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LEARJET 30 SERIES PILOT TRAINING MANUAL

Record of Revision No. .01

This is a complete reprint of Revision .01 of the *Learjet 30 Series Pilot Training Manual, Volume 2*.

The portion of the text or figure affected by the current revision is indicated by a solid vertical line in the margin. A vertical line adjacent to blank space means that material has been deleted. In addition, each revised page is marked “Revision .01” in the lower left or right corner.

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

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LEARJET
30 SERIES

PILOT TRAINING MANUAL
VOLUME 2
AIRCRAFT SYSTEMS

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Courses for the Learjet 30 Series are taught at the following FlightSafety learning centers:

Tucson Learning Center
1071 E. Aero Park Boulevard\
Tucson, AZ 85706
(800) 203-5627
FAX (520) 918-7111

Wichita (Learjet) Learning Center
Two Learjet Way
Wichita, KS 67209
(800) 491-9807
FAX (316) 943-0314

West Palm Beach Learning Center
3887 Southern Boulevard
West Palm Beach, FL 33406
(800) 769-6763
FAX (561) 515-2899

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 FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety 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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SYLLABUS

LEARNING CENTER INFORMATION

FlightSafety International is an aviation training company that provides type-specific training programs for over 50 different models of aircraft, using a fleet of 100 simulators. FlightSafety operates over 38 Learning Centers, including Centers in Europe and Canada.

Training for the Learjet is conducted at the FlightSafety Learning Centers in Wichita, Kansas; Tucson, Arizona; and West Palm Beach, Florida. The Centers are owned and operated by FlightSafety International and are located at the following address:

FlightSafety International
Learjet Learning Center
Two Learjet Way
P.O. Box 9320
Wichita, KS 67209

FlightSafety International
Tucson International Airport
1071 East Aero Park Boulevard
Tucson, Arizona 85706

FlightSafety International
West Palm Beach International Airport
3887 Southern Blvd.
West Palm Beach, Florida 33406-1431

DESCRIPTION OF TRAINING FACILITIES

Each classroom and briefing room is adequately heated, lighted, and ventilated to conform to local building, sanitation, and health codes. The building construction prevents any distractions from instruction conducted in other rooms or by flight operations and maintenance operations on the airport.

Classrooms are equipped for computer-based presentations, controlled from a specially designed lectern. A standard overhead projector is mounted on the lectern. Most lectern and student positions are equipped with a student responder system. Cockpit panel posters are on display.

Briefing rooms are equipped with cockpit panel posters, a white liquid chalkboard, a table, and chairs for individual or small-group briefings. A floor plan of the Centers follows.

Wichita Learning Centers

Classroom Number	Size in Feet	Student Capacity
1	20X25	18
2	20X25	12
3	20X25	18
4	20X20	12
5	20X30	24
6	20X20	12
7	20X25	18

Tucson Learning Centers

1	22X19	18
2	22X19	18
3	22X20	18
4	24X20	18
5	24X20	12
6	16X17	10
7	8X12	2

West Palm Beach Learning Centers

212	18X20	18
214	18X22	12
237	10X10	3

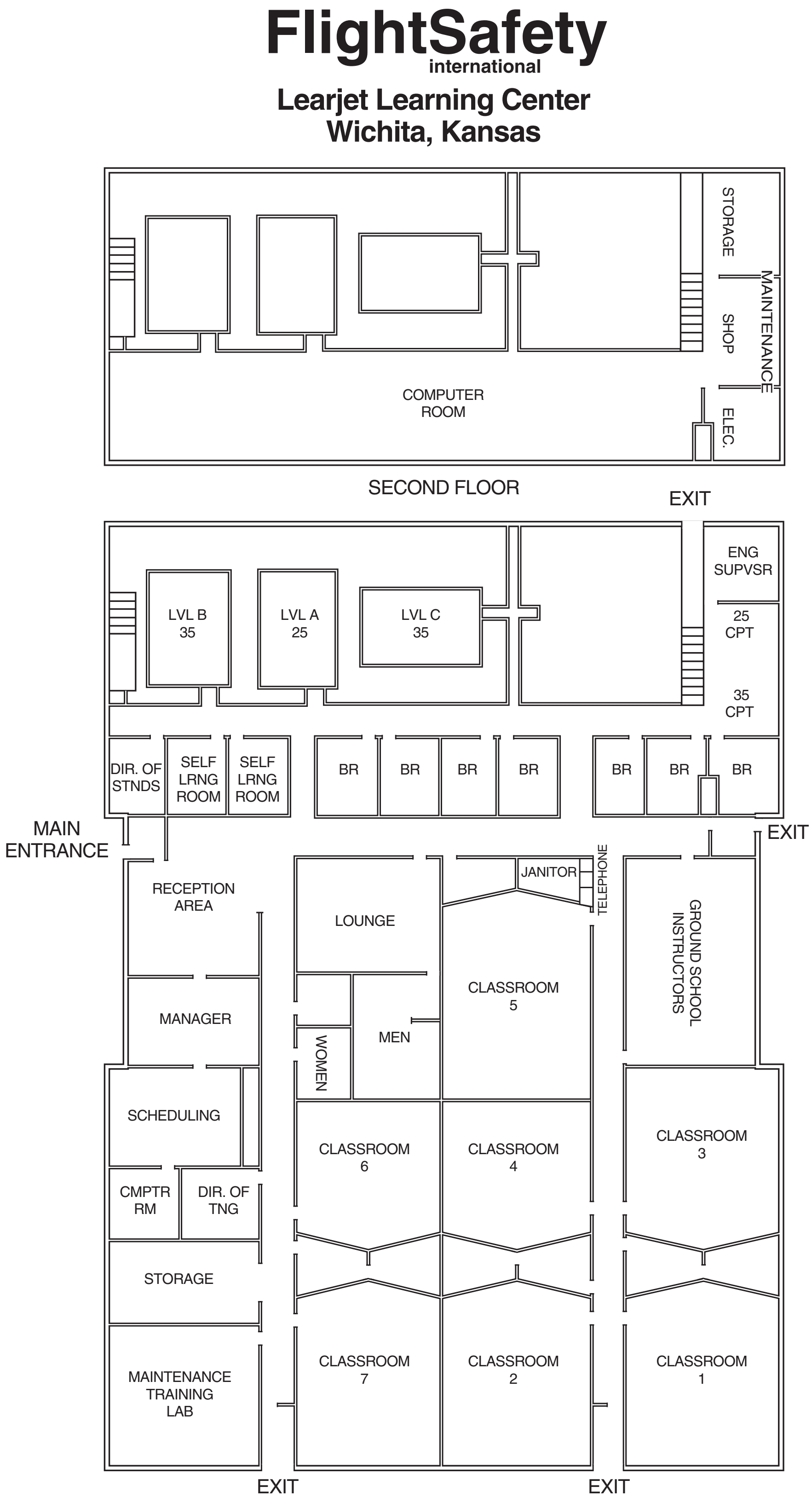

Figure SYL-1. Wichita Facility Floor Plan

FlightSafety international

Learjet Learning Center
Tucson, Arizona

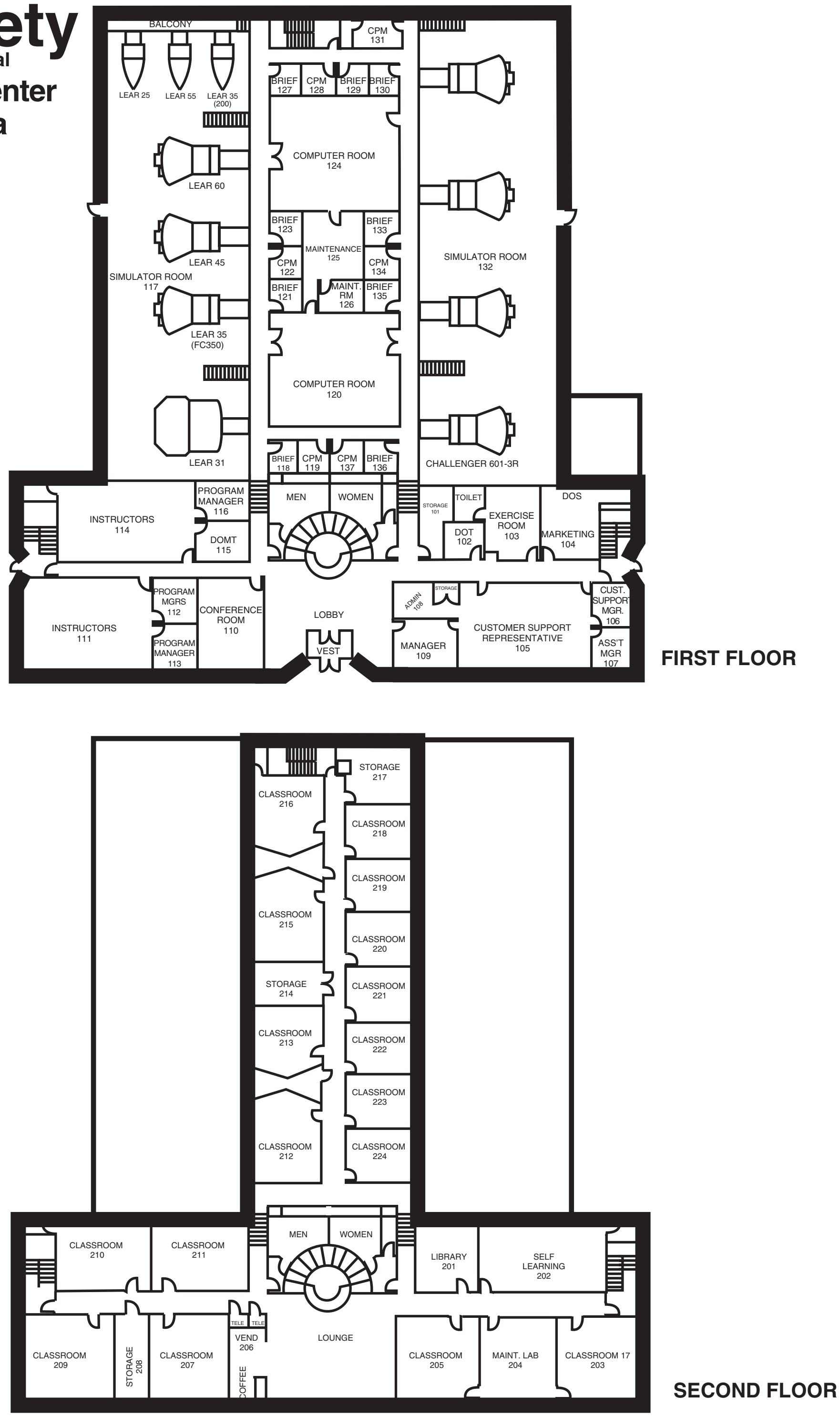

Figure SYL-2. Tucson Facility Floor Plan

FlightSafety international

West Palm Beach Learning Center

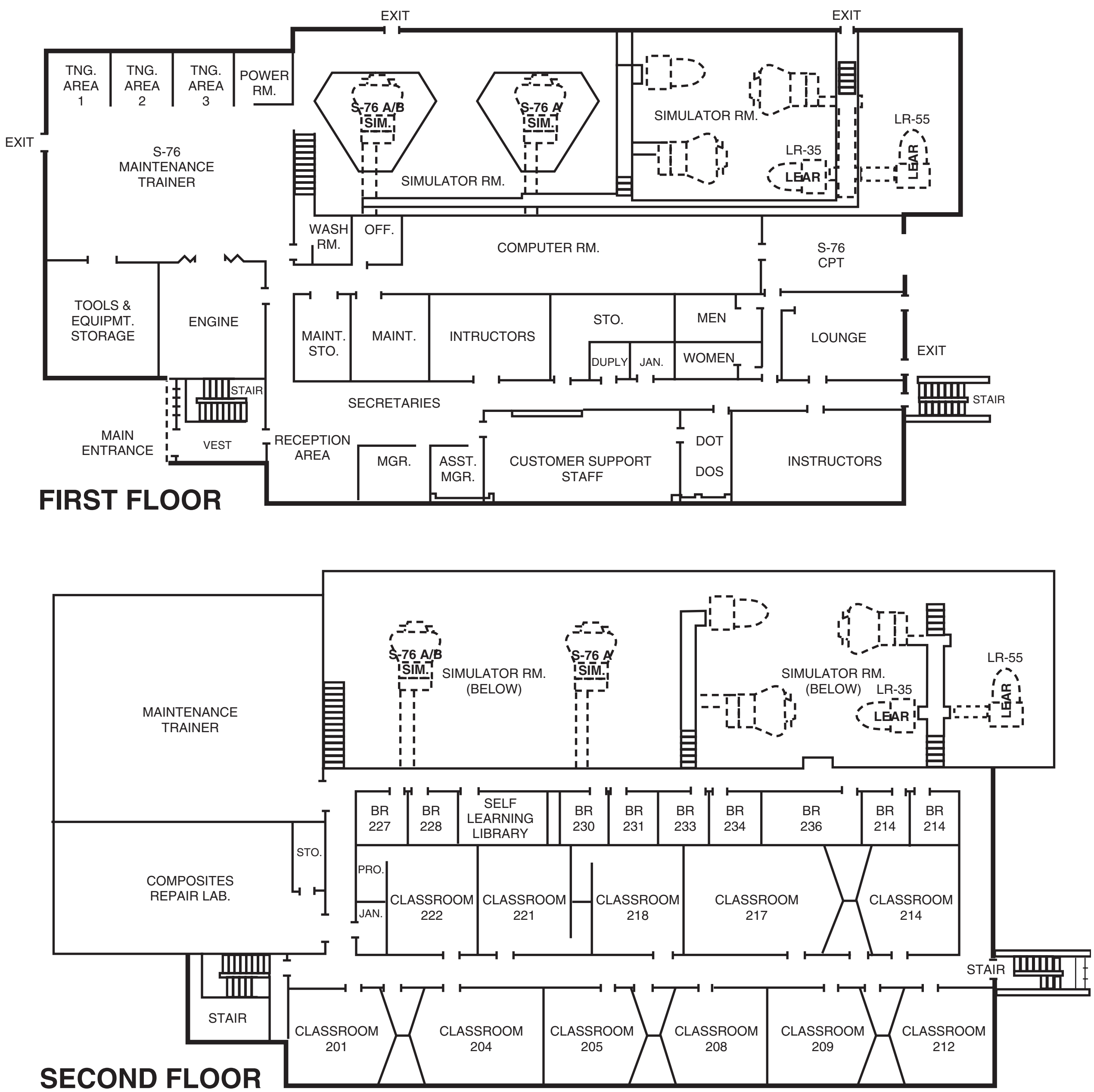

Figure SYL-3. West Palm Beach Facility Floor Plan

TYPE OF AIRCRAFT

LR-JET

CATEGORY OF TRAINING

Initial and Transition training for a LR-JET type rating added to an existing pilot certificate or the issuance of an Airline Transport Pilot Certificate with a LR-JET type rating.

DUTY POSITION

Pilot-in-Command (PIC)

CURRICULUM TITLE

LR-JET Series Pilot Training Course and Advance Simulation Training Program

CURRICULUM PREREQUISITES

CORE TRAINING CURRICULUM PREREQUISITES

§61.63

A pilot may enroll in this course and complete all of the items of the practical test required for a LR-JET type rating that are authorized to be accomplished in the flight simulator, then complete the items not approved for flight simulator in flight in a LR-JET Series airplane, if the pilot:

1. Holds a private pilot certificate with an airplane rating.

2. Holds an instrument rating.

3. Has a minimum of 1,000 hours flight experience in airplanes as a pilot. (May be waived at the discretion of the Center Manager).

4. Holds a MEL catagory rating without centerline thrust limitation.

§61.57

A pilot who meets the above requirements of §61.63 may concurrently apply for an Airline Transport Pilot certificate with a LR-JET type rating, providing the pilot:

1. Holds a commercial pilot certificate or an ICAO recognized Airline Transport Pilot or Commercial Pilot License without restrictions.

2. Meets the eligibility requirements of §61.153.

3. Has passed the written test required by §61.155.

4. Meets the experience requirements of §61.159.

Prerequisite Experience

The curriculum is designed to accommodate pilots with varied levels of experience. Depending on the pilot's experience and fight simulator approval level, the pilot may qualify for either 100% flight simulator curriculum or a combination curriculum using both flight simulator and aircraft. If a 100% flight simulator practical test is not accomplished, then aircraft training and testing will be required.

1. Pilots who meet the appropriate requirements in §61.63 (e)(4) or §61.157 (g)(3) may obtain an unlimited type rating.

2. Pilots who meet the appropriate requirements in §61.63 (e)(5) or §61.157 (g)(4) may be issued an added rating with pilot-in-command limitations. Fifteen hours of supervised operating experience as PIC accomplished IAW §61.63 (e)(8)(ii) or §61.15(g)(6)(ii) will be required to remove this limitation.

3. Pilots who do not meet the appropriate requirements in items 1 or 2 above may still be issued a type rating with pilot-in-command limitations. Twenty-five hours of supervised operating experience as PIC accomplished IAW §61.63 (e)(12)(ii) or §61.157 (g)(9)(ii) will remove this limitation.

4. Pilots completing training and testing who do not want an SOE limitation on their certificate may complete the following tasks on a static airplane or in flight, as appropriate:

    a. Preflight Inspection

    b. Normal Takeoff

    c. Normal ILS Approach

    d. Missed Approach; and

    e. Normal Landing

5. Pilots who receive training/testing in a Level A or B simulator must complete the practical test in the aircraft.

FLIGHTSAFETY TRAINING POLICY

The policy is to train to "Proficiency" based on need-to-know information.

DESCRIPTION OF INITIAL COURSE

The LR-JET Initial Course is scheduled for twelve days and consists of the following programmed hours:

GROUND TRAINING

General Operational Subjects	6.0
Aircraft Systems	31.0*
Systems Integration	2.0
Prebrief/Postbrief	9.0
Oral Exam and Pre/Postbriefings for Qualification	3.0
Total Ground Training	51.0

* If a pilot requires FMS training, he/she will receive an additional 4.0 hours of training.

FLIGHT TRAINING

Flight Training (Simulator)	10.0
Aircraft Flight Training (Typical) (If Necessary)	2.0
Total Flight Training	12.0

QUALIFICATION CHECK

Flight Simulator	2.0
Aircraft	0.5
Total Qualification Hours	2.5

An applicant may choose to take the entire practical test in the aircraft rather than in the simulator.

The 10 hours of flight simulator training is for left-seat training. Normally, a pilot is trained as a crew with another pilot. A pilot training not as a crew will receive an additional 5 hours of simulator training.

DESCRIPTION OF TRANSITION COURSE (FAR 135)

The LR-JET Initial Course is scheduled for twelve days and consists of the following programmed hours:

GROUND TRAINING

General Operational Subjects	6.0
Aircraft Systems	31.0*
Systems Integration	2.0
Prebrief/Postbrief (Simulator)	9.0
Oral Exam and Pre/Postbriefings for Qualification	3.0
Total Ground Training	51.0

* If a pilot requires FMS training, he/she will receive an additional 4.0 hours of training.

FLIGHT TRAINING

Flight Training (Simulator)	10.0
Total Flight Training	10.0

QUALIFICATION CHECK

Flight Simulator	2.0
Aircraft Preflight	0.5
Loft	4.0
Total Qualification Hours	6.5

The 10 hours of flight simulator training is for left-seat training. Normally, a pilot is trained as a crew with another pilot. A pilot training not as a crew will receive the hours specified in the operators approved training program.

COURSE OBJECTIVES

Upon the completion of this course, the pilot will have the necessary knowledge and skills to demonstrate that he/she is the master of the aircraft, with the successful outcome of a procedure or maneuver ***never*** in doubt, and to meet or exceed the requirements/standards listed in FAA-S-8081-5 Airline Transport Pilot and Type Rating Practical Test Standards.

Successful completion of the LR-JET Pilot Training Course will satisfy the requirements for the following:

- Second-in-Command qualifications as specified in FAR 61.55
- Pilot-in-Command recent flight experience as specified in FAR 61.57
- Pilot-in-Command Proficiency Check as specified in FAR 61.58
- Additional Aircraft Ratings as specified in FAR 61.63
- Category II Pilot Authorization as specified in FAR 61.67
- Airplane Rating Requirements as specified in FAR 61.157
- Practical Test Requirements for Airplane ATP Certifications and Associated Class and Type Ratings

Successful completion of the LR-JET Initial/Transition Pilot Training Course and the subsequent LR-JET practical test will also satisfy the requirements for the following:

- Instrument Competency Check as specified in FAR 61.57 (d)
- Initial Equipment/Transition training for FAR 135 certificate holders contracting with FlightSafety

TRAINING SCHEDULE (TYPICAL)

Listed below is a typical schedule for the pilot training curriculum. On occasion, the schedule may be rearranged to meet the needs of the client or Center. In addition, the times allotted for each lesson may vary due to pilot experience and class size. The schedule consists of 12 training days.

NOTE

Simulator hours reflect left-seat time for one pilot, performing all pilot flying duties. In addition, 1.0 hour for briefing and .5 hour for debriefing are allocated.

Day 1	FlightSafety Administration	0.5
	Classroom	7.0
	Aircraft General	
	Electrical	
	Lighting	
Day 2	Classroom	7.5
	Master Warning System	
	Fuel	
	Powerplant	
	Fire Protection	
Day 3	Classroom	7.5
	Pneumatics	
	Ice and Rain Protection	
	Air Conditioning	
	Pressurization	
	Hydraulic Systems	
Day 4	Classroom	7.5
	Landing Gear and Brakes	
	Flight Controls	
	Avionics	
Day 5	Classroom	7.0
	Miscellaneous Systems	
	Weight and Balance	
	Performance	
	Examination	
	CPT	2.0

Day 6	Simulator	2.0
	Simulator Period No. 1	
Day 7	Simulator	2.0
	Simulator Period No. 2	
Day 8	Simulator	2.0
	Simulator Period No. 3	
Day 9	Simulator	2.0
	Simulator Period No. 4	
	Aircraft Preflight Training	1.0
Day10	Simulator	2.0
	Simulator Period No. 5	
Day 11	Oral Equipment Examination	2.0
	Simulator-Type Rating Qualification Check	2.0
	Aircraft Preflight Check	0.5
Day 12	Simulator	2.5
	LOFT (FAR 135)	

CHAPTER 1
AIRCRAFT GENERAL

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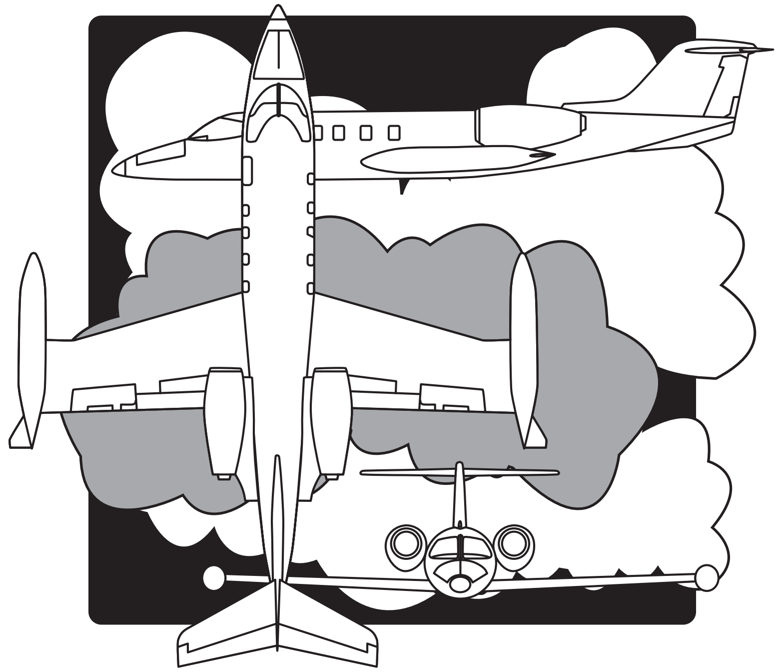
ILLUSTRATIONS

CHAPTER 1
AIRCRAFT GENERAL



INTRODUCTION

This training manual provides a description of the major airframe and engine systems installed in the Learjet 35/36.

This chapter covers the structural makeup of the airplane and gives a general description of the systems. No material is meant to supersede any of the manufacturer's system or operating manuals.

The material presented has been prepared from the basic design data, and all subsequent changes in airplane appearance or system operation will be covered during academic training and subsequent revisions to this manual.

The Annunciator Panel section in this manual displays all light indicators, and page ANN-1 should be folded out and referred to while studying this manual.

GENERAL

The Learjet 35/36 is certificated under FAR Part 25 as a two-pilot transport category airplane, approved for all-weather operation to a maximum altitude of 45,000 feet.

STRUCTURES

GENERAL

Figure 1-1 shows the Learjet 35/36. The structure consists of the fuselage, the wing, the empennage, and flight controls. The discussion on the fuselage includes all doors and windows. Figure 1-2 shows the general dimensions of the airplane.

Figure 1-3 displays the airplane turning radius.

Figure 1-4 displays the danger areas around the Learjet 35/36 presented by the weather radar emission cone, engine intakes, and engine exhaust cones.

Figure 1-1. Learjet 35/36

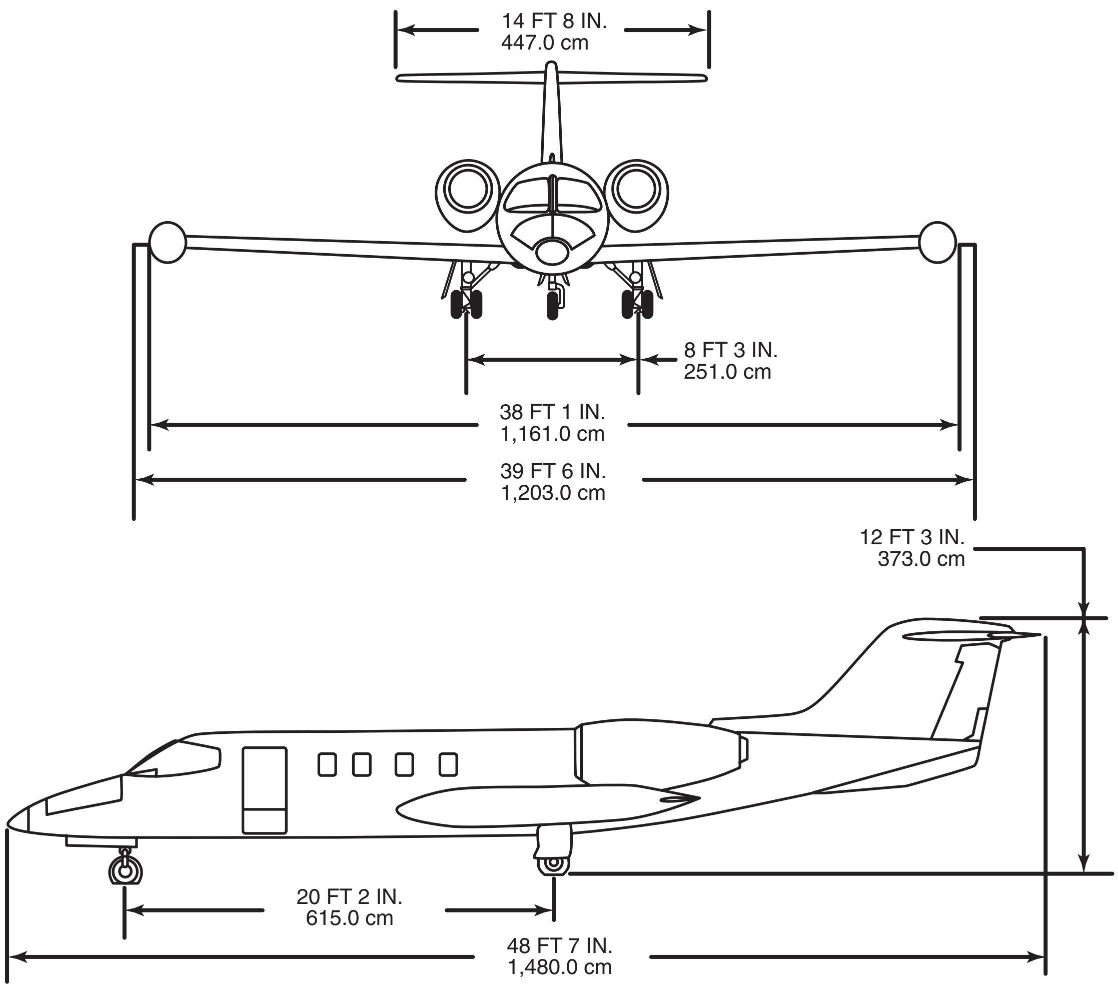


Figure 1-2. General Dimensions

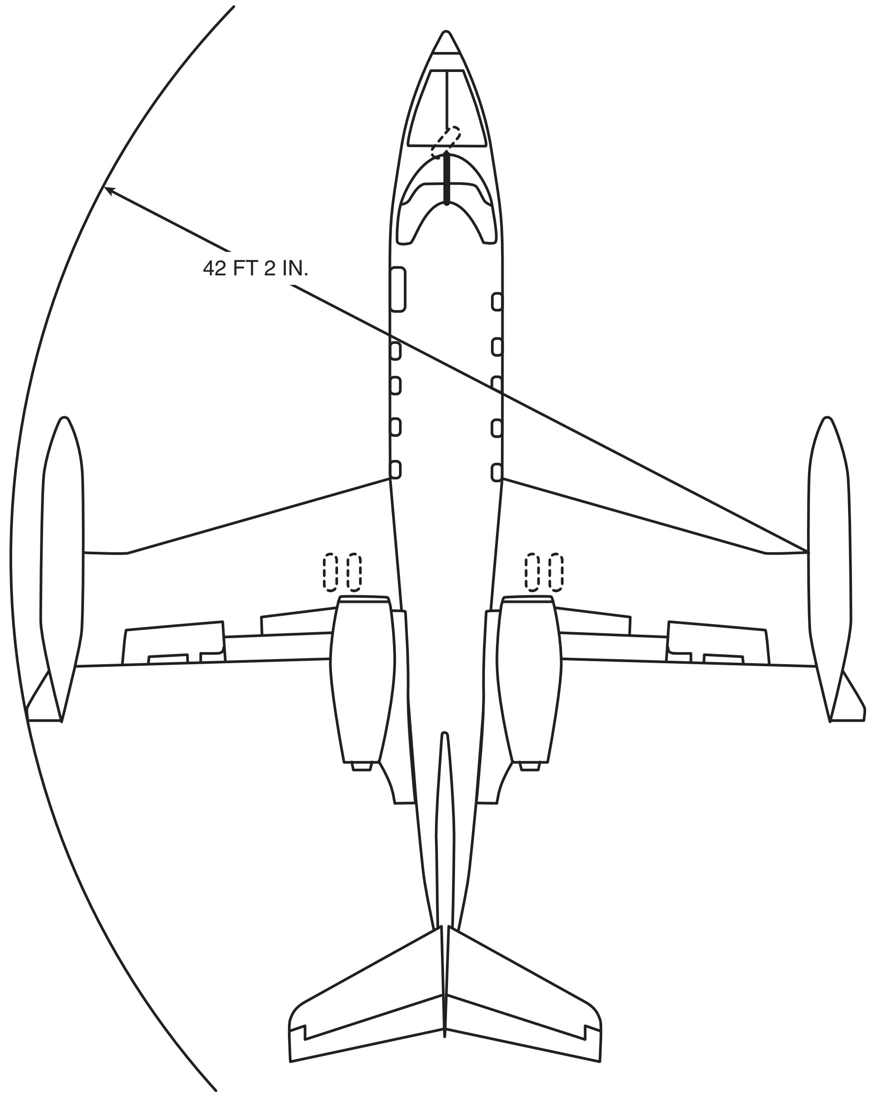


Figure 1-3. Turning Radius

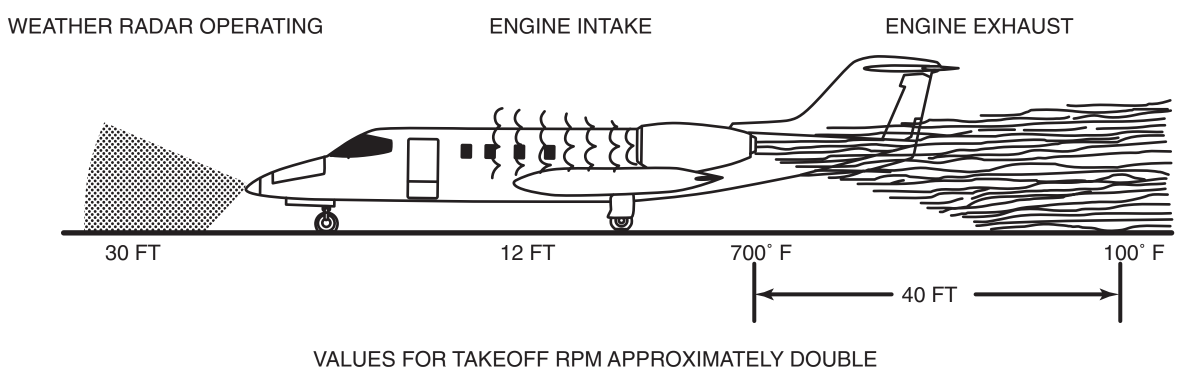


Figure 1-4. Danger Areas

FUSELAGE

General

The fuselage is constructed of stressed all-metal skin with stringers. It employs the area rule design to reduce aerodynamic drag, and has four basic sections. (See Figure 1-5.) They are:

1. The nose section which extends from the radome aft to the forward pressure bulkhead.
2. The pressurized section, which includes the cockpit and passenger areas, extends aft to the rear pressure bulkhead. On 36 models this bulkhead is further forward than on 35 models to provide space for the larger fuselage tank.
3. In both models, the fuselage fuel section starts just aft of the rear pressure bulkhead and extends to the tailcone.
4. The tailcone section extends aft of the fuel section.

The fuselage also incorporates attachments for the wings, tail group, engine support pylons, and the nose landing gear.

In addition to the pressurized cockpit and passenger compartments, the fuselage includes the nose wheel well, an unpressurized nose compartment, and a tailcone compartment used for equipment installation.

Nose Section

The nose of the fuselage (Figure 1-6) is formed by the radome. Aft of the radome is the nose compartment.

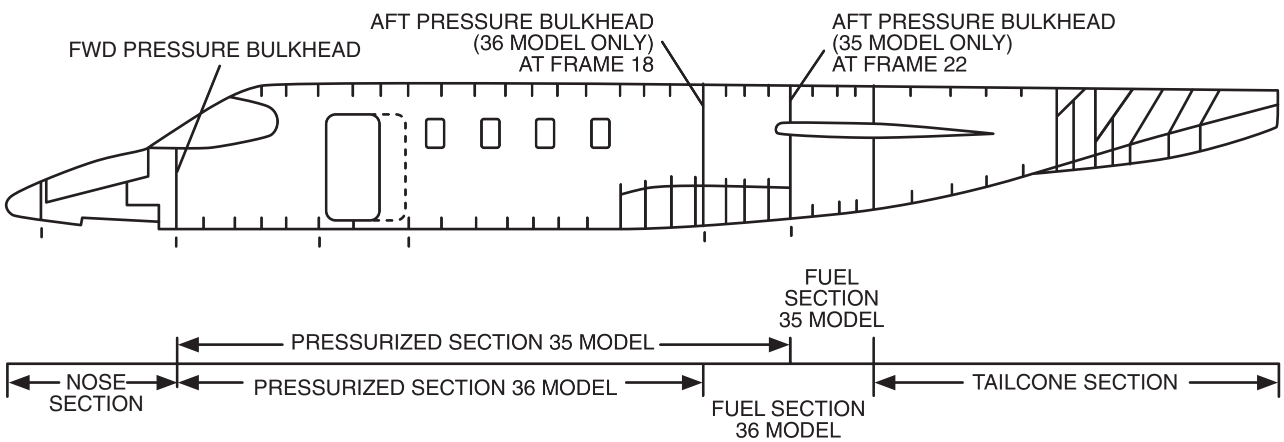


Figure 1-5. Fuselage Sections

Figure 1-6. Radome

Figure 1-7. Nose Compartment

The nose compartment access panels are on top of the fuselage (Figure 1-7), forward of the windshield. The panels must be removed for access to various electronic components, oxygen bottle (when installed in the nose), emergency air bottle, and the alcohol anti-icing reservoir.

Pressurized Section

The pressurized cabin lies between the forward pressure bulkhead and the aft pressure bulkhead, and includes the cockpit and passenger compartment. Within the passenger compartment is a 500-pound-capacity baggage area at the back of the cabin, a lavatory, a cabinet for storage of provisions, and galley equipment (depending on the airplane).

The passenger-crew door is located on the left side of the fuselage, just aft of the cockpit. One of the windows on the right side of the cabin serves as an emergency exit.

The cockpit seats two pilots and is fitted with a large, curved, two-piece windshield.

Passenger-Crew Door

The primary entrance and exit for passengers and crewmembers is through the clamshell door, located on the left side of the forward fuselage. (See Figure 1-8.) The standard entrance door is 24 inches wide, but there is an optional 36-inch door. The upper door serves as an emergency exit, and the lower door has integral entrance steps.

The upper portion of the door has both outside and inside locking handles connected to a common shaft through the door. Rotating either of these handles to the close position drives six locking pins into holes in the fuselage frame (three pins forward and three aft) and two pins through interlocking arms that secure the two door halves together.

The lower door has a single locking handle on the inside. Rotating the lower door handle to the closed (forward) position drives two pins into holes in the fuselage frame (one forward and one aft). There are a total of 10 locking pins on the two door sections.

To facilitate alignment of the upper door locking pins during closing, an electric actuator motor, torque tube assembly, and one or two hooks are installed in the lower door. The hooks engage rollers installed on the upper door and draw the two halves together. The actuator motor is operated from inside the airplane by a toggle switch on the lower door, and from the outside by a key switch. Should the motor fail, the hooks can still be operated

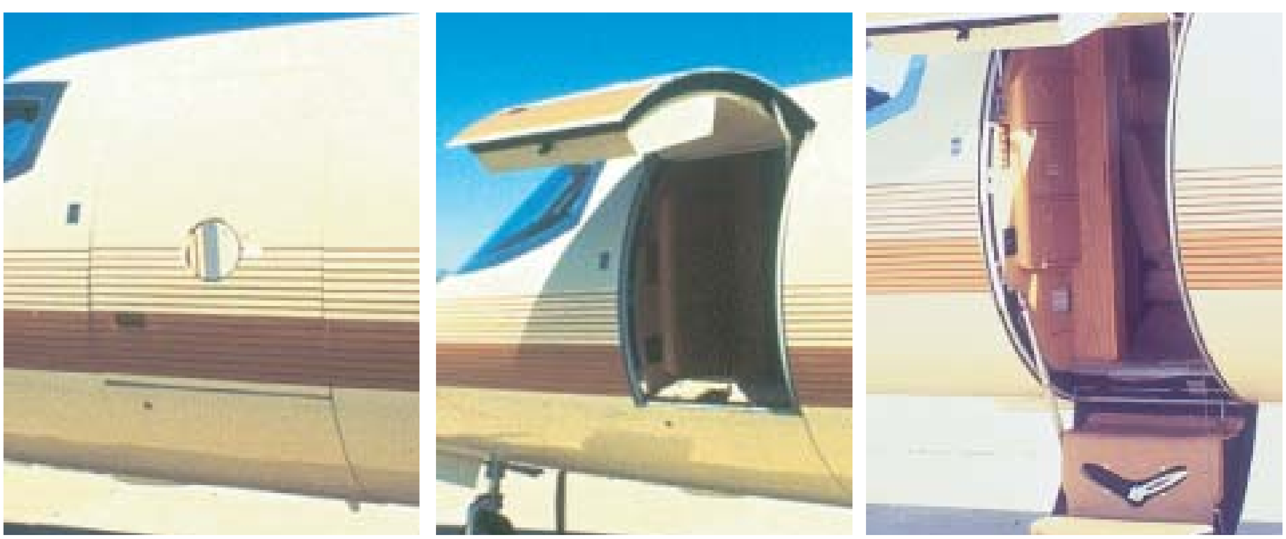


Figure 1-8. Passenger-Crew Door

manually from inside. Access is provided to the torque-tube mechanism through a panel in the lower door, and a ratchet handle provided in the airplane tool kit can be used to operate the torque-tube manually.

NOTE

One hook and roller is used on 24-inch doors, while two hooks and rollers are used on 36-inch doors.

When the door handles are in the closed position, the pins all contact microswitches. If any of the switches is not actuated, a red DOOR light illuminates on the annunciator panel. (See Annunciator Panel section.) If the light illuminates while the door is closed, eight inspection ports enable the crew to confirm the position of the door-frame latching pins by observing the position of two white alignment marks (Figure 1-9). The two latch pins which connect the upper and lower doors are visible through the upholstery gap at the interface and do not have white lines.

LOCKED

NOT LOCKED

Figure 1-9. Door Latch Inspection Port

When closing the doors from the inside, close and latch the lower door first. Then, close the upper door and actuate the door motor switch to the closed position. This engages the hooks over rollers in the upper door, and cinches the upper door down tight while allowing the locking pins to line up properly and meet the microswitches as the upper door handle is rotated to the closed position. The DOOR light will remain illuminated until the hooks are backed away from the upper door rollers by reverse operation of the door motor switch.

A secondary safety latch is installed on the lower door and is separate from the door-locking system. It consists of a notched pawl attached to the door. The pawl engages a striker plate attached to the frame when the door is closed.This engagement holds the lower door closed while the locking handle is being positioned to the locked position. Additionally, it prevents the door from falling open as soon as the door handle is opened. The latch is released by depressing the pawl.

Cables and hydraulic dampers are provided to stabilize the lower door when lowering it and when using it as a step. The 24-inch door has one cable and a hydraulic damper. The 36-inch door has two cables and may have an optional hydraulic damper. The cables are connected to takeup reels in the lower door and are also used to pull the door closed from inside the airplane.

The key switch is used to secure the door from the outside. By inserting a key into the switch and turning it in one direction, the actuator motor drives the hooks to engage the upper door rollers. Turning it in the other direction drives the hooks from the rollers to permit opening the door.

NOTE

Anytime the airplane is occupied with the entry doors locked, the hooks must be released. This permits opening the upper door for emergency egress.

The red DOOR light illuminated means:

- Any one of the 10 latch pins is not engaged with its respective microswitch.
- The hook drive mechanism is not completely retracted.
- The door is unsafe for takeoff.

A hollow neoprene seal surrounds the doorframe; the seal has holes to allow the entry of pressurized cabin air, forming a positive seal around the door.

Emergency Exit

A hatch near the right rear of the cabin (Figure 1-10) serves as an emergency exit for all occupants. A latching mechanism is accessible from inside and outside the cabin.

The inside latch handle, located at the top center of the window, is pulled inward to unlock. To open from the outside, depressing a PUSH button above the window releases a handle which must then be turned in the direction of the arrow stamped on the handle; then the hatch may be pushed inward.

Figure 1-10. Emergency Exit

Windows

Windshield

The windshield (Figure 1-11) is divided into two sections, the pilot's and copilot's halves, and is made up of three laminated layers of acrylic plastic. The windshield is approximately one inch thick. It is impact-resistant, heated or not, and was tested against 4-pound bird strikes at 350 knots.

Passenger Windows

The cabin windows (Figure 1-12), including the emergency exit window, are made up of two panes of stretched acrylic plastic with an air space between them. They are held apart and sealed air tight by a spacer.

Figure 1-11. Windshield

Figure 1-12. Windows Locations (Typical)

Fuel Section

The fuel section, located aft of the rear pressure bulkhead, contains the fuselage fuel cells.

As seen in Figure 1-5, the fuel section on 35 models is different from that on 36 models. On 36 models, the rear pressure bulkhead has been moved forward, allowing for four bladder cells rather than two, almost doubling fuselage fuel capacity.

Figure 1-13. Tailcone Door

Tailcone Section

The tailcone section extends aft from the fuel section to the empennage. The tailcone entry door (Figure 1-13) is located at the bottom of Page 1-8. The door is hinged at the forward edge and drops down when released by quick-release thumb latches, allowing access to the batteries, electrical components, fuel filters, fuel computers, refrigeration equipment, engine fire extinguishers, and hydraulic components.

There is an optional light switch in the tailcone equipment compartment. If inadvertently left on, it will be turned off by the door-closing action.

There is no cockpit indicator to warn the pilot if the door is open.

Figure 1-14. Learjet 35/36 Wing

WING

The Learjet 35/36 has a swept back, cantilevered, all metal wing (Figure 1-14) which is mounted to the lower fuselage and joined together at the fuselage. Most of the wing is sealed to form an integral fuel tank.

Eight fittings attaching the wings to the fuselage are designed to prevent wing deflections from inducing secondary loads in the pressurized fuselage. Ailerons are attached to the rear spar at three hinge points. The single-slotted Fowler flaps are attached to the inboard rear spar by tracks, rollers, and hinges. The spoilers are attached to the top of the wing surface by two hinges just forward of the flaps. The tip tanks are secured to the wing at two attach points.

The Learjet 35/36 wing is fitted with either vortex generators or boundary layer energizers. Whichever is used, they function to delay airflow separation over the ailerons at high Mach numbers.

Airplane Serial Nos. (SN) 35-002 through 35-278 and 36-002 through 36-044 (if not retrofitted with AAK 79-10) employ two rows for vortex generators bonded to the upper wing surface forward of both ailerons.

Subsequent serial-numbered airplanes and those modified with AAK 79-10 incorporate a "soft-flight" wing modification, which includes:

- Three rows of boundary layer energizers (BLEs) on each wing which perform the same function as vortex generators, but are more efficient. If ***any*** are missing, M_{MO} is reduced to 0.78 M_1 (FC200) or .77 M_1 (FC530).
- A full-chord stall fence on each wing, inboard of the aileron, which delays disruption of the airflow over the aileron at high angles of attack.

- A stall strip, affixed to the inboard section of each wing leading edge, which generates a buffet at high angle of attack to warn of an impending stall
- An aileron gap seal along the leading edge of each aileron

EMPENNAGE

The high-T-tail empennage (Figure 1-15) includes a vertical stabilizer with an attached rudder and a horizontal stabilizer with attached elevators.

The swept back vertical stabilizer is formed by five spars securely connected in the tailcone. It is the mounting point for the rudder and horizontal stabilizer. At the lower leading edge of the stabilizer is a dorsal fin which houses a ram-air scoop. Later model airplanes have the oxygen bottle located within the dorsal fin.

The horizontal stabilizer is a swept back, full span unit, constructed around five spars. It is attached to the vertical stabilizer at two points:

- The center aft edge attaches to a heavy-duty hinge pin.
- The center leading edge attaches to an electrically operated screwjack to provide pitch axis trim.

AIRPLANE SYSTEMS

ELECTRICAL POWER SYSTEMS

Primary DC electrical power is provided by two engine-driven generators. Secondary power is supplied by two 24-volt batteries. The airplane may be equipped with a single or dual emergency battery system. The airplane also has the capability of accepting DC power from a ground power unit.

DC power is used by either two or three solid-state static inverters which, in turn, supply AC power for equipment and instruments.

LIGHTING

Interior lighting is supplied for general cockpit use and for instrument illumination. Cabin lighting is supplied for the cabin overhead lighting, individual passenger positions, and cabin baggage compartment.

Exterior lighting includes the combination landing-taxi light on each main gear, navigation lights, anticollision lights, strobe lights, and a recognition light. A second recognition light and wing ice inspection light may be available.

Figure 1-15. Empennage

FUEL SYSTEM

Fuel is contained in integral wing tanks, tip tanks, and in a bladder cell fuselage tank just aft of the rear pressure bulkhead. The 36 model has a larger fuselage tank than the 35 model.

Fueling is accomplished through filler caps in the top of each tip tank.

POWERPLANT

The Learjet 35/36 is powered by two Garrett TFE731 turbofan engines. The TFE731 is a lightweight, two-spool, front fan-jet engine. It has a reverse-flow annular combustion chamber which reduces the overall length and results in more efficient combustion and cooler external surfaces of the turbine section.

The low-pressure rotor consists of a four-stage, axial compressor and a three-stage, axial turbine rotating on a common shaft. The axial-flow fan assembly is located at the forward end of the engine and is gear-driven by the low-pressure rotor.

The high-pressure spool incorporates a single-stage, high-pressure centrifugal compressor and a single-stage axial turbine constructed as a single unit. The high-pressure spool drives the accessory section.

The high-pressure spool is located between the low-pressure compressor and the low-pressure rotor shaft passing through its center.

The engines are mounted on external pylons and are accessed by upper and lower nacelle covers. An access door on the outboard side of each nacelle is provided to check engine oil quantity.

Fire detectors are located in each engine nacelle and two engine fire extinguisher bottles in the tailcone.

Each engine supplies both high-pressure (HP) and low-pressure (LP) bleed air which is used either independently or in combination for anti-icing, pressurization, cabin temperature control, and the Aeronca thrust reversers, if installed.

ICE AND RAIN PROTECTION

The anti-icing systems use engine bleed air, electric heating, and alcohol.

Bleed air is used to heat the wing leading edge, the horizontal stabilizer leading edge, windshields, nacelle lips, and on some airplanes, the engine fan spinners. Bleed air is also used to remove rain from the windshield.

Electrically heated systems include pitot tubes, static ports, P_2T_2 sensors, and the stall warning vanes.

An alcohol system is used for radome anti-icing and to back up the pilot's windshield bleed-air anti-icing.

AIR CONDITIONING AND PRESSURIZATION

Regulated engine bleed air is diluted into the pressurized compartment through a heat exchanger where it is cooled by ram air from the dorsal inlet. Cabin temperature is regulated by controlling the amount of bleed air allowed to bypass the heat exchanger.

Pressurization is regulated by controlling the amount of air that is exhausted from the cabin. Control is maintained by a pressurization controller module and an outflow valve. The controller module provides fully automatic control of pressurization as well as manual mode. It ensures that the airplane is depressurized on the ground, and causes automatic pressurization to occur on takeoff. Built-in safeguards prevent over-under pressurization.

A Freon refrigeration system and an optional auxiliary cabin heater supplement the normal air conditioning system; they may be used when the engines are not operating, provided a ground power unit is connected. Both systems are completely independent of the bleed-air pressurization system.

HYDRAULIC POWER SYSTEMS

The hydraulic system supplies pressure for the operation of the landing gear, gear doors, brakes, flaps, spoilers, and Dee Howard thrust reversers, if installed. A single reservoir supplies fluid to the two engine-driven pumps through fire shutoff valves.

An electric auxiliary pump can pressurize all systems except the spoilers. It draws fluid from the same reservoir. The auxiliary supply line is not affected by the fire shutoff valves.

LANDING GEAR AND BRAKES

The Learjet 35/36 has a retractable tricycle landing gear which is electrically controlled and hydraulically operated.

An emergency air bottle, located in the right side of the nose compartment, can be used to extend the landing gear or for emergency braking, or both, in case of hydraulic or electrical failure.

The self-centering nose gear has a single wheel and incorporates an electrical nosewheel steering system which has variable authority, depending upon taxi speed.

Each main gear has dual wheels, each equipped with multiple-disc brakes. Hydraulic braking is controlled from either the pilot's or copilot's station. A fully modulated antiskid system provide maximum braking performance while protecting against skids.

FLIGHT CONTROLS

The Learjet 35/36 uses manually actuated primary flight controls. Pilot inputs are transmitted via cables, bellcranks, and pushrods to the ailerons, rudder, and elevators. There are no hydraulic or electric power boosts for these systems. Primary control trims are electrically controlled and operated.

Secondary flight controls (spoiler/spoileron and flaps) are electrically controlled and hydraulically operated.

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

The automatic flight control system (AFCS) includes a flight director, autopilot, and yaw dampers.

The flight director system generates roll and pitch commands by means of a single-cue V-bar display in the pilot's attitude director indicator. Programming and annunciation of selected modes is accomplished on the AFCS control panel in the center glareshield.

The two-axis autopilot provides control of the roll and pitch axes. When engaged, the autopilot responds to the flight director as programmed, or the pilot may elect to operate the autopilot in a basic attitude-hold mode by canceling all flight director modes, in which case the command bars are biased out of view.

Dual yaw dampers are installed for control of the yaw axis. Intended for full-time in-flight operation, either yaw damper must be engaged after takeoff. Functioning to dampen yaw and provide turn coordination, the yaw damper(s) operate independently, whether or not the autopilot is engaged.

PITOT-STATIC SYSTEM

The type of system used to supply pitot and static pressure to the pilot's and copilot's instruments depends on whether the FC 200 or FC 530 automatic flight control system (AFCS) is installed.

FC 200 models use a conventional pitot-static system consisting of one heated pitot tube mounted on each side of the nose section and two heated static ports flush-mounted on each side of the nose compartment. The air data sensor uses the copilot's pitot line for pitot-pressure, while its static pressure is provided by two additional heated static ports installed on the nose, forward of the windshield. An alternate unheated static port inside the nose compartment is provided for the pilot's static system.

FC 530 models use a Rosemount-designed pitot-static system which physically integrates two static ports into each of two pitot tubes, one mounted on each side of the nose section. The air data sensor uses the copilot's pitot and static lines.

An unheated static port is located on the right side of the nose compartment to provide a static source for the pressurization control module.

OXYGEN SYSTEM

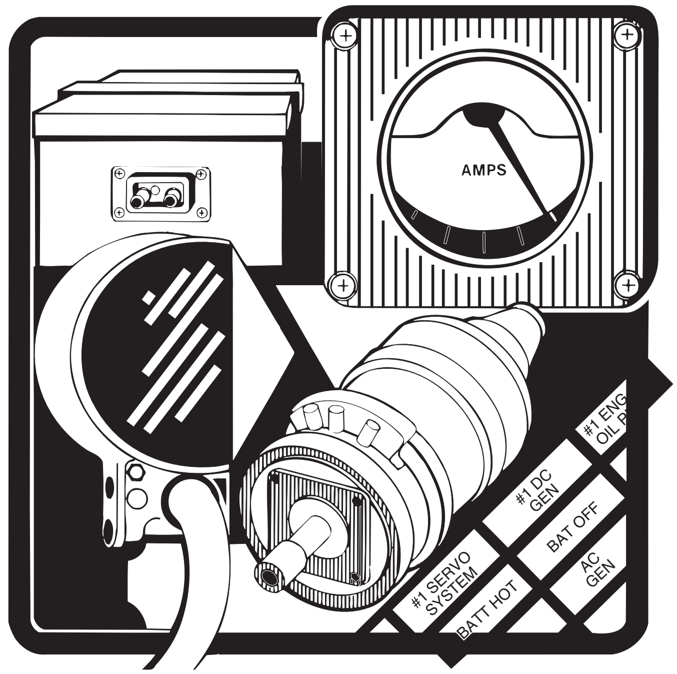
The oxygen system consists of the crew and passenger distribution systems connected to a high-pressure oxygen storage cylinder located in the nose compartment on early 35 and 36 models. On airplane SNs 35-492 and 36-051 and subsequent, the cylinder is located in the vertical stabilizer.

CHAPTER 2
ELECTRICAL POWER SYSTEMS

CONTENTS

ILLUSTRATIONS

CHAPTER 2
ELECTRICAL POWER SYSTEMS



INTRODUCTION

Primary DC electrical power is provided by two engine-driven brushless DC generators rated at 30 volts, 400 amperes each. A single generator is capable of sustaining normal DC load. Secondary DC electrical power is supplied by two batteries. In the event of a double generator failure, the airplane batteries will provide power for a limited period of time.

A ground power unit can also provide the DC electrical power needed for system operation or engine starting. Electrical power for AC-powered equipment is provided by two (or an optional third) solid-state static inverters located in the tailcone. The inverters require DC input power for operation. An emergency battery is provided in case of total airplane electrical failure to operate a standby attitude gyro and the landing gear and flaps. A second emergency battery may be installed at the customer’s option to power additional equipment such as an emergency communication radio, transponder, or emergency directional gyro.

GENERAL

The electrical system incorporates a multiple bus system for power distribution interconnected by relays, current limiters, overload sensors, and circuit breakers, which react automatically to isolate a malfunctioning bus. Manual isolation is also possible by opening the appropriate circuit breakers.

The batteries are capable of operating the minimum equipment for night instrument flight for approximately 30 minutes in the event both generators become inoperative. An emergency battery is provided to operate an emergency attitude gyro, and gear and flap systems in the event of total airplane electrical system failure.

It is possible to power the entire DC and AC electrical systems from the airplane batteries, an engine-driven generator, or GPU.

Figure 2-1 shows the major electrical power system component locations.

DC POWER

BATTERIES

Two batteries located in the tailcone (Figure 2-2) provide the secondary source of DC power.

Each battery has a removable cover and a case which is vented and cooled by overboard connections. The batteries are of sufficient capacity to supply normal ground electrical requirements and may be used for engine starting when external power is not available.

Lead-Acid

Lead-acid batteries are enclosed in a plastic case. Nickel-cadmium (nicad) batteries are enclosed in a stainless steel case. On airplane SNs 35-341 and 36-050 and subsequent equipped with lead-acid batteries, a sump jar has been added to contain any electrolyte spillover. A sponge saturated in a baking soda and water solution neutralizes the acid. AMK 81-5A makes this installation available in earlier airplanes.

Figure 2-2. Battery Location

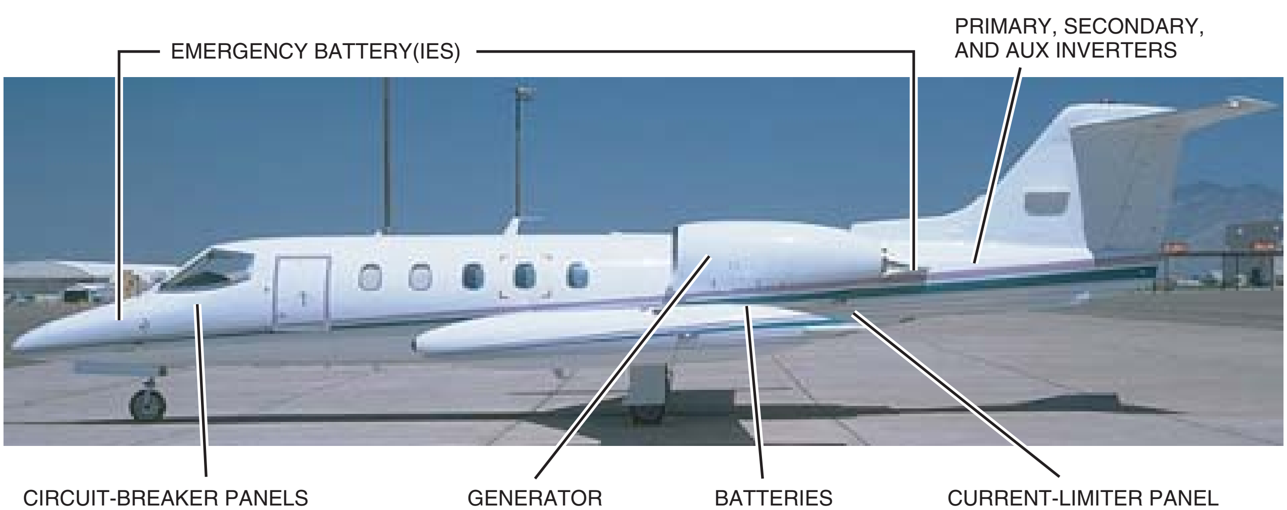


Figure 2-1. Component Locations

Each battery is connected to its respective battery bus through a 20-amp current limiter for hot-wired circuits.

Charging nicad batteries with a GPU is not recommended.

Charging lead-acid batteries in the airplane is not recommended because of poor GPU output regulation.

Controls

Two battery switches (Figure 2-3) are provided which connect the batteries in parallel to the battery-charging bus when the switches are on. The switches are labeled “BAT 1” and “BAT 2,” corresponding to the respective battery. Each switch is a two-position, ON–OFF switch which completes a ground circuit to close its respective battery relay in the ON position. (see Figure 2-13.)

The battery relays require approximately 16 volts (minimum) from the respective battery. If either battery voltage is less than 16 volts, the respective battery relay will not close, in which case that battery cannot be connected to the airplane electrical system for the purpose of operating electrical equipment (except equipment hot-wired to its battery bus), nor can it be charged by a GPU or the generators. The airplane batteries are always connected in parallel (including during engine starts) when both batteries switches are on.

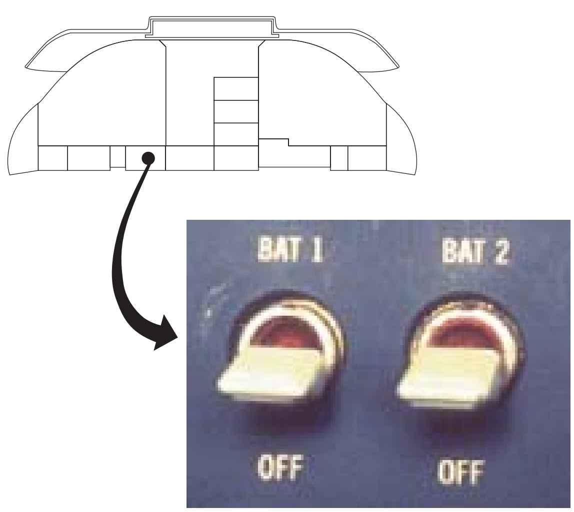


Figure 2-3. Battery Switches

Indicators

Electrical system indicators are grouped in a cluster on the upper portion of the center instrument panel. A single DC VOLTS meter, connected to the battery charging bus through a 5-amp current limiter, indicates the highest voltage input to the bus by batteries, generators, or a GPU. To read individual battery voltage, only one battery at a time may be connected to the battery-charging bus with the generators off and a GPU not connected. Airplane generators and GPUs normally put out a higher voltage than the batteries; therefore, when either of these is powering the battery-charging bus, generator or GPU voltage will be indicated (Figure 2-7).

Airplanes with nickel-cadmium batteries are equipped with battery temperature indicators and overheat warning light systems. The temperature indicators and warning lights are attached through two electrical connectors on the face of each battery case to temperature sensors and thermal switches on each battery.

A dual-indicating temperature gage is installed on the lower portion of the copilot’s instrument panel (Figure 2-4). Two red warning lights labeled “BAT 140” and “BAT 160” are installed in the annunciator panel and illuminate if either or both batteries reach 140 to 160° F, respectively.

GENERATORS

Two engine-driven DC generators, one on each engine (Figure 2-5), provide the primary source of DC power. Each brushless generator is rated at 30 VDC, 400 amperes.Cooling

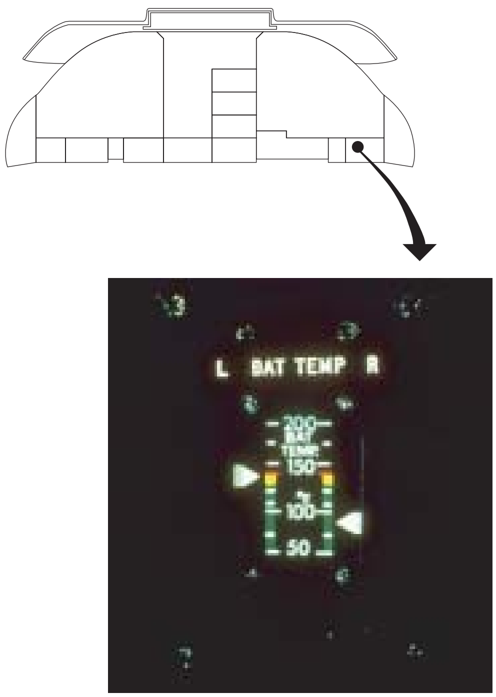


Figure 2-4. Battery Temperature Indicator

air is routed from a scoop on the engine nacelle to the associated generator. During normal operation, both generators operate in parallel through the solid-state voltage regulators located in the tailcone. As long as both battery switches are on, either generator charges both batteries through the associated 275-amp current limiter. The generators supply DC power to all DC-powered equipment on the airplane.

Generator voltage is regulated to 28.8 VDC for lead-acid batteries and to 28.5 VDC for nicad batteries. On airplanes SNs 35-148 and subsequent and 36-036 and subsequent, single-generator voltage is reduced as load increases during ground operation and any time a starter is engaged to limit amperage. This design feature protects the 275-amp current limiters during engine start. The generator control panel, located in the tailcone, contains relays for the batteries, starters, GPU over-voltage control, and an equalizer circuit for load sharing.

Controls

Two starter-generator switches are located on the center switch panel. They are three-position switches labeled "GEN," "OFF," and "START" (Figure 2-6). In the GEN position, current is provided to the generator field through the IGN & START circuit breaker, which automatically connects the generator bus, and the amber GEN caution light extinguishes.

Two generator reset buttons labeled "L GEN RESET" and "R GEN RESET" on the center switch panel (Figure 2-6) provide for resetting the generator in case of failure. Provided the GEN-OFF-START switch is in the GEN position, momentarily depressing the reset button

Figure 2-5. Generator Location

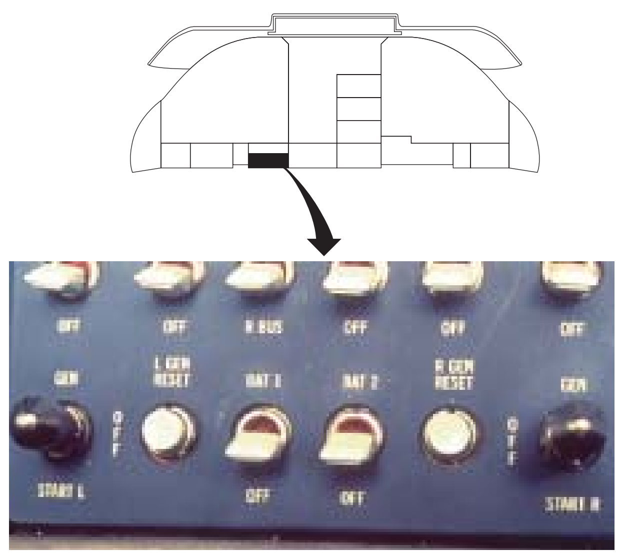


Figure 2-6. Generator Switches

resets the overvoltage relay, completes a power circuit to the voltage regulator, and restores the generator to normal operation.

Indicators

Two AMPS meters (one for each generator) indicate the load, in amperes, being carried by each generator (Figure 2-7). The load indication is measured at the voltage regulator.

Generator voltage is displayed on the DC VOLTS meter.

An amber L or R GEN caution light on the glareshield annunciator panel (Annunciator Panel Section) illuminates if the associated generator switch is turned off, if the generator fails, or if the generator is tripped off by the overvoltage cutout relay.

GROUND POWER

A ground power unit (GPU) can be connected to the airplane through the receptacle located on the left side of the fuselage, below the left engine (Figure 2-8). The receptacle connects GPU power to the battery charging bus through a power relay controlled by an overvoltage circuit. The overvoltage circuit samples GPU voltage provided through a control relay (Figure 2-9). At least one battery switch must be turned on to close the control relay, allowing the overvoltage circuit to sample GPU voltage, and, if below 33 volts, the power relay closes to complete the GPU-to-battery charging bus connection. The GPU should be regulated to 28 volts. Due to tower shaft torque limits, it must be limited to 1,100 amps for engine starts. It should be capable of producing at least 500 amps. If GPU voltage exceeds 33 volts, the overvoltage circuit causes the power relay to open, thereby disconnecting the GPU from the electrical system to prevent damage to voltage-sensitive equipment.

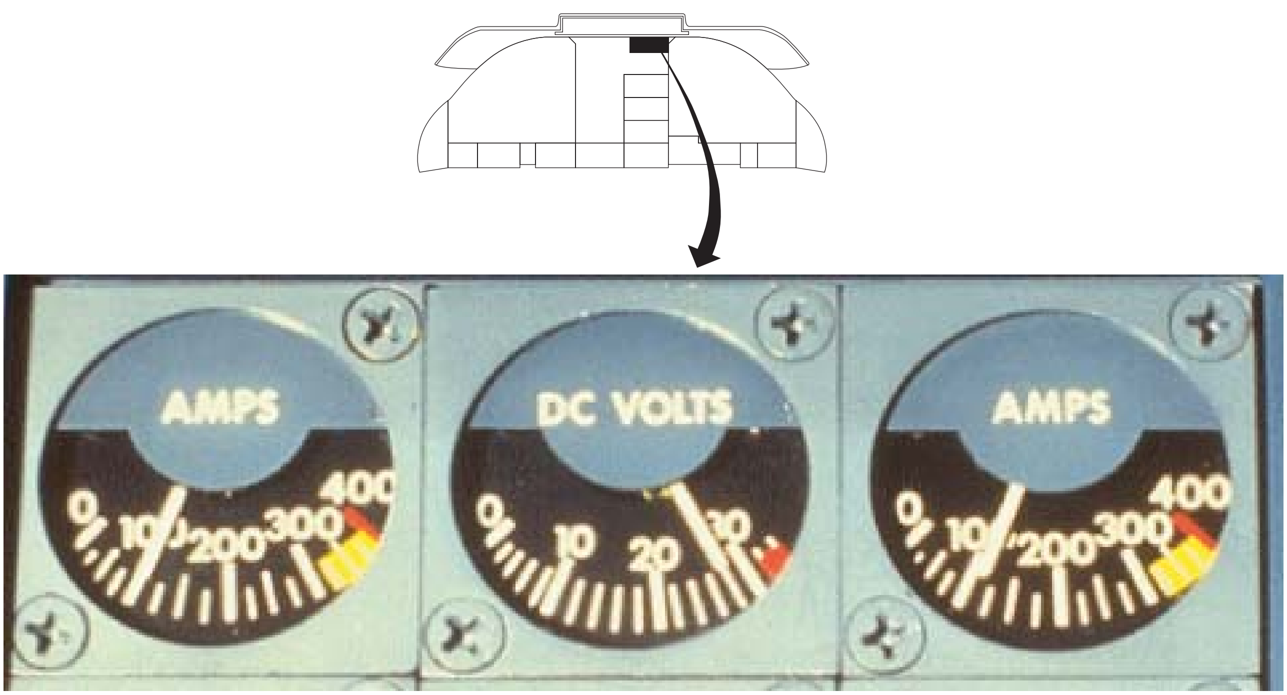


Figure 2-7. Generator Indicators

CIRCUIT COMPONENTS

Current Limiters

Throughout the electrical distribution system, various-size current limiters are placed at strategic locations to prevent progressive total electrical failure. A current limiter is similar to a slow-blow fuse in that it will carry more than its amp-rated capacity for short periods of time. Extreme or prolonged overloading will cause a current limiter to fail, thus isolating that particular circuit and precluding progressive failure of other electrical components. Current limiters are not resettable. When a current limiter has blown, it must be replaced. It should be replaced if it shows discoloration or other signs of heating or overloading. The current-limiter panel is located in the tailcone (Figure 2-10).

There are two current limiters (one on each generator) that are not located on the current-limiter panels in the tailcone. A 10-amp current limiter on each generator is part of the paralleling circuit.

Two types of current limiters are used in the system. The lower amperage current limiters (50 amps or less) are red and have a pin that protrudes if it is blown. The higher amperage current limiters are made of a gray ceramic material with a small window that allows visual inspection of current-limiter integrity.

There are two 275-amp current limiters in the main current-limiter panel which connect the generator buses with the battery-charging bus. On airplane SNs 35-002 through 35-147 and 36-002 through 36-035, testing of these current limiters is accomplished manually. On airplanes SNs 35-148 through 35-389, except 35-370, and 36-036 through 36-047, testing of the current limiters is accomplished using the rotary systems test switch. For all of the above airplanes, AMK 80-17 provides two amber lights, one for each current limiter, which allows continuous monitoring. On airplane SNs 35-370, 35-390, and 36-048 and sub-sequent, a single red CUR LIM light on the glareshield annunciator panel allows continuous monitoring of the 275-amp current limiters.

Figure 2-8. Ground Power Connector

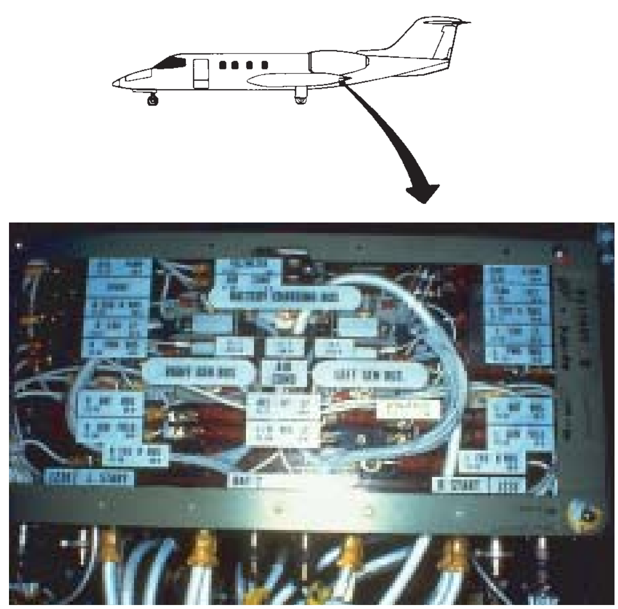


Figure 2-10. Current Limiter Panel

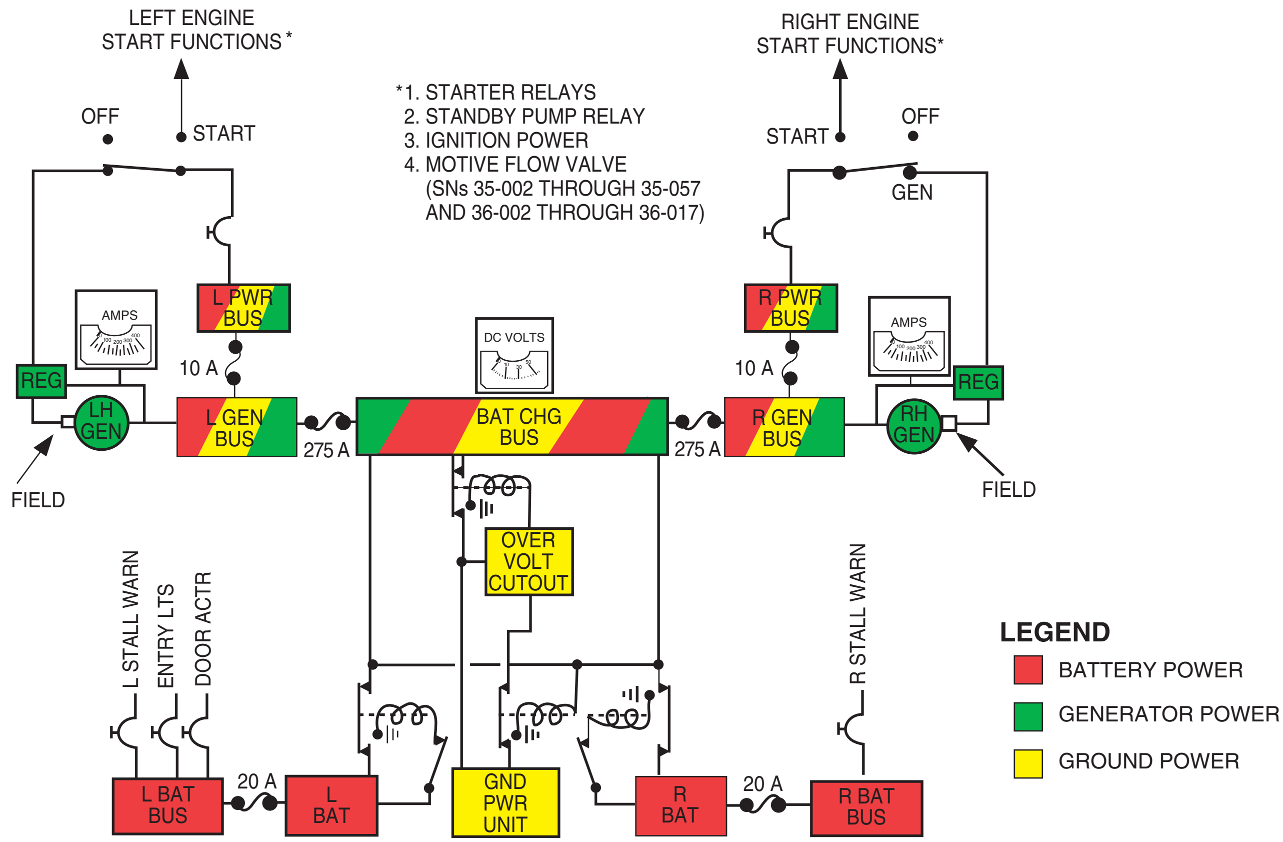


Figure 2-9. Basic DC Distribution

The 275-amp current-limiter annunciator light(s) are illuminated by 1-amp overload sensors wired across the current-limiter terminals. Failure of a current limiter results in a surge of current through the overload sensor causing it to trip, thereby illuminating the light. In flight, it is important to know if the current limiters have blown. On all airplanes with or without the current limiter annunciator light(s), current-limiter status may be determined by close observation of voltmeter and ammeter indications. If only one has failed, no difference will be noted on either indicator since power from each generator still flows to the battery charging bus through the opposite current limiter. Failure of both current limiters, however, could be recognized since the voltmeter will read battery voltage (less than 25 volts). On airplanes prior to SNs 35-509 and 36-054 not modified by AMK 85-1, this failure will eventually result in the depletion of the batteries since they are the only source of power to the essential buses. The generators have been separated from the load of the essential buses and are now supplying power to only the main buses and the generator buses. This greatly reduced loading will be reflected by abnormally low ammeter readings on both generators.

On airplane SNs 35-509 and 36-054 and subsequent or earlier airplanes with AMK 85-1 installed, a failure of both 275 amp current limiters will not result in the separation of the generators from the essential buses. Generator loads, therefore, will remain relatively normal. The generators have, however, been separated from the battery buses and battery charging bus. Consequently, the batteries are no longer being charged and will be slowly depleted by electrical equipment which draws power from either battery bus or the battery charging bus. Battery condition should be monitored using the DC voltmeter. On airplanes with the single CUR LIM annunciator light, if one limiter blows in flight, DC volts and amps should be monitored closely since the CUR LIM light remains illuminated and will not alert the pilot to subsequent failure of the other limiter.

Relays

Relays are used at numerous places throughout the electrical distribution system, particularly in circuits with heavy electrical loads. The relays function as remote switches to make or break power circuits. This arrangement allows the control circuit wiring to be a lighter gage since less current is required to operate the relay. Relays control the power circuits for the batteries, GPU, starters, generators, inverters, and left and right main buses. Instrument panel switches or circuit breakers complete the control circuits to operate the relays.

Overload Sensor

Overload sensors are used in the power circuits to the left and right main buses and in the power circuits to each inverter. These overload sensors react thermally to electrical loads in excess of their design capacity. In reacting, they electrically ground the relay control circuit causing the associated control circuit to trip, which causes the relay to open and break the power circuit. Once the overload condition has been removed, the overload sensor cools and resets automatically; however, the control circuit breaker must be reset manually. The overload sensors in the main bus power circuits are rated at 70 amps, and the overload sensors for the inverter power circuits are rated at 60 amps.

Circuit Breakers

Circuit breakers are located on two circuit-breaker panels in the cockpit, one left of the pilot's seat and one right of the copilot's seat. On FC 200 AFCS airplanes, three additional circuit breakers, located under the pilot's seat on the autopilot electric box, provide power for the autopilot flight director, and yaw damper annunciator lights. The DC circuit breakers are the thermal type, and the AC circuit breakers are the magnetic type. Amperage ratings are stamped on the top of each circuit breaker. The circuit breakers are arranged in rows according to the buses which serve them to simplify the isolation of individual buses or circuits. Basically, all circuit breakers in the top row (both sides) are on the 115-VAC and 26-VAC buses; in the second row they are on the main DC buses (except three which are power bus circuit breakers). Additionally, thrust reversers (if installed) are controlled by main bus circuit breakers which are physically installed on the left and right panels, third and fourth rows. The third and fourth rows on airplane SNs 35-002 through 35-201 and 35-205, and 36-002 through 36-040 are on the DC essential bus. On airplane SNs 35-202 and subsequent, except 35-205 and 36-041 and subsequent, and earlier airplanes incorporating AMK 78-13, the third and fourth rows are on the essential A and B DC buses and subsequent, and earlier airplanes incorporating AMK 78-13, the third and fourth rows are on the essential A and B DC buses.

Circuit breakers located on the third and fourth rows, but not powered by the essential buses, are:

- L STALL WARN, DOOR ACTR and ENTRY LTS (left battery bus items)
- R STALL WRN (right battery bus item)
- T/R EMER STOW and T/R POS IND (left main bus item—Aeronca)

- T/R CONT (right main bus item—Aeronca)
- T/R POWER and T/R CONT (left and right main bus items—Dee Howard)

See Figures 2-11 and 2-12 for typical representations of the circuit-breaker panels.

DISTRIBUTION

The airplane uses a multiple bus, multiple conductor, electrical distribution system. Buses and major circuits are protected by relays, current limiters, overload sensors, and circuit breakers to preclude total failure. This arrangement also allows isolation of malfunctioning buses. All circuit breakers are accessible to the crew during flight.

Battery Buses

The left and right battery buses are connected to the left and right batteries, respectively, through 20-amp current limiters (Figure 2-9). The battery buses are always "hot," provided the battery quick-disconnects are connected. The battery buses supply power to the following items:

- Left battery bus
  - Left stall warning system
  - Entry lights (step lights, baggage compartment lights, and tailcone inspection light)
  - Door actuator motor
- Right battery bus
  - Right stall warning system

Battery bus items must be turned off before leaving the airplane to prevent battery discharge.

Battery-Charging Bus

The battery-charging bus enables the generators or GPU to charge the batteries and is the central distribution point for the DC electrical system. It is powered by the batteries and GPU through their associated power relays by either generator through the respective left and right 275-amp current limiters (Figure 2-9).

One or both batteries can power the entire electrical system for a limited period of time, with the exception of the Freon air conditioner and auxiliary heater. Because their high amperage requirement would quickly deplete the batteries, these items are isolated by an open relay which will not close until a GPU or generator is on and operating.

On airplanes SNs 35-002 through 35-508 and 36-002 through 36-053, when not incorporating AMK 85-1, the essential buses are connected directly to the battery charging bus (Figure 2-15 and 2-16).

On all airplanes, the following equipment is directly connected to the battery charging bus (Figure 2-13):

- DC VOLTS meter
- Freon air conditioner and auxiliary heater
- Recognition light(s)
- Auxiliary hydraulic pump
- Fuel flow indicating system
- Auxiliary inverter (if installed)
- Utility light (if installed)
- Primary pitch trim motor (FC-530 AFCS only)
- Left and right engine starters

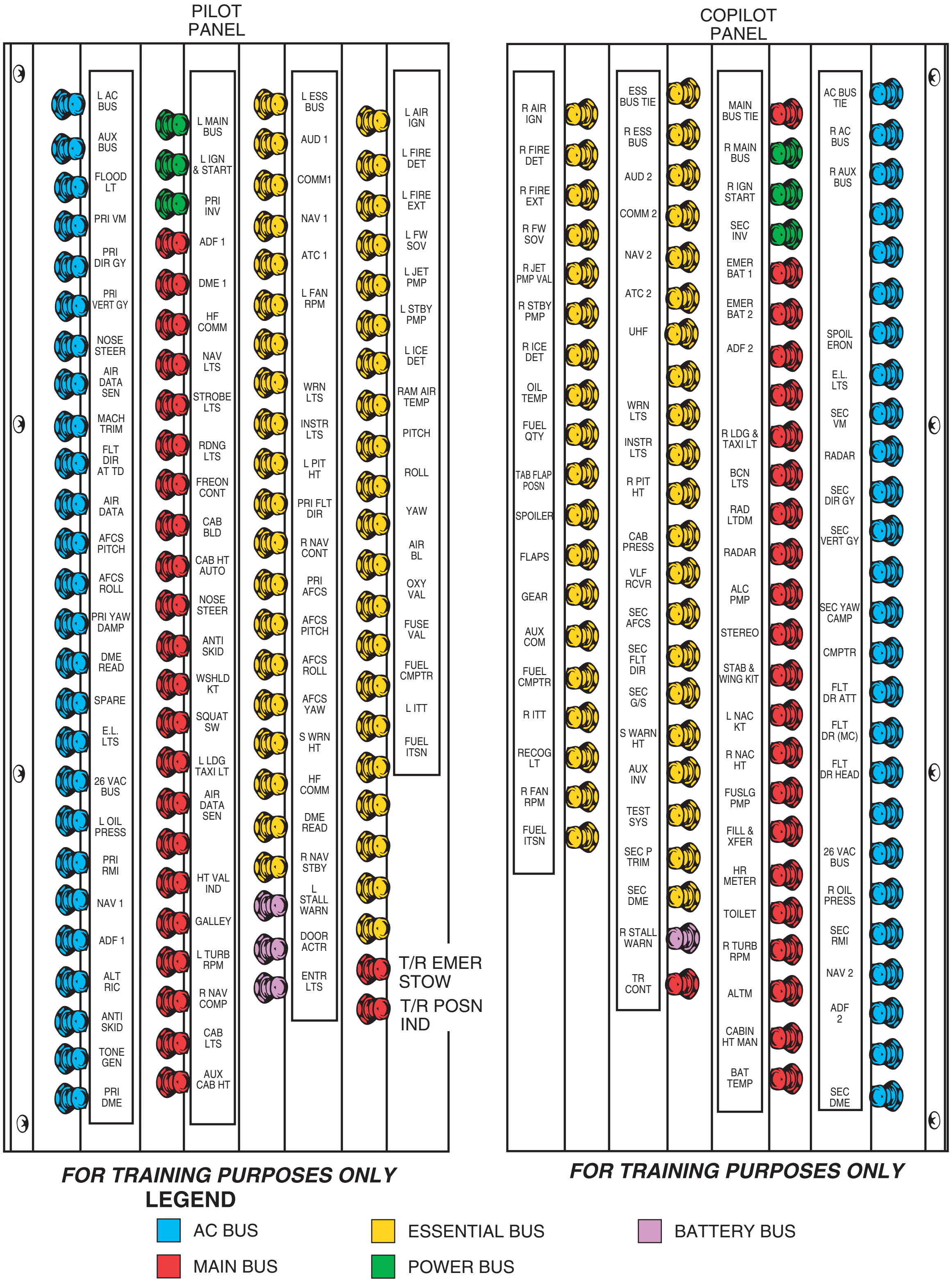


Figure 2-11. Typical Circuit-Breaker Panels—SNs 35-002 through 35-201 and 35-205, and 36-002 through 36-040 (Not Incorporating AMK 78-13)

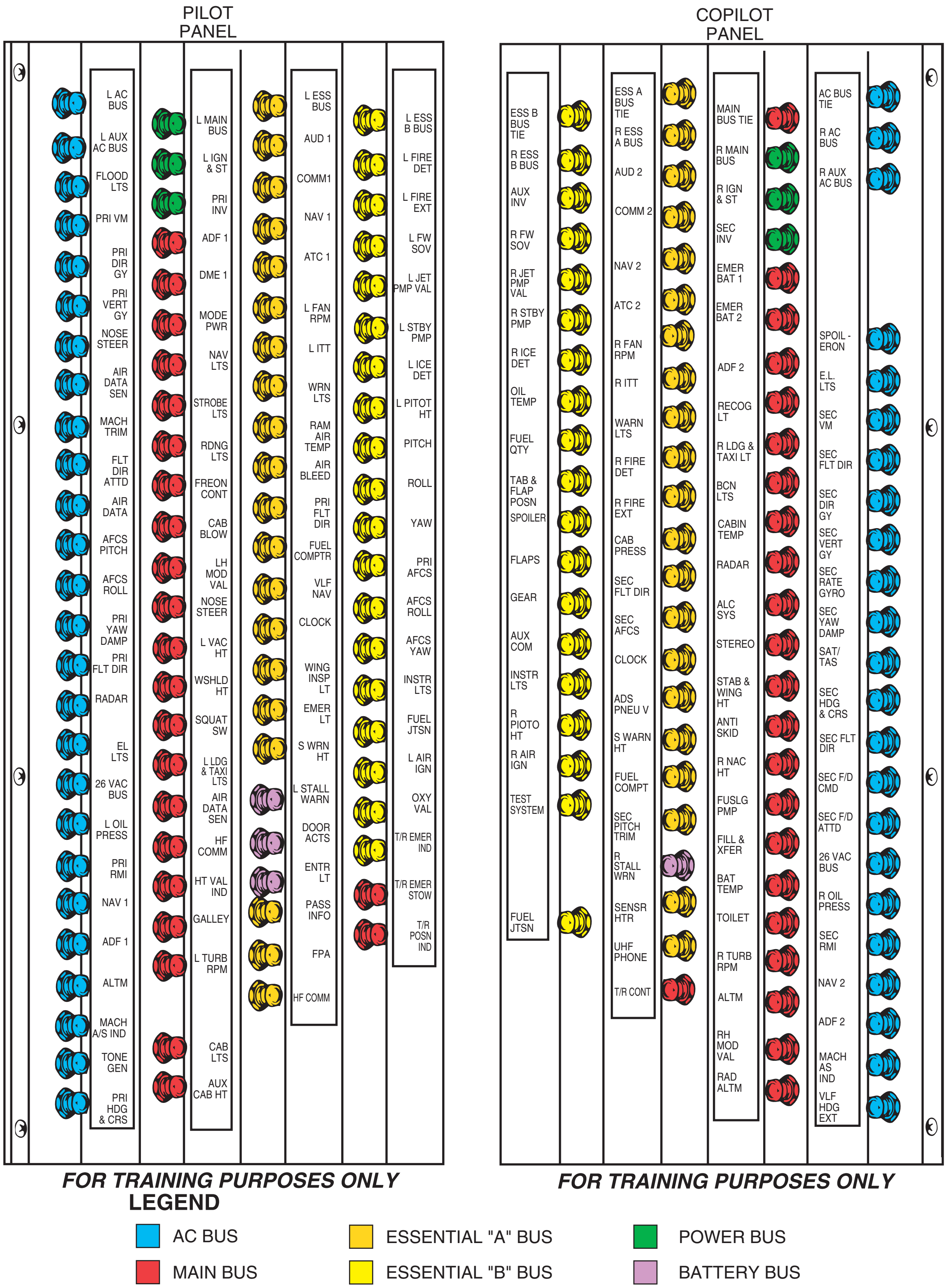


Figure 2-12. Typical Circuit-Breaker Panels—SNs 35-202 and Subsequent, except 35-205, 36-041 and Subsequent, and Airplanes Incorporating AMK 78-13

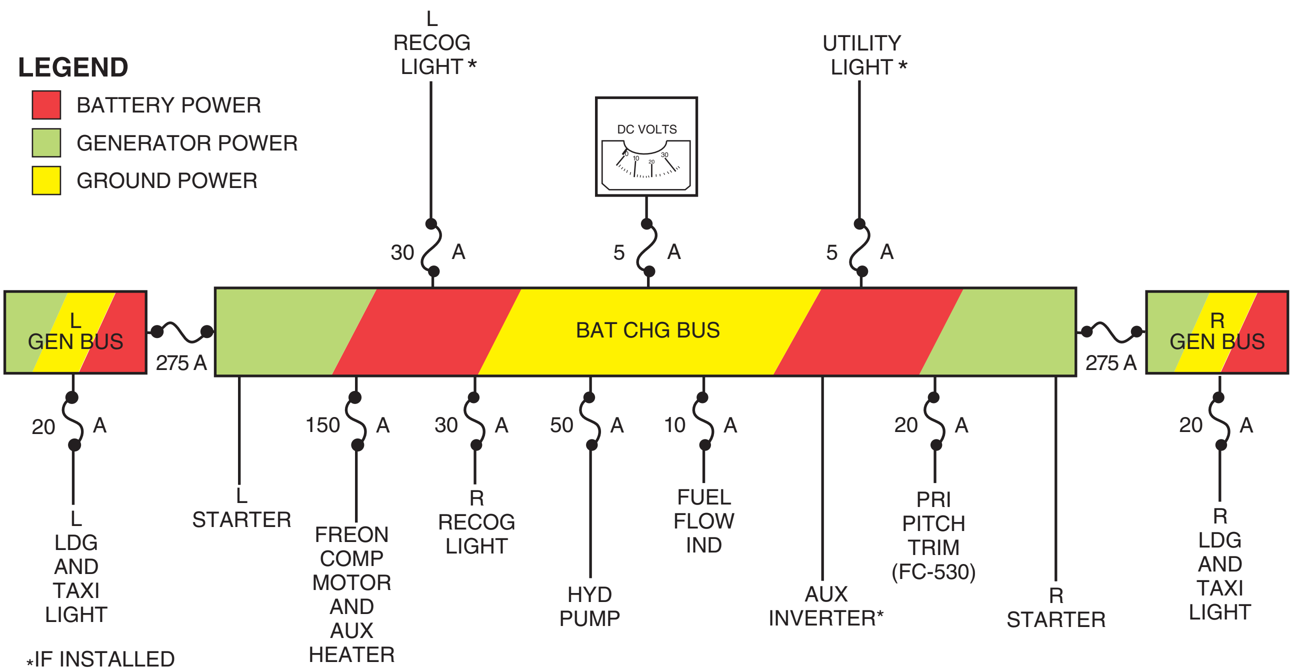


Figure 2-13. Equipment Powered by Battery Charging Bus and Generator Buses

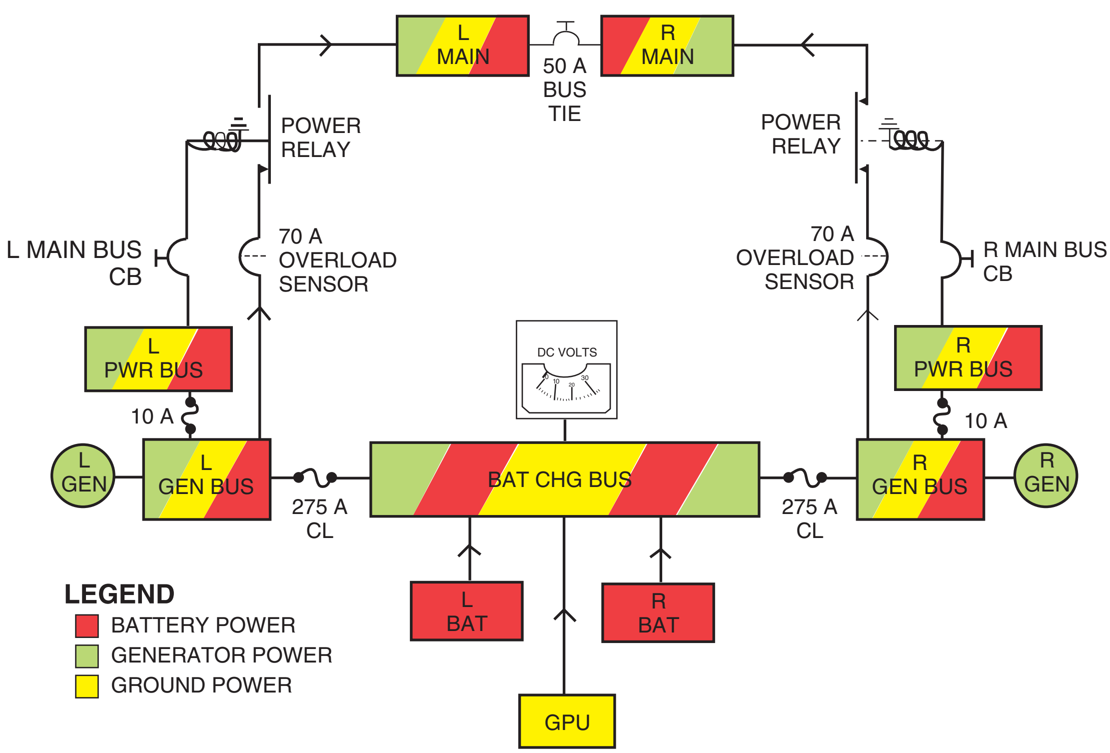


Figure 2-14. Main DC Bus Power

Generator Buses

The left and right generator buses distribute power to the right and left main buses, the primary and secondary inverters, and the left and right power buses (Figures 2-14 and 2-21). On airplanes SNs 35-509 and subsequent, 36-054 and subsequent, and prior airplanes incorporating AMK 85-1, the generator buses also power the respective essential A and B buses (Figure 2-17). On all airplanes, the landing/taxi lights are connected to the respective generator bus (Figure 2-13). The generator buses can be powered by the batteries, a GPU, or either generator.

Power Buses

The left and right power buses are powered from the respective generator bus through a 10-amp current limiter. Each power bus provides power to three circuit breakers which control the respective engine starting and generator functions, main bus power, and inverter power, as follows:

- The L or R MAIN BUS circuit breaker controls the respective main bus power relay which connects the respective generator bus to the main bus anytime DC power is available (Figure 2-14).

- The L or R IGN & START circuit breaker: (1) controls the respective starter relays and standby fuel pump relay and provides starting ignition power (through the thrust lever idle switch) when the GEN-OFF-START switch is in the START position; (2) provides power to the generator field when the switch is in the GEN position (Figure 2-13).

- The PRI or SEC INV circuit breaker controls the respective inverter power relay which connects the respective generator bus to the inverter when the inverter switch is turned on (Figure 2-21).

The power bus circuit breakers are located at the forward end of the respective circuit-breaker panels on what is generally referred to as the “main bus row,” those labeled “L” and “R IGN & START” and “PRI” and “SEC INV” are in no way related to, or affected by, the main buses; however, the L and R MAIN BUS circuit breakers are, in that they control the relays which power the main buses (Figure 2-14).

Main DC Buses

The left and right main buses are powered from the respective left and right generator buses through a 70-amp overload sensor and a power relay. The power relay is energized closed whenever there is power on the respective power bus and the associated MAIN BUS circuit breaker is closed. The left and right main buses are connected to each other by a 50-amp MAIN BUS TIE circuit breaker which is normally closed for load equalization (Figure 2-14).

In the event of an overload on either main bus, the respective overload sensor causes the affected MAIN BUS circuit breaker to trip. This deenergizes the power relay which opens to break the power circuits; then the MAIN BUS TIE circuit breaker opens when it is forced to accept the overload and cannot, resulting in automatic isolation of the faulty bus.

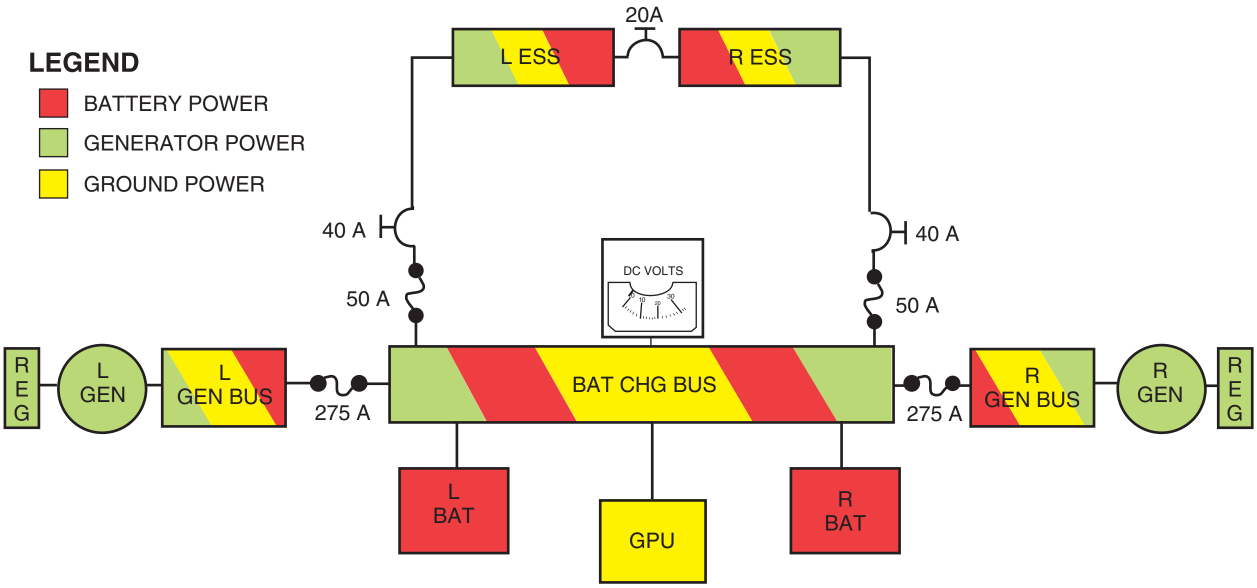


Figure 2-15. Essential DC Bus Power—SNs 35-002 through 35-201 and 35-205, and 36-002 through 36-040 (Not Incorporating AMK 78-13)

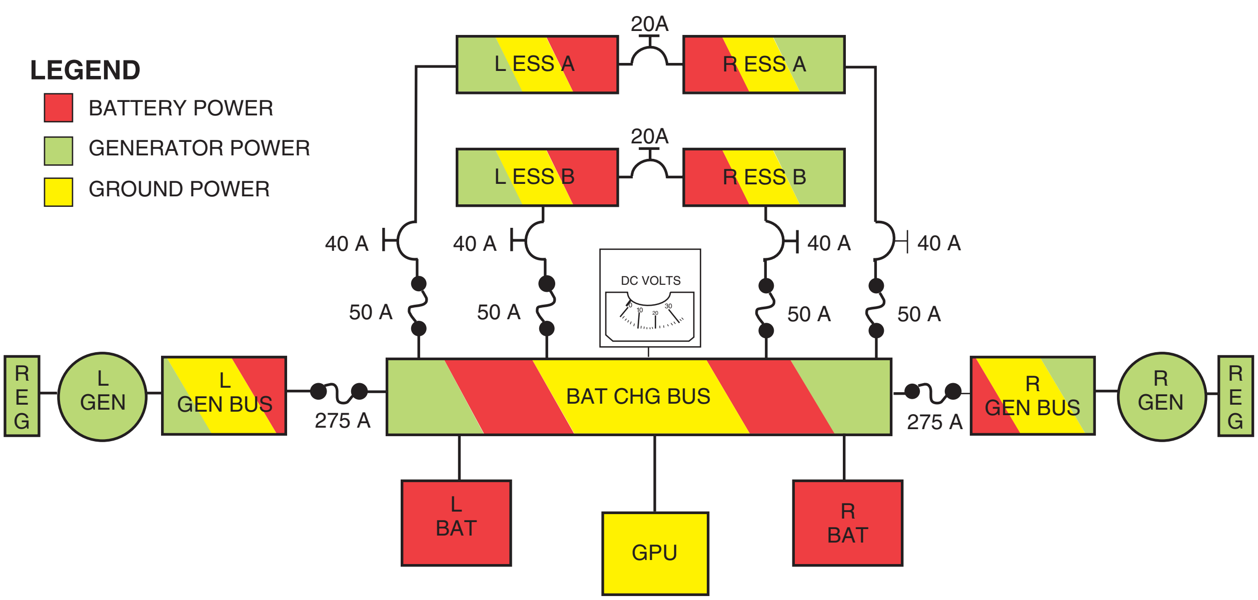


Figure 2-16. Essential DC Bus Power—SNs 35-202 through 35-508, except 35-205, 36-041 through 36-053, and Prior Airplanes Incorporating AMK 78-13

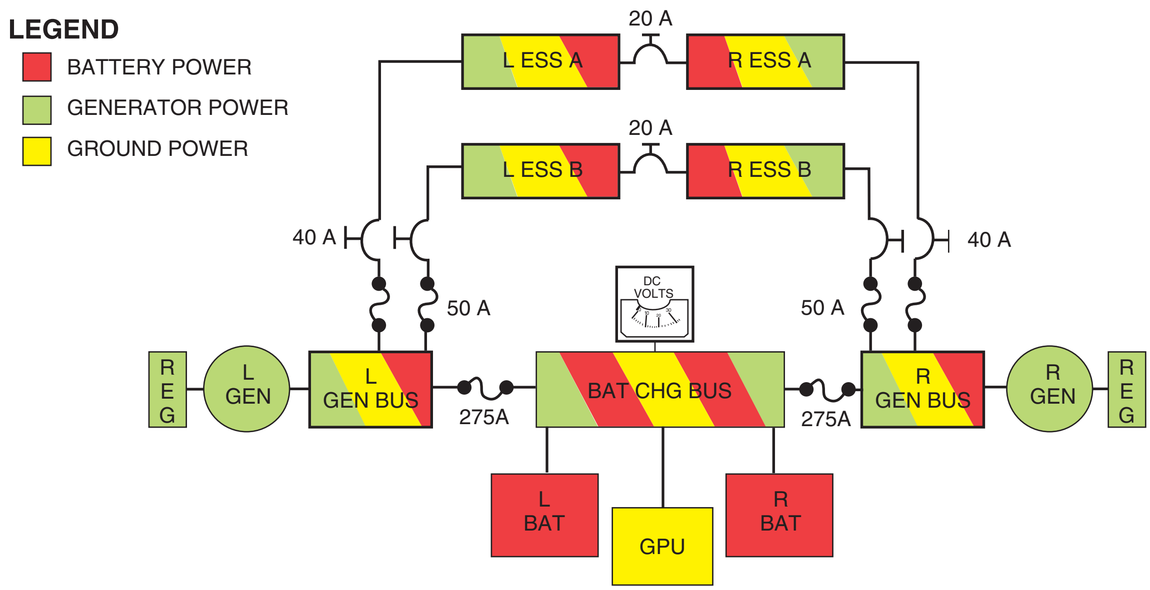


Figure 2-17. Essential DC Bus Power—SNs 35-509 and Subsequent and 36-054 and Subsequent, and Prior Airplanes Incorporating AMK 85-1

Essential DC Buses

One of three different bus configurations will apply to a given airplane, depending on production serial number and AMK applicability (Figures 2-15, 2-16, and 2-17).

The left and right essential buses are powered from the battery charging bus, or from the respective left or right generator buses (as applicable) through a 50-amp current limiter and a 40-amp ESS BUS circuit breaker, and are connected to each other by a 20-amp ESS BUS TIE circuit breaker which is normally closed for load equalization.

In the event of an overload on one of the essential buses, the respective ESS BUS circuit breaker opens, followed by the ESS BUS TIE circuit breaker which is forced to accept the overload and cannot, resulting in automatic isolation of the faulty bus. The current limiters provide backup for their respective ESS BUS circuit breakers.

AC POWER

INVERTERS

Alternating current to the AC electrical instruments and electronic equipment is provided by two or three 1,000-VA, solid-state static inverters located in the tailcone (Figure 2-18). The third (auxiliary) inverter is optional. During normal operation both, or all three, inverters are on and operate in parallel. It is recommended that the auxiliary inverter, if installed be operated in conjunction with the primary and secondary inverters to extend inverter life.

Figure 2-18. Inverter

The primary and secondary inverters are powered from the respective left and right generator buses through a 60-amp overload sensor and a power relay. The power relay is energized closed whenever there is power on the respective power bus, the associated PRI or SEC INV circuit breaker is closed, and the inverter switch is on (Figure 2-21).

In the event an inverter becomes overloaded, i.e., a shorted inverter, the respective overload sensor causes the affected PRI or SEC INV circuit breaker to trip. This energizes the power relay which opens to break the power circuit, resulting in automatic isolation of the faulty inverter. If installed, the auxiliary inverter circuits differ only in that they are powered from the battery charging bus, and the power relay is controlled by the AUX INV circuit breaker on the right essential bus (Figure 2-21).

Inverter output is 115-volt, 400-Hz, single-phase, alternating current. Some of the instruments and avionics require 26-VAC power. This 26-volt power is furnished by two step-down transformers located in the cockpit just aft of the circuit-breaker panels. These two transformers take 115-VAC input from the respective 115-VAC buses and step it down to 26-VAC output. Other components in the system include power relays, a paralleling box, overload sensors, circuit breakers, and inverter lights on the glareshield for primary, secondary, and auxiliary inverters.

The paralleling box is the central control unit for the AC electrical system. It incorporates load equalizer and frequency synchronizer/phaser circuits through which it maintains inverter load balance and frequency/phase synchronization. It also causes illumination of the associated annunciator lights for certain malfunctions.

CONTROLS

Two (or three) inverter switches, one for each inverter (PRI, SEC, and optional AUX) are installed on the pilot's lower instrument panel. The primary and secondary inverter switches have two positions, respectively, labeled "PRI–OFF" and "SEC–OFF." The auxiliary inverter switch, if installed, is labeled "ON––OFF" (Figure 2-19). If the optional auxiliary inverter is installed, an additional switch labeled "L BUS–R BUS" is also installed. This switch is used to direct the auxiliary inverter output to either the left or right AC bus as needed. In case of an inverter failure, the auxiliary inverter will not automatically assume the operation of the failed inverter unless the auxiliary inverter is turned on and the L/R BUS switch is properly positioned.

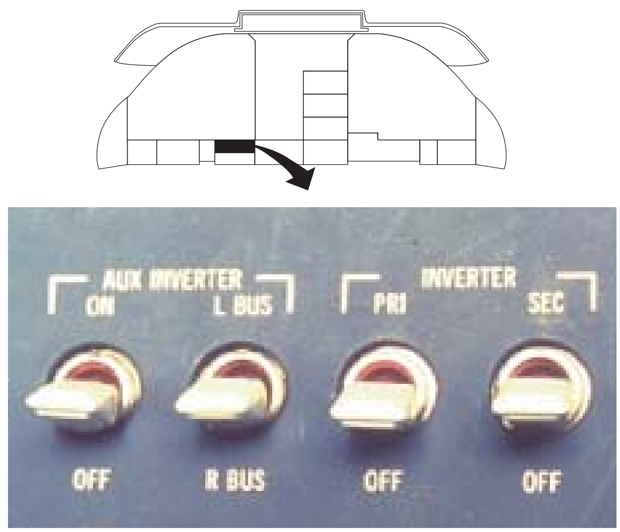


Figure 2-19. Inverter Switches

INDICATORS

Two red inverter warning lights labeled "PRI INV" and "SEC INV" are installed on glareshield. If the optional auxiliary inverter is installed, there is an amber AUX INV light on the glareshield (Annunciator Panel section).

The corresponding inverter annunciator light illuminates when inverter output is below 90 volts or if bus load is less than 10 volt-amps.

The primary and secondary inverter lights also illuminate when the respective inverter switch is turned off. The AUX INV light, however, illuminates only for an auxiliary inverter failure with the switch turned on.

A single AC voltmeter (Figure 2-20) indicates the voltage on the LH or RH AC bus, depending on the position of the AC BUS switch. This two-position switch labeled "PRI–SEC" selects which bus the AC voltage is being measured from. To check individual inverter voltage, only the inverter to be checked should be turned on.

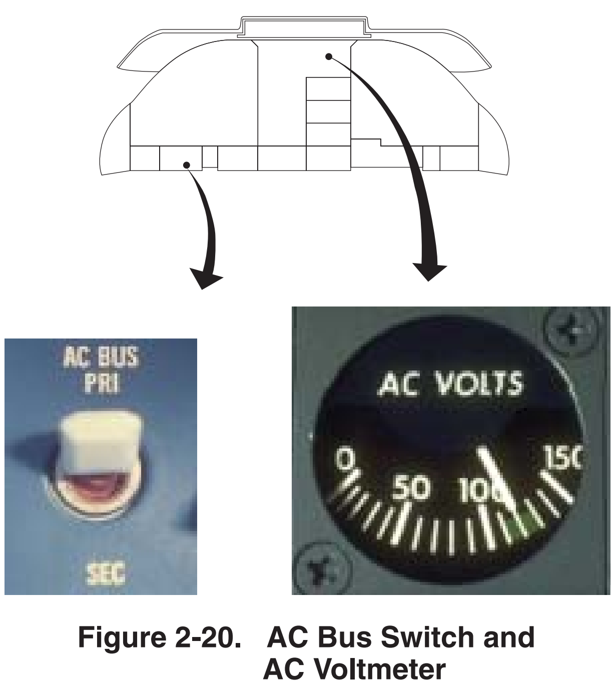


Figure 2-20. AC Bus Switch and AC Voltmeter

DISTRIBUTION

115-Volt AC Buses (L and R)

Alternating current from the inverters is distributed through the paralleling box to the respective left and right AC buses (Figure 2-21). Primary inverter output goes to the left bus; secondary to the right bus. Auxiliary inverter output (if installed) may be selected to either the left or the right bus.

All circuit breakers on the left 115-VAC bus are located on the top row of the left circuit-breaker panel. The right 115-VAC bus circuit breakers are on the top row of the right circuit-breaker panel. The first circuit breaker on the top row of the right panel is the 7 1/2-amp AC bus-tie circuit breaker. The second circuit breaker on the top row of the right panel and the first circuit breaker on the top row of the left panel are the L and R AC BUS 10-amp bus feeder circuit breakers.

26-Volt AC Buses (L and R)

Two step-down transformers draw 115-VAC power from the left and right 115-VAC buses, reduce the voltage output to 26 VAC, and connect to the 26-VAC buses for equipment requiring 26-VAC power.

The 26-VAC BUS breakers are approximately two-thirds of the way aft on the top row of each panel. All circuit breakers aft of the respective 26-VAC BUS breakers power 26-VAC equipment.

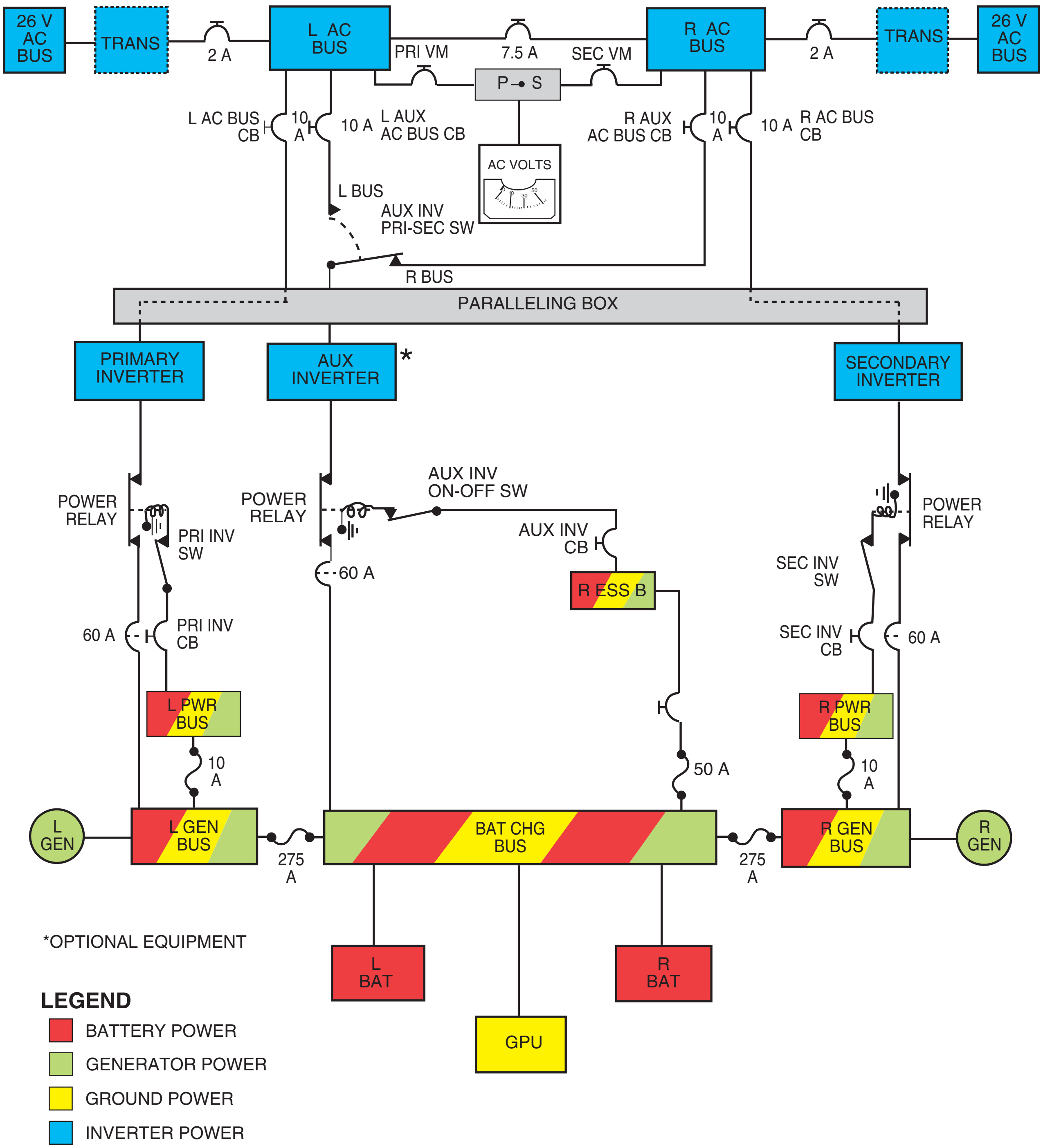


Figure 2-21. AC Distribution

EMERGENCY BATTERY

GENERAL

The airplane may be equipped with either a single (standard) or a dual (optional) emergency battery system. The battery, or batteries, may be installed in either the nose compartment or the tailcone, and provide an emergency electrical power source for selected equipment in the event of total airplane electrical system failure.

Emergency batteries may be nickel-cadmium (nicad) or lead-acid. The nickel-cadmium battery is standard up to airplane SNs 35-462 and 36-052. These airplanes are equipped with AC-powered standby attitude indicators, and the battery packs contain a built-in inverter. On later airplanes, lead-acid batteries and a DC-powered standby attitude indicator are standard. Lead-acid batteries may be retrofitted to earlier airplanes.

The nicad battery provides 25 VDC at 3.8 ampere-hours and contains an inverter and transformer that provide 115 VAC and 4.6 VAC. The lead-acid battery provides 24 VDC at 5.0 ampere-hours.

Both NO. 1 and NO. 2 emergency batteries receive a trickle-charge from the normal airplane electrical system through the respective EMER BAT 1 and EMER BAT 2 circuit breakers on the right main bus anytime power is on the bus. The trickle-charge is provided even when the switches are off, but at a reduced rate. Controls and indicator location are illustrated in Figure 2-22.

SINGLE EMERGENCY POWER SYSTEM

If an airplane is equipped with a single emergency battery, the cockpit switch is labeled "EMER PWR." There is an amber EMR PWR annunciator light on the pilot's instrument panel that illuminates when power from the emergency battery is being used but the trickle-charge from the airplane electrical system has been lost.

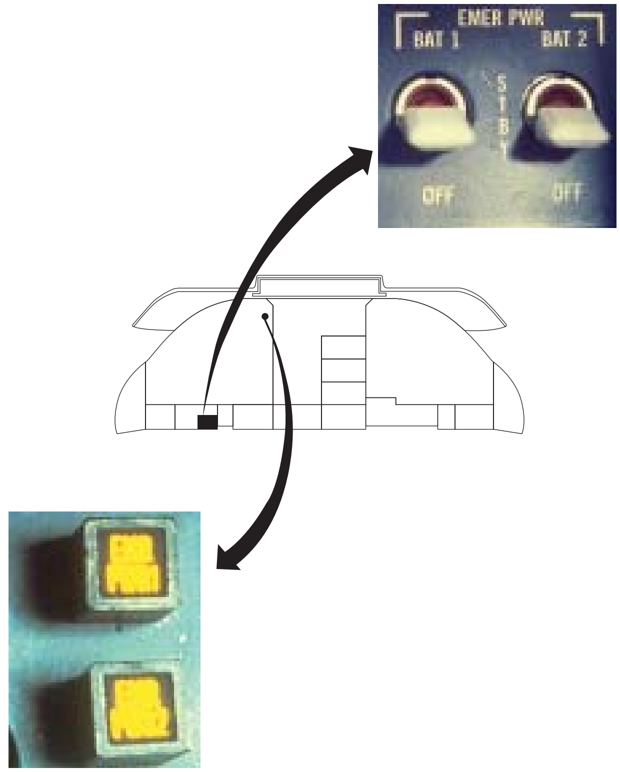


Figure 2-22. Emergency Battery Controls and Indicators

The EMER PWR switch has three positions—ON, STBY, and OFF. The emergency battery powers the following equipment with the switch in the ON or STBY position.

- ON
  - Standby attitude indicator, indicator lighting, and annunciator light
  - Landing gear control circuits and gear position lights
  - Flap control circuits
- STBY
  - Standby attitude indicator, indicator lighting, and annunciator light

With the switch in ON or STBY, the standby attitude indicator is powered from the emergency battery. If power is available from the airplane electrical system, the emergency battery is replenished as it provides power for the standby attitude indicator. The other equipment tied to the emergency battery is normally powered by the airplane electrical system and is powered by the emergency battery only when normal electrical power is off or has failed.

Normally, the EMER PWR switch is in the ON position. In the event of electrical system failure, the EMR PWR light will illuminate when power from the associated emergency battery is being used and is not receiving a trickle-charge. In the event of a total airplane electrical system failure, the approved AFM recommends that the EMER PWR switch be placed in STBY until gear or flap operation is required to conserve battery life. Since only the standby attitude indicator is powered in STBY, battery life is approximately 3 hours and 45 minutes versus 30 minutes in the ON position.

DUAL EMERGENCY POWER SYSTEM

The dual emergency battery system has two switches labeled “EMER PWR” (“BAT 1” and “BAT 2”). An amber EMR PWR annunciator light for each power supply is installed on the pilot’s instrument panel. The applicable EMR PWR light will illuminate when power from the associated emergency battery is being used and is not receiving a trickle-charge.

The BAT 1 switch operates the same systems as described under Single Emergency Power System. The BAT 2 switch has two positions—OFF and BAT 2. When turned on, power from the No. 2 emergency power supply is available to illuminate the EMR PWR 2 light and operate predetermined electrical equipment should the normal electrical system fail. The auxiliary communication radio is the most common equipment powered by BAT 2; however, its installation and use is optional. The pilot must turn off the emergency battery switch(es) before leaving the airplane. If the airplane power is turned off with the emergency battery switch(es) in ON or STBY, the emergency batteries will continue to power the emergency battery equipment and lose their charge.

SCHEMATICS

The following schematics (Figures 2-23, 2-24, and 2-25) are provided to show the three basic electrical circuit configurations, differing only with respect to the essential buses.

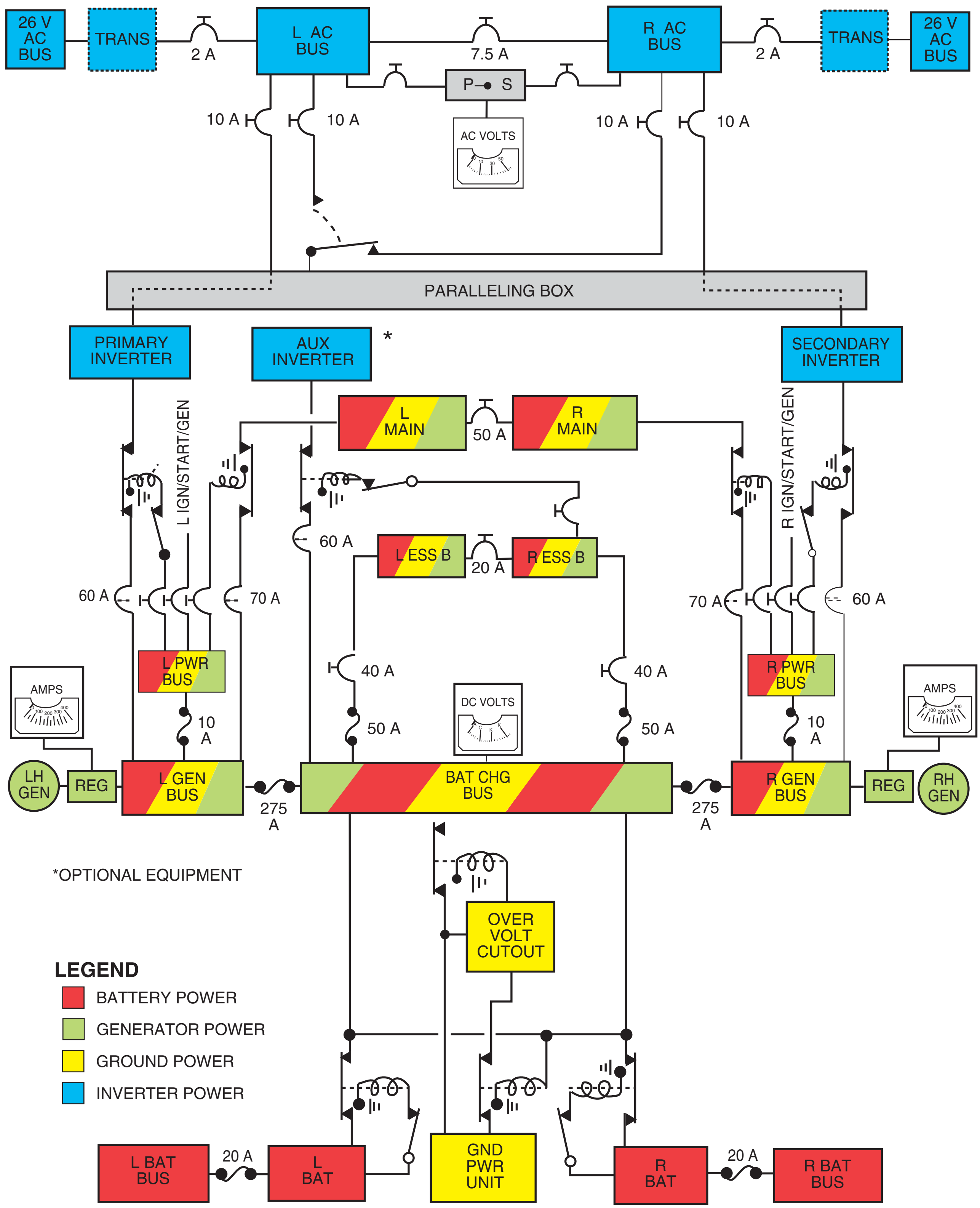


Figure 2-23. Electrical System—SNs 35-002 through 35-205 and 36-002 through 36-040 (Not Incorporating AMK 78-13)

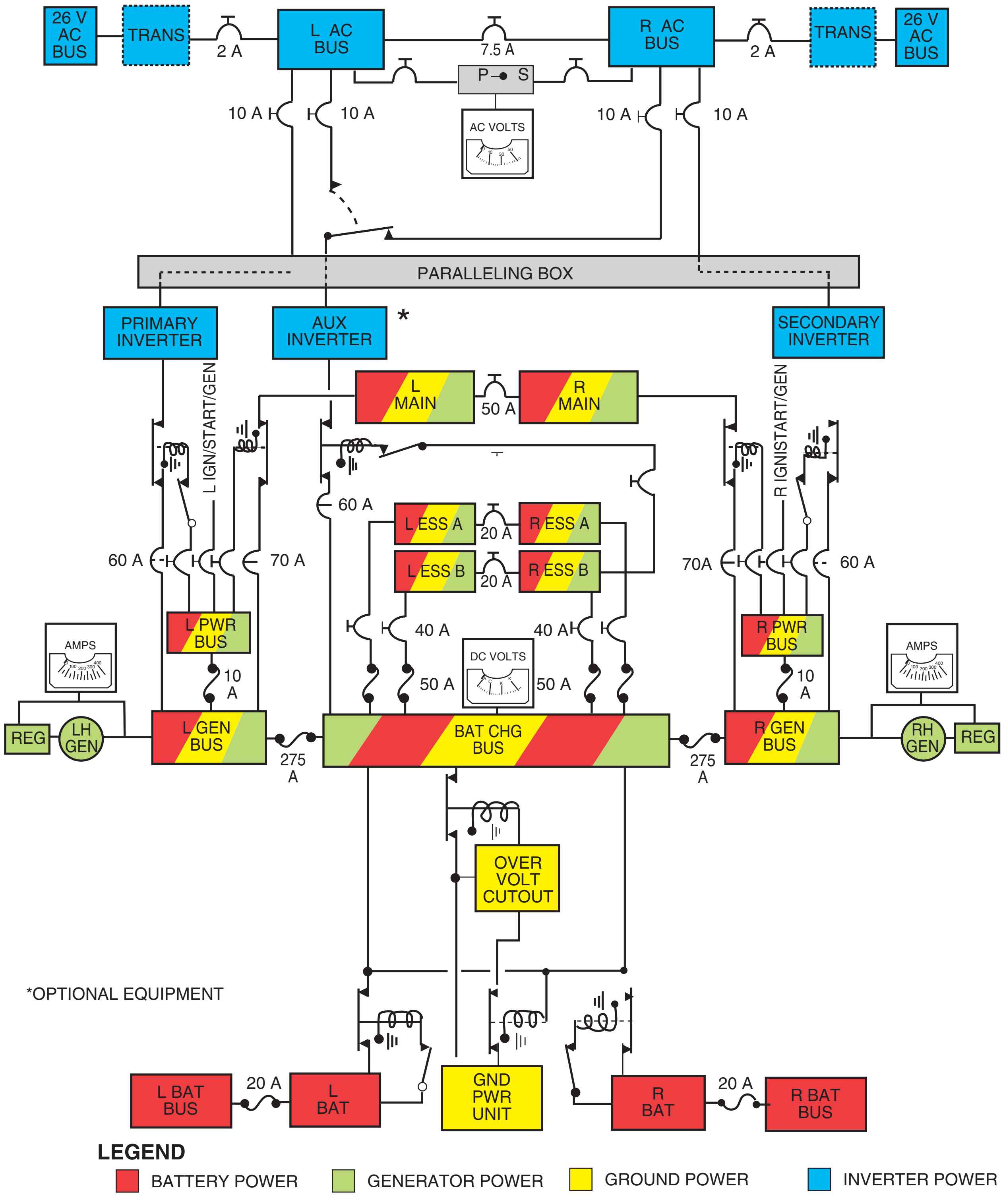


Figure 2-24. Electrical System—SNs 35-202 through 35-204, 35-206 through 35-508, 36-041 through 36-053, and Prior Airplanes Incorporating AMK 78-13

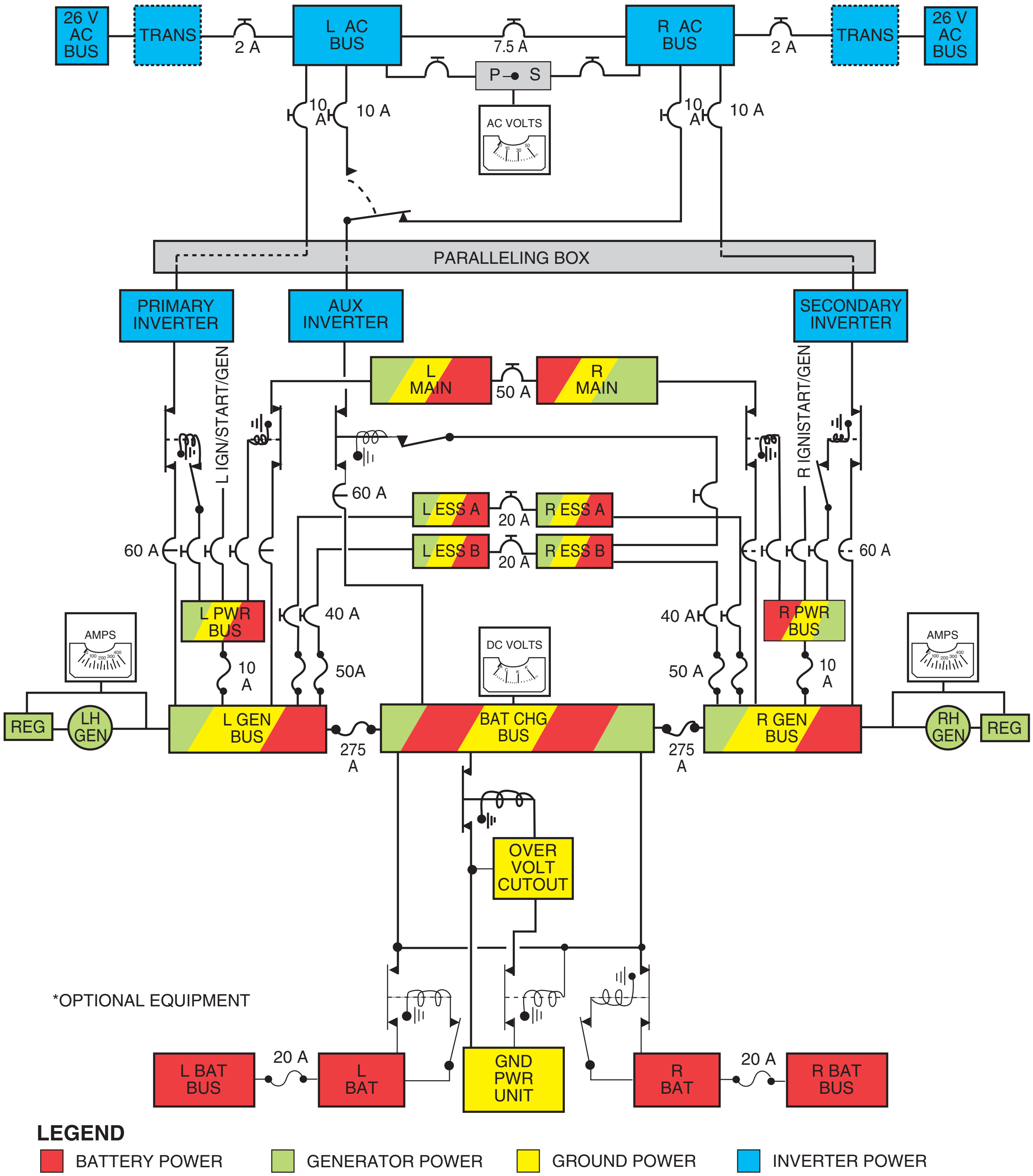


Figure 2-25. Electrical System—SNs 35-509 and Subsequent, 36-054 and Subsequent, and Prior Airplanes Incorporating AMK 85-1

QUESTIONS

1. The DC voltmeter indicates:

A. Battery voltage only
B. Generator voltage only
C. Voltage on the battery buses
C. Voltage on the battery charging bus

2. When a GPU is used for engine start, the output value should be:

A. Regulated to 24 volts
B. Regulated to 28 volts and limited to 1,100 amps
C. Regulated to 33 ± 2 volts
D. Regulated to 28 volts and limited to 500 amps

3. The buses that the airplane batteries power are:

A. Battery buses only
B. Battery and battery-charging buses only
C. All buses except the 115 VAC
D. All buses including AC if an inverter is on

4. A generator failure is indicated when:

A. One ammeter indicates less than 25 amps
B. The GEN switch is in the ON position and the GEN light remains illuminated after activating RESET.
C. The GEN light is extinguished.
D. The DC voltmeter reads less than 28 volts.

5. If airplane electrical power fails and the EMER PWR BAT 1 switch is ON, the systems powered by the emergency battery are:

A. Standby attitude gyro only
B. Flaps and gear only
C. Flaps, gear, and spoiler
D. Standby attitude indicator, gear, and flaps

6. If both 275-amp current limiters fail in flight:

A. The essential buses will remain powered by the airplane batteries.
B. The essential buses will remain powered by the generators.
C. The battery-charging bus will fail immediately.
D. Both inverters will fail.

7. Illumination of a PRI or SEC inverter light indicates:

A. The inverter is operating.
B. The inverter output is less than 90 VAC, or there is less than 10 volt-amps draw on the inverter.
C. The inverter switch is off.
D. B and C

8. The AC voltmeter will indicate:

A. Right AC bus voltage with the AC BUS switch in PRI
B. Left AC bus voltage when the AC BUS switch is in PRI
C. The AC load
D. The voltage on the 26-VAC buses

9. If an overload sensor shuts off power to a main bus, power may be restored by:

A. Resetting the control circuit breaker after the overload sensor resets
B. Changing the overload sensor
C. Automatic action after the current limiter cools
D. Automatic action after the overload sensor cools

10. To unlock the entrance door when the batteries are dead:

A. Plug in a GPU and use a key.
B. Plug in a GPU with 33 ±2 VDC or less on the small pin and use a key.
C. Remove both batteries for charging and reinstall.
D. Enter airplane through the emergency hatch, place the emergency battery switch to ON, and activate the interior door switch.

11. With a dual-generator failure in flight, the airplane batteries will support the minimum night IFR equipment load for approximately:

A. 60 minutes
B. 2 hours 45 minutes
C. 30 minutes
D. 30 minutes with fully charged emergency batteries and emergency BAT 1 in standby position

12. Inverter output is:

A. 115 VAC, 400 Hz
B. 115 VAC and 26 VAC, 400 Hz
C. 26 VAC, 400 Hz
D. 115 VAC and 26 VAC, 1,000 Hz

13. The approved AFM recommends that a GPU be used for engine start when the ambient temperature is:

A. 10° C or below
B. 0° F or below
C. 15° F or below
D. 32° F or below

14. When either primary or secondary inverter light illuminates, the first step of corrective action is:

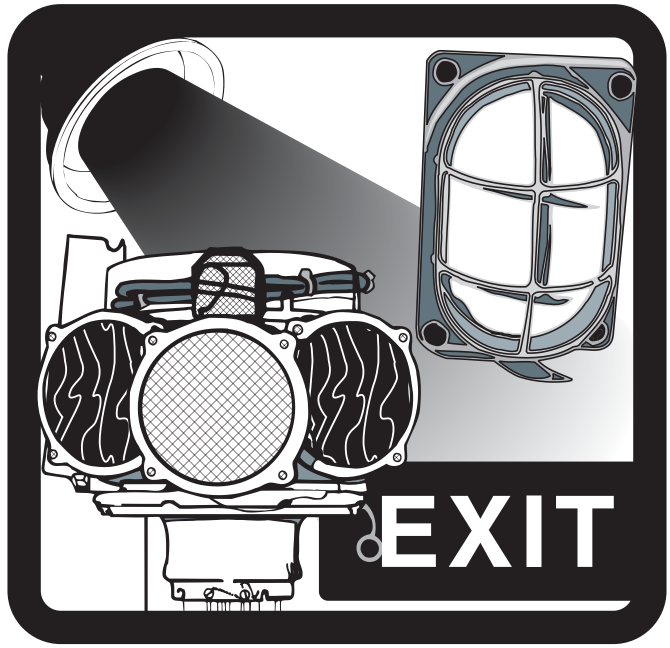
A. Pull the AC bus-tie circuit breaker
B. Turn the respective inverter switch off.
C. Check for open INV or AC BUS circuit breaker(s).
D. Reduce the load on the failed AC bus.

CHAPTER 3
LIGHTING

CONTENTS

ILLUSTRATIONS

CHAPTER 3
LIGHTING



INTRODUCTION

Airplane lighting is divided into interior, exterior, and emergency (if installed) lighting packages. Interior lighting provides illumination of both the cockpit and cabin areas under normal conditions. The cockpit area is provided with general illumination and specific lighting for instruments and map reading. Cabin area lighting provides illumination for the standard warning signs and specific area illumination for passenger safety and convenience. Exterior lighting consists of navigation, landing-taxi, anticollision, recognition, and strobe lights. An optional tailcone area inspection light and two lighting packages to illuminate the wing are available.

An emergency lighting system may be installed as optional equipment, serving to illuminate the cabin interior and egress points in the event of airplane electrical power failure. There are two basic configurations, depending on airplane serialization.

GENERAL

Cockpit lighting consists of the instrument lights, floodlight, electroluminescent lighting, and map lights, all adjustable for intensity with rheostat controls. The electroluminescent lighting illuminates the lettering on the various switch panels, pedestal, and circuit-breaker panels. Optional map lights may be installed, and consist of a flexible-neck light located on each pilot's sidewall panel or one of two overhead light installations, depending on airplane serialization. Cabin lighting consists of eight fluorescent upper center-panel lights (four on 36 models), two door entry lights, baggage compartment lights, individual reading lights, and the no smoking/fasten seat belts sign. The optional emergency lighting systems illuminate the fluorescent upper center-panel lights, and other lights at the exits. Exterior lights include landing-taxi lights, wing and tail navigation lights, anticollision beacons, one (or two optional) recognition light, and high-intensity strobe lights. A wing inspection and egress light (which may be part of the emergency lighting option) illuminates the right wing area to check for ice accumulation, and for emergency egress. An optional wing ice inspection light is available on late models which is ***not*** a part of the emergency lighting system. An optional light inside the tailcone does not require airplane battery switches to be on for operation.

INTERIOR LIGHTING

COCKPIT LIGHTING

General

Some cockpit lighting systems use both incandescent and fluorescent bulbs and, consequently, require both AC and DC power. Controls for lighting are either on the device or as illustrated in Figure 3-1.

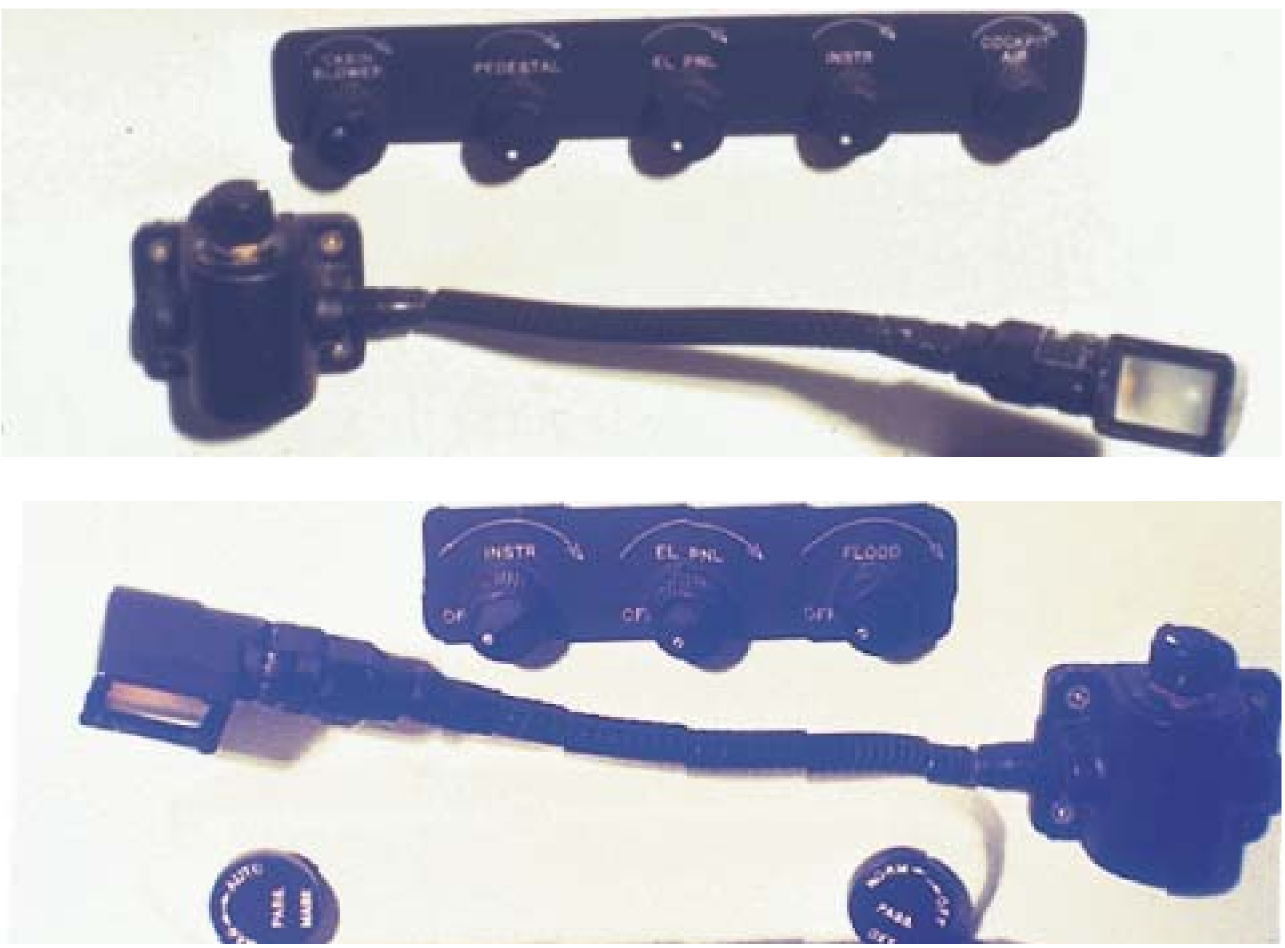


Figure 3-1. Interior Lighting Controls

Instrument Panel Floodlights

A single fluorescent light tube is installed under the glareshield to illuminate the instrument panel. It is controlled by the FLOOD rheostat switch on the pilot's side panel (Figure 3-1). Electrical power required is 115 VAC supplied through the FLOOD LT circuit breaker on the left (primary) AC bus.

Instrument Lights

Incandescent lighting is installed for pilot, copilot, and center instrument panels, pedestal indicators, and the magnetic compass. The lights are controlled with the INSTR rheostat on the pilot's side panel and both INSTR and PEDESTAL rheostats on the copilot's side panel. DC power for the lights is supplied through the respective INSTR LTS circuit breakers on the respective left and right essential buses.

Pilot's INSTR Lights

The pilot's INSTR rheostat provides lighting control for the pilot's flight instruments, engine instruments, clock, electrical indicators, oil temperature indicators, altitude indicator, and the radar edge lighting.

Copilot's INSTR Lights

The copilot's INSTR rheostat provides lighting control for the copilot's flight instruments, the magnetic compass, cabin temperature indicator, BAT TEMP indicator (if installed), landing gear control panel, EMERGENCY AIR and HYDRAULIC PRESSURE indicators, and the pressurization control panel.

PEDESTAL Lights

The PEDESTAL rheostat on the copilot's side panel provides lighting controls for the flight director panel and the pedestal.

Switch Panel Lighting

Electroluminescent lighting is used to illuminate the lettering on all switch panels and both circuit-breaker panels.

Electroluminescent (EL) lighting uses 115 VAC supplied through the EL LTS circuit breakers on the left (primary) and right (secondary) AC buses, respectively. The lights are controlled with the EL PANEL rheostat switches on the pilot's and copilot's side panels, respectively.

EL PANEL Rheostat (Pilot's Sidewall)

The EL rheostat controls all edge lighting on the switch panels to the left of a line running vertically between the radar and radio panels. This control includes dimming for the audio control panel, the left circuit-breaker panel and the pilot's microphone jack panel.

EL PANEL Rheostat (Copilot's Sidewall)

The EL PANEL rheostat controls all edge lighting on switch panels to the right of the vertical line established in the preceding paragraph. It also controls lighting for the copilot's microphone jack panel, audio panel, and the right circuit-breaker panel.

Maps Lights (Optional)

When installed, the airplane may have one or more of three different map light options: (1) flexible neck light on each pilot's sidewall panel, with an ON-OFF rheostat for intensity control (Figure 3-1); (2) a reading light and gasper assembly, installed in the cockpit headliner for each pilot, incorporating a rheostat for light intensity adjustment and a light pattern adjustment lever (Figure 3-2); (3) a dome light assembly, mounted on each side of the headliner just forward of the upper air outlets incorporating a rocker-operated

Figure 3-2. Cockpit Map Lights

Figure 3-2A. Reading Lights (Typical)

switch (labeled ON-REMOTE) with an unlabeled center off position (Figure 3-2) and a swivel-mounted light.

All installations are powered through the INSTR LTS circuit breakers on the left and right essential buses. In the REMOTE position, the dome lights are powered from the ENTRY LT circuit breaker on the left battery bus.

CABIN LIGHTING

General

Passenger compartment lighting consists of reading lights, overhead lights, entry lights, no smoking/fasten seat belt signs, and refreshment cabinet lights.

Reading Lights

The reading lights are mounted in the upper center panel above the seats on each side of

Figure 3-3. Overhead Lights Control (Typical)

the cabin. There are individual switches for each light. The lights are adjustable for position and use DC power supplied through the RDNG LTS circuit breaker on the left main bus (Figure 3-2A).

Overhead Lights

The cabin overhead light system consists of four (three on 36 models) fluorescent lights recessed in each side of the upper panel, a cabin lights power supply, a three-position switch, a cabin lights relay assembly, and a CAB LTS circuit breaker on the left main bus.

Normally, the lights are controlled with the three-position switch located on the left service cabinet forward of the entry door (Figure 3-3).

In the event of cabin depressurization, the lights automatically illuminate full bright when the cabin altitude reaches 14,000 feet. On airplanes equipped with the optional emergency lighting system, three overhead lights illuminate automatically in the event of airplane electrical power failure.

Entry Lights

The entry light system consists of a STEP LIGHT switch and light on the left service cabinet forward of the entry door (Figure 3-3), and another directly over the door opening. Power from the left battery bus is supplied through the ENTRY LT circuit breaker on the left battery bus; therefore, the lights are operable when the airplane BAT switch is in OFF.

Baggage Compartment Lights

Two lights are installed in the aft baggage compartment and, on 36 model airplanes, one light is installed in the forward baggage compartment. Aft baggage compartment lights are controlled by a switch on the left service cabinet forward of the entry door (Figure 3-3) and are powered through the ENTRY LT circuit breaker on the left battery bus. The forward baggage compartment light is controlled by a switch on the forward end of the upper center panel.

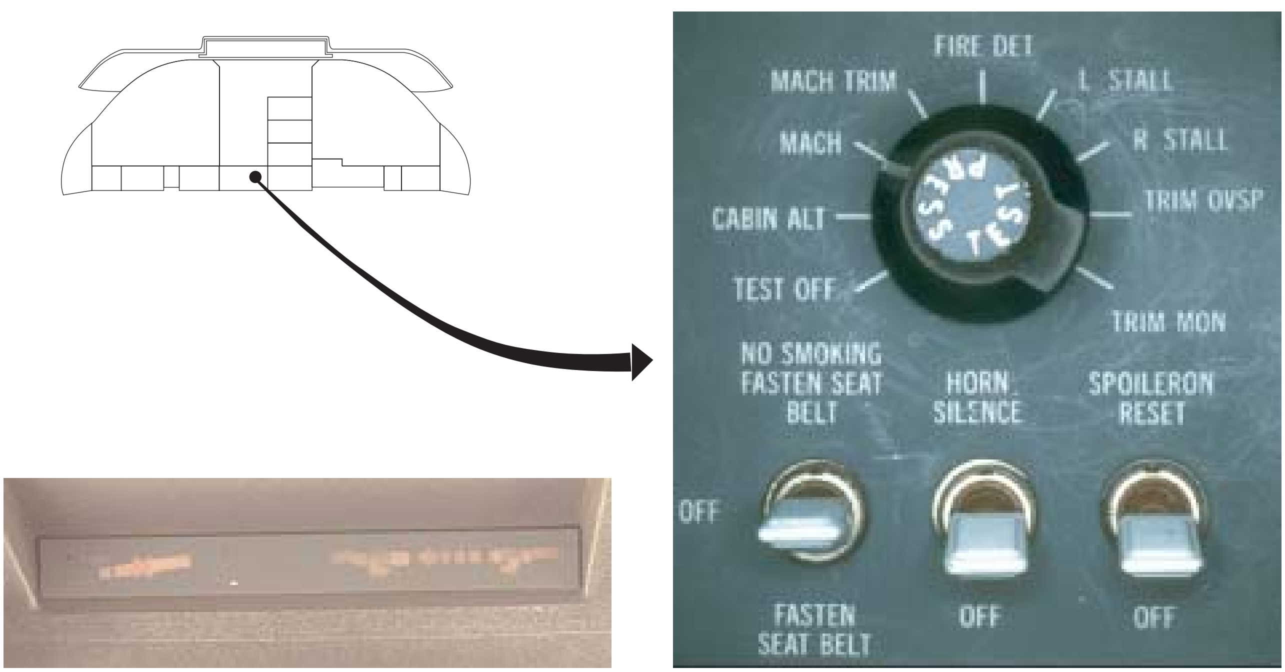


Figure 3-4. Advisory Lights and Controls

Passenger Advisory Lights

The no smoking/fasten seat belts advisory light system consists of two fixtures (one on 36 models) (Figure 3-4), a switch on the center switch panel, and the RDNG LT circuit breaker on the left main bus. The switch has three positions—NO SMOKING/FASTEN SEAT BELT–OFF–FASTEN SEAT BELT. When the switch is moved from OFF to either of the other positions, an audible tone sounds and the appropriate symbols illuminate. A RETURN TO SEAT light (if installed) in the lavatory is a part of the advisory light system. Location of the fixtures varies with cabin configuration.

Cabinet Lights

The cabinet light system varies with cabin configuration and consists of various lights within the refreshment cabinet, microswitches actuated by doors or drawers, power supplies, and a circuit breaker on the right essential bus.

EMERGENCY LIGHTING

Cabin Interior and Wing Inspection and Egress Lights

If these lights are installed, the airplane is equipped with two nickel-cadmium (nicad) battery power supplies and a control module, which function to illuminate selected areas automatically in the event of airplane DC power failure. An emergency light is installed in the upper cabin door (Figure 3-5) to illuminate the lower cabin door and the immediate door area. A second light illuminates the emergency exit window area. An exterior wing inspection/egress light optionally installed on the right side of the airplane is adjacent to the emergency exit window and illuminates the exterior egress area. The cabin upper-center panel (fluorescent) lights illuminate the cabin interior. When activated, one of the power supplies turns on the cabin upper-center panel lights, while the other power supply turns on the upper cabin door light, the emergency exit light, and the wing inspection/egress light. The nicad battery packs are charged through the EMER LTS circuit breaker on the right essential bus.

Control Module

The EMER LIGHT TEST switch on the pilot's (or center) switch panel (See Figure 3-6) provides the test function for the system and for automatic illumination of the emergency lights in the event of an interruption of normal DC electrical power. The switch has three positions: TEST, ARM and DISARM. Setting the switch to TEST simulates a failure of normal DC electrical power and illuminates the upper cabin entry door light, the emergency exit light, and the cabin overhead fluorescent lights. Setting the switch to ARM will arm the system to illuminate the emergency lights

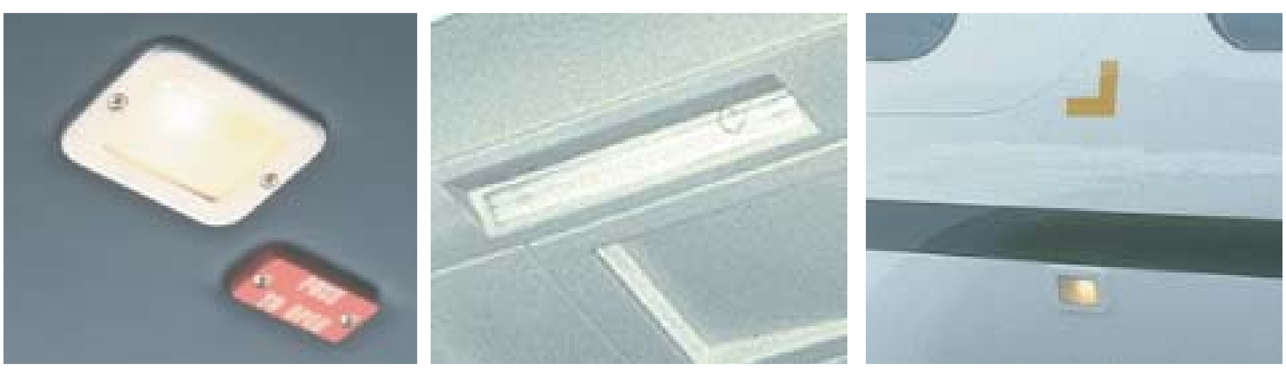


Figure 3-5. Emergency Cabin Door Light, Emergency Exit Light, and Wing Inspection/Egress Light

Figure 3-6. Emergency Lights Control

in the event of a failure of normal DC electrical power. Setting the switch to DISARM isolates the emergency lights from the emergency batteries. The switch should be set to ARM prior to takeoff. If the switch is in the DISARM position and at least one BAT switch is on, the amber light adjacent to the switch will illuminate to remind the pilot that the switch should be set to ARM. The switch should be set to DISARM prior to setting the BAT switches to OFF.

The WING INSPECTION light switch (included as part of the emergency lighting system), located adjacent to the EMERG LIGHT TEST–ARM–DISARM switch, may be used independently of the rest of the emergency lighting system to visually check for ice accumulation on the wing leading edge. Turning the switch on illuminates the exterior wing inspection/egress light.

The EMERGENCY LTS. switch on the left service cabinet near the entry door provides a means for manual illumination of the interior emergency lights. When the switch is set to EMERGENCY LTS., the upper cabin entry door light, the emergency exit light, the cabin overhead fluorescent lights, and the wing inspection/egress light (if installed) will illuminate. For normal operation, the

Figure 3-7. Exterior Lighting Locations

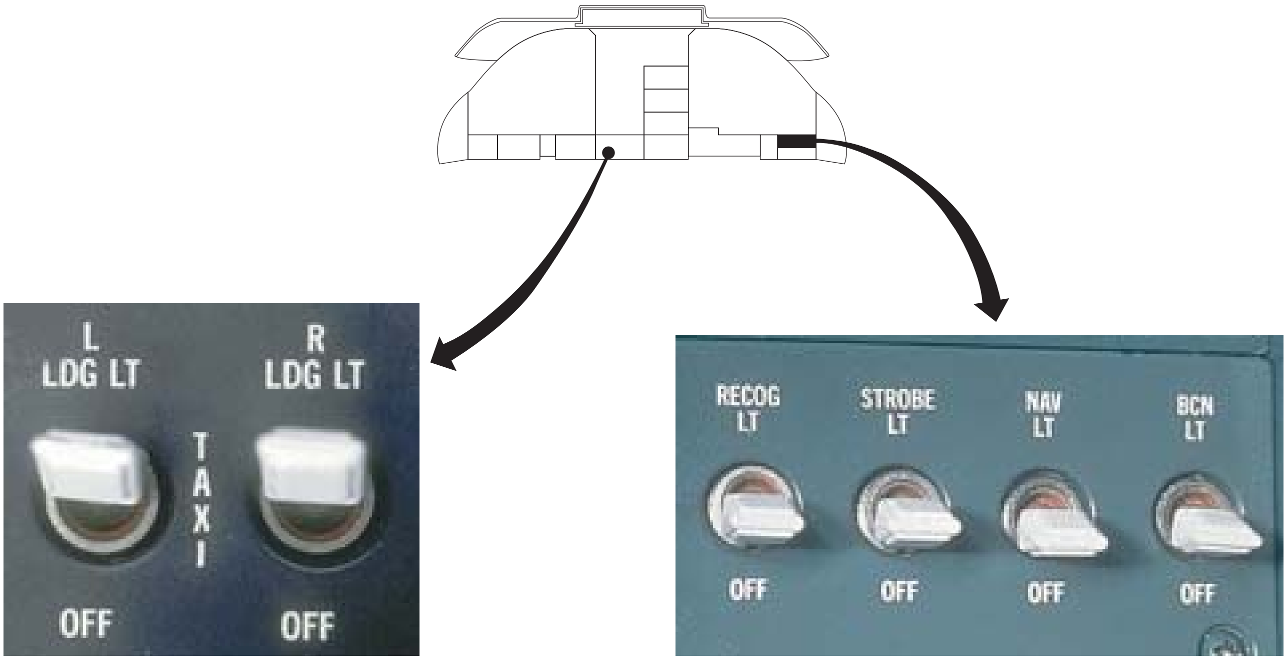


Figure 3-8. Exterior Lighting Controls

switch should be set to OFF, allowing automatic illumination of the emergency lights in the event of a failure of the normal electrical system.

EXTERIOR LIGHTING

GENERAL

The exterior lighting systems consist of the landing-taxi lights, navigation lights, anti-collision lights, recognition light(s), strobe lights, and an optional wing ice inspection

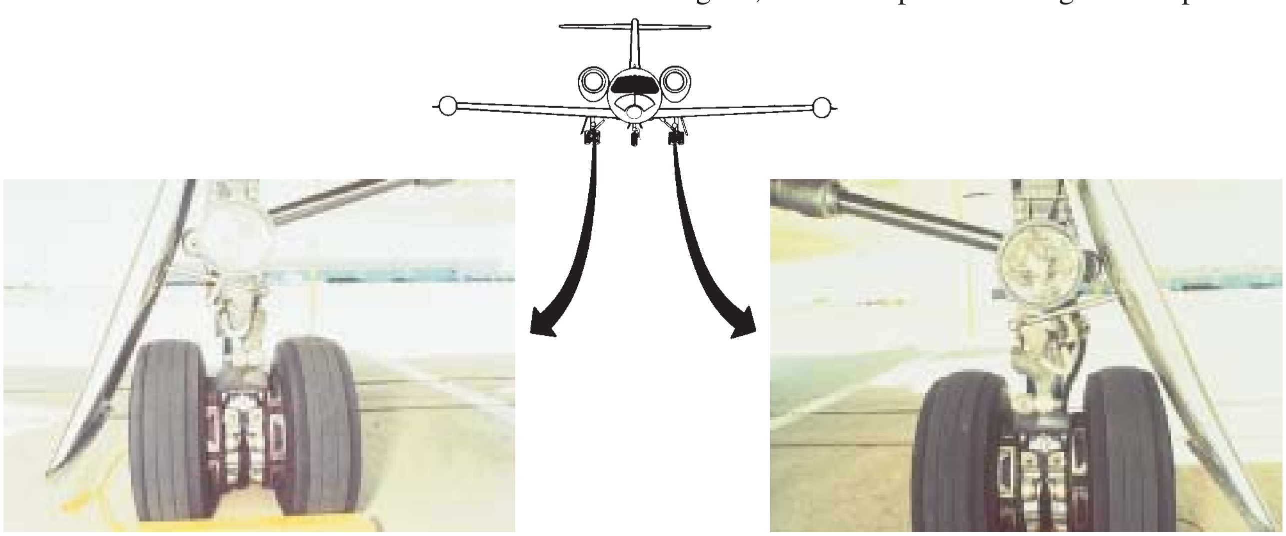


Figure 3-9. Landing-Taxi Lights

light (Figure 3-7). The exterior lighting controls are shown in Figure 3-8.

LANDING-TAXI LIGHTS

The landing light system consists of one 450-watt lamp mounted on each main landing gear strut (Figure 3-9), one 20-amp current limiter for each side in the current-limiter panel, relays, dimming resistors, and the L and R LDG LT switches on the center switch panel. The L and R landing light switches have three positions: OFF, TAXI and LDG LT. DC power to operate the relays comes from the left and right main buses, respectively.

Setting the L or R LDG LT switch to TAXI closes a relay which shunts DC power from the respective generator bus through a resistor which limits current to the lamp element. Moving the switch to LDG LT closes a second relay, allowing current flow to bypass the resistor, thereby increasing the brightness of the lamp. The 20-amp current limiters protect the power circuits between the respective generator bus and lamp filament.

Regardless of switch position, the lights will not illuminate unless the respective landing gear down-and-locked switches are closed and provide a ground. It is recommended that the lights be operated in the L and R LDG LT modes as sparingly as possible. Lamp service life is shortened in the LDG LT mode because of the higher current flow.

RECOGNITION LIGHT

A 250-watt recognition light is installed in the nose of the right tip tank (Figure 3-10). The light is controlled with the RECOG LT switch on the copilot's lighting control panel. When turned on, DC power, applied through the RECOG LT circuit breaker on the right essential bus, closes a control relay and connects power through a 30-ampere current limiter to the light. A second recognition light may be installed in the let tip tank as optional equipment.

Figure 3-10. Recognition Light

Figure 3-11. Strobe and Navigation Lights

STROBE LIGHTS

The strobe light system consists of a strobe light mounted inside each navigation light fixture, a power supply for each strobe (Figure 3-11), a STROBE LT switch on the copilot's lighting control panel, a DC STROBE LTS circuit breaker on the left main bus, and a timing circuit module that causes the strobes to flash. Each power supply is protected by an internal 3-ampere fuse.

NAVIGATION LIGHTS

The navigation light system consists of one lamp in the outboard side of each tip tank, two lamps in the upper aft tail fairing, a NAV LT switch on the copilot's lighting control panel, and a NAV LTS circuit breaker on the left main bus.

All three navigation lights are controlled by the NAV LT switch. Additionally, setting the NAV LT switch to ON automatically dims most instrument panel and pedestal "peanut" lights, and activates the landing gear position light dimmer rheostat.

ANTICOLLISION LIGHTS

Anticollision lights are installed on top of the vertical stabilizer and on the bottom of the fuselage (Figure 3-12). The lights are controlled by a BCN LT switch on the copilot's lighting control panel. Each light is a dual-bulb light and each bulb oscillates 180 degrees at 45 cycles per minute. The beam is concentrated by an integral lens, and an illusion of 90 flashes per minute occurs due to the oscillation. The lights operate on DC power through the BCN LT circuit breaker on the right main bus.

Figure 3-12. Anticollision Lights

WING INSPECTION LIGHTS

Two separate installations are designed to illuminate the wing area for signs of ice (Figure 3-13). Both are optional. One light is installed on the right side of the fuselage adjacent to the lower forward corner of the emergency exit window. This light is designed to illuminate the leading edge of the right wing and additionally serves as an illumination source for emergency egress over the wing. The light is designated the "wing inspection and egress light," and may be installed as an integral part of the earlier emergency lighting system, or as a selected option not involving the emergency lighting system. In either case, a second option may include a second light installed on the left side of the fuselage directly opposite the right-hand light, serving as an inspection light for the left wing. The WING INSPECTION control switch is located on the emergency lighting panel or on the instrument panel (Figure 3-6).

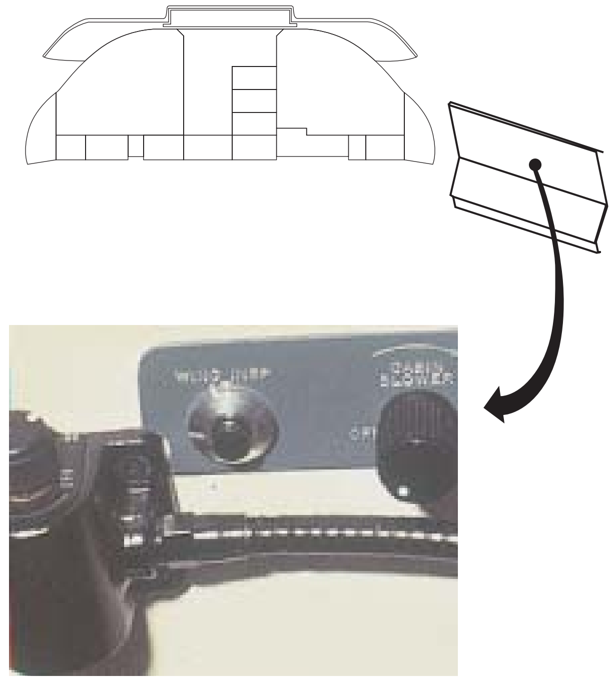


Figure 3-14. Wing Ice Inspection Light Control

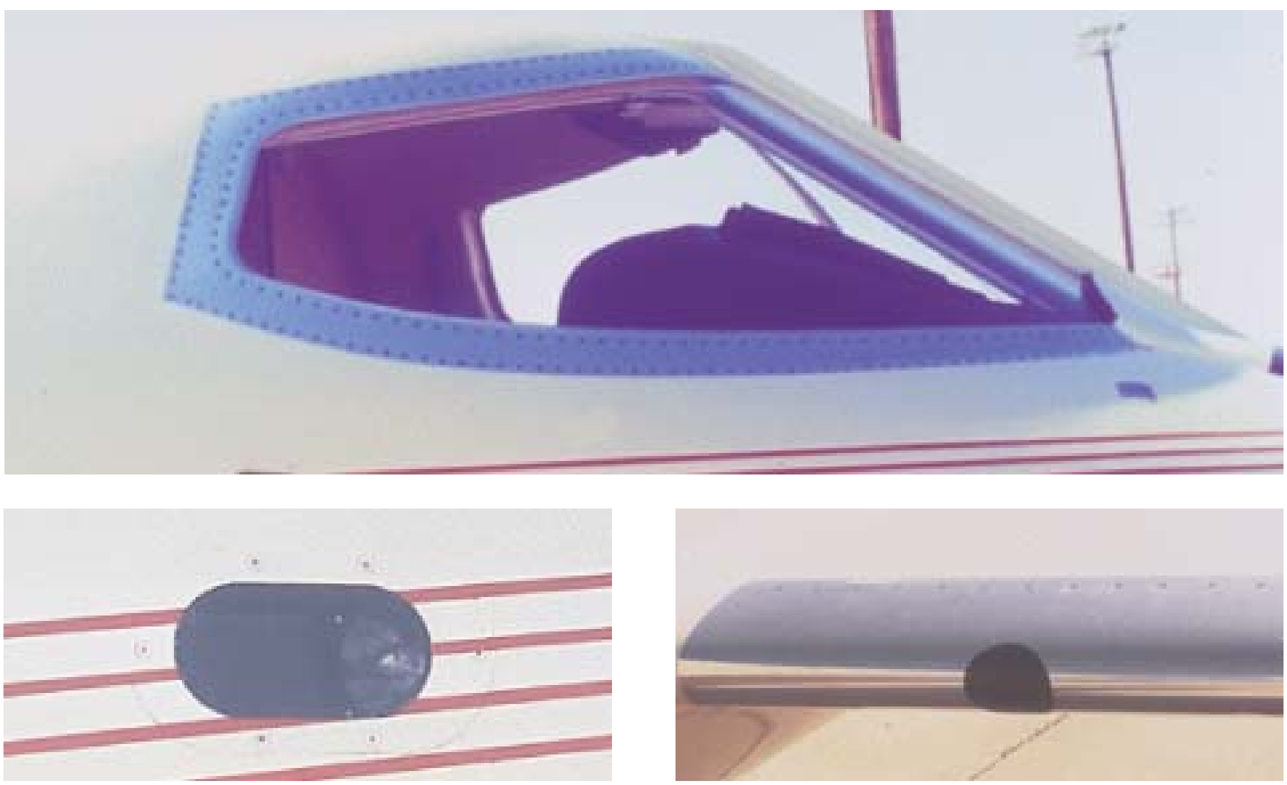


Figure 3-13. Wing Ice Inspection Light

On airplane SNs 35-416 and 36-048 and subsequent, another option provides a light installed in the fuselage below the copilot's window. It is designed to illuminate a black spot on the right wing leading edge. A covering of ice obscures the spot, enabling ice detection at night when the light is turned on. This light is designated the "WING INSP light" (Figure 3-13) and is operated by a push-button switch located forward of the rheostats on the copilot's right side panel (Figure 3-14).

TAILCONE AREA INSPECTION LIGHT (OPTIONAL)

When installed, this light is located in the tailcone, directly above the entry door. An ON—OFF switch is positioned inside the door at the forward left side of the opening. A microswitch installed on the forward right side of the opening breaks power to the light when the door is closed (Figure 3-15). Power for operating the tailcone area inspection light option is provided by the left battery bus through the ENTRY LT circuit breaker (pilot's circuit-breaker panel), which permits operation of the light without turning airplane power on. However, on some airplanes, the light is powered by the battery charging bus through a 5-amp current limiter, in which case an aircraft battery must be turned on to operate the light.

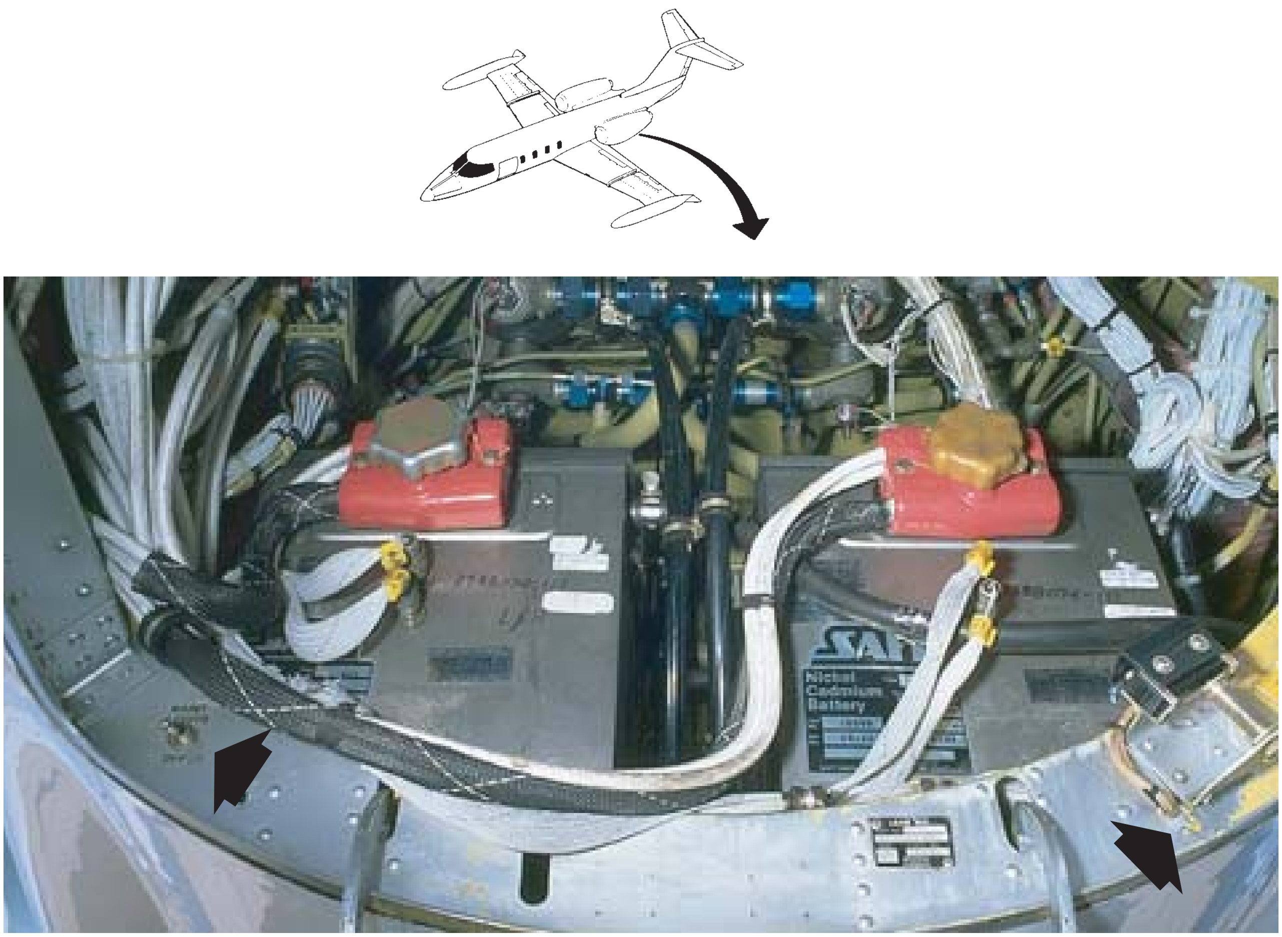


Figure 3-15. Tailcone Inspection Light Switches

QUESTIONS

1. The instrument panel flood light control is located:
   A. On the light
   B. Just forward of the warning panel
   C. On the pilot's side panel
   D. On the copilot's side panel

2. The cockpit map lights are controlled:
   A. With an ON-OFF switch on the copilot's side panel
   B. With the overhead map light rheostat on the copilot's side panel
   C. With an integral rheostat and a pattern lever
   D. Automatically, relative to ambient light

3. The cabin overhead light control switches are located on the:
   A. Right forward refreshment pedestal
   B. The entrance door threshold
   C. Left forward service cabinet
   D. Light assembly

4. When a cabin overhead light switch is turned on, first select:
   A. ON
   B. OFF
   C. DIM
   D. BRT

5. The lights that are illuminated by the emergency lighting system are the:
   A. Instrument panel floodlights and electroluminescent lights
   B. Cabin overhead lights, wing egress light, and emergency exit light
   C. Navigation lights
   D. Strobe lights

6. The emergency lighting switch position used during normal operation is:
   A. DISARM
   B. ARM
   C. TEST
   D. EMER LT

7. The lights that come on when cabin altitude reaches 14,000 feet or higher are the:
   A. Passenger advisory lights
   B. Lavatory lights
   C. Cabin overhead panel lights
   D. Reading lights

8. The wing ice inspection light switch (if installed) is located on the:
   A. Pilot's switch panel
   B. Light assembly
   C. Overhead panel
   D. Copilot's right sidewall

9. The lights that require inverter power are the:
   A. Cabin overhead lights
   B. FLOOD and EL lights
   C. INSTR lights
   D. NAV lights

10. The lights that can be operated with the airplane batteries turned off are the:
    A. Entry lights and baggage compartment light
    B. Overhead lights
    C. Passenger advisory lights
    D. Reading lights

CHAPTER 4
MASTER WARNING SYSTEM

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CHAPTER 4
MASTER WARNING SYSTEM

INTRODUCTION

The master warning system provides a warning for airplane equipment malfunctions,an indication of an unsafe operating condition requiring immediate attention, and an indication that a system is in operation.

GENERAL

The warning light system incorporates two horizontal rows of red, amber, and green lights (see Annunciator panel section), which alert the pilots to various conditions or switch positions, and are located on the center portion of the glareshield just above the autopilot-flight director panel. These lights are referred to as glareshield annunciator lights. Two MSTR WARN lights on the instrument panel, one in front of each pilot, flash when any red light on the glareshield panel illuminates. These flashing lights serve to draw pilot attention to the glareshield lights and, thereby, to the malfunctioning system.

Provision is made to test all glareshield annunciator lights with two switches, one located on either end of the glareshield just beneath the glareshield lights panel.

The intensity of the glareshield annunciator lights is controlled automatically.

There may be other annunciator lights located on the instrument panel, center pedestal, or thrust reverser control panel (if installed). These lights function as system advisory annunciators.

GLARESHIELD ANNUNCIATOR LIGHTS

The red, amber, and green glareshield lights receive power from the left and/or right essential DC buses through the respective WRN LTS circuit breakers. The red lights are used to indicate the more critical malfunctions. Amber lights denote cautionary items, and green lights indicate conditions which may be normal but need to be announced.

If a glareshield annunciator light illuminates and the condition is corrected, the light extinguishes; should the condition recur, the light again illuminates.

Five of the glareshield annunciator lights give a flashing indication under the following conditions:

1. SPOILER—If spoilers and flaps are both extended (flaps more than 13°)
2. STALL (L or R)—If the angle-of-attack indicators reach shaker limits (yellow band)
3. FIRE (L or R)—If the warning system detects a fire or overtemperature condition in the engine nacelle

NOTE

On airplane SNs 35-002 through 35-431 and SNs 36-002 through 36-049, the MSTR WARN lights may not cancel when any of these red glareshield lights are flashing.

MASTER WARNING LIGHTS

Anytime a red glareshield annunciator light illuminates, the red MSTR WARN lights on the pilot's and copilot's instrument panels also illuminate and flash. Pressing either MSTR WARN light causes both MSTR WARN lights to extinguish (except when triggered by a flashing red annunciator light on the early airplanes mentioned above. However, the red glareshield annunciator light remains illuminated as long as the causative condition exists.

TEST

Depressing either of the two test switches under the glareshield (See Figure 4-1) causes the following lights to illuminate:

- All glareshield annunciator lights and both MSTR WARN lights
- FIRE warning lights
- Marker beacon lights (if installed)
- Thrust reverser panel annunciator lights (if installed)
- AFCS/control panel annunciator lights (FC-530 AFCS)
- ANTISKID lights
- AIR IGN lights
- Fuel panel lights
- Copilot's flight director annunciator lights
- Dual PITOT HT indicator lights (if installed)
- Starter-engaged lights (if installed)
- Rotary test switch current limiter light (if installed)

Figure 4-1. Test Switch

INTENSITY CONTROL

A photoelectric cell outboard of each FIRE handle (Figure 4-1) automatically adjusts the glareshield annunciator light intensity for existing cockpit light conditions. The other instrument panel and pedestal annunciator lights dim when the navigation light (NAV LT) switch is turned on.

BULB CHANGE

Glareshield annunciator light lenses can be removed for bulb replacement.

ILLUMINATION CAUSES

The tabulation in Table 4-1 shows each annunciator light label, color, and cause for illumination.

NOTE

Some lights are optional, and arrangements may vary between airplanes.

Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES

ANNUNCIATOR	CAUSE FOR ILLUMINATION
DH	At or below altitude set on radio altimeter
LOW FUEL	Fuel is below 400-500 pounds in either wing tank
L FUEL PRESS R FUEL PRESS	Less than 0.25-psi fuel pressure to engine (Light extinguishes at 1 psi.)
SPOILER (FC 200) SPOILER (FC530)	Steady–Spoilers not locked down (normal if extended) Flashing–Spoilers deployed with 13° or more flaps extended (normal on landing roll)
DOOR	One of 10 latch pins not fully engaged or hook motor not fully retacted
AUG AIL	1. Spoilers split 6° or more 2. Spoiler and aileron split 6° or more in spoileron mode
PITOT HT	1. One or both pitot heaters is inoperative with the switches on. 2. One or both pitot heat switches is off.
FUEL FILTER	Differential pressure is 1.25 psi across one or both airframe fuel filters. Fuel is bypassing the filter.
L ENG ICE R ENG ICE	1. Switch ON–Insufficient pressure to nacelle or fan spinner or failure of valve(s) to open 2. Switch OFF–Nacelle or fan spinner valve(s) open
L FUEL CMPTR R FUEL CMPTR	1. Switch is off 2. Computer has failed with the switch on.
L STALL R STALL	1. Steady–system is off or failed. (During pusher actuation it is normal.) 2. Flashing–In shaker range
L VG MON R VG MON	One motor in the vertical gyro has failed.

Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES (Cont)

ANNUNCIATOR	CAUSE FOR ILLUMINATION
MACH TRIM	System is inoperative with speed above 0.69 Mach and autopilot disengaged. If above 0.74 Mach, the overspeed warning horn sounds.
PRI INV SEC INV	1. Inverter is off. 2. Inverter switch is on and output is less than 90 volts, or less than 10 volt-amps
AUX INV	Inverter has failed with the switch on.
LO OIL PRESS	Oil pressure on one or both engines is below 23 ±1 psi.
STAB OV HT	Stabilizer structural temperature is above 215° F.
WSHLD OV HT	Windshield heat has been shut off by a temperature limit. GND–High or low limit AIR–High limit only
STEER ON	Nosewheel steering is engaged
BLEED AIR L BLEED AIR R	1. Overtemperature of pylon (250° F) or duct (590° F/645°F) 2. Both lights–Manifold overpressure (47 psi) on SNs 35-082, 35-087 through 35-106, 35-108 through 35-112, 36-023 through 36-031, and AMK 76-7
L GEN R GEN	Indicated generator is off or has failed.
CAB ALT (Late ECS Only)	Cabin altitude has reached 8,750 ±250 feet and controller has automatically switched to manual control.

ANNUNCIATOR	CAUSE FOR ILLUMINATION
WING OV HT	Wing structural temperature is above 215° F.
WSHLD HT	The windshield anti-ice valve is open.
ALC AI	1. Late ECS–the alcohol tank is empty. 2. Early ECS–alcohol system pressure is low.
BAT 140	One or both batteries' temperature is 140° F or more.
BAT 160	One or both batteries' temperature is 160° F or more.
ENG SYNC	The engine sync switch is on with the nose gear down and locked.
TO TRIM	Airplane is on the ground and the horizontal stabilizer is not trimmed for takeoff.
CUR LIM	Failure of either or both 275-amp current limiter (SNs 35-370, 35-390 and subsequent and 36-048 and subsequent).
ARMED	Fire-extinguishing bottles are armed.
FIRE PULL	Fire/overheat is detected in associated engine.
MSTR WARN	A red light on the master warning panel is illuminated.
LOW HYD FUEL XFLO L LO OIL R LO OIL	LOW HYD–Hydraulic system pressure is 1,125 psi or less. FUEL XFLO–Fuel crossflow valve is open. L LO OIL, R LO OIL–Indicated engine oil pressure is low.
PITCH TRIM FC 530 AFCS	1. Primary pitch trim is running at fast rate with flaps up. 2. Primary pitch trim has a fault (potential runaway). 3. Wheel master switch is depressed

Table 4-1. ANNUNCIATOR ILLUMINATION CAUSES (Cont)

ANNUNCIATOR	CAUSE FOR ILLUMINATION
LH ENG CHIP RH ENG CHIP	Ferrous metal particles are detected in indicated engines oil.
EMER PWR 1 EMER PWR 2	Indicated emergency battery is powering the connected systems.
START L START R	Indicated starter is engaged.
COMPTR WARN	HSI headings are not within 7°.
NAC HEAT ON	L or R NAC HEAT switches are ON.
OR	
L NAC HEAT R NAC HEAT	Indicated NAC HEAT switch is ON.

ANNUNCIATOR	CAUSE FOR ILLUMINATION
AIR IGN L AIR IGN R	Ignition system is activated.
ANTI-SKID GEN L R	Indicated antiskid generator is inoperative.
L CUR LIMITER R CUR LIMITER (AMK 80-17)	Indicated 275-ampere current limiter has failed.
L PITOT HEAT R PITOT HEAT	1. Indicated pitot heat switch is off. 2. Switch is on and indicated pitot heat has failed.
PARK BRAKE	1. Parking brake is set. 2. Parking brake handle is not fully in after releasing parking brake.
WSHLD DEFOG L R	1. Illuminates momentarily when WSHLD DEFOG is set ON. 2. Indicates an over heat/underheat conditiion when ON.

QUESTIONS

1. All glareshield annunciator lights and system advisory annunciator lights can be tested by:
   A. The rotary test switch
   B. Depressing each individual light
   C. Depressing either glareshield TEST switch
   D. Shutting the represented system off

2. When a red glareshield annunciator light illuminates, another annunciation that occurs is:
   A. Only the pilot's MSTR WARN light flashes.
   B. Both MSTR WARN lights illuminate steady.
   C. Only the copilot's MSTR WARN light illuminates.
   D. Both MSTR WARN lights flash.

3. An illuminated glareshield annunciator light suddenly extinguishes, indicating:
   A. Five minutes have passed.
   B. The malfunction no longer exists.
   C. Three minutes have passed
   D. The MSTR WARN lights have been reset.

4. The glareshield annunciator light intensity is adjusted:
   A. Automatically by photoelectric cells
   B. By depressing the TEST button
   C. By depressing each individual capsule
   D. By depressing the DIM button

5. The flashing MSTR WARN lights can be reset by depressing either MSTR WARN light:
   A. Unless a red glareshield annunciator is flashing
   B. Anytime
   C. Unless a red glareshield annunciator is illuminated steady
   D. Unless an engine FIRE PULL light illuminated steady

CHAPTER 5
FUEL SYSTEM

CONTENTS

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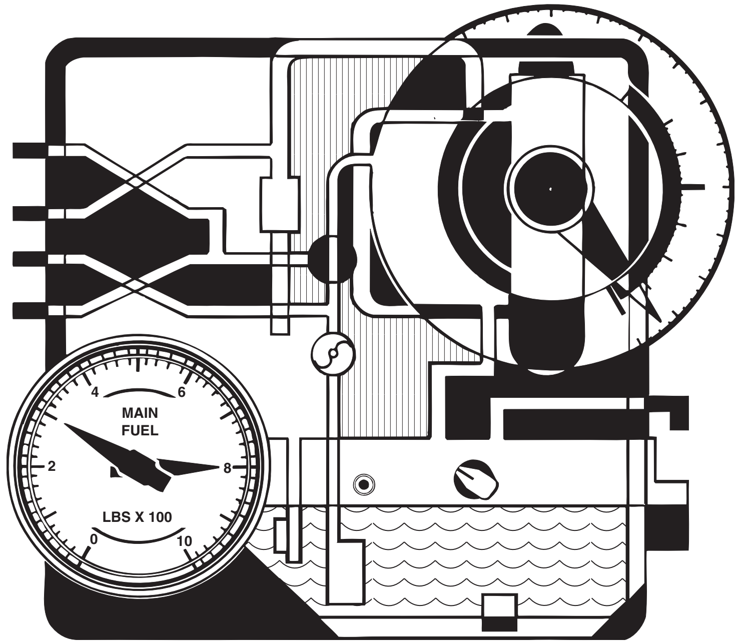
CHAPTER 5
FUEL SYSTEM



INTRODUCTION

The Learjet 35/36 series fuel system consists of the fuel tanks, tank venting, indicating, distribution, transfer, and jettison systems.

This chapter covers the operation of the fuel system up to the engine-driven fuel pumps. At that point, fuel system operation becomes a function of the engine. Refer to Chapter 7, "Powerplant," for additional information.

GENERAL

The fuel storage system consists of tip tanks, integral tanks in each wing, and a fuselage tank. A crossflow valve permits fuel transfer between the wings for fuel balancing.

Each wing tank contains a jet pump and an electric standby pump to supply fuel to the engine on the same side. Tip tank and fuselage tank fuel must be transferred into the wing tanks by jet pumps and an electric pump, respectively.

A ram-air system is used to vent all tanks. Drain valves are provided to remove condensation and contaminants from the low points in the fuel tanks, and to drain the contents of the vent system sump.

Tip tank fuel can be jettisoned, if required.

Figure 5-1 depicts the Learjet 35/36 series fuel systems.

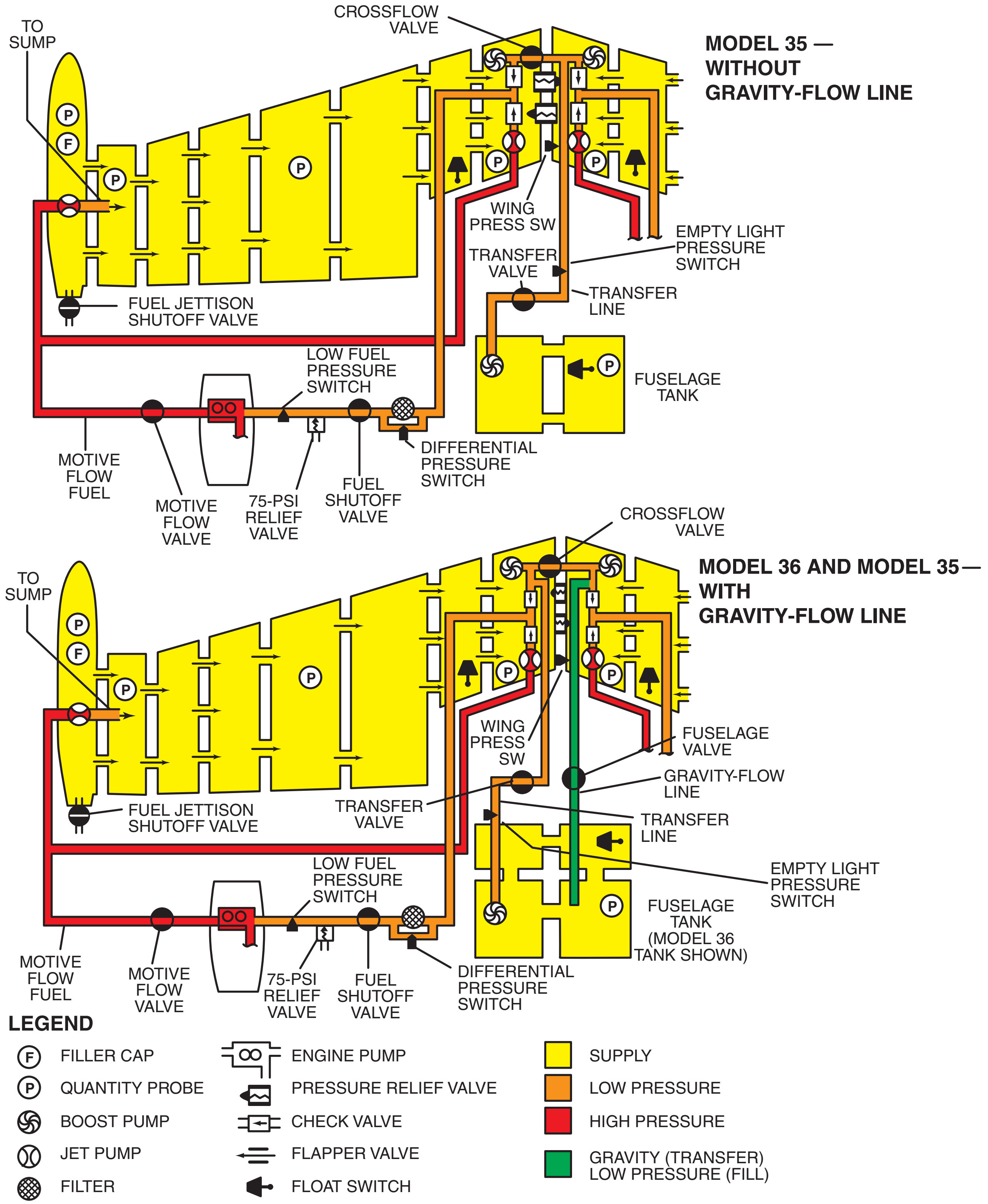


Figure 5-1. Fuel System

FUEL TANKS AND TANK VENTING SYSTEM

GENERAL

The total usable fuel capacity is approximately 6,238 pounds for the 35 model and approximately 7,440 pounds for the 36 model. Unusable (trapped) fuel is included in the airplane basic weight and is ***not*** reflected in any fuel quantity indications.

TIP TANKS

Each tip tank capacity is 1,215 pounds of usable fuel; capacity is reduced to 1,175 pounds with installation of a recognition light. The tanks are permanently attached to the wings and are positioned at 2 degrees nosedown relative to the airplane centerline. Baffles are installed to minimize slosh and prevent adverse effects on the airplane center of gravity during extreme pitch attitudes.

A jet pump installed in each tip tank transfers fuel into the wing tank. Approximately one-half of the fuel will gravity-flow through two flapper valves into the wing tank; however, any fuel at a level lower than one-half full must be transferred using the jet pump. A standpipe is installed in each jet pump transfer line to prevent fuel from being siphoned from the wing tank to the tip tank when the applicable engine is shut down.

The tip tank is vented through two vent float valves located in the forward and aft ends of the tank.

A fuel probe in each tip tank provides information to the fuel quantity indicating system.

All tip-tank fuel can be jettisoned through a valve in the tank tailcone, if required.

A filler cap on each tip tank is used to service the entire airplane fuel system.

WING TANKS

Each wing tank extends from the airplane centerline to the tip tank and holds 1,254 pounds of usable fuel. Areas which are not part of the wing fuel cell are the main landing gear wheel well, the leading edge forward of spar 1 (wing heat area), and the trailing edge between spars 7 and 8 (flap, spoiler, and aileron areas).

The 2.5-degree wing dihedral makes the inboard portions of the wing tanks the lowest areas. In each wing tank, a jet pump and an electric standby pump are located within these areas and will remain submerged in fuel until the tanks are nearly empty.

Wing tank ribs and spars act as baffles to minimize fuel shifting. Flapper valves located in the wing ribs allow unrestricted inboard flow of fuel and limit outboard flow. Two pressure-relief valves at the centerline rib equalize internal pressures between the two wing tanks. The wing tanks begin to fill through the two tip tank flapper valves as tip tank fuel increases beyond one-half full.

Three fuel probes in each wing tank provide information to the fuel quantity indicating system.

FUSELAGE TANK

The fuselage tank consists of rubber bladder fuel cells located between the aft pressure bulkhead and tailcone section. The 35 models are equipped with two fuel cells with a capacity of 1,340 pounds of usable fuel, while the 36 models are equipped with four fuel cells with a capacity of 2,542 pounds of usable fuel. Depending on the airplane, either one or two fuel lines connect the fuselage tank to the wing tanks for filling and transfer. This is explained in the Fuel Transfer Systems section.

One fuel probe provides information to the fuel quantity indicating system.

RAM-AIR VENT SYSTEM

A ram-air scoop located on the underside of each wing (Figure 5-2) supplies positive air pressure in flight to a manifold which directly vents the fuselage tank and both tip tanks. Each wing tank is indirectly vented to its own tip tank through a length of tubing, the ends of which extend to the uppermost area of each tank (Figure 5-3). The ram-air scoops, by design, do not require heating to remain ice-free. Two vent float valves are located in each tip tank, and one in the fuselage tank on 35 models. The float valves close when the fuel level reaches the vent ports, preventing fuel from entering the vent lines. A vacuum relief valve in each tip tank and the fuselage tank opens to allow air to enter the tanks should vacuum conditions occur. Each tip tank has two pressure relief valves which protect the tanks from excessive pressure. The pressure relief valves are set at 1.0 and 1.5 psi, the second valve providing a backup in case the first valve fails.

Thermal expansion of fuselage fuel in 35 models is accounted for by an open-ended vent line which bypasses the vent float valve (36 models use three open-ended vent lines) to vent pressures overboard through the ram-air scoops. A sump, installed in the vent manifold, located at the bottom center fuselage just aft of the main landing gear, collects any fuel which might enter the vent lines. A vent drain valve permits draining of the sump to ensure that the vent line to the fuselage tank is unobstructed.

FUEL INDICATING SYSTEMS

FUEL QUANTITY INDICATING SYSTEM/LOW FUEL WARNING

The fuel quantity indicating system includes an indicator and tank selector switch located on the fuel control panel (Figure 5-4). A red LOW FUEL warning light (Annunciator Section) illuminates when either wing tank fuel level is low.

The fuel quantity indicating system uses DC power from the right essential bus through the FUEL QTY circuit breaker. The six-position rotary selector switch enables the pilot to check the fuel quantity in each of the five tanks and the airplane total fuel quantity. The fuel quantity for the position selected is read on the fuel quantity indicator. The quantities printed beside each selector switch position indicate usable fuel capacities in pounds.

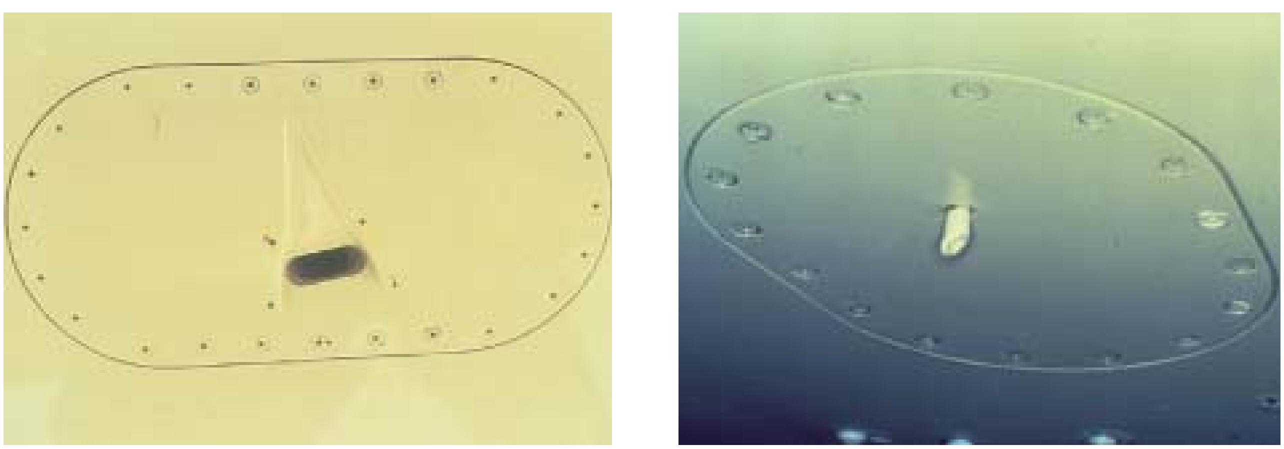


Figure 5-2. Ram-Air Scoop and Overboard Drain

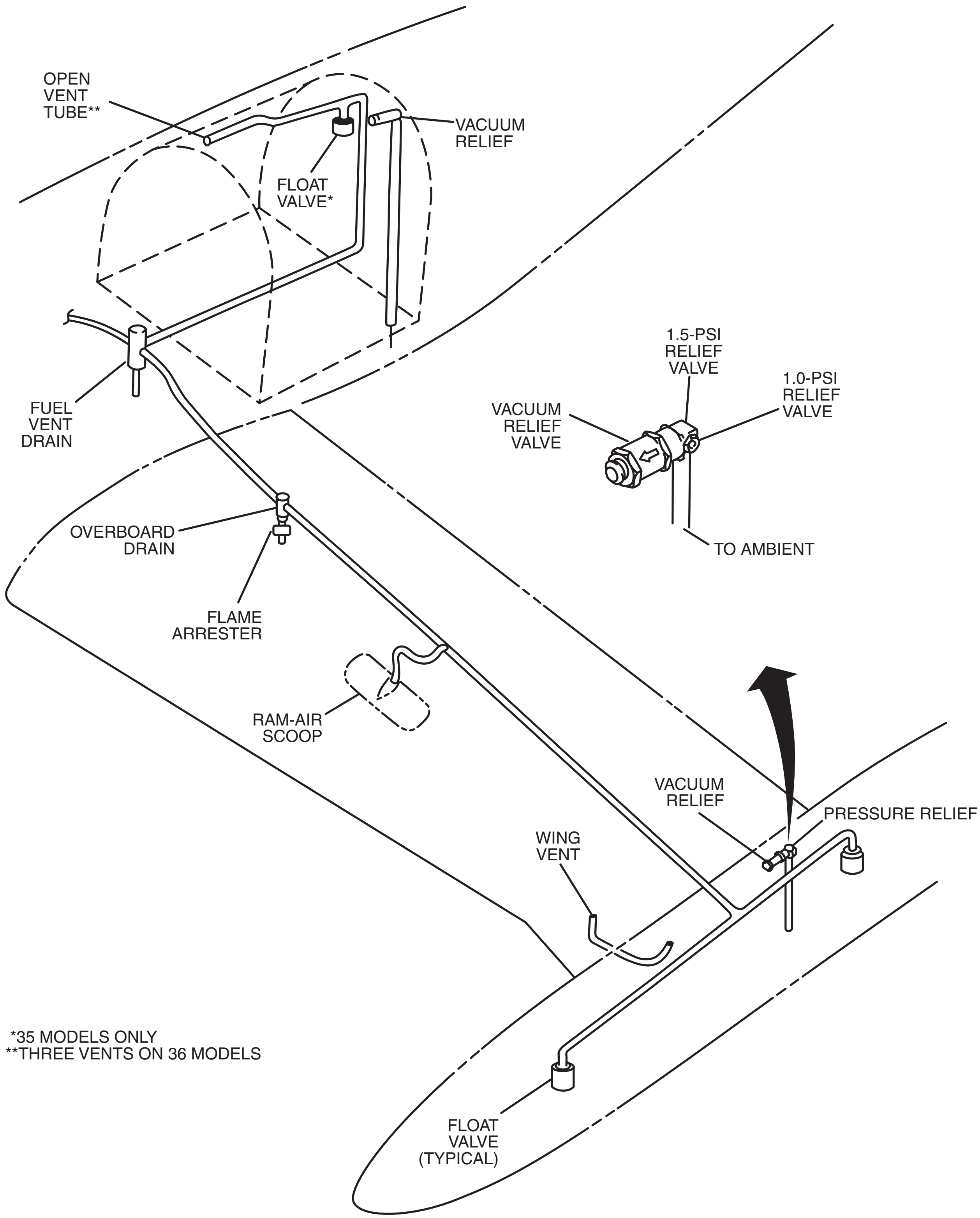
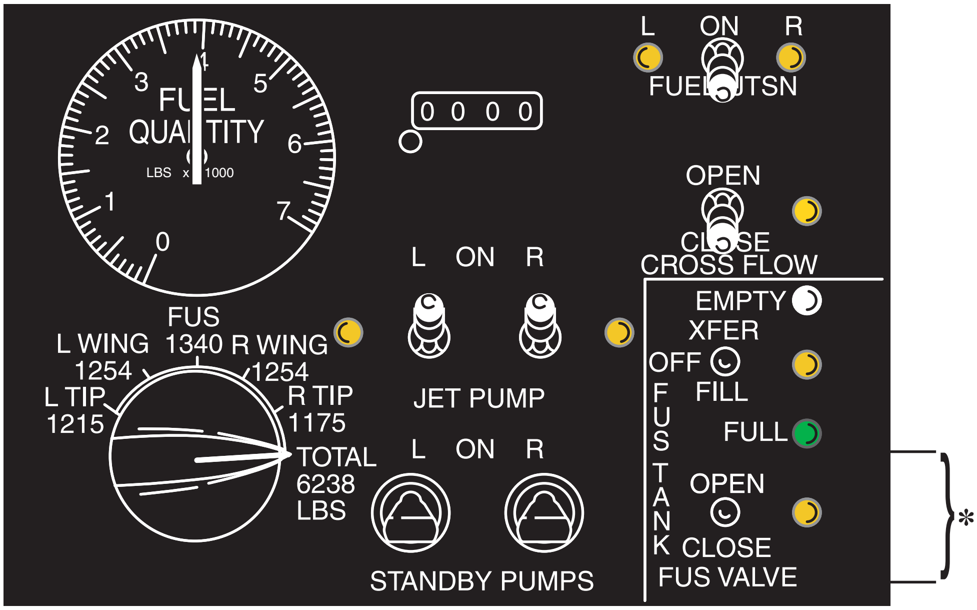
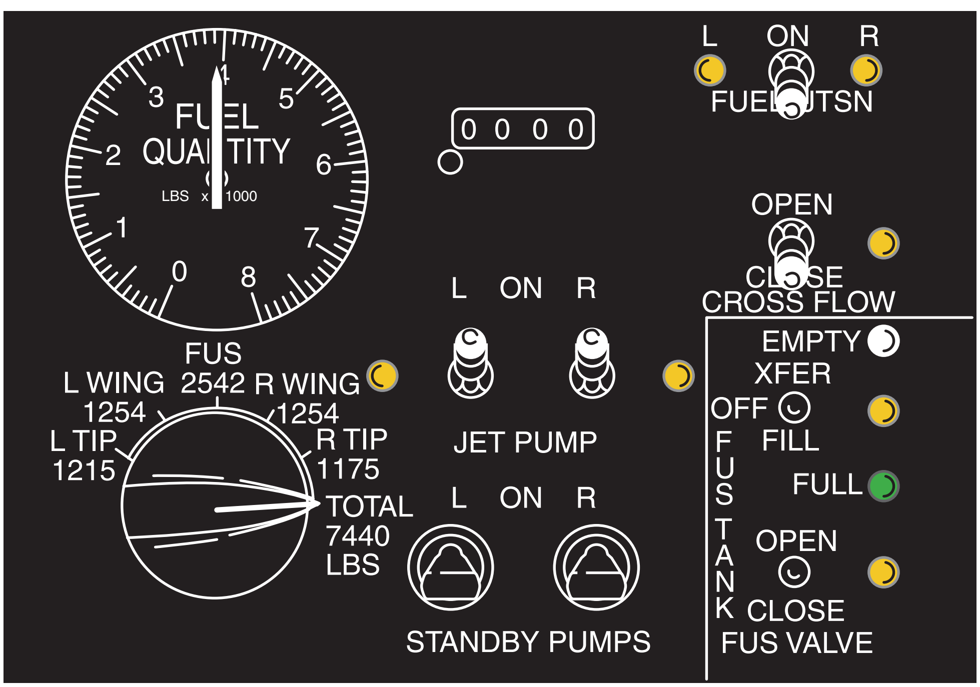


Figure 5-3. Fuel Vent System



MODEL 35

*OPTIONAL ON SNs 35-299 THROUGH 35-596.
STANDARD ON SNs 35-597 AND SUBSEQUENT.



MODEL 36

Figure 5-4. Fuel Control Panels

There are nine capacitance fuel probes. One fuel probe is located in each tip tank and in the fuselage tank. Each wing tank has three probes wired in parallel. The inboard probe in the left wing contains a temperature compensator which adjusts quantity readings for all switch selections for fuel density change due to temperature.

If the compensator probe is uncovered, erroneous fuel quantity indications could be encountered at all switch positions.

Each probe uses an electrical capacitance measuring system to sense the fuel level. It then transmits an electrical signal to the cockpit indicator where it is read as pounds x 1,000 on the dial.

Each wing tank has fuel low-level float switch. When ***either*** wing tank fuel level reaches 400 to 500 pounds remaining, the respective float switch actuates the red LOW FUEL light on the annunciator panel to indicate low wing fuel quantity (Annunciator Section). When flying with the LOW FUEL light on, limit pitch attitude and thrust to the minimum required.

FUEL FLOW INDICATING SYSTEM

A single fuel flow indicator, with two pointers (L and R) provides a readout of pounds of fuel flow per hour (Figure 5-5). A fuel counter (Figure 5-4) located on the fuel control panel provides a four-digit readout in pounds of fuel consumed by both engines. It should be reset to zero using the reset button adjacent to the counter before starting the first engine. Both indicators are powered from the battery charging bus through a 10-amp current limiter.

FUEL DISTRIBUTION

GENERAL

Each engine is supplied with fuel from its respective wing fuel system; there is no crossfeed capability. Either the wing standby pumps or the wing jet pumps supply fuel under

Figure 5-5. Fuel Flow Indicator

pressure to the engine-driven pumps. During engine start, the respective wing standby pump is automatically energized when the GEN–START switch is placed in the START position. When turbine speed (N_2) reaches 45% or 50%, or when the START switch is moved to OFF or GEN (computer off starts), the wing standby pump is deenergized and the wing jet pump then provides fuel to the engine. The wing jet pumps and standby pumps have check valves on the output side to prevent reverse flow when they are inactive.

BOOST PUMPS

Submerged DC-powered boost pumps are installed at three different locations—one standby pump in each wing adjacent to the jet pump, and one transfer pump in the fuselage tank.

The standby pumps are used:

- For engine start (automatically energized with starter switch activation)
- As a backup for the wing jet pumps
- For wing-to-wing crossflow
- For filling the fuselage tank (automatically energized with the XFER–FILL switch in the FILL position)

Both standby pumps are deactivated when the XFER–FILL switch is in the XFER position.

The transfer pump is used to transfer fuselage tank fuel to the wing tanks.

The standby pumps are powered by the respective L or R STBY PMP circuit breakers on left and right essential buses; the fuselage pump, from the FUSLG PMP circuit breaker on the right main bus.

MOTIVE-FLOW FUEL AND JET PUMPS

High-pressure fuel from the engine-driven fuel pumps is the source of motive-flow fuel to operate the jet pumps. The fuel is routed through the motive-flow valves to the jet pumps, where it passes through a small orifice into a venturi. The low pressure created in the venturi draws fuel from the respective tank, resulting in a low-pressure, high-volume output from the jet pump (Figure 5-6).

Motive-flow pressure varies with engine rpm and is regulated to 300 psi maximum. Consequently, jet pump discharge pressure also varies with engine rpm. At idle, discharge pressure is approximately 10 psi, while at full-power settings, discharge pressure is approximately 12 psi.

There are four jet pumps—one in each wing tank adjacent to the standby pump, and one in each tip tank. The wing tank jet pumps draw fuel from the wing tanks and supply low-pressure fuel to the engine-driven, high-pressure fuel pumps. Wing jet pump output can be supplemented by the wing standby pump to ensure positive pressure to an engine. The tip tank jet pumps draw fuel from the tip tanks and deliver it directly to the cavities where the standby pumps and jet pumps are located.

Jet pumps require no electrical power and have no moving parts. They are controlled by two jet pump switches (Figure 5-4) which electrically open and close the motive-flow

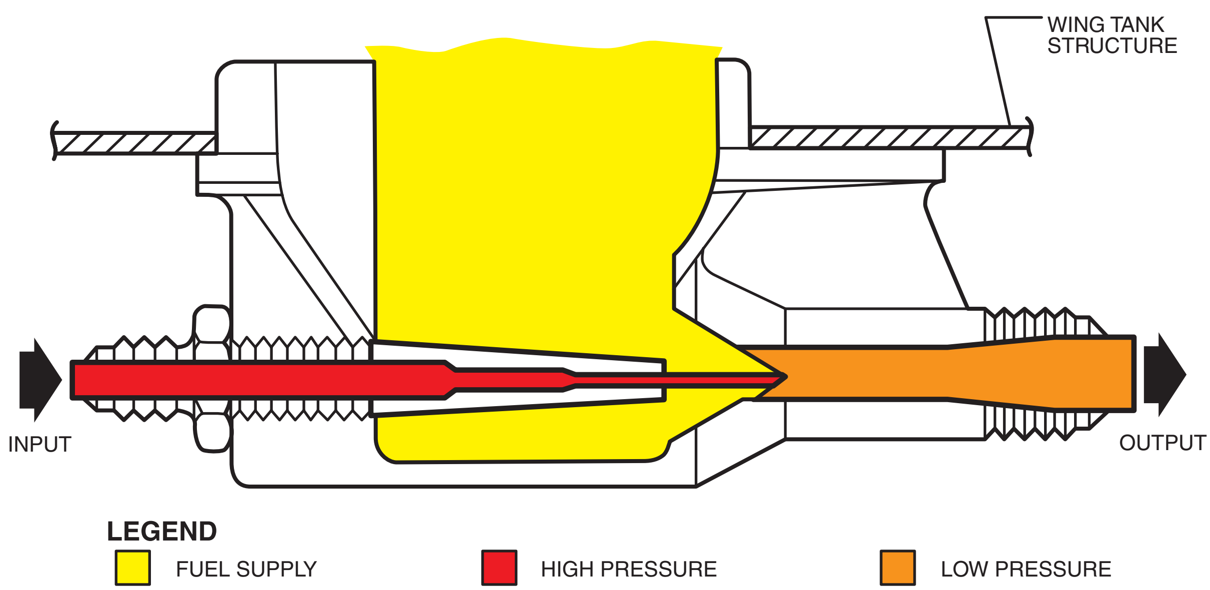


Figure 5-6. Jet Pump Schematic

valves. Power is provided by the respective L or R JET PUMP VAL circuit breaker on the left and right essential buses. The amber indicator lights next to the switches illuminate when the motive-flow valves are in transit or are not in the position selected on the switch. Each jet pump switch (and motive-flow valve) controls both jet pumps (wing and tip) on that side.

FILTERS

A fuel filter is installed in each engine feed line to filter the fuel before it enters the engine-driven fuel pump. Should the filters become clogged, the fuel is allowed to bypass them. A differential pressure switch installed in each filter assembly illuminates the ***one*** amber FUEL FILTER annunciator light if either or both filters are bypassing fuel (Annunciator Panel section).

MAIN FUEL SHUTOFF VALVES (FIREWALL)

The fuel shutoff valves are powered from the essential buses through the L and R FW SOV circuit breakers and are controlled by the FIRE handles on the glareshield. Pulling either FIRE handle closes the associated valve, and pushing the FIRE handle in opens the valve. The valves will remain in their last positions should DC power fail.

LOW FUEL PRESSURE WARNING LIGHTS

A low fuel pressure switch is located between the fuel shutoff valve and the engine-driven fuel pump in each engine feed line. The switches cause illumination of the appropriate red L or R FUEL PRESS annunciator light when fuel pressure drops below 0.25 psi. The light extinguishes when pressure increases above 1.0 psi. Illumination of a FUEL PRESS warning light is an indication of loss of fuel pressure to the engine. The probable cause is failure of the affected wing jet pump.

The engine-driven pump is capable of suction-feeding enough fuel to sustain engine operation without either the wing standby pump or jet pump. However, 25,000 feet is the highest altitude at which continuous operation should be attempted in this event.

PRESSURE-RELIEF VALVES

A 75-psi relief valve is installed in each main fuel line on the engine side of the main shutoff valve. The valves relieve pressure buildup caused by thermal expansion of trapped fuel when the engines are shut down by venting fuel overboard.

FUEL DRAIN VALVES

Drain valves are located at low points throughout the fuel system for draining condensation or sediment. A small amount of fuel should be drained from each valve during the exterior preflight inspection. The valves, spring-loaded to the closed position, are located as follows: one for each tip tank sump, one for the crossflow line, one for each wing sump, one for each engine line, one for each fuel filter, one (or two) for the fuselage tank line(s), and one (or two) for the fuselage tank sump(s) (Figure 5-7).

There is one drain valve located at the fuel vent sump. This valve ***must*** be completely drained during the exterior preflight inspection to prevent possible blockage of the fuselage ram-air vent line.

FUEL TRANSFER SYSTEMS

CROSSFLOW SYSTEM

A DC motor-driven valve is installed in the crossflow manifold connecting the two wing tanks (Figure 5-1). It is opened during fuselage fuel transfer and filling operations, and for wing-to-wing fuel balancing. The valve is

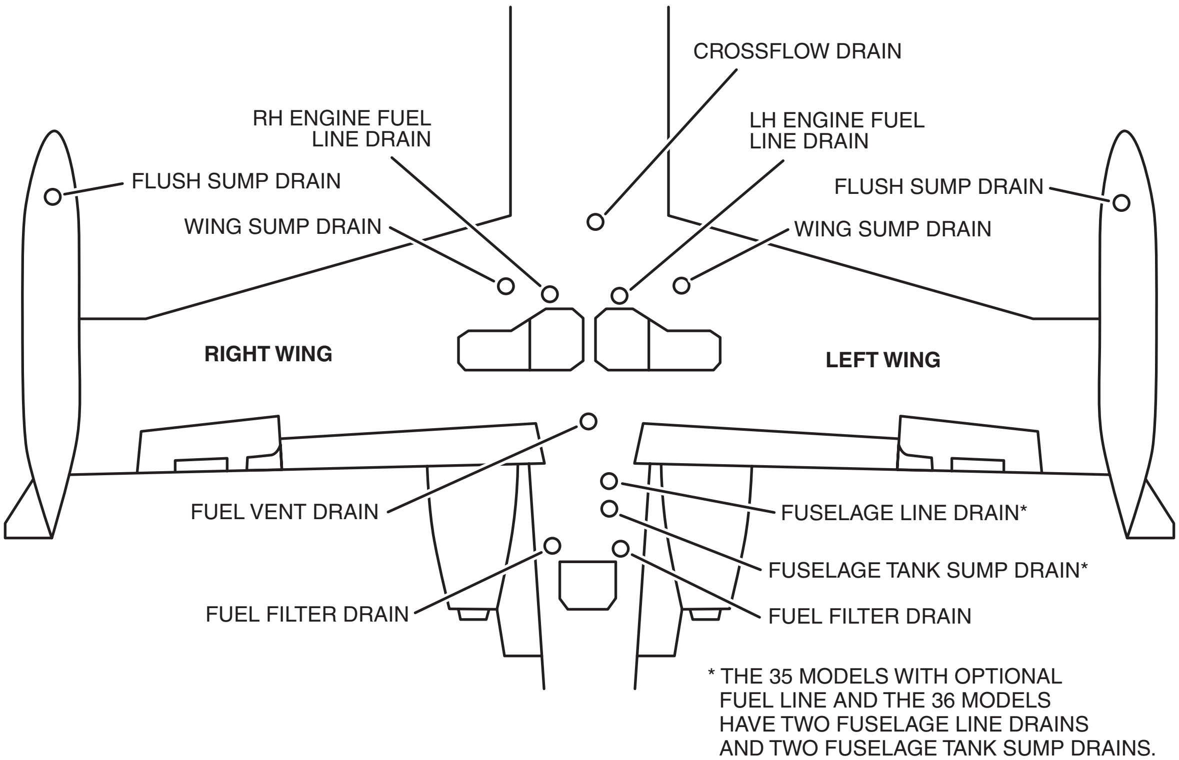


Figure 5-7. Fuel Drain Locations

controlled by the CROSS FLOW switch or the XFER–FILL switch on the fuel control panel (Figure 5-4) and is powered through the right main bus FILL & XFER circuit breaker. Additionally, on those airplanes equipped with the gravity-flow transfer line, the valve is controlled by the FUS VALVE switch which is powered from the left essential bus FUS VALVE (or FUSE VAL) circuit breaker.

The amber light (Figure 5-4) adjacent to the CROSS FLOW switch illuminates when the valve is in transit or is not in the position selected. A green or amber FUEL XFLO annunciator light (Annunciator Panel section) on the glareshield illuminates continuously whenever the crossflow valve is fully open.

If wing fuel imbalance occurs, as in single-engine operation, crossflow is accomplished by opening the crossflow valve and turning on the standby pump in the heavy wing, while ensuring that the opposite standby pump is off. The transfer rate is approximately 50 pounds of fuel per minute.

With both engines operating, opening the crossflow valve to balance fuel should not be attempted when a red FUEL PRESS light is illuminated unless it can be accomplished below 25,000 feet. To do so would divert pressure from the affected engine-driven pump to the crossflow line. Instead, asymmetric power settings may be used to balance fuel, if necessary. The above considerations do not apply to single-engine operations, and normal crossflow operations may be performed as usual.

NORMAL TRANSFER SYSTEM

The Learjet models 35/36 each have a fuel transfer line connecting the fuselage tank transfer pump with the crossflow manifold (Figure 5-1). A DC motor-driven transfer valve installed in the line controls fuel movement between the fuselage and wing tanks. The valve is controlled by the XFER-FILL switch located on the fuel control panel. When the switch is positioned from OFF to XFER, the transfer and crossflow valves are sequenced open and the transfer pump is energized automatically, while both standby pumps are deactivated. When the switch is positioned from OFF to FILL, the transfer and crossflow valves are sequenced open, and both standby pumps are energized automatically. When the switch is positioned from either XFER or FILL to the OFF position, the transfer pump or standby pumps (whichever the case may be) are deenergized and the transfer and crossflow valves are sequenced closed. The amber light adjacent to the XFER–FILL switch illuminates when the valve is in transit or is not in the position selected (Figure 5-4). The valve is powered through the right main bus FILL & XFER circuit breaker.

On 35 models without the optional gravity-flow line, the ***transfer*** line is connected to the ***right*** side of the crossflow valve. On all 36 models, and 35 models with the optional gravity-flow line, the ***transfer*** line is connected to the ***left*** side of the crossflow valve.

GRAVITY-FLOW TRANSFER SYSTEM

As an option of SNs 35-299 through 35-596, and as standard equipment on 35-597 and subsequent, and on all 36 models, a DC motor-driven fuselage valve is installed in a second fuel line, connecting the fuselage tank with the crossflow manifold on the ***right*** side of the crossflow valve (Figure 5-1). The valve is controlled by the FUS VALVE switch on the fuel control panel. When the FUS VALVE switch is positioned to OPEN, both the fuselage valve and the crossflow valve simultaneously open, allowing fuel to gravity-flow from the fuselage tank to both wings. When fuselage fuel is transferred in this manner, 162 pounds of fuel remain in the fuselage tank. The fuselage valve is also controlled by the XFER–FILL switch. When placed to FILL, the transfer valve, fuselage valve, and crossflow valve are sequenced open, and the standby pumps are energized to pump wing tank fuel through both fuel lines into the fuselage tank. The fuselage valve remains closed when the XFER–FILL switch is positioned to XFER. The amber light adjacent to the FUS VALVE switch will illuminate when the fuselage valve is in transit or is not in the position selected (Figure 5-4). If either standby pump switch is on, the FUS VALVE switch is rendered inoperative, and neither the fuselage valve nor the crossflow valve will open if the FUS VALVE switch is moved to OPEN. Conversely, if the FUS VALVE switch is already in the OPEN position (fuselage valve and crossflow valve open), turning either standby pump switch on will automatically cause the fuselage valve and crossflow valves to sequence closed. The fuselage valve is powered through the left essential bus FUSE VAL (or FUS VALVE) circuit breaker.

FLOAT AND PRESSURE SWITCHES

Fuselage Fuel Tank Float Switch

When filling the fuselage tank, a float switch mounted inside the tank actuates when the tank is full. The switch:

- Illuminates the green FULL light on the fuel control panel
- Deenergizes the standby pumps
- Closes the transfer and crossflow valves
- Closes the fuselage valve (all airplanes equipped with the gravity-flow transfer line)

The green FULL light on the fuel control panel (Figure 5-4) remains illuminated until the XFER–FILL switch is turned off.

Fuselage Tank Low-Pressure Switch

The fuselage tank low-pressure switch is installed in the fuselage transfer line to alert the pilot when fuselage fuel is depleted. With the XFER–FILL switch in the XFER position, the switch senses low pressure in the line and illuminates the white EMPTY light on the fuel control panel (Figure 5-4) when either of two conditions exists:

- The tank is empty.
- The fuselage transfer pump fails.

The switch actuates when pressure drops below 2.75 psi and resets at 3.75 psi as pressure increases.

Wing Fuel Pressure Switch

A wing fuel pressure switch is installed to prevent internal overpressurization of the wings during transfer of fuselage tank fuel. The switch, located in the right main wheel well, deenergizes the fuselage transfer pump when wing fuel pressure reaches 5 psi; the switch resets and energizes the pump again when the pressure drops below 2.5 psi.

PRESSURE-RELIEF VALVES

Two one-way pressure relief valves are located at wing rib 0.0, which separates the left and right wing fuel tanks. Each valve, relieving in the opposite direction, opens at 1 PSID to equalize fuel pressure between the wing tanks when crossflowing or transferring fuel.

FUSELAGE FUEL FILL-TRANSFER OPERATIONS

Fill Operation

Fuel may be pumped from the wings to the fuselage tank using the FILL position on the XFER–FILL switch. The FILL position may be used for CG considerations in flight; however, it is normally used only during fuel servicing.

When the XFER–FILL switch is placed to the FILL position:

- The transfer valve opens, then
- The crossflow valve opens, then
- The standby pumps are energized, and
- The fuselage tank float switch is enabled.

If the tank is to be filled to capacity, the float switch actuation automatically:

- Deactivates the standby pumps
- Closes all valves
- Illuminates the green FULL light, which will go out when the XFER–FILL switch is turned off

The filling process may be terminated at any point by turning the XFER–Fill switch off.

Transfer Operations

The ***normal*** method of transferring fuselage fuel in flight is accomplished by using the XFER position on the XFER–FILL switch. When the switch is placed in the XFER position:

1. The transfer valve opens.
2. The crossflow valve opens.
3. The fuselage transfer pump is energized.
4. The standby pumps are disabled.
5. The white EMPTY light (pressure switch) is enabled.

Gravity-flow is also possible on all airplanes through the ***normal*** transfer line, should the transfer pump fail. The amount of fuel trapped (unusable) is approximately 162 pounds. The rate of gravity transfer will, however, be slower than when using the fuselage valve, if installed.

When the XFER–FILL switch is placed in the OFF position:

- The transfer pump is deenergized, and
- Operation of the standby pumps is once again possible.
- The transfer valve closes, then
- The crossflow valve closes.

The ***alternate*** method of transferring fuselage fuel in flight is only possible on airplanes equipped with the gravity-flow line by using the OPEN position on the FUS VALVE switch. However, prior to doing so, it is essential to first assure that the XFER–FILL switch is off, and that ***both*** standby pump switches are off. Then, when the FUS VALVE switch is placed in the OPEN position, the fuselage valve and crossflow valve open simultaneously. The valves are not sequenced as they are when using the XFER–FILL switch.

When the amount of fuel in the wing tanks begins to decrease, the FUS VALVE switch may be turned off, and the transfer process may be completed using the ***normal*** transfer procedure. On airplanes with the gravity-flow line, approximately 162 pounds of fuel will be trapped (unusable) if the gravity-flow line ***only*** is used to transfer fuselage fuel.

TIP-TANK FUEL JETTISON SYSTEM

A DC motor-driven valve in the tailcone of each tip tank provides the capability of jettisoning tip-tank fuel. One FUEL JTSN switch on the fuel control panel (Figure 5-4) controls both tip-tank jettison valves. When the FUEL JTSN switch is placed to the ON position, the jettison valves open and two amber lights illuminate continuously on the fuel control panel, indicating that the valves are fully open. The jettison tubes are scarfed, which creates a low-pressure area that helps pull the fuel out of the tank(s). This, in combination with the force of gravity, enables the entire contents of both tanks to be jettisoned. Fuel jettison is faster while the airplane is in a noseup attitude. It takes approximately five minutes to jettison fuel from the tip tanks. The left- and right-hand jettison valves are protected independently by the FUEL JTSN circuit breakers located on the left and right essential buses, respectively.

FUEL SERVICING

GENERAL

Fuel servicing includes those procedures necessary for fueling and adding anti-icing additives.

Fueling is accomplished through a filler cap in the top of each tip tank. Fuel then begins to flow by gravity from the tip tanks into the wing tanks as the tip tanks reach one-half full. The wing standby pumps pump fuel to the fuselage tank when the XFER–FILL switch is set to FILL.

At normal temperatures some water is always in solution (dissolved) with fuel. At high altitudes, fuel undergoes a cold-soaking process and small amounts of water come out of solution and subsequently freeze. The anti-icing additives specified for use in the Learjet 35/36 are Hi-Flo Prist and QUELL. Either additive prevents the growth of microbiological organisms in the fuel. Fuel containing anti-icing additive conforming to MIL-I-27686 requires no further treatment.

SAFETY PRECAUTIONS

Refueling and defueling should be accomplished only in areas which permit free movement of fire equipment.

Figure 5-8 shows the airplane grounding points.

When adding anti-icing additives (Figure 5-9), follow the manufacturer's instructions for blending.

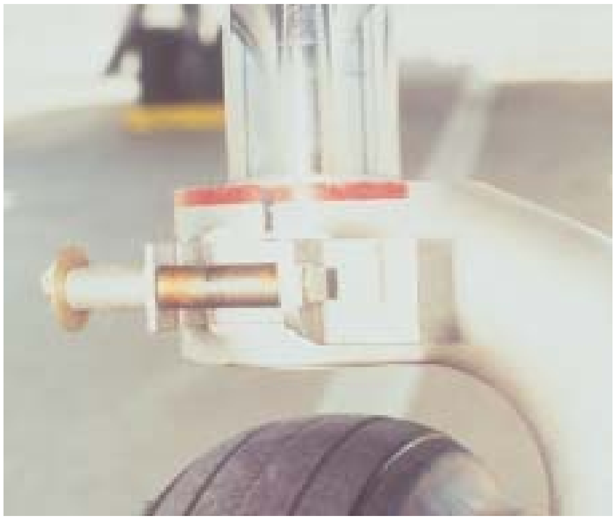


Figure 5-8. Airplane Grounding Points

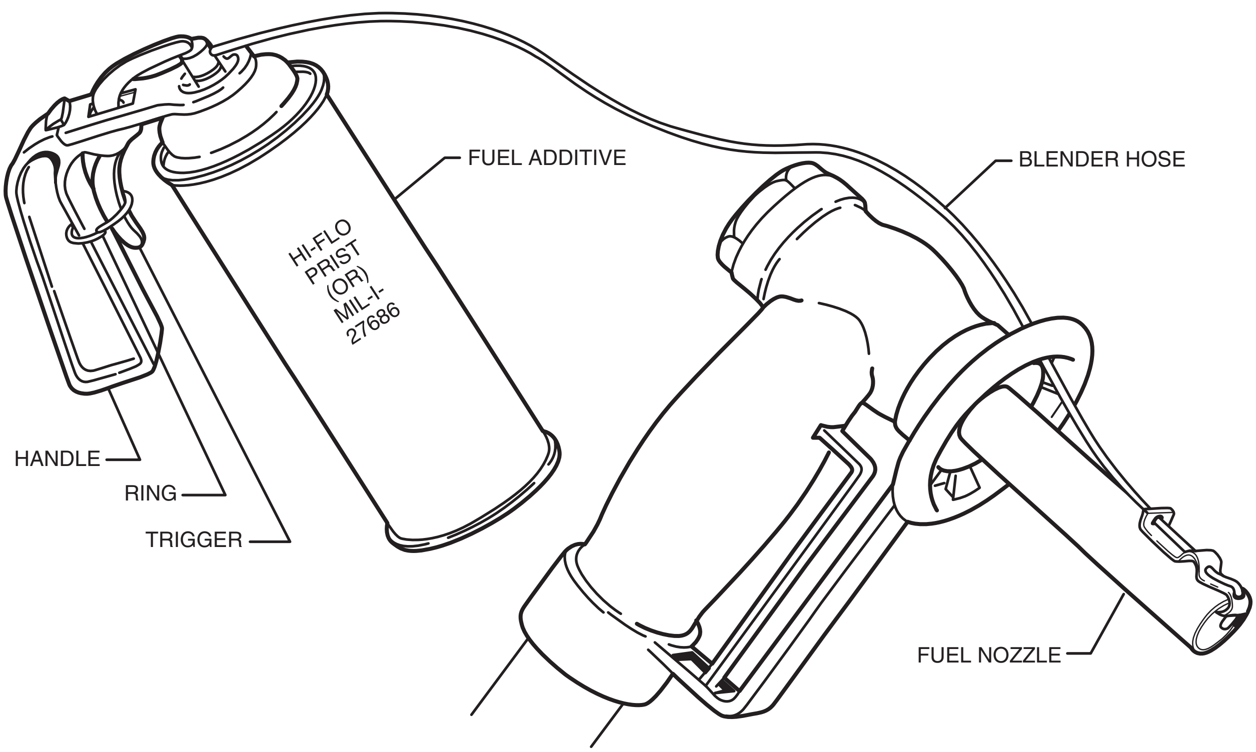


Figure 5-9. Prist Blending Apparatus

AVIATION GASOLINE

Aviation gasoline (MIL-D-5572D, Grades 80/87, 100/130, and 115/145) may be used as an emergency fuel and mixed in any proportion with the various approved jet kerosene-base fuels.

ANTI-ICING ADDITIVE

All fuels used must have an approved blended anti-icing additive. Depending upon fueling location and type of fuel, the additive may or may not be blended at the refinery. If not blended at the refinery, the additive must be blended at the time of fueling. Refer to the *AFM* for the approved MIL-Specs. Compare the MIL-Spec of the anti-icing additive to be blended with the referenced MIL-Specs in the *AFM* to determine the correct blending amounts.

REFUELING

Refueling is accomplished through the tip-tank filler caps (Figure 5-10). The fuel begins to flow by gravity into the wing tanks as the tip tanks reach one-half full. The standby pumps are used to fill the fuselage tank. (See Fuel Transfer Systems, this chapter.) A ground power unit should be used, if possible, because of the requirement to operate the standby pumps. Refer to the approved *AFM* for detailed refueling procedures.

Figure 5-10. Refueling Filler Cap

QUESTIONS

1. Trapped fuel weight:

A. Must be added to the weight of fuel taken on board when servicing the airplane
B. Is included in the airplane basic weight for airplanes certified in the U.S.
C. Must be accounted for in the fuselage tank for CG purposes
D. May be disregarded since it is less than 200 pounds

2. With the exception of the FUEL JTSN lights, all other amber lights on the fuel control panel, when illuminated steady, indicate that the respective:

A. Valves are cycling or the pumps are properly operating.
B. Valves are in the correct position; the pumps are inoperative.
C. Switch position agrees with the valve position or pump operation.
D. Valve position disagrees with the switch position.

3. The red LOW FUEL light illuminates when:

A. 350 pounds total fuel remains.
B. 250 to 350 pounds remain in either wing, depending on the airplane SN.
C. 400 to 500 pounds total fuel remains
D. 400 to 500 pounds remain in either wing

4. The standby pumps are used for all the following functions except:

A. Engine start
B. As a backup for the main jet pumps
C. Wing-to-wing crossflow with a wing tank jet pump inoperative
D. Wing-to-fuselage transfer of fuel

5. The crossflow valve opens:

A. Only when the CROSS FLOW switch is set to OPEN
B. Only when the CROSS FLOW switch is set to OPEN or the XFER–FILL switch is set to XFER
C. Anytime electrical power is lost
D. Whenever the CROSS FLOW, XFER–FILL, or FUS VALVE switches are moved from the OFF or CLOSE position

6. Steady illumination of an amber transfer valve light indicates:

A. The valve has failed to close.
B. The valve has failed open.
C. The valve has operated correctly.
D. The valve has failed to move to the position commanded by the XFER–FILL switch.

7. Illumination of the red L or R FUEL PRESS light indicates:

A. Fuel pressure to the respective engine-driven fuel pump is low.
B. Fuel pressure to the respective engine is too high for safe operation.
C. A fuel filter is bypassing.
D. Fuel pressure to the respective engine is optimum for engine start.

8. When the XFER–FILL switch is placed to the FILL position, the:

A. Float switch is disabled.
B. Wing standby pumps are disabled.
C. Fuselage valve closes.
D. Crossflow valve opens.

9. Motive-flow fuel for the jet pumps is supplied by the:

A. Engine-driven fuel pumps
B. Wing standby pumps
C. Fuselage transfer pump
D. Motive-flow control unit

10. The amber FUEL FILTER light indicates:

A. Low fuel pressure to the engine-driven pump; the standby pumps should be turned on
B. That both fuel filters are being bypassed; the light does not illuminate if only one filter is bypassed
C. That one or both fuel filters are being bypassed
D. That only the secondary fuel filters are being bypassed

11. The amount of fuel trapped in the fuselage tank after completion of gravity transfer via the fuselage valve is approximately:

A. 62 pounds
B. 162 pounds
C. 262 pounds
D. None

12. The wing fuel pressure switch:

A. Turns off the fuselage transfer pump when wing fuel pressure reaches 5 psi
B. Turns on the fuselage transfer pump when wing fuel pressure is below 5 psi
C. Turns off the wing standby pumps when wing fuel pressure reaches 5 psi
D. Turns on the wing standby pumps when wing fuel pressure is below 5 psi

13. When using any mixture of aviation gasoline:

A. Do not take off with fuel temperature lower than –54° C (–65° F).
B. Restrict flights to below 15,000 feet.
C. Both jet pumps and both standby pumps must be on and the pumps must be operating.
D. All of the above answers are correct.

14. The Learjet 35/36 requires anti-icing additive:

A. At all times
B. Only when temperatures of –37° C and below are forecast
C. Only for flights above 15,000 feet
D. Only for flights above FL 290

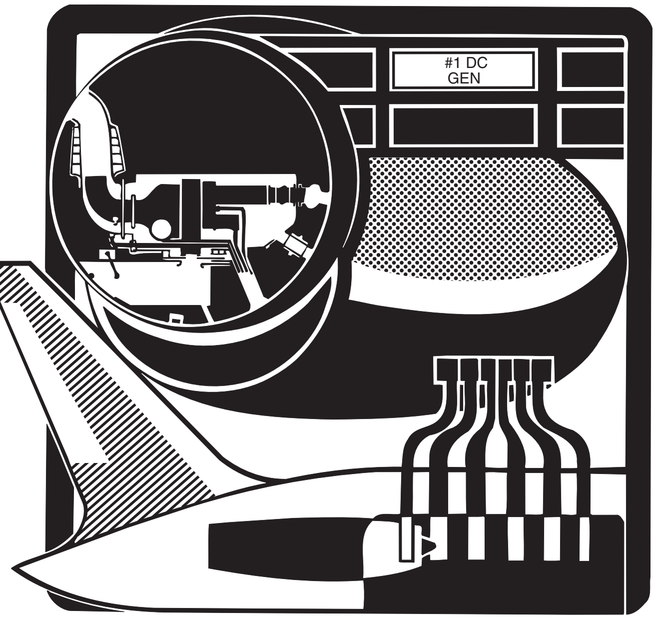
The information normally contained in this chapter is not applicable to this particular aircraft.

CHAPTER 7
POWERPLANT

CONTENTS

ILLUSTRATIONS

CHAPTER 7
POWERPLANT



INTRODUCTION

This chapter describes the powerplants installed on Learjet 35/36 series airplanes. In addition to the powerplant, the chapter describes such engine-related systems as oil, fuel, ignition, engine controls and instrumentation, engine synchronization, the Aeronca and the Dee Howard thrust reversers, and all pertinent powerplant limitations.

GENERAL

All 35/36 series airplanes are powered by two aft fuselage-mounted TFE731-2-2B turbofan engines. Optional thrust reversers are available either as a factory installation or as a retrofit.

The TFE731 series engine is manufactured by the Garrett Turbine Engine Company at Phoenix, Arizona. The engine is a lightweight, twin-spool turbofan. The fan is front mounted and gear driven.

Each engine develops 3,500 pounds of thrust, static at sea level, up to 72° F (+22°C).

The modular design concept of the engine facilitates maintenance.

MAJOR SECTIONS

For descriptive purposes, the engine (Figure 7-1) is divided into seven major sections as follows:

1. Air inlet
2. Fan
3. Compressor
4. Combustor
5. Turbine
6. Exhaust
7. Accessory

AIR INLET SECTION

The air inlet section is a specially designed, sound-reducing structure enclosing the fan and its associated planetary gear-drive. The fan shroud is armored for blade containment.

FAN SECTION

The fan section includes the single-stage axial fan, an integral spinner, and the fan planetary gear assembly which is driven by the low-pressure rotor. The rpm of the LP rotor is designated "N_1" (commonly referred to as "fan speed").

The planetary gear provides the required gear reduction for the fan. The rpm of the LP rotor (N_1) is read on the FAN SPEED indicator (Figure 7-5). Engine thrust is set using this instrument.

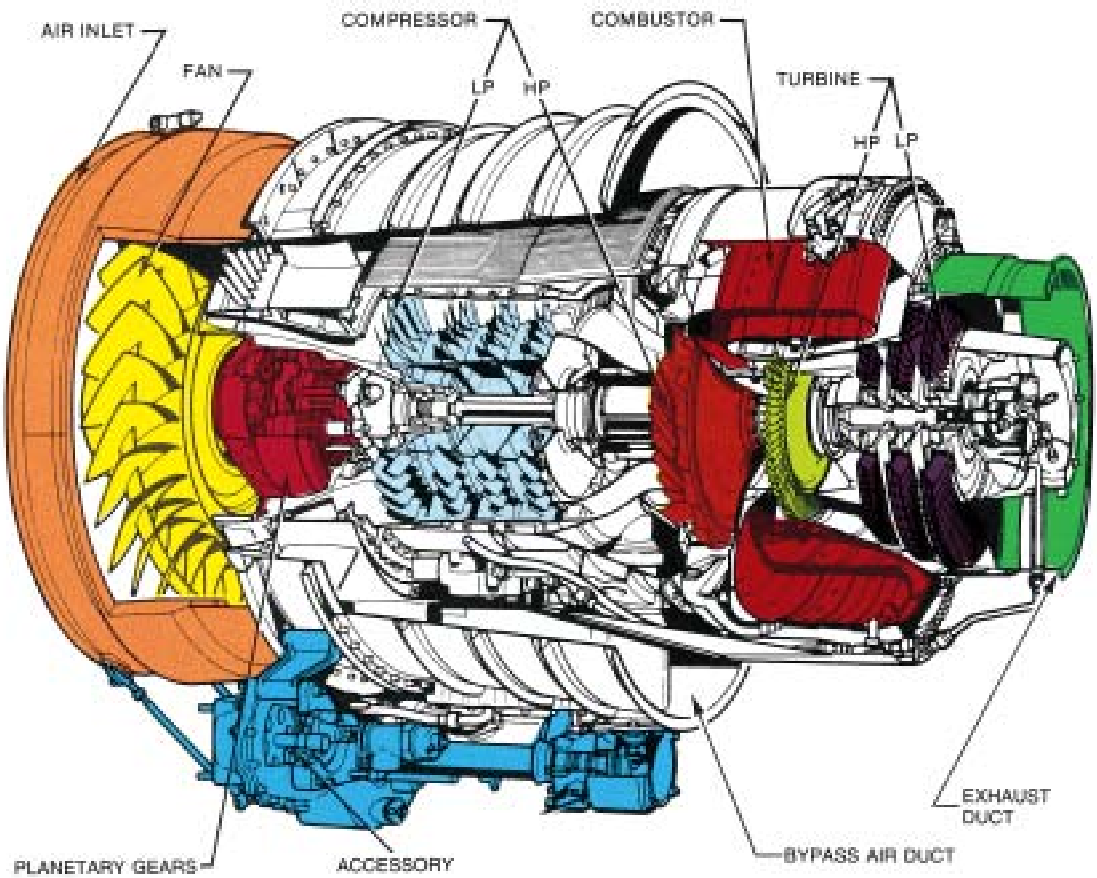


Figure 7-1. Major Sections

The fan performs two functions: (1) its outer diameter accelerates a large air mass at a relatively low velocity into the full-length bypass duct, and (2) the inner diameter of the fan accelerates a smaller air mass to the four-stage axial-flow compressor.

COMPRESSOR SECTION

The compressor section includes a low-pressure (LP) compressor and a high-pressure (HP) compressor.

The LP compressor incorporates four axial stages. Stall-surge protection is provided for the LP compressor by an automatically controlled surge bleed valve.

The HP compressor consists of a single-stage centrifugal impeller driven by the HP turbine.

COMBUSTOR SECTION

The combustor section includes an annular reverse-flow combustion chamber enclosed in a plenum. (Two 180° directional changes in airflow take place through the combustor section.)

Twelve duplex fuel atomizers (spray nozzles) and two igniter plugs are located in the combustion chamber.

TURBINE SECTION

The turbine section, consisting of a single-stage axial HP turbine and a three-stage axial LP turbine, is located in the path of the exhausting combustion air.

The single-stage HP turbine, rigidly joined with the HP compressor, forms the HP spool which rotates independently about the LP rotor shaft. The rpm of the HP spool is designated N_2 (commonly referred to as turbine speed). The rpm of the turbine (N_2 rpm) is read on the TURBINE SPEED indicator (Figure 7-5). This is a supporting engine operation instrument.

The three-stage LP turbine assembly is rigidly connected to the LP compressor assembly by a common shaft, forming the LP rotor. The forward end of the rotor shaft is geared to the planetary gear assembly which drives the fan.

EXHAUST SECTION

The exhaust section consists of the primary and bypass air exhaust ducts. The primary exhaust section directs the combustion gases to the atmosphere. The bypass air exhaust directs the fan bypass air to the atmosphere.

ACCESSORY SECTION

The accessory section consists of a transfer gearbox and an accessory drive gearbox located on the lower forward side of the engine. The transfer gearbox is driven by a tower shaft and bevel gear from the HP spool. A horizontal drive shaft interconnects the transfer gearbox to the accessory drive gearbox to drive the following accessories:

- Oil pump
- Fuel pump and mechanical governor within the fuel control unit (FCU).
- Hydraulic pump
- DC generator

In addition to these accessories, a DC starter motor is mounted on the accessory drive gearbox to turn the HP spool for engine starting.

OPERATING PRINCIPLES

The fan (Figure 7-2) draws air through the engine nacelle air inlet. The outer diameter of the fan accelerates a moderately large air mass through the fan bypass duct to provide direct thrust. The inner diameter of the fan accelerates a smaller air mass into the LP compressor.

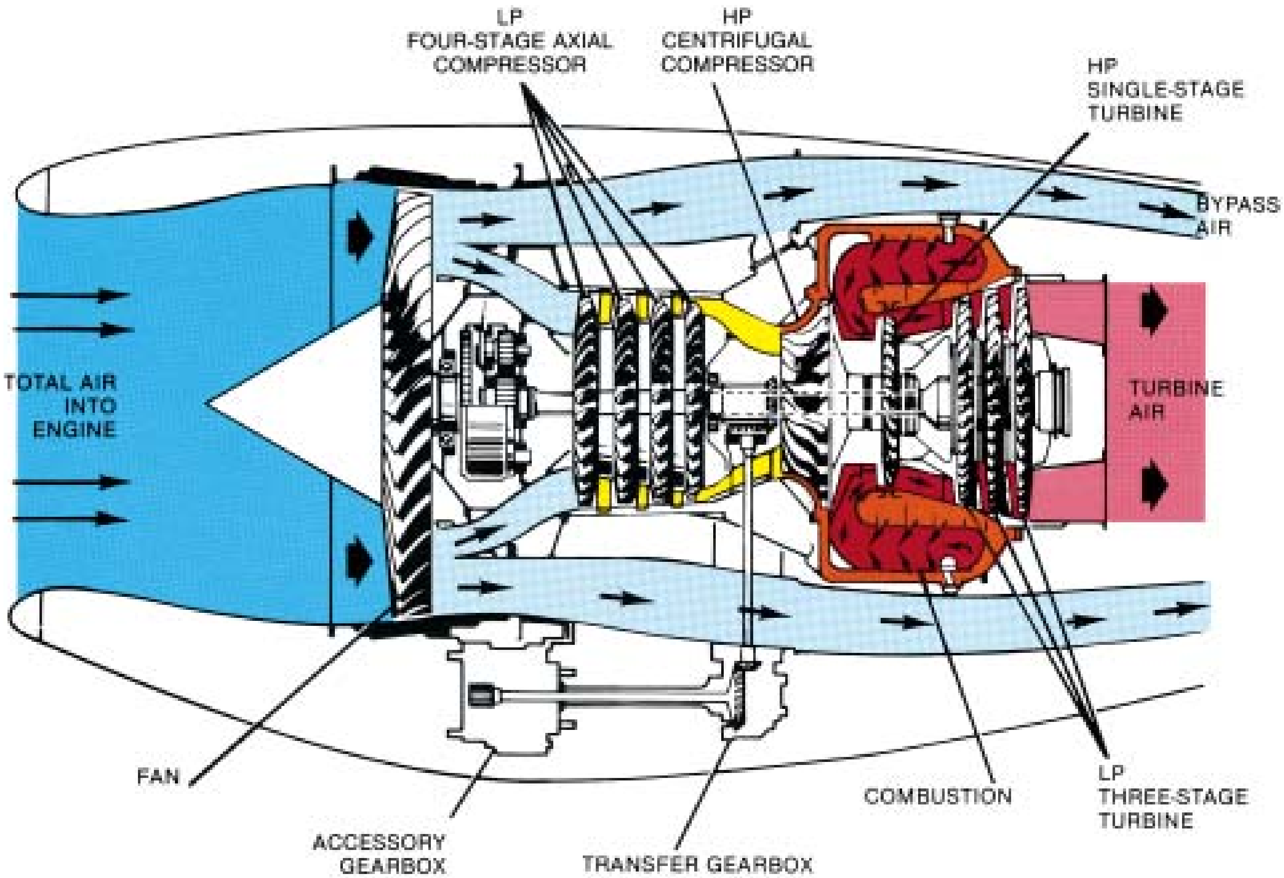


Figure 7-2. Airflow Diagram

Air is progressively compressed as it passes through the LP compressor, then to the HP compressor where a substantial increase in pressure results. Air leaving the HP compressor is forced through a transition duct into a plenum chamber surrounding the combustor. The compressed air is allowed to enter the combustor through holes and louvers designed to direct the flow of combustion air and to keep the flame pattern centered within the combustor. Each of the duplex fuel nozzles sprays fuel in two distinct patterns, resulting in efficient, controlled combustion. The mixture is initially ignited by the two igniter plugs. The expanding combustion gases, generating extremely high pressures, are directed to the HP turbine which extracts energy to drive the integral HP compressor and the accessory section through the tower shaft. The combustion gases continue to expand through the three-stage LP turbine which extracts energy to drive the LP compressor through the LP rotor shaft and the fan through the planetary gear. The combustion gases are then exhausted through the exhaust duct. The resulting thrust created by the combustion air adds to the thrust generated by the fan through the bypass air duct to produce the total propulsion force. At sea level, the fan contributes 60% of the total rated thrust, diminishing as altitude increases. At 40,000 feet, the fan contributes approximately 40% of the total thrust. Engine core rotation (looking forward) is clockwise, and fan rotation is counterclockwise.

OIL SYSTEM

GENERAL

The oil system provides cooling and lubrication of the engine main bearings, the planetary gear, and the accessory drive gear.

Oil is contained in a tank on the right side of the engine. Access for servicing and level checking (Figure 7-3) is located on the outboard side of each nacelle.

The engine-driven oil pump incorporates one pressure element, four scavenge elements, and a pressure regulator (Figure 7-6).

The pressure element draws oil from the tank and provides pressure lubrication for all bearings and gears. The scavenge elements return oil to the tank.

A bypass oil filter removes solids from the oil. A red pop-out ΔP indicator provides visual indication of a clogged filter. It can be checked through a spring port (Figure 7-4) on the right side of each engine nacelle. The indicator button should be flush with the housing; if it is not, maintenance is required before flight.

Figure 7-4. ΔP Indicator

LEFT ENGINE ACCESS **RIGHT ENGINE ACCESS**

Figure 7-3. Oil Servicing Access

Oil cooling is fully automatic and is achieved by a combination of sectional air-oil coolers in the fan bypass duct and a fuel-oil cooler mounted on the engine. Temperature and pressure bypass protection is provided for the oil coolers.

Oil venting is provided and controlled by an altitude compensating breather-pressurizing valve.

INDICATION

Oil pressure is displayed on a single indicator with dual (L-R) needles (Figure 7-5) on the engine instrument panel which require 26-VAC electrical power from the L and R OIL PRESS circuit breakers located on their respective L and R 26-VAC bus.

A single red LO OIL PRESS light on the annunciator panel provides warning of low oil pressure (Annunciator Panel). An optional installation provides for dual lights labeled "L LO OIL" and "R LO OIL," usually located outboard of either engine FIRE handle. The light(s) is illuminated by a pressure switch on each engine when pressure drops to 23 psi.

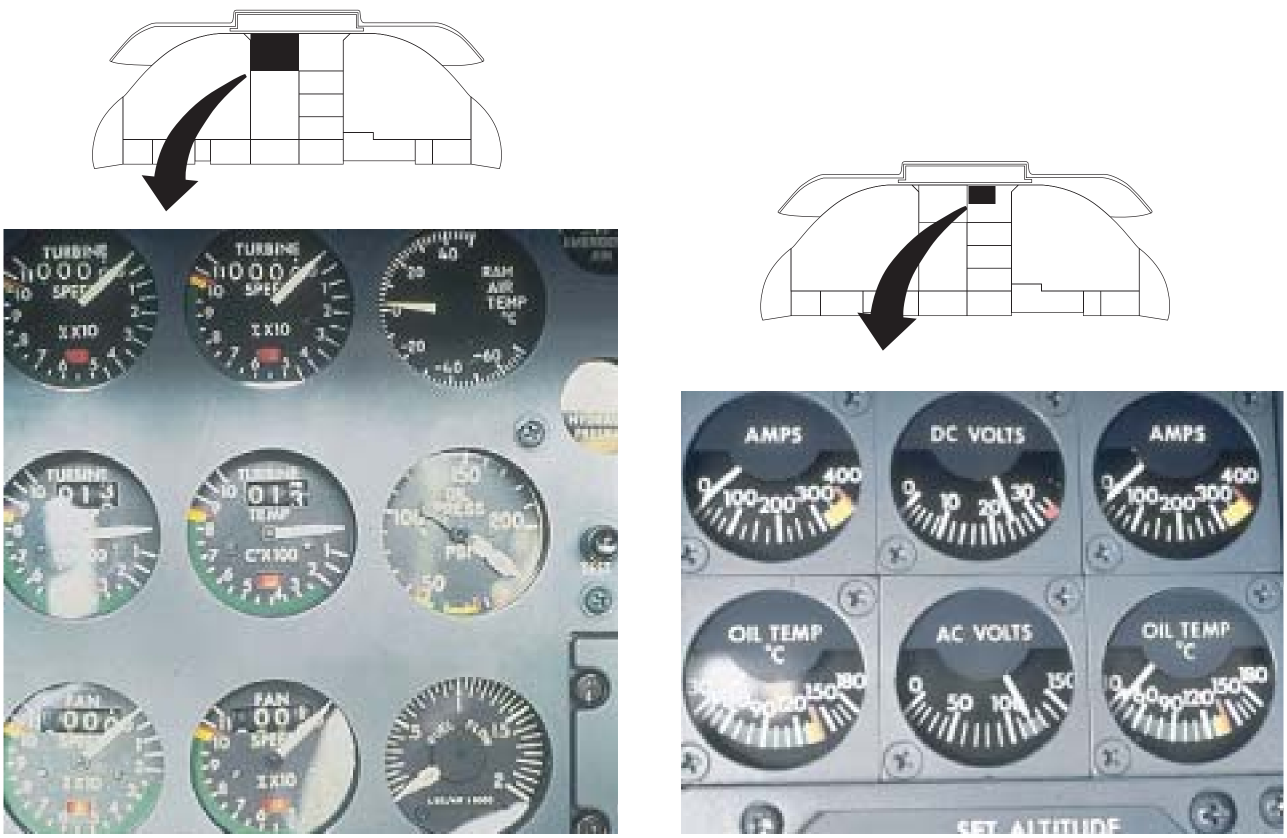


Figure 7-5. Engine Instruments

With the single LO OIL PRESS light installation, the light is wired in parallel from the pressure switch on each engine. When this light illuminates, the affected engine must be determined by checking the oil pressure indicator.

Oil temperature is displayed on individual gages (Figure 7-5) on the upper right side of the engine instrument panel. Power for these gages is supplied through the OIL TEMP circuit breaker located on the right essential bus.

A chip detector is installed in the scavenge return line. It is used by maintenance to check for the presence of ferrous particles in the oil. As optional equipment, the detectors may be connected to amber LH and RH ENG CHIP lights installed on the glareshield just to the right of the right-hand engine FIRE handles (Annunciator Panel).

OPERATION

Figure 7-6 illustrates operation of the engine oil system.

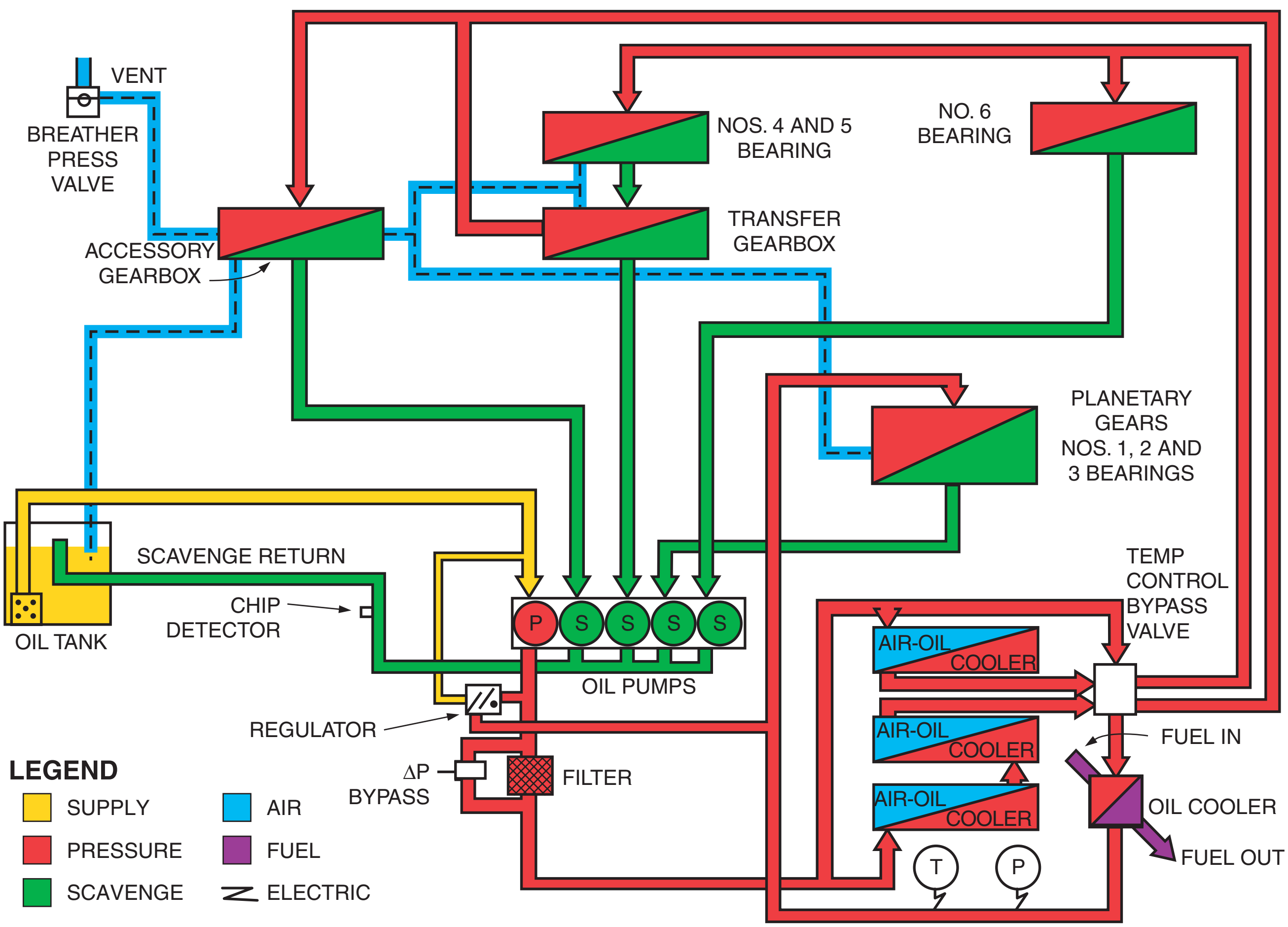


Figure 7-6. Oil System Schematic

FUEL SYSTEM

GENERAL

The engine fuel system provides for fuel scheduling during engine starting and acceleration to idle, operational acceleration and deceleration, and steady-state operation throughout the entire operating envelope of the airplane.

FUEL PRESSURE

Engine fuel pressure is generated by a two-stage engine-driven pump. The centrifugal LP stage increases inlet fuel pressure from the airplane fuel system and directs fuel through a bypassable fuel filter with a ΔP indicator button to the HP stage. The HP pump increases fuel pressure to the valve required for efficient operation of the fuel control unit (FCU). In addition, the HP fuel pump supplies the motive-flow fuel for operation of the fuel tank jet pumps. (See Chapter 5, “Fuel System.”)

MOTIVE-FLOW LOCKOUT VALVE AND PRESSURE REGULATOR

The lockout valve remains closed initially during engine start to ensure sufficient pressure to the FCU. The valve gradually opens fully as fuel pressure increases during the start. On earlier airplanes the motive-flow shutoff valve is also closed when the START-GEN switch is moved to START. A pressure regulator maintains motive-flow line pressure for efficient jet pump operation.

FUEL CONTROL UNIT (FCU)

The FCU schedules fuel flow to the fuel nozzles. Its primary mode of operation is the ***automatic mode*** (fuel computer on). In automatic the FCU responds to electrical signals from the fuel computer. The secondary mode of operation is the ***manual mode*** (fuel computer off or failed). In manual the FCU responds mechanically to thrust lever movement. The FCU includes (Figure 7-11):

- A mechanical fuel shutoff valve, operated by thrust lever movement between CUT-OFF and IDLE
- A DC potentiometer, mechanically positioned by thrust lever movement, which electrically transmits this as power lever angle (PLA) to the computer for automatic operation
- A manual mode solenoid valve which is normally energized open by the fuel computer for automatic mode operation. It is deenergized closed for manual mode operation.
- A DC torque motor which schedules fuel flow in automatic mode in response to electrical signals from the computer.
- A mechanical flyweight governor, driven by the engine fuel pump to (1) limit engine overspeed to 105% N_2 in the automatic mode and (2) govern engine rpm relative to thrust lever position in the manual mode.
- A pneumatically controlled metering valve, which (1) restricts fuel flow in the event of engine overspeed and (2) schedules fuel flow in manual mode.
- Pneumatic circuits to channel and control P_3 bleed-air pressure to pneumatically position the metering valve.
- An ultimate overspeed solenoid valve energized by the fuel computer at 109% N_1 or 110% N_2 to shut off fuel.

ELECTRONIC FUEL COMPUTER

General

Two electronic fuel computers are located in the tailcone area (Figure 7-7). They operate on DC power from the L and R FUEL CMPTR circuit breakers on the left and right essential buses, respectively.

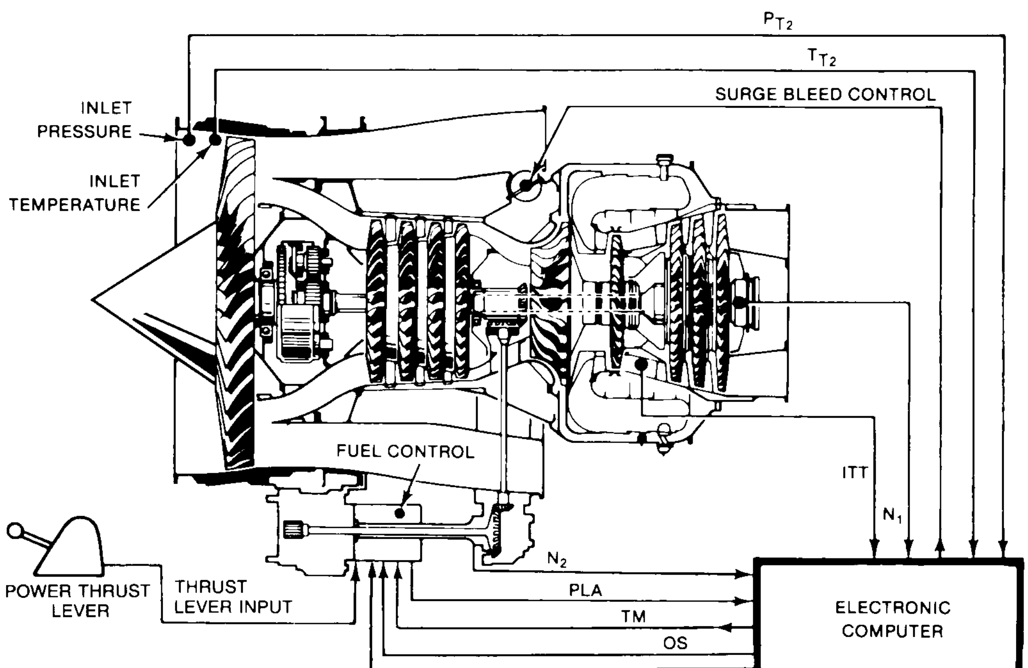
Automatic Mode Operation

The computer controls the fuel flow based on thrust lever position (PLA) and atmosphere conditions, while automatically maintaining N_1, N_2, and ITT within prescribed limits to permit optimum engine acceleration rates. The computer provides engine overspeed protection and also controls the surge bleed valve to prevent compressor stalls and surges. During engine start, the computer provides automatic fuel enrichment, starter disengagement, and termination of ignition and standby fuel pump operation.

The computer receives input signals representing the following engine parameters (Figure 7-8):

- N_1 (fan speed)
- N_2 (turbine speed)
- PLA (power level angle)
- P_{T2} (inlet pressure)
- T_{T2} (inlet temperature)
- ITT (interstage turbine temperature)

Figure 7-7. Electronic Fuel Computer



LEGEND

N_1 = LOW-PRESSURE SPOOL SPEED
N_2 = HIGH-PRESSURE SPOOL SPEED
P_{T2} = ENGINE INLET TOTAL PRESSURE
T_{T2} = ENGINE INLET TOTAL TEMPERATURE
ITT = INTERSTAGE TURBINE TEMPERATURE
PLA = POWER LEVER ANGLE
TM = DC TORQUE MOTOR
OS = OVERSPEED SOLENOID
MMS = MANUAL MODE SOLENOID

Figure 7-8. Computer Inputs and Outputs

The computer analyzes these signals and produces output signals which are sent to the torque motor (to control fuel flow) and to the surge bleed valve (to control compressor airflow). Thrust lever movement mechanically moves a power lever angle potentiometer, which furnishes a variable electrical signal (PLA) to the computer. This is the command input for a specific thrust setting. Fuel flow is metered by the torque motor to produce and maintain the desired thrust. Inlet temperature and pressure (P_{T2}/T_{T2}), N_1, N_2, and ITT signals are used to optimize engine acceleration rates and limit thrust and temperature within normal limits. By powering one or the other of the two surge bleed valve control solenoids, the computer opens or closes the surge bleed valve during engine acceleration and deceleration to prevent compressor stalls and engine surges.

In automatic operation, the mechanical flyweight governor section limits engine overspeed to 105% N_2 rpm. Should the 105% governing function fail, the computer energizes the ultimate overspeed solenoid valve closed at 109% N_1 or 110% N_2 to shut off fuel flow to the engine.

Indication

The computer constantly monitors input and output signals and, with the exception of ITT input loss, automatically reverts to manual mode if these signals are lost. In this case, or if computer power is lost, the amber "L" or "R FUEL CMPTR" annunciator light illuminates. In some cases it may be possible to regain normal operation. Refer to Section IV, Abnormal Procedures, of the *AFM*.

Manual Mode Operation

When the computer fails or is turned off, the fuel control unit assumes manual control of the fuel metering to the engine. The torque motor valve is deenergized and opens fully. The fuel flow is controlled by the mechanical flyweight governor section, functioning as an ***onspeed*** governor, utilizing the metering valve. The surge bleed valve automatically goes to the 1/3-open position and remains there.

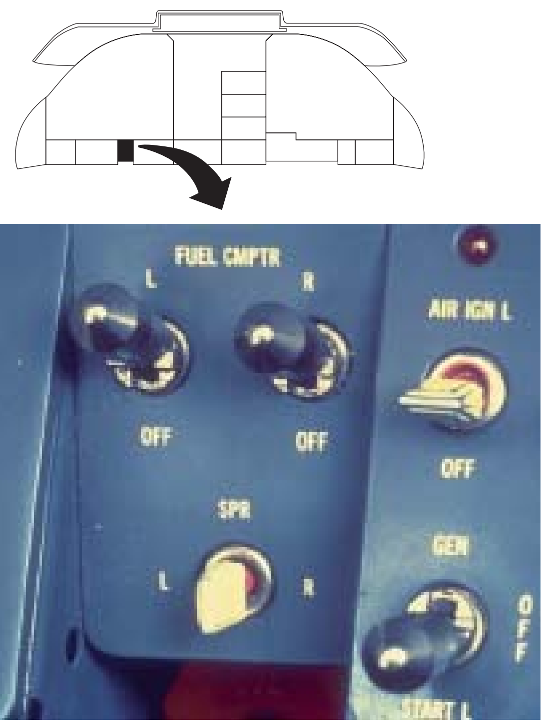


Figure 7-9. Fuel Computer and SPR Switches

START PRESSURE REGULATOR (SPR)

Fuel enrichment is automatically controlled by the fuel computer during starting up to 200°C. It may be extended manually to assist engine acceleration during starting in cold ambient temperatures (below 0° F) or during airstart at low altitude/high airspeed if light-off does not occur withing 5 seconds after moving the thrust lever to IDLE. This additional fuel is controlled by a three-position switch (Figure 7-9) labeled "SPR L" and "R." The switch is spring-loaded to the center (off) position. When additional start fuel is required, the switch must be held in the L or R position and released when ITT indicates between 300° C and 400° C. SPR is a computer function and is available only in the computer-on mode. Manual SPR overrides the automatic temperature-limiting feature of the computer. Therefore, ITT monitoring during SPR operation is extremely important. It should be used only during starting and discontinued when ITT is in the 300° C to 400° C range.

SURGE BLEED VALVE

The surge bleed valve functions to maintain a safe surge margin in the LP compressor by spilling some LP compressor air into the bypass duct, thus preventing LP compressor stalls and surges during acceleration and deceleration when large LP-HP rpm mismatch occurs. The surge bleed valve has three positions: FULL OPEN, FULL CLOSED, and 1/3 OPEN. Surge valve position is controlled by two fuel computer-operated solenoid valves which route P_3 bleed air to a respective port on the surge valve. By energizing one solenoid valve, the computer opens the surge valve, while energizing the other solenoid valve closes it. By deenergizing both solenoid valves, the surge valve assumes the 1/3 OPEN position which automatically occurs if the computer fails or is switched off, thereby providing some surge margin continuously while operating in manual mode. In addition, the surge bleed valve will assume the FULL OPEN position in the computer-on mode whenever the PLA is 26° or less (42° on early model computers). During acceleration, the computer first signals the surge bleed valve to assume the 1/3 OPEN position; if the surge margin cannot be maintained in this position, the computer will command the FULL OPEN position. The opposite is true during deceleration. In summary, surge bleed valve position is a function of the fuel computer, relative to N_1, N_2, and thrust lever angle.

FUEL FLOW

Fuel flow is sensed downstream of the FCU and is displayed on a dual-needle gage on the center instrument panel (Figure 7-5). The needles are labeled "L" and "R," and the gage is calibrated in pounds per hour times 1,000. Electrical power is supplied directly from the battery-charging bus through a 10-amp current limiter.

A resettable digital fuel counter (Figure 7-10) is located on the fuel control panel on the center pedestal. The indicator is operated by the fuel flow indicating system and displays pounds of fuel consumed. The indicator should be reset prior to engine starting.

FLOW DIVIDER

The flow divider splits fuel flow between the primary and secondary manifolds to which the fuel nozzles are connected. During engine

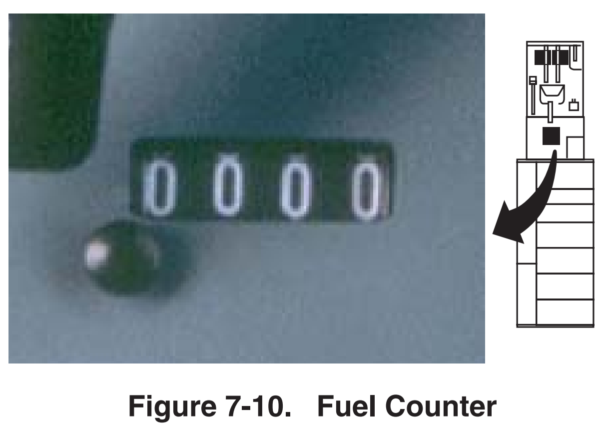


Figure 7-10. Fuel Counter

starts, the flow divider blocks the secondary manifold until fuel flow reaches 150 pounds per hour.

FUEL SPRAY NOZZLES

The twelve duplex fuel spray nozzles in the combustion chamber consist of concentric primary and secondary orifices that function to atomize the fuel delivered by the primary and secondary fuel manifolds.

OPERATION

Figure 7-11 illustrates the operation of the engine fuel system in simplified format.

IGNITION SYSTEM

GENERAL

A solid-state, high-energy ignition system consists of a dual ignition exciter (mounted on the engine) and two igniter plugs in the combustion chamber. Two ignition modes are available: (1) automatic and (2) selective.

AUTOMATIC MODE

Automatic ignition occurs during engine starting when the START–GEN switch on the center switch panel (Figure 7-12) is positioned to START and the thrust lever is moved from

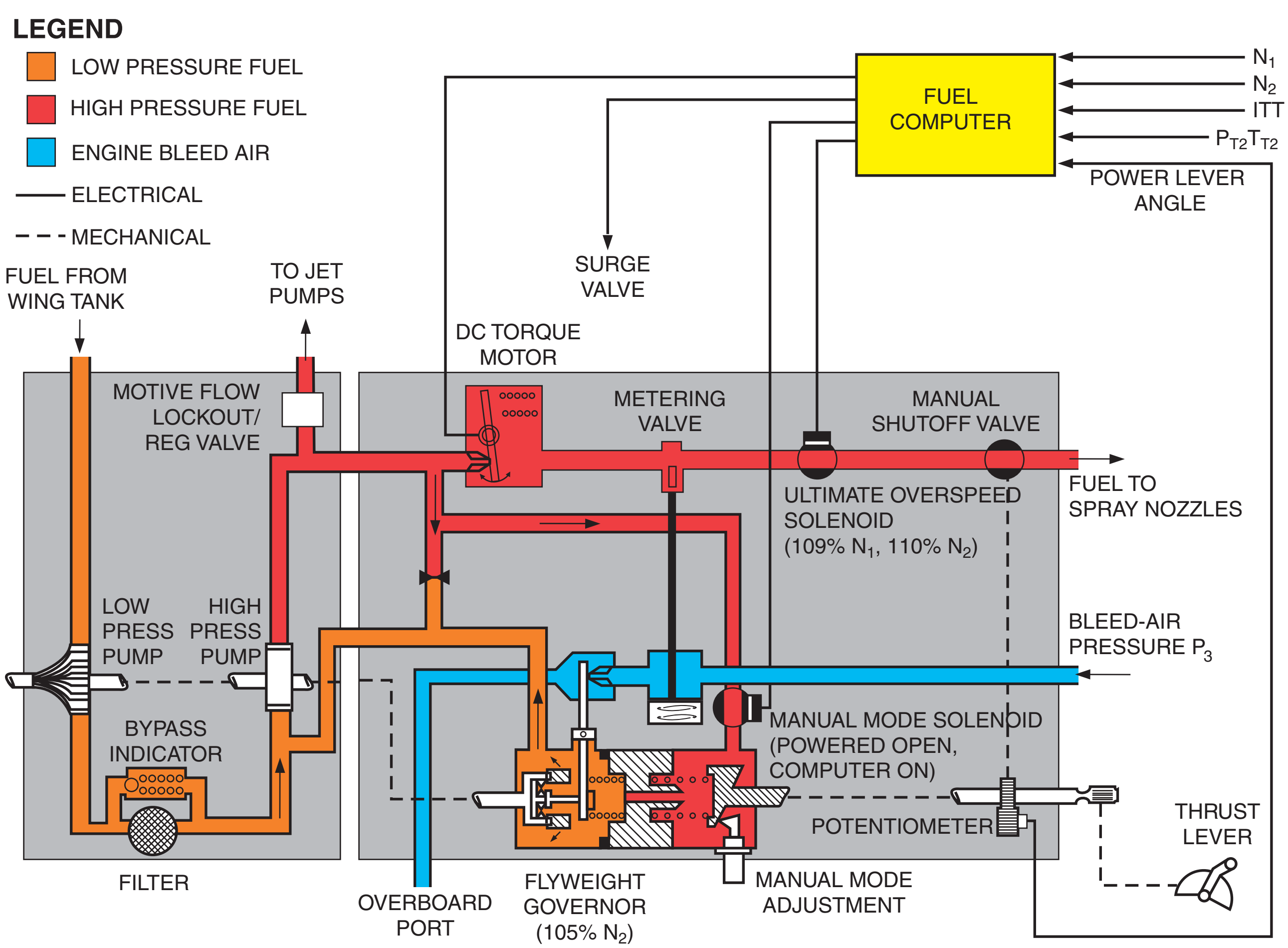


Figure 7-11. Engine Fuel System

CUT–OFF to IDLE position. Ignition is automatically terminated (in a computer-on mode) by an electronic speed switch in the computer at 45% or 50% N_2 as determined by the computer model installed. Power for automatic ignition is provided by the L and R IGN & START circuit breakers on the left and right power buses respectively.

If the computer switch is off during a starter-assisted start or if the computer reverts to manual mode during start, ignition will continue until the START–GEN switch is moved out of the START position. Ignition will also terminate (computer on or off) if the thrust lever is moved forward to a position representing approximately 70% N_2.

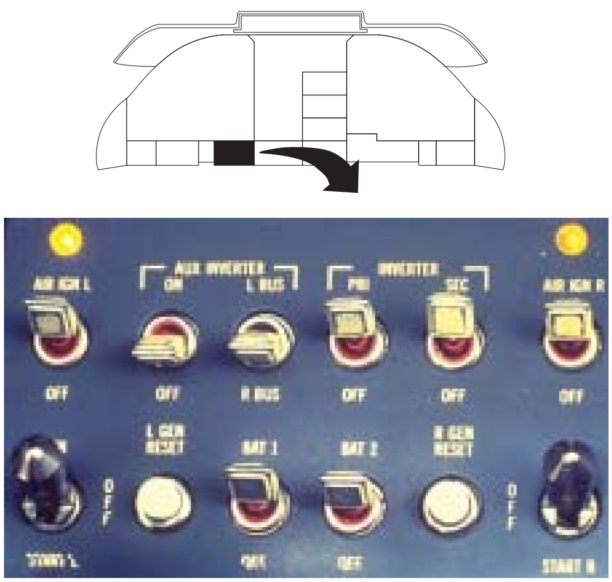


Figure 7-12. Center Switch Panel

SELECTIVE MODE

Selective ignition is controlled by two-position switches labeled " AIR IGN L" and " AIR IGN R"(Figure 7-12) located on the center switch panel. When the switch is positioned to AIR IGN, the igniters will operate continuously. Ignition power is supplied by the L or R AIR IGN circuit breakers on the left and right essential buses, respectively.

Selective use of air ignition is required for all takeoffs and landings, and also for windmilling airstarts. It may by used continuously when flying in heavy precipitation, icing conditions, or turbulent air.

INDICATION

An amber light (Figure 7-12 and Annunciator Panel section) located above each AIR IGN switch will be on whenever power is being supplied to the associated ignition exciter. The ignition lights (if on) will dim if the NAV LTS switch, located on the right switch panel, is on.

ENGINE CONTROLS

Engine control is achieved by thrust levers (Figure 7-13) mounted on a quadrant on the center pedestal. The levers can be moved from the fully aft or CUT–OFF position through the IDLE position to the fully forward, maximum power position. A stop is provided at the IDLE position which requires raising a release trigger on the outboard side of each lever before the lever can be moved to CUT–OFF.

The thrust lever is connected to the FCU by a cable. In the automatic mode, thrust lever position is relayed to the computer as an electrical signal from a potentiometer inside the FCU that represents thrust lever angle. In manual mode, thrust lever movement changes P3, which operates the metering valve. The thrust lever also mechanically operates a rotary fuel shutoff valve.

Figure 7-13. Throttle Quadrant

Optional thrust reverser levers are piggy-back mounted on the thrust lever. (See Thrust Reversers, this chapter).

STARTERS

GENERAL

Each engine starter is powered through relays controlled by the GEN–OFF–START switch and the fuel computer (during computer-on starts). A "soft start" feature incorporates a resistor to minimize the effect of the initial torque on the mechanical drive components. After a 1.5-second delay, a relay operates to allow the starting current to bypass the resistor so that full electrical potential is available to complete the start. Automatic starter disengagement occurs at 50% N_2 (45% for SNs 35-245 and subsequent, 36-045 and subsequent, and earlier airplanes equipped with 1142 fuel computers). On SNs 35-370, 35-390, and 36-048 and subsequent, illumination of a red light under the appropriate GEN–OFF–START switch indicates that the starter is engaged. On earlier airplanes modified by AMK 80-17, the red lights may be installed elsewhere on the instrument panel.

The GEN–OFF–START switches are locking-lever switches. They must be pulled out to move to the START position. It is not necessary to pull out for movement to any other position.

When either GEN–OFF–START switch is positioned to START for a normal computer-on start, the start sequence is initiated for that engine. The start sequence and circuitry for the left engine are presented herein; they are identical with those for the right engine.

There are three different designs for the relay circuits which route power to the starter.

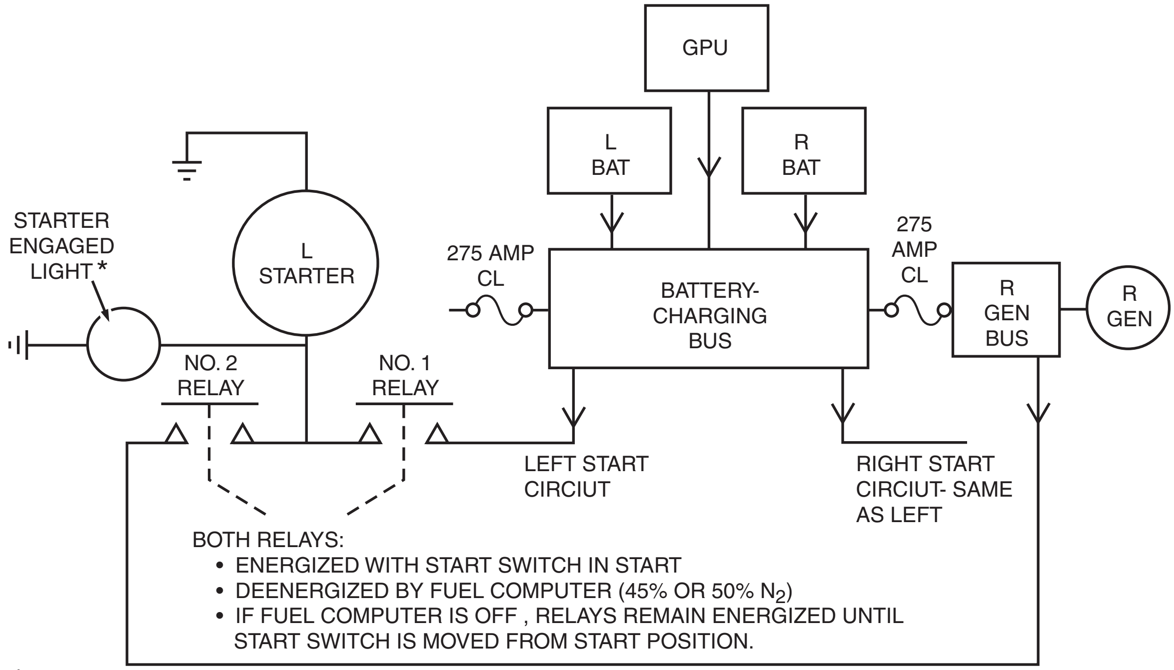
For SNs 35-002 through 35-147 and 36-002 through 36-035, the relays are wired in parallel.

One relay is connected to the opposite generator bus and the other to the battery-charging bus. This arrangement was designed to protect the 275-amp current limiters during initiation of each engine start sequence (Figure 7-14).

For SNs 35-148 through 35-389, except 35-370, and 36-036 through 36-047, the relays are again wired in parallel, but both are connected to the battery-charging bus (Figure 7-15). This design change includes automatic single-generator voltage reduction on the ground and during airstarts, resulting in 275-amp current limiter protection when the first generator is switched on and during initiation of the start sequence on the second engine.

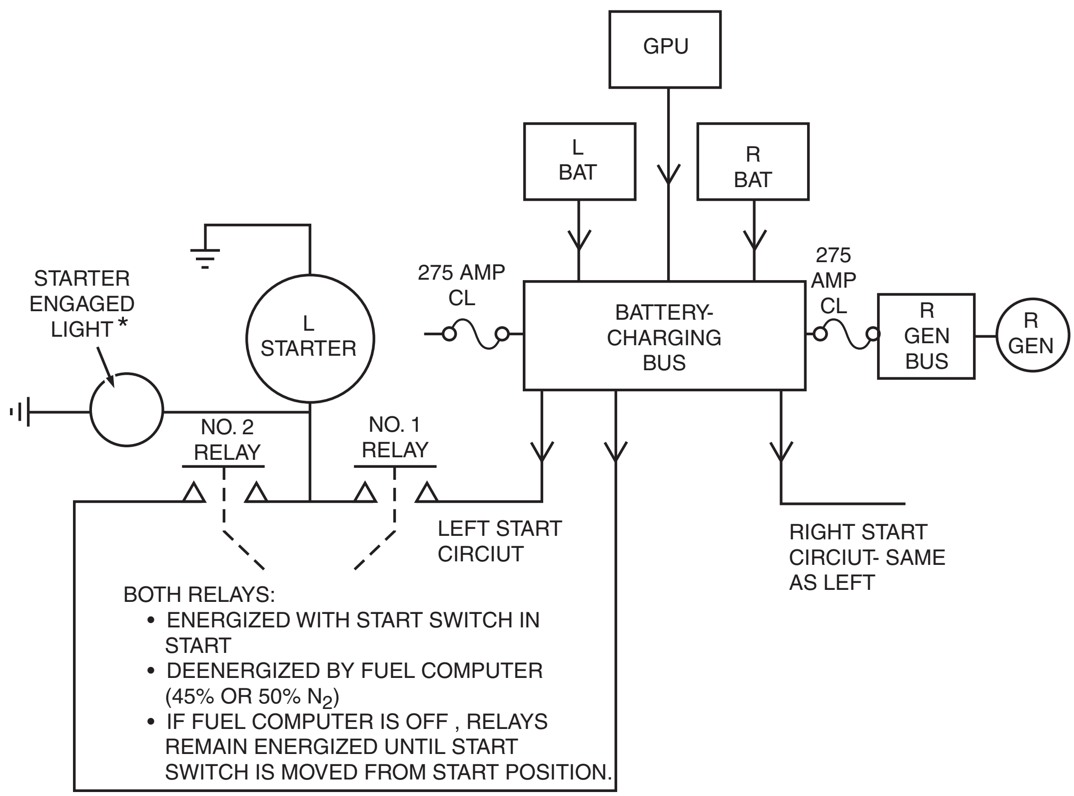
For SNs 35-002 through 35-389, except 35-370, and 36-002 through 36-047, two separate modifications have been introduced to the starting circuits:

- AMK 80-17 provides a red starter-engaged light for each starter to provide indication of starter engagement (Figures 7-14 and 7-15). Location of the lights is left to customer specification.



*WHEN INSTALLED BY AMK 80-17

Figure 7-14. Left Start Circuit—SNs 35-002 through 35-147 and 36-002 through 36-035



*WHEN INSTALLED BY AMK 80-17

Figure 7-15. Left Start Circuit—SNs 35-148 through 35-389, except 35-370 and 36-036 through 36-047

- AAK 81-1 installs a third starter relay in series between the two existing relays and the starter motor; the circuits which energize the relays are redesigned. AMK 80-17 is a prerequisite or concurrent requirement for this modification (Figure 7-16).

For SNs 35-370, 35-390, and subsequent, and 36-048 and subsequent, two starter relays are wired in series to the battery-charging bus, and the red starter-engaged lights are standard. (Figure 7-17).

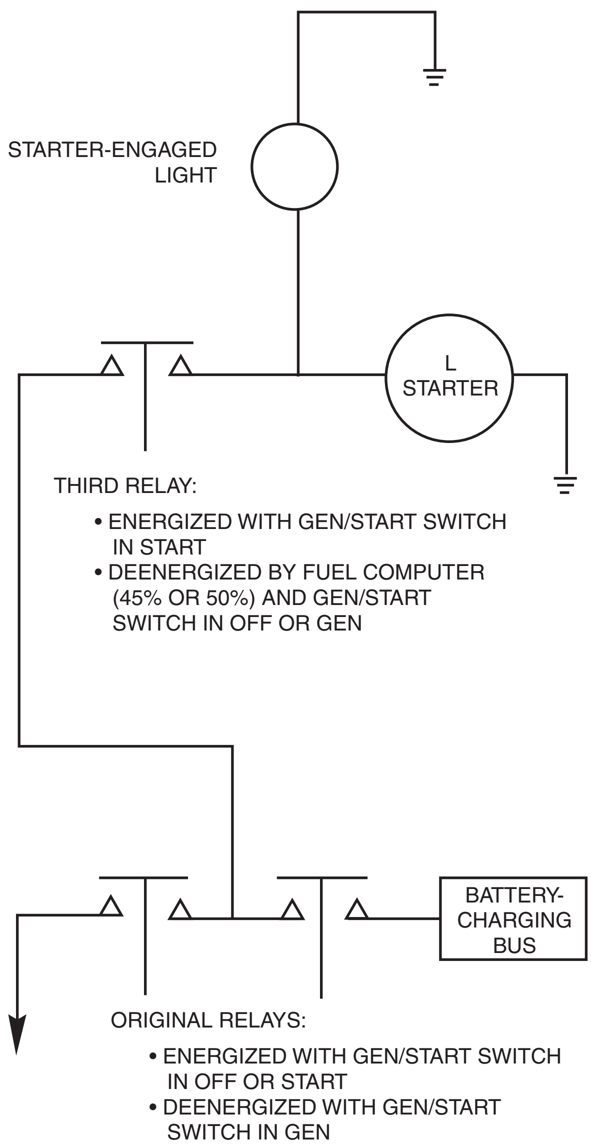


Figure 7-16. Installation of AAK 81-1

OPERATION

SNs 35-002 through 35-389, except 35-370, and 36-002 through 36-047 with or without AMK 80-17

With the airplane battery switches on, moving the GEN–OFF–START switch to START connects DC power through the IGN & START circuit breaker to energize the starter relays. Starter engagement occurs along with illumination of the starter-engaged light if AMK 80-17 is installed. With the fuel computer on, starter disengagement occurs automatically when power is removed from the starter relay circuit. At this time the starter-engaged light (if installed) extinguishes.

SNs 35-002 through 35-389, except 370, and 36-002 through 36-047 when incorporating AMK 80-17 and AAK 81-1

All three starter relays must be energized to power the starter and illuminate the starter-engaged light. With the airplane battery switches on, the two parallel relays are energized closed through the IGN & START circuit breaker anytime the GEN–OFF–START switch is in the OFF or START position. The third relay is also energized from the IGN & START circuit breaker, but only when the start switch is in the START position. If the fuel computer is on for the start, it will automatically deenergize the third relay when N_2 reaches 45% (or 50%, depending on which computer is installed). The starter is then disengaged and the starter-engaged light extinguishes. Moving the GEN–OFF–START switch to GEN deenergizes the two parallel relays to back up the release of the third relay. If either of the two parallel relays, plus the third relay, remain in the closed position, the starter-engaged light remains in the closed position, the starter-engaged light remains illuminated and the starter remains powered. The only way to disengage the starter in this event is to remove electrical power from the battery-charging bus by turning off both batteries and both generators. If the starter-engaged light remains illuminated after start, consult Section IV, Abnormal Procedures, of the approved *AFM*.

SNs 35-370, 35-390 and Subsequent, and 36-048 and Subsequent

There are two relays in series between the battery-charging bus and the starter (Figure 7-17). Both must be energized to power the starter and illuminate the starter-engaged light. With the airplane battery switches on, the No. 1 relay is energized through the IGN & START circuit breaker anytime the GEN–OFF–START switch is in the OFF or START position. The No. 2 relay is also energized from the IGN & START circuit breaker, but only when the GEN–OFF–START switch is in the START position. If the fuel computer is on for the start, it will automatically deenergize the No. 2 relay when N_2 reach 45%. The starter-engaged light extinguishes. Moving the GEN–OFF–START switch to GEN deenergizes the No. 1 relay to back up the release of the No. 2 relay. If both relays fail in the energized position, the starter-engaged light remains illuminated, and the starter remains powered. The only way to disengage the starter in this event is to remove electrical power from the battery-charging bus by turning off both batteries and both generators. If the starter-engaged light remains illuminated after start, consult Section IV, Abnormal Procedures, of the approved *AFM*.

OTHER START FUNCTIONS

In addition to the starter, a number of other circuits are affected when the GEN–OFF–START switch is placed in START. The standby fuel pump in the associated wing is energized, the ignition is armed, and the Freon air-conditioning and auxiliary heating systems are disabled.

Additionally, on SNs 35-002 through 35-057 and 36-002 through 36-017, the motive-flow

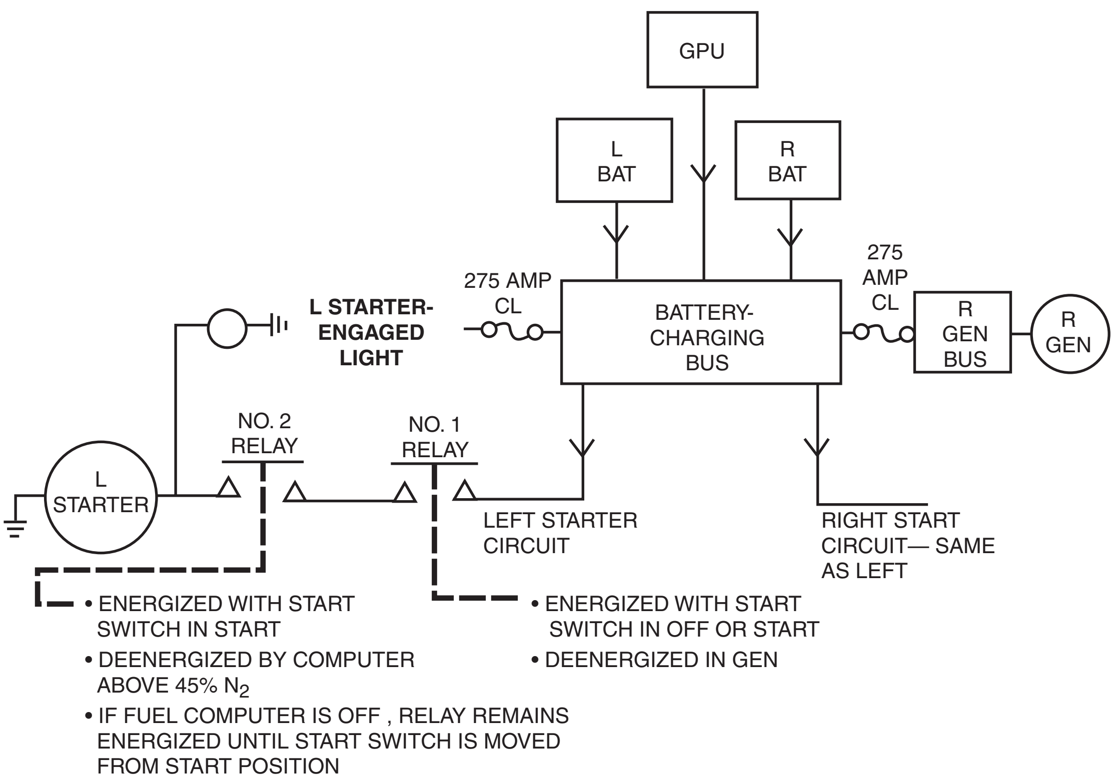


Figure 7-17. Left Start Circuit—SNs 35-370, 35-390, and Subsequent, and 36-048 and Subsequent

control valve must automatically cycle closed, or the starter relays will not energize. When the associated thrust lever is moved from CUT-OFF to IDLE, a switch in the throttle quadrant closes and activates the ignition system, causing the ignition light to illuminate. When turbine speed reaches 45% or 50% (depending on computer model), the fuel computer removes power from the start relay(s). This causes the starter to disengage and terminates ignition and standby pump operation. The start sequence can be aborted at any point by placing the thrust lever to CUT-OFF and the GEN–OFF–START switch to OFF. If engine start is accomplished with the fuel computer off, the starter is not automatically disengaged after starting. The pilot must position the GEN–OFF–START switch to OFF to terminate starter engagement and ignition.

After the engine reaches idle rpm, the GEN–OFF–START switch may be placed to the GEN position. The generator may be turned on when a GPU is connected; however, it is preferable to place the GEN–OFF–START switch to OFF after starting engines until the GPU is disconnected.

On SNs 35-002 through 35-147 and 36-002 through 36-035 (during battery start), after the first engine is started, one battery switch must be turned off prior to selecting GEN on the GEN–OFF–START switch. This action reduces the initial load on the generator and protects the 275-amp current-limiter. On later airplanes this procedure is not required, and the GEN position may be selected immediately after start.

When the GEN–OFF–START switch is moved from START, those systems which were disabled during the start can now be operated.

ENGINE INSTRUMENTATION

GENERAL

The primary engine instruments are mounted in two vertical rows on the center instrument panel (Figure 7-5). From top to bottom these instruments are:

- Turbine speed (N_2 rpm)
- Turbine temperature (ITT)
- Fan speed (N_1 rpm)

TURBINE SPEED (N_2)

Turbine speed (N_2 rpm) is remotely sensed by a dual monopole transducer installed in the transfer gearcase. One output signal is sent to the turbine speed (N_2) indicator, and another to the fuel computer. This indicator includes an analog scale and pointer calibrated in percentage of maximum design rpm, and a digital counter, calibrated in tenths of percent. A red OFF flag will appear on the face of the indicator to indicate loss of DC power to the indicator. The indicators are powered through the L R TURB RPM circuit breakers located on the left and right main buses, respectively.

TURBINE TEMPERATURE (ITT)

Turbine temperature is sensed by ten parallel-wired thermocouples located between the HP and LP turbines. An averager circuit provides two output signals—one to the turbine temperature indicator, and the other to the fuel computer. The indicator includes an analog scale and pointer, calibrated in degrees Celsius, and a digital counter, calibrated to the nearest whole degree. A red OFF flag will appear on the face of the indicator. The indicators are powered through the L and R ITT circuit breakers located on the left and right essential buses, respectively.

FAN SPEED (N_1)

Rotation of the LP rotor is sensed by a dual monopole transducer installed under a cover plate at the aft end of the LP rotor shaft. One output signal is sent to the fan speed (N1) indicator, and the other to the fuel computer. All other operational aspects of the indicator are identical with the turbine speed indicator except that the indicators are powered through the L and R FAN RPM circuit breakers located on the left and right essential buses, respectively.

NOTE

The fan speed (N_1) indicators are the ***primary*** power indicators.

ENGINE SYNCHRONIZER SYSTEM

GENERAL

The engine synchronizer system is installed on airplane SNs 35-067 and subsequent, and 36-018 and subsequent as standard equipment. It incorporates a synchronizer control box which uses N_1 or N_2 inputs from both engine fuel computers to enable automatic or manual synchronization of the engines.

CONTROL

The system incorporates a single R ENG SYNC indicator located on the pilot's lower instrument panel (Figure 7-18), and two ENG SYNC switches located immediately below the thrust levers (labeled "SYNC–OFF" and "TURB–FAN," respectively (Figure 7-19). The system operates manually (with the SYNC–OFF switch in the OFF position) or automatically (with the SYNC–OFF switch in the SYNC position) to maintain the right engine fan or turbine in sync with the left engine fan or turbine as determined by the TURB-FAN switch.

INDICATION

An amber ENG SYNC light (Annunciator Panel section) on the glareshield annunciator panel will be illuminated anytime the nose gear is down and locked with the SYNC–OFF switch in the SYNC position. The R ENG SYNC (SLOW/FAST) indicator indicates right engine rpm deviation from that of the left engine.

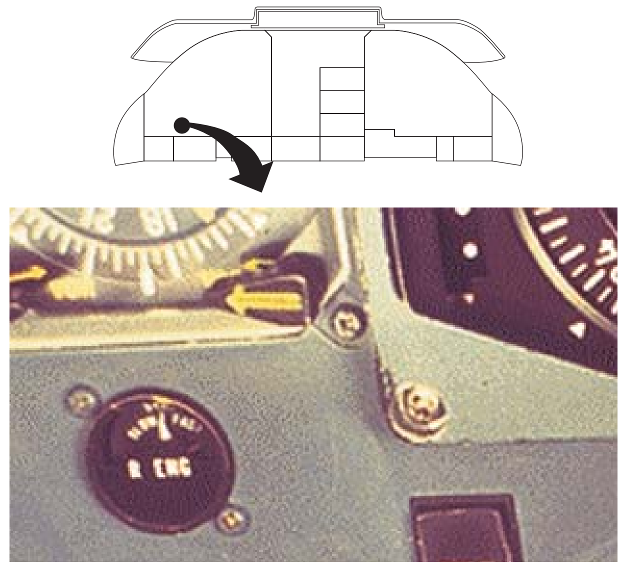


Figure 7-18. ENG SYNC Indicator

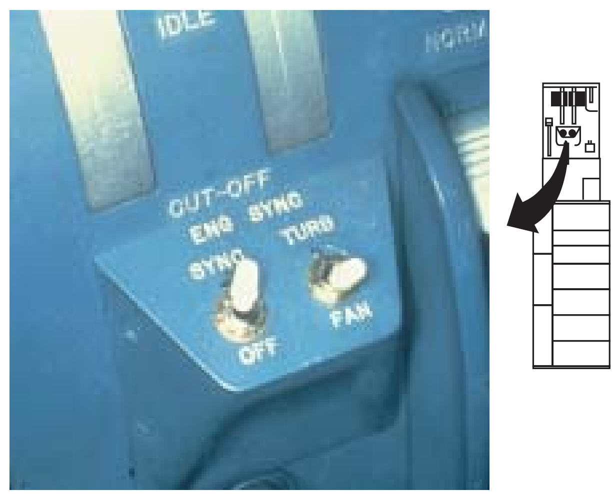


Figure 7-19. ENG SYNC Control Switches

OPERATION

Manual synchronization is accomplished by selecting OFF on the SYNC–OFF switch. The R ENG SYNC indicator shows SLOW or FAST out-of-sync condition of the right-hand engine (slave engine) relative to the left-hand engine (master engine). The pilot has the option of selecting either N_2 or N_1 as the rpm reference by using the TURB–FAN switch.

Automatic synchronization is accomplished by selecting SYNC on the SYNC–OFF switch. If the engines are within approximately 2.5% rpm of each other, the right engine will automatically synchronize to the left engine. It is necessary, therefore, to manually sync to within 2.5% initially. As in manual sync, either the N_2 OR N_1 may be selected as the rpm reference.

DC electrical power is supplied to the system from the left essential bus through the left FUEL CMPTR circuit breaker to the L FUEL CMPTR switch.

The amber ENG SYNC annunciator light serves as a reminder that the system should be turned off.

The engine sync system is inoperative if either fuel computer is off or failed.

THRUST REVERSERS (OPTIONAL EQUIPMENT)

GENERAL

The Learjet 35/36 series airplanes may be equipped with either a cascade thrust reverser system, manufactured by Aeronca, Inc., or a target reverser system (TR 4000), manufactured by the Dee Howard Co. Effective with airplane SNs 35-507 and 36-054, either system is available for retrofit, but only the target system is available during production.

AERONCA THRUST REVERSERS

General

The Aeronca thrust reverser system incorporates a translating structure (Figure 7-20) which forms the after body of the engine nacelle. When deployed, it exposes cascade vanes, while simultaneously operating two blocker doors that block engine exhaust ducts, thereby deflecting all exhaust in a forward direction through the cascade vanes.

The translating structure is deployed and stowed by an air motor using HP bleed air from the associated engine and sequenced electrically by microswitches operated by the reverser levers (Figure 7-13). The system incorporates automatic stow and stow-prevention features to minimize the possibility of inadvertent deployment on the ground and in flight, and inadvertent stow at high reverse thrust settings. The system is self-arming on the ground through control circuits operating through the landing gear squat switch relay box.

Figure 7-20. Thrust Reverser (Aeronca)

Control

The reverser levers control the deploy and stow cycles and engine power when the reversers are deployed. The thrust reverser control panel (Figure 7-21) is located in the center of the glareshield above the annunciator panel. It incorporates a rocker selector switch for normal and emergency operations, seven annunciator lights which provide visual evidence of normal sequencing and certain abnormal conditions, and a test switch for performing system test functions.

Thrust Reverser Control Panel

TEST Button

The TEST button provides a means of checking operation of the bleed valve and, on some airplanes, also checks the blocker door position indicating circuits. When depressed, the white BLEED VALVE lights should illuminate, and, on airplanes incorporating AMK 81-6 (installation of blocker door position indicator [DPI] switches), the white UNLOCK lights will flash to indicate that the blocker doors are correctly stowed.

BLEED VALVE Lights

In addition to the test function above, the white BLEED VALVE lights illuminate as reverse thrust is increased to indicate that HP bleed air to the air motors is shut off. This prevents inadvertent stow commands.

DEPLOY Lights

The two white DEPLOY lights illuminate when the corresponding thrust reverser is fully deployed. ***Both*** DEPLOY lights must be illuminated; otherwise, the reverser lever solenoid interlocks will not release to permit thrust increase.

UNLOCK Lights

In addition to the test function above, the two white or amber UNLOCK lights illuminate ***steady*** while the translating assembly is in transit during the deploy and stow cycles; that is, the reversers are not fully deployed or locked in the stowed position.

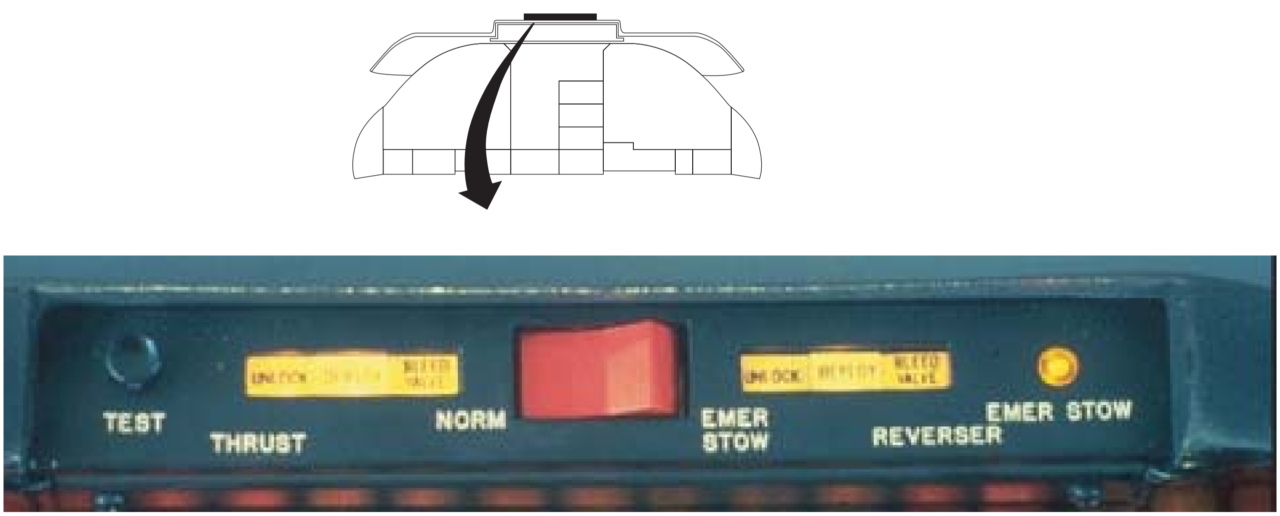


Figure 7-21. Thrust Reverser Control Panel (Aeronca)

NORM-EMER STOW Switch

In the NORM position, the red rocker switch provides the electrical circuitry for all normal and automatic functions. In the EMER STOW position, all normal electrical circuits are bypassed, and a separate circuit applies stow commands to the reversers.

EMER STOW Light

The amber EMER STOW light illuminates whenever the NORM–EMER STOW switch is in the EMER STOW position and the emergency stow circuits have been activated, thus rendering the normal system inoperative.

System Operation

Arming

The reversers are automatically armed for normal operation when the following conditions exist:

- The T/R circuit breakers are closed.

NOTE

The T/R POS and T/R EMER STOW circuit breakers are located on the left main bus, and the T/R CONT circuit breaker is located on the right main bus.

- The airplane is on the ground (squat switch relay box is in the ground mode).
- The NORM-EMER STOW switch is in the NORM position.
- ***Both*** thrust levers are at the IDLE position.

Electrical power for deployment will not be available unless ***both*** thrust levers are at IDLE and ***both*** reverser levers are raised to the deploy position.

Deploy

When the reverser levers are moved to the deploy position (the first "hard stop"), the main-thrust levers are locked in the IDLE position, and N_1 rpm increases to approximately 55%-60%. Switches are operated by each reverser lever to complete circuits that energize pneumatic latch releases (two per reverser) to unlock the translating assembly. Switches on each latch function to:

1. Illuminate the UNLOCK lights.
2. Shut off bleed air to the windshield heat, nacelle heat, and wing/stabilizer heat systems (for approximately 3 seconds).
3. Energize the air motor directional control solenoid valve which routes HP bleed air through an air inlet valve, into the air motor on the respective reverser.

The air motor transmits torque to drive the translating structure aft, exposing the cascade vanes. As the assembly approaches its aft limit, the blocker doors are closed, the DEPLOY light illuminates, the UNLOCK light extinguishes, and the reverser lever solenoid-operated interlock releases.

The reverser lever solenoid-operated interlock prevents movement of the reverser levers aft of the idle-deploy position until ***both*** DEPLOY lights are illuminated. If the pilot is applying excessive aft pressure on the reverser levers when the DEPLOY lights illuminate, the solenoid-operated interlock will not release, and reverse thrust above approximately 55%–60% N_1 will not be possible. The interlock will release when aft pressure is relaxed.

For single-engine reversing, ***both*** thrust levers must be at IDLE, and ***both*** reverser levers must be raised to the deploy position in order to deploy the reverser on the operating engine. Since the reverser on the inoperative engine will not deploy, the solenoid interlock will not release; therefore, reverse thrust on the operating engine is limited to reverse idle (55% to 60% N_1).

Reverse Thrust

After both DEPLOY lights illuminate (two-engine operation) and the solenoid-operated interlocks release, the reverser levers can be pulled further aft to increase engine power. There is no limitation on engine thrust when using reverse except that the normal forward thrust limitations still apply.

Stow Prevention

As reverse thrust (N_1) is increased, a pressure switch in each reverser system causes the bleed valve on the corresponding system to open and illuminate the BLEED VALVE light. This isolates the bleed-air system from the air motors (by closing the air inlet valve) until stow is commanded by the reverser levers or with the EMER STOW switch, thus preventing inadvertent stow on either engine which could cause significant thrust asymmetry.

At 60 KIAS the reverser levers should be smoothly returned to the idle-deploy position.

When the engines have reached the reverse-idle rpm (approximately 55%–60% N_1), the pilot may stow the reversers by moving the reverser levers to the full forward position.

Normal Stow

When the reverser levers are moved from the idle-deploy position to the full forward and down position (stow), they operate switches that send a stow signal to the directional control solenoid of the air motor. The bleed valve closes, admitting bleed air into the air motor, causing it to drive the translating structure toward the stow position. The DEPLOY lights extinguish and, simultaneously, the UNLOCK lights illuminate. When the thrust reversers are fully stowed and the pneumatic latches engage the translating structure, the UNLOCK lights will extinguish. As in the deploy cycle, bleed air is shut off to the windshield, nacelle, and wing/stabilizer heat systems for approximately 3 seconds when the stow cycle is initiated.

Abnormal Indications

UNLOCK Light (Steady)

If either thrust reverser fails to completely stow, or if any of the pneumatic latches fails to engage after stowing, the corresponding UNLOCK light will remain illuminated. Also, if a pneumatic latch disengages at any time, the corresponding UNLOCK light will illuminate.

The automatic stow circuit is activated anytime an UNLOCK light illuminates with the reverser levers in the stowed position. Stow pressure will be applied until the UNLOCK light extinguishes.

UNLOCK Light (Flashing)

A flashing UNLOCK light is a function of modification AMK 81-6 (installation of blocker door position indicator [DPI] switches). Proper stowing of the blocker doors is essential for continued operation. An -undetected jammed blocker door could result in inadvertent deployment of the affected thrust reverser. Each blocker door (upper and lower) actuates a DPI switch when in the properly stowed position. If the stow cycle is complete (latches engaged) and one of the DPI switches is not actuated, the corresponding UNLOCK light will flash to indicate the jammed blocker door. Since damage to the system has occurred, repairs are required prior to the next takeoff.

A flashing UNLOCK light at any other time indicates a malfunctioning DPI switch, but the blocker doors are still properly stowed. This does not preclude operating the reversers on landing.

BLEED VALVE Light

When the reversers are stowed, illumination of a BLEED VALVE light means that the bleed valve is open. This isolates the bleed-air-system from the air motor, and deployment of the affected reverser will not be possible.

Automatic Stow

The thrust reversers incorporate an auto-stow provision. If any of the pneumatic latches release (UNLOCK light illuminates) when the reverser levers are stowed, electrical power from the T/R CONT circuit breaker is applied to open the bleed-air valve and to the directional solenoid, causing the air motor to stow the translating structure. Stow pressure will be maintained until the UNLOCK light extinguishes.

Emergency Stow

The NORM–EMER STOW switch is normally left in the NORM position. The EMER position is designed for inadvertent UNLOCK or DEPLOY conditions when the reverser levers are stowed. Power is provided by the TR EMER STOW circuit breaker on the left main bus.

In the case of the UNLOCK or DEPLOY condition in flight, the EMER position on the switch in ***not*** functional with the thrust levers set at any power setting above approximately 70% N_1. It is therefore imperative that if the EMER selection is made for any reason due to a reverser malfunction, the amber EMER STOW indicator light be monitored. If the power setting is sufficiently high to prelude the emergency stow circuits from functioning, the amber light will not illuminate, and the appropriate thrust lever must be retarded until the light illuminates. Illumination of the EMER STOW light gives visual indication that the emergency stow circuits have, in fact, been activated.

In the event of a system malfunction while intentionally operating in the reversing range, there is nothing to preclude use of the EMER STOW selection at any time, and doing so will immediately command all components to stow, and illuminate the amber EMER STOW light.

All thrust reverser normal, abnormal, and emergency procedures are contained in the supplement section of the approved *AFM*.

DEE HOWARD TR 4000 THRUST REVERSERS

General

The Dee Howard thrust reversers incorporate a hydraulically operated system (Figure 7-22) consisting of a pair of clamshell doors forming the afterbody of the engine nacelle. When deployed, the doors deflect all exhaust in a forward direction. The reverser hydraulic system is integral with the airplane's hydraulic system for normal operation. It is equipped with a separate accumulator and a one-way check valve which enable one deploy and stow

Figure 7-22. Thrust Reverser (Dee Howard TR 4000

cycle in the event of airplane hydraulic system failure. The accumulator preload pressure is 900–1,000 psi.

An automatic emergency stow system, which includes an automatic throttle-retard feature, is incorporated to provide protection against inadvertent deployments.

Two pairs of spring-loaded latches (one pair each side) secure the doors when stowed. Hydraulic actuators operate each pair of latches, the doors, and a throttle-retard mechanism. Hydraulic pressure is supplied by a selector valve which incorporates four separate solenoid valves that are electrically sequenced by microswitches. One of the solenoid valves (the isolation valve) blocks hydraulic pressure at the selector valve inlet until the system is fully armed. The other three solenoid valves are for latch release, door stow, and door deploy.

Control

The reverser levers control the deploy and stow cycles and engine power when the reversers are deployed. The thrust reverser control panel (Figure 7-23) is located in the center of the glareshield above the annunciator panel. It incorporates two ARM–OFF–TEST switches (one for each reverser) which provides system arming, disarming, and testing. Four annunciator lights (two for each reverser) provide visual indication of normal sequencing and certain abnormal conditions.

Thrust Reverser Control Panel

ARM–OFF–TEST Switches

Arming, disarming, and testing are accomplished for each reverser by use of the respective ARM–OFF–TEST switch. The ARM position is wired in series with the ground mode of the squat switch relay box, as well as an IDLE switch on the respective thrust lever. The system, therefore, will only ARM when the airplane is on the ground and the thrust levers are at IDLE.

The TEST position provides a means of checking operation of the hydraulic isolation valve. When TEST is selected, the isolation valve is energized open, and hydraulic pressure is applied to a pressure switch that illuminates the ARM light.

The Arm position enables all sequencing microswitches and energizes the isolation valve open. Illumination of the ARM light indicates that the isolation valve has opened and hydraulic pressure is available to the other three solenoid valves for normal sequencing. The OFF position

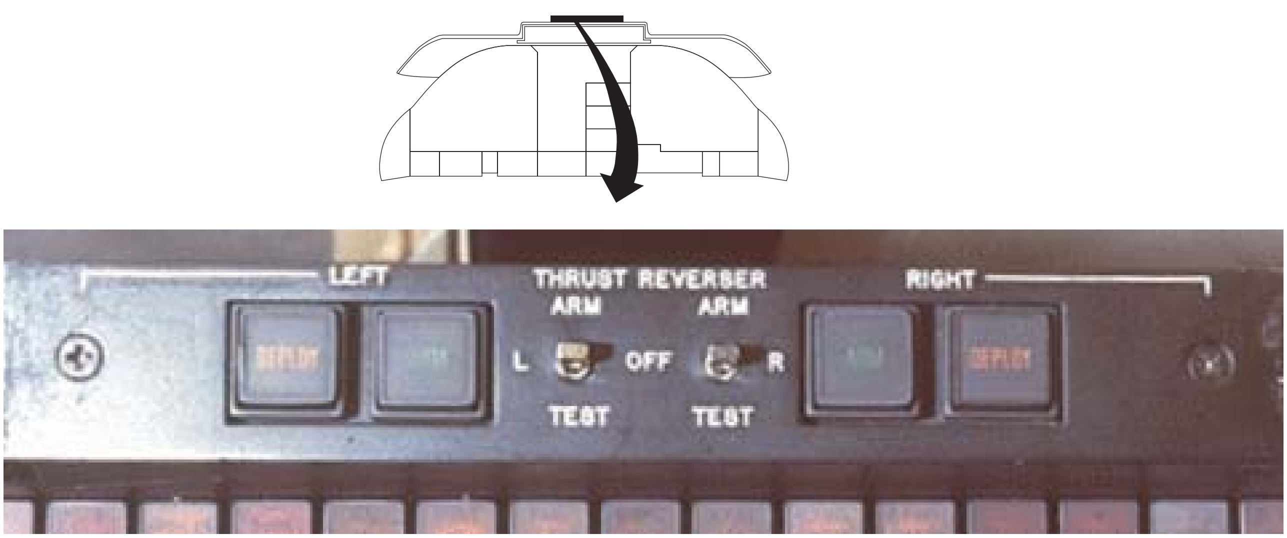


Figure 7-23. Thrust Reverser Control Panel (Dee Howard)

completely disarms the deploy circuits without disarming the automatic emergency stow system.

ARM Lights

The green ARM lights illuminate in conjunction with the TEST and ARM functions as described above. However, should the ARM light illuminate at any other time (i.e., in flight with the ARM–TEST switch in the OFF position), it indicates that two inboard (or outboard) door latches are unlocked, and automatic activation of the emergency stow circuit has occurred. This will be annunciated by a flashing DEPLOY light.

DEPLOY Lights

The amber DEPLOY lights flash during all stow/overstow cycles and illuminate ***steady*** when the respective reverser is in the fully deployed position during a normal deployment. A flashing DEPLOY light at any other time indicates that one or more of the door latches are unlocked (Automatic Emergency Stow this chapter).

System Operation

Arming

The reversers are armed for normal operation as follows:

- The T/R circuit breakers (two for each reverser) are closed.
- The airplane is on the ground (squat switch relay box is in the ground mode).
- The respective ARM–TEST switch is in the ARM position.
- The respective thrust lever is at the IDLE position.
- The respective green ARM light is illuminated.

Deploy

Raising the respective reverser lever to the idle deploy position (the first hard stop) locks the main thrust lever at IDLE and contacts a deploy switch that energizes the latch and stow solenoid valves open. This directs hydraulic pressure to both latch release actuators, the stow side of the door actuator, and the throttle-retard actuator. The resulting door "overstow" condition unloads the spring-loaded latches so that the latch release actuators can release them, and simultaneously, the throttle-retard actuator is operated by the stow pressure. When the latch release actuators engage their respective unlock switches, the stow solenoid valve is deenergized closed, the latch solenoid valve remains energized open, and the deploy solenoid valve is energized open. This directs hydraulic pressure to deploy the doors, while pressure is maintained on the latch release actuators. When fully deployed, the doors contact a switch that illuminates the DEPLOY light steadily, deenergizes the latch solenoid valve closed, and energizes the reverser lever solenoid-operated lock, which releases to allow the reverser lever to be pulled further aft to increase reverse thrust.

Reverse Thrust

When the DEPLOY light(s) illuminates and the reverser lever solenoid-operated lock(s) releases, the reverser lever(s) can be pulled further aft to increase N_1 to achieve the desired results. A second hard stop limits N_1 rpm to approximately 75%, which constitutes maximum reverse thrust.

At 60 KIAS the reverser levers should be smoothly started toward the idle deploy position.

Use of maximum reverse power below 50 KIAS could cause reingestion of exhaust gases or possible foreign object damage.

After the engines have reached reverse-idle rpm (approximately 30% N_1), the pilot can stow the reverser levers by returning them to the full forward position.

Normal Stow

Returning the respective reverser lever to the full forward and down position unlocks the main thrust lever and contacts a stow switch. This deenergizes the deploy solenoid valve closed and energizes the stow solenoid valve open, directing hydraulic pressure to stow the doors and operate the throttle-retard actuator. The "overstow" condition allows the four spring-loaded latches to lock into place and break contact with their respective latch switches. This deenergizes the stow solenoid valve closed, which shuts off hydraulic pressure to the door actuator and the throttle-retard actuator. Exhaust gas pressure and springs return the doors to their normal position against the latching hooks.

Abnormal Indications

ARM Light Fails to Illuminate during Test

If the ARM light fails to illuminate when TEST is selected on the ARM–TEST switch, the isolation valve has failed to respond correctly, hydraulic pressure is not available, or the pressure switch is faulty; also, the affected reverser will be inoperative.

ARM Light Fails to Illuminate during Normal Arming (On the Ground at Idle)

If the ARM light fails to illuminate when ARM is selected on the ARM–TEST switch (on the ground with thrust levers at IDLE), possible malfunctions are:

- Isolation valve failure
- No hydraulic pressure available
- Pressure switch failure
- Thrust lever IDLE switch failure
- Faulty squat switch relay circuitry

Steady ARM Light (ARM–TEST Switch Off)

Steady illumination of the ARM light with the ARM–TEST switch off indicates that two door latches on the same side (inboard or outboard) are unlocked. Illumination of the ARM light indicates activation of the automatic emergency stow circuit. This will be accompanied by a ***flashing*** DEPLOY light.

Flashing DEPLOY Light

A flashing DEPLOY light indicates that one or more of the door latches are unlocked.

Automatic Emergency Stow

The automatic emergency stow system is designed to prevent inadvertent deployment at any time (ARM–TEST switch off or on). If two latch position switches on the same side (inboard or outboard) indicate an unlatched condition for the doors, the result is as follows:

- The isolation valve opens, which illuminates the ARM light.
- The DEPLOY light begins to flash.
- The stow solenoid valve is energized open, which applies stow pressure to the door actuator and to the throttle retard actuator, which retards the thrust lever to the idle position.

The steady ARM light and flashing DEPLOY light remain on until the latches return to the latched position or until power is removed from the control circuits.

Automatic Throttle Retard

Automatic throttle retard is designed primarily to minimize severe thrust asymmetry which may occur as a result of inadvertent deployment of a reverser during high thrust settings. This is accomplished by use of the overstow cycle hydraulic pressure to operate a throttle retard actuator, resulting in mechanical repositioning of the thrust lever to the IDLE position.

This feature can be checked on the ground by deploying the reversers, pulling the reverser levers toward a higher power position, then quickly returning the reverser levers to the stow position and pushing forward on the thrust levers. Resistance to thrust lever movement will be felt until completion of the stow cycle.

All thrust reverser normal, abnormal, and emergency procedures are contained in the supplement section of the approved *AFM*.

Do not use thrust reversers to back up the airplane, and do not deploy the drag chute and thrust reversers simultaneously.

Adequate airplane control has been demonstrated with a 20-knot crosswind component, but this value is not considered to be limiting.

QUESTIONS

1. The TFE731-2-2B engine provides 3,500 pounds of thrust at:
   A. Sea level up to 72˚ F (22˚ C)
   B. All altitudes and temperatures
   C. Sea level at any temperature
   D. All altitudes up to 72˚ F (22˚ C)

2. The engine LP rotor (N_1) consists of:
   A. A four-stage, axial-flow compressor and a single-stage certrifugal compressor
   B. A single-stage fan and a three-stage, axial-flow compressor
   C. A single-stage fan, a four-stage, axial-flow compressor, and a three-stage, axial-flow turbine
   D. A four-stage, axial-flow compressor and a four-stage, axial-flow turbine.

3. During a normal ground start, the ignition light should come on when:
   A. N_2 reaches 10%
   B. The START–GEN switch is moved to START
   C. The thrust lever is moved to idle
   D. N_1 reaches 10%

4. The engine HP spool (N_2) consists of a:
   A. Three-stage axial compessor and a four-stage radial turbine
   B. Single-stage centrifugal compressor and a two-stage axial turbine
   C. Two-stage axial compressor and a single-stage axial turbine
   D. Single-stage centrifugal compressor and a single-stage axial turbine

5. The engine instruments (N_1, N_2 and ITT) are powered by:
   A. Self-generating tachometers
   B. The 26-VAC buses
   C. The essential buses
   D. The DC main and essential buses

6. Electrical power for engine oil pressure indication is provided by the:
   A. Left and right essential buses
   B. Inverters through the 26-VAC bus
   C. Battery charging bus
   D. Pilot's and copilot's 115-VAC buses

7. The primary engine thrust indicating instrument is the:
   A. Turbine (N_2)
   B. ITT
   C. Fan (N_1)
   D. Fuel flow

8. The maximum ITT during engine start is:
   A. 832˚ C
   B. 870˚ C for ten seconds
   C. 795˚ C
   D. 860˚ C

9. The maximum transient ITT during takeoff is:
   A. 860˚ C for five minutes
   B. 870˚ C for ten seconds
   C. 880˚ C for five seconds
   D. 865˚ C for five minutes

10. What is the maximum acceptable engine oil temperature?
    A. 140˚ C
    B. 70˚ C
    C. 130˚ C
    D. 127˚ C

11. During computer-on operation, the surge bleed valve:
    A. Is controlled by the fuel computer
    B. Remains closed
    C. Remains at 1/3 OPEN position
    D. Has no function

12. During computer-on operation, what engine overspeed protection is provided?

A. Only 109% N_1 and 110% N_2 ultimate overspeed shutoff
B. Only 105% N_2 mechanical governor
C. Only 109% N_1 ultimate overspeed shutoff
D. Only 105% N_2 mechanical governor and 109% N_1/110% N_2 ultimate overspeed shutoff

13. Which of the following statements regarding fuel control is true in the event of airplane electrical failure?:

A. Fuel control remains in the NORMAL mode, but overspeed protection is lost.
B. Fuel control reverts to the MANUAL mode, and ultimate overspeed protections is lost.
C. Fuel control reverts to the MANUAL mode, but 109% N_1 overspeed protection is still available if the computer switch is on.
D. Fuel control remains in the NORMAL mode with no loss of overspeed protection.

14. If the SPR switch is used during engine start, it should be released to OFF when:

A. ITT begins to rise.
B. ITT reaches 200˚ C
C. ITT reaches 300˚ to 400˚ C
D. Engine idle rpm stabilizes.

15. The ENG SYNC light indicates:

A. Engine sync is not turned on or has failed.
B. Engine sync is operating properly.
C. Engine sync is turned on, and the nose landing gear is locked in the DOWN position.
D. The engines are synchronized.

16. When performing a fuel control governor check, N_2 rpm increases rapidly. The pilot must:

A. Turn on the fuel computer switch immediately, allow rpm to stabilize at idle, shut down the engine, and have the system checked.
B. Pull the associated fire T-handle, set the fuel computer switch to manual, and restart the engine.
C. Wait until N_2 rpm stabilizes at 105%, then turn on the fuel computer switch, and, when N_2 drops to idle, shut the engine down.
D. Turn on the fuel computer switch, and, if the rpm drops to idle, no further action is necessary.

17. The major portion of total thrust at low altitudes is developed by the:

A. Fan
B. LP turbine
C. Core engine
D. HP turbine

18. The maximum allowable N_1 under all operating conditions is:

A. 101% to 103% maximum continuous
B. 105% for one minute
C. 103% to 105% for five seconds
D. 101.5% for five minutes

CHAPTER 8
FIRE PROTECTION

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CHAPTER 8
FIRE PROTECTION

INTRODUCTION

The Learjet 35/36 series airplanes are equipped with engine fire detection and fire-extinguishing systems as standard equipment. The systems include detection circuits which give visual warning in the cockpit and controls to activate one or both fire-extinguisher bottles. There is a test function for the fire detection system. One or two portable fire-extinguishers are provided.

GENERAL

The engine fire protection system is composed of three sensing elements, two control units (one for each engine) located in the tailcone, one warning indicator light for each engine, two fire-extinguisher bottles which are activated from the cockpit, and a fire detection circuit test switch. The fire-extinguishing system is a two-shot system; if an engine fire is not extinguished with actuation of the first bottle, the second bottle is available for discharge into the same engine. The fire bottles are located in the tailcone of the airplane. Exterior discharge indicators provide a visual indication if either fire bottle has been discharged manually or by thermal expansion.

ENGINE FIRE DETECTION AND INDICATORS

SENSING ELEMENTS AND CONTROL UNITS

Within each engine cowling are three heat-sensing elements—one mounted on the engine pylon firewall, one mounted around the lower engine accessory section, and one surrounding the engine combustion section. The elements are connected to a control unit which monitors the electrical resistance of the sensing elements. The sensing elements are made of Inconel metal tubing filled with a pliable, heat-sensitive ceramic material which, in turn, encloses a conductor wire at its center that carries the DC power through the detection circuit. The electrical resistance of the ceramic material is relatively high at normal temperatures; consequently, there is little current flow from the conductor wire through the ceramic material to ground (outer tubing). At high temperatures, however, the electrical resistance decreases and allows increased current flow.

The control unit detects the increased current flow and illuminates the red FIRE or ENG FIRE light in the T-handle when current flow equates to 890° F at the hot section sensor, or 410° F at the engine accessory and/or firewall sensors (Figure 8-1). DC essential bus electrical power for the system is supplied through the L and R FIRE DET circuit breakers on the pilot's and copilot's circuit-breaker panels.

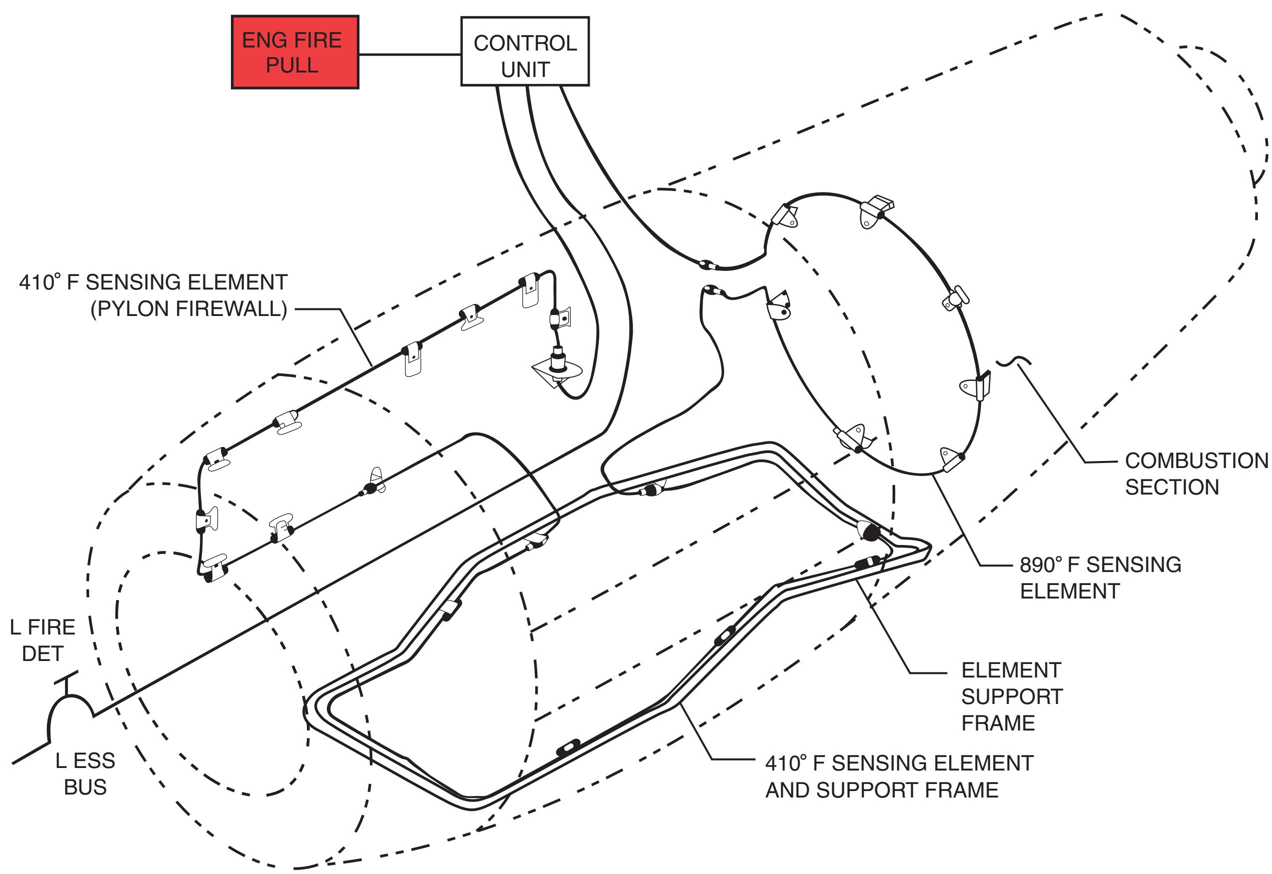

Figure 8-1. Engine Fire Detection System

FIRE AND ENG FIRE LIGHTS

The red FIRE PULL or ENG FIRE PULL warning lights are part of the T-handles, one located at each of the glareshield annunciator panel (Figure 8-2). In the event of an engine fire, the warning light in the T-handle will flash until the fire or overheat condition ceases to exist. Operation of the T-handles is explained under Engine Fire-Extinguishing.

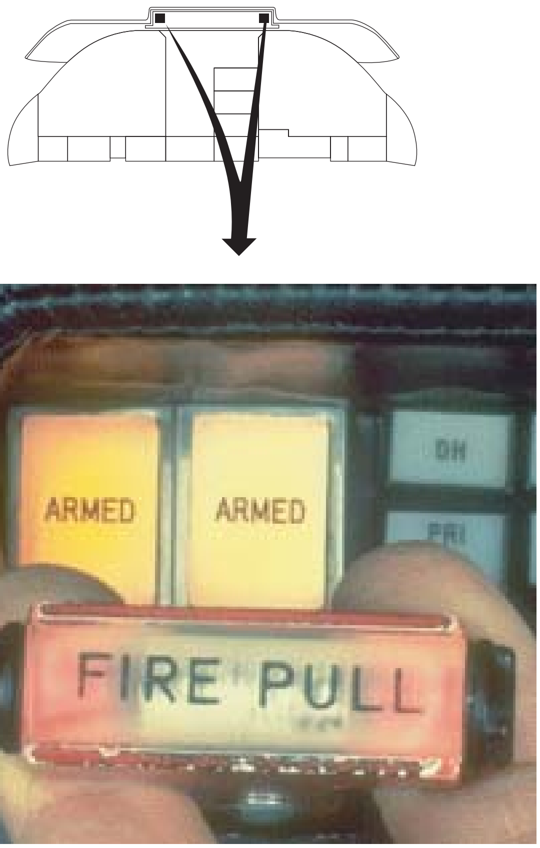

Figure 8-2. Engine Fire Warning Lights and Controls (LH)

FIRE DETECTION SYSTEM TEST

The rotary system test switch (Figure 8-3) on the center switch panel is used to test the fire detection system. Rotating the switch to FIRE DET and depressing the switch test button tests the continuity of the sensing elements and control units. A satisfactory test is indicated by both FIRE or ENG FIRE lights flashing until the test button is released.

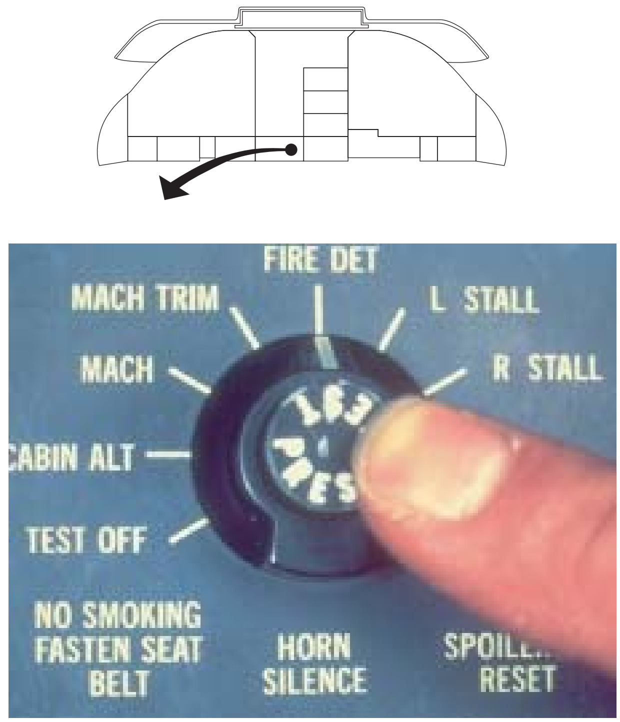

Figure 8-3. System Test Switch

ENGINE FIRE-EXTINGUISHING

EXTINGUISHER CONTAINERS

Two spherical extinguishing agent containers are located in the tailcone area. Both containers use common plumbing to both engine cowlings via shuttle valves, providing the airplane with a two-shot system. The agent used in the fire-extinguishing system is variously known as monobromotrifluoromethane, bromotrifluoromethane, or by the more common trade name of Halon 1301. It is noncorrosive, so no cleanup is necessary after use. The agent is stored under pressure, and a pressure gage is installed on each container. The pressure gages indicate approximately 600 psi at 70° F when the containers are properly serviced.

A thermal relief valve one each container is plumbed to a common discharge port (red disc) on the outside of the fuselage below the left engine pylon. The thermal relief valves will release bottle pressure at approximately 220˚ F.

FIRE OR ENG FIRE T-HANDLES AND ARMED LIGHTS

When a FIRE PULL or ENG FIRE PULL light begins to flash, it indicates a fire or overheat condition in the respective engine cowling. Following *AFM* procedures, the pilot should first place the affected engine thrust lever to CUT-OFF and then pull the corresponding T-handle. Pulling out on the T-handle closes the main fuel, hydraulic, and bleed-air shutoff valves for that engine. DC essential bus electrical power to close these valves is provided through the L and R FW SOV (firewall shutoff valve) circuit breakers on the pilot's and copilot's circuit-breaker panels, respectively.

There are two ARMED lights above each T-handle. Pulling either T-handle arms the fire extinguisher system, which is indicated by illumination of the two ARMED lights above the handle which was pulled. Depressing an illuminated ARMED light momentarily supplies DC power to the explosive cartridge which discharges the contents of one fire-extinguisher bottle and allows it to flow into the affected engine nacelle. When the ARMED light is depressed, a holding relay is also engaged which extinguishes the ARMED light, indicating that the associated bottle has been discharged. Either ARMED light may be depressed to extinguish the fire. Should one container control the fire, the other container is still available to either engine. (See Figure 8-4.)

NOTE

If the red warning light goes out, the continuity of the detection circuit should be tested using the rotary system test switch.

EXTERIOR EXTINGUISHER DISCHARGE INDICATORS

Two colored disc indicators are flush-mounted in the side of the fuselage below the left engine pylon (Figure 8-5). The red disc covers the thermal discharge port. It will be ruptured if one or both thermal relief valves have released bottle pressure. The yellow disc will be ruptured if either bottle is discharged by depressing an illuminated ARMED light. The integrity of the two discs is checked during the external preflight inspection.

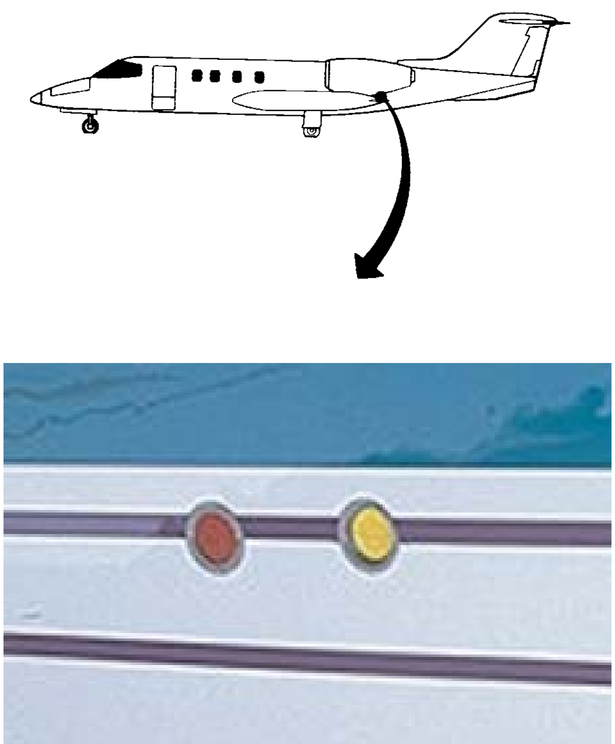

Figure 8-5. Fire-Extinguisher Discharge Indicators

PORTABLE FIRE-EXTINGUISHERS

One (standard) or two (optional) hand-held fire-extinguishers provide for interior fire protection. Location of the extinguisher(s) varies with airplane configuration.

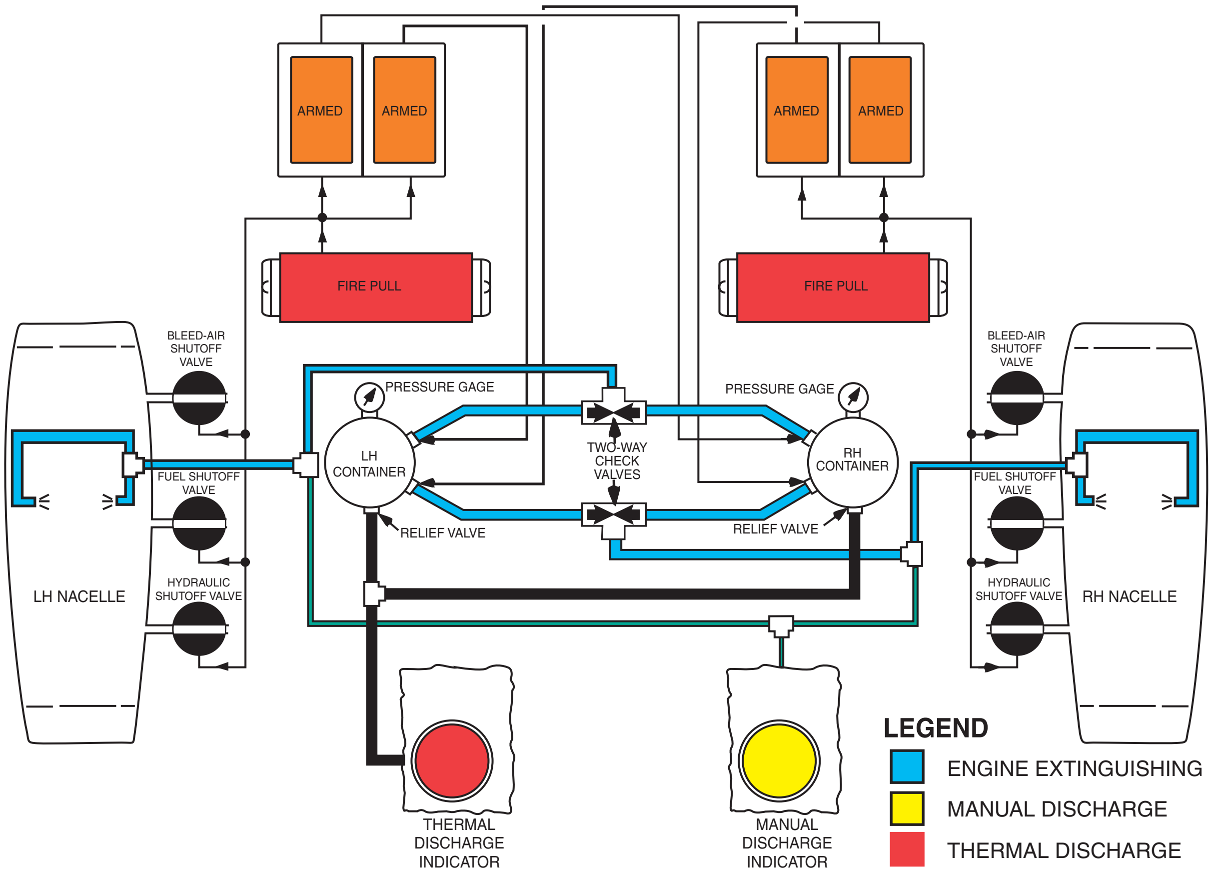

Figure 8-4. Engine Fire-Extinguishing System

Figure 8-6. Portable Fire-Extinguisher

QUESTIONS

1. Engine fire-extinguisher bottles are located in:
   A. The nacelles
   B. The engine pylons
   C. The tailcone
   D. The baggage compartment

2. The power-off preflight check of the engine fire-extinguishers includes:
   A. Checking the condition of one yellow and one red blowout disc
   B. Checking the condition of two yellow and two red blowout discs
   C. Checking blowout discs and extinguisher charge gages, all on the left side of the fuselage
   D. Activating the system TEST switch to FIRE DET

3. When the left FIRE or ENG FIRE T-handle is pulled:
   A. It discharges one extinguisher into the left nacelle.
   B. It closes the main fuel, hydraulic, and bleed-air shutoff valves for the left engine and arms both extinguishers.
   C. It discharges one extinguisher and arms the second.
   D. It ruptures the yellow discharge indicator disc.

4. When an engine fire occurs, the control unit:
   A. Arms the fire-extinguishing system
   B. Illuminates the MSTR WARN light and sounds the warning horn
   C. Automatically discharges the respective fire-extinguisher system
   D. Causes the respective FIRE or ENG FIRE light in the T-handle and both MSTR WARN lights to flash

5. The fire-extinguisher agent is discharged by:
   A. A temperature switch
   B. A mechanically fired pin at the base of the supply cylinder
   C. The FIRE T-handle electrical circuits
   D. Pushing an illuminated ARMED light

6. If fire persists after activating a fire bottle:
   A. The second fire bottle can be discharged into the affected area.
   B. The second fire bottle can only be used on an opposite-side fire.
   C. The first fire bottle can be discharged a second time.
   D. No further activation of the system is possible; both bottles discharge simultaneously when either ARMED button is pressed.

CHAPTER 9
PNEUMATICS

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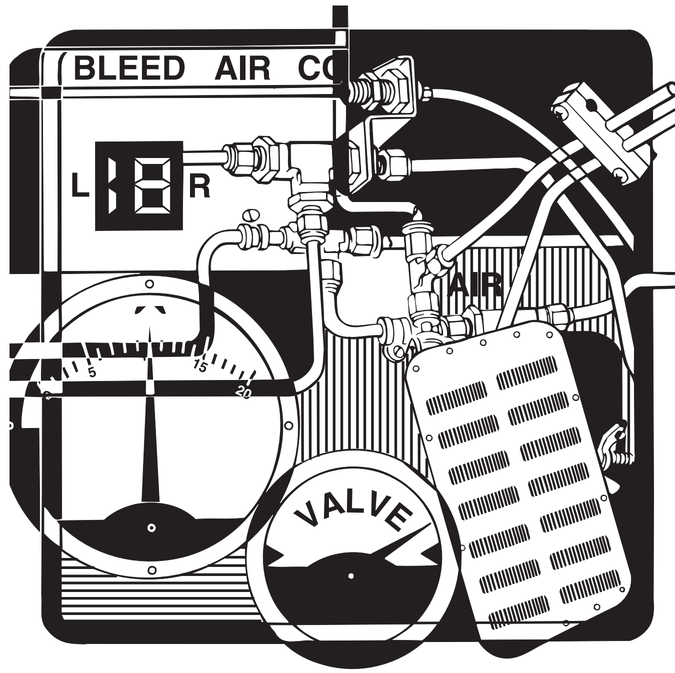
PNEUMATICS



INTRODUCTION

The airplane pneumatic system uses bleed air extracted from the engine compressor sections. It includes controls for regulation and distribution of low-pressure (LP) air from the fourth-stage axial compressor and high-pressure (HP) air from the centrifugal compressor. Pneumatic air is used for cabin pressurization and heating, anti-icing systems, hydraulic reservoir pressurization, and Aeronca thrust operation (if installed).

There are two basic pneumatic system configurations—airplane SNs 35-002 through 112, and 36-002 through 031, referred to in the text as “early” airplanes; and airplanes SNs 35-113 and subsequent, and 36-032 and subsequent which incorporate a major design change, including installation of the emergency valves, referred to as “current” airplanes.

GENERAL

Bleed air from both the LP and HP engine compressors is provided to a shutoff and regulator valve on each engine. When open, these valves regulate air pressure by selecting either LP or HP air, which is ducted to a common manifold which supplies most of the pneumatic systems. Some systems use only HP air which is tapped from the high-pressure compressor prior to the shutoff and regulator valve. Regulated bleed-air pressure is used for cabin pressurization and heating, windshield anti-icing, engine nacelle anti-icing, wing and stabilizer anti-icing, and to pressurize the hydraulic reservoir. HP air is used for fan spinner anti-icing and Aeronca thrust reversers, if they are installed. On current airplanes, HP air is used for the alcohol anti-icing system, operation of the emergency pressurization valves, as servo pressure for the cabin pressurization and temperature control systems, and for control of modulating valves on airplanes with AAK 85-6.

Control of pneumatic bleed air is accomplished with the L and R BLEED AIR switches on the copilot's lower right switch panel and by the engine FIRE T-handles. Provision is made for detection of overheat conditions within the engine pylon, the pylon bleed-air duct itself, and, on some airplanes, manifold overpressure. Visual indication is given by illumination of warning lights on the glareshield annunciator panel.

DESCRIPTION AND OPERATION

BLEED-AIR SHUTOFF AND REGULATOR VALVES

The bleed-air shutoff and regulator valves, one on each engine, are often called "mod valves" because they modulate air pressure (Figures 9-2 and 9-3). The valves are electrically controlled by the BLEED AIR switches and may also be closed by pulling the respective engine FIRE T-handle. When open, the valves operate pneumatically to maintain downstream pressure in the manifold of 27-35 psi.

Both HP and LP bleed air are available to the valves. As long as enough LP air is available to meet system demands, the valves will use only LP air. If there is not enough LP air available to meet system demands, the valves will automatically use HP air to maintain the required pressure.

The shutoff function of each shutoff and regulator valve is provided by a solenoid-operated shutoff valve which is spring-loaded open; DC power is required to close it. With loss of electrical power, the shutoff and regulator valves fall open. However, on airplane SNs 35-113 and subsequent and 36-032 and subsequent, an HP solenoid valve, which is springloaded closed, is installed (Figure 9-3). On these airplanes, if electrical power is lost, the shutoff and regulator valve fails open, but the HP solenoid valve fails closed so that only LP air will be available.

BLEED AIR SWITCHES

On airplanes SNs 35-002 through 112 and 36-002 through 031, the L and R BLEED AIR switches are located on the copilot's lower right switch panel (Figure 9-1). They are two-position, ON-OFF, switches, powered by the AIR BL circuit breaker on the left essential bus. In the ON position, the bleed-air shutoff valve (Figure 9-2) is open. In the OFF position, the valve is closed.

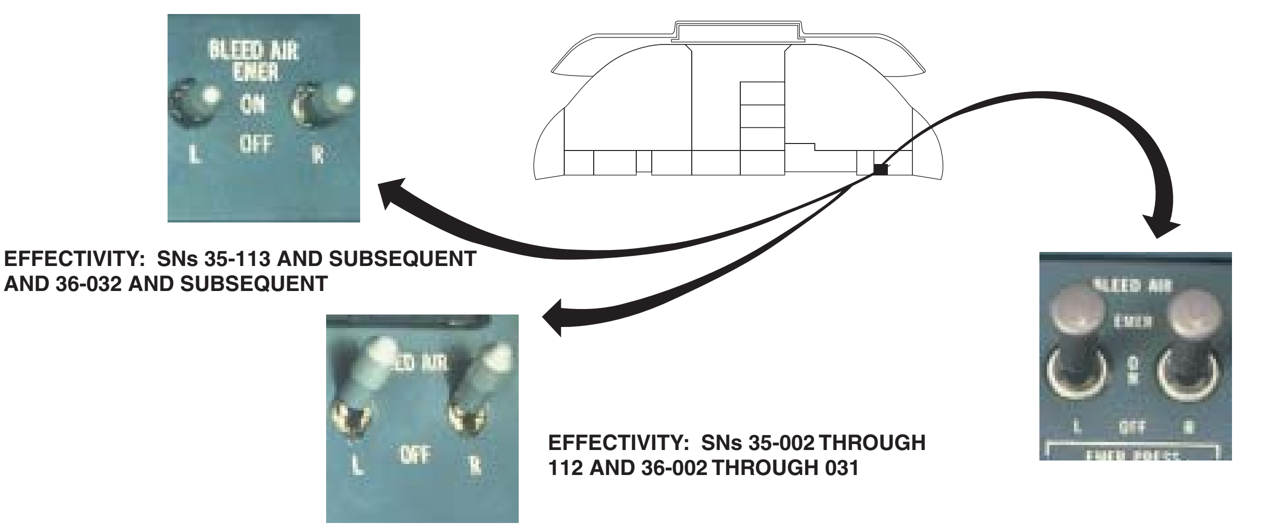
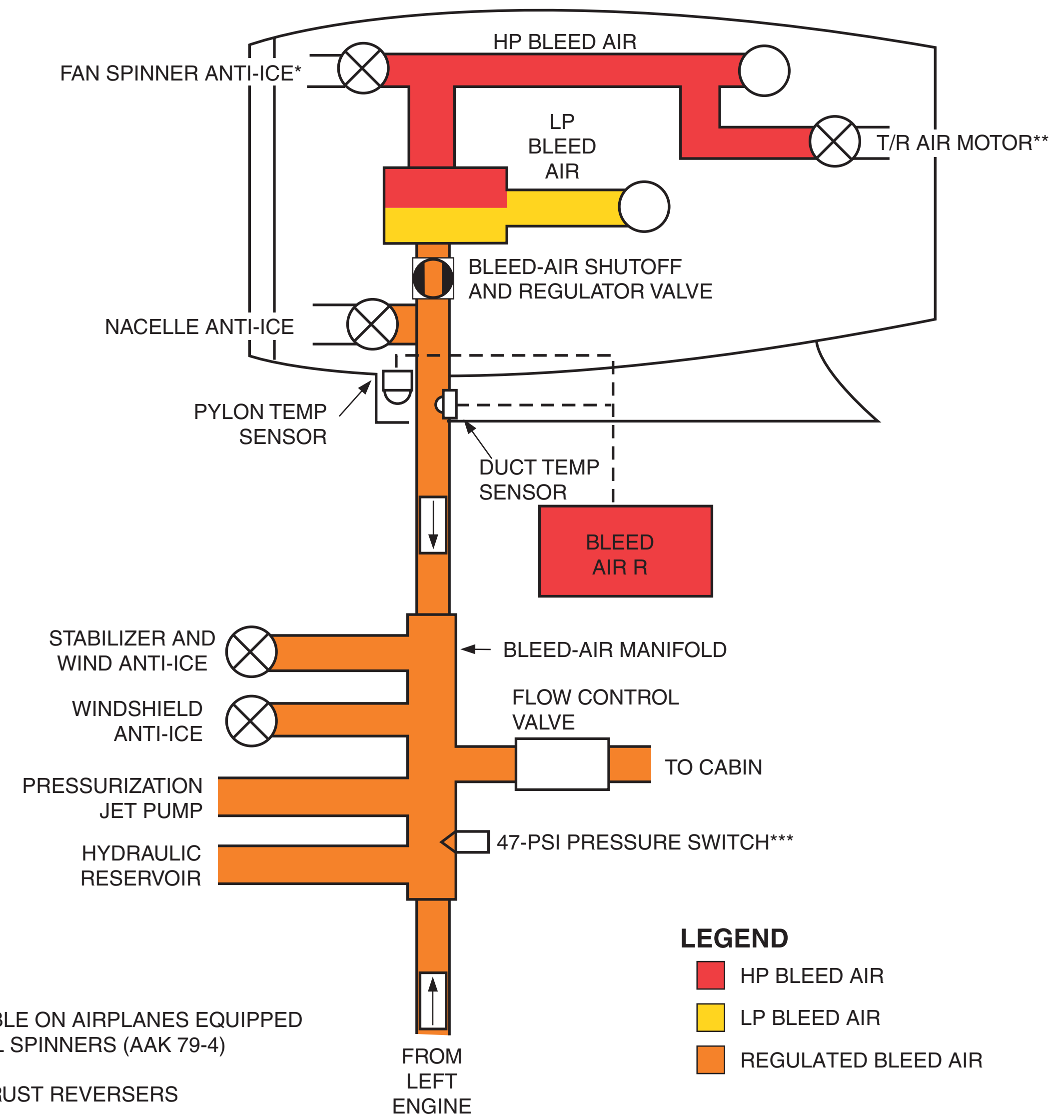


Figure 9-1. BLEED AIR Switches

On airplane SNs 35-113 and subsequent and 36-032 and subsequent, the L and R BLEED AIR switches are located on the copilot's lower right switch panel (Figure 9-1). They are three-position, OFF–ON–EMER, switches which control their respective bleed-air shutoff and regulator valves and their respective emergency pressurization valves.

In the OFF position, the bleed-air shutoff and regulator valve is closed, and the emergency pressurization valve is in its normal position (Figure 9-3). In the ON position, the bleed-air shutoff and regulator valve is open, and the emergency pressurization valve remains in its normal position. In the EMER position, the bleed-air shutoff and regulator valve is open, and the emergency pressurization valve is moved to the emergency position. At the same time, the HP solenoid valve is closed, restricting the shutoff and regulator valve output to LP air.



* NOT APPLICABLE ON AIRPLANES EQUIPPED WITH CONICAL SPINNERS (AAK 79-4)

** AERONCA THRUST REVERSERS

*** SNs 35-082, 087 THROUGH 112, AND 36-023 THROUGH 031 AND EARLIER SNs INCORPORATING AMK 76-7

Figure 9-2. Pneumatic System—SNs 35-002 through 35-112 and 36-002 through 36-031

On airplanes SNs 35-113 through 658 and 36-032 through 063, not modified by AMK 90-3, the L and R BLEED AIR switches use DC electrical power from the L and R MOD VAL circuit breakers on the left and right main DC buses. These circuit breakers provide power for control of the bleed-air shutoff and regulator valves and the emergency pressurization valves.

On airplanes SNs 35-659 and subsequent and 36-064 and subsequent, and earlier airplanes modified by AMK 90-3, the L and R BLEED AIR switches use DC electrical power from the L and R BLEED AIR and L and R EMER PRESS circuit breakers on the left and right main DC buses. The BLEED AIR circuit breakers provide power for control of the bleed-air shutoff and regulator valves. The EMER PRESS circuit breakers provide power for control of the emergency pressurization valves.

See Chapter 12, "Pressurization," for additional information on the emergency pressurization valves.

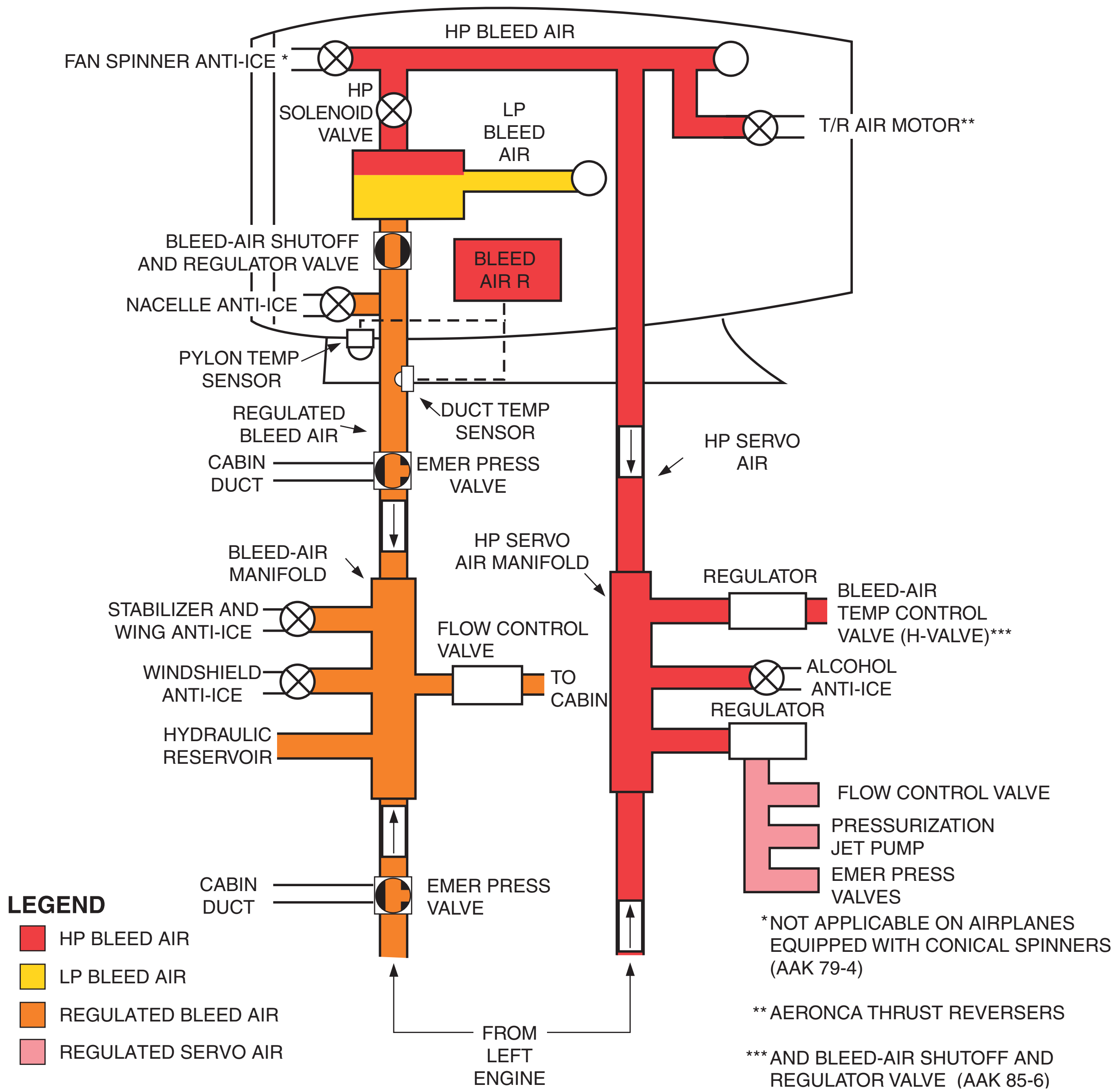


Figure 9-3. Pneumatic System—SNs 35-113 and Subsequent and 36-032 and Subsequent

BLEED-AIR CHECK VALVES

A check valve is installed in the bleed-air ducting from each engine. Each check valve allows airflow in one direction and blocks airflow applied in the opposite direction. The check valves prevent loss of bleed air during single-engine operation.

BLEED-AIR MANIFOLD

The bleed-air manifold serves as a collection point for regulated air pressure from either or both engines. From the manifold, bleed air is distributed to the flow control valve for cabin pressurization and heating, the pressurization jet pump (on early airplanes), the windshield anti-ice (defog) valve, the wing and horizontal stabilizer anti-ice pressure regulator valve, and the hydraulic reservoir regulator.

BLEED-AIR WARNING LIGHTS

The red BLEED AIR L and R warning lights on the glareshield annunciator panel illuminate when an associated pylon senor or pylon duct temperature sensor detects excessive temperatures. On some airplanes, a pressure sensor in the manifold causes both lights to illuminate for an overpressure condition.

A temperature sensor in each engine pylon operates when pylon structure temperature exceeds 250° F and illuminates the respective red L or R BLEED AIR light on the glareshield (Annunciator Panel).

A temperature sensor installed in each engine pylon bleed-air duct causes the respective red L or R BLEED AIR light to illuminate if duct temperature is excessive. Airplane SNs 35-002 through 35-064, and 36-002 through 36-017 use 590° F sensors. Later production airplanes use 645° F sensors.

On airplanes SNs 35-082, 087 through 112, and 36-023 through 031, a pressure sensor in the regulated bleed-air manifold causes both BLEED AIR warning lights to illuminate if pressure in the manifold exceeds 47 psi. This also applies to earlier airplanes incorporating AMK 76-7 (relocation of cabin air distribution flow control valve). (Figure 11-2 in Chapter 11.)

HP SERVO AIR

On airplane SNs 35-113 and subsequent, and 36-032 and subsequent (Figure 9-3), HP bleed air is tapped off the HP centrifugal compressor. The air from this tap flows through a check valve to the HP servo air manifold. From the manifold, air is ducted directly to the alcohol anti-icing system and through two regulators. The air from one regulator is used to control the position of the hot-air bypass valve (H-valve) and the bleed-air shutoff and regulator valve on airplanes modified per AAK-85-6. The other regulator provides air to (1) modulate the flow control valve, (2) control position of the emergency valves, and (3) operate the pressurization jet pump.

QUESTIONS

1. Pneumatic air is extracted from:
- A. The LP compressor
- B. The HP compressor
- C. Ram air
- D. Both A and B

2. With loss of DC electrical power, the shutoff and regulator valves:
- A. Fail closed
- B. Fair open
- C. Remain in their last position
- D. Can be closed only by pulling a FIRE/ENG FIRE T-handle

3. The L and R BLEED AIR ON–OFF switches are located:
- A. On the copilot's lower right switch panel
- B. On the left side panel
- C. On the pilot's lower left switch panel
- D. On the overhead panel

4. The temperature of the bleed air in the duct between the engine and the bleed- air manifold is monitored by the:
- A. Pylon overheat thermostat
- B. Aft fuselage equipment section thermostat
- C. Duct temperature sensor
- D. Duct overheat thermostat

5. The BLEED AIR L annunciator illuminates:
- A. When the temperature in the left pylon or the left bleed-air duct is too high
- B. When the pressure in the left pylon is below the system's operational limit
- C. When the left half of the bleed-air system is operating
- D. When the left half of the bleed-air system has failed

CHAPTER 10
ICE AND RAIN PROTECTION

CONTENTS

ILLUSTRATIONS

CHAPTER 10
ICE AND RAIN PROTECTION

INTRODUCTION

Anti-icing equipment on the Learjet 35/36 is designed to prevent buildup of ice on:

- The engine nacelle lip, early model fan spinner, and the inlet pressure-temperature probe
- The windshield and the radome
- The leading edge of the wings and the horizontal stabilizer
- Pitot probes, static ports, AOA vanes, shoulder static ports (if installed), and the total temperature (Rosemount) probe (if installed)

This system is certified for flight into known icing conditions.

GENERAL

Airplane anti-icing is accomplished through the use of electrically heated anti-ice systems, engine bleed-air heated anti-ice systems, and an alcohol anti-ice system.

Electrically heated components include pitot tubes, static ports, shoulder static ports (FC-200), the engine inlet air pressure-temperature (P_{T2} T_{T2}) sensors, stall warning vanes, and a total temperature (Rosemount) probe, if installed.

Engine bleed air is used to heat the windshields, wing and horizontal stabilizer leading edges, nacelle inlets, and the engine fan spinners on earlier airplanes with the elliptical (dome-shaped) spinners.

An alcohol system is installed for radome anti-icing and as a backup to the pilot's windshield bleed-air anti-icing.

On airplane SNs 35-643 and subsequent and 36-058 and subsequent, an auxiliary windshield defog heat system is installed.

All anti-icing equipment must be turned on before icing conditions are encountered. To delay until ice buildup is visually detected on airplane surfaces constitutes an unacceptable hazard to safety of flight.

If anti-ice systems are required during takeoff, they should be turned on prior to setting takeoff power. Appropriate takeoff power and performance charts must be used.

Icing conditions exist when there is visible moisture and the indicated ram-air temperature (RAT) is + 10° C or below. Takeoff into icing conditions is permitted with all bleed-air anti-icing systems on. The air temperature gage (RAT) should be checked frequently when flying in or entering areas of visible moisture.

During descents, the cabin altitude may increase unless sufficient engine rpm is maintained to compensate for the additional bleed-air use.

Anti-ice system switches are located on the anti-ice control panel (Figure 10-1).

ICE DETECTION

During daylight operation, ice accumulation can be visually detected on the windshield, the wing leading edges, and tip tanks.

WINDSHIELD ICE DETECTION

During night operations, the windshield ice detection lights indicate ice or moisture formation on the windshield. Two probes, one on the pilot's side of the glareshield and one on the copilot's side, contain red lights which continuously shine on the inside of the windshield surface. The ice detection lights normally shine through unseen. However, they will reflect red spots approximately 1 1/2 inches in diameter if ice or moisture has formed on the windshield.

The ice detection light on the pilot's side is inside the anti-ice airstream; the light on the copilot's side is located outside the anti-ice airstream. For this reason, the copilot's light should be monitored when flying in icing conditions (anti-icing equipment on). The ice detection lights are illuminated whenever airplane electrical power is on. The lights are powered through the L and R ICE DET circuit breakers on the pilot's and copilot's essential buses respectively.

WING ICE DETECTION

During daylight conditions, ice formation on the wing leading edges and tip tanks may be observed visually.

During darkness, the recognition light can be used to check for ice buildup.

On airplanes with the emergency light system, the wing inspection/egress light below the emergency exit, ice buildup may be detected on the inboard leading edge. On some airplanes, an optional wing ice inspection light is installed on the forward right side of the fuselage and is focused on a three-inch black dot on the wing leading edge next to the tip tank. The light is operated by a switch located on the copilot's sidewall panel (Figures 3-13 and 3-14 in Chapter 3.)

ANTI-ICE SYSTEMS

ENGINE ANTI-ICE SYSTEM (NACELLE HEAT)

The engine anti-ice system provides anti-icing for the engine nacelle inlet lips, the elliptical fan spinners, and the $P_{T2}T_{T2}$ probes. The nacelle lips are heated with regulated bleed air. The $P_{T2}T_{T2}$ probe is heated electrically. On airplane SNs 35-002 through 244 and 36-002 through 044, ***not*** incorporating AAK 79-4, the elliptical spinner is anti-iced by high pressure bleed air. On aircraft SNs 35-245 and subsequent, and 36-045 and subsequent, ***and*** earlier airplanes incorporating AAK 79-4, a conical spinner replaces the elliptical spinner and no anti-icing is required.

Nacelle Heat Switches

Each engine anti-ice system is independently controlled by the L and R NAC HEAT switches located on the anti-ice control panel (Figure 10-1).

When the NAC HEAT switch is turned on (L or R position), electrical power is supplied to heat the $P_{T2}T_{T2}$ probe. The switch also energizes the fan spinner shutoff valve open (if applicable) and deenergizes the nacelle lip shutoff valve open. Selecting the OFF position deenergizes the fan spinner shutoff valve closed and energizes the nacelle shutoff valve closed. Figure 10-2 is a schematic portrayal of the engine anti-ice systems.

DC electrical power to operate the systems is provided through the L and R NAC HT circuit breakers on the L and R main buses.

Bleed air for nacelle lip anti-icing is taken from the regulated bleed-air line just downstream from the bleed-air shutoff and regulator valve (Figure 10-2). It is ducted through the nacelle heat shutoff valve to a diffuser tube which distributes it around the inner surface of the nacelle lip and then exhausts it overboard through a hole at the bottom of the nacelle lip.

The source of fan-spinner heat is high-pressure (HP) bleed air.

Engine Ice Lights

The amber L and R ENG ICE lights on the glareshield annunciator panel (Annunciator Panel section) provide a visual indication of fan spinner or nacelle lip anti-ice system malfunction. The lights are operated by pressure switches in the associated fan spinner

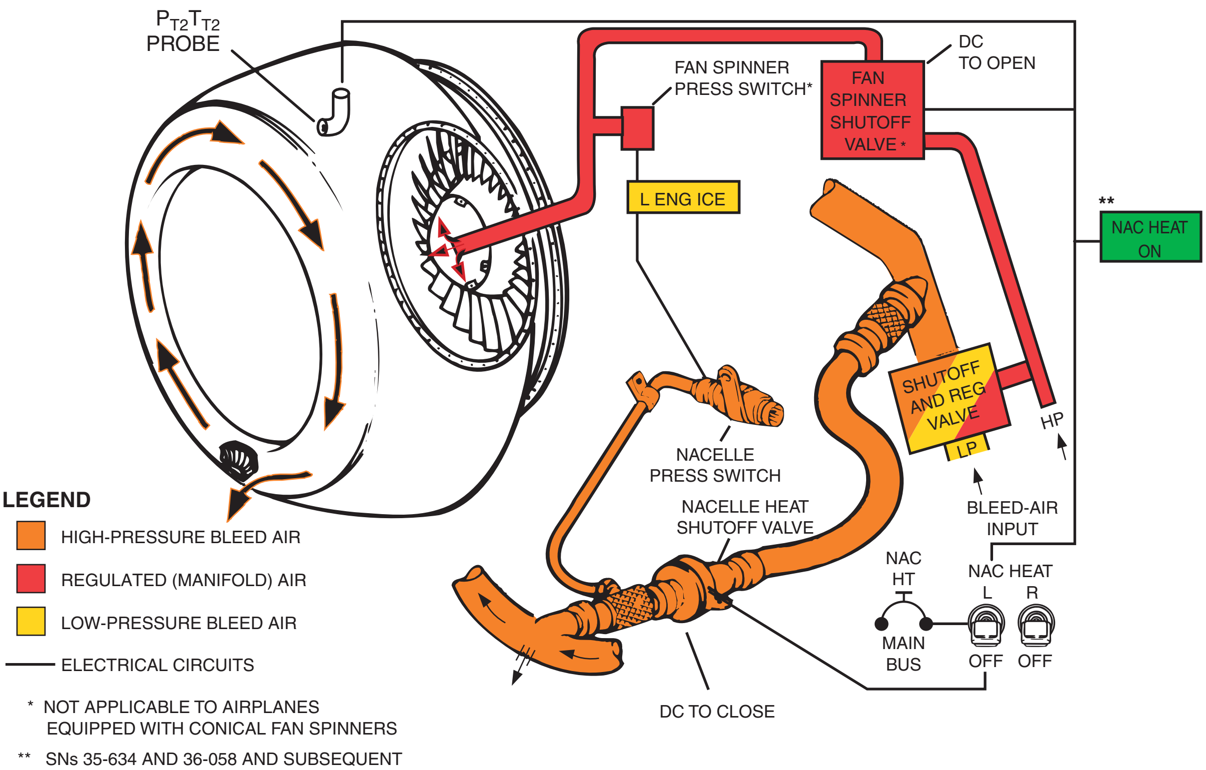


Figure 10-2. Nacelle and Fan Spinner Anti-ice Flow

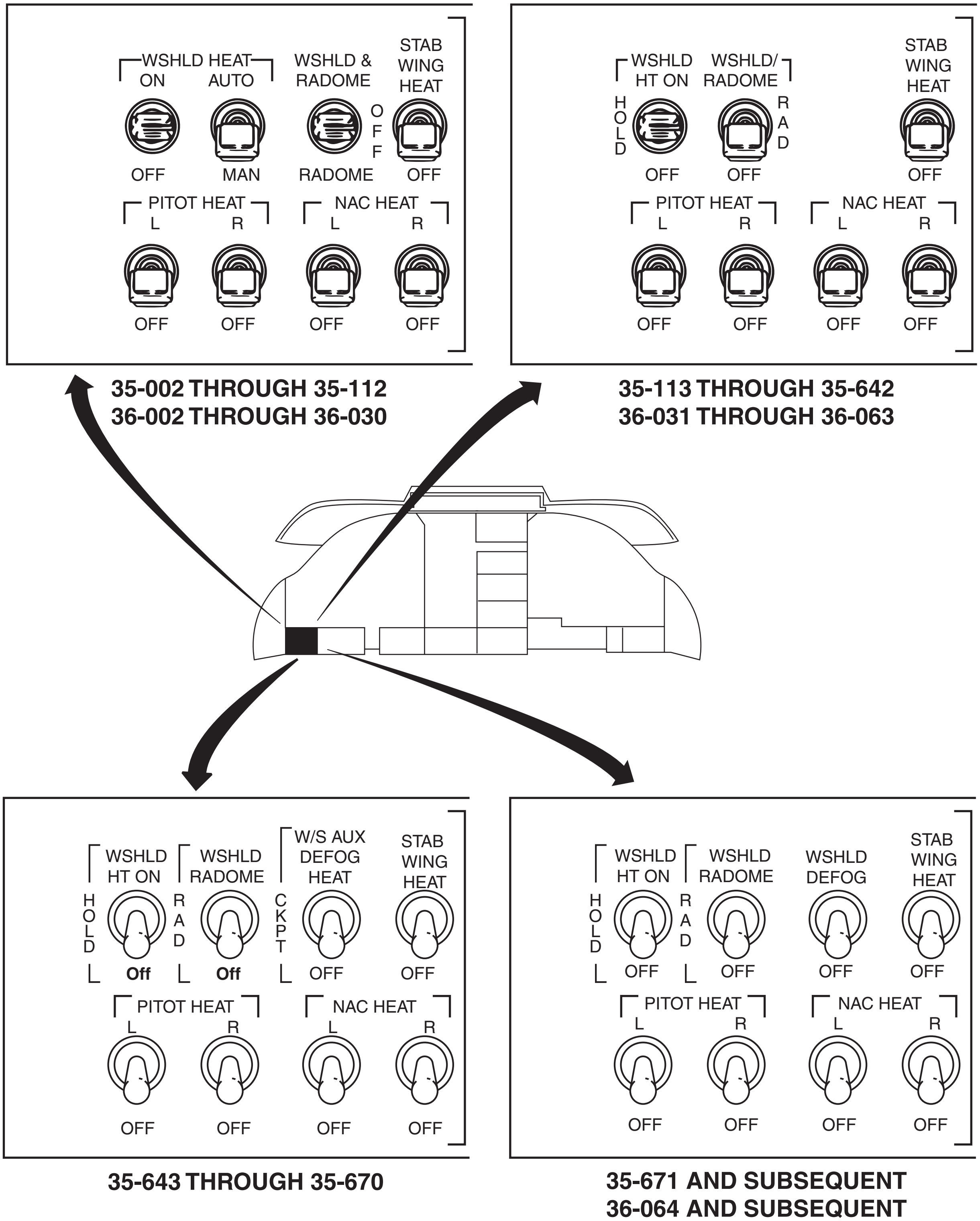


Figure 10-1. Anti-ice Control Panel

and nacelle lip bleed-air plumbing. Illumination of an ENG ICE light with the associated NAC HEAT switch on indicates that bleed-air pressure to either the fan spinner or to the nacelle lip is not sufficient to provide satisfactory anti-ice protection.

When a NAC HEAT switch is turned on or off, the respective ENG ICE light illuminates momentarily until bleed-air pressure at the pressure switch agrees with the switch command. Under some conditions, bleed-air pressure may not be sufficient at idle rpm to keep the pressure switches from illuminating the ENG ICE light. In this event, advance the thrust levers to check proper nacelle heating operation.

Illumination of either ENG ICE light NAC HEAT switches in the OFF position indicates the presence of bleed-air pressure in the nacelle lip or fan spinner plumbing due to a malfunction of the nacelle lip or fan spinner anti-ice shutoff valve. Cycling the NAC HEAT switch on and back to OFF may close the open valve.

GREEN NAC HT ON Light (Airplane SNs 35-634 and Subsequent, and SNs 36-058 and Subsequent)

A single green NAC HT ON annunciator light is installed on the glareshield annunciator panel. The light illuminates when either NAC HEAT switch is on as a reminder that the nacelle heat system is operating.

EXTERIOR WINDSHIELD DEFOG, ANTI-ICE, AND RAIN REMOVAL SYSTEM

General

There are five different systems used in the Learjet 35/36 to provide exterior windshield anti-icing, defogging, and rain removal. They will be covered individually. All systems operate on DC power from the WSHLD HT circuit breaker on the left main bus.

Airplane SNs 35-002 through 086, except 082, and 36-002 through 022, without AAK 76-7A or AMK 91-2

The exterior windshield heat/defog system can be controlled either automatically or manually (Figure 10-4). It is also used to supplement cockpit heating through the pilots' footwarmers, and to provide an alternate bleed-air source for emergency pressurization.

An IN–NORMAL/OUT–DEFOG knob, located below the instrument panel to the left of the pedestal (Figure 10-3), manually controls a valve which directs bleed-air either to the windshield or to the cockpit footwarmers.

When the knob is pushed into the IN–NORMAL position, with the windshield anti-ice on, bleed air is directed into the cockpit through the pilot's and copilot's footwarmers. This provides

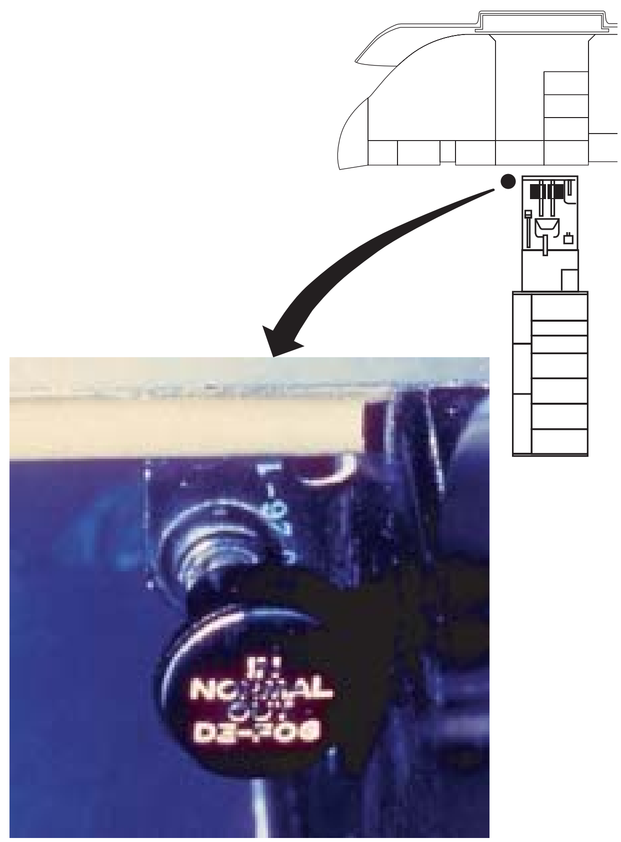


Figure 10-3. Defog Control Knob

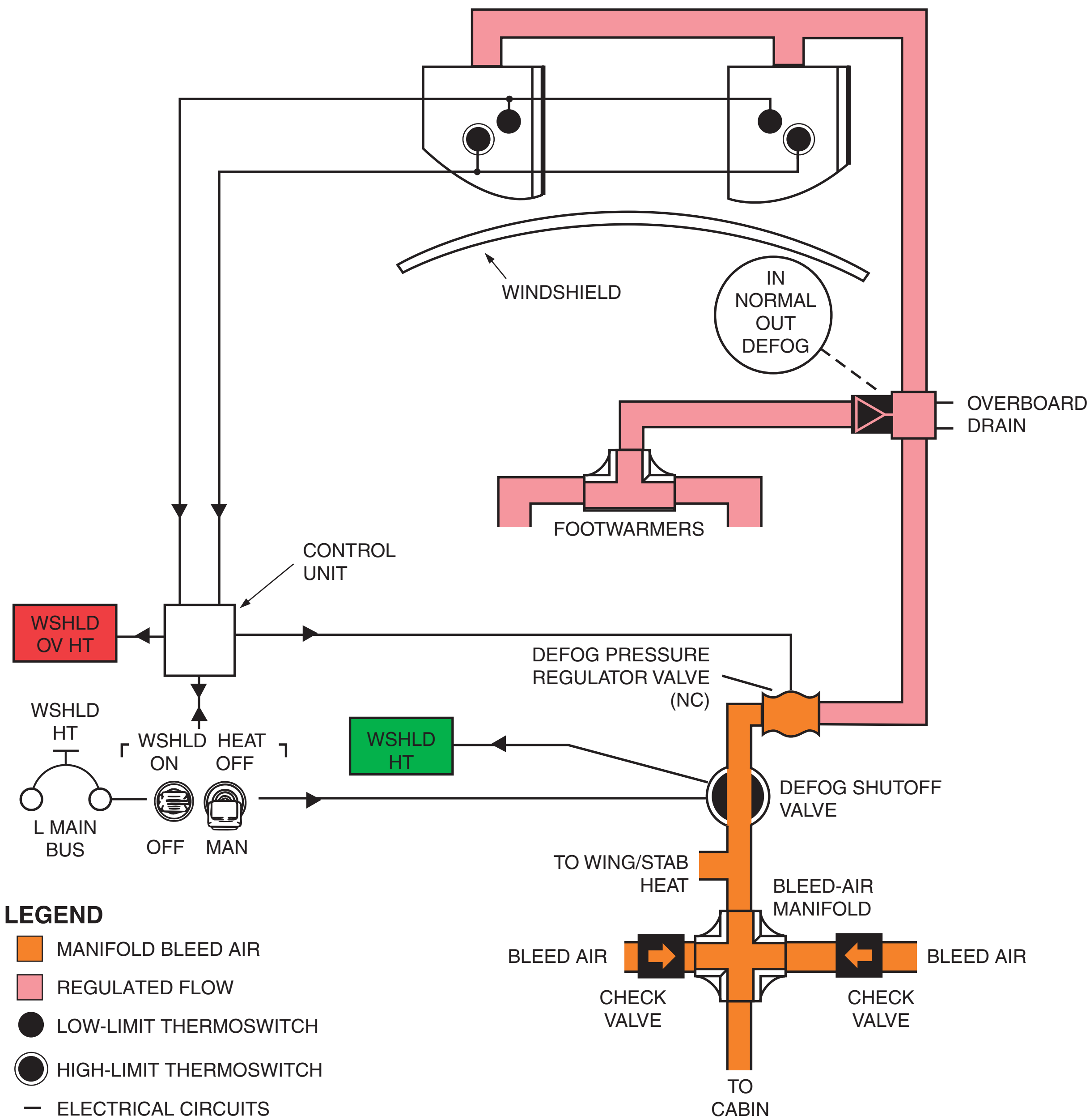


Figure 10-4. Windshield Anti-ice System (Airplane SNs 35-002 through 086, except 082, and 36-002 through 022, without AAK 76-7A or AMK 91-2)

additional heat in the cockpit and an alternate source of bleed-air flow into the cabin for emergency cabin pressurization. The knob is normally left in the IN–NORMAL position.

When the knob is pulled out to the OUT–DEFOG position, the bleed air is directed to the external windshield duct outlets for windshield defog, anti-ice, and rain removal.

Two windshield heat switches are located on the anti-ice panel. One is a three-position switch labeled “ON” and “OFF,” and is spring-loaded to the center (neutral) position. The other switch has two positions labeled “AUTO” and “MAN.”

Bleed air from the regulated bleed-air manifold is routed through two valves: the shutoff valve and the pressure-regulator valve.

The shutoff valve is motor-driven and controlled by either of the two switches on the anti-ice control panel. It takes four to five seconds to cycle fully. Selecting AUTO will open the shutoff valve and illuminate the green WSHLD HT light. The light will be on whenever the shutoff valve is not fully closed. If MAN is selected, the shutoff valve may be opened or closed with the ON–OFF switch. Since this switch is spring-loaded to neutral, it must be held in the ON position while the valve drives toward the fully open position. The switch may be released before the valve reaches full open. The shutoff valve will then stop and remain in an intermediate position. The shutoff valve can be closed only by holding the ON–OFF switch to OFF (with MAN selected) for at least four seconds.

The pressure-regulator valve is solenoid-operated and is deenergized closed. Its function is to regulate the engine bleed air from the manifold to 16 psi. It is energized open when DC electrical power is applied to the airplane and will be deenergized and closed to shut off windshield anti-ice in case of windshield overheat.

Automatic Operation

The flow of bleed air to the windshields is controlled in the auto mode by the high (250˚ F) and low (215˚F) temperature thermoswitches installed in each windshield outlet nozzle.

NOTE

AAK 77-6 provides for changing the high- and low-limit thermoswitches to 290˚ F and 250˚ F, respectively.

For ground operation, when the low-limit thermoswitch senses 215˚ F, it will close the shutoff valve, which extinguishes the green WSHLD HT light. It will also illuminate the red WSHLD OV HT light. If the low-limit switch fails, or the shutoff valve fails to close, the temperature may rise sufficiently to trigger the high-limit thermoswitch which removes power from the pressure-regulator valve. The red WSHLD OV HT light will illuminate, and the green WSHLD HT light will remain illuminated because the shutoff valve is not fully closed.

During flight, through the squat switch relay box, the low-limit switch will close the shutoff valve which extinguishes the WSHLD HT light. However, the red WSHLD OV HT light will not illuminate because the system is designed to cycle on the low-limit switches. If the high-temperature limit is reached in flight due to failure of the low-limit switches, the pressure-regulator valve will close, the red WSHLD OV HT light will illuminate, and the green WSHLD HT light will remain illuminated.

Manual Operation

Selecting MAN enables the spring-loaded ON–OFF switch to control the shutoff valve and, therefore, the amount of bleed air supplied to the windshields.

On the ground, in manual mode, a low-limit thermoswitch will illuminate the red WSHLD OV HT light, but will not close either the regulator valve or the shutoff valve. However, the high-limit thermoswitch does close the pressure-regulator valve. Therefore, an overheat condition is indicated by illumination of both the green and red lights, regardless of which limit is exceeded. In flight, the low-limit thermoswitch is disabled.

Airplane SNs 35-082, 087 through 112, and 36-023 through 031, and Earlier Airplanes Incorporating AAK 76-7A

The exterior windshield heat/defog system can be controlled either automatically or manually (Figure 10-5). It is also used to supplement cockpit heating through the pilot's footwarmers and to provide an alternate bleed-air source for emergency pressurization.

An IN–NORMAL/OUT–DEFOG knob, located below the instrument panel to the left of the pedestal (Figure 10-3), manually controls a valve which directs bleed air either to the windshield or to the cockpit footwarmers.

When the knob is pushed in to the IN–NORMAL position, with the windshield anti-ice on, bleed air is directed into the cockpit through the pilot's and copilot's footwarmers. This provides additional heat in the cockpit and an alternate source of bleed-air flow into the

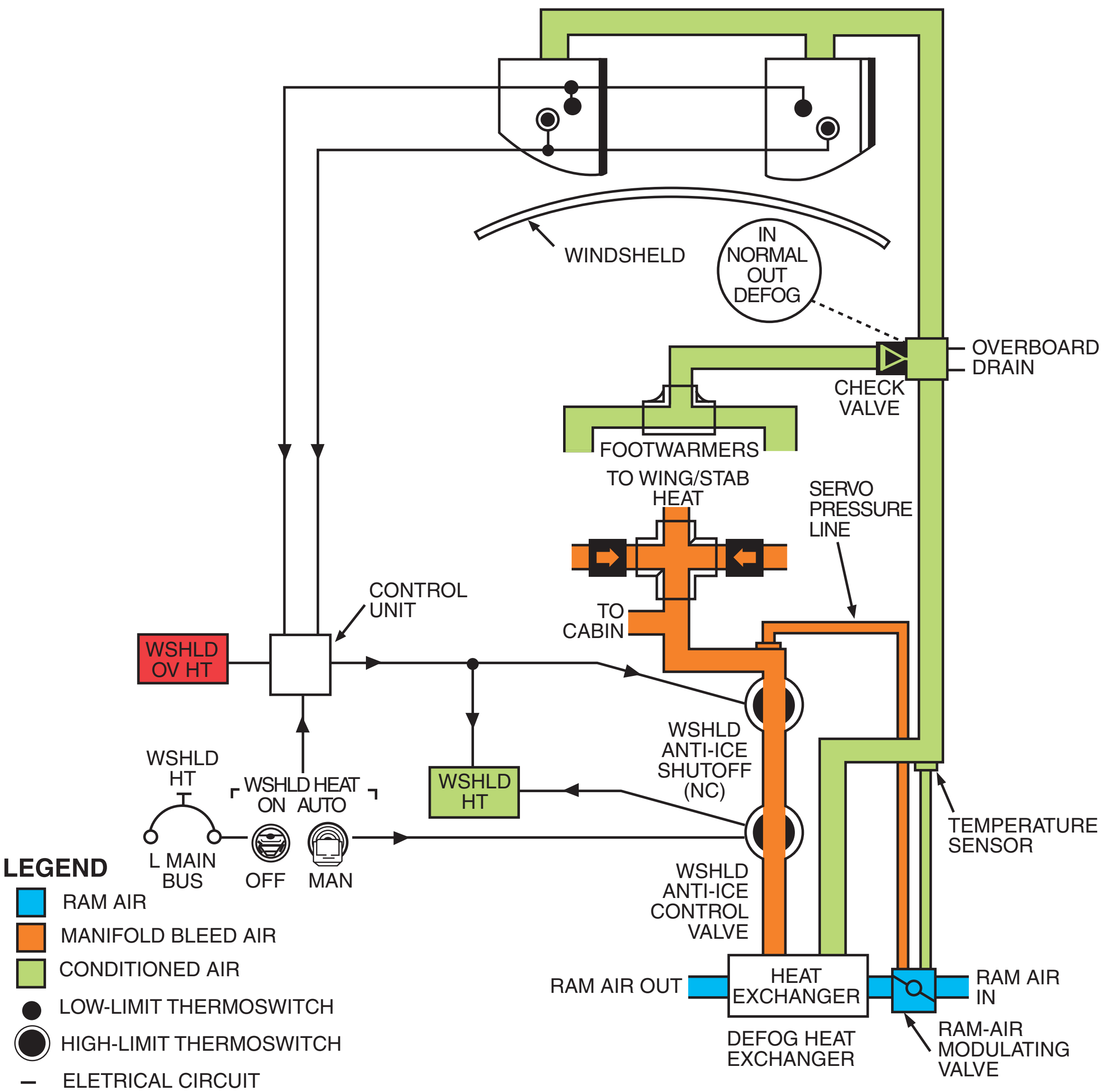


Figure 10-5. Windshield Anti-ice System (Airplane SNs 35-082, 087 through 112, 36-023 through 031, and Earlier SNs Incorporating AAK 76-7A)

cabin for emergency cabin pressurization. The knob is normally left in the IN–NORMAL position.

When the knob is pulled out to the OUT–DEFOG position, the bleed air is directed to the external windshield duct outlets for windshield defog, anti-ice, and rain removal.

Two windshield heat switches are located on the anti-icing panel. One is a three-position switch labeled “ON” and “OFF,” and is spring-loaded to the center (neutral) position. The other switch has two positions labeled “AUTO” and “MAN.”

Bleed air from the regulated bleed-air manifold is routed through two valves: the anti-ice shutoff valve and the anti-ice control valve.

The shutoff valve is solenoid-operated and is deenergized closed. Its function is to regulate the engine bleed air from the manifold to 16 psi. It is energized open when DC electrical power is applied to the airplane and will be deenergized and closed to shut off windshield anti-ice in case of windshield overheat.

The control valve is motor-driven and controlled by either of the two switches on the anti-ice control panel. It takes four to five seconds to cycle fully. Selecting AUTO will open the control valve and illuminate the green WSHLD HT light. If MAN is selected, the control valve may be opened or closed with the ON–OFF switch. Since this switch is spring-loaded to neutral, it must be held in the ON position while the valve drives toward the fully open position. The switch may be released before the valve reaches full open. The control valve will then stop and remain in an intermediate position. The control valve can be closed only by holding the ON–OFF switch to OFF (with MAN selected) for at least four seconds.

Operation

With windshield anti-ice on, bleed air flows through the open shutoff valve and anti-ice control valve, and through a heat exchanger from which it is ducted to the outlet nozzles at the base of each windshield. The anti-ice heat exchanger cools the bleed air with ram air. A ram-air modulating valve operates to maintain a 300° F duct temperature downstream of the heat exchanger by using a duct temperature sensor and a regulated bleed-air servo line. The subsequent heat loss occurring in the duct as the bleed air reaches the outlet nozzles keeps the outlet airflow temperature within the limits of windshield beat operation. During ground operation, ram air is not available to cool the bleed air.

Under normal conditions, the windshield heat bleed-air temperature is automatically controlled. However, an overheat warning system alerts the pilot and automatically shuts off windshield heat in the event of an overheat condition. A low-limit (approximately 250° F) and a high-limit (approximately 290° F) thermoswitch is installed in each windshield outlet nozzle. The low-limit switches function only on the ground and are cut out by the squat switch relay box when airborne. The high-limit switches are installed primarily to limit temperature during airborne operation, but will also function on the ground as a backup to the low-limit switches.

If either outlet nozzle temperature reaches the 250° F limit (ground) or 290° F limit (airborne), the thermoswitch will illuminate the red WSHLD OV HT light on the glareshield annunciator panel and cause the solenoid shutoff valve to close. The anti-ice control valve will remain in the position it was in, but the green WSHLD HT light will be extinguished while the solenoid shutoff valve is closed. The WSHLD OV HT light will extinguish and the shutoff valve will open again when the temperature at the thermoswitch cools. If the windshield heat has not been turned off, airflow will resume to the windshield, the green WSHLD HT light will illuminate, and the red WSHLD OV HT light will extinguish.

Through the squat switch relay box, the low-limit thermoswitches are disabled for 10 seconds after touchdown. This prevents automatic shutoff of bleed air at the moment of touchdown, which could restrict the pilot’s vision due to loss of rain-removal capability.

Airplane SNs 35-113 through 662 and 36-032 through 063, without AMK 91-2)

The WSHLD HT switch controls flow of engine bleed air to the exterior of the windshield for anti-icing, defogging, and rain removal (Figure 10-6). This three-position switch is labeled "ON," "HOLD," and "OFF," and is located on the anti-ice control panel.

Engine bleed air from the regulated bleed-air manifold is routed through two valves: the anti-ice shutoff valve and the anti-ice control valve. The shutoff valve is solenoid-operated and is deenergized closed. It is

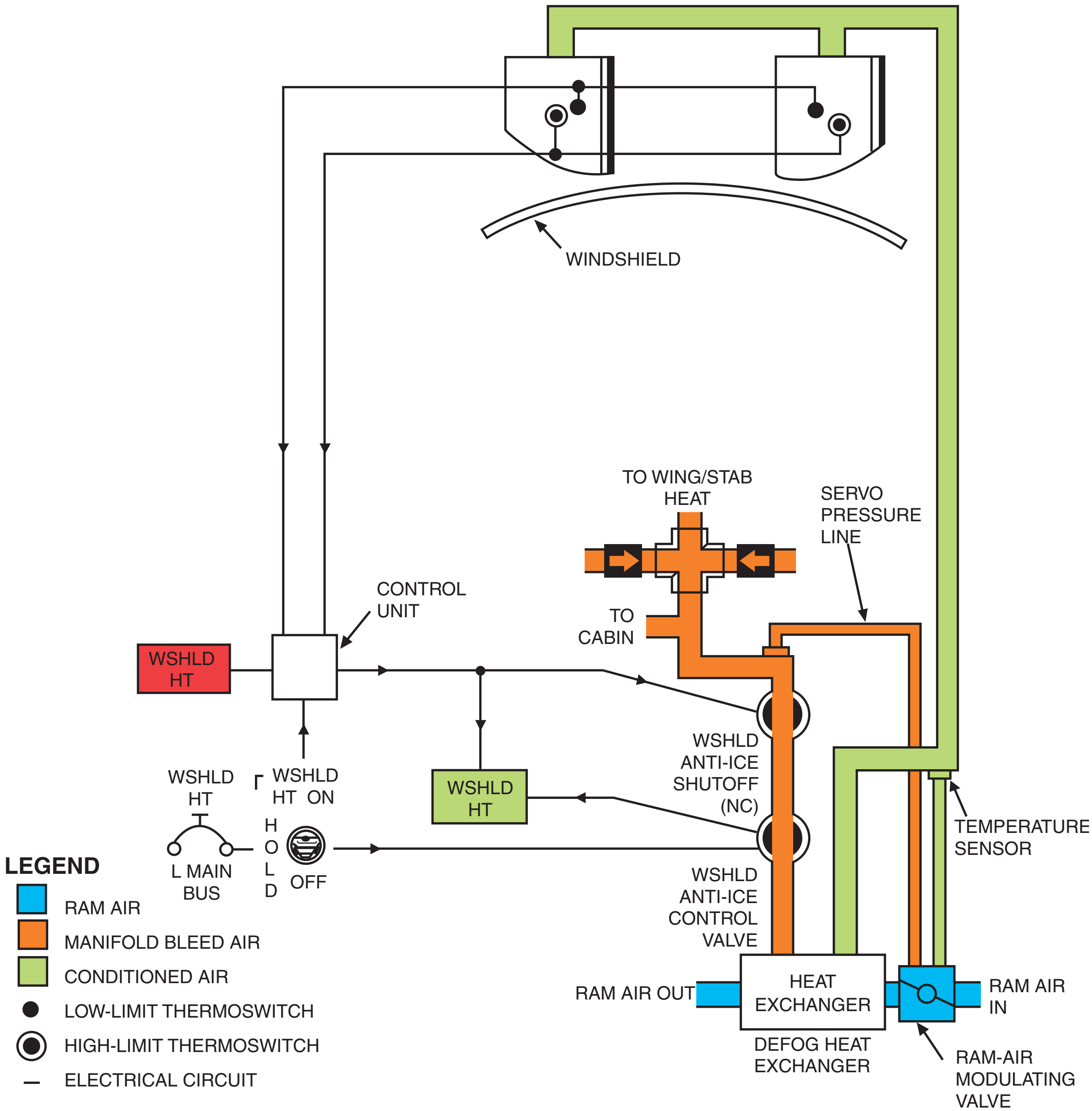


Figure 10-6. Windshield Anti-ice System (Airplane SNs 35-113 through 662, and 36-032 through 063, without AMK 91-2)

energized open whenever DC electrical power is applied to the airplane. The control valve is motor-driven and is controlled by the WSHLD HT switch.

When the WSHLD HT switch is positioned to ON, the anti-ice control valve begins to open and the green WSHLD HT light on the glareshield annunciator panel illuminates. The control valve drives to the fully open position within five to eight seconds after the WSHLD HT switch is turned to ON.

For reduced airflow to the windshield, the control valve may be stopped at any intermediate position by positioning the WSHLD HT switch to HOLD.

With both valves open, bleed air flows through a heat exchanger from which it is ducted to the outlets at the base of each windshield. The anti-ice heat exchanger cools the bleed air with ram air. A ram-air modulating valve operates to maintain a 300° F duct temperature downstream of the heat exchanger by using a duct temperature sensor and a regulated bleed air servo line. The subsequent heat loss occurring in the duct as the bleed air reaches the outlet nozzles keeps the outlet airflow temperature within the limits of windshield heat operation. During ground operation, ram air is not available to cool the bleed air.

Under normal conditions, the windshield heat bleed-air temperature is automatically controlled. However, an overheat warning system alerts the pilot and automatically shuts off windshield heat in the event of an overheat condition. A low-limit (approximately 250° F) and a high-limit (approximately 290° F) thermoswitch is installed in each windshield outlet nozzle. The low-limit switches function only on the ground and are cut out by the squat switch relay box when airborne. The high-limit switches are installed primarily to limit temperature during airborne operation, but will also function on the ground as a backup to the low-limit switches.

If either outlet nozzle temperature reaches the 250° F limit (ground) or 290° F limit (airborne), the thermoswitch will illuminate the red WSHLD OV HT light on the glareshield annunciator panel and cause the solenoid shutoff valve to close. The anti-ice control valve will remain in the position it was in, but the green WSHLD HT light will be extinguished while the solenoid shutoff valve is closed. The WSHLD OV HT light will extinguish and the shutoff valve will open again when the temperature at the thermoswitch cools. If the WSHLD HT switch has not been turned off, airflow will resume to the windshield, the green WSHLD HT light will illuminate, and the red WSHLD OV HT light will extinguish.

Through the squat switch relay box, the low-limit thermoswitches are disabled for 10 seconds after touchdown. This prevents automatic shutoff of bleed air at the moment of touchdown, which could restrict the pilot's vision due to loss of rain-removal capability.

Bleed air is not available for windshield anti-icing with both the left and right emergency pressurization valves in the emergency position.

Airplane SNs 35-663 and Subsequent, 36-063 and Subsequent, SNs 35-113 through 662, and 36-032 through 062 Incorporating AMK 91-2

The exterior windshield defog, anti-ice, and rain removal system is shown in Figure 10-7.

With the engines running and the BLEED AIR switches ON, engine bleed air from the regulated bleed-air manifold is available to two windshield anti-ice system valves: the anti-ice shutoff valve and the anti-ice control valve. The shutoff valve is solenoid-operated and is normally energized open whenever electrical power is applied to the airplane. The control valve is motor-driven and is controlled by the WSHLD HT switch.

The three-position (OFF–HOLD–ON) WSHLD HT switch is located on the anti-ice control panel. When the WSHLD HT switch is positioned to ON, the anti-ice control valve

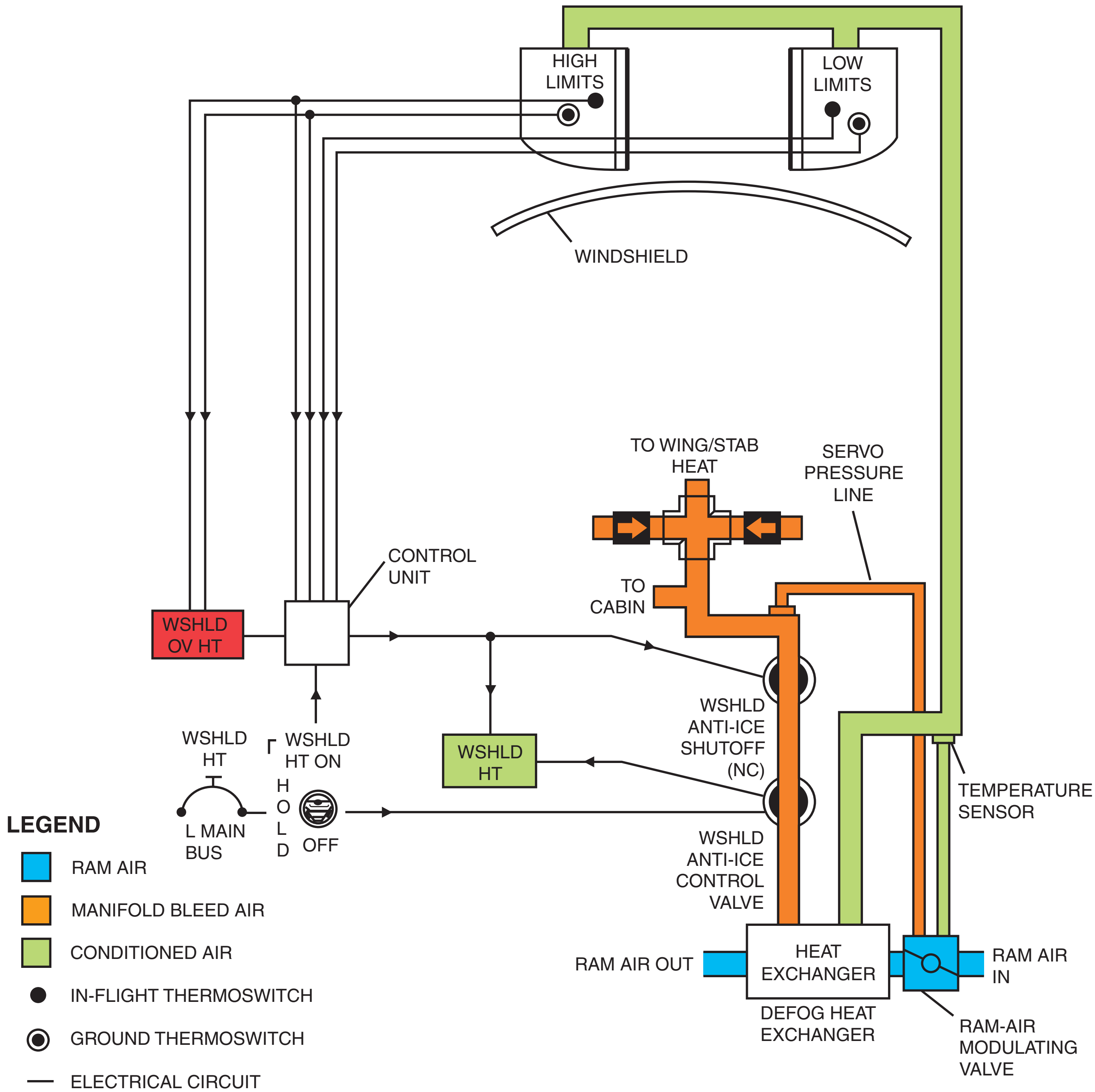


Figure 10-7. Windshield Anti-ice System (Airplane SNs 35-663 and Subsequent, 36-064 and Subsequent, SNs 35-113 through 662, and 36-032 through 063 Incorporating AMK 91-2)

begins to open, and the green WSHLD HT light on the glareshield annunciator panel illuminates. If the WSHLD HT switch is left in the ON position, the control valve will drive full open in approximately five to eight seconds. For reduced airflow to the windshield, the WSHLD HT switch may be positioned to HOLD before the control valve reaches full open. The control valve will then stop and remain in an intermediate position.

With both valves open, regulated engine bleed air flows through a heat exchanger in which it is cooled by ram-air. The ram-air flow is controlled by a pneumatically actuated modulating valve. The modulating valve senses bleed-air temperature, downstream of the heat exchanger, through a temperature sensor, and positions itself automatically to maintain an air temperature of approximately 300˚ F. From the heat exchanger, the temperature controlled bleed air is directed forward and dispensed over the outside of both the pilot's and copilot's windshields through outlets at the base of each windshield.

Normally, the windshield anti-ice bleed-air temperature is maintained at a safe level by the ram air modulating valve. However, an automatic shutdown and warning system has been provided to prevent windshield damage from an overheat condition. The system uses signals from four thermoswitches, two under the windshield heat air outlets at the base of each windshield.

One thermoswitch on each side operates only on the ground while the other operates on the ground and in the air. High-limit thermoswitches are located on the left side and low- limit thermoswitches are on the right.

If the bleed-air temperature at the windshield reaches a low limit (250˚ F in flight or 215˚ F on the ground), the anti-ice shutoff valve is deenergized closed and the green WSHLD HT light is extinguished. When the overheat cools, the thermoswitches will reset and the anti-ice shutoff valve will reopen. If the anti-ice control valve is still open, the green WSHLD HT light will illuminate and windshield anti-ice airflow will be restored.

If the bleed-air temperature at the windshield reaches a high limit (270˚ F in flight or 250˚ F on the ground; 215˚ F on the ground on airplanes with electrically heated windshields), the anti-ice shutoff valve is deenergized closed, the green WSHLD HT light is extinguished, and the red WSHLD OV HT light illuminates. When the overheat cools, the thermoswitches will reset, the red WSHLD OV HT light extinguishes, and the anti-ice shutoff valve will reopen. If the anti-ice control valve is still open, the green WSHLD HT light will illuminate and windshield anti-ice airflow will be restored.

The ground limit thermoswitches are disabled for approximately 10 seconds after landing. This prevents automatic shutoff of bleed air, which could restrict the pilot's visibility due to loss of rain-removal, if the outlet temperature is between the in-flight and ground limits at the moment of touchdown.

With loss of electrical power, the windshield anti-icing system will be inoperative since the anti-ice shutoff valve will be deenergized and will close. The control valve will remain in its last position.

Bleed air is not available for windshield anti-icing with both the emergency pressurization valves in the emergency position.

Airplane SNs 35-002 through 112 and 36-002 through 031 Incorporating AAK 76-7A and AMK 91-2

The exterior windshield heat/defog system can be controlled either automatically or manually (Figure 10-8). It is also used to supplement cockpit heating through the pilot's footwarmers, and to provide an alternate bleed-air source for emergency pressurization.

An IN–NORMAL/OUT–DEFOG knob, located below the instrument panel to the left of the pedestal (Figure 10-3), manually controls a valve which directs bleed air either to the windshield or to the cockpit footwarmers.

When the knob is pushed in to the IN–NORMAL position, with the windshield anti-ice on, bleed air is directed into the cockpit through pilot's and copilot's footwarmers. This provides additional heat in the cockpit and an alternate source of bleed-air flow into the cabin for emergency cabin pressurization. The knob is normally left in the IN–NORMAL position.

When the knob is pulled out to the OUT–DEFOG position, the bleed air is directed to the external windshield duct outlets for windshield defog, anti-ice, and rain removal.

Two windshield heat switches are located on the anti-icing panel. One is a three-position switch labeled "ON" and "OFF," and is spring-loaded to the center (neutral) position. The other switch has two positions labeled "AUTO" and "MAN."

Bleed air from the regulated bleed-air manifold is routed through two valves: the anti-ice shutoff valve and the anti-ice control valve.

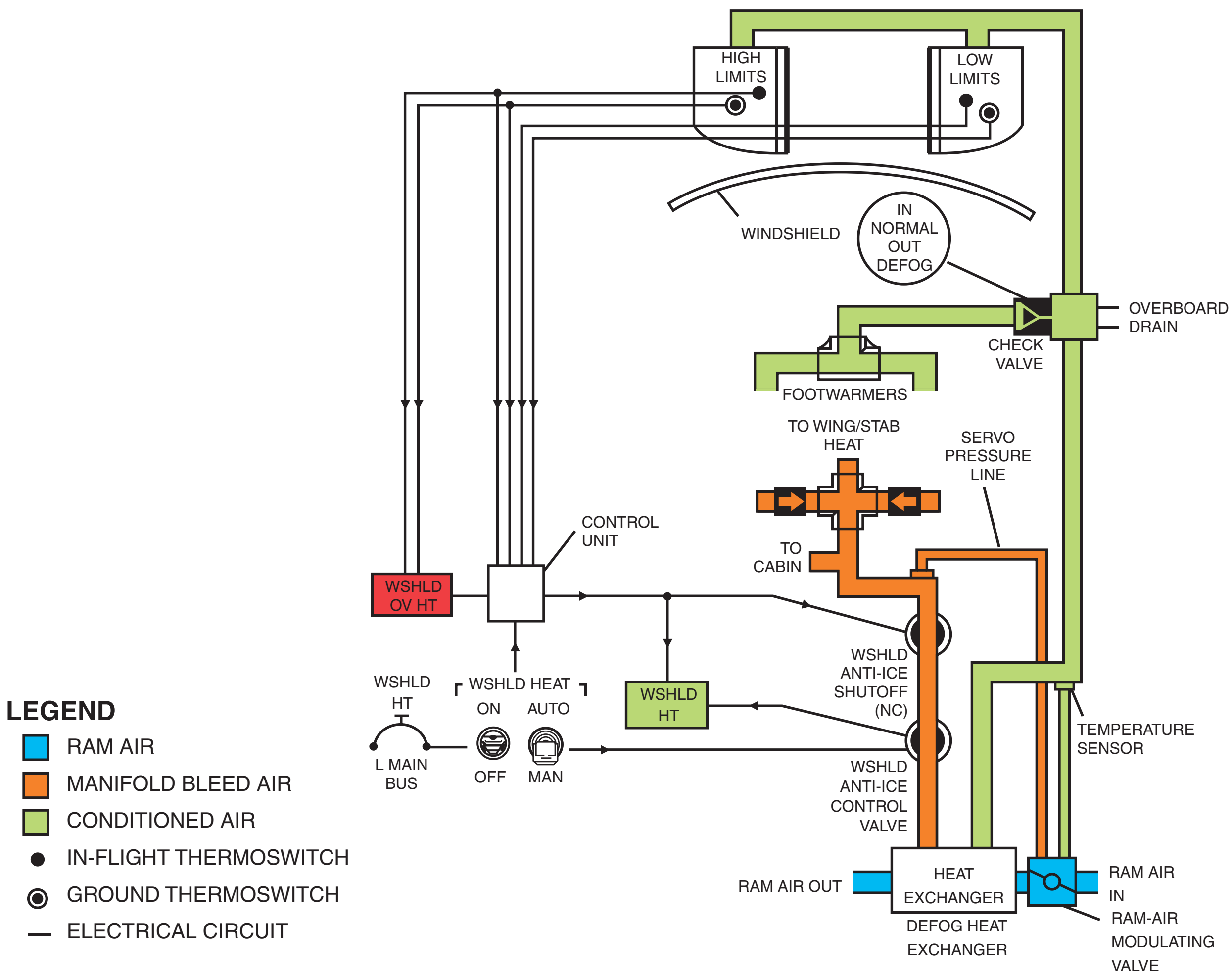


Figure 10-8. Windshield Anti-ice System (Airplane SNs 35-002 through 112 and 36-002 through 031 Incorporating AAK 76-7A and AMK 91-2)

The shutoff valve is solenoid-operated and is deenergized closed. Its function is to regulate the engine bleed air from the manifold to 16 psi. It is energized open when DC electrical power is applied to the airplane and will be deenergized and closed to shut off windshield anti-ice in case of windshield overheat.

The control valve is motor-driven and controlled by either of the two switches on the anti-ice control panel. It takes four to five seconds to cycle fully. Selecting AUTO will open the control valve and illuminate the green WSHLD HT light. If MAN is selected, the control valve may be opened or closed with the ON–OFF switch. Since this switch is spring-loaded to neutral, it must be held in the ON position while the valve drives toward the fully open position. The switch may be released before the valve reaches full open. The control valve will then stop and remain in an intermediate position. The control valve can be closed only by holding the ON–OFF switch to OFF (with MAN selected) for at least four seconds.

Operation

With both valves open, regulated engine bleed air flows through a heat exchanger in which it is cooled by ram air. The ram-air flow is controlled by a pneumatically actuated modulating valve. The modulating valve senses bleed-air temperature downstream of the heat exchanger through a temperature sensor and positions itself automatically to maintain an air temperature of approximately 300˚ F. From the heat exchanger, the temperature-controlled bleed air is directed forward and dispensed over the outside of both the pilot's and copilot's windshield through outlets at the base of each windshield.

Normally, the windshield anti-ice bleed-air temperature is maintained at a safe level by the ram-air modulating valve. However, an automatic shutdown and warning system has been provided to prevent windshield damage from an overheat condition. The system uses signals from four thermoswitches, two under the windshield heat air outlets at the base of each windshield.

One thermoswitch on each side operates only on the ground while the other operates on the ground and in the air. High-limit thermoswitches are located on the left side and low-limit thermoswitches are on the right.

If the bleed-air temperature at the windshield reaches a low limit (250˚ F in flight or 215˚ F on the ground), the anti-ice shutoff valve is deenergized closed and the green WSHLD HT light is extinguished. When the overheat cools, the thermoswitches will reset and the anti-ice shutoff valve will reopen. If the anti-ice control valve is still open, the green WSHLD HT light will illuminate and windshield anti-ice airflow will be restored.

If the bleed-air temperature at the windshield reaches a high limit (270˚ F in flight or 250˚ F on the ground; 215˚ F on the ground on airplanes with electrically heated windshields), the anti-ice shutoff valve is deenergized closed, the green WSHLD HT light is extinguished, and the red WSHLD OV HT light illuminates. When the overheat cools, the thermoswitches will reset, the red WSHLD OV HT light extinguishes, and the anti-ice shut-off valve will reopen. If the anti-ice control valve is still open, the green WSHLD HT light will illuminate and windshield anti-ice airflow will be restored.

The ground limit thermoswitches are disabled for approximately 10 seconds after landing. This prevents automatic shutoff of bleed air, which could restrict the pilot's visibility due to loss of rain-removal, if the outlet temperature is between the in-flight and ground limits at the moment of touchdown.

With loss of electrical power, the windshield anti-icing system will be inoperative since the anti-ice shutoff valve will be deenergized and will close. The control valve will remain in its last position.

INTERNAL WINDSHIELD DEFOG

General

All airplanes use conditioned engine bleed air for internal windshield defog (See Chapter 11, "Air Conditioning," for additional information). On late model airplanes, auxiliary internal windshield defog systems have been provided.

Internal Windshield Defog (Airplane SNs 35-643 through 670)

The internal windshield defog system on these airplanes uses an electrically heated coil, in the bleed-air duct leading into the cockpit, and the Freon air-conditioning system. It is controlled by a three-position (OFF–CKPT–W/S AUX DEFOG HEAT) switch on the anti- ice control panel.

To avoid damage to the electrically heated coil, the crew should ensure that adequate bleed-air flow is available in the duct to cool the coil before using the auxiliary windshield defog system.

Positioning the switch to CKPT applies DC power to the coil, heating all the air coming into the cockpit.

Positioning the switch to W/S AUX DEFOG HEAT again applies DC power to the coil, heating all the air coming into the cockpit. It also arms the Freon air-conditioning system so it will turn on automatically as the airplane descends through 18,000 feet. When the Freon air-conditioning system turns on, electrically actuated diverter doors on the cabin blower assembly will open and direct the cold air into the space between the cabin headliner and the fuselage skin. This dehumidifies the cabin air without lowering the cabin temperature excessively. (See Chapter 11 for additional information on the Freon air-conditioning system.)

DC electrical power to heat the auxiliary windshield defog coil is provided by the battery charging bus through two, 20-amp current limiters. DC control power for the auxiliary windshield defog system is provided by the AUX DEFOG circuit breaker on the left essential A bus.

Internal Windshield Defog (Airplane SNs 35-671 and Subsequent, and 36-064 and Subsequent)

The internal windshield defog system on these airplanes is shown in Figure 10-9. It uses 163-VAC power from the auxiliary and secondary inverters and is controlled by a two-position (OFF–WSHLD–DEFOG) switch located on the anti-ice control panel (Figure 10-1).

When the switch is positioned to WSHLD DEFOG, DC control power is applied to a windshield defog relay box. The relay box receives 163-VAC power, through 5-amp current limiters, from the auxiliary and secondary inverters and directs it to the heating elements in the windshield. Each heating element is a thin, gold film laminated in the windshield. The auxiliary inverter powers the element on the left side and secondary inverter powers the element on the right side.

Both heating elements are turned on and off together, but, once operating, the two elements are controlled separately by the relay box.

Two temperature sensors are located on each side of the windshield. One sensor is set to look for a windshield temperature of approximately 110˚ F. When the windshield reaches 110˚ F, the sensor will signal the relay box, which removes electrical power from the heating element on that side. As the temperature cools, the relay box will reapply power to maintain a constant windshield temperature of approximately 110˚ F.

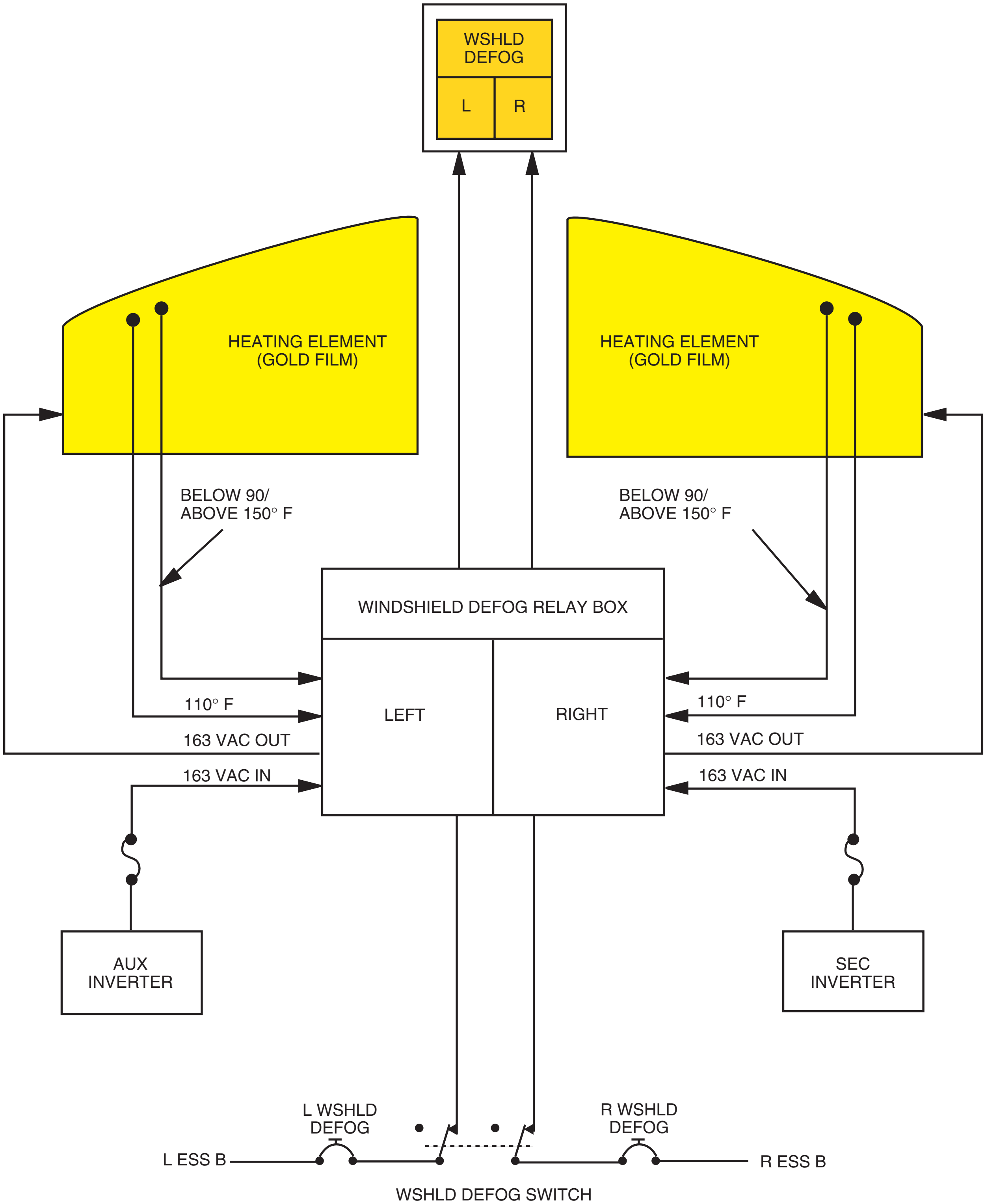


Figure 10-9. Electric Windshield Defog System—Models 35-671 and Subsequent and 36-064 and Subsequent)

The second sensor will signal the relay box in the event of an underheat or an overheat condition. If the windshield temperature is approximately 90˚ F or below, or approximately 150˚ F or above, the sensor will signal the relay box. In either situation, the relay box will illuminate an amber WSHLD DEFOG annunciator light. If an overheat condition exists, the relay box will also remove electrical power from the heating element in the affected windshield.

The difference between an overheat or underheat temperature condition may be determined by touching the windshield. If an overheat temperature condition is suspected, and the windshield does not cool off, the relay box has not removed electrical power from the heating element and the system should be turned off.

A windshield temperature of 90˚ F or below is common when the defog system is first turned on, and the annunciator light will illuminate. However, the light should soon extinguish as the windshield warms up.

The WSHLD DEFOG annunciator light, located to the left of the left ENG FIRE PULL T-handle, consists of three separate lights and is controlled by the windshield defog relay box. The upper WSHLD DEFOG light will illuminate when either of the lower lights illuminate. The lower L and R lights will illuminate to indicate which side of the windshield has malfunctioned.

The WSHLD DEFOG annunciator light will illuminate in the event of an underheat or overheat condition, as explained above. It will also illuminate with loss of DC or AC electrical power if the defog system switch is in the WSHLD DEFOG position.

The electric windshield defog system uses 163-VAC power as explained previously. DC control power for the system is provided by the L and R WSHLD DEFOG circuit breakers on the L and R essential B buses.

WINDSHIELD/RADOME ALCOHOL ANTI-ICE SYSTEM

General

Methyl alcohol from a reservoir located in the left side of the nose compartment is provided to prevent ice formation on the radome and, if necessary, the ***pilot's*** windshield as a backup for the windshield anti-ice (defog) system. The systems are operated by DC power from the right main bus.

There are two different systems in use.

Airplane SNs 35-002 through 112 and 36-002 through 031

A DC motor-driven pump supplies filtered alcohol from a 2 1/4-gallon reservoir to the radome only, or to the radome and pilot's windshield, depending on the position selected on the WSHLD/RADOME switch on the pilot's anti-icing control panel.

When the switch is positioned to RAD, the pump is energized and alcohol is delivered to the radome only due to a normally-closed solenoid valve in the windshield supply line. In this case, a fully serviced reservoir should dispense alcohol for approximately 1 hour and 30 minutes.

When the switch is positioned to WSHLD/RADOME, the pump is energized and the solenoid valve in the windshield supply line is energized open so that alcohol is delivered to both surfaces. Flow to the windshield is dispensed through an orifice assembly integrated with the pilot's defog outlet. In this case, duration is reduced to approximately 43 minutes.

A pressure switch installed in the radome supply line actuates the amber ALC AI light when the reservoir is empty or if the pump fails. The light will extinguish when the control switch is turned off (Figure 10-10).

The reservoir is vented through an open vent tube located in the same area as the pitot-static drains on the left-hand side of the nose compartment. A pressure relief valve operates to relieve excessive supply line pressure by returning it to the reservoir. Some airplanes are equipped with a siphon-break valve to prevent the siphoning of fluid from the tank after the system has been turned off (Figure 10-10).

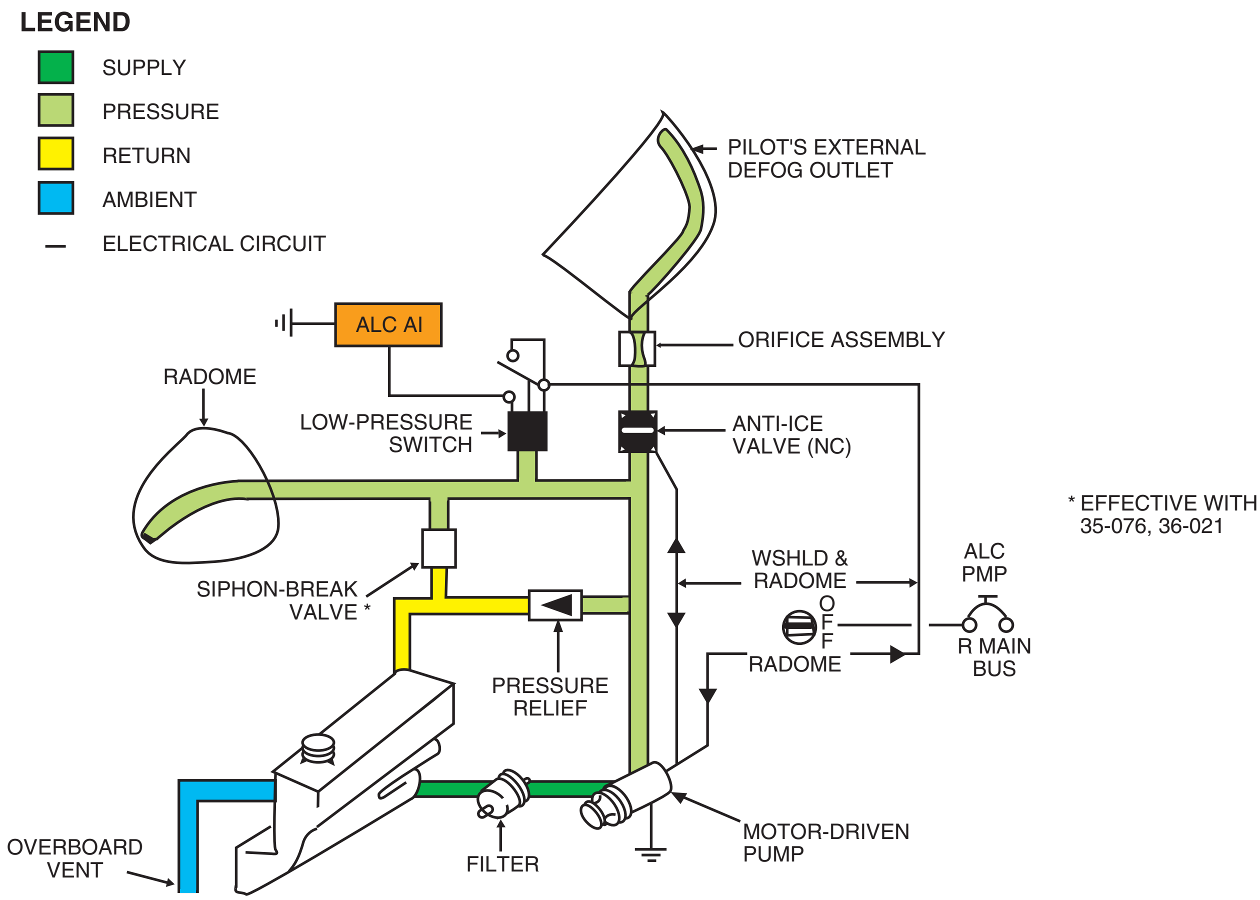


Figure 10-10. Alcohol Anti-ice System (Airplane SNs 35-002 through 112 and 36-002 through 031)

Airplane SNs 35-113 and Subsequent and 36-032 and Subsequent

Methyl alcohol is stored in a 1.75-gallon reservoir. When the cockpit control switch is positioned to WSHLD/RADOME or to RAD, circuits are completed to position a 3-way valve in the fluid supply line (Figure 10-11) and also to open the shutoff valve and pressure regulator in the servo bleed-air supply line.

Servo bleed air tapped from the high-pressure bleed-air manifold passes through the shutoff valve and pressure regulator where it is regulated to 2.3 psi and sent to pressurize the alcohol reservoir.

The alcohol is forced through a filter to the three-way valve which is positioned according to the selected switch position.

The pressure relief valve, set at 2.6 psi, relieves any overpressure in the reservoir should the pressure regulator fail, and bleeds off residual pressure when the control switch is turned off.

The float switch in the reservoir illuminates the ALC AI annunciator when the tank is empty. The light stays on even if the switch is off as a reminder to service the reservoir.

If the RAD position is selected, a fully serviced reservoir supplies only the radome with approximately 2 hours and 9 minutes of alcohol. When selected to the WSHLD/RADOME position, alcohol is also dispensed to the pilot's defog outlet via the three-way valve, and duration of the supply is reduced to approximately 45 minutes. This system is still operational if both emergency pressurization valves are in emergency (provided DC power is available).

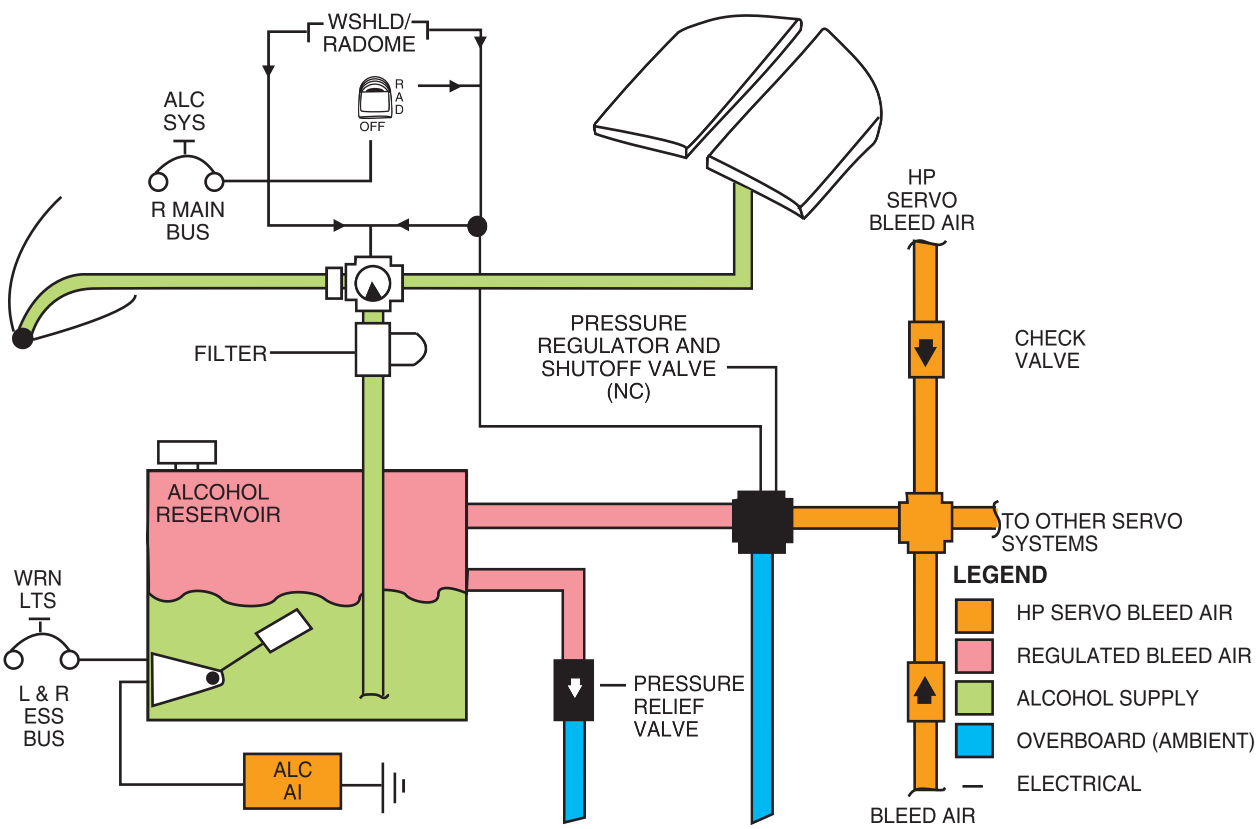


Figure 10-11. Alcohol Anti-ice System (Airplane SNs 35-113 and Subsequent and 36-032 and Subsequent)

WING AND HORIZONTAL STABILIZER ANTI-ICE SYSTEM

General

Bleed air is used to prevent ice formation on the wing and horizontal stabilizer leading edges. The bleed air is directed from the regulated engine bleed-air manifold through a solenoid-operated pressure regulator valve (Figure 10-12) to the respective leading-edge surfaces.

Controls and Indications

The STAB WING HEAT switch located on the pilot's anti-icing control panel controls the valve. When the switch is moved up (on), the valve is energized open. With the switch off, or if DC power fails, the valve deenergizes closed.

With the valve open, manifold bleed air is then routed through the wing-stabilizer pressure-regulator valve where it is regulated

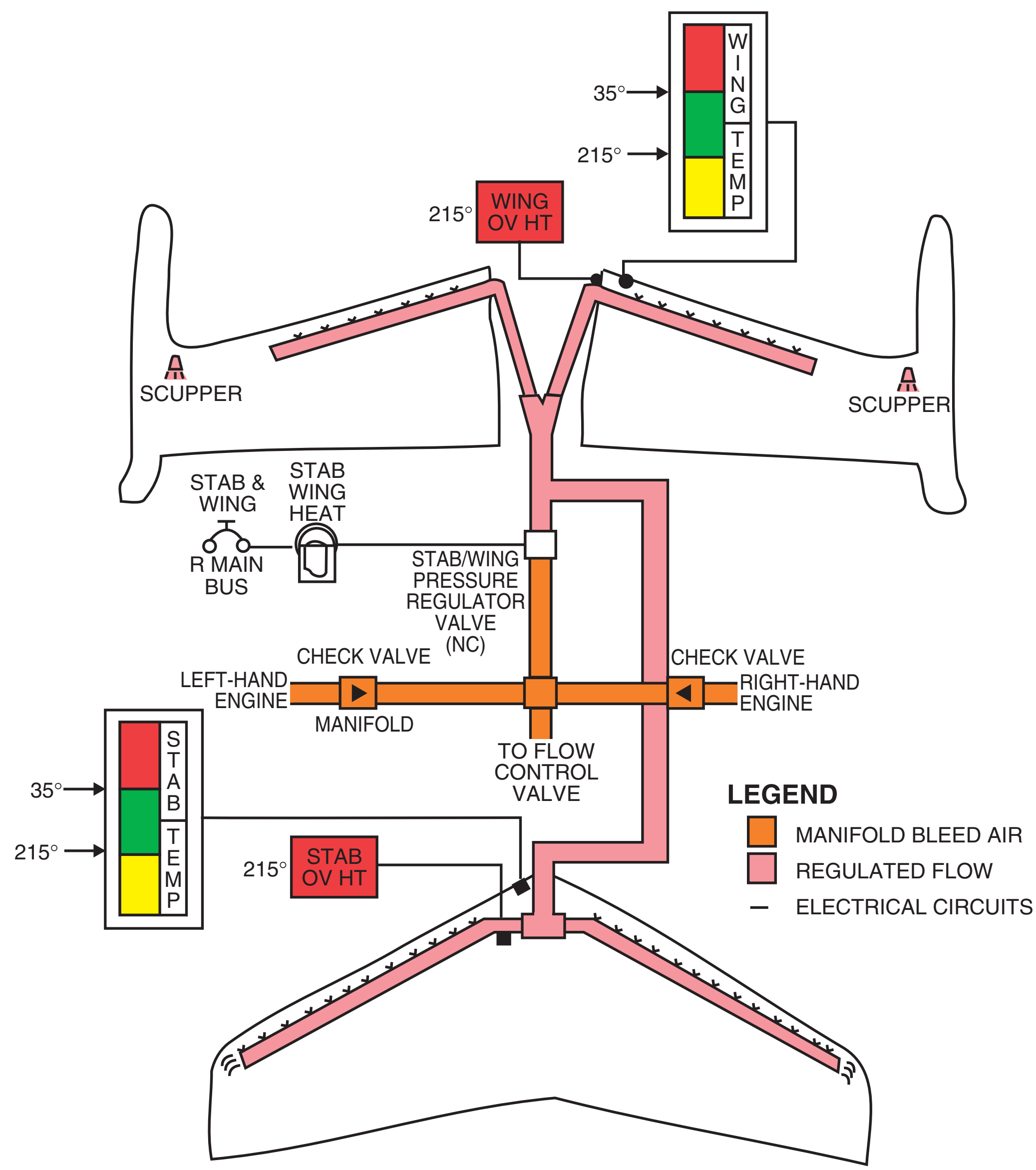


Figure 10-12. Wing and Horizontal Stabilizer Anti-ice System

to 16 psi, and finally to piccolo tubes in the leading edges of the wing and the horizontal stabilizer. After the bleed air has heated its respective leading edge, it continues outboard where it is vented overboard; each wing has a scupper vent on the underside of the leading edge, while the horizontal stabilizer has holes at each tip.

On the glareshield annunciator panel, red WING OV HT and STAB OV HT lights are illuminated should their respective sensors (Figure 10-12) detect 215° F.

Separate WING TEMP and STAB TEMP indicators, located on the center instrument panel (Figure 10-13), indicate leading-edge skin temperature and are color-coded as follows:

Red—Temperature below 35° F (danger of icing in visible moisture)

Green—Temperature between 35–215° F (normal operation)

Yellow—Temperature above 215° F (possible overheat)

When either overheat light comes on, and the system is turned off, the light will remain on until the temperature drops to within limits. The STAB WING HEAT switch may be turned back on, but the pilot must visually monitor the applicable skin temperature indicator and cycle the system on and off to maintain temperature in the green arc.

Stabilizer heat and wing heat are not available when both emergency pressurization valves (if installed) are in EMERGENCY. This is covered in Chapter 12, "Pressurization." DC power for system operation is through the STAB & WING HT circuit breaker on the right main bus.

PITOT, STATIC, AND ANGLE-OF-ATTACK VANE ANTI-ICE SYSTEM

Pitot and Angle-of-Attack Vane Anti-icing

The left and right pitot tubes and angle-of-attack (AOA) vanes contain electrical heating elements. The L and R PITOT HEAT switches located on the pilot's anti-icing control panel (Figure 10-1) each supply essential bus power to both respective heating elements. Even though each set of heating elements is controlled by the same switch, separate circuit protection for the AOA vane heater is provided; the L and R PITOT HT circuit breakers (for pitot heaters) and S WRN HT circuit breakers (for the AOA vane heaters) are all located on the left and right essential buses, respectively.

On FC 530 airplanes, one heating element in each pitot-static probe heats all of the pitot and static ports.

Dual amber L and R PITOT HEAT monitor lights are available as an optional feature and are located on either outboard side of the glareshield panel or on the instrument panel. On airplane SNs 35-271 and 36-045 and subsequent, a single amber PITOT HT light is standard equipment and is located on the annunciator warning light panel (Annunciator Panel section). In either case, the light(s) illuminates when the pitot heat switches are turned off or to indicate failure of power to a pitot tube element (the AOA vanes are not monitored).

Static Port Heating (FC 200 Only)

There are five static ports—two on the left side fuselage and three on the right side. Pilot instruments are supplied static pressure by the interconnected left front and right center ports which are heated. The interconnected left rear and right front static ports supply the copilot's static pressure and are also heated.

The right rear port, interconnected with an alternate port inside the nose compartment, is used by the altitude controller and does not require heat.

Two additional shoulder-static ports, located forward of the windshield, are also heated. These ports are used by the air data sensor.

All static port heating elements are connected directly to their respective L or R PITOT HT circuit breakers. Consequently, they are heated whenever airplane DC power is available, provided the circuit breakers are closed (in).

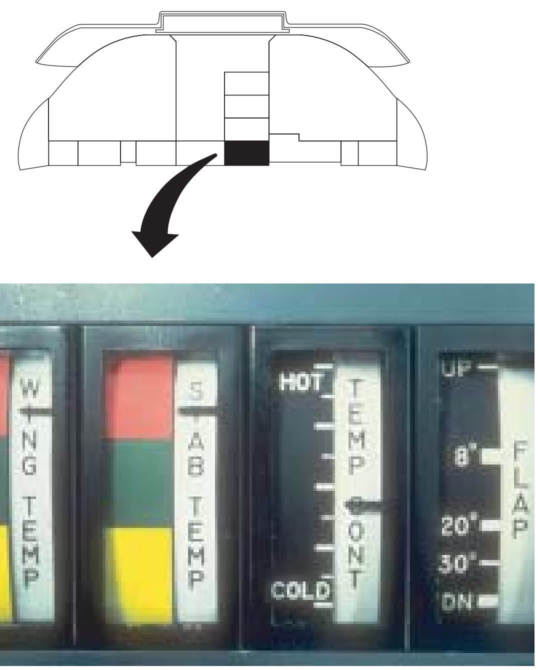


Figure 10-13. WING TEMP and STAB TEMP Indicators

QUESTIONS

1. Bleed air is used for anti-icing on:
   A. Pitot tubes and static ports
   B. P_{T2} T_{T2} sensors
   C. Wing and horizontal stabilizer leading edges
   D. Conical fan spinners

2. The L or R PITOT HEAT switches also supply heating element power for:
   A. The angle-of-attack vanes
   B. The shoulder static ports
   C. The instrument static ports
   D. P_{T2} T_{T2} probe heater

3. The crew action required when the red WING OV HT light illuminates is:
   A. No action is required; the system is automatic.
   B. Position the STAB WING HEAT switch to STAB.
   C. Turn the STAB WING HEAT switch to OFF or reduce power.
   D. Turn one BLEED AIR switch to OFF until the light goes out.

4. The internal windshield defog system uses:
   A. 230-VAC power
   B. 163-VAC power
   C. An electrically heated coil and the Freon air-conditioning system
   D. Engine bleed-air pressure

5. Anti-icing equipment must be turned on:
   A. When in icing conditions
   B. Before entering icing conditions
   C. Before takeoff
   D. During climbout

6. With the loss of airplane electrical power, anti-icing will be lost on:
   A. All systems
   B. Pitot, static, and P_{T2} T_{T2} probes only
   C. All systems ***except*** the nacelle inlet lips
   D. All systems ***except*** the windshield and radome alcohol system

7. The L NAC HEAT switch in the up (on) position provides anti-icing to all of the following ***except*** the:
   A. Nacelle lip
   B. Dome spinner (early models)
   C. $P_{T2}T_{T2}$ probe
   D. Conical spinner (late models)

8. The alcohol anti-ice system may be used to anti-ice the:
   A. Radome
   B. Copilot's windshield
   C. Pilot's windshield
   D. Both A and C

CHAPTER 11
AIR CONDITIONING

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CHAPTER 11
AIR CONDITIONING

INTRODUCTION

Air conditioning in the Learjet 35/36 is furnished primarily by regulated engine bleed air, which is temperature controlled and distributed throughout the cabin and cockpit areas. This is the same bleed air that is used for cabin pressurization.

Additional cooling and heating is provided by a Freon refrigeration system and an optional auxiliary electrical heating system. These systems share a separate distribution network through which cabin air is recirculated by a cabin blower and a cockpit fan.

GENERAL

Primary heating and cooling is accomplished by controlling the temperature of the bleed air entering the cabin by circulating it through an air-to-air heat exchanger. The cabin and cockpit distribution systems differ slightly, based on airplane serial number.

Additional refrigeration cooling by the Freon system is available for ground operations and in flight at altitude up to a ***maximum*** of 18,000 feet or 35,000 feet, depending on compressor motor part number.

Additional heating by the auxiliary electrical heating system (if installed) can be obtained for ground operations and at any altitude in flight.

ENGINE BLEED-AIR CONDITIONING AND DISTRIBUTION

GENERAL

This section addresses the conditioning process that the engine bleed air is subjected to before it enters the cabin area, beginning at the flow control valve. Chapter 9, "Pneumatics," describes the bleed-air supply system. Chapter 12, "Pressurization," describes how conditioned bleed air is used for cabin pressurization.

Regulated engine bleed air, supplied to a manifold located in the tailcone section, is ducted to the flow control valve. From the flow

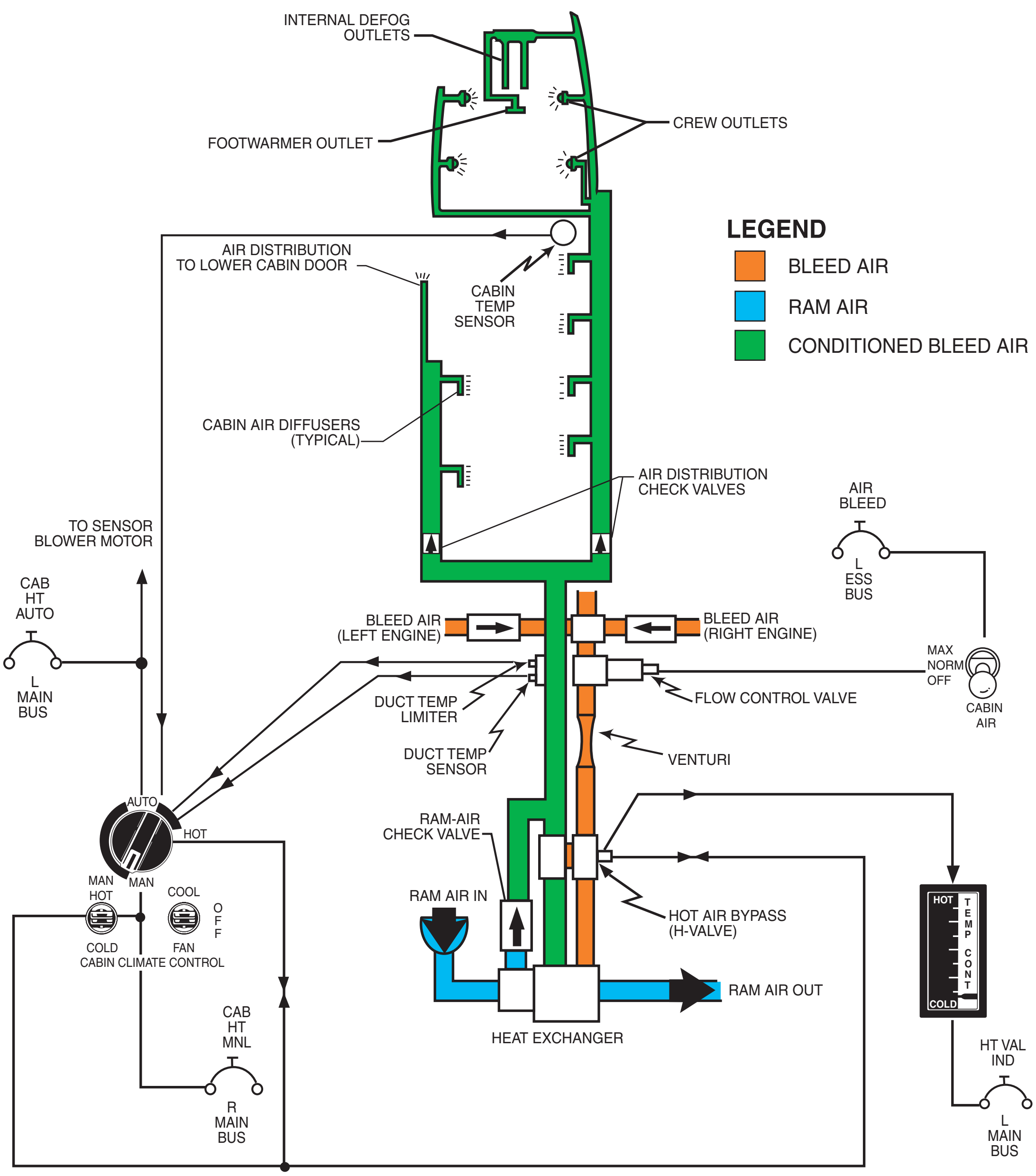


Figure 11-1. Engine Bleed-Air Conditioning System (SNs 35-002 through 35-086, except 35-082, and 36-002 through 36-022)

control valve there are three slightly different cabin and cockpit distribution configurations, each performing the same basic functions, but differing in component arrangement. Figures 11-1, 11-2, and 11-3 depict the three basic configurations by airplane serial number.

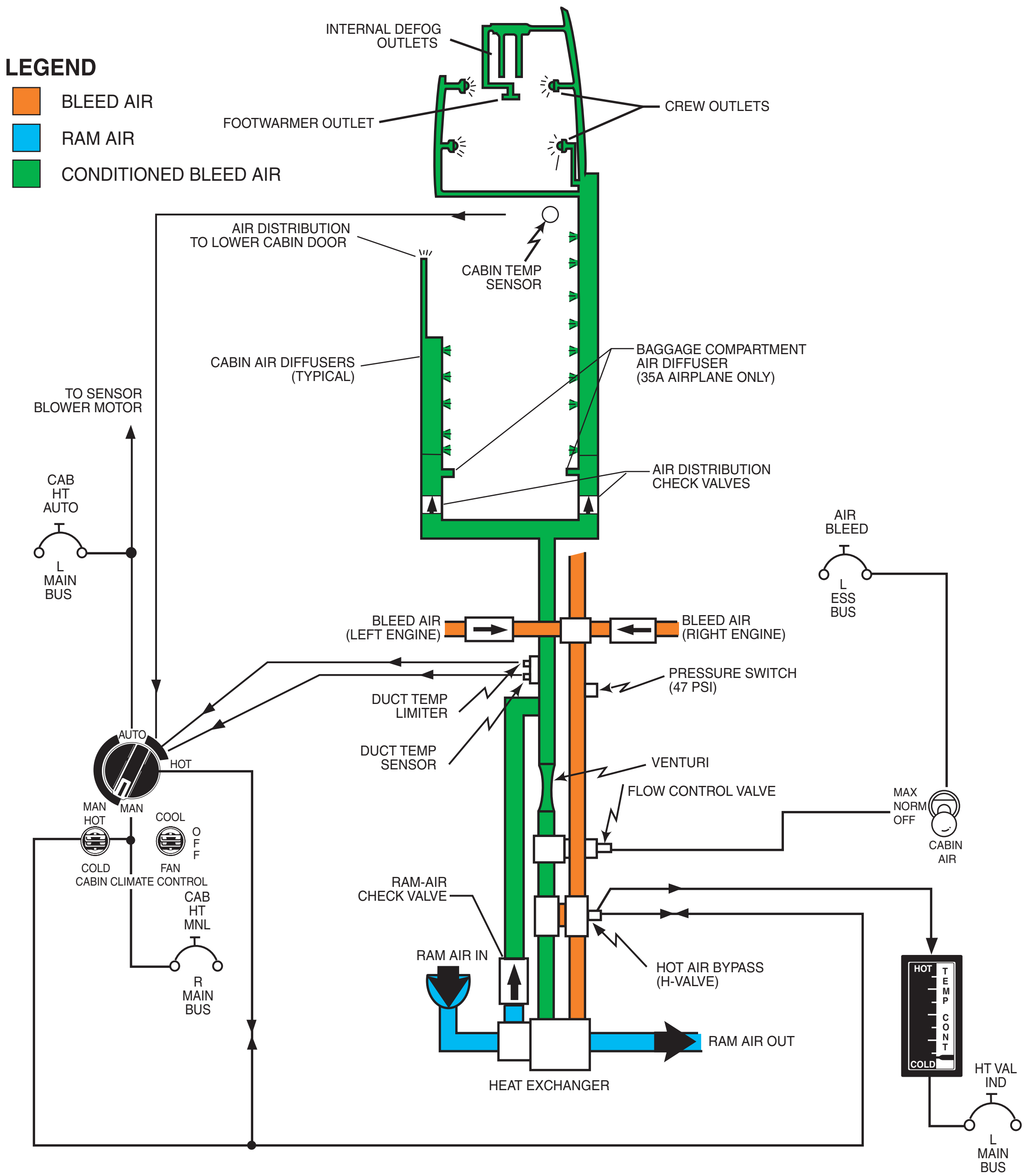
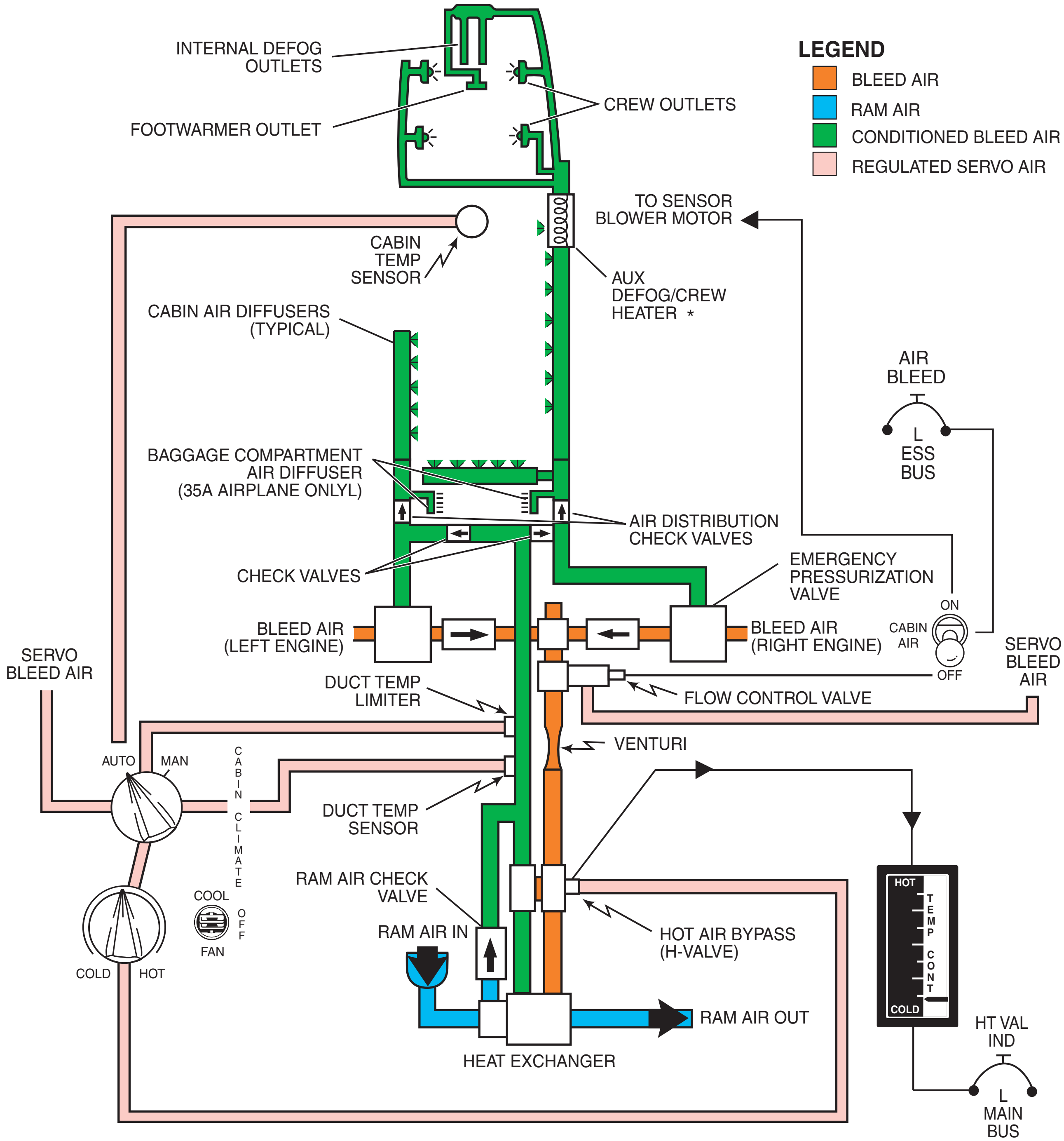


Figure 11-2. Engine Bleed-Air Conditioning System (SNs 35-082, 35-087 through 35-112, 36-023 through 36-031, and Earlier Airplanes Incorporating AMK 76-7)

All three configurations use the flow control valve to control the flow of bleed air through a hot air bypass valve and an air-to-air heat exchanger before it enters the cabin and cockpit distribution ducting.



* SNs 35-643 AND SUBSEQUENT AND SNs 36-064 AND SUBSEQUENT

Figure 11-3. Engine Bleed-Air Conditioning System (SNs 35-113 and Subsequent and 36-032 and Subsequent)

FLOW CONTROL VALVE

The flow control valve is a solenoid-operated valve which controls and regulates the flow of bleed air into the cabin. The position of the valve is determined by the CABIN AIR switch (Figure 11-4). The most current airplanes (Figure 11-3) use a two-position OFF–ON switch. Earlier airplanes (Figures 11-1 and 11-2) use a three-position OFF–NORM–MAX switch. When the CABIN AIR switch is in OFF, the valve is energized and closes. When the switch is in ON or NORM, the valve is deenergized and opens. In the MAX position, the valve opens fully to provide an increase in airflow to the cabin. DC power for valve operation is provided through the AIR BLEED circuit breaker on the left essential bus.

A venturi, located downstream of the flow control valve adjusts the valve to smooth out the flow of bleed air as it enters the cabin. Airflow through the venturi is measured by pneumatic sensing lines connected to a modulating mechanism in the flow control valve which ensures that airflow remains constant when engine power changes occur.

HOT AIR BYPASS VALVE (H-VALVE)

A butterfly bypass valve, more commonly referred to as the "H-valve," is located in the bleed-air duct upstream of the heat exchanger. Its function is to split the flow of bleed air, directing some to the heat exchanger for cooling, and some to bypass the heat exchanger. The result is a mixture of the two airflows, thereby conditioning the bleed air before it enters the cabin area. The position of the H-valve is indicated on the TEMP CONT indicator located in the lower center instrument panel (Figure 11-5).

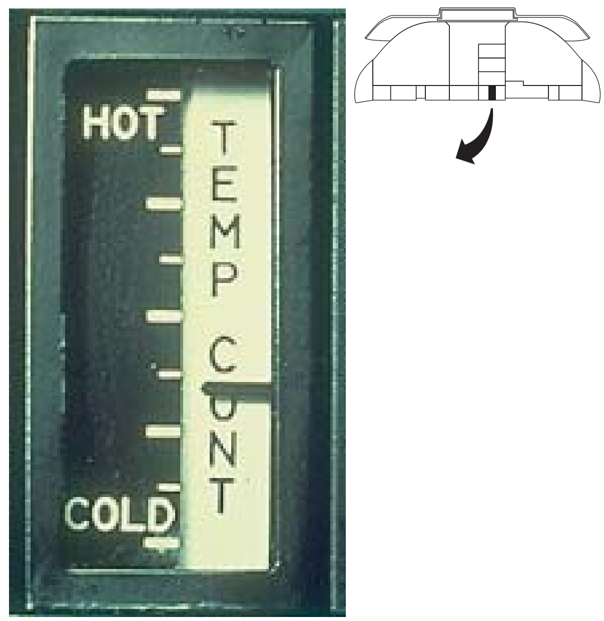
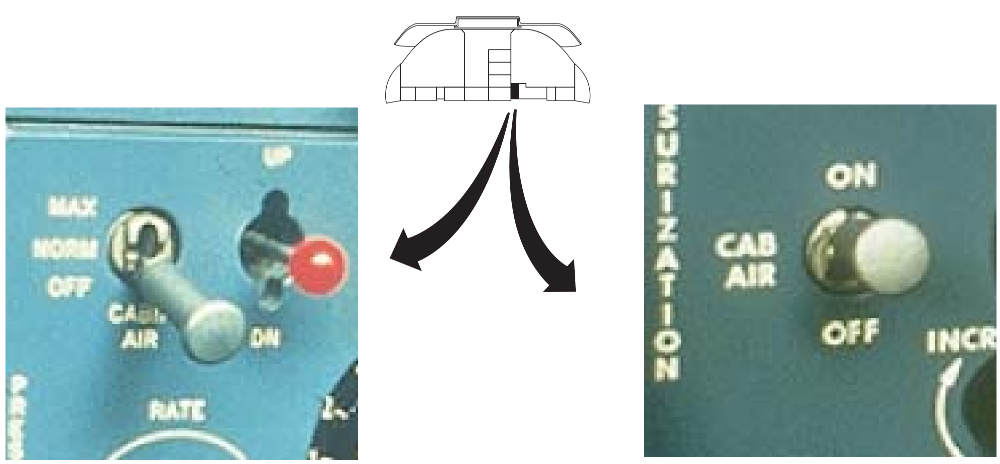


Figure 11-5. Temperature Control Indicator



SNs 35-002 THROUGH 35-112 AND 36-002 THROUGH 36-031

SNs 35-113 AND SUBSEQUENT AND 36-032 AND SUBSEQUENT

Figure 11-4. CABIN AIR Switch

On SNs 35-002 through 35-112 and 36-002 through 36-031, the H-valve butterfly is positioned by a DC electric motor operated by the climate control system. Approximately 25 seconds is required for the valve to travel from one extreme to the other. The valve will remain in its existing position in the event DC power is lost.

On SNs 35-113 and subsequent and 36-032 and subsequent, the H-valve butterfly is positioned pneumatically by servo bleed air (Chapter 9, "Pneumatics") from the climate control system. No electrical circuits are involved except that the TEMP CONT indicator requires DC power. Approximately 8 seconds is required for the valve to travel from one extreme to the other. The valve is spring-loaded to the full cold position anytime servo air pressure is not available.

RAM-AIR HEAT EXCHANGER

The heat exchanger is located inside the tailcone. It consists of a bleed-air core surrounded by a ram-air plenum. Cool air enters the ram-air inlet in the dorsal fin and flows through the plenum, across the bleed-air core, thus cooling the bleed air. The ram air then exhausts overboard through a port in the lower left side of the fuselage.

The cooled bleed air flowing out of the heat exchanger core is ducted back to the bypass side of the H-valve where it mixes with hot bypassed bleed air. The resulting conditioned air is then directed into the cabin and cockpit distribution system.

When the airplane is on the ground, do not perform extended engine operation above idle with the CABIN AIR and BLEED AIR switches positioned to ON. Since there is no ram air for cooling of the bleed air, possible damage to the air-conditioning components could result. Damage might also occur to interior cabin furnishings, as well as overheating the tailcone area.

On SNs 35-082, 35-087 through 35-112, and 36-023 through 36-031, and earlier airplanes incorporating AMK 76-7, the flow control valve is located downstream of the heat exchanger. Engine bleed air is available to the heat exchanger whenever an engine is operating and the BLEED AIR switches are one. Because of this, a pressure switch is installed in the tailcone ducting prior to the heat exchanger. Should this pressure switch actuate (at approximately 47 psi), ***both*** red BLEED AIR L and R annunciator lights illuminate to indicate the overpressure condition.

RAM-AIR VENTILATION

In the event that the airplane is unpressurized in flight, air for circulation and ventilation of the cabin and cockpit areas is provided by ram air, which is ducted into the conditioned bleed-air distribution system.

During normal operation, a one-way check valve in the connecting ram-air duct prevents loss of conditioned pressurization bleed air through the ram-air plenum exhaust port.

CABIN AND COCKPIT AIR DISTRIBUTION

Conditioned airflow distribution to the cabin and cockpit areas is essentially the same for all airplanes (Figure 11-6). The conditioned air is routed from the tailcone into the cabin area through two ducts, one on each side of the cabin. The left duct ends at the entry door, and the right duct continues forward to the cockpit.

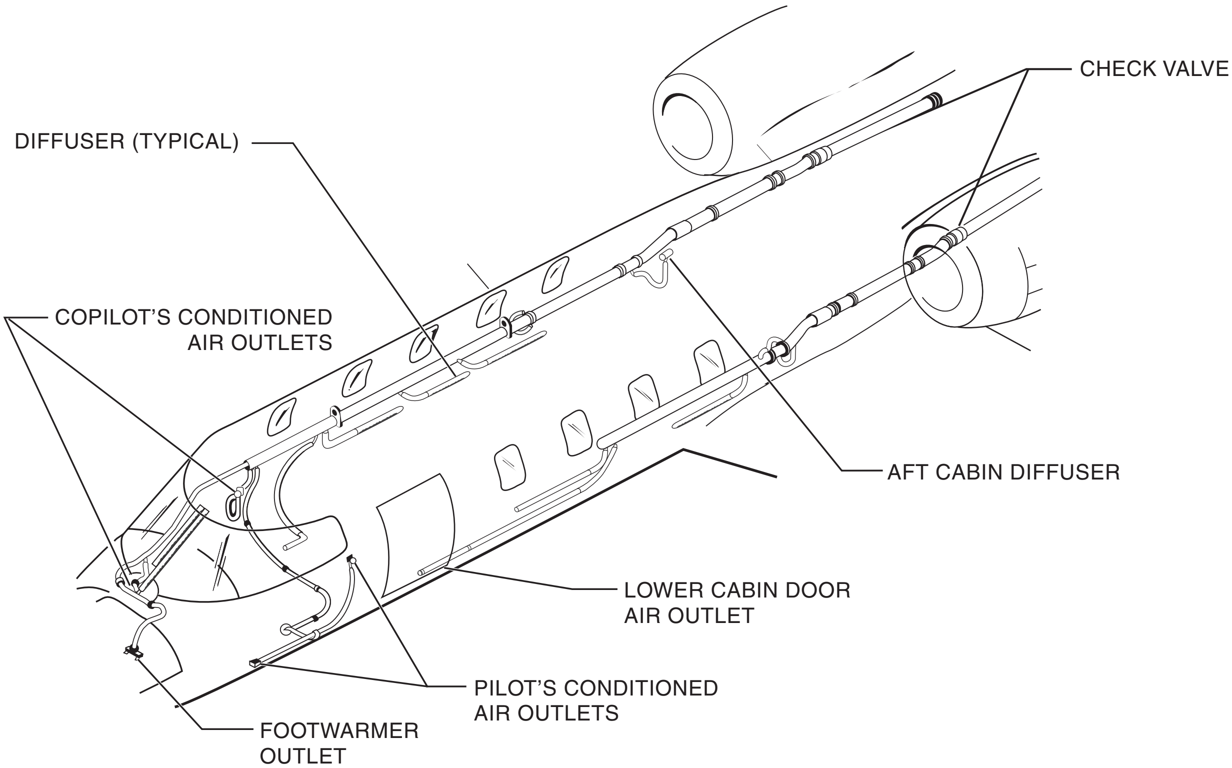


Figure 11-6. Conditioned Bleed-Air Distribution

Cabin Air Distribution

Cabin air distribution is furnished by diffusers installed at intervals along the two ducts, and they direct airflow toward the floor.

A one-way distribution check valve is located at the aft end of each cabin duct. These valves are functionally related to the pressurization system, as described in Chapter 12, "Pressurization."

On SNs 35-113 and subsequent and 36-032 and subsequent (Figure 11-3), distribution of air changes when either (or both) emergency pressurization valves are positioned to emergency.

If only one emergency valve is positioned to emergency, all bleed air from that engine is routed directly into only that side's cabin distribution duct, and temperature control of that air is lost. However, bleed air from the opposite engine is still subject to the normal conditioning process. One-way check valves in the normal distribution ducting prevent the emergency airflow from being lost through the normal distribution system.

If both emergency valves are positioned to emergency, all bleed air from both engines is routed directly into the respective left and right distribution ducts. Temperature control is then sacrificed for pressurization.

Cockpit Air Distribution

Cockpit air distribution is provided by the ducting connected to the forward end of the right hand cabin duct. Four WEMAC outlets (two on each side of the cockpit), located on the sidewall panels and adjacent to the outboard rudder pedals, enable the pilots to control and direct the airflow as desired. A footwarmer diffuser, located below the instrument panel just forward of the center pedestal, directs continuous condition air along the center floor. Two piccolo tubes installed vertically on each side of the windshield center support structure direct a continuous flow of conditioned air across the forward section of each pilot's windshield for the interior windshield defogging.

On SNs 35-328 and subsequent, and 36-050 and subsequent increased continuous interior windshield defogging capability has been provided. Two additional piccolo tubes are installed, one for each windshield. They are positioned horizontally along the lower edge and extend forward from the aft corner of the windshield. This position results in improved interior defogging for the sides of the windshield.

Interior windshield defogging can be maximized by closing the four WEMAC outlets to divert the maximum amount of conditioned air to the windshield piccolo tubes.

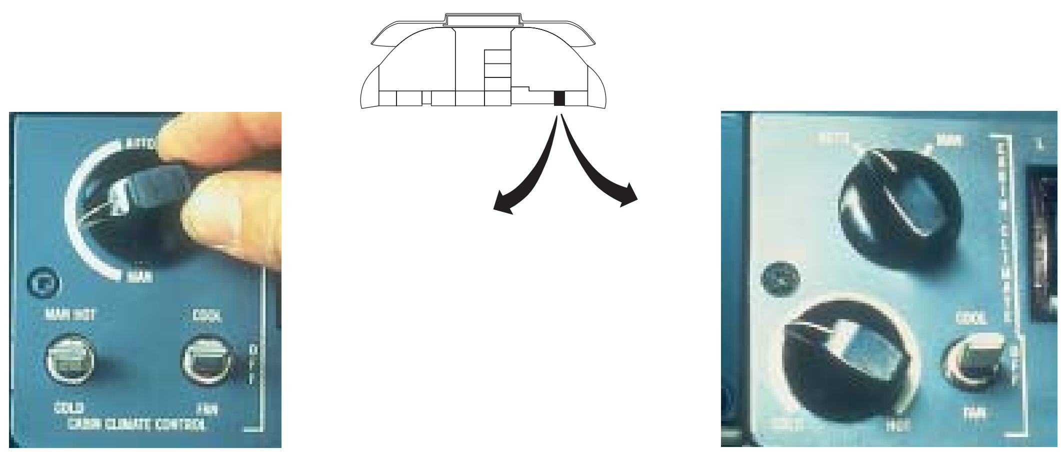
On SNs 35-002 through 35-112 and 36-002 through 36-031, additional heat is available to the cockpit via separate footwarmers that operate from the windshield heat/defog system discussed in chapter 10, "Ice and Rain Protection."

TEMPERATURE CONTROL

Temperature control of the engine bleed-air entering the cabin area is accomplished by varying the position of the H-valve butterfly. As the valve opens, less bleed air is directed to the heat exchanger for cooling, while more bleed air is bypassed, and mixed with the cooled air. Manual and automatic operation of the H-valve is achieved by controls on the CABIN CLIMATE switch panel, located on the copilot's lower instrument panel (Figure 11-7).

On SNs 35-002 though 35-112 and 36-002 through 36-031, the climate control system is operated electrically. System control is accomplished with a rheostat and a HOT–COLD toggle switch (spring load to center). Other system components include a temperature sensor located behind the copilot's seat, a duct temperature sensor and duct temperature limiter (both located in the air duct downstream of the H-valve) (Figure 11-1 or 11-2, as applicable), and a control unit.

If the rheostat is turned fully counterclockwise to the MAN detent, the cabin temperature sensor and duct temperature sensor are both off. The H-valve is then controlled manually by actuating the spring-loaded HOT-COLD switch. The TEMP CONT indicator (Figure 11-5) displays the position of H-valve. DC power



SNs 35-002 THROUGH 35-112
AND 36-002 THROUGH 36-031

SNs 35-113 AND SUBSEQUENT
AND 36-032 AND SUBSEQUENT

Figure 11-7. CABIN CLIMATE CONTROL Panel

for manual operation is provided by the CABIN HT MAN circuit breaker on the right main bus. The TEMP CONT indicator is powered from the HT VAL IND circuit breaker on the left main bus.

If the rheostat is out of the MAN detent, the H-valve position is determined automatically by the control unit, which evaluates inputs from the rheostat, the cabin temperature sensor, and the duct temperature sensor. The control system then responds by continuously modulating the H-valve to maintain the desired temperature. DC power for automatic operation is provided by the CAB HT AUTO circuit breaker on the left main bus.

Whether the system is being operated manually or automatically, the duct temperature limiter signals the control unit if the duct temperature increases to approximately 350° F. The control unit's response is to drive the H-valve to the full cold position, and direct all bleed air through the heat exchanger.

On SNs 35-107, 35-113 and subsequent, and 36-032 and subsequent, the H-valve is positioned pneumatically by servo bleed air (Chapter 9, "Pneumatics"), and no electrical circuits are involved. The CLIMATE CONTROL panel (Figure 11-7) incorporates two control knobs. The AUTO–MAN knob is actually a servo bleed-air selector valve. The COLD–HOT knob is a needle valve that controls the servo air pressure applied to the H-valve butterfly (spring-loaded to the full cold position). Other system components include a temperature sensor located in the upper forward cabin, a duct temperature sensor , and a duct temperature limiter located in the air duct downstream of the H-valve (Figure 11-3). The control system consists of an interconnected servo bleed-air network.

With the AUTO–MAN knob in the MAN position, the selector valve isolates the control system from the influences of the cabin temperature sensor and the duct temperature sensor. Servo air pressure is routed directly through the needle valve (controlled by the COLD–HOT knob) to the H-valve butterfly, which is spring-loaded to the full cold position. Changing the COLD–HOT knob position simply changes the servo air pressure on the H-valve, butterfly. The TEMP CONT indicator (Figure 11-5) displays the relative position of the H-valve, which is the only component in the system that requires DC electrical power. DC power is provided through the HT VAL IND circuit breaker on the left main bus.

With the AUTO–MAN knob (selector valve) in the AUTO position, the servo pressure control network samples the needle valve setting (COLD/HOT knob position), the cabin temperature sensor (existing cabin temperature), and the duct temperature sensor (actual temperature of the bleed air inside the duct). Servo air pressures are modulated by the control system, which causes the H-valve butterfly to modulate, thereby keeping the cabin temperature constant.

For manual or automatic operation, in the event of a duct overheat, the duct temperature limiter causes the control system to shut off servo air pressure to the H-valve butterfly. This allows it to spring to the full cold position, directing all bleed air through the heat exchanger.

A CABIN TEMP indicator may be installed on the center pedestal or instrument panel to indicate the temperature in the cabin from a remote sensor (Figure 11-8).

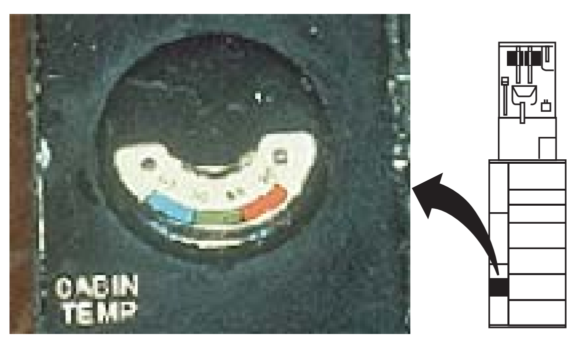


Figure 11-8. CABIN TEMP Indicator

AUXILIARY AIR-CONDITIONING SYSTEMS

GENERAL

Additional air circulation is provided by a cabin blower and a cockpit fan, ducted though distribution networks that are also used by the Freon refrigeration system (auxiliary cooler) and the optional electrical heating system (auxiliary heater).

The cabin blower and cockpit fan may be used simply to recirculate air within the cabin and cockpit areas, or by using the auxiliary cooler or auxiliary heater, to cool or to heat the recirculated air.

For operational requirements on the ground (subject to certain limitations), it is possible to precool or preheat the cabin prior to engine start.

DISTRIBUTION SYSTEM

The heart of the distribution system is the evaporator and blower assembly, installed in the cabin ceiling above the baggage compartment (Figure 11-9). The assembly houses the ducting, the cabin blower assembly, the cockpit fan assembly, the Freon system evaporator, and the optional electrical heating elements (when installed).

Cabin Blower Distribution

The cabin blower assembly consists of two squirrel-cage blowers driven by a single DC motor. The blowers draw air from the baggage compartment area though the evaporator and force it through separate ducts to a louvered grille at the front the ducts. The air is diffused as it blows out directly into the cabin. When installed, the optional heating elements are located within these ducts.

Diverter doors are installed in the ducting forward of the cabin blower. On SNs 35-002 through 642 and 36-002 through 063, the doors are located in the bottom of each duct and are manually controlled and actuated by the OPEN–CLOSE knob adjacent to the louvered grill (Figure 11-10). On SNs 35-643 through 35-646, electrically controlled and actuated diverter doors are installed in the top of each duct along with the mechanically controlled doors on the bottom. On SNs 35-647 and subsequent, and 36-064 and subsequent, only the electrically controlled doors are installed.

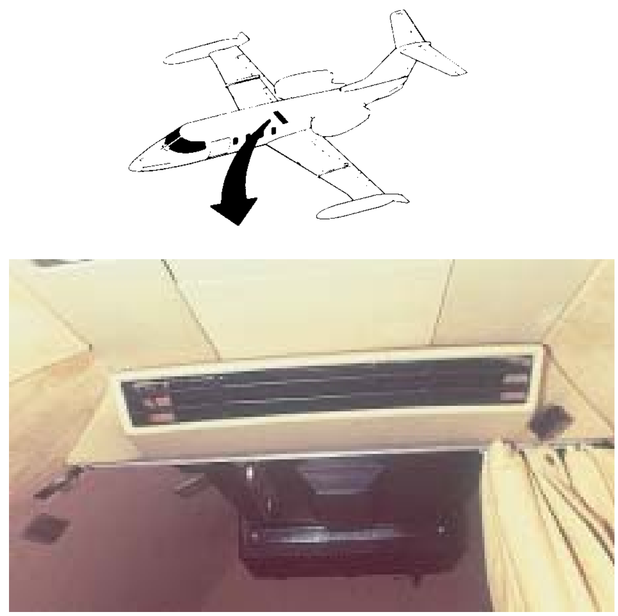
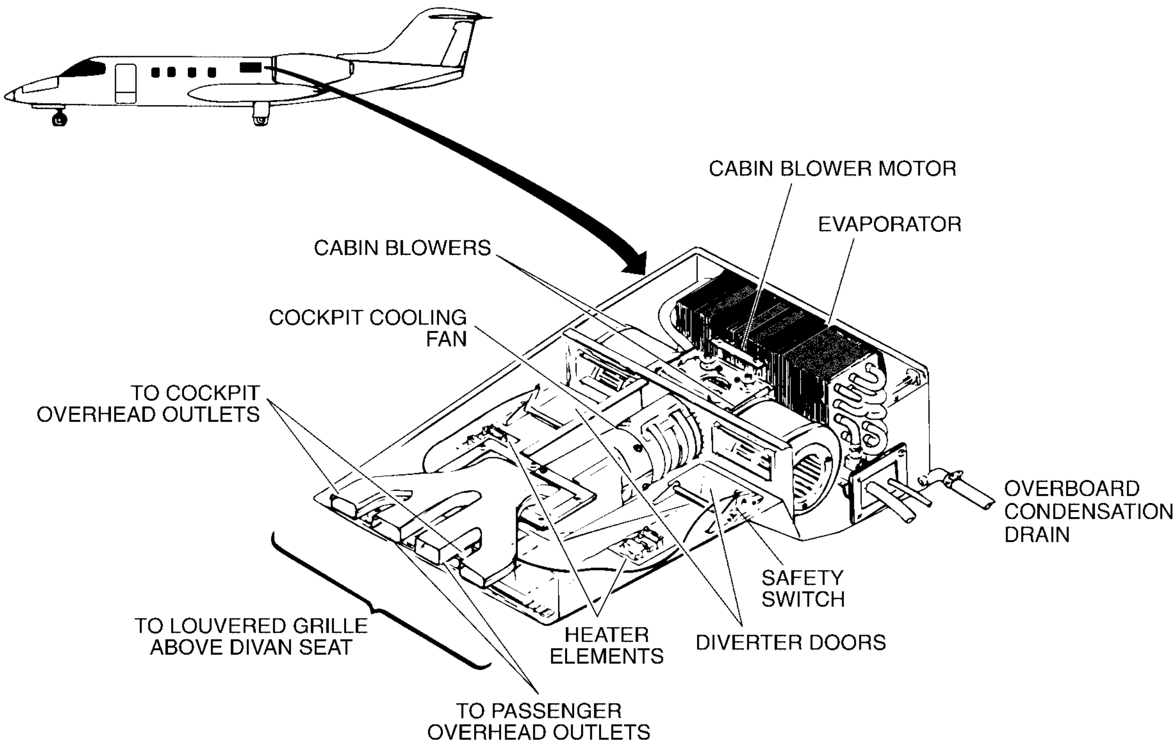
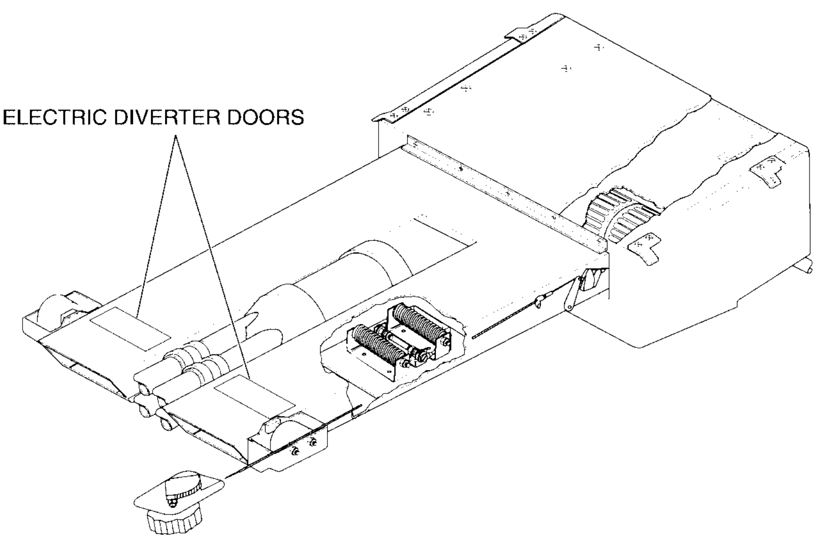


Figure 11-10. Cabin Blower Grille Outlet

On airplanes with the manual diverter doors, when the knob is rotated to the OPEN position, the diverter doors are raised up into the airflow from the cabin blower and divert the air down into the baggage compartment. When the knob is positioned to CLOSE, the doors are flush with the bottom of the ducting and the airflow from the cabin blower is directed into the cabin.



SNs 35-002 THROUGH 35-642



SNs 35-643 AND SUBSEQUENT; EFFECTIVE 35-647 AND SUBSEQUENT THE MANUAL DIVERTER DOORS HAVE BEEN DELETED.

Figure 11-9. Evaporator and Blower Assembly

On airplanes with the electric diverter doors, the doors are controlled by a two-position, ON–OFF, rocker switch located below the cabin blower air outlet. When the switch is positioned to OFF, the diverter doors are lowered into the airflow from the cabin blower and direct the air into the space between the cabin headliner and the fuselage skin. When the switch is positioned to ON, the diverter doors are flush with the top of the ducting and the airflow from the cabin blower is directed into the cabin. On SNs 35-643 through 35-670, the doors may also be controlled by the auxiliary internal windshield defog system. See Chapter 10, “Ice and Rain Protection,” for additional information.

When used simply for additional air circulation, the cabin blower is turned on by selecting FAN on the three-position FAN–OFF–COOL switch, located on the CABIN CLIMATE CONTROL panel (Figure 11-7). DC electrical power is provided by the CAB BLO circuit breaker on the left main bus. On SNs 35-113 and subsequent and 36-032 and subsequent, variable blower speed control is afforded through the CABIN BLOWER rheostat, located on the copilot’s sidewall panel (Figure 11-11). Earlier airplanes do not have this feature unless AMK 77-4 is incorporated.

Figure 11-11. COCKPIT AIR and CABIN BLOWER Rheostats

Figure 11-12. Cockpit Upper Air Outlets

Figure 11-13. Passenger Overhead Outlets (WEMACS)

Cockpit Fan Distribution

Between the two ducts fed by the cabin blowers is another duct which encloses the axial cockpit fan. This fan draws air from the baggage compartment area through the evaporator, but its output is furnished directly to four smaller ducts concealed in the cabin overhead paneling. Two of these ducts run directly to the two louvered overhead outlets in the cockpit (Figure 11-12). On SNs 35-092 and 36-025 and subsequent, two additional ducts (one on each side) are connected to the individual overhead WEMAC outlets above each of the passenger seats (Figure 11-13). Air volume and directional control is provided at each outlet. The fan motor is cooled by the air it moves through the ducting.

The cockpit fan is controlled by the COCKPIT AIR rheostat on the copilot's sidewall panel (Figure 11-11) using DC power from the CAB BLO circuit breaker on the left main bus.

On SNs 35-002 through 35-112, and 36-002 through 36-031, the OFF detent is at the full clockwise position, and the fan speed is increased by rotating the rheostat in a counterclockwise direction. On SNs 35-113 and subsequent and 36-032 and subsequent, the OFF detent is at the full counterclockwise position, and speed is increased by rotating the rheostat in the clockwise direction.

If all the cockpit and overheat outlets are closed, the cockpit fan must be operated because no cooling airflow for the motor is available and the motor will overheat.

AUXILIARY COOLING SYSTEM

A freon refrigeration system (auxiliary cooler) is installed to provide supplemental cooling for ground and in-flight operations and can also be used for dehumidification.

System components, identified schematically (Figure 11-14), are conventional. The compressor (belt-driven by a 3 3/4 horsepower motor), the condenser, and the dehydrator are located inside the tailcone. The compressor motor is cooled by air from the tailcone ventilation airscoop on the left side of the fuselage. The evaporator and expansion valve are located inside the evaporator and blower assembly (above the baggage compartment).

Operation

Electrical power for system operation must be supplied by either a GPU or an engine-driven generator. The system is turned on by selecting the COOL position on the FAN–OFF–COOL switch. DC power is applied simultaneously to the compressor motor and the cabin blower motor.

For best results on the ground, the CABIN AIR switch should be off to keep warm bleed air from entering the cabin while the engines are running.

Cool air is drawn through the evaporator and circulated as already described in Cabin Blower Distribution, except that the blower motor will run continuously at its maximum speed (the CABIN BLOWER rheostat, if installed, is rendered inoperative). The compressor motor is powered from the battery charging bus through a 150-amp current limiter and a control relay powered from the FREON CONT circuit breaker on the left main bus.

The diverter doors may be positioned as desired to control airflow into the cabin through the louvered grille above the divan seat. If desired, the cockpit fan may also be used to provide wider circulation of the cooled air to the cockpit and passenger WEMAC outlets.

When the cooling system is being powered by a GPU, it is possible in some conditions for the airplane batteries to be depleted if GPU failure occurs.

The compressor motor is automatically deenergized when the START–GEN switch is selected to START. However, normal operating procedures require that the FAN–OFF–COOL switch be in the OFF or FAN position prior to engine start to preclude possible electrical system damage.

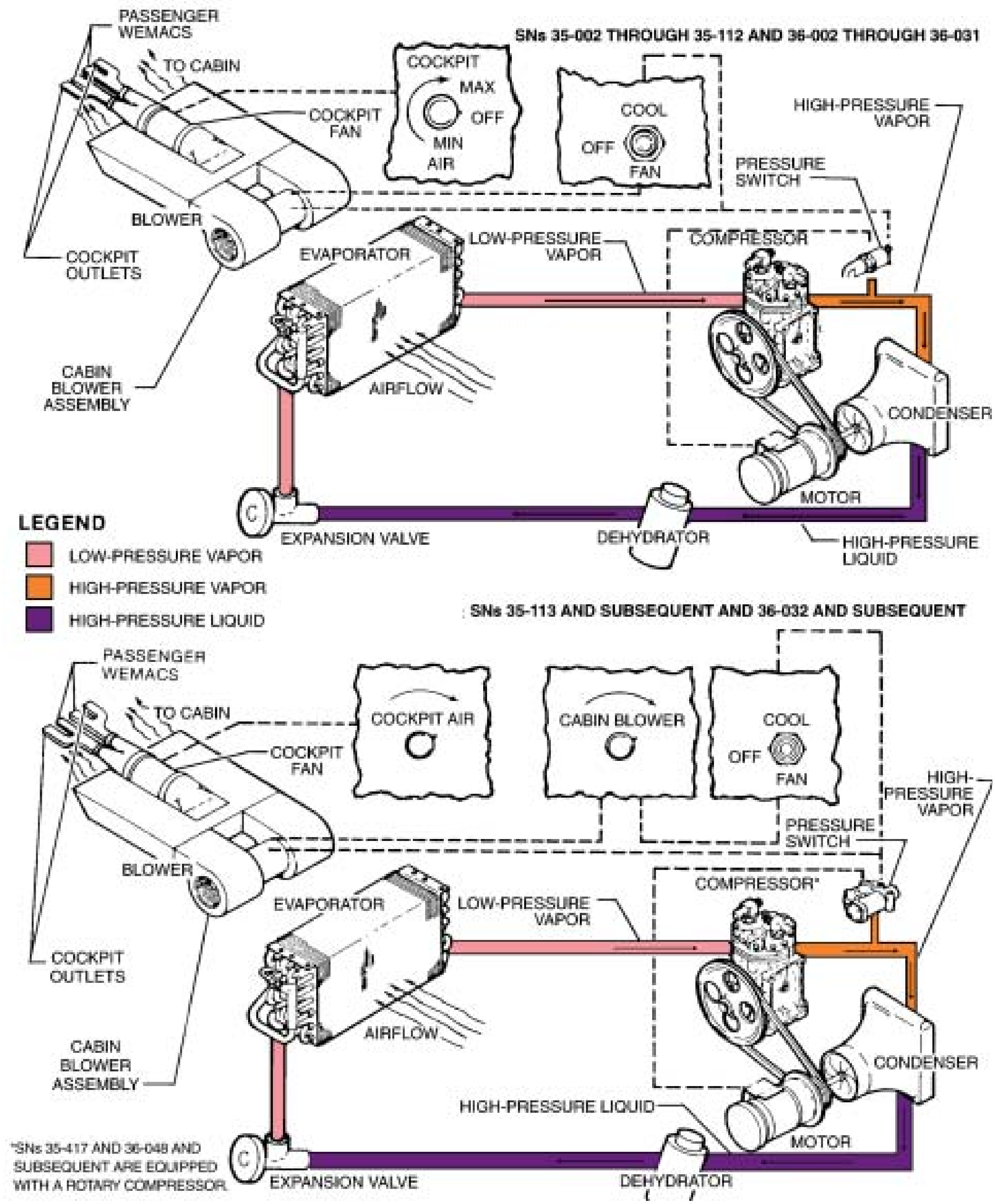


Figure 11-14. Freon Refrigeration System Schematic.

AUXILIARY HEAT SYSTEMS (OPTIONAL)

Two optional electric auxiliary heat systems are available: one for the cabin and one for the cockpit. Both systems may be used to provide additional heating on the ground or in flight.

Auxiliary Cabin Heat System

General

The auxiliary cabin heat system uses the cabin blower to circulate heated air. It also incorporates two, dual-coil heating elements, one located in each of the cabin blower ducts (Figure 11-15). Each heating element contains a thermoswitch set for high and low limits (150° F and 125° F), and a thermal fuse for overheat protection.

On SNs 35-002 through 35-646, and 36-002 through 36-063, if the manual diverter doors are open (air being diverted into the baggage compartment), the cabin heat system is inoperative. On SNs 35-643 and subsequent, and 36-064 and subsequent, if the electrical diverter doors are open (air being diverted above the headliner), the diverter doors will close when the auxiliary cabin heat system is turned on.

Because of the high amperage required by the heating coils, they cannot be powered with only airplane battery power. Either a ground power unit or an engine-driven generator must be supplying power to operate the auxiliary cabin heat system.

The auxiliary cabin heat system will ***not*** automatically shut down when a START–GEN switch is positioned to START. Therefore, it is recommended that the system be turned off during engine start to avoid possible 275-amp current limiter failure.

The Freon air-conditioning system has priority over the auxiliary cabin heat system. If the Freon air-conditioning system is operating, the auxiliary cabin heat is inoperative. If the auxiliary cabin heat system is operating, turning on the Freon air-conditioning system will turn off the auxiliary cabin heat system.

Operation

On SNs 35-002 through 35-670, and 36-002 through 36-063, the auxiliary cabin heat system is controlled by a three-position (LO–OFF–HI) AUX HT switch on the copilot's lower, right switch panel. Selecting the LO position powers the cabin blower and one heating coil on each element. The HI position powers the cabin blower and all four coils.

On SNs 35-671 and subsequent and 36-064 and subsequent, the cabin auxiliary heat system is controlled by a three-position (OFF–CREW–CAB & CREW) AUX HT switch on the copilot's lower, right switch panel. The CREW position of the switch energizes the crew auxiliary heater explained later in this section. Selecting the CAB & CREW position energizes the cabin blower and all four auxiliary cabin heating coils.

Initially, the cabin blower will run at one-tenth of its normal speed until one of the thermoswitches senses a high limit. At that time, the cabin blower will come up to full speed and electrical power to the heating coils will be removed. The coils will cool until the thermoswitch senses a low limit. Electrical power will then be reapplied to the heating coils and they will continue to cycle, on and off, between the high and low limits controlled by the thermoswitch. The cabin blower will continue to operate at full speed as long as the auxiliary cabin heat system is in operation.

DC electrical power to the heating coils is provided by the same 150-amp current limiter on the battery charging bus used to power the Freon air-conditioning compressor motor. Control power for the auxiliary cabin heating system is provided by the AUX CAB HT circuit breaker on the left main DC bus.

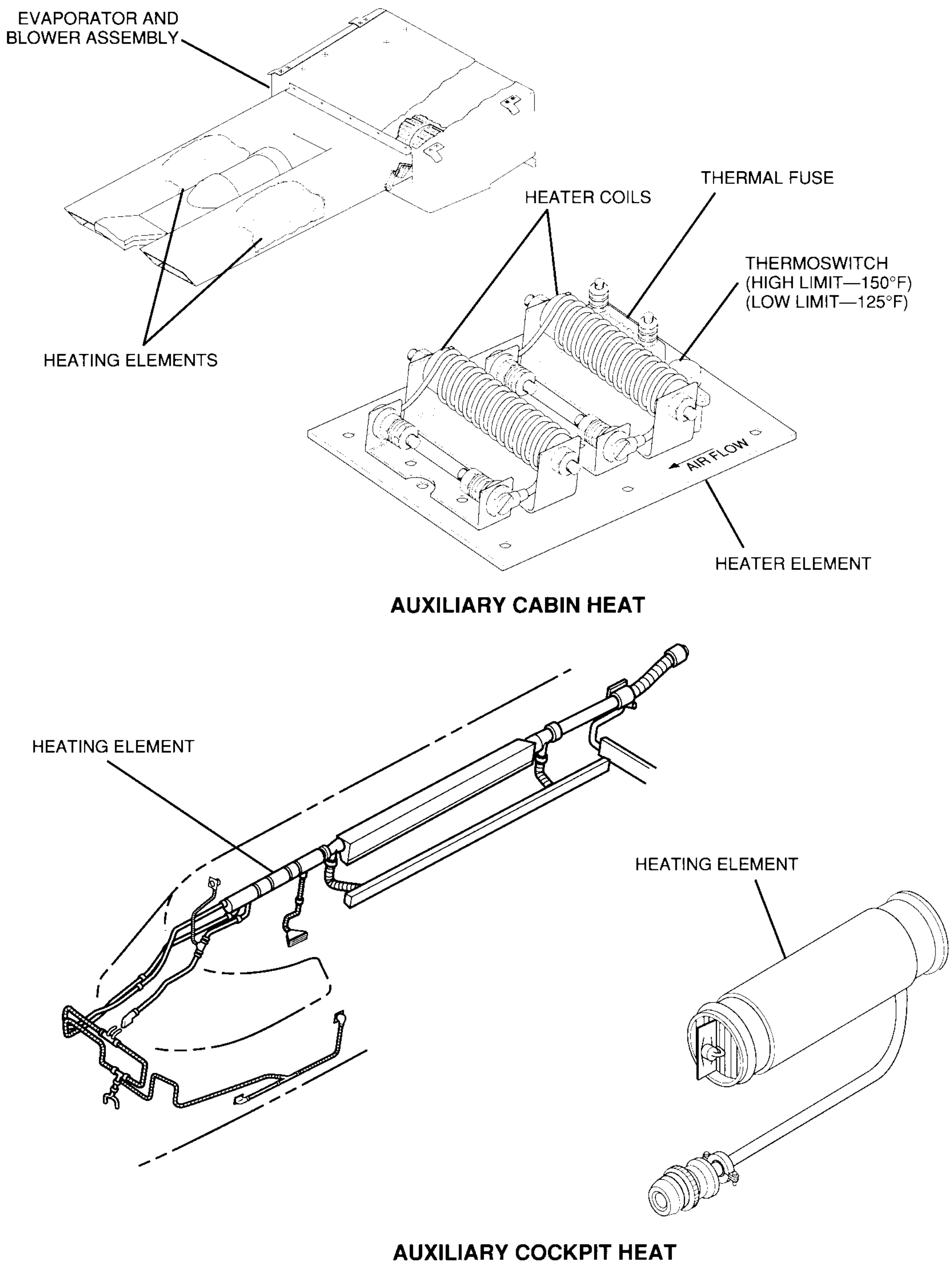


Figure 11-15. Auxiliary Heating System Components

Auxiliary Cockpit Heat System (SNs 35-643 and Subsequent, and 36-064 and Subsequent)

General

The auxiliary cockpit heat system provides additional heat for crew comfort and interior windshield defogging. It includes an electric heater installed in the forward end of the right-hand cabin bleed-air duct, where it connects to the cockpit air distribution ducting and uses condition bleed airflow to circulate heated air (Figure 11-15).

Operation

The heating element for the auxiliary cockpit heat system requires bleed airflow through it for cooling. Because of this, on SNs 35-671 and subsequent, and 36-064 and subsequent, the CABIN AIR switch must be ON, at least one engine must be running and its bleed air shut off and regulator valve must be open before electrical power can be applied to the heating element. If only the left engine is running, the left emergency pressurization valve must be in normal.

Despite these safeguards, on all airplanes, the crew should ensure the CABIN AIR switch is ON, at least one engine is running, and there is adequate airflow in the right-hand cabin bleed-air duct to cool the heating element before activating the auxiliary cockpit heating system.

On SNs 35-643 through 35-670, the auxiliary cockpit heating system is controlled by a three-position (OFF–CKPT–W/S AUX DEFOG HEAT) switch on the ANTI–ICE control panel. Selecting either CKPT or W/S AUX DEFOG HEAT will power the heater element. (See Chapter 10, "Ice and Rain Protection," for additional information on the W/S AUX DEFOG HEAT function.)

On SNs 35-671 and subsequent, and 36-064 and subsequent, the auxiliary cockpit heating system is controlled by a three-position (OFF–CREW–CAB & CREW) AUX HT switch, located on the copilot's lower, right switch panel. Selecting either CREW or CAB & CREW will power the heater element, as long as the CABIN AIR switch is ON and the other conditions described above are met.

With the heater element powered, all the air coming through the bleed-air outlets in the cockpit will be heated. A thermoswitch, located between the windshield defog diffusers and the center footwarmer, monitors the temperature of the airflow. The thermoswitch will cycle electrical power to the heater element off and on between approximately 155° and 160° F. In the event of an overheat, a 295° thermoswitch in the heater should remove power to the element. Finally, a thermal fuse on the heater will melt at approximately 415° F and remove power to the element.

Power for the auxiliary cockpit heat element is provided by two 20-amp current limiters from the battery charging bus. Control power for the auxiliary cockpit heat system is provided by a circuit breaker on the left essential A bus. On SNs 35-643 through 670, the circuit breaker is labeled AUX DEFOG. On SNs 35-671 and subsequent, and 36-064 and subsequent, it is labeled AUX CREW HT.

QUESTIONS

1. The manual diverter doors must be fully closed:
A. To operate the cockpit fan
B. To operate the Freon system
C. To operate the auxiliary heating system
D. Airplane does not have manual diverter doors

2. Equipment which can be operated with airplane battery power only is:
A. The auxiliary defog system
B. The Freon air-conditioning system
C. The cabin blower and cockpit fan
D. The auxiliary heating system

3. When the airplane is unpressurized on the ground, air circulation is provided by:
A. Ram air
B. The cockpit fan and the cabin blower
C. Bleed-air system
D. Auxiliary defog system

4. The primary air conditioning in flight is provided by:
A. Engine bleed air
B. The heat pump
C. The auxiliary heater
D. The Freon refrigeration system

5. When using the auxiliary cabin heater, the heated air blows out through:
A. The conditioned air outlets
B. The louvered grille above the divan seat
C. The overheat cockpit air outlets
D. The overheat passenger WEMAC outlets

6. The Freon system should not be used above:
A. 5,000 feet
B. 8,000 feet
C. 18,000 feet
D. 35,000 feet

7. The Freon system automatically disengages:
A. During engine start
B. Upon touchdown
C. When unpressurized
D. If the main door is opened

8. When the Freon system is operating, it cools:
A. Ram air
B. Cabin air
C. Outside air
D. Bleed air

9. When operating the Freon system on the ground with engines running, the switch that should be in OFF for maximum cooling effectiveness is the:
A. GEN–START
B. CABIN BLOWER
C. CABIN AIR
D. COCKPIT AIR

10. In order to operate the auxiliary cabin heater:
A. Engines cannot be running.
B. CABIN AIR switch must be off.
C. Either a GPU or an engine-driven generator is required.
D. Airplane must be on the ground.

11. If DC power fails, the flow control valve:

A. Closes completely
B. Modulates from open to closed
C. Remains open
D. Is bypassed

12. The temperature control indicator shows:

A. Cabin air temperature
B. Cockpit air temperature
C. The temperature of the bleed air in the plenum chamber
D. The position of the H-valve

CHAPTER 12
PRESSURIZATION

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ILLUSTRATIONS

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CHAPTER 12
PRESSURIZATION

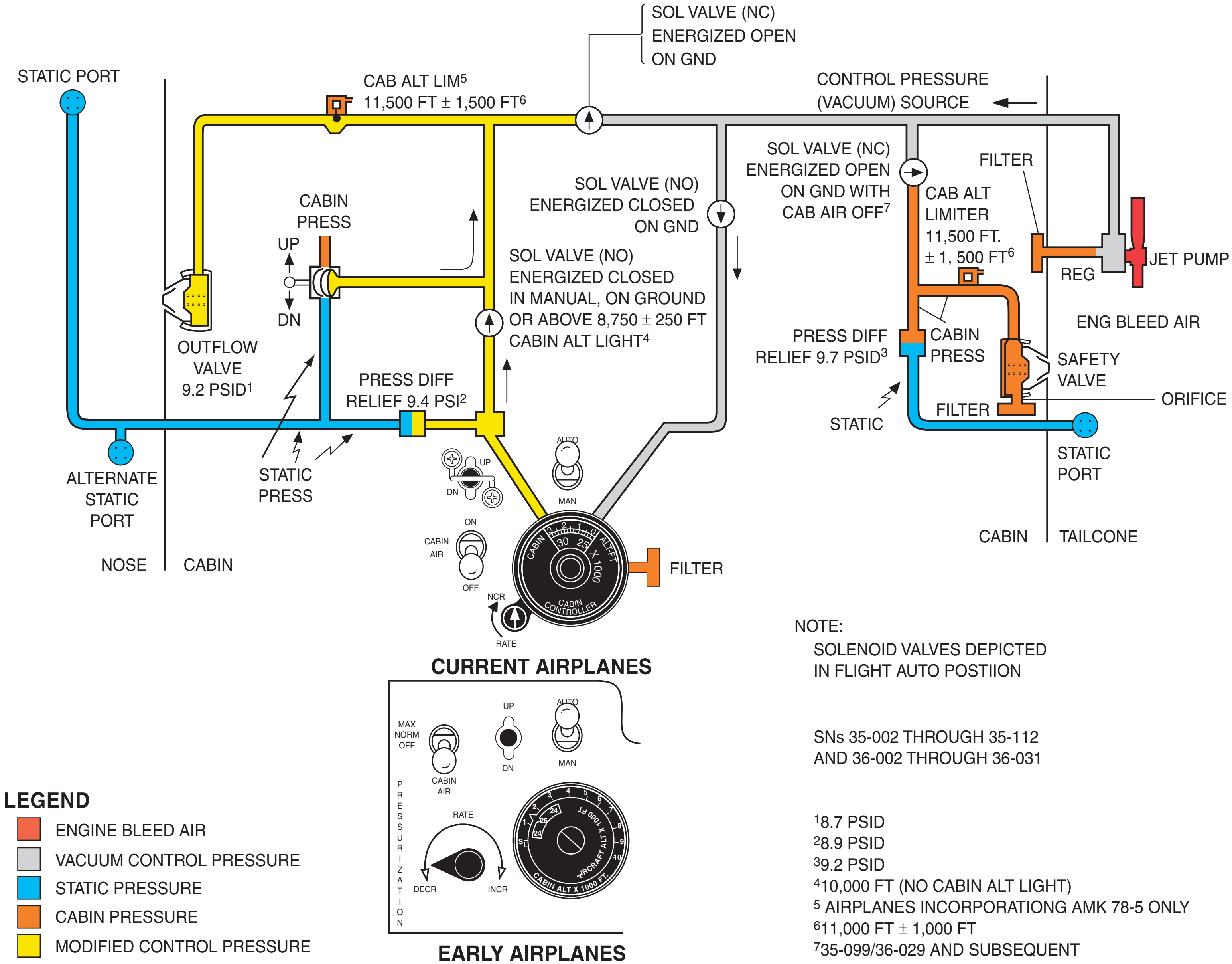
INTRODUCTION

The Lear 35/36 airplane incorporates a pressurization system which maintains a specified level of pressure consistent with built-in limits. Cabin pressure is regulated by controlling the exhaust of conditioned bleed air supplied by the engines. During normal operation, the system functions automatically to provide crew and passenger comfort within the operational envelope of the airplane. Cabin pressure is controlled by an outflow valve which is pneumatically operated to maintain a specified differential between cabin and ambient pressures. Inward and outward relief for both negative and excess positive differential conditions are incorporated to protect the airplane's structure. A control module provides a full range of manual control in the event of a malfunction of the automatic controls. The purpose of the pressurization system is to ensure crew and passengers comfort at all altitudes.

GENERAL

The pressurization control system is completely pneumatic during normal in-flight automatic operation. Pneumatic pressure is provided by a vacuum jet pump. Control pressure (vacuum) is applied to the outflow valve through the pressurization control module. The pressurization controller provides for both automatic and manual capabilities. Electrically actuated solenoid valves and switches are incorporated for ground and manual operation.

During climbs and descents the controller regulates the outflow discharge rate. This rate control is necessary to maintain a cabin change rate that is comfortable regardless of the airplanes rate of climb or descent.



NOTE:
SOLENOID VALVES DEPICTED IN FLIGHT AUTO POSTIION

SNs 35-002 THROUGH 35-112 AND 36-002 THROUGH 36-031

[1] 8.7 PSID
[2] 8.9 PSID
[3] 9.2 PSID
[4] 10,000 FT (NO CABIN ALT LIGHT)
[5] AIRPLANES INCORPORATIONG AMK 78-5 ONLY
[6] 11,000 FT ± 1,000 FT
[7] 35-099/36-029 AND SUBSEQUENT

Figure 12-1. Pressurization System Control

The “Pneumatics” and “Air Conditioning” chapters describe how the cabin and cockpit are pressurized, heated, and cooled. This chapter deals primarily with how the pressure is regulated.

MAJOR COMPONENTS

The pressurization control system (Figure 12-1) incorporates the following major components:

- Cabin outflow valve
- Vacuum jet pump and regulator assembly
- Pressurization control module
- Cabin safety valve

CABIN OUTFLOW VALVE

The pneumatically operated outflow valve is located on the forward pressure bulkhead in front of the copilot’s position. Excess cabin air pressure is relieved into the unpressurized nose section through the outflow valve as necessary to maintain the desired cabin pressure.

VACUUM JET PUMP AND REGULATOR ASSEMBLY

The pneumatic pressure source for control of the outflow valve is established by a vacuum jet pump and regulator assembly in the tailcone section of the airplane. Engine bleed air is routed through a venturi (jet pump) which generates a negative pressure, while a regulator ensures that the negative pressure maintains a constant differential pressure with respect to cabin pressure. This negative pressure (vacuum) is furnished to the pressurization control module which uses it to control the outflow valve.

PRESSURIZATION CONTROL MODULE

General

The pressurization control module is located on the copilot’s lower instrument panel. The controls on the front of the module are located on what is referred to as the pressurization control panel. Figure 12-2 illustrates a typical airplane pressurization control module configuration.

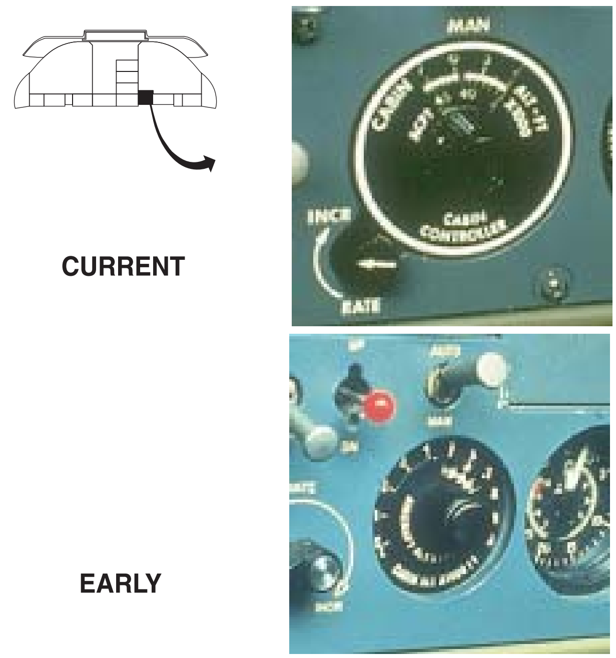


Figure 12-2. Pressurization Control Module

AUTO-MAN Switch

Pressurization control is normally accomplished in the automatic mode. With the AUTO-MAN switch in the AUTO position, the cabin controller automatically adjusts the pneumatic pressure sent to the outflow valve to regulate cabin pressure. If there is a malfunction in the cabin controller, the automatic pneumatic circuit can be isolated from the outflow valve by selecting MAN. The outflow valve is then manually controlled with the UP–DN control knob to regulate cabin pressure.

Cabin Controller

In the AUTO mode of operation, the cabin controller regulates cabin pressure in relation to the altitude that is set on the altitude selector knob. Rotating the knob on the face of the cabin controller either turns a dial or aligns a window to indicate two scales with a fixed index between them. The outer scale represents cabin altitude, and the inner scale represents airplane altitude.

For the current ECS, the cabin controller is capable of maintaining the cabin pressure at sea level with airplane altitudes up to approximately 24,000 feet. If the airplane is flown to an altitude higher than 24,000 feet, the cabin altitude must increase in order to maintain the same pressure differential. At an altitude of 45,000 feet, the cabin altitude will normally be approximately 6,700 feet.

Rate Control

A RATE knob is installed to the lower left of the CABIN CONTROLLER to control the rate at which the cabin climbs and descends. The RATE control knob allows variable control within the approximate limits of 175 feet/minute and 2,500 feet/minute. In AUTO mode, the CABIN CONTROLLER maintains the desired rate of climb or descent until the selected altitude is attained.

Manual Cabin Altitude Control Valve

The UP–DN lever can be used to pneumatically control the outflow valve. Because of the red knob on the end of the lever, it is frequently referred to as the "cherry picker."

The UP–DN lever is spring-loaded to the center position and is wire guarded on later airplanes to prevent inadvertent activation.

The UP–DN lever can be used to increase or decrease cabin altitude in either the AUTO or MAN mode. However, if it is used in the AUTO mode, the CABIN CONTROLLER will also attempt to control the outflow valve, and, as soon as the UP–DN lever is released to neutral, the cabin controller will return the cabin pressure to the original value.

Differential Pressure Relief Valve (Primary)

The primary differential pressure relief valve functions in association with the CABIN CONTROLLER. Its purpose is to relieve excessive control pressure to the outflow valve in the event that cabin pressure should exceed the normal limit when operating in AUTO mode.

NOTE

The primary differential pressure relief valve does not function in MAN control.

On airplane SNs 35-002 through 35-112 and 36-002 through 36-031, the relief valve is set for 8.9 psid.

On airplane SNs 35-113 and subsequent and 36-032 and subsequent, the valve is set for 9.4 psid.

During a rapid airplane climb, with a low setting on the RATE knob, it is possible to reach the differential pressure relief setting prior to reaching the selected airplane altitude, at which time the cabin climb rate will approximate the airplane climb rate.

Cabin Altitude Limiter (For Outflow Valve)

A cabin altitude limiter is installed on airplane SNs 35-113 and subsequent, and 36-032 and subsequent, and earlier SNs incorporating AMK 78-5. It functions to limit the loss of cabin pressure due to malfunctioning controller or inadvertent operation of the primary differential pressure relief valve.

If cabin altitude reaches 11,000 ± 1,000 feet on early airplanes, or 11,500 ± 1,500 feet on later airplanes, the altitude limiter forces modulation of the outflow valve by introducing cabin pressure into the control line, thereby restricting cabin altitude to the listed level.

Controller Solenoid Valves

Three solenoid-operated valves are installed in the controller which are used to control the routing of pneumatic control pressure to the outflow valve. All three valves are energized on the ground by the squat switch relay box, causing the outflow valve to open, thereby depressurizing the cabin.

One of the valves is used in flight to effect manual control of the outflow valve, and is referred to as the "manual-mode solenoid valve" (see Flight Operation-Manual, this section).

For normal automatic in-flight operation, all three solenoid valves are deenergized.

On the early SNs, valve actuation requires DC power from the AIR BLEED circuit breaker on the left essential bus. Later SNs require DC power from the CAB PRESS circuit breaker on the right essential bus.

Aneroid Switches

Either one or two aneroid switches are installed in the pressurization system depending on airplane serial number. Early airplanes use a single aneroid switch for both warning horn and manual solenoid operation. Later airplanes use two aneroid switches, one for the warning horn and another for manual solenoid operation.

Manual Pressurization Aneroid Switch

The pressurization aneroid switch is located inside the pressurization module.

On airplane SNs 35-002 through 35-112 and 36-002 through 36-031, if cabin altitude increases to 10,000 feet or above, the aneroid switch completes a power circuit to the normally open manual control solenoid valve. The solenoid valve is energized closed, isolating all automatic pneumatic circuits from the outflow valve. The outflow valve, now isolated holds its last attained position. When cabin altitude decreases to 8,000 feet or below, the aneroid resets and deenergizes the solenoid valve open, provided the AUTO MAN switch is in AUTO.

On airplane SNs 35-113 and subsequent and 36-032 and subsequent, the description of operation is the same as early SNs, except that the aneroid switch actuates at 8,750±250 feet and resets at 7,200 feet and, when actuated, causes the amber CAB ALT annunciator to illuminate (Annunciator Panel section). When the aneroid resets, the annunciator extinguishes.

Should the above cabin altitudes be reached or exceeded, the “cherry picker” is the only way to control the outflow valve.

Cabin Altitude Warning Horn Aneroid Switch

Early SNs use the manual pressurization aneroid just described, while later SNs use a separate 10,100-foot cabin aneroid to sound a cabin altitude warning horn. A spring-loaded HORN SILENCE switch on the center switch panel (Figure 12-3) may be used to silence the horn. However, the horn will reactivate approximately 30 seconds after being silenced with the HORN SILENCE switch.

The horn will continue to reactivate after each use of the HORN SILENCE switch until the aneroid resets at a cabin altitude of 8,000 feet (early SNs) or 8,590 feet (later SNs).

The rotary system TEST switch on the center switch panel (Figure 12-3) is used to test the cabin altitude warning horn. Rotating the switch to CAB ALT and depressing the TEST button provides a ground, simulating the altitude warning horn aneroid switch actuation. This test does not illuminate the CAB ALT light (if installed). During the test sequence, HORN SILENCE switch operations should also be checked.

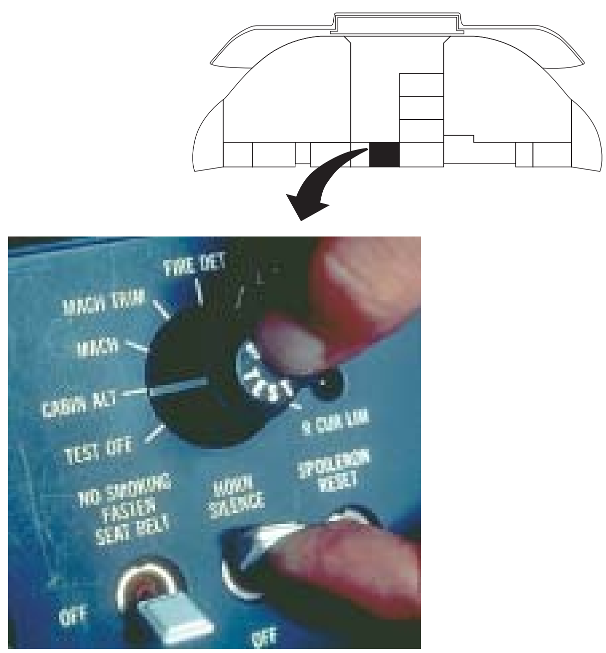


Figure 12-3. HORN SILENCE and Test Control

CABIN SAFETY VALVE

General

A pneumatically operated cabin safety valve is installed in the aft pressure bulkhead. Its purpose is to relieve a cabin overpressure or a negative pressure differential caused by a malfunction in the normal control system. In flight it normally remains fully closed unless acted upon by the secondary differential pressure relief valve, causing it to open (due to an overpressure). In the case of a negative differential pressure condition, ambient pressure unseats the safety valve, allowing an inward flow to equalize the differential.

Operation

Operation of the safety valve is automatic in flight; there is no crew control. On SNs 35-099 and subsequent and 36-029 and subsequent, a fourth solenoid valve is installed in the pneumatic control circuit to allow control of the safety valve on the ground only (engine running and BLEED AIR switches at ON). The solenoid valve is energized open when the CABIN AIR switch is turned OFF (to open the safety valve), and is deenergized closed 10 seconds after the CABIN AIR switch is turned to ON (to close the safety valve). The solenoid is deenergized in flight regardless of CABIN AIR switch position.

On the earlier SNs, the safety valve does not open on the ground.

Differential Pressure Relief Valve (Secondary)

The secondary pressure relief valve functions in association with the safety valve. Should the primary pressure relief valve not function properly, the secondary pressure relief valve forces the safety valve open to limit cabin pressure. The safety valve will relieve pressure at 9.2 psid on SNs 35-002 through 35-112 and 36-002 through 36-031. On SNs 35-113 and subsequent and 36-032 and subsequent, the pressure is relieved at 9.7 psid.

Cabin Altitude Limiter (Secondary)

The cabin altitude limiter for the cabin safety valve serves the same purpose as the cabin altitude limiter for the outflow valve. If the secondary differential pressure relief valve malfunctions (causing the safety valve to open) and cabin altitude reaches 11,000 + 1,000 feet on early airplanes (11,500 + 1,500 feet on current airplanes), the cabin altitude limiter introduces cabin air pressure into the safety valve. This causes the valve to modulate and maintain cabin altitude at the listed value.

CABIN AIR SWITCH

The CABIN AIR switch primarily controls the flow control valve as previously described in Chapter 11 (“Air Conditioning”). Additionally, the ON position (for current airplanes) provides electrical power for the cabin temperature sensor blower. Selecting the OFF position on airplanes subsequent to SNs 35-098 and 36-028, opens the safety valve if the airplane is on the ground. Early airplanes have a MAX position which opens the valve to full flow for smoke and fume elimination. Current airplanes have no position equivalent to MAX; increased airflow is achieved by positioning the BLEED AIR switches to EMER. The CABIN AIR switch uses DC power from the AIR BLEED circuit breaker on the left essential bus.

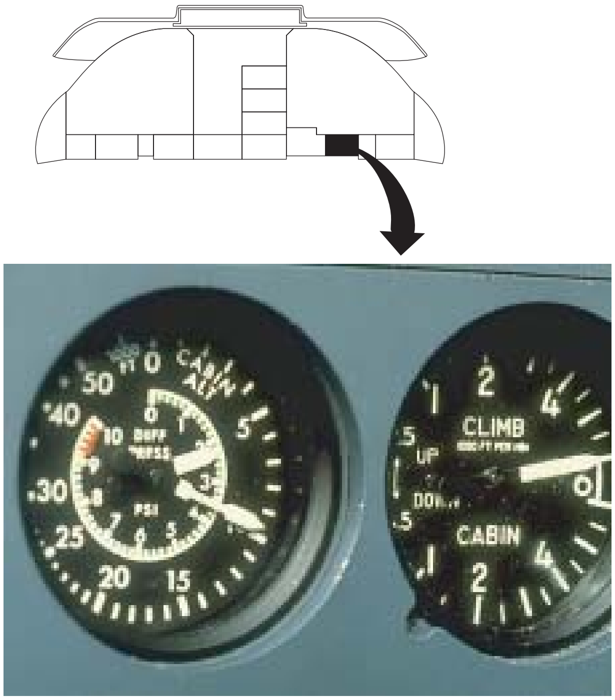


Figure 12-4. CABIN ALT and DIFF PRESS Indicator

INDICATORS

CABIN ALT and DIFF PRESS Indication

Cabin altitude and differential pressure are indicated on a single indicator incorporating two scales and two pointers (See Figure 12-4).

Cabin altitude is indicated by a single pointer and a circular scale on the outer edge with CABIN ALT markings from 0 to 50,000 feet.

The cabin differential pressure is indicated by a circular scale on the inner portion of the indicator and a single pointer. The scale represents differential pressure from 0–10 psi, with a green band from 0–8.9 psi on early airplanes and 0–9.4 psi on current airplanes, a yellow band from 8.9–9.2 psi on early airplanes and 9.4–9.7 psi on current airplanes, and a red band above 9.2 psi on early airplanes and 9.7 psi on current airplanes. Cabin altitude should always be equal to or less than the airplane altitude; therefore, cabin pressure should always be equal to or greater than atmospheric pressure at the airplane altitude. The indicator should normally read approximately .2 psi below the yellow arc.

Cabin Vertical Speed Indicator

The cabin vertical speed indicator is positioned to the right of the cabin altimeter. It provides an indication of cabin climb or descent rates of between 0 and 6,000 feet per minute.

NORMAL SYSTEM OPERATION

BEFORE TAKEOFF

During ground operation, the CABIN AIR switch is normally not turned to ON until just prior to takeoff unless engine bleed air is desired for cabin heating.

When accomplishing the Before Starting Engines checklist in the approved AFM, the crew will normally (1) set the AUTO-MAN switch to AUTO, (2) set the expected cruise altitude on the ACFT (inner) scale of the CABIN CONTROLLER dial, and (3) set the RATE knob to approximately the 9 o'clock position.

When the CABIN AIR switch is turned on prior to takeoff, the flow control valve is opened, allowing engine bleed air to enter the cabin. On SNs 35-099 and subsequent and 36-029 and subsequent, there is a delay of approximately 10 seconds before the safety valve closes.

FLIGHT OPERATION—AUTOMATIC

At liftoff, the squat switch relay box deenergizes all pneumatic solenoids and pressurization begins. The cabin altitude begins to climb at a rate based on the RATE knob setting. It should be adjusted, as necessary, to maintain a comfortable cabin altitude climb rate of approximately 600 feet per minute. As the airplane climbs to cruise altitude, the cabin controller automatically adjusts the outflow valve to give the desired cabin climb rate until the cabin altitude reaches the altitude set on the cabin controller dial. As the airplane continues its climb, the differential pressure increases while the cabin altitude remains constant until the airplane arrives at the selected ACFT altitude. If it is observed that the DIFF PRESS indicator is riding on the yellow/red line, a slightly higher cabin altitude should be selected. Adjust the cabin controller as necessary when changing cruise altitude.

Monitor cabin pressure and differential pressure throughout the flight.

FLIGHT OPERATION—MANUAL

If the cabin controller is not functioning properly, follow the Manual Mode Operation procedures in Section 2 of the approved *AFM*.

Manual mode operation is established when the AUTO–MAN switch is placed to MAN. This closes the manual mode solenoid valve, blocking the automatic pneumatic circuit. The UP–DN lever (cherry picker) is then used to control the outflow valve directly by using the static air source or existing cabin pressure to change position of the outflow valve, causing the cabin to climb or descend, respectively.

The manual control valve is very sensitive, and even small, momentary displacements of the lever will generate significant cabin climb or descent rates.

In manual mode, the cabin altitude must be monitored much more closely than in automatic mode, and the outflow valve position must be adjusted frequently during climbs and descents and when making power adjustments.

DESCENT

During descent for landing, destination field elevation should be set on the CABIN scale of the CABIN CONTROLLER dial. The airplane rate of descent should be controlled so that the descent rate is comfortable (approximately 600 feet/minute).

LANDING

As the airplane descends and reaches the preselected cabin altitude, the outflow valve modulates toward the open position. The cabin should be unpressurized at landing. At touchdown, the squat switch relay box actuates the three pneumatic solenoid valves in the controller, causing the outflow valve to open completely to ensure cabin depressurization. In addition, when the CABIN AIR switch is placed to OFF, the flow control valve closes, and on SNs 35-099 and subsequent and 36-029 and subsequent, an additional solenoid valve is energized open, causing the safety valve to open.

EMERGENCY PRESSURIZATION

An emergency source of pressurization bleed air is provided to increase the flow of air into the cabin in the event of a leak.

SYSTEM OPERATION— SNS 35-002 THROUGH 35-112 AND 36-002 THROUGH 36-031

Emergency pressurization is provided by use of the windshield anti-ice/defog system (Chapter 10). This is accomplished by pushing the IN–NORMAL/OUT–DEFOG knob in, then positioning the WSHLD HT AUTO–MAN switch to AUTO. This causes the defog shutoff valve to fully open and also illuminates the WSHLD HT light. These actions introduce air directly into the cabin area through the pilot's foot warmer and bypass possible leaks in the conditioned bleed-air distribution system. To isolate such a leak, the CABIN AIR switch must then be selected OFF to close the flow control valve (Chapter 11, AIR CONDITIONING, Figures 11-1 through 11-3).

On SNs 35A-082, 35A087 through 35A-112, and 36A-023 through 36A-031, and earlier airplanes incorporating AMK 76-7, the flow control valve is located downstream of the heat exchanger. Engine bleed air is available to the heat exchanger whenever an engine is operating and the BLEED AIR switches are on. Because of this, a pressure switch is installed in the tailcone ducting prior to the heat exchanger. Should this pressure switch actuate (at approximately 47 psi), both red BLEED AIL L and R annunciator lights illuminate to indicate the overpressure condition.

To deactivate emergency pressurization, select MAN and toggle the spring-loaded WSHLD HT switch to OFF until the valve is closed.

SYSTEM OPERATION— SNS 35-113 AND SUBSEQUENT AND 36-032 AND SUBSEQUENT

Emergency pressurization is accomplished by routing bleed air directly into the cabin from either (or both) engine(s) through the emergency pressurization valves. This air completely bypasses the entire manifold and conditioned bleed-air distribution system. See Chapter 9, "Pneumatics."

The valves are spring-loaded to the emergency position and require both servo bleed-air pressure and DC power to cause them to position to normal. Cockpit control of the valves is provided by the three-position (OFF–ON–EMER) BLEED AIR switches, while automatic positioning occurs as a result of excessive cabin altitudes or DC power failure.

With the BLEED AIR switches on, a solenoid on each emergency valve is energized, causing servo bleed-air pressure to move the valve to the NORMAL position.

Positioning either BLEED AIR switch to EMER deenergizes the respective solenoid, causing the servo bleed-air pressure to be blocked; the valve repositions to emergency by spring pressure. At the same time, HP air input to the shutoff and regulator valve is blocked so that only LP air is allowed to enter the cabin.

The emergency pressurization valves are also controlled by two cabin aneroid switches (one for each valve). The aneroids are set to operate at 9,500 feet ±250 feet cabin altitude. Should the cabin altitude reach 9,500 feet ±250 feet, the aneroid switches deenergize the solenoids on the emergency pressurization valves and the valves move to the emergency position. The aneroids reset when the cabin altitude decreases to approximately 8,300 feet; however, the approved *AFM* requires that the cabin altitude be at, or below 7,200 feet before attempting to reset the emergency pressurization valves.

To reset the emergency pressurization valves after they have been positioned to emergency, the BLEED AIR switches, one at a time, must be positioned to OFF momentarily, then back to ON.

On airplane SNs 35-113 through 35-658, and 36-032 through 06-063, not incorporating AMK 90-3, the emergency pressurization valves are powered by the L and R MOD VAL circuit breakers on the left and right main DC buses. These circuit breakers also provide electrical power to the L and R bleed-air shutoff and regulator valves. With a MOD VAL circuit breaker open, the emergency pressurization valve will position to emergency, the bleed-air shutoff and regulator valve will fail open, and HP air to the shutoff and regulator valve will be blocked so only LP air will be allowed to enter the cabin. In this case, positioning the BLEED AIR switch to OFF will not stop airflow into the cabin since DC electrical power is required to close the bleed-air shutoff and regulator valve.

On airplane SNs 35-659 and subsequent, 36-064 and subsequent, and earlier airplanes modified by AMK 90-3, the emergency pressurization valves are powered by the L and R EMER PRESS circuit breakers on the left and right main DC buses. On these airplanes, the bleed-air shutoff and regulator valves are powered by separate circuit breakers labeled L and R BLEED AIR, also located on the left and right main DC buses. With an EMER PRESS circuit breaker open, the emergency pressurization valve will position to emergency and the bleed-air shutoff and regulator valve will remain open. In this case, positioning the

BLEED AIR switch to OFF will stop airflow into the cabin since DC electrical power, from the BLEED AIR circuit breaker, will be available to close the bleed air shutoff and regulator valve.

See Chapter 9, "Pneumatics," for additional in formation on the bleed-air shutoff and regulator valves.

During the first engine start, the valves will automatically shift position from emergency to normal as HP servo air pressure from the engine becomes available. A slight rush of air into the cabin is normal during start.

Tables 12-1 and 12-2 provide a description of the automatic protection and warning features for cabin depressurization.

Table 12-1. AUTOMATIC PROTECTION AND WARNING FEATURES—SNs 35-002 THROUGH 35-112 AND 36-002 THROUGH 36-031

CABIN ALTITUDE	PROTECTION AND WARNING
10,000 ± 250 feet	• Pressurization aneroid automatically switches the system to manual control. • Cabin altitude warning horn sounds—initiate emergency descent.
11,000 ± 1,000 feet	• Cabin altitude limiters actuate.
14,000 ± 750 feet	• Passenger oxygen masks are deployed and cabin overhead lights are illuminated.

* The differential pressure relief for the outflow valve is 8.9 psid, and the differential pressure relief for the safety valve is 9.2 psid.

Table 12-2. AUTOMATIC PROTECTION AND WARNING FEATURES—SNs 35-113 AND SUBSEQUENT AND 36-032 AND SUBSEQUENT

CABIN ALTITUDE	PROTECTION AND WARNING
8,700 ± 250 feet	• Pressurization aneroid automatically switches the system to manual control. • CABIN ALT caution light illuminates.
9,500 ± 250 feet	• Emergency pressurization valves are activated by aneroid switches, directing engine bleed air directly into the cabin.
10,100 ± 250 feet	• Cabin altitude warning horn sounds—initiate emergency descent.
11,500 ± 1,500 feet	• Cabin altitude limiters actuate.
14,000 ± 750 feet	• Passenger oxygen masks are deployed and cabin overhead lights are illuminated.

* The differential pressure relief for the outflow valve is 9.4 psid, and the differential pressure relief for the safety valve is 9.7 psid.

EMERGENCY PRESSURIZATION OVERRIDE SWITCHES

On SNs 35-605 and subsequent and 36-056 and subsequent and earlier SNs incorporating AAK 84-4, two emergency pressurization override switches (Figure 12-5) allow the crew to override the 9,500-foot cabin aneroids to facilitate landing at high-elevation airports.

The guarded switches are labeled “L” and “R EMER PRESS” and have positions labeled “OVERRIDE” and “NORMAL.” With the guards down, the switches are in the NORMAL position. Lifting the guards and moving the switches to the OVERRIDE position disconnects the 9,500-foot aneroids from the system. The switches can also be used:

- To reset an emergency valve which has inadvertently positioned to emergency due to a malfunctioning aneroid

- To reset the emergency valves in order to restore windshield and stab/wing anti-icing (at any altitude)

In either case, selecting OVERRIDE must be followed by cycling the BLEED AIR switch(es) to OFF and then to ON, provided DC power is available and the MOD VAL (or EMER PRESS, as applicable) circuit breaker(s) are in.

Figure 12-5. Emergency Pressurization Override Switches

QUESTIONS

1. To regulate cabin pressure, the cabin controller modulates the:
   A. Cabin safety valve
   B. Flow control valve
   C. Outflow valve
   D. Primary differential pressure relief valve

2. Illumination of the amber CABIN ALT light (if installed) indicates:
   A. Cabin altitude is at or above 8,750 ±250 feet, and the pressurization control system is in manual mode.
   B. Cabin altitude is at or above 8,750 ±250 feet, and the pressurization control system may be in either AUTO or MAN mode.
   C. Cabin altitude is at or above 9,500 ±250 feet, and the emergency pressurization mode has activated.
   D. The CABIN AIR switch is in the OFF position.

3. On airplanes with emergency pressurization valves, if DC power fails:
   A. Cabin pressurization must be controlled manually with the UP–DN knob.
   B. Cabin pressure will dump.
   C. The emergency pressurization valves automatically actuate to provide emergency cabin pressure.
   D. The flow control valve fails closed.

4. The cabin altitude warning horn sounds when cabin altitude reaches approximately:
   A. 8,750 feet
   B. 9,500 feet
   C. 10,100 feet
   D. 11,500 feet

5. To dump residual cabin pressure on touchdown:
   A. The outflow valve opens automatically.
   B. The cabin safety valve opens automatically.
   C. The flow control valve closes automatically.
   D. The bleed-air shutoff and regulator valves close automatically.

6. On airplanes without the emergency pressurization valves, if DC power fails:
   A. The windshield anti-ice/defog system can be used in the event of a pressurization failure.
   B. The cabin will remain pressurized, but emergency pressurization capability will be lost.
   C. The flow control valve fails closed.
   D. The bleed-air shutoff and regulator valves fail closed.

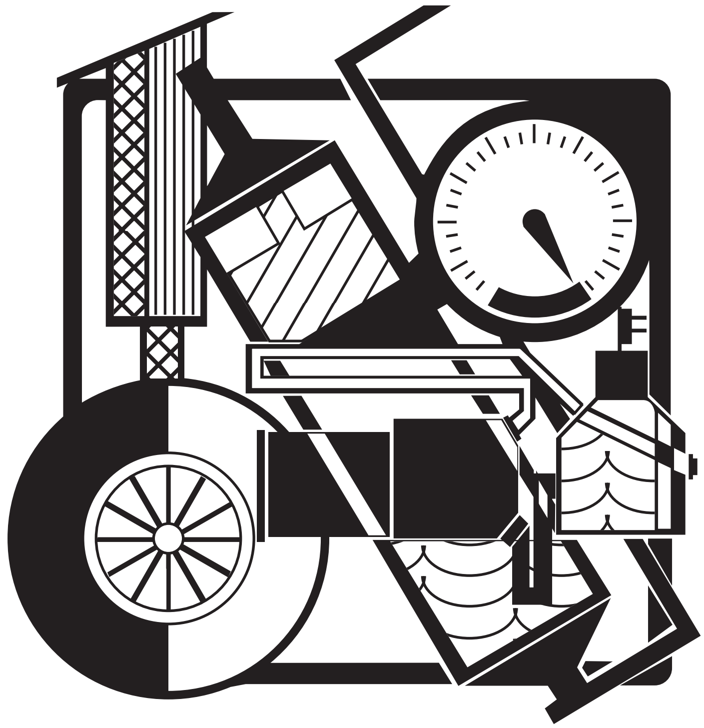
7. On all airplanes, if DC power fails:
   A. Pressurization control reverts to manual control.
   B. The manual mode of pressurization control cannot be selected or maintained.
   C. Cabin pressure is not controlled.
   D. The cabin slowly depressurizes.

CHAPTER 13
HYDRAULIC POWER SYSTEMS

CONTENTS

ILLUSTRATIONS

CHAPTER 13
HYDRAULIC POWER SYSTEMS



INTRODUCTION

Two engine-driven pumps normally provide hydraulic pressure for operation of the landing gear, wheel brake, flap, spoiler/spoileron, and Dee Howard TR-4000 thrust reverser (if installed) subsystems. An electrically driven auxiliary pump incorporated for use in the event of system failure is normally used only on the ground for operation of the brakes and flaps when both engines are shut down. It ***cannot*** be used to operate the spoiler/spoileron system.

GENERAL

A 1.9 gallon reservoir pressurized by regulated engine bleed air ensures a positive supply of MIL-H-5606 fluid to both engine-driven pumps and to the auxiliary pump. The 4-gpm, variable-volume, engine-driven pumps are supplied from supply lines connected to the side of the reservoir at approximately the .4-gallon level. This limits the amount of fluid the engine-driven pumps can deliver to a system leak, reserving fluid for the auxiliary pump that is connected to the bottom of the reservoir.

Hydraulic shutoff valves installed at the reservoir in each engine-driven pump supply line can be closed from the cockpit in the event of fire or when maintenance is to be performed.

An accumulator precharged with dry air or nitrogen dampens pressure surges and helps maintain system pressure. A direct-reading indicator on the center instrument panel displays system pressure. An amber annunciator light warns of low pressure.

There are three filters in the system—one in each pressure line, and one in the return line.

A system relief valve set to relieve at 1,700 psi prevents system damage by relieving excessive pressure into the return line.

Pressure from the engine-driven pumps is available to actuate the spoilers/spoilerons, flaps, landing gear, brakes, and TR-4000 thrust reversers (if installed). A check valve prevents auxiliary pump actuation of the spoilers/spoilerons.

The reservoir and the accumulator are located in the tailcone. Reservoir fluid level should be just above the sight glass with zero system pressure. Fluid is low if the level can be seen in the glass or if fluid is not visible.

Accumulator precharge, indicated by the gage on the accumulator, should be 850 psi when hydraulic pressure is zero.

Controls and indicators for the system are shown-in Figure 13-1.

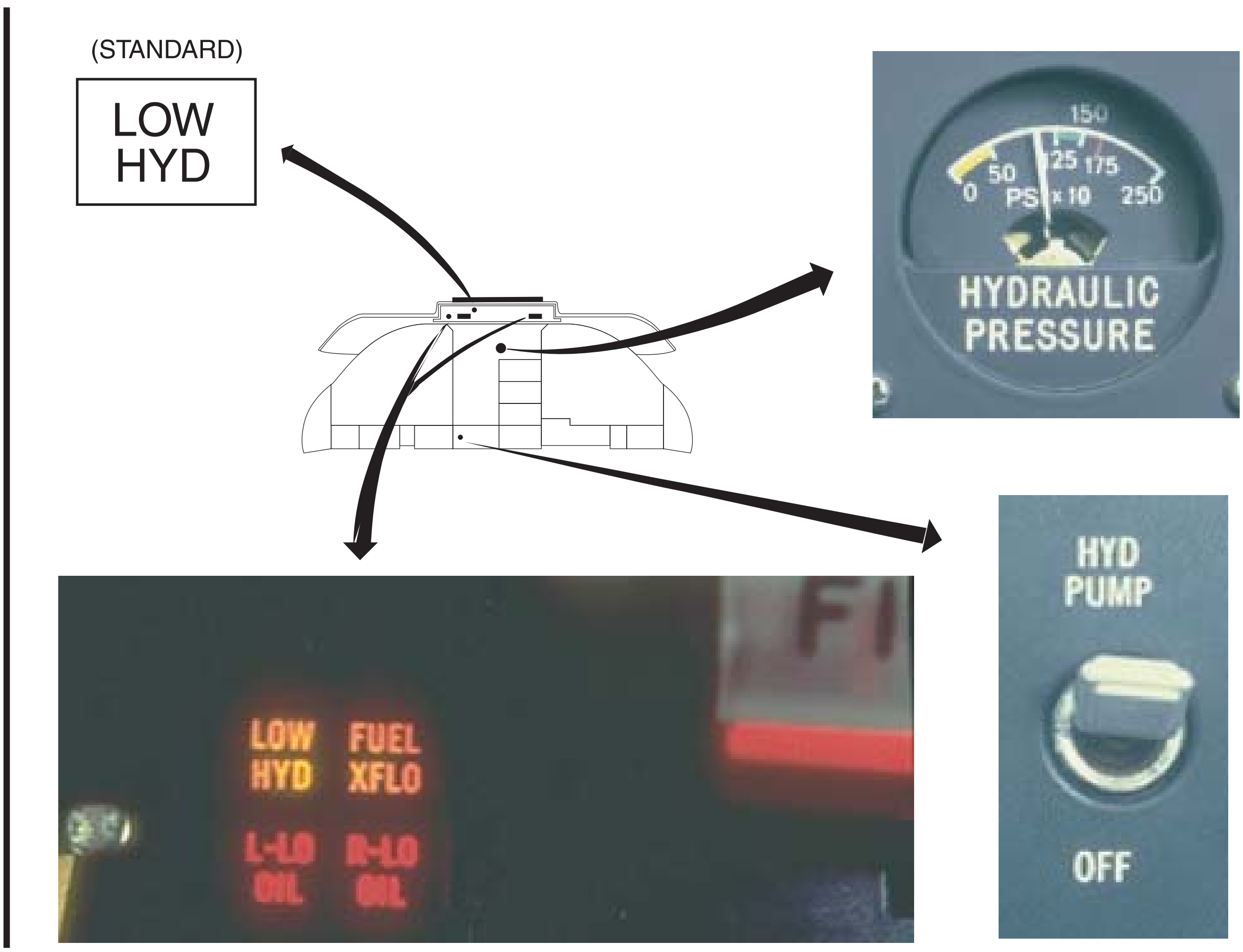


Figure 13-1. Controls and Indicators

HYDRAULIC SYSTEM OPERATION

Unless there is residual hydraulic system pressure, the auxiliary hydraulic pump must be operated to provide pressure for setting the parking brakes prior to engine start. Placing the HYD PUMP switch in the on (HYD PUMP) position starts the auxiliary pump, assuming both engines are shut down and system pressure is below 1,125 psi (Figure 13-2).

As pressure increases, a pressure switch actuates at 1,250 psi to extinguish the amber LOW HYD light on the annunciator panel. (All annunciators are shown in Annunciator Panel section.) At approximately 1,250 psi, the pressure switch stops the auxiliary pump. The HYD PUMP switch should then be positioned to OFF, where it normally remains unless flap operation is required prior to engine start. The LOW HYD light will illuminate if pressure drops below 1,125 psi.

If the HYD PUMP switch is left on, the pressure switch will cycle the pump between 1,125 psi and 1,250 psi.

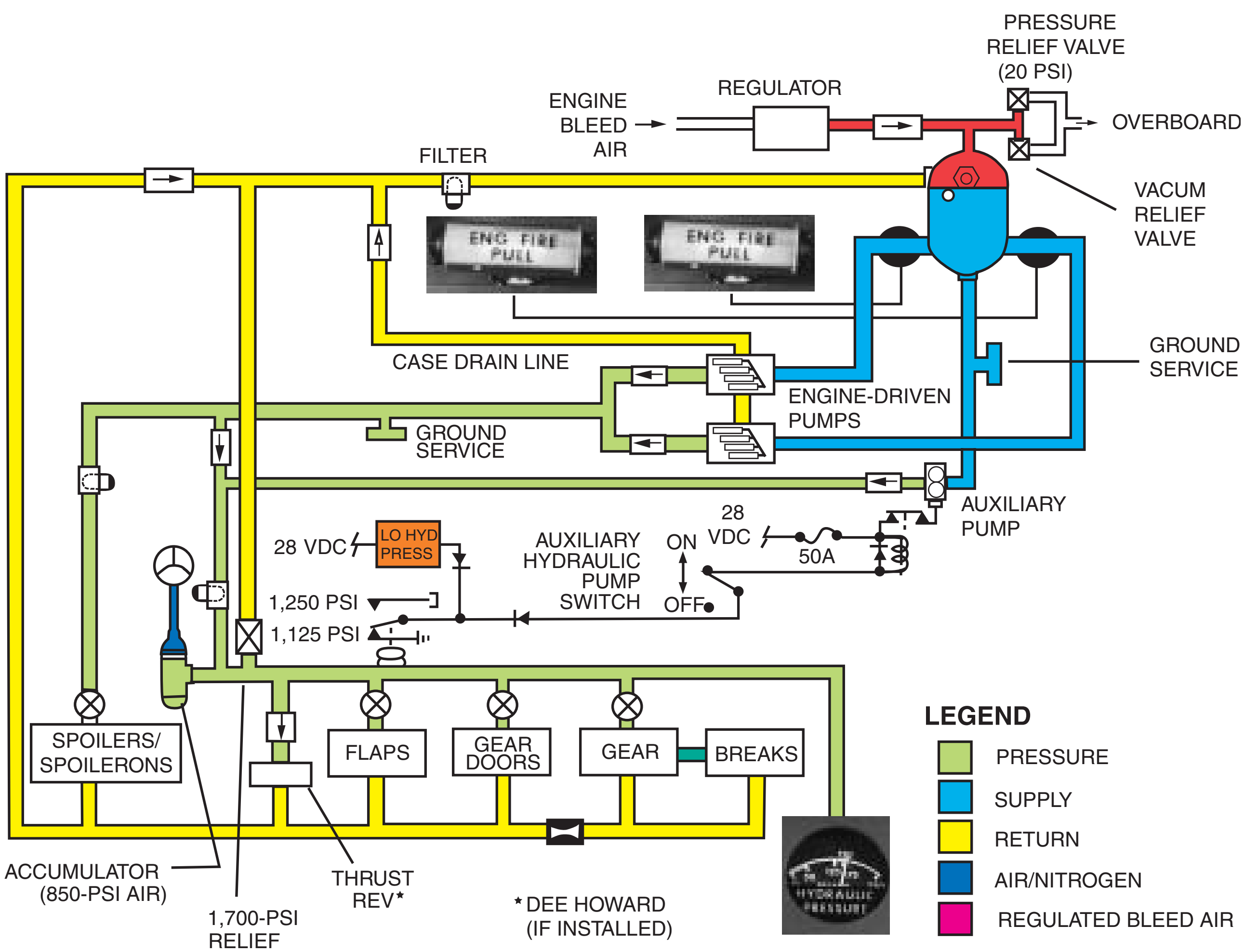


Figure 13-2. Hydraulic System Schematic

In the event of engine fire or when maintenance is to be performed, the DC motor-driven shutoff valves can be closed by pulling the appropriate FIRE handle on the glareshield. Pulling either handle also arms the fire-extinguisher system; therefore, if valve closing is to facilitate maintenance, the applicable FIRE EXT circuit breakers(s) should be pulled to prevent accidental discharge of the bottles. The shutoff valves are opened by pushing in the appropriate handle(s). The shutoff valves operate on DC power supplied through the L and R FW SOV circuit breakers on the left and right essential buses, respectively.

After starting the first engine, the HYDRAULIC PRESSURE indicator should be checked to verify engine-driven pump operation. Pressure should stabilize at 1,550 ±25 psi, indicating that the engine-driven pump is operating properly.

When the second engine is started, there is no change in pressure indication, but capacity is doubled. There is no positive indication that the second pump is operating properly; therefore, after landing, operation of the second pump can be verified by shutting down the engine started first and actuating a hydraulic subsystem.

If an engine-driven pump fails in flight, the other engine-driven pump is capable of meeting system demands.

Loss of fluid due to a system leak is the most probable cause of complete loss of hydraulic pressure. If all hydraulic system pressure is lost, the LOW HYD light will illuminate as pressure decreases below 1,125 psi. Do not operate the auxiliary pump until alternate landing gear extension procedures have been accomplished, as directed by the approved *AFM*. Otherwise, the auxiliary pump may discharge the .4 gallon of reserve fluid through the same leak.

There is no circuit-breaker protection in the cockpit for the auxiliary pump; it is powered directly from the battery charging bus through a 50-ampere current limiter.

HYDRAULIC SUBSYSTEMS

Operation of the hydraulic subsystems is presented in Chapter 14, “Landing Gear and Brakes,” Chapter 15, “Flight Controls” (flaps and spoiler/spoilerons), and Chapter 7, “Powerplant” (Dee Howard TR-4000 thrust reversers).

QUESTIONS

1. Normal hydraulic system pressure with the engine-driven pumps operating is:

A. 1,400 ±50 psi
B. 1,550 ±25 psi
C. 1,650 psi
D. 1,700 psi

2. The hydraulic shutoff valves are closed:

A. By pulling the engine FIRE handles
B. Automatically when the FIRE warning light comes on
C. By the GEN switch in the OFF position
D. By the BLEED AIR switches.

3. In the event of hydraulic system pressure failure in flight:

A. Immediately turn the HYD PUMP switch on.
B. Turn the HYD PUMP switch on when the LOW HYD light illuminates.
C. Refer to the Abnormal Landings checklist.
D. Refer to the Hydraulic System Failure/Alternate Gear Extension checklist.

4. In the event of hydraulic system failure, the LOW HYD light will illuminate at:

A. 1,125 psi
B. 1,500 psi
C. 1,250 psi
D. 850 psi

5. During a hydraulic system failure, ***do not*** operate the following subsystem using the auxiliary hydraulic pump:

A. Landing gear
B. Spoilers
C. Brakes
D. Flaps

6. The approved fluid for the hydraulic system is:

A. MIL-H-5606
B. MIL-O-M-332
C. Skydrol
D. MIL-H-2380

7. The operational time limit of the auxiliary pump is:

A. 5 minutes on, 15 minutes off
B. 5 minutes on, 25 minutes off
C. 3 minutes on, 20 minutes off
D. 2 minutes on, 30 minutes off

8. The auxiliary hydraulic pump will provide approximately:

A. 1,200 psi
B. 1,550 psi
C. 1,700 psi
D. 1,250 psi

9. If DC electrical power is applied to the airplane and residual hydraulic pressure is 1,450 psi:

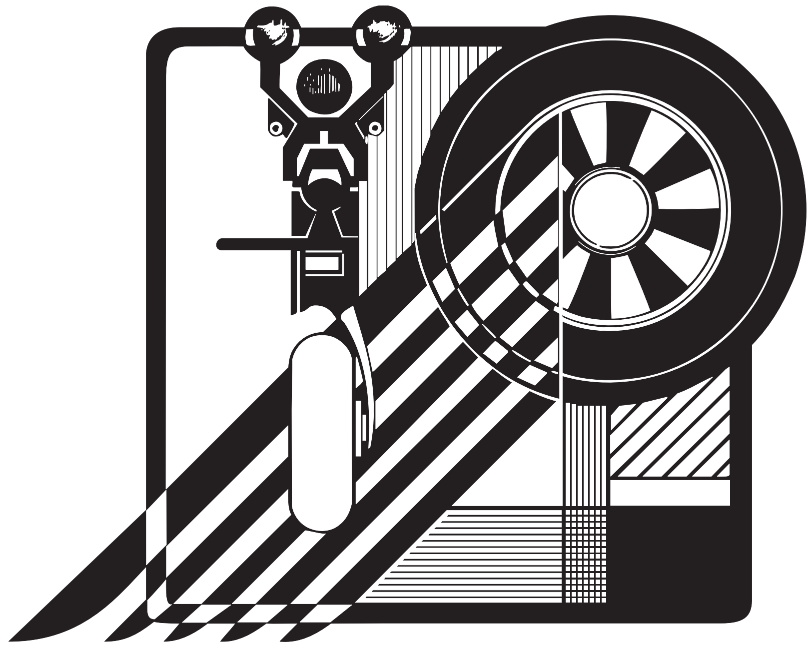
A. The auxiliary hydraulic pump will not operate when the HYD PUMP switch is on.
B. The LOW HYD light will be out.
C. 1,450 psi will be shown on the HYDRAULIC PRESSURE indicator.
D. All the above

CHAPTER 14
LANDING GEAR AND BRAKES

CONTENTS

ILLUSTRATIONS

CHAPTER 14
LANDING GEAR AND BRAKES



INTRODUCTION

The retractable landing gear is electrically controlled and hydraulically operated. The main gear incorporates dual wheels equipped with individual hydraulic brakes and retracts inboard. The single-wheel, self-centering nose gear incorporates an electrical steering system and retracts forward. Alternate gear extension and emergency braking are pneumatic. An antiskid system is incorporated into the normal hydraulic braking system.

GENERAL

The landing gear has three air-hydraulic shock struts. The main gear outboard doors are mechanically linked to the gear and move with it. The inboard doors are hydraulically operated and close when the gear is fully extended or retracted. An air bottle is provided for alternate gear extension and emergency braking. The gear actuators incorporate integral downlocking devices; downlock pins are not required. Gear position indications are displayed on the copilot's instrument panel.

The hydraulic brake system is controlled by four valves (two for each pilot) linked to the rudder pedals. Hydraulic system pressure is metered to the self-adjusting multiple disc brake assemblies in proportion to pedal deflection.

The antiskid system provides maximum deceleration without skidding the tires. When the system is operating, wheel speed transducers (generators) furnish wheel speed information to a control box which signals the antiskid servo valves to modulate braking pressure. The parking brake is set by pulling a handle on the throttle quadrant, and depressing the brake pedals, trapping hydraulic pressure in the brake assemblies.

The variable-authority, electric nosewheel steering system operates only on the ground. When the system is engaged, a computer determines the amount of nosewheel deflection allowable, based on rudder pedal movement and taxi speed, and uses a DC electric motor to deflect the nosewheel accordingly. Maximum authority is 45º either side of center at slow speeds, decreasing as speed increases.

LANDING GEAR

INDICATING SYSTEM

General

The landing gear position indicating system consists of three red lights and three green lights, a test switch, and an aural warning horn.

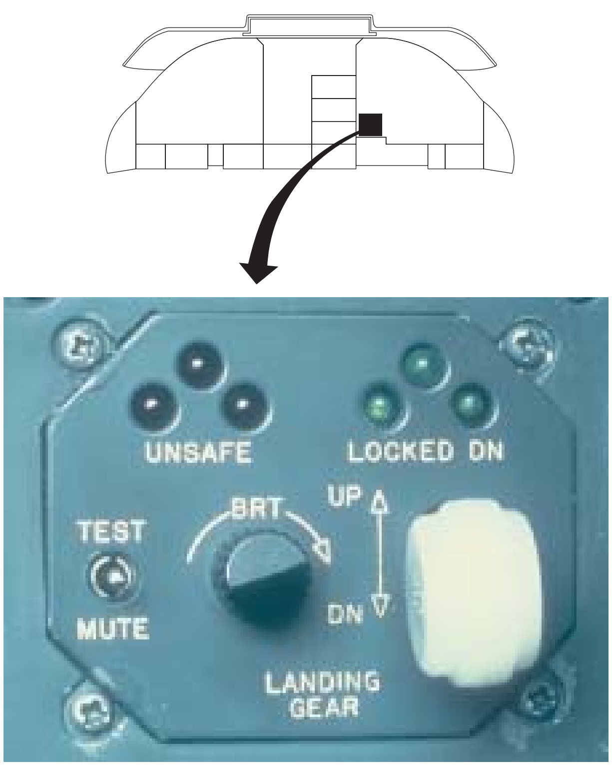


Figure 14-1. Gear Position Indicator Lights

Gear Position Lights

The three green LOCKED DN lights (Figure 14-1) are illuminated by their respective downlock switches on the gear actuators.

As each gear locks down, the corresponding green LOCKED DN light illuminates. During gear retraction the lights extinguish when the downlocks are hydraulically released.

The nose gear red UNSAFE light is illuminated when the nose gear is in transit (neither down and locked nor up and locked). When the nose gear is locked in either the up or the down position, the light extinguishes.

The two main gear red UNSAFE lights illuminate whenever the respective main gear door is unlocked. As each inboard door latches up (during extension or retraction), the corresponding red light extinguishes.

Indications for gear down-and-locked, up-and-locked, and in-transit conditions are shown in Figure 14-2.

If the gear is extended with the alternate (pneumatic) system, all three green lights and the two main gear red lights will be illuminated (both main gear doors will remain fully extended).

The position lights are tested by holding the TEST/MUTE switch on the landing gear control panel in the TEST position. All six lights will illuminate and the warning horn will sound. The lights can be dimmed with the dimming rheostat (Figure 14-2), provided the navigation lights are on; otherwise they will be at maximum intensity.

Circuitry related to the left and right main gear green position lights is common with the landing/taxi light for that side. Confirmation of main gear downlocking (after bulb testing) can be made by switching on the respective LDG LTS switch.

Nose gear green light circuitry is common with the engine synchronizing system (if installed). Confirmation of nose gear downlocking (after

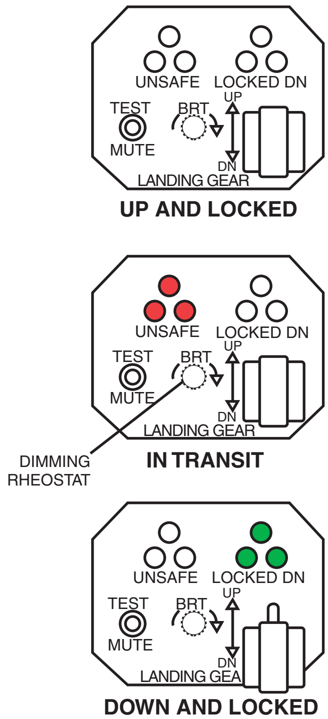


Figure 14-2. Gear Position Indications

bulb testing) is made by positioning the ENG SYNC switch on the pedestal to ENG SYNC (on) and observing that the amber ENG SYNC light on the annunciator panel illuminates.

Landing Gear Warning System

The aural warning horn will sound and three red UNSAFE lights will illuminate when all of the following conditions are present:

- Landing gear is not down and locked.
- Altitude is less than 14,500 ±500 feet.
- Either thrust lever is retarded below approximately 55-60% N_1.
- Airspeed is below 170 KIAS (FC 530 airplanes only)

At altitudes above 14,500 ±500 feet, the horn will ***not*** sound when the thrust levers are retarded, and the UNSAFE lights ***may*** illuminate. The horn also sounds when the flaps are extended beyond 25º if the landing gear is not down and locked, regardless of thrust lever position or altitude.

Holding the TEST/MUTE switch in the TEST position illuminates all six position indicator lights and sounds the horn. Momentarily positioning the switch to MUTE silences the horn when the thrust levers are retarded, and the gear is not down and locked.

The horn ***cannot*** be muted when the gear is not down and locked and the flaps are extended beyond 25º.

GENERAL

Main Gear Components

Each main gear consists of a conventional air-hydraulic shock strut, dual wheels, scissors, squat switch, main gear actuator, inboard and outboard doors, and an inboard door actuator (Figure 14-3).

Figure 14-3. Main Gear

The main gear hydraulic actuator also serves as a side brace when the gear is extended. It features an integral downlock mechanism that can be unlocked only by hydraulic pressure on the retract side; therefore, no downlock pins are provided. Each main gear scissors link actuates a squat switch.

The main gear is hydraulically held in the retracted position and is enclosed by an outboard and an inboard door. The outboard door is mechanically linked to, and travels with, the gear. The inboard door is hydraulically actuated, electrically sequenced by microswitches, and held retracted by hydraulic pressure and a spring-loaded, overcenter uplatch that is released by a hydraulic actuator.

Proper shock strut inflation is an important consideration. When the airplane weight is on the gear, the amount of strut extension will vary with the airplane load. With a full fuel load and no passengers or baggage aboard, 3 to 3½ inches of bright surface should be visible on the lower portion of the main gear strut.

Main Gear Wheel and Tires

Each main gear wheel incorporates a fusible plug that prevents tire blowout caused by excessive heat resulting from hard braking. Tires must be changed when the tread has worn to the base of any groove at any location or if the cord is exposed. Main gear tire pressure is determined by airplane gross weight certification.

Nose Gear Components

The nose gear consists of an air-hydraulic shock strut incorporating a self-centering device, a nosewheel steering actuator, and mechanically operated doors (Figure 14-4).

The nose gear strut is conventional, with two exceptions: it does not have a scissors and the nosewheel steering actuator motor is mounted on top of the strut housing.

The nose gear actuator incorporates an integral downlock mechanism to maintain a positive downlocked condition; therefore, a downlock pin is not required. As with the main gear actuator, the locking mechanism can be released only by hydraulic pressure on the retract side. The gear is held retracted by hydraulic pressure and a spring-loaded uplatch hook that engages the uplatch roller on the forward side of the strut. The uplatch hook is released by a hydraulic actuator.

Figure 14-4. Nose Gear

When retracted, the nose gear is enclosed by two doors that are linked to, and travel with, the gear.

An improperly centered nosewheel could jam in the wheel well; therefore, the nose strut incorporates a self-centering mechanism. At liftoff, two cams within the strut are engaged by strut air pressure to center the wheel (Figure 14-5).

Since nosewheel centering depends on air pressure in the strut, proper inflation of the strut is especially important. When the airplane weight is on the gear, the amount of strut extension will vary with airplane load. With a full fuel load and no passengers or baggage aboard, 5¼ to 5¾ inches of bright surface should be visible on the lower portion of the nose gear strut.

Because the cams cannot center the wheel if it is swiveled 180º from the normal position, the nose gear should be checked on the exterior inspection to ascertain that the gear uplatch roller (Figure 14-4) is facing forward.

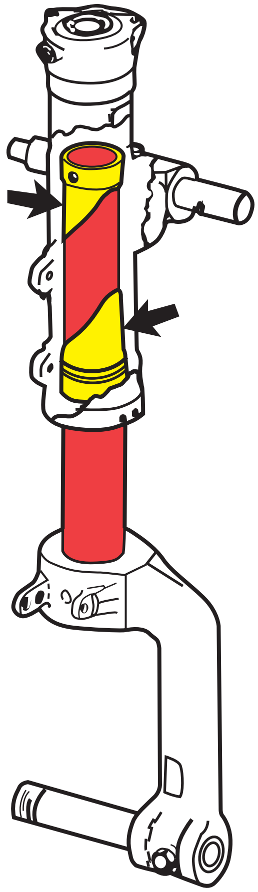


Figure 14-5. Nose Gear Centering Cams

Nose Gear Wheel and Tire

The nosewheel tire is chined to deflect water or slush spray (up to ¾-inch deep) away from the engine intakes during takeoff or landing.

Nosewheel tire pressure should be maintained at from 104 to 114 psi when the airplane is loaded and the crew is in the cockpit.

OPERATION

The landing gear system incorporates two solenoid-operated hydraulic control valves—one for operation of the main gear inboard doors and one for gear operation. ***Both*** inboard doors must be fully open before the gear can be extended or retracted.

The door control valve is energized to the door–open position when the landing gear selector switch is placed in either the UP or the DN position. It is energized to the door–close position by main gear-operated switches when ***both*** gear are fully retracted or down and locked.

The gear control valve is energized to the extend or the retract position by switches sensing the full open position of ***both*** main gear inboard doors. During retraction, the circuit is routed through ***both*** squat switches to ensure that the airplane is off the ground before the valve can be energized to the retract position.

Normal landing gear operation requires DC power supplied through the gear circuit breaker on the right essential bus.

Normal Retraction

Positioning the landing gear selector switch to UP energizes the door control valve to the open position, directing pressure to release the main gear inboard door uplatches and to open the doors. The two red main gear UNSAFE lights illuminate simultaneously with uplatch release.

When both inboard doors are fully open, the door–open switches are actuated. When both door–open switches are actuated and both squat switches are in the airborne position, the gear control valve energizes to the retract position, and hydraulic pressure is directed to retract the landing gear (Figure 14-6). The three green LOCKED DN lights extinguish, and the red nose gear UNSAFE light illuminates.

When the nose gear has fully retracted, the red nose gear UNSAFE light extinguishes. When ***both*** main gear are fully retracted, two gear-up trunnion switches are actuated to energize the door control valve to the closed position. Pressure closes the gear inboard doors, which lock in position by spring tension on the door uplatches, and the two red main gear UNSAFE lights extinguish.

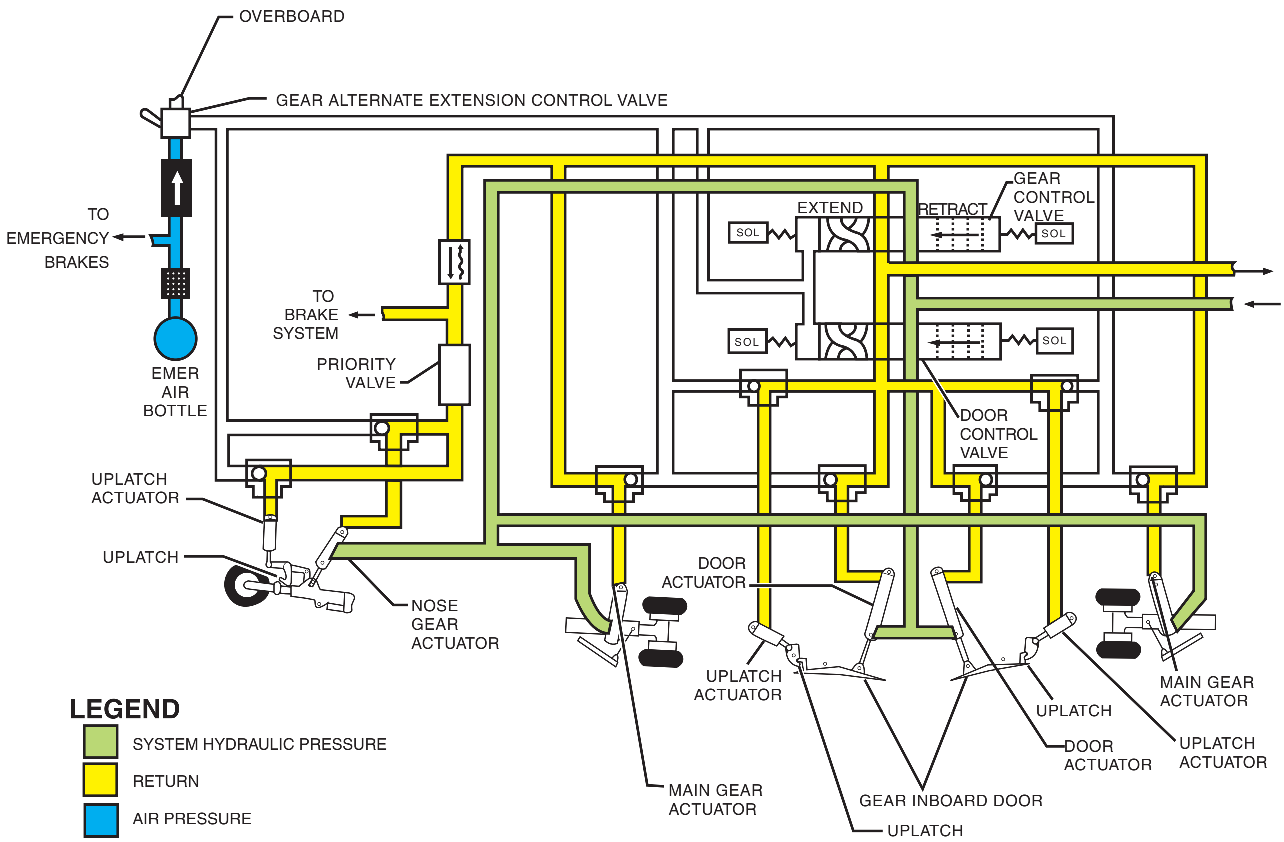


Figure 14-6. Landing Gear Retracted

Normal Extension

Positioning the landing gear selector switch to DN energizes the door control valve to the open position, directing pressure to release the main gear inboard door uplatches and to open the doors. The two red main gear UNSAFE lights illuminate simultaneously with uplatch release.

When ***both*** inboard doors are fully open, the door–open switches are actuated to energize the gear control valve to the down position. This directs pressure to release the nose gear uplatch and extend the nose and main gear (Figure 14-7). The red nose gear UNSAFE light illuminates.

When the gear is fully down and locked, the three green LOCKED DN lights illuminate and the red nose gear UNSAFE light extinguishes.

Circuitry is completed by ***both*** main gear downlock switches to energize the door control valve to the closed position. Pressure closes the gear inboard doors (Figure 14-7), which lock in position by spring tension on the door uplatches, and the two red main gear UNSAFE lights extinguish.

Alternate Extension

General

The alternate gear extension system is pneumatically operated by a bottle charged to 1,800–3,000 psi with dry air or nitrogen. Bottle pressure is shown on the direct-reading EMERGENCY AIR indicator on the center instrument panel (Figure 14-8). The bottle also provides pressure for emergency braking.

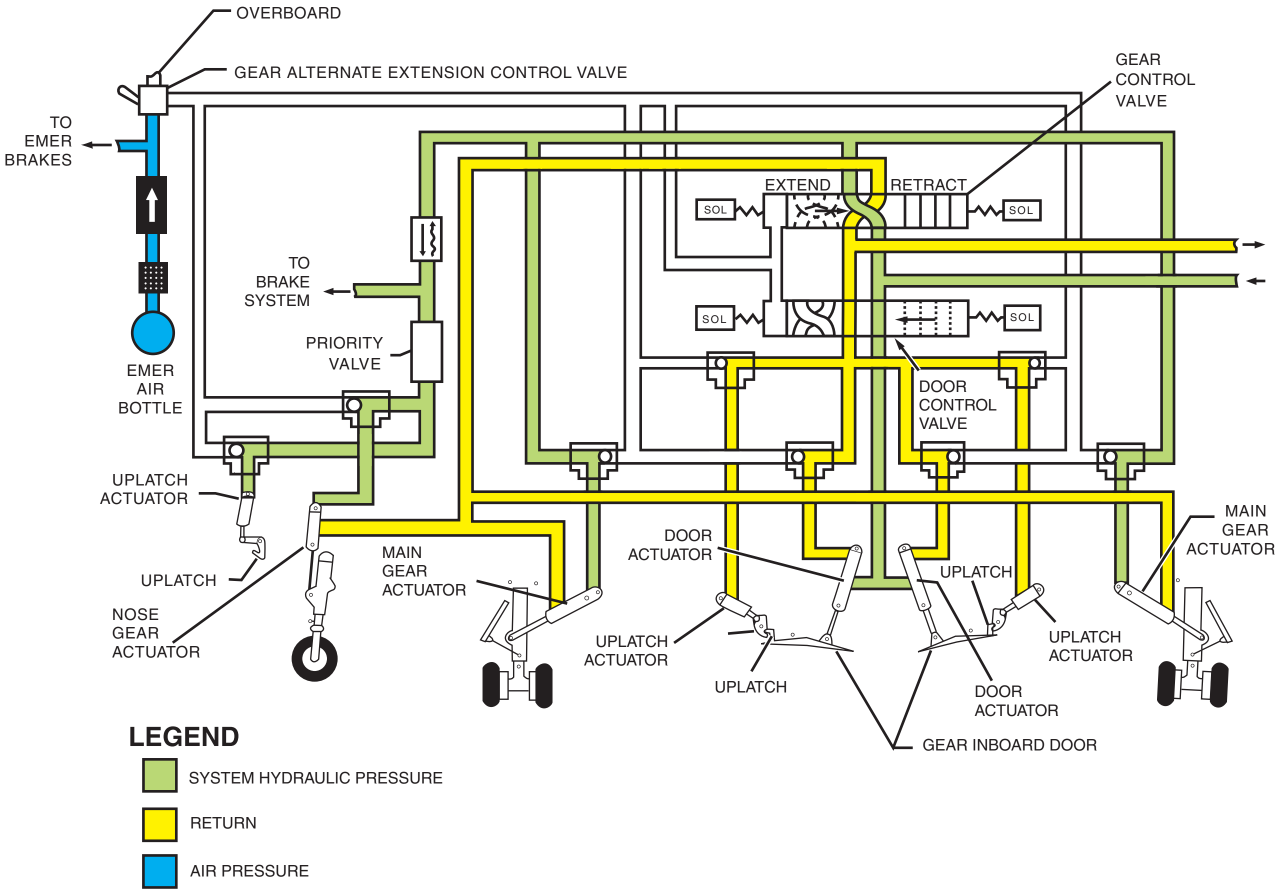


Figure 14-7. Landing Gear Extended

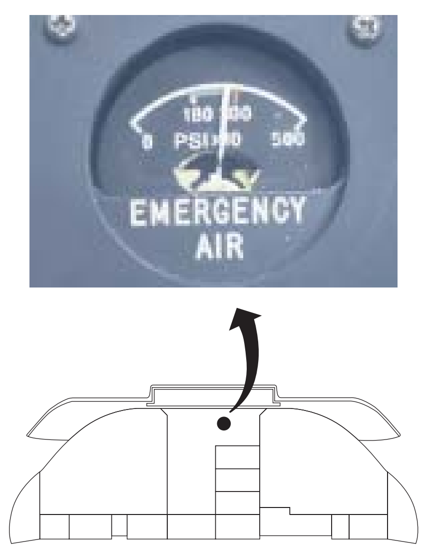


Figure 14-8. Emergency Air Pressure Indicator

Prior to using the system, the landing gear selector switch (Figure 14-2) should be placed in the DN position and the GEAR circuit breaker on the right essential bus should be pulled. This will prevent inadvertent gear retraction subsequent to a successful extension. The system is activated by pushing down the emergency gear lever on the right side of the pedestal (Figure 14-9). The lever has a ratchet to keep it in the down position, once activated, and can be raised only by manually actuating the release tab while simultaneously lifting the emergency gear lever.

Operation

Pushing the emergency gear lever down opens a valve to release air bottle pressure to position the gear control and door control valves to the extend position (Figure 14-10). This provides a return flow path for fluid in the retract side of the gear and door actuators. The air pressure

Figure 14-9. Alternate Extension Controls

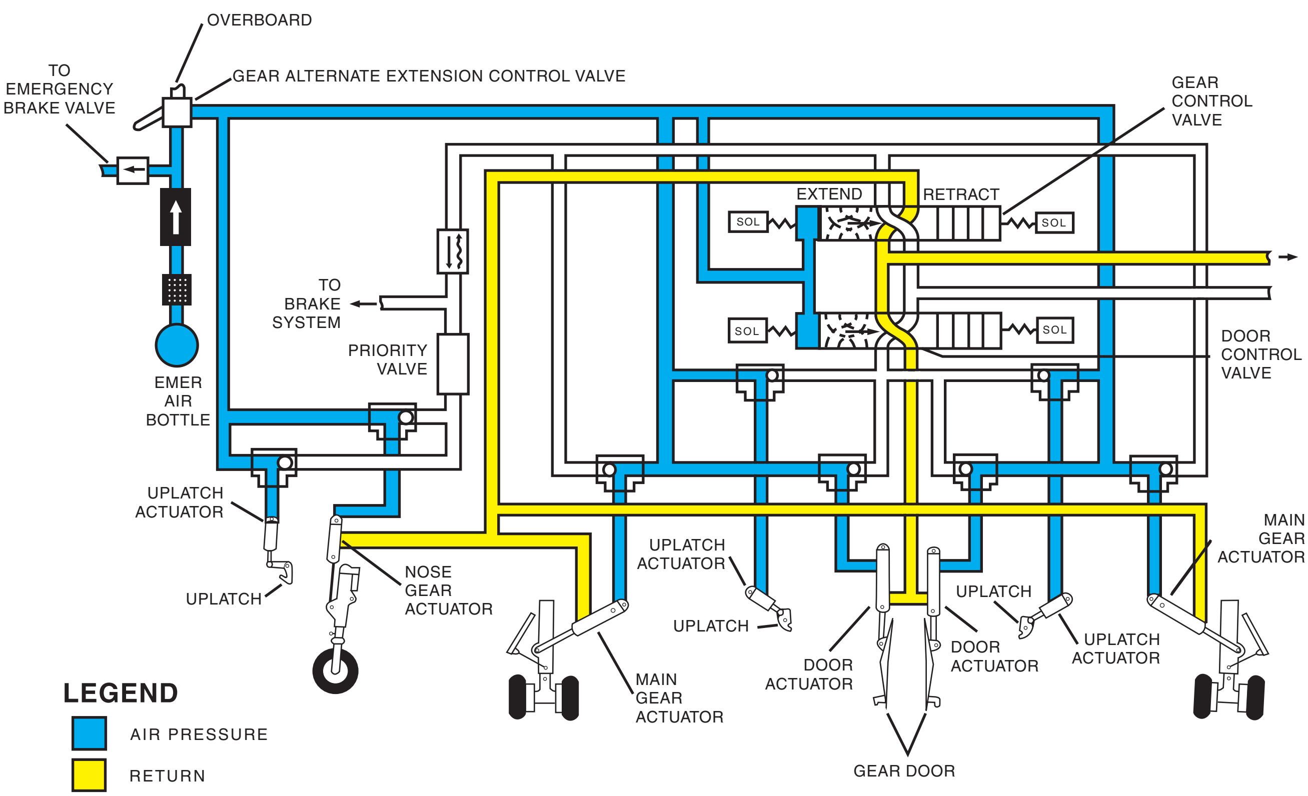


Figure 14-10. Alternate Landing Gear Extension

also repositions the shuttle valves to accomplish the following:

- Release the nose gear uplatch and the main gear door uplatches.
- Open the main gear inboard doors.
- Extend all three gear.

Since no provision is made to close the main inboard doors, the two main gear red UNSAFE lights will remain illuminated. The three green LOCKED DN lights will illuminate.

In a hydraulic failure situation, after the gear is down and locked, air pressure must be bled from the gear system by lifting the release tab (Figure 14-9) and raising the emergency gear lever to the normal position. This closes the valve on the emergency air bottle and isolates air pressure from the gear system, preventing a possible leak in the gear system from depleting air pressure that might be required for emergency braking.

If alternate extension is required due to an electrical fault, the emergency gear lever must remain in the down position to prevent subsequent inadvertent retraction of the gear.

BRAKES

GENERAL

The brake system is powered by hydraulic system pressure from the nose gear down (extend) line. The brakes can be applied by either pilot. The system has four multidisc, self-adjusting brake assemblies, one for each main gear wheel, operated by power brake valves linked to the top section of the rudder pedals. The left pedals control both brake assemblies on the left gear; the right pedals control the brake assemblies on the right gear. Braking force is in direct proportion to pedal application unless modulated by the antiskid system. The antiskid system, monitored by the red ANTISKID GEN warning lights, permits stopping in the shortest possible distance for any given runway condition. (Warning and annunciator lights are shown in Annunciator Panel section.) Parking brakes can be set by pulling a handle on the center pedestal. A pneumatic emergency brake system is used to stop the airplane if hydraulic pressure is lost. Neither antiskid protection nor differential braking is available during emergency braking.

NORMAL OPERATION

When either pilot depresses a brake pedal, the associated brake valve meters system hydraulic pressure through shuttle valves (one in each main pressure line), parking brake valves, antiskid valves, brake fuses, and a second set of shuttle valves, one for each of the four brake assemblies (Figure 14-11). The first set of shuttle valves determines whether the pilot or the copilot has control of the brakes (highest pressure predominating).

Pistons in each brake assembly move a pressure plate, forcing the stationary and rotating discs together against a backing plate to produce the braking action. Depressing one pedal applies both brakes on the corresponding main gear; therefore, differential braking is available, if required.

Releasing pedal pressure repositions the brake valve, and springs in the brake assembly force fluid back through the brake valves to the reservoir, thereby releasing the brakes.

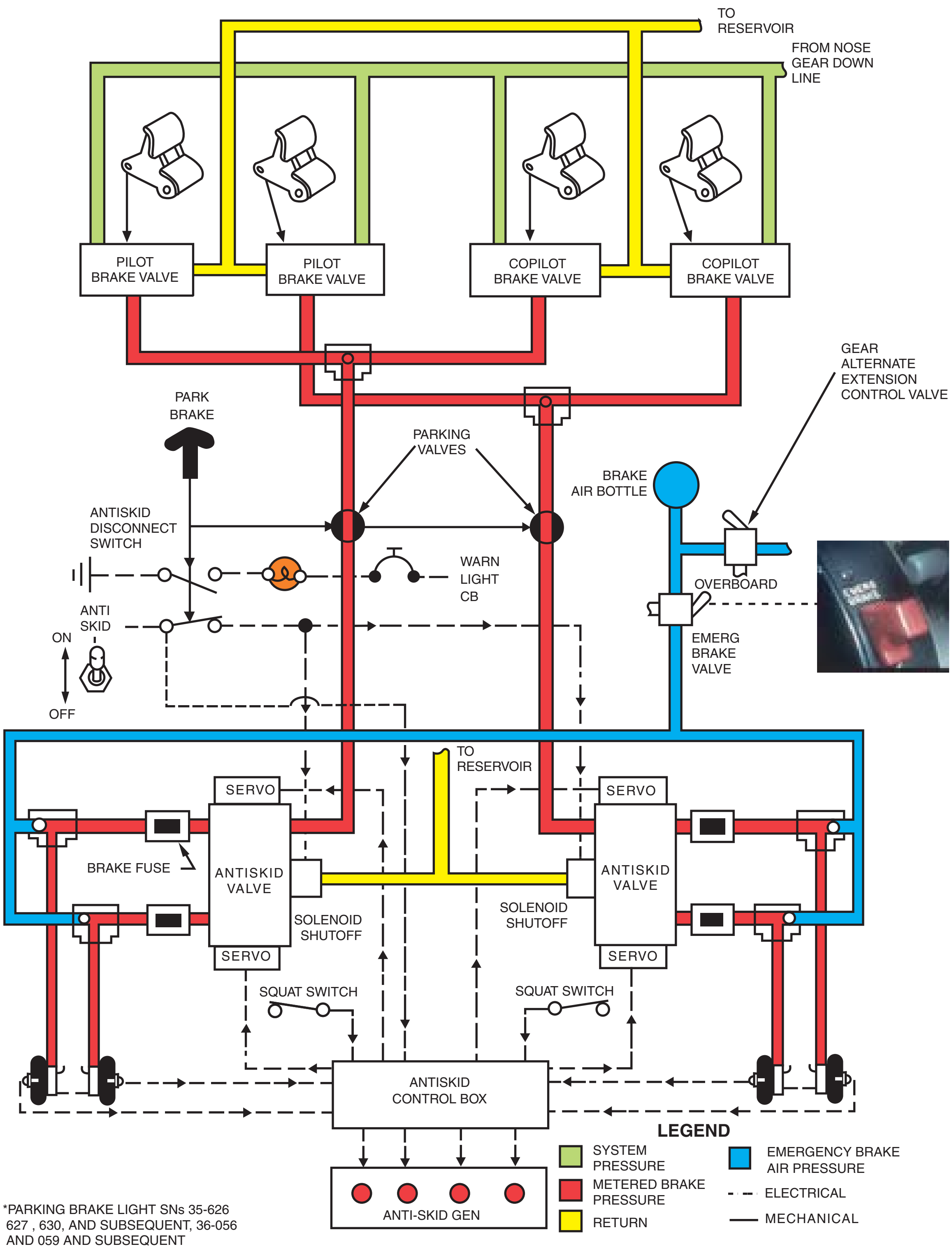


Figure 14-11. Brake System Schematic

During gear retraction, a restrictor in the nose gear return line creates back pressure on the brakes which is sufficient to stop the wheels from rotating prior to their entering the wheel well.

A priority valve, also in the nose gear downline, ensures proper gear sequencing during retraction by restricting hydraulic pressure applied to the nose gear actuator while full system pressure is being applied to the main gear actuators.

When taxiing through slush or snow, frequent brake application creates friction heat which may prevent the brakes from freezing.

If a takeoff is made in slush or snow, the wheels should be allowed to spin down for approximately 1 minute prior to gear retraction. This minimizes the possibility of brake freezing by slinging off accumulated slush. If frozen brakes are suspected after the gear is extended for landing, the ANTISKID switch should be positioned to OFF, and the brakes applied 6 to 10 times to break up any possible ice formations. The ANTISKID switch should be turned back to ON prior to landing.

ANTISKID OPERATION

General

One of two antiskid systems may be installed. The early system was standard on airplane SNs 35-002 through 35-066 and 36-002 through 36-017. The later system is standard on airplane SNs 35-067 and subsequent and 36-018 and subsequent, and may be retrofitted to early airplanes by AAK 76-4. The two systems are similar and are discussed together with the differences being noted.

The antiskid system limits braking on each main gear wheel independently to allow maximum braking under all runway conditions without tire skidding.

The system consists of four wheel speed transducers (one on each main wheel), two antiskid control valves, a control box, monitor lights, and a lever-locking ANTISKID switch on the center instrument panel.

Airplanes with the early antiskid system have test provisions on the system's rotary test switch. On these airplanes, the system is tested during the Before Taxi check in accordance with the approved *AFM*. The ANTISKID switch should be positioned to OFF after testing unless the airplane incorporates AAK 75-1 or AMK 76-3, in which case it can be left on. On airplanes with the later system, no testing is required and the switch is normally left in the ON position.

The antiskid system is not required to be operational for flight. However, if a malfunction is indicated by illumination of a red ANTI-SKID GEN light(s), it must be assumed that antiskid protection is lost on the associated wheel. Takeoff and landing data must be computed accordingly.

The system uses DC power from the ANTISKID circuit breaker on the right main DC bus.

Operation

The following conditions must exist for operation of the antiskid system:

- The ANTISKID switch must be on.
- Both squat switches must be in the ground mode (left for outboard, right for inboard).
- The parking brake must be released.
- Taxi speed must be above 8 to 10 knots (wheel speed, 150 rpm).

At high speed, with the ANTISKID switch on and brakes applied, the control box receives and analyzes wheel speed inputs from the transducer on each main wheel (Figure 14-11). If any wheel deceleration rate reaches a predetermined limit, the applicable servo valve will modulate braking force on the corresponding brake by diverting pressure into the return line.

With the gear extended in flight, the braking system is disabled. When the main gear squat switches go airborne, all braking pressure is diverted into the return line (as though all wheels were in a full-skid condition). This precludes the possibility of touching down on the next landing with brakes applied inadvertently. Further, at the moment of touchdown, the squat switches initiate a requirement for a 150-rpm wheel spinup or a 1- to 2-second delay, thus enabling the control box to sense realistic wheel speeds before normal braking can begin.

If the brakes are to be applied in flight to break up suspected accumulations of ice on the brakes, the ANTISKID switch must first be positioned to OFF. Position the switch to ON prior to landing.

At low taxi speeds (wheel speed below 150 rpm, 8–10 knots), the antiskid system is inoperative. The system is automatically disconnected when the parking brakes are set; however, the red ANTI-SKID GEN lights will not illuminate.

Four red ANTI-SKID GEN lights monitor circuitry from each wheel speed transducer and will individually illuminate if a fault is detected. Cycling the ANTISKID switch to OFF then back to ON may clear the fault. All four lights illuminate if power to the control box is lost or if the ANTISKID switch is off.

EMERGENCY BRAKES

Pneumatic emergency brakes are provided for use in the event of normal brake system failure. Antiskid protection, differential braking, and parking brakes are ***not*** available while using the emergency brakes.

To apply brakes with the emergency system, the EMER BRAKE handle must be pulled out of its recess (Figure 14-11) and pressed downward. This meters pressure from the emergency air bottle through four shuttle valves to the brake assemblies in proportion to handle movement. Releasing the handle stops flow from the bottle and allows applied air pressure to be vented overboard, releasing the brakes.

PARKING BRAKES

Normal hydraulic system pressure from either engine-driven pump or the auxiliary pump can be used to set the parking brakes. Pulling the PARKING BRAKE handle on the center pedestal mechanically closes both parking brake valves (Figure 14-11). The closed valves function as one-way check valves, allowing pressure from the pilot or copilot brake valves to be trapped in the brake assemblies.

To set the parking brakes, pedal pressure must be applied and the PARKING BRAKE handle pulled out (but not necessarily in that order). Setting the parking brake opens the antiskid disconnect switch (Figure 14-11) to disconnect the antiskid system and prevent inadvertent loss of brake pressure.

To release the parking brakes, the PARKING BRAKE handle must be pushed in all the way to the stop. If the PARKING BRAKE handle is not pushed in to the stop, the parking brakes may be released but the antiskid disconnect switch may not actuate to enable the antiskid system. The ANTI-SKID GEN lights will not illuminate, and subsequent heavy braking will result in wheel skids.

Airplanes SNs 35-626, 627, 630 and subsequent, 36-056, and 059 and subsequent have an additional PARK BRAKE light just above the ANTI-SKID GEN lights. The PARK BRAKE light will illuminate if the parking brake handle is not in the completely forward (released) position.

NOSEWHEEL STEERING

GENERAL

The electrical actuated nosewheel steering system has variable authority, as determined by signals from the left inboard and both right wheel

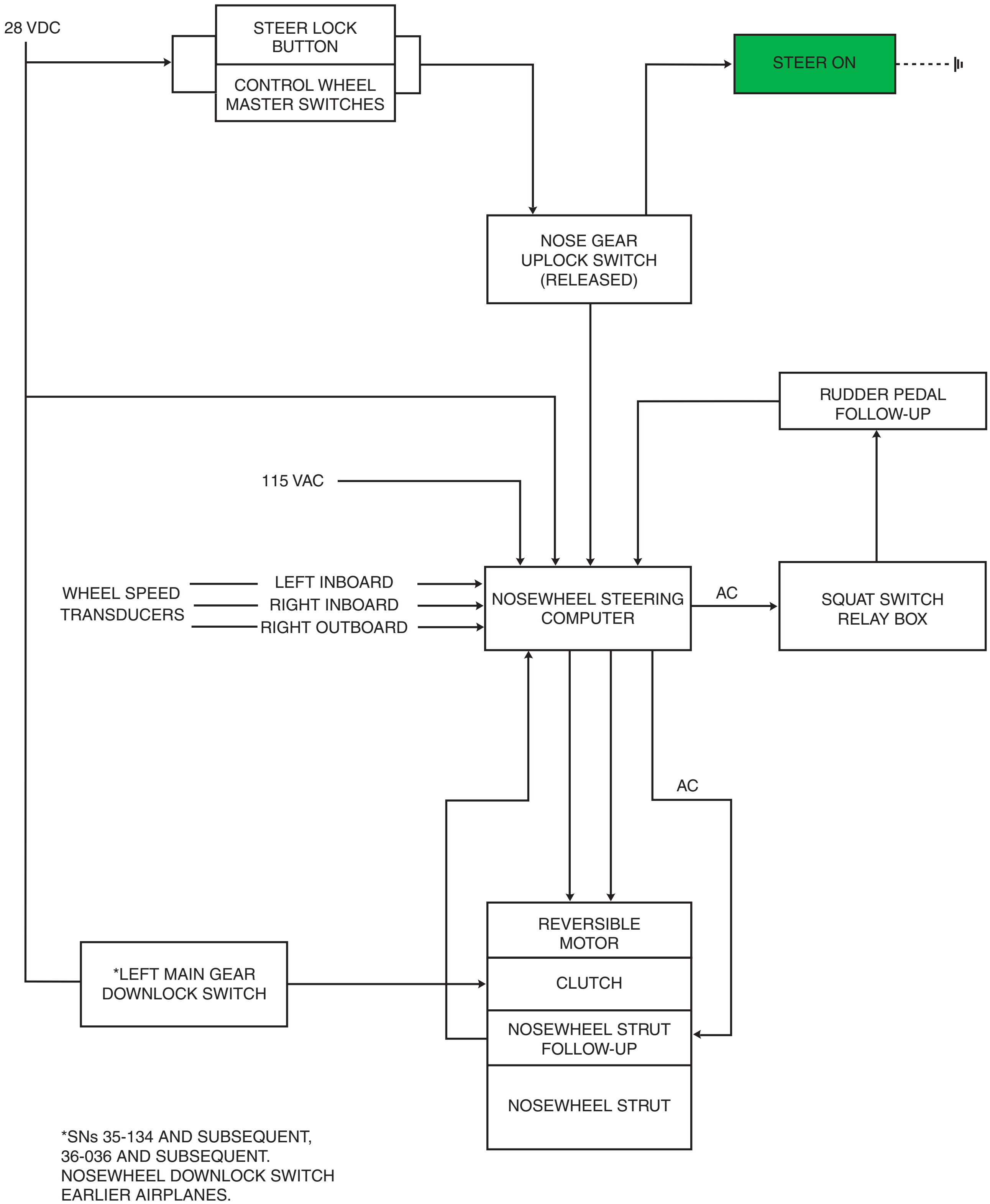


Figure 14-12. Nosewheel Steering System

speed transducers. System components also include a rudder pedal follow-up, a computer-amplifier, and a DC steering actuator motor (Figure 14-12). AC and DC power is supplied through the NOSE STEER circuit breakers on the left AC and left main buses, respectively.

The steering actuator, mounted on top of the nose strut, steers the nosewheel through a gearbox and an electrical clutch. When the airplane is on the ground, the clutch engages whenever DC power is applied to the electrical system allowing the steering actuator to function as a shimmy damper, even with steering disengaged. If DC power is lost or the DC NOSE STEER circuit breaker is out, the nosewheel is free to swivel, and the shimmy damper is inoperative.

Prior to towing, electrical power should be removed from the airplane. It is possible to misalign the nosewheel more that 90º from normal during towing; therefore, the nose gear uplock roller on the nose gear strut must be pointing forward prior to flight.

Steering authority varies from a maximum of 45º either side of center at speeds below 10 knots and decreases as ground speed increases. At the maximum steering speed of 45 knots, authority has been reduced to approximately 8º.

OPERATION

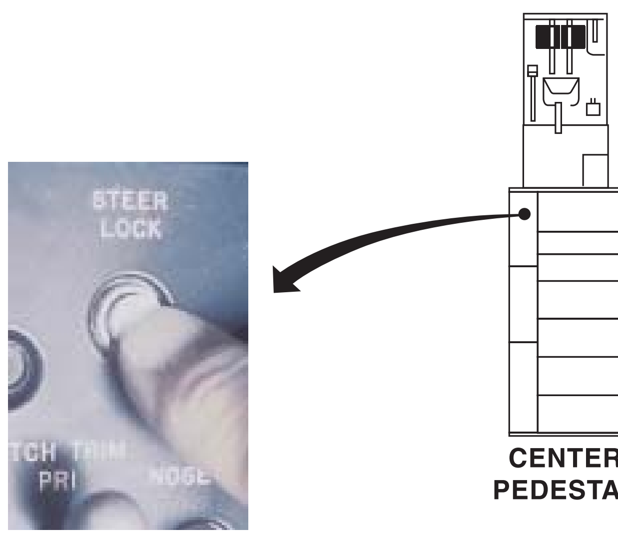
With the squat switches in the ground mode, nosewheel steering can be engaged by momentarily depressing the STEER LOCK switch or by depressing and holding the control wheel master switch (MSW) on either control wheel (Figure 14-13). STEER LOCK is disengaged by momentarily depressing either control wheel master switch.

When steering engages, the green STEER ON annunciator illuminates. A rudder pedal followup provides the displacement and directional signals modified by the computer-amplifier input from the wheel speed transducers. The computer-amplifier drives the steering actuator in the appropriate direction until it is stopped by a signal from a follow-up located in the drive gearbox.

If the nosewheel steering system is inoperative, differential power and braking can be used to taxi the airplane.

Since variable-authority steering is dependent upon wheel speed transducer signals, steering should not be used above 10 knots if any two of the following three ANTI-SKID GEN lights are illuminated: two inboard and right outboard.

CONTROL WHEELS



CENTER PEDESTAL

Figure 14-13. Nosewheel Steering System Controls

QUESTIONS

1 Emergency air pressure can be used for:
   A. Gear extension and parking brake
   B. Gear, flaps, spoilers, and brakes
   C. Gear extension and brakes
   D. Gear extension, flaps, and brakes

2. Prior to takeoff, the EMERGENCY AIR pressure indicators should indicate:
   A. 1,800 to 3,000 psi
   B. Minimum 1,700 psi
   C. 3,000 to 3,350 psi
   D. Maximum 1,750 psi

3. During normal gear operation, main gear inboard doors and the main gear are sequenced by:
   A. Microswitches
   B. Emergency air pressure
   C. Mechanical linkage
   D. Both A and B

4. Automatic brake snubbing is provided during gear retraction by restricting return fluid from the:
   A. Antiskid system
   B. Engine-driven pumps
   C. Squat switches
   D. Landing gear system

5. After an emergency gear extension, the gear position light indication should be:
   A. Three green
   B. Three green, two red
   C. Three red, two green
   D. Three red, three green

6. Three gear UNSAFE lights will be on and the gear warning horn sounding when the:
   A. Gear is retracted and no green LOCKED DN lights are on.
   B. Gear is down, thrust levers are above approximately 70% N_1, and altitude is below 14,500 ±500 feet.
   C. Gear is up, thrust levers are below approximately 55%–60% N_1, altitude is below 14,500 ±500 feet and, on FC 530 airplanes, airspeed is below 170 KIAS.
   D. Flaps are extended below 25º, regardless of altitude.

7. With the flaps extended beyond 25º and the gear not down and locked, the warning horn:
   A. Will sound but can be muted
   B. Will not sound
   C. Will sound but cannot be muted
   D. None of the above

8. Illumination of a red main gear UNSAFE light indicates:
   A. The corresponding main gear is not down and locked.
   B. The corresponding main gear is not up and locked.
   C. The corresponding main gear inboard door is not fully closed.
   D. The corresponding main gear inboard door is locked in the closed position.

9. The red nose gear UNSAFE light will be on when:
   A. The nose gear is unsafe or in transit.
   B. Nosewheel steering is inoperative.
   C. The nose gear doors are open.
   D. The nose gear doors are closed.

10. Parking brakes can be set with the:
- A. Pilot's brake pedals only
- B. Copilot's brake pedals only when the ANTISKID switch is on
- C. Pilot's or copilot's brake pedals
- D. Pilot's or copilot's brake pedals only with the ANTISKID switch off

11. If the first three ANTI-SKID GEN lights are illuminated:
- A. Takeoff weight is limited to 17,000 pounds.
- B. Nosewheel steering should not be engaged above 10 kts.
- C. Takeoff (V_R) will be affected.
- D. Both A and B are correct.

12. Normal brake pressure is provided by:
- A. Main hydraulic system pressure from the nose gear down line
- B. Brake accumulator
- C. Emergency air bottle through the antiskid control valves
- D. Emergency air bottle

13. Related to nosewheel steering, the precautions that should be taken prior to towing the airplane are:
- A. Keep rudder pedals centered.
- B. Do not exceed the 55º turning limits.
- C. Pull the NOSE STEER DC circuit breaker if the battery switches are on.
- D. Turn off the ANTISKID switch.

14. If the green main gear LOCKED DN light is burned out, positive down-and-locked condition can be confirmed by:
- A. GND IDLE light illuminated
- B. ENG SYNC light illuminated
- C. Illumination of the corresponding landing light when the switch is turned on
- D. Red UNSAFE lights illuminate.

15. The electrical requirements for nosewheel steering are:
- A. 24 VAC and 28 VDC
- B. Only 28 VDC
- C. Only 115 VAC
- D. 28 VDC and 115 VAC

16. When STEER LOCK is engaged:
- A. Nosewheel steering is engaged and full steering is available up to 45 kts.
- B. The nosewheel is locked in whatever position it is in at the time.
- C. Up to 45º left or right steering is available, with decreasing authority at higher speeds.
- D. Nosewheel becomes free swiveling.

17. STEER LOCK is disengaged by:
- A. Depressing the OFF button
- B. Depressing the STEER LOCK button a second time
- C. Momentarily depressing either wheel master switch
- D. Depressing the ANTISKID release button

CHAPTER 15
FLIGHT CONTROLS

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CHAPTER 15
FLIGHT CONTROLS

INTRODUCTION

The manually operated primary flight controls incorporate electrical trim in all three axes. Secondary flight controls consist of hydraulically actuated spoilers/spoilerons and flaps. Other systems related to flight controls are the yaw damper, stall warning, Mach overspeed warning, and Mach trim systems.

GENERAL

The primary flight controls (ailerons, elevator, and rudder) are mechanically operated through the dual control columns, control wheels, and rudder pedals. They are incorporated into both the FC 200 and the FC 530 automatic flight control system (AFCS). Both systems also incorporate a rudder/aileron interconnect.

The ailerons incorporate mechanical balance tabs to provide aerodynamic assistance. Trim systems (roll, yaw, and pitch) are electrically operated and controlled. Trim tabs are installed on the left aileron and the rudder. The movable horizontal stabilizer provides pitch trim.

The flaps and spoilers are hydraulically actuated and electrically controlled.

Aileron augmentation is provided by a spoileron system which increases roll authority when the airplane is configured for landing.

A dual yaw damper system provides yaw stability.

A dual stall warning system provides an indication of impending stall by vibrating the control column and, if no corrective action is taken, induces a forward control column movement to reduce the airplane angle of attack.

A Mach overspeed warning system warns of overspeed and induces an aft control column movement to raise the nose of the airplane.

A Mach trim system provides automatic pitch trim to compensate for Mach tuck.

All flight control surfaces are shown in Figure 15-1.

A flight controls gust lock is provided to prevent wind gust damage to the primary flight control surfaces. When installed, the lock holds full left rudder, full left aileron, and full down elevator displacement (Figure 15-2).

PRIMARY FLIGHT CONTROLS

ELEVATORS

The elevators are hinged to the aft edge of the horizontal stabilizer and are positioned by fore-and-aft movement of the control column. Three scuppers are located near the aft edge of each elevator for moisture drainage, and three static dischargers are attached to the trailing edge of each elevator.

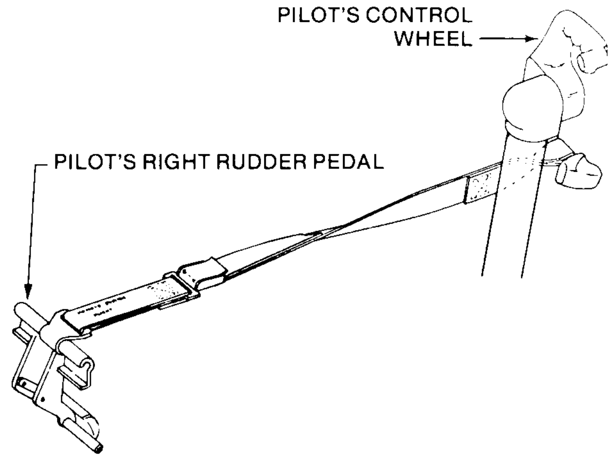

Figure 15-2. Flight Controls Gust Lock

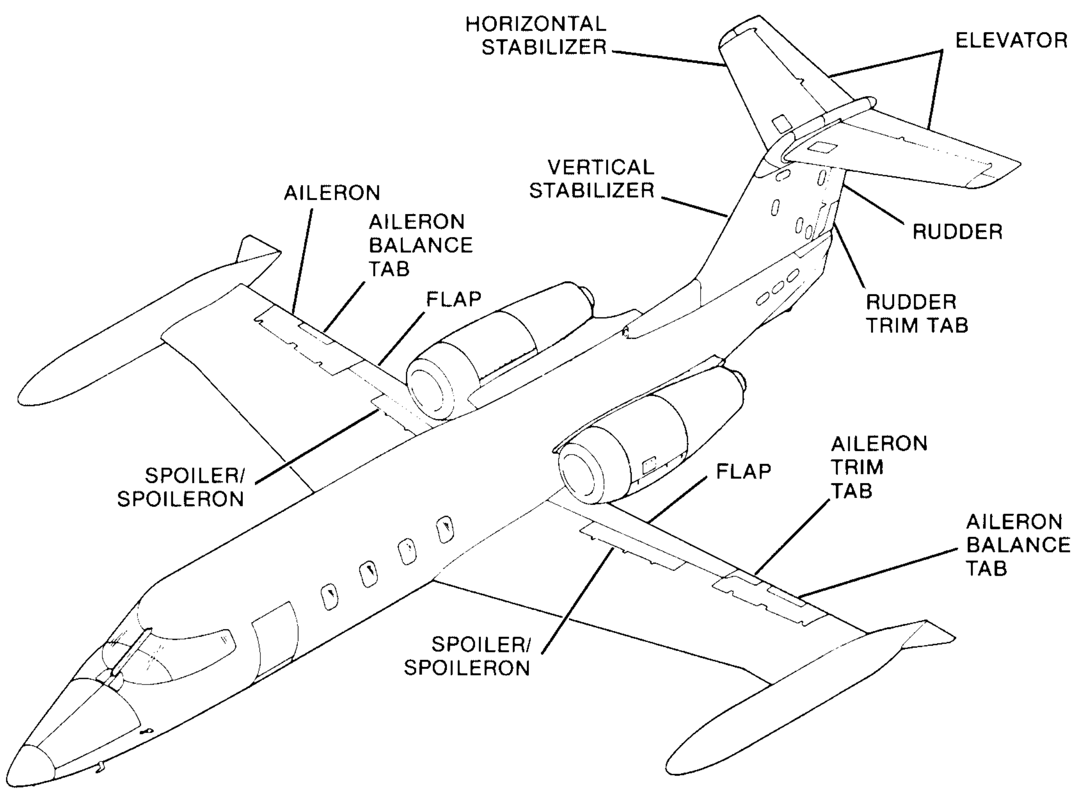

Figure 15-1. Flight Control Surfaces

The elevators can also be positioned by an electrically actuated pitch servo.

A bob weight attached to the control column and a downspring assembly in the elevator control linkage are incorporated to enhance pitch stability.

Pitch Servo

The pitch servo (torquer) is DC operated. It is mechanically connected to the elevator control linkage through a capstan mechanism incorporating an electric clutch and a mechanical slip clutch. Three flight control systems use the pitch servo to operate the elevators:

- Autopilot—When engaged, the autopilot can alter noseup or nosedown attitude by commanding the servo to torque the elevator up or down, as required.
- Both stall warning systems—Either system will cause the servo to torque the elevator nose ***down*** in the event of an impending stall (stick pusher). On FC 530 models, pulsating nose down torque signals are used for the "nudger."
- Mach overspeed warning system—Operating through the L STALL WARNING switch, the system will command the servo to torque the elevator nose ***up*** (stick puller) due to an overspeed.

On FC 200 AFCS airplanes, the electric clutch must be engaged to couple the servo to the elevator linkage. The clutch engages when any one of the following switches is in the ON position:

- L STALL WARNING
- R STALL WARNING
- AUTOPILOT master

With all three of the above switches in the OFF position, the electric clutch is disengaged, disconnecting the servo from the elevators. This enables the pilot to gain manual control of the elevator by eliminating the servo in the event of a malfunction.

By exerting sufficient force on the control column to slip the mechanical clutch, the pilot can also override any undesirable servo inputs to the elevators, if necessary.

On FC 530 AFCS airplanes, the electric clutch remains deenergized until the servo is signalled by either the autopilot, L or R stall warning system, or the overspeed puller system.

On these airplanes, the servo can be eliminated as a cause of malfunction by simply depressing and holding the wheel master switch. The pilot can also, by exerting the required force on the control column to slip the mechanical clutch, override any undesirable servo operation.

Autopilot operation is described in Chapter 16, "Avionics."

AILERONS

The ailerons, mechanically positioned with either control wheel, provide primary roll control. Aileron effectiveness is augmented by spoilerons when the airplane is configured for landing.

Spoileron (aileron augmentation) operation is automatically activated when the flaps are lowered beyond 25°. In the spoileron mode, when an aileron is moved up to initiate airplane roll, the spoiler on the same wing automatically rises the same number of degrees to provide additional roll.

Roll Servo (Autopilot Function Only)

The ailerons can also be positioned by the autopilot roll servo. The roll servo is similar to the pitch servo but does not incorporate an electric clutch. A mechanical slip clutch allows the pilot to override undesired roll servo inputs; the servo can also be disconnected by disengaging the autopilot.

Balance Tab

The balance tab on each aileron (Figure 15-3) provides aerodynamic assistance in moving the aileron, thus reducing control wheel forces.

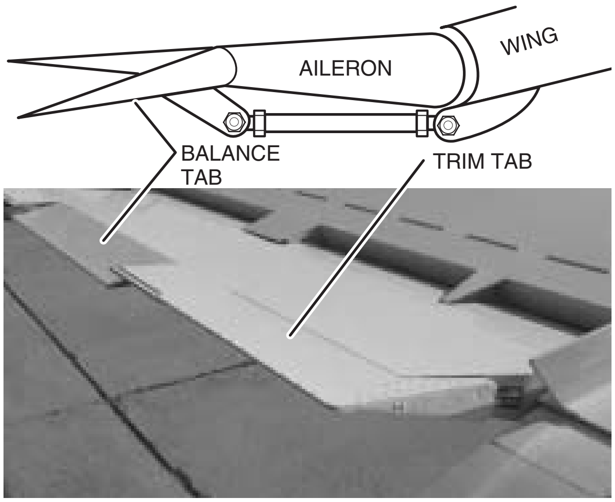

Figure 15-3. Aileron Tabs

Trim Tab

The electrically operated aileron trim tab is attached to the inboard trailing edge of the left aileron (Figure 15-3). The tab is positioned by either the pilot's or copilot's control wheel trim switch. Aileron trim tab position is indicated on the cockpit center pedestal.

Aileron Follow-ups

Aileron follow-up mechanisms, driven by the aileron control linkage, provide aileron displacement information to the spoileron computer, yaw damper, and the autopilot.

RUDDER

The rudder can be manually positioned with either set of rudder pedals, or by either of two yaw damper servos (primary and secondary). The crew can manually override the yaw damper through a mechanical slip clutch in the event of a malfunction. The yaw damper can be disengaged by depressing either wheel master switch or the corresponding yaw damper OFF button.

Rudder Trim Tab

A trim tab, mounted on the bottom trailing edge of the rudder, is controlled by a trim switch on the center pedestal. Trim position is also indicated on the center pedestal.

TRIM SYSTEMS

GENERAL

The ailerons and rudder are trimmed with conventional tabs on the control surfaces as previously described.

The airplane is trimmed in the pitch axis by changing the angle of incidence of the movable horizontal stabilizer. A dual-motor (primary and secondary) actuator moves the leading edge of the horizontal stabilizer up or down in response to pitch trim inputs. Controls and indicators for the trim systems are shown in Figure 15-4.

The trim position indicators for pitch, roll, and yaw are all DC powered through the TAB & FLAP POSN circuit breaker on the right essential bus.

RUDDER (YAW) TRIM

Control

Rudder (yaw) trim is controlled by the rudder trim switch on the center pedestal, (Figure 15-4) spring-loaded to the OFF position.

The switch knob is split into an upper and a lower half. Both halves must be rotated simultaneously to initiate rudder trim tab motion. This is a safety feature to reduce the possibility of inadvertent trim actuation. The rudder trim system is DC powered through the YAW circuit breaker on the left essential bus.

Rudder Trim Indicator

Rudder trim tab position indication is provided by the RUDDER TRIM indicator (Figure 15-4).

AILERON TRIM

Control

Aileron (roll) trim is controlled with either control wheel trim switch located on the outboard horn of each control wheel (Figure 15-4). Each control wheel trim switch is a dual-function (trim and trim arming) switch which controls roll and primary pitch trim. Each switch has four positions—LWD, RWD, NOSE UP, and NOSE DN, and is spring-loaded to the neutral position. The arming button on top of the switch must be depressed and held while simultaneously moving the trim switch in the direction of desired trim action. Actuation of either control wheel trim switch to

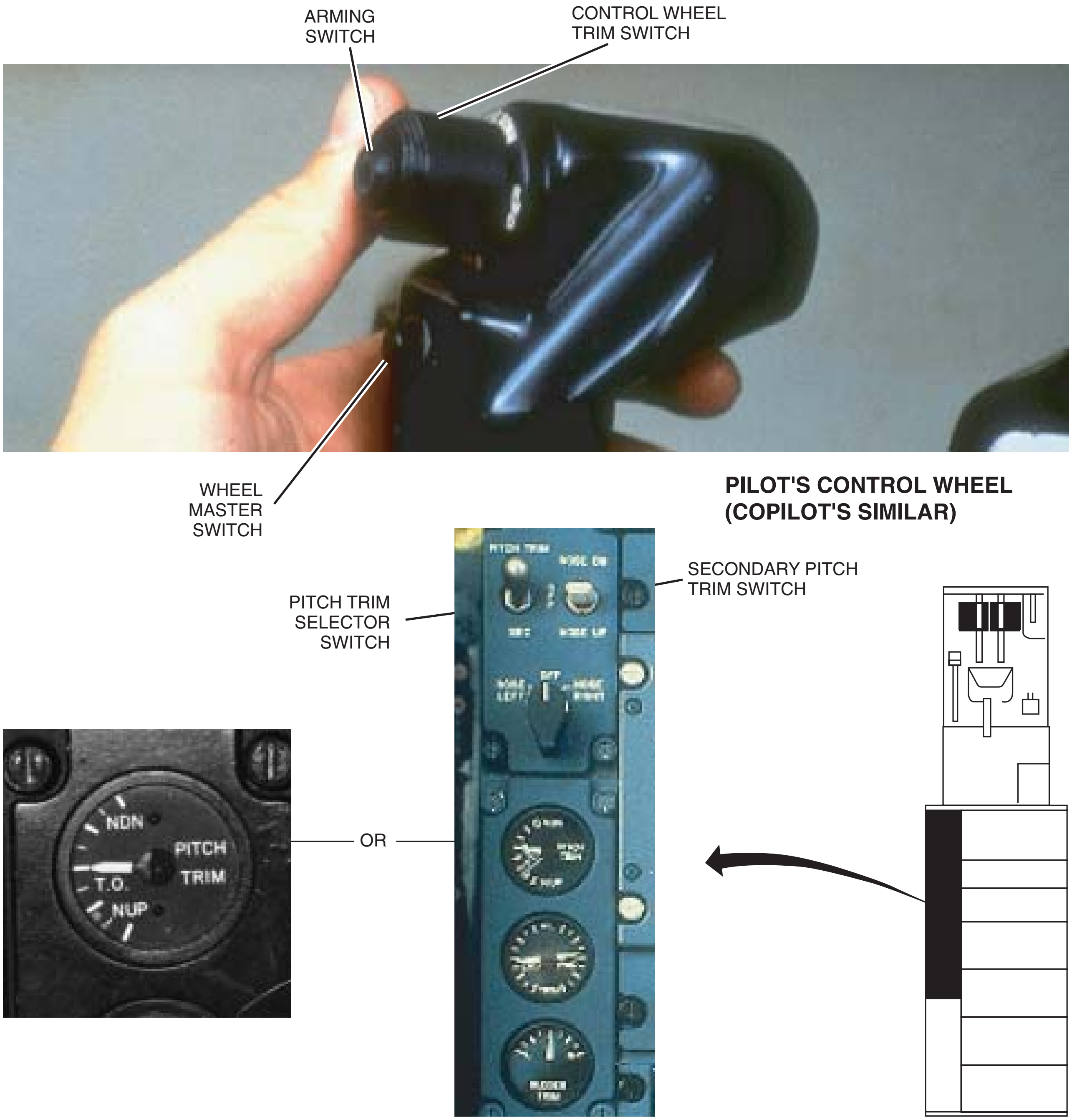

Figure 15-4. Trim Systems Controls and Indicators

LWD or RWD (with arming button depressed) will signal the trim tab actuator motor in the left aileron to move the trim tab in the appropriate direction. Actuation of the pilot's trim switch will override actuation of the copilot's switch.

The aileron trim motor is DC powered through the ROLL circuit breaker on the left essential bus.

Aileron Trim Indicator

Aileron trim tab position indication is provided by the AIL TRIM indicator (Figure 15-4).

PITCH TRIM

General

Pitch trim is accomplished by repositioning the horizontal stabilizer to the desired trim setting with a dual-motor (primary and secondary) actuator that operates in four modes:

1. Primary pitch trim mode 2. Mach trim mode	Primary trim motor
3. Secondary pitch trim mode 4. Autopilot pitch trim mode	Secondary trim motor

The pilot-operated primary pitch trim and secondary pitch trim systems are electrically independent systems. Mode selection (primary or secondary) is made with the PITCH TRIM selector switch (Figure 15-4).

Primary pitch trim is pilot-controlled through either of the control wheel trim switches; secondary pitch trim is controlled through the secondary pitch trim toggle switch on the center pedestal (Figure 15-4).

Airplanes with the FC 530 automatic flight control system (AFCS) incorporate a ***two-speed*** primary trim motor, a trim monitor system, and an audible clicker that signals trim in motion.

Mach trim automatically engages at approximately 0.69 M_1 if the autopilot is not engaged, and uses the primary trim motor to adjust pitch trim. Autopilot operation uses the secondary motor to adjust pitch trim.

NOTE

The PITCH TRIM selector switch must be in the PRI position to enable the Mach trim system. It may be in either the PRI or SEC position during autopilot operation.

Horizontal stabilizer position is displayed on the PITCH TRIM indicator (Figure 15-4).

Pitch Trim Actuator

The pitch trim actuator is operated by either of two DC-powered motors, either of which can move the horizontal stabilizer. On FC-200 AFCS airplanes, the primary trim motor and control circuits are powered through the PITCH circuit breaker on the left essential bus. On FC-530 AFCS airplanes, the motor is powered by the battery charging bus, and the PITCH circuit breaker on the left essential bus controls a relay in the power circuit. The secondary trim motor and control circuits are powered through the SEC PITCH TRIM (or SEC P TRIM) circuit breakers on the right essential bus.

On FC 200 AFCS airplanes, the secondary trim motor operates at approximately one-half the speed of the primary trim motor.

On airplanes with the FC 530 AFCS, the two-speed primary trim motor operates at a considerably slower rate (approximately one-fourth speed) with the flaps up. A 3° flap switch is used for speed switching. On these airplanes, operating speed of the secondary trim is approximately the same as the speed of the primary trim with flaps up.

PITCH TRIM Selector Switch

The PITCH TRIM selector switch provides the primary and secondary mode selections (Figure 15-4). In the PRI (forward) position, primary pitch trim is available from both of the control wheel trim switches and from the Mach

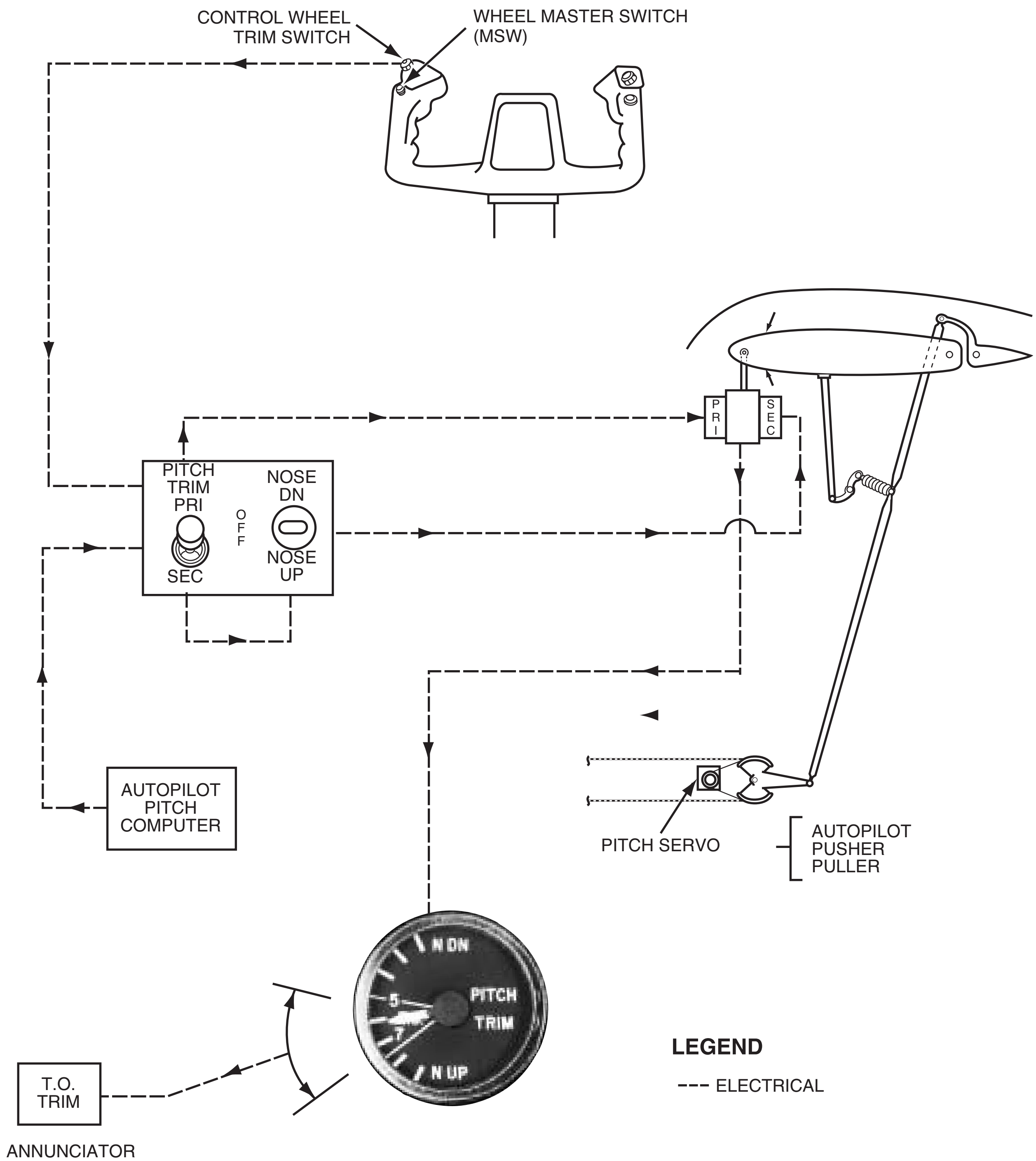

Figure 15-5. Pitch Trim System Schematic (Airplanes with FC 200 AFCS)

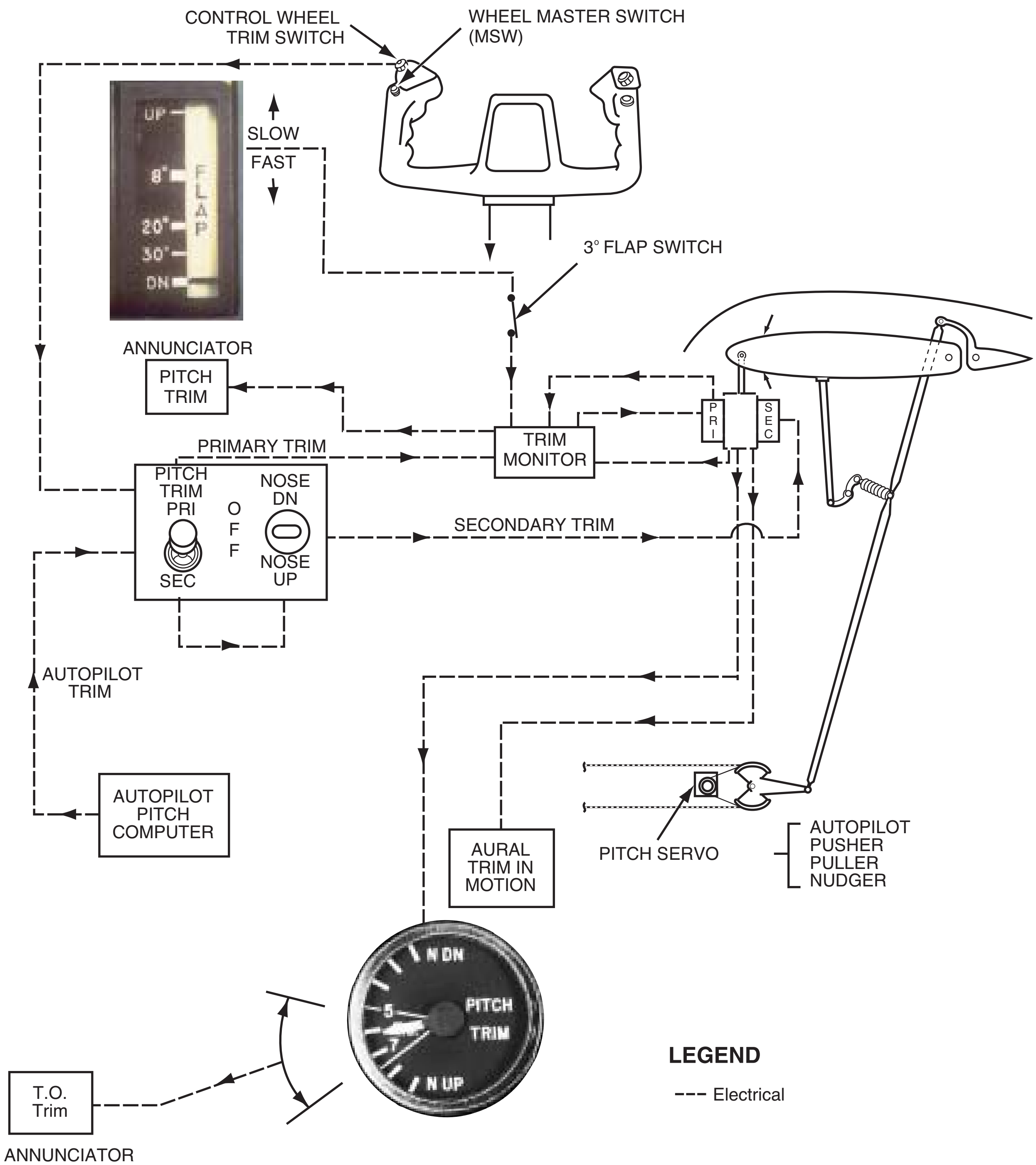

Figure 15-6. Pitch Trim System Schematic (Airplanes with FC 530 AFCS)

trim system. In the OFF position, both trim motors and control circuits are deenergized. In the SEC (aft) position, secondary pitch trim is available from the secondary trim switch (Figure 15-4); this renders the pilot's primary trim and Mach trim inoperative. The secondary pitch trim switch is spring-loaded to the OFF position.

The autopilot always uses the secondary trim motor whether the PITCH TRIM selector switch is in the PRI or SEC position; however, if either control wheel trim switch is actuated with the arming button depressed (Figures 15-5 and 15-6) or if the secondary trim switch is actuated, the autopilot will disengage.

In the event of inadvertent primary pitch trim operation (runaway trim), depressing and holding the wheel master switch will:

- Stop ***only*** the primary pitch trim motor (airplanes with FC 200 AFCS)

- Stop ***both*** the primary ***and*** the secondary trim motors (airplanes with FC 530 AFCS)

The control wheel trim switches (Figure 15-4) were described in this chapter under Aileron Trim.

Pitch Trim Indicator

Horizontal stabilizer trim position indication is provided by one of two types of PITCH TRIM indicators (Figure 15-4). On each indicator, a T.O. (takeoff) trim segment is marked to indicate the takeoff trim limits for center-of-gravity extremes. On early airplanes, the segment is marked by a green band on the edge of the indicator; on later airplanes, by white lines. Late model indicators may be retrofitted on early airplanes. In either case, whenever the pitch trim is not set within the T.O. trim segment, the amber T O TRIM annunciator light will illuminate (on the ground only). (All annunciator lights are shown in Annunciator Panel section.)

Pitch Trim Monitor System (Airplanes with FC 530 AFCS)

General

A monitor system incorporated in these airplanes provides a visual indication of certain faults in the ***primary*** trim system.

Though not physically a part of the monitor system, a clicker provides audible evidence of trim in motion (primary or secondary trim system) when the flaps are up.

Operation

The monitor system monitors the primary trim system, the 3° flap switch, and the horizontal stabilizer actuator mechanism. Faults are indicated by illumination of the amber PITCH TRIM light.

With flaps up (slow trim required), the monitor system will illuminate the PITCH TRIM light if it senses that primary trim is running at the ***fast*** rate (trim overspeed).

Regardless of flap position, the monitor system will also illuminate the PITCH TRIM light if it senses certain electrical faults in the primary system that create the ***potential*** for uncommanded motion of the stabilizer actuator.

When the PITCH TRIM light illuminates, the secondary trim system must be selected by placing the PITCH TRIM selector switch in the SEC position (unless it illuminates while holding the wheel master switch depressed, which is normal).

The audio clicker will sound anytime the stabilizer actuator is in motion with flaps up, whether trimming is being accomplished with the primary or secondary motor. However, to preclude the clicker from sounding every time trim is commanded, a delay of approximately $^{1}/_{4}$ second must follow each in-motion signal, thereby eliminating nuisance signals when the pilot uses short trim inputs.

The monitor system and trim-in-motion clicker are tested in accordance with procedures outlined in Section 2 of the approved *AFM*. Either a three-position switch decaled "TRIM OVSP–OFF–TRIM MON" (spring-loaded to OFF) or the TRIM OVSP and TRIM MON positions of the rotary systems test switch are used to perform the test.

MACH TRIM

General

The Mach trim system is an automatic pitch trim system that uses the primary trim motor to enhance longitudinal stability during accelerations/decelerations at high Mach numbers to compensate for Mach tuck. There is no switch to engage the system; it automatically becomes active at approximately 0.69 M_I if the autopilot is not engaged.

Since the Mach trim system requires the use of the primary pitch trim motor, the PITCH TRIM selector switch must be in the PRI position for system operation.

If the autopilot is engaged, the Mach trim system assumes a passive (standby) mode. In this case, the PITCH TRIM selector switch can be in either the PRI or SEC position, since the autopilot can utilize the secondary trim motor in both switch positions.

The Mach trim system consists of a computer, an air data sensor, a follow-up on the horizontal stabilizer, and a red MACH TRIM annunciator light, Mach overspeed warning horn and a monitor circuit. The system is powered by 115 VAC supplied by the MACH TRIM circuit breaker on the left AC bus, and DC power supplied by the PITCH circuit breaker on the left essential bus.

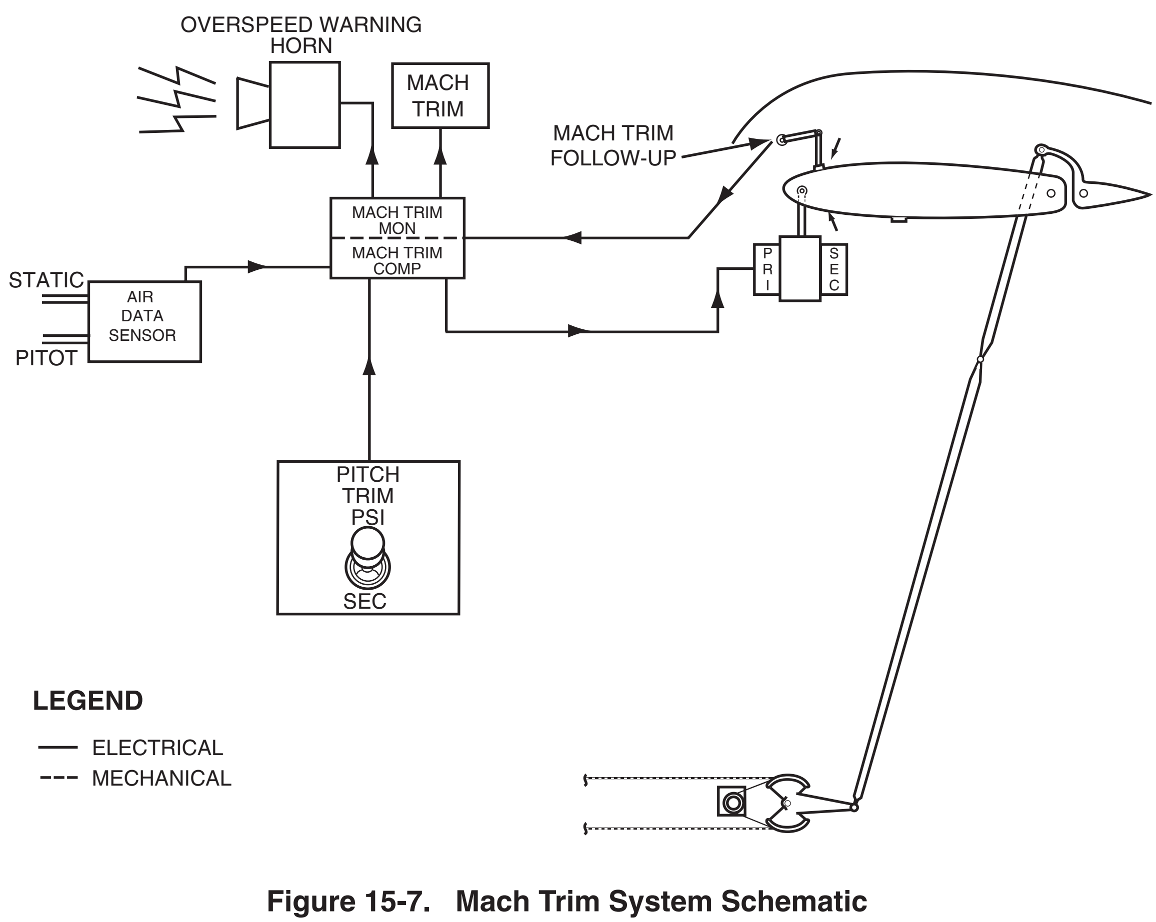

Figure 15-7. Mach Trim System Schematic

Operation

During flight, the air data sensor receives static pressure inputs from the left and right shoulder static pressure ports (FC 200 AFCS) and a pitot pressure input from the right pitot tube (Figure 15-7). On FC 530 AFCS airplanes, static pressure is provided by the RH static 1 and LH static 2 lines. This will be shown in Chapter 16, “Avionics.”

The air data sensor electrically transmits this information to the Mach trim computer. With the autopilot disengaged, the Mach trim system becomes active at approximately 0.69 M_I. The Mach trim computer will command the appropriate pitch trim changes (noseup trim for increasing Mach, nosedown for decreasing Mach) through the primary motor of the pitch trim actuator. The follow-up on the horizontal stabilizer provides the nulling signal to the computer.

Mach trim is interrupted whenever the airplane is manually trimmed. The system resynchronizes to function about the new horizontal stabilizer position when manual trim is released. In flight, synchronization may also be accomplished by selecting the MACH TRIM position on the SYS TEST switch and depressing the TEST button (applies to airplane SNs 35-247 and subsequent, 36-045 and subsequent, and earlier airplanes incorporating SB 35/36 22-4).

Mach Trim Monitor

The Mach trim monitor circuit continuously monitors input signals and power to the Mach trim computer, and compares signal inputs from the air data sensor (Mach) and the Mach trim follow-up on the horizontal stabilizer. A malfunction exists if the Mach trim monitor does not receive a corresponding signal change from the Mach trim follow-up when the air data sensor signals change (Mach change). A malfunction is also indicated in the event of power loss to the Mach trim computer, loss of input signals, or a Mach number/horizontal stabilizer trim position error. In either case, the Mach trim monitor will disengage Mach trim and illuminate the MACH TRIM light. If speed is above 0.74 M_I, the Mach overspeed warning horn will also sound. If the fault clears or power is restored, the system can be resynchronized by selecting the MACH TRIM position on the SYS TEST switch and depressing the TEST button (applies to airplane SNs 35-247 and subsequent, 36-045 and subsequent, and earlier airplanes incorporating SB 35/36-22-4). If the warning horn continues to sound, airspeed must be reduced below 0.74 M_I or the autopilot (if operational) may be engaged. Engaging the autopilot will cancel all warnings, and the airplane can be accelerated to M_{MO}.

SECONDARY FLIGHT CONTROLS

FLAPS

General

The single-slotted Fowler flaps are electrically controlled and hydraulically actuated. The left and right flaps are interconnected by cable to minimize asymmetrical effects in the event of a malfunction.

Position switches mechanically connected to each flap provide flap position information to the landing gear warning, stall warning, spoiler warning, spoileron, and autopilot systems. On airplanes SNs 35-067 and subsequent, SNs 36-018 and subsequent, and earlier airplanes incorporating AAK 76-4, the flap position switches actuate at 3°, 13°, and 25° of flap extension. On earlier airplanes the switches actuate only at 13° and 25°. On airplanes with the preselect flap system, flap limit switches automatically maintain flap position at the selected setting.

If hydraulic system pressure is lost, the flaps will probably remain in their last position. However, if the flaps are extended and hydraulic pressure is lost due to a leak in the flap downline, airloads on the flaps may cause some flap retraction.

The flaps can also be operated from EMER BAT 1 (ON position) in the event of electrical failure; however, the flap indicator is not powered by the emergency battery.

Flap Selector Switch

The flap selector switch may be one of three types. On SNs 35-002 through 35-010, the switch has three positions (up, neutral, and down), and is spring-loaded to the neutral position. The selector switch on later airplanes is not spring-loaded to neutral and will remain in the selected position. Airplane SNs 35-417, 419, 477, 479, and 483 and subsequent, and SNs 36-051 and subsequent incorporate the preselect flap system. On these airplanes the flap selector switch has four positions: UP, 8°, 20°, and DN (40°), with detents at the 8°, and 20° positions (Figure 15-8). The flap system is powered by the FLAPS circuit breaker on the right essential bus. Earlier SNs may be retrofitted with the preselect system by AAK 83-7.

Flap Position Indicator

A vertical-scale FLAP position indicator is mounted on the center switch panel (Figure 15-8).

Left flap position is electrically transmitted to the indicator. The indicator is DC powered by the TAB FLAP POSN circuit breaker on the right essential bus. The indicator will indicate DN with loss of electrical power, regardless of actual flap position.

Operation (Preselect Flaps)

When the flap selector switch is placed in the DN position, the down solenoid positions the flap control valve to direct pressure to the extend side of both flap actuators. The down solenoid remains energized, and the control valve maintains down pressure on the flap actuators to hold the flaps full down (40°). A check valve at the control valve inlet prevents flap retraction in the event of upstream hydraulic system failure.

Moving the selector switch to an intermediate (8° or 20°) position energizes the down or up solenoid, as appropriate, which repositions the control valve to extend or retract the flaps. The appropriate flap limit switch deenergizes the affected solenoid and the control valve closes, thereby stopping flap motion (9° and 21° during extension, 19° and 7° during retraction).

When extended, the flaps are protected from excessive airloads (due to excessive airspeed) by a relief valve in the downline. Pressure is relieved into the return line, causing the flaps to creep upward. The limit switches will energize the down solenoid to return the flaps to the selected position when the airspeed is reduced appropriately.

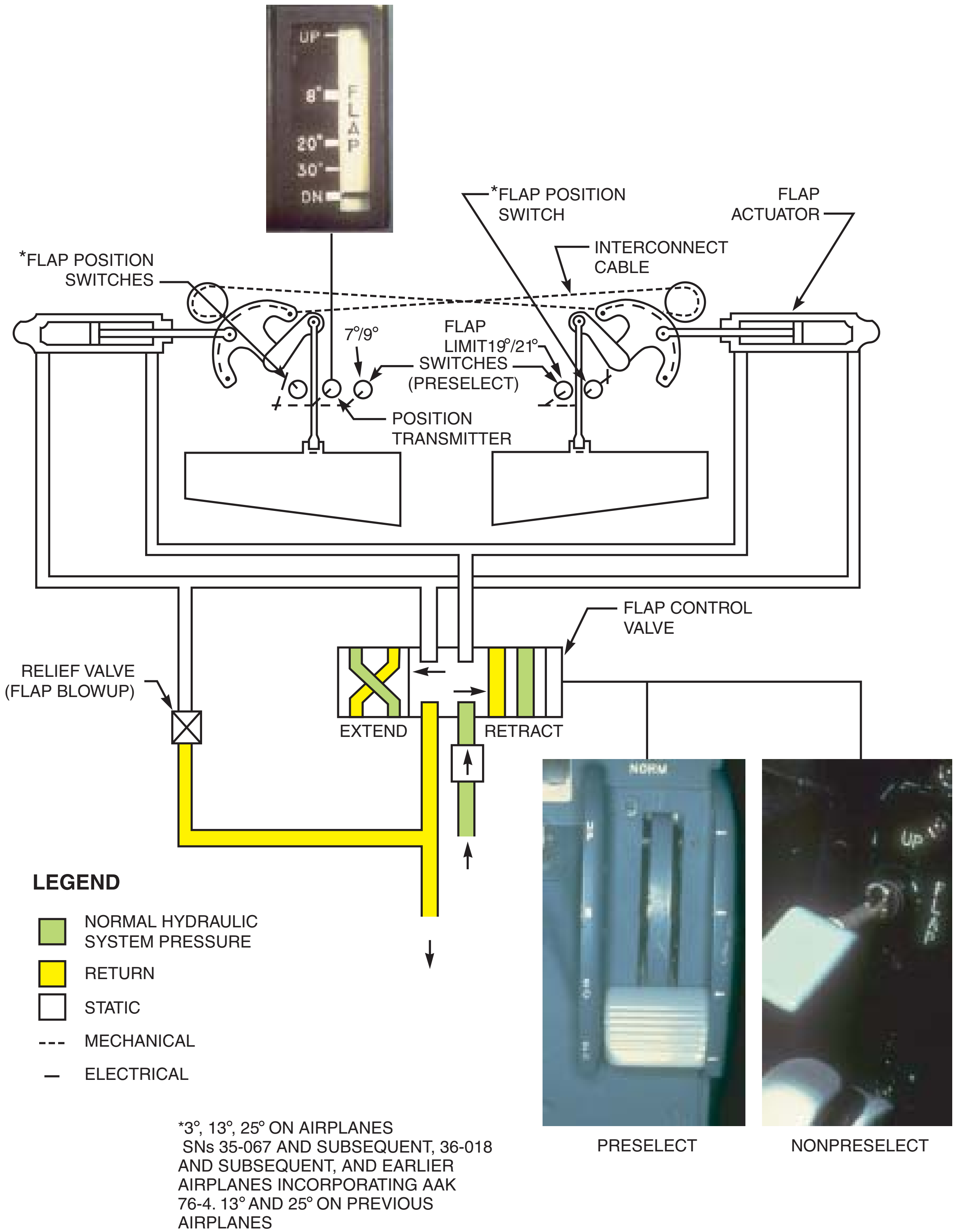

Figure 15-8. Flap System

When the selector switch is moved from the DN position toward the UP position, an intermediate stop is encountered at the 20° position to facilitate retraction in a go-around situation. Further movement of the selector switch toward UP or 8° requires that the switch lever be pulled out to clear the stop.

When the flap selector switch is placed in the UP position, the up solenoid positions the flap control valve to direct pressure to the retract side of both flap actuators. In the fully retracted position, the up solenoid remains energized and the control valve maintains positive pressure on the retract side of both flap actuators.

Operation (Nonpreselect Flaps)

When the flap selector switch is placed in the DN position, the down solenoid positions the flap control valve to direct pressure to the extend side of both flap actuators. The flaps may be stopped in any intermediate position by placing the selector switch in the center neutral position. This deenergizes the down solenoid which repositions the control valve to the neutral position, trapping fluid between the control valve and the actuators to hold the flaps in the selected position.

When extended, the flaps are protected from excessive airloads (due to excessive airspeed) by a relief valve in the downline, and the flaps will creep up until the airspeed is reduced appropriately.

If the flap selector switch is left in the DN position, the down solenoid remains energized, and the control valve maintains extend pressure on the flap actuators. A check valve at the control valve inlet prevents flap retraction in the event of an upstream hydraulic system failure.

Placing the selector switch in the UP position energizes the up solenoid, and the control valve repositions to direct pressure to the retract side of both actuators. In the fully retracted position, the up solenoid remains energized, and the control valve maintains retract pressure on the flap actuators. Returning the selector switch to the neutral position deenergizes the up solenoid and the control valve repositions to neutral.

SPOILERS

The spoilers, located on the upper surface of the wings forward of the flaps, may be extended symmetrically for use as spoilers (spoiler mode) or asymmetrically for aileron augmentation when the flaps are extended beyond 25° (spoileron mode).

The spoilers are hydraulically actuated by a solenoid-operated spoiler selector valve and two servo valves (one for each spoiler). Electrical control of the system is accomplished by the SPOILER switch (for spoiler mode) or by the spoiler computer (spoileron mode).

Both modes require DC and 115-VAC electrical power through the SPOILER and SPOILERON circuit breakers, respectively, on the right essential and AC buses. If either circuit breaker is pulled or either power source is lost in flight, the spoilers will “slam down” (if extended) and will be inoperative in both modes. Spoiler mode operation does not require 115-VAC power on the ground.

A spoiler annunciator light illuminates during normal spoiler deployment or when an uncommanded unlocked condition exists on either spoiler. On FC 200 AFCS models, the light is red. On FC 530 AFCS models, the light is amber.

In the event of main system hydraulic failure, the spoilers, if extended, will blow down and be inoperative. Spoilers cannot be operated with hydraulic pressure from the auxiliary hydraulic pump.

The spoiler mode, when selected, will override the spoileron mode if it is operating.

While airborne, flaps and spoilers should not be extended simultaneously. To do so may cause damage to the flaps and create excessive drag and loss of lift, resulting in increased stall speed for which the stall warning system is not compensated. If the spoilers are extended while the flaps are being extended, the SPOILER annunciator light will flash as the flaps extend beyond the 13˚ position.

Operation (Spoiler Mode)

The spoilers can be symmetrically extended or retracted with the SPOILER switch (Figure 15-9).

When the SPOILER switch is positioned to EXT, the spoiler selector valve is energized, the servo valves meter pressure to the extend side of the spoiler actuators, and the SPOILER light illuminates steady. Full extension is limited to approximately 40˚. Returning the switch to RET deenergizes the spoiler selector valve which repositions to route pressure to the retract side of the actuators, and the servo valves neutralize. The SPOILER light extinguishes when both spoilers are locked down by locks within the actuators (Figure 15-9A).

Spoiler extension and retraction times vary, depending on whether the airplane is airborne or on the ground, and whether the FC 200 AFCS or the FC 530 AFCS is installed. Ground deploy and retract times (***all*** airplanes) is 1–2 seconds and 3–4 seconds, respectively. In-flight deployment times are 3–4 seconds (FC 200) and 5–7 seconds (FC 530). Retract times are 3–4 seconds for ***all*** airplanes.

Spoiler deployment and retraction causes significant nosedown and noseup pitching (respectively). This should be anticipated and offset by application of elevator control pressure and pitch trim, as necessary.

Operation (Spoileron Mode)

During the spoileron (aileron augmentation) mode of operation, the spoilers are independently extended and retracted on a one-to-one ration with the upgoing aileron to increase lateral control in the landing configurations. Aileron augmentation (spoilerons) increases roll control authority up to 50%.

The spoileron mode is automatically engaged when the flaps are lowered beyond 25˚ and the SPOILER switch is in RET position. The spoileron computer continuously monitors aileron position. When the ailerons are displaced from neutral, the computer signals the servo valve to extend the spoiler on the wing with the raised aileron. The spoiler on the opposite wing is held retracted by its servo valve. Spoiler extension is limited to approximately

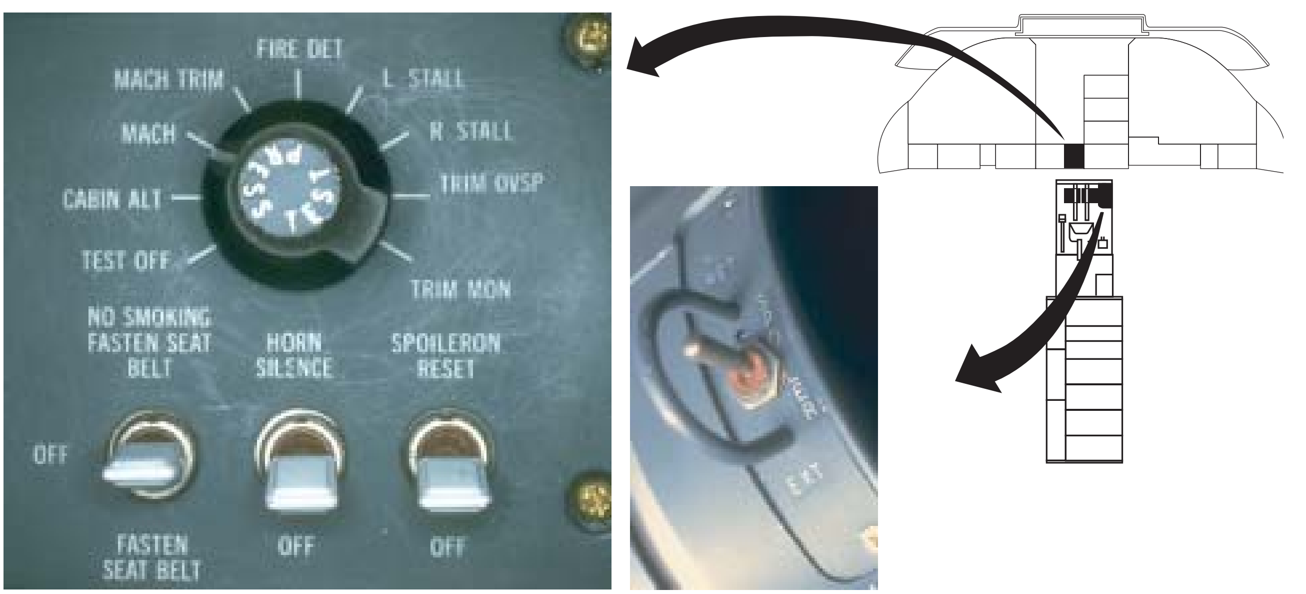

Figure 15-9. Spoiler System

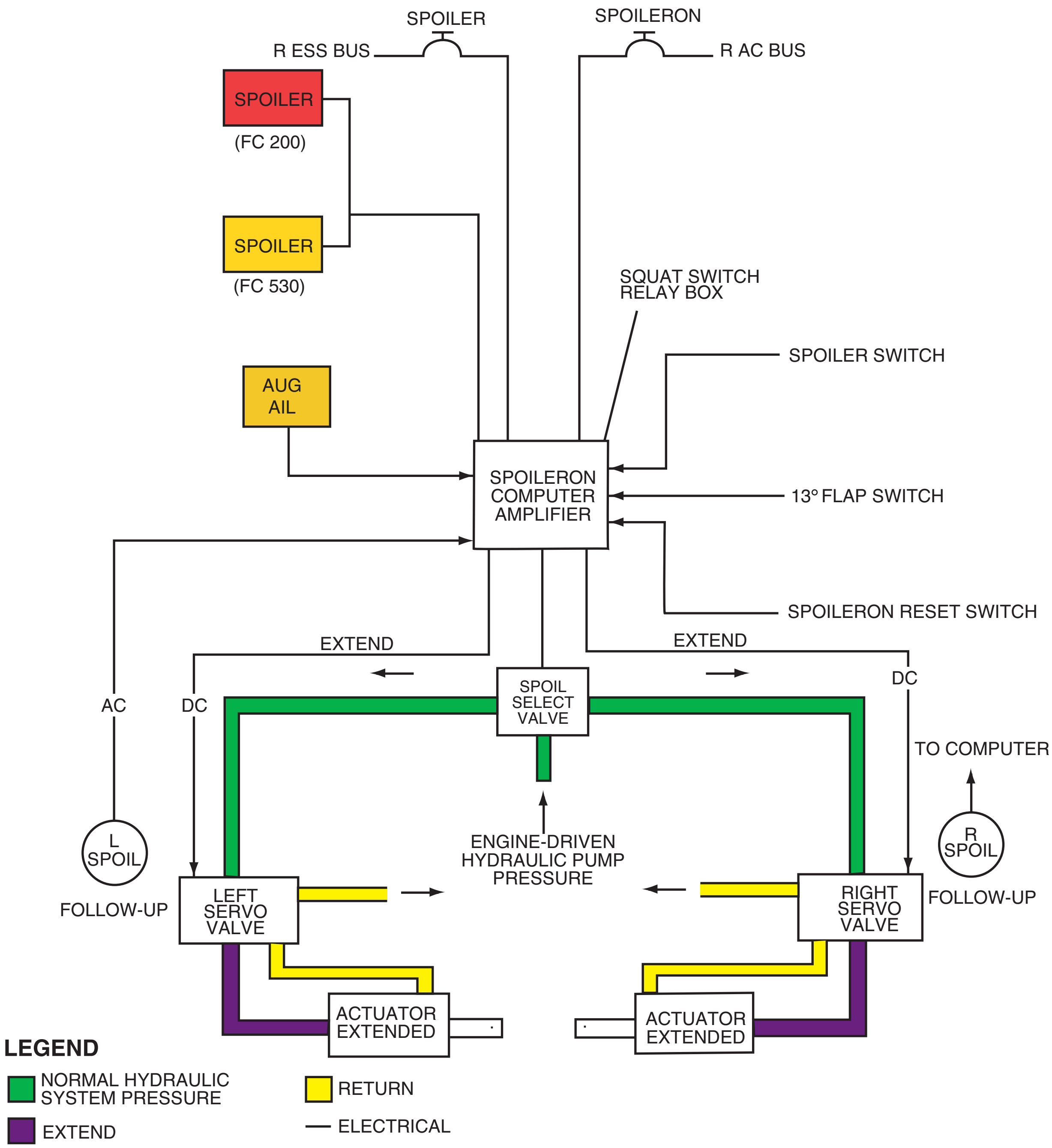

Figure 15-9A. Spoiler Operation

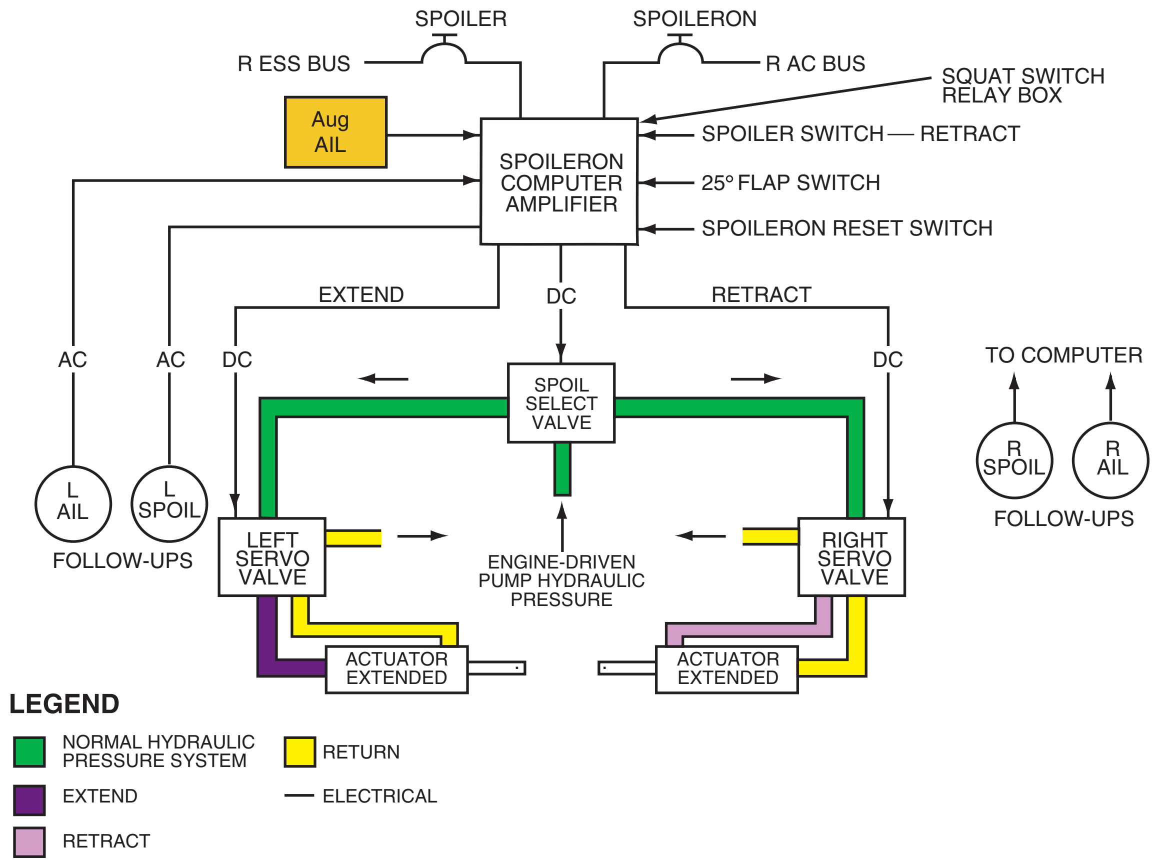

Figure 15-10. Spoileron Operation (Left Aileron Up)

15° during spoileron operation (full up aileron). The SPOILER light will not illuminate during spoileron operation.

Spoileron operation is shown in Figure 15-10.

Spoileron Monitor System

The computer monitors spoiler and spoileron modes of operation by a followup in each spoiler and each aileron. In flight, if a split of more than 6° occurs between the two spoilers (spoiler mode) or between the aileron and spoiler (spoileron mode), the amber AUG AIL light will illuminate and the spoilers will slam down. Both modes will remain inoperative (in flight) as long as the AUG AIL light is illuminated. However, the spoiler mode may be operative on the ground.

Spoileron Reset Switch

The SPOILERON RESET switch (Figure 15-9) is spring-loaded to the OFF position. If a malfunction occurs in either mode (AUG AIL light on), moving the SPOILERON RESET switch momentarily to RESET may restore spoiler/spoileron operation, provided the malfunction has cleared. If the AUG AIL light does not go out, both modes are inoperative in flight.

The SPOILERON RESET switch is also used during the spoileron/spoiler preflight check of monitor circuit operation. On the ground with flaps down, holding the switch in RESET induces a fault that inhibits spoileron movement. Therefore, if the control wheel is turned while holding the switch in RESET, the AUG AIL light should come on after the aileron has deflected approximately 6°. The system can be reset by releasing the SPOILERON RESET switch to OFF and then momentarily moving it back to RESET. Refer to the approved *AFM* for the complete spoileron/spoiler check.

YAW DAMPERS

GENERAL

Either of two yaw damper systems may be installed, depending on whether the airplane is equipped with the FC 200 AFCS, or the FC 530 AFCS. Both systems are described herein.

Two separate, independent (dual) yaw damper systems are installed in all airplanes to provide yaw stability. Either system provides full-time yaw damping in flight (whether or not the autopilot is engaged) by applying rudder against transient motion in the yaw axis, while coordinating the rudder during turns. Switching logic is such that only one yaw damper may be engaged at a time. Each system consists of a yaw rate gyro, a lateral accelerometer, a computer-amplifier, an aileron follow-up, and a DC rudder servo-actuator. Additionally, FC 530 AFCS models use a yaw damper force sensor, a calibration assembly, and a three-axis disconnect box.

The rudder servo actuator incorporates a capstan mechanism (slip clutch) which allows the pilot to override the yaw damper at any time, if required, by applying sufficient rudder pedal force.

When the stall warning indicators are in the shaker range, the yaw damper effectiveness is reduced. The reduction signal for the primary yaw damper comes from the left stall warning system and for the secondary yaw damper from the right stall warning system.

The primary yaw damper uses DC and AC power supplied by the AFCS YAW and PRI YAW DAMP circuit breakers, respectively, on the left AC and essential buses. The secondary yaw damper uses DC and AC power supplied by the SEC AFCS and SEC YAW DAMP circuit breakers, respectively, on the right AC and essential buses.

Both yaw dampers must be operational for flight, with one engaged at all times while airborne. The yaw damper should be disengaged while trimming the rudder, then reengaged. Ground testing of the yaw dampers must be accomplished in accordance with the approved *AFM,* Section 2.

YAW DAMPER CONTROL PANEL

The yaw damper control panel located on the center pedestal (Figure 15-11) provides the yaw damper selection, test, and indicating functions. The dual systems are independent, but share a common control panel.

On airplanes with the FC 200 AFCS, two PWR/TEST buttons, one for each yaw damper, are used to apply power to the respective controller-amplifier, and for system testing. The two green PWR/TEST lights illuminate to indicate that the associated system is powered. The two ENG buttons provide the means of engagement. The two green ENG lights illuminate to indicate an engaged yaw damper. Yaw damper disengagement may be accomplished by depressing the associated inboard OFF button, while power may be removed from the systems by depressing the associated outboard OFF button. A single servo force indicator provides indication of the amount of rudder force being applied by whichever yaw damper happens to be engaged, with clockwise deflection indicating a right rudder force.

On airplanes with the FC 530 AFCS, a single TST button provides simultaneous testing of both yaw damper systems. Two PWR buttons, one for each yaw damper, are used to apply and remove power to their respective controller-amplifiers. Two ENG buttons, one for each yaw damper, are used to engage and disengage the selected yaw damper. The two green ON annunciators illuminate to indicate that the associated system is powered. The two green ENG annunciators illuminate to indicate an engaged yaw damper. A servo force indicator is provided for each yaw damper, providing indication of rudder force being applied by its respective yaw damper, with clockwise deflection indicating right rudder force.

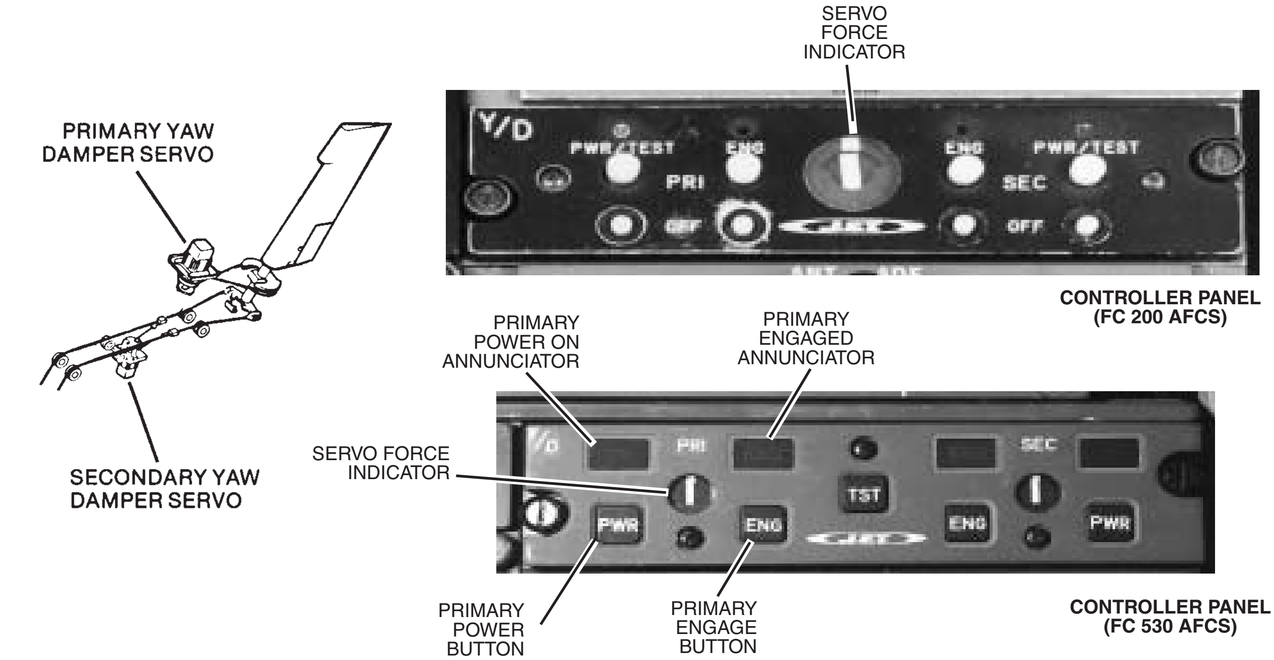

Figure 15-11. Yaw Damper Systems

OPERATION (AIRPLANES WITH FC 200 AFCS)

When the AUTOPILOT master switch is on, electrical power is applied to both yaw damper amplifiers, causing both green PWR/TEST lights to illuminate. However, if the AUTO-PILOT master switch is off, the PWR/TEST buttons, when individually depressed, will apply power to their respective systems, causing the associated PWR/TEST light to illuminate.

With power on (PWR/TEST lights illuminated), depressing either ENG button will engage the corresponding yaw damper and illuminate the associated green ENG light. If one yaw damper is engaged, depressing the opposite ENG button will automatically disengage the first yaw damper and engage the second.

Disengagement of either yaw damper may be accomplished by depressing the corresponding OFF button or by momentarily depressing either pilot's wheel master switch (MSW). On these airplanes, there is no audible annunciation of disengagement.

When a PWR/TEST button is held depressed (during ground testing), the respective PWR/TEST and ENG lights should illuminate. Simultaneously, the force indicator should suddenly move toward the side being tested, then slowly drift past neutral. Releasing the PWR/TEST button should extinguish the ENG light, and the force indicator should suddenly move in the opposite direction, then slowly drift back to neutral. The sudden movement of the force indicator tests the rate gyro circuitry, while the slow drift of the indicator tests the lateral accelerometer. A 5-second waiting period should be observed if retesting is desired.

OPERATION (AIRPLANES WITH FC 530 AFCS)

On these airplanes, the PWR buttons must be depressed in order to apply power to the individual amplifiers. Depressing a PWR button a second time will remove power from the amplifiers.

With power on (PWR annunciators illuminated), depressing either ENG button the first time will engage the corresponding yaw damper and illuminate the associated ENG annunciator. Depressing the ENG button a second time will disengage the yaw damper. If one yaw damper is engaged, depressing the opposite ENG button will automatically disengage the first yaw damper and engage the second.

Disengagement of either yaw damper may also be accomplished by momentarily depressing either pilot's wheel master switch (MSW). On these airplanes, the audible autopilot disconnect tone will always sound to signal yaw damper disengagement.

The TST button provides simultaneous testing of both yaw dampers. With power on (PWR annunciators illuminated), depressing and holding the TST button should illuminate both ENG annunciators. Simultaneously, both force indicators should suddenly move to the right, then slowly drift toward the left. Releasing the TST button should extinguish both ENG annunciators. The sudden movement of the force indicators tests the rate gyro circuitry, while the slow drift of the indicators tests the lateral accelerometers. A 5-second waiting period should be observed if retesting is desired.

On these airplanes, when flaps are extended beyond 25°, the amount of rudder pedal force required to override the yaw damper is significantly reduced. This enables the pilot to apply cross-control pressures without encountering noticeable yaw damper opposition. Because of this, the yaw damper must be engaged all the way to touchdown except when landing must be made with 0°, 8° or 20° flaps, in which case it should be disengaged in the flareout prior to touchdown.

STALL WARNING SYSTEMS

GENERAL

One of two stall warning systems may be installed on the airplane. Airplane SNs 35-067 and subsequent, and 36-018 and subsequent, and earlier airplanes incorporating AAK 76-4, have the "Alpha Dot" system. Earlier unmodified airplanes have the non-Alpha Dot system. Both are dual systems that provide visual and tactile warning of an impending stall and are equipped with the following dual (left and right) components: stall vane/transducer assemblies, computer-amplifiers, red STALL warning lights, stick shaker motors, ANGLE OF ATTACK indicators, and STALL WARNING switches. Both systems use the elevator pitch servo for stick pusher/nudger operation (Figure 15-12).

The Alpha Dot system uses flap position switches, aneroid switches, and rate sensors to provide "bias" information to the computer, which accounts for changes in stall speed in relation to flight conditions and flap configurations. Flap bias is provided by flap switches at the 3°, 13°, and 25° positions. Altitude bias is provided by the aneroid switches at 22,500 feet. The rate sensors establish the rate of change of increasing angle of attack, as in an accelerated approach to a stall. The non-Alpha Dot system is biased only for flap position at 13° and 25° and is not equipped with the aneroid switches or rate sensors.

The left and right systems are completely independent. The systems operate on DC power supplied from the L and R STALL WARN circuit breakers on the left and right battery buses; therefore, each system can be powered even when the battery switches are off. The L and R STALL warning lights are the only components that do not take power directly from the battery buses.

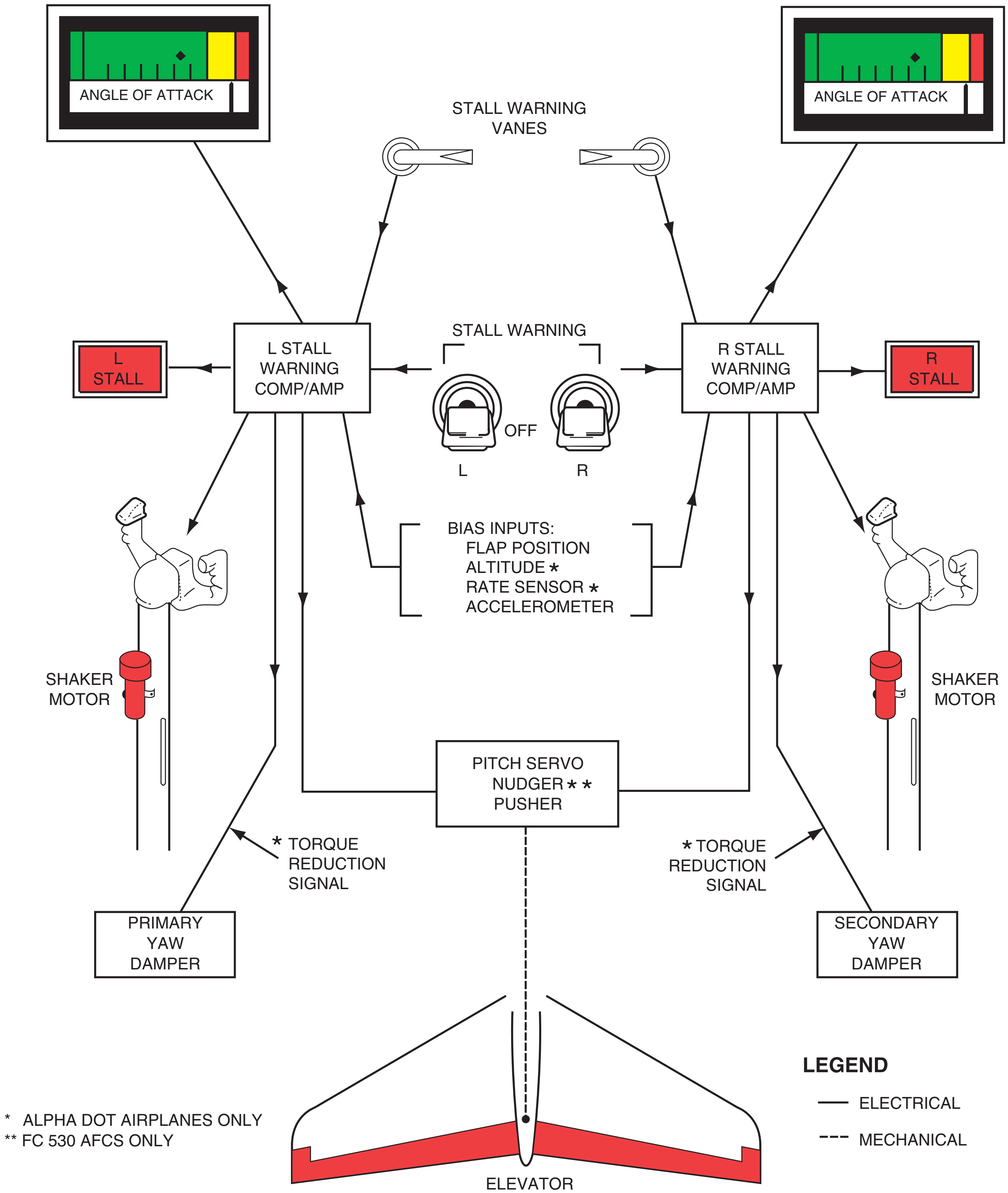

Figure 15-12. Stall Warning System

ANGLE OF ATTACK Indicators

The computers translate signals from the stall vane transducers into visual indications of stall margin on the ANGLE OF ATTACK indicators. The face of the indicators is divided into three color segments—green, yellow, and red. The green segment represents the normal operating range. The yellow segment warns of an approaching stall condition. Tactile warning occurs in this area, alerting the pilot to take positive action. The red segment signifies that aerodynamic stall is imminent or has occurred. The stick pusher is engaged in this area, thereby forcing a reduction in angle of attack.

Warning Lights

The L and R STALL warning lights begin to flash when the respective ANGLE OF ATTACK indicator pointers enter the shaker range, as described above. The STALL WARN lights illuminate steady in the red segment (pusher range). Steady illumination of the lights at any other time indicates a computer power loss or a circuitry malfunction. Cycling the STALL WARNING switch(es) off, then on, may restore normal operation. The lights illuminate whenever the STALL WARN switches are OFF.

Stick Shaker

Stick shaker motors are attached to the front side of each control column. Actuation of the shakers causes a high-frequency vibration in the control columns.

Pusher

The stick pusher function utilizes the elevator pitch servo to reduce angle of attack by decreasing airplane pitch attitude. Pusher activation provides elevator down motion, causing a sudden abrupt forward movement of the control column. The mechanical slip clutch on the pitch servo allows the pilot to override an inadvertent pusher actuation due to malfunction. Additionally, on airplanes equipped with the FC 530 AFCS, depressing and holding the wheel master switch will cancel an inadvertent pusher. See the approved *AFM* for appropriate corrective action.

Nudger (Airplanes with FC 530 AFCS)

On these airplanes, a nudger is incorporated into the stall warning system. As angle of attack increases slightly beyond the point of shaker motor operation (but prior to pusher operation), a gentle pulsating forward push command is applied to the pitch servo (the same servo that operates the pushers).

If the nudger fails to operate, a pulsating nudger monitor horn will sound to alert the pilot. In this case, angle of attack must be decreased immediately because the pusher has also failed.

OPERATION

During flight, the stall warning vanes align with the local airstream. The vane-operated transducers produce a voltage proportional to airplane angle of attack. These signals, biased by information from the flap position switches, altitude switches, and rate sensors (as applicable) are sent to the respective computer. As angle of attack increases, the indicator point moves to the right. As it crosses the green/yellow line, activation of the flashing STALL lights, stick shaker, and stick nudger (if installed) begins. If angle of attack is allowed to increase further, the pusher is activated as the pointer crosses the yellow/red line. Assuming an unaccelerated entry to a stall condition of altitudes below 22,500 feet, the green/yellow line approximates 7 knots or 7% above pusher speed, whichever is higher. The yellow/red line approximates 5% above stall speed (non-Alpha Dot); 1 knot above stall speed (Alpha Dot, except FC 530 AFCS airplanes) or; stall speed ±3 knots (Alpha Dot airplanes with FC 530 AFCS). The 22,500-foot aneroids on all Alpha Dot airplanes cause warning and pusher functions to occur approximately 15 knots earlier at high altitudes in the flaps-up configuration.

MACH OVERSPEED WARNING/STICK PULLER

GENERAL

The Mach overspeed warning system provides audible overspeed warning in the event airplane speed reaches V_{MO} or M_{MO}. The stick puller function signals the pitch servo to torque the elevator noseup if M_{MO} is exceeded. On FC 530 AFCS models, the puller also operates if high-altitude V_{MO} is exceeded.

The stick puller utilizes the autopilot pitch axis circuitry to control the elevator servo force applied. The resultant noseup force on the control column during puller actuation is approximately 18 pounds. If the autopilot is engaged, puller actuation cancels any selected flight director pitch modes and inhibits autopilot use of the pitch servo until the puller is released. System control circuits require 28 VDC and 115 VAC supplied through the L STALL WARN and AFCS PITCH circuit breakers, respectively, on the left essential and AC buses. Power for the stick puller system is controlled through the L STALL WARN switch. The system will be inoperative if the switch is in the OFF position.

OPERATION

The overspeed warning horn is functional whenever the airplane electrical system is powered and either WARN LTS circuit breaker is engaged (essential buses). The stick puller system becomes functional when the L STALL WARN switch is positioned to the on (STALL WARN) position. The STALL WARN switches should remain on at all times in flight except as directed by the approved *AFM* Emergency Procedures and Abnormal Procedures sections. With the stick puller inoperative, speed is limited to 0.74 M_I. The mechanical slip clutch on the pitch servo allows the pilot to override an inadvertent puller actuation due to malfunction. Additionally, on airplanes equipped with the FC 530 AFCS, depressing and holding the wheel master switch will cancel an inadvertent puller. See the approved *AFM* for appropriate corrective action.

QUESTIONS

1. The airplane systems that use the pitch servo to position the elevator are:
- A. Autopilot, Mach trim, stick puller
- B. Autopilot, stick pusher, stick puller
- C. Pusher, stick puller, Mach trim
- D. Yaw damper, stick pusher, stick puller

2. The airplane is trimmed in the pitch axis by:
- A. The elevator trim tab
- B. Canards
- C. The movable horizontal stabilizer
- D. The elevator downspring

3. To enable pitch trim through the control wheel trim switches, the PITCH TRIM selector switch must be in the:
- A. PRI or SEC position
- B. PRI, OFF, or SEC position
- C. PRI position
- D. SEC position

4. Illumination of the red MACH TRIM light indicates:
- A. Mach trim is not operating.
- B. The secondary trim motor is inoperative.
- C. The autopilot is engaged above 0.74 M_I.
- D. The trim speed controller/monitor has detected a trim speed error.

5. The systems that can function with the PITCH TRIM selector switch in the SEC position are:
- A. Primary pitch trim and Mach trim
- B. Secondary pitch trim and Mach trim
- C. Secondary pitch trim and primary pitch trim
- D. Secondary pitch trim and autopilot pitch trim

6. In the event of runaway trim, both trim motors can be disabled by:
- A. Depressing and holding either control wheel master switch
- B. Moving the PITCH TRIM selector switch of OFF
- C. Moving the PITCH TRIM selector switch to EMER position
- D. A or B

7. The MACH position on the rotary system test switch is used to test:
- A. Mach trim and Mach trim monitor
- B. Mach overspeed warning horn and stick puller
- C. Mach monitor
- D. The HORN SILENCE switch

8. In the event of airplane electrical failure, the flap position indicator will:
- A. Be powered by the EMER BAT and indicate actual position of the flaps
- B. Not be powered and will freeze at last flap position
- C. Fail, indicating DN regardless of flap position
- D. None of the above

9. A flashing SPOILER light indicates:
- A. Spoilers are split more than 6°.
- B. Spoiler-aileron relationship has exceeded 6°.
- C. Spoiler system is inoperative.
- D. Spoilers are extended, and flaps are down more than 13°.

10. The SPOILERON RESET switch is used to:

A. Retract the spoilers in the event of a malfunction.
B. Extend the spoilers in the event of a malfunction.
C. Reset the spoiler/spoileron system when the AUG AIL light illuminates.
D. Test the monitor system in flight.

11. If one yaw damper is found inoperative prior to takeoff:

A. The airplane may be flown, but altitude is restricted to 20,000 feet.
B. The airplane may be flown, but altitude is restricted to 41,000 feet.
C. The airplane may be flown, but the YAW DAMP circuit breaker for the inoperative system must be pulled.
D. The airplane must not be dispatched.

12. When the ANGLE OF ATTACK indicator pointers are in the yellow segment:

A. The pusher engages, and the horn sounds.
B. STALL WARN lights illuminate steady.
C. The shakers (and nudgers on FC 530) activate and the STALL WARN lights flash.
D. The shakers activate and the stall warning horn sounds.

13. The electrical power source for the stall warning system is provided by:

A. Battery buses
B. Battery-charging bus
C. Main DC buses
D. Emergency battery

14. If either L or R stall warning system is found to be inoperative before takeoff:

A. The airplane can be flown provided the STALL WARN circuit breaker is pulled for the inoperative system.
B. The airplane can be flown provided the pilot has an ATP rating.
C. The airplane may be flown provided the autopilot and yaw damper systems are operating.
D. The airplane must not be flown.

15. The switch used to turn the stick puller system on and off is the:

A. STICK PULLER switch
B. AUTOPILOT master switch
C. L STALL WARN switch
D. R STALL WARN switch

CHAPTER 16
AVIONICS

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CHAPTER 16
AVIONICS

INTRODUCTION

The Learjet 35/36 avionics consists of, but is not limited to, the navigation system, the automatic flight control system (AFCS), and the comm/nav system. This chapter includes the standard avionics used in the Learjet 35/36. The user should consult applicable supplements in the approved *AFM* and vendor manuals for additional information and information on specific systems not included in this chapter.

GENERAL

The basic navigation system consists of the pitot-static system and air data sensor, and the ram-air temperature gage.

The AFCS includes the flight director, autopilot, dual yaw damper, and Mach trim system. The standard automatic flight control systems installed on the Learjet 35/36 are the Jet Electronics and Technology, Inc. (J.E.T.) FC 200 on the early models, and the FC 530 on the late models. The flight directors can be used independently with the pilot steering the airplane to satisfy the flight director commands as programmed, or the autopilot may be engaged to automatically steer the airplane to satisfy flight director commands as programmed. The dual yaw damper system operates independently of the autopilot and may be engaged with or without the autopilot engaged. The Mach trim system operates at high Mach numbers when the autopilot is disengaged. The yaw damper and Mach trim systems are described in Chapter 15, “Flight Controls.”

The Communication System section of this chapter discusses the static discharge wicks.

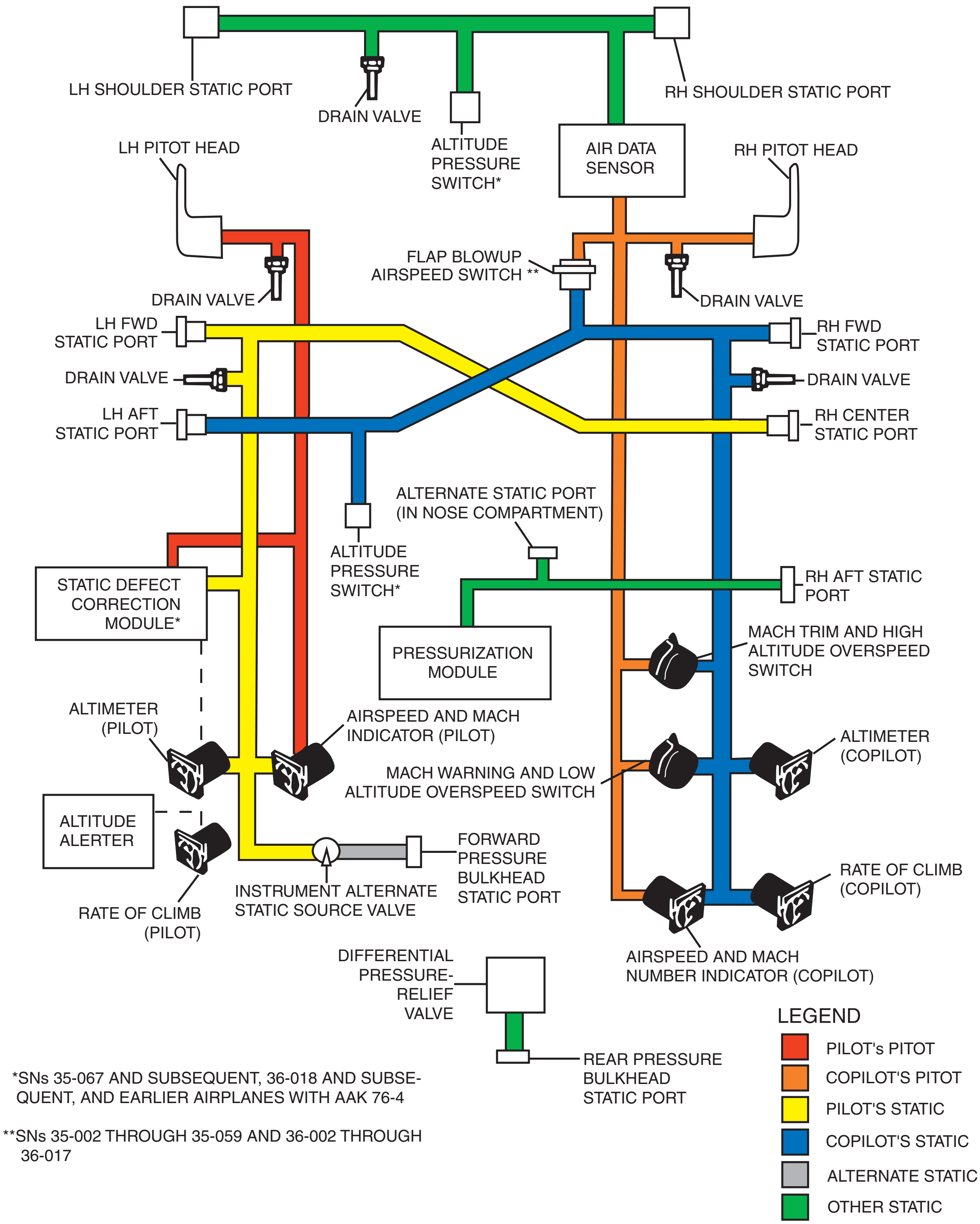


Figure 16-1. Pitot-static System (FC 200 AFCS)

NAVIGATION SYSTEM

PITOT-STATIC SYSTEM (FC 200 AFCS)

The pitot-static system supplies pitot and static air pressure for operation of the airspeed and Mach indicators, the high- and low-altitude overspeed switches, the air data sensor, and the static defect correction module. Static pressure is also supplied to the copilot's vertical velocity indicator, both altimeters, the pressurization control module, and the aft differential pressure relief valve (Figure 16-1).

A heated pitot head is located on each side of the fuselage just forward of the cockpit (Figure 16-2). Pitot heat switches are located on the pilot's anti-icing control panel. They also supply heat to both stall warning vanes. Refer to Chapter 10, "Ice and Rain Protection," for additional information.

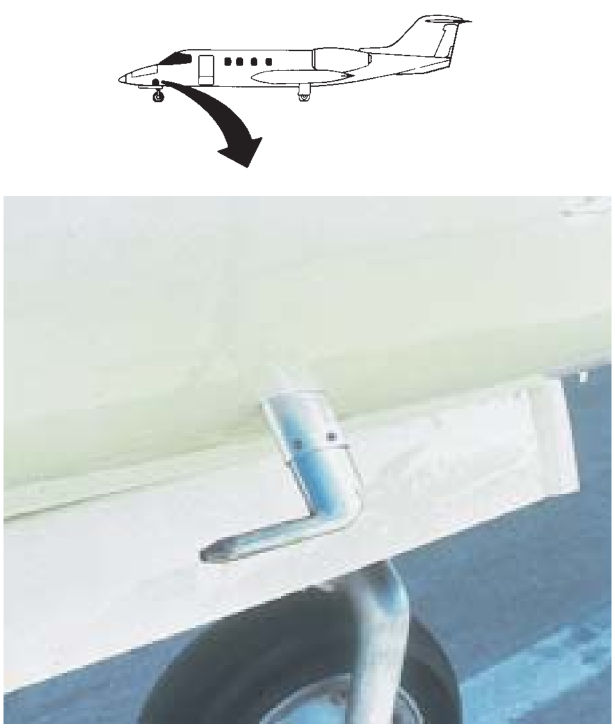


Figure 16-2. Pitot Head (Typical)

The normal static system provides independent sources of static pressure to the pilot's and copilot's instruments. Each static source (pilot or copilot) has one static port on each side of the airplane nose (Figure 16-3). The dual static ports are provided for redundancy and to reduce sideslip effects on the instruments which use static air.

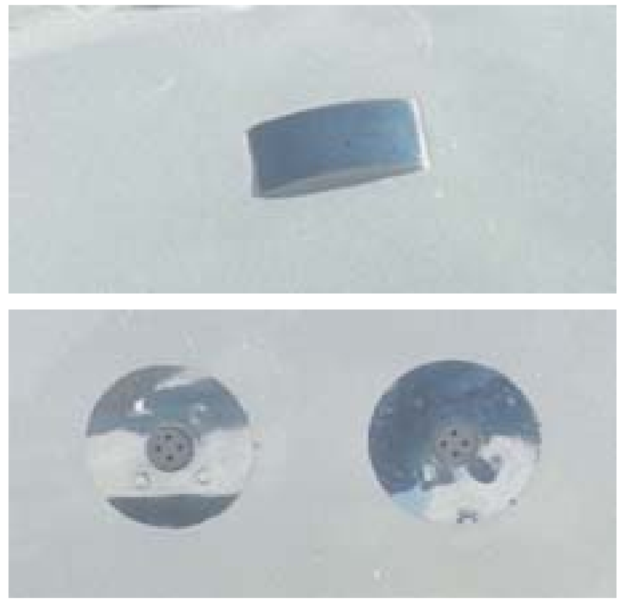


Figure 16-3. Static Ports (Typical)

The left front and right center static ports (both heated) are connected to the pilot's instruments. The left rear and right front static ports (both heated) are connected to the copilot's instruments. The right rear static port (not heated) is connected with an alternate static port inside the nose compartment to provide the pressurization module with a static source. Refer to Chapter 12, "Pressurization," for additional information.

Two heated shoulder static ports are located on top of the fuselage nose in front of the windshield. These ports provide static pressure to the air data sensor and the copilot's FD 108/FD 109 altitude controller (if installed).

An ALTERNATE STATIC SOURCE valve is located below the pilot's instrument panel (Figure 16-4). For normal operation, the lever remains down (CLOSED); for alternate air, the lever is moved up (OPEN).

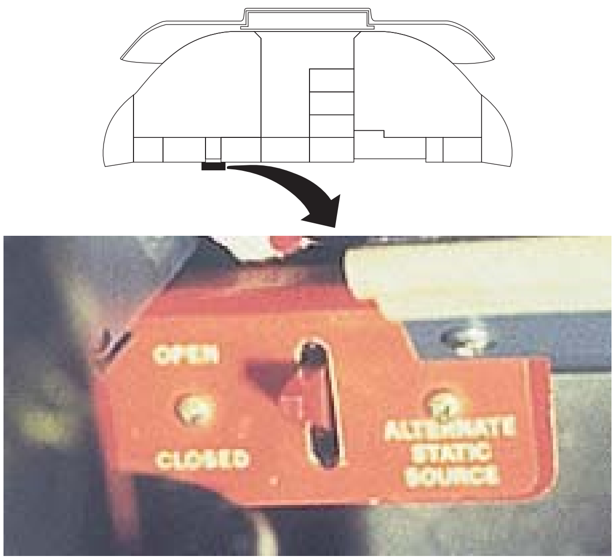


Figure 16-4. ALTERNATE STATIC SOURCE Valve

When the ALTERNATE STATIC SOURCE valve is positioned to OPEN, the ***pilot's*** instruments are connected to an alternate port inside the unpressurized nose section. With OPEN selected, the altimeter and Mach indicators will read slightly lower than normal.

Condensation drain valves for the pitot and static air lines are located adjacent to the nose wheel well doors.

PITOT-STATIC SYSTEM (FC 530 AFCS)

Pitot and static pressure for instruments and systems is obtained from two pitot-static probes, one on each side of the nose section (Figure 16-5). Each probe contains a pitot port in the tip and two static ports on the side. The probes also contain electrical heating elements controlled by the L and R PITOT HEAT switches. (Refer to Chapter 10, "Ice and Rain Protection.") Four drain valves located near the aft end of the nose gear doors (two on each side) are installed at the system's low points to drain moisture from the system.

Figure 16-5. Pitot-static Head (Typical)

The pitot systems (Figure 16-6) are independent. The left probe provides pitot pressure for the pilot's Mach/airspeed indicator, and the right probe head provides pitot pressure for the copilot's Mach/airspeed indicator, the Mach switch (0.74 M_I), gear warning airspeed switch (170 KIAS), air data unit, and other optional equipment.

There are four static ports in the main pitot-static system—two on each pitot-static probe. The ports on one probe are interconnected with those on the other probe to provide redundance. Four solenoid-operated shutoff valves enable the pilot to select the source of static pressure.

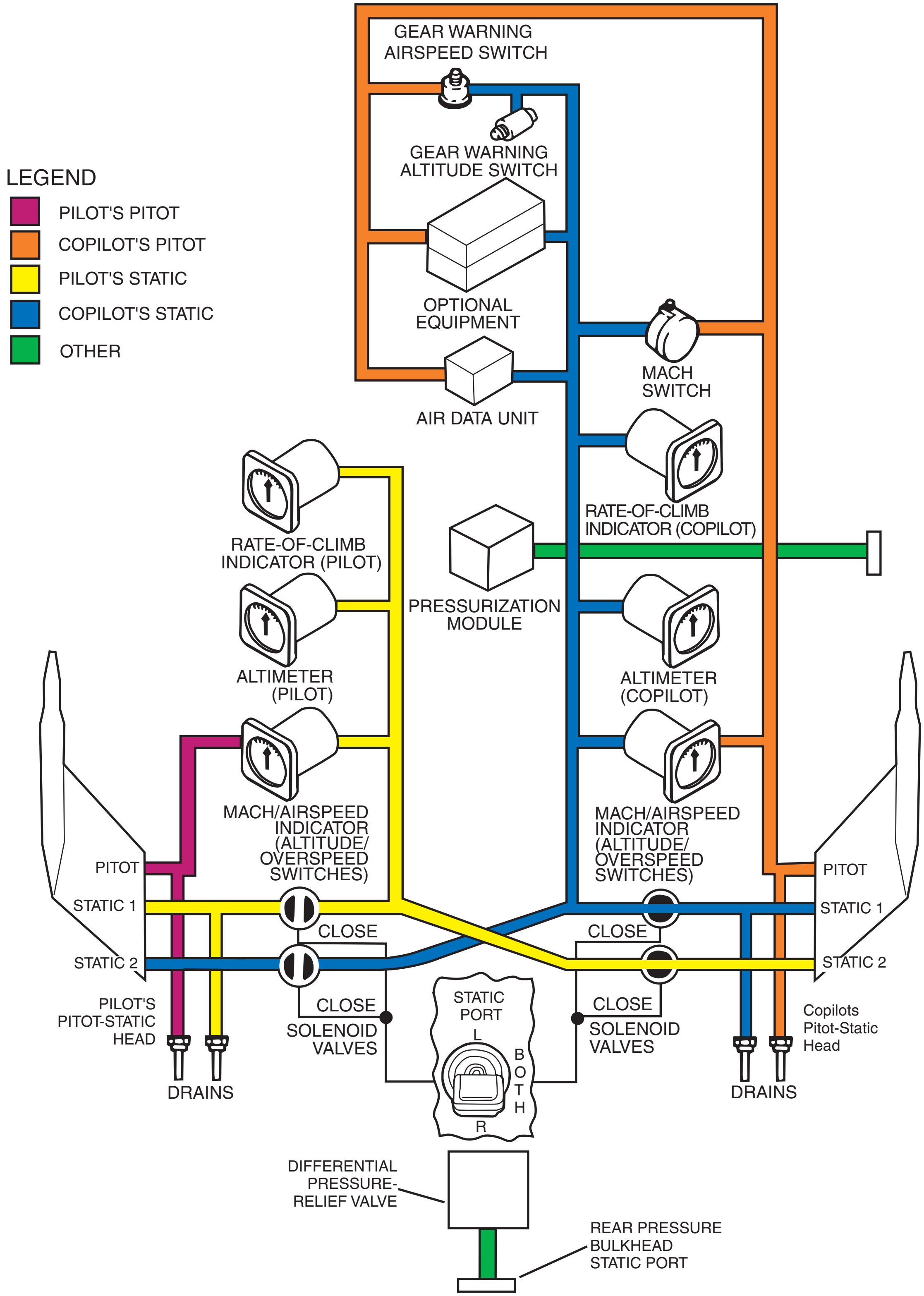


Figure 16-6. Pitot-static System (FC 530 AFCS)

The source of static pressure is controlled with the STATIC PORT switch located on the pilot's switch panel. The STATIC PORT toggle switch has three positions: L (left), BOTH, and R (right). This switch is normally set to the BOTH position except in the event one of the pitot-static heads becomes inoperable or unreliable (Figure 16-7).

In the BOTH position, the pilot's instruments receive static pressure from the forward port on the left head and the aft port on the right head. The copilot's instruments, the Mach switch, the gear warning altitude switch (14,500 feet), the gear warning airspeed switch, the air data unit, and other optional equipment receive static pressure from the front port on the right head and the aft port on the left head. This cross connection eliminates yaw error.

When the STATIC PORT switch is placed in the L or R position, solenoid-operated shutoff valves are energized to shut off the static source from the opposite side static ports. (See Figure 16-6.)

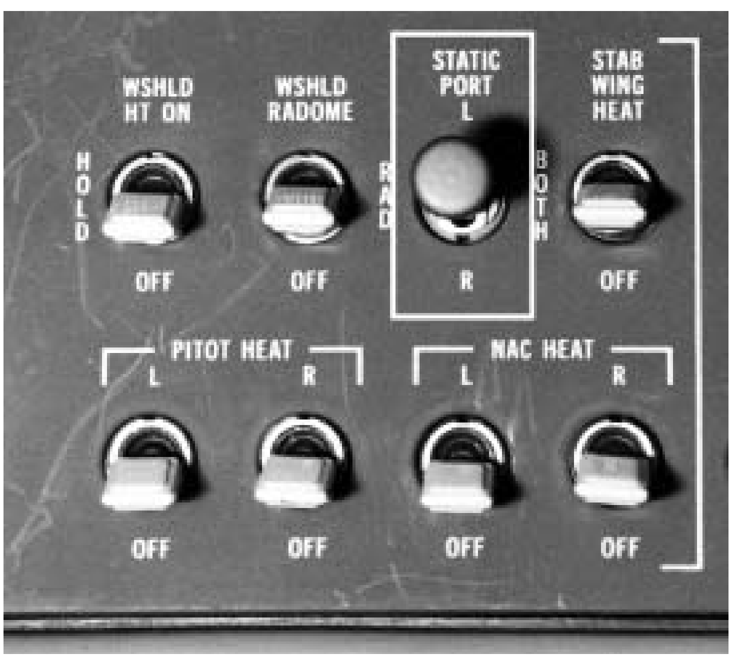
When the STATIC PORT switch is in the L position, static pressure is provided to all user systems only from the two static ports on the left pitot-static head. In the R position, static pressure is provided to all user systems only from the two static ports on the right pitot-static head.

The shutoff valves operate on DC power supplied through the STATIC SOURCE circuit breaker on the left main bus. In the event of electrical failure, all shutoff valves will be open regardless of the STATIC PORT switch position.

A separate unheated static port is flush mounted on the right side of the nose section to provide static pressure to the pressurization control module. Refer to Chapter 12, "Pressurization," for additional information.

AIR DATA

The air data sensor provides air data to the autopilot computer and to the Mach trim computer. On airplanes equipped with the FC 200 automatic flight control system, static input to the air data sensor is from the shoulder static air ports. The FC 530-equipped airplanes use the copilot's static air system for air-data-unit input. On all airplanes, the pitot input is from the copilot's pitot system. The unit is located inside the nose compartment.



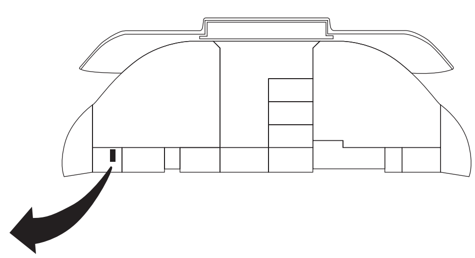


Figure 16-7. STATIC PORT Switch

RAM AIR TEMP INDICATOR

Ram-air temperature is displayed on the RAM AIR TEMP indicator located on the center instrument panel (Figure 16-8). The indicator is calibrated in degrees celsius and requires DC power from the RAM AIR TEMP circuit breaker on the left essential bus. For conversion to outside air temperature (OAT), refer to the Ram Air To Outside Air Temperature Conversion (RAT to OAT) figure in Section V of the approved *AFM*.

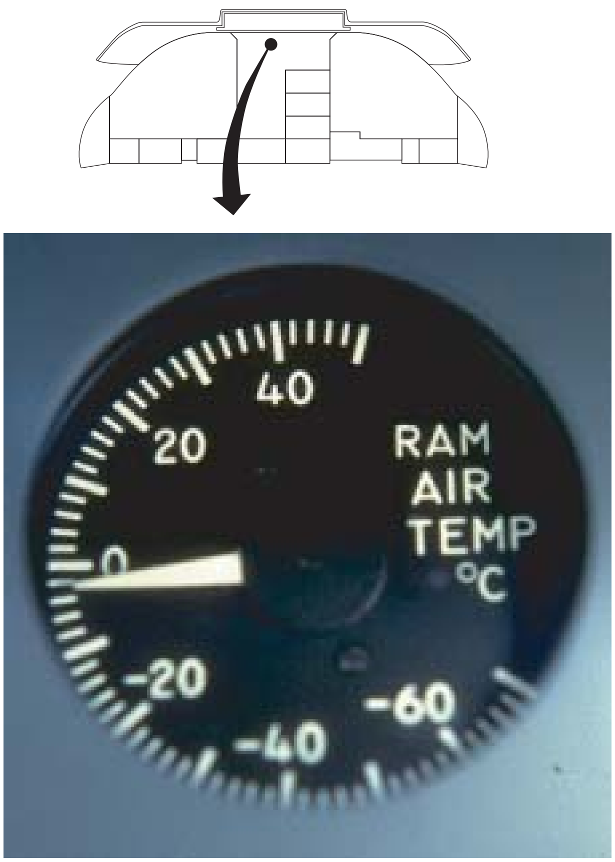


Figure 16-8. RAM AIR TEMP Indicator

AUTOFLIGHT SYSTEM

GENERAL

Either the J.E.T. FC 200 or the J.E.T. FC 530 AFCS may be installed, depending on production serial number. The FC 530 AFCS is installed on SNs 35-408, 35-447, 35-468, 35-506 and subsequent, and 36-054 and subsequent, and earlier SNs incorporating AAK 83-2.

NOTE

The yaw axis is controlled by the dual yaw damper system which operates independently of the autopilot and flight director.

Both systems incorporate a dual-channel AFCS computer which integrates the autopilot pitch and roll axes with the customer-specified flight director system. The AFCS control panel, located in the center of the glareshield, provides pilot access to the autopilot and to the AFCS computer for the flight director programming (mode selection).

The AFCS computer processes information received from the primary vertical and directional gyros, horizontal situation indicator (HSI), the NAV 1 receiver, and the air data sensor. The resulting computed roll and/or pitch command(s) is applied by the computer to the flight director indicator (FDI) command bars, which are built into the pilot's attitude director indicator (ADI).

When engaged, the autopilot is always coupled to the flight director command bars. The pilot has the option of using the flight director with the autopilot disengaged.

Additional controls available to the pilot for control of the autopilot and flight director functions are:

- Both four-way trim switches
- Both maneuver control switches
- The pilot's pitch SYNC switch
- The go-around switch (left thrust lever knob)
- The pilot's HSI heading (HDG) and COURSE selector knob
- The altitude alerter and pilot's altimeter (FC 530 AFCS only)

All of these controls are described in detail in this section.

FLIGHT DIRECTOR SYSTEMS

General

Several different flight directors are available for installation on the Learjet 35/36. The most common installations are the Collins FD 108, FD 109, FIS 84, and FDS 85. Either system includes an attitude director indicator (ADI) and a horizontal situation indicator (HSI) which provide conventional, "raw-data" attitude and heading reference, and glide slope and course deviation displays. The basic airplane attitude and heading references are energized whenever DC and AC power is applied to the airplane.

The flight director system is connected to the AFCS when the AUTO PILOT master switch is turned on.

When the auto pilot master switch is positioned to auto pilot (on), the PWR annunciator illuminates on the AFCS control panel, indicating that power is available to the autopilot and flight director. The AFCS control panel provides for flight director mode selection and annunciation whether the autopilot is engaged or disengaged. Autopilot engagement is accomplished by depressing the ENG button.

Refer to Figures 16-9 through 16-11 for typical installations.

Figure 16-9. ADI and HSI (Typical)

Attitude Director Indicator (ADI)

The pilot's ADI provides a visual presentation of the airplane attitude, as furnished by the remote primary vertical gyro. The flight director indicator (FDI) is built into the ADI and consists of a set of computer-positioned command bars which provide a single-cue command reference for both pitch and roll. The bars move up or down to command pitch, and rotate counterclockwise and clockwise to command roll. When a flight director mode(s) has been selected, the command bars appear in view to provide the computed pitch and roll commands. When the autopilot is engaged, it automatically responds to the command bars. If the autopilot is disengaged, the pilot must

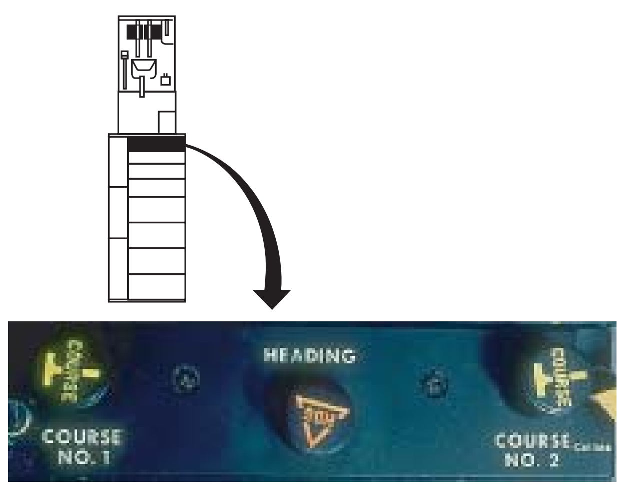


Figure 16-10. Remote Heading and Course Selector (Typical)

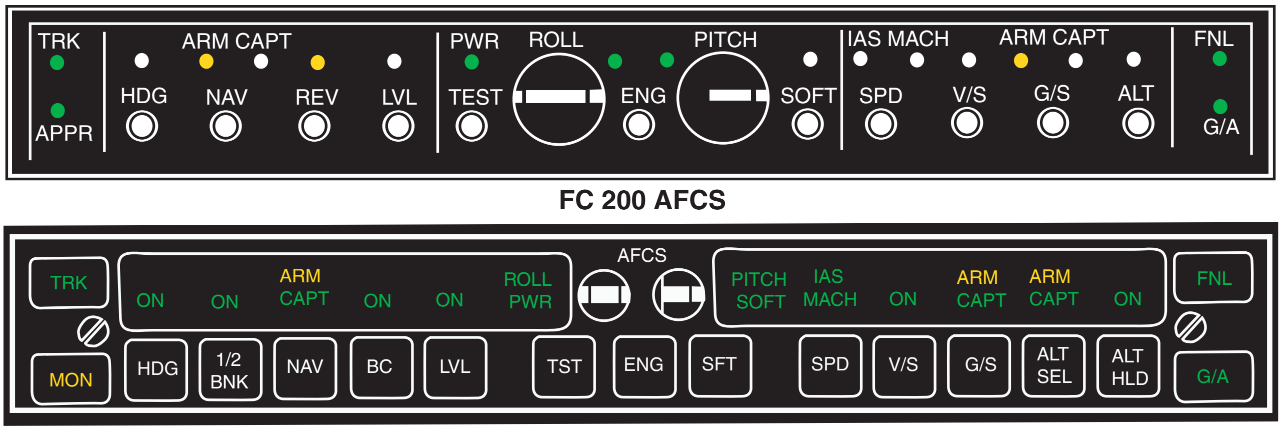
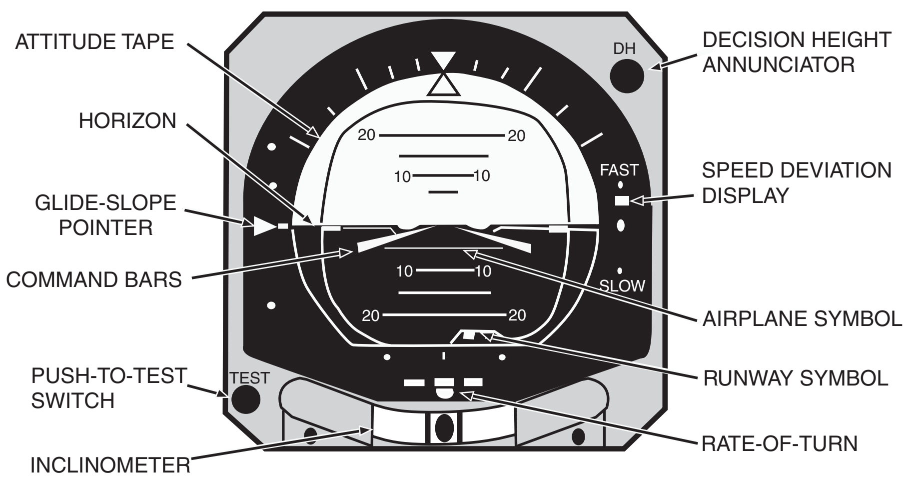


Figure 16-11. Autopilot and Flight Director Control Panels



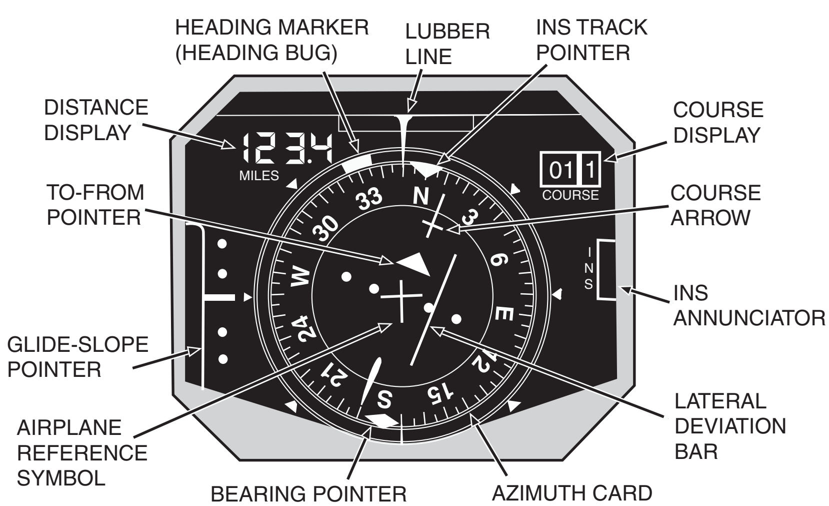


Figure 16-12. ADI and HSI Indications

perform the roll and pitch maneuvers necessary to align the airplane symbol with the command bars. Figure 16-12 illustrates the visual indications provided by the ADI and HSI. The ADI also provides for indication of localizer and glide-slope deviation and turn and slip.

Horizontal Situation Indicator (HSi)

The HSI provides a pictorial presentation of airplane position relative to VOR radials and localizer and glide-slope beams. Heading reference with respect to magnetic north is provided by a remote directional gyro which is slaved to a remote fluxgate compass. The SLAVE-FREE switch on the lower instrument panel allows unslaved operation by selecting FREE, in which case the magnetic reference (flux-gate compass) is removed.

The HSI provides the AFCS computer information regarding existing heading, heading marker reference, selected course, and course deviation. The heading marker (bug) is used to direct the airplane to turn to and maintain the heading selected with the heading (HDG) control knob. The course deviation indicator is used to intercept and track a VOR or LOC course which is set with the course control knob.

AUTOPILOT/FLIGHT DIRECTOR

General

The autopilot will automatically fly the airplane to, and hold, desired heading, attitudes, and altitudes. The autopilot system can also capture and track VOR/LOC/ILS radio beams. The system provides modes for speed control and vertical rate control as well.

On Learjet 35/36 airplanes with the standard avionics installation, the flight director is integrated with the autopilot by a computer through the AFCS control panel on the glareshield. Autopilot and flight director modes are engaged by depressing the applicable mode selector buttons on the control panel. Flight-director-only mode selection is accomplished by depressing the desired mode selectors on the control panel (Figure 16-11), but with the autopilot disengaged.

When the autopilot is not engaged, the ADI command bars indicate the deviation from the desired flight path, enabling the pilot to manually fly the airplane in response to the flight director system. When the autopilot is engaged, it will align the airplane with the command bars automatically to maintain the desired flight path.

Description

Airplane SNs 35-462, 35-447, 35-506 and subsequent and 36-054 and subsequent are equipped with the FC 530 AFCS. Earlier SNs are equipped with the FC 200. Both are manufactured by J.E.T. AAK 83-2 is available to retrofit the earlier models with the FC 530. Both systems include: an autopilot/flight director computer, an electric box, and interface, all located under the pilot's seat; the AFCS control panel mounted in the center glareshield; the roll and pitch servoactuators and follow-ups; the customer-specified flight director system; a roll-rate gyro; the NAV 1 receiver; the primary (pilot's) vertical gyro, directional gyro and HSI; and the air data sensor. The FC 530 also uses the altitude alerter and the pilot's altimeter for its altitude preselect feature.

AFCS Control Panel

The control panel (Figure 16-11) is mounted in the center of the glareshield. It is accessible to both pilots and provides the switches required for autopilot engagement and flight director mode selection. Annunciator lights are green, amber, blue, or white and appear above the mode select switches. The legend (white lettering) on the panel is back-lighted. On FC 200 models, intensity of the annunciator lights and the legend lighting is controlled by the PEDESTAL lights rheostat on the copilot's right sidewall. On FC 530 models, intensity of the annunciator lights is fixed so that they are legible in daylight, while the NAV LTS switch must be turned on for fixed illumination of the legend lighting.

The autopilot engage (ENG) pushbutton is used only to engage the autopilot, while all other pushbutton switches operate with alternate action. The first depression engages a mode; the second depression cancels it. Automatic cancellations also occur. Annunciation of the mode selected appears above the pushbutton. Any operating mode not compatible with a newly selected mode is automatically canceled in favor of the latest selection. This allows the pilot to advance along the flight sequence without the inconvenience of having to deselect modes manually.

Computer

The two-channel (roll and pitch) computer continuously monitors input signals from all AFCS component sensors. The computer is programmed by depressing the desired mode selector button(s) on the AFCS control panel. The computer computes the roll and pitch attitudes necessary to comply and signals the flight director V-bars to position accordingly, while also applying simultaneous signals to the roll and pitch servoactuators (if the autopilot is engaged).

Operation

The autopilot and flight director system controls airplane movement about two axes (pitch and roll). The yaw damper provides independent, automatic control of the yaw axis in the same way as when the airplane is being flown manually.

Pitch Axis Control

The computer pitch channel processes information furnished by the primary (pilot's) vertical gyro, which establishes the basic pitch reference; the air data sensor, which supplies altitude, vertical velocity, and airspeed/Mach information; glide-slope signals from the NAV 1 receiver; and a follow-up device in the pitch servoactuator which signals elevator movement. The FC 530 also uses the altitude alerter and pilot's altimeter for its altitude preselect feature and a vertical accelerometer which monitors G forces.

When a pitch mode is selected on the AFCS control panel, the computer responds by positioning the flight director V-bars accordingly. If the autopilot is engaged, a signal is also applied to the elevator pitch servo which adjusts elevator position. Feedback of elevator movement is provided by the servo follow-up. When the new pitch attitude has been established, the computer zeroes the servo effort by applying horizontal stabilizer trim via the secondary pitch trim motor, thereby preventing any airplane pitching motion when disengaging the autopilot. Pitch changes can also be induced by either pilot's wheel trim switch (without depressing the center button).

The computer uses the servo follow-up to control pitch changes to a rate of 1º per second, and limits pitch attitudes to ±25º (FC 200), and +20º and –10º (FC 530).

Roll Axis Control

The computer roll channel processes information furnished by the primary (pilot's) vertical gyro, which establishes the basic roll reference; the primary (pilot's) directional gyro and HSI, which supply heading and course references; VOR bearing and ILS/LOC course references from the NAV 1 receiver; a roll rate gyro, which provides roll rate data; and a follow-up on the left-hand aileron sector, which signals aileron position.

When a roll mode is selected on the AFCS control panel, the computer responds by positioning the flight director V-bars accordingly. If the autopilot is engaged, a signal is also applied to the aileron roll servo which adjusts aileron position. Feedback of aileron position is provided by the aileron followup. Roll changes can also be induced by either pilot's wheel trim switch when moved to LWD or RWD (without depressing the center button).

The autopilot does not apply trim in the roll axis as it does in the pitch axis. Therefore, if the airplane is out of trim in the roll axis, the autopilot must apply continuous roll servo effort to hold the desired roll attitude. This condition will be noticed by a continuously deflected roll force meter and control wheel.

The computer uses the roll rate gyro to control roll rates to 6º per second (FC 200), and 4º to 5º per second (FC 530). Bank angles are limited to a maximum of 30º.

The FC 200 uses a 13° flap position switch to increase autopilot roll authority when the airplane is configured for approach. This provides more lateral authority at the slower speeds and is annunciated by the green APPR light on the AFCS control panel. The FC 530 uses a 3° flap position switch to desensitize VOR and LOC signals which enhances close-in stability during approaches. It does not affect autopilot roll authority nor is it annunciated.

Electrical Requirements

The autopilot requires DC and AC electrical power. The DC power is supplied through the AFCS, AFCS PITCH, and AFCS ROLL circuit breakers on the left essential bus; 115 VAC is supplied through AFCS PITCH and AFCS ROLL circuit breakers on the left AC bus. All autopilot circuit breakers are located on the pilot's circuit-breaker panel; however, on FC 200 AFCS airplanes, there are three circuit breakers on the front side of the autopilot electric box under the pilot's seat for autopilot and yaw damper annunciator lights and edge lights.

Controls and Indicators

The autopilot and flight director control panel contains most of the controls and indicators used for the autopilot system. Additional controls and indicators are found on the control wheels, the pilot's switch panel, the HSI, the remote heading and course selector, the ADI, the altitude alerter, and the thrust levers.

Autopilot Master Switch

Power is provided to the autopilot and flight director systems when the AUTO PILOT master switch (located on the pilot's lower switch panel) is placed in the AUTO PILOT position, the green PWR (power) annunciator on the autopilot controller illuminates, and the red CMPTR flag on the pilot's ADI goes out of view.

Control Wheel Trim Switch

Either control wheel trim switch (NOSE UP/NOSE DN/LWD/RWD) functions as a manual autopilot controller when moved in any one of four directions ***without*** depressing the trim arming button (Figure 16-13). When an attitude change is made in this manner, the appropriate servo changes the attitude of the airplane and disengages any modes previously selected in the affected axis (except NAV ARM, G/S ARM, and ALT SEL ARM). The autopilot reverts to basic attitude hold in the affected axis when the switch is released.

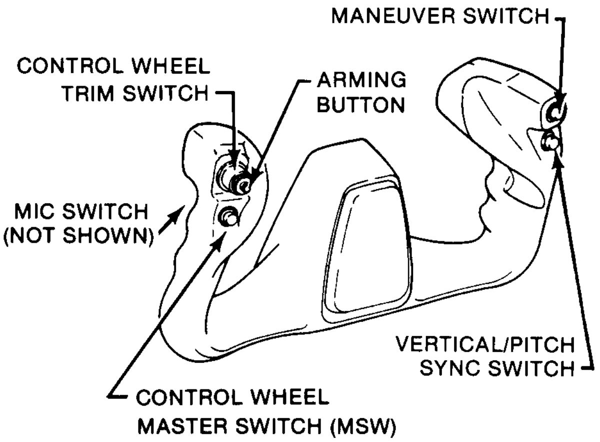


Figure 16-13. Control Wheel Switches (Typical)

Depressing the trim arming button and moving the trim switch in any of the four directions disengages the autopilot, and the autopilot disengagement tone will sound. This is considered the normal means of disengaging the autopilot since it does not disengage the yaw damper. Previously selected flight director modes are not disengaged when the autopilot is disengaged. Autopilot disengagement is further described in this chapter under "Autopilot Disengagement."

Control Wheel Master Switch

Depressing either pilot's control wheel master switch (MSW) disengages the autopilot and the yaw damper. The switch is referred to as the autopilot release/nose steer switch on the FC 200 AFCS.

Control Wheel Maneuver Switch

The control wheel maneuver control switch is referred to as the MANEUVER switch on the FC 200 AFCS and as the MANUV/RP switch on the FC 530 AFCS.

On FC 200 airplanes, depressing and holding either the pilot's or copilot's MANEUVER switch (see Figure 16-13) temporarily releases autopilot access to the pitch and roll servos, biases the command bars out of view, and cancels the ROLL and PITCH modes if engaged previously. This enables either pilot to change the airplane attitude in both pitch and roll axes manually. When the switch is released, the autopilot assumes basic attitude hold functions.

During flight-director-only operation, the maneuver switch will simply cancel all selected flight director modes and bias the command bars out of view.

On FC 530 airplanes, depressing and holding either the pilot's or copilot's MANUV/RP switch temporarily releases autopilot access to the pitch and roll servos and extinguishes the green ROLL and PITCH annunciators, but does not cancel any previously selected flight director roll or pitch modes. This enables either pilot to change the airplane attitude in both pitch and roll axes manually. When the switch is released, the autopilot will resynchronize to and hold the original roll mode and the existing (new) values in the SPD, V/S, or ALT HLD modes, and the green ROLL and PITCH annunciators will illuminate again.

Control Wheel SYNC Switch

On FC 200 airplanes, the pilot's pitch SYNC switch:

- Releases autopilot access to the pitch servo
- Allows the pilot to use manual elevator control to establish a new pitch attitude
- Cancels any selected pitch modes (except G/S ARM), but does ***not*** affect any roll modes
- Causes the command bars to synchronize to the new pitch attitude
- Causes the autopilot to hold the pitch attitude existing at the moment of switch release

On FC 530 airplanes, the pilot's PITCH SYNC switch:

- Is a flight director function only, and has no effect if the autopilot is engaged
- Will cancel any selected pitch modes (except G/S ARM and ALT SEL ARM)
- Synchronizes the command bars to the existing pitch attitude

In the case of a dual flight director installation, the copilot's pitch SYNC switch synchronizes ***only*** the copilot's command bars to the existing attitude and cancels the copilot's G/A mode, if selected. It does not affect the autopilot in any way (as the maneuver switch does).

Autopilot Engagement

The AUTO PILOT master switch must be placed on to accomplish system ground checks prior to flight and normally remains on throughout the flight. When the PWR annunciator is illuminated, the autopilot can then be engaged at any time (except during takeoff and landing) by depressing the ENG button. Illumination of the PITCH and ROLL annunciators indicate engagement of the respective axes.

On FC 200 airplanes, initial autopilot engagement will cancel all previously selected flight director modes (if bank angle happens to be more than 5°), the command bars will disappear, and the autopilot will hold the existing roll and pitch attitudes (if within normal limits). If bank angle is less than 5° at the moment of initial engagement, the LVL light will illuminate and the command bars will be presented, commanding the autopilot to maintain wings level at the existing pitch attitude. If the roll or pitch attitude(s) happens to be beyond the normal limit(s), the autopilot will (at normal rates) roll and/or pitch the airplane to the normal limit(s).

If the PITCH TRIM selector switch is in the OFF position, the autopilot may engage, but will disengage when it attempts to adjust secondary pitch trim and cannot.

On FC 530 airplanes, autopilot engagement will automatically couple to any previously selected flight director mode(s) (except G/A, in which case the G/A light will extinguish and the autopilot will maintain the existing attitude at the moment of engagement). If the autopilot is engaged without any previously selected flight director mode(s), the autopilot will maintain the existing roll and pitch attitudes (if within normal limits), and the command bars will remain out of view. If bank angle is less than 5° at the moment of engagement, the LVL light will annunciate and the command bars will be presented, commanding the autopilot to maintain wings level at the existing pitch attitude. The autopilot will not engage at bank angles in excess of 38° ±2° (regardless of pitch attitude). However, if bank angle happens to be between 30° and 38° ±2° and/or pitch angle is greater than –10° or +20°, the autopilot will (at normal rates) roll and/or pitch the airplane to the normal limit(s).

If the PITCH TRIM selector switch is in the OFF position, the autopilot will not engage.

Attitude Hold Mode

The autopilot is in pitch attitude hold when the PITCH annunciator is illuminated and all other pitch axis annunciators are extinguished (except G/S ARM and, for FC 530, ALT SEL ARM). The autopilot is in roll attitude hold when the ROLL annunciator is illuminated and all other roll axis annunciators are extinguished (except NAV ARM). When the autopilot is in both pitch and roll attitude hold, the flight director command bars will be out of view. Autopilot roll (bank) limit is a nominal 30°, while pitch limits are +25° and –25° (FC 200 airplanes), and +20° and –10° (FC 530 airplanes).

Extended autopilot operation in roll attitude hold or LVL will cancel the automatic erection feature of the vertical gyro. As the vertical gyro precesses, the autopilot will bank the airplane to maintain a zero-bank indication on the attitude indicator.

When the autopilot is in the basic attitude hold mode, attitude commands are accepted by the autopilot through either pilot's control wheel trim switch (arming button not depressed), and the autopilot will hold the attitude that exists when the command is released.

Autopilot/Flight Director Mode Selection

Autopilot and flight director modes are engaged by depressing the applicable mode selector button on the autopilot control panel. The engaged modes may be disengaged by depressing the selector button (except for the SPD mode on the FC 530 AFCS) a second time or by selecting another pitch mode.

Flight-director-only mode selection is made by depressing the applicable mode selector with the autopilot disengaged.

The roll axis modes are LVL (level), HDG (heading), NAV (navigation), VOR or LOC (used in conjunction with the NAV mode), BC (back course—FC 530), REV (back course—FC 200), and ½ BNK (half bank—FC 530 only).

The pitch modes are SPD (speed), V/S (vertical speed), G/S (glide slope), ALT SEL (altitude select—FC 530 only), ALT HOLD (altitude hold), and SFT (soft). The SPD submodes of IAS and MACH, and the V/S, G/S CAPT, ALT SEL CAPT, and ALT HLD modes cancel each other when one is selected. G/S ARM is compatible with a previously selected SPD. V/S or ALT mode, while ALT SEL is compatible with a previously selected SPD, or V/S mode.

Refer to Tables 16-1 and 16-2 for a further description of each mode, the applicable annunciator, and the function of each mode selector switch and annunciator.

Table 16-1. FC 200 AUTOPILOT SYSTEM MODES AND ANNUNCIATORS

MODE	ANNUNCIATOR	FUNCTION
	PWR	Indicates electrical power is available for autopilot/flight director operation (circuit breakers are in and the AUTO PILOT master switch is in the ON position.
TEST		When pressed during ground check, all autopilot controller annunciators illuminate. Failure to light indicates a malfunction in the AFCS or a burned out lamp. Force meters oscillate. When pressed in flight, only the annunciators illuminate.
ENG	ROLL PITCH	When depressed, the autopilot engages and the ROLL and PITCH annunciators illuminate.

Table 16-1. FC 200 AUTOPILOT SYSTEM MODES AND ANNUNCIATORS (Cont)

MODE	ANNUNCIATOR	FUNCTION
SOFT	SOFT	When depressed, the autopilot provides softer response in the pitch and roll axes for flying through turbulence. No function during flight-director-only operation. NOTE SOFT mode is locked out when an ILS frequency is tuned on NAV 1.
HDG	ON	When selected, flight director commands are generated to maneuver the airplane to fly a heading selected with the pilot's HSI heading "bug" using up to 25° of bank. NOTE The turn will be commanded in the shortest direction. It is recommended that the heading "bug" initially be set to not more than 135° in the direction of the desired turn when the turn is more than 135°.
NAV		When selected, it activates the flight director function that captures and tracks VOR and LOC. Functional only when the NAV 1 receiver is tuned to the appropriate frequency, NAV flag is out of view, and desired course is set on the pilot's HSI. The HDG mode may be used to intercept the course provided the intercept angle is less than 90°.
	ARM	Illuminates when NAV mode is selected. Goes out when the CAPT light illuminates. The ARM light will flash if NAV CAPT disengages due to a noisy or failed receiver signal, and in the cone of silence over VOR stations. NOTE When the ARM light is flashing, the flight director will assume a heading hold.
	CAPT (Capture)	Illuminates when the airplane approaches the desired course. Extinguishes if the receiver signal becomes noisy or fails, or while in the cone of silence over VOR stations.
	TRK	In the NAV CAPT mode, illuminates to indicate the airplane has acquired the center of a VOR or LOC beam. Crosswind compensation begins and maximum bank angle will be limited to 15° when it illuminates.
	APPR	The APPR light illuminates when the flaps are lowered beyond 13° and increases the autopilot roll torque limit to compensate for slower airspeed.
REV (BACK COURSE)		Functional only with NAV mode selected for localizer backcourse approach with ILS frequency tuned in. When selected, course information to the flight director is reversed and the glide-slope signal is locked out. The published inbound (front) course must be set in the pilot's HSI course window.
	ON	Indicates that the backcourse mode is selected. NOTE REV may also be used to fly outbound on an ILS front course.

Table 16-1. FC 200 AUTOPILOT SYSTEM MODES AND ANNUNCIATORS (Cont)

MODE	ANNUNCIATOR	FUNCTION
LVL (LEVEL)		When engaged, wings level is commanded by the flight director only if the autopilot is engaged.
	ON	Indicates the level mode is engaged. It is also a function of G/A mode, but has no other flight-director-only functions.
SPD (SPEED)		When selected, the flight director will command a pitch attitude that will maintain the airspeed existing at the time of mode selection. Power must be set by the pilot.
	IAS	Illuminates at altitudes up to approximately 29,000 feet.
	MACH	Illuminates at altitudes above approximately 29,000 feet.
V/S (VERTICAL SPEED)		When selected, the flight director commands a pitch attitude that will maintain the existing vertical speed. Power must be set by the pilot.
	ON	Illuminates when V/A mode is selected. NOTE Before engaging this mode, maintain the desired rate long enough (approximately 15 seconds) for vertical speed indicator lag to diminish.
G/S (GLIDE SLOPE)		When selected, activates the flight director function that captures the glide slope. Functional only when the NAV 1 receiver is tuned to an ILS frequency, an active glide-slope signal is present, the G/S flag is out of view, and the REV mode is **not** selected.
	ARM	Illuminates when the G/S mode is selected and the airplane is not on the glide-slope beam. Goes out when the airplane captures the beam.
	CAPT	Illuminates when the airplane intercepts and captures the glide-slope beam.
	FNL (FINAL)	Illuminates during an ILS or a localizer approach when the beam signal is being desensitized for close-in stability. NOTE The FNL mode will be activated when passing over the outer marker. If the outer marker signal is not available, depressing the NAV 1 TEST button momentarily will activate the FNL mode. This should be accomplished at the final approach fix. The flaps must be down 13° or more to initiate FNL.
ALT (ALTITUDE HOLD)		When selected, the flight director will command an airplane pitch attitude that will maintain the existing altitude.
	ON	Illuminates when ALT hold is engaged.
	G/A (GO-AROUND)	Flight-director-only mode, selected by depressing the GO-AROUND button on the left thrust lever knob. Illuminates the G/A and LVL annunciators, and positions command bars to 9° pitch up, wings level. On SNs 35-002 through 35-009 and 36-002 through 006, the G/A mode is coupled to the autopilot when N_1 is above 80%.

Table 16-2. FC 530 AUTOPILOT SYSTEM MODES AND ANNUNCIATORS

MODE	ANNUNCIATOR	FUNCTION
	PWR	Indicates electrical power is available for autopilot/flight director operation (circuit breakers are in and AUTO PILOT master switch is in ON position).
TST (TEST)		When depressed, all autopilot controller annunciators illuminate (light test only). When depressed simultaneously with ENG button, a system self-test is performed.
	MON (MONITOR)	Illuminates during self-test. Flashes if fault is detected.
ENG	ROLL PITCH	When depressed, the autopilot engages and the ROLL and PITCH annunciators illuminate.
SFT	SOFT	When depressed, the autopilot provides softer response in the pitch and roll axes for flying through turbulence. No function during flight-director-only operation. NOTE SFT mode is locked out when in NAV localizer CAPT, NAV VOR APPR, and ALT SEL CAPT.
HDG	ON	When selected, flight director commands are generated to maneuver the airplane to fly a heading selected with the pilot's HSI heading "bug" using up to 25° of bank. NOTE The turn will be commanded in the shortest direction. It is recommended that the heading "bug" initially be set to not more than 135° in the direction of the desired turn when the turn is more than 135°.
½ BANK	ON	Functional only with HDG or NAV VOR mode selected. Limits bank to a maximum of 13°.
NAV		When selected, it activates the flight director function that captures and tracks VOR and LOC courses. Functional only when the NAV 1 receiver is tuned to the appropriate frequency, NAV flag is out of view, and desired course is set on the pilot's HSI. The HDG mode may be used to intercept the course provided the intercept angle is less than 90°.
	ARM	Illuminates when NAV mode is selected. Goes out when the CAPT light illuminates. The ARM light will flash if NAV CAPT disengages due to a noisy or failed receiver signal, or while in the cone of silence over VOR stations. NOTE When the ARM light is flashing, the flight director will command a heading equal to the selected course plus the computed wind drift correction angle.
	CAPT (Capture)	Illuminates when the airplane approaches the desired course. Extinguishes if the receiver signal becomes noisy or fails, or while in the cone of silence over VOR stations. NOTE When flying in VOR approach, the flaps must be set at 8° or more in order to achieve signal desensitization for close-in stability. This function is provided by the 3° flap switch.
	TRK	In the NAV CAPT mode, illuminates to indicate the airplane is nearing the VOR or LOC beam. Crosswind compensation begins and maximum bank angle will be limited to 15° when it illuminates.

Table 16-2. FC 530 AUTOPILOT SYSTEM MODES AND ANNUNCIATORS (Cont)

MODE	ANNUNCIATOR	FUNCTION
BC (BACK-COURSE)		Functional only with NAV mode selected for localizer backcourse approach. When selected, course information to the flight director is reversed and the glide-slope signal is locked out. The published inbound (front) course must be set in the pilot's HSI course window.
	ON	Indicates that the backcourse mode is selected. Is also a function of G/A mode. NOTE BC may also be used to fly outbound on an ILS front course.
LVL (LEVEL)		When the LVL button is depressed (autopilot engaged or not), the flight director will command wings level, and any previously selected roll mode will be canceled. If a pitch mode happens to be engaged, pitch commands for that mode will not be affected; otherwise, the command bars will assume the existing pitch attitude.
	ON	Indicates the level mode is engaged. NOTE During flight-director-only operation, selecting SPD, V/S, or ALT HLD without a prior roll mode selection will automatically engage the LVL mode.
SPD (SPEED)		When selected, the flight director will command a pitch attitude that will maintain the airspeed existing at the time of mode selection. Power must be set by the pilot.
	IAS	Illuminates when the SPD mode selector is first depressed. The existing IAS is maintained.
	MACH	Illuminates when the SPD mode selector is depressed a second time. The existing Mach number is maintained. NOTE The switch will cycle between IAS and MACH, always starting with IAS upon initial engagement. Therefore, to disengage the mode, another pitch mode must be engaged, or momentarily move either control wheel trim switch (without depressing arming button) in the noseup or nosedown direction. In the flight-director-only mode, SPD is disengaged with activation of the pitch sync switch.
V/S (VERTICAL SPEED)		When selected, the flight director commands a pitch attitude that will maintain the existing vertical speed.
	ON	Illuminated when V/S mode is selected. NOTE Before engaging this mode, maintain the desired rate long enough(approximately 15 seconds) for vertical speed indicator lag to diminish.

Table 16-2. FC 530 AUTOPILOT SYSTEM MODES AND ANNUNCIATORS (Cont)

MODE	ANNUNCIATOR	FUNCTION
G/S (GLIDE SLOPE)		When selected, activates the flight director function that captures and tracks glide slope. Functional only when the NAV 1 receiver is tuned to an ILS frequency, an active glide-slope signal is present, the G/S flag is out of view, and the BC mode is not selected.
	ARM	Illuminates when the G/S mode is selected and the airplane is not on the glide-slope beam. Goes out when the airplane captures the beam.
	CAPT	Illuminates when the airplane captures the glide-slope beam.
	FNL (FINAL)	illuminates during an ILS or a localizer approach when the LOC and G/S beam signals are being desensitized for close-in stability. NOTE If the radio altimeter signal is valid, the FNL light will illuminate at approximately 1,200 feet AGL. If the radio altimeter is not valid, the FNL mode will be activated when passing over the outer marker. If the radio altimeter and outer marker are not valid, depressing the NAV 1 TEST button will activate the FNL mode. This should be accomplished at the final approach fix. The flaps must be down 3° or more to initiate desensing (FNL) manually.
ALT HLD (ALTITUDE HOLD)		When selected, the flight director will command an airplane pitch attitude that will maintain the existing altitude. Vertical velocity should be less than 1,000 ft/min.
	ON	Illuminates when ALT HLD is engaged.
ALT SEL (ALTITUDE SELECT)		When selected, the flight director will capture preselected altitudes.
	ARM	Illuminates when ALT SEL is activated. The desired altitude is set on the altitude alerter and any pitch mode (except ALT HLD) may be used to attain that altitude. Upon nearing the selected altitude, the ARM light goes out and any other pitch mode in use disengages.
	CAPT	Illuminates when an altitude interception begins. When the airplane is within 20 feet of the selected altitude and vertical speed within limits, the ALT HLD mode engages, the ALT HLD ON light illuminates, and the ALT SEL CAPT light extinguishes.
	G/A (GO-AROUND)	Flight-director-only mode, selected by depressing the GO-AROUND button on the left thrust lever knob. Disengages autopilot (if engaged), illuminates the G/A and LVL annunciators, and positions command bars to 9° pitch up, wings level.

Autopilot Disengagement

Whenever the autopilot and/or roll axes disengage, the applicable PITCH and/or ROLL annunciators will extinguish and the autopilot disengage tone will sound, as defined below:

- Either control wheel trim switch, with arming button depressed and moved in any of the four directions (NOSE UP, NOSE DN, LWD, or RWD), will disengage both autopilot axes.
- Either control wheel master switch (MSW), when depressed, will disengage both autopilot axes and the yaw damper.
- The AUTO PILOT master switch, when set to OFF, will disengage both autopilot axes.
- The PITCH TRIM selector switch, when moved to the OFF position, will disengage both autopilot axes, but only when it attempts to trim the horizontal stabilizer and cannot (FC 200). On FC 530 airplanes, autopilot disengagement is immediate.
- With the PITCH TRIM selector switch in either the PRI or SEC position, moving the pedestal NOSE DN–OFF–NOSE UP switch to NOSE UP or NOSE DN will disengage both autopilot axes.
- Individual axes may be disengaged by pulling the applicable axis AC or DC circuit breakers (pilot's AC and essential buses).

NOTE

On the FC 530 AFCS, if the AC AFCS PITCH circuit breaker is out, the puller system is also rendered inoperative and airspeed must be limited to 0.74 M_1.

- Depressing the pilot's VG ERECT button or actuating the pilot's L-R SLAVE switch will disengage both autopilot axes.
- On the FC 530 AFCS, depressing the GO-AROUND button (left thrust lever knob, will disengage the autopilot and select flight director G/A (go-around) and LVL modes. This positions the command bars at a wings level 9° noseup pitch position.

NOTE

On SNs 35-002 through 35-009, and 36-002 through 36-006, the G/A mode is coupled to the autopilot if engaged when power is advanced to approximately 80% N_1.

Servo Force Meters

Two servo force meters are located in the center of the control panel. The indicators provide an indication of what autopilot servo forces are present when the autopilot is engaged. The left one indicates roll force and the right, pitch force. If the force meter(s) is deflected, the appropriate axis should be trimmed to center the meter(s) prior to engaging the autopilot. If the autopilot is engaged, and the meter(s) indicates a steady deflection, the autopilot should be disengaged and the appropriate axis retrimmed. Small deflections before and after engagement are normal.

Roll Monitors

The computer uses the roll rate gyro and the pilot's vertical gyro to control the rate of roll and bank angle, respectively.

On FC 200 airplanes, excessive roll rate will disengage the roll axis, sound the disengage tone, and extinguish the ROLL light.

On FC 530 airplanes, excessive roll rate or bank angle in excess of approximately 40° will disengage both axes, sound the disengage tone, and extinguish the ROLL and PITCH lights.

Pitch Trim Monitor

The autopilot maintains pitch trim using the airplane's secondary pitch trim system. Whenever the autopilot is engaged and the secondary trim runs in a direction opposite the elevator servo force, a monitor will disengage both axes, sound the disengage tone, and extinguish the ROLL and PITCH lights.

Out-of-trim Monitors (FC 530 AFCS Only)

With the autopilot engaged, the out-of-trim monitors cause the applicable PITCH or ROLL annunciator to ***flash*** if an out-of-trim condition exists to a degree that servo force is continuously applied for more than approximately 20 seconds. The light continues to flash until either the trim is restored or the axis is disengaged.

G-force Monitor (FC 530 AFCS Only)

G forces are sensed by the vertical accelerometer with the autopilot engaged. The G-force monitor causes the elevator to streamline whenever the G level reaches 1.6 G or 0.6 G. The pitch axis remains engaged, but keeps the elevator streamlined. Previously engaged pitch modes also remain on. When the airplane is within the G limits, the pitch axis resumes normal elevator inputs.

Autopilot/Stick Nudger/Pusher/ Stick Puller Interface

If the autopilot is engaged and the stick nudger (FC 530 AFCS), pusher, or puller actuates, any selected pitch mode disengages. The autopilot then maintains a synchronous standby mode until the nudger, pusher, or puller releases. Upon this release, the autopilot maintains the existing pitch attitude.

Altitude Alerter

The altitude alerter provides automatic visual and aural signals announcing approach to and departure from a selected altitude. The alerter is a direct-reading instrument with a five-digit display (Figure 16-14).

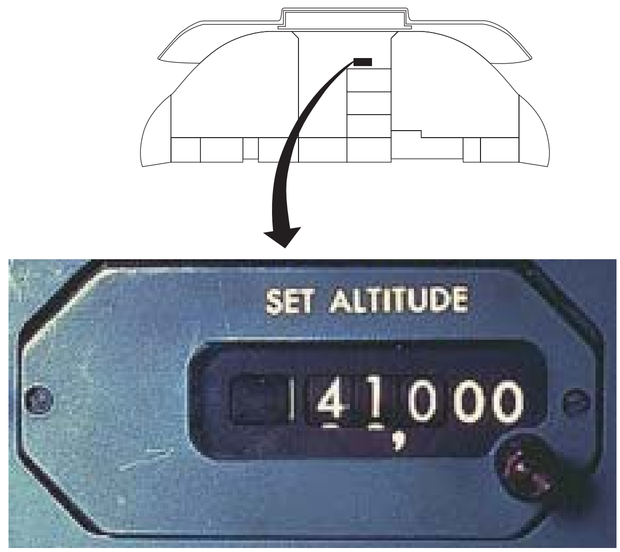


Figure 16-14. Altitude Display

The altitude alerter located in the center instrument panel functions in conjunction with the pilot's altimeter. An OFF flag adjacent to the altitude display will be in view whenever power is not available to the alerter. During flight, as the airplane passes within approximately 1,000 feet of the selected altitude, the amber ALT annunciators on the pilot's and copilot's altimeters will illuminate and an alert bell will sound. The point at which the approach to the preselected altitude is annunciated depends upon airplane vertical speed. The annunciators will extinguish when the airplane is within 300 feet of the preselected altitude. Should the altitude subsequently deviate more than ±300 feet from the selected altitude, the ALT annunciators will illuminate and the alert bell will sound.

The altitude alerter is also used to program the flight director altitude select (ALT SEL) mode on the FC 530 AFCS.

COMMUNICATION SYSTEM

STATIC DISCHARGE WICKS

A static electrical charge, commonly referred to as "P static" (precipitation static), builds up on the surface of an airplane while in flight and causes interference in radio and avionics equipment operation. The charge may be dangerous to persons disembarking after landing as well as to persons performing maintenance on the airplane. The static wicks are installed on all trailing edges (Figure 16-15) to dissipate static electricity.

Figure 16-15. Static Wicks (Typical)

QUESTIONS

NAVIGATION SYSTEM

1a. FC 200 AFCS—The static ports for flight instrument operation are located:

A. In the unpressurized nose section
B. On the top and bottom of the pitot-static heads
C. Flush mounted on the left and right sides of the fuselage nose section
D. On both sides of the aft fuselage

1b. FC 530 AFCS—The static ports for flight instrument operation are located:

A. In the unpressurized nose section
B. In the pitot-static heads
C. Flush mounted on the left and right sides of the nose section
D. On both sides of the aft fuselage

2a. FC 200 AFCS—The pilot controls the static pressure source for the pilot's flight instrument operation:

A. Electrically with the STATIC PORT switch
B. Mechanically with the STATIC PORT switch
C. Electrically with the ALTERNATE STATIC SOURCE switch
D. Mechanically with the ALTERNATE STATIC SOURCE valve lever

2b. FC 530 AFCS—The pilot controls the static pressure source for the pilot's flight instrument operation:

A. Electrically with the STATIC PORT switch
B. Mechanically with the STATIC PORT switch
C. Electrically with the ALTERNATE STATIC SOURCE switch
D. Mechanically with the ALTERNATE STATIC SOURCE switch

3a. FC 200 AFCS—The air data sensor receives pitot information from:

A. The left pitot head
B. The right pitot head
C. Both pitot-static heads
D. The right pitot-static head

3b. FC 530 AFCS—The air data unit receives pitot information from:

A. The left pitot head
B. The right pitot head
C. Both pitot-static heads
D. The right pitot-static head

4a. FC 200 AFCS—The air data sensor receives static information from:

A. The shoulder static air ports
B. The pressurization module static air port
C. The right pitot-static head
D. Both pitot-static heads

4b. FC 530 AFCS—The air data unit receives static information from:

A. The shoulder static air ports
B. The pressurization module static air port
C. The right pitot-static head
D. Both pitot-static heads

AUTOFLIGHT SYSTEM

5. During flight-director-only operation, depressing the pilot's SYNC switch:

A. Disengages G/S ARM and ALT SEL ARM (FC 530)
B. Inhibits the roll and pitch axes
C. Disengages any pitch mode except G/S ARM and ALT SEL ARM (FC 530)
D. Cages the ADI to airplane centerline reference

6. The ADIs and HSIs are energized when:

A. An inverter is turned on.
B. The AUTO PILOT master switch is positioned to ON.
C. The TEST switch is depressed.
D. The VG ERECT switch is depressed.

7. To control the airplane in the pitch axis, the autopilot uses the:

A. Pitch servo only
B. Pitch servo and trim tabs on the elevators
C. Horizontal stabilizer trim actuator only
D. Pitch servo and secondary pitch trim motor

8. If the stick nudger or puller engages during autopilot operation:

A. Selected pitch modes will be canceled.
B. The autopilot maintains a synchronous standby mode in the pitch axis until the nudger or puller releases.
C. Selected roll modes remain engaged.
D. All the above

9. When using the flight director REV (or BC) mode during a localizer back course approach, the:

A. Reciprocal of the front course must be set in the HSI course window.
B. Glide-slope receiver signal is captured.
C. Published inbound (front) course must be set in the HSI course window.
D. Both B and C are correct.

10. When using the autopilot, the following limitation applies:

A. The pilot and copilot must be in their respective seats with seat belts fastened.
B. The pilot or copilot must be in his respective seat with seat belt fastened.
C. The autopilot must be operative for airplane flight if the Mach trim system in inoperative.
D. Do not extend or retract gear or flaps with autopilot engaged.

COMMUNICATION SYSTEM

11. The static wicks are important because they:

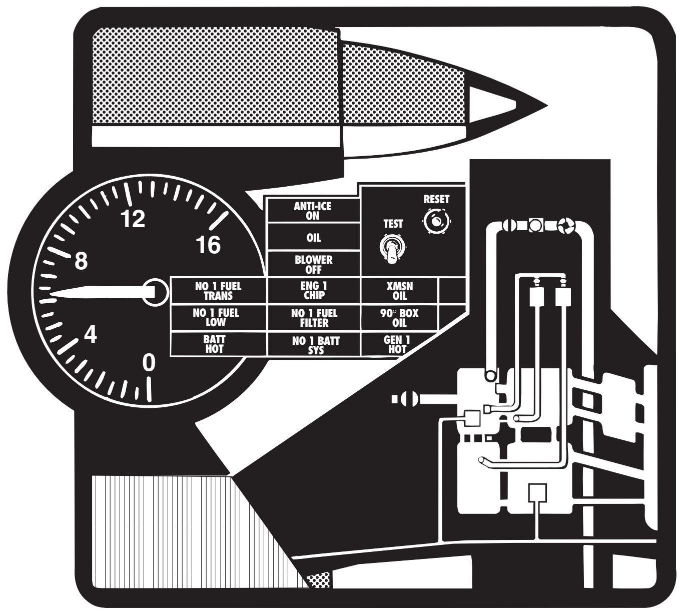
A. Collect static electricity
B. Function as an aerodynamic aid
C. Dissipate lightning strikes
D. Dissipate static electricity

CHAPTER 17
MISCELLANEOUS SYSTEMS

CONTENTS

ILLUSTRATIONS

CHAPTER 17
MISCELLANEOUS SYSTEMS



INTRODUCTION

Miscellaneous systems covered in this section include the oxygen system, the drag chute, and the squat switch system. The airplane uses high-pressure oxygen stored in a cylinder located in either the right nose section or the dorsal fin. Optional long-range oxygen installations are available. The drag chute is offered as optional equipment. The squat switch system provides the airborne and ground signals which activate or deactivate certain systems during takeoff and landing.

GENERAL

The 35/36 series oxygen system consists of the crew distribution system and the passenger distribution system. Oxygen is available to the crew at all times and can be made available to the passengers either automatically above 14,000 feet cabin altitude or manually at any altitude by the cockpit controls. The system is primarily designed for use in the event of rapid decompression or pressurization system failure. It is ***not*** designed for planned extended unpressurized flight at high cabin altitudes requiring the use of oxygen.

The optional drag chute is used to improve deceleration on the ground. It is most effective when deployed at higher speeds, but can still be effective when deployed at speeds below 60 knots.

The squat switch system includes two switches, one located on each of the main gear scissors, and a relay box.

OXYGEN SYSTEM

The oxygen system components include an oxygen storage cylinder and a shutoff valve-regulator assembly, an overboard discharge indicator, an oxygen pressure gage, and the distribution systems for the crew and passengers. Figure 17-1 depicts the oxygen system.

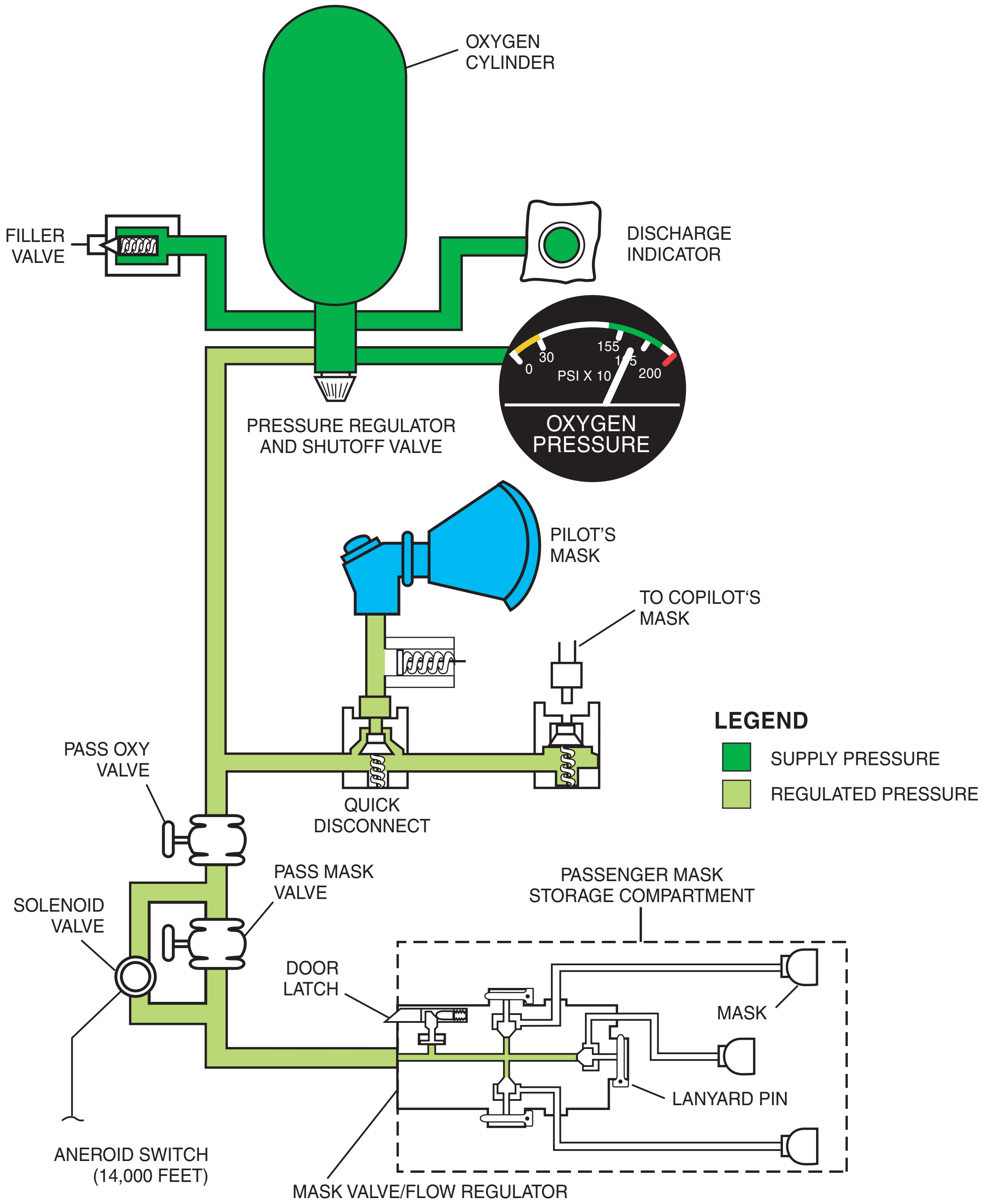


Figure 17-1. Oxygen System

OXYGEN CYLINDER

The system is supplied with oxygen from a storage cylinder located in the right nose section on airplane SNs 35-002 through 35-491 and 36-002 through 36-050 (Figure 17-2). On airplane SNs 35-492 and 36-051 and subsequent, the cylinder is located in the dorsal fin. An optional long-range installation incorporating two cylinders is available (location of the cylinders varies).

Each oxygen cylinder has a storage capacity of 38 cubic feet at 1,800 psi. The shutoff valve and pressure-regulator assembly is attached to the storage cylinder and provides for pressure regulation, pressure indication, and servicing. Oxygen pressure for the passenger and crew distribution system is regulated at 60 to 80 psi. The cylinder, along with its shutoff valve and regulator assembly, can be reached through an access door. Under normal conditions, this valve should always be left in the on (open) position (a specified item on the exterior preflight inspection). The pilot should be aware that if the oxygen cylinder shutoff valve is closed, oxygen pressure will still be read on the OXY PRESS gage in the cockpit. During the interior preflight inspection, ensure that the shutoff valve is open by checking for oxygen flow through both crew oxygen masks, using the 100%, or EMER, position.

OVERBOARD DISCHARGE INDICATOR

The overboard discharge indicator (green blowout disc) (Figure 17-2) provides the pilot with a visual indication that there has not been an overpressure condition in the oxygen storage cylinder. The disc blows out if the cylinder pressure reaches 2,700 to 3,000 psi, releasing all oxygen pressure. System pressure should normally be between 1,550 and 1,850 psi. The green blowout disc is located on the right side of the dorsal fin or the lower right side of the nose section.

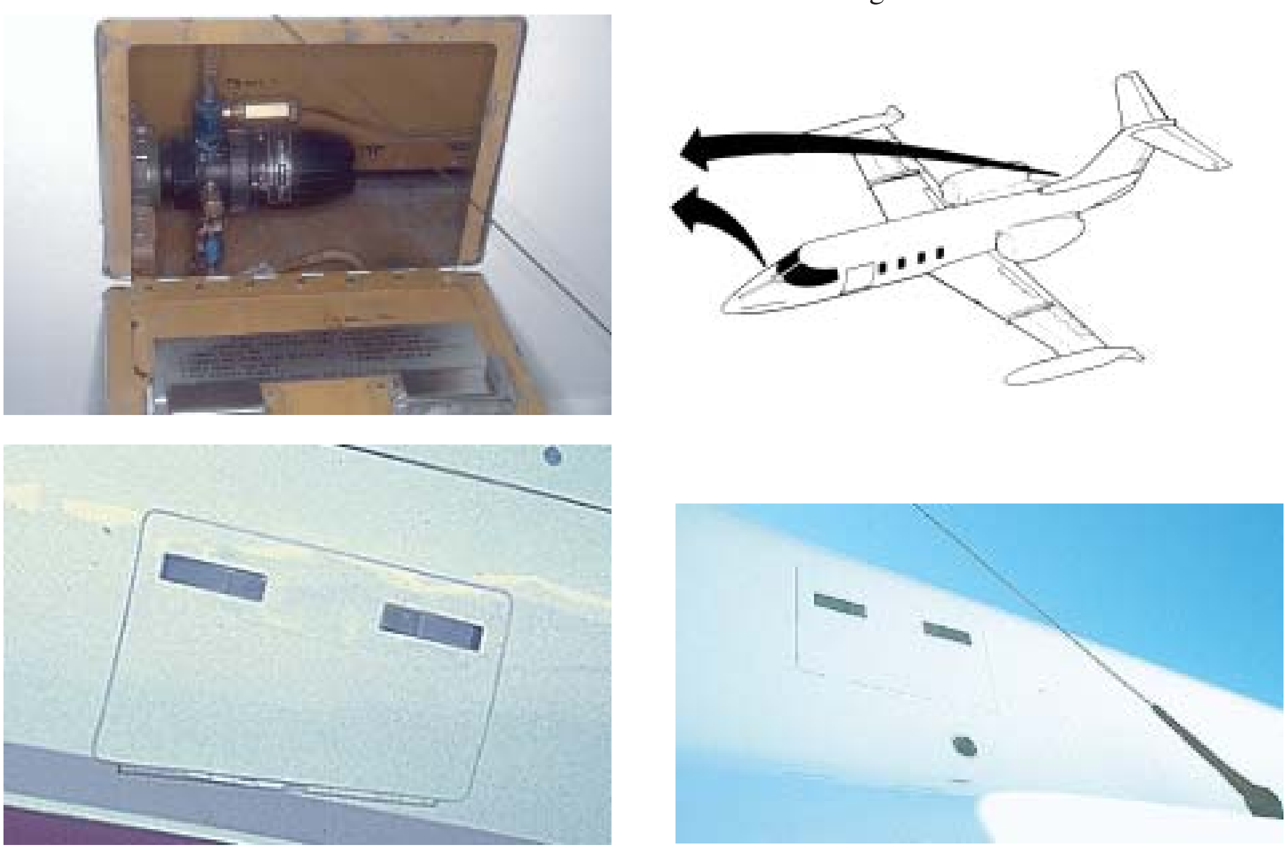


Figure 17-2. Oxygen Cylinder and Overboard Discharge Indicator

OXYGEN PRESSURE Gage

The OXYGEN PRESSURE gage (Figure 17-3) provides a direct reading of oxygen cylinder pressure, which is necessary to ensure that an adequate supply of oxygen is aboard. The gage is marked as follows:

- Yellow arc ....0–300 psi
- Green arc ....1,550–1,850 psi
- Red line ....2,000 psi

The gage is located on the pilot's side panel on late model airplanes; on early models it is mounted on the instrument panel.

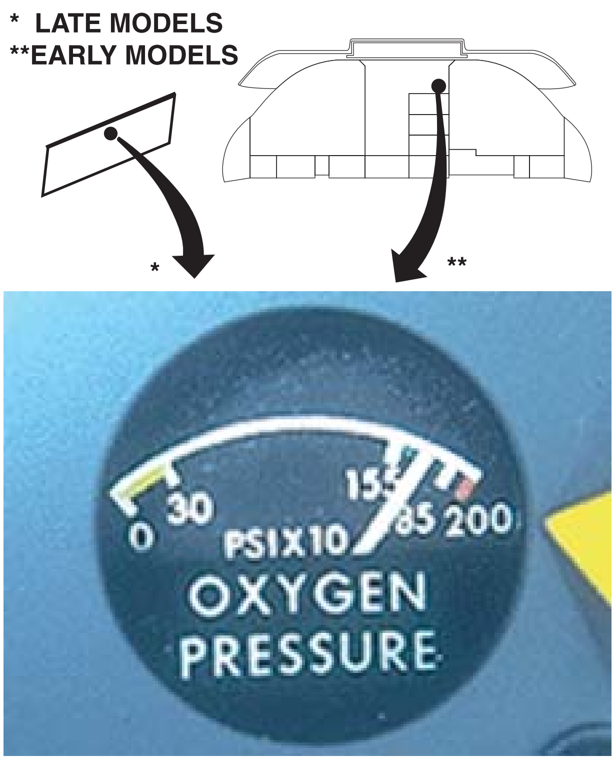


Figure 17-3. OXYGEN PRESSURE Gage

CREW DISTRIBUTION SYSTEM

The crew distribution system (see Figure 17-1) consists of the pilot's and copilot's oxygen masks with mask-mounted regulators for diluter-demand or 100% operation. Oxygen is available to the crew anytime the storage bottle shutoff valve is open and the mask are plugged in.

The crew masks (Figure 17-4) are stowed on the pilot's and copilot's sidewalls. The mask oxygen lines are connected to quick-disconnect receptacles located on the cockpit sidewalls. Optional oxygen-flow detectors may be installed in the mask oxygen lines.

NOTE

Headsets, eyeglasses, or hats worn by crewmembers may interfere with the quick-donning capabilities of the oxygen mask.

Figure 17-4. Crew Oxygen Mask

Three different oxygen mask/regulator configurations are available on the 35/36 model airplanes.

The ZMR 100 series diluter-demand mask regulator has a NORMAL-100% oxygen selector lever. With NORMAL selected, the regulator delivers diluted oxygen, on demand, up to 20,000 feet cabin altitude. Above 20,000 feet cabin altitude, the 100% oxygen position must be selected. With the selector in the 100% position, 100% oxygen is delivered at any cabin altitude. The 100% position should be used when smoke or fumes are present in the pressurized compartment.

The Robertshaw diluter-demand mask/regulator has two controls—the NORMAL-EMERGENCY selector and the 100% lever. With NORMAL selected, the regulator delivers diluted oxygen on demand, up to 30,000 feet cabin altitude. Above 30,000 feet, the regulator delivers 100% oxygen under a slight positive pressure. Depressing the 100% lever will deliver 100% oxygen at any time. With EMERGENCY selected (at any altitude) and the 100% lever depressed, the regulator delivers 100% oxygen and maintains a slight positive pressure for respiratory protection from smoke and fumes.

The Puritan-Bennett pressure demand mask/regulator incorporates a three-position selector knob labeled "NORM," "100%," and "EMER." With NORM selected, the regulator delivers diluted oxygen on demand, up to 33,000 feet cabin altitude. Above 33,000 feet, the regulator automatically delivers 100% oxygen. At 39,000 feet, it provides positive-pressure breathing. To obtain 100% oxygen at any time, 100% must be selected on the pressure-regulator control. With EMER selected, the regulator delivers 100% oxygen and maintains a slight positive pressure in the mask cup at all times for respiratory protection from smoke and fumes.

The Scott ATO MC 10-15-02 mask, in the normal pressure regulator position with the 100% lever extended, will deliver diluted oxygen up to 30,000 feet cabin altitude, 100% oxygen above 30,000 feet cabin altitude, and automatic pressure breathing above approximately 37,000 feet cabin altitude. To obtain 100% oxygen at any time, depress the 100% lever on the mask pressure regulator. With EMERGENCY selected, the mask will deliver 100% oxygen and maintain a positive pressure in the mask cup at all times for respiratory protection from smoke and fumes.

Each mask assembly includes a microphone and has an electrical cord which is plugged into the OXY-MIC jack on the respective OXY-MIC panel (Figure 17-5) located on each side panel. To operate the mask microphone, the OXY-MIC switch must be in the ON position and the microphone keyed, using the microphone switch on the outboard horn of the control wheel. Communication between crewmembers can be accomplished by using the INPH function of the audio control panel and increasing the MASTER VOL level.

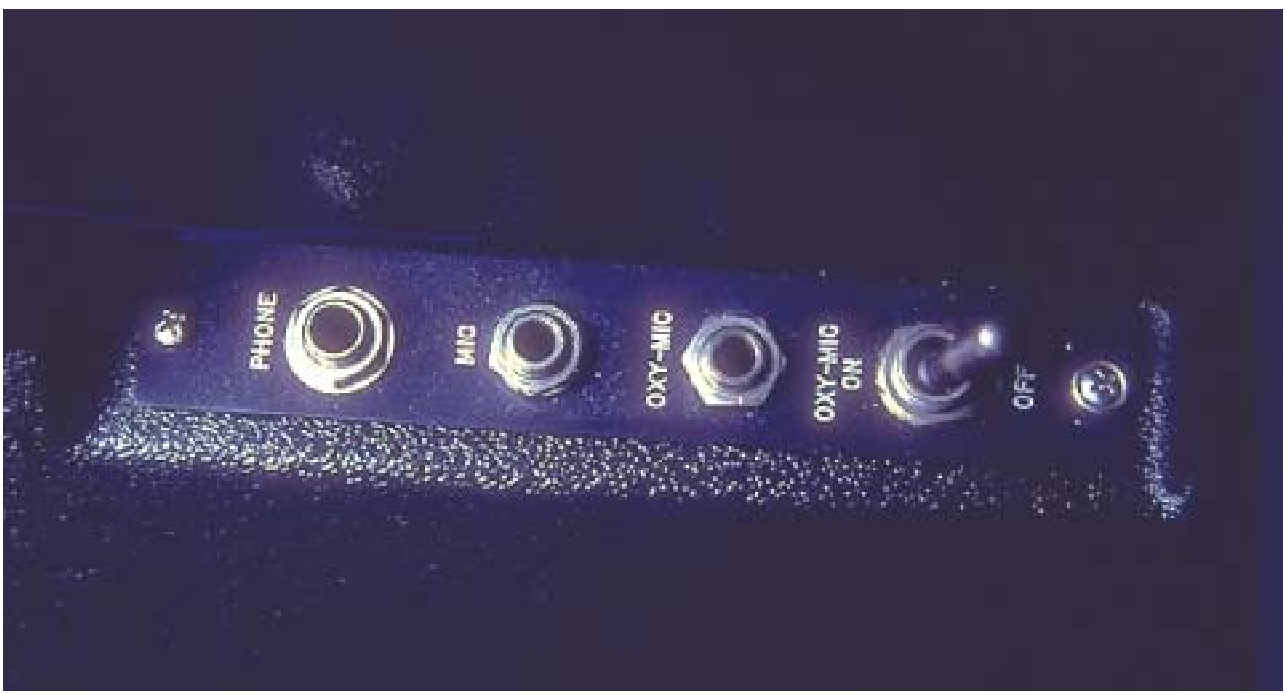


Figure 17-5. OXY-MIC Panel (Typical)

PASSENGER DISTRIBUTION SYSTEM

The passenger distribution system (Figure 17-6) is used to provide oxygen to the passengers in case of a pressurization system failure or any other time that oxygen is required. Oxygen is available in the crew oxygen distribution lines whenever the oxygen cylinder shutoff valve is open; however, oxygen is not available to the passenger distribution system until required.

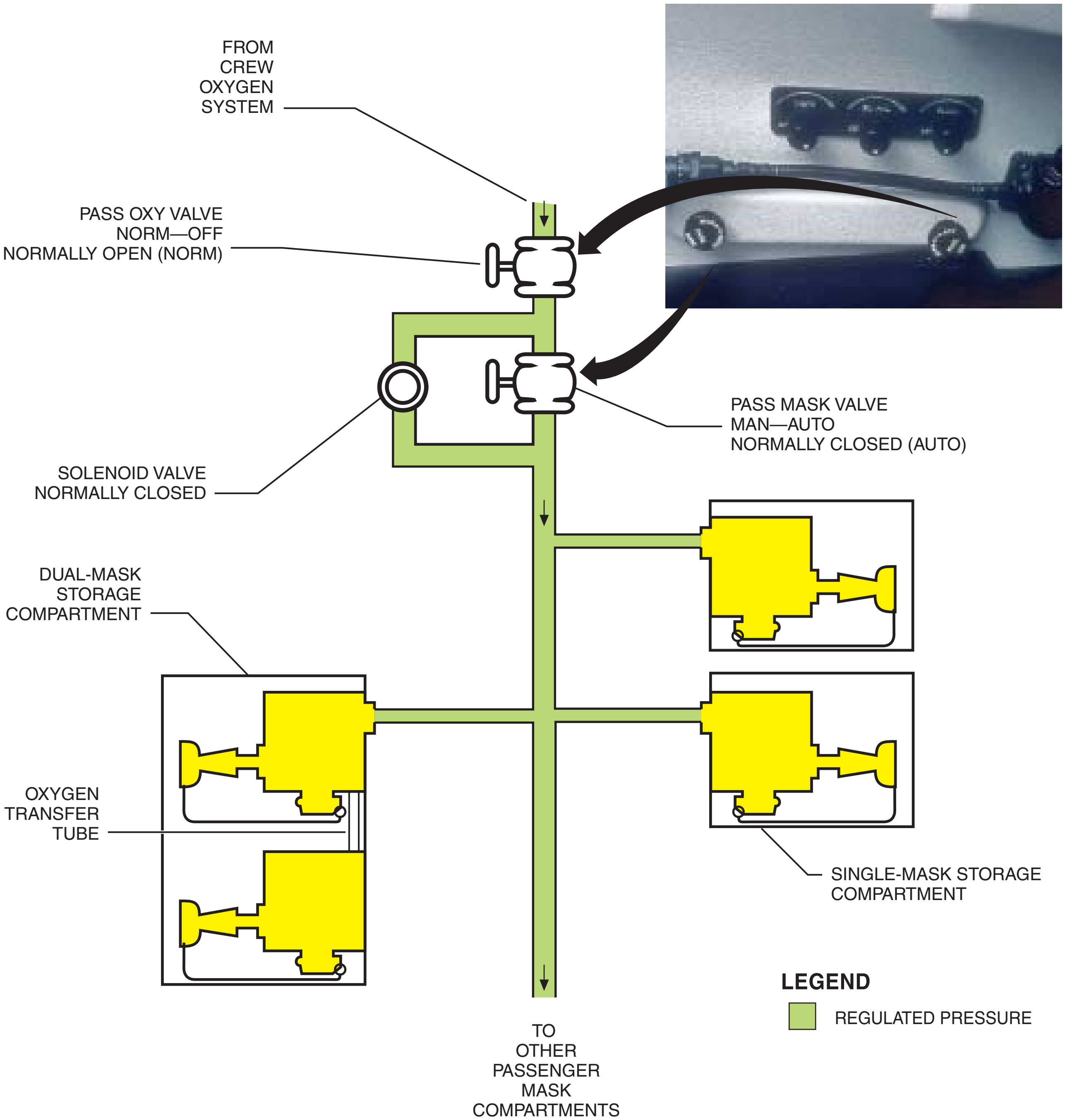


Figure 17-6. Passenger Distribution System

Oxygen supply to the passengers' system is controlled with three valves. Two valves are manually operated with control knobs on the pilot's sidewall, and the third is solenoid-operated by an aneroid switch. The manually controlled PASS OXY valve is normally in the NORM (open) position, which allows oxygen up and to the manually controlled PASS MASK valve and to the aneroid-controlled solenoid valve. Oxygen can be admitted to the passenger distribution system through either of these passenger mask valves, both of which are normally in the closed position.

With the PASS OXY valve is the OFF (closed) position, oxygen will not be available to the passenger distribution system in any event. This position may be used only when no passengers are being carried.

With the PASS OXY valve in the NORM (open) position, oxygen will be automatically admitted to the passenger distribution system through the aneroid-controlled solenoid valve if the cabin reaches 14,000 ±750 feet. The aneroid switch opens the solenoid valve and deploys the passenger masks. It also illuminates the cabin overhead lights.

In the event of airplane electrical failure, automatic deployment of the passenger masks is not possible. The oxygen solenoid valve requires DC power through the OXY VAL circuit breaker on the left essential bus for automatic mask deployment.

With the PASS OXY valve in the NORM (open) position, rotating the PASS MASK valve from AUTO to the MAN position admits oxygen into the passenger distribution system and causes the passenger oxygen masks to drop. This position can be used to deploy the passenger masks at any altitude, but ***will not*** cause the cabin overhead lights to illuminate.

The passenger oxygen masks (Figure 17-7) are stowed in compartments in the convenience panels above the passenger seats. The compartments may contain as many as three masks, depending on the airplane seating configuration. There will be at least one spare mask.

The passenger mask storage compartment doors are held closed by latches. When oxygen is admitted into the passenger distribution system, the oxygen pressure causes the door latches (plungers) to open each compartment door. When the doors open, the passenger

Figure 17-7. Passenger Mask

masks fall free and are available for passenger use. As the passenger pulls down on his mask to don it, an attached lanyard withdraws a pin from the supply valve which releases oxygen into the mask breather bag at a restricted, constant-flow rate. The rebreather bag may seem to inflate slowly, but this is normal. When inhaling, 100% oxygen is delivered to the mask cup. The breath is then exhausted into the rebreather bag.

Should the doors be inadvertently opened from the cockpit, oxygen pressure must be bled from the passenger distribution system before the masks can be restowed. This is accomplished by pulling one of the passenger mask lanyards after ensuring that the PASS MASK valve is closed (AUTO). If the doors open due to malfunction of the solenoid-operated valve, the PASS OXY valve must be turned off to permit stowage of the passenger masks.

The compartment doors can be opened manually for mask cleaning and servicing.

DRAG CHUTE

GENERAL

The optional drag chute may be used to shorten stopping distances. The greatest deceleration rate is produced at the highest speed; however, the chute is still effective at speeds below 60 knots. The chute is stored in a removable canister which is mounted inside the tailcone access door. The canister lid is released from the canister when the drag chute handle is pulled, allowing the pilot chute to deploy. The pilot chute then pulls the main chute canopy out of the canister.

The main chute riser attaches to the airplane at the chute control mechanism just forward of the canister (Figure 17-8). The loop at the end of the main riser slips over a recessed metal pin which is held in position by spring pressure when the drag chute handle is stowed. Therefore, if the chute should inadvertently deploy (handle in stowed position), the main chute riser will slip free of the pin and separate from the airplane. When the drag chute handle is pulled, the pin is mechanically locked

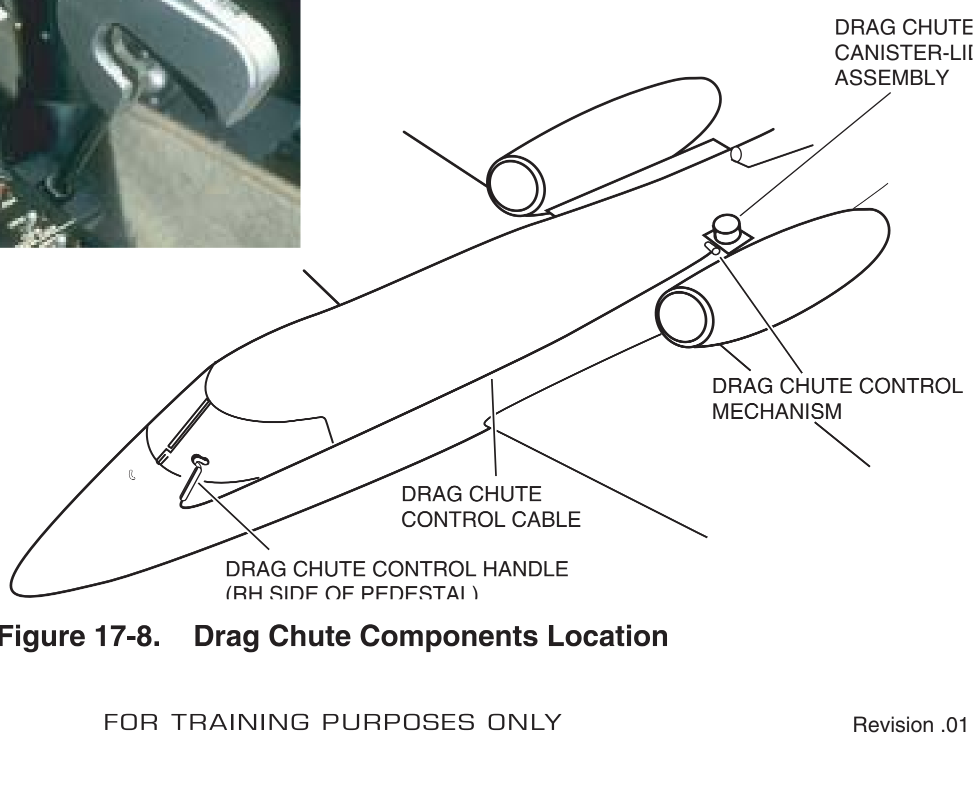


Figure 17-8. Drag Chute Components Location

in position to retain the chute riser while the mechanical canister control mechanism operates to release the canister lid, thereby deploying the chute.

The drag chute can be used:

- When landing on wet or icy runaway
- During any landing emergency involving no-flap hydraulic or brake failure, or loss of directional control
- During takeoff if the decision is made to abort

Do not deploy the drag chute under the following conditions:

- In flight
- If the nose gear is not on the ground
- When the indicated airspeed is above 150 knots
- With thrust reversers deployed

OPERATION

As the nosewheel touches down, the copilot, on the pilot's command, deploys the drag chute by squeezing the drag chute control handle (Figure 17-8) and pulling it up to its full extension (a pull force of approximately 50 pounds will be required). With the chute deployed, the pilot should keep the airplane well clear of the runway and taxiway lights, markers, and obstructions on the upwind side. Taxiing downwind should always be avoided.

The drag chute can be jettisoned after deployment at anytime. Normally, the pilot heads the airplane into the wind as much as possible to jettison the chute after the airplane clears the runway. The copilot jettisons the drag chute by squeezing the control handle grip safeties and pushing the handle down to the stowed position to release the chute. If the chute has collapsed prior to jettisoning, the chute riser must be pulled free after stowing the handle. Because the possibility always exists that jettisoning the chute might be required during the landing roll, any planned deployment should be coordinated with the control tower.

SQUAT SWITCH SYSTEM

GENERAL

Some airplane systems operate only on the ground while others operate only in the air. The squat switch system is designed to provide the necessary ground or airborne signals to these systems. The squat switch system consists of two squat switches, one on each main landing gear strut scissors, and a relay box located under the cabin floor. When the airplane is on the ground, and the main landing gear struts are compressed, the squat switches close to provide a ground mode signal. When the airplane lifts off the ground, and the main landing gear struts extend, the squat switches open, interrupting the ground mode signals, thereby shifting to air mode.

SQUAT SWITCHES

Each squat switch provides ground or air signals to the following components:

- Stall Warning System
    - The switches disable the stall warning test feature in the air.
    - The switches disable the stall warning rate sensor on the ground. The rate sensor remains disabled for approximately five seconds after lift-off.
    - The left squat switch controls the left stall warning system while the right squat switch controls the right stall warning system.
- Antiskid System
    - The switches disable the wheel brakes in the air with the antiskid system on.

The wheel brakes remain inoperative until the wheels' spinup requirements have been met on landing.

- The left squat switch controls the outboard wheel brakes while the right squat switch controls the inboard wheel brakes.

- Gear Control Valve
  - The switches disable the gear-up solenoid on the ground to prevent inadvertent landing gear retraction.
  - Either squat switch in ground mode will disable the gear-up solenoid. Both squat switches must be in the air mode to allow landing gear retraction.

- Squat Switch Relay Box
  - Either squat switch in the ground mode puts the relay box in ground mode.
  - Both squat switches must go to air mode to put the relay box in air mode.

The position of the SQUAT SW circuit breaker has no effect on landing gear, antiskid, or stall warning system operation. These systems receive signals directly from the squat switches as explained previously.

SQUAT SWITCH RELAY BOX

The squat switch relay box is necessary because of the limited number of electrical contacts available on the main landing gear squat switches. Sensing signals from both squat switches, the relay box provides ground or air mode signals to the components listed below. The squat switch relay box uses DC power from the SQUAT SW circuit breaker on the left main DC bus to provide ground mode signals. With the SQUAT SW circuit breaker open, all the relay box functions go to air mode.

The squat switch relay box provides ground or air mode signals to the following:

- Nosewheel steering—Disabled in the air
- Spoiler/spoileron system—Disables the monitor system on the ground. Slows the rate of spoiler deployment in the air.
- Cabin pressurization
- Safety valve vacuum solenoid closes in the air (SNs 35-099 and subsequent and 36-029 and subsequent only).
- Amber CAB ALT light (if installed) is disabled on the ground.
- Control module solenoids shift from ground to air mode.
- Amber TO TRIM light—Disabled in the air
- Windshield heat system—Shifts from ground to air mode. (See Chapter 10, "Ice and Rain Protection," for additional information.)
- Hourmeter and Davtron clock flight time function (if installed)—Disabled on the ground
- Mach trim test—Operates only on the ground
- Thrust reversers—Operates only on the ground
- Generator load limiting—Limits the output of a single, engine-driven generator on the ground only (SNs 35-148 and subsequent and 36-036 and subsequent only).
- Air data unit—TAS disabled on the ground (AFCS 530 autopilot airplanes only).
- Mach overspeed warning/stick puller—Test function disabled in the air (AFCS 530 airplanes only).
- Yaw damper—Disconnects at touchdown (AFCS 530 airplanes only).

QUESTIONS

1. During preflight the pilot can determine if the oxygen bottle is turned on by:

A. Reading the pressure indicated on the oxygen pressure gage in the cockpit
B. selecting 100% on the mask regulator and taking several deep breaths through the mask
C. Placing the OXY-MIC switch to the OXY position
D. Visually checking for the green flow indicator on the mask supply hose

2. With the PASS OXY valve in the NORM position, selecting MAN on the PASS MASK valve:

A. Causes passenger masks to drop and turns on the cabin overhead lights
B. Prevents oxygen from entering the passenger oxygen distribution lines
C. Disarms the 14,000-foot cabin aneroid
D. Admits oxygen to the passenger distribution lines and causes the passenger oxygen masks to drop

3. With the PASS OXY valve in the NORM position and the PASS MASK valve in the AUTO position:

A. Oxygen is supplied to the passenger masks if the cabin altitude reaches 10,000 feet.
B. Passenger masks will automatically deploy in the event of electrical failure.
C. Passenger masks will automatically deploy and the cabin overhead lights will illuminate if cabin altitude reaches 14,000 feet.
D. The aneroid-controlled passenger mask drop valve is disabled.

4. The OXY PRESS gage reads:

A. Direct pressure of the cylinder
B. Electrically derived system high pressure
C. Direct pressure of the pilot's supply line
D. Electrically derived system low pressure

5. The maximum demonstrated crosswind component for drag chute deployment is:

A. 10 knots
B. 15 knots
C. 20 knots
D. 25 knots

6. The drag chute is deployed by:

A. Squeezing the control handle
B. Rotating the control handle fully clockwise and pulling it up to its full extension
C. Squeezing the control handle and pulling it up to its full extension
D. Squeezing the control handle and pushing it completely forward

7. The maximum indicated airspeed for drag chute deployment is:

A. 120 knots
B. 130 knots
C. 140 knots
D. 150 knots

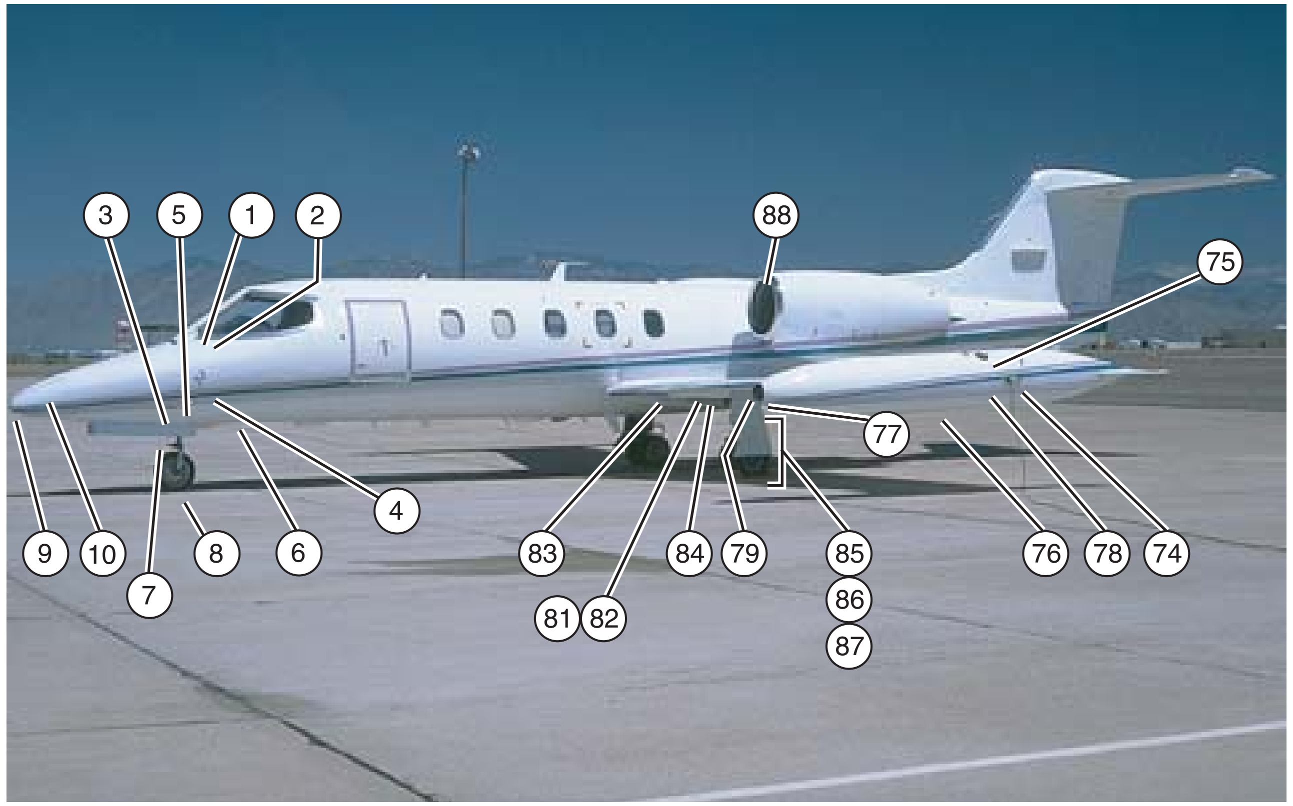
8. If either main landing gear squat switch remains in ground mode after takeoff:

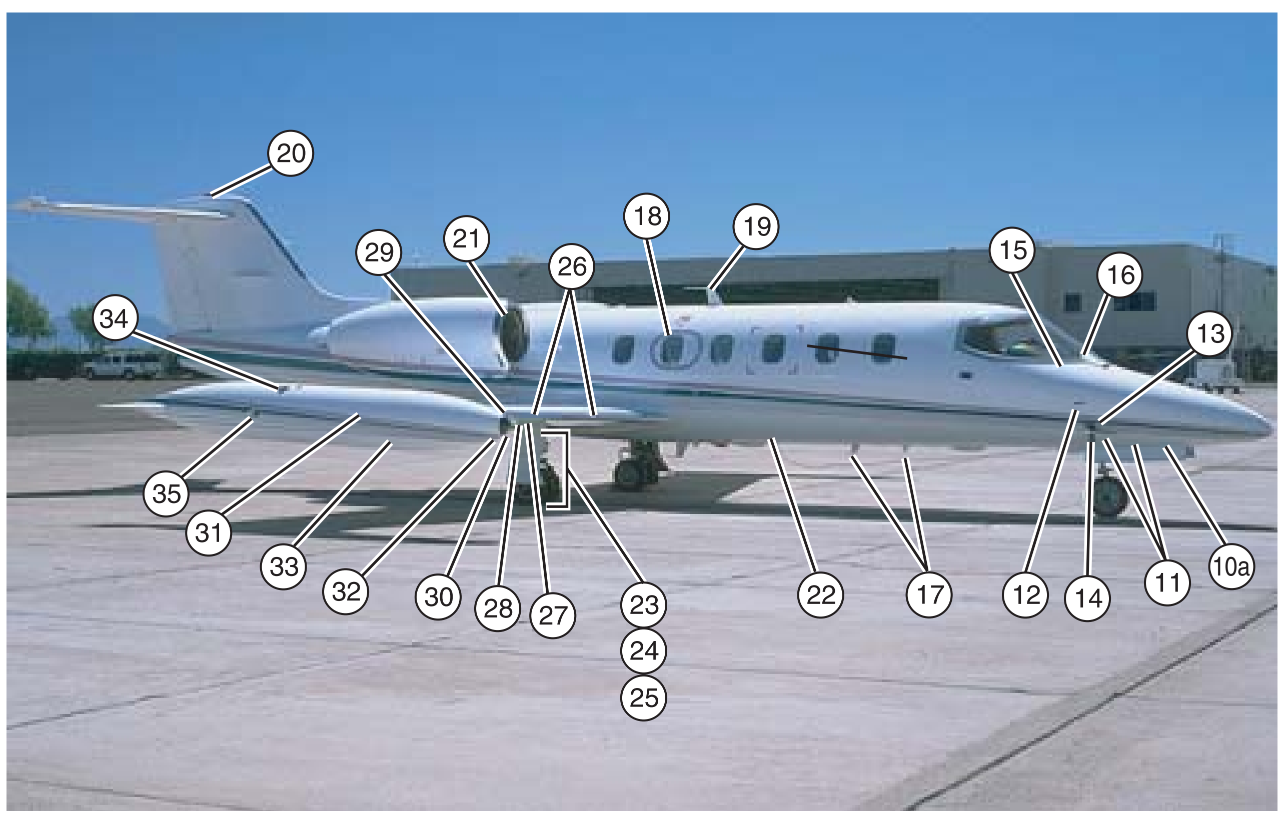
A. The landing gear will not retract.
B. The airplane will not pressurize.
C. The amber TO TRIM light may illuminate.
D. All of the above

WALKAROUND

The following section is a pictorial walkaround. It shows each item called out in the exterior power-off preflight inspection. The fold-out pages, WA-2 and WA-15, 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.





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LEARJET 30 Series PILOT TRAINING MANUAL

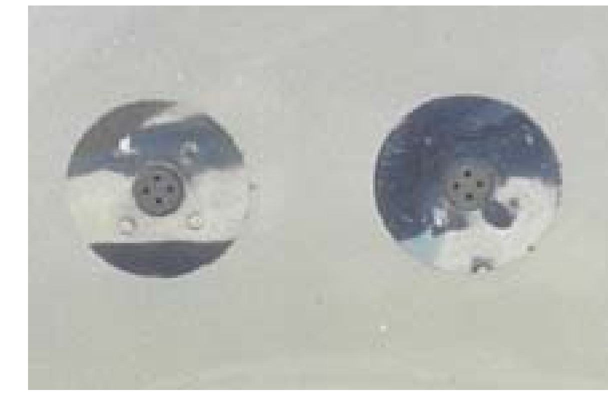
WALKAROUND INSPECTION

1. **PILOT'S WINDSHIELD ALCOHOL DISCHARGE OUTLETS AND PILOT'S DEFOG OUTLET**—CLEAR OF OBSTRUCTIONS
2. **LEFT SHOULDER STATIC PORT (FC 200)** —CLEAR OF OBSTRUCTIONS

3. **LEFT PITOT HEAD (FC 200)**—COVER REMOVED, CLEAR OF OBSTRUCTIONS

3a. **LEFT PITOT-STATIC PROBE (FC 530)**—COVER REMOVED, CLEAR OF OBSTRUCTIONS

4. **LEFT STALL WARNING VANE**—FREEDOM OF MOVEMENT, LEAVE IN DOWN POSITION



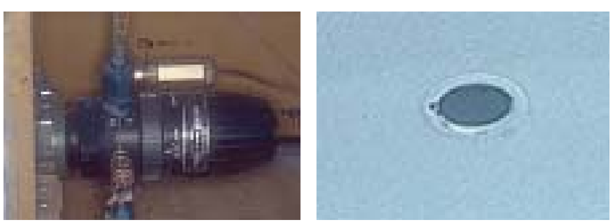
5. **LEFT STATIC PORTS (2) (FC 200)**—CLEAR OF OBSTRUCTIONS

6. **SHOULDER STATIC (1) (FC 200) AND LEFT PITOT-STATIC (2) DRAIN VALVES**—DRAIN

7. **NOSE GEAR AND WHEEL WELL**—HYDRAULIC LEAKAGE AND CONDITION

8. **NOSEWHEEL AND TIRE**—CONDITION AND NOSE GEAR UPLOCK FORWARD

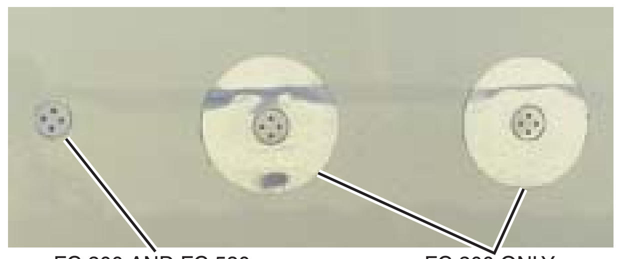
9. **RADOME ALCOHOL DISCHARGE PORT**—CLEAR OF OBSTRUCTION
10. **RADOME AND RADOME EROSION SHOE**—CONDITION



10a. **OXYGEN BOTTLE SUPPLY VALVE**—ON
OXYGEN PRESSURE RELIEF DISC—INTACT

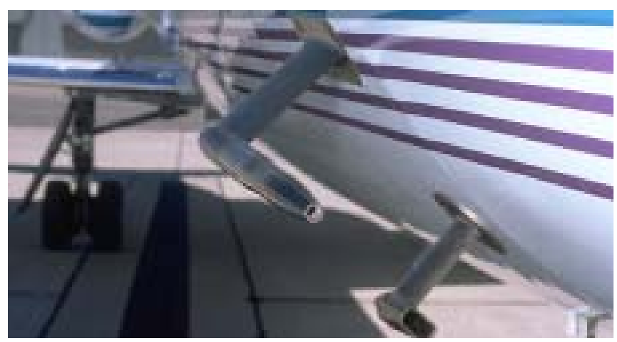
11. **RIGHT PITOT HEAD (FC 200) AND TEMPERATURE PROBE**—COVERS REMOVED, CLEAR OF OBSTRUCTIONS

12. **RIGHT STALL WARNING VANE**—FREEDOM OF MOVEMENT, LEAVE IN DOWN POSITION



FC 200 AND FC 530 FC 200 ONLY

13. **RIGHT STATIC PORTS FC 200 (3) OR FC 530 (1)**—CLEAR OF OBSTRUCTIONS



13a. **RIGHT PITOT-STATIC PROBE AND TEMPERATURE PROBE (FC 530)**—COVER REMOVED, CLEAR OF OBSTRUCTIONS

14. **RIGHT PITOT-STATIC DRAIN VALVES (2)**—DRAIN

15. **RIGHT SHOULDER STATIC PORT**—CLEAR OF OBSTRUCTIONS (FC0-200)
16. **COPILOT'S WINDSHIELD DEFOG OUTLET**—CLEAR OF OBSTRUCTIONS

17. **LOWER FUSELAGE ANTENNAE, ROTATING BEACON LIGHT AND LENS**—CONDITION

18. **EMERGENCY EXIT**—SECURE
19. **UPPER FUSELAGE ANTENNAE**—CONDITION

20. **ROTATING BEACON LIGHT AND LENS (ON VERTICAL FIN)**—CONDITION

21. **RIGHT ENGINE INLET AND FAN**—CLEAR OF OBSTRUCTIONS AND CONDITION

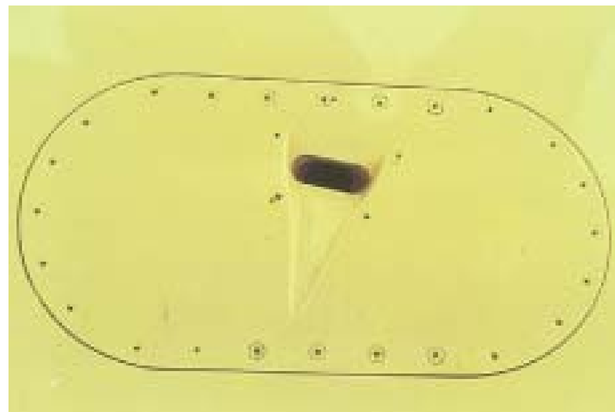
22. **FUEL CROSSOVER, LEFT WING SUMP, LEFT ENGINE FUEL, RIGHT WING SUMP, AND RIGHT ENGINE FUEL DRAIN VALVES**—DRAIN

23. **RIGHT MAIN GEAR AND WHEEL WELL**—HYDRAULIC/FUEL LEAKAGE AND CONDITION

24. **RIGHT MAIN GEAR LANDING LIGHT**—CONDITION
25. **RIGHT MAIN GEAR WHEELS, BRAKES, AND TIRES**—CONDITION

26. **STALL STRIP, WING LEADING EDGE, AND STALL FENCE**—CONDITION

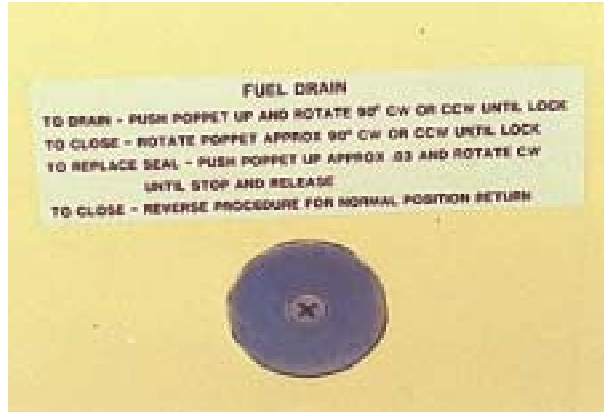
27. **RIGHT WING ACCESS PANELS (UNDERSIDE OF WING)**—CHECK FOR FUEL LEAKAGE



28. **RIGHT FUEL VENT (UNDERSIDE OF WING)**—PLUG REMOVED, CLEAR OF OBSTRUCTIONS

29. **VORTEX GENERATORS OR BOUNDARY LAYER ENERGIZERS**—CONDITION

30. **RIGHT WING HEAT SCUPPER (UNDERSIDE OF WING FORWARD)**—CLEAR OF OBSTRUCTIONS



33. **RIGHT TIP TANK SUMP DRAIN VALVE**—DRAIN

31. **RIGHT TIP TANK**—CONDITION

34. **RIGHT TIP TANK FUEL CAP**—CONDITION AND SECURE
35. **RIGHT TIP TANK NAVIGATION LIGHT, STROBE LIGHT, AND LENS**—CONDITION

32. **RIGHT TIP TANK RECOGNITION LIGHT AND LENS**—CONDITION

36. **RIGHT TIP TANK FIN AND STATIC DISCHARGE WICKS (2)**—CONDITION

37. RIGHT TIP TANK FUEL JETTISON TUBE—CLEAR OF OBSTRUCTIONS

38. SCUPPER (UNDERSIDE OF RIGHT WING AFT)—CLEAR OF OBSTRUCTIONS, NO FUEL LEAKAGE

39. RIGHT AILERON—CHECK FREE MOTION, BALANCE TAB LINKAGE, BRUSH SEAL CONDITION

40. RIGHT SPOILER AND FLAP—CONDITION

41. RIGHT ENGINE OIL QUANTITY—CHECK
FILLER CAP AND ACCESS DOOR—SECURE

42. RIGHT ENGINE OIL BYPASS VALVE INDICATOR—CHECK, NOT EXTENDED

43. **RIGHT ENGINE THRUST REVERSER**—CONDITION AND STOWED (AERONCA)

43A. **RIGHT ENGINE THRUST REVERSER**—CONDITION AND STOWED (DEE HOWARD)

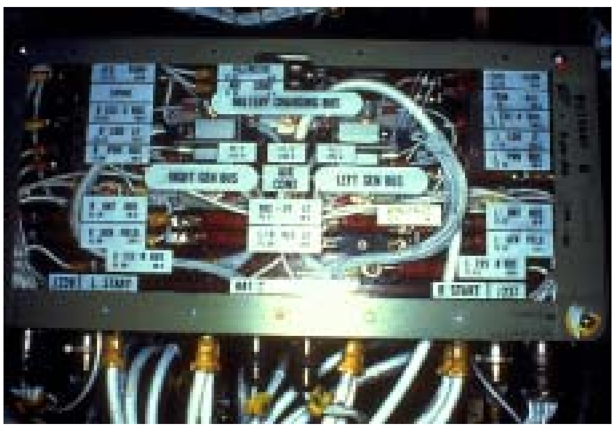
44. **RIGHT ENGINE TURBINE EXHAUST AREA**—CONDITION, CLEAR OF OBSTRUCTION, BLOCKER DOORS STOWED (AERONCA)

45. **RIGHT ENGINE FUEL BYPASS VALVE INDICATOR**—CHECK, NOT EXTENDED

46. **FUEL VENT DRAIN VALVE, TRANSFER LINE DRAIN VALVE, FUSELAGE TANK SUMP DRAIN VALVE**—DRAIN

47. **LEFT AND RIGHT FUEL FILTER DRAIN VALVES**—DRAIN

48. TAILCONE ACCESS DOOR—OPEN



49. TAILCONE INTERIOR—CHECK FOR FLUID LEAKS, SECURITY, AND CONDITION OF INSTALLED EQUIPMENT

49A. TAILCONE INTERIOR—CHECK FOR FLUID LEAKS, SECURITY, AND CONDITION OF INSTALLED EQUIPMENT
HYDRAULIC ACCUMULATOR AIR CHARGE—750 PSI MINIMUM

49B. TAILCONE INTERIOR—CHECK FOR FLUID LEAKS, SECURITY, AND CONDITION OF INSTALLED EQUIPMENT

50. DRAG CHUTE—CHECK FOR PROPER INSTALLATION

50A. DRAG CHUTE—CHECK FOR PROPER INSTALLATION

51. **TAILCONE ACCESS DOOR**—CLOSED AND SECURE

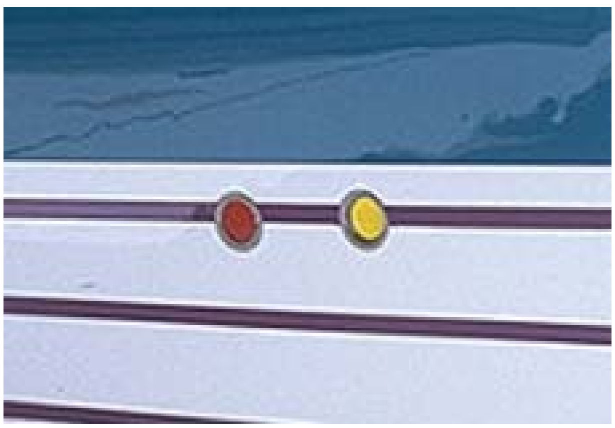
55. **RIGHT FUEL COMPUTER DRAIN VALVE**—DRAIN (DRAIN VALVES ARE RECESSED ON AIRPLANES EQUIPPED WITH DRAG CHUTE.)

52. **OXYGEN BOTTLE SUPPLY VALVE**—OPEN

56. **RIGHT VOR/LOC ANTENNA**—CONDITION
57. **VERTICAL STABILIZER, RUDDER, HORIZONTAL STABILIZER, AND ELEVATOR**—CONDITION, DRAIN HOLES CLEAR
58. **STATIC DISCHARGE WICKS (6 ON HORIZONTAL STABILIZER, 1 ABOVE NAV LIGHT, 1 ON VERTICAL FIN)**—CONDITION
59. **VERTICAL FIN NAVIGATION LIGHTS, STROBE LIGHT AND LENS**—CONDITION
60. **VLF H-FIELD ANTENNA**—CONDITION
61. **LEFT VOR/LOC ANTENNA**—CONDITION

53. **OXYGEN SERVICING DOOR**—SECURE
54. **OXYGEN DISCHARGE DISC**—CONDITION

62. LEFT FUEL COMPUTER DRAIN VALVE—DRAIN (DRAIN VALVES ARE RECESSED ON AIRPLANES EQUIPPED WITH DRAG CHUTE.)



63. FIRE EXTINGUISHER DISCS—CONDITION

64. LEFT ENGINE OIL BYPASS VALVE INDICATOR—CHECK, NOT EXTENDED

65A. LEFT ENGINE THRUST REVERSER—CONDITION AND STOWED (DEE HOWARD)

65. LEFT ENGINE TURBINE EXHAUST AREA—CONDITION, CLEAR OF OBSTRUCTIONS AND BLOCKER DOORS STOWED (AERONCA)

66. LEFT ENGINE TRUST REVERSER —CONDITION AND STOWED (AERONCA)

67. LEFT ENGINE FUEL BYPASS VALVE INDICATOR—CHECK, NOT EXTENDED

68. LEFT ENGINE OIL QUANTITY—CHECK
FILLER CAP AND ACCESS DOOR—SECURE

69. LEFT SPOILER AND FLAP—CONDITION

70. LEFT AILERON—CHECK FREE MOTION, BALANCE, AND TRIM LINKAGE, AND BRUSH SEAL CONDITION

71. SCUPPER (UNDERSIDE OF LEFT WING AFT)—CLEAR OF OBSTRUCTIONS, NO FUEL LEAK

72. LEFT TIP TANK FUEL JETTISON TUBE—CLEAR OF OBSTRUCTIONS

73. LEFT TIP TANK FIN AND STATIC DISCHARGE WICKS (2)—CONDITION

74. LEFT TIP TANK NAVIGATION LIGHT, STROBE LIGHT AND LENS—CONDITION

75. LEFT TIP TANK CAP—CONDITION AND SECURE

76. LEFT TIP TANK SUMP DRAIN VALVE—DRAIN

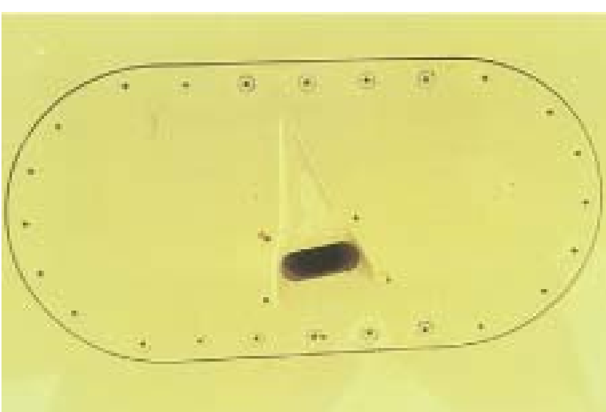
77. LEFT TIP TANK RECOGNITION LIGHT AND LENS (IF INSTALLED)—CONDITION

78. LEFT TIP TANK—CONDITION

79. LEFT WING HEAT SCUPPER (UNDERSIDE OF WING FORWARD)—CLEAR OF OBSTRUCTIONS

80. **VORTEX GENERATORS OR BOUNDARY LAYER ENERGIZERS**—CONDITION

81. **LEFT WING ACCESS PANELS (UNDERSIDE OF WING)**—CHECK FOR FUEL LEAKAGE



82. **LEFT FUEL VENT (UNDERSIDE OF WING)**—PLUG REMOVED, CLEAR OF OBSTRUCTIONS

83. **STALL STRIP (IF INSTALLED) AND WING LEADING EDGE**—CONDITION

84. **STALL FENCE (IF INSTALLED)**—CONDITION

85. **LEFT MAIN GEAR AND WHEEL WELL**—HYDRAULIC/FUEL LEAKAGE AND CONDITION

86. LEFT MAIN GEAR LANDING LIGHT—CONDITION

87. LEFT MAIN GEAR WHEELS, BRAKES, AND TIRES—CONDITION

88. LEFT ENGINE INLET AND FAN—CLEAR OF OBSTRUCTIONS AND CONDITION

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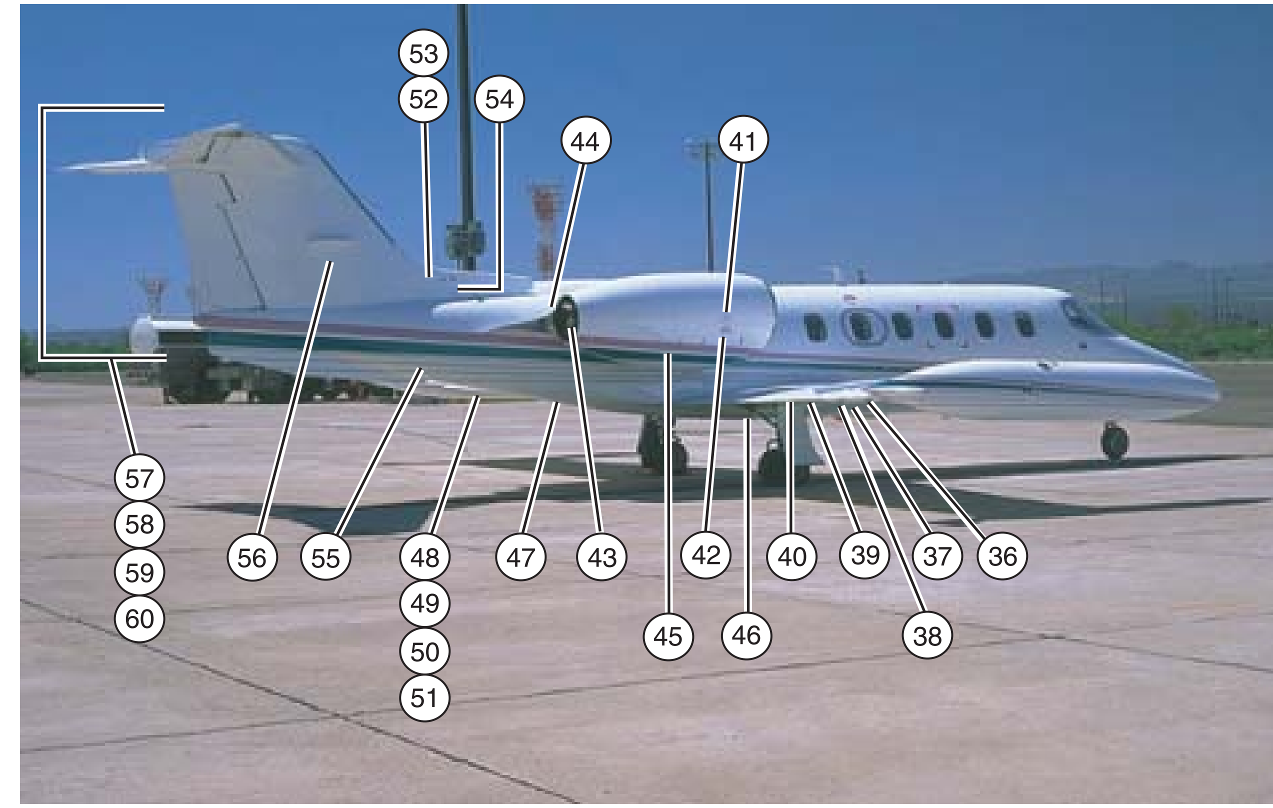

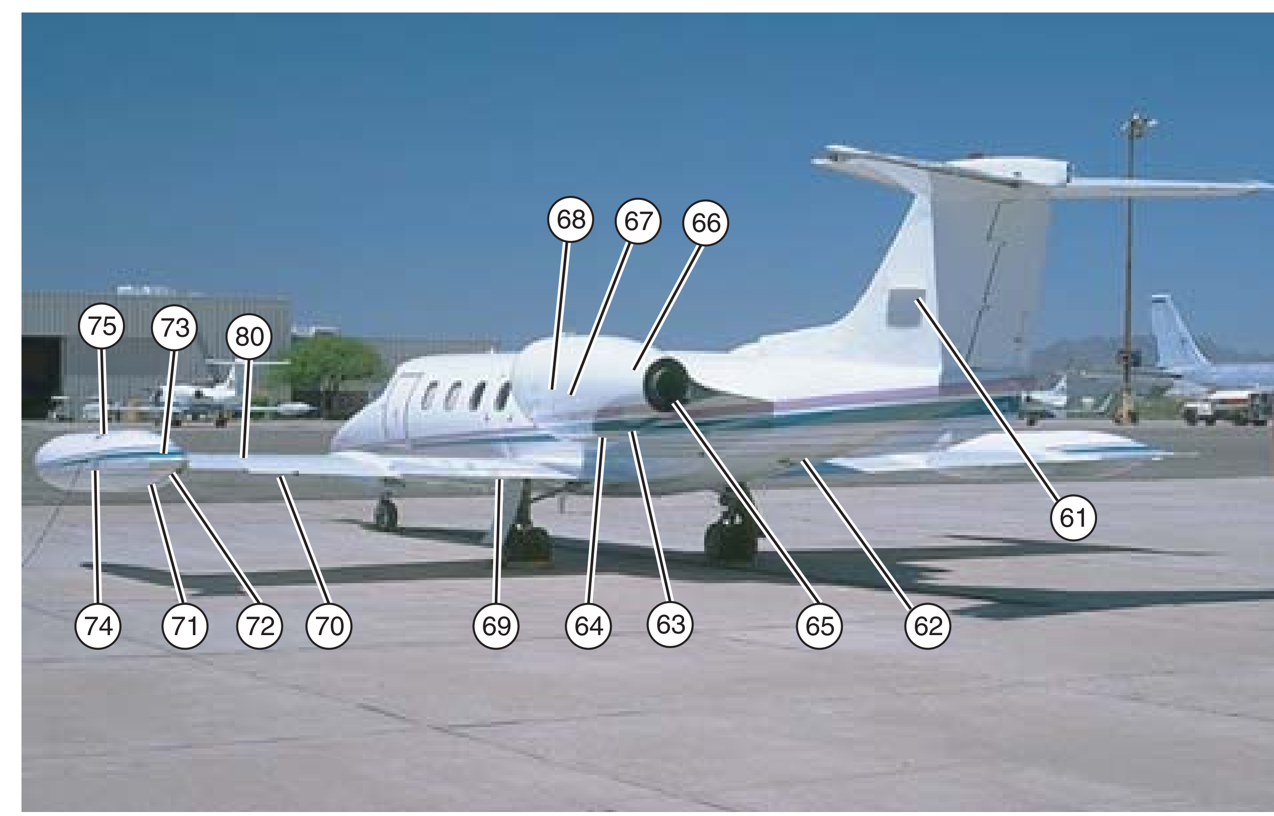

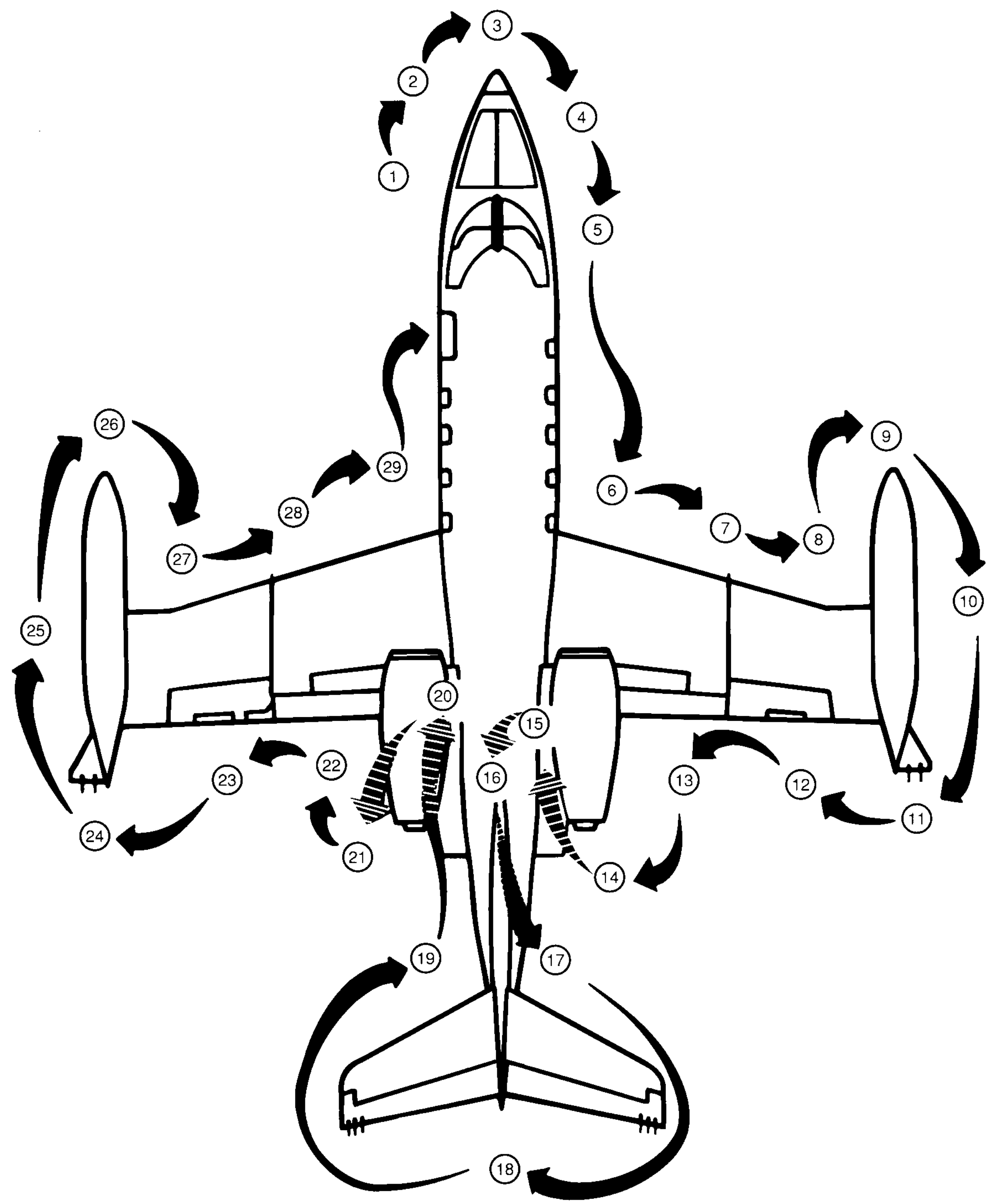

NOTE:
THE NUMBERS ON THIS
DIAGRAM CORRESPOND TO
THE PREFLIGHT POSITIONS
DEPICTED IN THE *AIRPLANE FLIGHT MANUAL.*

This appendix contains the following conversion tables:

Table APP-1. CONVERSION FACTORS

MULTIPLY	BY	TO OBTAIN
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

Table APP-2. FARENHEIT AND CELSIUS CONVERSION

−459.4 to −220

C.	C. or F.	F.
−273	**−459.4**	
−268	**−450**	
−262	**−440**	
−257	**−430**	
−251	**−420**	
−246	**−410**	
−240	**−400**	
−234	**−390**	
−229	**−380**	
−223	**−370**	
−218	**−360**	
−212	**−350**	
−207	**−340**	
−201	**−330**	
−196	**−320**	
−190	**−310**	
−184	**−300**	
−179	**−290**	
−173	**−280**	
−169	**−273**	−459.4
−168	**−270**	−454
−163	**−260**	−436
−157	**−250**	−418
−151	**−240**	−400
−146	**−230**	−382
−140	**−220**	−364

−210 to 0

C.	C. or F.	F.
−134	**−210**	−346
−129	**−200**	−328
−123	**−190**	−310
−118	**−180**	−292
−112	**−170**	−274
−107	**−160**	−256
−101	**−150**	−238
− 95.6	**−140**	−220
− 90.0	**−130**	−202
− 84.4	**−120**	−184
− 78.9	**−110**	−166
− 73.3	**−100**	−148
− 67.8	**− 90**	−130
− 62.2	**− 80**	−112
− 56.7	**− 70**	− 94
− 51.1	**− 60**	− 76
− 45.6	**− 50**	− 58
− 40.0	**− 40**	− 40
− 34.4	**− 30**	− 22
− 28.9	**− 20**	− 4
− 23.3	**− 10**	14
− 17.8	**0**	32

1 to 25

C.	C. or F.	F.
−17.2	**1**	33.8
−16.7	**2**	35.6
−16.1	**3**	37.4
−15.6	**4**	39.2
−15.0	**5**	41.0
−14.4	**6**	42.8
−13.9	**7**	44.6
−13.3	**8**	46.4
−12.8	**9**	48.2
−12.2	**10**	50.0
−11.7	**11**	51.8
−11.1	**12**	53.6
−10.6	**13**	55.4
−10.0	**14**	57.2
− 9.44	**15**	59.0
− 8.89	**16**	60.8
− 8.33	**17**	62.6
− 7.78	**18**	64.4
− 7.22	**19**	66.2
− 6.67	**20**	68.0
− 6.11	**21**	69.8
− 5.56	**22**	71.6
− 5.00	**23**	73.4
− 4.44	**24**	75.2
− 3.89	**25**	77.0

26 to 50

C.	C. or F.	F.
− 3.33	**26**	78.8
− 2.78	**27**	80.6
− 2.22	**28**	82.4
− 1.67	**29**	84.2
− 1.11	**30**	86.0
− 0.56	**31**	87.8
0	**32**	89.6
0.56	**33**	91.4
1.11	**34**	93.2
1.67	**35**	95.0
2.22	**36**	96.8
2.78	**37**	98.6
3.33	**38**	100.4
3.89	**39**	102.2
4.44	**40**	104.0
5.00	**41**	105.8
5.56	**42**	107.6
6.11	**43**	109.4
6.67	**44**	111.2
7.22	**45**	113.0
7.78	**46**	114.8
8.33	**47**	116.6
8.89	**48**	118.4
9.44	**49**	120.2
10.0	**50**	122.0

51 to 75

C.	C. or F.	F.
10.6	**51**	123.8
11.1	**52**	125.6
11.7	**53**	127.4
12.2	**54**	129.2
12.8	**55**	131.0
13.3	**56**	132.8
13.9	**57**	134.6
14.4	**58**	136.4
15.0	**59**	138.2
15.6	**60**	140.0
16.1	**61**	141.8
16.7	**62**	143.6
17.2	**63**	145.4
17.8	**64**	147.2
18.3	**65**	149.0
18.9	**66**	150.8
19.4	**67**	152.6
20.0	**68**	154.4
20.6	**69**	156.2
21.1	**70**	158.0
21.7	**71**	159.8
22.2	**72**	161.6
22.8	**73**	163.4
23.3	**74**	165.2
23.9	**75**	167.0

76 to 100

C.	C. or F.	F.
24.4	**76**	168.8
25.0	**77**	170.6
25.6	**78**	172.4
26.1	**79**	174.2
26.7	**80**	176.0
27.2	**81**	177.8
27.8	**82**	179.6
28.3	**83**	181.4
28.9	**84**	183.2
29.4	**85**	185.0
30.0	**86**	186.8
30.6	**87**	188.6
31.1	**88**	190.4
31.7	**89**	192.2
32.2	**90**	194.0
32.8	**91**	195.8
33.3	**92**	197.6
33.9	**93**	199.4
34.4	**94**	201.2
35.0	**95**	203.0
35.6	**96**	204.8
36.1	**97**	206.6
36.7	**98**	208.4
37.2	**99**	210.2
37.8	**100**	212.0

101 to 340

C.	C. or F.	F.
43	**110**	230
49	**120**	248
54	**130**	266
60	**140**	284
66	**150**	302
71	**160**	320
77	**170**	338
82	**180**	356
88	**190**	374
93	**200**	392
99	**210**	410
100	**212**	413
104	**220**	428
110	**230**	446
116	**240**	464
121	**250**	482
127	**260**	500
132	**270**	518
138	**280**	536
143	**290**	554
149	**300**	572
154	**310**	590
160	**320**	608
166	**330**	626
171	**340**	644

341 to 490

C.	C. or F.	F.
177	**350**	662
182	**360**	680
188	**370**	698
193	**380**	716
199	**390**	734
204	**400**	752
210	**410**	770
216	**420**	788
221	**430**	806
227	**440**	824
232	**450**	842
238	**460**	860
243	**470**	878
249	**480**	896
254	**490**	914

491 to 750

C.	C. or F.	F.
260	**500**	932
266	**510**	950
271	**520**	968
277	**530**	986
282	**540**	1004
288	**550**	1022
293	**560**	1040
299	**570**	1058
304	**580**	1076
310	**590**	1094
316	**600**	1112
321	**610**	1130
327	**620**	1148
332	**630**	1166
338	**640**	1184
343	**650**	1202
349	**660**	1220
354	**670**	1238
360	**680**	1256
366	**690**	1274
371	**700**	1292
377	**710**	1310
382	**720**	1328
388	**730**	1346
393	**740**	1364
399	**750**	1382

751 to 1000

C.	C. or F.	F.
404	**760**	1400
410	**770**	1418
416	**780**	1436
421	**790**	1454
427	**800**	1472
432	**810**	1490
438	**820**	1508
443	**830**	1526
449	**840**	1544
454	**850**	1562
460	**860**	1580
466	**870**	1598
471	**880**	1616
477	**890**	1634
482	**900**	1652
488	**910**	1670
493	**920**	1688
499	**930**	1706
504	**940**	1724
510	**950**	1742
516	**960**	1760
521	**970**	1778
527	**980**	1796
532	**990**	1814
538	**1000**	1832

1001 to 1250

C.	C. or F.	F.
543	**1010**	1850
549	**1020**	1868
554	**1030**	1886
560	**1040**	1904
566	**1050**	1922
571	**1060**	1940
577	**1070**	1958
582	**1080**	1976
588	**1090**	1994
593	**1100**	2012
599	**1110**	2030
604	**1120**	2048
610	**1130**	2066
616	**1140**	2084
621	**1150**	2102
627	**1160**	2120
632	**1170**	2138
638	**1180**	2156
643	**1190**	2174
649	**1200**	2192
654	**1210**	2210
660	**1220**	2228
666	**1230**	2246
671	**1240**	2264
677	**1250**	2282

1251 to 1490

C.	C. or F.	F.
682	**1260**	2300
688	**1270**	2318
693	**1280**	2336
699	**1290**	2354
704	**1300**	2372
710	**1310**	2390
716	**1320**	2408
721	**1330**	2426
727	**1340**	2444
732	**1350**	2462
738	**1360**	2480
743	**1370**	2498
749	**1380**	2516
754	**1390**	2534
760	**1400**	2552
766	**1410**	2570
771	**1420**	2588
777	**1430**	2606
782	**1440**	2624
788	**1450**	2642
793	**1460**	2660
799	**1470**	2678
804	**1480**	2696
810	**1490**	2714

1491 to 1750

C.	C. or F.	F.
816	**1500**	2732
821	**1510**	2750
827	**1520**	2768
832	**1530**	2786
838	**1540**	2804
843	**1550**	2822
849	**1560**	2840
854	**1570**	2858
860	**1580**	2876
866	**1590**	2894
871	**1600**	2912
877	**1610**	2930
882	**1620**	2948
888	**1630**	2966
893	**1640**	2984
899	**1650**	3002
904	**1660**	3020
910	**1670**	3038
916	**1680**	3056
921	**1690**	3074
927	**1700**	3092
932	**1710**	3110
938	**1720**	3128
943	**1730**	3146
949	**1740**	3164
954	**1750**	3182

1751 to 2000

C.	C. or F.	F.
960	**1760**	3200
966	**1770**	3218
971	**1780**	3236
977	**1790**	3254
982	**1800**	3272
988	**1810**	3290
993	**1820**	3308
999	**1830**	3326
1004	**1840**	3344
1010	**1850**	3362
1016	**1860**	3380
1021	**1870**	3398
1027	**1880**	3416
1032	**1890**	3434
1038	**1900**	3452
1043	**1910**	3470
1049	**1920**	3488
1054	**1930**	3506
1060	**1940**	3524
1066	**1950**	3542
1071	**1960**	3560
1077	**1970**	3578
1082	**1980**	3596
1088	**1990**	3614
1093	**2000**	3632

2001 to 2250

C.	C. or F.	F.
1099	**2010**	3650
1104	**2020**	3668
1110	**2030**	3686
1116	**2040**	3704
1121	**2050**	3722
1127	**2060**	3740
1132	**2070**	3758
1138	**2080**	3776
1143	**2090**	3794
1149	**2100**	3812
1154	**2110**	3830
1160	**2120**	3848
1166	**2130**	3866
1171	**2140**	3884
1177	**2150**	3902
1182	**2160**	3920
1188	**2170**	3938
1193	**2180**	3956
1199	**2190**	3974
1204	**2200**	3992
1210	**2210**	4010
1216	**2220**	4028
1221	**2230**	4046
1227	**2240**	4064
1232	**2250**	4082

2251 to 2490

F.	C. or F.	F.
1238	**2260**	4100
1243	**2270**	4118
1249	**2280**	4136
1254	**2290**	4154
1260	**2300**	4172
1266	**2310**	4190
1271	**2320**	4208
1277	**2330**	4226
1282	**2340**	4244
1288	**2350**	4262
1293	**2360**	4280
1299	**2370**	4298
1304	**2380**	4316
1310	**2390**	4334
1316	**2400**	4352
1321	**2410**	4370
1327	**2420**	4388
1332	**2430**	4406
1338	**2440**	4424
1343	**2450**	4442
1349	**2460**	4460
1354	**2470**	4478
1360	**2480**	4496
1366	**2490**	4514

2491 to 2750

C.	C. or F.	F.
1371	**2500**	4532
1377	**2510**	4550
1382	**2520**	4568
1388	**2530**	4586
1393	**2540**	4604
1399	**2550**	4622
1404	**2560**	4640
1410	**2570**	4658
1416	**2580**	4676
1421	**2590**	4694
1427	**2600**	4712
1432	**2610**	4730
1438	**2620**	4748
1443	**2630**	4766
1449	**2640**	4784
1454	**2650**	4802
1460	**2660**	4820
1466	**2670**	4838
1471	**2680**	4856
1477	**2690**	4874
1482	**2700**	4892
1488	**2710**	4910
1493	**2720**	4928
1499	**2730**	4946
1504	**2740**	4964
1510	**2750**	4982

2751 to 3000

C.	C. or F.	F.
1516	**2760**	5000
1521	**2770**	5018
1527	**2780**	5036
1532	**2790**	5054
1538	**2800**	5072
1543	**2810**	5090
1549	**2820**	5104
1554	**2830**	5126
1560	**2840**	5144
1566	**2850**	5162
1571	**2860**	5180
1577	**2870**	5198
1582	**2880**	5216
1588	**2890**	5234
1593	**2900**	5252
1599	**2910**	5270
1604	**2920**	5288
1610	**2930**	5306
1616	**2940**	5324
1621	**2950**	5342
1627	**2960**	5360
1632	**2970**	5378
1638	**2980**	5396
1643	**2990**	5414
1649	**3000**	5432

NOTE: — The numbers in bold face type refer to the temperature either in degrees Celsius or Fahrenheit which it is desired to convert into the other scale. If converting from degrees Fahrenheit to degrees Celsius the equivalent temperature will be found in the left column, while if converting from degrees Celsius to degrees Fahrenheit, the answer will be found in the column on the right.

$°F = 9/5 (°C) + 32$

$°C = 5/9 (°F - 32)$

INTERPOLATION FACTORS

C.		F.	C.		F.
0.56	1	1.8	3.33	6	10.8
1.11	2	3.6	3.89	7	12.6
1.67	3	5.4	4.44	8	14.4
2.22	4	7.2	5.00	9	16.2
2.78	5	9.0	5.56	10	18.0

S 47283 (B)

Table APP-3. INCHES TO MILLIMETERS (0.0001 INCH TO 10 INCHES)

INCHES	0.0000	0.0001	0.0002	0.0003	0.0004	0.0005	0.0006	0.0007	0.0008	0.0009
					MILLIMETERS					
0.000		0.0025	0.0050	0.0076	0.0101	0.0127	0.0152	0.0177	0.0203	0.0228
0.001	0.0254	0.0279	0.0304	0.0330	0.0355	0.0381	0.0406	0.0431	0.0457	0.0482
0.002	0.0508	0.0533	0.0558	0.0584	0.0609	0.0635	0.0660	0.0685	0.0711	0.0736
0.003	0.0762	0.0787	0.0812	0.0838	0.0863	0.0889	0.0914	0.0939	0.0965	0.0990
0.004	0.1016	0.1041	0.1066	0.1092	0.1117	0.1143	0.1168	0.1193	0.1219	0.1244
0.005	0.1270	0.1295	0.1320	0.1346	0.1371	0.1397	0.1422	0.1447	0.1473	0.1498
0.006	0.1524	0.1549	0.1574	0.1600	0.1625	0.1651	0.1676	0.1701	0.1727	0.1752
0.007	0.1778	0.1803	0.1828	0.1854	0.1879	0.1905	0.1930	0.1955	0.1981	0.2006
0.008	0.2032	0.2057	0.2082	0.2108	0.2133	0.2159	0.2184	0.2209	0.2235	0.2260
0.009	0.2286	0.2311	0.2336	0.2362	0.2387	0.2413	0.2438	0.2463	0.2489	0.2514

INCHES	0.000	0.001	0.002	0.003	0.004	0.005	0.006	0.007	0.008	0.009
					MILLIMETERS					
0.00		0.025	0.050	0.076	0.101	0.127	0.152	0.177	0.203	0.228
0.01	0.254	0.279	0.304	0.330	0.355	0.381	0.406	0.431	0.457	0.482
0.02	0.508	0.533	0.558	0.584	0.609	0.635	0.660	0.685	0.711	0.736
0.03	0.762	0.787	0.812	0.838	0.863	0.889	0.914	0.939	0.965	0.990
0.04	1.016	1.041	1.066	1.092	1.117	1.143	1.168	1.193	1.219	1.244
0.05	1.270	1.295	1.320	1.346	1.371	1.397	1.422	1.447	1.473	1.498
0.06	1.524	1.549	1.574	1.600	1.625	1.651	1.676	1.701	1.727	1.752
0.07	1.778	1.803	1.828	1.854	1.879	1.905	1.930	1.955	1.981	2.006
0.08	2.032	2.057	2.082	2.108	2.133	2.159	2.184	2.209	2.235	2.260
0.09	2.286	2.311	2.336	2.362	2.387	2.413	2.438	2.463	2.489	2.514

INCHES	0.00	0.01	0.02	0.03	0.04	0.05	0.06	0.07	0.08	0.09
					MILLIMETERS					
0.0		0.254	0.508	0.762	1.016	1.270	1.524	1.778	2.032	2.286
0.1	2.540	2.794	3.048	3.302	3.556	3.810	4.064	4.318	4.572	4.826
0.2	5.080	5.334	5.588	5.842	6.096	6.350	6.604	6.858	7.112	7.366
0.3	7.620	7.874	8.128	8.382	8.636	8.890	9.144	9.398	9.652	9.906
0.4	10.160	10.414	10.668	10.922	11.176	11.430	11.684	11.938	12.192	12.446
0.5	12.700	12.954	13.208	13.462	13.716	13.970	14.224	14.478	14.732	14.986
0.6	15.240	15.494	15.748	16.002	16.256	16.510	16.764	17.018	17.272	17.526
0.7	17.780	18.034	18.288	18.542	18.796	19.050	19.304	19.558	19.812	20.066
0.8	20.320	20.574	20.828	21.082	21.336	21.590	21.844	22.098	22.352	22.606
0.9	22.860	23.114	23.368	23.622	23.876	24.130	24.384	24.638	24.892	25.146

INCHES	0.00	0.1	0.2	0.3	0.4	0.5	0.6	0.7	0.8	0.9
					MILLIMETERS					
0.		2.54	5.08	7.62	10.16	12.70	15.24	17.78	20.32	22.86
1.	25.40	27.94	30.48	33.02	35.56	38.10	40.64	43.18	45.72	48.26
2.	50.80	53.34	55.88	58.42	60.96	63.50	66.04	68.58	71.12	73.66
3.	76.20	78.74	81.28	83.82	86.36	88.90	91.44	93.98	96.52	99.06
4.	101.60	104.14	106.68	109.22	111.76	114.30	116.84	119.38	121.92	124.46
5.	127.00	129.54	132.08	134.62	137.16	139.70	142.24	144.78	147.32	149.86
6.	152.40	154.94	157.48	160.02	162.56	165.10	167.64	170.18	172.72	175.26
7.	177.80	180.34	182.88	185.42	187.96	190.50	193.04	195.58	198.12	200.66
8.	203.20	205.74	208.28	210.82	213.36	215.90	218.44	220.98	223.52	226.06
9.	228.60	231.14	233.68	236.22	238.76	241.30	243.84	246.38	248.92	251.46

Table APP-4. WEIGHT (MASS): OUNCES OR POUNDS TO KILOGRAMS

(1 oz = 0.028 349 52 kg) (1 lb = 0.453 592 4 kg)

	0	1	2	3	4	5	6	7	8	9
	kg	kg	kg	kg	kg	kg	kg	kg	kg	kg
oz										
0	–	0.028	0.057	0.085	0.113	0.142	0.170	0.198	0.227	0.255
10	0.283	0.312	0.340	0.369	0.397	0.425	0.454	0.482	0.510	0.539
lb										
0	–	0.45	0.91	1.36	1.81	2.27	2.72	3.18	3.63	4.08
10	4.5	5.0	5.4	5.9	6.4	6.8	7.3	7.7	8.2	8.6
20	9.1	9.5	10.0	10.4	10.9	11.3	11.8	12.2	12.7	13.2
30	13.6	14.1	14.5	15.0	15.4	15.9	16.3	16.8	17.2	17.7
40	18.1	18.6	19.1	19.5	20.0	20.4	20.9	21.3	21.8	22.2
50	22.7	23.1	23.6	24.0	24.5	24.9	25.4	25.9	26.3	26.8
60	27.2	27.7	28.1	28.6	29.0	29.5	29.9	30.4	30.8	31.3
70	31.8	32.2	32.7	33.1	33.6	34.0	34.5	34.9	35.4	35.8
80	36.3	36.7	37.2	37.6	38.1	38.6	39.0	39.5	39.9	40.4
90	40.8	41.3	41.7	42.2	42.6	43.1	43.5	44.0	44.5	44.9
100	45	46	46	47	47	48	48	49	49	49
	0	10	20	30	40	50	60	70	80	90
200	91	95	100	104	109	113	118	122	127	132
300	136	141	145	150	154	159	163	168	172	177
400	181	186	191	195	200	204	209	213	218	222
500	227	231	236	240	245	249	254	259	263	268
600	272	277	281	286	290	295	299	304	308	313
700	318	322	327	331	336	340	345	349	354	358
800	363	367	372	376	381	386	390	395	399	404
900	408	413	417	422	426	431	435	440	445	449
1000	454	458	463	467	472	476	481	485	490	494

Table APP-5. WEIGHT (MASS): THOUSAND POUNDS TO KILOGRAMS

(1 lb = 0.453 592 4 kg)

lb	0	100	200	300	400	500	600	700	800	900
(000)*	kg	kg	kg	kg	kg	kg	kg	kg	kg	kg
1	454	499	544	590	635	680	726	771	816	862
2	907	953	998	1043	1089	1134	1179	1225	1270	1315
3	1361	1406	1451	1497	1542	1588	1633	1678	1724	1769
4	1814	1860	1905	1950	1996	2041	2087	2132	2177	2223
5	2268	2313	2359	2404	2449	2495	2540	2585	2631	2676
6	2722	2767	2812	2858	2903	2948	2994	3039	3084	3130
7	3175	3221	3266	3311	3357	3402	3447	3493	3538	3583
8	3629	3674	3719	3765	3810	3856	3901	3946	3992	4037
9	4082	4128	4173	4218	4264	4309	4354	4400	4445	4491
10	4536	4581	4627	4672	4717	4763	4803	4853	4899	4944
11	4990	5035	5080	5126	5171	5216	5262	5307	5352	5398
12	5443	5488	5534	5579	5625	5670	5715	5761	5806	5851
13	5897	5942	5987	6033	6078	6123	6169	6214	6260	6305
14	6350	6396	6441	6486	6532	6577	6622	6668	6713	6759
15	6804	6849	6895	6940	6985	7031	7076	7121	7167	7212
16	7257	7303	7348	7394	7439	7484	7530	7575	7620	7666
17	7711	7756	7802	7847	7893	7938	7983	8029	8074	8119
18	8165	8210	8255	8301	8346	8391	8437	8482	8528	8573
19	8618	8664	8709	8754	8800	8845	8890	8936	8981	9026
20	9072	9117	9163	9208	9253	9299	9344	9389	9435	9480

*Multiply lb value by 1000

ANNUNCIATORS

The Annunciator Section presents a color representation of all the annunciator lights in the airplane.

Please unfold page ANN-1 to the right and leave it open for ready reference as the annunciators are cited in the text.

FlightSafety international
LEARJET 30 Series PILOT TRAINING MANUAL

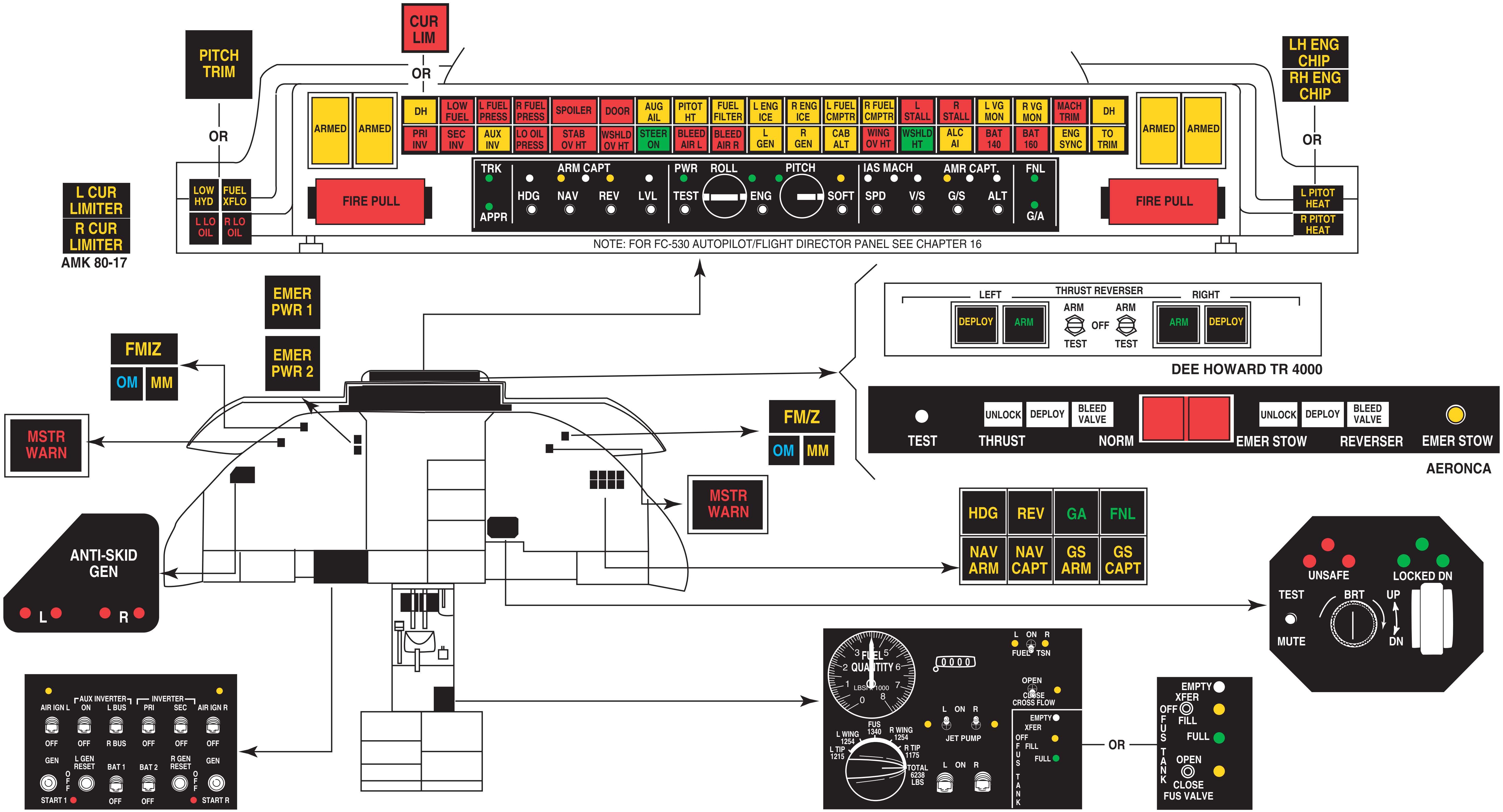

Figure ANN-1. Annunciators