frequency rate of ac power output. An independent pressure-type oil system for the drive unit has a 1.6 US gallon tank on the upper right side of the engine compressor section. (See figure 1-41 for oil specification.)

Standby Instrument Inverter.

The standby instrument inverter powers the 3-phase and single-phase instrument busses, if the ac generator is not supplying power. The inverter is powered by the dc primary bus and is energized when the standby inverter switch is ON. When the air start switch is ON, ac generator power is removed from the instrument busses and the standby instrument inverter is energized automatically to power these busses for instrument operation during an air start. When the standby instrument inverter is operating, the instrument busses are disconnected from the main 3-phase ac bus.

ELECTRICALLY OPERATED EQUIPMENT.

See figures 1-25 and 1-26.

EXTERNAL ELECTRICAL POWER RECEPTACLES.

The ac and dc external electrical power receptacles (figure 1-41) are within access doors on the lower surface of the fuselage, behind the main gear wells. All dc busses except the battery bus are energized automatically when a dc external power source is connected to the receptacle. (The battery bus can be energized by external power when the battery switch is ON, provided there is enough battery power to close the tie-in relay that connects the battery bus to the primary bus.) All ac busses are energized by the ac external source, provided the ac generator is not operating.

NOTE

If only an ac external power source is available, the primary bus must be hot to apply external ac power to the airplane.

CIRCUIT BREAKERS AND FUSES.

All the dc electrical distribution circuits are protected by push-to-reset type, trip-free circuit breakers which cannot be reset as long as the overload condition exists. (Refer to Circuit-breaker Use in section VII.) All the ac distribution circuits are equipped with fuses. The power supply circuits to the dc and ac busses are not protected. Circuit breakers and fuses that are accessible in the cockpit are on panels at each side of the cockpit. Circuit breakers and fuses are not duplicated in each cockpit on F-100F Airplanes. (See figures 1-27 and 1-28.)

ELECTRICAL POWER SUPPLY SYSTEM CONTROLS.

Battery Switch.

With the battery switch (figure 1-29) ON, the primary bus is connected to the battery bus and is powered by the battery, provided no other source (dc generator or external power) is supplying the electrical system. (On F-100F Airplanes, the rear cockpit switch is safetied ON to place complete control of the battery in the front cockpit. Both battery switches must be ON for the circuits controlled by these switches to be effective.) The battery bus can be energized by the primary bus when the battery switch is ON if enough battery power is available to close the battery-to-primary-bus tie-in relay. When the battery switch is OFF, the primary bus is disconnected from the battery bus and battery power is furnished only to those units connected to the battery bus.

DC Generator Switch.

A three-position switch (figure 1-29) controls the dc generator. (On F-100F Airplanes, the rear cockpit switch is safetied ON to place complete control of the dc generator in the front cockpit, as both switches must be at ON for generator operation. The safety wire must be broken to move the rear cockpit generator switch to OFF or RESET.) When engine speed is above about 40% rpm, and the switch is ON, generator output is supplied to the dc system. If a malfunction cuts out the generator, the dc generator switch should be held momentarily at RESET and then returned to ON to restore normal generator operation. The dc generator is "off the line" when the switch is OFF. The switch is spring-loaded from RESET to OFF.

NOTE

When the air start switch is turned ON to supply ignition for an air start, the dc generator is automatically cut out of the electrical system. If dc generator switch is ON, generator output is automatically restored when the air start switch is returned to OFF.

F-100D ELECTRICAL

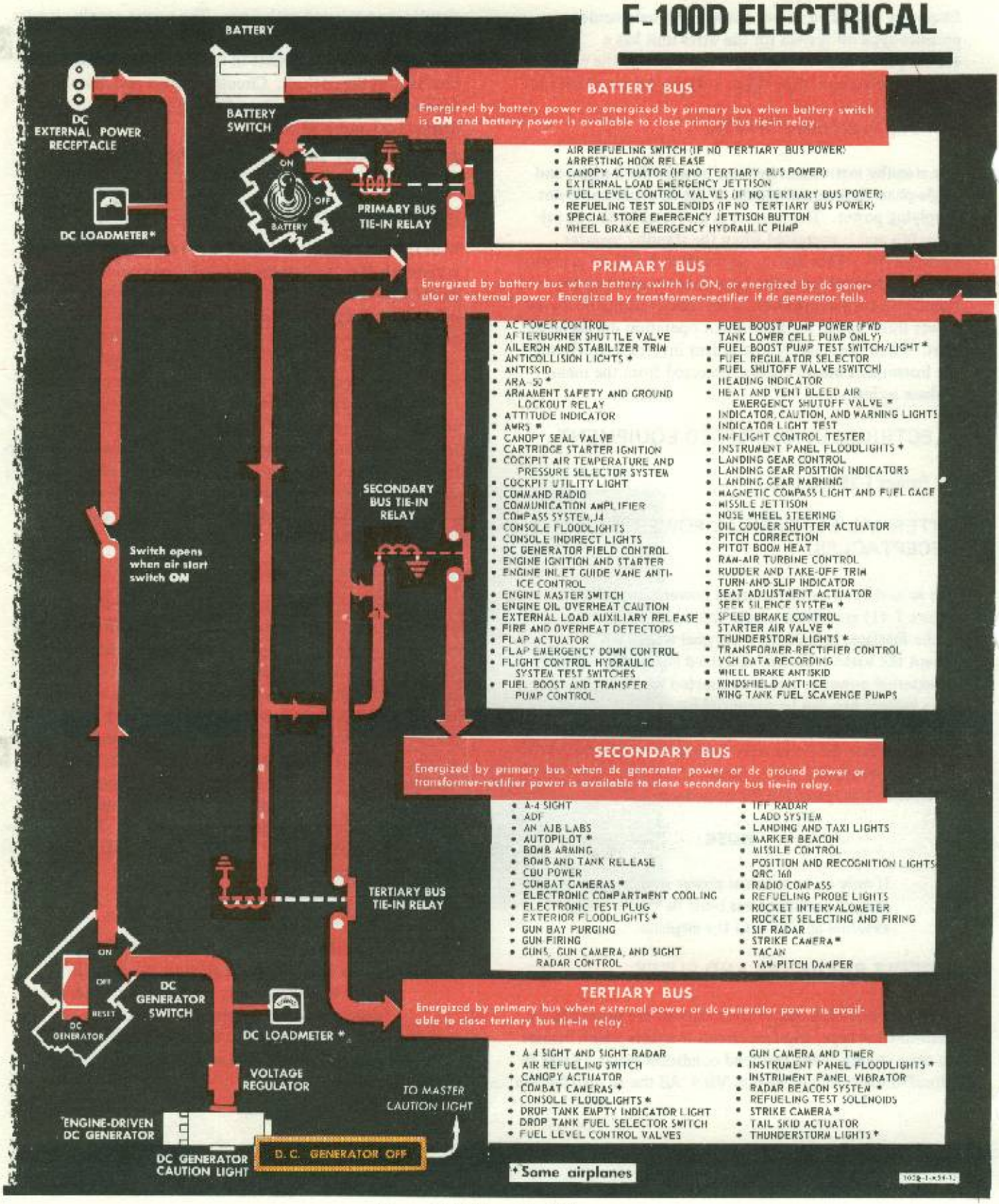


Figure 1-25 (Sheet 1 of 2)

POWER DISTRIBUTION

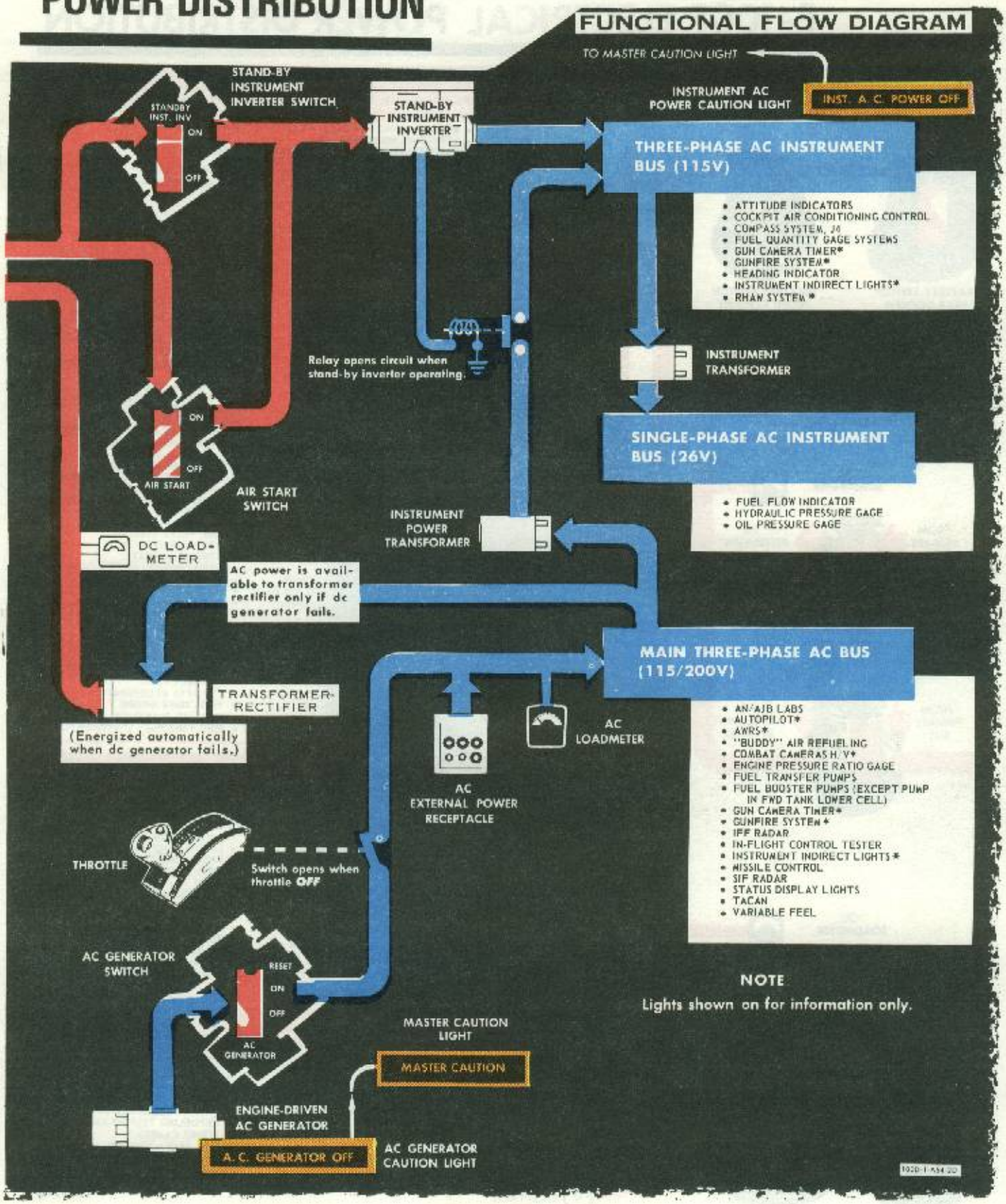


Figure 1-25 (Sheet 2 of 2)

F-100F ELECTRICAL POWER DISTRIBUTION

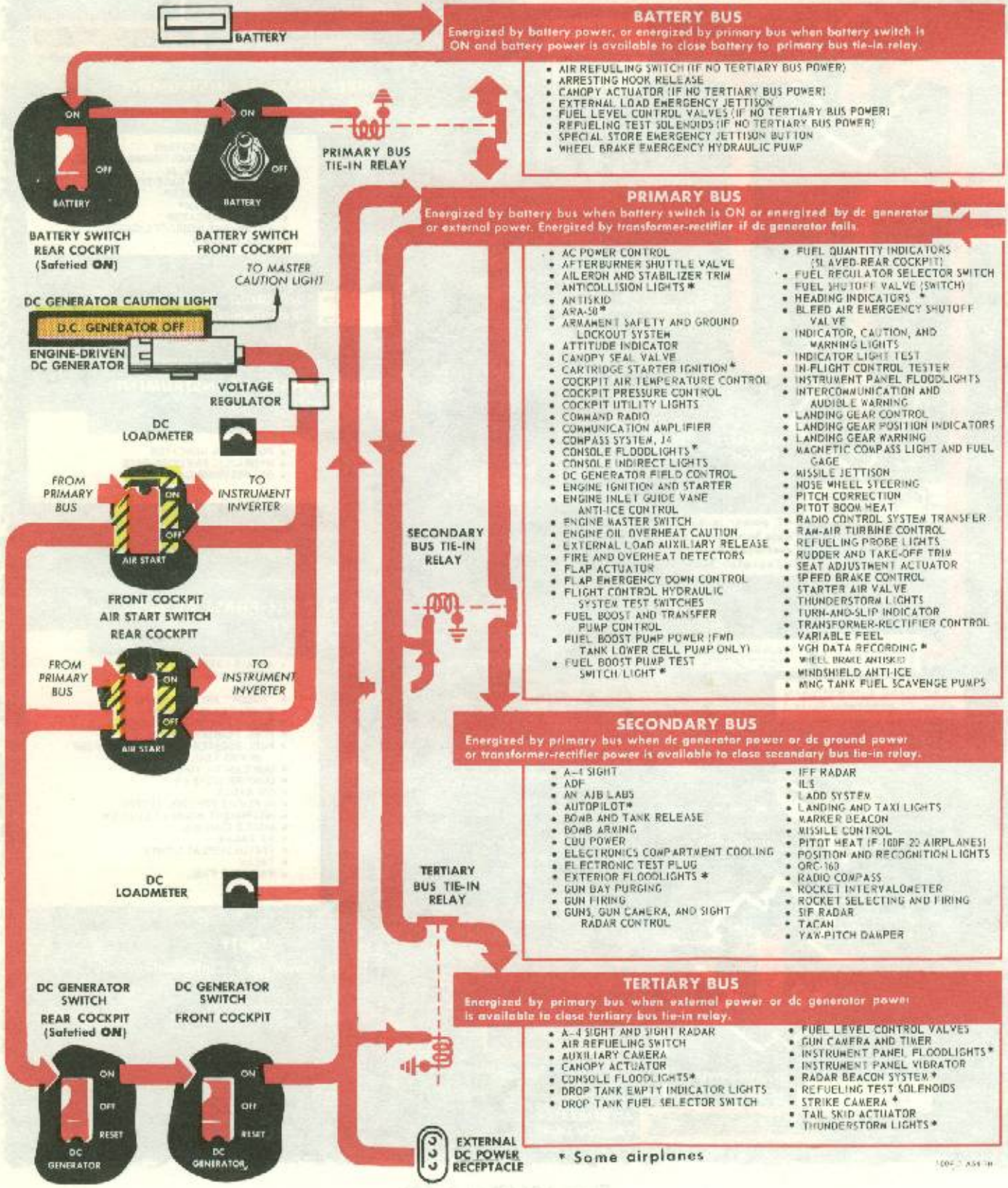
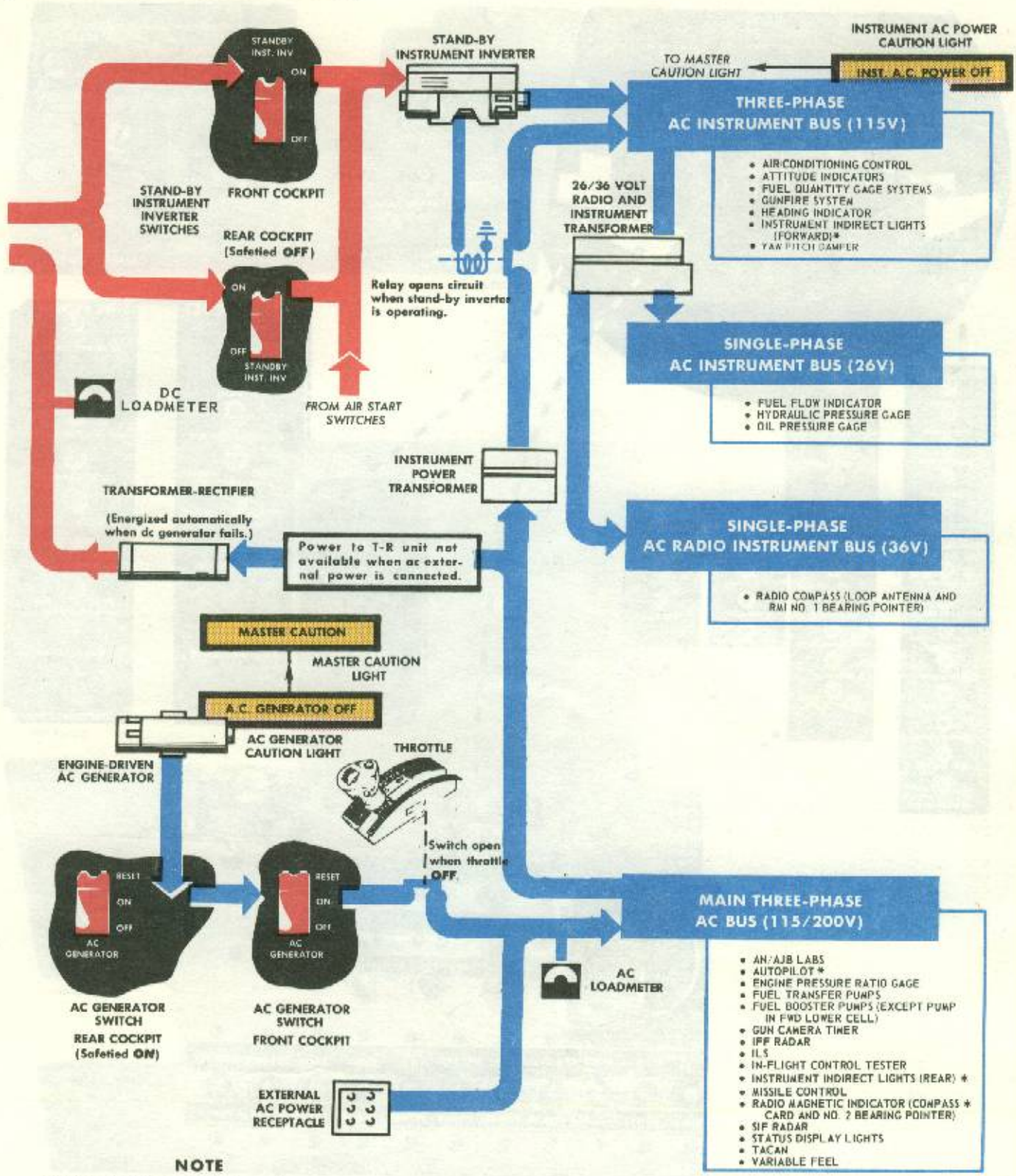


Figure 1-26 (Sheet 1 of 2)

FUNCTIONAL FLOW DIAGRAM



NOTE

Lights shown on for information only.

Figure 1-26 (Sheet 2 of 2)

F-100D CIRCUIT -BREAKER PANELS

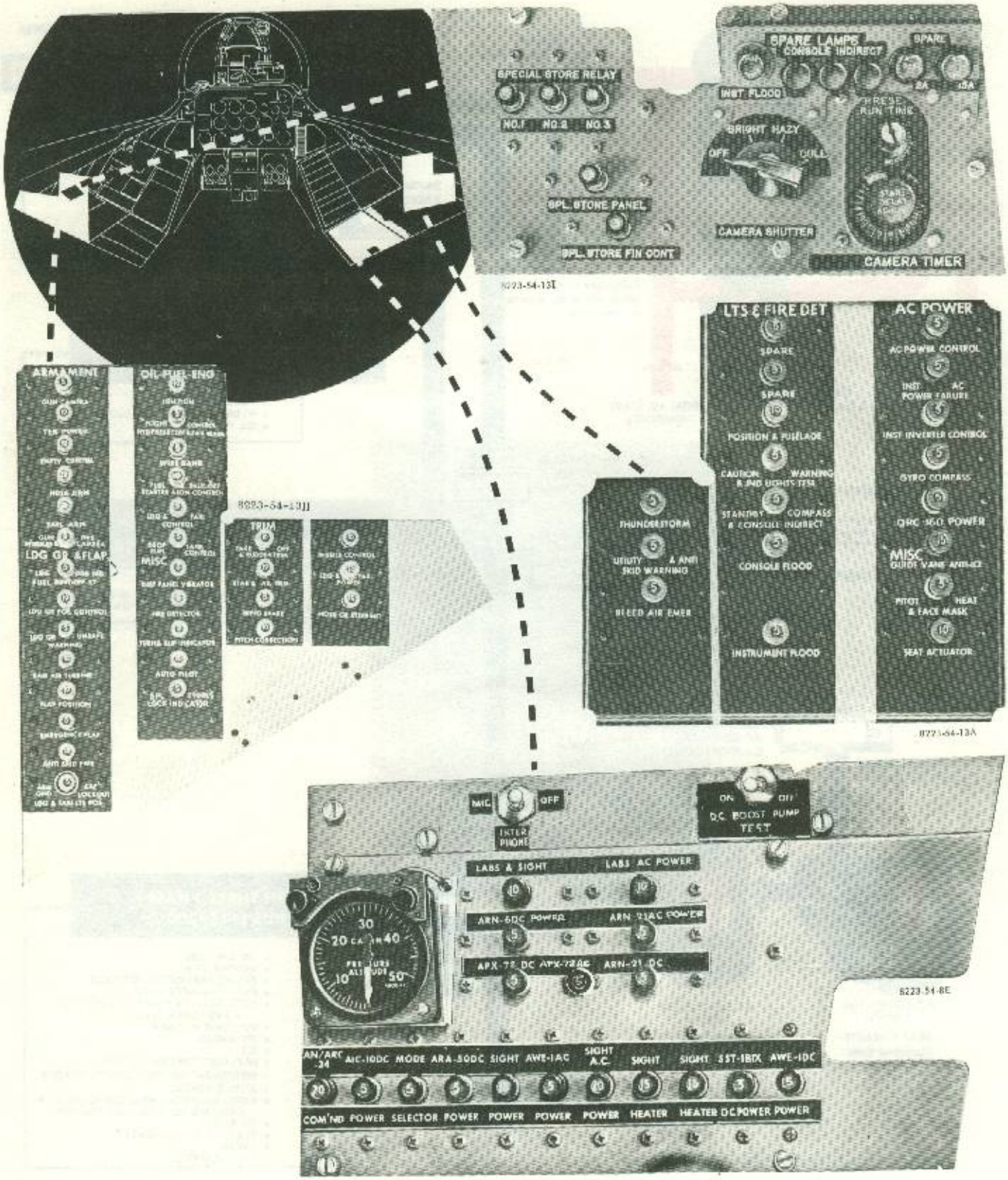


Figure 1-27 (Sheet 1 of 2)

(TYPICAL)

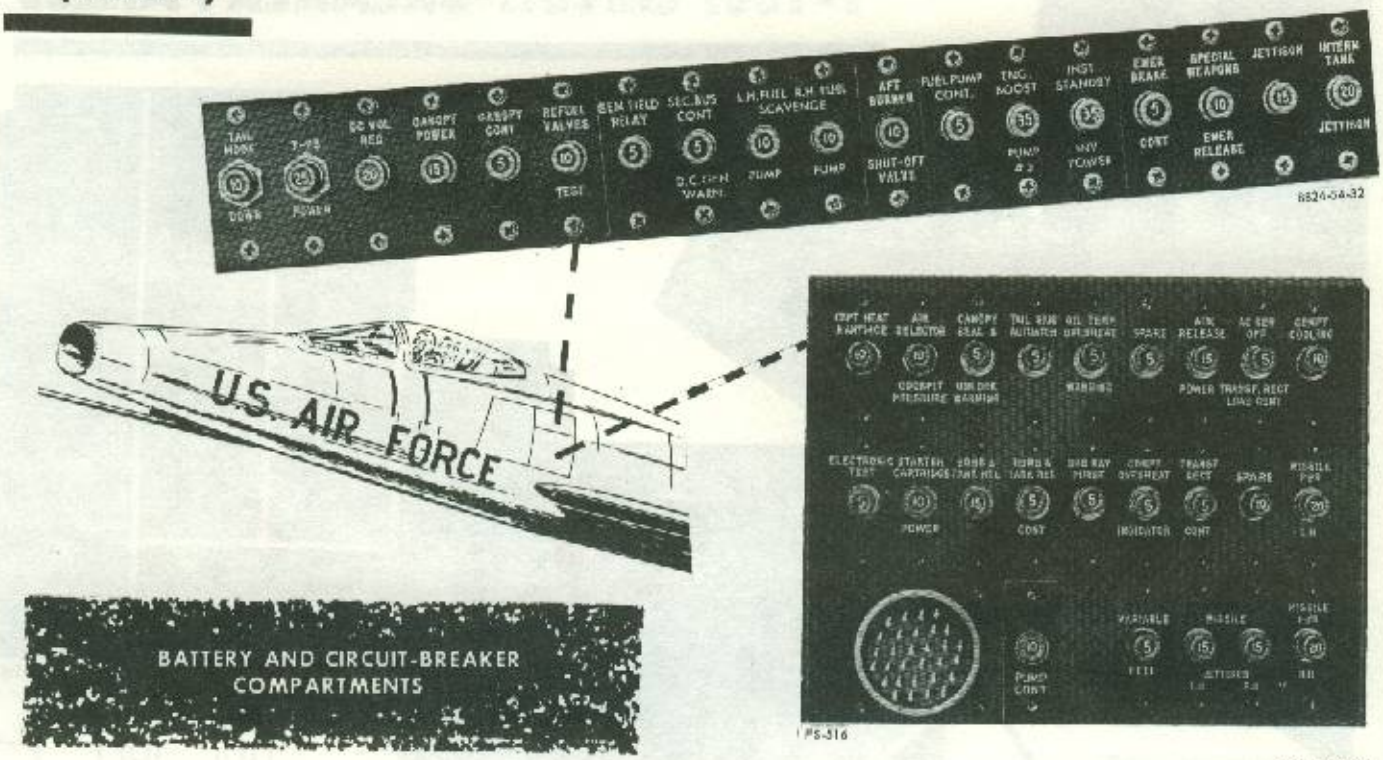


Figure 1-27 (Sheet 2 of 2)

AC Generator Switch.

A three-position switch (figure 1-29) controls the ac generator by means of primary bus power. (On F-100F airplanes, the rear cockpit switch is safetied ON to place complete control of the generator in the front cockpit because both switches must be ON for generator operation. The safety wire must be broken to move the rear cockpit generator switch to OFF or RESET.) When the ac generator switch is ON, the ac generator powers the main ac bus, provided the throttle is advanced from OFF. The ac generator is "off the line" when the ac generator switch is OFF, or the throttle is moved to the OFF position.

To restore ac generator output after it has been taken "off the line" by the generator switch being placed to OFF, generator switch should be held at RESET momentarily and released to ON. The RESET position is also used in an attempt to return the generator to the circuit, if a temporary malfunction has interrupted generator output. If it is desired to have ac external power when the engine is running, the ac generator switch must be OFF. The switch is spring-loaded from RESET to ON.

Standby Instrument Inverter Switch.

The standby instrument inverter switch (figure 1-29) controls primary bus power to energize the standby instrument inverter. The switch must be moved to ON to energize the instrument busses if the ac generator is not supplying power. (On F-100F airplanes, to shut down the inverter, the standby instrument inverter switches in both cockpits must be OFF. The switch in the rear cockpit is safetied at OFF.)

NOTE

When the air start switch is ON, the power source for the ac instrument busses are transferred automatically from the ac generator to the standby inverter, even though the inverter switch is OFF.

F-100F CIRCUIT-BREAKER PANELS

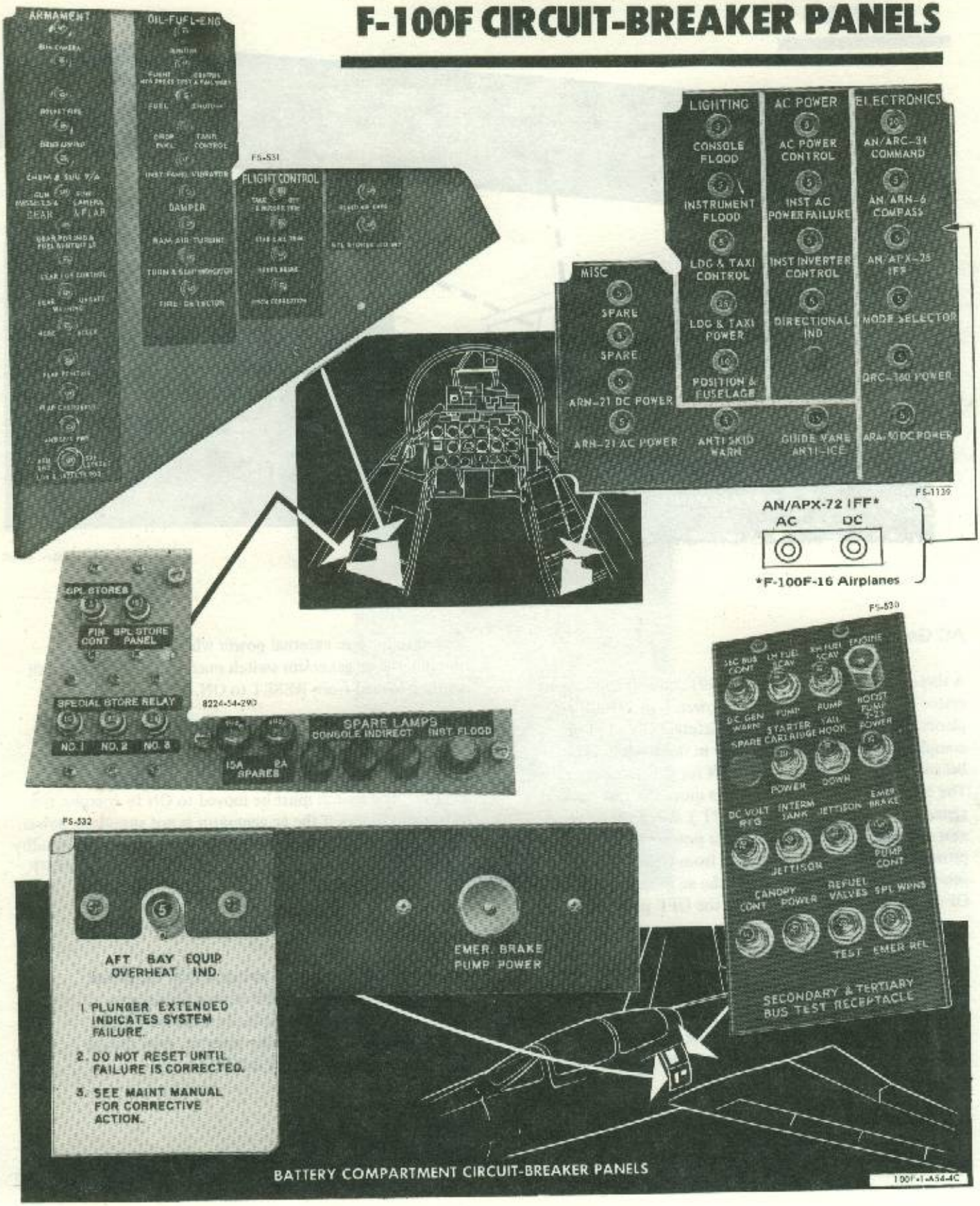


Figure 1-28 (Sheet 1 of 2)

(TYPICAL)

F-100F REAR COCKPIT

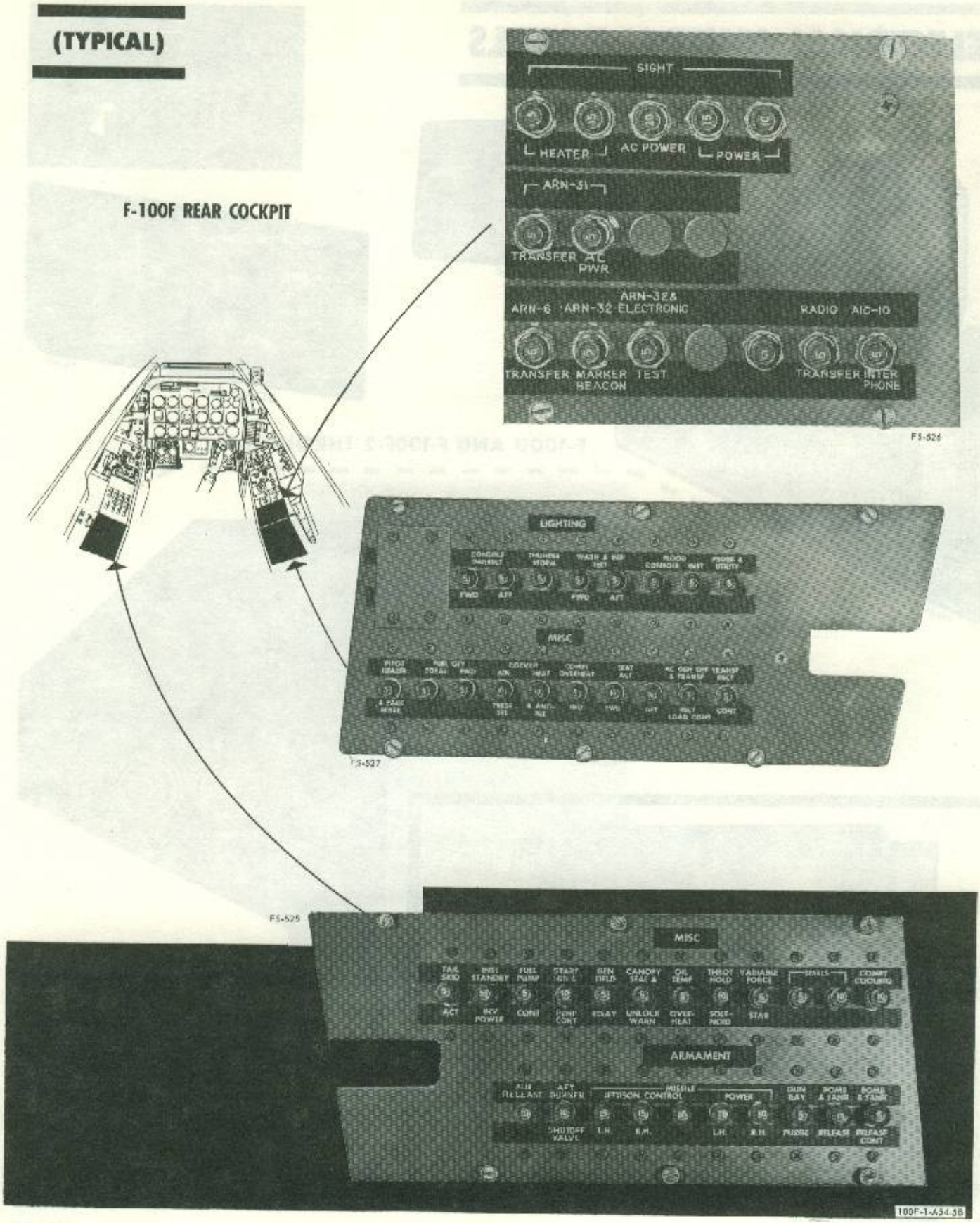
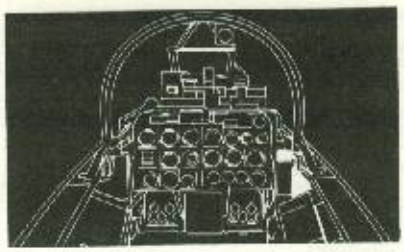
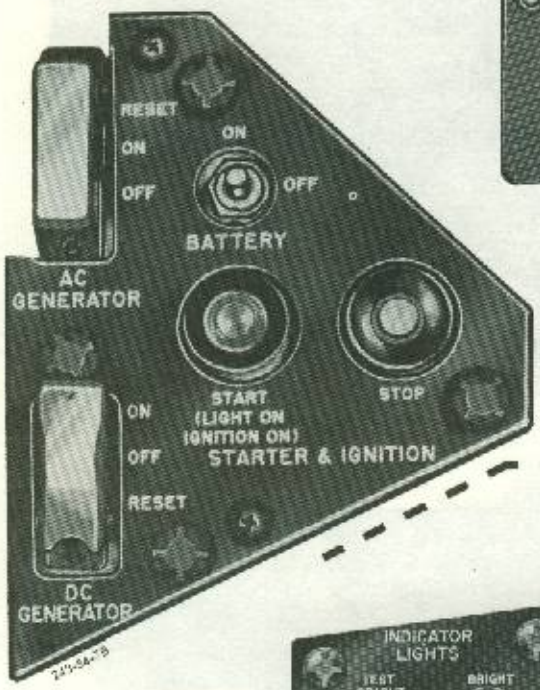


Figure 1-28 (Sheet 2 of 2)

ELECTRICAL CONTROL PANELS

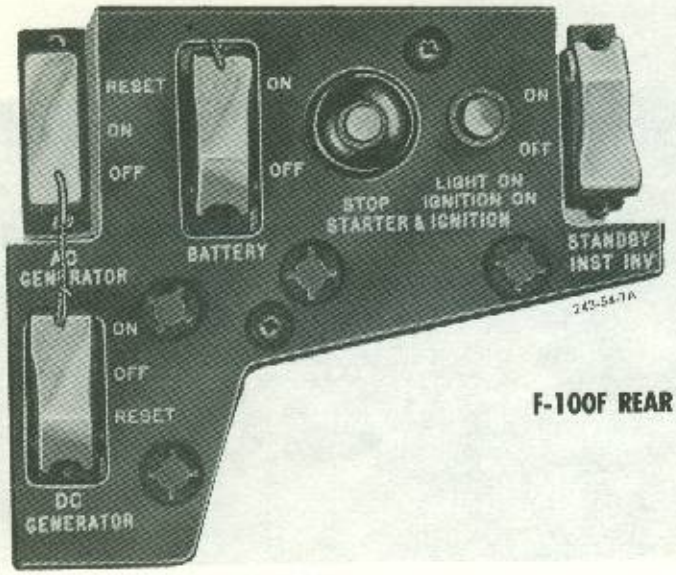




SOME AIRPLANES

F-100D AND F-100F-2 THROUGH-16

F-100F-20



F-100F REAR COCKPIT

Figure 1-29

ELECTRICAL POWER SUPPLY SYSTEM INDICATORS.

DC Generator Loadmeter.

The dc loadmeter (5, figures 1-6 and 1-7; 26, figure 1-12; 3, figure 1-13) shows the electrical load being drawn from the dc generator in terms of percentage of total generator output. The loadmeter also shows the dc load being drawn whenever the transformer-rectifier (standby dc system) is in operation.

AC Generator Loadmeter.

The ac loadmeter (3, figure 1-6; 4, figure 1-7; 25, figure 1-12; 1, figure 1-13) shows the electrical load being drawn from the ac generator in terms of percentage of total generator output.

DC Generator Caution Light.

The placard-type dc generator caution light (figure 1-19) is illuminated by primary bus power whenever the reverse-current relay is open. This shows a relay or generator failure or generator undervoltage (generator voltage less than battery voltage). Should generator voltage output become excessive (over 31 volts), the generator is automatically cut out of the circuit and the dc generator caution light comes on. Illumination of this light shows that the tertiary bus is inoperative and that the primary and secondary busses are being powered from the main ac bus through the transformer-rectifier. Bulbs within the dc generator caution light can be checked by the indicator light test circuit.

AC Generator Caution Light.

The placard-type caution light (figure 1-19) is illuminated by power from the primary bus, whenever the ac generator is "off the line." Illumination of the light shows that the main 3-phase ac bus is inoperative. Bulbs within the ac generator caution light can be checked by the indicator light test circuit.

Instrument AC Power-off Caution Light.

This placard-type caution light (figure 1-19) is powered by the primary bus. The light comes on when the 3-phase instrument bus voltage is less than 90-95 volts, or when the bus is not energized. If the standby instrument inverter is engaged because of ac generator failure, illumination of this light shows inverter failure. Bulbs within the caution light can be checked by the indicator light test circuit.

NOTE

When the air start switch is moved to ON to provide ignition for an air start, the standby instrument inverter is engaged automatically. The instrument ac power-off caution light blinks during this condition as the inverter comes up to speed.

HYDRAULIC POWER SYSTEMS.

The airplane has three separate, constant-pressure-type hydraulic systems: the utility hydraulic system, and two flight control hydraulic systems (identified as flight control system No. 1 and flight control system No. 2). Each system is independently pressurized by an engine-driven, 3000 psi hydraulic pump. In addition, flight control system No. 2 has a ram-air turbine-driven hydraulic pump for emergency power. The hydraulic fluid specification and hydraulic system ground test panel are shown in figure 1-41.

UTILITY HYDRAULIC SYSTEM.

The utility hydraulic system (figure 1-30) is a 3000 psi, pressure-type (closed center) system. It supplies hydraulic pressure to the following units: rudder, landing gear and fairing doors, wheel brakes, wing flaps, nose wheel steering, speed brake, automatic flight control system, gun and purge door, and the doors of the ram-air-driven turbine that powers the flight control hydraulic system emergency pump. Fluid is supplied to the system from a 2.7-gallon reservoir, and system pressure is maintained by a variable-volume, engine-driven hydraulic pump. On F-100F airplanes, fluid is supplied to the system from a 4.2-gallon reservoir. The rudder is powered by the utility hydraulic system. If this pressure fails or becomes too low, No. 2 flight control hydraulic system pressure is automatically supplied to the rudder.

HYDRAULIC PRESSURE GAGE AND GAGE SELECTOR SWITCH.

The hydraulic pressure gage (15, figure 1-6; 17, figure 1-7; 13, figure 1-12; 10, figure 1-13) and the gage selector

UTILITY HYDRAULIC

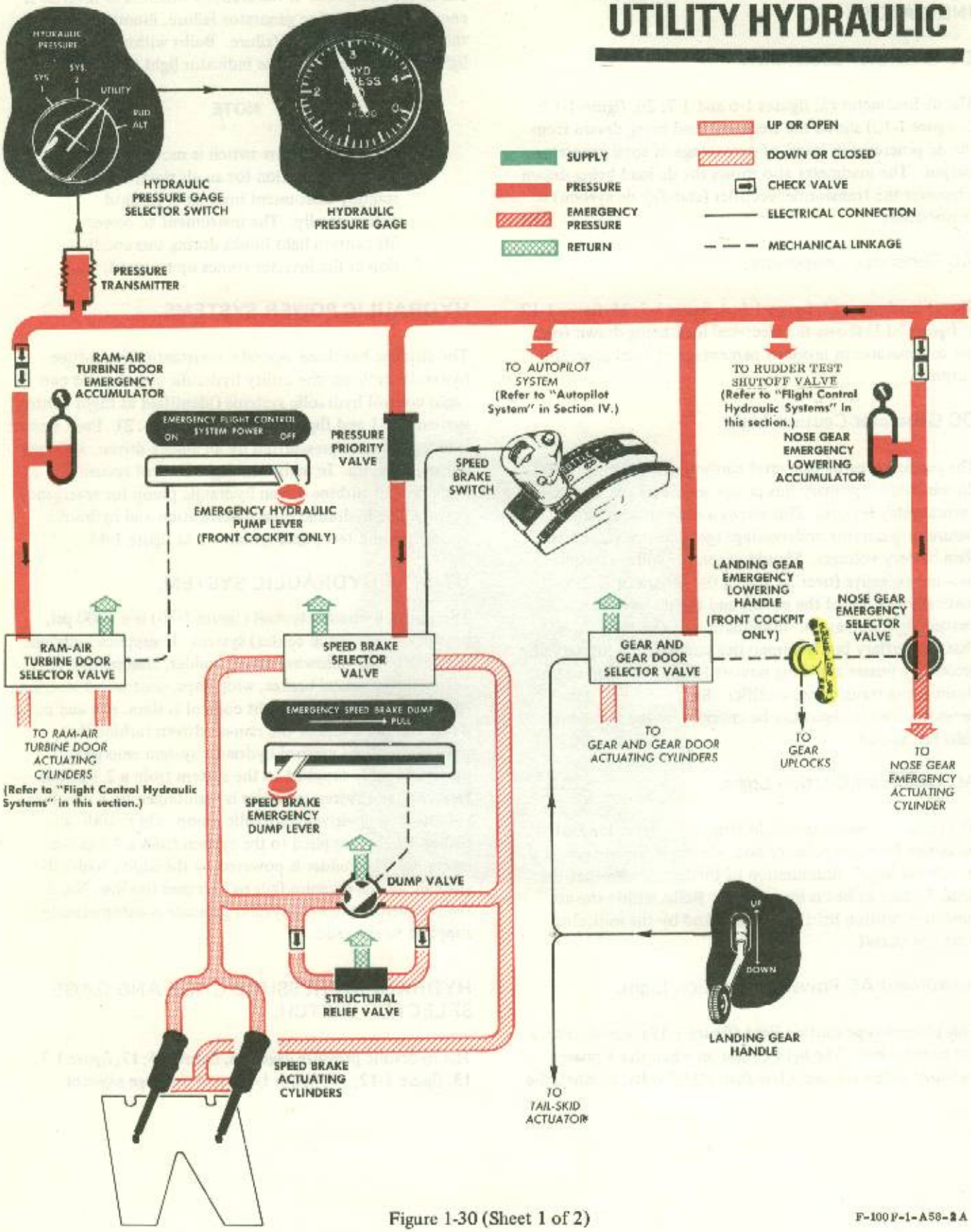


Figure 1-30 (Sheet 1 of 2)

F-100F-1-A58-2A

SYSTEM

NOTE
Caution lights shown illuminated for information only.

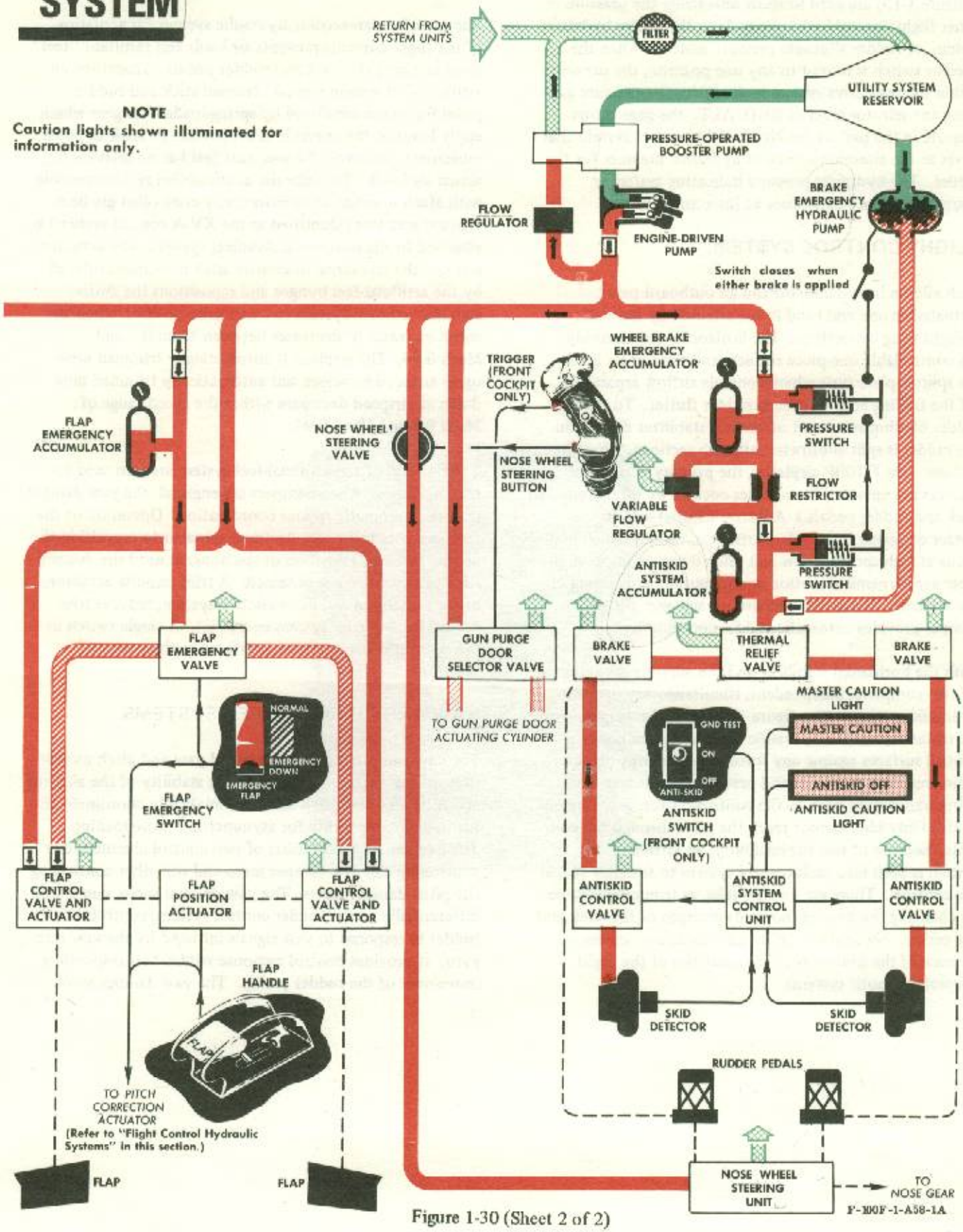


Figure 1-30 (Sheet 2 of 2)

switch (14, figure 1-6; 15, figure 1-7; 12, figure 1-12; 9, figure 1-13) are used to show selectively the pressure in either flight control hydraulic system, the utility hydraulic system, or rudder alternate pressure system. When the selector switch is moved to any one position, the pressure within the respective system is shown by the pressure gage. When the selector is set at RUD. ALT., the gage shows pressure in the part of the No. 2 flight control system that serves as the alternate source of hydraulic pressure for the rudder. The hydraulic pressure indicating system is energized by the single-phase ac instrument bus.

FLIGHT CONTROL SYSTEM.

Each aileron has an inboard and an outboard panel (actuated as one unit) and is not affected by normal in-flight wing deflections. The horizontal tail assembly is a controllable one-piece surface, and the rudder is of the splitter-plate type which controls airflow separation off the trailing edge to reduce rudder flutter. To prevent rudder binding as a result of vertical stabilizer deflection, the rudder is split chordwise into two sections, joined by a hinge. (On F-100F airplanes, the primary flight control surfaces are operated from either cockpit by interconnected stick and rudder pedals.) A flutter damper unit in the rudder dampens any rudder flutter (vibration) which may occur at high speeds. Yaw and pitch dampers improve the basic aerodynamic direction and longitudinal damping of the airplane. When the dampers are engaged, the yaw damper provides automatic rudder coordination.

Both the horizontal stabilizer and the ailerons are actuated by two complete, independent, simultaneously operating hydraulic systems. (See figure 1-33.) The irreversible characteristics of the hydraulic control system hold the control surfaces against any forces that do not originate from control movements and prevent these forces from being transmitted back to the controls. Thus, aerodynamic loads of any kind cannot reach the pilot through the controls. Because of this irreversibility, an artificial-feel system is built into each control system to simulate feel at the controls. There are no trim tabs, as trimming is done by changing the neutral (no-load) position of the stick and the pedals. No control surface gust locks are necessary, because of the irreversible characteristics of the flight control hydraulic systems.

ARTIFICIAL-FEEL AND TRIM SYSTEMS.

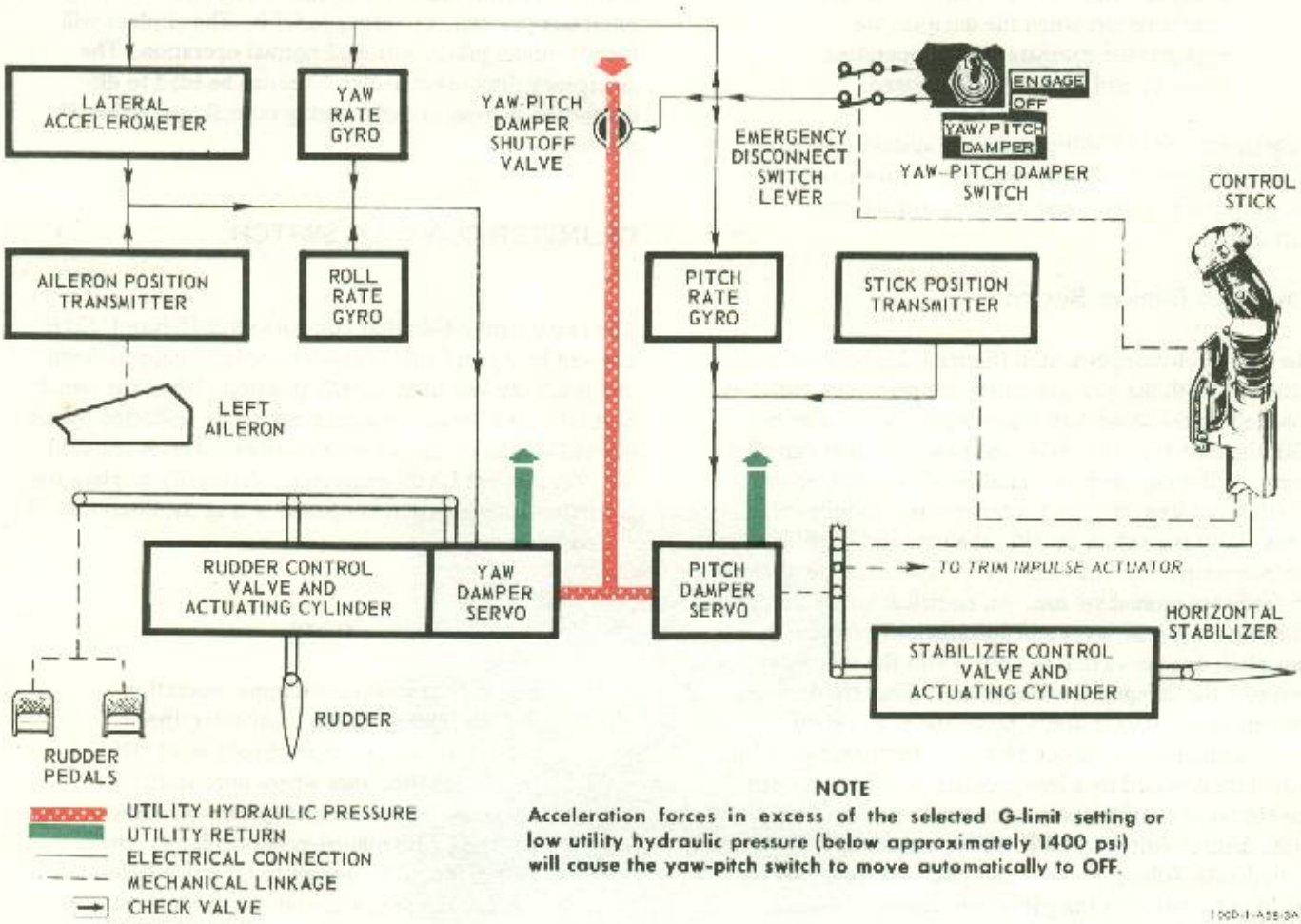
The use of an irreversible hydraulic system for actuation of the flight controls prevents air loads and resultant "feel" from reaching the stick and rudder pedals. Therefore, an artificial-feel system is used. Normal stick and rudder pedal forces are simulated by spring-loaded bungees which apply loads to the controls in proportion to stick or pedal movement; however, the resultant feel has no relation to actual air loads. To make the artificial-feel system variable with Mach number, an electronically controlled gradient-changer actuator (identified as the XV-A control system) is attached to the horizontal stabilizer system. The actuator changes the resistance to control stick movement offered by the artificial-feel bungee and repositions the entire stabilizer control system to a new trim position when airspeed increases or decreases between Mach 0.8 and Mach 0.94. The airplane is automatically trimmed nose up as airspeed increases and automatically trimmed nose down as airspeed decreases within the speed range of Mach 0.8 and Mach 0.94.

The bungees of the artificial-feel system are also used for trim purposes. When dampers are engaged, the yaw damper provides automatic rudder coordination. Operation of the trim switches causes an electrical actuator to reposition the neutral (no-load) position of the bungees until the desired stick or pedal force is obtained. A trim impulse actuator, in the horizontal stabilizer control system, reduces trim system lag and trim system overshoot. A single switch in the cockpit trims all control surfaces to their takeoff positions.

YAW AND PITCH DAMPER SYSTEMS.

The yaw and pitch dampers control yaw and pitch oscillations of the airplane. They increase stability of the airplane about its yaw and pitch axes, automatically coordinate all turns, and compensate for asymmetrical store loading. The damper system consists of two control channels, one controlling the yaw damper servo and the other controlling the pitch damper servo. The yaw damper servo, connected differentially to the rudder control system, controls the rudder in response to yaw signals initiated by the yaw rate gyro. It provides control response without corresponding movement of the rudder pedals. The yaw damper servo

YAW AND PITCH DAMPER SYSTEMS



NOTE

Acceleration forces in excess of the selected G-limit setting or low utility hydraulic pressure (below approximately 1400 psi) will cause the yaw-pitch switch to move automatically to OFF.

Figure 1-31

also controls the rudder to provide turn coordination in response to signals from the lateral accelerometer. However, the yaw damper system can be overridden by moving the rudder pedals to provide an intentional sideslip.

The pitch damper servo is connected differentially to the horizontal stabilizer control system and controls the stabilizer in response to pitch signals initiated by the pitch rate gyro. It provides control response without a corresponding movement of the control stick. The control stick position transmitter also supplies signals to the pitch damper. This signal is provided to prevent the damper from opposing stick pressure during pilot-initiated maneuvers. When the dampers are engaged, normal rudder trimming is not required. The rudder trim switch should not be used with dampers engaged. Pitch and roll trimming when only the dampers are engaged is accomplished in the normal manner by use of the lateral and longitudinal trim switch on the stick grip.

CAUTION

If any mistrim in yaw or pitch attitude change occurs when the dampers are engaged, the dampers are not operating normally and should be disengaged.

A check valve in the utility hydraulic system return part of the pitch damper servo prevents excessive return pressure surges that could cause a pitch control system malfunction.

Yaw-Pitch Damper Switch.

The yaw-pitch damper switch (figure 1-22) controls simultaneously both the yaw and pitch dampers. The switch is powered by the secondary bus and main 3-phase ac bus. With the switch at ENGAGE, the yaw and pitch dampers damp oscillations, increase stability of the airplane about its pitch and yaw axes, and automatically coordinate all turns. With the switch at OFF, the yaw and pitch dampers are inoperative, but the electrical components are warmed up ready for immediate use. An electrical power failure de-energizes the dampers and automatically returns the yaw-pitch damper switch to OFF. With the switch in this position, the dampers are inoperative. The dampers also become inoperative if utility hydraulic pressure drops below a minimum of about 1400 psi. Pressure below this minimum is sensed by a low-pressure switch that opens the electrical circuits and automatically positions the yaw-pitch damper switch to OFF. After momentary electrical or hydraulic failure, the yaw and pitch dampers may be restored by resetting the yaw-pitch damper switch to ENGAGE 1-1/2 minutes after electrical power has been restored or immediately after utility hydraulic pressure returns to about 1900 psi. If complete electrical or hydraulic failure occurs in the yaw and pitch dampers, the yaw and pitch damper servos recenter automatically and lock mechanically. The servos are then fixed and serve as conventional control links to allow directional control for continued flight. The yaw and pitch dampers can be disengaged at any time by moving the yaw-pitch damper switch to OFF, or by use of the damper emergency disconnect switch lever.

Damper Emergency Disconnect Switch Lever.

The damper emergency disconnect switch lever (figure 1-32) is on the control stick. Pressing this switch causes the yaw-pitch damper switch to move to OFF. The airplane will then be under pilot-controlled normal operation. The emergency disconnect switch lever can be used to disengage the damper systems during normal operations, if desired.

G-LIMITER CONTROL SWITCH.

The two-position G-limiter control switch (figure 1-22) is covered by a guard that is safety-wired and clipped down and holds the switch in the ON position. When the switch is at OFF, the damper system is no longer restricted by the preset G-limits of the G-limiting system. This is required only for combat LABS maneuvers. Authority to place the G-limiter control switch in operation is at the discretion of the commander.

NOTE

For normal yaw-pitch damper operation and for training LABS maneuvers, the G-limiter control switch should be at ON. In combat situations where unrestricted operation of the damper system throughout the LABS maneuver is mandatory, the switch should be moved to OFF just before the LABS run-in is initiated and returned to ON immediately after completion of the maneuver.

CAUTION

It is possible to exceed the G-limits of the airplane or external stores when the switch is at OFF.

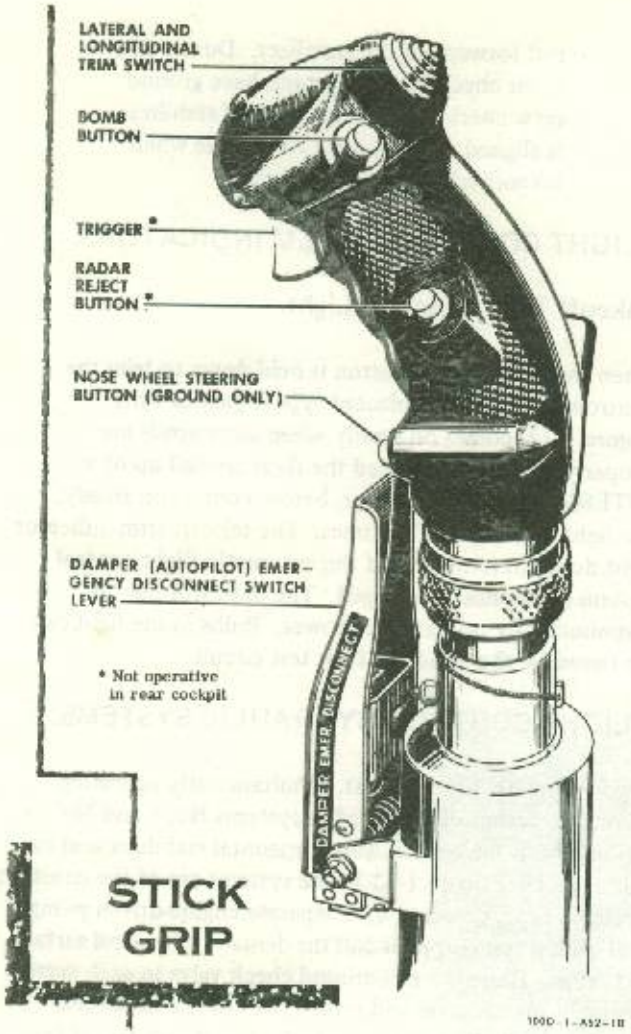


Figure 1-32

FLIGHT CONTROLS.

Control Stick.

The control stick is mechanically connected to hydraulic control valves at the aileron and horizontal stabilizer hydraulic actuators. The control stick grip (figure 1-32) incorporates the lateral and longitudinal trim switch, nose wheel steering button, trigger, bomb button, and radar reject button. The damper emergency disconnect switch lever is on the control stick below the grip. On F-100F airplanes, both sticks are interconnected and the trigger and radar reject button in the rear cockpit are inoperative.

Rudder Pedals.

The hanging-type rudder pedals are mechanically linked to hydraulic control valves at the rudder actuator. The wheel brakes are applied by toe action on the rudder pedals. On F-100F airplanes, the rudder pedals in each cockpit are interconnected. Rudder pedal movement also controls nose wheel steering.

PEDAL ADJUSTMENT. The rudder pedals can be individually adjusted fore and aft by an adjustment lever outboard of each pedal. Coordinated pedal alignment during adjustment is obtained by index numbers on each pedal adjustment ratchet. When the index numbers at each pedal correspond, the pedals are evenly adjusted.

Lateral and Longitudinal Trim Switch.

Lateral and longitudinal trim is controlled by a five-position thumb-actuated switch. (See figure 1-32.) The trim circuit is powered by the primary bus. Holding the trim switch to the right or left energizes the electrical lateral trim actuator; holding the switch forward or aft energizes the longitudinal trim actuator. When a trim actuator is energized, it repositions its respective artificial-feel bungee to a new neutral or no-load position. The trim impulse actuator, in the stabilizer trim system to reduce trim system lag and overshoot, is actuated by the longitudinal trim switch. When the stabilizer trim circuit is energized, pressure from flight control hydraulic system No. 2 causes the impulse actuator to move the stabilizer control valve before the trim actuator motor responds to the trim switch.

NOTE

When the switch is positioned for longitudinal trim, a slight jolt in the stick may be noticed. This is normal and is caused by action of the trim impulse actuator.

The trim switch is spring-loaded to OFF (center). When the switch is released, the trim action stops, and the trim impulse actuator returns to its normal position. This reduces overtravel of the stabilizer electrical trim actuator.

Rudder Trim Switch.

Rudder trim operation is controlled by primary bus power through a three-position switch. (See figure 1-22.) When

the switch is held at either LEFT or RIGHT, the rudder trim actuator is energized accordingly and repositions the rudder system artificial-feel bungee to a new neutral (no-load) position. The rudder trim switch is spring-loaded to OFF (center), and trim action stops when the switch is released. The rudder trim switch remains operative when the yaw damper is engaged.

Takeoff Trim Button.

All control surfaces can be trimmed at the same time to proper position for takeoff by a pushbutton switch. (See figure 1-22.) Pressing the button supplies primary bus power to the trim actuators, which reposition the artificial-feel bungees to obtain the correct control surface settings for takeoff. (The takeoff trim position of the ailerons and rudder is within ±1 degree of neutral; the leading edge of the horizontal stabilizer is set down about 4-1/2 degrees (5-1/2 degrees*) from neutral to give an airplane nose-up condition.) The airplane is then trimmed for about 260 knots IAS. This speed is for a clean airplane of about 27,000 pounds (29,000 pounds*) gross weight. (The trim speed can vary considerably with changes in gross weight, center of gravity, altitude, external store loading, and friction within the control system.) Above this speed, it may be necessary to retrim the airplane because of the trim tolerance.

When the button is held down, an indicator light comes ON, indicating the control surfaces have reached the proper position for takeoff if the wing flaps are full up or at INTERMEDIATE. To stop trim operation, the button must be released.

CAUTION

Illumination of the takeoff trim light indicates flight control actuators are positioned. Malfunctions are possible where the takeoff trim light is on and control surfaces are not properly aligned.

NOTE

The takeoff trim setting of the horizontal stabilizer is identified by a white triangle painted on the left side of the fuselage, just forward of the stabilizer. During preflight check of trim system, have ground crew check that leading edge of stabilizer is aligned with aft apex of triangle when takeoff trim button is pressed.

FLIGHT CONTROL SYSTEM INDICATORS.

Takeoff Trim Indicator Light.

When the takeoff trim button is held down to trim the controls for takeoff, a placard-type indicator light (figure 1-22) comes on steady when all controls are properly trimmed, provided the flaps are full up or at INTERMEDIATE. However, before coming on steady, the light may flash a few times. The takeoff trim indicator light does not remain on if the automatic flight control system (autopilot) is engaged. The indicator light is illuminated by primary bus power. Bulbs in the light can be tested by the indicator light test circuit.

FLIGHT CONTROL HYDRAULIC SYSTEMS.

Two complete, independent, simultaneously operating hydraulic systems (identified as systems No. 1 and No. 2) actuate both the controllable horizontal stabilizer and the ailerons. (See figure 1-33.) The systems are of the constant-pressure type, powered by a separate engine-driven pump, and each system supplies half the demand of control surface actuators. There is a run-around check valve in each system between the pressure and return lines of the horizontal stabilizer actuator to prevent lockup of the system if for any reason the return side becomes blocked. Failure of one system does not affect operation of the other system, which then assumes the entire load of flight control operation. Under such a condition, maximum force output is limited to half that of normal dual system operation. With a frozen engine, or if No. 2 engine-driven pump failure occurs, power in this system can be maintained by a ram-air turbine-driven emergency pump. The No. 2 flight control hydraulic system supplies pressure through the hydraulic summing valve to the rudder actuator to supplement the utility system power. Flight control hydraulic system No. 2 pressure is also used for the horizontal stabilizer trim impulse system.

The flight surface hydraulic actuator control valves are positioned mechanically by stick or rudder pedal

*F-100F Airplanes

movement. These valves then direct hydraulic power to the actuating cylinders to move the control surfaces. (The dual control valves, and tandem-type actuating cylinders are hydraulically independent of each other.) When the control surface moves, a follow-up mechanism returns the control valve to a neutral position, so that hydraulic flow to the actuating cylinder is shut off. The pressure in the actuating cylinder serves to hold the control surface in the desired position, and maintains irreversibility by means of check valves.

Flight Control Hydraulic System Emergency Pump.

The emergency hydraulic pump in flight control system No. 2 provides pressure for this system in case of engine failure or if failure of the system No. 2 engine-driven pump occurs. The emergency pump is powered by a ram-air driven turbine, in the upper part of the fuselage, behind the canopy. (See figure 7-3.) The ram-air turbine is automatically engaged in flight when engine rpm falls below about 40% rpm or can be engaged manually by a lever in the cockpit. When the emergency pump is selected, utility hydraulic system pressure opens air inlet doors in the engine intake duct, and an air outlet door on top of the fuselage. Ram-air from the engine inlet duct then flows through and drives the turbine. (Refer to Flight Control System Emergency Hydraulic Pump in section VII.)

NOTE

> The manual control actuates the turbine door hydraulic selector valve mechanically; the automatic system actuates the valve by primary bus power.

When the ram-air turbine doors are open, an audible warning sounds in the pilot's headset, and the warning light in the landing gear handle comes on when the airplane is below 10,000 feet with landing gear up. If utility hydraulic system pressure is not available to open the doors, the ram-air turbine door emergency accumulator in the system provides positive door opening. If the pump is no longer needed, it must be shut down manually, at which time utility hydraulic pressure closes the ram-air inlet and outlet doors to stop the turbine. The pump can be shut down only when engine rpm is above 45% rpm for sufficient output from the engine-driven pump in flight control hydraulic system No. 2. The nose gear load switch prevents automatic operation of the emergency pump when the airplane is on the ground. However, an externally mounted switch button permits the load switch to be overridden for a postflight operational check of the pump automatic starting system during engine shutdown.

Flight Control Emergency Hydraulic Pump Lever.

The flight control emergency hydraulic pump can be controlled manually by a lever (11, figure 1-10; 12, figure 1-11; 10, figure 1-15, and 9, figure 1-16), not in the rear cockpit. Pushing the lever forward to ON mechanically positions a hydraulic selector valve, so that the utility hydraulic system power opens the ram-air turbine air inlet and air outlet doors. This allows ram-air from the engine air intake duct to drive the turbine for operation of the emergency pump. (When the flight control hydraulic system test switch is held at its RAM TURB ON & SYS 2 OFF position for in-flight test of the pump or the test button is pushed, the lever moves automatically to the ON position.) To shut down the pump, the lever must be manually returned to OFF. When the lever is moved aft to OFF, utility hydraulic pressure closes the inlet and exhaust outlet doors to stop the turbine and shut down the pump.

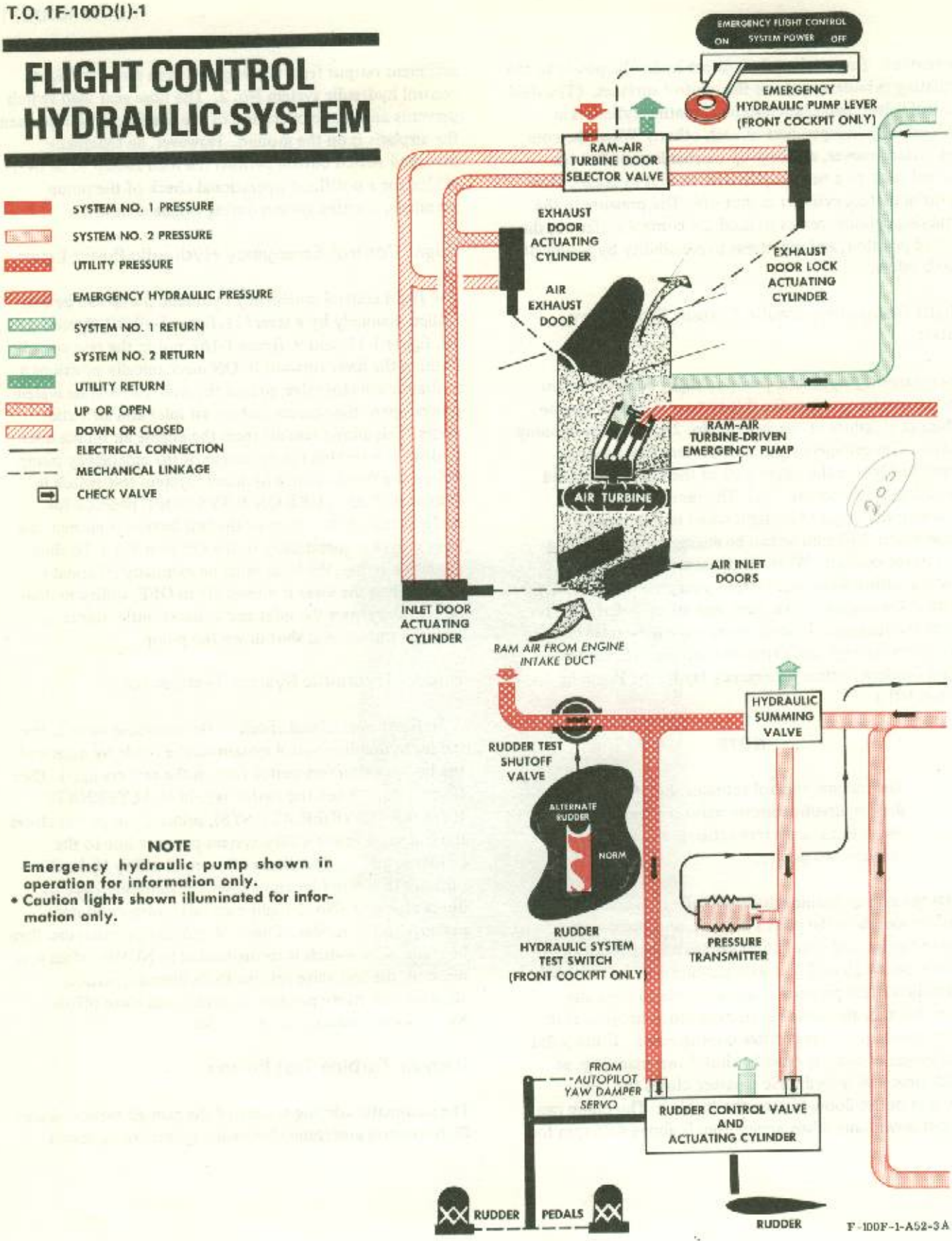
Rudder Hydraulic System Test Switch.

A preflight operational check of the summing valve in the rudder hydraulic control system can be made by means of the two-position test switch (not in the rear cockpit). (See figure 1-22.) When the switch is held at ALTERNATE RUDDER (RUDDER ALT SYS), primary bus power closes the test valve in the utility system pressure line to the rudder actuator. This shuts off normal utility hydraulic pressure to the rudder and allows the summing valve to direct alternate (No. 2 flight control) hydraulic system pressure to the rudder. Check of rudder operation can then be made. The switch is spring-loaded to NORM; when it is released, the test valve returns to its normal position, allowing the utility pressure to return and close off the No. 2 system pressure to the rudder.

Ram-air Turbine Test Button.

The automatic starting system of the ram-air turbine-driven flight control emergency hydraulic pump can be tested

FLIGHT CONTROL HYDRAULIC SYSTEM



NOTE

Emergency hydraulic pump shown in operation for information only.

• Caution lights shown illuminated for information only.

Figure 1-33 (Sheet 1 of 2)

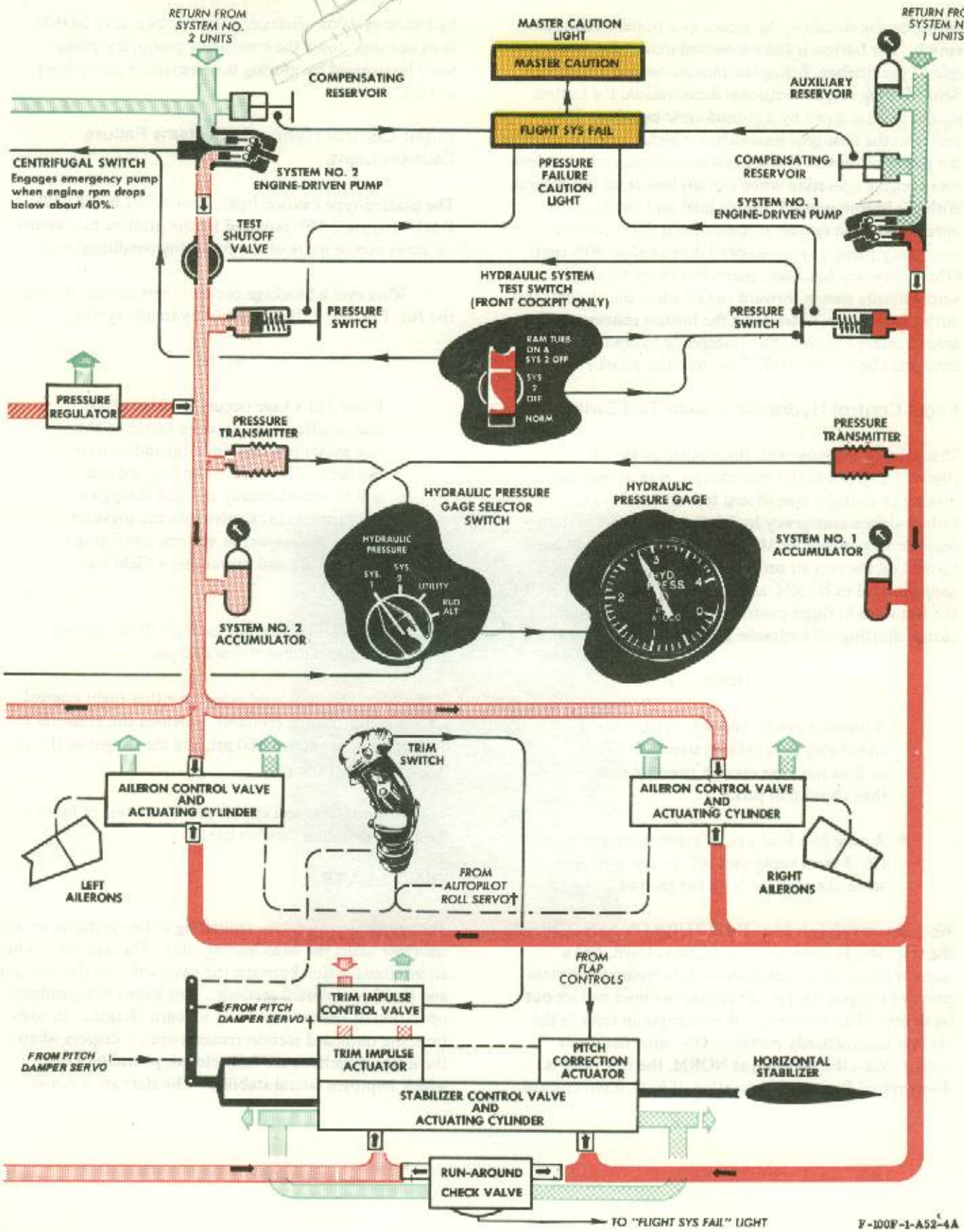


Figure 1-33 (Sheet 2 of 2)

during engine shutdown by means of a pushbutton-type switch. The button is flush-mounted externally on the left side of the fuselage, below the ram-air turbine air outlet door. During engine shutdown deceleration, the button should be held down by a ground crew member. This overrides the nose gear load switch which normally prevents the pump automatic starting system from operating at low-speed engine operation when the airplane is on the ground. With the button pressed, the air inlet and outlet doors open to start the ram-air turbine-driven flight control emergency pump as engine speed drops below 40% rpm. (The emergency hydraulic pump lever in the cockpit automatically moves forward to ON when the air inlet and outlet doors open.) Releasing the button restores the ground safety circuit. (The emergency hydraulic pump lever must be moved to OFF to close the doors.)

Flight Control Hydraulic System Test Switch.

This primary-bus-powered, three-position switch (figure 1-22), not in the rear cockpit, is used only for making an in-flight operational test of the ram-air turbine-driven emergency hydraulic pump after certain periodic inspections, or after maintenance has been performed on the ram-air turbine system. The switch is spring-loaded to NORM, and when it is held at SYS 2 OFF, the test valve in flight control hydraulic system No. 2 closes, shutting off hydraulic pressure in this system.

NOTE

A pressure switch prevents system No. 2 from being shut off if system No. 1 fails or does not have enough pressure (less than about 650 psi).

- A nose gear load switch prevents system No. 2 from being shut off for test purposes while the airplane is on the ground.

When the switch is held at RAM TURB ON SYS 2 OFF, the test valve in system No. 2 remains closed, and a selector valve opens that allows utility hydraulic system pressure to open the ram-air turbine air inlet and air outlet doors. (The emergency hydraulic pump lever in the cockpit automatically moves to ON when the doors open.) When the switch is at NORM, the test valve is de-energized for normal operation of both flight control hydraulic systems. Releasing the test switch to NORM does not shut down the emergency pump; the pump must be stopped by moving the emergency pump lever to OFF.

Flight Control Hydraulic Systems Failure Caution Light.

The placard-type caution light, labeled "FLIGHT SYS. FAIL" (figure 1-19), powered by the primary bus, comes on when one or more of the following conditions exist.

1. Whenever a blockage occurs in return line of either the No. 1 or No. 2 flight control hydraulic system.

NOTE

When a blockage occurs in either return line, a differential pressure switch in the line senses pressure build-up and turns on the light. The fluid in the blocked line will be automatically relieved through a run-around check valve into the pressure side of the respective system, equalizing the pressure and preventing a flight control lockup.

2. When pressure in either or both flight control hydraulic systems drops below 650 psi.

3. When the fluid level is low in either flight control system compensating reservoir, provided the pressures in both systems are above 650 psi, and the weight of the airplane is on the nose gear.

The indicator light test circuit provides a test of bulbs in the pressure failure caution light.

WING SLATS.

The moveable wing slats, consisting of five sections, extend spanwise along the wing leading edge. The slats do not have an interconnection between the two outboard slat sections and the three inboard sections. This allows independent operation of the outboard and inboard sections. In addition, the outboard section remains open 2 degrees when the inboard sections are fully closed, providing a slot which improves lateral stability. The slats are actuated

automatically by aerodynamic forces. An increase in airspeed closes the slats; the slats extend when airspeed is reduced. When the slats extend, the slot formed along the leading edge changes airflow characteristics to reduce stalling speeds in both accelerated and unaccelerated flight. Depending on the angle of attack, the slats float to either closed, partly open, or full open positions. Except at extreme altitudes, the slats remain mostly closed in climbing or cruising flight to offer minimum drag for maximum flight performance.

WING FLAP SYSTEM.

The wing flaps are electrically controlled and hydraulically operated and cover a spanwise area from the fuselage to the inboard aileron on each wing. The flap hydraulic actuator valves are interconnected for proper synchronization of the flaps. The flaps extend full down (45 degrees) in about 10 seconds, and retract in about 14 seconds. Flap extension varies with the airspeed at which the flaps are operated. There is a preset 20 degree intermediate flap position to improve air refueling and takeoff capabilities. Change in airplane attitude due to fully lowering or raising of the flaps is compensated for by a pitch correction actuator which changes the horizontal stabilizer position relative to control stick position during flap operation. There is no flap position indicator.

NOTE

> Pitch correction actuation depends upon full DOWN flap handle movement or the emergency switch actuation and not on actual wing flap operation.

F-100F-20 airplanes have ducted wing flaps which have a full flap travel of 40 degrees. When the ducted wing flaps are lowered, a spanwise duct, built into the leading edge, directs air from the lower surface of the flaps to the upper surface for reducing wing flap buffet when in landing configuration. The flap retraction time on these airplanes is about 4 seconds. Flap extension varies with the airspeed at which the flaps are operated.

WING FLAP HANDLE.

The airfoil-shaped wing flap handle (12, figures 1-8 and 1-9; 13, figure 1-14; 9, figure 1-17) controls flap operation. On F-100F airplanes the wing flap handles are interconnected mechanically so that positioning the handle in either cockpit has a corresponding movement on the other. To position the flaps full up, intermediate, or full down, the flap lever is moved to UP, INTERMEDIATE, or DOWN. This directs primary bus power to the flap position actuator which mechanically positions hydraulic actuator valves to permit utility hydraulic pressure to position the flaps. When the flap handle is moved into or out of DOWN, the pitch correction actuator is energized by the primary bus to provide pitch trim change by altering the position of the horizontal stabilizer. Pitch trim change occurs when the handle is moved to or from DOWN even if a malfunction prevents flap operation. The INTERMEDIATE position does not energize the pitch correction actuator.

WING FLAP EMERGENCY SWITCH.

A two-position switch (14, figures 1-8, 1-9, and 1-14; 11, figure 1-17) permits the flaps to be lowered if use of the flap handle does not lower the flaps. The switch controls primary bus power to the solenoid-operated flap emergency hydraulic valve. When the switch is at NORMAL, the emergency valve is de-energized and the flaps operate in the normal manner. If the flaps fail to lower when the flap handle is moved to DOWN, because of utility system failure or electrical failure in the normal flap control system, the flap emergency switch should be placed at EMERGENCY DOWN. The emergency valve then opens, permitting pressure from the flap emergency accumulator to fully lower the flaps if the airspeed is below about 180 knots IAS. The flap emergency accumulator is sufficient to lower the flaps only once. Pitch correction remains operative when the flaps are lowered by the emergency switch.

SPEED BRAKE SYSTEM.

A hydraulically operated, electrically controlled speed brake is on the lower surface of the fuselage, behind the nose gear well. During normal operation, the speed brake opens in about 2-1/2 seconds and closes in about 1-3/4 seconds. With engine at 60% rpm, the speed brake retraction time should not exceed 7 seconds.

A pressure priority valve, in the speed brake hydraulic pressure line, prevents speed brake operation if utility hydraulic system pressure drops below 1300 psi. (This is the minimum pressure needed for wheel brake and

nose wheel steering systems.) Speed brake operation when utility hydraulic pressure is 1300 to 1850 psi is slow; at pressures above 1850 psi, speed brake operation is normal.

CAUTION

If the speed brake operates when utility hydraulic system pressure is below 1300 psi, the pressure priority valve is faulty and should be replaced before the next flight.

The speed brake system has an emergency control to close the speed brake to trail position in flight, in case of utility hydraulic system failure. There is no speed brake position indicator.

NOTE

Although the speed brake can be used at any speed, a relief valve in the speed brake hydraulic system prevents speed brake extension or allows speed brake to retract, as necessary, under excessive aerodynamic loads, to prevent structural damage. It is possible at extremely high speeds that the speed brake may not open sufficiently to be effective.

SPEED BRAKE GROUND SAFETY LOCKS.

A removable ground safety lock may be installed on each of the two speed brake actuating cylinders to keep the speed brake in the out position when the airplane is on the ground. (See figure 1-34.) The lock assemblies, which have conventional red warning streamers, must be removed before flight.

SPEED BRAKE SWITCH.

A serrated switch on the throttle (figure 1-21) controls speed brake operation. The OFF (center) position is indicated by a white alignment mark on the switch guide. Moving the switch to IN or OUT supplies primary bus power to position the speed brake control valve accordingly so that utility hydraulic system pressure actuates the speed brake. After the speed brake has been opened or closed, the switch must be returned to OFF (center) to preclude loss of fluid, in case of line failure.

On F-100F airplanes, either switch in the out position overrides the other switch. The speed brake switch in the rear cockpit is spring-loaded to OFF (center). When both switches are in OFF (center), the control valve is in neutral, which holds the brake in the desired position. The emergency dump lever overrides both speed brake switches. Failure to return the forward cockpit speed brake switch to OFF (center) after the OUT position has been selected also removes electrical control of the brake from the rear cockpit.

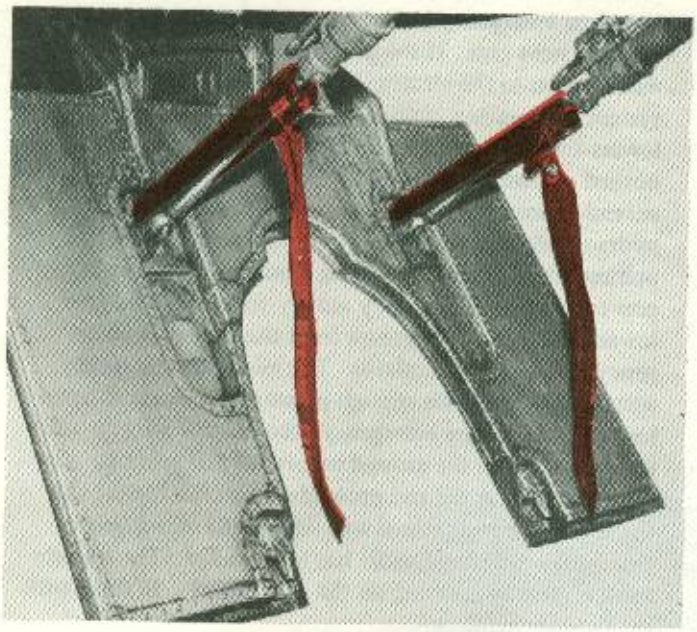
SPEED BRAKE EMERGENCY DUMP LEVER.

The speed brake can be retracted in flight, if normal operation fails, by means of a dump lever. (See 6, figures 1-8 and 1-9; 3, figure 1-14; 6, figure 1-17.) Moving the lever forward mechanically opens a dump valve, which relieves hydraulic pressure from the speed brake actuating cylinders. Air loads then return the speed brake to a trail position. This occurs regardless of speed brake switch position, because the emergency dump lever overrides the speed brake switch. The speed brake system is reset for normal operation when the emergency dump lever is returned to the off position. No emergency method for opening the brake is provided.

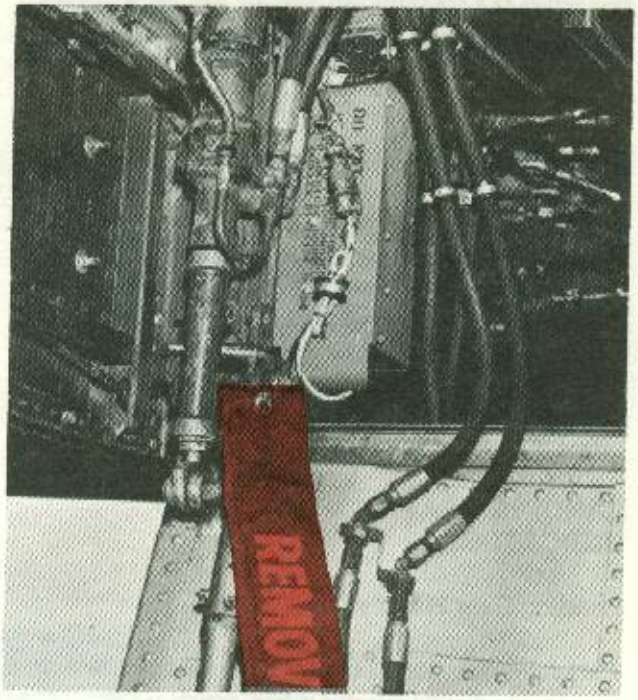
LANDING GEAR SYSTEM.

The retractable tricycle landing gear is electrically controlled and hydraulically actuated. The main gear retracts inboard into the lower surface of the wing and fuselage and the dual-wheel nose gear retracts aft into the fuselage. An electrically actuated retractable tail skid operates simultaneously with the landing gear. The wheel well doors are closed after the gear is down and locked. A load switch on each main gear shock strut prevents the landing gear from being retracted when the weight of the airplane is on the gear. A nose gear load switch prevents engagement of the nose wheel steering system when the weight of the airplane is off the nose gear. On airplanes changed by T.O. 1F-100-1031, the nose gear load switch is also used for identifying inputs to the VGH data recording system. An emergency gear lowering system is provided. During normal or emergency operation, gear lowering time is about 6 to 8 seconds (it should not exceed 10 seconds); normal gear retraction

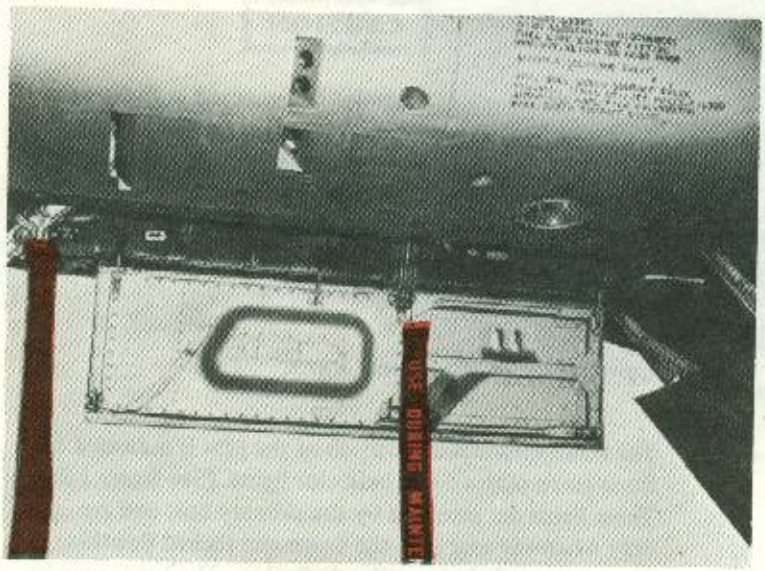
GROUND SAFETY LOCKS



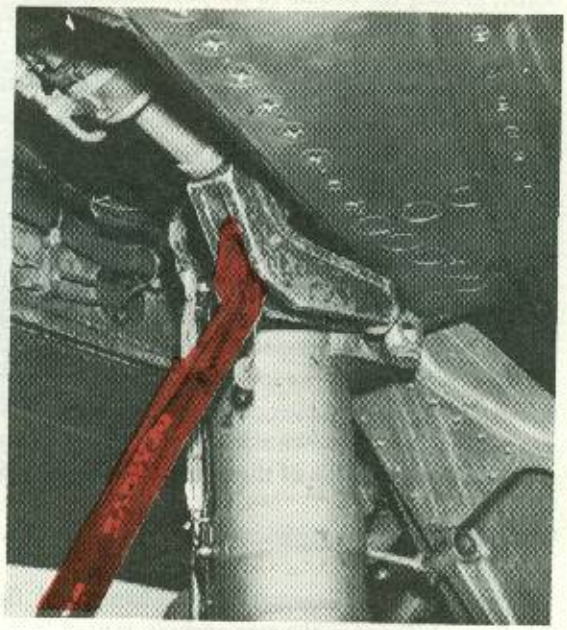
SPEED BRAKE



MAIN GEAR DOOR (TYPICAL)



NOSE GEAR AND NOSE GEAR DOOR



MAIN GEAR (TYPICAL)

WARNING

Remove all ground safety devices before flight.

Figure 1-34

requires about the same time. A steering unit on the nose gear serves as a conventional shimmy damper when the steering is not engaged. The main gear wheels have hydraulically operated, multiple-disk type brakes with an antiskid system.

LANDING GEAR GROUND SAFETY LOCKS.

Removable ground safety locks may be installed in the main and nose gear assemblies to prevent possible collapsing of the gear while the airplane is on the ground. (See figure 1-34.) Ground safety locks are also provided for the open position of the main and nose gear wheel well doors. The locks have regulation red warning streamers. All gear ground safety locks must be removed before flight.

LANDING GEAR CONTROLS AND INDICATORS.

Landing Gear Handle.

The landing gear handle (figure 1-35) controls the gear, the gear door hydraulic selector valves, which are primary bus powered, and the tail skid actuator, by tertiary bus power. On F-100F airplanes, the handles in each cockpit are mechanically interconnected. When the airplane is airborne, moving the handle to UP positions the door selector valve so that utility hydraulic pressure opens the wheel well doors. After the doors are open, the gear selector valve applies pressure to retract the gear. When the gear is up and locked, the door selector valve is repositioned, to close the doors. After the doors are closed and locked, the landing gear system is automatically depressurized. When the landing gear system handle is moved to DOWN, both the door and gear selector valves are energized to permit hydraulic pressure to open the wheel well doors and lower the gear. After the gear is down and locked, the door selector valve is repositioned to close and lock the doors. At the completion of the lowering cycle, the gear and door selector valves remain energized, maintaining hydraulic pressure against the down side of the gear actuating cylinder and the closed side of the door actuating cylinders.

Landing Gear Emergency Lowering Handle.

The landing gear emergency lowering handle (23, figure 1-6; 24, figure 1-7) not in the rear cockpit, is used to lower the gear in case of failure of the normal lowering system. Pulling the handle full out (after the landing gear handle has been moved to DOWN) mechanically unlocks all gear and wheel well door uplocks, and positions the hydraulic selector valves to open the doors and lower the gear. To provide positive nose gear lowering, pulling the emergency lowering handle also directs hydraulic pressure from the nose gear emergency lowering accumulator to lower and lock the nose gear. If the gear fails to lower because of a malfunction in the electrical system, and the hydraulic system is operating, pulling the emergency lowering handle lowers and locks the gear by hydraulic pressure as in a normal extension cycle. (The handle must be held fully extended until the gear is down and locked.) If utility hydraulic system failure prevents normal gear lowering, pulling the emergency lowering handle allows the main gear to fall free and lock by gravity while the nose gear lowers and locks by pressure from the nose gear emergency lowering accumulator. The nose gear emergency accumulator provides enough pressure for one emergency lowering only. The emergency accumulator selector valve must be reset on the ground before normal gear operations can continue. A red rod protrudes from the fuselage at the left of the nose wheel well when the landing gear emergency lowering handle has been pulled. The selector valve is reset by pushing on the rod until it is in flush with the fuselage skin.

CAUTION

The nose gear cannot be retracted in flight after being lowered by the emergency lowering handle. The emergency accumulator selector valve must be reset manually (on the ground) before the next flight, if the gear has been lowered by the emergency handle.

Landing Gear Position Lights.

The down-and-locked position of the gear is indicated by three green push-to-test indicator lights. (See figure 1-35.) These lights are powered by the primary bus, and are on only when the gear is in the down-and-locked position. There is no tail skid position indicator. The lights may be dimmed by the indicator light dimmer switch.

Landing Gear Warning Light.

A primary-bus-powered red light in the plastic knob of the landing gear handle (figure 1-35) warns of an unsafe condition of the landing gear or wheel well door. When the landing gear handle is moved to UP, the light comes on immediately and remains on until the gear is up and locked and the wheel well doors are closed and locked. When the landing gear handle is moved to DOWN, the light remains on until the gear is down and locked. The light also comes on whenever the audio signal is actuated, or whenever the landing gear and landing gear handle positions do not coincide. On some airplanes, the landing gear warning light comes on whenever the throttle is retarded below 85%, regardless of other conditions, if the landing gear is not down and locked. This light may be dimmed by the indicator light dimmer switch when the instrument panel indirect lights are turned on. The bulb of the landing gear warning light can be tested by the indicator light test circuit.

Landing Gear Audio Warning Signal.

An audio warning in the headset provides an indication of unsafe landing gear condition when all of the following conditions are met: landing gear not down and locked; altitude below 10,000 feet; airspeed below 205 (±5) knots IAS; and throttle not in AFTERBURNER (some airplanes*), or throttle retarded below 85% rpm. The warning signal is powered by the primary bus. On some airplanes*, the gear audible warning signal system also operates with the ram-air turbine door open, if the landing gear is up and the airplane is below 10,000 feet, regardless of airspeed.

If a Military Thrust takeoff is made, the warning signal sounds as the first gear leaves its down-locked position and continues to sound until an airspeed of 220 knots IAS is reached.

Warning System Cutout Button.

A warning system cutout button (figure 1-35) is used to disconnect the audible warning signal at any time and under any condition. An increase in airspeed above 220 knots IAS or an increase in altitude above 10,000 feet automatically resets the signal circuit.

NOSE WHEEL STEERING SYSTEM.

The nose wheel steering system affords directional control of the airplane during taxiing, takeoff, and landing. It is engaged by primary bus power and actuated by utility hydraulic pressure. Steering is mechanically controlled by the rudder pedals and is engaged or disengaged by a pushbutton on the stick grip. The irreversible hydraulic steering unit permits the nose wheels to be turned 35 degrees each side of center. A nose gear load switch prevents engagement of the steering system when the weight of the airplane is off the nose gear. As the gear retracts, a mechanically operated valve depressurizes the steering unit so that it cannot remain engaged upon gear retraction if the nose gear load switch or hydraulic shutoff valve fails. When the steering system is not engaged, the steering unit serves as a conventional shimmy damper.

NOSE GEAR TORQUE LINK.

The nose gear shock strut and nose wheel assembly are connected by a torque link (scissors), which keeps the strut and wheel assemblies aligned. (See figure 1-36.) The torque link has a quick-release pivot pin, to disconnect the link for towing the airplane. When the link is disconnected, the nose wheels can free-swivel. The torque link is disconnected by pulling out on the spring-loaded safety pin and then unscrewing the pivot pin. The pivot pin should be replaced in the upper section of the torque link when the link is disconnected. The torque link is connected by pulling out on the safety pin and inserting the pivot pin through both sections of the link. The pivot pin is then tightened finger-tight, to the nearest safety pin hole and safetied by releasing the spring-loaded safety pin into the hole.

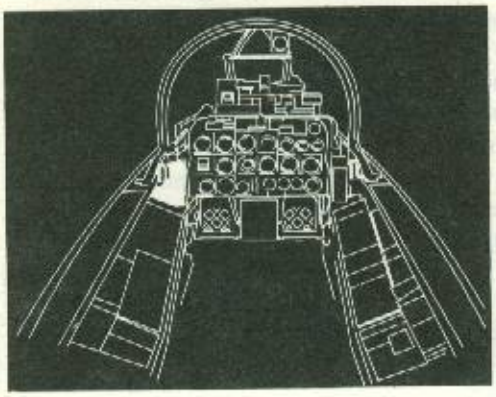
NOSE WHEEL STEERING BUTTON.

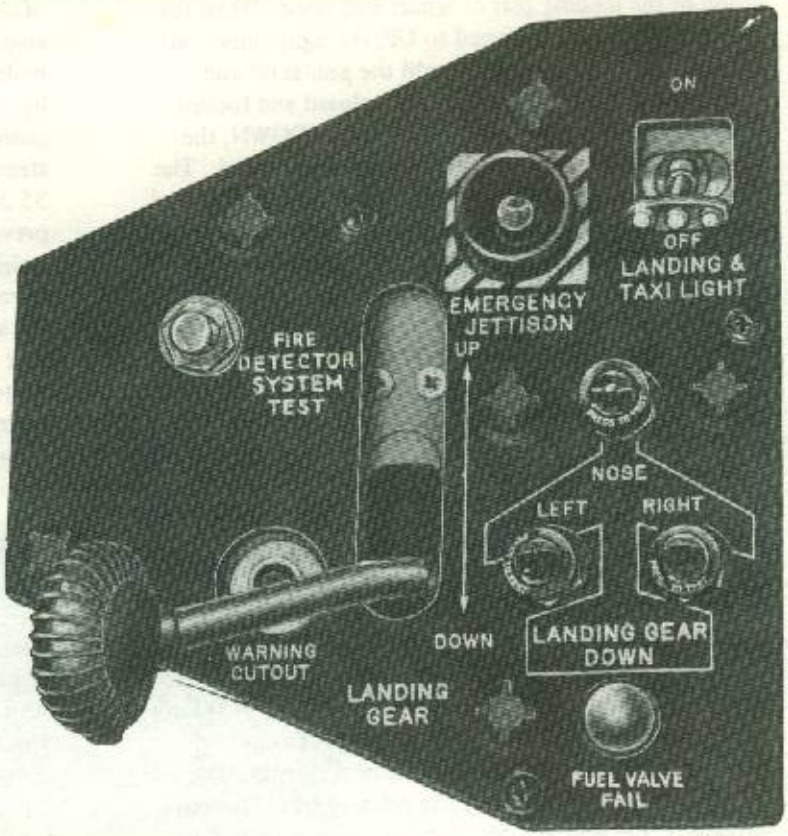
The nose wheel steering system is engaged or disengaged by a pushbutton switch. (See figure 1-32.) Momentarily pressing the button energizes the steering shutoff valve

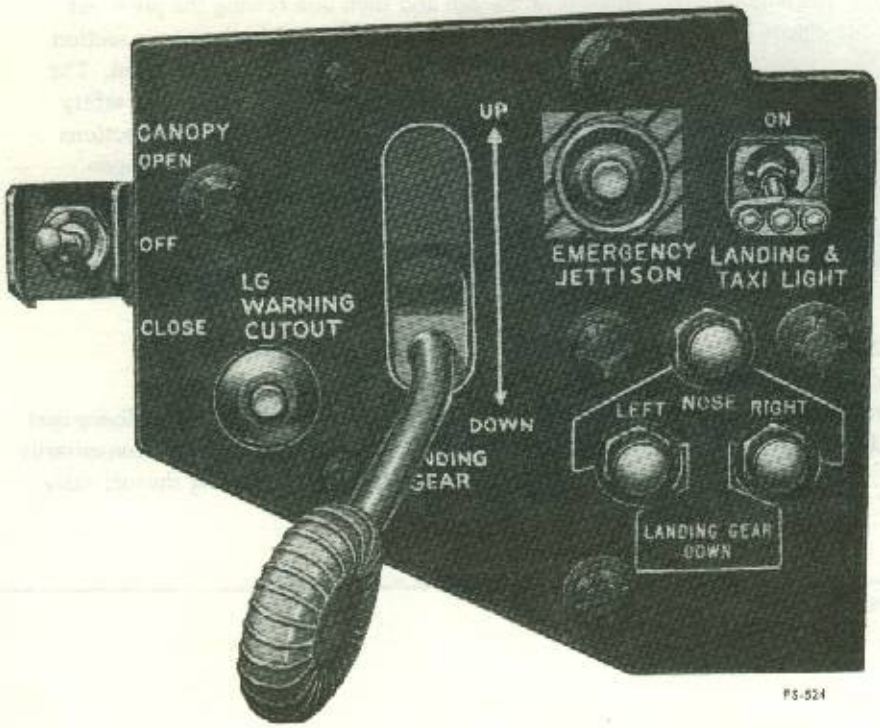
*F-100D-61 and later, and F-100F Airplanes

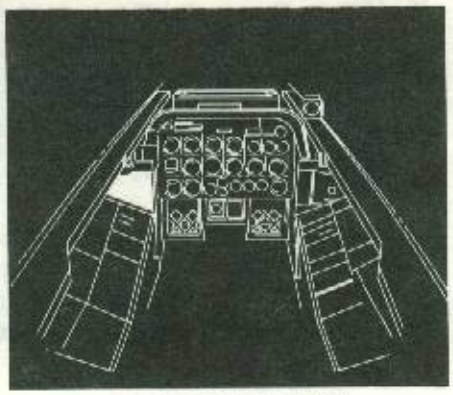
LANDING GEAR CONTROL PANEL F-100D AND F-100F-2 THROUGH-16

(TYPICAL)









F-100F REAR COCKPIT

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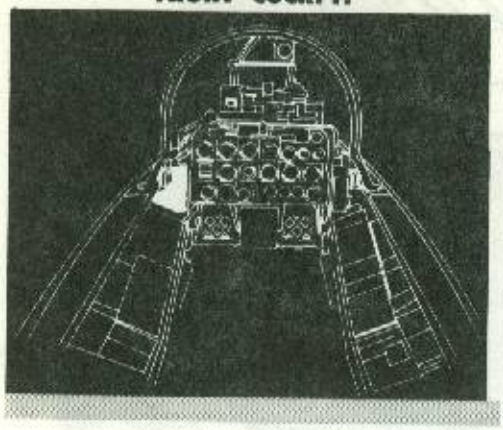
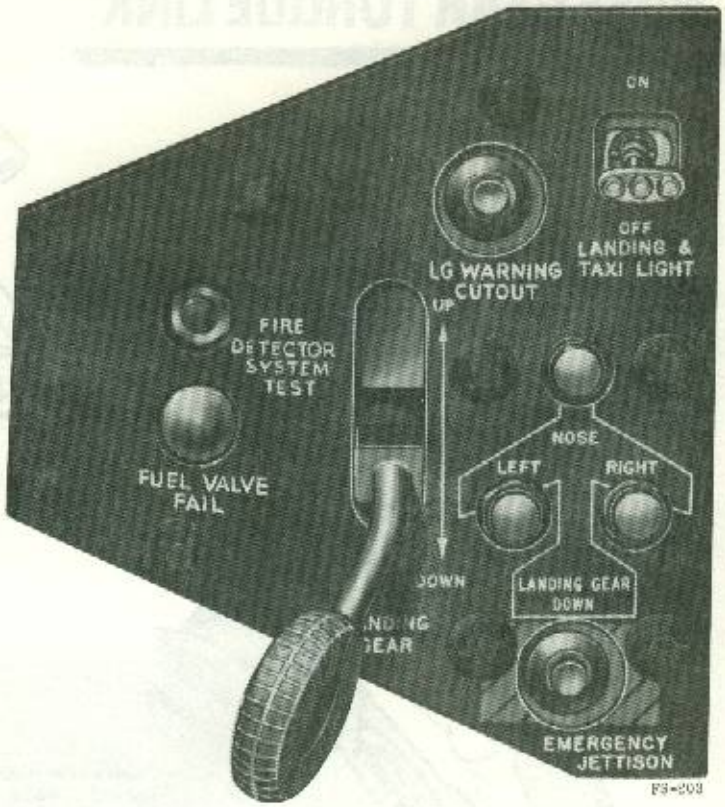
Figure 1-35 (Sheet 1 of 2)

F-100F-20 AIRPLANES

FRONT COCKPIT





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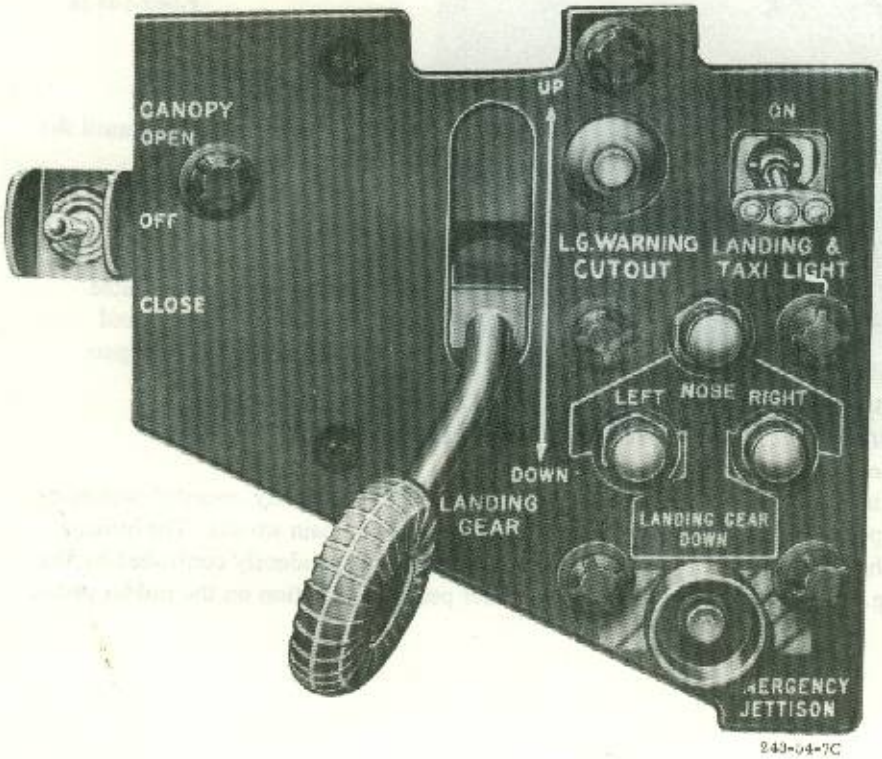

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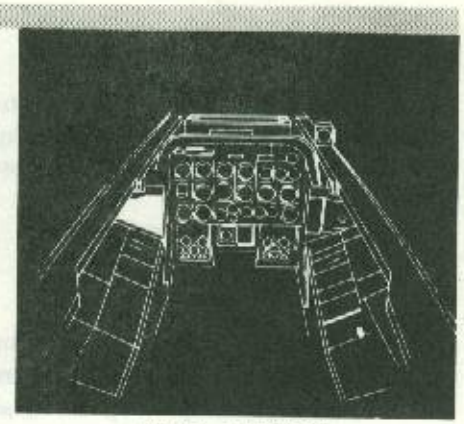

REAR COCKPIT

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Figure 1-35 (Sheet 2 of 2)

NOSE GEAR TORQUE LINK

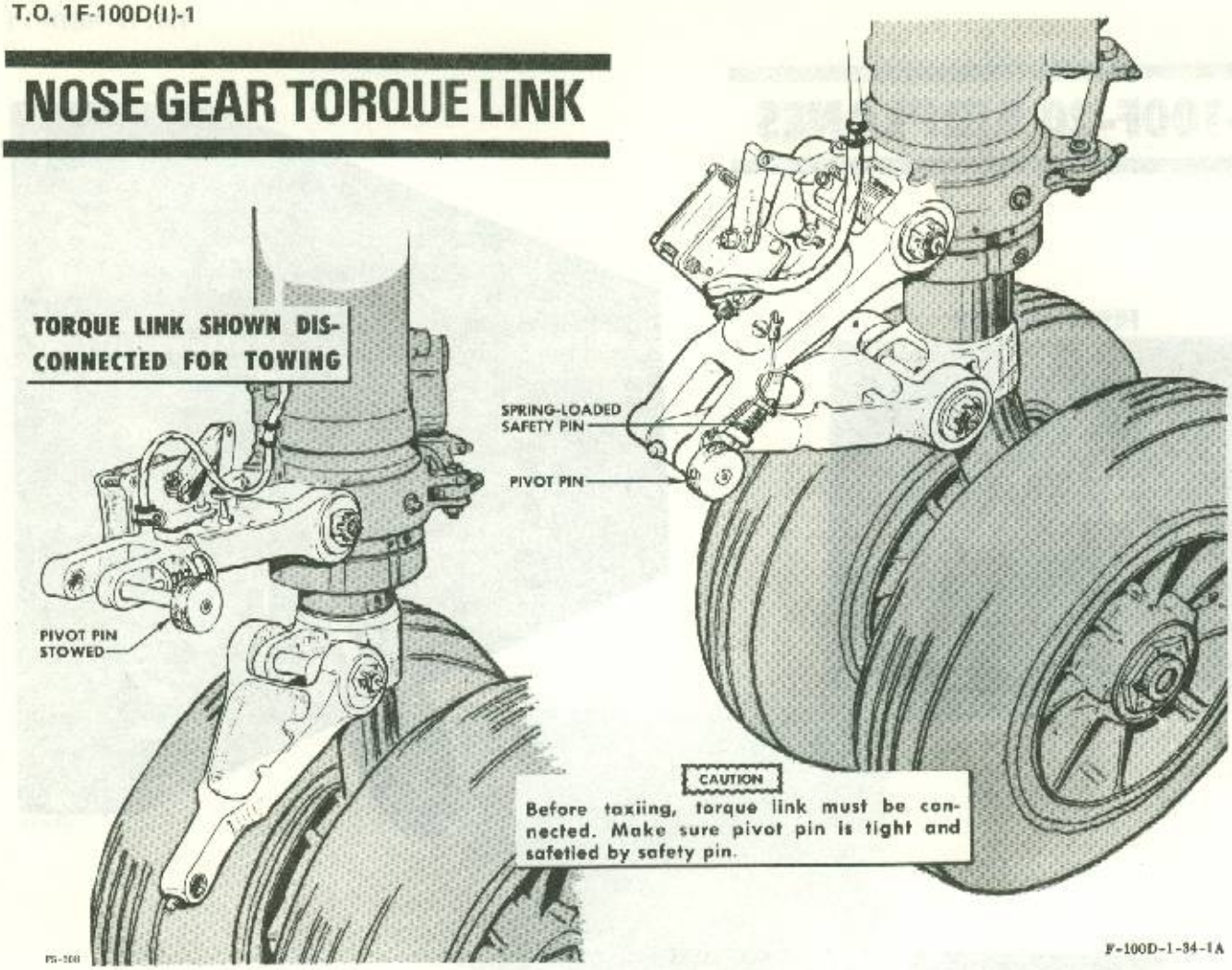


Figure 1-36

which directs utility hydraulic pressure to the nose wheel steering unit. A clutch is then engaged hydraulically to link the rudder pedal cables with the steering unit.

CAUTION

If rudder pedals are not at neutral when button is pressed, the steering may or may not engage (depending on engagement of clutch in steering unit) and move the nose wheels to agree with the pedal position. If the steering does not engage, the pedals must be moved in the direction of the nose wheel setting to obtain steering.

After the system is engaged, it remains engaged until the button is depressed and released a second time.

NOTE

The nose wheel steering button is operable only if primary bus power is available and the weight of the airplane is on the nose gear.

WHEEL BRAKE SYSTEM.

The multiple-disk type hydraulically operated brakes are on the outboard side of the main wheels. The braking action on each wheel is independently controlled by the respective rudder pedal. Toe action on the rudder pedals

actuates the brake valves, which meter utility hydraulic pressure to apply the brakes. The hydraulic pressure applied to the brakes is proportional to the force applied at the pedal, up to the maximum pressure admitted through the brake valve, and an increase in pedal pressure does not increase the pressure to the brakes. The wheel brake emergency accumulator furnishes power for the brake valves in case of a pressure drop in the brake system or utility hydraulic system failure. The wheel brake system includes an antiskid system which automatically prevents excessive wheel skidding. If the pressure to the brake valves drops below 500 psi, a variable-flow regulator ("run-around valve") opens to permit additional pressure (flow) to bypass the flow restrictor and maintain pressure to the brake valves.

An electrically driven emergency hydraulic pump, powered from the battery bus, provides brake operating pressure (for power braking) in case of utility system failure, as long as fluid is available from the utility reservoir. Emergency pump operation is controlled by pressure switches in the brake system which sense low brake operating pressures. If, because of antiskid cycling, repeated application of brakes or utility system failure, the wheel brake emergency accumulator pressure drops to 750 psi and the antiskid system accumulator pressure drops below 450 psi, the pressure switches close. With the pressure switches closed and with either brake pedal depressed (so that either or both pedal switches are actuated), the emergency pump operates to supply brake pressure. The pump shuts off when pressure is again built up to 750 psi in the antiskid system accumulator. A time-delay relay in the emergency brake system enables the emergency pump to operate if the modulating restrictor and/or variable flow regulator becomes clogged because of contamination. (Refer to section VII for wheel brake operation.)

NOTE

> An operational check of the emergency pump can be made before the engine is started, by operating the brake pedals and listening for pump operation.

WHEEL BRAKE ANTISKID SYSTEM.

An electrically controlled, hydraulically operated antiskid system in the wheel brake system detects the start of a skid condition or a near-locked condition and automatically releases hydraulic pressure at the brakes. The system is not an automatic braking system, and, therefore, maximum brake pedal pressure should not be applied at touchdown and maintained throughout the landing roll during a normal landing. The system functions automatically, when engaged, to release the brakes as the wheels approach a skid. A skid detector unit, on each main gear, senses the rate of change of wheel speed as well as rotation of the wheel. These detectors supply electrical signals to the system control unit, which controls the antiskid control valves in the brake lines. The control valves regulate the hydraulic pressure applied by the pilot to the brakes. (See figure 1-30.) If either wheel approaches a skid condition after the brakes are applied, the detector sends a "skid" signal to the control unit. The control unit actuates the antiskid control valve to release the brakes by shutting off the applied pressure and dumping the fluid for both brakes to return. As the skid condition is corrected by this automatic brake release, the detector stops sending a "skid" signal. This causes the control valves to open and restore pressure to the brake. The antiskid system maintains this "on-off" cycling of the brakes as long as the brakes are applied and skid conditions prevail.

The antiskid system is effective from the highest possible landing speed to a lower limit of about 10 knots. The system should not be depended upon at taxi speeds below 10 knots, because changes in the rate of wheel rotation are not sufficient to provide reliable antiskid control. Touchdown with the wheels locked is prevented by an "arming" circuit that energizes the antiskid control valves when the landing gear handle is UP. The brake system is kept inoperative until the wheels touch down and start to rotate.

Lock-wheel protection is provided for 4 to 16 seconds after initial touchdown so that the brakes will not lock up during a bounce. At touchdown, as the wheels begin to rotate, the skid detector units sense the acceleration and simultaneous wheel rotation and permit brake application within 0.6 second maximum. If either brake is released by the system for a continuous period of 3 seconds, the related control unit turns on an antiskid caution light. If this occurs, the antiskid switch should be turned OFF and normal braking techniques should be used, since antiskid protection cannot be ensured.

Antiskid Switch.

Primary bus power to the antiskid system is controlled by a three-position switch (not in rear cockpit). (See

figure 1-22.) When the switch is ON, antiskid protection is available. In addition, with the switch ON, moving the landing gear handle to UP after takeoff engages the arming circuit that prevents landing with the brakes on. This gives locked-wheel protection for about 4 to 16 seconds after initial touchdown. Moving the switch to OFF shuts off the antiskid system and normal braking is available immediately. However, the antiskid caution light may not come on until 1-1/2 to 4 seconds later. (Power braking is still available when the switch if OFF.) The ARM (GRD TEST) position is used to "arm" the system for maintenance and adjustment purposes only. The switch is spring-loaded from ARM (GRD TEST) to ON.

Antiskid Caution Light.

When the skid protection has been lost on either wheel, the antiskid caution light (figure 1-19) is illuminated by primary bus power. The bulbs in the light can be tested by the indicator light test circuit.

DRAG CHUTE SYSTEM.

The 16-foot, ring-slot type parachute, packed in a deployment bag, is stowed in a compartment in the lower surface of the aft fuselage, outboard of the tail skid. A riser cable (stowed externally in a faired recess on the left side of the fuselage) joins the drag chute to a coupling in the trailing edge of the vertical stabilizer, below the rudder. The drag chute is mechanically controlled and can be jettisoned, if desired, after being deployed. The drag chute release mechanism has a safe arming device that automatically releases the chute from the airplane if it is deployed by any action other than movement of the drag chute handle.

DRAG CHUTE HANDLE.

A drag chute handle (51, figure 1-6; 3, figure 1-7; 1, figure 1-12; 35, figure 1-13) is used to control the drag chute. (On F-100F airplanes, the cockpit initiating the action has primary control of the drag chute handle.) When the handle is pulled straight back about 3 inches to deploy the chute, the spring-loaded drag chute compartment doors are mechanically unlocked. A pilot chute springs from the compartment when the doors open, and pulls the drag chute out of the deployment bag into the air stream, where it is inflated. To jettison the drag chute at any time after it has been deployed, the handle should be rotated 90 degrees counterclockwise and then pulled back another 2 inches from the deploy position. A release mechanism is mechanically unlatched to release the chute and riser cable.

ARRESTING HOOK.

The externally mounted arresting hook, forward of the tail skid, is used to stop the airplane during takeoff or landing emergencies. It is held against the fuselage in a cocked (stowed) position by a solenoid latch that is ground-safetied by a red streamered safety pin. A hook point guard is installed on the rear fuselage to prevent a rebounding barrier cable from engaging the stowed arresting hook. When released, the flat spring steel hook is held against the runway by spring tension.

WARNING

Stay clear of the hook travel arc, as it is always cocked when stowed and inadvertent release could cause serious injuries.

ARRESTING HOOK RELEASE BUTTON.

The arresting hook release button (21, figures 1-8 and 1-9; 21, figure 1-14; 15, figure 1-17) is powered by the battery bus. The release button has a light in the cap that comes on when the arresting hook is released. Pressing this button actuates the solenoid latch that releases the hook. Once the hook is released, it must be manually cocked by ground personnel. There is a small post light adjacent to the arrestor hook release button guard. This light is controlled by the instrument panel light rheostat for night operations.

INSTRUMENTS.

Most of the instruments are powered by the airplane electrical systems. The exhaust temperature gage and tachometer systems, however, are of the self-generating type. An instrument panel vibrator, energized automatically by the tertiary bus, prevents instrument lag or sticky pointer indications.

NOTE

For information regarding instruments that are an integral part of a particular system, refer to applicable paragraphs in this section and in section IV.

The following paragraphs contain information pertaining to basic instrument systems and power requirements. For information on instrument displays, refer to Instrument Flying Manual, AFM 51-37.

PITOT-STATIC BOOM.

Pitot and static pressures for various flight instruments are obtained from the pitot-static boom. (See figures 1-1 and 1-2.) Because the length of the boom makes it vulnerable during towing or other ground operations, the boom is hinged forward of its attachment point to allow it to be folded upward. Anti-icing protection is provided for the boom. (Refer to Air Conditioning, Pressurization, Defrosting, Anti-icing, and Rain Removal Systems in section IV.)

ACCELEROMETER.

The three-pointer accelerometer (13, figure 1-6; 18, figure 1-7; 11, figure 1-12, figure 1-13) shows positive and negative G-loads and has two recording pointers (one for positive G-loads and one for negative G-loads) which follow the indicating pointer to its maximum attained travel. (Refer to VGH Data Recording System — Airplanes Changed by T.O. 1F-100-1031 in section IV.)

AAU-19/A ALTIMETER.

The AIMS system incorporates a servoed altimeter that functions in conjunction with the AIMS system altitude automatic reporting feature. (See figure 1-37.) The servoed altimeter, which replaces the standard altimeter, also functions in the standby mode as a conventional altimeter. In the servoed mode, the altimeter presents aircraft altitude corrected for static system installation error. When the altimeter is functioning in the servoed mode, use of the standard altimeter correction card (on the canopy frame) is not necessary. The altitude indica-

AIMS SYSTEM SERVOED ALTIMETER

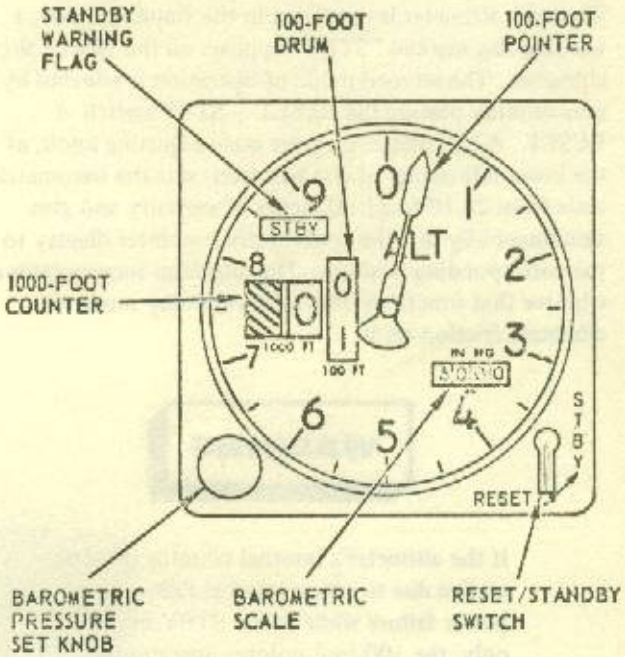


Figure 1-37

tion is presented on a counter, drum, and pointer display covering a range of -1000 to +80,000 feet. The counter-drum portion of the display provides a 3-digit indication. The first two digits (counter display) indicate altitude X 1000 feet. The third digit (drum display) indicates altitude X 100 feet. The pointer also indicates altitude X 100 feet with a full revolution equivalent to 1000 feet. The pointer display enables more precise reading because the scale is graduated in increments of 50 feet.

Altitude data is processed by the AIMS system altitude computer and is encoded for transmission in response to mode C interrogations with altitude reported in increments of 100 feet. When functioning in the servoed mode, output

data from the altitude computer is also fed to the altimeter. The switch-over to the standby mode can be selected by a RESET – STBY switch located at the lower right corner of the altimeter or the switch-over will occur automatically in case of power failure or altitude computer malfunction. When the altimeter is operating in the standby mode, a warning flag marked "STBY" appears on the face of the altimeter. The servoed mode of operation is selected by momentarily placing the RESET – STBY switch at RESET. A barometric pressure scale adjusting knob, at the lower left corner of the altimeter, sets the barometric scale from 28.10 to 31.00 inches of mercury and also simultaneously sets the counter-drum-pointer display to the corresponding position. The altimeter incorporates a vibrator that functions only in the standby mode to eliminate friction error.

WARNING

If the altimeter's internal vibrator is inoperative due to either internal failure or dc power failure while in the STBY mode only, the 100-foot pointer may momentarily hang up when passing through "0" (12 o'clock position). If the vibrator has failed, the 100-foot pointer hangup can be minimized by tapping the case of the altimeter. Be especially watchful for this failure when minimum approach altitude lies within the 800- to 1000-foot part of the scale (1800 to 2000 feet, 2800 to 3000 feet, etc.), and use any appropriate altitude backup information available for altitude cross-check; e.g., cockpit pressure altitude indicator.

NOTE

During rapid accelerations or decelerations above 300 KIAS, the vertical velocity indicator will give erroneous indications. The AAU-19/A altimeter in the RESET mode should be used for correct indication of ascent or descent.

Altimeter Correction Card.

An altimeter correction card is on the canopy frame. (Refer to Altimeter Correction in T.O. 1F-100C(I)-1-1.)

NOTE

Zero altitude position correction should be used for the AAU-19/A altimeter in the RESET mode.

Airspeed/Mach Indicator.

The airspeed/Mach indicator (46, figures 1-6 and 1-7; 3, figure 1-12; 34, figure 1-13) shows indicated airspeed within a range of 80 to 850 knots.

NOTE

The maximum allowable airspeed/Mach pointer indicates a speed greater than the clean airplane limit airspeed; therefore, it should be ignored.

MAGNETIC COMPASS.

A conventional magnetic compass, suspended from the windshield bow, provides standby heading indications for navigation in case of instrument or electrical system failure. A compass correction card is on the right canopy rail.

ATTITUDE INDICATOR.

An ARU-13A type attitude indicator (9, figures 1-6, 1-7, and 1-12; 5, figure 1-13) as a pictorial-type instrument that displays information from an electrically operated, remote MD-1 gyro control assembly. Power is supplied from the 3-phase ac instrument bus. A window in the lower left corner of the instrument shows "OFF" only with loss of power and not for internal failure. The system starts operating when the engine master switch is ON. The "OFF" will not be covered until after a warmup period

of approximately 1-1/2 to 2-1/2 minutes. "OFF" also will appear in case of failure of 3-phase ac instrument bus power.

WARNING

After airplane maneuvers exceeding 82 degrees climb or dive, attitude indications may not be reliable until the gyro has been re-erected.

- There is no indication for internal failure of the attitude indicator system.

STANDBY ATTITUDE INDICATOR.

The standby attitude indicator, a 2-inch vertical gyro indicator (VGI), is a pictorial-type instrument (44, figure 1-6; 43, figure 1-7) that displays information from an electrically operated remote MD-1 gyro control assembly. Power is supplied from the 3-phase ac instrument bus. Operation of the standby attitude indicator and ARU-13A type attitude indicator is identical.

ATTITUDE INDICATOR FAST ERECTION BUTTON.

The square attitude indicator fast erection pushbutton (11, figures 1-6 and 1-7; 8, figure 1-12; 25, figure 1-13) is marked "PUSH VGI ERECT." The button is powered by the 3-phase ac instrument bus and permits fast erection of the attitude gyros only while the button is depressed. A light in the button comes on when the instrument panel indirect lights are on.

NOTE

Do not use the fast erection button in flight until the airplane is in a wings-level, unaccelerated condition. The "OFF" flag may or may not appear during fast erection. If the "OFF" flag remains present after the erection button is released, then it must be assumed the gyro has failed.

- Do not hold the fast erection button in longer than 1-1/2 minutes without a 2-minute cooling period between applications.

TURN-AND-SLIP INDICATOR.

A conventional (4-minute) turn-and-slip indicator (10, figure 1-6; 16, figure 1-7; 36, figure 1-12; 7, figure 1-13) is electrically driven by the primary bus. Because of the mounting tilt of the turn-slip indicator, the indicator will show a turn in the opposite direction to that of actual aircraft attitude during a high rate of roll. Refer to Instrument Flight Procedures in section IX.

INDICATOR LIGHT TEST CIRCUIT.

The indicator light test circuit provides a means of testing the operation of the bulbs in the indicator, caution, and warning lights simultaneously. All indicator, caution, and warning light bulbs (except the fuel system shutoff valve fail light, plus those in the engine compartment fire- and overheat-warning lights) are included in this test circuit which is powered by the primary bus. In addition, the status display lights may be tested provided the main ac bus is energized.

NOTE

Use of the indicator light test circuit is not an operational check-out of any indicator, caution, status display, or warning system.

INDICATOR LIGHT TEST SWITCH.

A three-position switch (figures 1-29 and 4-15) permits testing the illumination and brilliancy of the indicator, caution, status display, and warning lights. When the switch is held at TEST BRIGHT or TEST DIM, primary bus power illuminates the lights at the selected brilliancy. The switch is spring-loaded to OFF (center).

NOTE

Releasing the indicator light test switch from either position changes the indicator light circuit to bright.

INDICATOR LIGHT DIMMER SWITCH.

A three-position switch controls the brilliancy of the indicator, caution, status display, and warning lights when the instrument panel lights are on. Moving the switch from its spring-loaded center off position to BRIGHT or DIM sets

up circuitry so that the indicator, caution, and warning lights will come on at the selected brilliancy when illuminated. (See figures 1-29 and 4-15.) All indicator, caution, and warning lights can be dimmed by means of the switch except those on the in-flight control tester panel, which have an integral control.

NOTE

Use of the indicator light test switch changes the indicator light circuit to bright when released. To dim the indicator lights after using the indicator light test switch, move the indicator light dimmer switch to BRIGHT, then to DIM, and release to the center off position.

MASTER CAUTION LIGHT.

The master caution light (figure 1-19), powered by the primary bus, comes on whenever any of the placard type caution lights comes on. This alerts the pilot to check the placard lights to determine in which system the trouble is occurring. To enable the master caution light to show the illumination of any additional lights, it must be put out each time it comes on. This is done by pressing the lighted indicator or caution light into its socket. The master caution light then goes out and it is readied to relight should another indicator or caution light come on. The lighted indicator or caution light does not go out when it is pressed. On F-100F airplanes, duplicate indications are received in the front and rear cockpits; however, resetting the master caution light can only be done in the front cockpit. The bulbs in the master caution light can be tested by the indicator light test switch.

EMERGENCY EQUIPMENT.

ENGINE COMPARTMENT FIRE AND OVERHEAT WARNING SYSTEMS.

Fire and overheat detector systems detect and indicate overheating and fire in the forward or aft engine compartment. One of four different configurations, which can be identified by the system test buttons, may be installed. Most rewired airplanes and airplanes changed by T.O. 1F-100-965 have a single-wire system connected to single-terminal detectors with a single pushbutton switch for testing circuits. Airplanes with a dual-wire system connected to dual-terminal detectors have four pushbutton switches for testing the circuits. Airplanes with a dual-wire system connected to single terminal detectors have two pushbutton switches for testing the circuits. Some airplanes have a single-wire system connected to single-terminal detectors. These airplanes have two combination pushbutton lights.

Engine Compartment Fire- and Overheat-warning Lights.

An abnormal rise in temperature in either engine compartment is shown by primary-bus-powered placard-type warning lights. (See figure 1-19.) Two lights, one for the forward and one for the aft compartment, come on to show an excessive temperature condition or fire in the respective engine compartment. The forward compartment light shows "FIRE ENG. COMP." when it comes on; the aft compartment light, "OVERHEAT ENG BURNER."

Engine Compartment Fire- and Overheat-warning System Test Buttons.

Some airplanes have a single primary-bus-powered test button. (See figure 1-35.) This button permits testing the continuity of the fire- and overheat-warning systems, and provides an operational test of the bulbs within the fire- and overheat-warning lights. Other airplanes have four primary-bus-powered test buttons. Pressing the "MAIN" or "AUX" test button for either the fire- or overheat-warning

*F-100F Airplanes

system tests the main or auxiliary circuit of the respective system. The fire- or overheat-warning light should come on as each circuit and system test button is pressed.

NOTE

Press the "MAIN" and "AUX" buttons separately in order to get an individual check of each circuit.

Failure of each bulb in either warning light to come on bright during these tests indicates a malfunction of the respective system, and the system should be checked before flight. On airplanes with the partial dual-wire system, both "AUX" pushbuttons are inoperative and are covered with a cap marked "INOP." The operation of the fire- and overheat-warning lights can only be tested by the engine compartment fire- and overheat-warning system test buttons. Some airplanes use combination pushbutton lights. (See 7, figure 1-7.) Pressing the FIRE ENG COMP or OHEAT ENG BURNER light will test the respective system.

CANOPY.

The one-piece clamshell-type canopy (figure 1-38) has an electromechanical mechanism for normal operation and a cartridge-type canopy remover for emergency force-jettisoning. The canopy is hinged at the rear; in opening, it moves directly back about one inch and then rises to about 30 (23*) degrees. Normal operation of the canopy is controlled by a switch in the cockpit and external switches on both sides of the fuselage. Emergency force-jettisoning of the canopy is accomplished when either of the ejection seat handgrips is raised, or when the canopy alternate emergency jettison handle is pulled. F-100D-61 and later airplanes and F-100F airplanes also have an external handle that permits the canopy to be jettisoned by ground personnel for emergency entrance. If the canopy has been opened more than about 4 (8*) inches at the windshield bow, firing the canopy remover will not force-jettison the canopy. (On F-100F airplanes, the canopy remover tube mechanically locks to the canopy structure whenever the canopy remover is fired, to prevent the possibility of air loads causing the remover tube to strike the rear seat occupant.) Handles in the canopy frame permit mechanical release of the canopy in emergencies, both from within the cockpit and externally. The mechanical release is independent of the ejector charge or the normal electrical actuator.

CANOPY SEAL.

An inflatable rubber seal (figure 1-38) in the edge of the canopy frame and bow, seats against the mating surfaces of the fuselage and windshield bow to provide sealing for cockpit pressurization. The seal pressurization switch is actuated just before the complete locking of the canopy. The switch controls primary bus power to a valve that directs engine compressor air to inflate the seal. (This air is passed through the primary heat exchanger of the air conditioning system.) Operation of the canopy mechanism, during the opening cycle, dumps the seal pressure during the initial aft movement of the canopy.

NOTE

If primary bus failure occurs, the canopy seal deflates.

WINDSCREEN (REAR COCKPIT) – F-100F AIRPLANES.

A windscreen (figure 1-38) above the rear cockpit instrument panel shroud, protects the rear seat occupant from wind blast when the canopy is jettisoned in flight. A pip pin and hook that retain the windscreen in the stowed (lowered) position are released either by canopy remover firing or by a manual release after the canopy is manually jettisoned. The windscreen is then forced upward and into position by spring action. If the canopy bubble is broken or fails, pulling the windscreen manual emergency release knob will raise the windscreen and afford the rear seat occupant some protection. A windscreen ground safety pin is inserted through a bracket on the left side of the windscreen just ahead of the hinge line to prevent inadvertent release.

Windscreen Manual Emergency Release Knob.

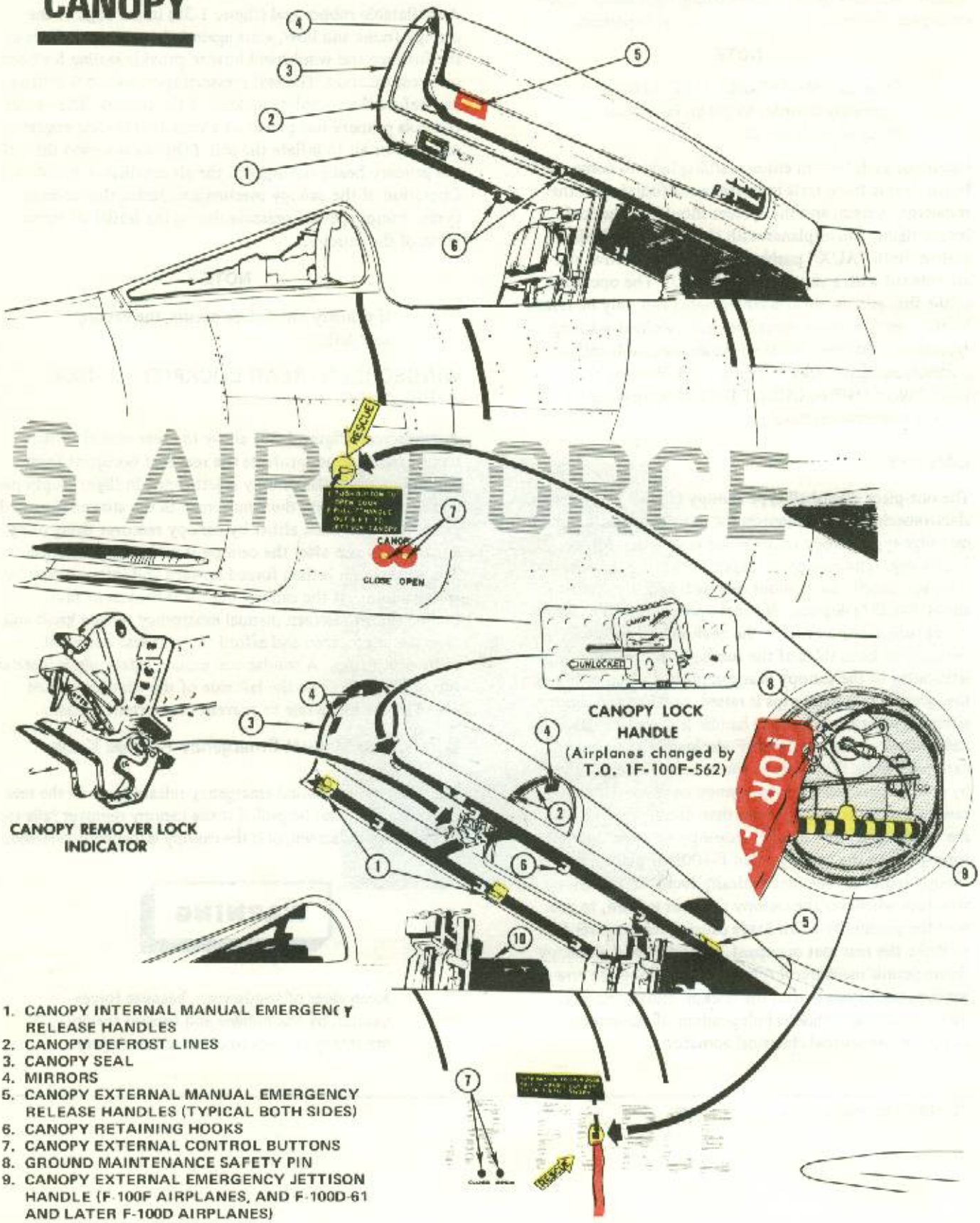
The windscreen manual emergency release knob in the rear cockpit only, must be pulled if the canopy remover fails to release the windscreen, or if the canopy is manually released.

WARNING

Keep clear of windscreen, because forces exerted by the bungee and torsion spring are strong enough to cause serious injury.

*F-100F Airplanes

CANOPY



1. CANOPY INTERNAL MANUAL EMERGENCY RELEASE HANDLES
2. CANOPY DEFROST LINES
3. CANOPY SEAL
4. MIRRORS
5. CANOPY EXTERNAL MANUAL EMERGENCY RELEASE HANDLES (TYPICAL BOTH SIDES)
6. CANOPY RETAINING HOOKS
7. CANOPY EXTERNAL CONTROL BUTTONS
8. GROUND MAINTENANCE SAFETY PIN
9. CANOPY EXTERNAL EMERGENCY JETTISON HANDLE (F-100F AIRPLANES, AND F-100D-61 AND LATER F-100D AIRPLANES)
10. WINDSCREEN

Figure 1-38

CANOPY CONTROLS AND INDICATOR.

NOTE

The canopy actuator is normally powered by the tertiary bus. If this power is not available, the battery bus automatically becomes the power source for the canopy actuator.

Canopy External Control Buttons.

Two pushbuttons (figure 1-38), on each side of the airplane, marked "OPEN" and "CLOSE," control electrical power to the canopy actuator. The selected button must be held down until the canopy reaches the desired position, then released. Limit switches in the canopy actuator automatically cut off power to the actuator when the canopy reaches the full open or full closed position.

Canopy Switch.

A three-position switch (19, figures 1-8 and 1-9; 18, figure 1-14) controls canopy operation from the cockpit. The switch is spring-loaded to OFF from both OPEN and CLOSE positions. To fully open the canopy, the switch must be held at OPEN until canopy travel is completed, at which time electrical power is automatically cut off. Releasing the switch to OFF stops the canopy at any point. To obtain a full closed position, the switch is held at CLOSE until the canopy is fully closed and locked, at which time electrical power is automatically cut off. Should the canopy fail to jettison during in-flight emergency procedures, it can be opened and blown off by the slipstream. However, at certain flight conditions, air loads may prevent canopy from opening by this method.

Canopy Lock Handle.*

The two-position canopy lock handle (figure 1-38) in the front cockpit, is used to mechanically lock and unlock the canopy. The canopy must be locked manually in the front cockpit after it is closed and manually unlocked before it is opened. Pushing the canopy lock handle forward to LOCKED disengages the canopy switches from the canopy actuating circuit and manually locks the canopy. A guard on the canopy lock handle prevents it from being moved unintentionally to UNLOCKED. When the lock handle is pulled aft to UNLOCKED, it is possible to open the canopy by selecting OPEN with the canopy switch in either cockpit. Actuating the canopy lock handle does not disturb the manual emergency release handles.

NOTE

Emergency removal of the canopy by use of the canopy remover or manual emergency release handles automatically releases the canopy lock handle. Use of the manual emergency release handle to unlock the canopy does not render the electrical canopy actuator inoperative.

Canopy Alternate Emergency Jettison Handle.

A canopy alternate emergency jettison handle (26, figure 1-10; 27, figure 1-11; 27, figure 1-15; 20, figure 1-18) jettisons the canopy without arming the seat catapult. This handle is either a "T" or "D" handle and has a spring clip over the top which must be released before the handle can be pulled. (See figure 3-5.) When this handle is pulled to its full extended position (about 1 inch), a mechanical linkage fires a cartridge in the canopy initiator. This causes the canopy remover to fire, jettisoning the canopy.

NOTE

This handle is intended only as an alternate means of removing the canopy when it is desired to jettison the canopy only.

Canopy Internal Manual Emergency Release Handle.

The canopy internal manual emergency release handle right side only (figure 1-38) is used to unlock and open the canopy manually when the airplane is on the ground, if the canopy cannot be opened electrically. Opening the canopy internal manual emergency release handle to its extended position unlocks the manual lock,† deflates the canopy seal, and allows the canopy to be pulled back manually (by means of the handle) to an unlocked position. The forward end of the canopy may then be raised manually or electrically.† Although this handle is intended for use while the airplane is on the ground, it may be used in an attempt to open the canopy if the canopy fails to jettison during emergency procedures.

*F-100F-2 Airplanes AF56-3726 and -3727; F-100F-11 and later airplanes; and F-100F Airplanes changed by T.O. 1F-100F-562

†F-100F Airplanes

Canopy External Manual Emergency Release Handles.

Two canopy external manual emergency release handles (figure 1-38), one flush-mounted outside on each side of the canopy frame, are used to gain access to the cockpit in case of an emergency when the airplane is on the ground. Releasing either handle to an extended position allows the canopy to be moved back manually about one inch, thereby unlocking it. The canopy may then be lifted at its forward end to allow entrance to the cockpit. It is possible to then latch the canopy in the open position, or to remove it from the airplane (as is recommended for emergency entrance) by lifting the canopy up and rotating it aft as far as possible.

Canopy External Emergency Jettison Handle — F-100D-61 and Later F-100D Airplanes, and F-100F Airplanes.

The canopy external emergency jettison handle (figure 1-38), housed within an access door on the left side of the fuselage below the windshield, permits ground rescue personnel to jettison the canopy from the airplane for emergency entrance. The circular access door is secured by a spring-loaded push-type latch, and is identified by an arrow on the fuselage labeled "RESCUE."

When the door is unlatched and removed, the jettison handle and 6 feet of cable are exposed. To jettison the canopy, the handle must be pulled until the slack is removed from the cable, and then pulled another inch. This causes the canopy remover to fire and jettison the canopy.

Canopy Remover Lock.*

The canopy remover lock (figure 1-38) is marked "OPEN" and "CLOSED." In the initial phase of the canopy jettison operation, the head of the canopy remover, in its upward travel, strikes the forward "OPEN" portion of the canopy remover lock. This causes the lock to rotate and lock the remover to the canopy. The "OPEN" portion should be exposed so that the canopy remover will be locked to the canopy during canopy jettison. If the aft "CLOSED" portion of the lock is exposed, the canopy remover will not be retained by the lock during canopy jettison, but will be left in the cockpit to possibly endanger the rear cockpit occupant.

Canopy-Not-Locked Warning Light.

A placard-type canopy-not-locked warning light (figure 1-19) is illuminated by primary bus power when the canopy is in any position other than full closed and locked. On some F-100F airplanes, the canopy-not-locked warning lights come on the moment the canopy lock handle is placed at UNLOCKED. The bulbs can be checked by the indicator light test circuits.

CAUTION

If canopy-not-locked warning light comes on during flight, do not actuate the canopy switch, as this may cause the canopy to leave the airplane. A landing should be made as soon as practical after the canopy-not-locked light comes on.

EJECTION SEAT.

WARNING

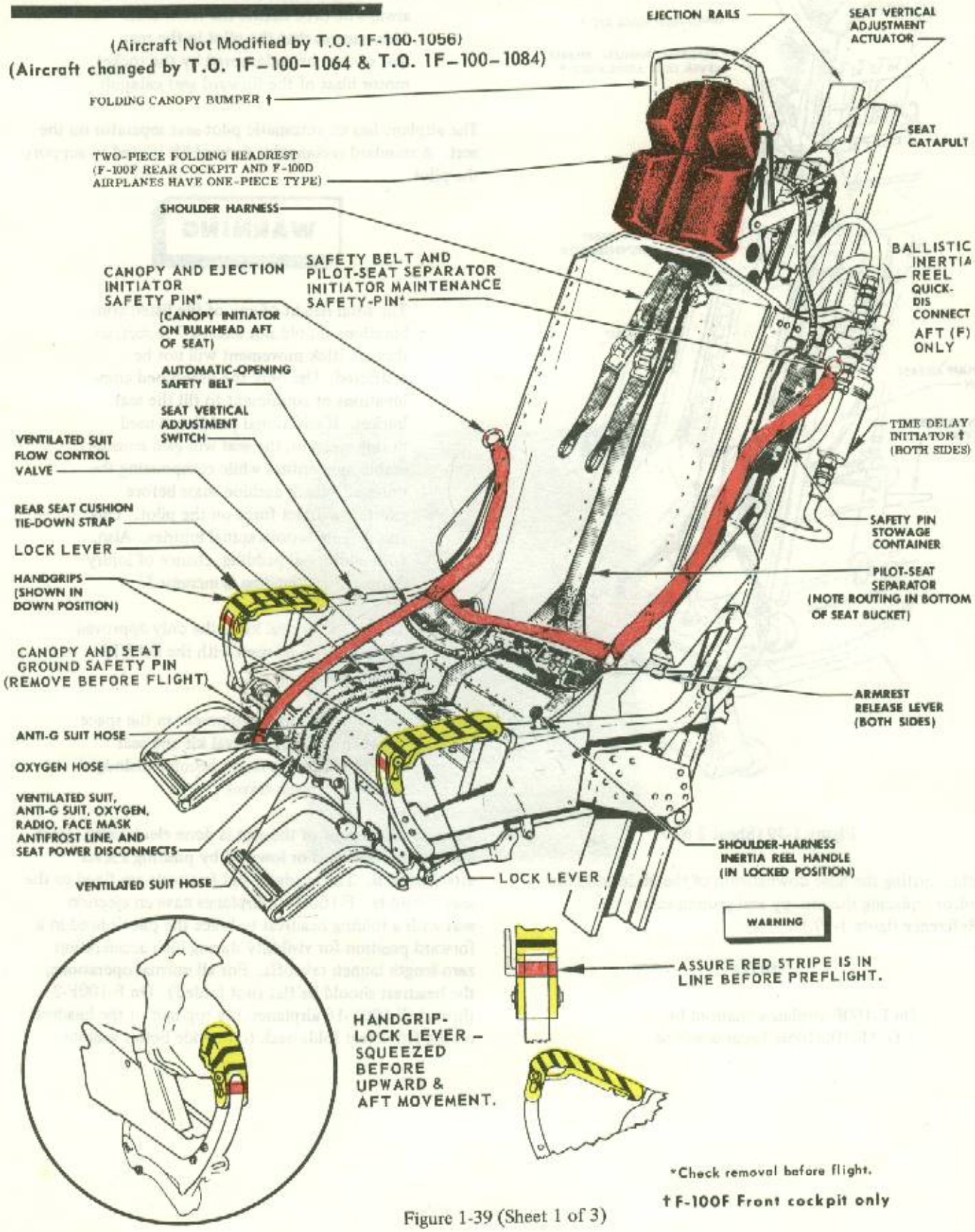
On aircraft changed by T.O. 1F-100-1064, inspect the red stripe on the forward face of both handles and guards for alignment. If the red stripes are not in alignment, the ejection seat is unsafe for flight.

The ejection seat (figure 1-39) permits ejection at any speed or flight attitude. A ballistic rocket ejection seat catapult ejects the seat and pilot from the airplane. The seat has an automatic opening safety belt and accommodates a back-type parachute. On aircraft modified by T.O. 1F-100-1064, the quick disconnect has been removed. Therefore, the seat should be rendered inoperative during emergency rescue by

*F-100F-1 Airplanes AF56-3726 and -3727, and F-100F-11 and later airplanes

EJECTION SEAT (TYPICAL)

(Aircraft Not Modified by T.O. 1F-100-1056)
(Aircraft changed by T.O. 1F-100-1064 & T.O. 1F-100-1084)



*Check removal before flight.

† F-100F Front cockpit only

Figure 1-39 (Sheet 1 of 3)

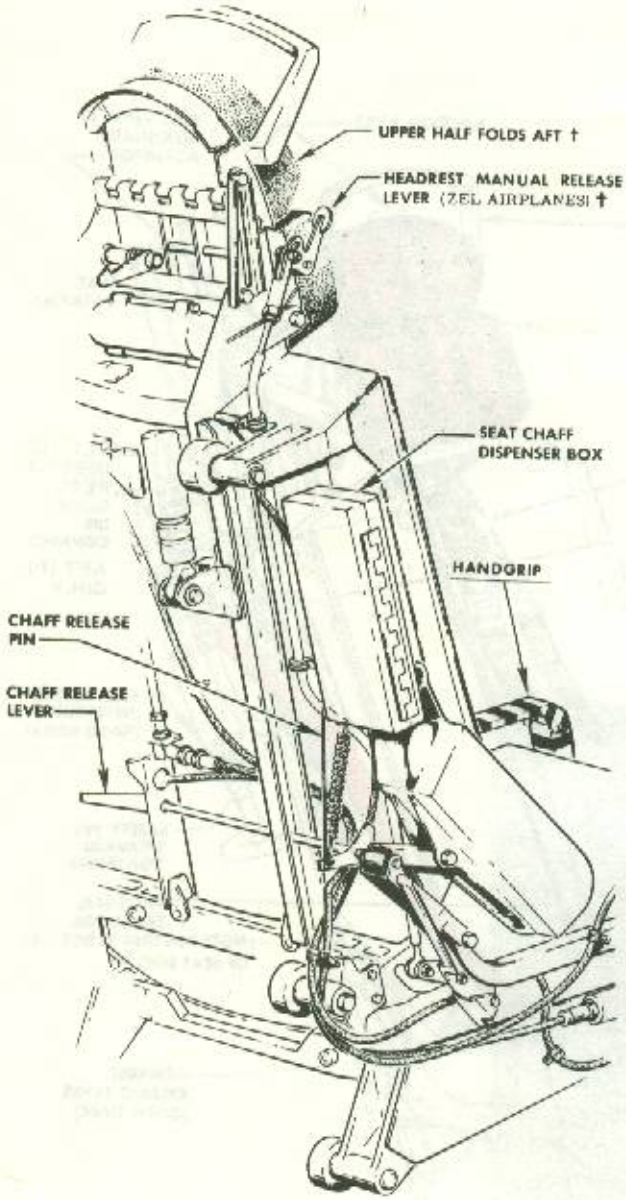


Figure 1-39 (Sheet 2 of 3)

either cutting the hose downstream of the M-26 initiator and/or replacing the canopy and ground safety pin. (Reference figure 1-39.)

NOTE

On F-100F airplanes changed by T.O. 1F-100-1056, because of the tandem location of the pilots' seats, a 1-second time-delay system sequences the seat ejection so that the aft seat will always be fired before the front seat. This ensures that the pilot in the rear seat cannot be endangered by the rocket motor blast of the forward seat catapult.

The airplane has an automatic pilot-seat separator on the seat. A standard rectangular survival kit is used to support the pilot.

WARNING

The total height of any of the listed combinations should not exceed 7 inches, so that aft stick movement will not be restricted. Use only the prescribed combinations of equipment to fill the seat bucket. If additional padding is used during ejection, the seat will gain considerable momentum while compressing the unusually thick cushion mass before exerting a direct force on the pilot. This could cause serious spinal injuries. Also, with additional padding, chance of injury during forced landing is increased.

- The MD-1 survival kit is the only approved survival kit to be used with the DART/Snubber seat.
- Do not stow loose equipment in the space between front of survival kit and seat bucket because it might become dislodged and restrict stick travel.

Vertical adjustment of the seat is done electrically. Each armrest can be raised or lowered by pushing a lever attached to it. The headrest and footrests are fixed to the seat assembly. F-100D-91 airplanes have an ejection seat with a folding headrest to brace the pilot's head in a forward position for visibility during high acceleration zero-length launch takeoffs. For all normal operations, the headrest should be flat (not folded). On F-100F-2 through F-100F-16 airplanes, the top half of the headrest on the front seat folds back to provide better visibility

DART/SNUBBING EJECTION SEAT

(Airplanes changed by T.O. 1F-100-1056 & T.O. 1F-100-1064)

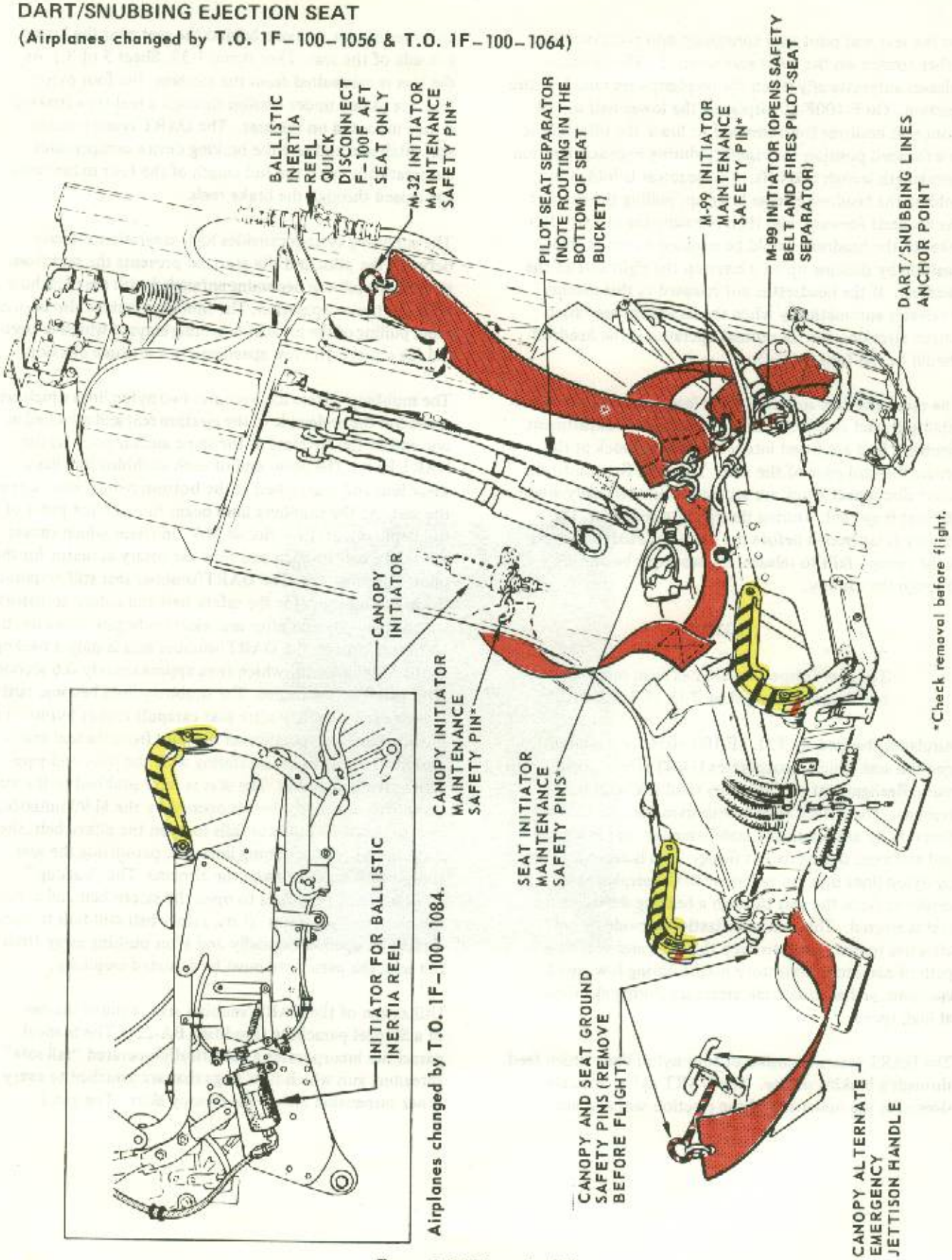


Figure 1-39 (Sheet 3 of 3)

for the rear seat pilot, and springs up into position when either armrest on the front seat is raised. The headrest releases automatically when the handgrips are raised before ejection. On F-100F-20 airplanes, the lower half of the front seat headrest folds forward to brace the pilot's head in a forward position for visibility during high-acceleration zero-length launch takeoffs. The headrest is folded by holding the headrest release lever up, pulling the sides of the headrest forward, and then releasing the lever. After takeoff, the headrest should be released to the normal position by pushing up on a lever on the right side of the headrest. If the headrest is not released in this manner, it releases automatically when the handgrips are raised before ejection. For all normal operations, the headrest should be flat (not folded).

The radio and face mask antifrost leads; the oxygen, ventilated suit, and anti-G suit hoses; and the seat adjustment electrical lead are fitted into a disconnect block at the center forward edge of the seat. The lines fitted into the lower disconnect block are separated automatically when the seat is ejected. During the ejection sequence, the canopy is jettisoned before the seat is ejected; however, if the canopy fails to release, the seat may be ejected through the canopy.

NOTE

The chaff dispenser box has been removed on airplanes changed by T.O. 1F-100-1072.

Airplanes changed by T.O. 1F-100-1056 have a modified ejection seat which incorporates DART (Directional Automatic Realignment of Trajectory) and seat snubbing systems. The DART feature improves the ejection seat function by automatically stabilizing the seat in the pitch and roll axes, during rocket firing. This is accomplished by nylon lines that are anchored to the airplane and deployed from the seat through a braking device as the seat is ejected. This seat stabilization provides more effective rocket thrust to keep the pilot and seat on a path of maximum trajectory height during low speed ejections, and adequate tail clearance during ejection at high speed.

The DART system consists of four nylon lines which feed through a braking device. The DART nylon lines are stowed at the underside of the ejection seat and are anchored to the airframe behind the seat near the lower left side of the seat. (See figure 1-39, Sheet 3 of 3.) As the seat is catapulted from the airplane, the four nylon lines are pulled under tension through a reel-type braking device mounted on the seat. The DART system senses seat pitch and roll and the braking device compensates automatically until the full length of the four nylon lines has passed through the brake reels.

The snubbing system provides high separation velocity between the pilot and the seat and prevents the seat from striking the pilot or becoming entangled with the parachute after pilot/seat separation. The snubbing feature also insures rapid pulling of the parachute arming lanyard which is essential for a successful low speed and low altitude ejection.

The snubbing system incorporates two nylon lines which are stowed at the underside of the ejection seat and attached at one end to the airframe at the same anchor point as the DART lines. The other end of each snubbing line has a shear link and is attached at the bottom rear of each side of the seat. As the snubbing lines begin to reach the point of full deployment, they fire an M-99 initiator which causes the safety belt to open and fires the rotary actuator for the pilot/seat separator. The DART/snubber seat still employs the M-32 initiator (for the safety belt and rotary actuator) which fires 1 second after seat ejection begins; however, the M-32 function on the DART/snubber seat is only a backup to the M-99 initiator which fires approximately 0.6 second after seat ejection begins. The snubbing lines become taut (which occurs shortly after seat catapult rocket burnout) to provide positive separation of the pilot from the seat and to prevent the seat from interfering with the pilot and parachute after separation. The seat is fully snubbed to the airplane after the safety belt is opened by the M-99 initiator. In case the M-99 initiator fails to open the safety belt, the shear links in the snubbing lines fail, permitting the seat and pilot to separate from the airplane. The "backup" M-32 initiator then fires to open the safety belt and actuate the pilot/seat separator. If the safety belt still fails to open, it must be opened manually and after pushing away from the seat, the parachute must be actuated manually.

Utilization of the DART/snubber seat requires the use of a Stencel parachute (modified BA-22). The Stencel parachute incorporates a ballistically-operated "fail safe" spreading gun which fires slugs that are attached to every other suspension line at the canopy skirt. The gun is

fired by a lanyard when a certain amount of tension is imposed on the suspension lines as the parachute deploys. This causes the parachute canopy to be spread symmetrically and rapidly.

The symmetrical spreading of the canopy distributes the airloads equally which prevents stress concentration on individual lines or panels.

WARNING

Due to the symmetrical and rapid spreading of the canopy, the opening shock forces exerted on the pilot are increased considerably; therefore, the leg straps of the parachute harness must be adjusted to a snug fit.

If the spreading gun does not fire, the slugs are released and the parachute canopy inflates aerodynamically in the usual manner. The Stencel parachute also utilizes low speed and high speed external pilot chutes. These external pilot chutes assist the internal pilot chute in pulling the canopy from the pack and reduce the time from release of the internal pilot chute to firing of the "fail safe" spreading gun. The low speed external pilot chute is designed to break away at speeds in excess of 190 knots. When this occurs, the high speed external pilot chute is deployed.

The zero-delay lanyard hook to the parachute ripcord handle is not utilized on the Stencel parachute. The Stencel parachute incorporates a 3/4-second timer to delay chute opening after the arming lanyard is pulled instead of the 1-second delay timer that is used on the unmodified BA-22 parachute.

Because of the relationship of the seat trajectory with the pilot trajectory after separation using the DART/snubber system compared to the trajectories without DART/snubbing, the Stencel parachute must be used with the DART configuration but MUST NOT be used with airplanes that do not have the DART configuration.

WARNING

Use of incorrect parachute for the ejection seat configuration may result in seat/parachute entanglement during ejection.

SHOULDER HARNESS INERTIA REEL.

A shoulder harness inertia reel, on the back of the ejection seat, has a manual control to permit pilot selection of reel locking. The multidirectional-type reel automatically locks the shoulder harness when the harness cable is pulled from the reel with an acceleration of 2 G to 3 G in any direction. This locking depends only on the speed at which the cable is pulled from the reel, and not upon the inertia imposed on the reel. Movement of the pilot in any direction that pulls that cable from the reel at the preset speed locks the reel. The inertia reel is locked automatically when either ejection seat handgrip is raised during seat ejection sequence. On F-100F-20 airplanes, the shoulder harness has a multidirectional type reel which automatically locks the shoulder harness by the speed the harness is pulled from the reel or when the reel is subjected to a side acceleration exceeding 2G. The inertia reel is locked automatically when either ejection seat handgrip is raised during the seat ejection sequence.

Airplanes changed by T.O. 1F-100-1084 have incorporated a Ballistic Powered Inertia Reel (BPIR). The BPIR automatically retracts when either ejection seat handgrip is raised at the start of the seat ejection sequence. The quick-disconnect on the F-100F aft seat is incorporated to allow seat/aircraft separation. If the quick-disconnect is not properly engaged, a red ring should be visible at the separation point.

PILOT-SEAT SEPARATOR.

An A-shaped web harness automatically separates the pilot from the seat after the seat belt opens. The harness lies in the bottom of the ejection seat bucket, and extends up the back of the seat over a roller under the headrest. The

forward ends of the webbing are attached to the front of the seat and the other end is attached to a rotary actuator under the parachute shelf at the rear of the seat. The rotary actuator is powered by gases from the safety belt initiator. When the safety belt initiator fires, gas pressure is provided to both the safety belt, which opens, and the rotary actuator, which starts reeling in the pilot/seat separator webbing. This pulls the webbing tight between the front edge of the seat bucket and the headrest roller, forcing the pilot from the seat.

EJECTION SEAT CONTROLS.

Ejection Seat Handgrips.

Ejection is initiated by squeezing the locking lever on either handgrips inward and raising the leg guards. The handgrips are latched in the down position by the locking lever which is engaged into horizontal slots on the guard and down lock lugs. Squeezing the locking lever inward toward the center line of the airplane releases the handgrips for the upward and aft movement necessary to jettison the canopy and initiate the ejection. Since the seat handgrip assemblies are linked together by a torque-tube, pulling up on either handgrip automatically raises the other to the full up position. Raising either handgrip to full up fires a cartridge in an initiator unit behind the seat. The expanding gases are discharged through a flexible hose to an exactor unit on the canopy remover and to an M-26, 0.3-second delay initiator. This gas pressure pulls the seat pin from the canopy remover, causing the remover to fire and jettison the canopy and fire the M-26 initiator. The gas pressure generated by the M-26 initiator is directed 0.3 seconds later into the ejection seat catapult. When either or both ejection seat handgrips are raised the shoulder harness is automatically locked and on aircraft changed by T.O. 1F-100-1084, the shoulder harness is automatically retracted placing the pilot into the optimum position for ejection. In addition, the front seat folding canopy bumper on the F-100F airplanes is automatically raised into position by spring action. A pull of approximately 30 pounds is required to raise either handgrip.

On F-100F airplanes, ejection of the front and rear seats is normally controlled by their respective handgrips (airplanes changed by T.O. 1F-100-1064). However, the rear seat can also be fired from the front cockpit. If the rear seat has not already been ejected, a time-delay system sequences ejections so that raising either front seat handgrip will eject the rear seat first, followed 1/2 to 1 second later by the front seat. The front seat time delay is 1/2 second on airplanes not changed by T.O. 1F-100-1056 and 1 second in airplanes changed by T.O. 1F-100-1056.

Seat Vertical Adjustment Switch.

The seat is vertically adjusted by primary bus power through a three-position switch (figure 1-39) which is spring-loaded to OFF (center). The seat may be stopped at the desired position by releasing the switch.

CAUTION

Caution should be exercised to prevent continued operation of the switch in the DOWN position after the seat reaches the FULL DOWN position. In order to remove any possible strain, it is recommended that the switch be returned to the UP position momentarily to assure that the catapult mechanism is not in a prestressed position.

NOTE

On airplanes changed by T.O. 1F-100-1103, an electrical down travel limit switch has been added to provide clearance for the DART/snubber.

Shoulder Harness Inertia Reel Handle.

On airplanes changed by T.O. 1F-100-1084, F-100F rear seat only, check the quick disconnect to insure that the red stripe is NOT visible.

The shoulder harness inertia reel is manually controlled by a handle. (See figure 1-39.) Forward is INERTIA REEL LOCKED; aft is INERTIA REEL UNLOCKED. It is recommended that the shoulder harness be locked manually during maneuvers and flight in rough air, or as a safety precaution in case of a forced landing. The shoulder harness locks automatically when the handle is in the INERTIA REEL UNLOCKED position. Whenever the reel is locked, any movement of the pilot toward the seat

releases tension on the cable and it automatically retracts. As the cable retracts it locks at each 1/2 inch of retraction. Once the reel has been locked automatically, the handle must be moved forward to INERTIA REEL LOCKED and then back to INERTIA REEL UNLOCKED to unlock the reel and permit the pilot to move forward.

NOTE

Adjust shoulder harness with inertia reel handle in the LOCKED position before placing handle at UNLOCKED; otherwise, it may become necessary to unfasten the safety belt in flight to unlock the shoulder harness reel.

AUTOMATIC-OPENING SAFETY BELT.

The ejection seat has an automatic-opening safety belt. In high-altitude ejections (above 14,000 feet), use of the automatic belt, *in conjunction with the automatic-opening parachute*, avoids parachute deployment at an altitude where sufficient oxygen would not be available to permit safe parachute descent. In a low-altitude ejection, use of the automatic belt greatly reduces the time required for separation from the seat and full parachute deployment. *Under no circumstances should the automatic belt be manually opened before ejection, regardless of altitude.* (The M-32 automatic belt initiator opens the belt one second after ejection.) Since the drag-to-weight ratio of the seat is considerably greater than that of the pilot, immediate separation would result if the belt were opened manually before ejection. This could result in the parachute pack accidentally blowing open, and the high opening shock of the parachute could cause serious or fatal injuries.

HBU AUTOMATIC-OPENING SAFETY BELT – AIRCRAFT CHANGED BY T.O. 1F-100-1072.

The HBU belt has been thoroughly tested and proven to eliminate the occurrence of inadvertent lap belt opening or binding of the automatic release. Manual release of the lap belt is accomplished by rotation of the rotary mechanism on the belt. Automatic operation is accomplished during ejection by gas pressure from a separate automatically controlled initiator. A description of the hook-up, use, and operation of the HBU lap belt when used in conjunction with the automatic parachute is as follows:

1. Manual Operation: Refer to figure 1-40, for proper sequence of opening and closing the HBU lap belt, shoulder harness, automatic parachute arming lanyard anchor (Gold Key).

2. Automatic Operation: Automatic lap belt opening and man/seat separation are accomplished as part of the seat ejection sequence and requires no additional effort on the part of the pilot.

WARNING

Under no circumstances should the belt be manually opened before ejection regardless of altitude.

3. Within 1 second after seat ejection begins, an automatically-fired initiator causes the safety belt to open (releasing the shoulder harness) and fires the rotary actuator for the pilot/seat separator. The automatic arming lanyard (Gold Key) is retained by the left half of the safety belt and actuates the automatic parachute as the pilot separates from the seat.

LOW-ALTITUDE ESCAPE EQUIPMENT.

Low-altitude escape capability is provided on airplanes without the DART/snubber seat (i.e., airplanes not changed by T.O. 1F-100-1056), by means of an M-32 initiator and use of a zero-delay parachute arming lanyard hook. The M-32 initiator fires 1 second after seat ejection begins in order to open the safety belt and fire the pilot/seat separator actuator. Zero-delay deployment of the parachute is accomplished by use of a lanyard which is fastened to the parachute arming lanyard and has a detachable hook that can be attached to the parachute ripcord handle. At low altitude, the zero-delay hook must be connected to the parachute ripcord handle, to automatically deploy the parachute immediately after separation from the ejection seat. At higher altitudes, the hook must be disconnected from the parachute ripcord handle, thus allowing the parachute aneroid and 1-second

timer to delay parachute deployment until below 14,000 feet and to reduce opening shock at high speed. A ring, attached to the parachute harness, is provided to stow the zero-delay hook when the hook is not connected to the ripcord handle.

Improved low altitude escape capability is provided on airplanes with the DART/snubber seat (i.e., airplanes changed by T.O. 1F-100-1056) by means of nylon stabilization and snubbing lines, an M-99 initiator (with the 1-second delay M-32 initiator retained as a backup) to open the safety belt and fire the pilot/seat separator actuator, and use of a Stencel parachute. Refer to Ejection Seat in this section for detailed information concerning the Stencel parachute. Although the parachute automatic arming lanyard anchor is inserted in the notch at the top of the HBU safety belt, the zero-delay lanyard hook to the parachute ripcord handle is not utilized on the Stencel parachute. This eliminates the need to manually attach or disconnect the lanyard hook to the ripcord handle during various phases of flight.

Refer to Ejection in section III for emergency minimum ejection conditions and zero-delay lanyard connection requirements.

AUXILIARY EQUIPMENT.

Information concerning auxiliary equipment is supplied in section IV.

AUTOMATIC-OPENING SAFETY BELT, TYPE HBU

(Aircraft changed by T.O. 1F-100-1072)

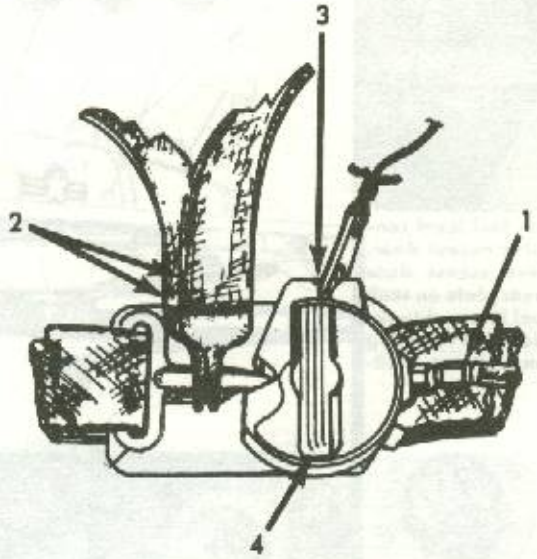
LOCKED

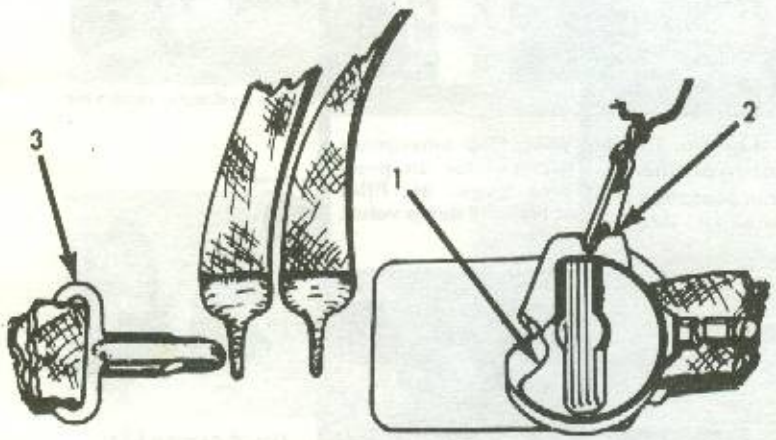
1. Initiator hose to automatic release mechanism.
2. Shoulder harness loops over link.
3. Anchor (from automatic parachute arming lanyard) inserted in notch at top of buckle.
4. Manual release handle is closed.

WARNING

Lap belt cannot be fastened until zero delay lanyard anchor is properly inserted into slot at top of buckle. After lap belt is fastened perform a pull check of the arming lanyard to insure that the Gold Key is secured.

- Lanyard must be outside parachute harness and not fouled on any equipment, to permit clean separation from seat.
- To ensure a positive lock of link and anchor into lap belt buckle mechanism, do not rotate manual release handle while fastening lap belt.
- When tightening the lap belt, assure that belt webbing is under stress and not the ballistic hose. Compensate as necessary by adjusting the webbing on the link half of the belt to relieve tension on the ballistic hose.





AUTOMATICALLY OPENED

1. Automatic release mechanism actuated by gas pressure from initiator, detaching link from lap belt buckle mechanism.
2. Anchor (from automatic parachute arming lanyard) retained by buckle mechanism.
3. Link released.

MANUALLY OPENED

1. Link released by manual release lever (automatic release mechanism not actuated).
2. Anchor (from automatic parachute arming lanyard) freed from buckle mechanism.

WARNING

If belt is manually opened during ejection, the parachute will not open automatically upon separation from the seat.

3. Manual release rotated to open position.

NOTE

Manual release handle can be used to unlock belt at any time, even if automatic-opening sequence has been initiated.

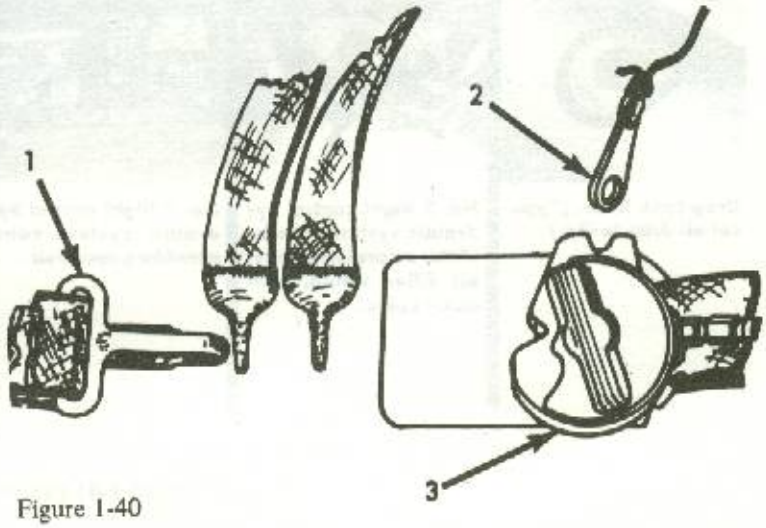


Figure 1-40

SERVICING DIAGRAM (TYPICAL)

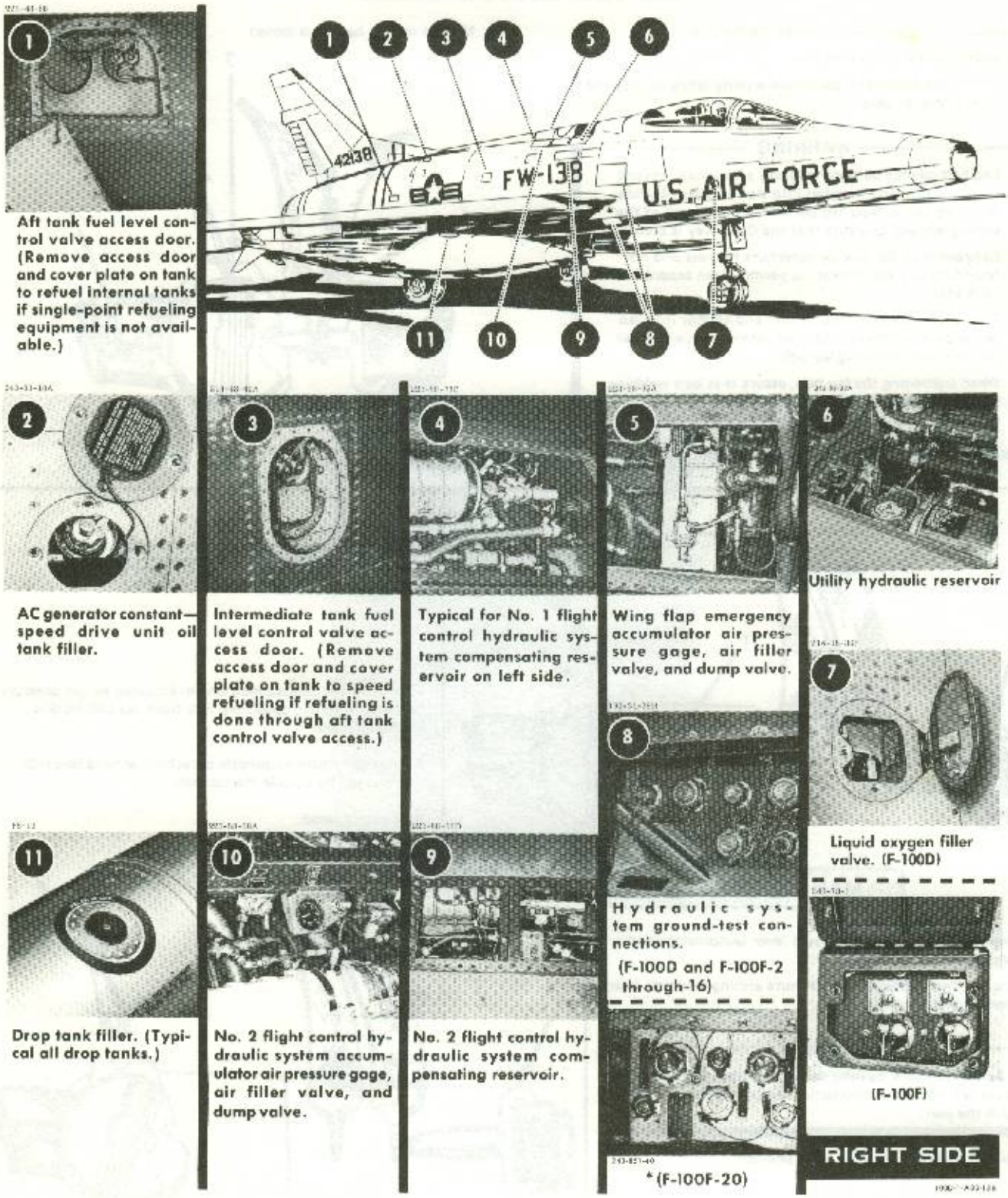


Figure 1-41 (Sheet 1 of 3)

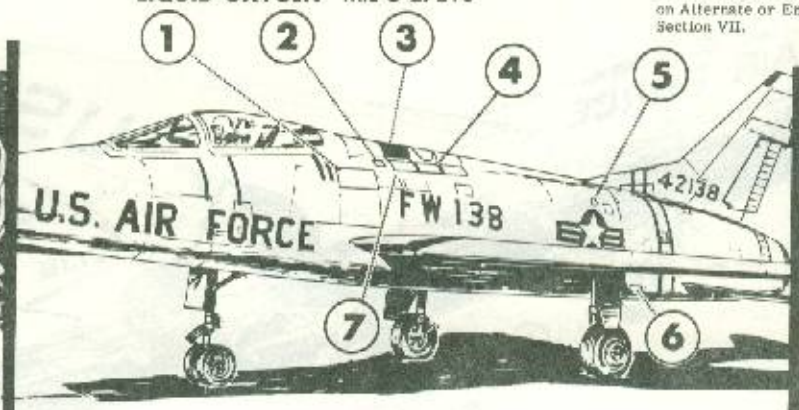
SPECIFICATION

FUEL JP-4 (MIL-J-5624)
OIL MIL-L-7808
HYDRAULIC FLUID MIL-H-5606
LIQUID OXYGEN MIL-O-27210

NOTE

For alternate and emergency fuels, refer to "Fuel Grade Properties and Limits" in Section VII.

When using alternate or emergency fuels, refer to "Alternate and Emergency Fuel Limitations" in Section V and "Operation on Alternate or Emergency Fuel" in Section VII.

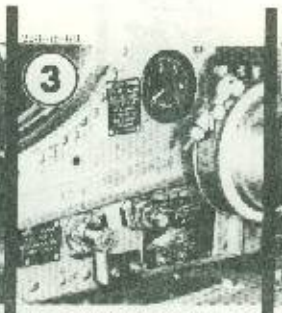


Battery and battery compartment circuit-breaker panel. (F-100D)

Battery and battery compartment circuit-breaker panel. (F-100F)

Cockpit pressure and canopy seal ground test, and pressure gage connections. (F-100D)

Cockpit pressure and canopy seal ground test, and pressure gage connections. (F-100F)



No. 1 flight control hydraulic system accumulator air pressure gage, air filler valve, and dump valve. (F-100D)

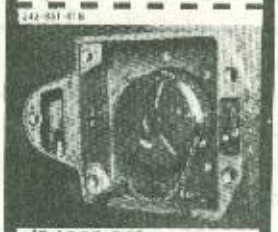
No. 1 flight control hydraulic system accumulator air pressure gage, air filler valve and dump valve. (F-100F)

Ram-air turbine door emergency accumulator air pressure gage, air filler valve, and dump valve. (F-100D)

No. 1 flight control hydraulic system auxiliary reservoir air pressure gage and air filler valve. (F-100F)

No. 1 flight control hydraulic system auxiliary reservoir air pressure gage and air filler valve. (F-100D)

Single-point refueling receptacle. (F-100D and F-100F-2 through-16)



(F-100F-20)

Engine oil tank filler.

Ram-air turbine door emergency accumulator air pressure gage, air filler valve, and dump valve.

Hydraulic hand-pump and hand-pump selector valve. (F-100F)

LEFT SIDE

Figure 1-41 (Sheet 2 of 3)

SERVICING DIAGRAM (TYPICAL)

Antiskid and wheel brake emergency accumulator air pressure gages, air filler valves, and dump valve. (in right side of nose wheel well.)

Nose gear emergency lowering accumulator air pressure gage, air filler valve, and dump valve.

External power receptacles for starter air supply and ac and dc electrical power.

NOTE

Suitable external dc power units are A3, A4, C-21, C-22, C-26, C-27, and V1. NC-5, MA-1, MA-1A, or MA-2 power units can supply both ac and dc power.

Cartridge starter loading breech and cartridge power-on warning light

Drag chute compartment doors.

ACCUMULATOR PRECHARGE PRESSURE (PSI)

PLACARDED ACCUMULATOR PRECHARGE

TEMPERATURE (°F)	−20	−10	0	10	20	30	40	50	60	70	80	90	100	110	120
WING FLAP EMERGENCY ACCUMULATOR	860	882	904	924	945	966	988	1010	1030	1050 (±50)	1072	1094	1114	1135	1155
NOSE GEAR EMERGENCY LOWERING ACCUMULATOR	980	1015	1025	1055	1080	1105	1130	1150	1175	1200 (±50)	1225	1250	1275	1300	1320
WHEEL BRAKE EMERGENCY ACCUMULATOR	415	420	430	440	450	460	470	480	485	500 (−50)	510	520	530	540	550
ANTISKID ACCUMULATOR	255	260	268	275	280	285	290	298	305	310 (±15)	315	322	330	335	340
RAM-AIR TURBINE DOOR EMERGENCY ACCUMULATOR	1480	1515	1550	1585	1620	1660	1695	1730	1765	1800 (±50)	1835	1870	1905	1945	1980
NO. 1 AND NO. 2 FLIGHT CONTROL HYDRAULIC SYSTEM ACCUMULATORS	495	510	520	530	545	555	565	580	590	600 (+50)	610	625	635	645	655

Figure 1-41 (Sheet 3 of 3)