Description SECTION I

AIRPLANE.

The X-15 is a single-place research airplane, specifically designed to obtain data on flight at extremely high altitudes and speeds and on the physiological and psychological effects of such flight conditions on the pilot. Built by North American Aviation, Inc, the airplane has an inertial all-attitude (gyro-stabilized platform) flight data system and is powered by one XLR99 liquid-propellant rocket engine. The 25-1/2 degree swept-back wing has hydraulically operated flaps on the inboard trailing edge of each wing panel. All aerodynamic control surfaces are actuated by irreversible hydraulic systems. The horizontal stabilizer has a 15-degree cathedral. The two sections move simultaneously for pitch control, differentially for roll control, and in compound for pitch-roll control. The upper and lower vertical stabilizers are in two sections, a movable outer span for yaw control and a fixed section adjacent to the fuselage. The lower movable section (ventral) is jettisonable for landing. Each fixed section incorporates a split-flap speed brake. For changes in airplane attitude relative to flight trajectory at altitudes where aerodynamic controls are relatively ineffective, the airplane incorporates a ballistic control system, wherein the metered release of gas through small rockets in the nose and wing causes the airplane to move about each axis as required. Two auxiliary power units drive the airplane hydraulic pumps and ac electrical generators. Fuel for the rocket engine is carried internally. The airplane is not designed for normal ground take-off, but is air-launched by a B-52 Airplane. The landing gear consists of a dual-wheel nose gear and two main landing skids. The gear is lowered in flight by gravity and air loads.

AIRPLANE DIMENSIONS.

The over-all dimensions of the airplane (in-flight configuration with gear up and ventral retained) are as follows:

Length	49 feet 2 inches
Span	22 feet 4 inches
Height	13 feet 1 inch

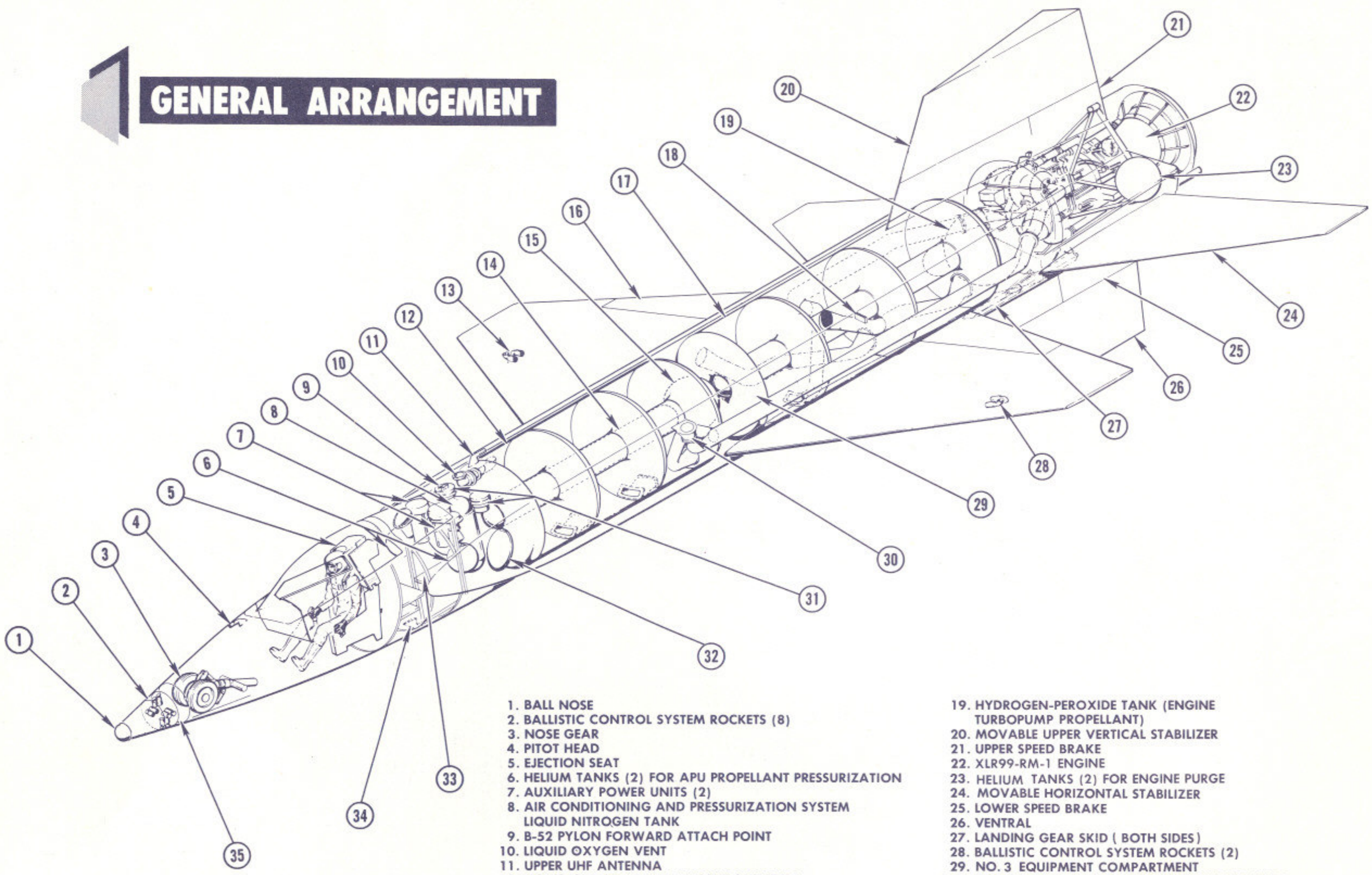
NOTE

> In the landing configuration (landing gross weight and gear down, with specified nose tire and strut inflation and with ventral jettisoned), height is 11 feet 6 inches.

AIRPLANE GROSS WEIGHT.

The approximate launch gross weight of the airplane (including full internal load and pilot) is 32,900 pounds. However, this can vary a few hundred pounds, depending on the type of instrumentation carried.

GENERAL ARRANGEMENT



1. BALL NOSE
2. BALLISTIC CONTROL SYSTEM ROCKETS (8)
3. NOSE GEAR
4. PITOT HEAD
5. EJECTION SEAT
6. HELIUM TANKS (2) FOR APU PROPELLANT PRESSURIZATION
7. AUXILIARY POWER UNITS (2)
8. AIR CONDITIONING AND PRESSURIZATION SYSTEM LIQUID NITROGEN TANK
9. B-52 PYLON FORWARD ATTACH POINT
10. LIQUID OXYGEN VENT
11. UPPER UHF ANTENNA
12. LIQUID OXYGEN TANK (ENGINE OXIDIZER)
13. BALLISTIC CONTROL SYSTEM ROCKETS (2)
14. HELIUM TANK (FOR ENGINE PROPELLANT PRESSURIZATION)
15. HELIUM TANK FOR PRESSURIZATION CONTROL, ENGINE CONTROL, AND ENGINE HYDROGEN PEROXIDE PRESSURIZATION
16. RIGHT-HAND WING FLAP
17. AMMONIA TANK
18. B-52 PYLON REAR ATTACH POINT (BOTH SIDES)
19. HYDROGEN-PEROXIDE TANK (ENGINE TURBOPUMP PROPELLANT)
20. MOVABLE UPPER VERTICAL STABILIZER
21. UPPER SPEED BRAKE
22. XLR99-RM-1 ENGINE
23. HELIUM TANKS (2) FOR ENGINE PURGE
24. MOVABLE HORIZONTAL STABILIZER
25. LOWER SPEED BRAKE
26. VENTRAL
27. LANDING GEAR SKID (BOTH SIDES)
28. BALLISTIC CONTROL SYSTEM ROCKETS (2)
29. NO. 3 EQUIPMENT COMPARTMENT
30. LIQUID OXYGEN TANK (ENGINE OXIDIZER) FILLER
31. HYDRAULIC SYSTEM RESERVOIRS (2)
32. HYDROGEN-PEROXIDE TANKS (2) APU AND BALLISTIC CONTROL SYSTEMS PROPELLANT
33. NO. 2 EQUIPMENT COMPARTMENT
34. LOWER UHF ANTENNA
35. NO. 1 EQUIPMENT COMPARTMENT

X-15-1-00-1B

Figure 1-1

AIRPLANE SERIAL NUMBERS.

The Air Force serial numbers for X-15 Airplanes covered by this manual are AF56-6670, -6671, and -6672.

ENGINE.

Thrust is provided by one XLR99 turborocket engine. It has a single thrust chamber, a two-stage, continuous-igniter starting system, a turbopump, and a gas generator. Propellants are liquid oxygen and anhydrous ammonia, supplied from the airplane propellant system. (See figure 1-6.) The engine is of variable-thrust design, capable of operating over the range of 50 to 100 percent of full rated thrust. The gas generator decomposes a monopropellant fuel, 90 percent hydrogen peroxide (H_2O_2), to provide a high-pressure gas mixture for driving the turbopump, which in turn drives the two centrifugal pumps that supply the propellants to the engine. Upon discharge from the pumps, the propellants are delivered to the two igniters and the thrust chamber where they are burned. At the first-stage igniter, the oxygen (in gaseous form) and ammonia are mixed and then ignited by three spark plugs. Liquid oxygen and ammonia, meanwhile, also are routed to the second-stage igniter. When the pressure created by the hot gases in the first-stage igniter actuates a pressure switch, propellants are allowed to enter the second-stage igniter. Here, the propellants are mixed and ignited by the incoming first-stage gases at greatly increased pressure. When a pressure switch in the second-stage igniter is actuated, propellants are allowed to enter the thrust chamber itself. They are again mixed and ignited by the gases coming from the second-stage igniter and build to the tremendous pressures needed for required thrust. The thrust chamber is an assembly of small welded, wire-wound tubes preformed as segments of the chamber. Before injection into the thrust chamber, the ammonia passes through these tubes to cool the chamber. Exhaust gases are discharged through a venturi-shaped sonic nozzle.

ENGINE COMPARTMENT.

The engine compartment, in the extreme aft end of the fuselage, houses the tubular steel engine mount which supports the engine and turbopump. The engine compartment is completely isolated from the airframe by a mono-fire-wall. A large access door is provided in the forward end of the engine compartment for access to the engine compartment from the hydrogen peroxide storage tank area. The engine compartment also houses the instrumentation pick-offs, fire detection system sensors, and helium release line. A fire seal closes out the compartment and protects against the entry of exhaust gases and expelled propellants into the engine compartment. For engine compartment purging, refer to "Engine Compartment Purging System" in this section.

Engine Compartment Fire Detection System.

A detection circuit is provided to detect and indicate a fire condition in the engine compartment. This circuit is of the continuous-element type, which detects excessive temperatures anywhere along its length. The system is powered by the battery bus and continuously monitors the resistance of the circuit. The resistance of the material used in the circuit varies inversely with temperature and total length of the sensing circuit. Whenever temperatures in the engine compartment reach 1100°F (594°C) or higher, the resistance of the sensing element falls below a preset value because of the excessive temperature, and the warning system is energized. A placard-type warning light, a system selector switch, and a test switch are in the cockpit. For emergency procedures in case of a fire-warning indication, refer to "Fire or Explosion" in Section III.

Fire-warning Light. An abnormal rise in engine compartment temperature is shown by a placard-type warning light (70, figure 1-2), on the instrument panel. The light is powered by the primary dc bus and has a red plastic cap which shows the word "FIRE" when the light is on. The light may be tested by a push-to-test switch on the instrument panel right wing.

Fire-warning Light Test Button. A fire-warning light and detection circuit test button (30, figure 1-2) is on the instrument panel right wing. The button is powered by the primary dc bus. When the button is pressed, the fire-warning light should come on, verifying the continuity of the detection circuit.

Engine Compartment Purging System.

The engine compartment can be purged by releasing an inert gas (helium) under pressure into the area to extinguish a fire or relieve an overheat condition. Three cubic feet of helium is stored in two spherical containers under 3600 psi pressure. The containers are on either side of the engine compartment in the left and right wing root fairing tunnels. Either automatic or manual release of helium can be selected by the pilot. Because of the location of the two containers adjacent to the engine compartment, any high-temperature condition in the compartment will affect these containers and create a potential explosion hazard. As the helium is released into the engine compartment, it inhibits any fire condition and at the same time eliminates the explosion hazard.

Helium Release Selector Switch. This three-position switch (69, figure 1-2), labeled "HE REL SW," is on the left side of the instrument panel. It permits the pilot to select the type of engine compartment purging (either automatic or manual) in case of a fire indication. The switch is powered by the battery bus. The AUTO position sets up an entirely automatic sequence if a fire occurs in the engine compartment, as indicated by illumination of the fire-warning light. The engine is shut down, and the helium from the two containers outboard of the engine is jettisoned into the engine compartment to inhibit the fire or overheat condition and to prevent overpressurization of the containers by extreme temperature increase. The OFF position sets up the fire detection system for illuminating the fire-warning light only, in case a fire occurs. It will then be necessary either to move the switch to ON to release the helium into the engine compartment without affecting

INSTRUMENT PANEL

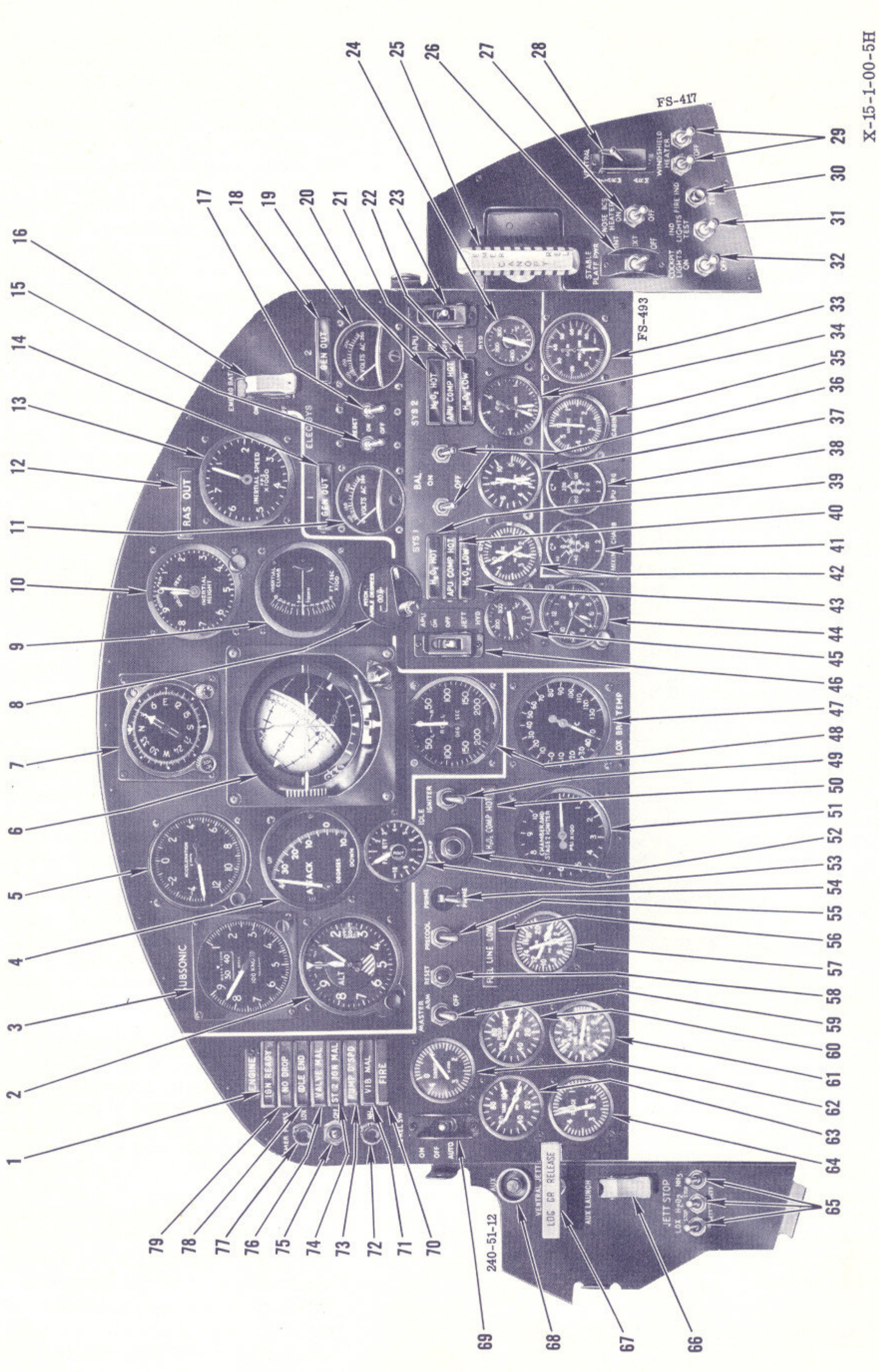


Figure 1-2

1. IGNITION-READY CAUTION LIGHT
2. ALTIMETER
3. AIRSPEED INDICATOR
4. ANGLE-OF-ATTACK INDICATOR
5. ACCELEROMETER
6. ATTITUDE INDICATOR
7. AZIMUTH INDICATOR
8. PITCH ANGLE SET CONTROL
9. VERTICAL VELOCITY INDICATOR
10. INERTIAL HEIGHT (ALTIMETER) INDICATOR
11. NO. 1 GENERATOR VOLTMETER
12. RAS-OUT INDICATOR LIGHT
13. INERTIAL SPEED (VELOCITY) INDICATOR
14. NO. 1 GENERATOR-OUT LIGHT
15. NO. 1 GENERATOR SWITCH
16. EMERGENCY BATTERY SWITCH
17. NO. 2 GENERATOR SWITCH
18. NO. 2 GENERATOR-OUT LIGHT
19. NO. 2 GENERATOR VOLTMETER
20. NO. 2 APU HYDROGEN PEROXIDE OVERHEAT WARNING LIGHT
21. NO. 2 APU COMPARTMENT OVERHEAT CAUTION LIGHT
22. NO. 2 APU HYDROGEN PEROXIDE-LOW CAUTION LIGHT
23. NO. 2 APU SWITCH
24. NO. 2 HYDRAULIC TEMPERATURE GAGE
25. CANOPY INTERNAL EMERGENCY JETTISON HANDLE
26. STABLE PLATFORM SWITCH
27. NOSE BALLISTIC ROCKET HEATER SWITCH
28. VENTRAL ARMING SWITCH
29. WINDSHIELD HEATER SWITCHES (2)
30. FIRE-WARNING LIGHT TEST BUTTON
31. INDICATOR, CAUTION, AND WARNING LIGHT SWITCH
32. COCKPIT LIGHTING SWITCH
33. CABIN PRESSURE ALTIMETER
34. HYDRAULIC PRESSURE GAGE
35. CABIN HELIUM SOURCE PRESSURE GAGE
36. NO. 1 AND NO. 2 BALLISTIC CONTROL SWITCHES
37. HYDROGEN PEROXIDE TANK PRESSURE GAGE
38. APU BEARING TEMPERATURE GAGE
39. NO. 1 APU HYDROGEN PEROXIDE OVERHEAT WARNING LIGHT
40. NO. 1 APU COMPARTMENT OVERHEAT CAUTION LIGHT
41. MIXING CHAMBER TEMPERATURE GAGE
42. APU SOURCE PRESSURE GAGE
43. NO. 1 APU HYDROGEN PEROXIDE-LOW CAUTION LIGHT
44. CLOCK
45. NO. 1 HYDRAULIC TEMPERATURE GAGE
46. NO. 1 APU SWITCH
47. LIQUID OXYGEN BEARING TEMPERATURE GAGE
48. RATE-OF-ROLL INDICATOR
49. IGNITER IDLE SWITCH
50. H_2O_2 COMPARTMENT-HOT CAUTION LIGHT
51. CHAMBER AND STAGE 2 IGNITER PRESSURE GAGE
52. TURBOPUMP IDLE BUTTON
53. FUEL QUANTITY GAGE
54. ENGINE PRIME SWITCH
55. ENGINE PRECOOL SWITCH
56. FUEL LINE-LOW CAUTION LIGHT
57. PROPELLANT MANIFOLD PRESSURE GAGE
58. ENGINE RESET BUTTON
59. ENGINE MASTER SWITCH
60. PROPELLANT PUMP INLET PRESSURE GAGE
61. H_2O_2 TANK AND ENGINE CONTROL LINE PRESSURE GAGE
62. PROPELLANT SOURCE PRESSURE GAGE
63. PROPELLANT TANK PRESSURE GAGE
64. H_2O_2 SOURCE AND PURGE PRESSURE GAGE
65. JETTISON STOP SWITCHES
66. AUXILIARY LAUNCH SWITCH
67. LANDING GEAR HANDLE
68. VENTRAL JETTISON BUTTON
69. HELIUM RELEASE SELECTOR SWITCH
70. FIRE-WARNING LIGHT
71. ENGINE VIBRATION MALFUNCTION CAUTION LIGHT
72. AMMONIA TANK PRESSURE-LOW CAUTION LIGHT
73. TURBOPUMP OVERSPEED CAUTION LIGHT
74. STAGE 2 IGNITION MALFUNCTION CAUTION LIGHT
75. PROPELLANT EMERGENCY PRESSURIZATION SWITCH
76. VALVE MALFUNCTION CAUTION LIGHT
77. LIQUID OXYGEN TANK PRESSURE-LOW CAUTION LIGHT
78. IDLE END LIGHT
79. NO-DROP CAUTION LIGHT

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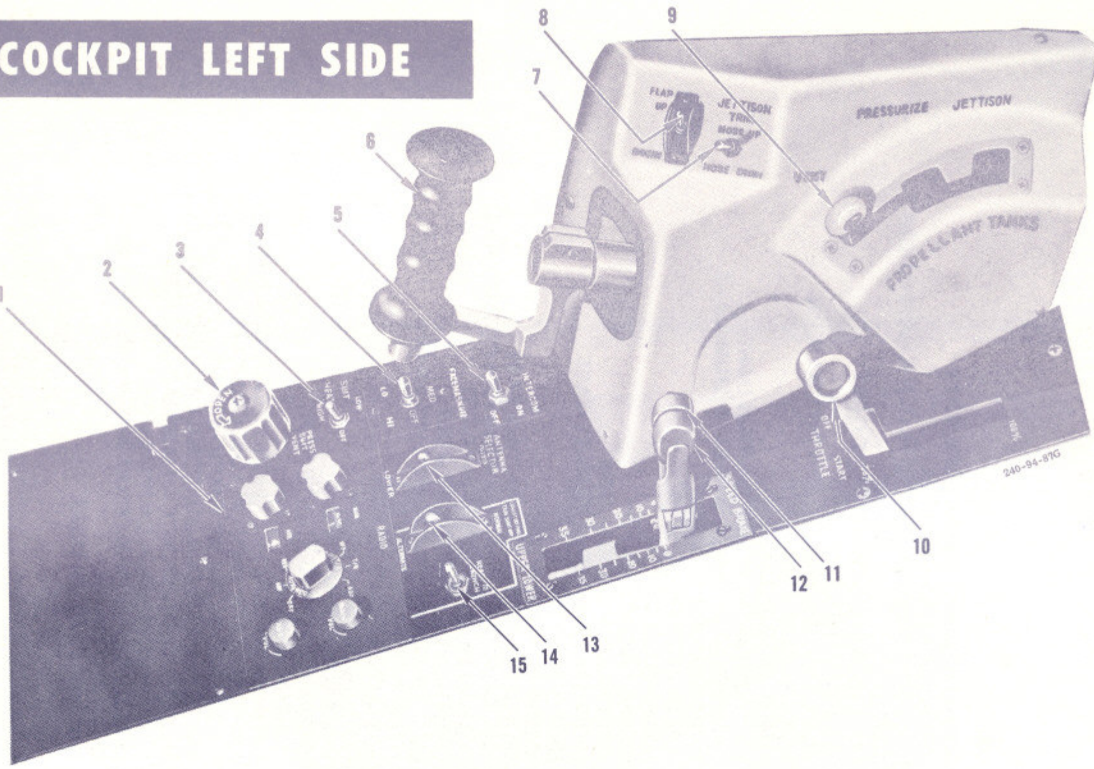
COCKPIT LEFT SIDE



1. UHF CONTROL PANEL
2. PRESSURE SUIT VENTILATION KNOB
3. VENT SUIT HEATER SWITCH
4. FACE MASK HEATER SWITCH
5. INTERCOMMUNICATION SWITCH
6. BALLISTIC CONTROL STICK
7. JETTISON TRIM SWITCH
8. WING FLAP SWITCH
9. VENT, PRESSURIZATION, AND JETTISON LEVER
10. ENGINE THROTTLE
11. UPPER SPEED BRAKE HANDLE
12. LOWER SPEED BRAKE HANDLE
13. ANTENNA SELECTOR SWITCH
14. TRIM CONTROL SWITCH
15. READY-TO-LAUNCH SWITCH

X-15-1-00-6E

Figure 1-3.

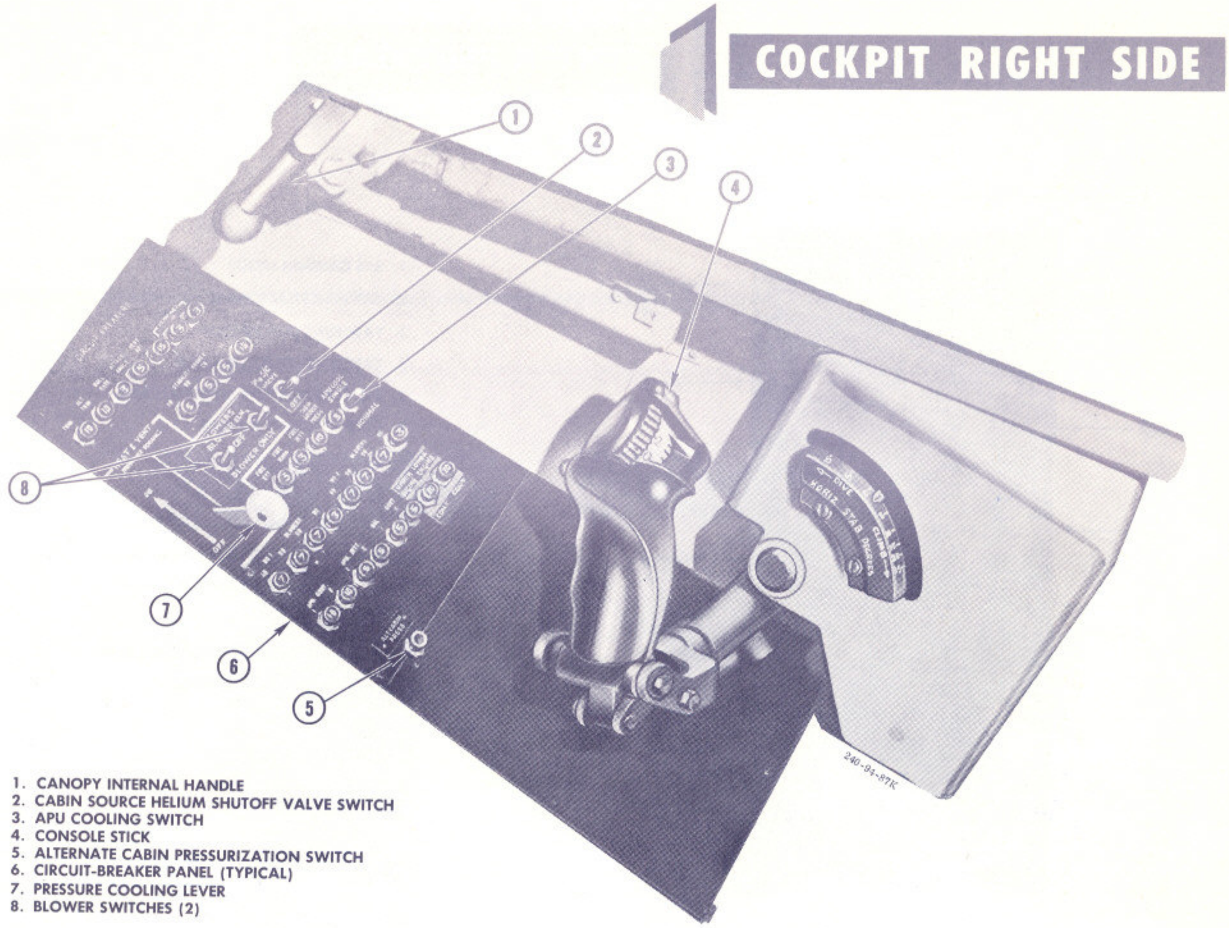
engine operation, or to move the switch to AUTO to release the helium and simultaneously shut down the engine. Moving the switch to ON will release the helium into the engine compartment whenever the battery bus is energized. Once energized, the helium release valve is locked in the jettison position and must be electrically unlocked by ground personnel.

ENGINE TURBOPUMP.

A turbopump, mounted on the front of the engine, delivers the propellants in the desired quantities and at the proper pressures to the engine. The turbopump contains a gas generator, a two-stage, axial-flow turbine, and two centrifugal pumps on a common shaft. Each pump supplies one of the propellants to the engine. The monopropellant for driving the turbine is 90 percent hydrogen peroxide (H_2O_2). An electrohydraulic power servo system governs turbine speed to the selected power requirement.

Turbopump Propellant (H_2O_2) System.

The hydrogen peroxide monopropellant (H_2O_2) used to drive the turbopump is contained in a 10-cubic-foot spherical supply tank (19, figure 1-1) with a capacity of 854 pounds (77.5 US gallons). A swivel-type pickup feed line allows positive feeding of the monopropellant regardless of airplane attitude. The system includes a combination vent, pressure relief, and tank pressurization valve; a jettison valve; a hydrogen peroxide throttle control metering valve; a safety valve; a shut-off valve; and a gas generator. This system is controlled by switches and a control lever in the cockpit and is put into operation whenever the engine starting sequence is begun. For a description of these controls, refer to "Engine Controls" in this section. When the engine is not operating, the tank is vented to atmosphere if the vent, pressurization, and jettison control lever is at VENT and control gas is available. The tank is pressurized with helium control gas, to feed the H_2O_2 to the gas generator, which provides steam power for turbopump operation. Tank pressure can be read from a gage in the cockpit. Refer to "H_2O_2 Tank and Engine Control Line Pressure Gage" in this section. The system also includes a jettison feature that permits the H_2O_2 to be forcibly expelled overboard.



1. CANOPY INTERNAL HANDLE
2. CABIN SOURCE HELIUM SHUTOFF VALVE SWITCH
3. APU COOLING SWITCH
4. CONSOLE STICK
5. ALTERNATE CABIN PRESSURIZATION SWITCH
6. CIRCUIT-BREAKER PANEL (TYPICAL)
7. PRESSURE COOLING LEVER
8. BLOWER SWITCHES (2)

X-15-1-00-7D

Figure 1-4.

H_2O_2 Compartment-hot Light. An amber H_2O_2 compartment-hot caution light (50, figure 1-2), on the instrument panel, comes on when temperature in the upper area of the turbopump propellant compartment reaches 538°C (1000°F) or when temperature in the lower area of the compartment reaches 427°C (800°F). When illuminated, the light reads "H_2O_2 COMP HOT." The light is powered by the primary dc bus and may be tested through the indicator, caution, and warning light test circuit.

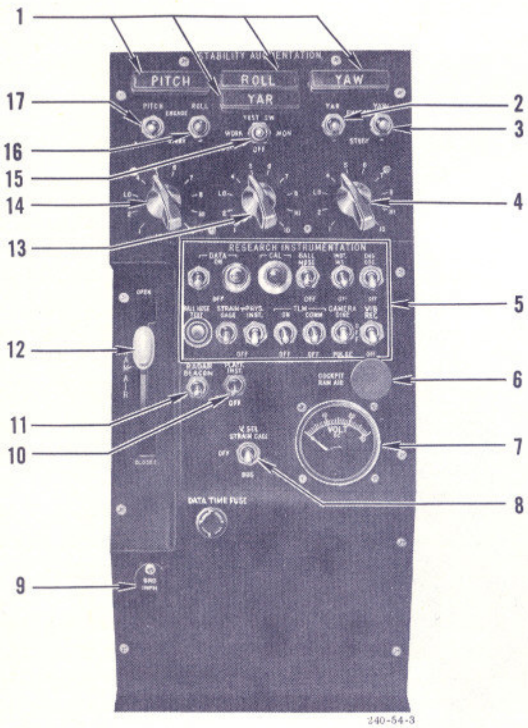
Turbopump Speed Control.

An electrohydraulic servo system is used as an actuation and reference system between the turbine speed and H_2O_2 flow. Its main components are a power package, a governor, a throttle synchro, a servo amplifier, a governor actuator, and an H_2O_2 throttle control metering valve. Pressurized oil from an electrically driven hydraulic pump is supplied to the governor and metering valve. When the engine throttle (throttle synchro) is moved, the governor speed adjustment lever is set to the desired position by the governor actuator. The speed of the turbopump is sensed by the governor, and the hydraulic pressure balance between the governor and metering valve is adjusted to control peroxide flow into the gas generator. Decrease or increase of the turbopump speed from that required for the selected thrust causes a hydraulic imbalance between governor and metering valve. As the governor reacts to restore the hydraulic balance, hydraulic pressure to the metering valve is increased or decreased, as necessary, to alter the rate of H_2O_2 flow to the gas generator and thus restore the turbopump to the desired speed.

ENGINE PROPELLANT AND CONTROL SYSTEM.

The two propellants, anhydrous ammonia and liquid oxygen, are routed from their respective fuel tanks to the main feed valves, which are operated by helium pressure. From the main feed valves, the fuel is routed to the turbopump. The fuels, pressurized by a low-pressure inert gas (helium), flow from the respective tanks to the turbopump. Prime orifices allow propellants to circulate and cool the engine and prime the propellant pumps. The turbopump begins operation when the hydrogen peroxide supply upstream safety valve and downstream shutoff valves are opened. The

CENTER PEDESTAL



1. SAS CAUTION LIGHTS
2. YAR FUNCTION SWITCH
3. YAW FUNCTION SWITCH
4. YAW GAIN SELECTOR KNOB
5. INSTRUMENTATION CONTROL PANEL
6. COCKPIT RAM-AIR KNOB
7. DC VOLTMETER
8. DC VOLTMETER SWITCH
9. GROUND INTERPHONE RECEPTACLE
10. STABLE PLATFORM INSTRUMENT SWITCH
11. RADAR BEACON SWITCH
12. RAM-AIR LEVER
13. ROLL AND YAR GAIN SELECTOR KNOB
14. PITCH GAIN SELECTOR KNOB
15. SAS TEST SWITCH
16. ROLL FUNCTION SWITCH
17. PITCH FUNCTION SWITCH

X-15-1-00-13F

Figure 1-5.

H_2O_2 then flows to a gas generator, where it is converted to a high-pressure gas mixture of superheated steam and oxygen to drive the turbine wheel, which in turn drives the propellant pumps. The propellants are then supplied to the first-stage and second-stage igniters and to the main thrust chamber. After priming is completed, the turbopump is operating, and the first-stage igniter propellant valve is opened, the liquid oxygen to the first-stage igniter is routed inside the turbine exhaust duct, whose hot gases heat the liquid oxygen and change it to a gaseous state. The gaseous oxygen and ammonia then enter the first-stage igniter. Three spark plugs in the first-stage igniter fire the fuel and oxidizer mixture. When the pressure switch in the first-stage igniter is actuated, the second-stage igniter start valves open, allowing liquid oxygen and ammonia to flow into the second-stage igniter. First-stage igniter flames ignite the second-stage fuel mixture. Combustion pressure in the second-stage igniter then actuates a switch which signals the main propellant valve to open. When the main propellant valve opens, fuel and liquid oxygen are injected into the main thrust chamber, where they are ignited by second-stage flames. Before entering the main thrust chamber, the ammonia is routed through the chamber tubes in order to cool the main thrust chamber. Opening of the main propellant valve stops the flow of propellants to the prime valves. Once engine operation has been initiated, thrust output is varied between 50% and 100% according to the throttle position selected by the pilot. The engine propellant control system is shown schematically in figure 1-7.

ENGINE CONTROLS.

Throttle.

The throttle (10, figure 1-3) controls thrust output of the engine. The throttle quadrant has three marked positions: OFF, 50%, and 100%. The throttle controls an electromechanical servo system, which includes a synchro transmitter attached to the throttle, a servo amplifier, and an actuator position transmitting synchro linked to the turbopump governor. Turbopump and first- and second-stage igniter operation is accomplished with the throttle at OFF (full aft and outboard). During the 30-second idle operation period with the throttle OFF, the turbopump is automatically maintained at idle speed. Within 30 seconds after igniter idle operation is begun, the throttle must be moved to 50% to open the main propellant valves to the main thrust chamber or the start sequence must be terminated. After main thrust chamber operation is begun, movement of the throttle between 50% and 100% will vary engine thrust accordingly.

Vent, Pressurization, and Jettison Lever.

This lever (9, figure 1-3) controls the pressurization system selector valve. The valve is a manually controlled pneumatic selector valve. The lever has three positions: VENT, PRESSURIZE, and JETTISON. With the lever at VENT, helium pressure (from the helium pressure control system) is applied to all tank control valves in the propellant system. The pressurization valves close and the vent valves open, venting the H_2O_2, liquid oxygen, and ammonia tanks. In order to obtain engine operation, the lever must be moved to PRESSURIZE, opening the propellant system pressurization valves and closing the vent valves. This allows helium to enter and pressurize the turbopump H_2O_2 supply tank and the liquid oxygen and ammonia tanks. When the lever is positioned to JETTISON, helium pressure is applied to open three jettison valves and pressurize the H_2O_2, liquid oxygen, and ammonia tanks. The three propellants will then begin to dump overboard.

NOTE

The propellants will not jettison if the jettison test switches are OFF.

Jettison Trim Switch.

This switch (7, figure 1-3) is on the left vertical side panel. It has three positions: NOSE UP, NOSE DOWN, and an unmarked, center off position. The switch is powered by the 28-volt primary dc bus. With the vent, pressurization, and jettison lever at JETTISON, the jettison stop switches in the JETT position, and this switch at the unmarked off position, simultaneous jettisoning of H_2O_2, liquid oxygen, and ammonia will occur. Moving this switch to NOSE UP (when nose-down trim is felt) stops the flow of the ammonia. Moving the switch to NOSE DOWN (when nose-up trim is felt) stops the flow of liquid oxygen. In either case, when the airplane returns to trim, the switch must be released and allowed to return to the unmarked off position.

Jettison Stop Switches.

Three jettison stop switches (65, figure 1-2), on the instrument panel left wing, have a STOP and a JETT position. These switches, powered by the primary dc bus, are normally left in the STOP position until the prelaunch cruise portion of the flight. To perform a test of the turbopump H_2O_2, liquid oxygen, and ammonia jettison system, the vent, pressurization, and jettison control lever should be placed at JETTISON. The systems then can be tested by placing the switches to JETT. The jettison line of each system should then emit a vaporous cloud. Flow will cease when the switches are returned to the STOP position or when the vent, pressurization, and jettison control lever is moved to PRESSURIZE. (See figure 1-15 for location of jettison, drain, and bleed outlets.)

Engine Master Switch.

The engine master switch (59, figure 1-2), on the instrument panel, is powered by the primary dc bus. With the switch at OFF, primary dc bus power for engine control and engine indicator lights is interrupted. With the switch at ARM, primary dc bus power is applied to the engine indicator lights and engine control switching units.

Engine Reset Button.

The engine reset button (58, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. For a normal engine start or if a malfunction causes automatic shutdown during any phase of operation, depressing this button positions the engine control circuits to the armed position. However, if the malfunction which caused shutdown persists, engine control circuits will not be armed.

Engine Precool Switch.

The engine precool switch (55, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. It has two maintained positions: PRECOOL and OFF (down). With an engine start sequence initiated, moving the switch to PRECOOL opens the liquid oxygen main feed valve and precools the system up to the main propellant valve. The precooling flow dumps overboard through the engine liquid oxygen prime valve.

NOTE

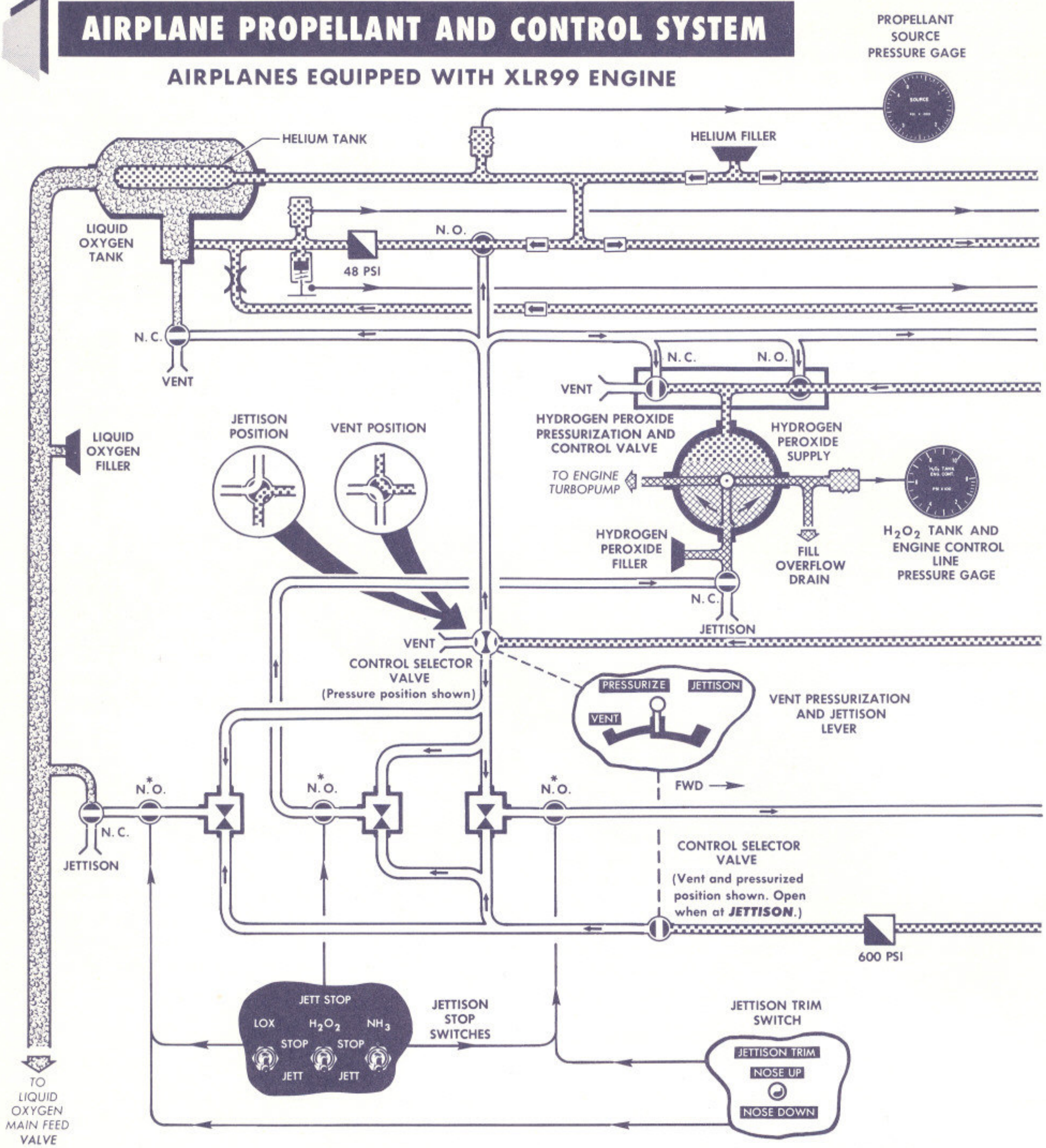
About 10 minutes is required to precool the engine liquid oxygen system. After precooling is completed, the engine can be maintained in a precooled condition for an extended period by the following schedule: engine precool switch at OFF for 20 minutes, then at PRECOOL for 7-1/2 minutes, repeating this cycle as often as necessary.

Engine Prime Switch.

The engine prime switch (54, figure 1-2), on the instrument panel, has three positions: an unmarked, maintained center position; a momentary PRIME position; and a maintained STOP PRIME position. With an engine start sequence initiated, moving the switch momentarily to PRIME opens the liquid oxygen and ammonia main feed valves and the turbopump H_2O_2 upstream safety valve and admits helium to the engine control and purge systems. Approximately 30 seconds is required for priming at high-flow rate, and when the engine precool switch is placed at OFF, prime continues at low-flow rate until an actual start stops the prime or until the engine prime switch is placed momentarily at STOP PRIME. Engine operation may be terminated during any phase by moving this switch to STOP PRIME.

AIRPLANE PROPELLANT AND CONTROL SYSTEM

AIRPLANES EQUIPPED WITH XLR99 ENGINE



NOTE

Move jettison trim switch to ***NOSE DOWN*** to stop liquid oxygen jettison. Move switch to ***NOSE UP*** to stop ammonia jettison.

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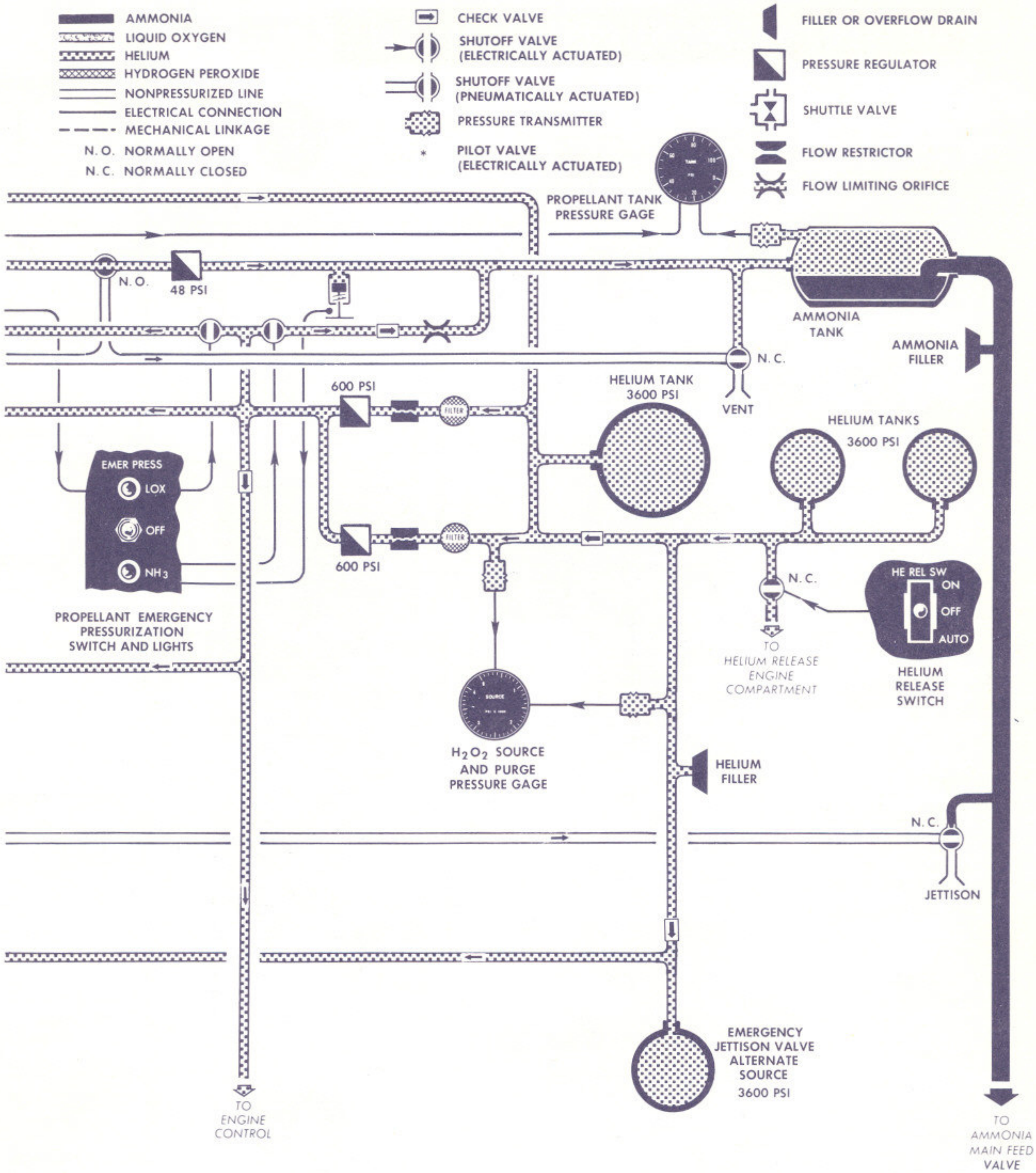
Figure 1-6



NOTE

When the vent, pressurization, and jettison control lever is in the ***VENT*** position, all vent valves are open, and pressurization and jettison valves are closed. When the lever is in the ***PRESSURIZED*** position, the pressurization valves are open, and the jettison and vent valves are closed. When the lever is in the ***JETTISON*** position, the jettison and pressurization valves are open, and the vent valves are closed.

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ENGINE PROPELLANT CONTROL SYSTEM

NOTE

Liquid oxygen, ammonia, and hydrogen peroxide are supplied to the feed valves under pressure when the vent, pressurization, and jettison control lever is placed at ***PRESSURIZE.***

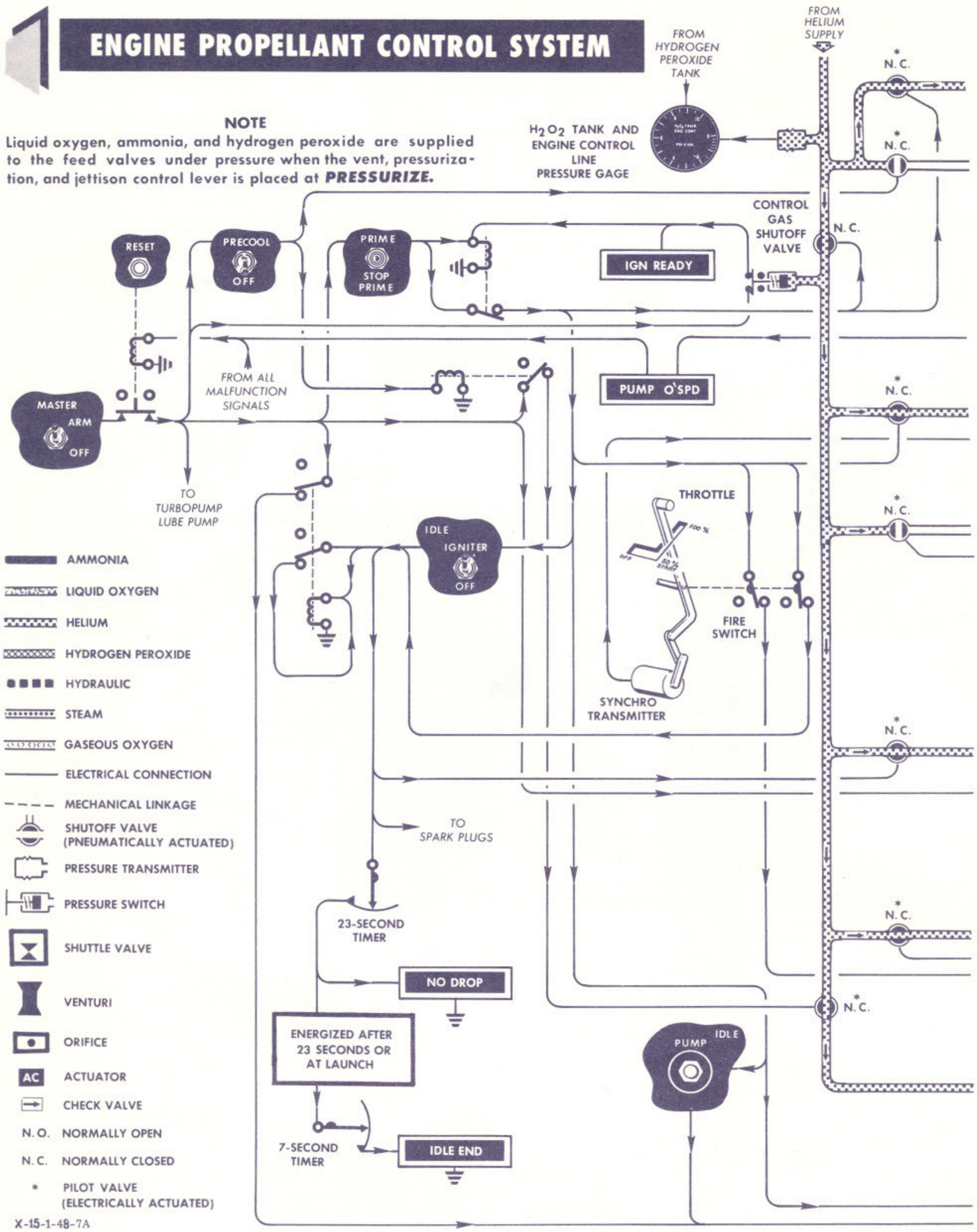
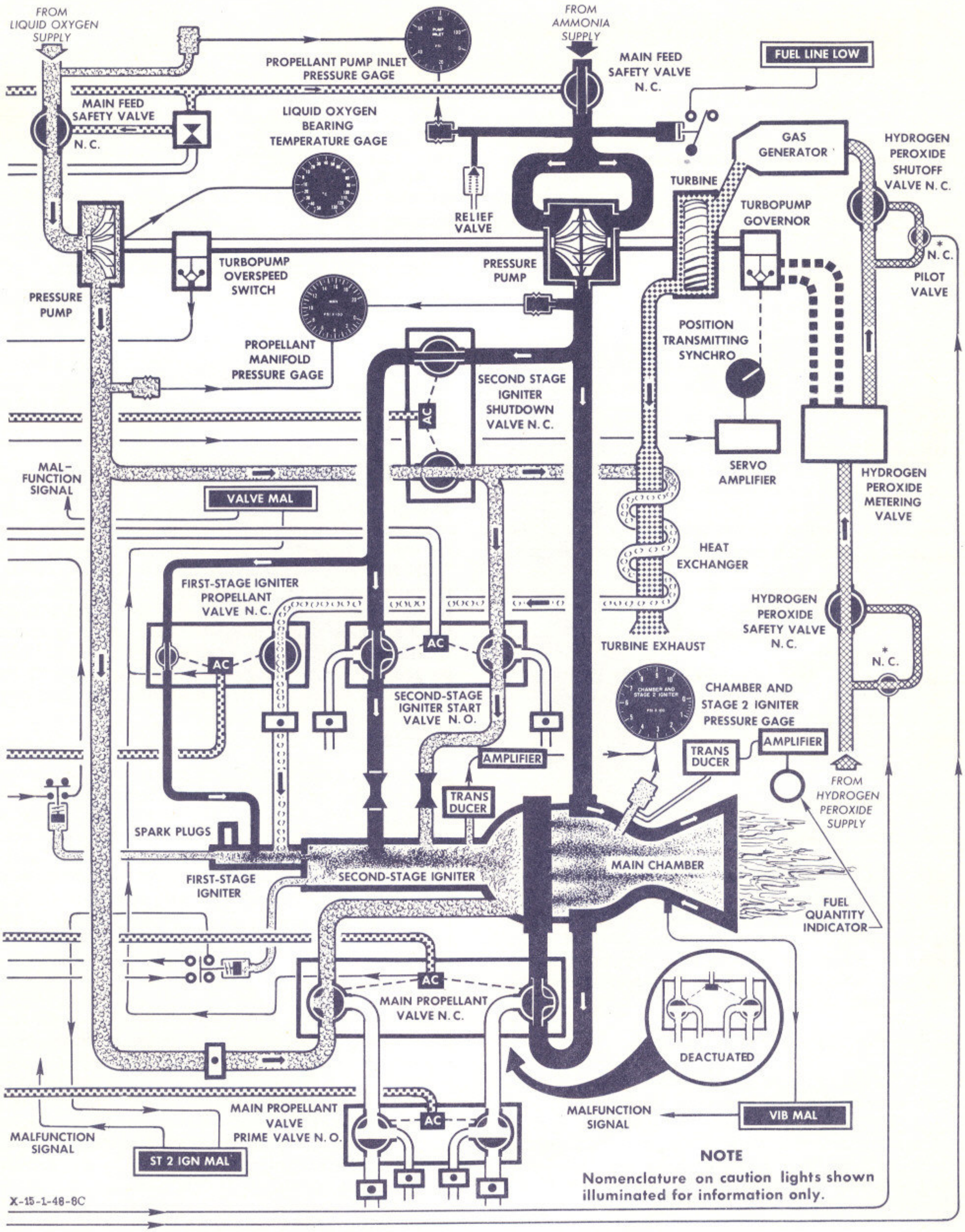


Figure 1-7


FROM LIQUID OXYGEN SUPPLY
PROPELLANT PUMP INLET PRESSURE GAGE
FROM AMMONIA SUPPLY
MAIN FEED SAFETY VALVE N.C.
FUEL LINE LOW
MAIN FEED SAFETY VALVE
N.C.
LIQUID OXYGEN BEARING TEMPERATURE GAGE
GAS GENERATOR
HYDROGEN PEROXIDE SHUTOFF VALVE N.C.
RELIEF VALVE
TURBINE
TURBOPUMP GOVERNOR
TURBOPUMP OVERSPEED SWITCH
PRESSURE PUMP
N.C.
PILOT VALVE
PRESSURE PUMP
POSITION TRANSMITTING SYNCHRO
PROPELLANT MANIFOLD PRESSURE GAGE
SECOND STAGE IGNITER SHUTDOWN VALVE N.C.
AC
SERVO AMPLIFIER
HYDROGEN PEROXIDE METERING VALVE
MAL-FUNCTION SIGNAL
VALVE MAL
HEAT EXCHANGER
FIRST-STAGE IGNITER PROPELLANT VALVE N.C.
HYDROGEN PEROXIDE SAFETY VALVE N.C.
AC
AC
TURBINE EXHAUST
N.C.
SECOND-STAGE IGNITER START VALVE N.O.
CHAMBER AND STAGE 2 IGNITER PRESSURE GAGE
AMPLIFIER
TRANS DUCER
AMPLIFIER
TRANS DUCER
FROM HYDROGEN PEROXIDE SUPPLY
SPARK PLUGS
MAIN CHAMBER
FIRST-STAGE IGNITER
SECOND-STAGE IGNITER
FUEL QUANTITY INDICATOR
AC
MAIN PROPELLANT VALVE N.C.
DEACTUATED
MALFUNCTION SIGNAL
VIB MAL
MAIN PROPELLANT VALVE PRIME VALVE N.O.
AC
MALFUNCTION SIGNAL
ST 2 IGN MAL
NOTE
Nomenclature on caution lights shown illuminated for information only.
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Turbopump Idle Button.

The turbopump idle button (52, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch and the engine overspeed reset button. With an engine start sequence initiated and the prime phase completed, depressing this button for one second opens the turbopump H_2O_2 downstream shutoff valve, which starts turbopump operation. When pressure of the ammonia in the propellant manifold builds to approximately 210 psi, the turbopump speed control system begins operation and maintains the turbopump at idle speed.

Igniter Idle Switch.

This switch (49, figure 1-2), on the instrument panel, receives power from the primary dc bus through the engine master switch and engine overspeed reset button. With an engine start sequence initiated and the engine turbopump operating at idle speed, moving this switch from OFF (down) to IGNITER causes the following sequence of actions: a 2-second helium purge is initiated, and the spark plugs are energized; the first-stage igniter propellant valves open, and igniter and idle timing starts; gaseous nitrogen flow (from the carrier airplane) starts when first-stage igniter pressure reaches a specified value; propellants flow to the second-stage igniter; and second-stage ignition occurs.

WARNING

This phase of operation (igniter idle) is limited to 30 seconds. Either igniter idle operation must be terminated (by placing the engine prime switch to STOP PRIME) or the launch accomplished at the end of the 30-second idle period.

ENGINE INDICATORS.

Propellant Tank Pressure Gage.

This dual-indicating gage (63, figure 1-2), on the instrument panel, is powered by the 26-volt ac bus. The gage indicates the two propellant tank pressures. The gage is graduated from 0 to 100 psi in increments of 5 psi. One pointer of the gage has the letter "L," indicating the liquid oxygen tank pressure; the other pointer has the letter "A," indicating the ammonia tank pressure.

Propellant Pump Inlet Pressure Gage.

This dual-indicating gage (60, figure 1-2) is powered by the 26-volt ac bus. It indicates liquid oxygen and ammonia pressures at the engine turbopump inlets. This gage is graduated from 0 to 100 psi in increments of 25 psi. The pointer labeled "L" reads the liquid oxygen pressure; the pointer labeled "A" indicates the ammonia pressure.

Liquid Oxygen Bearing Temperature Gage.

This gage (47, figure 1-2) is powered by the 26-volt ac bus and indicates the temperature of the bearing for the liquid oxygen centrifugal pump segment of the engine turbopump. The gage is graduated from -60° F to 260° F in 5-degree increments.

Propellant Manifold Pressure Gage.

This dual-indicating gage (57, figure 1-2), on the instrument panel, is powered by the 26-volt ac bus. The gage indicates propellant pump discharge pressures. It is graduated from 0 to 2000 psi in increments of 50 psi. One pointer is labeled "L" and indicates liquid oxygen pump discharge pressure; the other pointer is labeled "A" and indicates ammonia pump discharge pressure.

Chamber and Stage 2 Igniter Pressure Gage.

This dual-indicating gage (51, figure 1-2), on the instrument panel, is powered by the 26-volt ac bus. The gage is graduated from 0 to 1000 psi in increments of 20 psi from 0 to 100, and 50 psi from 100 to 1000. The short hand indicates pressure in the second-stage igniter. The long hand indicates pressure in the main thrust chamber.

Ignition-ready Indicator Light.

This green indicator light (1, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. When illuminated, it reads "IGN READY," indicating that the engine electrical circuits and purge gas network have been energized. In the normal starting sequence, this light will go out for 2 seconds when the igniter idle switch is moved to IGNITER, then come on again. During all helium purges, this light will go out. This light may be tested by the indicator, caution, and warning light test circuit.

No-drop Caution Light.

This amber caution light (79, figure 1-2) on the instrument panel, is powered by the primary dc bus through the engine master switch. During a normal engine start sequence, this light will come on when 7 seconds remains in the igniter idle phase of operation. The light, when illuminated, reads "NO DROP" and serves to warn the pilot to terminate igniter idle operation or to continue on to the launch phase. This light may be tested by the indicator, caution, and warning light test circuit.

Idle-end Caution Light.

This amber caution light (78, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. This light will illuminate, reading "IDLE END," when the 30-second igniter idle phase of engine operation is complete. When this light comes on, engine shutdown must be accomplished or operation continued into the main chamber phase (after launch). This light may be tested by the indicator, caution, and warning light test circuit.

Valve Malfunction Caution Light.

This amber caution light (76, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. When illuminated, this light reads "VALVE MAL." This light will come on when malfunction shutdown occurs because the main or first-stage propellant valve is improperly positioned during the starting sequence. The light will also come on momentarily whenever a malfunction shutdown occurs with the main chamber operating. This light may be tested by the indicator, caution, and warning light test circuit.

Stage 2 Ignition Malfunction Caution Light.

This amber caution light (74, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. When illuminated, this light reads "ST 2 IGN MAL." The light will come on when a malfunction shutdown occurs during the starting sequence because of failure of the second-stage igniter to reach operating pressure. The light will also come on momentarily whenever a malfunction shutdown occurs with the main chamber operating. This light may be tested by the indicator, caution, and warning light test circuit.

Vibration Malfunction Caution Light.

This amber light (71, figure 1-2), on the instrument panel, comes on when the engine shuts down because of excessive vibration. Excessive engine vibration causes a shutdown signal to be transmitted from either of two sensors to the engine control box. The signal causes the actuation of two malfunction relays in the control box that de-energize the prime, precool, and firing circuits to shut down the engine and turn on the light. The light is powered by the primary dc bus. If the light comes on during powered flight, an engine restart may be attempted.

Turbopump Overspeed Caution Light.

This amber caution light (73, figure 1-2), on the instrument panel, is powered by the primary dc bus through the engine master switch. When illuminated, this light reads "PUMP O'SPD." The light will come on if a malfunction shutdown occurs because of turbopump overspeed which is not corrected by the turbopump governor. This light may be tested by the indicator, caution, and warning light test circuit.

Fuel Quantity Gage.

A fuel quantity gage (53, figure 1-2), on the instrument panel, is calibrated in percent of total fuel load. The gage is graduated from 0 to 100 in increments of 5 and operates on the principle that chamber pressure in the engine is essentially proportional to the flow of fuel and oxidizer to the engine. During engine operation, the pointer moves counterclockwise on the dial at a rate proportional to the engine thrust selected (which also is proportional to the rate of fuel consumed). Before an engine start is initiated, the pointer is adjusted (by the center reset knob) to indicate the percent of total fuel on board. During engine operation, the gage indicates the percentage of fuel load remaining. The fuel quantity gage receives a signal from the fuel quantity control unit, which is powered by the primary dc and No. 2 primary ac busses.

NOTE

The fuel quantity indicating system can be adjusted for an accurate indication at burnout time by selection of the proper start point with the reset knob. This start point is determined by calibration of the system to the airplane in which it is installed.

Fuel Line Low Caution Light.

An amber "FUEL LINE LOW" caution light (56, figure 1-2) is on the left side of the instrument panel. This light, powered by the primary dc bus through the engine master switch, is actuated by a pressure switch installed in the fuel (ammonia) line downstream of the main safety valve. If fuel pressure at the turbopump inlet drops to 32 (±2) psi, the light will come on. Illumination of the light indicates that partial cavitation of the pump is likely to occur.

CAUTION

When the "FUEL LINE LOW" caution light is on, there is not sufficient cooling around the main chamber of the engine for temperature protection. Thrust settings above 50% may increase the temperature and cause damage to the engine.

If the light comes on before the engine is started, the start will be aborted. (Refer to "Fuel Line Pressure Low" in Section III.) When this light has been illuminated, it will remain on until the engine master switch has been placed in the OFF position. The light may be tested by the indicator, caution, and warning light test switch.

PROPELLANT SUPPLY SYSTEM.

The propellant supply system consists of the liquid oxygen supply (oxidizer), anhydrous ammonia (fuel) supply, valves, and associated plumbing. The liquid oxygen and ammonia are fed under a low inert gas pressure from the supply tanks to the turbopump for engine operation. The helium supply systems, which furnish gas pressure for tank pressurization, pneumatic valve operation, and system purging, are described in separate paragraphs in this section. See figure 1-16 for the liquid oxygen and ammonia specifications.

LIQUID OXYGEN TANK.

The liquid oxygen supply is carried in a triple-compartmented tank (12, figure 1-1), just aft of the No. 2 equipment compartment. The center section area of the cylindrical tank is hollow and forms a case for a gaseous helium high-pressure storage tank. When the liquid oxygen tank is not under pressurization, it is vented to atmosphere. The tank compartments are check-valve-vented. Each compartment feeds rearward toward the airplane center of gravity. The liquid oxygen is fed from the rear compartment under 48 psi of helium pressure to the turbopump or jettison line through a series of control valves. The total volume of the tank is 1034 US gallons; of this amount, 14 gallons is residual at a liquid surface angle of 38 degrees, and 17 gallons is vent and expansion space. The total usable liquid oxygen is 1003 gallons. The tank is filled for flight through the carrier airplane's supply system. The tank incorporates a liquid oxygen fluid level sensing switch that permits the tank to be topped off automatically whenever fluid drops below a predetermined level. For ground operational checks, the tank is serviced through the receptacle mounted on the engine feed line. The tank filler is on the topside of the wing fairing tunnel forward of the left wing root leading edge.

AMMONIA TANK.

The ammonia supply is carried in a triple-compartmented cylindrical tank (17, figure 1-1), just aft of the No. 3 equipment compartment and ahead of the turbopump hydrogen peroxide tank. The center section area is hollow and closed at both ends. The rear compartment center section is perforated to allow storage of ammonia within the center section area. The compartments are check-valve-vented. This aids in the pressure feed of the fluid transfer from the rear tank compartment forward toward the airplane center of gravity. The rear compartment empties first; then the middle compartment empties into the front compartment, with the ammonia fed from the front compartment under 48 psi of helium pressure to the turbopump or jettison line through a series of control valves. The total volume of the tank is approximately 1445 US gallons. The tank is ground-serviced only. The filler receptacle for the tank is on the underside of the right wing root fairing tunnel.

ENGINE AND PROPELLANT CONTROL HELIUM SYSTEM.

Helium to pressurize the turbopump hydrogen peroxide supply tank and to supply pneumatic pressure for engine and propellant control is contained in four spherical tanks. One tank (15, figure 1-1) is between the liquid oxygen and ammonia tanks. Two tanks (23, figure 1-1) are in the left and right wing root fairing tunnels outboard of the engine. These three tanks are interconnected, supplying 3600 psig pressure to two pressure-reducing regulators in parallel. The fourth tank is just to the right of the turbopump H_2O_2 supply tank and supplies helium at 3600 psig to a single pressure-reducing regulator for emergency or secondary pneumatic control of the propellant jettison valves. This tank is interconnected with the other three tanks for filling purposes only. From the parallel pressure-reducing regulators of the main supply, helium at 575 to 600 psig is supplied to the engine helium manifold for operation of engine control valves, to the turbopump H_2O_2 supply tank for tank pressurization, and to propellant control jettison and main feed valves. Two of the tanks supply helium directly to the helium dump valve, for engine compartment purging. The dump valve is solenoid-operated and controlled by the helium release selector switch. For information on operation of this switch, refer to "Helium Release Selector Switch" in this section. The helium to the engine helium manifold is in turn routed to a control gas valve and the two gas regulators in the purge valve network at a pressure of 550 to 600 psig. The control gas valve is energized during the prime period and admits helium at a pressure of 550 to 600 psig to the pilot valves for the prime valve, first-stage igniter start valve, second-stage igniter start and shutoff valves, and main propellant valve. Helium at a pressure of 125 to 200 psig is routed from the two purge gas regulators and to the return side of the second-stage igniter start and shutoff valves. Another regulator supplies helium from the helium manifold at 7.5 psig to the lubrication system accumulator, engine control box, and hydraulic power package.

HELIUM RELEASE SELECTOR SWITCH.

Refer to "Engine Compartment Purging System" in this section.

H_2O_2 SOURCE AND PURGE PRESSURE GAGE.

A dual-indicating H_2O_2 source and purge pressure gage (64, figure 1-2), on the instrument panel, is powered by the 26-volt ac bus. This gage indicates the helium pressure available from three of the engine and propellant helium system tanks. Needle 1 indicates pressure in the large tank between the liquid oxygen and ammonia tanks. Needle 2 indicates pressure in the two smaller tanks in the wing root fairing tunnels. The gage is calibrated from 0 to 4000 psi in increments of 100 psi. Normally, the two pointers will indicate the same pressure. However, if there is a malfunction of the emergency jettison system helium supply or if helium is dumped into the engine compartment, the pointers will not indicate the same pressure. There is no gage in the cockpit which indicates pressure in the emergency jettison system helium supply tank.

H_2O_2 TANK AND ENGINE CONTROL LINE PRESSURE GAGE.

This dual-indicating gage (61, figure 1-2), on the instrument panel, is powered by the 26-volt ac bus. One pointer, labeled "C," indicates engine control line (helium) pressure downstream of the two parallel pressure regulators. The other pointer, labeled "T," indicates pressure in the turbopump H_2O_2 supply tank. The gage is calibrated from 0 to 1000 psi in increments of 50 psi, except that the range 0 to 100 is in increments of 20 psi.

PROPELLANT PRESSURIZATION HELIUM SYSTEM.

The propellant pressurization helium system supplies gas to pressurize the liquid oxygen and ammonia tanks. This helium is contained in the supply tank (14, figure 1-1) within the center section of the liquid oxygen tank and is pressurized to 3600 psi. The helium flows to the normally open pressure regulators of the liquid oxygen and ammonia supply tanks. The two regulators are actuated by helium pressure (from the engine and propellant helium control system) to the closed position when the vent, pressurization, and jettison control lever is at VENT. When the control lever is placed at PRESSURIZE or JETTISON, the regulators open and helium pressure flows to the liquid oxygen and ammonia tanks.

The regulators reduce the helium pressure to 48 psi before it enters the liquid oxygen and ammonia tanks. when the liquid oxygen and ammonia tanks are pressurized, the propellants are forced through the feed lines to the main feed shutoff valves.

PROPELLANT SOURCE PRESSURE GAGE.

The propellant source pressure gage (62, figure 1-2), on the instrument panel, is powered by the 26-volt ac bus. The gage indicates pressure in the cylindrical helium tank for liquid oxygen and ammonia tank pressurization. The gage is calibrated from 0 to 4000 psi in increments of 100 psi.

PROPELLANT EMERGENCY PRESSURIZATION SYSTEM.

The propellant emergency pressurization system can be used to pressurize either the liquid oxygen or the ammonia tank in case of a failure in the normal pressurization system. **This will permit continued low thrust engine operation or propellant jettisoning. The emergency system can supply pressurizing gas to only one propellant tank at a time. The emergency system uses helium from the three interconnected tanks in the engine and propellant control helium system. The system includes a switch and two caution lights.**

Propellant Emergency Pressurization Switch.

This three-position switch (75, figure 1-2), on the instrument panel, controls primary dc bus power to the two emergency pressurization system solenoid-operated shut-off valves. With the switch at OFF, the valves are de-energized closed. The switch must be pulled straight out of a detent to move it from OFF to either of the other positions. With the switch at LOX, electrical power is applied to open the shutoff valve which controls emergency helium pressure to the liquid oxygen tank. With the switch at NH_3, electrical power is applied to open the shutoff valve which controls emergency helium pressure to the ammonia tank. All three switch positions are maintained.

Liquid Oxygen and Ammonia Tank Pressure-low Caution Lights.

These lights (72 and 77, figure 1-2), on the instrument panel, are powered by the primary dc bus. The liquid oxygen tank pressure-low caution light is labeled "LOX." The ammonia tank pressure-low caution light is labeled "NH_3." (The nomenclature for the lights also serves as position nomenclature for the propellant emergency pressurization switch.) After the vent, pressurization, and jettison lever is placed at PRESSURIZE, the related light will come on when pressure in the affected tank drops to 34 (±2) psi. If a light comes on during powered flight, it may remain on even after emergency pressurization of the affected tank has been initiated, indicating that the affected tank pressure is not above 40 psi.

NOTE

During the transitional period when the vent, pressurization, and jettison lever is moved from VENT to PRESSURIZE, the lights should come on and remain on for approximately 6 seconds (during build-up of pressure in the propellant tanks).

AUXILIARY POWER UNITS.

The airplane is equipped with two auxiliary power units (7, figure 1-1) that are set side-by-side in a compartment in the forward fuselage. Each unit is a completely automatic, constant-speed, turbine drive machine that transmits power to, and provides structural support for, an ac generator and a hydraulic pump. Propellant for each auxiliary power unit is provided by an independent feed system, using helium pressure to move the monopropellant, hydrogen peroxide. The two auxiliary power units with their respective feed systems are identified as system No. 1 and system No. 2. Their operation is completely independent of each other, and each furnishes one half of the power required. If one unit should fall, the other will provide sufficient electrical and hydraulic power for limited flight capabilities. Each auxiliary power unit is started and stopped by a switch in the cockpit. When an APU is turned on, a solenoid-type shutoff valve is opened to allow hydrogen peroxide from the propellant feed system to flow into the unit. The propellant is routed first through a gear case for cooling purposes (nitrogen gas is also introduced into the upper turbine bearing area for additional cooling) and then to a modulating flow control valve. The flow control valve is modulated to open or close to provide stabilization through a speed control system consisting of a tachometer generator and a frequency detector. Any turbine overspeed condition is sensed by an overspeed sensing element in the speed control system which will automatically act to close the shutoff valve. When the shutoff valve is closed, fuel flow stops and the unit shuts down. The APU shutoff valve is fitted with a drain that opens when the valve is closed, to relieve any excess pressure in the line downstream of the shutoff valve. After passing through the flow control valve, the hydrogen peroxide enters a decomposition chamber containing a catalyst bed. This catalyst bed is made up of a series of silver and stainless-steel screens which act to decompose the hydrogen peroxide into a high-pressure gas mixture of superheated steam and oxygen. The decomposition chamber is heated electrically from the carrier airplane to ensure a fast start under "cold-soak" conditions in case of an emergency. The superheated steam and oxygen mixture enters a nozzle box in the turbine housing. Here, five nozzles convert pressure energy of the fluid into kinetic energy and direct the flow of gas against a turbine wheel. The turbine, acting through a reduction gear train, transmits power to the ac generator and hydraulic pump. The turbine wheel is housed within an exhaust casing which is designed to contain any buckets that might separate from the wheel during an overspeed operation. The exhaust casing collects spent gases that have passed through the turbine wheel and exhausts them overboard. A gear casing assembly contains the reduction gearing, accessory drive pads, cooling passages, provisions for lubrication, and a drive for the tachometer generator. A typical auxiliary power unit and its propellant feed system are shown schematically in figure 1-8. For information on nitrogen cooling of the upper turbine bearing of each APU, refer to "APU Cooling Switch" in Section IV.

APU SPEED CONTROL.

The speed control for each auxiliary power unit provides positive speed control, starting and stopping, and

AUXILIARY POWER UNIT AND BALLISTIC

NOTE

- Only the No 1 supply system and No 1 APU are shown; the No. 2 supply system and No. 2 APU are identical.
- Nomenclature on indicator lights shown illuminated for information only.

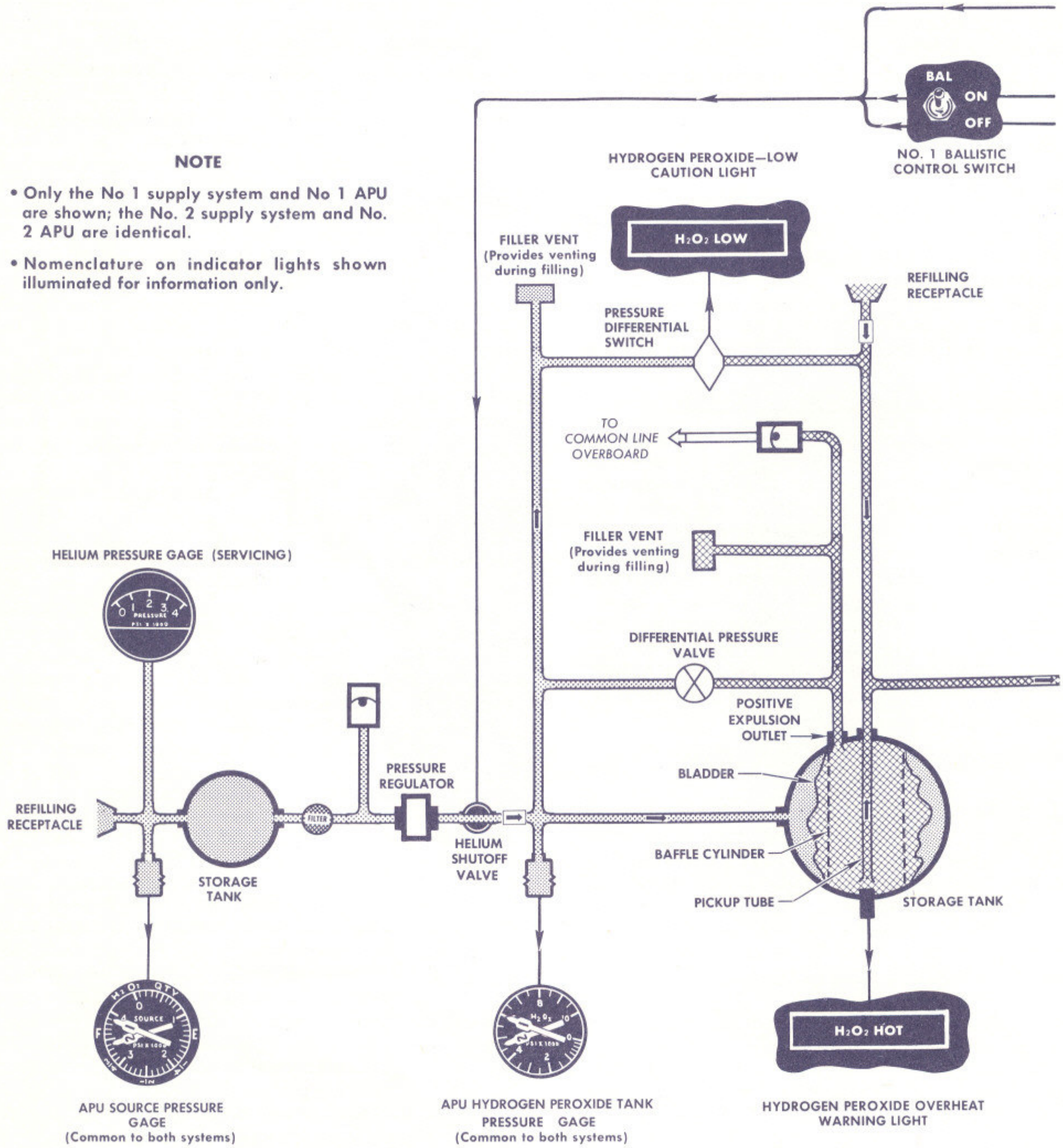

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Figure 1-8 (Sheet 1 of 3)

CONTROL AND REACTION AUGMENTATION SYSTEMS

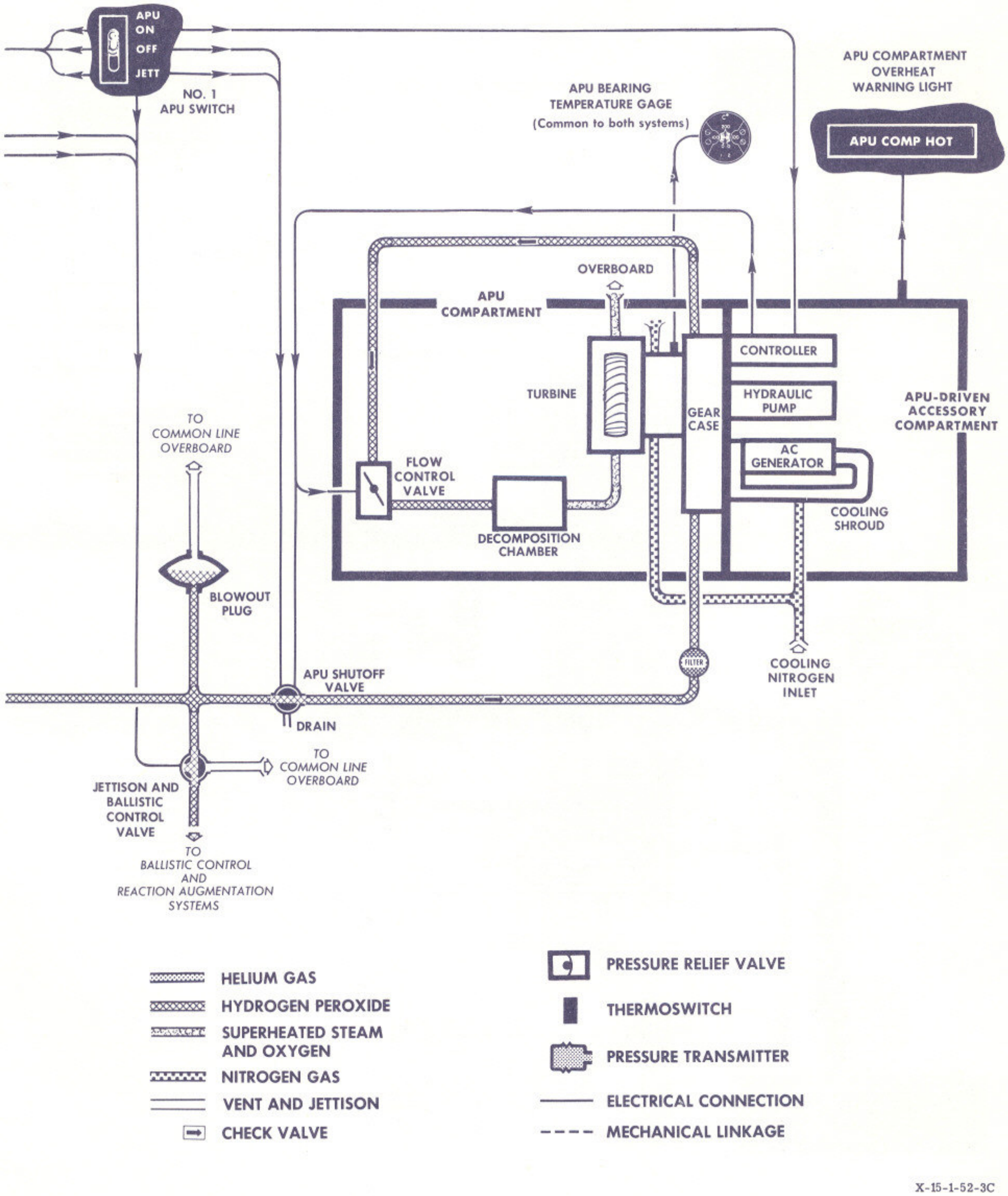

Figure 1-8 (Sheet 2 of 3)

AUXILIARY POWER UNIT AND BALLISTIC CONTROL SYSTEMS

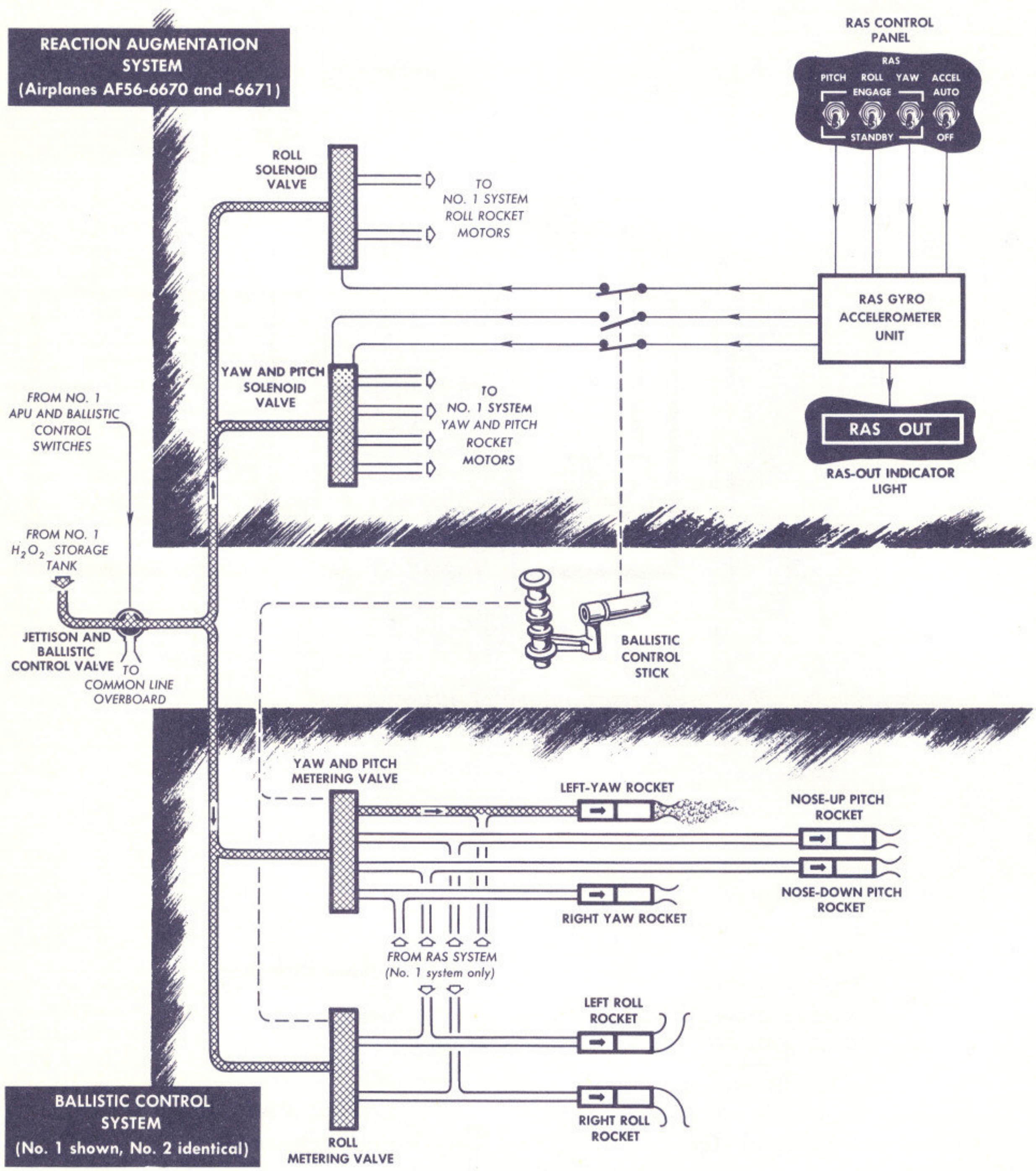

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Figure 1-8 (Sheet 3 of 3)