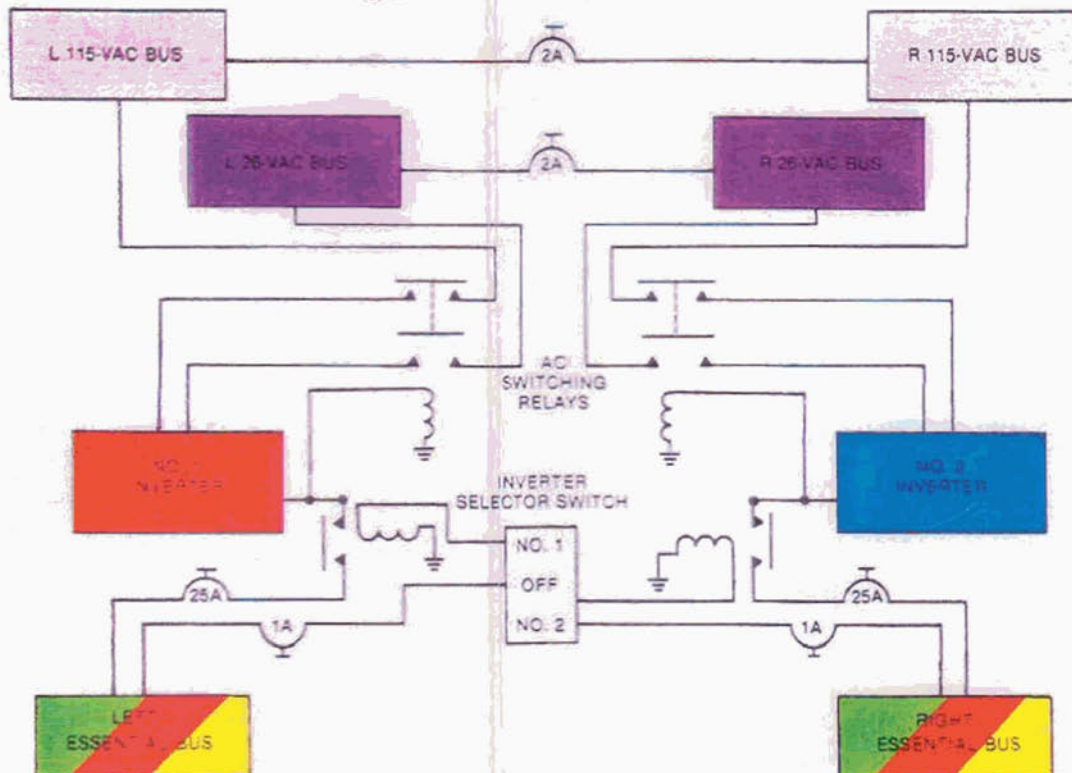


Figure 1-12. AC Distribution

LIGHTING

Interior lighting comprises cockpit and cabin lights. Some of the interior lighting controls are seen in Figure 1-13.

Appendix B in this manual displays all light indicators, and page B-1 should be folded out and referred to while studying this manual.

Annunciator lights alert the pilot of system malfunctions and operating status. The lights are red, amber, or green, except for the BY-PASS OPEN lights, which are blue. All annunciators can be tested with the PRESS TO TEST button, seen in Figure 1-14.

Exterior lighting is illustrated in Figure 1-15 and is controlled from the pilot's lower switch panel.

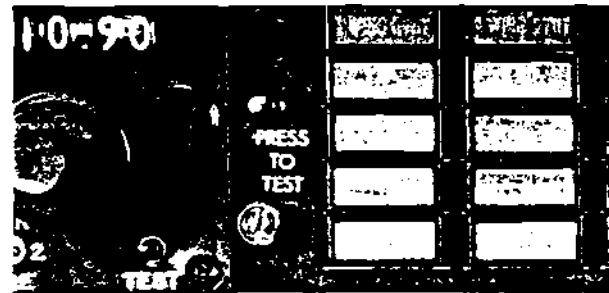


Figure 1-14. Annunciator Panel Press-to-Test Button

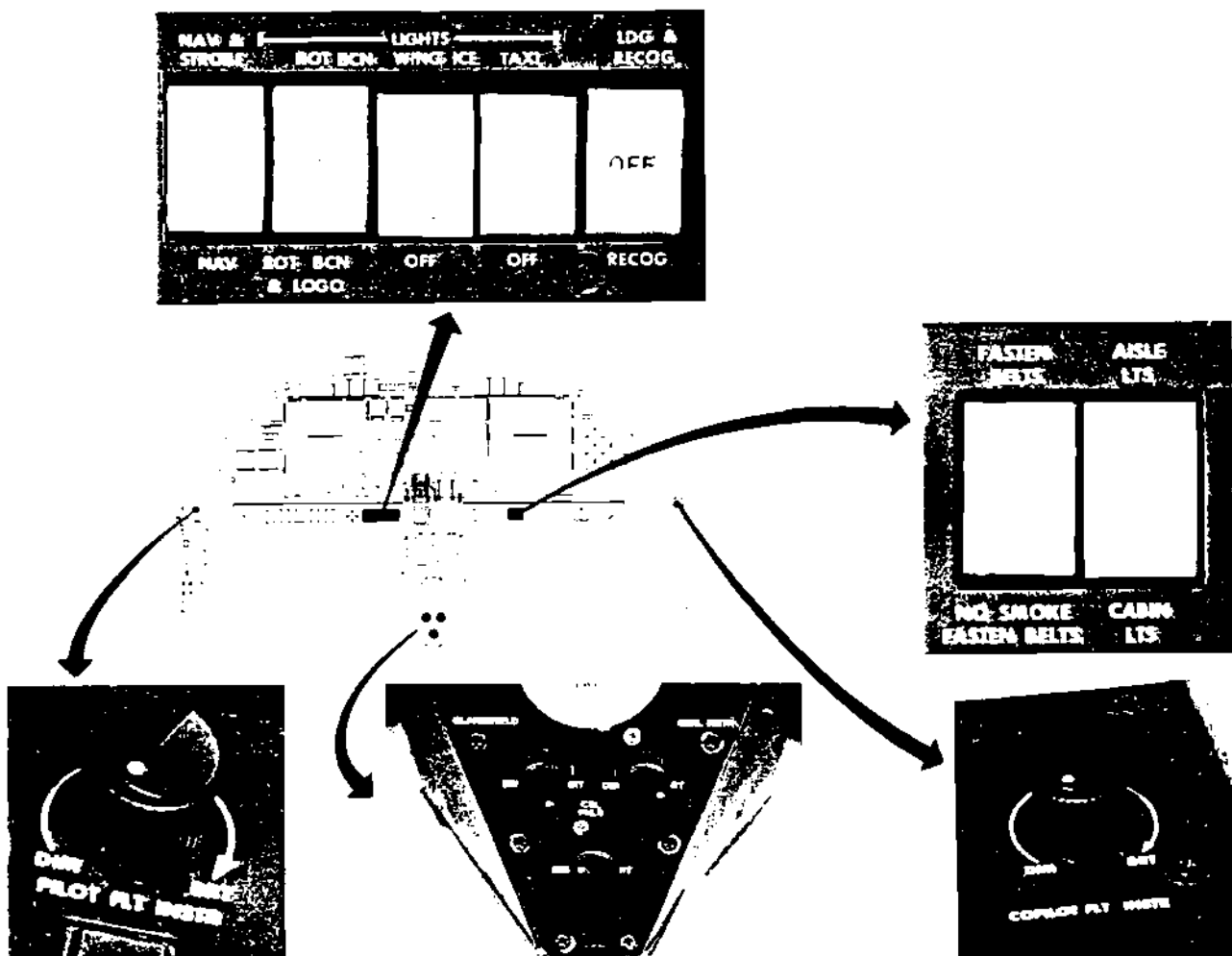


Figure 1-13. Lighting Controls



FUEL SYSTEM

Fuel for each engine is stored in an integral wing tank, with a gravity-fed crossflow system. The crossflow capability allows either engine to use all fuel on board, and is also used for fuel balancing. The crossflow valve is powered by one of the essential bus transfer circuits.

Each wing tank capacitance fuel probe system sends an input to the dual fuel quantity indicator on the cockpit instrument panel (Figure 1-16). The indicator is calibrated in hundreds of pounds. Total usable fuel capacity is 648 gallons (4,342 pounds).

The airplane is gravity-fueled through **overwing** filler holes. A quick-release drain valve on the crossflow line allows rapid single-point defueling.

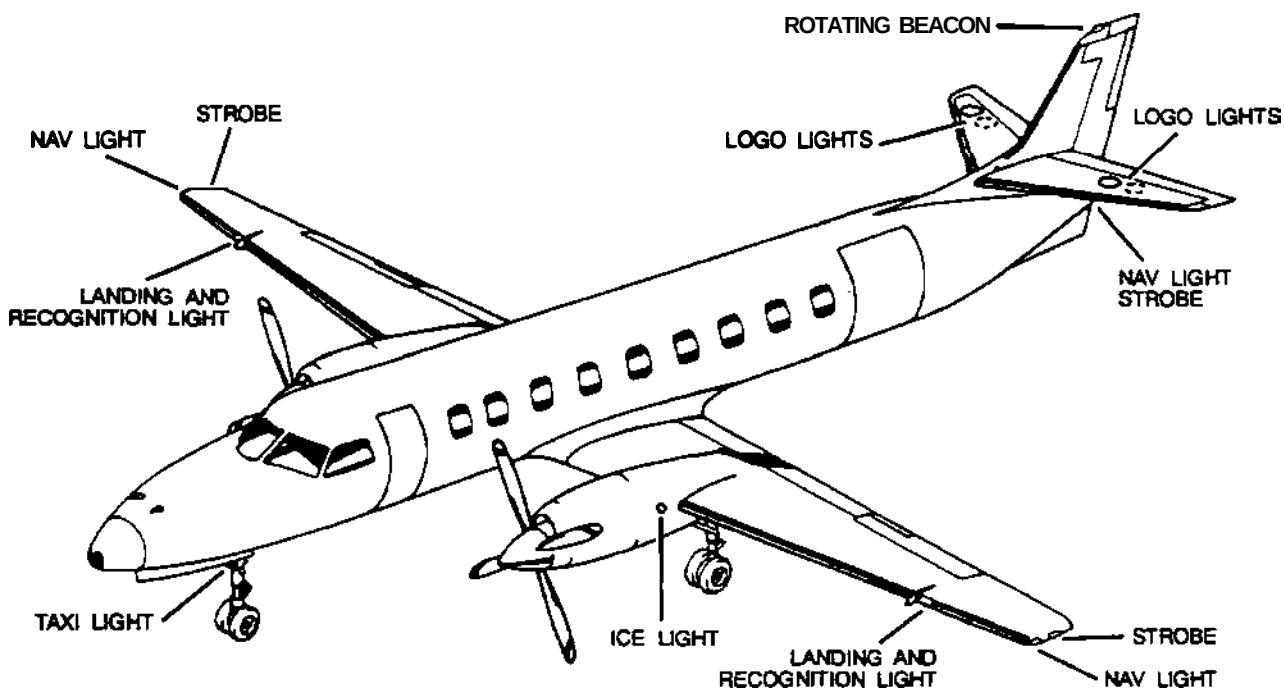


Figure 1-15. Exterior Lights



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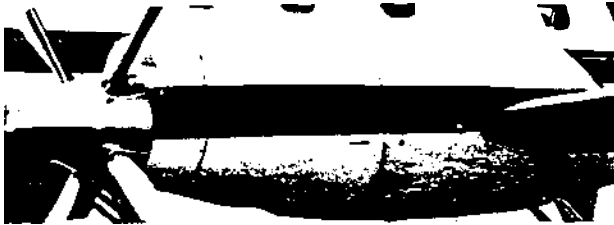


Figure 1-17. TPE 331 Engine

The engine power section consists of a **two-stage centrifugal compressor**, an annular reverse-flow combustion chamber, and a three-stage axial-flow turbine.

The propellers are oil-operated, constant-speed, full feathering, and reversible. The blades move to feather when oil pressure is not **present** or is less than the propeller feathering spring pressure.

POWERPLANT

The airplane is powered by the 1,000-shp Garrett TPE 331-11U-611G or 612G turbo-prop engine (Figure 1-17).

Engine controls consist of power levers, speed levers, automatic negative torque sensing, a single red line computer, and a **temperature-limiting system**. The controls are shown in Figure 1-18.

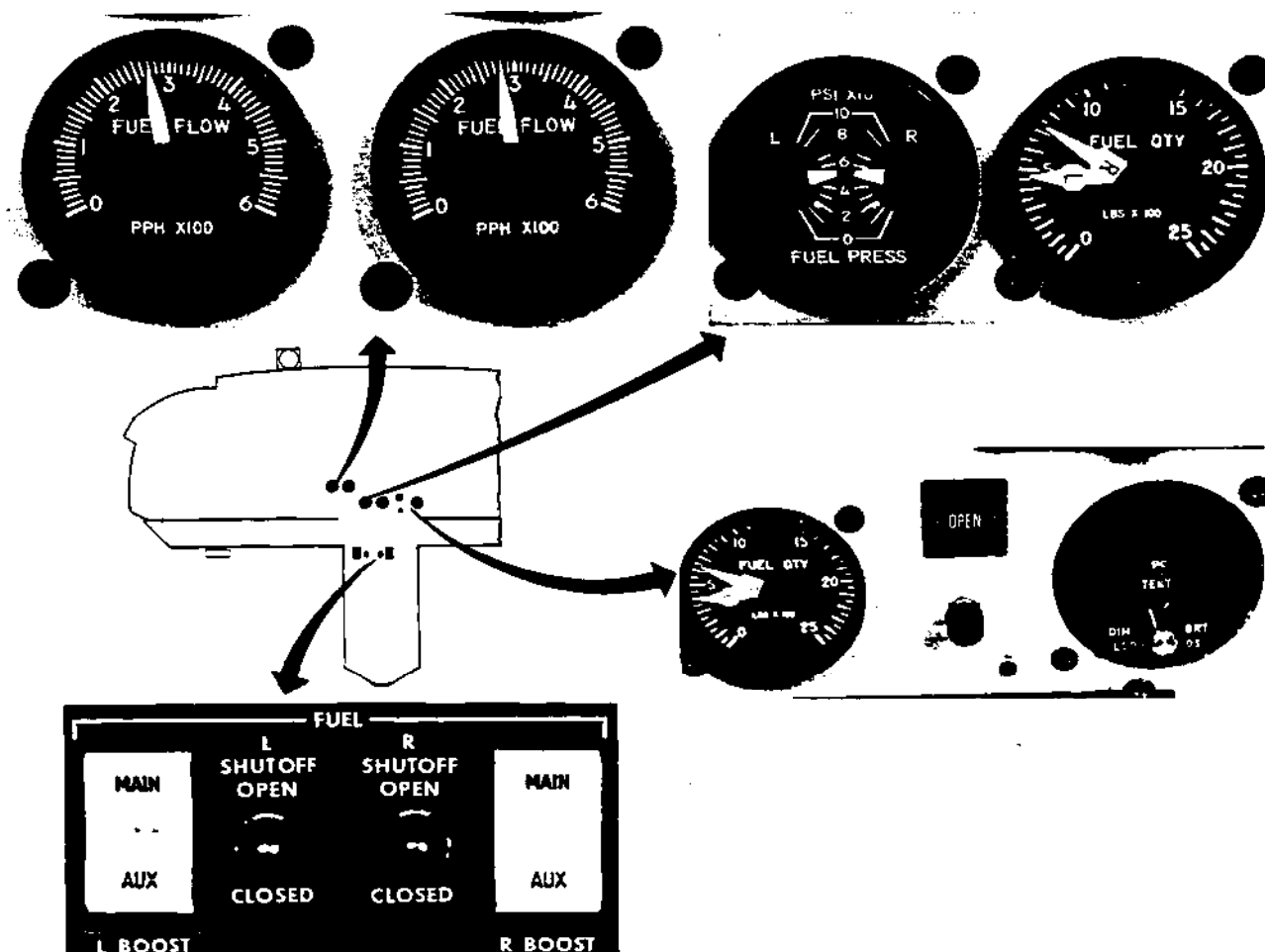


Figure 1-16. Fuel Controls and Indicators

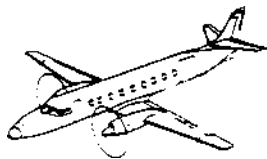


Figure 1-18. Engine Controls

The engine indicators, located on the instrument panel, are shown in Figure 1-19. The indicators for each engine include:

- EGT
- Torque
- Percent rpm
- Fuel flow
- Combined oil temperature and oil pressure
- Fuel pressure

FIRE PROTECTION

There are overheat detectors for the engines and wings and fire extinguishers inside the engine nacelles. Detection of an overheat situation illuminates cockpit warning lights on the annunciator panel and on the fire extinguisher control assembly (Figure 1-20). The fire extinguishers are discharged from the cockpit. During preflight, the pressure gage for each fire extinguisher bottle should be checked (Figure 1-21).

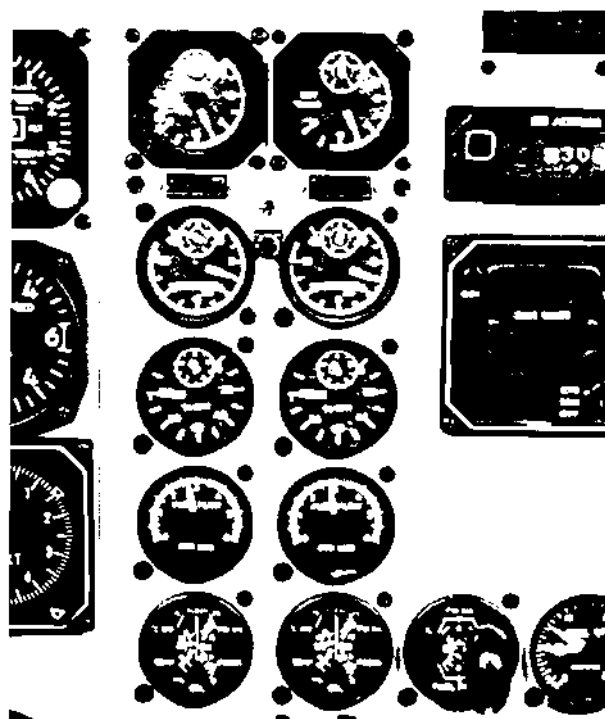


Figure 1-19. Engine Indicators



Figure 1-20. Fire Protection Controls and Indicators



Figure 1-21. Engine Fire Extinguisher Pressure Gage



PNEUMATICS

The pneumatic system uses regulated and unregulated bleed air and vacuum. Either engine is sufficient to meet all requirements, but both are normally used.

Regulated bleed air is used for pressurization, air-conditioning and deice boot inflation, window defogging, hydraulic reservoir pressure, door seal inflation, and vacuum generation. Unregulated bleed air is used for engine and nacelle inlet anti-icing.

Bleed-air shutoff valves, controlled by left and right toggle switches on the copilot's switch panel, control the air for pressurization and air-conditioning systems (Figure 1-22).

Bleed air is routed through an air ejector to provide vacuum for pressurization control, deice boot deflation, and some instrument operation. A suction indicator on the instrument panel (Figure 1-23) and an amber LOW SUCTION annunciator light (Appendix B) allow the pilot to monitor the vacuum system operation.

ICE AND RAIN PROTECTION

Metro/Merlin airplanes are supplied with electrical deicing for the propellers and oil cooler duct inlets, electrically heated pitot heads and SAS vane, heated windshield panels, pneumatic deice boots on wings and horizontal stabilizer leading edges, compressed bleed air for engine nacelle inlets, ice-free static sources, and electrically powered windshield wipers.

To prevent moisture formation between the dual windshield and the side window panes, a window purge (defog) system taps bleed air from the door seal inflation plumbing.

Figure 1-24 shows the airplane location of typical ice and rain protection devices. On

the 16,000-pound versions of the airplane, the wing deice boots extend inboard of the nacelles.

The controls for ice protection devices are shown in Figure 1-25.

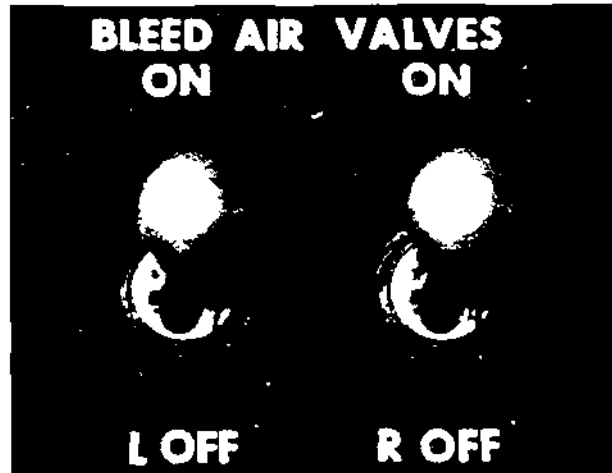


Figure 1-22. Bleed-Air Valve Switches

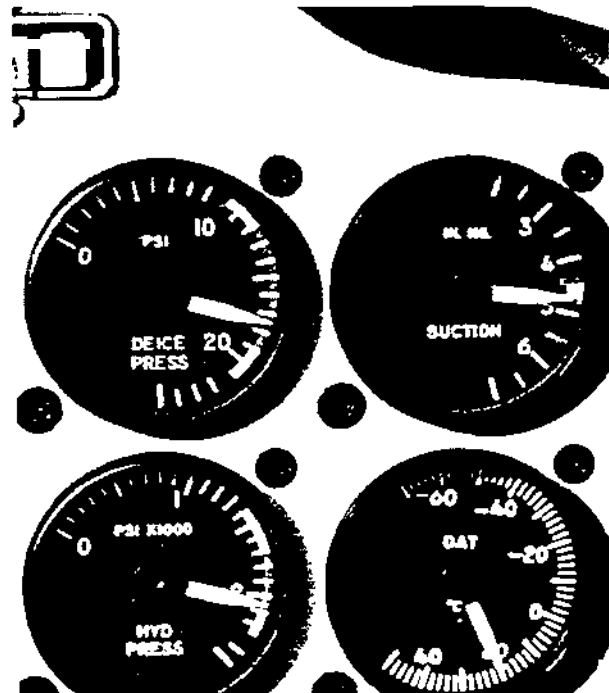


Figure 1-23. Suction Indicator



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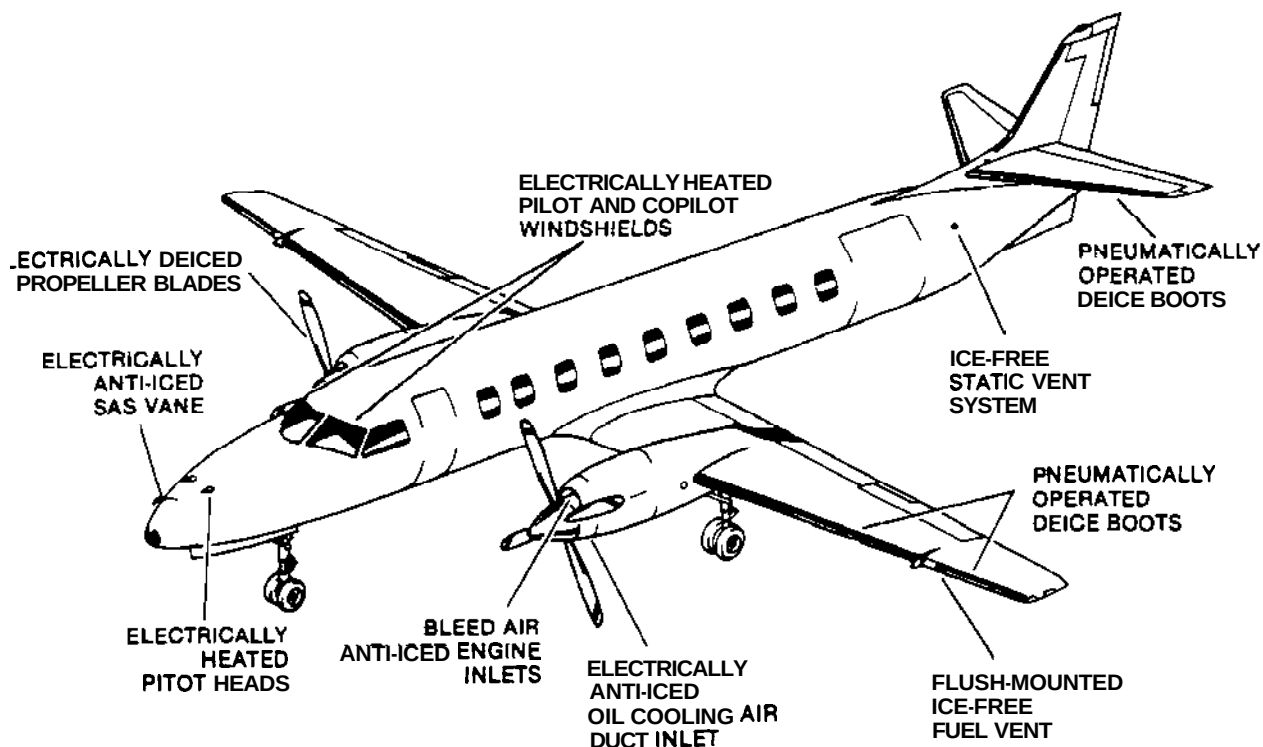


Figure 1-24. Ice and Rain Protection Devices

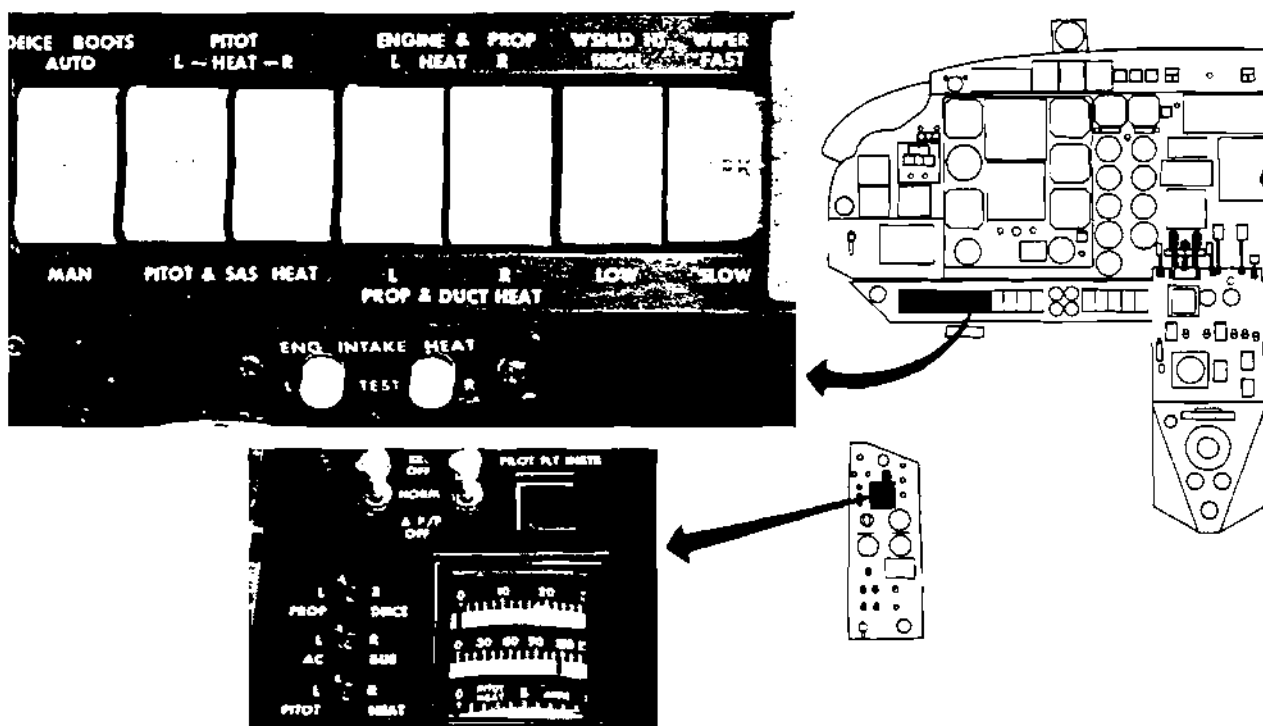


Figure 1-25. Ice and Rain Protection Controls



AIR CONDITIONING

The air-conditioning system supplies cold, hot, and fresh air to the cabin and cockpit (Figure 1-26). Each engine supplies bleed air to a cooling turbine, providing two independent cooling systems, each capable of providing total airplane requirements. Either bleed-air system may be operated on the ground when the respective engine is operating.

Hot bleed air is routed to the airplane center section where it is mixed with cold bleed air to provide temperature-controllable conditioned air. An automatic temperature control system senses and regulates the temperature within the airplane.

A fresh air fan system is provided primarily for cockpit ventilation during ground operation. The blower and motor are located in the nose baggage compartment. While this system is normally deactivated during flight, an override is provided to allow operation during flight

if neither air-conditioning system is operating properly. However, it should *not* be operated during pressurized flight.

PRESSURIZATION

The pressurization system is designed to provide a 7.0-psi differential cabin pressure which allows a sea level cabin altitude up to a 16,800-foot pressure altitude, and a 7,400-foot cabin altitude at a 31,000-foot pressure altitude. Two safety relief valves limit cabin pressure differential to approximately 7.25 in the event of pressure controller failure.

The pressurization system is based on the air-conditioning system flow into the pressure vessel. The volume of air passed overboard through the outflow valve located on the aft pressure bulkhead, or through the emergency dump valve on the forward pressure bulkhead (Figure 1-27), will determine the actual cabin pressure.

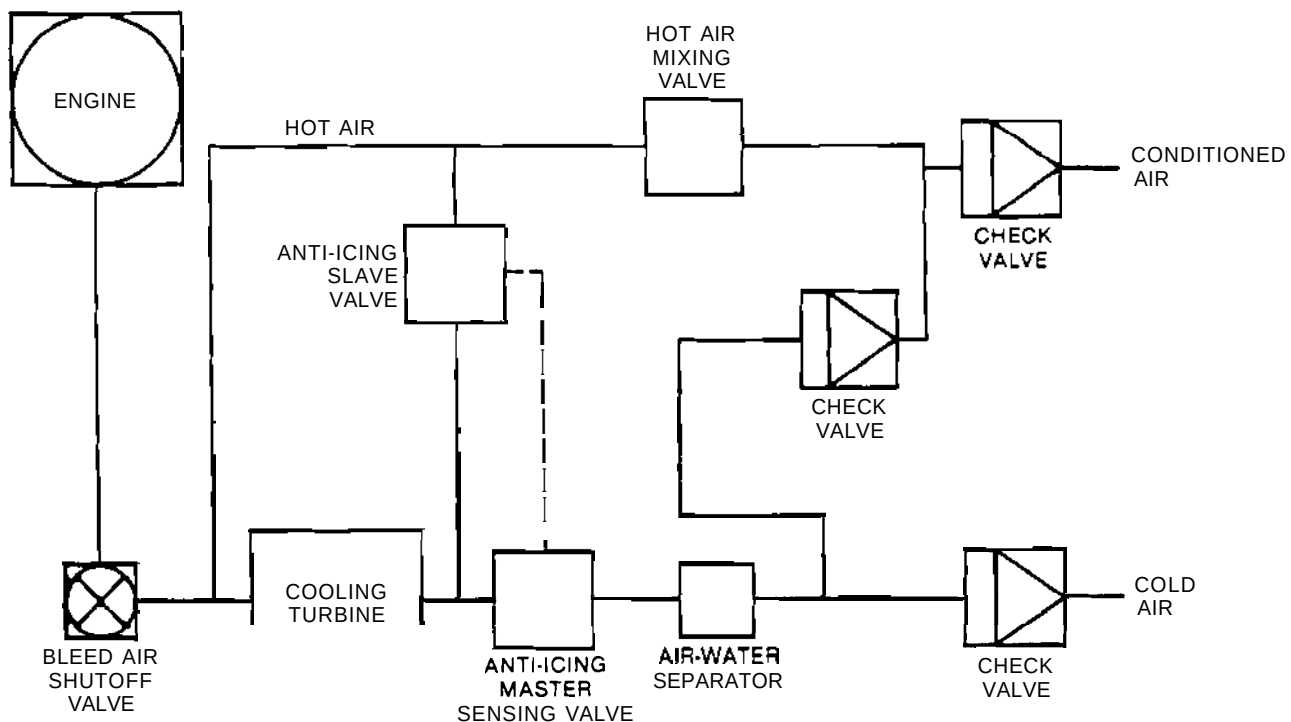


Figure 1-26. Environmental Control Distribution

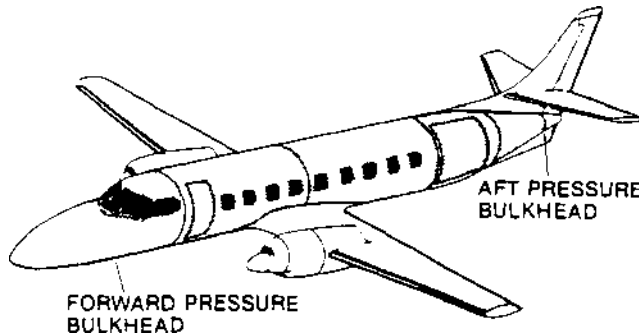


Figure 1-27. Pressurized Vessel

The cabin pressure automatic controller is located on the instrument panel and contains a RATE control knob and a CABIN ALTitude selector knob. A CABIN PRESS MANUAL CONTROL and a CABIN PRESS SELECTOR are both used during manual operation and are located on the pilot's left console. These controls are shown in Figure 1-28.

HYDRAULIC POWER SYSTEMS

The airplanes have a 2,000-psi hydraulic system powered by two engine-driven pumps. Loss of a single engine or its pump will not preclude hydraulic operation, but the systems will function at a reduced rate. Hydraulic pressure actuates the landing gear, flaps, power brakes, and nosewheel steering. Pressure surges within the system are dampened by an accumulator.

A hydraulic hand pump, which draws reserve fluid from the same reservoir as the engine-driven pumps, provides pressure for emergency landing gear extension.

Low hydraulic pressure annunciator panel lights warn of pump failure or low pressure caused by any other malfunction. A single electrically operated hydraulic pressure indicator displays

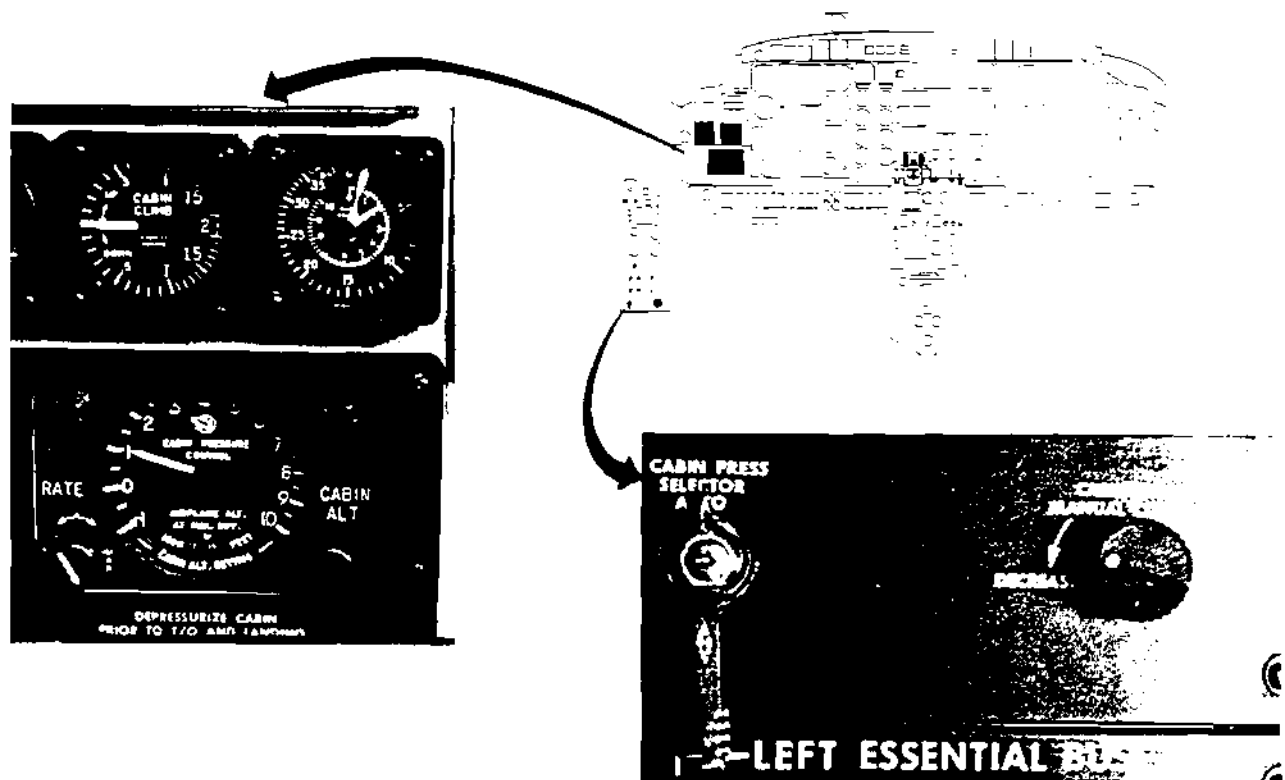
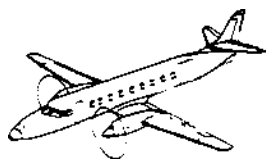


Figure 1-28. Pressurization Controls



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normal system or hand pump pressure, whichever is higher. Figure 1-29 shows the hydraulic controls and indicators.

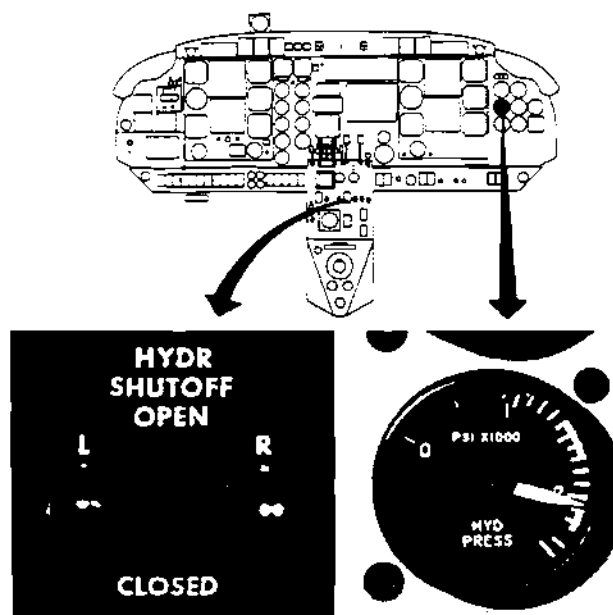


Figure 1-29. Hydraulic Controls and indicators

LANDING GEAR AND BRAKES

The airplanes have fully retractable landing gear, each equipped with dual wheels. Self-adjusting hydraulically actuated disc brakes are installed on each main landing gear wheel.

Extension and retraction are electrically controlled and hydraulically actuated. Normal operation is controlled by a landing gear handle located on the pedestal (Figure 1-30). Electrical power is supplied by one of the bus transfer switches.

Emergency free-fall gear extension is accomplished by use of an emergency release lever beside the copilot's seat. There is no backup for gear retraction.

Gear position is indicated by three green lights and three red lights. A green light indicates that its respective gear is locked down; red signifies gear in transit. When all three gear are up and locked, no lights are illuminated. The test button illuminates all six indicators.

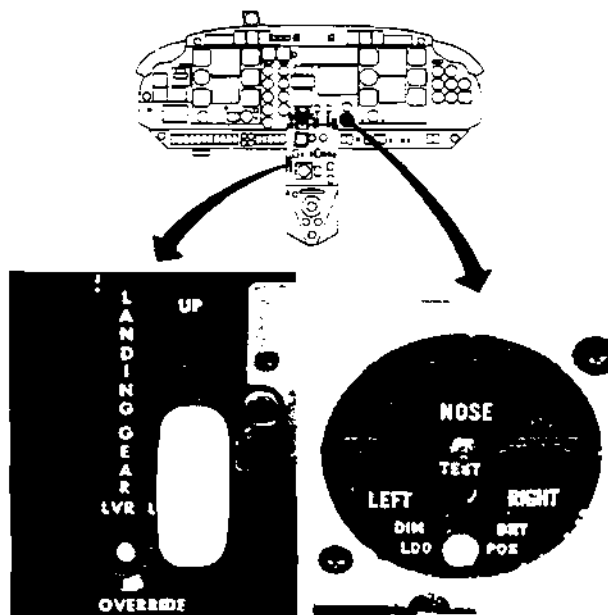


Figure 1-30. Gear Position Indicators and Control Handle

Gear doors operate mechanically by gear action. Main gear doors are closed after gear extension and retraction. The nose gear doors remain open while the gear is extended.

The nosewheel is automatically centered when airplane weight is lifted off the nosewheel.

Nosewheel steering is electrically controlled and hydraulically powered. It is armed by the NOSE GEAR STEERING switch on the left console (Figure 1-31). When armed, it is activated by pressing the nose steering button on the left power lever or by positioning the right speed lever to the LOW position.



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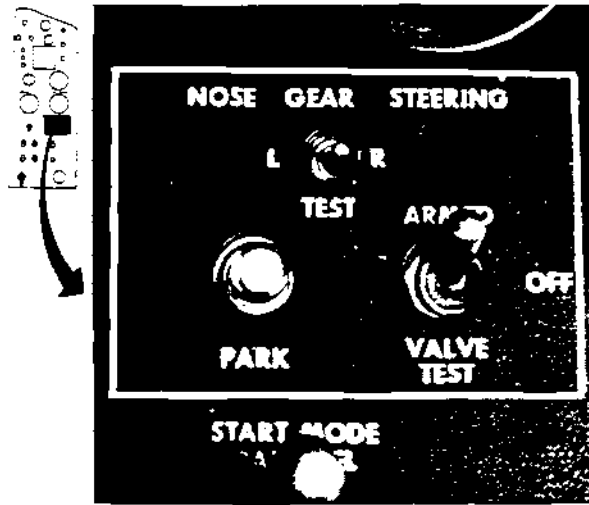


Figure 1-31. Nose Gear Steering Panel

Manual brakes, without antiskid, are activated by toe brakes on the pilot and copilot rudder pedals. Hydraulic fluid for this system is contained in a brake system hydraulic reservoir, independent of the main hydraulic system.

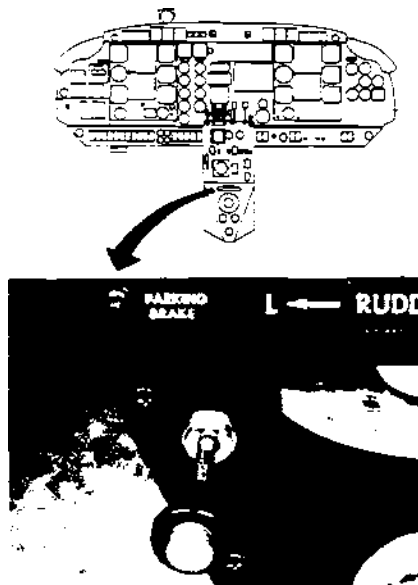


Figure 1-32. Parking Brake Control

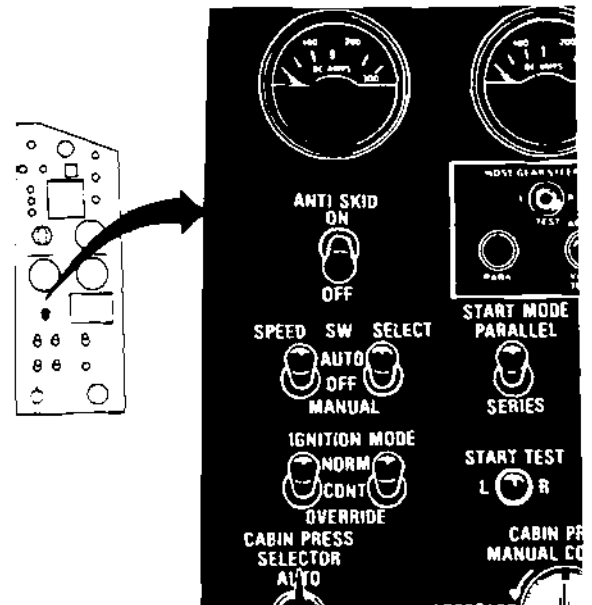


Figure 1-33. Antiskid Control Switch

Shuttle valves transfer the braking function to either the pilot or copilot brake master cylinders, whichever set is actuated first, but prevents simultaneous actuation of a brake by both pilot and copilot.

A parking brake control (Figure 1-32) is located on the pedestal. Depressing the brake pedals while holding the parking brake control out traps the brake pressure within the lines, locking the brakes.

An antiskid power brake system (Figure 1-33) is optional, operating from the airplane hydraulic system. If the antiskid system or hydraulic system fails, conventional braking is available. The antiskid system will function only when the power brake system is operating. An antiskid control box takes the signal from each wheel transducer. If one wheel slows abnormally, all brakes will be released.



FLIGHT CONTROLS

The primary flight controls, ailerons, rudder, and elevators (Figure 1-34) are manually operated by either the pilot or the copilot, using a conventional yoke and rudder pedal arrangement. Rudder and aileron trim tabs are mechanically controlled from trim wheels on the cockpit pedestal. The elevators do not use trim tabs. Instead, the horizontal stabilizer is electrically moved to provide pitch trim.

An internal, cable-operated gust lock system locks the rudder and ailerons in the neutral position when engaged. The power levers are also blocked from going forward of FLT IDLE. The handle (Figure 1-35) is located forward of the power levers on the pedestal.

Stabilizer trim control is transferable to switches on either the pilot's or copilot's control wheels by the TRIM SELECT switch (Figure 1-36) on the pedestal. An auxiliary trim switch on the pedestal facilitates the pilot's operation of the copilot's trim should a **malfunction** occur in the pilot's trim circuitry.

One of two trim-in-motion sonalerts, mounted overhead in the cockpit, sounds when a stabilizer trim is actuated.

A stabilizer trim indicator is located on the pilot's instrument panel.

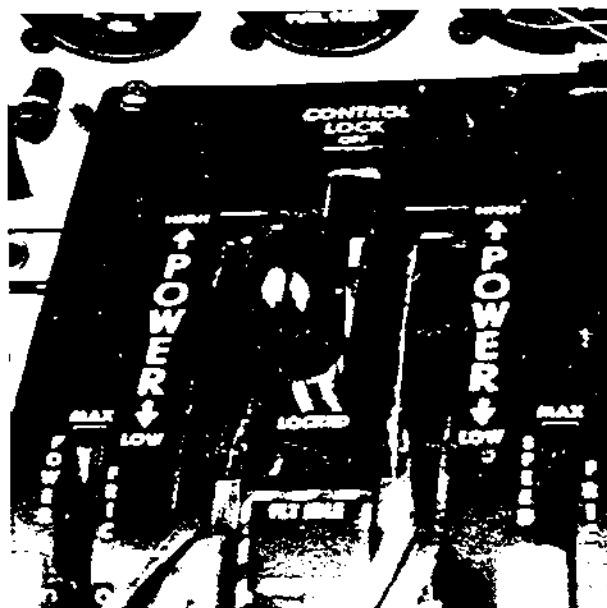


Figure 1-35. Gust Lock Handle

The wing flaps are electrically controlled, hydraulically actuated, and mechanically interconnected to ensure symmetrical operation. They are controlled by operation of a flap control handle (Figure 1-37) located on the pedestal. There is no provision for emergency flap extension or retraction in the event of DC power or hydraulic system failure. A flap position indicator on the copilot's instrument panel displays position sensed on the left flap.

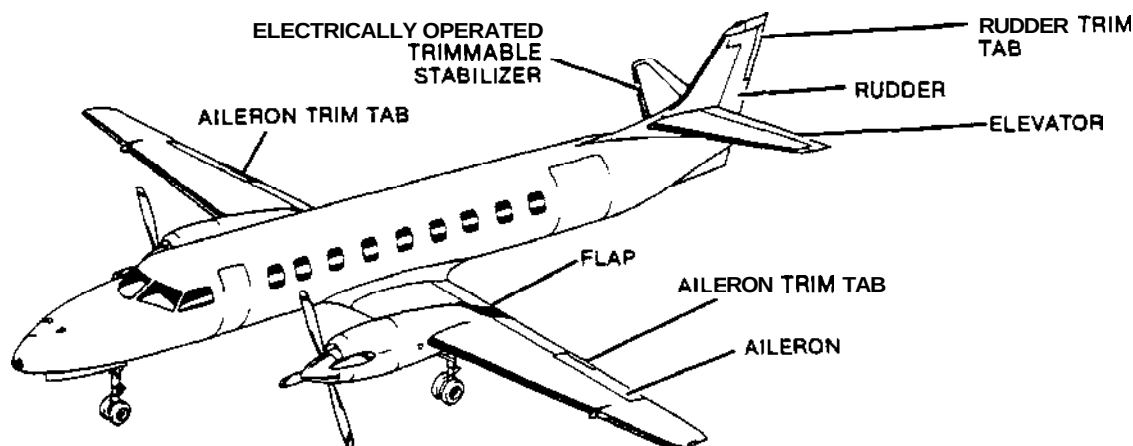


Figure 1-34. Flight Control Surfaces

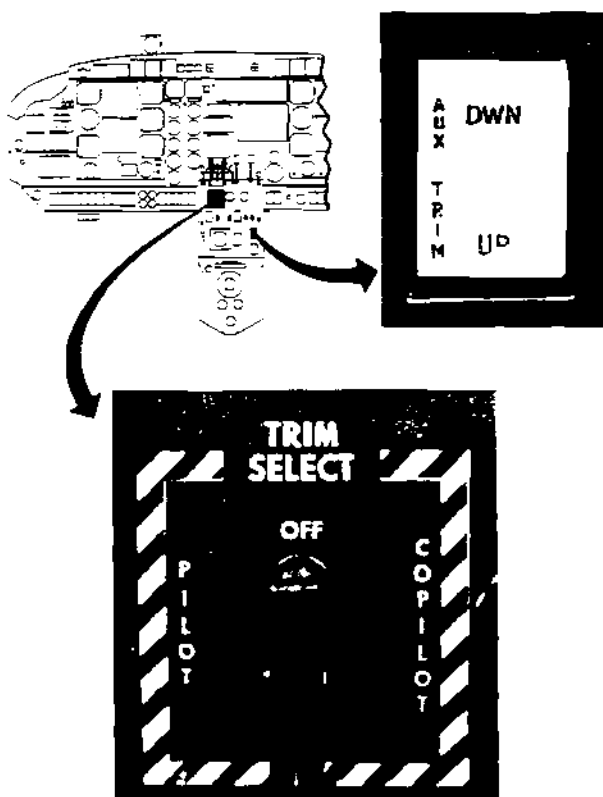


Figure 1-36. Pitch Trim Control Switches

PITOT-STATIC SYSTEM

Pitot

A separate pitot mast is installed on the pilot's and the copilot's side of the airplane nose. Each supplies its respective airspeed indicator with independent pitot reference pressure.

Static System

Separate balanced static systems provide reference pressure to the pilot's and copilot's rate-of-climb, altimeter, and airspeed indicators. Two static ports are located on each side of the aft fuselage, aft of the cargo door entrance.

Alternate Static System

The pilot can select an alternate static source by positioning the handle on the lower left

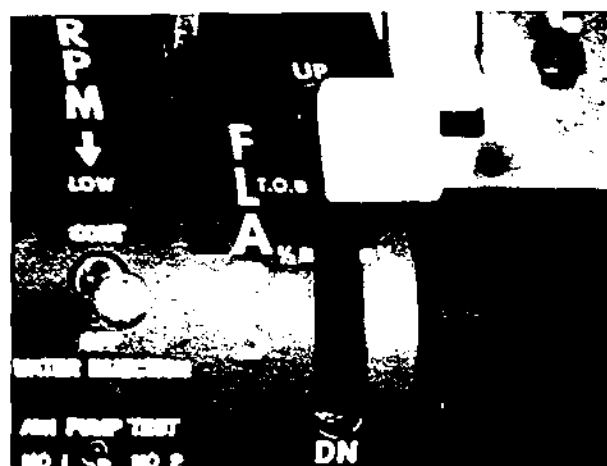


Figure 1-37. Flap Lever

side of the instrument panel. Selection of the alternate source provides its reference only to the pilot's instruments.

OXYGEN SYSTEM

The oxygen system, installed as standard equipment, is designed for use in the event of pressurization failure, smoke, or for medical needs.

Two diluter/demand crew oxygen masks, sufficient passenger masks, an oxygen cylinder, and crew controls and indicators are provided.

Merlin IVC—A 1,850-psi cylinder is located behind the aft baggage compartment bulkhead. The overpressure rupture disc is located on the right side of the airplane tail section. Passenger masks are stowed in nine overhead compartments.

Metro III—One or two 1,850-psi cylinders are located behind the aft compartment bulkhead. An overpressure rupture disc is located on the

right side of the airplane tail section. Passenger masks can either be of the dropout type or will be stowed in a seat-back pocket and must be plugged into outlets before use.



WALKAROUND

The following section is a pictorial walk-around. It shows each item called out in the exterior power-off preflight inspection. The fold-out pages at the beginning and end of the walkaround section should be unfolded before starting to read.

The general location photographs do not specify every checklist item. However, each item is portrayed on the large-scale photographs that follow.

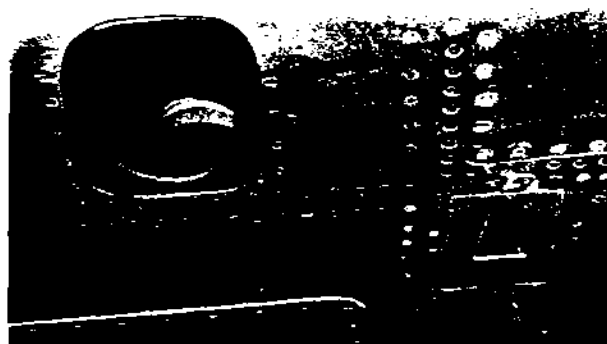


WALKAROUND INSPECTION

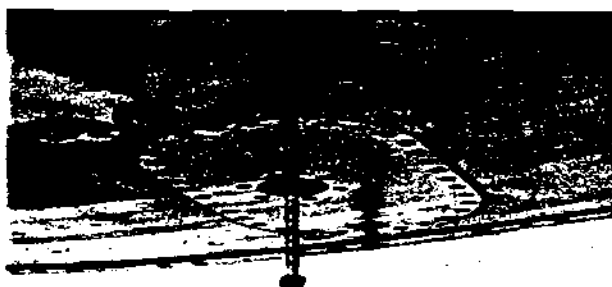
LEFT WING



1. FUEL SUMPS—DRAIN



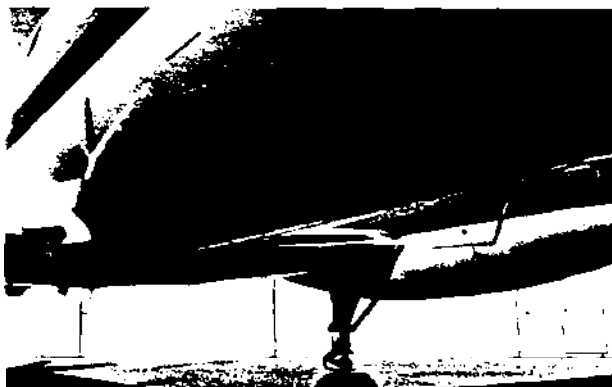
4. LEADING-EDGE RAM-AIR SCOOP—CLEAR



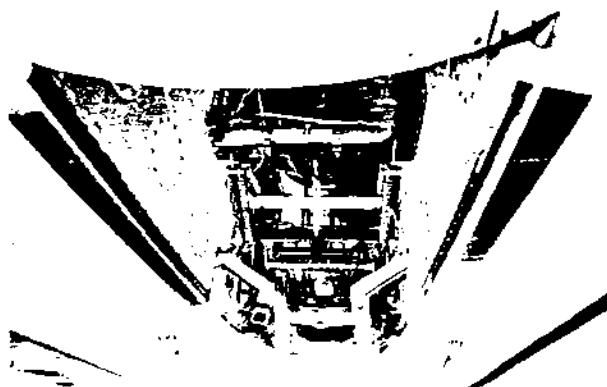
2. MAGNA STICK—CHECK TANK QUANTITY



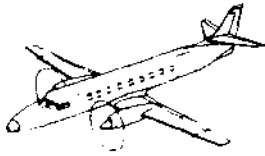
5. GEAR DOORS (FIRST FLIGHT OF DAY)—OPEN



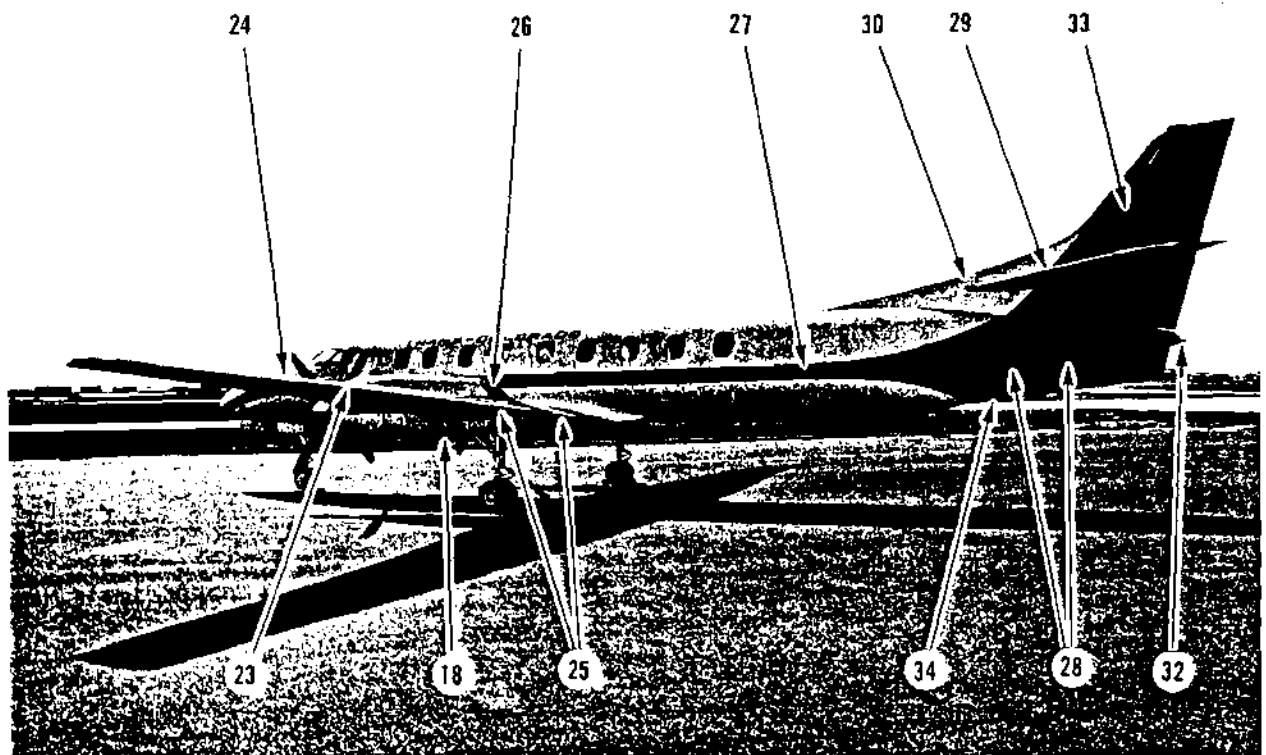
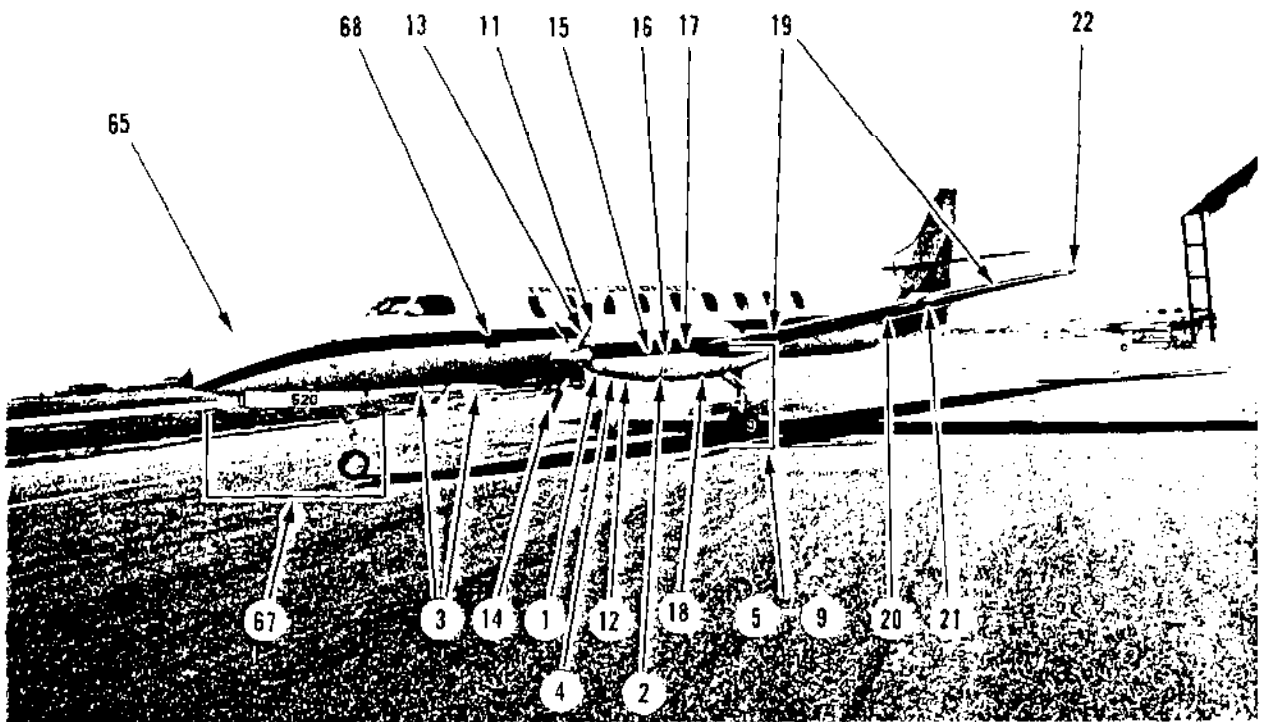
3. LOWER ANTENNAS—CONDITION



6. LANDING GEAR, BRAKES, TIRES, HUB CAPS, AND WHEEL WELL—CONDITION

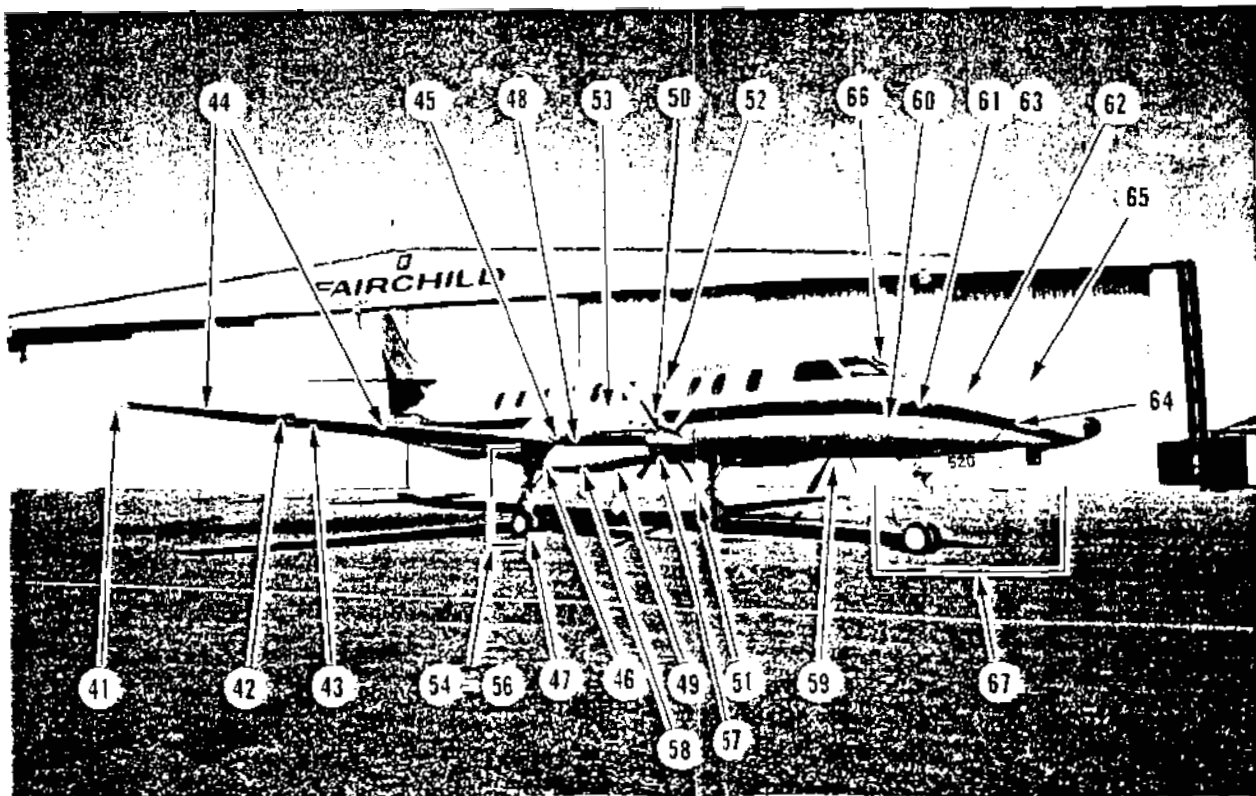
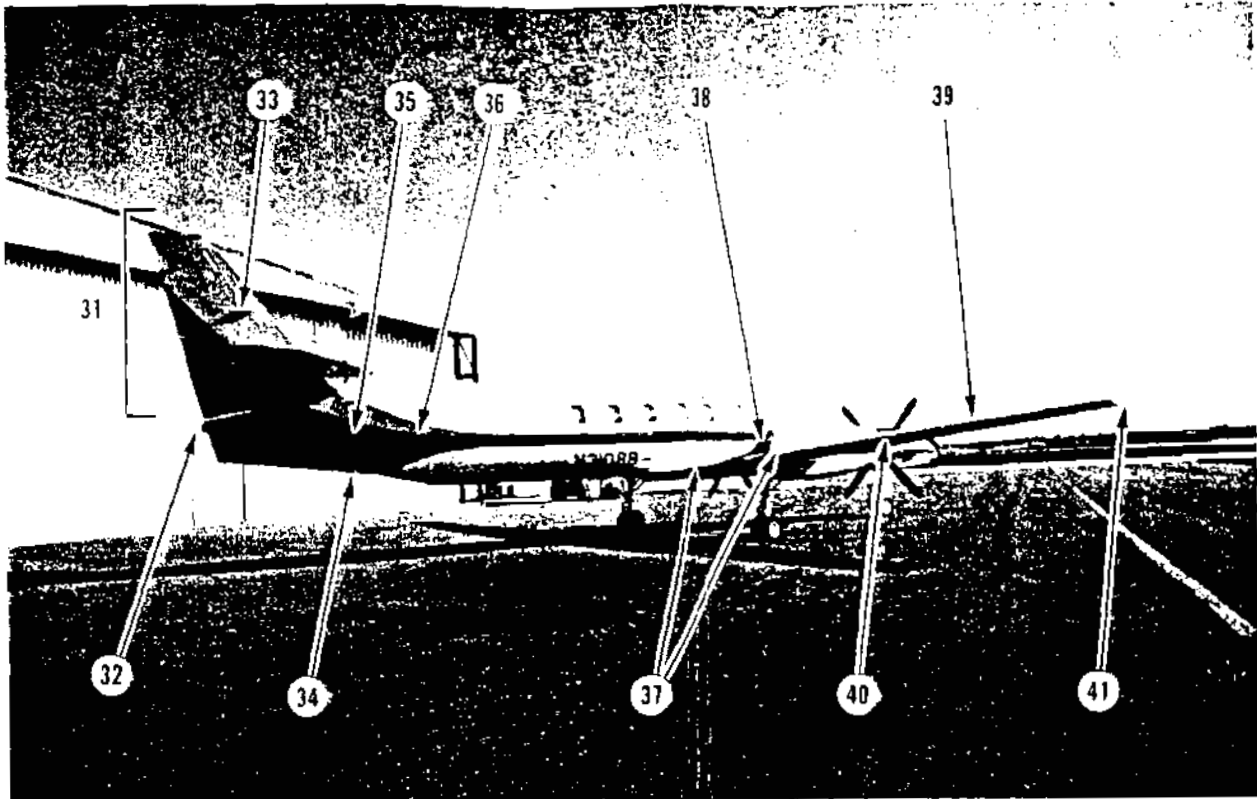


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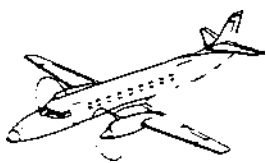




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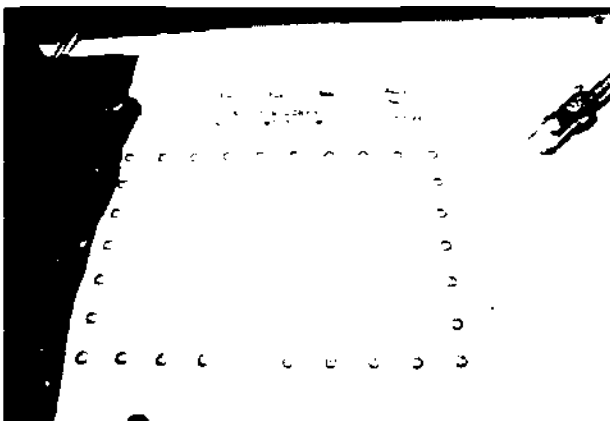


FOR TRAINING PURPOSES ONLY

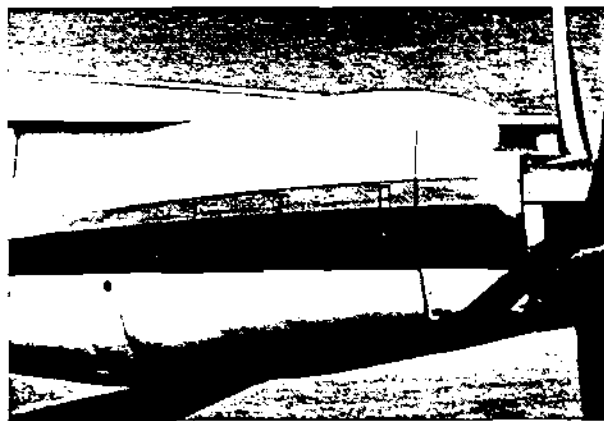


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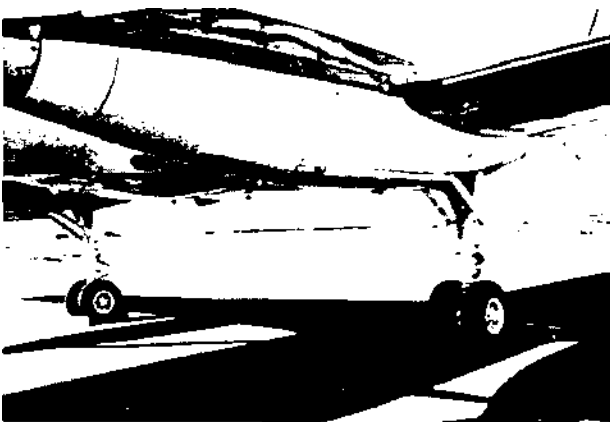
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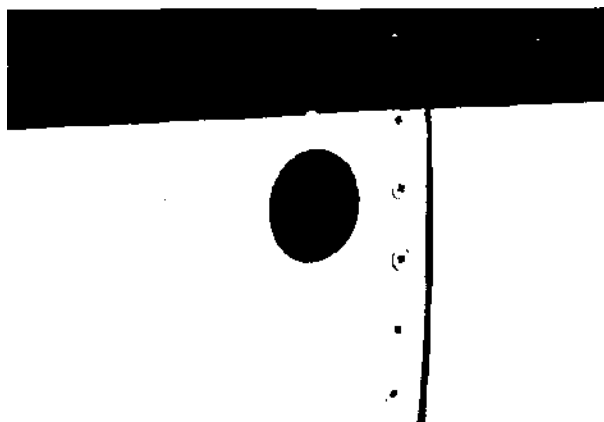
7. GENERATOR CIRCUIT BREAKERS—IN



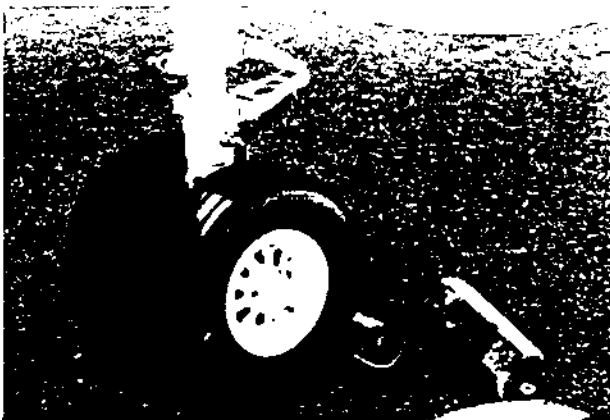
10. COWLING AND DOORS—SECURE



8. GEAR DOORS—CLOSED



11. FIRE EXTINGUISHER BOTTLE PRESSURE—CHECK



9. TIE DOWNS AND CHOCKS—REMOVE



12. OIL COOLER INLET—CLEAR AND CONDITION

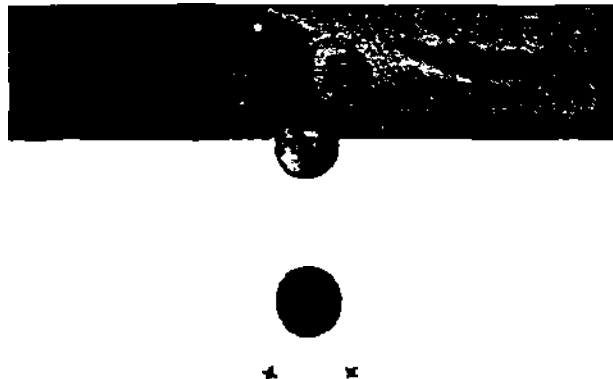


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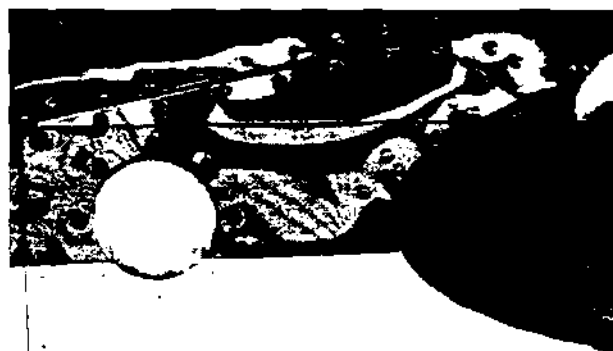
13. ENGINE INLET AND SENSORS—CLEAR AND CONDITION



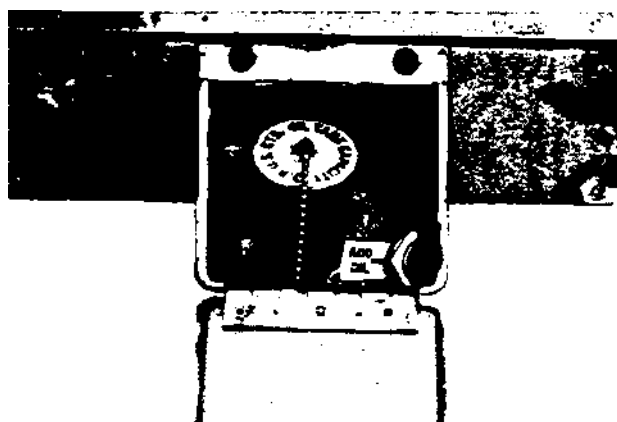
16. HYDRAULIC RESERVOIR SIGHT GLASSES—CHECK



14. PROPELLER AND PROPELLER DEICE BOOTS—CHECK FREE ROTATION AND CONDITION



17. WING ICE DETECTOR LIGHT—CHECK



15. ENGINE OIL QUANTITY AND FILLER CAP—CHECK AND SECURE

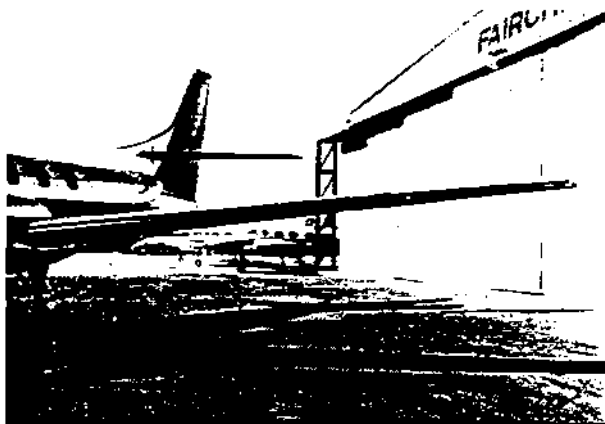


18. FUEL SUMP—DRAIN

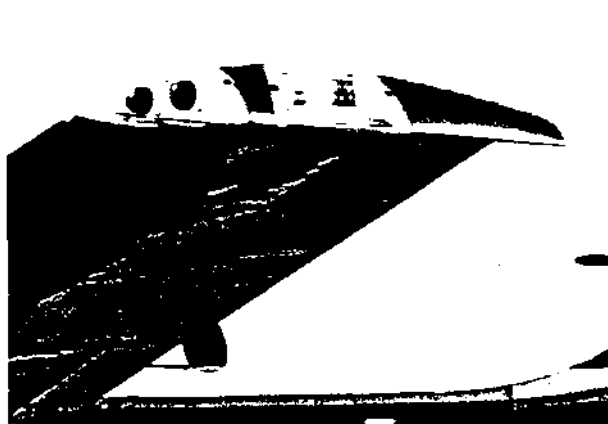


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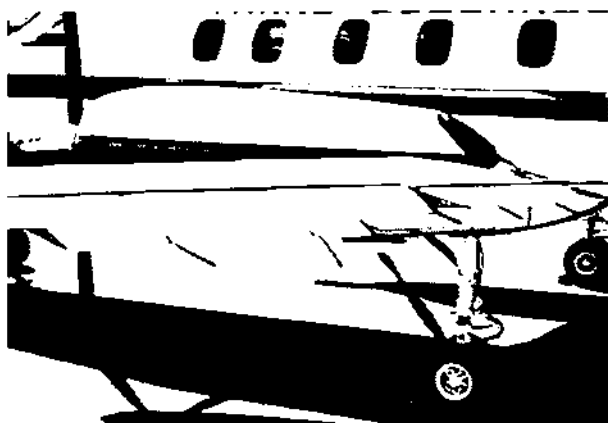
19. WING DEICE BOOTS—CONDITION



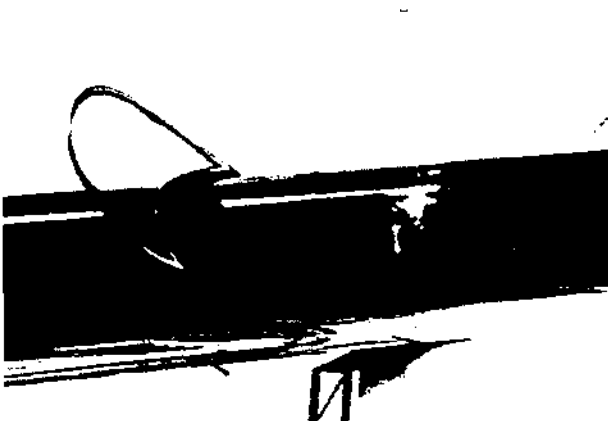
22. NAVIGATION LIGHTS—CHECK



20. FUEL VENT—CLEAR



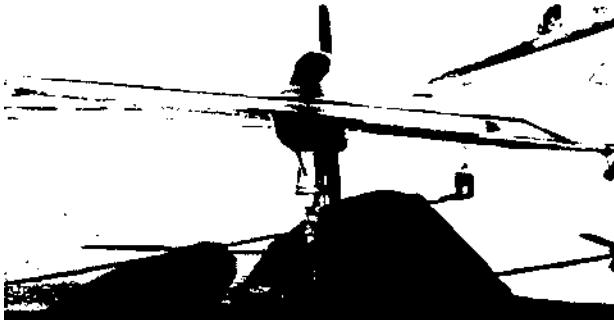
23. AILERON AND TAB—CONDITION



21. LANDING AND RECOGNITION LIGHTS AND SHIELD—CHECK



24. WING FUEL CAP—SECURE



25. FLAPS—CONDITION

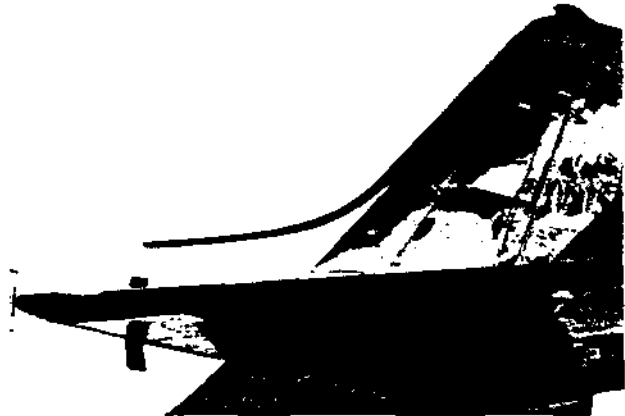


26. EXHAUST—CLEAR

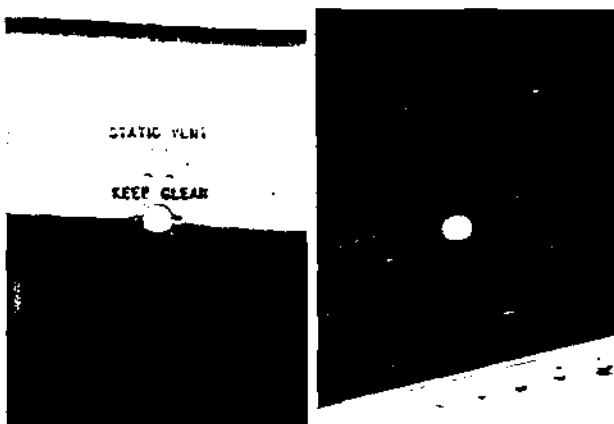
TAIL SECTION



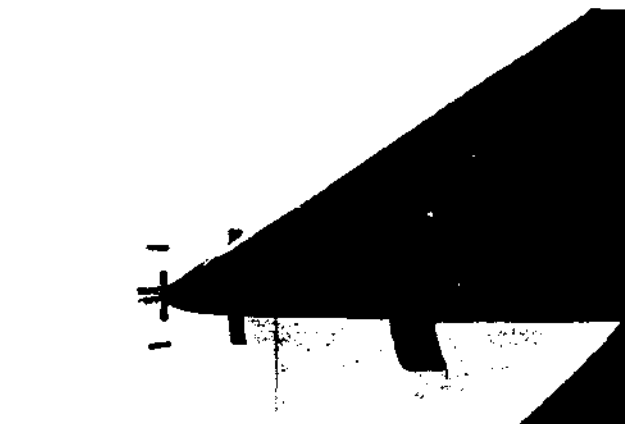
27. CARGO DOOR—SECURE



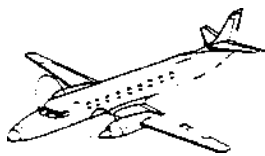
29. DEICE BOOTS—CONDITION



28. STATIC SOURCES—CLEAR



30. STABILIZER SETTING—CHECK IN AGREEMENT WITH COCKPIT INDICATORS

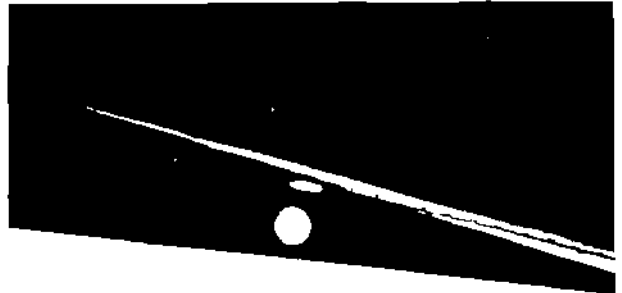


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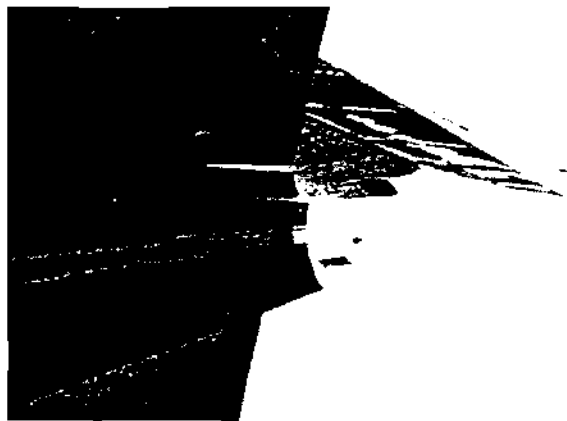
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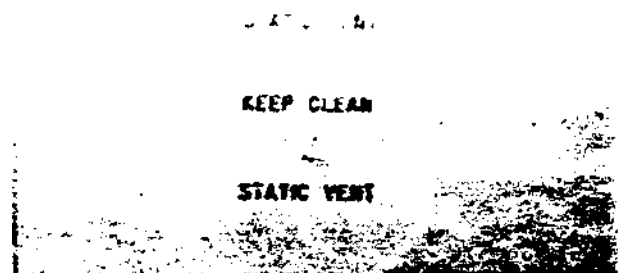
31. CONTROL SURFACES AND RUDDER TAB—
CONDITION



34. TIE DOWN—REMOVE



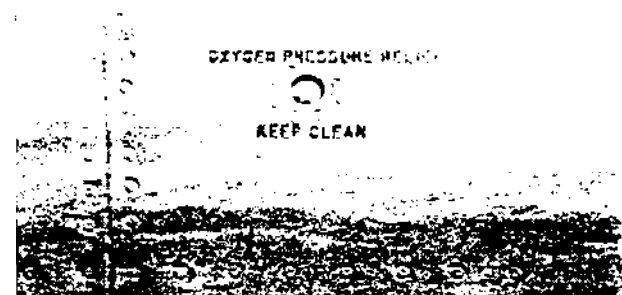
32. NAVIGATION LIGHTS—CHECK



35. STATIC VENTS—CLEAR



33. UPPER ANTENNAS—CONDITION



36. OXYGEN BOTTLE THERMAL RELIEF DISC—
CONDITION



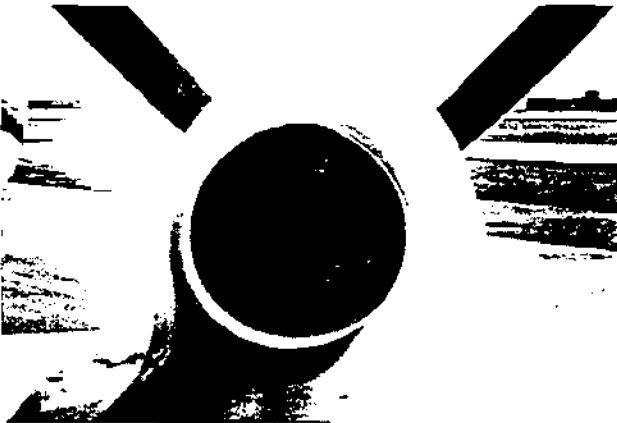
RIGHT WING



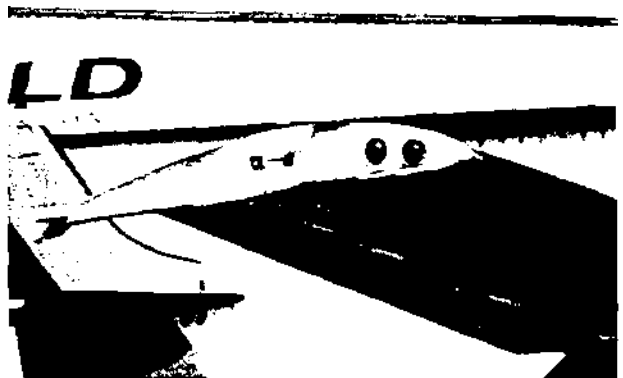
37. FLAPS—CONDITION



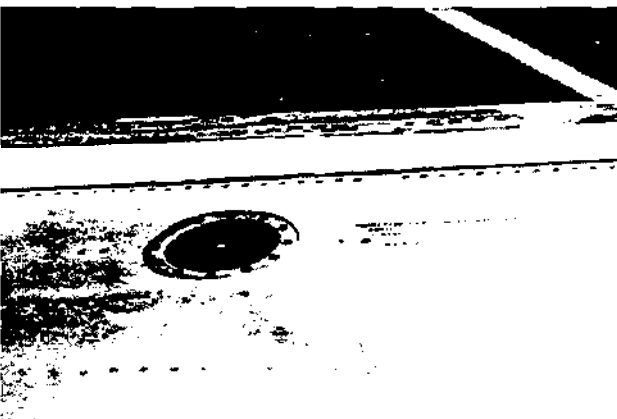
40. AILERON AND TAB—CONDITION



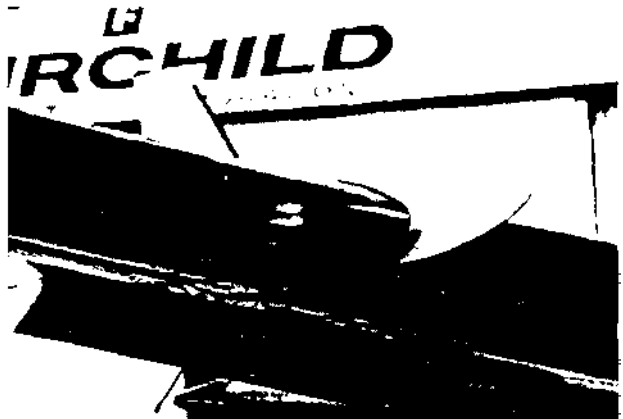
38. EXHAUST—CLEAR



41. NAVIGATION LIGHTS—CHECK



39. WING FUEL CAP—SECURE

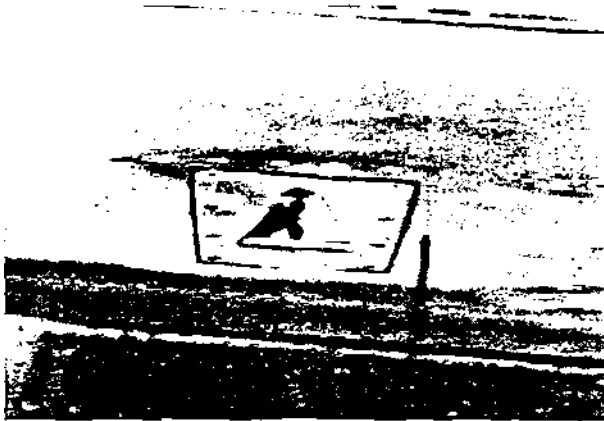


42. LANDING AND RECOGNITION LIGHTS AND SHIELD—CHECK

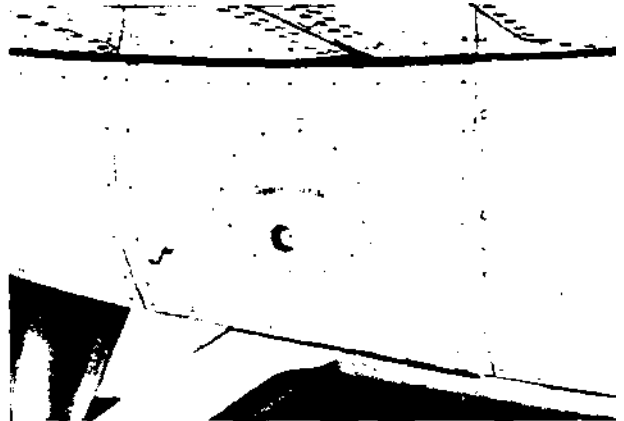


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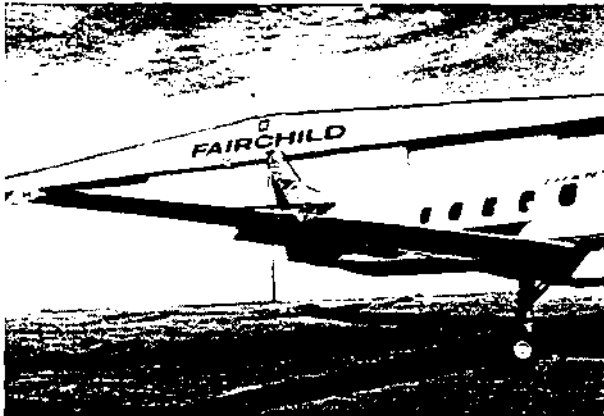
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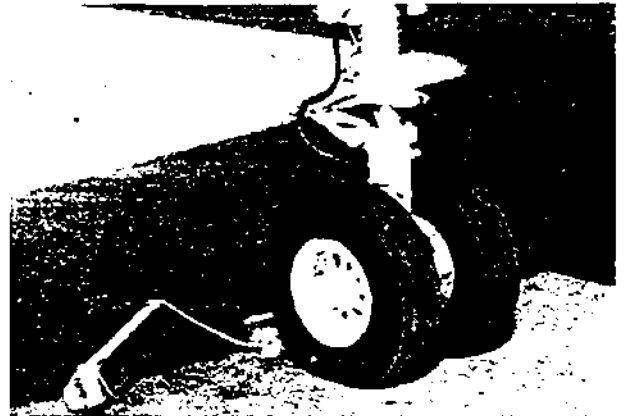
43. FUEL VENT—CLEAR



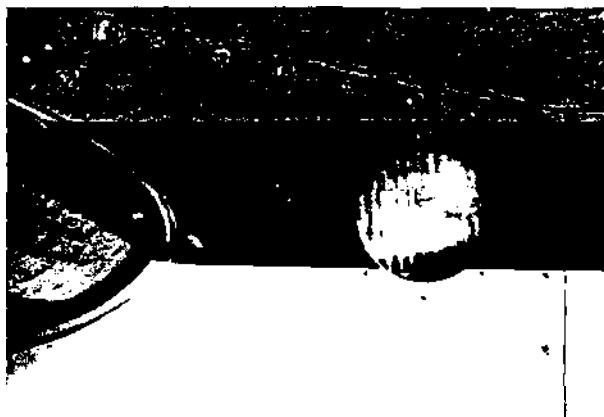
46. FUEL SUMP—DRAIN



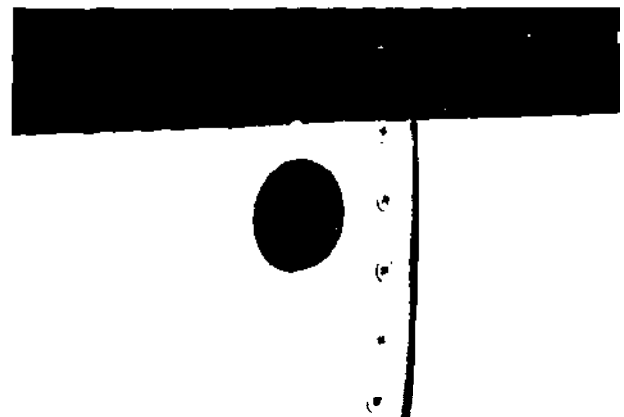
44. WING DEICE BOOTS—CONDITION



47. TIE DOWNS AND CHOCKS—REMOVE



45. WING ICE DETECTOR LIGHT—CHECK

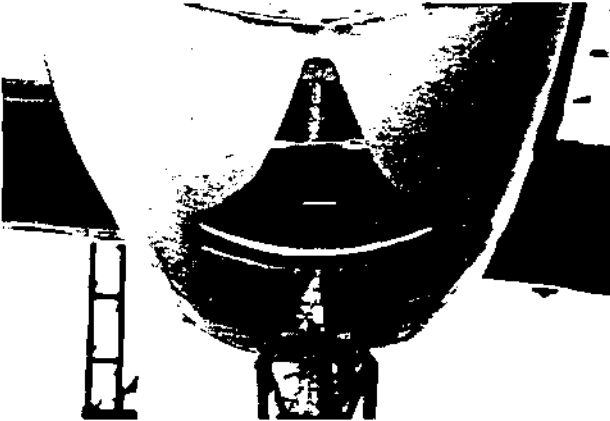


48. FIRE EXTINGUISHER BOTTLE PRESSURE—CHECK



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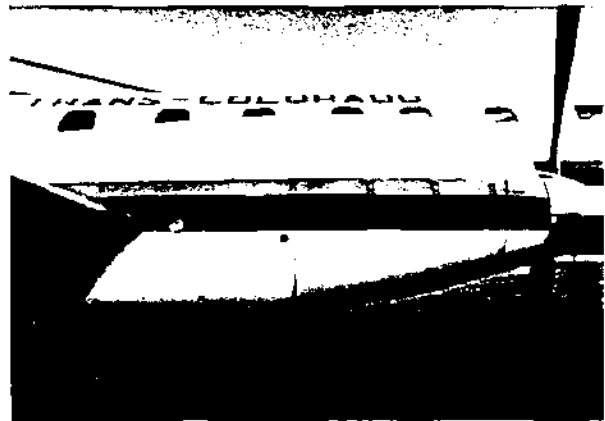
49. OIL COOLER INLET—CLEAR AND CONDITION



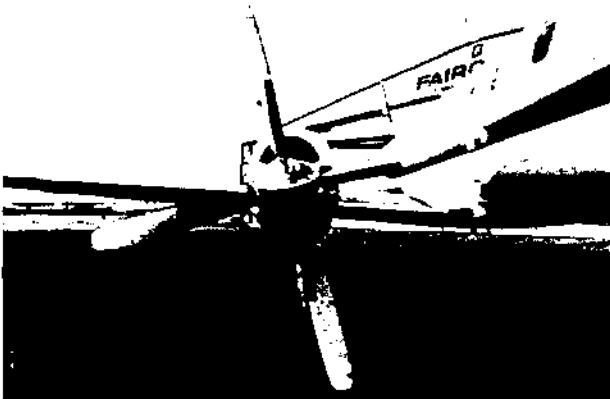
52. ENGINE OIL QUANTITY AND FILLER CAP—CHECK AND SECURE



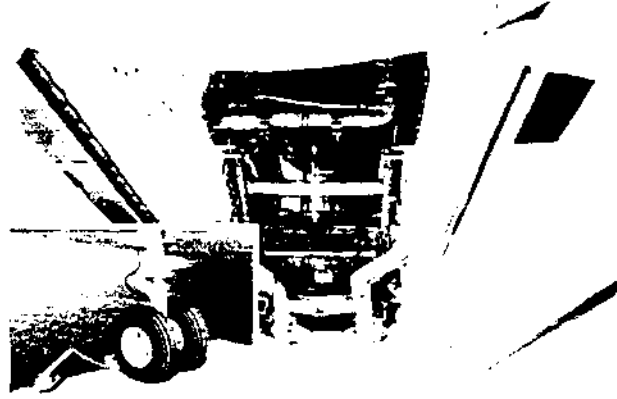
50. ENGINE INLET AND SENSORS—CLEAR AND CONDITION



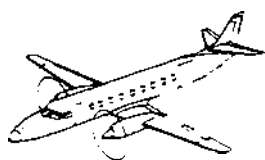
53. COWLING AND DOORS—SECURE



51. PROPELLER AND PROPELLER DEICE BOOTS—CHECK FREE ROTATION AND CONDITION

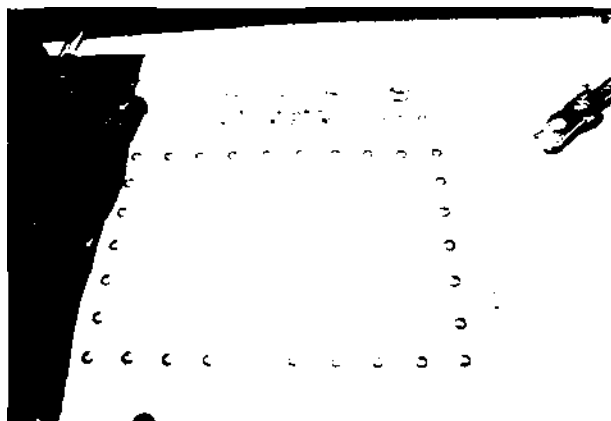


54. LANDING GEAR, BRAKES, TIRES, HUB CAPS, AND WHEEL WELL—CONDITION

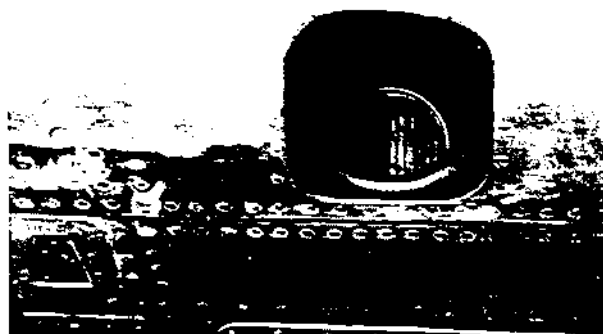


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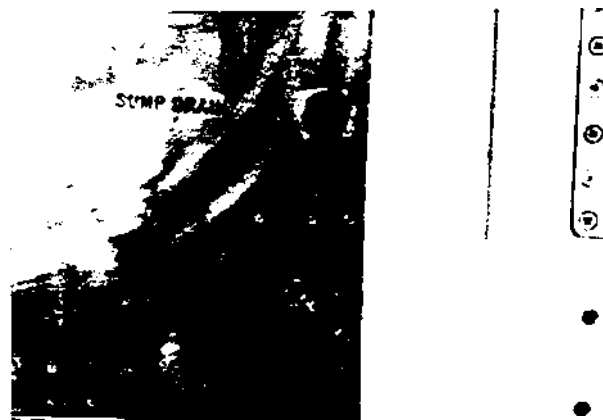
55. GENERATOR CIRCUIT BREAKERS—IN



57. LEADING-EDGE RAM-AIR SCOOP—CLEAR



56. GEAR DOORS—CLOSED



58. FUEL SUMPS—DRAIN

NOSE SECTION



59. OUTSIDE AIR TEMPERATURE SENSOR—CLEAR

STATIC VENT
●
KEEP CLEAR

STATIC VENT
●
KEEP CLEAR

60. STATIC SOURCES—CLEAR



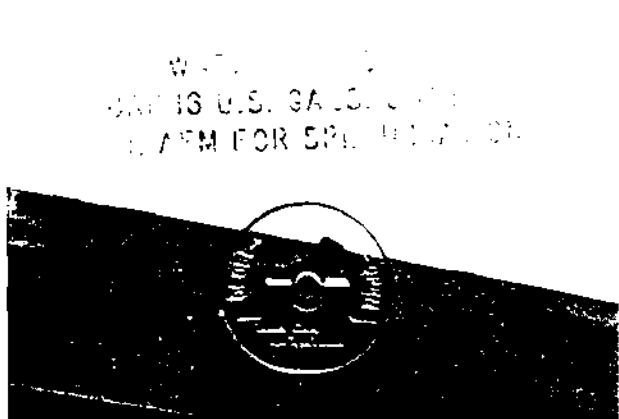
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61. CAWI TANK SIGHT GAGE—CHECK QUANTITY



64. SAS VANE—CHECK



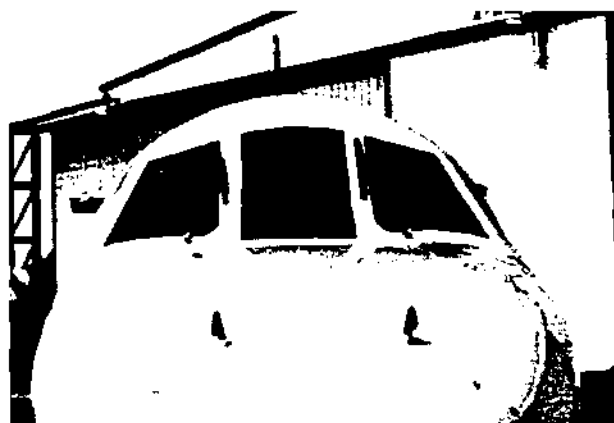
62. CAWI TANK FILLER CAP—SECURE



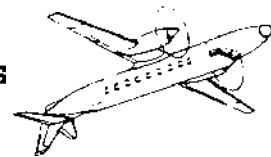
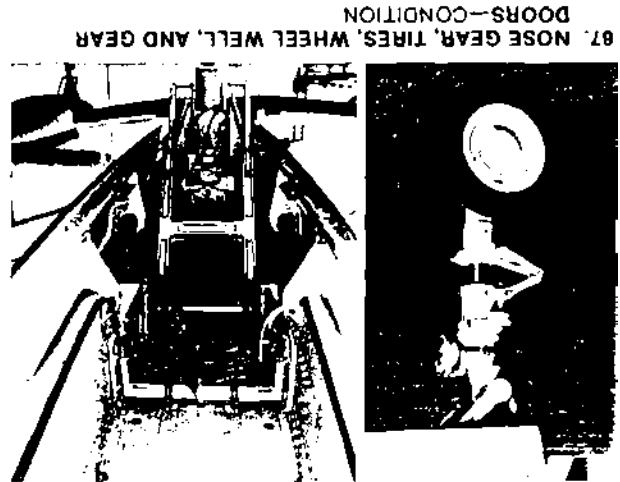
65. PITOT COVERS—REMOVE



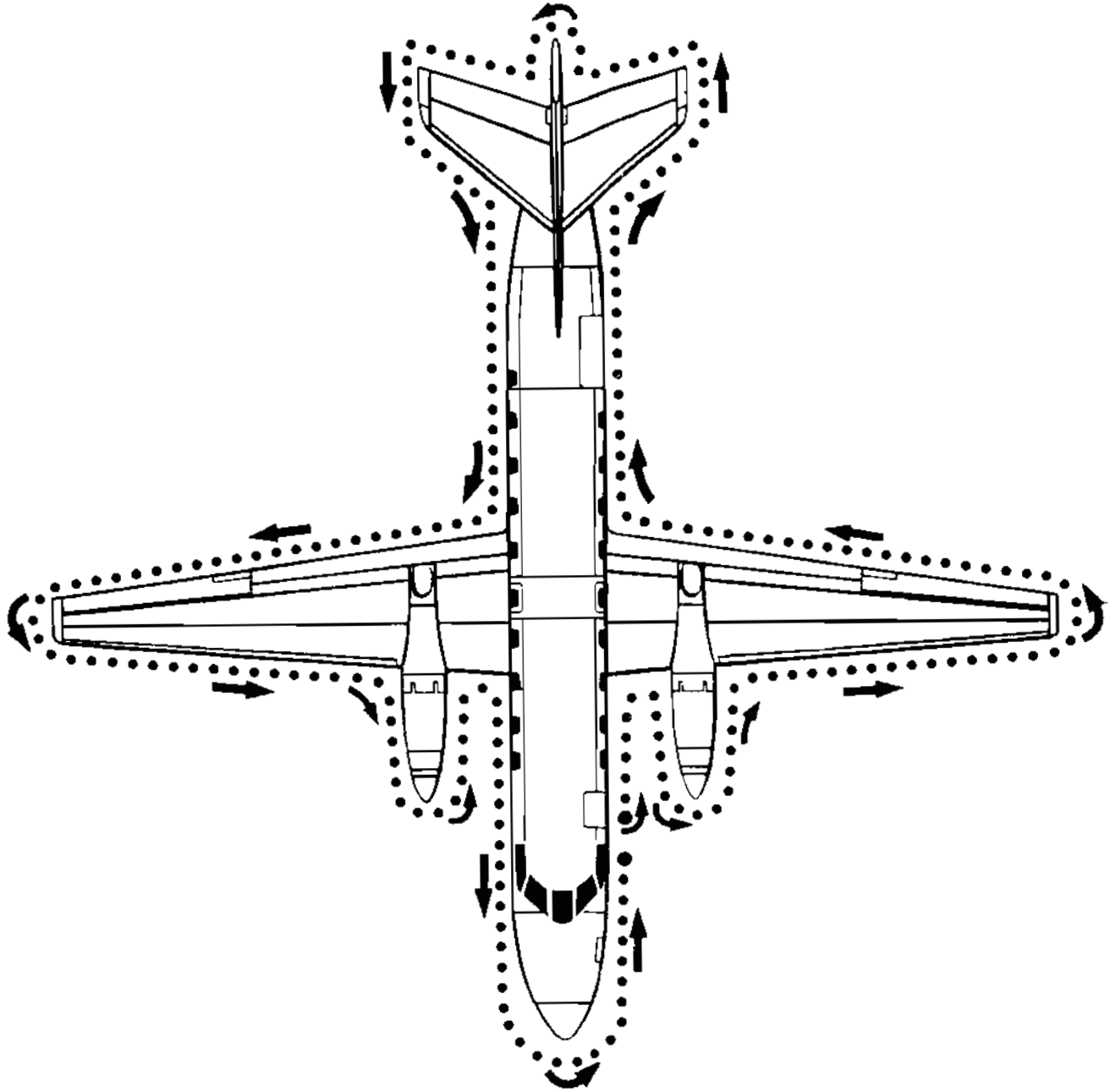
63. BAGGAGE DOORS—SECURE



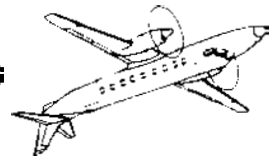
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CHAPTER 2

ELECTRICAL POWER SYSTEMS

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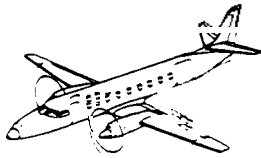


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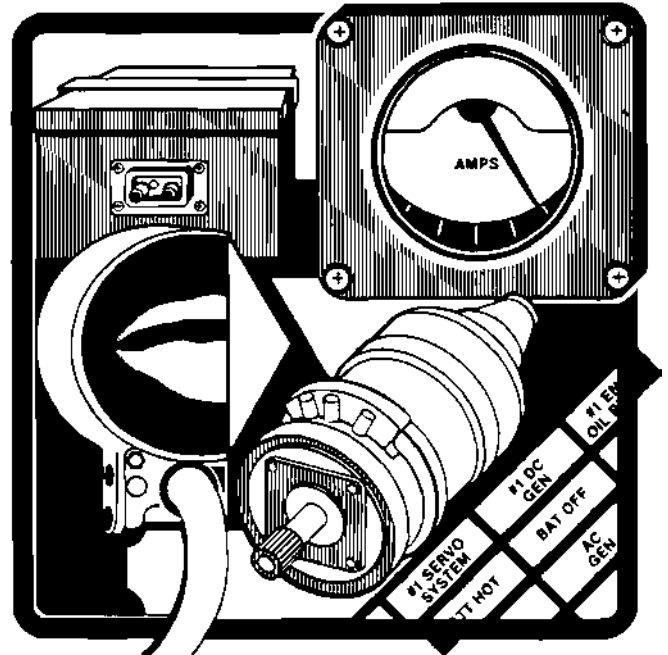
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CHAPTER 2

ELECTRICAL POWER SYSTEMS



INTRODUCTION

The electrical power system provides 28-volt DC, 115-volt AC, and 26-volt AC power for all airplane electrical requirements. An external power source, engine-driven starter-generators, and nickel-cadmium batteries supply the DC power for the majority of aircraft systems. AC power is provided by two static inverters which supply the avionics systems, and flight instruments. DC and AC power are distributed through two independent bus systems. Monitoring and warning devices are provided to inform the pilot of the systems' operating status.

GENERAL

Basic electrical power for the airplane is provided by the DC power system. This system consists of two starter-generators, two nickel-cadmium batteries, a DC ground power unit, and protective and indicating components. The

AC power system consists of two static inverters which provide 115- and 26-volt power to the airplane. Figure 2-1 shows the basic electrical system component locations, and Figure 2-2 shows the basic electrical system.



DC POWER

BATTERIES

Two nickel-cadmium (nicad) batteries each supply 24-volt DC electrical power for engine starts and standby power in the event of generator failure. The batteries are vented overboard to prevent fumes and liquids from accumulating within the airplane.

One battery is installed in each wing in a well located inboard of the nacelle and forward of the front spar. Each battery is accessed by removing a panel on the wing upper surface.

See Figure 2-3 for battery locations.

Battery switches, one for each battery, are located on the left switch panel (Figure 2-4). They are three-position switches labeled "BATTERY" (L or R), "OFF," and "RESET." L BAT DISC

and R BAT DISC warning lights (Appendix B) on the annunciator panel illuminate when the left or right battery relays are disconnected.

Battery voltage may be monitored by selecting the appropriate voltmeter switch position. The battery switch should be off when checking battery voltage. If the battery switch is on, the voltmeter displays the highest electrical source voltage connected to the system. The voltmeter and selector switch are installed on the pilot's side console (Figure 2-5).

The battery temperature indication system consists of an indicator (Figure 2-6), two switches on the instrument panel, and a temperature sensor for each battery. The indicator contains a temperature meter for each battery, an amber WARM light, which illuminates if either battery temperature exceeds 120° F, and a red HOT light which illuminates if either battery temperature exceeds 150° F. The temperature scales on the meters read from

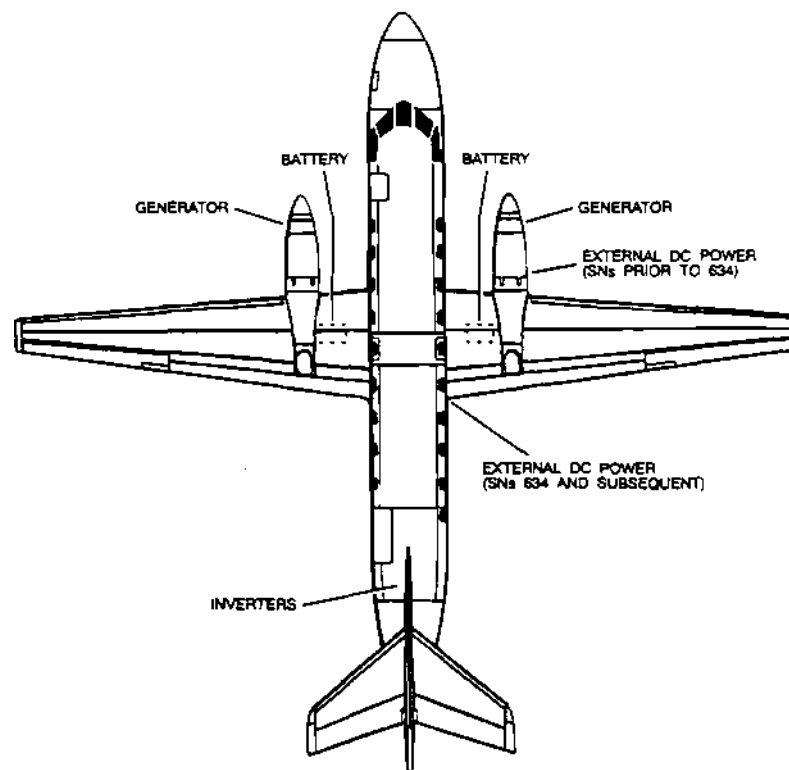


Figure 2-1. Component Locations



100 to 190° F. Below 120° F, the scales are marked in green. Between 120 and 150° F, they are marked in yellow, and above 150° F they are marked in red. Two switches adjacent to the indicator are labeled "BAT TEMP IND TEST" and "**RANGE EXTEND**."

Temperatures between 50 and 100° F can be read by pressing the **RANGE EXTEND** switch. This switch adds 50° F to the battery temperatures and displays the result. The actual temperature will be the scale reading minus 50° F.

The temperature indicator can be tested by pressing the BAT **TEMP** IND TEST switch. Both needles **should** travel from the bottom of the scale to the top smoothly and evenly in approximately five seconds. Both the WARM and HOT lights should illuminate as the needles travel up the scale.

Fault transformers on the battery feeder lines are used in conjunction with the battery fault detector to provide ground fault protection for the battery feeder lines.

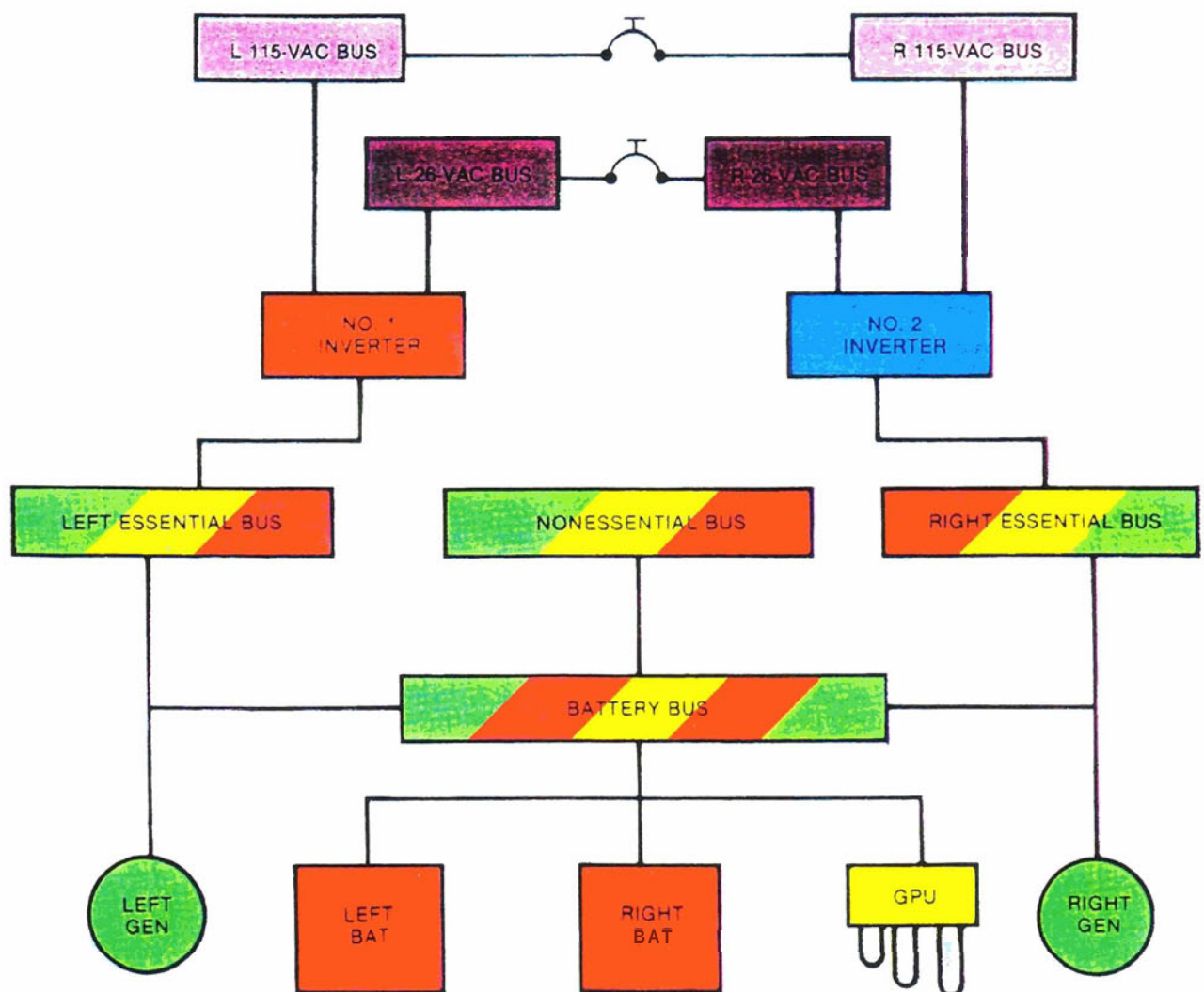


Figure 2-2. Basic Electrical System

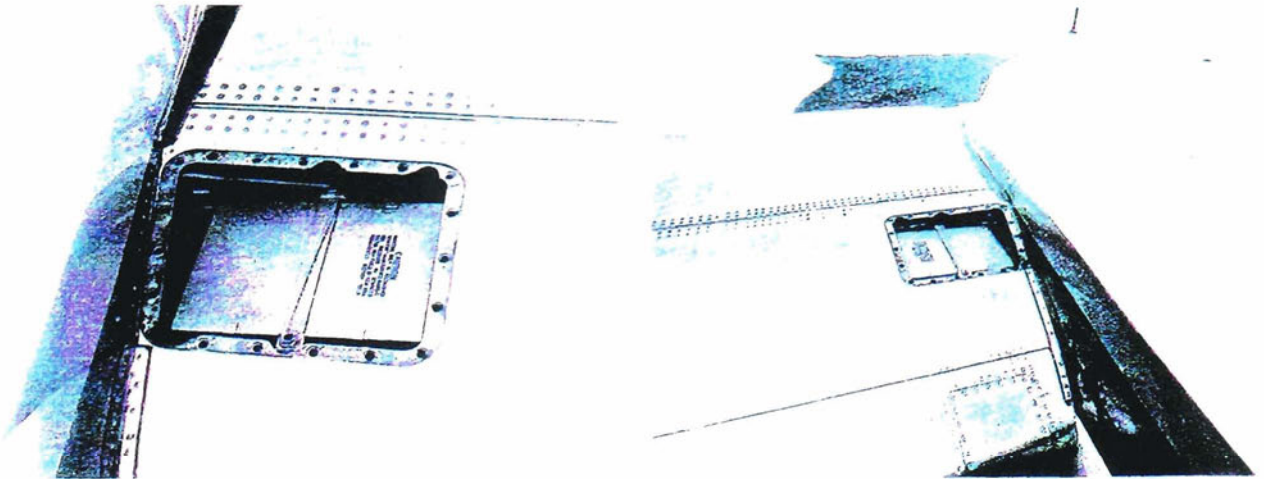


Figure 2-3. Battery Locations

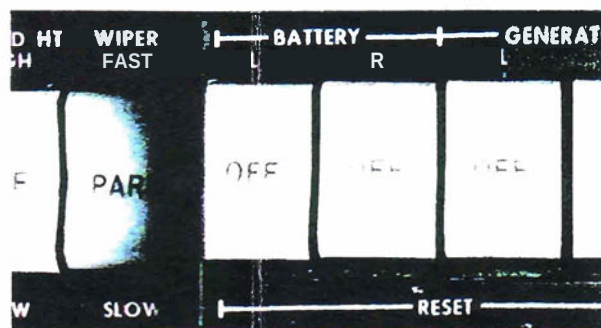


Figure 2-4. Battery Switches

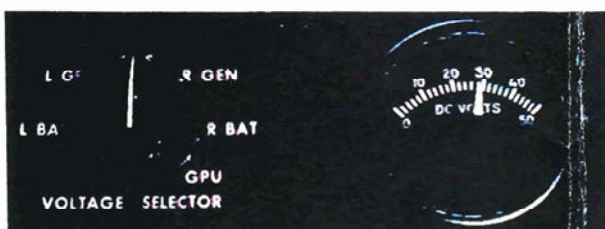


Figure 2-5. Voltage Selector and Meter



Figure 2-6. Battery Temperature Indicator



Wiring must be monitored from input end to output end. One of these ends will be called the source and one will be called the load. The purpose of the fault detection system is to verify that the current provided by the source is delivered through the wires to the load.

Beginning with a wire, as shown in Figure 2-7, current flows as the source supplies the load. The wire carrying the current passes through the center of a transformer installed at each end of the wire. The transformers are connected in opposition to each other, and their composite output is connected to a meter. As the current flow varies in the wire, small voltages are induced in each transformer. Since the transformers are connected in opposition, their output sums to zero, and the meter needle remains centered. It is important to note that the direction of current or magnitude of current flowing through the wire makes no difference; the transformer outputs oppose each other and sum to zero. For example, the batteries could be either a source or a load, depending on whether or not the generators are on the line.

If the wire shorts to ground, the source is still supplying a load, but it is not the load being monitored by the transformer system. One transformer now has no output and cannot cancel the output of the opposite transformer. A voltage exists in the system as indicated by the meter needle.

Using a meter, as in the example, would complicate cockpit proceedings and require continual monitoring by the crew. Instead of a meter, an automatic control circuit is used to continuously monitor the transformer outputs.

When a current imbalance in a battery feeder line is sensed, both batteries are disconnected by the battery fault detector. The BATTERY FAULT annunciator (Appendix B) and both BAT DISC lights illuminate to indicate a battery fault has been detected and that both batteries are disconnected. The pilot may attempt to get the batteries back on line by first pressing both battery switches to OFF, then by alternately pressing each battery switch to RESET, then to ON.

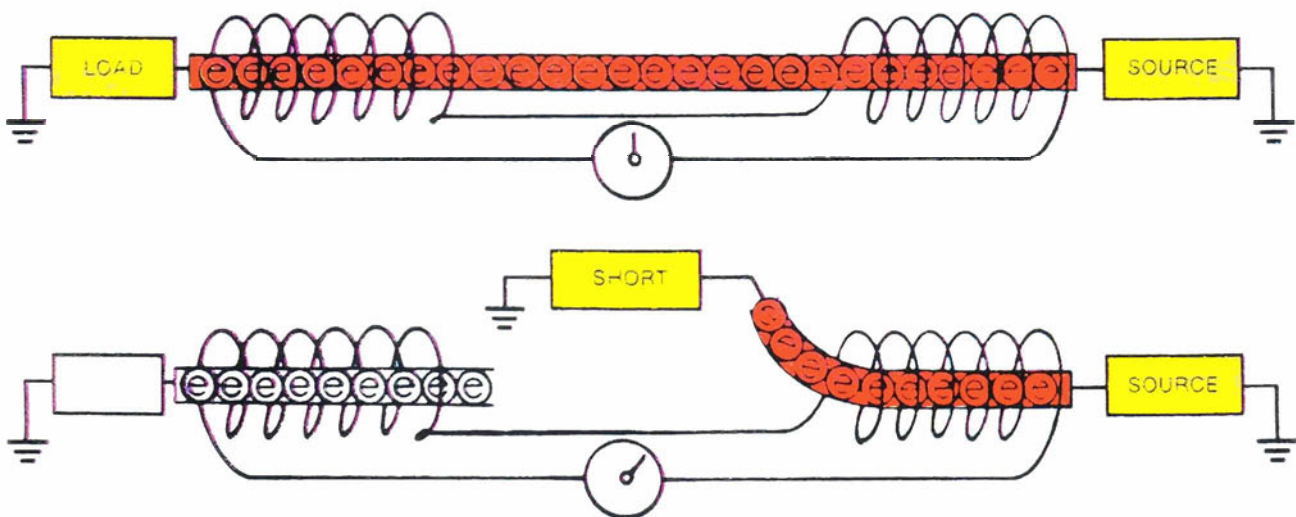
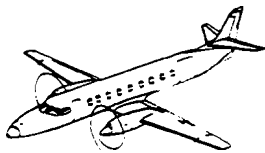


Figure 2-7. Source versus Load



Two conditions must be satisfied for the battery fault detection circuit to be operative. Either both generator switches must be on, or the left generator switch must be on and a GPU plug must be in the external power receptacle.

If the battery fault circuitry is active during an engine start, the fault detector will detect a fault and disconnect the batteries. To avoid a battery fault disconnect during engine start, the generator switches are turned OFF.

GENERATORS

Two engine-driven starter-generators, mounted on the lower right side of each engine, provide the airplane's primary source of DC power. (Figure 2-8). Each generator's output is regulated at 28.5 volts and, on some aircraft, electrically limited to 305 amperes at 71% engine rpm. Permissible generator continuous ground load is 200 amperes. More current is allowed during cross-generator starts and during battery recharging after engine start. Allowable generator load in flight is 300 amperes in Merlin IVCs and later Metro IIIs. Earlier Metro III's are limited to a 200-ampere generator load in flight.

The generator control switches are located on the left switch panel (Figure 2-9) and have



Figure 2-8. Generator Location

three positions labeled "L," "OFF," and "RESET" or "R," "OFF," and "RESET." DC power is routed through the generator relays and current limiters to the battery bus in the junction box behind the pilot's seat.

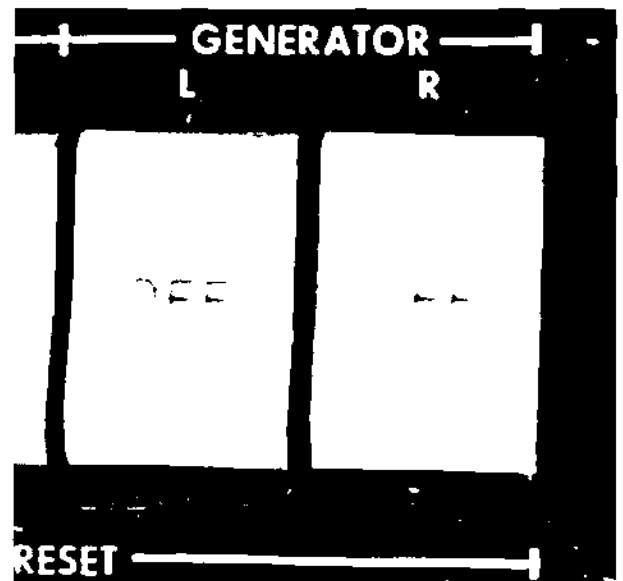
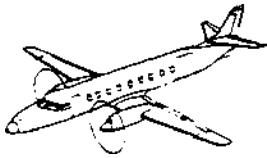


Figure 2-9. Generator Switches

Each generator is controlled by a generator control panel located on the left side of the respective wheel well. The panel controls voltage regulation, generator paralleling, relay control, reverse-current control, ground fault protection, and overvoltage protection. Paralleling and reverse-current control functions receive power from the GEN CONT circuit breakers on the respective essential bus circuit-breaker panels. On airplanes SNs 595 and subsequent, the generator control circuit breakers have been moved to the respective wheel wells. Electrical power for the generator control switch comes from the START CONT 2 circuit breaker on the respective essential bus. On aircraft SNs 734 and subsequent (or earlier aircraft modified by Service Bulletin 227-24-012), the generators are self-exciting, and the control panels are powered from the GEN CONT circuit breaker on the respective essential bus circuit-breaker panel.



Fault transformers on the generator feeder lines are used in conjunction with the generator control panel to provide ground fault protection. When a current imbalance in a generator feed line is sensed, the affected generator relay opens. Placing the generator switch to RESET should reset the protection circuit if the fault no longer exists. It is not unusual for a fault to be detected during an engine start. Normal operating technique is to use the RESET position after an engine start prior to turning on the generator.

L or R GENERATOR FAIL lights on the annunciator panel (Figure 2-10 and Appendix B) illuminate whenever the respective generator relay is disconnected (open).

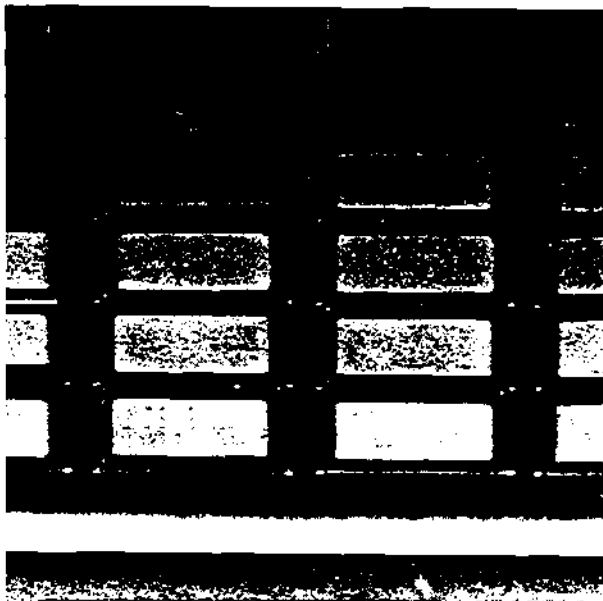


Figure 2-10. Generator Failure Lights

If a generator becomes disconnected during operation, the pilot should turn the generator off and select that generator with the voltmeter selector. If the generator voltage is not normal, the pilot should reset the generator. If the generator voltage appears normal, the

pilot may try to connect it. If the generator will not connect after being reset, no further action is possible.

A voltmeter and a selector switch located on the left side console allow reading of the voltage for each battery, generator, GPU, or battery bus (Figure 2-5). To read the output of the selected battery or generator, the corresponding battery or generator must be in the OFF position. If the selected battery or generator is connected to the DC bus, the voltmeter displays DC bus voltage. To avoid eventual drain of the batteries, the voltmeter selector switch should be left in the BUS position after engine shutdown. Circuit protection for the voltmeter circuit to each generator is provided by a circuit breaker located on the right side of the respective wheelwell. The voltmeter circuit to the battery bus and the GPU are protected by circuit breakers on the J-box behind the pilot's seat. Protection for the voltmeter circuit to each battery is provided by a circuit breaker located in each respective wing battery well.

Two DC ammeters, connected as loadmeters, are installed on the left side console (Figure 2-11) to indicate the respective generator's output.

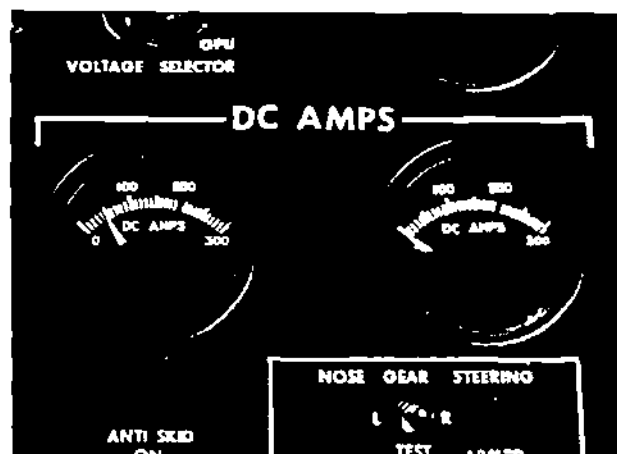


Figure 2-11. DC Ammeters



If an engine is shut down without turning its generator off, a reverse current sufficient to blow the respective 325-ampere current limiter might be generated. The blown current limiter would make it impossible to restart the engine. During cross-generator starts, a generator produces 305 amperes at 71% rpm and may produce sufficient current to blow the operating generator's 325-ampere current limiter if the rpm is greater than 71%.

Current limiter operation can be checked after a cross-generator start by determining that all buses can be powered from a single source. Normally, both batteries and one generator are turned off for the current limiter check. Conduct an annunciator panel test or the stabilizer trim check, both of which use electrical power from all three DC buses. If the test is normal, then all three DC buses are powered, and all the current limiters are good.

Current limiter integrity should be verified after a cross-generator start or before deliberate shutdown of an engine in the air if an airstart is planned.

GROUND POWER

The ground power circuit consists of a ground power unit (GPU) receptacle (Figure 2-12), a connection to the right battery feeder line, a GPU plug-in switch, and a GPU PLUG IN light (Appendix B) on the annunciator panel. Power supplied by the GPU goes to the battery bus relay and the battery bus (Figure 2-13). The GPU PLUG IN light illuminates when a GPU plug is inserted into the GPU receptacle.

CAUTION

Do not operate the airplane generators with a ground power unit connected to the bus. **Internal** damage to the airplane wiring system may result.

A battery switch must be on to allow GPU use. Do not operate avionics without the airplane batteries on.

Certain types of ground power units produce voltage variations or spikes in their output. These spikes do not harm the systems installed in the airplane, but cause solid-state avionics to behave erratically or, in some cases, do permanent damage to these avionics. The batteries serve as large capacitors and smooth out these voltage spikes.

EXTERNAL DC POWER

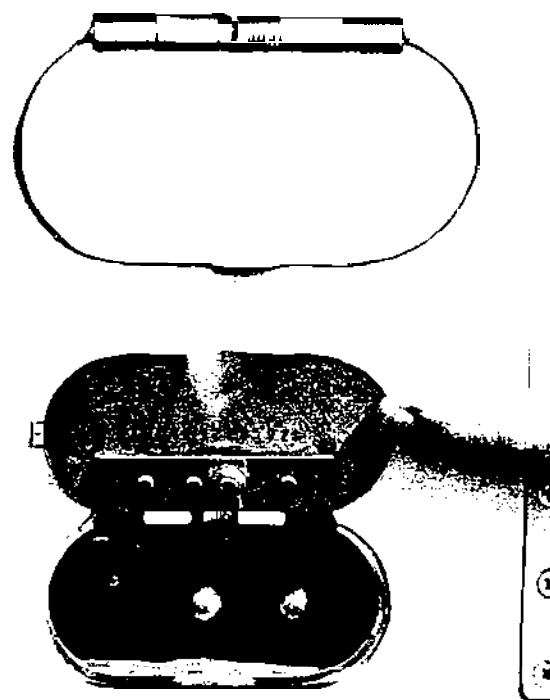


Figure 2-12. GPU Receptacle

The large pin on one end of the GPU plug must be negative. The large pin in the center of the GPU plug must be positive and must be jumpered to the small pin. If power is not supplied to the airplane sensing circuits through the small pin, the batteries may shift into series during engine start even though the GPU is supplying power; and it will not be possible to monitor GPU voltage.



The battery bus is located in the junction box (J-box), as shown in Figure 2-13, and forms the central distribution point for power. Each battery is connected through a battery relay to a battery bus relay, and then to the battery bus. Power is supplied to the nonessential bus through a 150-ampere circuit breaker and a bus tie switch. The left and right essential buses are powered through 225-ampere current limiters and bus tie switches. The generators supply power to their respective essential buses and to the battery bus.

Power supplied to each bus is further distributed to the various circuits by circuit breakers. When either battery, either generator, or a GPU is operating and the associated battery or generator switch is on, DC power is available to the battery bus. Power is distributed from the battery bus to all of the three DC distribution buses through the bus-tie switches. The left essential bus is located in the left console. The right essential and the nonessential buses are located in the right console. Each





bus is usually connected to the distribution system with the bus tie switch mounted on the respective console.

Ten essential items, normally powered by the left essential bus, can be powered by the right essential bus by actuating the BUS TRANSFER switches located on the aft end of the left circuit-breaker panel (Figure 2-14). These items are the following:

- Pilot's DC instruments (some airplanes)
- Fuel crossflow valve
- Pilot's turn and bank (except C-26)
- Surface deicer boots
- Landing gear control
- Landing gear position indicator
- Cabin pressure dump
- Left engine intake heat
- Right engine intake heat
- Left windshield heat

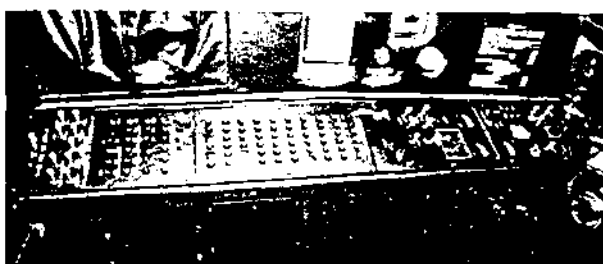


Figure 2-14. Left Circuit Breaker Panel and Bus Transfer Switches (Typical)

On the C-26 a transfer switch allows the standby attitude indicator to be powered by the left essential bus or the left battery.

Additionally, 26-VAC and 115-VAC power may be produced by selecting the No. 1 or No. 2 inverter powered by the left or right essential bus, respectively.

AC POWER

INVERTERS

AC electrical power is supplied by single-phase, solid-state inverters. Two inverters are installed, but only one is used at a time. The inverter selector switch position (Figure 2-15) determines which one is used. The inverters, located on the aft equipment rack, produce 115-volt and 26-volt AC power.

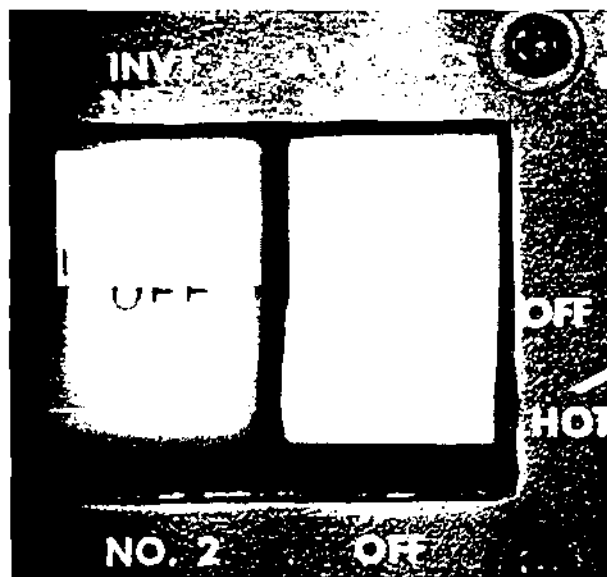


Figure 2-15. Inverter Selector Switch



CONTROL AND INDICATION

The AC warning and monitoring system includes a bus selectable voltmeter on the left console (Figure 2-16) and two bus failure warning lights on the annunciator panel (Appendix B).

The AC voltmeter can be selected to monitor either the left or the right 115-volt bus. If power to either 115-VAC bus is lost, the respective AC BUS warning light illuminates to alert the pilot. Illumination of only one AC BUS warning light is usually an indication that the 115-VAC bus-tie circuit breaker has opened. Illumination of both warning lights is usually an indication of an inverter failure, and the other inverter should be selected.

DISTRIBUTION

DC electrical power for control and operation of the No. 1 inverter is supplied from the left essential bus (Figure 2-17). The right essential bus supplies the No. 2 inverter. The No. 1 inverter supplies power to the left 115-VAC bus and the left 26-VAC bus. The No. 2 inverter supplies power to the right 115-VAC bus and the right 26-VAC bus. The 115-volt buses are tied through a circuit breaker as are the 26-volt buses. Consequently, when either inverter is operating, power is connected to all four AC buses.

AC circuit breakers are located on the left and right forward and aft console panels.

Figures 2-18 and 2-19 show the electrical system in detail.

LIMITATIONS

ENGINE STARTER DUTY CYCLES

The starter duty cycle limitations are located in Table 2-1.

MAXIMUM RECOMMENDED STARTING CURRENT

Due to the possibility of damage to airplane starter wiring during engine start, it is recommended that the maximum starting current from a ground power source be limited to 1,000 amperes.

Maximum continuous load for each generator is limited as follows:

Ground operations 200 amps

In flight..... 300 amps

Earlier Metro IIIs..... 200 amps

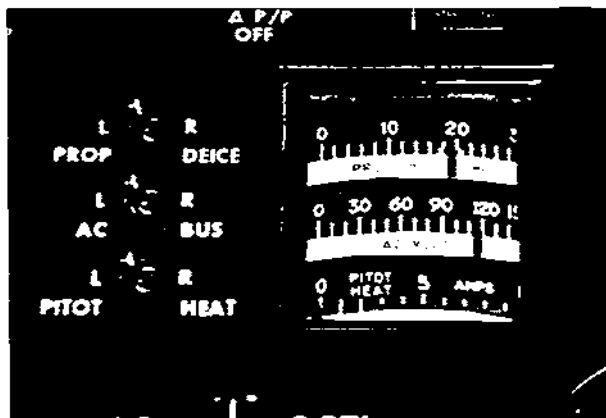


Figure 2-16. AC Voltmeter

TABLE 2-1. ENGINE STARTER DUTY CYCLES

Start Attempt	Starter ON Time	Starter OFF Time
1	30 seconds	60 seconds
2	30 seconds	60 seconds
3	30 seconds	15 minutes



NOTE

indicated load on the operating generator will exceed 300 amperes **during** cross-generator engine starts while **the starting engine is in** the 0 to 60% rpm, **starter-energized** range.

Following battery engine starts, indicated load on the operating generator will exceed 200 amperes

until the batteries have had time to recharge, **nor to exceed 2 minutes duration.**

BATTERY TEMPERATURE RED WARNING LIGHT ILLUMINATES

Takeoff is prohibited. The battery **must be removed and bench-checked** prior to further use.

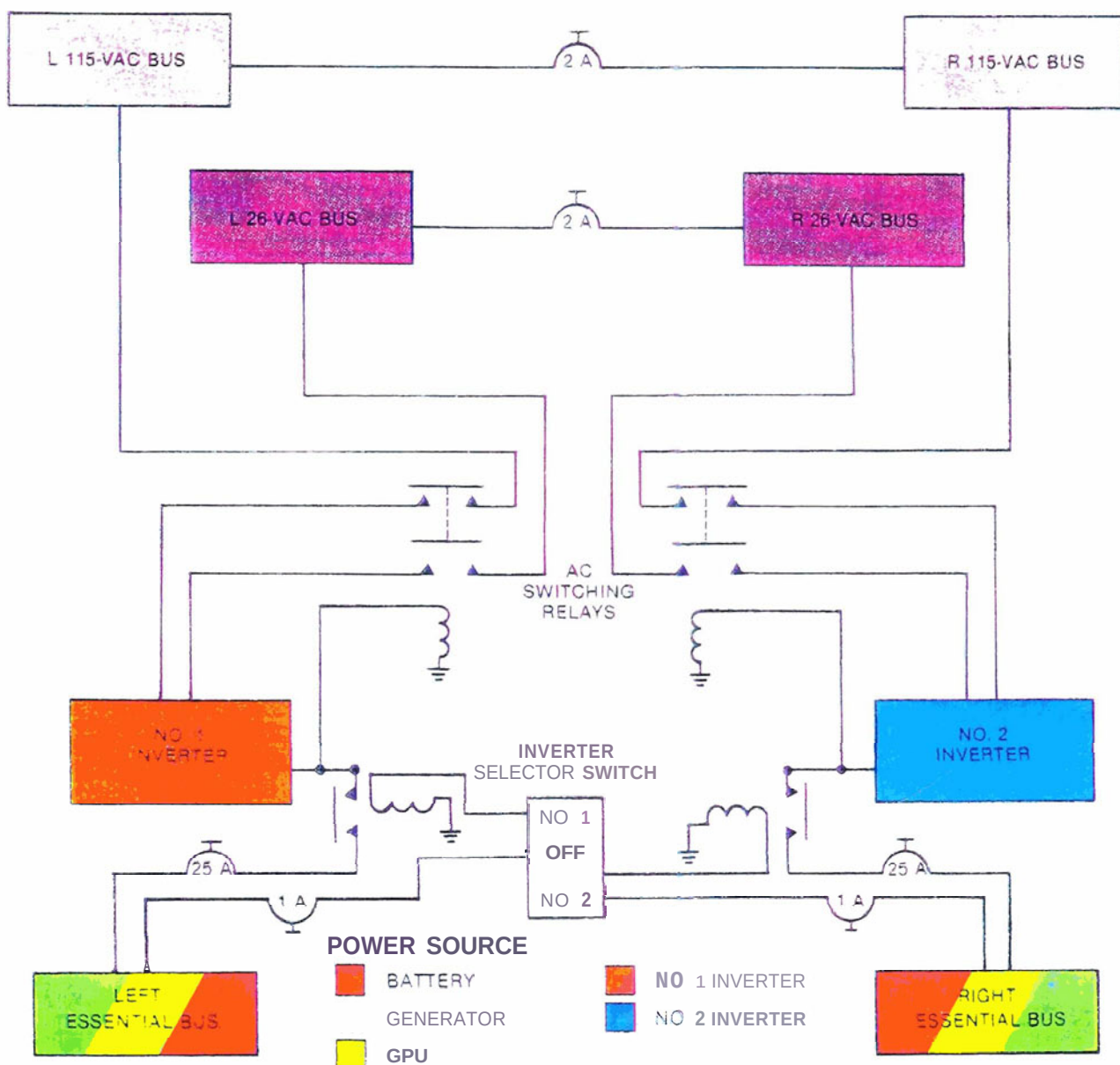


Figure 2-17. AC Distribution

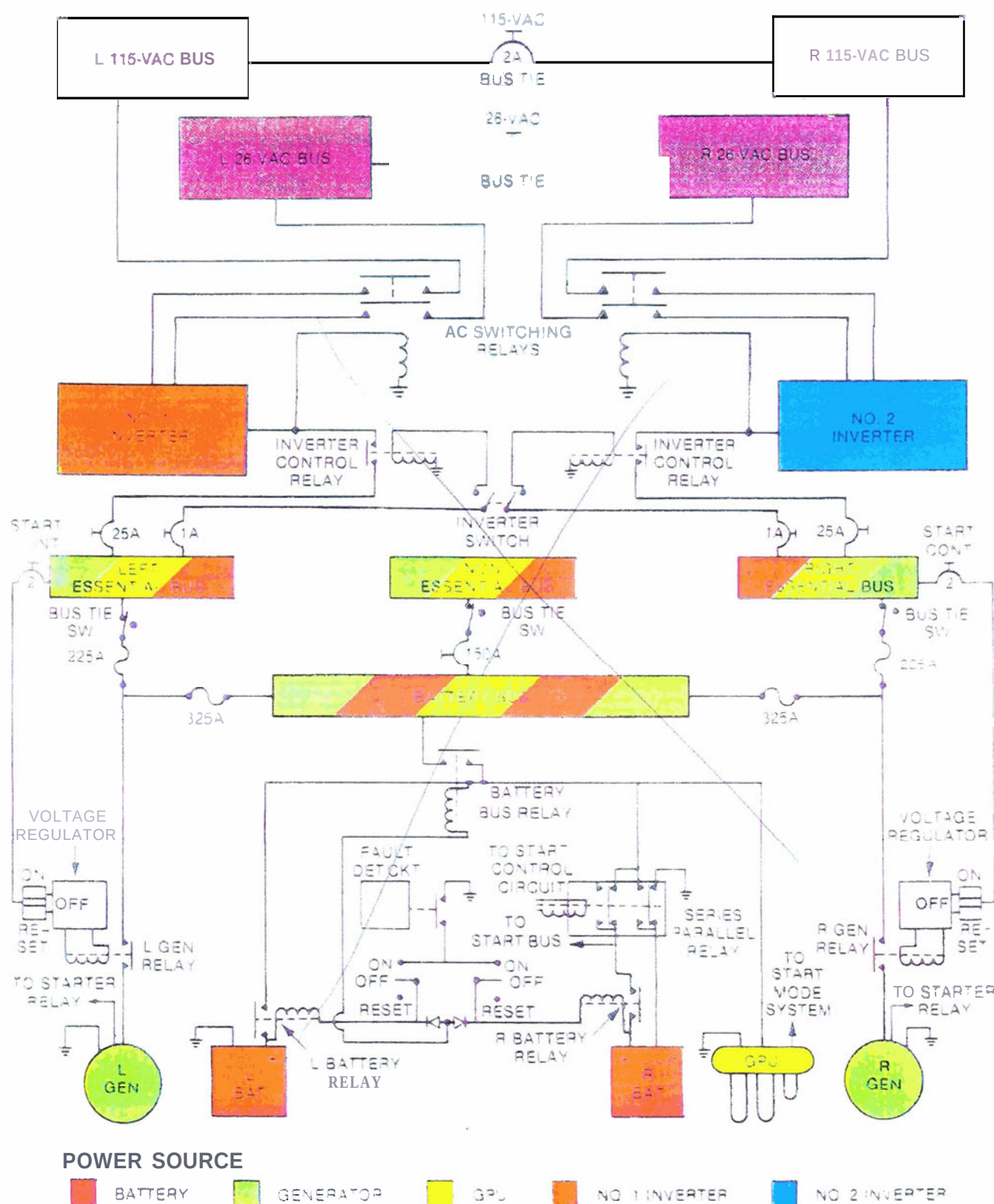


Figure 2-18. Detailed Electrical System (SNs Prior to 734)

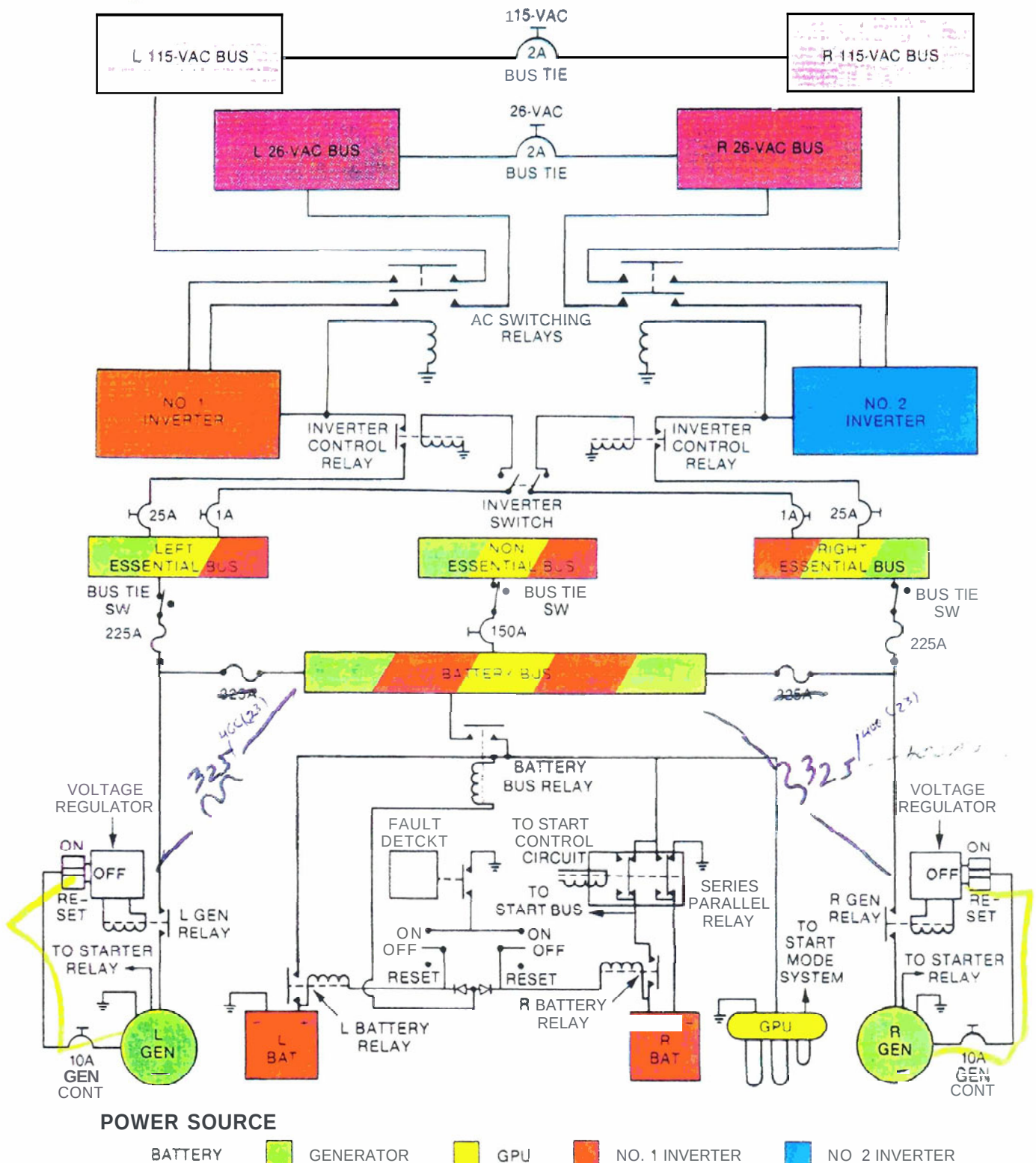


Figure 2-19. Detailed Electrical System (SNs 734 and Subsequent or Airplanes Modified by SB 227-24-012)



QUESTIONS

1. The electrical rating for the airplane batteries is:
A. 24 volts
☒ B. 28 volts
C. 30 volts
D. 34 volts
2. A temperature of 70° can be displayed on the battery temperature indicator by:
A. Pressing the annunciator TEST switch
B. Selecting BATT on the DC voltmeter
C. Pressing the BAT TEMP IND TEST switch
D. Pressing the RANGE EXTEND switch
3. The battery temperature indicator HOT light illuminates at:
A. 120° F
B. 130° F
C. 140° F
☒ D. 150° F
4. The in-flight electrical rating for each generator on the Merlin IVC and late Metro IIIs is:
A. 24 volts, 200 amperes
B. 28.5 volts, 300 amperes
C. 24 volts, 300 amperes
D. 28.5 volts, 200 amperes
5. The in-flight electrical rating for each generator on early Metro IIIs is:
A. 24 volts, 200 amperes
B. 28.5 volts, 300 amperes
C. 24 volts, 300 amperes
D. 28.5 volts, 200 amperes
6. If a generator fault is detected during a start, to turn the generator on:
A. Move the generator switch to OFF.
B. Turn the battery switch to OFF.
C. Disconnect the GPU.
☒ D. Move the generator switch to RESET.
7. To read just one battery's voltage on the DC voltmeter:
A. Position the voltmeter select switch to BUS.
B. Shut off all other power sources.
☒ C. Turn that battery switch off.
D. Pull the battery circuit breaker on the pilot's J-box.
8. During taxi, an engine flames out due to water ingestion. Prior to restart:
A. Turn off the batteries.
B. Turn off the associated generator switch.
C. A thermal overload check must be made.
D. A start circuit check must be made.
9. Electrical power from all three DC buses is used by the:
A. Battery temperature indicator test
B. Landing gear light test
C. Pressurization system test
☒ D. Annunciator panel test
10. For the battery fault detection circuit to be operative:
A. Have both generator switches on.
☒ B. Operate only one avionics system at a time.
C. Remove power from the nonessential bus.
D. Disconnect the left battery.
11. If the left essential bus fails, to restore power to essential system circuits:
A. Depress the bus-tie circuit breaker.
B. Reset the generators.
☒ C. Use the BUS TRANSFER switches.
D. Use emergency battery power.



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12. The voltage output of each inverter is:
- ☒ A. 115-volt AC and 26-volt AC
 - ☐ B. 115-volt AC and 350-volt AC
 - ☐ C. 125-volt AC and 30-volt AC
 - ☐ D. 28-volt AC and 120-volt AC
13. The number of inverters that can be operated at a time is:
- A. Four
 - B. Three
 - C. Two
 - D. One
14. If just the left 115-volt AC bus loses power, to restore it:
- A. Use the BUS TRANSFER switches.
 - ☐ B. Select the other inverter.
 - ☒ C. Attempt to reset the 115-volt AC bus-tie circuit breaker.
 - D. Recycle the bus-tie switch.
15. 115-volt AC bus voltage can be read:
- A. With a meter on the inverter
 - ☐ B. On the AC voltmeter
 - ☐ C. Only by maintenance personnel
 - D. With the DC voltmeter



CHAPTER 3 LIGHTING

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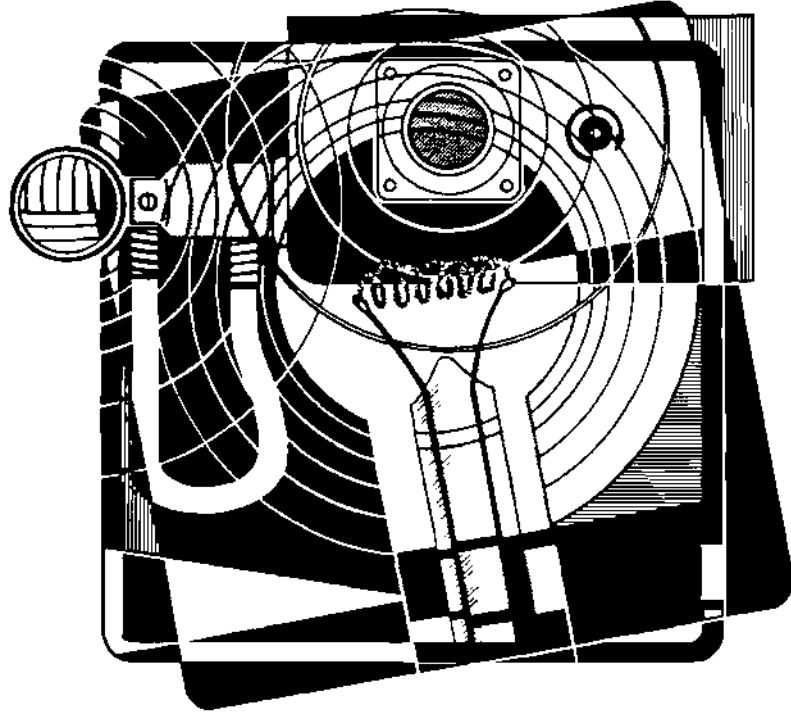
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CHAPTER 3 LIGHTING



INTRODUCTION

A standard lighting package is used on the Merlin/Metro series to illuminate the cockpit area, all flight instruments, the entrance, cabin area, nose compartment, and baggage areas. The majority of the instruments are internally lighted. For general illumination, either fluorescent or incandescent floodlights are used. Standard warning signs are provided for the cabin area. Exit signs are electroluminescent. Exterior lighting consists of navigation, rotating beacon, wing ice, landing, taxi, strobe, logo, and recognition lights.

GENERAL

Interior lighting consists of cockpit, cabin, and emergency lighting. Cockpit lighting is powered from all three DC buses. All cabin lighting is powered from the nonessential bus with the exception of the entrance, cargo, and

baggage compartment lights. The cargo and baggage compartment lights are powered from the right battery. The entrance door light is powered from the left battery.



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The exterior lighting system is equipped with:

- Five or six navigation lights (two red on the left **wingtip**, two green on the right **wingtip**, and one or two clear in the tail cone)
- One red rotating beacon mounted on top of the vertical stabilizer
- Two wing ice lights, one in the outboard side of each engine nacelle
- Two landing lights located under a cover on the leading edge of each wing
- One taxi light located on the nose landing gear
- Three strobe lights, one on each **wingtip** and one on the tail
- Four optional logo lights, one on the top and one on the bottom of each horizontal stabilizer
- Two recognition lights located under a cover on the leading edge of each wing

All exterior lighting is powered from the nonessential bus.

Lighting controls (Figure 3-1) are on the lower switch panels, the left and right forward consoles, the bottom of the pedestal, and on the light itself or in proximity to it. Emergency lights above each exit are self-energized.

INTERIOR LIGHTING

COCKPIT LIGHTING

Lighting for the cockpit area consists of general illumination of the instrument panel from the underside of the glareshield, overhead floodlights, pilot and copilot instrument lights, engine and auxiliary instrument lights, map lights, and console and pedestal lights.

Pilot and copilot flight instrument lights are controlled by individual dimmers on the left and right forward consoles. The control on the left forward console is labeled "PILOT FLT INSTR" and controls light intensity for the

pilot's basic flight instruments. The pilot's lights are powered from the left essential bus through the PLT INSTR LIGHTS circuit breaker. The control on the right forward console is labeled "COPILOT FLT INSTR" and controls light intensity for the copilot's basic flight instruments. The copilot's lights are powered from the right essential bus through the C/PLT INSTR LIGHTS circuit breaker.

Engine and auxiliary instrument lights are controlled by the GENL INSTR dimmer on the bottom of the pedestal. All internal instrument lighting except pilot and copilot flight instruments is controlled with this dimmer. Power for the lights comes from the right essential bus through the GENL INSTR LIGHTS circuit breaker.

Pilot and copilot map lights are controlled by a switch on each light. The lights are powered from the nonessential bus through the COCKPIT MISC LTS circuit breaker.

Glareshield lights are controlled by the GLARESHIELD dimmer on the bottom of the pedestal. The lights are powered from the nonessential bus through the COCKPIT MISC LTS circuit breaker.

Overhead floodlights are controlled by the LH and RH OVHD light controls. The lights are powered from the nonessential bus through the COCKPIT MISC LTS circuit breaker. The left overhead floodlight is turned on when the cabin entry light switch is on.

Console, lower switch panels, and pedestal lights are controlled by the CSL PNLS dimmer on the bottom of the pedestal. Power for the lights is routed from the left essential bus through the CSL LIGHTS circuit breaker.

CABIN LIGHTING

Lighting for the cabin consists of reading, aisle, galley, buffet, lavatory, window, entrance, warning, nose, and baggage compartment lights. Controls are located on the right lower switch panel in the cockpit and on or



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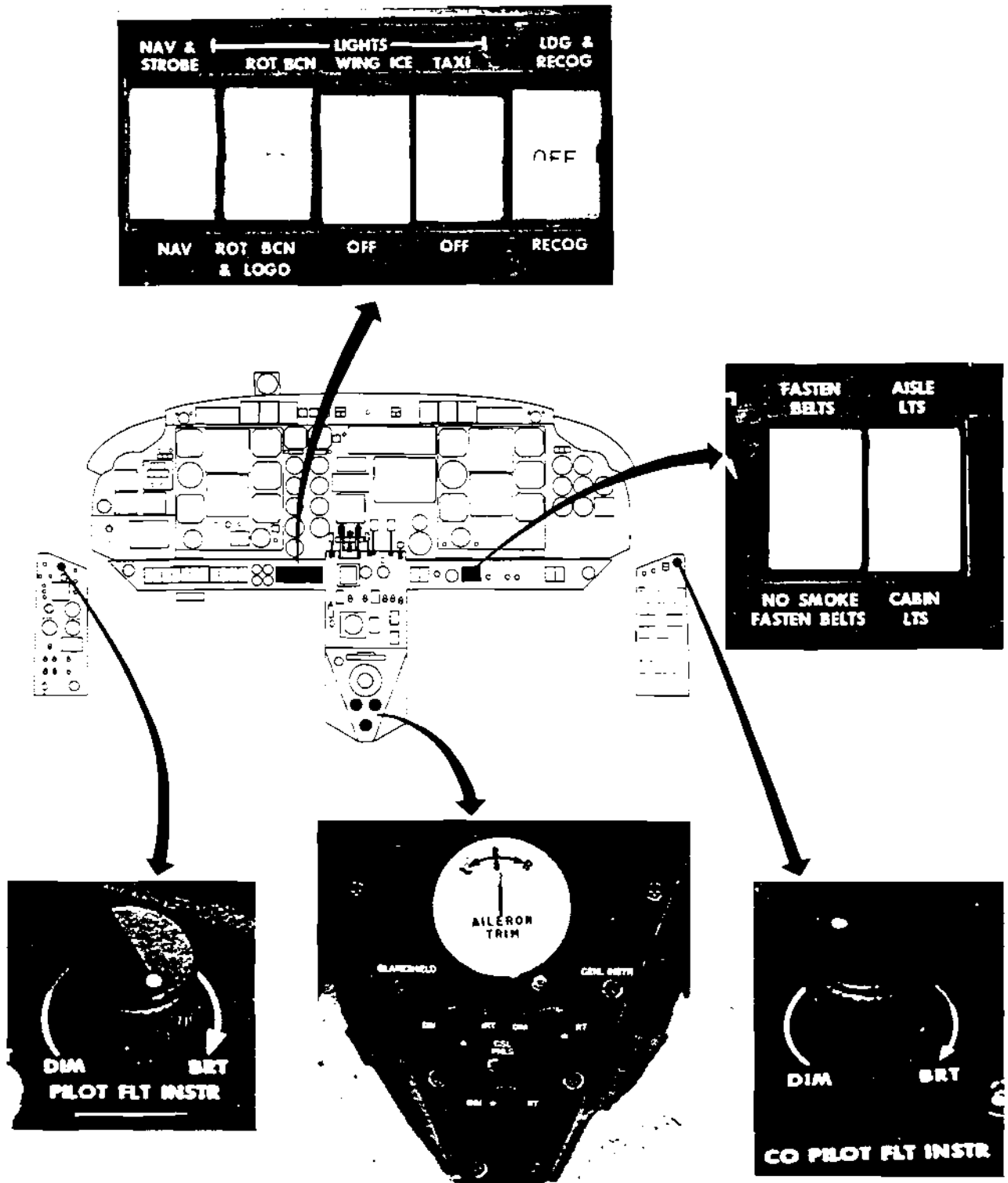


Figure 3-1. Lighting Controls



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near the lights. All circuit breakers are on the nonessential bus unless otherwise noted.

Reading lights are controlled by individual switches next to each light. On airplanes with an aisle and cabin lights switch, power is routed through the switch—before going to the lights.

Aisle lights are controlled by a single switch, or by two switches, depending on the airplane configuration. On airplanes with a single switch, it is labeled either "AISLE LTS" or "AISLE LTS—CABIN LTS." On airplanes with two switches, the labeling is "FWD AISLE LTS" and "AFT AISLE LTS."

Galley, buffet, and lavatory lights on the Merlin IVC are controlled by switches installed during the airplane interior completion, so the labeling varies between airplanes.

The window lights on the Merlin IVC are fluorescent and are controlled by WINDOW LTS dimming switches (Figure 3-2). Depending on the installation, there are one or two switches.

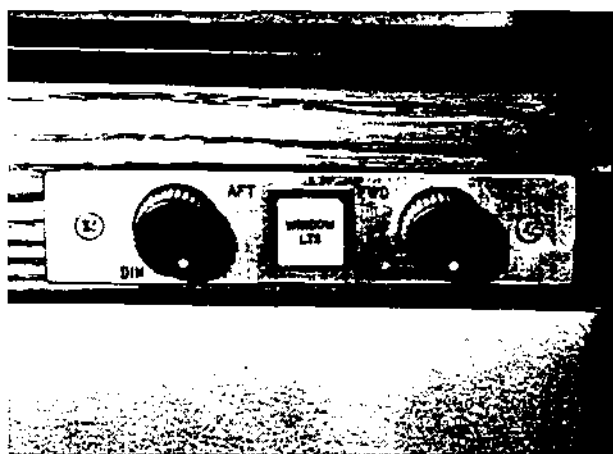


Figure 3-2. Window Light Controls

On airplanes with one switch, it is located on the cabinet. If two switches are provided, one is on the cabinet and the other on the bar.

Entrance lights are controlled by the ENTRANCE LGT switch (Figure 3-3). Power from the hot side of the battery relay is routed through an ENT LIGHTS circuit breaker located near the battery.



Figure 3-3. Entrance Light Switch

The seat-belt and no-smoking sign is controlled with a switch on the right lower switch panel. In the FASTEN BELTS position, just the fasten-belts portion of the sign is illuminated. In the NO SMOKE FASTEN BELTS position, both portions of the sign are illuminated. If the airplane has more than one sign, they are all controlled by the same switch.

Compartment lights in the nose and rear cargo areas are controlled with a switch in the respective compartment. Power for the lights is routed from the hot side of the right battery relay through a 5-minute timer. The timer is reset whenever the LDG GR CONT circuit breaker on the right essential bus is powered. It may also be reset by pressing a reset button inside the left nose baggage door.



EMERGENCY LIGHTING

Self-contained emergency exit lights are installed (Figure 3-4). The emergency exit labeling and placards are constructed of an electroluminescent material.

EXTERIOR LIGHTING

Exterior lighting consists of navigation and strobe, rotating beacon and logo, wing ice, taxi, landing, and recognition lights (Figure 3-5). All circuit breakers for exterior lights are located on the nonessential bus.



Figure 3-4. Emergency Exit Placard

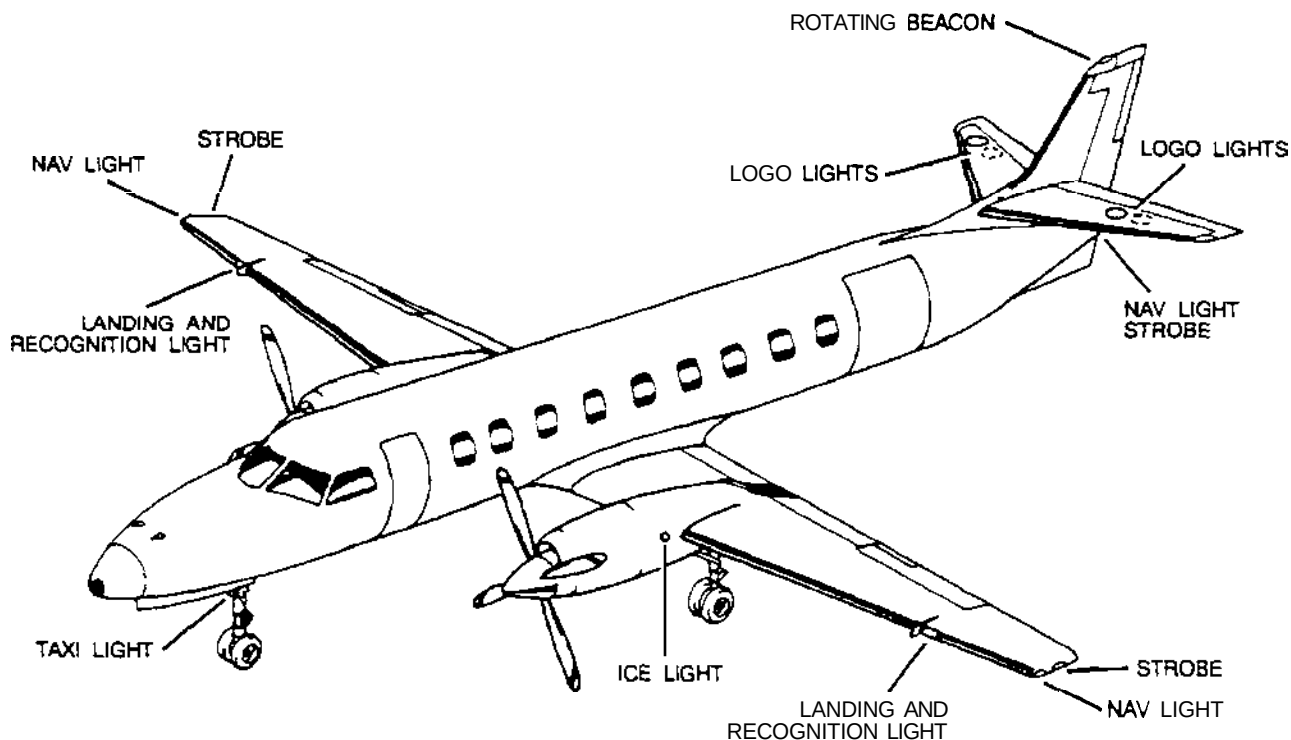


Figure 3-5. Exterior Lights



NAVIGATION AND STROBE LIGHTS

The navigation and strobe lights (Figure 3-6) are controlled with a single switch on the pilot's switch panel. It has two powered positions: NAV & STROBE and NAV. In the NAV position, the two right wingtip lights (green), the two left wingtip lights (red), and the tail-cone light (white) are illuminated. When the switch is in NAV & STROBE, all navigation lights, a strobe on each wingtip, and a strobe on the tail are illuminated; additionally, all the green lights on the annunciator panel are dimmed whenever the navigation lights are on. Power for the strobe lights is

boosted and cycled by a power supply for each light.

NOTE

Neither anticollision nor strobe lights should be used when flying through clouds or overcast; the flash effect reflected from water particles in the atmosphere, particularly at night, could produce vertigo (loss of orientation). Also, as a consideration to other pilots, the strobe lights should be left off during taxi near other occupied airplanes.



Figure 3-6. Navigation and Strobe Lights



ROTATING BEACON AND LOGO LIGHTS

The rotating beacon is controlled by a switch labeled "ROT BCN—ROT BCN & LOGO." In the ROT BCN position, the lightweight, oscillating anticollision light mounted on top of the vertical stabilizer is illuminated (Figure 3-7). It has an aerodynamic shape to reduce drag and is shielded to prevent radio interference. The light is a dual-lamp unit with the lamps oscillating 180° out of phase.

In the ROT BCN & LOGO position, a light on the top and a light on the bottom of each horizontal stabilizer is illuminated in addition to the beacon (Figure 3-7). These lights are designed to illuminate the vertical stabilizer logo area. Logo lights are optional.

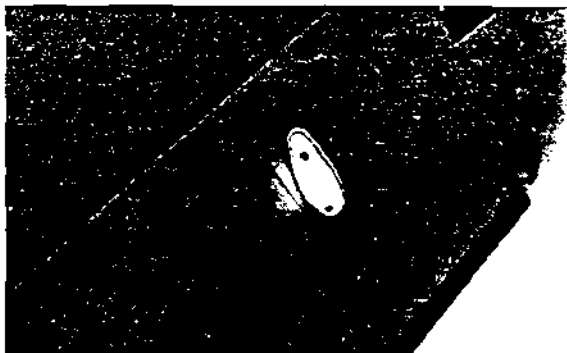


Figure 3-7. Rotating Beacon and Logo Lights

WING ICE LIGHTS

The wing ice lights are controlled with the WING ICE switch on the pilot's switch panel. A sealed-beam light is located in the outboard side of each engine nacelle to illuminate the wing leading edge (Figure 3-8).

TAXI LIGHT

The taxi light is controlled with the TAXI switch on the pilot's switch panel. The taxi light is mounted on the nose landing gear (Figure 3-9). A microswitch in the nose wheel

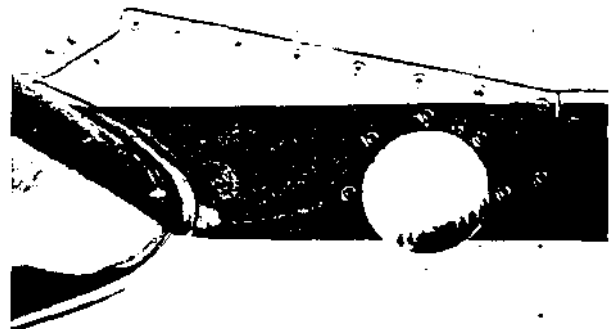


Figure 3-8. Wing Ice Light

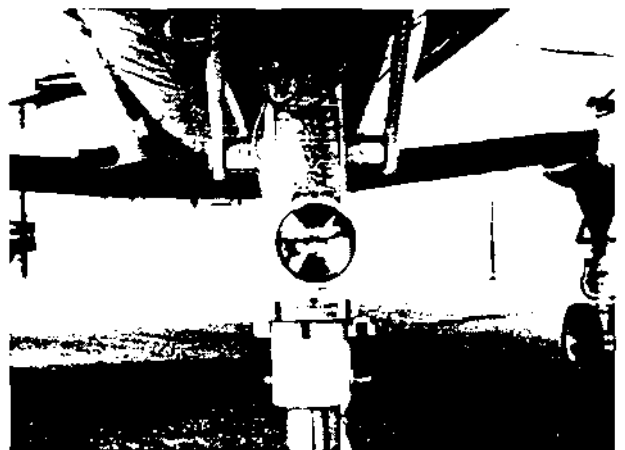
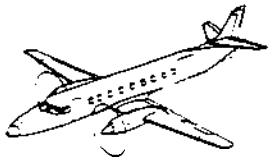


Figure 3-9. Taxi Light



well turns off the light when the nose landing gear is retracted.

NOTE

Ensure that the TAXI switch is off except when in actual use. Should a malfunction allow the light to remain on with the nose gear retracted, heat damage within the wheel well could occur.

LANDING LIGHTS

The landing lights and recognition lights are controlled with a single switch labeled "RECOG-LDG & RECOG." In the **RECOG** position, the recognition lights are illuminated, and in **LDG & RECOG**, both landing and recognition lights are illuminated. The landing and recognition lights are housed under a common cover in the leading edge of each wing (Figure 3-10). A glareshield is installed inboard of the lights to prevent the flight crew from being distracted by glare from the lights.

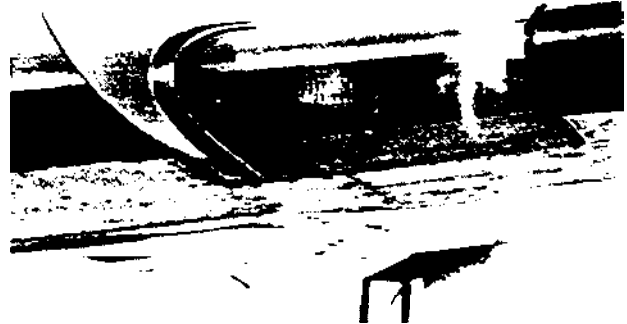
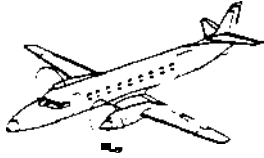


Figure 3-10. Landing Lights and Recognition Lights



QUESTIONS

1. The controls for the pilot's and copilot's flight instrument lights are located on the:
 - A. Lower switch panel
 - ☒ B. Forward side consoles
 - C. Bottom of the pedestal
 - D. Aft side consoles
2. The lighting intensity for the copilot's airspeed, altimeter, vertical speed, and radio heading instruments is varied by the use of the:
 - ☒ A. COPILOT FLT INSTR control
 - B. CSL PNLS control
 - C. GENL INSTR control
 - D. GENL FLT INSTR control
3. The following lights can be turned on with the battery switches off:
 - A. Passenger warning lights
 - B. Navigation lights
 - C. Overhead floodlights
 - D. Entrance lights
4. The nose baggage and cargo compartment timer can be reset:
 - A. Only by maintenance
 - B. By cycling the entrance door light switch
 - ☒ C. By turning on the cargo door light
 - D. By pressing the RESET switch inside the left nose baggage door
5. All exterior lights are powered from the:
 - A. Nonessential Bus
 - B. Left Essential Bus
 - C. Right Essential Bus
 - D. All of the Above

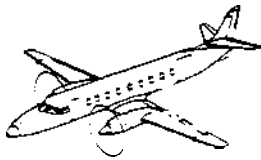


CHAPTER 4

MASTER WARNING SYSTEM

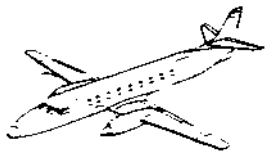
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CHAPTER 4

MASTER WARNING SYSTEMS



INTRODUCTION

The master warning system consists of an annunciator panel, a valve position annunciator panel, and various other lights. Most lights are located on the annunciator panel. An illuminated light alerts the pilot to a system malfunction (red), a system operating parameter (amber), or a system normal operating condition (green). When a light illuminates, the pilot should follow the approved checklist procedure.

GENERAL

System annunciators are grouped as follows: the annunciator panel, the valve position annunciator panel, fire extinguisher annunciators, oil cooler inlet duct heat cycle lights, fuel crossflow switchlight, fuel bypass lights, cargo door warning and test system, the nose steer fail light, and the fuel filter bypass lights. Provisions are made to test the warning and indication lights or circuits.

The annunciator panel, located in the upper center section of the instrument panel, is interconnected to numerous airplane systems and utilized to monitor system operation. The validity of several system warning lights may be checked by referring to the associated system's gage. Appendix B shows all lights illuminated.



ANNUNCIATOR PANEL

The annunciator panel contains red warning lights to advise the pilot of serious system conditions, amber caution lights to indicate system conditions of a less serious nature, and green lights to indicate other specific system conditions. The panel is powered by both the left and right essential buses.

TEST

A **PRESS TO TEST** switch is located on the left side of the panel. Pressing the switch checks the continuity of the annunciator panel lights, the valve position lights, and complete continuity of the fire warning circuit. The **BYPASS**

OPEN lights and **DUCT HEAT CYCLE** lights are also tested with the same switch

DIMMING

The green system annunciators are automatically dimmed when the navigation lights are turned on. The red warning and amber caution annunciators cannot be dimmed. The fuel bypass lights can be dimmed by a **BRT-DIM** switch located between the lights.

ILLUMINATION CAUSES

Table 4-1 gives each annunciator's legend, color, and reason for illumination. The annunciators are listed by color, starting at the top left and reading to the right within that colored section.

Table 4-1. ANNUNCIATOR PANEL

ANNUNCIATOR	REASON FOR ILLUMINATION	ANNUNCIATOR	REASON FOR ILLUMINATION
L ENG FIRE	Excessive temperature is detected in associated engine nacelle	SAS FAULT	Steady SAS computer power has failed or that power has failed in combination with servo failure. On the ground, the SAS vane has deflected full up. Check SAS indicator needle. Flashing SAS servo or servo clutch has failed.
R ENG FIRE			
TEST	Illuminates any during test	HYDR PRESS	Pump output pressure is low.
DOOR	Open switch in cabin door warning system—cabin door not properly closed.	HYDR PRESS	
WING OVHT	Steady Temperature exceeding 350°F in the wheel well or 450°F in the air-conditioning duct	CARGO DOOR	Open switch in the cargo door click-clack warning system
R WING OVHT	Flashing Temperature exceeding 250°F in the wing leading edge		Illuminates only during test.
BATTERY FAULT	A battery ground fault has been detected.	GEAR DOOR POSITION	One of the main gear doors is not latched closed (illuminates on ground only)
L OIL PRESSURE	Oil pressure is below 40 psi		
R OIL PRESSURE			



Table 4-1. ANNUNCIATOR PANEL (Cont.)

ANNUNCIATOR	REASON FOR ILLUMINATION	ANNUNCIATOR	REASON FOR ILLUMINATION
L BETA	Prop pitch control oil pressure is sufficient to command reverse operation.	GENERATOR FAIL	The generator relay is open.
R BETA		R GENERATOR FAIL	
LOW SUCTION	Insufficient suction.	ANTI-SKID	The antiskid switch is off or there is a system fault (NOSE STEER FAIL if no antiskid installed).
L CHIP DET	Metal particles detected in the engine oil.	INTAKE HEAT	If the intake heat switch is on, the engine anti-ice valve is open. If the closed switch is pressed, the valve is closed.
R CHIP DET		R INTAKE HEAT	
CABIN ALTITUDE	Cabin altitude is above 10,000 feet.	L W S HT	Thermostat is calling for respective windshield heat.
FUEL LOW	Fuel level in hopper tank is low.	R W S HT	
FUEL PUMP		SAS ARM	The SAS airspeed switch has armed the SAS.
GPU PLUG IN	A GPU plug is plugged into the external power receptacle.	SAS DEICE	The SAS heat relay has operated.
L BAT DISC	The Indicated battery relay is disconnected.	AWI NO 1 PUMP ON	The applicable AWI pump is operating.
R BAT DISC		AWI NO 2 PUMP ON	
L SRL OFF	The SRL computer is not operating. Normal with less than 80% rpm.	NOSE STEERING	Steady: Power is available to nose steering relay. Flashing: A nose steering fault has been detected, and the nose steering is disconnected.
R SRL OFF			
L AC BUS	115-VAC bus is deenergized.		Illuminates only during test.
R AC BUS			

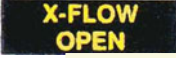


MISCELLANEOUS LIGHTS

VALVE POSITION ANNUNCIATOR PANEL

The valve position annunciator panel is located on the instrument panel (Table 4-2). These lights are tested with the main annunciator panel PRESS TO TEST switch.

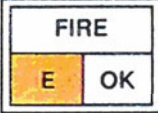
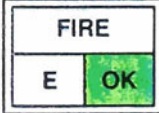
Table 4-2. VALVE POSITION ANNUNCIATOR PANEL

ANNUNCIATOR	REASON FOR ILLUMINATION
	The engine fuel shutoff valve is not in the position selected by the switch or is in transit.
	The engine hydraulic shutoff valve is not in the position selected by the switch or is in transit.
	The fuel crossflow valve is not closed; it is in transit or open.

FIRE EXTINGUISHER ANNUNCIATORS

The annunciators for the fire system are located on the instrument panel. Each is part of a three-lens control switchlight (Table 4-3). The FIRE portion of the annunciator may be tested with either the main annunciator panel test switch or with the FIRE EXT TEST switch located between the lights. The main annunciator panel test switch verifies the engine fire light and fire detector circuit integrity, while the FIRE EXT TEST switch verifies the lights and fire extinguisher circuit integrity.

Table 4-3. FIRE EXTINGUISHER ANNUNCIATORS

ANNUNCIATOR	REASON FOR ILLUMINATION
	1. Excessive temperature in the associated engine nacelle 2. The annunciator panel PRESS TO TEST is actuated. 3. The FIRE EXT TEST is actuated.
	The associated fire extinguisher bottle has been discharged and is empty or the FIRE EXT TEST is actuated.
	The FIRE EXT TEST is good.

OIL COOLER INLET DUCT HEAT CYCLE LIGHTS

The heat cycle lights for oil cooler inlet ducts are located on the left forward side console (Table 4-4).

Table 4-4. DUCT HEAT CYCLE ANNUNCIATORS

ANNUNCIATOR	REASON FOR ILLUMINATION
	Indicated oil cooler inlet duct anti-ice thermostat is operating to heat the oil cooler inlet duct.

FUEL CROSSFLOW SWITCHLIGHT

The crossflow switchlight is located on the pilot's lower instrument panel next to the fuel quantity indicator (Table 4-5).

Table 4-5. CROSSFLOW SWITCHLIGHT

ANNUNCIATOR	REASON FOR ILLUMINATION
	The switch is selected to open the crossflow valve.



FUEL BYPASS LIGHTS

The fuel bypass lights are located on the pilot's instrument panel just below the EGT indicators (Table 4-6). Fuel bypass light intensity is controlled with a BRT and DIM switch located between the lights.

Table 4-6. FUEL BYPASS LIGHTS

ANNUNCIATOR	REASON FOR ILLUMINATION
BYPASS OPEN	The associated engine fuel bypass valve is open.

CARGO DOOR WARNING AND TEST SYSTEM

The DOOR UNSAFE and SWITCHES NORMAL lights are located on the right forward console. The system indicates door latch positions (Table 4-7).

Table 4-7. CARGO DOOR WARNING AND TEST SYSTEM

ANNUNCIATOR	REASON FOR ILLUMINATION
DOOR UNSAFE	The cargo door is not closed or safe.
	The door handle is in the open position and the click-clack warning switches are operating correctly.
NORMAL	

NOSE STEER FAIL LIGHT

The NOSE STEER FAIL light is located on the pilot's instrument panel, between the EGT indicators and the annunciator panel (Table 4-8). If an antiskid system is not installed, it is located on the annunciator panel.

Table 4-8. NOSE STEER FAIL LIGHT

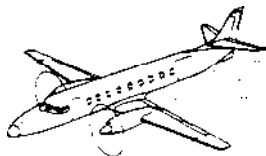
ANNUNCIATOR	REASON FOR ILLUMINATION
NOSE STEER FAIL	Nose steer arming valve has failed open.

FUEL FILTER BYPASS LIGHTS

The FUEL FILTER bypass lights, located on the annunciator panel, are installed on aircraft with -12 engines (Table 4-9).

Table 4-9. FUEL FILTER BYPASS LIGHTS

ANNUNCIATOR	REASON FOR ILLUMINATION
L FUEL FILTER	The associated fuel filter is being bypassed.
R FUEL FILTER	



QUESTIONS

1. The annunciators can be tested:
 - A. Only by individual system test switches
 - B. By pressing each light capsule
 - C. With the annunciator panel PRESS TO TEST switch
 - D. By using the dimmer switches
2. The color(s) of annunciators on the annunciator panel that can be dimmed is:
 - A. Red, amber, and green
 - B. Red
 - C. Amber
 - D. Green
3. The annunciators are dimmed:
 - A. When the TEST switch is depressed
 - B. When the NAV lights are on
 - C. When the dimmers are off
 - D. When the ambient light is greater than 50 candlepower
4. The L and R FUEL shutoff valve annunciators are located:
 - A. On the valve position annunciator panel
 - B. On the annunciator panel
 - C. On the center console
 - D. On the left forward console
5. The crossflow switchlight is located:
 - A. On the pilot's instrument panel beside the crossflow switch
 - B. On the pilot's lower instrument panel next to the fuel quantity indicator
 - C. On the left forward side console
 - D. On the right forward side console
6. Illumination of the R HYD annunciation in the valve position annunciator indicates:
 - A. The right hydraulic shutoff valve is closed.
 - B. The right hydraulic shutoff valve is open.
 - C. The right hydraulic shutoff valve is not in the position selected by the right hydraulic shutoff switch.
 - D. The right hydraulic shutoff valve is in the position selected by the right hydraulic shutoff switch.
7. The annunciator panel is powered by the:
 - A. Left and right essential buses
 - B. Left essential bus only
 - C. Right main bus only
 - D. Left and right main buses
8. The fuel bypass lights can be dimmed:
 - A. Automatically when the navigation lights are on
 - B. With a switch located between the lights
 - C. Automatically by a photoelectric cell
 - D. With a dimmer rheostat



CHAPTER 5

FUEL SYSTEM

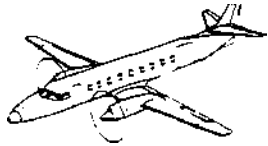
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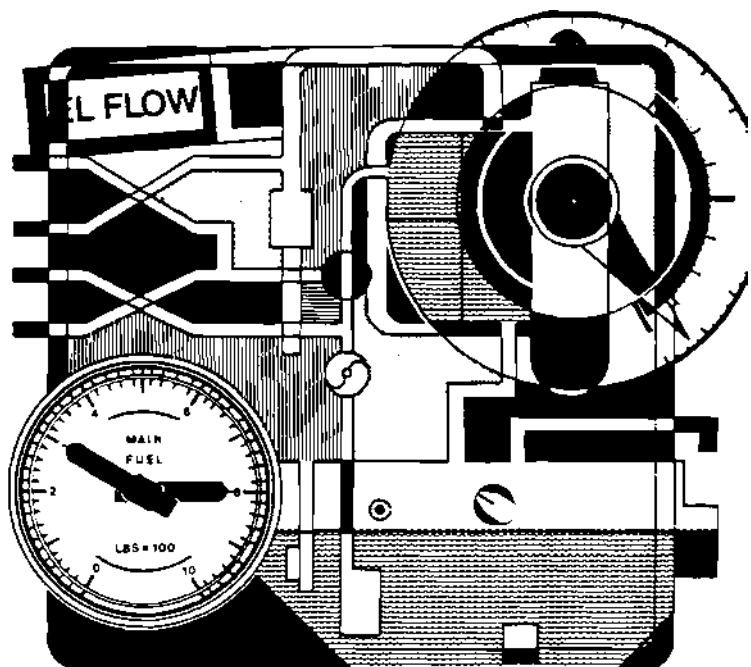


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CHAPTER 5 FUEL SYSTEM



INTRODUCTION

The fuel system consists of the fuel storage and vent, fuel transfer and engine feed, and indicating systems.

The fuel system is covered in this chapter from the fuel tanks to the engine-driven low-pressure fuel pumps, at which point fuel system operation becomes a function of the powerplants. Refer to Chapter 7, "Powerplant," for additional information.

The airplane is fueled by **overwing** gravity fueling. Defueling is accomplished through a single-point defueling drain.

The total usable fuel capacity is 4,342 pounds (648 U.S. gallons at 6.7 pounds per gallon).

GENERAL

The fuel system provides for fuel storage and low-pressure fuel distribution to the engine-driven low-pressure fuel pumps.

The fuel storage system consists of two integral wet-wing tanks and a vent system. A fuel filler port is located on each upper outboard wing surface.



The crossflow valve enables transfer of fuel between wings for fuel balancing. In each wing tank, the 5° wing dihedral and gravity flow ensure supply of fuel to the hopper tank and the two collector tanks located in the inboard section of each wing. The hopper tank contains two electrically driven boost pumps, main and auxiliary.

Each wing tank contains a capacitance fuel gaging system. The dual fuel quantity indicator indicates the amount of usable fuel in each wing tank if the boost pumps are operating.

The fuel flow indicating system shows the amount of fuel that each engine is consuming on the fuel flow gages. Total fuel consumed by both engines is shown on the fuel totalizer.

The fuel pressure indicating system shows the pressure in the line between the engine-driven low- and high-pressure fuel pumps.

FUEL STORAGE

WING TANK STORAGE

The area of the integral wet-wing tank is depicted in Figure 5-1. Intermediate wing ribs serve as baffles to prevent in-flight lateral fuel surging. Flapper valves allow fuel to gravity-flow through the wing tanks and into the hopper tank and prevent reverse flow of fuel.

WING TANK VENT SYSTEM

A flush-mounted ram-air scoop tank vent is located on the lower outboard surface of each wing between the forward and aft spars (Figure 5-1). A vent balance line is routed aft of the aft wing spar to a drain at the center wing section. This vent system provides cross-venting between the tanks. The vents are designed so that they are not affected by airframe icing. A

spark arrester is installed in the vent line at each wing vent and at the vent drain.

WATER DRAIN VALVES

A forward drain is located in the forward collector tank, and an aft drain is located in the hopper tank. Poppet drains, one located on the outboard side of each nacelle, enable draining of condensation and contaminants from the leading-edge wing tank area. (See Figure 5-1.)

FUEL TRANSFER AND ENGINE FEED SYSTEM GENERAL

Fuel to each engine is supplied by an independent fuel system from its respective wing tank. A crossflow line interconnects the left and right wing tanks. Two collector tanks located at the inboard end of each fuel tank refill by gravity flow. A hopper tank between the collector tanks is kept full by a jet pump system, providing a boost pump is on. Two submerged fuel pumps located in the hopper tank supply fuel through the fuel supply line and fuel shutoff valve to the engine.

HOPPER TANKS

A hopper tank is located in the inboard section of each wing tank between the nacelle and the fuselage (Figure 5-2). A main and an auxiliary boost pump in each hopper tank provide low-pressure fuel to the engine-driven low-pressure fuel pumps. Four jet transfer pumps (two for each hopper) are driven by the boost pumps during boost pump operation and keep the hopper tanks filled to capacity from the collector tanks. Each hopper tank has a capacity of approximately 94 pounds (14 U.S. gallons), of which approximately 13 pounds (2 U.S. gallons) are unusable.



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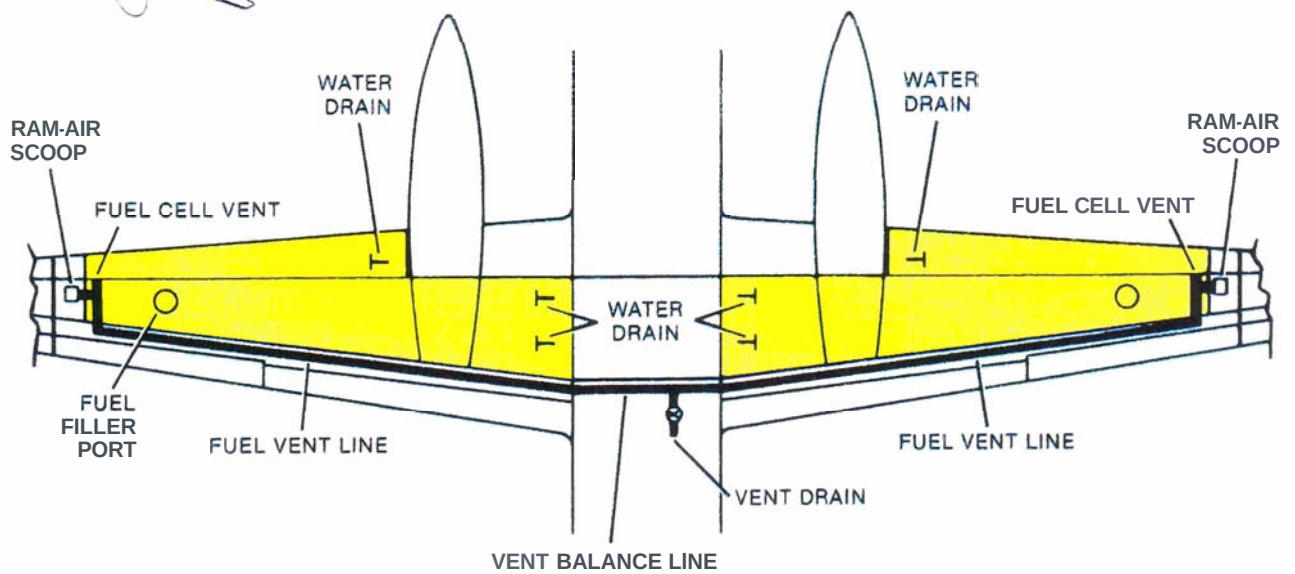


Figure 5-1. Fuel Storage and Tank Vent System

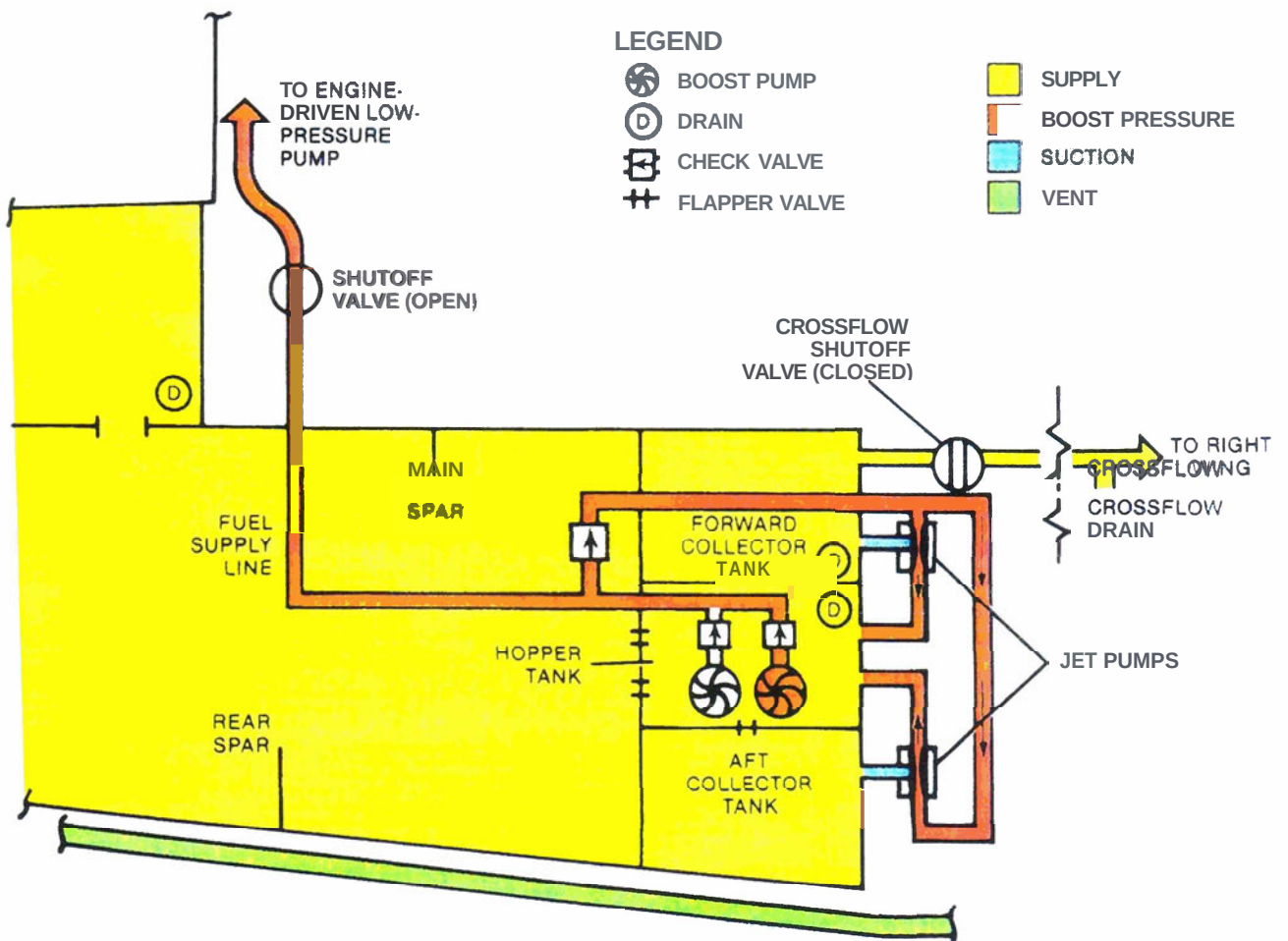
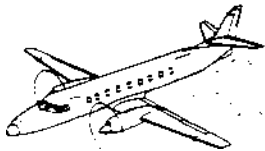


Figure 5-2. Fuel System



BOOST PUMPS

The two boost pumps in each hopper tank are controlled by the L BOOST and R BOOST switches on the pedestal. The positions of these rocker switches are labeled "MAIN," "OFF," and "AUX" (Figure 5-3).

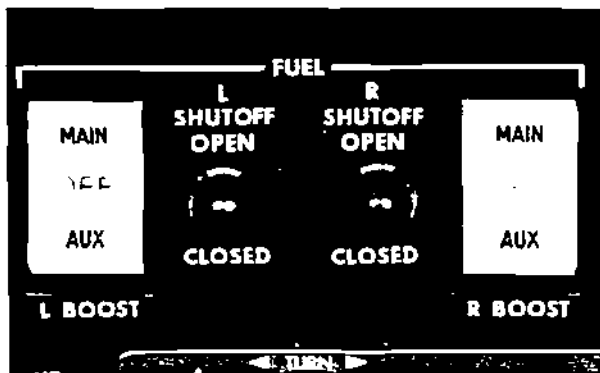


Figure 5-3. Fuel Boost Pump and Shutoff Valve Controls

These boost pumps deliver low-pressure fuel to the engine-driven low-pressure fuel pumps. Check valves, one for each boost pump, prevent reverse flow from one boost pump to the other.

Each main boost pump is powered from its respective 28.5-volt DC essential bus. Each auxiliary boost pump is powered from the opposite 28.5-volt DC essential bus.

JET TRANSFER PUMPS

Turning on either boost pump forces fuel through the two jet transfer pumps for each hopper tank. The jet transfer pumps use the venturi principle to draw fuel from the forward and aft collector tanks and return it to the hopper tank. (See Figure 5-4.)

A float switch in each hopper tank illuminates the respective R XFER PUMP or L XFER PUMP caution lights on the annunciator panel (see Appendix B) when the hopper tank quantity is less than approximately 75 pounds.

Since the jet transfer pumps are operated by the boost pumps, turning on the applicable boost pump should extinguish the XFER PUMP light. If not, the jet transfer pumps may be inoperative. The fuel quantity gage readings represent the total amount in pounds of **usable** fuel available with the boost pumps and jet transfer pumps operative and energized. With boost pumps and/or jet transfer pumps inoperative, 75 pounds of the fuel indicated on the fuel quantity gage for that wing tank will be unusable, and the pilot should plan accordingly.

If the boost pumps are not used or are inoperative, wing dihedral will no longer keep the hopper tank full, and with approximately 600–700 pounds of fuel remaining in the tank, and the XFR PUMP caution light will illuminate.

If the boost pumps and jet pumps are not energized or are inoperative when the XFER PUMP light illuminates, remaining fuel in the wings and hopper will decrease at an even rate until the wing dihedral ceases to gravity-fill the hopper tank. The fuel quantity indicator will indicate that 75 pounds of fuel remain in the tanks; however, that fuel is unusable and engine flameout is impending.

With boost pumps and jet transfer pumps working normally, the hopper tank will stay full until the wing tank fuel level reaches approximately 75 pounds. At this time, the level in the hopper tank will start to decrease and the XFR PUMP caution light will illuminate. Approximately 75 pounds will be indicated on the fuel quantity gage and is all that is available for use.

Figure 5-5 illustrates operation with and without boost pumps.

CROSSFLOW SYSTEM

A crossflow line (Figure 5-2) interconnects the left and right forward collector tanks and incorporates a crossflow valve for fuel balancing by gravity feeding fuel from one wing tank to the other.



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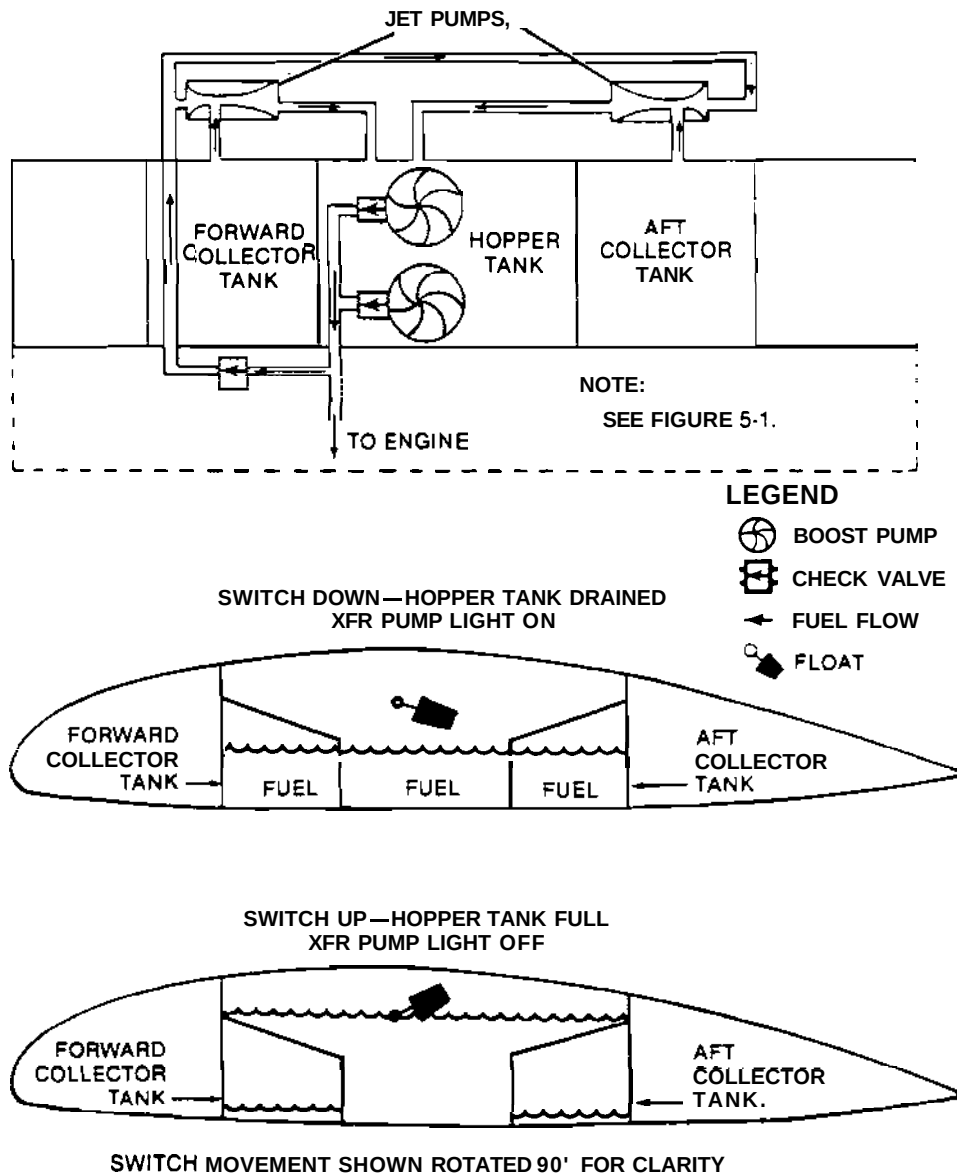
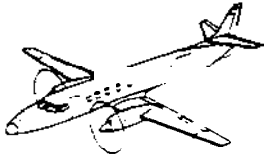


Figure 5-4. Jet Pump System



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The crossflow valve is controlled by the CROSS-FLOW SWITCH, a push-on/push-off switch-light located on the pilot's instrument panel (Figure 5-6). The amber light in the switch illuminates when the switch is selected to OPEN. The light indicates only the switch selection. The X-FLOW OPEN light (See Appendix B) on the valve position annunciator panel illuminates when the valve is fully or partially open.

The fuel crossflow switch and valve are powered from either the left or right 28.5-volt DC essential bus as selected on the ESSENTIAL BUS TRANSFER panel on the left aft console (Figure 5-6A). This switch is normally selected to the left essential bus.

A quick-release drain valve is installed in the crossflow line to allow for rapid single-point defueling of the airplane.

FUEL SHUTOFF VALVE SYSTEM

A fuel shutoff valve in each wheel well (Figure 5-2) is controlled by its respective FUEL SHUT-OFF switch on the pedestal (Figure 5-3). These normally open valves allow fuel to pass through the firewall. The amber L or R FUEL shutoff valve disagreement light (See Appendix B) illuminates when the valve on its respective side is not in the position selected by the switch. The system on each side is powered from its respective essential DC bus.

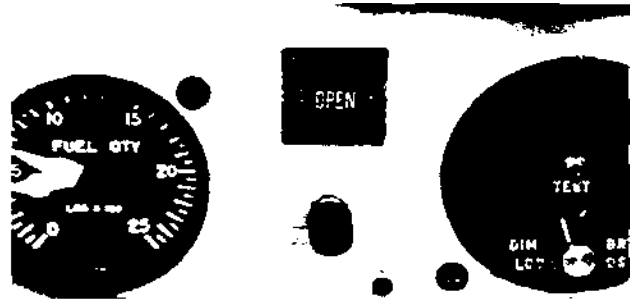
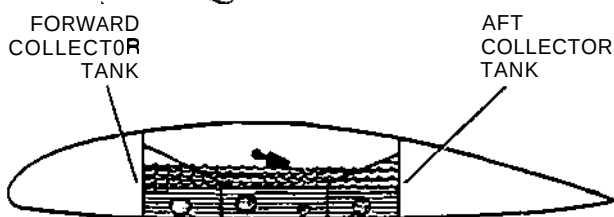


Figure 5-6. Fuel Crossflow Switch

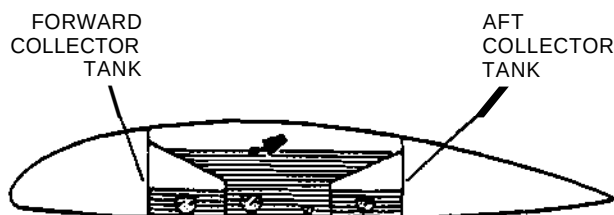


Figure 5-6A. Essential Bus Transfer Switches

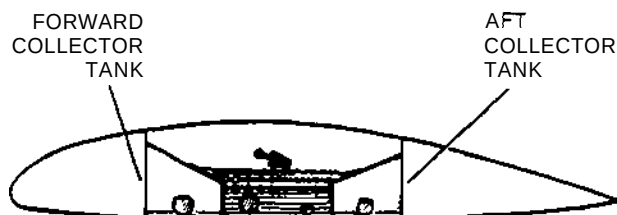
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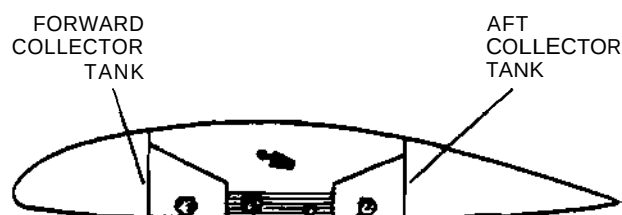
BOOST PUMP IS OFF,
AT 600 TO 700 POUNDS OF FUEL REMAINING,
THE TRANSFER PUMP LIGHT ILLUMINATES.



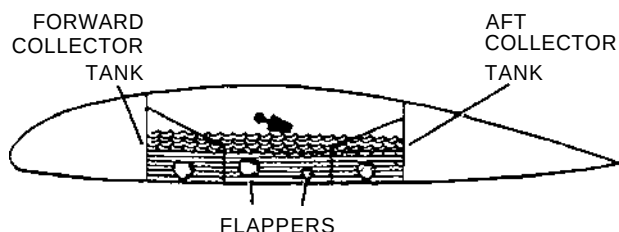
BOOST PUMP IS ON,
THE TRANSFER PUMP LIGHT EXTINGUISHES.



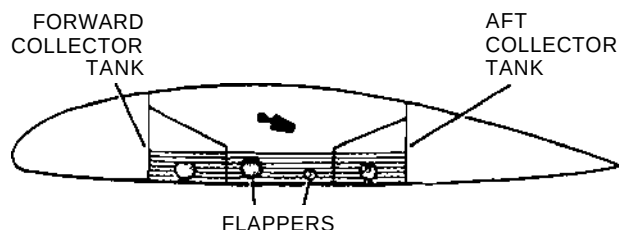
WITH 75 POUNDS OF FUEL REMAINING,
THE TRANSFER PUMP LIGHT
ILLUMINATES AGAIN.



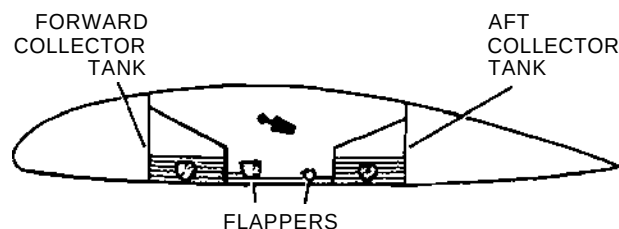
**FUEL QTY GAGE READS
LESS THAN 75 POUNDS,**
ENGINE FLAMEOUT IS IMPENDING.



**WITH 600 TO 700 POUNDS OF FUEL
REMAINING, THE XFER PUMP LIGHT
ILLUMINATES.**



**REMAINING FUEL DROPS AT
AN EVEN LEVEL.**



**FUEL QTY GAGE READS 75 POUNDS
REMAINING (UNUSABLE FUEL). ENGINE
FLAMEOUT IS IMPENDING.**

Figure 5-5. Internal Fuel Transfer System



FUEL QUANTITY INDICATING

The fuel quantity capacitance gaging systems operate on input from five fuel probes in each wing tank, and each gage is powered from its respective engine instrument bus.

The FUEL QTY gage (Figure 5-7) on the instrument panel is calibrated in pounds X 100. This gage is equipped with two pointers, one for each wing tank. The gage is tested using the PRESS-TO-TEST push-button switch adjacent to it. When pressed, the pointers should move to 1,250 pounds; when released, the pointers should return to the pre-test quantity. The airplane must be in coordinated flight or wings level on the ground to obtain an accurate reading.

A mechanical fuel level indicator (standard on all aircraft after SN 682 and optional on all others) allows the checking of the fuel quan-

tity, in gallons, of each wing tank. The indicator (magna-stick) is located under each wing tank inboard of the nacelle (Figure 5-8). A reading is taken by pushing up the lock tab, turning 90° in either direction, and allowing the indicating scale to drop down.

FUEL FLOW INDICATING

The FUEL FLOW indicators and the fuel totalizer (Figure 5-7) operate on input from a fuel flow transmitter installed on each engine.

The fuel flow indicators are calibrated in pounds per hour and indicate the amount of fuel that each engine is consuming.

The fuel-consumed totalizer indicates the total amount of fuel consumed by both engines since the counter was last zeroed. Zeroing is accomplished by pressing the mechanical reset button on the totalizer.

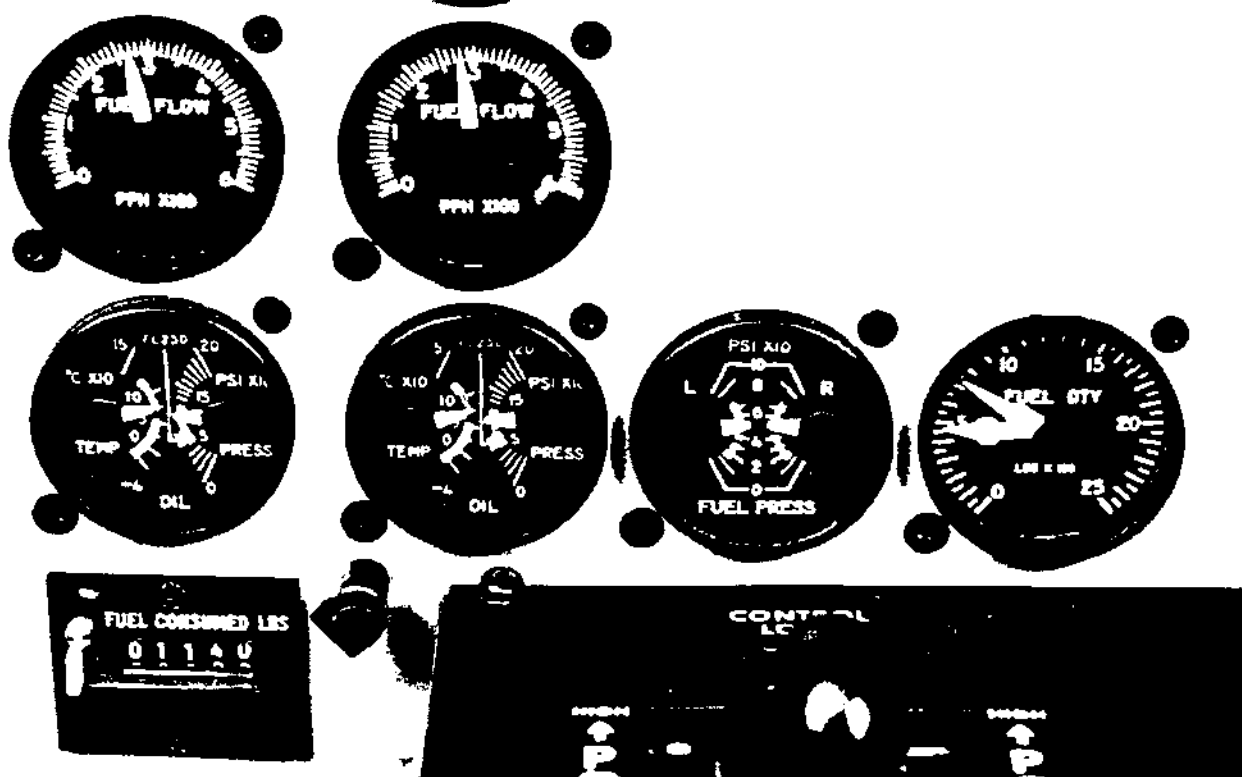
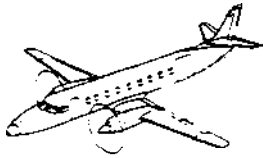


Figure 5-7. Fuel System Indicators



The left and right engine FUEL FLOW indicators are each powered from the respective 28-volt DC engine instrument bus. The FUEL totalizer is powered from the nonessential bus.

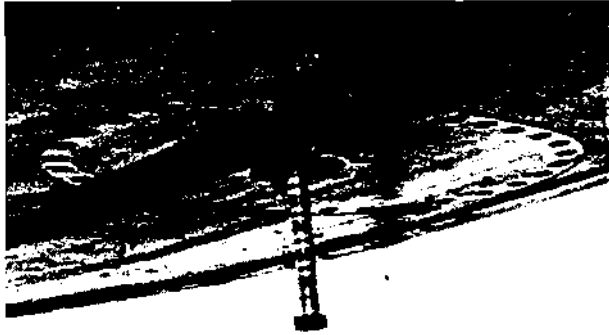


Figure 5-8. Magna-Stick

FUEL PRESSURE INDICATING

The dual FUEL PRESSure gage (Figure 5-7) indicates fuel pressure on each engine between the engine-driven low-pressure pump and the engine-driven high-pressure pump within the fuel control of each engine. An inoperative boost pump is detected by zero fuel pressure prior to engine start and a pressure 15 to 20 psi lower than normal with the engine in operation. This system is powered from the 28-volt DC engine instrument bus.

LIMITATIONS

APPROVED FUELS

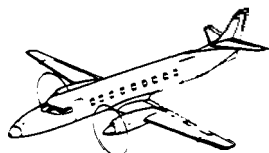
Aviation fuels conforming to Garrett AiResearch *Installation Manual IM 5117* (Jet A, Jet A-1, Jet B, JP-1, JP-4, and JP-5) may be used. Grade 100LL aviation gasoline may be used, provided the restrictions specified in the Limitations section of the *AFM* are observed.

FUEL ADDITIVE

Use of icing inhibitor fuel additive is approved subject to restrictions specified in the Limitations section of the *AFM*.

FUEL IMBALANCE

The aileron trim setting and control wheel force requirements for operation with a fuel imbalance depend on total fuel loading and the airspeed. The control wheel force and aileron trim requirements increase with increased imbalance. For all 16,000-pound aircraft and for Metro IIIs with McCauley props, the maximum allowable fuel imbalance for takeoff and landing is 200 pounds if total fuel quantity is more than 2,000 pounds and 400 pounds if total fuel quantity is less than 2,000 pounds. For all other Metro III and Merlin IVC aircraft, the maximum demonstrated fuel imbalance for takeoff and landing is 500 pounds.



FUEL BOOST PUMP AVAILABILITY

Boost pumps must be on for all flight operations outside the No Boost Pumps Required envelope. Takeoff with the XFER PUMP light(s) illuminated is prohibited. Figure 5-9 shows the fuel boost pump availability requirements.

NOTE

Two operable boost pumps per wing fuel tank are required for all operations with aviation gasoline, JP-4, or Jet B fuel.

When using Jet A, Jet A-1, JP-1, or JP-5, takeoff and flight operations without boost pumps are permitted within the limitations indicated in Figure 5-9. For all other operations, all fuel boost pumps must be operable.

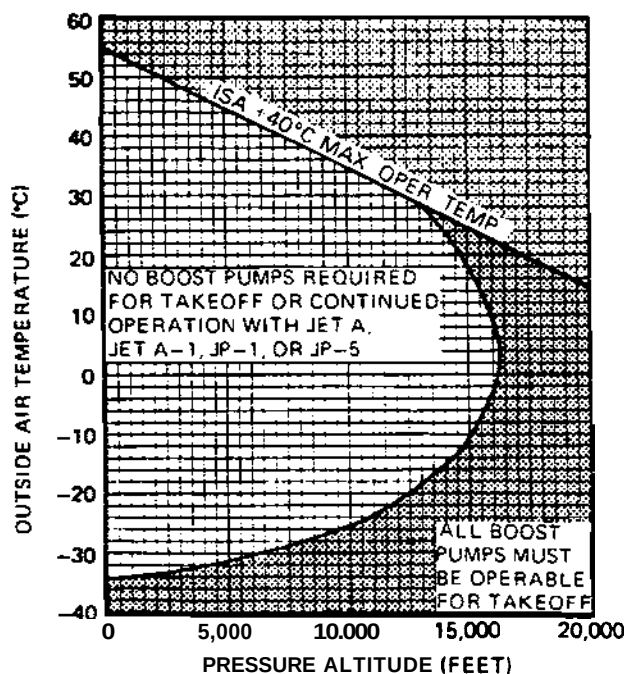


Figure 5-9. Fuel Boost Pump Availability Requirements

REFUELING

Overwing refueling (Figure 5-10) must be accomplished with electrical power off and the airplane, fuel truck, and fuel hose nozzle properly grounded.

CAUTION

Do not attempt to completely fill one tank before adding fuel to the opposite tank as excessive lateral imbalance will occur. When fueling, fill both tanks at the same time or alternately add 125 gallons to each tank until the desired amount is in each tank.



Figure 5-10. Overwing Refueling Receptacle

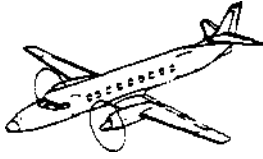


QUESTIONS

1. The following statement is true:
 - A. The fuel cell vents are located in the wing center tank.
 - B. The jet transfer pumps deliver fuel from the collector tanks to the hopper tank.
 - C. The jet transfer pumps are driven by the engine-driven fuel pumps.
 - D. The auxiliary boost pumps on each side are powered from their respective 28.5-volt DC essential bus.
2. With the boost pumps inoperative, the wing tank fuel quantity indicated but unusable is:
 - A. Approximately 94 pounds
 - B. Approximately 88 pounds
 - C. Approximately 75 pounds
 - D. Approximately 13 pounds
3. The boost pumps:
 - A. Are driven by the engine.
 - B. Deliver fuel to the engine-driven low-pressure fuel pumps.
 - C. Keep the collector tanks filled with fuel.
 - D. Use high-pressure fuel tapped from the engine.

Assuming that the jet transfer pumps are operative, the amber L and R XFER PUMP lights are extinguished by:

 - A. Turning on the boost pumps
 - B. Opening the crossflow valve
 - C. Turning on the boost pumps when fuel remains in the wing tanks
5. Power for the following is transferable from the left essential bus to the right essential bus:
 - A. The main boost pumps
 - B. The auxiliary boost pumps
 - C. The jet transfer pumps
 - D. The crossflow valve
6. The quick-release drain valve for rapid single-point defueling is located in the:
 - A. Hopper tank in the wing
 - B. Crossflow line under the fuselage
 - C. Forward collector tank in the wing
 - D. Aft collector tank in the wing
7. An accurate fuel quantity reading is obtained in flight by:
 - A. First pressing the PRESS-TO-TEST push-button and verifying that the indicators move to the 12 o'clock position
 - B. First establishing coordinated flight
 - C. Balancing the fuel
 - D. All of the above
8. Illumination of the R FUEL light indicates that the:
 - A. Right wing has less than 75 pounds of fuel remaining.
 - B. Right fuel shutoff valve is closed.
 - C. Right fuel pump pressure is low.
 - D. Right fuel shutoff valve is not in the position of the right fuel shutoff switch.
9. During refueling, avoid:
 - A. Refueling with the airplane's electrical power on
 - B. Filling both tanks at the same time
 - C. Grounding of the airplane, fuel truck, and fuel nozzle
 - D. Use of aviation gasoline



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The information normally contained in this chapter is not applicable to this particular aircraft.

FOR TRAINING PURPOSES ONLY



CHAPTER 7 POWERPLANT

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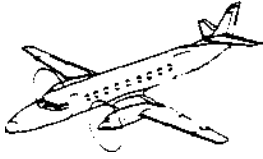
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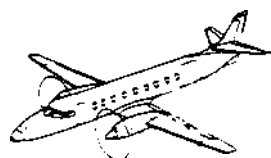


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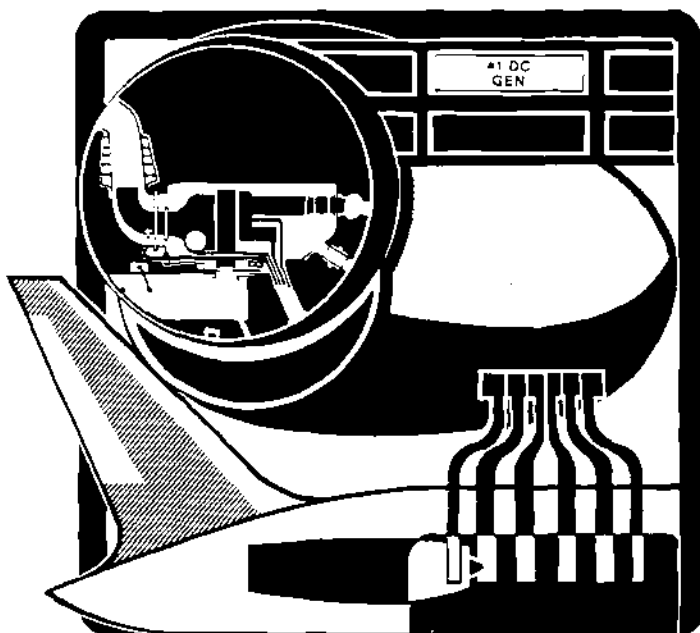
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CHAPTER 7 POWERPLANT



INTRODUCTION

This chapter features descriptions and operation of engine systems such as the oil, fuel, ignition, engine controls and instrumentation, engine temperature-limiting, and water-alcohol injection.

GENERAL

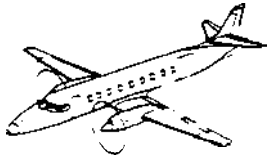
The airplane is powered by two turboprop engines driving four-blade constant-speed propellers. The propellers include full feathering and reversing capabilities, as well as Beta mode control for ground handling and Beta mode follow-up during flight to limit blade angle to a safe minimum if malfunctions occur.

The engines are manufactured by the Garrett Turbine Engine Company of Phoenix, Arizona and are designated **TPE331**. They are lightweight, fixed-shaft turboprops designed

to operate as constant-speed engines, although various speed ranges can be selected.

The propellers are manufactured by Dowty RotoI Company, Limited, of England or the **McCauley** Accessory Division of Cessna Aircraft, Vandalia, Ohio.

The Metro III and Merlin IVC are each powered by two **TPE331-11U** engines, incorporating a factory-installed alcohol-water injection system. This engine is dry-rated at 1,000 shaft horsepower for takeoff and



continuous operation. It is wet-rated at 1,100 shaft horsepower for a maximum of five minutes from the start of takeoff roll.

NOTE

Modifications applicable to the -11U engines on the Metro III and Merlin IVC are each identified by specific configuration dash numbers and Application-Aircraft alphabetical codes. These must be determined from the applicable *AFMs*.

ENGINES

GENERAL

The TPE331 engine (Figure 7-1) includes a two-stage, centrifugal compressor and a three-stage, axial-flow turbine mounted on a single shaft. An annular, reverse-flow combustor surrounds the turbine. A two-stage reduction gear with an integral inlet duct is located at the front of the engine. The reduction gear forms the power conversion section of the engine. An exhaust duct, located at the rear of the engine, directs the combustion gases to the atmosphere.

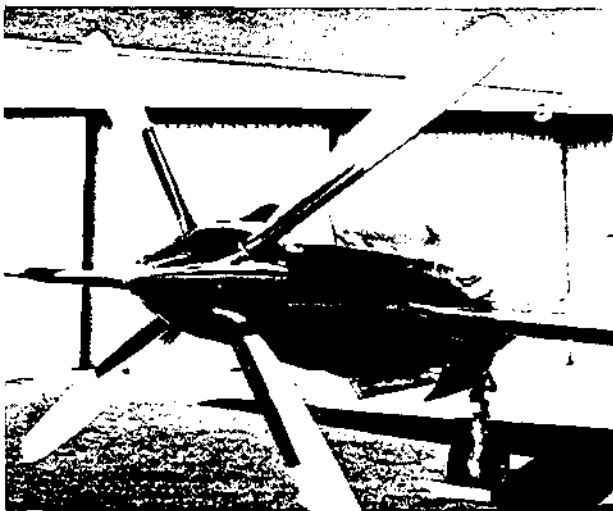


Figure 7-1. Wing-Mounted Engines

AIRFLOW STATIONS

The engine manufacturer assigns station numbers to assist in determining various functions and locations within the airflow path. Figure 7-2 illustrates the station numbers assigned to the TPE331 engine: station 1 represents the ambient conditions existing outside of the engine, station 2 the compressor inlet, station 3 the compressor discharge, station 4 the turbine inlet, and station 5 the turbine discharge or exhaust.

When these numbers are combined with alphabetical symbols, identification of various operations within the engine is simplified. For example, combining P (pressure) and 3 (station 3) produces the symbol P_3 , or compressor discharge pressure, while P_5 signifies exhaust discharge pressure downstream of the turbine. T (temperature) and 4 (station 4) form the symbol T_4 , or turbine inlet temperature.

MAJOR SECTIONS

GENERAL

The engine is divided into five major sections (see Figure 7-2) from front to rear, as follows:

1. Reduction gear
2. Compressor
3. Combustor
4. Turbine
5. Exhaust

REDUCTION GEAR

The reduction gear consists of an air accessory drive gear and a planetary gear. At 100%

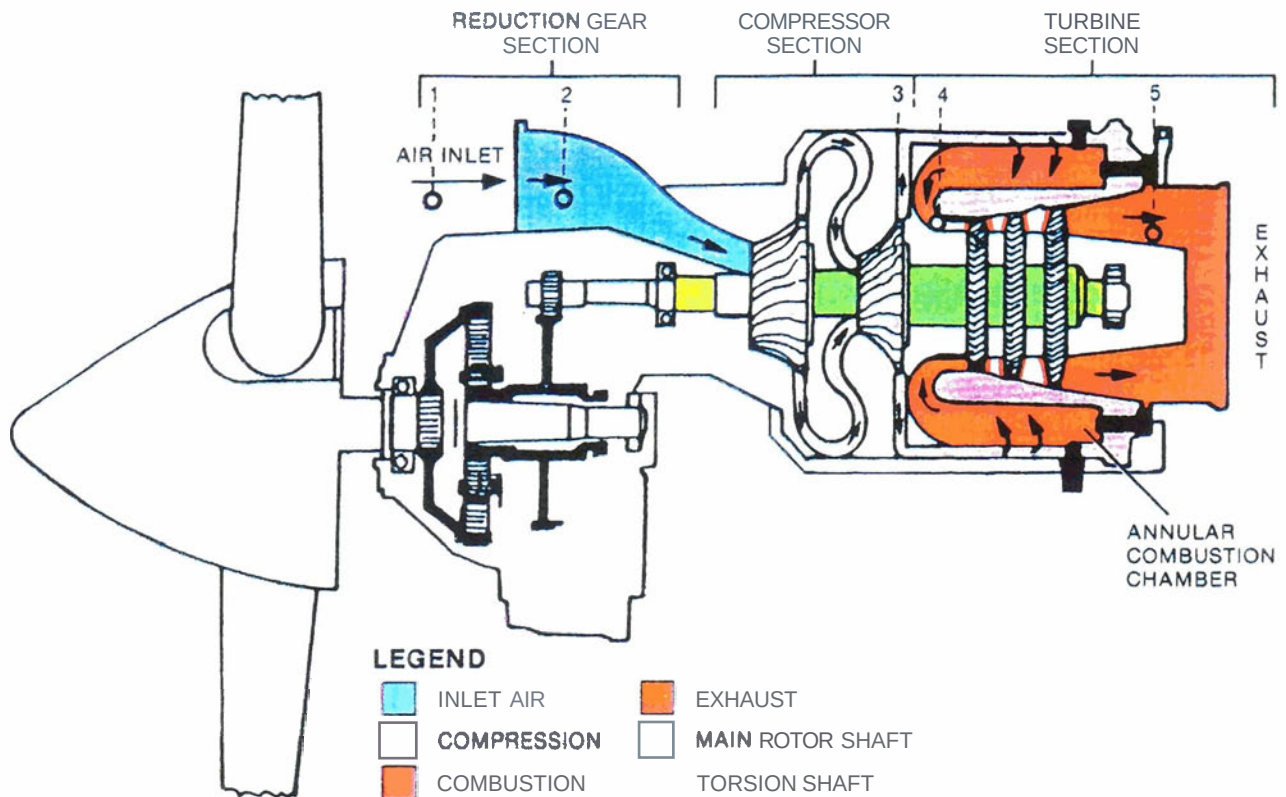


Figure 7-2. Engine Stations, Major Sections, and Gas Flow

engine rpm (41,730). the propeller rotates at 1,591 rpm. The accessories driven by the reduction gear include:

- Fuel control and engine-driven fuel pumps
- Hydraulic pump
- Starter-generator
- Tach generator
- Propeller governor
- Oil pump

COMPRESSOR

The two-stage centrifugal compressor has an overall compression ratio of approximately 10 to 1. About one-third of the compressor

discharge air is used to support the actual fuel combustion. The remaining air is used primarily to control the location of the fireball within the combustion chamber and to reduce the temperature of the combustion products prior to their entering the turbine.

COMBUSTOR

The combustor is an annular, reverse-flow type with ten duplex spray nozzles (primary and secondary) and two igniter plugs.

TURBINE

The three-stage, axial-flow turbine extracts energy from the combustion gases to drive the compressor, accessories, and propeller.

About two-thirds of the power produced by the turbine is used to drive the compressor



and the accessories. The remaining energy is applied to the propeller shaft through the reduction gear.

EXHAUST

The exhaust section directs the exhaust gases to the atmosphere through an overwing duct.

OPERATING PRINCIPLES

Figure 7-2 illustrates the operating principles of a gas turbine engine. Ambient air is directed to the engine inlet through the nacelle inlet duct. The diffusion process in the two-stage compressor increases air pressure and directs it rearward to the combustor. The airflow is turned 180° and flows forward. A precise portion of this airflow enters the combustion chamber, where fuel is added by the duplex fuel nozzles.

The gas mixture is initially ignited by the two high-energy igniter plugs, after which combustion is self-sustaining. The expanding combustion gases flow forward in the combustion chamber, turn 180°, and flow aft to the turbine. The turbine extracts nearly all the energy to drive the compressor. The remaining high rpm-low torque developed by the turbine is transmitted forward to the reduction gear. The reduction gear converts this high rpm-low torque to low rpm-high torque to drive the accessories and the propeller. The spent gases leaving the turbine are directed to the atmosphere by the overwing exhaust pipe. The available thrust from these gases is very low.

ENGINE SYSTEMS

GENERAL

The engine systems include the following:

- Instrumentation
- SRL autostart computer

- Oil system
- Fuel system
- Temperature-monitoring system
- Torque-monitoring system
- Ignition system

ENGINE INSTRUMENTATION

The engine instruments (Figure 7-3) are located in two vertical rows on the center instrument panel. The left and right engine

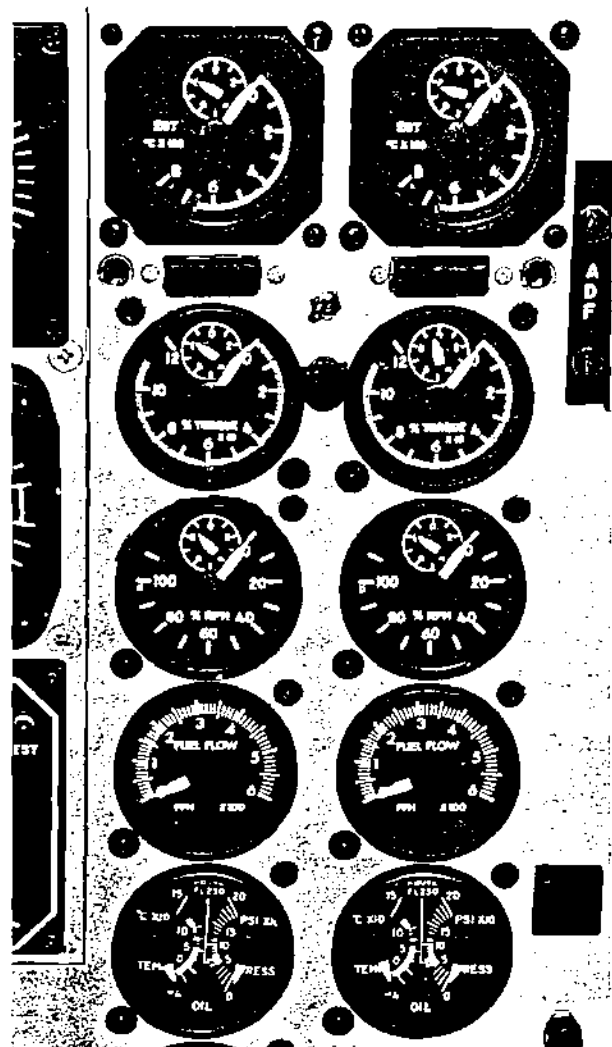


Figure 7-3. Engine Instruments



instruments normally receive 28.5-VDC power from the **ENG INST BUS** circuit breakers on their respective essential buses (Figure 7-4).

If either essential bus loses power, the failed engine instrument bus will automatically be powered by the opposite essential bus.

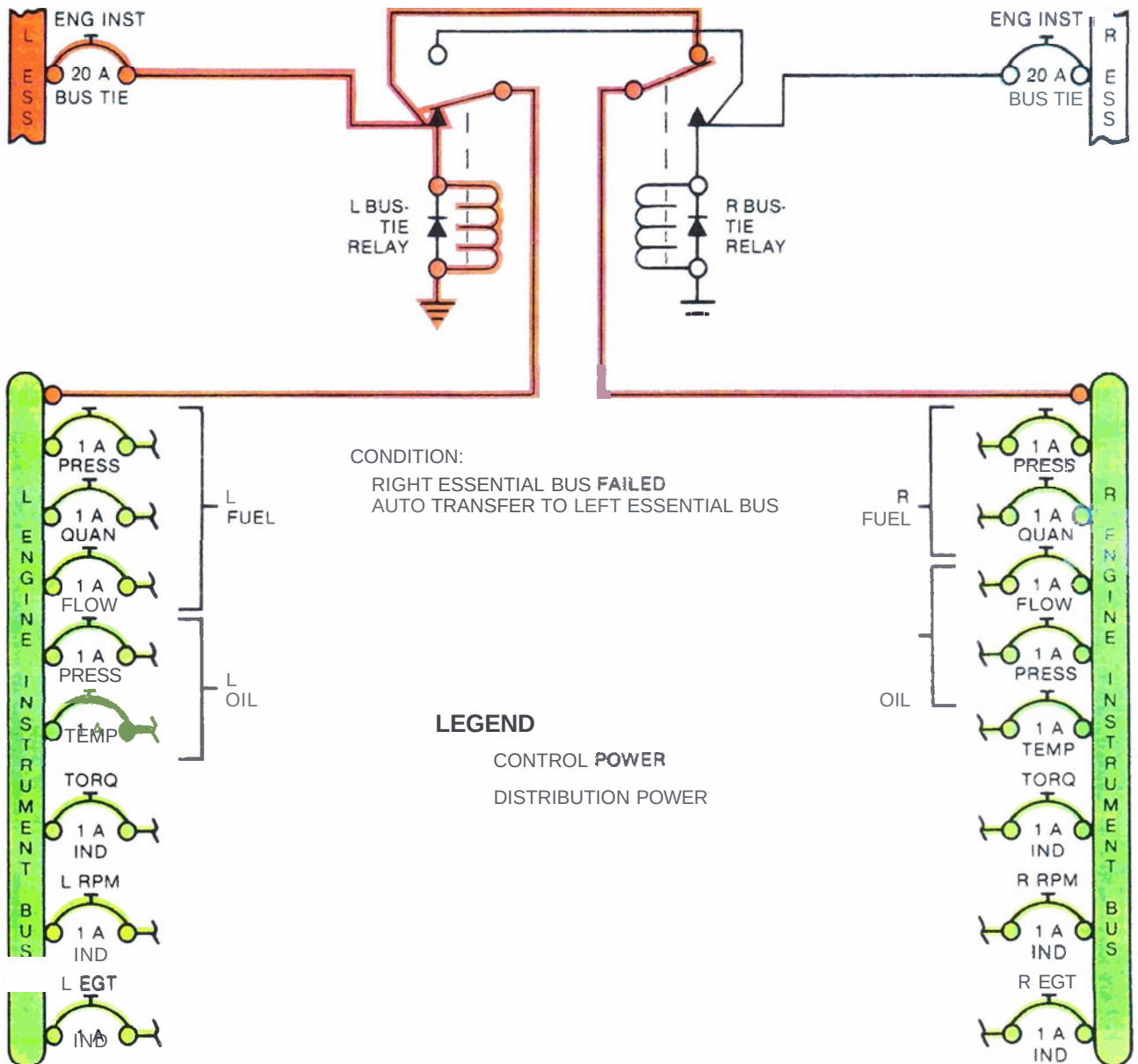


Figure 7-4. Engine Instrument Power Schematic



From top to bottom, the gages include:

- Exhaust gas temperature (EGT)—The EGT gages are marked in degrees Celsius. When rpm is below 80%, the EGT gage shows compensated EGT. When rpm is above 80%, the EGT gage shows an EGT computed by the single red line (SRL) computer.
- Engine torque— The torque gages are marked in percentage of torque from 0 to 120%. 100% is equal to the maximum continuous rating of 1,000 shaft horsepower (shp) at 100% rpm. 11090 is equal to the maximum takeoff (wet) rating of 1,100 shp at 100% rpm. Input signals to the torque gages are from an electronic torque-measuring system in the reduction gear, described later in this chapter.
- RPM— The engine rpm gages are calibrated in percentage of maximum continuous rpm, which is 100%. Inputs to the gages are from tach generators driven by the accessory gear system.
- Fuel flow — The fuel flow system is calibrated in pounds per hour. The input to the gages is from fuel flow transducers in the fuel lines.

SINGLE RED LINE (SRL) AUTOSTART COMPUTER

Description

The SRL autostart computer controls three functions: engine speed switching functions, automatic start fuel enrichment, and single red line EGT computation. Inputs to the SRL autostart computer include compensated exhaust gas temperature (T_e), engine rpm, compressor inlet temperature (T_c), airspeed and altitude from the A P/P (P_2/P_5) transducer, and whether or not alcohol-water injection (CAWI) is being used.

The speed switching functions of the autostart computer include: (1) automatic control of the start sequence from 1090 to 60% rpm, (2) SRL EGT computation above 80%

rpm, and (3) enabling CAWI operation and EGT temperature limiting above 90% rpm.

The automatic start fuel enrichment modulates the start fuel enrichment valve to provide an appropriate amount of fuel during the start.

The SRL EGT computation results in an indicated EGT with a single maximum EGT of 650° for all operating conditions.

Control

A pair of three-position SRL $\Delta P/P$ power switches (Figure 7-5) on the left console are provided for testing the SRL computation.

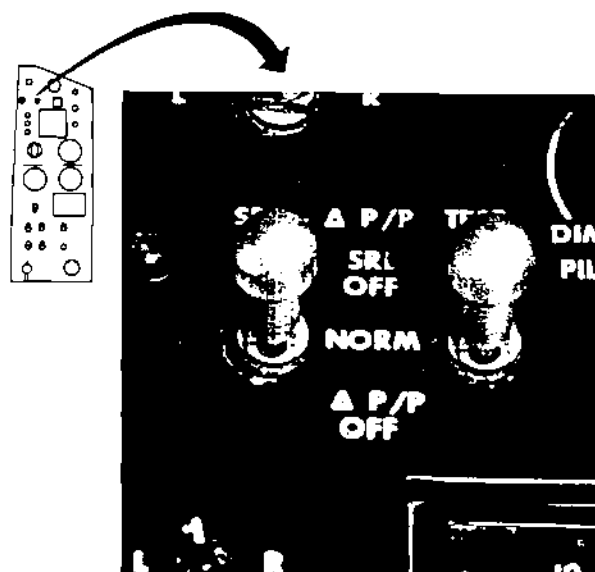


Figure 7-5. SRL $\Delta P/P$ Power Switches

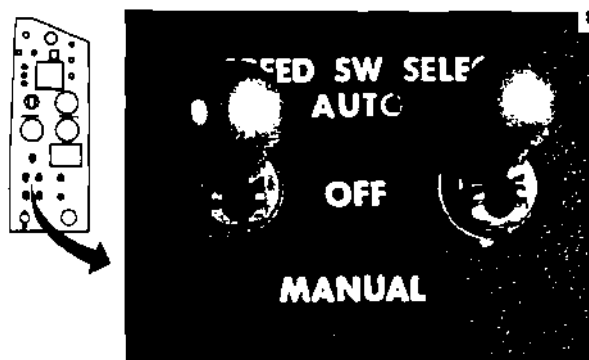


Figure 7-6. Speed Switch **Select** Switches



The other switches (Figure 7-6) marked "SPEED SW SELECT." with three positions labeled "AUTO," "OFF," and "MANUAL," control power to the 10% and 60% speed switches and may be used to start the engine if control of the automatic start sequence malfunctions.

ENGINE OIL SYSTEM

General

The engine oil system provides for cooling and lubrication of the main engine bearings and the reduction gear. In addition to these functions, the engine oil system supplies oil for propeller control, negative torque sensing (NTS), and the unfeathering pump.

The engine oil system is a dry-sump pressure-scavenge system consisting of one pressure pump and three scavenge pumps, all of which are engine-driven.

The oil tank is mounted on the firewall to the left of the engine. The tank includes a filler neck, screen, and sight gage. Servicing and checking can be done through an access door (Figure 7-7) on the left side of each nacelle.

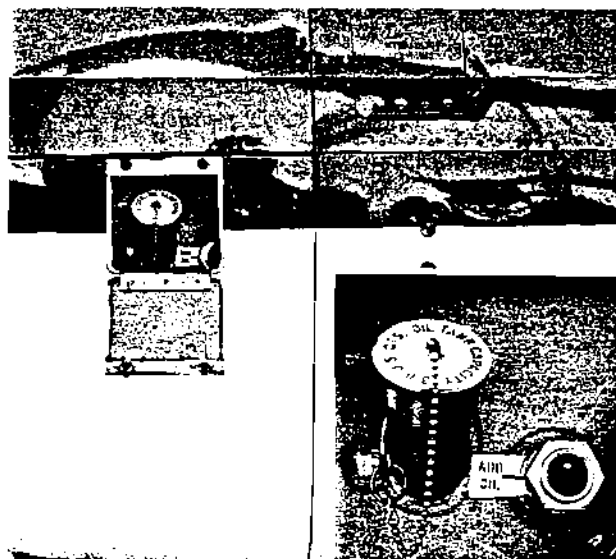


Figure 7-7. Oil Tank Access and Sight Gage

A fully automatic oil cooler is located in the nacelle. Ram air is supplied to the cooler through an inlet duct on the bottom of each nacelle.

An oil filter includes a pop-out pin to indicate bypass of a clogged element.

An oil vent valve introduces air into the inlets of the pressure pump and gearcase scavenge pumps during engine ground starting to reduce drag during start acceleration. Vent valve control is automatic through the engine start system.

Indication

Oil pressure is sensed by a DC transmitter to provide cockpit indication on a combination oil pressure-temperature gage (Figure 7-8) located on the engine instrument panel. DC power is supplied from the associated engine instrument bus.

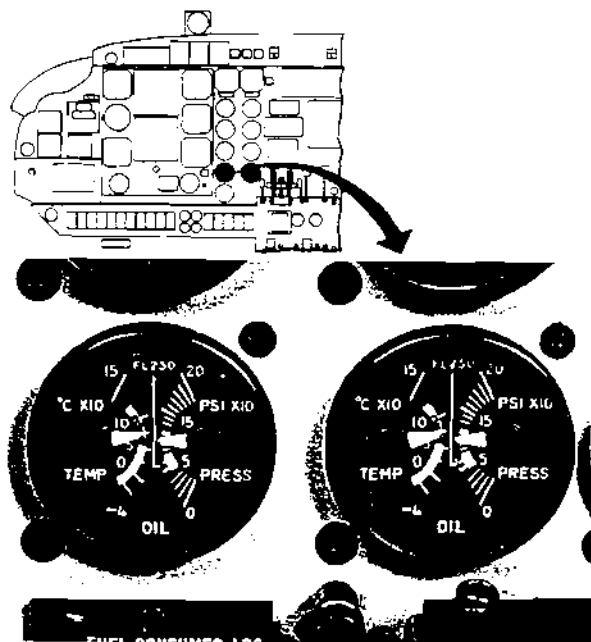


Figure 7-8. Oil Pressure-Temperature Gages



Oil pressure is also sensed by a pressure switch. If oil pressure drops below 40 psi, the pressure switch illuminates the appropriate L or R OIL **PRESSURE** warning light (see Appendix B) on the annunciator panel.

Oil temperature is sensed by a DC resistance bulb which transmits to the respective combination oil pressure-temperature gage (Figure

7-8). The temperature scale is calibrated in degrees Celsius, and the indicator is powered from the respective engine instrument DC bus.

A magnetic chip detector in each gearcase is connected to the respective L or R CHIP DET warning light (see Appendix B) on the annunciator panel. When a chip detector warning light illuminates, it indicates metal particles in the oil.

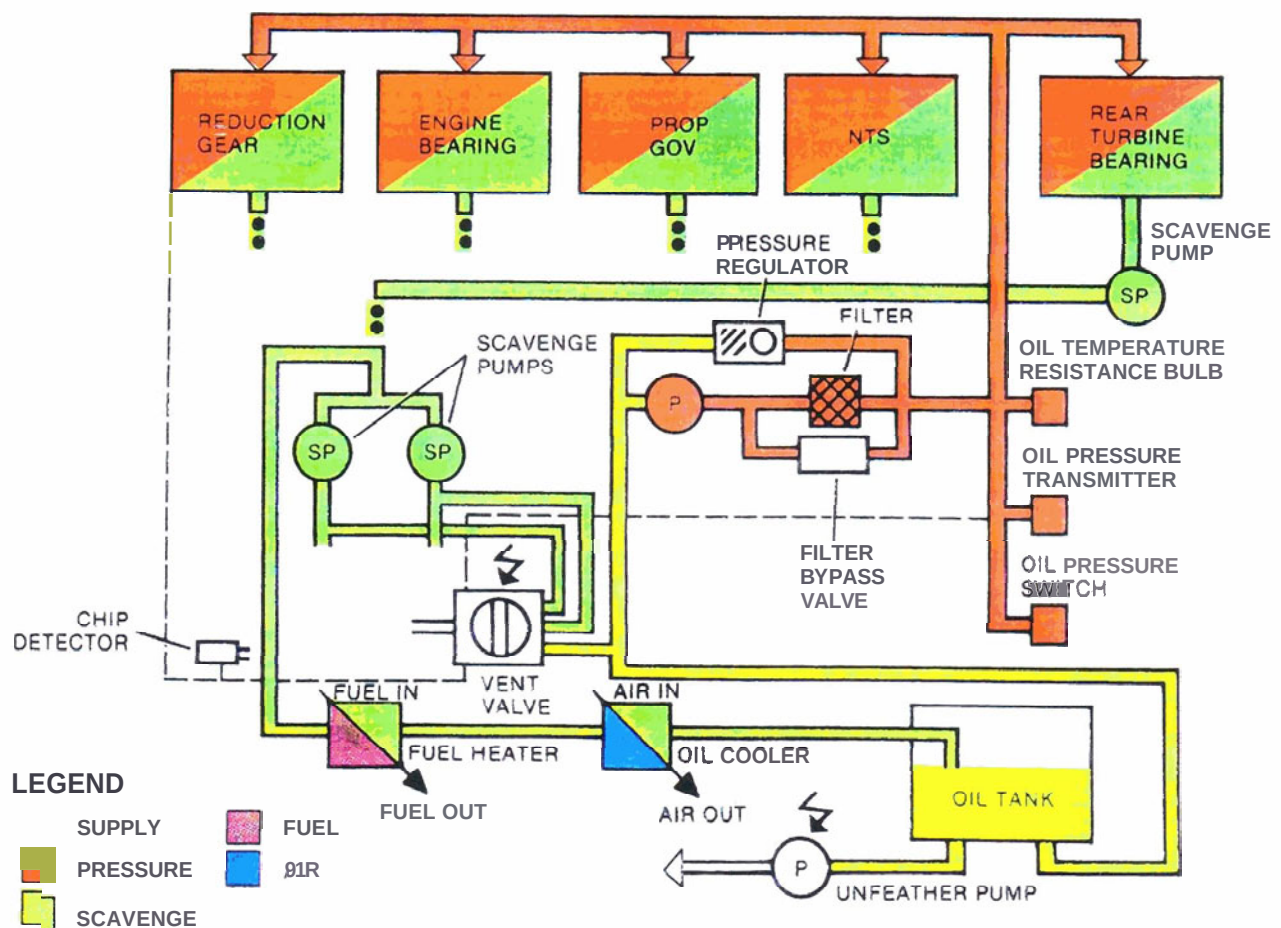
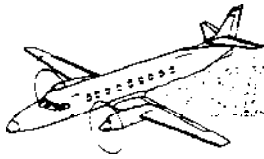


Figure 7-9. Oil System Schematic



Approved Oils

Engine oil must be a Type II oil meeting military specification MIL-L-23699B.

NOTE

Do not mix brands of oil.

Operation

The pressure pump draws oil from the tank and supplies it under pressure to the oil filter. Distribution oil pressure is limited by a pressure regulator.

Two scavenge pumps in the gearcase are used to scavenge the oil back to the tank. The rear turbine bearing has its own scavenge pump to return oil to the gearcase. The gearcase scavenge oil passes through a fuel heater and an oil cooler on the way back to the oil tank. Both the fuel heater and oil cooler are automatically controlled. Figure 7-9 shows the operation of the engine oil system.



Figure 7-10. P_2T_2 Sensor

The best time to check oil quantity is within ten minutes after engine shutdown. After that time, oil tends to siphon from the tank to the engine. Oil can be returned to the tank by motoring the engine to approximately 15% rpm.

ENGINE FUEL SYSTEM

General

The engine fuel system provides the proper amount of metered fuel to the combustion chamber under three phases of operation:

- Starting and acceleration to idle
- Ground operation
- Flight operation

The engine fuel system includes engine-driven, low- and high-pressure pumps, a fuel control unit, a fuel solenoid valve, a flow divider, fuel manifolds, and primary and secondary fuel nozzles. In addition to these major components are the start fuel enrichment valve, a primaries-only solenoid valve, a fuel anti-icing valve, an anti icing lockout valve, and a fuel bypass valve.

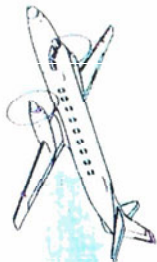
Fuel Control Unit

The Woodward fuel control unit hydromechanically meters the correct amount of fuel to the fuel nozzles.

Power lever position and speed (rpm) lever position are inputs to the fuel control unit and affect the fuel metered to the fuel nozzles. Altitude and ambient temperature compensation are a result of compressor inlet pressure and temperature (P_2T_2) sensing (Figure 7-10). Other inputs include engine rpm and compressor discharge pressure (P_3).

The fuel control unit contains an underspeed governor, an overspeed governor, and manual control of fuel metering.

The underspeed governor operates to prevent the rpm from going below a value set by the speed lever. If engine speed should decrease below the selected setting, the underspeed



TRAINING PURPOSES

Revision 4—September

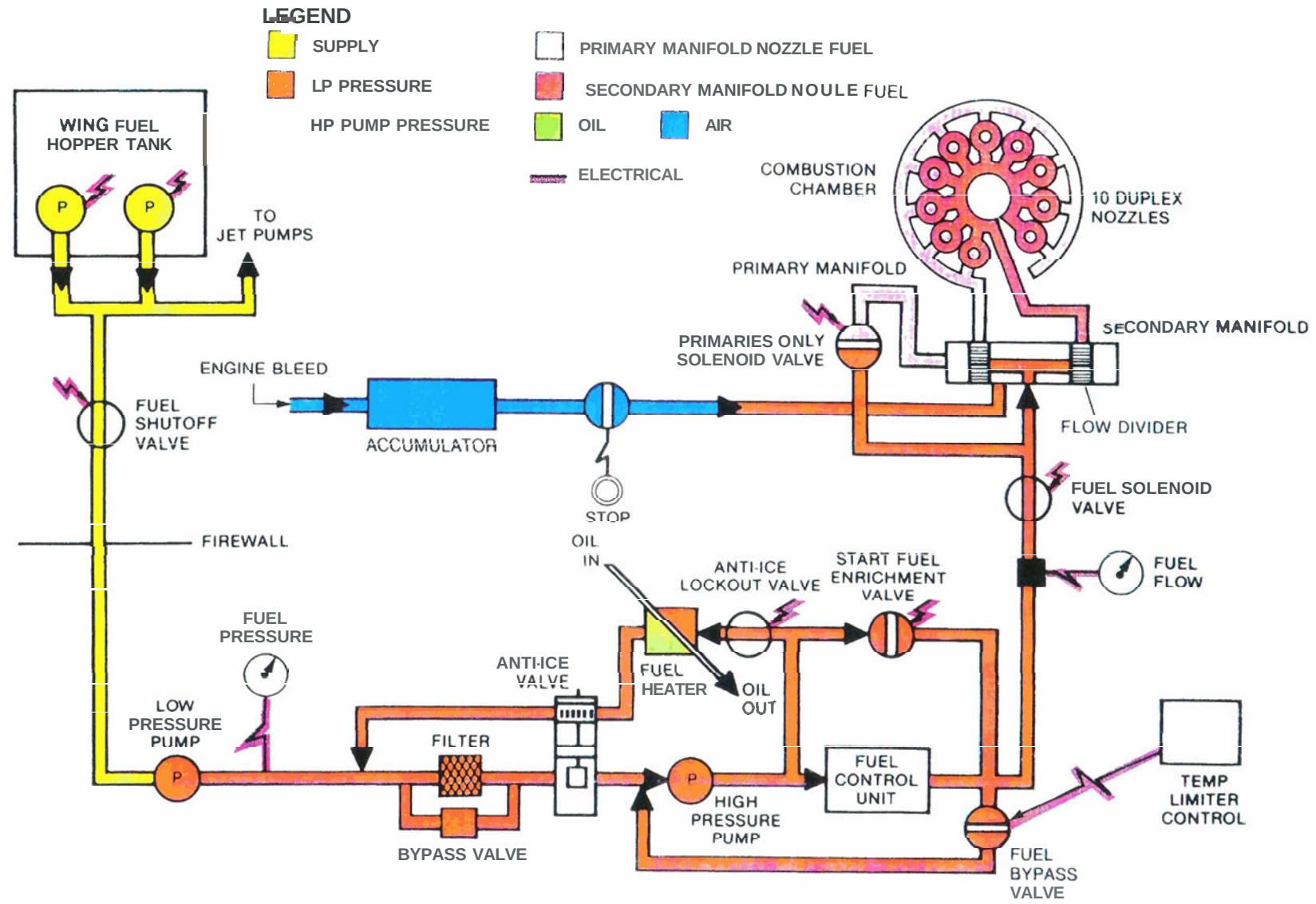


Figure 7-11. Fuel System Schematic



governor increases the fuel flow to oppose the speed decrease. The underspeed governor maintains engine rpm during Beta mode operation (ground operation). The underspeed governor can be set between 71 and 97% rpm by the speed lever.

Manual control of fuel metering by the power lever occurs when the power lever is moved forward of FLT IDLE, the point at which it meters more fuel than the underspeed governor.

The overspeed governor is a safety device to prevent excessive engine overspeed in the event of propeller malfunction. Excess engine rpm limits fuel metering to prevent additional rpm increase. The overspeed governor is adjusted by maintenance to operate from 103 to 105% rpm with the propeller at flat pitch (minimum load).

Fuel Solenoid Valve

The fuel solenoid valve is located in the metered fuel line between the fuel control unit and the flow divider (Figure 7-11). The fuel solenoid valve is electrically opened by the start system at 10% rpm and electrically closed by pressing the STOP button (Figure 7-12). The fuel solenoid valve can be mechanically closed by pulling the ENGINE STOP AND FEATHER control (Figure 7-13). Pulling this control first closes the fuel solenoid valve mechanically; it then feathers the propeller. If the STOP AND FEATHER control is pushed in, the fuel solenoid valve can be opened electrically by the start system. The fuel solenoid valve cannot be opened mechanically.

Flow Divider and Primaries-Only Solenoid Valve

The flow divider is a mechanical valve that distributes the fuel to the primary and secondary fuel manifolds (see Figure 7-11).

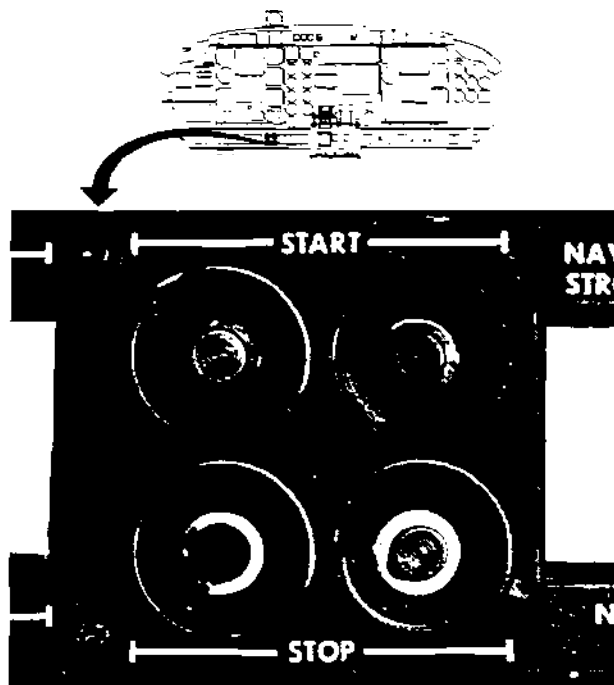


Figure 7-12. Engine START and STOP Buttons

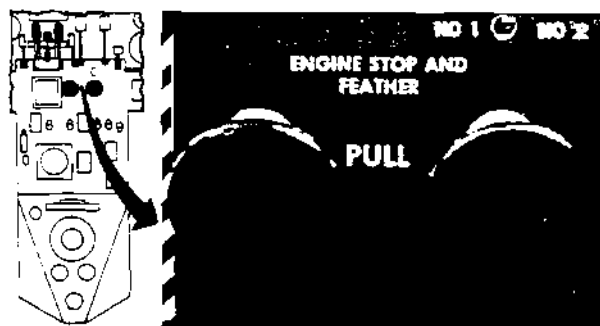
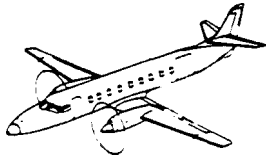


Figure 7-13. ENGINE STOP AND FEATHER Controls

To improve the fuel spray pattern in the combustion chamber during low airflow conditions, the primaries-only solenoid valve is opened during engine starts at 10% rpm to increase the proportion of fuel metered to the primary fuel manifold. The valve closes at 60% rpm.

Operation of the primaries-only solenoid valve is controlled by the SRL-autostart computer.



Start Fuel Enrichment Valve

The start fuel enrichment system provides for automatic **and/or** manual fuel enrichment to assist engine start acceleration.

Automatic fuel enrichment is controlled by the SRL autostart computer, which modulates the start fuel enrichment valve within the rpm range of 10 to 60% to maintain start EGT at approximately 690° C.

Manual fuel enrichment (when required) is controlled by the engine START button (Figure 7-12) between 10 and 60% rpm. The START button is normally released when the EGT rises. If the START button is pressed between 10 and 60%, rpm the start fuel enrichment valve opens fully and remains open as long as the START button is held. Manual fuel enrichment may be used as necessary to maintain start EGT at 650° C.

Fuel Anti-icing Valve

The fuel anti-icing valve is modulated to allow some engine-driven high-pressure pump output to be routed through the fuel heater; this maintains the temperature of the fuel above a temperature which might result in ice crystals clogging the fuel filter.

Anti-icing Lockout Valve

The anti-icing lockout valve is closed during engine start to prevent fuel from bypassing through the fuel heater. The valve closes as soon as the START button is pushed and opens at 60% rpm.

Fuel Bypass Valve

Automatic engine temperature limiting is provided by a fuel bypass valve controlled by the electronic temperature limiter (Figure 7-14). When rpm is above 90%, if EGT tends to exceed 650° C, the temperature limiter modulates the fuel bypass valve to bypass some of the metered fuel and limit the EGT to 650° C.

Fuel System Operation

Fuel is supplied from the hopper tank through the fuel shutoff valve to the engine-driven, low-pressure fuel pump (see Figure 7-11).

Fuel pressure is increased by the low-pressure pump and sensed by a transmitter that provides pressure indication in the cockpit. Low-pressure pump output fuel is supplied through a fuel filter and the fuel anti-icing valve to the engine-driven, high-pressure fuel pump. Fuel filter heat is controlled by the fuel anti-icing valve. High-pressure pump fuel is supplied to the fuel control unit which, in turn, meters the fuel being used by the engine. The fuel then goes through the fuel flow transducer to the fuel solenoid valve, the flow divider, and the primaries-only solenoid valve. The flow divider supplies fuel to the primary and secondary fuel nozzles.

During start, between 10 and 60% rpm, the fuel enrichment valve supplements the fuel metered by the fuel control unit, and the primaries-only solenoid valve supplements the fuel being delivered to the primary nozzles by the flow divider. Above 90% rpm, the fuel bypass valve is modulated by the temperature limiter to prevent the EGT from exceeding 650° C.

ENGINE TEMPERATURE-MONITORING SYSTEM

General

An EGT gage is provided for each engine for engine temperature monitoring (Figure 7-15). Below 80% rpm, compensated EGT is displayed on the EGT indicators. Above 80% rpm, a computed EGT is displayed.

The signal from the engine exhaust probes is first compensated for differences in the exhaust probe harness. The signal then goes to the SRL-autostart computer. Below 80% rpm, the signal proceeds without modification to the EGT gage. Above 80% rpm, the signal from the compensator is used as one of the inputs to the SRL computer.

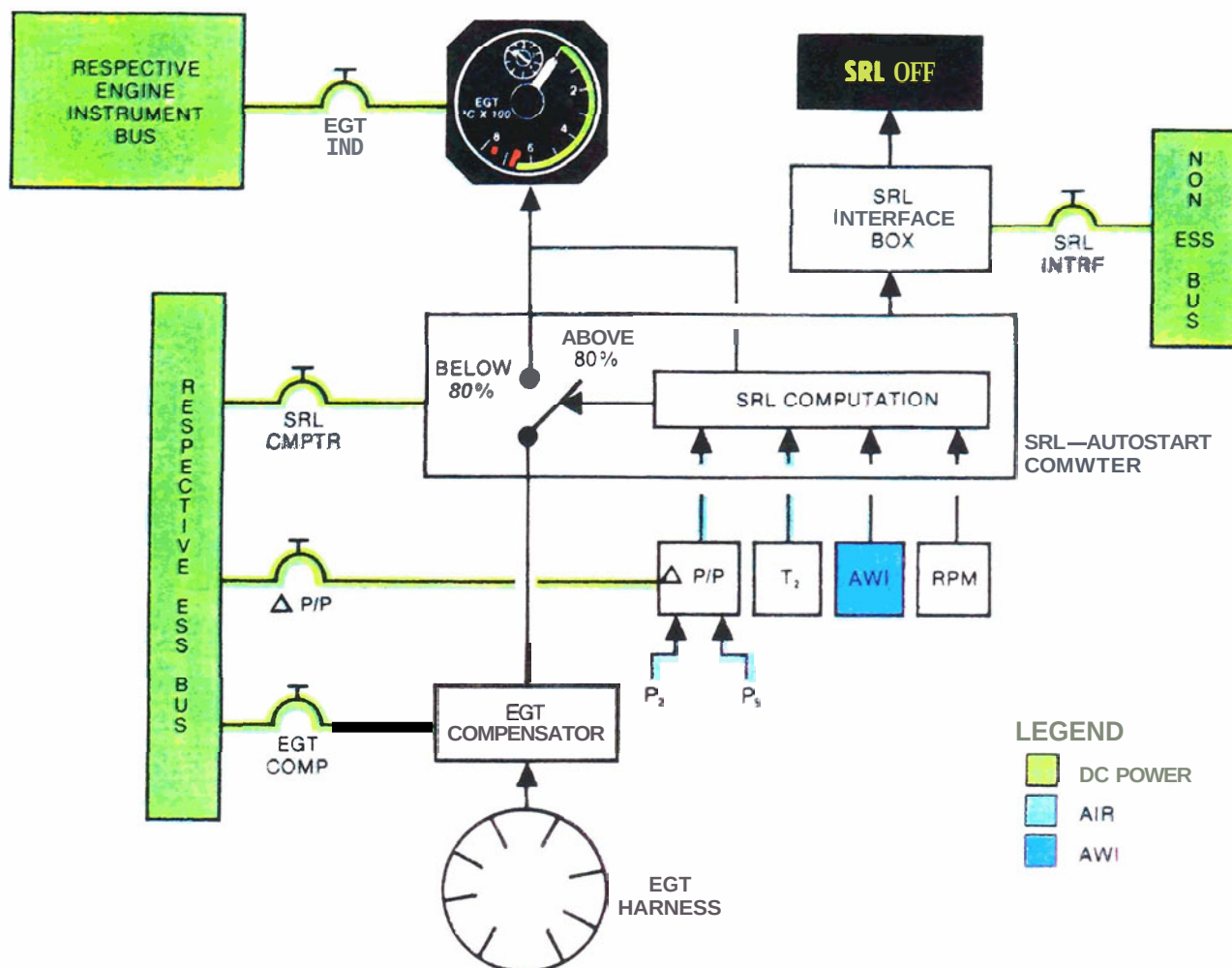


Figure 7-15. EGT Indicating Schematic

Maximum permissible exhaust gas temperature for takeoff and flight operations varies **with** outside air temperature, airspeed, altitude, engine rpm, and whether or not alcohol-water injection (AWI) is being used. Inputs to the SRL computation include compensated EGT from the EGT probes, the temperature of the **compressor** inlet air (T_2), engine rpm, altitude and airspeed in the form of a signal from the $\Delta P/P$ (P_1/P_5) transducer, and whether or not AWI is being used. The computed EGT shows the pilot how close the engine temperature is to the maximum **allowable**. The maximum permissible computed EGT is 650° C for all operations except starting.

Control

Display of compensated EGT for operations below 80% rpm and computed EGT for operations above 80% rpm is controlled automatically by the SRL-autostart computer. Two **SRL A P/P** power switches (Figure 7-5) are provided for testing the SRL/EGT computation.

Indication

Amber lights marked "L SRL OFF" and "R SRL **OFF**" (see Appendix B), located on the annunciator panel, advise the pilots when compensated EGT is displayed or when the SRL

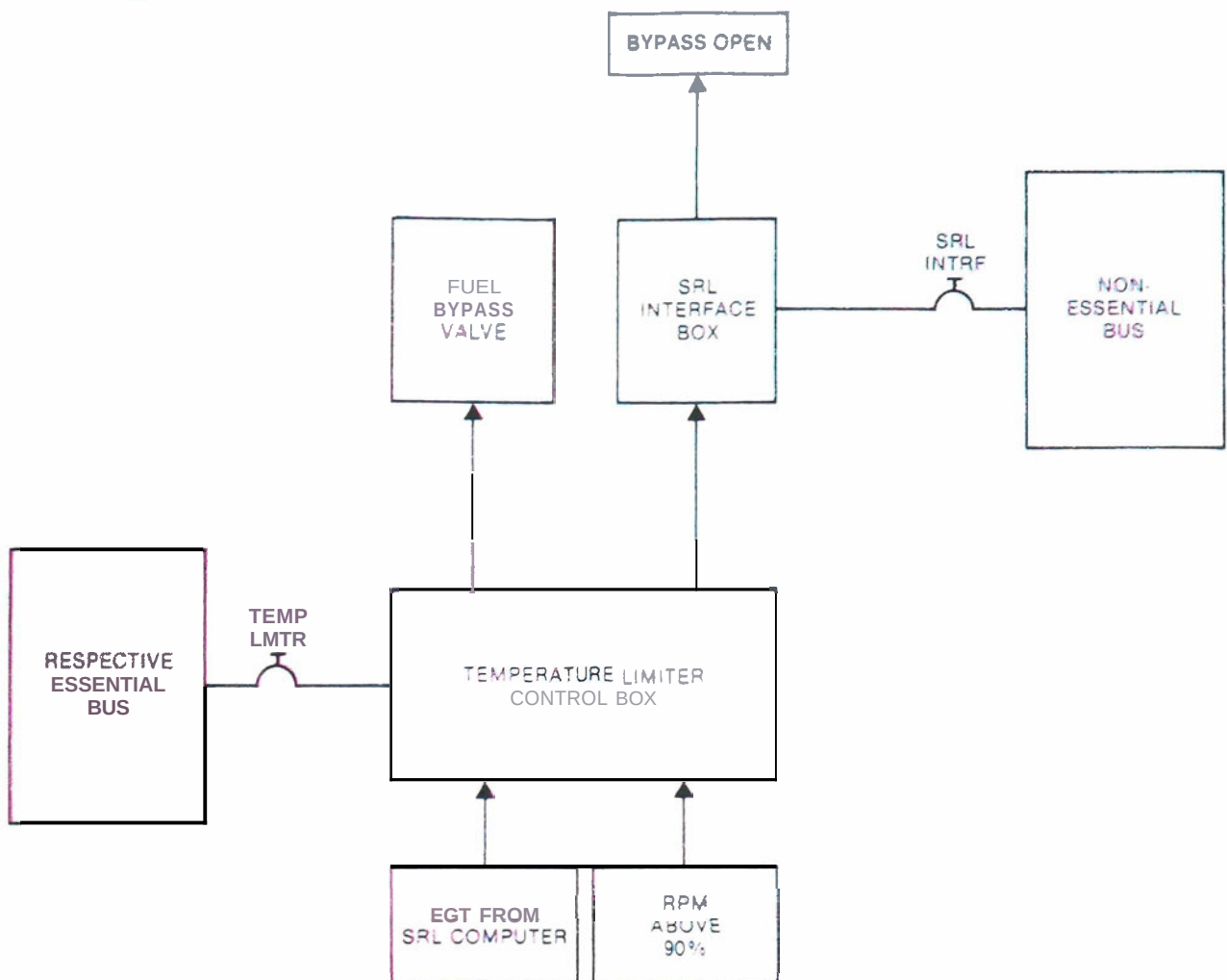


Figure 7-14. Temperature Limiter Schematic

computation is unreliable. The SRL OFF light is normally illuminated below 80% rpm; if an **SRL OFF** light is illuminated when rpm is over 808, a malfunction is indicated.

TEMPERATURE LIMITER

General

The system consists of a temperature limiter controller and a fuel bypass valve (Figure 7-14). The system functions automatically to limit EGT to 650° C by opening the fuel bypass valve and bypassing metered fuel back to the inlet of the high-pressure fuel pump (Figure 7-11).

Control

The temperature limiter is integrated with the SRL computer. It is armed whenever the SRL switch (Figure 7-5) is in the **NORM** position and rpm is above 90%.

Operation

The temperature limiter receives computed EGT inputs from the SRL computer. If rpm is 904% or greater and computed EGT tends to exceed 650° C, the temperature limiter signals the fuel bypass valve to modulate a bypass flow back to the fuel pump inlet. The reduction in metered fuel prevents EGT from exceeding the SRL value of 650° C.



Indication

Operation of the temperature limiter is indicated by illumination of the respective BYPASS OPEN light on the instrument panel (see Appendix B). The bypass valve may operate during takeoff. If the bypass valve opens during climb or cruise, the power lever should be retarded until the BYPASS OPEN light extinguishes.

A switch marked "TEMP LMTR TEST," "L" and "R" (Figure 7-16) is located on the pilot's left-forward console. It is used during the ground test of the SRL/temperature limiter system described in the Normal Procedures section of the AFM.

WARNING

Do not test the temperature limiter in flight. Flameout may result.



Figure 7-16. Temperature Limiter Test Switch

ENGINE TORQUE-MONITORING SYSTEM

The electronic torque-measuring system monitors the twisting force being applied to the propeller through the reduction gear. Strain gages in the reduction gearcase send an electronic signal to a torque signal conditioner, which processes the signal and drives the torque indicator (Figure 7-3) in the cockpit.

Two independent sets of strain gages are installed in the reduction gearcase. The second set is provided as a backup in case the first set malfunctions. Maintenance actions are required to switch from one set to the other.

IGNITION SYSTEM

General

The ignition system is a high-energy type, consisting of an engine-mounted ignition exciter and two igniter plugs located in the combustion chamber. An independent ignition system is provided for each engine.

Ignition operation is indicated by amber lights located below the EGT gages (Figure 7-3). The associated light is on whenever power is applied to the ignition exciter.

Ignition is controlled by two switches on the left console. On airplanes prior to SN 660, they are labeled "IGNITION MODE" (Figure 7-17). On airplanes SNs 660 and subsequent, they are labeled "AUTO/CONT IGNITION" (Figure 7-18).

Operation

Figure 7-19 shows the operation of the ignition system on aircraft prior to SN 660.

When the IGNITION MODE switch is in the NORMAl position, ignition is controlled during the autostart sequence by the SRL-autostart computer, which turns on the ignition at 10% rpm and turns it off at 60% rpm. The ignition is also operated during a manual start while the SPEED SW SELECT is in the MANUAL position.

When the switch is in the CONTInuous position, ignition operates while the airplane is on the ground. At lift-off, the ignition stops operating.

The ignition system operates continuously as long as the switch is in the OVERRIDE position.

Figure 7-20 shows the operation of the auto/continuous ignition system, which became standard on aircraft SNs 660 and subsequent.

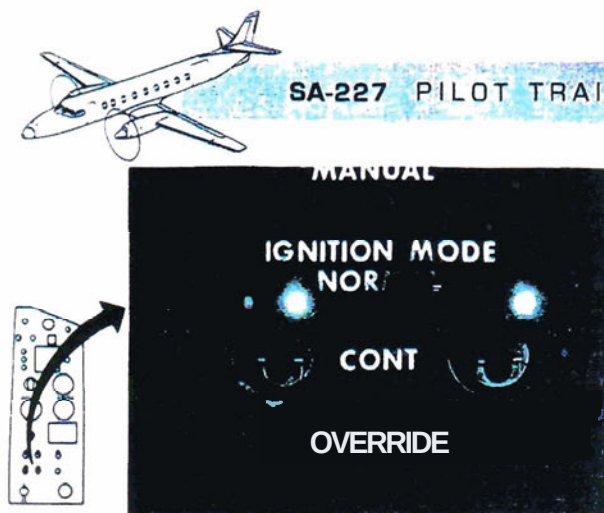


Figure 7-17. Ignition Mode Switches



Figure 7-18. Auto/Continuous Ignition Switches

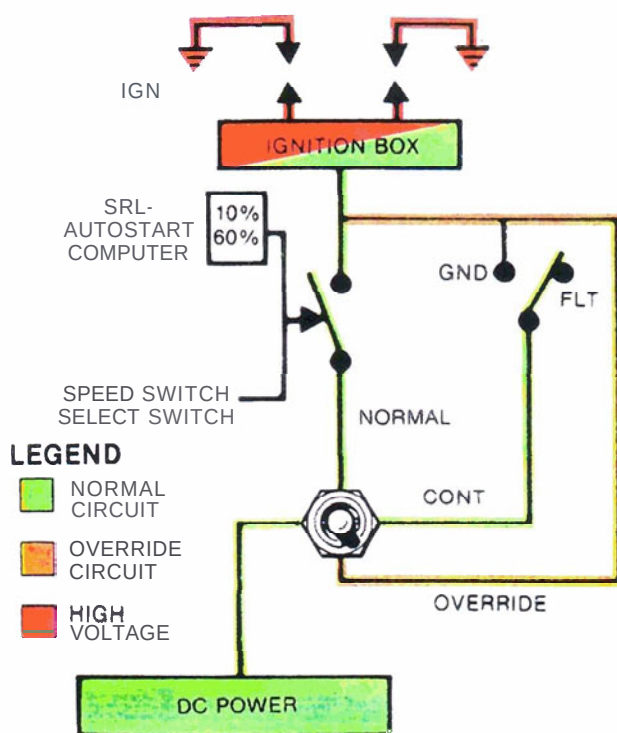


Figure 7-19. Ignition System Schematic

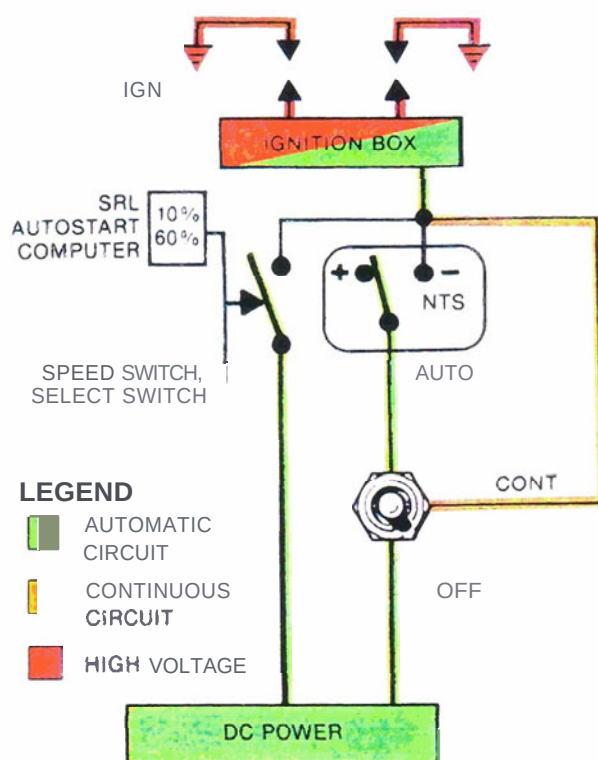
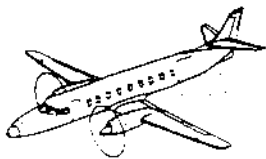


Figure 7-20. Auto/Continuous Ignition System Schematic



The AUTO position allows the SRL-autostart computer to control ignition during start, turning it on at 10% rpm and off at 60% rpm. An additional feature of the AUTO position energizes the ignition if the negative torque sensing (NTS) system is activated. Engine failure or flameout permits the propeller to drive the engine (negative torque), causing the ignition to be energized as long as the negative torque continues, and up to 30 seconds after the negative torque ceases.

Ignition operates continuously when the switch is in the CONTinuous position. The OFF position permits normal engine starting but lacks the NTS feature. It is not normally used.

PROPELLER

GENERAL

The four-blade, constant-speed propellers are manufactured by Dowty-Rotol, Limited or McCauley Accessory Division. All Metro III airplanes SNs 634 and subsequent have McCauley propellers as standard equipment with Dowty-Rotol propellers as an option. The propellers incorporate full feather and reverse capabilities, in addition to Beta mode control for ground operation.

Propeller start locks are provided to maintain minimum blade angle to reduce propeller drag during engine start.

While the engine is running, unfeathering and reversing functions are accomplished by using propeller governor oil pump pressure. Feathering and pitch change toward a more positive blade angle is accomplished by springs in the propeller dome, assisted by counterweights.

An unfeather pump is provided for airstarts. It is also used, when necessary, to place the propeller on the start locks for ground start.

PROPELLER MODES

The propellers operate in two modes: the Beta, or ground mode and the propeller governing mode.

In the Beta mode, blade angles are hydromechanically selected by the pilot to facilitate ground handling of the airplane. The Beta mode must never be selected in flight. Operation in the Beta mode is indicated by amber lights labeled "L BETA" and "R BETA" on the annunciator panel (see Appendix B).

The propeller governing mode is the constant-speed mode. In this mode (at a constant power setting), the blade angle automatically varies (as a function of true airspeed) to maintain any selected rpm.

CONTROL

The propeller is normally controlled by the interaction of (1) a power lever, (2) a speed lever, and (3) the propeller STOP AND FEATHER control, which is used only under abnormal or emergency conditions. Operation of these levers is explained in detail later in this chapter.

A propeller is **onspeed** when the actual rpm equals the selected rpm. Underspeed exists when the actual rpm is less than the selected rpm. Overspeed exists when the actual rpm is greater than the selected rpm. Figure 7-21, Sheets 1 through 3, shows propeller **onspeed**, **underspeed**, and **overspeed** operation. In turboprop engines, rpm is a sole function of blade angle; and in flight at a constant power setting, blade angle is a sole function of true airspeed.

In the propeller governing mode of operation, the propeller is controlled by a mechanical governor. The governor operates on the principle of balancing two opposing forces: a speeder spring force and a flyweight force.

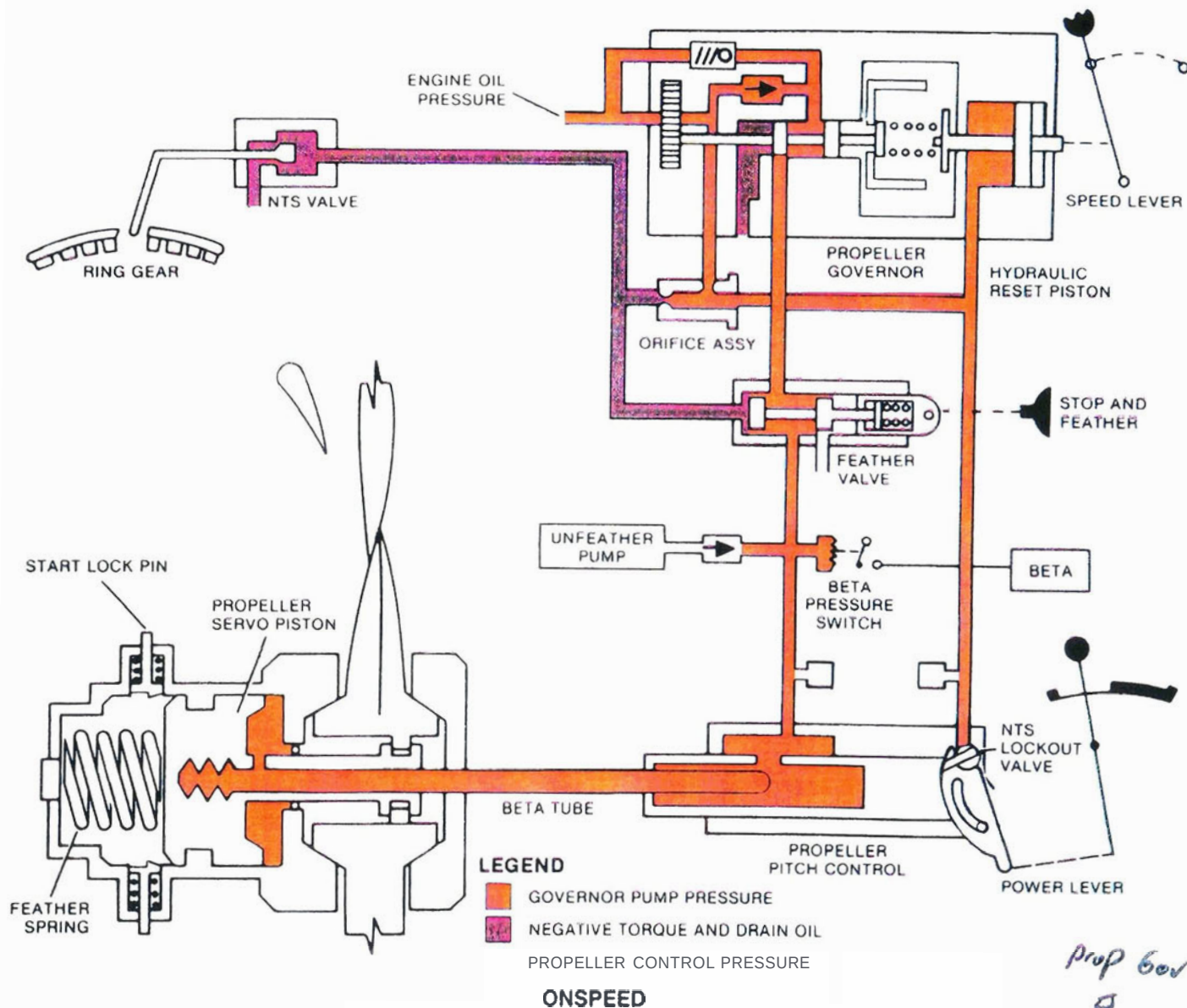


Figure 7-21. Propeller Operation, Sheet 1

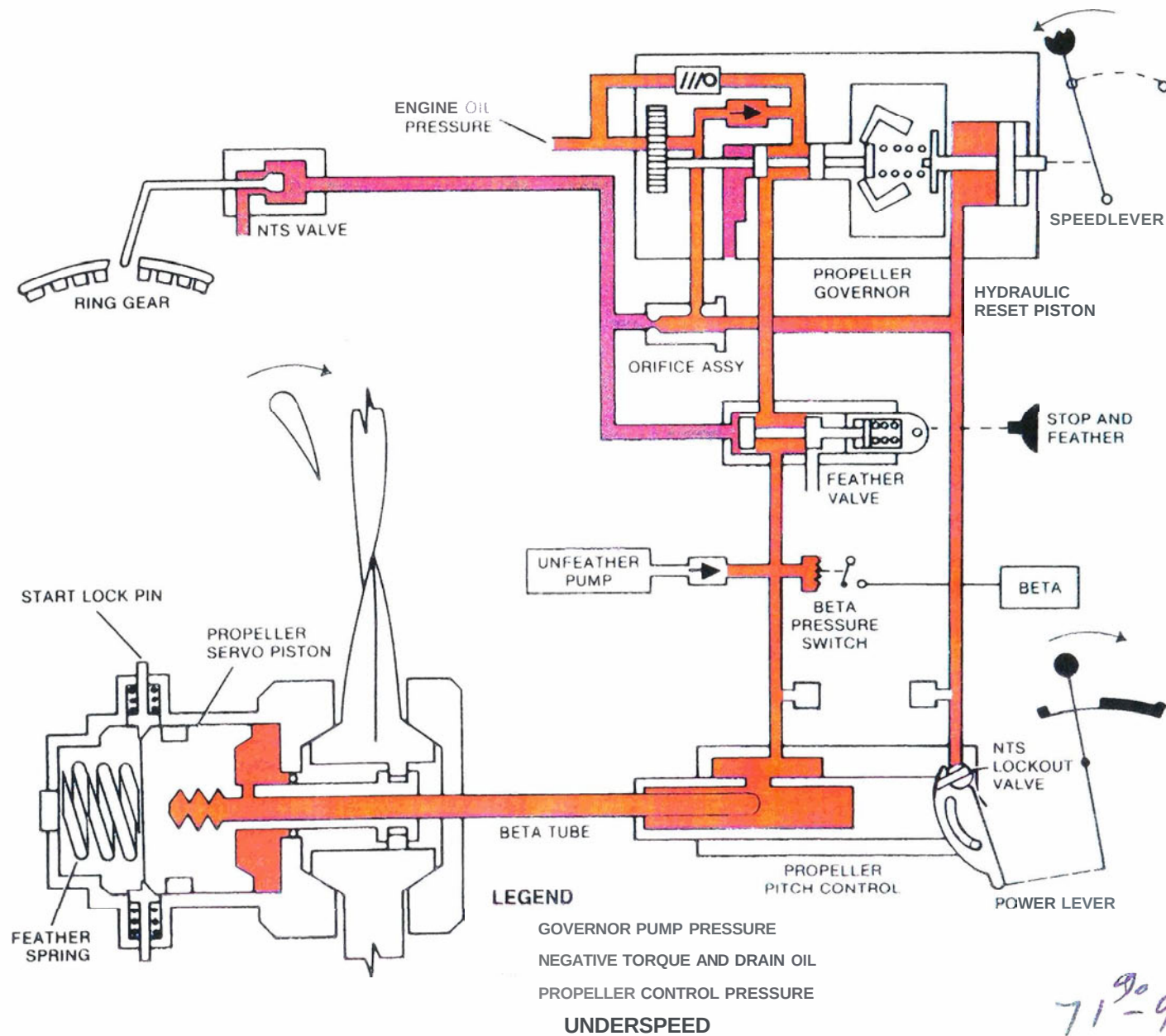


Figure 7-21. Propeller Operation, Sheet 2



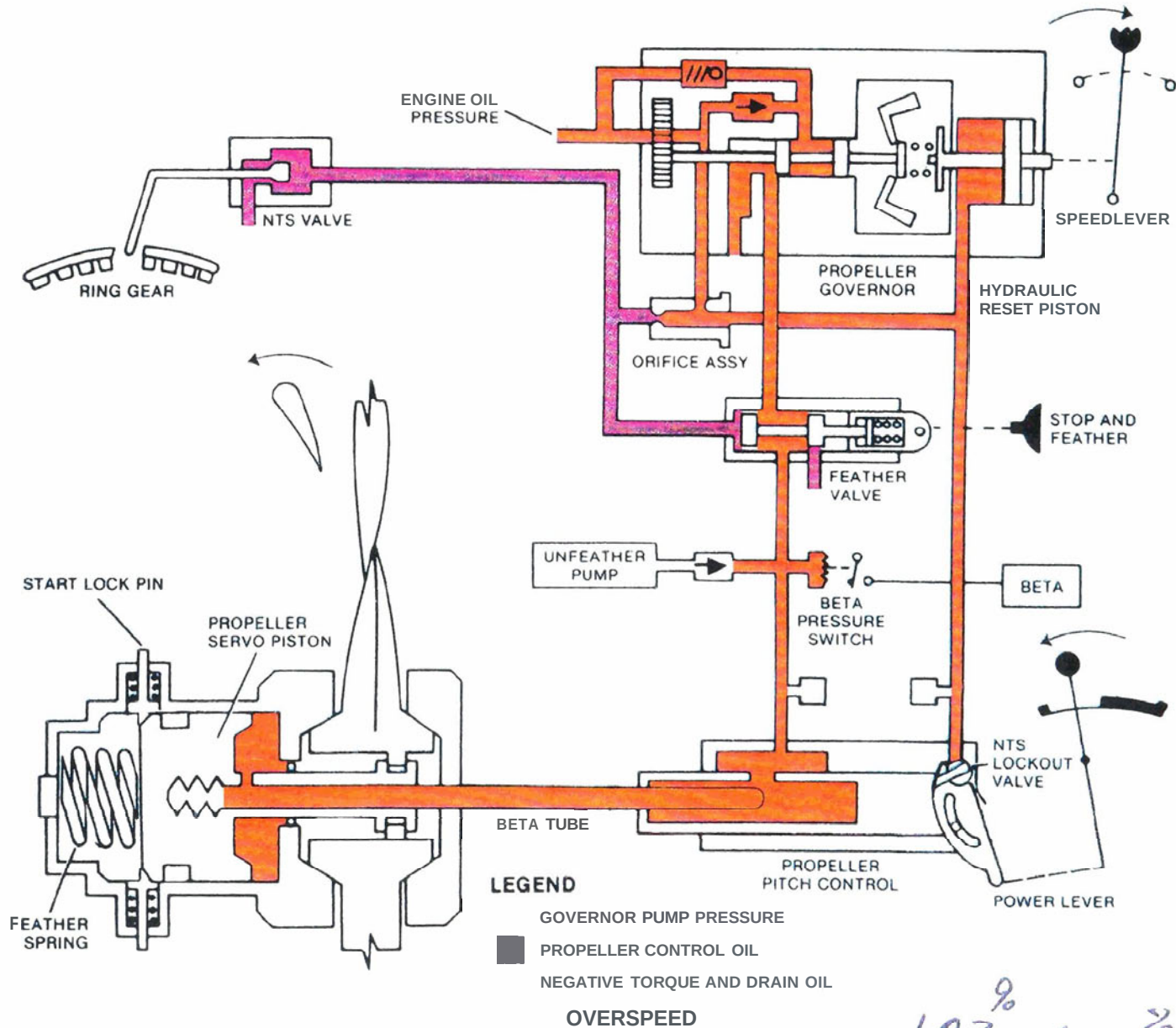
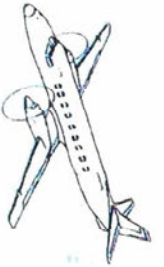


Figure 7-21. Propeller Operation, Sheet 3



The speeder spring force is a function of, and varies according to, speed lever position. The flyweight force is a function of, and varies with, engine rpm.

If the speeder spring force exceeds the flyweight force, the result is underspeed. If the flyweight force exceeds the speeder spring force, the result is overspeed. When these two forces balance, the propeller is onspeed.

Any imbalance between the speeder spring and the flyweight force results in the repositioning of a pilot valve. For example, if an underspeed exists, (Figure 7-21, Sheet 2) the pilot valve is positioned to direct governor pump pressure through a Beta tube to the propeller piston to reduce the angle of the blades and permit the engine to accelerate to the selected onspeed condition determined by the balance of the speeder spring and flyweight forces (Figure 7-21, Sheet 1).

Conversely, if the engine accelerates above the selected rpm (Figure 7-21, Sheet 3), the flyweight force increases and positions the pilot valve to shut off governor pump pressure from the propeller piston and, at the same time, directs the oil to the reduction gearcase sump. Then the propeller counterweights and springs increase the blade angle, reducing rpm until the speeder spring and flyweight forces are again in balance and onspeed results (Figure 7-21, Sheet 1). This is a continuous process in any constant-speed propeller because changes in airspeed, altitude, attitude, temperature, and power lever or speed lever position must result in a change of blade angle to maintain any selected rpm.

INDICATION

Moving the power lever behind FLT IDLE causes an underspeed condition of the propeller governor, which directs governor oil pressure to the propeller pitch control. This pressure closes a pressure switch, turning on a light marked "L BETA" or "R BETA" (see Appendix B) when the pressure is adequate for full reverse operation.

FEATHERING

The feathered angle of the propeller is the angle that produces minimum drag. When oil pressure is drained from the propeller servo piston, the propeller is feathered by a spring and counterweight forces.

Pulling the ENGINE STOP AND FEATHER control (Figure 7-13) shuts off fuel and drains propeller oil, in that sequence, and the propeller feathers and stops. In case of engine failure in flight, operation of the negative torque sensing (NTS) system moves the propeller blade angle toward feather to minimize the drag of the windmilling propeller.

UNFEATHERING

An electrohydraulic unfeather pump, supplied with oil from the low point of the engine oil tank, is used to unfeather the propeller in flight to produce windmilling of the engine for airdrops.

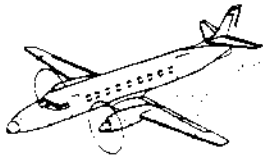
The unfeathering pump can also be used on the ground to unfeather the propeller and decrease blade angle all the way to full reverse when the engine is not running.

The unfeather pump is actuated for airdrops by pressing the engine START button when weight is off the gear squat switch.

The unfeather pump may also be operated by an UNFEATHER TEST switch (Figure 7-22)



Figure 7-22. UNFEATHER TEST Switch



with two positions, L and R. Holding this switch in the L or R position causes the selected left or right unfeather pump to run until the switch is released.

START LOCKS

The propeller start locks are used to maintain a blade angle that will produce the minimum resistance to start acceleration. The propeller must always be on the start locks prior to attempting a ground start.

The propeller start locks consist of two spring-loaded pins that engage the propeller piston and, consequently, lock the blades in a flat pitch position. The start locks remain engaged following engine start because of the shear loads applied to the pins by the propeller piston. The start locks are released when ready to taxi by selecting REVERSE with the power lever to remove the shear loads. The centrifugal forces produced by the rotating propeller overcome the spring load and disengage the pins from the piston.

During a normal shutdown, the stop button is first pushed to shut off fuel; then the power lever is moved into REVERSE as the rpm decays below 50%. As the propeller decelerates, the spring pressure of the pins overcomes centrifugal force and re-engages the start locks. After the rpm decays below 10%, the power lever may be released.

NEGATIVE TORQUE-SENSING (NTS) SYSTEM

General

Negative torque occurs when the propeller drives the engine, as opposed to the positive torque developed when the engine is driving the propeller. Loss of engine power during flight results in loss of positive torque, and the windmilling propeller produces negative torque, which results in drag that decreases performance and increases yaw.

The engine has a negative torque-sensing (NTS) system that provides for automatic drag reduction without any action on the pilot's part. It is not an autofeather system, but rather a system that increases blade angle to reduce windmilling drag. The pilot must still feather the propeller. A small amount of negative torque is unavoidable; otherwise, it would not be possible to windmill the engine for an airstart.

More negative torque than required for airstart causes closing of an NTS valve that is normally open to drain oil pressure to the gear case. When the valve is closed by sensing negative torque, oil pressure builds up and moves the feathering valve, dumping oil from the propeller servo piston, which allows the spring and counterweight to increase blade angle. The increasing blade angle reduces negative torque, and the NTS valve opens again, resulting in the feather valve moving back to normal under the influence of its spring. This condition, usually called NTSing, is repetitive until the pilot feathers the propeller.

CAUTION

As NTSing will rapidly reduce engine rpm, the pilot must not allow the engine to windmill in the critical vibration range of 18 to 28%. Garrett recommends that the propeller be feathered when the rpm decays to 30%, if not sooner.

NTS Lockout and Propeller Governor Reset

An NTS lockout system prevents windmilling propeller drag from being sensed as an engine failure during landing rollout. This, of course, would cause an increase in blade angle and reduce drag at a time when the pilot requires the opposite. When the pilot moves the power lever aft of FLT IDLE, the NTS lockout valve in the propeller pitch control is mechanically opened and drains oil from the



propeller governor reset piston, which resets the propeller governor 5% higher. The oil line that is drained is common to the NTS valve, and, consequently, NTS oil pressure cannot build up to operate the feather valve. The resetting of the propeller governor ensures that it will sense an underspeed and supply maximum oil pressure to the propeller pitch control, resulting in power lever control of propeller blade angle.

NTS Test

The WIS system must be checked prior to the first flight each day. To test the system, select low rpm with the speed lever, and advance the power lever slowly until the rpm stabilizes. Stabilized rpm should not exceed 94.5%. If it does, it indicates that either the propeller governor low setting is misrigged or the oil supply to the NTS system is malfunctioning. A maintenance check of the NTS system is required at regular intervals.

SYNCHROPHASING

After takeoff and when climb power is established, a propeller synchrophaser can be used to synchronize the propeller rpm and establish a blade phase relationship (called synchrophasing). Combined synchronizing and synchrophasing help reduce propeller noise and result in greater passenger comfort.

The system consists of a biasing coil on each propeller governor and a control box that receives signals from each propeller representing rpm. The system operates to match the rpm of the slow engine to that of the fast engine over a very limited range. It is recommended that the engines be manually synchronized before turning the system on.

The synchrophasing system is controlled by a two-position switch (Figure 7-23) marked "TAKEOFF & LANDING" (off) and "CLIMB & CRUISE" (on).

When the synchrophaser system is turned on after takeoff (Figure 7-24), the magnetic sen-

sors for each propeller system transmit inputs (representing rpm) to the sync controller. The signals are analyzed by the controller, and an output is sent to a biasing coil on the governor of the slow engine. This changes the governor setting until its rpm precisely matches that of the other engine. In addition, the phase relationship between the left and right propeller blades is adjusted so that noise level is kept to a minimum. The synchrophaser system does not affect normal governor operation.

If an engine fails or is feathered with the sync system on, the rpm loss on the operating en-

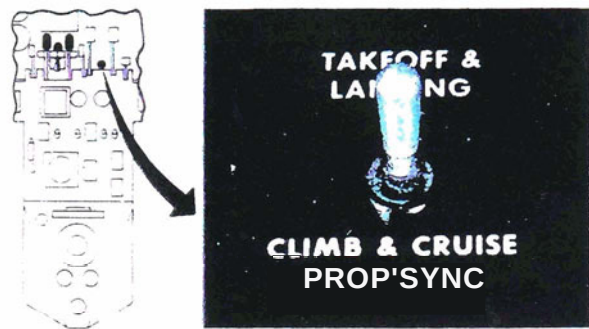


Figure 7-23. Propeller Synchrophaser Switch

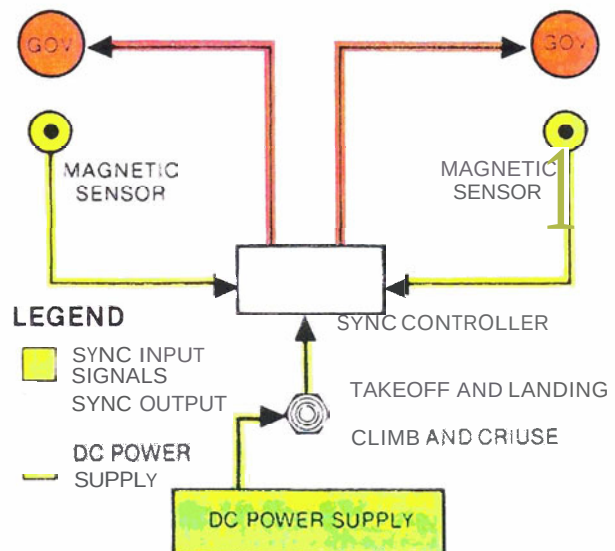


Figure 7-24. Syncrophaser System Schematic



gine is very limited, and under no conditions could rpm drop below the governor setting of the speed lever.

POWERPLANT CONTROL

GENERAL

The powerplant is controlled by the interaction of the power levers and the speed (rpm) levers (Figure 7-25).

POWER LEVER

Two power levers operate in a quadrant on the center pedestal which is marked "FLT IDLE," "GROUND IDLE," and "REVERSE."

The power lever connects to the propeller pitch control and the manual fuel valve of the fuel control unit.

When the power lever is between FLT IDLE and REVERSE, any movement positions the propeller pitch control to provide a blade angle proportionate to power lever movement without affecting the position of the manual fuel valve in the fuel control unit.

When the power lever is positioned forward of FLT IDLE, it controls fuel flow by operating the manual fuel valve in the fuel control unit. The power lever can move freely between HIGH (full forward) and FLT IDLE positions. The power levers must be lifted over a detent before they can be moved aft of FLT IDLE.

SPEED (RPM) LEVER

Two speed levers operate in a quadrant on the center pedestal marked "RPM HIGH"

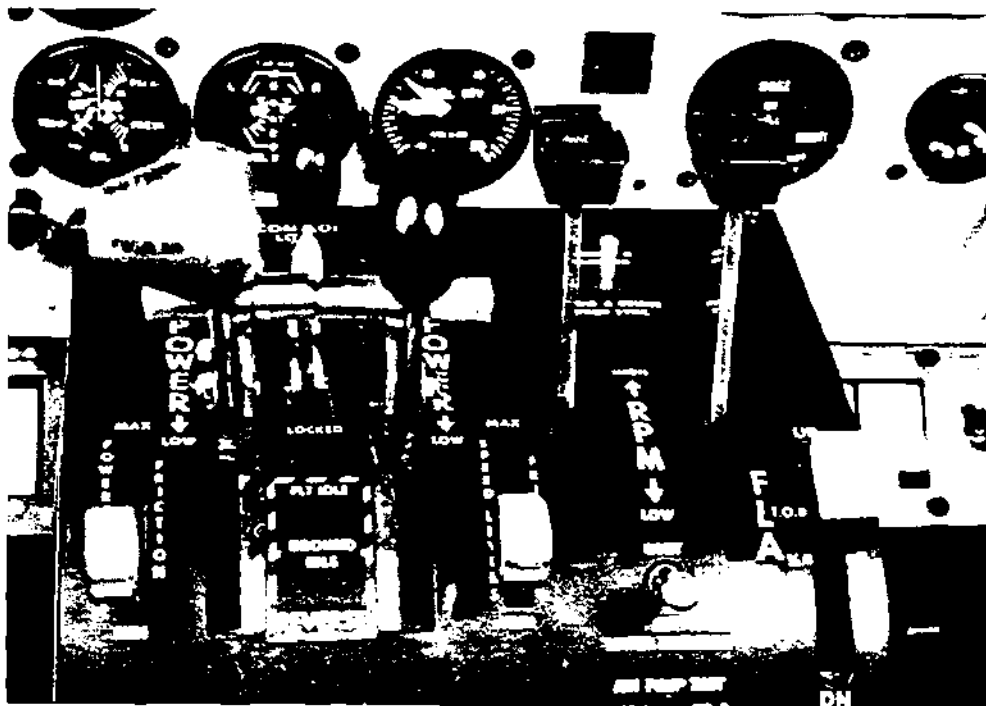
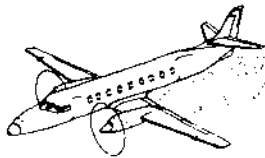


Figure 7-25. Powerplant Control



(forward) and "LOW" (aft). The speed lever connects to the propeller governor and to the underspeed governor in the fuel control unit.

FRICTION LEVERS

Two friction control levers are also located on the pedestal. One lever adjusts the friction for both power levers, and the other lever controls the speed lever friction.

STOP AND FEATHER CONTROL

A two-position push-pull knob (Figure 7-13) for each engine, marked "L" or "R ENGINE STOP AND FEATHER," is mechanically connected to the fuel solenoid valve and the propeller feather valve. When the knob is pulled out, the fuel solenoid valve will fully close before the feather valve begins to open. This is important because combustion must be terminated before the feathering propeller reduces rpm, and, consequently, airflow through the engine. When the feather valve is open (feather position), oil is dumped from the propeller servo piston, and the springs and counterweights rapidly feather the propeller blades. Pushing the ENGINE STOP AND FEATHER control in allows the spring-loaded feather valve to assume its normal position, but the fuel solenoid valve remains closed. The only means of opening the fuel solenoid valve is electrically, during start.

OPERATION

There are two modes of powerplant operation: propeller governing mode and Beta mode. The position of the power lever determines the mode of operation. When the power lever is forward of FLT IDLE, the powerplant is operating in propeller governing mode. When the power lever is aft of FLT IDLE, the powerplant is operating in Beta mode.

The power lever determines the direction and value of the power developed by the propeller. The speed lever determines the operating rpm.

When the power lever is forward of GROUND IDLE, the propeller is developing forward thrust. When the power lever is aft of GROUND IDLE, the propeller is developing reverse thrust.

In propeller governing range (forward of FLT IDLE) the power lever controls engine power by controlling the manual fuel valve in the fuel control unit. The propeller governor maintains engine rpm by adjusting the propeller blade angle. The propeller governor rpm is set by the speed lever.

In Beta mode, the power lever controls the propeller blade angle: forward thrust from GROUND IDLE forward and reverse thrust from GROUND IDLE aft. The underspeed governor in the fuel control unit maintains engine rpm by providing fuel as required by the propeller load. The underspeed governor rpm is set by the speed lever.

Movement of the speed lever between high and low rpm adjusts either the propeller governor or the underspeed fuel governor. The position of the power lever determines which governor is active. When the power lever is forward of FLT IDLE (propeller governing mode), the propeller governor is controlling rpm. When the power lever is aft of FLT IDLE (beta mode), the underspeed governor is controlling rpm. The propeller governor is set to control rpm from 94 to 100%. The underspeed governor is set to control rpm from 71 to 97%.

When the speed lever is moved to HIGH prior to takeoff, the engine rpm should increase to 96–97%, controlled by the underspeed governor. As the power lever moves forward of FLT IDLE, the manual fuel valve function of the fuel control meters more fuel, and the rpm increases to the propeller governor setting of 100%.



During flight, propeller governor rpm should be set at either 97 to 100% with the speed lever. Normally, 100% rpm is used for climb, and 97% is used for cruise.

Prior to landing, the speed lever is moved forward, setting the propeller governor at 100% and the underspeed governor at 97%. It is important that the speed lever be set at HIGH, not only to have maximum rpm available in case of a rejected landing, but also to set the underspeed governor to maximum rpm to ensure proper Beta mode operation during the landing roll.

WARNING

In the event there is an indication of improper operation of a fuel or propeller control, it is recommended that the affected engine be shut down and a single-engine landing accomplished.

After touchdown, as the power lever is moved behind FLT IDLE into the Beta range, the power lever is controlling propeller blade angle and load, and the underspeed governor is metering fuel to maintain rpm.

NOTE

Check that both BETA lights are illuminated before moving the power levers into the REVERSE range during landing roll.

CAUTION

Do not use full reverse above 90 knots. Reduce airspeed 1 knot for each 1° F above 90° F prior to using maximum reverse power.

CAUTION

Attempted reverse with the speed levers aft of the HIGH rpm position may result in an engine overtemperature condition.

After landing, do not retard speed levers while power levers are aft of GROUND IDLE. Severe engine damage may result.

ENGINE STARTING

GENERAL

Ground starts and airstarts differ only slightly. The rotational force for ground starts is provided by the starter-generator on the accessory gearbox. Rotational force for airstarts is provided by unfeathering the propeller and allowing airflow to windmill the engine.

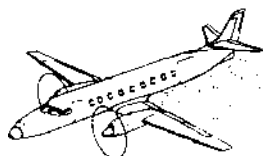
GROUND STARTS

General

Starts may be automatic or manual.

If possible, a ground power unit (GPU) should be used especially at low temperatures. If used, the GPU should be set to provide no more than 1,000 amps.

If a battery start is accomplished, normally the right engine is started first, and then the right generator is used to recharge the batteries and assist in a cross-generator start of the left engine.



Automatic Ground Start

Set the speed lever at LOW and the power lever at FLT IDLE.

NOTE

When setting the power lever for the start, move it fully forward to remove slack in the linkage and then back to FLT IDLE.

Select the START MODE switch (Figure 7-26) as required (SERIES or PARALLEL).

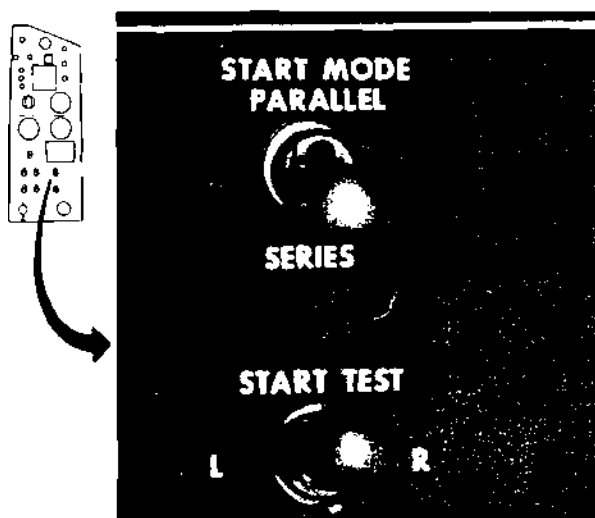


Figure 7-26. START MODE and START TEST Switches

For the first battery start of the day or subsequent battery starts with the oil temperature near ambient temperature, it is recommended that the START MODE switch be in SERIES so that the batteries are placed in series at 10% rpm for increased starting voltage.

NOTE

It is recommended that the right engine be started first.

For GPU or cross-generator starts, the START MODE switch should be placed in PARALLEL. Check to see that the propellers are clear and on the start locks. Press and hold the START button, and verify that rotation takes place.

At 10% rpm, the automatic start sequence should open the fuel solenoid valve and power the ignition unit. Check that fuel flow is indicated and the ignition light is illuminated. When the EGT rises, release the START button. Continue monitoring the EGT to see that the automatic fuel enrichment increases the EGT to 690° C and holds it there. The maximum allowable EGT is 770° C for one second. The engine should accelerate smoothly. As the rpm passes 60%, the ignition light should extinguish, signaling that the automatic start sequence is completed.

As the rpm passes 71%, move the power lever back toward GROUND IDLE. If overspeed governor, SRL, or temperature limiter checks are required, avoid moving the power lever below 75% rpm to prevent inadvertent release of the start locks.

CAUTION

If EGT rise is not obtained within ten seconds after reaching 10% rpm or prior to reaching 20% rpm, abort the start.

CAUTION

If rpm stops increasing and EGT is above 700° C and rising rapidly, immediately abort the start.

NOTE

During a battery start, the voltage drop following starter engagement may cause erratic torque indications.



If acceleration is sluggish between 10 and 60%, pressing and holding the START button provides manual fuel enrichment and assists acceleration. Monitor EGT carefully, and release the start button if the EGT reaches 650° C.

CAUTION

If a GPU is used, the BATTERY switches should be turned off until the GPU is plugged in and started, and its output voltage checked. At least one BATTERY switch must be ON to utilize GPU power.

CAUTION

Do not turn on either engine generator with the GPU plugged in. Keep the engine generators off while the GPU is plugged in and operating.

Manual Ground Start

If no fuel flow or ignition is observed at 10% rpm, and there is no EGT rise, it is possible that the automatic start sequence failed. In this case, it is permissible to attempt a manual start as described in the Abnormal Procedures section of the *AFM*.

The SPEED SW SELECT switch (Figure 7-6) is placed in the OFF position prior to a manual start. While pressing the START button, the SPEED SW SELECT switch is moved to MANUAL as the rpm reaches 10%. This action opens the fuel solenoid valve, powers the ignition unit, and initiates the other events that normally occur at 10%.

If the manual start is successful, return the SPEED SW SELECT switch to AUTO after the rpm stabilizes at 71%.

All the notes and cautions of an automatic ground start apply to a manual ground start. After accomplishing a manual ground start, the SRL/temperature limiter checks in the Normal Procedures section of the *AFM* must be accomplished prior to flight.

Residual Heat Starts

The START TEST switch (Figure 7-26) should be used to cool an engine prior to a start if EGT is above 200° C. Hold the START TEST switch to the desired L or R position up to 15% rpm. Then momentarily push and release the START button and monitor the start as previously described.

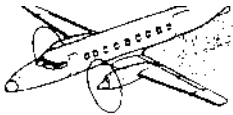
Before Start Unfeathering

An engine start must not be attempted if the propeller is not on the start locks. A feathered propeller must be unfeathered and placed on the start locks prior to starting. While holding the affected power lever in full REVERSE, move the UNFEATHER TEST switch (Figure 7-22) to the appropriate L or R position to operate the unfeathering pump. Keep the pump operating until the propeller reaches full reverse; then release the UNFEATHER TEST switch, move the power lever full forward of FLT IDLE, and then position it for the start.

ABORTED STARTS

To abort an engine start, push the STOP button (Figure 7-12) and pull the ENGINE STOP AND FEATHER control (Figure 7-13).

Following an aborted start, the engine should be cleared prior to attempting another start. Clearing an engine means cranking without fuel and ignition. This is accomplished by holding the START TEST switch (Figure 7-26) to the appropriate L or R position for ten seconds or 15% rpm, whichever occurs first.

**CAUTION**

Do not operate in the 18 to 28% rpm range during engine clearing.

AIRSTARTS

Airstarts are the same as ground starts except for power lever and speed lever positions and the time allowed observe an EGT rise. Airstart procedures are detailed in the Abnormal Procedures section of the *AFM*.

CAUTION

If an engine has been shut down because of an obvious failure, as indicated by the engine instruments or excessive vibration, an airstart should not be attempted.

The power lever is placed $\frac{1}{4}$ inch forward of FLT IDLE and the speed lever at a position corresponding to 97% propeller governor rpm. If the EGT does not rise by 25% rpm, abort the start by pulling the ENGINE STOP AND FEATHER control.

ENGINE SHUTDOWN**NORMAL SHUTDOWN**

A fuel purging system (Figure 7-27) is installed to purge residual fuel in the flow divider and fuel manifolds into the combustion chamber during a normal engine shutdown.

The air accumulator is charged by engine bleed air when engine rpm increases to approximately 96%. During shutdown, pushing the STOP button opens the purge valve, and the residual fuel under accumulator air pressure enters the combustor and burns.

To accomplish a normal shutdown, press the engine STOP button for a minimum of five seconds to ensure complete fuel purging. The rpm will increase about 5% with actuation of the stop circuit as the fuel is purged into the combustor, and then the rpm will decay as the manifold fuel is depleted. Hold the power lever in reverse from 50 to 10% rpm to place the propeller on the start locks.

NOTE

Observe a 3-minute engine cooldown period before stopping engines.

PREPLANNED SHUTDOWN IN FLIGHT

When an engine is to be shut down in flight for training, or following maintenance or adjustments, or for any reason external to the engine, the Preplanned Engine Shut Down in Flight from the Abnormal Procedures section of the approved *AFM* should be used. Stresses due to temperature gradients will be reduced and engine life prolonged.

In the event of an actual engine failure or engine fire, implement the appropriate emergency procedure as stated in the Emergency Procedures section of the *AFM*.

EMERGENCY SHUTDOWN IN FLIGHT

If an emergency engine shutdown is necessary, pull the ENGINE STOP AND FEATHER control. Perform any other appropriate memory actions, and then refer to the appropriate emergency checklist.

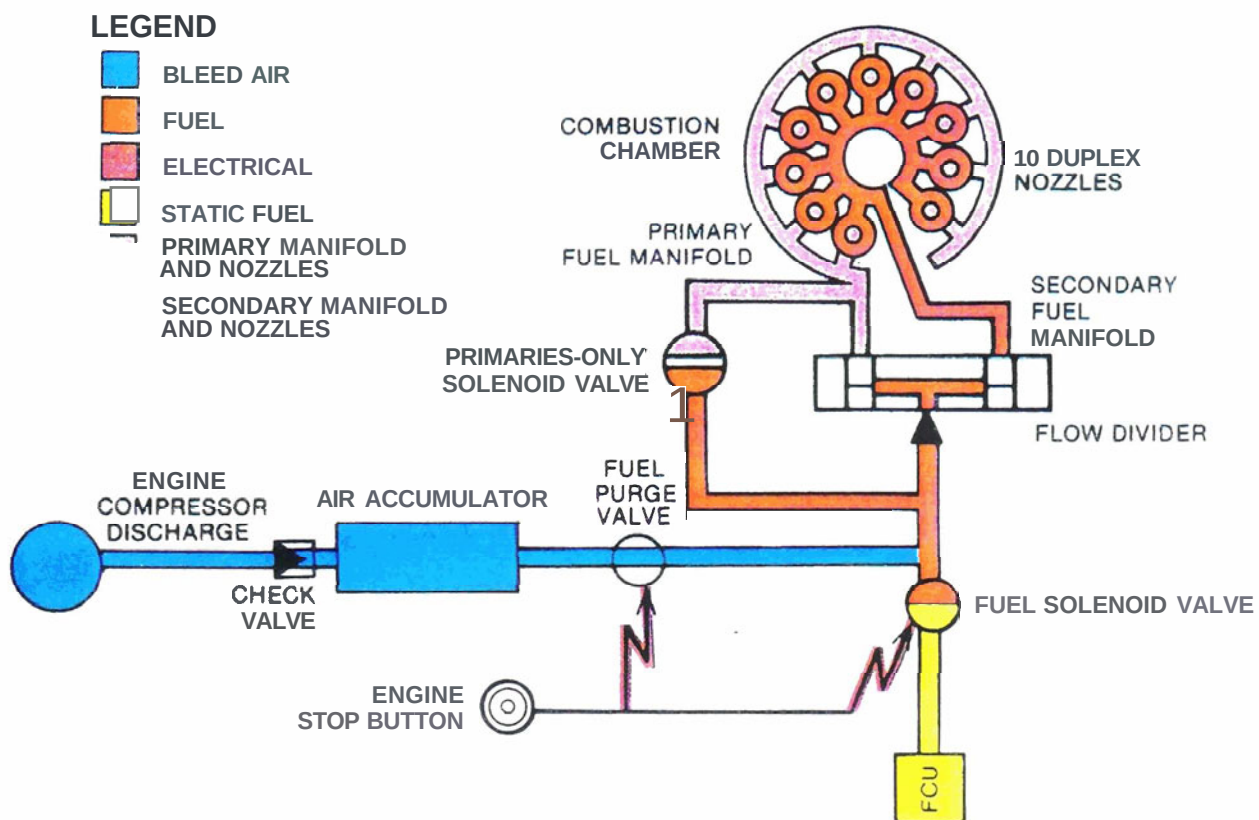
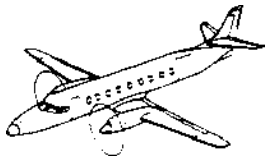


Figure 7-27. Engine Stop System Schematic



CONTINUOUS ALCOHOL-WATER INJECTION (CAWI)

GENERAL

The CAWI system is used during takeoff to recover power lost at high-density altitudes. The mixture used is 40% methyl alcohol and 60% distilled or demineralized water.

CAUTION

Methanol and its vapor are toxic and flammable.

The CAWI system includes a storage tank in the nose section, two pumps, and a spray ring and nozzles in the engine inlet.

CAWI may be used only for takeoff and for a maximum of five minutes. In-flight use of CAWI is prohibited.

The CAWI storage tank holds 16 gallons of usable AWI fluid. A cockpit gage (Figure 7-28) shows AWI quantity.

CONTROL

The CAWI system is controlled by a two-position WATER INJECTION switch (Figure 7-29) with positions marked "CONT" and "OFF," located on the center pedestal. Another switch on the center pedestal (Figure 7-29) is marked "AWI PUMP TEST." It has two labeled positions, NO 1 and NO 2, and an unlabeled center off position.

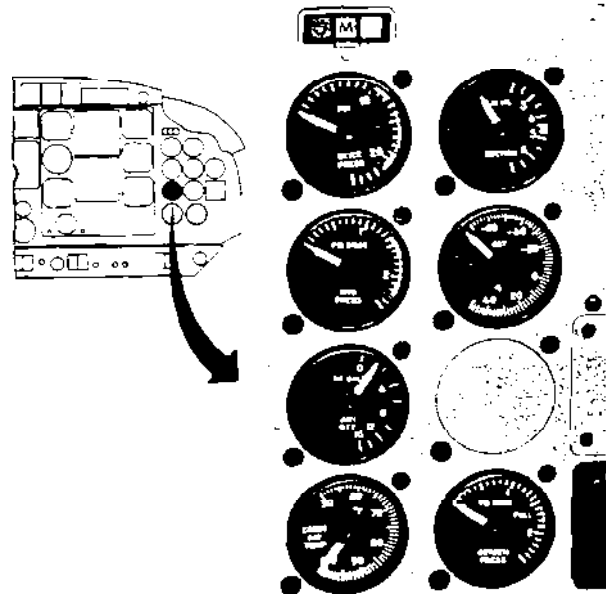


Figure 7-28. AWI Gage

INDICATION

Two annunciator lights (Appendix B) marked "AWI NO. 1 PUMP ON" and "AWI NO. 2 PUMP ON" are illuminated whenever the as-

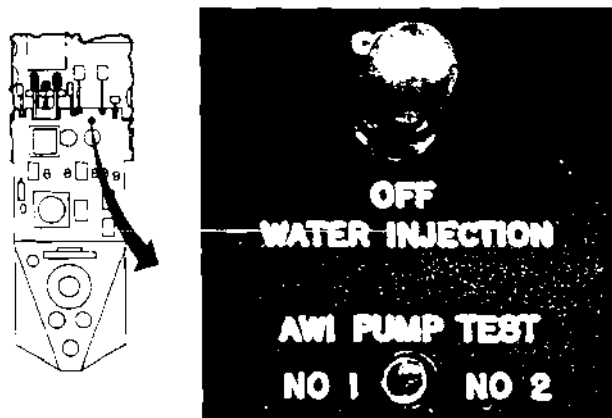
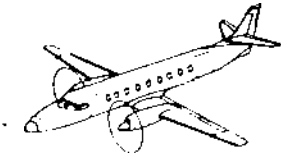


Figure 7-29. AWI Switch and AWI PUMP TEST Switch



sociated AWI pump is running and developing acceptable pressure. If the WATER INJECTION switch is OFF, holding the AWI PUMP TEST switch to the NO 1 or NO 2 position should turn on the associated AWI pump and light. This verifies that the pump is operational and that the check valve of the other pump is properly seated. Illumination of both lights indicates that the check valve of the opposite system is faulty, and the condition should be corrected before a wet takeoff is attempted.

OPERATION

Figure 7-30 shows operation of the AWI system. To activate the CAWI system for takeoff:

- Engine RPM must be above 90%.
- The power lever must be approximately 1 inch forward of FLT IDLE (to close a microswitch in the power lever quadrant).
- The WATER INJECTION switch must be placed in CONT.

When these conditions are met, the pumps operate, the bleed air valves close, and the AWI shutoff valves open, allowing fluid flow to the engines. The pilot should note an immediate torque increase of 30–35% and illumination of the AWI PUMP ON lights. AWI fluid is now flowing to the spray manifold at

the top of each engine inlet. When the pressure in the manifold reaches a specific value, a pressure switch closes and sends a signal to the SRL computer. AWI flow continues until the AWI switch is turned off, the AWI fluid runs out, or engine rpm decreases below 90%.

If one engine fails, its AWI flow stops when rpm decreases below 90%. Flow continues for the operating engine.

NOTE

When the WATER INJECTION switch is turned off, torque may decrease as much as 35%.

LIMITATIONS

GENERAL

The following is a partial listing of the powerplant limitations; see the appropriate AFM for a complete listing.

OIL

Mixing oil types or brands is prohibited. Refer to Garrett AiResearch Specification EMS 53110 Type II for a current list of approved oils.

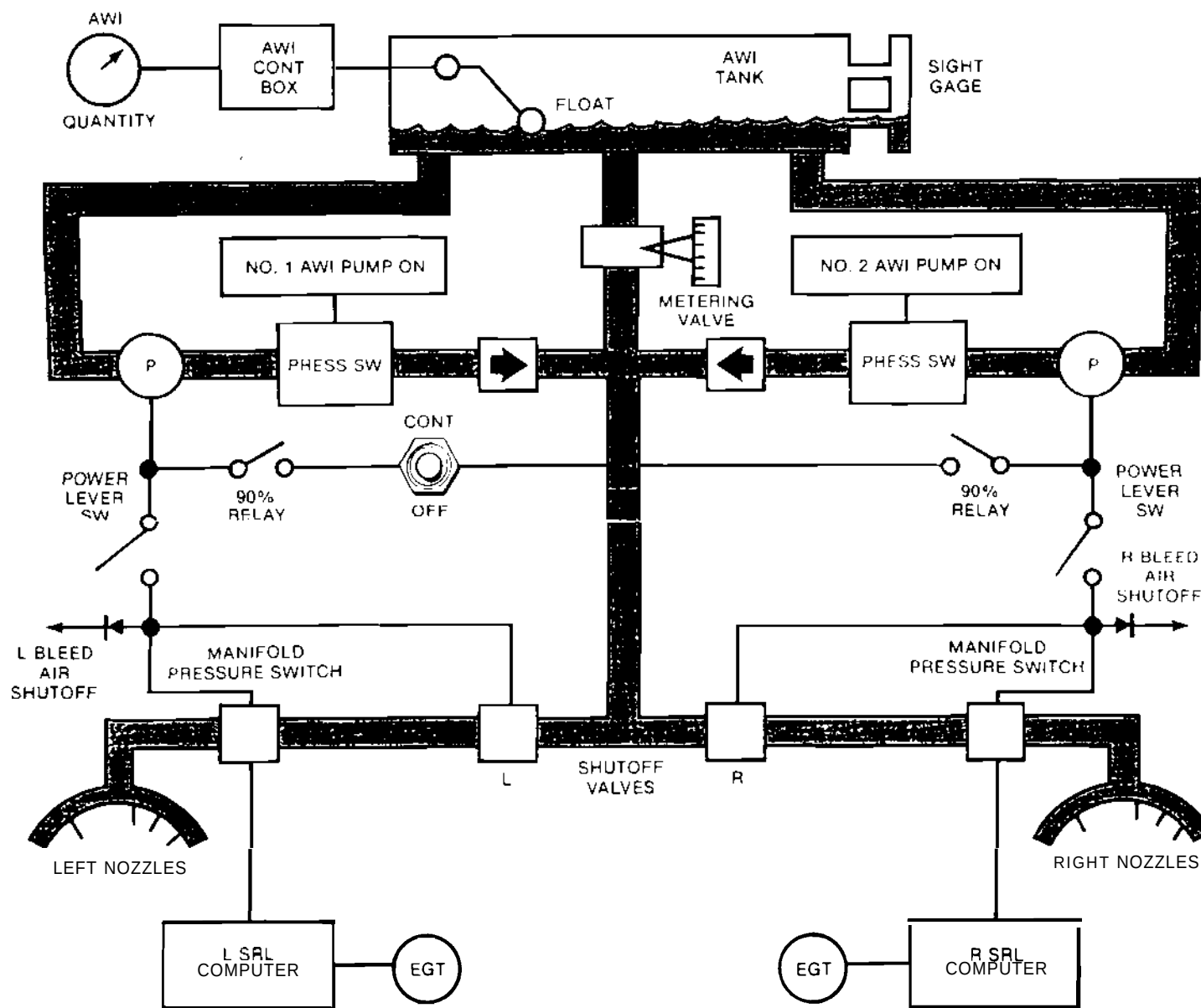
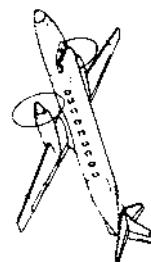
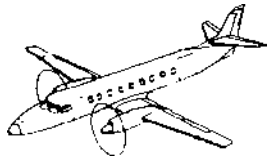


Figure 7-30. CAWI Schematic



Maximum Recommended Starting Current

Due to the possibility of excessively high current surge during engine start, it is recommended that the maximum starting current from an external power source be limited to 1,000 amperes.

ENGINE STARTER DUTY CYCLES

General

The specified starter-on times (Table 7-2) assume no ignition but do include engine clearing time. Starter-on time may be extended if ignition occurs in accordance with the AFM.

ENGINE OPERATION WITH CONTINUOUS ALCOHOL-WATER INJECTION (CAWI)

The CAWI system may not be operated if the AWI fluid has been exposed to ambient tem-

perature below -24°C within the preceding one hour. Table 7-1 shows CAWI limitations.

Table 7-1. CAWI LIMITATIONS

Time limit	5 minutes
Maximum torque	110%
Maximum EGT.....	650" C
Minimum OAT for CAWI operation	-6" C
Maximum usable AWI fluid quantity	16 U.S. Gallons
Minimum AWI fluid quantity for takeoff.....	See AFM
CAUTION CAWI use is limited to takeoff operations only. In-flight use of CAWI may result in exceeding the engine operating limits.	

Table 7-2. STARTER LIMITATIONS

START ATTEMPT	STARTER-ON TIME	STARTER-OFF TIME
1	30 seconds	60 seconds
2	30 seconds	60 seconds
3	30 seconds	15 minutes



Single Red Line Computer (SRL)

Operation of the airplane with the SRL inoperative is prohibited except as stipulated in the applicable **AFM**.

Temperature Limiter—Bypass Valve Open

Maximum altitude for prolonged operation is 15,000 feet.

RPM must be maintained between 99 and 101% or 96 and 98%.

Do not take off with the bypass valve failed in the open position.

REQUIRED ENGINE CHECKS

The NTS system must be checked:

- Operationally before the first flight of the day
- At intervals not to exceed 250 flight hours

The SRL computer and temperature limiter system should be checked operationally:

- At intervals not to exceed 50 flight hours
- Prior to any flight when manual engine start has been necessary
- When there is any indication of SRL computer malfunction
- After any engine fuel control or SRL computer maintenance or adjustment

The overspeed governors and the propeller unfeathering pumps should be checked:

- At intervals specified in the **AFM**
- Prior to any flight for which intentional airtasks are planned
- When there is any indication of malfunction
- After engine control system maintenance or adjustment

AIRSTART ENVELOPE

Maximum pressure altitude for airtasks:

- With boost pumps operating... 20,000 feet
- Without boost pumps operating 12,000 feet

Airspeed limits for
airstart..... 100 to 180 KIAS

PROPELLERS

Propeller Reversing

Full reverse operations (landing rollout, taxi, and ramp operations) are limited to speeds below 90 knots. Reduce this speed by 1 knot for each 1° F above 90° F prior to using full reverse power.

WARNING

Propeller reversing in flight is prohibited.

Do not retard power levers aft of FLT IDLE in flight.



TEMPERATURE LIMITS—ALL ALTITUDES

Minimum ambient temperature:

- For engine ground starting..... -40°C
- For engine operation..... -54°C

NOTE

Successful engine starts may not be possible if the fuel has cold-soaked at temperatures below -40°C .

Maximum ambient temperature..... $\text{ISA} + 40^{\circ}\text{C}$

ENGINE INSTRUMENT MARKINGS

Table 7-3. ENGINE INSTRUMENT MARKINGS

INSTRUMENT	RED RADIAL (MIN)	YELLOW ARC	GREEN ARC	YELLOW ARC	RED RADIAL (MAX)	RED DOT or DIAMOND
EGT ($^{\circ}\text{C}$)			0 to 650		650	7702
Torquemeter (% torque)			0 to 100	100 to 110	110	
Tachometer (% rpm)			96 to 100		101	
Fuel Pressure (psi)	15 ¹	15 to 20	20 to 80		80	
Oil Pressure (psi)	40 ¹	40 to 70 ¹	50 to 70 ² 70 to 120		120	
Oil Temperature ($^{\circ}\text{C}$)	4 0	4 0 to 55	55 to 110	110 to 127 ³	127	
CAWI Quantity (gallons)	8/9 ⁴					
A t 71% rpm ² Above 23,000 feet, minimum oil pressure is 50 psi. ³ Ground operation only ⁴ 16,000-pound aircraft requires 9-gallon minimum CAWI for wet takeoff.						



ENGINE LIMITATIONS

Table 7-4. POWER SETTING AND OPERATING LIMITS FOR TPE331-11U-611G OR -612G TURBOPROP ENGINES WITH REVERSING PROPELLERS

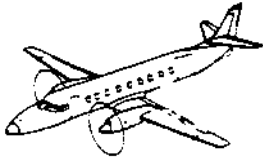
POWER SETTING	TIME	MAXTORQUE (%)	MAX EGT (° C)	RPM (%) ¹	OIL PRESS (PSI)	FUEL PRESS (PSI)	OIL TEMP (° C)
Takeoff (Dry)	...	100 ²	6503	100 ⁴	70 to 120	20 to 80	55 to 110
Takeoff (Wet)	5 minutes	110 ⁵	6503	100 ⁴	70 to 120	20 to 80	55 to 110
Maximum Continuous	...	100	650 ³	100	70 ⁶ to 120	20 to 80	55 to 110
Flight Idle	...	...	...	96 to 100	70 ⁶ to 120	20 to 80	55 to 110
Ground Idle	...	...	...	70 or 97 ⁷	40 to 120	15 to 80	-40 to 127
Starting	...	...	770	...	...	...	-40 minimum
Reverse High	...	...	...	95.5 to 97	70 to 120	20 to 80	55 to 110
Reverse Low	...	...	...	75 minimum	70 to 120	15 to 80	40 to 127
Shutdown	3 minutes ⁸	...	...	...	...	...	...

- ¹ Avoid operation between 18 and 28% rpm except for transients occurring during engine start and shutdown. 96% is the minimum rpm during flight. At 100% rpm, propeller speed is 1,591 rpm.
- ² Static takeoff power should be limited to 97% dry torque to preclude overtorque condition occurring due to ram effects during takeoff (100% equals 3,301 ft-lb).
- ³ 650 to 675° C, reduce power; 675 to 685° C for less than 20 seconds, reduce power; 675 to 685° C for more than 20 seconds, conduct power check; 685 to 687° C for up to 5 seconds, conduct power check; 685 to 687° C for more than 5 seconds or in excess of 687° C, remove engine.
- ⁴ 101 to 101.5% for 5 minutes, 101.5 to 105.5% for 30 seconds—If rpm time limits are exceeded, conduct power checks to determine satisfactory engine performance. Record time in excess of time limits in engine logbook. 105.5 to 106% for 5 seconds—If 5-second limit or 106% is exceeded, remove engine.
- ⁵ Static takeoff power should be limited to 107% wet torque to preclude overtorque condition occurring due to ram effects during takeoff.
- ⁶ Above 23,000 feet, minimum oil pressure is 50 psi.
- ⁷ Typical engine speeds for low and high rpm speed lever position.
- ⁸ Three-minute cooldown period prior to stopping engines. Descent, approach, landing roll, and taxi times are included if power during those phases does not exceed 20% torque. If reverse power is used and exceeds 20% power (stabilized), the 3-minute cooling period commences at engine power reduction.

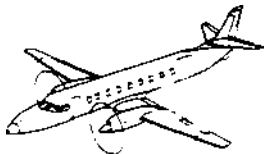


QUESTIONS

1. One of the following best describes the four-blade propellers:
 - A. Double-acting hydraulic
 - B. **Variable-speed**, autofeathering
 - C. Reversible, constant-speed, full-feathering
 - D. Constant-speed, double-acting
2. A heated inlet on the lower side of the nacelle supplies air for:
 - A. The gas generator
 - B. Internal engine cooling
 - C. A fuel heater
 - D. An automatic oil cooler
3. A vent valve associated with the engine oil system is used for:
 - A. Reducing oil pump drag during engine starting
 - B. Venting the gear case to the atmosphere
 - C. Air-oil separation
 - D. Lubrication of the engine bearings
4. When the engine stop and feather control is pulled out, it will perform one of the following sequence of functions:
 - A. Feather the propeller and close the fire shutoff valve.
 - B. Feather the propeller and shut off fuel and **hydraulic** fluid.
 - C. Feather the propeller and close the fuel shutoff valve.
 - D. **Close** the fuel solenoid valve and feather the propeller.
5. The primaries-only solenoid valve functions to:
 - A. Limit engine torque when power is over 90% rpm.
 - B. Provide fuel enrichment between 10 and 60% rpm.
 - C. Increase primary nozzle fuel during start.
 - D. Prevent compressor stall during starting.
6. If the fuel bypass open light of the left engine comes on during takeoff, the proper action to take is:
 - A. Abort the takeoff if below lift-off speed.
 - B. **Take no** action until the critical point of the takeoff is complete.
 - C. Reduce power on both engines.
 - D. Turn off the AWI system if operating.
7. If a manual engine start was made, the proper action to take after starting is:
 - A. Leave the SRL switch in the OFF position.
 - B. Pull the ignition circuit breaker at 60% rpm.
 - C. Leave the SPEED SW SELECT in MANUAL.
 - D. Determine that the SRL function is operational.
8. The maximum acceptable difference in EGT indications between engines in the SRL on and off modes is:
 - A. 30° C
 - B. 15° C
 - C. 10° C
 - D. 0° C
9. When setting dry static takeoff power, the proper action to take is:
 - A. Never exceed 770° C EGT.
 - B. Limit torque to 97% maximum.
 - C. Do not exceed 675° C for more than five minutes.
 - D. Limit rpm to 100%.
10. The maximum altitude for sustained operation with the fuel bypass valve open is:
 - A. 30,000 feet
 - B. 25,000 feet
 - C. 20,000 feet
 - D. 15,000 feet



11. If the fuel bypass valve has failed in the open position, the proper action to take is:
- A. Maintain rpm at **96%** maximum.
 - B. Do not take off.
 - C. Maintain rpm at **100%** if EGT is greater than **650° C**.
 - D. Do not exceed **600° C** EGT for operation above **15,000** feet.
12. The maximum forward speed for full reverse thrust application is:
- A. Unlimited on paved surfaces
 - B. **100** knots, all conditions
 - C. **90** knots on unpaved surfaces
 - D. **90** knots if temperature is **90°** or less
13. The maximum EGT during engine starting is:
- A. **770° C**
 - B. **650° C**
 - C. **580° C**
 - D. **560° C**
4. Except for transients during starting and shutdown, engine operation must be avoided in the **rpm** range of:
- A. **96** to **98%**
 - B. **10** to **60%**
 - C. **70** to **97%**
 - D. **18** to **28%**
15. The **CAWI** system functions to:
- A. Reduce compressor outlet temperature and permit higher EGT limits.
 - B. Provide engine cooling during periods of high-power demands.
 - C. Restore power when OAT is high or when taking off from high-altitude fields.
 - D. Prevent ice formation at the nacelle inlet and in the compressor inlet duct.
16. The minimum **AWI** quantity for a wet power takeoff is:
- A. 4 gallons
 - B. **8** gallons
 - C. **9** gallons
 - D. **16** gallons

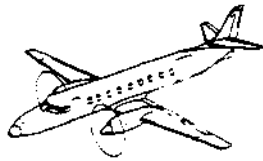


CHAPTER 8

FIRE PROTECTION

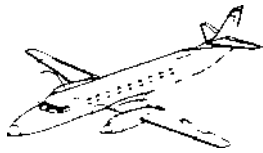
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CHAPTER 8

FIRE PROTECTION



INTRODUCTION

The fire protection system consists of engine and wing overheat detection systems, engine fire-extinguishing systems, flammable fluids protection, and heat-protective sleeving on wiring bundles in areas which might be subject to overheat. There are cockpit warning lights located on the annunciator panel and on the fire extinguisher control assembly.

GENERAL

Heat sensors which complete an electrical circuit are located in each engine nacelle and within each wing. There are overheat detectors within the wing leading edges, the wheel wells, and the conditioned air ducts.

A fire extinguisher, which is discharged from the cockpit is located within each engine nacelle. Attached to each extinguisher bottle is

a pressure gage which is checked during airplane preflight.

Flammable fluids which are ducted through wheel wells, the wing center section, the cockpit, and the nose baggage compartment are shrouded to isolate them from possible ignition sources.



ENGINE FIRE DETECTION

Four heat sensors are located in each engine compartment (Figure 8-1) to provide warning of overheating. Each functions independently, but they are connected in parallel to activate the appropriate FIRE warning light when any one detects an overheat condition.

Each fire detector is a stainless steel tube containing two contacts. If excessive temperature is detected, a contact is made, illuminating the red FIRE warning light in the fire extinguisher switchlight. The detector can withstand flash fires. When the temperature drops, the detector cools and opens the contacts, extinguishing the FIRE light.

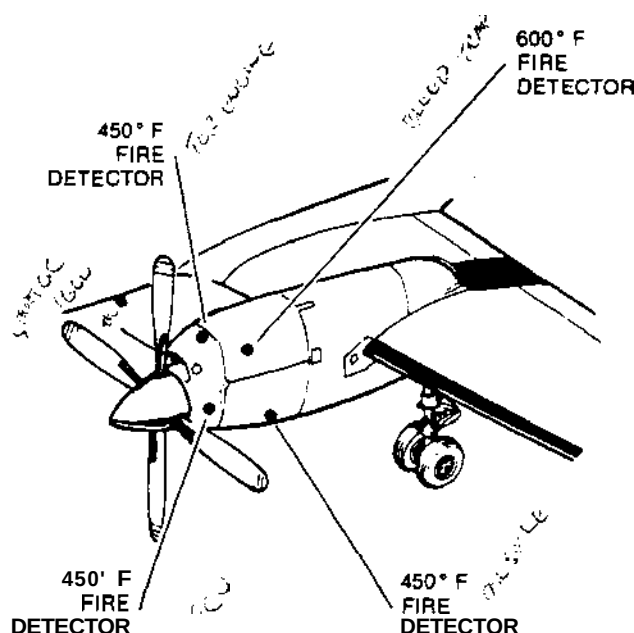


Figure 8-1. Engine Fire Detectors

The PRESS TO TEST switch on the annunciator panel verifies the continuity of the wiring to the eight engine fire detectors by illuminating the red FIRE warning lights in the fire extinguisher switchlights for both engines. All annunciators are shown in Appendix B.

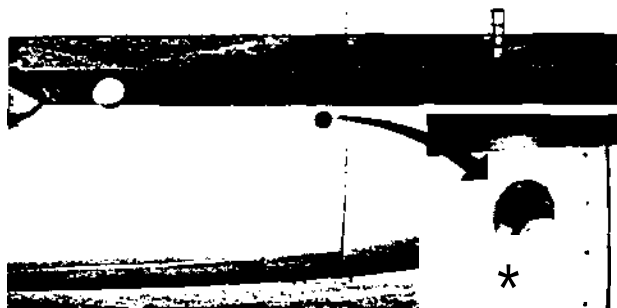


Figure 8-2. Engine Fire Extinguisher Pressure Gauge

NOTE

In addition to the two switchlights, some aircraft have two red ENG FIRE lights on the annunciator panel which function the same as the switchlights.

ENGINE FIRE EXTINGUISHING

Each engine nacelle contains a fire extinguisher bottle located just behind the firewall. When the noncorrosive agent (Halon) is released, it is directed to several areas in the nacelle. On the right side of the nacelle, the pressure gauge is visible for preflight check (Figure 8-2).

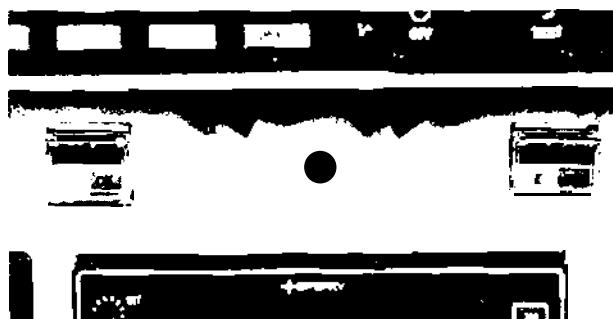


Figure 8-3. Fire Extinguisher Control Assembly



The control assembly mounted on the instrument panel contains a logic control module, two combination fire extinguisher system switchlights, and a test switch (Figure 8-3). The two switchlights (one for each bottle) contain three indicator lights and the bottle actuation switch. Pushing the lens of the switchlight discharges the bottle. All engine fire detectors and both extinguishers are powered as long as there is power on either essential bus. The lens is covered by a clear plastic, spring-loaded guard to avoid accidental discharge. The indicator comprises a red FIRE light, an amber E (empty) light, and a green OK light. The lights are operated by the logic module. The FIRE EXT TEST switch is located between the two switchlights on the instrument panel.

Pressing the annunciator panel test switch will **test the engine fire warning system for continuity by illuminating the red FIRE lights on the switchlights**. Should the fire detectors on the engine sense an **overheat condition**, the FIRE light will illuminate.

Normally, no switchlights are illuminated. If a bottle is thermally discharged or discharged from the cockpit, the amber E light will illuminate. When the FIRE EXT TEST switch is pressed, all three lights on both switchlights should illuminate. Illumination of the green OK light indicates that the bottle discharge circuit is good.

FLAMMABLE FLUID PROTECTION

All flammable fluids are isolated from possible ignition sources by shrouding certain fluid carrying lines and assemblies and by shielding electrical wiring and terminals. Most of these protected areas are in the wheel wells, wing center section, cockpit, and nose baggage compartment.

Electrical wiring throughout the center section is wrapped with fire sleeving. All terminal

strips and connectors are potted or sealed. The main wire bundles in the fuselage are also covered by fire sleeving.

Hydraulic lines, valves, and master cylinders in the cockpit and nose baggage compartment, as well as fuel lines and valves in the main wheel wells, are neoprene-shrouded and drained overboard.

WHEEL WELL AND WING OVERHEAT WARNING

GENERAL

Two red annunciator panel lights labeled "L WING OVHT" and "R WING OVHT" indicate **overheat conditions in the respective wing leading edges, wheel wells, or conditioned air ducts**.

Depending on the source of the overheat signal, the respective light will either flash or be

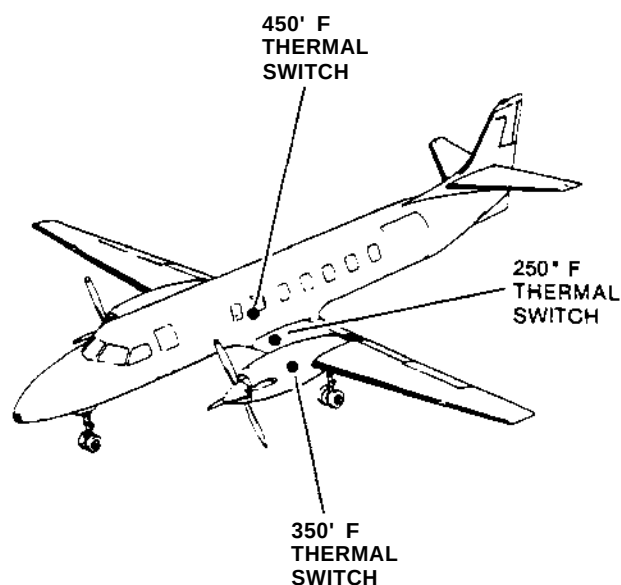
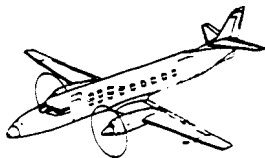


Figure 8-4. Wing Overheat Sensor Locations



on steady. When testing the annunciator lights, keep the test button depressed until the wing overheat lights begin to flash.

LEADING-EDGE OVERHEAT (FLASHING LIGHT)

Bleed-air lines supplying the air-conditioning and ice protection systems are routed inside the wing leading edge. There are also wire bundles with battery and generator power cables in the same area. A temperature sensor which can detect a hot wire or a broken bleed-air line, is set to actuate at 250° F, and is secured to the electrical wire bundle. If a temperature above 250° F is sensed, a signal is sent to the annunciator panel, causing the respective WING OVHT light to flash.

WHEEL WELL OVERHEAT (STEADY LIGHT)

A temperature sensor is installed near the bleed-air lines within each wheel well. It is set to actuate at 350° F and cause the respective WING OVHT light to illuminate.

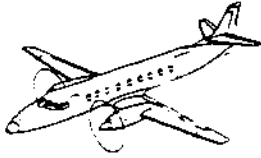
CONDITIONED AIR DUCT OVERHEAT (STEADY LIGHT)

A sensor is installed in the duct from each air-conditioning system to detect overheat of the cooling turbine. The sensor will detect a temperature of 450° F and illuminate the respective WING OVHT light.



QUESTIONS

1. Overheat sensing within the engine nacelle will:
 - A. Illuminate the red FIRE and amber E indicators.
 - B. Not illuminate any light unless two or more sensors detect the overheat condition.
 - C. Illuminate a red FIRE light on the main annunciator panel and automatically discharge the extinguisher.
 - ☒ D. Illuminate the red FIRE light on the respective fire extinguisher switchlight.
2. Pushing in on the left engine fire extinguisher switchlight:
 - ☒ A. Discharges the left engine fire extinguisher bottle as long as electrical power is available.
 - B. Only activates the extinguisher if the left FIRE light is illuminated.
 - C. Tests all three lights and the detector circuitry.
 - D. Discharges both fire extinguishers into the left engine nacelle.
3. A wing overheat warning is shown by:
 - A. A flashing FIRE light on the control panel.
 - ☒ B. Either a steady or a flashing WING OVHT light on the annunciator panel, depending on what is causing the condition.
 - C. A L or R WING OVHT temperature gage on the center instrument panel.
 - D. A L or R WING OVHT annunciator light which is on steady until the temperature reaches 350° F and then begin to flash.



CHAPTER 9 PNEUMATICS

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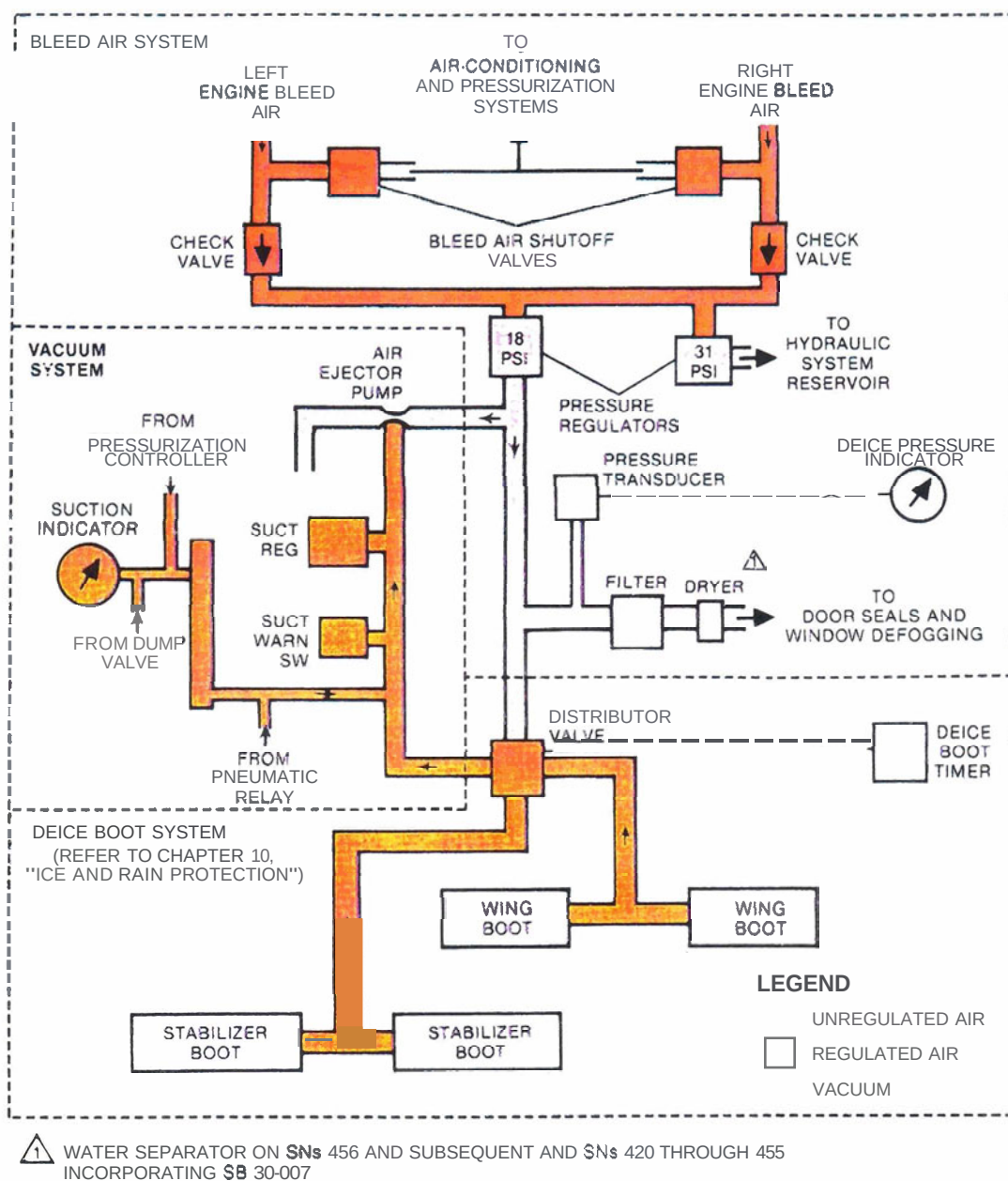
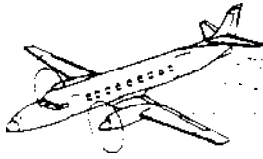


Figure 9-1. Pneumatic System Overall Diagram



The bleed air shutoff valves regulate the airflow to the respective air-conditioning system. Since the extraction of bleed air causes a loss in engine power, the amount of air extracted must be carefully regulated. The bleed air shutoff valve is calibrated to maintain a preset amount of bleed air from the engine under all operating and ambient conditions.

When an engine is not running, the respective bleed air shutoff valve is spring-loaded closed. When an engine is running, the unregulated bleed air pressure causes the bleed air shutoff valve to open and function as a pressure regulator. The valves may be closed electrically by operation of the BLEED AIR VALVES toggle switches (Figure 9-2) located on the lower right switch panel. Turning a switch OFF stops airflow to the respective air-conditioning and pressurization systems. (The bleed air supplied to the engine anti-icing and the 18- and 31-psi pressure regulators is not affected

by the position of the bleed air shutoff valves.) Electrical power to close the bleed air shutoff valves comes from the BLEED AIR circuit breakers on the nonessential bus.

The deice pressure indicator (Figure 9-31, located on the far right side of the instrument panel, allows the pilots to monitor the 18 psi system pressure. An electrical signal from the deice pressure transducer drives the indicator.

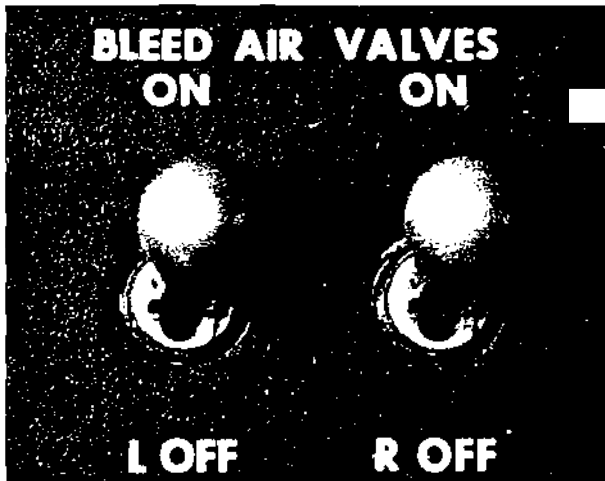


Figure 9-2. BLEED AIR VALVES Toggle Switches

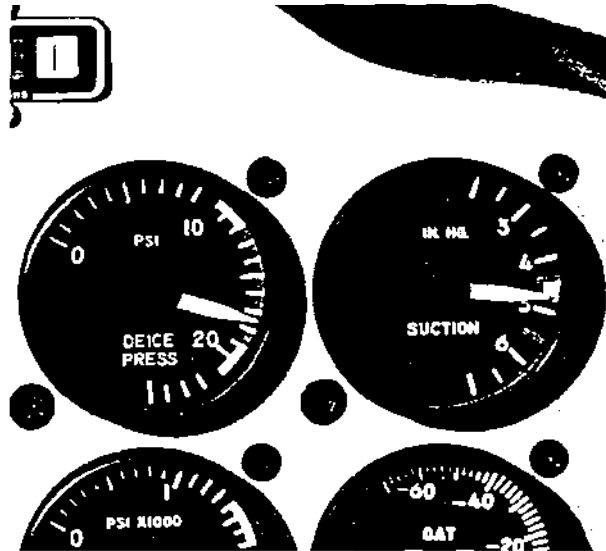


Figure 9-3. Deice Pressure and Suction Indicators

VACUUM SYSTEM DESCRIPTION AND OPERATION

The vacuum system (Figure 9-4) supplies the necessary suction for operation of (1) the vacuum instruments (if installed), (2) the hold-down suction phase of the surface deice boots, and (3) control of the pressurization system. Bleed air from either engine is sufficient to maintain full system operating capacity.

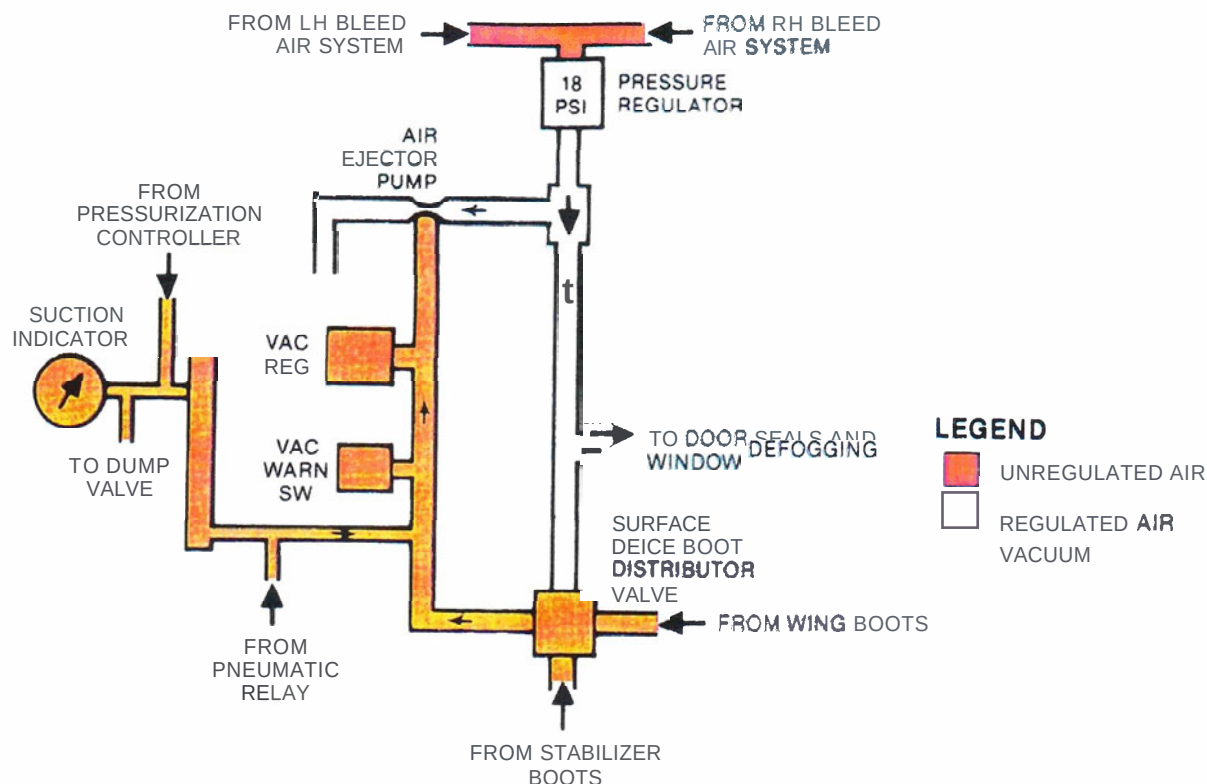


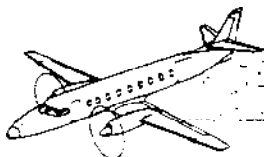
Figure 9-4. Vacuum System Flow Diagram

The air ejector is the primary component of the vacuum system. Regulated 18-psi air flows to the air ejector, where vacuum is created by venturi action.

A vacuum regulator is installed in the suction side of the air ejector. It maintains a vacuum of 4.4 to 4.8 inches Hg in the system. This regulator incorporates a filter for outside air that is entering the vacuum system.

A pressure switch illuminates an amber LOW SUCTION warning light if the suction drops below limits (see Appendix B). The LOW SUCTION warning light uses electrical power from the SUCTION circuit breaker on the nonessential bus.

The suction indicator (Figure 9-3) on the instrument panel is direct-reading and does not require electrical power.



QUESTIONS

1. Bleed air supplied to the engine and nacelle inlet anti-icing system is:
☒ A. Unregulated
☐ B. Regulated to 18 psi
☐ C. Regulated to 31 psi
☐ D. Regulated by the bleed air shutoff valve
2. The following is calibrated to provide pressure-regulated bleed air to the air-conditioning system:
☒ A. Check valve
☐ B. Bleed air shutoff valve
☐ C. Filter
☐ D. Pressure regulator
3. The indicator used to monitor the 18-psi system pressure is the:
☐ A. Bleed air pressure indicator
☒ B. Suction indicator
☐ C. Deice pressure indicator
☐ D. LOW SUCTION warning light
4. The indicator used to monitor the vacuum system is the:
☐ A. Bleed air pressure indicator
☐ B. Pressure regulator
☐ C. Deice pressure indicator
☒ D. Suction indicator
5. Bleed air supplied to the vacuum system is:
☐ A. Unregulated
☒ B. Regulated to 18 psi
☐ C. Regulated to 31 psi
☐ D. Regulated by the bleed air shutoff valve



CHAPTER 10

ICE AND RAIN PROTECTION

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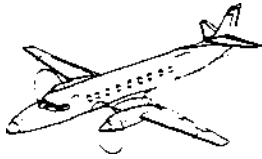
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CHAPTER 10

ICE AND RAIN PROTECTION



INTRODUCTION

Ice and rain protection is provided for the following components:

- Wings and horizontal stabilizer
- Engine and nacelle inlets
- Propellers
- Oil cooler ducts
- Windshields
- Pitot tubes and SAS vane
- Fuel system

An optional window defog system is available.

GENERAL

The leading edges of the wing and horizontal stabilizer are protected by electrically controlled and pneumatically operated deice boots.

The engine inlets are heated by hot bleed air tapped from the engines. The associated valves are electrically controlled.



SA-227 PILOT TRAINING MANUAL

The windshields, propellers, oil cooling air duct inlets, pitot tubes, and SAS vane are electrically heated. The windshield wipers are electrically operated.

The window defog system (if installed) taps bleed air from the door seal inflation plumbing and operates whenever an engine is operating.

Fuel is automatically heated as required by hot scavenge oil from the reduction gearcase.

WING AND HORIZONTAL STABILIZER DEICE BOOT SYSTEM

The leading edges of the wing and horizontal stabilizers are protected from icing by deice boots containing built-in inflation tubes. The deice boots are pneumatically operated by

regulated 18-psi air pressure from the bleed air system. They are cemented in place and surfaced with conductive neoprene to dissipate static electric charges.

Two boots per wing are installed between the engine nacelle and wingtip on the 14,500-pound airplanes, while the 16,000-pound airplanes have an additional wing deice boot between the engine nacelle and fuselage fairing (Figure 10-1).

This system is controlled by the three-position DEICE BOOTS switch on the pilot's lower switch panel (Figure 10-2). When positioned to AUTO, the deice boots are cycled by an electronic timer in the following sequence: the wing boots inflate for six seconds, then deflate while the horizontal stabilizer boots inflate for four seconds. All boots deflate under suction while the system rests for 170 seconds. One complete cycle of the deice boot timer lasts three minutes.

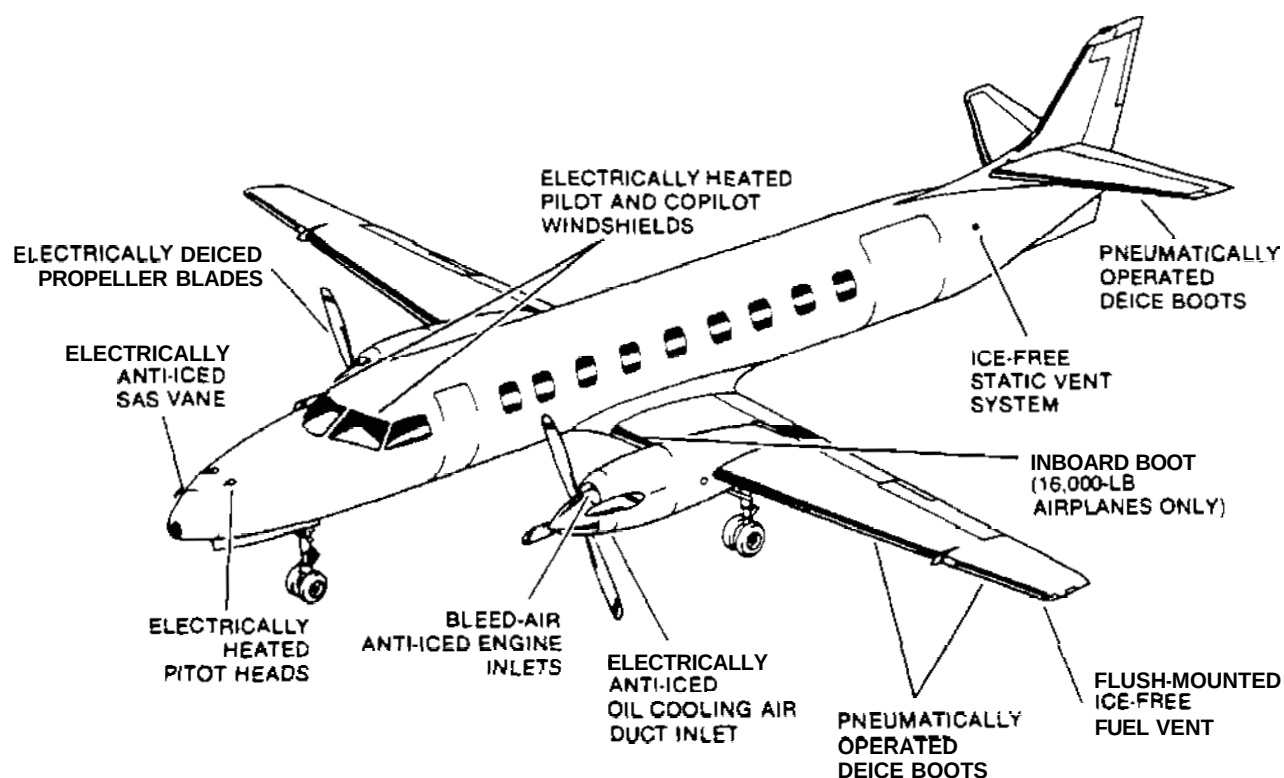


Figure 10-1. Icing Protection



SA-227 PILOT TRAINING MANUAL

When positioned to MANual, all boots are inflated simultaneously. The boots remain inflated as long as the switch is depressed. When the switch is released, the boots deflate and are held flat by vacuum.

When an inflation cycle ends, pressurized air which remains in the boot is exhausted overboard, and the boots are held to the leading edges by vacuum.

CAUTION

Do not turn on the deice system until sufficient ice (1/2 to 1 inch) has built up on the leading edge so that effectiveness of the system is assured.

Deice boot operation is checked by positioning the DEICE BOOTS switch to AUTO and monitoring the DEICE PRESSure gage on the right side of the instrument panel. Pressure will fluctuate twice as first the wing boots

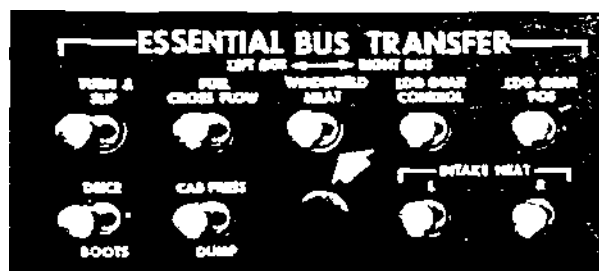


Figure 10-3. ESSENTIAL BUS TRANSFER Switches (Typical)

and then the horizontal stabilizer boots inflate. Approximately three minutes should be allowed for one full cycle of the timer.

Electrical power for the deice boot system is selectable to the left or right 28.5-VDC essential bus through the DEICE BOOTS switch on the ESSENTIAL BUS TRANSFER panel (Figure 10-3). The ESSENTIAL BUS TRANSFER switches are located on the left-aft console and are normally selected to the left essential DC bus.

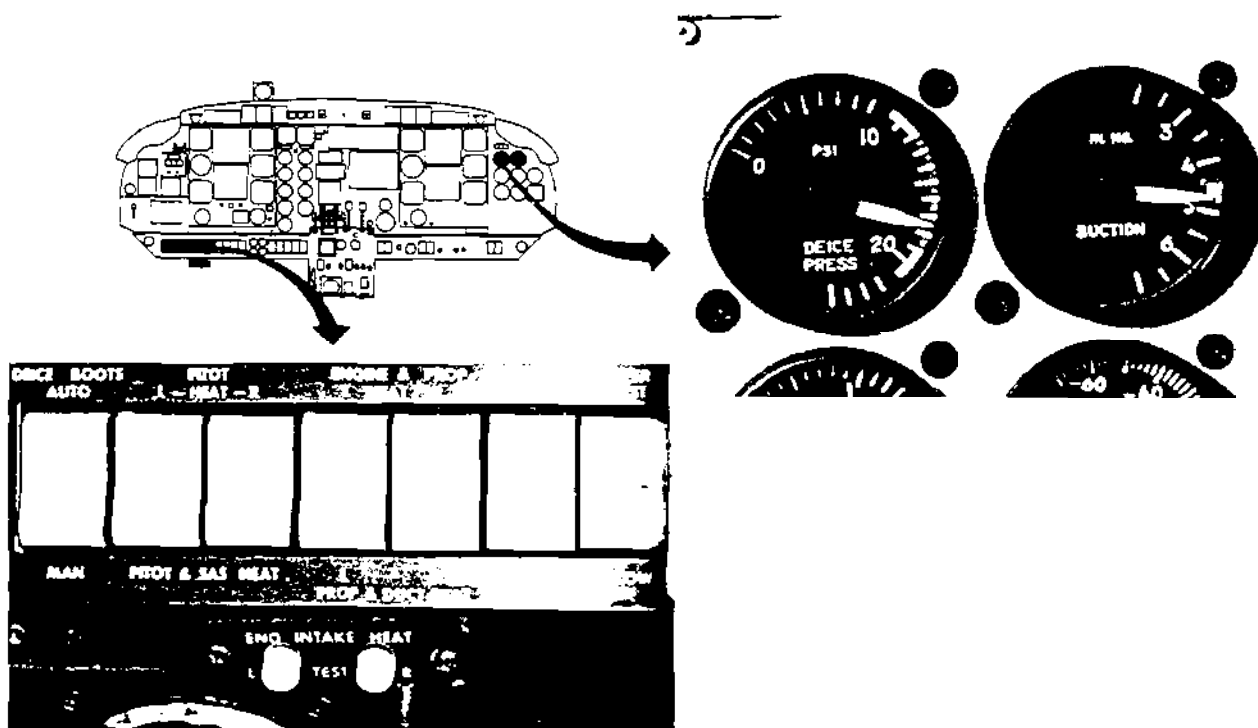


Figure 10-2. Ice and Rain Protection Switches and Deice Pressure Gage



ENGINE AND NACELLE INLET ANTI-ICING SYSTEM

The engine and nacelle inlet anti-icing system taps hot bleed air from each engine compressor and routes it to an inlet anti-ice valve for that **engine**. The valves are controlled by a pair of three-position switches labeled "ENGINE & PROP HEAT," located **on** the pilot's lower switch panel (**Figure 10-2**).

Placing a switch in the ENGINE & PROP HEAT position routes hot bleed air to the engine intake and $P_2 T_2$ probe and illuminates the appropriate INTAKE HT light on the annunciator panel (see Appendix B). The lower portion of the engine air inlet is kept free of ice by heat transfer from engine oil. The ENGINE & PROP HEAT position also activates the propeller deice and oil cooler duct anti-ice systems.

CAUTION

Use of engine inlet heat on the ground must be restricted to a maximum of ten seconds and 71% rpm when OAT is above +5° C.

NOTES

- Engine and propeller heat must be on prior to entering icing conditions and should be used continuously during icing conditions (OAT less than +5° C and visible moisture).
- EGT will increase slightly and torque will decrease slightly when ENGINE & PROP HEAT is selected.

This system is powered from the left or right 28.5-VDC essential bus, and is transferable by the L and R INTAKE HEAT switches on the ESSENTIAL BUS TRANSFER panel (**Figure 10-3**). The switches are normally selected to the left essential DC bus.

The INTAKE HT lights are valve agreement lights. With the system energized, illumination of the respective INTAKE HT light indicates that the intake heat valve is open. When the system is deenergized, the intake heat valve **position** is tested by depressing the ENG INTAKE HEAT TEST push-button switches (**Figure 10-2**). If the anti-ice valves are in the correct (closed) position, the INTAKE HT lights illuminate. It is important to press the ENG INTAKE HEAT TEST push-buttons any time the valves have been open, to verify closure.

When the ENGINE & PROP HEAT switches are positioned to ENGINE & PROP HEAT or PROP & DUCT HEAT, each propeller and oil cooling air duct inlet is electrically heated from its respective essential bus (**Figures 10-1 and 10-2**). The oil cooling air duct inlets (one per engine) are protected by a heating element and temperature sensor which are bonded to the inlet scoop lip. A thermostat maintains heating element temperature between approximately 90 and 120° F. The green DUCT HEAT CYCL lights on the left-forward console illuminate when heat is applied to the intake by the thermostat.

The DUCT HEAT CYCL lights (see Appendix B) should be checked for intermittent illumination, and the generator load ammeters should indicate corresponding load increases of approximately 30 amperes per duct when the thermostats are applying heat.



PROPELLER DEICE SYSTEM

Airplanes may be equipped with a single heating element or a dual-element boot on each blade. To conserve electrical power, a deice timer alternately cycles power between symmetrical heating elements approximately every 34 seconds.

The propeller deice timers do not have a home position. As a result, either set of boots will heat when the system is activated, depending on the point at which the timing cycle was interrupted. When the L and R ENGINE & PROP HEAT switches (Figure 10-2) are in the upper or lower position, each propeller deice system is powered from its respective 28.5-VDC essential bus.

CAUTION

The engine and propeller heat should be turned on before entering icing conditions to avoid damage from heavy pieces of loosened ice.

The propeller deice timers are checked by monitoring the PROP DEICE AMPS ammeter in the trimeter on the left-forward console (Figure 10-4) for at least one minute each in the L and R PROP DEICE positions. A small momentary needle deflection approximately

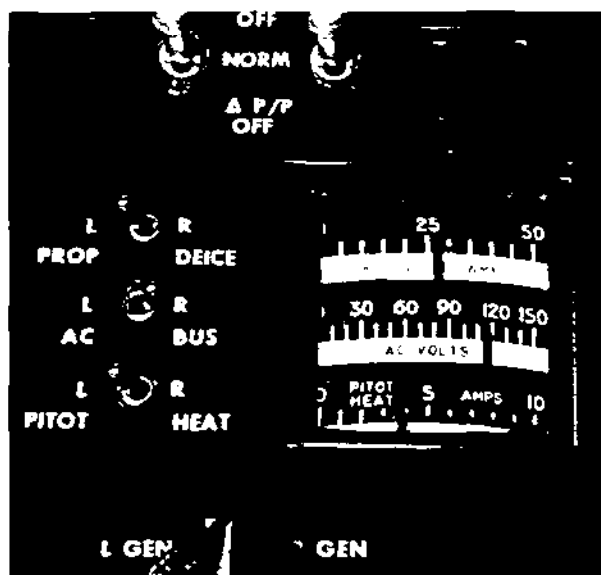


Figure 10-4. Trimeter Assembly

every 34 seconds indicates proper system operation. Propeller deice amperage should be in the green arc.

The propeller spinners are not provided with ice protection.

CAUTION

Propeller deice boots and oil cooler ducts must not be operated when the propellers are static.

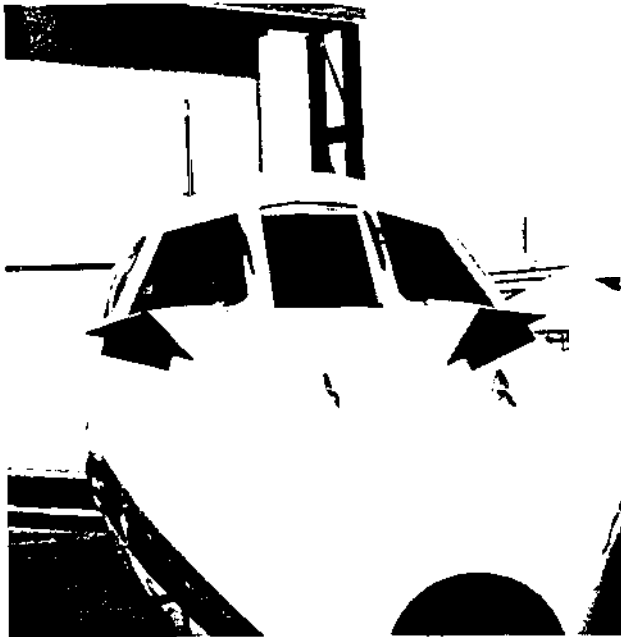


Figure 10-5. Heated Windshield Panels and Windshield Wipers

FUEL ANTI-ICING SYSTEM

Fuel is anti-iced automatically by hot scavenge oil through an oil-fuel heat exchanger mounted on each engine. A temperature-controlled anti-icing valve is opened when fuel temperature is low. An anti-ice lockout valve closes during engine start and opens at 60% rpm. The lockout valve prevents diversion of engine-driven fuel pump output during engine start.

WINDSHIELD HEAT SYSTEM

The pilot's and copilot's windshields are electrically heated (Figure 10-5). When the WSHLD HT switch (Figure 10-2) is positioned to LOW, power from the right essential bus activates a relay which causes heating elements in the windshields to heat in series. A

temperature controller for each windshield is powered from its respective bus. A temperature sensor in each windshield is connected to the respective temperature controller. The temperature controllers monitor the windshield temperature and attempt to keep the windshield temperature between approximately 90 and 100° F. When either temperature controller is calling for heat, the respective left or right W/S HT light on the annunciator panel is illuminated. Since the windshields operate in series in the LOW position, both windshield temperature controls must be calling for heat for heating current to flow.

When the WSHLD HT switch is positioned to HIGH, heat for each windshield is supplied from its respective essential bus. In the HIGH position, the two windshield heat systems operate independently. If the left essential bus fails, power for left windshield heat can be transferred to the right essential bus through the WINDSHIELD HEAT switch on the ESSENTIAL BUS TRANSFER panel (Figure 10-3).

Turning the heat on (HIGH or LOW) improves the windshield resistance to bird strikes.

WINDOW DEFOGGING SYSTEM

The window defogging system prevents condensation from forming between the dual panes of the cockpit center windshield and side windows. Beginning with SN 578, this system is installed on the Merlin IVC and is available as an option for the Metro III. Dried 18-psi bleed air is routed to the cockpit center and side windows whenever at least one engine is operating.

An optional center windshield defogging fan motor can be installed under the instrument panel with an outlet to divert air into the center windshield. The fan is controlled by a three-position switch labeled "HIGH," "OFF," and "LOW" on the copilot's switch panel.



WINDSHIELD WIPER SYSTEM

Separate windshield wiper systems are provided for the pilot and copilot. Each system is powered from its respective essential DC bus. The motor-driven systems are energized when the WIPER switch on the pilot's lower switch panel is positioned to FAST or SLOW (Figure 10-2).

In the SLOW position, the wipers operate at approximately half the maximum speed. When the switch is moved to the center (PARK) position, the windshield wiper blades automatically park on the windshield divider posts (Figure 10-5). The wipers **should not** be operated at speeds above 125 KIAS or on a dry windshield.

PITOT AND SAS ANTI-ICE SYSTEMS

Two pitot tubes are located on the upper nose section (Figure 10-1). Power for heating each pitot tube is from the respective essential bus.

Two dual switches on the pilot's lower switch panel labeled "PITOT HEAT" and "PITOT & SAS HEAT" energize the left and right systems individually (Figure 10-2). The PITOT HEAT position energizes the respective pitot heat system; the PITOT & SAS HEAT position energizes the respective pitot heat and SAS vane heat simultaneously. Either switch can energize SAS vane heat.

The PITOT HEAT AMPS ammeter is part of the trimeter and is located on the left-forward console (Figure 10-4). A selector switch is provided for checking the left or right pitot heat circuits for a reading in the green arc. SAS heating current does not register on this ammeter.

When either of the switches is in the PITOT & SAS HEAT position, the green SAS DEICE light on the annunciator panel should be illu-

minated, but the light is **not** positive indication that the vane is being heated.

The SAS vane is located aft of the radome on the right side of the nose cone.

CAUTION

Extended ground use will damage the pitot and SAS vane heating elements.

LIMITATIONS

For flight in icing conditions (visible moisture and OAT below +5° C), the following equipment **must** be on and operative:

- Engine and propeller heat
- Pitot tube and SAS vane heat
- Windshield heat (high)
- Wing and horizontal stabilizer deice boots as required
- Continuous ignition in the OVERRIDE or AUTO position (see applicable AFM)

NOTE

Continuous ignition is provided for use during takeoff or landing on wet or slush-covered runways when engine ingestion of ice or water is possible. It may also be used in flight. Refer to Chapter 7, "Powerplant," for additional information on this system.

CAUTION

Icing conditions affect airplane operation. See the appropriate AFM for V_{MC} and approach speed adjustments when flying in icing conditions.



QUESTIONS

1. The following components have pneumatically operated deicing systems:
 - A. Windshields
 - B. Propellers
 - ☒ C. Wings and horizontal stabilizers
 - D. Pitot tubes
2. One complete cycle of the deice boots timer takes:
 - A. 6 seconds
 - B. 10 seconds
 - C. 170 seconds
 - ☒ D. 3 minutes
3. Power for control of the following ice protection system is transferable from the left essential bus to the right essential bus:
 - A. Engine intakes
 - B. Left windshield
 - C. Wing and horizontal stabilizer deice boots
 - ☒ D. All the above
4. Annunciators which function as valve agreement lights when the applicable system is energized and when that system is deenergized and tested are:
 - A. W/S HT
 - B. DUCT HEAT CYCLE
 - ☒ C. INTAKE HEAT ON
 - D. SAS DEICE
5. Systems which should be used continuously during icing conditions are:
 - A. Engine and propeller heat
 - B. Windshield heat
 - ☒ C. Pitot and SAS heat
 - D. All the above
6. Use of engine inlet heat on the ground must be restricted to a maximum of:
 - A. 5 seconds, when OAT is above +5° C
 - ☒ B. 10 seconds, when OAT is above +5° C
 - C. 5 seconds, when OAT is above +10° C
 - D. 10 seconds, when OAT is above +10° C
7. When the WSHLD HT switches are positioned to LOW:
 - A. Heating elements in each windshield are heated in series, and the W/S HT lights illuminate while heat is being applied.
 - B. Heating elements in each windshield are heated in series, the W/S HT lights illuminate, and heat for either windshield system may cut off without affecting the other system.
 - C. Heating elements in each windshield are heated in series, the W/S HT lights illuminate, and heat cycling off for either windshield removes power from both and extinguishes both lights.
 - ☒ D. Heating elements in each windshield are heated in series, the W/S HT lights illuminate, and heat cycling off for either windshield removes power from both, but extinguishes only the light on the side cutting off.



CHAPTER 11

AIR CONDITIONING

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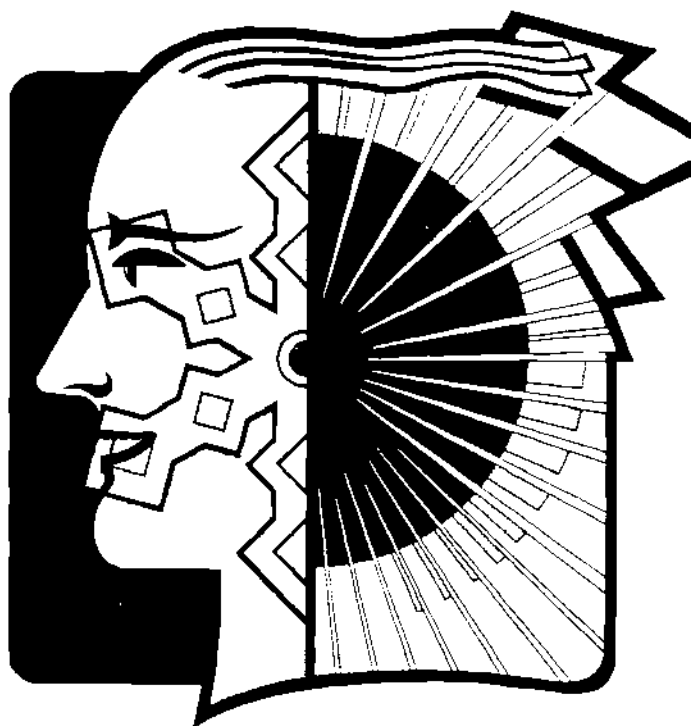
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CHAPTER 11

AIR CONDITIONING



INTRODUCTION

The environmental control system consists of three major subsystems: the air-conditioning system, the fresh air fan system, and the pressurization system.

Bleed air from the engines is cooled and temperature regulated in the air-conditioning system. The air is then routed to the cabin and provides a comfortable cabin temperature for the occupants. The air is discharged from the cabin through the pressurization system's outflow valve at a **controlled** rate to regulate the cabin pressure. Refer to Chapter 12, "Pressurization," for more information on the pressurization system.

The fresh air fan system provides cockpit ventilation during ground operation. It may also be used as a source of ventilation air during unpressurized flight.

AIR-CONDITIONING SYSTEM

GENERAL

The air-conditioning system supplies cold air and conditioned air to the cabin and the cock-

pit. Cold air and conditioned air are supplied by two independent systems, each capable of



providing complete air conditioning. Bleed air is supplied by the engines to drive cooling turbines which provide cold air for the airplane. Hot bleed air is mixed with a portion of the cold air to produce conditioned air. Ducts within the fuselage distribute the airflow to the passengers and crew.

DESCRIPTION

The airplane has **two** identical air-conditioning systems. Each system (Figure 11-1) has an air cycle machine, a water separator, a mixing valve, cold-air ducts, and conditioned air ducts.

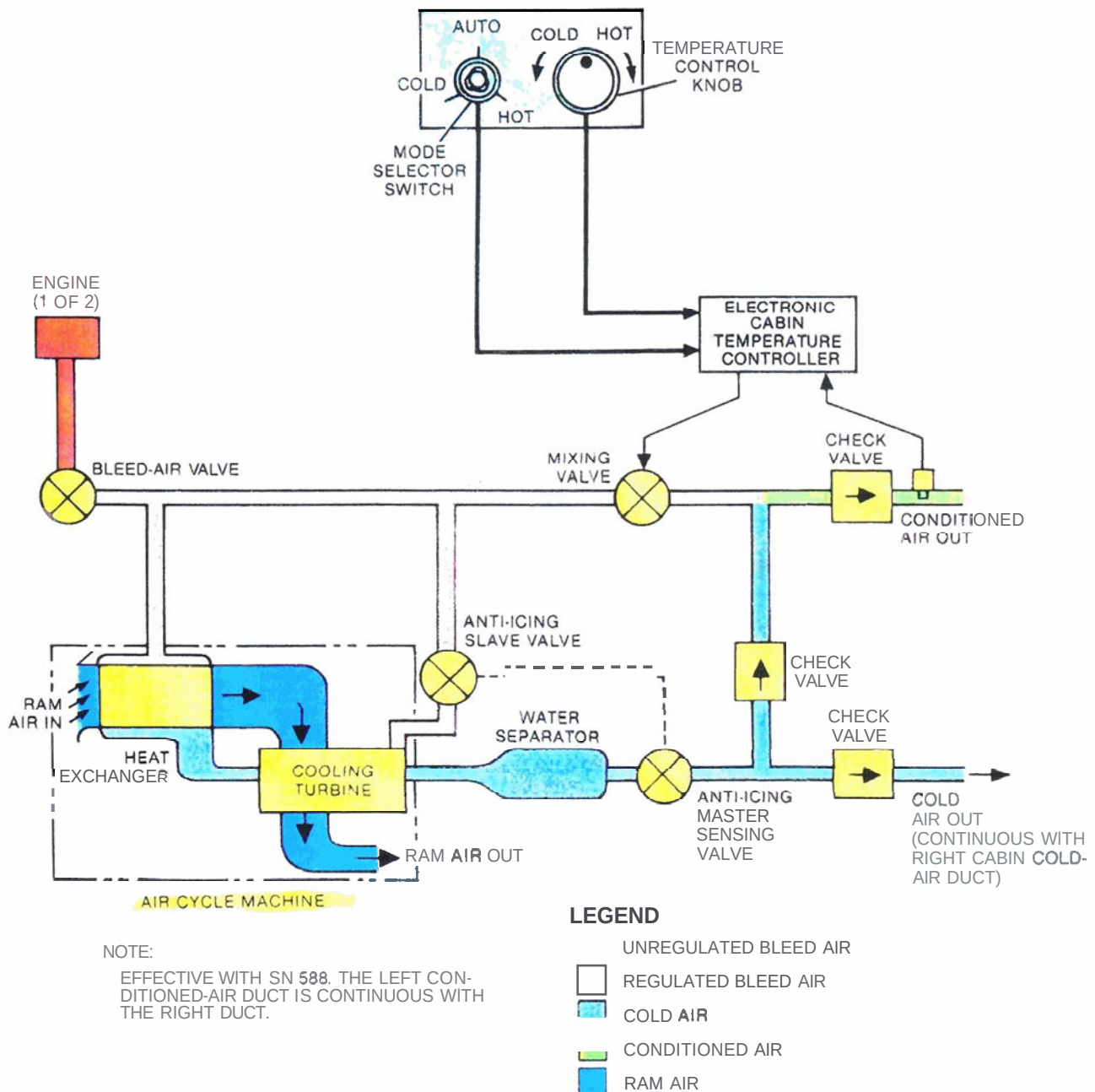


Figure 11-1. Air-Conditioning System Simplified Flow Diagram



Both systems are controlled by a single temperature control system which may be operated in automatic or manual mode.

OPERATION

In order for the air-conditioning system to operate, one or both engines must be running, and the applicable bleed air switch must be ON.

As shown in Figure 11-1, hot engine bleed air is routed through the opened bleed air valve and is regulated for volume as it passes through the valve. Once through the valve, the air goes to the mixing valve and the air cycle machine.

The air cycle machine contains a cooling turbine and an air-to-air heat exchanger. Hot bleed air first passes across the heat exchanger which uses ram air as the cooling medium. The pre-cooled bleed air then enters the cooling turbine which can cool the air to 30° to 60° F below outside air temperature.



Figure 11-2. Typical Eyeball Outlet

From the air cycle machine, the cold bleed air goes to a water separator which dehumidifies it.

Some of the dehumidified cold air is expelled through the eyeball outlets (Figure 11-2). The rest is mixed with hot bleed air from the mixing valve and becomes conditioned air. The conditioned air exits from the conditioned-air ducts and the pilots' footwarmers.

TEMPERATURE CONTROL SYSTEM

A mode selector switch and an automatic temperature control (Figure 11-3) are installed on the copilot's switch panel. The mode selector has four positions: OFF, AUTO, HOT, and COLD. In the OFF position, all power is removed from the temperature control system. In the HOT position, the mixing valves open fully, allowing the maximum amount of hot air into the conditioned air ducts. In the COLD position, the mixing valves close fully, resulting in only cold air and no hot air entering the ducts. In the AUTO position, power is applied to the temperature controller, allowing this unit to maintain the cabin air temperature.

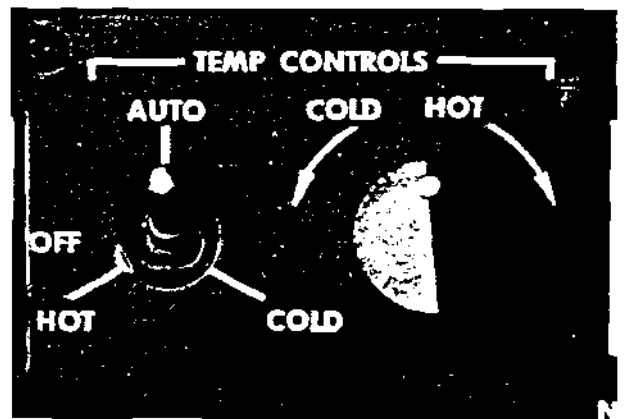


Figure 11-3. Temperature System Controls



cockpit Air Control

Two COCKPIT CONDitioned AIR knobs (Figure 11-4) on the pilot and copilot switch panels regulate the volume (but not the temperature) of the conditioned air delivered to the cockpit footwarmers. On later airplanes, conditioned air is also available through two adjustable eyeball outlets.

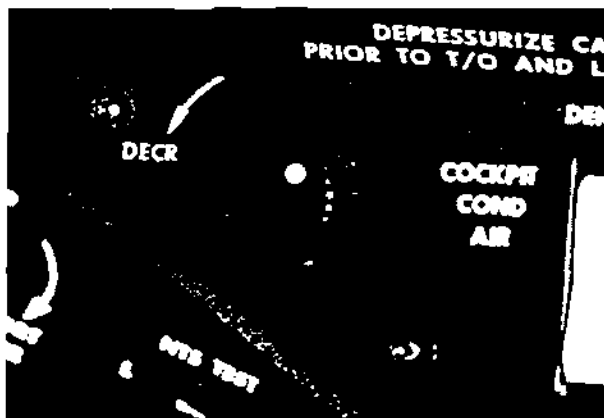


Figure 11-4. Cockpit Conditioned Air Knobs

Air Conditioning Control

Operation of the temperature control system is shown in Figure 11-5. The temperature

controller monitors temperatures in the cabin and in each conditioned-air duct. If the mode selector is in AUTO, the temperature controller responds to commands from the automatic temperature control by modulating the mixing valve in each bleed-air duct toward open (hot) or closed (cold).

As long as the mode selector remains in AUTO, a fan draws cabin air over a temperature sensor in the controller, which then maintains cabin temperature at a level selected by the temperature control knob (Figure 11-3). When the mode selector is not in AUTO, the automatic temperature control is bypassed, and the mixing valves can be modulated by moving the mode selector to HOT or COLD.

Cabin Cold Air Control

Two knobs located at floor level behind the copilot's seat control the volume of cold air delivered through the eyeball outlets in the cabin. Formerly optional but now standard equipment, the knobs operate a pair of cable-controlled butterfly valves in the cold-air ducts. When opened, the valves allow much of the cold eyeball air to be exhausted overboard after circulating through the interior of the airplane. By pulling the knobs outward, the valves can be partially or fully closed, forcing more cold air through the eyeball outlets.

FRESH AIR FAN SYSTEM

The FRESH AIR switch, labeled "NORM," "OFF," and "OVERRIDE," located on the copilot's switch panel operates the fresh air fan system (Figure 11-6). With the switch in NORMAL, a blower under the floor of the forward baggage compartment circulates ambient air through the cockpit when the airplane is on the ground, provided the nonessential bus is powered.

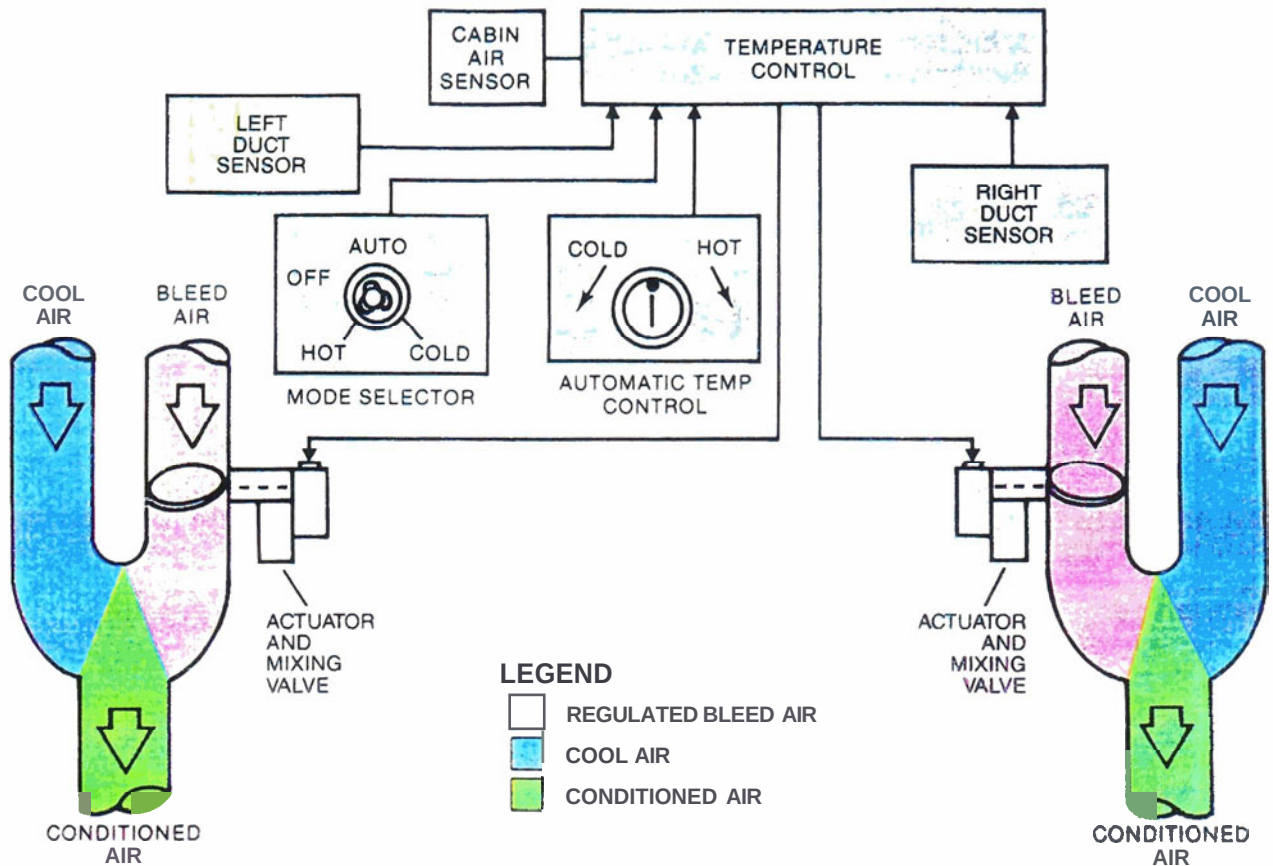


Figure 11-5. Conditioned-Air System Control Diagram

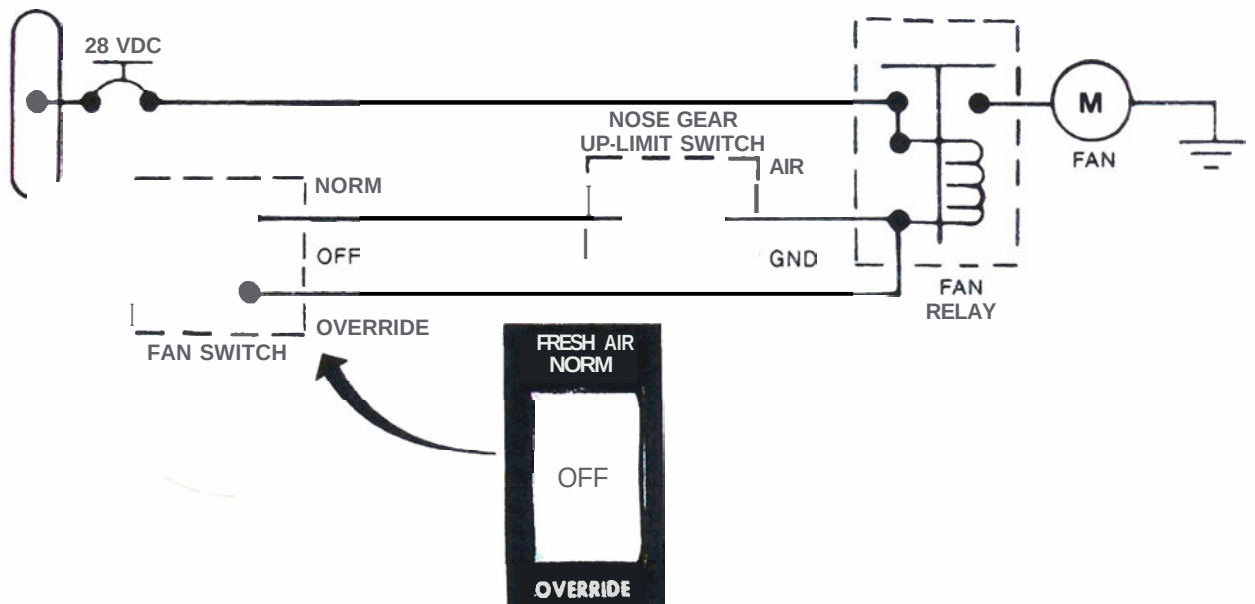
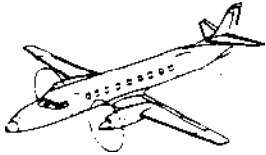


Figure 11-6. Fresh Air Fan System Block Diagram



The blower automatically stops when the nose gear retracts. Placing the switch in **OVER-RIDE** bypasses the the nose gear up-limit switch to permit cockpit ventilation during unpressurized flight. The system should not be operated in **OVERRIDE** while the airplane is pressurized.

FREON AIR-CONDITIONING SYSTEM (OPTIONAL)

GENERAL

An optional Freon air-conditioning system consisting of an electrically powered compressor/condenser unit and two evaporators with fans provides ground cooling while the engines are not operating. The system also supplements the standard air-conditioning system after engine start. The Freon system may be used below 17,500 feet MSL, provided a ground power unit or both generators are powering the nonessential bus. The evaporator fans can be used independently of the air-conditioner.

OPERATION

Control of the system is provided by three **FREON AIR CONDITIONING** rocker switches

on the copilot's switch panel (Figure 11-7). The air conditioning switch has three positions: **AIR CONDITION**, **OFF**, and **FAN**. In the **AIR CONDITION** position, the compressor/condenser forces cold air through the fore and aft evaporators and into the cabin through outlets at the front and rear of the cabin ceiling. A green light near the switch illuminates to indicate operation of the compressor/condenser unit. The **FAN** position can be used if air circulation without cooling is desired. The two other rocker switches control the forward and aft evaporator fans and can be positioned to either **HI** or **LOW** to select fan speed. The fans operate when the air conditioner switch is in the **AIR CONDITION** or **FAN** position.

AUXILIARY GROUND HEAT SYSTEM (OPTIONAL)

An auxiliary cabin heater for ground use only is available as an option. The heater, controlled by a two-position rocker switch labeled "**AUX HEAT**" and "**OFF**" on the copilot's side console (Figure 11-8), will operate only if a ground power unit is powering the electrical system. A green light near the switch illuminates when the heater is in use. Heater ducts are located at floor level in the front and rear of the cabin.



SA-227 PILOT TRAINING MANUAL

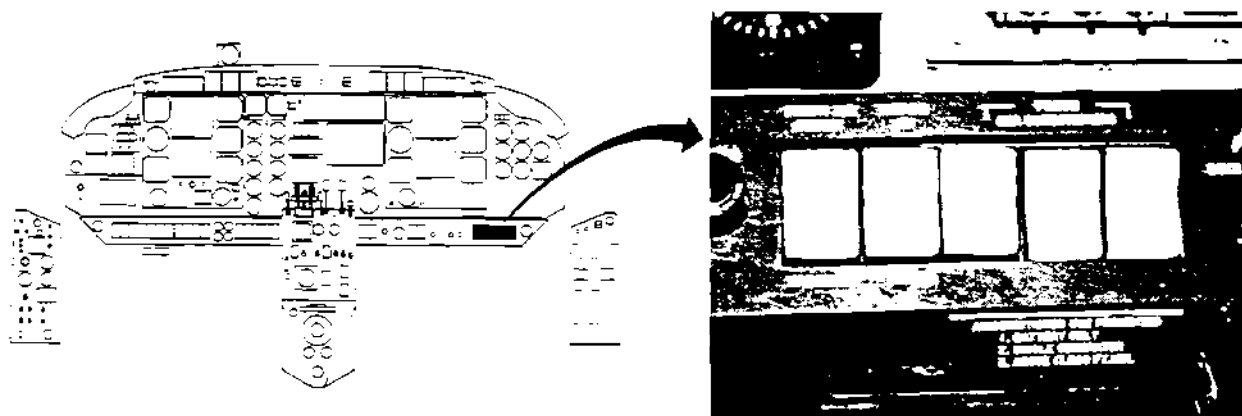


Figure 11-7. Freon Air-Conditioning System Controls

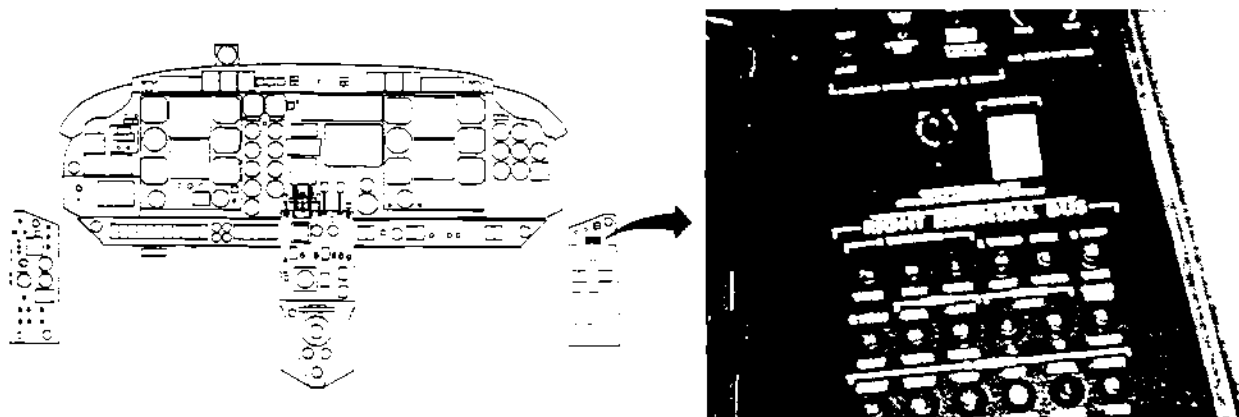
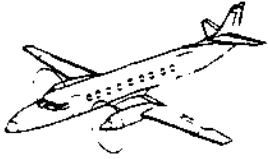


Figure 11-8. Auxiliary Ground Heat Control



QUESTIONS

1. The temperature control system maintains _____ temperature at a level selected by the _____.
A. Cockpit, crew
B. Cabin and cockpit, crew
☒ C. Cabin, crew
☐ D. Cabin, mode selector switch
2. What is bypassed when the HOT or COLD position is selected by the mode selector switch?
☒ A. Automatic temperature control
B. Bleed air valves
C. Mixing valves
D. Temperature controller
3. To increase or decrease the conditioned air flow to the pilot's foot warmers use the:
4. FRESH AIR FAN switch
☒ B. COCKPIT COND AIR knobs
C. TEMPERATURE CONTROL knob
D. mode selector switch
4. With the mode selector switch in the _____ position, the mixing valves close fully, resulting in no hot air entering the conditioned air ducts.
A. HOT
B. AUTO
C. OFF
☒ D. COLD
5. With the FRESH AIR switch in NORM, the blower will circulate air to the cockpit:
A. If either bleed valve is open
B. If either bleed valve is open and the associated engine is operating
C. When the airplane is on the ground
D. Whenever electrical power is available



CHAPTER 12

PRESSURIZATION

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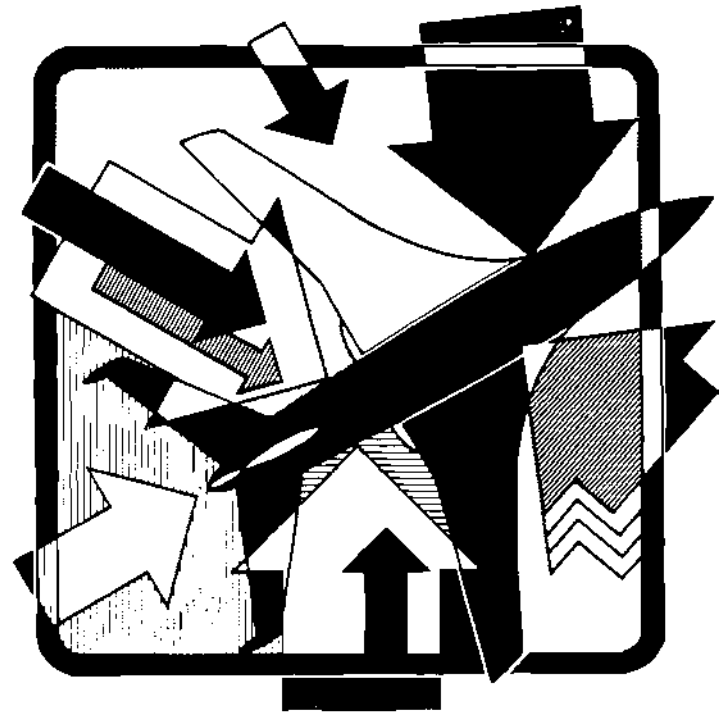
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CHAPTER 12

PRESSURIZATION



INTRODUCTION

The pressurization system maintains the cabin of the airplane (within specified limits) at any selected pressure altitude equal to or lower than the airplane altitude. During normal operation, the system automatically controls the cabin pressure as well as the rate-of-pressure changes. Safety features prevent the cabin pressure from exceeding maximum limits and also relieve negative pressure (cabin pressure less than ambient pressure). A combination safety and dump valve is provided to manually dump cabin pressure and prevent overpressurization.

GENERAL

The airplane is pressurized by controlling the volume of conditioned air being exhausted from the cabin through the outflow valve. Either air-conditioning system alone can maintain cabin pressurization. Operation of the air-conditioning system is covered in Chapter 11, "Air Conditioning."

The pressurization system develops a normal maximum cabin differential pressure of 7.0 psi, allowing a sea level cabin pressure altitude at airplane altitudes up to 16,800 feet gradually increasing to 7,000 feet at an airplane altitude of 30,000 feet.



Controls are provided to select the desired cabin altitude and to adjust the rate of cabin pressure changes. Indicators display the cabin altitude, differential pressure, and the rate of change.

The entire fuselage, with the exception of the nose baggage compartment, is pressurized. Figure 12-1 depicts the pressurized areas.

COMPONENTS DESCRIPTION AND OPERATION

GENERAL

The major components of the pressurization system consist of an outflow valve, pneumatic relay, cabin pressure selector (mode selector), pressure controller, manual control valve, and safety (dump) valve (Figure 12-2).

OUTFLOW VALVE

The outflow valve (Figure 12-3), installed on the aft pressure bulkhead, is used to control the flow of air out of the airplane pressure vessel. The valve responds to pressure commands supplied by the pressurization control system through the pneumatic relay. The outflow valve is opened by vacuum; it is closed by spring force. If differential pressure between the cabin and the ambient air exceeds approximately 7.15 psi, the valve opens, regardless of the command being supplied by the pressure control system. The outflow valve also opens to relieve negative pressure (ambient air pressure greater than cabin pressure).

PNEUMATIC RELAY

Because of the distance from the cockpit to the outflow valve, the pneumatic relay (Figure 12-3) is used to speed up the reaction time of the outflow valve to commands provided by the pressurization control system. The pneumatic relay repeats the input from the pressure control system and uses vacuum to open the outflow valve.

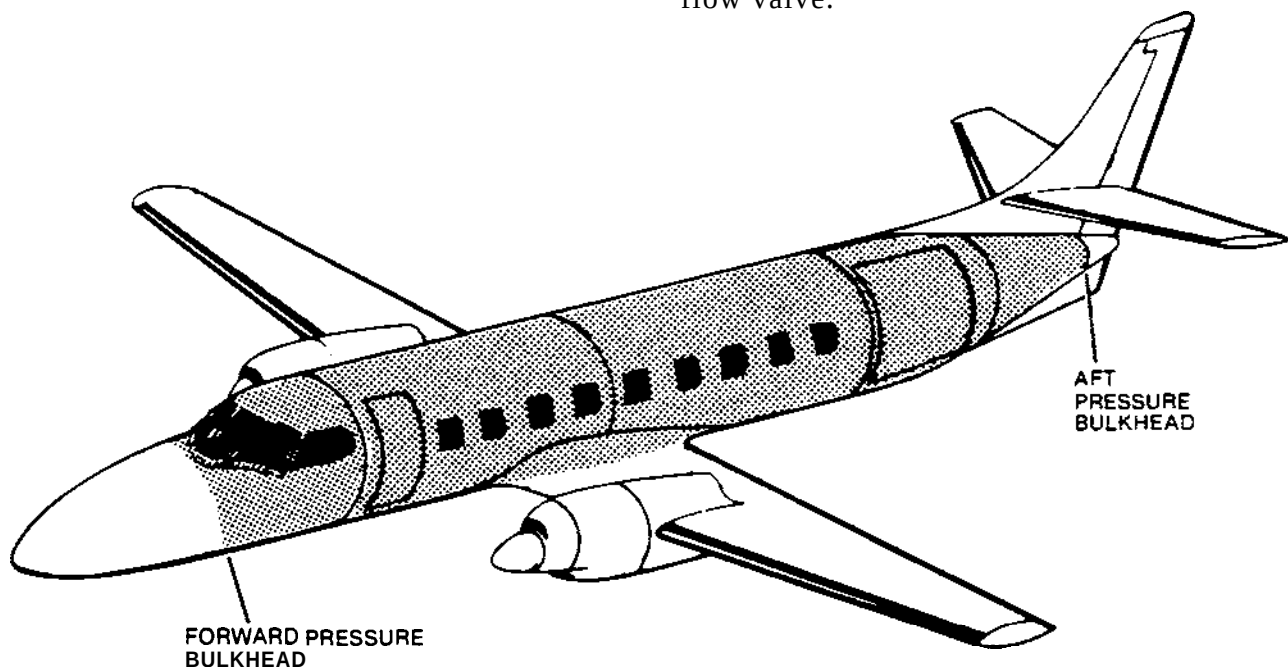


Figure 12-1. Pressurized Areas

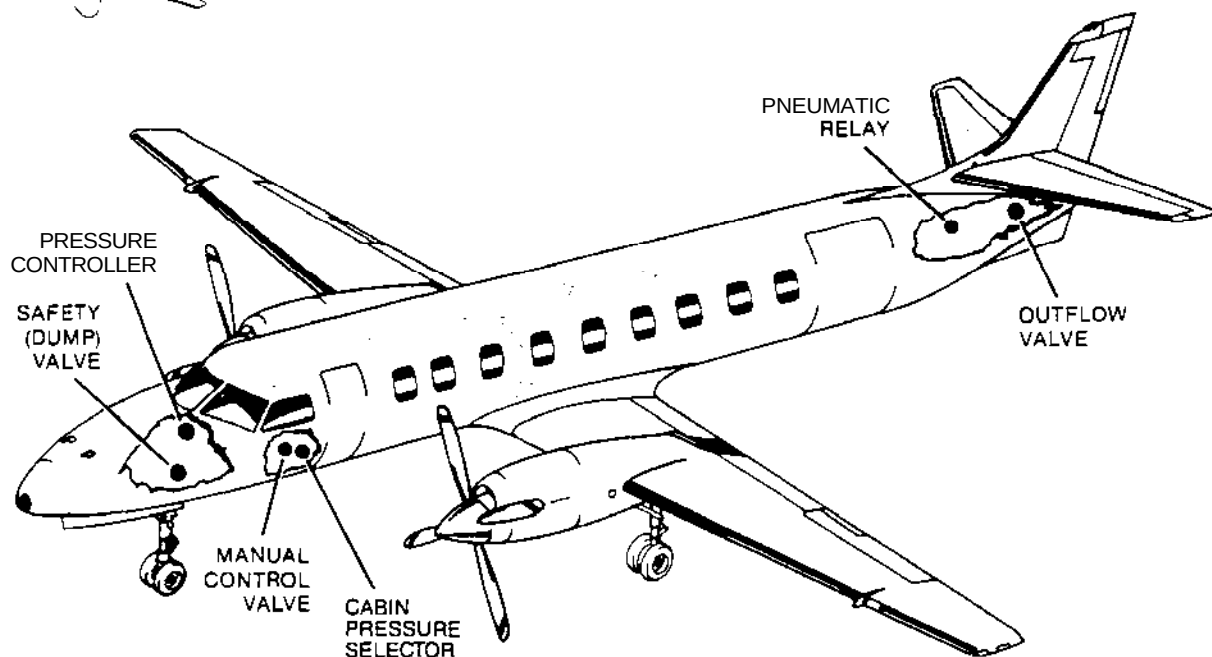


Figure 12-2. Pressurization System Major Components

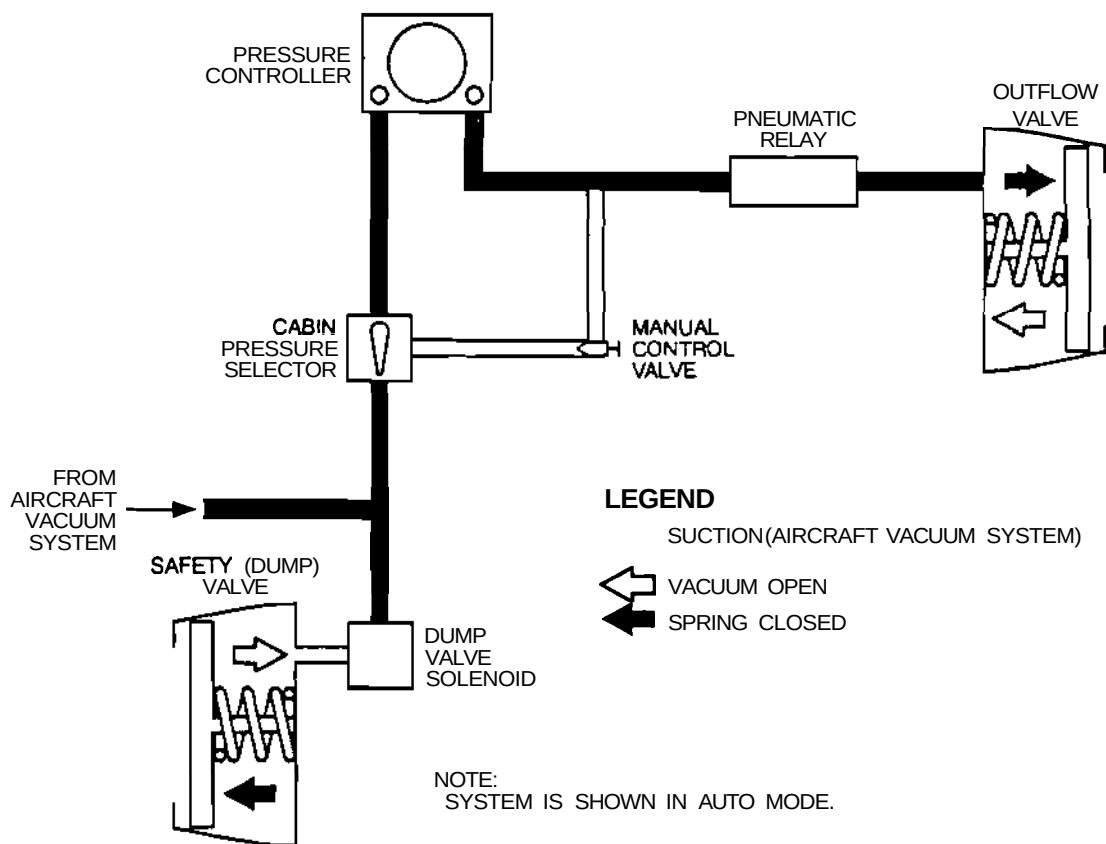


Figure 12-3. Pressurization System Schematic



PRESSURE CONTROL SYSTEM

Cabin Pressure Selector (Mode Selector)

A two-position cabin pressure selector valve on the left console is used to select the source of vacuum signal to the pneumatic relay. In the AUTO position, the pneumatic relay is controlled by the automatic pressure controller. In the MANUAL position, the pneumatic relay is controlled by the adjacent CABIN PRESS MANUAL CONTROL knob (Figure 12-4).

Automatic Mode

In the automatic mode, the cabin pressure controller (Figure 12-5) on the instrument panel regulates the outflow valve through the pneumatic relay to vary cabin pressure altitude. Desired cabin altitude between -1,000 feet and +10,000 feet is selected by rotating the CABIN ALT knob on the face of the controller. A small window in the bottom of the controller indicates the airplane altitude in thousands of feet at which maximum differential pressure (7.0 psi) is reached.

The RATE control knob allows the pilot to select a rate-of-cabin pressure change within the limits of approximately 50 to 2,000 fpm. A white triangular-shaped marker set directly on an arc above the control knob indicates a setting of approximately 500 fpm.

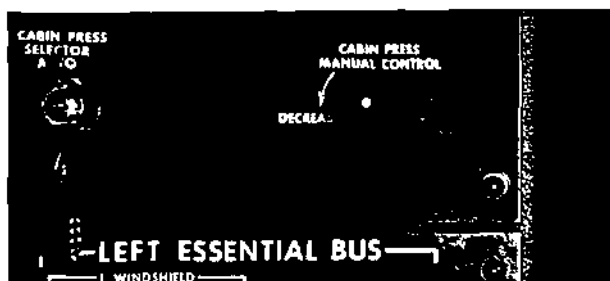


Figure 12-4. Cabin Pressure Selector and Manual Control

Manual Mode

If the automatic controller malfunctions, or if rapid depressurization is desired, the manual mode can be selected. With the CABIN PRESS SELECTOR (Figure 12-4) in MANUAL, rotation of the CABIN PRESS MANUAL CONTROL operates a needle valve, providing a very coarse adjustment of cabin altitude and rate of change. Moving the manual control counterclockwise toward DECREASE opens the outflow valve, decreases differential pressure, and increases cabin altitude. Clockwise rotation of the manual control increases differential pressure and decreases cabin altitude.

The normal position of the CABIN PRESS MANUAL CONTROL is full clockwise to ensure that maximum differential pressure will be preselected in case the pilot needs to use the manual control valve at high altitude. However, excessive force can damage the manual control valve.

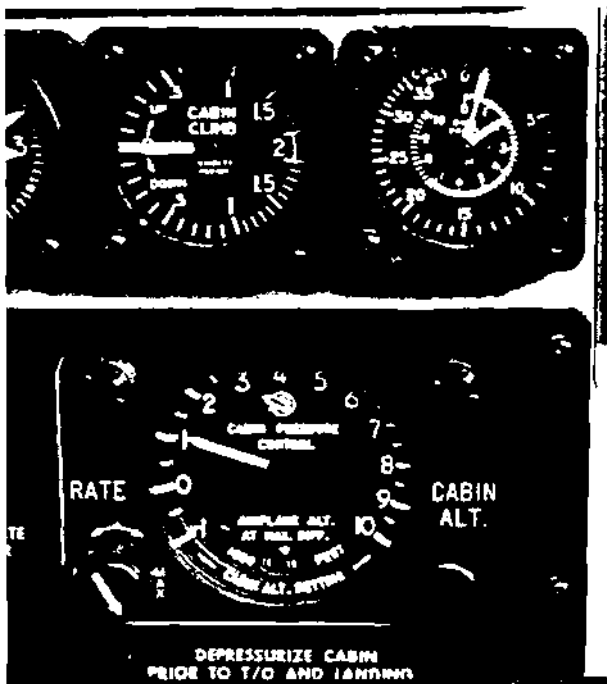
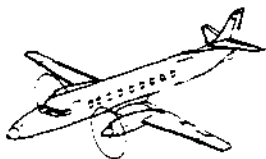


Figure 12-5. Cabin Pressure Controller and Indicators

CABIN PRESSURE INDICATORS

A cabin rate-of-climb indicator and a cabin differential pressure indicator (Figure 12-5) are located immediately above the automatic pressure controller.

The cabin rate-of-climb indicator displays the speed at which the cabin is pressurizing or depressurizing. The cabin differential pressure indicator displays both the cabin pressure altitude and the pressure differential between cabin and ambient pressure. Cabin altitude is indicated on the outer scale; differential pressure is indicated on the inner scale.

PRESSURIZATION DUMP SYSTEM

Safety (Dump) Valve

The safety dump valve (Figure 12-3) is located on the left side of the forward pressure bulkhead. The valve is opened by vacuum and is spring-loaded closed. Vacuum is applied by opening a solenoid-operated control valve. When the control valve closes, the vacuum is relieved slowly to prevent a pressure "bump" immediately after takeoff. If the outflow valve fails closed, the safety (dump) valve opens to relieve pressure when cabin pressure exceeds approximately 7.25 psi.



Dump Valve Electrical Controls

The dump valve solenoid is controlled by the CABIN PRESS switch on the copilot's lower switch panel (Figure 12-6). The red switch has two positions labeled "NORM" and "DUMP".

Placing the switch in the NORMal position energizes the dump valve solenoid through the right landing gear squat switch. When energized, the dump valve solenoid opens, and

vacuum is applied to the dump valve. As a result, the dump valve opens and the airplane remains unpressurized while on the ground.

After takeoff, opening of the squat switch deenergizes the dump valve solenoid, causing the dump valve to close, allowing normal pressurization of the cabin. Placing the CABIN PRESS switch in the DUMP position bypasses the squat switch and energizes the dump valve solenoid. This applies vacuum to the dump valve and dumps the cabin pressure.

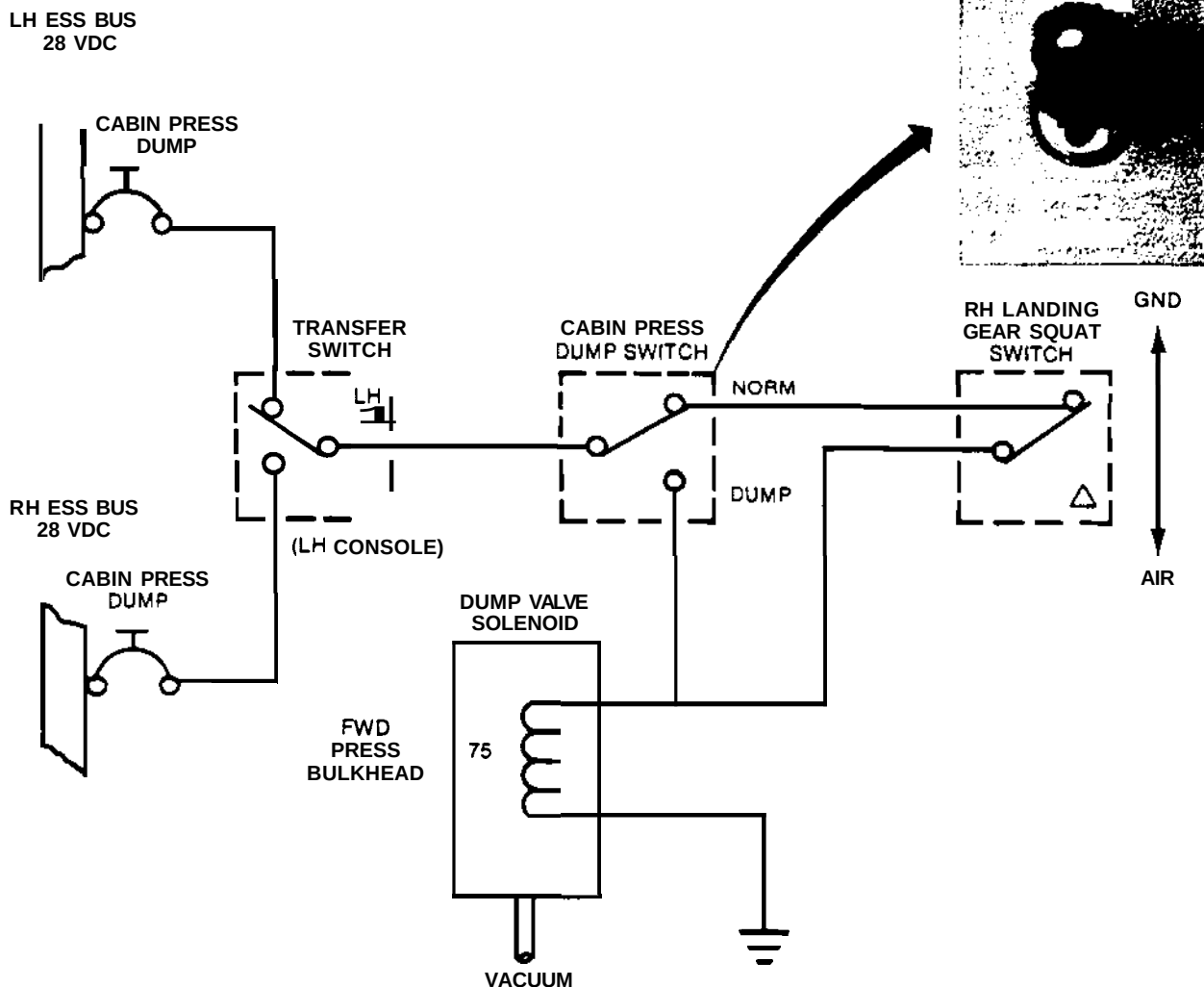


Figure 12-6. Cabin Pressure Dump System Diagram



28.5-VDC power for the dump solenoid is available from either essential bus through the CAB PRESS DUMP switch on the ESSENTIAL BUS TRANSFER panel (see Chapter 10, Figure 10-3).

CABIN ALTITUDE WARNING SYSTEM

The cabin altitude warning system informs the pilot that the cabin altitude has exceeded approximately 11,000 feet and that supplemental oxygen is required. A pressure switch causes illumination of the amber CABIN ALTITUDE light on the annunciator panel (see Appendix B).

SYSTEM OPERATION

With either engine operating and the respective bleed-air valve open, conditioned air enters the cabin. On the ground, the landing gear safety (squat) switch opens the dump valve. Thus, cabin air passes through the valve and to the atmosphere. At lift-off, the dump valve closes if the CABIN PRESS switch on the copilot's switch panel is in the **NORMal** position.

If the CABIN ALT selector is set to the airport elevation prior to takeoff, the cabin pressure is maintained at that elevation during flight. This is possible as long as the airplane does not ascend to an altitude which would create a differential pressure of more than 7.0 psi. If the airplane continues to climb, the pressure controller causes the cabin to climb enough to maintain a constant differential pressure of 7.0 psi. Normally, after takeoff, the CABIN

ALT selector is set for maximum expected cruise altitude +1,000 feet, or landing field pressure altitude +200 feet, whichever is higher.

The cabin pressure rate of change is controlled by the RATE control knob. The rate of change is adjustable from 50 to 2,000 fpm. The RATE control setting determines how quickly or slowly the cabin altitude is reached. Once reached, the cabin altitude is maintained at a constant pressure. The reduced size of the outflow opening restricts the flow of cabin air exiting the fuselage in order to maintain an essentially constant cabin pressure. In actual practice, the outflow valve never closes completely but is modulated by negative pressure from the pneumatic relay opposed by spring tension and atmospheric pressure.

Prior to landing, the desired rate of change and the desired cabin altitude at the destination airport should be set on the cabin pressure controller. The controller now functions to control the cabin pressure rate of change in accordance with the selected rate until either the selected pressure altitude is reached or the cabin becomes unpressurized. The airplane must be landed unpressurized.

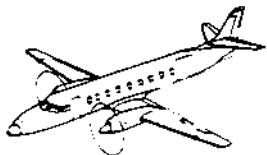
LIMITATIONS

- Maximum normal cabin differential pressure is 7.0 psi.
- = The safety valve is set at 7.25 psi.
- The cabin must be depressurized during takeoff and landing.



QUESTIONS

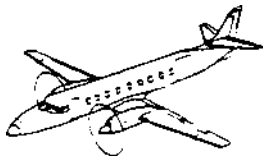
1. One of the functions of the cabin pressure controller is to control cabin pressure to a selected value within the isobaric range of -1,000 feet to:
 - A. 15,000 feet
 - ☒ B. 10,000 feet
 - C. 5,000 feet
 - D. 1,000 feet
2. The RATE control knob allows the pilot to select a rate-of-cabin pressure change within the limits of approximately:
 - A. 30 to 2,000 fpm
 - B. 40 to 4,000 fpm
 - ☒ C. 50 to 2,000 fpm
 - D. 50 to 5,000 fpm
3. One of the functions of the _____ is to limit the cabin pressure differential to 7.25 psi in case of a(n) _____ failure.
 - A. Dump valve, bleed air
 - B. Pressure controller, outflow valve
 - ☒ C. Dump valve, outflow valve
 - D. Pressure controller, bleed air
4. With the CABIN PRESS SELECTOR in the MANUAL position, cabin pressure is controlled with the:
 - A. RATE control knob
 - B. CABIN ALT selector knob
 - C. CABIN AIR control knob
 - ☒ D. CABIN PRESS MANUAL CONTROL knob



CHAPTER 13 HYDRAULIC POWER SYSTEMS

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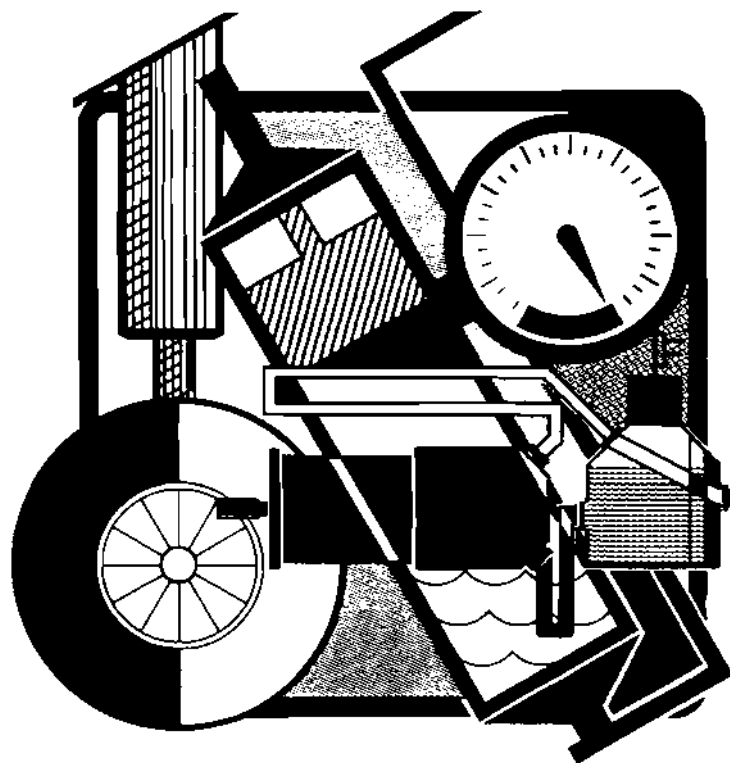
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CHAPTER 13

HYDRAULIC POWER SYSTEMS



INTRODUCTION

The main hydraulic system is pressurized by two engine-driven pumps, one on each engine. The system provides pressure for actuation of the landing gear, flaps, and nose-wheel steering on all aircraft, and for the power brakes on those aircraft so equipped. Aircraft without power brakes have a separate hydraulic system for braking. (Refer to Chapter 14, "Landing Gear and Brakes.")

The auxiliary system, pressurized by a hand pump, supplies pressure only for emergency extension of the landing gear.

GENERAL

The output of two variable-volume pumps is manifolded together to pressurize the main hydraulic system. Either pump is capable of actuating all the subsystems but at a reduced rate. The pumps draw MIL-H-83282 (Brayco) fluid from a reservoir through shutoff valves controlled from the cockpit. Fluid returning

from the subsystems is routed through a filter prior to entering the reservoir.

Two warning lights on the annunciator panel warn of low pump pressure or pump failure. Pressure surges in the system are dampened by an accumulator.



A hydraulic hand pump, drawing reserve fluid from the bottom of the reservoir, pressurizes the auxiliary system for emergency extension of the landing gear. One electrically operated hydraulic pressure indicator shows main or auxiliary pressure, whichever is higher.

MAIN HYDRAULIC SYSTEM

GENERAL

The reservoir is pressurized by regulated 31-psi engine bleed air to assure adequate supply flow to the pumps during peak flow demands. The engine-driven pumps draw fluid from above the standpipes in the reservoir, reserving a quantity of fluid for hand pump operation during emergency extension of the landing gear.

Supply fluid to the engine-driven pumps passes through shutoff valves controlled by two HYDR SHUT OFF switches on the center pedestal (Figure 13-1). These switches are normally in the OPEN position and are CLOSED only in the event of fire, engine shutdown in flight, or when maintenance is to

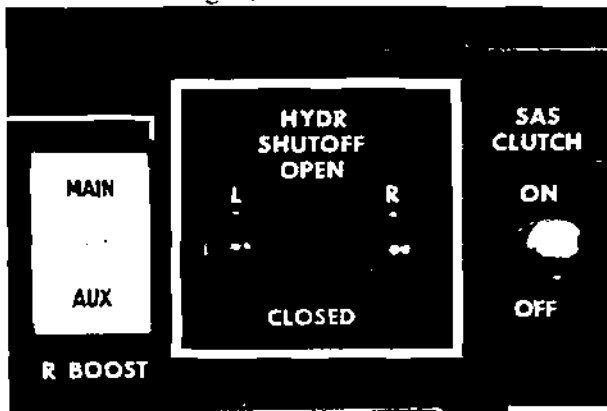


Figure 13-1 Hydraulic Shutoff Switches

be performed. If either of the switches is moved from one position to the other, an amber L HYD or R HYD position annunciator disagreement light (see Appendix B) will illuminate until the valve is in the proper po-

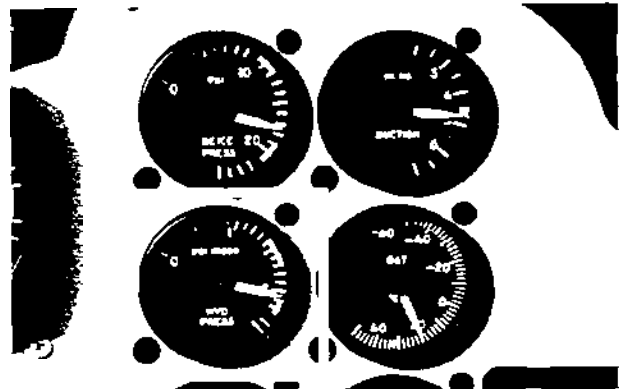


Figure 13-2. HYD PRESS Indicator

sition. When the valve is fully opened or closed, the light will extinguish. If the light does not extinguish, the valve has not fully opened or closed and may not have moved at all.

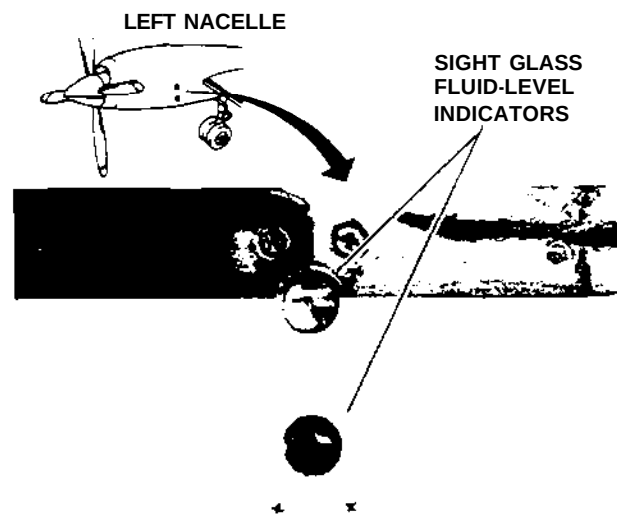


Figure 13-3. Hydraulic Reservoir Sight Glass (Typical)

A shuttle valve allows the HYD PRESS indicator (Figure 13-2) to serve both the main and auxiliary hydraulic systems. Red L HYD PRESS and R HYD PRESS warning lights (see Appendix B) illuminate when pump output is low.

Reservoir fluid level can be checked by observing the sight glass fluid-level indicators (Figure 13-3). The FULL line on the upper



indicator shows the normal level. If no fluid is visible in the upper indicator, maintenance action is required.

OPERATION

When the right engine is started, the pump draws fluid from above the reservoir standpipe through the open shutoff valve. As pres-

sure increases the R HYD PRESS light extinguishes (Figure 13-4).

Pressure will stabilize at 2,000 psi. The system is now capable of actuating all subsystems, but at a reduced rate. If pump malfunction allows pressure to become excessive, the relief valve opens. If the pump fails, the R **HYD PRESS** light will illuminate.

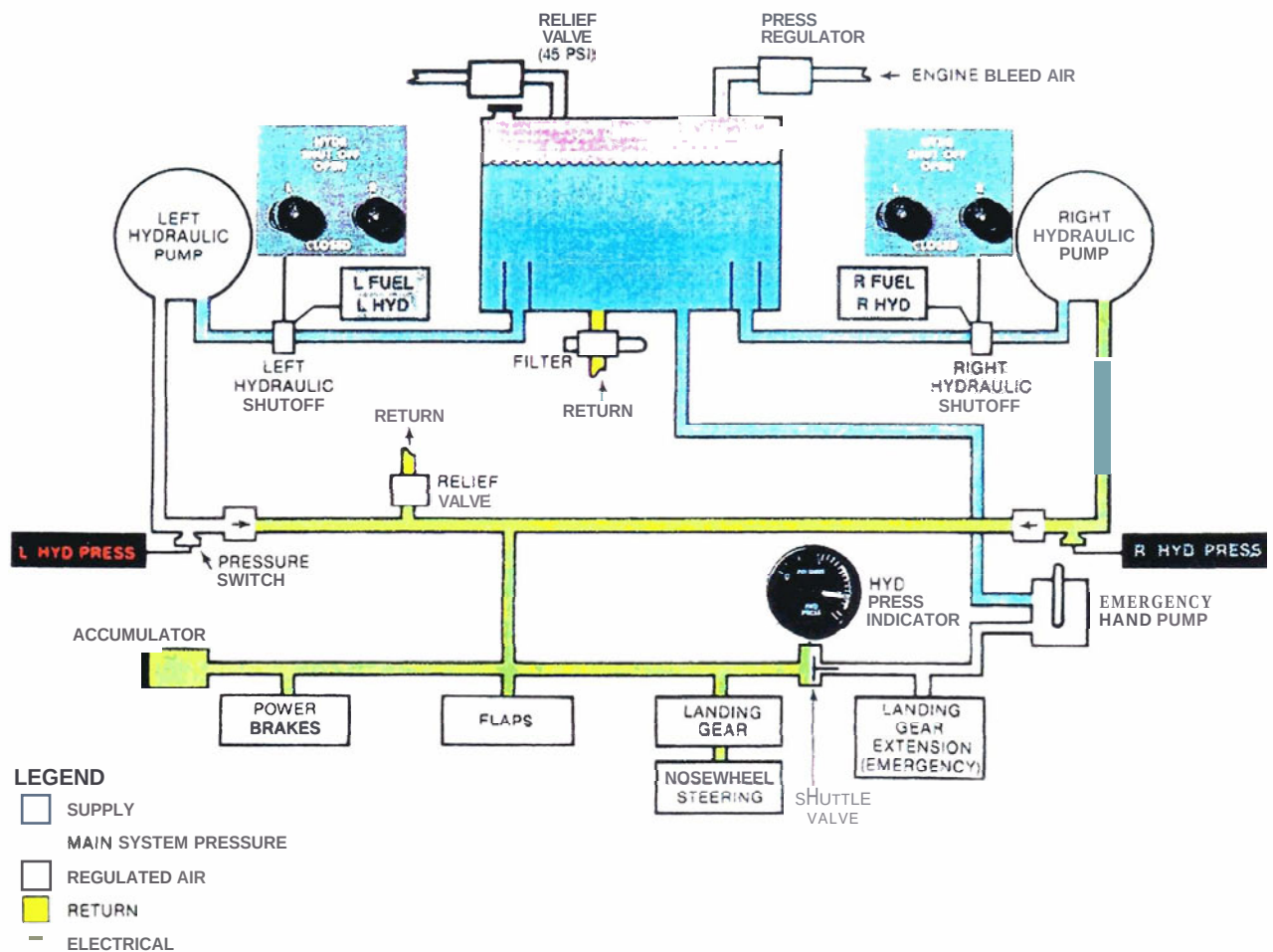


Figure 13-4. Right Hydraulic Pump Operating



As the left engine is started, the pump draws fluid from the other standpipe in the reservoir, through the open shutoff valve, and adds its pressurized flow to the system, extinguishing the L HYD PRESS light (Figure 13-5).

The system is now at full operating potential. If either pump fails, the applicable low-pressure light illuminates as pressure decreases. The remaining engine-driven pump continues to supply 2,000 psi to the system, as indicated by the HYD PRESS indicator.

AUXILIARY HYDRAULIC SYSTEM

GENERAL

The auxiliary hydraulic system, powered by the emergency hand pump, provides pressure for emergency extension of the landing gear. The gear cannot be retracted with the auxiliary system.

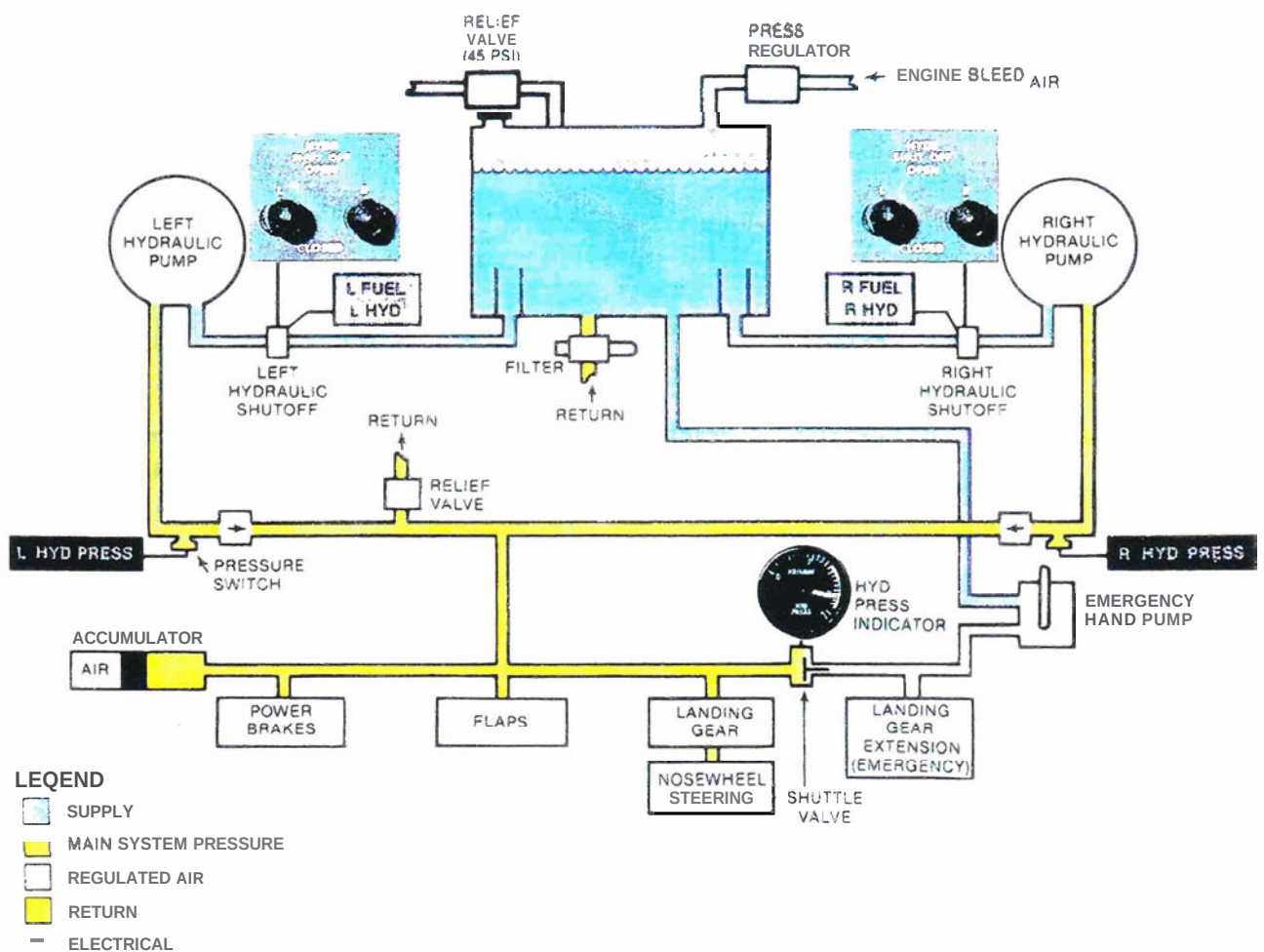


Figure 13-5. Both Hydraulic Pumps Operating



OPERATION

The hand pump handle is normally blocked by positioning the hand pump engage valve handle in the NORM GEAR position (Figure 13-6). Pulling the pip pin and rotating the handle 90° forward (counterclockwise) frees the hand pump for operation and positions the hand pump engage valve to direct pressure to the landing gear (Figure 13-7). Emergency landing gear operation is presented in Chapter 14, "Landing Gear and Brakes."

HYDRAULIC SUBSYSTEMS

Landing gear, brakes, nosewheel steering, and flaps are hydraulically powered. These systems are described in Chapter 14, "Landing Gear and Brakes," and Chapter 15, "Flight Controls."

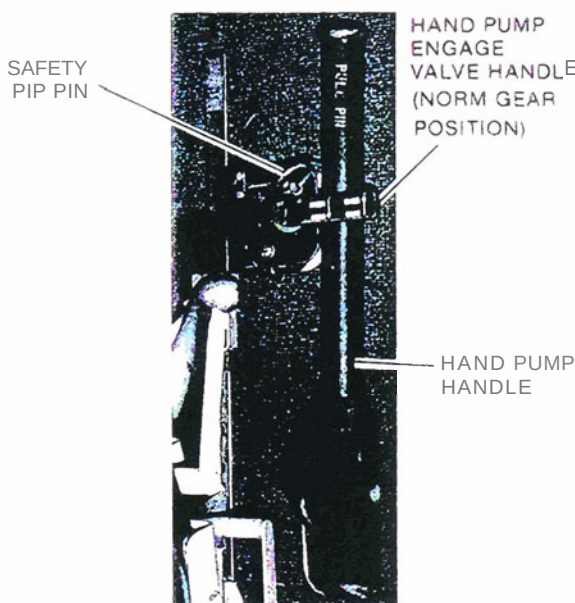


Figure 13-6 Auxiliary Hydraulic System Controls

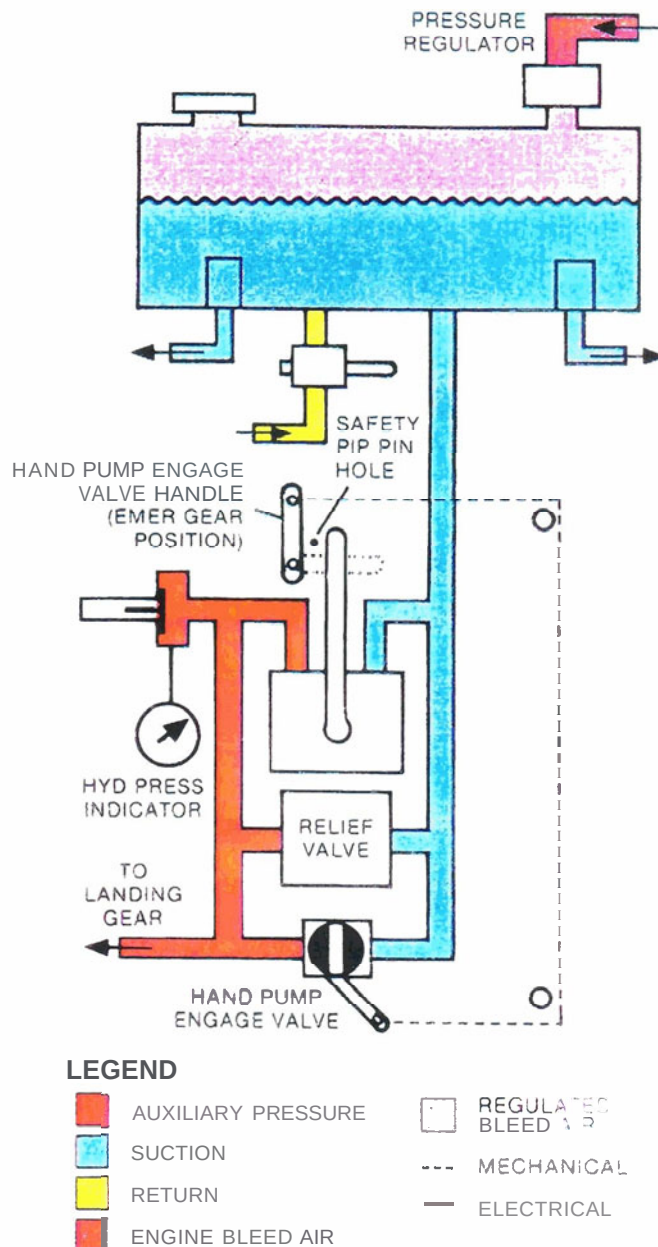
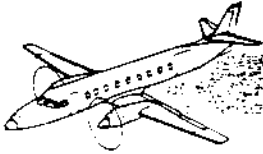


Figure 13-7. Auxiliary Hydraulic System



QUESTIONS

1. The L HYD PRESS warning light, illuminating during flight with the HYD PRESS indicator showing 2,000 psi, indicates:
 - A. The left engine-driven pump has failed or its pressure output is low.
 - B. The pressure switch controlling the light is malfunctioning.
 - C. The light is shorted out.
 - D. Either engine-driven pump may have failed.
2. The purpose of the accumulator in the hydraulic system is to:
 - A. Store pressure for emergency use
 - ☒ B. Dampen pressure surges
 - C. Pressurize the reservoir
 - D. None of the above
3. A HYDR SHUT OFF switch should be placed in the CLOSED position:
 - A. After engine shutdown on the ground
 - ☒ B. Only in case of engine fire, engine shutdown in flight, or when maintenance is to be performed
 - C. When a low-pressure (L HYD PRESS or R HYD PRESS) warning light illuminates
 - D. When reservoir fluid has been depleted
4. A hydraulic shutoff valve (L HYD or R HYD) disagreement light illuminates:
 - A. When the valve is fully open
 - B. When the valve is fully closed
 - C. When system pressure is low
 - ☒ D. When the valve is not in the position selected by the switch
5. When the hydraulic reservoir is properly serviced with fluid:
 - A. Both sight glass indicators show a solid mass of fluid.
 - B. The lower sight glass indicator shows a solid mass of fluid; no fluid can be seen in the upper indicator.
 - C. No fluid can be seen in either sight glass indicator.
 - ☒ D. The fluid level line will be at the FULL mark on the upper sight glass indicator.



CHAPTER 14

LANDING GEAR AND BRAKES

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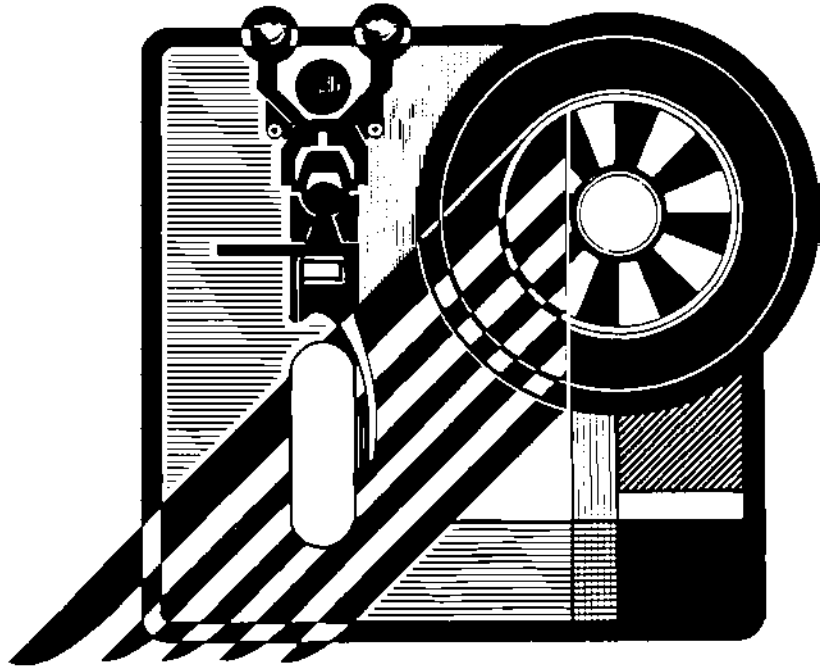
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CHAPTER 14

LANDING GEAR AND BRAKES



INTRODUCTION

The airplane has a dual-wheel, retractable, tricycle landing gear enclosed by mechanically actuated doors. Gear position and warning are provided by indicator lights and a warning horn.

The nosewheel steering system provides directional control while taxiing. It dampens nosewheel shimmy and casters freely when not engaged.

The standard braking system is manual; antiskid power brakes are optional.

GENERAL

The forward-retracting tricycle landing gear is normally controlled electrically and actuated by the two hydraulic actuators attached to each gear. All gear doors are linked to the gear and are mechanically actuated by gear movement. An emergency extension system is provided in the event of electrical and/or hydraulic failure.

Gear position indication is provided by lights on the instrument panel. In addition, a gear warning horn sounds intermittently if all three gear are not down and locked when flaps are extended more than one half travel or either power lever is retarded to flight idle.

Nosewheel steering is electrically controlled and hydraulically actuated to provide variable-authority directional control. When disengaged, it reverts to the caster mode and provides shimmy dampening.



Standard disc brakes are provided from a set of master cylinders independent of the main hydraulic system. An option provides power brakes and antiskid backed up by master cylinders. The power brake system is actuated by main hydraulic system pressure and controlled by master cylinder pressure.

LANDING GEAR

GENERAL

The gear is electrically controlled. The main landing gear assembly is shown in Figure 14-1.

Each gear is operated by two hydraulic actuators. Both actuators are pressurized during retraction; only one, during extension. All gears are locked down by an overcenter condition of the drag braces and are mechanically locked up when retracted. Unlocking of the gear from the retracted or extended position is accomplished by motion of the hydraulic actuators. The gear can also be manually released from the uplocks.

All gear doors are mechanically actuated by gear movement and are mechanically locked when closed. The main gear doors are closed with the gear extended or retracted. Nose gear doors remain open when the gear is ex-

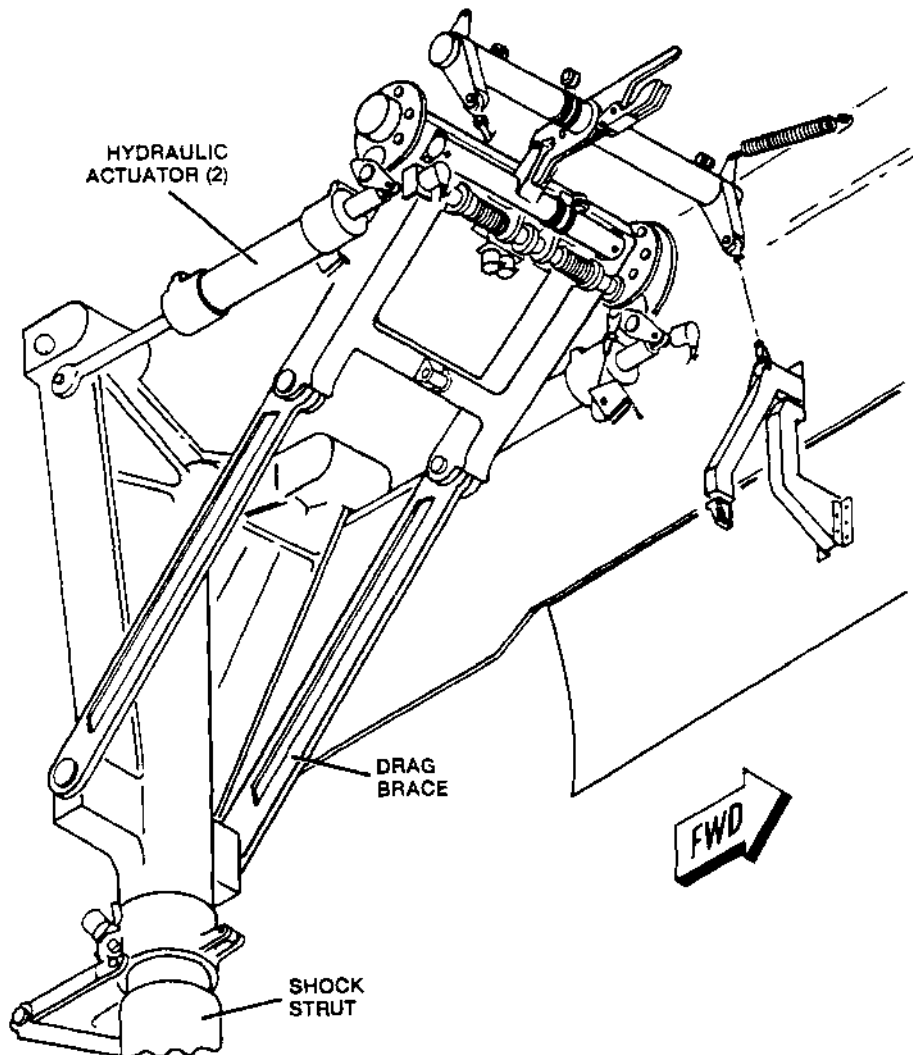
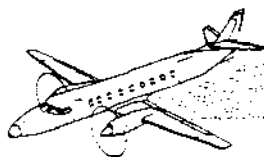


Figure 14-1. Main Gear Assembly (Typical)



tended. The main gear doors can be opened on the ground for preflight inspection.

The nose gear strut incorporates a mechanical device to center the nosewheels at lift-off.

CONTROLS AND INDICATORS

The landing gear is controlled by the **LANDING GEAR** lever on the center pedestal. Gear position is indicated by three green lights and three red lights on the landing gear position indicator (Figure 14-2). A warning horn and a **GEAR DOOR POSITION** light provide warning of abnormal conditions. (All annunciators are shown in Appendix B.)

Controls

The landing gear lever controls circuitry to the solenoid-operated gear selector valve. 28.5-VDC power from either the left or right essential bus is available through the **LDG GEAR CONTROL** switch on the **ESSENTIAL BUS TRANSFER** panel (see Chapter 2, Figure 2-14). On the ground, a solenoid-actuated locking device engages to hold the lever in the **DOWN** position. Airborne, a squat switch on the left main gear completes circuitry to release the locking device. If the solenoid fails or electrical power is lost, pushing down on the **LVR LOCK OVERRIDE** lever releases the locking device (Figure 14-2).

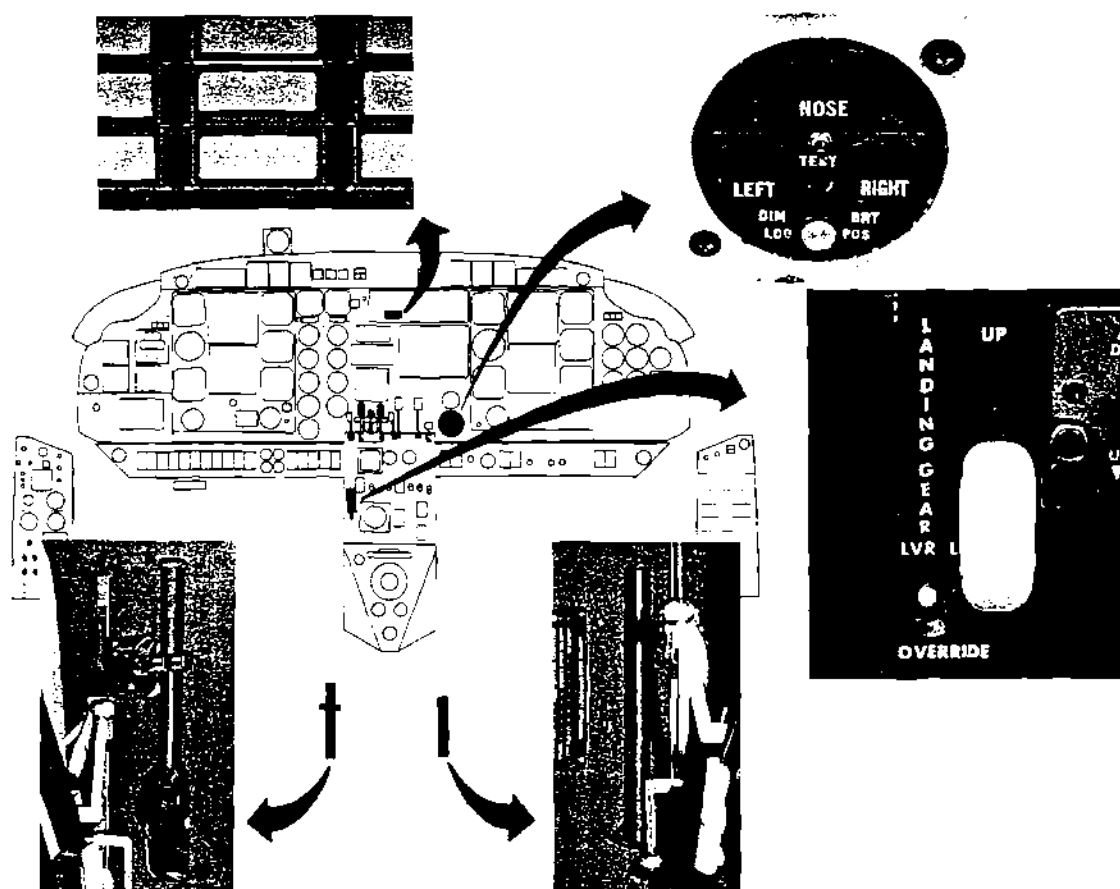


Figure 14-2. Landing Gear Controls and Indicators



Indicators

Electrical power for the gear indicator is available from either essential bus through the **LDG GEAR POS** switch on the **ESSENTIAL BUS TRANSFER** panel (see Chapter 2, Figure 2-14).

The landing gear position indicator features a green light and a red light for each gear. An illuminated green light indicates that the gear is down and locked. An illuminated red light indicates that the gear is in transit. With the gear up and locked, neither light is on.

Figure 14-3 shows indicator light displays for various gear conditions.

All six indicator lights are tested by pressing the **TEST** button on the position indicator panel. The intensity of the green lights can be varied with the dimmer knob (Figure 14-3).

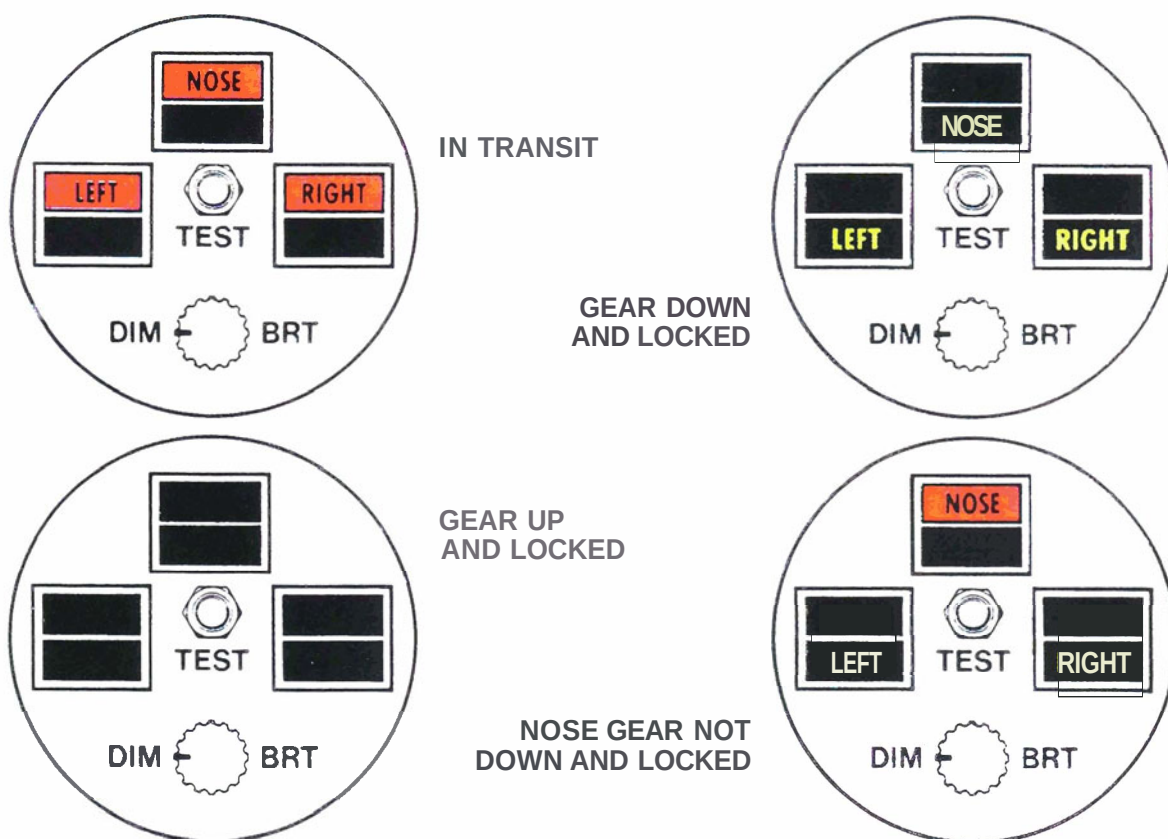


Figure 14-3. Gear Position Indications

Warning System

The landing gear warning system consists of an intermittent gear warning horn and a red **GEAR DOOR POSITION** light on the annunciator panel. The light is wired through a squat switch and illuminates when the airplane is on the ground and any main gear door

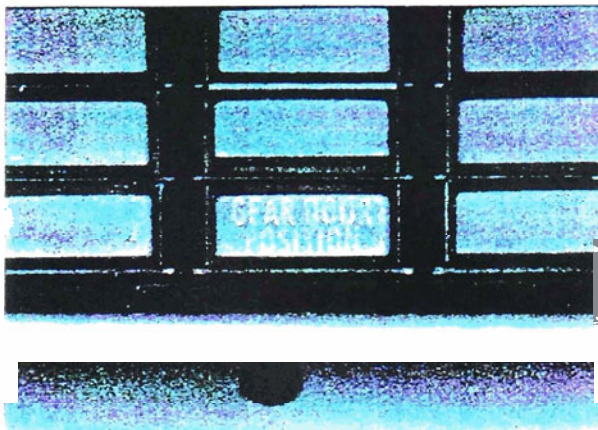


Figure 14-4. Gear Warning System



is not closed (Figure 14-4). The corrective action is to close the applicable gear door.

With any landing gear not down and locked, the gear horn sounds, provided one of the following conditions exists:

- Either power lever is retarded to flight idle.
- Flaps are lowered slightly beyond 1/2

The horn continues to sound until the condition is corrected.

OPERATION

Extension

Placing the landing gear lever DOWN energizes the gear selector valve to the gear extend position (Figure 14-5). Pressure is directed to one actuator at each gear. When the gear is down and locked, the selector valve remains energized and pressure is constantly

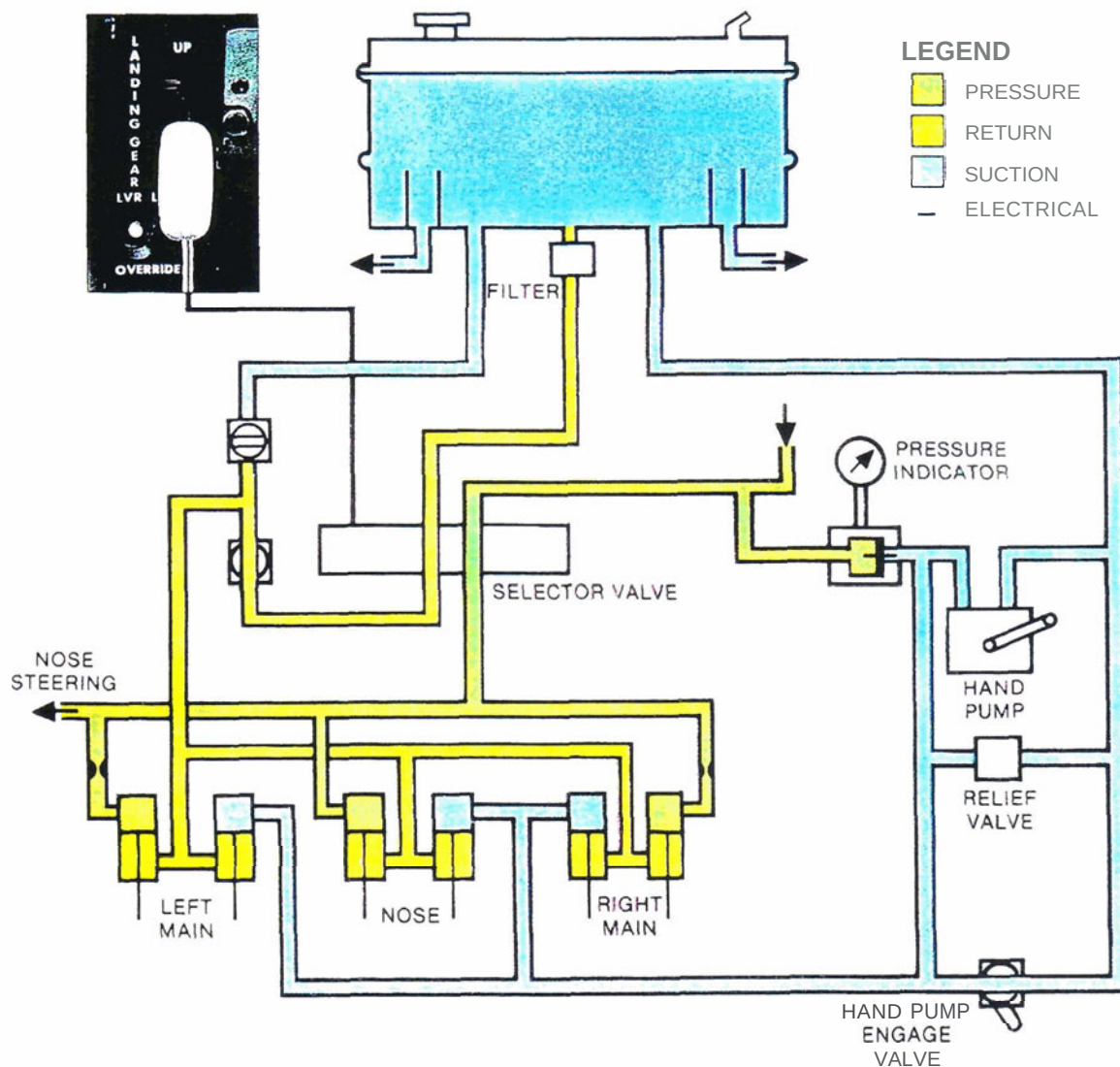


Figure 14-5. Normal Extension



applied as long as electrical power and hydraulic pressure are available. The gear drag braces move overcenter to lock the gear down,

the selector valve returns to neutral, forming a fluid lock. The gear is also held retracted by mechanical uplocks.

Retraction

Placing the landing gear lever in the UP position energizes the selector valve to retract the gear (Figure 14-61, provided the left main gear squat switch senses an airborne condition. Pressure is applied to both actuators on each gear. When all three gear are up and locked,

Emergency Extension

Emergency extension of the landing gear (Figure 14-7) is necessary if DC power or hydraulic pressure is lost, or if the gear selector valve malfunctions.

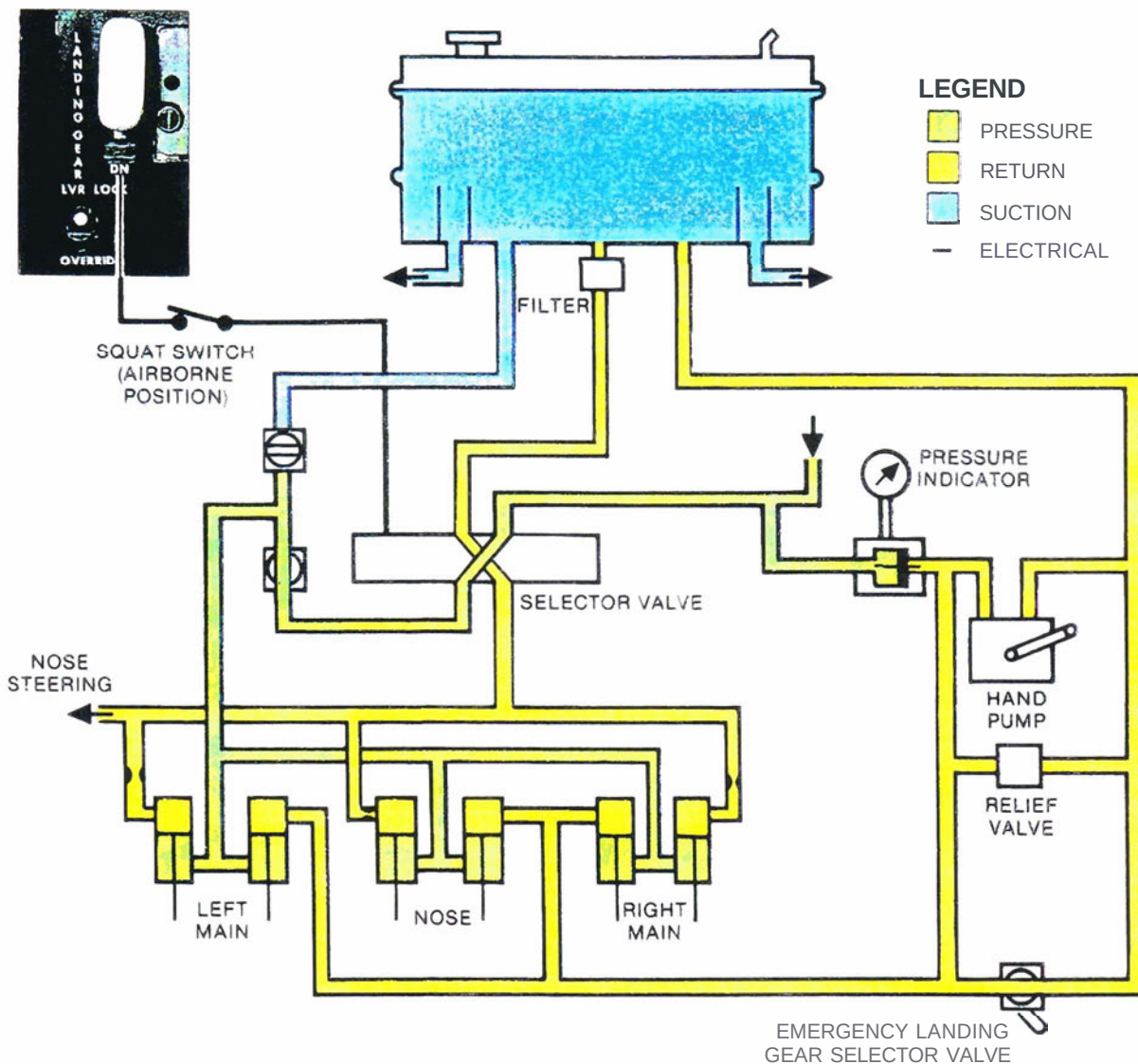


Figure 14-6. Normal Retraction



SA-227 PILOT TRAINING MANUAL

Prior to emergency extension of the landing gear, airspeed should not exceed gear extended speed, and the gear lever should be DOWN,

Moving the emergency gear release lever (Figure 14-7) 90° counterclockwise mechanically releases the gear uplocks and positions valves to bypass return fluid around the selector valve.

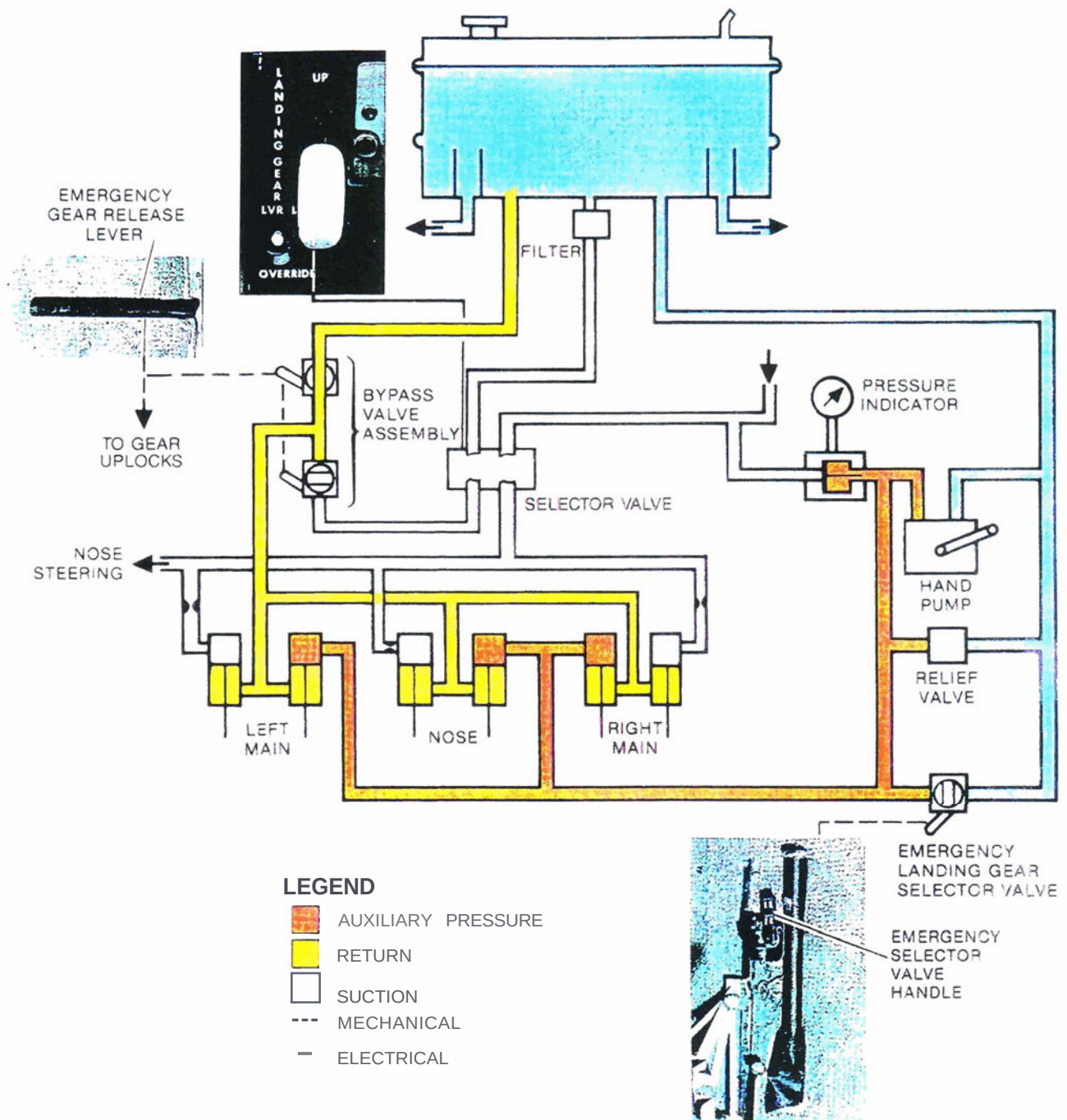
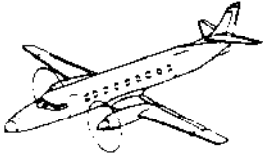


Figure 14-7. Emergency Gear Extension



With the uplocks released, the gear free-falls, aided by the airstream. After safety pin removal, moving the hand pump engage valve handle forward (counterclockwise) mechanically positions the hand pump engage valve and frees the hand pump handle for operation. Hand pump pressure is applied to one actuator on each gear to ensure that the overcenter drag brace goes overcenter, locking the gear down. The gear position indicators show normal down-and-locked indications. There is no provision for emergency retraction of the gear.

VARIABLE-AUTHORITY NOSEWHEEL STEERING

GENERAL

The rack-and-pinion nose steering actuator is electrically controlled with 28.5 VDC from the nonessential bus and is hydraulically actuated when the landing gear lever is placed in the DOWN position. Steering input signals are generated by rudder pedal deflection. **Normal** maximum steering angle is 10" left or right of center. **Variable authority**, when selected, can increase the steering angle to 63° left or right of center. After takeoff a cam on the rear of the nose strut mechanically centers the nosewheel as the strut extends.

OPERATION

Selecting the NOSE GEAR STEERING switch to the ARMED position provides electrical power to the nosewheel steering system. A green NOSE STEERING light on the annunciator panel illuminates when the system is armed (Figure 14-8). Nosewheel steering can be engaged by depressing a button on the left power lever or by placing the right speed lever fully aft (Figure 14-9). Either method provides power steering 10" left or right of center as commanded by rudder pedal movement. When the system is not engaged, the nose gear will be in the caster mode and can, if required, caster up to 63" left or right of center. The

nose steering actuators function as a shimmy damper when the system is not engaged.

When the system is engaged, increased authority may be obtained by depressing and holding the PARK button (Figure 14-9). Holding the PARK button in for several seconds increases the turning authority up to 63" right or left of center, as commanded by the rudder pedals. During this time the PARK button illuminates with a brilliance in proportion to the increased authority. When the button is released, authority and brilliance both decrease over a period of several seconds.

The green NOSE STEERING light serves two purposes. A steady light indicates that the NOSE GEAR STEERING switch is in the ARMED position; flashing indicates that the nosewheel has turned more than 3" beyond the angle selected by the rudder pedals. If this occurs, the nosewheel steering system should disengage, and appropriate action should be taken as outlined in the *AFM*.

The amber NOSE STEER FAIL (Figure 14-9) light illuminates if hydraulic pressure is available to the nosewheel steering actuator when the system is not engaged or the NOSE GEAR STEERING switch is not in the ARMED position. The NOSE STEER FAIL light is normally installed on the annunciator panel. However, if antiskid brakes are installed, the NOSE STEER FAIL light is a square amber light on the instrument panel. Should the NOSE STEER FAIL light illuminate, take the appropriate action outlined in the *AFM*.

BRAKES

GENERAL

The standard brake system is completely independent of the main hydraulic system and uses manual hydraulic disc brakes. An optional system uses main hydraulic system pressure for power-assisted brakes with antiskid, along with an unboosted backup system. Brakes are applied by depressing the rudder pedals.



SA-227 PILOT TRAINING MANUAL

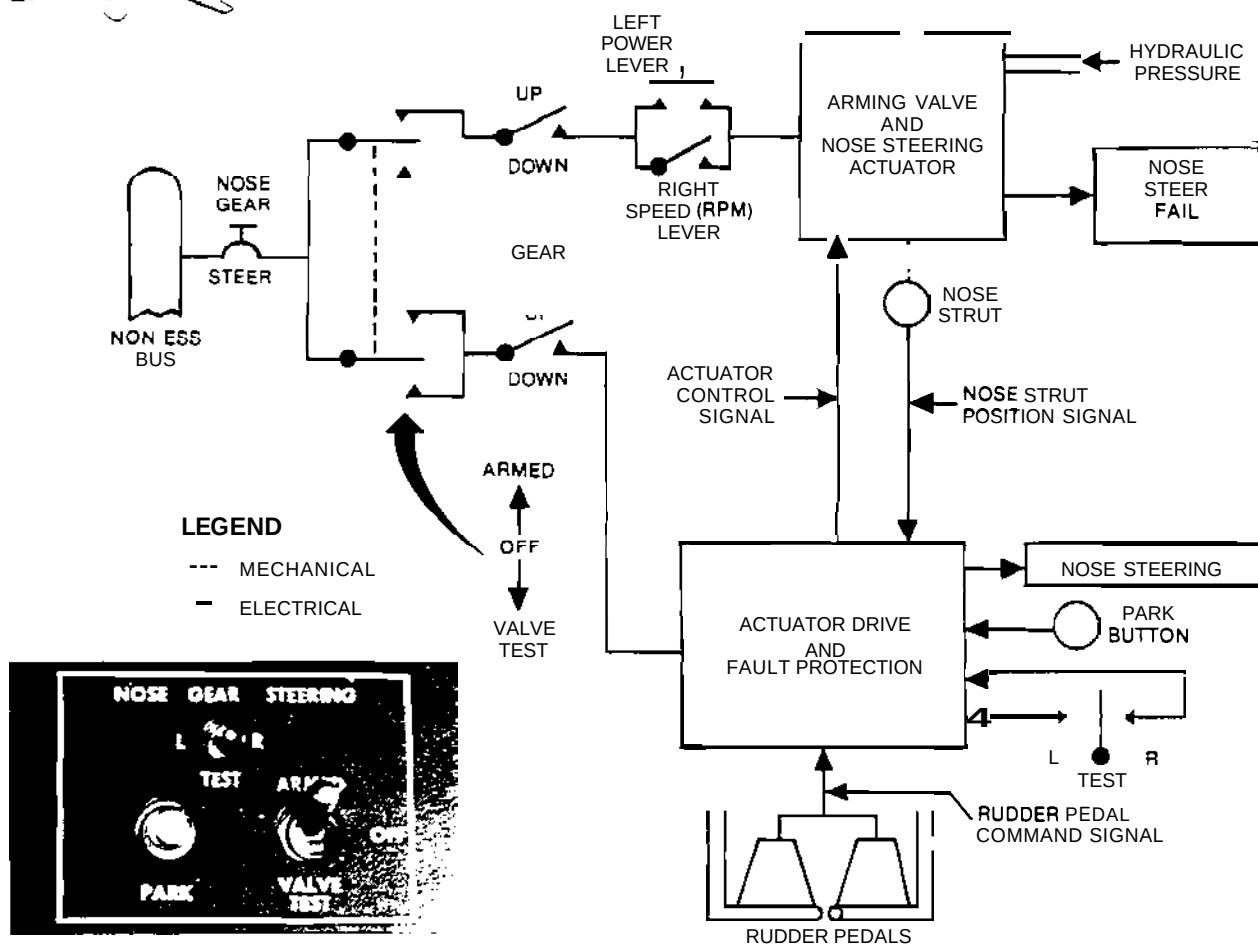


Figure 14-8. Nosewheel Steering Simplified Schematic

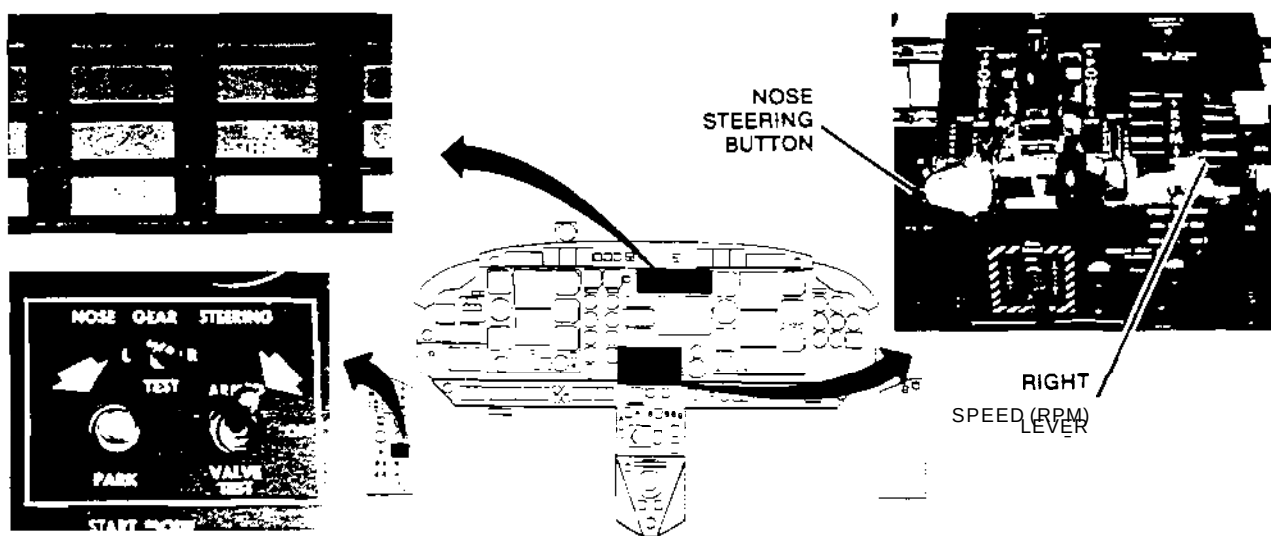


Figure 14-9. Nosewheel Steering Controls and Indicators



MANUAL (UNBOOSTED) BRAKE SYSTEM

Pressure generated by either the pilot's or copilot's set of master cylinders is directly applied to the brake assembly in each wheel through shuttle valves and parking brake valve; (Figure 14-10). The shuttle valves transfer braking function to the first set of master cylinders actuated, preventing simultaneous brake activation by both pilots. Braking force is in direct proportion to pedal deflection. Each pedal applies the corresponding set of brakes which allows differential braking.

To apply the parking brakes, pull the PARKING BRAKE knob out and then depress the rudder pedals while continuing to hold the knob. To release, press the button on the knob, push in the knob, and depress the brakes. After releasing the parking brakes, the brakes should be applied to assure proper operation.

POWER BRAKE SYSTEM (OPTIONAL)

The power brake system utilizes main hydraulic system pressure, controlled by master cylinder pressure. Power braking is available only when the ANTI-SKID switch is in the ON position.

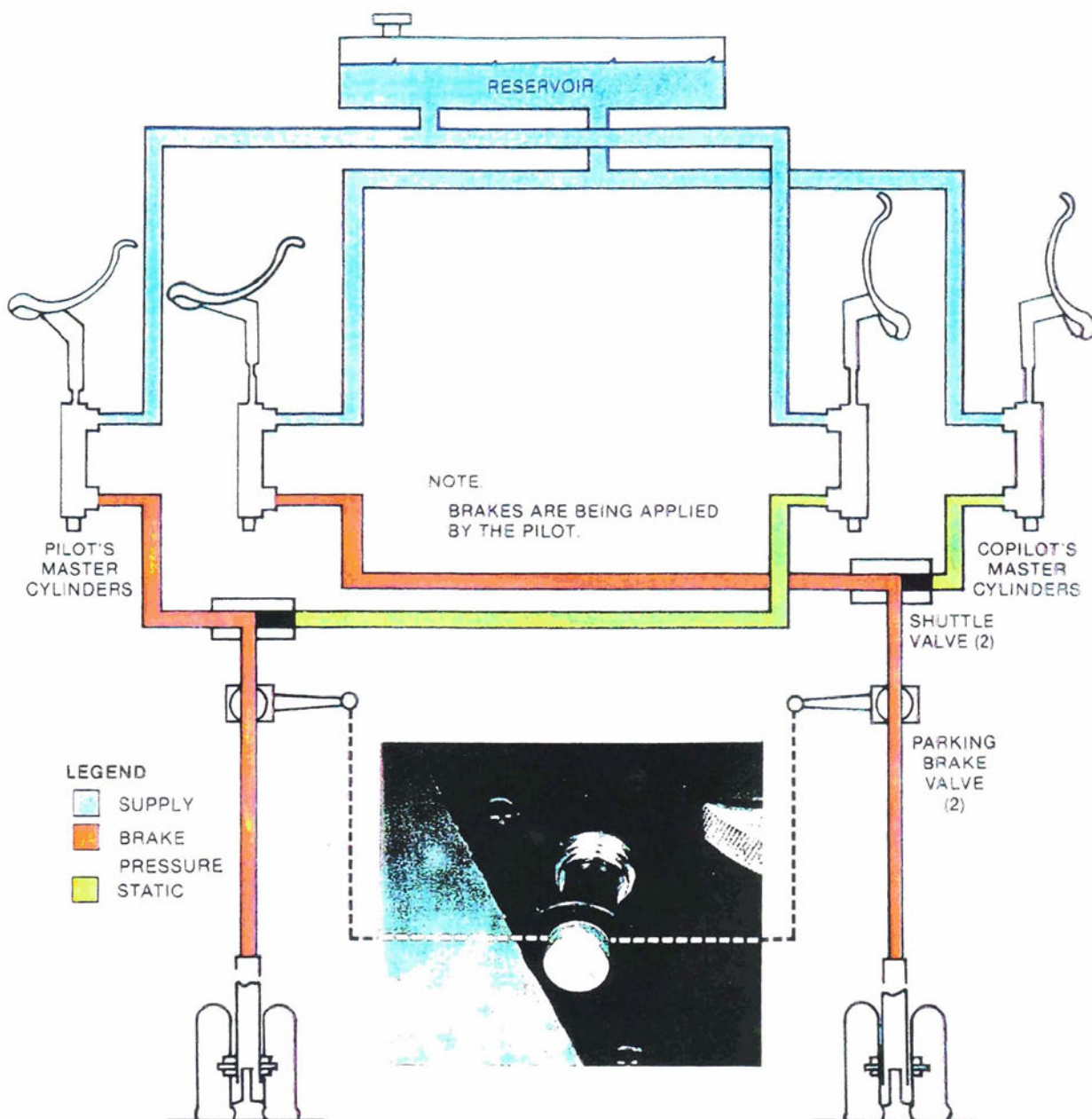
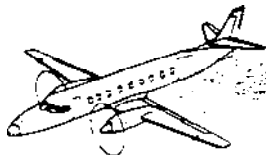


Figure 14-10. Manual Brake System



Operation (Antiskid On)

With the ANTI-SKID switch ON, the anti-skid system is armed and the solenoid shut-off valve is opened, applying main system hydraulic pressure to the brake metering valve (Figure 14-11).

Pressure on either set of rudder pedals applies master cylinder control pressure to the brake metering valve, metering main system pressure to the brakes in direct proportion to pedal pressure. If excessive wheel deceleration is sensed by the antiskid control box, the anti-skid servo valve dumps braking pressure, momentarily releasing all brakes. Brake cycling can be detected by the pilot. Wheel rotation speed must initially be above 10 knots for the antiskid to operate.

If a fault develops in the antiskid system, the amber ANTI-SKID annunciator light illuminates (see Appendix B). The ANTI-SKID switch should then be turned OFF to prevent power brake operation without antiskid protection.

Operation (Antiskid Off Or No Main System Pressure)

With the ANTI-SKID switch OFF or with no pressure in the main hydraulic system, the

brake system reverts to master cylinder operation (Figure 14-12), and the ANTI-SKID light illuminates. Operation of the brakes is essentially the same as previously described under Manual Brake System.

LIMITATIONS

NOSEWHEEL STEERING

Takeoff is prohibited when there has been a hydraulic system failure.

ANTISKID BRAKE SYSTEM

1. Use of the antiskid brake system is prohibited when there has been a hydraulic system failure.
2. Use of the antiskid brake system is prohibited when the amber ANTI-SKID caution light fails to extinguish.
3. Use of power brakes without antiskid is prohibited.



NOTE.

WITH ANTISKID OPERATING, THE
ANTI-SKID WARNING LIGHT
WILL BE EXTINGUISHED.

NOTE,

BRAKES ARE BEING APPLIED
BY THE PILOT.

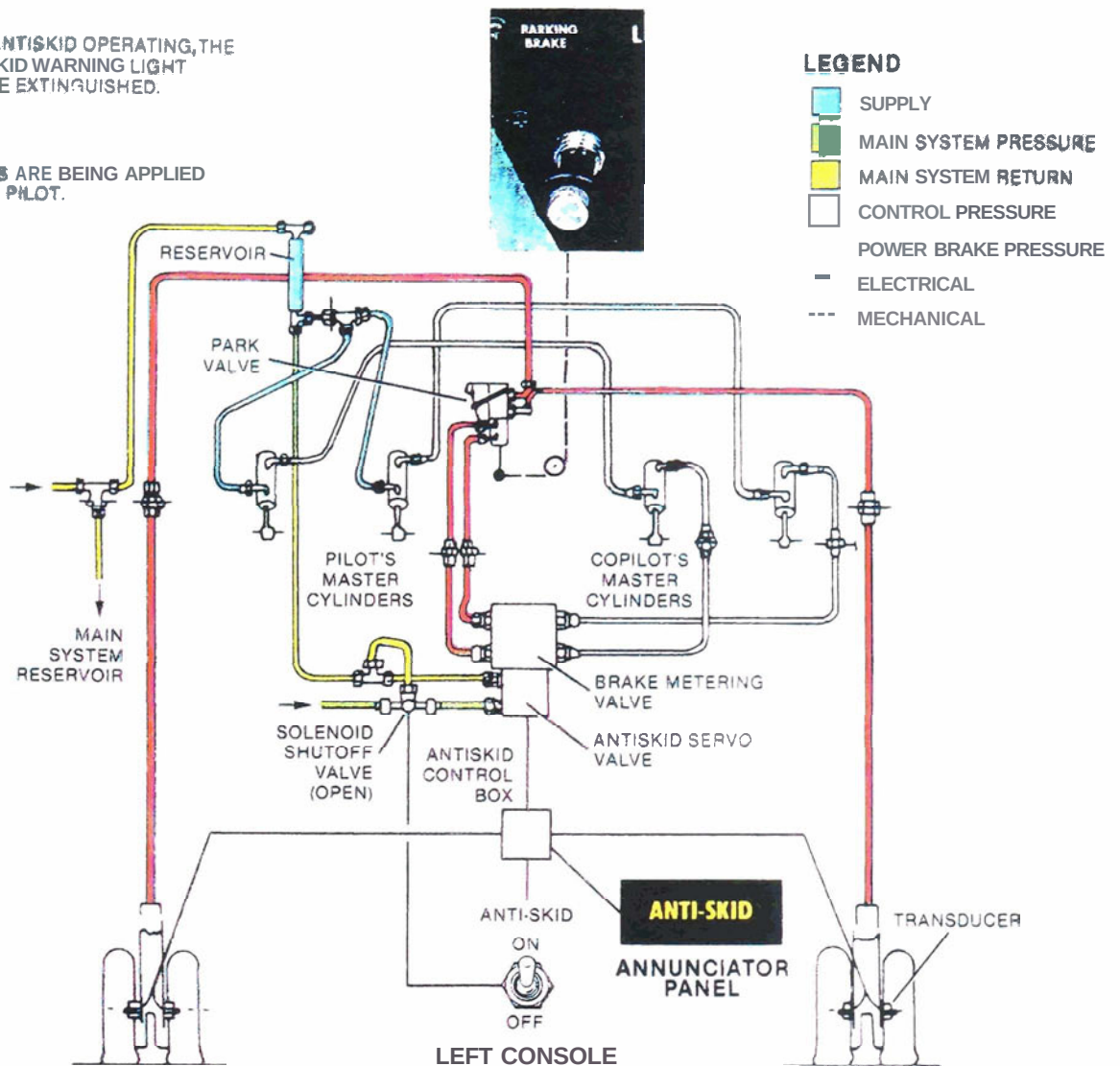


Figure 14-11. Power Brake System (Antiskid On)



NOTE:

BRAKES ARE BEING APPLIED
BY THE PILOT.

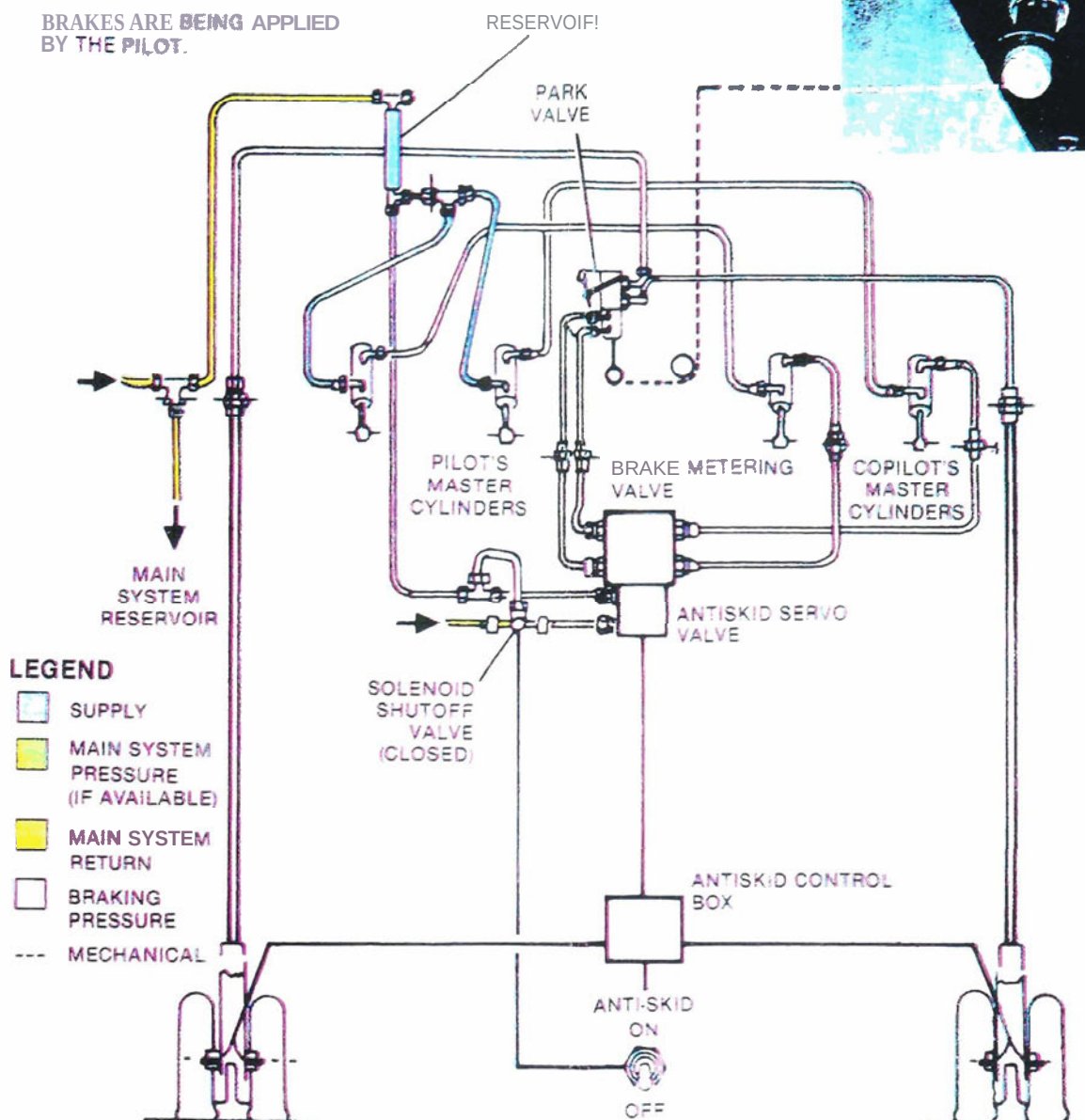
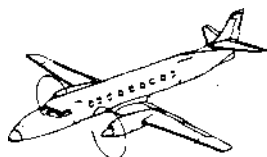


Figure 14-12. Power Brake System (Antiskid Off or No Main System Pressure)



QUESTIONS

1. At lift-off, the nosewheels are centered by:
 - A. An internal mechanical centering device
 - B. An external mechanical centering device
 - ☒ C. The nosewheel steering actuator
 - D. Scissors
2. The landing gear is maintained down and locked after a normal extension by:
 - ☒ A. Overcenter drag braces and hydraulic pressure
 - B. Mechanical locks in all gear actuators
 - C. Overcenter drag braces
 - D. Hydraulic pressure only
3. The landing gear is maintained down and locked after an emergency extension by:
 - A. Hydraulic pressure only
 - ☒ B. Overcenter drag braces and hydraulic pressure
 - C. Mechanical locks in all gear actuators
 - D. Overcenter drag braces
4. The position of the landing gear doors with the gear extended is:
 - A. Main gear doors open, nose gear doors closed
 - B. All doors closed
 - C. All doors open
 - ☒ D. Main gear doors closed, nose gear doors open
5. The landing gear warning horn sounds when:
 - A. Any gear is not down and locked.
 - B. All gears are not down and locked.
 - C. Any gear is not down and locked and flaps are extended beyond 1/4 travel.
 - ☒ D. Any gear is not down and locked and either power lever is retarded to FLT IDLE, or flaps are extended beyond 1/2 travel.
6. Nosewheel steering authority is increased to 63° by:
 - ☒ A. Holding the PARK button depressed
 - B. Advancing the right speed lever out of the LOW position
 - C. Advancing either power lever forward of the FLT IDLE position
 - D. Momentarily pressing and then releasing the PARK button
7. A flashing NOSE STEERING light indicates:
 - A. The system is armed but not engaged.
 - B. A fault exists but the system is still engaged.
 - ☒ C. A fault exists and the system has disengaged.
 - D. The system is engaged and is operating properly.
8. Nosewheel steering operation after gear retraction is prevented by:
 - ☒ A. Control circuits being opened by gear retraction
 - B. The nose gear centering device
 - C. Hydraulic pressure being supplied to the actuators
 - D. The transmission mechanism in the strut being disengaged
9. On airplanes with the standard manual brake system, the shuttle valves:
 - A. Block out the set of master cylinders not being used
 - B. Aid in setting the parking brakes
 - C. Prevent fluid loss in the event a line ruptures
 - D. Shuttle braking pressure from brakes on one gear to brakes on the other gear



- 10.** The antiskid system prevents wheel skid by:
- A. Reducing pressure being applied to the brakes
 - B. Shutting off main system pressure from the brake metering valve
 - C. Bypassing master cylinder control pressure
 - D. Preventing more pressure from being applied to the brakes and, at the same time, releasing the pressure already applied to the brakes
- 11.** The amber ANTI-SKID light comes on when:
- A. The ANTI-SKID switch is placed in the ON position.
 - B. A fault develops in the system.
 - ☒ C. The ANTI-SKID switch is placed in the OFF position.
 - D. Both B and C



CHAPTER 15

FLIGHT CONTROLS

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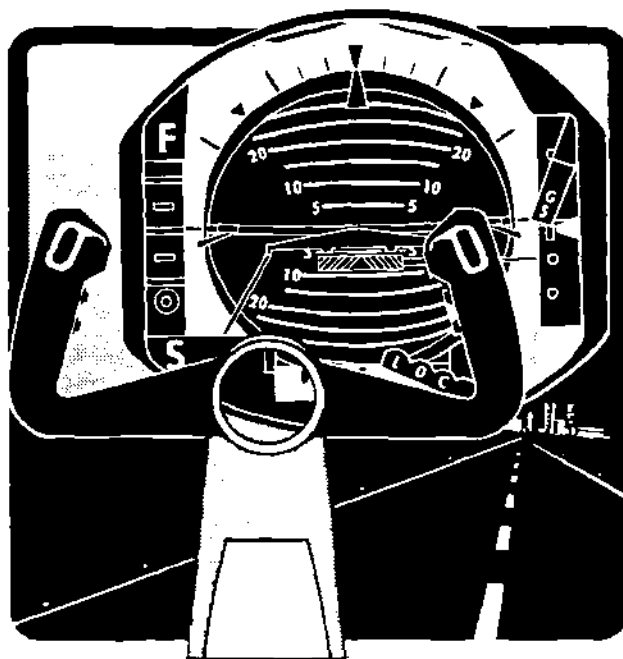
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CHAPTER 15 FLIGHT CONTROLS



INTRODUCTION

The primary flight controls are manually actuated by the rudder pedals, control wheels, and control columns and can be immobilized by a control lock system when on the ground.

Trim is available in all three axes. Aileron and rudder trim is mechanical; pitch trim is electrical.

Secondary flight controls consist of hydraulically operated, electrically controlled flaps.

PRIMARY FLIGHT CONTROLS

A stall avoidance system warns of impending stalls and initiates recovery prior to an actual stall.

An optional yaw damper system compensates for yaw tendencies by automatically applying rudder as required.

The ailerons, rudder, and elevators are manually operated by either pilot through a conventional control column and rudder pedal arrangement. Control inputs are transmitted to the control surfaces through cables, push-pull rods, and bellcranks.



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The ailerons and the rudder have mechanically actuated trim tabs. The horizontal stabilizer is electrically trimmed.

A bob weight attached to the forward side of the control column applies a **nosedown** preload on the pitch control system. Elevator down springs also aid in **nosedown** preload.

All flight control surfaces, including primary, secondary, and trim control surfaces, are shown in Figure 15-1.

CONTROL LOCK SYSTEM

GENERAL

When the control lock system is engaged, **lockpins** are mechanically inserted into the actuating linkage of the rudder and ailerons, and both power levers are restricted to positions aft of the FLT IDLE position. The system is fail-safe; that is, **lockpins** are spring-loaded to disengage if the control lock cable breaks. The control locks cannot be engaged in flight with the power levers forward of FLT IDLE due to mechanical blockage by

the power lever linkage. If desired, the elevator can be secured on the ground by using a seat belt extension to hold the control column fully aft.

OPERATION

To release the control locks, lift the CONTROL LOCK lever out of the LOCKED detent (Figure 15-2) and move it forward to the OFF position, retracting the **lockpins** and freeing both power levers. If the control column has been restrained with the seat belt, hold the control column prior to releasing the seat belt to prevent a sudden forward movement against the instrument panel. Allow the control column to move forward gently.



Figure 15-2. Control Lock Lever

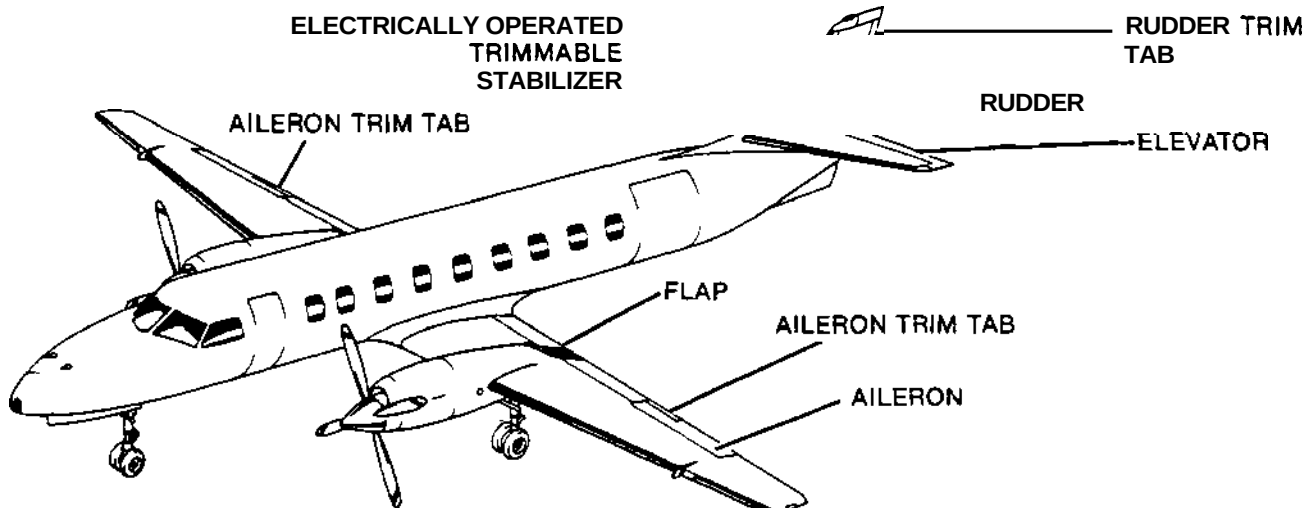
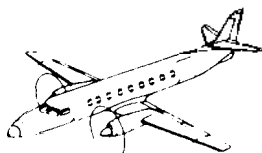


Figure 15-1. Flight Control Surfaces



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After engine shutdown, with both power levers in the **GROUND IDLE** position, the control locks can be engaged by lifting the **CONTROL LOCK** lever out of the **OFF** detent and moving it aft to the **LOCKED** position. Actuate the rudder and ailerons until the locks engage. With the locks engaged, both power levers are mechanically restrained to behind **FLT IDLE**.

TRIM SYSTEMS

GENERAL

Aileron and rudder trim are mechanical; pitch trim is electrical.

AILERON

The trim tab on each aileron is cable-connected to the **AILERON TRIM** wheel on the pedestal (Figure 15-3). Left or right rotation of the trim wheel positions both tabs for lateral trim. As the outer trim wheel is rotated, the inner indicator rotates in the same direction but at a reduced rate. Aileron trim tabs are in the neutral position when the index lines on the outer wheel and the inner indicator are aligned vertically.

The trim tabs also function as servo tabs. When the ailerons are deflected by the control wheel, the tabs move to assist desired aileron movement.

RUDDER

The rudder trim tab is actuated by cables connected to the **RUDDER TRIM** wheel on the pedestal (Figure 15-3). The trim wheel has an index mark for neutral rudder trim tab position.

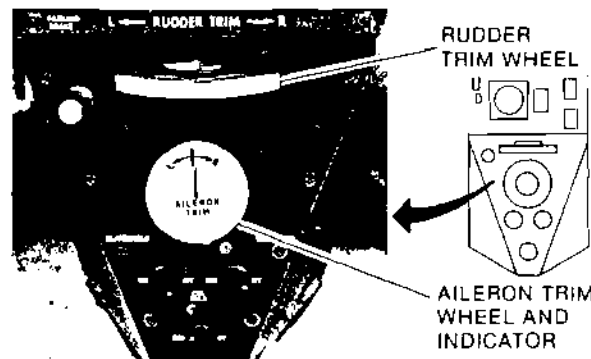


Figure 15-3. Aileron and Rudder Trim

HORIZONTAL STABILIZER (PITCH TRIM)

General

There are no trim tabs on the elevators; pitch trim is accomplished by moving the horizontal stabilizer. An actuator containing two DC motors and two screwjack mechanisms moves the horizontal stabilizer leading edge up or down about a pivot point near the stabilizer rear spar. The motor in the pilot's portion of the system is powered by the left essential DC bus and is controlled by the trim switch on

the pilot's control wheel. The other motor receives power from the right essential DC bus and is controlled by the trim switch on the copilot's control wheel or by the **AUX TRIM** switch on the center pedestal. Stabilizer position is displayed on a **PITCH TRIM** indicator on the instrument panel. The indicator is powered from the nonessential DC bus. The pilot can actuate pitch trim through the copilot's system, if necessary. Pitch (stabilizer) trim controls and indicators are shown in Figure 15-4.

Operation

Control of the pitch trim system is selected with the **TRIM SELECT** switch on the center pedestal (Figure 15-4). With the switch in the **OFF** position, both pitch trim systems are inoperative. With the **PILOT** or **COPILOT**

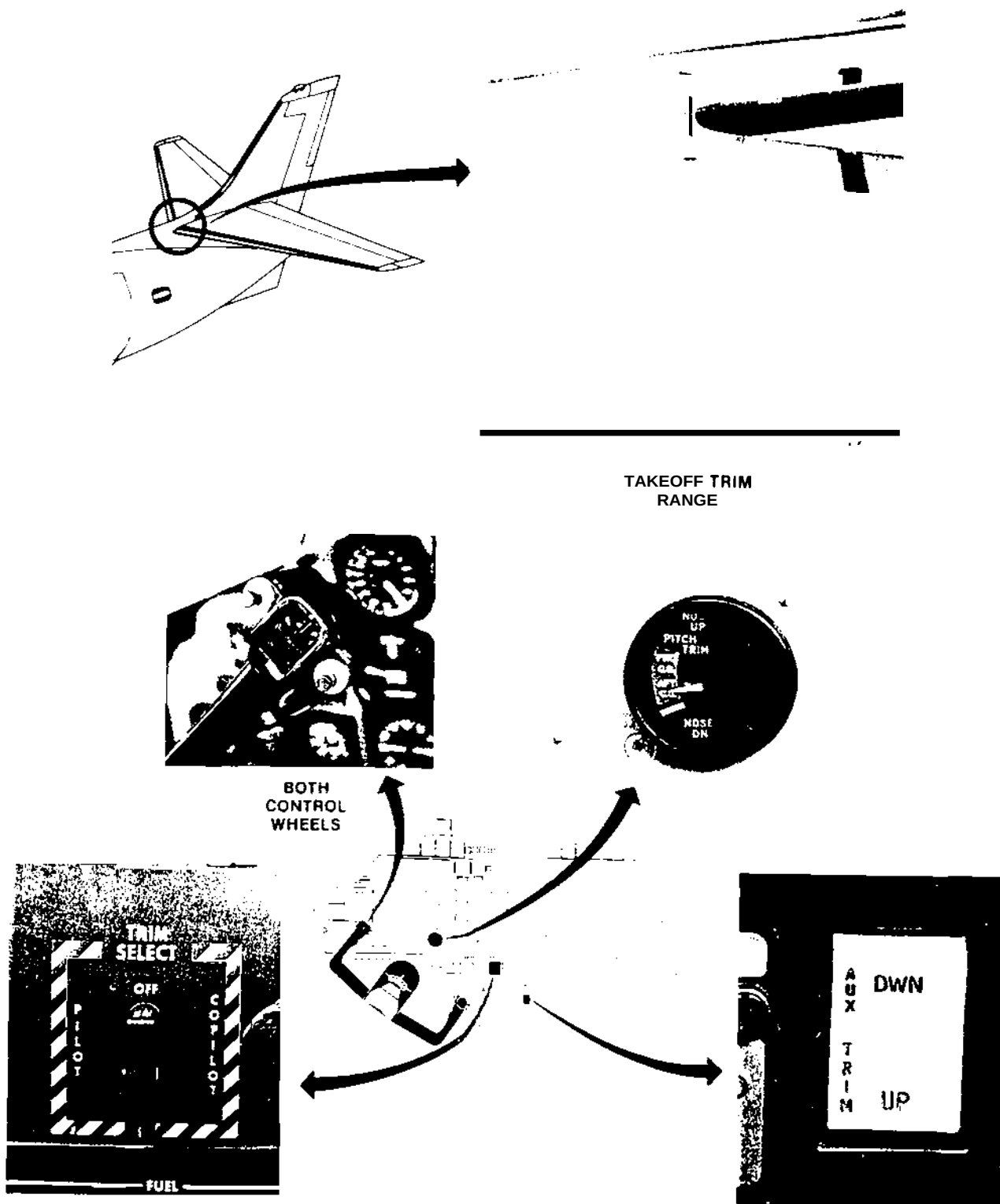


Figure 15-4. Stabilizer Trim Controls and Indicators



position selected, pitch trim can be actuated with the dual-element pitch trim switch on the respective control wheel.

Both elements of the trim switch must be actuated simultaneously to move the stabilizer. Trim actuation with one element of the switch indicates a malfunction. (Refer to the Normal Procedures section of the AFM). As the stabilizer moves, an aural trim-in-motion horn sounds and stabilizer position is shown on the PITCH TRIM indicator. Permissible trim setting for takeoff is indicated by the green band on the PITCH TRIM indicator and the TAKE-OFF range markings at the stabilizer leading edge.

If the pilot's trim system is inoperative, the copilot's trim system can be operated by the pilot with the AUX TRIM rocker switch (Figure 15-4), provided the TRIM SELECT switch is in the COPILOT position. Actuation of the AUX TRIM switch overrides inputs from the copilot's control wheel switch. A diagram of the stabilizer trim system is shown in Figure 15-5.

Warning System

If the stabilizer is out of the TAKEOFF trim range (Figure 15-4) and both power levers are advanced for takeoff, a warning horn will sound. The warning circuit is routed through a squat switch and is inoperative when airborne.

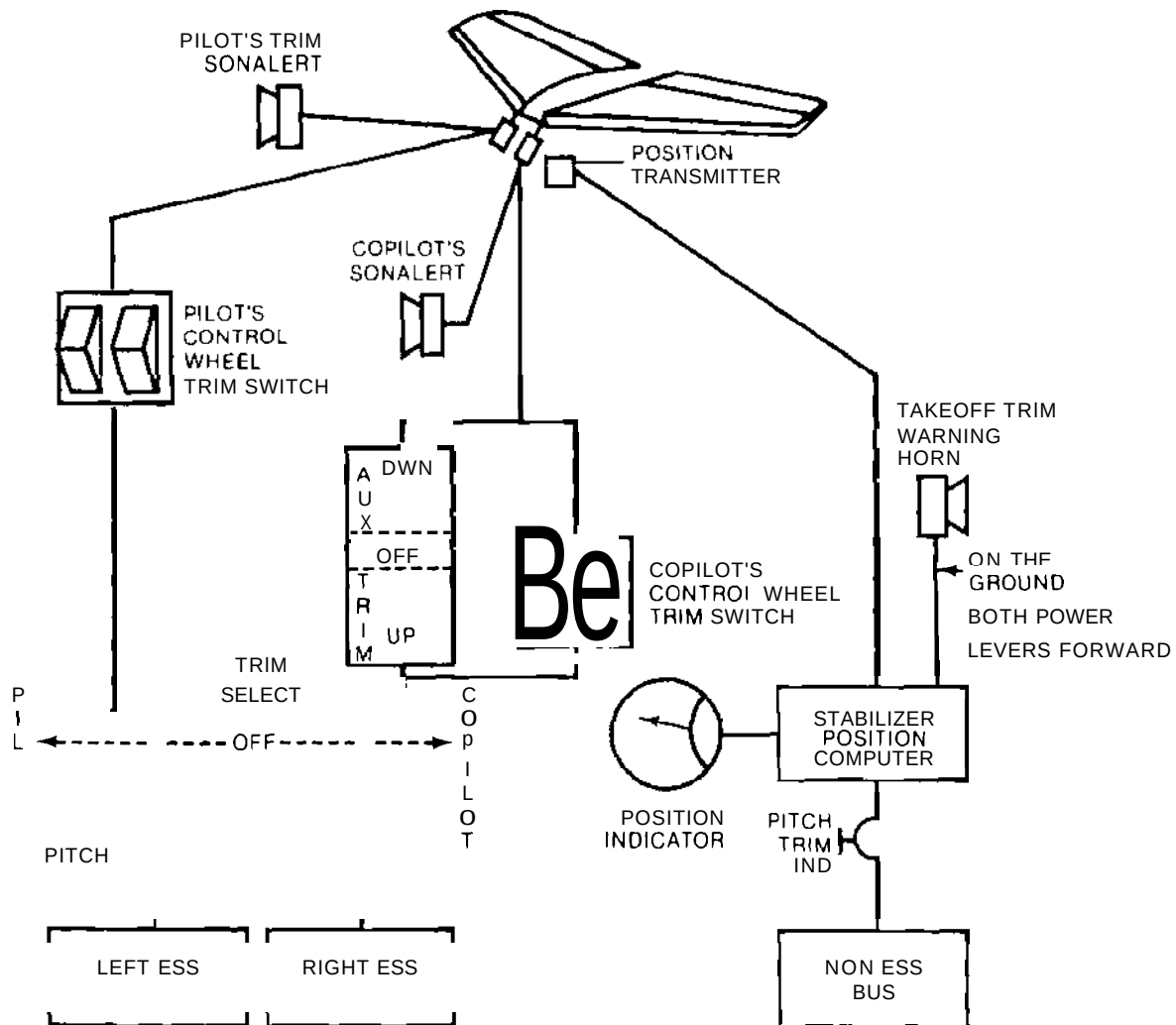


Figure 15-5. Stabilizer Trim System



SECONDARY FLIGHT CONTROLS

FLAPS

Double-slotted, Fowler flaps are electrically controlled and hydraulically actuated. The right and left **wing** flaps are mechanically interconnected to prevent asymmetrical operation **and** to allow either the left or right hydraulic actuators to drive both flaps. Electrical power and hydraulic pressure are required to actuate the solenoid-operated selector valve. There are no provisions for emergency operation of the flaps. Flap system controls and indicators are shown in Figure 15-6. Electrical power for flap control and indication is supplied through the FLAP CONT circuit breaker on the nonessential bus.

Operation

The FLAP lever on the pedestal is used to preselect any flap position from full up to full down. The lever has detents at the $\frac{1}{4}$ (**T.O.**) and $\frac{1}{2}$ flap positions for pilot convenience. Moving the FLAP lever to any position, except up, completes a circuit through the logic box to the selector valve (Figure 15-7). Pressure is directed through lock valves to the actuators, driving the flaps to the selected position. The deenergized selector valve and the lock valves each create liquid locks to hold the flaps in position.

If the flaps do not operate properly, moving the FLAP lever to the full UP position should retract the flaps. The circuit is direct, bypassing the logic box and permitting retraction in the event certain logic circuits fail.

Flap position is shown on the FLAP POSition indicator (Figure 15-6).

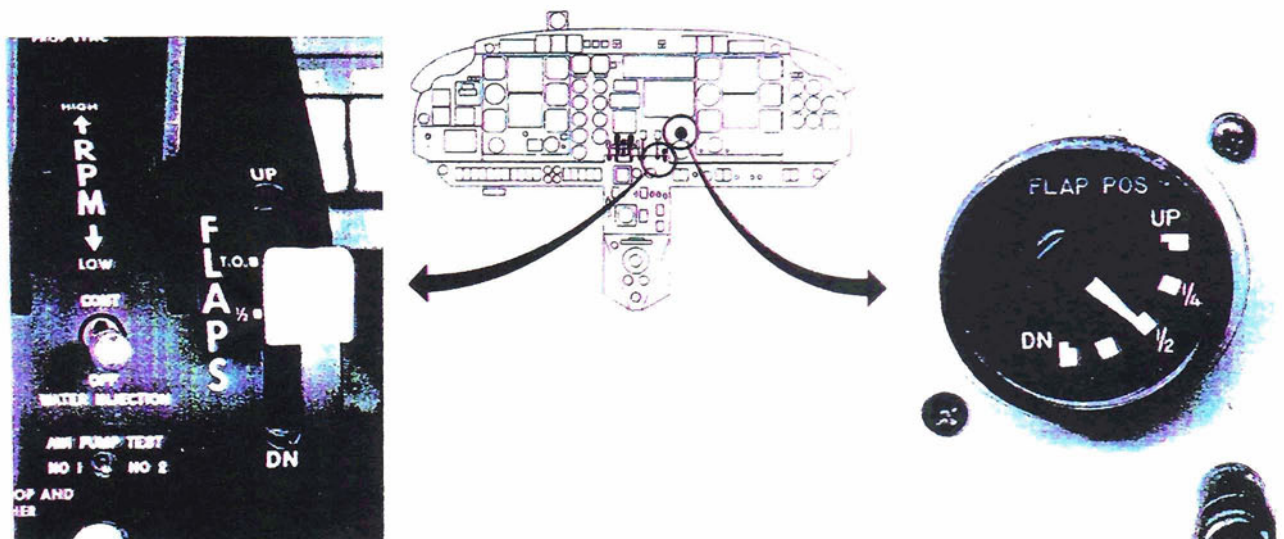


Figure 15-6. Flap System Controls and Indicators

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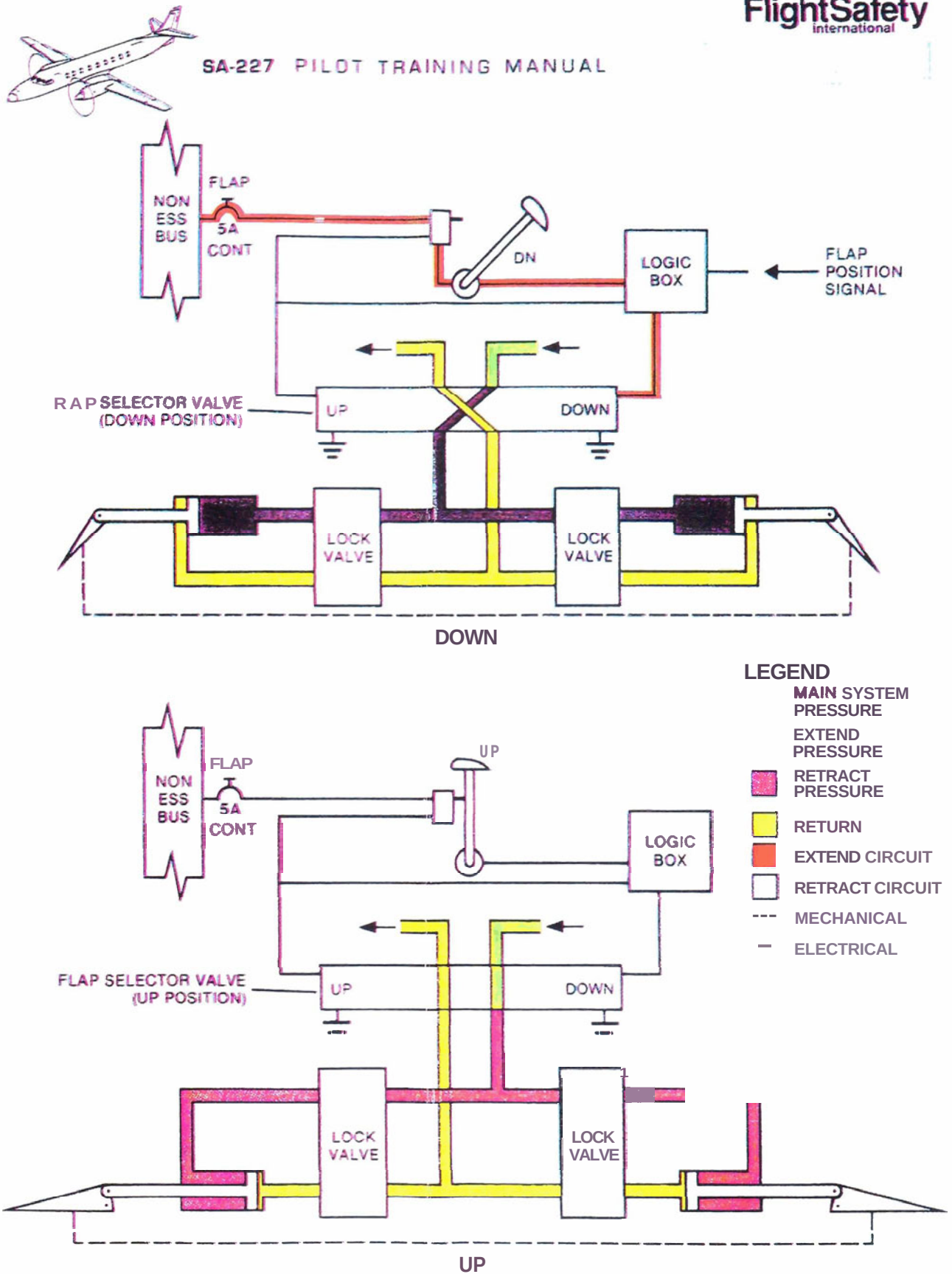


Figure 15-7. Flap Operation

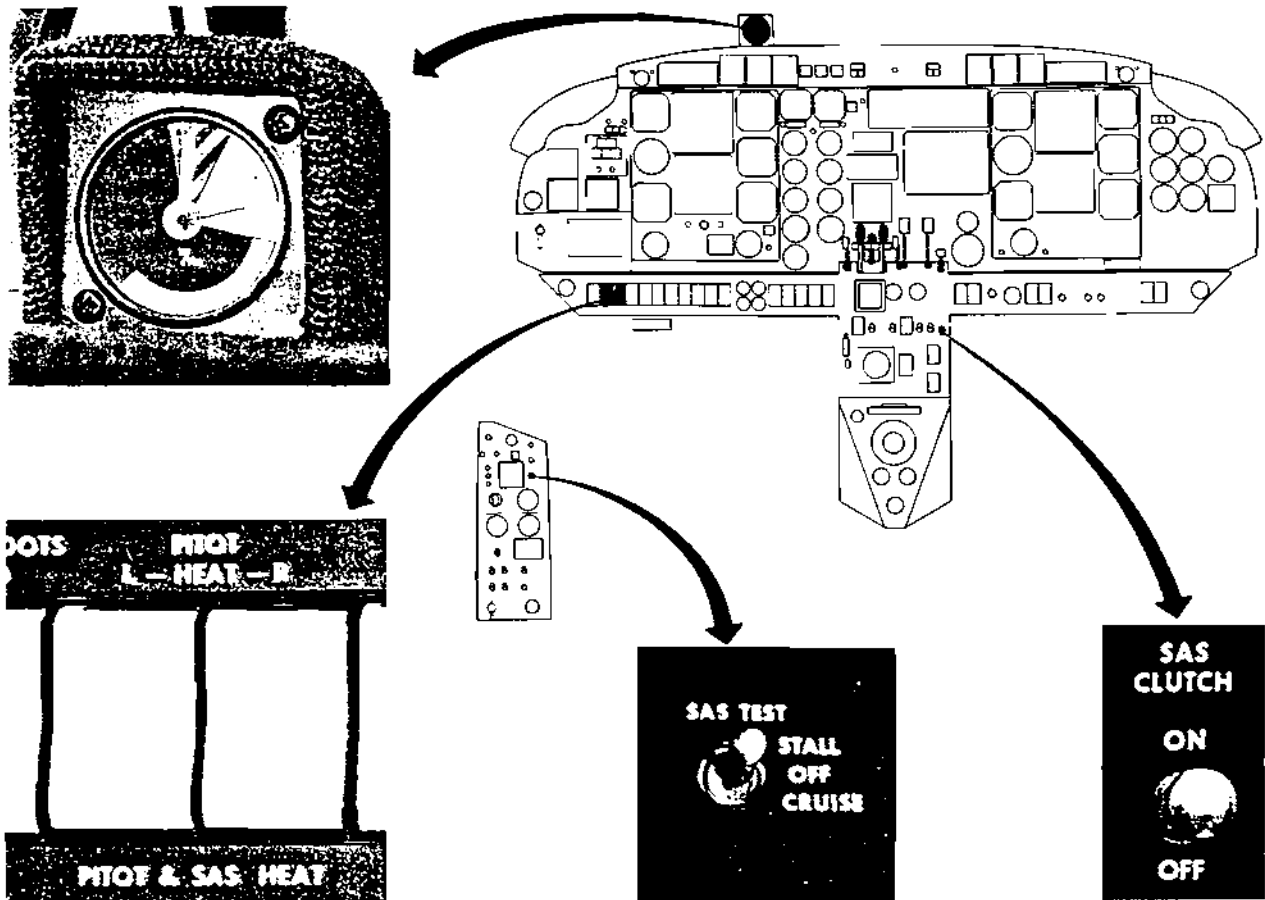
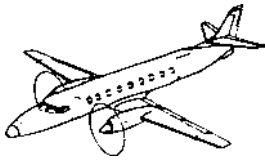


Figure 15-8. Stall Avoidance Controls and Indicators

STALL AVOIDANCE SYSTEM (SAS)

GENERAL

The Metro III and Merlin IVC are equipped with a stall avoidance system (SAS) which is armed at liftoff and disarmed as airspeed increases.

Inputs from an angle-of-attack (SAS) vane on the forward fuselage and a flap position transmitter are evaluated by the SAS computer. The computer then determines operation of the SAS indicator, the stall warning horn, and the control column (stick) pusher.

Aural and visual warning of an impending stall are provided by a warning horn and the SAS indicator. If visual and aural warnings go unheeded, the control column is **pushed** forward by an electrical servo as the stall approaches. The SAS system must be in operation for flight. Controls and indicators for the systems are shown in Figure 15-8.

OPERATION

The SAS stick pusher and stall warning horn are disabled on the ground by a squat switch. The SAS is armed by an airspeed switch in the copilot's pitot-static system. A green SAS ARM light on the annunciator panel (see Appendix B) illuminates at airspeeds below 140 ± 5 KIAS to indicate that the SAS is armed.



Above 140 KIAS the airspeed switch disables the stick pusher and stall warning horn, and the SAS ARM light extinguishes. When airspeed falls below 140 KIAS, the SAS arm light re-illuminates, and the pusher and horn are enabled again,

Approaching a stall, the increasing angle of attack drives the SAS indicator pointer near the edge of the **red/black** crosshatch area (approximately $1.1 V_{S1}$) (Figure 15-9), and the warning horn will sound. If airspeed continues to decrease, just before the stall the control column will be pushed forward with approximately **60** pounds of force until increasing airspeed drives the pointer back into the yellow zone. The column force can be overcome by the pilot.

If the stick pusher actuates at an inappropriate time, it can be disengaged by turning the SAS **CLUTCH/SERVO** switch OFF (Figure 15-8).

A red SAS **FAULT** light on the annunciator panel (see Appendix B) warns the pilot of SAS malfunctions. When the light flashes, it indicates that the pusher servo is inopera-

tive or that the servo clutch is disengaged. It is normal for the light to flash if the SAS **CLUTCH/SERVO** switch is OFF.

Steady illumination of the SAS **FAULT** light indicates that there is no electrical power to the SAS computer or that the computer and pusher have both failed. The SAS indicator and stall warnings are unreliable when the light is illuminated.

A block diagram of the stall avoidance system is shown in Figure 15-10.

Either **PITOT HEAT** switch (Figure 15-8), when placed in the **PITOT & SAS HEAT** position, heats the SAS vane to prevent vane icing. Selection of SAS heat is indicated by illumination of the green SAS **DEICE** annunciator light.

The SAS **TEST** switch (Figure 15-8) is used for ground testing of the SAS system. (Refer to the Normal Procedures section of the AFM.)

The SAS indicator is not certified for use as an in-flight approach indicator, but is to be used for ground testing of the system and for in-flight indication of approaching stalls.

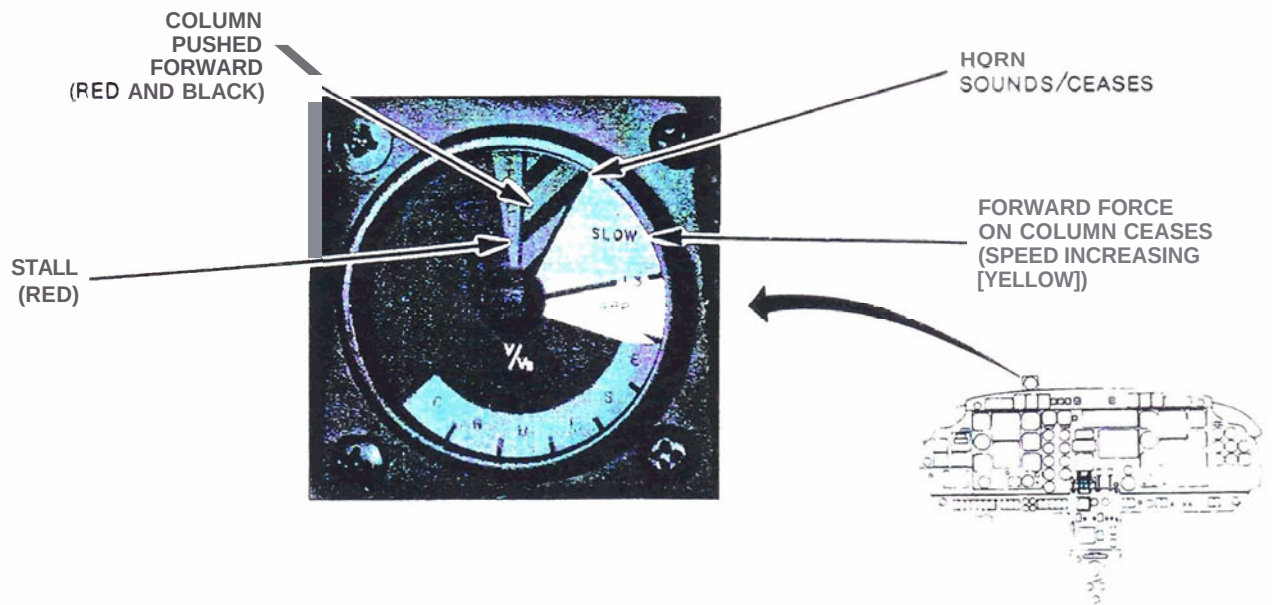


Figure 15-9. SAS Indicator

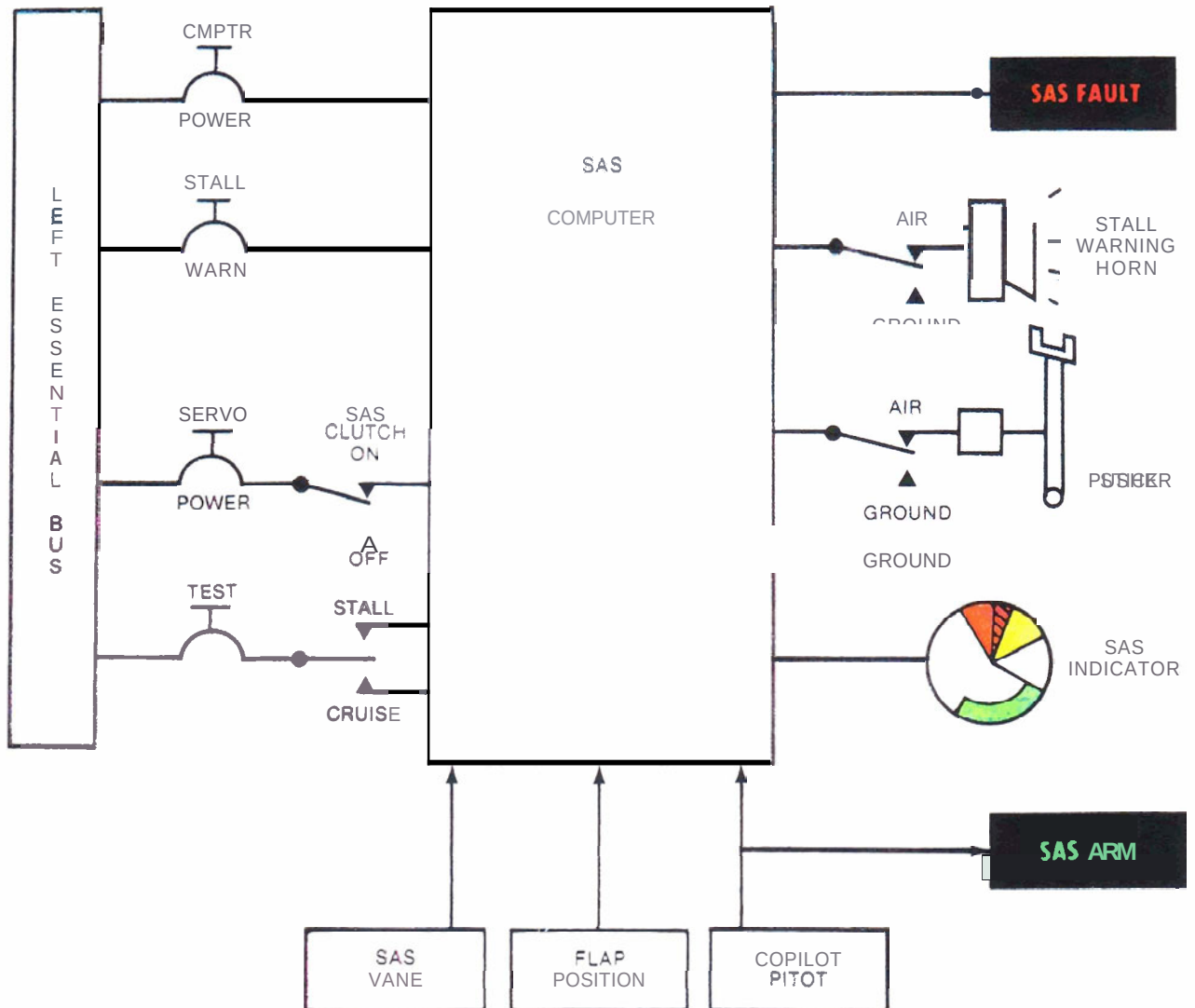


Figure 15-10. Stall Avoidance System



WARNING

Do not stall the airplane with SAS inoperative.

YAW DAMPER

GENERAL

The optional yaw damper improves stability, especially during low airspeed or cross-controlled flight during crosswind landing. Yaw signals applied to an amplifier result in electrical servo action to dampen yaw motion. The factory-installed yaw damper is usually

replaced when an autopilot is installed. In that case, autopilot operating information explains yaw damper operation.

OPERATION

The system is enabled by placing the YAW DAMP switch in the ON position (Figure 15-11), provided the left main squat switch senses an airborne condition. Yawing detected by the pilot's turn and bank indicator and a slip/skid sensor results in servo actuation of the rudder. A slip clutch on the servo allows pilot override of the system in the event of a system malfunction. Use of the yaw damper system is optional.

The YAW DAMP TEST switch can be used to test the system prior to takeoff. (Refer to the Normal Procedures section of the AFM.)

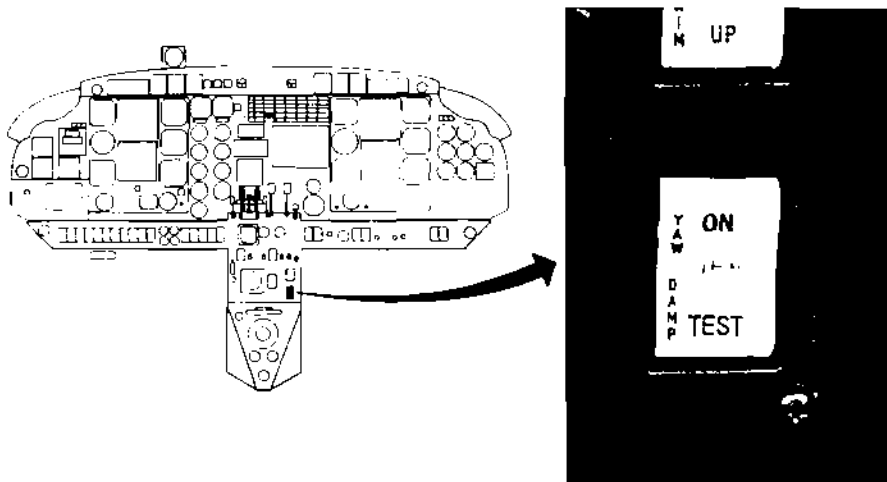


Figure 15-11. Yaw Damper System



LIMITATIONS

REQUIRED EQUIPMENT LIST

The SAS system must be operational for all flights.

ICING CONDITIONS (VISIBLE MOISTURE BELOW +5° C)

The Pitot heat/SAS heat must be *on* and *operative*.

AIRSPEED LIMITS

See Table 15-1 for a listing of airspeed limitations.

Table 15-1. AIRSPEED LIMITS

SPEED	KCAS	KIAS'	REMARKS
V_{MO}/M_{MO} (Maximum Operating Speed)	24810.52	246	This speed applies from sea level through 17,800 feet. At pressure altitudes above 17,800 feet, use the maximum allowable airspeed indicator to remain below the M_{MO} limit. V_{MO}/M_{MO} must not be exceeded deliberately.
V_A (Maneuvering Speed)	See Limitations Section of AFM		Maximum speed at maximum gross weight at which individual application of full available aerodynamic control will not overstress the airplane. This speed decreases approximately 7 KIAS per 1,000 pounds reduction in weight.
V_{FE} (Flaps Extension Speed)			Maximum speed for extending the flaps or operating with flaps extended: 114 flaps (9°) 112 flaps (18°) Full flaps (36°)
V_{MCA} (Minimum Control Speed)			Minimum control speed in flight in the following configuration: gear up and flaps at takeoff position, takeoff power on the operating engine, windmilling propeller on the inoperative engine with NTS operative, no more than 5° bank into good engine.

'KIAS' is based on the normal static system.



QUESTIONS

1. The surfaces classified as primary flight control surfaces are the:
 - A. Flaps, ailerons, and rudder
 - ☒ B. Ailerons, rudder, and elevators
 - ☒ C. Ailerons, rudder, and horizontal stabilizer
 - D. Flaps, rudder, and horizontal stabilizer
2. If a control lock cable breaks, control surface lockup is prevented by:
 - A. Dual cables in the system
 - B. Oversize cables in the system
 - ☒ C. Lockpins which are spring-loaded to disengage
 - D. Nothing prevents control surface lockup due to cable breakage.
3. In-flight engagement of the control locks is prevented by:
 - ☒ A. Mechanical obstruction by power lever linkage
 - B. Airload on the control surfaces
 - C. A squat switch-controlled lockout device
 - D. Nothing prevents in-flight engagement of the gust locks
4. Rudder trim is accomplished:
 - ☒ A. By the yaw damper
 - B. Manually, by rotation of the RUD-
DER TRIM wheel on the pedestal
 - C. Electrically, by lateral movement of the dual-element trim switches on either control wheel
 - D. Electrically, by rotation of the RUD-
DER TRIM wheel on the pedestal
5. If the trim switch on the left control wheel is inoperative, the pilot can apply pitch trim:
 - A. Only by constant pressure on the control column
 - B. Only by instructing the copilot to apply trim
 - ☒ C. By placing the TRIM SELECT switch to the COPILOT position and using the AUX TRIM switch on the center pedestal
 - D. He cannot apply pitch trim.
6. Simultaneous pitch trimming by the pilot and copilot is prevented by:
 - ☒ A. Positioning of the TRIM SELECT switch
 - B. Pilot inputs overriding those made by the copilot
 - C. Each input canceling the other input
 - D. Nothing prevents it; it is possible
7. Asymmetrical flap operation is prevented by:
 - A. Equal airload on the flaps
 - B. Equal pressure on the flap actuators
 - ☒ C. Mechanical interconnection
 - D. Nothing prevents asymmetrical operation; it can occur.
8. The flaps are held in any position by:
 - A. A liquid lock created by the flap lock valves
 - B. A liquid lock created by the selector valve
 - C. A liquid lock created by the selector valve and the flap lock valves.
 - D. A pressure-operated internal locking device within the actuators



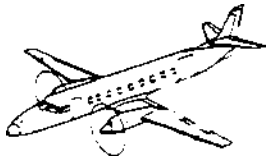
9. The relationship of airspeed to stall speed when the stall warning horn begins to sound is approximately:
- A. 1.0 V_S
 - B. 1.1 V_S
 - ☒ C. 1.3 V_S
 - D. 1.75 V_S
10. The SAS CLUTCH/SERVO switch should be placed in the OFF position when:
- ☒ A. A forward force on the control column is experienced at an inappropriate air-speed.
 - B. The warning horn ceases to sound.
 - C. The warning horn sounds.
 - D. Practicing stalls
11. The indication of pusher servo failure is:
- A. The SAS FAULT light flashes.
 - ☒ B. The SAS ARM light flashes.
 - C. Steady illumination of the SAS FAULT light
 - D. The SAS ARM light comes on.
12. The source of input signals to the factory-installed yaw damper is the:
- A. Autopilot
 - ☒ B. Slip/skid sensor and turn-and-bank indicator
 - C. Slip/skid sensor only
 - D. Turn-and-bank indicator only



CHAPTER 16 AVIONICS

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CHAPTER 16

AVIONICS



INTRODUCTION

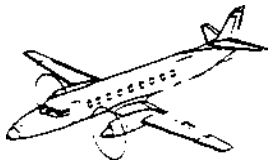
The avionics packages are provided on a custom basis for each airplane. Refer to the applicable vendor manuals for information on all avionics equipment. The pitot-static system will be discussed in this chapter.

GENERAL

The left and right **pitot** heads supply **pitot** pressure to their respective airspeed indicators. The static system incorporates two sets of static ports: one set for the pilot's instruments and the other set for the copilot's. In addition, an alternate static port can supply static pressure to the pilot's instruments only.

The airplanes incorporate additional static ports for other systems, including one used as

a reference for the safety relief function of the outflow valve in the pressurization system. Another for the cabin door is located underneath the handle. Additional ports are used for the cabin pressure controller (pressurization system), the automatic flight control system, the alternate static port and the safety relief function of the dump valve (pressurization system).



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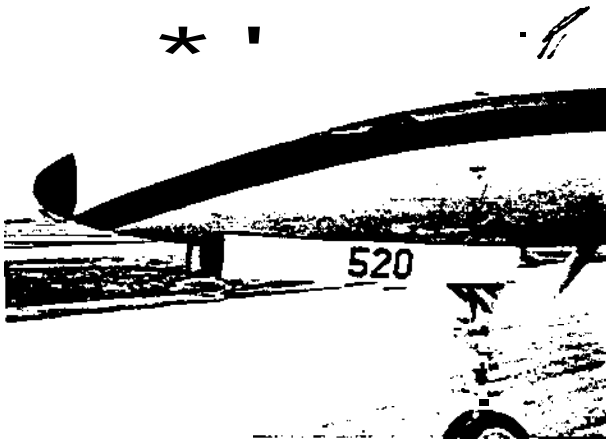


Figure 16-1. Pitot Heads

PITOT SYSTEM

A pitot head (mast) is installed on the left and right sides on the upper half of the nose section (Figure 16-1).

Each is electrically heated for anti-icing purposes. Refer to Chapter 10, "Ice and Rain

Protection." for more information.

As shown in Figure 16-2, the left pitot head is the pitot source for the pilot's airspeed indicator and the right is used for the copilot's airspeed. Additionally, the copilot's pitot head provides a pitot source for the airspeed switch in the SAS system. Refer to Chapter 15, "Flight Controls," for more information on the airspeed switch.

STATIC SYSTEM

DESCRIPTION AND OPERATION

Separate, balanced static systems for each pilot (Figure 16-2) provide static reference to the pilot's and the copilot's airspeed indicators, vertical speed indicators, and altimeters. The copilot's static system provides the static reference to the SAS airspeed switch. The pilot's instruments can also use the alternate static source.

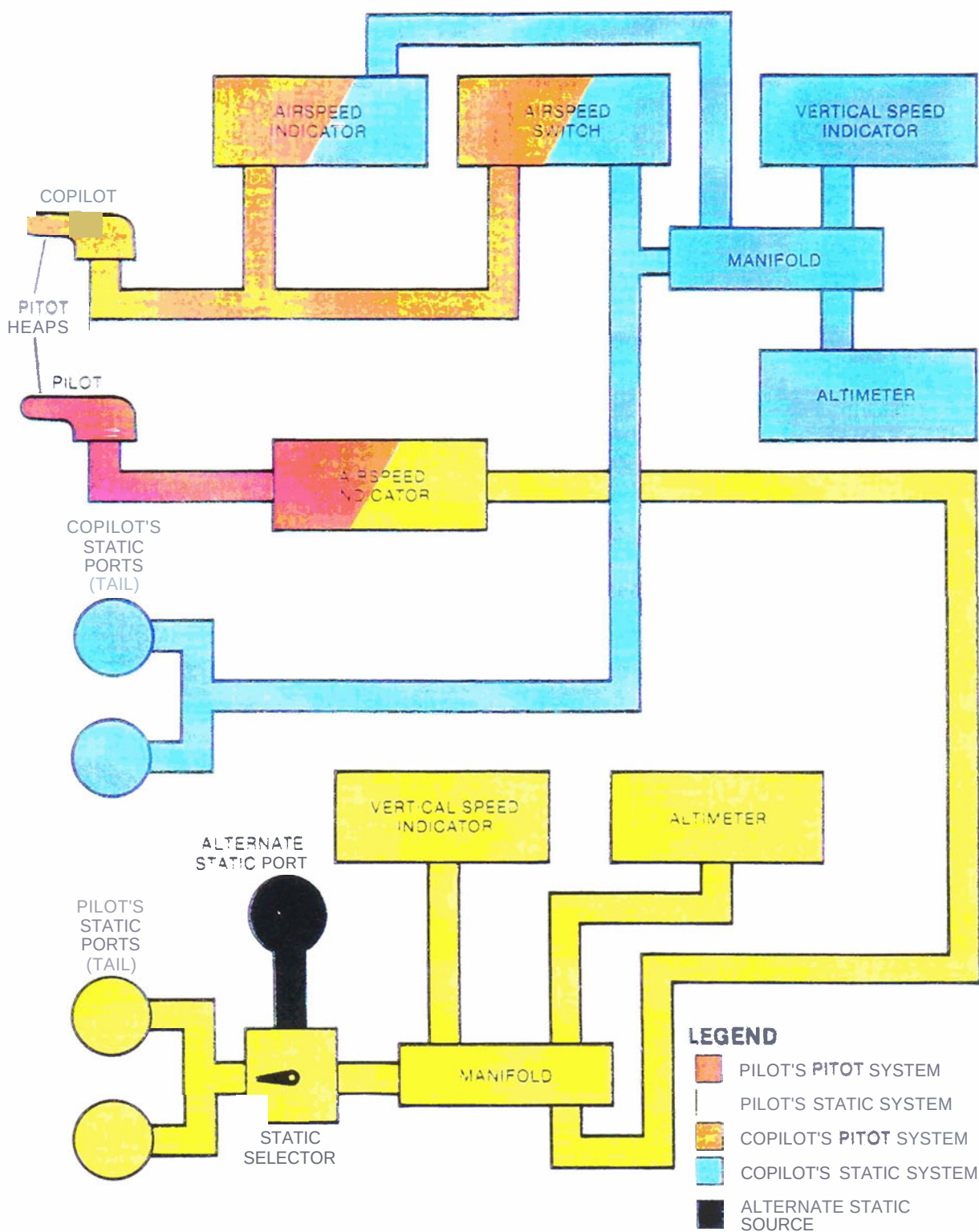
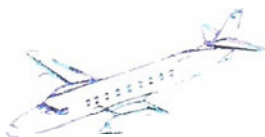
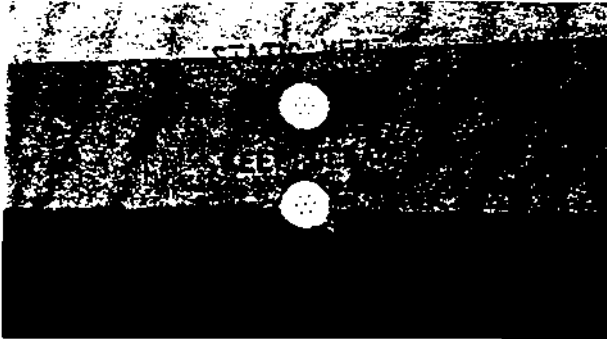
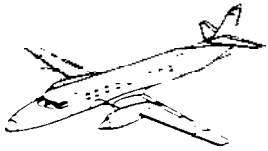


Figure 16-2. Pitot-Static System Overall Diagram



Two static ports are located on each side of the fuselage aft of the cargo door entrance (Figure 16-3). A condensation drain is provided at each side console, as well as inside the aft fuselage.

ALTERNATE STATIC SYSTEM DESCRIPTION

NOTE

- The copilot's static pressure instruments are not connected to the alternate static pressure source.
- Do not dump pressurization when using the alternate static pressure source.
- The altitude and airspeed corrections are not valid if the dump valve is open.

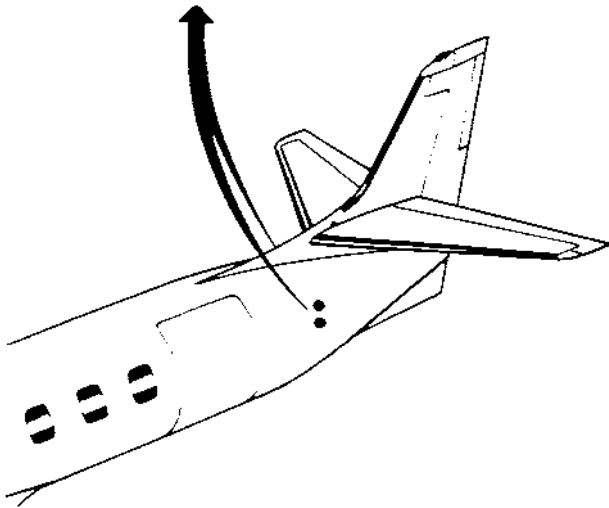


Figure 16-3. Static Ports

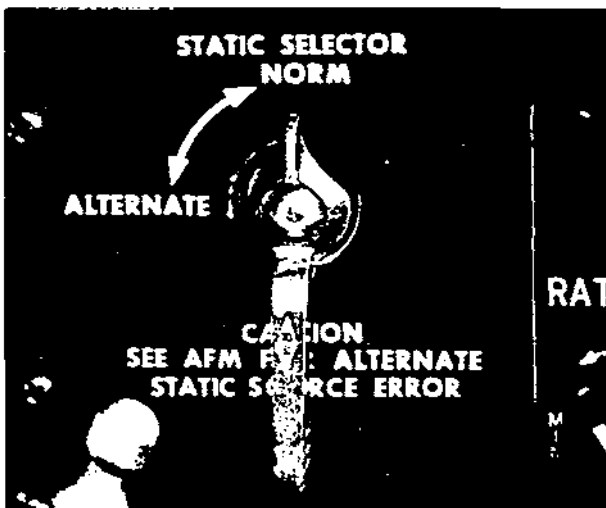


Figure 16-4. STATIC SELECTOR Handle

The alternate static port is located in the *unpressurized* nose baggage compartment, and can be selected for the pilot's static instruments by placing the STATIC SELECTOR handle in the ALTERNATE position (Figure 16-4). The selector handle and valve are located on the lower left side of the pilot's instrument panel.

The alternate static system is used whenever one of the static ports fails or when the static system plumbing develops a leak. Under normal operating conditions, the STATIC SELECTOR handle is left in the NORMAL position. Refer to the AFM for altitude and airspeed corrections when using the alternate static system.



QUESTIONS

1. The pitot heads are located:
 - A. On the lower half of the nose section
 - B. On the upper half of the nose section
 - C. One on top of the nose, one on the bottom
 - D. One on each side of the nose section
2. Besides the copilot's airspeed indicator, the other component that receives pitot pressure from the copilot's pitot is the:
 - A. Rate-of-climb indicator
 - B. Altimeter
 - C. Air data computer
 - D. Airspeed switch for the SAS system
3. The number of static ports that can supply a static reference to the copilot's altimeter is:
 - A. One
 - B. Two
 - C. Three
 - D. Four
4. The _____ can be selected by placing the STATIC SELECTOR handle in the _____ position.
 - A. Alternate static port, EMER
 - B. Alternate static port, NORM
 - C. Emergency static port, EMER
 - D. Alternate static port, ALTERNATE

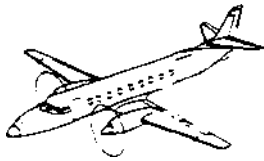


CHAPTER 17

MISCELLANEOUS SYSTEMS

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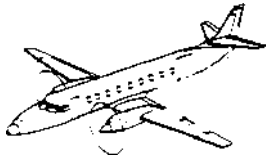


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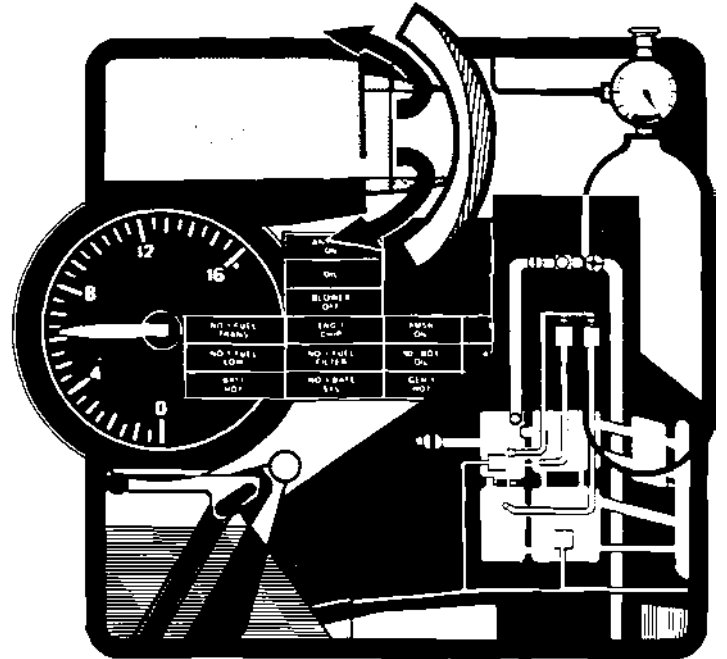
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CHAPTER 17

MISCELLANEOUS SYSTEMS



INTRODUCTION

The Metro and Merlin airplanes have an oxygen system installed as standard equipment. Functionally, the oxygen system is the same on all Metros and Merlins, except for differences in location and quantity of passenger masks and in the **number** and size of oxygen cylinders. The oxygen system is intended to be used in the event of a pressurization failure, smoke in the cockpit or cabin, and for medical purposes, as required.

OXYGEN SYSTEM

DESCRIPTION

The oxygen systems (Figures 17-1 and 17-2) consist of: two crew oxygen masks and outlets, the **PASS OXYGEN** toggle control, an **OXYGEN PRESS** gage, oxygen cylinder(s), and passenger masks and outlets.

Crew Oxygen Masks

Two crew oxygen masks plug into flush-mounted outlets (Figure 17-3). The crew oxygen masks are the quick-donning, diluter-demand type equipped with a **NORMAL/100%** selector. The selector is kept in the 100% position in case the mask must be donned in emergency conditions.



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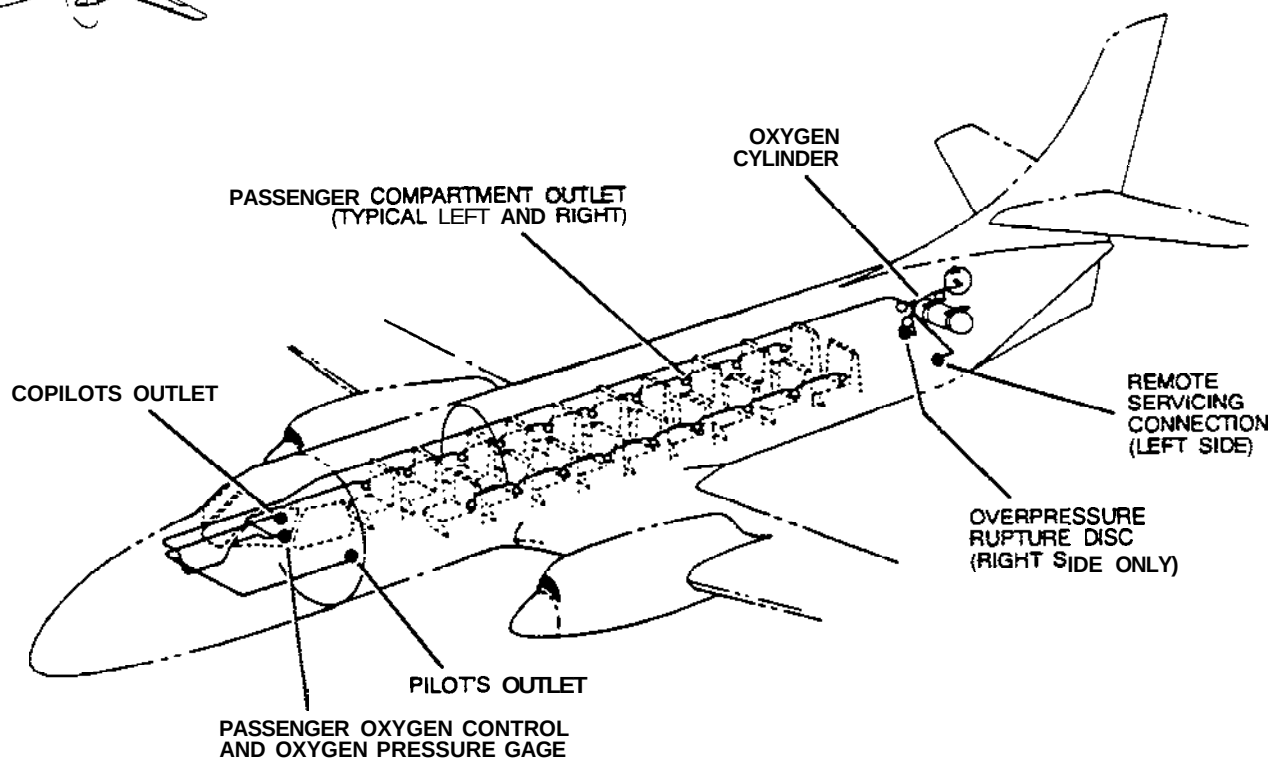


Figure 17-1. Oxygen System Component Locations–Metro III

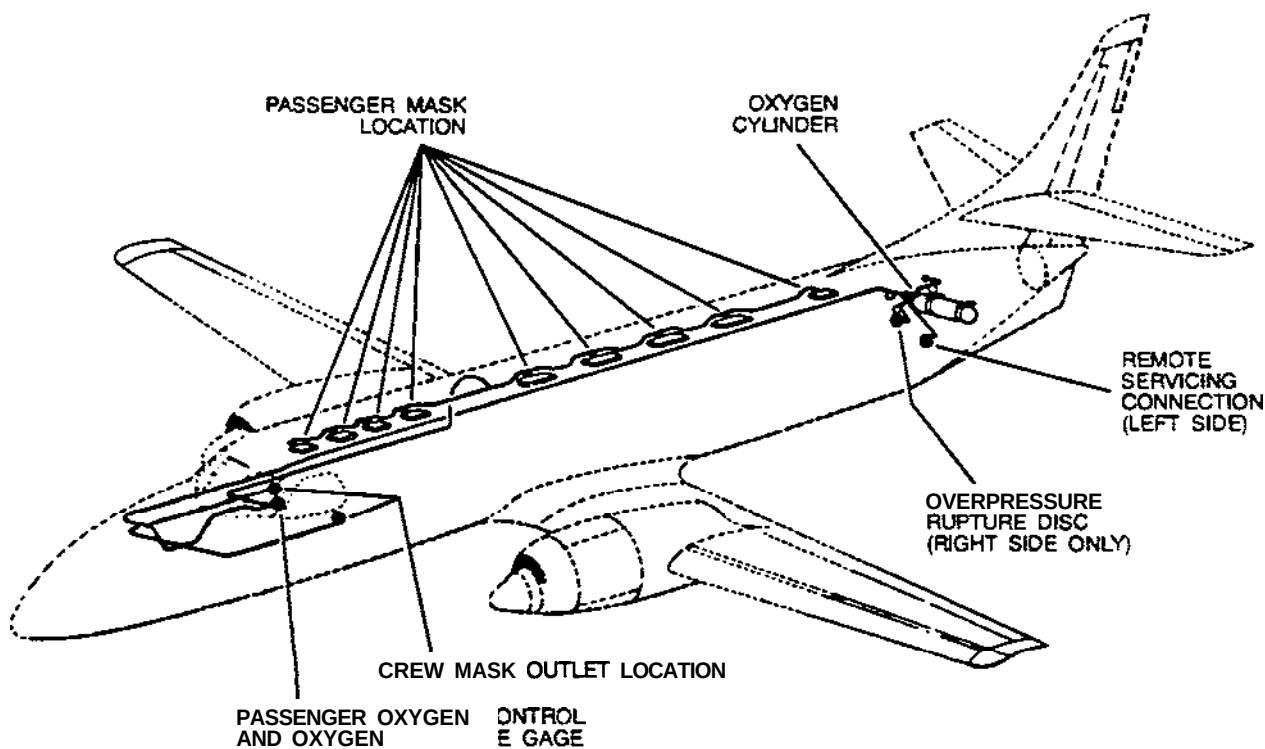


Figure 17-2. Oxygen System Component Locations–Merlin IVC



Oxygen is available at the crew mask outlets anytime the manual shutoff valve on the oxygen cylinder is open. The outlets are located to the left and right of the pilot and copilot

above the side consoles. The crew masks may be left plugged into the outlets because oxygen does not flow unless the pilots inhale.

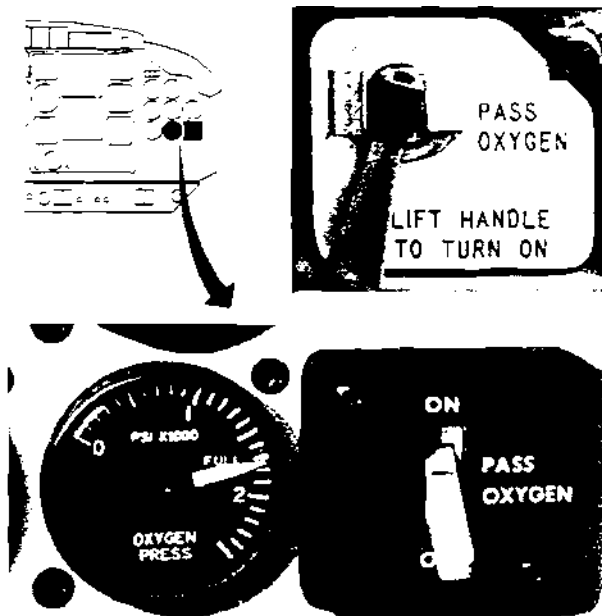


Figure 17-4. Passenger Oxygen Control and Oxygen Pressure Gage

Passenger Oxygen Toggle Control and Oxygen Pressure Gage

A PASS OXYGEN toggle control (Figure 17-4) is located on the lower right side of the instrument panel or under the copilot's side window. The toggle control is used to control the oxygen flow to the passengers. The electrically powered oxygen pressure page (Figure 17-4), mounted on the right side of the instrument panel, allows pilots to monitor the pressure within the oxygen cylinder.

Remote Servicing and Overpressure Rupture Disc

All airplanes incorporate a remote servicing connection and a manual shutoff valve as a part of the cylinder and its pressure regulator. If the manual shutoff valve is closed, pressure will **still** be indicated on the oxygen pressure gage, but no oxygen will be available to the airplane occupants.

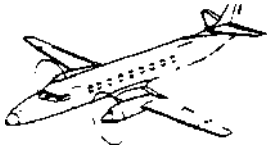


OUTLET

READY FOR USE

IN USE

Figure 17-3. Crew Oxygen Mask and Outlet (Typical)



If the oxygen cylinder becomes overpressurized, the regulator dump valve activates, venting the entire contents of the oxygen cylinder overboard through a rupture disc indicator. If the green disc is missing, it indicates that the system has been discharged and should be serviced.

Component Locations

A 49- or 115-cubic foot, 1,850-psi oxygen cylinder is located behind the aft baggage compartment bulkhead. Additionally, the Metro III may have two bottles for a total of 179 cubic feet. The remote servicing connection (Figure 17-5) is also located in the vicinity of the oxygen cylinder. The overpressure rupture disc is located on the right side of the tail section (Figure 17-6).

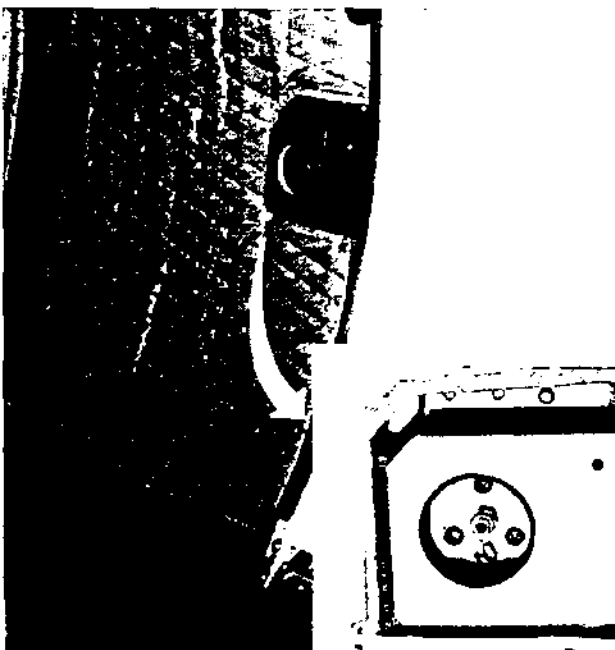


Figure 17-5. Oxygen Cylinder Filler

Passenger Oxygen Masks

Passengers have constant-flow oxygen masks. Oxygen flow to the passengers is controlled by the PASS OXYGEN toggle control in the cockpit. There are enough masks for all the passengers.

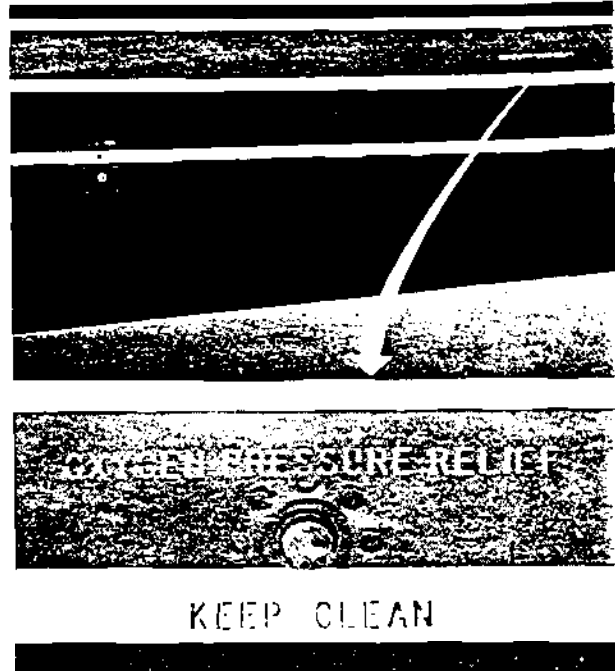


Figure 17-6. Overpressure Rupture Disc

Merlin IVC Mask Location

In the Merlin IVC, the passenger masks (Figure 17-7) are stowed in nine overhead compartments.

Metro III Mask Location

The Metro III has a flush-mounted passenger oxygen mask outlet at each passenger seat. Masks are usually stowed in seatback pockets and must be plugged into the outlet before use. Drop-out masks similar to this one are available as an option.

OPERATION

See Figure 17-8 for a diagram of the oxygen system. The systems for the Merlin IVC and Metro III differ slightly but the basic operation of both oxygen systems is the same.



Figure 17-7. Passenger Oxygen Mask—
Merlin IVC

The oxygen cylinder incorporates a pressure regulator along with the manual shutoff valve. The regulator reduces the high-pressure oxygen to low pressure for use by the crew and passengers. Oxygen is delivered through low-pressure tubing to overhead compartments in the Merlin IVC and to flush-mounted outlets at each passenger seat in the Metro III. **On all** airplanes, the oxygen is also routed to outlets next to the pilot and copilot seats.

On the **Merlin IVC**, the passenger masks drop out of the overhead compartments when the **PASS OXYGEN** toggle control is placed in the **ON** position. The access panels open automatically, deploying the masks. When the passenger pulls down the mask, it pulls the release pin on its shutoff valve, and oxygen begins to flow.

In the **Metro III**, when the **PASS OXYGEN** toggle control is placed in the **ON** position, oxygen is available at each of the passenger outlets. The passenger must remove the passenger oxygen mask from its stowed position and plug the mask's oxygen line into the outlet. Once the **mask** has been connected to the outlet, oxygen begins to flow at a constant rate.

Oxygen is available at the crew mask outlets whenever the manual shutoff valve is open. Each crewmember has a diluter-demand mask with a **NORMAL/100%** flow selector.



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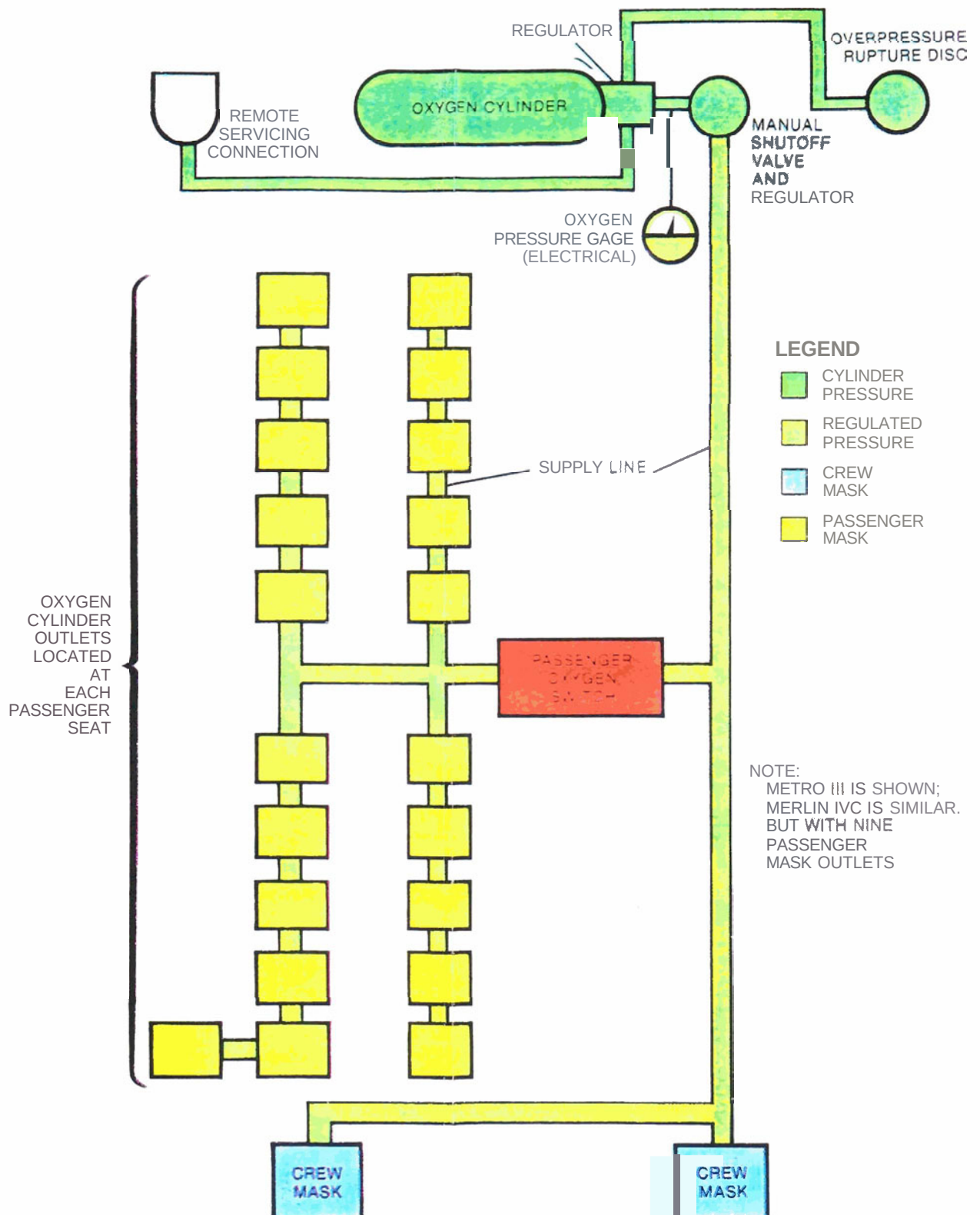


Figure 17-8. Oxygen System Diagram

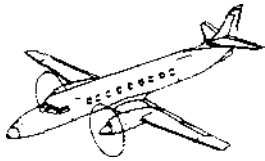


Table 17-1 is included as a typical example of the effects of hypoxia at various altitudes when pressurization is lost.

LIMITATIONS

NOTE

For oxygen duration under various conditions and passenger loads, consult the appropriate AFM.

REQUIRED EQUIPMENT

Flight Above 15,000 Feet

- Oxygen system, including a mask for one crewmember
- Supplemental oxygen system, including one mask for the passengers

Flight Above 25,000 Feet

- Supplemental oxygen system, including a mask for each crewmember and passenger.

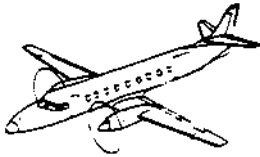
TABLE 17-1. AVERAGE TIME OF USEFUL CONSCIOUSNESS

35,000 feet.....	1/2 to 1 minute
30,000 feet.....	1 to 2 minutes
28,000 feet.....	2 to 3 minutes
25,000 feet.....	3 to 5 minutes
22,000 feet.....	5 to 10 minutes
12,000 to 18,000 feet.....	30 minutes or more



QUESTIONS

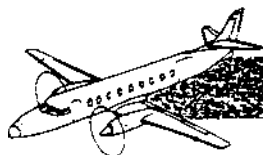
1. Any time the manual shutoff valve is open, oxygen is available at the:
 - A. Passenger mask compartments
 - B. Oxygen cylinder
 - C. Crew mask outlets
 - D. Crew oxygen masks
2. Oxygen flow to the passenger masks is controlled by the:
 - ☒ A. PASS OXYGEN toggle control
 - B. MANUAL shutoff valve control switch
 - C. Cabin altitude controller
 - D. Bleed-air valves
3. Upon a loss of pressurization, the average time of useful consciousness at an altitude of 25,000 feet is:
 - A. $\frac{1}{2}$ to 1 minute
 - B. 30 minutes or more
 - C. 1 to 2 minutes
 - D. 3 to 5 minutes
4. When the oxygen shutoff valve is positioned to OFF:
 - A. Oxygen is available to the crew only.
 - B. The oxygen pressure gage reads 0 psi.
 - C. The oxygen pressure gage reads low-pressure in the system.
 - D. The oxygen pressure gage reads cylinder high pressure.



APPENDIX A

Table A-1. CONVERSION FACTORS

<u>Multiply</u>	<u>By</u>	<u>To Obtain</u>
centimeters	0.3937	inches
kilograms	2.2046	pounds
kilometers	0.621	statute miles
kilometers	0.539	nautical miles
liters	0.264	gallons
liters	1.05	quarts (liquid)
meters	39.37	inches
meters	3.281	feet
millibars	0.02953	in. Hg (32° F)
feet	0.3048	meters
gallons	3.7853	liters
inches	2.54	centimeters
in. Hg (32° F)	33.8639	millibars
nautical miles	1.151	statute miles
nautical miles	1.852	kilometers
pounds	0.4536	kilograms
quarts (liquid)	0.946	liters
statute miles	1.609	kilometers
statute miles	0.868	nautical miles



APPENDIX B

Appendix B presents a color representation of all the annunciator lights in the airplane.

Please unfold page B-1 to the right and leave it open for ready reference as the annunciators are cited in the text.

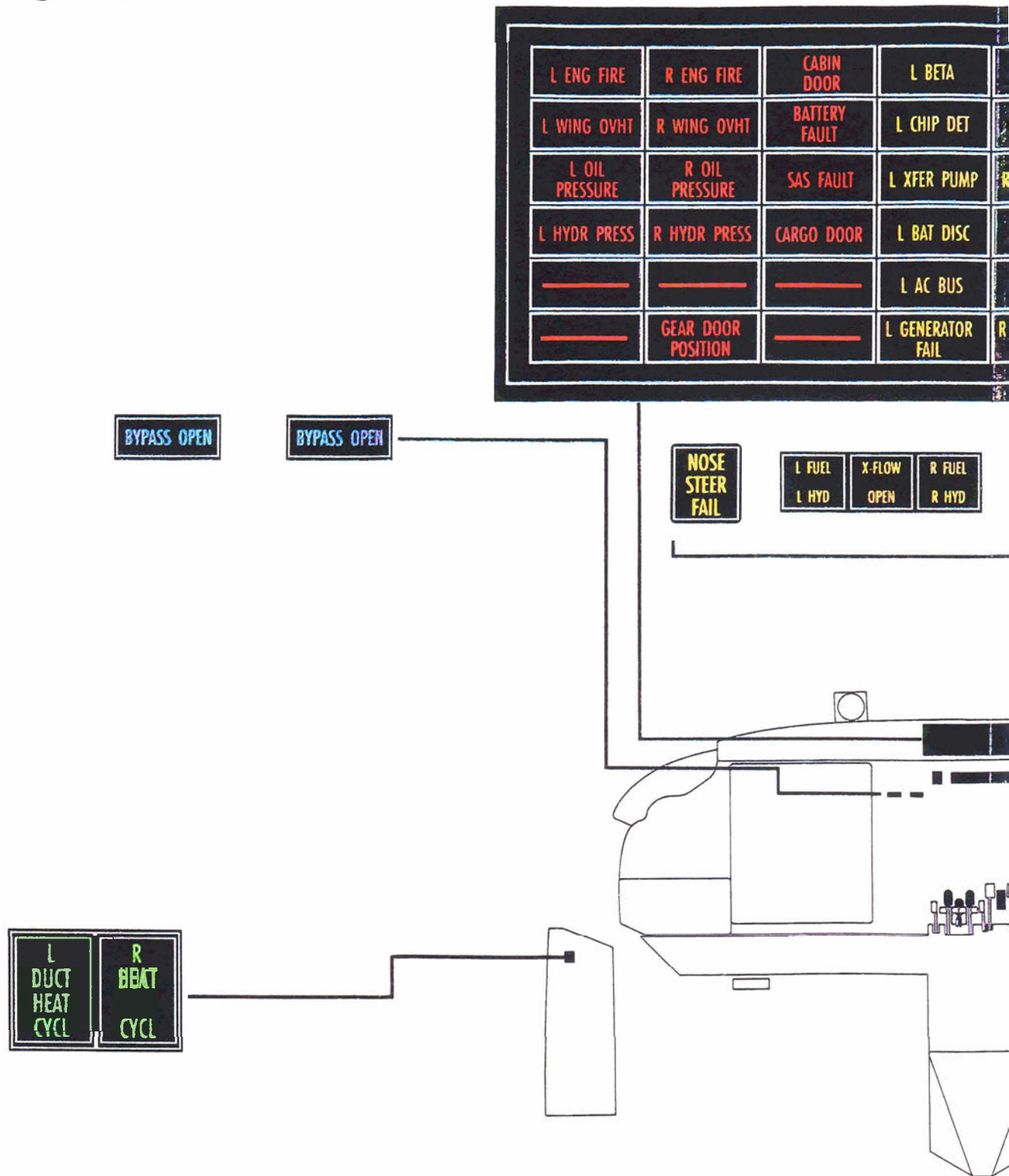


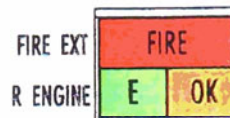
Figure B-1.



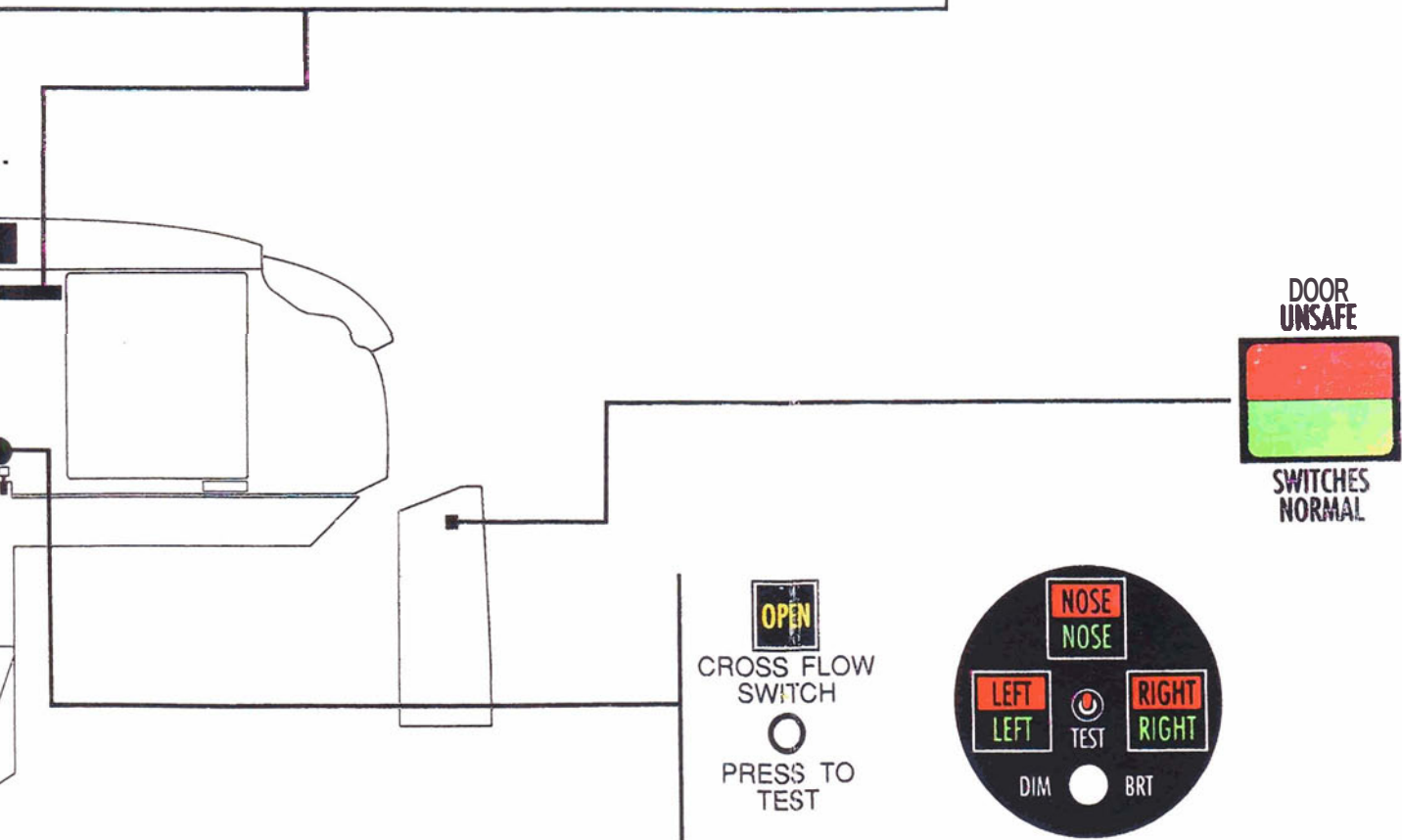
R BETA	LOW SUCTION	L INTAKE HT	R INTAKE HT
CHIP DET	CABIN ALTITUDE	L W/S HT	R W/S HT
XFER PUMP	GPU PLUG IN	_____	_____
BAT DISC	L SRL OFF	SAS ARM	SAS DEICE
AC BUS	R SRL OFF	AWI NO. 1 PUMP ON	AWI NO. 2 PUMP ON
GENERATOR FAIL	ANTI-SKID	NOSE STEERING	_____



FIRE EXT
L ENGINE

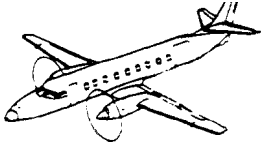


FIRE EXT
R ENGINE



Annunciators

PURPOSES ONLY



APPENDIX C

C-26 TPE331-12UA DIFFERENCES

Two versions of the C-26 are currently in use, the primary distinction being in the replacement of the standard engine with the Garrett TPE331-12UA. Changes in the -12UA engine airplane are described below.

ELECTRICAL SYSTEM

Maximum continuous load for each generator is 200 amps on the ground. 300 **amps** in flight up to FL 250. and 250 **amps** above FL 250. Louvers on the lower **right side** of each nacelle improve generator cooling.

POWERPLANT

The engine is a Garrett TPE331-12UA-701G, flat-rated at 1,100 shp wet and 1,000 shp dry. **Engine instrument markings, operations, and limitations are unchanged from those of the TPE331-11U engine.**

An access door on the left of each nacelle permits visual inspection of the oil filter pop-out pin.

Two amber caution lights on the annunciator panel, labeled "**R FUEL FILTER**" and "**L FUEL FILTER**," illuminate if the fuel filter is being bypassed due to blockage.

PNEUMATIC SYSTEM

The bleed-air switches have three positions: HIGH, LOW, and OFF. HIGH provides the normal amount of engine bleed air to the air-conditioning and pressurization systems. LOW provides approximately $\frac{2}{3}$ the flow of HIGH.

The bleed air will normally be in HIGH or OFF for takeoff. Certain performance charts are predicated on the basis of using LOW bleed for improved cruise performance.