overspeed protection by regulating the flow of hydrogen peroxide to the decomposition chamber. The key component of each speed control is a controller which contains the necessary circuits for sensing unit operation through the frequency output of a tachometer generator. During normal operation, the frequency of the power generated by the tachometer generator, being proportional to the turbine speed, matches a preset frequency of the controller. Any change in turbine speed due to a change in load, or from any other disturbance, causes a proportional change in frequency of the tachometer generator. This frequency change is sensed by the controller, which in turn adjusts the opening of the flow control valve to bring turbine speed back to the normal operating level. During normal operation, the speed of the unit is automatically controlled to maintain 51, 200 rpm by the speed-sensing element of the controller. Should an overspeed condition occur (56, 000 rpm or greater), the overspeed sensing element of the controller automatically acts to energize the solenoid-operated APU shutoff valve to the closed position, thereby shutting off the flow of the propellant. The unit then decelerates and stops. It cannot be restarted until the APU switch is first cycled to the OFF position.

APU SWITCHES.

There are two APU switches (23 and 46, figure 1-2) on the instrument panel, one for control of each auxiliary power unit and its associated feed system. When either switch is turned to ON, battery-bus power is used to open the helium shutoff valve in the related propellant feed system, allowing helium pressure to move the hydrogen peroxide through the feed lines. At the same time, power is applied to the opening circuit of the related APU shutoff valve. This permits the propellant to flow to the auxiliary power unit. Turning the switch to OFF closes the helium shutoff valve (if the ballistic control switch is at OFF), shutting off the helium supply.

NOTE

> If the ballistic control switch is at ON, the OFF position of the APU switch will not close the helium shutoff valve.

At the same time, the APU shutoff valve closes, shutting off the flow of the propellant to the unit. The JETT position, powered by the primary dc bus, is used if an emergency arises and it is desired to jettison the propellant overboard. The switch is guarded to prevent it from being accidentally moved to the JETT position. When the switch is turned to JETT, the following occurs: The helium shutoff valve in the feed system opens, or remains open if it is already so, allowing the helium to continue to force the hydrogen peroxide through the feed lines. Concurrently, the APU shutoff valve closes and a jettison and ballistic control valve in the feed system opens to the jettison position. The jettison and ballistic control valve serves both as a shutoff valve for propellant supply to the ballistic control system and as a propellant jettison control. As this valve opens to the jettison port, the propellant is routed through a line that dumps overboard at the aft end of the airplane.

APU COMPARTMENT OVERHEAT CAUTION LIGHTS.

Two APU compartment amber overheat caution lights (21 and 40, figure 1-2) are adjacent to their related APU switches on the instrument panel. The lights are powered by the battery bus. A thermoswitch in each APU accessory drive compartment is set to energize the light when the temperature in the compartment rises to approximately 525° F. The lights read "APU COMP HOT" when on. If either light comes on, the related auxiliary power unit should be shut down immediately.

APU AND BALLISTIC CONTROL PROPELLANT FEED SYSTEMS.

Two completely independent feed systems provide propellant for the auxiliary power units and the ballistic control system. (Refer to "Ballistic Control System" in this section.) System 1 is in the left side of the fuselage; system 2 is in the right side of the fuselage. The systems are identical. Helium gas under pressure moves the monopropellant hydrogen peroxide to its basic function of providing fuel to these units at the required flow rates and pressures. Each propellant feed system includes a high-pressure, spherical, helium storage tank and a positive expulsion-type hydrogen peroxide storage tank. Both tanks are below the related auxiliary power unit in the forward fuselage. Helium and hydrogen peroxide filler valves and helium high-pressure gages for ground servicing are in each side of the fuselage side fairings. Helium and hydrogen peroxide pressure gages common to both systems are in the cockpit. When the ballistic control switch is turned ON, or when the APU switch is turned ON (or to JETT), a shutoff valve is opened to allow helium to flow from the storage tank. The helium tank contains enough helium to expel all the hydrogen peroxide in the hydrogen peroxide storage tank. Helium pressure is reduced from 3600 psi at the tank to 550 psi as it passes through a pressure regulator. A relief valve upstream of the pressure regulator prevents overpressurization due to overcharging or pressure build-up during high-temperature conditions. From the pressure regulator, the helium passes through the shutoff valve and pressurizes the hydrogen peroxide tank. The positive-expulsion type hydrogen peroxide tank contains a baffle cylinder, perforated to allow the propellant to flow to a pickup tube inside the baffle cylinder. The inlet of the pickup tube is very close to the bottom of the tank to prevent it from being uncovered during normal flight attitudes when only approximately 20 percent of the propellant supply remains in the tank. Between the baffle cylinder and tank wall is a collapsible plastic bladder. The helium enters the tank between the wall and the bladder where pressure on the bladder forces the hydrogen peroxide into the baffle cylinder through the pickup tube and out of the tank. A check valve upstream of the tank prevents hydrogen peroxide from backing into the helium system in case of a bladder failure. When the tank is emptied to the extent that the bladder collapses against the baffle cylinder, the feed pressure will drop off. This pressure drop creates a pressure differential between the helium and hydrogen peroxide. When this pressure differential increases to approximately 35 psi, a differential pressure switch in the system actuates a low-level caution light in the cockpit. Pressure differential is also sensed by a valve that opens at approximately 55 psi and allows helium to flow to the top of the baffle cylinder, expelling the remaining hydrogen peroxide. Protection of the hydrogen peroxide tank against rupture due to overpressurization is provided by a pressure relief valve and a blowout plug. The relief valve is designed to open at approximately 650 psi (100 psi over normal tank pressure). In case of a malfunction of the pressure relief valve or an abnormal rate of pressure increase, the blowout plug will rupture at approximately 900 psi. If the blowout plug should rupture, the affected system will be deactivated by loss of the propellant through the plug. The flow of both of these relief devices is routed overboard through a vent and jettison line. A thermoswitch at the base of the tank energizes a warning light in the cockpit if the hydrogen peroxide in the tank becomes overheated. From the storage tank, the propellant is routed into feed lines through shutoff valves to the auxiliary power unit and the ballistic control system. Temperature of the propellant at the APU inlet must be a minimum of 40° F during starting. To prevent freezing in the feed lines, warm air from the carrier airplane is pumped into the compartment containing the propellant feed system hydrogen peroxide components to maintain a temperature of approximately 120° F. The system is designed to dump helium and hydrogen peroxide overboard if an emergency arises.

APU and Ballistic Control Propellant Feed System Controls and Indicators.

APU Source Pressure Gage. A dual-movement helium pressure gage (42, figure 1-2), common to both propellant feed systems, is on the right side of the instrument panel. The gage, labeled "SOURCE," is calibrated in pounds per square inch and has two pointers, marked "1" and "2" for system identification. The gage includes a slip ring, with calibration markings of "F," "3/4," "1/2," "1/4," and "E." The slip ring is used to indicate the amount of H_2O_2 available to the APU's. The quantity of H_2O_2 available from the tanks is proportional to source (helium) pressure. Therefore, the slip ring should be rotated just before APU start so that the "F" mark is aligned with the No. 1 pointer; then the position of the No. 1 pointer in relation to the slip ring calibrations will indicate the amount of H_2O_2 available for APU operation once they are started. The pressure indicating system is powered by the 26-volt ac bus. Operating pressure will vary from 3600 down to 550 psi, depending upon the helium supply in the storage tank.

APU Hydrogen Peroxide Tank Pressure Gage. A dual-movement hydrogen peroxide pressure gage (37, figure 1-2), common to both propellant feed systems, is on the right side of the instrument panel. The gage is labeled "H_2O_2" and is calibrated in pounds per square inch. The pointers are marked "1" and "2" for system identification. The gage shows tank pressure in the hydrogen peroxide storage tanks, sensed by a pressure transmitter in each feed system. Normal operating pressure is approximately 550 psi. The pressure indicating system is powered by the 26-volt ac bus.

APU Bearing Temperature Gage. A dual-pointer APU bearing temperature gage (38, figure 1-2) is on the instrument panel. The gage shows in degrees centigrade the temperature of No. 1 and No. 2 APU upper turbine bearings. The temperature indicating system is powered by the No. 1 primary ac bus. The gage is calibrated from zero to 200 in increments of 20 degrees. The left pointer indicates No. 1 APU upper turbine bearing temperature; the right pointer, No. 2 APU upper turbine bearing temperature.

Hydrogen Peroxide-low Caution Lights. Each propellant feed system has a low-level caution light (22 and 43, figure 1-2) that reads "H_2O_2 LOW" when on. The two amber, placard-type lights, on the right side of the instrument panel, are powered by the primary dc bus. A pressure-differential switch in each system becomes energized, causing the related light to come on when the differential pressure between helium and hydrogen peroxide rises to approximately 35 psi. The light will come on at this instant, when approximately 20 percent of the hydrogen peroxide supply is left in the storage tank. If either light comes on, extreme maneuvers should be avoided to prevent uncovering the inlet of the pickup tube in the tank, thus allowing helium to flow into the hydrogen peroxide line.

Hydrogen Peroxide Overheat Warning Lights. The No. 1 and No. 2 hydrogen peroxide overheat warning lights (20 and 39, figure 1-2) are on the right side of the instrument panel. The red, placard-type lights are powered by the battery bus and read "H_2O_2 HOT" when on. A thermoswitch at the base of each system hydrogen peroxide storage tank energizes the related light if the temperature of the contents of the tank rises to approximately 160° F. If either light comes on, the contents of the affected tank should be jettisoned. Concurrently, the related auxiliary power unit will automatically shut down.

APU Switches. Refer to "Auxiliary Power Units" in this section.

Ballistic Control System Switches. Refer to "Ballistic Control System" in this section.

ELECTRICAL POWER SUPPLY SYSTEMS.

The airplane is equipped with an alternating-current and a direct-current electrical power system. (See figure 1-9.) Power for the ac system is supplied by two alternator-type generators. The dc system normally is powered from the ac system through two transformer-rectifiers. A 24-volt battery is available for use in an emergency to supply dc power to essential equipment. During ground operation, ac and dc power can be supplied to the airplane by an external power source. During captive flight, the carrier airplane can supply ac and dc power to the airplane. Both external power sources supply minimum dc power for initial relay or valve operation only. Large amounts of dc power then are supplied from the ac system through the transformer-rectifiers.

AC ELECTRICAL POWER DISTRIBUTION.

Two ac generators supply 200/115-volt, 400-cycle, three-phase ac power to the two primary ac busses. Each generator is driven through a gear train by an auxiliary power unit. (Refer to "Auxiliary Power Units" in this section.) Two 26-volt ac busses are powered by the No. 2 primary ac bus. Automatic frequency control and voltage regulation are provided for the ac generators.

AC Generators.

Each ac generator is driven through a gear train by an auxiliary power unit and supplies 200/115-volt, 400-cycle, three-phase ac power to its respective primary ac bus. Failure of an APU causes failure of the ac generator it drives. If one ac generator fails for any reason, the other ac generator automatically supplies power to both primary ac busses. If either generator drops "off the line" because of a momentary malfunction, it can be reset "onto the line."

Primary AC Busses.

The No. 1 primary ac bus normally is powered by the No. 1 ac generator; the No. 2 primary ac bus, by the No. 2 ac generator. However, if either generator fails, the remaining generator will power both primary ac busses. External power, on the ground or from the carrier airplane, will power the primary ac busses, but only when neither ac generator is on.

26-volt AC Busses.

The two 26-volt ac busses are powered through two parallel transformers by the No. 2 primary ac bus. The 26-volt busses are powered as long as either ac generator is operating. In addition, when external power is applied to the airplane on the ground or from the carrier airplane (and both ac generators are off), the 26-volt busses are powered.

DC ELECTRICAL POWER DISTRIBUTION.

Direct-current power is distributed from the 28-volt primary dc bus and the battery bus.

28-volt Primary DC Bus.

The 28-volt primary dc bus is powered by both primary ac busses through two transformer-rectifiers. The primary dc bus, in addition to powering certain equipment, normally powers the battery bus. Failure of one ac generator will not de-energize the primary dc bus. The primary dc bus also is energized when external power is applied on the ground or from the carrier airplane.

Battery Bus.

The battery bus normally is powered by the primary dc bus. However, the emergency battery can be connected to the battery bus to provide emergency dc power. In addition, external power on the ground or from the carrier airplane can be applied to the battery bus.

Emergency Battery.

A stand-by, 24-volt, emergency battery is available to provide emergency power to the battery bus.

ELECTRICALLY OPERATED EQUIPMENT.

See figure 1-9.

EXTERNAL ELECTRICAL POWER RECEPTACLE.

The external power receptacle is on the upper surface of the fuselage, aft of the canopy. (See figure 1-16.) When ac and dc external power is applied to the airplane by a ground unit, an adapter must be used. When external power is applied from the carrier airplane, a single plug-in unit in the carrier airplane pylon is used.

CIRCUIT BREAKERS AND FUSES.

The electrical distribution circuits are protected by circuit breakers and fuses on the electrical power panel in the No. 2 equipment compartment and on the circuit-breaker panel (6, figure 1-4) on the right console in the cockpit. All of the circuit breakers on the right console panel are of the push-pull type. The circuit breakers in the No. 2 equipment compartment must be properly positioned before carrier take-off, because they are not accessible in flight.

CAUTION

> If the two external power circuit breakers in the No. 2 equipment compartment are not closed before carrier take-off, carrier airplane electrical power cannot be applied to the X-15 Airplane.

ELECTRICAL POWER SUPPLY CONTROLS.

No. 1 Generator Switch.

A three-position switch (15, figure 1-2) on the instrument panel controls operation of the No. 1 ac generator by means of battery bus power. The switch is spring-loaded from the RESET position to ON. When the switch is OFF, the generator is taken "off the line." When the generator is "off the line," because the switch is at OFF or because of a momentary generator malfunction, the switch must be moved to RESET momentarily to bring the generator "on the line" and then released to ON to maintain the generator "on the line."

NOTE

> Neither generator will operate unless the APU for the respective generator is also operating and driving the generator.

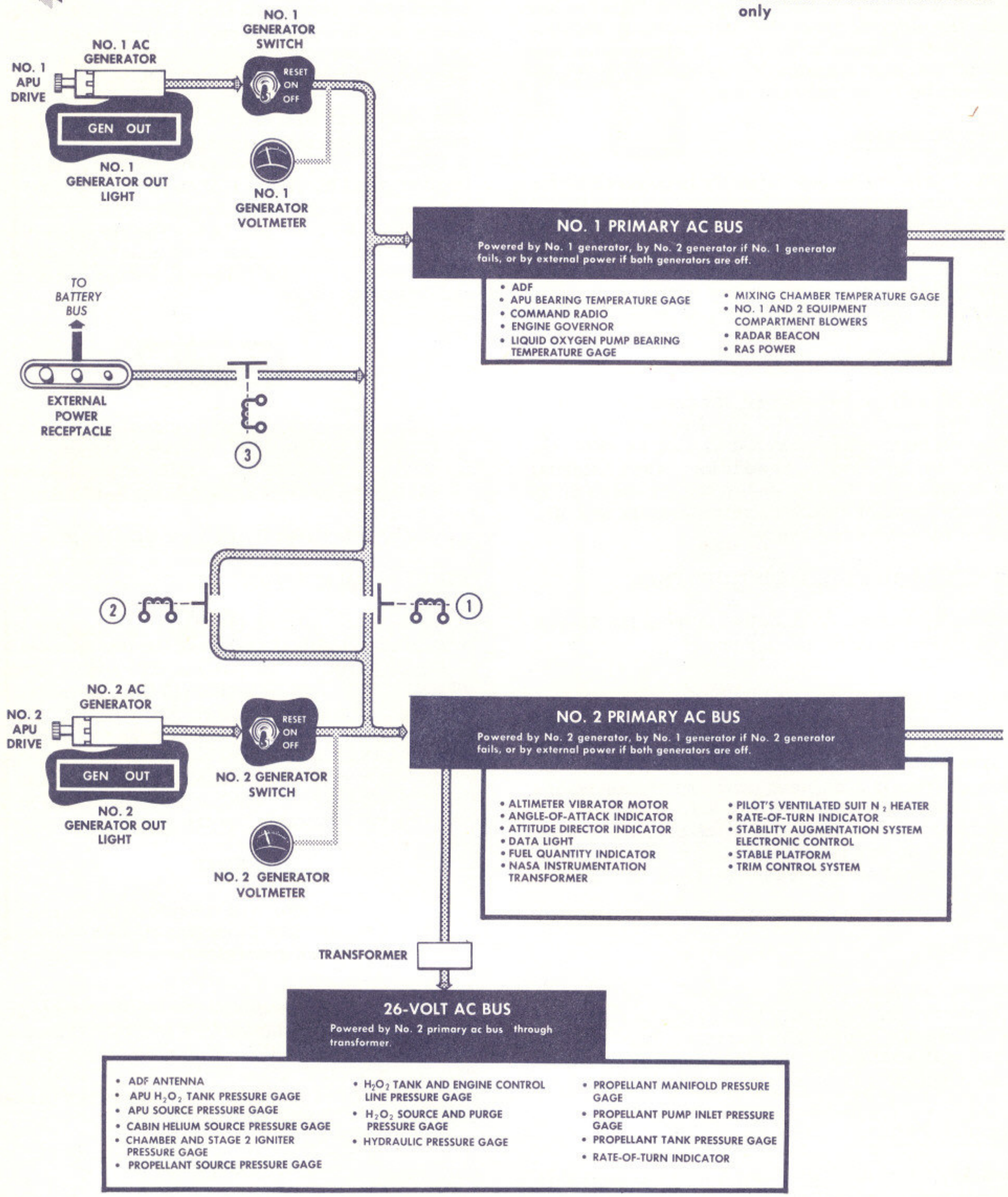
To bring the No. 1 generator "on the line" initially, the switch must be moved from OFF to RESET momentarily and then released to ON. It is not necessary to move the switch to OFF when the No. 1 APU is shut down, because the No. 1 generator underfrequency protective relay will have tripped the generator off.

NOTE

> If either generator is operating and "on the line," most external power is automatically disconnected. Power to the battery bus, certain heaters, ready-to-launch light, liquid oxygen level probe, and stabilization of stable platform remains on.

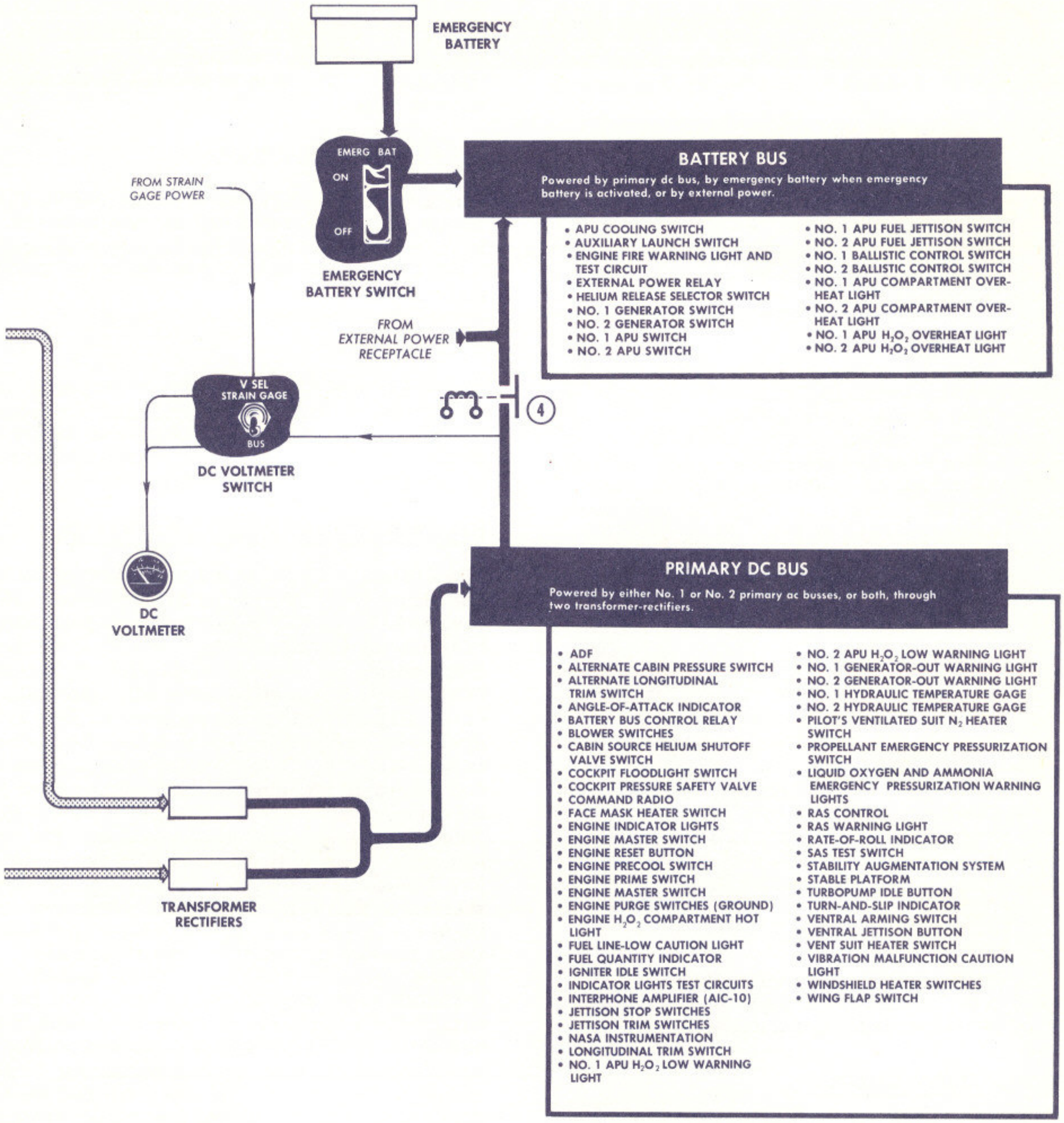
ELECTRICAL POWER DISTRIBUTION

NOTE
Nomenclature on caution lights shown illuminated for information only



X-15-1-54-1F

Figure 1-9



(1) No. 1 generator control relay. Energized open when No. 1 generator is operating.

(2) No. 2 generator control relay. Energized open when No. 2 generator is operating.

(3) External power relay. Energized closed when external power is applied and both generators are off.

(4) Battery bus control relay. Energized closed when either generator is operating.

X-15-1-54-2G

No. 2 Generator Switch.

Switch positions and operation of this switch (17, figure 1-2), on the instrument panel, are identical to those for the No. 1 generator control switch, except that, obviously, the No. 2 switch controls the No. 2 generator.

Emergency Battery Switch.

This guarded, two-position switch (16, figure 1-2), on the instrument panel, controls connection of the emergency battery to the battery bus. Normally, the switch is in the guarded OFF position. Raising the guard and moving the switch to ON connects the emergency battery directly to the battery bus.

DC Voltmeter Switch.

This three-position switch (8, figure 1-5), on the center pedestal, allows strain gage or primary dc bus voltage to be monitored. When the switch is at OFF, the dc voltmeter is disconnected. Moving the switch to STRAIN GAGE connects the dc voltmeter to the strain gage power. When the switch is at BUS, the voltmeter is connected to the primary dc bus.

ELECTRICAL POWER SUPPLY SYSTEM INDICATORS.

No. 1 Generator Voltmeter.

An ac voltmeter (11, figure 1-2), on the instrument panel, shows the No. 1 generator voltage. The instrument has a range of 0 to 250 volts, graduated in increments of 20 volts. The voltmeter reads line-to-line, rather than line-to-neutral or line-to-ground. Thus, the reading under normal conditions should be 200 volts and would check availability of two phases rather than only one phase. (When both primary ac busses are energized by one generator or by external power, both generator voltmeters should indicate the same voltage).

NOTE

If either generator fails, both the No. 1 and No. 2 generator voltmeters will show the output of the remaining generator.

No. 2 Generator Voltmeter.

An ac voltmeter (19, figure 1-2), on the instrument panel, shows the No. 2 generator voltage. The instrument has a range of 0 to 250 volts, graduated in increments of 20 volts. The voltmeter reads line-to-line, rather than line-to-neutral or line-to-ground. Thus, the reading under normal conditions should be 200 volts and would check availability of two phases rather than only one phase. (When both primary ac busses are energized by one generator or by external power, both generator voltmeters should indicate the same voltage.)

NOTE

If either generator fails, both the No. 1 and No. 2 generator voltmeters will show the output of the remaining generator.

No. 1 Generator-out Light.

Whenever the No. 1 generator drops "off the line," the amber No. 1 generator-out caution light (14, figure 1-2), on the instrument panel, comes on and reads "GEN OUT." The light is powered by the 28-volt primary dc bus.

No. 2 Generator-out Light.

Whenever the No. 2 generator drops "off the line," the amber No. 2 generator-out caution light (18, figure 1-2), on the instrument panel, comes on and reads "GEN OUT." The light is powered by the 28-volt primary dc bus.

DC Voltmeter.

The dc voltmeter (7, figure 1-5), on the center pedestal, has a range of 0 to 30 volts, graduated in increments of one volt. This voltmeter will indicate dc battery bus and strain gage power voltage. The voltage reading selection is through the dc voltmeter switch.

HYDRAULIC POWER SUPPLY SYSTEMS.

The airplane has a No. 1 and a No. 2 hydraulic system. (See figure 1-10.) Airplanes AF56-6670 and -6671 also have an SAS (stability augmentation system) emergency hydraulic system. The No. 1 and No. 2 systems are independent, but operate simultaneously to supply hydraulic pressure to all hydraulically operated systems of the airplane. Fluid is supplied to each hydraulic system by a reservoir, and pressure for each system is maintained by a variable high-displacement pump and a constant low-displacement pump. Each pump is driven through a gear train from the APU. (Refer to "Auxiliary Power Units" in this section.) The hydraulic system No. 1 pump is driven by APU No. 1; the system No. 2 pump is driven by APU No. 2. The hydraulic systems supply power for operation of the aerodynamic flight control system, speed brakes, and wing flaps. Dual, tandem hydraulic actuators are used, so that failure or shutdown of one hydraulic system will still permit the other hydraulic system to operate the various units. Each hydraulic actuator is capable of holding half of the maximum design hinge moment during single-system operation, which is adequate for control and landing of the airplane. The SAS pitch-roll servo cylinders are powered by the No. 2 hydraulic system. The SAS emergency hydraulic system will provide pressure to the pitch-roll servo cylinders in case of No. 2 system failure or shutdown of APU No. 2. The stability augmentation system yaw servo cylinder is powered only by hydraulic system No. 1. The No. 1 or No. 2 hydraulic system is automatically in operation whenever its respective APU is operating.

SAS EMERGENCY HYDRAULIC SYSTEM.

The SAS emergency hydraulic system (figure 1-10), on Airplanes AF56-6670 and -6671, provides hydraulic power for the SAS pitch-roll servo cylinders in case of failure of the No. 2 hydraulic system or its related APU. The 3000 psi system consists of a hydraulic motor, powered by the No. 1 hydraulic system, which drives a variable-displacement, constant-pressure hydraulic pump. A pressure-operated selector valve directs No. 1 hydraulic system pressure to the hydraulic motor in case of No. 2 system failure. The emergency system has its own reservoir.

HYDRAULIC PRESSURE GAGE.

A dual-movement, synchro-type gage (34, figure 1-2), on the instrument panel, indicates pressure in the No. 1 and No. 2 hydraulic systems. There are two pointers, numbered 1 and 2, for indicating respective system pressure. The gage has a range of 0 to 4000 psi in increments of 100 psi. The pressure indicating system receives power from the 26-volt ac bus.

HYDRAULIC TEMPERATURE GAGES.

Both the No. 1 and No. 2 hydraulic systems have a gage (24 and 45, figure 1-2) which indicates hydraulic fluid temperature. The gages are calibrated in degrees centigrade through the range of -100° C to 300° C in increments of 20° C. The temperature indicating systems receive power from the primary dc bus.

FLIGHT CONTROL SYSTEMS.

The airplane has two control systems. The aerodynamic flight control system (figure 1-10) consists of a mechanical system including hydraulically actuated control surfaces for use at altitudes where these surfaces are effective for maneuvering the airplane. The ballistic control system (figure 1-8) is used to control the airplane attitude at altitudes where the aerodynamic surfaces are relatively ineffective. The ballistic control system uses a monopropellant which is released through rockets at high velocity to rotate the airplane about its pitch, roll, and yaw axes as required for re-entry, and to correct oscillation.

AERODYNAMIC FLIGHT CONTROL SYSTEM.

The aerodynamic flight control system incorporates hydraulically actuated yaw and pitch-roll control surfaces. The irreversible characteristics of the hydraulic system hold the control surfaces against any forces that do not originate from pilot control movement and prevent these forces from being transmitted back to the pilot controls. Thus, aerodynamic loads of any kind cannot reach the pilot through the controls. An artificial-feel system is built into the control system to simulate feel at the pilot controls. In-flight trimming in pitch is accomplished by changing the neutral (no-load) position of the artificial-feel system and repositioning the control sticks. Yaw control is provided by movable upper and ventral vertical stabilizers. The left and right horizontal stabilizers provide pitch and roll control, simultaneous operation for pitch control, and differential operation for roll control. On Airplanes AF56-6670 and -6671, an assist to the aerodynamic damping in pitch, roll, and yaw is provided by a stability augmentation system (SAS).

Flight Control Hydraulic Systems.

When both hydraulic systems fail, the aerodynamic control surfaces will remain in the position at which failure occurred. However, the surfaces may be moved in the direction in which they are driven by aerodynamic loads by repositioning of the pilot controls in the direction to streamline the surface. The pedals, center stick, and console stick are mechanically linked to the control valves on their respective actuators. Movement of a pilot control results in corresponding movement of its actuator control valve. As the actuator moves, the control valve is repositioned to a neutral position so that flow to the actuator is shut off. The pressure remaining in the actuator serves to hold the control surface in the desired position. Control cable rig tension is maintained throughout a wide temperature and deflection range by thermal expansion and contraction tension regulators.

Artificial-feel and Trim Systems.

The artificial-feel system gives a sense of control feel to the pilot under all flight conditions where the aerodynamic controls are used. Aerodynamic stick and rudder pedal forces are simulated by spring-loaded bungees in the control system. The bungees apply loads to the pilot controls in proportion to stick or pedal movement, but the resultant feel has no relation to actual air loads. A nonlinear stick-to-stabilizer displacement ratio is incorporated in the pitch control linkage to minimize sensitivity. Pitch trim is obtained by shifting the neutral "no-load" position of the feel bungee to a stick position corresponding to the desired horizontal stabilizer position. Roll and yaw trim is adjustable only on the ground to compensate for airplane asymmetrical conditions.

Horizontal Stabilizer (Roll-Pitch Control).

The horizontal stabilizer consists of two all-movable, one-piece surfaces which can be moved simultaneously, differentially, or in compound. Aerodynamic control in pitch is obtained by simultaneous displacement of the left and right stabilizer surfaces. Roll control is obtained by differential displacement of the stabilizer surfaces. Combined pitch-roll control is obtained by compound movement of the stabilizer surfaces. A series of mixer bell cranks sum pilot control and SAS inputs to the two stabilizer actuator valves to obtain the desired pitch or roll-pitch control surface displacement.

Vertical Stabilizers (Yaw Control).

Aerodynamic control in yaw mode is obtained through displacement of the upper and ventral vertical stabilizers, which are actuated simultaneously through the coupled linkage between the upper and ventral vertical stabilizer actuator valves. Pilot pedal displacement and SAS yaw inputs are transmitted by mechanical linkage and cables to the synchronized upper and ventral stabilizer control valves. Since the ventral extends below the main landing skids, it must be jettisoned before landing. Four explosive bolts and a piston containing an explosive charge are electrically fired when the ventral jettison button is depressed. If this is not done, or if the ventral fails to jettison when the button is depressed, the ventral will jettison automatically when the landing gear is lowered. For either method of jettisoning, however, the ventral arming switch must be at ARM, to arm the jettison circuits.

HYDRAULIC SYSTEMS AND

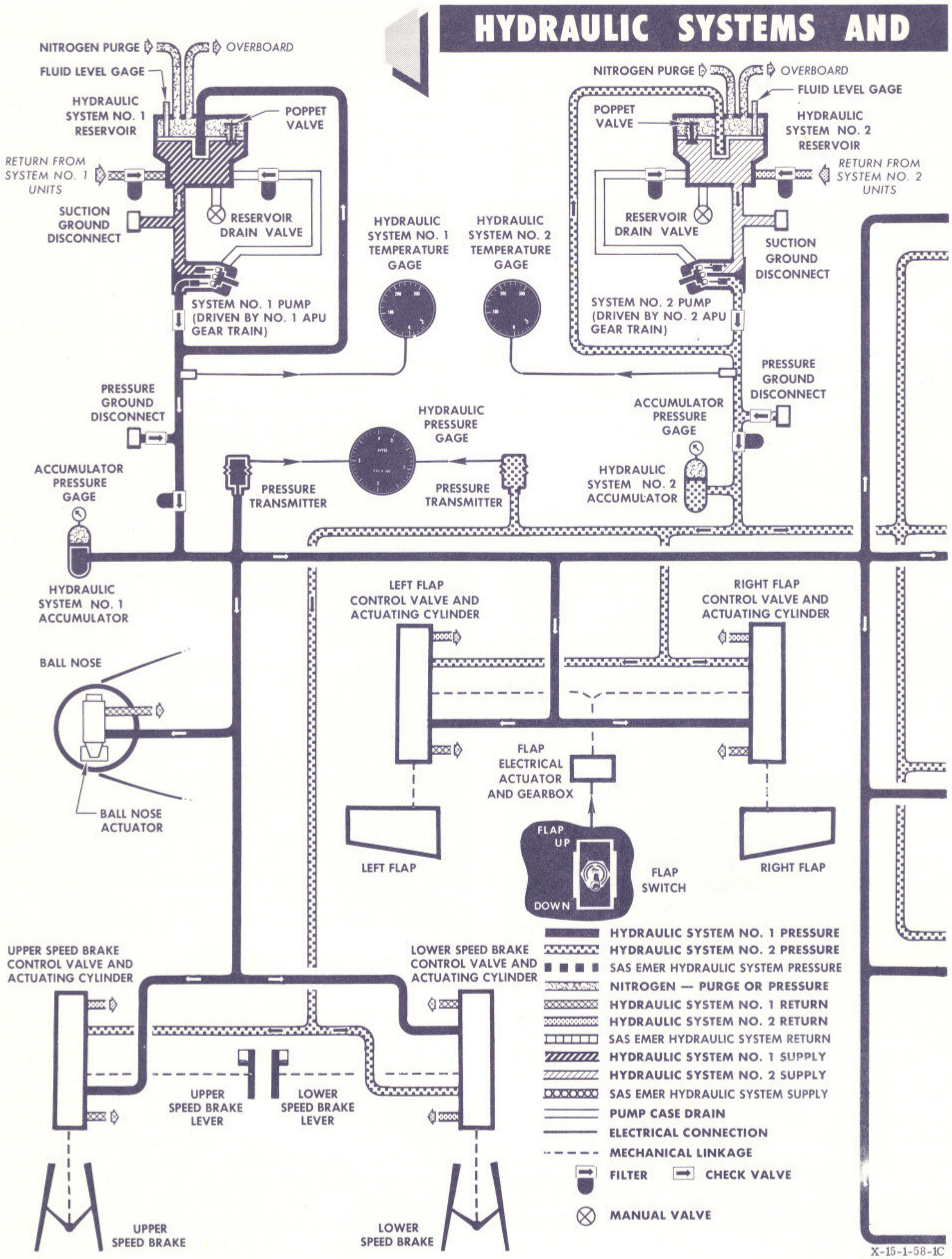


Figure 1-10. (Sheet 1 of 3)

AERODYNAMIC FLIGHT CONTROL SYSTEM

NOTE
Only Airplanes AF56-6670 and -6671 have the SAS and the SAS emergency hydraulic systems.

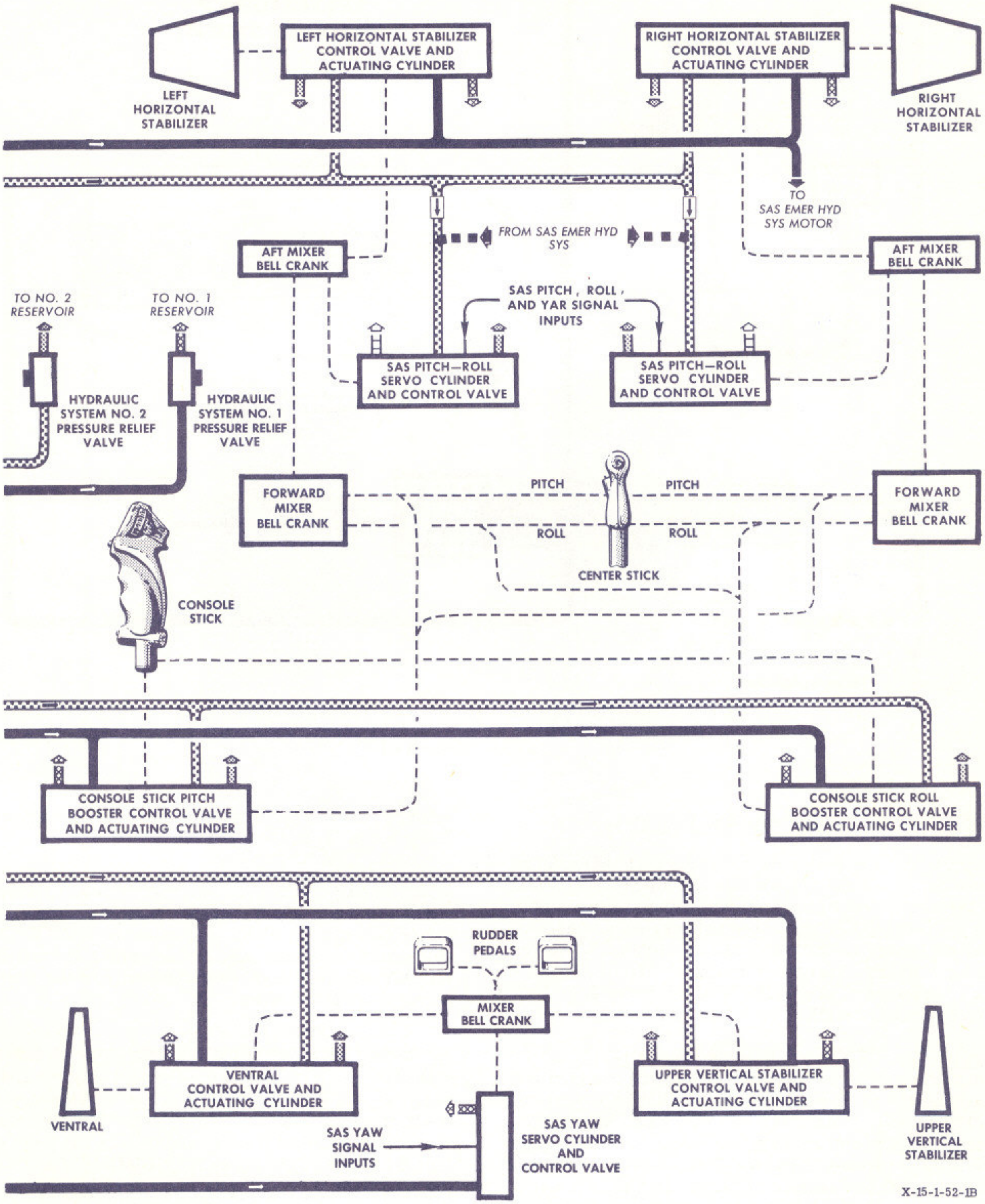


Figure 1-10. (Sheet 2 of 3)

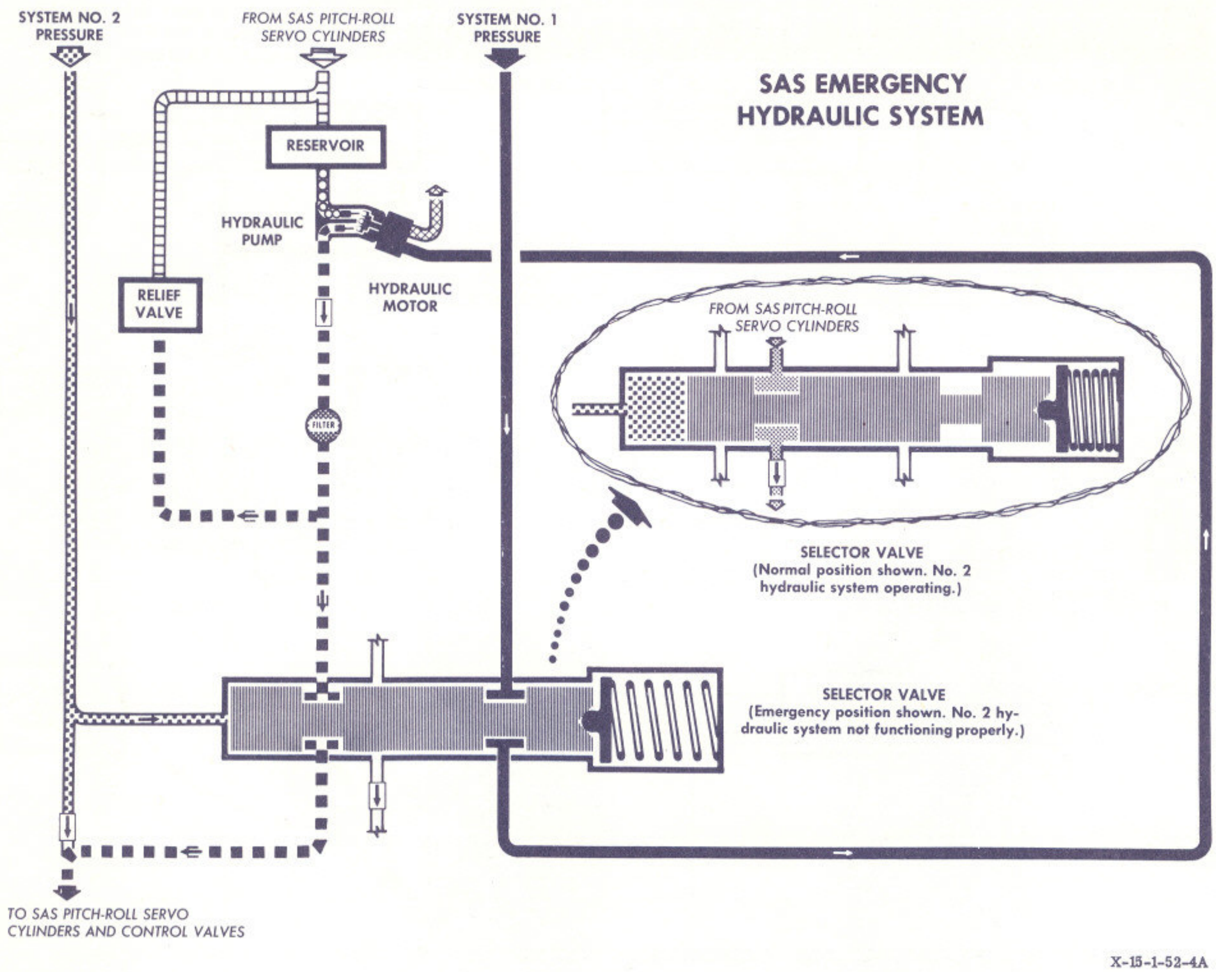


Figure 1-10. (Sheet 3 of 3)

Aerodynamic Flight Controls and Indicator.

Center Stick. The center stick (figure 1-11) is designed for use during normal periods of longitudinal and vertical acceleration. Pilot pitch and roll inputs to the center stick are summed by the mixer bell cranks and applied to the horizontal stabilizer actuator valves. A microphone button and alternate trim switch are on the stick grip. The button is in parallel with the microphone button on the console stick.

Console Stick. The console stick (figure 1-11), on the right console, enables the pilot to control the airplane throughout the periods of high longitudinal and vertical accelerations. This stick has full range of surface control in pitch and roll and is coupled to the center stick linkage through separate pitch and roll hydraulic boost actuators to reduce console stick pilot control forces and to synchronize displacement of the center and console sticks. The console stick has a pitch trim knob and a microphone button. The trim knob is graduated in degrees. When the trim control switch is in NORMAL, moving the trim knob causes corresponding movement of the pitch trim actuator. A microphone button on the console stick is in parallel with the microphone button on the center stick.

Rudder Pedals. Conventional rudder pedals, which are adjustable, are mechanically linked to the yaw system mixer bell crank. Pedal movement and SAS inputs are summed by a mixer bell crank, which in turn transmits the summed signal mechanically to the stabilizer actuator control valves.

Horizontal Stabilizer Position Indicator. A horizontal stabilizer position indicator (figure 1-11), on the right vertical side panel, provides a quick reference to the position of the horizontal stabilizer before re-entry. The forward end of the inboard face of the indicator is labeled "DIVE"; the aft end, "CLIMB." On the top of the indicator is a scale calibrated in increments of 5 degrees, from 0 to 15 degrees stabilizer leading-edge up, and from 0 to 35 degrees stabilizer leading-edge down. A red index marker is attached to the console stick shaft so that as the stick is moved, the marker will point to the corresponding horizontal stabilizer position on the scale.

FLIGHT CONTROLS AND INDICATORS

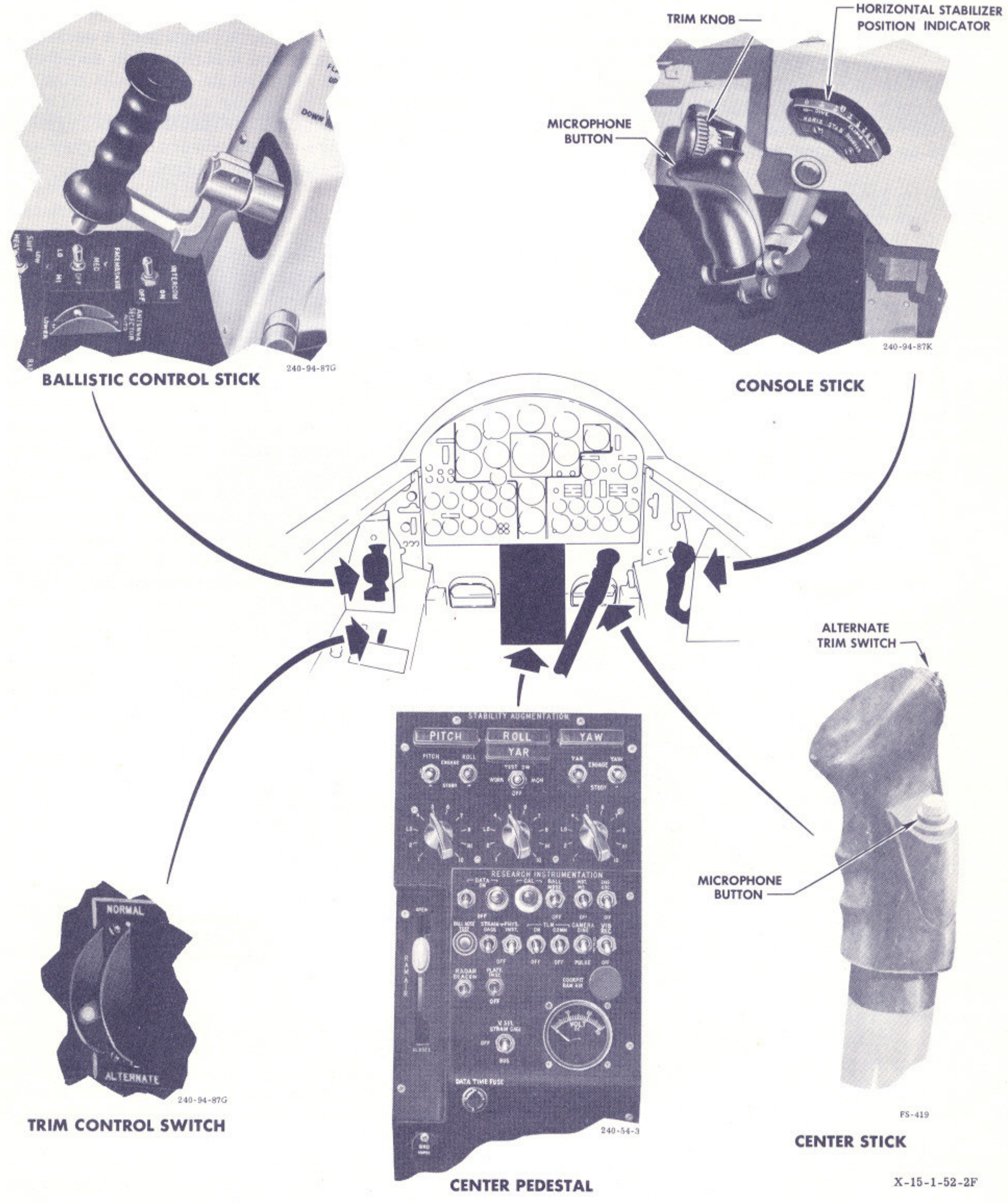


Figure 1-11.

Trim Control Switch. This two-position switch (14, figure 1-3), on the left console, controls selection of normal or alternate trim. With the switch at the maintained NORMAL position, the center stick trim control knob is inoperative and primary ac and dc power is applied to the electronic trim amplifier and relay unit to activate the console stick pitch trim knob. Movement of the pitch trim knob on the console stick then causes primary dc power to be applied to the pitch trim actuator. With the trim control switch at the maintained ALTERNATE position, output of the electronic trim amplifier is cut off, the pitch trim knob is inoperative, and pitch trim is accomplished through the alternate trim switch on the center control stick.

Alternate Trim Switch. This three-position switch on the center stick grip (figure 1-11) controls primary dc bus power to the pitch trim actuator. The switch is spring-loaded to the center position, and all positions are unmarked. With the switch in the center position and the trim control switch at NORMAL, primary dc bus power is applied to the electronic trim amplifier and relays, and pitch trim is accomplished by movement of the pitch trim knob. Moving the alternate trim switch forward (NOSE DOWN) or aft (NOSE UP) bypasses the electronic trim amplifier and applied primary dc bus power directly to the pitch trim actuator, causing the actuator to move for nose down or nose up trim, as selected.

CAUTION

If the trim control switch is moved from ALTERNATE to NORMAL, the pitch trim actuator will run to the preset position of the pitch trim knob.

Ventral Arming Switch. This guarded, two-position switch (28, figure 1-2) is on the instrument panel right-hand wing and is labeled "VENTRAL JETT." With the switch at DE-ARM, the circuit which controls primary dc bus power to the explosive bolts and explosive charge in the ventral jettison piston is interrupted and the ventral cannot be jettisoned either selectively through the ventral jettison button or automatically when the landing gear is lowered. With the switch at ARM, the ventral jettison circuit is armed.

Ventral Jettison Button. This jettison button (68, figure 1-2), labeled "VENTRAL JETT," is recessed within a circular guard on the instrument panel left wing. The button is powered by the primary dc bus and is the primary means of jettisoning the ventral. When the button is depressed, with the ventral arming switch at ARM, four explosive-type bolts and a piston containing an explosive cartridge are fired. At the instant the bolts separate, the piston is driven downward to forcibly jettison the ventral.

Hydraulic Pressure Gage. Refer to "Hydraulic Power Supply Systems" in this section.

STABILITY AUGMENTATION SYSTEM (SAS).

The stability augmentation system (SAS), installed on Airplane AF56-6670 and -6671 provides damping inputs to the aerodynamic flight control system about the pitch, roll, and yaw axes. (See figure 1-10.) Major components of the system are a three-axis gyro, an electronic case assembly, two pitch-roll servo cylinders, one yaw servo cylinder, and a gain selector switch assembly. In-flight testing of the SAS channels can be done by use of an in-flight test unit controlled from the cockpit. The SAS has fundamentally three semi-independent channels, each comprising a working circuit, a monitor circuit, and a malfunction detector. Each working circuit received its commands from the gyro assembly in its particular axis and in turn commands associated servo cylinder displacement. The monitor circuits receive commands from the gyro assembly identical to those of the control circuits. The malfunction detectors compare the commands passing through the monitor circuits with the associated servo cylinder displacement. If a predetermined error between the working and monitor circuits is exceeded, the malfunction detector locks out the particular channel in which the error occurred. The working circuit command to the associated servo cylinders is an electrical signal which drives an electro-hydraulic transfer valve on the servo cylinder. The transfer valve controls hydraulic pressure on each side of the servo cylinder piston. The pitch-roll servo cylinders, powered by the No. 2 hydraulic system, are mechanically linked to the horizontal stabilizer control linkage by mixer bell cranks and move surfaces by simultaneous movement of the bell cranks. Roll is achieved by differential movement of the bell cranks. In addition, loss of No. 2 hydraulic system pressure will automatically engage the SAS emergency hydraulic system. Should an abrupt loss of No. 2 system pressure occur, the pitch and roll damping channels may trip. If this occurs, they must be reset. The yaw servo cylinder, powered by the No. 1 hydraulic system, is mechanically linked to the vertical stabilizer control linkage through a mixer bell crank and moves surfaces by movement of the bell crank. An interaction of the yaw and toll damping working circuits is provided whereby signals from the yaw axis of the gyro are fed into the roll circuit to augment roll damping. This is referred to as the "yar" function. The SAS is powered from the No. 2 primary ac bus and the primary dc bus and is in ready status continuously as long as these busses are powered.

CAUTION

- The 28-volt dc SAS circuit breaker must be closed (pushed in) before the airplane is fueled, and must be kept closed until the end of the flight, so that the SAS gyro heaters will be energized to prevent freezing damage to the gyro
- Airplane AF56-6672 has an MH-96 adaptive flight control system installed for test purposes, in place of the SAS and reaction augmentation system. For information on the MH-96 system, refer to MH Aero Report No. 2373-TM1, Volumes I through VIII.

Stability Augmentation System Controls and Indicators.

All stability augmentation system controls and indicators are on the SAS control panel on the center pedestal.

Pitch Function Switch. This two-position switch, (17, figure 1-5), labeled "PITCH," controls the SAS pitch channel circuitry. Both switch positions are maintained. With the switch at STDBY, the pitch channel is functioning, but the input signals to the servo cylinder control valves and hydraulic pressure to the servo cylinders are shut off. With the switch at ENGAGE, pitch damping signals are applied to the pitch-roll servo cylinder control valves, which in turn permits hydraulic power to be applied to the servo cylinders. The switch is powered by the primary dc bus.

NOTE

If both the pitch and roll function switches are at STDBY, the pitch-roll servo cylinders are centered and locked.

Roll Function Switch. This two-position switch, (16, figure 1-5), labeled "ROLL," is powered by the primary dc bus. It controls the SAS roll channel circuitry. With the switch at STDBY, the roll channel is operating, but the input signals to the servo cylinder control valves and hydraulic pressure to the servo cylinders are shut off. With the switch at ENGAGE, roll damping signals are applied to the pitch-roll servo cylinder control valves, which in turn permits hydraulic power to be applied to the servo cylinders. Both switch positions are maintained.

NOTE

If both the pitch and roll function switches are at STDBY, the pitch-roll servo cylinders are centered and locked.

Yaw Function Switch. The yaw function switch (3, figure 1-5), labeled "YAW," has two maintained positions. The switch is powered by the primary dc bus. With the switch at STDBY, the SAS yaw channel is operating, but the input signals to the servo cylinder control valve and hydraulic pressure to the servo cylinder are shut off and the yaw servo cylinder is centered and locked. With the switch at ENGAGE, yaw damping signals are applied to the yaw servo cylinder control valve, which in turn permits hydraulic power to be applied to the servo cylinder.

NOTE

Shutoff or failure of hydraulic system No. 1 automatically causes the yaw servo cylinder to recenter and lock.

"Yar" Function Switch. This two-position switch, (2, figure 1-5), labeled "YAR," is powered by the primary dc bus. The switch controls the yaw signal input to the SAS roll control circuit. With the switch at STDBY, the "yar" signal circuit is inoperative. With the switch at ENGAGE while the roll function switch is at ENGAGE, yaw signals are applied to the SAS roll control circuit. Both switch positions are maintained.

NOTE

If the roll function switch is at STDBY or the roll control circuit does not function properly, yaw input to the SAS roll control circuit will neutralize roll and result in a "no roll" output.

Gain Selector Knobs. Three gain selector knobs (4, 13, and 14, figure 1-5) control, through selection of fixed resistors, the ratio of pitch, roll, and yaw damping signal to servo cylinder displacement. There are 10 switch positions; the third position is designated as LO, and the ninth is designated as HI. Pitch gain is controlled by the left knob, roll gain by the center knob, and yaw gain by the right knob. For a given gyro signal, servo cylinder displacement is lowest with a knob at position 1, and the highest at position 10.

NOTE

Gain selector knob positions to be used in flight depend on damping requirements the pilot considers necessary. However, the initial setting depends on the particular flight conditions.

SAS Caution Lights. Four placard-type amber caution lights (1, figure 1-5) indicate operating status of the SAS control circuits. The lights are powered by the primary dc bus and can be tested through the indicator, caution, and warning light test circuit. There is one light for each of the pitch, roll, yaw, and "yar" channels. When the lights are on, they read "PITCH," "ROLL," "YAW," and "YAR," respectively. The pitch, roll, and yaw lights are on when the pitch, roll, and yaw function switches are at STDBY. When any one of these switches is at ENGAGE and its control circuit is operating normally, the associated caution light is out. When the circuit error exceeds the predetermined limit, the associated caution light blinks at approximately a 4-cycle-per-second rate, and the affected channel is automatically disabled. If the error returns to within limits, the function switch must be moved to STDBY to reset the channel and then back to ENGAGE in order to restore the channel to operation. If the error is still out of limits, the function switch may be returned to STDBY and the caution light will be constantly on. The "YAR" caution light shows only the "yar" function switch position. When the switch is at STDBY, the light is on. When the switch is at ENGAGE and the "yar" signal is available to the roll channels, the light is out. Any error in the "yar" circuit exceeding the established limits is monitored by the roll channel malfunction detector and causes the roll caution light to come on.

SAS Test Switch. The three-position SAS test switch (15, figure 1-5) is on the center pedestal. The switch is powered by the primary dc bus. It has momentary WORK and MON positions and a spring-loaded OFF position. While the pitch, roll, and yaw damping channels are engaged, singly or in combination, their working and monitor circuits can be tested during captive or free flight. Placing the switch at either WORK or

MON opens the associated rate gyro ground circuits (working or monitor) and inserts a calibrated test voltage in series with the pick-offs. This voltage is added to the normal output of the gyro and, if the gain selector knobs are set to prespecified positions, the test voltage unbalances the SAS channels beyond the expected trip level of the SAS malfunction detectors. If the SAS is functioning properly, the SAS caution lights for the channels being tested will blink, signifying malfunction circuit operation. After each set of circuits (working or monitoring) is tested, the SAS channels must be re-engaged.

NOTE

The "yar" function switch must be at STDBY during the tests.

BALLISTIC CONTROL SYSTEM.

The ballistic control system is used to control airplane attitude at flight altitudes where aerodynamic flight controls are ineffective. Ballistic control is provided by two independent systems which normally are operated simultaneously. Each system uses a monopropellant which is converted by catalytic action to superheated steam and oxygen and is released through small rockets in the nose section and wings. Figure 1-8 illustrates schematically the operation of one system of the ballistic controls. Operation of the other system is identical. The reaction of the escaping gas causes the airplane to move about the selected axis or combination of axes. The monopropellant, hydrogen peroxide, is supplied from the APU and ballistic control propellant feed system. (Refer to "APU and Ballistic Control Propellant Feed Systems" in this section.) The hydrogen peroxide tank which supplies the No. 1 APU also supplies the No. 1 system of the ballistic controls. The tank which supplies the No. 2 APU also supplies the No. 2 system of the ballistic controls. Movement of the ballistic control stick in the cockpit opens a metering valve, allowing the monopropellant to enter the selected rockets. In the rockets, the hydrogen peroxide enters catalyst chambers, where it is decomposed into a high-pressure gas mixture of superheated steam and oxygen. The gases then exhaust through the nozzles. The reaction of the escaping gases causes the airplane to move about the selected axis in a direction opposite to that of the escaping gases. There are six rockets in each system. One system includes four rockets in the nose and the two left wing rockets. The other system includes the remaining four rockets in the nose and the two right wing rockets. A dual metering valve controls flow of the monopropellant to the eight nose rockets for pitch and yaw, and a dual metering valve controls flow of the monopropellant to the four wing rockets for roll. Flow of the monopropellant to the metering valves is controlled by two switches in the cockpit, one for each system. With both systems operating, a nose-down selection from the cockpit causes operation of the two rockets in the top of the nose section. A nose-right selection from the cockpit causes operation of the two rockets in the left side of the nose. A right-roll selection from the cockpit causes operation of the rocket whose nozzle is in the bottom of the left wing and the rocket whose nozzle is in the top of the right wing. Stick force gradients are maintained for all three axes of operation by spring bungees. For the pitch and yaw axes, an increase in force versus deflection rate of the ballistic control stick marks half of the maximum control travel and half of the maximum force. Thus, the pilot feels the mid-point of maximum opening of the metering valve. The acceleration and velocity of airplane movement about an axis vary with the amount and duration of ballistic control stick application. The velocity tends to sustain itself after the stick is returned to the neutral position. A subsequent stick movement opposite to the initial one is required to cancel the original attitude change. The No. 1 ballistic control system rockets also are used by the reaction augmentation system, which is installed on Airplanes AF56-6670 and -6671. Refer to "Reaction Augmentation System (RAS)" in this section. If either ballistic control system fails, the other system provides adequate power to control the airplane. A transitional altitude band exists wherein it will be necessary to use the ballistic control system and the aerodynamic flight controls simulianeously for maneuvering and controlling airplane attitude. The size of this transitional band is somewhat affected by airplane speed and the amount of maneuvering or attitude change required.

Ballistic Control System Controls.

Ballistic Control Switches. The No. 1 and No. 2 ballistic control switches (36, figure 1-2), on the instrument panel, are powered by the primary dc bus. Moving either switch to ON simultaneously opens the helium shutoff valve and the propellant feed system jettison and ballistic control valve for the respective system and allows hydrogen peroxide to flow to the metering valves. With either switch at OFF, the propellant feed system jettison and ballistic control valve for the respective system is turned off. However, the helium shutoff valve will not close as long as either APU switch is at ON.

NOTE

When either APU switch is at JETT, the feed system jettison and ballistic control valve for the related system of the ballistic controls closes and shuts down that system.

Ballistic Control Stick. The ballistic control stick (figure 1-11), above the left console, is mechanically connected to the system metering valves. When the ballistic control system is operating, moving the stick positions the metering valves to allow flow of hydrogen peroxide to the selected rockets in the nose and wings. Yaw control is obtained by direct left or right movement of the stick. For left yaw, the stick must be moved directly to the left. Roll control is obtained by rotation of the stick. For left roll, the stick must be rotated to the left (counterclockwise). Pitch control is obtained by direct up or down movement of the stick. An airplane nose-up pitch change is obtained by raising the stick.

Nose Ballistic Rocket Heater Switch. The nose ballistic rocket heater switch (27, figure 1-2), on the instrument panel right-hand wing, receives electrical power from the carrier airplane. Moving this switch

REACTION AUGMENTATION SYSTEM CONTROLS

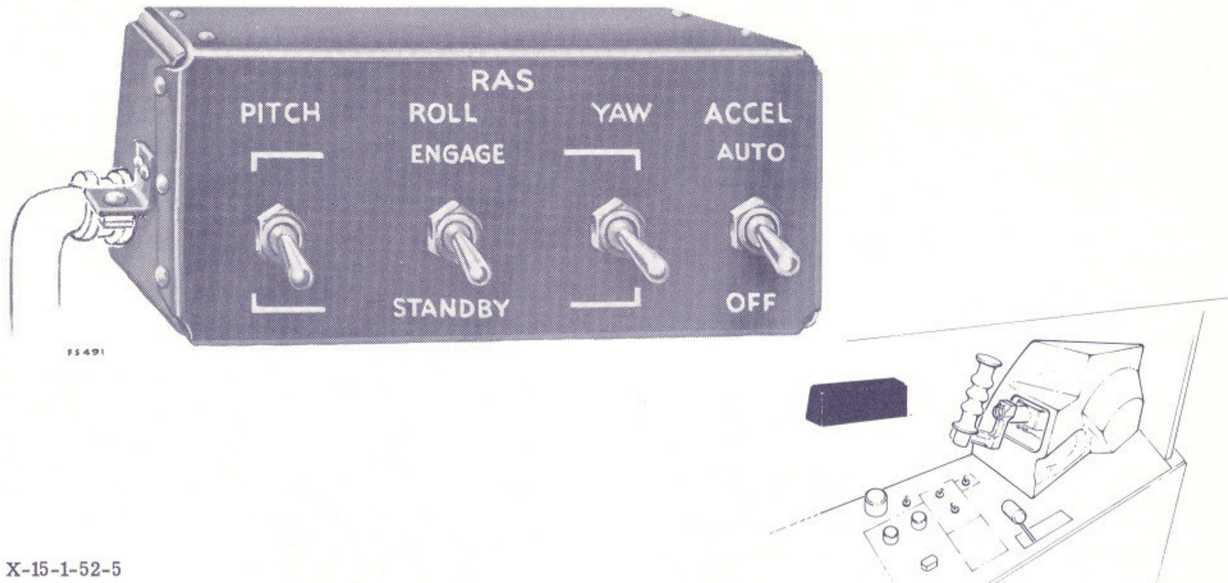


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Figure 1-12.

to ON during captive flight starts the electric heaters on the ballistic control system rockets in the nose of the X-15 to preheat the rockets. After launch, this switch is inoperative.

REACTION AUGMENTATION SYSTEM (RAS).

The reaction augmentation system (RAS) installed on Airplanes AF56-6670 and -6671, provides rate damping to aid airplane control and minimize pilot overcontrol when the ballistic control system is used. The RAS conserves APU and BCS H_2O_2 supply by minimizing pilot overcontrol tendencies at low acceleration levels. The RAS is incorporated in the No. 1 ballistic control system and uses the rocket motors associated with that system. Pilot control of both system No. 1 and No. 2 rocket motors is still maintained. Whenever the ballistic control stick is in neutral, the RAS gyro accelerometer assembly senses angular rate about the three axes and at a preset angular rate, a switching system is energized to fire the proper No. 1 system rocket motor to reduce the sensed angular acceleration. The RAS is automatically turned off by means of an acceleration-sensing device upon a build-up of G, which corresponds to an increase in the effectiveness of the aerodynamic controls. The RAS can be overridden by the pilot in case of a failure. The RAS is in a stand-by condition whenever power is applied to the airplane and is engaged by switches in the cockpit. Any combination of the three axes can be selected individually as desired. An indicator light in the cockpit shows when the RAS is in stand-by. Electrical power for the RAS is obtained from the No. 1 primary ac bus and primary dc bus. The system is protected by circuit breakers in the equipment bay.

Reaction Augmentation System Controls and Indicator.

All reaction augmentation system controls are on the RAS control panel (figure 1-12) above the left console aft of the ballistic control stick.

Accelerometer Switch. The two-position accelerometer switch on the RAS control panel is labeled "ACCEL." Moving the switch to AUTO engages the automatic cutoff. With the switch at AUTO, the automatic cutoff of the RAS will occur when a preset level of normal acceleration is sensed. When automatic cutoff occurs, an indicator light comes on. To reset the automatic cutoff feature, the switch is moved to OFF, then returned to AUTO. The OFF position of the switch disengages only the automatic cutoff feature.

Pitch Switch. The two-position pitch switch on the RAS control panel has ENGAGE and STANDBY positions. Moving the switch to ENGAGE closes the pitch circuit and permits operation of the proper pitch rocket motor by the RAS. With the switch at STANDBY, the pitch output circuit is opened.

Roll-Switch. The two-position roll switch on the RAS control panel has ENGAGE and STANDBY positions. Moving the switch to ENGAGE closes the roll circuit and permits the operation of the proper roll rocket motor by the RAS. With the switch at STANDBY, the roll output circuit is opened.

Yaw Switch. The two-position yaw switch on the RAS control panel has ENGAGE and STANDBY positions. Moving the switch to ENGAGE closes the yaw circuit and permits operation of the proper yaw rocket motor by the RAS. With the switch at STANDBY, the yaw output circuit is opened.

RAS-out Indicator Light. The RAS-out indicator light (12, figure 1-2), on the instrument panel, receives primary dc bus from the gyro accelerometer assembly and comes on to show "RAS OUT" whenever automatic cutoff occurs or if the control switches of all three axes are at STANDBY. Moving any one of the three switches to ENGAGE turns off the light if the automatic cutoff circuit has not been tripped. If the light is on because of automatic cutoff, the automatic cutoff circuitry can be reset by moving the accelerometer switch to OFF, then to AUTO, to put out the light. The light can be tested by means of the indicator, caution, and warning light switch.

MH-96 ADAPTIVE FLIGHT CONTROL SYSTEM.

The MH-96 adaptive flight control system is installed for test purposes on Airplane AF56-6672. For information on description and operation of this system, refer to MH Aero Report 2373-TM1, Volumes I through VIII.

WING FLAP SYSTEM.

Flap position is controlled by an electromechanical actuator, containing two electric motors that are coupled together by a set of differential gears. The output of this actuator drives a push-pull cable system which opens the valves in the dual, tandem hydraulic flap actuators and hydraulically positions the flaps to either full up or full down. Flap extension is possible even if one motor fails; however, the extension time is approximately twice the normal time. Normal extension from full up to full down position requires about 8 to 10 seconds. No provisions have been made for automatic pitch correction with flap extension, nor for intermediate positioning of the flaps. However, because of the incorporation of hydraulic relief valves to limit the maximum air load on the surfaces, the flaps may partially close at speeds above 250 knots. No flap position indicator is provided.

WING FLAP SWITCH.

A two-position switch (8, figure 1-3) is on the left vertical side panel. The switch, labeled "FLAP," controls flap operation, and is powered by the primary dc bus. It has two positions, UP and DOWN. No intermediate positioning is provided.

SPEED BRAKE SYSTEM.

The airplane has two speed brakes, one on the fixed portion of the upper vertical stabilizer and the other on the fixed lower vertical stabilizer. Each speed brake consists of two symmetrical panels, hinged at the forward end. Each speed brake is operated by a dual, tandem hydraulic actuator. One segment of an actuator is powered from hydraulic system No. 1; the other segment, by hydraulic system No. 2. Failure of one hydraulic system still permits operation of the speed brakes, however, at reduced rate under any particular air load. The actuator valves are controlled from the cockpit by a system of cables and mechanical linkage. Follow-up mechanisms permit positioning each speed brake to any position between fully closed and fully open.

NOTE

Each speed brake actuator incorporates a relief valve which prevents the speed brake from extending or allows the speed brake to retract under excessive air loads, to prevent structural damage.

WARNING

The speed brakes on this airplane were not designed for use as a low-speed drag device; opening them at subsonic speeds can be dangerous. Their design function is to provide necessary drag conditions for control of the airplane at supersonic speeds and relatively high altitudes.

SPEED BRAKE HANDLES.

The speed brake handles (11 and 12, figure 1-3) are on the left console. The inboard handle controls the lower speed brake; the outboard handle controls the upper speed brake. The handles normally are locked together by an interconnecting bolt at the forward cable sectors to ensure symmetrical operation of the speed brakes. A spring-loaded lock lever on the inboard handle is designed to unlock the handles for independent speed brake operation and lock them for symmetrical operation if the interconnecting bolt is not installed. Speed brake position is indicated by a scale for each handle on the speed brake handle quadrant. The scales are calibrated in increments of 5 degrees. When the handles are moved to a given setting, the speed brakes will open to the position selected.

LAUNCH SYSTEM.

The dropping of this airplane from the carrier airplane is normally performed by the pilot of the carrier airplane. However, if this cannot be done, the captive airplane is equipped with an auxiliary launch switch to perform this function from a separate power source. A switch in the captive airplane controls an indicator light in the carrier airplane to indicate that the pilot of the captive airplane is ready for launch.

READY-TO-LAUNCH SWITCH.

The ready-to-launch switch (15, figure 1-3), on the left console, is moved forward (ON) to light the "READY TO LAUNCH" indicator light in the carrier airplane when the captive airplane is ready to be dropped. There is no indicator light in the captive airplane. Moving the ready-to-launch switch aft (OFF) turns out the indicator light. This circuit is powered by the carrier airplane's electrical system.

AUXILIARY LAUNCH SWITCH.

The guarded, battery-bus-powered auxiliary launch switch (66, figure 1-2), on the instrument panel left wing, uses primary dc bus power to operate the hydraulic launch system in the carrier airplane. Lifting the guard and moving the switch up (ON) supplies power directly to the solenoid valve in the normal launch hydraulic system.

LANDING GEAR SYSTEM.

The two main landing gears are of the skid type and lie adjacent to the lower aft fuselage and parallel to the airplane centerline when retracted. The skids (27, figure 1-1) are mounted on inflexible struts with an air-oil shock absorber attached to the upper end which permits some outward rotation when the weight of the airplane is on the landing gear. The nose gear (3, figure 1-1) is a conventional, nonsteerable, dual-wheel type and retracts forward, fairing into the fuselage nose section. Both the main and nose gear, when unlocked, extend by gravity and air loads. However, the nose gear lowering system includes an initiator to ensure positive nose gear lowering. They cannot be retracted by the pilot. No gear-down indication is provided. Gear retraction must be accomplished manually by ground personnel. When the main gear is released, a microswitch on the left main gear activates an explosive charge, causing the ventral to jettison, provided the ventral arming switch is at ARM and if the ventral had not been previously jettisoned by use of the ventral jettison button.

LANDING GEAR HANDLE.

The T-type landing gear handle (67, figure 1-2) is on the instrument panel left wing. The handle is mechanically linked to the main gear uplocks and the nose gear and nose gear door uplocks. When the handle is pulled straight aft approximately 11 inches, the uplocks are released, the spring-loaded scoop door in the nose gear door swings downward into the airstream, and the nose gear extension initiator fires. Gravity and air loads cause the main gear to extend and lock. Air loads on the nose gear scoop door force the nose gear down and locked. (The initiator actuates a piston which forces the nose gear door open under flight attitudes where air loads tend to hold the door closed.)

CAUTION

The landing gear handle should be manually stowed to prevent possible damage to the instrument panel.

INSTRUMENTS.

Most of the instruments are powered by the ac or dc electrical systems or a combination of both.

NOTE

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

PITOT-STATIC SYSTEM.

Pitot pressure for the conventional airspeed indicator and altimeter is supplied by the fuselage-mounted pitot head. Static pressure is supplied by ports on each side of the fuselage forward of the cockpit area. The ball nose (1, figure 1-1) is a sphere-shaped, pitot-pressure, flow-direction sensor. The ball simultaneously measures angle of attack and angle of sideslip through two complete and independent servo systems. One system controls the vertical and one the horizontal axis. As the airplane encounters a sideslip condition or change in angle of attack, the respective system will turn the ball (electrohydraulically) into the relative wind. The difference between airplane heading and relative wind is then transmitted through sensors to cockpit instruments to read angle of attack and angle of sideslip. Electrically, the ball nose is powered through the instrumentation transformer, which in turn is powered by the No. 2 primary ac bus. Hydraulic power for the ball nose is provided from the No. 1 hydraulic system. If hydraulic system No. 1 fails, the ball nose will not turn in response to sensor signals. In this event, the angle-of-attack and sideslip indicators, although inoperative, will provide erroneous indications, as they will continue to register the conditions at the time of the power failure. In case of electrical failure, the indicators may show a continually unsafe condition.

Ball Nose Test Button. This push-button type switch on the center pedestal (figure 1-5) is labeled "BALL NOSE TEST" and is used to test operation of the ball nose in flight. Depressing and holding the button applies an error signal to the ball nose transducers. This error signal electrically simulates a predetermined airplane sideslip angle and angle of attack, which are presented on the angle-of-attack indicator and vertical and horizontal pointers on the attitude indicator. When the button is released, the ball nose should drive rapidly to an extreme position, resulting in full-scale deflection of the angle-of-attack indicator pointer and vertical and horizontal pointers on the attitude indicator. This indication should be maintained for about 2 to 3 seconds; then the ball nose should drive rapidly without overshoot to indicate the actual airplane sideslip angle and angle of attack.

NOTE

Full-scale deflection of the angle-of-attack indicator pointer and vertical and horizontal pointers on the attitude indicator when the button is released and the subsequent return to normal readings, are positive indications of proper operation of the ball nose.

ACCELEROMETER.

A three-pointer accelerometer (5, figure 1-2), on the instrument panel, shows positive and negative G-loads.

One pointer continuously indicates acceleration forces. Two recording pointers indicate maximum positive and negative G encountered. The recording pointers may be reset by clockwise movement of the reset knob on the lower left corner of the instrument.

ALTIMETER.

The altimeter (2, figure 1-2), on the instrument panel, has standard 1000- and 100-foot pointers and a 10,000-foot pointer extending from a movable center disk to the edge of the dial, so that it cannot be obscured by the other pointers. The center disk also has a wedge-shaped cutout through which a set of warning strips appear at altitudes below 16,000 feet. This altimeter offers improved readability and gives warning when an altitude of less than 16,000 feet is entered.

AIRSPEED INDICATOR.

The airspeed indicator (3, figure 1-2), on the instrument panel, shows indicated airspeed within a range of 100 to 900 knots with a conventional-type pointer. Visible through a window on the face of the indicator is a vernier drum which has a range of 0 to 100 knots. This permits reading of airspeed to the nearest knot through a range of 0 to 1000 knots. The airspeed indicator is the primary flight instrument for indicating speed during landing.

ANGLE-OF-ATTACK INDICATOR.

A remote-type angle-of-attack indicator (4, figure 1-2), on the instrument panel, is electrically driven by power from both the No. 2 primary ac and primary dc busses. The ball nose measures the angle between the relative wind and the fuselage reference line. The attack angle so determined is then transmitted to the indicator. The indicator has a range from 10 degrees nose down to 40 degrees nose up.

RATE-OF-ROLL INDICATOR.

A rate-of-roll indicator (48, figure 1-2), on the instrument panel, is electrically powered by the primary dc bus. The rate-of-roll indicator indicates the roll rate in degrees per second for right and left roll from 0 to 200 degrees per second.

INERTIAL ALL-ATTITUDE FLIGHT DATA SYSTEM (GYRO-STABILIZED PLATFORM).

NOTE

For convenience of presentation, the inertial all-attitude flight data system henceforth is referred to as the stable platform system.

The stable platform system is essentially a navigating system designed to function over the earth within a high range area, approximately 720 miles long and 240 miles wide. Primarily, the system provides an attitude, velocity, and height record in a flight environment in which conventional flight reference instruments cease to function (that is, during prolonged operation at high altitudes and Mach numbers). The equipment in this system is operated with no angular range limitations or tumbling effects inherent in conventional gyro equipment. The system displays primary flight data to the X-15 pilot and transmits this data to recorders in the X-15 Airplane. The system is divided functionally into two groups. The first group, in the X-15, consists of the platform, computer, and flight instruments, which present attitude, velocity, and altitude indications. This equipment supplies all the required data after the X-15 is launched. The platform itself has four independent gimbals that permit unlimited maneuvers about the longitudinal, lateral, and vertical axes. It also has three gyros and three accelerometers. The second group, in the carrier airplane, is used to supply the proper initial conditions to the computer and thus align and stabilize the platform before launch. A control panel in the carrier airplane enables an operator to continuously monitor performance of the flight data system before launch, and to preset the required calibrations to the X-15 through an umbilical cord. In captive flight, the system is slaved to compass and velocity-measuring equipment in the carrier airplane. During free flight, the system dead-reckons from the launch point.

STABLE PLATFORM SYSTEM CONTROLS AND INDICATORS.

NOTE

In referring to electrical power sources in the following paragraphs, it is considered that the stable platform switch is positioned at INT and that the system controls and indicators are being powered from the X-15 Airplane.

Stable Platform Power Switch.

The stable platform power switch (26, figure 1-2) provides a means for selecting stable platform power, either from the carrier airplane or from the X-15. The switch is on the instrument panel right wing and is labeled "STABLE PLATF PWR." It has three maintained positions: INT, EXT, and OFF. The OFF position is a detent position. To move the switch from OFF, the switch must be pulled out of the detent. Moving the switch to INT energizes the platform with No. 2 primary ac bus power from the APU's of the X-15.

NOTE

Do not turn the stable platform switch to INT when the APU's are off.

When the switch is moved to EXT, the system is energized by power units in the carrier airplane.

NOTE

The switch should be moved from EXT to INT just before launch. However, if this is not done, power will be automatically transferred from the carrier airplane to the X-15 at time of launch.

Turning the switch to OFF shuts off all power to the platform system.

Stable Platform Instrument Switch.

This two-position switch on the center pedestal (figure 1-5) is labeled "STABLE PLATF INST." The switch controls power to the stable platform indicators in the cockpit. It is used during ground checks of the stable platform system to interrupt power to the instruments when their operation is not required to perform the check-out. With the switch at OFF, power to the stable platform instruments is interrupted. Except during ground checks, the switch must be at ON.

Attitude Indicator.

The attitude indicator (6, figure 1-2), on the instrument panel, is powered by the primary dc bus and the No. 2 primary ac bus. It is a pictorial-type instrument that combines displays of attitude and azimuth on a universally mounted sphere displayed as the background for a miniature reference airplane. The sphere (remotely controlled by the stable platform) is free to rotate 360 degrees in pitch, roll, and azimuth. The miniature reference airplane is always in proper physical relationship to the simulated earth, horizon, and sky areas of the background sphere. The horizon on the sphere is represented as a solid white line. On this horizon line is an azimuth scale graduated in 5-degree markings from 0 through 360 degrees. Above the horizon line, the sky is indicated by a light-gray area. Below the horizon line, the earth is indicated by a dull-black area. The sphere is marked by meridian lines spaced every 30 degrees. Pitch angle is referenced to the center dot of the fixed miniature airplane by horizontal marks spaced every 10 degrees on the meridians. A pitch-adjustment knob on the lower right side of the instrument electrically rotates the sphere to the proper position in relation to the miniature airplane to correct for pitch attitude changes. Clockwise rotation of the knob causes the horizon line to deflect upward from the airplane index. Rotating the knob counterclockwise causes the horizon line to deflect downward. Trim setting is automatically and gradually cancelled as airplane attitude approaches the vertical in climb or dive to ensure a true vertical indication. It returns automatically when level flight is resumed. Bank angles are read from a semicircular bank scale on the lower quarter of the instrument. Two long pointers project across the sphere. Movement of these pointers shows airplane displacement with respect to the air in which it is flying (small angles of attack and sideslip). The horizontal long pointer is a vernier indication of the angle-of-attack slip indicator on the instrument panel. This horizontal pointer moves upward when the angle of attack is increased and downward when the angle of attack is decreased. The vertical long pointer moves to the right to indicate a left sideslip, and to the left to indicate a right sideslip. The range of either pointer movement is adjustable (on the ground) to operate within ±5 to ±10 degrees. A short horizontal pointer is on the left side of the instrument. This pitch pointer is a vernier indication for the pitch axis of the sphere and moves in the same direction as the sphere. The pointer indicates displacement in a range of ±5 degrees of that angle selected on a pitch angle set control. Immediately after launch, the pilot rotates the airplane to the pitch angle that is preset on the pitch angle set control. Initially, the sphere is used to approximate this angle. As the airplane approaches to within 5 degrees of the preselected pitch angle, the small pitch pointer moves toward the center index (0). The pilot then switches his attention from the sphere to the pointer for fine adjustments. The pointer will remain at the zero position as long as the preset pitch angle is maintained. A turn-and-slip indicator is on the lower portion of the instrument below the bank angle scale. The rate-of-turn needle is powered by the 26-volt ac bus and the No. 2 primary ac bus. Immediately after electrical power is applied, an "OFF" flag on the lower left section of the indicator retracts. Failure of either dc or ac power causes the "OFF" flag to reappear.

Pitch-angle Set Control.

A pitch-angle set control (8, figure 1-2), powered by the primary dc bus and the No. 1 primary ac bus, is on the instrument panel, next to the attitude indicator. The control is used in conjunction with the small pitch pointer on the attitude indicator. The angle that is set on this control is the pitch angle the pilot will attain for either the climb or the re-entry phase of a mission. The instrument consists of four counters and a pitch angle set controller knob and lever. Rotating the knob clockwise sets up the desired pitch angle on three of the counters. The number on the far right counter is preceded by a dot to indicate the reading is in tenths of a degree. Counter range is from 0 to 90 to permit selection of any pitch angle up to 90 degrees. Rotating the knob counterclockwise returns the three counters to 0. The lever, adjacent to the knob, can be rotated upward or downward to change the sign (negative or positive) of the selected pitch angle. When the lever is moved upward, a minus (-) sign shows on the left counter; downward movement produces a plus (+) sign.

Azimuth Indicator.

The aximuth indicator (7, figure 1-2), on the instrument panel, displays azimuth reference with respect to the surface of the earth. Its display is presented on a movable compass card and a single pointer. The compass card is synchronized with the stable platform of the flight data system. A push-to-set synchronizer knob at the lower right side of the indicator permits adjustment of the compass card to read airplane azimuth displacement either from magnetic north or from the high range centerline. The knob may be turned to accurately synchronize the card with the stable platform during initial operation of the system. Turning the knob toward + causes the compass card to rotate clockwise. Turning the knob toward - causes the compass card to rotate counterclockwise. The pointer is synchronized with the antenna of the automatic radio direction finder (ADF) system. It reads displacement between the airplane centerline and a course to the ADF ground station. A knob labeled "SET HDG," at the lower left side of the indicator, is used to set the index marker at the periphery of the compass card to a prebriefed position. The azimuth indicator is powered by the primary dc bus and the No. 1 primary ac bus.

Inertial Height Indicator.

The inertial height indicator (10, figure 1-2), powered by the primary dc bus and No. 1 primary ac bus, is on the instrument panel. The indicator is coupled to the stable platform system computer to automatically indicate any change in height. The computer, in turn, determines the height from vertical accelerations measured by a vertical accelerometer in the stable platform. Height is measured from an arbitrary reference line. This reference line may be determined in one of several ways - pressure altitude, ground control radar, etc. The inertial height indicator has a long 10,000-foot pointer and a short 100,000-foot pointer. Before launch, the pilot should readjust the height indicator (by turning a knob labeled "SET" on the face of the instrument) to his height in relation to the arbitrary reference line to eliminate possible cumulative errors.

Inertial Speed (Velocity) Indicator.

The speed (velocity) indicator (13, figure 1-2) is a single-pointer instrument coupled to the stable platform system computer to indicate airplane trajectory velocity. The computer determines this trajectory velocity from accelerations measured by accelerometers in the stable platform. The indicator, powered by the primary dc bus and No. 1 primary ac bus, is on the instrument panel and reads in thousands of feet per second from 0 to 7.

Vertical Velocity Indicator.

This single-pointer indicator (9, figure 1-2), on the instrument panel, displays inertial ascent and descent in hundreds of feet per second from 0 to 10. The vertical velocity indicator, powered by the primary dc bus and the No. 1 primary ac bus, is coupled to the stable platform system computer. The computer determines this vertical velocity from accelerations measured by the vertical velocity accelerometer in the stable platform.

INSTRUMENTATION SYSTEM

The instrumentation system records a wide variety of data on the basic airframe and airplane systems. The system is powered from the No. 2 primary ac bus and the primary dc bus. Controls and indicators for the system are on the center pedestal. (See figure 1-5.)

INSTRUMENTATION SYSTEM CONTROLS AND INDICATORS.

Instrumentation Master Power Switch.

Turning this two-position switch to ON energizes all instrumentation heater circuits, energizes air-borne recording instrumentation dc power circuits, and arms the gyro cage switch, strain gage power switch, and data switch. Moving the switch to OFF electrically de-energizes all instrumentation equipment.

Strain Gage Power Switch.

When the two-position strain gage power switch is turned from OFF to ON, all transducers in the instrumentation system requiring strain gage battery power and that record on air-borne oscillographs, are energized.

Data Switch and Light.

This two-position switch energizes all air-borne recording media and arms the camera and calibrate switches when it is turned from OFF to ON. The switch also turns on a neon data light next to the switch. The data light blinks in synchronization with the camera timer, thus providing a positive indication of correct timer operation and energizing of recorder film magazines.

Calibrate Button and Light.

This is a spring-loaded, push-button type switch with an integral signal light. Momentarily depressing the button triggers the automatic in-flight calibration circuits used with the strain gage transducers to record the air-borne oscillographs. The signal light in the button comes on green when the button is depressed, to indicate that automatic calibration is in progress. The light continues to glow after the button is released, remaining on until the calibration is completed.

Camera Switch.

The camera switch provides a means of selecting cine or pulse operation for the recording cameras. The switch has three positions: CINE, PULSE, and OFF.

Telemeter Master Power Switch.

This two-position switch, when turned from OFF to ON, energizes all power circuits in the telemeter system.

Telemeter Commutator Motor Switch.

Moving this two-position switch from OFF to COMM energizes the commutator motor.

Engine Oscillograph Record Switch.

The engine oscillograph record switch is on the center pedestal instrumentation panel (5, figure 1-5) and is labeled "ENG OSC." With instrumentation system electrical power available (instrumentation master and data switches ON), moving the engine oscillograph record switch to ON (up) applies primary dc bus power to operate the engine oscillograph. When the switch is at OFF, the oscillograph is off and will not record system parameters. Both switch positions are maintained. Use of the switch depends on the type of research mission to be flown.

Engine Vibration Recorder Switch.

The engine vibration recorder switch, on the instrumentation panel (5, figure 1-5), is labeled "VIB REC."

Moving the switch to ON (up) turns on the engine vibration recorder. Although a vibration sensor on the engine continuously sends engine vibration signals through a signal box to the recorder, no record is made until the engine vibration recorder switch is moved to the ON position. The recorder, which has its own battery power, is shut down when the switch is moved to OFF.

Physiological Instrumentation Switch.

A physiological instrumentation switch is on the instrumentation control panel (5, figure 1-5), on the center pedestal, on modified airplanes. The switch, labeled "PHYS INST," allows the physiological instrumentation system to be turned ON (switch moved up) or OFF. The switch controls primary dc bus power to the physiological instrumentation system.

INDICATOR, CAUTION, AND WARNING LIGHT SYSTEM.

Malfunctions or operating conditions for various airplane systems are indicated by placard-type indicator, caution, and warning lights on the instrument panel and center pedestal. For information on the functions of these lights, refer to the applicable systems. Except for the engine compartment fire-warning light, all the lights can be tested by a switch in the cockpit.

INDICATOR, CAUTION, AND WARNING LIGHT SWITCH.

An indicator, caution, and warning light switch (31, figure 1-2), powered by the primary dc bus, is on the instrument panel right wing. This switch has two positions, TEST and NORMAL. When the switch is placed in the TEST position, all indicator, caution, and warning lights (except the fire-warning light) on the instrument panel and center pedestal come on; this is only a test of the bulbs. The switch must be at NORMAL for all normal operations.

CANOPY.

The one-piece, pneumatically counterbalanced, clamshell-type canopy (figure 1-13) is manually operated and mechanically locked in the down position. The canopy is hinged at the rear and opens about 45 degrees after moving aft to unlock. The canopy can be manually locked from either inside or outside. The canopy has a double-pane window with an air space between the panes for defrosting air. A retractable, adjustable head support can be lowered from the top of the canopy to restrain the pilot's head during deceleration. A canopy seal, incorporated in the rim of the canopy, contacts the canopy sill and a bulkhead at the rear of the cockpit to allow pressurization of the cockpit. A cartridge-type canopy remover is fired by an initiator when the ejection seat armrests are raised. Also, this initiator directs some of its expanding gases to extend a thruster at the forward end of the canopy. When the canopy leaves the airplane, it fires the ejection seat catapult initiator to eject the seat. The canopy can also be ejected with the canopy internal emergency jettison "T" handle which fires a separate initiator that does not arm and fire the ejection seat when the canopy leaves. A similar "T" handle is provided externally, behind a door on the right side of the fuselage, just below the canopy split line, for ground emergencies.

CANOPY SEAL.

An inflatable rubber seal, built into the edge of the canopy frame, seats against mating surfaces of the canopy sill and a bulkhead at the aft end of the cockpit to provide sealing for cockpit pressurization. The seal pressurization valve is mechanically actuated just before the complete locking of the canopy to permit gaseous nitrogen to inflate the seal. When the canopy handle is actuated to open the canopy, the seal pressurization is dumped and the nitrogen valve is closed.

CANOPY CONTROLS.

Canopy External Handle.

The canopy external handle (figure 1-13), on the right side of the fuselage, below the forward end of the canopy, is behind a flush door that is opened by pushing a flush-mounted button just aft of the door. Pulling the black long-hinged handle out of its spring clip and rotating it upward unlocks and moves the canopy aft about one inch to permit manual raising of the canopy. Rotating the handle forward and down moves the canopy forward and locks it. Pushing the handle into the spring clip stows the handle.

Canopy Internal Handle.

The canopy internal handle (1, figure 1-4), for locking or unlocking the canopy, is on the right side of the cockpit, just below the canopy sill. Locking of the canopy requires that the canopy be lowered manually until it is tight against the canopy sill, then the handle pushed forward until it is against the stops and then rotated outboard to lock. Unlocking the canopy is accomplished by pulling the handle inboard and aft all the way back against the stops until the canopy is unlocked. The canopy can then be raised manually. During the ejection sequence, the handle automatically moves aft when the canopy unlocks.

WARNING

Keep hands and arms clear of canopy internal handle when canopy is jettisoned, because the canopy handle moves aft with considerable force.

Canopy Internal Emergency Jettison Handle.

The canopy internal emergency jettison handle (25, figure 1-2), on the instrument panel right wing, jettisons the canopy without firing the seat. The "T" handle fires a separate initiator that fires the canopy remover and

CANOPY

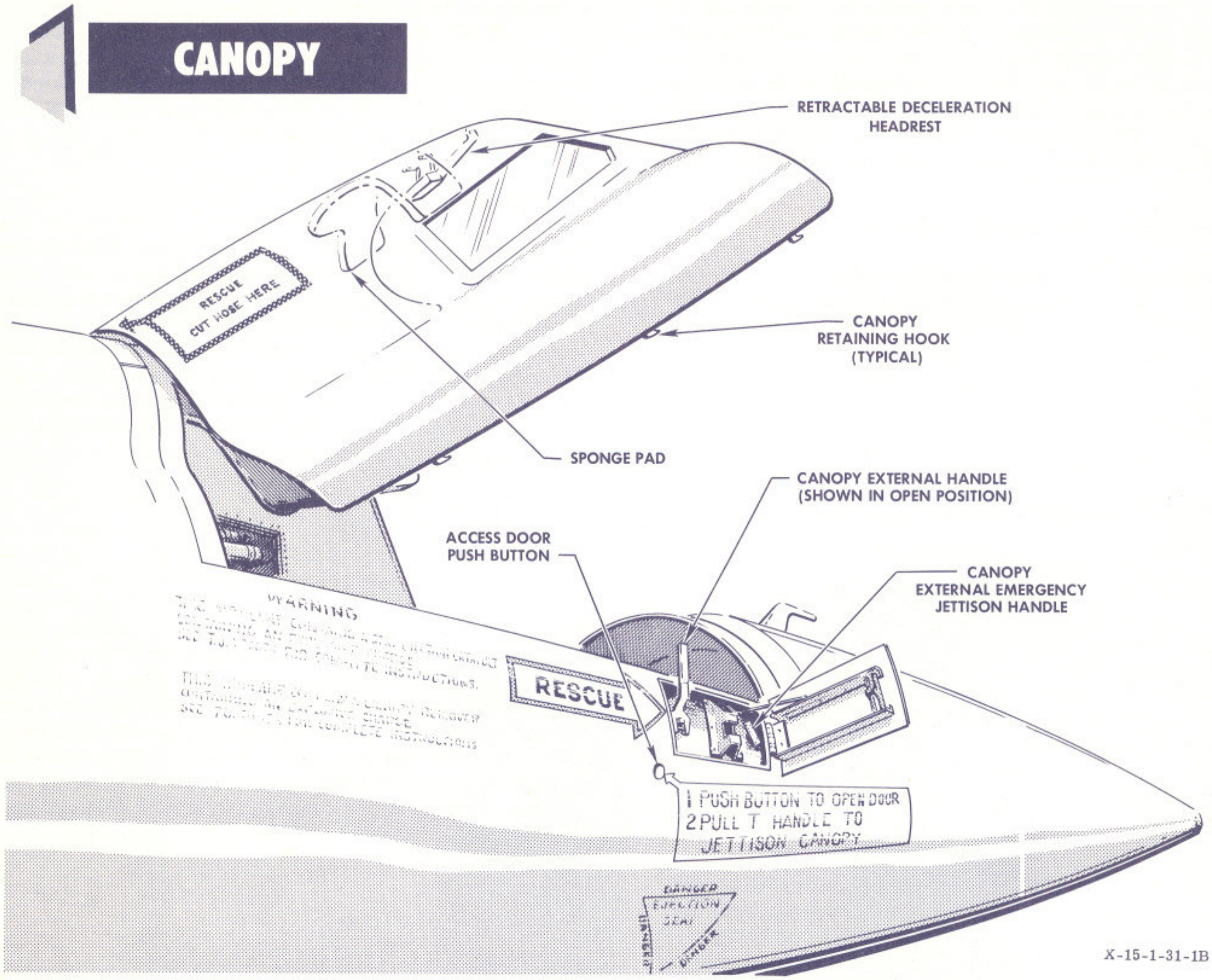


Figure 1-13

extends the canopy thruster. On the ground, the handle is safetied with a safety pin inserted through the handle.

WARNING

Keep hands and arms clear of canopy internal handle when pulling the emergency jettison handle, because the canopy internal handle moves aft with considerable force.

Canopy External Emergency Jettison Handle.

The canopy may be jettisoned from the outside by the canopy external emergency jettison handle. (See figure 1-13.) This yellow, T-type handle is just forward of the canopy external handle in a recess behind a flush door on the right side of the fuselage, below the forward end of the canopy. When the handle is pulled straight out approximately 4 inches with a force of about 10 pounds, a canopy initiator is fired. This in turn fires a canopy remover that forcibly jettisons the canopy. The seat is not armed when the canopy is removed by this means.

WARNING

- The canopy external emergency jettison handle does not have an extension lanyard. Therefore, extreme care should be taken to ensure that no part of a person's body is directly over any portion of the canopy when it is jettisoned.
- If the ejection handles are raised and the canopy has not jettisoned, the ejection seat must be deactivated before the canopy is either manually opened or jettisoned, to prevent seat ejection.

SAFETY PINS.

Because of the interdependence of the seat and canopy ejection systems, the seat and canopy safety pins are discussed together in this paragraph. To safety the canopy and seat ejection systems adequately, seven safety pins are required. The two initiators on the canopy deck, just aft of the right side of the seat, and the two initiators on the right side of the seat near the headrest are safetied by a safety pin inserted through a hole in the initiator sear pin of each initiator. The right-hand ejection handle, on the ejection seat, restraint emergency release handle on the ejection seat, and canopy internal emergency jettison handle on the instrument panel right wing are safetied with a safety pin through a hole in each handle.

WARNING

All safety pins must be removed before flight and replaced after landing.

EJECTION SEAT.

The ejection seat (figure 1-14) is designed to permit safe pilot ejection up to Mach 4.0, in any attitude, and at any altitude up to 120,000 feet. Firing of the seat is initiated by jettisoning the canopy. The seat cannot be ejected unless the canopy has left the airplane. A ballistic-rocket type catapult supplies the necessary propulsion force to eject the seat and pilot from the airplane. During ejection, stabilizing fins and booms automatically extend to stabilize the seat. Restraint devices are provided for the pilot's body and legs to prevent injuries and separation from the seat above 15,000 feet. At this altitude, an aneroid device fires three initiators to free the restraint devices and permit pilot separation from the seat. A manual handle is provided to permit the pilot to release the restraints if the aneroid device fails. If ejection below 15,000 feet is accomplished, there is a 3-second delay after ejection before automatic pilot-seat separation is initiated. The breathing oxygen supply is contained in cylinders mounted to the underside of the seat and is used when the airplane is launched from the carrier airplane. While in captive flight, breathing oxygen is supplied by the carrier airplane. An oxygen selector valve and gage on the left side of the seat permits selection of either carrier airplane oxygen or the seat-contained oxygen. The personal leads (radio, oxygen, and ventilated suit) are attached to a disconnect block that is fitted into a disconnect fitting on the left side inside the seat bucket. The ejection seat also has a quick-disconnect receptacle to plug in the pilot's physiological instrumentation wiring harness. The receptacle is on the top of the seat pan in the forward right-hand corner. The manually adjustable shoulder harness straps are fastened to the integrated parachute restraint harness with quick-disconnect fittings. During the ejection sequence, the shoulder harness is released when the headrest ejects. The pilot's parachute is carried in a container attached to the pilot's integrated harness, with a pilot chute in a separate container; the pilot chute is released when the headrest is ejected. The operation sequence of the aneroid device is actuated at 15,000 feet, and it fires the three initiators that fire the headrest and release the seat belt, personal leads, ejection handles, and all restraints, to permit pilot separation from the seat.

EJECTION SEAT CONTROLS.

Ejection Seat Ejection Handles.

Unlatching and raising either ejection handle on the seat to within 15 degrees of its full travel fires an initiator that fires the canopy remover. As the canopy leaves the airplane, it fires the seat catapult initiator that fires the two-stage seat catapult. The ejection handle release latch is in the top portion of each ejection handle and is actuated when either ejection handle is grasped. Since the ejection handle assemblies are linked together by a linkage to a torque tube, pulling up on either ejection handle automatically raises the other. The ejection handles lock in the full travel position until unlocked by the restraint release system. If the canopy is inadvertently lost in flight, the seat will not eject unless the ejection handles are raised. During pilot separation from the seat after ejection, the ejection handle assemblies are automatically unlocked and swing out board to permit unrestricted pilot separation from the seat. Modified ejection seats and pressure suits provide automatic actuation of the pilot's emergency oxygen supply when the ejection handles are raised. As the handles are raised, a cable pulls the pin from the emergency oxygen supply valve. The cable is attached in parallel with the manual actuation cable.

Restraint Emergency Release Handle.

A restraint emergency release handle, on the right side of the ejection seat, is pulled up to afford a quick release from the seat on the ground when the manual release would be too slow, or after ejection if the aneroid device fails to actuate the automatic restraint release system. Pulling this handle releases the foot restraints, lap belt, personal leads, and armrest assemblies, and fires the headrest which releases the shoulder harness.

Foot Restraint Release Buttons.

The foot restraint release buttons are on the top front corners of the seat, above each foot restraint. Depressing each button unlocks its respective foot restraint, which releases the pilot's feet. During automatic pilot-seat separation, the foot restraints unlock automatically.

SAFETY PINS.

Refer to "Safety Pins" under "Canopy" in this section.

PARACHUTE, INTEGRATED RESTRAINT EQUIPMENT, AND PRESSURE SUIT.

The personal parachute for the pilot is contained within a fiber glass parachute container that is attached to the pilot's integrated harness at the top and bottom. The

EJECTION SEAT

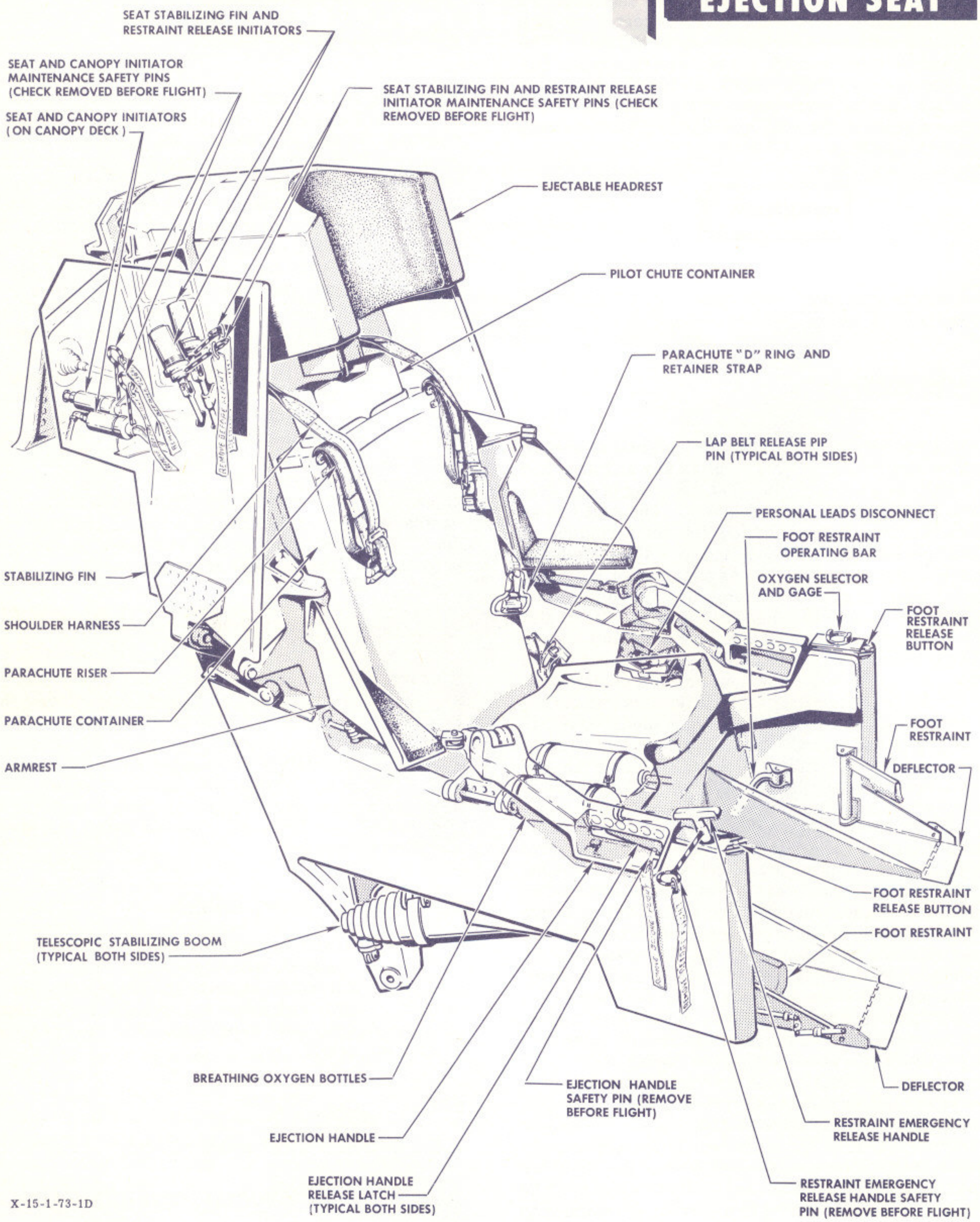


Figure 1-14

DRAIN, JETTISON, AND VENT CONFIGURATION

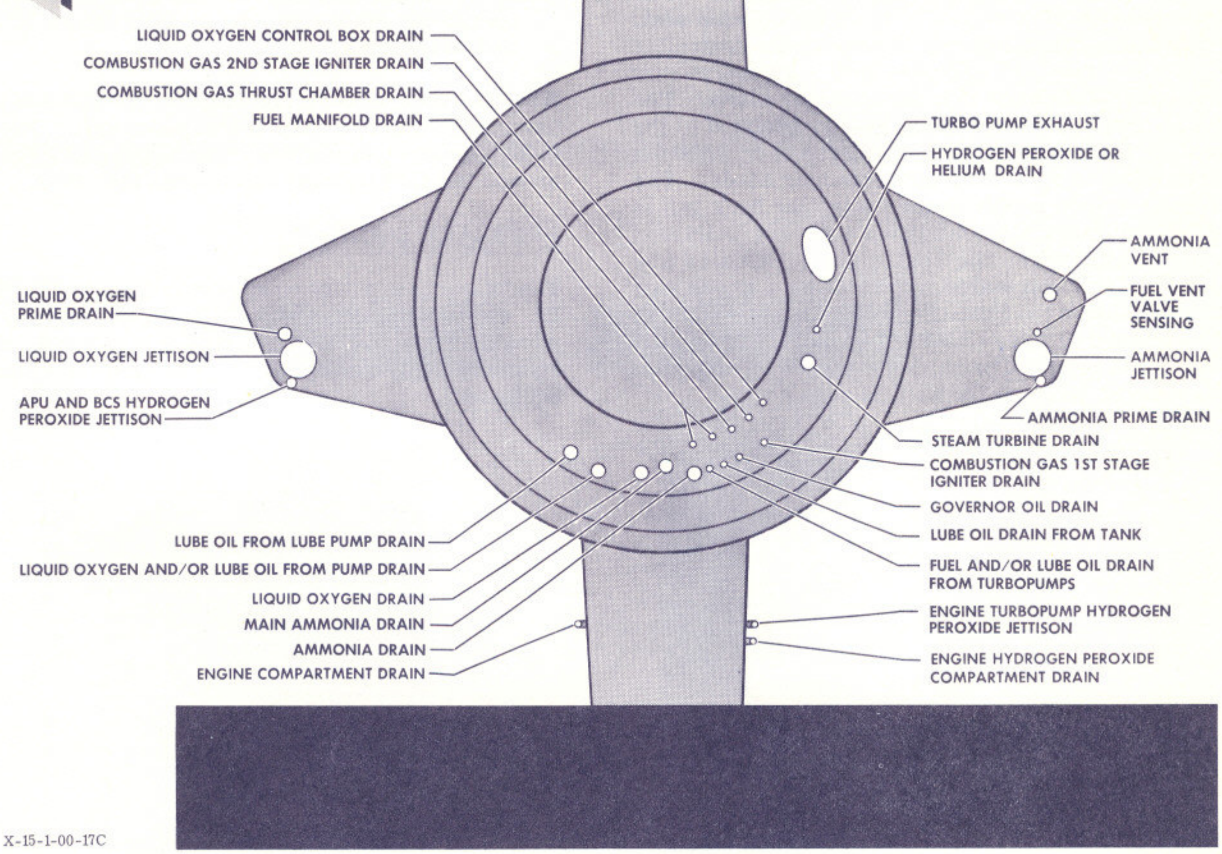


Figure 1-15

parachute may be either a 24-foot or a 28-foot type. Another fiber glass container contains the pilot chute and, along with the parachute container lids, closes the top of the parachute container. The pilot chute container is held in place by a retaining pin that locks into the manual ripcord pin. On the top of the container, a retainer pin fits into the ejection seat headrest. During ejection, the pilot chute container is locked to the headrest as the seat leaves the airplane, and is pulled off the parachute container to deploy the pilot chute when the headrest is fired. This in turn pulls out the main parachute canopy. A rescue beacon transmitter, installed in the parachute container, is automatically energized into continuous operation when the pilot's parachute deploys. The transmitter antenna is attached to one of the parachute straps. Transmission is on the X-15 telemetering frequency of 244. 3 megacycles. The transmitter permits ground stations to obtain position fixes on the pilot after an ejection. The parachute container is attached to the seat by a strap on the lower corner of each side. This strap is attached to a fitting which is held by a lap belt release on each side in the seat bucket. This release hinges on a pip pin during automatic release, but can be removed manually to permit removal of the parachute container without actuating the restraint release system. When pilot-seat separation occurs during ejection, the parachute container remains with the pilot. The parachute riser quick-release buckles also fasten the shoulder harness and parachute container to the pressure suit fittings, on the pilot's chest, just below the shoulders. The rip cord "D" ring is attached to a strap fastened to the left side of the parachute container. This strap also has a quick-release buckle which is fastened to a fitting on the left side of the integrated restraint harness, just below the arm. The lap belt portion of the integrated restraint harness consists of two straps, one over each hip, fitted with hooks that are snapped into a ring portion of the lap belt release fitting on either side of the seat. The lap belt has double adjuster straps, one through each side of the buckle. The adjuster straps must be pulled tight to keep the pilot firmly in the seat. This attaches the pilot to the seat and parachute container. The full pressure suit was modified for the X-15 Airplane and has the restraint straps and parachute harness designed as an integral part of the suit. A neck seal is used to keep the suit pressurization nitrogen and breathing oxygen separated. Attached to the back of the restraint harness is a controller back pan which incorporates an oxygen regulator, suit pressure regulator, anti-G valve, and the emergency oxygen supply for breathing and suit emergency pressurization. This emergency oxygen supply is sufficient for about 20 minutes after the pilot separates from the ejection seat. The oxygen and communication lines from the controller back pan are internal and plug into a pressure suit mating receptacle over the left shoulder blade. The personal-lead disconnect block is also attached to the controller back pan. The suit regulator ventilation exhaust valve is over the right shoulder blade. The emergency oxygen supply in the controller back pan is actuated automatically (modified ejection seats and pressure suits) when the seat ejection handles are raised, or can be actuated manually by a green ball on the right side of the suit. The helmet is free-rotating and is fastened to the suit with a snap connector ring that seals the joint. The helmet visor is locked down with a squeeze latch on the bottom edge.

WARNING

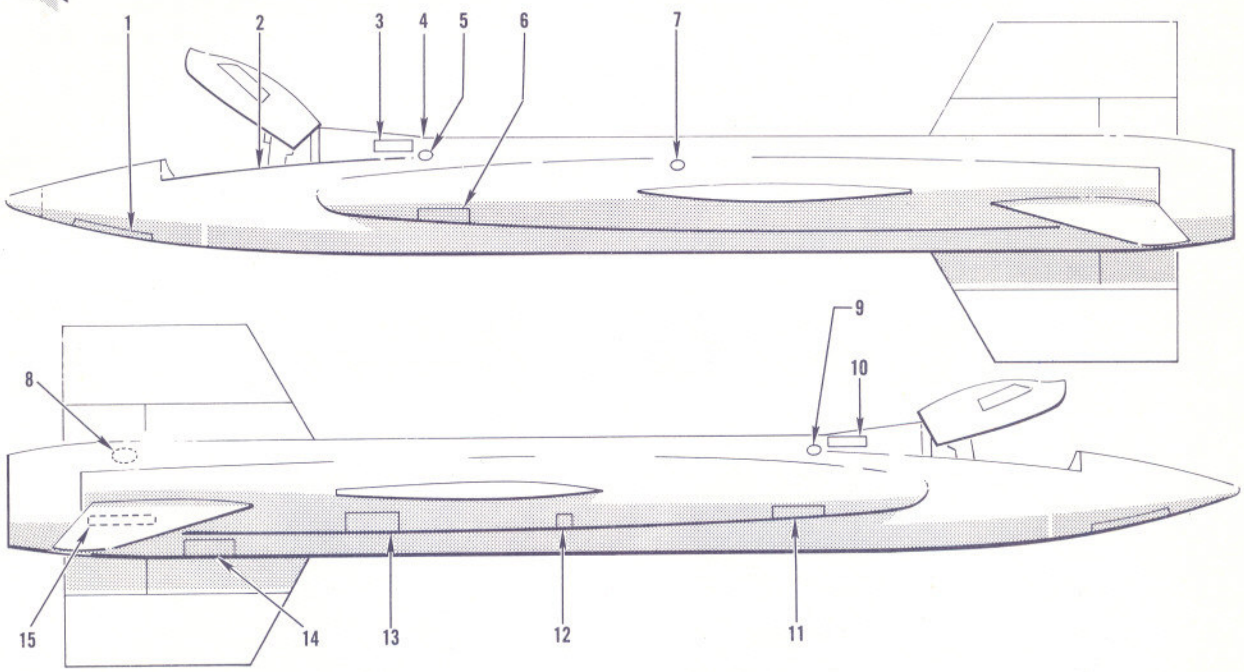
If the pilot prebreathes 100 percent oxygen before entering the cockpit, the visor must not be opened at any time during ground or flight operations; otherwise, the denitrogenation effect will be nullified.

When the visor is down and latched, the suit and helmet are completely sealed in a leakproof unit. The visor is kept fog-free by the breathing oxygen as it comes in around the face opening and across the visor.

AUXILIARY EQUIPMENT.

The following auxiliary equipment is described in Section IV: air conditioning and pressurization system, communication and associated electronic equipment, lighting equipment, pilot's oxygen system, and miscellaneous equipment.

SERVICING DIAGRAM



1. Nitrogen filler—nose gear strut (in wheel well)
2. Breathing oxygen filler (on seat)
3. Lubricating oil filler—APU No. 1
4. B-52 disconnects (7)
5. No. 1 hydraulic reservoir level sight gage
6. Helium filler and pressure gage—APU No. 1
   Hydrogen peroxide filler—APU No. 1
   No. 1 hydraulic pressure disconnect
   No. 1 hydraulic suction disconnect
7. Liquid oxygen filler—engine oxidizer
8. Engine control hydraulic oil filler
9. No. 2 hydraulic reservoir level sight gage
10. Lubricating oil filler—APU No. 2
11. Liquid nitrogen filler—air conditioning and pressurization system
    Helium filler—air conditioning and pressurization system
    Helium filler and pressure gage—APU No. 2
    Hydrogen peroxide filler—APU No. 2
    No. 2 hydraulic pressure disconnect
    No. 2 hydraulic suction disconnect
12. Liquid nitrogen filler—helium tank cooling
    Pressure test—propellant system controls
    Helium filler—propellant system
13. Ammonia filler
14. Helium filler—engine purge system and No. 1 and No. 2 hydraulic accumulators
    Hydrogen peroxide filler—engine turbopump propellant
15. Engine lubrication system filler

SPECIFICATIONS

HYDROGEN PEROXIDE	NA2-2103 Grade A
HELIUM	High-grade, oil-free, with purity greater than 99.9 percent, and maximum dew point of −70°F
GASEOUS NITROGEN	Grade A Type I MIL-N-6011
LIQUID NITROGEN	Grade A Type II MIL-N-6011
LIQUID OXYGEN (OXIDIZER)	MIL-P-25508
GASEOUS OXYGEN (BREATHING)	BB-O-925 Grade A Type I
HYDRAULIC FLUID	NA2-2078A (Oronite 8515)
LUBRICATING OIL (APU)	MIL-L-7808C
LUBRICATING OIL (ENGINE)	Halo carbon oil 4-11V (RMD Spec 4043)
ANHYDROUS AMMONIA	JAN-A-182
HYDRAULIC OIL (ENGINE CONTROL)	RMD Spec 4041

X-15-1-00-8D

Figure 1-16