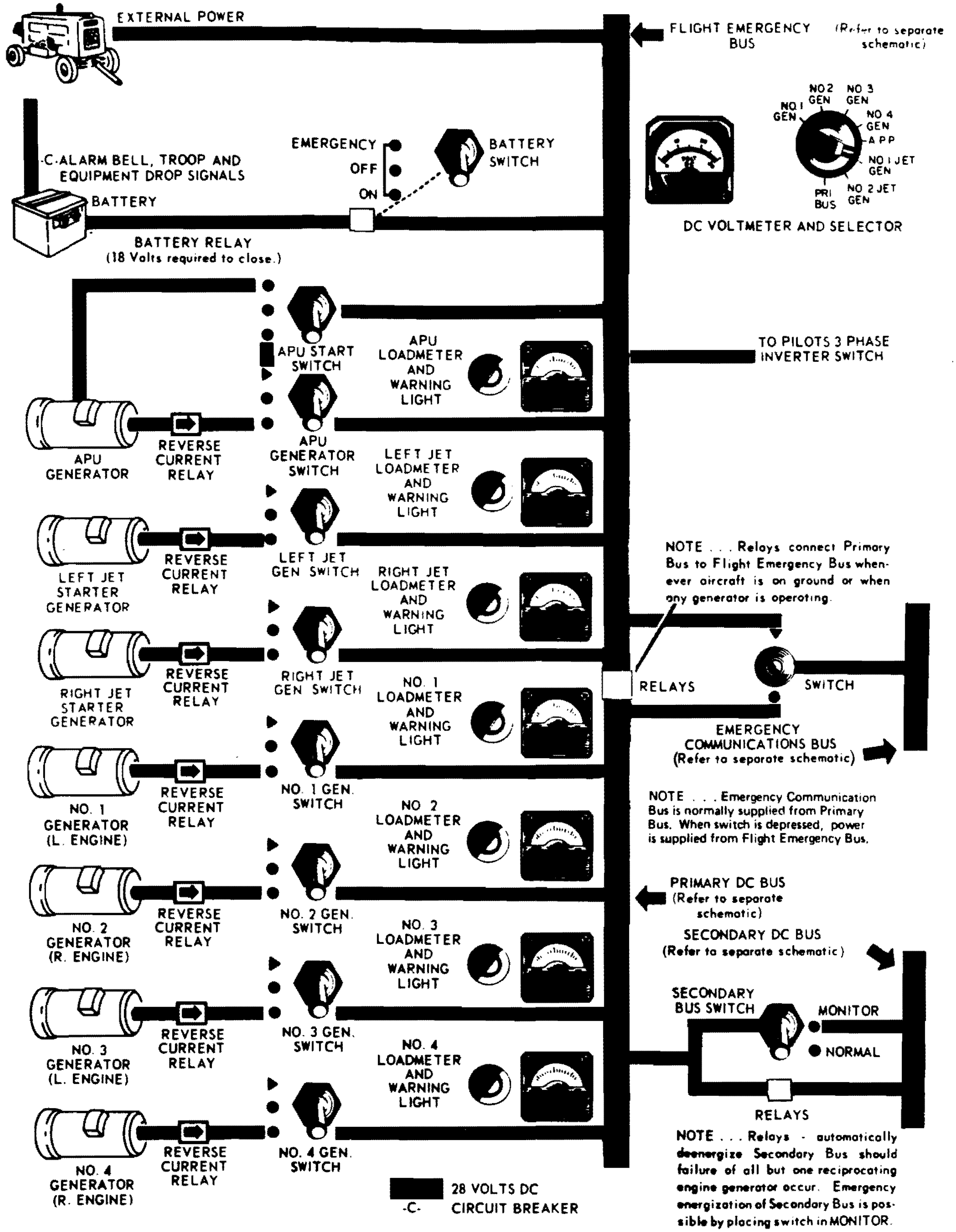


DC ELECTRICAL power supply

Figure 1-19

ELECTRICAL equipment

(TYPICAL)

1. ENGINE GENERATOR
2. NACELLE JUNCTION BOX
3. GROUND BLOWER RELAY JUNCTION BOX
4. RIGHT MAIN TERMINAL JUNCTION BOX
5. FIELD CONTROL RELAYS
6. MAIN ELECTRICAL EQUIPMENT PANEL
7. APU GENERATOR
8. CARGO LIGHTS BOX
9. HEATER CONTROL BOX
10. VOLTAGE REGULATORS (RECIP ENGINES)
11. SINGLE PHASE INVERTER
12. IFF JUNCTION BOX
13. SPARE THREE PHASE INVERTER
14. PILOT'S AND COPILOT'S THREE PHASE INVERTERS
15. J-2 COMPASS BOX
16. LIFT COMPUTER
17. HEATER RELAY-FIRE DETECTOR CONTROL BOX
18. AC RELAY AND CIRCUIT BREAKER PANEL
19. LEFT MAIN TERMINAL JUNCTION BOX
20. COWL FLAP-FIRE EMERGENCY RELAY BOX
21. BATTERY
22. EXTERNAL POWER RECEPTACLE
23. MISCELLANEOUS EQUIPMENT AND APP RELAY PANEL
24. RADIO AC POWER PANEL
25. INTERPHONE JUNCTION BOX
26. VOR JUNCTION BOX
27. DIMMING RELAY PANEL
28. RADIO SECONDARY BUS RELAY BOX
29. IGNITION ANALYZER JUNCTION BOX
30. OVERHEAD PANELS
31. AILERON DEICING JUNCTION BOX

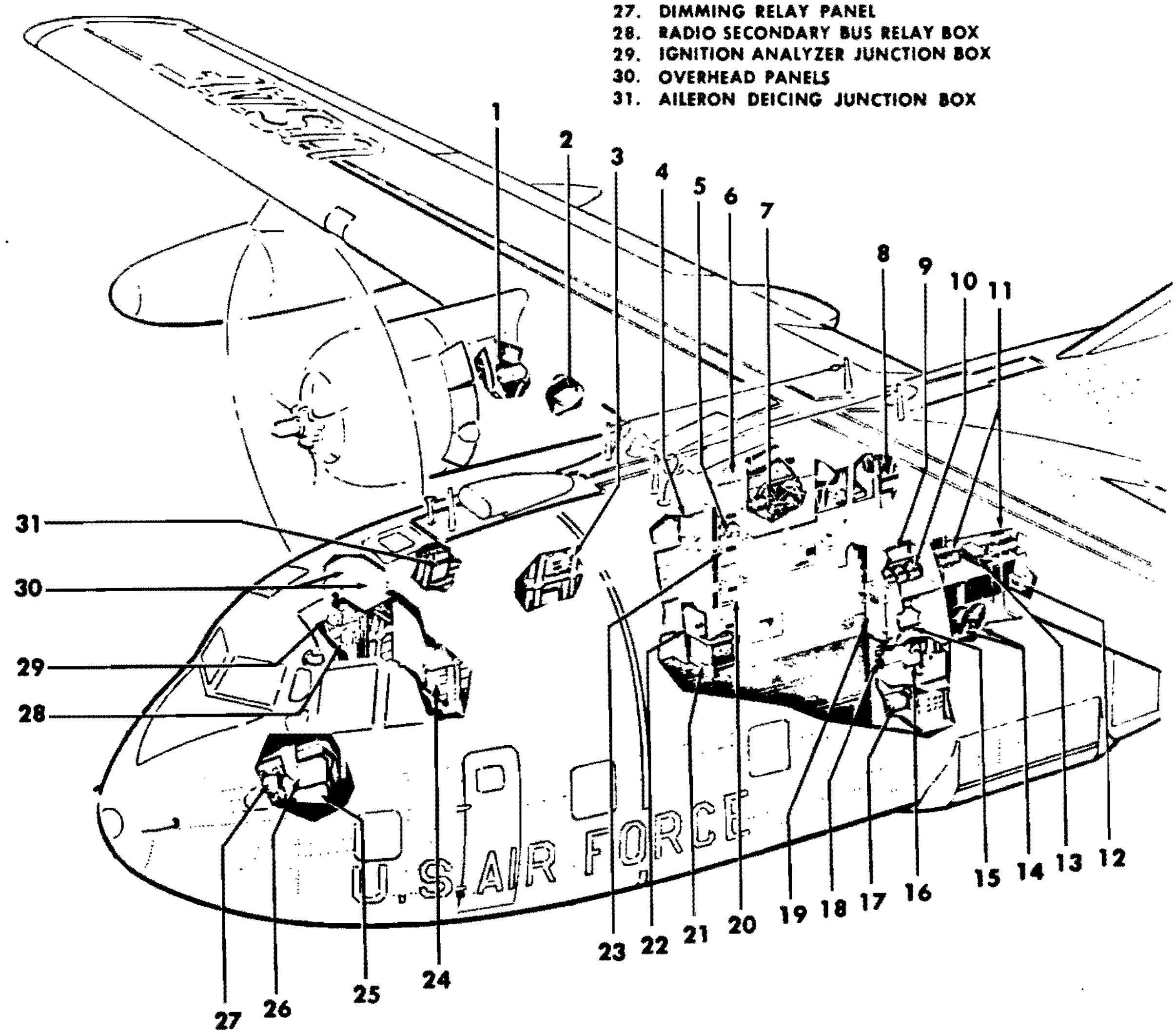


Figure 1-20

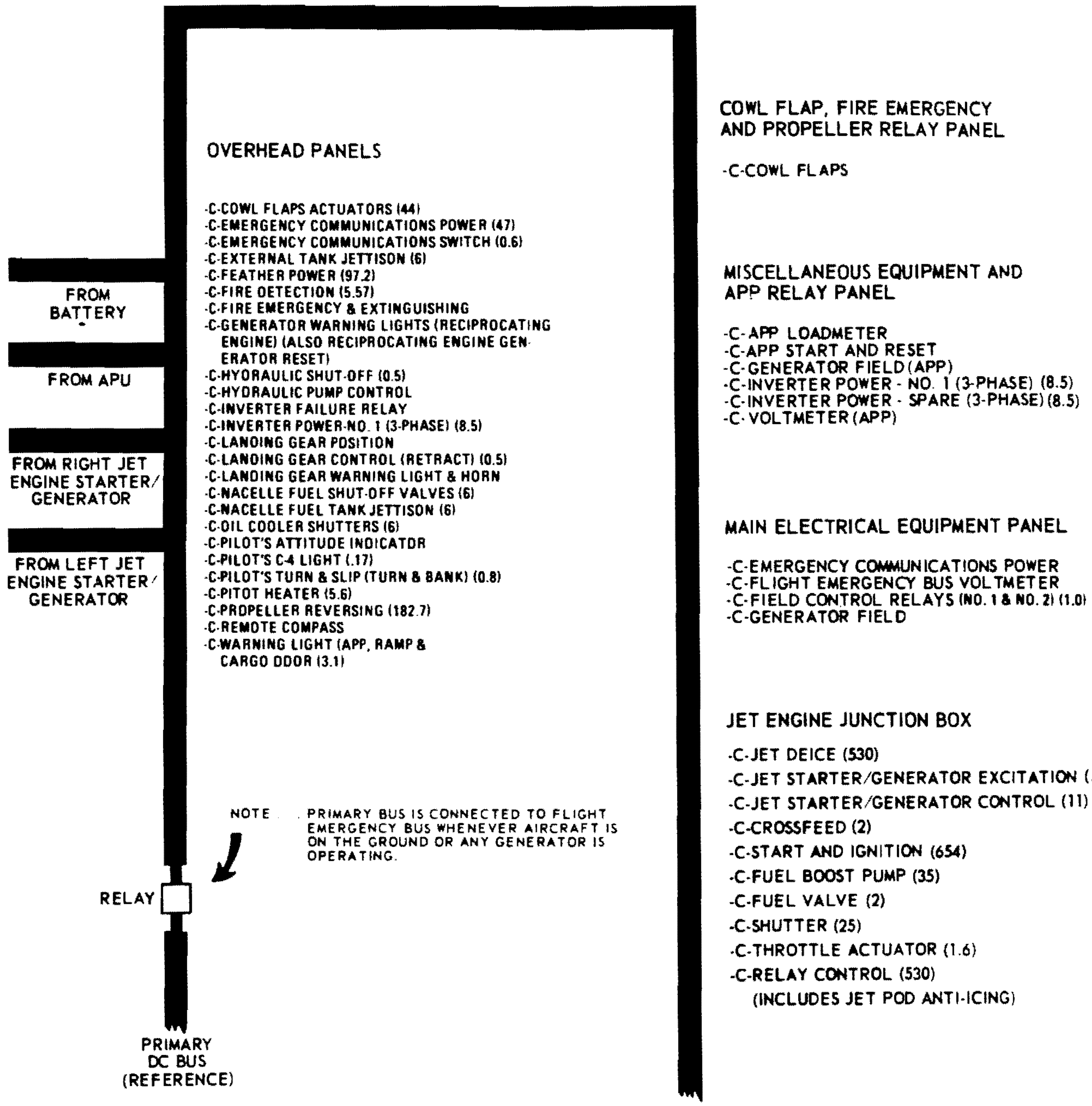
17516

Voltmeter Selector Switches and Voltmeter.

A multi-position, rotary voltmeter selector switch (figure 1-58), and a 30-volt dc voltmeter (figure 1-58), are mounted on the copilot's instrument panel. The placarded positions are No. 1 GEN, No. 2 GEN, No. 3 GEN, No. 4 GEN, APP, No. 1 JET GEN, No. 2 JET GEN, PRI BUS and OFF. The voltmeter will indicate the voltage output of the component selected by the selector switch.

Reciprocating Engine Generator Field Control Relay Buttons.

Any of the generator field control relays located in the right electrical compartment forward of the right main gear well may be manually reset if the RESET position of the generator control switch fails to per-



NOTE . . .

- MORE CIRCUIT BREAKERS MAY BE FOUND LISTED HERE THAN OCCUR IN A SPECIFIC PANEL OR JUNCTION BOX. THIS LISTING REPRESENTS A COMPLETE COMPOSITE OF ALL C-123K AIRCRAFT. ON ANY SPECIFIC AIRCRAFT, THE CIRCUIT BREAKERS ARE CLEARLY PLACARDED AT THE JUNCTION BOX OR PANEL IN WHICH THEY ARE LOCATED.
- CURRENT REQUIREMENTS ARE NOTED IN PARENTHESES FOLLOWING EQUIPMENT DESIGNATION AND ARE BASED ON NORMAL OPERATION.

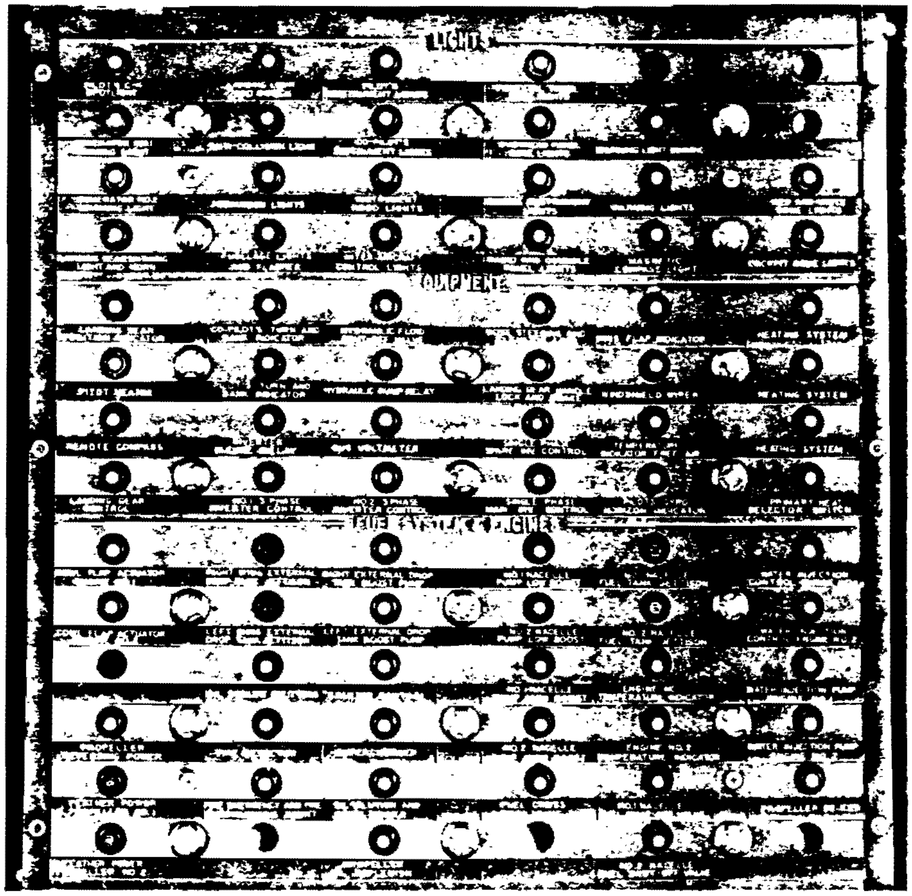
(black bar) 28 VOLTS DC

-C- CIRCUIT BREAKER

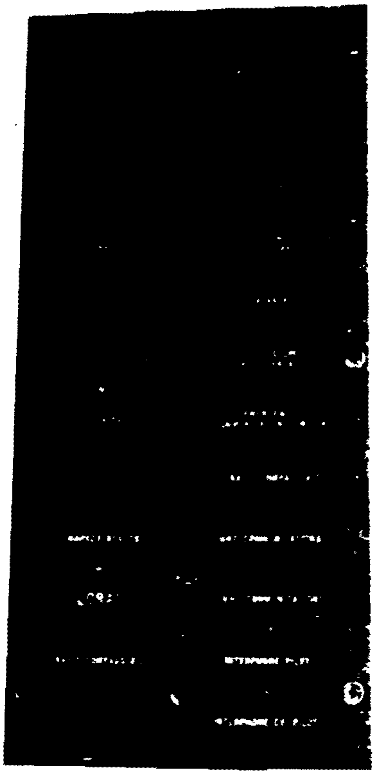
DC ELECTRICAL flight emergency bus . . Typical

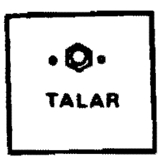
Figure 1-21

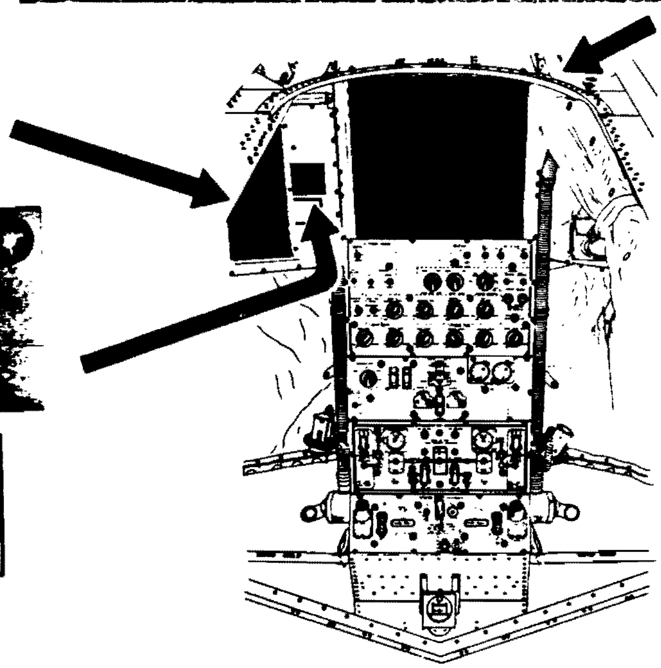
PILOTS' OVERHEAD PANEL



PILOTS' OVERHEAD PANEL (Radio)

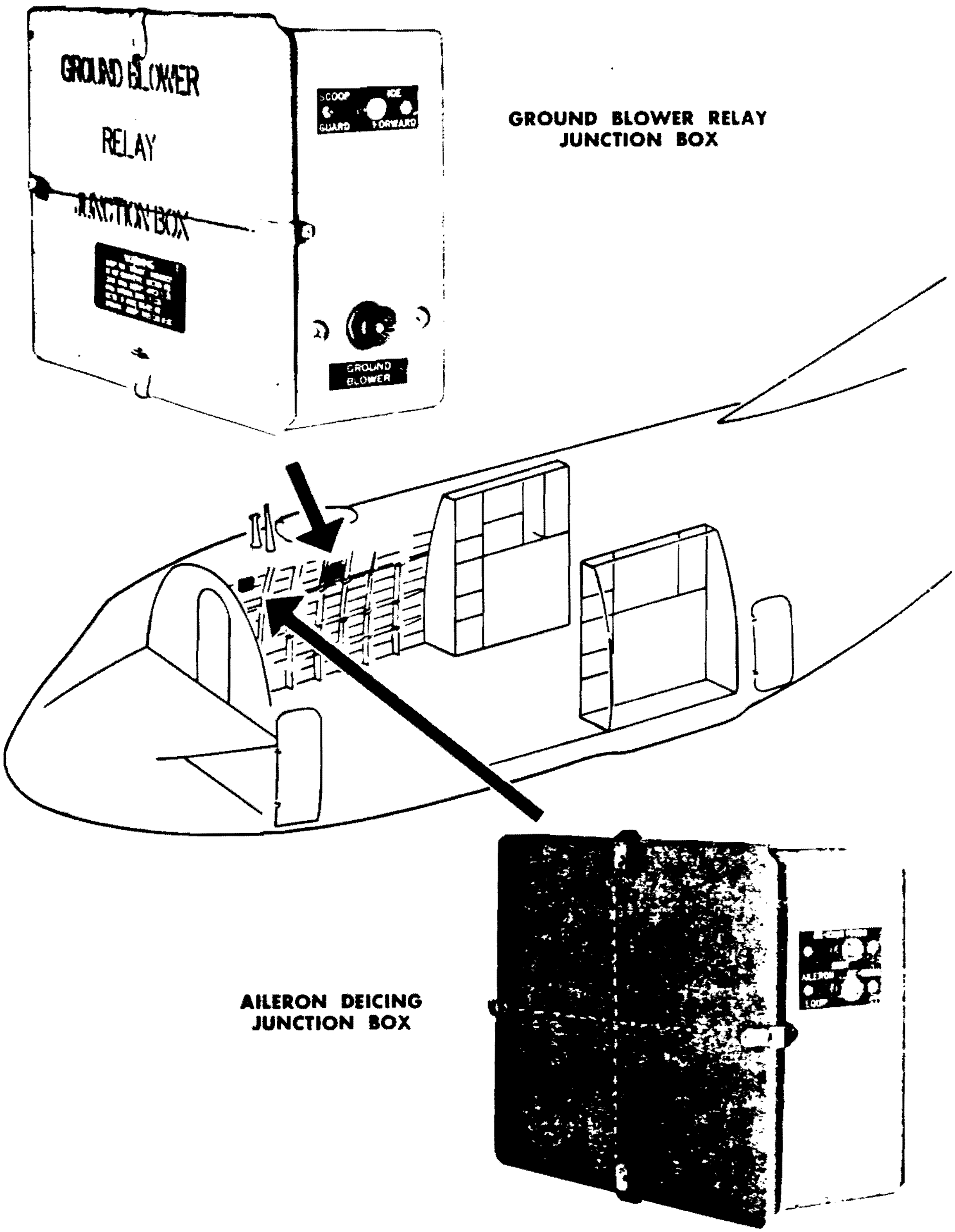






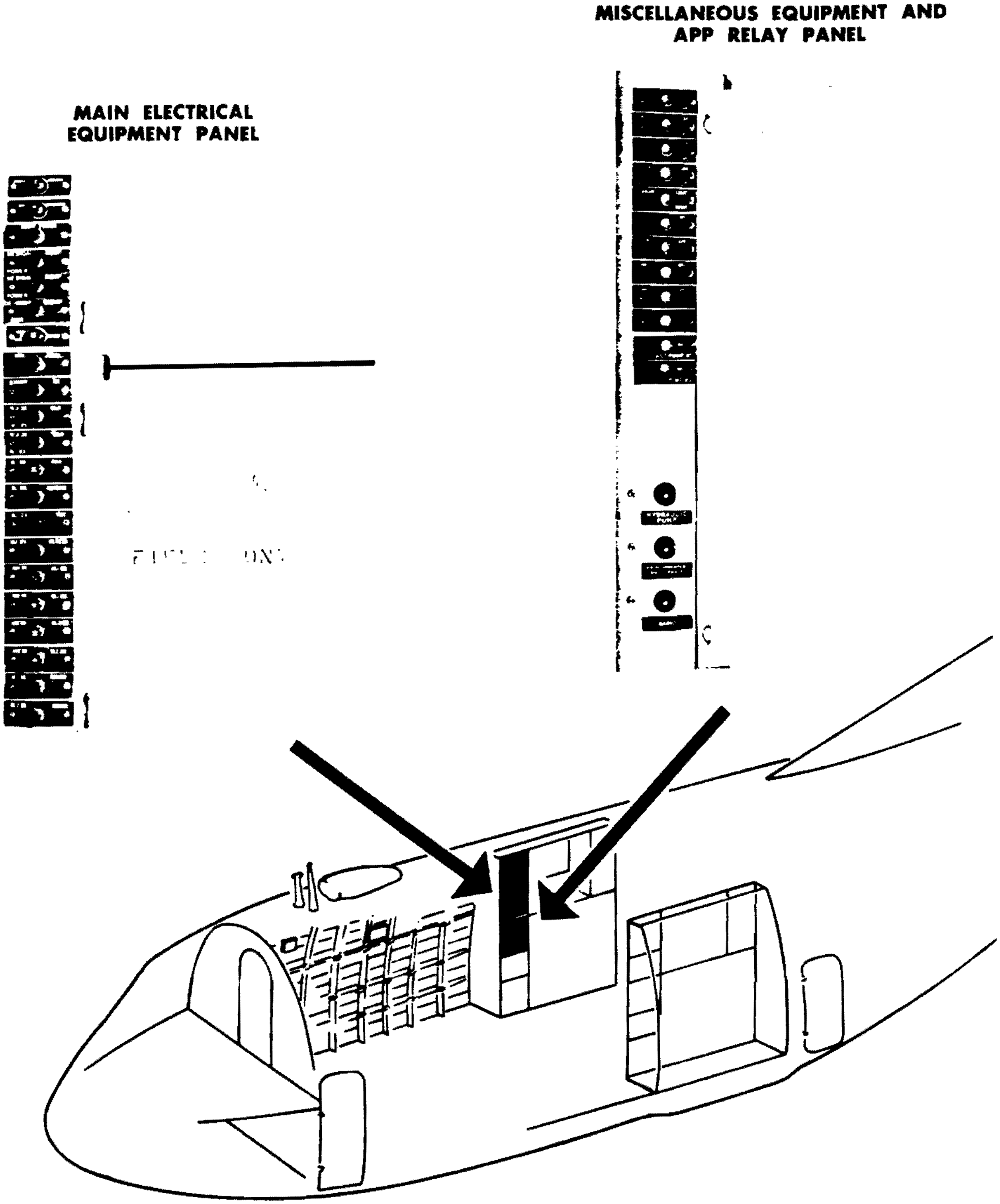
CIRCUIT BREAKER

Figure 1-22 (Sheet 1)



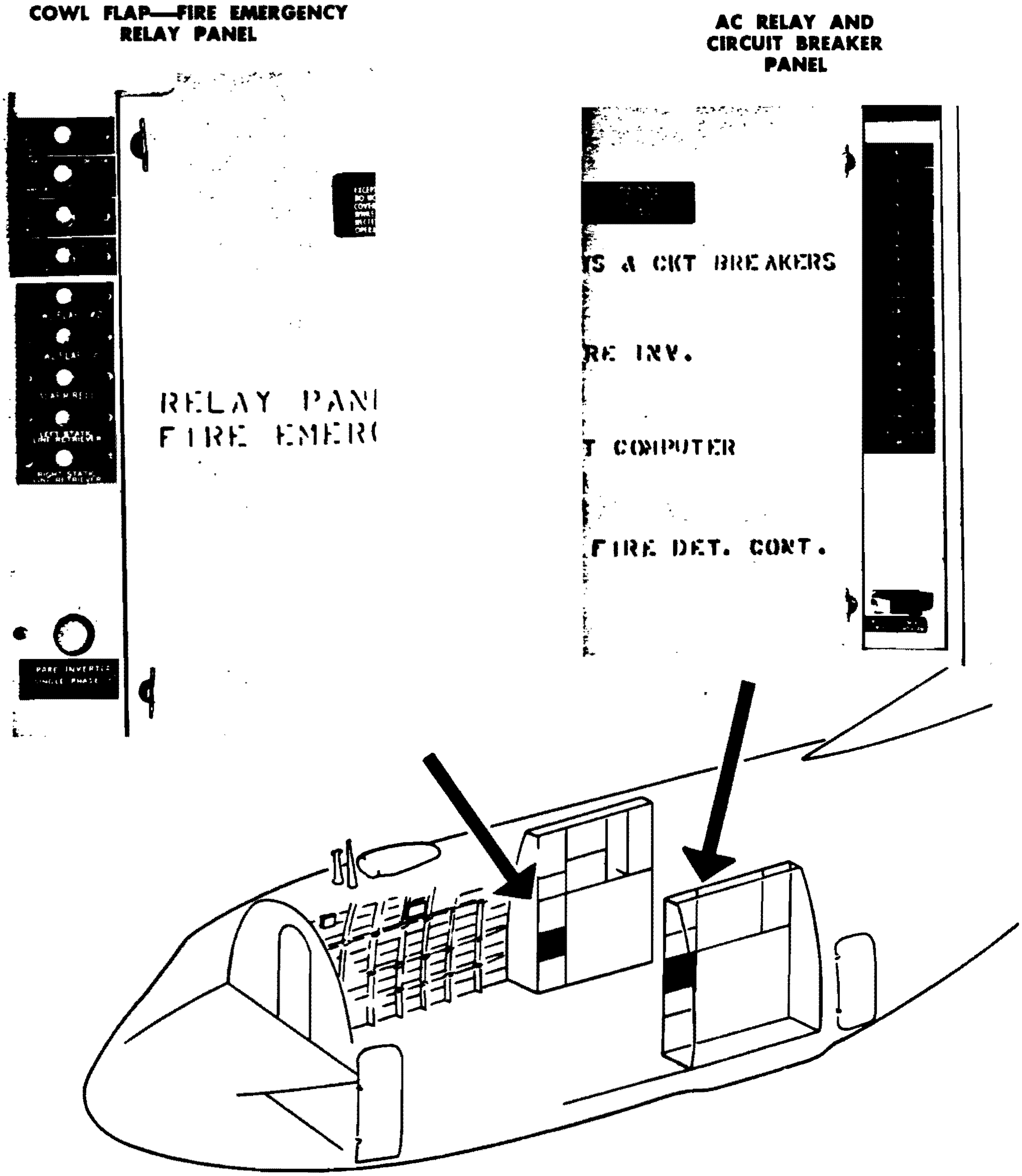
panels ..... typical

Figure 1-22 (Sheet 2)



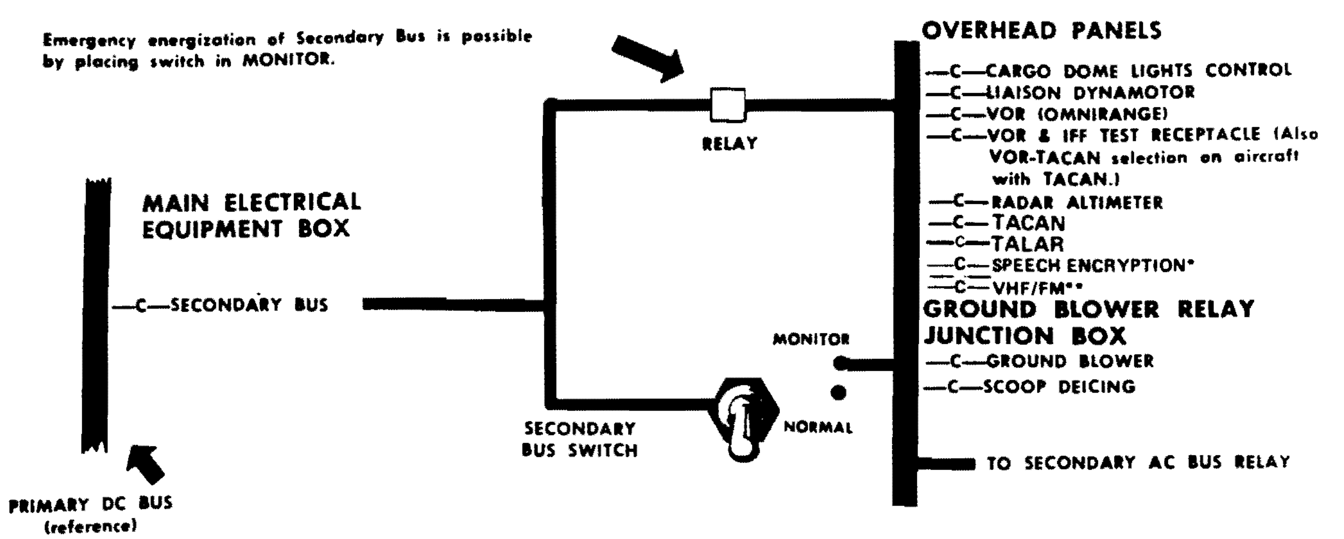
CIRCUIT BREAKER

Figure 1-22 (Sheet 3)



panels ..... typical

Figure 1-22 (Sheet 4)

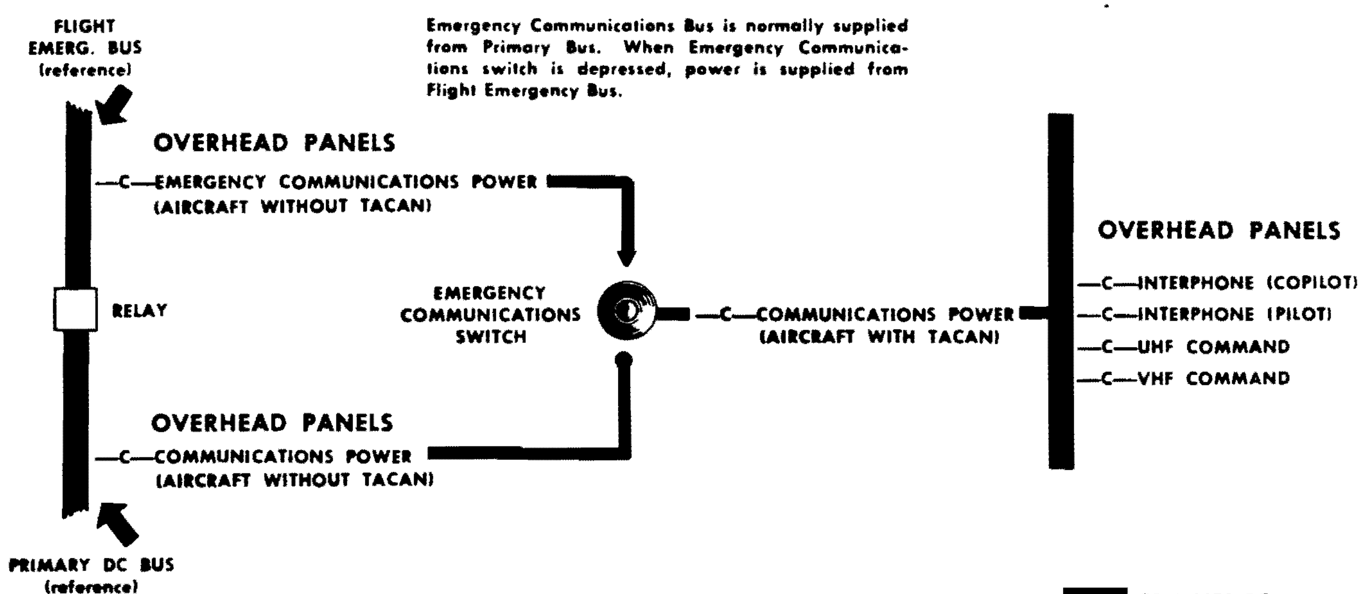


*AIRCRAFT MODIFIED IN ACCORDANCE WITH TCTO 1C-123K-584.

**AIRCRAFT MODIFIED IN ACCORDANCE WITH TCTO 1C-123K-589.

DC ELECTRICAL
secondary bus

Figure 1-23



DC ELECTRICAL
emergency communications bus

Figure 1-24

DC ELECTRICAL PRIMARY BUS (TYPICAL)

FROM NO.1 ENGINE GENERATOR

FROM NO.2 ENGINE GENERATOR

FROM NO.3 ENGINE GENERATOR

FROM NO.4 ENGINE GENERATOR

OVERHEAD PANELS

-C-ANTI-COLLISION LIGHT (3.7)
-C-APP MAG CUT OUT
-C-BUS VOLTMETER
-C-CARGO DOOR & RAMP WARNING
-C-COCKPIT DOME LIGHTS
-C-COMMUNICATIONS POWER
-C-COPILOT'S INSTRUMENT LIGHTS
-C-COPILOT'S TURN & SLIP (TURN & BANK)
-C-EMERGENCY KEYER
-C-ENGINE INSTRUMENTS LIGHTS
-C-ENGINE TEMPERATURE
-C-EXTERNAL TANK BOOST PUMPS
-C-FORMATION LIGHTS (1.9)
-C-FREE AIR TEMPERATURE
-C-FUEL CROSSFEED (6)
-C-FUSELAGE LIGHTS
-C-HEATING SYSTEM*
-C-HEATING SYSTEM*
-C-HEATING SYSTEM*
-C-HYDRAULIC PUMP RELAY
-C-HYDRAULIC SHUT-OFF
-C-IFF RADAR (2.4)
-C-INSTRUMENT HOOD LIGHTS (3.6)
-C-INTERAIRCRAFT CONTROL LIGHT (5.3)
-C-INTERPHONE (COPILOT'S) (10)
-C-INVERTER POWER-NO. 2 (3-PHASE)
-C-INVERTER SWITCH-MAIN (1-PHASE)
-C-LANDING GEAR HANDLE LOCK & BRAKE (INCLUDES ANTI-SKID)
-C-LANDING LIGHT AND SWITCH (42.8)
-C-MAGNETIC COMPASS LIGHT
-C-MARKER BEACON (1.5)
-C-NACELLE BOOST PUMP (56)
-C-NOSE STEER-ELEVATOR LOCK
-C-OIL DILUTION AND FUEL PRIME
-C-OVERHEAD PANEL EDGE LIGHTS
-C-PEDESTAL EDGE LIGHTS
-C-PILOT'S INSTRUMENTS LIGHTS
-C-PRIMARY HEAT SELECTOR SWITCH
-C-PROPELLER OIL REPLENISH
-C-PROPELLER DEICING (433)
-C-RADAR ALTIMETER (1.5)
-C-RADIO COMPASS (4.5)
-C-RADIO COMPASS NO. 2 (4.5)
-C-RADIO AND CONSOLE PANEL LIGHTS
-C-STARTER CONTROL
-C-SUPERCHARGER (1.5)
-C-UHF DIRECTION FINDING
-C-WARNING LIGHTS DIMMING
-C-WATER INJECTION CONTROL
-C-WATER INJECTION PUMP
-C-WING FLAP INDICATOR
-C-WINDSHIELD WIPER (5.2)
-C-WINDSHIELD HEATER (173.5)
-C-AERIAL SPRAY

MISCELLANEOUS EQUIPMENT AND APP RELAY PANEL

-C-APP AMMETER
-C-APP COMPARTMENT LIGHT
-C-APP START AND RESET
-C-CARGO LIGHTS (WHITE 20) (RED 37.5)
-C-FUEL BOOSTER (56)
-C-HYDRAULIC PUMP
-C-INVERTER POWER-MAIN (1-PHASE) (100)
-C-INVERTER-NO. 2 (3-PHASE) (8.5)
-C-LANDING LIGHTS (42.8)
-C-RADIO

MAIN ELECTRICAL EQUIPMENT PANEL

-C-ANGLE OF ATTACK
-C-STALL WARNING
-C-LOADMETERS
-C-ANTI-ICING VALVES (1.3)
-C-CARGO HEAT VALVES (1.3)
-C-GENERATOR VOLTMETER
-C-HEATER VALVES (2.6)
-C-IGNITION ANALYZER
-C-INDUCTION VIBRATOR
-C-ANGLE OF ATTACK DEICE
-C-MEDICAL RECEPTACLE POWER
-C-NAVIGATOR'S EQUIPMENT LIGHTING
-C-SECONDARY BUS
-C-DEICING EQUIPMENT POWER
-C-SPRAY SYSTEM DUMP VALVE
-C-SPRAY BUS CONTROL

AILERON DEICING JUNCTION BOX

-C-AILERON DEICING EQUIPMENT
-C-DEICING PRESSURE CONTROL

COWL FLAP, FIRE EMERGENCY AND PROPELLER RELAY PANEL

-C-INVERTER POWER-SPARE (1-PHASE)
-C-PROPELLER DEICING LOADMETERS
-C-STATIC LINE RETRIEVERS

SPRAY CONTROL PANEL

-C-STARTER AND CHOKE
-C-THROTTLES
-C-INDICATORS
-C-SPRAY VALVE
(B)

OPERATOR CONSOLE PANEL ASSEMBLY (B)

-C- HI-VOLUME ENGINE IGNITION
-C- SPRAY CONTROL
-C- FLOWMETER
-C- ULV BY-PASS VALVES
-C- HI-VOLUME SPRAY – ULV FLOW VALVES
-C- ULV PUMP CONTROL
-C- SPRAY ARMING IND
-C- ULV SPRAY VALVE
-C- SPRAY SYSTEM RECIRC VALVES
-C- ULV PUMP

EXTERNAL POWER BOX

-C-INVERTER POWER-SPARE (1-PHASE) (100)

WINDSHIELD HEATER CIRCUIT BREAKER JUNCTION BOX

-C-INVERTER POWER (173.5)

■ 28 VOLTS DC

-C- CIRCUIT BREAKER

NOTE . . .

- MORE CIRCUIT BREAKERS MAY BE FOUND LISTED HERE THAN OCCUR IN A SPECIFIC PANEL OR JUNCTION BOX. THIS LISTING REPRESENTS A COMPLETE COMPOSITE OF ALL UC-123K AIRCRAFT. ON ANY SPECIFIC AIRCRAFT, THE CIRCUIT BREAKERS ARE CLEARLY PLACARDED AT THE JUNCTION BOX OR PANEL IN WHICH THEY ARE LOCATED.
- CURRENT REQUIREMENTS ARE NOTED IN PARENTHESES FOLLOWING EQUIPMENT DESIGNATION.

* The three HEATING SYSTEM circuit breakers are an integral part of the complete control system; it is not possible to assign a specific function to each of them individually.

Figure 1-25

FIELD CONTROL RELAYS . . . . . .

TYPICAL

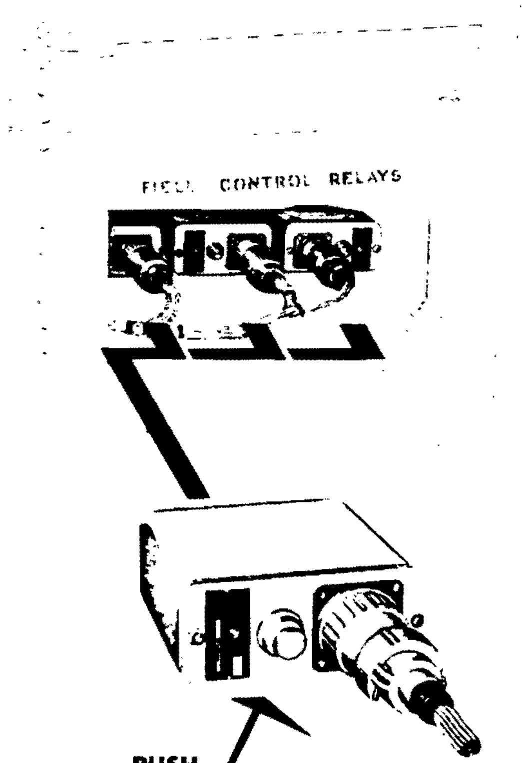
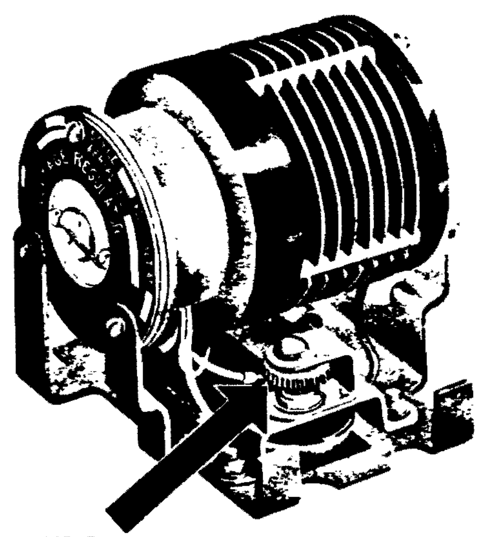


Figure 1-26



voltage regulator . . . RHEOSTAT

Figure 1-27 17519

form this function electrically. Depressing the reset button which protrudes from the case of the relay unit resets the relay.

Battery Switch.

A three-position switch (figure 1-7), is located on the engine starting panel to connect the battery to the 28-volt dc system. In the OFF position, the battery is disconnected from the system. With the switch in the ON or EMERGENCY positions, battery power is connected to the flight emergency bus and may be used for normal starting of the APU. A battery relay, which operates in conjunction with the battery switch, must be closed before the generator can recharge the battery. A minimum power supply of 18 volts from the battery is required to close the relay. The battery switch need not be on for operation of the alarm bells, troop signal or equipment drop signals.

Note

- The ON and EMERGENCY positions are connected by a jumper wire.
- The signal lights, if left on, will drain the battery.

Reciprocating Engine Generator Loadmeters.

Generator loadmeters (figure 1-58), one for each engine-driven generator, are mounted on the copilot's instrument panel. These loadmeters indicate the percent of output of the corresponding generator as compared with its rated maximum output. Thus a 0.4 reading for a 300-ampere rated generator would mean an output of 120 amps.

Jet Engine Generator Loadmeters.

Two jet engine generator loadmeters (figure 1-58), located on the copilot's instrument panel, indicate the percentage of output of the corresponding generator as compared with its rated maximum output.

Voltage Regulator Rheostats.

Four voltage regulators, located in the left main electrical panel, control the voltage output of the aircraft's reciprocating engine generators. The regulators are placarded No. 1 VOLTAGE REGULATOR through No. 4 VOLTAGE REGULATOR. Each regulator is adjusted by a rheostat knob on the side of the regulator case. Except for in-flight emergencies, generator voltage adjustment is a ground maintenance item. Refer to Section III.

Jet Engine Generator Control Units.

The generator control units (GCU) are multi-purpose and provide the following functions: voltage regulation, field weakening during jet engine starting, generator parallel equalization, overvoltage protection and control, and speed sensing for automatic transfer of the start cycle to generating mode. Two external adjustments for overvoltage and paralleling of the generators are provided. The field weakening circuit of the GCU supplies more positive torque and starting assistance for the starter/generator at high jet rotor speed. The GCU unit replaces the jet engine generator field control relays. The jet engine generator control

Figures 1-28, 1-29 and 1-30 deleted.

switch resets the GCU rather than the generator field relays, when positioned to RESET. The reset/trip switch on the GCU is not used in lieu of the RESET position of the generator control switch, each has its specific function.

The generator control units, one for each jet starter/generator are located on the right and left side of the cargo compartment forward of the main wheel well.

AC POWER SYSTEM.

Alternating current for the operation of flight instruments, engine instruments, and radio equipment on the aircraft is supplied by 2 single-phase and 3 three-phase, 115-volt, 400-cycle inverters located in compartments near the left wheel well in the cargo compartment. The inverters are 28-volt dc compound-wound motors which drive ac generators built on the same shaft. The ac supply is dependent, therefore, upon the dc system. The inverters are self-contained units incorporating the necessary controls for governing rpm and voltage. A secondary ac bus arrangement is incorporated into the single-phase ac power system. The secondary ac bus is energized only when its counterpart, the dc secondary bus, is energized. Should the output of three of the reciprocating engine-driven generators be excluded from the bus for any reason, the secondary dc bus, and correspondingly the secondary ac bus is deenergized. Should use of the equipment on the secondary bus become necessary, the automatic disconnection feature may be bypassed by positioning the secondary bus switch to MONITOR. Aircraft modified in accordance with T.O. 1C-123K-524 require the loss of all but one engine-driven generator before the secondary ac bus is de-energized.

Single-phase Inverters.

Two 1500-volt ampere, 400 cycle-per-second, 115-volt ac, single-phase inverters (figure 1-20), are mounted on a rack over the left wheel well. One inverter is the main, the other the spare. Normally, the main inverter supplies the 115-volt ac, single-phase power requirements of the aircraft. Should the main single-phase inverter fail, an automatic changeover relay will function to turn on the spare single-phase inverter. If desired, operation of the spare single-phase inverter may be manually selected. The operating single-phase inverter provides ac power for fuel quantity gages, fuel pressure gages, oil pressure gages, torque pressure gages, aileron deicing pressure gage, course indicator, pilot's heading indicator, navigator's heading indicator, radio magnetic indicators, slaved gyro magnetic compass, liaison radio, radar altimeter, IFF radar, TACAN, glide slope ignition analyzer, and the single-phase transformer.

Single-phase Inverter Switch And Warning Lights.

A three-position switch (figure 1-58), on the copilot's instrument panel controls the selection of the main or spare single-phase inverter. In the center OFF position of the switch neither single-phase inverter is energized. The MAIN position is normally used and permits the main single-phase inverter to supply the power for the single-phase ac system. Should the main single-phase inverter fail while the switch is in the MAIN position, a single-phase inverter failure warning light (figure 1-58), labelled MAIN on the copilot's instrument panel, will come on. At the same time an automatic changeover relay will function to permit spare single-phase inverter to assume the responsibility of supplying the single-phase ac system.

Note

> When the single-phase inverter switch is in MAIN and the main single-phase inverter fails, a second single-phase inverter failure warning light (figure 1-58), labelled BOTH, will momentarily light until the automatic changeover is accomplished.

The SPARE position of the single-phase inverter switch is used to manually connect the spare single-phase inverter into the system should the automatic changeover fail to operate. If the switch is in the SPARE position, the MAIN single-phase inverter warning light will glow. Should the spare single-phase inverter fail while the switch is turned to SPARE, the automatic changeover relay will not automatically place the main single-phase inverter in operation even though the main single-phase inverter is still operable. The BOTH light will glow. Manual operation of the single-phase inverter switch is required to place the main single-phase inverter in the ac system as a source of power; the MAIN and BOTH single-phase inverter failure lights will then go out. If both single-phase inverters fail, the condition is indicated by the glowing of the BOTH single-phase inverter failure warning light on the copilot's instrument panel. For equipment affected by loss of power in the single-phase ac system, refer to INVERTER FAILURE, Figure 3-14.

Pages 1-47 and 1-48 deleted.

Single-phase Transformer.

A transformer located in the ac circuit breaker and relay panel (figure 1-20), reduces part of the output of the single-phase inverter to 26-volt, single-phase power for operation of the torque pressure, fuel pressure, oil pressure and aileron deicing pressure indication systems.

Three-phase Inverters. *

Three 115-volt, 400 cycle-per-second, three-phase inverters (figure 1-20), rated at 100 volt amperes on some aircraft and at 250 volt amperes on other aircraft, are provided to supply three-phase ac power for the pilot's and copilot's instruments. The pilot's and copilot's inverters are located over the left wheel well and under the single-phase inverter. A third, or spare three-phase inverter is located in the left electrical equipment compartment on some aircraft † or over the left wheel well aft of the pilot's and copilot's inverters on other aircraft ††. The pilot's inverter powers the pilot's attitude indicator and the slaved gyro magnetic compass system. The copilot's inverter powers the copilot's attitude indicator and heading indicator. The spare inverter can assume the load of either the pilot's or copilot's inverter. Under emergency conditions the pilot's inverter or spare inverter can be driven directly by the battery to operate the pilot's instruments, since both receive power from the flight emergency bus. The copilot's inverter is driven by power from the primary bus and will be inoperative if primary bus power is lost. In event of single-phase power failure, the slaved gyro magnetic compass automatically receives single-phase power from the "c" leg of the three-phase inverter supplying power to the pilot's instruments.

** AF 54-552 thru 54-706*
† AF 54-552 thru 54-686
†† AF 54-687 thru 54-706

Three-phase Inverter Switches And Warning Lights. *

Two three-position inverter switches, one on the pilot's and one on the copilot's instrument panel, control operation of the three-phase inverters. Switch positions are ON, OFF, and SPARE. Positioning either switch to ON will start operation of the corresponding inverter. Positioning the switch to OFF will cause the corresponding inverter to become inoperative. In the SPARE position of the pilot's switch, the spare inverter will assume the load of the pilot's inverter. In the SPARE position of the copilot's switch, the spare inverter will assume the load of the copilot's inverter. If both switches are in the SPARE position, the spare inverter will assume only the load of the pilot's inverter as the pilot's switch overrides the copilot's switch in SPARE position. An instrument power failure red warning light is located on the instrument panel adjacent to its respective switch. If the pilot's or copilot's three-phase inverter fails, the corresponding warning light will glow. When the spare inverter is energized to carry the load of the inoperative inverter, the warning light will go out, indicating that power is being supplied by the spare inverter.

Note

> On aircraft, AF 54-687 through 54-706, the three-phase inverters require a 20-second warm-up period when first turned ON. During the warm-up, the warning lights will glow. The spare inverter is warmed up simultaneously with the pilot's, copilot's, or both inverters and requires no additional time in event of switch over.

Three-phase Inverter Switches And Warning Lights. **

One three-position ON-OFF-MANUAL OVERRIDE switch (figure 1-58) and one two-position ON-OFF switch (figure 1-58) control operation of the three-phase inverters. The pilot's switch, mounted on the pilot's instrument panel, is a three-position switch controlling operation of the pilot's or spare three-phase inverters. When positioned to ON, the inverter, after approximately one minute warm-up time, will be ready for operation. The warm-up period requires 60 seconds, during which time a red warning light (figure 1-58), labelled INSTRUMENT POWER FAILURE adjacent to the pilot's inverter switch will illuminate. After the warm-up period the pilot's three-phase inverter is energized and the pilot's INSTRUMENT POWER FAILURE warning light will go out. Should the pilot's three-phase inverter output be unsatisfactory, the pilot's INSTRUMENT POWER FAILURE warning light will remain on, and an automatic changeover relay will select the spare inverter to supply the pilot's instrument power requirements. After the spare inverter has been selected, the pilot's INSTRUMENT POWER FAILURE warning light will go out and an amber warning light (figure 1-58), labelled SPARE INVERTER ON will illuminate, indicating that the spare three-phase inverter is supplying power to the pilot's instruments. Should the voltage output of the spare three-phase inverter become unsatisfactory, the pilot's INSTRUMENT POWER FAILURE warning light will glow, and the automatic changeover relay will again try the pilot's inverter. The SPARE INVERTER ON light will then go out. This cycling will continue until a satisfactory inverter output is found. During the cycling, the pilot's INSTRUMENT POWER FAILURE light will remain illuminated, while the SPARE INVERTER ON light will alternately illuminate for four seconds and be out for four seconds. Direct operation of the spare inverter, bypassing the automatic changeover relay, may be accomplished by positioning the pilot's three-phase inverter switch to MANUAL OVERRIDE. If the MANUAL OVERRIDE position is selected upon initially starting the pilot's three-phase inverter, a 20-second warm-up period will be required. During the warm-up, the pilot's INSTRUMENT POWER FAILURE light will illuminate. When the spare three-phase inverter begins supplying power for the pilot's instruments, the pilot's INSTRUMENT POWER FAILURE light will go out and the SPARE INVERTER ON light will illuminate. The copilot's three-phase inverter switch, mounted on the copilot's instrument panel, is a two-position switch controlling operation of the copilot's three-phase instrument inverter. In the ON position, the copilot's and spare inverters are permitted to warm up. The warm-up period requires 60 seconds, during which time a red warning light (figure 1-58), labelled INSTRUMENT POWER FAILURE adjacent to the copilot's inverter switch will illuminate. After the warm-up period, the copilot's three-phase inverter is energized and the co-pilot's INSTRUMENT POWER FAILURE warning light will go out. Should the copilot's three-phase inverter output be unsatisfactory, the copilot's INSTRUMENT POWER FAILURE warning light will remain on, and after four seconds of operation, an automatic changeover relay will select the spare inverter to supply the copilot's instrument power requirements provided the spare inverter is not being used to supply the pilot's instruments.

Note

> The spare inverter will at all times fill pilot instrument requirements in event of failure of both pilot's and copilot's inverters.

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* *AF 54-552 thru 54-706*
** *AF 54-707 and Subsequent*

AC ELECTRICAL
three-phase power supply

(AF 54-552 thru 54-706)

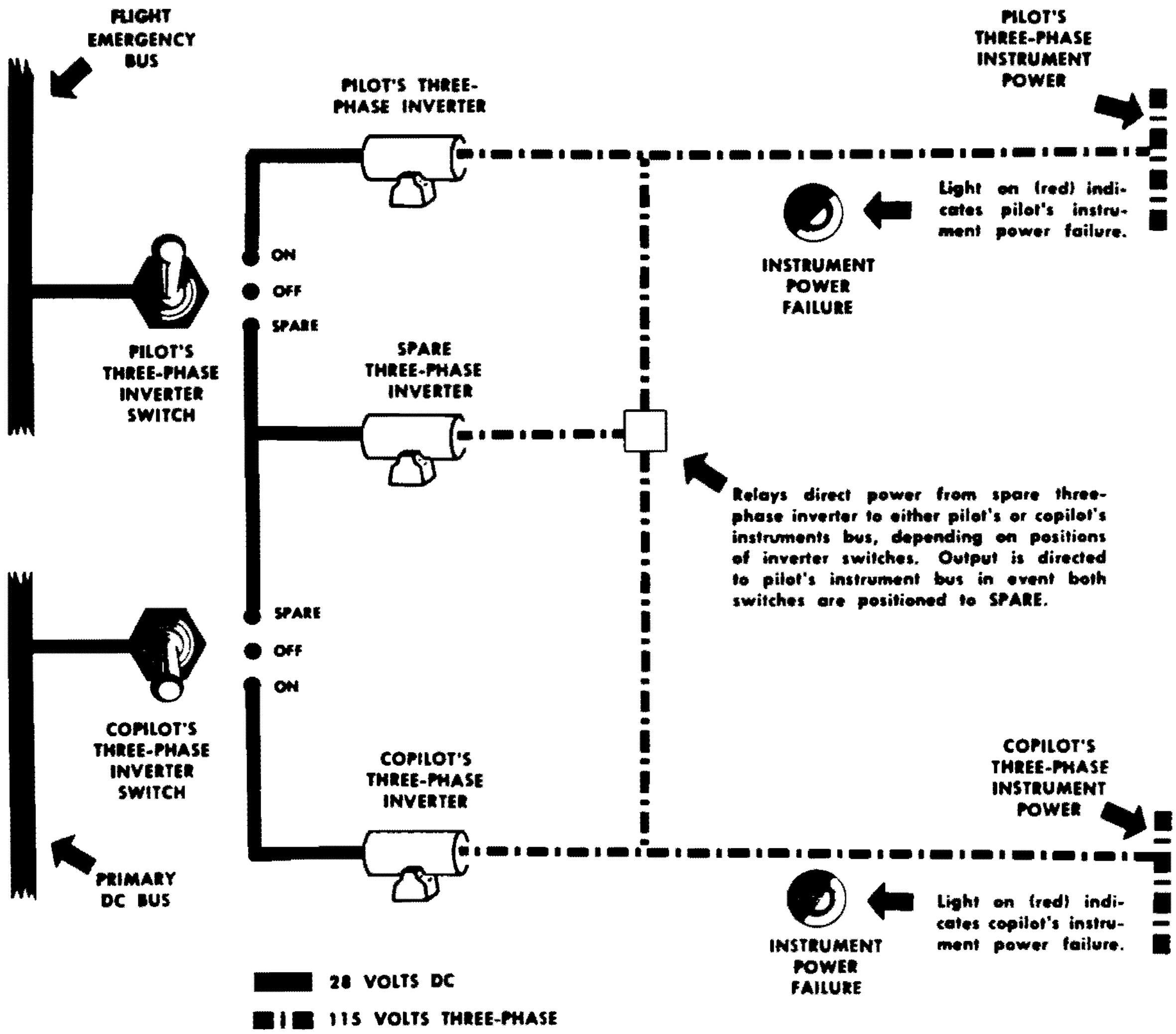


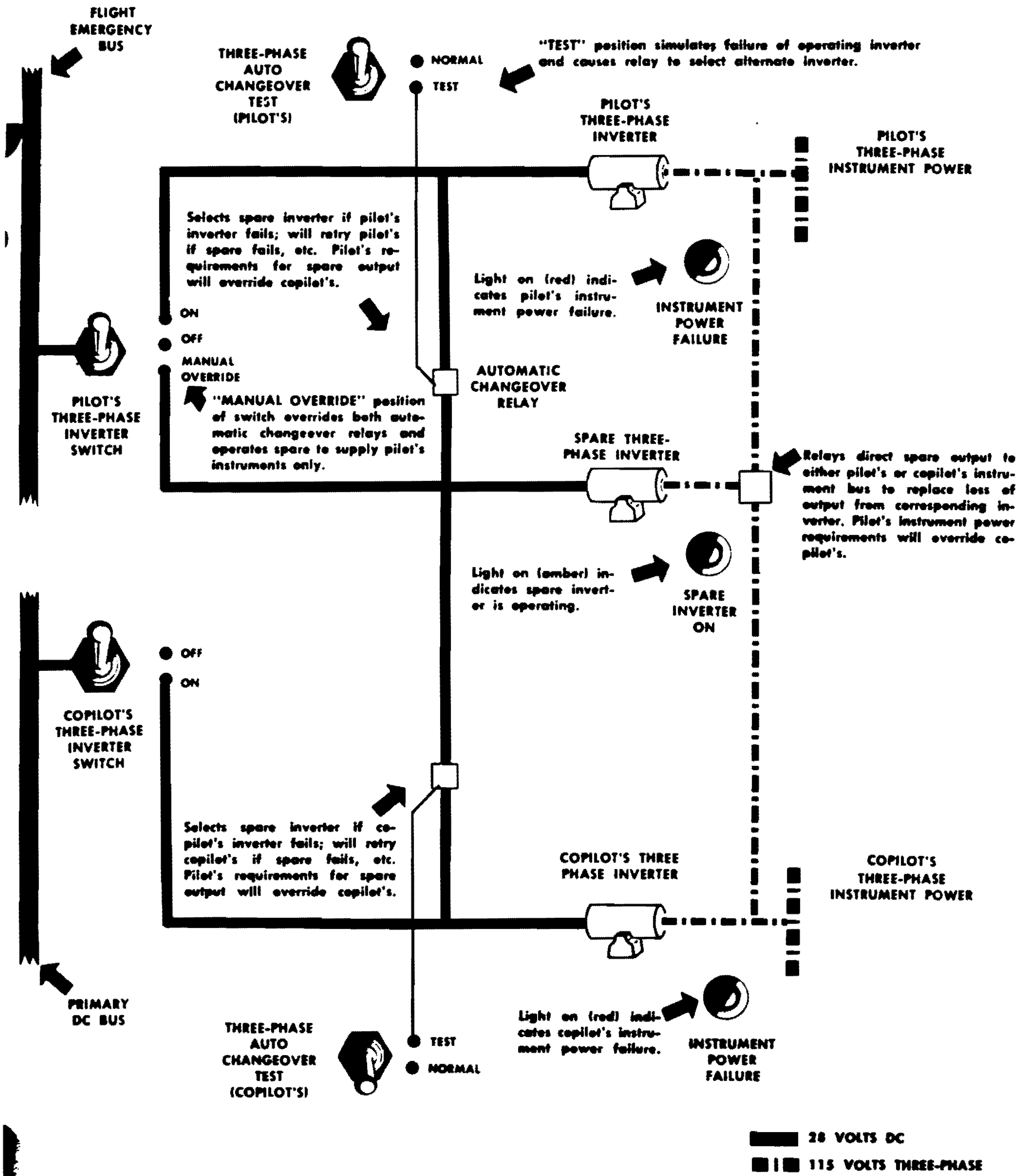
Figure 1-35

If the spare inverter is available for supplying the copilot's instruments, the copilot's INSTRUMENT POWER FAILURE warning light will go out and the SPARE INVERTER ON indicator light will illuminate, indicating that the spare three-phase inverter is supplying power to the copilot's instruments. Should the voltage output of the spare three-phase inverter become unsatisfactory, or should the spare inverter be required to supply power for the pilot's instruments, the copilot's INSTRUMENT POWER FAILURE warning light will illuminate, and the automatic change-over relay will again try the copilot's inverter. The SPARE INVERTER ON light will then go out, unless the spare inverter is operating to supply power for the pilot's instruments.

Note

If the spare inverter is being utilized to meet pilot instrument requirements, the copilot's INSTRUMENT POWER FAILURE light will remain illuminated and power for that system will not be available.

This cycling will continue until a satisfactory inverter output is found. During the cycling, the copilot's INSTRUMENT POWER FAILURE warning light will remain illuminated, while the SPARE INVERTER ON light will either remain illuminated, indicating that the spare inverter is operating to supply power for the pilot's instruments, or will alternately illuminate for four



AC ELECTRICAL

three-phase power supply

(AIRCRAFT AF 54-707 and SUBSEQUENT)

Figure 1-36

seconds and be out for four seconds. The pilot's INSTRUMENT POWER FAILURE warning light uses power from the flight emergency bus; the copilot's INSTRUMENT POWER FAILURE warning light uses power from the primary bus; the SPARE INVERTER ON light uses flight emergency bus power when the spare three-phase inverter is operating to supply power for the pilot's instruments and primary bus power when the spare three-phase inverter is operating to energize the copilot's instruments. For equipment affected by loss of three-phase power, refer to INVERTER FAILURE, SECTION III.

Three-phase Inverter Automatic Changeover Test Switches. *

Two, two-position switches (figure 1-58), one on the pilot's instrument panel and one on the copilot's instrument panel (figure 1-58), provide a means to test the pilot's and copilot's automatic changeover relays. In the NORMAL position, the switches are inoperative. Holding the pilot's switch to the momentary-contact TEST position for four seconds will simulate failure of the inverter operating to supply the pilot's instruments, and will cause the automatic changeover relay to shift to the alternate power source. Similar operation of the copilot's switch will cause similar response of the copilot's automatic changeover relay. In event the spare inverter is supplying the pilot's instruments when the copilot's switch is held to TEST, the automatic changeover relay will unsuccessfully attempt to switch to the spare inverter, but, will cycle back to the copilot's inverter after four seconds. Proper operation of the changeover relays in response to the test switches will be indicated by interpreting the operation of the three-phase inverter warning and indicator lights.

Secondary Bus Switch.

A guarded, two-position, secondary bus switch (figure 1-58), is mounted on the copilot's instrument panel. The switch positions are NORMAL and MONITOR. In the NORMAL position of the switch, the secondary ac bus is automatically energized by primary ac bus whenever two or more reciprocating engine generators are operating. Loss of output from all but one of the engine generators will automatically cause the secondary ac bus to be deenergized. However, should operation of the equipment supplied by the secondary bus become necessary, the secondary bus may be reenergized by positioning the switch to MONITOR, thus bypassing the automatic disconnect feature. The secondary ac bus supplies 115-volt ac single-phase power for the radar altimeter, TACAN, glide slope, and liaison sets.

* *AF 54-707 and Subsequent*

Windshield Heat Inverter.

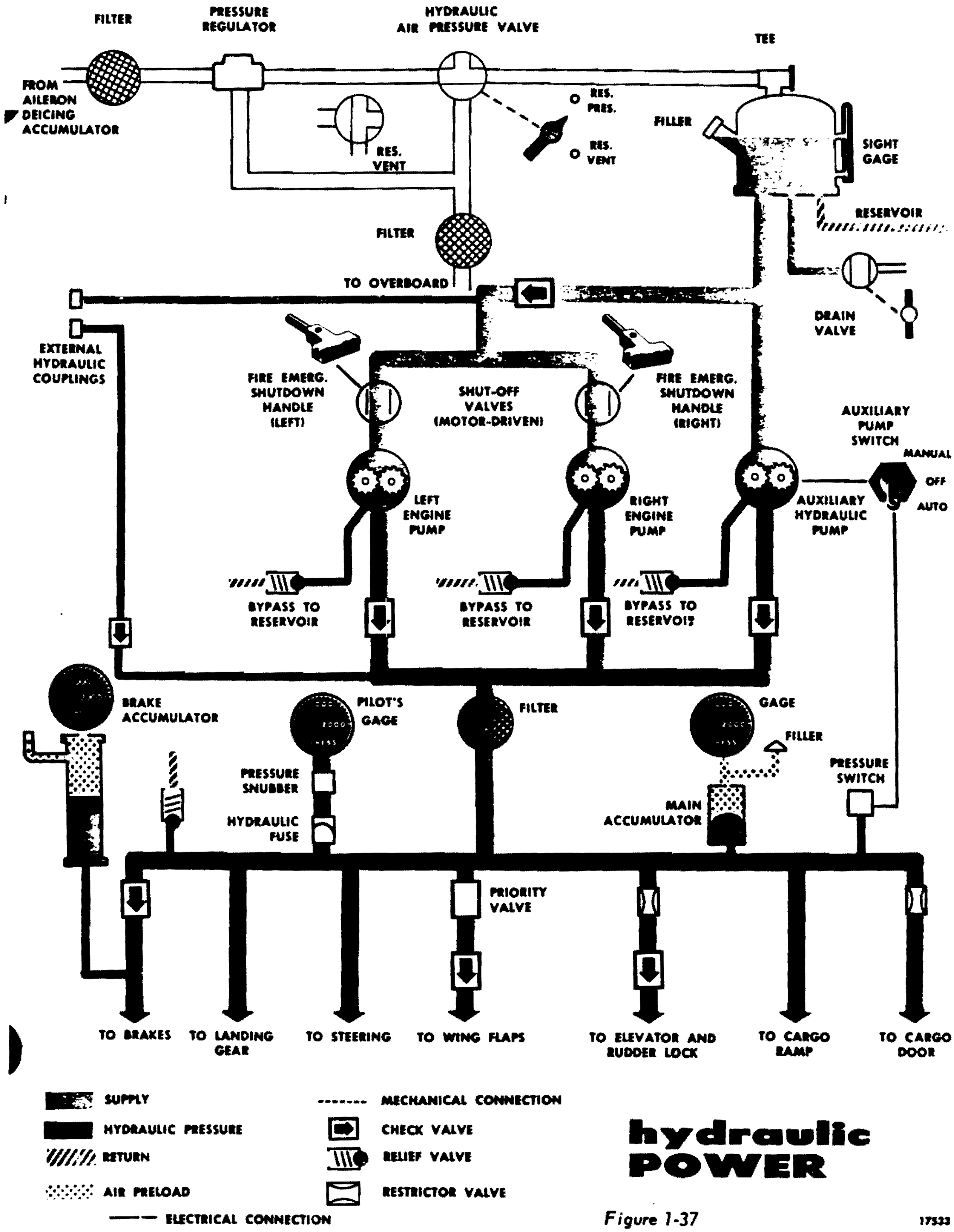
A 115-volt ac, 400-cycle, three-phase inverter is installed in the right side of the aircraft and aft of the right troop door. This inverter is controlled by the windshield heat switch which energizes the inverter from the primary dc bus. The operating inverter supplies 115-volt ac, 400-cycle, single-phase power to the pilot's and copilot's windshield heaters. For additional information on windshield heat, refer to Section IV.

HYDRAULIC SYSTEM.

Hydraulic pressure is used as the medium by which the landing gear, wing flaps, nose wheel steering, main gear brakes, rudder and elevator lock, and cargo door and ramp systems are operated.

SYSTEM COMPONENTS.

Hydraulic supply consists of a three-gallon capacity reservoir located above the right wheel well in th cargo compartment. A sight gage and filler point a provided on the reservoir for determining the flui level and replenishing as required. The reservoir contains an integral filtering system and is vented to the atmosphere. The total capacity of the hydraulic system is 11.5 gallons. Refer to Servicing, Figure 1-62, for hydraulic oil specification. Pressurization of the hydraulic system is normally accomplished by the operation of two variable-volume pumps, one driven by each of the engines. Hydraulic fluid, drawn from the reservoir by the pumps, is directed under pressure through a line filter and check valves to the main pressure line. A compressed air, preloaded accumulator in the main pressure line supplies initial surge pressure requirements when any of the hydraulically operated systems are employed. A separate accumulator and check valve in the brake system assures a supply of pressure for brake application and prohibits dissipation of brake system reserve pressure by othe systems. Relief valves are utilized to prevent the build-up of abnormal pressures in the system. An automatic pressure switch in the main pressure line controls operation of the auxiliary hydraulic pum whenever automatic operation of the pump is select The 28-volt dc motor-driven auxiliary pump may used to augment system pressure during emergencies or to pressurize the hydraulic system on the ground External hydraulic couplings are provided to permit the use of a portable hydraulic pump during groun maintenance. Hydraulic fluid supply to the correspond ing engine pump is shut-off when the fire emergency shut-down handle is pulled (figure 1-57).



hydraulic POWER

Figure 1-37

The reservoir is pressurized with air from the aileron deicing accumulator to prevent loss of prime of the hydraulic pumps. This air is filtered and regulated, then fed through a two-position selector valve to the top of the reservoir. With this valve in the RES. PRES. position, the reservoir is pressurized any time the aileron deicing compressor is operating; however, when filling the reservoir or in the event of a malfunction of the pressurizing system, the valve should be placed in the RES. VENT position, thereby venting the reservoir to atmosphere through a filter.

CAUTION

Before removing the reservoir filler cap, the selector valve (hydraulic air pressure) must be placed in the RES. VENT position to prevent escape of hydraulic fluid.

Hydraulic Reservoir Drain Valve.

A two-position, manually-operated drain valve is included in the drain line immediately beneath the hydraulic reservoir. Normally, the valve remains in the CLOSED position. In the OPEN position, the hydraulic reservoir will drain through a line which ports beneath the aircraft aft of the right main gear well.

Auxiliary Hydraulic Pump Switch.

A 28-volt dc motor-driven auxiliary hydraulic pump is located on the hydraulic power panel directly above the right main gear wheel well. Operation of the pump is controlled by a three-position toggle switch (figure 1-7), located on the engine starting panel. The switch positions are AUTO, OFF, and MANUAL. When the switch is in the OFF position, the pump is inoperative. In the AUTO position of the switch, the pump is permitted to operate in response to the automatic pressure switch in the main hydraulic pressure line. When the switch is held in the momentary-contact MANUAL position, the pump is energized directly - bypassing the pressure switch-and operates continuously.

Pressure Switch.

A 28-volt dc pressure switch (figure 4-28), mounted adjacent to the main hydraulic system accumulator above the right main gear wheel well automatically controls the auxiliary hydraulic pump in response to the pressure in the main hydraulic system whenever the auxiliary hydraulic pump switch is in the AUTO position. The switch turns the auxiliary pump on whenever pressure falls below 1250 psi and turns the pump off whenever the pressure reaches 1400 psi.

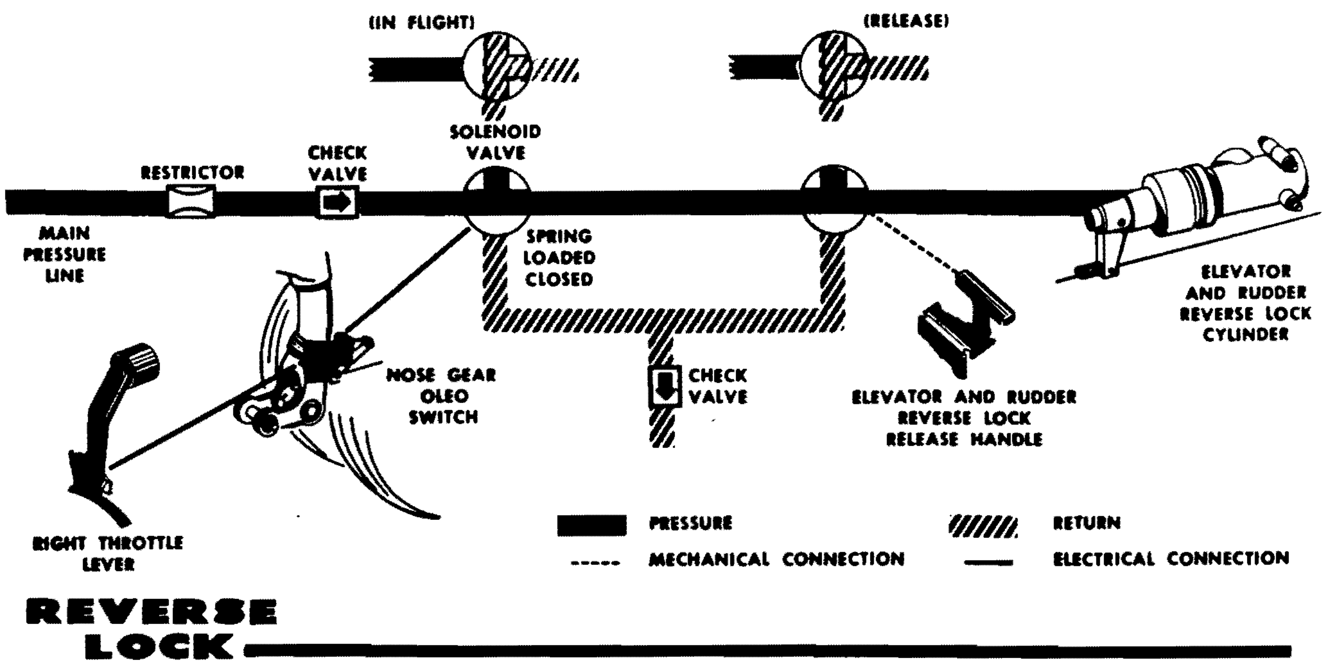


Figure 1-38

Hydraulic Pressure Gage.

A pressure-operated gage (figure 1-58), on the pilot's instrument panel indicates main hydraulic system pressure in pounds per square inch (psi). The gage is protected by an orifice-type pressure snubber which prevents surge pressure differences from damaging the gage. A hydraulic fuse, which permits transfer of pressure but which closes when the flow rate exceeds a few drops per minute, prevents vapor flooding of the crew compartment by hydraulic fluid in the event a rupture of the pressure line to the gage should occur.

Note

> Lack of hydraulic pressure indication on the hydraulic pressure gage does not indicate complete loss of pressure for brake application since brake accumulator pressure is isolated by a check valve. Refer to Figure 1-50, Hydraulic Brakes.

Main Accumulator Gage.

A pressure-operated gage (figure 4-28), above the right main wheel well indicates in pounds per square inch (psi) the air pressure existing in the main system accumulator. During normal operation, when the system is pressurized, this gage should read the same as the gage on the pilot's instrument panel. When the hydraulic system is shut down, the accumulator gage will continue to indicate the preload pressure to which the accumulator was charged. Refer to Figure 5-1, Instrument Range Markings, for accumulator preload pressure.

Hydraulic Shut-off Valves.

Flow of hydraulic fluid to the engine-driven pumps may be shut off at the firewall in each nacelle by a motor-driven valve. These valves operate on 28-volt dc power from the flight emergency bus, and are controlled by the corresponding fire emergency shutdown handle (figure 1-56). When the handle is pulled, the valve closes, and remains closed until the handle is pushed back in.

External Hydraulic Power Couplings.

To permit the use of a portable hydraulic pump during ground maintenance, external power couplings (figure 1-62), are provided which connects to the supply and pressure lines of the aircraft's hydraulic system. The external couplings are located behind a hinged access door on the right side of the fuselage.

Pressure Regulator.

A pressure regulator is mounted at the top of the hydraulic reservoir compartment to reduce and regulate the air supplied by the aileron deicing accumulator to the hydraulic reservoir. The regulator also contains a vacuum relief valve that will maintain atmospheric pressure in the reservoir if a malfunction of the air supply occurs or if there is a sudden drop in air pressure.

Hydraulic Air Pressure Valve.

A two-position manual selector valve is located on the forward side of the hydraulic reservoir compartment with the positions placarded RES. PRES. and RES. VENT. With the valve in RES. PRES. position, air pressure from the aileron deicing system is directed to the hydraulic reservoir. When in RES. VENT, air from the deicing system accumulator is shut off and the reservoir is vented to atmosphere. The valve should remain in the RES. PRES. position for all normal use and positioned to RES. VENT only in the event of an air supply malfunction or when servicing the reservoir.

FLIGHT CONTROLS SYSTEM.

Two conventional control columns and rudder pedals operate the primary flight control surfaces. Rudder, elevator and aileron trim tabs are actuated manually from the control pedestal. Spring tabs are incorporated into the elevator and rudder surfaces to reduce control system loads and assist pilot effort during flight. A flight controls lock, when applied, will simultaneously lock all the flight surfaces. Automatic locking of the rudder and elevator during reversing is also provided. The ailerons are equipped with pneumatically operated deicing boots.

Note

> The trim tab setting should be checked with no cargo, fuel evenly divided in the right and left nacelle tanks and at an airspeed of 145 ±5 knots in straight and level hands off flight. The following values apply:
>
> a. Aileron Tab ±5 degrees.
>
> b. Rudder Tab ±3 degrees.
>
> c. Elevator Tab ±10 degrees.

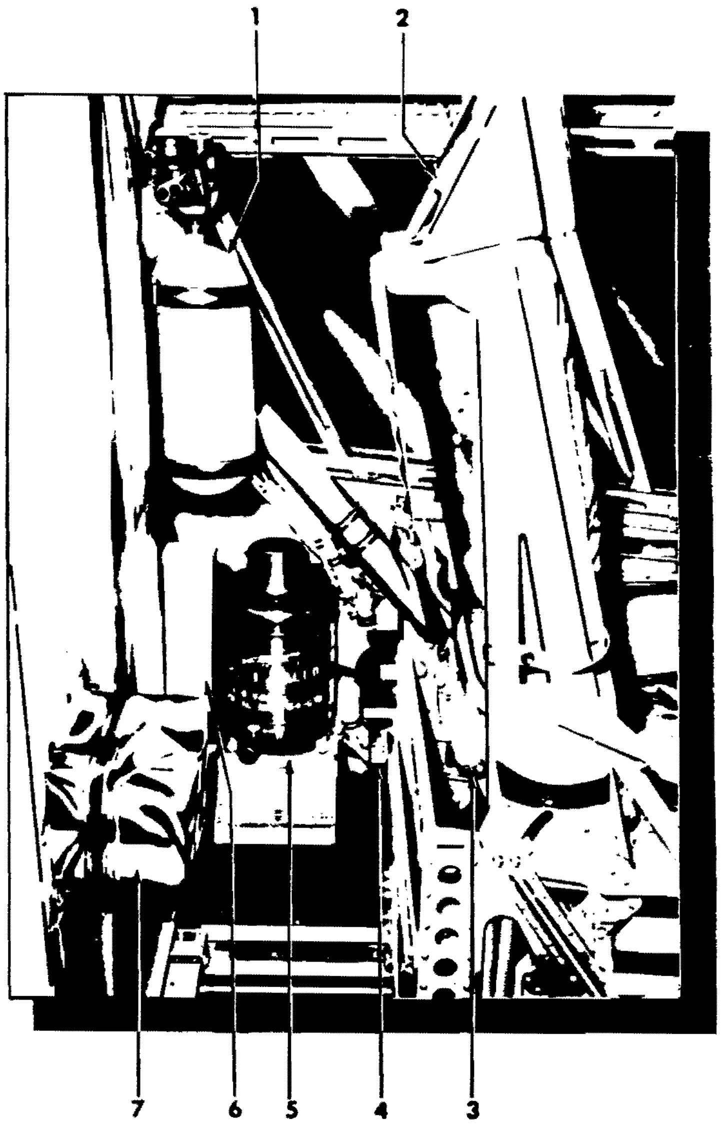
AILERON CONTROLS.

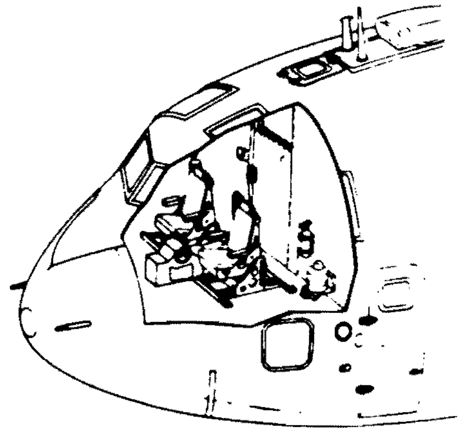
Control wheels for pilot and copilot and associated cable assembly are used to effect mechanical aileron control. A balanced, slotted-type, fabric or fiberglas covered aileron is located in each wing between the outboard flap and the wing tip. Full deflection of the ailerons is obtained by 135-degree rotation of the control wheels each side of the neutral position. Positive stops are provided so that travel beyond limits is restricted.

cockpit . . . . . ARRANGEMENT

pilot's

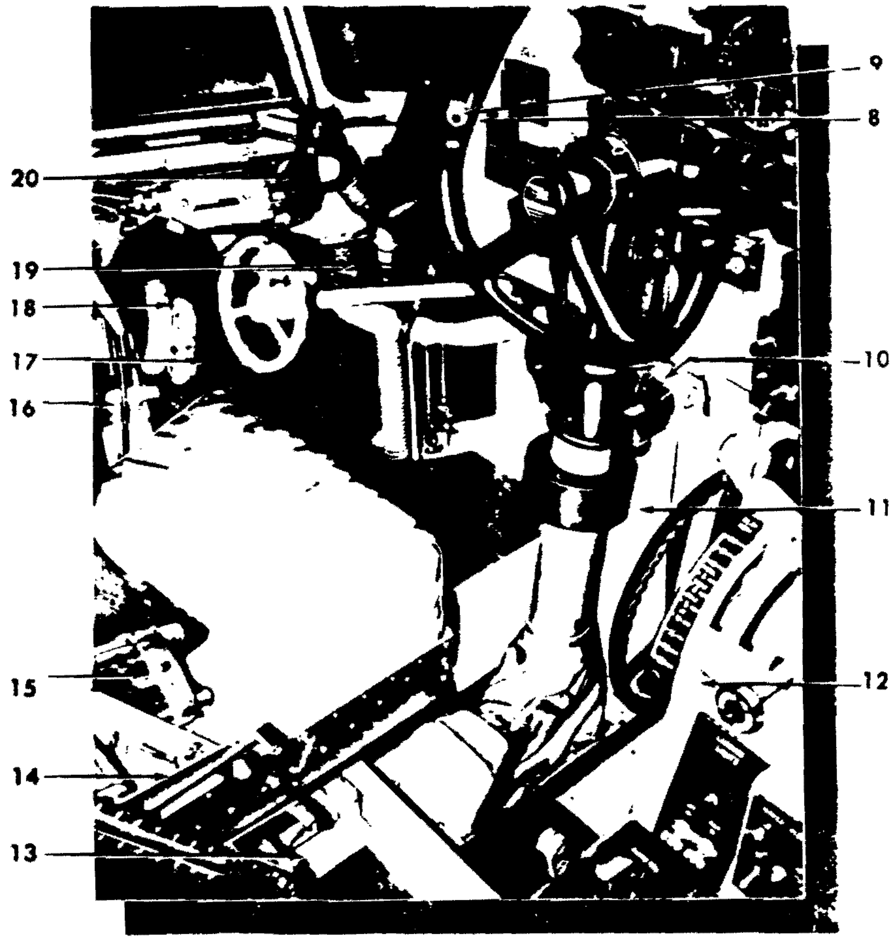
—typical—





1. PORTABLE OXYGEN BOTTLE (PILOT)
2. PILOT'S SEAT
3. INERTIA REEL LOCK CONTROL (PILOT)
4. ALDIS LAMP
5. DRINKING WATER CONTAINER
6. DRINKING CUP DISPENSER
7. FIRST AID KIT
8. MICROPHONE SWITCH (PILOT)
9. INTERPHONE SWITCH (PILOT)
10. RUDDER PEDALS ADJUSTMENT KNOB (PILOT)
11. CONTROL COLUMN SHAKER
12. CONTROL QUADRANT
13. SEAT VERTICAL ADJUSTMENT LEVER (PILOT)
14. SEAT RECLINING ADJUSTMENT LEVER (PILOT)
15. SAFETY BELT (PILOT)
16. ELEVATOR AND RUDDER REVERSE LOCK RELEASE HANDLE
17. NOSE STEERING WHEEL
18. HEAD SET (PILOT)
19. UPPER SIDE WINDOW DEFROSTING VALVE (PILOT)
20. EMERGENCY AIR BRAKE VALVE HANDLES AND GUARD
21. DILUTER DEMAND OXYGEN REGULATOR (PILOT)
22. INTERPHONE RANGE FILTER PANEL (PILOT)
23. CONSOLE LIGHTS RHEOSTAT (PILOT)
24. SIDE WINDOW LOCKING HANDLE (PILOT)
25. LOWER WINDOW DEFROSTING VALVE
26. INERTIA REEL LOCK HANDLE (PILOT)
27. FLIGHT CONTROLS LOCK LEVER
28. ALDIS LAMP FILTER CASE
29. SEAT FORE AND AFT ADJUSTMENT LEVER (PILOT)
30. FIRE FIGHTER'S SMOKE MASK
31. INTERPHONE CONTROL PANELS (PILOT)

Figure 1-39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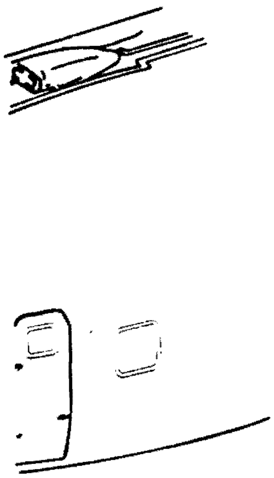

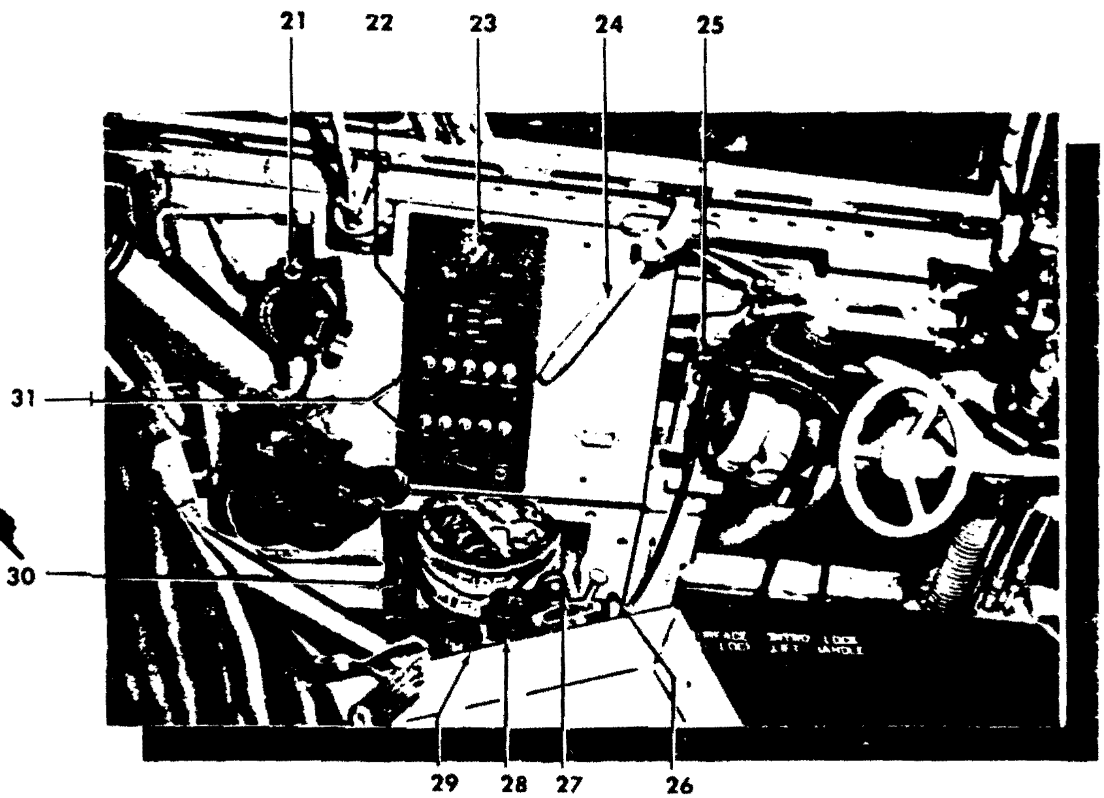
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cockpit . . . . . ARRANGEMENT

copilot's

—typical—

1. COPILOT'S SEAT
2. PORTABLE OXYGEN BOTTLE (COPILOT)
3. APU CONTROLS
4. SPARE LAMPS BOX
5. VHF COMMAND TRANSMITTER
6. CREW COMPARTMENT ENTRANCE CURTAIN
7. VHF COMMAND RECEIVER
8. HAND FIRE EXTINGUISHER
9. FLARE STOWAGE BAG
10. PYROTECHNIC PISTOL
11. INERTIA REEL LOCK
12. SIDE WINDOW LOCKING HANDLE (COPILOT)
13. CONSOLE LIGHTS RHEOSTAT (COPILOT)
14. INTERPHONE RANGE FILTER PANEL (COPILOT)
15. INTERPHONE CONTROL PANELS (COPILOT)
16. DILUTER DEMAND OXYGEN REGULATOR (COPILOT)
17. ALDIS LAMP POWER RECEPTACLE (COPILOT)
18. HEADSET (COPILOT)
19. FIRE FIGHTER'S SMOKE MASK
20. SEAT RECLINING ADJUSTMENT LEVER
21. SIDE WINDOW HANDLE (COPILOT)
22. MICROPHONE SWITCH (COPILOT)
23. UPPER SIDE WINDOW DEFROSTING VALVE (COPILOT)
24. RUDDER PEDALS ADJUSTMENT KNOB (COPILOT)
25. INTERPHONE SWITCH (COPILOT)
26. VHF/FM RADIO CONTROL PANEL FM-622A (AIRCRAFT WITH T.O. 1C-123-589)
27. SPEECH ENCRYPTION CONTROL PANEL TSEC/KY-8 (AIRCRAFT WITH T.O. 1C-123-584)

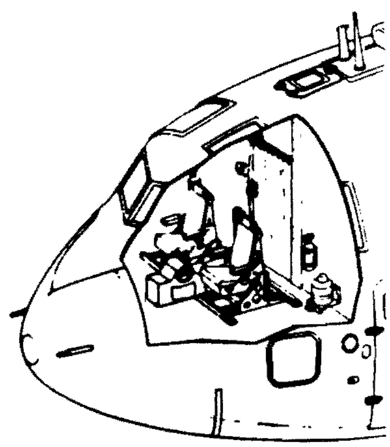
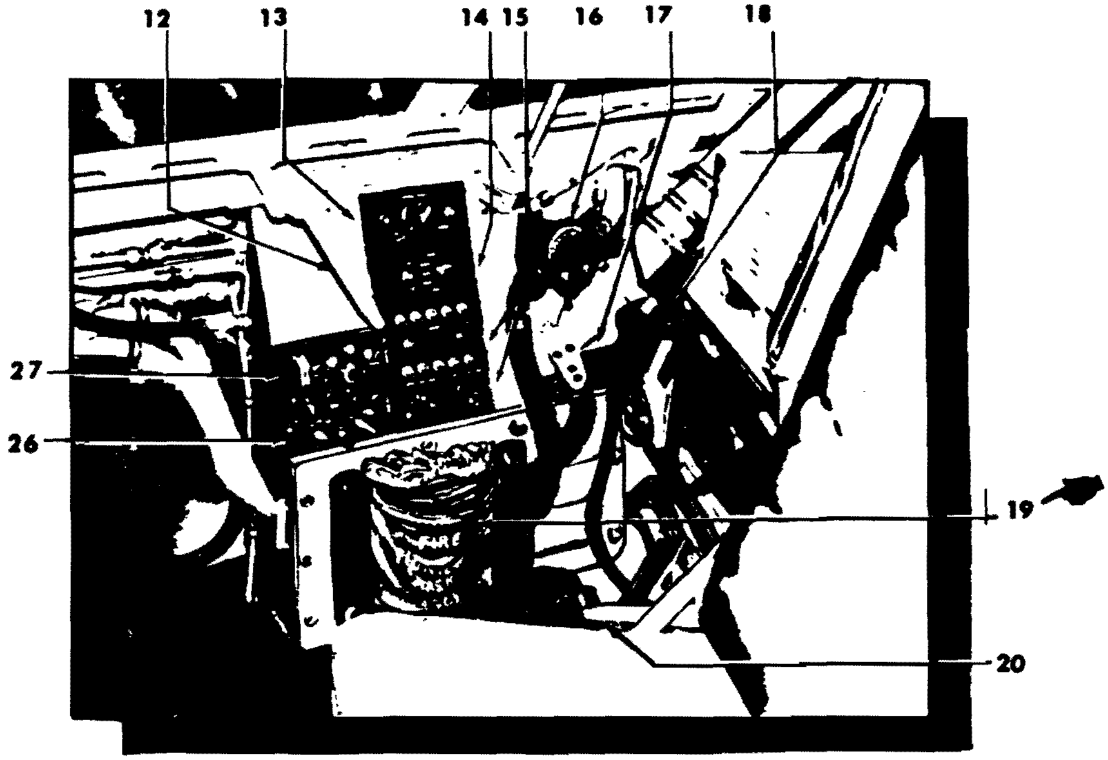


Figure 1-40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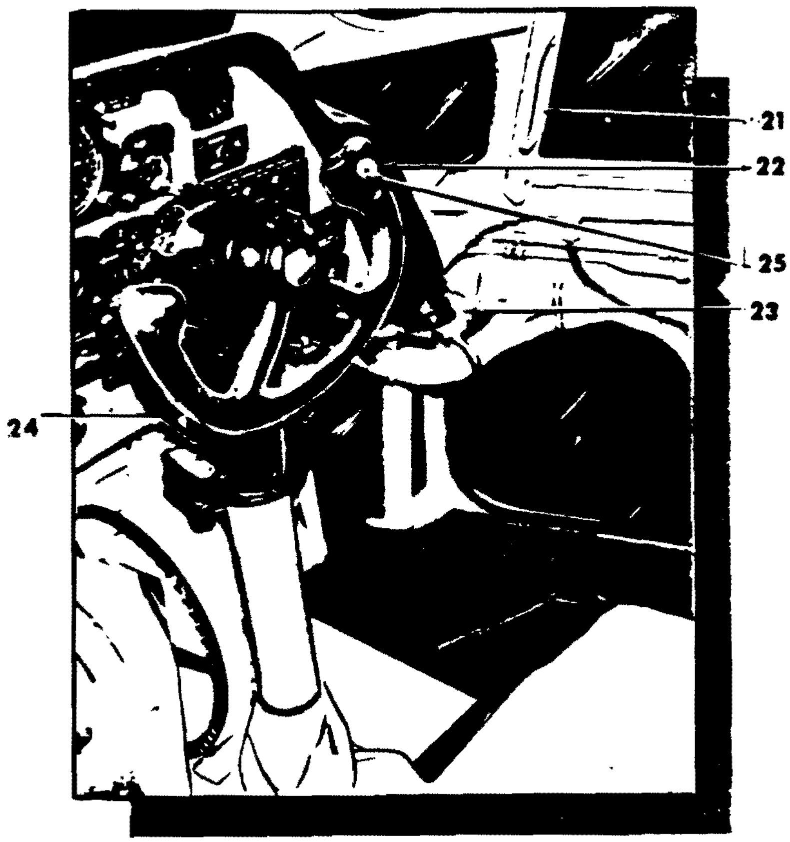
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Aileron Trim Tab Wheel And Indicator.

An all-metal aileron trim tab, which provides aileron adjustment effecting lateral balance of the aircraft, is located on the left aileron. The aileron trim tab wheel (figure 1-4), is located on the pilots' control panel. Trim tab position is indicated mechanically by a pointer immediately above the tab-control wheel.

ELEVATOR CONTROLS.

Dual, fabric or fiberglas covered elevator surfaces at the trailing edge of the stabilizer control the aircraft about its lateral axis. Fore and aft movement of the control column controls the positioning of the elevators. Full-up elevator is obtained by approximately ten inches of aft control column movement and full-down elevator by approximately six inches of forward control column movement. Positive stops are provided so that travel of the elevator control system beyond prescribed limits is restricted. Each elevator surface incorporates a metal spring tab located inboard of each elevator trim tab. The elevator spring tabs operate automatically with control column movement assisting elevator movement during flight and lessening pilot control effort necessary for maneuvering.

Elevator Trim Tab Wheel And Indicator.

An all-metal trim tab is incorporated into each elevator surface. The elevator trim tab wheels (figure 1-3), are installed vertically on each side of the control quadrant. The control wheels are mechanically interconnected. The elevator trim tab indicator located on the pilot's side of the quadrant indicates the position of the tabs in degrees of deflection. and is accurate within $\pm$ 3 degrees.

RUDDER CONTROLS.

The fabric or fiberglas covered rudder hinged to the trailing edge of the vertical fin is the means by which the aircraft is controlled about its vertical axis. Each set of rudder pedals can be adjusted to accommodate pilots of different stature. Positive stops in the system prevent rudder travel beyond prescribed limits. A metal spring tab located on the lower trailing edge of the rudder operates automatically with rudder deflection and lessens the effort necessary to move the rudder surface during flight.

Rudder Pedal Adjustment Knobs.

A rudder pedal adjustment knob (figures 1-39 and 1-40), mounted above each set of rudder pedals, mechanically varies the position of the pedals to accommodate pilots of different stature. Turning either adjustment knob clockwise will move its respective set of rudder pedals aft; counterclockwise adjustment will move the pedals forward.

Rudder Trim Tab And Indicator.

An all-metal rudder trim tab installed on the trailing edge of the rudder surface immediately above the spring tab provides a means of trimming the aircraft directionally. The action of the tab enables the pilot to obtain the desired directional stability of the aircraft without exerting pressure on the rudder pedals. Rudder trim tab control is accomplished by turning a trim tab wheel (figure 1-4) on the pilot's control panel. Rudder trim tab deflection is limited by a stop arrangement within the rudder trim tab assembly. Trim tab position indication is afforded by a mechanical indicator adjacent to the control wheel.

FLIGHT CONTROLS LOCK.

A mechanically operated flight controls lock is provided to prevent movement of the control surfaces when the aircraft is parked.

Flight Controls Lock Lever.

The flight controls lock system is a manually operated, cable-controlled system actuated by a lever (figure 1-39), on the crew compartment floor at the left of the pilot's seat. When the flight controls lock lever is pulled up and aft, sector brakes mounted adjacent to each control surface engage lock sectors attached to the aileron, rudder, and elevator surfaces. Positive lock of the surfaces may be checked by applying pressure to the control wheel and rudder pedals. The flight controls locks are released by returning the locking lever to its full forward position. To insure that the flight controls are unlocked prior to any attempt at flight operation, a mechanical linkage between the flight controls lock lever and throttles is provided. If the lock lever is in the LOCKED position, a stop arrangement on the control quadrant prevents the run-up of both engines simultaneously to Maximum Power. Refer to FLIGHT CONTROLS THROTTLE STOP, this section.

Note

> The flight controls lock may be applied with the controls in any position. Normally, however, application will be made with the flight controls in neutral. Because of mechanical linkage between the throttles and flight controls lock, the throttles should be retarded prior to the application of the lock.

REVERSE LOCK
. . . elevator & rudder

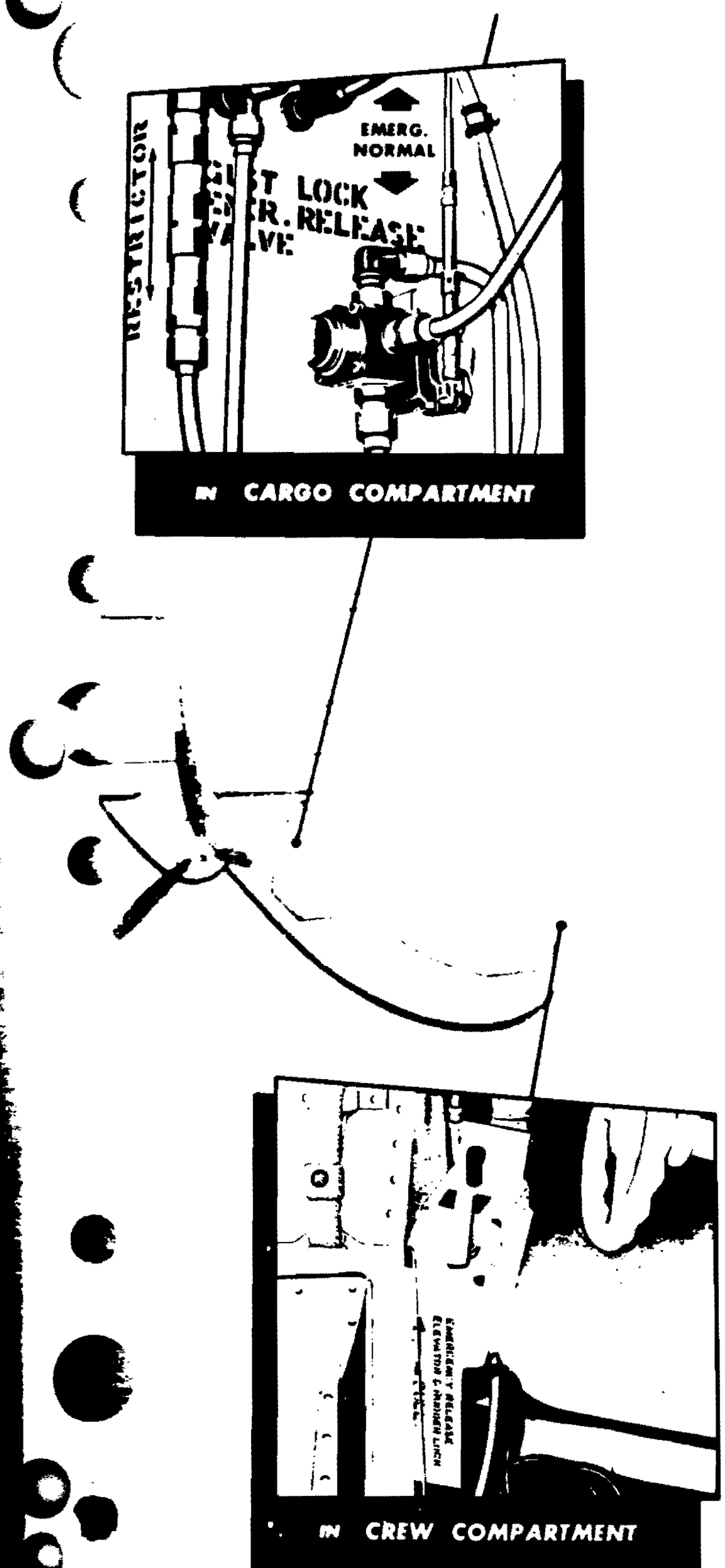


Figure 1-41

REVERSE LOCK.

Automatic locking of the rudder and elevators during reverse pitch operation is accomplished by moving the right throttle lever aft into REVERSE THRUST. Movement of the throttle lever operates a cam and torque tube arrangement. At approximately the same time as the propeller reversing circuits are energized, the cam actuates a switch which closes the circuit to a 28-volt dc solenoid valve. Energization of the solenoid valve directs hydraulic pressure to an actuator, the extension of which pulls the flight controls lock cable (aft of the aileron locks attachment) and applies the locks at the rudder and elevator surfaces. When the right throttle is returned to the forward thrust position, cam and switch action is such that the solenoid valve circuit is not deenergized until engine manifold pressure reaches approximately 35 inches of mercury. This permits ground checks and taxiing to be accomplished under gusty conditions, while the rudder and elevator surfaces are locked.

Note

An oleo switch on the nose gear strut permits the elevator and rudder reverse lock to be engaged only when the gear is extended and the strut compressed.

Reverse Lock Release Handle.

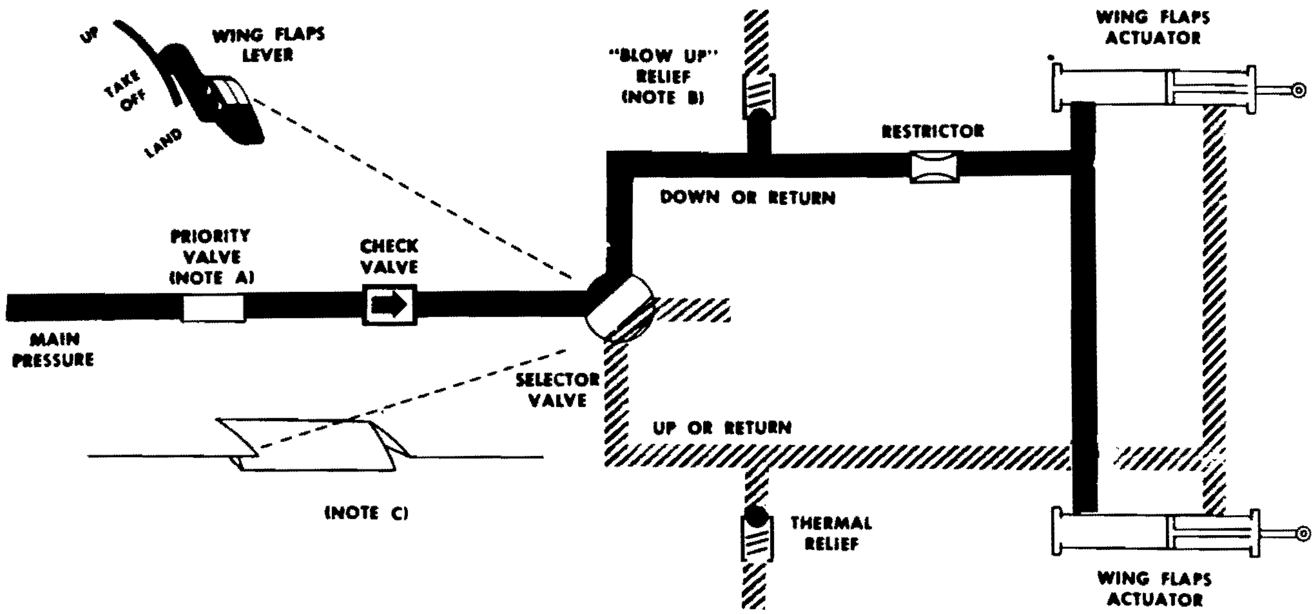
A means of manually releasing the elevator and rudder reverse lock is provided for the pilot. Should the lock fail to release or inadvertently engage during flight, the pilot pulls the emergency release handle (figure 1-39 and 1-41), located at the left of the pilot's seat, which opens the release valve and releases lock actuator pressure, thereby freeing the elevator and rudder lock.

Reverse Lock Release Valve.

A valve located above the right main landing gear well is incorporated into the hydraulic portion of the elevator and rudder reverse lock system to release the locks should they fail to disengage or inadvertently engage during flight. The valve (figure 4-28) may be operated manually from the cargo compartment or by pulling the reverse lock release handle at the pilot's station. The valve actuating lever in the cargo compartment has both NORMAL and EMERGENCY positions. The lever is down when the valve is in the NORMAL position and up when it is in the EMERGENCY position. The valve must be manually repositioned back to NORMAL if it has been positioned to EMERGENCY.

WING FLAPS SYSTEM.

The wing flaps are composed of four mechanically interconnected single-slotted type sections. Two



NOTES . . . .

A. The priority valve is provided to assure sufficient pressure and flow for adequate nose gear steering while raising flaps on the ground.

B. The "blow up" relief valve in the down line prevents excessive air loads. Refer to WING FLAPS, Section I.

C. Selector valve is closed by mechanical follow-up when flaps reach desired position.

WING FLAPS

Figure 1-42

panels on each wing extend from the inboard side of the aileron to the fuselage less the nacelle area. Retraction and extension of the wing flaps are accomplished by hydraulic pressure which is controlled by the mechanical positioning of the wing flaps selector valve. Should the hydraulic system fail, the wing flaps system is inoperative, as no equipment for the emergency extension or retraction of the wing flaps is provided. Extension of the wing flaps to 45 degrees at maximum airspeed for wing flaps operation requires approximately 20 seconds; retraction requires approximately 14 seconds.

WING FLAPS LEVER.

The wing flaps lever (figure 1-3) is located on the engine control quadrant. Wing flap control is accomplished by moving the wing flap lever to the desired setting which, by a cable arrangement, positions a four-way selector valve. The selector valve directs hydraulic pressure to a pair of double-acting wing flap actuators which position the flaps. With the wing flaps lever positioned at UP, flap deflection is 0 degrees; TAKE-OFF, 19 (±2) degrees; LAND, 45 (±1) degrees. Intermediate flap settings may be selected. Detents are provided on the quadrant for UP, TAKE-OFF and LAND positions.

Note

> A "blow" up valve adjusted to 1050 psi has been installed in the wing flaps hydraulic system. If LAND position is selected and the air speed is increased to the maximum limit, the flaps may blow up to approximately 35 degrees.

FLAP RELIEF VALVE.

A "blow-up" relief valve prevents excessive flap air-loads by relieving hydraulic pressure from the "down" side of the wing flap system, thus allowing upward movement or stopping downward travel when the aircraft is over "flaps-down" speed. As the speed of the aircraft decreases and pressure within the hydraulic lines falls below the relief valve setting, the relief valve will reseat itself and allow the flap actuator to lower the flaps to the down position again, provided that the flap selector valve is retained in one of the flaps-extended positions.

FLAP PRIORITY VALVE.

A priority valve is installed in the wing flaps system to restrict operation of the wing flaps so that sufficient pressure and flow are assured for braking and nose wheel steering. When these three systems are operated simultaneously - a condition which frequently occurs after landing - the priority valve closes at anytime the pressure in the hydraulic system falls below 1100 psi. At pressures above 1100 psi, the priority valve permits flow in a restricted amount until 1300 psi in the hydraulic system is obtained. A full rate-of-flow in the wing flaps system requires 1300 psi or above. The priority valve is installed in the wing flaps pressure line just upstream of the wing flaps selector valve and functions during both extension and retraction of the wing flaps.

Flap Position Indicator.

A wing flap position indicator (figure 1-58 or 1-59), which provides continuous indication of wing flaps position in degrees of travel from the UP position, is located on the right side of the instrument panel. The indicator reflects flap travel as sensed by the transmitter mechanically connected to the wing flap interconnect torque tube. The following flap position indications are acceptable at detent positions: Up ±3 degrees; TAKE-OFF 19 (±2) degrees. In the LAND position, the pointer on the flap indicator should be on the edge of the DOWN flag ±3 degrees. Operating power is taken from the 28-volt dc bus.

LANDING GEAR SYSTEM.

The tricycle-type landing gear on the aircraft consists of a dual-wheel nose gear, which is steerable 60° in either direction, and a single wheel on each of the two main gears. The main struts are mounted on either side of fuselage, in a nearly vertical position, and are connected, by means of drag links, to the forward portion of the wheel wells. All three gear are operated simultaneously with the retraction requiring approximately nine seconds, and the extension, approximately six seconds. Electrical power for control of the hydraulically operated landing gear is supplied by the 28-volt dc flight emergency bus. In the event of an electrical power failure, the gear may be raised and lowered; however, when a hydraulic failure occurs, operation of the gear is limited to extension by free fall.

UPLOCKS AND DOWNLOCKS.

The main landing gear uplocks and downlocks are normally unlocked by the initial movement of the actuators through mechanical linkage. Locking action is accomplished by spring pressure. A single locking mechanism on the nose gear is employed as an uplock and down lock.

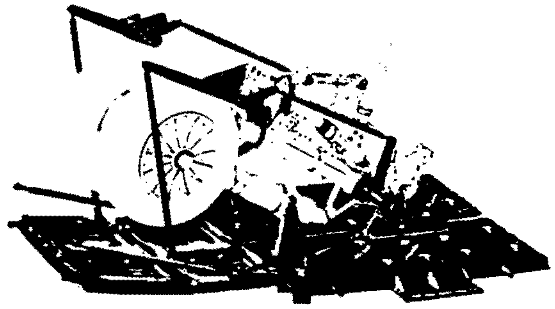
LANDING GEAR DOORS.

The main landing gear doors consists of one door for each gear which is mechanically connected by a swivel rod to the main gear strut. The doors open mechanically when the gear are extended and close when the gear retract. The nose gear doors (figure 1-43), are also mechanically operated through a system of bell-cranks and push-pull rods connected to nose gear trunnions. During the nose gear extension cycle all doors start opening, the front doors fold and swing forward, side doors swing down and out allowing clearance for the tire. During the later part of the extension cycle the front doors unfold and return to the normal position, the side doors continue outward to approximately parallel with the fuselage. During the nose gear retraction cycle the front doors fold and swing forward, the side doors start moving downward, the strut retracts pulling the side doors closed and the front doors unfold and return to the normal position. Ground clearance with the side doors open is approximately 19 inches.

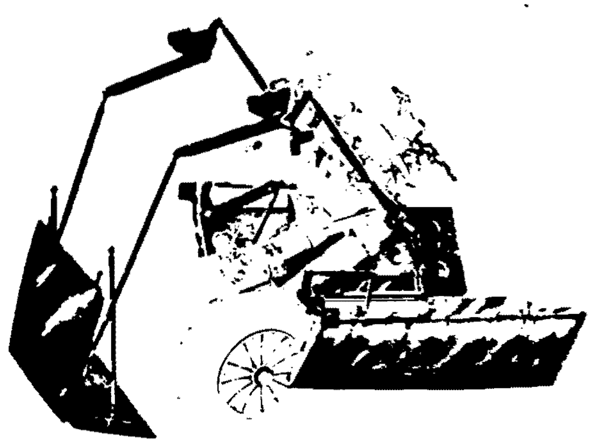
NORMAL OPERATION.

Retraction and extension of the landing gear is accomplished hydraulically by electrically positioning the landing gear hydraulic control valve. When the gear are placed in operation, hydraulic system pressure is automatically supplemented by the auxiliary hydraulic pump. The additional flow required is supplied through the action of a pressure switch.

DOOR OPERATION
nose gear



UP AND LOCKED



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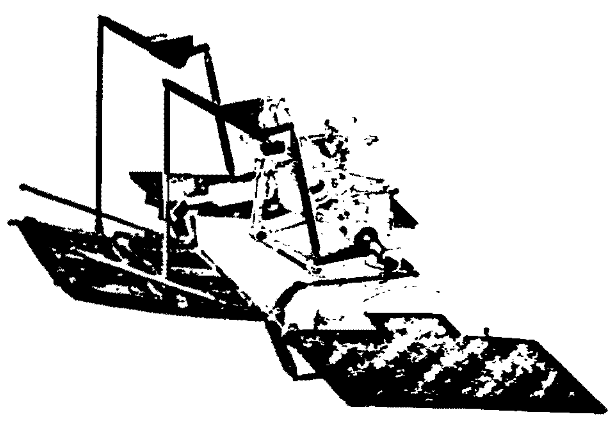

DOWN AND LOCKED

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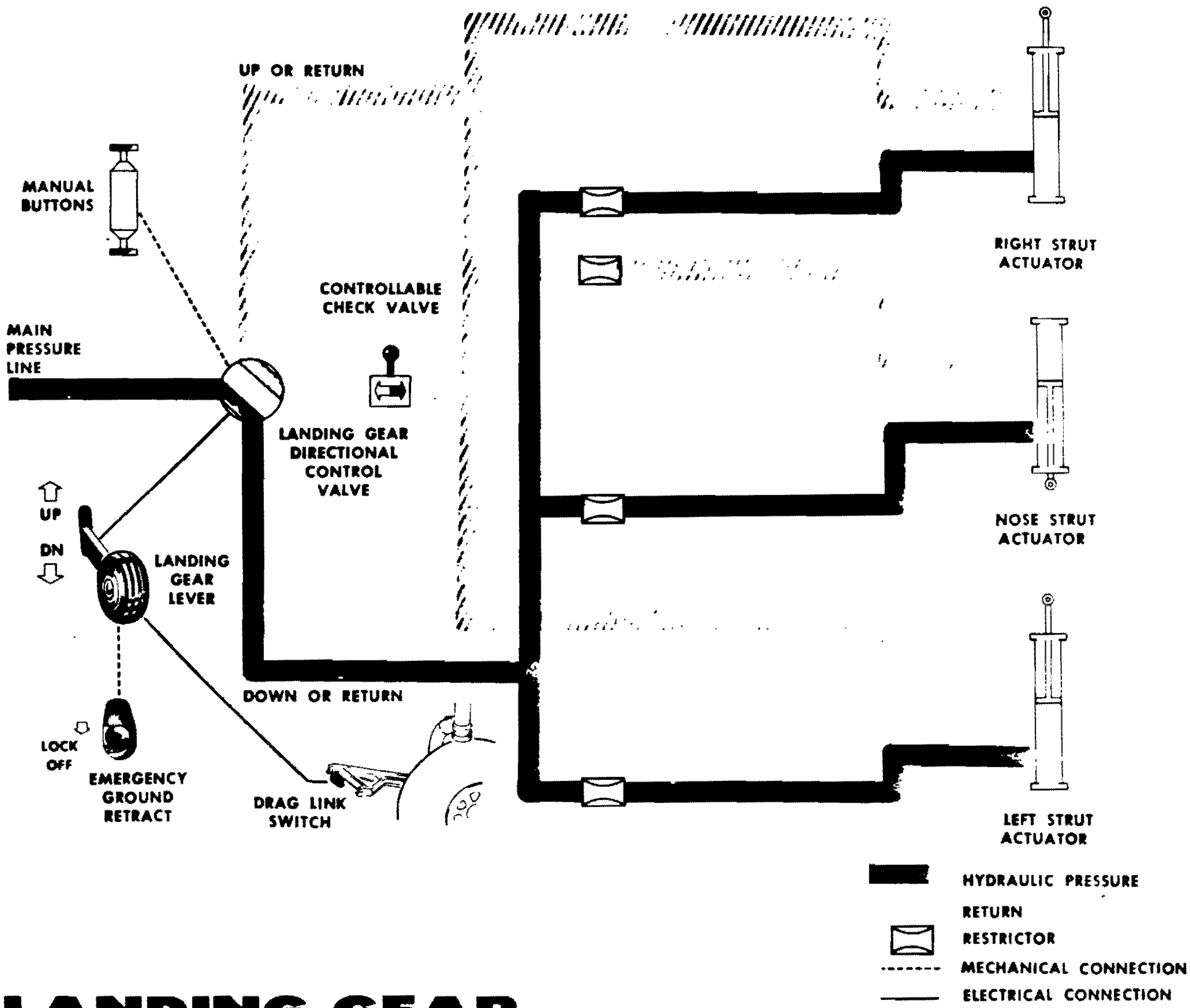
Figure 1-43



LANDING GEAR

Figure 1-44

Note

The pressure switch activates the auxiliary hydraulic pump only when the auxiliary hydraulic pump switch is in the AUTO position. Refer to HYDRAULIC SYSTEM, this section.

MAIN LANDING GEAR GROUND LOCK PINS.

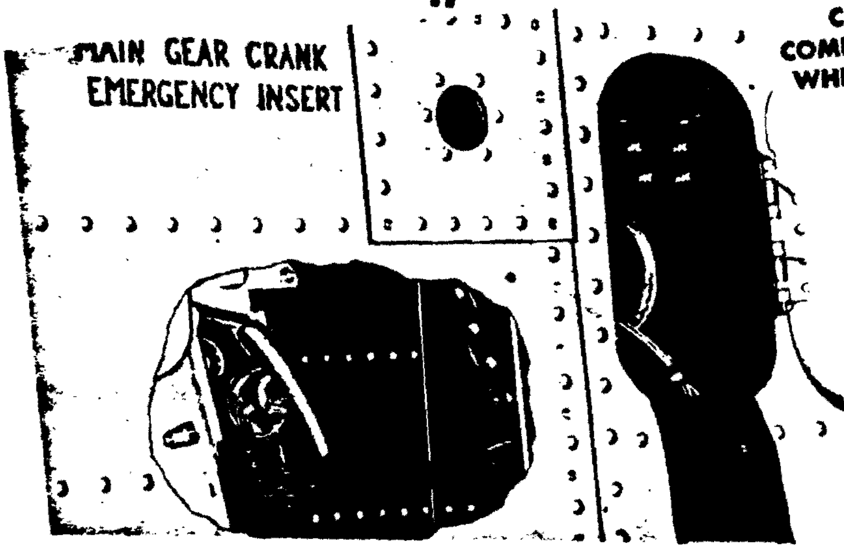
Ground locking pins, manually installed and removed, are a safety device to prevent inadvertent retraction of the landing gear struts during maintenance and ground handling. The main landing gear ground locking pins (figure 1-45), are inserted through ground lock pin access doors on the inboard side of the wheel wells. A red flag is attached to each pin and is draped through the access door when the ground lock pin is installed. When not in use, the pins are stored in a stowage case located over the left main wheel well.

GROUND LOCK PINS . . . . . landing gear

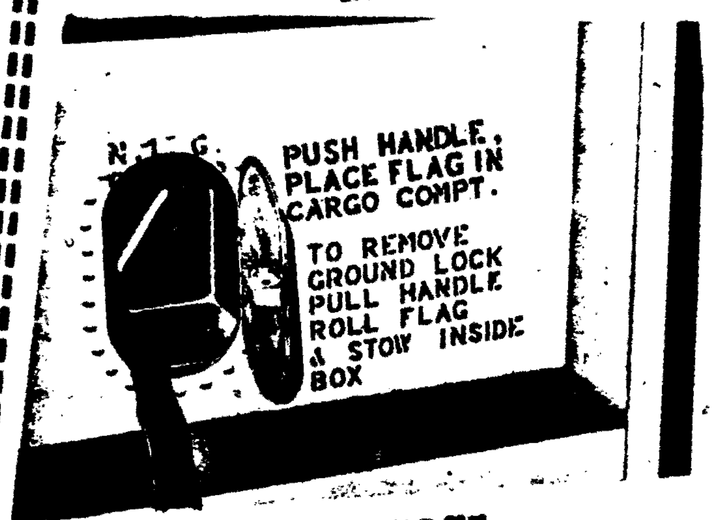
main gear . . .

(right side shown, left side similar).

AT CARGO COMPARTMENT WHEEL WELLS



AT CREW COMPARTMENT ENTRANCE LADDER



nose gear . . .

17540

Figure 1-45

NOSE LANDING GEAR GROUND LOCK PIN.

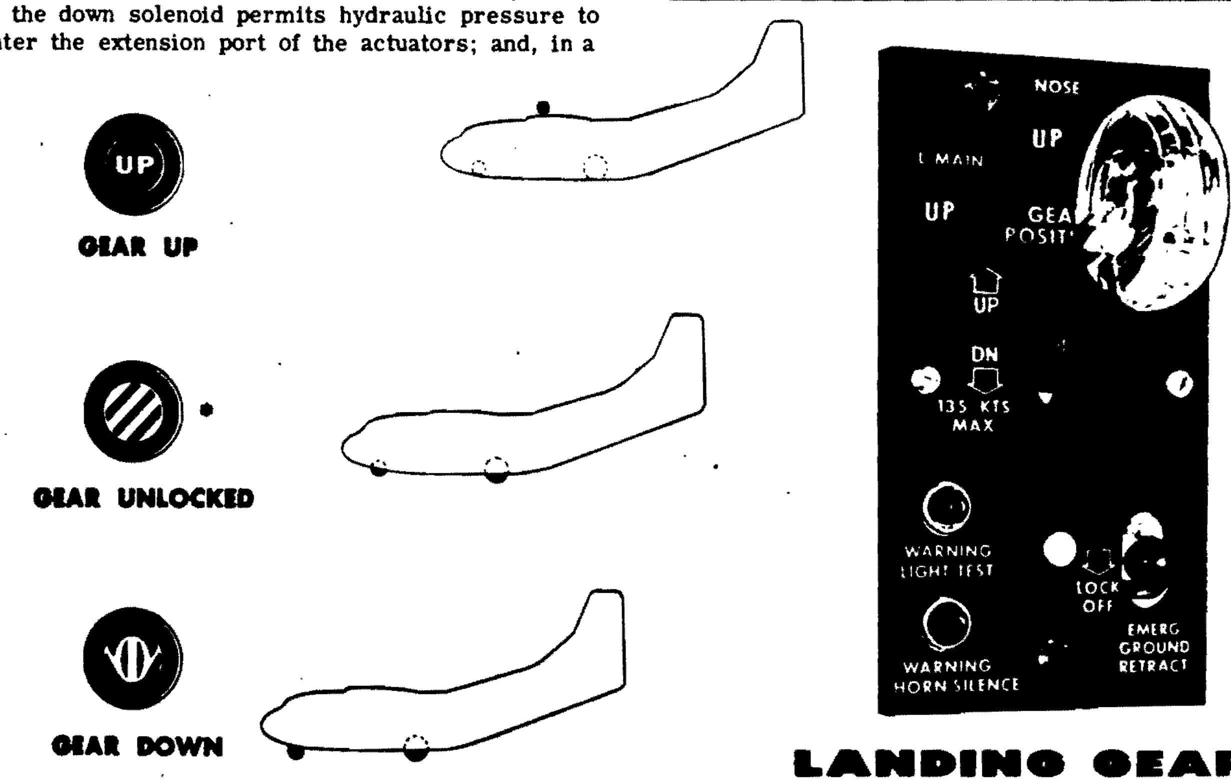
A permanently installed ground lock (figure 1-45), is provided for the nose gear, with an access door in the forward bulkhead of the cargo compartment. The pin is inserted by pushing forward on the "T" handle, and is withdrawn by pulling aft. A red flag is attached to the pin and is draped through the access door to give a visual indication that the pin is inserted. After the pin has been withdrawn, the flag is folded and stowed in the access compartment and the access door closed.

Landing Gear Lever.

Normal operation of the landing gear is controlled by the landing gear lever (figure 1-46), located on the copilot's instrument panel. The positions of the lever are UP and DOWN. When the lever is placed in the UP position, the up solenoid of the 28-volt dc landing gear control valve is energized and directs hydraulic pressure to the retraction port of the landing gear actuators; the initial movement of each actuator positions a mechanical linkage which releases its downlock. In the DOWN position of the lever, energization of the down solenoid permits hydraulic pressure to enter the extension port of the actuators; and, in a manner similar to the retraction cycle, the initial movement of the actuator positions a mechanical linkage which releases the uplock. To prevent inadvertent retraction of the gear, the landing gear lever is locked in the DOWN position by a positive mechanical lock as long as the aircraft is on the ground. When the main gear leaves the ground and the shock struts extend, a drag link switch energizes a solenoid which releases the lock and permits movement of the lever to the UP position. Provisions are also incorporated into the lever assembly to override the lock should an emergency require retraction of the gear when the aircraft is on the ground. As an auxiliary function, the landing gear lever, when placed in the UP position, closes the brake shut-off valve, rendering the hydraulic brake system inoperative during flight. Refer to HYDRAULIC BRAKE SHUT-OFF VALVE, this section.

Landing Gear Lever Emergency Release Knob.

A release knob (Figure 1-46), located on the copilot's instrument panel adjacent to the landing gear lever, is provided to unlock the lever should an emer-



* Striped marker appears when landing gear lever is moved to UP or DN or when the power is off; disappears when the gear are locked in position and power is ON.

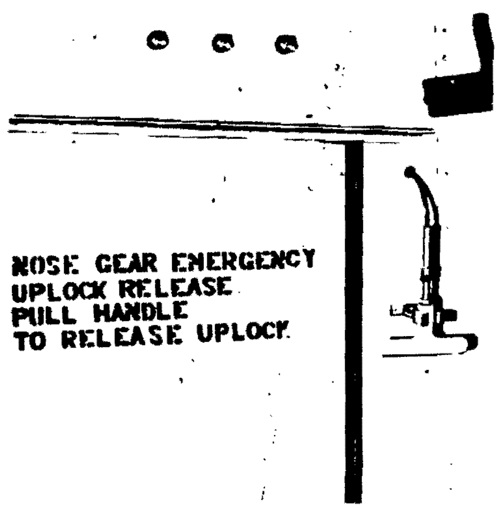
LANDING GEAR CONTROL & INDICATION

17538

Figure 1-46

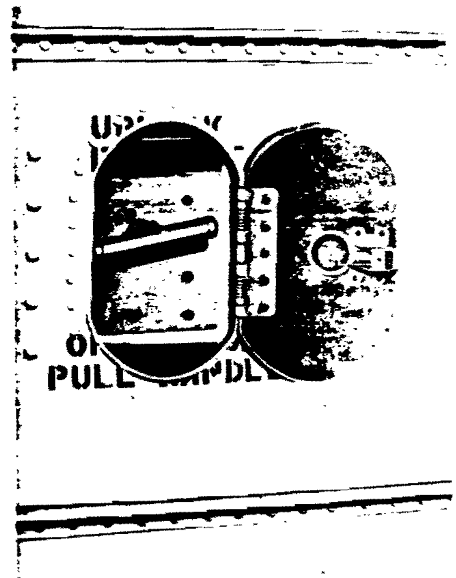
UPLOCK RELEASES . . . landing gear

nose gear . . .



left side crew compartment entrance

main gear



. . . main wheel well compartments

Figure 1-47

gency ground retraction of the gear be deemed necessary. The solenoid lock is normally disengaged electrically when the main gear struts extend as the aircraft leaves the ground. The emergency release knob, when pressed down, mechanically releases the lock and permits the landing gear lever to be moved to the UP position.

Note

The solenoid lock is operated by 28-volt dc power from the primary bus. Therefore, in the event of failure of all generators and the APU, the emergency release knob must be used to raise the landing gear.

CAUTION

Do not manually unlock the landing gear lever on the ground unless an emergency retraction is required.

Landing Gear Warning Horn And Silence Button.

An electrically operated landing gear warning horn is located on the forward face of the pedestal. If, on some aircraft, both throttles are moved to a setting less than 24 to 26 inches Hg. MAP at 2400 rpm (set for 3000 feet density altitude) and any landing gear is not locked in the fully extended position, a warning horn will sound. A warning horn silence button (Figure 1-46) is installed on the landing gear control panel to disconnect this signal, but movement of the throttles to a setting above approximately 26 inches Hg. MAP or complete extension of the landing gears will silence the warning horn and automatically reset the signal circuit. On other aircraft, the warning horn is connected to both throttles in such a manner that retarding either one will cause the horn to sound. Power for the operation of the landing gear warning horn is supplied by the 28-volt dc flight emergency bus.

Landing Gear Warning Light.

A red warning light, mounted in the landing gear lever (figure 1-46), illuminates whenever any landing gear is not locked in position. This light also comes on whenever either throttle is advanced and any landing gear is not in the position selected by the landing gear lever. Furthermore, whenever both throttles are retarded (either throttle on some aircraft †), the light will glow until all three landing gear are down and locked. Electrical power for the landing gear warning light is supplied by the 28-volt dc flight emergency bus.

Landing Gear Warning Light Test Button.

A push button switch located on the left side of the landing gear control panel (figure 1-46), is provided for testing the warning light in the landing gear lever. Depressing the switch will illuminate the red warning light only as long as the button is held in.

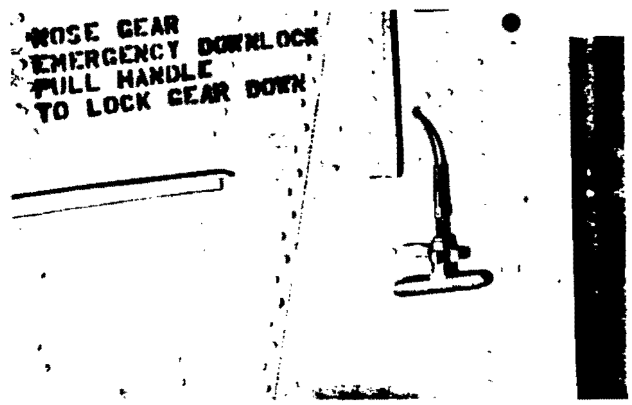
Landing Gear Position Indicators.

Position indicators for each landing gear assembly are provided on the landing gear control unit (figure 1-46), which is located on the copilot's instrument panel. Indicators are powered by the 28-volt dc flight emergency bus and will show whether the landing gear is UP and locked, UNLOCKED, or DOWN and locked. Refer to Landing Gear Indication, Figure 1-46.

EMERGENCY OPERATION.

In the event of electrical failure, the gear may be raised or lowered hydraulically by manual positioning of the control valve. Should the main hydraulic system fail, the uplocks may be released manually allowing the gear to fall free. If the gear fails to lock down, emergency extension is then completed by pulling the nose gear emergency down lock and manually cranking the main landing gear into the down lock with a hand crank (figure 3-15).

† AF 54-607 and Subsequent



NOSE GEAR EMERGENCY DOWNLOCK HANDLE

Figure 1-48

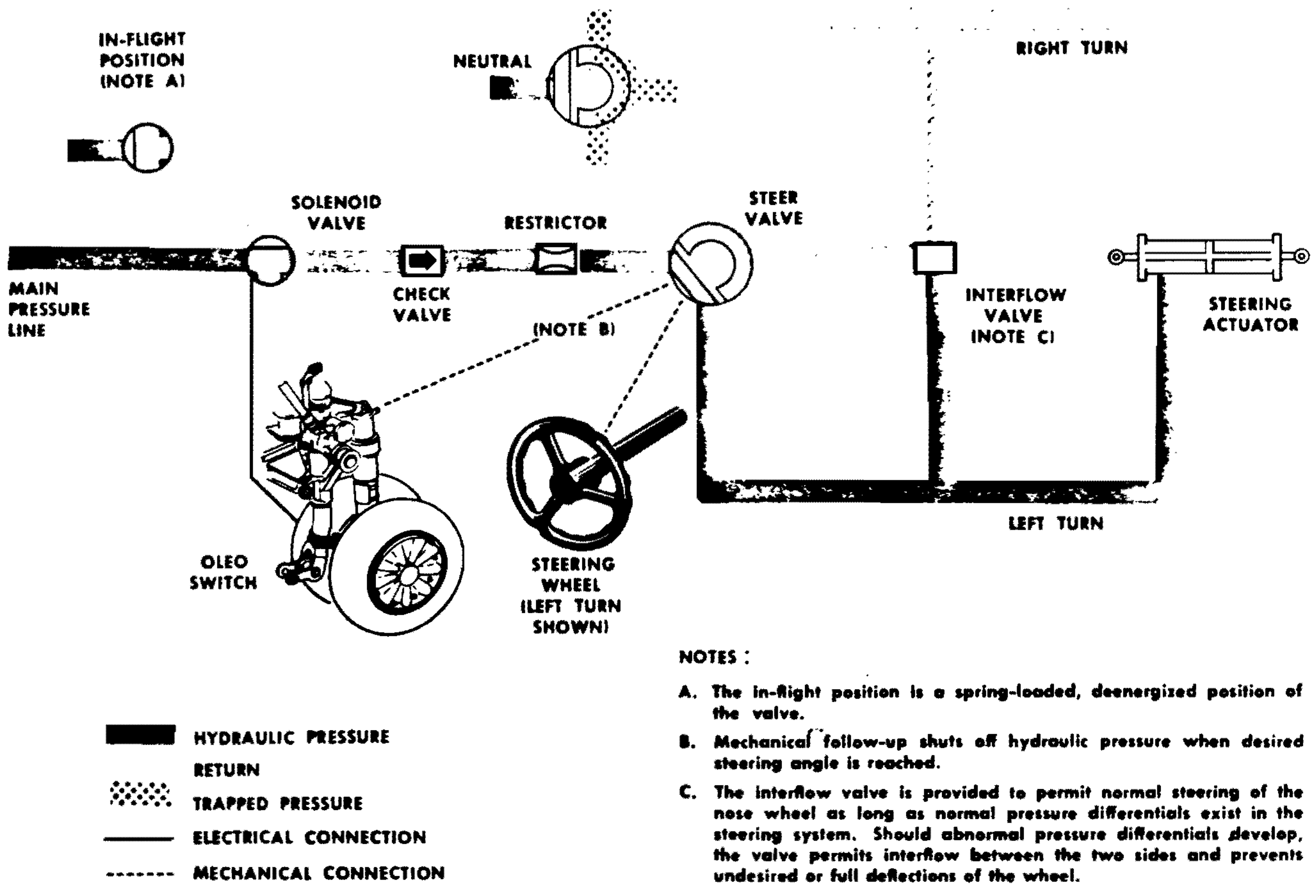
Landing Gear Manual Control Buttons.

For manual positioning of the landing gear hydraulic control valve during electrical system emergency, two manual buttons (figure 4-28), are provided on the valve itself. One button placarded DOWN, when pushed, positions the valve for extension of the gear; the other, unmarked on the opposite side of the valve, will position the valve for gear retraction. The valve is mounted on the hydraulic power panel over the right wheel well.

CAUTION

When ground lock pins are not installed and the accumulator is hydraulically charged, movement of the manual control button to the UP position will retract the landing gear.

NOSE GEAR STEERING



NOTES:

A. The in-flight position is a spring-loaded, deenergized position of the valve.

B. Mechanical follow-up shuts off hydraulic pressure when desired steering angle is reached.

C. The interflow valve is provided to permit normal steering of the nose wheel as long as normal pressure differentials exist in the steering system. Should abnormal pressure differentials develop, the valve permits interflow between the two sides and prevents undesired or full deflections of the wheel.

Figure 1-49

Landing Gear Controllable Check Valve.

A controllable check valve (figure 4-28), installed in the landing gear control valve return line to the reservoir is located on the hydraulic power panel over the right main gear well. This valve serves two purposes. In the NORMAL position, it allows hydraulic fluid to flow only from the landing gear control valve to the reservoir, thus preventing the possible release of the uplocks and downlocks due to surges in the return line caused by operation of other hydraulic components. In the EMERGENCY position of the valve the return line is open and affords an additional supply of fluid to fill the down line completely during a free fall of the landing gear. This is necessary because the area of the bottom (up) side of the piston is less-due to the area of the piston rod-than the area of the top (down) side.

Main Landing Gear Uplock Release Handle.

An emergency uplock release handle (figure 1-47), is provided for each of the gears. Each main gear uplock release handle is located behind an access cover at the rear of its respective wheel well. If the normal hydraulic system should fail to extend the gear, these handles, when pulled, release the locks holding the main gear in the up position and allow them to fall free.

Nose Landing Gear Uplock Release Handle.

The nose landing gear release handle (figure 4-34), is the upper of the two yellow painted "T" handles located just to the left of the crew compartment entrance ladder. The handle is cable-connected to the nose gear uplock so that pulling on the handle releases the uplock and allows the nose gear to fall free.

Nose Gear Emergency Downlock Handle.

A nose gear emergency downlock handle (figure 4-34), is mounted on the forward bulkhead of the cargo compartment, to the left of the crew compartment ladder. The handle is cable-connected to the nose gear strut in such a manner that, when the nose gear has been manually released, pulling aft on the handle forces the strut into the down and locked position. During normal operation of the landing gear, the handle should be stowed in the bracket provided on the bulkhead.

Landing Gear Emergency Handcrank.

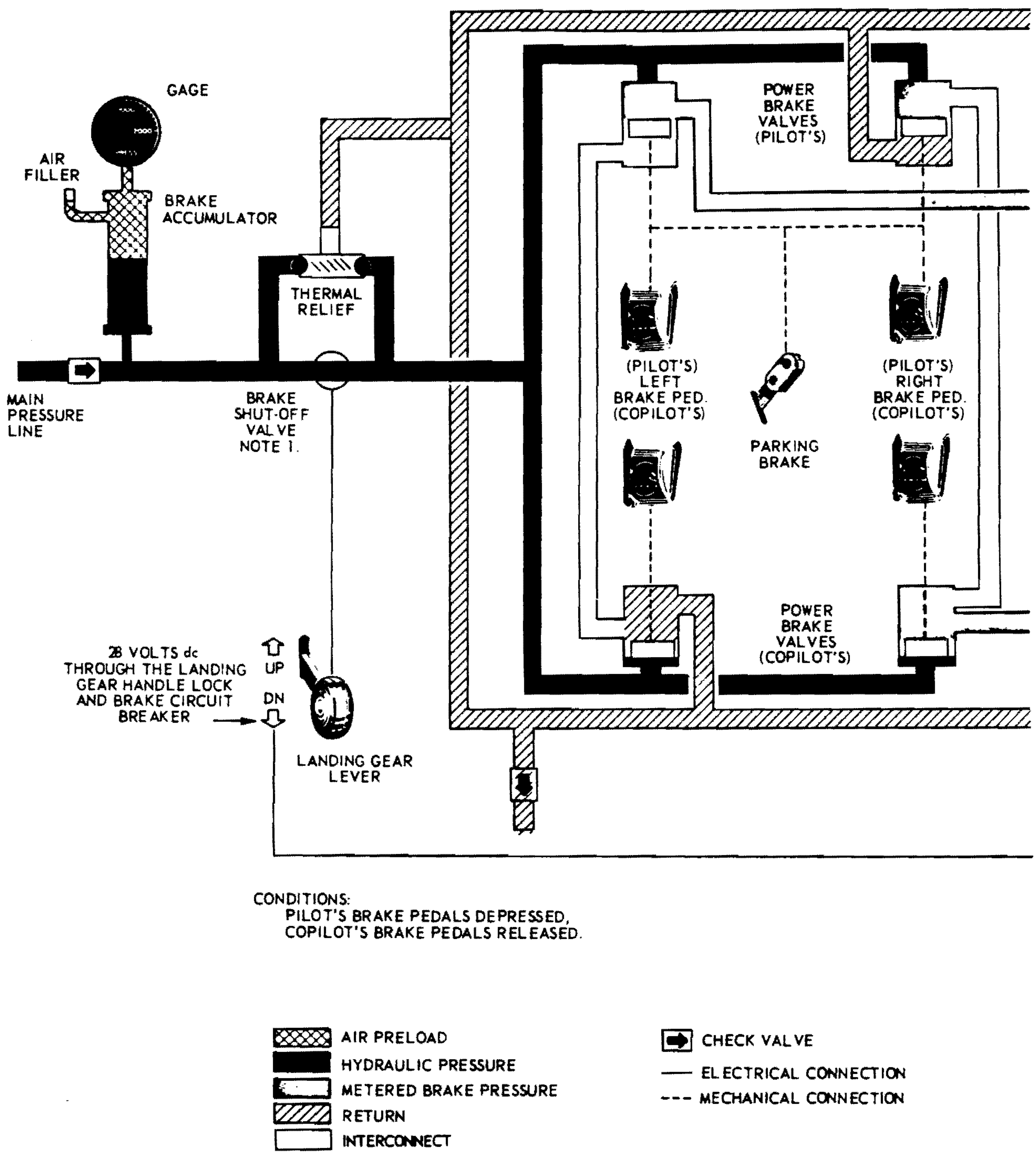
The landing gear emergency handcrank (figure 4-34), is used to crank the main gear into the down and locked position. If the main gear do not lock down after free fall, the ratchet-operated handle can be inserted through access doors into a mating socket on each gear trunnion shaft and used to lock each gear down manually. Should cranking fail to force the gear entirely down and locked, the tapered end of the handcrank may be used to pry the gear to the fully locked position.

STEERING SYSTEM.

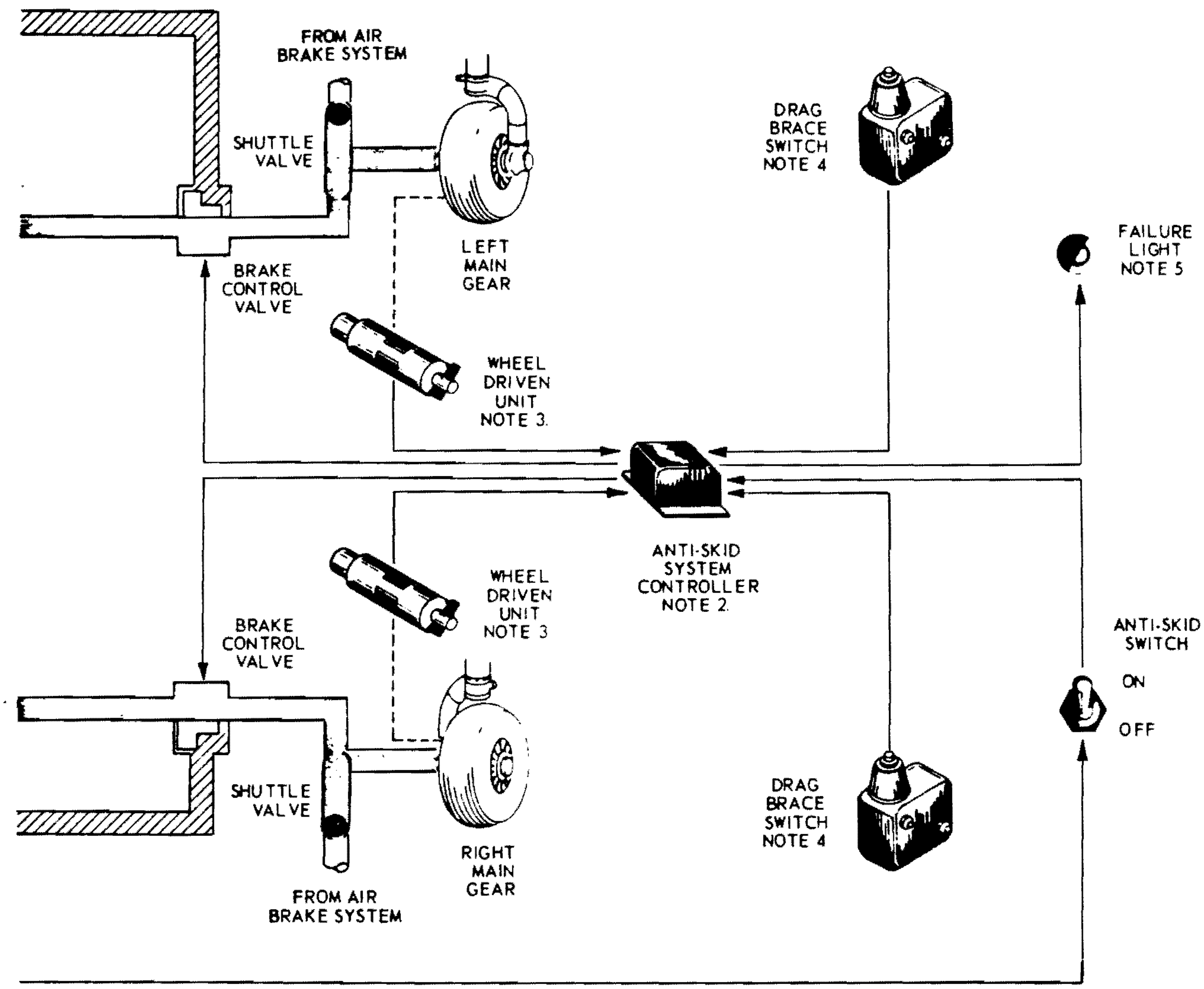
A nose gear steering system, operable only when the aircraft is on the ground, is provided to turn the nose gear 60 degrees in either direction.

STEERING WHEEL.

When the nose gear strut is compressed, a 28-volt dc (primary bus) electrically controlled solenoid valve opens, making hydraulic pressure available to the steering control valve. This valve controls the steering action by directing hydraulic pressure to the appropriate side of the steer cylinder. The valve is mechanically positioned by the movement of the nose gear steering wheel. The nose gear steering wheel protrudes from beneath the left side of the pilot's instrument panel. A notch in the steering wheel indicates the position of the nose gear. When the notch is in the top centered position, the nose gear is aligned fore and aft.



HYDRAULIC BRAKE AND ANTI-SKID



NOTES:

1. BRAKE SHUT-OFF IS A SOLENOID VALVE; SPRING LOADED TO OPEN. REFER TO BRAKE SYSTEM, SECTION 1.
2. RECEIVES 28 VOLTS DC WHEN THE LANDING GEAR LEVER IS IN THE DOWN POSITION, RECEIVES SIGNALS FROM WHEEL DRIVEN UNITS. CONTROLS THE BRAKE CONTROL VALVES.
3. SUPPLIES SIGNAL REPRESENTATIVE OF INDIVIDUAL WHEEL SPEED TO THE CONTROLLER.
4. SWITCHES ARM THE SYSTEM WHEN MAIN GEARS ARE DOWN BUT DO NOT ACTIVATE SYSTEM UNTIL STRUTS ARE COMPRESSED AND AN INITIAL GROUND SPEED OF 22 KNOTS IS ATTAINED.

   EITHER SWITCH COMPRESSED WILL ALLOW SYSTEM TO BE ACTIVATED.
5. INDICATES COMPONENT OR SYSTEM FAILURE. ANTI-SKID SYSTEM IS DEENERGIZED.

CONTROL SYSTEM

Figure 1-50

CAUTION

Do not use strong and forceful pressure at any time when turning the steer wheel. A gentle force on the wheel will actuate the steer valve to the maximum limit in either direction and may be held as desired on "follow through" as the gear turns, and may be reversed at any point as desired. If for some reason gear turning appears sluggish, extra steer wheel force will NOT increase steering power. The steer valve is either open or closed (using a 1500 psi source) as you turn or release the steering wheel. Violation of this procedure will shorten the life of or break steer cable. HANDLE GENTLY WITH ONE HAND.

FAIL-SAFE FEATURE. **

The control valve is permitted to leak pressure to both sides of the steering actuator to prevent nose gear shimmy whenever the gear is not being turned. An interflow valve prevents uncontrolled turning of the nose gear in event of loss of shimmy dampening pressure from one side of the actuator, such as might result from a ruptured hydraulic line.

Note

On those aircraft without the interflow valve, if a nose wheel steering system failure occurs, steering (taxiing) is not possible as the nose gear locks into position.

FLOW REGULATOR.

On some aircraft, †† a flow regulator is installed in the nose gear steering system to prevent pressure surges.

BRAKE SYSTEMS.

Hydraulic and air brake systems are provided for individual braking of the main landing gear wheels. The two systems are independent except for the lines between the shuttle valves and the brake pistons. The shuttle valves, which operate in response to pressure, separate the systems and permit only one system to affect the brakes at a time. During all normal operations, the hydraulic brakes are used. If hydraulic pressure fails, the emergency air brakes may be employed.

HYDRAULIC BRAKE PEDAL SYSTEM.

The main landing gear hydraulically operated brakes are controlled either by the pilot's or by the copilot's pedals. Depressing the toe portion of the rudder pedals mechanically operates the corresponding power brake valve, which directs hydraulic pressure through the brake system lines, anti-skid control valves, and landing gear shuttle valves to the pistons of the main gear brake assemblies. The anti-skid control valve for each wheel prevents skidding by metering pressure to the brake in response to the anti-skid system, which senses rapidity of deceleration of the wheel. Refer to HYDRAULIC BRAKE SYSTEM, Section VII. The pilot's and copilot's brake valves are hydraulically interconnected so that the brakes will respond to whichever pedals are depressed most. Whenever the landing gear lever is in the UP position, a solenoid shut-off valve closes and prohibits use of the hydraulic brakes. This prevents damage to the gear structure which could occur if the brakes were applied during the retraction cycle. The air-preloaded hydraulic brake accumulator, air filler valve, and pressure gage are located in the radio compartment. The accumulator supplies hydraulic pressure to the brake system for the initial braking surge, parking pressure, and reserve in event of failure of the hydraulic power supply. A brake accumulator gage reading of 850 psi will assure the pilot of braking pressure for at least one full application of the brakes.

Brake Accumulator Gage.

A pressure-operated gage adjacent to the brake accumulator in the radio compartment indicates, in pounds per square inch (psi), the air pressure existing in the brake accumulator. During normal operation, when the system is pressurized, this gage should read the same as the pilot's hydraulic pressure gage on the instrument panel. When brake system pressure is depleted, the accumulator gage will continue to indicate the preload pressure to which the accumulator was charged. Refer to Figure 5-1, INSTRUMENT RANGE MARKINGS, for accumulator preload pressure. The gage is located near the accumulator filler point in order that it may be used when replenishing the air preload.

Hydraulic Brake Shutoff Valve.

A hydraulic brake shutoff valve is provided to shut off hydraulic pressure upstream of the power brake control valves. The 28-volt dc solenoid valve is energized when the landing gear handle is placed in the UP position. This prevents the application of brakes at any time the landing gear handle is UP and power is applied to the primary dc bus. The valve is spring-loaded to return to the open position should failure of the electrical system occur.

*** AF 54-587 and Subsequent*
†† AF 54-687 and Subsequent

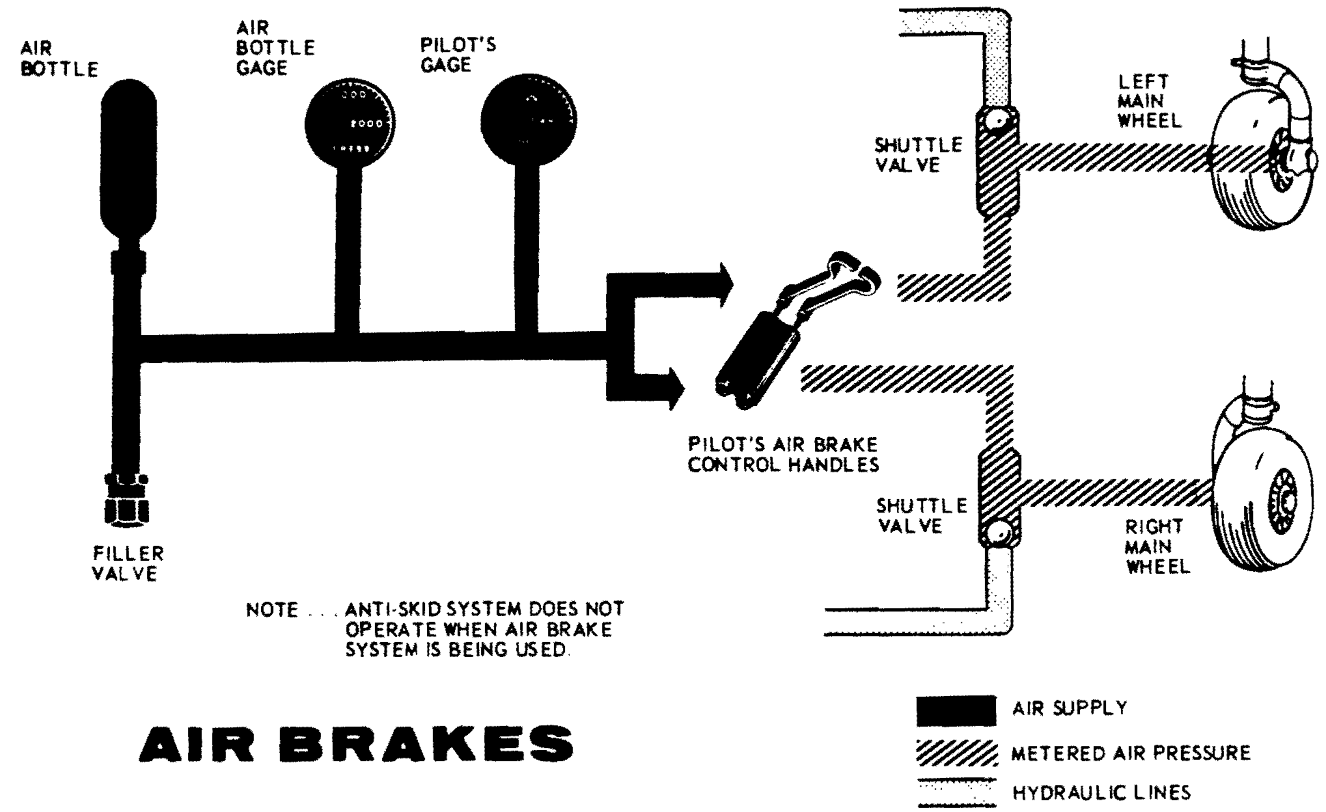


Figure 1-51

Anti-skid Switch and Warning Light.

In the event of an electrical failure in the brake anti-skid system, an amber warning light on the pilot's instrument panel is illuminated by fail-safe circuits and the brake system is automatically reverted to manual control. An on-off switch is located next to the warning light so that the pilot may remove the primary 28-volt dc power from the anti-skid control circuits, rendering them inoperative. Electrical power for the anti-skid system is obtained from the LANDING GEAR HANDLE LOCK AND BRAKE circuit breaker.

WARNING

In the event of main hydraulic system failure prior to landing, turn the anti-skid system OFF. This will insure positive braking action of at least one full application of the brakes.

Note

It is not uncommon for the anti-skid warning light to illuminate when the landing gear is lowered. If this happens, the anti-skid switch should be cycled to the OFF position and then back to the ON position. If the light goes out, the system is operative.

PARKING BRAKE.

The parking brake, controlled by a parking brake handle on the pilot's side of the pedestal, utilize the hydraulic brake pedal system for braking action. The parking brake is set by holding the pilot's brake pedals depressed while moving the parking brake handle aft to the ON position. This causes the pilot's power brake valves to be held open continuously. The parking brake may be set regardless of whether hydraulic pressure is available or not. If the parking brake is set while hydraulic pressure is not available, the brakes will be applied whenever hydraulic pressure becomes available. The parking brake is released by easing the handle forward while holding the brake pedals depressed.

EMERGENCY AIR BRAKE SYSTEM.

An emergency air brake system is installed to provide braking action in the event of hydraulic system failure. The air brake system consists of an air bottle, filler valve, two pressure gages, dual brake valves, and the related interconnecting pressure lines. To apply the brakes, air from the bottle is metered through the air brake valves to the shuttle valves of the brake system. The air pressure then is applied to the brake pistons through the same lines normally used by the hydraulic brakes. Fully charged, the air brake system is sufficient for three full applications of the brakes. The supply bottle, filler valve and one gage are located in the oxygen compartment. For servicing information, refer to Servicing.

Emergency Air Brake Valve Handles.

Dual, guarded, emergency air brake valve handles, located on the side window frame to the left of the pilot, meter air from the bottle to the piston in the brake assemblies. To operate the air brake valves, the guard is pushed forward and the handles pulled aft. The valves can be operated separately or together to direct air to the left or right brake independently, or to both simultaneously. Moving the handle forward releases the brakes by permitting the braking air pressure to escape.

CAUTION

The anti-skid system does not operate when the air brakes are used.

Emergency Air Brake Pressure Gages.

Two air pressure gages are installed in the emergency air brake system to provide visual indication of system pressure. One gage is located on the pilot's instrument panel; the other, adjacent to the emergency air bottle in the oxygen compartment for servicing. The gages are calibrated in pounds per square inch (psi).

Moisture Drain Valve.

A moisture drain valve is provided adjacent to the outlet of the air bottle. A standpipe to the lowest portion of the bottle permits moisture to be driven out by air pressure in the bottle whenever the drain valve is opened. The valve is provided to drain the bottle each time the air bottle is recharged with air.

INSTRUMENTS.

Other than the conventional engine and flight instruments installed on the instrument panels (figures 1-58 and 1-59), a slaved gyro magnetic compass system and a stall warning system are provided. For additional information. refer to SLAVED GYRO MAGNETIC COMPASS SYSTEM, Section IV and angle of attack/stall warning system in Section VII.

ENGINE INSTRUMENTS.

The instrument panel is arranged with the reciprocating and jet engine instruments (figure 1-58) in the center, visible to both the pilot and copilot.

RECIPROCATING ENGINE INSTRUMENTS.

For a description of the reciprocating engine instruments, refer to ENGINES, this section.

JET ENGINE INSTRUMENTS.

Refer to JET ENGINES, this section, for a description of the jet engine instruments provided.

FLIGHT INSTRUMENTS.

On the left side of the instrument panel, directly in front of the pilot, are the pilot's flight instruments which consist of an angle of attack indicator, airspeed indicator, heading indicator, attitude indicator, altimeter, turn-and-slip indicator and a vertical velocity indicator. The copilot's flight instruments are located on the right side of the instrument panel and consist of an angle of attack indicator, airspeed indicator, heading indicator, attitude indicator, altimeter, turn-and-slip indicator, and a vertical velocity indicator. This instrumentation is similar to the pilot's except that the pilot's heading indicator is a functional component of the gyro magnetic compass system, whereas the copilot's corresponding instrument serves only as a heading gyro.

WARNING

- With partial or complete loss of the ME-1A Compass Amplifier, it is possible to have erroneous indications on the pilot heading indicator, the ID-250. There will be no "OFF" flags or warning lights displayed.
- It is possible to have partial electrical failure or failure of certain components within the attitude indicating systems, that will not cause the (OFF) flag to appear. Therefore, the attitude indications given during flight should periodically be checked against other flight instruments.
- A slight amount of pitch error in the indication of the Type MF-2 or B-1A attitude indicator will result from accelerations or decelerations. It will appear as a slight climb indication after a forward acceleration and as slight dive indication after deceleration when the aircraft is flying straight and level. This error will be most noticeable at the time the aircraft breaks ground during the take-off run. At this time, a climb indication error of about 1-1/2 bar widths will normally be noticed; however, the exact amount of error will depend upon the acceleration and elapsed time of each individual take-off. The erection system will automatically remove the error after acceleration ceases.

A magnetic compass is attached to the lower edge of the overhead panel in the center.

Figure 1-52

17801

ALTIMETER.

ALTIMETER SETTING.

It is possible to set the altimeter in error by 10,000 feet. This happens when the barometric set knob is continuously rotated after the barometric scale is out of view. The knob can be rotated until eventually the numbers will reappear in the window from the opposite side. If the correct altimeter setting is then established, the altimeter will read approximately 10, 000 feet in error. See figure 1-52 for an example of correctly set and misset altimeters.

††Altimeter-Encoder AAU-21/A.

One altimeter-encoder (figure 1-52A) is installed in the altimeter position on the pilot's instrument panel. The altimeter-encoder combines a conventional barometric type altimeter, possessing a counter-drum-pointer display, with an altitude-reporting encoder in one self-contained unit. The 10,000 and 1000-foot counters and 100-foot drum provide a direct digital output and readout of altitude in increments of 100 feet, from -1000 to 38,000 feet. The digital output is referenced to 29.92 inches of mercury and is not affected by changes of barometric setting. The pointer repeats the indications of the 100-foot drum, and serves both as a vernier for the drum and as a quick indication of the rate and sense of altitude changes. Two methods may be used to read indicated altitude on the counter-drum-pointer altimeter: (1) read the counter-drum window, without reference to the pointer, as a direct digital readout in thousands and hundreds of feet, or (2) read the thousands of feet on the two counter indicators, without referring to the drum, and then add the 100-foot pointer indication.

NOTE

The altimeter mechanism which provides this feature also causes a characteristic behavior of the pointer. This is a noticeable pause or hesitation of the pointer caused by the additional intermittent friction and inertia loads applied to the mechanism in order to turn over the counter at thousand foot intervals as the pointer completes each revolution. This momentary pause is followed by a noticeable acceleration as the altimeter mechanism overcomes the counterwheel load and rolls the dial over to the next thousand foot digit. This effect will be more pronounced at ten-thousand foot intervals where both counters are turned over simultaneously. The pause occurs during the "9" to "1" portion of the scale. The pause-and-acceleration behavior is more pronounced at high altitudes and high rate of ascent and descent. During normal rates of descent at low altitudes, the effect will be minimal.

The self-contained servo driven encoder provides altitude encoder in 100-foot increments for automatic transmission when the IFF transponder is interrogated on mode C. In case of power loss to the encoder servo system, a CODE OFF flag appears automatically in a window in the upper left portion of the display, indicating that altitude information is no longer being transmitted. In this condition, the instrument continues to function as a barometric altimeter.

The barometric pressure is entered by use of a barometric set knob in the lower left front of the instrument case. The altimeter setting appears on counters in the window at the lower right of the display and has a range of settings from 28.1 to 31.0 inches of mercury. An internal vibrator operates continuously whenever aircraft dc power is turned on. The vibrator minimizes internal mechanical friction, enabling the instrument to provide a smoother display during changing altitude conditions. Should vibrator failure occur, the altimeter will continue to function pneumatically, but a less-smooth movement of the instrument display will be evident with changes in altitude.

WARNING

If the internal vibrators of the altimeter-encoder or altimeter are inoperative due to either internal failure or dc power failure, the 100-foot pointers may momentarily hang up when passing through 0 (12 o'clock position). If the vibrators have failed, hangup of the 100-foot pointers can be minimized by tapping the case of the altimeters. Pilots should be especially watchful for this failure when the minimum approach altitude lies within the 800 to 1000-foot part of the scale (1800 to 2000 feet, etc.).

††ALTIMETER AAU-27/A.

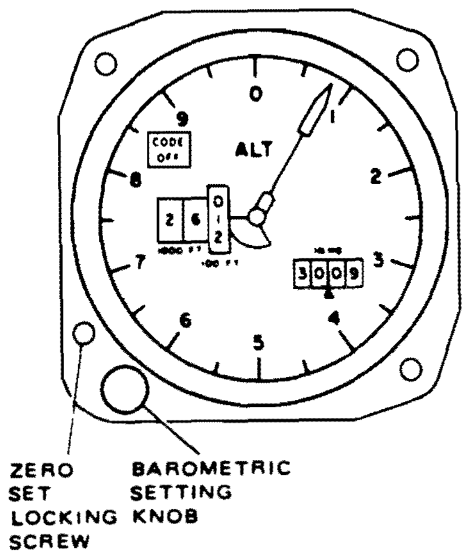
One altimeter (figure 1-52A) is installed in the co-pilot's instrument panel. The instrument is similar to the altimeter-encoded except it does not have an altitude encoder or the CODE OFF display mechanism. The indicated altitude on the altimeter is from -1000 to 50,000 feet. The altitude display, altimeter setting, and vibrator considerations described for the altimeter-encoder also apply to the altimeter.

NOTE

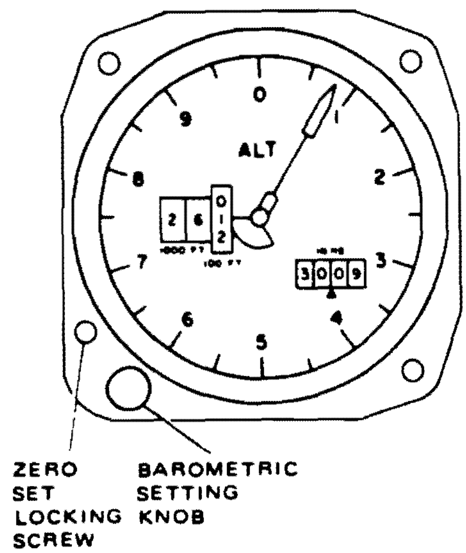
If altimeter AAU-27/A is not installed as part of T.O. 1C-123-609 modification, the co-pilot's existing altimeter will be retained.

††Aircraft modified in accordance with TCTO 1C-123-609.

ALTIMETER

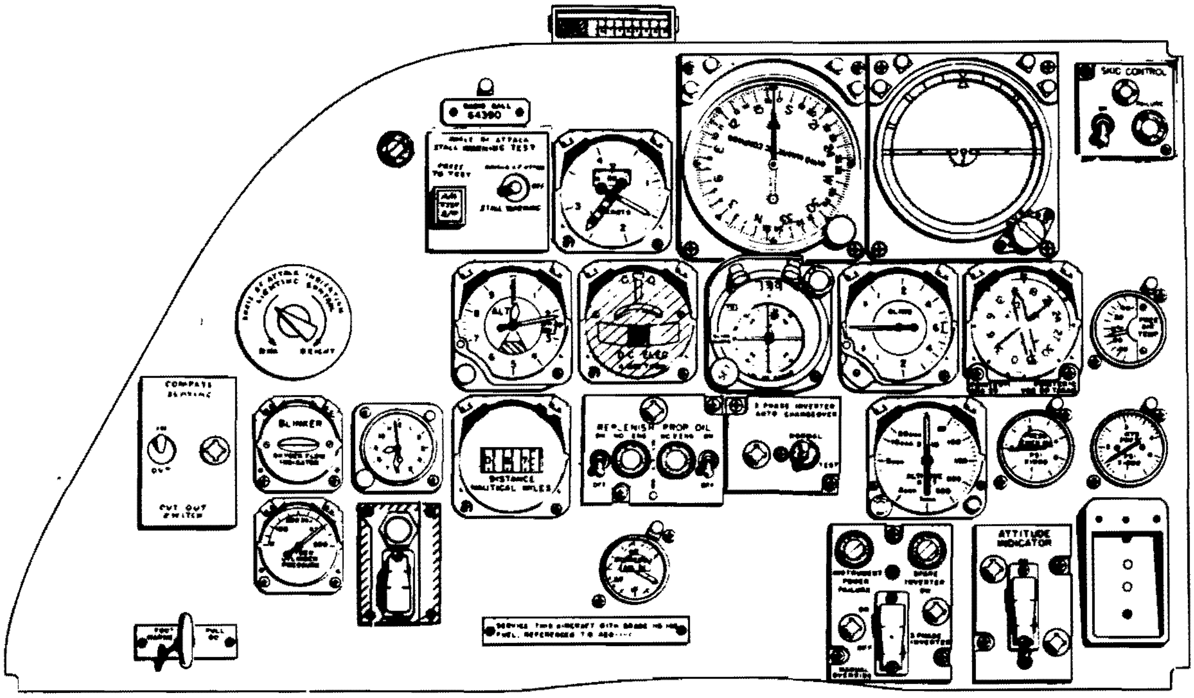


AAU-21/A

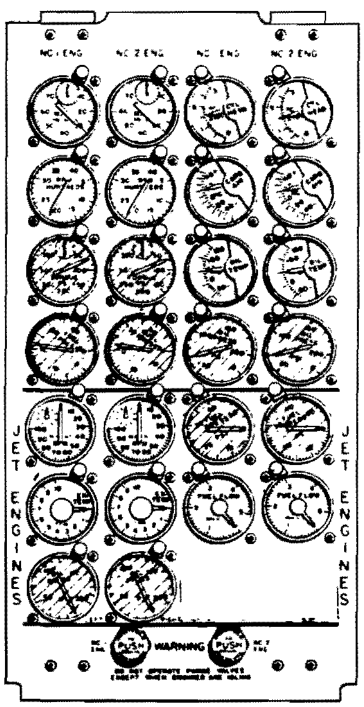


AAU-27/A

Figure 1-52A.



pilot's



engine

28 VOLTS DC

28 VOLTS DC (flight emergency bus)

115 VOLTS AC 3-PHASE

115 VOLTS AC 1-PHASE

26 VOLTS AC 1-PHASE

This illustration indicates the basic power sources required for the operation of the instruments. Operation of instruments requiring AC power presumes the availability of 28 volt DC power. For a detailed discussion of those instruments which employ more complex power inputs from electronic equipment or subordinate power sources, refer to COMPASS SIGNAL POWER AMPLIFIER and NAVIGATIONAL FLIGHT INSTRUMENTS, Section IV.

instrument electrical POWER REQUIREMENTS typical

Figure 1-53. (Sheet 1 of 2)

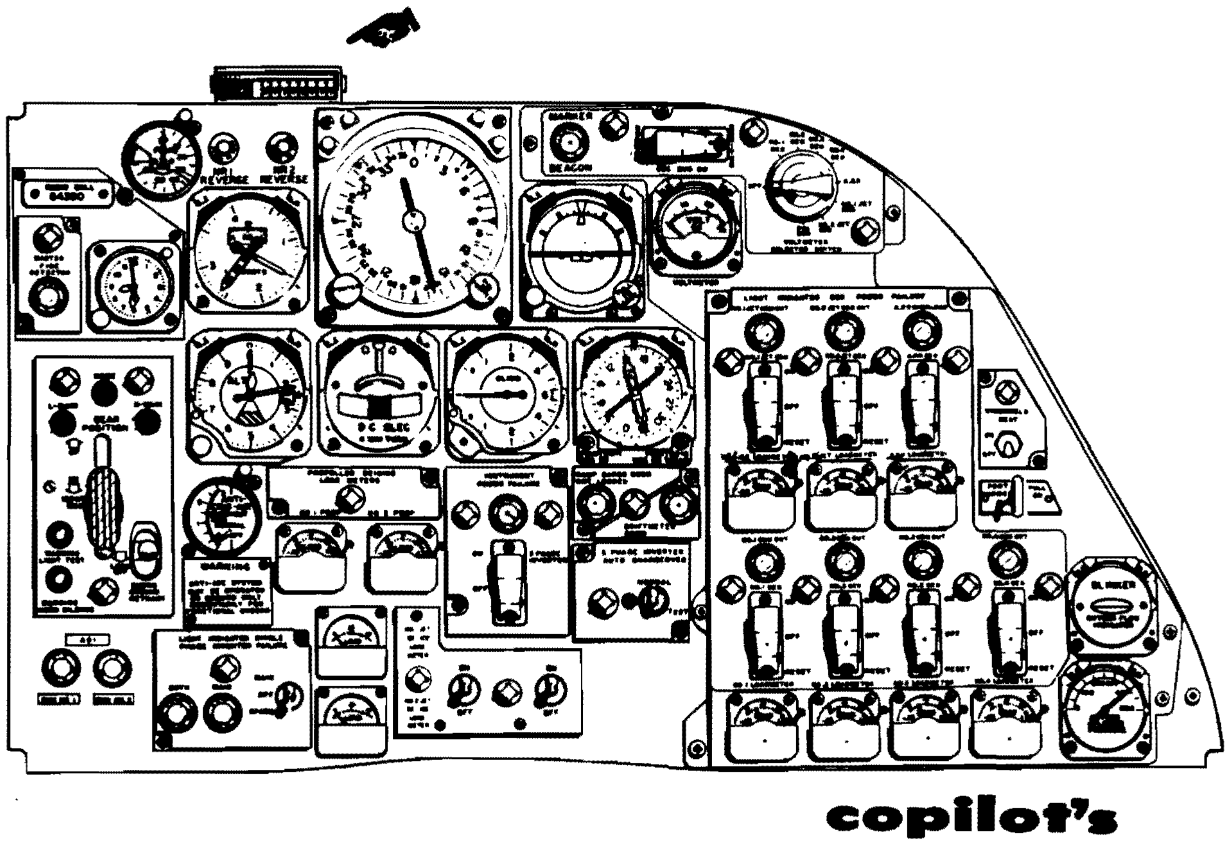


Figure 1-53. (Sheet 2 of 2)

Instrument Power Requirements.

Of the above instruments, the airspeed indicators, altimeters, and vertical velocity indicators are operated by the pitot-static system and require no electrical power. The turn-and-slip indicators require 28-volt dc for the turn needle. The pilot's indicator receives power from the flight emergency bus and the copilot's from the primary dc bus. The inclinometer requires no electrical power. The forces acting on the ball are gravity and centrifugal force. The pilot's B-1A attitude indicator requires 28-volt dc power from the flight emergency bus and 115-volt three-phase ac power from the pilot's or spare three-phase inverter. Three-phase, 115-volt ac power is required for the pilot's and copilot's MF-2 attitude indicators, the pilot's heading indicator, and the copilot's heading indicator. The 115-volt three-phase ac power is supplied to the pilot's instruments by either the pilot's or spare three-phase inverter and to the copilot's instruments by the copilot's or spare three-phase inverter. In addition, a 75-volt source of single-phase ac power is required for the pilot's heading indicator. Normally this is supplied by the compass signal power amplifier when the single-phase inverter is operating. Refer to COMPASS SIGNAL POWER AMPLIFIER, Section IV. If the single-phase inverter should fail, a similar voltage is automatically obtained from one winding of the pilot's or spare three-phase inverter, whichever is operating at the time. The angle of attack indicator is powered from the 28-volt dc primary bus.

Attitude Indicator Switch.

A two-position, OFF-ON, attitude indicator switch (figure 1-58), guarded in the ON position, is mounted on the pilot's instrument panel to control operation of the pilot's B-1A attitude indicator. On some aircraft, the switch and provisions for the equipment are installed but inoperative until the B-1A indicator and K-4B control assembly are installed. On aircraft with B-1A indicators, when the switch is positioned to OFF, the attitude indicator is inoperative; when the switch is positioned to ON, the equipment will operate after a two-minute ± 30 second warm-up time delay, provided 115-volt three-phase ac power and 28-volt dc power are available. Completion of the warm-up cycle is signalled by the disappearance of the attitude warning (OFF) flag from the face of the indicator.

ANGLE OF ATTACK/STALL WARNING SYSTEM.

An angle of attack/stall warning system is installed to provide continuous angle of attack indication and physical warning of impending stall. The system consists of a vane-operated transmitter in the right wing outer panel, a computer unit, a shaker mounted on the pilot's control column, two angle of attack indicators mounted on the instrument panel glare shield, and a test switch and light on the instrument panel. Wing flap position compensation is provided by a potentiometer connected to the wing flaps torque tube. A compensating signal for below zero thrust throttle setting is provided by throttle-actuated microswitches. Airflow direction ahead of the leading edge of the wing displaces a vane which positions two potentiometers. The vane displacement signal is directed to the computer unit where it is modulated by flap and throttle position signals. The output of the computer unit energizes the angle of attack indicator and activates the control column shaker when a stall is imminent. The shaker circuit is routed through the left main landing gear downlock and drag link switches to ensure that the shaker is not energized when the gear is down and the strut is compressed. Power for system operation is provided by the 28-volt dc primary bus. Refer to ANGLE OF ATTACK/STALL WARNING SYSTEM, Section VII, for more detailed information.

Angle of Attack Indicator.

Two angle of attack indicators are mounted on the instrument panel glare shield (figure 1-58.) When the needle is held on the selected increment of stall speed, the correct angle of attack for that speed is maintained. If the needle moves to the left of the selected position, the speed is too low; to the right, the speed is too high. In cruising flight, the needle moves into the green cruise range. Refer to figure 1-54. A rheostat (figure 1-58), on the left side of the pilot's instrument panel controls illumination of the indicator.

Angle of Attack/Stall Warning Test Switch and Light.

The test switch (figure 1-58), on the pilot's instrument panel is provided to check angle of attack calibration and the stall warning shaker. When placed in the STALL WARNING position, the left main landing gear downlock and drag link switches in the shaker circuit are bypassed, permitting the shaker to be energized directly. The test light indicates electrical malfunction when illuminated.

FREE AIR TEMPERATURE INDICATOR.

A free air temperature indicator (figure 1-58), is mounted on the pilot's instrument panel. Free air temperature is measured by a bulb mounted flush with the fuselage skin and is transmitted to the indicator, which registers free air temperature in degrees centigrade. The free air temperature bulb, which contains a thermal resistance element, is mounted in the nose section, flush with the fuselage skin forward of nose wheel door on the underside of nose. Power for operation of the system is obtained from the 28-volt dc primary bus.

ANGLE OF ATTACK INDICATOR AND TEST PANEL

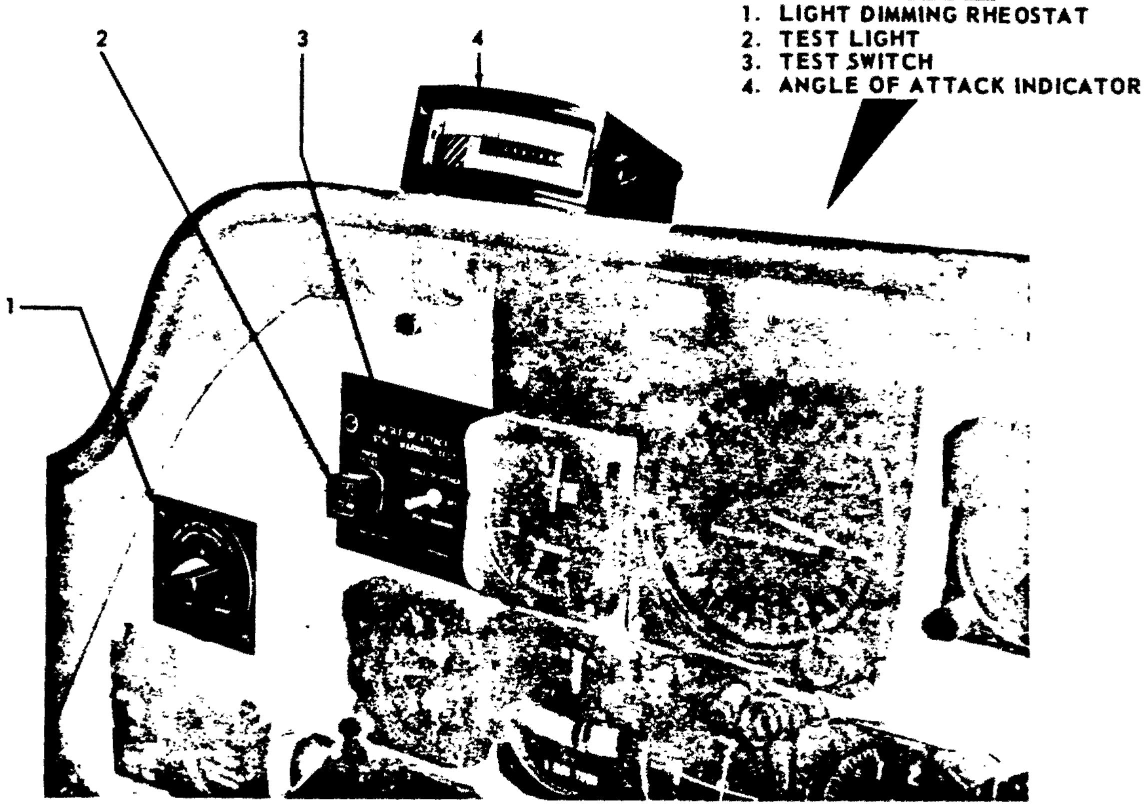


Figure 1-54

RECIPROCATING ENGINE AND APU FIRE DETECTION SYSTEM.

Two different type fire detector systems are installed in the aircraft; the thermocouple type in the reciprocating engine nacelles and APU compartment, and the continuous cable-type installed in the jet engine pods. Both types respond to an abnormal increase in temperature by illuminating appropriately labeled warning lights on the engine emergency panel and if installed, the master fire warning light on the copilot's instrument panel. Electrical power for both systems is obtained from the 28-volt dc flight emergency bus.

THERMOCOUPLE DETECTOR UNITS.

Ninteen fire-sensitive thermocouple detector units (twenty on aircraft AF 57-6289 through 57-6294) are installed in each nacelle. Placement of thermocouples includes eight on the baffle section, nine in the engine mount (ten on aircraft AF 57-6289 through 57-6294), and two in the top cowling panel. The thermocouples are divided into two independent loops; identified as Loop No. 1 and Loop No. 2. There are two circuits in each loop, one in each nacelle, providing a total of four fire detection circuits each containing thermocouples dispersed throughout the nacelle and engine mount areas. The circuits operate independently, therefore, failure of one will not affect the others. Five thermocouple units are installed throughout the APU compartment. When a flame contacts a thermocouple, the thermocouple generates a current which is transmitted to a sensitive relay. When energized, this relay allows 28-volt dc from the aircraft power supply to activate a slave relay. The slave relay closes the circuit to the applicable warning light, indicating the location of the fire. The fire detection system is unaffected by ambient temperature. Refer to figure 1-55. The system operates on 28-volt dc from the flight emergency bus supplied through a five ampere circuit breaker located on the overhead circuit breaker panel.

FIRE EMERGENCY SHUTDOWN HANDLE.

Two fire emergency shutdown handles (figure 1-57), one for each engine installation, are located on the engine emergency panel. Fire detector warning lights are imbedded in each of the T-shaped plastic handles. Either handle, which is pulled aft when the warning lights signal the existence of a fire, automatically accomplishes all procedures necessary to shut down its respective engine except mixture, ignition and generator shut-off. The handle, when pulled, energizes a series of 28-volt dc relays which accomplish the following in its respective nacelle: propeller feathering; fuel, hydraulic, and water shut-off; opening the oil cooler exit door; opening the cowl flaps; and arming the extinguishing agent circuit. When the handle is returned to its normal position, the positioning of the cowl flaps, oil cooler exit doors, fuel shut-off valve and water shut-off valve is dependent upon the setting of the respective control switches. The hydraulic shut-off valve will open. The propeller will unfeather only when the appropriate unfeathering procedures are employed.

MASTER FIRE DETECTOR WARNING LIGHT

A red master fire detector warning light (figure 1-58), is mounted on the copilot's instrument panel to provide fire warning in the normal field of vision of the pilot and copilot. Glowing of the light indicates fire in one of the engines or the APU. To determine the location of the fire it is necessary to observe the fire detector warning lights on the engine emergency panel.

FIRE DETECTOR WARNING LIGHTS.

The red lights imbedded in the fire emergency shutdown handles indicate the presence of fire in the reciprocating engines. Separate lights on the engine emergency panel indicate fires in the jet engines and APU compartment. A master fire warning light on the copilot's instrument panel is energized by the reciprocating engine and APU fire warning systems. The jet engine fire warning system is also connected to the master fire warning light.

Note

Operation of the fire emergency shutdown handles does not in any way affect the operation of the jet engine.

ENGINE FIRE DETECTOR LOOP TEST SWITCH.

A three-position fire detector loop test selector switch (figure 1-57), is located on the engine emergency panel. It provides a means of checking not only the warning lights of the emergency shutdown handles, but the complete engine fire detection system. When the selector switch is placed in the LOOP NO. 1 position, circuit No. 1 of the left engine and circuit No. 2 of the right engine are functionally checked. The detector thermocouple circuits in the nacelles are energized, thus, illuminating the fire emergency shutdown handle lights within 15 seconds. When the switch is moved to the LOOP NO. 2 position, circuit No. 3 of the left engine and circuit No. 4 of the right engine are tested in a similar manner.

ENGINE FIRE DETECTOR WARNING LIGHT TEST BUTTON.

A press-to-test button (figure 1-57), is located on the engine emergency panel. The function of the button is merely that of providing a check of the warning lights in the fire emergency shutdown handle.

JET ENGINE FIRE DETECTION SYSTEM.

Jet engine fire detection (figure 1-60) consists of a special coaxial cable with hermetically sealed, high-temperature connector plugs. A single, solid conductor forms the core of the cable and the space between the conductor and the outer sheath contains a highly compacted material whose resistance decreases with increased temperatures. Each jet engine detector cable circumvents the forward and aft end of the compressor-turbine area of the engine. A section of the cable is routed across the top and bottom of this same compartment. When a fire or overheat condition exists, the temperature of the nearest detector unit increases causing a decrease in resistance and a warning light, located on the engine emergency panel, illuminates. The system is powered from the 28-volt dc flight emergency bus.

JET ENGINE FIRE DETECTOR TEST BUTTONS.

Testing of the fire detector cable elements and warning lights in the jet engine fire detector system is accomplished by depressing the test buttons on the engine emergency panel. Proper operation of the system is indicated by immediate illumination of the warning lights.

APU FIRE DETECTOR TEST BUTTON.

Testing of the thermocouples and warning light in the APU fire detector system is accomplished by depressing the test button (figure 1-57), on the engine emergency panel. Proper operation of the system is indicated by illumination of the light within 15 seconds.

ENGINE FIRE EXTINGUISHING SYSTEMS.

A separate fire extinguishing system (figure 1-55) is provided for each reciprocating engine and each jet engine with each system controlled from the engine emergency panel. These controls consist of fire emergency shutdown handles for the reciprocating engines only, a single fire extinguishing agent discharge switch