NASA

PROJECT GEMINI

familiarization manual

SEDR 300 **COPY NO. _____**

RENDEZVOUS and DOCKING CONFIGURATIONS

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FOREWORD

Initiated by the NASA and implemented by McDonnell Aircraft Corporation, Project Gemini is the second major step in the field of manned space exploration. Closely allied to Project Mercury in concept and utilizing the knowledge gained from the Mercury flights, Project Gemini utilizes a two man spacecraft considerably more sophisticated than its predecessor. The Gemini spacecraft is maneuverable within its orbit and is capable of rendezvous and docking with a second orbiting vehicle.

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Reviewed by Arthur C. Esker
Supervisor - Technical Data

Reviewed by for A. Holbrow
NASA - Resident Manager

PROJECT GEMINI

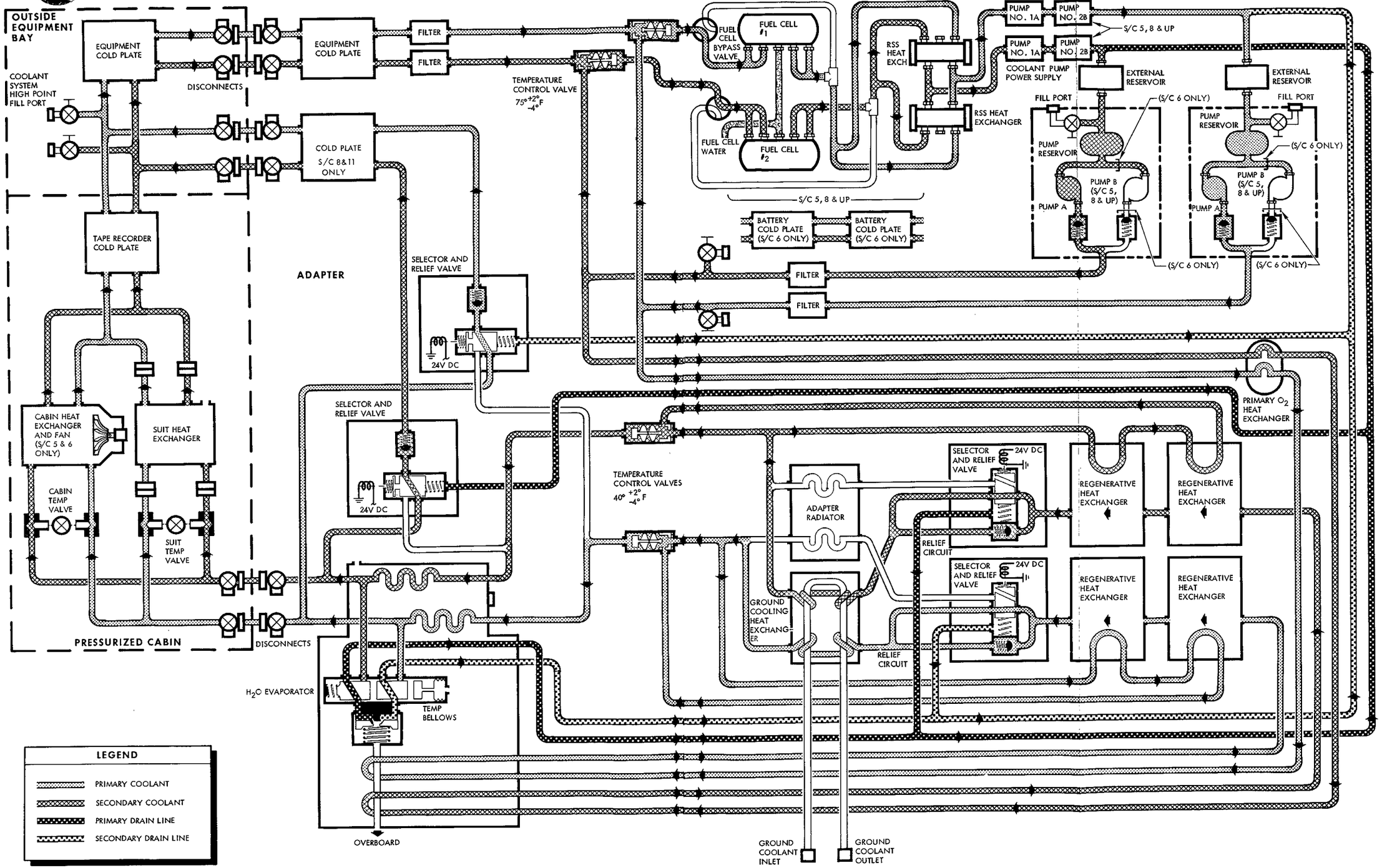

Figure 7-2 Cooling System Flow Schematic Pre-Launch and Launch

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PROJECT GEMINI

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INTRODUCTION

The purpose of this manual is to describe the Gemini Spacecraft systems and major components. The manual is intended as a familiarization-indoctrination aid and as a ready reference for detailed information on a specific system or component. The manual is sectionalized by spacecraft systems or major assemblies. Each section is as complete as is practical to minimize the need for cross referencing.

The information contained in this manual (SEDR 300, VOL II) is applicable to rendezvous missions only and is accurate as of 1 April 1966.

For information pertaining to long range or modified (non-rendezvous) configurations of the spacecraft, refer to SEDR 300, VOL. I.

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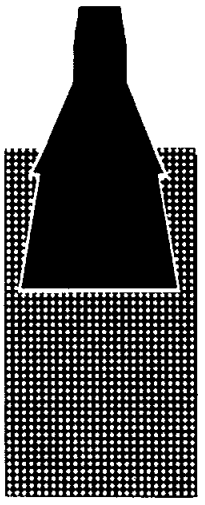

SPACECRAFT MISSION

Section I

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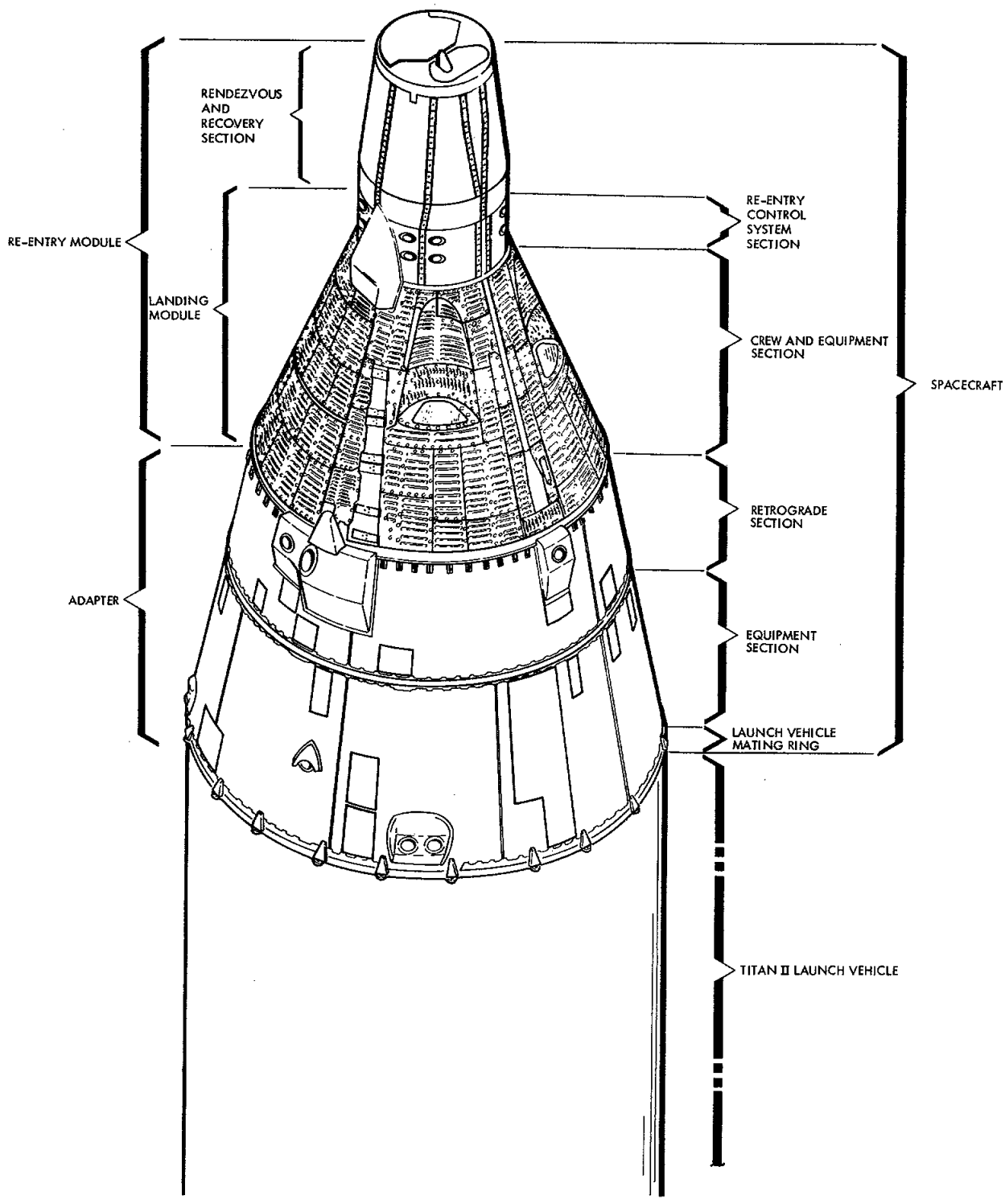


Figure 1-1 Spacecraft Pre-Launch Configuration

SECTION I SPACECRAFT MISSION

MISSION DESCRIPTION

Fundamentally, the mission of Project Gemini is the insertion of a two man spacecraft into a semi-permanent orbit about the earth, the study of man's ability to rendezvous and dock with another orbiting vehicle, and the subsequent safe return of the spacecraft and its occupants to the earths surface. Previous missions included manned and unmanned flights to study human capabilities during extended missions in space. Rendezvous and docking with an orbiting Agena Target Vehicle or Augmented Target Docking Adapter and Extra-Vehicular Activities are planned for most missions.

MISSION OBJECTIVES

Specifically, the project will seek to:

1. Demonstrate the ability of the spacecraft to perform in manual and/or automatic modes of operation.
2. Evaluate the adequacy of major systems in the spacecraft.
3. Verify the functional relationships of the major systems and their integration into the spacecraft.
4. Determine man's requirements and performance capabilities in a space environment.
5. Determine man's interface problems, and develop operational techniques for the most efficient use of on-board capabilities.
6. Evaluate system performance during rendezvous and docking.
7. Demonstrate the ability of the pilots to perform Extra-Vehicular Activities.

8. Develop operational techniques required for rendezvousing and docking with another orbiting vehicle.
9. Develop controlled re-entry techniques required for landing in a predicted touchdown area.
10. Develop operational recovery techniques of both spacecraft and pilots.

SPACECRAFT DESCRIPTION

GENERAL

The Gemini Spacecraft (Figure 1-1) is a conical structure 19 feet long and weighs approximately 7000 lbs. Basically it consists of a re-entry module and an adapter.

RE-ENTRY MODULE

The re-entry module consists of the heat shield, the crew and equipment section, Re-entry Control System section and the rendezvous and recovery section. The crew and equipment section contains a pressurized area suitable for human occupation, and a number of non-pressurized compartments for housing equipment. External access doors are provided for equipment compartments. The Re-entry Control System section contains the major Re-entry Control System components. The rendezvous and recovery section contains the rendezvous radar equipment, the drogue parachute and pilot parachute assemblies, and the main parachute assembly. The rendezvous and recovery section is jettisoned after re-entry along with the drogue parachute.

ADAPTER

The adapter consists of the launch vehicle mating ring, the equipment section and the retrograde section. The launch vehicle mating ring is bolted to the launch vehicle. A portion of the ring remains with the launch vehicle at spacecraft-

launch vehicle separation. The equipment section contains major components of the Electrical, Propulsion, and Cooling Systems. The primary oxygen supply for the Environmental Control System is also located in the equipment section. The retrograde section contains the retrograde rockets and some components of the Cooling System.

LAUNCH VEHICLE DESCRIPTION

The vehicle used to launch the Gemini Spacecraft is the Gemini - Titan II, built by the Martin Company. The Titan II is modified structurally and functionally to accept the Gemini adapter and to provide for the interchange of electrical signals.

The Titan II is a two stage launch vehicle 90 feet long and 10 feet in diameter from the thrust chamber to the spacecraft adapter. The first stage is 70 feet long and develops approximately 430,000 pounds of thrust. The second stage is 20 feet long and develops about 100,000 pounds of thrust.

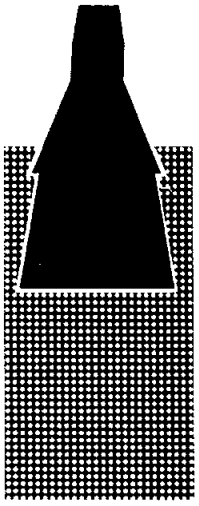
Titan II uses hypergolic (self-igniting when mixed) propellants. Nitrogen tetroxide is the oxidizer and unsymmetrical dimethylhydrazine is the fuel. The propellants can be stored within the launch vehicle indefinitely and ignite automatically when they are mixed in the propulsion chamber. The hypergolic propellants will burn (although at a very rapid rate) rather than explode, which is a significant safety advantage.

CREW REQUIREMENTS

The Gemini Spacecraft utilizes a two-man crew seated side by side. The crew member on the left is referred to as the command pilot and functions as spacecraft commander. The crew member on the right is referred to as the pilot. Crew members are selected from the NASA astronaut group.

SPACECRAFT RECOVERY

The Gemini landing module will make a water landing in a pre-determined area. A task force of ships, planes, and personnel will be standing by for locating and retrieving the spacecraft and crew. In the event an abort or other abnormal occurence results in the spacecraft landing in a remote location, electronic and visual recovery aids and survival kits are provided in the spacecraft to facilitate spacecraft retrieval and crew survival, respectively.



MAJOR STRUCTURAL ASSEMBLIES

Section II

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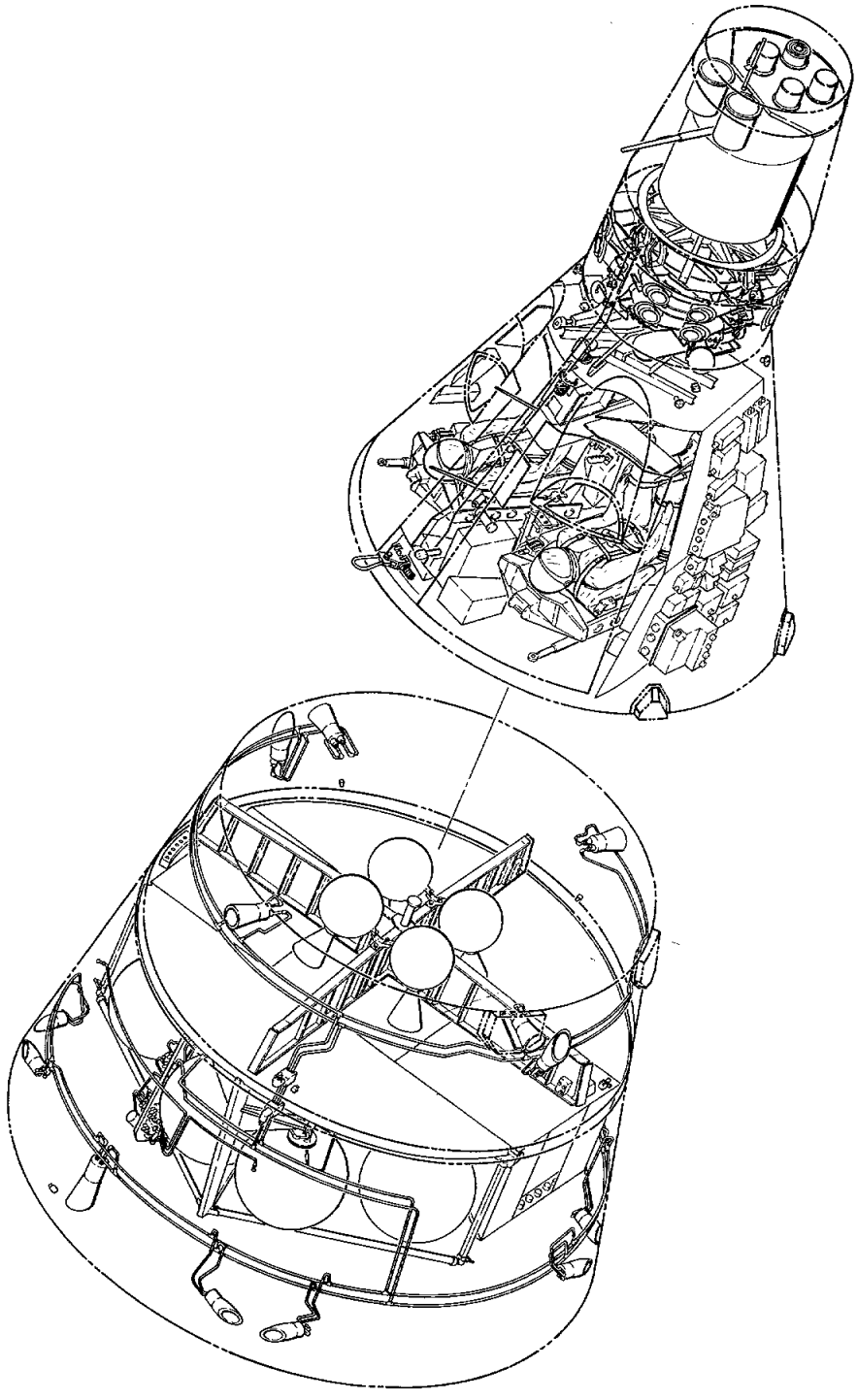


Figure 2-1 Interior Arrangement (Typical)

SECTION II MAJOR STRUCTURAL ASSEMBLIES

GENERAL INFORMATION

The Gemini Spacecraft is basically of a conical configuration (Figure 2-1) consisting of a re-entry module and an adapter as the two major assemblies. Spacecraft construction is semimonocoque, utilizing titanium for the primary structure. It is designed to shield the cabin pressure vessel from exessive temperature variations, noise and meteorite penetration (Figure 2-2). See Figures 2-3 and 2-4 for spacecraft orientation.

RE-ENTRY MODULE

The re-entry module (Figure 2-5) is separated into three primary sections which include the Rendezvous and Recovery section (R and R), Re-entry Control System section (RCS) and the cabin section. Also incorporated in the re-entry module is the heat shield which is attached to the cabin, and a nose fairing which is attached to the forward end of the R and R section. The nose fairing is ejected during launch.

RENDEZVOUS AND RECOVERY SECTION

The (R and R) section (Figure 2-5), the forward section of the spacecraft, is semiconical in shape and is attached to the Re-entry Control System section with twenty-four bolts. Incorporated in this joint is a pyrotechnic device which severs all bolts causing the rendezvous section to separate from the RCS section on signal for parachute deployment. A drogue parachute will assist in the removal of this section. The R and R section utilizes rings, stringers and bulkheads of titanium for its primary structure. The external surface is composed of beryllium shingles, except for the nose fairing. The nose fairing is composed of fiberglass reinforced plastic laminate.

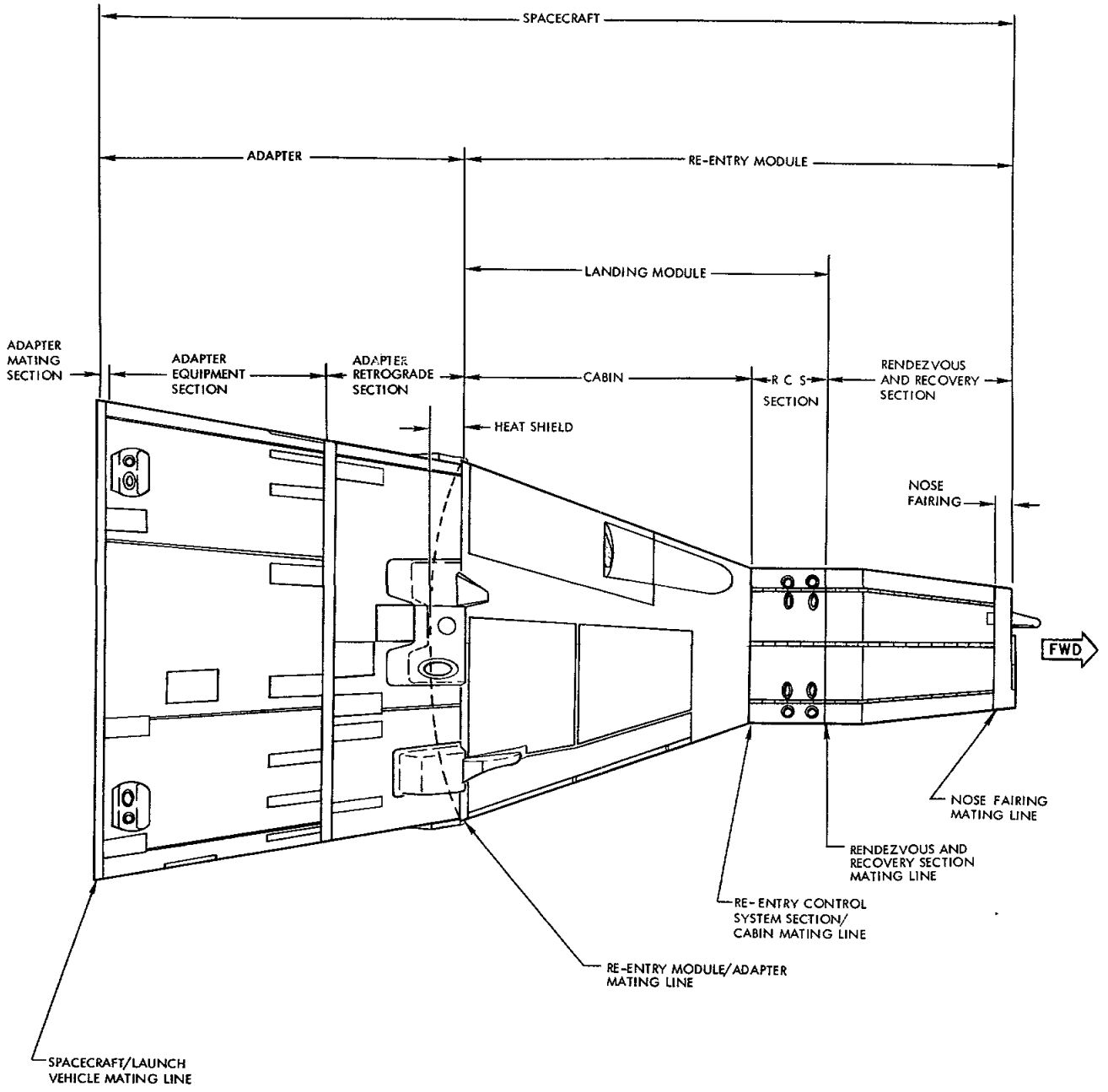
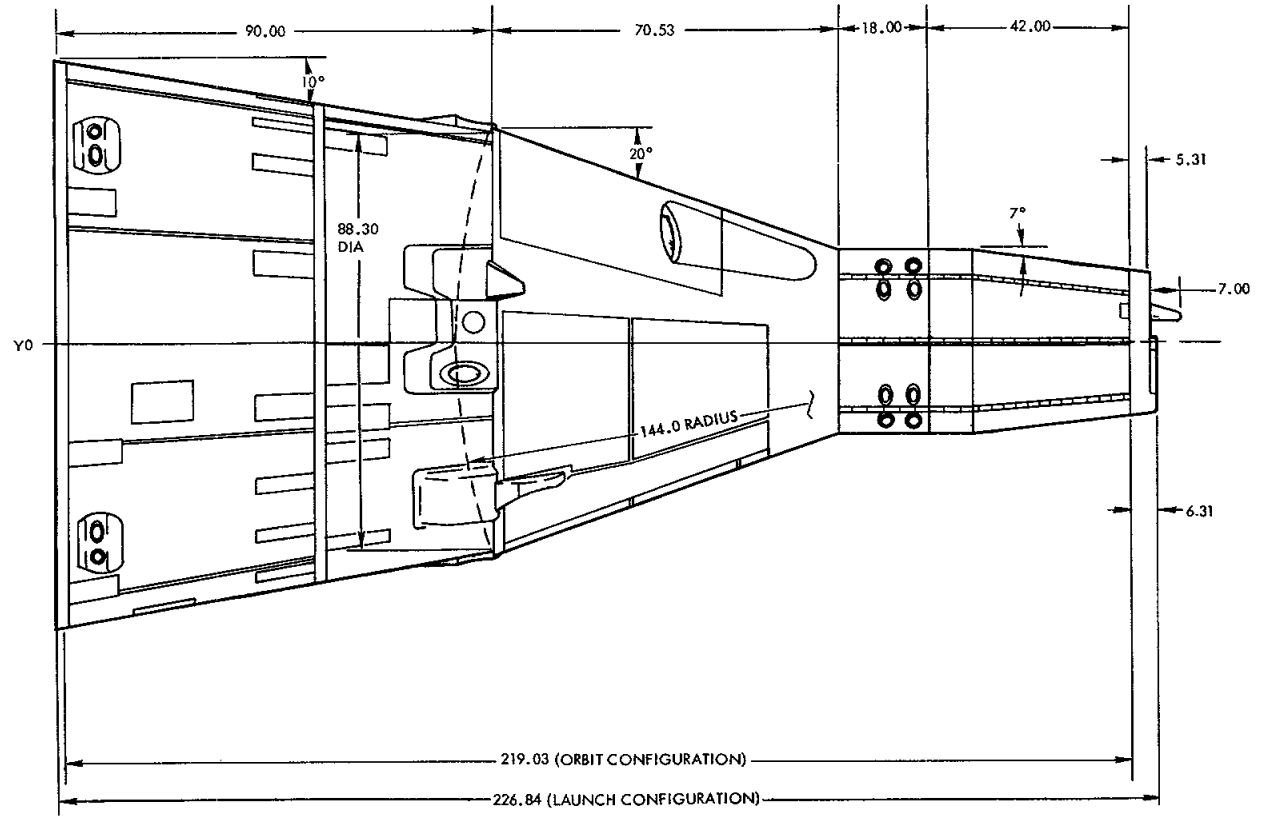


Figure 2-2 Spacecraft General Nomenclature



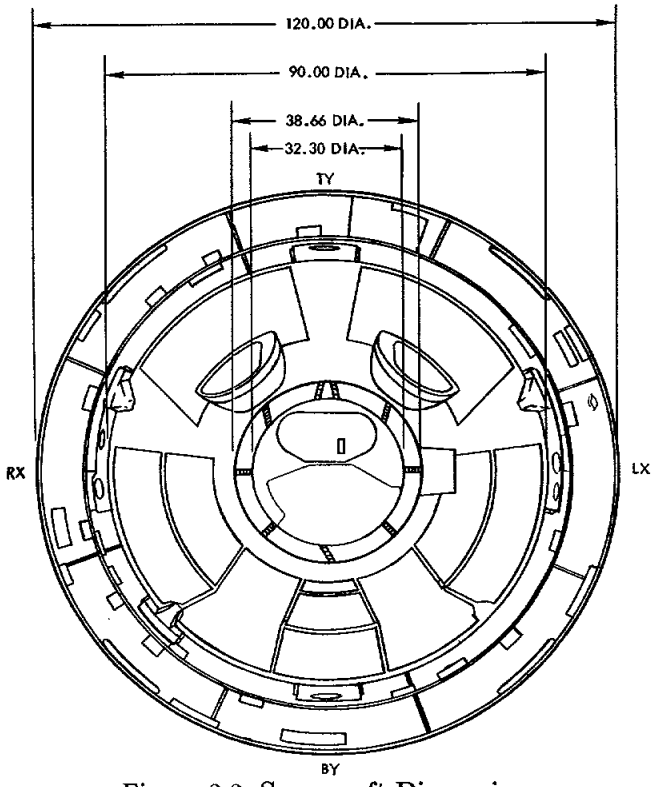
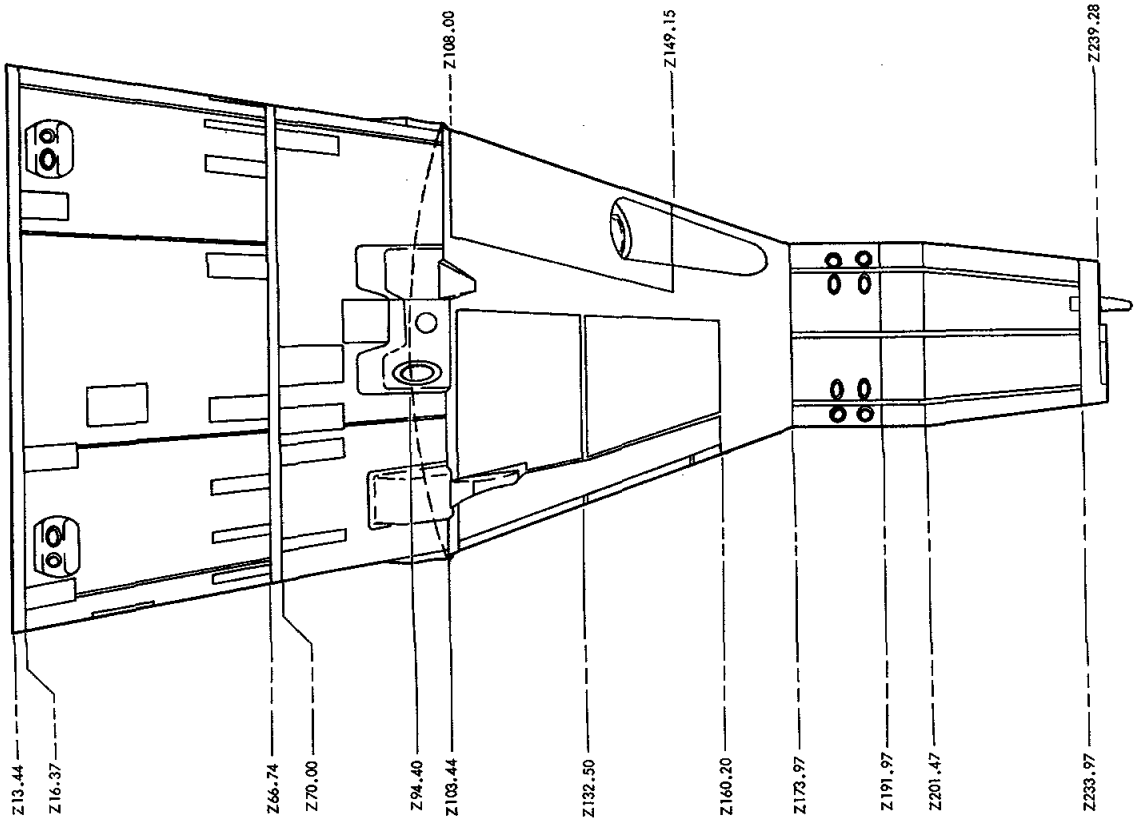


Figure 2-3 Spacecraft Dimensions



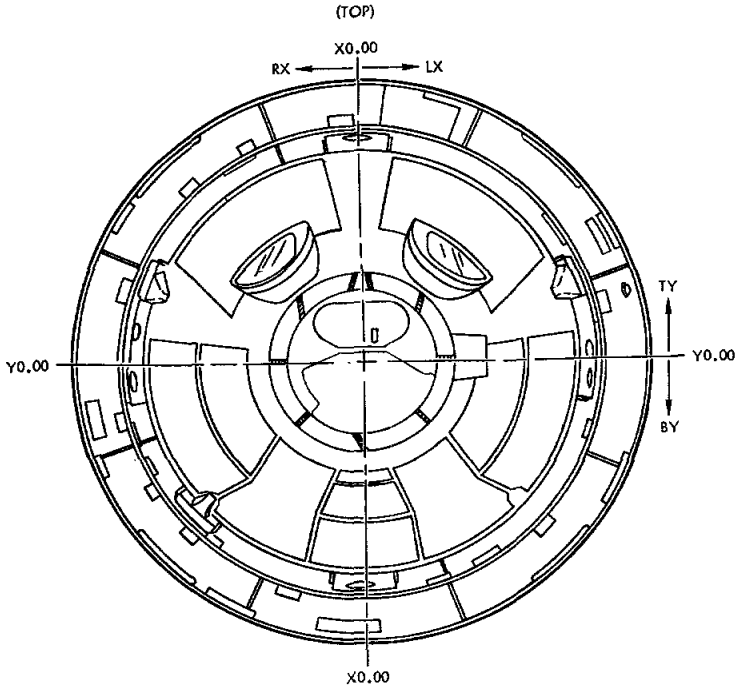


Figure 2-4 Stations Diagram

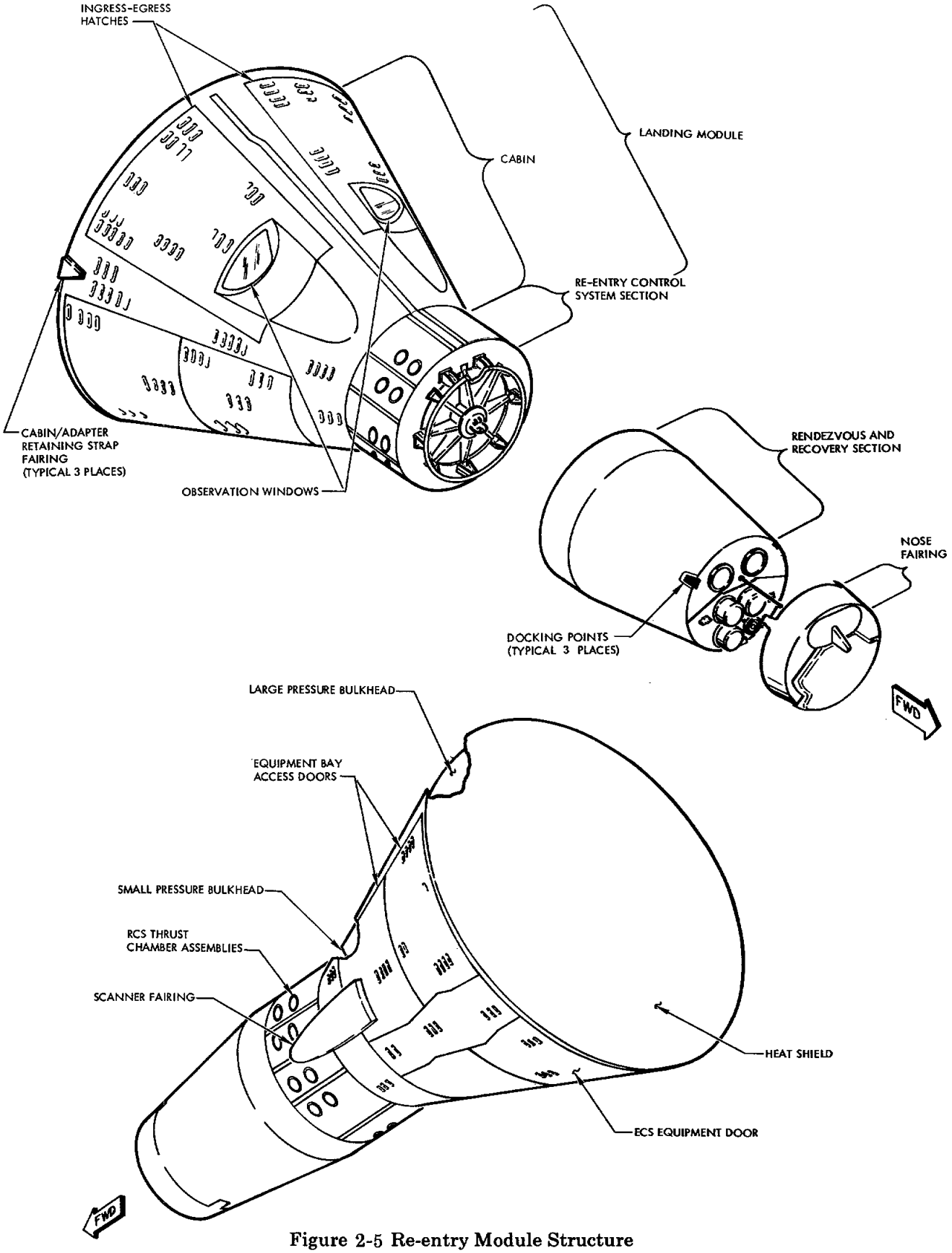


Figure 2-5 Re-entry Module Structure

RE-ENTRY CONTROL SYSTEM SECTION

The RCS section is located between, and mated to, the R and R and cabin sections of the spacecraft (Figure 2-5). This section is cylindrical in shape and is constructed of an inner titanium alloy cylinder, eight stringers, two rings and eight beryllium shingles for its outer skin. The RCS section is designed to house the fuel and oxidizer tanks, valves, tube assemblies, and thrust chamber assemblies for the RCS.

A parachute adapter assembly is installed on the forward face of the RCS section for attachment of the main parachute.

CABIN

The cabin (Figure 2-5), similar in shape to a truncated cone, is mated to the RCS section and the adapter. The cabin has an internal pressure vessel (Figure 2-6) shaped to provide an adequate crew station with a proper water flotation attitude. The shape of the pressure vessel also allows space between it and the outer conical shell for the installation of equipment.

The basic cabin structure consists of a fusion welded titanium frame assembly to which the side panels, small and large pressure bulkheads and hatch sill are seam welded. The side panels, small and large pressure bulkheads are of double skin construction and reinforced by stiffeners spotwelded in place. Two hatches are hinged to the hatch sill for pilot ingress and egress. For heat protection, the outer conical surface is covered with Rene' 41 shingles and an ablative heat shield is attached to the large end of the cabin section.

A spring loaded hoist loop, located near the heat shield between the hatch openings, is errected after landing to facilitate engagement of a hoisting hook for spacecraft retrieval.

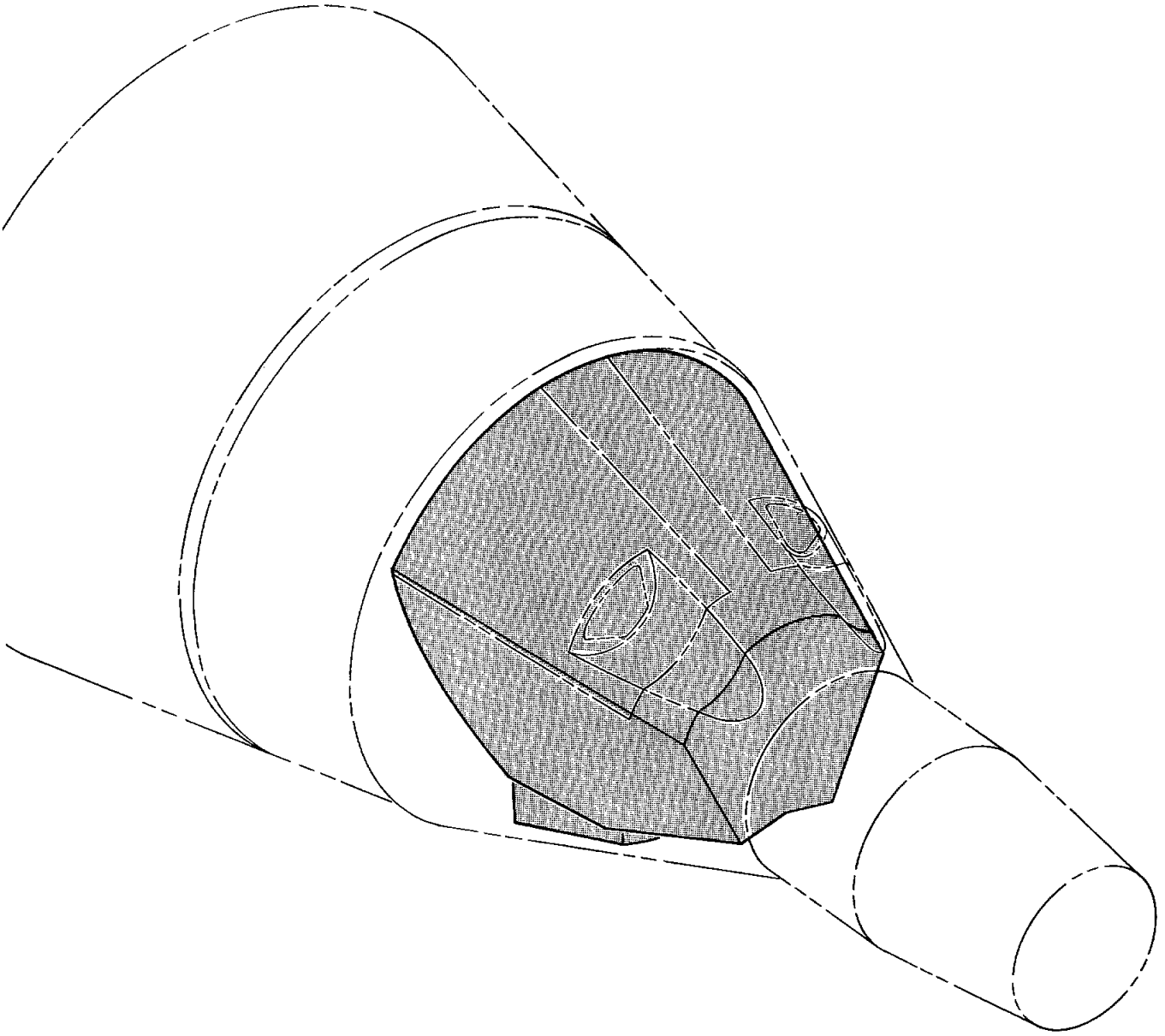


Figure 2-6 Cabin Pressure Vessel

Equipment Bays

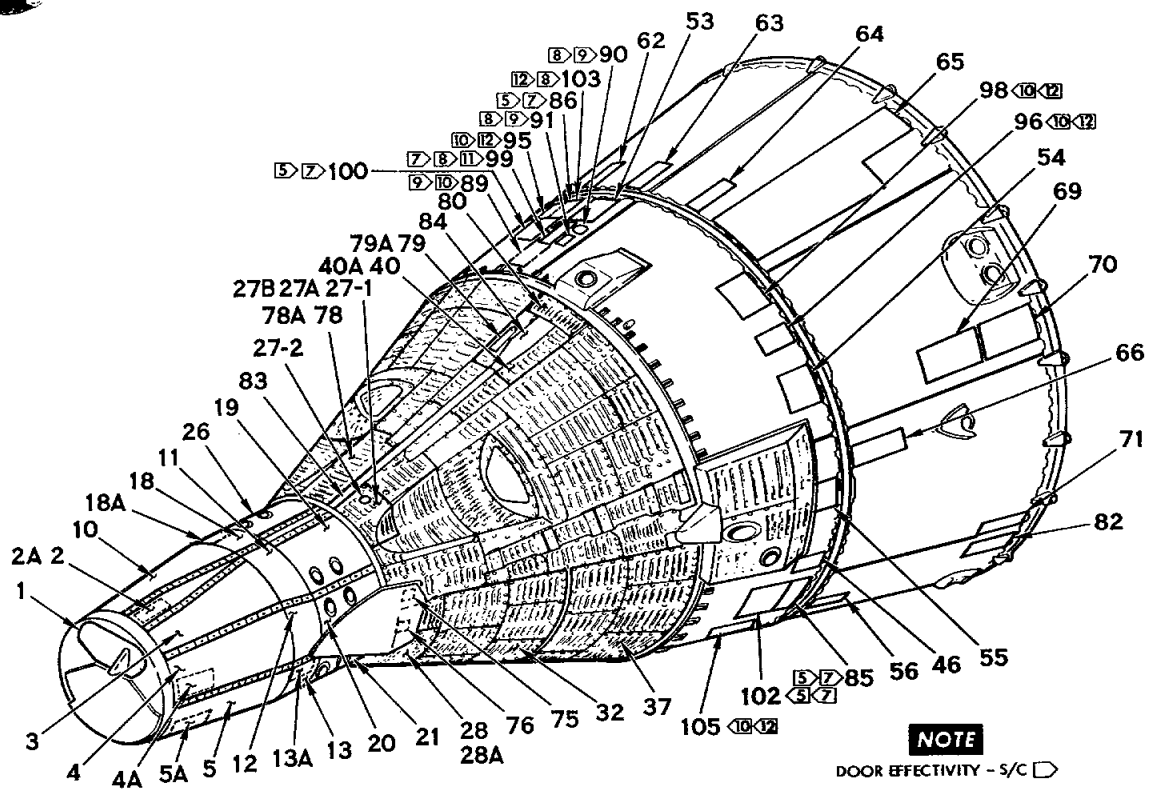
The equipment bays are located outside the cabin pressure vessel (Figure 2-7). Two bays are located outboard of the side panels and one bay beneath the pressure vessel floor. The bays are structurally designed for mounting of the equipment.

Doors

To enclose the side equipment bays, two structural doors are provided on each side of the cabin (Figure 2-7). These doors provide access to the components installed in the equipment bays. The main landing gear bays, located below the left and right equipment bays, are each enclosed by one door. The landing gear is not installed but fittings are provided for the attachment of the gear for future spacecraft. On the bottom of the cabin, between the landing gear doors, two additional doors are installed. The forward door allows access to the lower equipment compartment and the aft door provides access to the Environmental Control System compartment which is a portion of the pressure vessel.

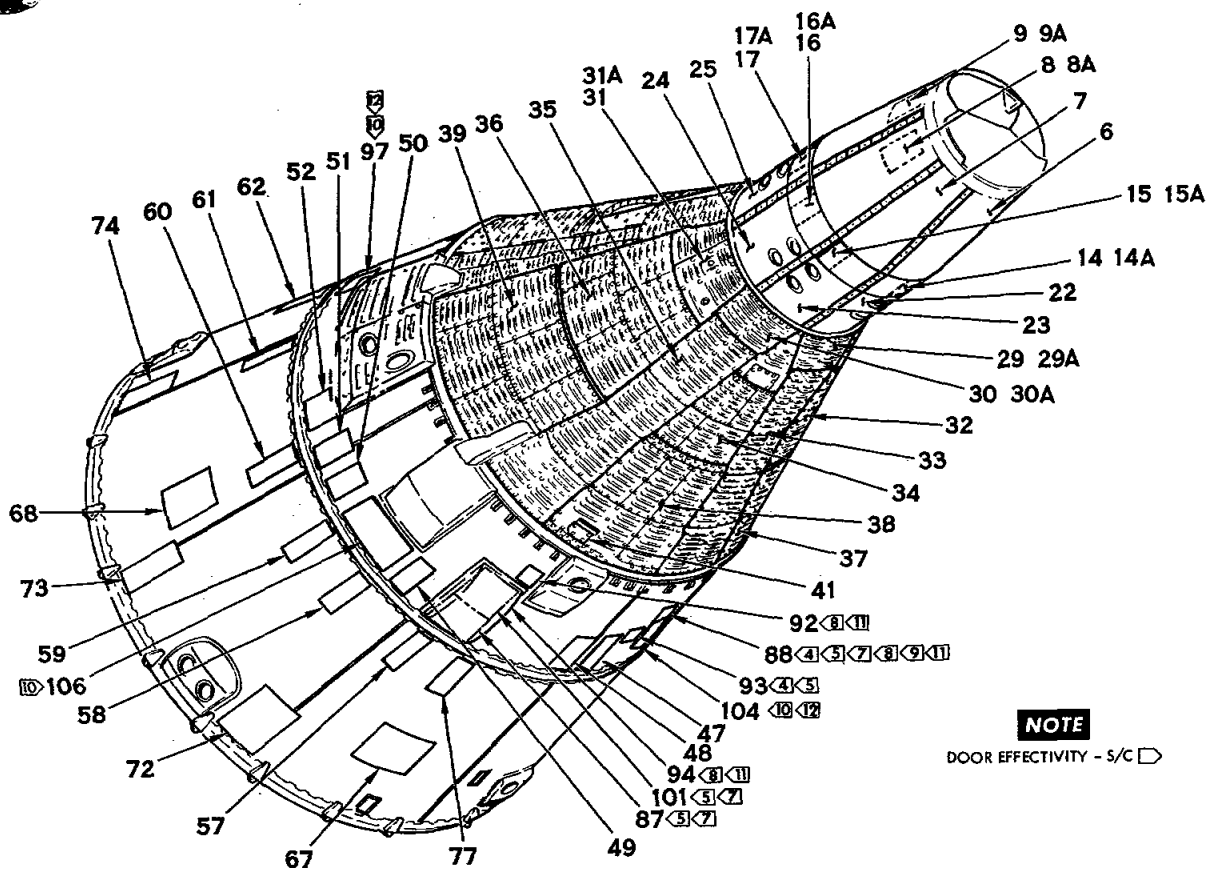
Hatches

Two large structural hatches (Figure 2-8) are incorporated for sealing the cabin ingress or egress openings. The hatches are symmetrically spaced on the top side of the cabin section. Each hatch is manually operated by means of a handle and mechanical latching mechanism. Each is hinged on the outboard side. In an emergency, the hatches are opened in a three sequence operation employing pyrotechnic actuators. When initiated, the actuators simultaneously unlock and open the mechanical latches, open the hatches and supply hot gases to ignite the ejection seat rocket catapults. An external hatch linkage fitting is incorporated to allow a recovery hatch handle to be inserted for opening the hatches from the outside. The recovery hatch handle is stowed on the main parachute adapter



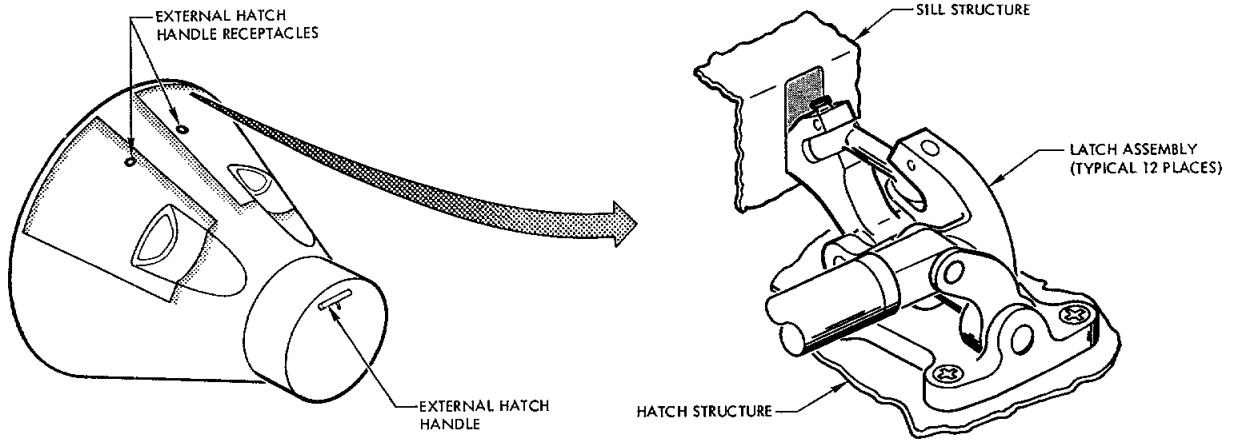
NO.	DESCRIPTION	NO.	DESCRIPTION	NO.	DESCRIPTION
1	DROGUE CHUTE DOOR	28	SHINGLE	79	RECOVERY LIGHT DOOR
2	DOCKING BAR CARTRIDGE ACCESS	28A	Z160.20 EQUIPMENT ACCESS	79A	RECOVERY LIGHT DOOR RELEASE MECHANISM
2A	PYRO ELECTRICAL DISCONNECT ACCESS	32	FORWARD EQUIPMENT BAY DOOR - LEFT	80	HOIST LOOP DOOR
3	SHINGLE	37	AFT EQUIPMENT BAY DOOR - LEFT	82	SHAPED CHARGE DETONATOR ACCESS
4	EMERGENCY DOCKING RELEASE CARTRIDGE AND GUILLOTINE CARTRIDGE ACCESS	40	SHINGLE	83	COVER ASS'Y - PARACHUTE CONTROL CABLES
4A	PILOT CHUTE DEPLOY SENSOR SWITCH ACCESS	40A	RECOVERY LIGHT AND HOIST LOOP RIGGING AND CARTRIDGE ACCESS	84	COVER ASS'Y. - PARACHUTE CONTROL CABLES
5	SHINGLE	46	SEPARATION SENSING SWITCH ACCESS	85	RADIOMETER
5A	RADAR ACCESS	53	OAMS LINE GUILLOTINE ACCESS	86	CRYO SPECTROMETER/INTERFEROMETER
10	SHINGLE	54	F.L.S.C. TUBING CUTTER ACCESS	89	MICROMETEORITE EXPERIMENT
11	INTERFACE ACCESS	55	FORWARD MANEUVERING ENGINE ACCESS	90	UHF - VHF POLORIZATION
12	INTERFACE ACCESS	56	FUEL CELL SERVICE ACCESS	91	UHF - VHF POLORIZATION ACCESS
13	INTERFACE ACCESS	62	OAMS OXIDIZER PURGE ACCESS	95	PITCH ION SENSOR ACCESS
13A	GUILLOTINE CARTRIDGE ACCESS	63	OAMS LINE GUILLOTINE ACCESS	96	YAW ION SENSOR ACCESS
18	INTERFACE ACCESS	64	OAMS OXIDIZER PURGE ACCESS	98	YAW SENSOR SYSTEM
18A	PYROTECHNIC SWITCH CARTRIDGE AND BRIDLE DISCONNECT CARTRIDGE ACCESS	65	OAMS MODULE SERVICE ACCESS	99	NUCLEAR EMULSION
19	RE-ENTRY CONTROL SYSTEM ACCESS	66	ECS SERVICE ACCESS	100	CRYO SPECTROMETER/INTERFEROMETER ACCESS
20	RE-ENTRY CONTROL SYSTEM ACCESS	69	ECS PUMP MODULE SERVICE ACCESS	102	RADIOMETER ACCESS
21	RE-ENTRY CONTROL SYSTEM ACCESS	70	ECS PUMP MODULE SERVICE ACCESS	103	SPACE POWER TOOL ACCESS
26	RE-ENTRY CONTROL SYSTEM ACCESS	71	SEPARATION SENSING SWITCH ACCESS	105	BETA SPECTROMETER
27-1	SHINGLE	75	ELECTRICAL DISCONNECT ACCESS		
27-2	FRESH AIR DOOR	76	ELECTRICAL DISCONNECT ACCESS		
27A	Z160.20 EQUIPMENT ACCESS	78	SHINGLE		
27B	Z160.20 EQUIPMENT ACCESS	78A	Z160.20 EQUIPMENT ACCESS		

Figure 2-7 Access Doors Spacecraft 5,6,8 and Up (Sheet 1 of 2)

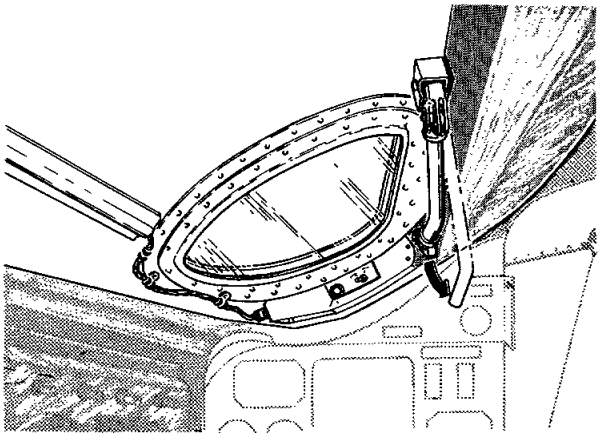


NO.	DESCRIPTION	NO.	DESCRIPTION	NO.	DESCRIPTION
6	EMERGENCY DOCKING RELEASE CARTRIDGE AND GUILLOTINE CARTRIDGE ACCESS	29A	Z160.20 EQUIPMENT ACCESS	51	B. I. A. RELAY PANEL ACCESS
7	SHINGLE	30	SHINGLE	52	FORWARD MANEUVERING ENGINE ACCESS
8	SHINGLE	30A	Z160.20 EQUIPMENT ACCESS	57	SHAPED CHARGE DETONATOR ACCESS
8A	RADAR ACCESS	31	SHINGLE	58	FUEL CELL SERVICE ACCESS
9	EMERGENCY DOCKING RELEASE CARTRIDGE AND GUILLOTINE CARTRIDGE ACCESS	31A	Z160.20 EQUIPMENT ACCESS	59	GUILLOTINE CARTRIDGE ACCESS
9A	DROGUE CHUTE DEPLOY SENSOR SWITCH ACCESS	32	FORWARD EQUIPMENT BAY DOOR - LEFT	60	GUILLOTINE CARTRIDGE ACCESS
14	INTERFACE ACCESS	33	MAIN LANDING GEAR DOOR - LEFT	61	OAMS FUEL PURGE ACCESS
14A	GUILLOTINE ANVIL ACCESS	34	CENTER EQUIPMENT BAY DOOR - FORWARD	67	ENGINE TO SCUPPER INTERFACE ACCESS
15	INTERFACE ACCESS	35	MAIN LANDING GEAR DOOR - RIGHT	68	ELECTRONIC MODULE TEST ACCESS
15A	GUILLOTINE ANVIL ACCESS	36	FORWARD EQUIPMENT BAY DOOR - RIGHT	72	GUILLOTINE CARTRIDGE AND LAUNCH VEHICLE ELEC. CONN. ACCESS
16	INTERFACE ACCESS	37	AFT EQUIPMENT BAY DOOR - LEFT	73	SEPARATION SENSING SWITCH ACCESS
16A	GUILLOTINE CARTRIDGE ACCESS	38	E.C.S. BAY DOOR	74	SHAPED CHARGE DETONATOR ACCESS
17	INTERFACE ACCESS	39	AFT EQUIPMENT BAY DOOR - RIGHT	77	FUEL CELL PURGE ACCESS
17A	PARAGLIDER ELECT. CONTROL BOX ACCESS	41	PURGE FITTING ACCESS	87	SPECTROMETER/INTERFEROMETER
22	RE-ENTRY CONTROL SYSTEM ACCESS	47	RELAY PANEL ACCESS	88	ELECTROSTATIC CHARGE SENSOR
23	RE-ENTRY CONTROL SYSTEM ACCESS	48	RELAY PANEL ACCESS	92	LLTV ACCESS
24	RE-ENTRY CONTROL SYSTEM ACCESS	49	SEPARATION SENSING SWITCH ACCESS	93	ELECTROSTATIC CHARGE ACCESS
25	RE-ENTRY CONTROL SYSTEM ACCESS	50	GUILLOTINE CARTRIDGE ACCESS	94	LLTV FAIRING
29	SHINGLE			97	PITCH SENSOR SYSTEM
				101	SPECTROMETER/INTERFEROMETER ACCESS
				104	BETA SPECTROMETER ACCESS
				106	MAGNETOMETER ACCESS

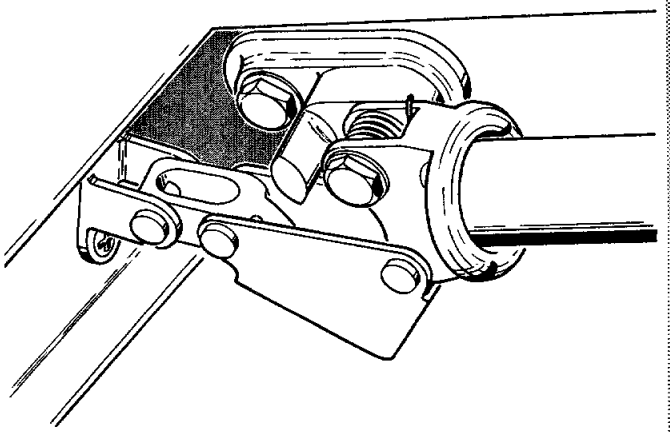
Figure 2-7 Access Doors Spacecraft 5, 6, 8 and Up (Sheet 2 of 2)



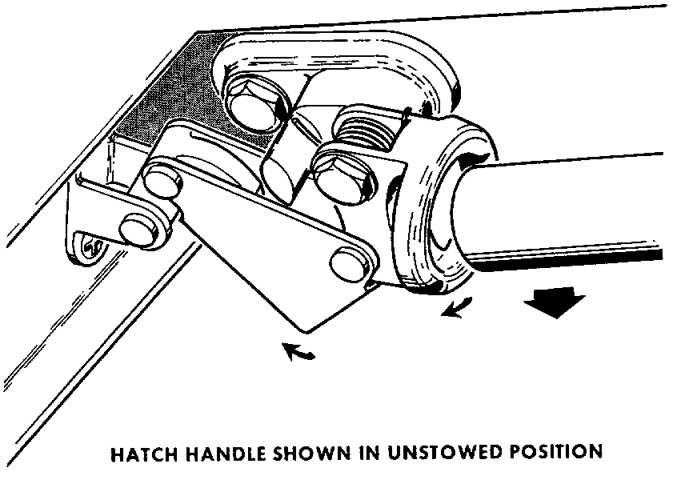
HATCH LATCH SHOWN IN LATCHED POSITION



HATCH INTERIOR VIEW



HATCH HANDLE SHOWN IN STOWED POSITION



HATCH HANDLE SHOWN IN UNSTOWED POSITION

Figure 2-8 Spacecraft Ingress/Egress Hatches

assembly located on the forward face of the RCS section. A hatch curtain (Figure 2-9) is stowed along the hinge of each hatch. After water landing, when the hatches are open, the curtains are installed to help prevent water from entering the cabin.

Windows

Each of the ingress/egress hatches incorporates a visual observation window (Figure 2-10). Each window consists of an inner and outer glass assembly. The outer assembly is a single flat pane and the inner panel assembly consists of two flat panes. The panes consist of Vycor (96% silica). The panes in the right window are optically ground for better resolution. Each surface of each pane, with the exception of the outer surface of the outer pane, is coated to lessen reflection and glare from cabin lights and to aid in impeding ultraviolet radiation into the cabin compartment.

Heat Shield

The heat shield is a dish-shaped structure composed of silicone elastomer filled, phenolic impregnated, fiberglass honeycomb. It is an ablative device, 90 inches in diameter with a spherical radius of 144 inches. The shield is designed to protect the re-entry module from extreme thermal conditions during re-entry into the atmosphere. The device is attached to the large diameter end of the cabin structure by 1/4 inch bolts.

Shingles

The external surface of the cabin is made up of beaded shingles of Rene' 41. The R and R and RCS section surfaces are made up of umbeaded shingles of beryllium. The shingles protect the re-entry module structure from excessive heat and provide additional rigidity for the cabin. The shingles are black on the outer surface

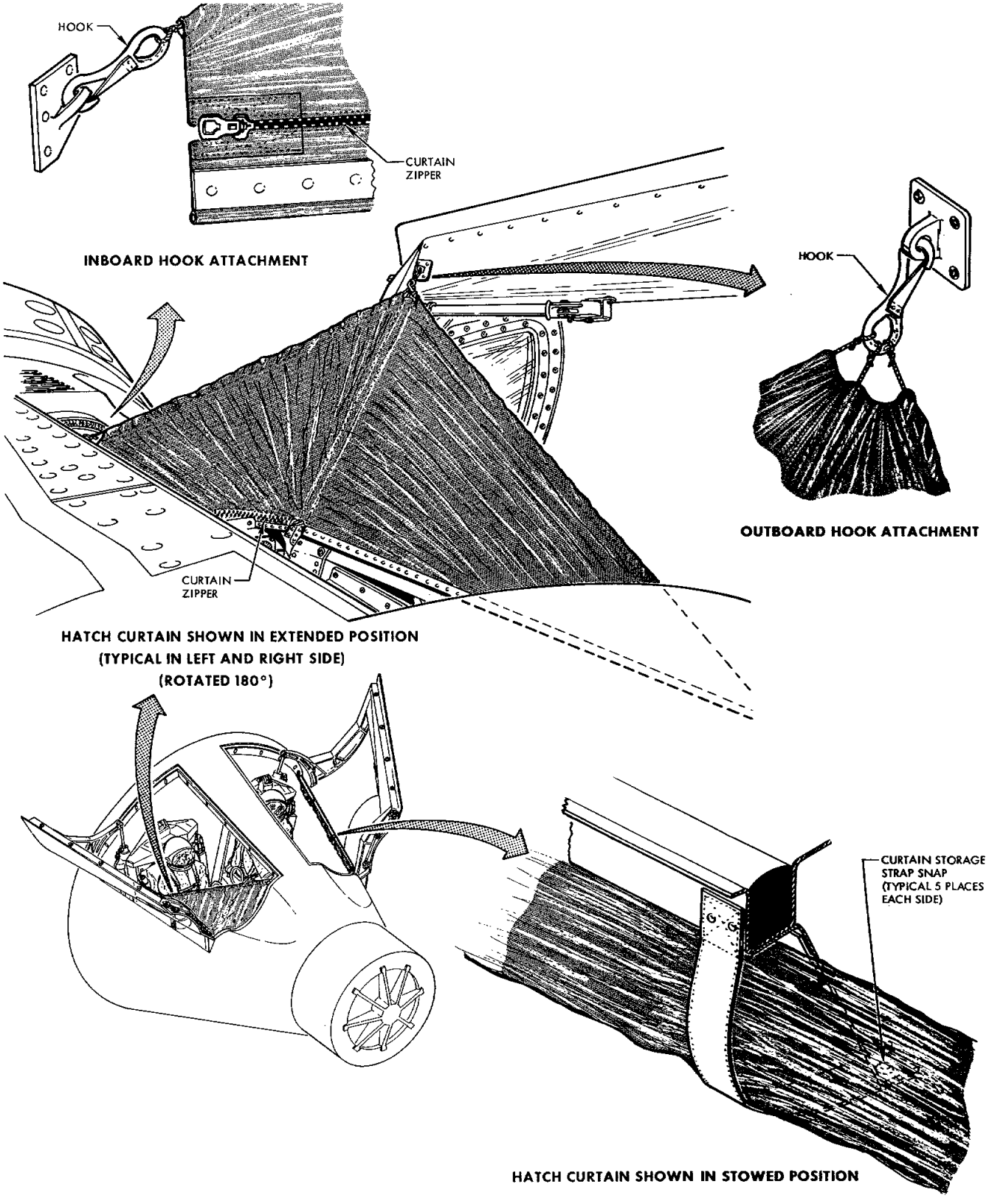


Figure 2-9 Hatch Curtain

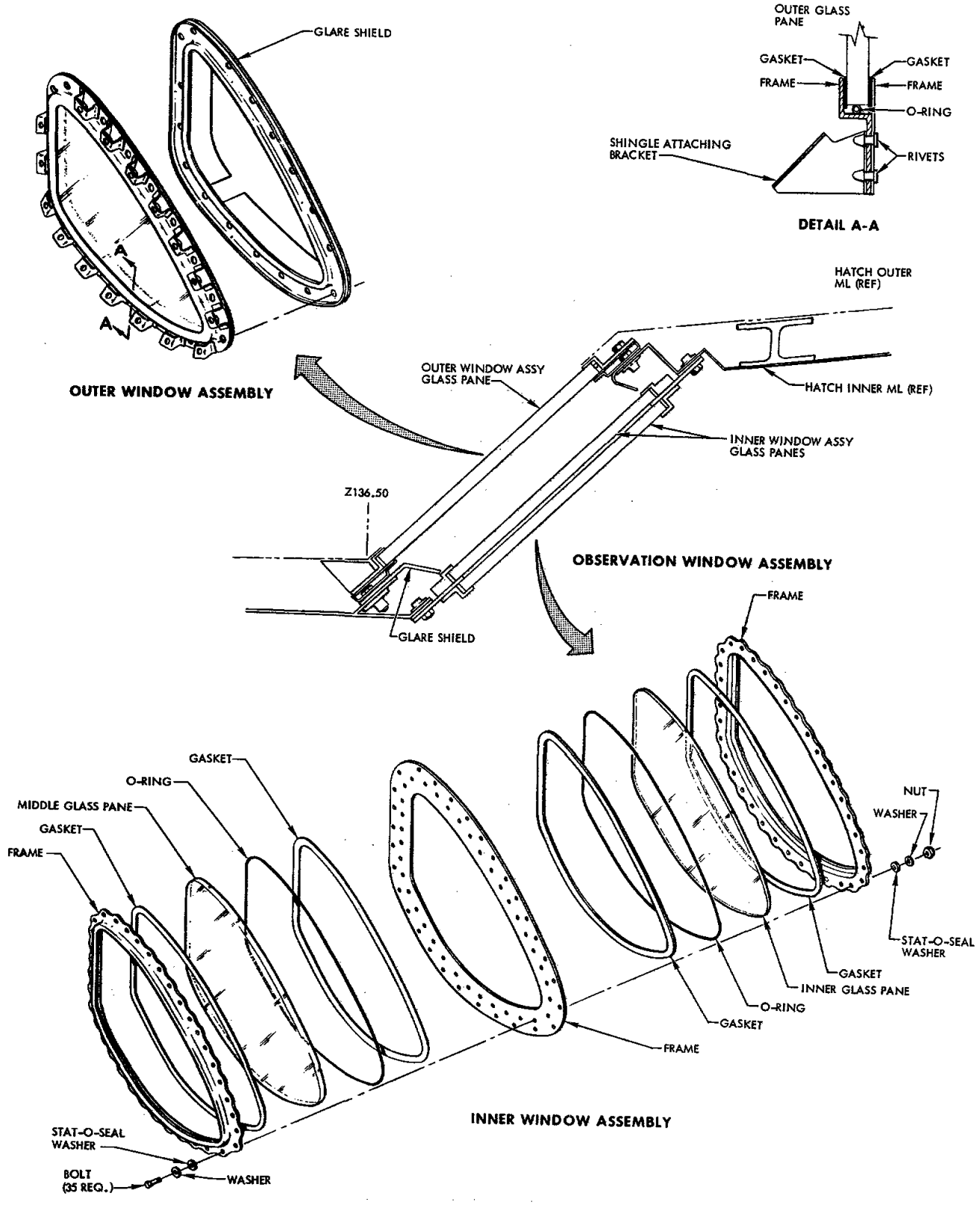


Figure 2-10 Observation Window

to control thermal radiation. The inner surface of the beryllium shingles are coated with gold to provide a low emissivity surface.

ADAPTER

The adapter functions to mate the spacecraft to the launch vehicle, to provide for mounting equipment and retrograde rockets , and to serve as a radiator for the spacecraft coolant system. The adapter (Figure 2-2) is a truncated cone-shape, simimonocoque structure consisting of circumferential aluminum rings, extruded magnesium alloy stringers, and magnesium skin. The extruded stringers are designed in a bulb-tee shape to provide a flow path for the liquid coolant which transfers heat to the adapter skin for radiation to space. The outer surface of the skin is coated with white ceramic type paint and the inner surface is covered with aluminum foil. The inner adapter surfaces of spacecraft 9 through 12 are gold plated. The forward end of the adapter is coupled to the aft end of the re-entry module by utilizing three titanium tension straps (Figure 2-11).

RETROGRADE SECTION

The retrograde section, the smaller end of the adapter, provides for installation of four retrograde rockets and six Orbital Attitude Maneuvering System thrust chamber assemblies. To provide for the installation of the retrograde rockets, the retrograde section employs an aluminun I beam support assembly. The I beams are assembled in the form of a cruciform with one retrograde rocket mounted in each quadrant.

EQUIPMENT SECTION

The equipment section is the larger diameter end of the adapter. The section provides hard points for the attachment of structural modules for the OAMS tanks, Environmental Control System primary oxygen supply, fuel cell (batteries on

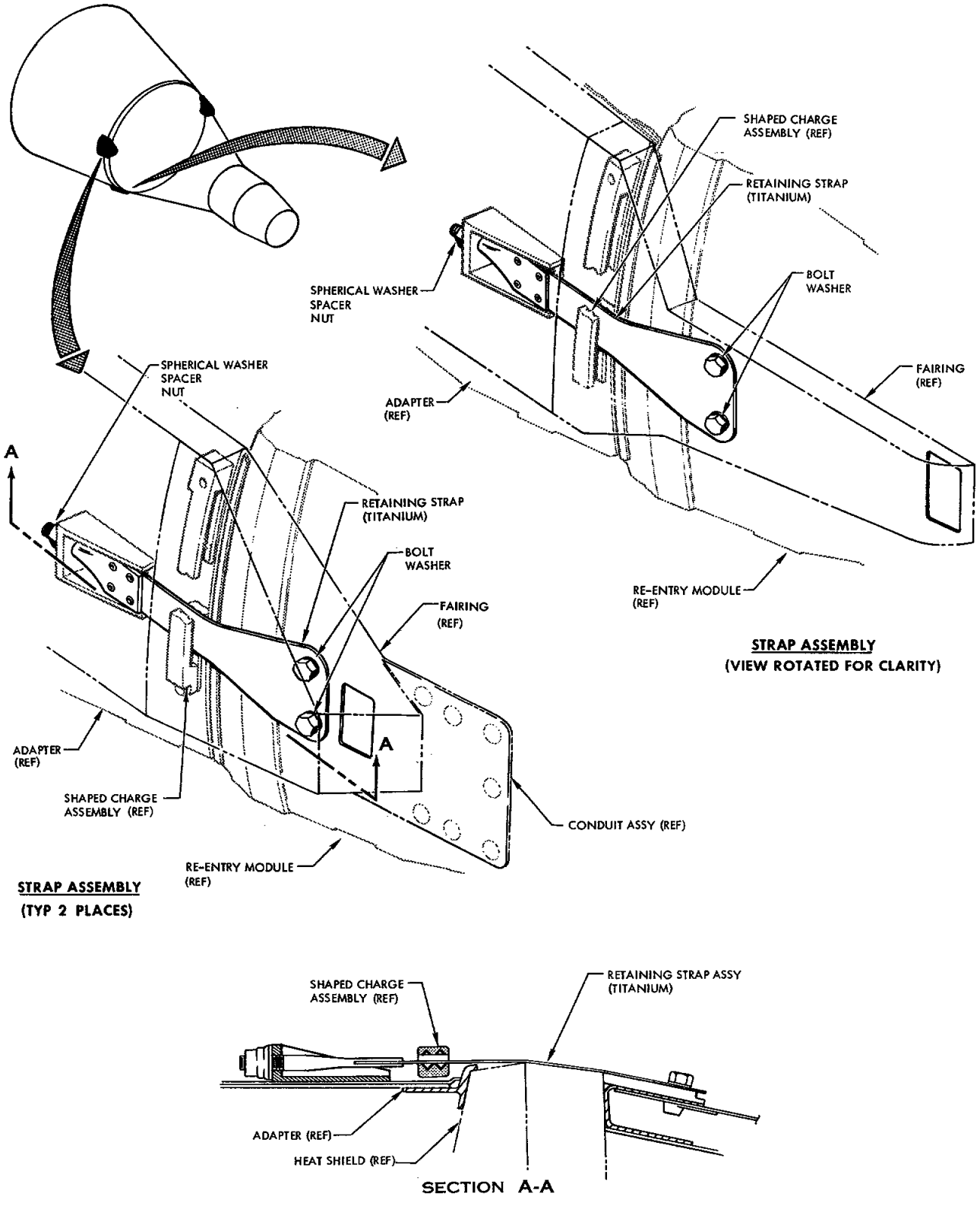


Figure 2-11 Re-Entry Module-Adapter Retaining Straps

spacecraft 6), coolant, electrical and electronic components, Extra Vehicular Activity (EVA) equipment on spacecraft 9 through 12, and Rendezvous Evaluation Pod on spacecraft 5 only. A honeycomb blast shield is provided above the modules to shield the equipment section and booster dome from excessive heat during retrograde rocket firing under abort conditions. Ten OAMS thrust chamber assemblies are mounted on the large diameter end of the equipment section. A gold deposited fiberglass temperature control cover protects the equipment from solar radiation through the open end of the adapter after separation from the launch vehicle.

SPACECRAFT/LAUNCH VEHICLE MATING

The spacecraft is mated to the Titan II Launch Vehicle with a machined aluminum alloy ring (Figure 2-12). This ring, 120 inches in diameter, mates with the launch vehicle mating ring. Twenty bolts secure the rings together. To provide for alignment, the launch vehicle incorporates one steel 3/26 inch diameter alignment pin located at TY and four index marks. To separate the spacecraft from the launch vehicle, a pyrotechnic charge is fired, severing the adapter section approximately $1\frac{1}{2}$ inches above the launch vehicle/spacecraft mating point.

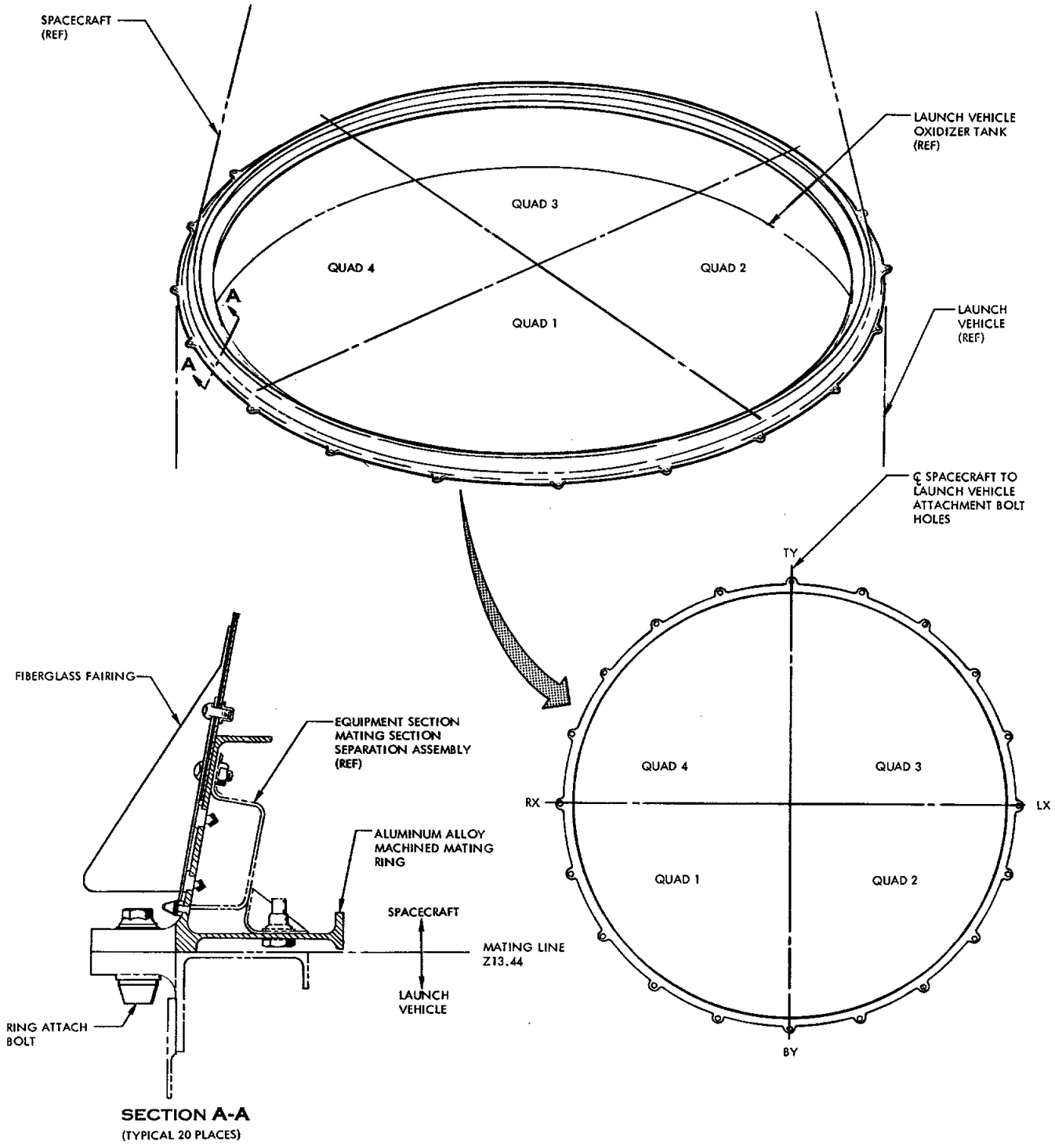


Figure 2-12 Spacecraft/Launch Vehicle Mating Ring

CABIN INTERIOR ARRANGEMENT

Section III

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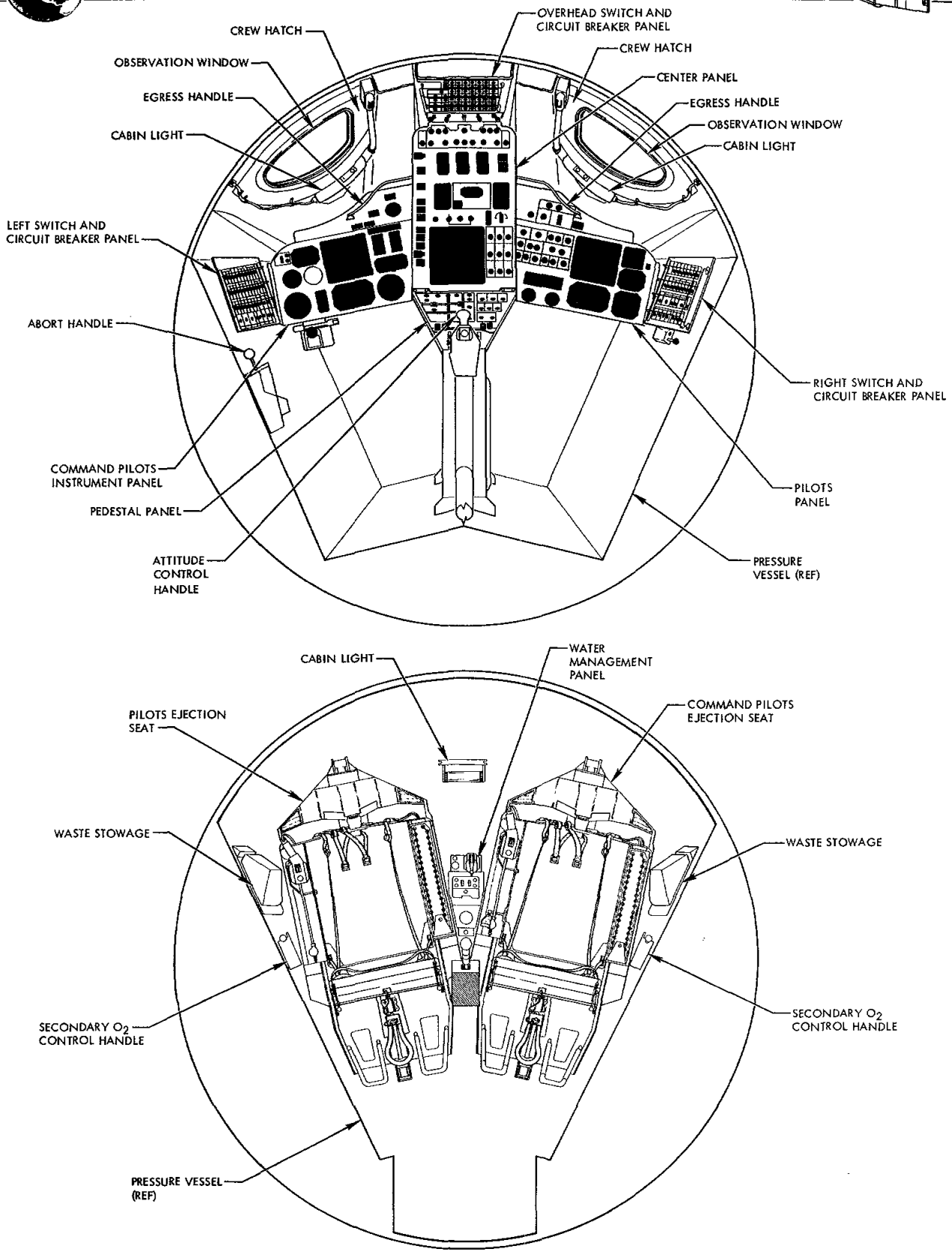


Figure 3-1 Cabin Equipment (Typical)

SECTION III CABIN INTERIOR ARRANGEMENT

GENERAL

The equipment within the cabin is arranged to permit the command pilot, seated to the left, and the pilot, seated to the right, to operate the controls and observe displays and instruments in full pressure suits in the restrained or unrestrained position. The cabin air outflow is regulated during launch to establish and maintain a 5.5 psi differential pressure between the cabin and outside ambient condition. The cabin is maintained at a nominal 5.1 psia throughout the flight by a cabin pressure regulator. The cabin equipment (Figure 3-1) basically consists of crew ejection seats, instrument panels and controls, lighting, food, water, waste collection, and miscellaneous equipment.

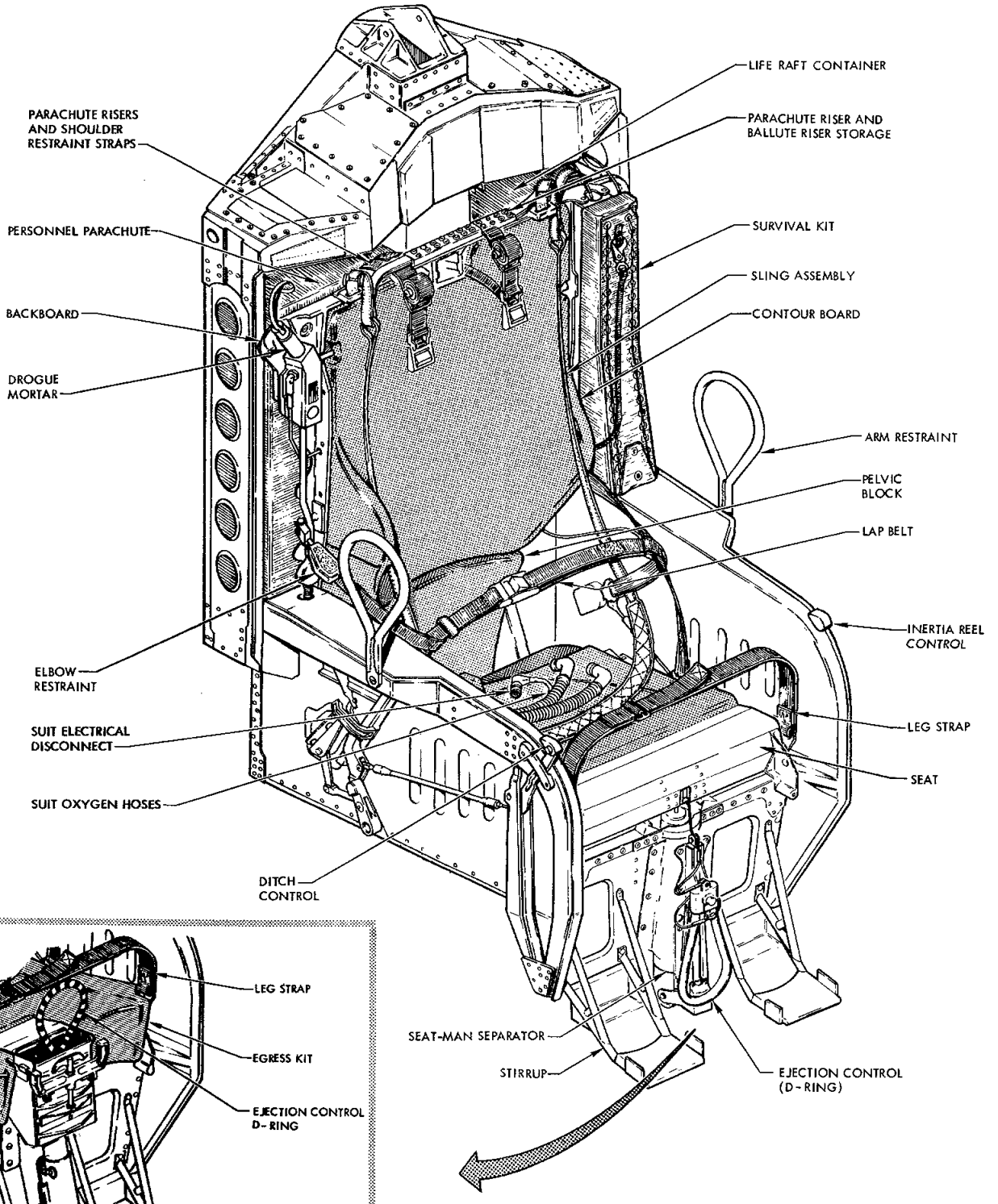
CREW SEATING

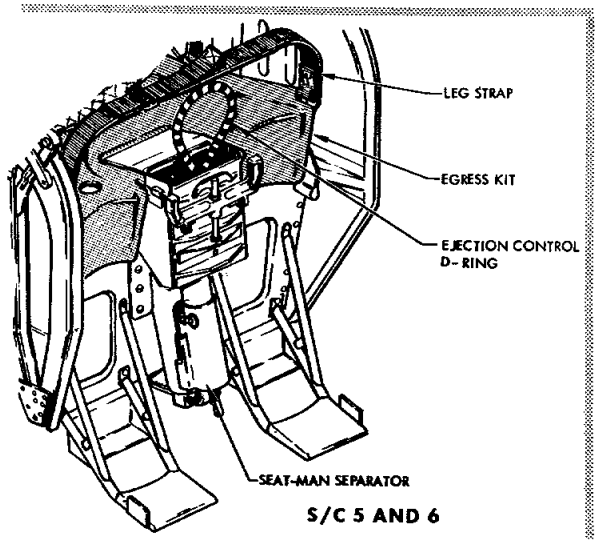
The crew members are seated in the typical command pilot and pilot fashion, faced toward the small end of the re-entry module. The seats are canted 12° outboard and 8° forward to assure separation and to provide required elevation in the event an off the pad ejection is necessitated.

Crew seating provisions include seats, restraint mechanisms, seat ejection devices, seat man separator, survival gear, and an egress kit assembly effective spacecraft 5 and 6 only.

SEAT DESCRIPTION

The crew seats (Figure 3-2) are all metal built-up assemblies consisting of a torque box framed seat bucket, channeled backs and arm rests. The seat has lateral and vertical stiffeners, designed for a single moment of thrust. The





NOTE

COMMAND PILOT EJECTION SEAT ILLUSTRATED. HARNESS RELEASE ACTUATOR IS LOCATED ON OUTBOARD SIDE OF SEAT.

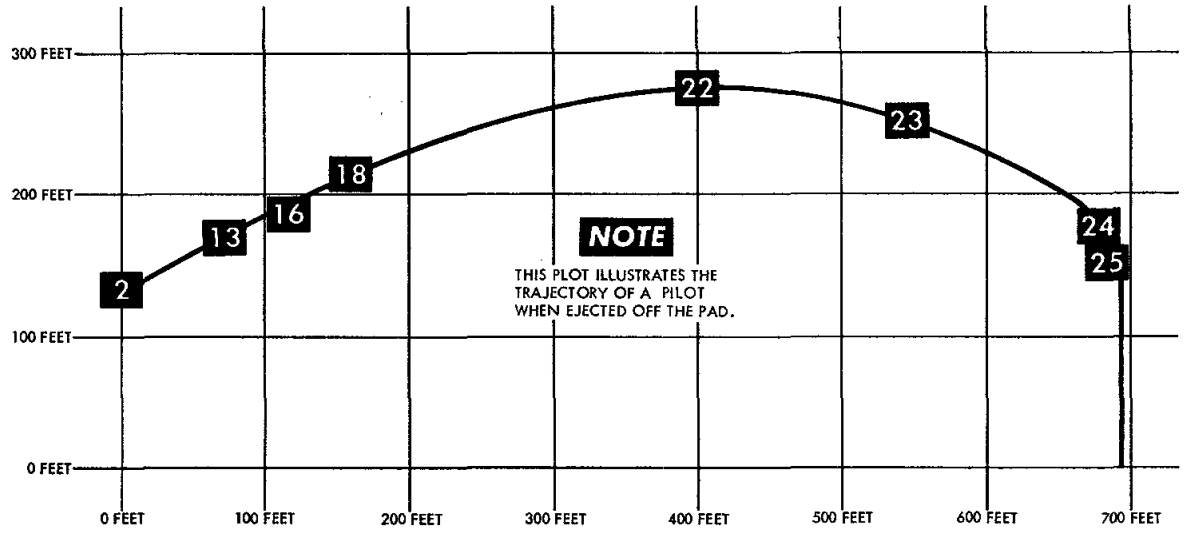
Figure 3-2 Gemini Ejection Seat Assembly

seat is supported at a single point at the top of the seat back. At this point, the seat bolts to the rocket/catapult. Each seat is supported against fore, aft, and side movement by slide blocks mounted on the seats and retained in tee type rail assemblies attached to the large pressure bulkhead. The seats incorporate a padded contoured headrest to support the pilots helmet. Each seat also incorporates a restraint system, harness release system and a seat/man separator.

SEAT EJECTION SYSTEM

The seat ejection system (Figure 3-3) provides the crew with a means of escaping from the vicinity of the spacecraft in the event of an abort or in an emergency condition during launch or re-entry. Crew member seats are ejected by means of rocket/catapults. Hot gas from each of the hatch actuators is routed to the appropriate seat catapult where dual firing pins strike dual percussion primers, thereby igniting the seat rocket/catapult main charge and ejecting the seats from the spacecraft. Hot gas from the rocket/catapult main charge ignites the sustainer rocket and the rocket provides additional separation from the spacecraft. In the event ejection becomes necessary, after deployment of main landing system parachute and while descending in the two point suspension, it is mandatory that the main landing system parachute be jettisoned before ejecting from the spacecraft.

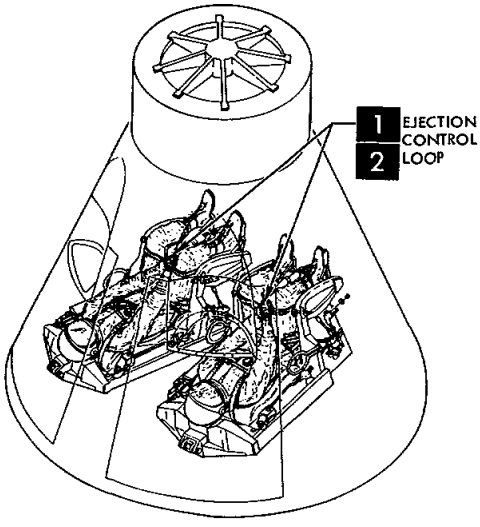
The ejection sequence is initiated by manually pulling either ejection control (D-ring) located on the front of the seat buckets. During the launch phase of flight each pilot erects and holds on the D-ring. This action aids in stabilizing the pilots arms and at the same time places them in a position for instant response. The D-rings are normally stowed at the front of the seat and are pinned in a



EJECTION SEAT TRAJECTORY PLOT

NOTE

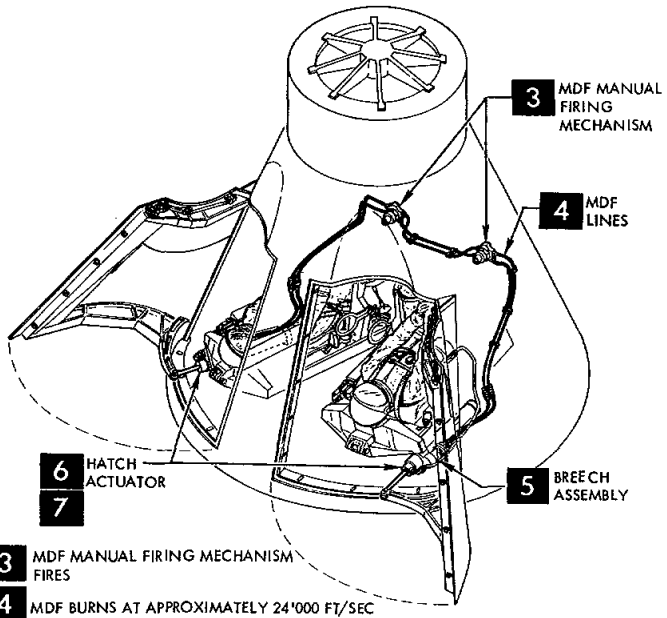
EITHER PILOT CAN EJECT SEATS.



1 BOTH PILOTS HOLD THE EJECTION CONTROL LOOP FIRMLY.

2 INITIATE EJECTION

EVENT TIME
0



3 MDF MANUAL FIRING MECHANISM FIRES

4 MDF BURNS AT APPROXIMATELY 24'000 FT/SEC

NOTE

MDF CROSSOVER NETWORK INITIATES SECOND SEAT INITIATOR. IMPULSE TRAVELS TO HATCH ACTUATORS IN 4 SEPARATE LINES.

EVENT TIME
.01

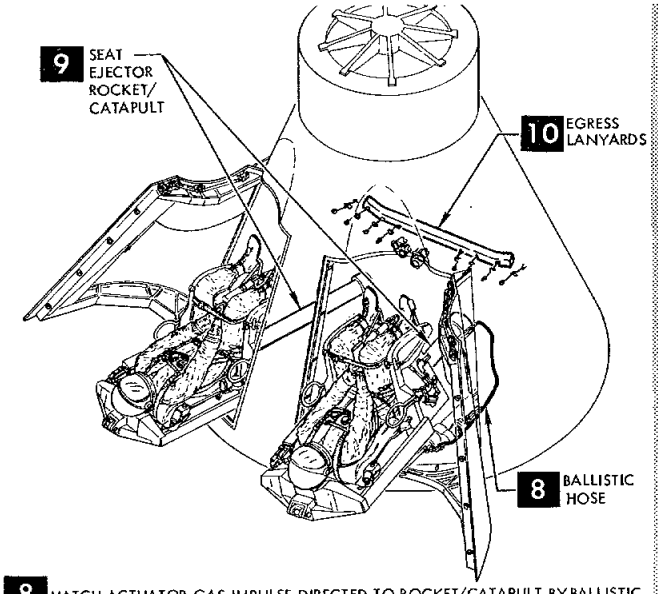
5 BREECH ASSEMBLY IGNITES.

6 HATCH IS UNLATCHED.

7 HATCH IS OPENED.

EVENT TIME
.24

Figure 3-3 Ejection Seat Sequence Of Operation (Sheet 1 of 4)



8 HATCH ACTUATOR GAS IMPULSE DIRECTED TO ROCKET/CATAPULT BY BALLISTIC HOSE

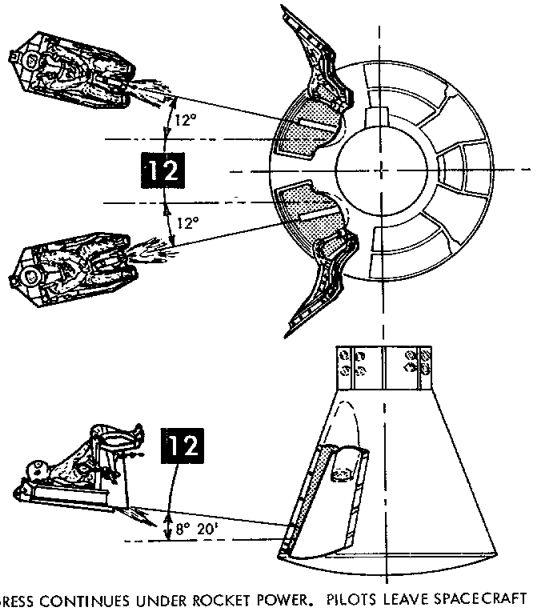
9 CATAPULT FIRES

10 HARNESS RELEASE ACTUATORS INITIATED. EGRESS LANYARDS PULLED, COMMUNICATION IS SEVERED.

EVENT TIME
.32

11 EJECTION SEAT MOVING UP, IGNITES EJECTION ROCKET APPROXIMATELY 4 INCHES FROM END OF RAIL TRAVEL.

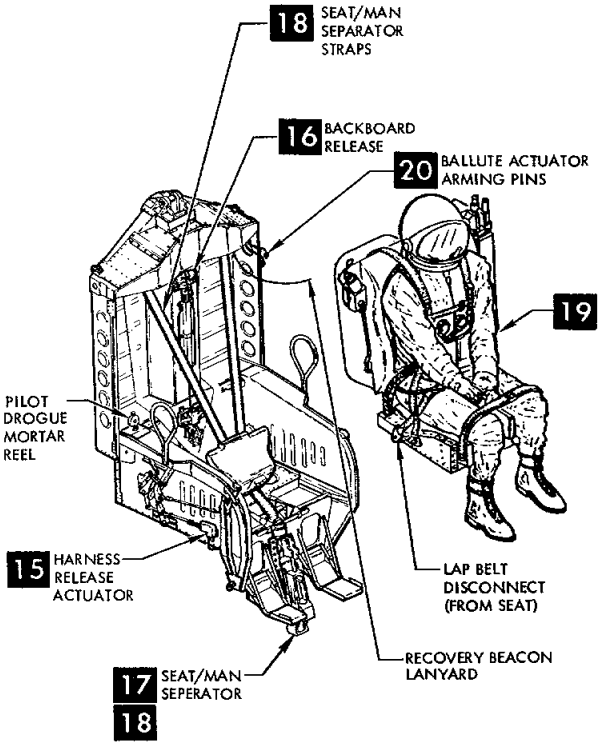
EVENT TIME
.39



12 EGRESS CONTINUES UNDER ROCKET POWER. PILOTS LEAVE SPACECRAFT AT 12° OUTBOARD OF "X" AXIS AND 8° 20' FORWARD OF "Z" AXIS.

13 ROCKET BURN OUT.

EVENT TIME
.66



14 EJECTION SEAT CONTINUES ON TRAJECTORY

15 HARNESS RELEASE ACTUATOR FIRES BY LANYARD PULL.

16 LAP BELT RELEASE ASSEMBLY ACTIVATED. BACKBOARD AND SURVIVAL GEAR ASSEMBLY RELEASED FROM SEAT.

EVENT TIME
1.41

17 HARNESS RELEASE ACTUATOR GAS IMPULSE DELIVERED TO SEAT/MAN SEPARATOR BY BALLISTIC HOSE.

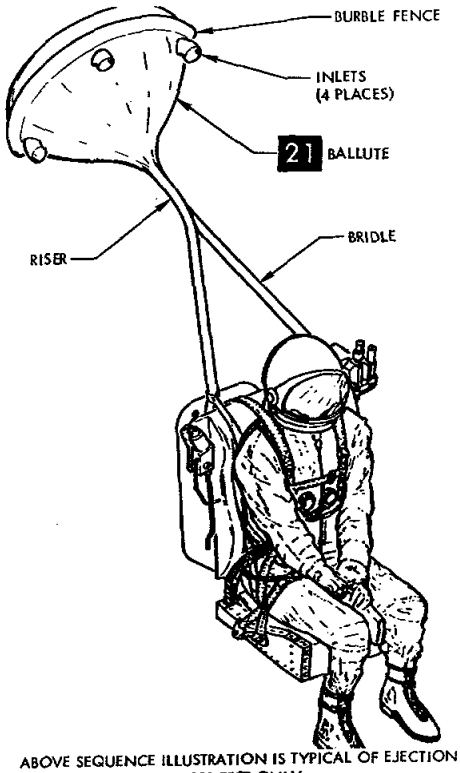
18 SEAT/MAN SEPARATOR SHOE EXTENDS AND REMOVES SLACK FROM STRAP ASSEMBLY.

19 PILOT WITH BACKBOARD AND SURVIVAL GEAR SEPARATE FROM SEAT

20 PILOT DROGUE MORTAR, BALLUTE SYSTEM AND RECOVERY BEACON INITIATED BY LANYARDS CONNECTED TO SEAT STRUCTURE.

EVENT TIME
1.50

Figure 3-3 Ejection Seat Sequence Of Operation (Sheet 2 of 4)



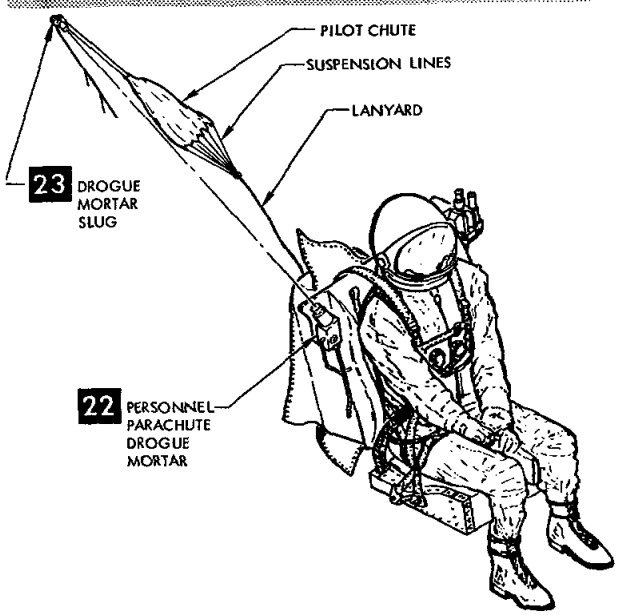
ABOVE SEQUENCE ILLUSTRATION IS TYPICAL OF EJECTION BETWEEN 7500 AND 40,000 FEET ONLY.

21 BALLUTE DEPLOYS AFTER A 5 SECOND DELAY.

NOTE

1. BALLUTE BAROSTAT HAS BEEN ACTIVATED TO JETTISON THE BALLUTE AT 7500 FEET.
2. TIME CHART APPLICABLE TO EJECTION ABOVE 7500 FEET ONLY.

EVENT TIME
6.50



NOTE

DROGUE MORTAR BAROSTAT IS ACTIVATED DURING SEAT/MAN SEPARATION TO DEPLOY THE PARACHUTE AT 5700 FEET OR BELOW.

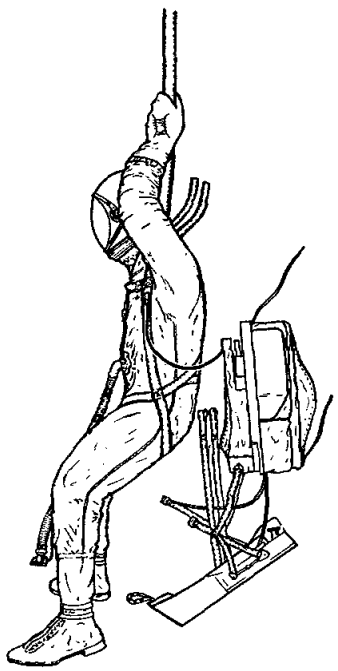
22 DROGUE MORTAR FIRES.

EVENT TIME
3.80

NOTE

EVENT TIMES ARE FOR EJECTION BELOW 5700 FEET ONLY.

23 MORTAR SLUG DEPLOYS AND INFLATES PILOT AND MAIN PARACHUTES.



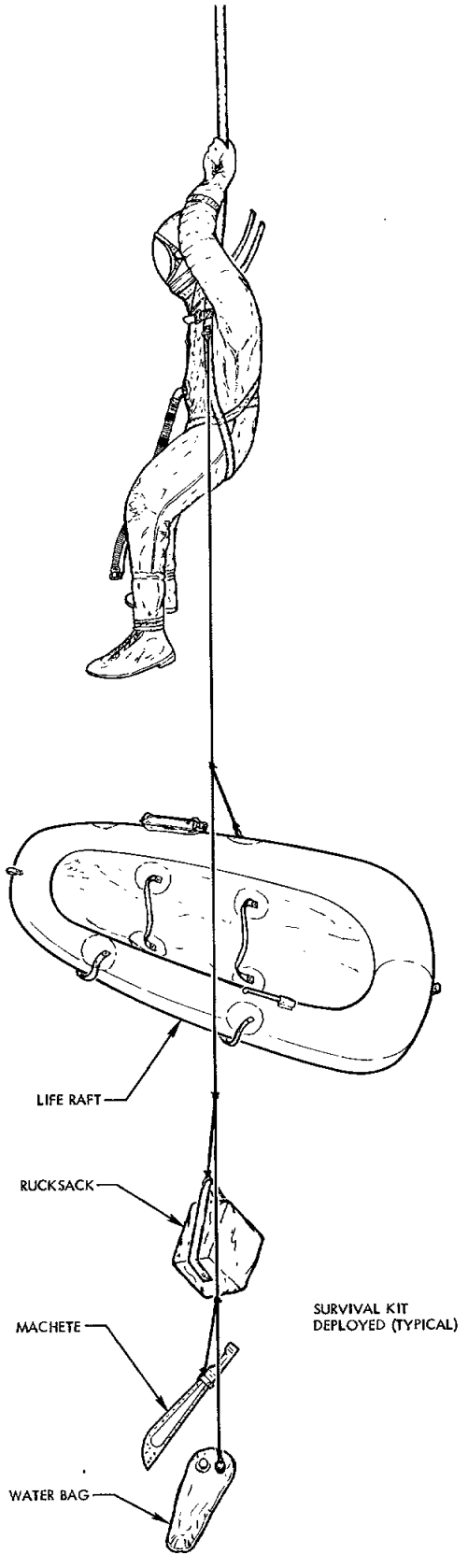
24 PARACHUTE FULLY INFLATED.

EVENT TIME
6.80

25 BACKBOARD AND SEAT SEPARATED FROM PILOT.

EVENT TIME
8.80

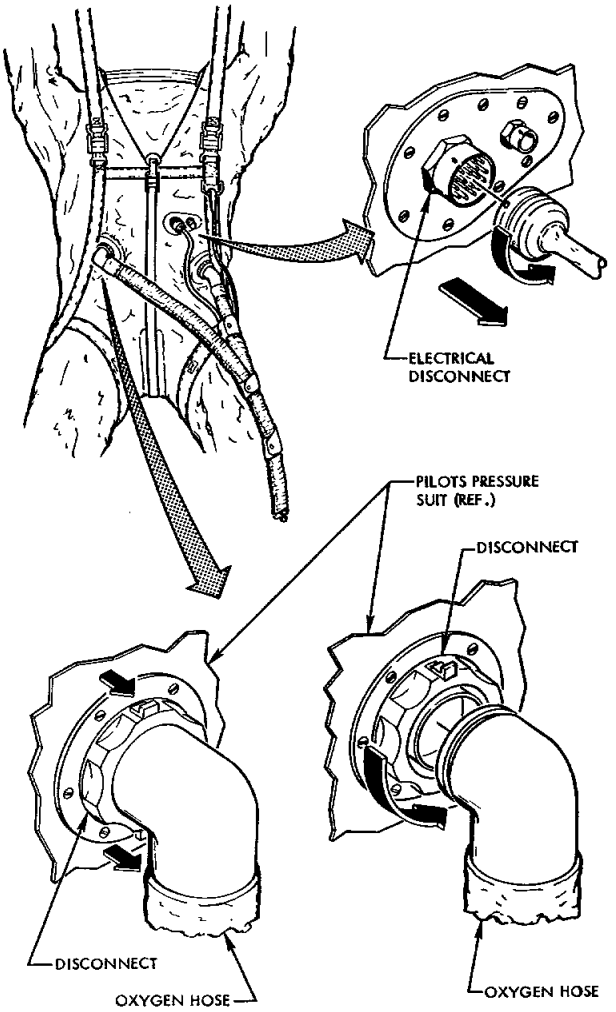
Figure 3-3 Ejection Seat Sequence Of Operation (Sheet 3 of 4)



SURVIVAL KIT DEPLOYED (TYPICAL)

NOTE

SURVIVAL EQUIPMENT DEPLOYED AS BACKBOARD FALLS FROM PILOT. THE LIFE RAFT IS INFLATED MANUALLY DURING PARACHUTE DESCENT ALL SURVIVAL EQUIPMENT IS SECURED TO PILOT BY A LANYARD.



NOTE

PILOT DISCONNECTS OXYGEN INLET AND OUTLET HOSES. OXYGEN CONNECTION IN PRESSURE SUIT IS SEALED CLOSED WHEN OXYGEN HOSE IS REMOVED.

Figure 3-3 Ejection Seat Sequence Of Operation (Sheet 4 of 4)

downward position at the front of the seat structure. The safety pin is removed during launch and re-entry and during orbit.

RESTRAINT SYSTEM

Each pilot is restrained in his ejection seat by a restraint system (Figure 3-4) consisting of personal harness, lap belt assembly, shoulder restraint, inertia reel and leg restraint. Other portions of the restraint system are part of the ejection seat (Figure 3-2). These seat restraints are the arm restraint loops, elbow restraint and foot stirrups. The restraint system provides adequate support and restraint during conditions of maximum acceleration and deceleration.

INERTIA REEL

The inertia reel (Figure 3-4) is a two position locking device, located on the rear of the backboard. Two straps connect the inertia reel and the personal harness to restrain the pilots forward movement. The inertia reel control handle is located on the front of the left arm rest and has two positions, manual lock and automatic lock. Orbital flight is accomplished with the inertia reel in the automatic lock position. Manual lock position is used during launch and re-entry. The manual lock position prevents the pilots shoulders from moving forward.

To release his shoulders, when the inertia reel is in the manual lock position, the pilot must position the control handle to the automatic position. The automatic lock allows the astronaut to move forward slowly a maximum of 18 inches but will lock with a 3 g deceleration. When the automatic lock has engaged, the lock will ratchet and permit movement back into the seat, but will not permit forward movement. The release of the automatic lock is accomplished by cycling the control handle to manual and back to automatic lock.

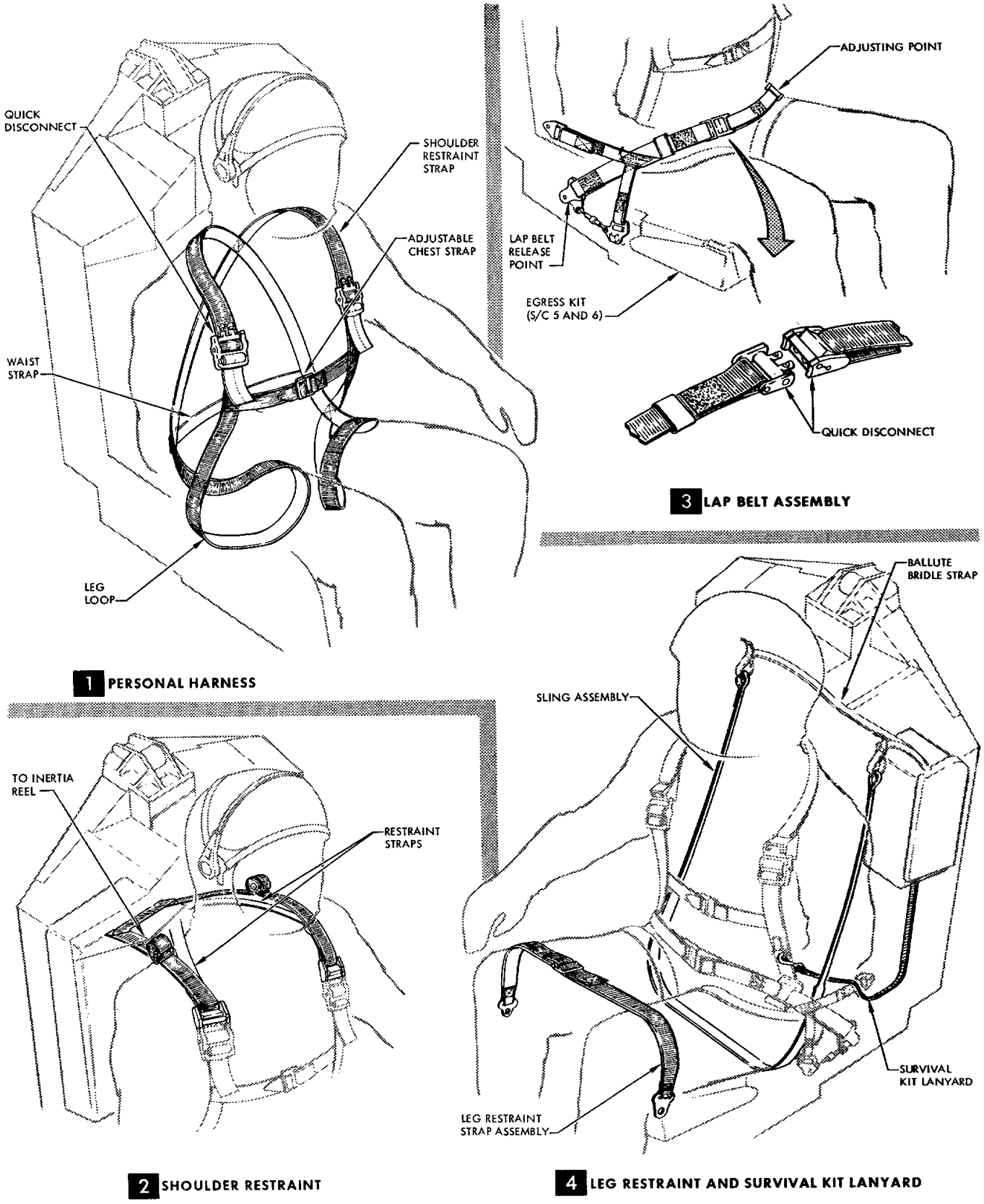


Figure 3-4 Restraint System (Sheet 1 of 2)

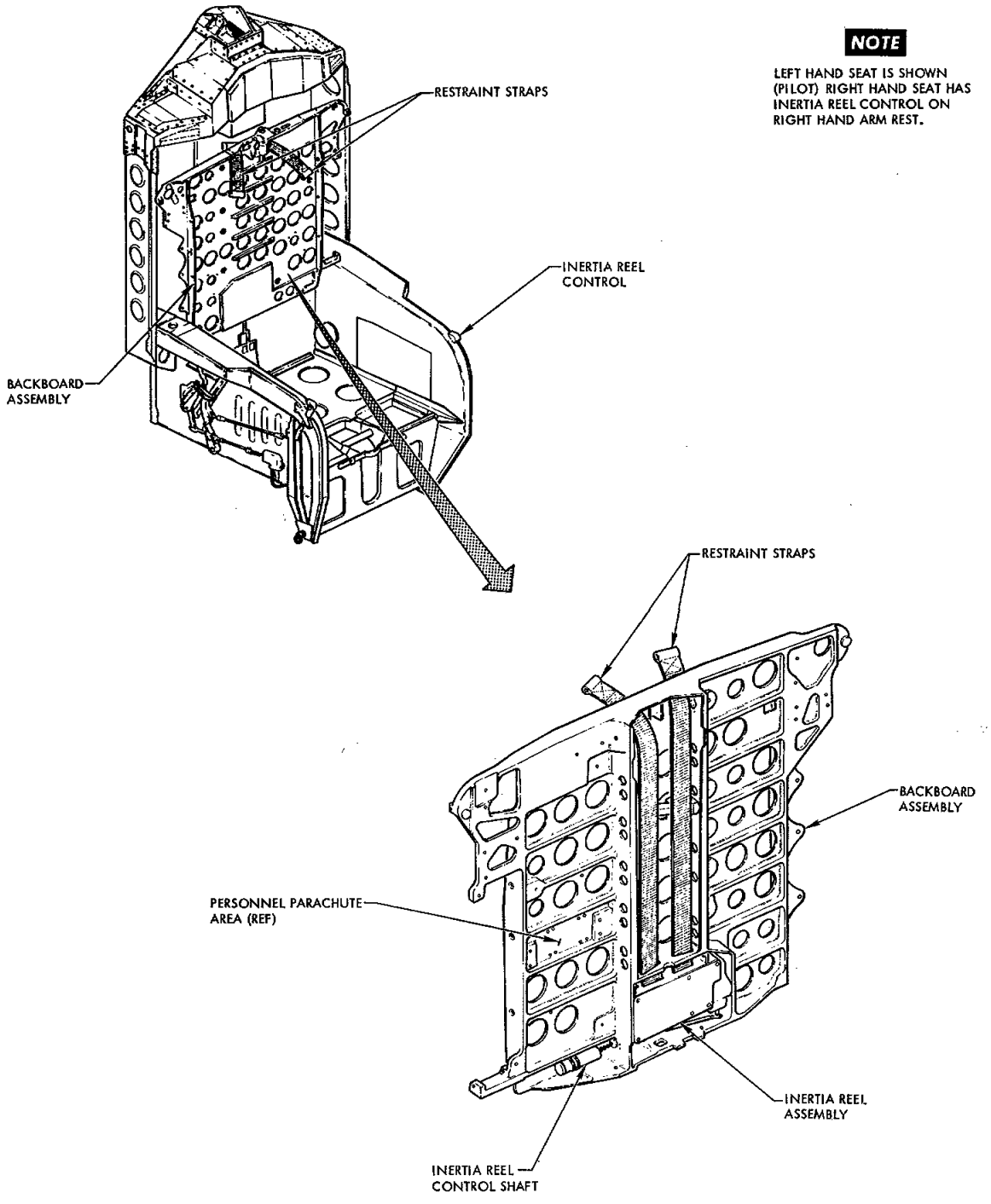


Figure 3-4 Restraint System (Sheet 2 of 2)

ARM RESTRAINT

The arm restraint (Figure 3-4) is a welded, 1/2 inch diameter tube assembly made up in the form of a loop. A loop is installed on each arm rest to retain the pilots arms within the ejection envelope. When the arm restraint loop is not required, it may be swung to the rear and down.

ELBOW RESTRAINT

An elbow restraint is provided for the command pilot only. It is used to stabilize his forearm during manual re-entry.

LEG RESTRAINT STRAP

The leg restraint (Figure 3-4) consists of two straps of dacron webbing with a connecting slide buckle. One end of each strap is secured to the seat by round metal eyelets. The left strap of each leg restraint has a metal end assembly that permits the right strap to fold back on itself. Velcro tape on the right strap is used to secure the strap end in position when the strap is drawn tight over the pilots legs. During seat/man separation, the restraint strap eyelets are automatically released from the base of the seat, freeing the restraint strap.

EJECTION SEAT FOOT STIRRUP

The ejection seat foot stirrups (Figure 3-2) consist of two welded frames attached to the front of the ejection seat. Each stirrup has a short protruding platform with small vertical edges rising along the outboard side. The stirrup is so constructed that the pilots shoe heel will lock in place and prevent forward movement of the foot while the small vertical edges will prevent side movement. During seat ejection, the pilots feet will stay in place.

LAP BELT

The lap belt (Figure 3-4) is an arrangement of dacron and nylon straps, designed to restrain the pilot in the seat structure. Load carrying straps from the lap belt are fastened to the backboard and seat. The lap belt has a manual quick disconnect and a pyrotechnic release fitting near the center of the pilots lap. The manual quick disconnect can be released with one finger. Lap belt tension is adjusted by sliding excess strap through the pyrotechnic release. During ejection, the lap belt ends attached to the seat structure are released just prior to seat/man separation. During separation, the lap belt remains with the pilot. Five seconds after the backboard drogue mortar fires, the pyrotechnic lap belt release activates and allows the lap belt, backboard and seat to fall free.

A second manual release for the lap belt is also available to the pilot. It is located forward on the right arm rest and is referred to as the ditch control. Releasing the lap belt with the ditch control allows the pilot to egress from the landing module with the backboard and seat.

EGRESS KIT (Effective Spacecraft 5 and 6)

The egress kit assembly contains the bail out oxygen for an ejected pilot. The egress kit rests in the ejection seat bucket and forms a mounting surface for the egress kit cushion. The egress kit contains an oxygen supply, for breathing and suit pressurization; a composite disconnect, which when separated closes the port and prevents escape of egress oxygen; a relief valve, to prevent pressure build up in the pressure suit; a regulator, to reduce high pressure to a controlled flow of low pressure oxygen, a pressure gage, for visually checking egress oxygen pressure; and connecting lines. Three lanyards are attached between the egress kit and

the spacecraft. These lanyards pull release pins to allow the composite disconnect to separate, allow the oxygen to flow through the pressure regulator and allow the relief valve to control the pilots suit pressure. When the drogue mortar deploys the pilot parachute, a 5-second pyrotechnic time delay is initiated and at burn out, the egress kit with the backboard is separated from the pilot.

EGRESS KIT CUSHION (Effective Spacecraft 5 and 6)

The egress kit cushion (Figure 3-2) has a universal type of contour and is attached to the top of the egress kit. The cushion is positioned forward of the pelvic block and up to the ejection control handle access door.

BACKBOARD ASSEMBLY

The backboard assembly (Figure 3-2) is machined aluminum, designed and stressed to retain the inertia reel, ballute, ballute release and deploy mechanism, drogue mortar, parachute and survival kit. A cushion, contoured to the individual pilots body requirements, is positioned on the forward surface of the backboard. The cushion is provided to supply support and comfort to the pilots back. The inertia reel straps and lap belt secures the pilot to the backboard. The backboard accompanies the pilot through seat ejection to parachute deployment. Five seconds after parachute deployment, the backboard with the seat is separated from the pilot.

PELVIC BLOCK

The pelvic block (Figure 3-2), contoured to the lower torso of each pilot, is positioned between the backboard assembly and the seat. The block supports the pilots lower vertebra and pelvic structure. It remains with the seat structure upon seat/man separation.

BALLUTE SYSTEM

The ballute system (Figure 3-2) consists of a barostat controlled pyrotechnic initiator, combined with a pyrotechnic gas generator, cutters and a packaged ballute. The ballute, located on the back and lower left side of the pilots backboard, is an aluminized nylon fabric enclosed cone. It is inflated by ram air passing through four inlets located symmetrically around the upper periphery. The ballute is connected to the backboard through an 8 inch riser, a 5 foot dual bridle, and by a one inch wide dacron webbing passing through a pyrotechnic actuated cutter. The ballute provides the pilot with a stabilized, feet into the wind, attitude for all ejections over 7,500 feet. The system is fully automatic and is actuated at seat/man separation. At altitudes below 7,500 feet, the barostat prevents deployment of the ballute.

PERSONNEL PARACHUTE

The personnel parachute (Figure 3-2) is a standard 28 ft dia nylon parachute. The parachute is located on the right rear of the pilots backboard. It is deployed by the drogue mortar slug and pilot chute. The parachute risers are attached to the pilots personal harness.

PARACHUTE DROGUE MORTAR

The parachute drogue mortar (Figure 3-2) is a pyrotechnic device designed to eject a 10 oz drogue slug with sufficient velocity to deploy the pilot chute of the personnel parachute. The drogue mortar is a barostat operated firing mechanism, but can be fired manually. It will fire and deploy the parachute at or below 5,700 feet plus a 2.3 seconds time delay from seat/man separation. An MDF chain is initiated by the drogue mortar and separates the backboard and seat from the pilot.

PERSONAL HARNESS ASSEMBLY

The personal harness assembly (Figure 3-4) provides a light, strong, and comfortable arrangement to attach the personnel parachute to the pilot. The harness is constructed from nylon webbing formed into a double figure-8. The two figure-8's are joined by two cross straps, the waist strap, and the chest strap. Only the chest strap is adjustable. A quick disconnect is placed forward and below each shoulder for connection of the parachute risers and shoulder restraint straps. Below the left quick disconnect, a small ring is incorporated to attach the survival kit lanyard.

SURVIVAL KIT

The survival kit (Figure 3-2) is a packaged group of specially designed equipment for the use of a downed pilot. Articles in this kit are intended to aid in preserving life under varying environmental conditions. Deployment of the survival kit is automatic if the pilot ejects and is also available to the pilot if he lands with the spacecraft.

Deployment of the survival kit during the ejection cycle takes place as the backboard and seat falls away from the parachuting pilot. As the backboard falls, the survival kit lanyard, connected to the pilots harness, pulls a pin on the life raft container. When the pin is removed, the daisy chain loops are disengaged and the life raft and rucksack are extracted from the container. The survival kit lanyard repeats the extraction process in removing the machete and water bottle from the second container. The machete and water bottle are stowed in a survival equipment container on the left front side of the backboard.

During seat/man separation, a lanyard between the seat structure and the rucksack activates the radio beacon. As the pilot descends on his parachute, the survival equipment is suspended below and the radio beacon transmits on an emergency frequency. Direction finding equipment on aircraft and aboard ship can plot the pilots position taking navigational fixes on the radio/beacon.

Survival equipment is divided into two major stowage containers. The life raft container mounted on the left rear of the backboard has the following items:

Life raft container (Typical)

1 Life Raft

1 Sea anchor

1 4 inch x 4 inch Foam rubber pad

1 CO_2 cylinder

1 Sea dye marker

1 Sun bonnet

Rucksack (Typical)

1 Survival light

1 Strobe light

1 Flash light

4 Fish hooks

Fish line

2 Sewing needles and thread

1 Magnetic compass

1 Fire starter

4 Fire fuel

1 Whistle

1 Signal mirror

14 Water purification tablets

1 De-salter kit (less can)

8 De-salter tablets

1 Water bag

1 Repair kit

1 Medication kit (Typical)

6 Tablet packets

1 Small injector (1 CC)

1 Large injector (2 CC)

1 3 inch x 3 inch compress

1 12 inch x 12 inch aluminum foil

1 Tube zinc oxide

1 pr Sun glasses

1 Radio beacon

The forward survival kit, mounted on the forward surface of the backboard to the left of the pilots shoulder, contains the following;

1 Water container with 3 lb of water

1 Machete with sheath

PYROTECHNIC DEVICES

There are 18 pyrotechnic devices incorporated in the cabin all of which pertain to seat ejection, restraint release and parachute deployment. The pyrotechnic devices are 2 hatch actuators, 2 seat rocket/catapults, 2 ballute deployment and release mechanisms, 2 backboard and seat jettison, 2 drogue mortars, 2 harness

release actuators, 2 seat/man separator actuators, 2 hatch actuator initiators and 2 hatch MDF (Mild Detonating Fuse) harnesses. The pyrotechnic devices, except the drogue mortar, are safetied by stowing the ejection control handle (D-ring) with a safety pin through the handle into the ejection control assembly. On spacecraft 8 only, a second ejection control pyrotechnic safety pin is also inserted in the side of the ejection control assembly to completely safety the MDF manual firing mechanism.

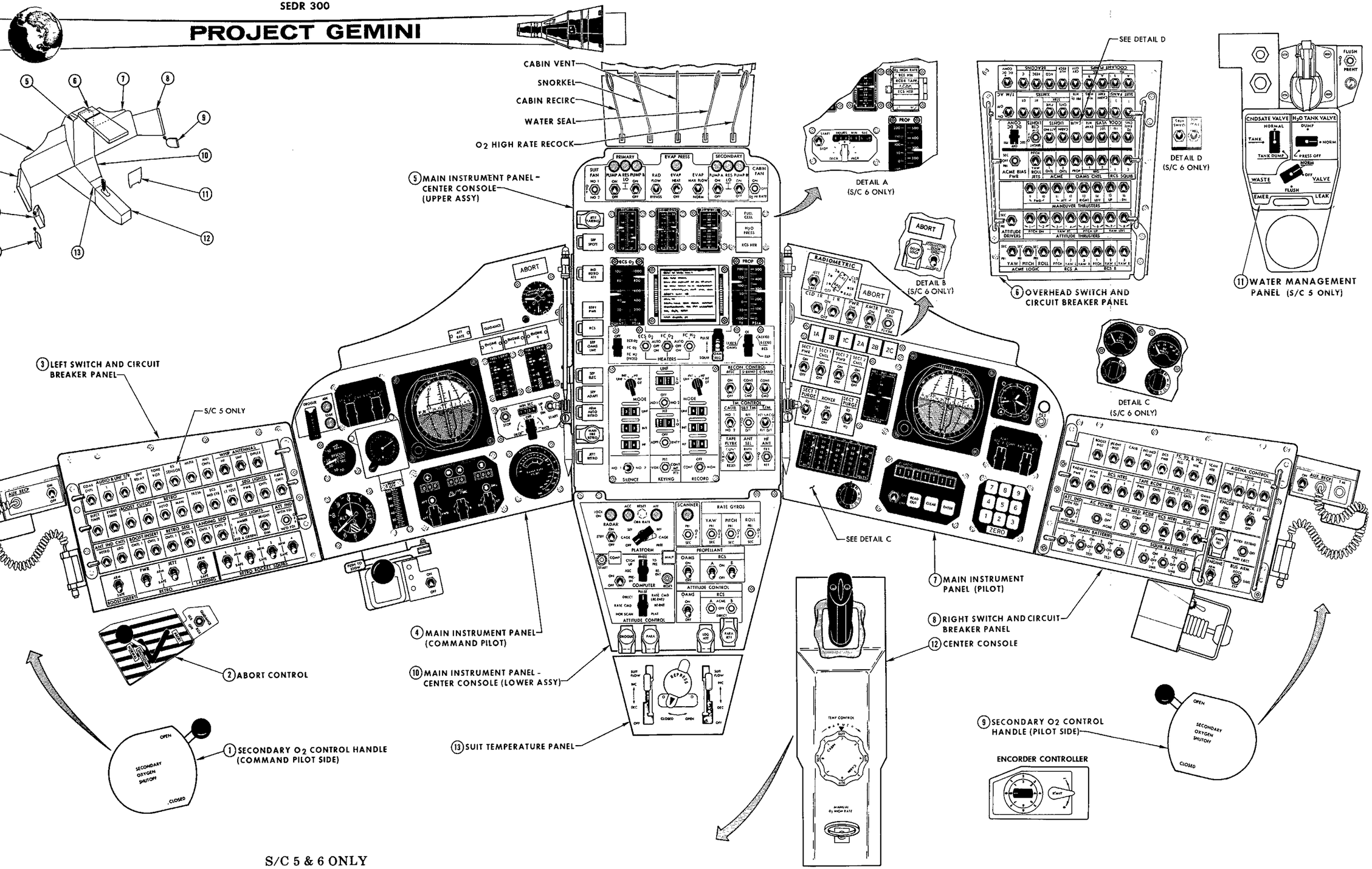
INSTRUMENT PANELS

Instrument panels, switch and circuit breaker panels and pedestal panels (Figure 3-5) are arranged to place controls and indicators within reach and convenient view of each crew member while in a full pressure suit. A swizzle stick, stowed by the overhead switch and circuit breaker panel, enables a pilot to position switches and rotate selectors on the opposite side of the cabin. With this arrangement, one pilot can control the complete spacecraft and temporarily free the second pilot of all duties.

CABIN INTERIOR LIGHTING

Cabin interior lighting is provided by three types of lights located in five separate locations, described as follows: Cabin flood lights are located aft and above the center-line stowage area. A DIM control is located under the light to control light intensity. Instrument flood lights are located at the forward inner edge of the hatches. Each instrument flood light installation contains two lamps, one lamp having a red filter and the other a white filter projecting downward. A DIM control and a RED-WHITE-OFF switch are provided at each of the lights.

Two utility lights attached to the ends of spiral extension cords are located on the left and right side walls of the spacecraft interior. The lights stow in



S/C 5 & 6 ONLY

Figure 3-5 Instrument Panels and Displays (Sheet 1 of 3)

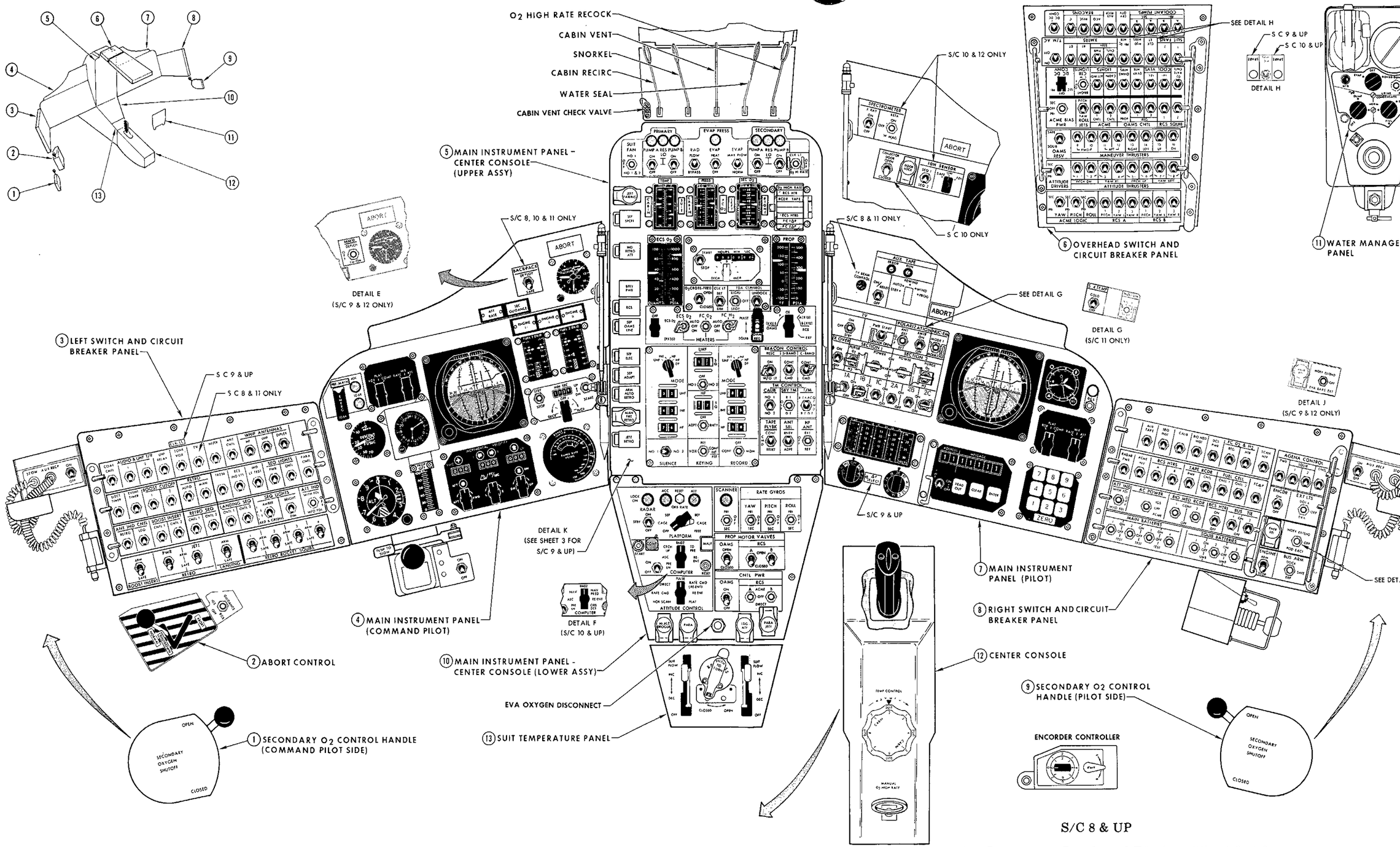


Figure 3-5 Instrument Panels and Displays (Sheet 2 of 3)

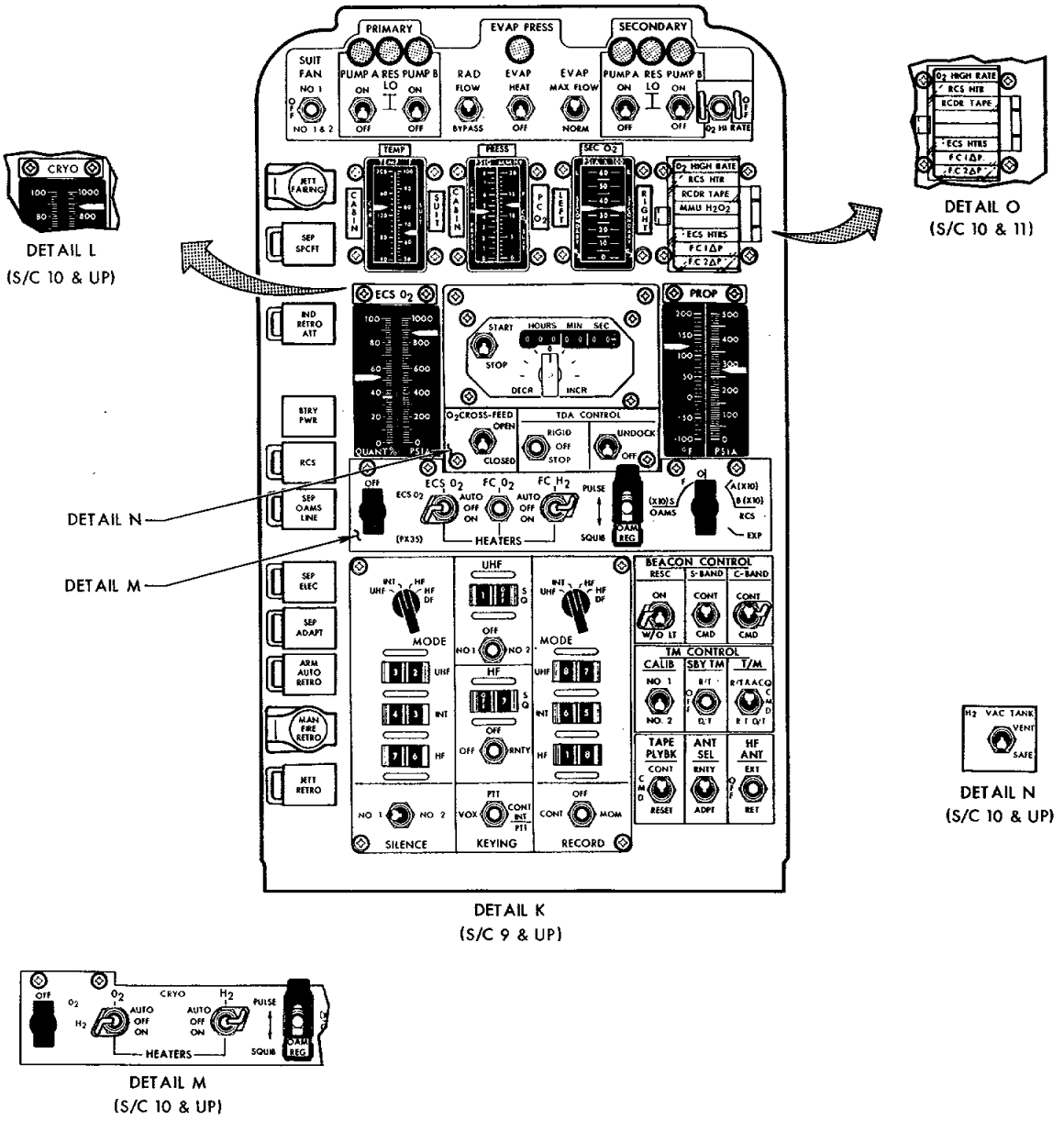


Figure 3-5 Instrument Panels and Displays (Sheet 3 of 3)

clips mounted on the side walls. An ON-OFF switch is located adjacent to the AUX RECEP panel on each of the spacecraft side walls. The CTR LIGHTS, BRIGHT-OFF-DIM switch and the CABIN LIGHTS switch-circuit breaker are located on the overhead switch and circuit breaker panel.

ELECTRICAL OUTLETS

The two receptacles, powered by the spacecraft electrical system, are installed on brackets immediately aft of the left and right switch/circuit breaker panels. These receptacles are controlled by adjacent ON-OFF switches and are used for powering the utility light or other electrical equipment.

STATIC SYSTEM

The static pressure system is employed to operate the rate of descent indicator, altimeter, and to supply pressure to the static pressure transducer for instrumentation. The static system is also utilized to provide a differential pressure for the cabin pressure transducer. The static ports (Figure 3-6), used for atmospheric pressure pick-up, are located in the small end of the spacecraft conical section. The static port (Figure 3-6), used for differential pressure pick-up, is located on the forward surface of the small pressure bulkhead.

FOOD WATER AND EQUIPMENT STOWAGE

Containers to left, right and aft of pilots (Figure 3-7) are provided for equipment and food storage. Although minor changes in storage containers are dictated by mission requirements, the main containers are as follows: Center-line stowage box, used for larger size camera containers and EVA (Extra-Vehicular Activity) chest pack; right aft pressurized stowage box, used to stow food initially and later, body waste materials; left aft stowage box, used to stow food packages;

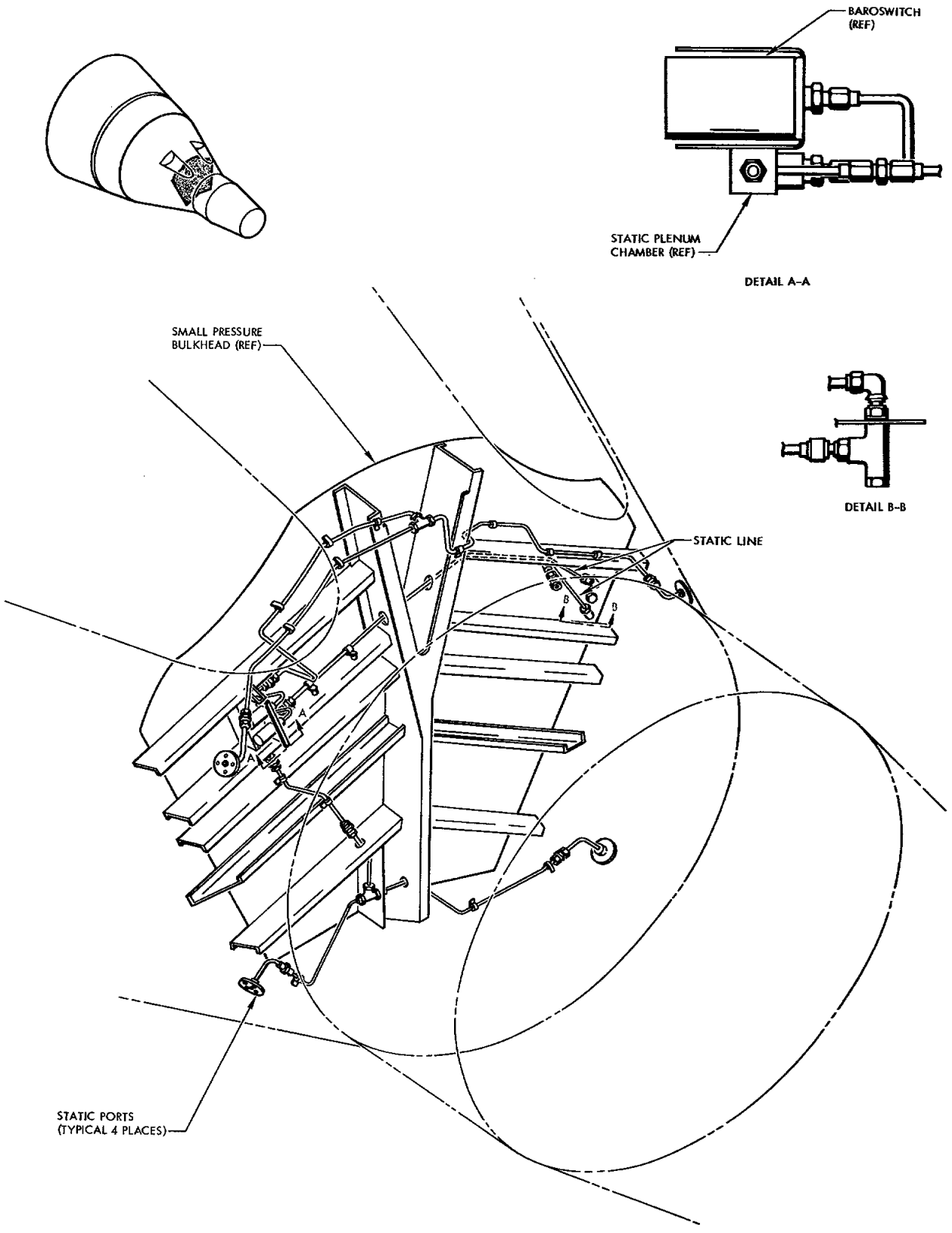


Figure 3-6 Static System

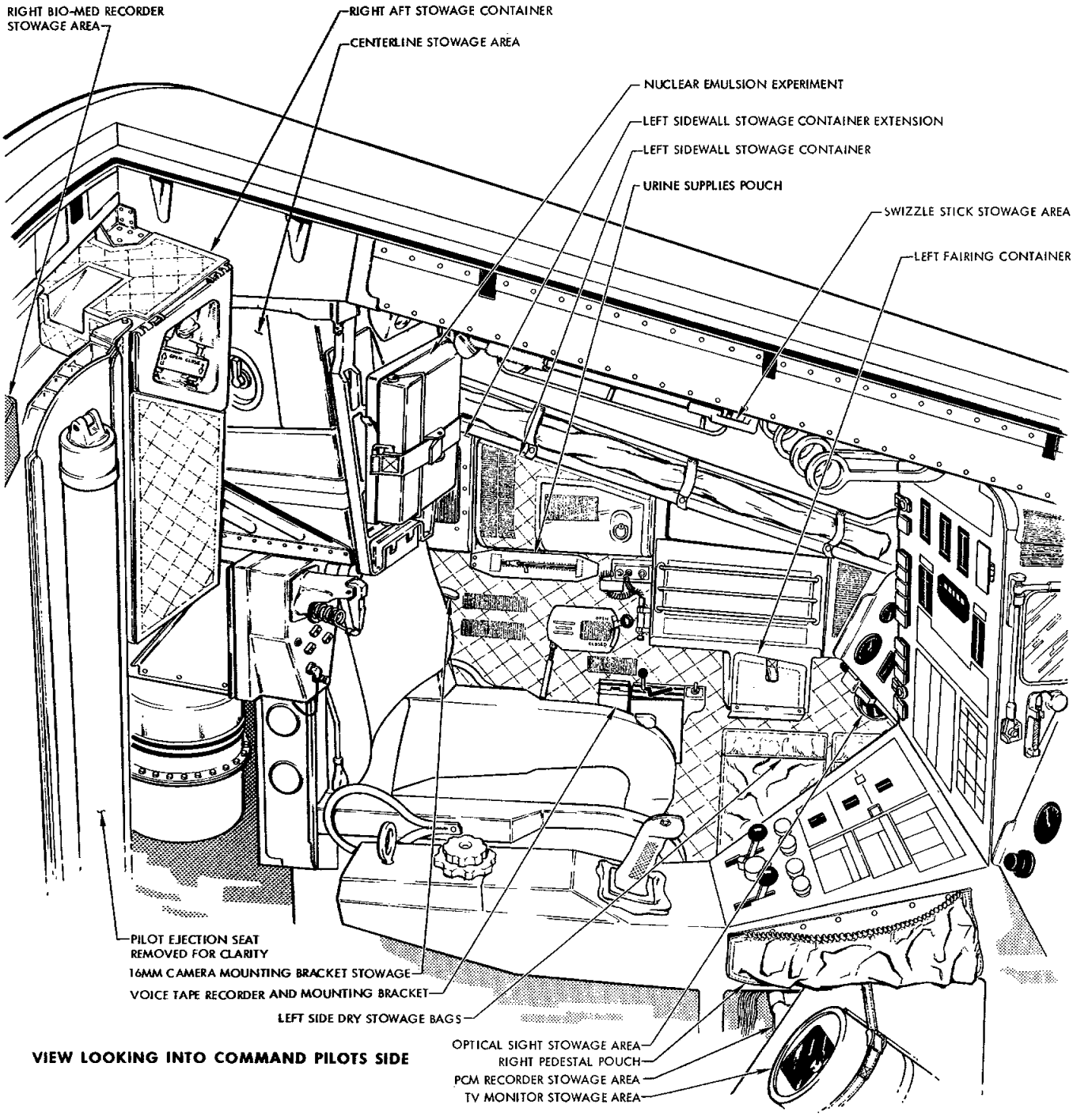


Figure 3-7 Spacecraft Interior Stowage Areas (Typical) (Sheet 1 of 2)

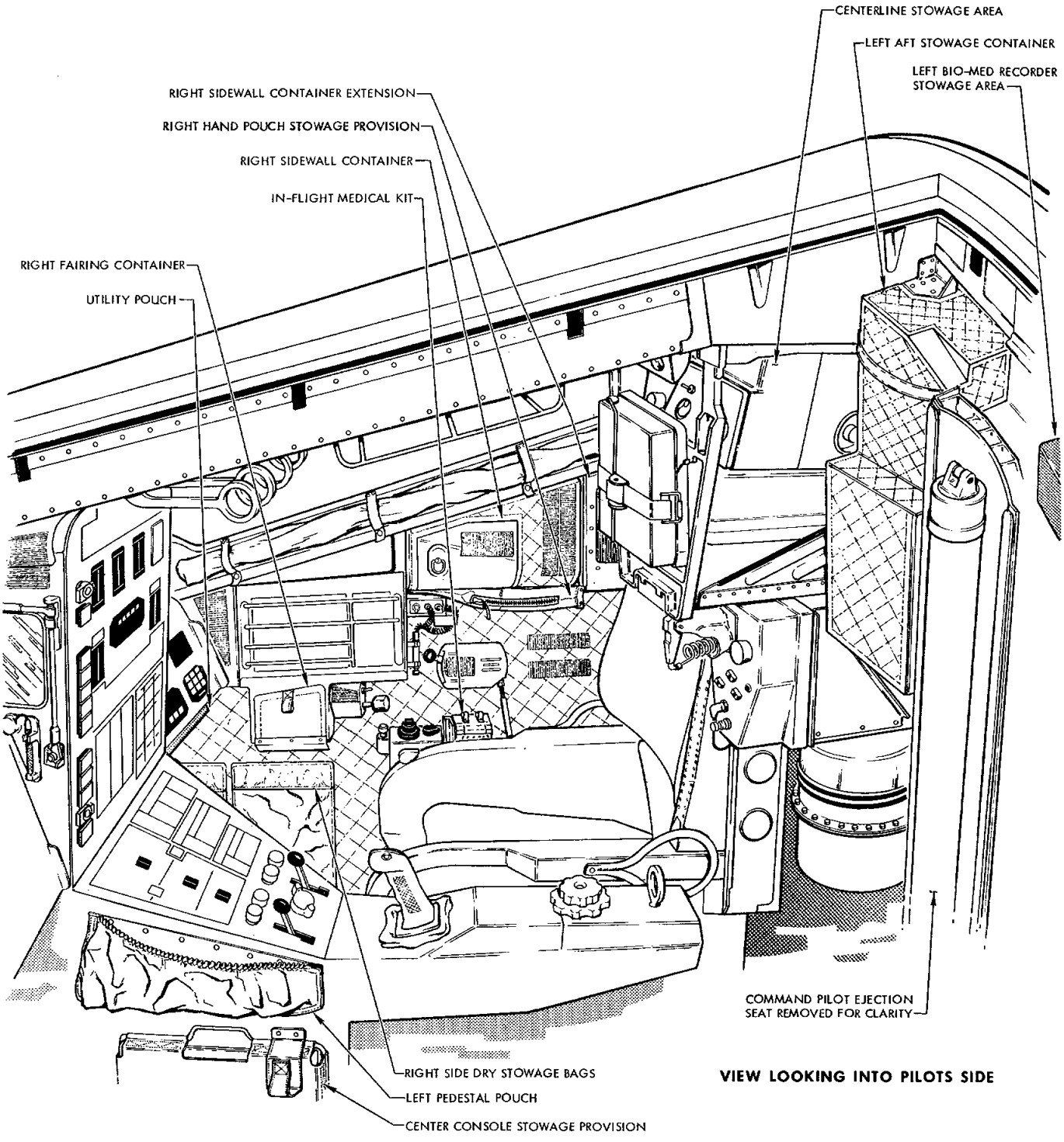


Figure 3-7 Spacecraft Interior Stowage Areas (Typical) (Sheet 2 of 2)

right and left sidewall stowage boxes, used to stow small pieces of equipment; left and right fabric covered sidewall stowage boxes, used to stow lightweight head sets; hatch food pouches used to stow large quantities of food; and sidewall stowage box extensions used to stow penlight, spotmeter, exposure dial and tape recorder cartridges. Equipment stowed in the above boxes may change with each mission.

Larger pieces of equipment, emergency equipment or equipment used on every flight, have special stowage brackets or fabric pouches positioned throughout the interior of the spacecraft. Examples of specific stowage brackets are as follows: inflight medical kit, stowed aft of abort control handle; and the optical sight, stowed under command pilots instrument panel. Without counting the food packages, stowage facilities are furnished for more than 125 pieces of equipment.

During flight, various pieces of frequently used equipment are removed from launch stowage areas and are stowed, with Velcro tape, on the spacecraft sidewalls, and on the inside surfaces of the hatch. As debris accumulates during flight, it is placed in the left aft debris area, located aft of the pilots seat. Prior to descent, the equipment is re-stowed. Only a general rule can be applied to stowage descriptions. Exposed film is placed in insulated containers, previously occupied by cameras and lens, in the center line stowage box. The left aft stowage box is filled and the remainder of the loose equipment is divided among the sidewall stowage boxes on a planned basis. The pressurized stowage box is used to store urine samples and waste containers.

A water storage container, with a 16-pound capacity, is located forward of the aft pressure bulkhead, between the seats. As the water is used from the main storage container, it is replenished by the water stowed in the adapter section.

Drinking is accomplished by means of a tube and manual valve system. Food and water will be sufficient for the mission and a postlanding period of 48 hours.

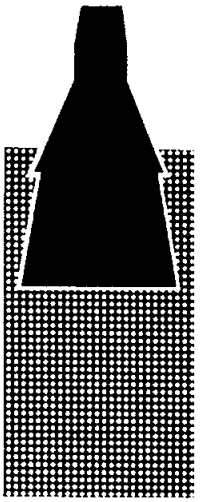
WASTE DISPOSAL

Feces will be collected in a glove-like plastic bag. Urine samples are taken, and the remainder disposed of by overboard dumping. The urine samples and feces waste containers are stowed in the right aft pressurized container which allows cabin depressurization without possible boiling off of the waste materials moisture content.

STOWAGE PROVISIONS

Personal stowage facilities are provided by retaining removed portions of the pressure suit and other equipment as required. These provisions consist of floor pouches, Velcro covered areas on the walls of the pressure vessel, adjacent to the pilots and attached to the structure in usable areas. Items to be stowed utilize the hook and pile principle of mating Velcro patches.

SEQUENCE SYSTEM



Section IV

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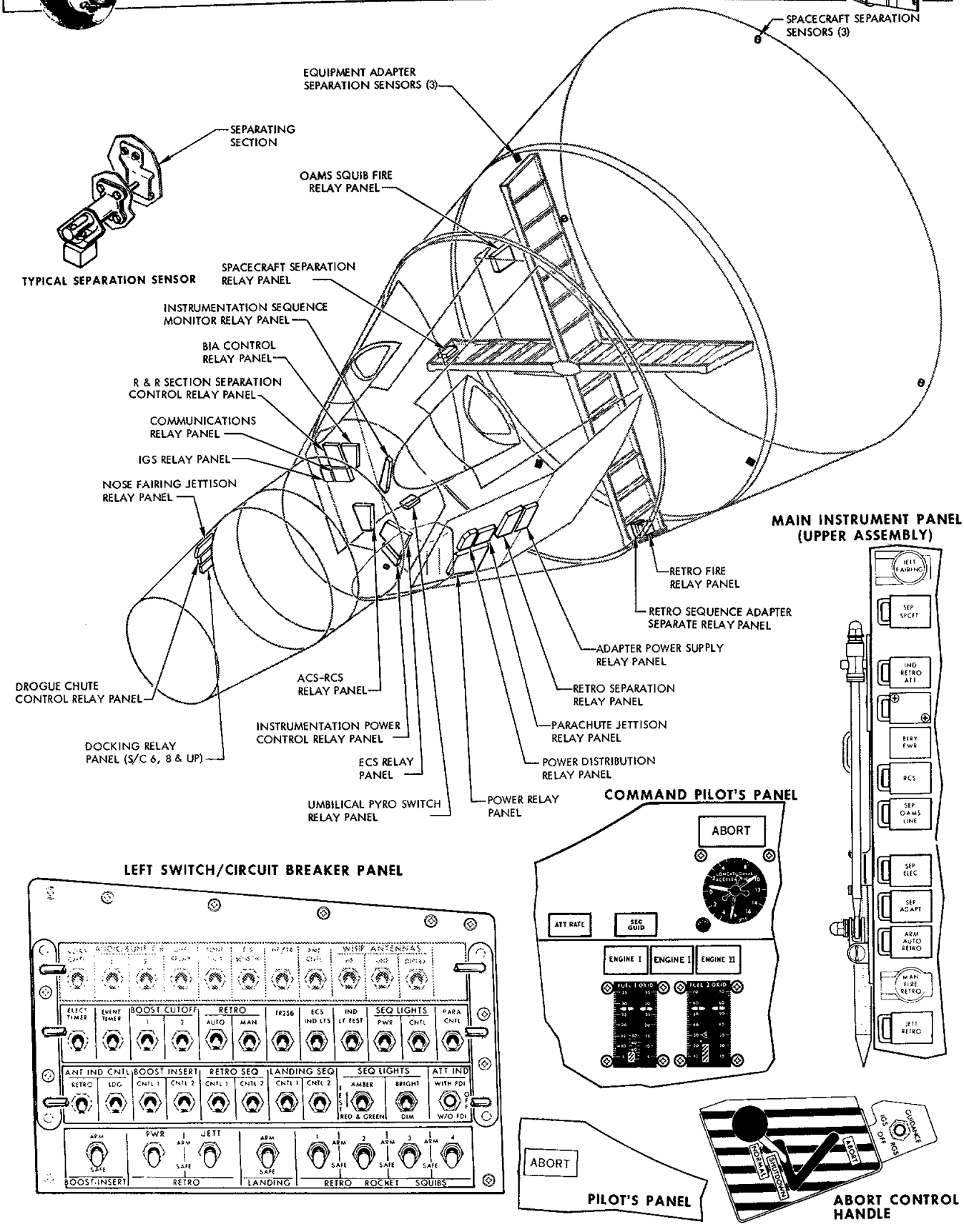


Figure 4-1 Sequential System

SECTION IV SEQUENCE SYSTEM

SYSTEM DESCRIPTION

The Sequence System of Gemini Spacecraft 5, 6, and 8 through 12 comprises those controls, indicators, relays, sensors and timing devices which provide semiautomatic control of the spacecraft and/or launch vehicle during the critical control times, but which are not part of other systems. (See Figure 4-1). The critical times are: the time from booster engine ignition through insertion into orbit; the time to prepare to go to retrograde through post-landing; and the time to abort.

The Gemini crew does not control the spacecraft during boost through Second Stage Engine Cutoff (SSECO). The spacecraft is controlled by Radio Guidance System (RGS) and the Digital Command System (DCS), or by the Inertial Guidance System (IGS) and the on-board computer. The crew does however, monitor certain indicators to keep informed of the operation of the launch vehicle, to anticipate a crisis if one should develops, and to know if and when mission abort is mandatory. After SSECO the command pilot takes necessary action to separate the spacecraft from the launch vehicle and applies final thrust to place the spacecraft in the desired orbit.

During orbit, the Sequence System is in standby. The electronic timer, however, which is part of the Time Reference System, is counting down the time-to-go to retrograde.

At 4 minutes and 16 seconds before retrograde, (T_R-256 seconds), a Sequence System relay is actuated, and several Sequence System indicators illuminate amber. These indicators provide the crew with cues for necessary operations. Again at 30 seconds before retrograde, the crew is reminded to separate the adapter equipment and arm the automatic retrograde rocket firing circuits. The Sequence System, if properly

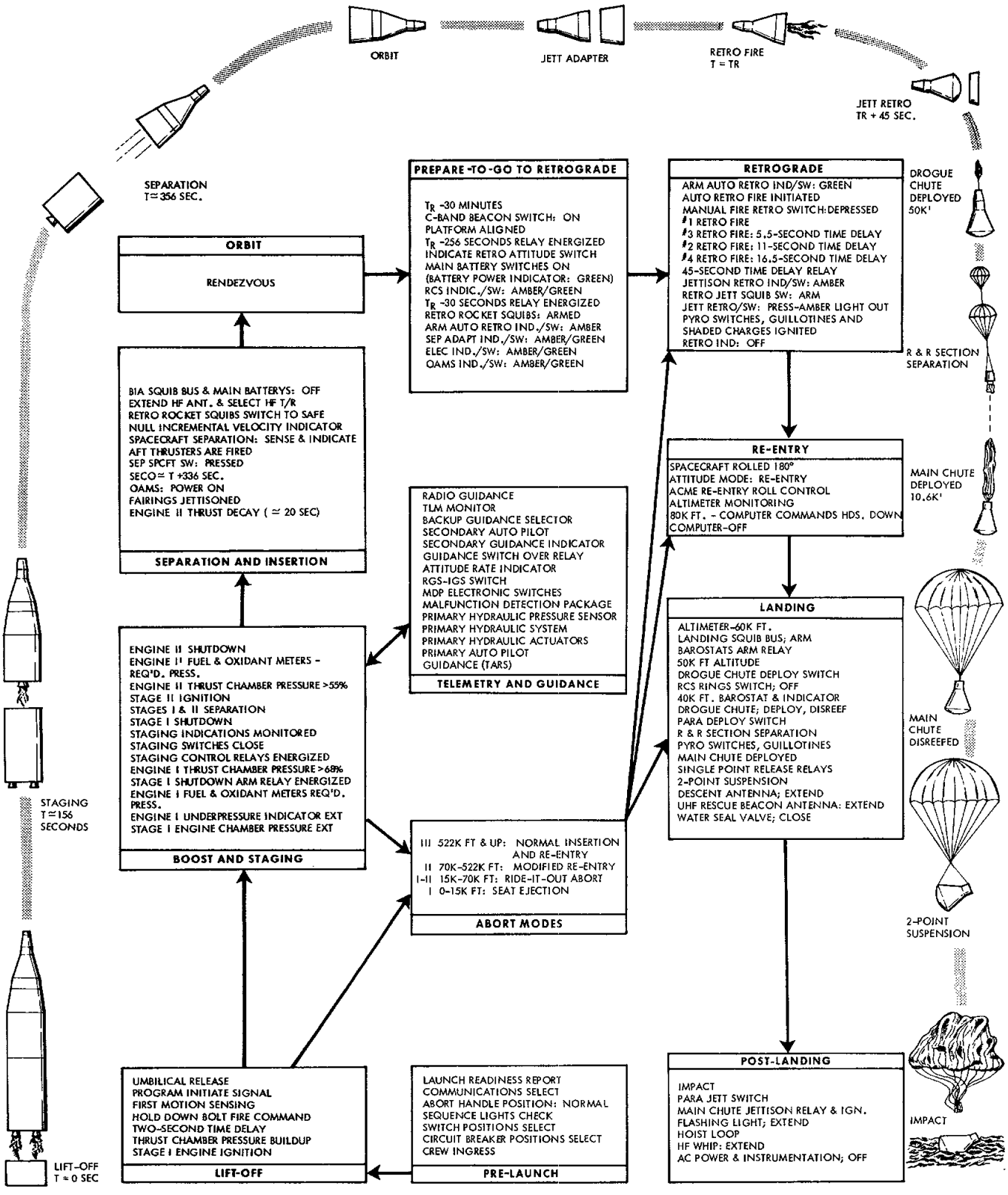


Figure 4-2 Sequential System Simplified Block Diagram

armed, will initiate retrograde automatically. The crew redundantly initiates retrograde manually as a safety precaution. During descent, altitude indicators illuminate as cues to deploy parachutes. After splash down, the main parachute is jettisoned, and all systems are shutdown.

Four abort modes comprise the abort sequence. They are: seat ejection (mode I); ride-it-out abort (mode I-II); modified re-entry (mode II); and normal re-entry (mode III). The mode selected for abort is related to the spacecraft altitude at the time the abort command is given.

SYSTEM OPERATION

To simplify explanation, the Sequence System is divided into eight stages. The eight stages are; pre-launch, lift-off, boost and staging, separation and insertion, prepare-to-go to retrograde, retrograde, re-entry, and abort. Figure 4-2 shows these sequence stages, and the detailed function of each stage. Telemetry guidance, landing and post-landing are related to but not part of the Sequence System. The simplified block diagram is explained in the following paragraphs.

Pre-launch, lift-off, boost and staging, and separation and insertion are explained first. Prepara-to-go to retrograde, retrograde, and re-entry are discussed next. Abort is discussed last.

PRE-LAUNCH

The command pilot and the pilot ingress the Gemini cabin and take their assigned crew stations. The hatches are closed and locked. The crew checks that both D-rings are unstowed. The command pilot makes sure that the abort control handle is in the NORMAL position; the maneuver controller is stowed; the altimeter is set; and the Incremental Velocity Indicator (IVI) is zeroed. He verifies that

the nine sequence indicators, the two ABORT indicator lights, the ATT RATE indicator light, the SEC GUIDANCE indicator light, both ENGINE I indicator lights, and the ENGINE II indicator light are extinguished. He places the top three rows of circuit breakers on the left switch/circuit breaker panel to the closed (up) position. He places the BOOST-INSERT and RETRO ROCKET SQUIB switches in the bottom bow to ARM, and the RETRO and LANDING switches to SAFE. He tests the nine sequence indicators with the SEQ LIGHTS TEST switch. He selects switches for gyro run-up and platform alignment, and performs on-board computer checkout.

The pilot places the four MAIN BATTERIES switches and the three SQUIB BATTERIES switches to ON. Both pilots select and check their intercom and uhf communications. The remaining controls and indicators are also monitored or positioned as required. The crew verifies and reports all systems ready for launch.

LIFT-OFF

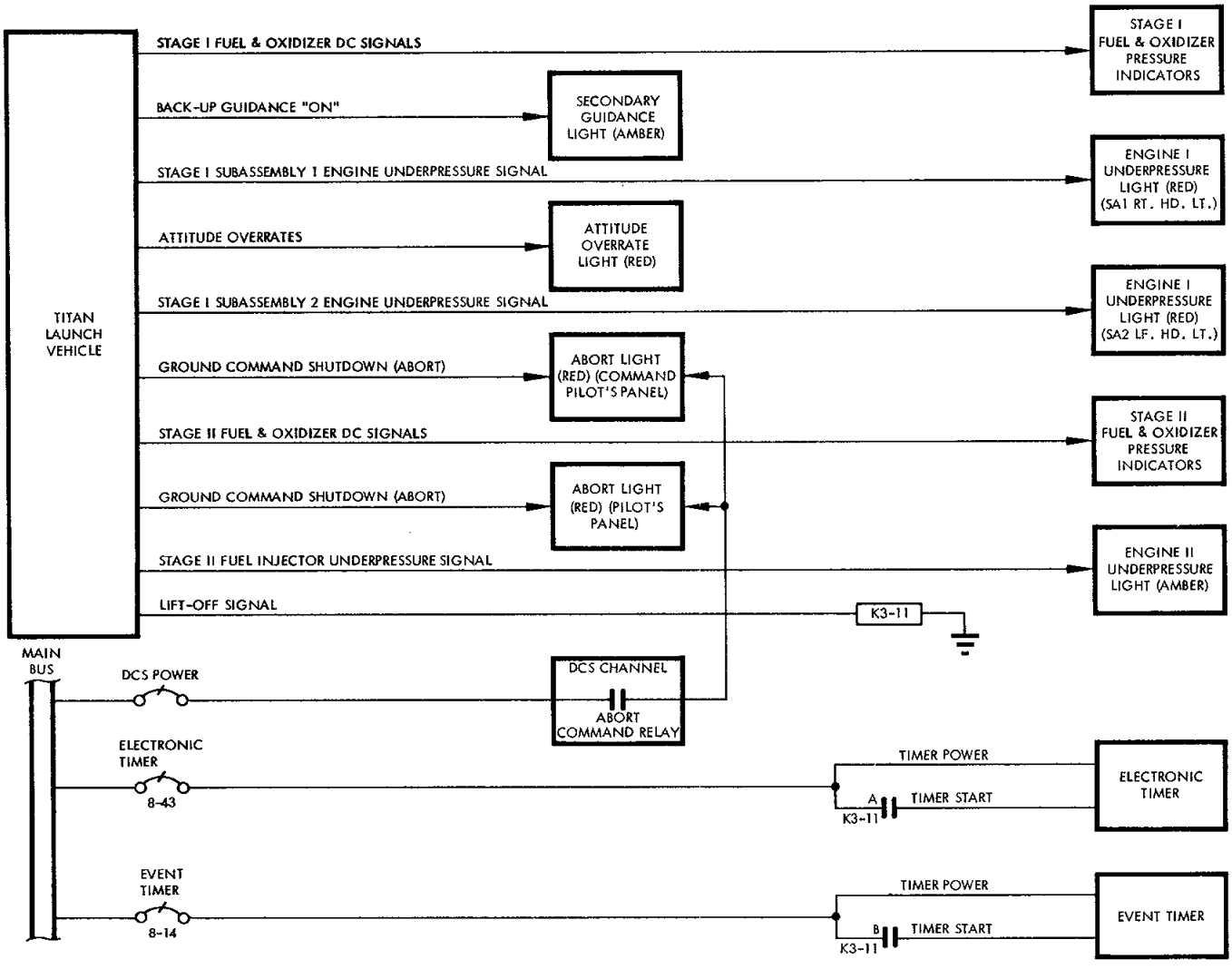
When the pre-launch countdown reaches zero, the first stage engine ignition signal is given from the blockhouse. Both first stage engines begin thrust chamber pressure buildup. Both ENGINE I indicators illuminate red but extinguish in about one second. When the thrust chamber pressure of these two engines exceeds 77 percent of rated pressure, a two-second time delay is initiated in the blockhouse. If all systems remain go during this delay, the hold-down-bolt fire command is given and the launch vehicle is committed to flight. First motion sensors detect vehicle ascent one and one-half inches off the pad, and energize time-zero relays in the blockhouse and in the spacecraft. A 145-second shutdown arm time delay is initiated to prevent accidental booster engine shutdown prior to the scheduled staging time. The umbilical release command is given, disconnecting the adapter, and re-entry umbilicals. The on-board computer is switched from the guidance inhibit mode to

the guidance initiate mode and enabled to accept acceleration data. The lift-off signal is also applied to the electronic timer and the event timer. The electronic timer begins to count down the time-to-go to retrograde. The event timer begins to count up the time from lift-off.

BOOST AND STAGING

As the missile continues to climb, the crew monitor the boost sequence and ABORT indicators. The two ENGINE I under pressure indicators, the ATT RATE indicator and both ABORT indicators must remain extinguished. The ENGINE II indicator illuminates amber. The STAGE I FUEL and OXIDIZER needles must indicate pressures within the required limits, and the LONGITUDINAL ACCELEROMETER must indicate an increasing acceleration within prescribed limits for the flight time indicated by the event timer. The pilots monitor their indicators and report via uhf link to the ground. Abort mode I prevails during the first 50 seconds of flight. Ground stations notify the pilots when abort mode I is no longer applicable, and when abort mode I - II becomes applicable. Abort mode I-II is in effect during approximately the next 45 seconds of flight. At T+95 seconds, the crew receives and acknowledges changeover to abort mode II.

At T+145 seconds, when the acceleration has climbed to nearly 6g's, the first stage engine shutdown arm relays are energized. At approximately T+153 seconds, the thrust chamber pressure drops to less than 68 percent. The two ENGINE I indicators illuminate red, and the staging control relays are energized. The staging switches are closed. The stage I shutdown solenoids energize and both engines are shutdown. Acceleration drops sharply to approximately 1.5g's. The booster sequential system immediately ignites the second stage engine. The explosive bolts which unite stage 1 and stage 2 are detonated, and the stages separate. Both



NOTE

RELAY K3-11, THE LIFT-OFF RELAY, IS LOCATED ON THE COMMUNICATIONS RELAY PANEL.

Figure 4-3 Boost and Staging Sequence

ENGINE I indicators are extinguished. Fuel injector pressure of the second stage engine rapidly increases above 55 percent, extinguishing the ENGINE II underpressure indicator. The LONGITUDINAL ACCELEROMETER begins to climb slowly. The crew reports the results of the staging sequence to the ground station. (Refer to Figure 4-3).

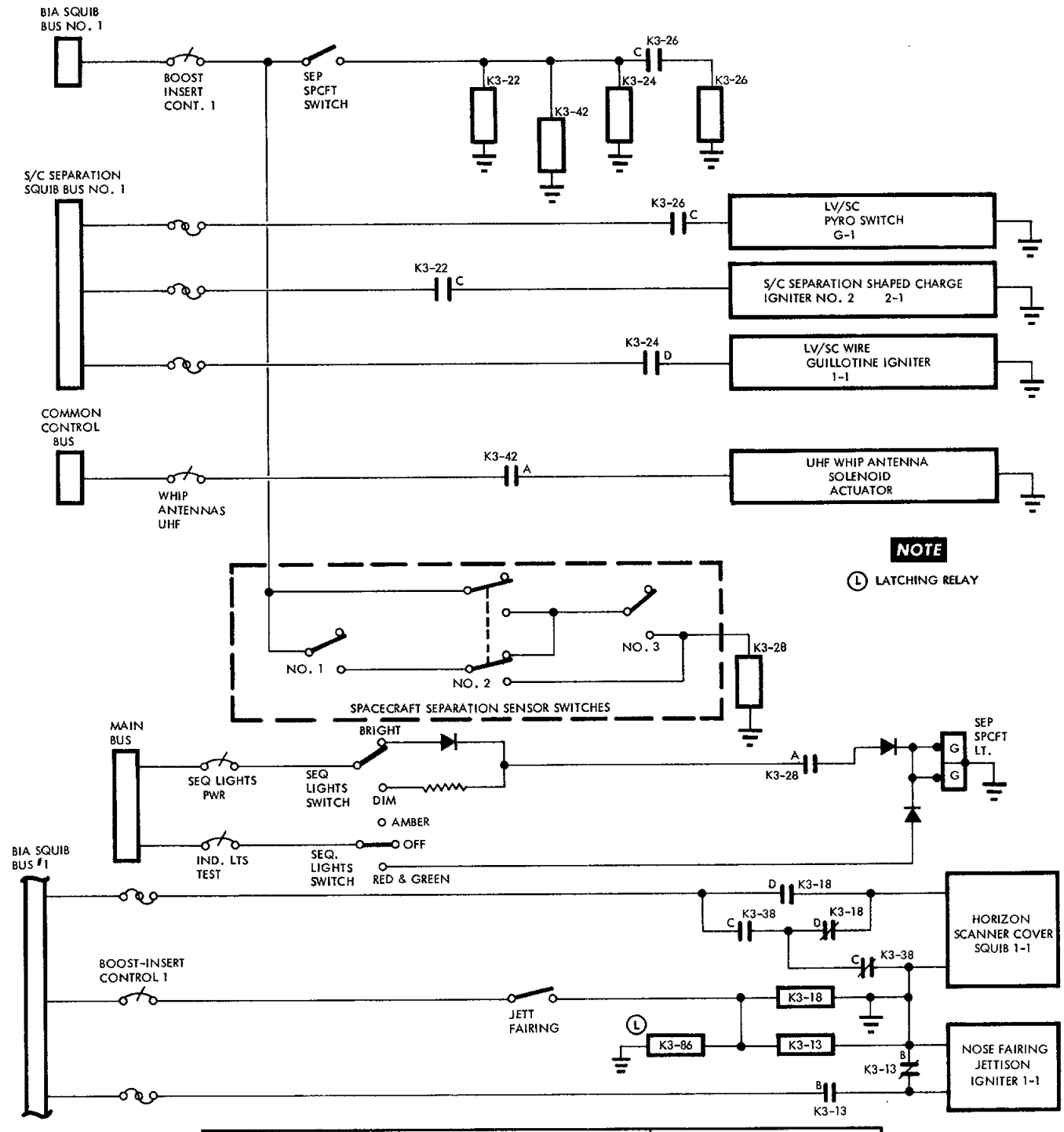
The ENGINE II underpressure indicator, the Attitude Overrate (ATT RATE) indicator, and the two ABORT indicators must remain extinguished. The STAGE 2 FUEL and OXIDIZER needles must indicate the required pressures, and the LONGITUDINAL ACCELEROMETER must show the required increase.

At approximately T+310 seconds, the spacecraft has climbed above 522,000 feet and its velocity exceeds 80 percent of orbital velocity. The ground station notifies the crew that abort mode III now replaces abort mode II. Both pilots acknowledge the change of abort modes.

SEPARATION AND INSERTION

At T+330 seconds, the acceleration has climbed to almost 7g's, and the spacecraft has nearly reached orbital velocity and altitude. Approximately 337 seconds after lift-off, the blockhouse computer transmits the SSECO command tones via the Digital Command System to the launch vehicle. The SSECO solenoids energize SSECO occurs, thrust decays, and acceleration falls rapidly. The on-board computer begins to compute the delta-V required for insertion.

The command pilot waits 20 seconds for launch vehicle thrust to decay. Near the end of the thrust decay period, the command pilot depresses and releases the JETT FAIRING switch on the main instrument panel. This switch energizes nose fairing jettison relays K3-13 and K3-17 and scanner cover jettison relays K3-18 and K3-19.



RELAY	REDUNDANT RELAY	NOMENCLATURE	RELAY PANEL
K3-22	K3-23	SPACECRAFT SHAPED CHARGE IGNITION	BIA S/C SEPARATION CONTROL
K3-24	K3-25	LAUNCH VEHICLE/SPACECRAFT GUILLOTINE	BIA S/C SEPARATION CONTROL
K3-26	K3-27	LAUNCH VEHICLE/SPACECRAFT PYRO SWITCH	BIA CONTROL
K3-28	K3-29	SPACECRAFT SEPARATION SENSOR	BIA CONTROL
K3-42	K3-43	UHF WHIP ANTENNA ACTUATOR	COMMUNICATIONS
K3-86	K3-87	NOSE FAIRING JETTISON LATCH	DOCKING
K3-13	K3-17	NOSE FAIRING JETTISON	BIA NOSE FAIRING JETTISON
K3-18	K3-19	SCANNER COVER JETTISON	ACS-RCS
K3-38	K3-39	SQUIB BUS ABORT	POWER DISTRIBUTION

Figure 4-4 Spacecraft Separation Sequence

These jettison relays arm the nose fairing squibs and scanner cover squibs. The squibs detonate explosive charges, which jettison the fairing and cover.

When thrust decay is complete, the command pilot, depresses and releases the SEP SPCFT switch-indicator on the main instrument panel shown on Figure 4-1. When the contacts of the SEP SPCFT switch-indicator closes, squib bus number 1 power is applied through the closed BOOST-INSERT CONT 1 circuit breaker to relays K3-22, K3-24, and K3-42. (Refer to Figure 4-4). K3-22 is the spacecraft shaped charge ignition relay. K3-24 is the launch vehicle/spacecraft wire guillotine relay. K3-42 is the uhf whip antenna extend relay. Redundant contacts of the SEP SPCFT switch-indicator energize redundant relays with power from squib bus number 2. For simplicity, redundant elements are not shown.

Time delays in the relays and pyrotechnics cause the separation events to occur in the following sequence. K3-24, contacts C energize the launch vehicle/spacecraft pyrotechnic switch relay K3-26. K3-26, contacts C immediately fire the pyrotechnic switch, open-circuiting the wires on the battery side of the guillotine. Next the wire guillotines are fired, severing the launch vehicle spacecraft wires at the interface. Finally the spacecraft shaped charges are ignited, breaking the structural bond between the launch vehicle and the spacecraft. The operation of all pyrotechnics mentioned in this section is explained in Section XI.

The launch vehicle may now separate from the spacecraft, or thrust from the Orbit Attitude and Maneuver System (OAMS) may be required to effect separation. When two inches of separation exist at the interface, the spacecraft separation sensors close. The spacecraft separation sensor relay K3-28 is energized when two of the three sensor switches are actuated. Contacts A of K3-28 apply main bus power

RELAY	REDUNDANT RELAY	NOMENCLATURE	RELAY PANEL
K1-29		MAIN BATTERY POWER INDICATOR	POWER
K7-3		EQUIPMENT DISCONNECT	ECS
K8-16		T_R-256 SECONDS	COMMUNICATIONS
K8-17		POWER SYSTEM T_R-256	POWER
K8-19		RCS AMBER LIGHT	ACS & RCS
K8-29		INDICATE RETRO ATTITUDE	IGS
K11-5		RCS AUXILIARY	ACS & RCS
K11-7	K11-9	RCS SQUIB FIRE	ACS & RCS
K11-8	K11-10	RCS SQUIB FIRE	ACS & RCS
K11-34	K11-35	RCS RING "B" SQUIB FIRE	ACS & RCS
K12-5		RETRO BIAS	IGS

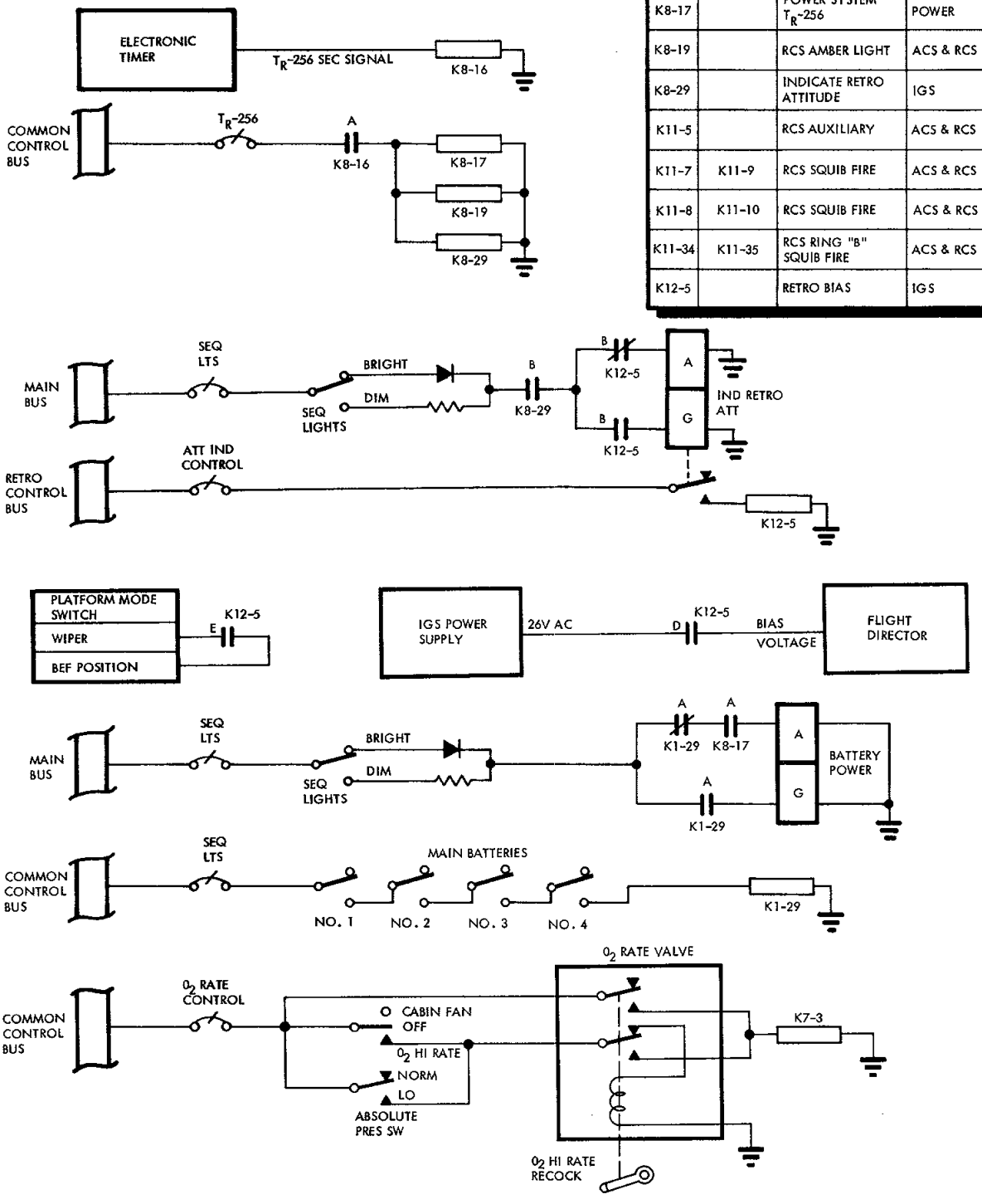


Figure 4-5 Time to Retrograde Minus 256 Seconds Sequence (Sheet 1 of 2)

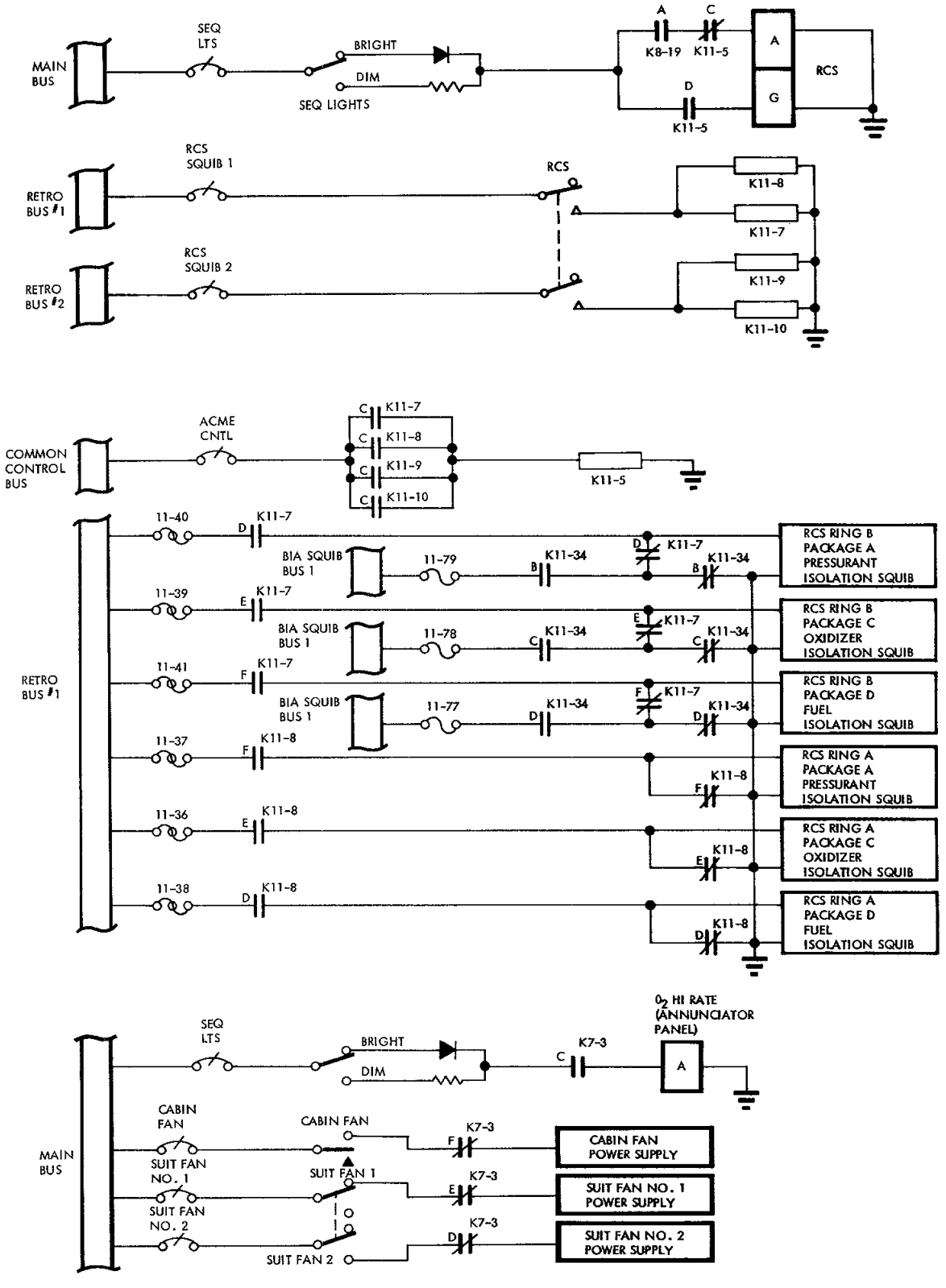


Figure 4-5 Time to Retrograde Minus 256 Seconds Sequence (Sheet 2 of 2)

through the closed SEQ LIGHTS PWR circuit breaker and the SEQ LIGHTS BRIGHT-DIM switch to the switch-indicators. The SEP SPCFT switch-indicator illuminates green.

The command pilot observes the delta-V required for insertion which is now displayed on the IVI. He fires the aft thrusters until the IVI is nulled. The spacecraft is in the required orbit. The crew places the following switches to these positions: RETRO ROCKET SQUIB to SAFE, BOOST-INSERT SQUIB to SAFE, and MAIN BATTERIES 1, 2, 3 and 4 to OFF. For the communication switches positioned at this time, refer to Section IX.

PREPARE-TO-GO TO RETROGRADE

Approximately 30 minutes before retrofire time, the crew places the C-band beacon switch to CONT and performs platform alignment procedures. Then maneuver the spacecraft to the Blunt End Forward (BEF) position.

At T_R-256 seconds (4 minutes and 16 seconds before retrofire time), the electronic timer energizes the T_R-256 second relay K8-16. (See Figure 4-5). The A contacts of K8-16 close and energize K8-17, K8-19 and K8-29. K8-17 is the Electrical Power System T_R-256 relay, and its A contacts now close to illuminate the BTRY PWR indicator amber. K8-19 is the Re-entry Control System (RCS) amber light relay, and illuminates the RCS indicator amber. K8-29 is the indicate retrograde attitude relay, and illuminates the IND RETRO ATT indicator amber.

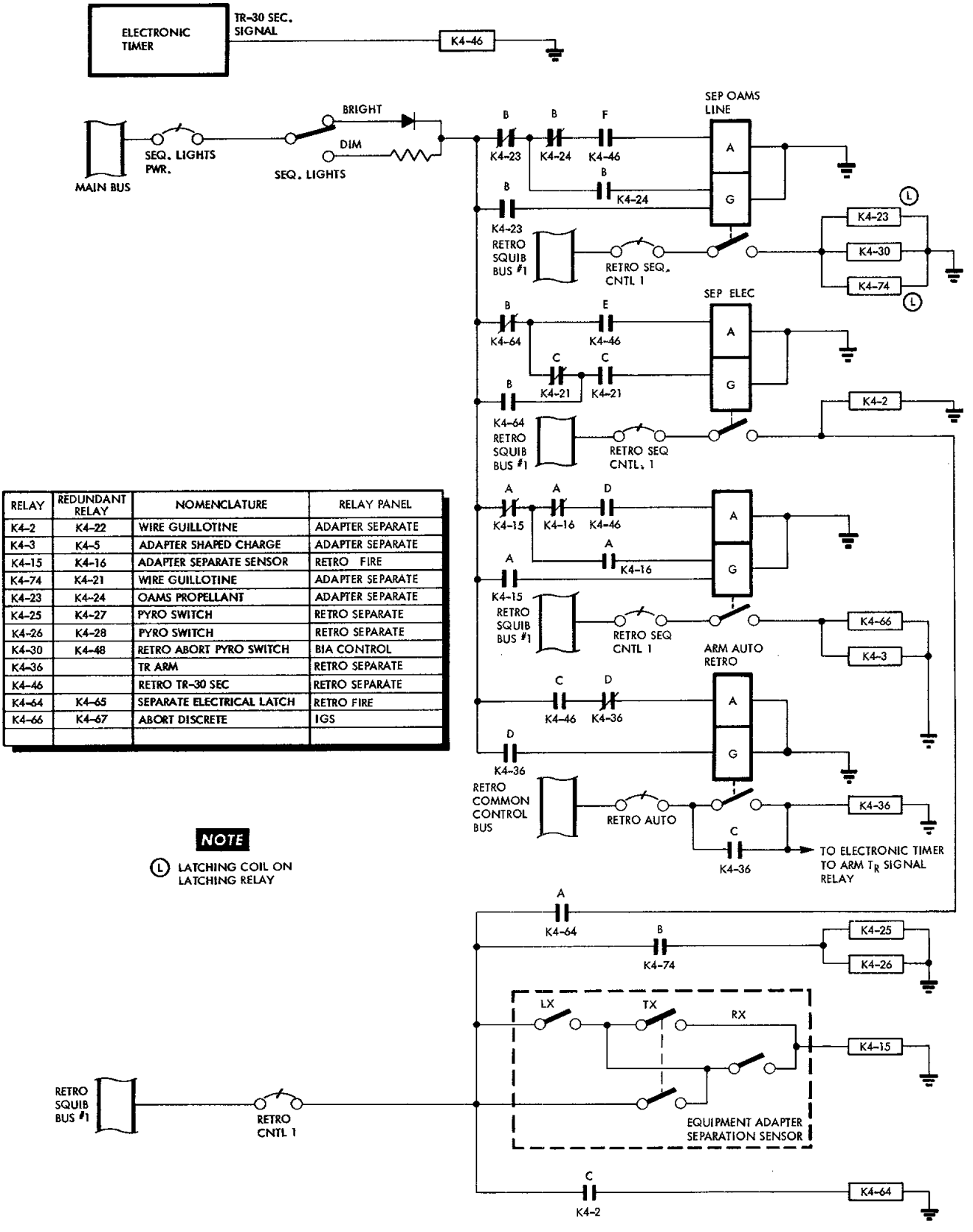
The amber BTRY PWR indicator reminds the pilot to turn on the main batteries by placing the four MAIN BATTERIES switches to the ON position. Relay K1-29 is energized through the ON position of the four battery switches. The BTRY PWR indicator illuminates green.

Depressing the amber IND RETRO ATT switch-indicator energizes the retrograde bias relay K12-5. K12-5 extinguishes the amber lamp and illuminates the green lamp of the indicator. K12-5 also applies the retrograde attitude bias voltage to the Flight Director Indicator (FDI), and electrically places the inertial platform in the BEF mode. The FDI needles can now be used to orient the spacecraft in this attitude.

Depressing the RCS switch-indicator energizes the four RCS squib fire relays K11-7, K11-8, K11-9, and K11-10. Relays K11-7 and K11-8 are energized from retrograde bus number 1 while K11-9, and K11-10 are energized from retrograde bus number 2. When any of the four RCS squib fire relays energize, the RCS auxiliary relay K11-5 is latched, changing the RCS indicator from an amber to a green indication. Relays K11-7 and K11-9 both fire the package A, C, D, pressure isolation, oxidizer isolation, and fuel isolation squibs of ring B. Relays K11-8 and K11-10 fire the package A, C, D, pressure isolation, oxidizer isolation, and fuel isolation squibs of ring A. The RCS RING A and RING B switches are now placed to ACME, and the attitude controller is operated to fire and test the RCS thrusters.

O_2 high rate flow is initiated after the T_R-256 second sequences at the option of the crew. When the CABIN FAN switch is placed to the O_2 HI RATE position, the disconnect relay K7-3 is energized. K7-3 removes power from the cabin fan power supply and the two suit power supplies, and illuminates the amber O_2 HI RATE indicator.

After the T_R-256 sequence, re-entry communications are selected, as discussed in Section IX.



RELAY	REDUNDANT RELAY	NOMENCLATURE	RELAY PANEL
K4-2	K4-22	WIRE GUILLOTINE	ADAPTER SEPARATE
K4-3	K4-5	ADAPTER SHAPED CHARGE	ADAPTER SEPARATE
K4-15	K4-16	ADAPTER SEPARATE SENSOR	RETRO FIRE
K4-74	K4-21	WIRE GUILLOTINE	ADAPTER SEPARATE
K4-23	K4-24	OAMS PROPELLANT	ADAPTER SEPARATE
K4-25	K4-27	PYRO SWITCH	RETRO SEPARATE
K4-26	K4-28	PYRO SWITCH	RETRO SEPARATE
K4-30	K4-48	RETRO ABORT PYRO SWITCH	BIA CONTROL
K4-36		TR ARM	RETRO SEPARATE
K4-46		RETRO TR-30 SEC	RETRO SEPARATE
K4-64	K4-65	SEPARATE ELECTRICAL LATCH	RETRO FIRE
K4-66	K4-67	ABORT DISCRETE	IGS

Figure 4-6 Time to Retrograde Minus 30 Seconds Sequence (Sheet 1 of 2)

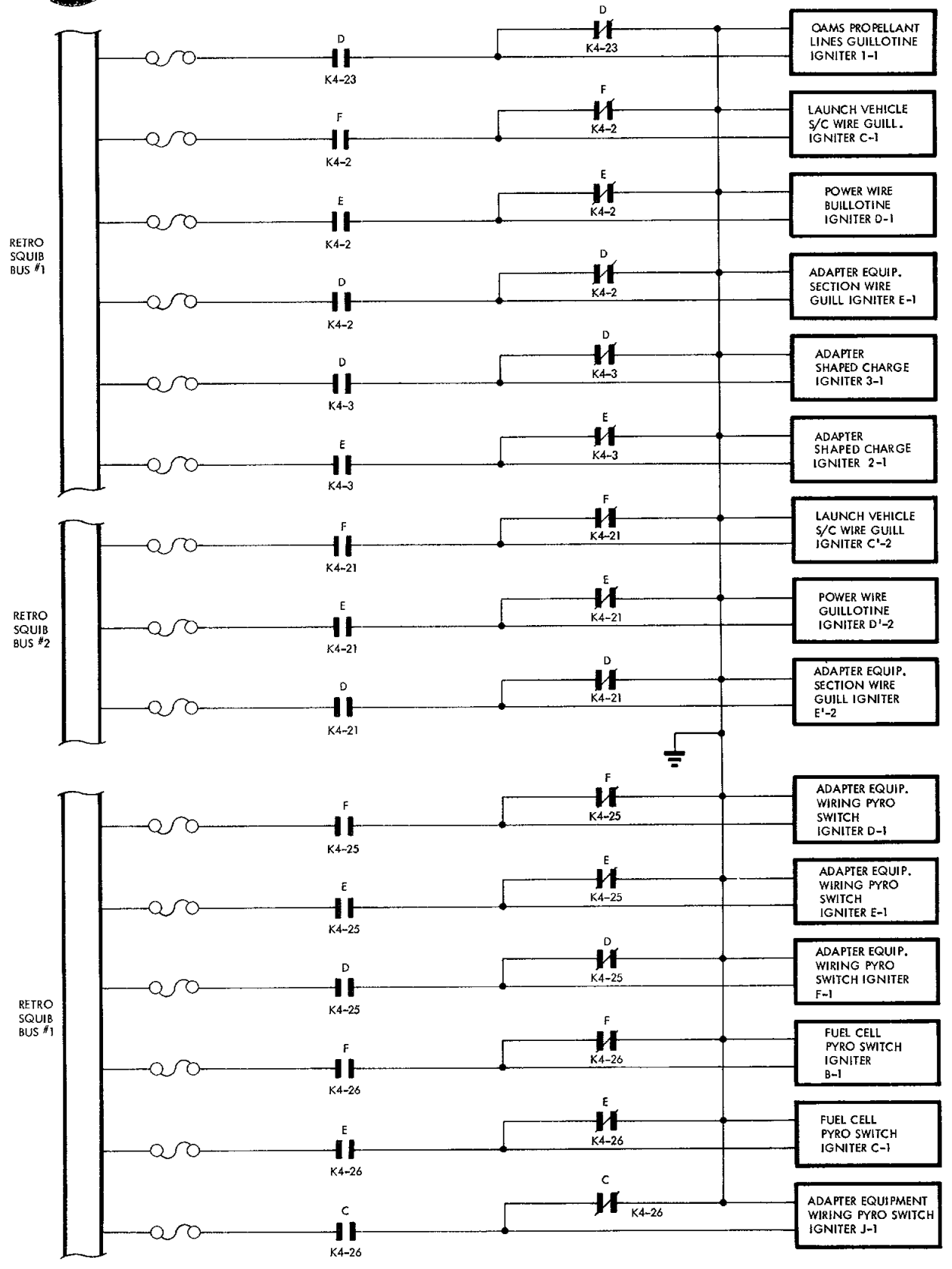


Figure 4-6 Time to Retrograde Minus 30 Seconds Sequence (Sheet 2 of 2)

RETROGRADE MINUS 30 SECONDS

Thirty seconds prior to retrograde (T_R-30 seconds), the electronic timer initiates a contact closure. This closure energizes the retrograde T_R-30 seconds relay K4-46, which illuminates the SEP OAMS LINE, SEP ELEC, SEP ADAPT, and ARM AUTO RETRO indicators amber. Figure 4-6 shows a logic presentation of the T_R-30 second sequence. Some of the sequences shown in Figure 4-6 such as SEP OAMS LINES, SEP ELECT, and SEP ADAPT are performed redundantly. However, for simplicity only the sequences powered from retrograde squib bus number 1 is shown.

As soon as the command pilot observes that the four indicators have illuminated amber, he depresses and releases the SEP OAMS LINE switch-indicator. This switch closure energizes the OAMS propellant line guillotine relay K4-23 and the retrograde abort pyrotechnic squib relay K4-30. K4-23 changes the SEP OAMS LINE indication from amber to green, fires the OAMS propellant lines guillotine igniter 1-1, and then energizes pyrotechnic switch relays K4-25 and K4-26. Relay K4-25 and K4-26 energize pyrotechnic switches B, C, D, E, F and J.

Next, the command pilot depresses and releases the SEP ELEC switch-indicator which energizes wire guillotine relay K4-2. K4-2 ignites wire guillotine C, D and E and energizes the separate electrical latch relay K4-64. When K4-64 energizes, the SEP ELEC switch-indicator changes from amber to green. Then, the command pilot initiates the equipment adapter separation sequence by depressing and releasing the SEP ADAPT switch-indicator. Closure of the SEP ADAPT switch energizes the adapter shaped charge relay K4-3 and abort discrete relay K4-66. K4-3 detonates shaped charge igniter 2-1 and 3-1. The adapter equipment section separates, and separation is sensed by three toggle sensor switches. The switches close when the physical separation is one and one half inches. The closure of any two switches

energizes the adapter separate sensor relay K4-15. K4-15 changes the SEP ADAPT switch-indicator from amber to green. The green SEP ADAPT light informs the crew that the adapter equipment section has been jettisoned from the spacecraft. K4-66 sends the abort transfer discrete to the on-board computer.

Lastly, the command pilot depresses and releases the ARM AUTO RETRO switch-indicator. The ARM AUTO RETRO switch latches the T_R arm relay K4-36. This relay changes the indication from amber to green and arms the electronic timer for the T_R relay contact closure. The four RETRO ROCKET SQUIB switches are now moved to the ARM position.

RETROGRADE SEQUENCE

A logic diagram of the retrograde sequence is shown in Figure 4-7. As discussed previously, whenever a sequence is initiated from retrograde squib bus number 1, there is an identical redundant sequence initiated from retrograde squib bus number 2. The retrograde sequence is initiated by the T_R signal from the electronic timer. The redundant sequence is initiated manually by the crew.

At Retrograde (T_R), the electronic timer latches the T_R signal relay K4-34. The T_R signal relay in the latched condition energizes the retrorocket automatic fire relay K4-7. K4-34 also energizes the 45-second time delay relay K4-4, initiates a 5.5-seconds, 11.0-seconds, and a 16.5-second time delay, and deactivates the IGS platform free mode. The retrorocket automatic fire relay redundantly fires retrorocket number 1 from retrograde squib bus number 1 and number 2. At the end of the 5.5-second time delay, the retrorocket automatic fire relay K4-9 is energized. K4-9 ignites retrorocket number 3 from retrograde squib bus number 1 and number 2. Retrorocket number 2 is redundantly ignited from retrograde squib

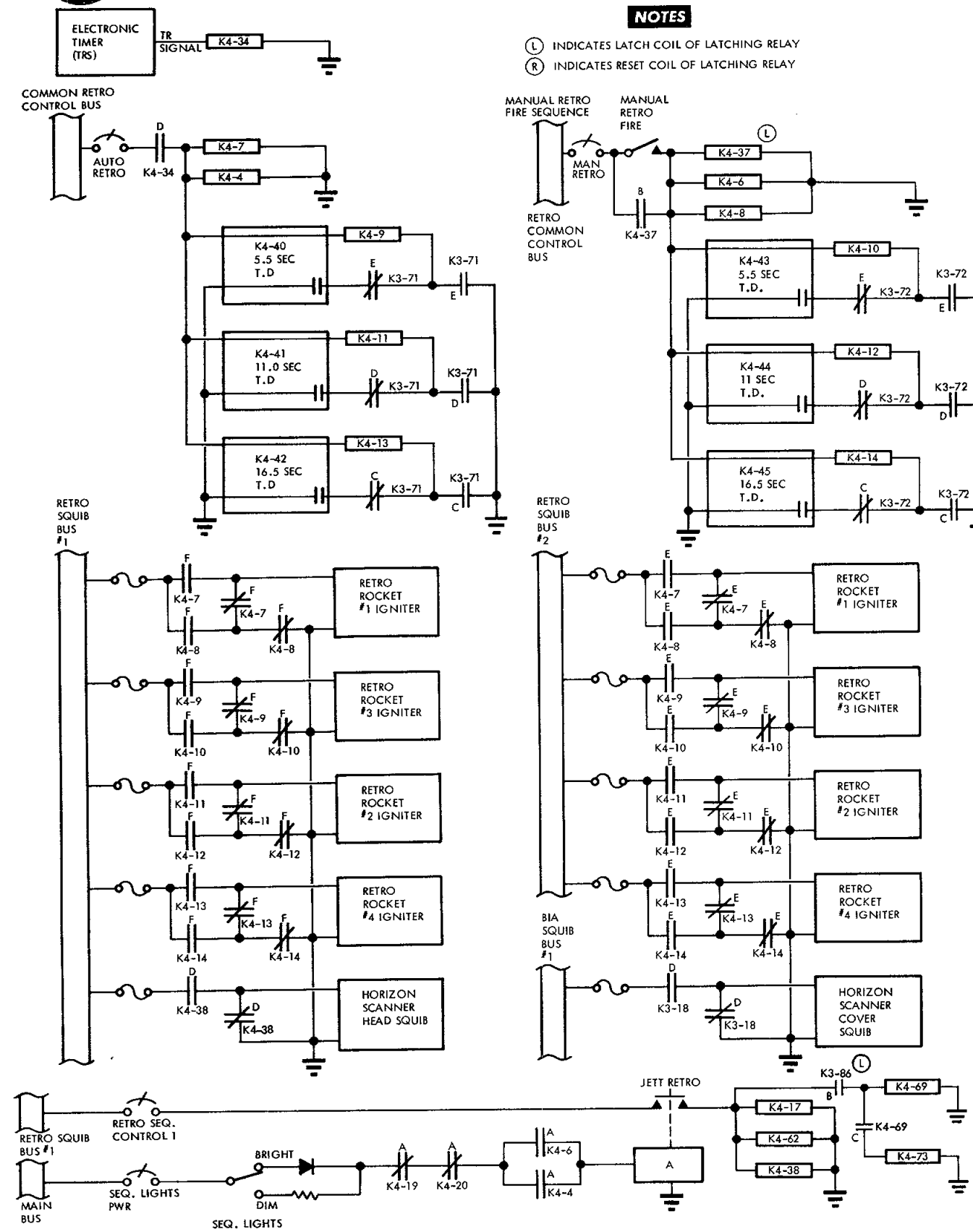
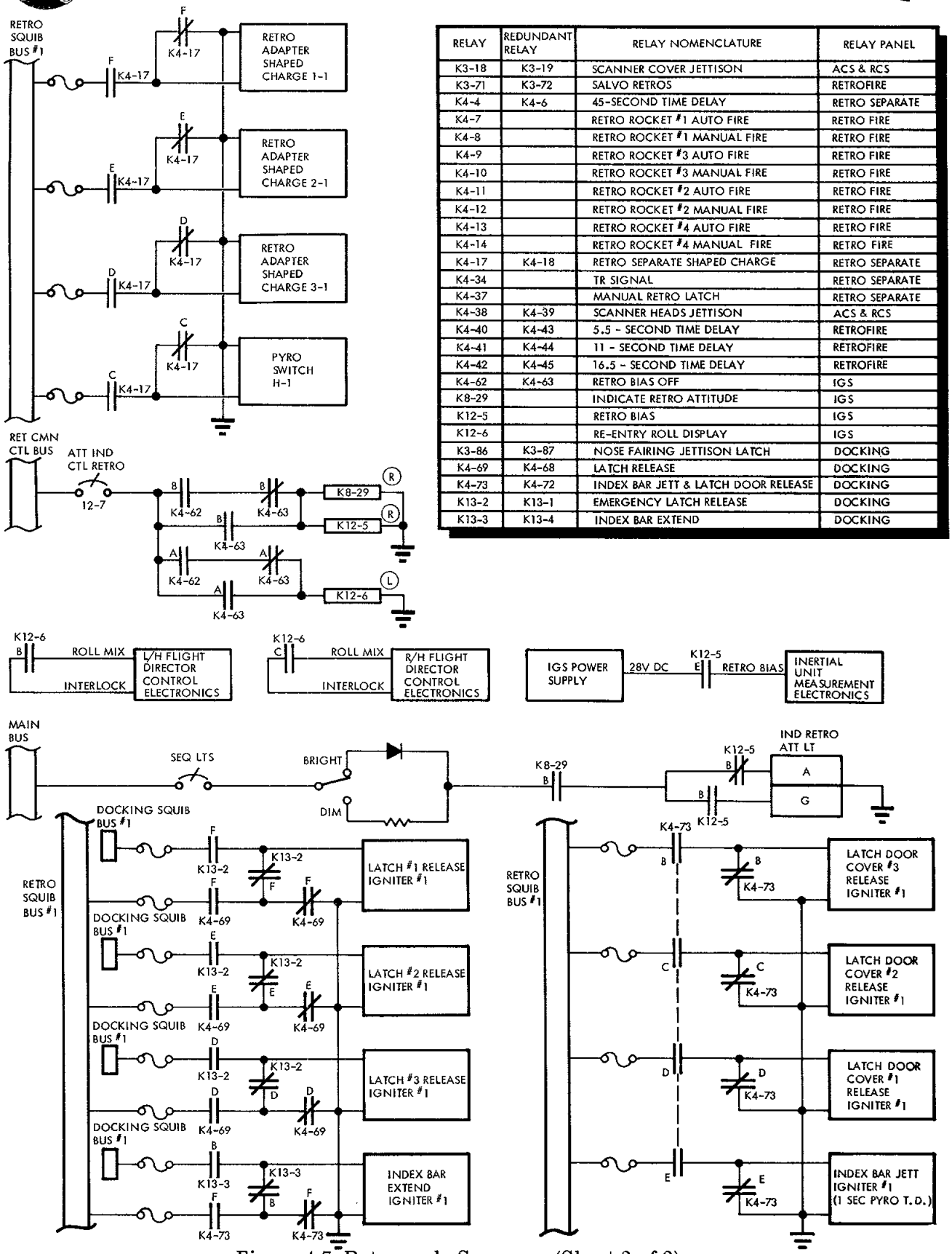


Figure 4-7 Retrograde Sequence (Sheet 1 of 2)



RELAY	REDUNDANT RELAY	RELAY NOMENCLATURE	RELAY PANEL
K3-18	K3-19	SCANNER COVER JETTISON	ACS & RCS
K3-71	K3-72	SALVO RETROS	RETROFIRE
K4-4	K4-6	45-SECOND TIME DELAY	RETRO SEPARATE
K4-7		RETRO ROCKET #1 AUTO FIRE	RETRO FIRE
K4-8		RETRO ROCKET #1 MANUAL FIRE	RETRO FIRE
K4-9		RETRO ROCKET #3 AUTO FIRE	RETRO FIRE
K4-10		RETRO ROCKET #3 MANUAL FIRE	RETRO FIRE
K4-11		RETRO ROCKET #2 AUTO FIRE	RETRO FIRE
K4-12		RETRO ROCKET #2 MANUAL FIRE	RETRO FIRE
K4-13		RETRO ROCKET #4 AUTO FIRE	RETRO FIRE
K4-14		RETRO ROCKET #4 MANUAL FIRE	RETRO FIRE
K4-17	K4-18	RETRO SEPARATE SHAPED CHARGE	RETRO SEPARATE
K4-34		TR SIGNAL	RETRO SEPARATE
K4-37		MANUAL RETRO LATCH	RETRO SEPARATE
K4-38	K4-39	SCANNER HEADS JETTISON	ACS & RCS
K4-40	K4-43	5.5 - SECOND TIME DELAY	RETROFIRE
K4-41	K4-44	11 - SECOND TIME DELAY	RETROFIRE
K4-42	K4-45	16.5 - SECOND TIME DELAY	RETROFIRE
K4-62	K4-63	RETRO BIAS OFF	IGS
K8-29		INDICATE RETRO ATTITUDE	IGS
K12-5		RETRO BIAS	IGS
K12-6		RE-ENTRY ROLL DISPLAY	IGS
K3-86	K3-87	NOSE FAIRING JETTISON LATCH	DOCKING
K4-69	K4-68	LATCH RELEASE	DOCKING
K4-73	K4-72	INDEX BAR JETT & LATCH DOOR RELEASE	DOCKING
K13-2	K13-1	EMERGENCY LATCH RELEASE	DOCKING
K13-3	K13-4	INDEX BAR EXTEND	DOCKING

Figure 4-7 Retrograde Sequence (Sheet 2 of 2)

bus number 1 and number 2 when the retrorocket automatic fire relay K4-11 energizes at the end of the 11.0-second time delay. Retrorocket automatic fire relay K4-13 is energized at the end of the 16.5-second time delay. K4-23 redundantly fires retrorocket number 4 from retrograde squib bus number 1 and number 2.

In order to assure retrograde rocket ignition, the command pilot initiates manual retrograde ignition by depressing and releasing the MAN FIRE RETRO switch-indicator approximately one second after automatic retrofire initiation. The MAN FIRE RETRO switch latches the manual retrograde latch relay K4-37, energizes retrorocket manual fire relay K4-8, and initiates the 45-second time delay relay K4-6. This switch also initiates the 5.5-second, 11-second and 16.5-second time delays. The 5.5, 11 and 16.5-second time delays energize retrorocket manual fire relays K4-10, K4-12 and K4-14 respectively, which in turn fire retrorockets number 3, number 2, and number 4 respectively. Retrorocket number 1 is fired by K4-8. As in automatic retrorocket fire, each retrorocket is fired from retrograde squib bus number 1 and number 2. Twenty-two seconds after retrofire is initiated, the last retrorocket ceases firing. The command pilot moves the JETT RETRO SQUIB ARM switch on the left switch circuit breaker panel from SAFE to ARM. Forty-five seconds after retrograde ignition, K4-4 or K4-6 energizes and illuminates the JETT RETRO lamp on the main instrument panel.

As soon as the command pilot observes the JETT RETRO indicator is amber, he depresses and releases this switch-indicator. The switch energizes the retrograde separate shaped charge relay K4-17, the retrograde bias off relay K4-62, and the horizon scanner heads jettison relay K4-38. Relay K4-17 fires retrograde adapter shaped charge igniter 1-1, 2-1, and 3-1 and pyrotechnic switch H-1. Relay K4-62 latches the re-entry roll display relay K12-6 removing roll mix

interlock from the flight director controller. K4-62 also resets two latch relays: the retrograde bias relay K12-5 and the indicate retrograde attitude relay K8-29. Relay K8-29 extinguishes the IND RETRO ATT indicator. K4-18 fires horizon scanner cover squib 1-1 if it was not fired previously during the boost phase. K4-38 ignites the horizon scanner head squib 1-1 through an 80-millisecond pyrotechnic time delay and jettisons the scanner head. The firing of pyrotechnic switch H-1 extinguishes the SEP ELEC, SEP ADAPT, SEP OAMS, ARM AUTO RETRO and JETT RETRO indicators.

On spacecraft 6 and 8 through 12, the JETT RETRO switch also energizes latch release relay K4-69 through the B contacts of the nose fairing jettison latch relay K3-86. K4-69 fires the release igniters of docking latches 1, 2 and 3 to jettison them. K4-69 also energizes the index bar jettison and latch door release relay K4-73. K4-73 fires three latch door cover release igniters. These igniters release the latch doors which cover the ports left by the jettisoned docking latches. K4-73 also jettisons the docking index bar. If the bar was not extended previously, it is first extended and then jettisoned. These functions are not a part of the retrograde sequence during an abort if the abort occurs prior to nose fairing jettison.

RE-ENTRY

After the retrograde adapter and horizon scanner heads have been jettisoned, the command pilot places the RETRO PWR and RETRO JETT squib switches to SAFE. Using the attitude controller and the FDI needles, he rolls the spacecraft 180 degrees so that the horizon is visible in the upper portion of his cabin window. He changes the ATTITUDE CONTROL mode select switch on the main instrument panel from PULSE to RATE CMD (RE-ENT). The command pilot uses attitude control and

maneuvering electronics and the attitude controller to control the roll attitude during approximately the next 10 minutes in which the altitude diminishes to 400,000 feet. As this altitude the FDI roll needles start to move, the computer START light illuminates, and the computer begins to calculate the point of impact. The command pilot changes the ATTITUDE CONTROL mode select switch from RATE CMD (RE-ENT) to RE-ENT. The computer now computes the roll attitude for optimum re-entry lift and also automatically controls the roll attitude. During approximately the next 10 minutes, the altitude decreases to 100,000 feet. At this altitude, the altimeter indicator begins to come off the peg. At 80,000 feet, the computer commands the spacecraft to assume the best attitude for drogue parachute deployment. Then the command pilot places all guidance and electronic switches to OFF.

ABORT MODES

An abort is an unscheduled termination of the spacecraft mission. An abort may be initiated at any time during the spacecraft mission. In all cases the actual abort sequence has to be initiated by the crew after an abort command has been received. An abort indication consists of illumination of the ABORT indicators located on the command pilot and pilot's panels. The ABORT indicator may be illuminated by three different methods. During pre-launch prior to umbilical disconnect, the ABORT indicator may be illuminated from the blockhouse via hardline through the launch vehicle tail plug connector. After umbilical release, the ABORT indicator may be illuminated by ground command to the spacecraft via a channel of the DCS or by ground command to the launch vehicle to shutdown the booster.

The abort sequence is part of the Sequence System. The abort sequence comprises the abort indicators, controls, relays, and pyrotechnics. The part of the abort sequence which the crew make use of is determined by the abort mode in effect at the time when the abort command is received or the decision to abort is made.

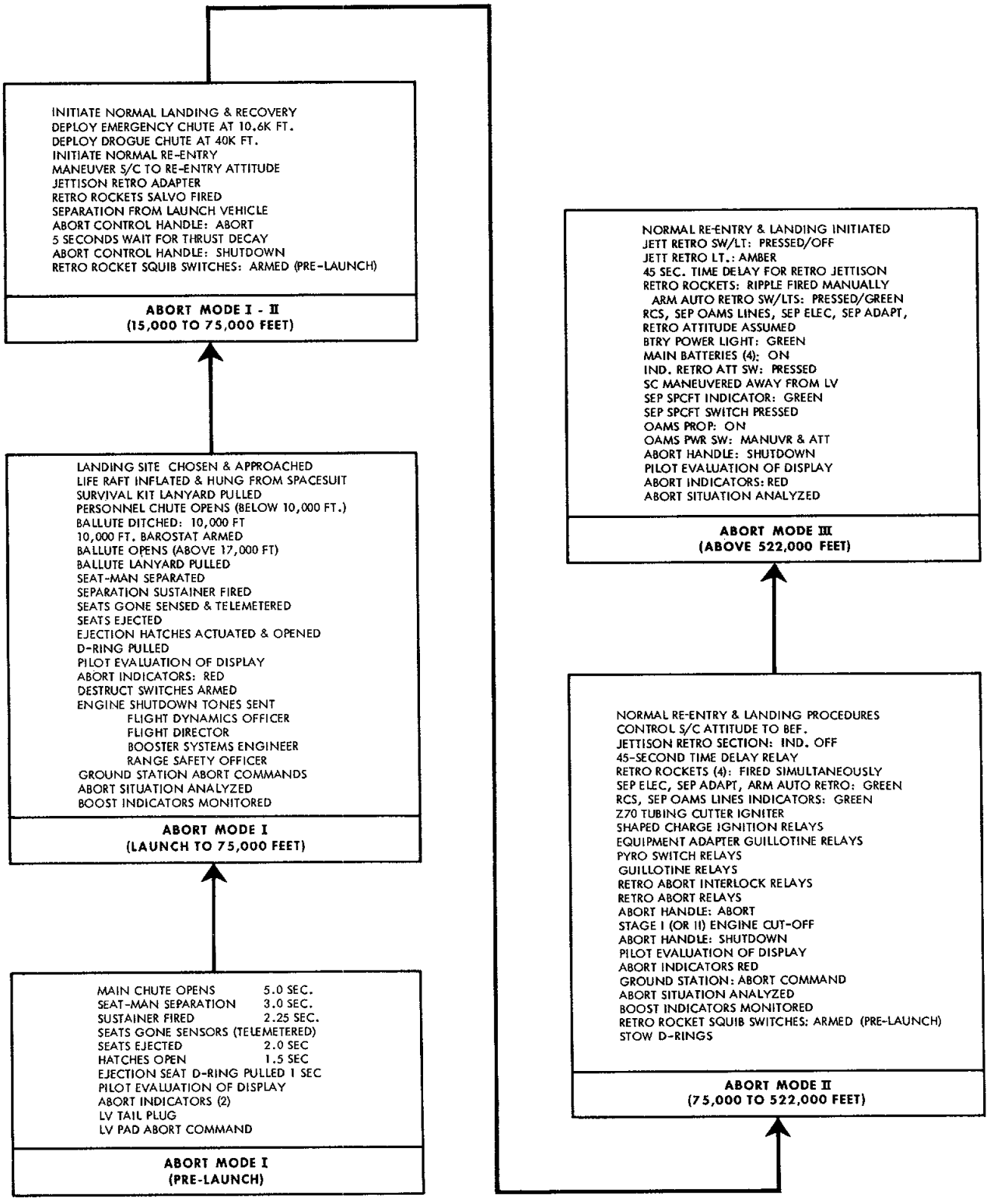


Figure 4-8 Abort Modes Simplified Block Diagram

The abort mode to be used at any time during the mission is determined by calculations made on the ground and depends on the altitude and velocity attained by the spacecraft. The critical abort altitudes are 15,000 feet, 75,000 feet, and 522,000 feet. The spacecraft reaches 15,000 feet approximately 50 seconds after lift-off, 75,000 feet approximately 100 seconds after lift-off, and 522,000 feet approximately 310 seconds after lift-off. Below 15,000 feet, seat ejection (mode I) is used. Between 15,000 and 75,000 feet, ride-it-out abort (mode I-II) is used. Between 75,000 and 522,000 feet, modified re-entry (mode II) is used. Above 522,000 feet normal re-entry (mode III) is used, except that the spacecraft electronic timer does not illuminate the sequential indicators amber when the time to press them occurs, unless the timer is updated by ground command. Figure 4-8 presents a simplified block diagram of the abort sequences in each of the three modes.

Abort Mode I

When an abort becomes necessary during pre-launch, it is accomplished by using abort mode I. The abort command is given from the blockhouse by hardline through the launch vehicle tail plug connector. The command lights both ABORT indicators on the command pilot and pilot's panels. When the pilots see this display, they immediately pull the D-rings attached to their ejection seats. When one D-ring is pulled, both ejection systems are energized. One-half seconds later, the hatches are open, and one-half second after that the seats have been ejected. Sensors detect the ejection of the seats and notify the blockhouse that the pilots are out of the spacecraft. One-quarter second after the seats are ejected, a sustainer rocket under each seat is fired, which extends the distance between the pilots and the launch vehicle. Then a pyrotechnic ignites and separates the ejection seat from the pilots. Two seconds after sustainer ignition, the main

parachutes have opened and the pilots are lowered safely to the ground. For illustrations and fuller descriptions of the equipment used for seat ejection abort, refer to Section III of this manual.

After normal lift-off, and before the Gemini-Titan reaches an alitutde of 15,000 feet, an abort condition could develop. The crew monitor their booster indicators so that they are aware at all times of the manner in which the flight is proceeding. Booster operation data is telemetered to the ground for analysis and interpretation. The range safety officer, the booster systems engineer, the flight director, or the flight dynamics officer, who are on the ground, amy decide that danger is imminent and an abort mandatory.

A channel of the DCS is used to send the abort command to the spacecraft and ground commands are sent to the launch vehicle to shutdown the booster engines. Then the engine shutdown tones are received, the destruct switches of the launch vehicle are armed. The two ENGINE I indicators and both ABORT indicators illuminate red. The command pilot and pilot evaluate these displays and pull the D-rings. The hatches open and the pilots in their seats are ejected. Refer to Section III for a description of the remainder of this sequence.

Abort Mode I - II

Abort mode I - II is the ride-it-out abort mode. It is effective at altitudes between 15,000 and 75,000 feet approximately 50 seconds to 100 seconds after lift-off. Abort mode I - II is used when a mode I abort is inadvisable and when a delay to permit entry into the mode II conditions is impractical. The crew however has the option to eject or to ride-it-out depending upon their assessment of the abort conditions. Therefore the D-rings are not stowed during the I - II mode.

Abort mode I - II begins during stage I boost approximately 50 seconds after lift-off. If an abort condition develops, and the crew elect to ride it out, the command pilot moves the abort control handle from NORMAL to SHUTDOWN. He waits 5 seconds for booster thrust to decay, then moves the handle from SHUTDOWN to ABORT.

The retrograde abort relays and the retrograde abort interlock relays are energized. These relays arm the buses needed for abort action. The retrograde common control bus is armed from the common control bus. Retrograde squib buses number 1 and number 2 are armed from OAMS squib buses number 1 and number 2. On spacecraft 5 only, spacecraft separation squib buses number 1 and number 2 are armed from Boost Insert Abort (BIA) squib buses number 1 and number 2. Two parallel circuits are used for redundancy. This arming of buses by means of relays eliminates the motion of throwing the switch ordinarily required to arm the buses. Then, in rapid sucession, wire guillotine relays, pyrotechnic switch relays, and shaped charge igniter relays are energized. The relays ignite the pyrotechnics at the equipment adapter/retrograde adapter mating line, and the vehicles separate. Then, the four retrorockets are salvo fired and the spacecraft thrusts away from the launch vehicle.

If the abort altitude is between 15,000 and 25,000 feet, the retrograde adapter is jettisoned 7 seconds after retrorocket salvo fire is initiated. If the abort altitude is between 25,000 and 75,000 feet, the retrograde adapter is jettisoned 45 seconds after salvo fire.

After retrograde adapter jettison, the spacecraft is maneuvered to the re-entry attitude. If the abort altitude is above 40,000 feet, the drogue parachute is deployed at 40,000 feet, and the main parachute at 10,600 feet. If the drogue

parachute fails or has not been deployed before the spacecraft descents to 10,600 feet, the emergency main parachute switch is used to deploy the main parachute.

If one of the two first stage engines should fail and the launch vehicle is above 40,000 feet, the pilots may elect to remain with the spacecraft until the operating engine has boosted them to 75,000 feet. At this altitude, abort mode I - II becomes inapplicable.

Abort Mode II

Abort mode II becomes effective above 75,000 feet. At approximately 100 seconds after lift-off on a normal mission, the launch vehicle has boosted the spacecraft to an altitude of 75,000 feet. Ground station computers calculate the time for changeover from abort mode I - II to abort mode II. The ground station notifies the crew via the uhf communications link of the change to abort mode II. Both the command pilot and pilot acknowledge the change via the same link, and stow the ejection seat handles (D-ring). Initiation of abort mode I above 75,000 feet could be disastrous.

Abort mode II begines during stage 1 boost before booster engine cutoff and ends during stage 2 boost before second stage engine cutoff. The crew continues to monitor the booster indicators. If they should notice an abort situation developing, they analyze it. The decision to abort may be theirs or it may come from the ground. If a ground station sends the command to abort, both ABORT indicators illuminate red. In abort mode II, the command pilot must act. He moves the abort handle to the SHUTDOWN position. The operating engine is cutoff. Since launch vehicle destruct is imminent and escape from the fireball is urgent, he moves the ABORT handle to ABORT. The spacecraft is separated from the launch vehicle at the equipment adapter/retrograde adapter mating line. The retrorockets,

armed by four RETRO ROCKET SQUIB switches during pre-launch checkoff, are salvo fired, propelling the spacecraft away from the launch vehicle.

Since orbital velocity could not have been reached below 522,000 feet, the spacecraft immediately begins a re-entry trajectory. The spacecraft is maneuvered to the retrograde blunt end forward attitude, the retrograde section is jettisoned, and normal landing procedures are initiated.

Abort Mode III

At approximately 310 seconds after lift-off, the launch vehicle reaches the altitude of 522,000 feet and a velocity of approximately 21,000 feet per second. The ground station commands a change from abort mode II to abort mode III via the uhf link.

If an abort after this time should become necessary, the ABORT indicators would be illuminated red. The command pilot responds and moves the ABORT handle to the SHUTDOWN position. The shutdown command is thus given to the second stage engine. The ABORT handle remains in the SHUTDOWN position. The command pilot then presses the SEP SPCFT switch-indicator on the main instrument panel. This switch fires the shaped charges and severs the wiring at the launch vehicle/spacecraft mating line as described earlier. OAMS thrust is applied to put distance between the second stage and the spacecraft. The crew perform the T_R-256 seconds and the T_R-30 seconds procedures, using the main instrument panel switch-indicators. After retrofire has been initiated manually, normal re-entry, landing, and post-landing procedures are followed.

ABORT SEQUENCE

The abort sequence described herein occurs during abort modes II and I - II. The

description covers the series of events which the abort control handle and Figure 4-9 shows the electrical circuits which cause the abort sequence to occur. Figure 4-9 includes the switches, circuit breakers, buses, relays, and pyrotechnic igniters. A table of Figure 4-9 gives the names, reference designations and relay panel locations of the relays and redundant relays of the abort sequence. The redundant relays, their buses, fuses, and squibs (with a few exceptions) are not shown, since the circuitry and end results are identical with those shown. The ommission is made to maintain clarity and simplicity.

Abort mode I, the seat ejection mode, is not covered here. The events of this mode are discussed in Section III of this Manual.

Abort mode III is executed by performing a launch vehicle engine shutdown, a spacecraft separation sequence and a retrograde sequence. Separation and retrograde in abort mode III differs from normal separation and retrograde in that the abort sequence is performed without cues from the indicators on the main instrument panel. The electrical circuits however are identical with those shown in the shutdown sequence (Figure 4-9), the spacecraft separation (Figure 4-4), the T_R-256 seconds sequence (Figure 4-5), the T_R-30 seconds sequence (Figure 4-6).

Shutdown

When the command pilot moves the abort control handle to SHUTDOWN, the SHUTDOWN switch is closed. BIA common control bus power is applied to the launch vehicle engine shutdown signal relays K3-28 and K3-49. This power is also applied to the engine shutdown relays in the Tital Launch Vehicle. The operating engine(s) are cut off. As K3-48 and K3-49 energize, common control bus power is applied through their B contacts to the spacecraft instrumentation programmer. The programmer

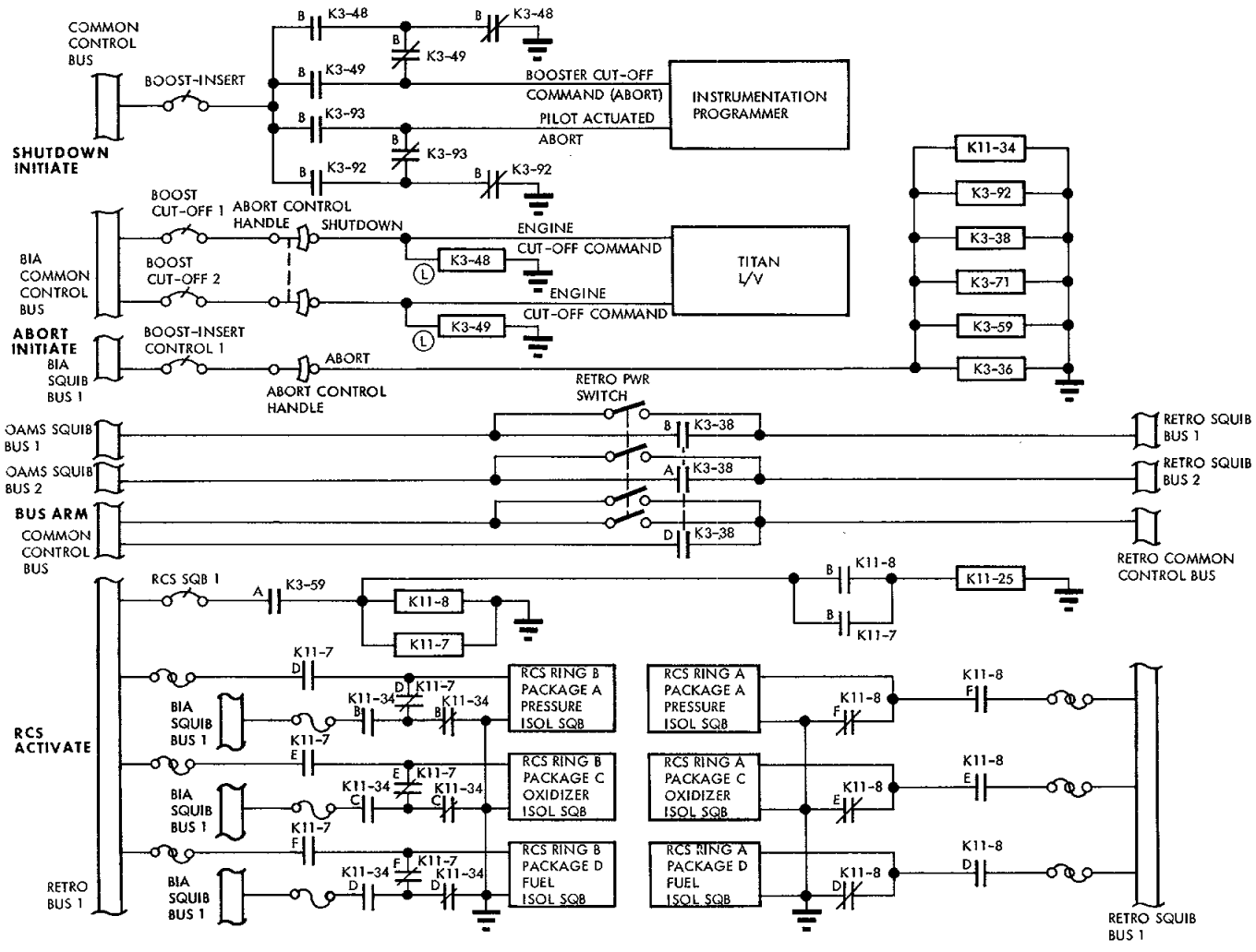


TABLE OF RELAYS

RELAY	REDUNDANT RELAY	RELAY NOMENCLATURE	RELAY PANEL
3-36	3-37	RETRO ABORT	RETRO SEQUENCE
3-38	3-39	SQUIB BUS ABORT	POWER DISTRIBUTION
3-48	3-49	L/V ENGINE SHUTDOWN SIGNAL	INSTRUMENTATION SEQUENTIAL MONITOR
3-59	3-60	ACS ABORT	ACS-RCS
3-71	3-72	SALVO RETRO	RETRO FIRE
3-92	3-93	INSTRUMENTATION ABORT	INSTRUMENTATION SEQUENTIAL MONITOR
4-2	4-22	WIRE GUILLOTINE	RETRO SEQUENCE
4-3	4-5	ADAPTER SHAPED CHARGE	RETRO SEQUENCE
4-4	4-6	45-SECOND TIME DELAY	RETRO SEPARATION
4-7	4-8	RETRO ROCKET 1 AUTO FIRE	RETRO FIRE
4-9	4-10	RETRO ROCKET 3 AUTO FIRE	RETRO FIRE
4-11	4-12	RETRO ROCKET 2 AUTO FIRE	RETRO FIRE
4-13	4-14	RETRO ROCKET 4 AUTO FIRE	RETRO FIRE
4-23	4-24	OAMS LINES GUILLOTINE LATCH	RETRO SEQUENCE
4-25	4-27	PYRO SWITCH	RETRO SEPARATION
4-26	4-28	PYRO SWITCH	RETRO SEPARATION
4-30	4-48	L/V PYRO SWITCH ABORT	BIA CONTROL
4-40	4-43	5.5 SECOND TIME DELAY	RETRO FIRE
4-41	4-44	11.0-SECOND TIME DELAY	RETRO FIRE
4-42	4-45	16.5-SECOND TIME DELAY	RETRO FIRE
4-64		SEPARATE ELECTRICAL LATCH	RETRO FIRE
4-66	4-67	ABORT DISCRETE	IGS
4-74	4-21	WIRE GUILLOTINE LATCH	RETRO SEQUENCE
11-7	11-9	RCS SQUIB FIRE	ACS-RCS
11-8	11-10	RCS SQUIB FIRE	ACS-RCS
11-25	11-26	RETRO ABORT INTERLOCK	RETRO SEQUENCE
11-34	11-35	RCS RING B SQUIB FIRE	ACS-RCS

Figure 4-9 Abort Sequence (Sheet 1 of 2)

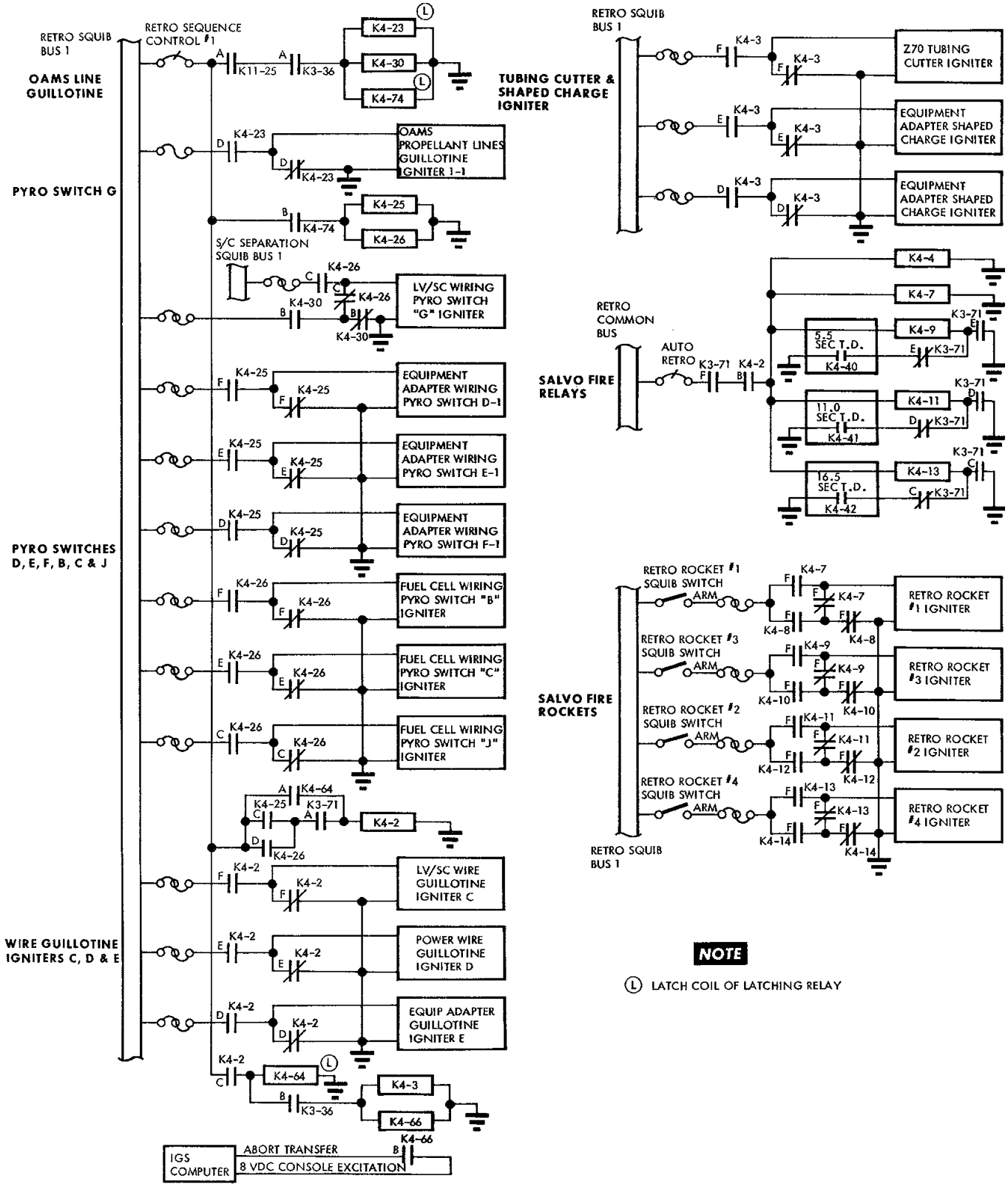


Figure 4-9 Abort Sequence (Sheet 2 of 2)

encodes the voltage from this bus as the booster cutoff command signal for telemetry transmission to the ground tracking station.

Abort Initiate

When the command pilots moves the abort control handle to ABORT, numerous relays are energized, as shown of Figure 4-9. However five of these relays are key relays in that they control the principal abort operations. These operations are: (1) telemetry of the abort action to the ground; (2) arming of the retrograde buses; (3) activation of the RCS; (4) separation of the spacecraft from the launch vehicle; and (5) salvo firing of the retro rockets.

The relays which control those operations are: (1) the instrumentation abort relay, K3-92; (2) the squib bus abort relay K3-38; (3) the Attitude Control System abort relay K3-59; (4) the retrograde abort relay K3-36; and (5) the salvo retrograde relay K3-71.

Abort Telemetry

When the instrumentation abort relay K3-92 is energized by the abort switch, its B contacts connect common control bus power to the spacecraft instrumentation programmer. The programmer encodes this signal as the pilot actuated abort signal for telemetery transmission to the ground.

Abort Squib Bus Arming

Abort, if it occurs, requires that power for the circuits used in the retrograde phase of the mission become immediately available. When the abort switch is closed, squib bus power is applied to K3-38. K3-38 arms the retrograde squib buses 1 and 2 and the retrograde common control bus.

Re-entry Control System (RCS) Activation

Re-entry immediately and automatically follows an abort. Re-entry requires the use of the RCS for control of the spacecraft during this phase. Hence the RCS is activated. Activation involves opening and pressurizing the RCS fuel and oxidizer lines. This is done by firing the squibs of the fuel, oxidizer, and pressurant packages.

In operation, the abort switch applies BIA squib bus power to the Attitude Control System abort relay K3-59. K3-59 applies retrograde squib bus power to RCS (ring A) squib fire relay K11-8 and the RCS (ring B) squib fire relay K11-7. K11-8 applies retrograde squib bus power to package A, C, and D igniters of RCS ring A. The squibs thus fired open the ring A fuel and oxidizer lines and pressurize them. K11-7 applies retrograde squib bus power to similar igniter of RCS ring B with similar results.

The B contact of K11-7 and K11-8 energize the retrograde abort interlock relay K11-22. K11-25, contact A initiates the station Z70 separation sequence.

OAMS Lines and Lower Wires Guillotine

Since the retrorockets are to be fired in the abort modes controlled by the abort switch, the spacecraft must separate from the launch vehicle at station Z70. Z70 is on the mating line between the spacecraft and the equipment adapter section. To make separation complete, the OAMS propellant lines which cross this station must be sealed and guillotined.

The abort switch energizes the retrograde abort relay K3-36 which arms K4-23, the OAMS lines guillotine latch relay; K4-30, the retrograde abort pyrotechnic switch relay; and K4-74, the wire guillotine relay. When K11-25 is energized, it energizes

K4-23, K4-30, and K4-74. The D contacts of K4-23 apply power to the OAMS propellant lines guillotine igniter. The guillotine now seals and cuts the lines. Pyrotechnic switch G fires, opening the launch vehicle/spacecraft interface circuits. The lower wire gundles are guillotined. The first step toward launch vehicle/spacecraft separation has been taken.

Pyrotechnic Switch Ignition

The second step in launch vehicle/spacecraft separation is the removal of power from the hot wires crossing station Z70. These wires like the propellant lines, must also be guillotined, and the guillotine blade could cause a short circuit of the spacecraft power. Pyrotechnic switches B, C, D, E, F, G and J must be operated to remove power from the wires to be guillotined.

K3-36 and K11-25 apply power to launch vehicle/spacecraft pyrotechnic switch abort relay K4-30 and to wire guillotine latch relay K4-74, initiating pyrotechnic switch ignition. K4-30 applies power to launch vehicle/spacecraft wiring pyrotechnic switch G igniter, opening pyrotechnic switch G. K4-74 energizes pyrotechnic switch relays K4-25 and K4-26. K4-25 ignites equipment adapter pyrotechnic switches D, E and F. K4-26 ignites fuel cell wiring pyrotechnic switch B, C and J. With the operation of the pyrotechnic switches, the second step in launch vehicle/spacecraft separation has been taken.

Upper Wire Guillotine Ignition

The third step in launch vehicle/spacecraft separation is the cutting of the upper wires that cross station Z70. This is accomplished by actuating the wire guillotines. Three wire guillotines igniters must be fired: the launch vehicle/spacecraft wire guillotine igniter C, the power wire guillotine igniter D, and equipment adapter wire guillotine igniter E.

When K4-25 and K4-26 energize, they apply power through the A contacts of K3-71 to wire guillotine relay K4-2. K4-2 fires the wire guillotine igniters C, D and E cutting the station Z70 wires.

K4-2, contact C energizes the separate electrical latch relay K4-64, the adapter shaped charge relay K4-3 and the abort discrete relay K4-66. K4-64, contact A latches K4-2 in the energized position. K4-66 changes the computer from the ascent mode to the re-entry mode and enables the computer to accept re-entry data and solve the re-entry problem. K4-3 prepares the way for the fourth step in the separation of the launch vehicle from the spacecraft.

Tubing and Structural Bond Cutting

The fourth and final step is to sever the adapter skin at station Z70 and break the launch vehicle to spacecraft structural bond.

When K4-2 causes K4-3, the adapter shaped charge relay, to energize, K4-3 fires the Z70 tubing cutter igniter and the equipment adapter shaped charge igniters. The pyrotechnics complete the task of launch vehicle/spacecraft separation.

Retrorocket Salvo Fire

The retrorockets are salvo fired at the same time that the tubing and structural bond is cut. To salvo fire the retrorockets, power must be applied simultaneously to the retrorocket automatic fire relays and thus to the retrorockets. Therefore the 5.5, 11.0, and 16.5-second time delay relays must be bypassed. Contacts C, D and E of K3-71 bypass the time delay relays. When K4-2 energizes, retrograde common bus power simultaneously energizes the retrorocket automatic fire relays K4-7, K4-9, K4-11 and K4-13. As these relays energize, retrograde squib bus power is applied to the igniters of retrorockets 1, 3, 2 and 4. Salvo burn lasts

approximately 5.5 seconds.

Retrograde Section Jettison

When the retrorocket automatic fire relays are energized by K4-2, the 45-second time delay relay K4-4 is also energized. When K4-4 energizes after 45 seconds, it illuminates the JETT RETRO indicator as shown on Figure 4-7. The JETT RETRO switch-indicator is then pressed, and the retrograde section is jettisoned in a mode II abort. However, in a mode I - II abort when the altitude is between 15,000 and 25,000 feet, the switch-indicator is pressed seven seconds after the retrorockets begin firing. After the retrograde section has been jettisoned, normal re-entry and landing procedures are initiated.

SYSTEM UNITS

The Sequence System as shown in Figure 4-1 comprises the following units;
Left switch/circuit breaker panel, consisting of three rows of circuit breakers and one row of switches.

Boost and staging indicators, consisting of seven lights and three meters on the top of the command pilot and pilot's panels.

Sequence controls, consisting of two pushbutton switches, eight switch-indicators, and one indicator are located on the left side of the main instrument panel.

Re-entry switches and indicators, consisting of four switches on the main instrument panel center console and one switch, two lights, and two meters on the command pilot's panel.

Abort controls, consisting of two D-rings on the ejection seats and one abort control handle on the left side of the cabin.

Relay panels, consisting of four relay panels in the re-entry module and four in the equipment adapter and retrograde sections, and two in the rendezvous and recovery section.

Separation sensing devices, consisting of three each in the equipment adapter section and the retrograde section.

The components of the Sequence System are described below:

LEFT SWITCH/CIRCUIT BREAKER PANEL

The switches and circuit breakers on the left switch and circuit breaker panel perform important functions in the operation of the Sequence System. The top tow of circuit breakers however pertain largely to communications. The second row of circuit breakers perform functions related to the operation of the Sequence System. Their functions are as follows:

ELECTRONIC TIMER Circuit Breaker

The electronic timer circuit breaker CB8-15 applies main bus power through contact A of lift-off relay K3-11 to start the electronic timer when the lift-off signal energizes the K3-11. The timer begins counting the time-to-go to retrograde.

EVENT TIMER Circuit Breaker

The event timer circuit breaker CB8-14 applies main bus power through contact B of lift-off relay K3-11 to start the event timer when the lift-off signal energizes K3-11. The event counter counts the time since lift-off occured.

BOOST CUTOFF 1 Circuit Breaker

The boost cutoff 1 circuit breaker CB3-8 applies BIA common control bus power to the booster shutdown switch on the abort control and to the secondary guidance

(RGS-IGS) switch. This circuit breaker arms the booster shutdown circuit and the secondary guidance manual switch-over circuit.

BOOST CUTOFF 2 Circuit Breaker

The boost cutoff 2 circuit breaker CB3-21 applies BIA common control bus power redundantly to the booster shutdown switch, and supplies power for the second stage engine cutoff signal input to the computer.

RETRO AUTO Circuit Breaker

The retrograde fire automatic circuit breaker CB4-1 applies retrograde common control bus power to the ARM AUTO RETRO switch. It provides power to salvo fire the retrorockets during the abort sequence. If CB4-1 is not closed, the electronic timer T_R contact closure will not automatically fire the retrorockets.

RETRO MAN Circuit Breaker

The retrograde manual circuit breaker CB4-2 provides retrograde common control bus power for manually firing the retrorockets, and salvo firing the retrorockets with the abort control handle.

T_R-256 Circuit Breaker

The retrograde minus 256 seconds circuit breaker CB8-16 applies common control bus power to relay contacts in the electronic timer and contacts of the T_R-256 second relay. CB8-16 enables the T_R-256 second signal to illuminate amber the IND RETRO ATT, BTRY PWR, and RCS indicators on the main instrument panel.

SEQ LIGHTS POWER Circuit Breaker

The sequence lights power circuit breaker CB6-1 applies main bus power to the sequence light BRIGHT-DIM switch and to open contacts on the barostat switch arm relay and the message acceptance pulse relay.

SEQ LIGHTS CONTROL Circuit Breaker

The sequence lights control circuit breaker CB1-13 applies common control bus power through the four MAIN BATTERIES switches to relay K1-29. When the main battery power indicator relay K1-29 is energized, the BTRY PWR indicator on the main instrument panel is illuminated green.

The third row of circuit breakers on the left switch/circuit breaker panel perform functions related to the Sequential System. The functions are the following:

ATT IND CNTL RETRO Circuit Breaker

The attitude indicate control retrograde circuit breaker CB12-7 applies retrograde common control bus power to the IND RETRO ATT switch-indicator and to contacts of retrograde bias off relays K4-62 and K4-63. Power from CB12-7 energizes retrograde bias relay K12-5 when the JETT RETRO indicator is pressed.

BOOST-INSERT CONTROL 1 Circuit Breaker

The boost-insert control 1 circuit breaker CB3-1 provides BIA squib bus number 1 power to initiate the abort sequence with the abort control handle, jettison the nose fairing and scanner cover, separate the spacecraft from the launch vehicle, sense launch vehicle/spacecraft separation, extend the uhf and diplexer whip antennas, and initiate several experiments.

BOOST-INSERT CONTROL 2 Circuit Breaker

The boost-insert control 2 circuit breaker CB3-11 connects BIA SQUIB BUS number 2 power redundantly to the same switches to which CB3-1 connects power.

RETRO SEQ CNTL 1 Circuit Breaker

The retrograde sequence control 1 circuit breaker CB4-3 connects the retrograde squib bus number 1 to the SEP OAMS LINES switch-indicator, the SEP ADAPT switch-

indicator, the SEP ELEC switch-indicator, and the JETT RETRO switch-indicator on the main instrument panel. It also arms the abort discrete relays and the equipment adapter separation sensor switches and relays.

RETRO SEQ CNTL 2 Circuit Breaker

The retrograde sequence control 2 circuit breaker CB4-28 connects the retrograde squib bus number 2 redundantly to the same switches to which the retrograde sequence control 1 circuit breaker connects power and arms the same circuits.

SEQ LIGHTS TEST (AMBER-OFF-RED & GREEN) Switch

The sequence lights test switch connects main bus power to all amber-colored sequence lights and to all lights on the annunciator panel in the AMBER positions, and to all red or green sequence lights in the RED & GREEN position.

SEQ LIGHTS (BRIGHT-DIM) Switch

The sequence light bright-dim switch is a single-pole, double-throw toggle switch. It connects the main bus through a diode to all sequence light circuits in the BRIGHT position. It connects the bus through a resistor to the same circuits in the DIM position.

The fourth row on the left switch/circuit breaker panel contains eight switches. These switches arm or safety the various squib buses used by the Sequential System. Their functions are as follows.

BOOST-INSERT (ARM-SAFE) Switch

The boost-insert squib bus arm-safe switch is a four pole, double throw toggle switch. In the ARM position, this switch arms the BIA squib buses 1 and 2 and the BIA common control bus. These buses arm the SEP SPCFT switch-indicator, the BOOST CUTOFF 1 and 2 circuit breakers, the BOOST INSERT CNTL 1 and CNTL 2 circut

breakers, and the relay contacts which fire the nose fairing jettison, scanner cover jettison, OAMS activate, RCS activate, spacecraft separate, guillotine and pyrotechnics.

RETRO POWER (ARM-SAFE) Switch

The retrograde power squib bus arm-safe switch is a four-pole, double-throw switch. In the ARM position, it arms retrograde squib bus 1 and 2 and the retrograde common control bus. Thru these buses it arms the RETRO JETT ARM-SAFE switch, the RETRO ROCKET SQUIBS ARM-SAFE 1, 2, 3, and 4 switches, the ATT IND CNTL RETRO, RETRO SEQ. 1 and 2, and RETRO AUTO and MAN circuit breakers on the left switch/circuit breaker panel, and the RCS SQUIB 1 and 2 circuit breakers on the overhead switch/circuit breaker panel.

RETRO JETT (ARM-SAFE) Switch

The retrograde jettison squib bus arm-safe switch is a two-pole double-throw toggle switch. In the ARM position, it arms retrograde jettison squib buses number 1 and number 2. From these buses, the retrograde jettison relays get the power to fire the retrograde adapter shaped charges and retrograde pyrotechnic switch H.

RETRO ROCKET SQUIB 1, 2, 3, 4 (ARM-SAFE) Switches

The four retrograde rocket squib arm switches apply the voltages which ignite the four retrofire rockets to open contacts of the retro rocket automatic and manual fire relays. In the safe position of these four switches, the ignition voltage is removed from the relays. When both the RETRO POWER squib arm switch and the four RETRO ROCKET SQUIB arm switches are placed to the ARM position, the OAMS squib buses 1 and 2 are connected redundantly to the retrorocket fire relays.

BOOST-INSERT-ABORT CONTROLS AND INDICATORS

Seven indicators, three meters and four controls are provided for the boost-insert-abort phase of the spacecraft mission.

ENGINE I Indicators

The two ENGINE I indicators are provided on the command pilot's panel to indicate thrust chamber underpressure of the first stage booster engines. Each indicator illuminates red when the thrust chamber pressure of the engine is 68 percent of rated pressure or less. Both indicators illuminate red at stage 1 ignition but extinguished 0.91 to 1.25 seconds later as the pressure increases above 68 percent. Both indicators illuminate red at booster engine cut-off and extinguish quickly at staging.

ENGINE II Indicator

The ENGINE II indicator on the command pilot's panel illuminates amber to indicate the fuel injector underpressure (or off) condition of the second stage engine. The critical pressure for engine 2 is 55 percent of rated value. The indicator illuminates when the first stage engine is ignited and stays amber through first stage boost. Approximately one second after both ENGINE I indicators extinguish, the ENGINE II indicator also extinguishes, indicating normal staging and engine 2 fuel injector pressure build up.

ATT RATE Indicator

The attitude rate indicator on the command pilot's panel indicates an evaluation of the launch vehicle attitude rates during the boost phase. The indicator is extinguishes if the attitude rates remain within acceptable limits, but illuminates red if the rates exceed these limits.

SEC GUID Indicator

The secondary guidance indicator on the command pilot's panel indicates which guidance system is in operation. The indicator is extinguished to indicate that primary guidance is being used. The indicator illuminates amber to indicate that secondary guidance has been selected.

ABORT Indicators

Two ABORT indicators are provided, one for each pilot. Both indicators illuminate red when the abort command is transmitted. When the ABORT indicator is illuminated, immediate and appropriate action is imperative. The indicator signals the crew to initiate immediately the abort mode appropriate for the altitude and velocity of the spacecraft. These modes are described under Sequence System Operation. During the boost phase, the crew has been remined via the uhf communications link of the abort mode in effect.

STAGE 1 FUEL/OXIDIZER Meters

The stage 1 fuel and oxidizer meters on the command pilot's panel enable the crew to monitor the current status and progress of the boost phase, and to anticipate an abort condition if one should develop. These meters indicate the gas pressures in psia of the stage 1 fuel and oxidizer tanks. Dual indicating needles are provided for redundancy. The range of the stage 1 meters is 35 to 5 psia. A time-versus-pressure scale near the bottom of the meter shows the minimum required pressure at 20, 40, and 60 seconds after lift-off. Critical fuel tank pressure is indicated by a shaded column at the low end of the scle. After staging with no signals applied, the meters indicate maximum psia.

STAGE 2 FUEL/OXIDIZER Meters

The stage 2 fuel and oxidizer meters on the command pilot's panel indicate stage 2 fuel and oxidizer tank pressure over a 70 to 10 psia range. Redundant pointers are used. Critical fuel tank pressures are indicated by a shaded column at the low end of the scale. The S-flag at the 30-psia mark indicates the minimum acceptable stored pressure in the tank before pressurization. After spacecraft separation, the meters indicate maximum psia.

Longitudinal Accelerometer

The accelerometer on the command pilot's panel indicates the rate in g's at which the launch vehicle engines are changing the velocity of the spacecraft. The range of the accelerometer is minus 6g's to 16g's. The meter has positive and negative memory pointers. The accelerometer enables the crew to monitor the effectiveness of the engines. It is a secondary indicator of staging.

RGS-IGS Guidance Switch

The guidance switch above the abort control handle permits the command pilot to manually change from primary guidance to secondary backup guidance. When back-up guidance has been selected either manually or automatically during stage 1 boost, and the ground station determines that primary guidance is feasible during stage 2 boost, primary guidance can be selected again by momentarily placing the guidance switch to the RGS position.

D-Rings

A D-ring is provided on the ejection seat of each pilot. These rings are pulled to initiate mode I abort at altitude below 70,000 feet. Refer to Section III of this volume for the location and operation of these devices.

Abort Control Handle

The abort control handle is located on the command pilot's side of the cabin. It is used for spacecraft re-entry in abort modes I-II, II and III. These modes are effective above 25,000 feet. The three positions of this handle are NORMAL, SHUTDOWN, and ABORT. In NORMAL, the handle is inoperative. When the handle is moved to SHUTDOWN, the engine cutoff command is sent to the operating launch vehicle engine. When the abort handle is moved to ABORT, an immediate spacecraft separation and retrograde sequence is performed. These sequences differs from the normal sequences in that they are performed without cues from the indicators on the main instrument panel.

SEQUENCE CONTROLS AND INDICATORS

The switches, indicators, and switches-indicators on the main instrument panel center console have the following nomenclature, place in the mission sequence, and functions.

JETT FAIRING Pushbutton Switch

The jettison fairing switch is used at the end of second stage engine thrust decay, by the command pilot to jettison the nose fairing, and the horizon scanner head cover.

SEP SPCFT Switch-Indicator

The separate spacecraft switch-indicator is used in the separation-insertion phase of the sequence. The command pilot presses the switch-indicator approximately 20 seconds after second stage engine cutoff when the IVI displays the delta-V required for insertion. Pressing the switch-indicator causes several things to happen. Primarily, it detonates pyrotechnic devices which separate the spacecraft from the

launch vehicle. Secondarily, it extends the uhf and diplexer antennas and readies the acquisition aid beacon for use. As the spacecraft moves away from the launch vehicle, separation sensors close and energize the spacecraft separation relays. The relays illumminate the indicator green.

IND RETRO ATT Switch-Indicator

The indicate retrograde attitude switch-indicator is illuminated amber when the electronics timer energizes the T_R-256 second relay. The amber light cues the crew to press the switch-indicator at this time. When pressured, a bias voltage is placed on the pitch needle of the FDI, and the inertial platform is electrically placed in the BEF mode. When released the amber light is extinguished and a green light is illuminated.

BTRY PWR Indicator

The battery power indicator illuminated amber by the T_R-256 second relay. The amber light cues the pilot to place the MAIN BATTERIES switch to ON, and the fuel cell switch or ADAPTER BATTERIES switch to OFF. This change must be made because the adapter section will be jettisoned at retrograde. When all of the main battery switches are on, the indicator changes from amber to green.

RCS Switch-Indicator

The RCS indicator is illuminated amber by the T_R-256 second relay. The amber light cues the command pilot to activate the RCS by firing the fuel, oxidizer, and pressurant isolation squibs. Pressing the switch-indicator energizes relays which fire the squibs. The indicator changes from amber to green, indicating that the RCS has been activated.

SEP OAMS LINES Switch-Indicator

The separate OAMS lines indicator is illuminated amber by the TR-256 second relay is the prepare-to-go to retrograde phase. The amber light cues the crew to seal and sever the OAMS lines before jettisoning the adapter. Pressing the switch-indicator energizes relays which ignite the pyrotechnics used to seal and sever the lines. The relays also fire pyrotechnic switches and wire guillotines severing some of the adapter-retrograde mating line wiring. The indicator changes from amber to green.

SEP ELEC Switch-Indicator

The separate electrical indicator is also illuminated amber by the T_R-256 second relay. The amber light cues the crew to sever all the wiring at the retrograde/adapter mating line. Pressing the switch-indicator energizes the wire guillotine relay. The pyrotechnics are detonated and the wiring is cut. The indicator changes from amber to green to indicate that electrical separation has been accomplished.

SEP ADAPT Switch-Indicator

The separate adapter indicator is illuminated amber by the T_R-256 second relay. The amber light cues the crew to jettison the adapter equipment section. Pressing the switch-indicator causes the adapter shaped charge and the Z70 tubing cutter pyrotechnic to be detonated, and the adapter section severed. Separation of the adapter section is sensed by the equipment adapter separation sensors. Two closed sensors energize the sensor relay and change the indicator from amber to green.

ARM AUTO RETRO Switch-Indicator

The arm automatic retrofire indicator is illuminated amber by the T_R-30 second relay. The amber light cues the crew to arm the automatic retrofire circuits so

that when the electronic timer closes the TR contacts at T_R time, the retro-rockets will fire automatically. Pressing the switch-indicator completes the patch from the retrograde common control bus to the timer T_R contact, and also energizes the T_R arm relay. The relay changes the light from amber to green. Contact closure at T_R time energizes the T_R signal relay. The signal relay energizes the 45-second time delay relay, fires the retro rockets at 5.5 second intervals, and puts the platform in the free mode.

MAN FIRE RETRO Pushbutton Switch

The manual fire retrorockets switch connects the retrograde common control bus to the manual retrograde latch relay. Contacts of this relay energizes the 45-second time delay relay, fire the retrorockets at 5.5-second intervals, and place the platform in the free mode operation.

JETT RETRO Switch-Indicator

The jettison retrograde adapter indicator is illuminated amber by the 45-second time delay relay 45 seconds after retrofire begins. The amber light cues the crew to jettison the retrograde adapter. Pressing the indicator ignites pyro-technic switch H and other pyrotechnic devices which disconnect and guillotine the wires at the retrograde adapter section/re-entry vehicle mating line. It fires the shaped charges which sever the retrograde adapter section from the re-entry vehicle. It energizes the Horizon Sensor System scanner head jettison relays which fire the jettison squibs and jettison the scanner head. It removes the retrograde attitude signals applied to the flight director needles at T_R-256 seconds. It switches the FDI roll channel to the mix mode for re-entry. Finally by igniting pyrotechnic switch H it extinguishes the IND RETRO ATT, SEP OAMS LINE, SEP ELEC, SEP ADAPT and ARM AUTO RETRO green indicators and the JETT RETRO amber

indicator.

RE-ENTRY VEHICLE RELAY PANELS

Ten Sequence System relay panels are installed in Gemini Spacecraft 5, 6, and 8 through 12. Four relay panels are located in the re-entry vehicle, three in the retrograde section, one in the equipment section, and two in the rendezvous and recovery section. See Figure 4-1. The following Sequence System relay panels are in the re-entry module.

BIA Control Relay Panel

The boost-insert-abort control relay panel contains six relays to perform spacecraft separation indicator control and launch vehicle/spacecraft pyrotechnic switch firing.

Retrograde Separation Relay Panel

The necessary functions required for adapter retrograde section separation are performed by the fourteen relays of the retrograde separation relay panel. The relays perform such functions as pyrotechnic switch and shaped charge ignition, T_R-30 second indication, automatic IGS free mode selection, and arming of the contacts of the Time Reference System.

ACS Scanner and RCS Squib Fire Relay Panel

Re-entry Control System squib firing, scanner cover and scanner heads jettison, abort interlock RCS amber light actuation, and RCS ring B squib firing test prior to launch are provided by the sixteen relays of the attitude control system scanner and RCS squib fire relay panel.

Umbilical Pyrotechnic Switch Relay Panel

The umbilical pyrotechnic switch relay panel contains two relays which apply landing squib bus 1 and 2 power to re-entry umbilical wiring pyrotechnic switch.

ADAPTER RELAY PANELS

The retrograde section contains the following three relay panels which control spacecraft separation, retrofire, and equipment section separation. The equipment section contains the Orbit Attitude Maneuver System squib fire relay panel.

Spacecraft Separation Control Relay Panel

The spacecraft separation control relay panel contains six relays to perform the following functions: shaped charge ignition, and launch vehicle/spacecraft guillotine firing.

Retrograde Fire Relay Panel

The retrofire relay panel has twenty relays. These relays control the automatic, manual and salvo firing of the retro rockets, and time the 5.5-second firing sequence.

Retrograde Sequence Adapter Separate Relay Panel

The retrograde sequence adapter separate relay panel contains twelve relays. The relays are used for equipment adapter shaped charge ignition, propellant line guillotine, electrical wire guillotine, and retrograde abort.

Orbit Attitude and Maneuver System Squib Fire Relay Panel

The OAMS squib fire relay panel contains six relays for firing the OAMS squibs and controlling the regulator valves.

Rendezvous and Recovery SECTION RELAY PANELS

The Rendezvous and Recovery section contains two Sequence System relay panels: the nose fairing jettison relay panel, and the docking relay panel.

Nose Fairing Jettison Relay Panel

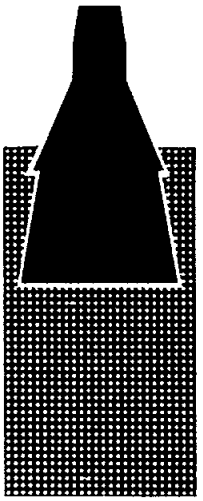
The nose fairing jettison relay panel contains two relays which control the jettisoning of the nose fiairng.

Docking Relay Panel

The docking relay panel has eleven relays which extend the docking index bar, illuminate the MSG ACPT light, effect emergency release of the docking latches, release and jettison the locking latches at retrograde, jettison the index bar, and cover the docking latch ports.

SEPARATION SENSORS

The Sequence System contains two sets of separation sensors. These are the launch vehicle/spacecraft separation sensors and the equipment adapter/re-entry vehicle separation sensors. Figure 4-1 shows their configuration and location. Figure 4-3 shows how the launch vehicle/spacecraft separation sensors operate. Figure 4-6 shows how the equipment section/re-entry vehicle separation sensors operate. Separation sensors are toggle switches which are normally open before separation is initiated. The separating structure will close the sensors as it moves away from the spacecraft re-entry module. The closure of any two of a set of three sensors is sufficient to sense and indicate separation.



ELECTRICAL POWER SYSTEM

Section V

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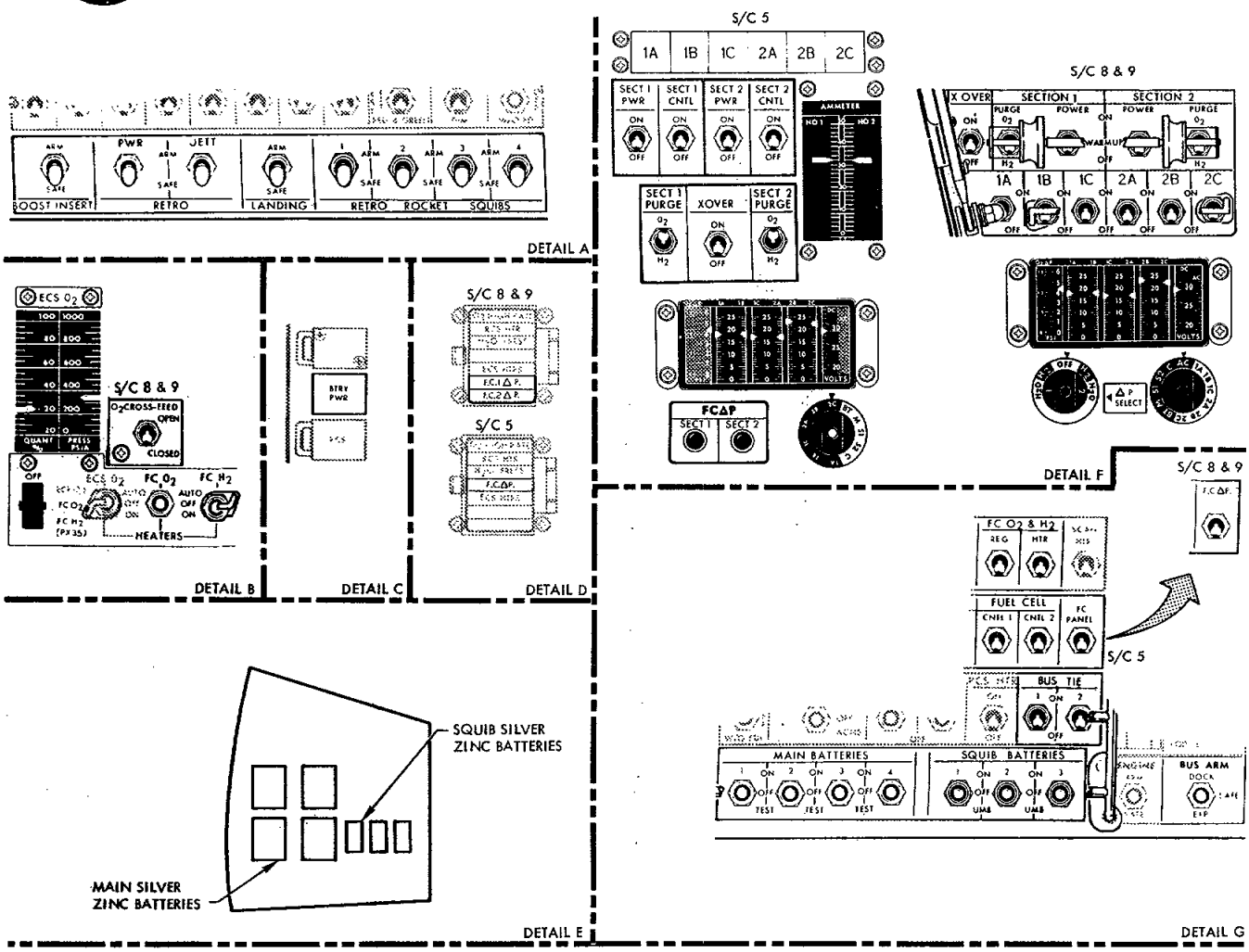


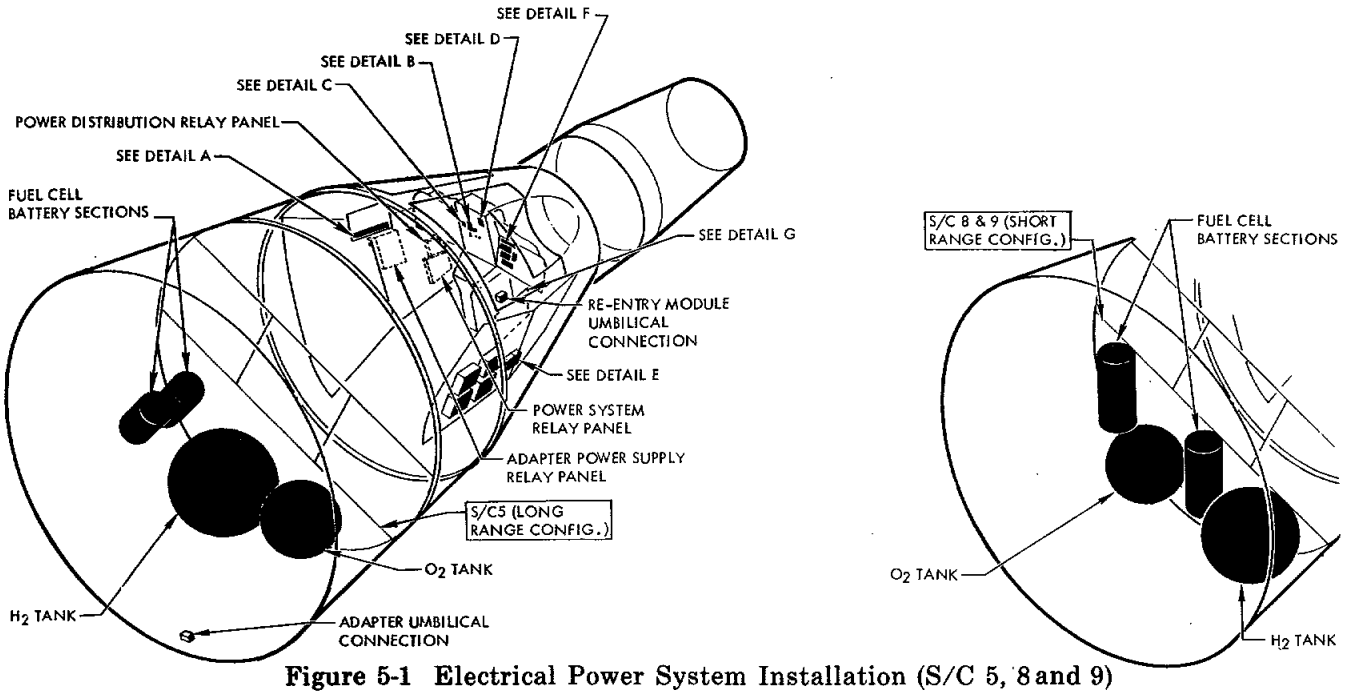


Figure 5-1 Electrical Power System Installation (S/C 5, 8 and 9)

SECTION V ELECTRICAL POWER SYSTEM

SYSTEM DESCRIPTION

The Electrical Power System for the Gemini Spacecraft basically consists of two fuel cell battery sections, four silver-zinc main batteries and three silver-zinc squib batteries (spacecraft 6 uses three 400 ampere/hour silver-zinc batteries in lieu of the fuel cell batteries). No primary ac electrical power system is provided for the spacecraft. Devices requiring ac power obtain this power from self-contained inverters within the individual systems. See Figure 5-1 for spacecraft 5, 8 and 9 configuration, Figure 5-2 for spacecraft 10, 11 and 12 configuration, and Figure 5-3 for spacecraft 6 configuration.

The Electrical Power System includes switches, circuit breakers, relay panels, ammeters, a voltmeter and telelights which provide control, distribution and monitoring for the system. Also included as an Electrical Power System subsystem is the Reactant Supply System (RSS) which provides storage and control of the reactants (hydrogen and oxygen) used for fuel cell battery operation (not applicable to spacecraft 6). Provisions are made for utilizing external power and remote monitoring of the spacecraft power buses during ground tests and pre-launch operations.

The two fuel cell battery sections and four main batteries provide dc power to the spacecraft main power bus (on spacecraft 6, the three adapter module batteries and the four main batteries provide dc power to the main bus). The squib batteries provide dc power to the common control bus and the two Orbital Attitude Maneuvering System (OAMS) squib buses. The OAMS squib buses in turn distribute dc power to the Boost-Insert-Abort (BIA), retrograde, landing and agena squib

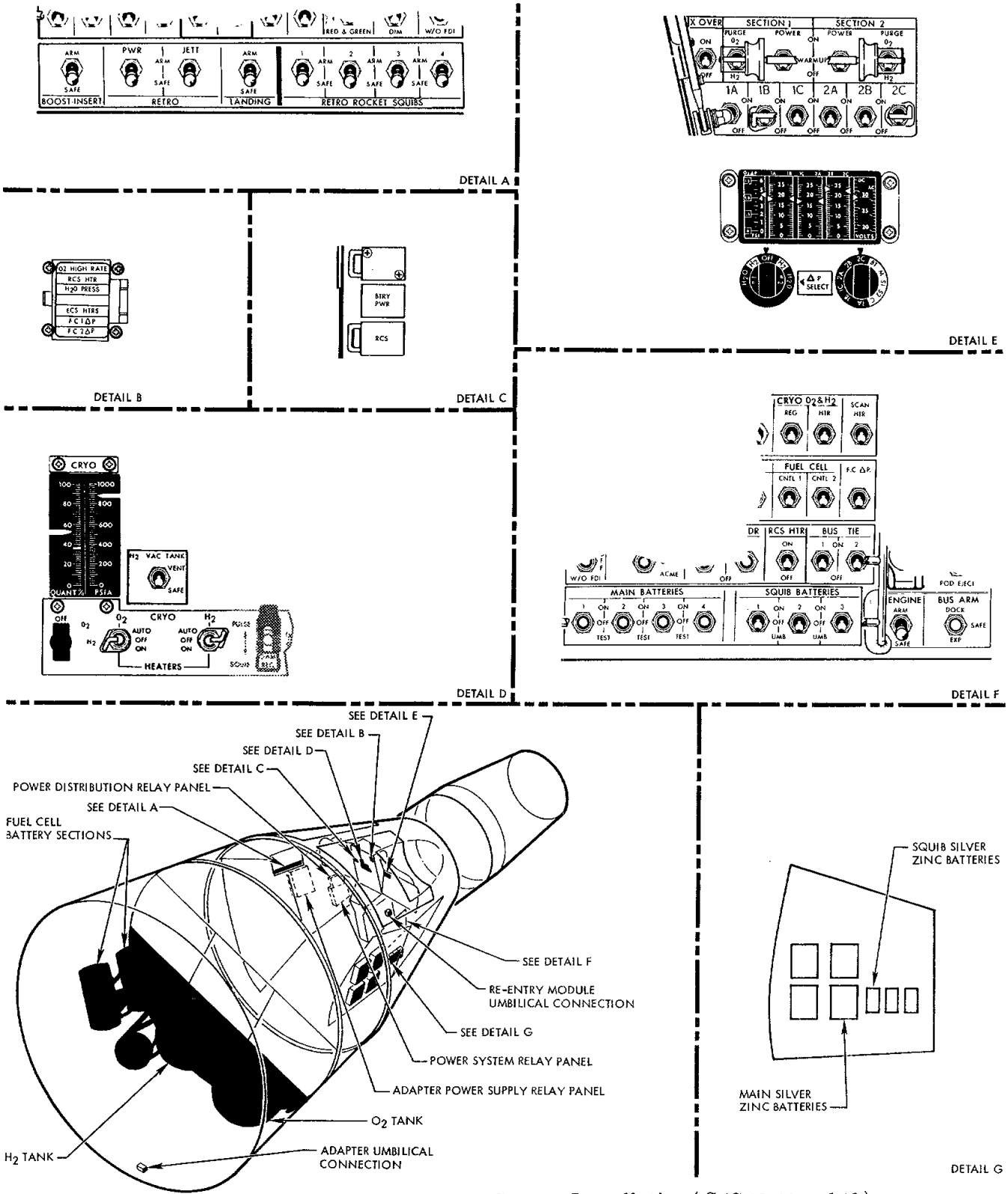
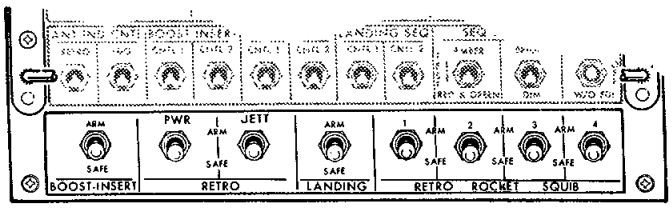
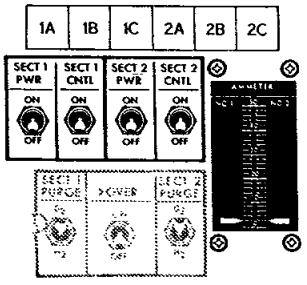
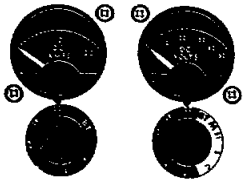
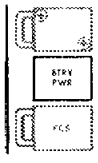


Figure 5-2 Electrical Power System Installation (S/C 10,11 and 12)









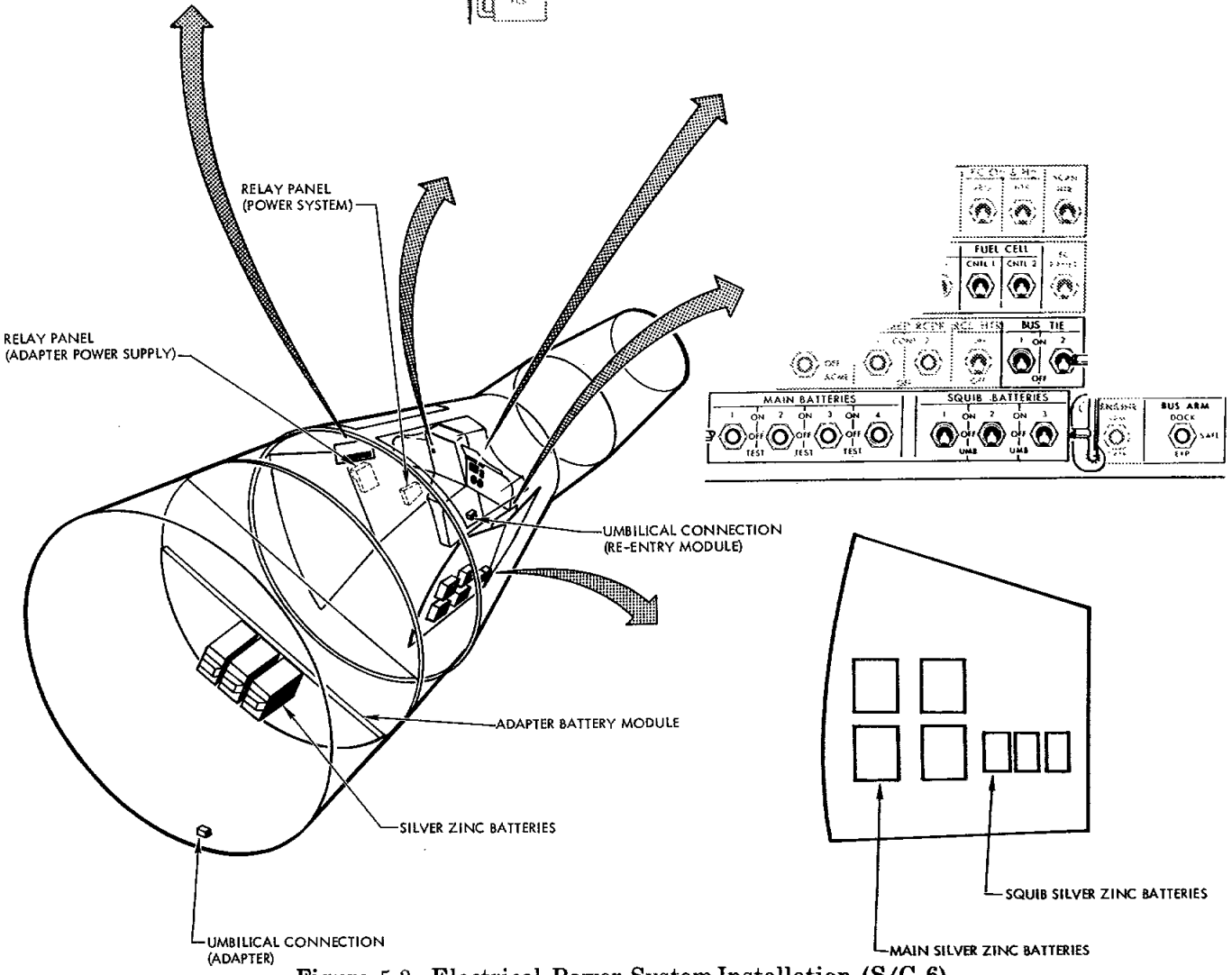


Figure 5-3 Electrical Power System Installation (S/C 6)

buses via the individual squib bus arming switches. See Figure 5-4 for spacecraft 5 and 8 thru 12 configuration and Figure 5-5 for spacecraft 6 configuration Electrical Power System schematics.

The fuel cell battery sections, along with the required RSS components, are installed in the RSS/fuel cell module (Figure 5-6, 5-7 and 5-8) which is located in the spacecraft equipment adapter section (on spacecraft 6, three silver-zinc batteries are installed in the adapter battery module (Figure 5-9) which is located in the spacecraft equipment adapter section.) The main and squib batteries are installed in the right cabin equipment bay.

The fuel cell SECTION 1 and SECTION 2 POWER (PWR and CNTL switches on spacecraft 5 and 6), PURGE, X-OVER and stack control switches (1A through 2C) are located on the right instrument panel. The fuel cell PWR and CNTL switches are used to control the battery module power on spacecraft 6. The PURGE and X-OVER switches are inoperative on spacecraft 6.

On spacecraft 5 and 6, a dual-vertical-readout ammeter is located on the right instrument panel. On spacecraft 5 and 8 through 12, a power system monitor consising of: a delta pressure indicator, three dual-vertical-readout ammeters and an ac/dc voltmeter, with associated selector switches, is located on the right instrument panel.

On spacecraft 5, two FCΔP indicator lamps are located on the right instrument panel. On spacecraft 6, a conventional voltmeter and ammeter with associated selector switches are located on the right instrument panel.

The MAIN BATTERIES switches, SQUIB BATTERIES switches, BUS TIE switches, agena

BUS ARM switch, FUEL CELL CONT, FCΔP (FC PANEL on spacecraft 5 and 6) and FC O_2 and H_2 (CRYO O_2 and H_2 on spacecraft 10, 11 and 12) regulator and heater circuit breakers are located on the right switch/circuit breaker panel. The FC O_2 and H_2 regulator and heater circuit breakers are inoperative on spacecraft 6. The squib bus arming switches are located on the left switch/circuit breaker panel.

The BTRY PWR sequence light, FCΔP telelights, O_2 and H_2 heater switches, O_2/H_2 quantity indicator (integral with Environmental Control system (ECS) O_2 indicator) and selector switch are located on the center instrument panel. The O_2 and H_2 heater switches, quantity indicator and selector switch are identified as CRYO switches and indicator on spacecraft 10, 11, and 12.

On spacecraft 8 and 9, an O_2 CROSS-FEED switch is also located on the center instrument panel. On spacecraft 10, 11 and 12, this switch is identified as H_2 TANK VAC. The power system relay panel, power distribution relay panel and adapter power supply relay panel are located in the left equipment area of the cabin.

SYSTEM OPERATION

PRE-LAUNCH

In order to conserve spacecraft battery power, external electrical power is utilized during the pre-launch phase of the mission. External power is supplied to the spacecraft common control, main and squib power buses through umbilical cables connected to the re-entry module and equipment adapter section umbilical receptacles. The external power source is provided by Aerospace Ground Equipment (AGE).

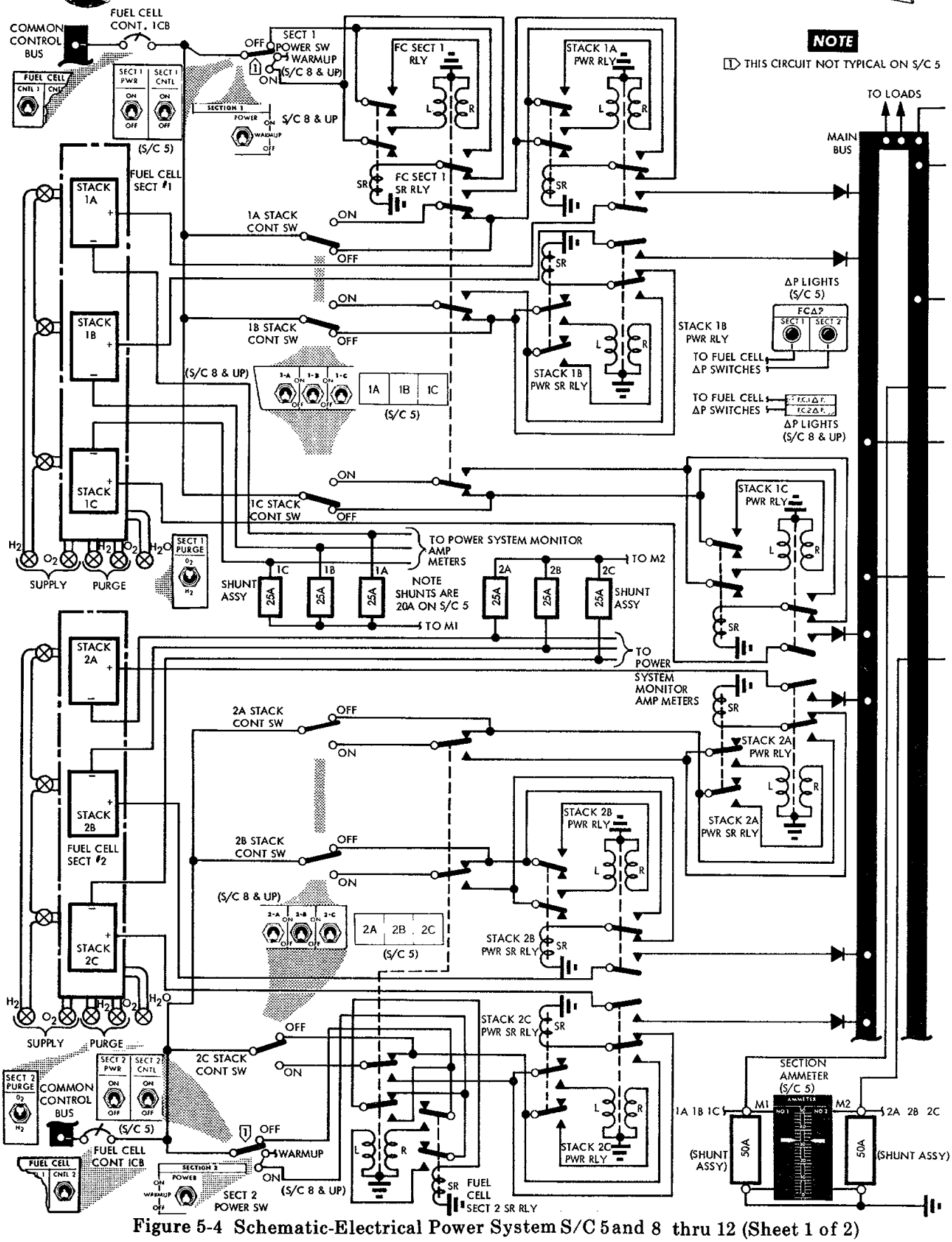


Figure 5-4 Schematic-Electrical Power System S/C 5 and 8 thru 12 (Sheet 1 of 2)

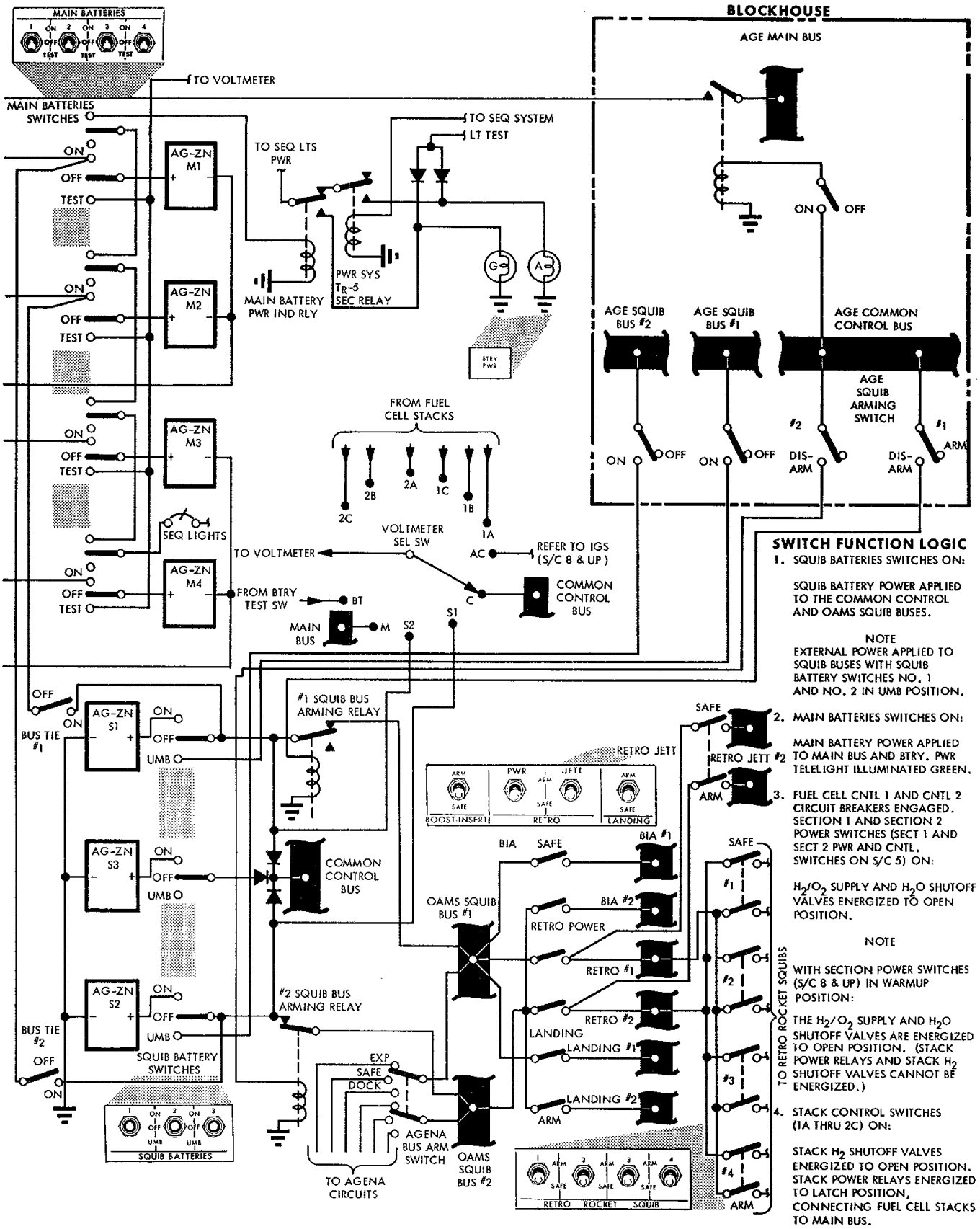


Figure 5-4 Schematic-Electrical Power System S/C 5 and 8 thru 12 (Sheet 2 of 2)

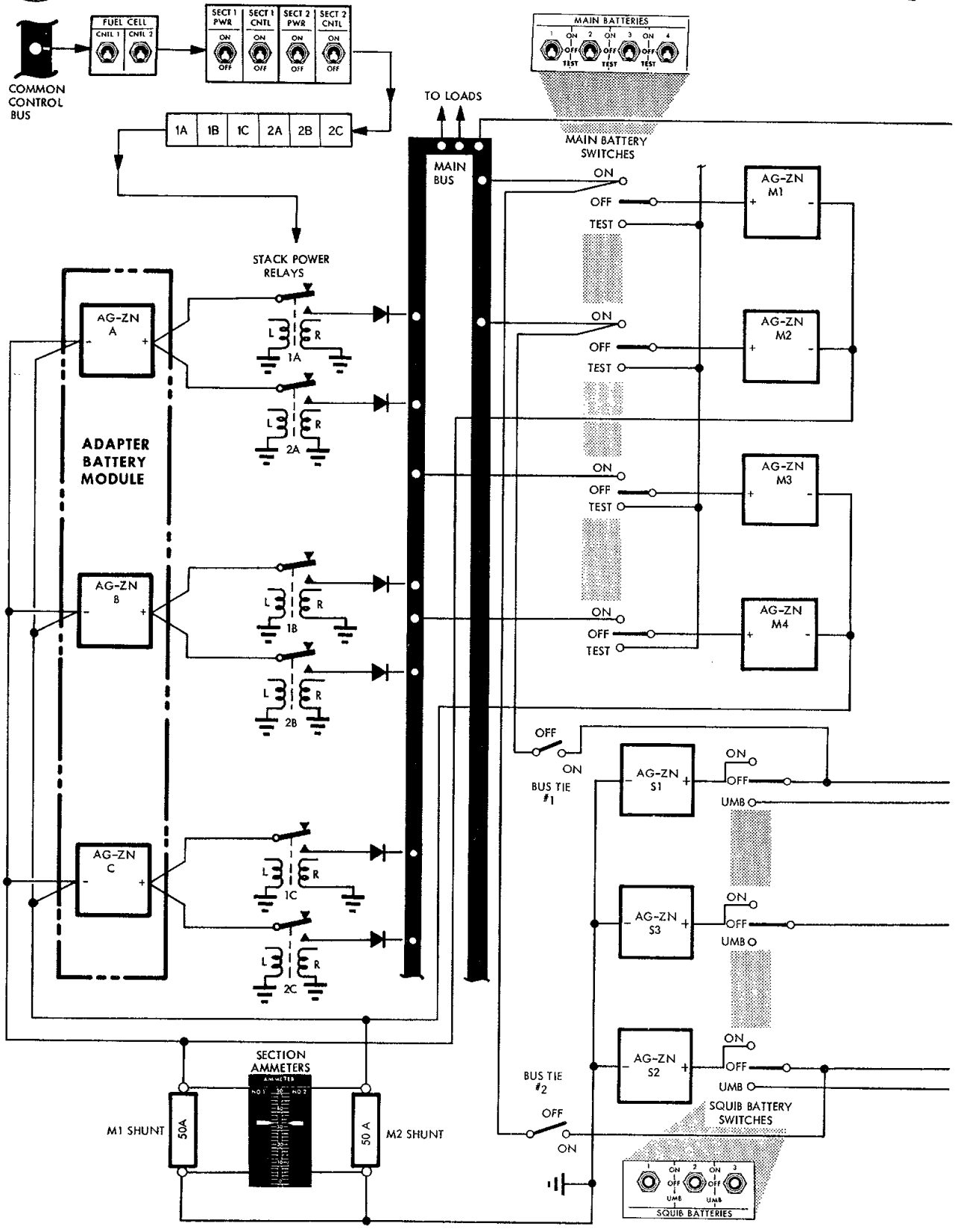
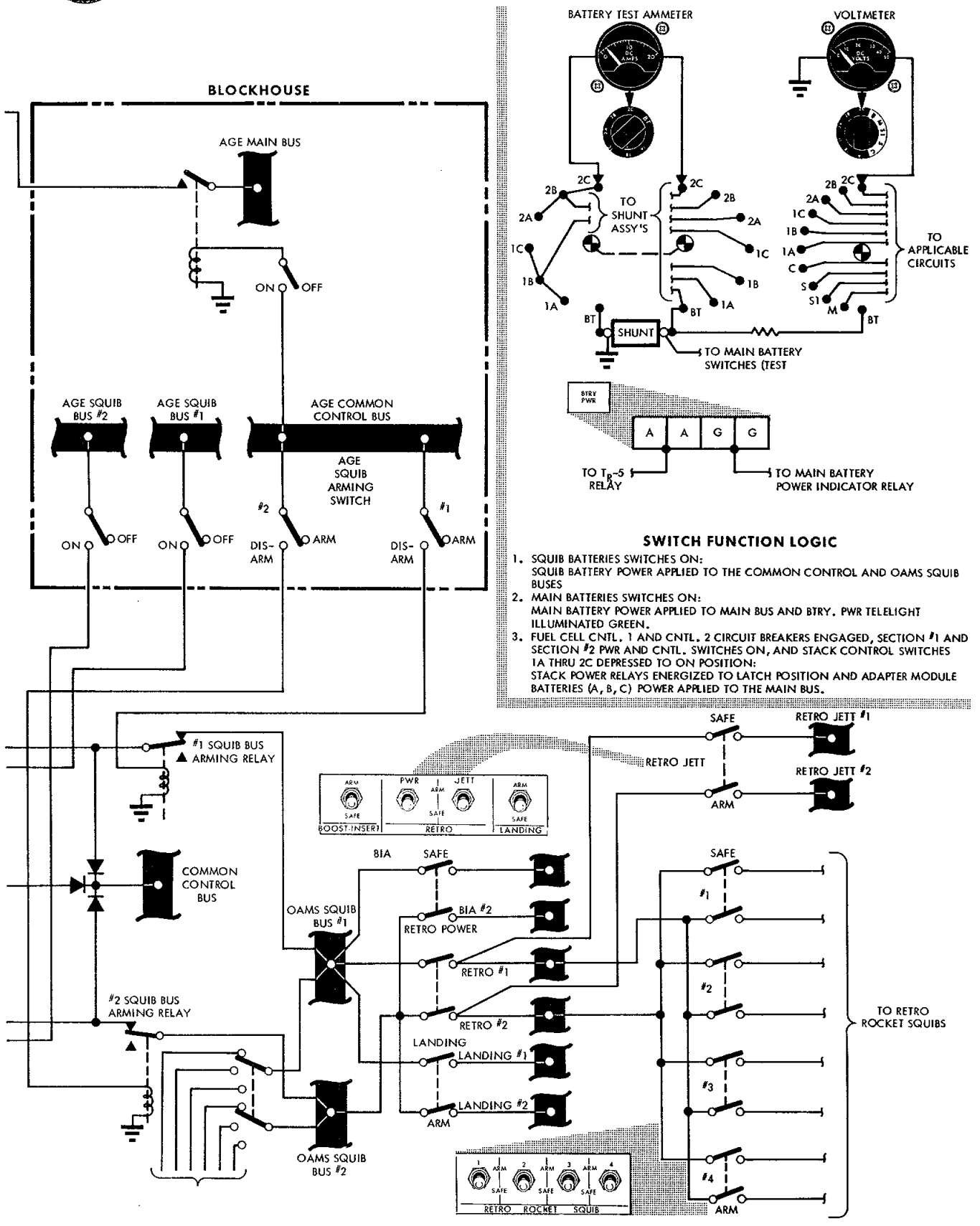


Figure 5-5 Schematic-Electrical Power System (Spacecraft 6) (Sheet 1 of 2)



SWITCH FUNCTION LOGIC

1. SQUIB BATTERIES SWITCHES ON:
   SQUIB BATTERY POWER APPLIED TO THE COMMON CONTROL AND OAMS SQUIB BUSES
2. MAIN BATTERIES SWITCHES ON:
   MAIN BATTERY POWER APPLIED TO MAIN BUS AND BTRY. PWR TELELIGHT ILLUMINATED GREEN.
3. FUEL CELL CNTL. 1 AND CNTL. 2 CIRCUIT BREAKERS ENGAGED, SECTION #1 AND SECTION #2 PWR AND CNTL. SWITCHES ON, AND STACK CONTROL SWITCHES 1A THRU 2C DEPRESSED TO ON POSITION:
   STACK POWER RELAYS ENERGIZED TO LATCH POSITION AND ADAPTER MODULE BATTERIES (A, B, C) POWER APPLIED TO THE MAIN BUS.

Figure 5-5 Schematic-Electrical Power System (Spacecraft 6) (Sheet 2 of 2)

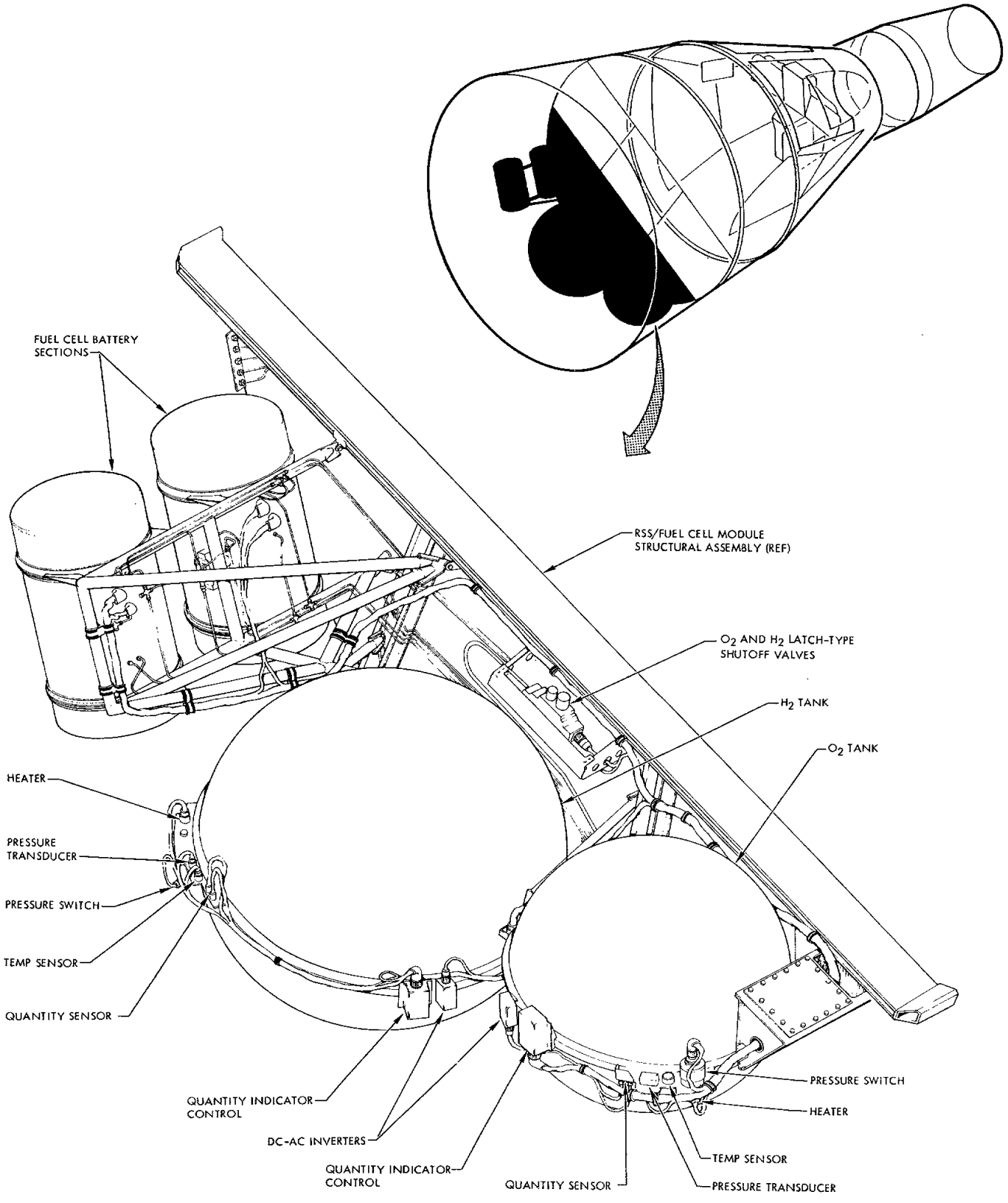


Figure 5-6 RSS/Fuel Cell Module (S/C 5)

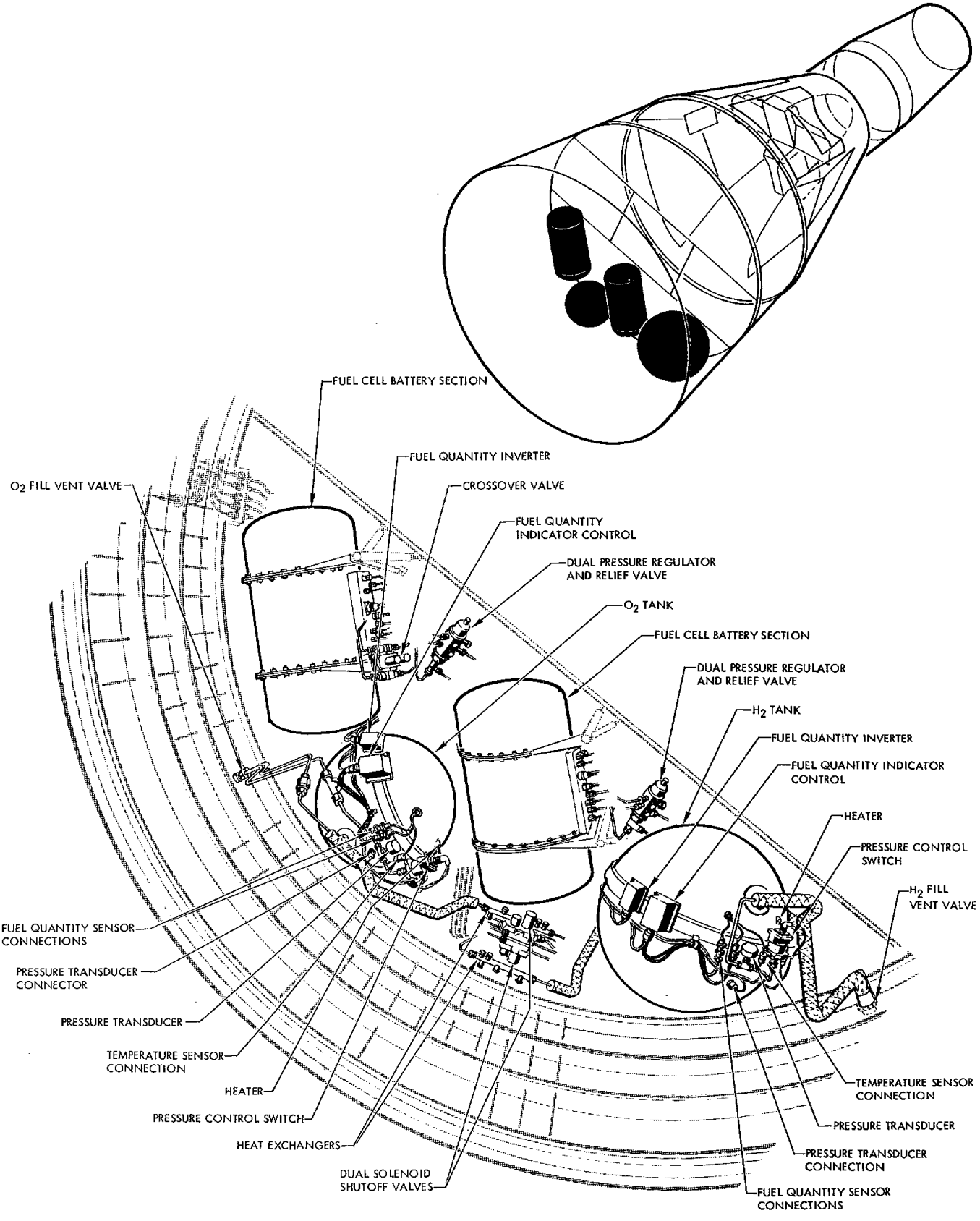


Figure 5-7 RSS/Fuel Cell Module (S/C 8 & 9)

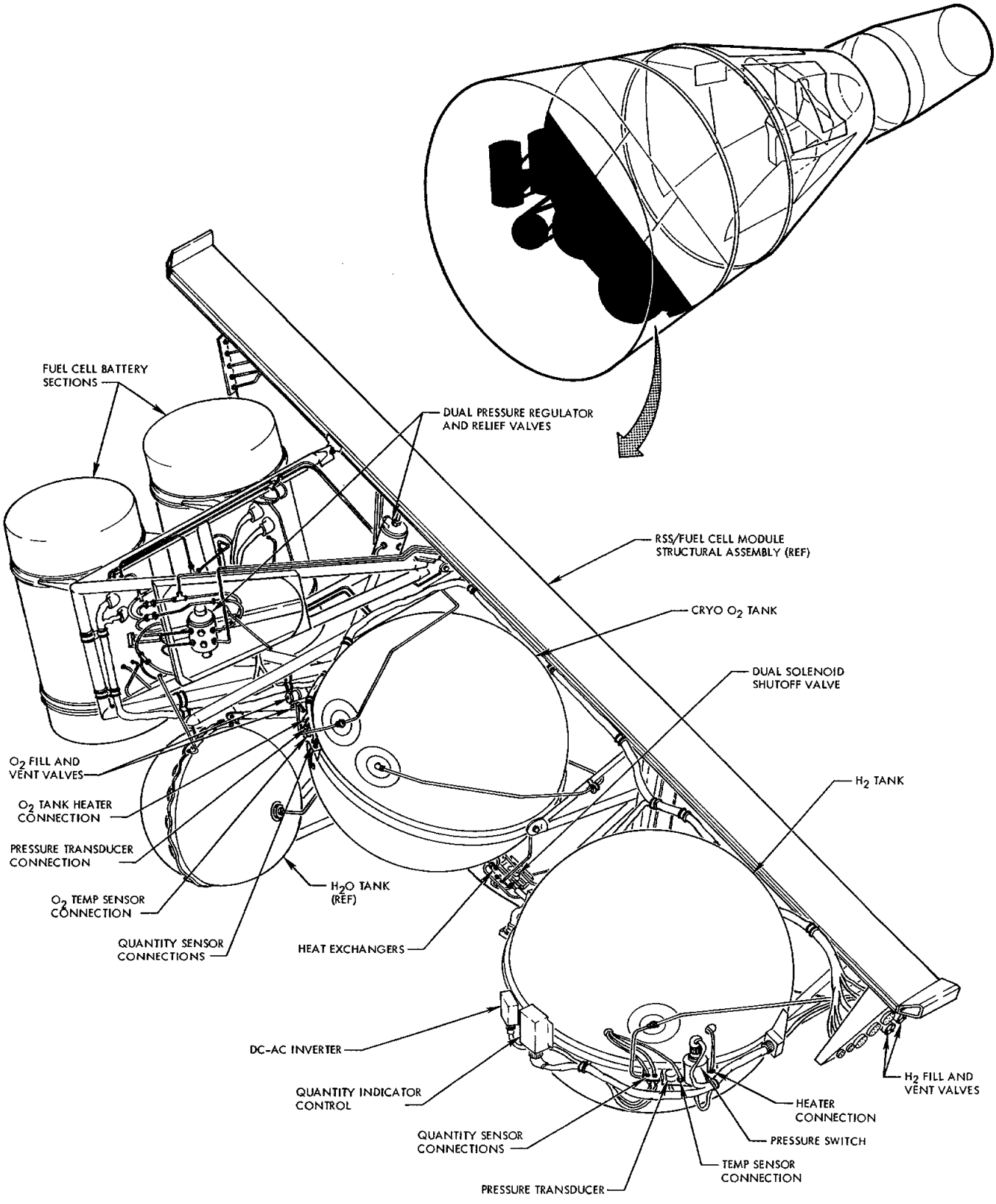


Figure 5-8 RSS/Fuel Cell Module (S/C 10, 11 & 12)

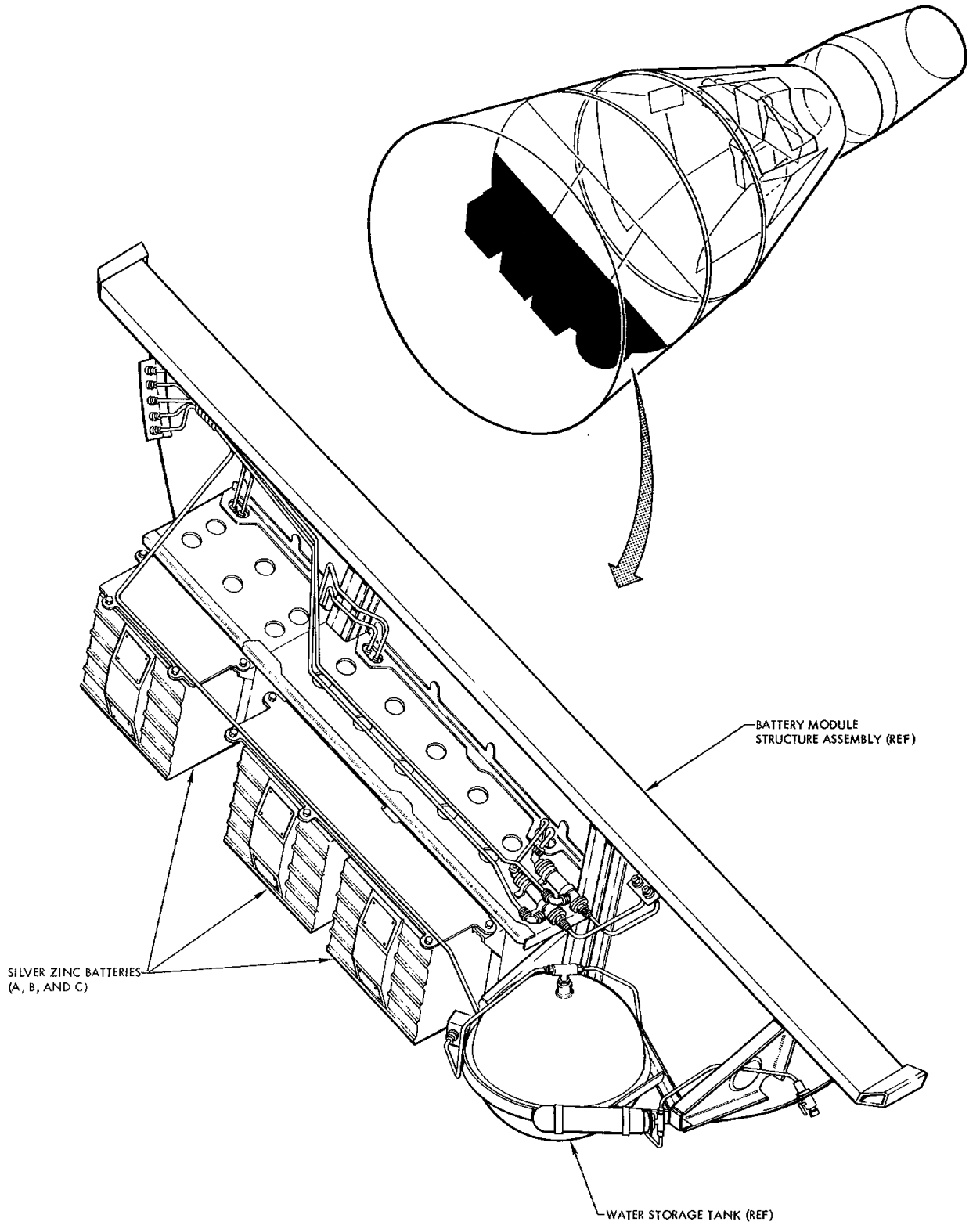


Figure 5-9 Adapter Battery Module (S/C 6)

SQUIB BATTERIES switches 1 and 2 must be placed in the umbilical (UMB) position in order to apply external power to the spacecraft squib buses. Remote control of the spacecraft squib bus arming relays, and remote monitoring of the spacecraft power buses is also accomplished through the re-entry and adapter umbilical cables.

Prior to launch, all MAIN BATTERIES and SQUIB BATTERIES switches, SECTION POWER switches (SECT PWR and CNTL switches on spacecraft 5 and 6) and stack control switches (1A through 2C) are set to the ON position to insure maximum redundancy of the Electrical Power System during the launch phase of the mission. On spacecraft 5 and 8 through 12, the fuel cell batteries are activated in sufficient time prior to launch, to insure launch readiness of the fuel cell batteries and RSS.

The common control bus and OAMS squib buses are switched from external power to the squib batteries in sufficient time prior to launch, to verify the squib battery circuits. The BIA buses are armed prior to launch by setting the BOOST-INSERT ARM/SAFE switch to ARM position.

The re-entry and equipment adapter section umbilicals are disconnected from the spacecraft just prior to lift-off. Normally, umbilical separation is accomplished by an electrical solenoid device. A backup method of separation is also provided by a lanyard initiated mechanism which is actuated by movement of the launch vehicle.

ORBIT

From launch time until booster separation and insertion into orbit, both the fuel

cell battery sections (module batteries on spacecraft 6) and the four main batteries are connected in parallel to the main power bus. After booster separation is accomplished, the MAIN BATTERIES switches are placed in the OFF position to conserve the main battery power. Also the pilots will disarm the BIA squib buses by setting the BOOST-INSERT ARM/SAFE switch to SAFE.

In the event of an abort, all squib buses required for the abort function, which are not armed prior to launch, are armed via the abort relays controlled by the Sequential System. These relays effectively bypass the applicable squib bus arming switches which normally arm these buses.

The SQUIB BATTERIES switches remain in the ON position throughout the entire mission until landing is accomplished. All three squib batteries are connected to the common control bus through diodes for individual fault protection. Squib batteries 1 and 2 are connected to the two OAMS squib buses via the de-energized squib bus arming relays.

The O_2 CROSS-FEED switch (spacecraft 8 and 9) remains in the CLOSED position except in the event of a loss of RSS O_2 tank pressure. This switch controls the O_2 crossfeed valve, which provides the capability of connecting the ECS O_2 supply to the fuel cell battery sections. (Refer to Figure 5-14).

The H_2 TANK VAC switch (spacecraft 10, 11 and 12) provides the capability of venting the area between the inner and outer wall of the RSS H_2 supply tank to outside vacuum in space. This switch, when in VENT position, initiates a pyrotechnic cutter device to perform this function. The H_2 TANK VAC switch remains in the SAFE position until the pilots elect to perform this function.

The BUS TIE switches remain in the OFF position unless the necessity arises where the pilots must use main bus power to fire the squibs. The BUS TIE switches provide a method of connecting the main bus to the OAMS squib and common control buses. The agena BUS ARM switch will be used according to the mission requirements.

On spacecraft 5 and 8 through 12, a small percentage of the reactant gases must be purged from the fuel cell batteries periodically to insure that the impurities contained in the feed gases do not restrict reactant flow to the cells and to remove any accumulation of product water in the gas lines. This purging function is performed by the pilots manually actuating the O_2 and H_2 PURGE switches. The pilots may increase the flow of gases to the fuel cell sections for more effective purging by setting the X-OVER switch to ON position.

RE-ENTRY

At 256 seconds before retrograde time (T_R-256) the pilots will arm the retrograde squib buses by setting the RETRO PWR ARM/SAFE switch and the individual RETRO ROCKET SQUIBS No. 1, 2, 3 or 4 ARM/SAFE switches to ARM position. The retrograde rockets are used according to the mission requirements.

The MAIN BATTERIES switches must be returned to the ON position at T_R-256 seconds to insure continuity of main bus power at the time of separation of the equipment adapter section, containing the RSS/fuel cell module (battery module on spacecraft 6), from the spacecraft. There is no automatic switching provided for this function.

The stack control switches (1A through 2C) and the SECTION 1 and SECTION 2 POWER switches (SECT 1 and SECT 2 PWR and CNTL switches on spacecraft 5 and 6) are

set to OFF position after the main batteries are properly connected to the main bus.

After retrograde rocket firing has been accomplished, the pilots will set the RETRO JETT ARM/SAFE switch to ARM. This switch provides a method of arming the JETT RETRO switch (center instrument panel), and is effectively an interlock to prevent inadvertent jettisoning of the retrograde section prior to firing of the retrograde rockets.

After the equipment adapter and retrograde sections are separated from the spacecraft, the pilots will disarm the retrograde squib buses by setting the RETRO PWR ARM/SAFE switch, RETRO JETT ARM/SAFE switch and RETRO ROCKET SQUIBS ARM/SAFE switches to SAFE position. At this time, the landing squib buses are armed by setting the LANDING ARM/SAFE switch to ARM position.

After landing is accomplished, the pilots will disarm the landing squib buses by returning the LANDING ARM/SAFE switch to SAFE. At this time, power will be removed from the OAMS squib buses by setting SQUIB BATTERIES switches 1 and 2 to OFF position. All unnecessary electrical equipment will be deactivated to conserve the remaining spacecraft batteries for recovery equipment operation. SQUIB BATTERIES switch 1 and the MAIN BATTERIES switches will remain in the ON position to power the main and control buses throughout the recovery phase of the mission.

MONITOR AND DISPLAY

Throughout the mission, visual displays of bus voltage and current are provided by the system voltmeter and ammeters. On spacecraft 5 and 8 through 12, a

power system monitor, which consists of a delta pressure indicator, three dual ammeters and an ac/dc voltmeter is utilized.

The ammeters monitor individual fuel cell stack current (1A through 2C). The voltmeter, used in conjunction with a selector switch, displays individual fuel cell stack voltage, common control bus voltage, OAMS squib bus 1 and 2 voltage, Inertial Guidance System (IGS) inverter output voltage (spacecraft 8 through 12 only), main bus voltage and individual main battery voltage with the selector switch in Battery Test (BT) position and a particular MAIN BATTERIES switch in TEST position.

The delta pressure indicator, in conjunction with a selector switch, provides a visual display of O_2 versus H_2 and O_2 versus H_2O differential pressure in the fuel cell battery sections. In the event that the differential pressure exceeds the prescribed limits, the pilots must evaluate the fuel cell battery performance, and if a malfunction exists, shut down the malfunctioning fuel cell battery section by setting the applicable fuel cell power and stack control switches to OFF position. The delta pressure indicator is inoperative on spacecraft 5.

An out of tolerance delta pressure indication is also provided by the fuel cell delta pressure (FCΔP) telelights on the center instrument panel. The lights are illuminated red when a malfunction exists. On spacecraft 5 only, two FCΔP indicator lamps on the right instrument panel are illuminated red when a malfunction exists.

The reactant (O_2 and H_2) supply quantities are displayed on the ECS O_2 quantity indicator (center instrument panel) when the associated selector switch is set to FC O_2 or FC H_2 positions. On spacecraft 10, 11 and 12, this indicator

and switch are identified as CRYO O_2 and H_2.

The BTRY PWR sequence light, located on the center instrument panel, is illuminated amber at T_R-256 seconds during the mission by action of the T_R-5 relay in the power system relay panel. This informs the pilots that they must return the MAIN BATTERIES switches to ON position to insure continuity of main bus power due to the impending separation of the equipment adapter section containing the adapter power supply (fuel cell battery sections on spacecraft 5 and 8 through 12 and silver-zinc batteries on spacecraft 6). With all main batteries properly connected to the main bus, the BTRY PWR sequence light is illuminated green.

On spacecraft 5 and 6 the dual-vertical-readout section ammeter provides a display of section 1 and 2 main bus current. Section 1 includes 50 percent of the adapter power supply current (fuel cell batteries or silver-zinc batteries as applicable) plus main batteries 1 and 2 current. Section 2 includes 50 percent of the adapter power supply current plus main batteries 3 and 4 current.

The stack ammeter (used for battery test ammeter on spacecraft 6) with selector switch in 1A, 1B, 1C or 2A, 2B, 2C positions, displays 50 percent of adapter module battery current. With the selector switch in BT position, the ammeter displays individual main battery test current as the appropriate MAIN BATTERIES switch is set to TEST position.

Displays of common control bus voltage, main bus voltage, OAMS squib bus voltage, and adapter module battery voltage is provided by the system voltmeter and selector switch. Individual main battery voltage (with the particular battery removed from the main bus) is monitored with the voltmeter selector switch in BT position and the applicable MAIN BATTERIES switch in TEST position.

The FCAP telelights and reactant quantity indications are not operative on spacecraft 6.

SYSTEMS UNITS

SILVER-ZINC BATTERIES

The four main batteries are 45 ampere/hour, 16 cell, silver-zinc batteries. The three squib batteries are 15 ampere/hour, 16 cell, silver-zinc batteries. The squib batteries are special high-discharge-rate batteries which will maintain a terminal voltage of 18 volts for one second under a 75 ampere load.

On spacecraft 6, three 400 ampere/hour, 16 cell, silver-zinc batteries are installed in the adapter battery module. These batteries are used in lieu of fuel cell batteries. All of the silver-zinc batteries have an open circuit terminal voltage of 28.8 to 29.9 volts DC.

The main and squib battery cases are made of titanium. The approximate activated (wet) weight for each squib battery is 8 lbs and each main battery 17 lbs. The adapter module battery cases (spacecraft 6) are constructed of magnesium and the approximate wet weight of each battery is 118 lbs.

The battery electrolyte consists of a 40 percent solution of reagent grade potassium hydroxide and distilled water. The main and squib batteries have a vent valve in each cell designed to prevent electrolyte loss and will vent the cell to atmospheric pressure in the event a pressure in excess of 40 psig builds up within the cell.

All of the silver-zinc batteries are equipped with relief valves which maintain a tolerable interior to exterior differential pressure in the battery cases. The batteries are capable of operating in any attitude in a weightless state. Prior

to installation into the spacecraft, the batteries are activated and sealed at sea level pressure. All of the batteries are coldplate mounted to control battery temperature.

POWER SYSTEM RELAY PANEL

The power system relay panel contains relays necessary for controlling and sequencing power system functions. The panel contains the control relays for the fuel cell battery system and RSS, main battery power sequence light relay, T_R-5 relay and the squib bus arming relays.

POWER DISTRIBUTION RELAY PANEL

The power distribution relay panel contains the relays required for arming the retrograde squib buses in the event of an abort. These relays are controlled by the Sequential System.

ADAPTER POWER SUPPLY RELAY PANEL

The adapter power supply relay panel contains relays necessary for controlling adapter module power to the main power bus. The relay panel contains the stack power relays which connect the individual fuel cell stacks to the main bus. (On spacecraft 6 the stack power relays connect the adapter module batteries to the main bus.) The panel also contains diodes used for reverse current protection between the adapter power supply and the spacecraft main power bus.

AMMETERS

The main bus section ammeter (spacecraft 5 and 6) is a dual-edge-readout vertical reading meter having a 0-50 ampere range with a total accuracy of two percent. The NO. 1 scale displays main batteries 1 and 2 and 50 percent of the adapter power supply current. The NO. 2 scale displays main batteries 3 and 4 and 50

percent of the adapter power supply current. The ammeter is shunt connected between the main power bus and spacecraft ground.

The fuel cell stack ammeter (used as a battery ammeter on spacecraft 6) with associated selector switch, provides a display of individual main battery test current with the selector in BT position and a particular MAIN BATTERIES switch in TEST position. With the selector switch in 1A, 1B, 1C or 2A, 2B, 2C positions, the ammeter displays 50 percent of the applicable adapter module battery current. The meter has a 0-20 ampere scale.

VOLTMETER

On spacecraft 6 the voltmeter, used in conjunction with a selector switch, displays main bus, common control bus and squib bus voltage. Individual main battery voltage may be monitored with the voltmeter selector switch set to BT position and a particular MAIN BATTERIES switch set to TEST position. The voltmeter displays applicable adapter module batteries (A, B and C) voltage when the selector switch is set to 1A, 1B, 1C or 2A, 2B, 2C positions. The voltmeter has a 0-50 vdc range.

POWER SYSTEM MONITOR

The power system monitor (not applicable on spacecraft 6) consists of five vertical reading indicators; a delta pressure indicator, three dual-readout ammeters and an ac/dc voltmeter. The delta pressure indicator (not operative on spacecraft 5) and voltmeter are used in conjunction with selector switches located just below on the instrument panel.

The ammeters provide a display of individual fuel cell stack (1A through 2C) current (reading must be multiplied by 0.8 on spacecraft 5). The voltmeter, with the selector switch in appropriate position, displays individual fuel cell stack

voltage, main bus, squib bus and common control bus voltage, individual main battery voltage (with a particular MAIN BATTERIES switch in TEST position) and IGS inverter output voltage with the selector switch in AC position. The AC position on the selector switch is inoperative on spacecraft 5. The voltmeter has a 20-33 ac volt range and an 18-33 dc volt range.

The delta pressure indicator has a 0-1.5 psi range with the selector switch in either H_2 position and a 0-6 psi range with the selector switch in either H_2O position. This indicator displays O_2 versus H_2 differential pressure and O_2 versus H_2O differential pressure for each fuel cell battery section.

FUEL CELL BATTERIES

Construction

The fuel cell batteries used in the Gemini Spacecraft are of the solid ion-exchange membrane type using hydrogen (H_2) for fuel and oxygen (O_2) for an oxidizer. The fuel cell battery system is comprised of two separate sections which are sealed in air tight pressure containers. Each section is made up of three interconnected fuel cell stacks with plumbing for transferring hydrogen, oxygen and product water. (See Figure 5-10).

Each fuel cell stack consists of 32 individual fuel cells. Each basic fuel cell is made up of two catalytic electrodes separated by a solid type electrolyte in laminated form. (See Figure 5-11).

The electrolyte is composed of a sulfonated styrene polymer (plastic) approximately 0.10 inches thick. Thin films of platinum catalyst, applied to both sides of the electrolyte, act as electrodes and support ionization of hydrogen on the anode side of the cell and oxidation on the cathode side of the cell.

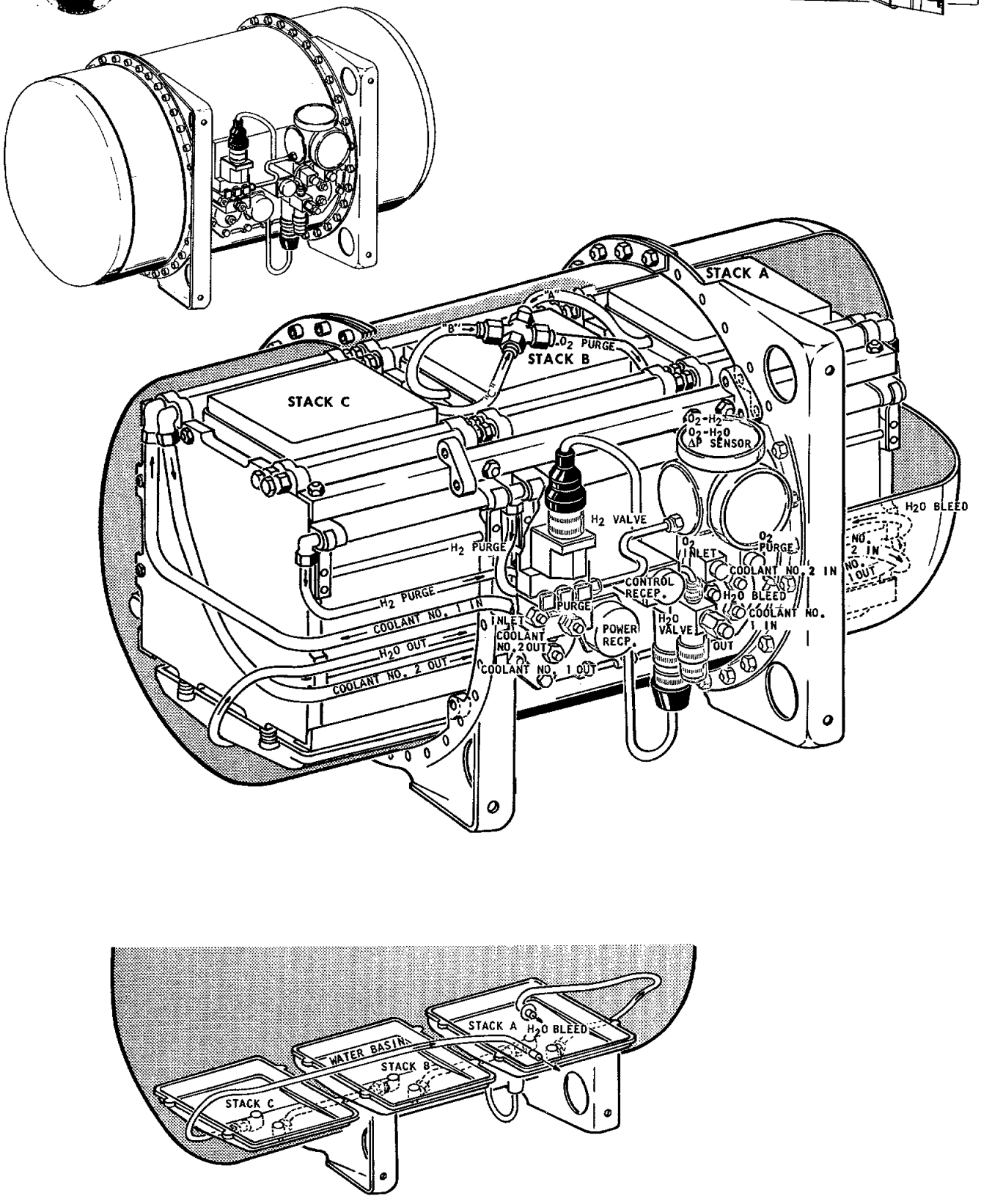


Figure 5-10 Fuel Cell Battery Section

A thin titanium screen, imbedded into the platinum catalytic electrode, reduces the internal resistance along the current flow path from the electrode to the current collector and adds strength to the solid electrolyte.

On the hydrogen side of the fuel cell, a current collector is attached by means of a glass-cloth-reinforced epoxy frame which assures a tight seal around the edges of the cell, forming a closed chamber. Ribs in the collector are in contact with the catalytic electrode on the fuel cell, providing a path for current flow.

The hydrogen fuel is admitted through an inlet tube in the frame of the current collector and enters each gas channel between the collector ribs by way of a series of slots in the tube. Another tube provides a purge outlet, making it possible to flush accumulated inert gases from the cell. The collector plate is made of approximately 0.003 inch thick titanium.

On the oxygen side of the cell, a current collector of the same configuration and material as the hydrogen side collector is attached. Its ribs, located at right angles to those of the other collector, provide structural support to the electrolyte-electrode structure.

A Dacron cloth wick, attached between the ribs, carries away the product water through capillary action, by way of a termination bar on one side of the assembly. Oxygen is admitted freely to this side of the fuel cell from the oxygen filled area of the section container.

The cell cooling system consists of two separate tubes bonded in the cavity formed by the construction of the oxygen side current collector and the back side of the hydrogen current collector. Each tube passes through six of the collector ribs and has the cooling capacity to maintain operating temperature. The cooling of

the oxygen current collector, which holds the product water transport wicks provides the coldplate for water condensation from the warmer oxygen electrode.

The individual fuel cell assemblies are arranged in series to form a stack as shown in Figure 5-12. When assembling the cells into a stack, the ribs of the oxygen side current collector contact the solid electrolyte of the fuel cell assembly. Titanium terminal plates are installed on the ends of the two outside cells to which connections are made for the external circuit. End plates, which are honey-comb structures of epoxy-glass laminate 0.5 inch thick, are installed on the outside of the terminal plates.

Stainless steel insulated tie rods hold the stack together and maintain a compression load across the area of each cell assembly. This assures proper contact of the solid electrolyte with the ribs of each current collector. The fuel cell stacks are packaged in a pressure tight container, together with the necessary reactant and coolant ducts and manifolds, water separator for each stack, and required electrical power and instrumentation wiring.

The hydrogen inlet line, hydrogen purge line, and the two coolant lines for each cell lead from their respective common manifolds running the length of the stack. The manifolds are made of an insulating plastic material and the individual cell connections are potted in place after assembly to provide a leak-tight seal. The oxygen sides of the cells are open to the oxygen environment surrounding the fuel cell assemblies within the container.

An accessory pad is mounted on the outside of the fuel cell section container. It includes the gas inlet and outlet fittings, purge and shutoff valves, water valve and electrical connectors. Structurally, the container is a titanium pressure

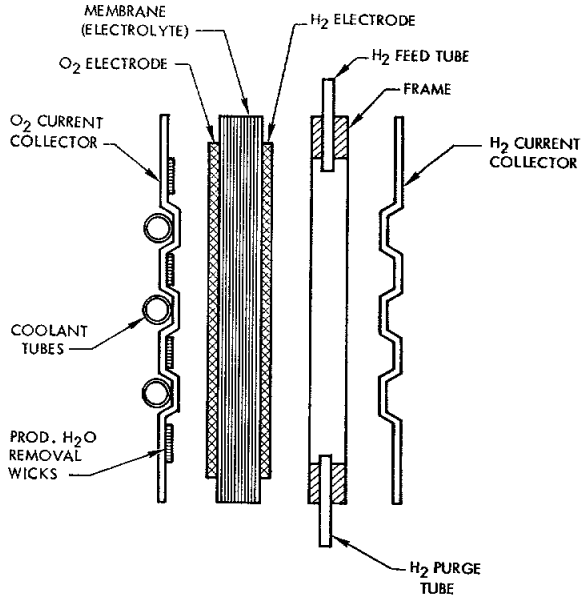


Figure 5-11 Basic Fuel Cell Assembly

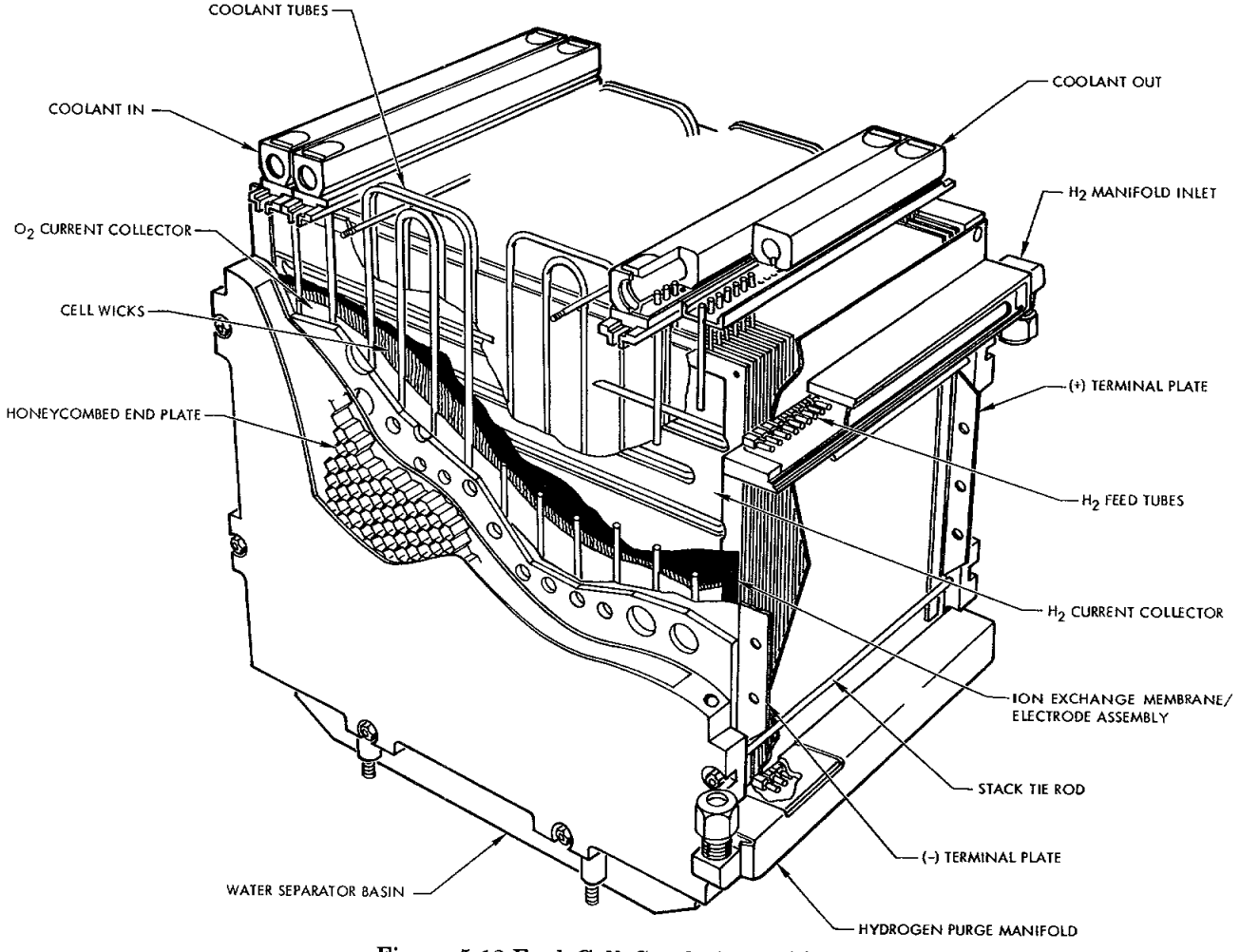


Figure 5-12 Fuel Cell Stack Assembly

vessel consisting of a central cylinder with two end covers and two mounting brackets. Within the container, the fuel cell stacks are mounted on fiberglass-impregnated epoxy rails by bolts which pass through the stack plates. These rails are in turn bolted to the mounting rings sandwiched between the two flanges on the section container.

The hydrogen manifolds on each stack within a section are parallel fed with a hydrogen shutoff valve and check valve in the feed line to each stack. Oxygen is fed into the section container so that the entire free volume of the container contains oxygen at approximately 22.5 psia. The coolant reaches the fuel cell battery sections by two separate isolated lines. Any malfunction in the coolant line in one section will not affect the cooling function of the coolant line in the other section.

Each stack in the section has its own water-oxygen separators which are manifolded into a single line coming out of the section container. All hydrogen, oxygen coolant, electrical and water storage pressure line connections at the section container are fastened to standard bulkhead fittings on the accessory pad.

After the stacks are completely assembled within the container, all void spaces are filled with unicellular foam. The purpose of this foaming is for vibration dampening, accoustical noise deadening and minimizing free gas volume to prevent possible fire propagation. Thin plastic covers are placed over the top and bottom of each stack to manifold oxygen to the stack and to keep the foam material from entering areas around the coolant manifolds and oxygen water separator.

Operation

The basic principle by which the fuel cell operates to produce electrical energy

and water, is the controlled oxidation of hydrogen. This is accomplished through the use of the solid electrolyte ion-exchange membrane. On the hydrogen side of the fuel cell, hydrogen gas disassociates on the catalytic electrode to provide hydrogen ions and electrons. The electrons are provided a conducting path of low resistance by the current collector, either to an external load or to the next series-connected fuel cell.

When a flow of electrons is allowed to do work and move to the oxygen side of the fuel cell, the reaction will proceed. By use and replacement, hydrogen ions flow through the solid electrolyte to the catalytic electrode on the oxygen side of the fuel cell. When electrons are available on this surface, oxygen disassociates and combines with the available hydrogen ions to form water. (See Figure 5-13).

The oxygen current collector provides the means of distributing electrons and condensing the product water on a surface to be transported away by the wick system through capillary action. The individual cell wicks are integrated into one large wick which routes the water to an absorbent material that separates the water from the gas.

By using the oxygen outlet pressure as a reference, a small pressure differential is obtained over the length of the water removal system. This pressure is sufficient to push the gas-free water toward the storage reservoir.

Waste heat, generated during the fuel cell battery operation, is dissipated by means of the recirculating coolant provided by the spacecraft cooling system. In addition, the total coolant flow provides the function of preheating the incoming reactant gases. In the spacecraft, the reactant gases are supplied to the fuel cell battery sections by the RSS. This system contains the reactant supply

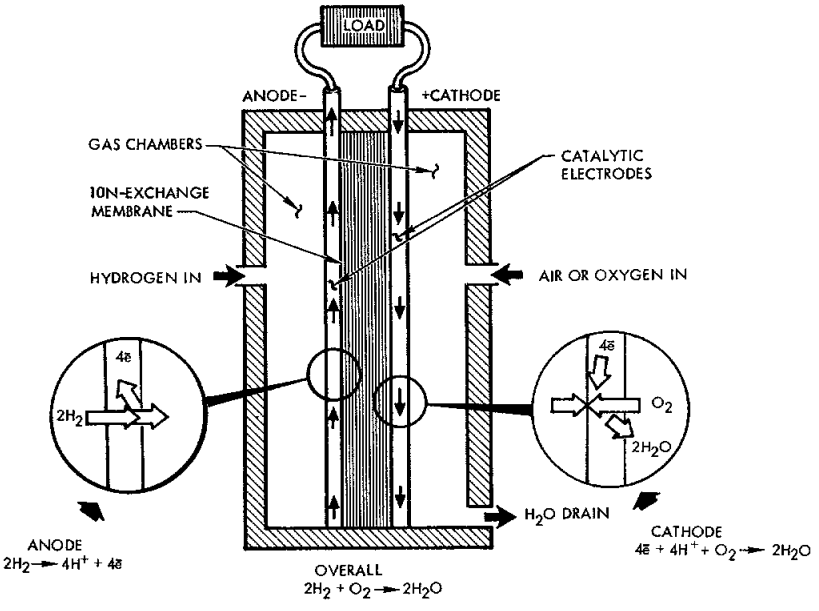


Figure 5-13 Principal of Operation.

tanks, control valves, heat exchangers, temperature sensors and heaters required for management of the fuel cell reactants. (See Figure 5-14).

REACTANT SUPPLY SYSTEM

The RSS is essentially a subsystem for the fuel cell battery sections. The system provides storage for the cryogenic hydrogen and oxygen, converts the reactants to gaseous form and controls the flow of the gases to the fuel cell battery sections. The RSS components are installed in the RSS/fuel cell module. On spacecraft 10, 11 and 12, the RSS O_2 requirements are supplied from a central cryogenic O_2 tank located in the RSS/fuel cell module. This vessel also supplies O_2 to the ECS. See Figure 5-6, 5-7 and 5-8 for component installation and Figure 5-14 for a functional diagram.

Components

Reactant Supply Tanks

Two tanks are utilized to separately contain the cryogenic hydrogen and oxygen required for the operation of the fuel cell battery sections. The tanks are thermally insulated to minimize heat conduction to the stored elements which would cause the homogeneous solution to revert to a mixture of gas and liquid. The tanks are capable of maintaining the stored liquids at supercritical pressures and cryogenic temperatures.

The approximate total amount of liquid stored in the hydrogen vessel is 22.25 lbs for long mission configurations and 5.80 lbs for short mission configurations. The approximate total amount of liquid stored in the oxygen vessel is 180 lbs on spacecraft 5, 106 lbs on spacecraft 10, 11 and 12 (for ECS and RSS) and 46.0 lbs on spacecraft 8 and 9.

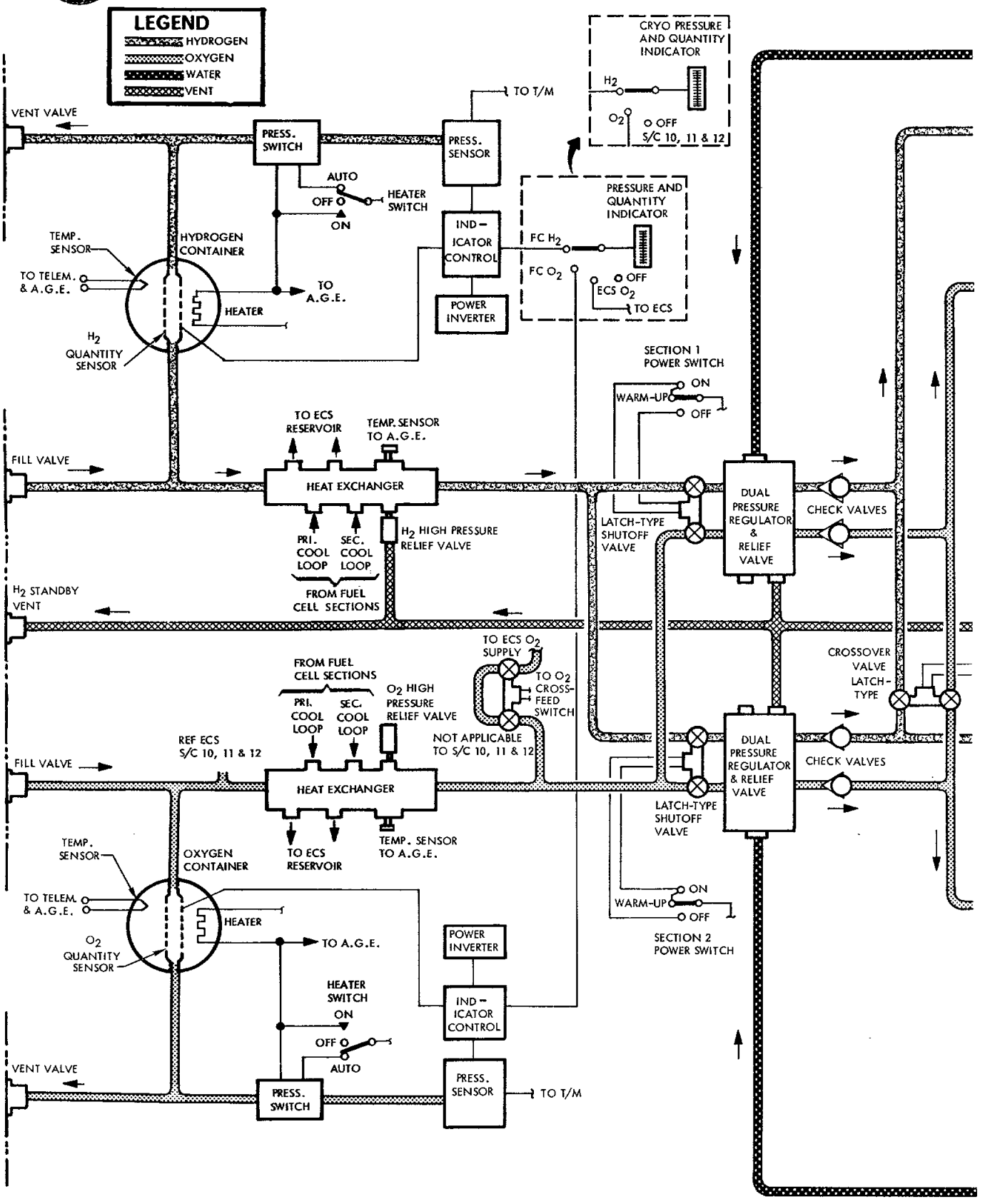


Figure 5-14 RSS/Fuel Cell System Functional Diagram (S/C 5 & 8 thru 12) (Sheet 1 of 2)

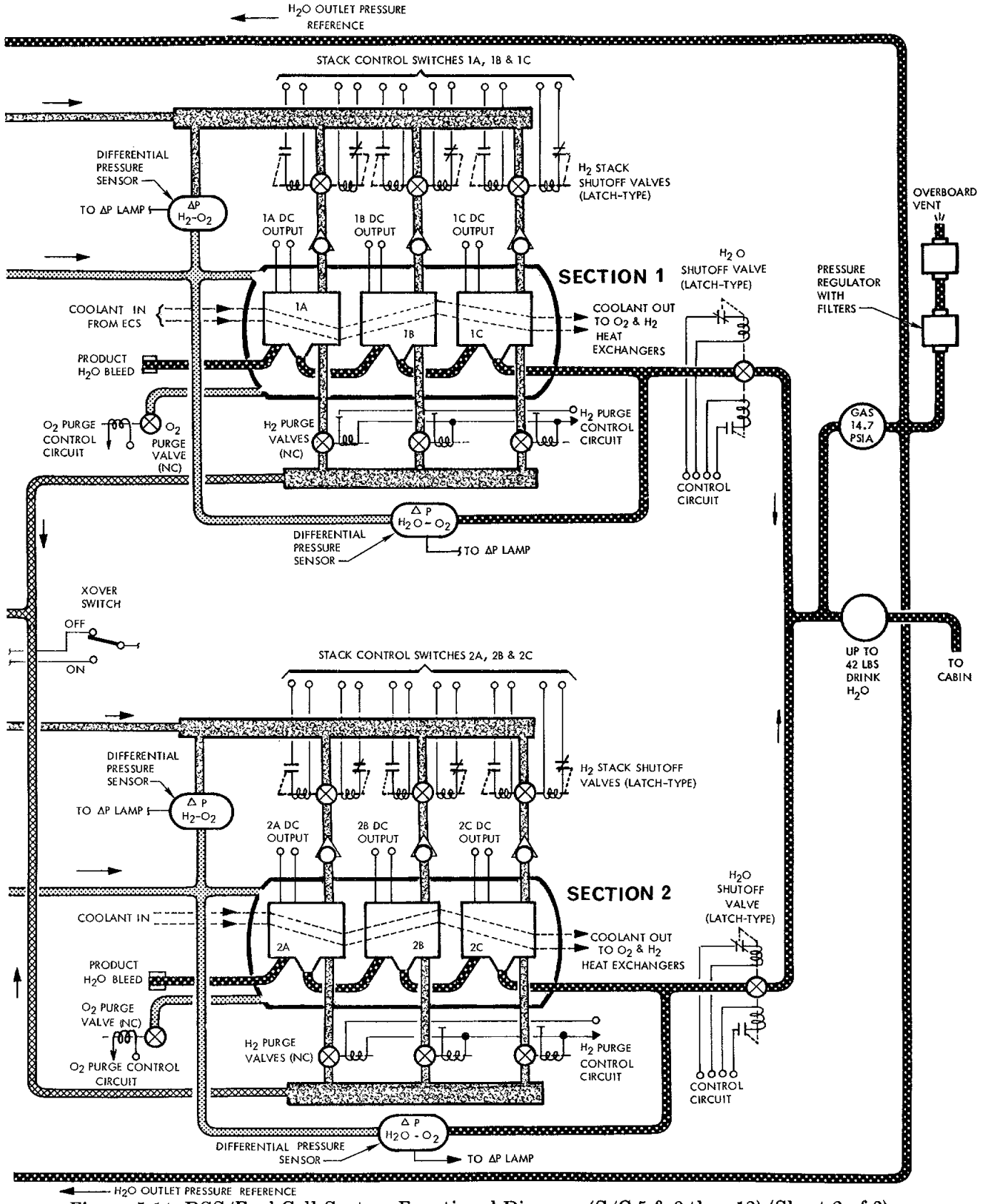


Figure 5-14 RSS/Fuel Cell System Functional Diagram (S/C 5 & 8 thru 12) (Sheet 2 of 2)

The hydrogen vessel is composed of titanium alloy and the oxygen vessel is made of a high strength nickel base alloy. Both vessels are spherical in shape and double walled. A vacuum space between the inner and outer wall (approximately one inch) provides thermal insulation from ambient heat conduction. The inner wall is supported in relation to the outer wall by an insulating material supplemented by compression loading devices.

Each storage tank contains a fluid quantity sensor, a pressure sensor, a temperature sensor and an electrical heater (the O_2 tank on spacecraft 10, 11 and 12 has two heaters) installed in the inner vessel in intimate contact with the stored reactants. The fluid quantity sensor is an integral capacitance unit which operates in conjunction with an indicator control unit containing a null bridge amplifier.

The sensor varies the capacitance (in proportion to fluid level) in a circuit connected to the null bridge amplifier. The amplified signal is then used to drive a servo motor, which in turn operates a visual indicator for quantity indication. Power inverters supply 400 cycle, 26 vac power to the fluid quantity circuits.

The temperature sensor is a platinum resistance device capable of transmitting a source signal to a balanced bridge circuit. The sensor provides cryogenic fluid temperature monitoring for telemetry and AGE.

The pressure sensor is a dual resistive element, diaphragm type transducer. The sensor provides signals for cryogenic fluid pressure monitoring on a spacecraft meter. The electrical heaters provide a method of accelerating pressure build-up in the reactant supply tanks. The heaters may be operated either in a manual or

automatic mode. In the automatic mode a pressure switch removes power from the heater element when the tank pressure builds up to a nominal 900 psig in the oxygen tank and a nominal 250 psig in the hydrogen tank. In the manual mode a spacecraft pressure meter indicates proper switch operation.

Fill and Vent Valves

The fill and vent valves provide a dual function in permitting simultaneous fill and vent operations. Quick disconnect fittings are provided for rapid ground service connection to both the storage tank fill check valve and the vent check valve. When fill connections are made, the pressure of the ground service connection against the fill and vent valve poppet shaft simultaneously opens both the fill and vent ports. When ground service equipment is removed, the valve poppet automatically returns to its normally spring-loaded-closed position. The vent check valve is a single poppet type, spring-loaded-closed valve which opens when system pressure exceeds 20 psig to relieve through the fill and vent valve vent ports.

Heat Exchangers

The supply temperature control heat exchangers are finned heat exchangers in which the supply fluid temperature is automatically controlled by absorbing heat from the recirculating coolant fluid of the spacecraft cooling system. The special double-pass design precludes freezing of the coolant and assures a reactant fluid supply at 50°F minimum and 140° maximum.

Dual Pressure Regulator and Relief Valves

The dual pressure regulator and relief valves are normally open poppet type regulators which control downstream pressure to the fuel cell battery sections. The

regulators maintain the hydrogen pressure at approximately 21.7 psia and the oxygen pressure at approximately 22.2 psia. The oxygen side of the regulators is referenced to hydrogen pressure. The hydrogen side of the regulators is referenced to produce H_2O pressure.

The relief valves provide overpressurization protection for the regulated pressure to the fuel cell battery sections. This valve is precalibrated to operate at a pressure of approximately 10 psia above the normal supply level.

High Pressure Relief Valves

The high pressure relief valves are single poppet type, spring-loaded-closed valves which provide system overpressurization protection. The valves vent system gas to ambient when pressure exceeds the system limits.

Solenoid Shutoff Valves

The solenoid shutoff valves are solenoid operated latching type valves which eliminate fluid loss during the nonoperating standby periods. The valves are normally open and are closed only during fill and standby periods by applying power to the solenoids.

Crossover Valve

The crossover valve is a solenoid operated latching type valve which provides the capability of selecting both dual pressure regulators to supply hydrogen and oxygen to a fuel cell battery section for the purpose of increasing flow rate for more effective purging. The crossover valve is controlled by the X-OVER switch on the right instrument panel.

O_2 Crossfeed Reactant Valve (spacecraft 8 and 9)

The O_2 crossfeed reactant valve is a solenoid operated, latching type valve which

provides the capability of pressurizing the RSS with O_2 from the ECS oxygen supply. This provides a redundant method of supplying the proper reactant O_2 pressure to the fuel cell battery sections in the event of a malfunction in the RSS oxygen supply. The crossfeed valve is controlled by the O_2 CROSS-FEED switch located on the center instrument panel.

Operation

During pre-launch, the two separate reactant supply tanks are serviced (using AGE equipment) with liquid hydrogen and oxygen. After the tanks are filled, in order to accelerate pressure buildup within the tanks, the internal tank heaters are operated, utilizing external electrical power. In approximately one hour the liquid is converted into a high density, homogeneous fluid at a constant pressure.

During the fill operation, the solenoid shutoff valves between the storage tanks and the dual pressure regulators are closed. Once operating pressure is obtained, the solenoid shutoff valves may be opened by applying power to the coil of the valves. The high density, homogeneous fluid will then flow upon demand.

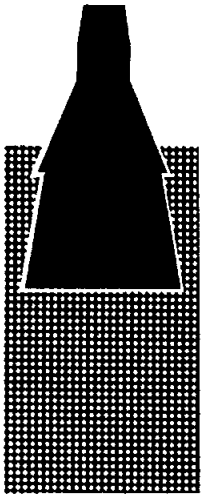
The fluid flows from the supply tanks to the heat exchangers. The fluid temperature, when entering the heat exchangers is approximately -279°F for the oxygen and approximately -423°F for the hydrogen. The heat exchangers absorb heat from the recirculating coolant fluid of the spacecraft cooling system. This heat, applied to the high density fluid, raises the temperature of the reactants to approximately 50°F to 140°F.

The reactants, now in gaseous form, flow through the heat exchangers, past the high pressure relief valves and AGE temperature sensors, to the supply solenoid shutoff valves. During fuel cell battery operation, if the demand on the fluid

flow is inadequate to keep tank pressures within limits, the high pressure relief valves will vent, externally, the excess fluid. The AGE temperature sensors on the heat exchangers are used for pre-launch checkout only.

The reactants flow through the supply solenoid shutoff valves to the dual pressure regulator and relief valves. The dual pressure regulators reduce the pressure of the reactants to approximately 21.7 psia for the hydrogen and approximately 20.5 psia for the oxygen. The gas now flows through the manual shutoff valves and is then directed to the fuel cell battery sections at a flow rate that is determined by both the electrical load applied and the frequency of purging. The flow rate of the gases may be increased for more effective purging by opening the crossover valve.

After launch, the supply tank heaters are operated by spacecraft power. The heaters operate as required to maintain proper system pressures.



ENVIRONMENTAL CONTROL SYSTEM

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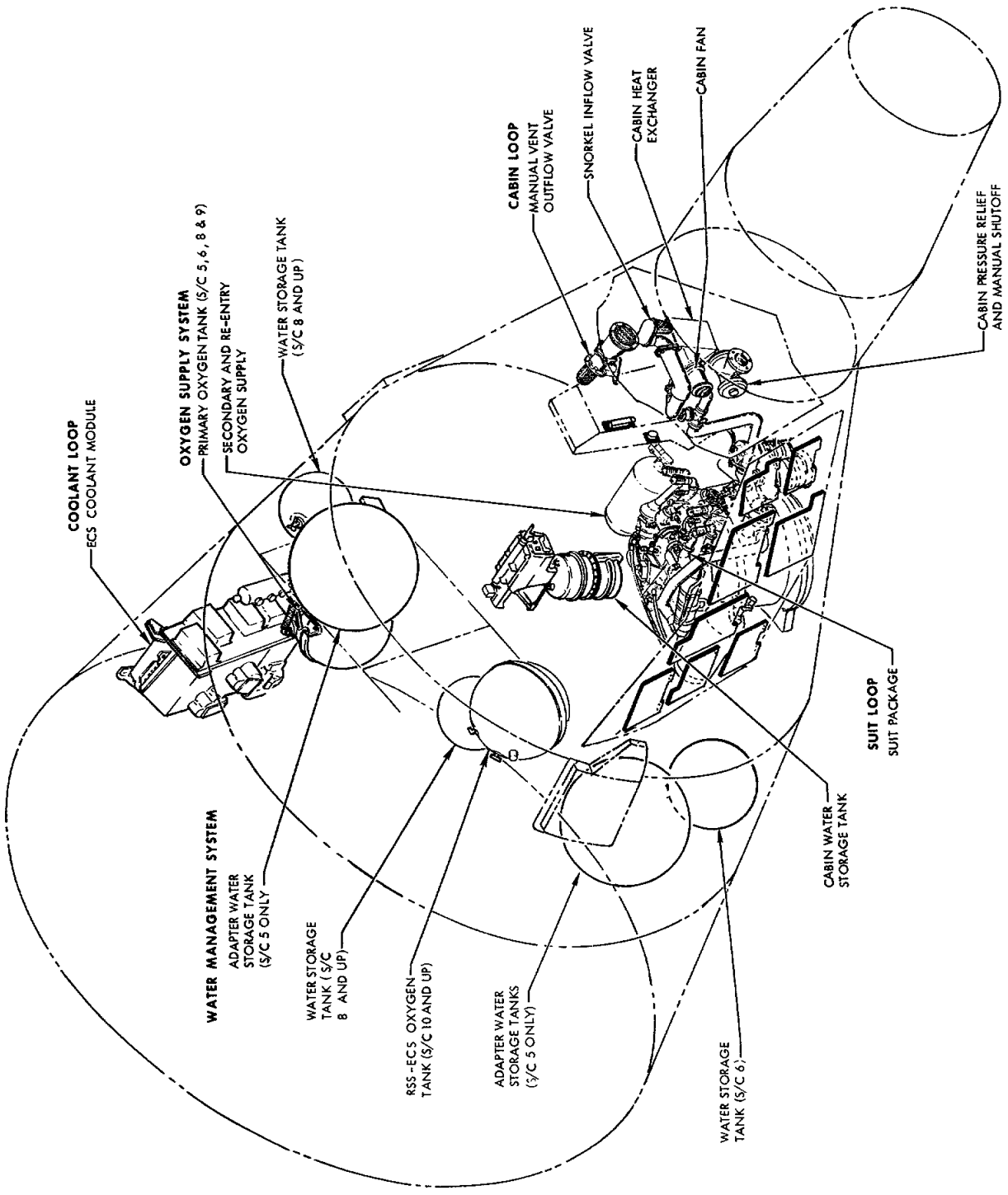


Figure 6-1 Environmental Control System

SECTION VI ENVIRONMENTAL CONTROL SYSTEM

SYSTEM DESCRIPTION

The Environmental Control System (ECS) (Figure 6-1, 6-2) may be defined as a system which provides a safe and comfortable gaseous atmosphere for the pilots. The system must perform such tasks as providing fresh oxygen, pressurization, temperature control, water removal and toxic gas removal. In addition to providing atmospheric control for the pilots, the system provides equipment cooling and regulated temperatures for certain pieces of equipment.

For ease of understanding, the Environmental Control System may be separated into four systems or loops which operate somewhat independent of each other. These loops are:

(1) The oxygen supply system.

(2) The cabin loop.

(3) The suit loop.

(4) The water management system.

OXYGEN SUPPLY SYSTEM

There are three oxygen systems: Primary, Secondary and Egress.

Primary Oxygen (Figure 6-3, 6-4)

This system stores and dispenses oxygen for breathing and for suit and cabin pressurization.

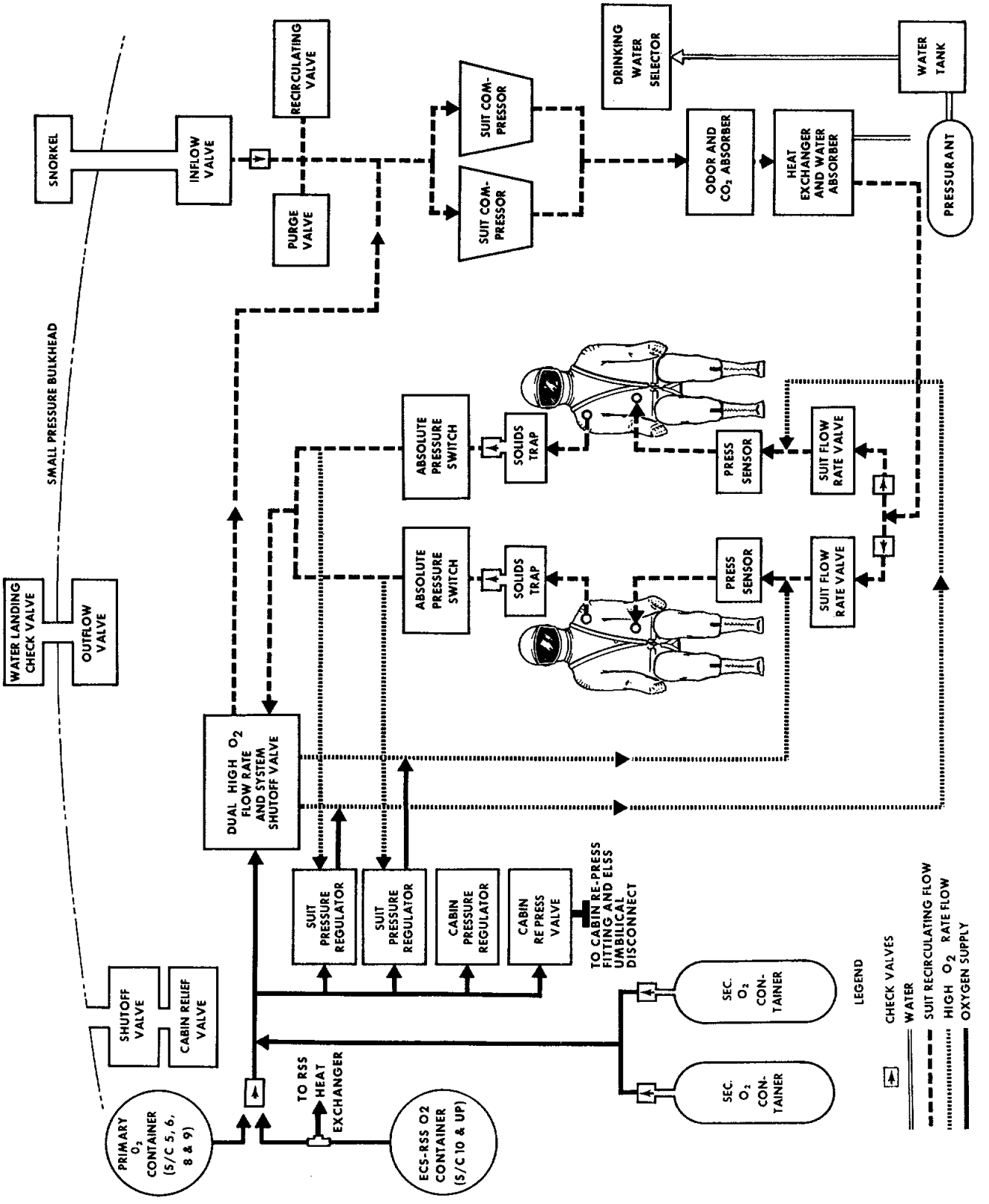


Figure 6-2 ECS Block Diagram

This system provides oxygen during the period commencing two hours prior to launch and terminating with jettison of the adapter section at retrograde.

The primary oxygen supply is stored at supercritical pressure in a cryogenic spherical container in the adapter section of the spacecraft. This container is filled with liquid oxygen at atmospheric pressure. Heat is supplied by thermal leakage through the container insulation and by activation of an electric heater in order to build pressure to the critical point of 736 psia. Above this point liquid oxygen becomes a homogeneous mixture, described for simplicity as a dense supercritical fluid. This fluid is warmed, regulated and filtered before it enters the suit or cabin loop.

On spacecraft 10 through 12, a long mission ECS oxygen tank will replace the short mission Reactant Supply System (RSS) oxygen tank and the ECS oxygen tank, allowing ECS breathing and the RSS oxygen to be stored in the same container. A tee fitting on the cryogenic line allows both the ECS and RSS systems to receive oxygen from the common container.

The primary loop consists of the following components: primary oxygen container, pressure control switch, pressure transducer, fill and vent valves, temperature discharge sensor, pressure relief valve, pressure regulator, shutoff valve, filter, check valves, and heat exchanger.

Secondary Oxygen (Figure 6-3, 6-4)

The secondary oxygen system is capable of performing the same functions as the primary oxygen system and operates when pressure in the primary system falls below 75 $\pm$ 10 psi. At retrograde, when the primary oxygen container is jettisoned

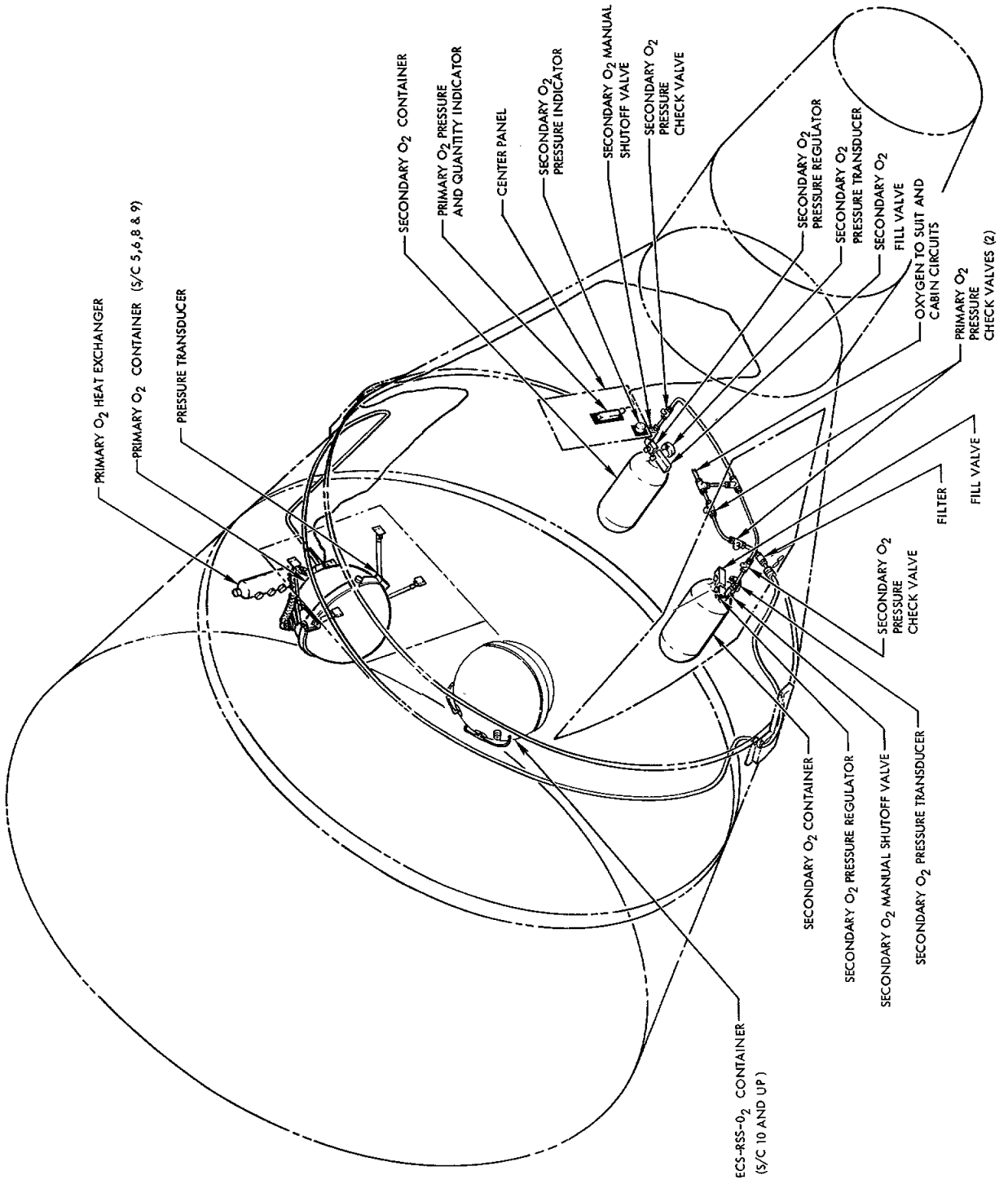


Figure 6-3 Primary and Secondary Oxygen System

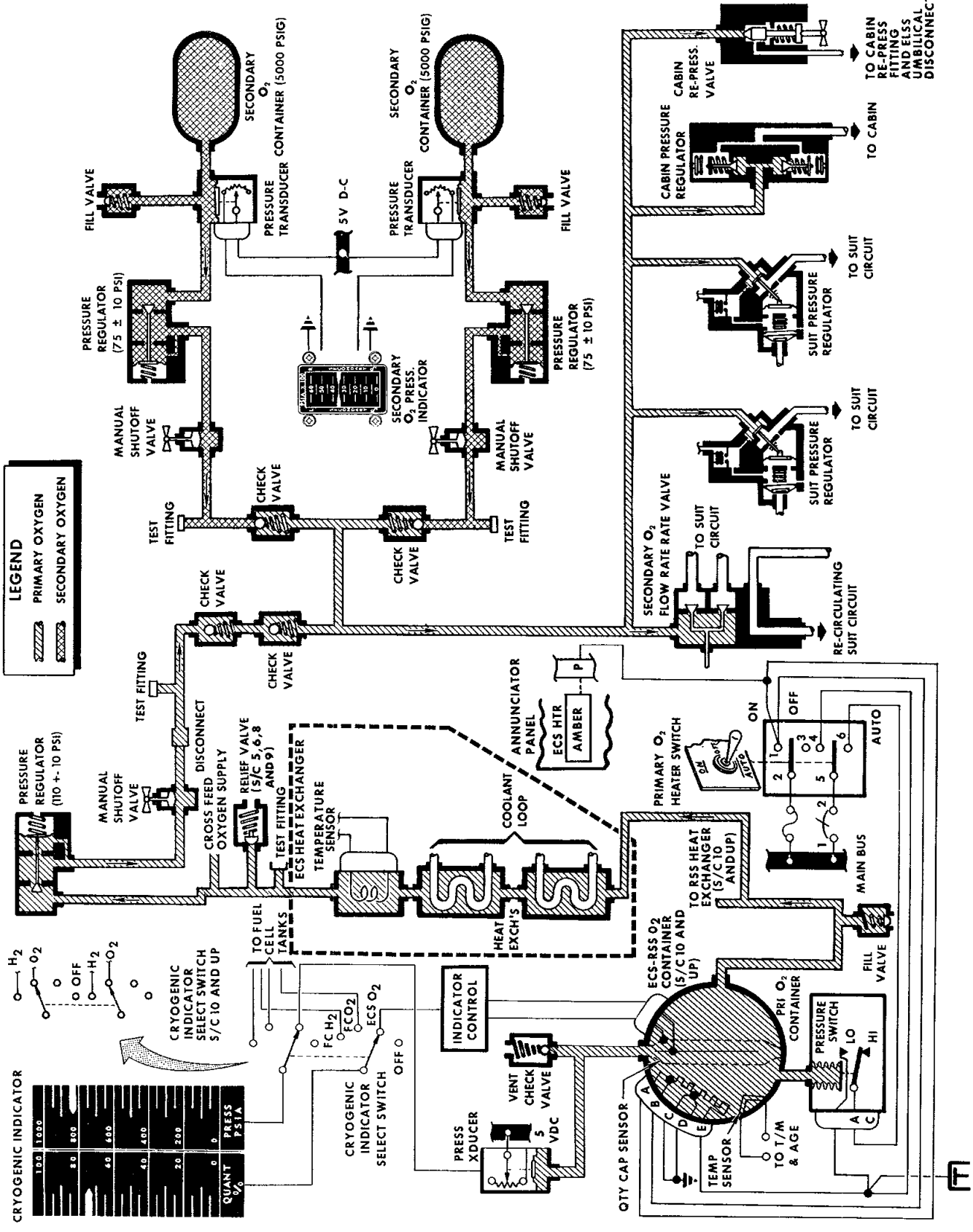


Figure 6-4 Primary and Secondary Oxygen Schematic

with the equipment adapter, the secondary oxygen system assumes the duties of the primary oxygen system.

The gaseous secondary oxygen supply is stored in two cylinders located in the re-entry module. Each tank contains 6.5 pounds of usable oxygen pressurized to 5000 psig maximum at 70°F. This oxygen supply is then regulated before it enters the suit or cabin loop.

The secondary system consists of two: tanks, fill valves, transducers, pressure regulators, shutoff valves and check valves.

Egress System (Figure 6-5)

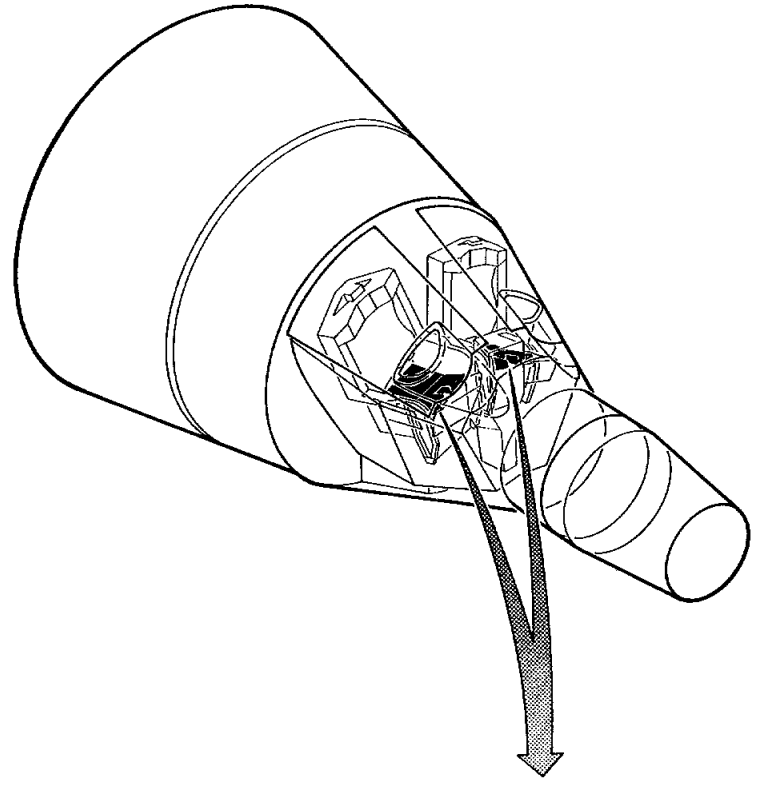
This system provides each pilot with oxygen for breathing and for suit pressurization in the event that they initiate ejection procedures at 70,000 feet or below, during launch or re-entry. The egress oxygen is provided on spacecraft 5 and 6 only.

The egress gaseous oxygen supply is stored in a tank located in each seat-mounted egress kit. Each tank contains 0.31 pound of usable pressurized oxygen.

Each egress system consists of a tank, pressure regulator, pressure gage, restrictor, check valve, shutoff valve, and composite disconnects.

CABIN LOOP (Figure 6-6)

Design cabin leakage at ground test conditions is 670 standard cubic centimeters per minute, (scc/min) of nitrogen at 5.0 psig. Makeup oxygen, to maintain cabin pressure at nominal 5.1 psia level, is called for by the cabin pressure regulator. In order to obtain maximum utilization of oxygen, it first passes



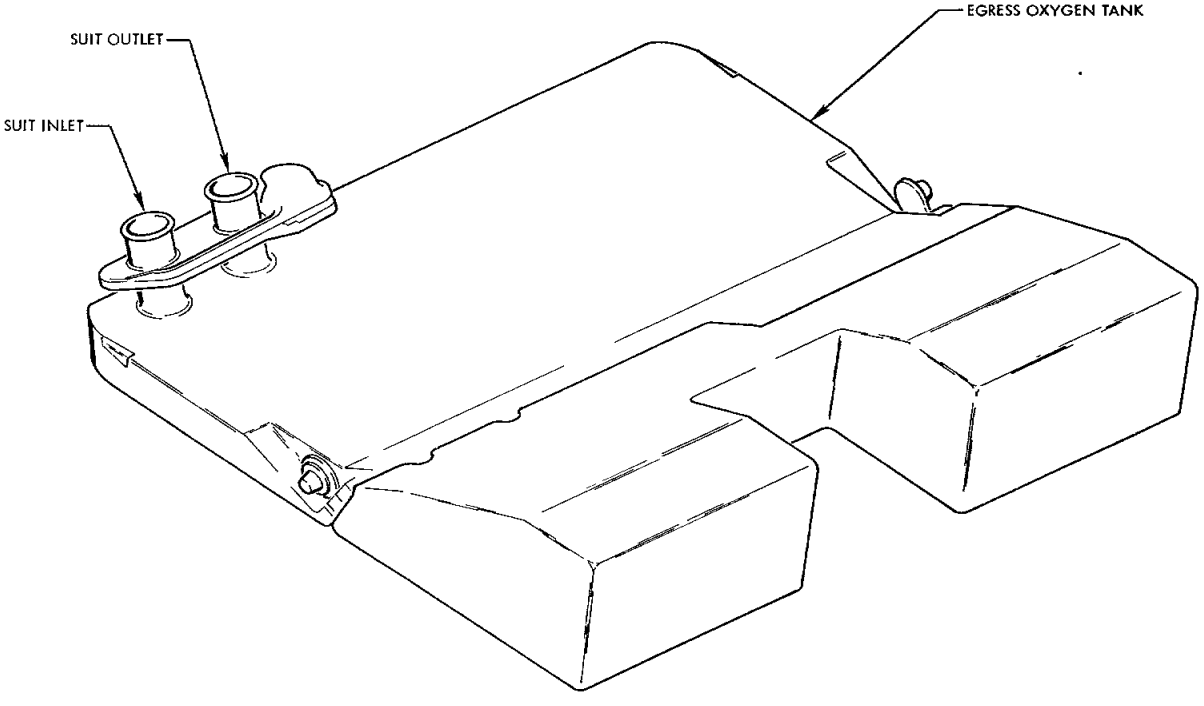


Figure 6-5 Egress Oxygen System (S/C 5 & 6)

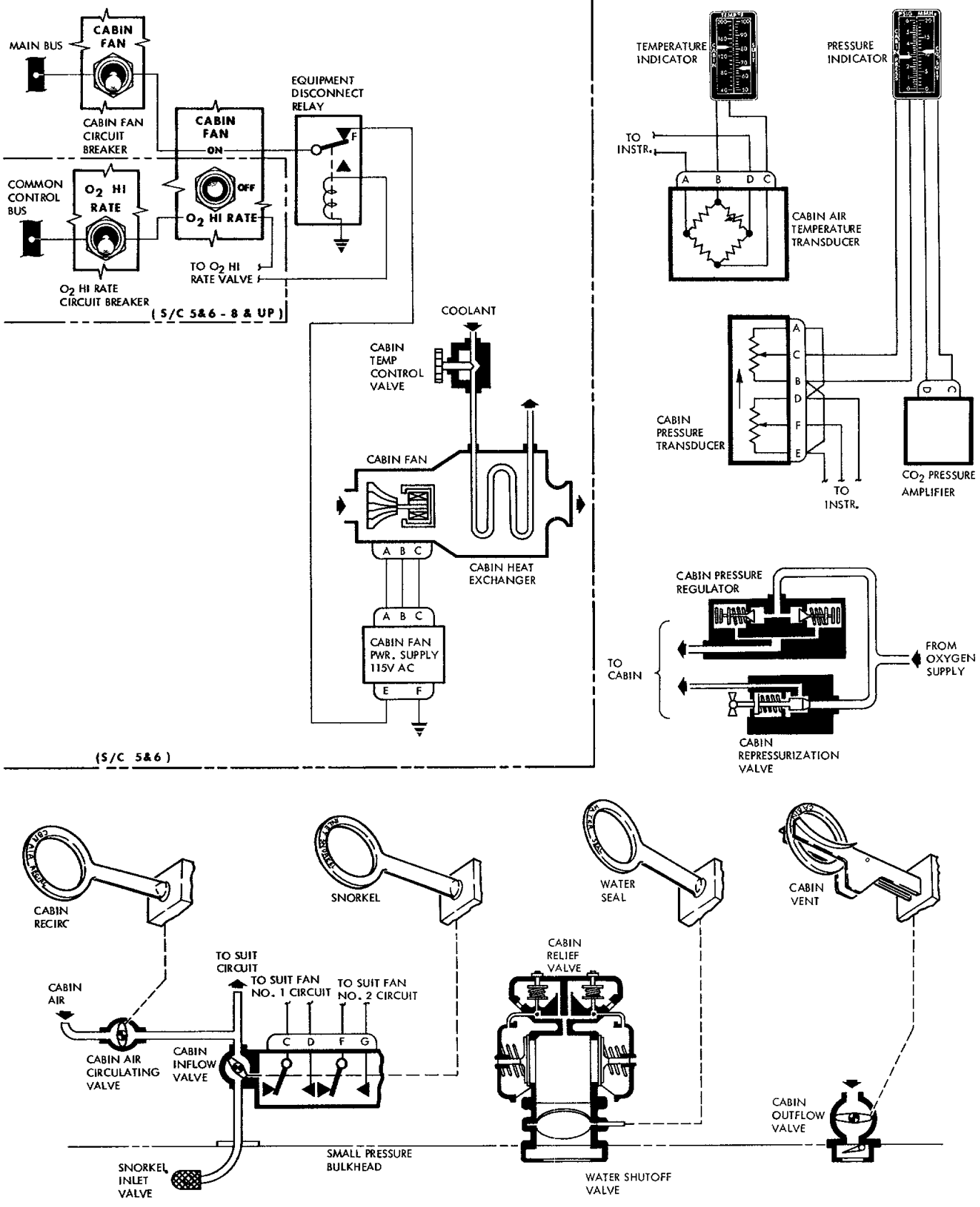


Figure 6-6 Cabin Environmental Control

through the suit loop before it is dumped into the cabin through the suit pressure relief valves.

Primary cabin components for spacecraft 5 and 6 are a cabin heat exchanger and a fan. These parts have been removed from spacecraft 8 through 12. This loop also contains a relief valve for both positive and negative pressure relief, a pressure regulator and manual valves to either dump cabin pressure or repressurize. In the latter operation, oxygen is supplied directly to the cabin.

SUIT LOOP (Figure 6-7, 6-8, 6-9)

The pilots are provided with redundant atmospheres by having a closed pressure suit circuit within the pressurized cabin. This suit circuit provides for cooling, pressurization, purification and water removal.

The suit loop is a closed system with two pressure suits operating in parallel. Circulation of oxygen through the suit is provided by a centrifugal compressor. Carbon dioxide and odors are removed by an absorber bed containing lithium hydroxide and activated charcoal. The gases are cooled in a heat exchanger by a liquid coolant, Monsanto MCS 198, to a temperature below the dew point. Water condensing within the heat exchanger is dumped overboard or routed to the water evaporator. The reconditioned oxygen is mixed with fresh makeup oxygen.

The suit circuit has two modes of operation, the normal recirculation mode which was discussed in the previous paragraph and the high rate mode which shuts off the recirculation system and dumps oxygen directly into the suit.

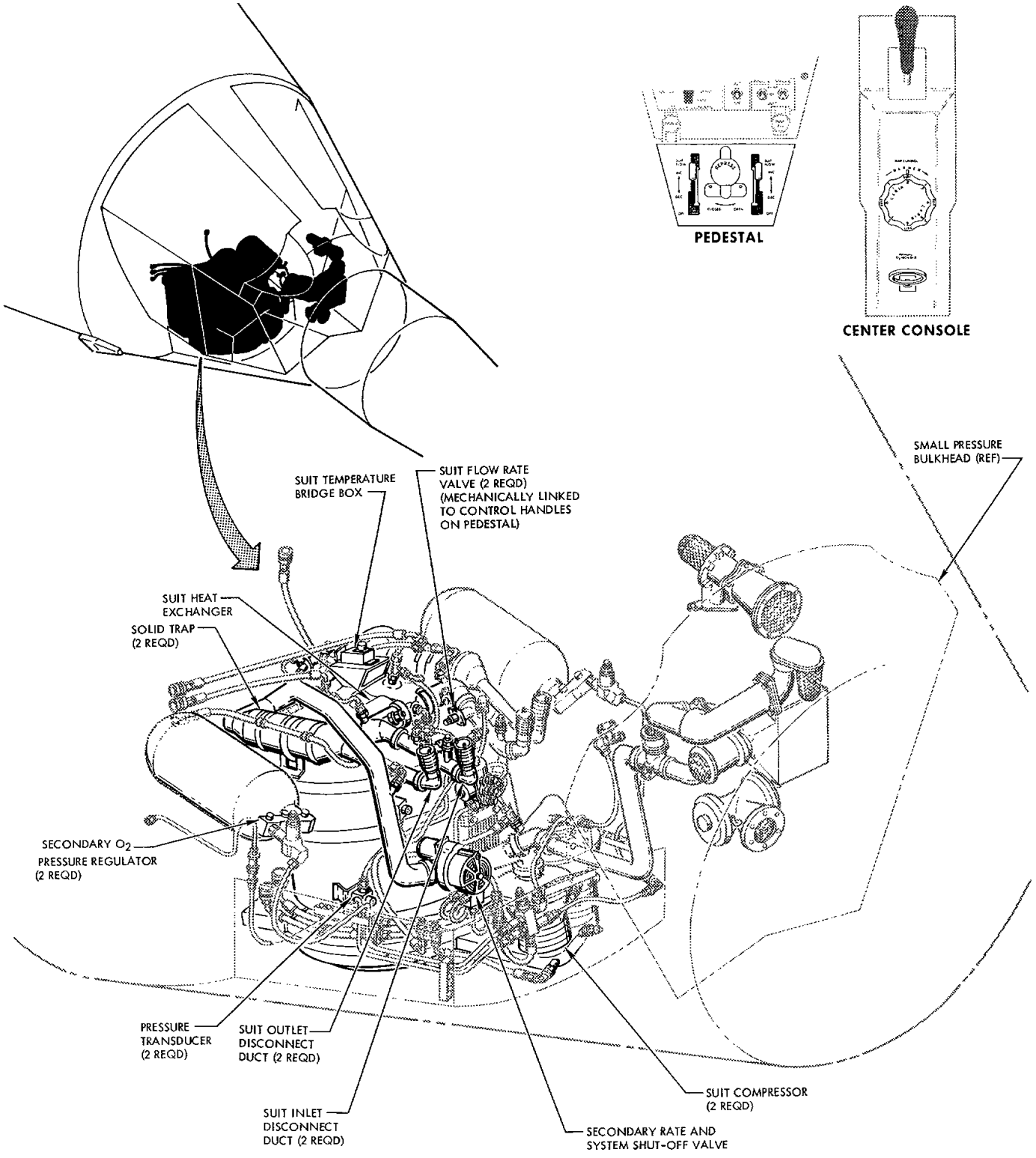


Figure 6-7 Suit Loop (ECS Package) (Sheet 1 of 2)

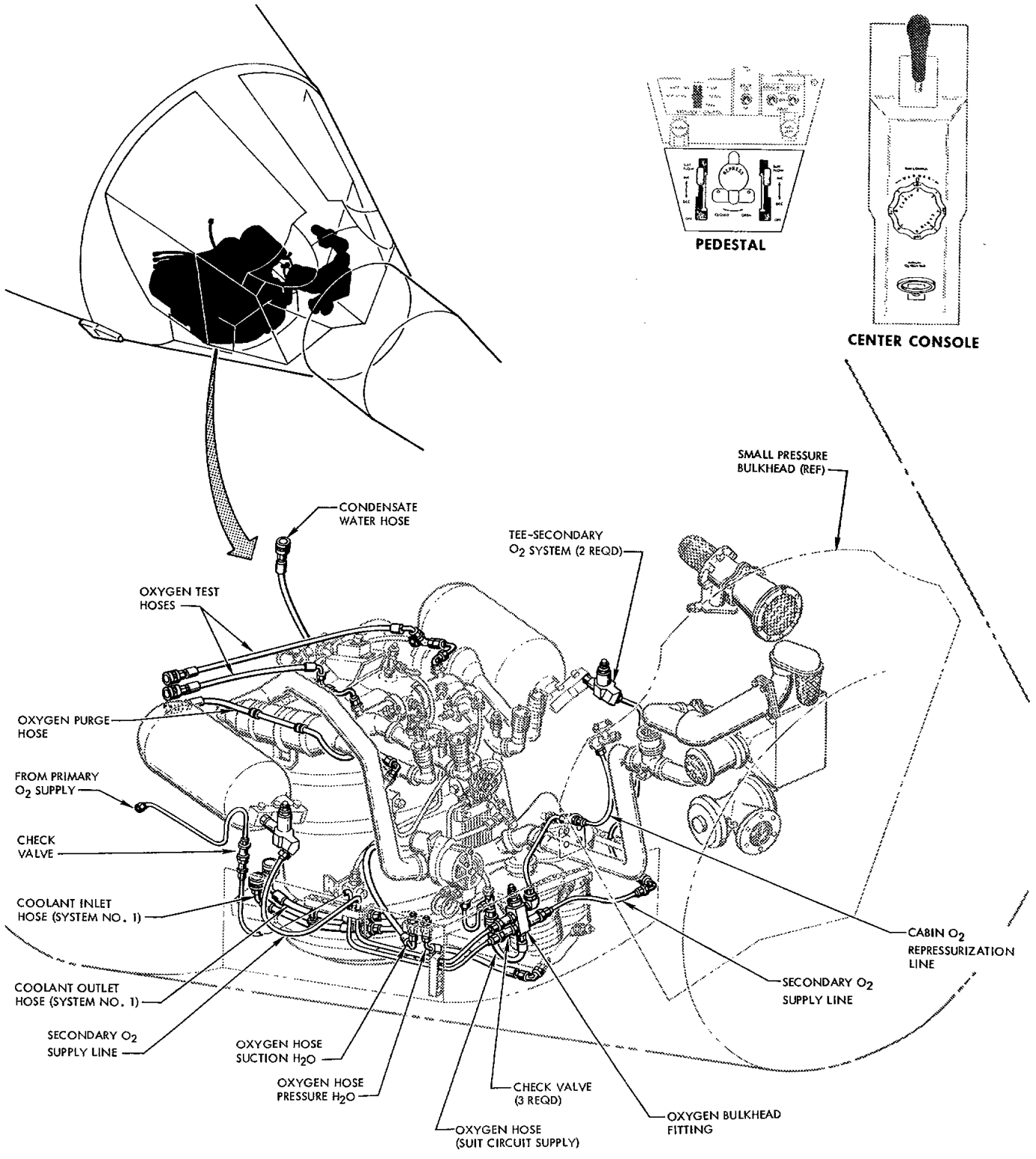


Figure 6-7 Suit Loop (ECS Package) (Sheet 2 of 2)

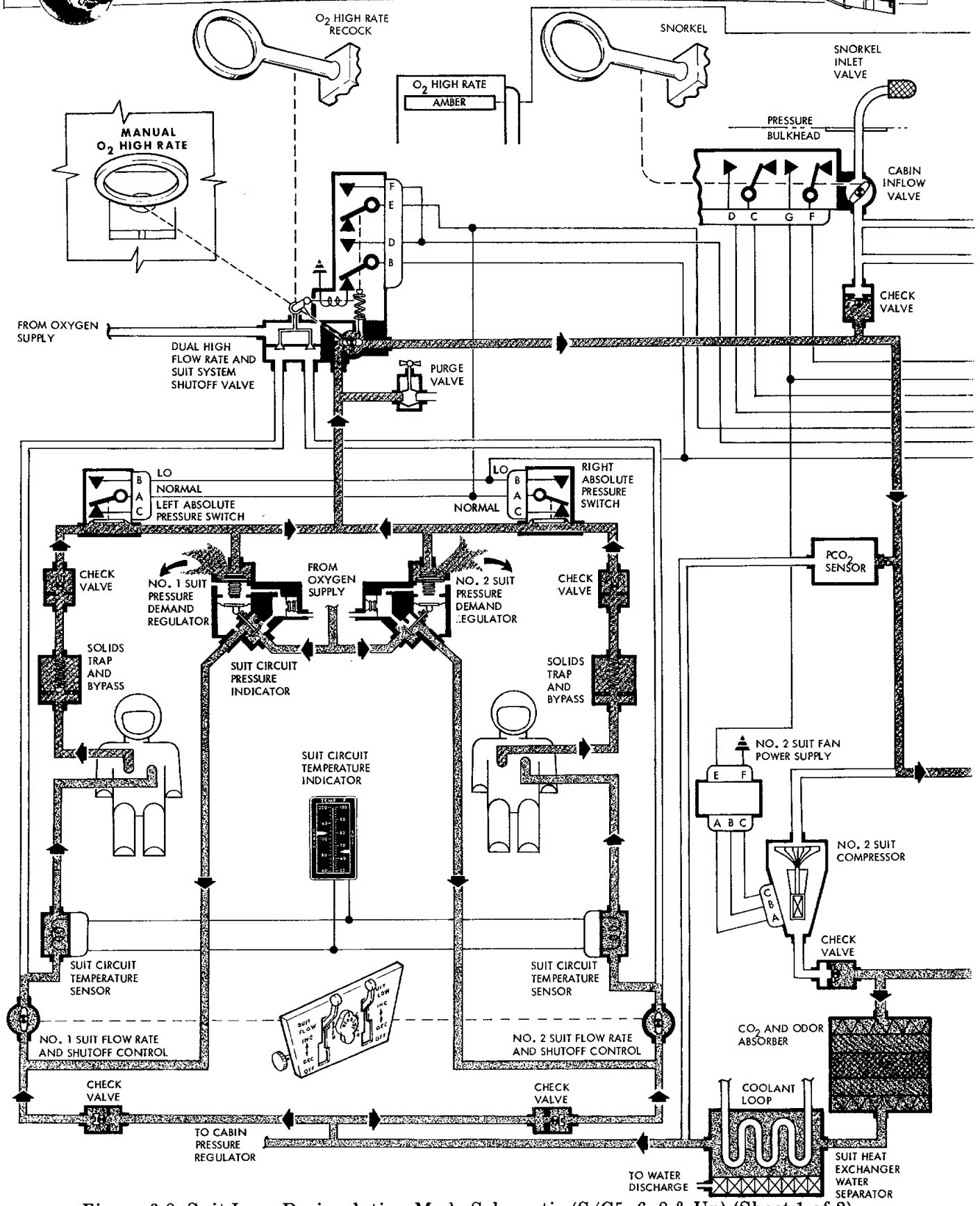


Figure 6-8 Suit Loop Recirculation Mode Schematic (S/C5, 6, 8 & Up) (Sheet 1 of 2)

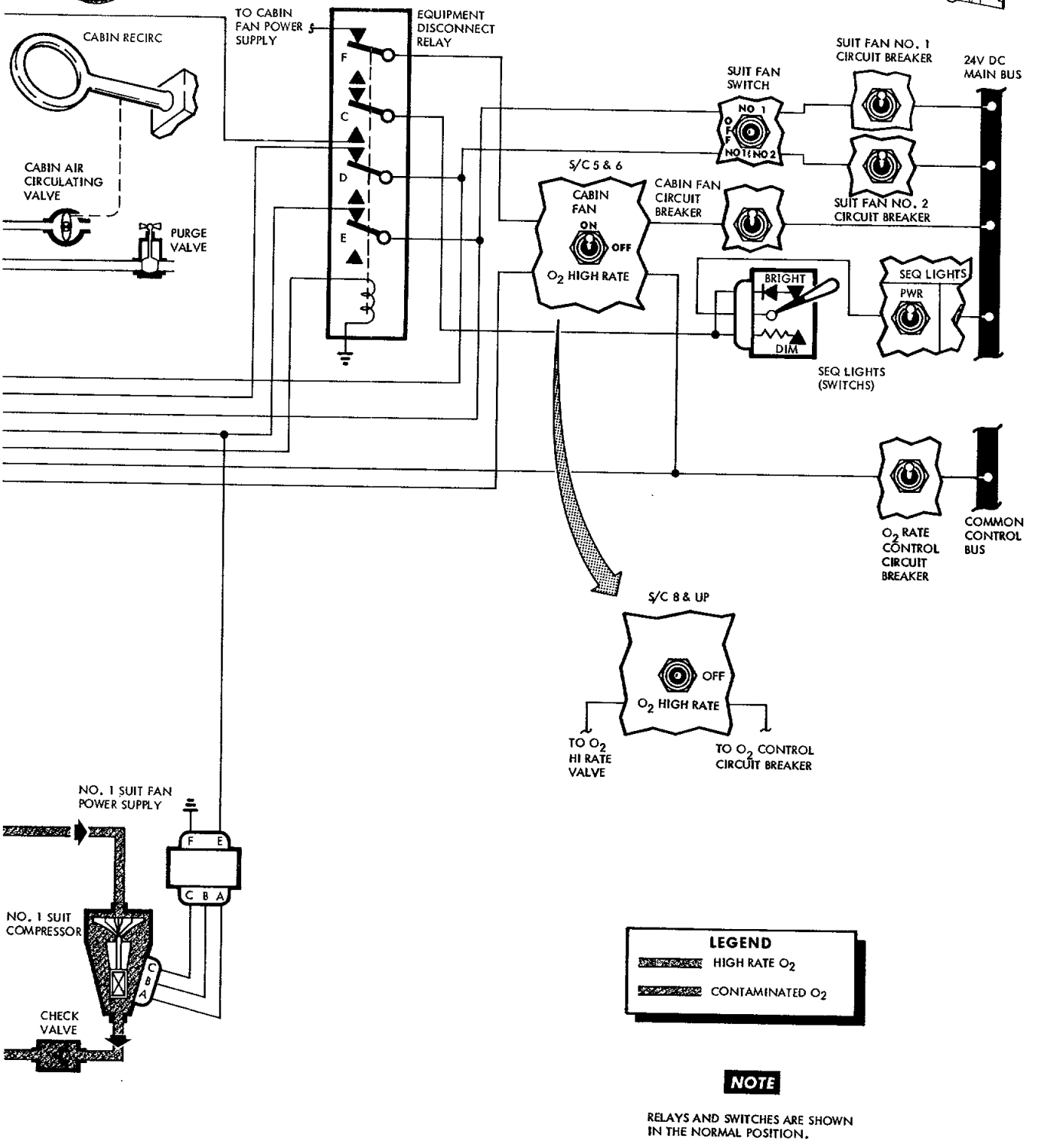


Figure 6-8 Suit Loop Recirculation Mode Schematic (S/C5, 6, 8 & Up) (Sheet 2 of 2)

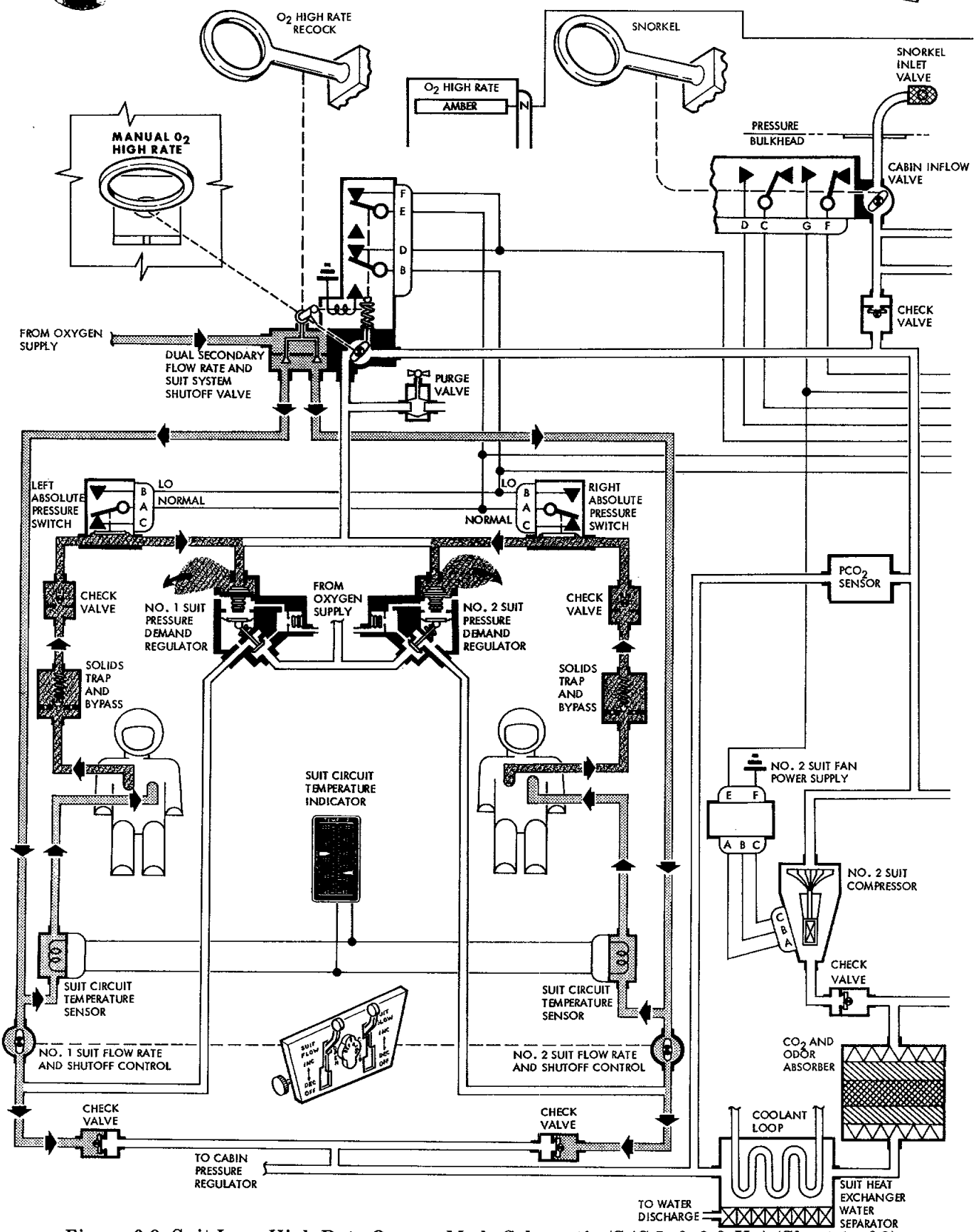


Figure 6-9 Suit Loop High Rate Oxygen Mode Schematic (S/C 5, 6, 8 & Up) (Sheet 1 of 2)

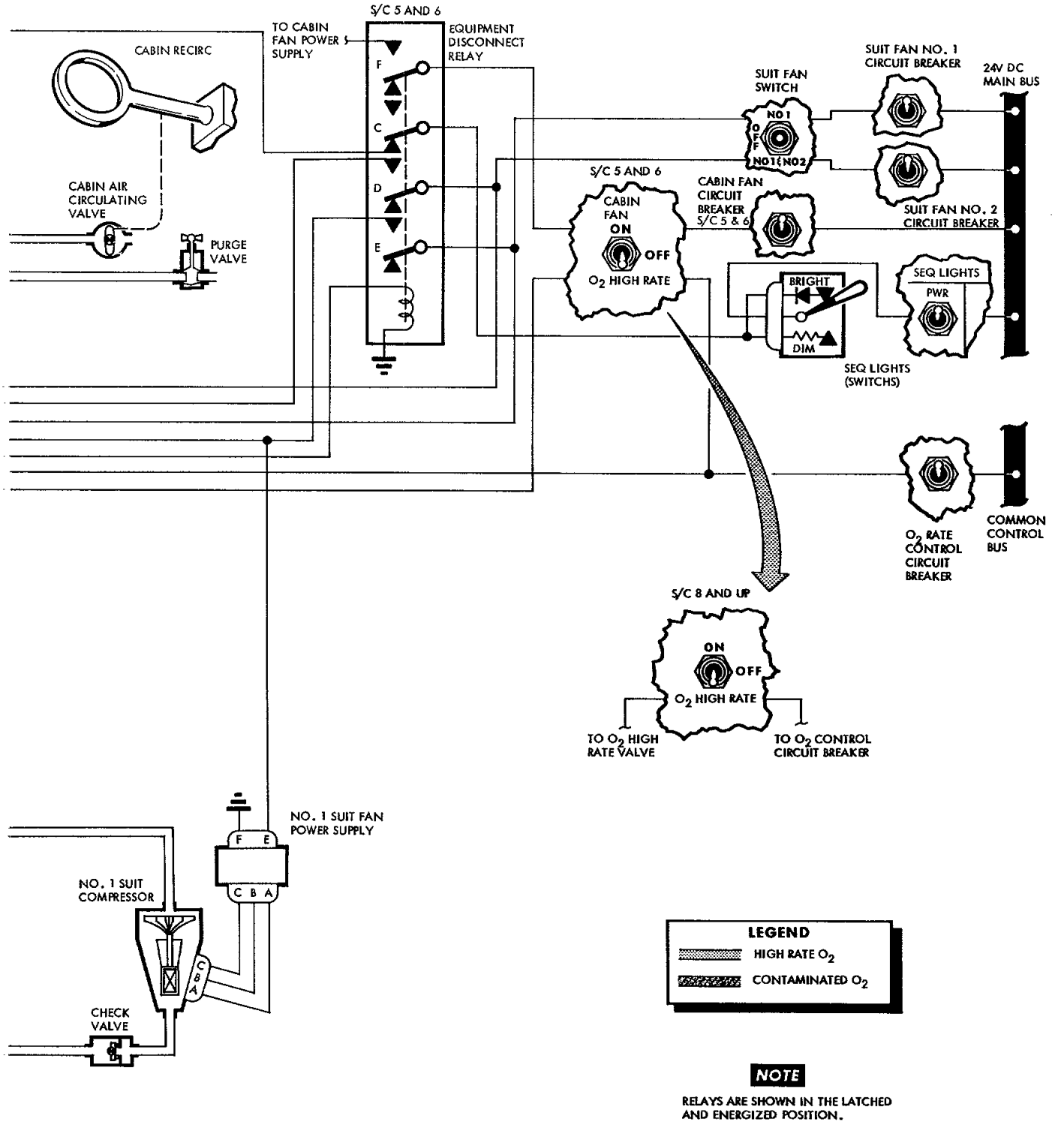


Figure 6-9 Suit Loop High Rate Oxygen Mode Schematic (S/C 5, 6, 8 & Up) (Sheet 2 of 2)

The suit loop consists of two suit pressure demand regulator valves, four check valves, two throttle valves, two solid traps, a system shutoff and high flow rate valve, two compressors, one carbon dioxide and odor absorber, and a suit heat exchanger.

WATER MANAGEMENT SYSTEM (Figure 6-10)

The purpose of the water management system is to store and dispense drinking water, collect and route unwanted water to the evaporator or dump overboard.

Drinking water is stored in a tank or tanks in the adapter. Each tank contains a bladder and is pressurized to supply water to the transparant tank in the re-entry module.

Spacecraft 5 utilizes two drinking water storage tanks which store both the drinking water and the fuel cell by-product water. One tank uses a combination of oxygen and fuel cell water as the pressurant while the other tank is pressurized with oxygen.

Spacecraft 6 has only one storage tank and uses oxygen for the pressurant.

Spacecraft 8 through 12 utilizes two storage tanks. Fuel cell by-product water is used as the pressurant for the drinking storage tank. The other tank is used to store fuel cell water.

Urine and condensated water from the suit circuit heat exchanger is absorbed by the wick in the water boiler or dumped overboard.

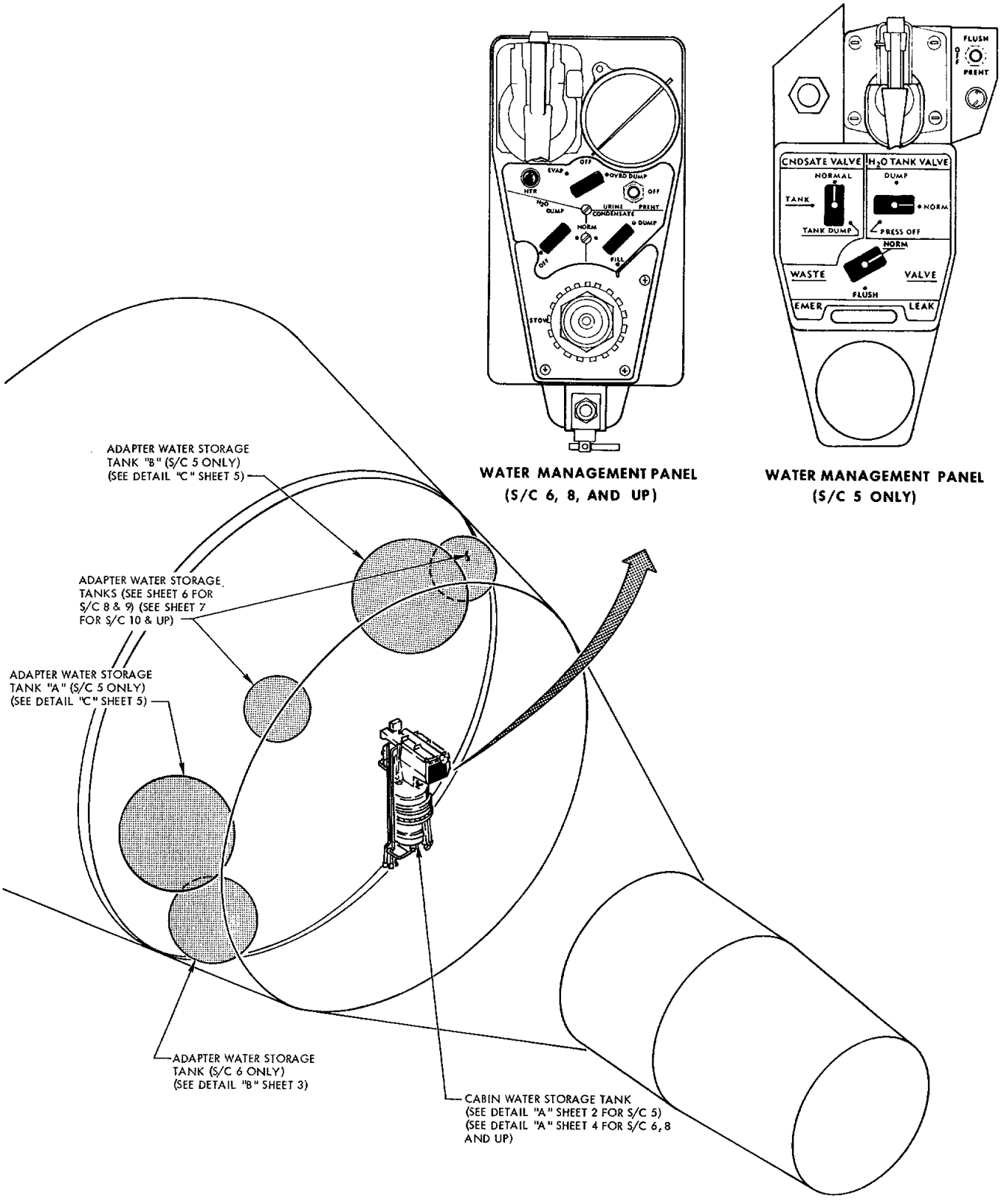


Figure 6-10 Water Management System (Sheet 1 of 7)

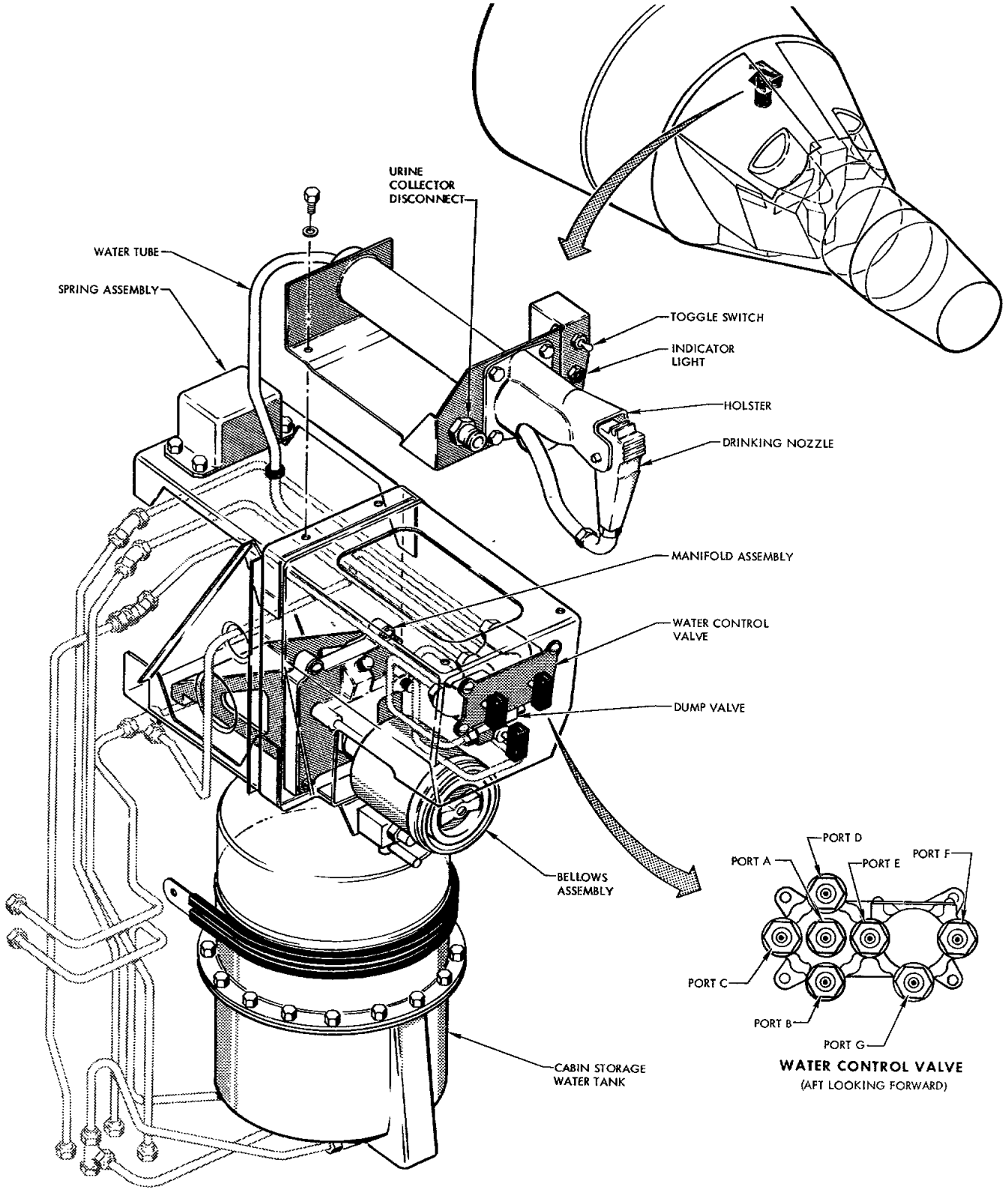


Figure 6-10 Water Management System (Sheet 2 of 7)

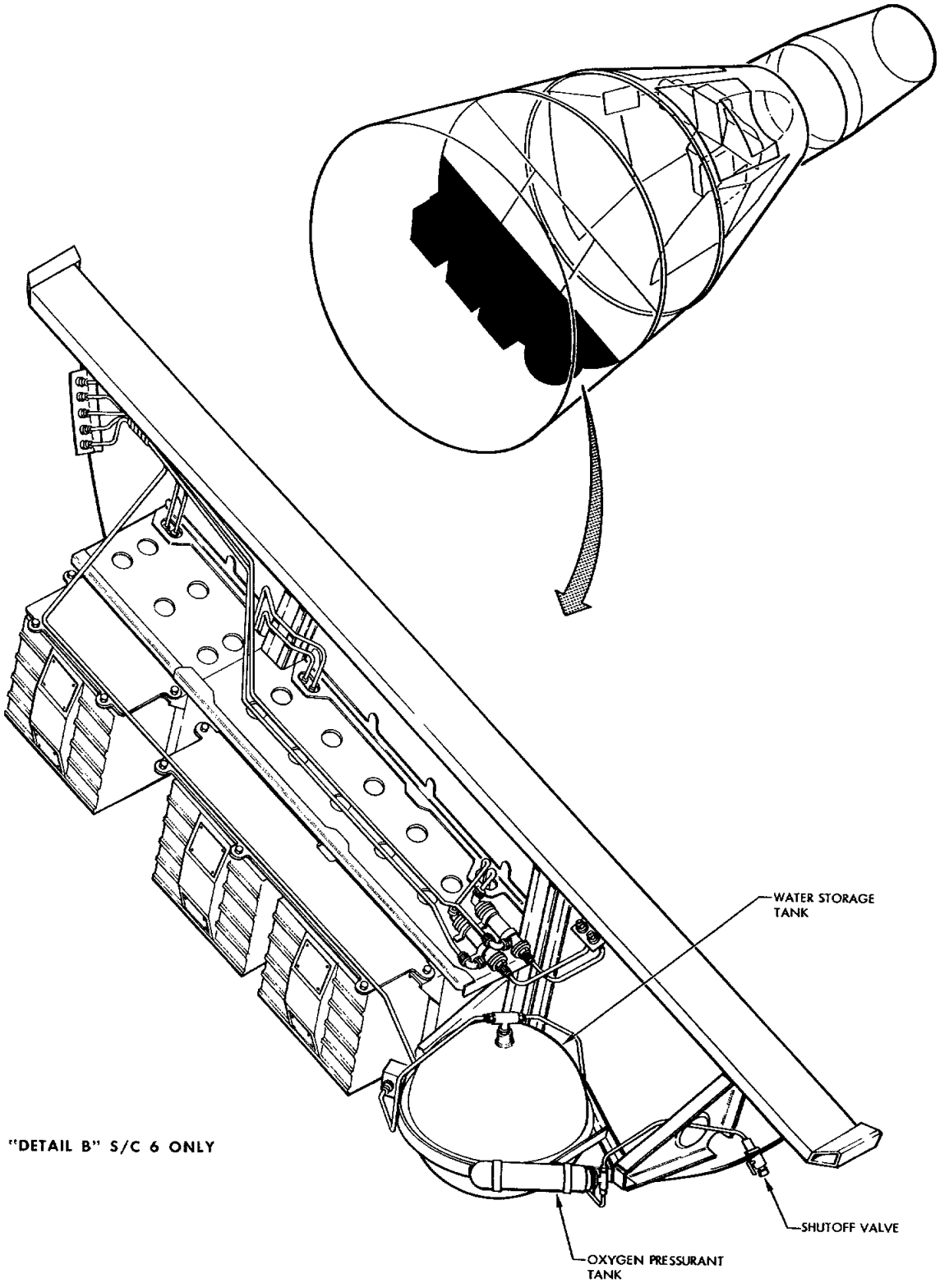


Figure 6-10 Water Management System (Sheet 3 of 7)

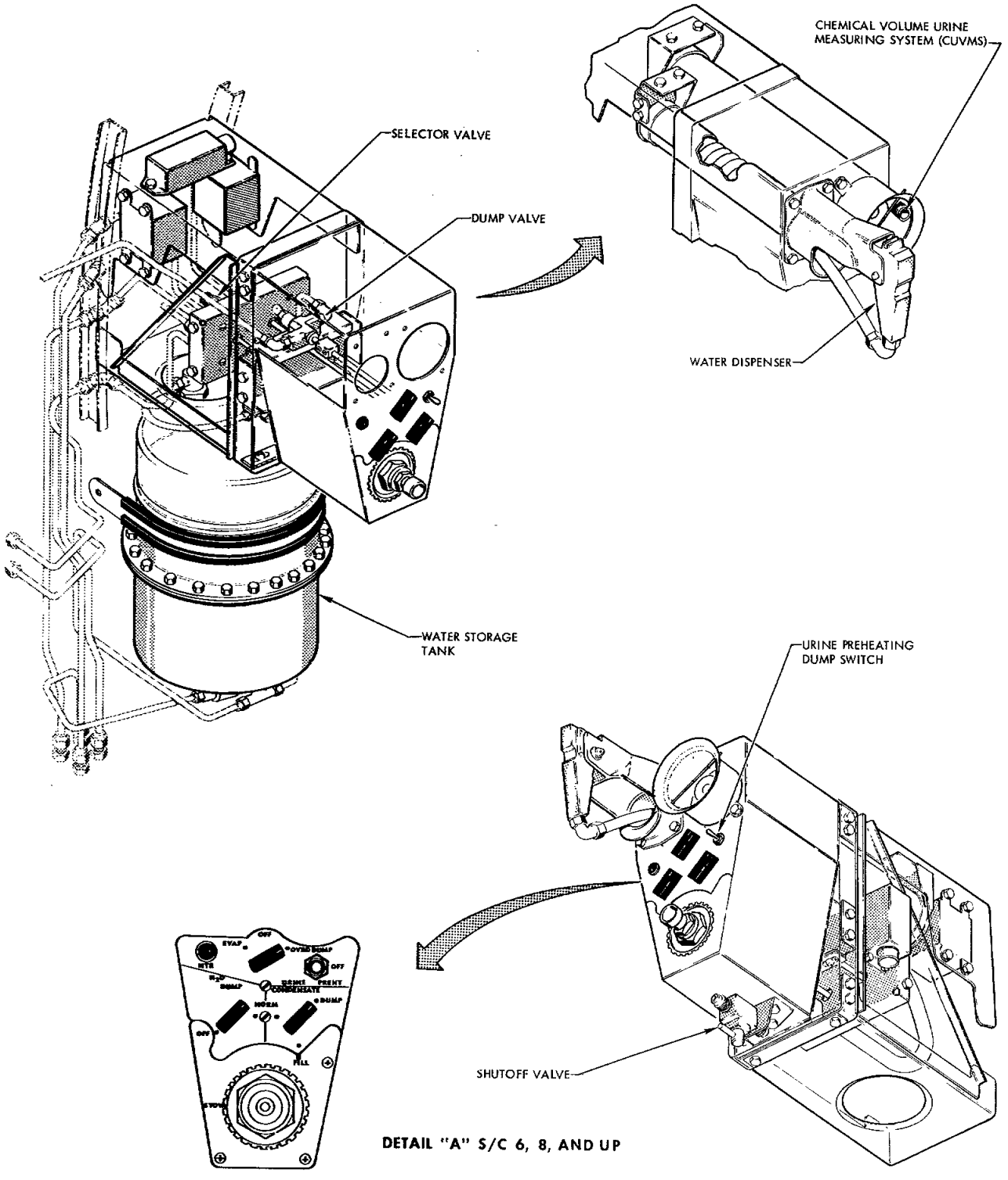


Figure 6-10 Water Management System (Sheet 4 of 7)

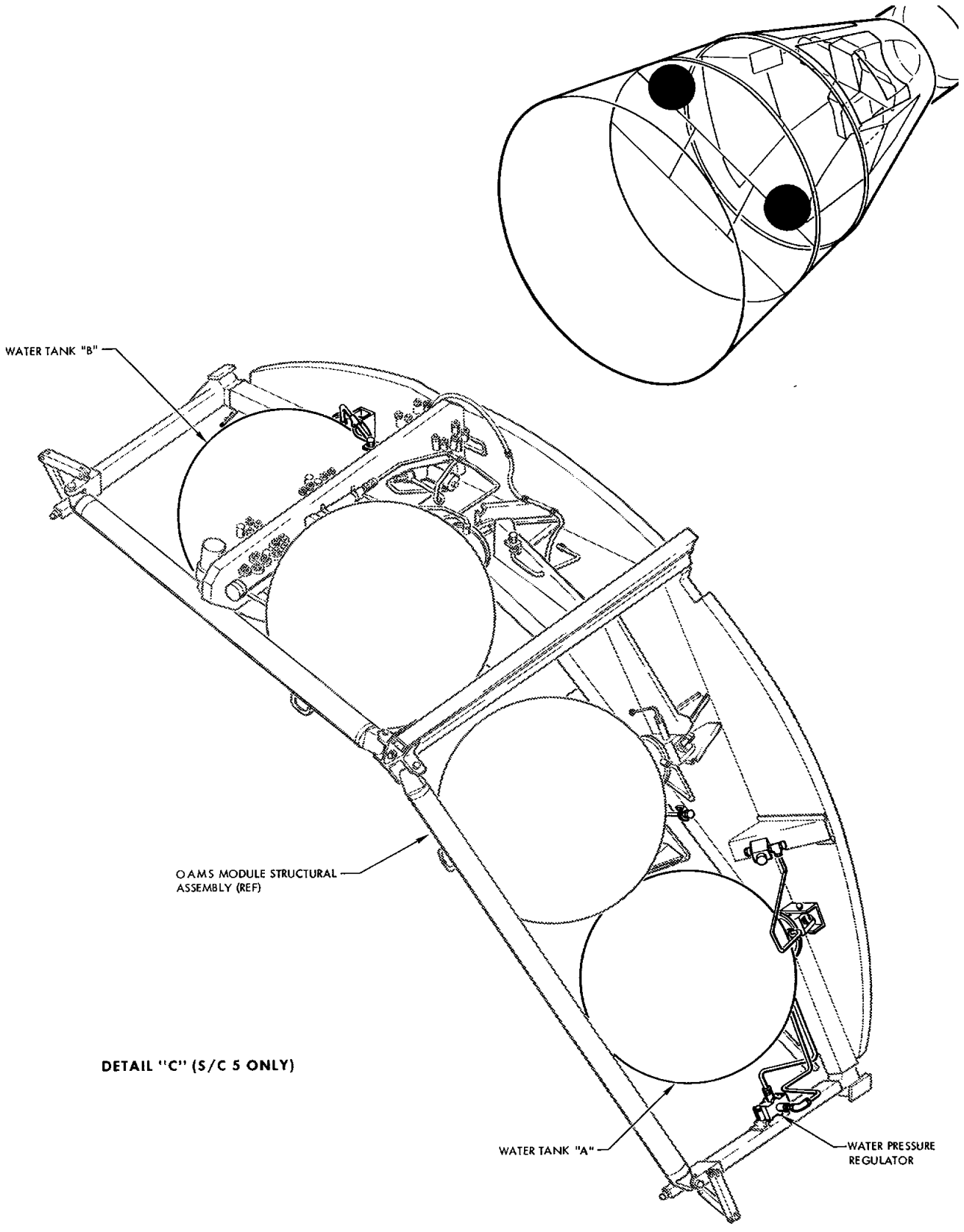


Figure 6-10 Water Management System (Sheet 5 of 7)

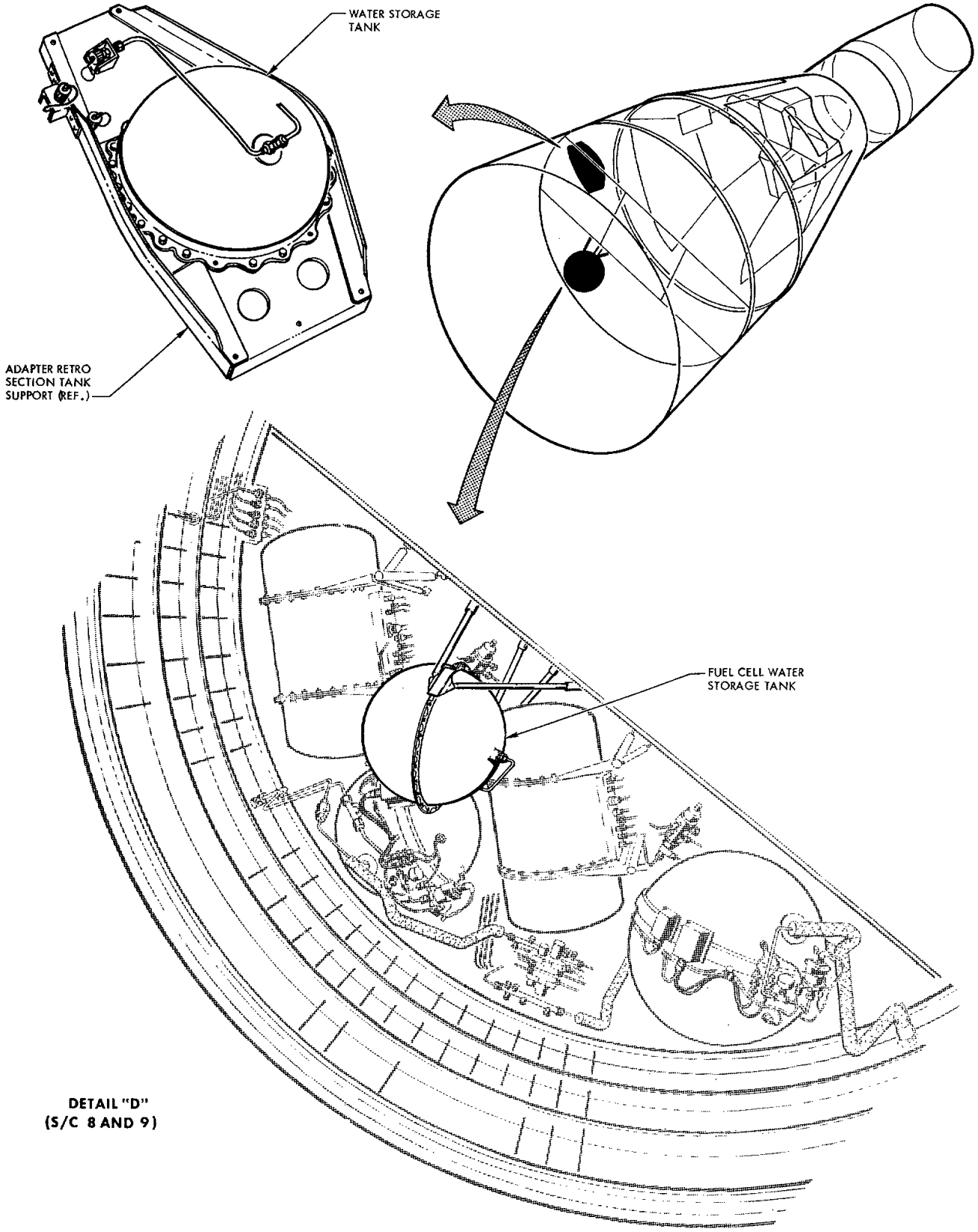


Figure 6-10 Water Management System (Sheet 6 of 7)

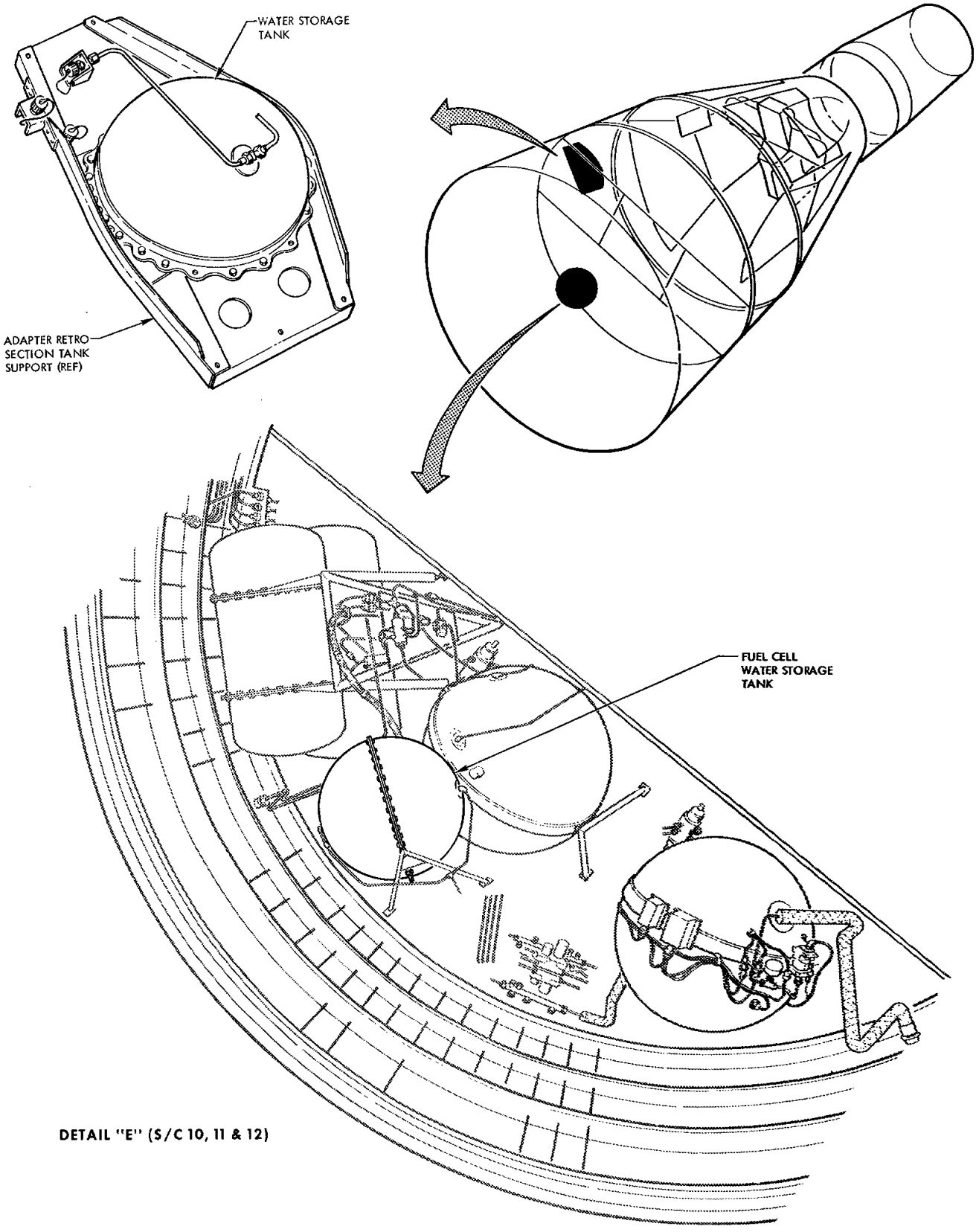


Figure 6-10 Water Management System (Sheet 7 of 7)

Components of the water management system, in addition to the water tanks, are a water control valve, condensate valve, water evaporator and two manual shutoff valves.

Urine System (Figure 6-11)

The urine system is designed for total management of the crewmens urinary output. It samples and determines total volume of every urination and provides for chemical analysis. Volume is determined by sampling every urination and using a tracer dilution technique. Three tenths (0.3) ml of tracer chemical is added to the urine and samples are taken. During postflight operations, the amount that the tracer has been diluted in each sample determines the quantity of urine voided by the crewmen. The urine system consists of the following components; a Chemical Urine Volume Measuring System (CUVMS) with selector valve, tracer storage accumulator and collection/mixing bag, a urine receiver assembly with collection bag, a urine quick-disconnect hose and urine solids trap filter, urine sampling bags and a roll-on cuff receiver assembly.

SYSTEM DISPLAYS AND CONTROLS

The displays and controls for the Environmental Control System are provided in the cabin and function as specified.

SECONDARY OXYGEN SHUTOFF Handle

A manual secondary oxygen shutoff handle is provided for each member of the flight crew for complete and positive shutoff of each secondary oxygen container. The handles are located aft of the right and left switch/circuit breaker panels. The position OPEN or CLOSED is noted.

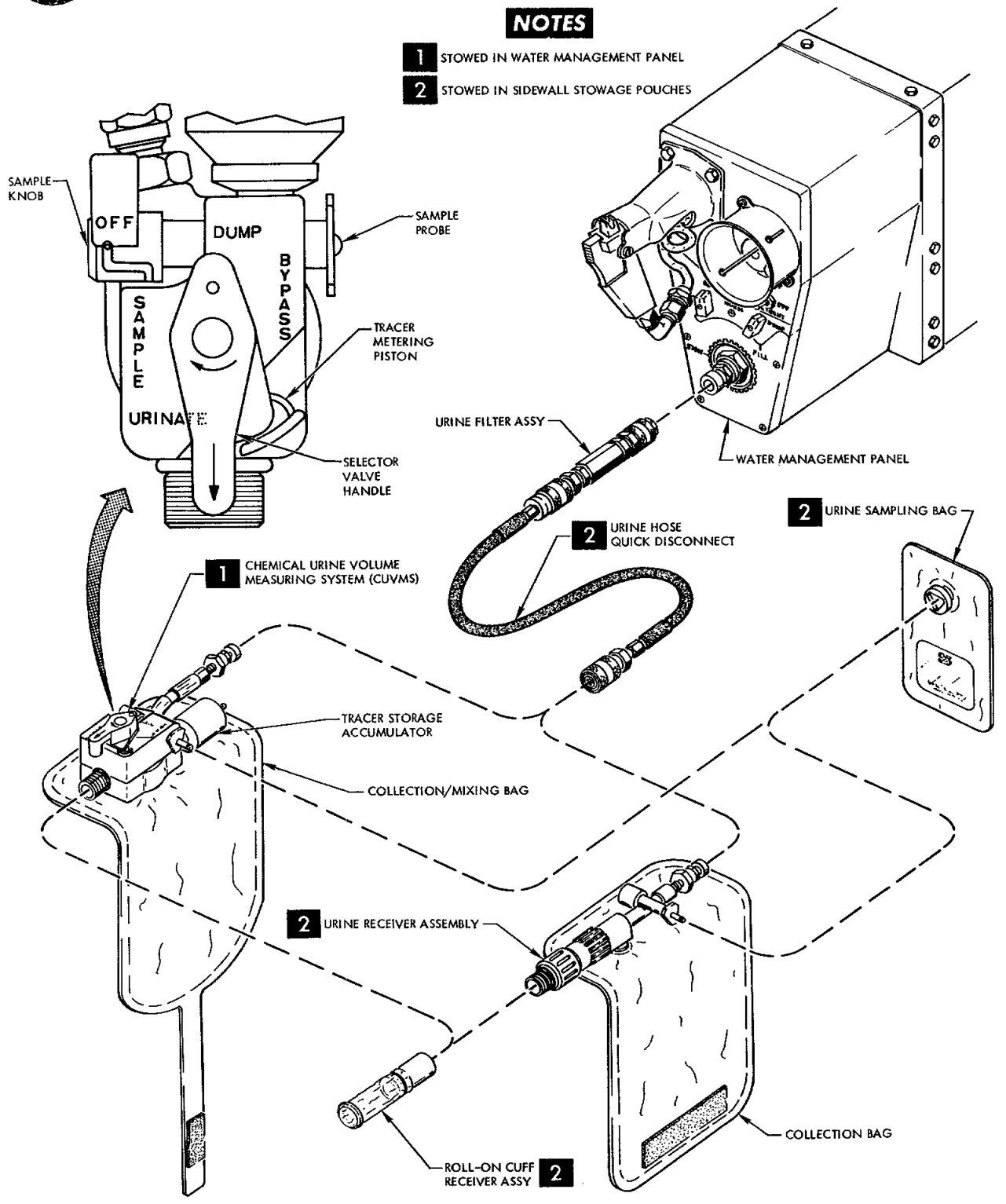


Figure 6-11 Urine System (Typical)

O_2 HIGH RATE Telelight

The O_2 HI RATE telelight located on the instrument panel illuminates when the high oxygen rate valve is opened manually or automatically.

The O_2 HI RATE switch on spacecraft 5 and 6 also activates the cabin fan. The switch has three positions; CABIN FAN, O_2 HI RATE, and OFF. Spacecraft 8 through 12 do not have the cabin fan, and that position of the switch is not used.

CBN AIR RECIRC Handle

The cabin air recirculation handle controls the recirculation valve which permits entry of cabin air into the suit circuit for removal of odors and carbon dioxide. This procedure will renovate cabin air without cabin decompression and reduces the possibility of carbon dioxide pockets by increasing circulation of the cabin atmosphere.

INLET SNORKEL Handle

This handle will control the cabin air inlet valve which provides for ventilation during landing and postlanding phases of the mission.

CABIN VENT Handle

This handle controls the operation of the cabin outflow valve to permit emergency decompression in orbit and cabin ventilation during the landing phase.

WATER SEAL Handle

This handle provides for watertight closure of the cabin pressure relief valve during a water landing.

O_2 HI RATE RECOCK Handle

This handle provides for the manual return of the oxygen high rate valve to the closed position, thereby restoring normal oxygen flow rate. Actuation of this handle also reestablishes the capability of initiating high rate oxygen flow when necessary.

CABIN And SUIT TEMP Indicator

A dual indicator provides for monitoring temperatures in the suit and cabin circuits. Range markings are calibrated in degrees Fahrenheit.

CABIN And P CO_2 PRESS Indicator

A dual indicator provides for monitoring cabin atmospheric pressure and the amount of carbon dioxide at the suit inlet. Cabin atmospheric pressure is calibrated in pounds per square inch. Carbon dioxide partial pressure is calibrated in millimeters of mercury.

SEC O_2 Indicator

A dual indicator is provided for monitoring pressure in the individual gaseous oxygen containers in the secondary oxygen subsystem. The indicator range is from 0 to 6000 psia, divided into 500-pound increments and numbered at each 1000-pound interval. Readings must be multiplied by 100 to obtain correct values.

ECS O_2 QUANT % and PSIA Meter

This indicator provides for monitoring quantity and pressure of cryogenic oxygen in the primary oxygen container. The quantity scale displays from 0 to 100 per cent in 2 per cent increments, numbered at 20 per cent intervals. The pressure scale ranges from 0 to 1000 psia in 20-pound increments, numbered at 200-pound intervals. Red undermarkings are incorporated on the oxygen meter to indicate the point at which thermal pressurization may be discontinued by de-energizing the heaters. Spacecraft 10 through 12 reidentifies the indicator as the CRYO meter, but it performs the same function as before.

Cryogenic Quantity Switch

This switch allows the same indicator to be used when monitoring the pressure and quantity of cryogen in any of the three cryogenic containers. The three containers are: the ECS primary oxygen supply, the RSS or fuel cell (FC) oxygen supply, and the RSS or FC hydrogen supply. The switch is located below the indicator on the center panel and has the following positions: ECS O_2, FC O_2, FC H_2 (PX35) and OFF. Spacecraft 10 through 12 have a three position switch. The positions are; O_2, H_2 (PX35) and OFF. The O_2 position allows the indicator to monitor the ECS-RSS oxygen supply and the H_2 (PX35) position allows monitoring the RSS or FC hydrogen supply.

ECS O_2 HEATER Switch

This switch is connected to the heaters in the ECS primary oxygen container. The switch has three positions; AUTO, OFF, and ON. It is located below the flight plan display on the center panel. Spacecraft 10 through 12 reidentifies the ECS O_2 HEATER switch as O_2 HEATER switch and connects to heaters in the

ECS-RSS oxygen container.

SUIT FAN Switch

The switch has three positions; NO 1, OFF, and NO 1 & 2. The switch is located in the upper left hand corner of the center panel. This allows suit fans number 1 and number 2 to operate together or independently. Suit fan NO 2 may be operated by placing switch in NO 1 & 2 position and placing suit fan NO 1 circuit breaker switch to OFF.

O_2 CROSS FEED Switch

The O_2 CROSS FEED Switch, spacecraft 6, 8 and 9 when in the OPEN position, permits oxygen from the primary oxygen supply module for the ECS to be used in the RSS in the event of RSS oxygen module failure. The reverse is also true.

Water Management Panel

A three knob panel is provided for managing, replenishing, and dumping waste water and urine overboard.

SUIT and CABIN TEMP Controls

Dual concentric knobs are mounted between the ejection seats for suit and cabin temperature control. These knobs control the operation of valves regulating the flow rate of primary and secondary coolant through the suit and cabin heat exchangers. Clockwise rotation results in increased temperatures.

MANUAL O_2 HIGH RATE Handle

This handle is located on the console between the members of the flight crew and provides for manual control of the dual high oxygen rate and suit system

shutoff valve. Actuation of the handle shall initiate the oxygen high flow rate and de-energize the suit compressor. Resumption of normal system operation shall be effected by actuation of the oxygen high rate recock handle.

SUIT FLOW Control Levers

An individual lever is provided for each member of the flight crew for regulation of circulatory oxygen flow through the suit circuits. The levers are located on the lower section of the pedestal and shall provide any selected flow valve setting from fully open to fully closed. A detent provides an intermediate position to prevent inadvertent shutoff of suit flow. This detent may be by-passed by moving the lever outboard.

Cabin REPRESS Control

A rotary handle control is provided for cabin repressurization after a decompression has occurred and for ELSS oxygen supply. The control rotates approximately 90° between fully OPEN (repressurize) and fully CLOSED (off) positions. This control is located on the center console between the suit flow control panels.

ECS HTR Telelight

This telelight, located on the annunciator panel of the center instrument panel, illuminates when the heater in the primary oxygen container has been manually activated.

SYSTEMS OPERATION

The Environmental Control System (Figure 6-1, 6-2) is semi-automatic in operation and provides positive control in all modes of operation. There are six

operational modes:

1. Pre-Launch
2. Launch
3. Orbit
4. Re-Entry
5. Postlanding
6. Emergency

Prior to the pre-launch mode, it is necessary to service and to check the system functionally.

SERVICE AND CHECKOUT

For this operation, it is assumed that the spacecraft has been mated with the booster on the launch pad and in the unserviced condition. The primary, secondary and egress oxygen storage tanks are filled. The water boiler and drinking water supply tank are supplied with water. The cartridge in the suit loop cannister is then replaced.

PRE-LAUNCH

The pre-launch phase is defined as the period after the servicing has been completed and prior to launch.

Suit Loop

The pilots in their suits, with face plates open, are connected to the suit

circuit. The suit circuit compressor is actuated and the suit temperature control valve is adjusted to satisfy the pilot desiring the cooler temperature. The other pilot becomes comfortable by adjusting his suit flow rate control valve toward the closed position to obtain a warmer setting. A ground supply of pure oxygen is connected to the pressure suit circuit purge fitting. Flow is initiated with the face plates closed. The suit circuit gas is sampled periodically until an acceptable oxygen content is attained. A suit circuit leakage test is conducted. After satisfactorily completing the suit circuit leakage test, the primary and secondary oxygen manual shutoff valves are opened and the suit circuit purge system is disconnected and removed.

Cabin Loop

The cabin hatches are closed. A ground supply of pure oxygen is connected to the cabin purge fitting, flow is initiated and the cabin is purged. The cabin fan is actuated and the recirculation valve is opened. A cabin leakage test is conducted. After satisfactorily completing the cabin purge and leakage test, the cabin purge system is disconnected and removed and the cabin purge fitting is capped.

Oxygen Loop (Figure 6-3, 6-4)

The primary and secondary oxygen manual shutoff valves are opened.

The liquid oxygen inside the primary supercritical container has been changing from a liquid to a supercritical fluid by thermal leakage and heater activation.

A pressure control switch provides for automatic or manual activation of these heaters. The manual control switch is located on the center control panel.

An indicator also on the center control panel indicates both pressure and quantity from a transducer and control unit that are attached to the container.

The oxygen gas flows from the container and is warmed to approximately 50°F in a heat exchanger. This heat exchanger also contains a relief valve that limits maximum pressure to 1000 psig. This valve opens, permitting full flow and reseats within the range of 945-1000 psig.

A discharge temperature sensor provides an indication, for telemetering only, of the temperature in the primary oxygen line downstream of the heat exchanger.

The oxygen gas is regulated from 1000 psia maximum to 110 $\pm$ 10 psig. Flow capacity of 0.35 lb/min with an inlet pressure from 800 to 1000 psia and an inlet temperature of 60°F. This regulator also contains a relief feature that limits downstream pressure to 215 psig in the event of a failed-open condition.

A 10-micron filter provides filtration of the primary oxygen supply before it enters the suit or cabin loop.

LAUNCH

Cabin Loop

The cabin pressure relief valve opens to limit the pressure differential between cabin and ambient to 5.5 $^{+.0}_{-.0}$ psi.

Suit Loop (Figure 6-8)

Oxygen is supplied to the suit loop through the suit pressure regulator. The suit pressure is controlled to between 2 and 9 inches of water above cabin pressure by the suit pressure regulator.

Suit circuit oxygen from the suit circuit demand regulator recirculates through the suit compressor, the carbon dioxide and odor absorber, the suit heat exchanger and water separator, the pressure suits, and the suit circuit solids traps. There are two compressors in the circuit. One is an alternate to be used if a compressor failure occurs. The alternate compressor is activated by positioning the SUIT FAN switch on the center panel. The cartridge of lithium hydroxide and activated charcoal remove carbon dioxide and odors of an organic nature that could have any ill effects on the pilots. As suit circuit oxygen flows through the suit heat exchanger, the temperature is controlled as selected by the pilots.

Solid traps, located in the oxygen outlet ducts of both pilots' suits, remove particulate solids, preventing contamination of the suit circuit system. An integral by-pass opens if the traps become choked with collected solids permitting continuous oxygen flow through the suit circuit.

ORBIT

Cabin Loop

Normal cabin leakage allows the cabin pressure to decay to a nominal value of 5.1 psia. The cabin pressure control valve maintains this value automatically.

A dual cabin pressure regulator supplies makeup oxygen through the pilots' pressure suits to the cabin on demand, as sensed by two aneroid elements within the regulator. The regulator supplies the makeup oxygen at a controlled pressure between 5.0 to 5.3 psia.

The cabin fan on spacecraft 5 and 6 circulates cabin air through the cabin heat exchanger. The cabin fan has been removed on spacecraft 8 through 12. One or both of the pilots may open their faceplates. The cabin air circulating valve is in the open position to provide for recirculation of the cabin oxygen through the suit circuit.

In the event of spacecraft depressurization, whether intentionally or by spacecraft puncture, the dual cabin pressure regulator closes when cabin pressure decreases to $4.1 \, ^{+0.2}_{-0.1}$ psia, preventing excessive loss of oxygen.

Suit Loop (Figure 6-8, 6-9)

The suit circuit demand regulator senses cabin pressure and maintains suit circuit pressure at 2.5 to 3.5 inches of water below to 2 to 9 inches of water above cabin pressure. Should cabin pressure decrease below 3.5 psia, the suit circuit demand regulators maintain the suit circuit pressure at $3.5 \, ^{+0.4}_{-0.0}$ psia by constant bleed orifices and sensing aneroids within the regulator. When cabin pressure is restored to $5.1 \, ^{+0.2}_{-0.1}$ psia, the suit circuit demand regulators return to normal operation.

In the event of cabin and suit circuit malfunction, the suit circuit will automatically revert to the high rate of operation when suit circuit pressure

decreases below 3.0 $^{+0.1}_{-0.0}$ psia. A suit circuit pressure sensing switch energizes the solenoid of the dual high flow rate and system shutoff valve. This initiates a high oxygen flow rate of 0.08 $\pm$0.008 lb/min per man (total flow: 0.16 lb/min). This high flow rate flows directly into the suits by-passing the suit demand regulators. The suit recirculating system is shut off and the suit compressors are de-activated when the solenoid of the dual high flow rate and system shutoff valve has been energized. The O_2 HI RATE light on the center panel illuminates when the suit circuit is on the high flow rate. There is also a manual control for the high flow rate and system shutoff valve located on the center console.

When the suit circuit pressure is restored to a level above 3.0 $^{+0.1}_{-0.0}$ psia, the high rate and system shutoff valve is reset manually by using the control marked O_2 HIGH RATE RECOCK located on the center panel. This returns the suit circuit to normal operation by opening the system shutoff valve and closing the high rate valve. The suit compressor is also reactivated.

Water Management System (Figure 6-12)

The drinking water system is pressurized and manually controlled by the pilots.

Water from the adapter supply is used to replenish the cabin tank water supply.

The water tank drink selector valve is set in the NORM position.

The pilots manually operate the drinking dispenser to provide drinking water from the cabin storage tank.

The water separator removes metabolic moisture through a wicking material positioned between the plates of the suit heat exchanger.

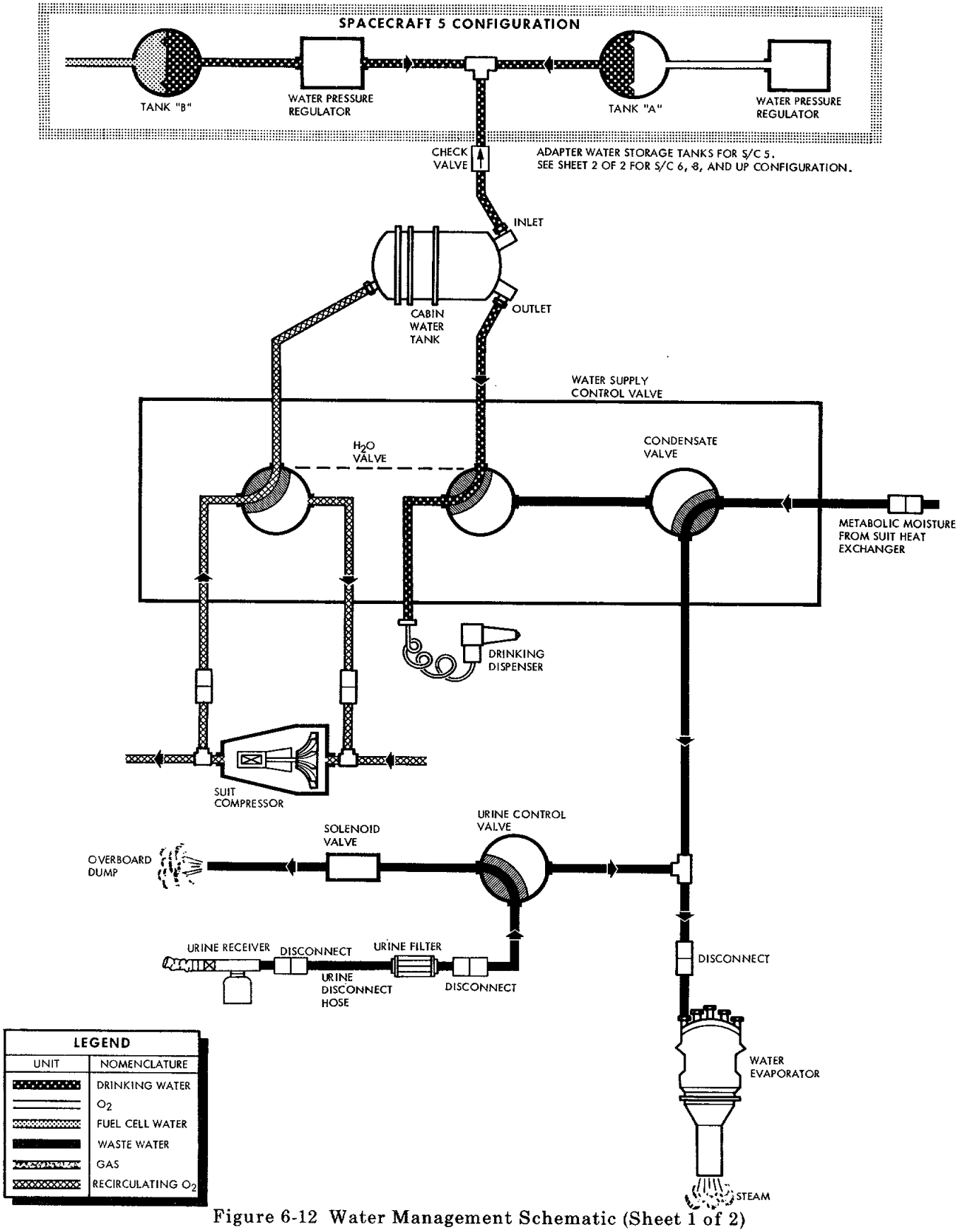
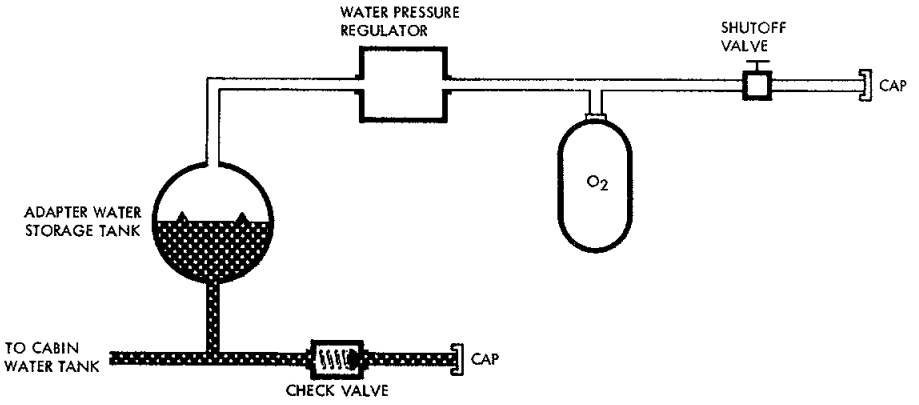
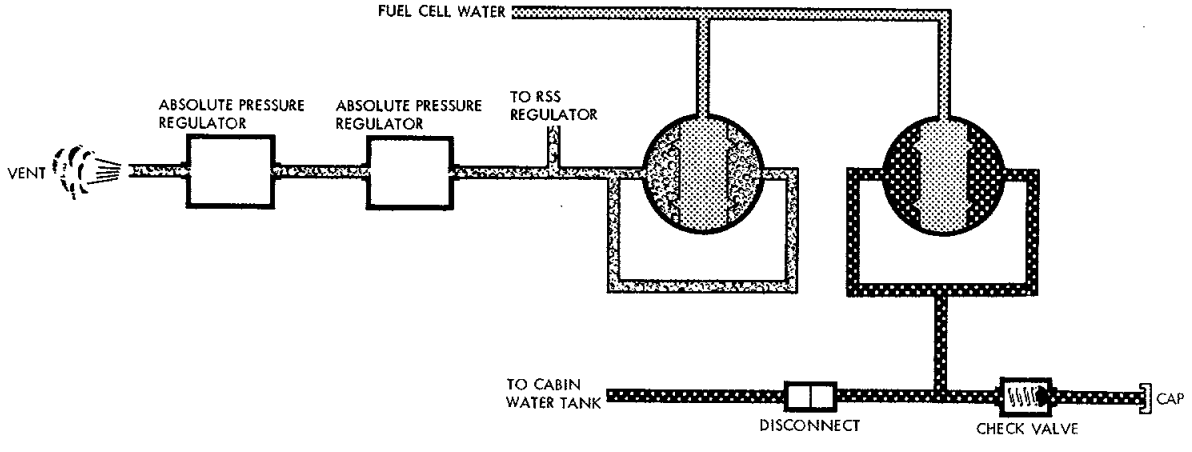


Figure 6-12 Water Management Schematic (Sheet 1 of 2)



SPACECRAFT 6 CONFIGURATION



SPACECRAFT 8 AND UP CONFIGURATION

Figure 6-12 Water Management Schematic (Sheet 2 of 2)

Urine Disposal System (Figure 6-11)

The Chemical Urine Volume Measuring System (CUVMS) has a four position selector valve labeled URINATE, SAMPLE, DUMP and BY-PASS. Position selection is made by rotating the selector valve which is attached to a multi-ported center plug. The selector valve includes a positive displacement tracer metering pump which is activated by the handle as it passes over a plunger between the BY-PASS and URINATE positions. This supplies a quantity of tracer solution to the passages to mix with the urine in the collection/missing bag. Therefore, volume measurement can only be accomplished when the selector is in the URINATE position and the tracer chemical is added. The urine receiver assembly and collection bag provides for collecting and sampling urine or overboard dump provisions but does not provide for volume measurement. The urine quick-disconnect hose and filter assembly consists of a section of flexible hose with quick disconnect couplers on each end. This assembly connects the CUVMS or the urine receiver assembly to the water management panel through an in-line urine solids trap filter for dumping urine overboard. The urine sampling bags are plastic laminate with a valve that connects to the sampler port on the CUVMS or the urine receiver assembly for taking urine samples. The urine receiver roll-on cuff is the interface between the crewmen and the CUVMS or the urine receiver assembly. It provides an air and liquid tight seal for direct urine transfer. Selector valve positions on the CUVMS direct urine flow as follows. The SAMPLE position directs urine and tracer chemical mixture from collection/mixing bag to the sample port and into the sampling bag. The DUMP position directs the urine/tracer chemical mixture overboard. The BY-PASS position (normal purge position) directs the urine flow to the water management panel for dumping overboard.

The dump selector valve on the water management panel is positioned to route the urine either to the water boiler or dumped overboard. The normal procedure is to dump. Before it is dumped the urine dump system is preheated by positioning its heater switch located on the water management panel. A urine dump heater light is also provided and located on the water management panel. This light illuminates when the heater is activated.

RE-ENTRY

Oxygen System

The primary oxygen system is disconnected when the adapter section is separated from the re-entry module. This removes the primary oxygen supply pressure which automatically activates the secondary oxygen supply.

The system shutoff and high rate valve is positioned to the high rate position before the adapter section is jettisoned.

Cabin Loop

The pressure in the suit and cabin remains constant at 5 psia (nominal) until an altitude of approximately 27,000 feet is reached.

As ambient pressure increases during descent, the cabin pressure relief valve admits ambient air into the cabin, preventing high differential pressures. The

cabin pressure relief valve begins to open when the ambient pressure is 15.0 inches of water greater than cabin pressure and opens to maximum flow when the pressure differential is 20 inches of water.

At an altitude of 25,600 feet, or below, the pilots manually open the cabin inflow and outflow valves to circulate external air through the cabin and suit circuit.

Maximum negative pressure on the cabin should not exceed 2 psi as controlled by the cabin relief valve.

Suit Loop

Prior to re-entry the face plates should be closed. The high flow rate of oxygen is flowing directly into the suit circuit.

When the cabin inflow valve is opened it activates the suit compressor and external air is circulated through the suit circuit.

POSTLANDING

Ventilation is provided by the suit compressor as long as electrical power is available (12 hours minimum).

Ambient air is drawn into the vehicle through the snorkel inflow valve, by the suit compressor, circulated through the suit circuit into the cabin, then discharged overboard through the outflow vent valve.

The snorkel inlet valve functions as a water check valve. When the snorkel inlet valve is above water level, the ball check is retained freely in a wire mesh cage, permitting ambient air to enter the suit circuit. Normal oscillations of the spacecraft in the sea may result in the snorkel valve being momentarily submerged. This will cause the ball check to seat and is held there by suction from the suit compressor. Opening the cabin air circulating valve allows the ball to drop from its seat.

To prevent water from entering the cabin through the cabin pressure relief valve, the manual shutoff section of the valve is closed.

EMERGENCY

Cabin Loop

If cabin depressurization becomes necessary due to toxic contaminants or fire, the cabin outflow valve is opened to depressurize the cabin. The cabin regulator will close, stopping the oxygen supply to the cabin, permitting the escape of toxic contaminants and preventing oxygen assistance to combustion in the event of fire. The cabin repressurization valve permits repressurization of the spacecraft cabin.

The control knob for the cabin repressurization valve is located on the lower console and is rotated counterclockwise to open the valve. It is rotated

clockwise to close the valve when cabin pressure is between 4.3 and 5.3 psia. Cabin pressure is then automatically controlled at 5.1 $^{+0.2}_{-0.1}$ psia by cabin pressure regulator valve.

Egress Oxygen (Figure 6-13)

This system is installed on spacecraft 5 and 6 only. Operation of the egress oxygen system is initiated by three of the four lanyards which are pulled when the seat leaves the spacecraft. One lanyard pulls a pin in the composite disconnect allowing it to separate and close the normal suit circuit. Two of the remaining lanyards open the container shutoff valve and circuit relief valve activating the egress oxygen system.

Each of the egress oxygen containers is pressurized to 1800 psig with gaseous oxygen. The oxygen flows from the containers through a pressure regulator, where the pressure is reduced to 40 psia. It then flows through a shutoff valve and a flow restrictor, which allows a flow of 0.052 to 0.063 lb/min, then through a check valve to the suit. After leaving the suit, oxygen flows through the shutoff and relief valve, which dumps the oxygen overboard, as well as controls the suit pressure to 3.5 $^{+0.6}_{-0.0}$ psia if ejection occurs at an altitude above 31,500 feet, and 2 to 8.23 inches of water above ambient at an altitude below 31,500 feet.

SYSTEM UNITS

DUAL SECONDARY OXYGEN RATE AND SUIT SYSTEM SHUTOFF VALVE (Figure 6-14)

The dual secondary oxygen rate and suit system shutoff valve provides a constant flow rate of oxygen directly to the pilot's suit during re-entry or in the event the suit circuit malfunctions during launch or orbit.

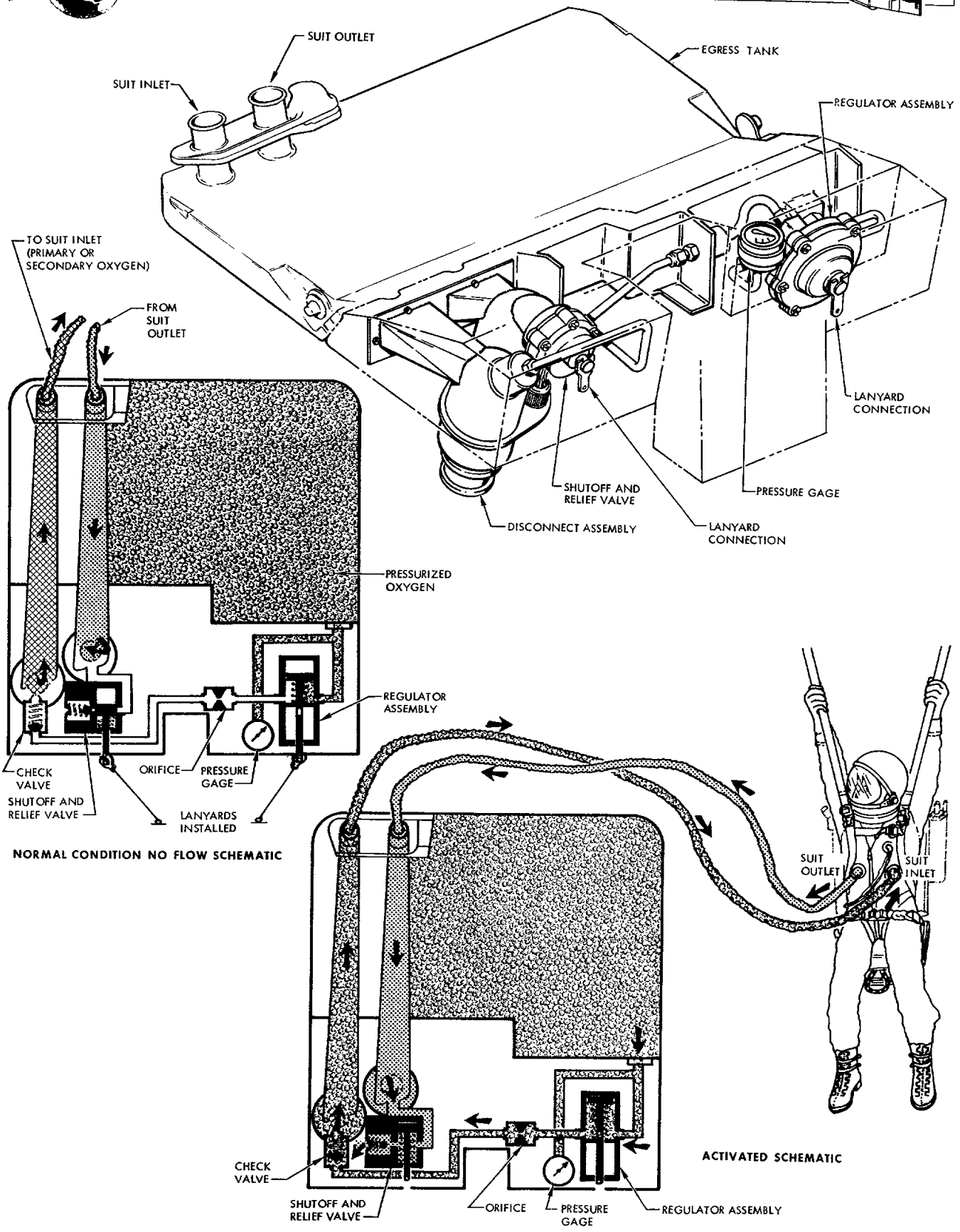
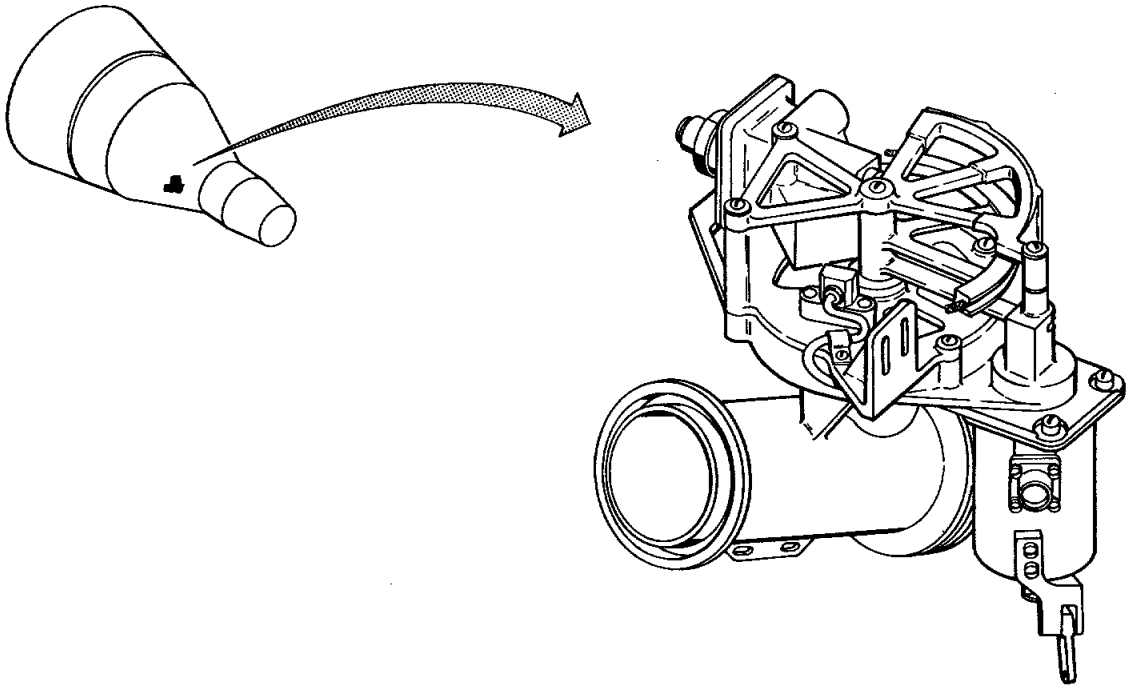


Figure 6-13 Egress Oxygen Flow Diagram (S/C 5 & 6)



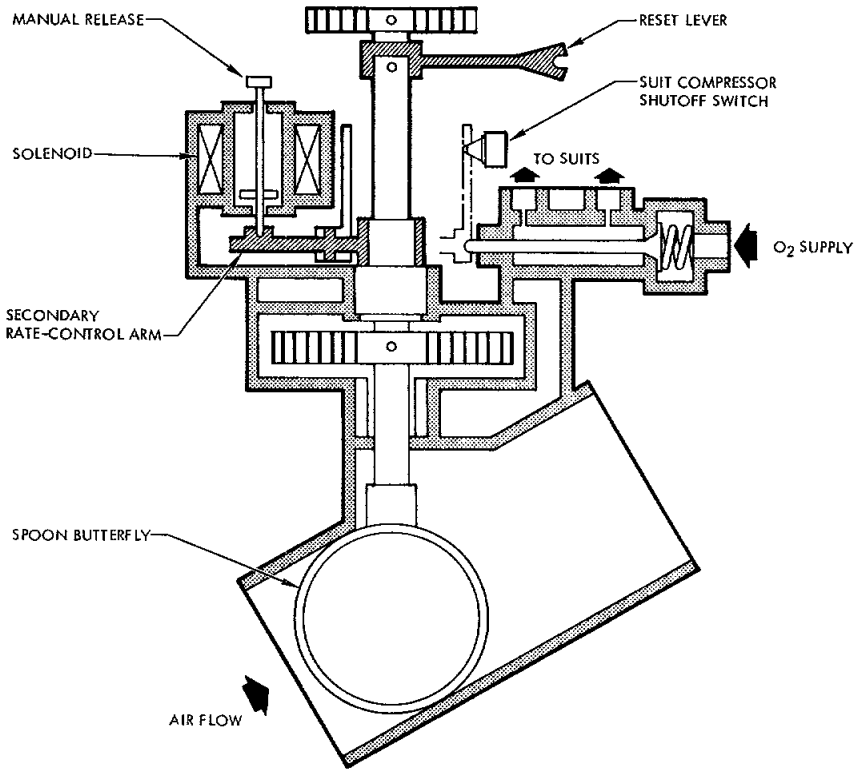


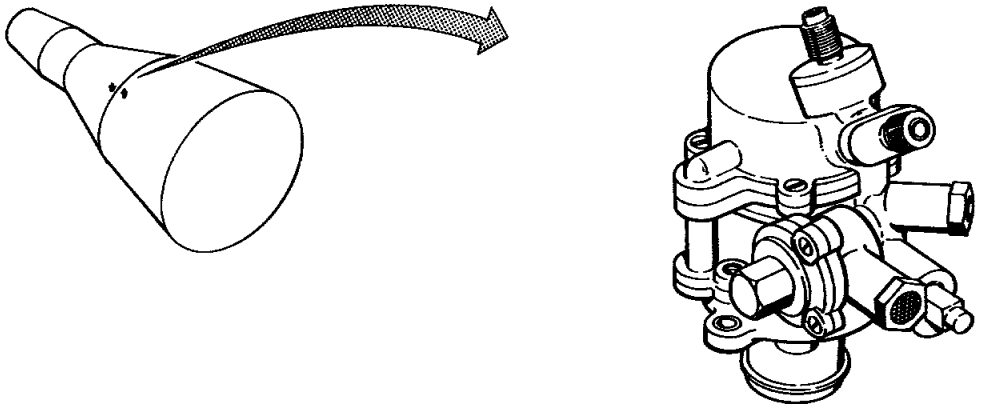
Figure 6-14 Dual Secondary Oxygen Rate and Shutoff Valve

The valve is designed for manual and automatic initiation. The recirculating suit oxygen circuit flows through the shutoff section of the valve, which is manually opened and is spring loaded to the closed position. The shutoff valve is held open by a 24 vdc solenoid pin, as long as the solenoid is de-energized.

The secondary flow poppet valve, held closed by spring tension, remains closed whenever the shutoff valve is in the open position. When the solenoid is energized, the butterfly arm is released and rotates by spring tension, closing the suit circuit valve and mechanically opening the secondary oxygen flow rate poppet valves. Opening the poppet valves allows oxygen to flow to each pilot's suit through fixed orifices at a rate of 0.08 $\pm$ 0.008 lb/min per man (total flow 0.16 lb/min.). The butterfly arm simultaneously actuates a switch that de-energizes the solenoid, turns off the suit compressor and cabin fan, and illuminates a SECONDARY FLOW RATE lamp on the pilots' center display panel. A pressure sensor switch attached to each pilot's suit circuit will energize the solenoid if the suit circuit pressure in either suit decreases below 3.0 $^{+0.1}_{-0.0}$ psia, automatically shutting off the suit circuit flow and initiating the secondary flow rate. A manual control is provided for resetting the valve to the normal position. The secondary flow rate is used during re-entry. Prior to retro-grade the pilots manually disengage the solenoid initiating the secondary flow rate.

SUIT OXYGEN DEMAND REGULATOR (Figure 6-15)

The suit oxygen demand regulator controls the oxygen to the suit circuit from the primary or secondary oxygen system and replenishes oxygen used by the pilots or lost by leakage.



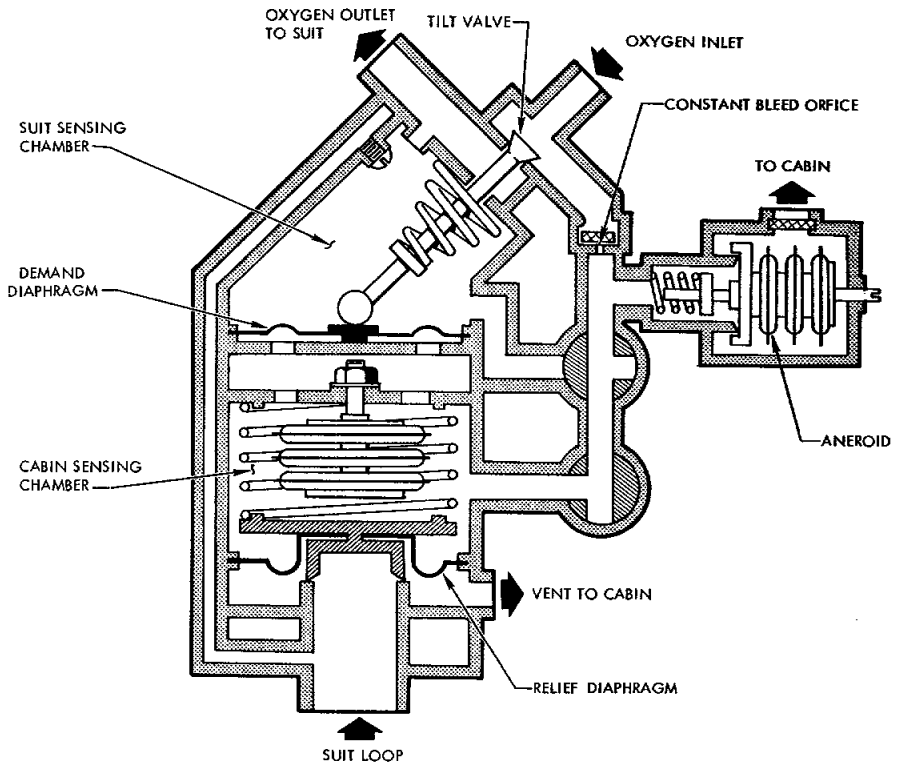


Figure 6-15 Suit Oxygen Demand Regulator

Cabin pressure is sensed on one side of the diaphragm and suit pressure is sensed on the opposite side of the diaphragm. The differential pressure across this diaphragm opens or closes a poppet valve admitting or stopping oxygen flow into the suit circuit. With cabin pressure of 5.0 psia the suit regulator maintains suit pressure at 2.5 to 3.5 inches of water below cabin pressure.

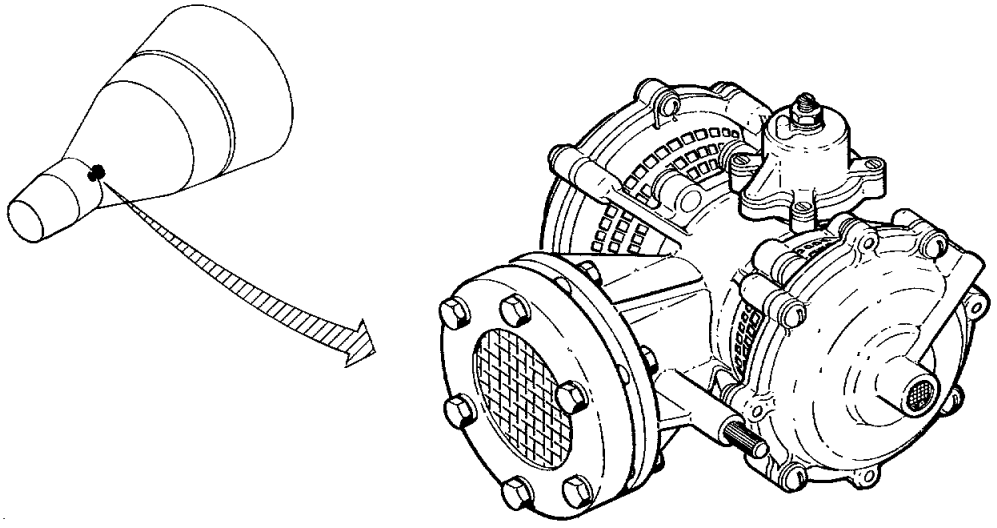
A resilient diaphragm type valve relieves pressure in the suit during ascent and limits excess pressure to between 2.0 and 9.0 inches of water above cabin pressure. During descent, the suit demand regulator relieves the secondary oxygen rate flow through the relief portion of the valve, maintaining suit pressure 2 to 9 inches of water above cabin pressure.

A constant bleed and aneroid elements maintain the suit pressure at $3.5\ ^{+0.4}_{-0.0}$ psia if cabin pressure decreases below this pressure. The bleed flow by-passes the tilt valve through a bleed orifice and is directed to the cabin pressure sensing side of the pressure demand diaphragm. A metering valve, controlled by an aneroid, regulates the reference pressure on the demand diaphragm. The regulator returns to normal operation when cabin pressure returns to $5.1\ ^{+0.2}_{-0.1}$ psia. In the event that cabin decompression and a ruptured relief diaphragm in the regulator occur simultaneously, an aneroid over the relief diaphragm extends to control suit pressure at 3.9 psia maximum.

CABIN PRESSURE RELIEF VALVE (Figure 6-16)

The cabin pressure relief valve automatically controls the cabin-to-ambient differential pressure during launch, orbit and re-entry. Duplicate spring loaded poppet valves are controlled by servo elements within the valve.

The servo elements control spring loaded metering valves which determine the



CABIN PRESSURE RELIEF VALVE

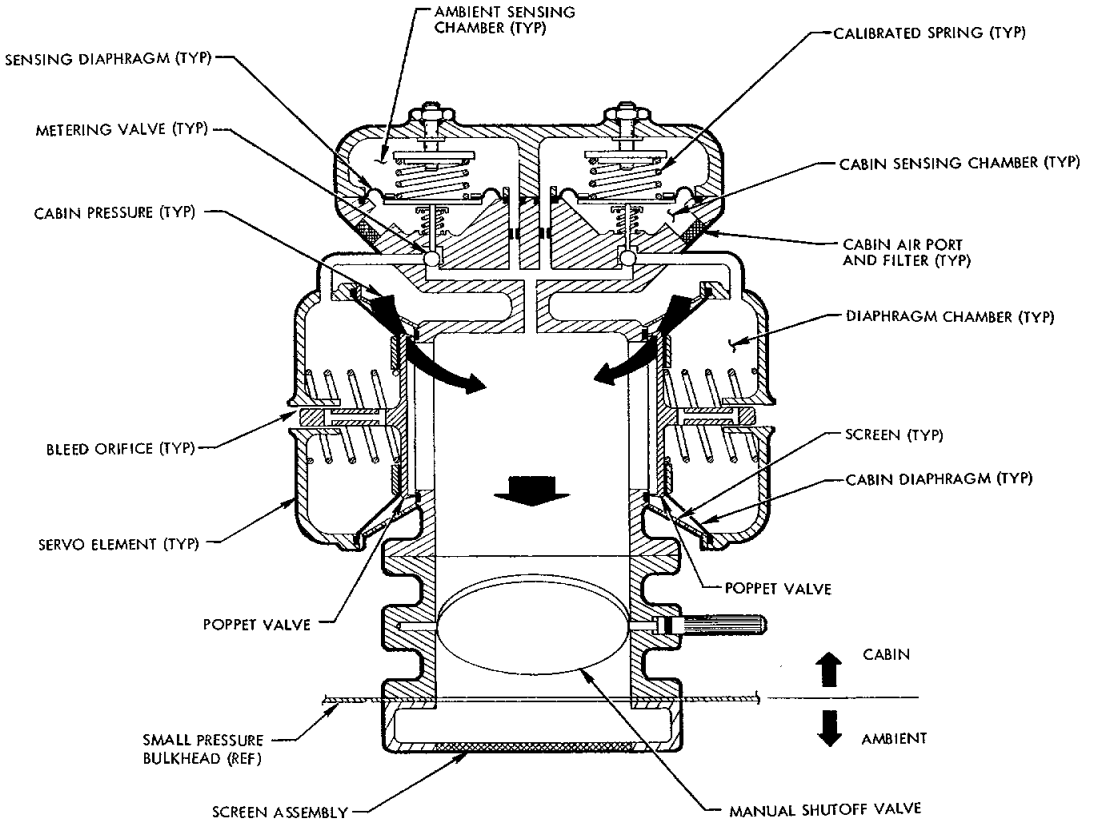


Figure 6-16 Cabin Pressure Relief Valve

pressure within the diaphragm chamber behind the poppet, controlling the poppet position. A small inlet bleed orifice admits cabin pressure to the diaphragm chamber. When the poppet opens, a large orifice permits rapid change in pressure ensuring quick closure of the poppet.

During ascent the valve will relieve cabin pressure as ambient pressure decreases until cabin differential pressure is 5.5 to 6.0 psia. The valve closes maintaining differential pressure in this range. When cabin pressure decreases below 5.5 psia the servo element closes the metering valve maintaining reference pressure within the diaphragm chamber at cabin pressure. The poppet is held closed by spring force and the zero differential between the diaphragm and the cabin prevents cabin pressure release. If cabin differential pressure exceeds 5.5 psia the zero element retracts, opening the metering valves, allowing the diaphragm chamber to discharge to ambient. The discharge port being larger than the inlet bleed orifice permits the diaphragm chamber to approach external pressure. The cabin pressure reacting on the diaphragm overrides the poppet spring force, which opens permitting cabin pressure relief to ambient. During descent, as external pressure increases, ambient air is admitted to the cabin by the valve to reduce the differential pressure. As external pressure increases above the cabin pressure the metering valves are held on their seats, preventing external pressure from entering the diaphragm chamber and retaining cabin pressure in the chamber. The poppet valve senses diaphragm chamber pressure versus ambient pressure. When the ambient pressure is 15 inches of water greater than cabin pressure the poppet begins to open permitting ambient air to enter the cabin. The poppet opens fully when the differential pressure is 20 inches of water.

To preclude water entering the cabin during postlanding, a manual shutoff valve

is provided.

SUIT CIRCUIT COMPRESSOR (Figure 6-17)

Two electric motor driven, single stage compressors are incorporated in the suit circuit. One compressor is utilized for circulation of the gases within the suit circuit, supplying both suits. The other compressor functions as a backup and is activated only by manual selection by the pilots. Either compressor can be manually selected by a switch on the center display panel, and both compressors can be selected simultaneously.

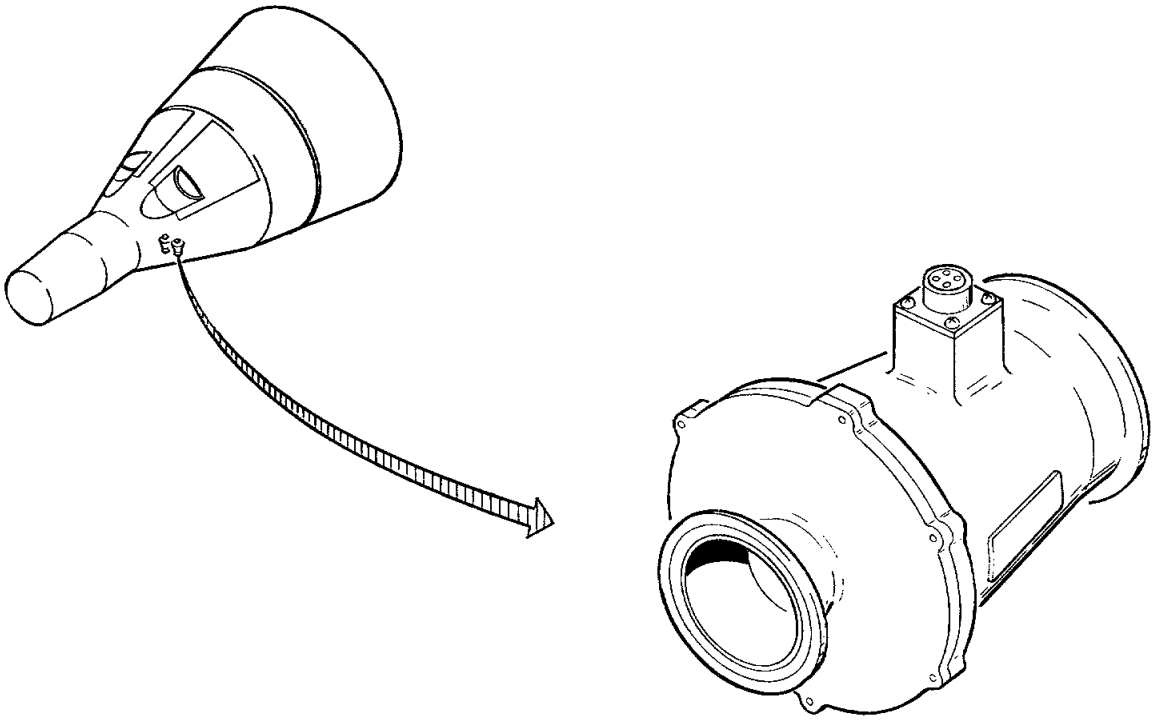
When secondary oxygen flow rate is selected, the compressor is automatically de-energized. Re-entry is made using the secondary rate. At an altitude of 25,600 feet or below the manual inflow valve is opened which re-energizes the compressor. The suit compressor provides ventilation during landing and for a twelve hour postlanding period, or until the batteries fail.

SOLIDS TRAP (Figure 6-18)

A solids trap is located in the oxygen outlet duct of each suit. A cylindrical 40 micron filter strains the gaseous flow in the suit circuit removing the solid matter. In the event that the trap becomes choaked with collected solids, an integral by-pass opens when the differential pressure across the screen exceeds 0.50 inches of water.

DUAL CABIN PRESSURE REGULATOR (Figure 6-19)

The cabin pressure regulator maintains cabin pressurization by providing makeup oxygen to the cabin on demand. The regulator contains two aneroid elements which individually sense cabin pressure. When cabin pressure decreases, the aneroids



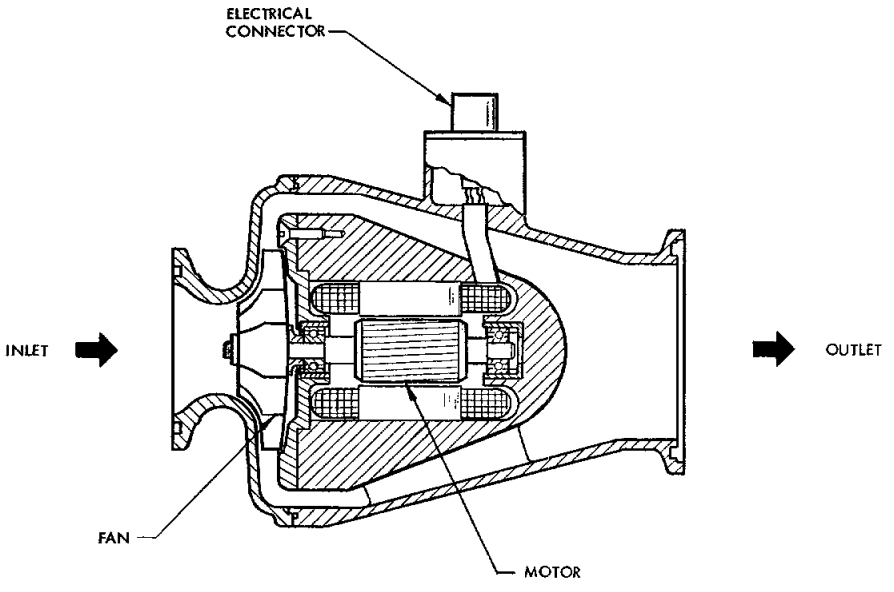
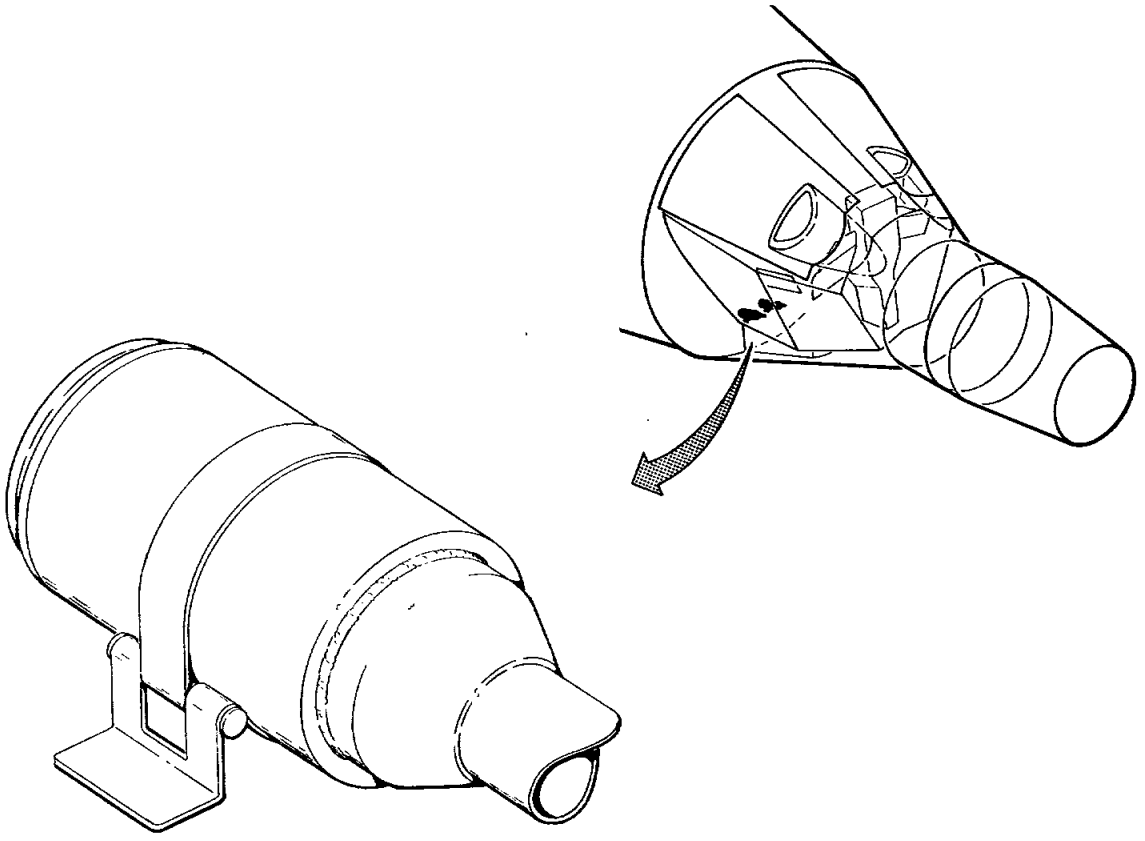


Figure 6-17 Suit Circuit Compressor



SOLIDS TRAP

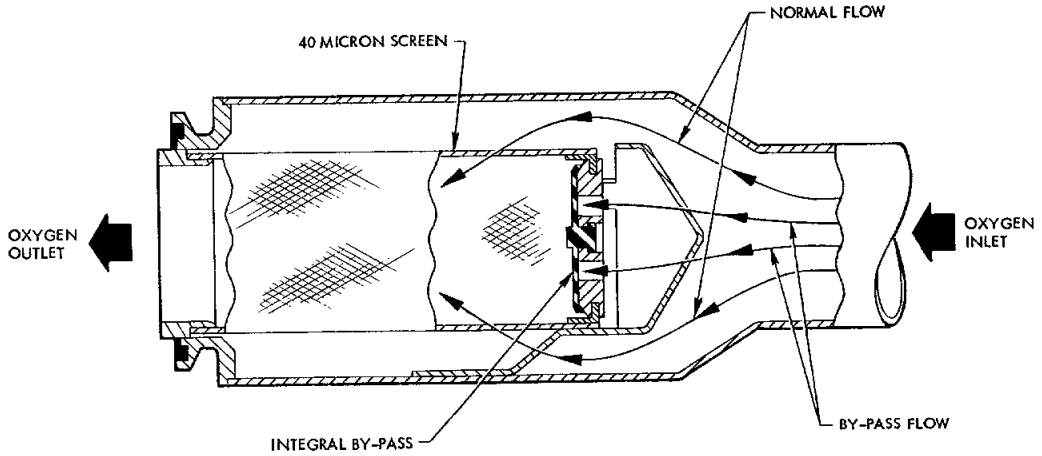
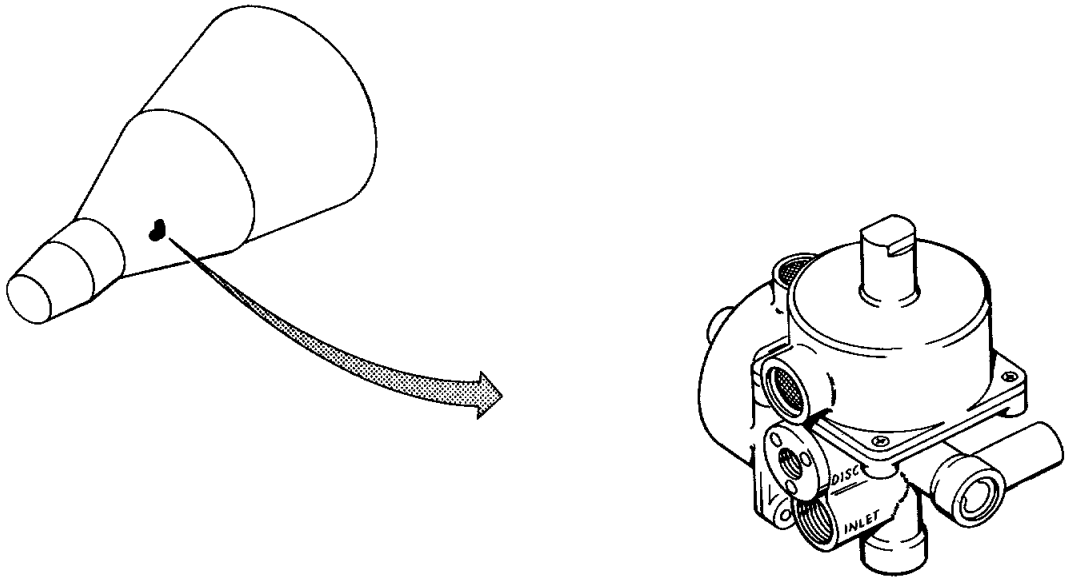


Figure 6-18 Suit Circuit Solids Trap



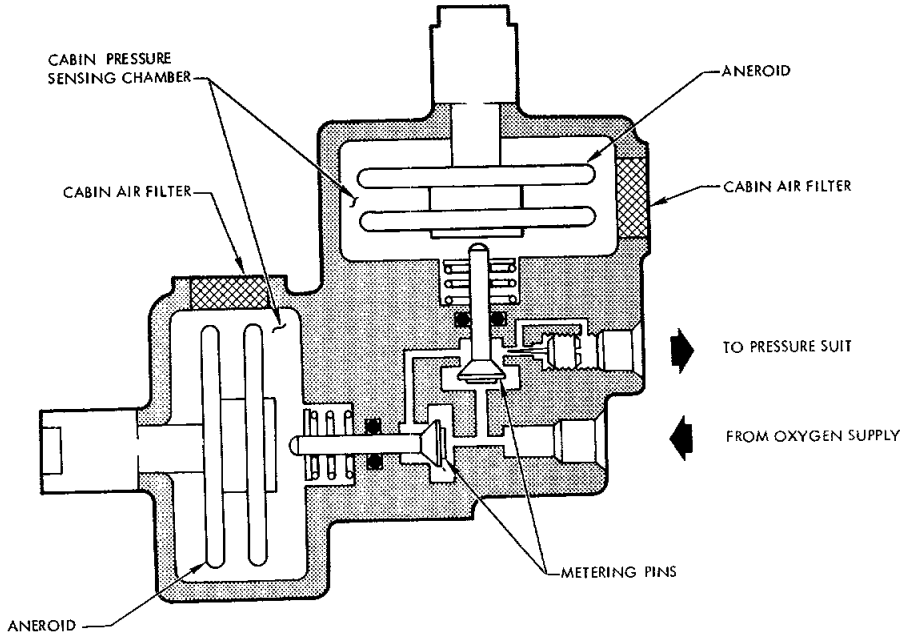


Figure 6-19 Dual Cabin Pressure Regulator

expand, forcing metering pins open and permitting oxygen flow into the cabin, maintaining cabin pressure at 5.1 $^{+0.2}_{-0.1}$ psia. If the cabin is punctured or develops leakage greater than the flow capacity of the valve (4.79 $\pm$ 0.48) 10-3 lb/min, oxygen flow to the cabin is stopped when the cabin pressure decreases to 4.0 $^{+0.2}_{-0.1}$ psia, by the aneroids expanding enough to cause the metering to close off the oxygen.

PRIMARY SUPERCRITICAL OXYGEN CONTAINER (Figure 6-20)

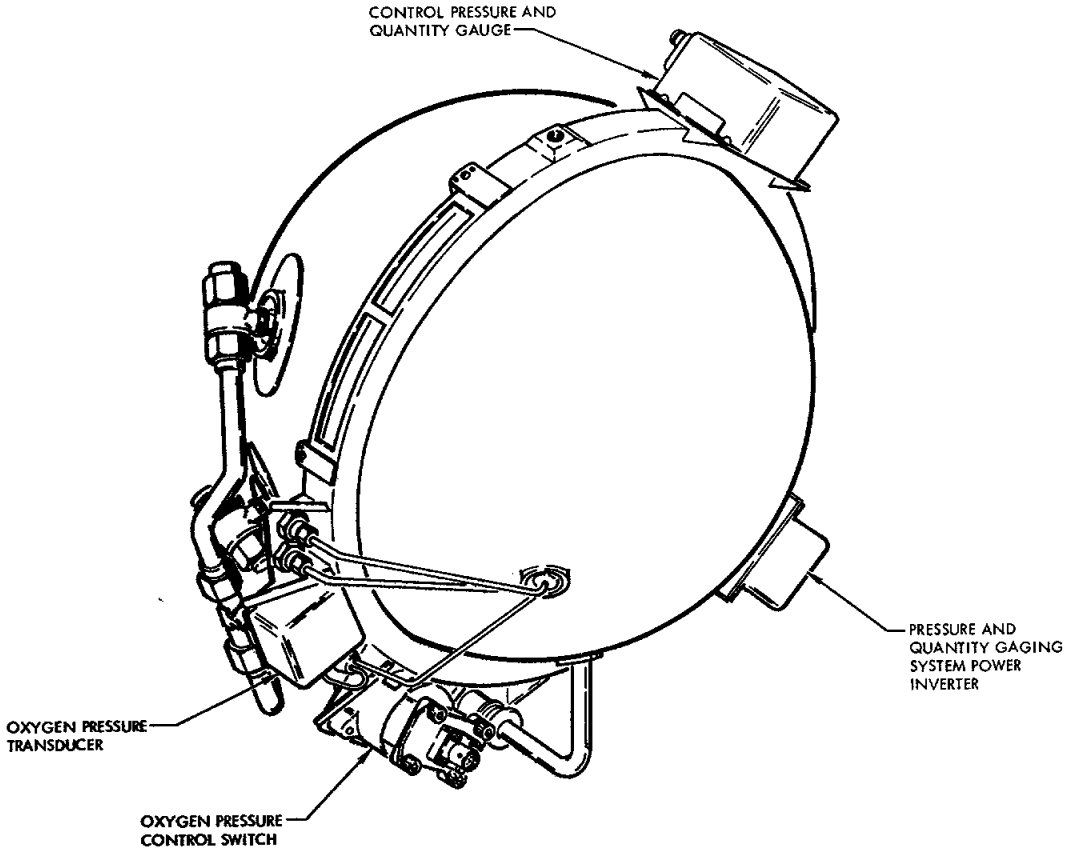
The primary oxygen container is a double walled tank. A dual concentric cylinder, quantity measuring devices, heaters and heat transfer spheres are internal to the container. The tank contains two heaters. The first is a 12.0 $\pm$ 2 watt heater which is activated either manually by a switch located on the center panel, or automatically by a pressure switch. The pressure switch controls the activation of the heating element in the tank to automatically maintain the cryogen in a supercritical state. The switch de-energizes the heater circuit when the pressure in the tank is between 875 to 910 psig, and closes the circuit 15 to 75 psig below the opening pressure. The second heater is a 325 $^{+50}_{-0}$ watt heater manually controlled by a switch located on the overhead switch/circuit breaker panel.

The pressure relief valve maintains the oxygen pressure within the container at 1000 $^{+0}_{-55}$ psig, and prevents overpressurization of the containers.

Provisions for servicing the primary oxygen container from a ground supply source of oxygen are provided.

SECONDARY OXYGEN CONTAINER (Figure 6-21)

The secondary oxygen container is a cylindrical shaped container, having a useful oxygen capacity of 6.5 pounds at an operating pressure of 5000 psig.



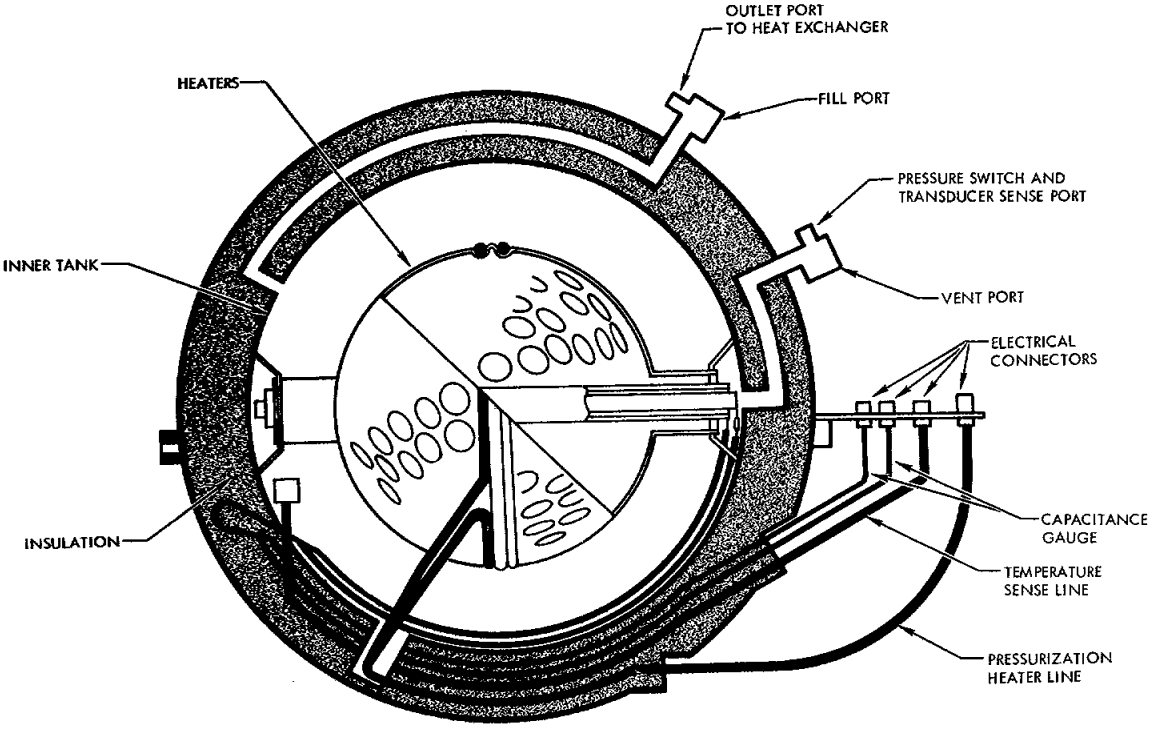


Figure 6-20 Supercritical Primary Oxygen Container

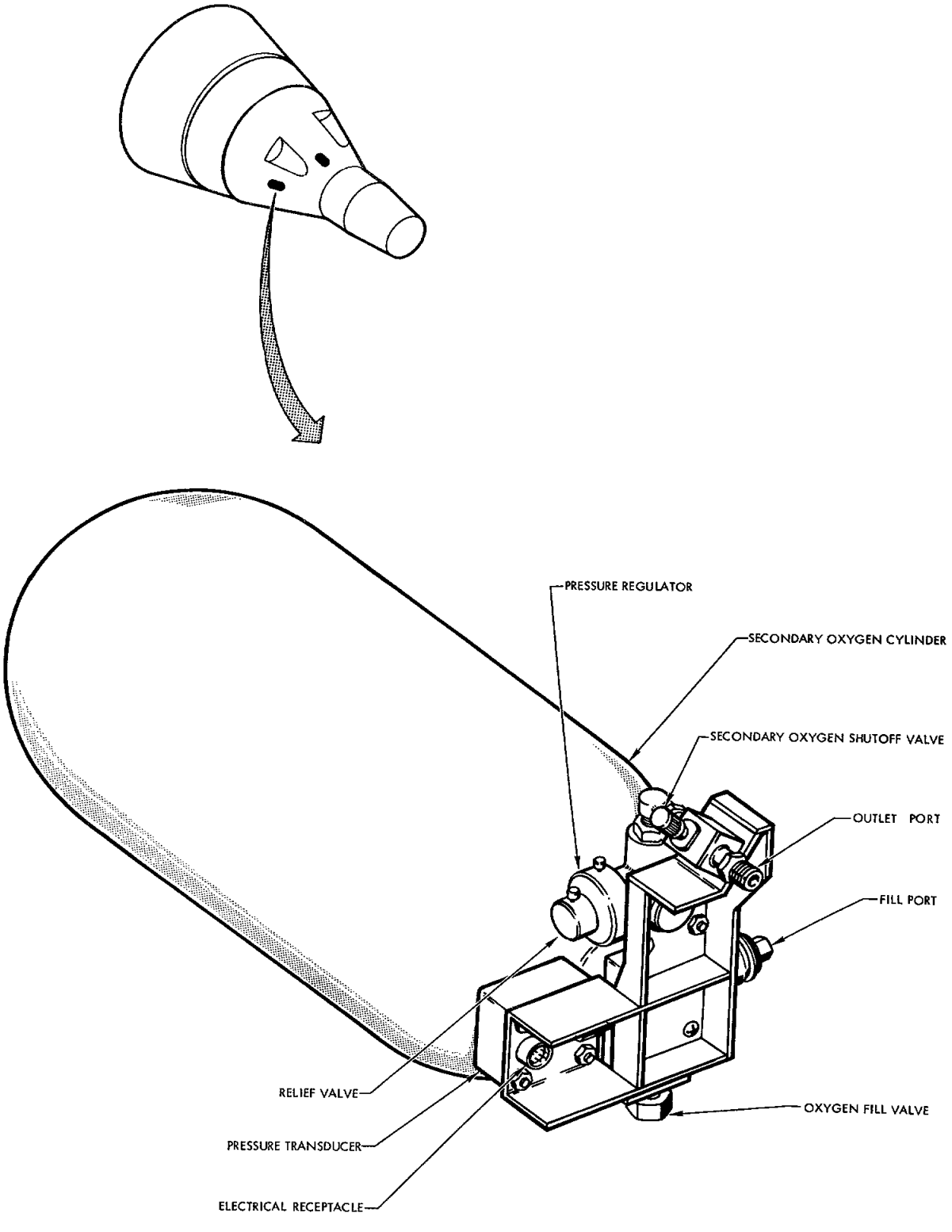


Figure 6-21 Secondary Oxygen Tank

COOLING SYSTEM

Section VII

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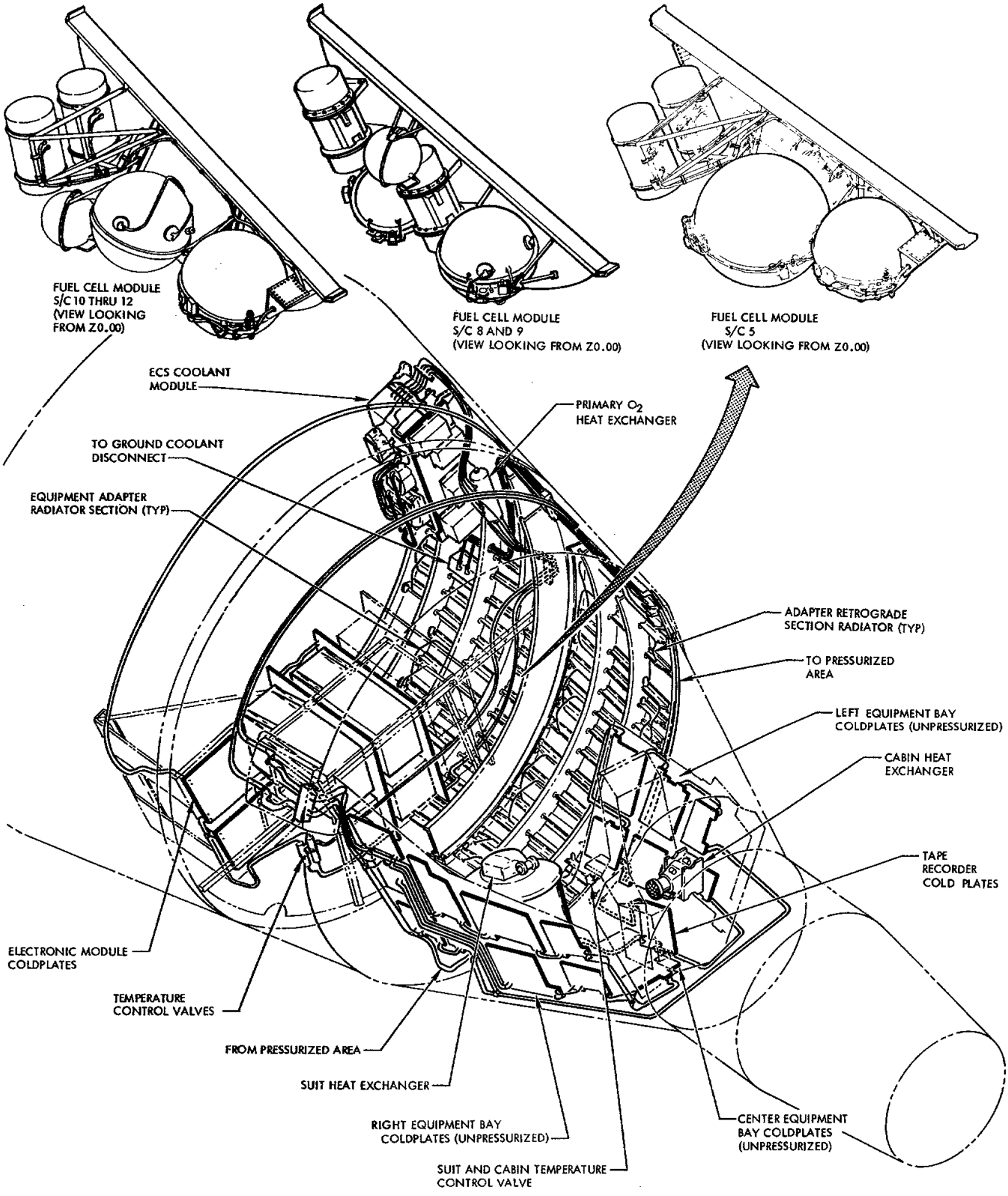


Figure 7-1 Spacecraft Coolant System

SECTION VII COOLING SYSTEM

SYSTEM DESCRIPTION (Figure 7-1)

The spacecraft cooling system consists basically of two identical temperature control circuits functioning independently of each other to provide the cooling requirements for the spacecraft. Each cooling circuit consists of a pump package, thermostatic and directional control valves, various type heat exchangers, radiators, filters, and the necessary plumbing required to provide a closed circuit. The cooling system may be operated in either the primary and/or secondary circuit, and is capable of carrying maximum heat loads in either circuit.

The equipment coldplates, cabin and suit heat exchangers are located in the re-entry module. The upper radiator panels are located in the retrograde section. The pump package, battery coldplates, filters, electronic equipment coldplates, ground launch cooling and regenerative heat exchangers and the lower radiator panels are located in the adapter equipment section. System manual controls are located on the pilots' pedestal console and the control switches, warning lights and indicators are located on the center panel.

The cooling systems in spacecraft 8 through 12 are provided with a means of by-passing the coolant around the fuel cells rather than through them. This provision is for ground operation when fuel cells are not in use.

During orbital flight, Monsanto MCS-198 coolant is supplied throughout the cooling system and thermostatic control valves regulate the coolant temperature.

Temperature sensors, located in the system, provided the necessary telemetering of system temperatures to ground stations.

SYSTEM DISPLAYS AND CONTROLS

Cooling system displays and controls are located on panels in the cabin as shown in section 3 and function as specified below.

SUIT And CABIN TEMP Controls

Dual concentric knobs are mounted between the ejection seats for suit and cabin temperature control. These knobs control the operation of valves regulating the flow rate of primary and secondary coolant through the suit and cabin heat exchangers for spacecraft 5 and 6. Spacecraft 8 does not have the cabin heat exchanger. Clockwise rotation results in increased temperatures.

CABIN And SUIT TEMP INdicator

A dual indicator provides for monitoring temperatures in the suit and cabin circuits. Range markings are calibrated in degrees Fahrenheit.

PRIMARY And SECONDARY Pump Switches

These switches are connected to the coolant pumps power supplies, one switch for each power supply. Each switch has two positions; ON and OFF. The switches are located on the center panel. On spacecraft 8 through 12 B pump switch in each loop changes the flow rate from 183 lb/hr to 140 lb/hr.

PRIMARY And SECONDARY Pump Lights

Pump lights illuminate when the pumps are activated. They are located above their respective switches near the top of the center panel. The RES LO lights illuminate when the coolant level in the reservoir is low.

EVAP PRESS Indicator

This light illuminates when pressure in the evaporator builds up to $4.0\ ^{+0.0}_{-0.3}$ psig and is extinguished when the pressure falls to $3.1\ ^{+0.3}_{-0.3}$ psig.

EVAP PRESS Heater Switch

This switch is connected to the evaporator heater and is used to heat the water in the evaporator before dumping.

SYSTEM OPERATION

The cooling circuit in which the cooling system operates is dependent upon the temperature loads generated by the equipment, spacecraft phase of flight and the temperature within the spacecraft cabin. Cooling is provided throughout the mission up to pre-retrograde firing. At this time the coolant pump packages are jettisoned with the adapter equipment section, terminating spacecraft cooling.

Spacecraft 5 and 8 through 12 require both loops to be operated continuously. In spacecraft 6 the primary circuit operates continuously providing the required cooling during low temperature loads. The secondary circuit is used, in conjunction with the primary circuit, during phases of high temperature loads; namely - launch, rendezvous, and pre-retrograde. Under normal heat loads, the

number 1 pump in the primary circuit provides the required cooling. Under peak heat loads, the number 1 pump in the secondary circuit is used with the primary circuit number 1 pump to provide maximum cooling. In the event of a number 1 pump malfunction in either circuit, the number 2 pump in that circuit is used. In the event of both pumps failing in one circuit, both pumps of the remaining circuit can be used to provide the required cooling. (Spacecraft 6 does not have the number 2 pump in either circuit.)

PRE-LAUNCH (Figure 7-2)

During pre-launch an external supply of Monsanto MCS-198 coolant is circulated through the spacecraft ground cooling heat exchanger providing temperature control of the cooling system coolant. The number 1 pumps of the primary and secondary cooling circuits are activated, using an external power source, to provide the required cooling for spacecraft equipment and cabin. The spacecraft radiator switch, located on the center panel, is placed in the BYPASS position so the cooling system coolant by-passes the radiators and is directed through the ground cooling heat exchanger.

Coolant is circulated through each coolant loop by a positive-displacement gear pump. Spacecraft 5, and 8 through 12 are provided with 2 pumps in each loop. Spacecraft 6 has only one pump in each loop. Selection of loops and number of pumps is controlled manually.

The coolant is filtered, as it leaves the pump, and simultaneously flows to the inlet of the battery coldplate or fuel cell temperature control valve and primary oxygen heat exchanger.

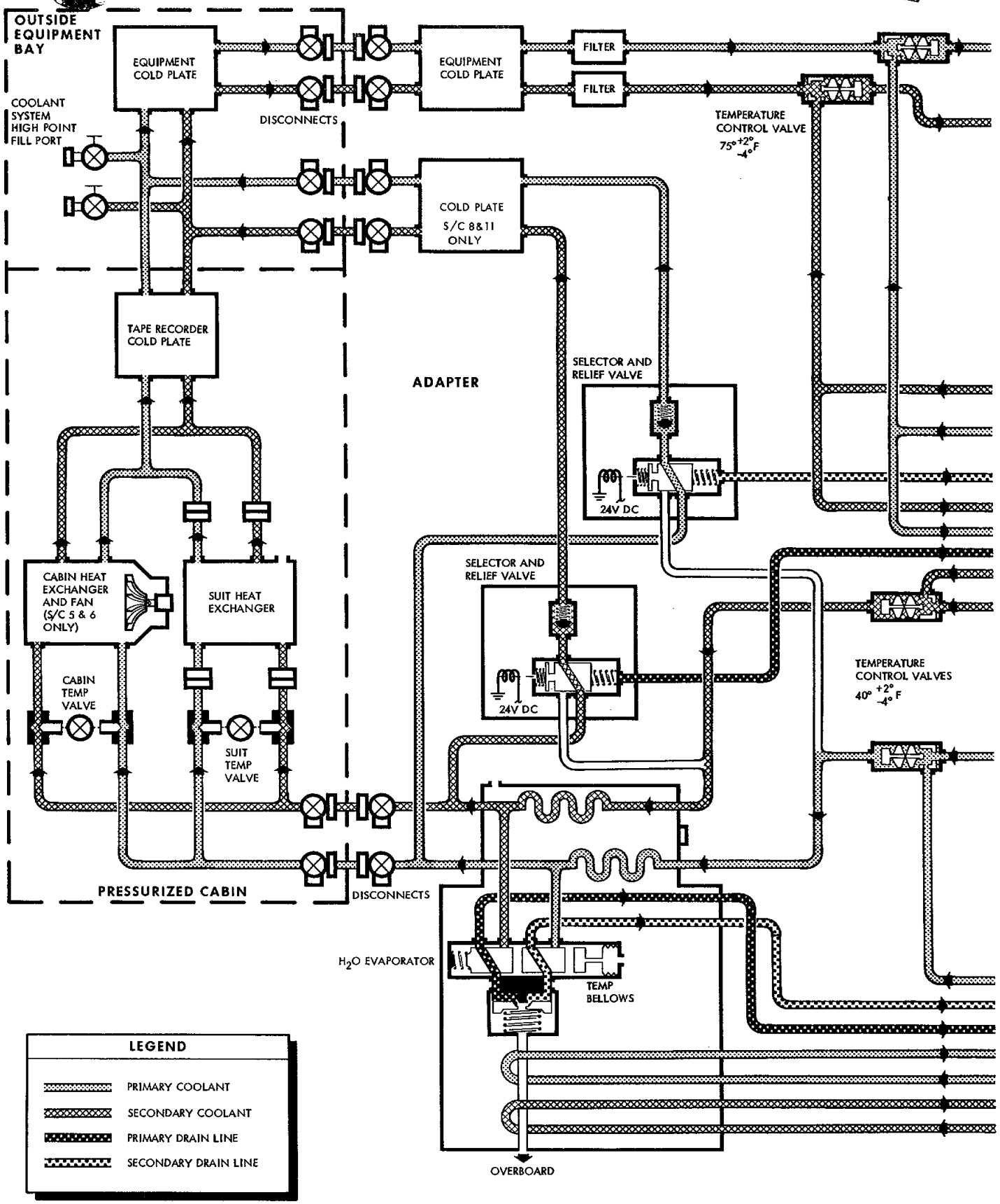


Figure 7-2 Cooling System Flow Schematic Pre-Launch and Launch (Sheet 1 of 2)

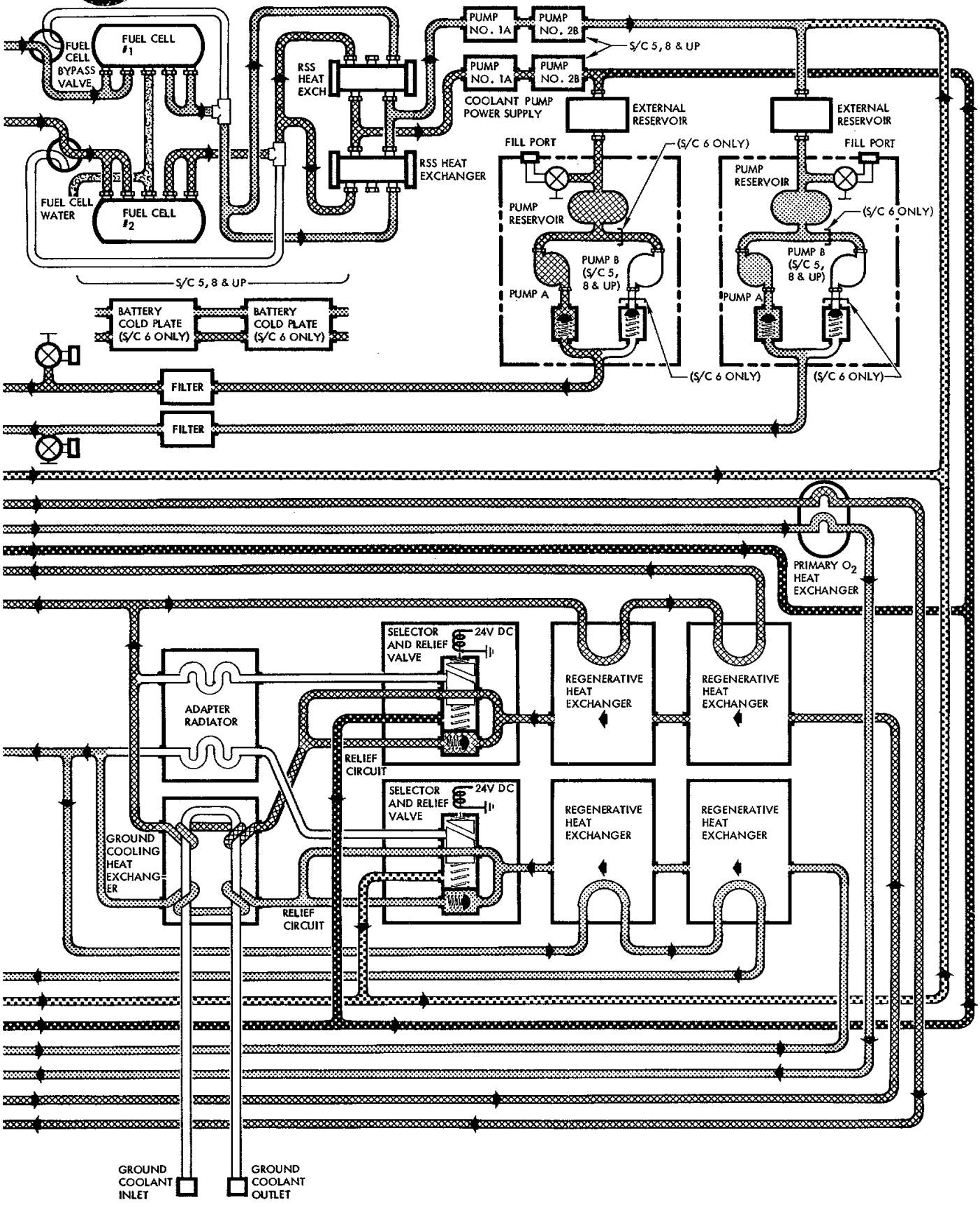


Figure 7-2 Cooling System Flow Schematic Pre-Launch and Launch (Sheet 2 of 2)

The temperature control valve maintains the cooling temperature at the fuel cell or battery coldplate inlet at $75^{\circ}\ ^{+2^{\circ}}_{-4^{\circ}}$ F. Temperature increasing above setting will reduce by-pass flow. Coolant temperature from by-pass line varies from 80°F to 165°F. Coolant temperature from equipment lines varies from 60°F to 125°F.

Coolant enters the primary oxygen heat exchanger and then is routed around the steam discharge lines in the water boiler before it passes through the regenerative heat exchanger. It then passes through the selector and pressure relief valve. This selector valve is electrically actuated and when in the radiator by-pass position allows the coolant to pass through the ground cooling heat exchanger where the external supply of coolant flowing through the ground cooling heat exchanger absorbs the heat from the spacecraft's coolant system.

The ground coolant heat exchanger has an airborne flow capacity of 336 lb/hr, per coolant loop, at 125°F. It has a ground coolant flow capacity of 425 lb/hr at 40°F.

The coolant is now ready to pass through the temperature control valve. This valve maintains the outlet temperature at $40^{\circ}\ ^{+2^{\circ}}_{-4^{\circ}}$ F. If the coolant entering the valve from the ground heat exchanger is below this range, a portion of the coolant is directed through the regenerative heat exchanger and then mixed at the valve.

The coolant then flows through the water evaporator to the cabin and suit manual temperature control valves. These valves meter the coolant flow through the cabin and suit heat exchangers. The evaporator selector valve relief portion allows part of the coolant to by-pass the cabin and suit heat exchangers depending

on the setting of the manual control valves. The selector portion of this valve allows the by-pass fluid to come from either downstream or upstream of the evaporator. The coolant continues through the various coldplates until it reaches the battery coldplates for spacecraft 6 or through the fuel cells on spacecraft 5, and 8 through 12. The coolant has now returned to the reservoir where the cycle is ready to be repeated.

Shortly before launch, the external cooling and electrical power are disconnected.

LAUNCH (Figure 7-2)

During launch, the launch cooling heat exchanger goes into operation in the following sequences. The heat transfer characteristics and capabilities of the ground cooling heat exchanger no longer exist. The Monsanto MCS 198 coolant fluid now with no place to dissipate its internal heat, which is constantly being generated by and absorbed from the loop components, circulates about the temperature control valve of the heat exchanger. When the coolant temperature exceeds $46\ ^{+4^{o}}_{-2^{o}}$ F the temperature control valve opens to pressurize a donut shaped bellows which unseats the poppet valve exposing the water in the heat exchanger core to reduced pressure as altitude increases during launch.

When spacecraft altitude exceeds 100,000 feet, water in the heat exchanger will boil absorbing heat from the coolant. This absorbed heat is then expelled overboard in the form of steam.

When the coolant reaches a temperature of 46^{o}F, the temperature control valve repositions to relieve pressure to the donut shaped bellows holding the poppet open. As this pressure diminishes, a spring behind the poppet will reposition it to the

closed position. The evaporator selector valve is positioned to allow all flow to go through the evaporator.

The water boiler water reservoir is constantly replenished from the suit heat exchanger water separator, and if the need arises, from the drinking water supply tank.

ORBIT (Figure 7-3)

After orbiting for approximately 30 minutes, to allow the radiator to cool after being subject to launch heating, the coolant flow is directed through the space radiators by manual selection of the radiator switch located on the center panel. This by-passes the ground cooling heat exchanger. The evaporator selector valve is also positioned so that only the flow to the suit and cabin heat exchangers pass through the evaporator.

Prior to retrograde firing, the coolant pump packages, radiators, batteries and various heat exchangers are jettisoned with the adapter equipment section. Prior to adapter jettisoning and retrograde firing the number 1 coolant pumps for both the primary and secondary coolant circuits are activated. The suit, cabin, and equipment bays are cooled to as low a temperature as possible, before the adapter equipment section is jettisoned.

SYSTEM UNITS

PUMP PACKAGE (Figure 7-4)

The pump package for each coolant circuit incorporates two constant displacement electrical pumps, two pump inverters, an external reservoir, filters, relief and

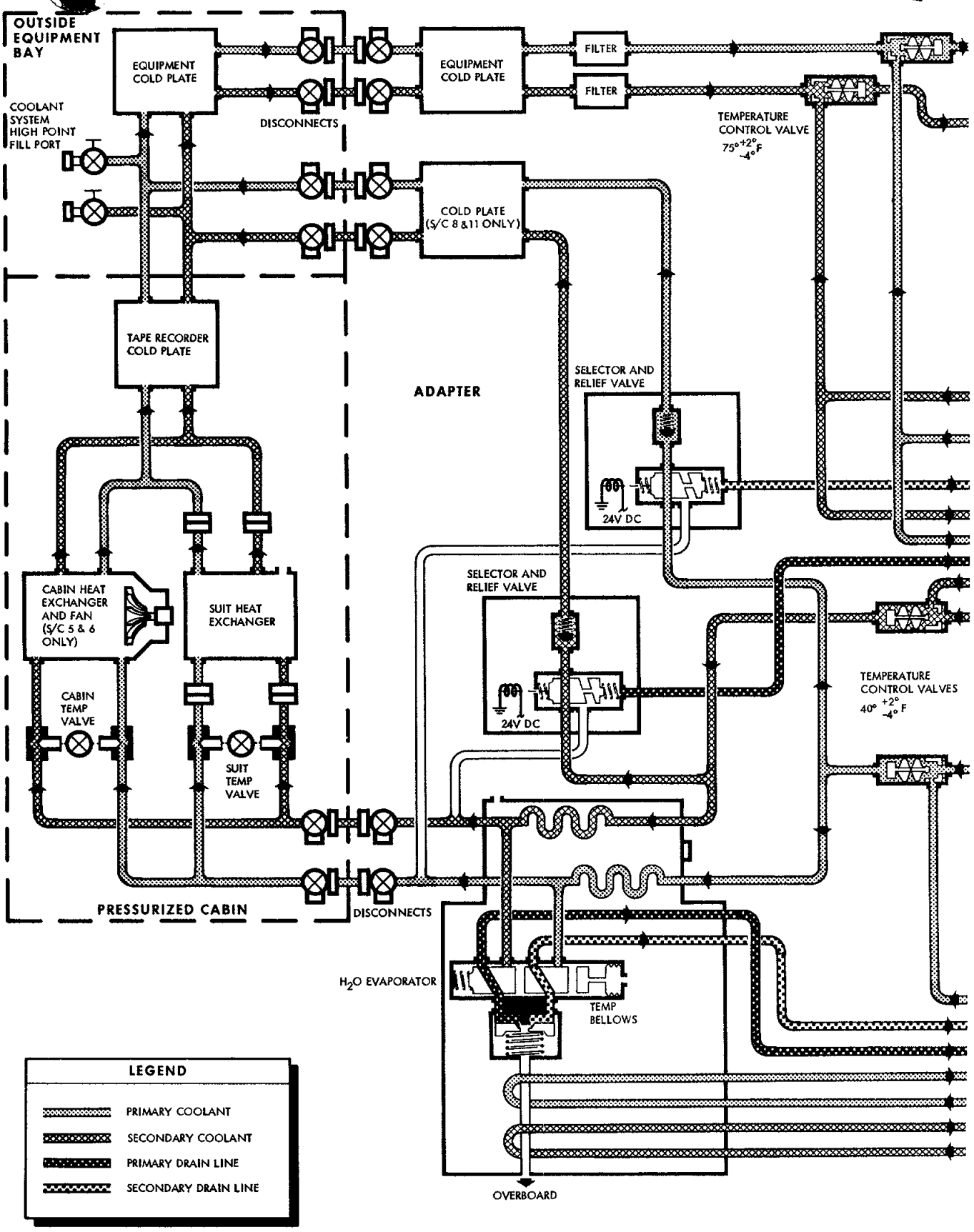


Figure 7-3 Cooling System Flow Schematic-Orbit (Sheet 1 of 2)

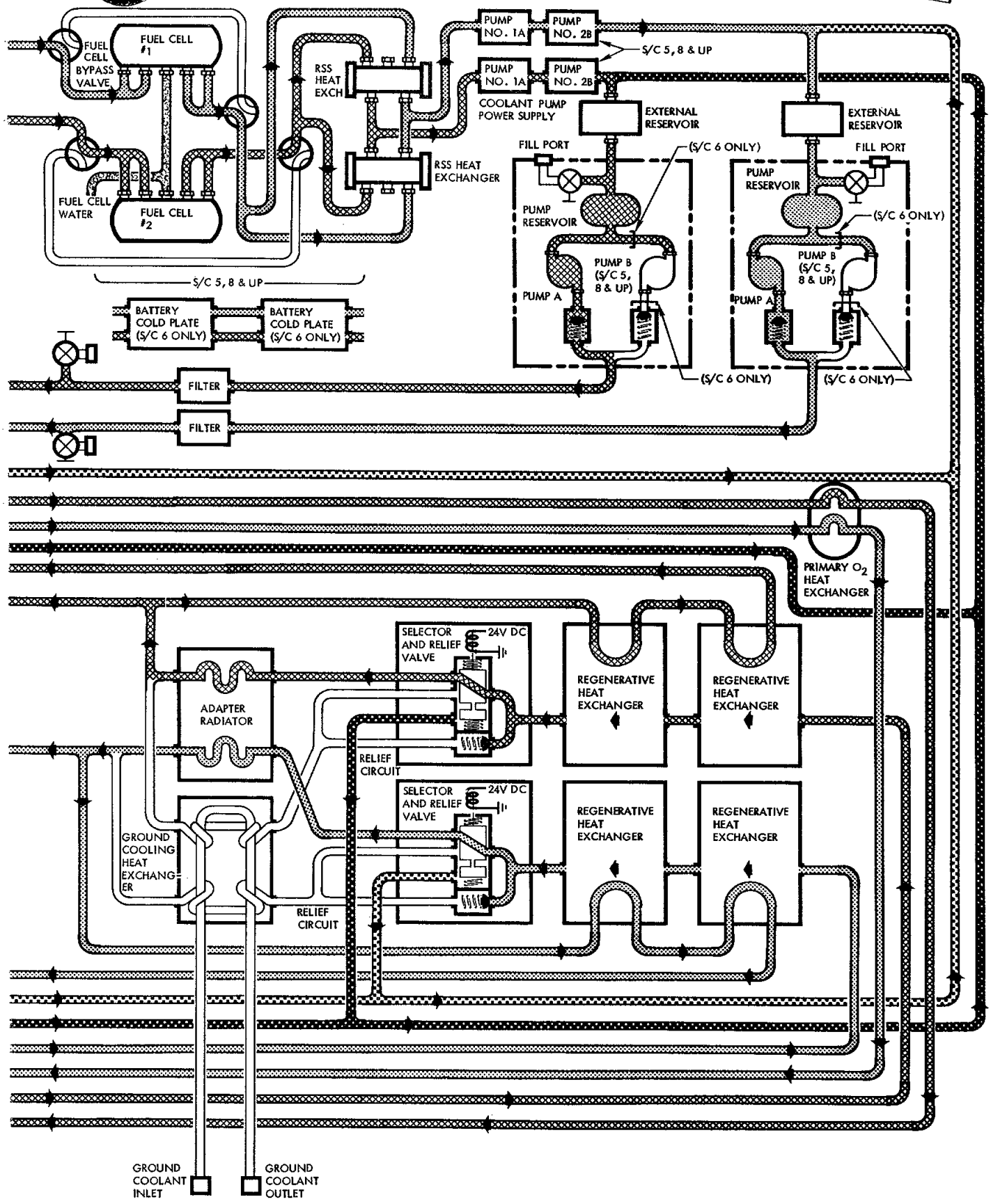


Figure 7-3 Cooling System Flow Schematic-Orbit (Sheet 2 of 2)

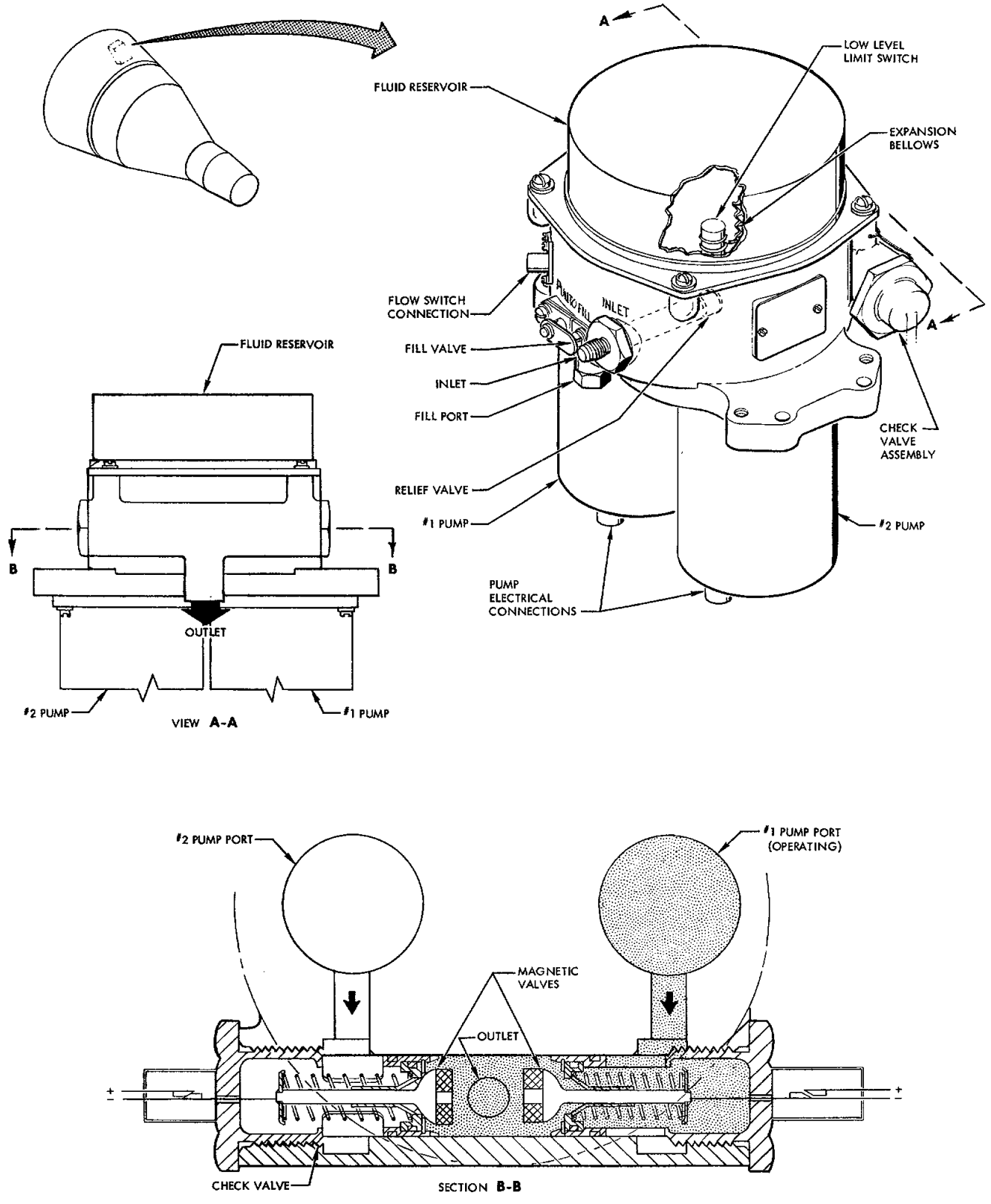


Figure 7-4 Coolant Pump Package

check valves. The pump package is located in the adapter equipment section. Pump selection is provided by switches on the pilots' center panel. A pump failure warning light is provided on the center panel. When a pump is activated the coolant flows from the reservoir to the pump, which circulates the coolant through the cooling circuit. The coolant returns to an external reservoir that compensates for thermal expansion, contraction, and leakage of the coolant. A 100 micron filter downstream of the pump prevents contamination of the cooling system. Check valves in the pump package prevent the operating pump from pumping coolant into the redundant pump. Flow sensing switches illuminate a pump failure lamp on the pilots' center panel in the event of pump failure.

RADIATOR (Figure 7-5)

The spacecraft radiator consists of two circumferential radiator panels made of 0.25 inch diameter cooling tubes. There are four sections of tubing to each radiator panel. The tubing is manufactured as part of the spacecraft structure. Each panel incorporates two parallel cooling circuits, one for the primary cooling circuit and the other for the secondary circuit. During orbit the cooling system coolant is circulated through the radiator. The heat of the coolant radiates into space, lowering the temperature of the coolant.

COLDPLATES (Figure 7-6)

The coldplates, other than the battery coldplates, are plate fin constructed units incorporating parallel coolant system passages. Coldplates are fabricated from aluminum. Battery, electrical, electronic and other heat generating

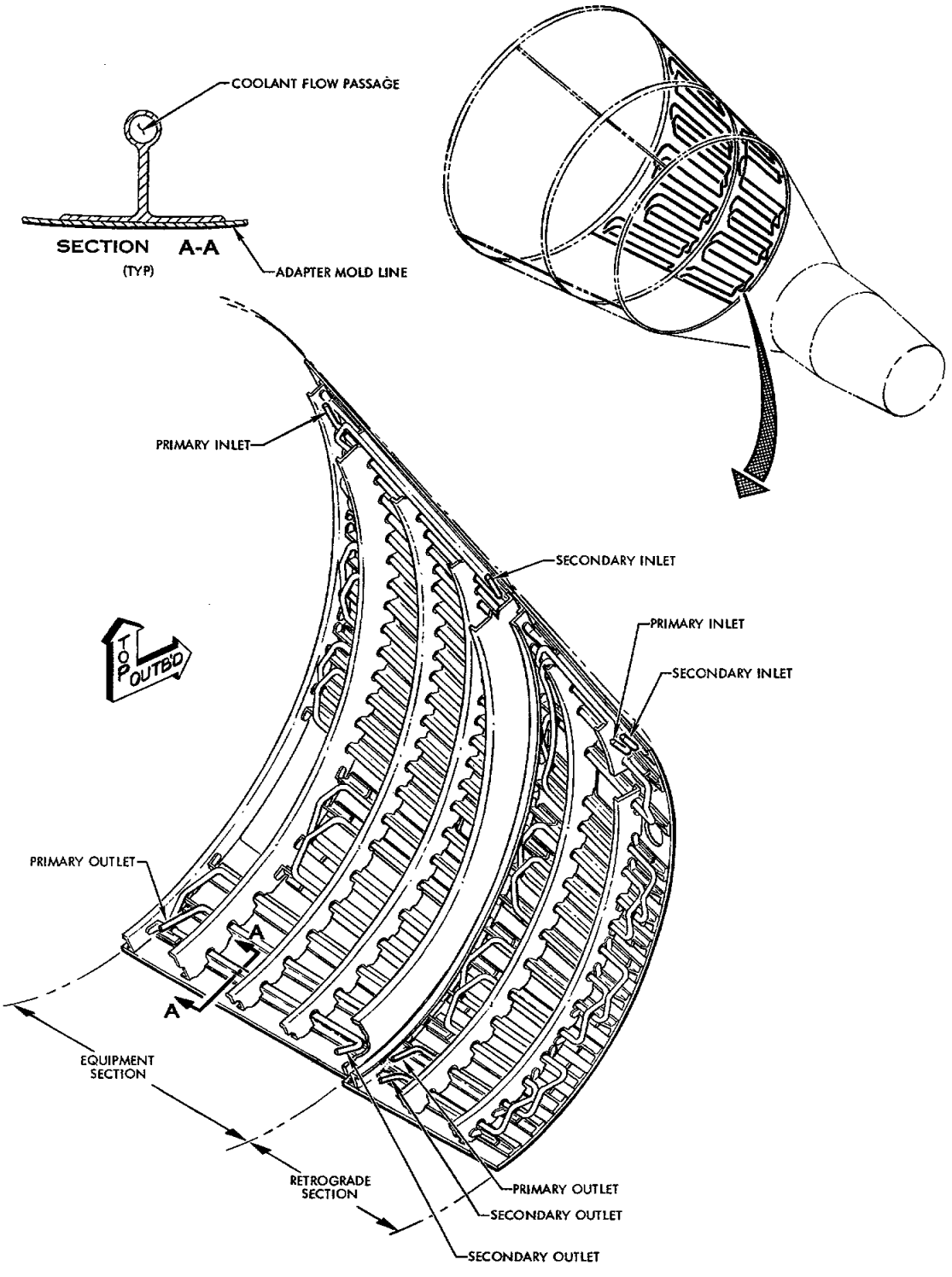
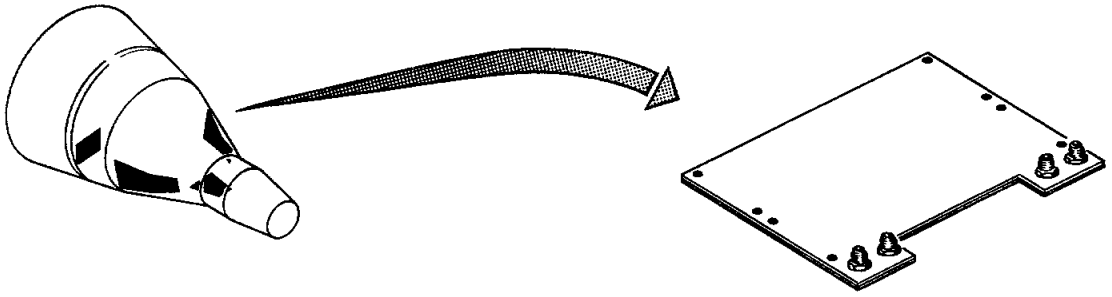
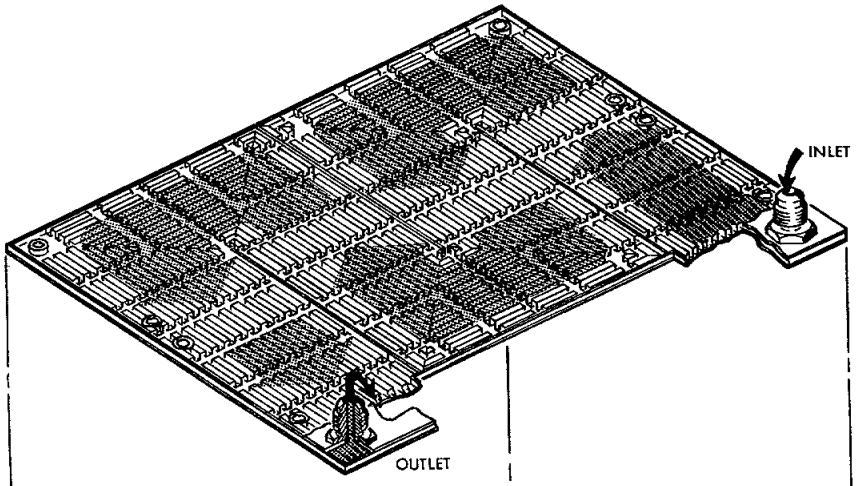


Figure 7-5 Radiator Stringer Assembly

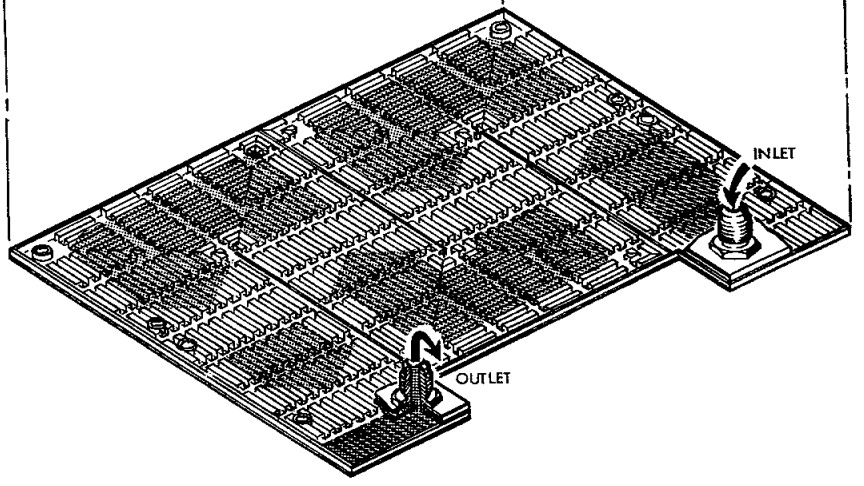


COLD PLATE (TYPICAL)



PRIMARY SECTION

VIEW OF COLD PLATE SEPARATED TO CLEARLY ILLUSTRATE THE PRIMARY AND SECONDARY FLOW.



SECONDARY SECTION

Figure 7-6 Cold Plate

components are mounted on coldplates. The coolant flowing through the coldplates absorbs the heat generated by the components, preventing overheating of the operating equipment.

HEAT EXCHANGERS (Figure 7-7, 7-8)

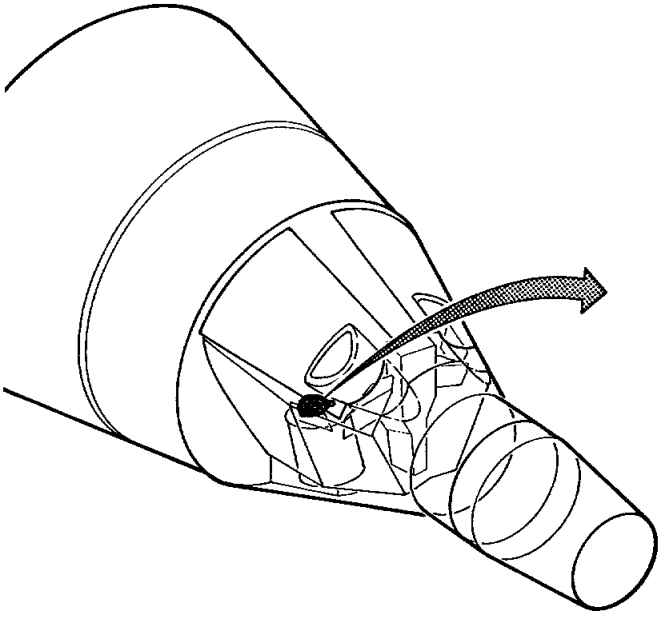
Two types of heat exchangers are used in the spacecraft; namely, plate fin constructed and shell and tube constructed heat exchangers. The suit, cabin, water evaporator, ground cooling and regenerative heat exchangers are of plate fin construction. The primary oxygen heat exchanger is of shell and tube construction. The coolant absorbs heat from the cabin, suit and regenerative heat exchangers. The ground cooling and water evaporator heat exchangers permit heat transfer to cool the coolant. The primary oxygen heat exchanger is designed so heat transfer will heat the primary oxygen to a desired temperature.

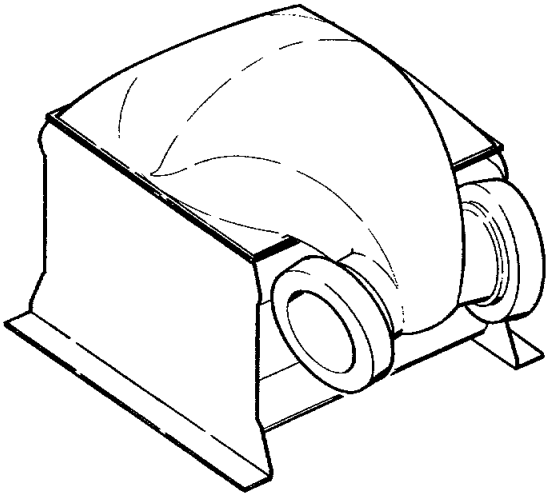
TEMPERATURE CONTROL VALVE (Figure 7-9)

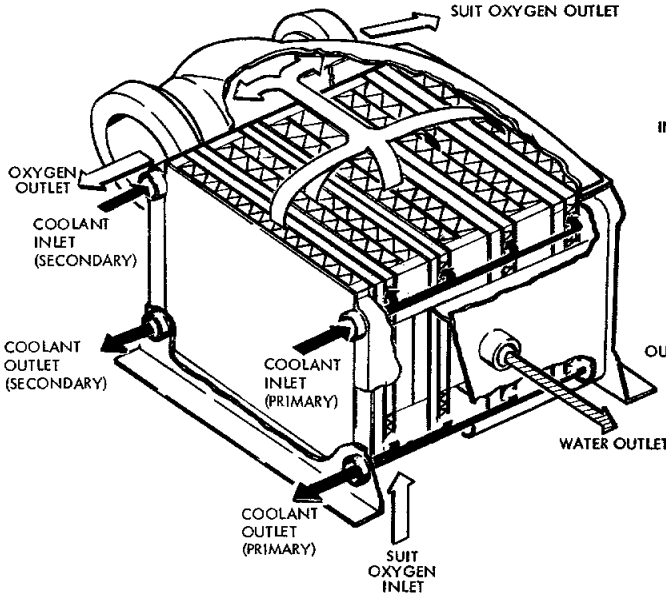
Temperature control valves are provided in both the primary and secondary cooling circuits. These valves are located at the radiator outlets and at the inlets to the battery coldplates or fuel cells.

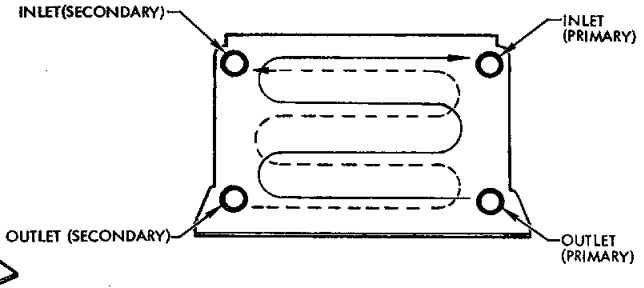
The temperature control valve located in the coolant system radiator outlet automatically maintains the coolant outlet temperature at $40\ ^{+2^{\circ}}_{-4^{\circ}}$F as long as the radiator capacity has not been exceeded.

The temperature control valve located in the battery coldplate inlet automatically maintains the coolant inlet temperature at $75\ ^{+3^{\circ}}_{-4^{\circ}}$ F or above.





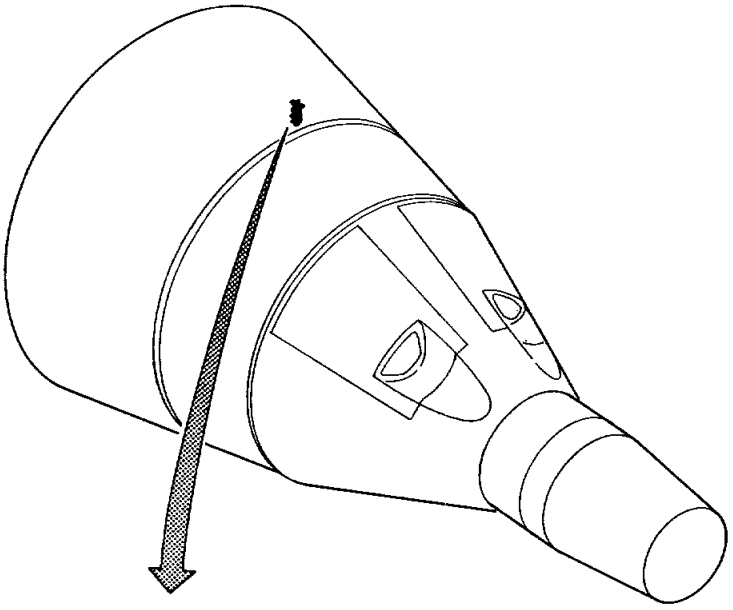




VEIW ROTATED 180°

TYPICAL PLATE FIN CONSTRUCTION

Figure 7-7 Heat Exchanger-Suit



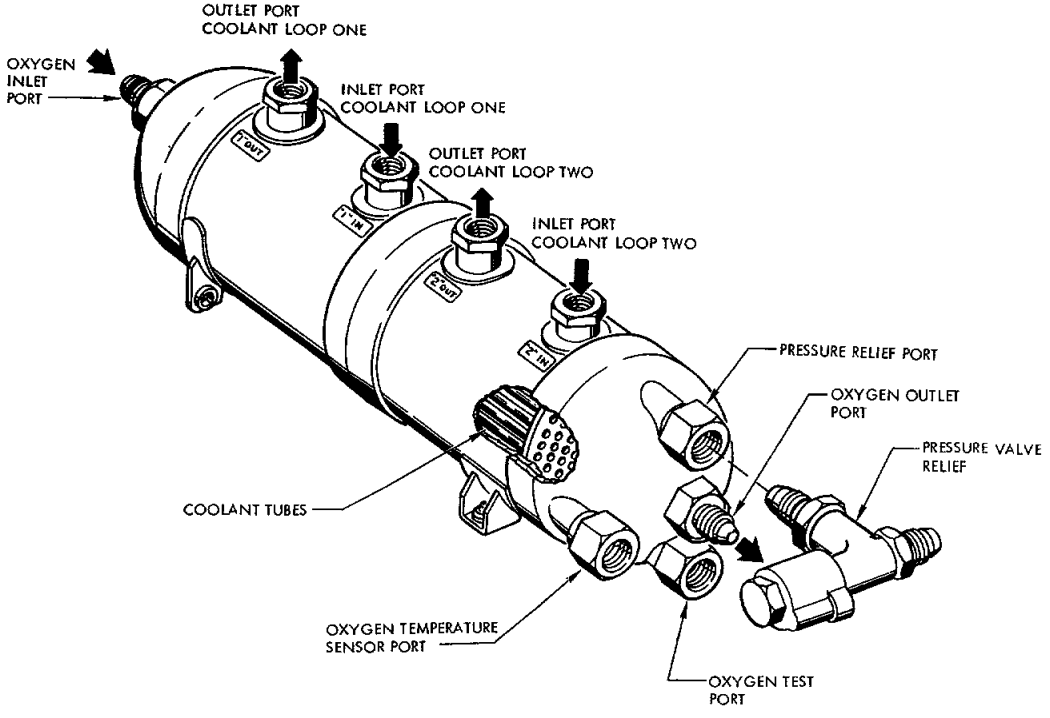
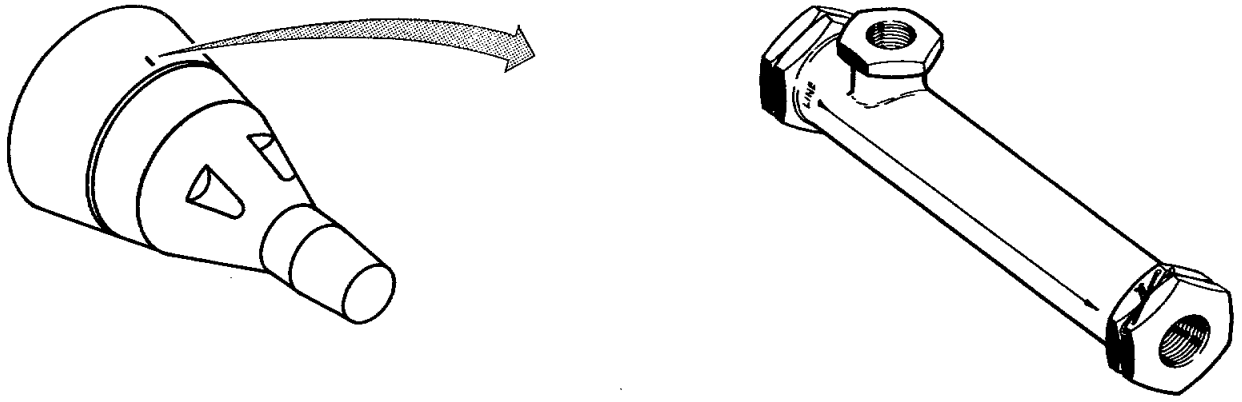


Figure 7-8 Coolant Tube Type Heat Exchanger



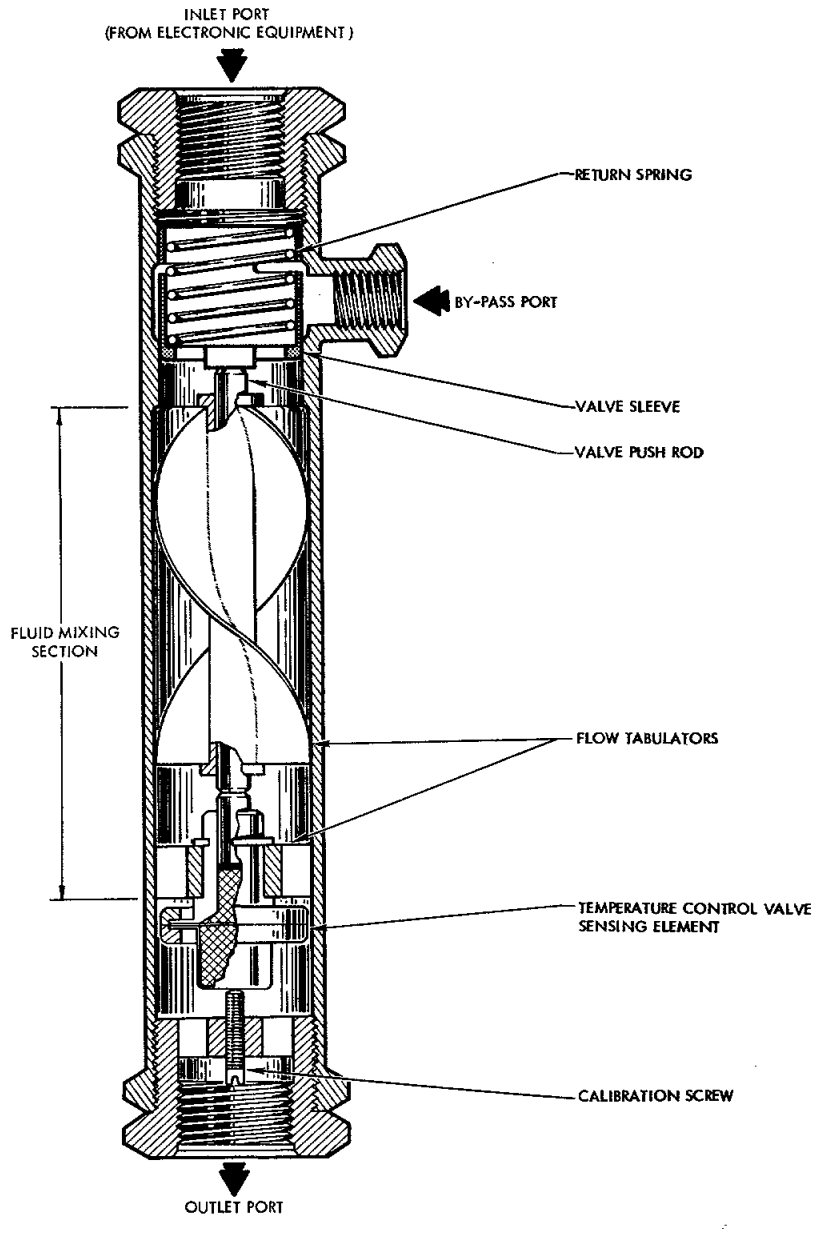


Figure 7-9 Coolant Temperature Control Valve

The temperature control valve contains a piston that regulates the inlet flow to the valve. The piston is spring loaded on one side. A thermostatic actuator on the opposite side of the piston determines piston movement, which in turn regulates the coolant flow through the valve. The thermostatic actuator, which is located to accurately sense mixing temperature, consists of an encapsulated wax pellet that expands or contracts as temperature varies. As temperature around the pellet increases, the wax expands exerting pressure on the diaphragm. The diaphragm moves a piston, which in turn controls the inlet flow to the valve. Temperature reduction around the wax decreases the pressure in the pellet cup allowing the spring to reposition the piston regulating the flow of coolant through the valve.

LAUNCH COOLING HEAT EXCHANGER (Figure 7-10, 7-11)

The launch cooling heat exchanger is located in the adapter section. Via its relief valve it can dump liquids overboard, or if the temperature control valve senses temperature greater than $50^{\circ}F$, it can control the outlet temperature of the primary and secondary coolants to $46\ ^{+4^{\circ}}_{-2^{\circ}}F$. In addition, it serves as a water reservoir, storing water until it is needed for cooling.

This evaporator consists of a wicking type heat exchanger and is capable of storing seven pounds of water. A temperature control valve has been set to control the outlet coolant temperature to $46^{\circ}\ ^{+4^{\circ}}_{-2^{\circ}}$ F. A relief valve opens and allows excess water to be dumped overboard at 2.75 ± 0.25 psi differential and reseats at 2.0 psi differential minimum. An electrical heater is provided in the poppet to prevent ice formation. Coolant flow capacity is 366 lb/hr at $40^{\circ}F$.

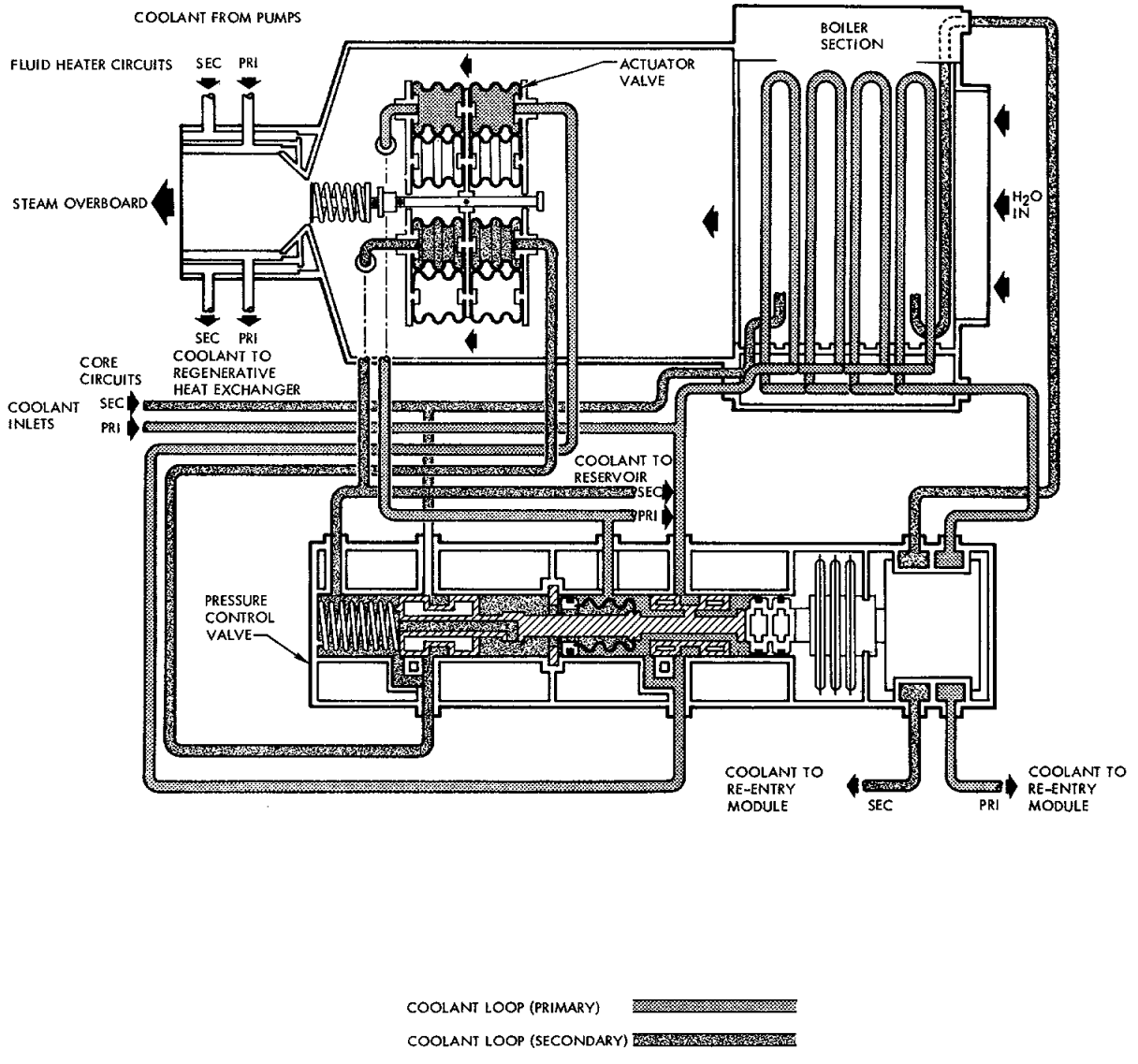


Figure 7-10 Launch Cooling Heat Exchanger Schematic

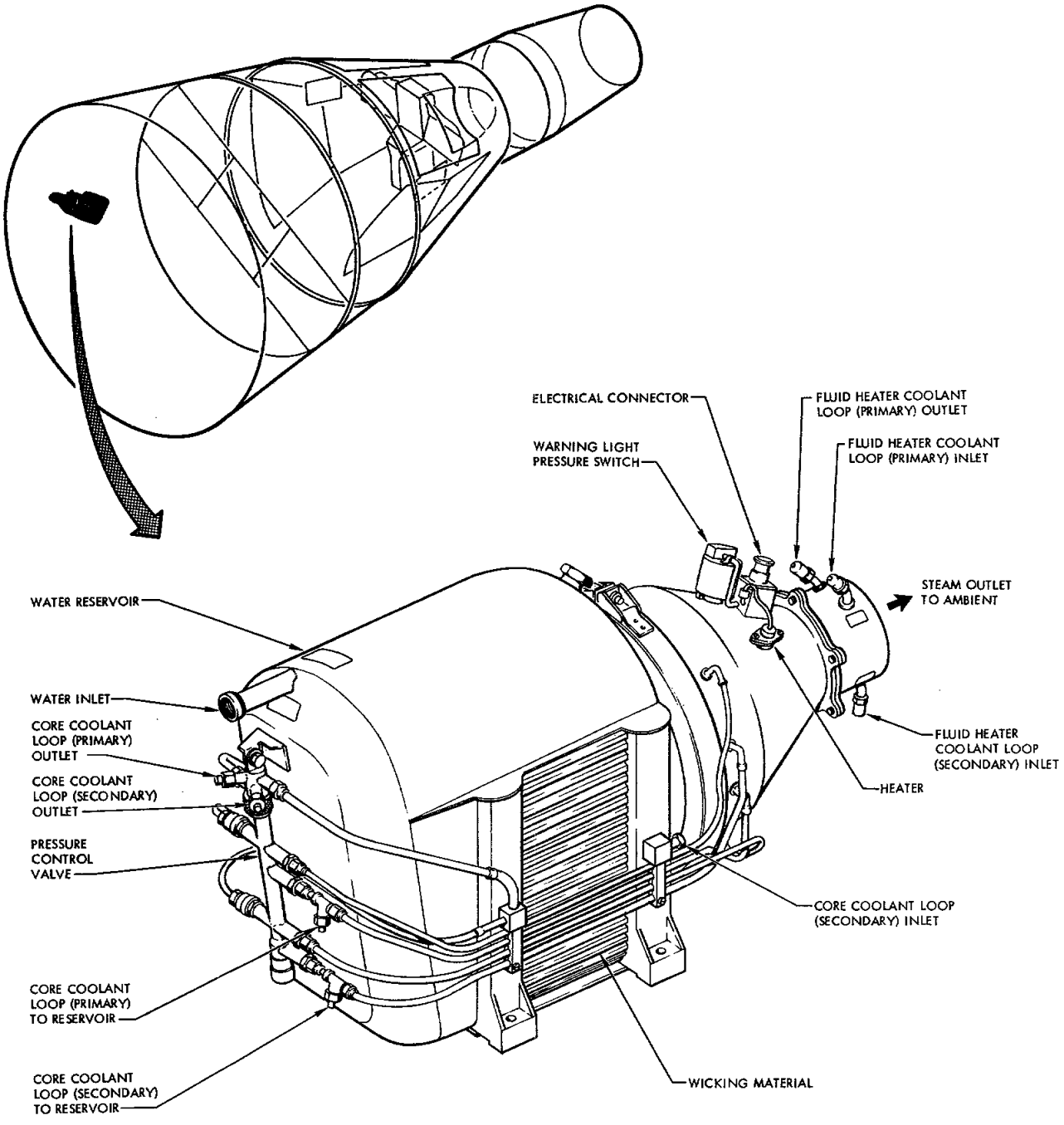


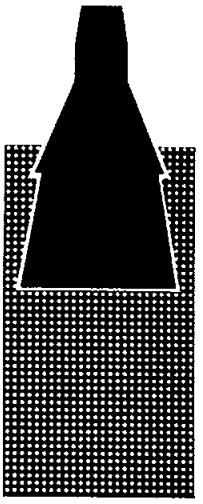
Figure 7-11 Launch Cooling Heat Exchanger

Water flow capacity is 3 lb/min when cooling is not required from the evaporator. Maximum operating pressure in the fluid heater coolant circuits is 230 psig, and 100 psig in the core circuits. Maximum operating pressure in the water circuit is 20 psig with exit port relief valve in normal operation.

The steam exit duct is continuously heated by coolant coming from the primary oxygen heat exchanger to prevent ice formation.

A loss of pressure in either coolant loop will not affect the operation of the valve.

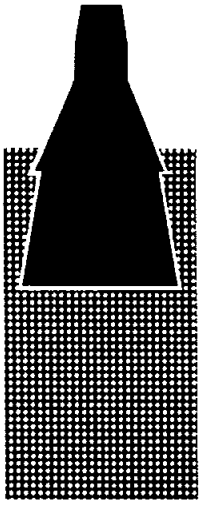
GUIDANCE and CONTROL SYSTEM



Section VIII

REFER TO THE SEDR 300 CONFIDENTIAL SUPPLEMENT FOR INFORMATION CONCERNING THE GEMINI GUIDANCE AND CONTROL SYSTEM.

COMMUNICATION SYSTEM



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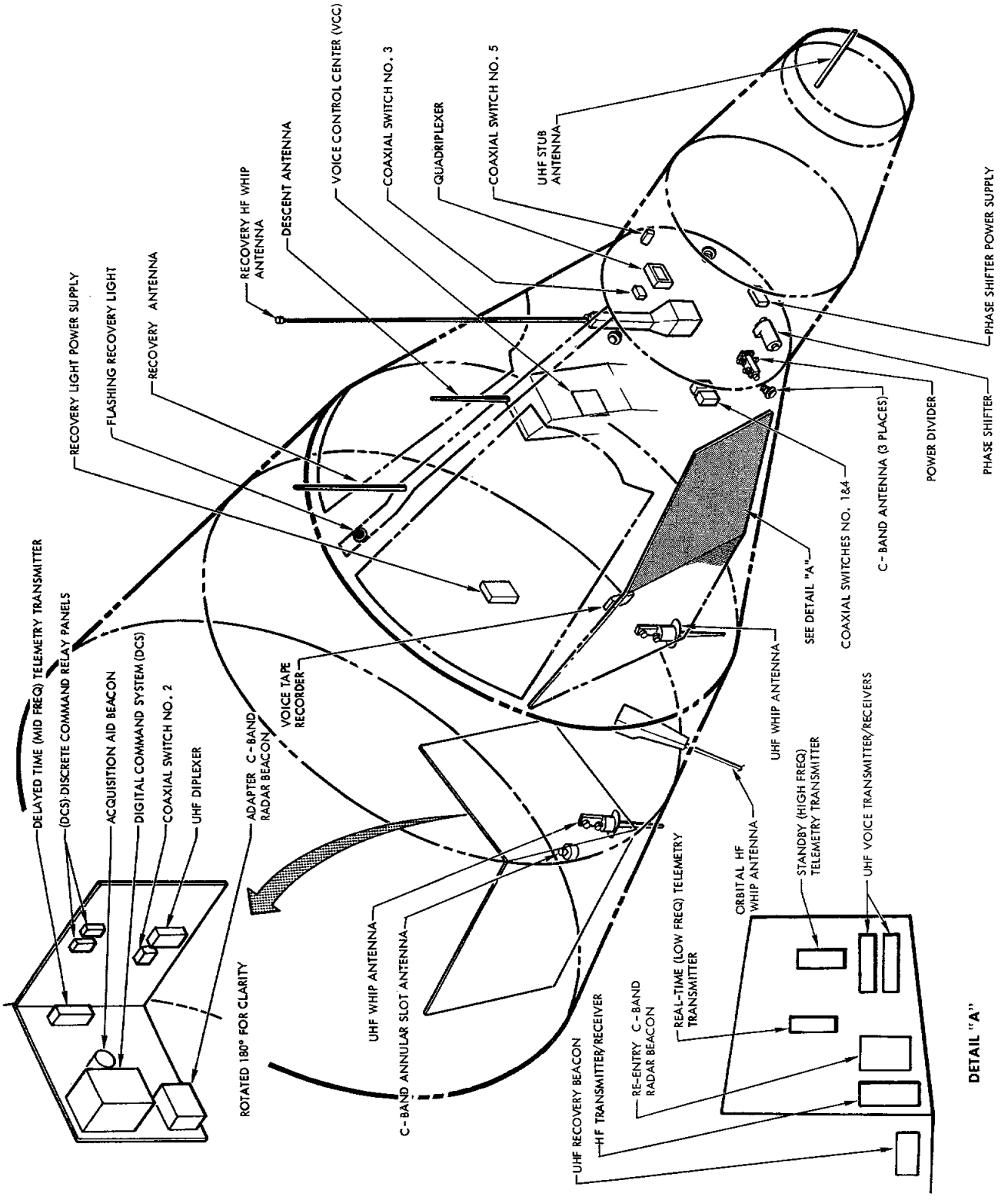


Figure 9-1 Communication System

SECTION IX COMMUNICATION SYSTEM

SYSTEM DESCRIPTION

The Communication System is the only communication link between the ground and the Gemini Spacecraft. The system has the following capabilities: radar tracking of the spacecraft; two-way voice communications between the ground and the spacecraft, and between the crew; ground command to the spacecraft; Instrumentation System data transmission; and postlanding and recovery aid data transmission. To make possible these various capabilities, the Communication System contains components that may be divided into the following categories: antennas, including multiplexers and coaxial switches; beacons; voice communications; telemetry transmitters; flashing recovery light; and Digital Command System. The flashing recovery light and the uhf recovery beacon are grouped together in a category called the Electronic Recovery Aids (ERA).

The Communication System components are located throughout the spacecraft with the largest concentration being in the right equipment bay of the re-entry module and the electronic module of the adapter equipment section as illustrated in Figure 9-1.

ANTENNAS

Eight antennas and one antenna system provide transmission and/or reception capabilities for the various Communication System components. The spacecraft Communication System (Figure 9-2) contains the following antennas: uhf recovery; uhf stub; uhf descent; two uhf whips; two hf whips; C-band annular slot; and a C-band antenna system consisting of a power divider, a phase shifter, a phase shifter power supply, and three helical antennas. Antenna usage is illustrated in Figure 9-3 and described in the individual antenna description.

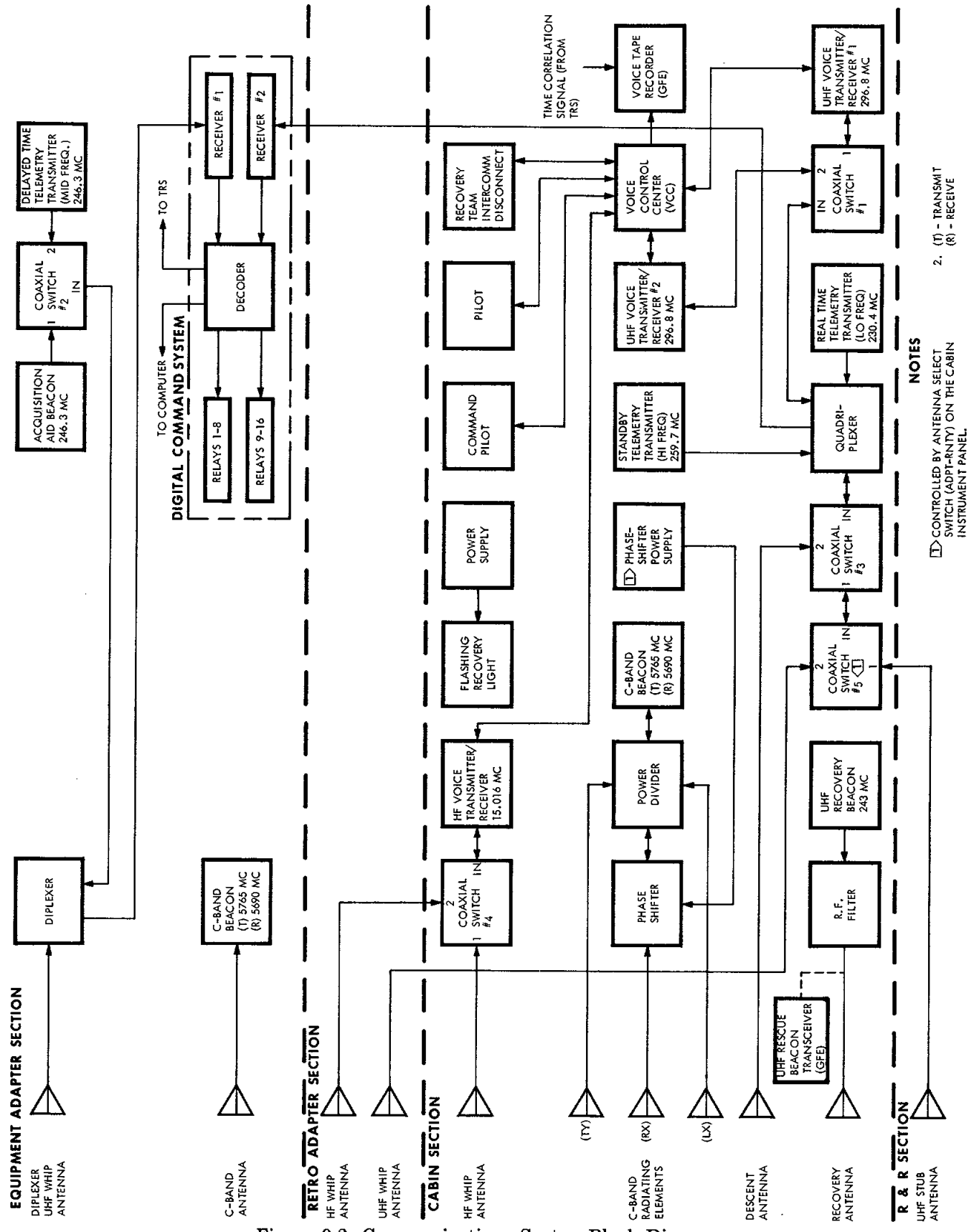


Figure 9-2 Communications System Block Diagram

To achieve the most efficient antenna usage, a diplexer and a quadriplexer are used with the uhf whips and the uhf stub antenna. The multiplexers make it possible to use more than one transmitter and/or receiver with a single antenna.

Five coaxial switches permit antenna and transmitter/receiver switching for best communication coverage during the various phases of the mission (launch, orbit, re-entry and recovery).

BEACONS

Four beacons in the Communication System establish the capability of locating and tracking the spacecraft during the mission. The four beacons are: An acquisition aid beacon and a recovery beacon used to locate the spacecraft, and two C-band beacons used to track the spacecraft. The acquisition aid beacon, operating on a fixed frequency, is used to determine when the spacecraft is within the range of a ground tracking station, and provides information for orientating the ground station antennas during the orbital phase of the mission. The recovery beacon is a transmitter that operates on the international distress frequency, and is used by the recovery forces to determine the spacecraft location. The C-band beacons are transponders which, when properly interrogated by a ground station, transmit signals for accurate spacecraft tracking.

During the recovery phase of the mission, emergency communications can be established by connecting one of the uhf rescue beacon transceivers to the uhf recovery antenna. The rescue beacon transceivers are Government Furnished Equipment (GFE), stowed in pilot's survival kits.

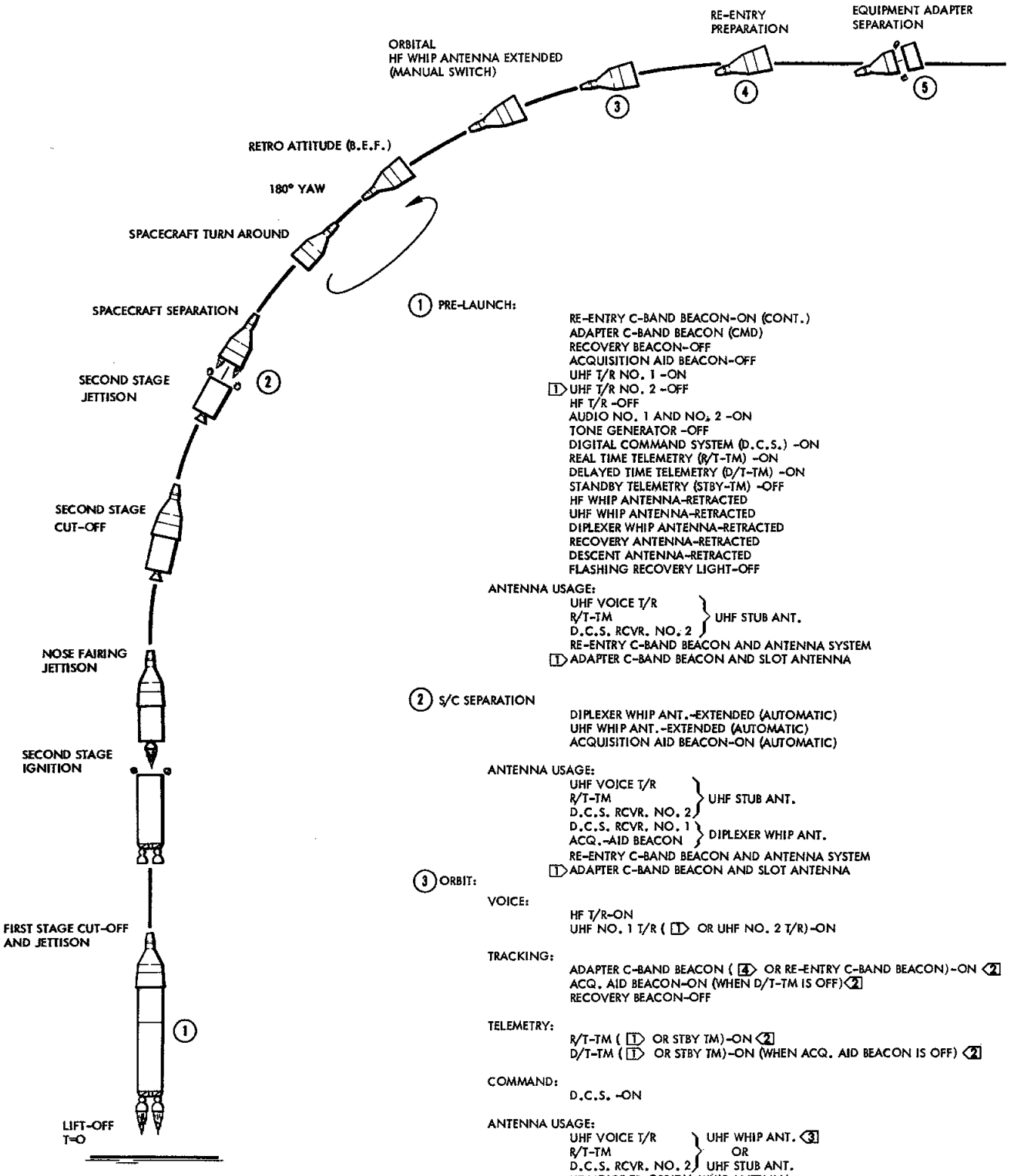
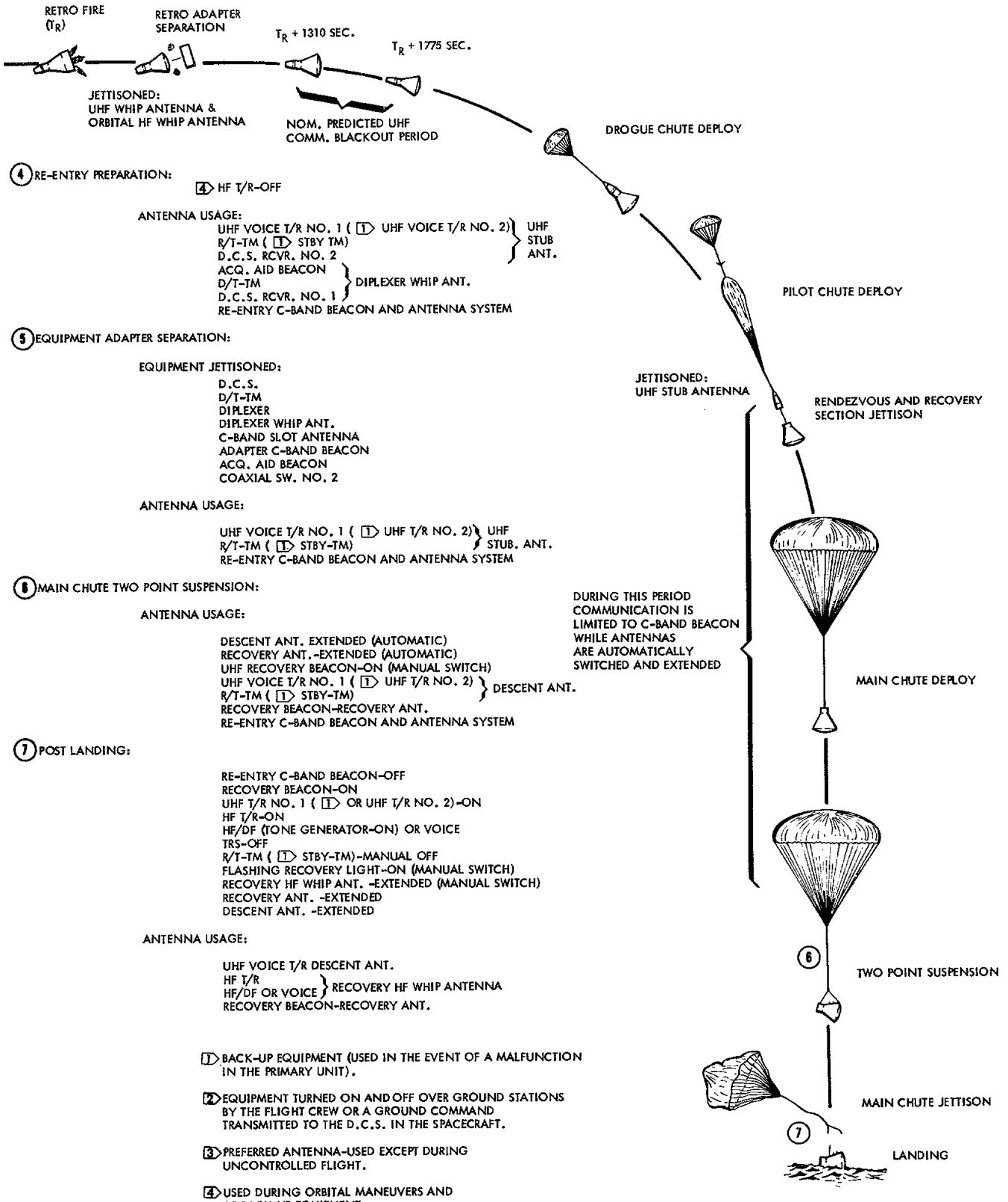


Figure 9-3 Communication System Sequential Diagram (Sheet 1 of 2)



(4) RE-ENTRY PREPARATION:

[4] HF T/R-OFF

ANTENNA USAGE:

UHF VOICE T/R NO. 1 ([1] UHF VOICE T/R NO. 2) } UHF STUB ANT.
R/T-TM ([1] STBY TM) } UHF STUB ANT.
D.C.S. RCVR. NO. 2 } UHF STUB ANT.
ACQ. AID BEACON } DIPLEXER WHIP ANT.
D/T-TM } DIPLEXER WHIP ANT.
D.C.S. RCVR. NO. 1 } DIPLEXER WHIP ANT.
RE-ENTRY C-BAND BEACON AND ANTENNA SYSTEM

(5) EQUIPMENT ADAPTER SEPARATION:

EQUIPMENT JETTISONED:

D.C.S.
D/T-TM
DIPLEXER
DIPLEXER WHIP ANT.
C-BAND SLOT ANTENNA
ADAPTER C-BAND BEACON
ACQ. AID BEACON
COAXIAL SW. NO. 2

ANTENNA USAGE:

UHF VOICE T/R NO. 1 ([1] UHF T/R NO. 2) } UHF STUB. ANT.
R/T-TM ([1] STBY-TM) } UHF STUB. ANT.
RE-ENTRY C-BAND BEACON AND ANTENNA SYSTEM

(6) MAIN CHUTE TWO POINT SUSPENSION:

ANTENNA USAGE:

DESCENT ANT. EXTENDED (AUTOMATIC)
RECOVERY ANT.-EXTENDED (AUTOMATIC)
UHF RECOVERY BEACON-ON (MANUAL SWITCH)
UHF VOICE T/R NO. 1 ([1] UHF T/R NO. 2) } DESCENT ANT.
R/T-TM ([1] STBY-TM) } DESCENT ANT.
RECOVERY BEACON-RECOVERY ANT.
RE-ENTRY C-BAND BEACON AND ANTENNA SYSTEM

(7) POST LANDING:

RE-ENTRY C-BAND BEACON-OFF
RECOVERY BEACON-ON
UHF T/R NO. 1 ([1] OR UHF T/R NO. 2)-ON
HF T/R-ON
HF/DF (TONE GENERATOR-ON) OR VOICE
TRS-OFF
R/T-TM ([1] STBY-TM)-MANUAL OFF
FLASHING RECOVERY LIGHT-ON (MANUAL SWITCH)
RECOVERY HF WHIP ANT. -EXTENDED (MANUAL SWITCH)
RECOVERY ANT. -EXTENDED
DESCENT ANT. -EXTENDED

ANTENNA USAGE:

UHF VOICE T/R DESCENT ANT.
HF T/R } RECOVERY HF WHIP ANTENNA
HF/DF OR VOICE } RECOVERY HF WHIP ANTENNA
RECOVERY BEACON-RECOVERY ANT.

[1] BACK-UP EQUIPMENT (USED IN THE EVENT OF A MALFUNCTION IN THE PRIMARY UNIT).

[2] EQUIPMENT TURNED ON AND OFF OVER GROUND STATIONS BY THE FLIGHT CREW OR A GROUND COMMAND TRANSMITTED TO THE D.C.S. IN THE SPACECRAFT.

[3] PREFERRED ANTENNA-USED EXCEPT DURING UNCONTROLLED FLIGHT.

[4] USED DURING ORBITAL MANEUVERS AND AS BACK-UP EQUIPMENT.

Figure 9-3 Communication System Sequential Diagram (Sheet 2 of 2)

VOICE COMMUNICATIONS

Voice communications is maintained by one hf and two uhf transmitter/receivers and the Voice Control Center (VCC). The VCC has all the necessary controls and switches required for various keying modes, transmitter/receiver selection, squelch, volume control, and voice recording. The hf voice transmitter/receiver may also be used for Direction Finding (DF) purposes during the postlanding phase of the mission.

An intercom connector is available for communication between the crew and recovery team, prior to opening the spacecraft hatches during the recovery phase of the mission. Lightweight headsets are supplied for use when the spacesuit helmets are removed during orbit, or during postlanding if the helmets or entire spacesuit is removed prior to recovery.

TELEMETRY TRANSMITTERS

Receiving inputs from the Pulse Code Modulated (PCM) programmer and the on-board tape recorder, three telemetry transmitters transmit vital spacecraft system parameters to the ground stations. The three transmitters operate on different frequencies and are identified as real-time, delayed-time, and stand-by transmitters. The stand-by transmitter is only used in case of real-time, or delayed-time transmitter failure.

FLASHING RECOVERY LIGHT

The flashing recovery light, used during the postlanding phase of the mission, contains its own power supply and improves visual spacecraft location.

DIGITAL COMMAND SYSTEM

The Digital Command System (DCS) is the command link between the ground and the

spacecraft. The DCS consists of two uhf receivers, a decoder, and two relay packages and is operational from pre-launch until adapter equipment section separation. Basically, the DCS receives and decodes two types of commands: a discrete or Real-Time Command (RTC) for spacecraft equipment utilization, and Stored Program Commands (SPC) that supply digital information to various spacecraft systems. Real-time commands operate DCS relays that control power directly or energize relays in the spacecraft Electrical System that determine equipment usage. Stored program commands are received and decoded for use by the Time Reference System (TRS), or the computer.

SYSTEM OPERATION

The Communication System is semi-automatic in operation. The sequence and theory of operation of the Communication System is described in the following paragraphs and referenced in Figures 9-2 and 9-3. Individual components are described in System Units.

VOICE TAPE RECORDER

Voice tape recordings are made during the mission by placing the RECORD switch on the VCC to the CONT or MOM position. The TONE VOX, AUDIO & UHF T/R 1 and 2 circuit breakers must be in the ON position. Each tape cartridge allows approximately one hour of recording time and is easily changed. An end-of-tape light on the voice recorder illuminates for two seconds when two minutes of recording time remains on the tape. The end-of-tape light will remain on when the end of the tape is reached. A digital timing signal is applied to one channel of the tape for time correlation of the voice recording.

PRE-LAUNCH

C-Band Radar Beacons

During pre-launch the BEACONS-C circuit breaker is placed to the ON position to arm the C-RNTY and C-ADPT BEACON CONTROL switches. The C-RNTY switch is placed in the CONT position during pre-launch to enable the re-entry C-band beacon to reply when properly interrogated by a ground station. The C-ADPT switch is placed in the CMD position during pre-launch. The CMD position enables the ground station during launch, to activate the adapter C-band beacon via a DCS channel if the need arises. After the adapter C-band beacon is activated, it will reply when properly interrogated by a ground station.

The C-band antenna system, used with the re-entry C-band beacon, is energized when the ANT SEL switch is placed in the RNTY position. The ANT SEL switch is armed when the COAX CNTL circuit breaker is positioned to ON. The ANT SEL switch controls application of power to the phase shifter power supply in the C-band antenna system.

UHF Transmitter/Receiver

The number 1 uhf voice transmitter/receiver will be utilized during pre-launch unless some malfunction occurs in which case the number 2 transmitter/receiver can be selected. For operation of either uhf voice transmitter/receiver, stand-by power is applied through the AUDIO & UHF T/R circuit breakers 1 and 2, which must be in the ON position. The selected transmitter/receiver will be powered by placing the UHF select switch to the number 1 or number 2 position and the MODE switch of number 2 AUDIO to the UHF position. The UHF select switch also controls coaxial switch 1 which connects the uhf transmitter/receiver to the quadriplexer. Coaxial switch power is obtained from the common control bus

through the ON position of the UHF RELAY circuit breaker. The method of keying the uhf transmitter/receiver is selected by positioning the KEYING switch on the VCC to VOX (voice operated relay), PTT (push-to-talk), or CONT INT/PTT (continuous intercom/push-to-talk transmitter keying).

The desired antenna usage is obtained by placing the ANT CNTL circuit breaker to the ON position. This places coaxial switch 3 to position 1; thus connecting the quadriplexer to the IN position of the coaxial switch 5. Coaxial switch 5 was placed in position 1 when the ANT SEL switch was placed in the RNTY position during the C-band beacon operation. With coaxial switch 5 in position 1, the uhf stub antenna is available for uhf voice transmission and reception. Prior to umbilical release, voice communication is maintained between the spacecraft and the ground complex through a hardline using the headset and microphone amplifiers of the VCC. After umbilical release, voice transmission to the ground complex is accomplished by means of the uhf voice transmitter/receiver.

Real-Time Telemetry Transmitter

The real-time telemetry transmitter will be operating during the pre-launch phase of the mission. The real-time telemetry transmitter is powered by placing the RT XMTR circuit breaker in the ON position and placing the TM CONTROL switch to the R/T & ACQ position.

The real-time telemetry transmitter uses the uhf stub antenna via the quadriplexer and coaxial switches 3 and 5, the same as the uhf transmitter/receiver.

In case of real-time telemetry transmitter failure, the stand-by telemetry transmitter may be used for real-time transmission. The STBY XMTR CNTL and PWR

circuit breakers must be in the ON position to operate the stand-by telemetry transmitter. If selection is made by the crew, the STBY TM CONTROL switch is placed to the R/T position. Selection can be made by a ground command via the DCS when the TM CONTROL switch is in the OFF position. When operating as the real-time telemetry transmitter, the stand-by transmitter uses the stub-antenna for transmission.

Non-Operation Components

The following Communication System components will be non-operational during the pre-launch phase of the mission. To assure the off condition of these components, the following switches should be in the position specified below:

HF (on VCC)	OFF
BEACON CONTROL - RESC	OFF
HF ANT	OFF

To assure proper sequential actuation of the various communication components, the following circuit breakers (in addition to those previously described) must be placed to the position listed prior to launch:

WHIP ANTENNAS	-	HF
WHIP ANTENNAS	-	UHF
WHIP ANTENNAS	-	DIPLEX
HF T/R	-	ON
BEACONS	-	ACQ
BEACONS	-	RESC
XMTRS	-	DT
TAPE RCDR	-	CNTL

SPACECRAFT/LAUNCH VEHICLE SEPARATION

Equipment usage after spacecraft/launch vehicle separation is identical to that described under Pre-Launch except for the following: Upon closure of any two of the three spacecraft separation sensors the acquisition aid beacon is energized. The uhf whip antenna solenoid actuators are powered and release the latch mechanism of the uhf whip antennas, allowing them to self extend.

The acquisition aid beacon transmits via the diplexer and uhf whip antenna on the adapter equipment section. Placing the TAPE RCDR-CNTL circuit breaker to ON and the TM CONTROL switch to R/T & ACQ during pre-launch places coaxial switch 2 in position 1 which connects the acquisition aid beacon to the diplexer.

ORBIT

During orbit, operation of the telemetry transmitters and beacons will normally be controlled by ground commands via DCS channels. To operate from ground commands the C-ADPT, C-RNTY and T/M CONTROL switches must be in the CMD position.

HF Voice Transmitter/Receiver

During orbit hf communications is via the hf whip antenna on the adapter retrograde section. At insertion the adapter hf whip is extended by placing the LANDING switch to the SAFE position, and the HF ANT switch to the EXT position. This will place coaxial switch 4 in position 2 and allow hf voice transmission and reception via the adapter hf whip. After extension (approximately one minute) the HF ANT switch is returned to the OFF position.

Stand-by power is applied to the hf transmitter/receiver by the HF T/R circuit breaker which was positioned to ON during pre-launch. The hf transmitter/receiver

is powered by positioning the HF select switch (on the VCC) to RNTY and audio MODE switch 1 or 2 to the HF position. The method of keying the hf transmitter/receiver is selected by positioning the KEYING switch on the VCC to VOX, PTT, or CONT INT/PTT. During orbit, any of the three keying modes may be selected.

UHF Voice Transmitter/Receiver

The uhf voice transmitter/receiver operation is identical to that described under Pre-Launch with the following exception. Preferred antenna usage during orbit for uhf transmission and reception is via the adapter retrograde uhf whip antenna. The retrograde uhf whip antenna is selected by placing the ANT SEL switch to the ADPT position which places coaxial switch 5 to position 2. Although preferred uhf transmission and reception is via the retrograde uhf whip antenna, the uhf stub antenna may be used during orbit by placing the ANT SEL switch to the RNTY position.

Delay Time Telemetry Transmitter

The acquisition aid beacon operates continuously during the orbital phase of the mission except when the delayed-time telemetry transmitter is operating. When the ground station receives the acquisition aid beacon signal, it initiates a DCS command for the delayed-time telemetry transmitter to transmit data stored by the on-board recorder while the spacecraft was between ground stations.

Delayed-time transmission may also be initiated by placing the T/M CONTROL switch to the R/T-D/T position. This will initiate real-time as well as delayed-time telemetry transmission.

Real-time and delayed-time transmission will normally be initiated from the ground station via DCS channels. At the time the delayed-time telemetry transmitter is selected, the acquisition aid beacon is turned off and coaxial switch 2 is placed in position 2, allowing telemetry transmission via the diplexer and uhf whip antenna on the adapter equipment section.

As the spacecraft goes out of range the delayed-time telemetry transmitter is turned off and the acquisition aid beacon resumes transmission. This is normally performed by the ground station but may be accomplished by placing the T/M CONTROL switch to the CMD, or the R/T & ACQ position. If the R/T & ACQ position is selected, the delayed-time transmitter is turned off and the real-time transmitter and the acquisition aid beacon begin transmitting. If the CMD position is selected, only the acquisition aid beacon will operate; however, the ground station has the capability of energizing the real-time telemetry transmitter via a DCS channel.

Any of the three previously described methods of disabling the delayed-time telemetry transmitter will place coaxial switch 2 to position 1, and allow acquisition aid beacon transmission via the diplexer and uhf whip antenna.

The stand-by telemetry transmitter may be used for delayed-time transmission should failure of the delayed-time telemetry transmitter occur. The stand-by transmitter is switched to delayed-time transmission by a ground command via a DCS channel (if the STBY TM CONTROL switch is in the OFF position), or by placing the STBY TM CONTROL switch to the D/T position. Delayed-time transmission via the stand-by telemetry transmitter uses the uhf stub or the uhf whip antenna on the retrograde adapter depending upon the setting of the ANT SEL switch.

Real-Time Telemetry Transmitter

Orbital operation of the real-time telemetry transmitter is similar to that of the delayed-time telemetry transmitter in that the real-time telemetry transmitter is operated only during the period that the spacecraft is within range of a ground station. The real-time telemetry transmitter is turned on by a DCS command from the ground station or by placing the T/M CONTROL switch to the R/T & ACQ or R/T-D/T position. Real-time transmission is by the uhf stub or the retrograde uhf whip antenna, depending upon the position of the ANT SEL switch. In case of failure of the real-time telemetry transmitter, the stand-by transmitter may be used for real-time transmission. The stand-by transmitter is switched to real-time transmission by a ground command via a DCS channel (if the STBY TM CONTROL switch is in the OFF position), or by placing the STBY TM CONTROL switch to the R/T position. The stand-by telemetry transmitter transmits via the uhf stub or the retrograde uhf whip antenna, depending upon the position of the ANT SEL switch.

It should be noted that the stand-by telemetry transmitter may be used for delayed-time, or real-time transmission, but may not be used simultaneously for both. In the event that both the real-time and delayed-time transmitters fail, it is up to the ground station to determine the purpose for which the stand-by transmitter will be used.

C-Band Radar Beacons

During orbit, the C-band beacons are used only while the spacecraft is within range of a ground station. Normally, the adapter C-band beacon will be used during stabilized flight and the re-entry C-band beacon used during roll maneuvers.

Operation of the beacons is similar to that described under Pre-Launch. The C-RNTY and C-ADPT BEACON CONTROL switches are normally kept in the CMD position. When the spacecraft comes within range of a ground station, as determined from the acquisition aid beacon signal, power to the desired C-band beacon is applied by ground command via a DCS channel. The desired beacon may also be selected by placing the C-ADPT or C-RNTY BEACON CONTROL switch to the CONT position. After power is applied, the selected C-band beacon will transpond when properly interrogated by a ground station. When the re-entry C-band beacon is selected, the ANT SEL switch should be placed in the RNTY position to energize the phase shifter and provide optimum radiation coverage.

ADAPTER SEPARATION

Prior to adapter equipment section separation, the re-entry module antennas are selected by placing the T/M CONTROL switch to R/T & ACQ, the ANT SEL switch to RNTY, and the C-RNTY BEACON CONTROL switch to the CONT position. Transmission and reception during re-entry is via the C-band antenna system and the uhf stub antenna. The acquisition aid beacon will operate until it is jettisoned with the adapter equipment section. The hf voice communications is disabled by placing the HF select switch to the OFF position. On spacecraft 5, the hf whip on the adapter retrograde section will remain extended. On later spacecraft, the hf whip may be retracted by holding the HF ANT switch in the RET position for approximately 1.5 minutes for complete retraction.

The following communications components will be jettisoned with the adapter section:

Digital Command System (DCS)

Delayed-time telemetry transmitter

Diplexer

C-band annular slot antenna

Adapter C-band radar beacon

Diplexer uhf whip antenna

Acquisition aid beacon

Coaxial switch 2

This limits telemetry data transmission to real-time, voice communication to uhf, and tracking data to the re-entry C-band beacon.

Following equipment section separation and retro firing, retrograde section separation will occur at which time the retrograde uhf whip and the adapter hf whip antennas will be jettisoned.

RE-ENTRY

During the re-entry phase of the mission, two short duration communication blackout periods occur. The first period, from approximately 1310 seconds after retrofire time (T_R) to 1775 seconds after T_R, is caused by an ionization shield around the spacecraft. This ionization is due to the extremely high temperatures created upon re-entry into the earths atmosphere. The second blackout period occurs at Rendezvous and Recovery (R & R) section separation when the uhf stub antenna is jettisoned. This period is terminated at two-point suspension which occurs shortly after main parachute deployment.

At R & R separation, energized parachute deploy time delay relays energize coaxial switch 3, placing it to position 2. This makes the uhf descent antenna available for real-time telemetry transmission and uhf voice communications.

At two-point suspension the uhf recovery and uhf descent antennas are automatically extended. The uhf recovery beacon is turned on by placing the RESC BEACON CONTROL switch to the W/O LT position.

Antenna usage during re-entry is as follows: Prior to R & R separation, real-time telemetry transmission and uhf voice communication is via the uhf stub antenna. After two-point suspension, the uhf descent antenna is used instead of the uhf stub. The re-entry C-band beacon and C-band antenna system is used for tracking and the uhf recovery beacon will use the uhf recovery antenna.

LANDING THROUGH RECOVERY

Upon impact the main parachute is jettisoned by actuating the PARA JETT switch. This extends the flashing recovery light. The light is energized by changing the RESC BEACON CONTROL switch from the W/O LT position to the ON position.

The re-entry C-band beacon and real-time telemetry transmitter is turned off by placing the C-RNTY BEACON CONTROL and the T/M CONTROL switch to the CMD position. If the stand-by telemetry transmitter was selected for real-time transmission, the stand-by transmitter will be turned off by placing the STBY TM CONTROL switch to the OFF position.

The recovery hf whip antenna is extended by placing the HF ANT switch to the PST LDG position for spacecraft 5, or on later spacecraft by holding the HF ANT switch in the EXT position for approximately one minute. Voice communication via the hf transmitter/receiver is then possible by placing the HF select switch to the RNTY position and either MODE switch to HF. The hf transmitter/receiver can also be used to transmit a direction finding signal by placing either MODE switch to HF/DF.

During the recovery phase of the mission the uhf rescue beacon transceiver may be connected to the uhf recovery antenna. The uhf recovery beacon can be turned off by positioning the RESC BEACON CONTROL switch to OFF. Lightweight headsets are provided to replace the spacesuit helmets if the helmets or spacesuits are removed and the crew remains inside the spacecraft. A recovery team disconnect is used for intercom conversation between the crew and recovery team prior to opening the spacecraft hatches.

SYSTEM UNITS

ANTENNAS

UHF Descent and UHF Recovery Antennas

Purpose: The uhf descent antenna is used for simultaneous transmission of the real-time and stand-by telemetry transmitters, and transmission and reception for the uhf voice transmitter/receiver. The uhf recovery antenna provides transmission capability for the uhf recovery beacon. The two antennas are used from two-point suspension of the main parachute through final recovery of the spacecraft.

Physical Characteristics: The two antennas, being similar in physical appearance, are shown in Figure 9-4. Both antennas are mounted in the parachute cable trough where they are stowed until main parachute two point suspension during the landing phase of the mission.

The antenna element consists of two one-half inch wide gold plated steel blades bolted together at two places. The uhf descent antenna is approximately 16 inches long. The uhf recovery antenna is approximately 18 inches long.

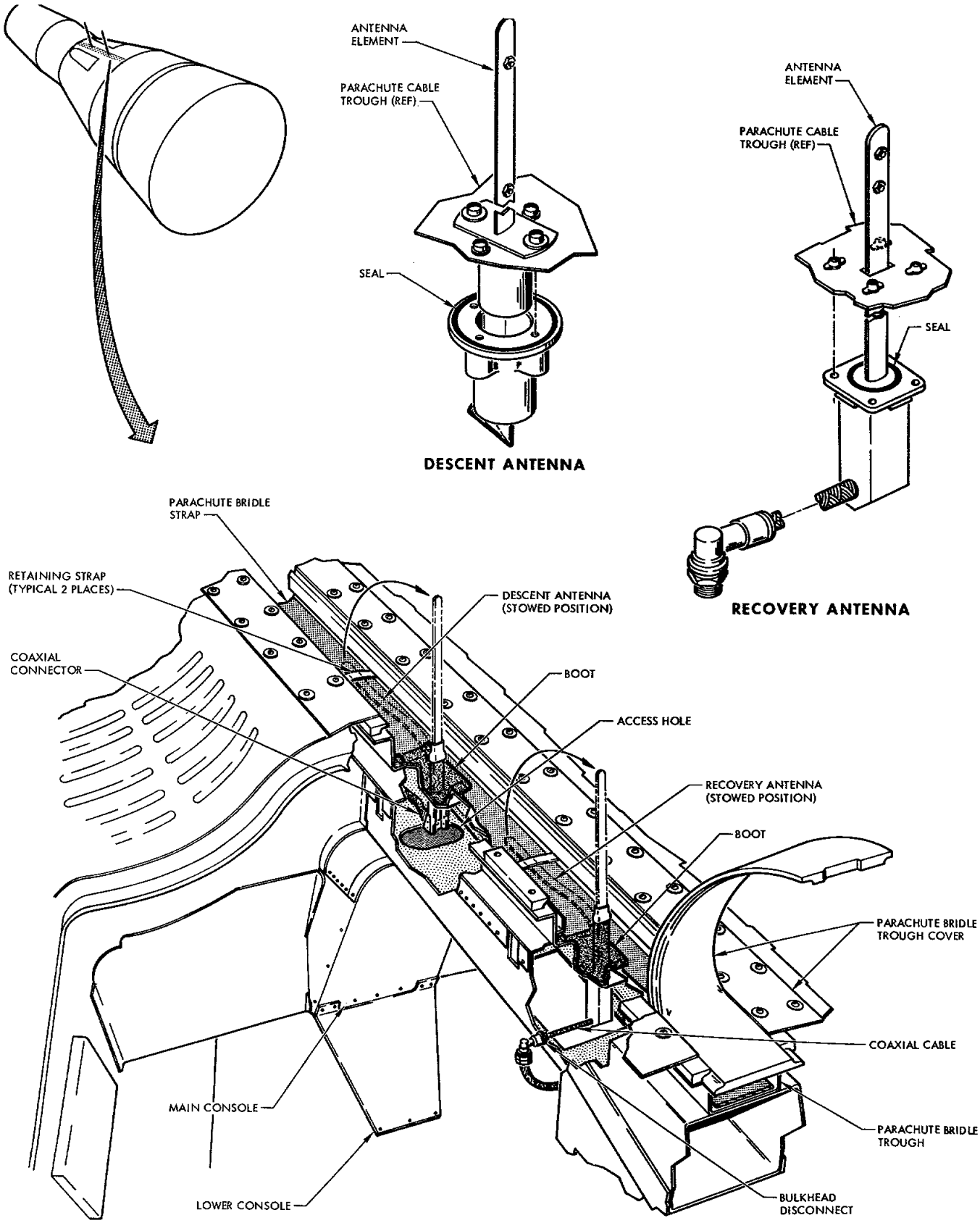


Figure 9-4 UHF Descent and Recovery Antennas

Mechanical Characteristics: For rigidity, the antenna element is shaped in a 0.5 inch wide arc having a radius of 1.5 inches. The two laminations of steel blades, compounding a single antenna element, are rigidly secured at the lower half of the antenna. To allow a slight displacement of the two laminations with respect to each other during stowage and deployment, two nuts and bolts placed through elongated holes secure the two laminations together at the upper half of the antenna element.

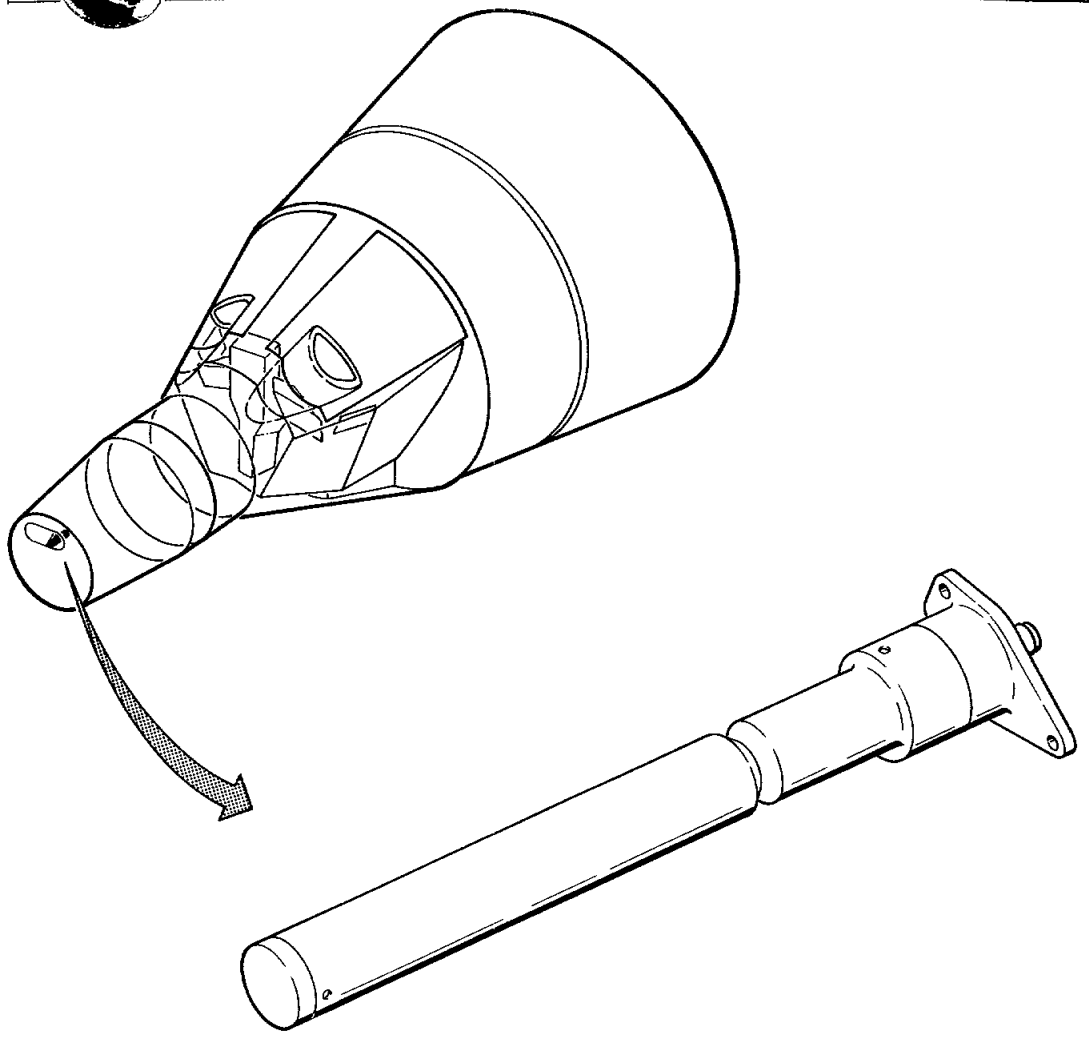
The antennas are bent towards the small end of the spacecraft for stowage and are held in place by a retaining strap. The strap is broken when the Landing System shifts from single point to two point suspension, allowing the antennas to extend.

Each of the two antennas have a radiation pattern which is identical to that of a quarter wave stub.

UHF Stub Antenna

Purpose: The uhf stub antenna (Figure 9-5) allows simultaneous transmission of the real-time and stand-by telemetry transmitters, transmission and reception for the uhf voice transmitter/receivers, and reception for DCS receiver number 2. The antenna may be used from pre-launch until separation of the R & R section during re-entry, but is normally used from pre-launch to insertion and from re-entry preparation to R & R section separation.

Physical Characteristics: The uhf stub antenna, physically constructed as illustrated in Figure 9-5, is mounted in the nose of the R & R section. The antenna protrudes forward from the R & R section and is covered by a nose fairing during the boost phase of the mission. The antenna consists of a mast and base which weighs approximately 1.1 pounds. The mast is constructed of 3/4 inch cobalt steel, machined to tubular form, and covered by a Teflon ablation shield for



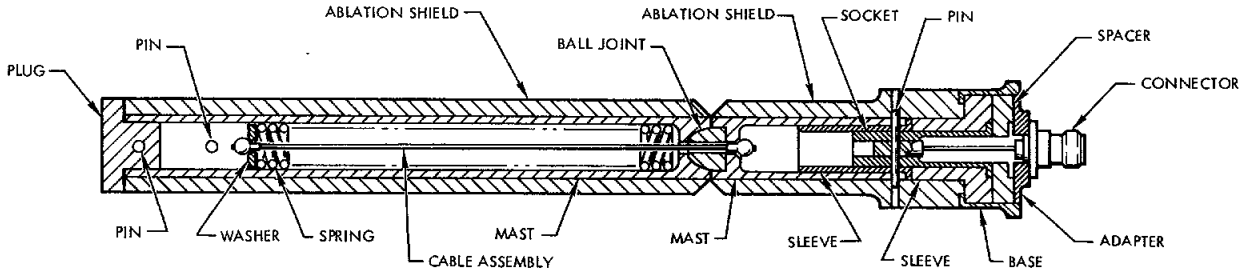


Figure 9-5 UHF Stub Antenna

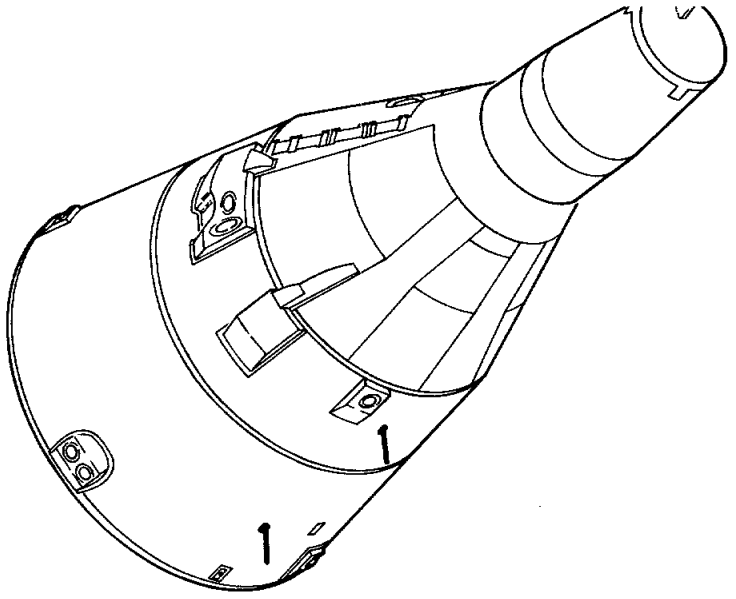
protection during re-entry. The antenna is approximately 13.5 inches long including the connector, and 1.25 inches in diameter over the ablation material. The mast consists of two sections. The front section is mounted on a cobalt steel ball joint and retained to the rear section by a spring loaded cable. Electrical contact between the mast sections is made through the ball joint and the spring loaded cable assembly. The ball joint allows the front section of mast to be deflected to approximately 90 degrees in any direction around the antenna axis. The spring of the cable assembly is pre-loaded to approximately 45 pounds to cause the front section, when deflected, to return to the erected position.

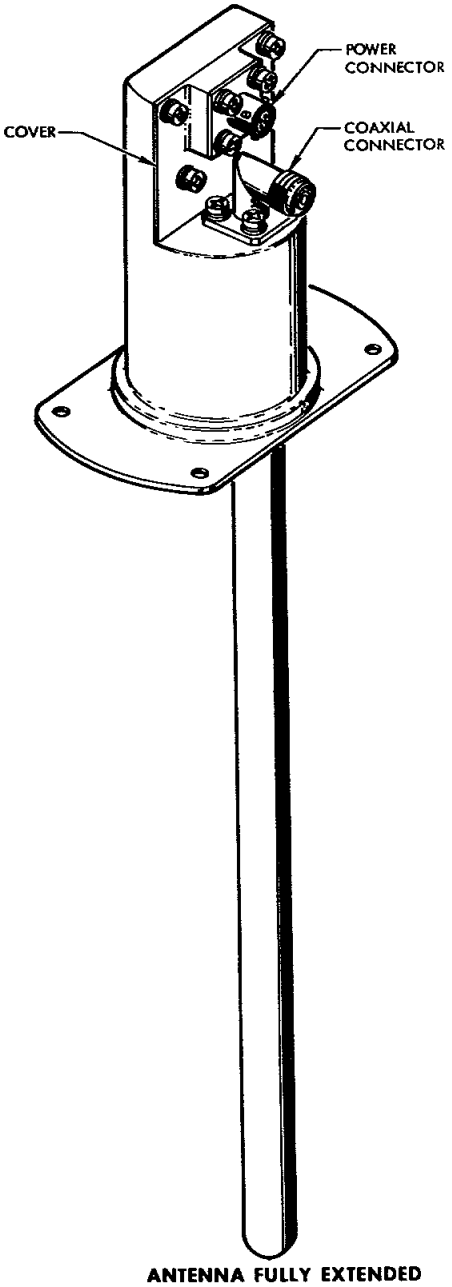
The rf connector is press fitted into a socket and makes contact to the mast through the socket and sleeve, which are the same material as the mast. The shell of the rf connector is mounted to the base which is isolated from the mast by a Teflon spacer and sleeve.

Mechanical Characteristics: The uhf stub is a quarter wave length antenna. The radiating length of the antenna is approximately 11.2 inches.

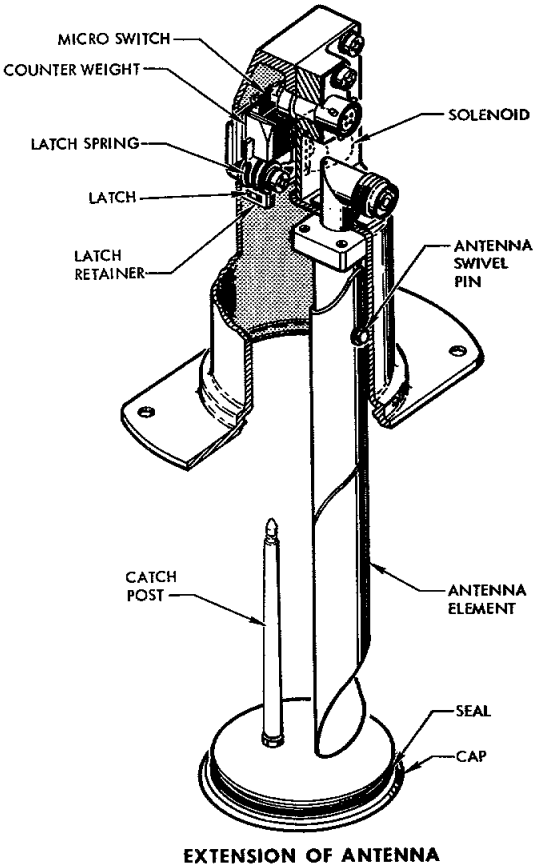
UHF Whip Antennas

Purpose: Two identical uhf whip antennas (Figure 9-6) supply the required uhf transmission and reception facilities during orbit. One of the uhf antennas is located on the adapter equipment section and serves the DCS receiver number 1, and the acquisition aid beacon or delayed-time telemetry transmitter. The second uhf antenna, mounted on the adapter retrograde section, serves the real-time and standby telemetry transmitters, the uhf voice transmitter/receivers, and DCS receiver number 2.

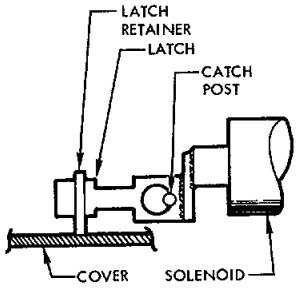




ANTENNA FULLY EXTENDED



EXTENSION OF ANTENNA



RELEASE MECHANISM
(SHOWN IN LATCHED CONDITION)

Figure 9-6 UHF Whip Antennas

Physical Characteristics: The uhf whip antenna is self extendable and requires no power other than that required for initial release. The antenna element is a tubular device made from a 2 inch wide beryllium copper strip processed in the form of a tube. The antenna, when fully extended, forms an element that is approximately 12 inches long and 1/2 inch in diameter. During stowage, the tube is opened flat, wound inside of a retaining drum, and latched in position. Upon release of the latch by a solenoid, the extension of the antenna depends entirely on the energy stored in the rolled strip material. This energy is sufficient to erect the antenna at a rate of 5 feet/second into its tubular form. In the stored condition, the antenna is flush with the outer skin of the spacecraft.

Mechanical Characteristics: The antenna element is retained inside the housing by a metal lid. A metal post is attached to the lid and passes through the center of the coiled antenna. The bottom of the post is grooved to accept a forked latch which holds the catch post assembly firmly in position prior to release. The forked latch is attached to a miniature pull-solenoid which is spring loaded in the extended position to ensure that launch shock and vibration loads will not cause inadvertent antenna extension. When a voltage from the Sequence System is applied to the antenna solenoid, the latch will be withdrawn allowing the antenna cap to eject and the antenna to extend. As the catch post assembly is ejected, a microswitch in series with the solenoid coil opens the circuit to the coil to prevent further current drain from the power source.

The two antennas are jettisoned with the corresponding adapter section.

HF Whip Antennas

Purpose: The hf whip antennas provide transmission and reception for the hf voice transmitter/receiver during the orbital and postlanding phases of the mission.

Physical Characteristics: The hf whip antennas are physically constructed as illustrated in Figure 9-7. The recovery hf whip antenna is mounted on the small pressure bulkhead, outside the pressurized area of the spacecraft re-entry module. The other hf whip antenna is located on the adapter retrograde section. The antenna mechanism housing, approximately 6.25 inches wide and 22.4 inches high, completely encloses all parts of the antenna, including storage space for the antenna elements.

The recovery hf whip antenna contains six elements which, when fully extended, comprise a single antenna mast approximately 13 feet 3 inches long. The adapter hf whip antenna contains three elements which, when fully extended, comprise a single antenna mast approximately 16 feet long on spacecraft 5 and 6, and approximately 13 feet long on later spacecraft. The mast is one inch in diameter on all spacecraft. Two connectors, supported by the antenna body, provide a means of applying power and connecting the antenna to the rf connector on the hf voice transmitter/receiver. The recovery hf whip antenna weighs approximately 9.0 pounds. The 16-foot version of the adapter hf whip antenna weighs approximately 7.5 pounds and the 13-foot version 6.0 pounds. The main supporting structure of the antenna mechanism housing is the antenna body consisting of a thin fiberglass shell.

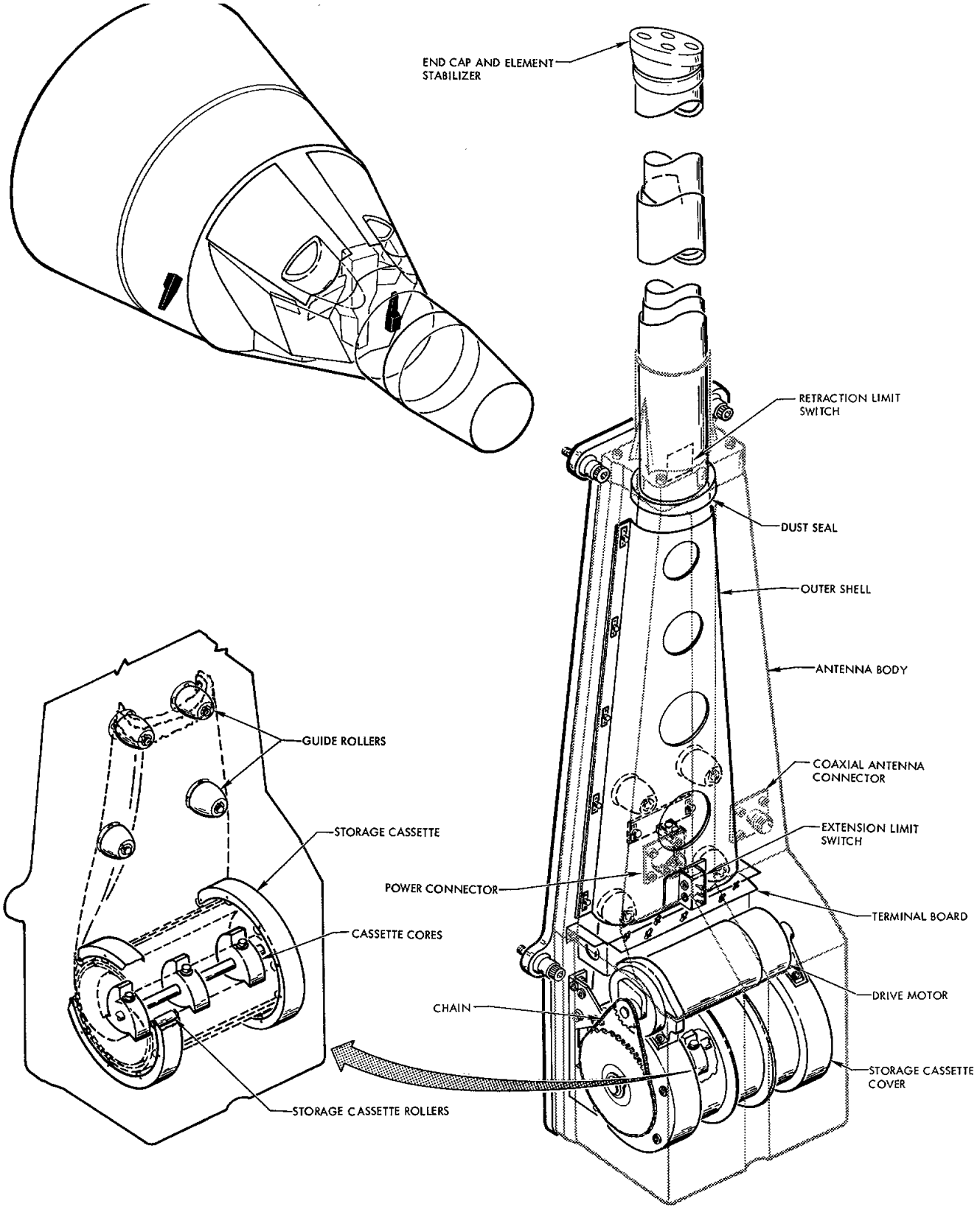


Figure 9-7 HF Whip Antennas

The outer shell is made in two sections which mate together and form a completely sealed envelope around all moving parts. The antenna mast elements are heat treated stainless steel strips and are stored in a dc motor driven cassette.

Mechanical Characteristics: The strip material comprising the antenna elements is heat treated into a material circular section in such a manner that the edges of the material overlap approximately 180 degrees. When the antenna is retracted, the tubular elements are continuously transformed by guide rollers into a flattened condition, and stored in a strained manner in a cassette. Extension and retraction of the antenna is accomplished by a motor which, by means of a chain, drives the storage cassette core. Because of the natural physical shape of the antenna elements, the antenna has a tendency to self-extend; thus giving an extension time of approximately 25 seconds. Retraction time is approximately 40 seconds. The antenna is stopped within its desired limits by two microswitches, one for extension and one for retraction, which automatically cut the power applied to the motor at the time of extreme limits of the antenna are reached.

The rf connection to the antenna is obtained by a wiper arm sliding on the cassette core drive shaft.

On spacecraft 5, the hf whip antennas are operated as follows: Spacecraft control bus voltage is supplied through the WHIP ANTENNAS -HF circuit breaker to the HF ANT switch. The adapter hf whip antenna is extended during orbit by positioning the HF ANT switch to EXT. The adapter hf whip antenna is not retracted during

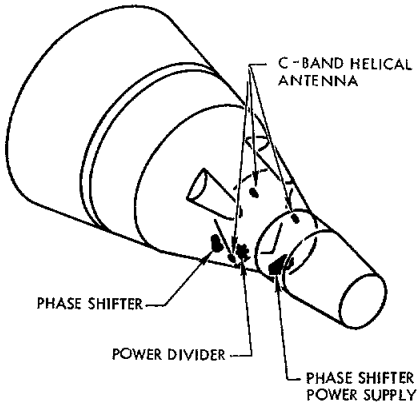
orbit, but is jettisoned in the extended position with the retrograde section. After landing, the recovery hf whip antenna is extended by positioning the HF ANT switch to PST LDG, and is retracted by positioning the HF ANT switch to EXT.

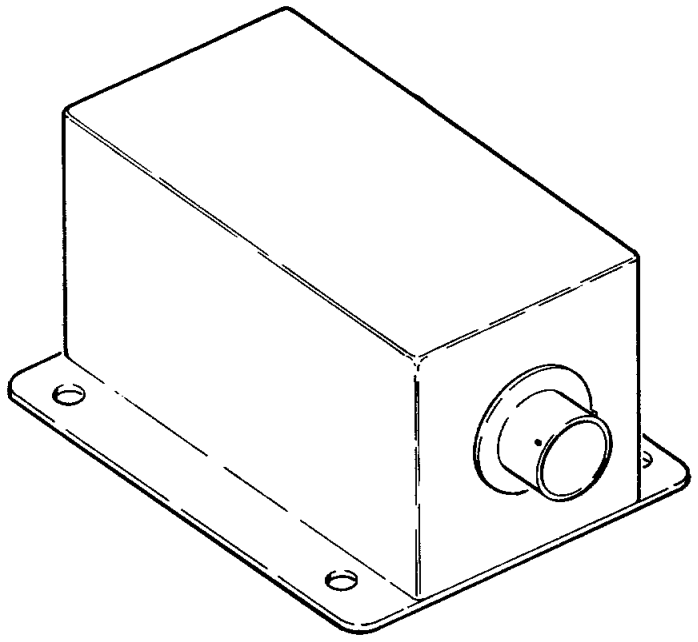
On spacecraft 6 through 12, extension of the hf whip antennas is controlled through the HF ANT switch and LANDING switch. The hf antennas are operated as follows: Spacecraft control bus voltage is supplied through the WHIP ANTENNAS - HF circuit breaker to the HF ANT switch, which has momentary type contacts. During orbit, the LANDING switch is in the SAFE position and adapter hf whip antenna can be extended or retracted by holding the HF ANT switch in the EXT or RET position respectively. During re-entry, the LANDING switch is placed in the ARM position. After landing, the recovery hf whip antenna can be extended or retracted by holding the HF ANT switch in the EXT or RET position respectively. The HF ANT switch should be held in the EXT position for approximately one minute for full extension of the antennas, and in the RET position for approximately 1.5 minutes for full retraction.

C-Band Annular Slot Antenna

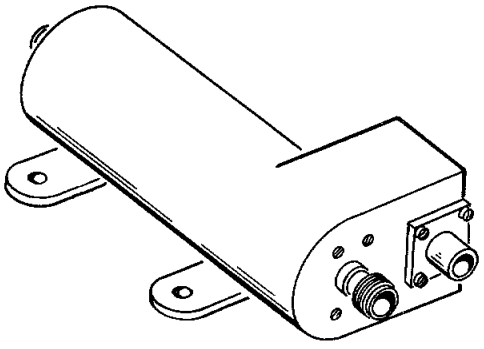
Purpose: The C-band annular slot antenna (Figure 9-8) serves the adapter C-band radar beacon and is normally used during stabilized flight.

Physical Characteristics: The C-band annular slot antenna is mounted on the equipment section of the adapter. The physical construction is such that the antenna is flush with the outer skin of the spacecraft. The antenna is approximately 1.4 inches in diameter, 1.35 inches long and weighs 8 ounces maximum.

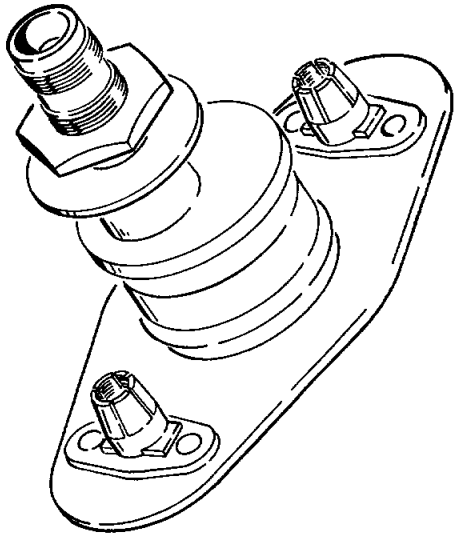




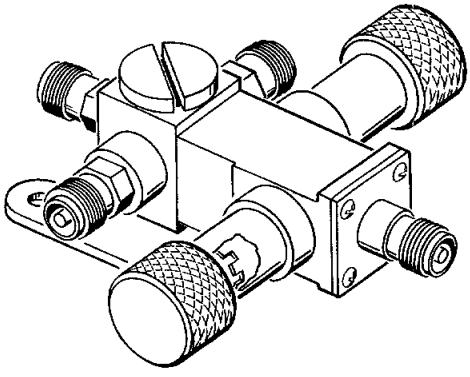
PHASE SHIFTER POWER SUPPLY



PHASE SHIFTER



C-BAND HELICAL ANTENNA



POWER DIVIDER

Figure 9-8 C-Band Antenna System

Mechanical Characteristics: The antenna radiation pattern is identical to that of a quarter wave stub on a ground plane. The antenna is used for both reception and transmission of the adapter C-band radar beacon during the orbital phase of the mission. The antenna is jettisoned with the equipment section of the adapter.

C-Band Antenna System

Purpose: The C-band antenna system, consisting of a power divider, a phase shifter, and three helical antennas, provides transmission and reception capability for the re-entry C-band radar beacon. The power divider supplies equal transmission power to the three helical antennas. A phase shifter is in series with one of the antennas to compensate for areas of low or no radiation coverage between lobes of the three individual radiation patterns. A phase shifter power supply supplies the phase shifter with 26 vac 453 cps power. The antenna system gives the circular radiation pattern around the spacecraft longitudinal axis required for ascent, descent and roll spacecraft attitudes.

Physical Characteristics: The power divider, phase shifter, phase shifter power supply, and helical antennas are shown in Figure 9-8. The power divider, phase shifter, and phase shifter power supply are mounted on the small pressure bulkhead, outside the pressurized area of the spacecraft. The power divider measures approximately 3.86 inches over the connectors, 4.0 inches over the tuning knobs and weighs approximately 6.5 ounces. The phase shifter is approximately 5.8 inches long, 2.84 inches wide at the large end, 1.4 inches high, has a diameter at the small end of about 1.5 inches, and weighs approximately 12 ounces. The phase shifter power supply measures approximately 1.5 inches wide, 1.75 inches high, 3.5 inches long over the connector, and weighs approximately 8 ounces.

The three C-band helical antennas are mounted flush with the outside skin of the spacecraft and spaced approximately 120 degrees apart. Each antenna unit is approximately 3.4 inches long, 1.8 inches wide, has a depth of 2.21 inches over the connector and weighs approximately 3.5 ounces.

Electrical Characteristics: The power divider, phase shifter, and helical antennas comprise an antenna system that satisfies the transmission and reception requirements for the re-entry C-band radar beacon during the launch and re-entry phases of the mission.

The power divider is basically a cavity type power splitter. During beacon transmission, power is delivered to the power divider where it is divided equally among the C-band helical antennas. The power divider compensates for loss of power due to the phase shifter in series with the right antenna. The power divider contains a double stub tuner to compensate for mismatch between the re-entry C-band beacon, the C-band helical antennas, and the phase shifter. Tuning is accomplished by means of a self-locking tuning shell located underneath each tuning stub cap.

The phase shifter has its own ac power supply. The input to the phase shifter, is half wave rectified and applied across a coil wound around a ferrite material. Due to the characteristics of the ferrite material, the rf signal from the power divider is delayed 0 to 180 degrees $\pm$ 20 degrees at the rate of 453 cycles per second. The changing phase shift of the rf power on one of the C-band helical antennas with respect to the other two, shifts the lobe of that antenna by approximately $\pm$ 45 degrees; thus giving the effect of an almost ideal circular radiation pattern around the longitudinal axis of the spacecraft. The combination of the three antenna elements gives a radiation pattern which extends in all directions except forward and aft of the spacecraft.

The phase shifter power supply is a dc-ac inverter which supplies a nominal 26 vac, 453 cps power to operate the phase shifter. The power supply is a hermetically sealed solid-state unit consisting of a voltage regulator, single-stage oscillator, buffer stage, and a push-pull output stage with transformer coupled output. The power supply provides a minimum output of 21 volts rms at 453 ± 17 cps with an input voltage range from 20 to 30 vdc. Input voltage is applied from the spacecraft main bus via the BEACON-C circuit breaker, C-RNTY BEACON CONTROL switch and the RNTY position of the ANT SEL switch. Maximum input current is 370 milliamperes.

Multiplexers (UHF Diplexer and UHF Quadriplexer)

Purpose: The uhf diplexer provides isolation between DCS receiver number 1, and the acquisition aid beacon or the delayed-time telemetry transmitter operating into a common antenna. The uhf quadriplexer provides isolation between the standby telemetry transmitter, the real-time telemetry transmitter, a uhf voice transmitter/receiver, and DCS receiver number 2 operating into a common antenna via coaxial switches.

Physical Characteristics: The physical representation and approximate location of the uhf diplexer and the uhf quadriplexer is shown in Figure 9-9. The diplexer is located on the electronic module of the adapter equipment section. The quadriplexer is located forward of the small pressurized bulkhead outside the pressurized area of the cabin.

The diplexer is approximately 4.5 inches wide, 4 inches high, and 2.7 inches deep; contains two input and one output connectors, and weighs approximately 1.25 pounds. The uhf quadriplexer is approximately 5.75 inches wide, 5.5 inches deep, and 4.1 inches high; weighs approximately 2.75 pounds, and has four input and one output connectors.

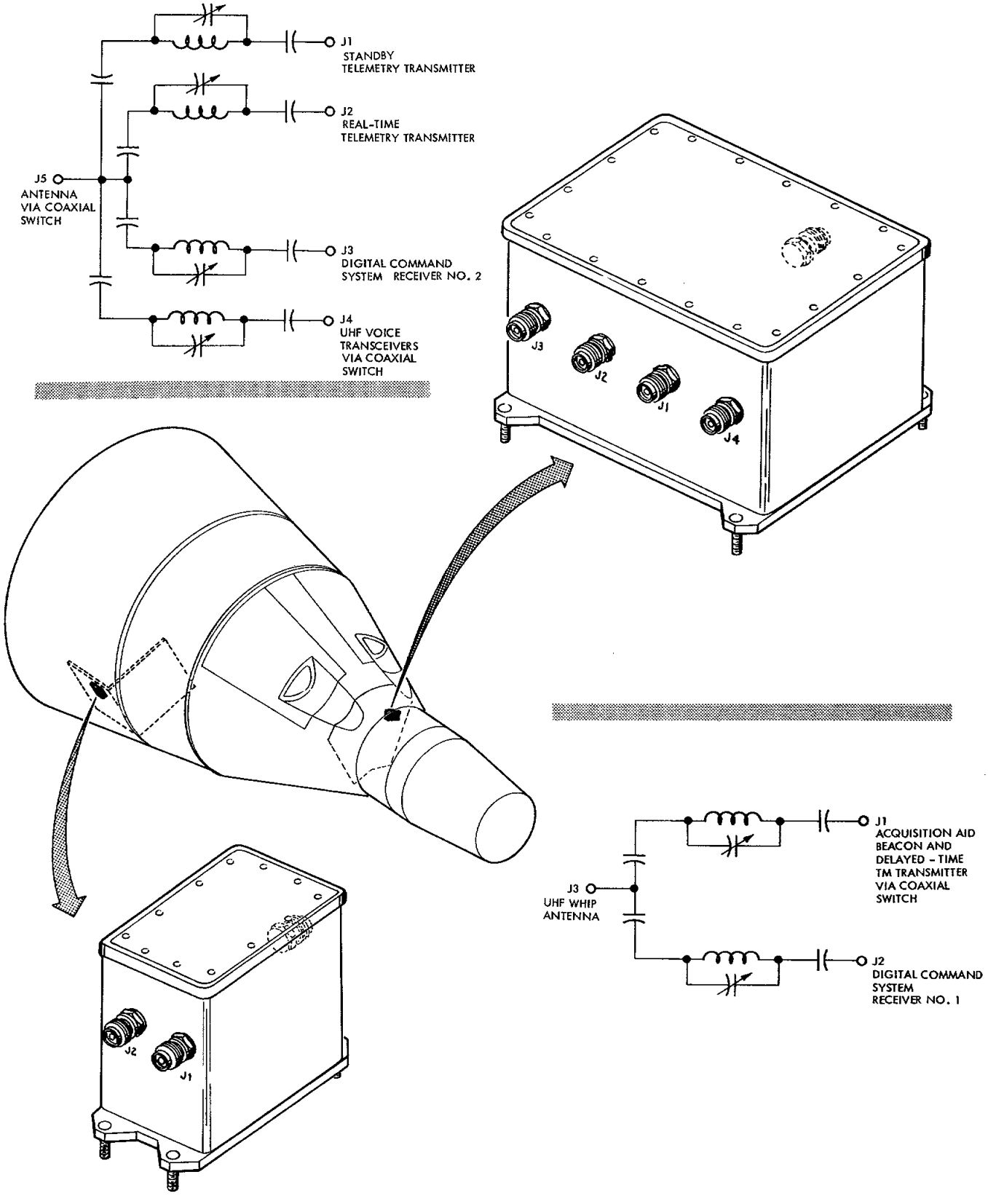


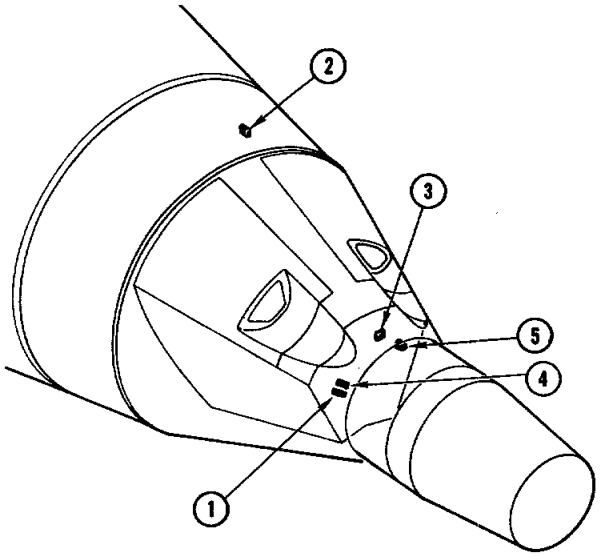
Figure 9-9 UHF Diplexer/UHF Quadriplexer

Electrical Characteristics: Figure 9-9 shows the schematic of the uhf diplexer and the uhf quadriplexer. Each channel consists of a high Q cavity, tuned to the corresponding operating frequency. All channels are isolated from each other without appreciably attenuating the rf signals passing through it. Each channel can be re-tuned if the assigned operating frequency is changed. The diplexer isolates DCS receiver number 1, and the acquisition aid beacon or the delayed-time telemetry transmitter, depending upon the position of coaxial switch number 2. The diplexer operates into the uhf whip antenna on the adapter equipment section.

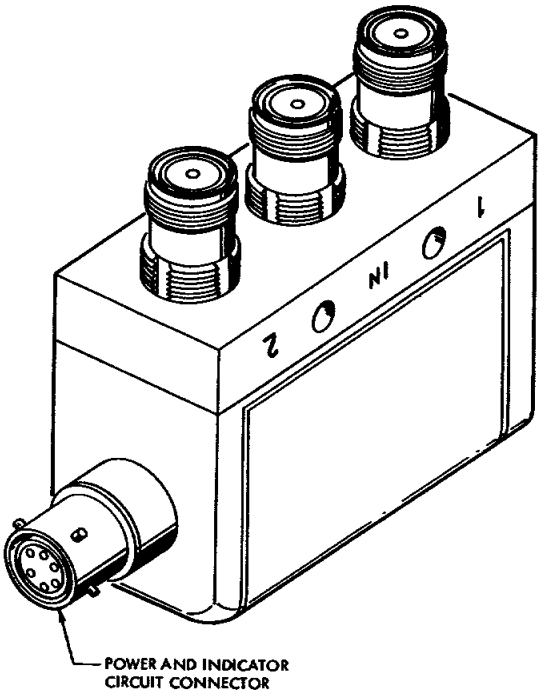
The uhf quadriplexer isolates the real-time telemetry transmitter, the stand-by telemetry transmitter, one of the two uhf voice transmitter/receivers, and DCS receiver number 2. The quadriplexer operates into one of the following three uhf antennas, depending on the position of the coaxial switches in series with the antennas: uhf stub antenna, uhf descent antenna, or the uhf whip antenna on the adapter retrograde section.

Coaxial Switches

Purpose: Five coaxial switches are used to perform the following functions: (1) select the acquisition aid beacon or the delayed-time telemetry transmitter output as the input to the diplexer; (2) select one of the two uhf voice transmitter/receiver outputs as the input to the quadriplexer; (3) connect the hf voice transmitter/receiver to the adapter hf whip antenna on the retrograde section, or to the recovery hf whip antenna on the re-entry module; (4) connect the output of the quadriplexer to the uhf descent antenna, or through coaxial switch 5 to the uhf stub or the retrograde adapter uhf whip antenna.



ITEM	FUNCTION
1	UHF TRANSCEIVERS
2	TELEMETRY TRANSMITTER/ACQUISITION AID BEACON
3	DESCENT ANTENNA
4	HF TRANSCEIVER
5	UHF WHIP/UHF STUB



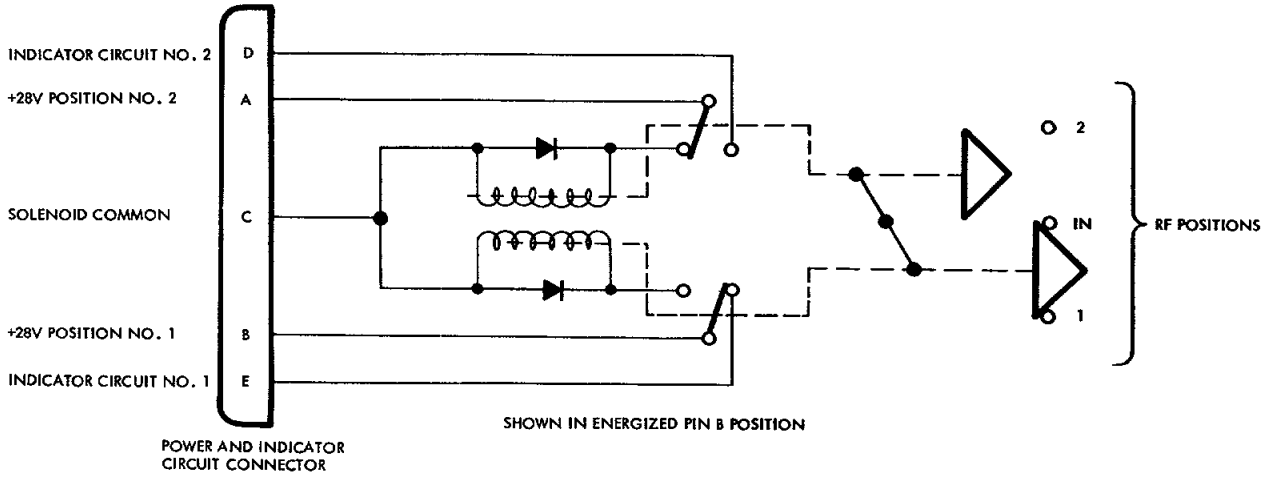


Figure 9-10 RF Coaxial Switches

Physical Characteristics: The physical construction and approximate location of the coaxial switches is shown in Figure 9-10. The location of the switches is as follows:

Coaxial switch 1: approximately five inches from the small end of the cabin, in the fourth quadrant.

Coaxial switch 2: approximately 10 inches from the forward (small) end of the adapter equipment section, in the third quadrant.

Coaxial switch 3: approximately 10 inches from the small end of the cabin, in the third quadrant.

Coaxial switch 4: located adjacent to coaxial switch 1

Coaxial switch 5: approximately 7 inches from the small end of the cabin, in the third quadrant.

Each switch contains a power connector, an input connector, two output connectors, and weighs approximately 0.5 pounds. The dimensions of each switch are approximately 2.65 inches long, 1.82 inches high, and 1 inch wide.

Electrical Characteristics: The five coaxial switches are identical and may be used interchangeably. Basically, the coaxial switches supply single pole double throw switching action as illustrated in Figure 9-10. The switch, having a 20 millisecond maximum operation time, operates on 3 amperes at 28 vdc and uses a latching solenoid break-before-make switching action. The coaxial switches are designed to operate from 15 mc to 500 mc, and from 5500 mc to 5900 mc. Pins D

and E of each switch are brought out to AGE test points to permit monitoring of the switch positions prior to lift-off. Pins A and B of each switch are utilized to accomplish the switching action.

BEACONS

Re-entry C-Band Radar Beacon

Purpose: The re-entry C-band radar beacon provides tracking capability of the spacecraft from lift-off to insertion and from retrograde to landing. The re-entry C-band beacon may be used during roll maneuvers or in the event of adapter C-band beacon failure.

Physical Characteristics: The re-entry C-band radar beacon is a sealed unit which measures approximately 7.64 x 6.14 x 3.02 inches and weighs about 8.3 pounds. As shown in Figure 9-11, the beacon has power, antenna, and test connectors. Located on the rear of the beacon are various adjustments for transmitter, preselector, and local oscillator tuning. Solid-state modular circuitry is used throughout the beacon with the exception of the transmitter magnetron and the local oscillator. The beacon is mounted on the right forward equipment bay, and uses the C-band antenna system for reception and transmission.

Electrical Characteristics: The re-entry C-band radar beacon is a transponder which upon reception of a properly coded interrogation signal from a ground radar tracking station, transmits a pulse modulated signal back to the tracking station. By measuring the elapsed time between transmission and reception at the tracking stations, and compensating for the time delay of the beacon, the position of the spacecraft can be determined. The block diagram of the beacon is shown in

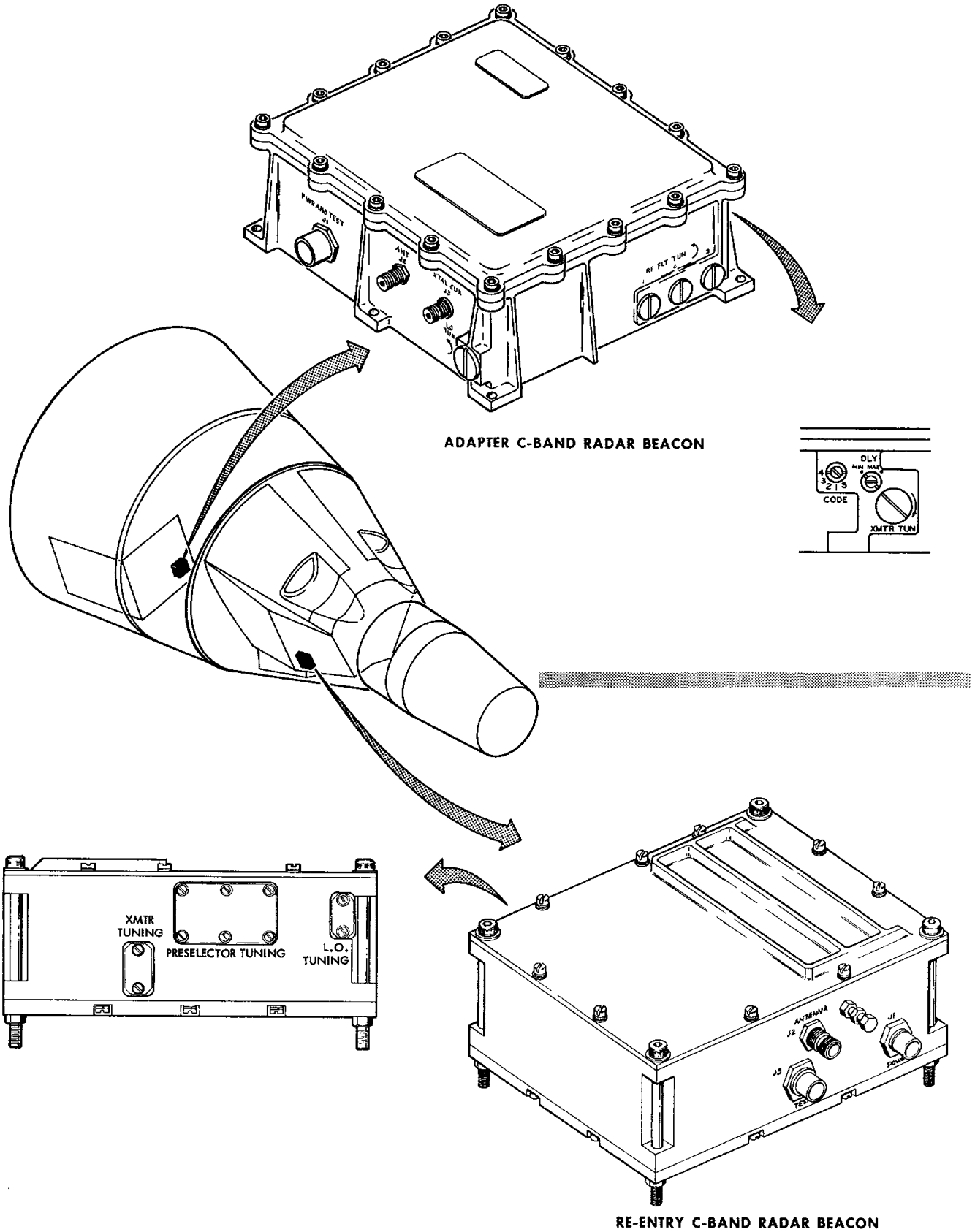


Figure 9-11 C-Band Radar Beacons

Figure 9-12. The signal arriving at the antenna is routed through the directional coupler to one half of a dual ferrite circulator. The ferrite circulator isolates the transmitter from the receiver, allowing a single antenna system to be used for both reception and transmission. The beacon utilizes a superhetrodyne receiver which is tunable, by means of a three stage preselector, over a range of 5600 mc to 5800 mc. The assigned receiver center frequency is 5690 mc.

The output of the preselector is combined with the local oscillator frequency in the crystal mixer to produce an output intermediate frequency of 80 mc. The local oscillator is of the metal-ceramic triode cavity type. The mixer contains a ferrite circulator for isolation between the local oscillator, mixer and preselector. The output of the mixer is amplified by three tuned intermediate frequency amplifier stages, followed by a video detector and a video preamplifier. Additional amplification is obtained by a pulse amplifier whose output is supplied to the decoder. The purpose of the decoder is to initiate triggering of the transmitter after a correctly coded signal has been received. The system delay, in conjunction with the delay variation correction circuitry, provides for a constant fixed delay used in determining the exact position of the spacecraft. The beacon incorporates a cw immunity circuit that prevents the transmitter from being triggered by random noise. The noise level is reduced below the triggering level of the transmitter by controlling the gain of the pulse amplifier. The transmitter uses a magnetron and provides a one kilowatt peak pulse modulated signal at a frequency of 5765 mc to the power divider. The beacon is powered by a dc-dc converter employing a magnetic amplifier and silicon controlled rectifiers. The converter provides voltage regulation for input voltage variations between 18 and

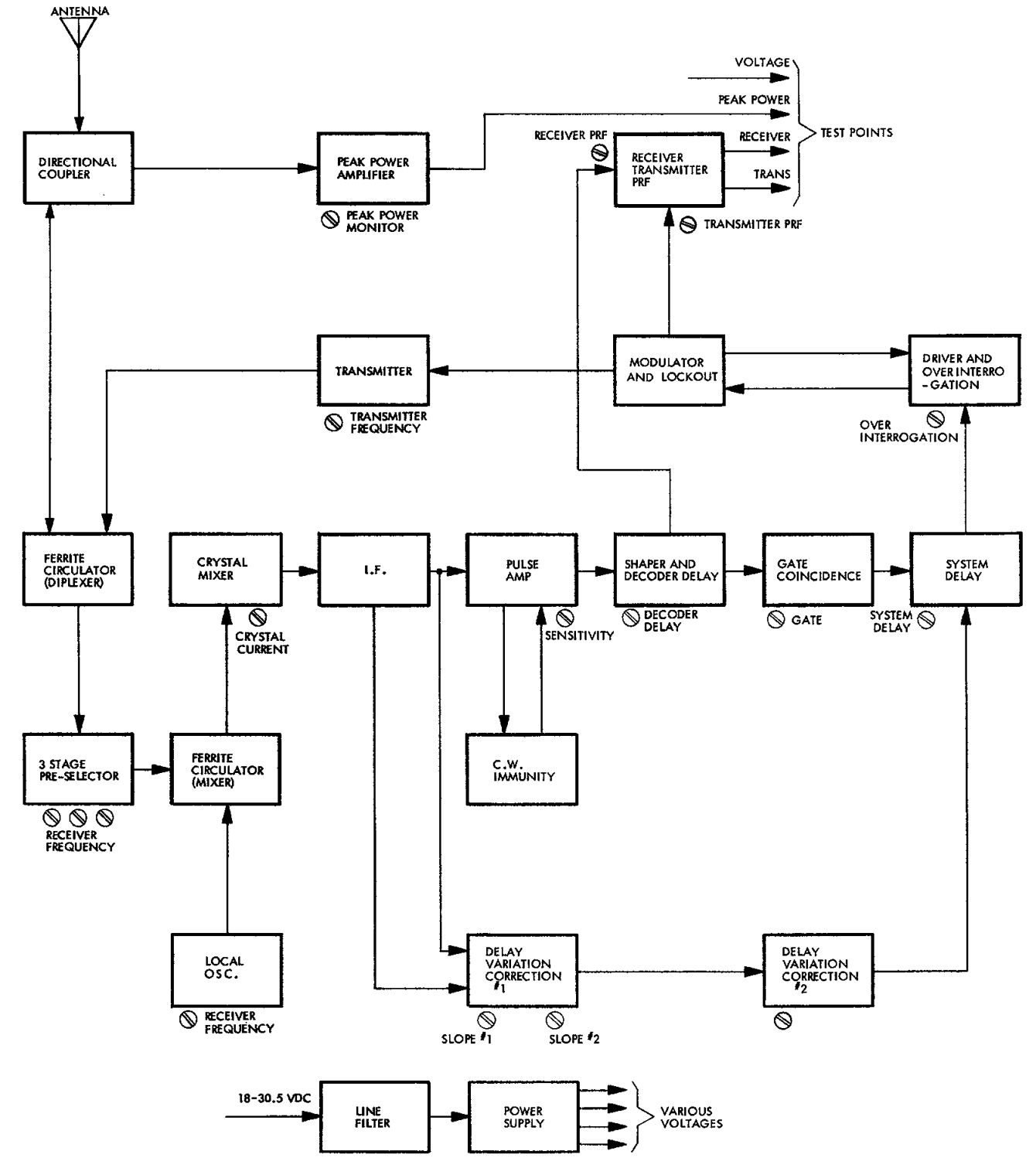


Figure 9-12 Re-Entry C-Band Radar Beacon Block Diagram

32.5 vdc. The input to the converter is filtered by a pi-type filter to minimize any line voltage disturbances.

Adapter C-Band Radar Beacon

Purpose: The adapter C-band radar beacon provides tracking capability of the spacecraft during the orbital phase of the mission and is jettisoned with the adapter equipment section.

Physical Characteristics: The adapter C-band beacon is a sealed unit and measures approximately 9.34 x 8.03 x 3.26 inches. As shown in Figure 9-11, the adapter beacon has a power and test connector, an antenna connector, and a crystal current test point connector. The beacon contains external adjustments for local oscillator, preselector (rf filter), and transmitter tuning; switches for selecting the desired interrogation code, and one of two preset transponder fixed delay times. These adjustments and switches are accessible by removing pressure sealing screws. The beacon employs solid-state circuitry, except for the transmitter magnetron and receiver local oscillator. The adapter beacon is located on the electronic module of the adapter equipment section and uses the C-band annular slot antenna for reception and transmission.

Electrical Characteristics: The adapter C-band radar beacon is a transponder, which employs the same basic operating principles as the re-entry C-band beacon to provide spacecraft location data upon receipt of a properly coded interrogation signal. A block diagram of the adapter C-band beacon is shown in Figure 9-13. The interrogation signal is fed from the antenna to the duplexer. The duplexer is a ferrite circulator which couples the received signal to the rf filter preselector and also isolates the receiver from the transmitter to permit

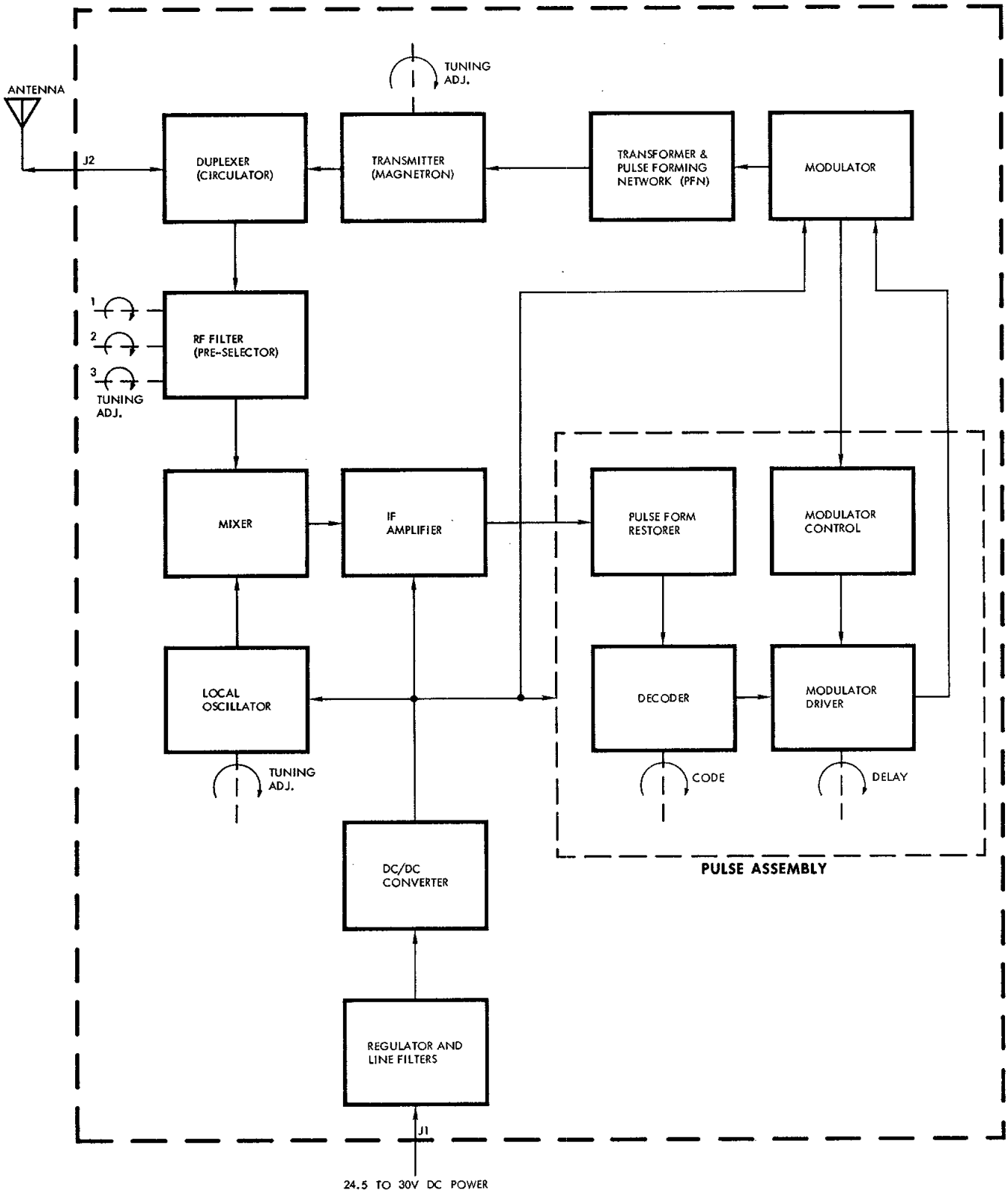


Figure 9-13 Adapter C-Band Radar Beacon Block Diagram

use of a common antenna for reception and transmission. The superhetrodyne receiver frequency is tunable from 5395 mc to 5905 mc. The assigned operating center frequency is 5690 mc and is selected by adjustment of the rf filter.

The rf filter is a three-stage preselector, employing three separately tuned coaxial resonator cavities to provide adequate rf selectivity and to protect the mixer crystal from damage due to transmitter power reflected by the antenna.

The output of the preselector is combined with the local oscillator output in the mixer stage to provide a 60 mc output to the intermediate frequency amplifier. The mixer consists of a coaxial directional coupler and a mixer crystal. The directional coupler isolates the local oscillator output from the antenna and directs it to the mixer crystal. The local oscillator is a re-entrant cavity type employing a planar triode to generate the cw signal required to operate the mixer.

The intermediate frequency amplifier is a high gain amplifier composed of an input stage, five amplifier stages, and a video amplifier. The amplified video output is fed to the pulse form restorer circuits which prevent a ranging error due to variations in receiver input signal levels, and also provides a standard amplitude pulse to the decoder for each input signal exceeding its triggering threshold. The decoder determines when a correctly coded signal is received and supplies an output to the modulator driver. The type code to be accepted is selected by the CODE switch. Single pulse, two pulse or three pulse codes may be selected. The modulator driver and control circuits initiate and control triggering of the transmitter modulator. The modulator driver supplies two fixed values of overall system

delay. The desired delay is selected by the position of the DLY switch. An alternate value of maximum delay is available by removing an internal jumper lead. The modulator control furnishes the trigger and turn-off pulse for the modulator and limits modulator triggers to prevent the magnetron duty cycle from being exceeded, regardless of the interrogating signal frequency. The modulator circuit employs silicon controlled rectifiers which function similar to a thyratron, but require a much shorter recovery time.

The associated modulator Pulse Forming Network (PFN) and transformer provide the necessary pulse to drive the transmitter magnetron. The desired pulse width is selected by the internal connections made to the PFN. The transmitter magnetron frequency is tunable from 5400 mc to 5900 mc. The assigned transmitter center frequency is 5765 mc. A minimum of 500 watts peak pulse power is supplied to the antenna under all conditions of rated operation.

The transponder power supply consists of input line filters, a series regulator, and a dc-dc converter. The power supply furnishes the required regulated output voltages with the unregulated input voltage between 21 and 30 vdc. The converter employs a multivibrator and full wave rectifier circuits.

Acquisition Aid Beacon

Purpose: Unlike the C-band beacons that supply accurate tracking data, the acquisition aid beacon is merely a transmitter used to determine when the spacecraft comes within range of a ground tracking station. When the spacecraft comes within the range of a ground tracking station, the acquisition aid beacon is disabled and remains off until the spacecraft is again out of range.

Physical Characteristics: The acquisition aid beacon, shown in Figure 9-14, is cylindrical, having a diameter of approximately 2.6 inches, and a height of approximately 3.5 inches. The acquisition aid beacon is located as shown in Figure 9-14. The beacon contains a power connector, a coaxial antenna connector and weighs approximately 17 ounces.

Electrical Characteristics: The acquisition aid beacon consists of a transmitter, dc-dc voltage regulator, and a low pass output filter.

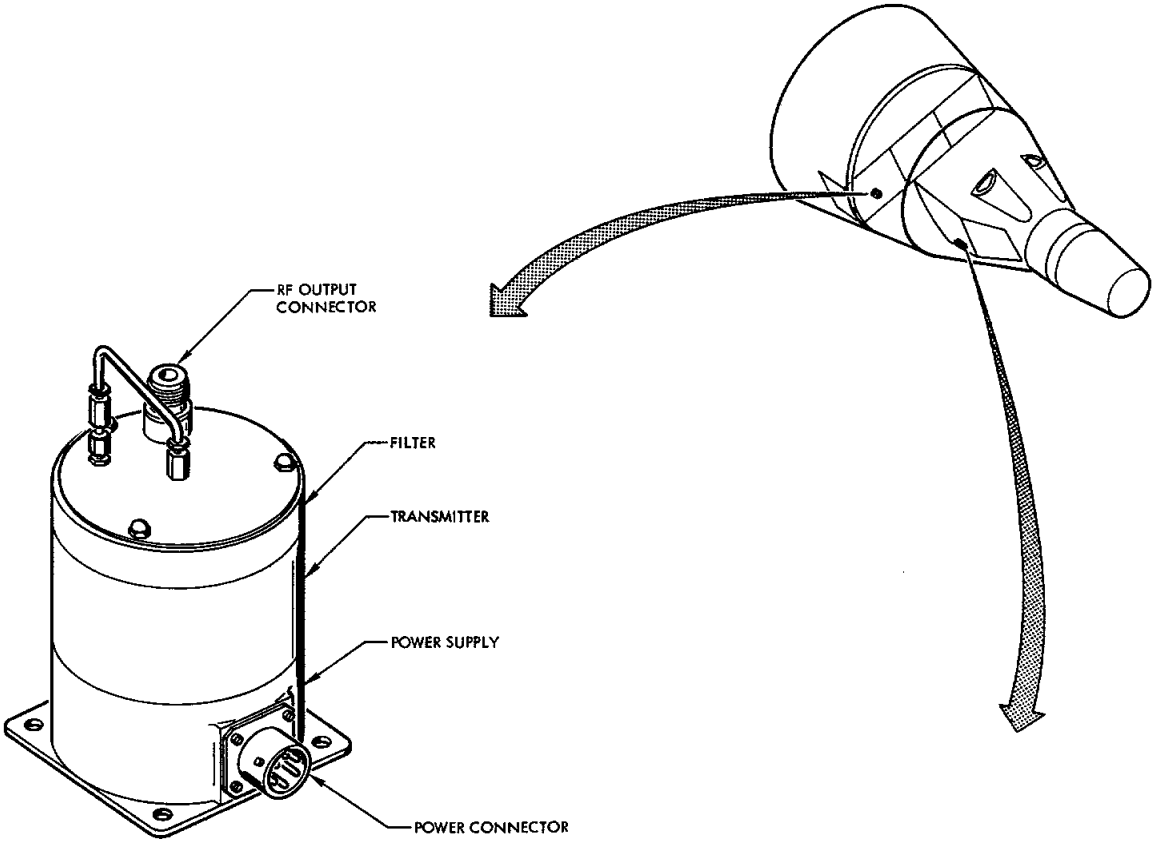
The transmitter is an all transistorized unit, containing a push-pull output stage to obtain a minimum output of 200 milliwatts at a frequency of 246.3 mc. The transmitter frequency is derived by taking the basic frequency of an oscillator and multiplying it through a series of tripler and doubler stages.

The transmitter is powered by a dc-dc voltage regulator. The regulator is completely transistorized and supplies a regulated output voltage of 28 vdc. To reduce the probability of obtaining a spurious output signal, a band pass filter is placed in the output circuit.

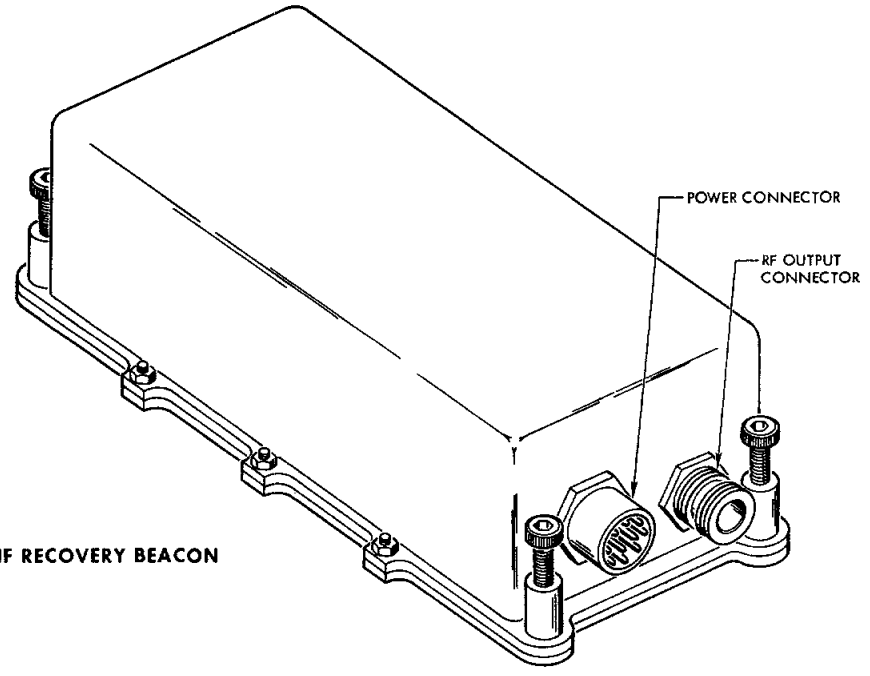
UHF Recovery Beacon

Purpose: The uhf recovery beacon, operating on the international distress frequency of 243 mc, serves as a recovery aid by providing information regarding location of the spacecraft.

Physical Characteristics: The uhf recovery beacon and its approximate location is shown in Figure 9-14. The beacon is mounted on the aft right equipment bay of the spacecraft re-entry module. The beacon is approximately 9.0 inches long,



ACQUISITION AID BEACON



UHF RECOVERY BEACON

Figure 9-14 Acquisition Aid and UHF Recovery Beacons

4.0 inches wide, 2.5 inches high, and weighs 3.9 pounds maximum. The beacon contains one multipin power connector and one coaxial connector.

Electrical Characteristics: The uhf recovery beacon consists of a spike eliminator, a regulator, a dc-dc converter, a pulse coder, a modulator, and a transmitter.

Spacecraft main bus voltage is fed to the switching type regulator through the spike eliminator filter. The voltage regulator provides a dc regulated output voltage of 12 vdc to the dc-dc converter, the transmitter tube filaments, and the pulse coder.

The dc-dc converter is a solid-state device providing two high voltage outputs to the transmitter and modulator. The pulse coder, a solid-state device, operates with the modulator to apply correctly coded high voltage pulses to the transmitter for plate modulation of the power amplifiers.

The transmitter consists of an oscillator stage, a doubler stage, and a power amplifier. The transmitter power amplifier provides a uhf pulse coded output having a peak power of at least 50 watts to the uhf recovery antenna. An external rf band-pass filter is installed between the transmitter output and the antenna to reduce spurious rf radiations, especially at the uhf voice transmitter frequencies.

VOICE COMMUNICATION

Voice Control Center

Purpose: The Voice Control Center contains switches and controls for selecting the type of voice communication and the desired operating mode. The VCC also contains microphone and heatset amplifiers, an alarm tone generator, and voice

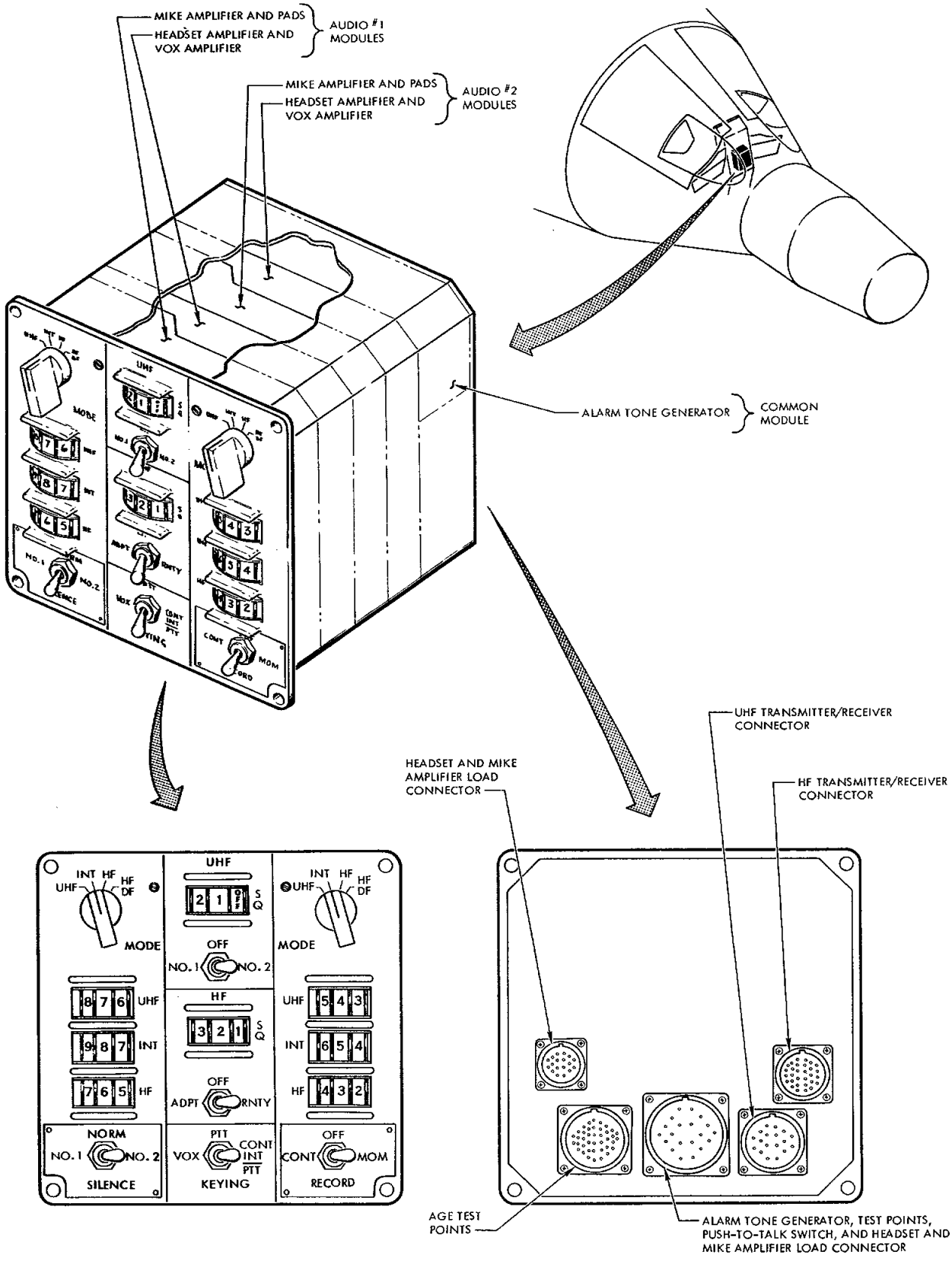


Figure 9-15 Voice Control Center

actuated transmitter keying circuitry.

Physical Characteristics: The VCC and its approximate location is shown in Figure 9-15. The VCC is mounted in the center instrument panel of the spacecraft cabin. The VCC is modular constructed, approximately 6.4 inches wide, 6.4 inches high, 5.5 inches deep, and weighs approximately 6.5 pounds.

Five connectors located on the rear of the unit provide connection to the other voice communication system components and test connectors. The function of each connector is listed on Figure 9-15.

The switches and controls of the VCC are located on the front panel. The number 1 and number 2 audio MODE switches are for selection of UHF, INT, HF, or HF/DF transmission. Below the MODE switches are three thumb-wheel-type multidetent volume controls, one for each of the above mentioned modes.

In the center is the KEYING switch, a HF select switch, a UHF select switch, and thumb-wheel-type squelch controls for uhf and hf circuitry. The KEYING switch provides for selection of PTT, VOX, or CONT INT/PTT for the voice transmitters. The UHF and HF select switches provide capability of selecting the desired transmitter/receiver. The ADPT position of the HF select switch is not used.

The record switch, lower right, permits recordings to be made in any mode of operation. Continuous (CONT) or Momentary (MOM) recording can be selected.

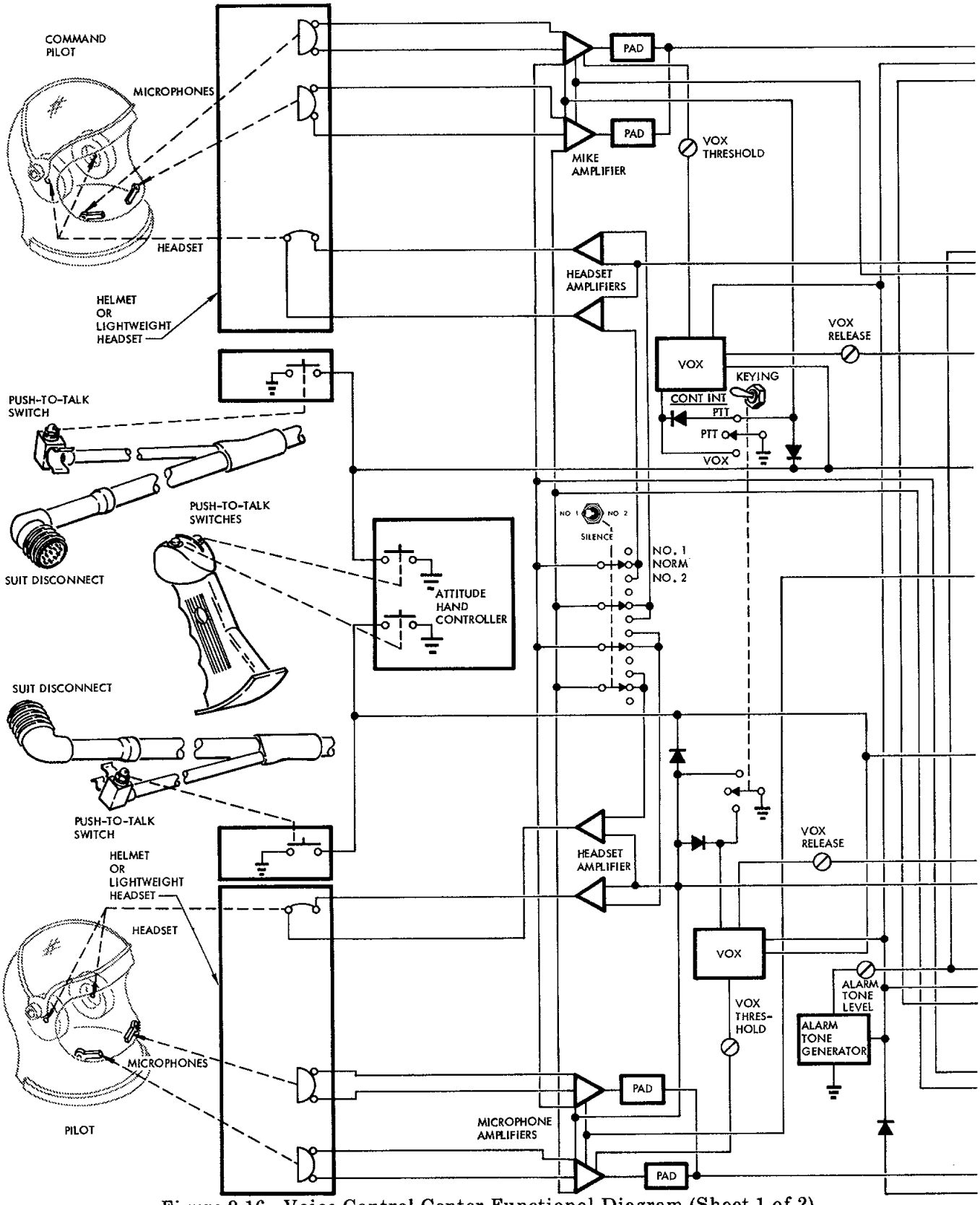


Figure 9-16 Voice Control Center Functional Diagram (Sheet 1 of 2)

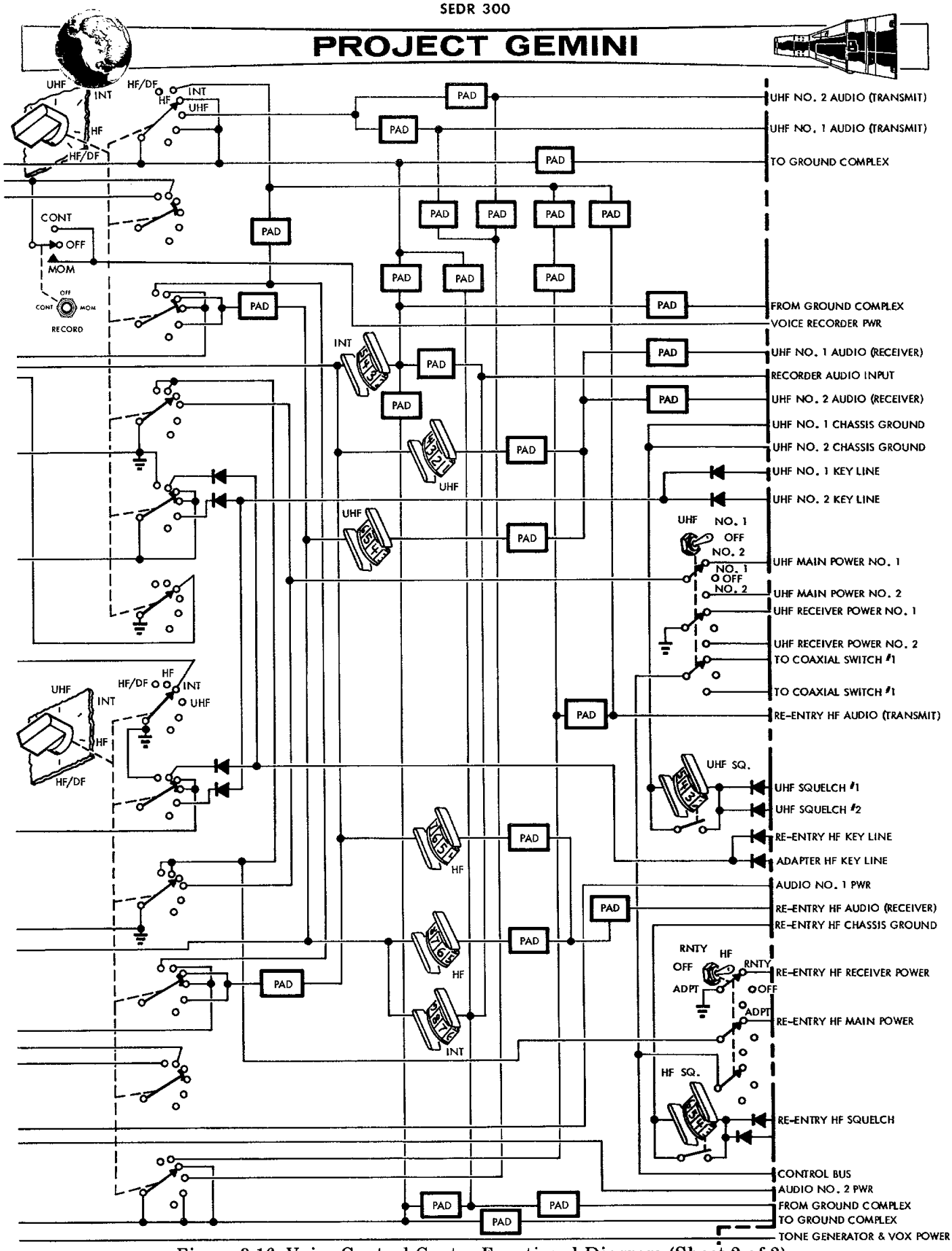


Figure 9-16 Voice Control Center Functional Diagram (Sheet 2 of 2)

The SILENCE switch, lower left, is to permit uninterrupted sleep during extended spacecraft missions. The NORM position allows reception for both pilots. The NO. 1 position removes power from the command pilot's headset amplifiers and the NO. 2 position removes power from the pilot's headset amplifiers; thus, making reception impossible.

Electrical Characteristics: The VCC contains two headset and two microphone amplifiers for each of the audio channels.

Figure 9-16 shows a functional block diagram of the VCC. An audio signal, from the microphone in the helmets or lightweight headsets, is amplified by two microphone amplifiers and then applied to the MODE switch. With the MODE switch in the HF position, the output of the microphone amplifiers is applied to the hf transmitter. When the MODE switch is in the INT position, the output of the microphone is applied to the four headset amplifiers, via the two INT volume controls. With the MODE switch in the UHF position, the output of the microphone amplifiers is applied to the uhf transmitters. The UHF switch selects uhf transmitter number 1 or number 2 and also operates coaxial switch 1 to connect the selected transmitter output to the uhf quadriplexer. The desired keying mode is selected by a common KEYING switch. Three methods may be selected to key the voice transmitters. The VOX position enables keying of the selected transmitter at the instant the microphone has an output signal. The PTT position eanbles keying of the transmitter when either push-to-talk switch, on the suit disconnect cables of the attitude control handle, is depressed. The CONT INT/PTT position gives continuous intercommunication between the crew and push-to-talk keying for transmission from

the spacecraft to the ground station.

The VCC also controls the power supplies of the transmitter/receivers by means of ground switching. With the MODE switch in a position other than HF and the HF select switch in the RNTY position, a ground is supplied to the hf transmitter/receiver auxiliary power supply to power the hf receiver.

With the HF select switch in RNTY and the MODE switch in the HF position, a ground is supplied to the hf transmitter/receiver main power supply to power the hf receiver and transmitter. The uhf circuitry operates on the same principle as the hf. The UHF select switch supplies power ground for the selected receiver.

The MODE switch (UHF position) together with the UHF select switch, supplies a power return for the uhf transmitter and receiver.

The HF/DF position of the MODE switch is used for direction finding purposes. With the MODE switch in HF/DF and the HF select in the RNTY position, the hf transmitter is modulated by a 1,000 cps tone which is utilized to determine spacecraft location.

UHF Voice Transmitter/Receivers

Purpose: Two uhf voice transmitter/receivers are provided for redundant line-of-sight voice communication between the spacecraft and the ground.

Physical Characteristics: The uhf voice transmitter/receivers and their approximate location is shown in Figure 9-17. Both transmitter/receivers are identical and are mounted side by side in the forward right equipment bay of the re-entry

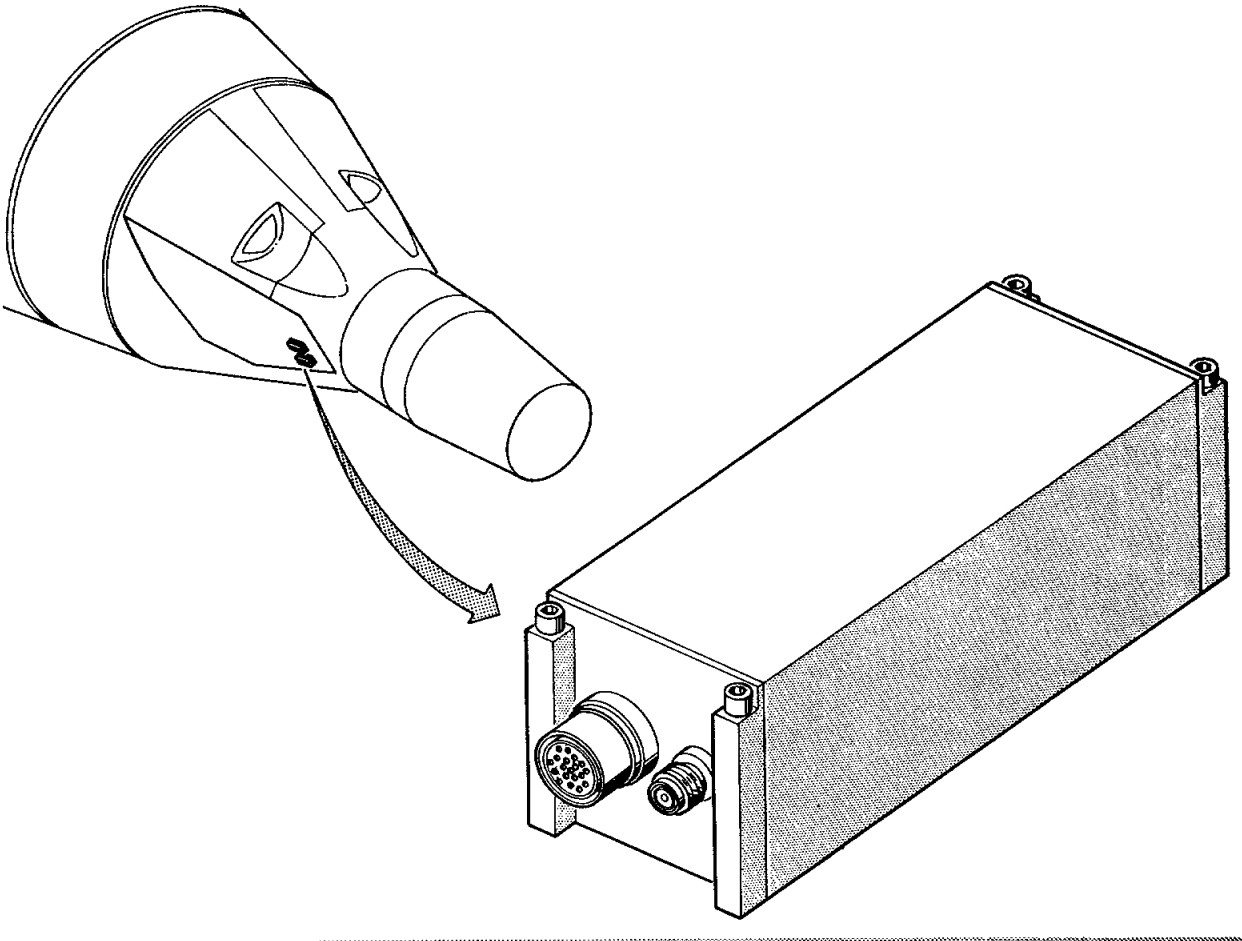
module. Each transmitter/receiver is a modular constructed, hermetically sealed unit approximately 7.7 inches long, 2.8 inches wide, 2.4 inches deep and weighs approximately 3.0 pounds. Each unit has a multipin audio and power connector, and a coaxial connector.

Electrical Characteristics: The uhf voice transmitter/receiver consists of a transmitter, receiver, and power supply.

The transmitter consists of a crystal controlled oscillator, two rf amplifiers, a driver, and a push-pull power amplifier. All stages except the driver and power amplifier are transistorized. The transmitter is fixed-tuned at 296.8 mc and is capable of producing an rf power output of 3.0 watts into a 50 ohm resistive load. The transmitter is amplitude modulated (am) by a transistorized modulator stage.

The am superhetrodyne receiver is fully transistorized, is fixed-tuned at a frequency of 296.8 mc, and contains a squelch circuit for noise limiting. The squelch threshold is manually controlled. An automatic volume control stage is also incorporated to provide a constant audio output with input signal variations.

The uhf voice transmitter/receiver is powered by two dc-dc converters comprising an auxiliary and a main power supply. Operating power for the two power supplies is limited by two circuit breakers located on the left switch/circuit breaker panel. One circuit breaker is provided for each unit. Actuation of the power supplies is accomplished by ground return switching through the Voice Control Center. If the UHF select switch is in the NO. 1 or NO. 2 position and the MODE switch is in a position other than UHF, a ground is supplied to the auxiliary power supply



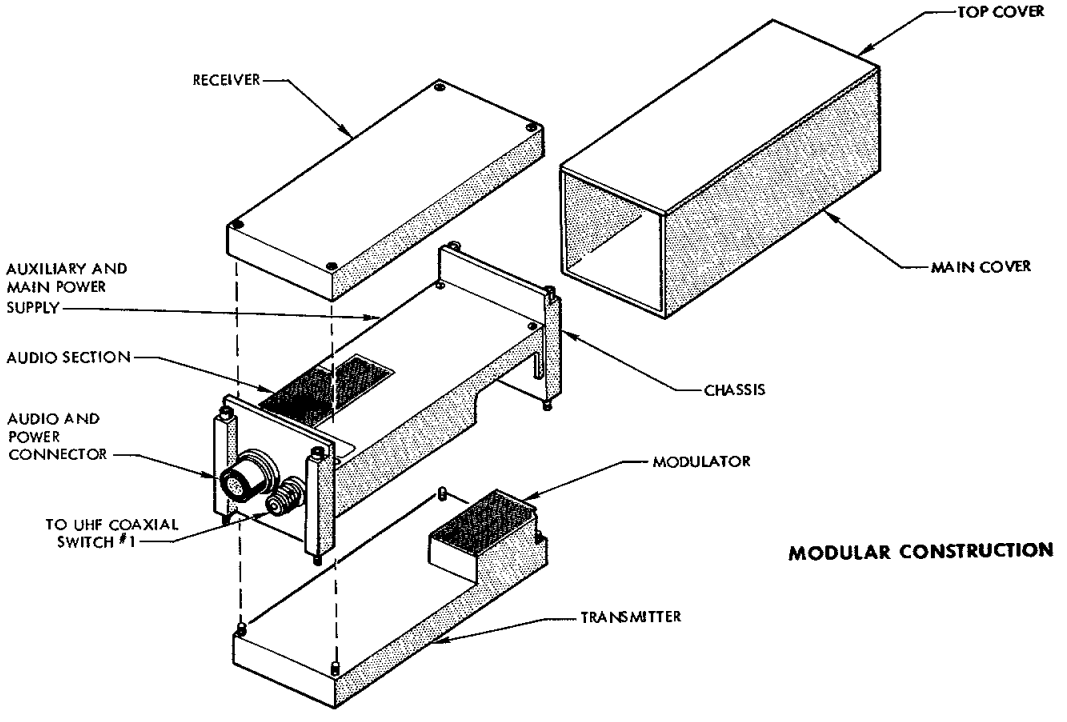


Figure 9-17 UHF Voice Transmitter/Receiver

only, placing the transmitter/receiver into a receive condition. With the MODE switch in the UHF position, a ground is supplied to the main power supply, placing the selected uhf voice transmitter/receiver into a receive and transmit condition.

It should be noted that when the uhf transmitter is keyed, the uhf receiver is disabled and uhf voice transmissions from the ground station can not be received.

HF Voice Transmitter/Receiver

Purpose: The hf voice transmitter/receiver is provided to enable beyond the line-of-sight voice communication between the spacecraft and the ground.

Physical Characteristics: Figure 9-18 shows the modular construction and approximate location of the hf voice transmitter/receiver in the forward right equipment bay of the re-entry module. The unit weighs approximately 62 ounces, is approximately 8.5 inches long, 3.3 inches wide, and 2.9 inches deep. One multipin audio connector and one rf connector are provided.

Electrical Characteristics: Basically, the hf voice transmitter/receiver is electrically identical to the uhf transmitter/receiver except for the operating frequency and power output. The hf transmitter and receiver are fixed tuned to a frequency of 15.016 mc and the hf transmitter provides an rf power output of 5 watts.

Actuation of the hf receiver and transmitter is accomplished through the VCC. If the HF select switch is in RNTY and the MODE switch is in a position other than HF, the hf transmitter/receiver is in a receive condition. With the MODE switch in the HF position, the hf transmitter/receiver is placed in a receive and transmit condition.

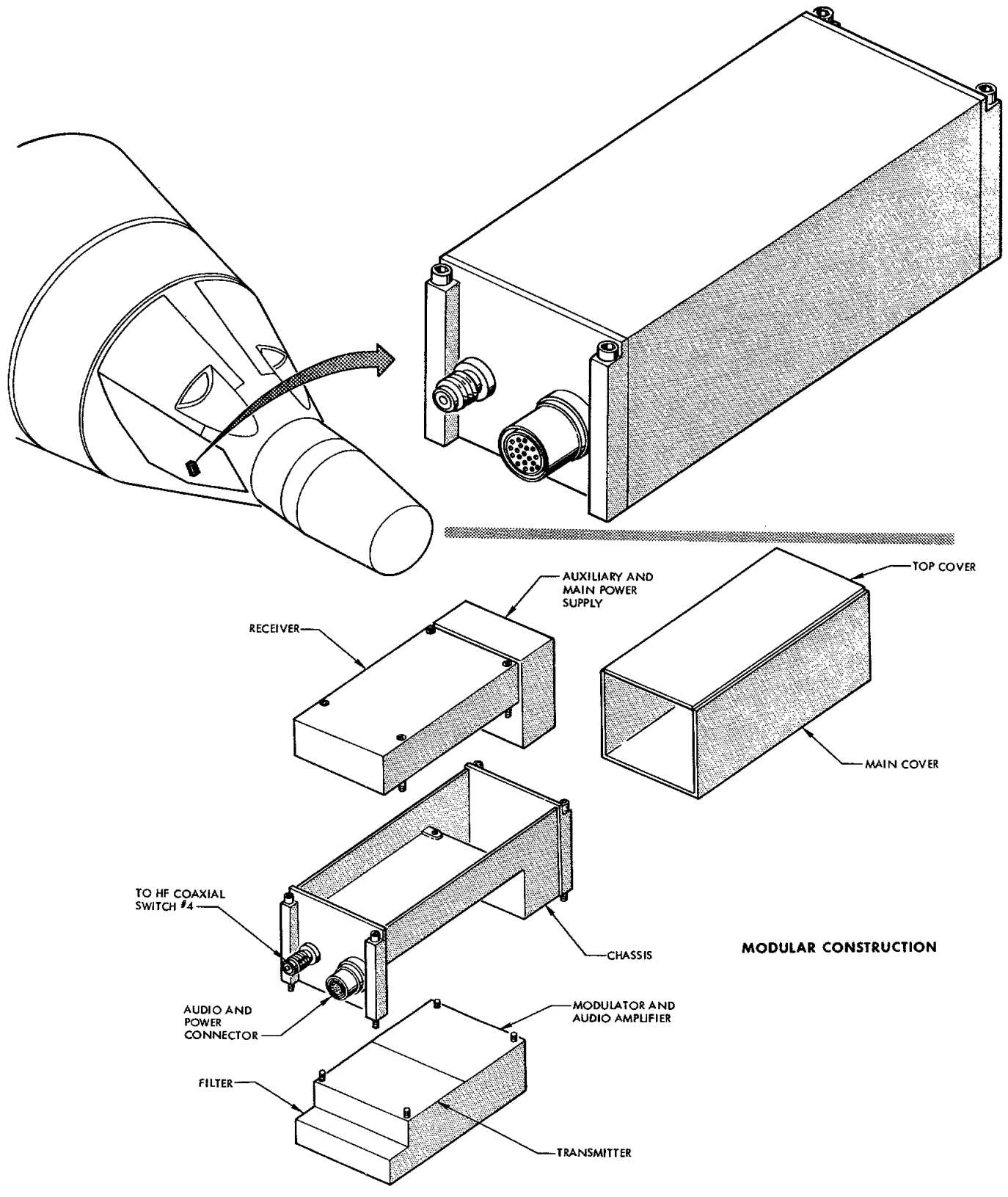


Figure 9-18 HF Voice Transmitter/Receiver

When the hf transmitter is keyed, the hf receiver is disabled.

Voice Tape Recorder

Purpose: The voice tape recorder is provided so recordings can be made during the spacecraft mission.

Physical Characteristics: The physical construction and approximate location of the voice tape recorder is shown in Figure 9-19. The voice tape recorder is located inside the cabin in a vertical position between the pilot's seat and the right-hand side wall on spacecraft 5 and 6. On spacecraft 8 through 12 the recorder is located on the left-hand side wall aft of the abort handle. The voice tape recorder assembly consists of the recorder, tape cartridge, and shock absorber mounting plate and is supplied as GFE equipment. The recorder is approximately 6.25 inches long, 2.87 inches wide, one inch thick, and weighs 30 ounces maximum without the tape cartridge. The shock absorber mounting plate is approximately 6.3 inches long, three inches wide, and weighs 20 ounces maximum. The tape cartridge is approximately 2.25 inches square, 3/8 inch thick, and weighs two ounces.

The recorder contains a power connector and a signal connector located on the end as shown in Figure 9-19. The recorder is retained in the shock mount by guides and two allen-head bolts for easy removal. The door contains a red plastic lens so that light from the end-of-tape bulb is visible. A safety latch prevents accidental opening of the door. The door is opened by pressing down on the latch and sliding it sideways. When the latch is released, the spring loaded hinge causes the door to open, exposing the cartridge tab. Flat pressure springs on the door hold the inserted cartridge in place and maintains tape contact with the

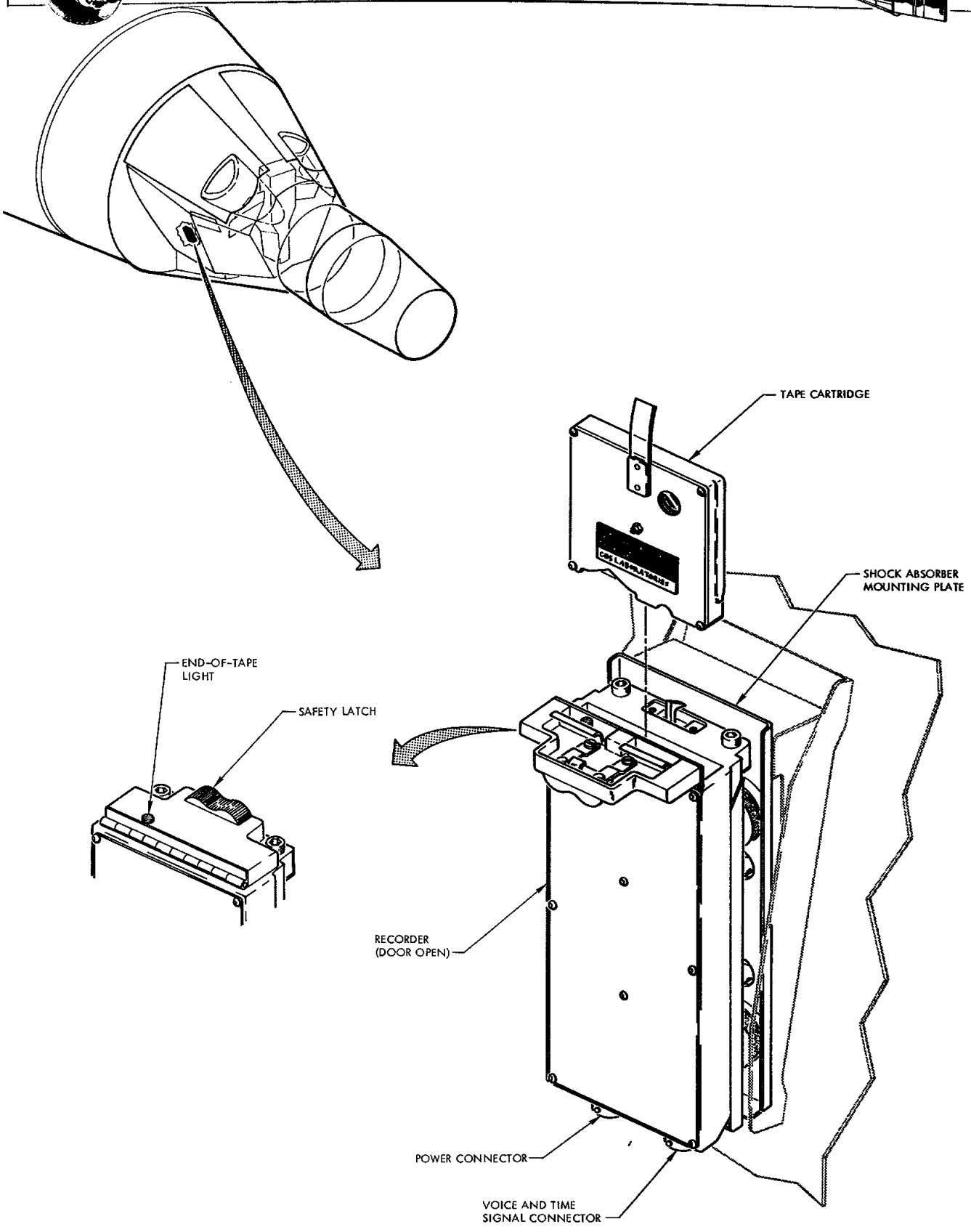


Figure 9-19 Voice Tape Recorder

recorder head and end-of-tape contact.

The tape cartridge is guided into the recorder by step rails on each side of the cartridge. When the recorder door is opened, a heavy tab on the cartridge springs up to provide easy removal. The cartridge contains approximately 180 feet of magnetic tape, a supply reel, take-up reel, and associated gears and clutches.

Electrical Specifications: The recorder is a two-channel transistorized unit consisting of the cartridge hold-down mechanism, voltage regulator, voice amplifier, time signal amplifier, bias oscillator, motor drive circuit, synchronous drive motor, speed reduction unit, capstan, magnetic record head, and end-of-tape circuit.

When the tape cartridge is inserted and secured in the tape recorder, the pressure roller in the cartridge contacts the capstan and the tape is pressed against the record head and the end-of-tape contact.

The voice tape recorder is energized by spacecraft main bus power applied through the TONE VOX circuit breaker and the CONT or MOM position of the RECORD switch on the VCC. The voltage regulator supplies 15 vdc to the motor drive circuits, bias oscillator and amplifiers. With the VCC and recorder energized, voice signals from the microphones are applied through microphone amplifiers in the VCC to the recorder voice amplifier. The voice signal is amplified and applied to the lower record head for recording on the magnetic tape. The time channel receives a digital timing signal from a time correlation buffer in the TRS. The timing signal is amplified by the recorder time signal amplifier and applied to the upper record head for recording on the magnetic tape.

Simultaneously with the voice or timing signal, a 20 kc bias current from the bias oscillator is applied to the recorder heads to make a linear recording.

The motor drive circuit consists of a 133 cps oscillator, a driver and push-pull output stage used to drive the synchronous motor. Phase-shift capacitors are connected to one motor winding for self-starting. The motor speed of 8000 rpm is reduced through the speed reduction unit to a capstan speed of 122 rpm.

The end-of-tape circuit is energized by conductive foil on the tape contacting the recorder head and end-of-tape contact, causing the end-of-tape light to illuminate. The end-of-tape light will illuminate for two seconds when two minutes of recording time remains on the tape. The light will remain on when the end-of-tape is reached. Recordings cannot be made when the light is illuminated. The pilot may remove the used tape cartridge, insert another cartridge and continue recording. Each cartridge provides approximately one hour of recording. The tape speed is approximately 0.6 inches per second.

TELEMETRY TRANSMITTERS

Purpose: The three telemetry transmitters provide a radio frequency (rf) link from the spacecraft to ground communication facilities for transmission of various data obtained by the Instrumentation System.

Physical Characteristics: The three telemetry transmitters are identical except for the operating frequency. The physical construction and approximate location of the transmitters in the spacecraft is shown in Figure 9-20. The transmitters are approximately 2.75 inches high, 2.25 inches wide, 6.5 inches long, and weigh

approximately 41 ounces. Each transmitter contains a dc power connector, an rf output power connector, and a video connector. Two of the transmitters are located in the right forward equipment bay of the re-entry module, the third is located on the electronic module in the adapter equipment section.

Electrical Characteristics: The three telemetry transmitters are classified by their operating frequency or by their function.

The real-time (low-frequency) telemetry transmitter operates at 230.4 mc. The delayed-time (mid-frequency) telemetry transmitter operates at a frequency of 246.3 mc. The stand-by (high-frequency) transmitter, operating at 259.7 mc, may be used for real-time or delayed-time transmission in case one of the transmitters fails.

The telemetry transmitters are solid-state fm transmitters. After a 30 second warm-up, the transmitters are capable of continuous uninterrupted operation for 500 hours. Information is transmitted to the ground in digital format by deviating the carrier frequency to the higher frequency deviation limit to transmit a 1, and to the lower deviation limit to transmit a 0.

The transmitters receive Non-Return to Zero (NRZ) PCM pulse trains from the PCM programmer and voice tape recorder. The real-time transmitter provides the ground monitoring stations with current real-time data at a rate of 51.2 kilobits per second. The delayed time transmitter provides the ground monitoring station with data stored on the tape recorder while the spacecraft was between ground stations. The delayed-time data is transmitted at a rate of 112.6 kilobits per second. The stand-by transmitter is used as backup for the real-time or delayed-time transmitters in event of a failure in either transmitter. Transmission of the real-time and delayed-time data provide essentially full-time

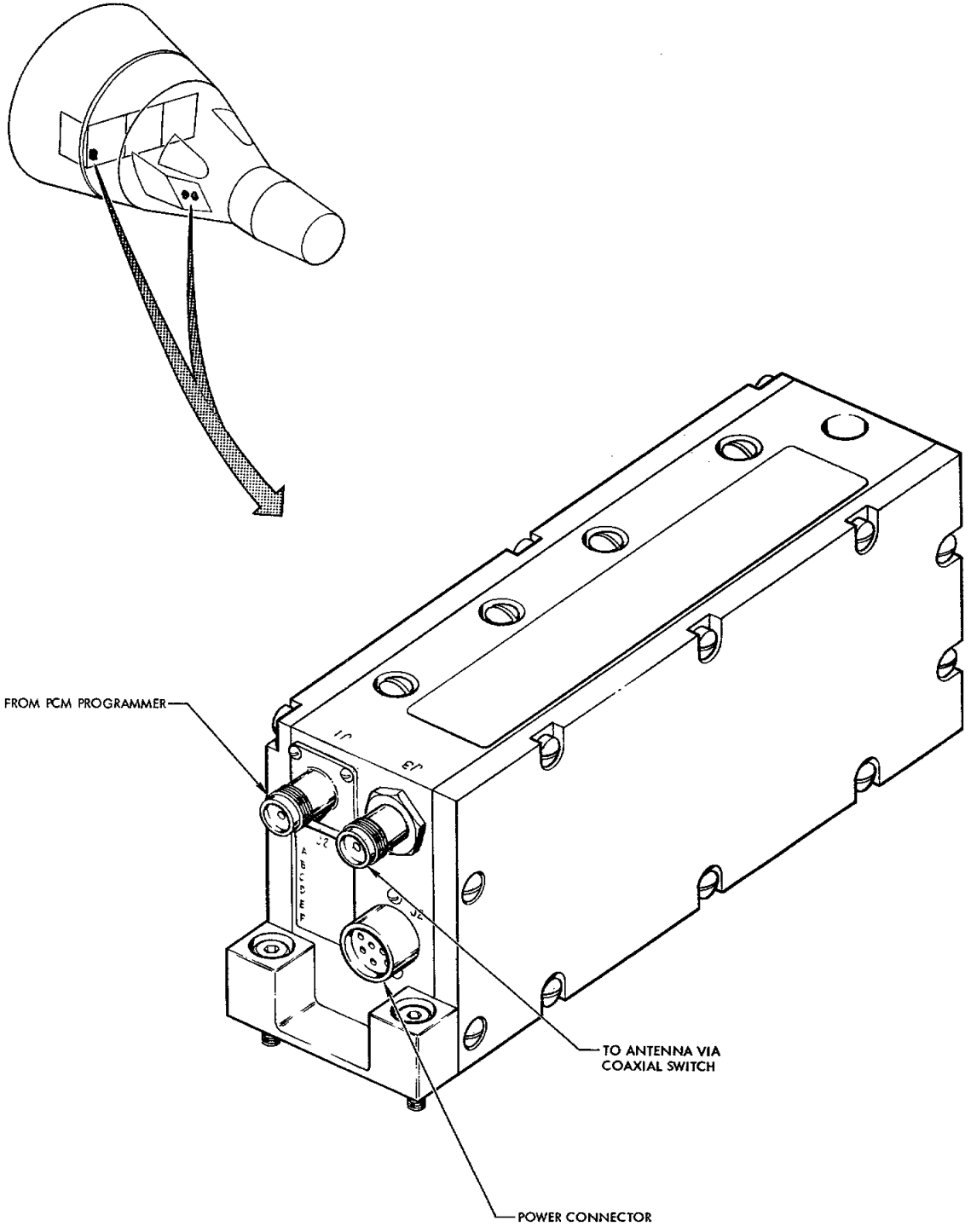


Figure 9-20 Telemetry Transmitters

coverage throughout the spacecraft mission. The transmitters can be energized by a command from the ground station via the DCS or by controls on the instrument panel.

Each transmitter consists of five subassemblies as shown in the block diagram of Figure 9-21. The subassemblies are an oscillator-modulator, a times 12 (x12) multiplier and power amplifier, a bandpass output filter, a line filter and a dc-dc converter. The oscillator-modulator and the times 12 multiplier and power amplifier subassemblies contain the variable resistors, inductors, transformers and trimmer capacitors for tuning the transmitter frequency and power output. The subassembly components are point-to-point wired.

The oscillator-modulator consists of a video amplifier, crystal controlled oscillator, phase shift networks and buffer amplifiers. The oscillator frequency is modulated by the video amplifier output. The phase shift networks provide impedance matching of the crystal oscillator to improve signal linearity for large deviations of frequency. The buffer amplifiers increase signal levels and isolate the crystal circuit from the frequency multipliers.

The times 12 multiplier and power amplifier consists of a buffer amplifiers, times 4 multiplier, power amplifier and times 3 multiplier which increase the carrier frequency and power to the desired output values. The power amplifier develops 6 to 7 watts of power at a frequency from 75 to 80 mc into the output tripler circuit.

The bandpass filter is used to minimize spurious radiations at the output of the transmitter. The real-time, stand-by and delayed-time transmitters each contain a filter with a different frequency bandpass. The filter has a minimum 3 db

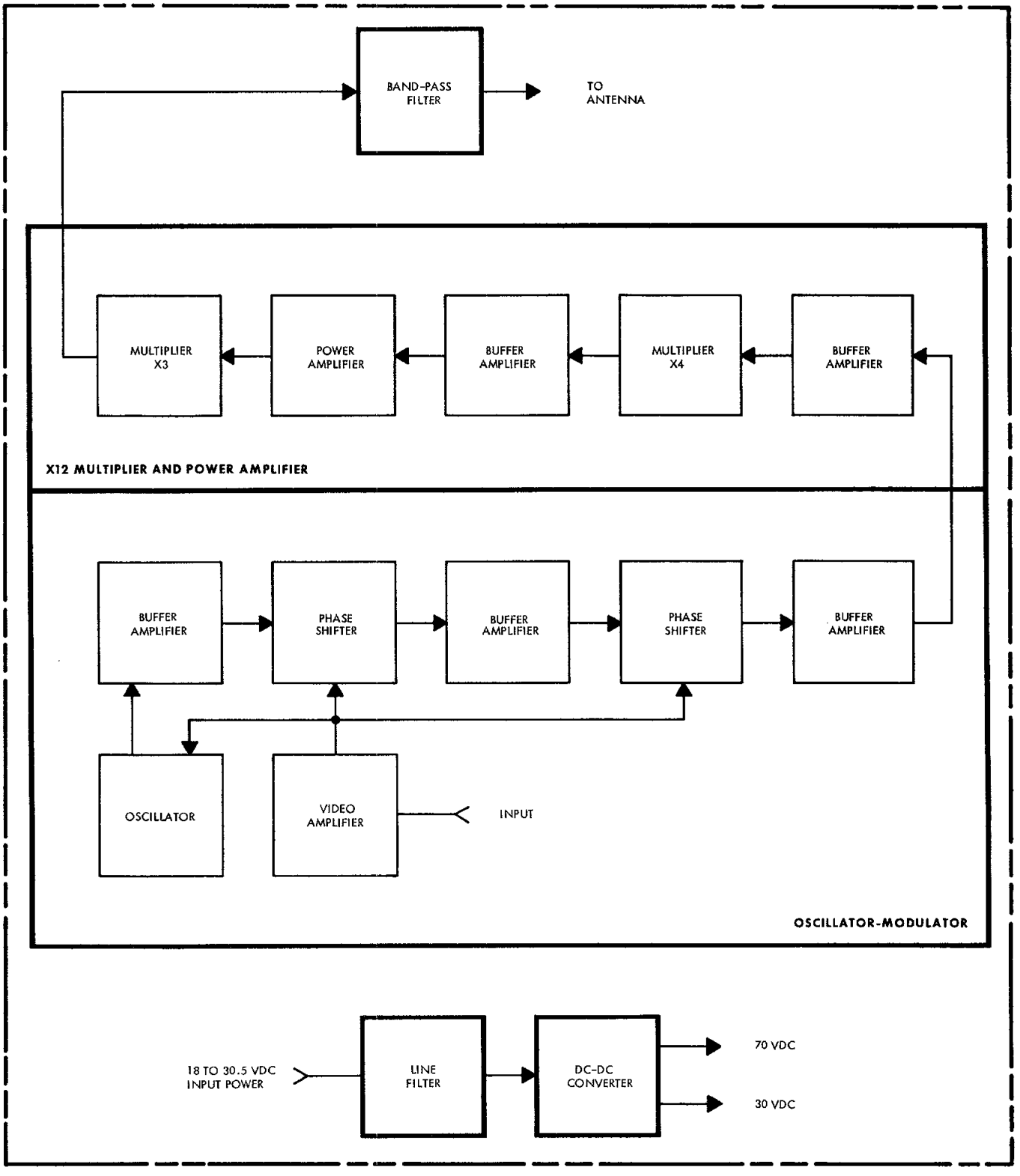


Figure 9-21 Telemetry Transmitter Block Diagram

bandwidth of 16 mc, an impedance of 50 ohms and a vswr of less than 1.5 to 1. The rf output connector J3 is an integral part of the bandpass filter.

The line filter prevents noise on the input power bus from affecting transmitter operation and prevents transients generated within the transmitter from feeding back to the input power bus. The multipin power connector J2 is an integral part of the line filter.

The dc-dc converter is a completely encapsulated unit employing transistors, diodes and a transformer to provide regulated outputs of 30 vdc and 70 vdc from an unregulated input voltage of 18 to 30.5 vdc. The converter is a constant power input type, thus minimizing the heat dissipation caused by high voltage inputs.

FLASHING RECOVERY LIGHT AND POWER SUPPLY

Purpose: The flashing recovery light and power supply provide visual spacecraft location information.

Physical Characteristics: Figure 9-22 shows the physical representation and approximate location of the flashing recovery light and its power supply. The light is self-extended by a torsion spring. The plug applying power to the light is kept in place by a compression spring. The recovery light will be automatically extended at the time the main parachute is jettisoned.

The flashing recovery light power supply is mounted in the cabin, aft of the ejection seats. The power supply is approximately 7 inches long, 4 inches wide, 3 inches deep and contains one connector. The flashing recovery light is approximately 1.25 inches wide, 0.75 inches thick, and 3.25 inches high, excluding tube and erecting mechanism. The overall length of the light and erecting mechanism is

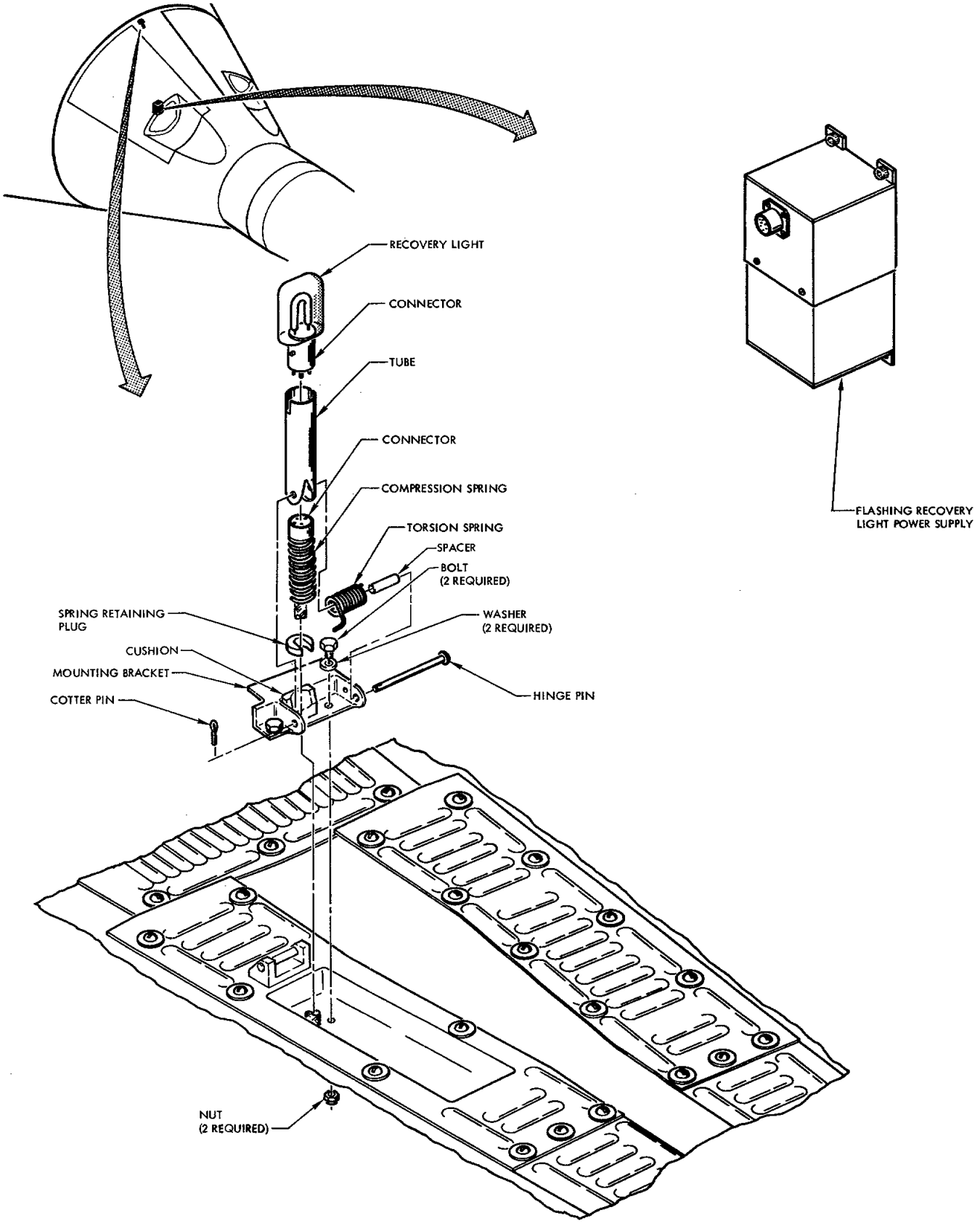


Figure 9-22 Flashing Recovery Light and Power Supply

approximately 6.5 inches.

Electrical Characteristics: The recovery light is automatically extended at main parachute jettison. The extended recovery light is energized by positioning the RESC BEACON CONTROL switch to ON.

The power supply consists of a battery pack and converter. The battery pack consists of several mercury cells to comprise a power source of 6.75 vdc to a dc-dc converter whose output is fed to a voltage doubler and a capacitive network. The 450 vdc output of the voltage doubler is used to power the flashing light while the capacitive network in conjunction with a thyratron, provides trigger pulses to accomplish switching or flashing action of the light. The trigger pulses occur at a rate of 15 triggers per minute.

DIGITAL COMMAND SYSTEM

Purpose

The DCS provides a discrete command link and a digital data updating capability for the computer and TRS.

The discrete command link enables the ground to control radar tracking beacons, selection of telemetry transmitters, instrumentation data acquisition, and abort indications.

The capability of digital data updating enables the mission control center to update the computer and TRS to bring about a controlled re-entry at a pre-determined point, and allows timed shutdown of equipment controlled by DCS relays.

Physical Characteristics

The DCS consists of a receiver/decoder package and two relay boxes as illustrated in Figure 9-23 and 9-24, respectively. The three components are located in the electronic module of the adapter equipment section.

The receiver/decoder package is approximately 8 inches high, 8 inches wide, and 12 inches long. Both relay boxes are identical. Each relay box is approximately 2.25 inches wide, 5 inches high, and 3 inches deep. The combined weight of the receiver/decoder package and the two relay boxes is approximately 23 pounds. The receiver/decoder package contains two uhf receivers and a decoder while each of the two relay boxes contain eight relays.

General Description

The DCS receives Phase Shift Keyed (PSK) signals composed of a reference and an information signal. The information signal is in phase with the reference for a logical 1 and 180 degrees out of phase with the reference for a logical 0; thus establishing the necessary requirements for digital data.

Types of Commands

The DCS receives two types of digital commands: Real Time Commands (RTC) and Stored Program Commands (SPC). RTC causes relays within the DCS to be actuated. Nine of the 16 relays available for RTC are utilized to perform the following functions:

(1) Select the stand-by telemetry transmitter for real-time transmission.

(2) Select the stand-by telemetry transmitter for delayed-time transmission.

(3) Select the real-time telemetry and acquisition aid beacon transmission.

(4) Select real-time and delayed-time telemetry transmission.

(5) Actuate the adapter C-band radar beacon.

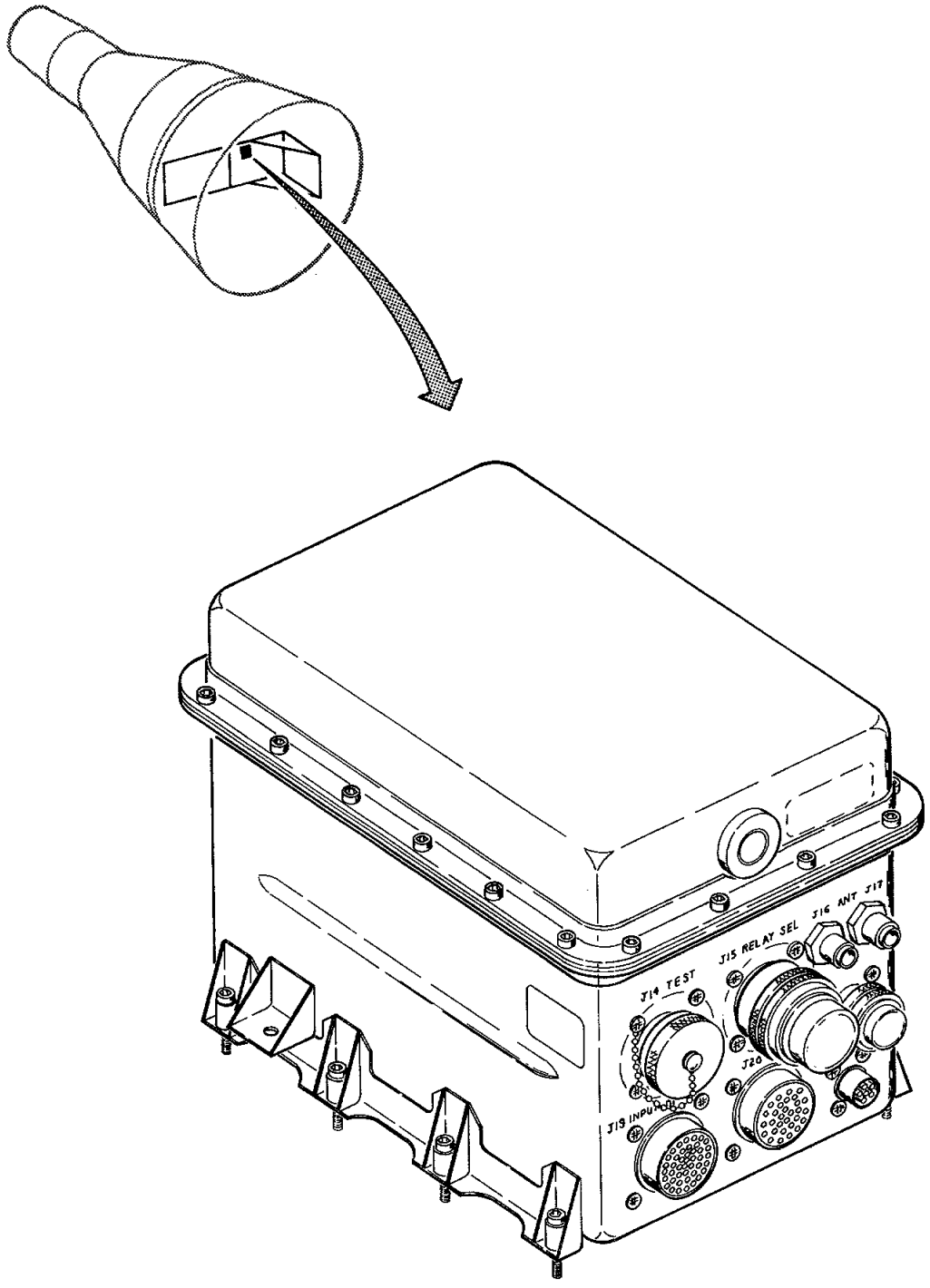


Figure 9-23 DCS Receiver/Decoder

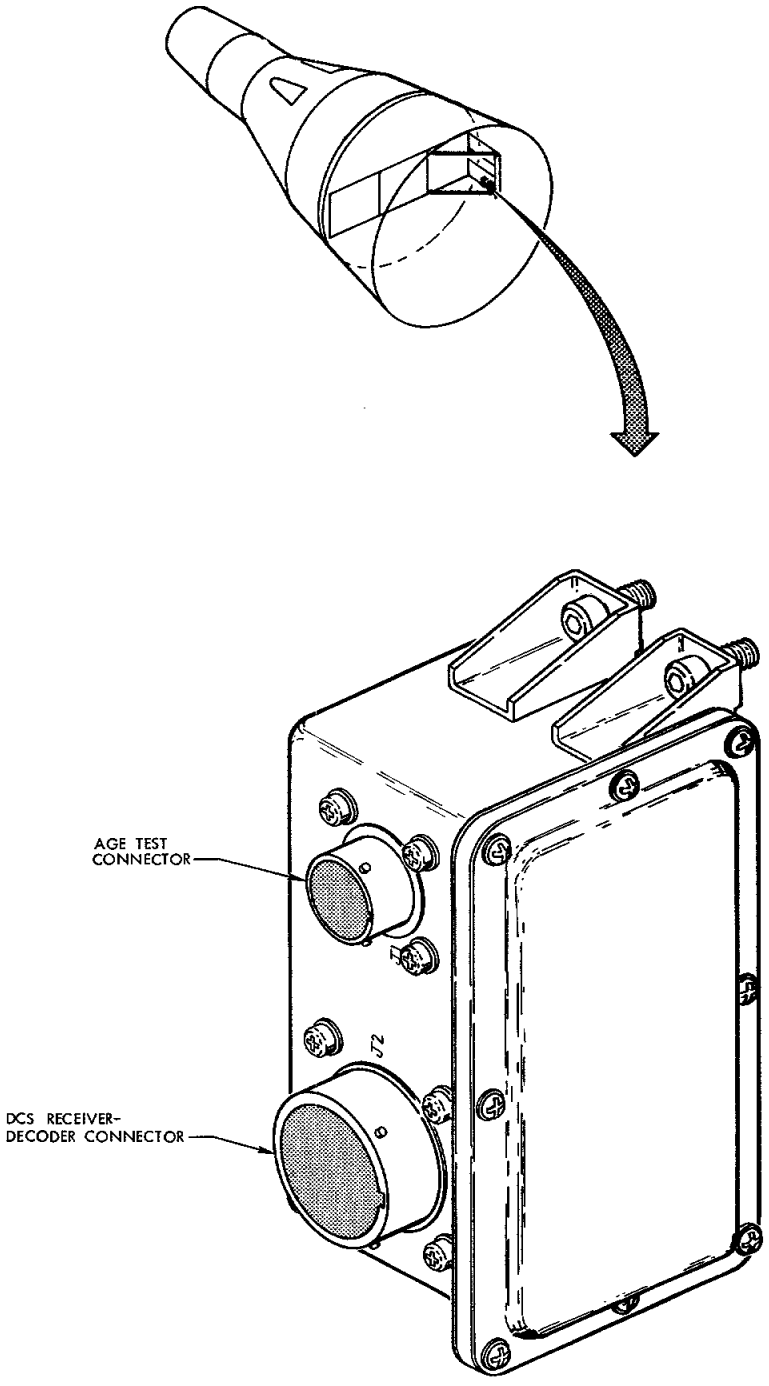


Figure 9-24 DCS Relay Box

(6) Actuate the re-entry C-band radar beacon.

(7) Illuminate the abort indicators.

(8) Actuate the playback tape recorder.

(9) Initiate calibration voltage for the PCM programmer.

The remaining seven relays are not utilized and perform no mission function. DCS channel assignments for the nine functions listed above may be different on each spacecraft.

When the spacecraft goes out of range of the ground station, equipment controlled by DCS channels may be shutdown by a signal applied from the TRS to reset the DCS relays. This condition is known as salvo. The DCS relays in one relay box may be reset by momentarily positioning the TAPE PLY BK switch to RESET.

Message Format and Modulation

The ground station transmits a 30-bit message for SPC and a 12-bit message for RTC. Each bit consists of five sub-bits. The five sub-bits are coded to represent a logical 1 or 0. The first three bits of each message designate the vehicle address. If the vehicle address is not correct, the DCS will reset itself and will not accept the message. If the vehicle address is accepted the sub-bit code will be automatically changed for the remainder of the message to reduce the probability of accepting an improper message.

The second three bits of each message designate the system address and identify the remainder of the message as being a RTC or one of the following SPC: computer update, TRS time to go (TTG) to T_R, or TRS TTG to equipment reset (T_X). If the message is a SPC, the last 24 bits will be a data word. If the SPC is a TRS TTG TO T_X command, the last eight bits are ignored by the TRS. In case of a

computer message, six bits of the data word contains the internal computer address and the remaining 18 contains information. Since a RTC consists of 12 bits, the six bits following the system address contain a 5-bit relay number and a 1-bit relay set-reset discrete.

The PSK modulation signals are 1 kc reference and a 2 kc information signal. The receiver output is the composite audio of the 1 kc and the 2 kc signals. The composite audio output is filtered to recover the 1 kc and the 2 kc signals. The phase comparator compares the 2 kc to the 1 kc signal. The output of the phase comparator is used to trigger a flip-flop to produce either a logical 1 or 0 sub-bit. The 1 kc reference signal is used to synchronize the DCS.

Operational Description

A block diagram of the DCS receiver/decoder is shown in Figure 9-25. Basically, the block diagram consists of a receiver, a decoder, and a power supply common to both sections.

The audio outputs of the two receivers are linearly summed in an emitter follower of the sub-bit detector module. The sub-bit detector converts the audio to sub-bits. The 5-stage shift register provides buffer storage for the output of the sub-bit code. When a proper sub-bit code exists in the shift register, the bit detector produces a corresponding 1 or 0 bit. The output of the bit detector is applied to the 24 stage shift register. The operation for RTC and SPC is identical up to the input to the 24 stage shift register.

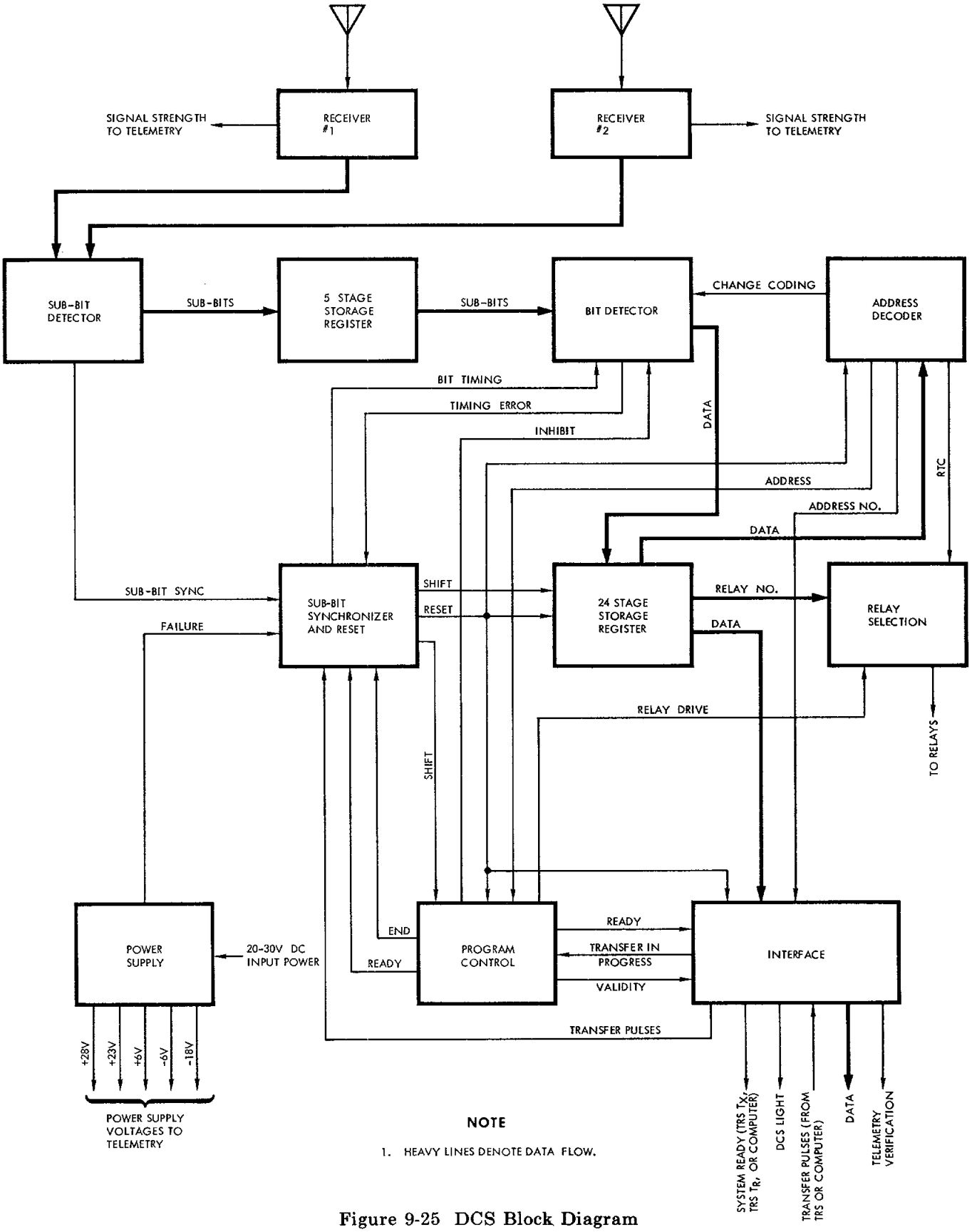
The sub-bit sync counter produces a bit sync output for every five sub-bits. The bit sync is used to gate the 24 stage shift register.

When a message is received, the vehicle address is inserted into the first three stages of the 24 stage shift register. If the vehicle address is correct, the vehicle address decoder circuit will produce an output to the bit detector which changes the acceptable sub-bit code for the remainder of the message. The next three bits of the message, the system address, are inserted into the first three stages of the 24 stage shift register, displacing the vehicle address to the next three stages. The system address decoder circuit identifies the specific address and sets up the DCS to handle the remainder of the message.

When the system address is recognized to be a RTC, the message is inserted into the first six stages of the 24 stage shift register and the system address and vehicle address are shifted into the next six stages. The RTC selection circuit recognizes the first stage of the 24 stage shift register to be a relay set or reset function and will apply a positive voltage to all set or reset relay coils, as applicable. The RTC selection gates select the proper relay from the relay number stored in the 24-stage shift register and provides an output which applies a power return to the coil of the selected relay.

When the system address is a SPC, the six address bits in the 24 stage shift registers are cleared and the remaining 24 bits of the message are placed into the register.

Assuming that the system address recognizes a TRS TTG to T_R message, the data flow would be as follows: The TRS T_R isolation amplifier, in the interface circuit, will apply a READY pulse to the TRS. The READY pulse sets up the TRS to transfer TRS TTG to T_R data from the DCS. When the TRS is ready to accept the data, it sends 24 shift pulses, at the TRS data rate, to the TRS input of the DCS.



NOTE

1. HEAVY LINES DENOTE DATA FLOW.

Figure 9-25 DCS Block Diagram

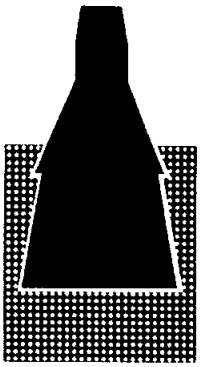
The data in the 24 stage shift register is then shifted out of the register through the DCS data isolation amplifier to the TRS. The DCS operations for computer updating and TRS TTG to T_X messages are similar to TRS TTG to T_R operations.

Salvo occurs when TRS TTG to T_X reaches zero. At $T_X = 0$, the TRS applies a signal to the TRS T_X input line of the DCS which causes the RTC selection circuits to reset the DCS relays.

After a SPC or RTC has been carried out by the DCS, a verification signal is supplied to the telemetry system for transmission to a ground station. The DCS indicator, on the instrument panel, illuminates when a SPC is transferred to the appropriate system.

Upon completion of data transfer or if the system to which the data was transferred fails to respond within 100 milliseconds, the DCS will reset in preparation for the next message. The DCS will also reset in the event of a timing error in transmission of data, or if the DCS power supply voltages become out of tolerance.

INSTRUMENTATION SYSTEM



Section X

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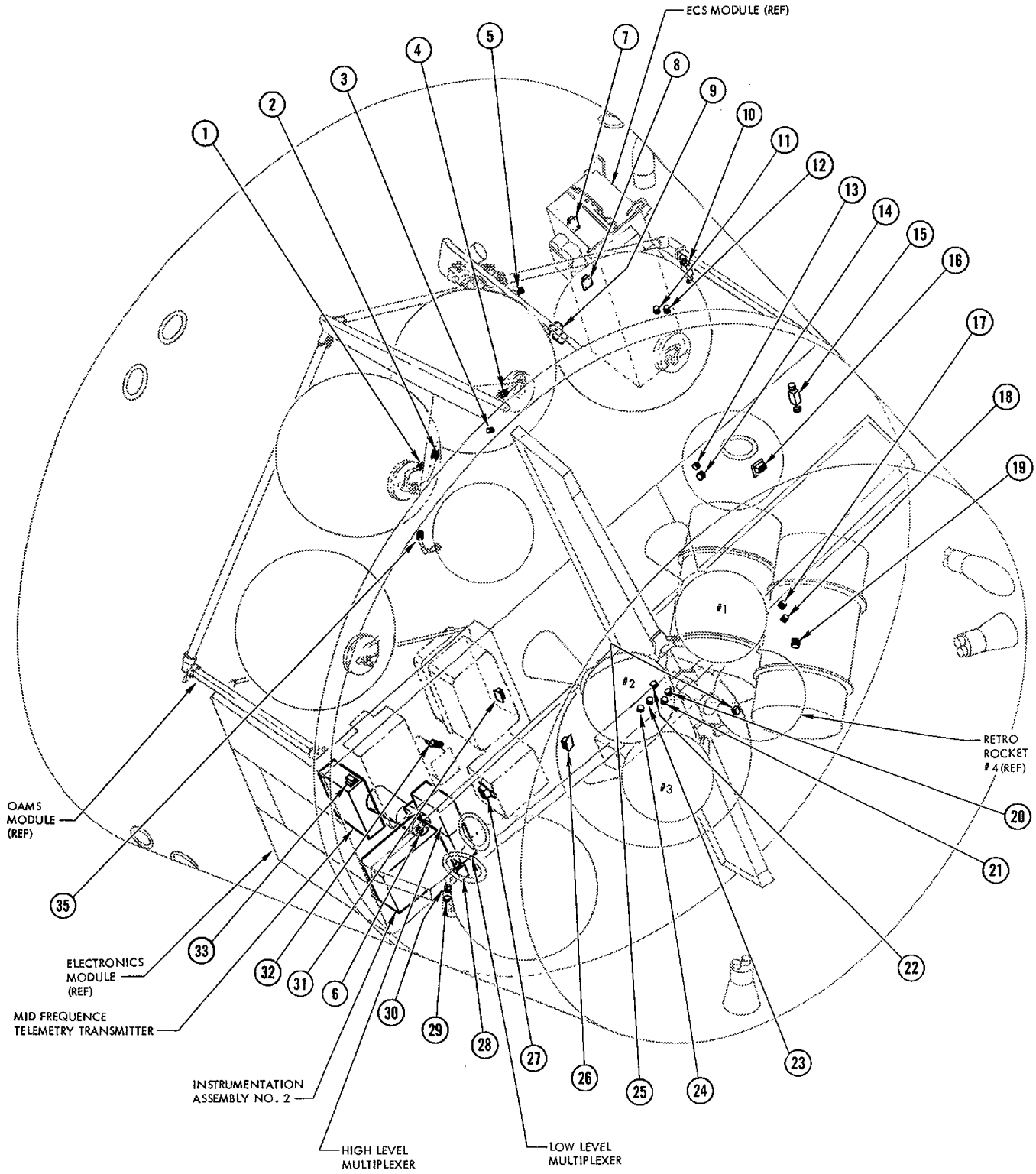


Figure 10-1 Instrumentation System Components (Sheet 1 of 3)

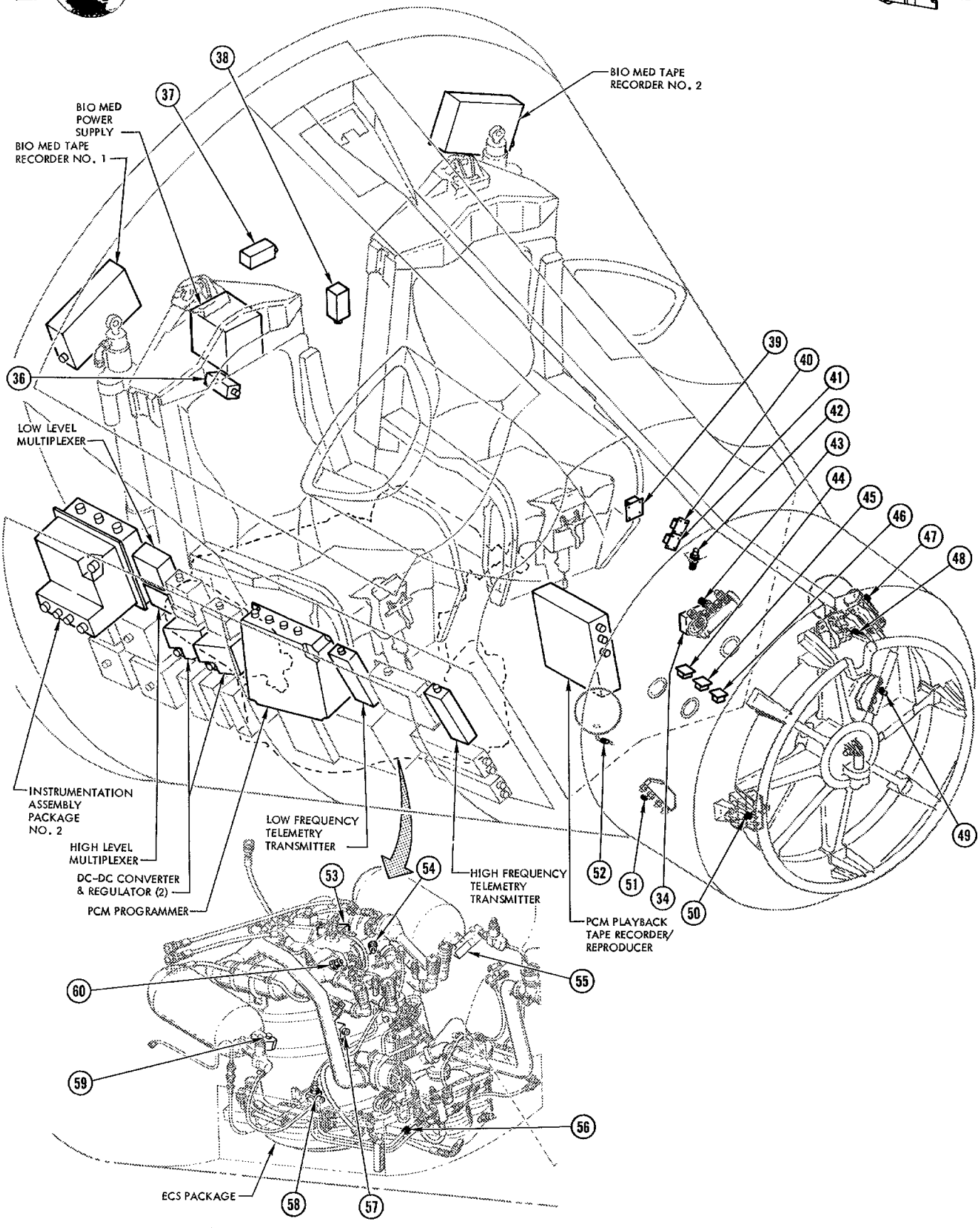
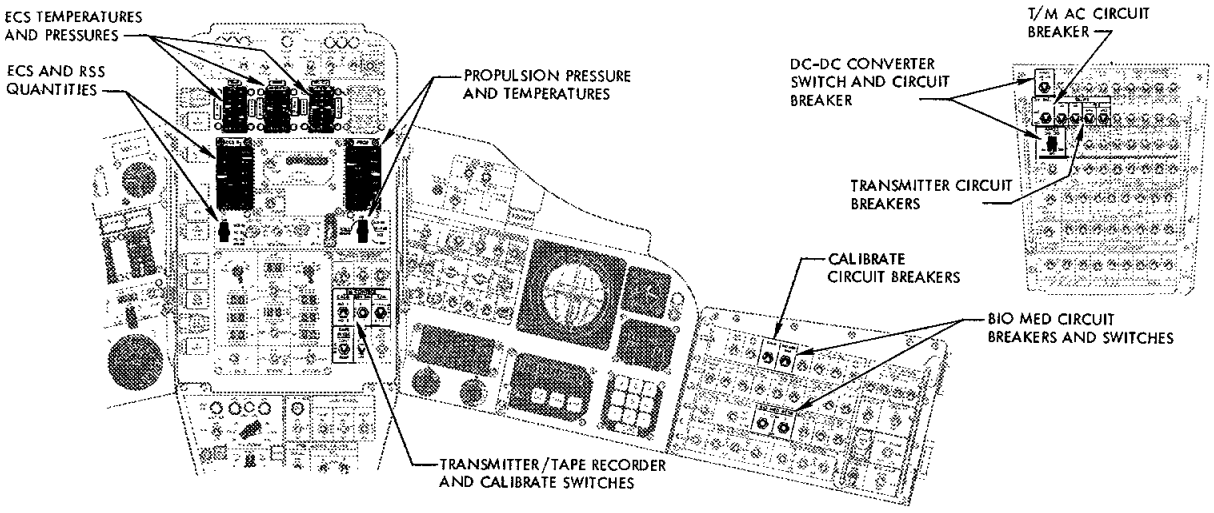


Figure 10-1 Instrumentation System Components (Sheet 2 of 3)



LEGEND

ITEM	PARAMETER	NOMENCLATURE
1	GC04	REG He AT OXID TANK TEMPERATURE
2	GB02	OXIDIZER FEED TEMPERATURE
3	GB01	FUEL FEED TEMPERATURE
4	GC03	REG He AT FUEL TANK TEMPERATURE
5	GC01	SOURCE He PRESSURE
6	GD06	TCA #10 HEAD TEMPERATURE
7	CJ01	PRIMARY COOLANT PUMP INLET PRESSURE
8	CJ02	SECONDARY COOLANT PUMP INLET PRESSURE
9	GC05	REGULATED He PRESSURE
10	CH03	SECONDARY COOLANT RADIATOR OUTLET TEMPERATURE
11	CD04	SECONDARY COOLANT TEMP AT OUTLET OF RADIATOR
12	CD03	PRIMARY COOLANT TEMP AT OUTLET OF RADIATOR
13	CA06	PRIMARY ECS O_2 SUPPLY BOTTLE TEMPERATURE
14	CA02	PRIMARY ECS O_2 TANK PRESSURE
15	CH02	PRIMARY COOLANT RADIATOR OUTLET TEMPERATURE
16	CA09	CRYO MASS QUANTITY (RSS-ECS)
17	CD02	SECONDARY COOLANT INLET TO F.C. SECT 2 TEMP
18	CD01	PRIMARY COOLANT INLET TO F.C. SECT 1 TEMP
19	CL01	WATER PRESSURE
20	BC03	F.C. H_2 TEMP AT HEAT EXCHANGER OUTLET
21	BA04	HYDROGEN TANK PRESSURE
22	BB05	F.C. O_2 TEMP AT HEAT EXCHANGER OUTLET
23	BA06	RSS H_2 SUPPLY BOTTLE TEMPERATURE
24	CA09	CRYO MASS QUANTITY (RSS-ECS)
25	HH01	RETRO ROCKET CASE TEMPERATURE
26	CA09	CRYO MASS QUANTITY (RSS-ECS)
27	LC09	ADAPTER C-BAND BCN PACKAGE TEMPERATURE
28	CA09	CRYO MASS QUANTITY (RSS-ECS)
29	BA05	RSS O_2 SUPPLY BOTTLE TEMPERATURE
30	BA02	OXYGEN TANK PRESSURE

LEGEND

ITEM	PARAMETER	NOMENCLATURE
31	LA05	DCS PACKAGE TEMPERATURE
32	LD01	ACQ AID BCN CASE TEMPERATURE
33	MC02	MID FREQ TM XMTR CASE TEMPERATURE
34	CC05	OXYGEN HIGH RATE
35	GC02	SOURCE He TEMPERATURE
36	KA01	Z ACCELERATION
37	KA02	X ACCELERATION
38	KA03	Y ACCELERATION
39	CB07	FWD COMPARTMENT ABSOLUTE PRESS
40	KB02	STATIC PRESSURE
41	CB01	CABIN PRESS TO FWD COMP.
42	CB02	CABIN AIR TEMPERATURE
43	HC03	REG N_2 PRESSURE-SYST A
44	DQ07	PITCH ATTITUDE-SYNCHRO REPEATER
45	DQ08	ROLL ATTITUDE-SYNCHRO REPEATER
46	DQ09	YAW ATTITUDE-SYNCHRO REPEATER
47	HC04	REG N_2 PRESSURE-SYST B
48	HC06	SOURCE N_2 PRESS-SYST B
49	HC02	N_2 SOURCE PRESS-RCS SYST B
50	HA02	RCS OXIDIZER FEED TEMP-SYST A
51	HC01	N_2 SOURCE PRESS-RCS SYST A
52	HC05	SOURCE N_2 PRESS-SYST A
53	CC06	CO_2 PARTIAL PRESSURE SENSOR
54	CC03	LEFT SUIT INLET AIR TEMP
55	CA03	ECS O_2 SUPPLY PRESS NO. 1-SEC
56	CK06	SUIT HEAT EXCHANGER INLET TEMP-PRI
57	CC01	LEFT SUIT PRESSURE
58	CC02	RIGHT SUIT PRESSURE
59	CA04	ECS O_2 SUPPLY PRESS NO. 2-SEC
60	CC04	RIGHT SUIT INLET AIR TEMP

Figure 10-1 Instrumentation System Components (Sheet 3 of 3)

SECTION X INSTRUMENTATION SYSTEM

SYSTEM DESCRIPTION

The Instrumentation System provides a means of data acquisition with respect to the performance and operation of the spacecraft throughout its mission. Data acquisition is defined as the sensing of specific conditions or events on board the spacecraft, displaying the derived data from these inputs to the crew and ground operation personnel, and recording and later processing this data for use in post flight reports and analysis. In this respect the data acquisition function is shared by all spacecraft systems, the ground operational support system, and the data processing facility.

Basically, the instrumentation parameters are divided into two categories: operational and nonoperational. Operational parameters are those which are necessary for determing the progress of the mission, assessing spacecraft status, and making decisions concerning flight safety. Nonoperational parameters are those which are required for post mission analysis and evaluation.

The basic components comprising the Instrumentation System are: sensors, signal conditioners, Multiplexer-Encoder System, and transmitters. Because the system is used to sense parameters of every spacecraft system, its components are located throughout the spacecraft as shown in Figure 10-1.

SYSTEM OPERATION

The purpose of the Instrumentation System is data acquisition with respect to the progress and condition of the spacecraft, necessitating its operation

throughout the mission. The Instrumentation System provides the capability of data acquisition and transmission to the ground stations. The data is supplied by all spacecraft systems. The basic operations by which the system fulfills its purpose are: to sense the various conditions and functions; convert them to proportional electrical signals (if applicable); condition the resulting signal (when necessary) to make it compatible with the encoding and multiplexing equipment; display pertinent data in the cabin; record data for delayed time (data-dump) transmission; and provide signals for real-time transmission to the ground station. An overall block diagram of the Instrumentation System is shown in Figure 10-2 and the power distribution is shown in Figure 10-3.

The system senses the prescribed parameters through the use of sensors which may be contained within the Instrumentation System or which may be an integral part of another system. Typical sensors include pressure transducers, accelerometers, and temperature sensors. Signals may also be obtained from such functions as switch and relay actuations, and from electronic package monitor points. Sensors and signal sources are shown in block diagram form on the applicable data source system illustration.

The majority of the signals acquired are usable for the spacecraft cabin indicators and/or the encoding equipment without alteration. Some of them, however, are routed to signal conditioning packages (instrumentation assemblies) where their characteristics and/or amplitudes are changed. The resulting signals, as well as those from the other sensors, are of four basic types: low-level

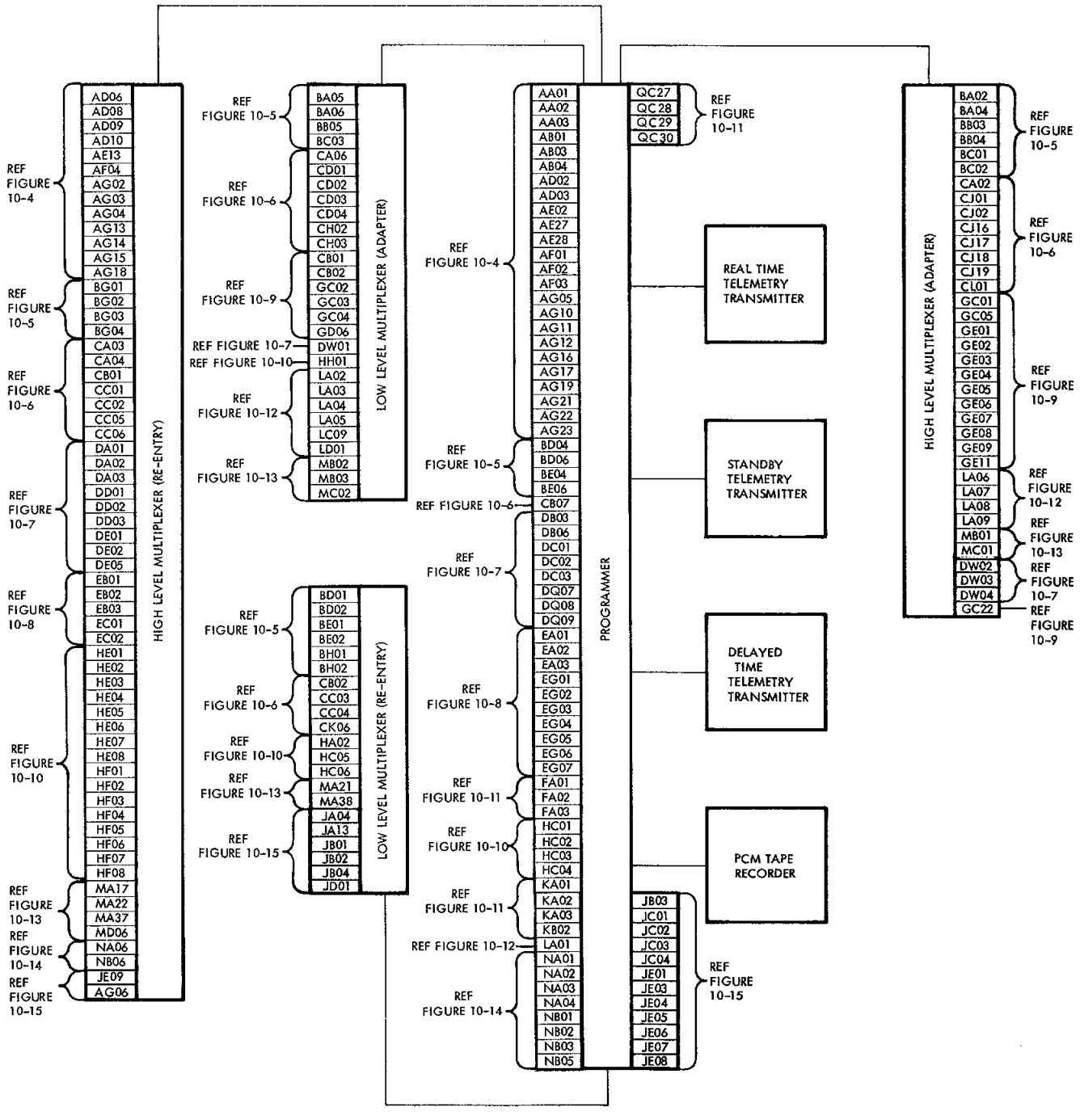


Figure 10-2 Instrumentation System Signal Flow Block Diagram

PROJECT GEMINI

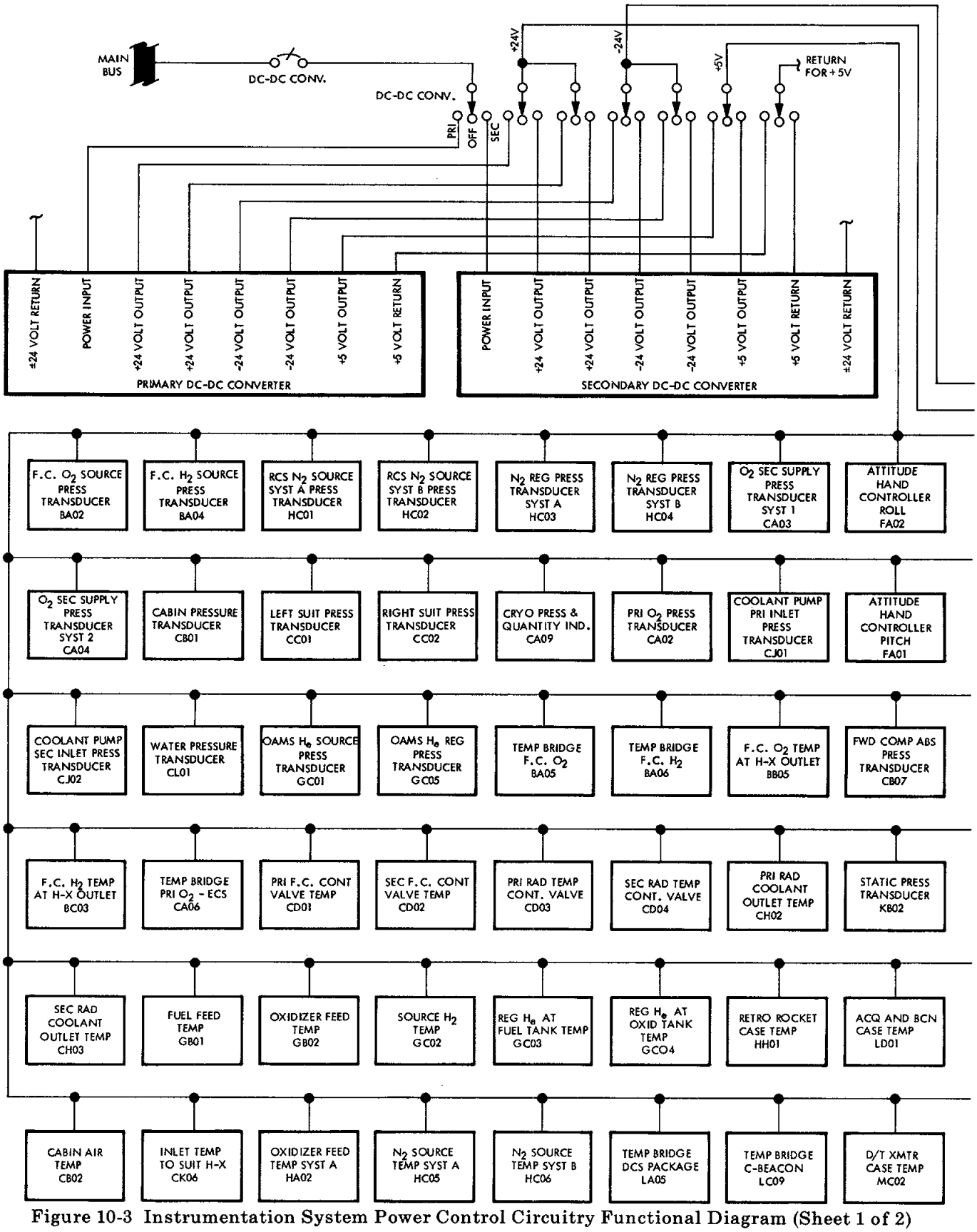


Figure 10-3 Instrumentation System Power Control Circuitry Functional Diagram (Sheet 1 of 2)

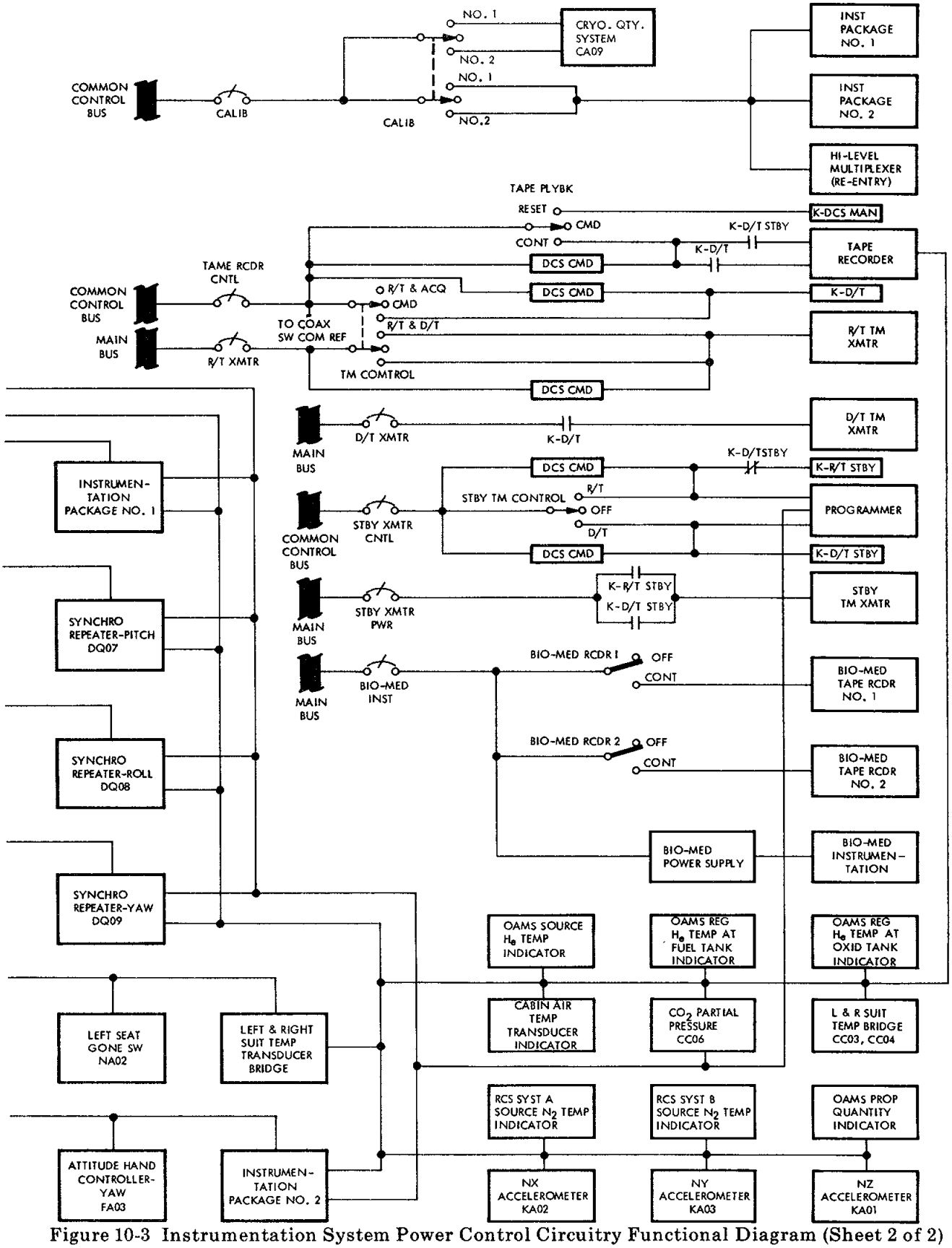


Figure 10-3 Instrumentation System Power Control Circuitry Functional Diagram (Sheet 2 of 2)

(0-20 mv dc), high-level (0-5 vdc), bi-level (0 or 28 vdc), and bi-level pulse (28 or 0 vdc). Signals of selected parameters are supplied to the cabin indicators, while signals of all parameters are supplied to the Multiplexer/Encoder System. The Multiplexer/Encoder System converts the various spacecraft analog and digital signals to a serial binary-coded digital signal for presentation to the data-dump tape recorder and the real-time telemetry transmitter. The tape recorder records a portion of the real-time data from the programmer at a tape speed of 1 7/8 inches per second and, upon command, will play back the data for transmission to a ground station, at a speed of 41.25 ips (22 times the recording speed).

Four physiological functions are monitored for each pilot. All of the measurements are supplied as real-time data, while only one is supplied as delayed-time data. In addition, most of the measurements are recorded by two special (biomed) tape recorders.

During pre-launch operations, data acquisition is accomplished by use of hardlines attached to the spacecraft umbilical and by telemetry. Between launch and orbital insertion, data acquisition is via the real-time telemetry transmitter. While the spacecraft is in orbit, data is acquired via the real-time telemetry transmitter for the period while the spacecraft is within range of a ground station. Data during the period while the spacecraft is out of range of a ground station is recorded on the PCM recorder and played back via the delayed-time telemetry transmitter while the spacecraft is within range of a ground station. A more detailed description of the telemetry transmitters is given in Section IX.

The paragraphs to follow, present a brief description of all instrumentation parameters. The parameters are described in groups identified by their applicable data source system. It should be noted that although most of the parameters are applicable to all spacecraft, the following parameters is for spacecraft 8 specifically.

SEQUENTIAL SYSTEM PARAMETERS

A functional diagram showing the Sequential System parameters is presented in Figure 10-4. The Instrumentation System monitors 41 sequential events and Sequential System parameters. Each parameter is described below individually, or as part of a group of related parameters.

The Time Reference System (TRS) supplies three 24-bit digital words to the 24-bit shift register of the PCM programmer. These three signals are: time since lift-off (AA01, AA02) and time to retrograde (AA03). Time since lift-off is referenced to the launch vehicle lift-off signal and provides time correlation for the data tape recorders. Time to retrograde (AA03) indicates the time remaining before retrofire initiation by the TRS. This signal is used to verify that the correct retrofire time has been inserted into the TRS by ground command or by the pilots.

Launch vehicle second stage cut-off (AB01) is monitored for ground station indication of this event. This parameter is provided by a signal from the spacecraft IGS computer to a bi-level channel of the programmer.

Launch vehicle/spacecraft separation (AB03) is indicated to the ground station when any two of the three spacecraft/launch vehicle limit switches close, ener-

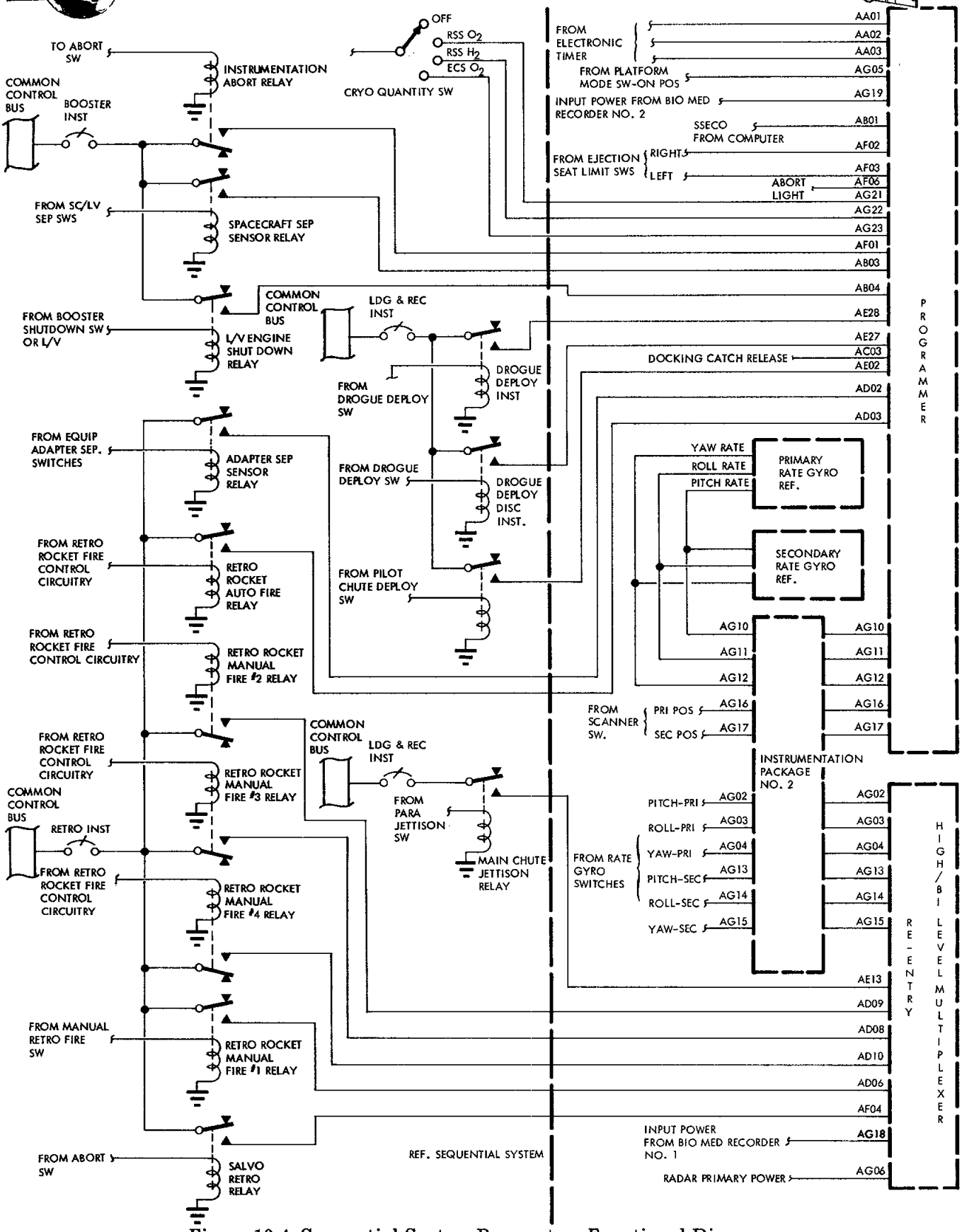


Figure 10-4 Sequential System Parameters Functional Diagram

gizing the spacecraft separation relays. Actuation of any two of the three relays applies 28 vdc to a bi-level channel of the programmer.

Rendezvous Radar primary power (AG06) is a high-level signal applied to the re-entry high-level multiplexer. This signal originates when the Rendezvous Radar primary power switch is energized. Docking catch release (AC03) originates during the separation sequence after docking has occured. Actuation of the release catch microswitch energizes the docking catch release relay in the instrumentation relay panel, and provides a bi-level signal to the programmer.

Equipment section separation (AD02) is monitored to indicate a safe condition for retrograde prior to manual initiation or ground command of retrofire as a backup to the automatic system. This signal is originated when any two of the three separation sensors close, energizing the equipment section separation relays. Actuation of two of the three relays applies 28 vdc to a bi-level channel of the programmer.

The retrorocket ignition commands are monitored by ground stations to obtain data for calculation of expected re-entry trajectory. Automatic (AD03) and manual (AD06) ignition commands are monitored. Parameters are obtained from the ignition command of the four retrorockets individually; AD09, rocket 2; AD08; rocket 3; AD10, rocket 4. The manual and automatic retrofire commands indicate retrorocket 1 fire. The signals, 28 vdc, are applied to the re-entry high-level multiplexer.

Channel 10 of the Digital Command System is used by the ground station to relay the abort command to the spacecraft. Verification of ABORT light illumination

is by (AF06) parameter.

Indication that the pilot actuated abort (AF01) is supplied to the ground station. The signal is originated when the abort handle is moved to the ABORT position actuating a limit switch which energizes the instrumentation abort relays. Actuation of one of the relays applies a signal to a bi-level channel of the programmer.

In case of pilot ejection during an abort, left (AF03) and right (AF02) ejection seat gone signals are relayed to the ground station. The signals are originated at the time the ejection seats leave the spacecraft closing the corresponding limit switch and applying the signals to the bi-level channels of the programmer.

Confirmation of salvo retrofire is given to the ground station in case of an abort. A signal is applied to a bi-level channel of the re-entry high-level multiplexer when the salvo retrograde relay is energized.

Indication of booster cut-off command (AB04) is given to the ground station when pilots move the ABORT handle to the SHUTDOWN position, actuating a limit switch. This energizes a relay applying 28 vdc to a bi-level channel of the programmer.

Ground indication of pilot parachute deployment (AE02) is provided via a bi-level channel of the programmer. The signal is originated when a lanyard from the parachute actuates a toggle switch, energizing the pilot parachute deployed instrumentation relay.

The parachute jettisoned (AE13) signal is initiated when the pilot depresses the CHUTE JETT switch energizing redundant main parachute jettison relays. The relays apply a 28-vdc signal to a bi-level channel of the re-entry high-level multiplexer.

Platform mode selection (AG05) is indicated to a ground station. Any position other than OFF on the PLATFORM mode switch will apply a signal to a bi-level channel of the programmer.

Primary (AG16) or secondary (AG17) horizon scanner operation can be monitored by the ground station via bi-level channels of the programmer.

Primary pitch (AG02), roll (AG03), and yaw (AG04) and secondary pitch (AG13), roll (AG14), and yaw (AG15), rate gyro operation is monitored to indicate an on or off condition. Each signal is applied to a signal conditioner whose output is applied to a bi-level channel of the high-level multiplexer.

Pitch (AG10), roll (AG11), and yaw (AG12) rate gyro (primary or secondary depending which is operational) outputs are applied to three signal conditioners. Each of the signal conditioners is a transistor switch providing no output for an input of 0-0.325 volts and a 16.5 volt output for an input greater than 0.325 volts. The conditioned signals are applied to bi-level input channels of the programmer.

Bio-medical tape recorder on-off signals (AG18, AG19) are used for time correlation of the recorded bio-medical data with the telemetry data. An on-off indication is provided to the playback recorder and to telemetry by a bi-level

signal to the programmer (AG19) and re-entry high-level multiplexer (AG18). Drogue parachute deployment (AE27) and drogue release (AE28) can be verified by the ground station via bi-level channels of the programmer. The signals are initiated when the HI-ALT DROGUE switch is depressed.

The selected cryogenic quantity switch position is indicated to the ground station by AG21 (Reactant Supply System oxygen), AG22 (Reactant Supply System hydrogen), and AG23 (Environmental Control System oxygen) to allow the ground station to identify the reading of CA09 described under Environmental Control System.

ELECTRICAL POWER SYSTEM PARAMETERS

Figure 10-5 shows a functional diagram of the Electrical Power System parameters. Approximately 24 Electrical Power System parameters are monitored by the Instrumentation System. The parameters are listed and described in the following subparagraphs.

Fuel cell oxygen (BA02) and hydrogen (BA04) tank pressures are monitored by dual potentiometer pressure transducers installed as part of the fuel cell system. Each dual transducer provides one output to the adapter high-level multiplexer and the other output drives an indicator on the instrument panel in the cabin.

To evaluate proper operation of the fuel cell, stack 1A (BD01), 1B (BD02), 2A (BE01), 2B (BE02) and section 1 (BH01) and 2 (BH02) currents are monitored and transmitted to the ground station. Stack C currents are obtained mathematically by subtracting section A and B currents from the corresponding section current. The signals being monitored originate from 50 millivolt shunts. The

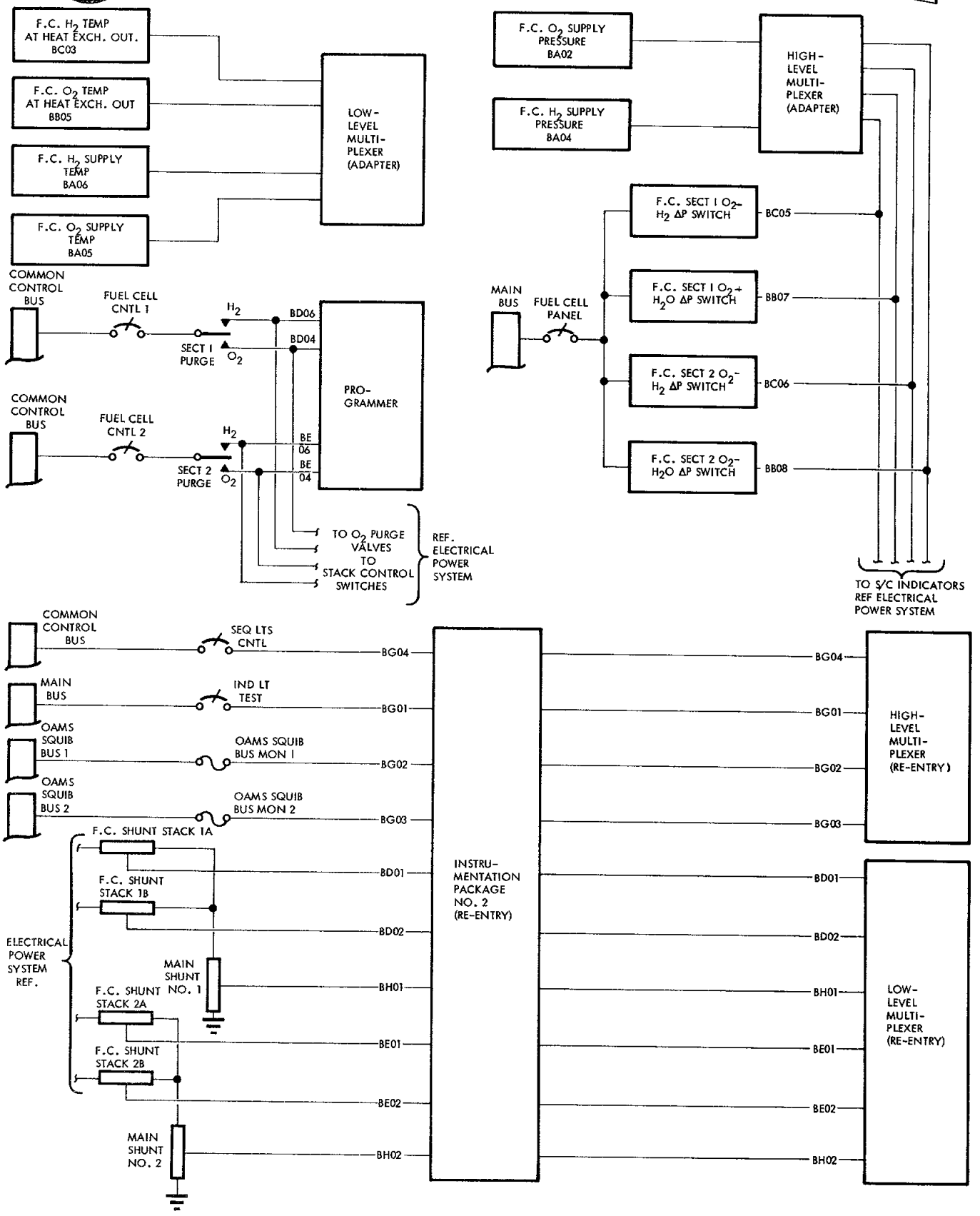


Figure 10-5 Electrical Power System Parameters Functional Diagram

shunts are installed at the main buses for the section, and in the lines from stacks A and B to the main buses for stack A and B currents. Each of these signals is conditioned to a 0 to 20 millivolt signal which is directly proportional to the input current and then applied to the re-entry low-level multiplexer.

The following parameters relate to the ground station information regarding spacecraft main, squib and control bus voltages: BG01 (main), BG02 (squib 1), BG03 (squib 2), BG04 (control bus). Each of these parameters is conditioned and then applied to the re-entry high-level multiplexer.

The Reactant Supply System (RSS) O_2 (BA05) and H_2 (BA06) supply bottle temperatures are monitored by means of two temperature sensors located on each supply bottle. The output of the sensors is applied to the adapter low-level multiplexer.

Fuel cell section 1 O_2 to H_2 (BC05), section 1 O_2 to H_2O (BB07), section 2 O_2 to H_2 (BC06), and section 2 O_2 to H_2O (BB08) differential pressures are monitored by a pressure-sensitive switch installed within the fuel cell to provide for safe operation monitoring capability of the fuel cell by the ground station. The outputs of the pressure switch is applied to bi-level channels of the adapter high-level multiplexer.

Oxygen (BB05) and hydrogen (BC03) temperatures at the outlet of the heat exchanger are monitored and relayed to the ground station via the adapter low-level multiplexer.

To provide an aid in evaluating fuel cell operation by the ground station,

section 1 O_2 (BD04), section 2 O_2 (BE04), section 1 H_2 (BD06), and section 2 H_2 (BE06) purging is monitored. The signals are actuated by the pilot by placing the corresponding section purge switch to the H_2 or O_2 position. The signals are applied to the bi-level channels of the programmer.

ENVIRONMENTAL CONTROL SYSTEM PARAMETERS

A functional diagram showing the Environmental Control System (ECS) instrumentation parameters is presented in Figure 10-6. Twenty-eight parameters and RSS/ECS quantities associated with the ECS are monitored by the Instrumentation System and relayed to the ground station for analysis.

The primary oxygen tank pressure (CA02) is telemetered to the ground station and displayed in the spacecraft cabin. The signals originate from a dual potentiometer pressure transducer installed as part of the ECS. The signal is relayed to the ground station via the adapter high-level multiplexer.

A differential pressure transducer is used to sense cabin to forward compartment pressure differential (CB01). The transducer has a dual output used for cabin indications and for transmission to the ground station via the re-entry high-level multiplexer.

Left (CC01) and right (CC02) suit to cabin differential pressure is displayed in the spacecraft cabin and telemetered to the ground station. Dual potentiometer pressure transducers serve as the signal source. The output of each transducer is applied to the cabin indicator and to the re-entry high-level multiplexer.

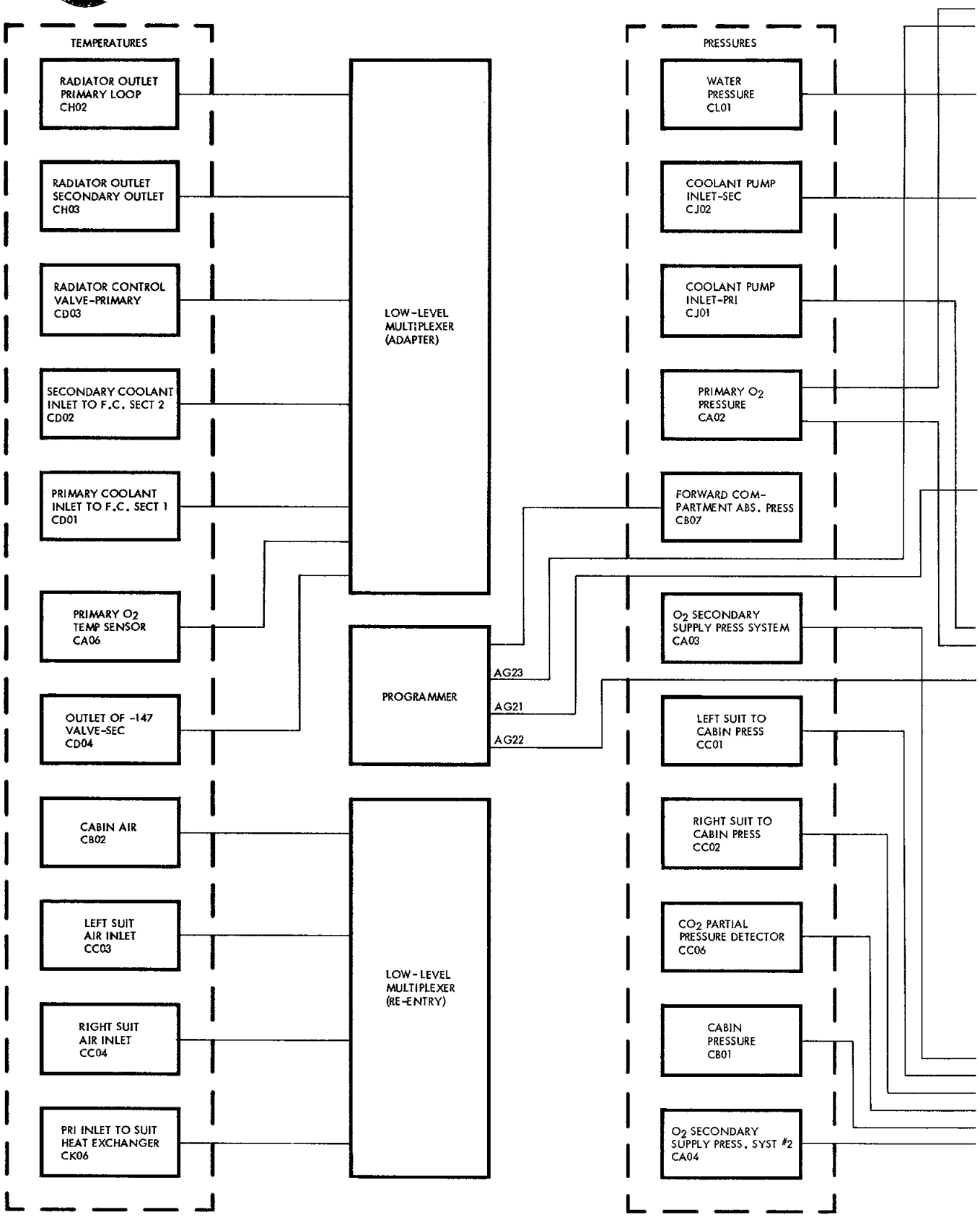


Figure 10-6 Environmental Control System Parameter Functional Diagram (Sheet 1 of 2)

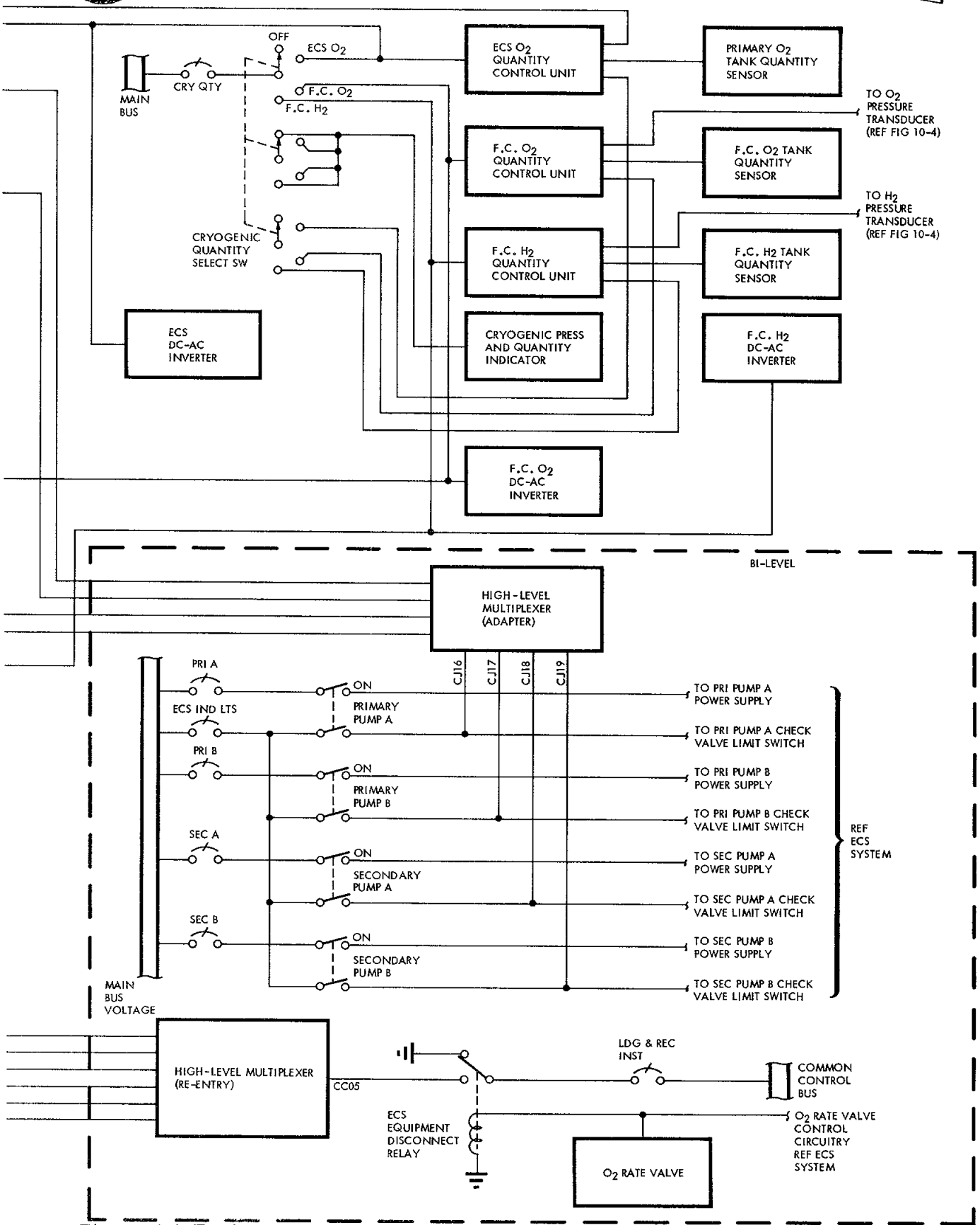


Figure 10-6 Environmental Control System Parameter Functional Diagram (Sheet 2 of 2)

The ground station is informed of an O_2 high rate condition by CC05. This signal is originated when the spacecraft CABIN FAN switch is placed in the O_2 HI RATE position, when manual O_2 high rate is selected by the pilot, or when the suit pressure drops below 3.3 psia and O_2 high rate is automatically selected. The signal is applied to a bi-level channel of the re-entry high-level multiplexer.

To assure that a safe supply of oxygen is available to the pilots, CO_2 partial pressure (CC06) is monitored indicating the percentage of carbon dioxide with respect to the total pressure of gas in the suits. CO_2 partial pressure is displayed in the spacecraft cabin and applied to the re-entry high-level multiplexer.

Primary and secondary coolant temperatures are monitored at various locations within the coolant loop to evaluate system performance. Coolant temperatures are monitored at the primary coolant inlet to section 1 of the fuel cell (CD01), secondary coolant inlet to section 2 of the fuel cell (CD02), the radiator control valve in the primary loop (CD03), the secondary loop (CD04), radiator outlet in the primary loop (CH02), and radiator outlet in the secondary loop (CH03).

To relay information concerning proper operation of the coolant loop and pumps, to the ground station, primary (CJ01) and secondary (CJ02) coolant pump inlet pressures are monitored. The outputs of the transducers are applied to the adapter high-level multiplexer.

The condition of the primary and secondary coolant pumps is monitored by CJ16 (primary pump A), CJ17 (primary pump B), CJ18 (secondary pump A), and CJ19

(secondary pump B). The signal is originated when the corresponding coolant pump is actuated, and is applied to bi-level channels of the adapter high-level multiplexer.

To insure safe operation of the fuel cell, water pressure (CL01) is monitored at the output of the fuel cell. The signal is applied to the adapter high-level multiplexer.

The coolant inlet temperature to the suit heat exchanger (CK06) is monitored to relay to ground stations information concerning the environmental condition of the pilots. The output of the temperature sensor is applied to the re-entry low-level multiplexer.

The position of the cryogenic quantity select switch is monitored to identify parameter CA09. The parameter CA09 indicates ECS O_2, RSS O_2, or RSS H_2 quantity depending upon the position of the cryogenic quantity select switch. The position of the selector switch is indicated to the ground station by AG21 (F.C. O_2), AG22 (F.C. H_2), and AG23 (ECS O_2). The signals are applied to bi-level channels of the programmer. The parameter CA09 is also applied to the programmer and is displayed in the spacecraft cabin.

Secondary O_2 supply pressures are monitored in the number 1 (CA03) and number 2 (CA04) systems. The transducers are installed as part of ECS secondary O_2 supply assemblies. The outputs of the pressure transducers are applied to the re-entry high-level multiplexer.

As an aid in calculating ECS O_2 quantity, the primary O_2 supply bottle tempera-

ture (CA06) is monitored and applied to the adapter low-level multiplexer.

To provide the capability for the ground station to monitor the environmental condition of the cabin and to provide an aid for evaluating suit pressure, a cabin air temperature transducer (CB02), and a forward compartment absolute pressure transducer (CB07) is provided. Absolute pressure is applied to the programmer and cabin temperature is applied to the re-entry low-level multiplexer. Cabin temperature is also displayed in the spacecraft cabin.

To further evaluate system performance and pilot environmental condition, the air entering the suit circuit is monitored with respect to temperature by 2 dual temperature sensors (1 for each suit circuit). The temperatures are displayed in the spacecraft cabin and are applied to the re-entry low-level multiplexer as CC03 (left suit), and CC04 (right suit).

INERTIAL GUIDANCE SYSTEM PARAMETERS

Figure 10-7 shows a block diagram of the Inertial Guidance System (IGS) parameters except the digital computer functions. The Instrumentation System monitors 8 IGS parameters and handles approximately 200 computer words.

The Instrumentation System monitors the computer modes of operation; pre-launch, ascent, catch-up, rendezvous, re-entry, and touchdown. Important functions or parameters (approximately 200) are monitored during each mode of operation. This information is used during post mission analysis and is applied to the programmer.

In addition to the digital computer words, the Instrumentation System monitors the following IGS parameters.

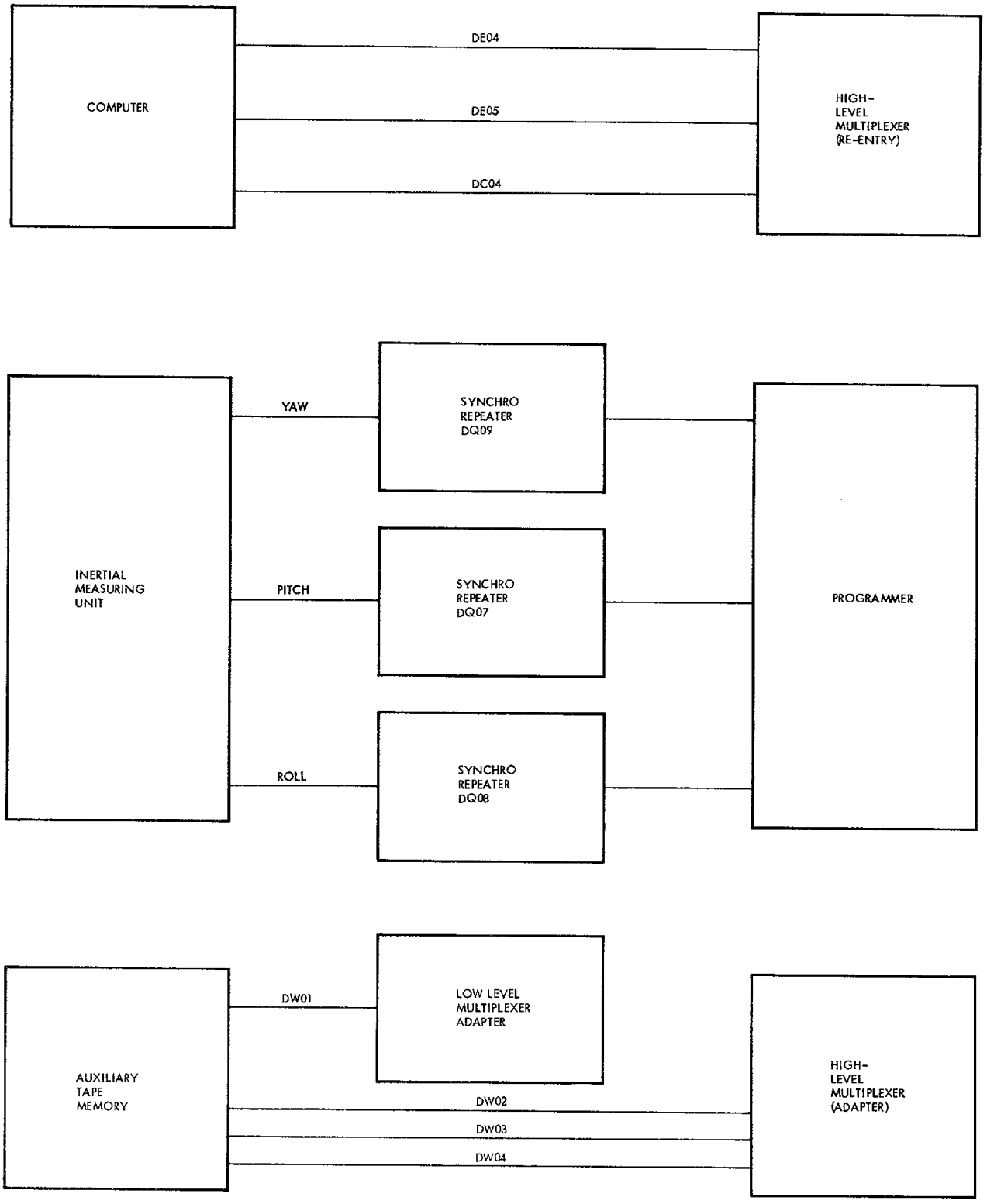


Figure 10-7 Inertial Guidance System Parameters Block Diagram

Inertial platform attitudes are monitored to provide ground stations with attitude data during flight. Roll (DQ08), pitch (DQ07), and yaw (DQ09) signals are taken from the Inertial Measuring Unit (IMU), conditioned by synchro repeaters, and applied to the programmer.

IGS regulated power is monitored at two points: 26 vac (DE04) and 10.2 vdc (DE05). These voltages are conditioned and then applied to the re-entry high-level multiplexer.

Computer START light (DC04) malfunction signal is the only malfunction detection parameter monitored. This signal is used for display in the spacecraft cabin and applied to the bi-level channel of the programmer.

Auxiliary Tape Memory environmental conditions are monitored by case temperature (DW01) and internal pressure (DW02). Motor drive inhibit (DW03) and verification-reproduction plus 20 volts (DW04) parameters indicate the mode of operation (off, standby, read, or write).

ATTITUDE CONTROL AND MANEUVERING ELECTRONICS PARAMETERS

A block diagram showing the Attitude Control and Maneuvering Electronics (ACME) System parameters is shown in Figure 10-8. Fifteen ACME parameters are monitored by the Instrumentation System.

Spacecraft rates in pitch (EA01), roll (EA02), and yaw (EA03) are monitored to allow evaluation of the rate control portion of the ACME. Each signal from the rate gyro package is conditioned by a phase sensitive demodulator and then applied to the high-level channels of the programmer. Primary and secondary rate gyro signals are parallel summed and monitored on the same channels.

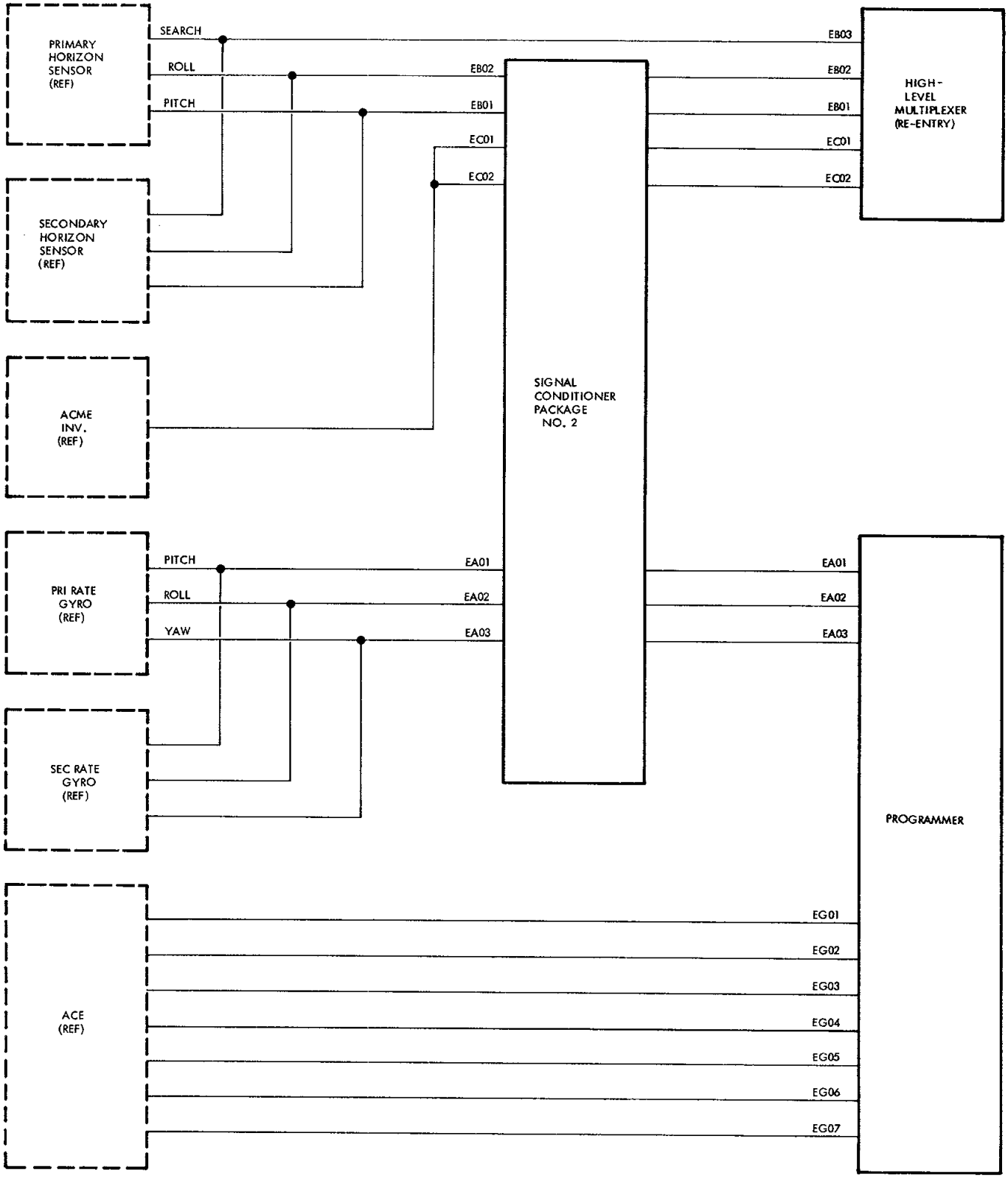


Figure 10-8 Attitude Control & Maneuvering Electronics Parameters Block Diagram

Horizon Sensor operation is monitored with respect to pitch (EB01) and roll (EB02) outputs, and search mode of operation (EB03). Pitch and roll parameters are monitored to verify inertial platform alignment for the retrograde phase of the mission. These parameters (EB01, EB02) provide pitch and roll attitudes from the Horizon Sensor during orbital flight when the platform has been shut down to conserve electrical power. The signals originate when the SCANNER switch is in the PRI or SEC position. The pitch and roll outputs are conditioned and then applied to the re-entry high-level multiplexer. The search mode of operation is monitored to determine whether the Horizon Sensor unit is in the search mode, or has sensed the horizon. This signal illuminates the SCANNER light in the cabin and is also applied to a bi-level channel of the re-entry high-level multiplexer.

ACME inverter 26 vac voltage (EC01) and frequency (EC02) is monitored for post mission analysis. The signals are conditioned and then applied to the re-entry high-level multiplexer.

The following attitude control modes are monitored depending upon the position of the ATTITUDE CONTROL switch; HOR SCAN (EG01), RATE CMD ORBIT (EG02), DIRECT (EG03), PULSE (EG04), RATE CMD RNTY (EG05), RE-ENTRY (EG06), and PLATFORM (EG07). The signals are applied to bi-level channels of the programmer.

ORBIT, ATTITUDE AND MANEUVERING SYSTEM PARAMETERS

The Orbit, Attitude and Maneuvering System (OAMS) parameters are shown in Figure 10-9 in block diagram form. A brief description of each of the 19 parameters is given in the paragraphs to follow.

To insure that adequate propellant pressure is available for OAMS, helium source

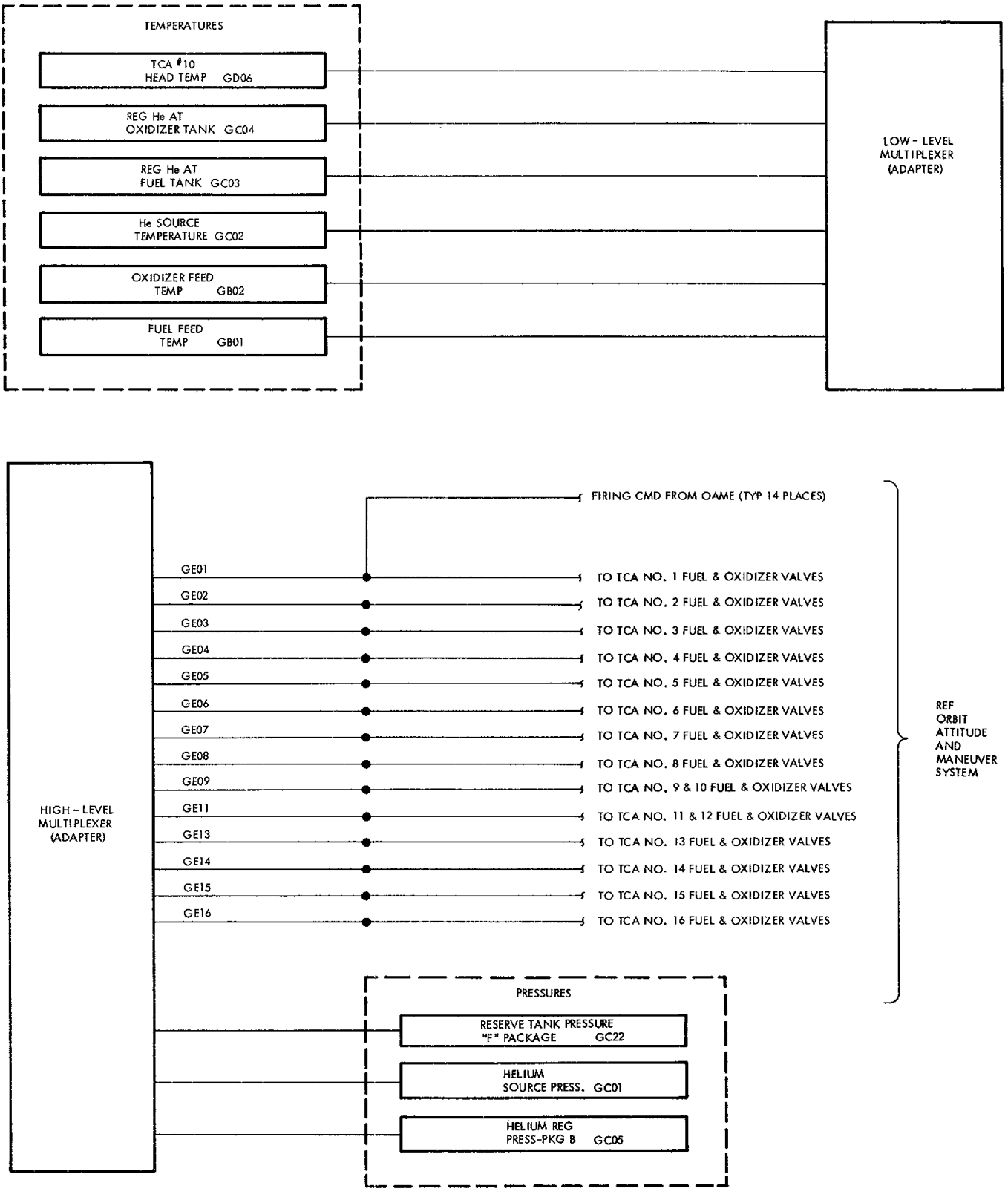


Figure 10-9 Orbit, Attitude & Maneuvering System Parameters Block Diagram

pressure (GC01) is monitored. The signal originates from a dual potentiometer pressure transducer at the helium pressurant tanks. One output is applied to the adapter high-level multiplexer and the other is used to drive an indicator in the spacecraft cabin.

The propellant feed temperature at the fuel (GB01) and oxidizer (GB02) feed lines is monitored to verify that propellant aboard is above freezing temperature and is available for use. The signals originate from two individual temperature sensors and are applied to the adapter low-level multiplexer.

To allow monitoring capability of the helium source temperature (GC02), a temperature sensor is installed on the helium supply line at the supply tank. The output is applied to the adapter low-level multiplexer. A separate sensor is installed to drive an indicator in the spacecraft cabin.

Temperature of the pressure regulated helium is monitored at the fuel (GC03) and oxidizer (GC04) tank inlet lines. The outputs of these temperature sensors is applied to the adapter low-level multiplexer. Two additional sensors are installed to drive indicators in the spacecraft cabin.

Regulated helium pressure (GC05) is monitored by a dual potentiometer pressure transducer. One of the outputs is applied to the adapter high-level multiplexer, and the other is used to drive a cabin indicator. Reserve tank pressure (GC22) monitors pressure available to the reserve fuel tank.

To provide an indication of maximum Thrust Chamber Assembly (TCA) temperature, TCA 10 (GD06) injector head temperature is monitored. This signal is applied to the adapter low-level multiplexer.

To provide ground station monitoring capability of TCA firing, the following TCA solenoid command signals are applied to bi-level channels of the adapter high-level multiplexer: GE01 (TCA 1), GE02 (TCA 2), GE03 (TCA 3), GE04 (TCA 4), GE05 (TCA 5), GE06 (TCA 6), GE07 (TCA 7), GE08 (TCA 8), GE09 (TCA 9, 10), GE11 (TCA 11, 12), GE13 (TCA 13), GE14 (TCA 14), GE15 (TCA 15), and GE16 (TCA 16).

RE-ENTRY CONTROL SYSTEM PARAMETERS

Figure 10-10 shows in block diagram form the Re-entry Control System parameters. Some 24 parameters are monitored by the Instrumentation System to provide for ground station observation of proper system performance.

Nitrogen source pressure, HC01 (system A) and HC02 (system B), and nitrogen source temperature, HC05 (system A) and HC06 (system B) are monitored. Pressure is sensed by two dual pressure transducers. One of the outputs of each transducer is used to drive a cabin indicator, and the other is applied to the programmer. Outputs of the temperature sensors are applied to the re-entry low-level multiplexer and are used to drive a spacecraft cabin indicator.

Because the oxidizer has a more critical temperature range than fuel, its temperature is measured to insure that both fuel and oxidizer are within the proper temperature range for use in the Re-entry Control System. The oxidizer feed temperature (HA02) is applied to the re-entry low-level multiplexer.

Regulated nitrogen pressure is monitored for system A (HC03) and system B (HC04). The outputs of the pressure transducers is applied to the programmer.

To provide for ground station monitoring capability of proper RCS TCA firing, firing commands are applied to bi-level channels of the re-entry high-level

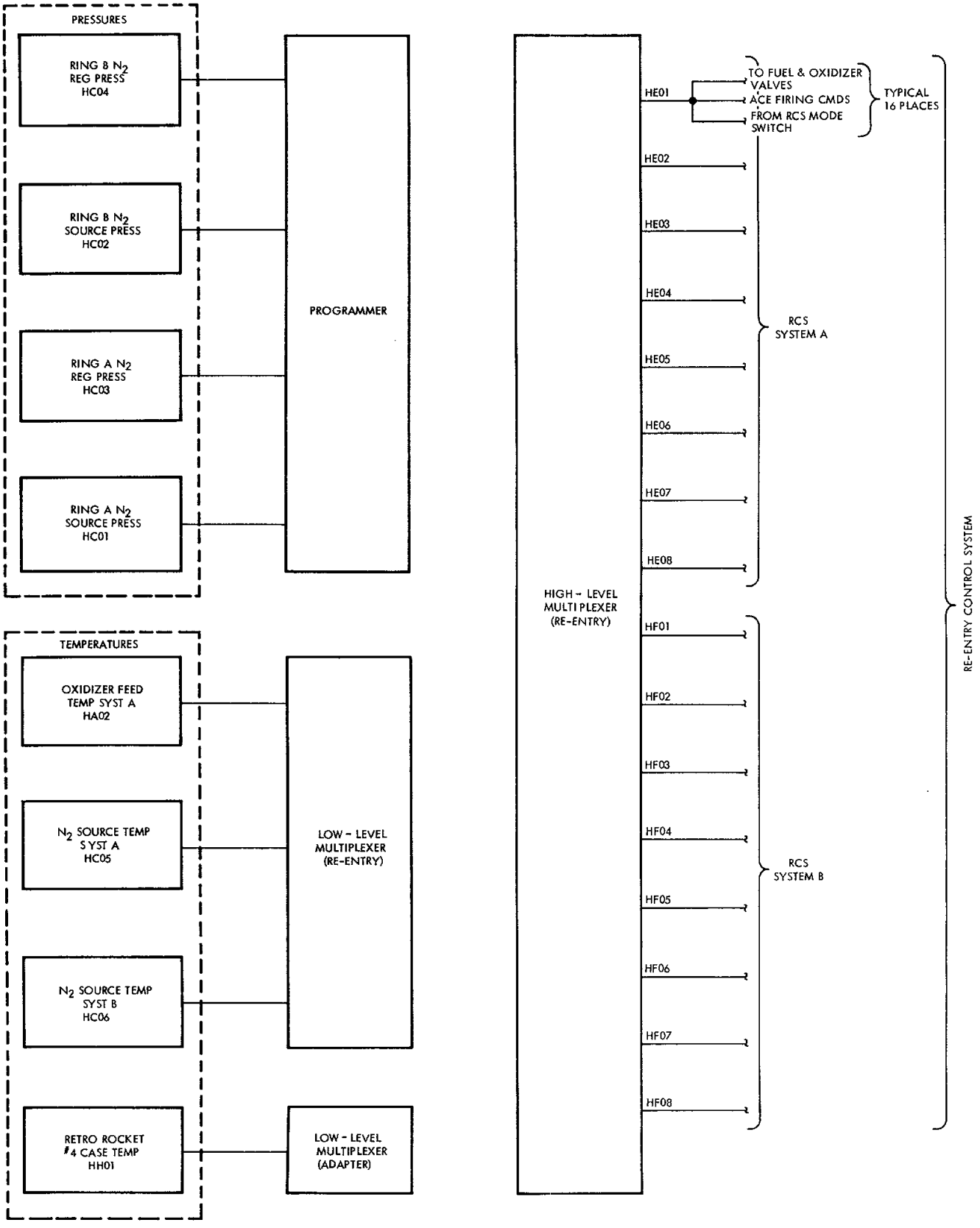


Figure 10-10 Re-Entry Control System Parameters Block Diagram

multiplexer. RCS system A thrusters, 1A thru 8A have been assigned parameters HE01 thru HE08 respectively and system B thrusters 1B thru 8B are designated by HF01 thru HF08 respectively.

The retrorocket case temperature (HH01) was monitored on spacecraft 5. The signal originated from a surface mounted temperature sensor located on retrorocket number 4 and was applied to the adapter low-level multiplexer.

AERODYNAMIC AND CREW CONTROL PARAMETERS

Aerodynamic and crew control parameters are monitored as shown in block diagram form in Figure 10-11.

Spacecraft longitudinal (KA01), lateral (KA02), and vertical (KA03) accelerations are monitored to provide ground station indications during the launch and re-entry phases of the mission. The accelerometer outputs are applied to the programmer.

Static pressure (KB02) is monitored by a potentiometer type absolute pressure transducer. Static pressure is obtained from four static pressure ports equally spaced around the forward part of the conical section and connected in parallel to the transducer. The transducer output is applied to the programmer.

Pitch (FA01), roll (FA02), and yaw (FA03) attitude control stick positions are monitored to indicate pilot manual control usage and to evaluate thruster operation. Signals originate from the attitude hand controller potentiometers and are applied to the programmer.

Two bi-level channels are reserved for events to be monitored as required by the experiments of each particular spacecraft mission. Electrical provisions for monitoring these parameters are provided at the right (FD01) and left

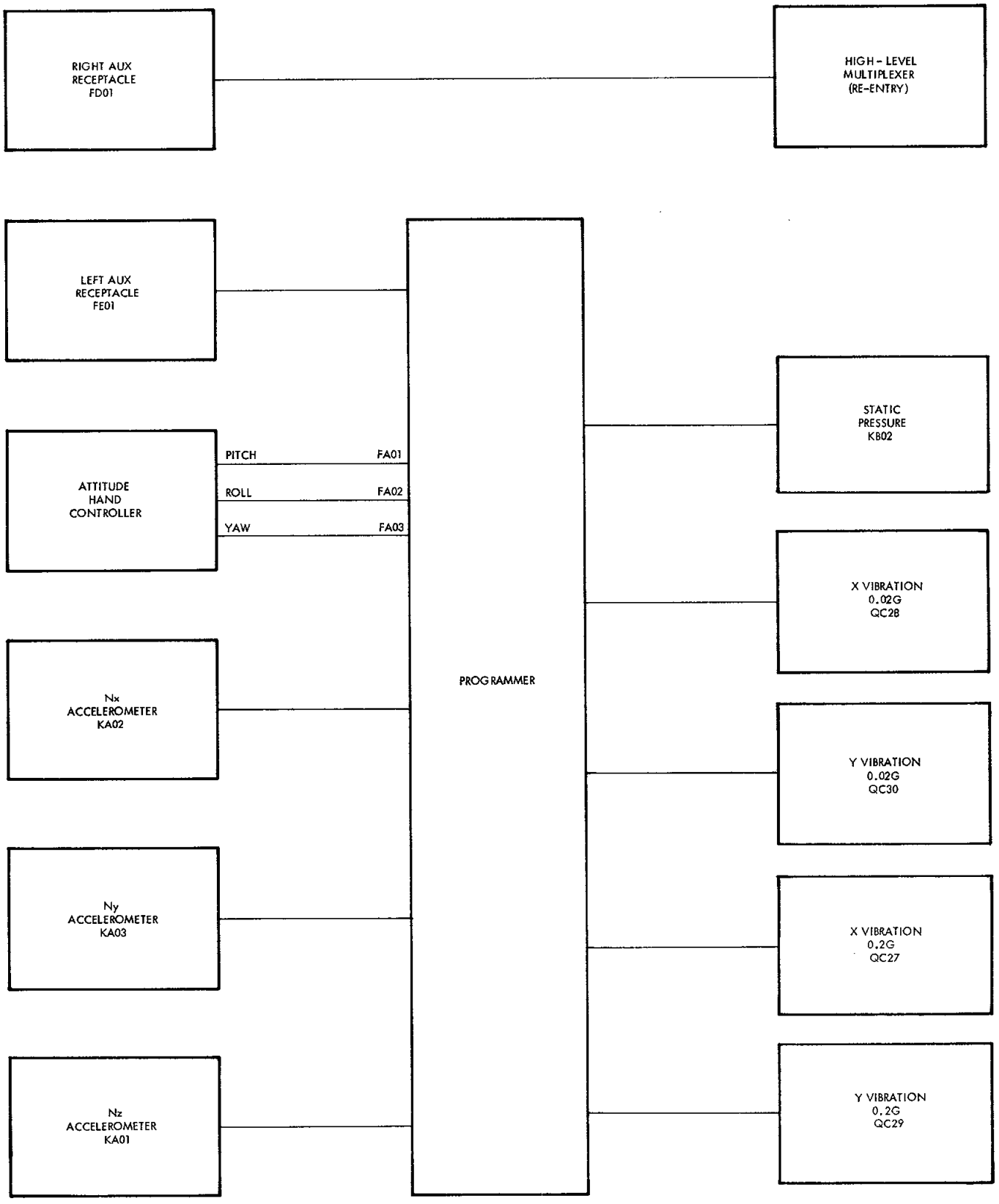


Figure 10-11 Aerodynamic and Crew Control Parameters Block Diagram

(FE01) utility receptacles.

COMMUNICATION SYSTEM PARAMETERS

The Instrumentation System monitors 11 Communication System parameters. These parameters are shown in block diagram form in Figure 10-12. A brief description of each of the parameters is presented in the paragraphs that follow.

To verify proper DCS performance and aid in malfunction isolation, the following DCS parameters are monitored: diplexer (LA04) and quadriplexer (LA03) receiver signal strength, package temperature (LA05), 6 vdc regulated power (LA02), 28 vdc regulated power (LA06), -18 vdc regulated power (LA07), 23 vdc regulated power (LA08), and -6 vdc regulated power (LA09). Parameters LA02, LA06, LA07, LA08, and LA09 are conditioned and then applied to the adapter high-level multiplexer except LA02 which is applied to the adapter low-level multiplexer. Parameters LA03, LA04, and LA05 are applied directly to the adapter low-level multiplexer.

Acquisition aid beacon (LD01) and adapter C-band beacon (LC09) case temperatures are also monitored to assure proper equipment performance. These temperature signals are applied to the adapter low-level multiplexer.

INSTRUMENTATION SYSTEM PARAMETERS

To insure proper operation of the Instrumentation System, various reference voltages and other pertinent data is telemetered to ground stations for analysis. The Instrumentation System parameters are shown in Figure 10-13 in block diagram form. A brief description of each parameter follows.

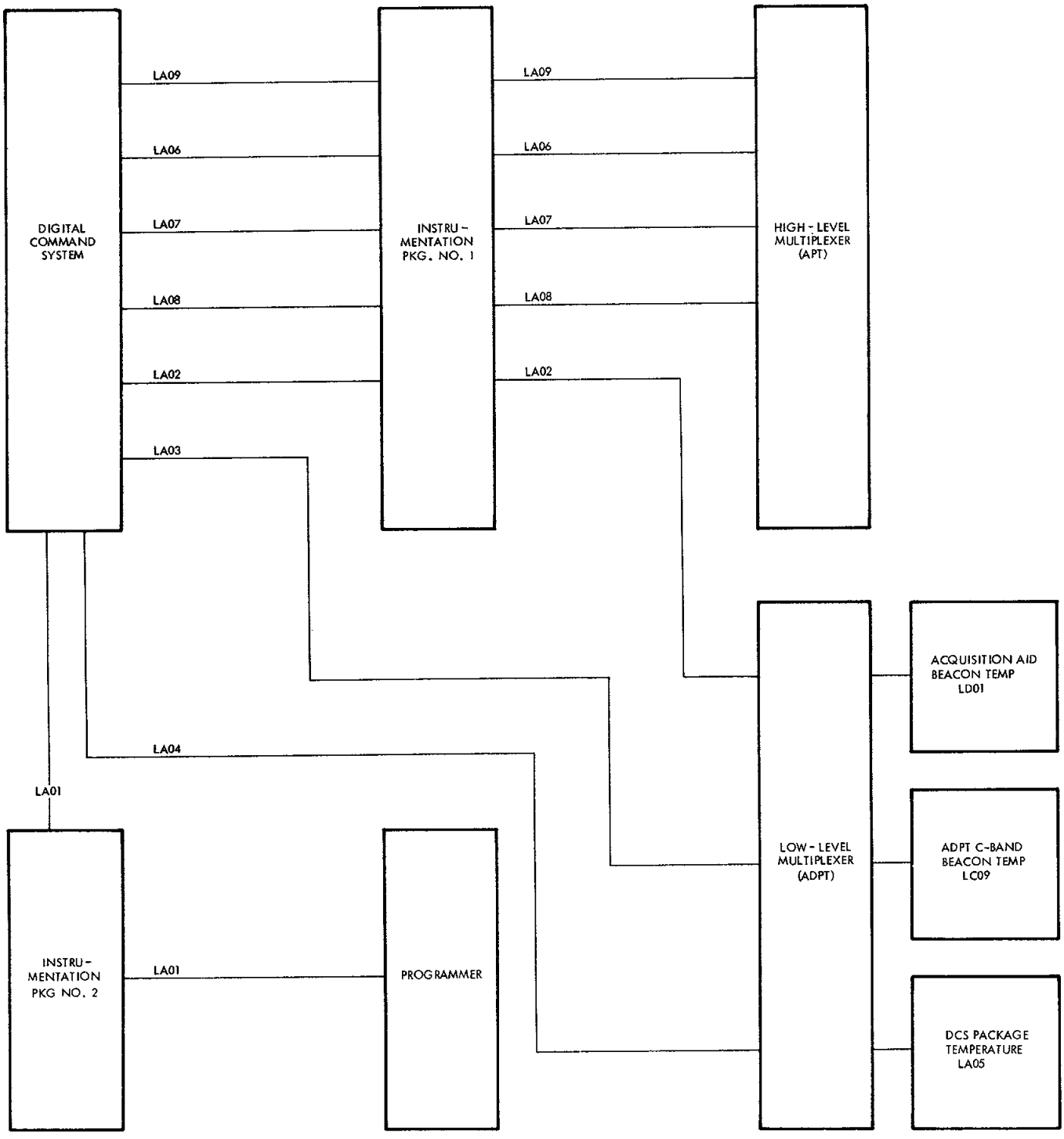


Figure 10-12 Communication System Parameters Block Diagram

High (MA17) and low (MA38) level zero reference voltages are monitored to insure that proper scaling is being employed by the Multiplexer/Encoding System. The low-level zero reference originates from the 5 vdc output of the dc-dc converter which is attenuated by a signal conditioner to 3 millivolts (the zero reference point) and is then applied to a channel in each of the multiplexers in the re-entry vehicle. This signal is also applied to the programmer as MA38.

High (MA37) and low (MA21) level full scale reference voltages, as the zero reference voltages, are required to insure that proper scaling is being employed by the Multiplexer/Encoding System. The 5 vdc output of the dc-dc converter is attenuated to 4.5 vdc and to 15 millivolts prior to application to channels of the high and low-level multiplexers in the re-entry vehicle, respectively. These parameters are required to provide a measurement of the reference voltage for potentiometer type transducers and resistive element temperature sensors.

Reference voltages for the high and low-level multiplexers located in the equipment adapter are provided by MB01 (full scale) MB02 (zero) and MB03 (full scale). These have the same characteristics as the reference parameters described above.

Parameter MA22 (calibrate signal) is provided to indicate that a calibration voltage is being applied, thus eliminating the confusion between a data and a calibrate signal. Parameter MA22 will exist whenever the CALIB switch in the spacecraft cabin is actuated or a calibration is commanded by the DCS.

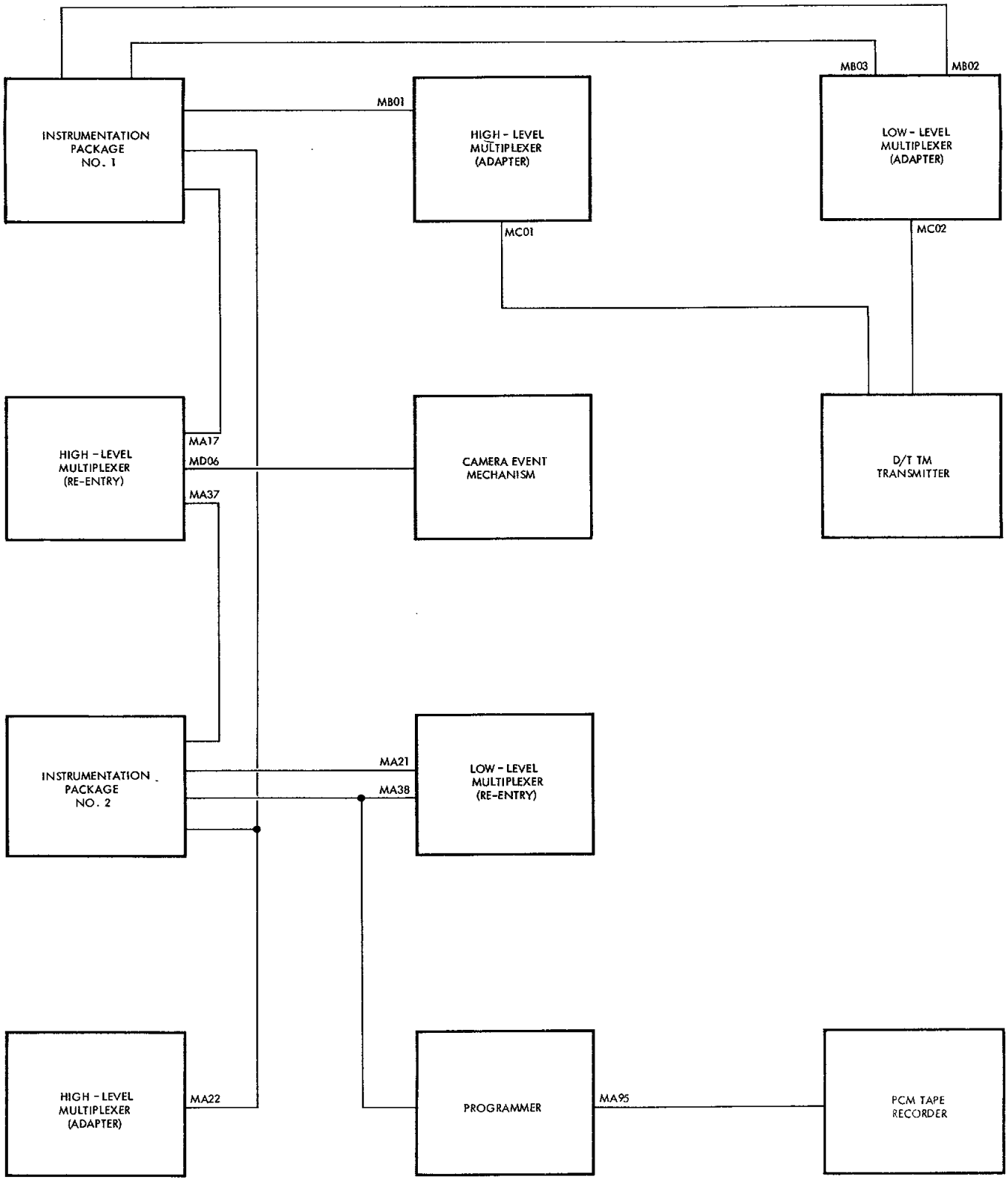


Figure 10-13 Instrumentation System Parameters Block Diagram

An indication of proper functioning of the PCM tape recorder is provided by monitoring tape motion (MA95). This is accomplished by providing a signal to a bi-level channel of the programmer when the recorder tape reels are in motion.

The rf power output (MC01) and the case temperature (MC02) of the delayed-time telemetry transmitter is monitored to provide an indication of transmitter operation. The transmitter physically located in the adapter is chosen for these measurements because it is subject to more extreme environmental temperature changes than the other two transmitters. Temperature signals are applied to the adapter low-level multiplexer, and the rf power output is applied to the adapter high-level multiplexer.

A camera event (MD06) is indicated to the ground station when the pilot initiates the camera event mechanism on the onboard camera. This signal is applied to a bi-level channel of the re-entry high-level multiplexer.

PHYSIOLOGICAL PARAMETERS

The physiological functions of the crew are monitored by sensors which are attached at various points to their skin. A block diagram showing the physiological parameters is shown in Figure 10-14. Signal conditioners, located in pockets of the underwear, condition the signals from the sensors to make them compatible with the recording and multiplexing equipment. All parameters except the oral temperature are recorded on bio-medical recorders. All signals, except oral temperature are applied to the programmer. Oral temperature is applied to the re-entry high-level multiplexer. The following command pilot parameters are monitored: electrocardiograms 1 and 2 (MA01, MA02), respiration rate and depth

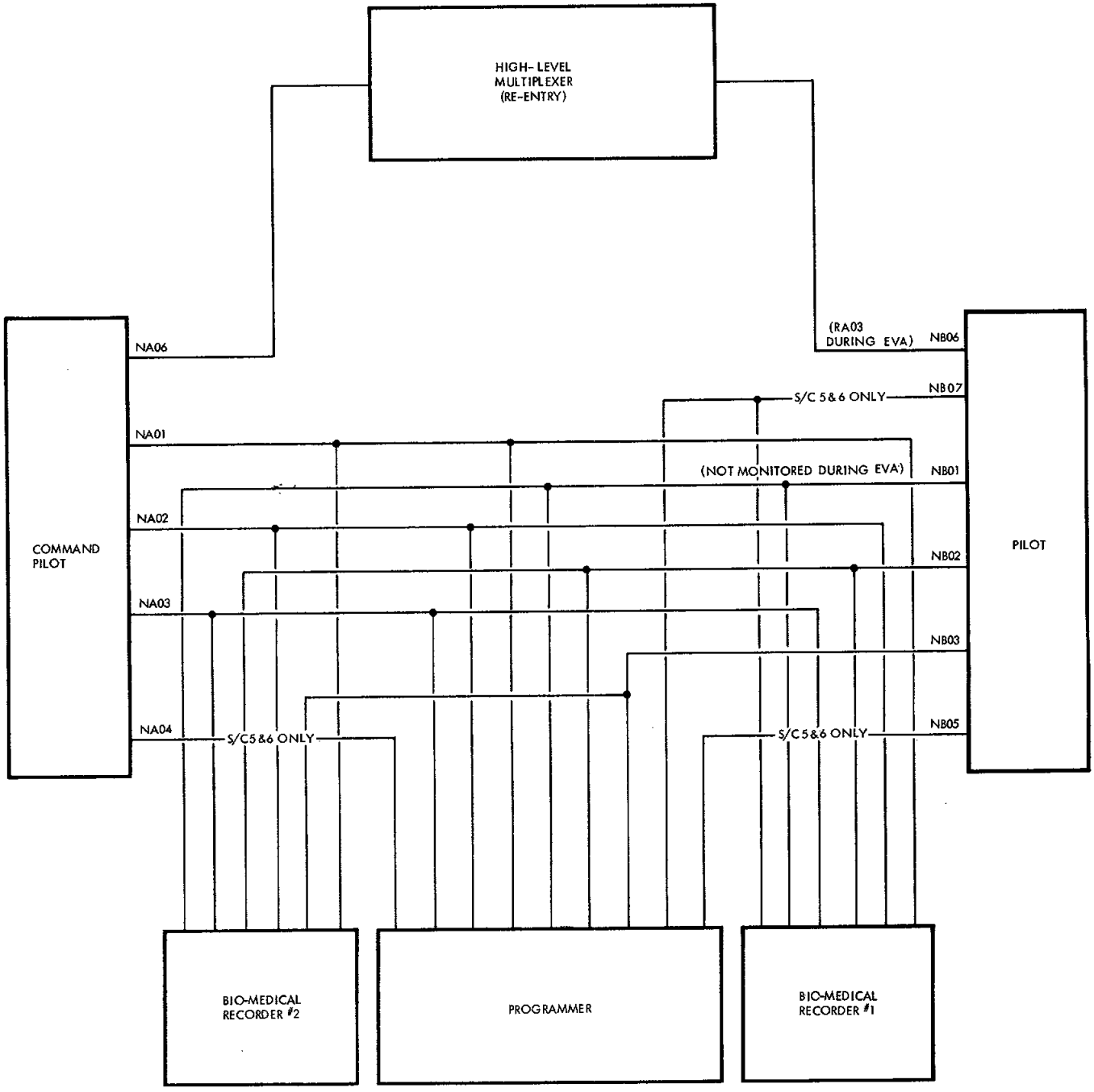


Figure 10-14 Physiological Parameters Block Diagram

(NA03), and oral temperature (NA06). The following pilot parameters are monitored: electrocardiograms 1 and 2 (NB01, NB02), respiration rate and depth (NB03), and oral temperature (NB06). The command pilot's blood pressure (NA04) and the pilot's blood pressure (NB05) and suit inlet temperature (NB07) was monitored on spacecraft 5 and 6 missions.

Electrocardiogram 1 (NB01) is not monitored during Extra-Vehicular Activities (EVA). To allow the ground station to evaluate the pilot's environmental condition while outside the spacecraft, suit pressure (RA03) is monitored in place of oral temperature (NB06). All other physiological parameters are monitored as previously described.

RENDEZVOUS RADAR PARAMETERS

To permit post-mission analysis of the Radar System operation during the rendezvous portion of the mission, 19 radar parameters are monitored. A functional block diagram of the radar parameters is shown in Figure 10-15.

The target verification (JC04) parameter provides telemetry information that the spacecraft radar has located the target vehicle. The radar LOCK-ON indicator on the pilot's pedestal will illuminate when this event occurs. Target range rate (JA04) information is relayed to the ground station via telemetry transmission and displayed to the pilots on the range rate indicator. During rendezvous, the range to target (JA13) parameter provides close-range data which is not available through the computer.

Radar inner skin (JB01), antenna faceplate (JB02), and transmitter tube (JB04) temperatures are sensed by resisitive-element temperature sensors. A output which is proportional to the temperature sensed is produced by integral bridges

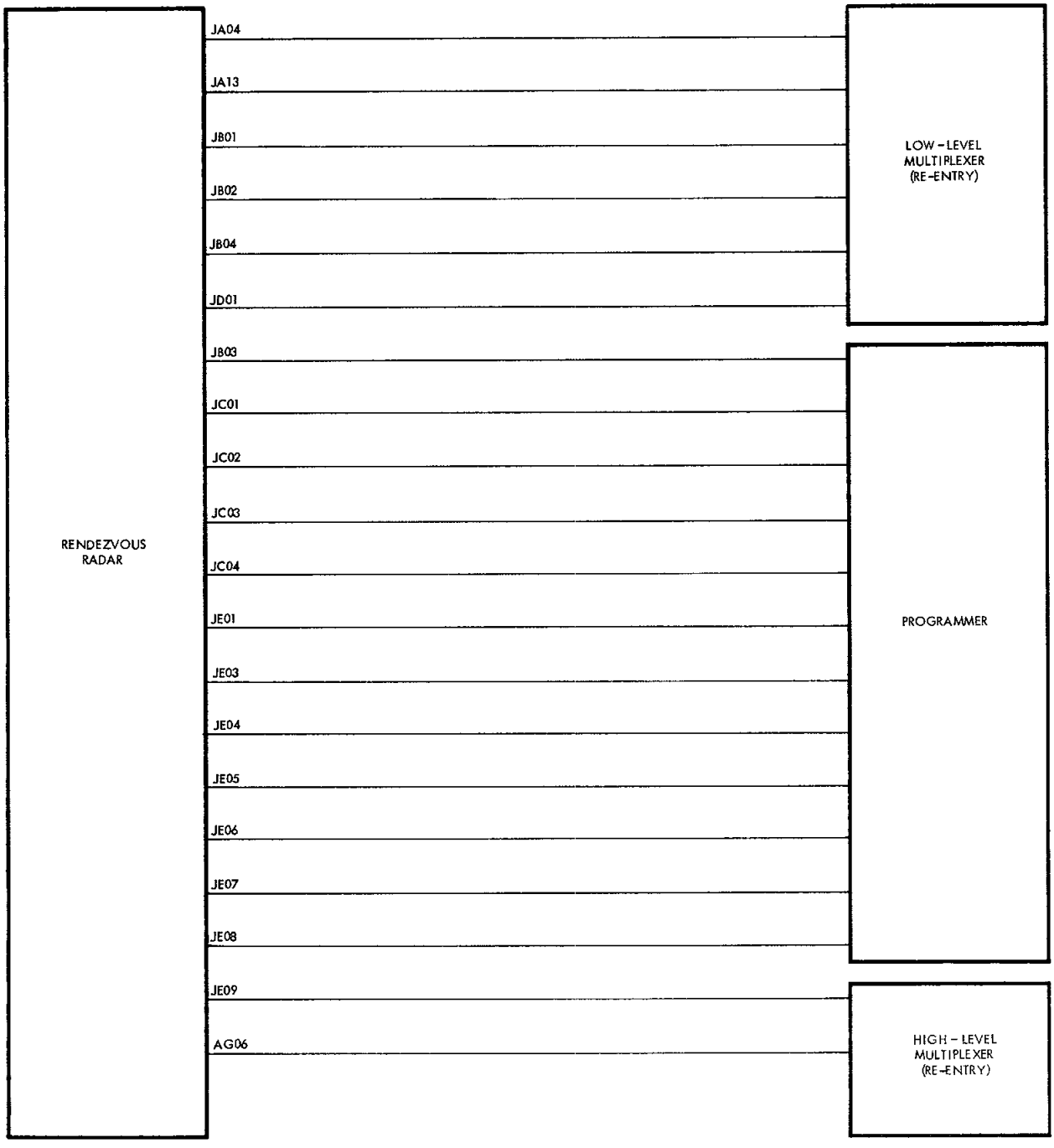


Figure 10-15 Rendezvous Radar Parameters Block Diagram

within the sensor. A pressure transducer within the radar package senses radar pressurization (JB03). This signal is used to determine the validity of the radar operation, which is dependent on the internal pressure being maintained.

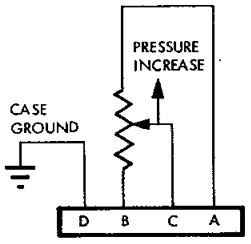
Oscillator crystal current (JC01) and radar regulated power (JE01 and JE03 through JE09) parameters provide information relative to the Radar System operation and monitor for specific functions which occur. Additional parameters monitored to evaluate the operation of the radar are: rf power (JC01), agc voltage (JC02), and narrow-band agc voltage (JC03).

SYSTEM UNITS

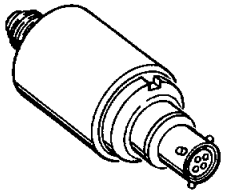
PRESSURE TRANSDUCERS

The purpose of the pressure transducer is to sense pressure, and to convert this pressure into a proportional electrical signal. There are six physically different configured pressure transducers as shown in Figure 10-16. Transducers have different physical appearances and different pressure ranges to accommodate the specific application of use. The numerical call outs below each transducer in Figure 10-16 identifies the location and application of the transducer as shown in Figure 10-1. The numbers correspond to those on Figure 10-1.

The sizes of the units vary from about 1 1/4 inches x 1 1/4 inches x 3 inches to 2 1/2 inches x 2 1/2 inches x 4 inches; the weights vary from approximately 5 ounce to 1 pound. The unit construction utilizes a bellows or Bourdon tube which varies the wiper position of a potentiometer, proportionally, with the input pressure. Two potentiometers are used in the dual-output units to separate the cabin

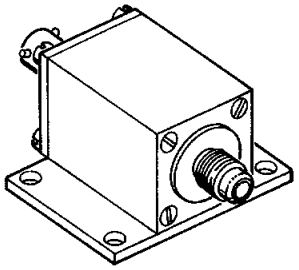


SINGLE POTENTIOMETER TRANSDUCER SCHEMATIC (TYPICAL)



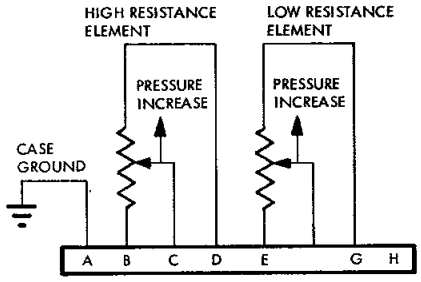
WATER PRESSURE TRANSDUCER

REF FIGURE 10-1 INDEX NO. 19 FOR LOCATION

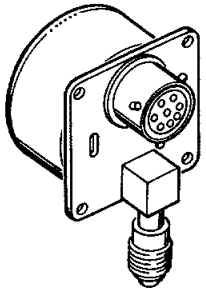


ABSOLUTE AND STATIC PRESSURE TRANSDUCER

REF FIGURE 10-1 INDEX NO. 39 & 40 FOR LOCATION

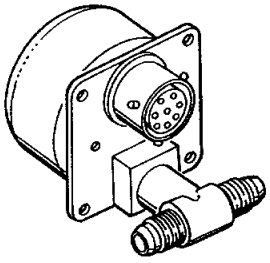


DUAL POTENTIOMETER TRANSDUCER SCHEMATIC (TYPICAL)



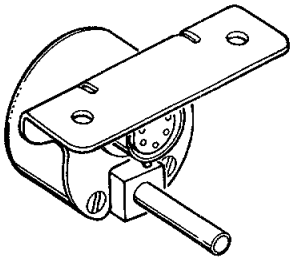
ECS SECONDARY SUPPLY PRESSURE TRANSDUCER

REF FIGURE 10-1 INDEX NO. 55 & 59 FOR LOCATION

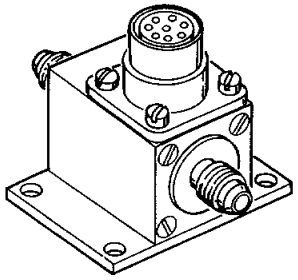


RSS AND PRIMARY ECS SUPPLY PRESSURE TRANSDUCER

REF FIGURE 10-1 INDEX NO. 14, 21 & 30 FOR LOCATION



OAMS PROPELLANT QUANTITY



CABIN AND SUIT PRESSURE TRANSDUCER

REF FIGURE 10-1 INDEX NO. 41, 57 & 58 FOR LOCATION

Figure 10-16 Pressure Transducers

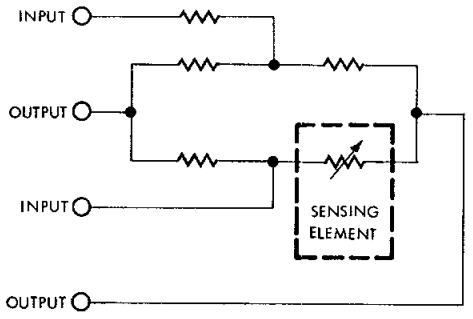
indicator circuit from the multiplexer/encoder (telemetry) circuit, thus avoiding a possible loading error in the latter. With one exception, pressure transducer outputs range from 0 to 5 vdc. The OAMS quantity system pressure-temperature sensor, driving a cabin indicator, has an output of 0 to 24 VDC.

TEMPERATURE SENSORS

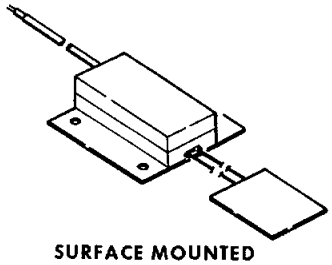
Temperature sensors are used to convert temperatures into directly proportional electrical signals. Basically there are two types of temperature sensors: a probe type and a surface mounted type. Variations exist within each type to accommodate specific mounting requirements. Nine physically different types of temperature sensors are shown in Figure 10-17. With respect to temperature range, approximately 20 different sensors are used. The numbers beneath each sensor in Figure 10-17 corresponds to the sensors locations and application as shown in Figure 10-1.

Spacecraft temperatures are monitored by platinum element temperature sensors. The sensors vary somewhat in size but are roughly 0.4 x 0.75 x 2.0 inches. There are two types of resistive-element sensors, a probe type and a surface-mounted type. Probes are used to monitor fluid temperatures, and surface-mounted sensors are used to monitor surface temperatures. Both types utilize a fully-annealed pure-platinum wire, encased in ceramic insulation. The sensors form one leg of a bridge network whose unbalance will produce an output of 0 to 20 mv dc or 0 to 400 mv dc. The 0-20 mv dc outputs are used for data transmission purposes and the 0-400 mv dc outputs for cabin displays.

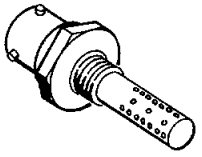
In some applications, mounting and space requirements necessitate that the bridge



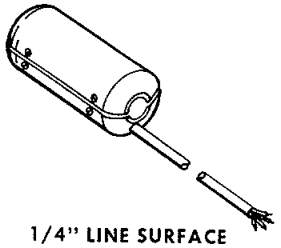
TYPICAL SCHEMATIC



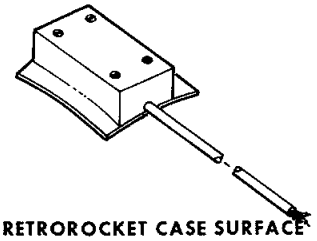
SURFACE MOUNTED SENSOR AND BRIDGE
REFER TO FIGURE 10-1, INDEX NO. 6, 27, 31 & 34 FOR LOCATION



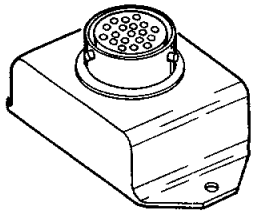
PROBE AIR TEMPERATURE SENSOR AND BRIDGE
REFER TO FIGURE 10-1 INDEX NO. 42, 54, & 60 FOR LOCATION



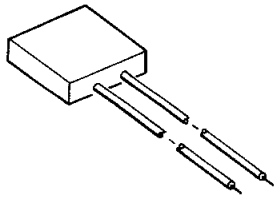
1/4" LINE SURFACE MOUNTED SENSOR AND BRIDGE
REFER TO FIGURE 10-1 INDEX NO. 2, 3, 11, 17, 20, 22, 35, 48, 50, 52 & 56 FOR LOCATION



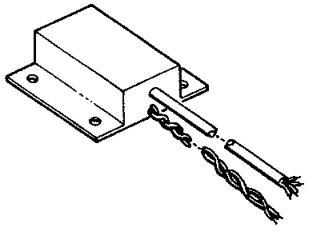
RETROROCKET CASE SURFACE MOUNTED SENSOR AND BRIDGE
REFER TO FIGURE 10-1 INDEX NO. 1, 4, 25, 32 & 33 FOR LOCATION



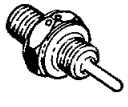
BRIDGE PACKAGE
REFER TO FIGURE 10-1 INDEX NO. 54 & 60 FOR LOCATION



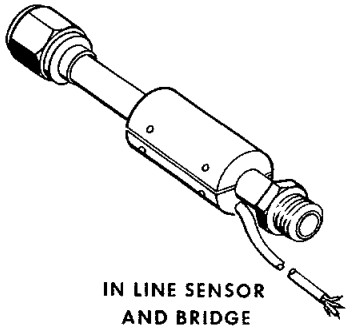
SURFACE MOUNTED SENSOR ELEMENT
OAMS QUANTITY



BRIDGE PACKAGE
REFER TO FIGURE 10-1 INDEX NO. 13, 23 & 29 FOR LOCATION



SUPPLY BOTTLE SENSING PROBE
REFER TO FIGURE 10-1 INDEX NO. 13, 23, 29 FOR LOCATION



IN LINE SENSOR AND BRIDGE
REFER TO FIGURE 10-1 INDEX NO. 10, 12, 15 & 18 FOR LOCATION

Figure 10-17 Temperature Sensors

is remotely located from the sensing element. In most cases, however, the bridge and sensing element are housed in the same case.

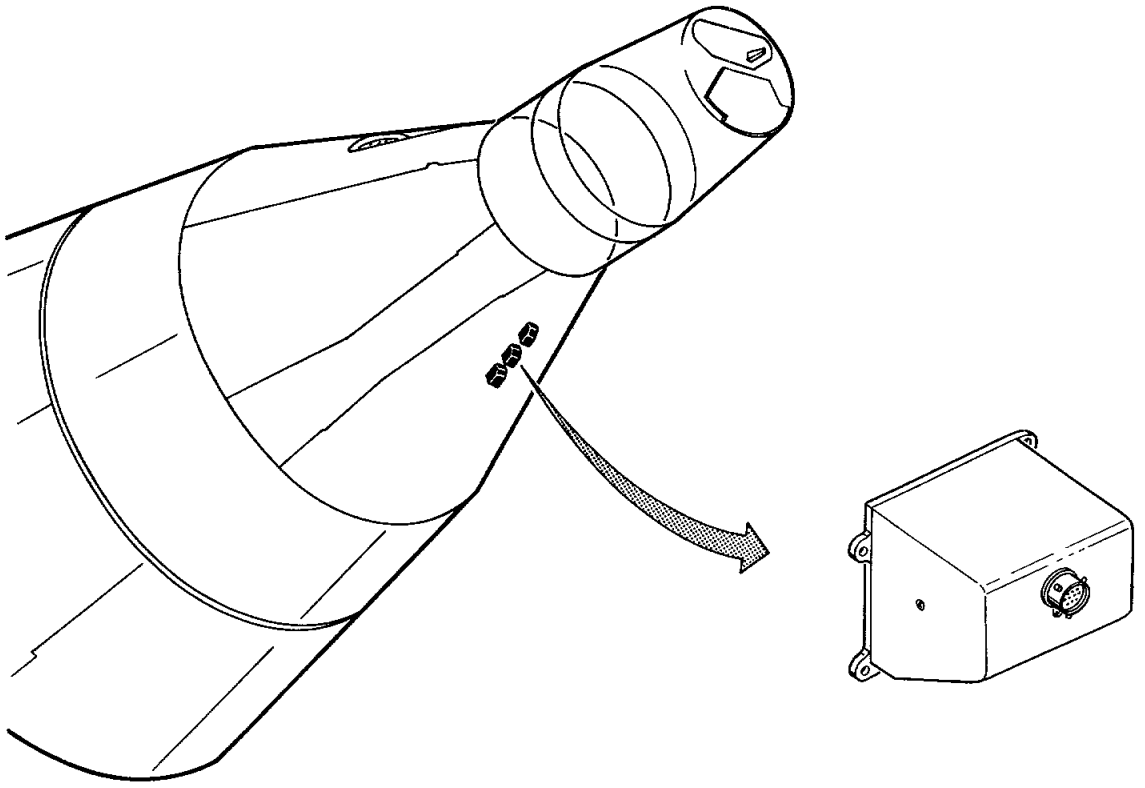
Regardless of how the bridge and sensing element are housed, combined, they comprise a schematic as shown in Figure 10-17.

SYNCHRO REPEATERS

Three synchro repeater assemblies, mounted in the upper portion of the aft landing gear will as shown in Figure 10-18, monitor the synchros on the IGS platform gimbals. Each synchro repeater output is a dc signal proportional to the spacecraft roll, yaw and pitch attitude in terms of platform coordinates. Two outputs are available per repeater; a course output, which provides 0-5 vdc output for 0-350 degrees of synchro travel and a fine output, which gives 0-5 output for every 35 degrees of synchro travel; only the coarse output is monitored as shown in Figure 10-18. A dead band of 10 degrees maximum exists, centered around the 135-degree position in the synchro repeater potentiometers. Control of the synchro repeaters is achieved by pilot actuation of the PLATFORM mode select switch.

CO_2 PARTIAL PRESSURE DETECTOR

A carbon dioxide partial pressure detector, as shown in Figure 10-19, is utilized to insure that there is a safe level of CO_2 in the pilots suit circuits. The detector is located in the ECS module. The gaseous mixture to be sampled is obtained as it exits from the ECS carbon dioxide and odor absorbers. The sample stream is divided through two separate passages, both filtering water vapor, but only one filtering carbon dioxide. The streams then pass into



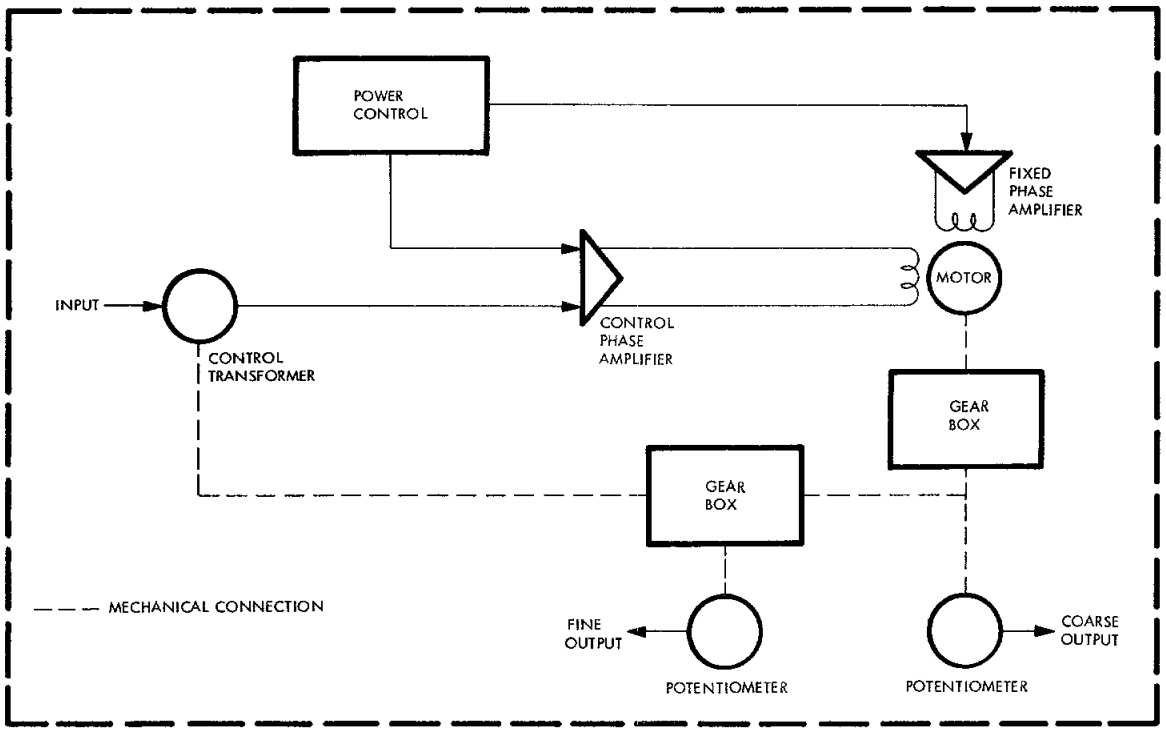
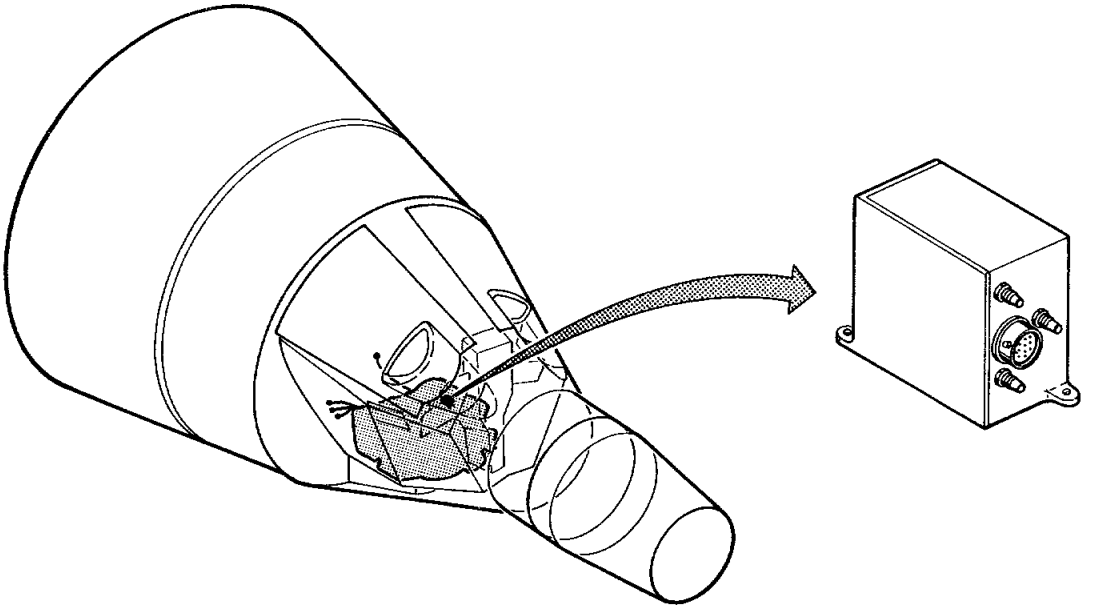


Figure 10-18 Synchro Repeaters and Schematic Diagram



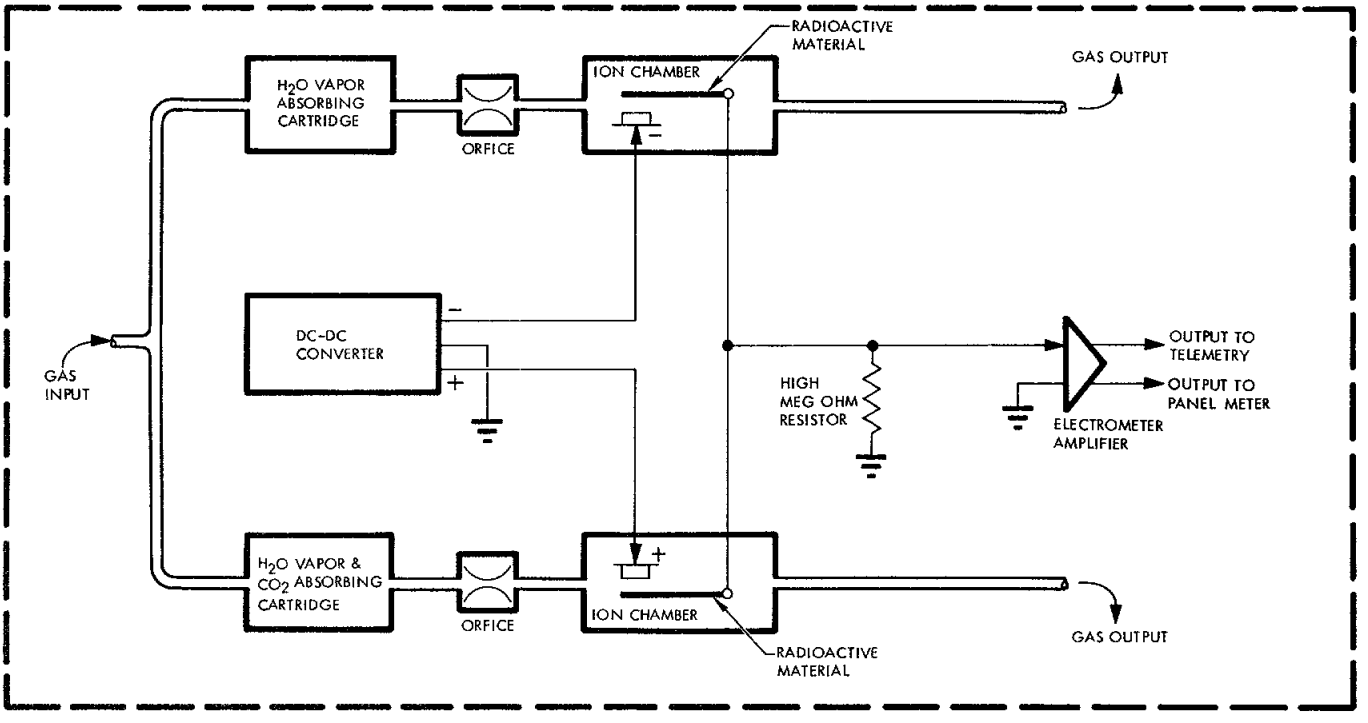


Figure 10-19 $C0_2$ Partial Pressure Detector and Schematic Diagram

identical ion chambers which are polarized with ± 50 vdc obtained from a dc-dc converter contained in the detector assembly; there a radioactive source ionizes the gases. The difference of the electrical outputs is amplified and provides a voltage which is proportional to the partial pressure of the mixture. The gas is then returned to the inlets of the suit compressors. The system provides two outputs: 0-5 vdc into a nominal 2.5 megohm load for telemetry use and 0-100 microamps into a 4000 ohm cabin indicator.

ACCELEROMETERS

Three linear accelerometers are provided to measure the accelerations along each of the spacecraft axes. The units are approximately 1.2 x 1.2 x 3 inches and are pictured in Figure 10-20. The accelerometers are electrically-damped, force-balance, servo-type units with outputs of 0-5 vdc. The unit which is used for longitudinal measurements has a range of -3 to +19 g and the other two have ranges of ± 3 g. The accelerometer is a torque-balanced, closed-loop system with a pendulous mass supported by an extremely low friction jewel bearing. The schematic of the accelerometer is shown in Figure 10-20. An electromagnetic position detector notes the slightest movement of the mass and supplies a directly proportional electrical signal to a servo amplifier. The output of the servo amplifier is applied to a torque generator which tends to restore the mass to its equilibrium position. The output of the accelerometer is obtained by sensing the voltage drop across the resistor in the system loop.

INSTRUMENTATION PACKAGES

A number of the signals in the various spacecraft systems are not compatible

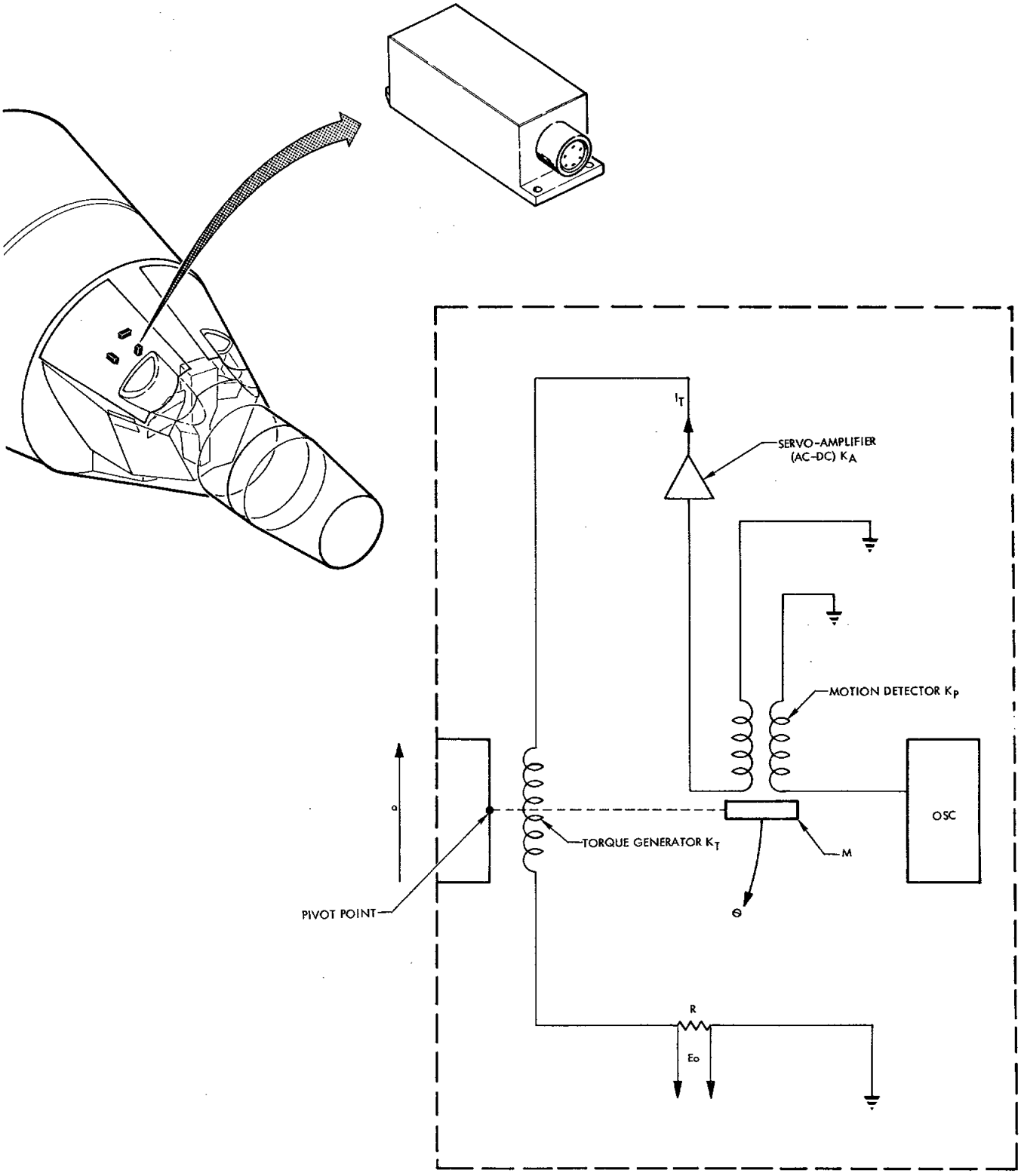


Figure 10-20 Servo Accelerometer and Schematic Diagram

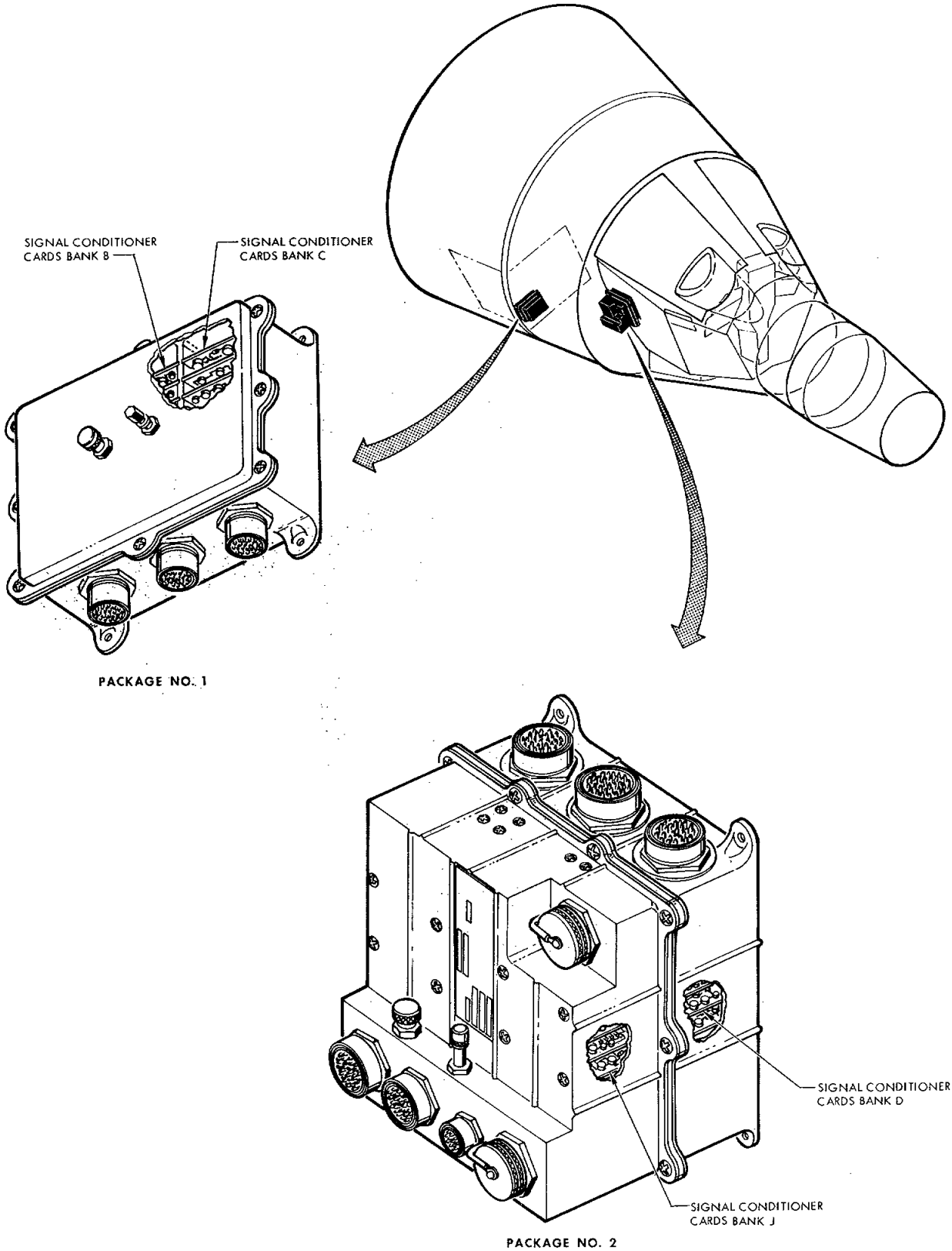


Figure 10-21 Instrumentation Package Assemblies

with the instrumentation circuitry, and therefore, must be conditioned for their use. Two signal-conditioning packages (instrumentation assemblies) are provided for this purpose. Instrumentation assembly number 1 is approximately 8 x 10 x 3 inches and is located in the adapter section. Instrumentation assembly number 2 is approximately 10 x 10 x 8 inches and is located in the upper right hand equipment bay of the re-entry section. Both units utilize sealed containers with an operating pressure of 4.5 psid and are shown in Figure 10-21. The assemblies employ a modular construction with plug-in modules that may be replaced, individually. A module consists of one or two standard printed circuit boards with the necessary component parts and a connector for attachment to a mother board within the package. There are 18 modules spaces in assembly number 1 and 51 in assembly number 2. Some modules provide for one data channel and others for two. There are six basic types of modules, and several of these have additional variations for different signal handling capabilities.

There are six variations of the Phase Sensitive Demodulators (PSD). Basically, the PSD accepts two input voltages: one signal voltage and one reference. It provides a dc output of five volts for a full scale input signal that is in phase with the reference and an output of zero volts for a full scale signal that is out of phase with the reference. The various configurations of this unit provide different full-scale sensitivities including special calibration curves for rate gyros.

The twelve types of dc voltage monitors are designed to accept various positive and negative dc voltage inputs and provide outputs of 0 to 5 vdc.

The ac voltage monitor accepts a signal ranging from 23 to 29 volts rms over a frequency range of 380 to 420 cycles. The output is from 0-5 vdc, varying only with the input voltage.

There are nine types of attenuator modules. These modules have various dc inputs which are changed to signals in the 0-20 mv dc range or the 0-5 vdc range. Some attenuator modules contain two data channels.

The dc millivolt monitor accepts an input of 0 to 50 mv dc and provides a proportional output of 0 to 20 mv dc.

The ac frequency sensor provides a 0 to 5 vdc output proportional to an input frequency varying from 380 cps to 420 cps. The voltage level of the input is 26 volts rms and does not affect the output of the module.

MULTIPLEXER/ENCODER SYSTEM

The multiplexer/encoder is divided into five packages to allow the data signals to be sampled near their sources. Its purpose is to accept signals from the signal pickups or signal conditioners, combine and convert these signals to a serial binary-coded digital signal, and supply this signal to the tape recorder and to the real-time telemetry transmitter. The units are shown in their respective locations in Figures 10-22 and 10-23.

The multiplexer/encoder consists of a programmer, two identical high-level multiplexers, and two identical low-level multiplexers. These five units are discussed as a single unit because of their close function relationship.

The low-level multiplexers each weigh 2.6 pounds and measure approximately 5.75 x 5.25 x 2.75 inches. The high-level multiplexers are 5.25 x 4.75 x 2.5 inches and weigh approximately 3.25 pounds each. The multiplexers might be considered as an

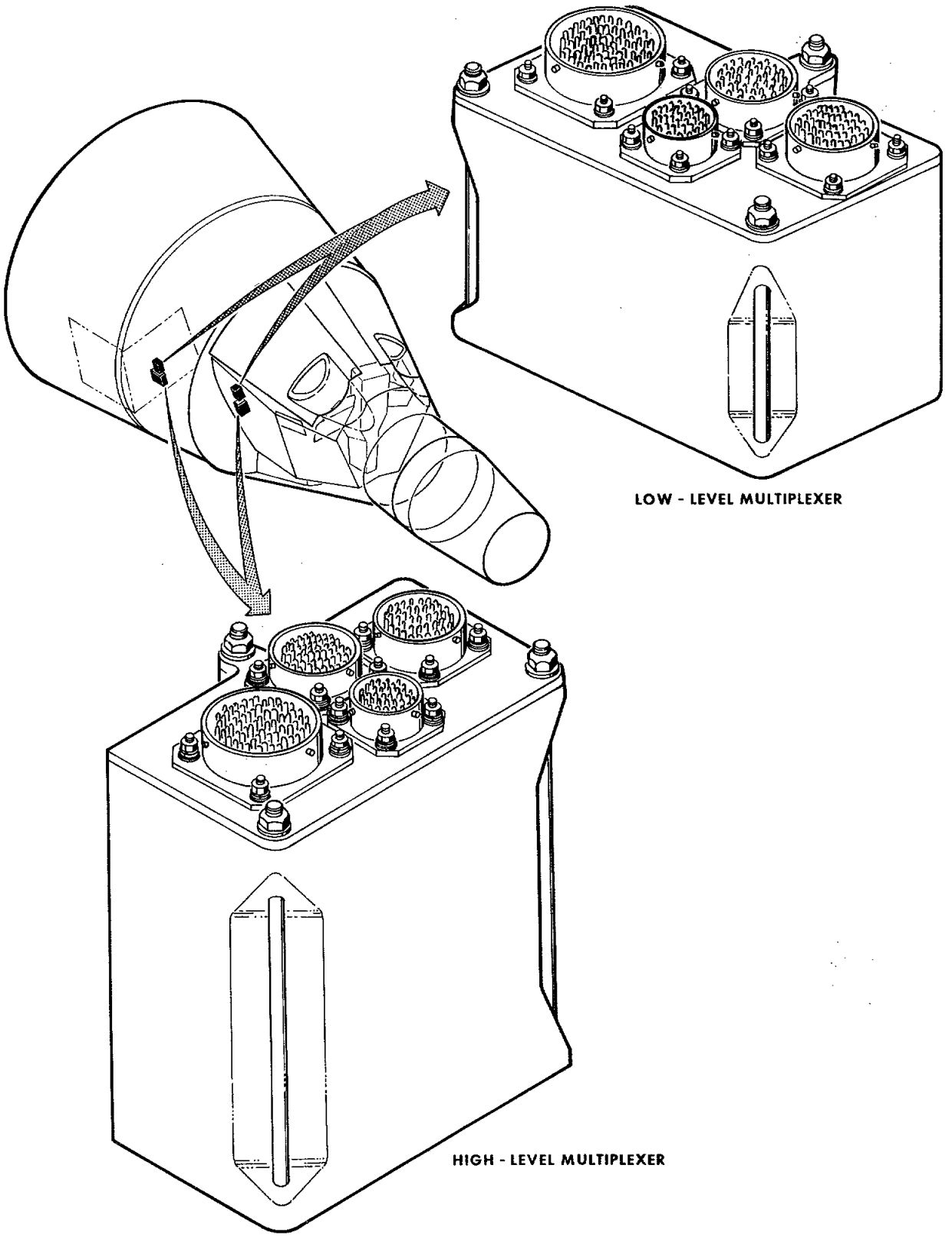
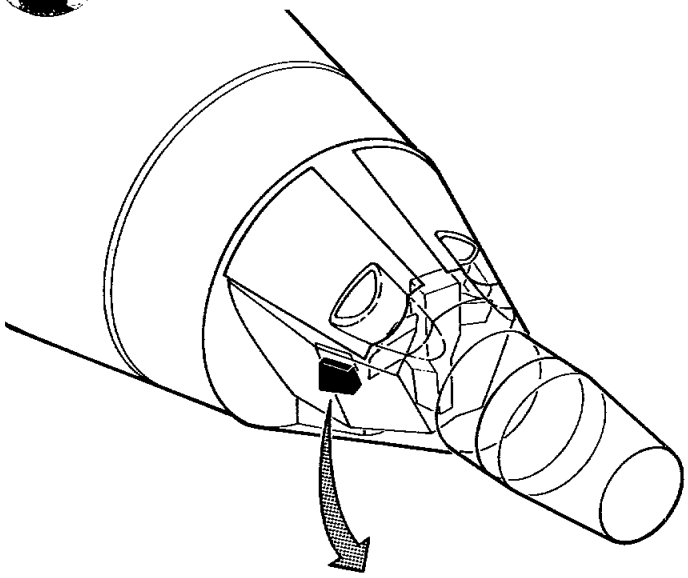


Figure 10-22 Instrumentation System Multiplexers



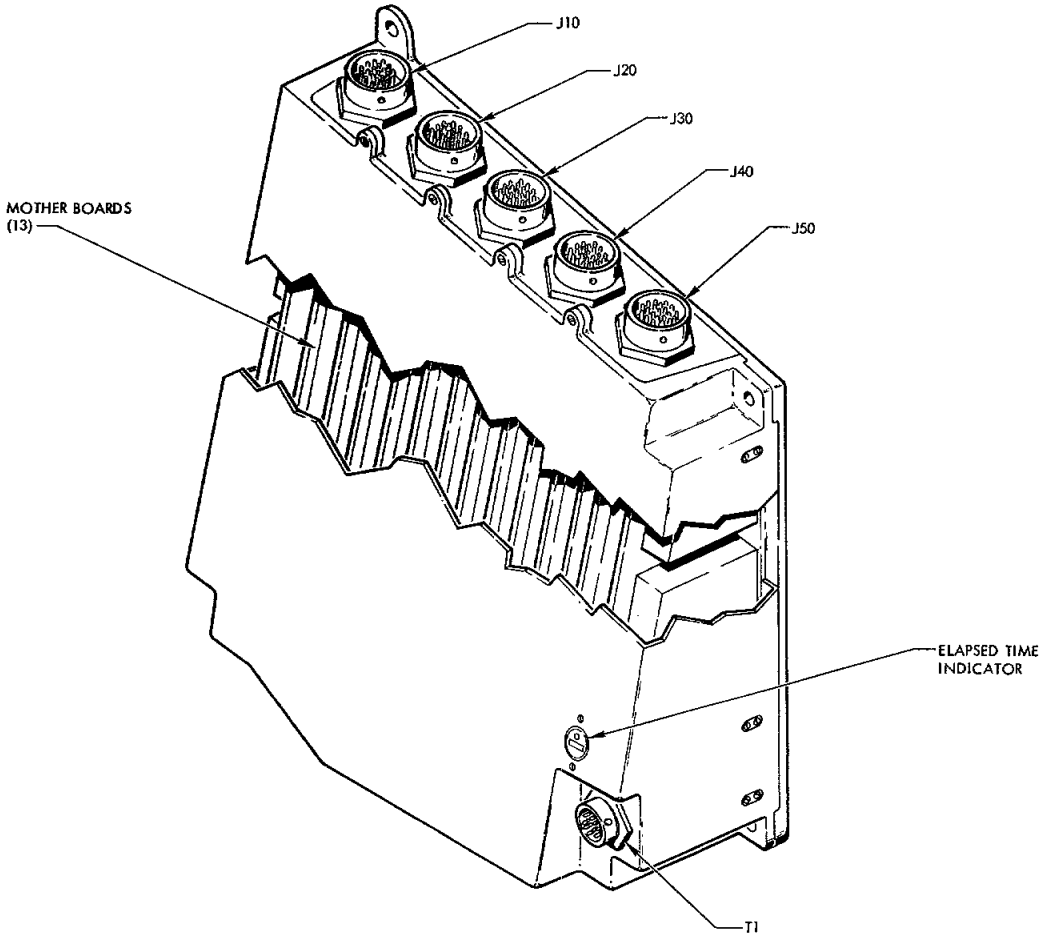


Figure 10-23 Instrumentation System Programmer

expansion to the data handling capability of the programmer and are dependent on the programmer for encoding and timing. The sampling rates of the multiplexer inputs are established by the timing chain from the programmer. The low-level multiplexers each sample 32 low-level (0-20 millivolts dc) signals: 24 at 0.416 samples per second and 8 at 1.25 samples per second. In the high-level multiplexer, 32 high-level (0-5 vdc) are samples at 1.25 samples per second, and the 24 bi-level and 16 bi-level-pulsed signals are sampled in sets of eight at a sample rate of ten per second. For the bi-level signals, a binary one (nominally 28 vdc, but at least 15 vdc) may indicate that an event or function has or has not taken place. For example, the indication that the bio-med tape recorders are on is a one but the indication that the computer is on is a binary zero (nominally zero vdc, but less than 5 vdc). For bi-level-pulsed signals, 15 vdc or more represents a binary zero, while 5 vdc or less for at least ten milliseconds is a binary one. The pulse conditioning circuitry in the multiplexer senses these pulses and holds the voltage level until it is sampled by the programmer.

The programmer weighs approximately 20 pounds and measures approximately 11 x 11 x 4.5 inches. The programmer may operate as a self-contained data handling unit.

The basic functions of the programmer are: data multiplexing; timing, to support the multiplexing functions; and analog to digital conversion. The programmer output is a PCM pulsetrain to the tape recorder and the real-time telemetry transmitter.

The basic components of the programmer are a high-level analog subcommutator, prime subcommutator, master commutator, analog to digital converter, output shift register, digital input circuitry and input selector, special timing, output filter, and tape recorder input converter (Figure 10-24).

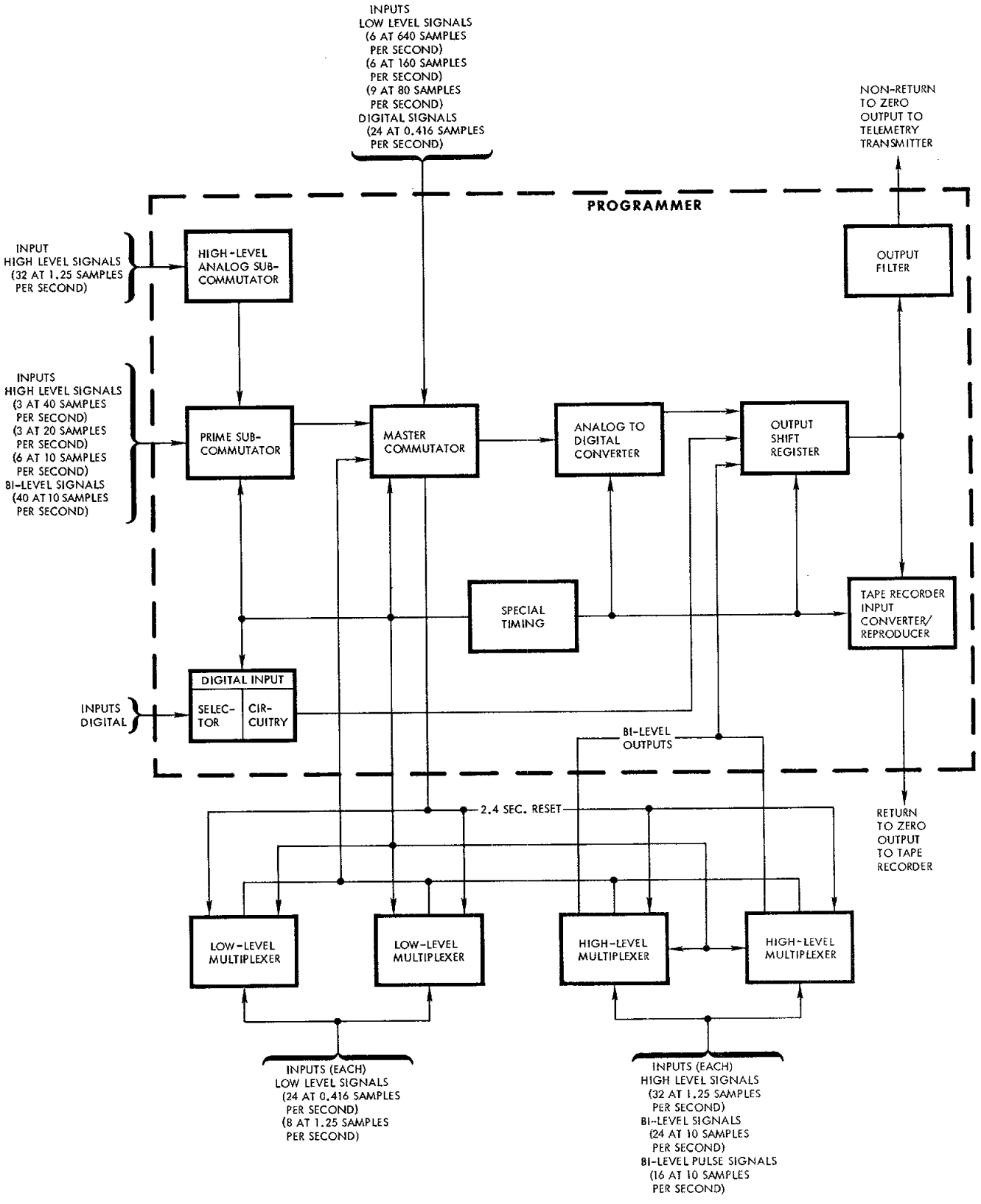


Figure 10-24 Multiplexer/Encoder System Block Diagram

The programmer high-level analog subcommutator segment has the capability of sampling 32 high-level signal inputs at 1.25 samples per second. The analog subcommutator receives its inputs directly from the signal sources, or from the signal conditioners. The output of the analog subcommutator is applied to the prime subcommutator for further multiplexing.

The prime subcommutator, in addition to accepting the sampled high-level subcommutator data output, has the capability of sampling 40 bi-level signals at 10 samples per second, 3 high-level signals at 40 samples per second, 3 high-level signals at 20 samples per second and 6 high-level signals at 10 sampler per second. The prime subcommutator supplies its output to the master commutator.

The inputs to the prime subcommutator, low-level outputs from the two low-level multiplexers, and high-level outputs from the two high-level multiplexers are combined into a prime subframe which is applied to the master commutator. In addition, the master commutator has the capability of sampling 6 low-level signals at 640 samples at 80 samples per second, and 24 digital signals at 0.416 samples per second.

The output of the master commutator is applied to the analog to digital converter where the analog output from the master commutator is converted to a digital presentation.

The digital data from the output of the analog to digital converter, digital input circuitry, and the two high-level multiplexers (bi-level signals) is combined in the output shift register into a continuous Non-Return-to-Zero (NRZ) Pulse Code

Modulated (PCM) wavetrain of 51.2 kilobits per second.

The output of the output shift register is applied to the real-time telemetry transmitter, and to the tape recorder input converter/conditioner.

The tape recorder input converter/conditioner selects data from the 51.2 kilobits per second NRZ output of the output shift register, converts the data to an Return-to-Zero (RZ) output and applies the data to the tape recorder input conditioner module at 5.12 kilobits per second.

The serial outputs, provided from the programmer, all have positive voltages for ones and zero or negative voltages for zeros. The output for the tape recorder is a 5.12 kilobit per second serial Return-to-Zero (RZ) signal with a +5 volt transition for data ones and a -5 volt transition for data zeros. A clock signal at 5.12 kilobits per second is also provided for the tape recorder. This output is a pulse train of 50 percent duty cycle at a peak amplitude of 5 volts. The timing of the positive excursion is coincident with data one pulses. The programmer output for the real-time transmitter is a 51.2 kilobit NRZ signal with a voltage which is adjustable between 0.1 volt and 1.0 volt peak. Separate hardline outputs are provided to allow various test equipment to be used without degradation of the transmitter or tape recorder outputs. The hardline outputs are real-time PCM signal, basic PCM clock rate signal, and master reset pulse signal. The signals are two volts peak-to-peak and are fed over twinex coaxial or video cables.

The programmer message format includes a master frame and prime subframe; the complete format is transmitted as real-time and only the prime subframe is

tape recorded. The master frame consists of 160 words, each word consisting of eight data bits, sampled 40 times per second. Ninety-six master frames are required to completely cycle through all data inputs. Every tenth word in the master frame contains prime subframe data. The prime subframe consists of 64 words sampled ten times per second. Twenty-four prime subframes are required to cycle through all data inputs of this part of the system. Information bits are obtained from analog data, arranged with the most significant bit first, digital data, broken into groups of eight bits with the most significant bit first, or bi-level data grouped as eight consecutive data bits (referred to as a bi-level set).

TRANSMITTERS

Three telemetry transmitters are used to transmit the Instrumentation System data to the ground stations. Although the transmitters serve the Instrumentation System, its antennas, and associated switching is part of the Communication System; therefore, the transmitters are described in detail in Section IX, Communication System.

PCM TAPE RECORDER

The tape recorder is designed for monitoring and for producing a recording of the signals received from the PCM programmer. The tape recorder records PCM data at a tape speed of 1 7/8 inches per second and playback, on command, of this recorded data, will, on command, stop, reverse tape direction and playback the recorded data at a tape speed of 41.25 inches per second. Erasure of data will occur only during record mode. The power control circuitry is described in detail in Section IX, Communication System.

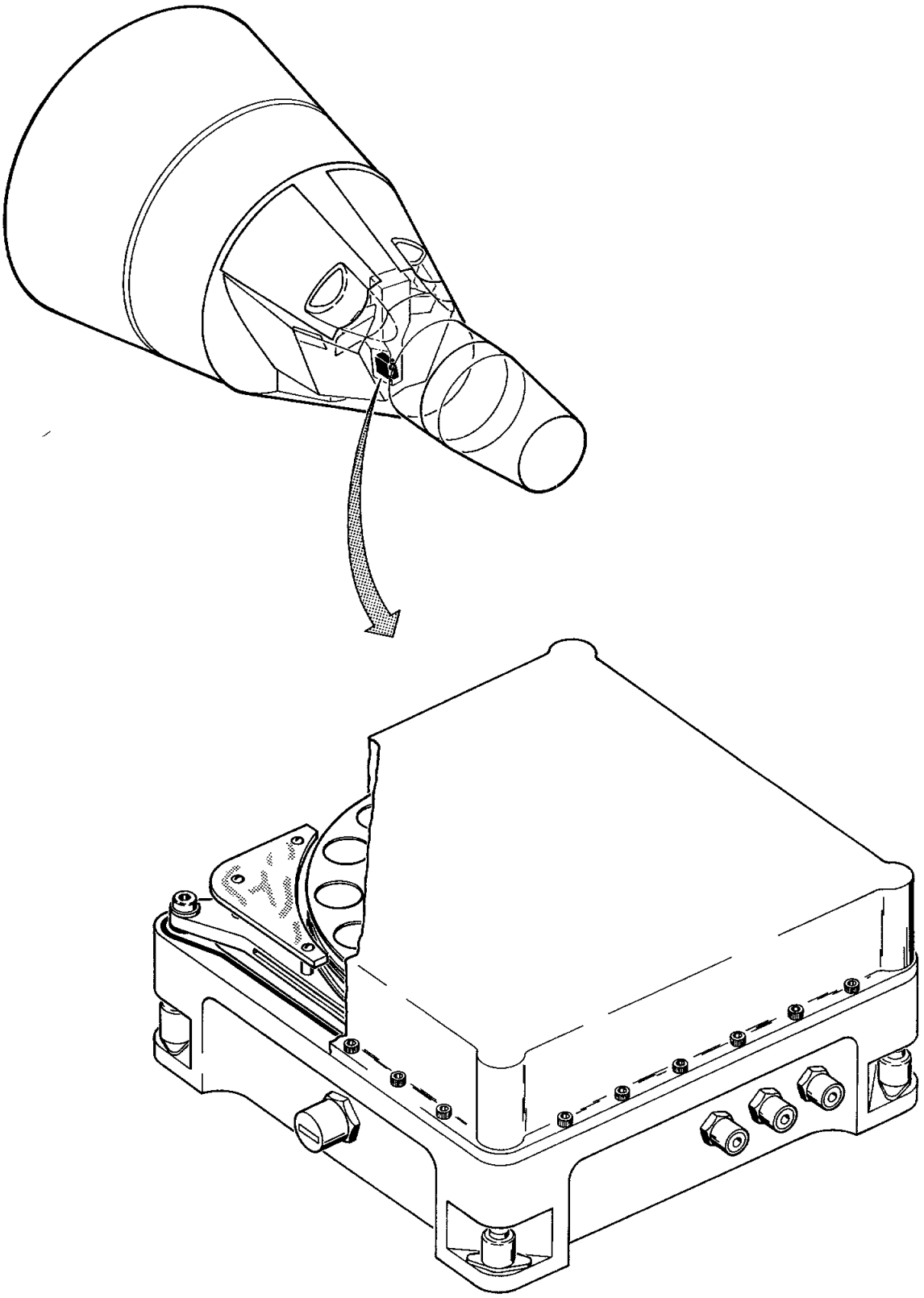


Figure 10-25 PCM Tape Recorder

Telemetered signals recorded are RZ. The PCM tape recorder reproducer shown in Figure 10-24 consists of one completely enclosed tape recorder which is approximately 4.3 inches high, 10.0 inches wide and 10.0 inches deep. The tape recorder consists of the cover assembly, capstan drive assembly and tape transport assembly. Connectors on the side of the case assembly provide signal connections, power connections and test connections.

Record

The magnetic tape recorder is capable of providing a minimum of four hours of recording time at a tape speed of 1 7/8 inches per second. Two tracks of simultaneous PCM data can be recorded at 1 7/8 inches per second. Four hours of RZ data at 5120 bits per second can be recorded at 1 7/8 inches per second.

Playback

On command, the recorder is to rewind the tape into the supply reel at 22 times the record speed (41.25 inches per second) while reading and playing back the information recorded on the tape. Final output of recorded data is in NRZ form.

Diphase System

The diphase signal processing technique permits the maximum tape utilization efficiency, while avoiding certain serious problems encountered with use of conventional NRZ recording at high packing density. It involves the encoding of the digital information prior to recording and decoding of the playback and conversion of the reproduced signal into standard NRZ form.

The diphase technique is essentially a phase-modulated carrier process. The digital data format to be recorded in RZ with an accompanying clock, and the desired output in the reproduce mode is of the standard NRZ form.

The diphase signal to be recorded is created in the following manner. Inverted RZ data and clock signal are OR gated into a binary flip-flop such that a transition of the flip-flop occurs on every negative going edge. A logical zero is represented in the diphase code by a square wave at 1/2 the data rate. Each time a logical one is received, a phase transition occurs in the center of the bit cell so that a logic one is represented by a square wave at the data rate. The output of the flip-flop is the diphase signal. This signal is then fed to the record amplifier which drives the diphase signal into the record head.

Record Mode

During the record mode, the input signal is sent to a preamplifier, encoder and amplifier. A clock signal is applied to the input of the triggerable flip-flop. The diphase code produced is recorded on magnetic tape.

The magnetic tape is dc erased prior to recording. The magnetic head utilized is a high-quality instrumentation recording head has a gap width approximately 1/3 the recorded wave length. The gap width is not critical, but if it is much wider than 1/3 of the recorded wave length, the high frequency playback components are attenuated and if it is much narrower, the high frequency components are accentuated, causing a difficult equalization problem.

Reproduce Mode

During the reproduce mode, the signal is picked up by the magnetic head and applied to the playback amplifier, where it is amplified approximately 60 db, filtered, and equalized to compensate for the effects of the head to tape system. The equalized signal is then fed to an input coupler where approximately 40 db of hard limiting is provided, thus providing extremely high immunity from amplitude variation in the reproduced signal.

The ability of this system to operate satisfactorily through such a large variation in playback signal amplitude assures a high degree of reliability and extremely low data drop out. The outputs of the input coupler, the recorded diphase signal and its complement, are fed to the one shot timing extractor circuitry and simultaneously to the decoder circuitry.

The function of the timing extractor and decoder is to produce timing pulses from the amplified and limited diphase playback signal. This circuitry detects the data, using the timing pulses and diphase signal, and produces the final NRZ output.

The output filter is fed the decoder output and filters out some of the higher harmonics of the NRZ output signal. The hardline output amplifiers produce a hardline output with good square wave characteristics at high frequencies.

DC-DC CONVERTERS

The two dc-dc converters (one of which is a standby unit) supply the Instru-

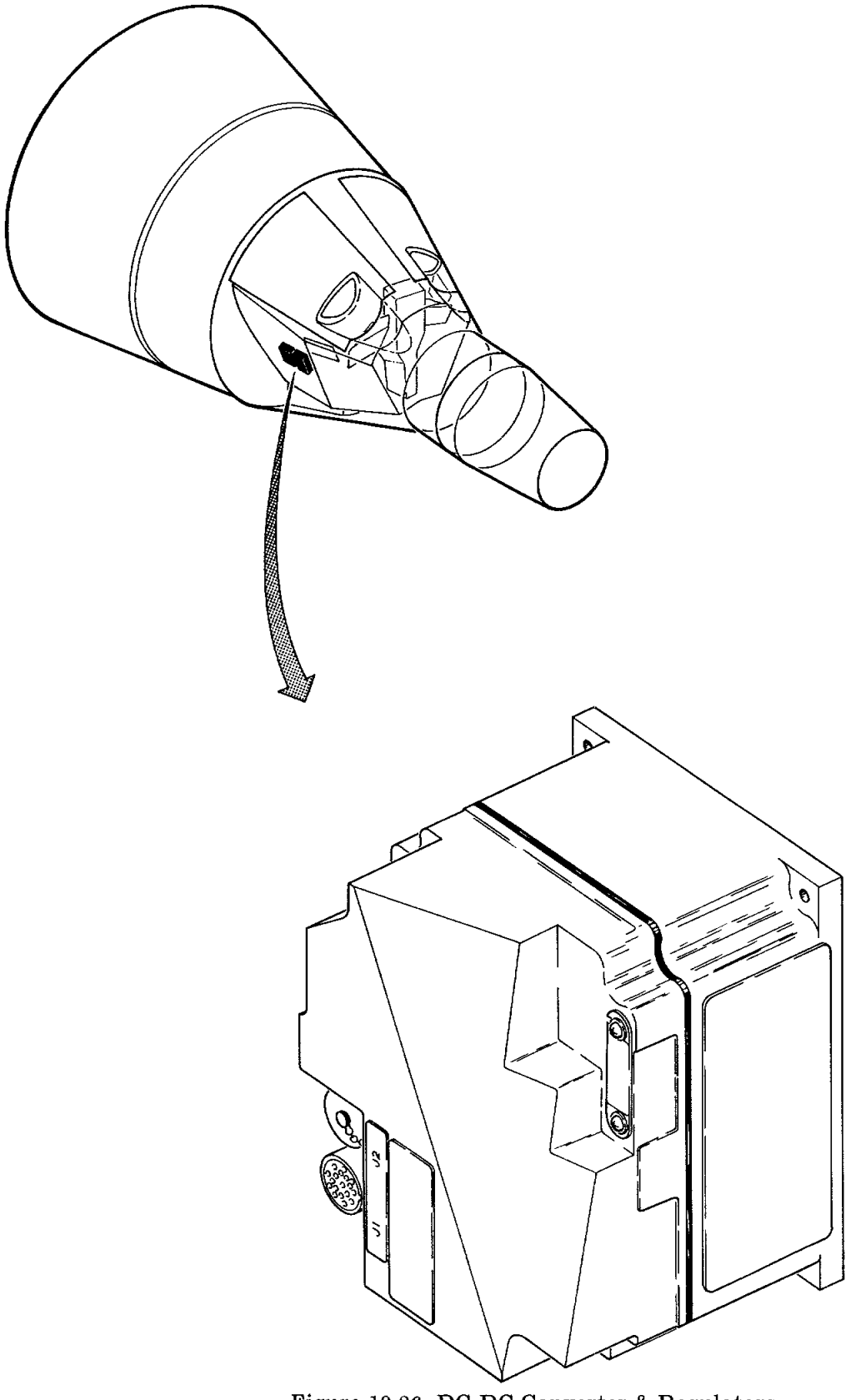


Figure 10-26 DC-DC Converter & Regulators

mentation System with regulated dc power. The units are approximately 5.5 x 5.5 x 7 inches weigh approximately seven pounds each, and are located in the right-hand equipment bay of the re-entry section as shown in Figure 10-25. The converters are essentially voltage regulators which operate on 18 to 30.5 vdc and supply output voltages of +5 vdc, +24 vdc and -24 vdc.

The power control circuitry for the dc-dc converters is shown in Figure 10-3. Essentially, input power to the dc-dc converters is supplied through the on position of the DC-DC CONV circuit breaker on the overhead switch/circuit breaker panel. This arms the DC-DC CONV switch. Placing the DC-DC CONV switch on the overhead switch/circuit breaker panel, to the SEC or PRI position, will apply power to the corresponding converter. Usage of the dc-dc converter regulated output voltages is illustrated in Figure 10-3.

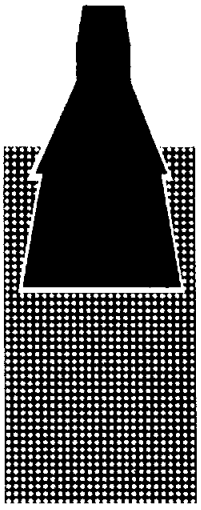
BIO-MED TAPE RECORDERS AND POWER SUPPLY

The two tape recorders used in the physiological Instrumentation System are identical. Each one is approximately 9 x 6 1/2 x 1 3/4 inches (excluding connector and mounting projections) and weighs about three pounds. One external connector provides termination points for all inputs and outputs. The circuitry is made up of 19 printed circuit boards with solid-state components. The recorder uses recording tape with a width of 0.497 $\pm$ 0.001 inches. The reel capacity is 880 feet. All physiological functions, except oral temperature and blood pressure of each pilot are recorded on the tape recorders. Each recorder has six data channels and one timing channel. The timing input is a pulse-coded pulsetrain derived from the Time Reference System through the Time Correlation Buffer

This signal is used for time correlation during post mission analysis. The recorders will operate for a total of 100 hours at a normal tape speed of 0.0293 inches per second. Recorder operation is controlled by the crew during the mission without playing back the data. Upon completion of the mission, the recorders are removed from the spacecraft so that the tape can be removed and the data extracted. The total power requirement of each recorder is 1.2 watts at 24 vdc.

The electrical control circuitry for the bio-med instrumentation is shown in Figure 10-2 and the location of the components is shown in Figure 10-1. The recorders are Government Furnished Equipment and are actuated from the spacecraft main bus through the BIO-MED INST circuit breaker and the CONT position of the BIO-MED RCDR switch (1 and 2).

The bio-med power supply, similar in construction to the dc-dc converters, supplies dc regulated voltage to the bio-med instrumentation. Input power for the converter is obtained from the main bus through the BIO-MED INST circuit breaker.



PYROTECHNICS and RETRO ROCKET SYSTEM

Section XI

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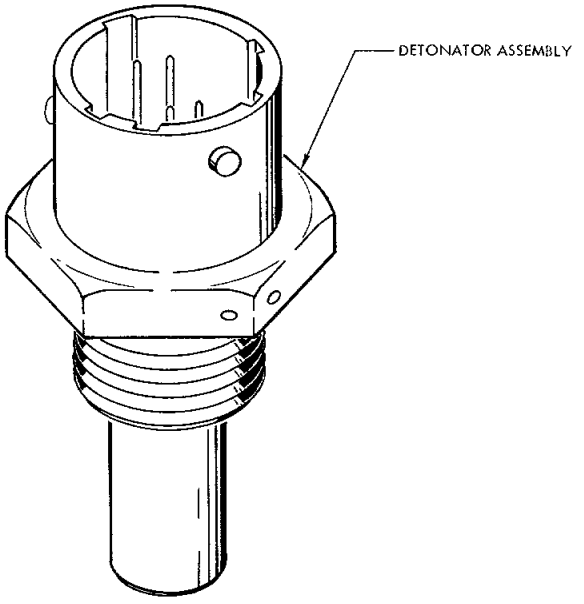
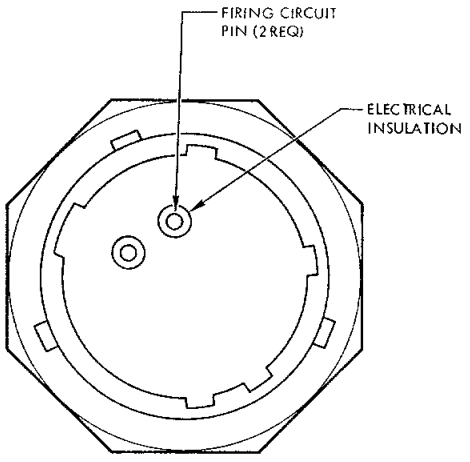
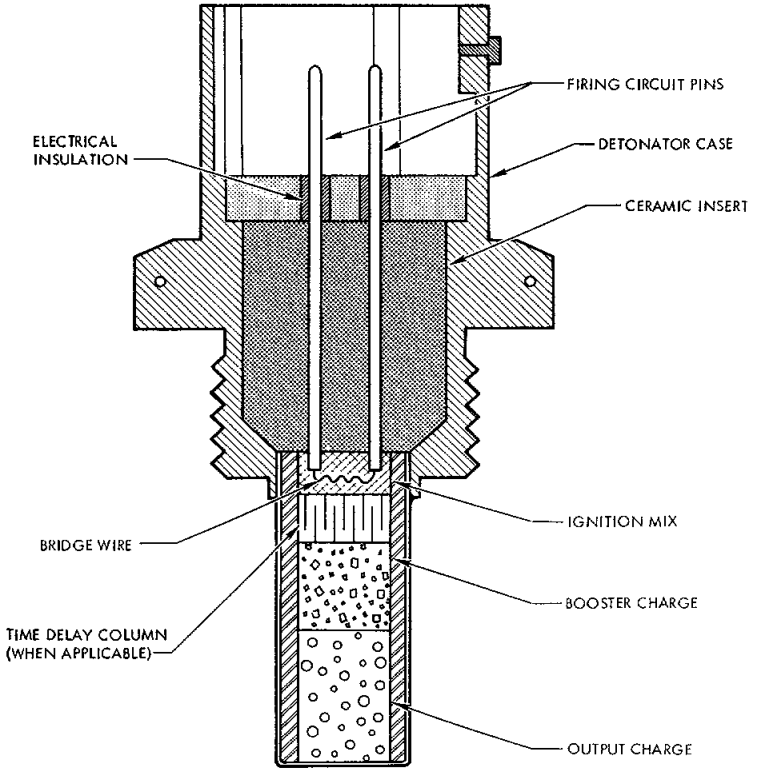
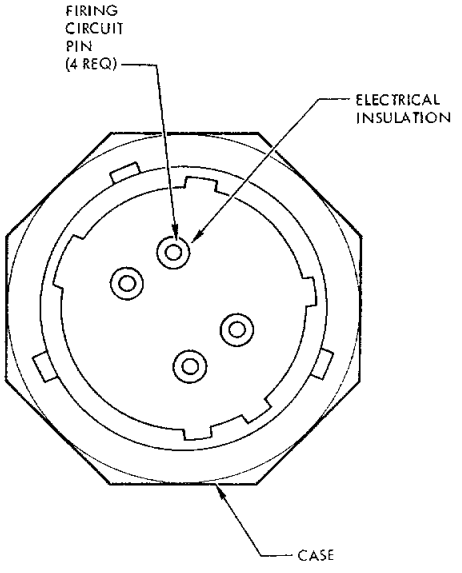




SINGLE BRIDGEWIRE DETONATOR
(ELECTRICAL RECEPTACLE)





DUAL BRIDGEWIRE DETONATOR
(ELECTRICAL RECEPTACLE)

Figure 11-1 Detonator (Typical)

SECTION XI PYROTECHNICS AND RETROGRADE ROCKETS

GENERAL INFORMATION

The pyrotechnic devices and retrograde rockets, installed in the Gemini Spacecraft, provide the escape system propulsion modes, enable and disable systems, and separate various sections and assemblies. Pyrotechnics are installed in each of the major sections and in numerous locations throughout the spacecraft. The retrograde rockets retard the spacecrafts orbital velocity to initiate re-entry into the earths atmosphere. The retrograde rockets are located in the retrograde section of the adapter.

RECURRENT COMPONENTS

Some pyrotechnic items are used extensively throughout the spacecraft. To avoid repetition in subsequent paragraphs, their description and operation will be presented at this time. When describing the various systems, these components shall be mentioned by name only.

DETONATOR

Description

The typical detonator (Figure 11-1) is a machined steel or aluminum cylinder containing an ignition mix, booster charge and an output charge. In some instances a pyrotechnic time delay column is used to provide a time delay between ignition and detonation. The case is threaded at one end for installation purposes. An electrical receptacle is provided at the other end. Electrically, the detonators are provided in two different configurations. One incorporates two independent, identical firing circuits. The other incorporates only one firing circuit. The

circuits of both detonators are electrically insulated from and independent of the detonator body. Each firing circuit consists of two electrical connector pins, across which a bridge wire is incorporated. The detonator is used to initiate high explosive components.

Operation

Upon receipt of a 28 vdc electrical signal, the firing circuit or circuits will cause the detonator to fire. Either circuit (detonators with dual circuits) will initiate the charge with the same performance characteristics as exist when both circuits are operative. The bridge wire ignites the ignition mix which in turn ignites the booster charge. The booster charge then propagates detonation to the output charge. If a delay column is installed, the ignition mix will ignite the delay column which ignites the booster charge. The output charge detonates and transmits the detonation wave to the assembly to which it is attached.

CARTRIDGE

Description

The typical cartridge (Figure 11-2) is a machined steel sylinder containing an ignition mix and an output charge. In some instances a pyrotechnic time delay column is used to provide a specific time delay between ignition and output. The cartridge is threaded at one end for installation purposes. An electrical receptacle is provided at the opposite end. Electrically, the cartridges are provided in two different configurations. One incorporates two independent, identical firing circuits. The other incorporates only firing circuit. The circuits of both cartridges are electrically insulated from and independent of the cartridge body. Each firing circuit consists of two electrical connector pins mounted in a high strength ceramic dielectric base and with a bridge wire

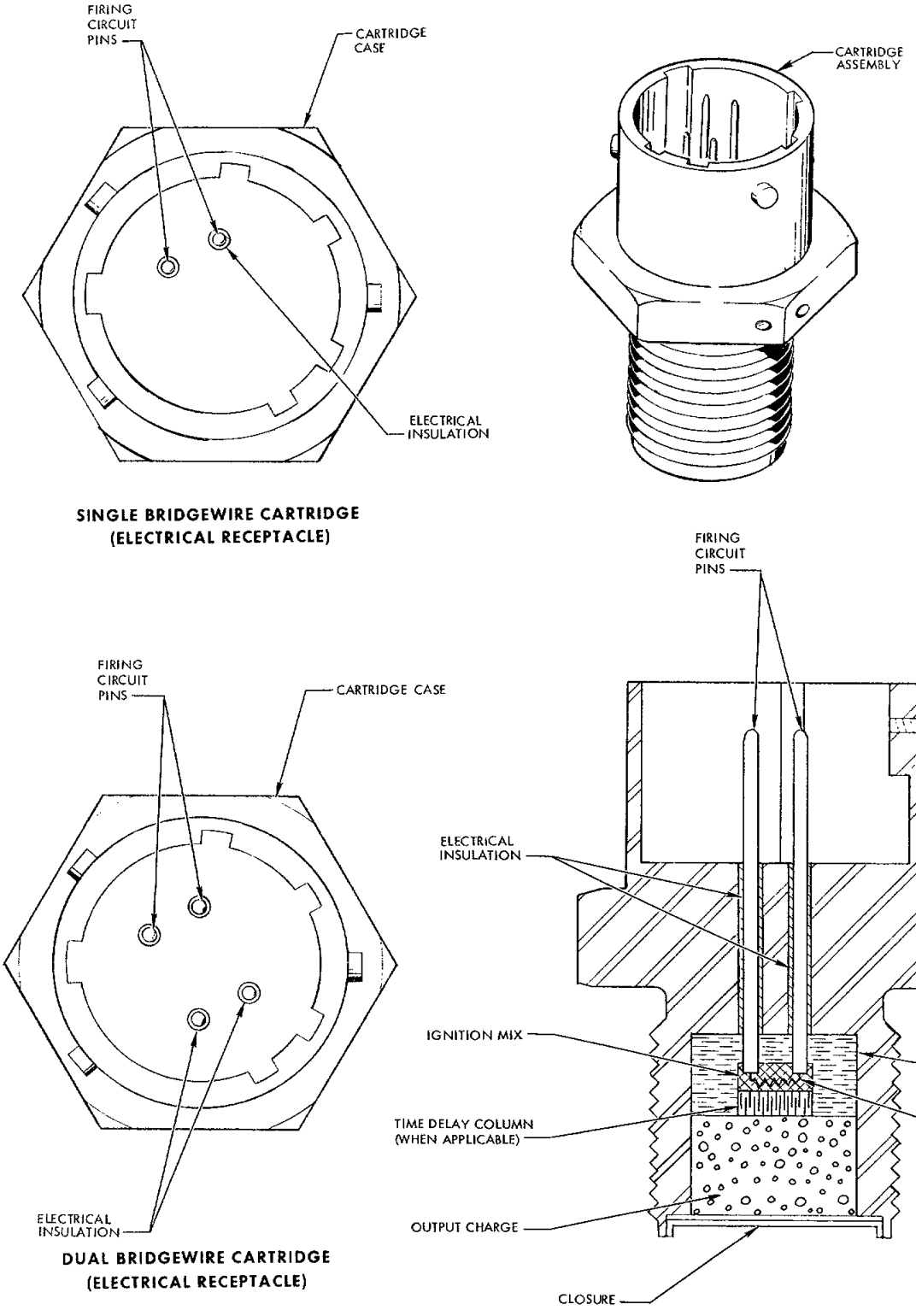


Figure 11-2 Cartridge (Gas Pressure)

connected between the two pins. The cartridge output is a hot gas pressure.

Operation

When initiated by a 28 vdc electrical signal, the firing circuits will cause the cartridge to function. Either circuit (cartridges with dual circuits) will fire the charge with the same performance characteristics as exist when both circuits are operative. The bridge wire ignites the ignition mix which propagates burning to the delay column, if applicable, and to the output charge. The output charge produces gas pressure that is used to operate the specific device in which the cartridge is installed.

FLEXIBLE LINEAR SHAPED CHARGE

Description

Flexible Linear Shaped Charge (FLSC) is a V-shaped, flexible lead sheathing containing a high explosive core. FLSC is used in separation assemblies to sever various types, thicknesses, and shapes of materials. The specific type, shape and thickness of the material to be separated, dictates the amount of explosive contained in the FLSC. In the Gemini Spacecraft and Agena Adapter, the FLSC is provided in four different core loadings: 7, 10, 20, and 25 grains per foot.

Operation

When installed, the open portion of the V-shaped FLSC is placed towards the item to be severed. The FLSC is detonated by a booster charge that has been initiated by a detonator. The explosive core of the FLSC detonates, resulting in collapse of the sheathing in the V groove, which produces a cutting jet composed of explosive products and minute metal particles. This jet produces extremely high localized pressures resulting in stress far above the yield strength of the target

material.

MILD DETONATING FUSE

Description

Mild Detonating Fuse (MDF) is a strand of high explosive encased in a lead sheathing with a circular cross section. MDF is used as a separation device and as an explosive interconnect. As a separation device, the strand contains 5 grains of explosive per foot. As an explosive interconnect, the strand contains 2 or 3.3 grains of explosive per foot. The interconnect type MDF is installed in either flexible woven steel mesh or nylon hose and rigid stainless steel tubing. Both rigid and flexible MDF have a small booster charge incorporated at each end. The booster charges are referred to as acceptor and donor. The acceptor being on the end that receives a detonation wave from an initiator. The donor being on the end that transmits a detonation wave to a component or other acceptor. The interconnects are attached to various devices by AN type or Bendix type electrical connectors.

Operation

The MDF used as a separation device is placed in a groove milled in a magnesium ring. The ring is formed to the shape of the items to be separated and is placed between the mating surfaces. The assembly to be jettisoned is attached to the main structure by frangible bolts. The bolts have been axially drilled to reduce tensile strength to a specified breaking point. When detonated, the MDF exerts a force against the mating surfaces greater than the tensile strength of the frangible bolts. The MDF, used as an explosive interconnect, is initiated when a detonator or booster charge propagates a detonation wave to the MDF booster. The

booster strengthens the wave and transmits it linearly through the length of the MDF strand. The booster, at the opposite end, propagates the detonation wave to the device to which it is attached.

SEPARATION ASSEMBLIES AND DEVICES

There are several different types of separation assemblies and devices used in the Gemini Spacecraft (Figure 11-3). These assemblies and devices are presented individually in the following paragraphs.

SPACECRAFT/LAUNCH VEHICLE SEPARATION ASSEMBLY

Description

The spacecraft/launch vehicle separation assembly (Figure 11-4) separates the spacecraft from the launch vehicle by severing the mating ring. The separation assembly primarily consists of two flexible linear shaped charges (FLSC) installed around the periphery of the mating ring, three detonators, three detonator blocks, three dual boosters, a molded backup retainer and a back blast shield. The dual boosters are inserted in the detonator blocks. The dual booster protrude into the molded backup retainer, indexed directly above the FLSC, when the detonator blocks are installed. The detonators are inserted in the detonator blocks with the output charge adjacent to the dual boosters. The back blast shield attaches the molded backup retainer and FLSC to the mating ring.

Operation

Upon receipt of the 28 vdc electrical signal, the detonators transmit a detonation wave that is propagated to the dual boosters. The dual boosters strengthen the detonation wave to achieve proper detonation of the FLSC The FLSC detonates and severs the mating ring redundantly. The backup retainer absorbs the shock in the

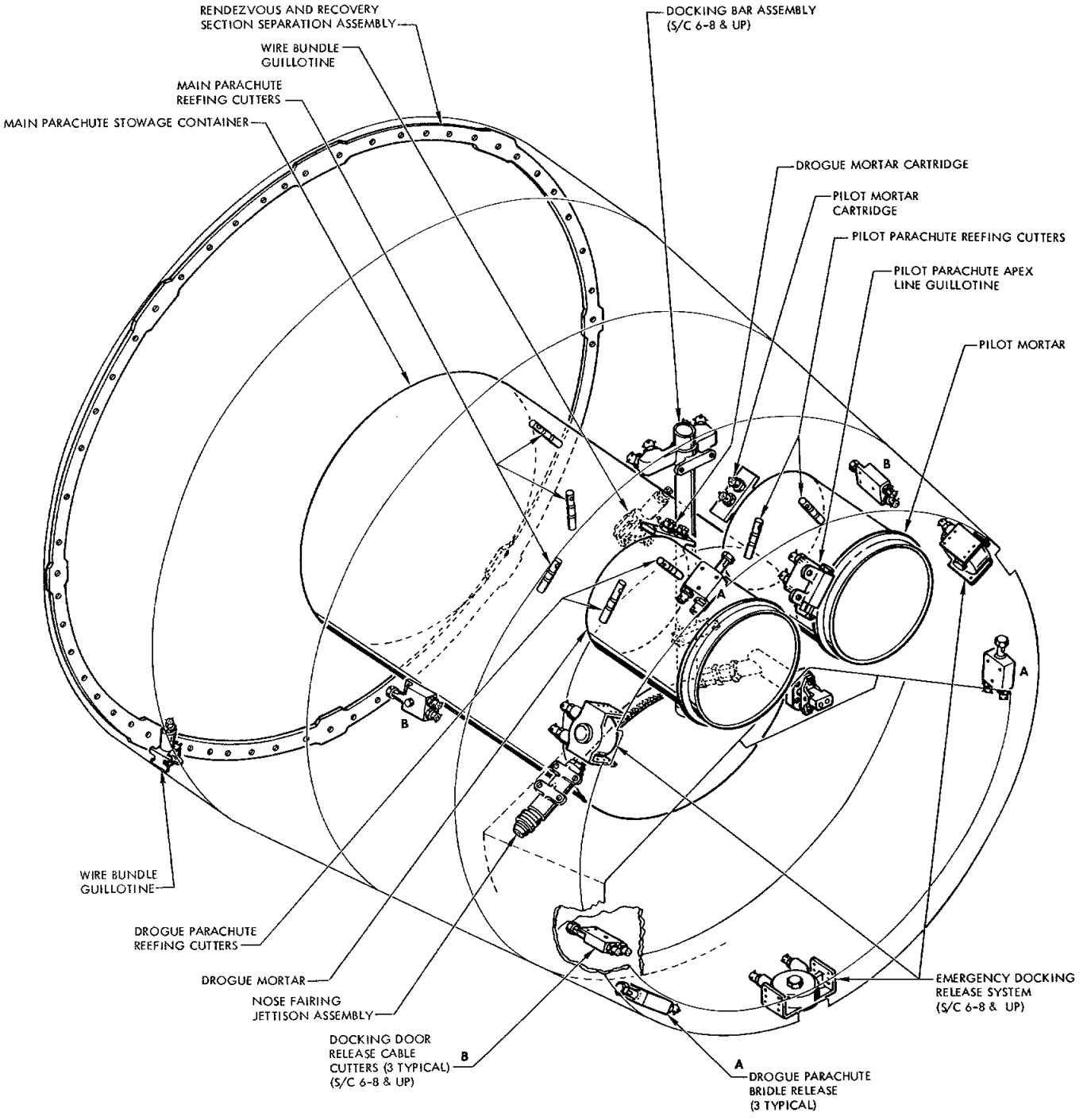


Figure 11-3 Spacecraft Pyrotechnic Devices (R & R Section) (Sheet 1 of 3)

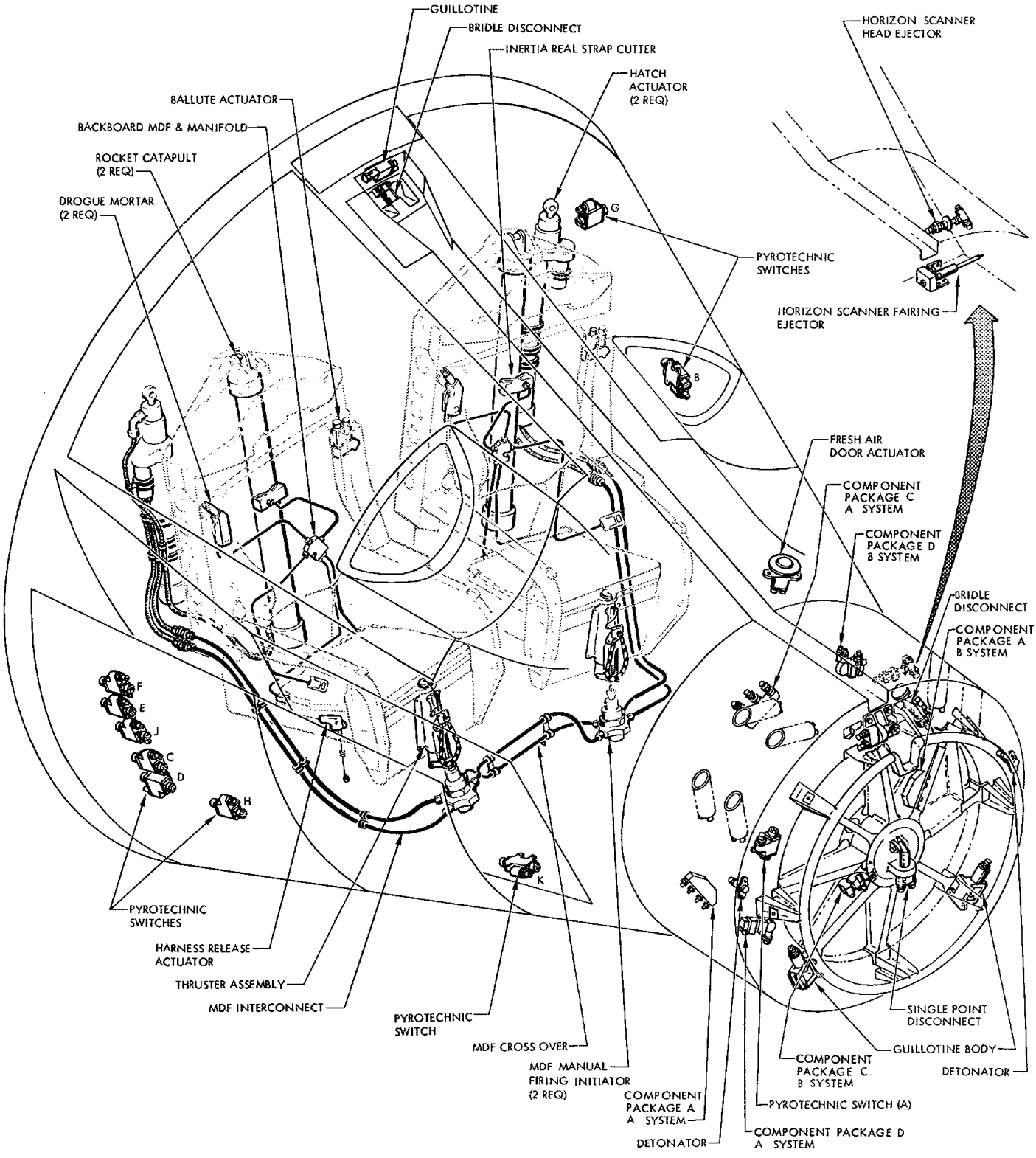


Figure 11-3 Spacecraft Pyrotechnic Devices (Landing Module) (Sheet 2 of 3)

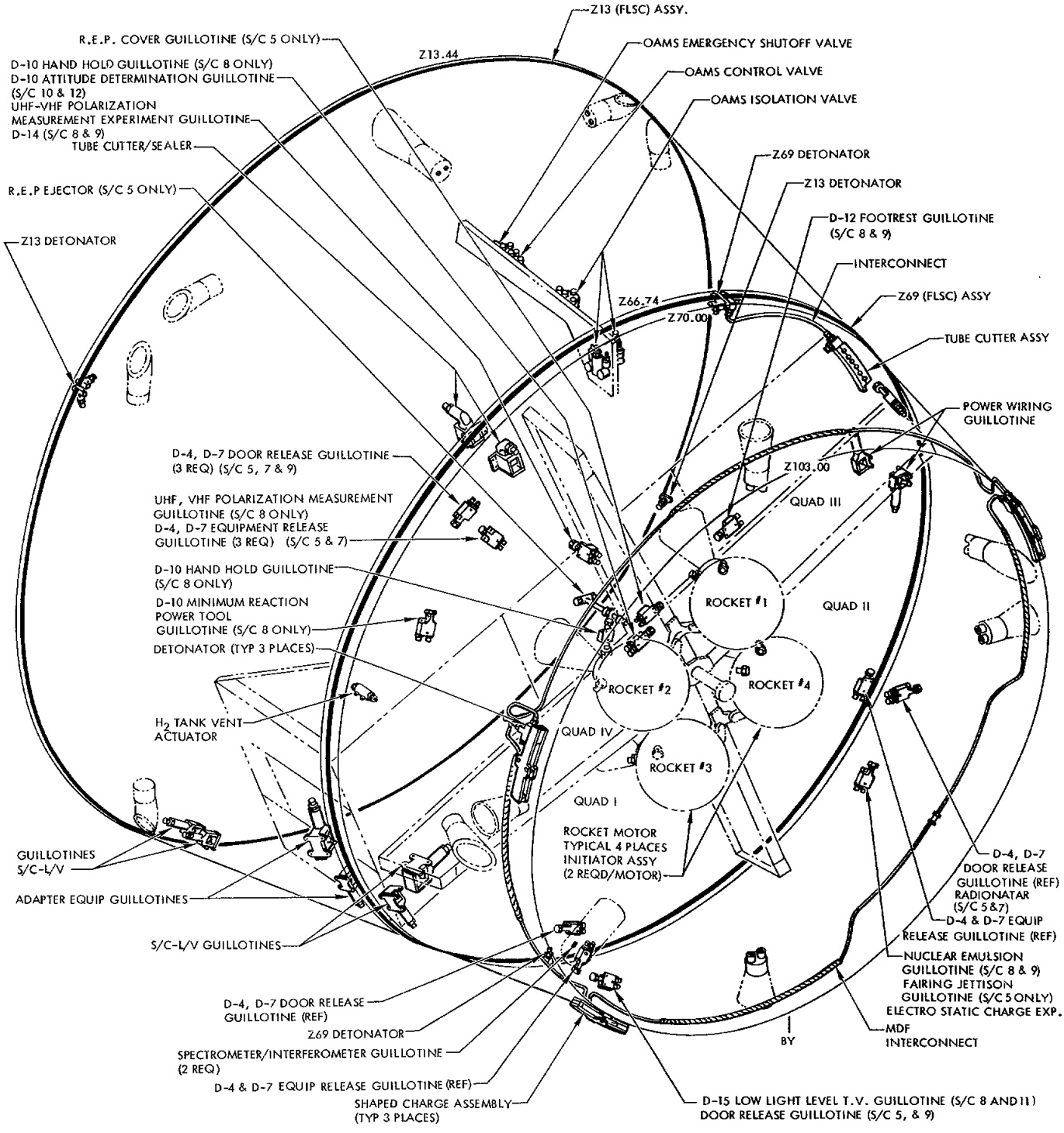


Figure 11-3 Spacecraft Pyrotechnic Devices (Adapter) (Sheet 3 of 3)

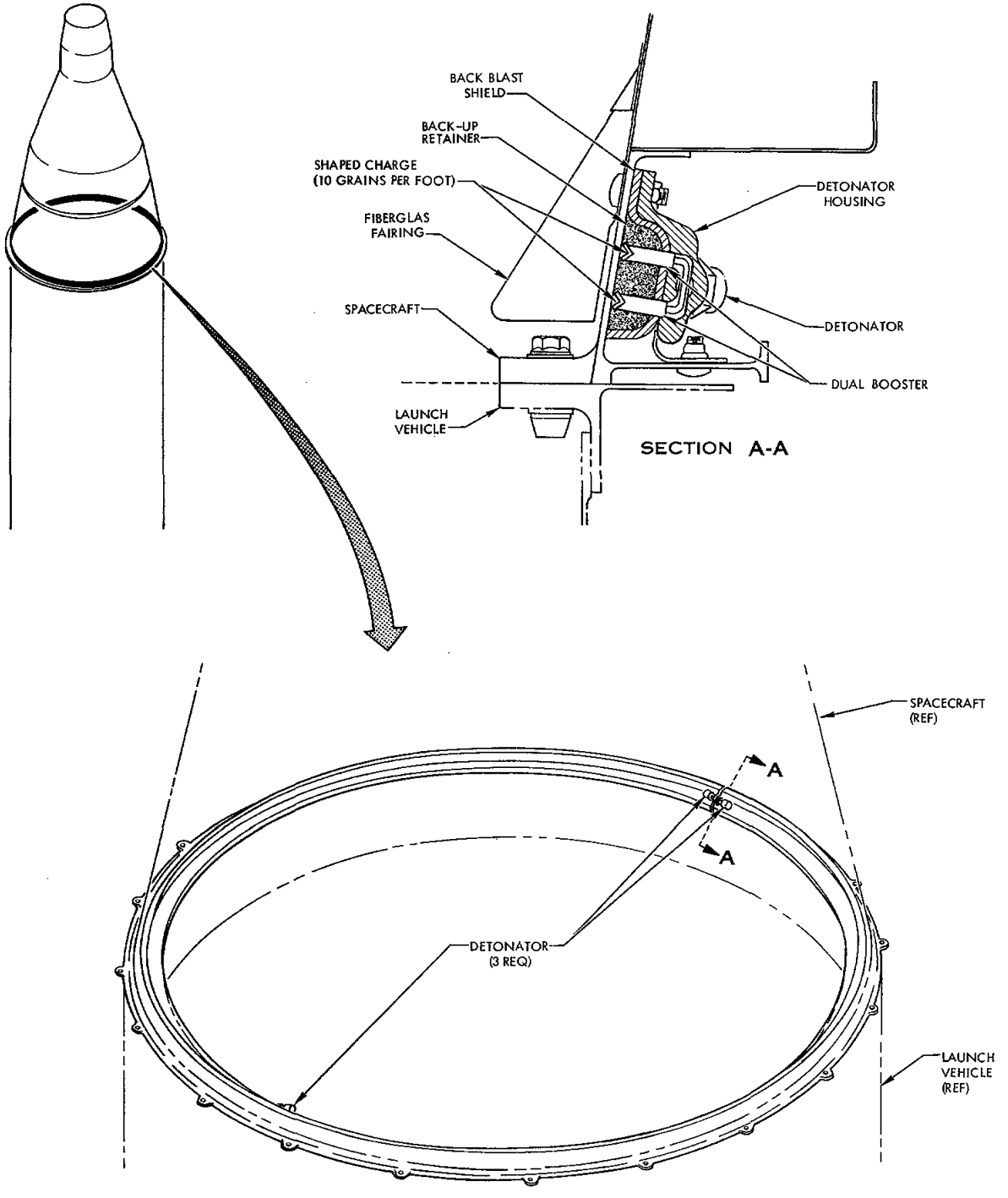


Figure 11-4 Spacecraft/Launch Vehicle Separation Assembly

back blast. The back blast shield protects the structure and equipment from shrapnel. Proper detonation of only one strand of FLSC is sufficient to sever the mating ring.

EQUIPMENT SECTION/RETROGRADE SECTION SEPARATION ASSEMBLY

Description

The equipment section/retrograde section separation assembly (Figure 11-5) separates the equipment section of the adapter from the retrograde section of the adapter. The assembly basically consists of two main units: the shaped charge assembly and the tubing cutter assembly. The shaped charge assembly primarily consists of two flexible linear shaped charges (FLSC), three detonator blocks, containing three crossovers and six boosters, three detonators, ten segmented backup strips and a molded backup retainer. The detonator blocks provide for installation of the detonators. One detonator block provides for the installation of the tubing cutter explosive interconnect. The tubing cutter assembly primarily consists of an explosive interconnect (MDF), two formed aluminum parallel housings, molded backup retainer, two flexible linear shaped charges with boosters attached, a detonator block and a detonator. The explosive interconnect (MDF) is a flexible nylon hose containing a strand of high explosive and end mounted booster charges. The interconnect has Bendix type connectors incorporated at each end for attaching the interconnect to the cutter and shaped charge detonator blocks. The interconnect is attached to the cutter detonator block with its booster charge adjacent to one of the boosters on the FLSC. The detonator is installed in the cutter detonator block with its output end adjacent to the other booster on the FLSC. The cutter assembly is bracket mounted to the inside of the retrograde section of the adapter, forward of the parting line. The shaped charge assembly is

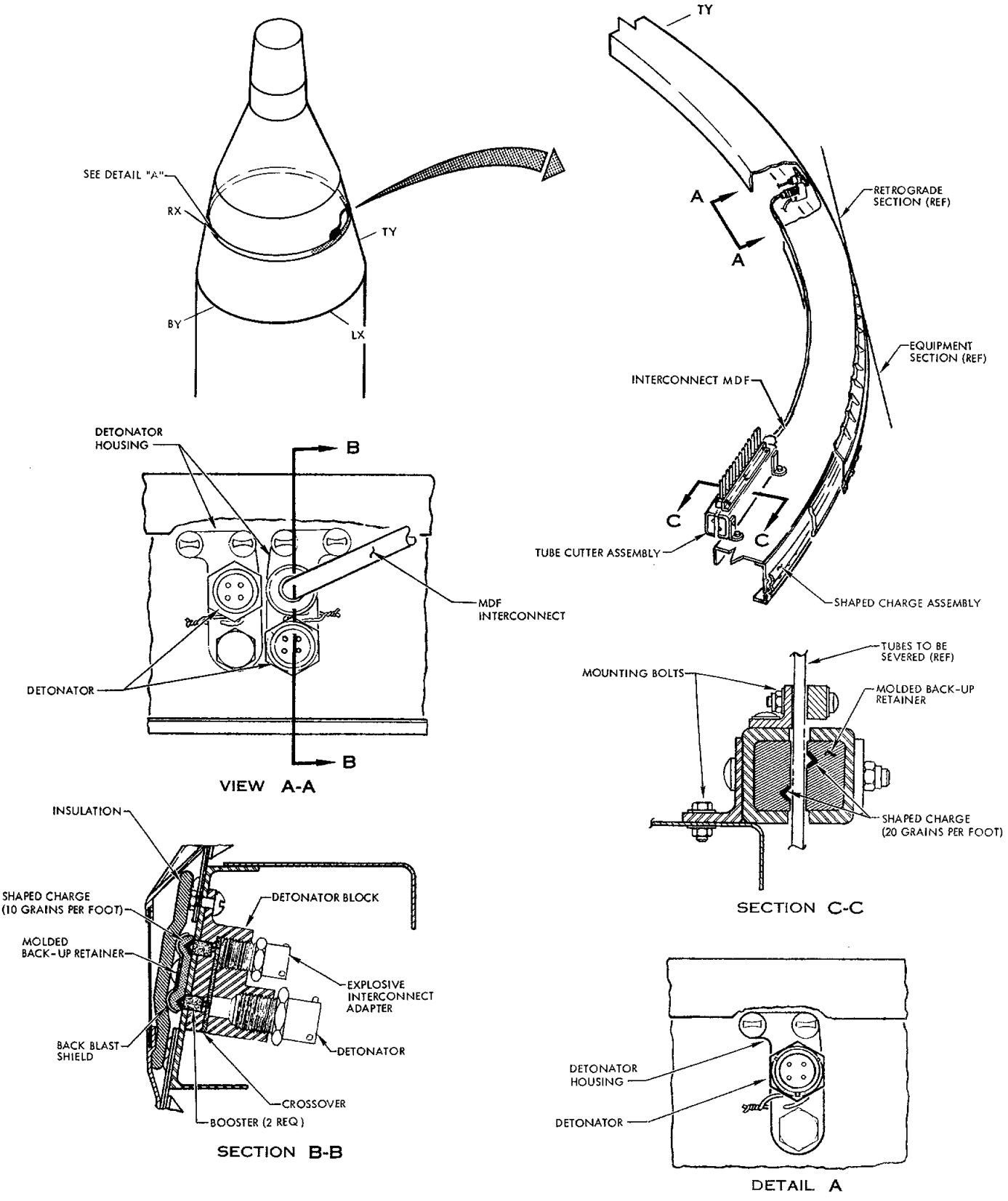


Figure 11-5 Separation Assembly-Equipment Section/Retrograde Section

installed around the outer periphery of the adapter at the equipment section and retrograde section parting line.

Operation

When initiated by a 28 vdc electrical signal, the detonators will fire. The detonators of the shaped charge assembly transmit a shock or detonation wave to the crossovers which in turn initiates the boosters. The boosters propagate the wave to the FLSC. The FLSC detonates and functions to sever the adapter at the parting line redundantly. The detonator of the tubing cutter assembly propagates detonation to the booster on one strand of FLSC in the cutter assembly. The explosive interconnect transmits detonation from the shaped charge assembly to the booster on the other strand of FLSC in the cutter assembly. The two boosters propagate the shock wave to the FLSC. The two strands of FLSC in the cutter assembly detonate and sever the twelve aluminum tubes and one nylon tube. Proper detonation of only one strand of FLSC, in both the shaped charge assembly and tubing cutter assembly, is sufficient to achieve separation.

RETROGRADE SECTION/RE-ENTRY MODULE SEPARATION ASSEMBLY

Description

The retrograde section/re-entry module separation assembly (Figure 11-6) functions to separate the retrograde section of the adapter from the re-entry module. Separation is accomplished by severing the three titanium straps and various tubes and wire bundles. The separation assembly primarily consists of three cutter assemblies, three detonator housings, three detonators, three parallel booster columns, six explosive interconnects, and three unions. The detonator housings contain a booster column and a parallel booster column. The cutter assemblies

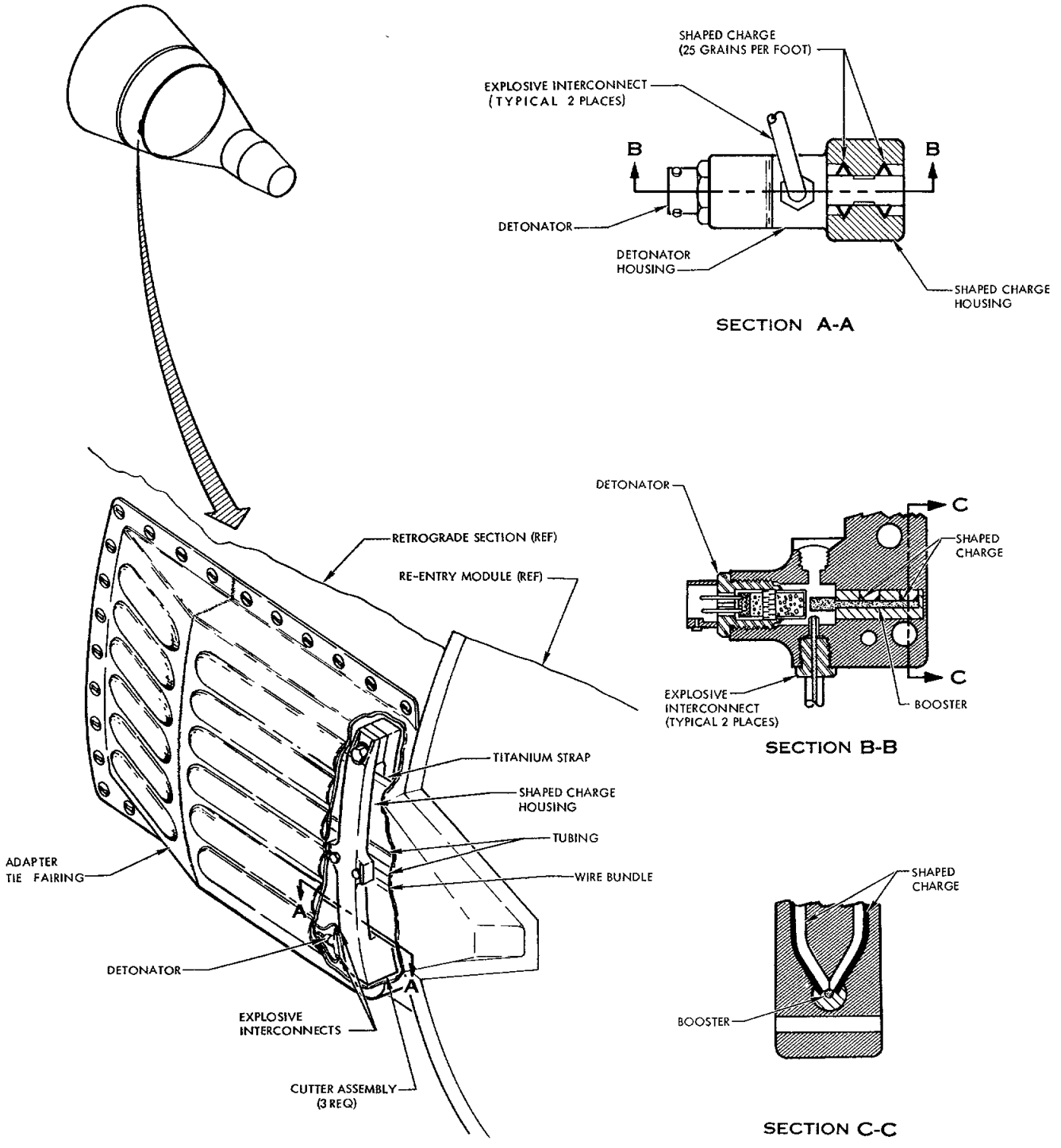


Figure 11-6 Retrograde Section/Re-Entry Module Separation Assembly

consist of two parallel machined aluminum bars that contain four strips of FLSC. The bars are joined by the detonator housings with the parallel boosters. A detonator is installed in each of the three detonator housings. The cutter assemblies are located in three places around the parting line and are linked by the explosive interconnects.

Operation

When initiated by a 28 vdc electrical signal, the detonators propagate a detonation or shock wave to the boosters which relay propagation to cutter FLSC and simultaneously the shock wave is propagated to the explosive interconnects. The interconnects transmit the wave to all three cutter assemblies. This is to ensure detonation of all three cutters FLSC, in the event one or even two detonators do not function. Detonation of the cutter FLSC completely severs the titanium straps, wire bundles and tubing redundantly. Proper detonation of only two opposing strips of FLSC in each cutter is sufficient to achieve separation.

RENDEZVOUS AND RECOVERY SECTION SEPARATION ASSEMBLY

Description

The rendezvous and recovery section separation assembly (Figure 11-7) separates the Rendezvous and Recovery (R & R) section from the Re-entry Control System (RCS) section. The assembly primarily consists of Mild Detonating Fuse (MDF), MDF housing ring, two detonators, two detonator housings and two booster charges. Two strands of MDF are installed in parallel grooves milled in the housing ring face. The grooves intersect at the booster charges which are installed approximately 180° apart. The R & R section is attached to the RCS section by frangible bolts, with the MDF ring fastened to the R & R section at the mating surface. The detonator

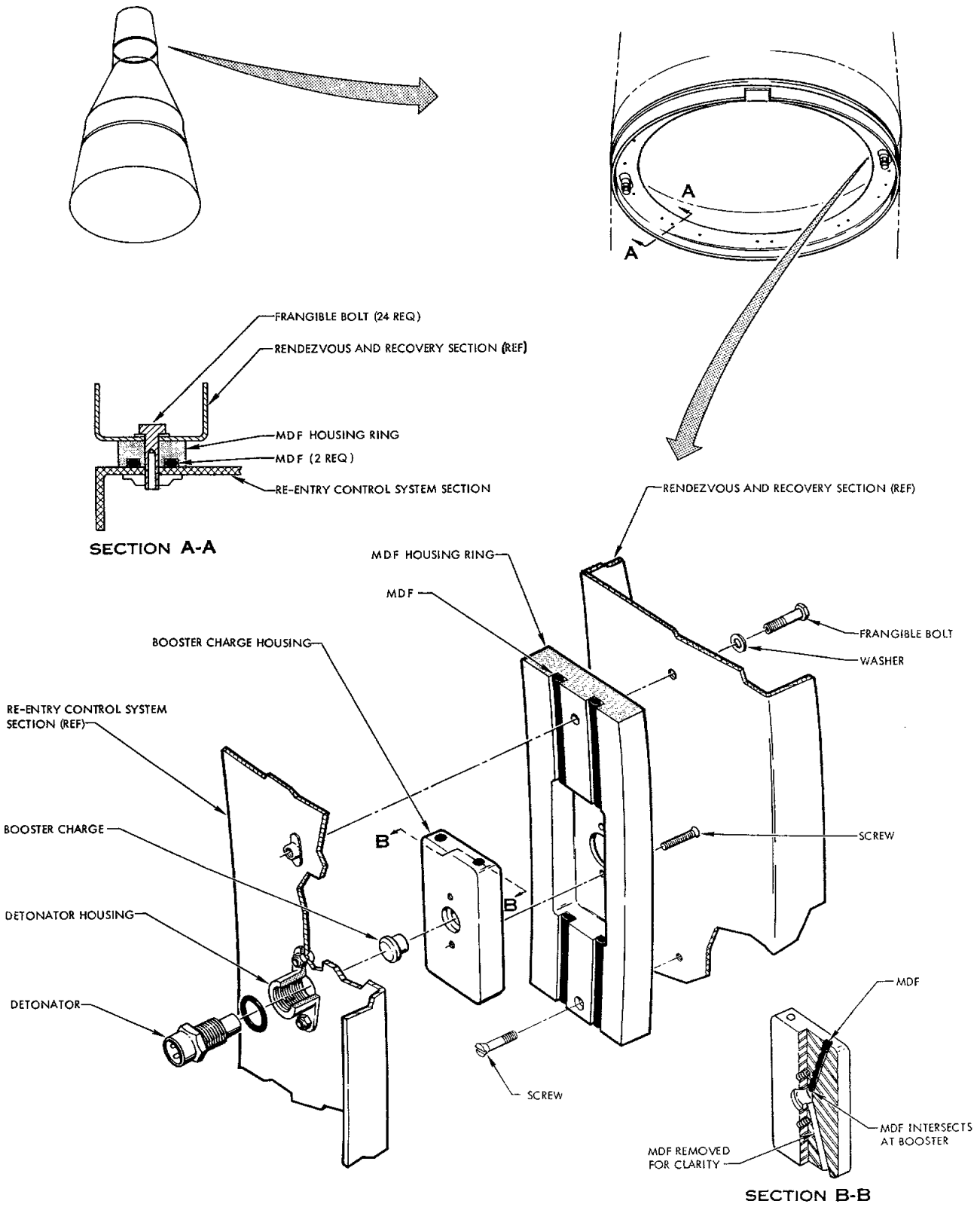


Figure 11-7 Rendezvous and Recovery Section Separation Assembly

housings are installed in the RCS section, with the detonators indexed directly above the booster charges, when the sections are mated.

Operation

When initiated by a 28 vdc electrical signal, the detonators propagate a detonation wave to the two booster charges. The booster charges strengthen the detonation wave and transmit it to the dual strands of MDF. The MDF detonates, exerting a force against the RCS and R & R section mating surfaces. The force breaks all the frangible bolts and allows the pilot chute to pull the R & R section free of the spacecraft. Satisfactory propagation of either strand of MDF will successfully separate the R & R section.

WIRE BUNDLE GUILLOTINE

Description

The wire bundle guillotine (Figure 11-8) is used throughout the spacecraft to sever various sized bundles of electrical wires. The guillotines are used in two sizes. One size can sever a wire bundle up to one and one quarter inches in diameter and the other can sever a wire bundle up to two and one half inches in diameter. Both sizes are similar in design, appearance and operation. The guillotines primarily consist of a body, end cap or anvil, piston/cutter blade, shear pin(s) and an electrically fired gas pressure cartridge. The body houses the piston/cutter blade, provides for installation of the cartridge, and attachment of the anvil. The anvil is removable to facilitate removal and installation of either the guillotine or wire bundle. Two guillotines are used on a wire bundle, one on each side of the separation plane. Lugs, for attaching the guillotine to the spacecraft structure, are an integral part of the guillotine body.

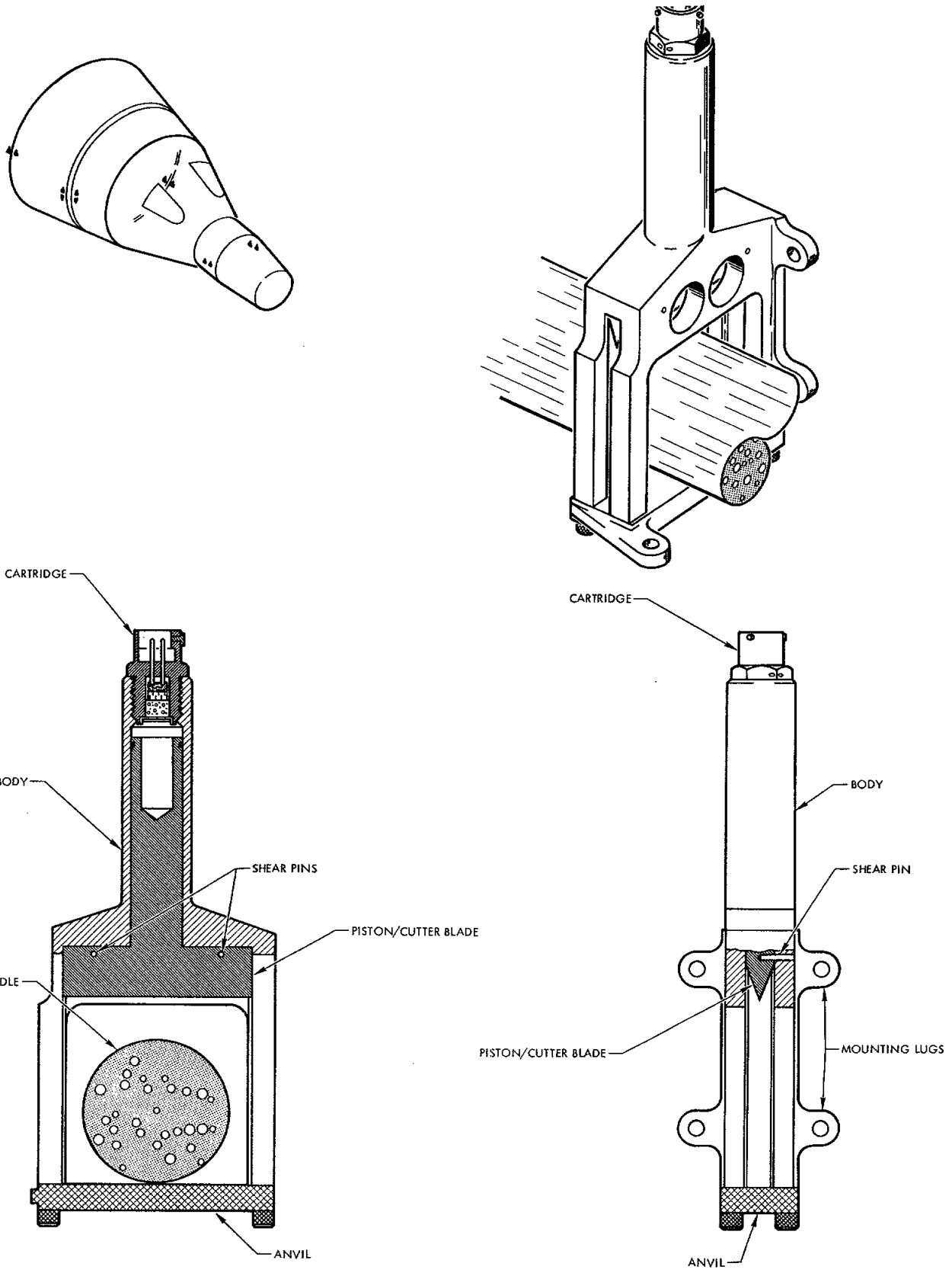


Figure 11-8 Wire Bundle Guillotine

Operation

When initiated by the proper electrical signal, the cratridge produces gas pressure. This gas pressure, exerts force on the piston cutter blade. When sufficient force is applied, the piston/cutter blade will sever the shear pin(s). As the pin(s) shear, the piston/cutter blade strokes, and completely severs the wire bundle. The wire bundle is then free to pull out of the guillotine body.

WIRE BUNDLE GUILLOTINE (CABLE CUTTING)

Description

The wire bundle guillotine (cable cutting) (Figure 11-9) is used to sever woven stainless steel cables. The guillotine primarily consists of the body, piston/cutter blade, shear pin, anvil and end cap, and two electrically fired gas pressure cartridges. The body provides a piston actuation area and provides for cartridge installation. The anvil is retained in the barrel section of the body by the end cap. The anvil and end cap is removable to permit guillotine and cable installation and removal. Lugs, for attaching the guillotine to the spacecraft structure, are an integral part of the body. The shear pin is provided to retain the piston/cutter blade in a retracted position.

Operation

When initiated by a 28 vdc electrical signal, the two cartridges fire and produce gas pressure. The gas pressure exerts force on the piston/cutter blade. When sufficient force is applied, the piston/cutter blade severs the shear pin. The piston/cutter blade travels the length of the barrel section and severs the cable installed in the guillotine. The cable is then free to pull out of the guillotine.

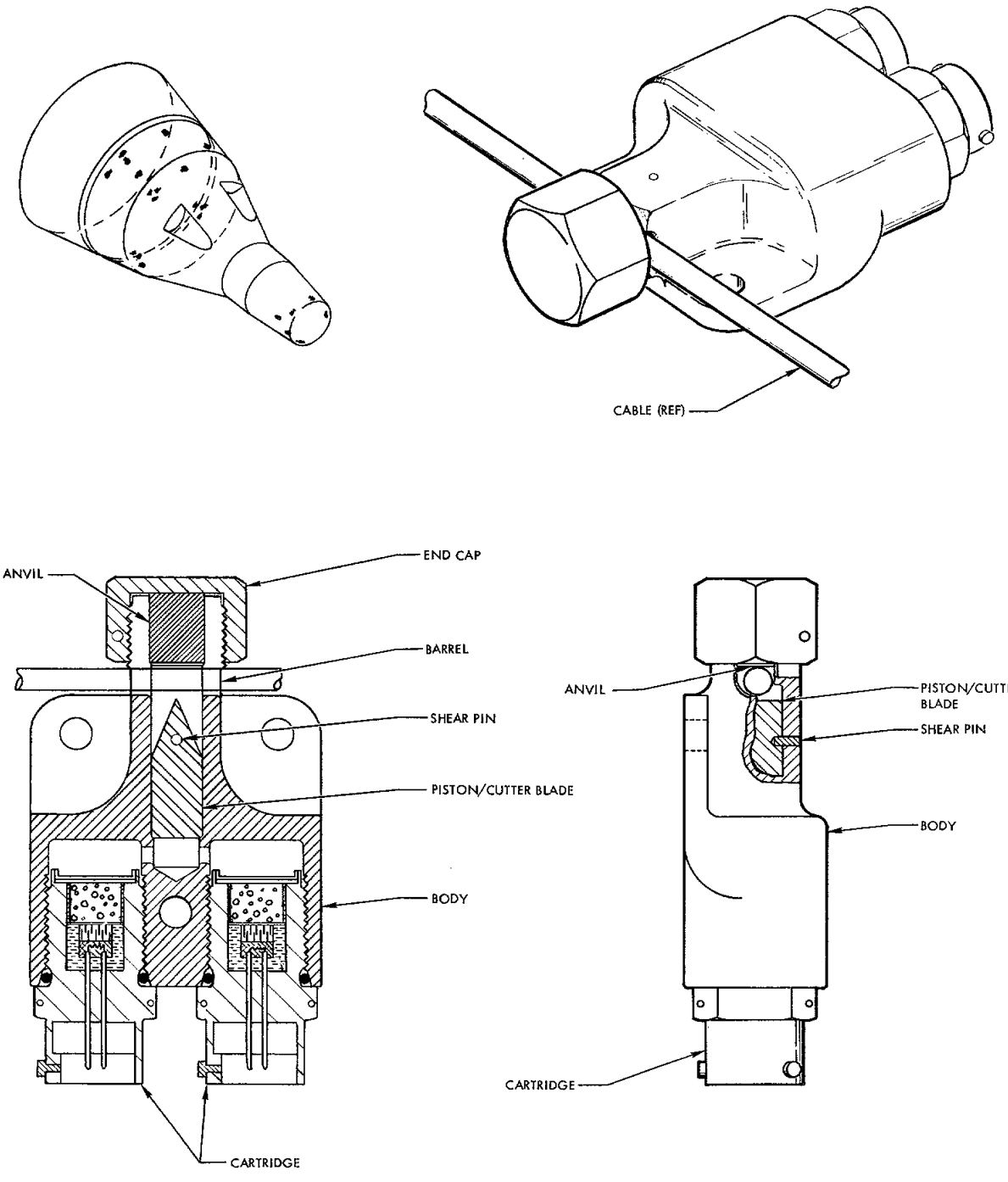


Figure 11-9 Wire Bundle Guillotine (Cable Cutting)

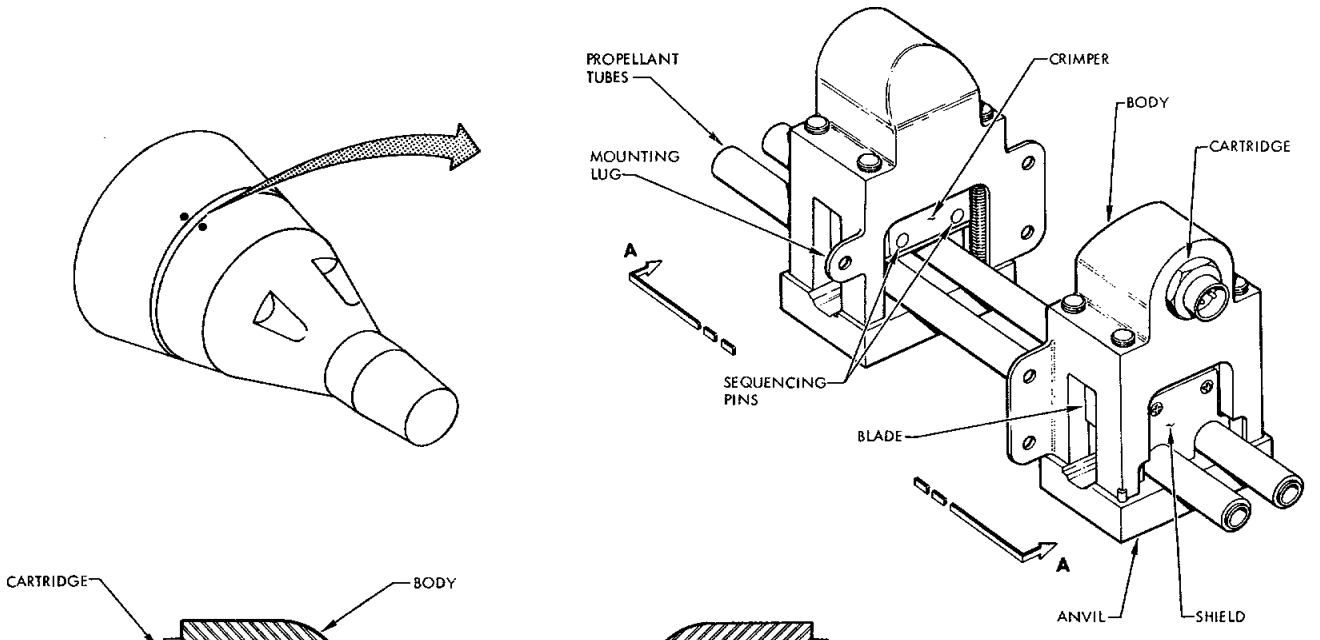
TUBING CUTTER/SEALER

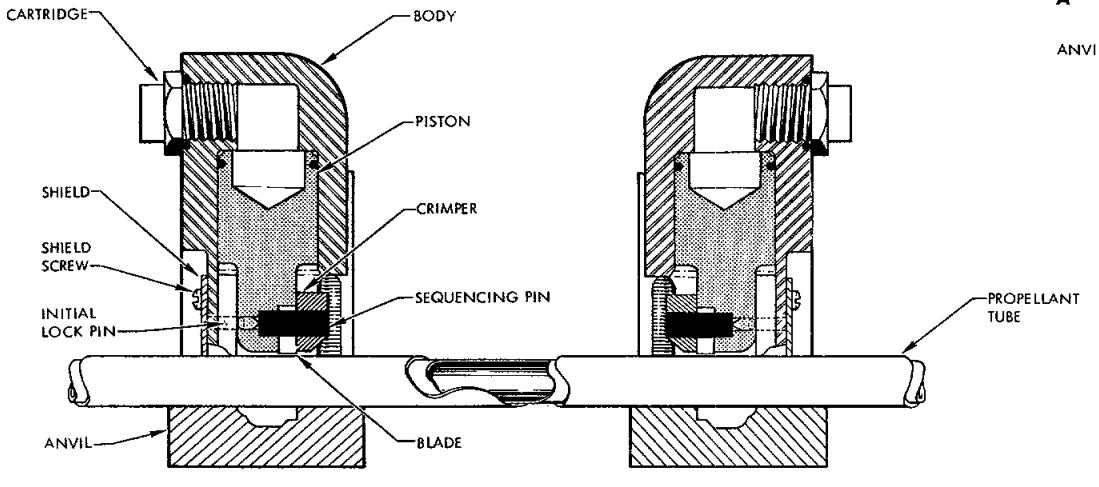
Description

The tubing cutter/sealer (Figure 11-10) is used to cut and seal two stainless steel, Teflon lined tubes. The tubes contain hypergolic propellants used in the Orbit Attitude and Maneuvering System (OAMS). Two tubing cutter/sealer assemblies are located in the adapter, one on each side of the retrograde/equipment section separation line. The tubing cutter/sealer assembly primarily consists of the body, anvil, one electrically fired gas pressure cartridge, four shear pins and cutter assembly. The cutter assembly consists of the piston, crimper and blade. The crimper and blade are attached to the piston by two of the shear pins, (sequencing pins). The piston is secured in the body by the other two shear pins, (initial lock pins). The body provides for the installation of the cartridge, attachment of the anvil, and housing for the cutter assembly. Lugs, for attaching the tubing cutter/sealer to the spacecraft structure, are an integral part of the body.

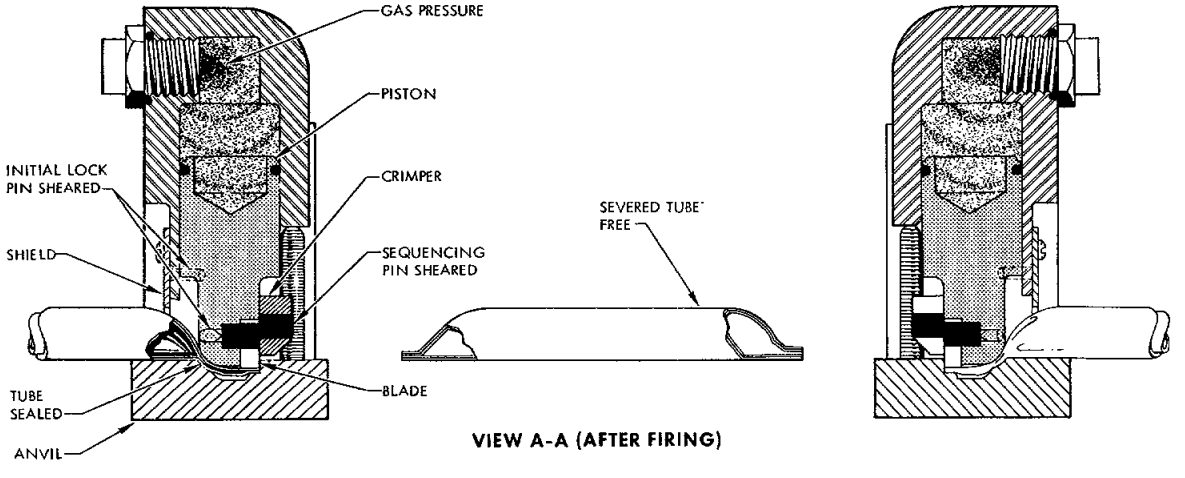
Operation

When initiated by a 28 vdc electrical signal, the cartridge fires and generates gas pressure. The gas pressure exerts a force on the piston of the cutter assembly. When sufficient force is applied to the piston, the initial lock pins are severed and the cutter assembly strokes to seal and cut the two tubes. The blade and crimper, extending past the end of the piston, contact the tubing first. As the cutter assembly moves down, the crimper flattens the tubing against the raised portion of the anvil. As the cutter assembly continues its travel, the sequencing pins are severed between the crimper and blade, stopping the travel of the crimper. The base of the piston and blade further crimp and seal the tubing with the blade severing the tubing. The sealed portion of the tubing remains in





VIEW A-A (BEFORE FIRING)



VIEW A-A (AFTER FIRING)

Figure 11-10 Tubing Cutter/Sealer

the tubing cutter/sealers at adapter separation. The severed portion of the tubing between the tubing cutter/sealers is free to pull out at adapter separation.

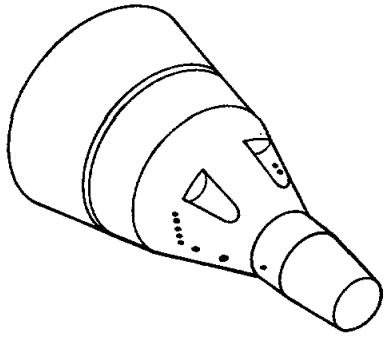
PYROTECHNIC SWITCH

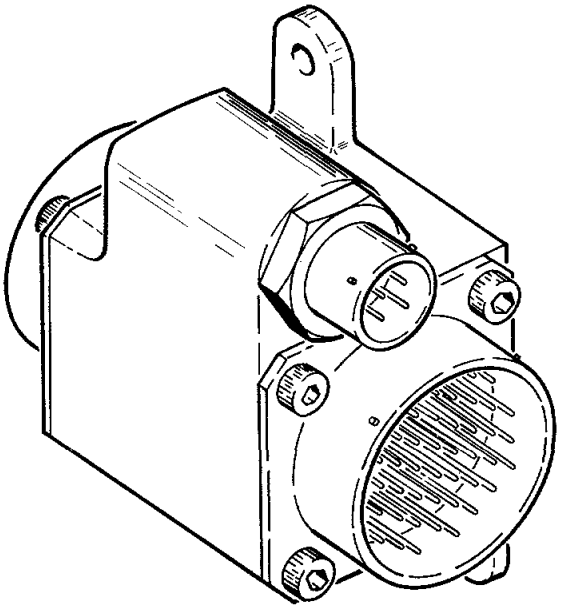
Description

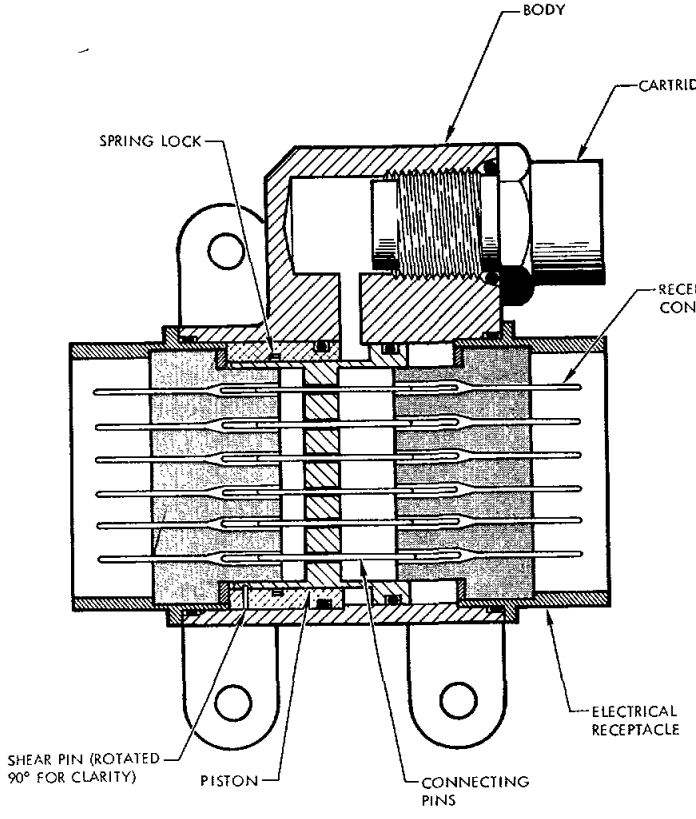
The pyrotechnic switch (Figure 11-11) functions to positively open electrical circuits and prevent current flow in various wire bundles prior to their being severed. The switches are located in various places throughout the re-entry module. The switches primarily consist of the body, actuator (piston), shear pin, spring lock, and electrically fired gas pressure cartridge. The shear pin secures the actuator in the switch closed position prior to switch actuation. Incorporated in opposite ends of the switch body are two electrical receptacles. The end mounted receptacles contain hollow spring leaf contacts. The contacts are axially connected by pins mounted in the actuator. All switches are identical in design and operation with the exception of the number of contacts in the receptacles. One model contains 41 contacts, and the other model contains 55 contacts. Lugs, for attaching the switch to the spacecraft structure, are an integral part of the body.

Operation

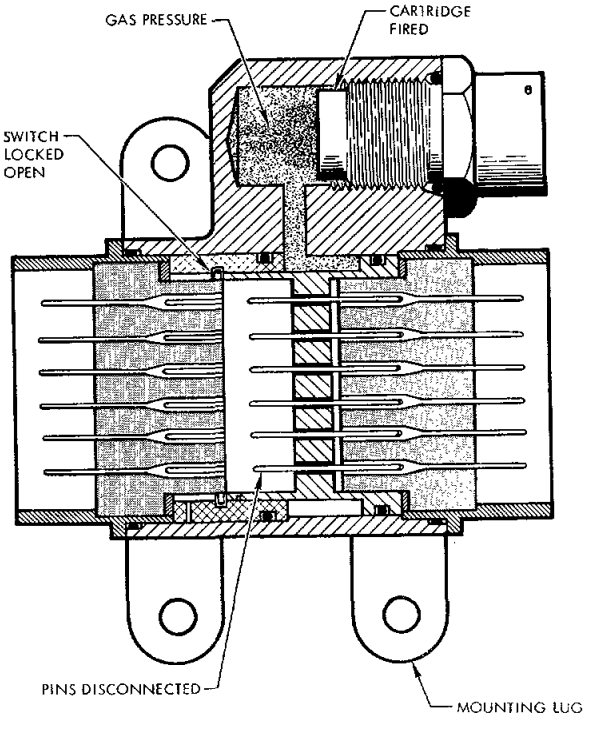
When initiated by a 28 vdc electrical signal, the cartridge fires and generates gas pressure that is ported through the switch body to the actuator. The pressure exerts a force against a flange of the actuator. The force causes the actuator to sever the shear pin and move axially in the body. As the actuator moves, the connecting pins mounted in the actuator are disengaged from the hollow contacts at one end and are driven further into the hollow contacts at the other end. The spring lock drops into place behind the actuator and prevents it from returning to







SWITCH CLOSED



SWITCH OPEN

Figure 11-11 Pyrotechnic Switch

its original position. The actuator is thus held in the electrically open position.

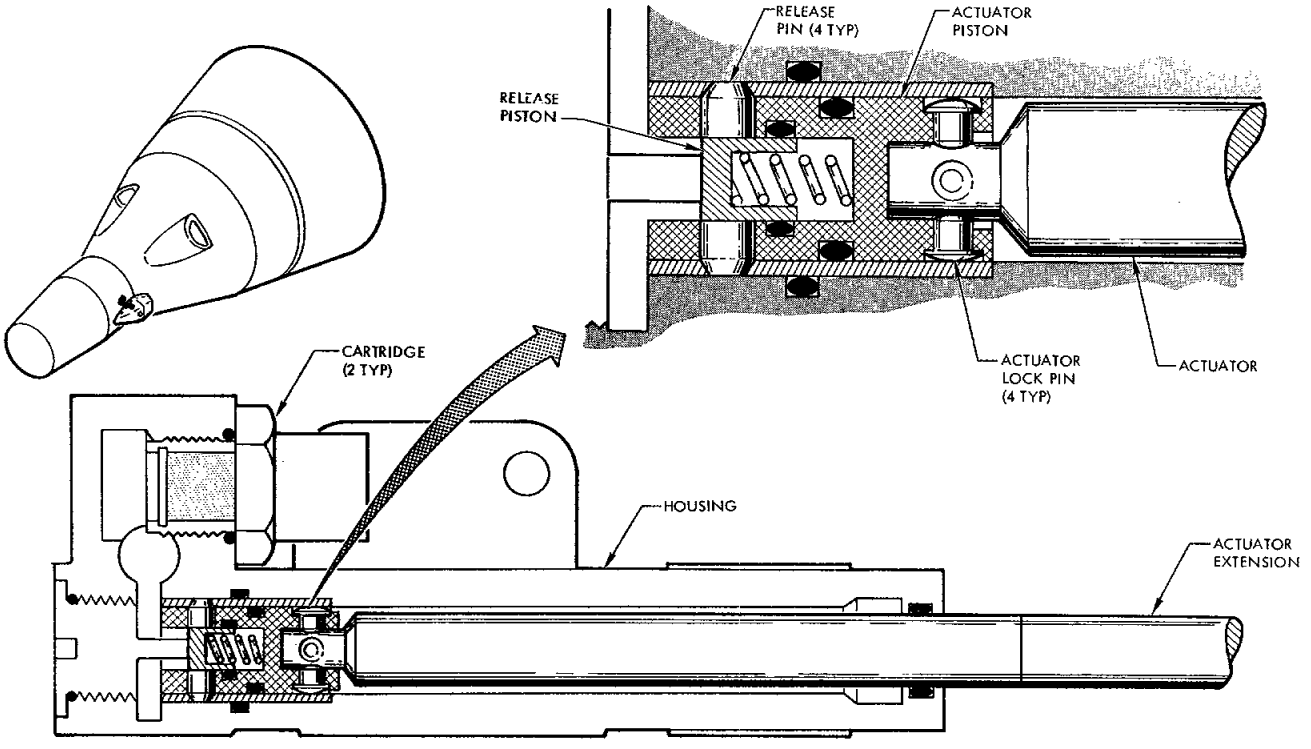
HORIZON SCANNER FAIRING RELEASE ASSEMBLY

Description

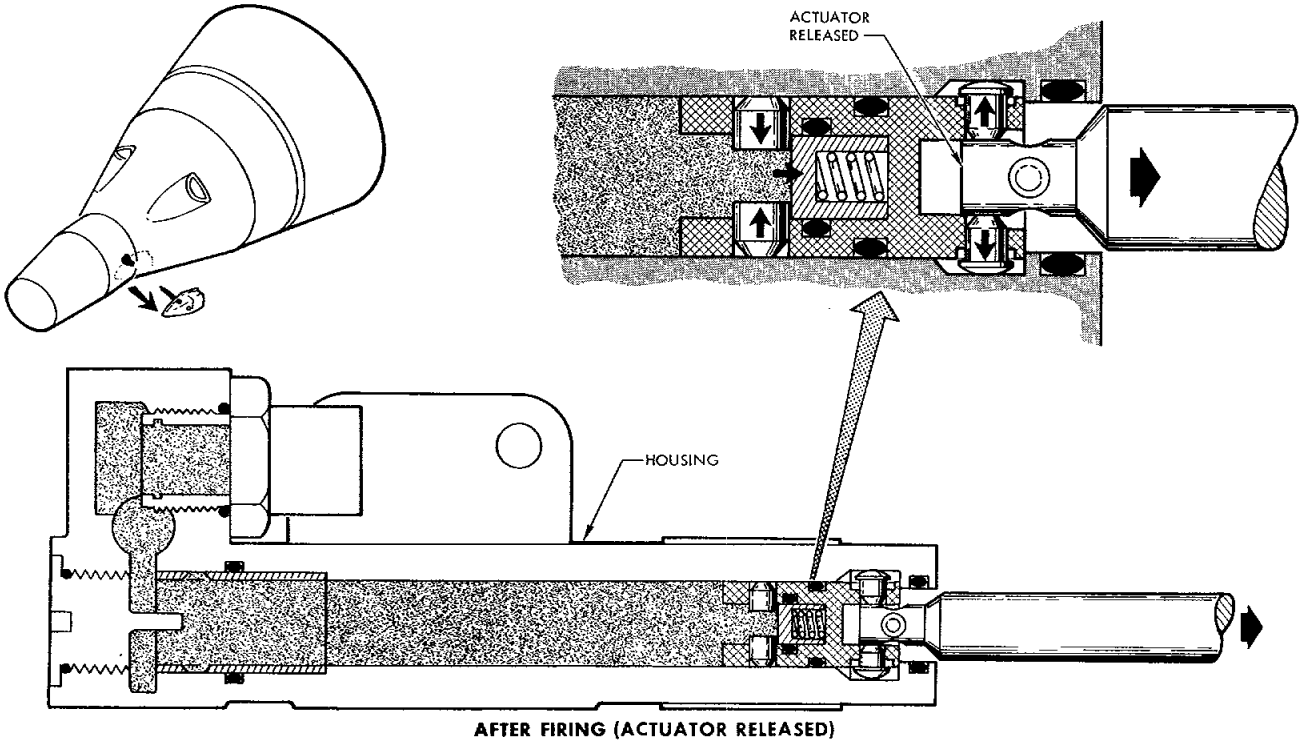
The horizon scanner fairing release assembly (Figure 11-12) secures the horizon scanner fairing to the spacecraft, and when initiated, jettisons the fairing. The assembly primarily consists of the actuator housing, actuator, actuator extension, main piston, release piston, eight locking pins and two electrically fired gas pressure cartridges. The actuator extension forms a positive tie between the actuator and the scanner fairing. The actuator is locked to the main piston by four locking pins. The main piston is locked in the base of the actuator housing by four locking pins, that are held in place by the release piston. The release piston is spring energized in the locked position. The actuator housing provides for installation of the cartridges and mounting for the assembly.

Operation

When initiated by a 28 vdc electrical signal, the cartridges fire and produce gas pressure. The pressure is ported through a milled passage in the actuator housing, to the base of the piston. The gas pressure moves the release piston forward, which enables the four locking pins to cam inboard, releasing the main piston. The gas pressure causes the main piston, with attached actuator, to move through the length of the actuator housing. As the piston reaches the end of the housing, a shoulder stops the piston travel. The four locking pins, securing the actuator extension to the piston, cam outboard into a recess and release the actuator extension. The actuator extension being thus freed is jettisoned with the scanner fairing attached.



BEFORE FIRING (ACTUATOR LOCKED)



AFTER FIRING (ACTUATOR RELEASED)

Figure 11-12 Horizon Scanner Fairing Release Assembly

HORIZON SCANNER RELEASE ASSEMBLY

Description

The horizon scanner release assembly (Figure 11-13) secures the horizon scanners to the spacecraft and jettisons the scanners when initiated. The horizon scanner release assembly primarily consists of the actuator housing, actuator, locking mechanism, cartridge housing, and two electrically fired gas pressure cartridges. The actuator is secured in the actuator housing by the locking mechanism. The locking mechanism consists of a tang lock, tang lock retainer and a shear pin. The tang lock is secured to and is located in the base of the actuator housing. The actuator housing is attached to and becomes a part of the spacecraft structure. The scanner base support and mounting platform are attached to the actuator prior to installing the cartridge housing on the actuator. The two cartridges are installed in the cartridge housing.

Operation

When initiated by a 28 vdc electrical signal, the two cartridges fire and produce gas pressure which is ported through the hollow actuator to the base of the actuator housing. Slots in the tang lock allow the gas pressure to flow to the base of the tang lock retainer. The gas pressure exerts a force against the base of the retainer. The retainer moves axially in the actuator housing, severing the shear pin and exposing the tines of the tang lock. The tines cam open, releasing the actuator and allowing the gas pressure to jettison the actuator and horizon scanners.

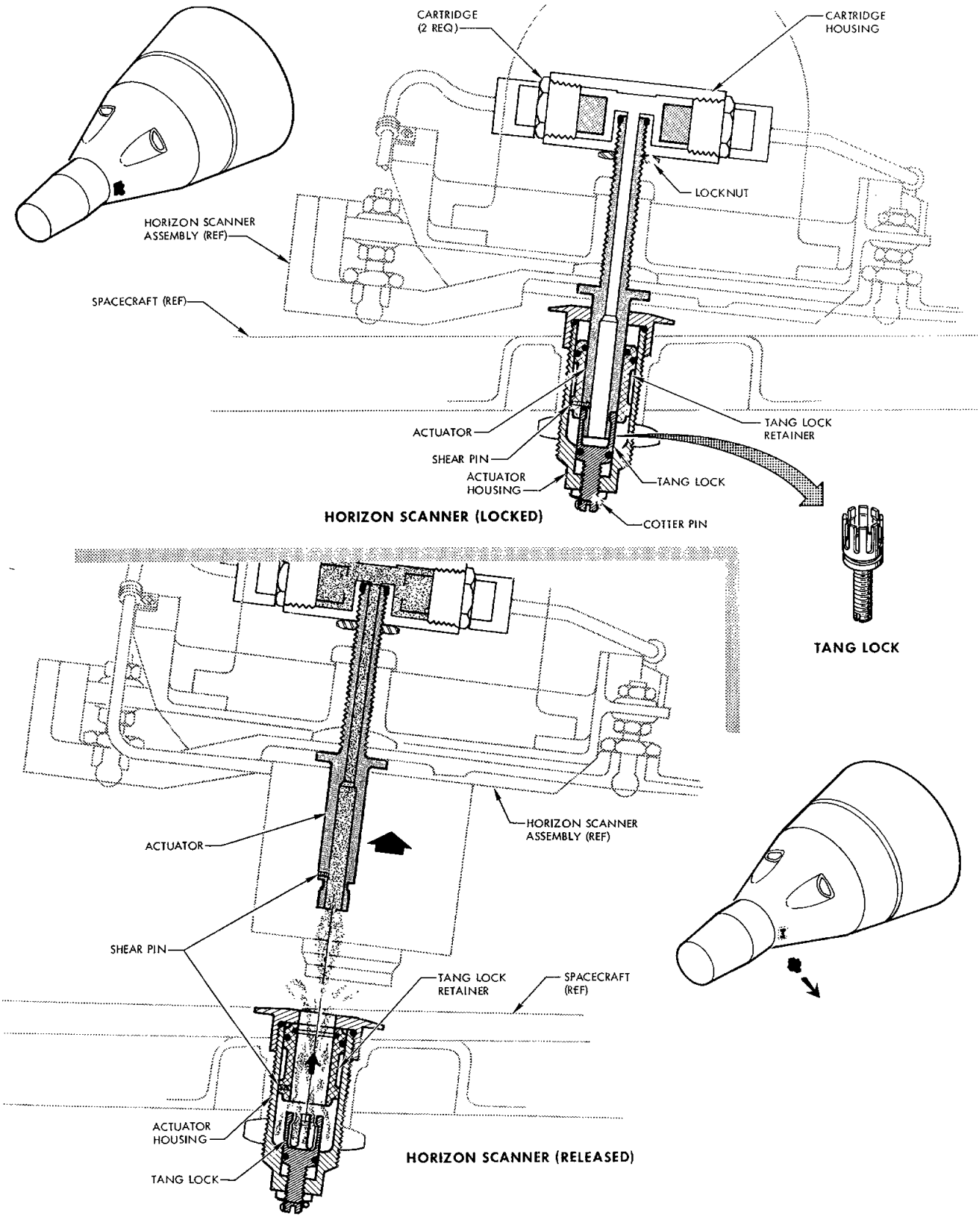


Figure 11-13 Horizon Scanner Release Assembly

FRESH AIR DOOR ACTUATOR

Description

The fresh air door actuator (Figure 11-14) is provided to retain the fresh air door to the spacecraft and to eject the door when initiated. The fresh air door actuator is located forward of the egress hatches, to the left of the spacecraft centerline and below the outer mold line. The actuator primarily consists of the breech, plunger, screw and two electrically fired gas pressure cartridges. The plunger forms a positive tie between the fresh air door and the breech. The plunger is retained in the breech by the screw which acts as a shear pin. The breech provides for installation of the two cartridges. Lugs, for attaching the actuator to the spacecraft structure, are an integral part of the breech.

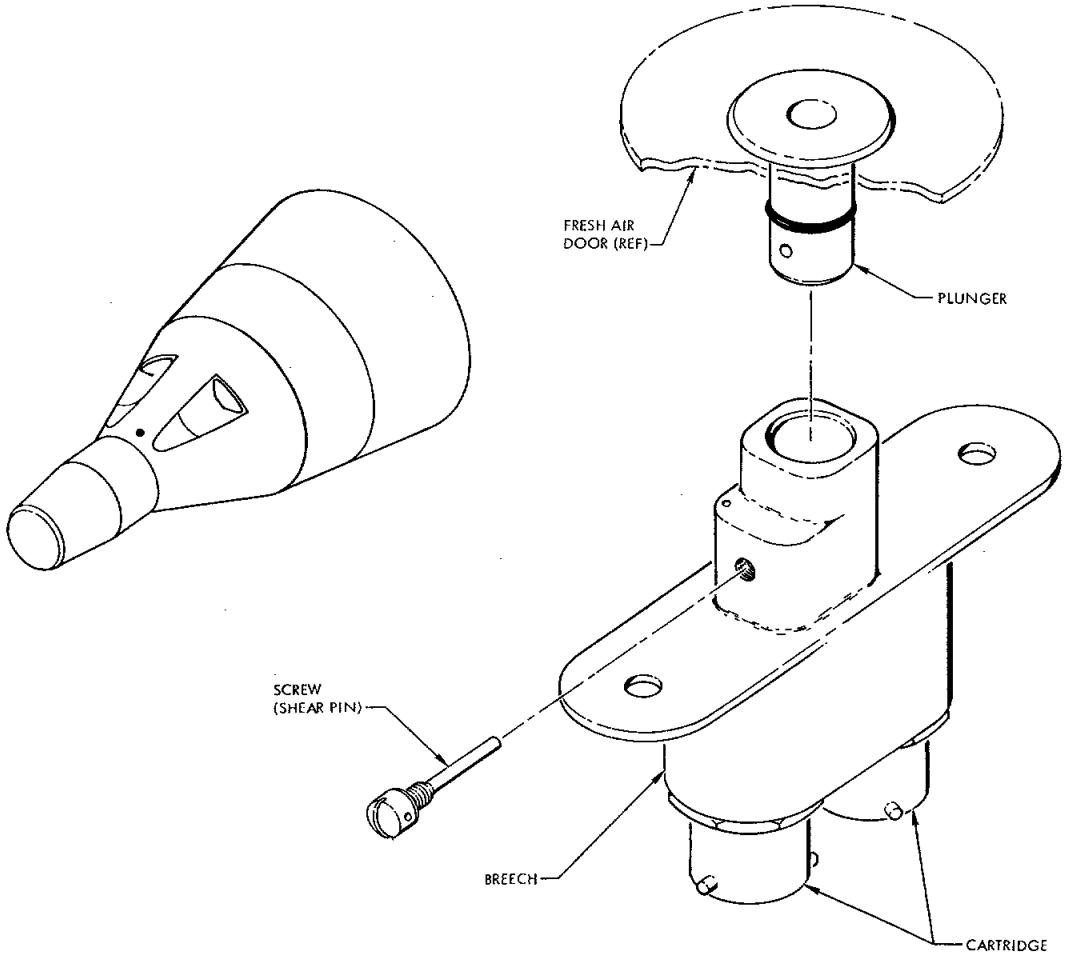
Operation

When initiated by a 28 vdc electrical signal, the cartridges are caused to fire and generate gas pressure that exerts a force on the plunger. When sufficient force is applied, the plunger severs the screw and is ejected out of the breech. The plunger and fresh air door are then jettisoned free of the spacecraft.

NOSE FAIRING EJECTOR

Description

The nose fairing ejector (Figure 11-15) is used to secure the rendezvous and recovery nose fairing to the spacecraft until initiated by a 28 vdc signal. When initiated the pyrotechnic ejector will positively jettison the nose fairing. The nose fairing ejector assembly consists of a breech, ballistic hose, actuator assembly, crank assembly, and an electrically fired gas pressure cartridge. The nose fairing is attached to the crank assembly. An actuator shaft forms a



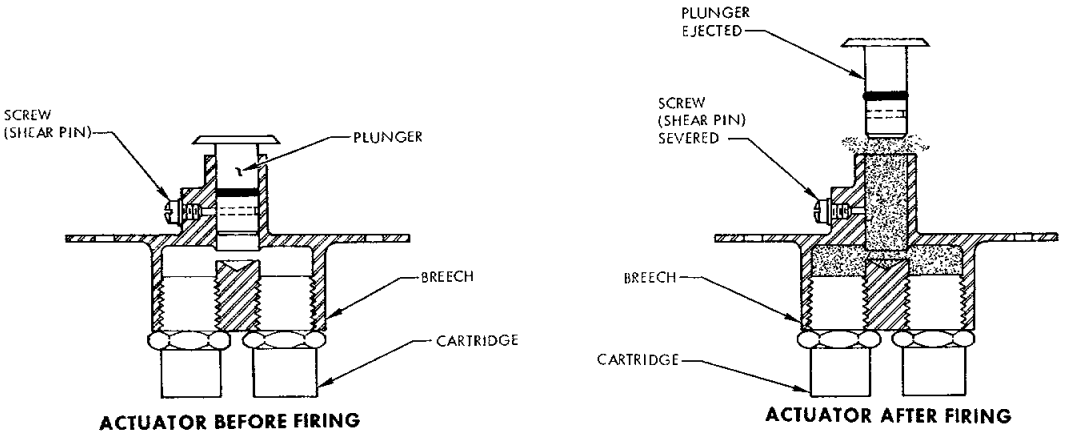
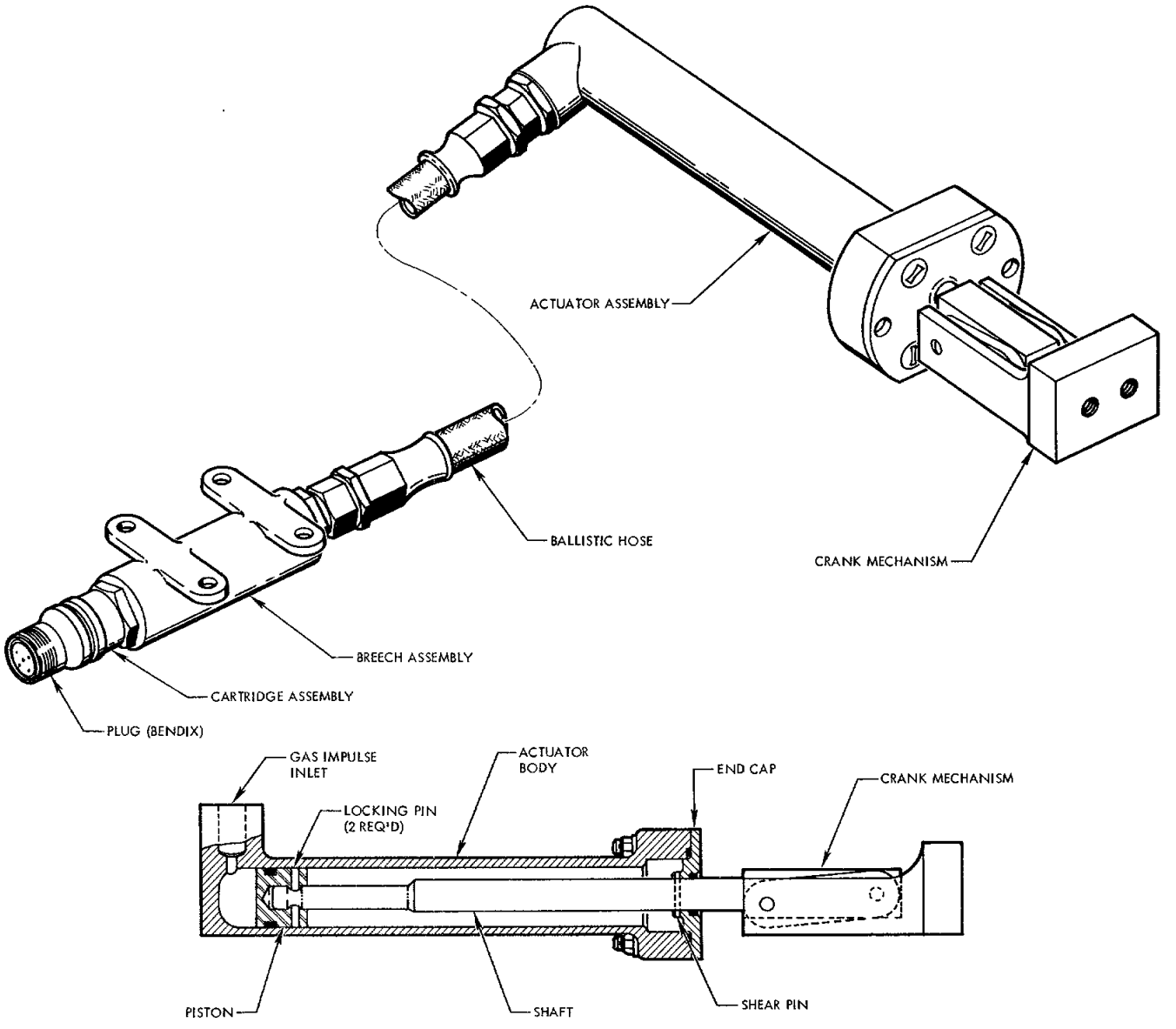
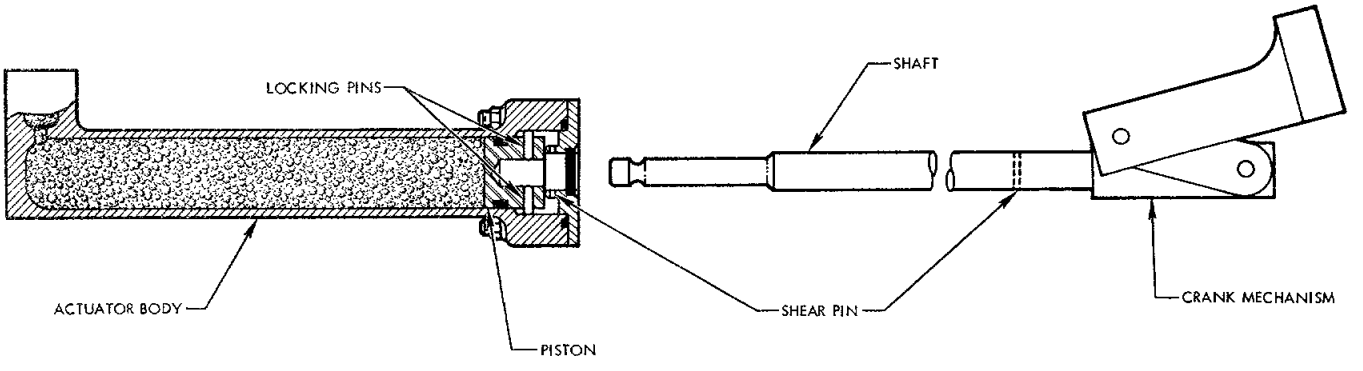


Figure 11-14 Fresh Air Door Actuator



NOSE FAIRING INSTALLED



NOSE FAIRING EJECTED

Figure 11-15 Nose Fairing Ejector Assembly

positive tie between the actuator body and the crank assembly. The actuator shaft is locked to a piston in the actuator by two locking pins and held in place by a shear pin in the end cap of the actuator. The actuator assembly is connected to a breech by a ballistic hose. The breech provides for installation of the cartridges and is positioned approximately nine inches from the actuator. The actuator is installed on the antenna support and fairing actuator fitting of the R and R section and is located on the X axis, five inches from Y zero.

Operation

When initiated by a 28 vdc signal, the cartridge generates gas pressure which is transferred through a ballistic hose to the actuator housing and exerts a force on the actuator piston. The gas pressure causes the piston, with attached shaft, to move, severing the shear pin and continuing through the length of the actuator housing. As the piston reaches the end of the housing, the two locking pins, securing the shaft to the piston, cam outboard into a recess and release the actuator shaft: The actuator shaft, now free, is jettisoned, with the nose fairing attached, by the crank mechanism. The crank mechanism provides an angled jettisoning of the fairing from the axial movement of the ejector shaft, without recontact with the spacecraft. A hinge on the nose fairing, located on the outer mold line, releases and directs the path of the fairing away from the spacecraft.

EGRESS SYSTEMS AND DEVICES

The egress systems and devices (Figure 11-16) provide the pilots with a rapid and positive method of escaping the spacecraft, should an emergency arise. The system is manually initiated and is used below an altitude of 15,000 feet only. Each system and device is presented in the sequence of their operation

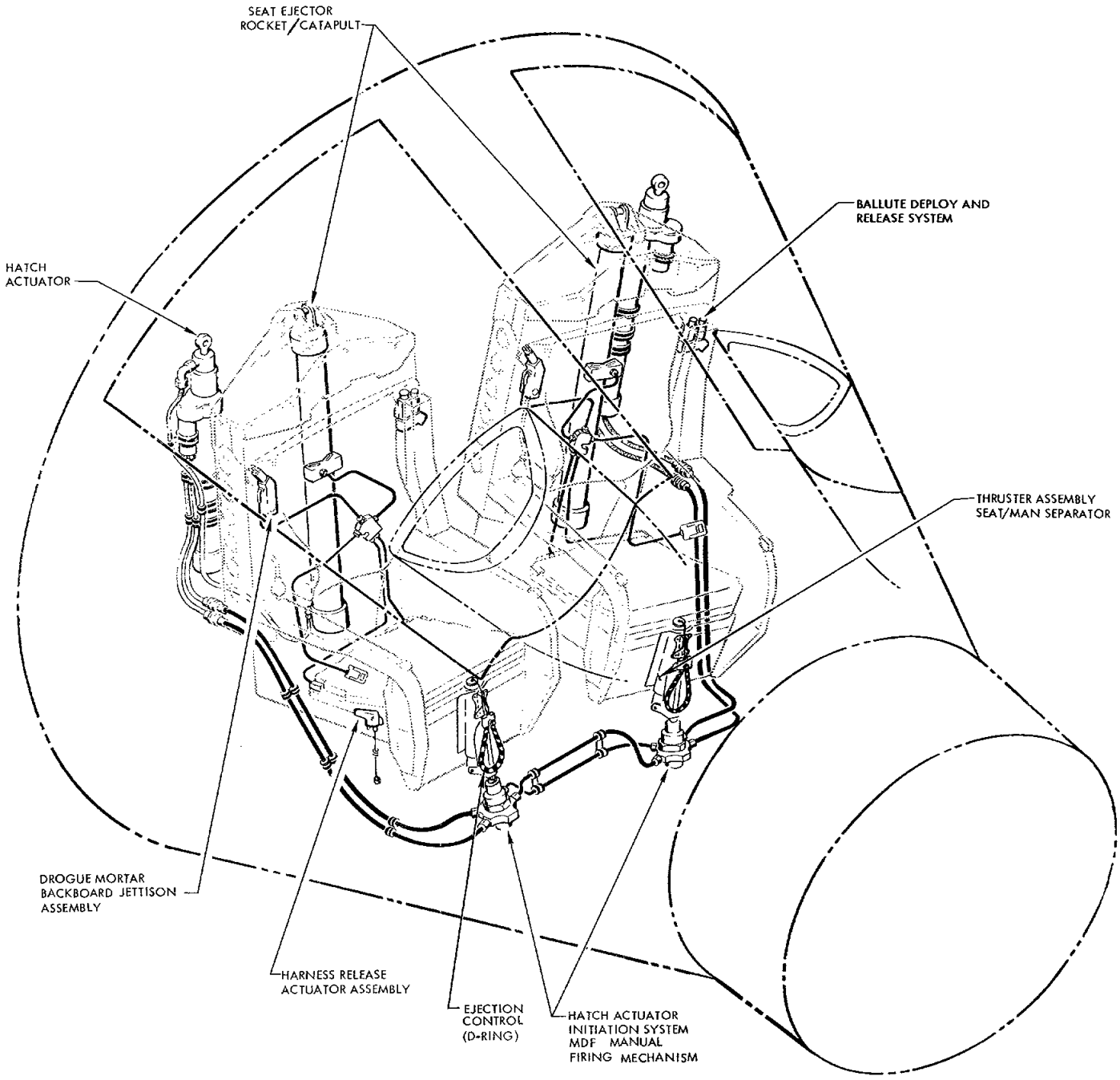


Figure 11-16 Egress System and Devices

HATCH ACTUATOR INITIATION SYSTEM (MDF)

Description

The hatch actuator initiation system (Figure 11-12) is used to initiate the firing mechanisms of both hatch actuators. The system is manually activated by either pilot. The system primarily consists of 8 MDF interconnects, two MDF crossovers and two manual firing mechanisms. The interconnects consists of four rigid and four flexible MDF assemblies that connect the firing mechanisms to the hatch actuators. The two crossovers are rigid MDF assemblies that cross connect the two initiation system firing mechanisms. The firing mechanisms each contain dual firing pins, dual percussion primers, and a booster charge. The firing mechanism is drilled and tapped for installing two MDF interconnects and two crossovers. The MDF interconnects and crossovers are installed so that the small booster on the end of each MDF is adjacent to the booster charge of the firing mechanism. The firing mechanism is attached to the spacecraft structure, located below the pilots feet.

Operation

The hatch actuator initiation system is activated when either pilot pulls the ejection control handle (D-ring) located between the pilots knees and connected to the firing mechanism. Approximately one-half inch travel and approximately a 40 pound pull of the lanyard connecting the ejection control (D-ring) to the firing mechanism will cock and release the dual firing pins. The firing pins strike the dual percussion primers, causing the booster charge to detonate. The firing mechanism booster charge propagates detonation to the four MDF ends. The interconnecting MDF propagates the detonation wave to the firing pins of the hatch actuator breech assembly. The crossover MDF propagates the detonation wave to the other pilots firing mechanism. This insures initiation of both hatch actuators.

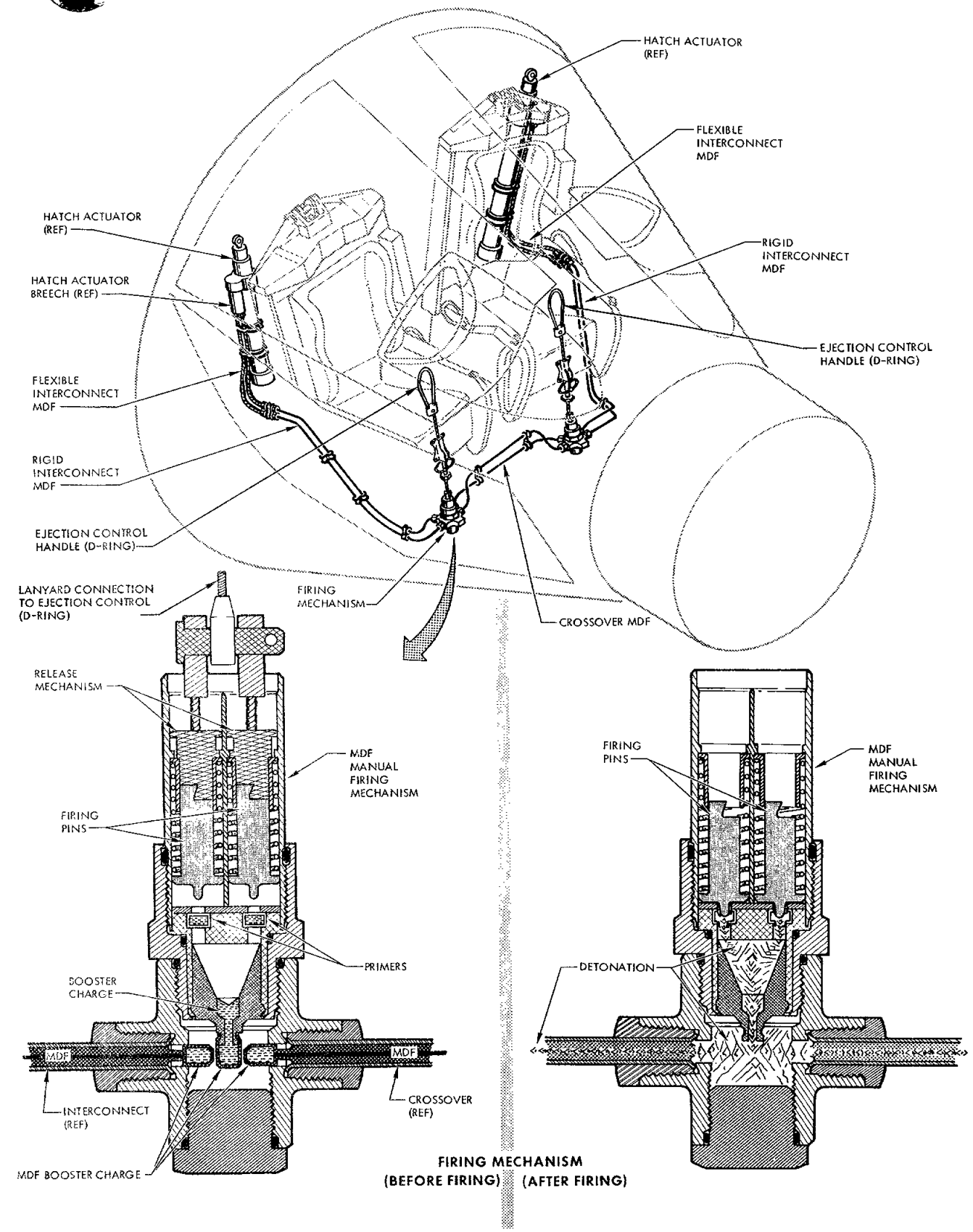


Figure 11-17 Hatch Actuator Initiation System

HATCH ACTUATOR ASSEMBLY

Description

The hatch actuator assembly (Figure 11-18) unlocks, opens and mechanically restrains the hatch in the open position. The assembly also furnishes sufficient pressure to initiate the firing mechanism of the seat ejector rocket/catapult. The assembly primarily consists of the breech end cap, breech, cylinder, stretcher assembly, end cap (base) and rod end assembly. The breech end cap assembly contains the locking mechanism for mechanically restraining the hatch in the open position; provides for installation of the seat ejector rocket/catapult ballistic hose; provides for installation of the breech assembly, and is thread mounted to the top of the cylinder. The breech contains two firing pins, two percussion fired cartridges, and a gas producing propellant charge. Two interconnects, from the hatch actuator initiation system, are attached to the breech adjacent to the firing pins. The stretcher assembly primarily consists of the piston and stretch link, and is located inside the cylinder. One end of the stretch link is attached to a web inside the piston. The other end is attached to the rod end assembly. The rod end assembly connects the stretcher assembly to the hatch. The end cap is attached to the lower end of the cylinder, and provides for attaching the hatch actuator assembly to the spacecraft structure. The end cap contains a latch piston that actuates the hatch unlock mechanism.

Operation

The hatch actuator functions when initiated by the initiation system MDF interconnects. The shock wave, propagated by the MDF interconnects, causes the two

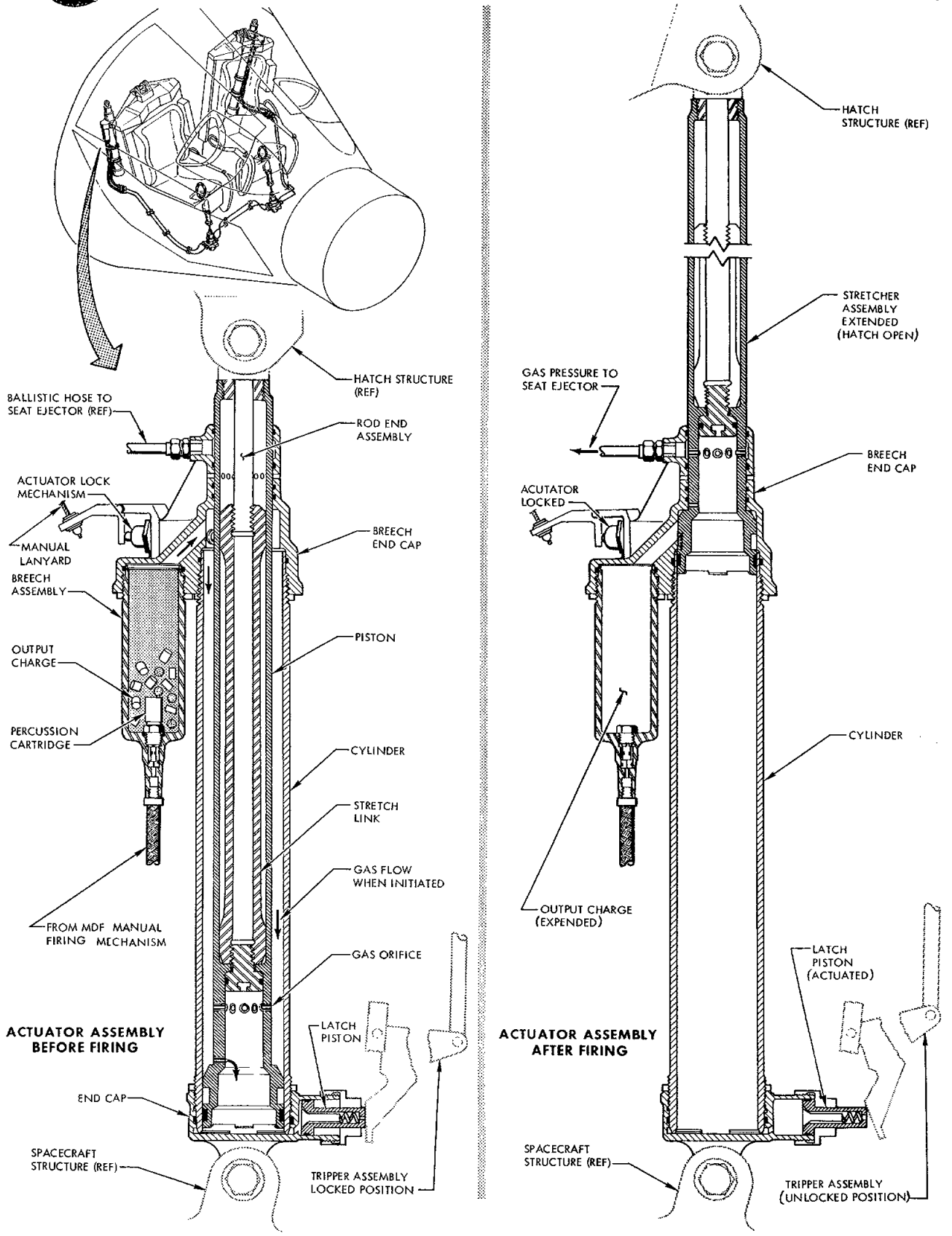


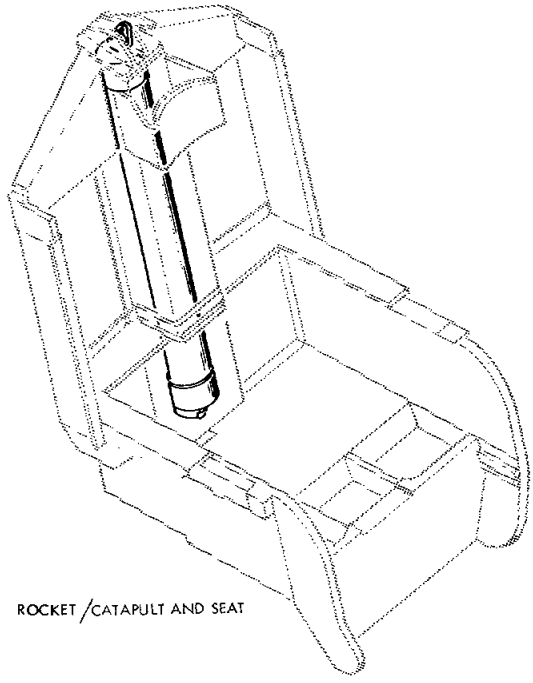
Figure 11-18 Hatch Actuator Assembly

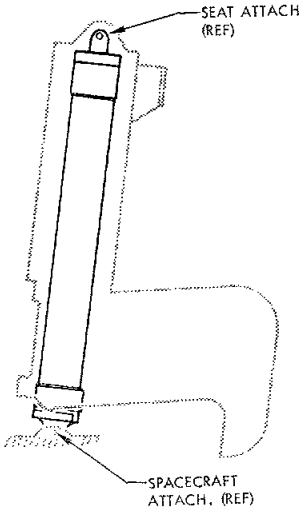
firing pins of the breech assembly to sever shear pins and strike the primers of the two percussion fired cartridges. The cartridges ignite and generate hot gas which ignites the main propellant charge of the breech. The propellant charge produces a large volume of high pressure gas. The gas pressure is exhausted into the area between the piston of the stretcher assembly and the cylinder. Orifices in the lower end of the piston wall admit the gas pressure to the base of the stretcher assembly. The gas pressure is ported through a drilled passage to the latch piston. The gas pressure extends the latch piston, which unlocks the hatch through a bellcrank/pushrod mechanism. The gas pressure then acts on the base of the stretcher assembly, moving it through the length of the cylinder. Immediately prior to the stretcher assembly reaching full extension, gas pressure is exhausted through a port to the ballistic hose. The ballistic hose delivers the pressure to the firing mechanism of the seat ejector-rocket/catapult. As the stretcher assembly reaches the fully extended position, the lock pin of the locking mechanism engages the piston of the stretcher assembly and holds the hatch open. The locking mechanism is also operative when the hatch is fully opened by hand. A lanyard, attached to the locking mechanism, permits the hatch to be unlocked, when manually actuated.

SEAT EJECTOR-ROCKET/CATAPULT

Description

The seat ejector-rocket/catapult (Figure 11-19) is used to eject the man-seat mass from the spacecraft. The seat ejector-rocket/catapult basically consists of the catapult assembly and the rocket motor assembly. The catapult assembly primarily consists of the catapult housing, firing mechanism, main charge (gas producer), and locking assembly. The catapult housing contains all of the listed components





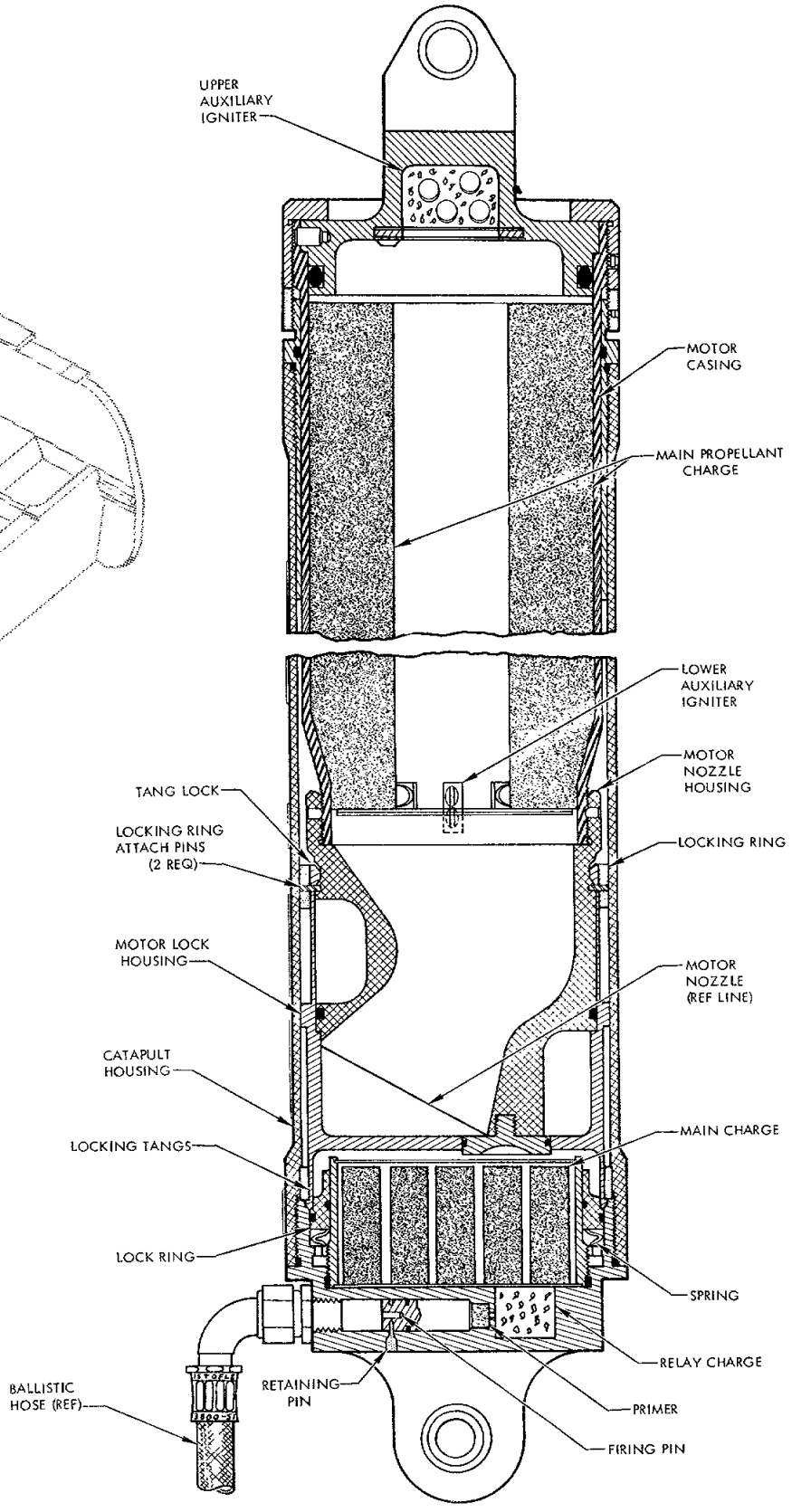


Figure 11-19 Seat Ejector - Rocket/Catapult

in its base. The firing mechanism consists of dual firing pins, dual percussion fired primers, and a relay charge. The firing pins are secured in place by retaining pins. The locking assembly consists of the lock ring and a spring to hold the ring in place. The base of the catapult assembly is attached to the spacecraft structure. The rocket motor assembly primarily consists of the motor case, nozzle, motor lock housing, lock ring, shear pins, upper and lower auxiliary igniters, and the main propellant charge. The nozzle is threaded to the motor case and is secured by four set screws. The nozzle is secured to the motor lock housing by locking tangs. The locking tangs are held in place by a lock ring that is retained by four shear pins. The motor lock housing is secured in the base of the catapult by tang locks. The tangs are held in place by the lock ring of the catapult. The main propellant charge is located in the motor case with an auxiliary igniter at each end of the charge. The top end of the rocket motor assembly is attached to the upper aft portion of the seat.

Operation

The seat ejection cycle is initiated when gas pressure is received via the ballistic hose from the hatch actuator. Sufficient gas pressure will cause the dual firing pins to shear their retaining pins and strike the dual percussion primers. The primers ignite the relay and main charges. Hot gas pressure, produced by the main charge, releases the motor lock housing by displacing the lock ring against the spring through piston action. With the motor lock housing released, the gas pressure propels the rocket motor through the length of the catapult housing. Prior to complete ejection from the catapult housing, the lock ring of the motor lock housing makes contact with a stop which severs its four shear pins. The tang locks of the motor lock housing cam open and release the rocket motor. Separation of the rocket motor from the motor lock housing allows the hot gas from the

catapult main charge to ballistic hose to initiate the thruster assembly.

THRUSTER ASSEMBLY-SEAT/MAN SEPARATOR

Description

The thruster assembly - seat man separator (Figure 11-21) is the active portion of the seat/man separation assembly. The thruster supplies a stroke of adequate length and power to a webbed strap that accomplishes seat/man separation. The thruster assembly primarily consists of the thruster body, thruster piston, firing mechanism and percussion fired gas pressure cartridge. The cartridge and firing mechanism is installed in the upper end of the thruster body. The firing mechanism contains a firing pin, retained by a shear pin. The ballistic hose from the harness release actuator is attached to the firing mechanism. The thruster piston is located in the thruster body and is retained in the retracted position by a shear pin. The thruster body is mounted on the front of the seat structure, between the pilots feet.

Operation

High pressure gas from the harness release actuator is transmitted through the ballistic hose to the thruster firing mechanism. The gas pressure causes the firing pin to sever its shear pin and strike the primer of the cartridge. The cartridge is ignited and generates gas pressure. The gas pressure exerts force on the thruster piston, causing the piston to sever its shear pin. As the piston extends out of the thruster body, the strap is pulled taut effecting seat/man separation.

BALLUTE DEPLOY AND RELEASE SYSTEM

Description

The ballute deploy and release system (Figure 11-22) primarily consists of the

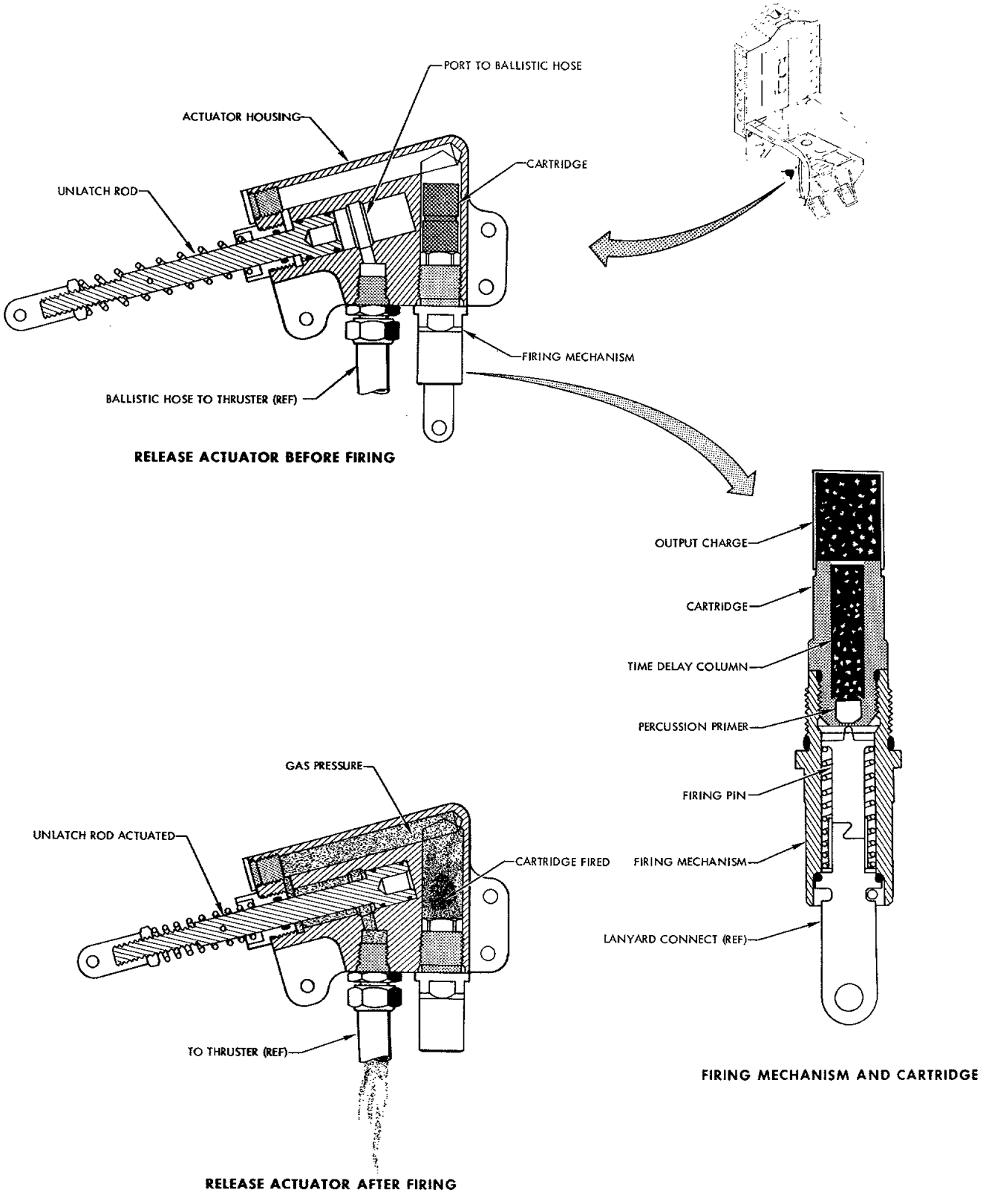


Figure 11-20 Harness Release Actuator Assembly

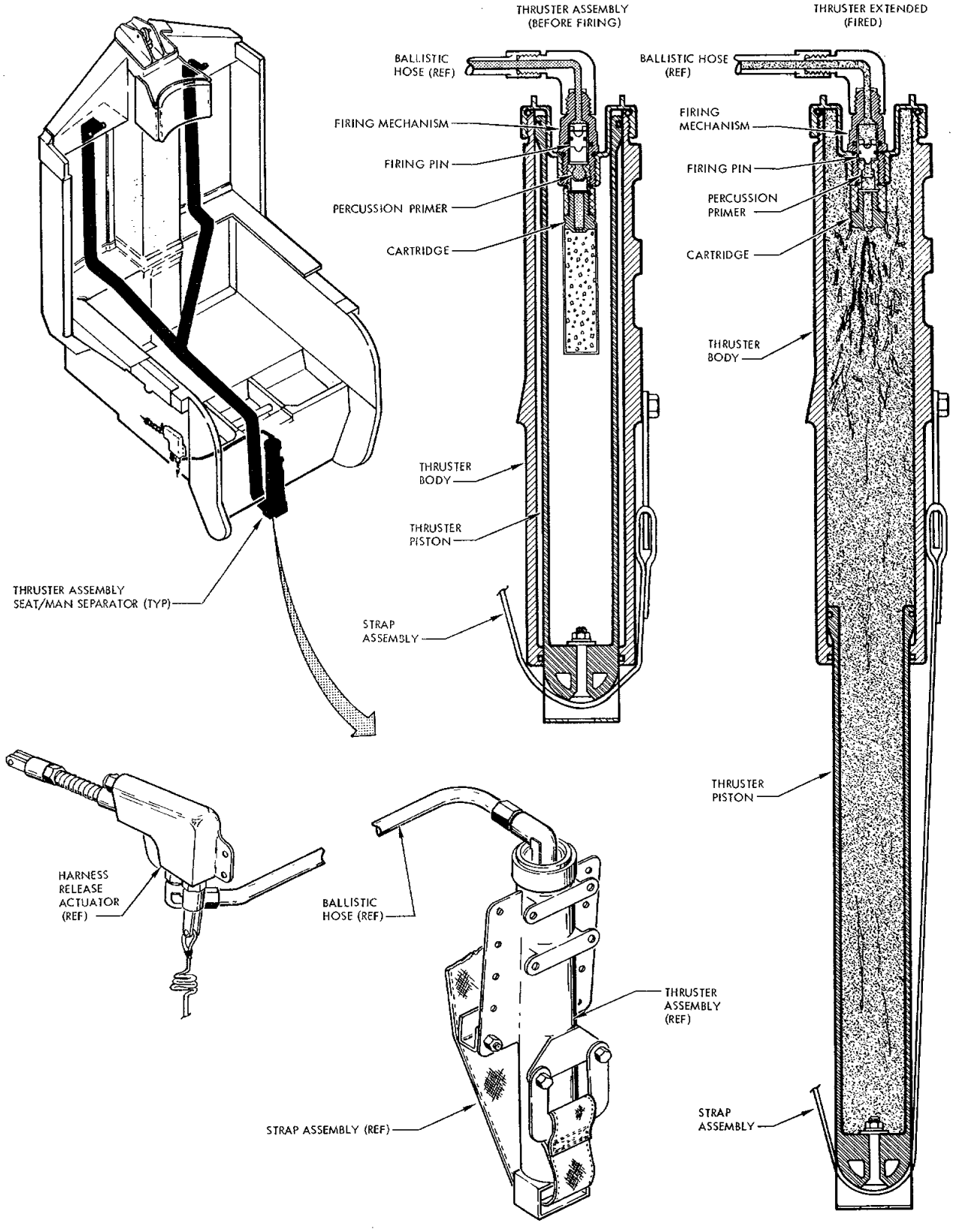


Figure 11-21 Thruster Assembly-Seat/Man Separator

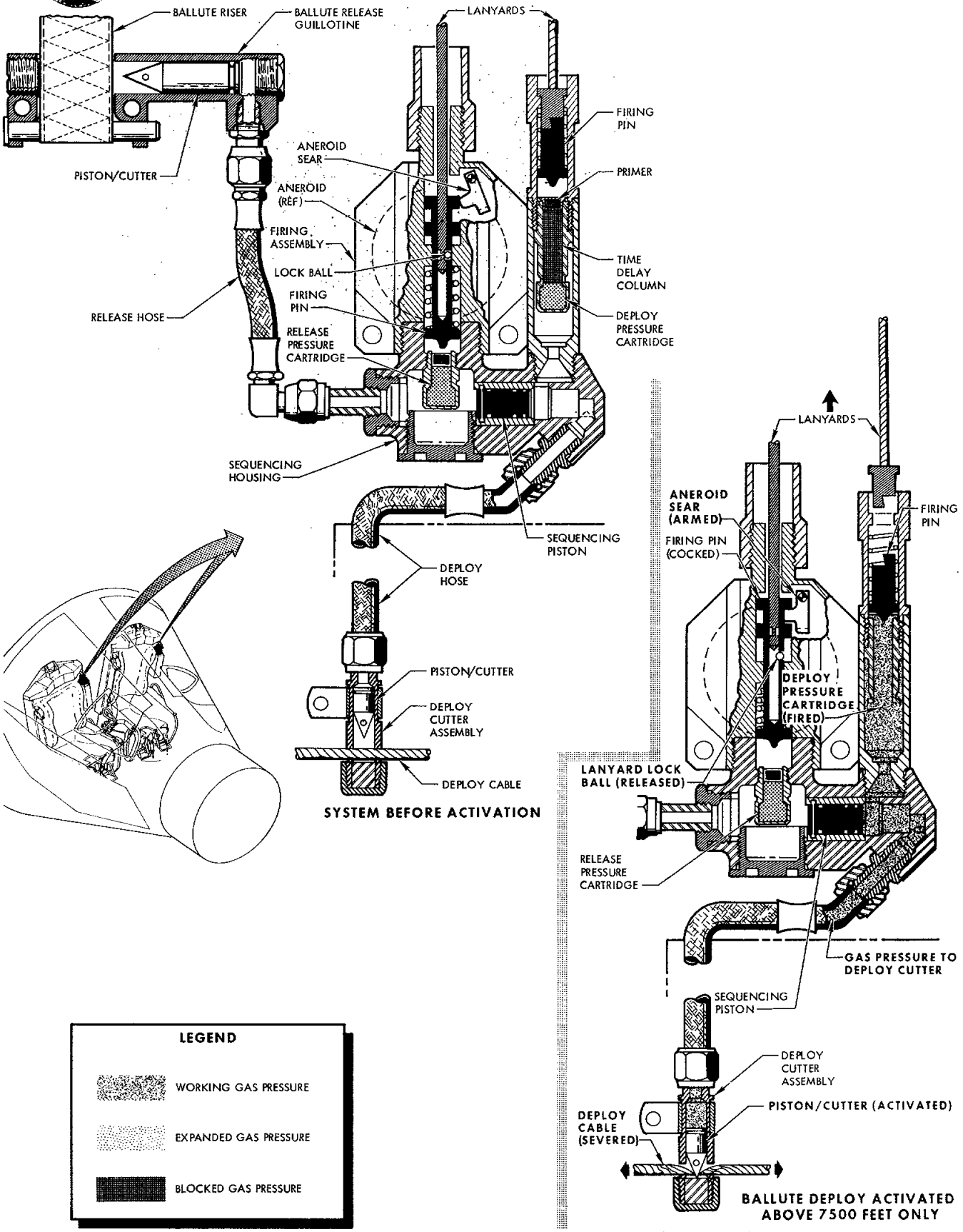


Figure 11-22 Ballute Deploy and Release System (Sheet 1 of 2)

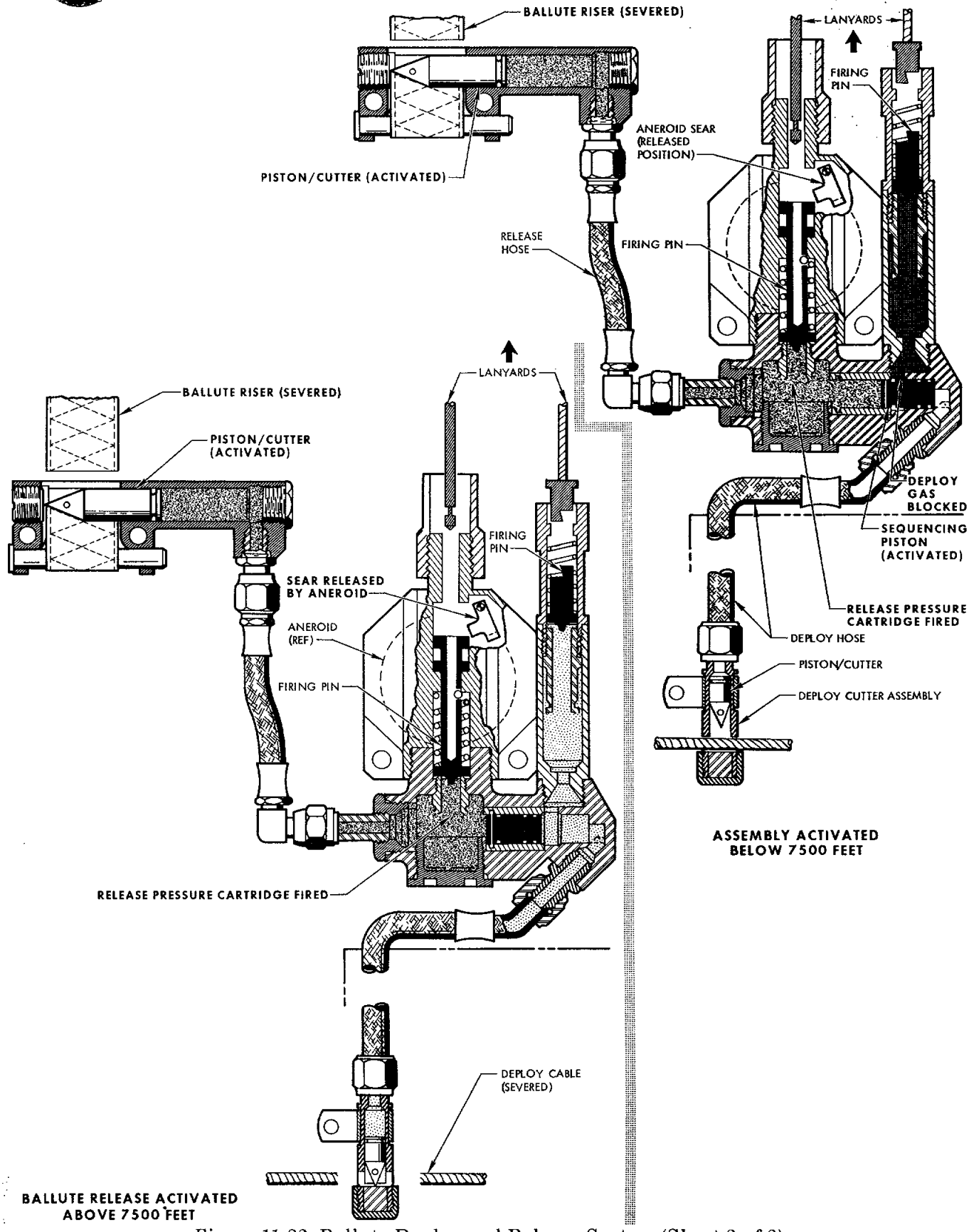


Figure 11-22 Ballute Deploy and Release System (Sheet 2 of 2)

firing assembly, deploy cutter and hose, and release guillotine and hose. Contained within the firing assembly, is the release aneroid firing mechanism and cartridge, the deploy firing mechanism and cartridge, and the sequencing housing and piston. The basic function of the system is to deploy and release the ballute between specified altitudes and prevent ballute deployment below specified altitudes. The system is located on the upper left side of each pilots backboard. The deploy firing mechanism and the release aneroid firing mechanism is linked to the pilots seat by individual lanyards.

Operation

The system is initiated by the lanyard pull as seat/man separation is effected. When initiated above 7500 feet, the release aneroid is armed and the deploy firing mechanism is activated. The firing pin of the deploy firing mechanism strikes the primer of the cartridge and causes ignition. The cartridge generates gas pressure after burning through the time delay column. The pressure is ported through the deploy hose to the deploy cutter assembly. The cutter severs a nylon strap that allows the ballute to deploy. The armed aneroid functions when an altitude pressure level of 7500 feet is reached. The aneroid sear releases the cocked firing pin of the ballute release firing mechanism. The firing pin strikes the primer, which ignites the cartridge and causes it to generate gas pressure. The pressure is ported through the release hose to the release guillotine. The guillotine severs the ballute riser strap and allows the ballute to be carried away. When the system is activated by the lanyard pull below 7500 feet, both cartridges are immediately initiated. The time delay incorporated in the deploy cartridge permits the release cartridge to generate gas pressure first. The pressure is ported through the release hose to the release guillotine, which severs the ballute riser

strap. Simultaneously gas pressure is ported to the sequencing housing and sequencing piston. The piston is actuated, causing it to block the gas exit of the deploy cartridge. The gas pressure, generated by the deploy cartridge, does not reach the deploy cutter, preventing deployment of the ballute.

DROGUE MORTAR-BACKBOARD JETTISON ASSEMBLY

The drogue mortar-backboard jettison assembly is provided to deploy the personnel drogue parachute and to separate the backboard and seat from the pilot.

Description

Drogue Mortar

The drogue mortar (Figure 11-23) functions to fire a weighted slug with sufficient velocity to forcibly deploy the personnel parachute and to initiate the backboard jettison assembly firing mechanism. The drogue mortar primarily consists of the mortar body, mortar barrel, drogue slug, main cartridge (gas pressure), initiator cartridge (detonator), aneroid assembly, main lanyard, manual lanyard, and the main and manual firing mechanisms. The mortar barrel is threaded into the mortar body and contains the drogue slug. The drogue slug is retained in the barrel by a shear pin. The aneroid assembly is attached to the mortar body and contains the main firing mechanism. The main lanyard is enclosed in a rigid housing to prevent inadvertant pulling of the lanyard. The housing is attached to the main firing mechanism housing at one end and to a take-up reel at the other. The main lanyard, a fixed length of cable, is attached to the main firing mechanism at one end and to the take-up reel at the other. The take-up reel incorporates an extendable cable that is attached to the ejection seat. The main cartridge is threaded into the mortar body, with the primer end, adjacent to the main firing mechanism, and the output end in the mortar body pressure cavity. The manual

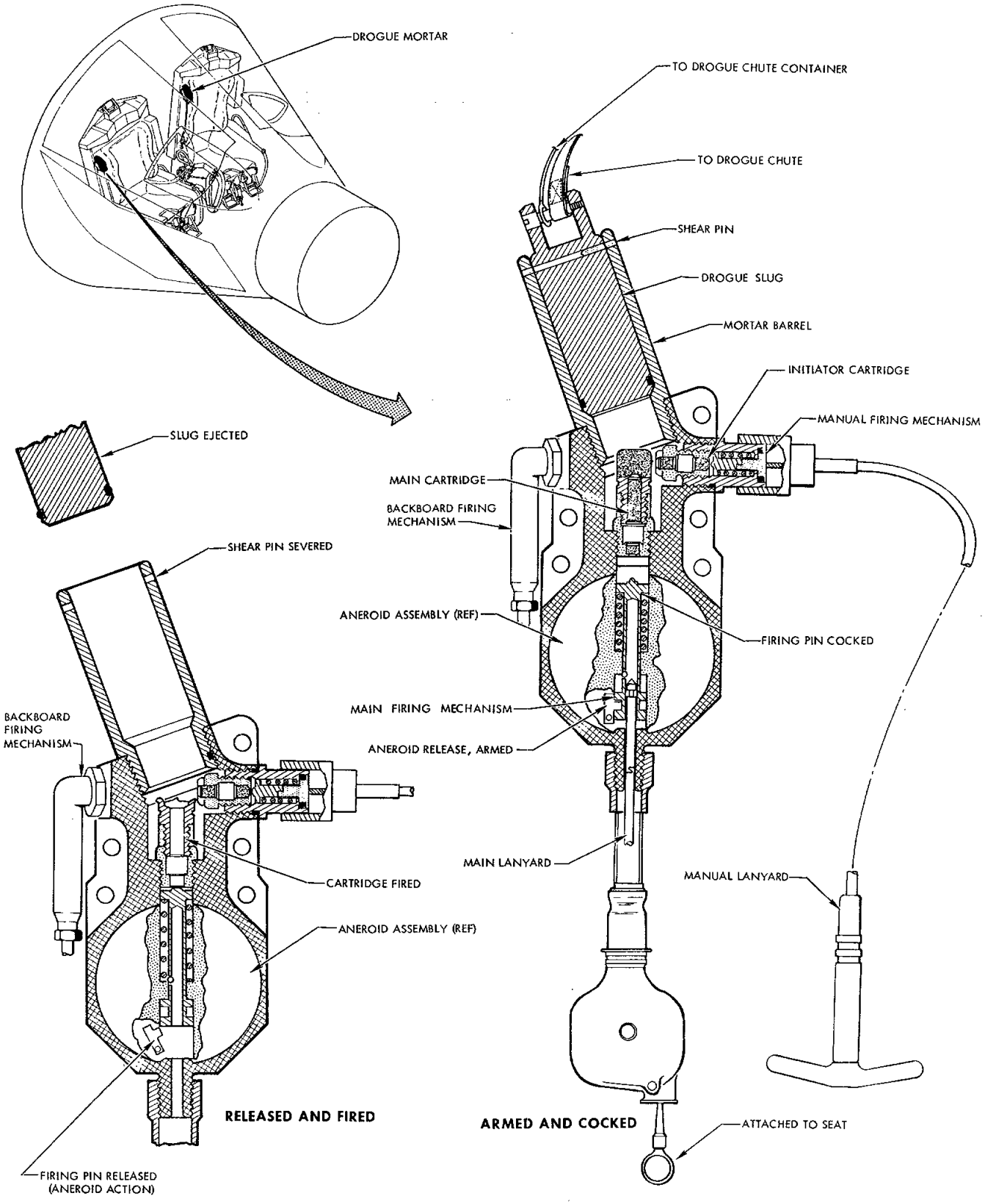


Figure 11-23 Drogue Mortar

lanyard is enclosed in a flexible conduit to prevent inadvertant pulling of the lanyard. The lanyard is attached to the manual firing mechanism at one end and to a manual pull handle at the other. The manual firing mechanism is threaded into the mortar body. The primer end of the detonator is threaded into the manual firing mechanism, and its output end 90 degrees and adjacent to the main cartridge output area. The drogue mortar is attached to the upper right side of each pilots backboard.

Backboard Jettison Assembly

The backboard jettison assembly (Figure 11-24), functions to separate the backboard and seat from the pilot, when initiated by the pressure from the drogue mortar. The backboard jettison assembly primarily consists of the MDF firing mechanism, MDF time delay cartridge (detonator), interconnect (time delay MDF), MDF manifold assembly, jetelox release pin, interconnect (jetelox pin MDF), lap belt disconnect, interconnect (belt disconnect MDF), restraint strap cutter (FLSC), and interconnect (strap cutter MDF). The MDF firing mechanism is attached to the drogue mortar body and contains a shear pin retained firing pin. The MDF time delay cartridge is a percussion fired cartridge and is installed in the MDF firing mechanism. The interconnect (time delay MDF) is connected to the MDF firing mechanism and the MDF manifold. The interconnect (jetelox pin MDF) is connected to the MDF manifold and the jetelox release pin. The interconnect (belt disconnect MDF) is connected to the MDF manifold and the lap belt disconnect. The interconnect (strap cutter MDF) is connected to the MDF manifold and the restraint strap cutter (FLSC). The three component interconnects terminate in the MDF manifold with their acceptor end adjacent to the interconnect (time delay MDF) donor end. The jetelox release pin retains the jetelox joint to the seat until

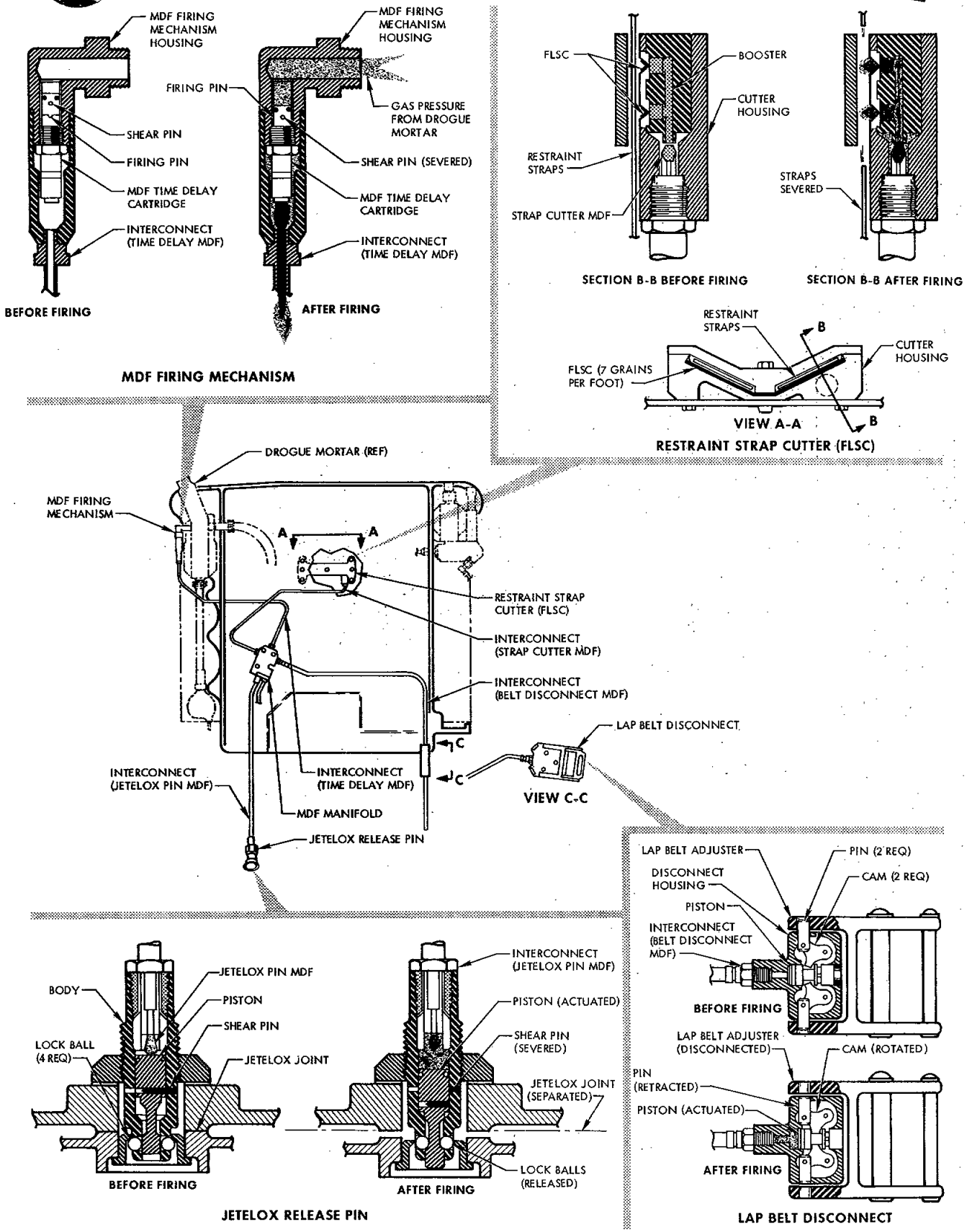
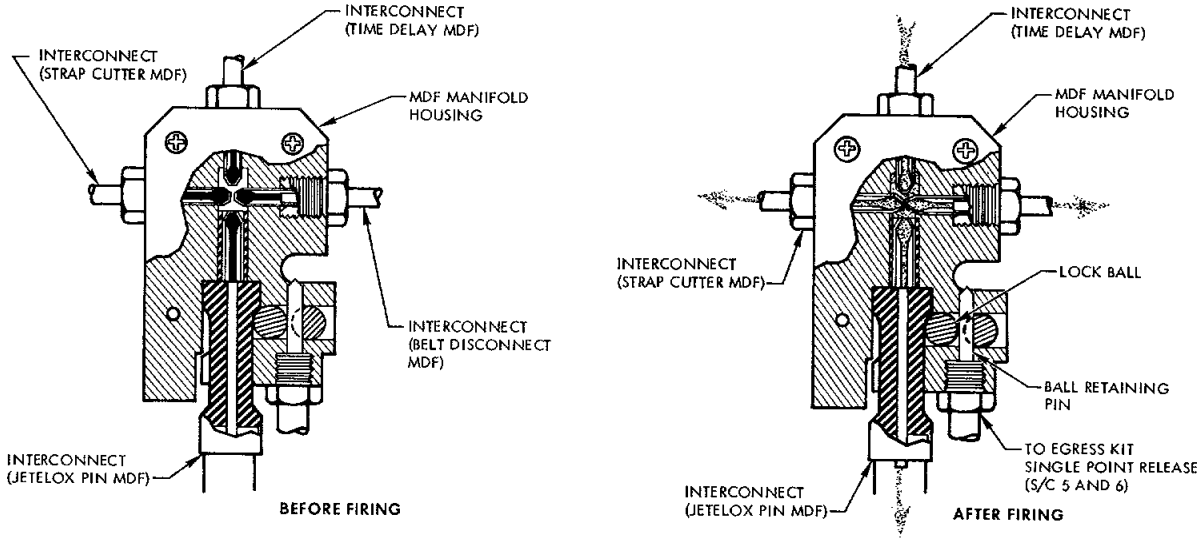


Figure 11-24 Backboard Jettison Assembly (Sheet 1 of 2)



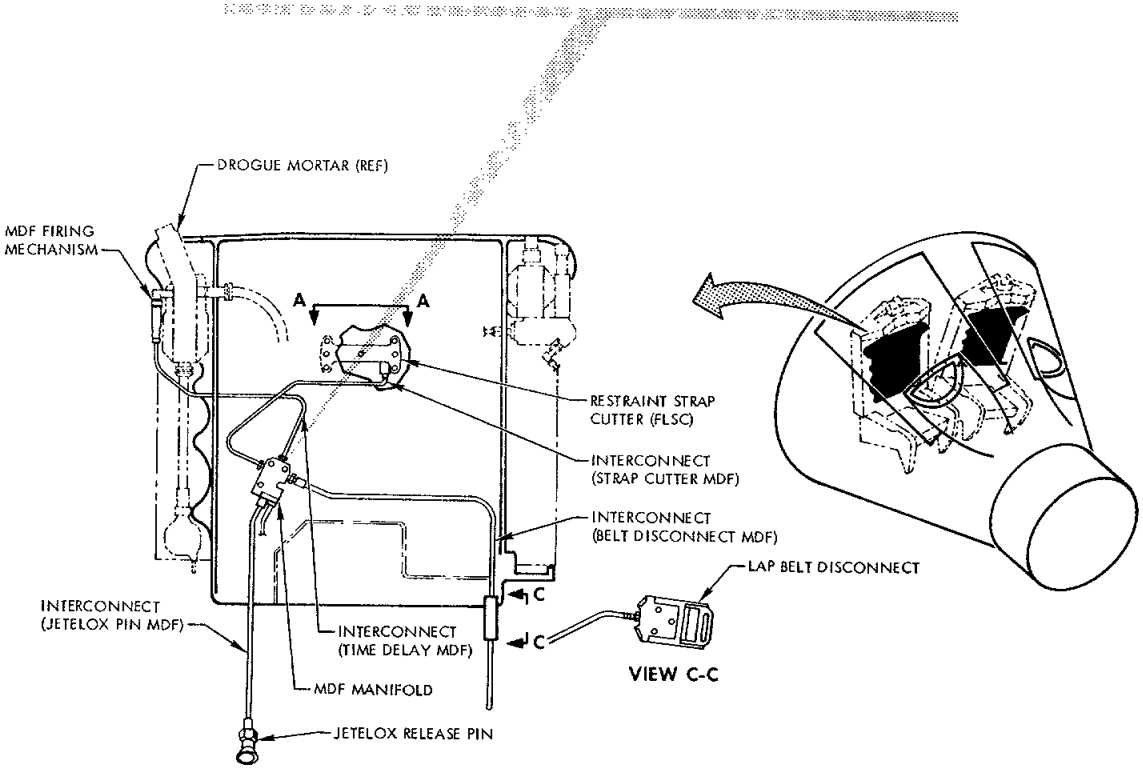


Figure 11-24 Backboard Jettison Assembly (Sheet 2 of 2)

initiated. The jetelox release pin primarily consists of the body, piston, four lock balls, and a shear pin. The lap belt disconnect is provided to unfasten the lap belt when properly initiated. The lap belt disconnect primarily consists of the housing, two lock pins, two cams, piston and a shear pin. The restraint strap cutter is provided to sever the pilots shoulder harness. The cutter primarily consists of the housing, two strips of FLSC and a booster.

Operation

Drogue Mortar

The drogue mortar is initiated by the pull of the main lanyard, at seat/man separation. The extendable cable, attached to the seat, uncoils from the take-up reel. Upon reaching the end of its travel, the cable pulls the take-up reel free of the rigid housing. The fixed length main lanyard attached to the reel is pulled, and if in excess of 5,700 feet, cocks the main firing mechanism and arms the aneroid. At an altitude pressure level of 5,700 feet, the aneroid releases the cocked main firing pin. The firing pin strikes the primer and ignites the main cartridge, which produces gas pressure. The gas pressure causes the drogue slug to sever its shear pin and travel out of the mortar barrel. Simultaneously, the gas pressure initiates the backboard firing mechanism. When initiated by the main lanyard below 5,700 feet, the main firing mechanism is cocked and immediately released to fire the main cartridge. The aneroid is in the release position because of the altitude pressure level, therefore is not armed and does not delay the cartridge firing. The drogue mortar may be initiated manually by pulling the manual lanyard handle at any altitude. The lanyard cocks and releases the manual firing pin, which strikes the primer of the initiator cartridge (detonator). The initiator cartridge detonates and ignites the output charge

of the main cartridge, which produces the gas pressure for drogue slug ejection and backboard firing mechanism initiation.

Backboard Jettison Assembly

The backboard jettison assembly is caused to function when the main cartridge of the drogue mortar is fired. Gas pressure from the drogue mortar main cartridge, causes the firing pin of the backboard firing mechanism, to sever its shear pin and strike the primer of the time delay cartridge. After the proper time delay, the cartridge propagates a detonation wave to the MDF interconnect, which transmits the wave simultaneously, to the three MDF interconnects attached to the MDF manifold assembly. Simultaneously, the detonation wave is propagated by the three MDF interconnects to the restraint strap cutter (FLSC), lap belt disconnect, and the jetelox release pin. The detonation wave propagated by the interconnect (jetelox pin MDF) acts upon the piston of the jetelox pin, causing it to sever the shear pin. As the piston moves, a recess in the piston is aligned with the lock balls. The pressure exerted by the jetelox joint, forces the lock balls into the piston recess, and releases the jetelox joint and egress kit. The detonation wave, propagated by the interconnect (belt disconnect MDF), is directed against the piston of the lap belt disconnect. The detonation wave moves the piston causing it to sever the shear pin. As the piston moves, the cams rotate and retract the pins from the lap belt adjuster. The lap belt separates and permits the pilot to be partially free of the backboard. The detonation wave, propagated by the interconnect (strap cutter MDF), is transmitted to the booster of the restraint strap cutter (FLSC). The booster strengthens and increases the reliability of the detonation wave for proper detonation of the two strips of FLSC. The FLSC detonates

and severs the two restraint straps allowing the pilot to be completely free of the backboard. The seat may be released manually by the pilot actuating the seat single point release handle. Effective spacecraft 5 and 6 a cable from the single point release, pulls a ball retaining pin from the MDF manifold. The pressure of the interconnect (jetelox pin MDF), moves the ball aside and pulls out of the MDF manifold.

PARACHUTE LANDING SYSTEM PYROTECHNICS

The Parachute Landing System (Figure 11-25) is provided to safely recover and land the re-entry module, after its entry into the earths atmosphere. The pyrotechnic portion of the system consists of the drogue, pilot, and main parachute reefing cutters; the drogue and pilot parachute mortars; the drogue parachute bridle release guillotines; the pilot parachute apex line guillotine; and the main parachute disconnects. Each of these pyrotechnic devices are presented in the following paragraphs.

DROGUE PARACHUTE MORTAR ASSEMBLY

The drogue parachute mortar assembly is provided to positively deploy the drogue chute. The assembly is similiar to the pilot parachute mortar assembly (Figure 11-25) in design and operation.

DROGUE PARACHUTE REEFING CUTTERS

The drogue parachute reefing cutters are provided to disreef the drogue chute. The cutters are similiar to the pilot parachute reefing cutters (Figure 11-26) in design, operation, and number.

DROGUE PARACHUTE BRIDLE RELEASE GUILLOTINES

The drogue parachute bridle release guillotines are provided to sever the bridle

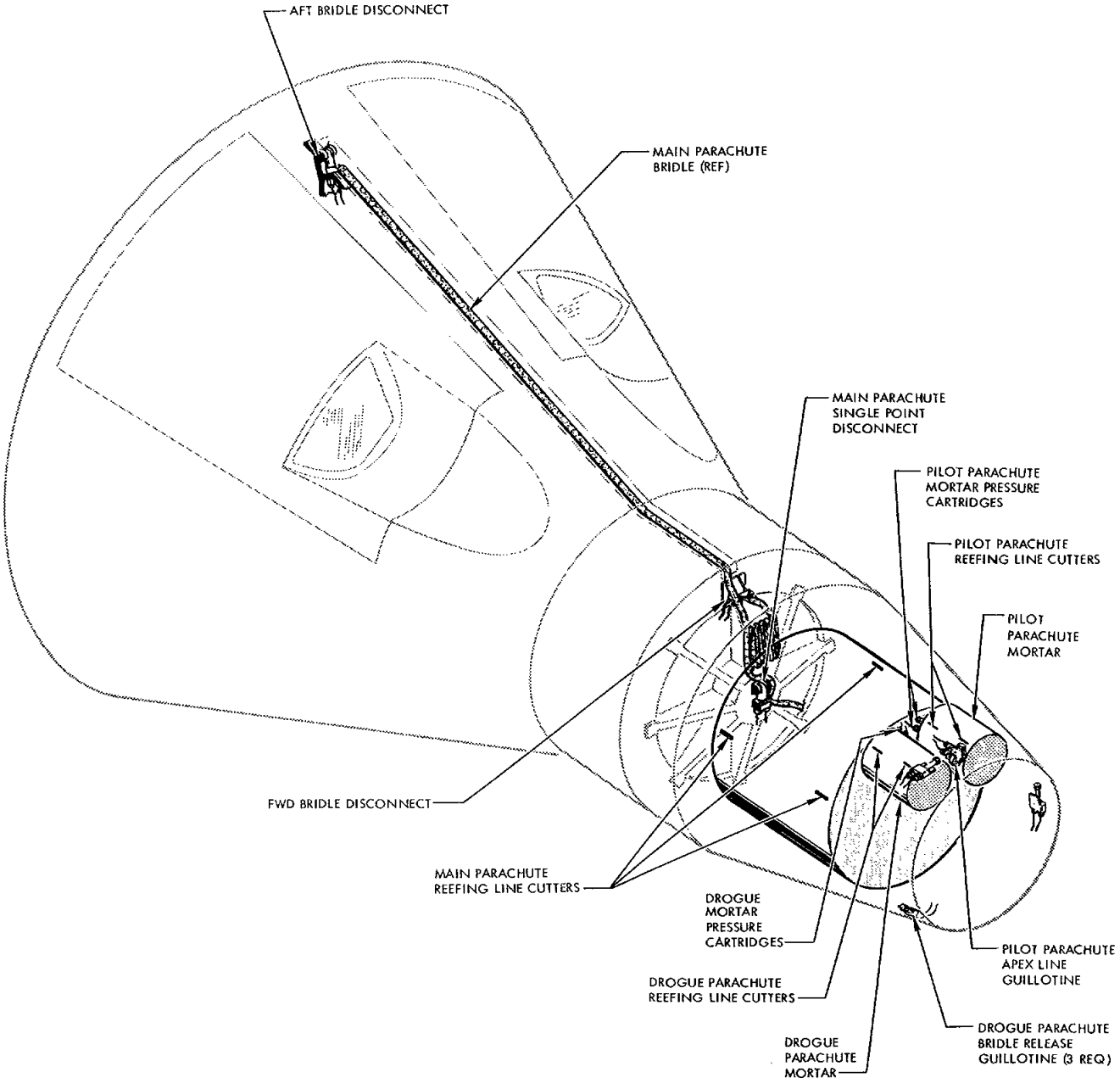


Figure 11-25 Parachute Landing System Pyrotechnics

at its three attach points. The release guillotines are similar in design and operation to the cable and wire bundle guillotine (Figure 11-9).

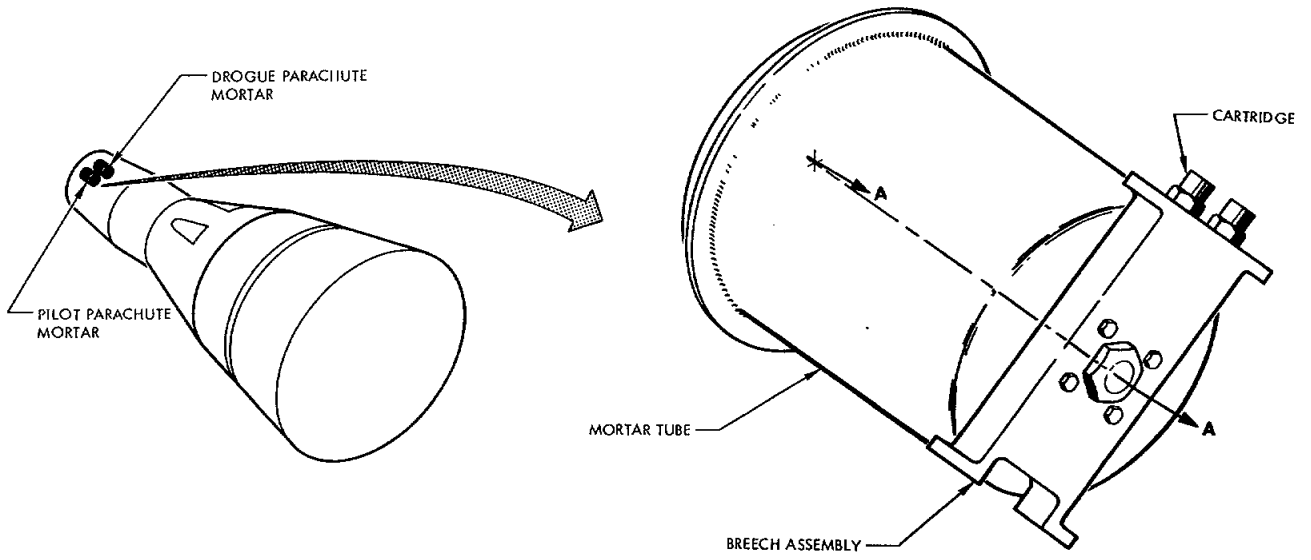
PILOT PARACHUTE MORTAR ASSEMBLY

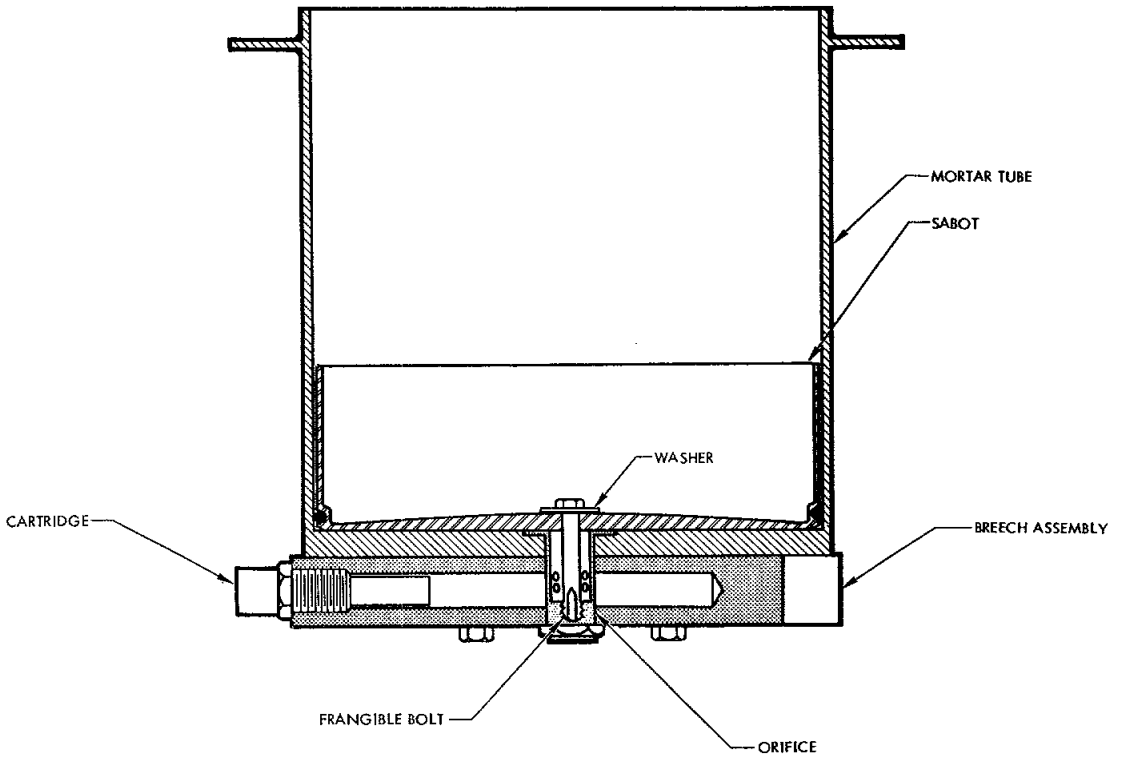
Description

The pilot parachute mortar assembly (Figure 11-26) functions to deploy the pilot parachute in the event of a malfunctioning drogue chute. The mortar assembly is located in the forward end of the rendezvous and recovery section. The mortar assembly primarily consists of the mortar tube, sabot, breech, orifice, frangible bolt, washer and two electrically fired gas pressure cartridges. The base of the mortar tube is attached to the breech and the breech is attached to the rendezvous and recovery section structure. The flanged orifice passes through the base of the mortar tube and breech. The orifice is secured beneath the breech by a lockmut. One end of the breech is drilled and tapped to provide for installation of the two cartridges. The sabot is located in the lower section of the mortar tube and is secured by a washer and a frangible bolt. The frangible bolt passes through the washers, the center of the sabot mortar tube base, and is threaded into the base of the orifice. The pilot chute is installed in the sabot.

Operation

The mortar functions when the cartridges are initiated by a 28 vdc electrical signal. The cartridges generate gas pressure that is ported through the breech and orifice to the base of the sabot. When sufficient pressure is exerted on the sabot, the frangible bolt will part and release the sabot. The gas pressure propels the sabot and pilot chute out of the mortar tube, thus effecting positive chute deployment.





SECTION A-A

Figure 11-26 Pilot Parachute Mortar Assembly

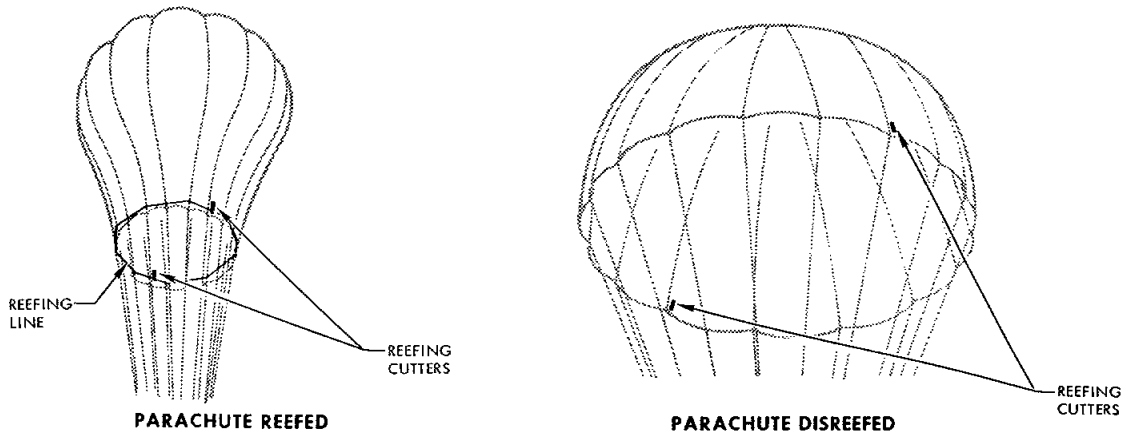
PILOT PARACHUTE REEFING CUTTERS

Description

The pilot parachute reefing cutters (Figure 11-27) are provided to disreef the pilot chute by severing the reefing line. The reefing cutters primarily consist of the cutter body, cutter blade, firing mechanism and percussion fired time delay cartridge. Two cutter assemblies are sewn to the inside of the parachute skirt band 180 degrees apart. The reefing cutter is a tubular device, with all its components contained within the cutter body. The firing mechanism is contained in one end of the cutter body and consists of a firing pin, lock ball, spring, and sear pin. The firing pin is retained in the cocked position by the lock ball. The lock ball is held in place by the sear pin. A lanyard is attached to the sear pin and to the parachute canopy. The spring is precocked and energizes the firing pin when the sear pin is pulled. The cartridge is installed in the center portion of the cutter body and is roll crimped in place. The cartridge consists of a percussion primer, time delay column and output charge. The cutter blade is stake locked in the cutter body, below and adjacent to the output end of the cartridge. A washer is crimp locked in the end of the cutter body, and serves as the anvil and stop for the cutter blade. A hole in each side of the cutter body, between the cutter blade and washer, permits installation of the reefing cable.

Operation

Deployment of the pilot chute causes the reefing cutters to be initiated. As the canopy extends, the lanyard is pulled taut and pulls the sear pin from the firing mechanism. The lock ball moves inboard and unlocks the firing pin. The spring energized firing pin is driven into the primer of the cartridge and ignites the



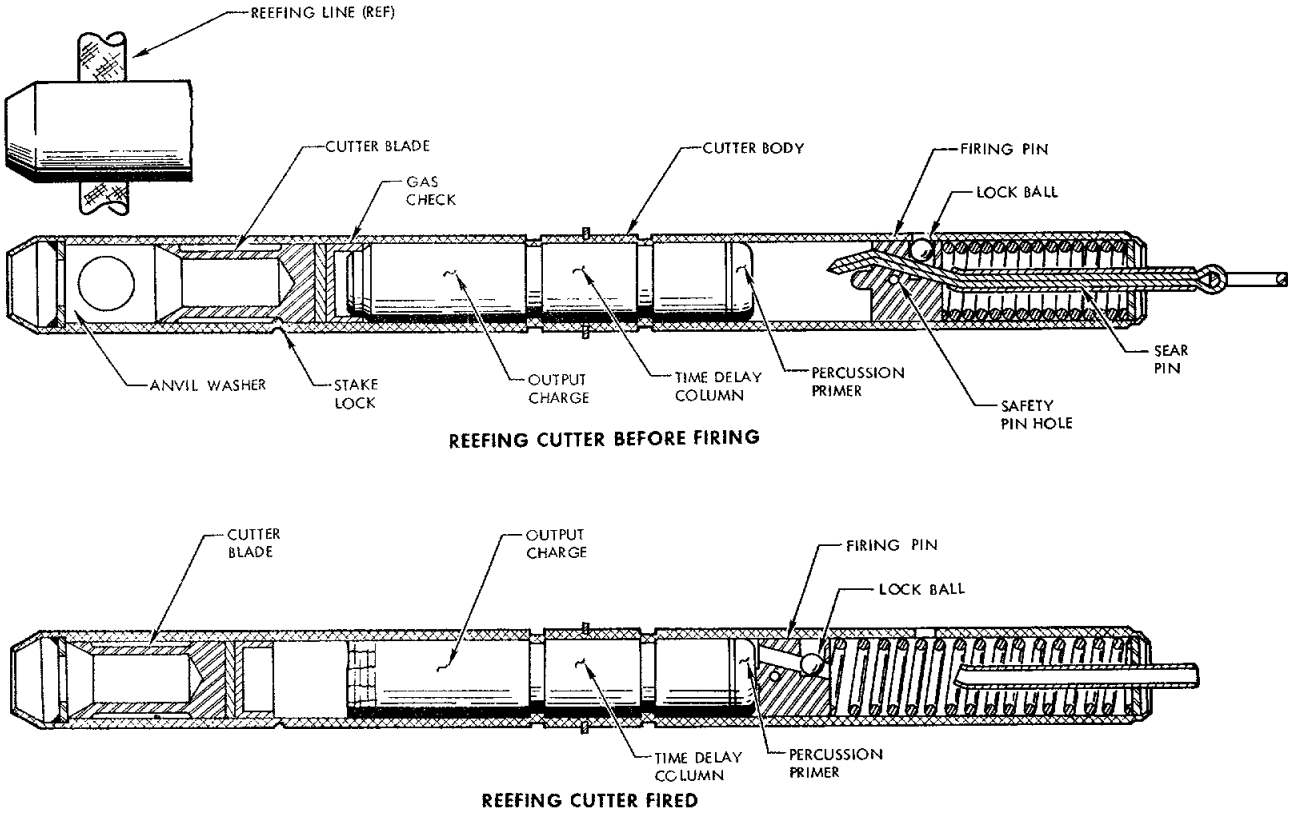


Figure 11-27 Pilot Parachute Reefing Cutters

time delay column. After the specified time delay, the cartridge produces gas pressure that exerts force on the cutter blade. When sufficient force is exerted, the cutter blade shears the stake lock and strokes to sever the reefing cable. Proper functioning of only one of the two cutters is sufficient to perform pilot chute disreefing.

PILOT PARACHUTE APEX LINE GUILLOTINE

Description

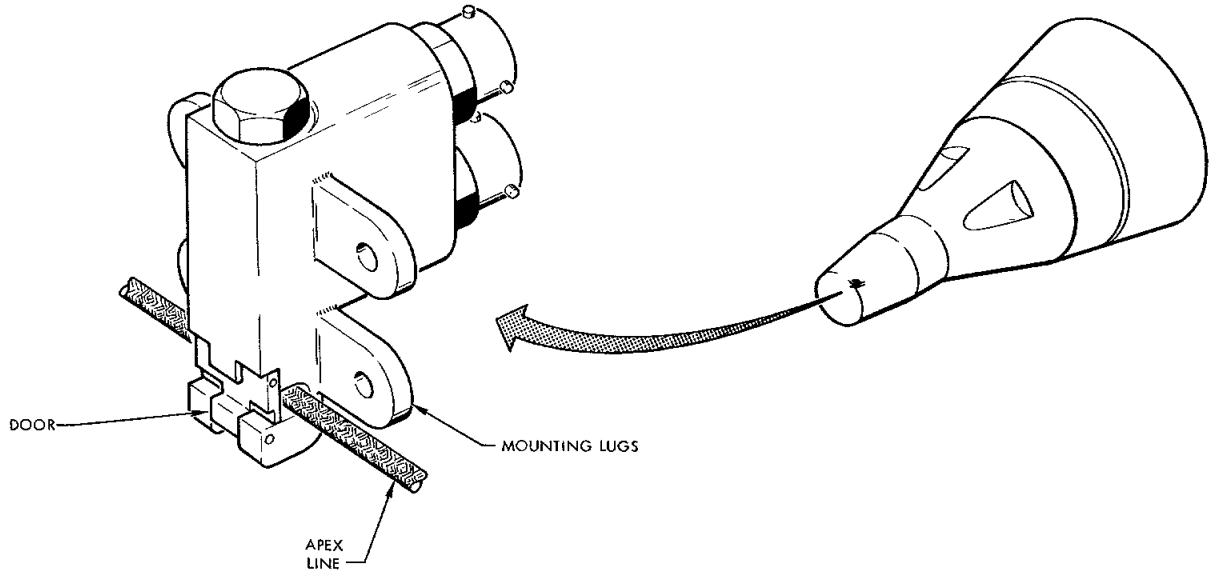
The pilot parachute apex line guillotine (Figure 11-28) is provided to sever the pilot chute apex line, in the event of a drogue chute malfunction. The guillotine primarily consists of the body, cutter blade, and two electrically fired gas pressure cartridges. The cutter blade is retained by a shear pin. The body provides for the installation of the two cartridges, and incorporates drilled passages from the cartridges to the cutter blade. Design of the guillotine allows the apex line to pull free, when the pilot chute is deployed by the drogue chute. The guillotine is located in the forward section of the rendezvous and recovery section.

Operation

When initiated by a 28 vdc electrical signal, the cartridges produce gas pressure. The pressure is ported through a drilled passage to the head of the cutter blade. Sufficient pressure causes the cutter blade to sever the shear pin and stroke to cut the apex line. The apex line is thus free of the malfunctioned drogue chute, permitting mortar deployment of the pilot chute.

MAIN PARACHUTE REEFING CUTTER

The main parachute reefing cutter (Figure 11-29) is provided to disreef the main



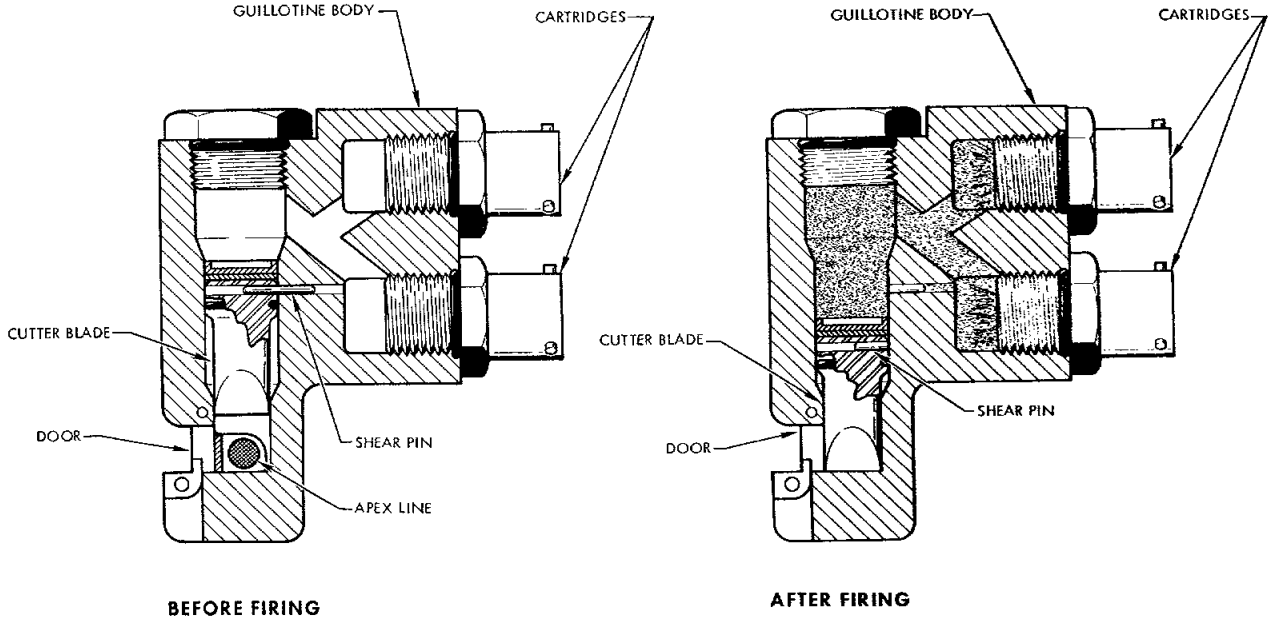
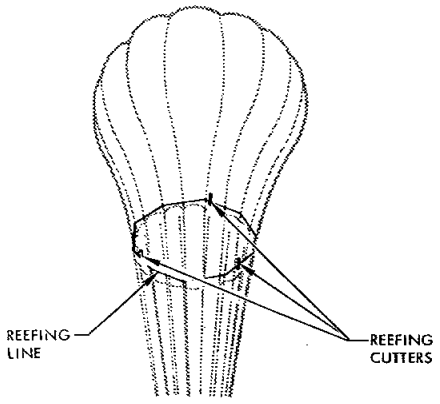
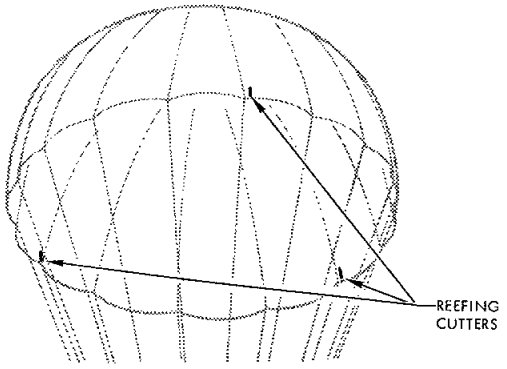


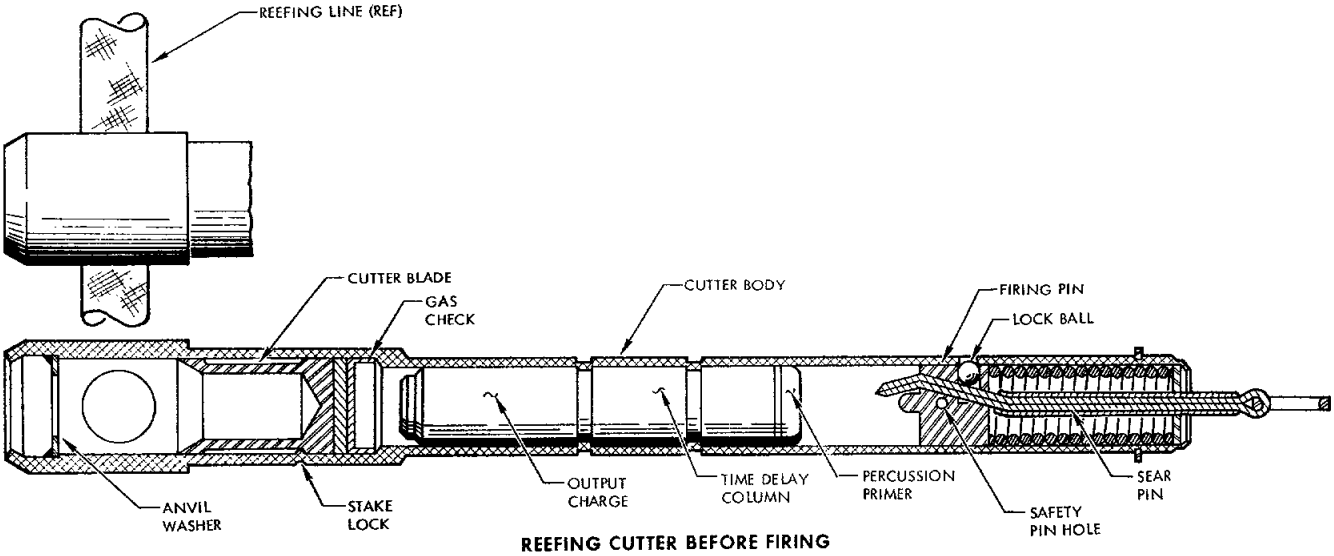
Figure 11-28 Pilot Parachute Apex Line Guillotine



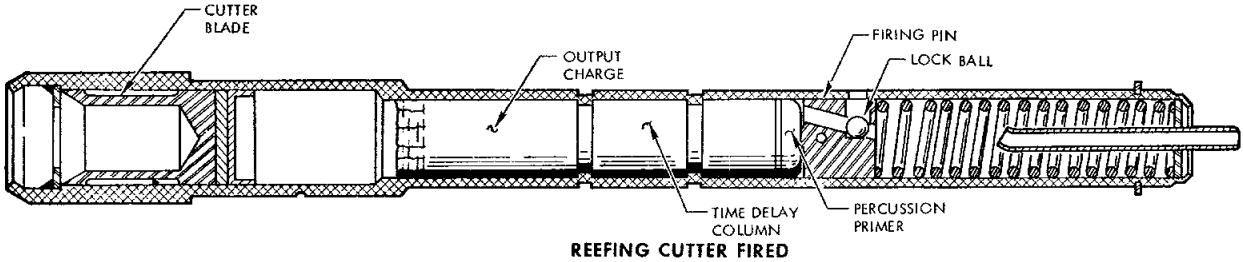
PARACHUTE REEFED



PARACHUTE DISREEFED



REEFING CUTTER BEFORE FIRING



REEFING CUTTER FIRED

Figure 11-29 Main Parachute Reefing Cutter

parachute. Three reefing cutters are located on the inside of the canopy skirt band 120 degrees apart. The reefing cutters are similar in design and identical in operation to the pilot chute reefing cutters. Proper operation of only one of the three cutters is sufficient to disreef the parachute.

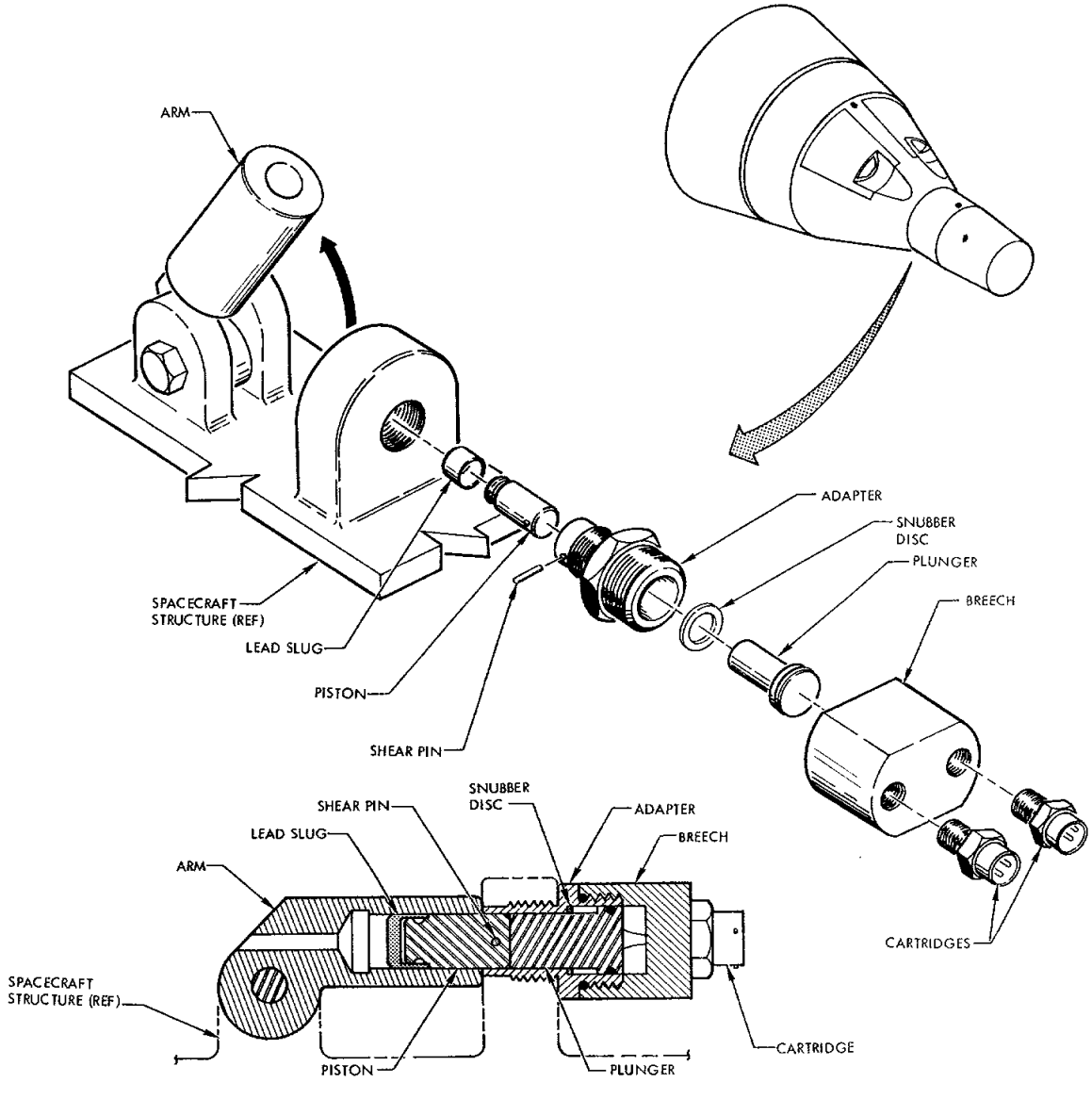
MAIN PARACHUTE DISCONNECT

Description

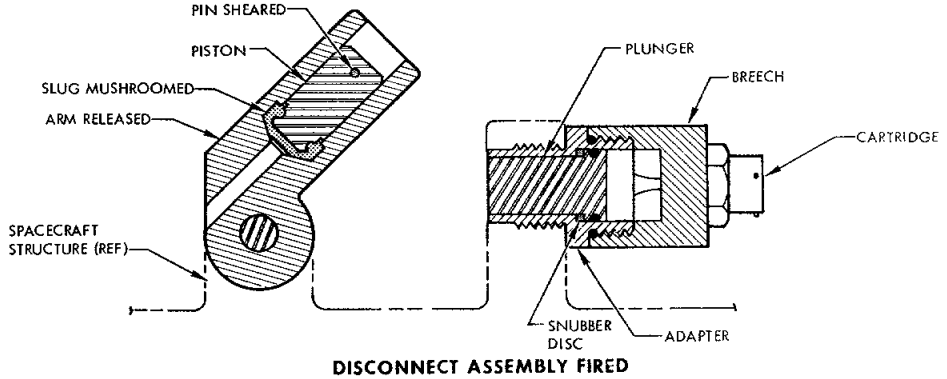
The main parachute disconnects (Figure 11-30) include the single point disconnect assembly and the forward and aft bridle disconnect assemblies. The disconnect assemblies are identical, in design and function. The disconnect assemblies primarily consist of the breech assembly, arm, and two electrically fired gas pressure cartridges. The breech assembly consists of the adapter, piston, lead slug, snubber disc, and plunger. The piston is retained in the adapter by a shear pin. The lead slug is located on the end of the piston. The snubber disc is located under the head of the plunger. The adapter is threaded into the spacecraft structure with the piston extending into the arm. The breech is threaded onto the adapter and the cartridges are installed in the breech. The single point disconnect is mounted on the hub of the main parachute adapter assembly. The adapter assembly is located on the forward ring of the Re-entry Control System section. The forward bridle disconnect is mounted at the top of the forward ring of the Re-entry Control System section. The aft bridle disconnect is located forward of the heat shield between the crew hatches.

Operation

When initiated by a 28 vdc electrical signal, the disconnect cartridges are ignited. The cartridges produce gas pressure that is ported through drilled passages



DISCONNECT ASSEMBLY BEFORE FIRING



DISCONNECT ASSEMBLY FIRED

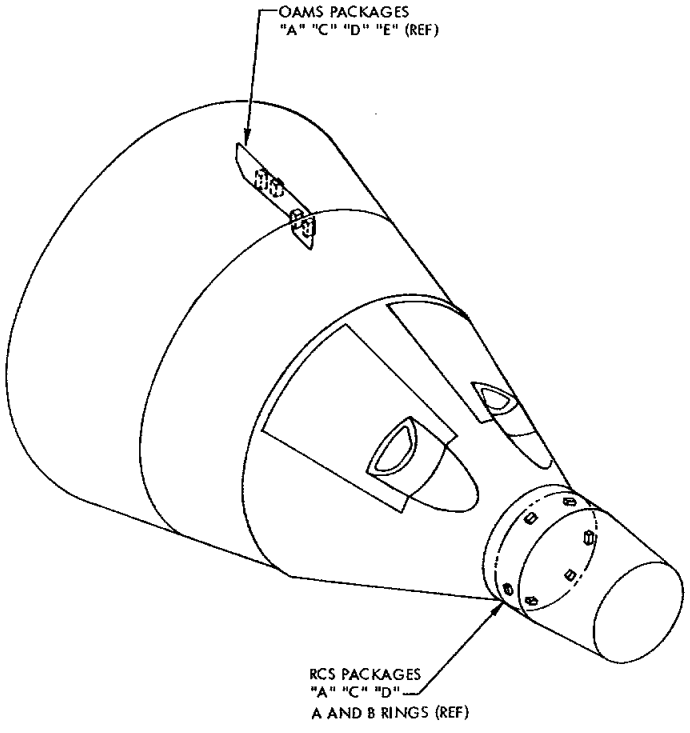
Figure 11-30 Main Parachute Disconnects

in the breech, to a common chamber at the head of the plunger. The gas pressure exerts a force on the head of the plunger, which in turn propels the piston by physical contact. The piston severs the shear pin and is driven into the arm. The plunger is prevented from following the piston, since the head of the plunger strikes a shoulder in the adapter. The snubber disc provides a cushioning effect, to prevent shearing the plunger head. As the piston strikes the back of the arm, the lead slug at the end of the piston mushrooms. Mushrooming of the slug, retains the piston in the arm, preventing the piston from hindering arm operation. The pull of the parachute causes the arm to cam open, thus releasing the riser or bridle.

PYROTECHNIC VALVES

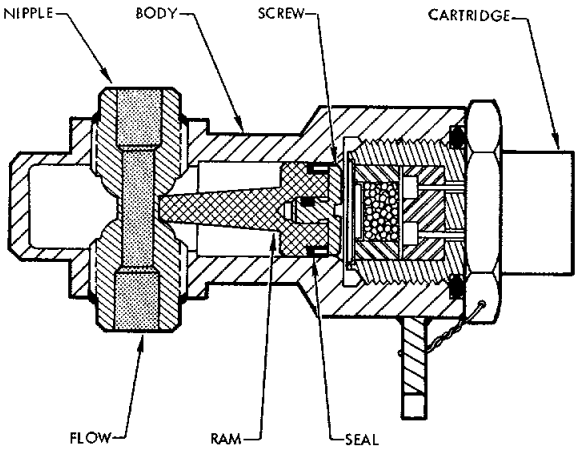
DESCRIPTION

Pyrotechnic valves(Figure 11-31)are installed in the Orbit Attitude and Maneuvering System(OAMS)and in the Re-entry Control System(RCS). The valves are one time actuating devices, used to control the flow of fluids. The spacecraft contains pyrotechnic valves that consist of the electrically fired high explosive cartridge, valve body, nipple, ram, seal, and screw. The nipple, either open or closed depending on the particular valve, is installed in the valve body and welded into place. The ram, incorporating the seal and screw at its head is located in the valve body, indexed directly above the center of the nipple. The cartridge is installed in the valve body at the top of the ram head. Two types of valves are used; normally open and normally closed. The "A" packages of the RCS and OAMS, contain a normally closed non-replaceable valve. The "E" package of the OAMS contain a normally open, and a normally closed, non-replaceable valve. If the valves in the "A" and "E"

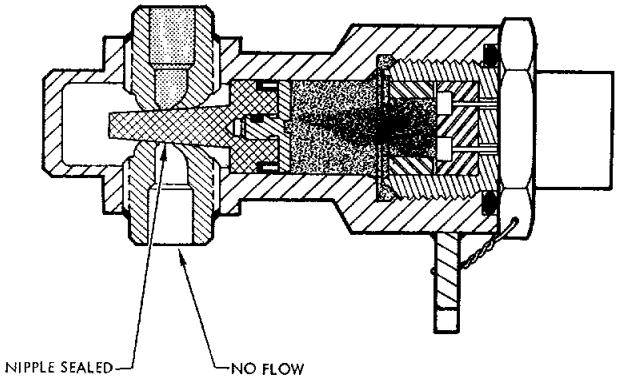


OAMS PACKAGE "E"

BEFORE FIRING



AFTER FIRING

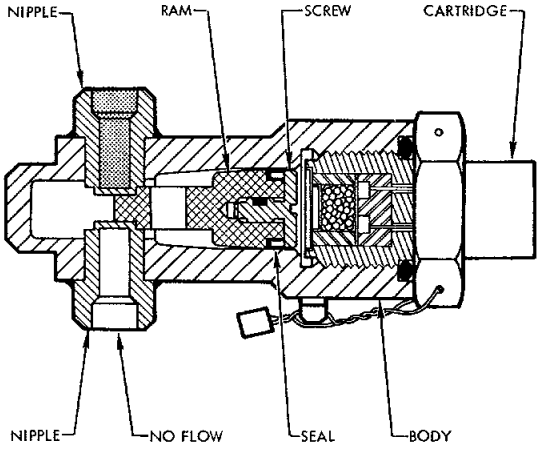


NORMALLY OPEN VALVE (NON-REPLACEABLE)

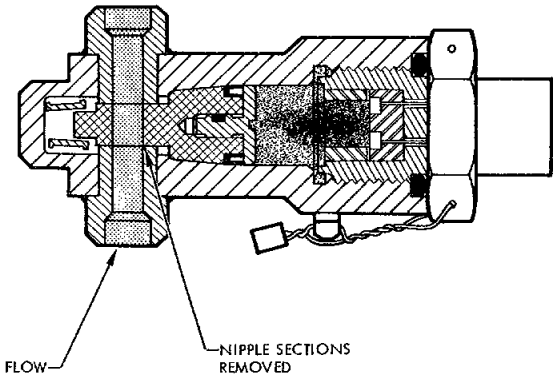
Figure 11-31 Pyrotechnic Valves (Sheet 1 of 2)

OAMS PACKAGE "A" AND "E"
RCS PACKAGE "A"

BEFORE FIRING



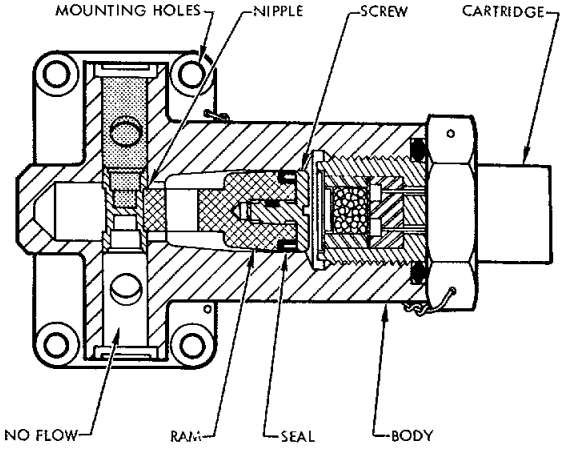
AFTER FIRING



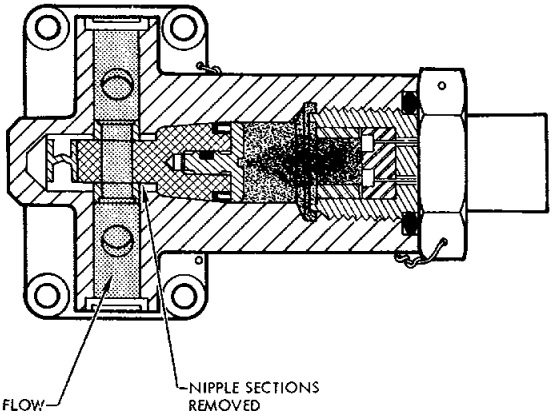
NORMALLY CLOSED VALVE (NON-REPLACEABLE)

OAMS PACKAGE "C" AND "D"
RCS PACKAGE "C" AND "D"

BEFORE FIRING



AFTER FIRING



NORMALLY CLOSED VALVE (REPLACEABLE)

Figure 11-31 Pyrotechnic Valves (Sheet 2 of 2)

packages are defective, or the cartridge has been fired, the packages must be changed. The "C" and "D" packages of the RCS and the OAMS contain normally closed replaceable valves. These valves are attached to the exterior of the package, and if defective or the cartridge fired, may be changed individually.

OPERATION

Normally open valve; the pyrotechnic valve is caused to function when the cartridge is initiated by a 28 vdc electrical signal. Ignition of the cartridge produces gas pressure that acts on the head of the ram. The ram is driven down on the nipple, severing and removing a section of the nipple. The ram, having a tapered cross section, is wedged in the nippled opening, completely sealing the nipple, thus stopping the flow of fluid. Normally closed pyrotechnic valves are all basically identical except for nipple design. The non-replaceable valve has two closed end nipples butted together. The ram severs and removes the end of each nipple and wedges itself between the ends. A hole is incorporated in the ram, allowing fluid flow after ram actuation. The replaceable pyrotechnic valve has a nipple installed with a bulkhead in the cross section that stops fluid flow. The ram removes the section of the nipple containing the bulkhead and wedges itself in place. A hole incorporated in the ram allows fluid to flow.

RETROGRADE ROCKET SYSTEM

The retrograde rocket system (Figure 11-32) primarily consists of four solid propellant rocket motors and eight igniter assemblies. The retrograde rockets are provided to retard spacecraft orbital velocity for re-entry and to provide distance and velocity to clear the launch vehicle in the event of an abort during ascent. The rocket motors are symmetrically located about the longitudinal axis

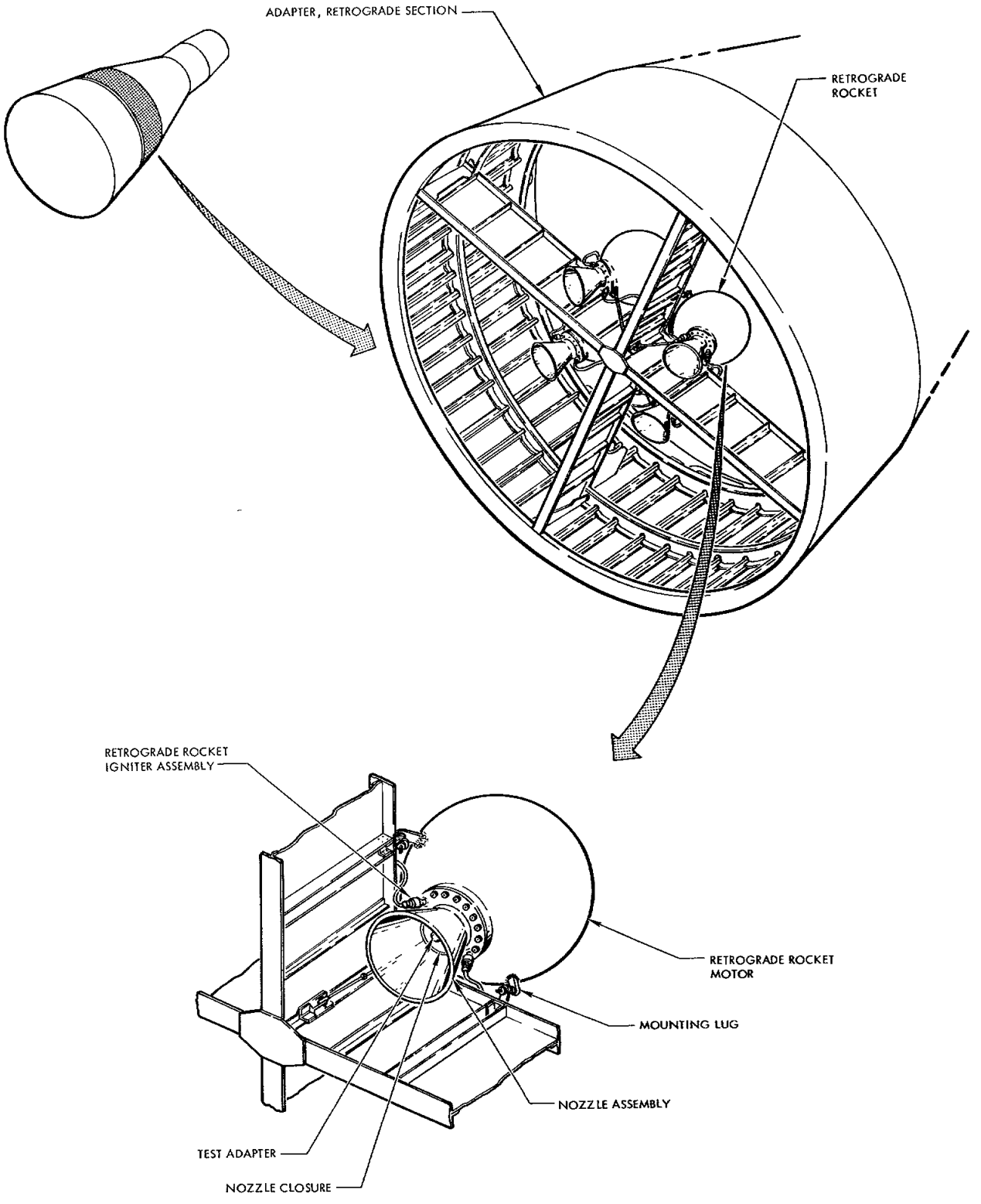


Figure 11-32 Retrograde Rocket System

of the spacecraft and are mounted in the retrograde section of the adapter. The rocket motors are individually, optically aligned prior to mating the adapter to the re-entry module.

RETROGRADE ROCKET MOTOR ASSEMBLY

Description

The spacecraft contains four retrograde rocket motor assemblies (Figure 11-33) that are identical in design and performance, spherical in shape, and are approximately 13 inches in diameter.

Rocket Motor Case

The motor case is formed from titanium alloy in two hemispherical sections. The halves are forged, machined, and welded together at the equator. Each hemisphere is insulated to reduce heat transfer during motor operation. The aft hemisphere is drilled and tapped to provide a mating flange for the nozzle assembly. The nozzle assembly, a partially submerged type, consists of the expansion cone, throat insert and nozzle bulkhead. The nozzle bulkhead is a machined titanium alloy, bolted to the flange at the aft end of the motor case. The bulkhead is threaded to provide for expansion cone installation. The expansion cone is compression molded of vitreous silica phenolic resin and is threaded into the nozzle bulkhead. The throat insert is machined from high density graphite and is pressed into the nozzle bulkhead. The throat insert is insulated from the bulkhead by a plastic material to reduce heat transfer during motor operation. The throat insert is recessed into the motor case to reduce nozzle assembly length. A rubber nozzle closure is sandwiched between the throat insert and the nozzle bulkhead. The closure incorporates a shear groove that permits ejection at a predetermined

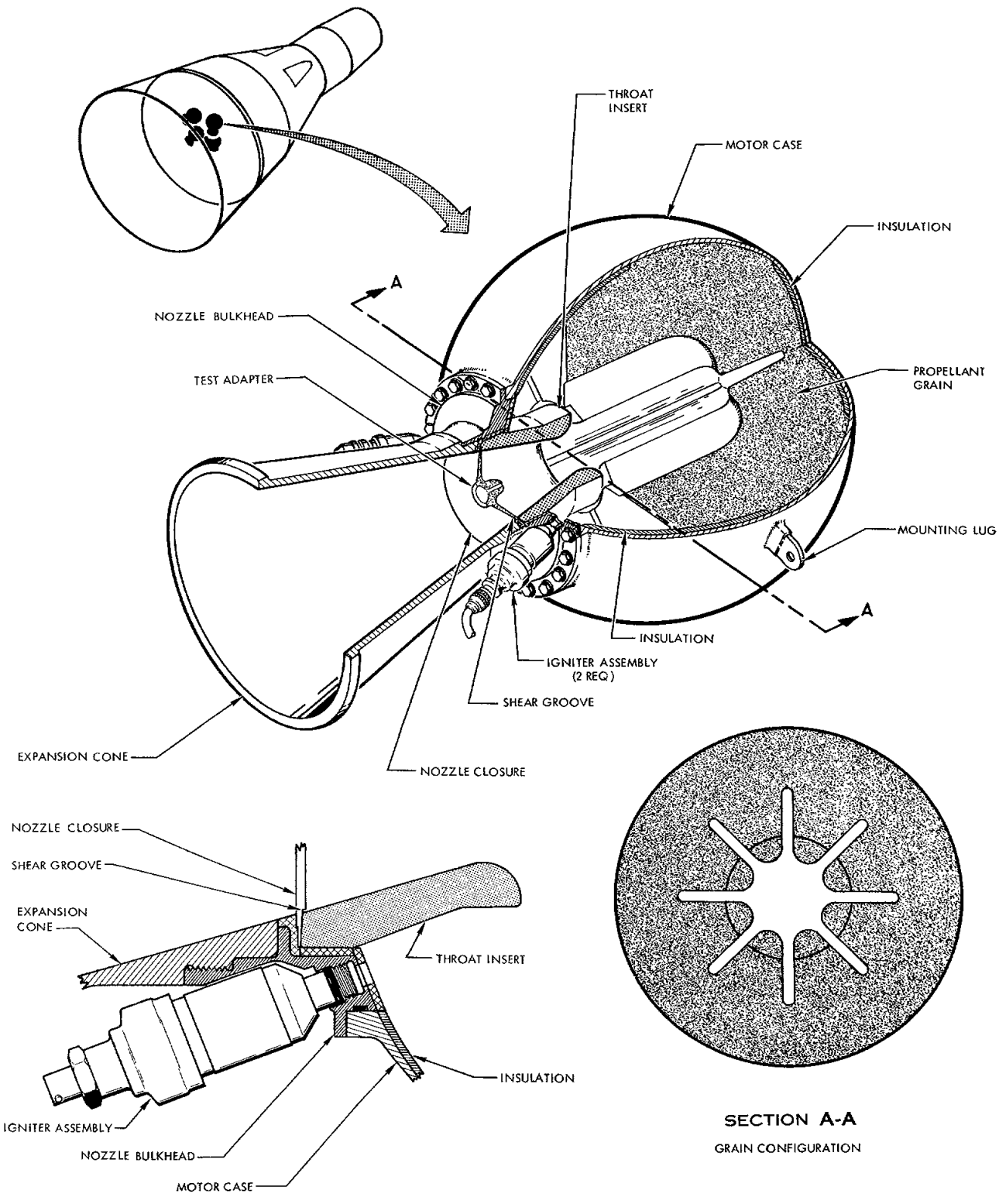


Figure 11-33 Retrograde Rocket Motor Assembly

internal pressure level, or basically at motor ignition. A test adapter fitting is incorporated in the closure to permit pressure checking of the rocket motor.

Rocket Motor Propellant

The motor case is lined with a rubber material that provides propellant grain to case adhesion. The rocket motor propellant is cast and cured in the motor case. The propellant grain is cast in an internal burning eight pointed star configuration. The propellant grain is ignited by the two igniter assemblies, mounted 180 degrees apart on the aft end of the motor case, adjacent to the nozzle assembly.

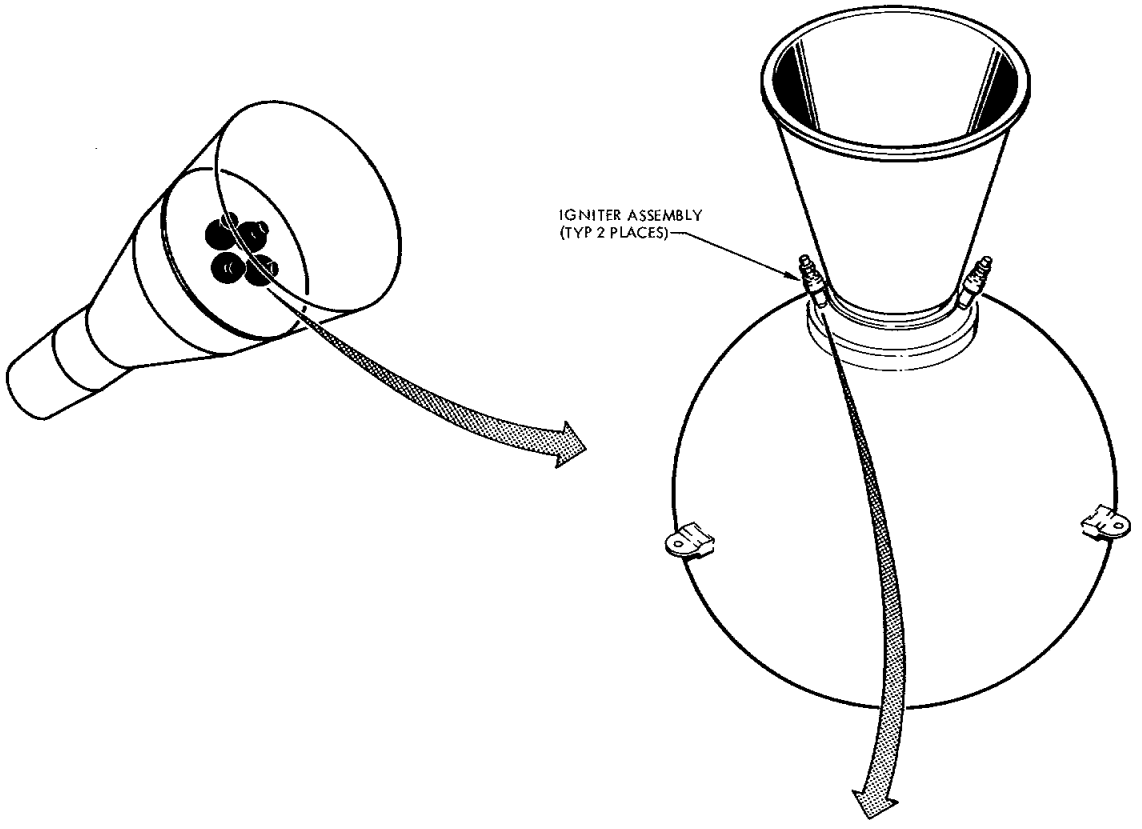
Operation

The retrograde rocket motors function in two modes: normal and abort. In the normal mode of operation, the rocket motors are used to initiate spacecraft re-entry. The rocket motors are fired at 5.5 second intervals in 1-2-3-4 order. The propellant grain of the rocket motor is ignited by the hot gases from the igniter assemblies. The propellant grain burns over the entire surface of its eight pointed star configuration until exhausted. The thrust produced by the motors is transmitted to the spacecraft structure and retards spacecraft velocity. In the abort mode of operation, the rocket motors are fired in salve or as mission requirements may direct.

RETROGRADE ROCKET IGNITER ASSEMBLY

Description

The retrograde rocket igniter assemblies (Figure 11-34) are used to ignite the propellant grain of the retrograde rocket motor. The spacecraft contains eight igniter assemblies that primarily consist of the case, head cap, grain, booster



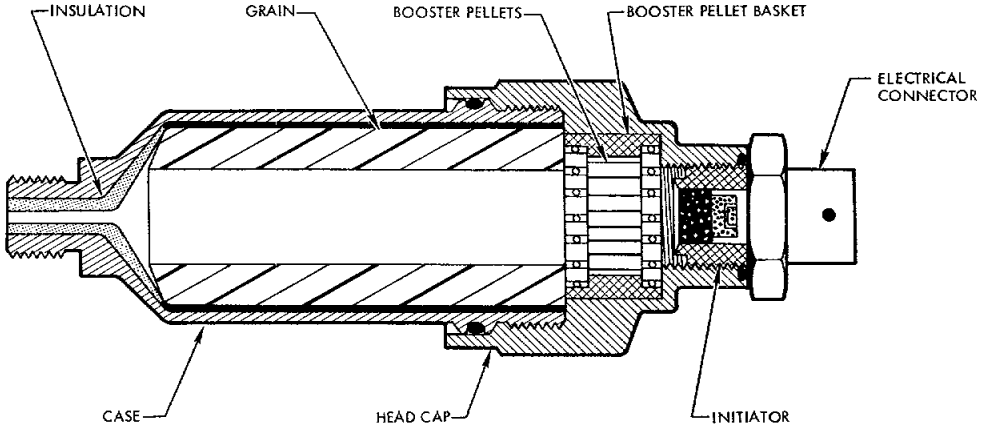


Figure 11-34 Retrograde Rocket Igniter Assembly

pellets, pellet basket, and initiator. The igniter assembly is essentially a small solid propellant rocket motor. The case and head cap are individually machined from a stainless steel alloy and have a threaded interface. On the internal surface of the case, at the gas exit, a silica-phenolic insulating material is bonded to reduce heat transfer during igniter firing. The grain is cast and cured in a phenolic paper tube. The grain is inserted into the igniter case prior to case and head cap assembly. The booster pellets, consisting of boron potassium nitrate, are contained in the pellet basket, located in the head cap. The pellet basket is held in place by the head cap and is installed prior to case and head cap mating. The initiator cartridge consists of the body, one firing circuit (bridge wire), ignition mix, and output charge. The basic function of the initiator is to fire the igniter. The initiator is threaded into the head cap of the igniter at the time of igniter assembly build up at the vendors.

Operation

When initiated by a 28 vdc electrical signal, the initiator of the igniter assembly is activated. The initiator ignites the boron pellets, which boosts the burning to the igniter grain. The igniter grain generates hot gas which is exhausted into the retrograde rocket motor cavity. The hot gases provide the temperature and pressure for retrograde rocket motor propellant grain ignition. Either igniter is sufficient to initiate burning of the rocket motor.

DOCKING SYSTEM PYROTECHNIC DEVICES

The pyrotechnic devices utilized for docking are located in the R and R section of the spacecraft (Figure 11-3). The pyrotechnic devices utilized for docking consist of:

(a) A pyrotechnically actuated indexing bar to aid the astronauts in mating the spacecraft with the TDA.

(b) Three pyrotechnically ejected latch receptacles.

(c) Three pyrotechnically actuated cable cutters to release the three latch covers.

DOCKING BAR ASSEMBLY

Description

The docking bar assembly (Figure 11-35) primarily consists of the cylinder, inner piston, inner piston extension, cutter piston, indexing bar, housing/manifold, two breeches, extension cartridge, two jettison cartridges, and a locking mechanism. The basic function of the assembly is to extend and lock the indexing bar, prior to the docking maneuver and to jettison the indexing bar after the docking operations are completed. The purpose of the indexing bar is to aid both visually and mechanically in the docking maneuver. The docking bar assembly is located in the rendezvous and recovery section and attached to the section structure. The housing/manifold, mounted at the top of the cylinder, contains the locking mechanism and provides for installation of two breeches. The breeches provide for the installation of cartridges. One extension cartridge is installed in the extension breech on the left hand side of the manifold and two jettison cartridges are installed in the jettison breech on the right hand side of the manifold.

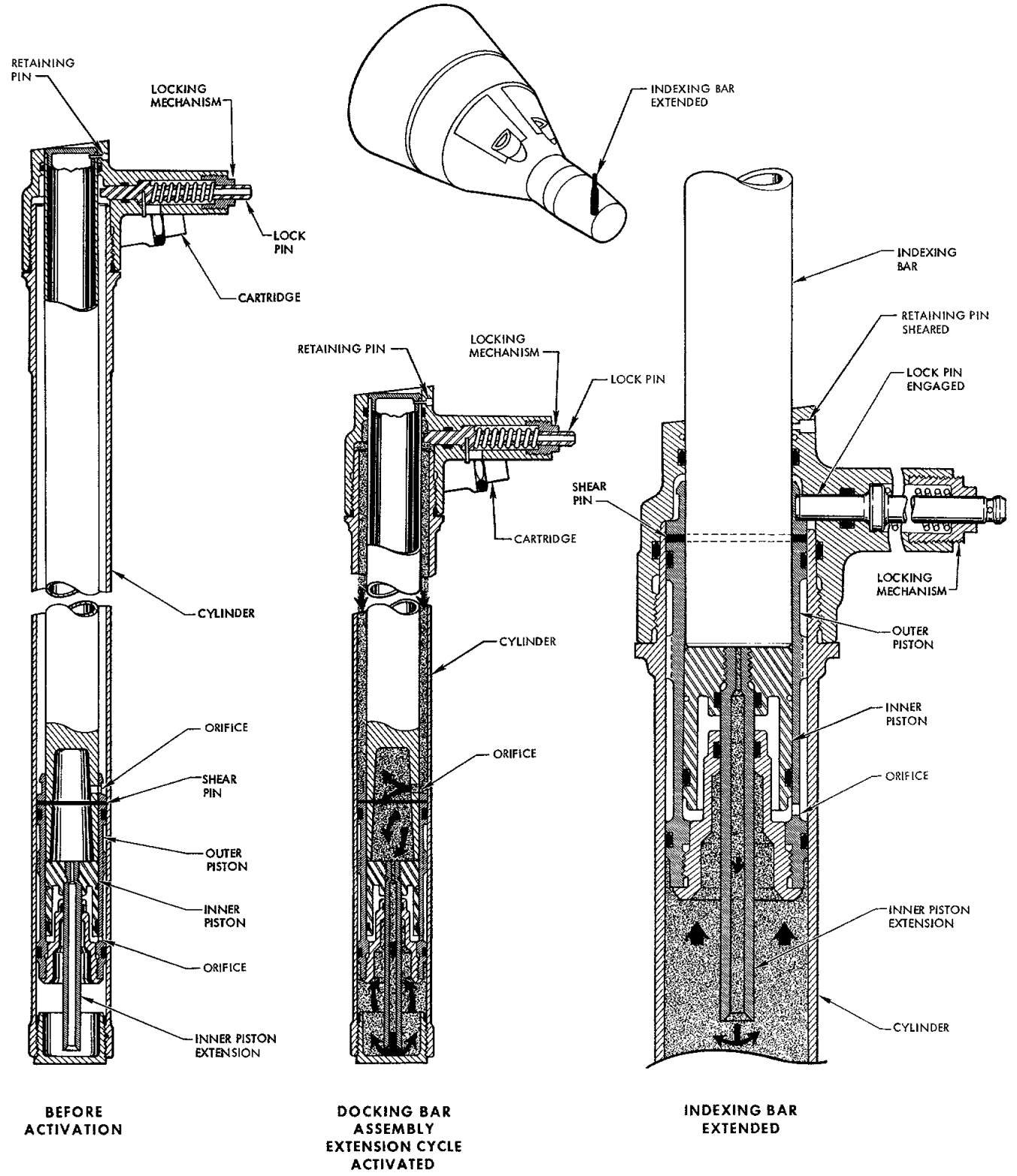


Figure 11-35 Docking Bar Assembly (Sheet 1 of 2)

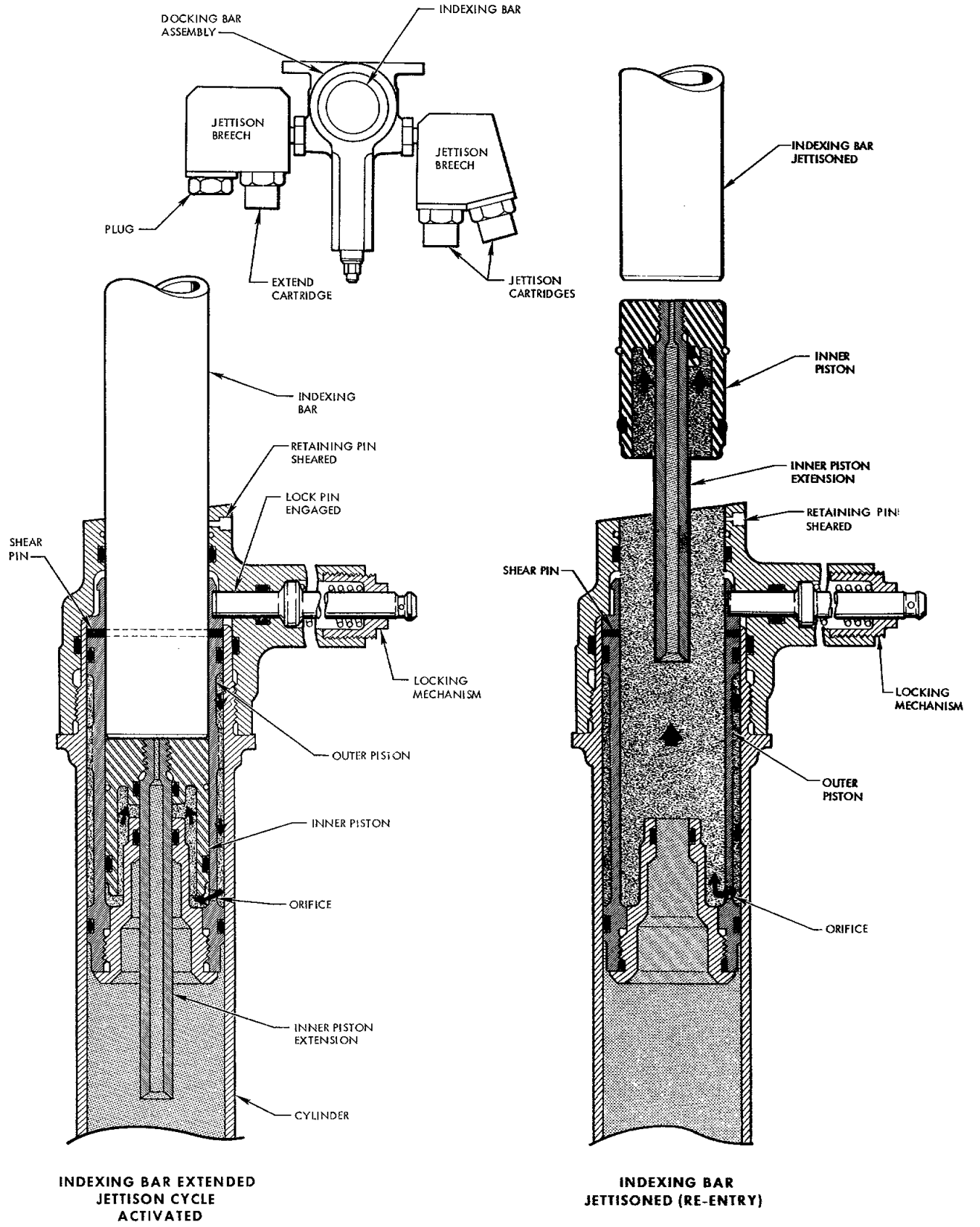


Figure 11-35 Docking Bar Assembly (Sheet 2 of 2)

The hollow inner piston extension protrudes through the cutter piston base and is attached to the inner piston. The indexing bar is secured to the outer piston by a shear pin and retained in the cylinder by a retaining pin.

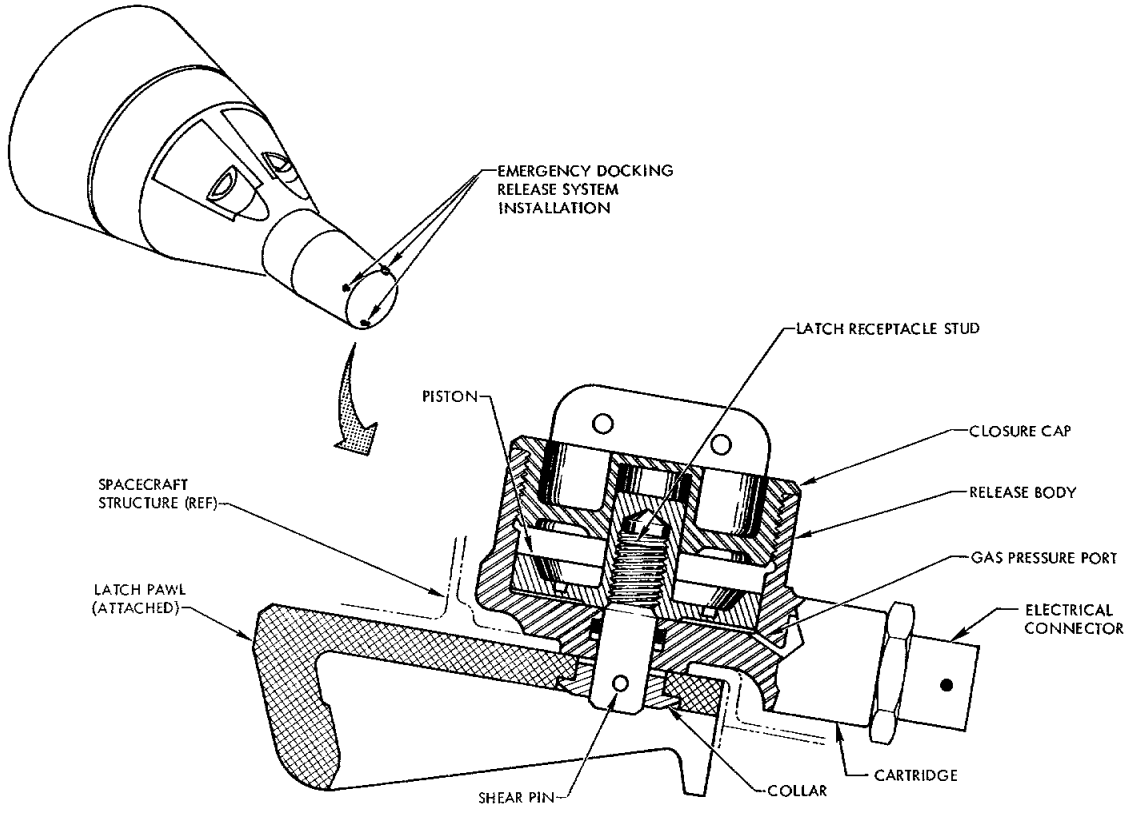
Operation

The indexing bar is extended by the gas pressure generated when the extension cartridge is initiated by a 28 vdc signal. The gas pressure is ported into the cylinder and enters an orifice in the base of the indexing bar. The gas pressure is then ported through the hollow inner piston extension and exerts force on the bottom of the outer piston, causing it to sever the retaining pin and extend the outer piston with indexing bar attached. When fully extended, the indexing bar is secured in place by the pin of the locking mechanism engaging a groove in the outer piston. The extension cartridge has dual bridge wires with a separate electrical circuit to each bridge wire for redundancy. The indexing bar is jettisoned when the dual jettison cartridges are initiated by a 28 vdc electrical signal. The gas pressure generated by the jettison cartridges is ported into the cylinder and through an orifice in the outer piston to a cavity between the inner and outer pistons. The thrusting action of the inner piston causes the indexing bar to sever the shear pin and be jettisoned. Both the inner piston and the indexing bar are jettisoned. Initiation of only one jettison cartridge is sufficient to jettison the indexing bar. The jettison cartridges have a two second pyrotechnic time delay to assure that during an abort mode the extend cartridge will be firing first to extend the bar before the jettison cycle is initiated.

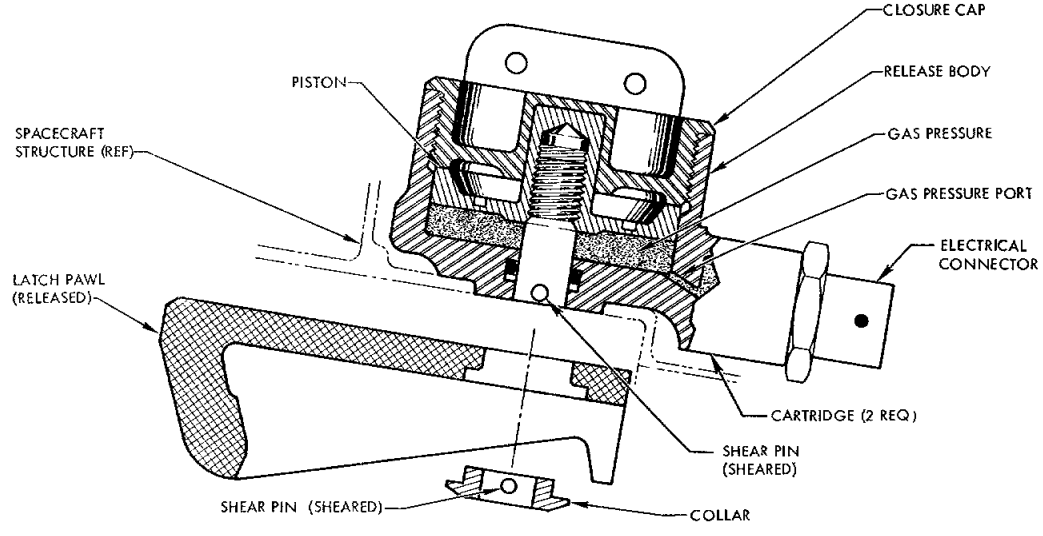
EMERGENCY DOCKING RELEASE SYSTEM

Description

The emergency docking release system (Figure 11-36) contains three release assem-



EMERGENCY DOCKING RELEASE ASSEMBLY (UNACTUATED)



EMERGENCY DOCKING RELEASE ASSEMBLY (ACTUATED)

Figure 11-36 Emergency Docking Release System

blies located symmetrically about the longitudinal axis and attached to the rendezvous and recovery structure at the outer mold line. The basic function of the release system is to positively release the spacecraft from the docking vehicle in the event of normal release system failure. The release assemblies primarily consist of a latch pan, body, piston closure cap stud assembly and two gas pressure cartridges. The release body is attached at the opposite side of a common structural member of the rendezvous and recovery section. A stud assembly, torqued into the docking release piston protrudes out of the unit body. The latch pan is secured to the stud by a collar and shear pin. Installation of the closure cap seals the area above the release piston. Each unit is activated by two pyrotechnic cartridges, installed in the base of the release body.

Operation

Malfunction of the normal docking release system requires the use of the emergency docking release system. Upon receipt of a 28 vdc electrical signal, the cartridges are ignited in all three release assemblies. The cartridges generate gas pressure that is ported through drilled passages in the body to the underside of the piston. The pressure displaces the piston, separating the shear pin from the latch pawl and moving the piston stud up into the body to clear the latch pawl. The latch pawl being held by the docking vehicle latch, is thus parted from the emergency release body. The spacecraft is then free to move out of the docked condition.

TRANSPONDER COVER

Description

The Target Docking Adapter has a single Flexible Linear Shape Charge (FLSC) ring assembly to open and jettison the transponder cover. The jettison assembly

consists of one FLSC, installed around the inside edge of the transponder cover, two detonator blocks, two detonators and an explosive ring. The explosive ring is bolted to the TDA under the transponder door. Detonators are inserted in the two detonator blocks, which are installed on the TDA under the explosive ring.

Operation

Upon receipt of a 28 vdc electrical signal, the detonators transmit a detonation wave that is propagated to the FLSC. The FLSC detonates, severs and jettisons the transponder door skin. The explosive ring absorbs the shock in back of the blast and protects the transponder and structure from shrapnel. One strand of FLSC is adequate to sever and jettison the transponder door.

FUEL CELL HYDROGEN TANK VENT ACTUATOR

Description

The fuel cell hydrogen tank vent actuator (Figure 11-37) is utilized to puncture the hydrogen tank pinch-off tube to allow any gas in the void between the inner and outer tank walls to escape after the spacecraft is in orbit, thus increasing the thermal efficiency of system. The hydrogen tank vent actuator assembly consists of a body, piston/cutter blade, shear pin, guard, breech, ballistic hose, and an electrically fired gas pressure cartridge. The body houses the piston/cutter blade which is retained by a shear pin and provides for attachment of the blade guard. The actuator body assembly is bonded and strapped in place on the fuel cell hydrogen tank and connected to the breech by a ballistic hose. The breech contains the gas generating cartridge and is positioned approximately nine inches from the actuator.

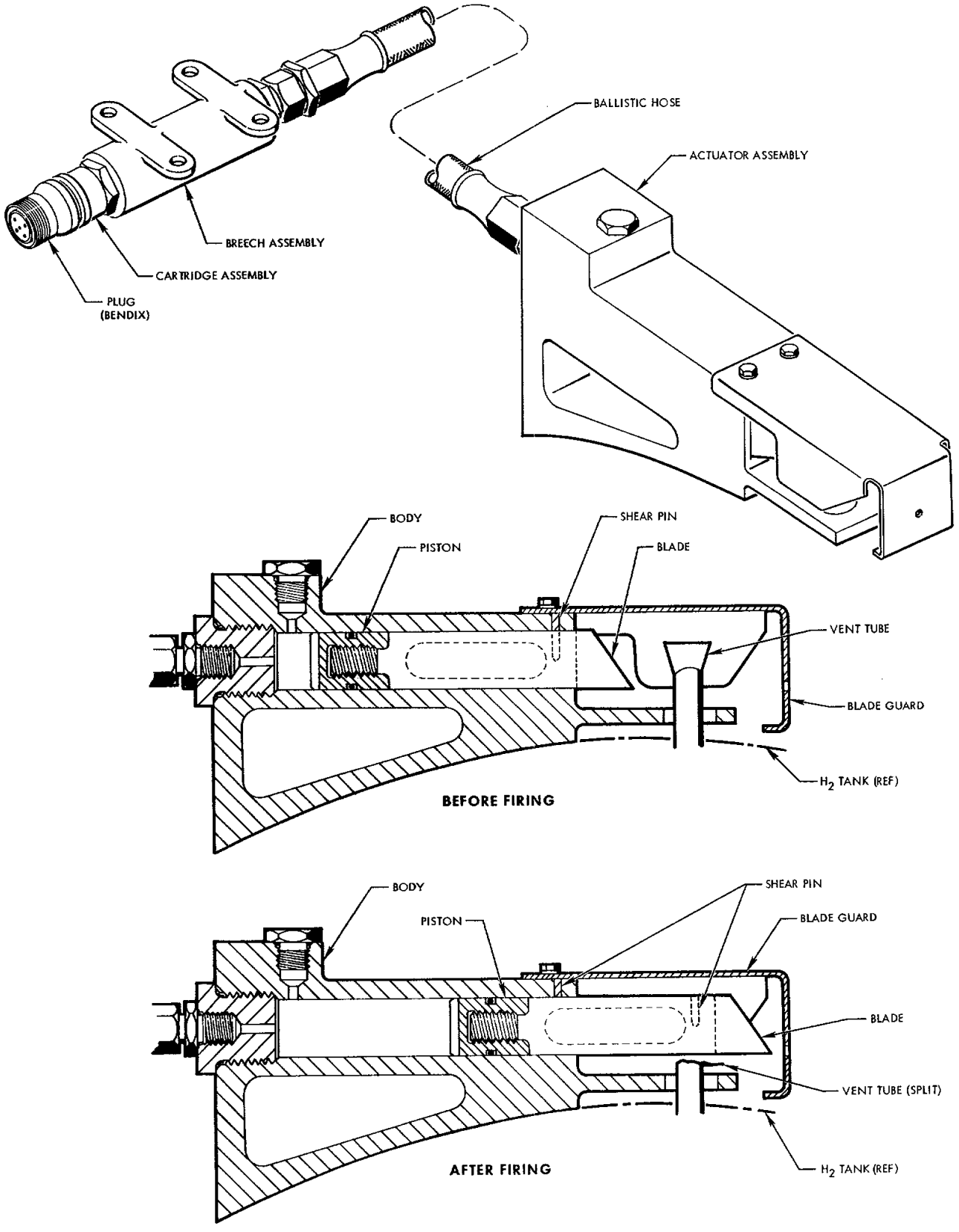
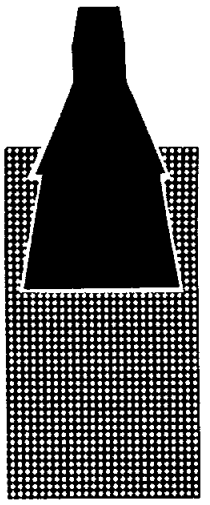


Figure 11-37 H_2 Tank Vent Actuator Assembly

Operation

When initiated by a 28 vdc electrical signal, the cartridge generates gas pressure that is transferred through a ballistic hose to the actuator body which exerts a force on the piston/cutter blade. The gas pressure causes the piston/cutter blade to move, severing the shear pin and spliting the vent tube. The blade guard is attached to the actuator body and forms a barrier which will stop the blade when the piston reaches the end of its stroke.

LANDING SYSTEM



Section XII

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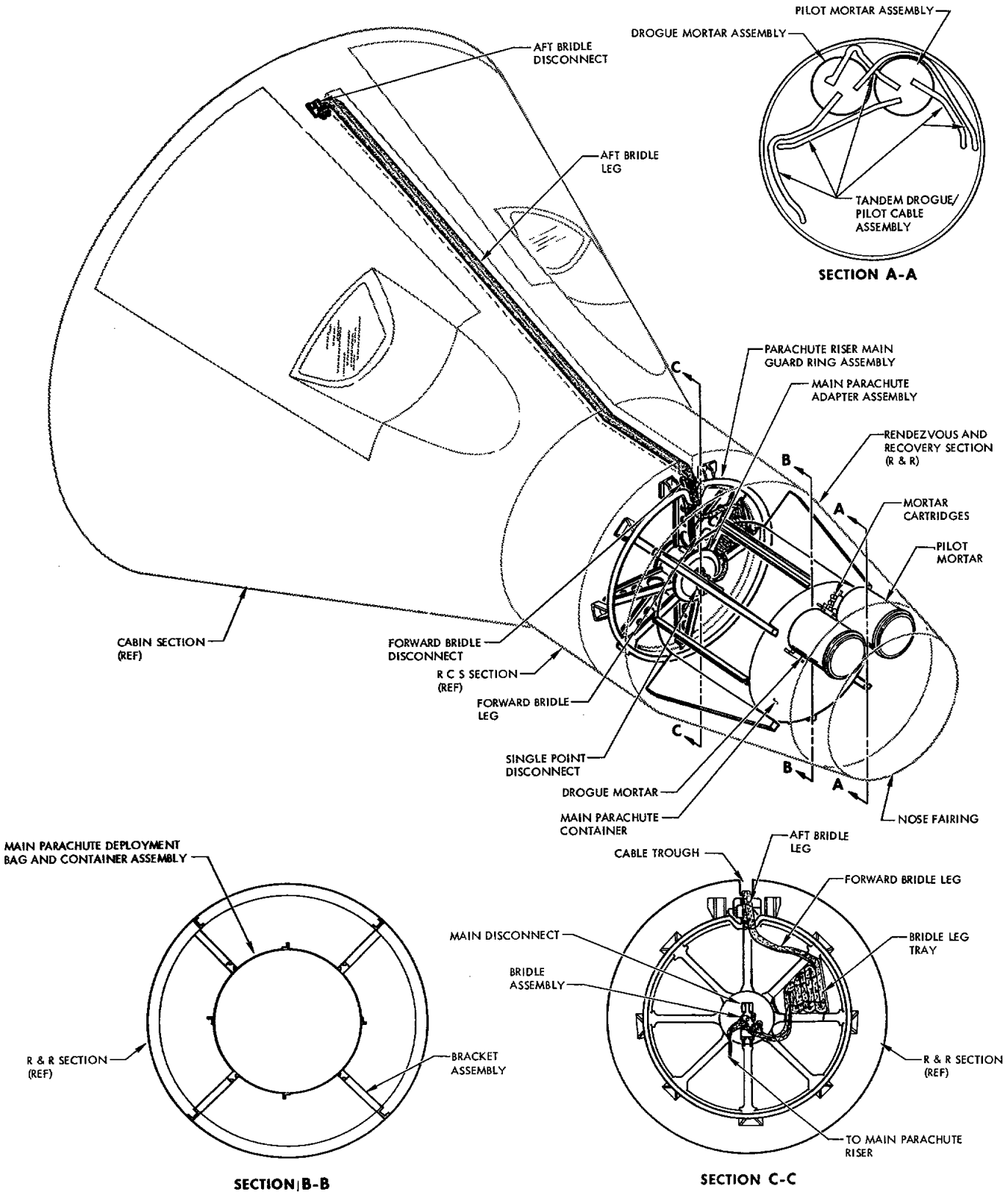


Figure 12-1 Parachute Landing System

SECTION XII LANDING SYSTEM

SYSTEM DESCRIPTION

The parachute Landing System (Figure 12-1) provides a safe rate of descent to return the re-entry module to the earth's surface and furnishes the proper attitude to the landing module for a water impact. A system of three parachutes in series is utilized for stabilizing and retarding the velocity of the spacecraft from 50,000 feet. During the final stage of descent, the main parachute suspension is changed from a single point to a two point system in order to achieve a more favorable attitude for a water landing. The Landing System consists of three parachute assemblies (a drogue parachute assembly, a pilot parachute assembly, and a main parachute assembly), two mortar assemblies, reefing cutters, disconnect assemblies, riser assemblies, and attaching hardware. The entire Landing System, with the exception of the aft bridle leg and disconnect assembly, is located in the rendezvous and recovery section of the spacecraft. Figure 12-2 illustrates the sequence of events from re-entry to impact in block diagram form. Figure 12-3 illustrates the electrical sequence of the landing system.

SYSTEM OPERATION

Prior to re-entry, the landing and postlanding common control electrical buses are armed by positioning the LANDING switch to ARM. This also applies power to the two barometric pressure switches for illumination of the 10.6K and 40K warning indicators.

When the altimeter indicates an altitude of 50,000 feet, the HI-ALT DROGUE switch is manually activated. The drogue switch energizes two single pyrotechnic cartridges in the drogue mortar. The drogue mortar deploys the reefed drogue parachute. The

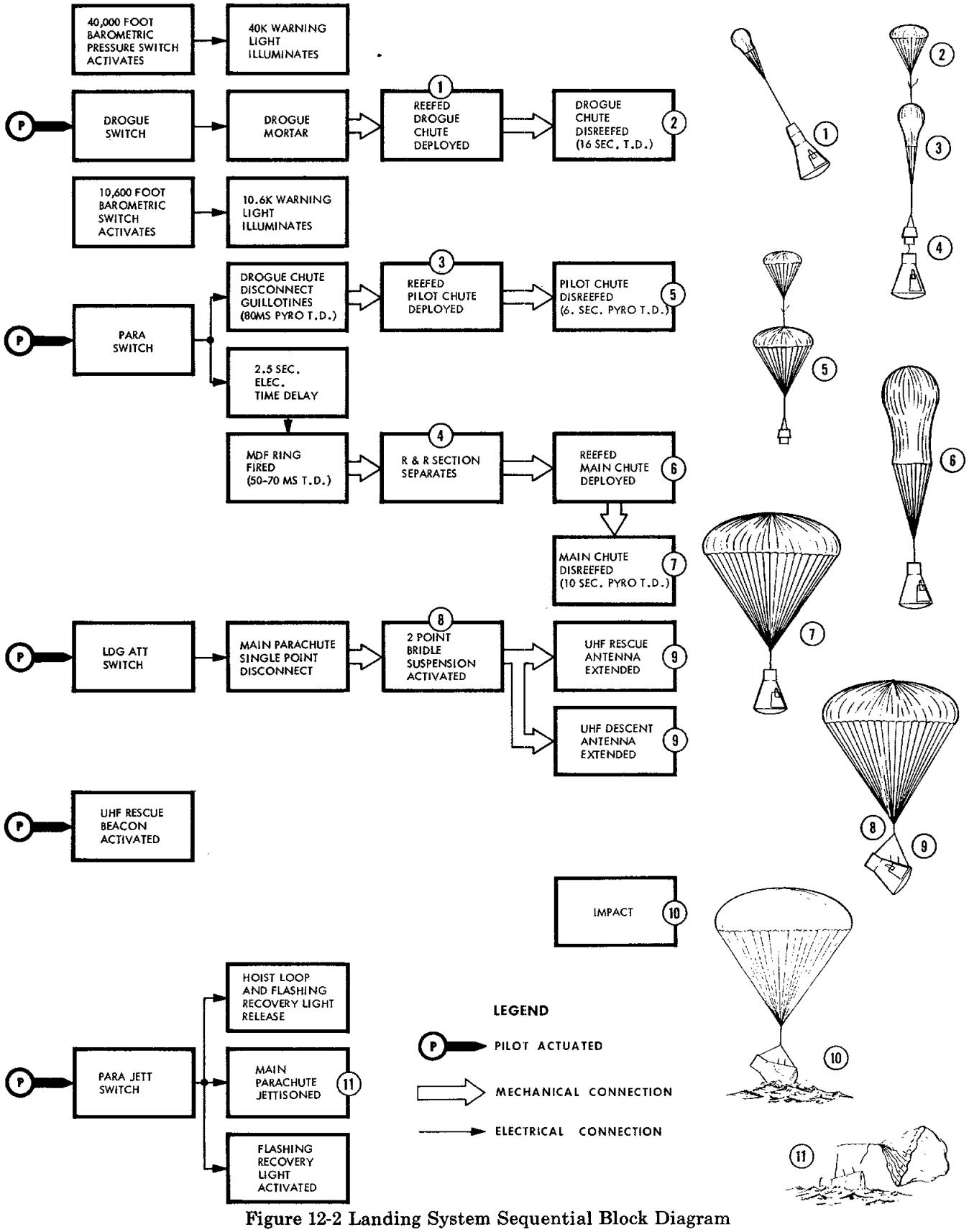


Figure 12-2 Landing System Sequential Block Diagram

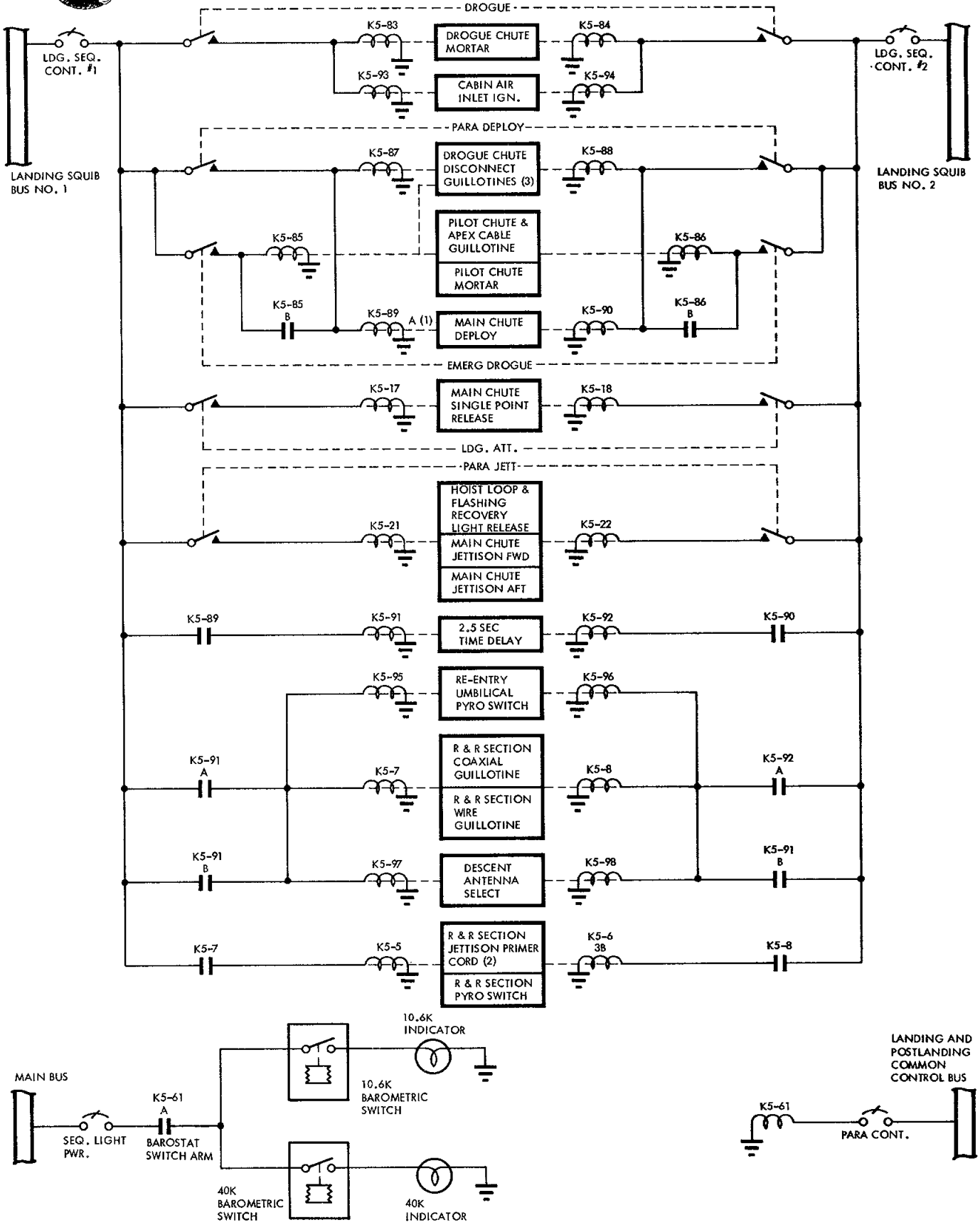


Figure 12-3 Landing System Schematic

parachute is reefed to limit the opening shock load. Sixteen seconds after deployment, two pyrotechnic reefing cutters disreef the drogue parachute. The drogue parachute stabilizes the re-entry module.

At approximately 10,600 feet, the PARA switch is activated. The PARA switch fires the three drogue cable guillotines and sets a 2.5 second time delay to the MDF ring detonators. After the drogue riser legs have been cut, the drogue parachute pulls away from the re-entry module extracting the pilot parachute from the pilot mortar tube with the apex line. When deployed, the pilot parachute is reefed to limit the initial shock load. Two lanyard initiated pyrotechnic reefing cutters disreef the pilot chute six seconds after deployment. 2.5 seconds after the pilot chute has been deployed, the MDF ring fires separating the rendezvous and recovery section from the landing vehicle. The pilot parachute functions to decelerate the re-entry module, remove the rendezvous and recovery section, and deploy the main parachute.

As the landing module falls away from the rendezvous and recovery section, the main parachute is deployed in a reefed condition. The main parachute is disreefed by three lanyard initiated pyrotechnic reefing cutters ten seconds after deployment. The two decelerations provided by the main parachute divide the retarding shock load. After the main parachute has been disreefed, the manually operated LDG ATT switch is actuated to change the single point suspension system to a two point suspension system. The two point suspension system provides a more favorable attitude for impact. As soon as the landing module contacts the ocean surface, the PARA JETT switch is activated. The PARA JETT switch energizes the forward and aft bridle disconnects releasing the main parachute. Upon completion of the

landing, the landing module is prepared for transmitting data and recovery information through the erected recovery antenna.

EMERGENCY OPERATION

In the event the drogue parachute does not deploy or deploys improperly, the PRE-MAIN 10.6K switch is actuated. The closure of this switch fires the three drogue cable guillotines, the apex line guillotine, and the pilot parachute mortar and starts the 2.5 second time delay to the MDF rings. The pilot mortar deploys the pilot parachute in a reefed condition. From this point, the emergency sequence of events is exactly the same as used during a normal landing. Figure 12-4 illustrates the emergency sequence of events in block diagram form, and Figure 12-5 illustrates the emergency deployment.

SYSTEM UNITS

DROGUE PARACHUTE ASSEMBLY

The drogue parachute assembly (Figure 12-6) stabilizes the re-entry module and deploys the pilot parachute. This assembly consists of an 8.3 ft diameter conical ribbon parachute with twelve 750-pound tensile strength suspension lines. A three legged riser assembly attaches the parachute assembly to the rendezvous and recovery section.

When initially deployed, the drogue chute is reefed to 43 percent of the parachute diameter in order to reduce the opening shock load. Sixteen seconds after deployment, two pyrotechnic reefing cutters disreef the drogue chute. Initiation of the PARA switch fires three cable guillotines located at the base of the three riser legs. As the drogue chute pulls away from the rendezvous and recovery section, an

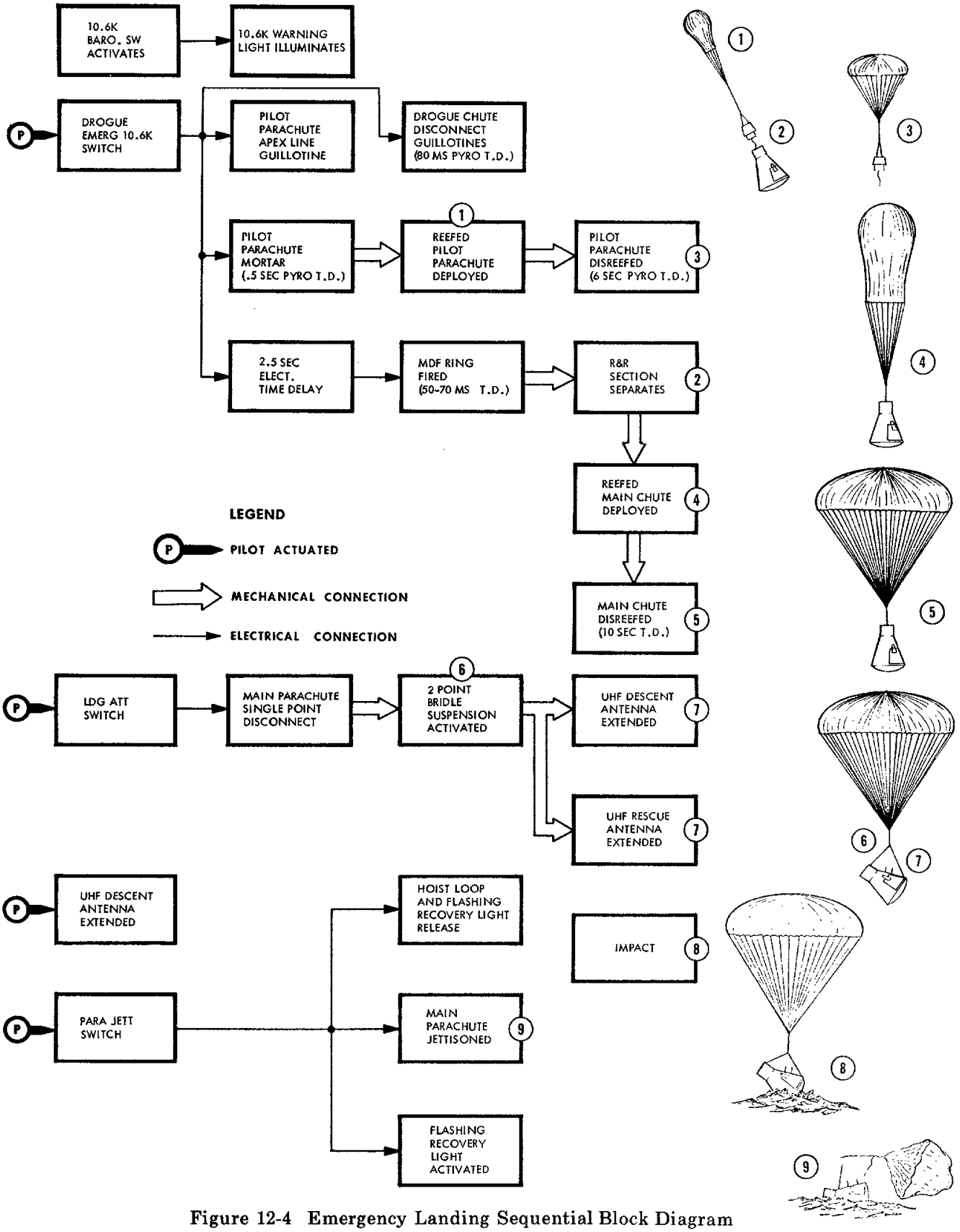


Figure 12-4 Emergency Landing Sequential Block Diagram

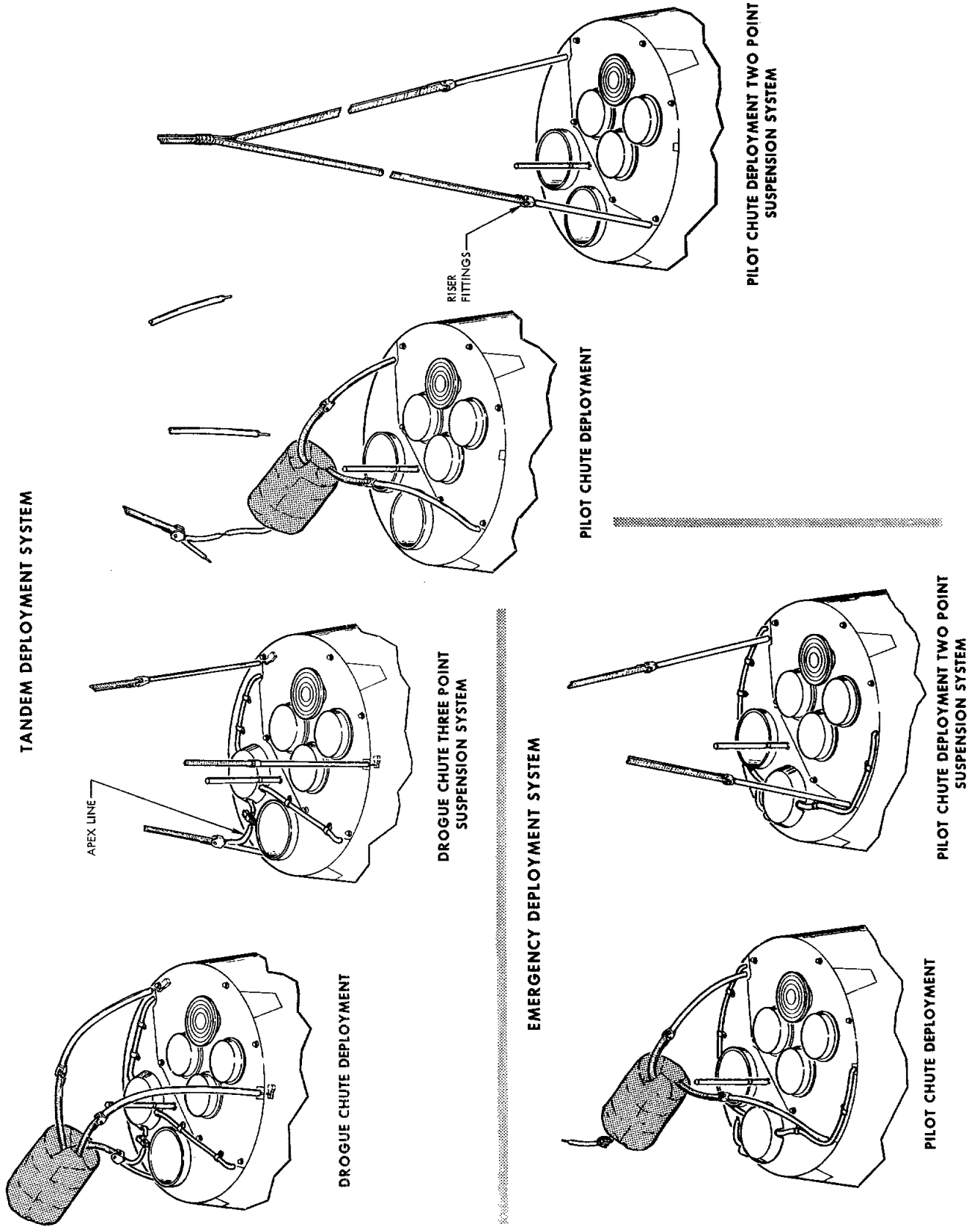


Figure 12-5 Tandem and Emergency Deployment System Operation

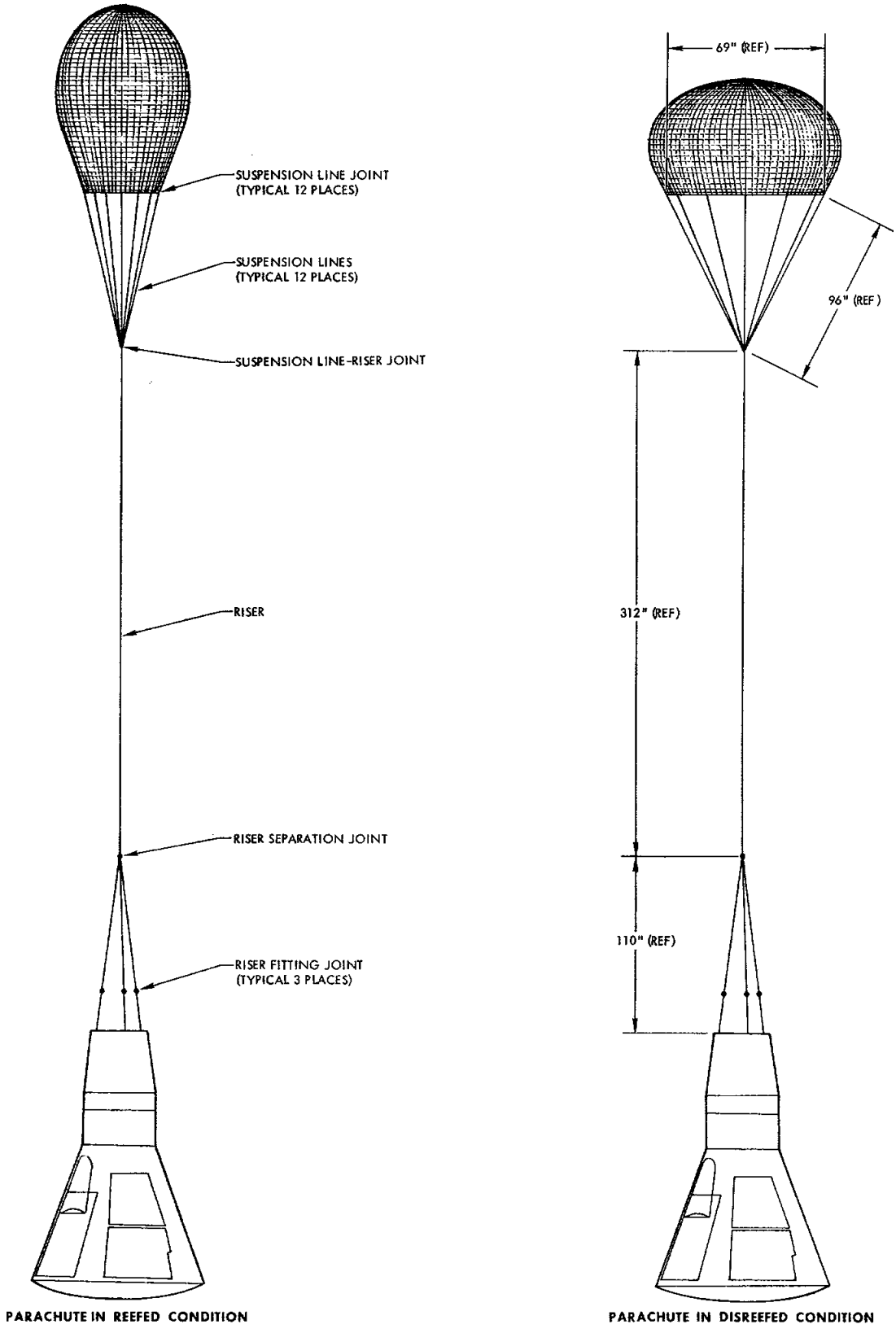


Figure 12-6 Drogue Parachute Assembly

apex line, which is attached to one of the riser legs, extracts the pilot parachute from the pilot mortar tube. The drogue parachute remains attached to the pilot parachute during the entire descent of the rendezvous and recovery section of the re-entry module.

Drogue Parachute Mortar Assembly

The drogue parachute mortar assembly stores and protects the drogue parachute during flight and deploys the drogue parachute when activated by the HI-ALT DROGUE switch. An insulated metal pan retains the parachute in the mortar tube which has a diameter of 7.15 inches and is 9.12 inches long. The breech assembly, located at the base of the mortar tube, contains two electrically actuated pyrotechnic cartridges and an orifice. The cartridges generate gases which enter the mortar tube through the orifice and eject the drogue parachute and sabot.

Drogue Mortar Sabot

The drogue mortar sabot is an aluminum cup located in the base of the mortar and functions to eject the drogue parachute with a piston like action. In order to insure the most effective ejection, the sabot is fastened to the base of the orifice by a frangible bolt, and an O-ring, located near the base of the sabot which contacts the inner wall of the mortar tube to prevent any escape of gases generated by the two pyrotechnic cartridges. When enough pressure to break the frangible bolt has built up, the sabot and parachute are expelled from the mortar tube. After ejection, the sabot remains attached to the parachute bag and aids in stripping the bag from the parachute.

Drogue Parachute Deployment Bag

The drogue parachute deployment bag protects the drogue parachute during deployment

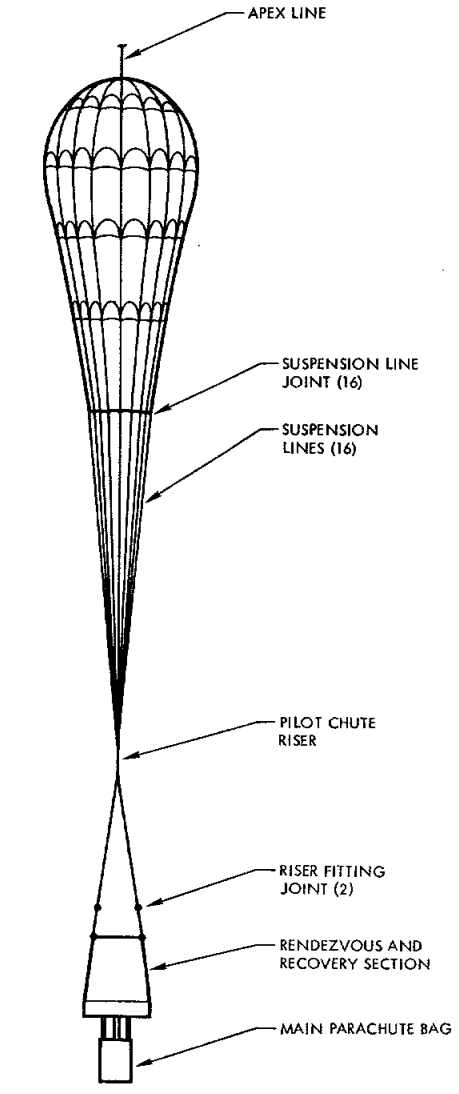
and allows for an orderly deployment of the parachute. The bag is fabricated from cotton sateen and nylon. A 0.35 pound aluminum plate, sewn into the top of the bag, aids in stripping the bag from the canopy during the deployment.

PILOT PARACHUTE ASSEMBLY

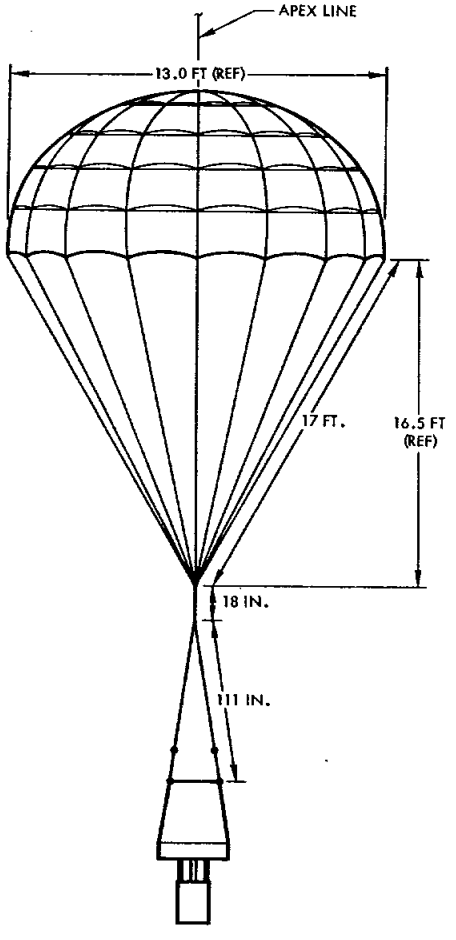
The pilot parachute assembly (Figure 12-7) decelerates the re-entry module and removes the rendezvous and recovery section from the landing module which results in the deployment of the main parachute. During flight, the pilot parachute assembly is stowed in the pilot mortar tube. The 18.3 foot diameter canopy is of the ringsail type having 16 gores and fabricated from 1.1 and 2.25 ounce per square yard nylon. Sixteen nylon cord suspension lines, which are 17 foot long and have a tensile strength of 550 pounds each, attach the canopy to the riser assembly. A 10.7 foot long split riser, constructed of four layers of 2600 pound tensile strength dacron webbing, holds the pilot parachute assembly to the rendezvous and recovery section of the spacecraft. When initially deployed, the pilot parachute is reefed to 11.5 percent in order to limit the opening shock load to 3000 pounds. Two pyrotechnic reefing cutters disreef the parachute 6 seconds after deployment. The pilot parachute remains attached to the rendezvous and recovery section throughout the entire descent.

Pilot Parachute Mortar Assembly

The pilot parachute mortar assembly is similar in design and operation to the drogue parachute mortar assembly. During normal operation of the Landing System, this assembly serves only to store and protect the pilot parachute. In the event of a failure in the deployment of the drogue parachute, the pilot parachute mortar can be activated to deploy the pilot parachute by initiation of the PRE-MAIN 10.6K



PARACHUTE IN REEFED CONDITION



PARACHUTE IN DISREEFED CONDITION

Figure 12-7 **Pilot Parachute Assembly**

switch. Actuation of the PRE-MAIN 10.6K switch fires the three drogue cable guillotines, the apex line guillotine, and the pilot parachute mortar. After the pilot parachute has been deployed, the landing is completed through the normal sequence of events. Figure 12-5 illustrates the pilot parachute deployment.

Pilot Mortar Sabot

The pilot mortar sabot functions are the same as those of the drogue mortar sabot. Refer to the description of the Drogue Mortar Sabot.

Pilot Parachute Deployment Bag

The pilot parachute deployment bag is similar to the drogue parachute deployment bag in design and use, except for the bag handles attached to the apex line for extraction by the drogue parachute.

MAIN PARACHUTE AND RISER ASSEMBLY

The main parachute (Figure 12-8 and 12-9) is of the ringsail type with a diameter of 84.2 feet. The nylon canopy has seventy-two gores alternating in colors of international orange and white. Seventy-two suspension lines are attached to eight legs of a single integral riser. Each suspension line has a tensile strength of 550 pounds. The 3.25 foot integral riser consists of eight layers of 5,500 pound tensile strength nylon webbing. The canopy is fabricated from 1.1 and 2.25 ounce per square yard nylon and is designed to operate at a dynamic pressure of 120 pounds per square foot. However, by reefing the main parachute, a maximum load of 16,000 pounds is experienced at deployment. When initially deployed, the parachute is reefed to 10.5 percent. The disreefed main parachute allows a maximum rate of descent of 31.6 feet per second for a module weight of 4,400 pounds.

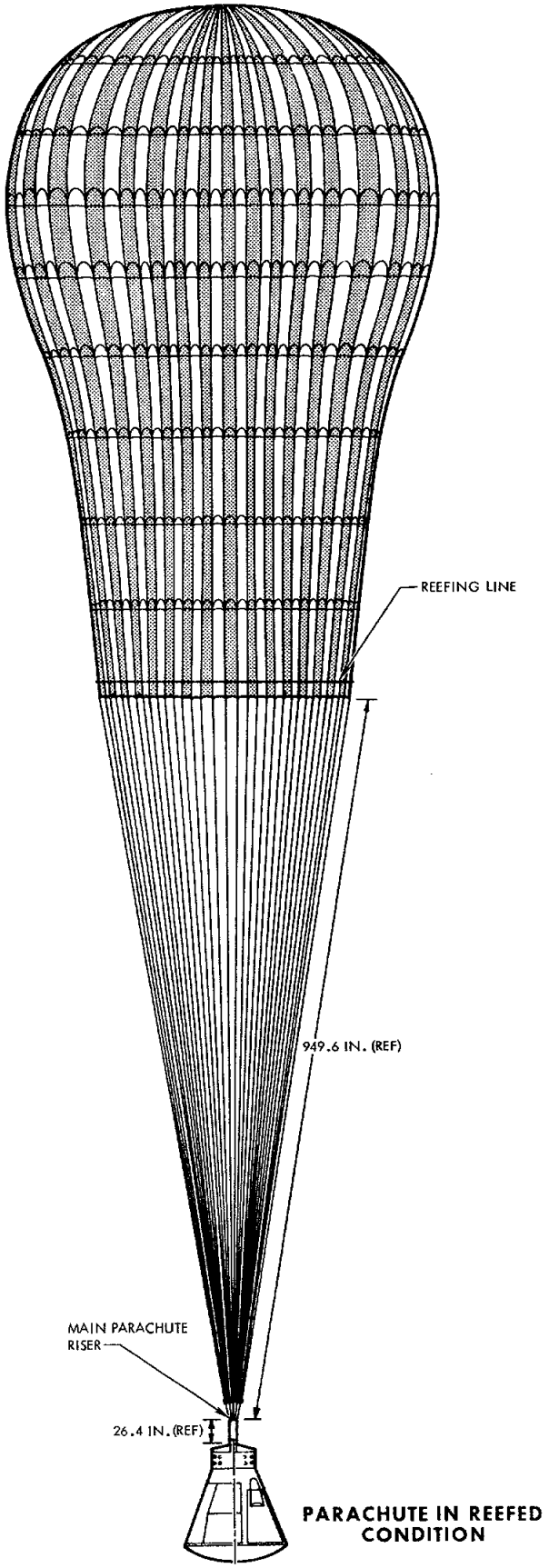


Figure 12-8 Main Parachute and Single Point Suspension System

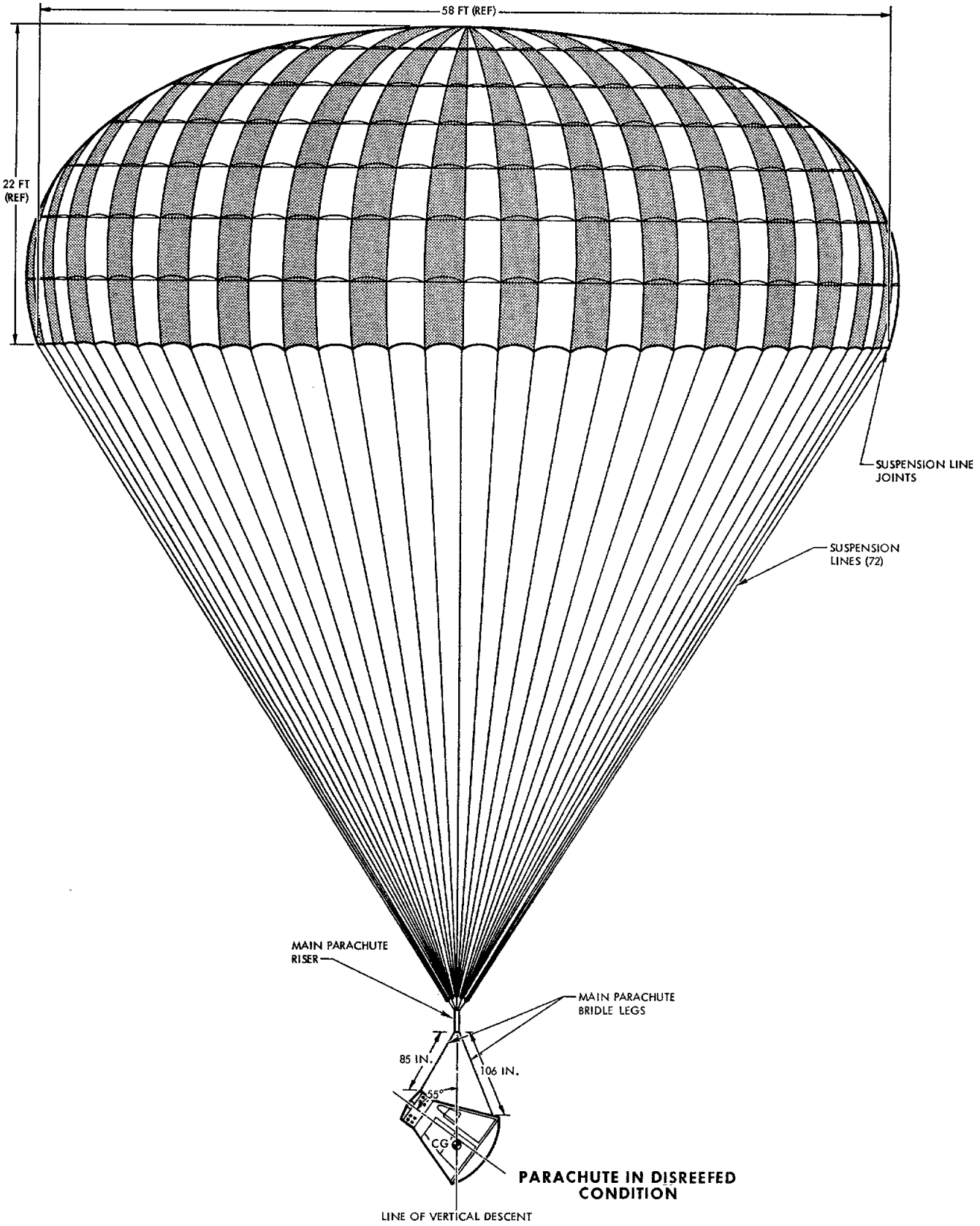


Figure 12-9 Main Parachute and Two Point Suspension System

Main Parachute Deployment Bag and Container Assembly

The main parachute deployment bag and container assembly (Figure 12-1) stows the main parachute. This assembly is located in the aft end of the rendezvous and recovery section of the spacecraft. The deployment bag is fabricated from a cotton sateen material reinforced with nylon webbing. In order to insure a full and orderly deployment of the main parachute, the suspension lines must be stretched out prior to the release of the canopy. Therefore, transverse locking flaps are incorporated in the bag to separate the canopy from the suspension lines. Four restraining straps hold the deployment bag in the container until deployment.

The main parachute container is 22.25 inches in diameter and 21.32 inches long. The container is closed on the forward end and is secured to the rendezvous and recovery section by four vertical reinforcing brackets. At deployment, the restraining straps of the deployment bag are unlocked, the risers and suspension lines are extended, and the canopy is pulled from the deployment bag. The deployment bag remains attached to the container by four bag handles.

Main Parachute Bridle Assembly

The main parachute bridle assembly (Figure 12-9) provides a two point suspension system in order to achieve the optimum attitude for a water landing. Two separate bridle straps constitute the main parachute bridle assembly. The forward bridle strap is an 85 inch long nylon strap with a looped end connected to the forward bridle disconnect. Prior to single point release, the forward bridle is stowed in the bridle tray (Figure 12-10). The aft bridle is 106 inches long and connects to the aft disconnect which is located immediately forward of the single point

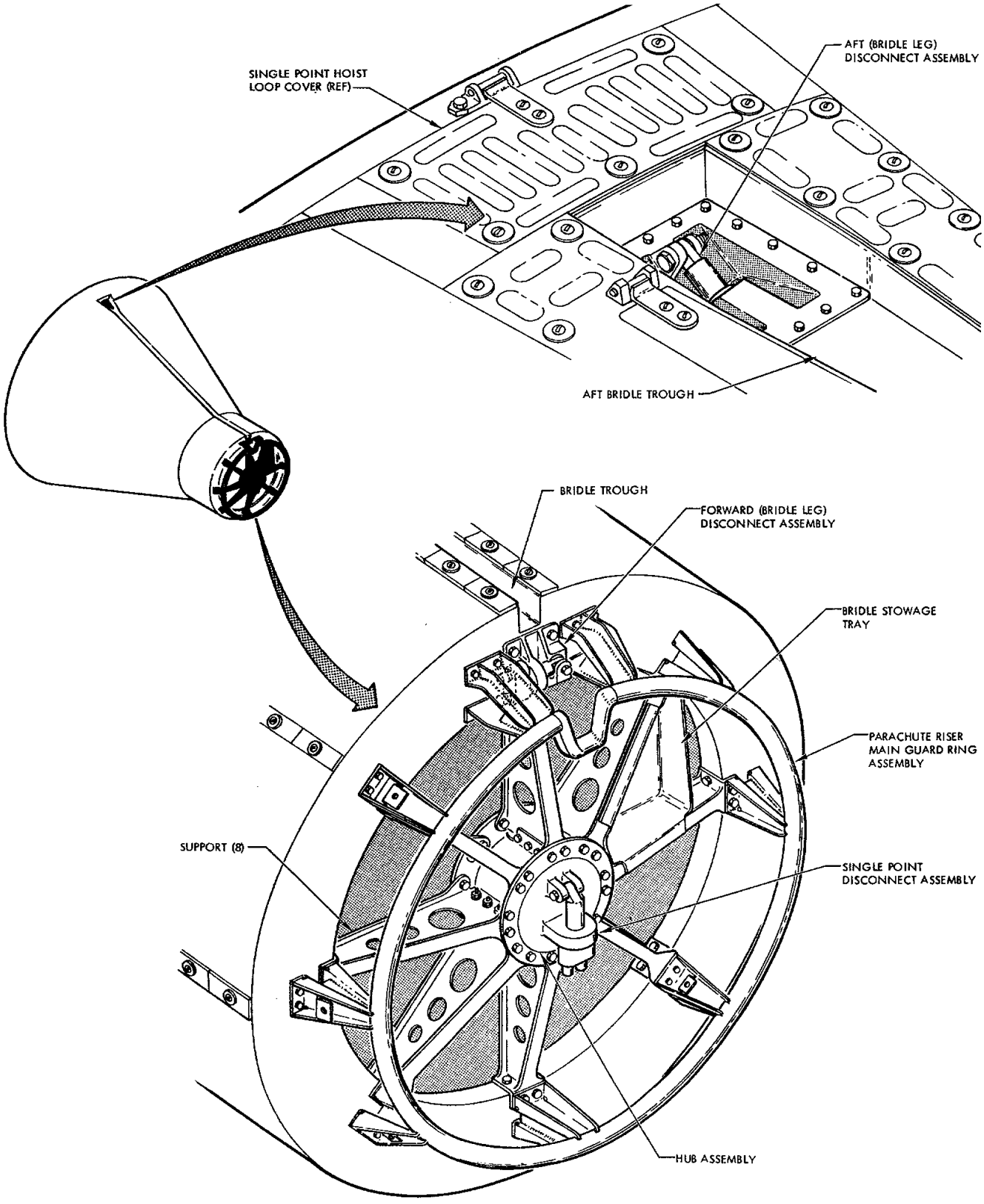
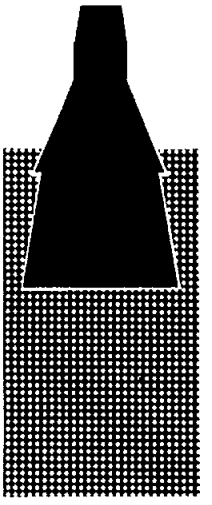


Figure 12-10 Main Parachute Support Assembly

hoist loop (Figure 12-10). Constructed of heat resistant nylon, the aft bridle is stowed in a trough that extends from the front of the Re-entry Control System section to the aft disconnect during flight. An insulating cover shields the aft strap in the cable trough until the single point suspension is released, at which time the bridle leg tears through the insulation.

Main Parachute Release

Upon landing in the water, the main parachute is released from the landing module by activation of the PARA JETT switch. This initiates the forward and aft disconnect pyrotechnics and allows the chute to pull away from the landing module.



DOCKING SYSTEM

Section XIII

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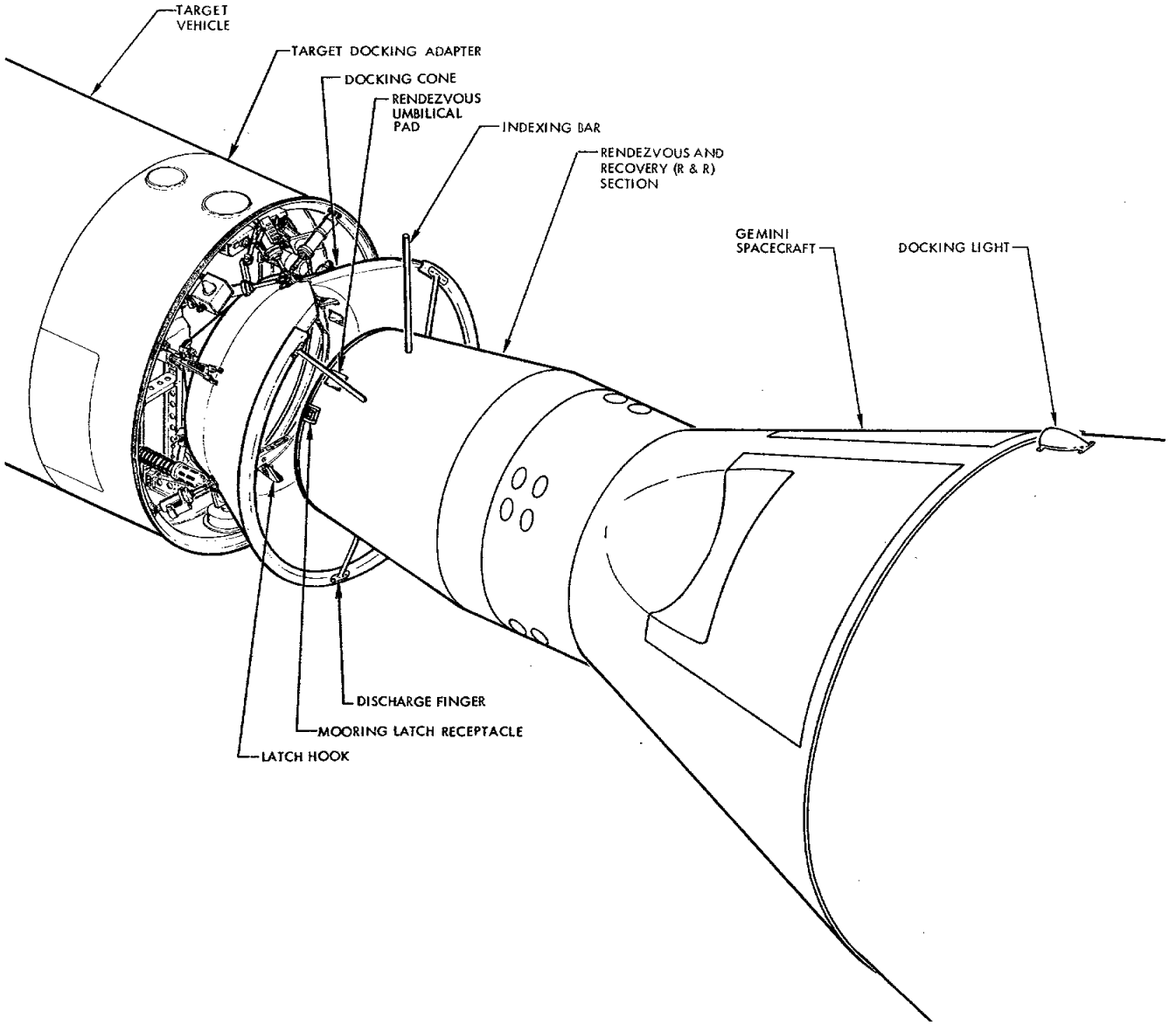


Figure 13-1 Docking System

SECTION XIII DOCKING SYSTEM

SYSTEM DESCRIPTION

The Docking System (Figure 13-1) is used on spacecraft 8 through 12 and utilizes two major assemblies: the spacecraft Rendezvous and Recovery (R & R) section and the Target Docking Adapter (TDA) assembly. These assemblies permit the spacecraft to mate with the target vehicle during orbital flight. The TDA assembly may be mounted on either the Agena Target Vehicle (ATV) or the Augmented Target Docking Adapter (ATDA). The docking cone receives the spacecraft R & R section during docking. Electromechanical devices in the spacecraft and the docking adapter control the docking operations automatically or as directed by the pilots or the ground control stations. For convenience, these major assemblies are considered separately. In this section, details of the R & R section of the spacecraft, approach and mooring, rigidizing and unrigidizing to the target vehicle are presented. Details of the TDA are presented in the Target Docking Adapter section.

RENDEZVOUS AND RECOVERY SECTION

The R & R section of the spacecraft (Figure 13-2) contains the following Docking System units: the support structure, a nose fairing, an index bar, 3 mooring latch receptacles, 3 latch-receptacle covers, an umbilical connector (receptacle), two rendezvous umbilical connectors (pads), pyrotechnic cable cutters and ejection devices. Other components of the R & R section are not concerned with docking. The main structure is secured to the spacecraft and serves as a mounting structure for other units.

The nose fairing is rigidly but temporarily fastened to the forward end of the

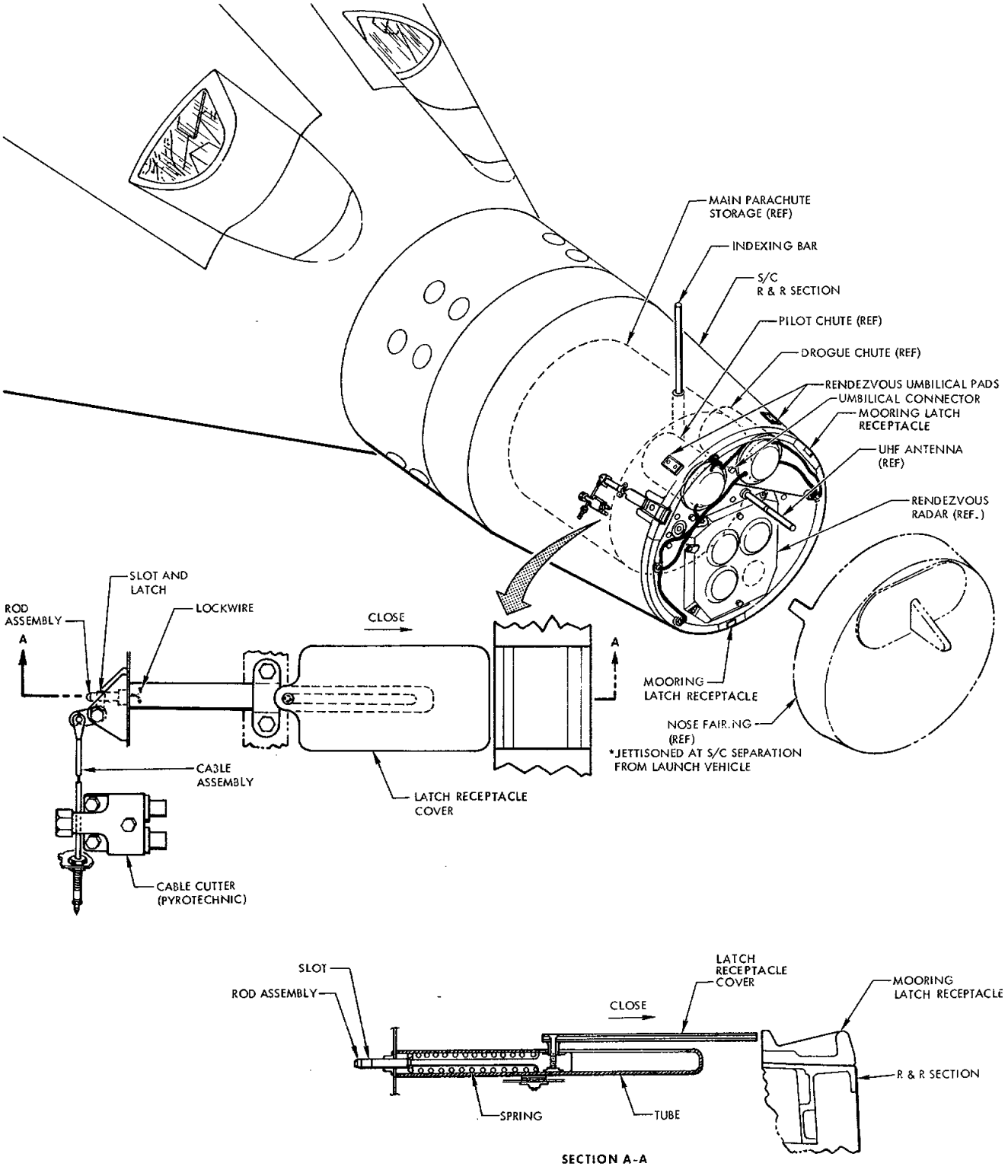


Figure 13-2 R & R Section, S/C

R & R section. The fairing is jettisoned at spacecraft separation from the launch vehicle. A pyrotechnic igniter severs a shear pin and swings the fairing clear of the spacecraft. The docking latch receptacles are exposed as the fairing is jettisoned. The index bar is so located that it coincides with the slot in the docking cone when properly aligned. The bar is extended pyrotechnically by the pilots and remains extended until jettisoned at retrograde adapter separation. The pilots use the bar to align the spacecraft with the docking cone. The pilots use another set of igniters to jettison the bar. The bar can neither be extended or jettisoned prior to nose fairing jettison.

The three latch receptacles are equally spaced around the forward perimeter of the R & R section. The lower latch serves as a pivot for the nose fairing when it is jettisoned. The exposed receptacles latch with the mooring latch hooks on the docking cone as the two units come in contact during the docking maneuver. The latch receptacles will remain latched until unlatched automatically by the separation sequence. The latch receptacles are jettisoned just before re-entry or during an emergency. The cavities left by the jettisoned receptacles are covered by pyrotechnically released covers before re-entry. Relays prevent these operations until after the nose fairing has been jettisoned.

The Rendezvous Radar is located in the front part of the R & R section and consists of the transmitter-receiver unit plus 3 receiving antennas and one transmitting antenna. The radar (and encoder) are used to control the target vehicle before docking. The pilots can initiate a number of encoded commands via the Command Link System to accomplish this.

The rendezvous umbilical pads mate with the rendezvous umbilical lever assemblies on the docking cone, at the same time the latch hooks engage the latch receptacles.

The main umbilical receptacle mates with its umbilical plug during the rigidizing sequence of the docking maneuver. The umbilical connection provides direct hard-line control of the target vehicle. This control includes simple switching and encoded commands.

Pyrotechnic igniters are used to extend the index bar, jettison the index bar and the mooring latches, jettison the nose fairing and release the latch covers. The pyrotechnic igniters are fired electrically from the crew station.

APPROACH AND MOORING SEQUENCE

The approach and mooring sequence is initiated by a ground station command to the target vehicle. Using the Digital Command System, the ground station will turn on electrical power and radar transponder circuits and extend the dipole antenna on the TDA. This allows the pilots of the spacecraft to make contact with the target vehicle using the Rendezvous Radar System.

The pilots will begin tracking and control the target vehicle at approximately 100 nautical miles separation. When the separation is about 20 nautical miles, the pilots can use the Command Link System to unrigidize the docking cone and turn on the acquisition lights. At about 500 feet separation, the pilots use the Command Link to turn on the approach lights and status display panel lights. The pilots will extend the index bar pyrotechnically and decrease the spacecraft closing rate to 0.7 feet per second. Later, the pilots turn on the spacecraft docking light. As the target vehicle is approached, the pilots use spacecraft

thrusters and the index bar to align the spacecraft with the docking cone.

When proper alignment is achieved the pilots begin the mooring sequence. The mooring sequence is completed when the spacecraft is firmly connected to the docking cone in the unrigidized position. First contact between the R & R section and the docking cone is with three electrical discharge fingers protruding from the docking cone. These discharge fingers are so wired that any static charge between the spacecraft and the target vehicle will be dissipated at a controlled rate. As the pilots thrust the R & R section further into the docking cone the spring loaded mooring latch hooks are pushed aside, then drop behind the latch receptacles to create the latched condition. At the same time the two rendezvous umbilical connectors are mated and provide direct hardline control to the TDA mooring drive system.

RIGIDIZING AND UNRIGIDIZING SEQUENCE

The rigidize sequence and the unrigidize sequence are automatic once they are initiated. The rigidize sequence is initiated when the spacecraft R & R section contacts the latch hooks, pushes them aside and then allows the hooks to lock on to the latch receptacles. Latch engagement is sensed by three limit switches, one at each latch position. The sensing switches route target vehicle electrical power through the rigidize limit switches to the rigidized power control relay. The relay energizes, connecting target vehicle electrical power to the mooring drive system motor. When the system is rigidized, the rigidized limit switches open, de-energizing the rigidized power control relay and stopping the motor. The main umbilical plug is extended during the rigidize sequence.

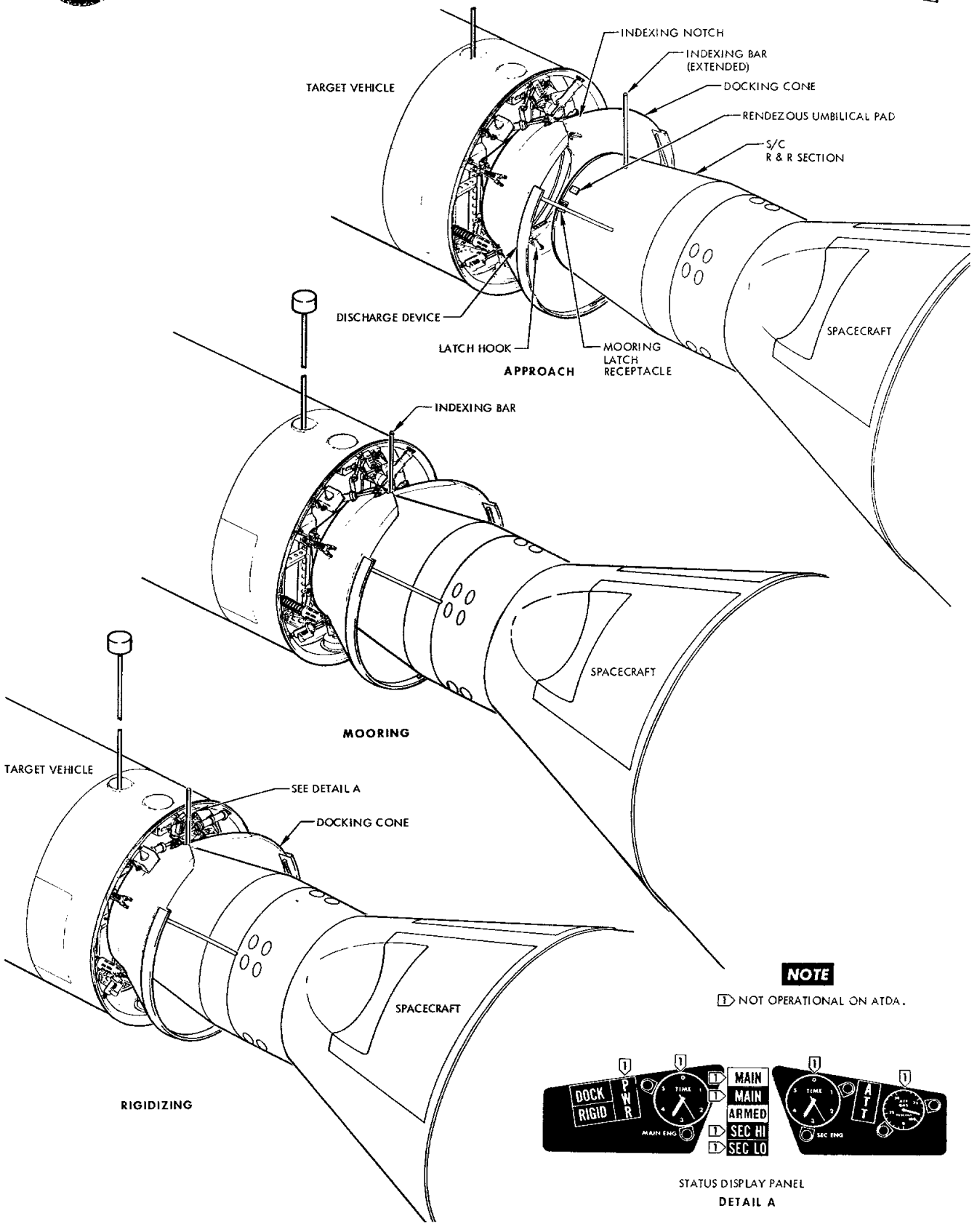


Figure 13-3 Docking System Operation

The unrigidize sequence is automatic once it is initiated by either ground command or by the pilots. When the mooring drive system is unrigidized, the unrigidized limit switches route target vehicle electrical power to the latch actuator. When the mechanism is unlatched, the unlatch sensing switch routes target vehicle electrical power to the separation timer. The timer, after a 30-second delay, applies power to the reset side of the latch actuator through the latch reset control relay and the latch reset relay. The latch hooks are reset so that the docking maneuver can be repeated. Normally the cone does not rigidize after reset of the latch hooks but if the pilot does not back away within the 30-second delay before reset, the cone will rigidize. The unrigidize sequence would then have to be repeated.

SYSTEM OPERATION

The Docking System operation (Figure 13-3) illustrates the sequence in which docking is accomplished. Control of the Docking System is primarily electrical but pyrotechnic devices and mechanical sensing switches are integrated into the system. The Docking System electrical block diagram (Figure 13-4) links together various assemblies used in the docking operation. The electrical system provides the capability for the pilots or ground station to control the docking operation.

The operation of the Docking System consists of three basic functions: (1) Docking sequence, (2) Maneuvering sequence, (3) Separation sequence.

DOCKING SEQUENCE

The docking sequence consists of approach, mooring, rigidizing and main umbilical connection.

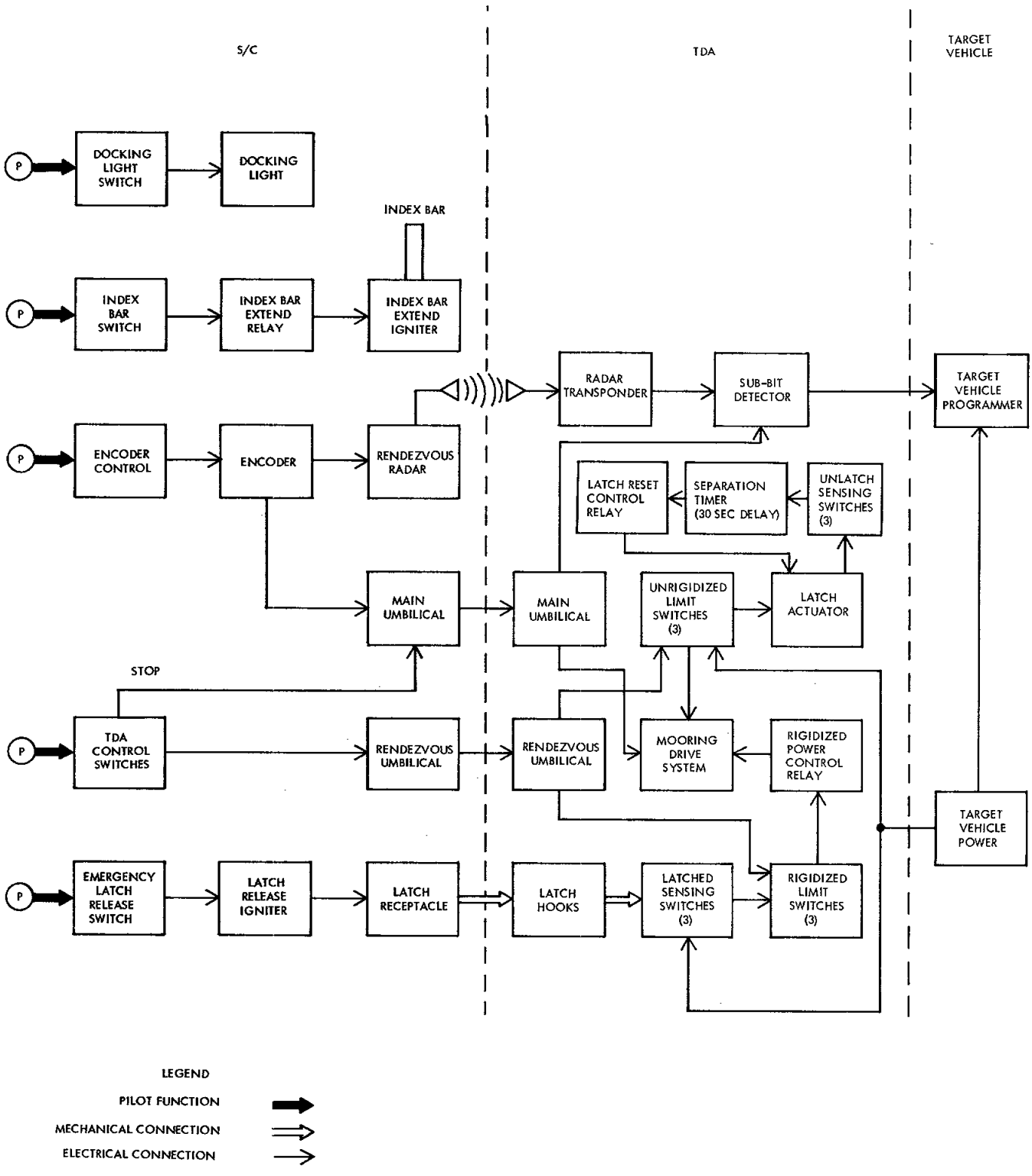


Figure 13-4 Docking System Electrical Block Diagram

Approach

The pilots in coordination with the ground station control the Docking System (Figure 13-3 and Figure 13-4).

During the approach phase, the Digital Command System is used to switch on the target vehicle command circuits, the docking adapter radar transponder and to extend the docking adapter dipole antenna boom. With this accomplished, the pilots and the ground stations can control other docking adapter/target vehicle functions via the Command Link System. Typical commands sent during the approach phase are UNRIGIDIZE and ACQUISITION LIGHTS ON at about 20 nautical miles separation; APPROACH LIGHTS ON, STATUS DISPLAY ON and STATUS DISPLAY BRIGHT at near approach. When the DOCK light on the status display panel is illuminated, the system is unrigidized and the latch hooks in docking position. If the light is not on, the UNRIGIDIZE command is given again, either by the pilots or the ground station. In addition, the pilots operate the following switches during the approach phase: DOCK LT (or EXT LT on spacecraft 8 through 12), BUS ARM and INDEX EXTEND/POD EJECT. In the ON position, the DOCK LT (or EXT LT) switch applies main bus power through the CABIN LIGHTS circuit breaker to the dock light. In the INDEX EXTEND position, the INDEX EXTEND/POD EJECT switch applies OAMS squib bus 1 and 2 power to the index bar extend relay coils through the BUS ARM switch. The energized index bar extend relays apply docking squib bus power to the extended igniters on the index bar assembly. The pilots use the extended index bar to align the spacecraft with the docking cone.

Mooring

As proper alignment is achieved, the pilots maneuver the spacecraft to slide the indexing bar into the slot (at the tip of the V-notch) in the docking cone (Figure

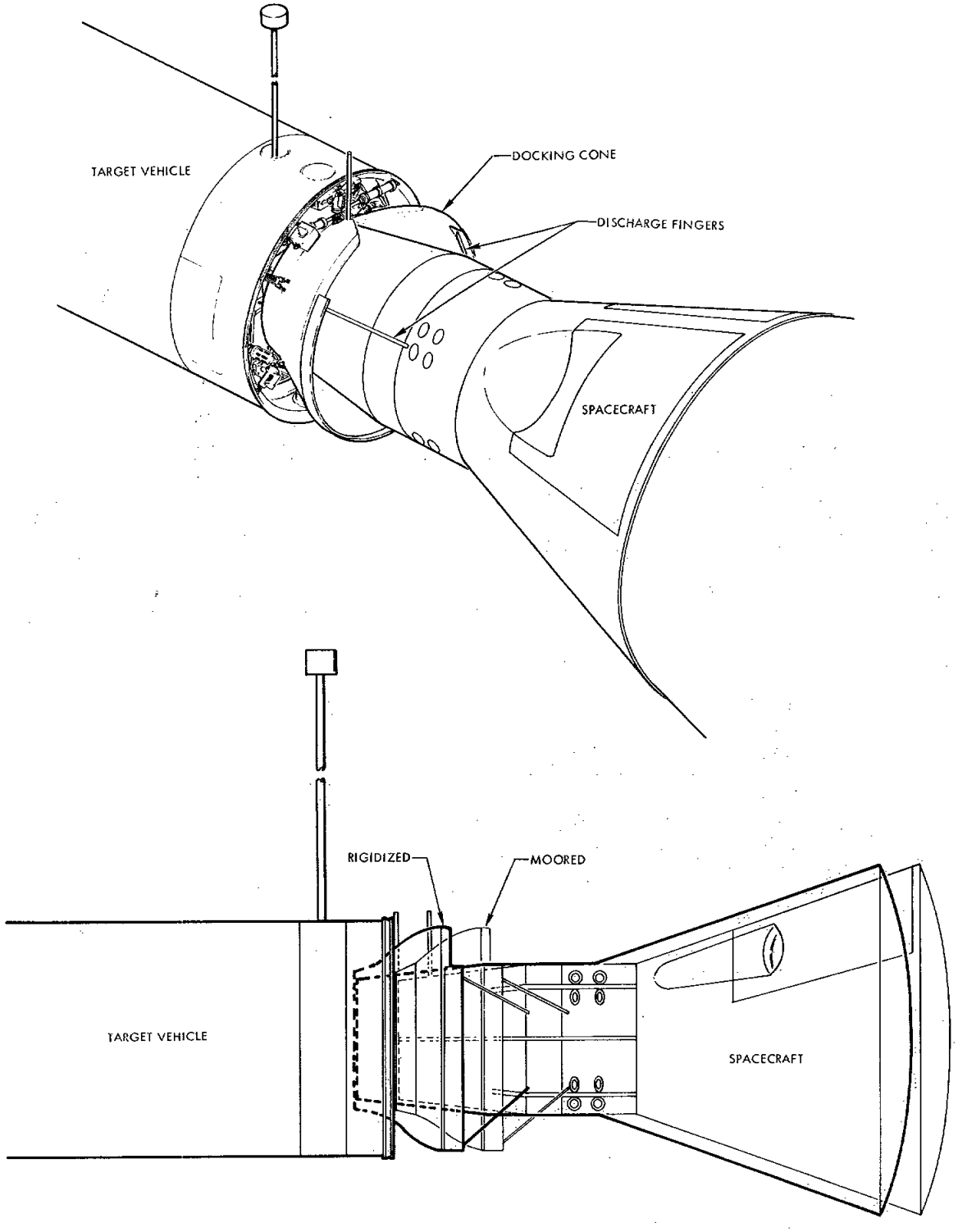


Figure 13-5 Rigidizing Operation

13-3). The pilots increase the forward thrust as the spacecraft enters the docking cone and makes contact with the electrical discharge fingers. This temporary thrust increase provides enough momentum for the spacecraft R & R section to actuate the spring loaded mooring latch hooks on the interior of the docking cone. Actuation of the latch hooks captivates the spacecraft R & R section in the docking cone and automatically initiates the rigidize sequence. At the same time the two rendezvous umbilical connectors are mated and provide direct hardline control to the TDA mooring drive system. The completed mooring operation connects the spacecraft firmly to the docking cone.

Rigidizing

Rigidizing (Figure 13-5) is the process of pulling the docking cone in toward the adapter main structure until it reaches a firm seating against the pads (part of the adapter main structure). This operation is required because the cone dampers form only a flexible connection between the cone and the adapter structure. Maneuvering the spacecraft/target vehicle combination is impractical under unrigidized conditions.

The docking cone is rigidized as soon as the mooring operation is completed. As mooring is completed, three limit switches mounted on the TDA latch assemblies automatically activate the mooring drive (rigidizing) mechanism motor, which in turn, drives the rigidizing linkages through the flexible drive cables, gear boxes and drive arms. The drive arms apply tension to the rigidizing linkages connected to the target docking cone. As the linkages are retracted, the docking cone is pulled toward the docking adapter main structure and the main umbilical plug is extended. When the docking cone bottoms on the adapter pads, the mooring drive mechanism stops and the RIGID light on the status display panel illuminates. The

firm connection between the spacecraft and the target docking vehicle enables joint maneuvers with the two vehicles.

The rigidizing sequence can also be initiated by the pilots or the ground stations as back-up to the automatic system. In this case, the pilots have two ways to rigidize the docking cone. (1) Position the RIGID-OFF-STOP switch, on the main instrument panel, to RIGID. This by-passes the dock latch limit switches and applies power to the mooring drive motor. (2) Send the RIGIDIZE command to the target vehicle command programmer via the Command Link System, which routes power to the mooring drive motor. The remainder of the sequence is the same as when initiated automatically.

If the rigidize limit switch should fail to remove power from the mooring drive motor once the docking cone is rigidized, the pilot can remove power by placing the RIGID-OFF-STOP switch to the STOP position.

Main Umbilical Connection

The main umbilical plug in the docking adapter assembly is actuated by the mooring drive mechanism. In the rigidized condition, the mooring drive mechanism extends the plug out of its casing so that it mates with the receptacle on the spacecraft R & R section. The plug remains extended and mated until the cone is unrigidized.

MANEUVERING SEQUENCE

The maneuvering sequence is subject to considerable variation except for a few common operations. Most of the maneuvering sequence is subject to individual mission requirements; however, the following operations are typical on most docking missions with the Agena Target Vehicle (ATV).

When the docking cone is rigidized, the pilots can use the Command Link System to turn off the acquisition lights, the approach lights and the target vehicle radar transponder. In addition, they will dim the status display lights and send maneuvering commands to the ATV. When the pilots have the ENGINE switch in the ARM position, the target vehicle propulsion systems can be used for attitude changes, orbit adjustments, etc. At the end of a maneuvering sequence, the ENGINE switch is set to STOP and the Command Link System is used to return the target vehicle to the pre-docking status if desired. A command via the Digital Command System may be substituted for any pilot command if necessary. However, ground command cannot override pilot-switched functions such as engine arm.

No maneuvering sequence has been considered when docked with the ATDA.

SEPARATION SEQUENCE

As the maneuvering phase ends, the pilots will perform the separation sequence. Existing conditions at the time will determine whether the mode is normal or emergency (Figure 13-6).

Normal

The normal separation sequence is initiated by a command to the target vehicle command programmer to unrigidize the docking cone. The pilots or ground control may initiate this command. Receipt of this command by the target vehicle command programmer starts the mooring drive mechanism motor and lights the MSG ACPT light in the crew station. When the cone is unrigidized, the mooring drive motor stops and the latch mechanism actuator is energized to retract the latch hooks and release the spacecraft. The unlatch limit switch, within the latch mechanism, stops the actuator and starts a 30-second time delay relay. The 30-second time delay normally allows the pilots sufficient time to back away from the docking cone

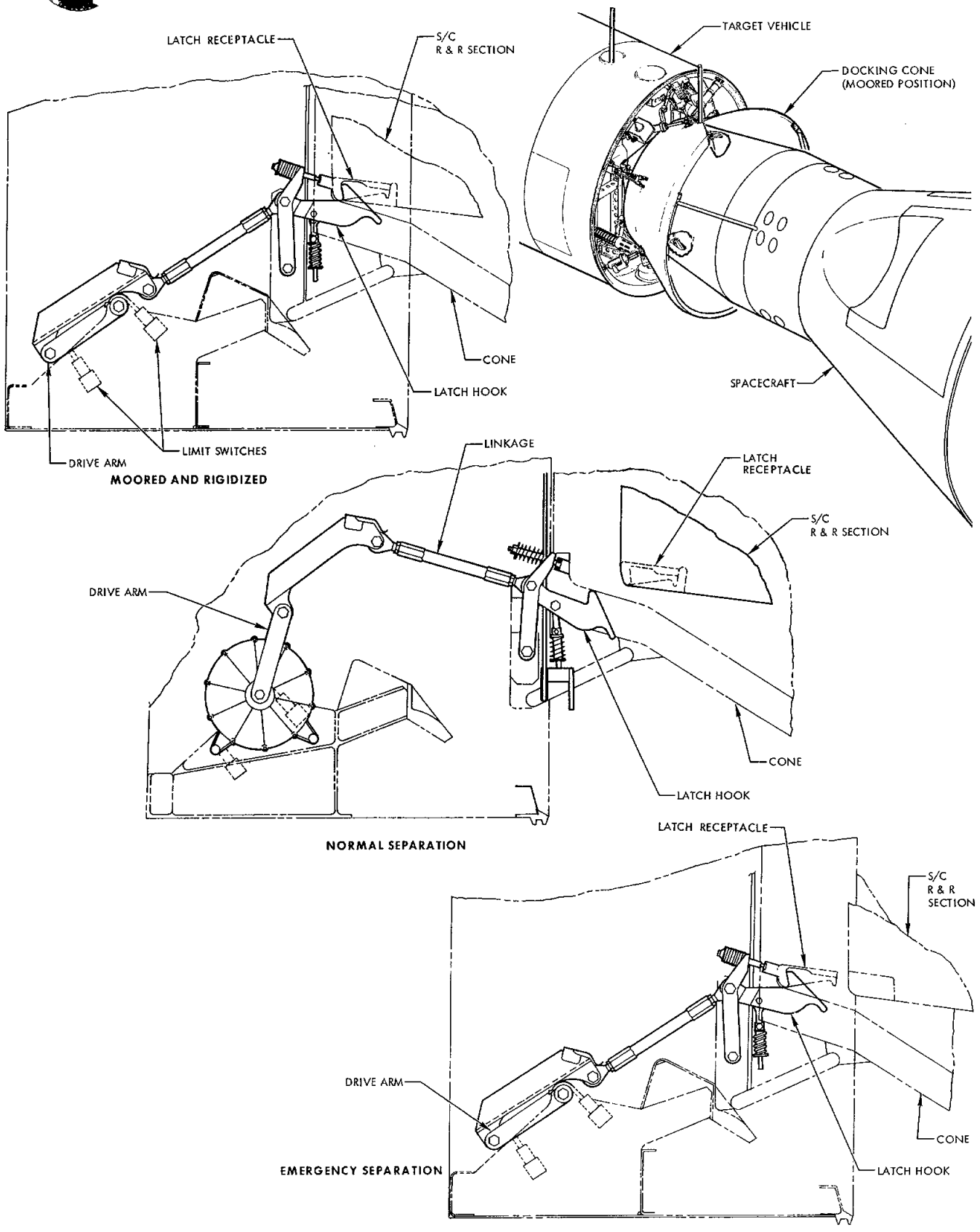


Figure 13-6 Separation Sequence

before the latch hooks are reset to the mooring position. At retrograde adapter jettison, the pilots will jettison the index bar and the docking latches and release the docking latch covers (Figure 13-2).

An exception to the normal sequence as presented can arise if the pilots should fail to thrust clear of the docking cone during the 30-second time delay before the latch hooks are automatically reset. In this case, the rigidize sequence is initiated automatically after the latch-reset function. The pilots would then have to send the UNRIGIDIZE command a second time. The pilots will then have another 30-second period in which to thrust free of the docking cone. These operations may be repeated several times if necessary.

A backup operation to the normal sequence may be initiated by the pilots if the Command Link System should fail. In this case, the pilots would place the UNDOCK-OFF switch on the main instrument panel to the UNDOCK position. This applies power to the mooring drive mechanism, which starts the unrigidized sequence. The remainder of the operation is the same as the normal sequence.

Emergency

If the pilots cannot thrust free of the docking cone by using the normal separation sequence or if emergency conditions exist, they may employ the emergency separation sequence (Figure 13-6) to rapidly separate the spacecraft from the target vehicle. The pilots initiate the emergency separation sequence by pressing an emergency release switch in the crew station. This switch fires pyrotechnic igniters at each of the three docking latch locations, separating the latch receptacle from the spacecraft. The pilots then fire thrusters to back out of the docking cone. The latch covers are released and the index bar jettisoned at

retrograde adapter jettison. The emergency separation sequence permanently removes the docking capabilities of the spacecraft.

SYSTEM UNITS

The Docking System units are those units which are concerned solely with docking and in addition, are not part of another system. Those units which have docking functions but are part of other systems are not discussed in detail in this section. Some units which are mounted on the R & R section of the spacecraft are not concerned with docking at all and therefore are not discussed.

MOORING LATCH RECEPTACLES

The mooring latch receptacles (Figure 13-2) are fixed units, shaped to mate with the latch hooks on the docking cone. They are securely but temporarily fastened to the main structure of the R & R section. Pyrotechnic igniters are electrically fired to separate the latch receptacles loose from the R & R section. When docking, the receptacles easily push the latch hooks aside as the spacecraft enters the docking cone and then let the hooks drop behind the catches to create the latched condition. The latch receptacles are protected by the nose fairing during the boost phase.

MOORING LATCH COVER ASSEMBLIES

Each mooring latch cover assembly (Figure 13-2) consists of a cover, a spring assembly, a shaft and latch assembly, a cable and a pyrotechnic cable cutter. The shaft, latch and cable keep the spring assembly compressed and the cover retracted until the cable is cut by the pyrotechnic cable cutter (guillotine). The guillotines are electrically fired. As each guillotine cuts it associated cable, the latch drops down, allowing the spring assembly to slide the cover over the cavity left by the jettisoned receptacles. The covers prevent overheating of

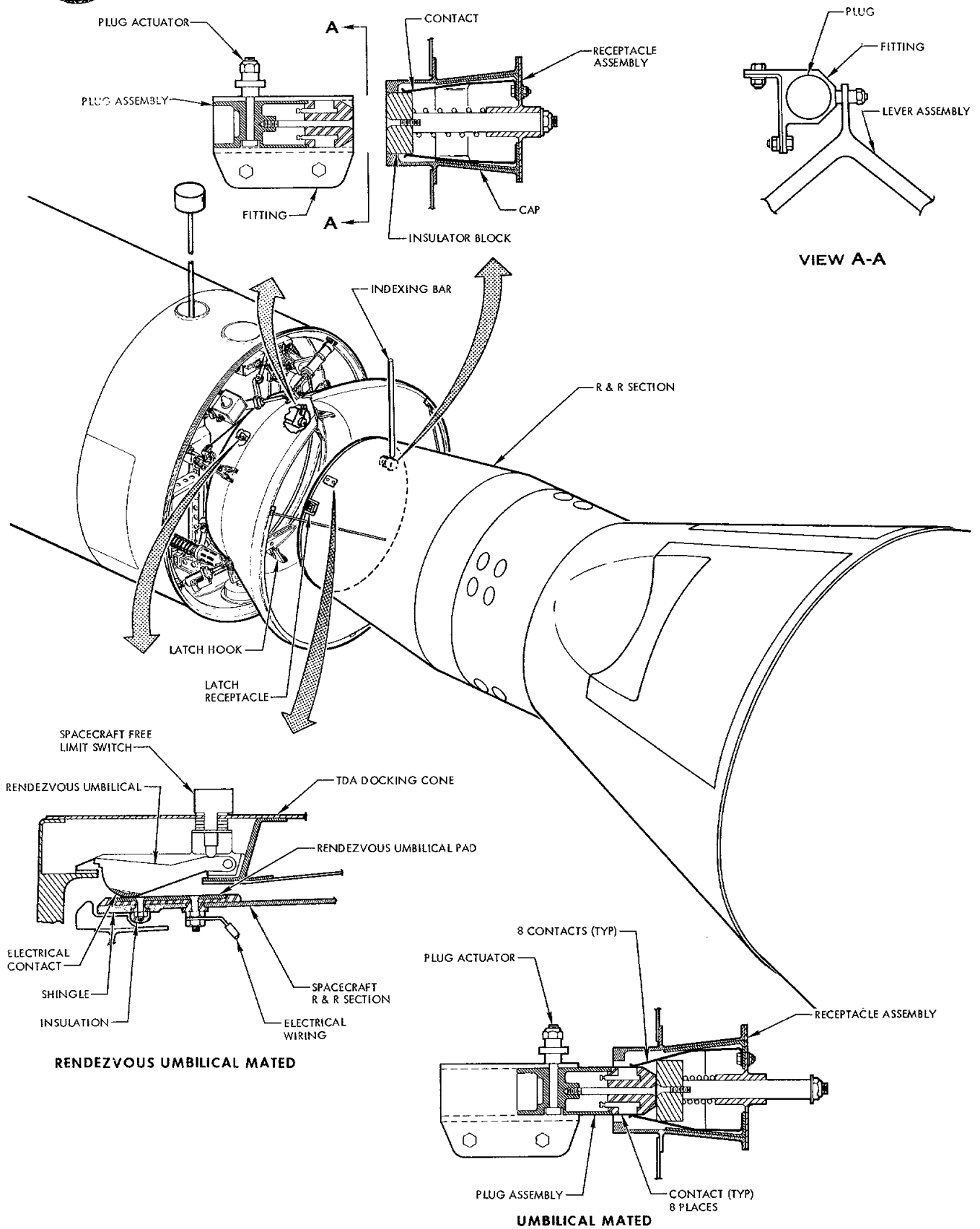


Figure 13-7 Umbilical Connectors

the latch cavities during re-entry.

UMBILICAL CONNECTORS

The main umbilical connector (Figure 13-7) is a 9 conductor receptacle-plug assembly. The receptacle is designed to mate with the plug as the Docking System is rigidized. The plug is extended and retracted by the rigidizing mechanism. As the plug enters the receptacle, it forces the insulator block into the receptacle. This action allows the receptacle and plug contacts to touch. When the plug is retracted, the spring returns the insulator block to the closed position. The insulator block covers the end of the receptacle.

The two rendezvous umbilicals are mated automatically when the R & R section latches to the TDA. The rendezvous umbilical consists of an electrical slide contact on the R & R section and a single pin lever assembly on the docking cone. During the docking sequence, the electrical slide contact depresses the spring loaded lever assembly slightly providing an electrical connection between the spacecraft and the TDA.

INDEX BAR ASSEMBLY

The index bar assembly (Figure 13-8) includes the bar, a housing, a pyrotechnic extension mechanism and a pyrotechnic jettison mechanism. The pyrotechnic igniters are fired electrically from the crew station. The extension igniters generate a gas pressure which forces the bar up until the socket in which the bar fits reaches a stop. This is the extended position of the bar. The bar is jettisoned in the same way, except that the gas pressure forces the bar out of the socket and ejects it into space. The jettison igniters also seal off the opening left by the jettisoned bar. The jettison igniters have a 1-second pyrotechnic delay to assure that the extend igniters will be fired first.

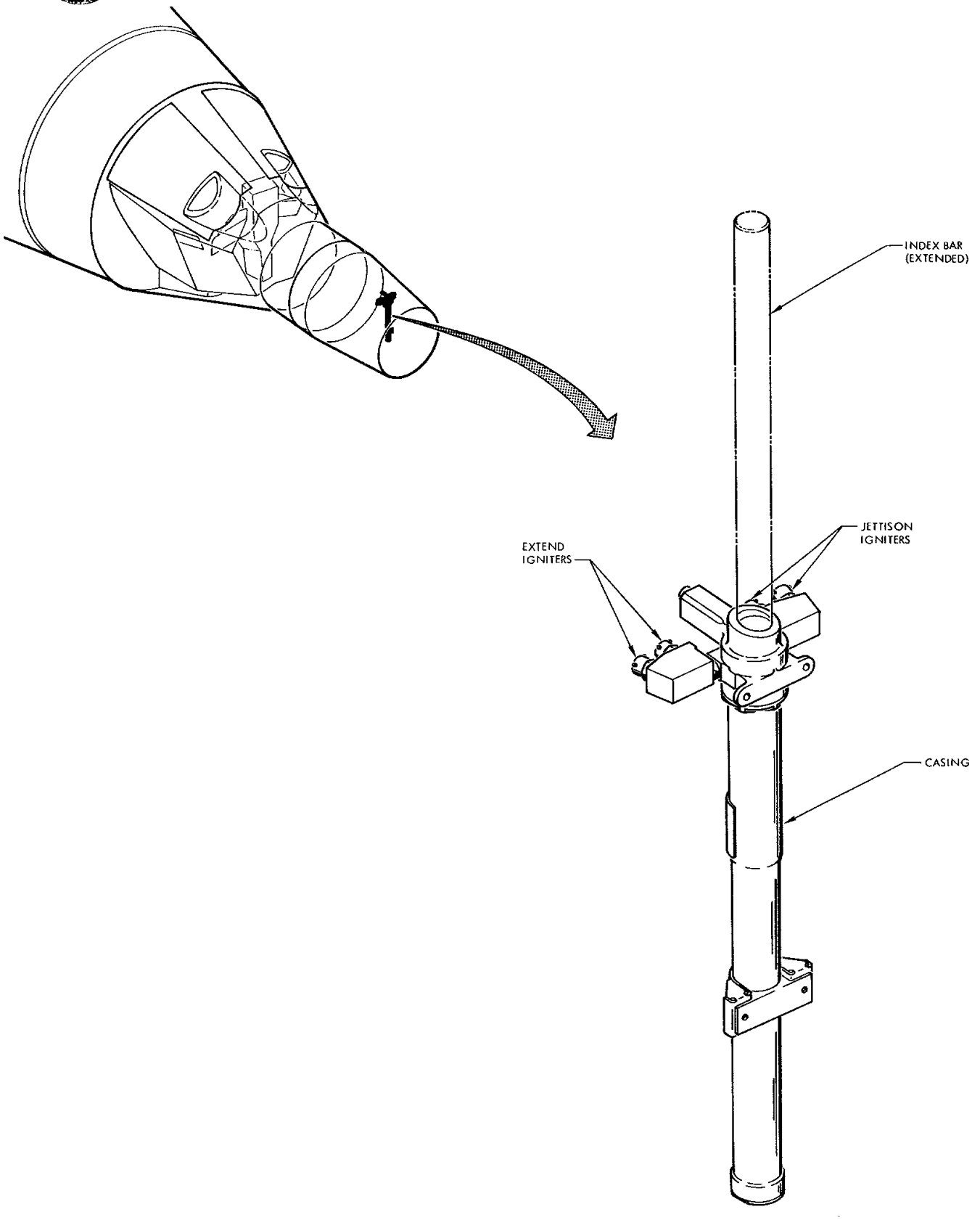


Figure 13- 8 Index Bar Assembly

RADAR SYSTEM AND PYROTECHNIC DEVICES

The details of the Rendezvous Radar System units and the Pyrotechnic Devices are presented in Section VIII and XI respectively. The necessary information to explain how these units fit into the Docking System operation is presented elsewhere in this section.

TARGET VEHICLE STATUS DISPLAY PANEL

The status display panel (Figure 13-3) is mounted on the TDA of the target vehicle. It is located on the docking adapter just back of the V-notch in the docking cone. This gives the pilots a full view of the panel during the approach and docking sequence. The displays on the panel and their functions when docking with either the ATV or the ATDA are explained in the following paragraphs.

ATV Displays

(1) DOCK (green light) when illuminated, indicates that the docking cone is unrigidized and that the latch hooks are reset.

(2) RIGID (green light) when illuminated, indicates that the docking cone is rigidized.

(3) PWR (green light) when illuminated, indicates that +28 volts dc unregulated, +28 volts dc regulated, -28 volts dc regulated, 115 volts 400 cps single phase and 115 volts 400 cps 3 phase power are operating.

(4) MAIN (red light) when illuminated, indicates the following:

a. With the main engine firing, the turbine has exceeded 27,000 rpm (1046 sensor level), the hydraulic pressure is below 1500 $\pm$ 20 psi, or the differential pressure between the fuel and oxidizer tanks is below 2 $\pm$ 2 psi.

b. With the main engine not firing, the differential pressure between the fuel and oxidizer tanks is below 2 $\pm$ 2 psi (fuel above oxidizer).

(5) MAIN (green light) when illuminated, indicates that the main fuel tank is above 15 ± 2 psia, the oxidizer tank is above 15 ± 2 psia and hydraulic pressure is above 50 ± 5 psia.

(6) ARMED (amber light) when illuminated, indicates that the engine control circuits are closed and either the main or secondary engines may be fired by command.

(7) SEC HI (green light) when illuminated, indicates that more than 1110 ± 20 psi expulsion gas pressure exists in both nitrogen spheres for a 50 second thruster firing and that more than 170 ± 5 psi regulated pressure exists in both propellant tank gas manifolds.

(8) SEC LO (green light) when illuminated, indicates that more than 360 ± 20 psi expulsion gas pressure exists in both nitrogen spheres for a 150 seconds thruster firing and more than 170 ± 5 psi regulated pressure exists in both propellant tank gas manifolds.

(9) ATT (green light) when illuminated, indicates that the Agena attitude control system is active.

(10) MAIN TIME (clock display) indicates by minute and second hands the time remaining for main engine burn. The regulated 28-volt dc power is applied to the display unit when the main engine is running. This causes the display unit to decrease the time remaining indication at a rate of one second per second of burning time.

(11) SEC TIME (clock display) indicates by minute and second hands the number of seconds of 200-pound secondary propulsion system burn time remaining. The regulated 28-volt DC power is applied on separate wires for high and low thrusters of the secondary propulsion system. This causes the display unit to decrease the time-remaining indication at a rate of one second per

second of burn time for the high consumption and a rate of one twelfth (1/12) second per second of burn time for the low consumption rates.

(12) ATT GAS (synchro display) indicates the percentage of total pressure remaining in the Agena attitude control system gas spheres.

ATDA Displays

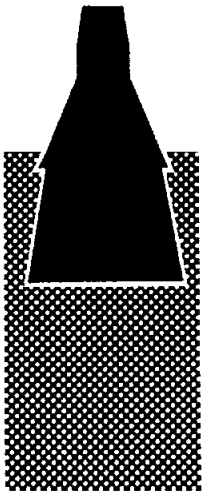
When docking with the ATDA only four displays are used, the others are inoperative.

(1) DOCK (green light) when illuminated, indicates that the docking cone is unrigidized and that the latch hooks are reset.

(2) RIGID (green light) when illuminated, indicates that the docking cone is rigidized.

(3) ARMED (amber light) when illuminated, indicates that ring A of the reaction control system has been activated.

(4) ATT (green light) when illuminated, indicates that 0 degree per second rate control has been selected in all three axes.



TARGET DOCKING ADAPTER (TDA)

Section XIV

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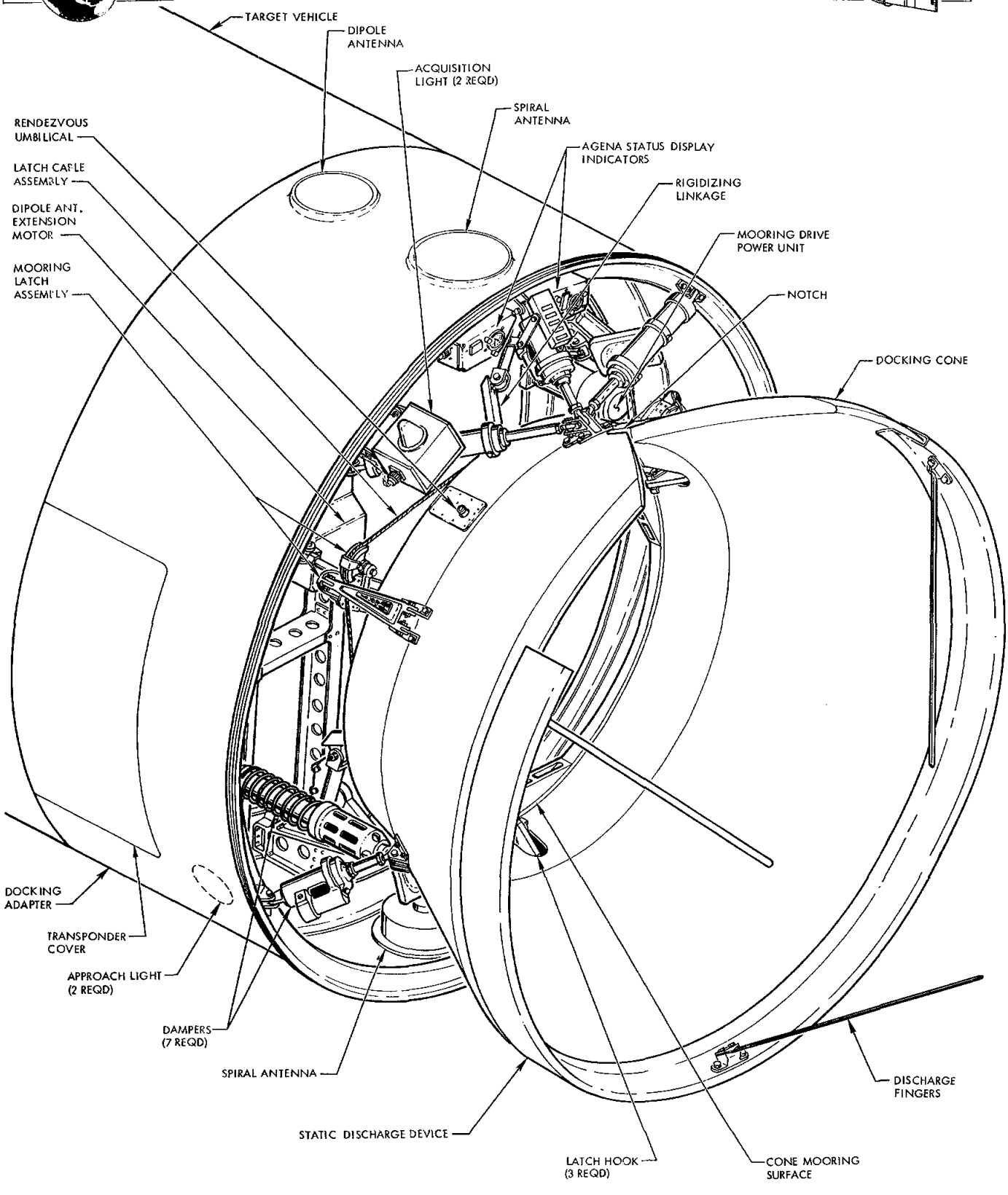


Figure 14-1 Target Docking Adapter Assembly

SECTION XIV TARGET DOCKING ADAPTER

SYSTEM DESCRIPTION

The Target Docking Adapter (Figure 14-1) may be mounted on either the Agena Target Vehicle or the Augmented Target Docking Adapter. The Target Docking Adapter is mounted on the forward equipment section of the target vehicle and becomes the mounting structure for all other components of the Target Docking Adapter.

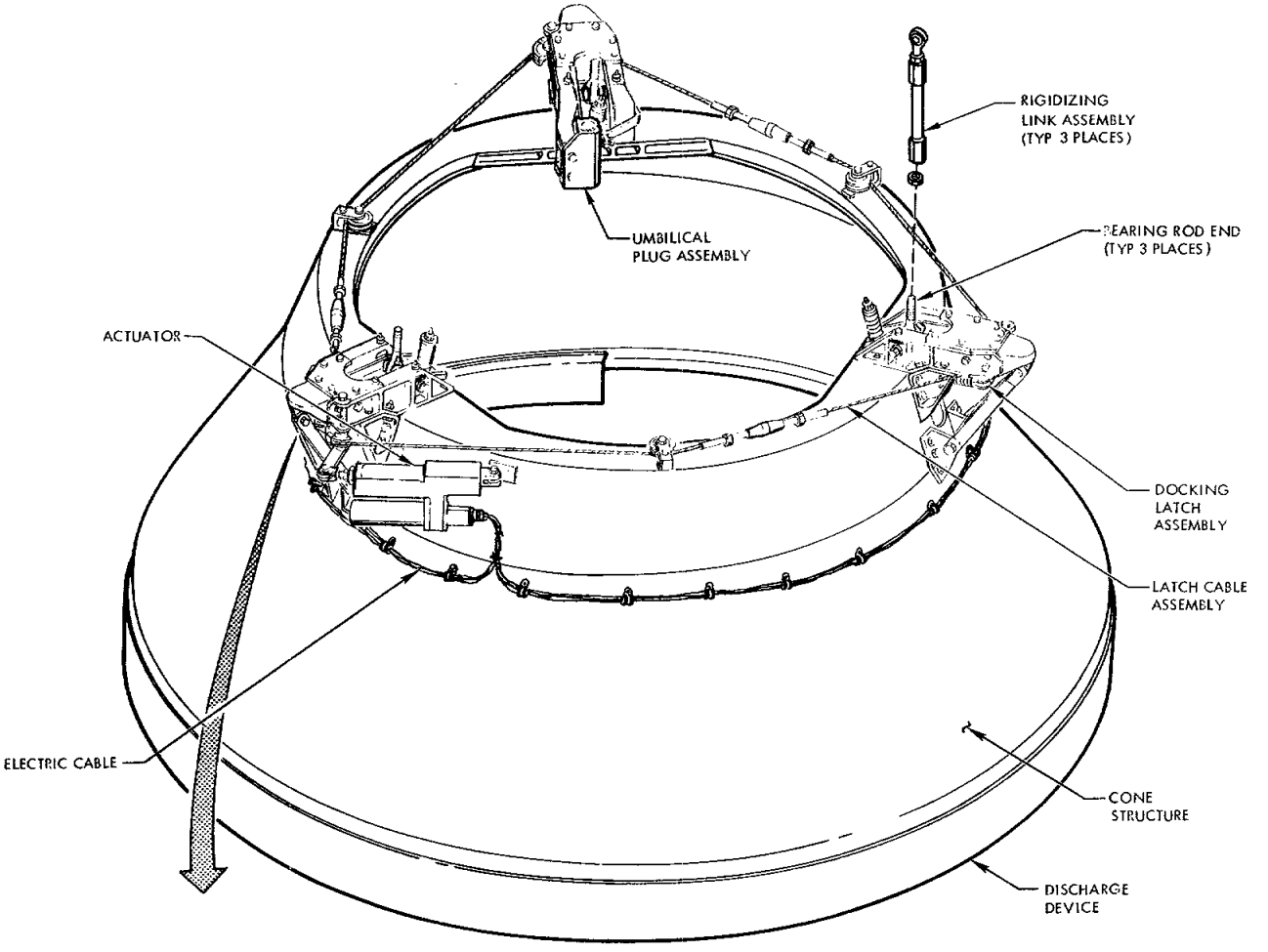
The docking cone receives the spacecraft during docking maneuvers. Electromechanical devices in the spacecraft and the docking adapter rigidize the docking cone as soon as the mooring operation is completed. Rigidizing operations may be automatic or as directed by the pilots or the ground control stations.

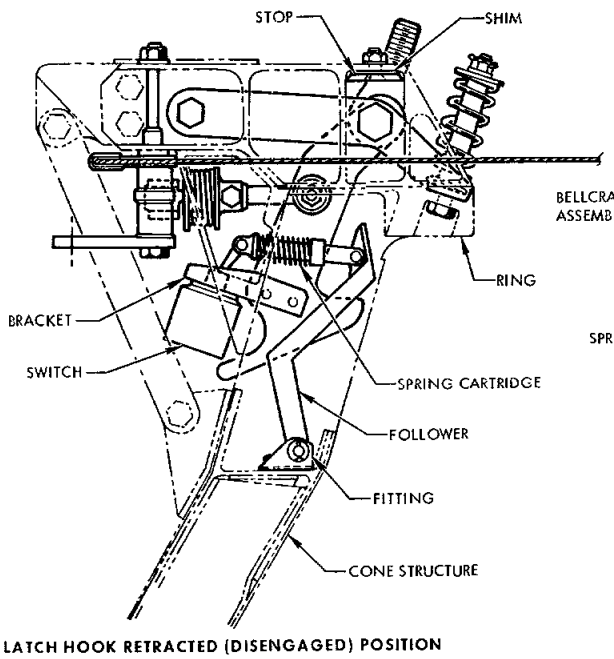
SYSTEM OPERATION

The Target Docking Adapter (TDA) consists of two major subassemblies (Figure 14-1): the docking cone and the docking adapter.

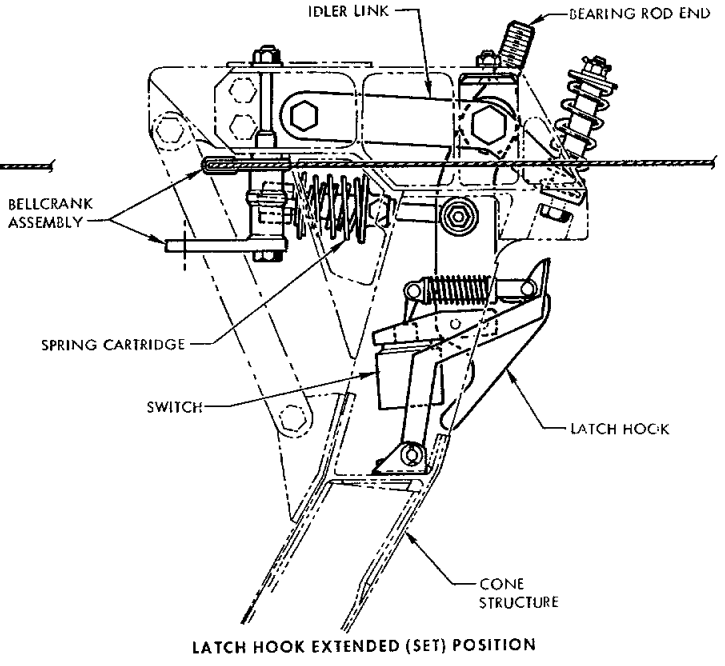
DOCKING CONE

The docking cone (Figure 14-1) is connected to the docking adapter by seven cone damper assemblies, three lateral and four longitudinal, and is the mounting structure for the mooring latch assemblies, the umbilical plug assembly, two rendezvous umbilical lever assemblies, the latch release actuator and a static discharge device. The latch assemblies are released and reset by the latch release actuator as a part of the unrigidize sequence. With the target vehicle control circuits operating, the cone unrigidized and





LATCH HOOK RETRACTED (DISENGAGED) POSITION



LATCH HOOK EXTENDED (SET) POSITION

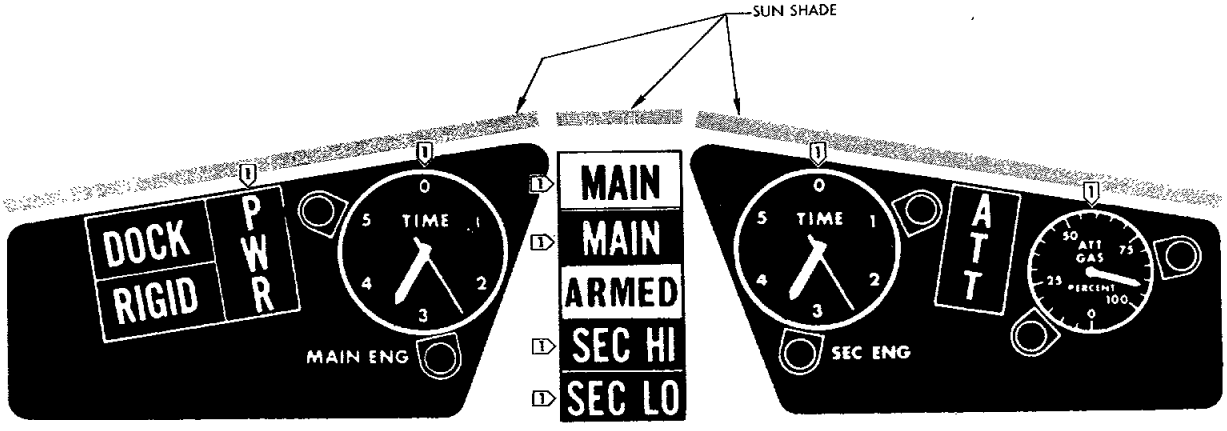
Figure 14-2 Cone Assembly

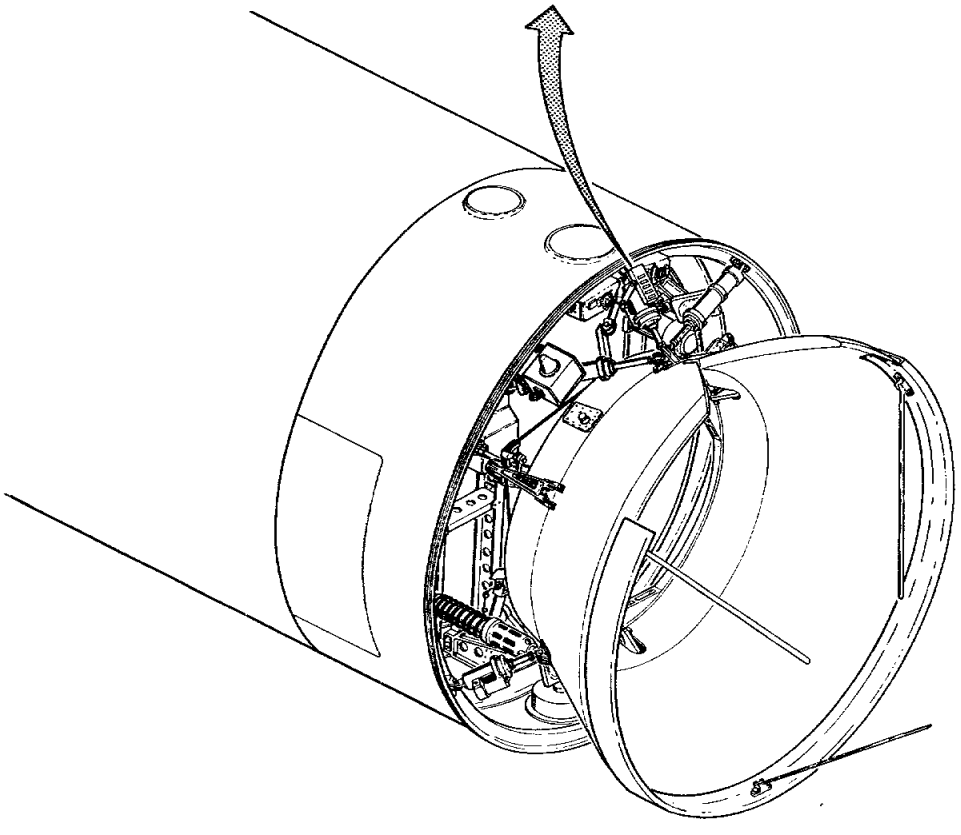
ready for docking, the latch assemblies are reset. In the reset condition, the latch assemblies are spring loaded so that when the Rendezvous and Recovery section enters the docking cone, the latch hooks engage the latch receptacles. The cone has a V-notch, with a slot at the bottom of the V, cut into the cone's upper surface. This notch and slot, together with the index bar, indexes the spacecraft to obtain proper alignment between the spacecraft and the docking target.

The seven cone damper assemblies are attached in three sets and are equally spaced at 120-degree intervals around the rear perimeter of the docking cone. One set, has a single lateral and two longitudinal damper assemblies and is connected to the cone behind the V-notch. Each of the two lower sets consists of one lateral and one longitudinal damper assembly. The dampers connect the cone to the adapter structure and absorb the shock of the actual docking thrusts. They are arranged to absorb thrust from any direction and to compensate for any misalignment of the two vehicles.

Three umbilical connectors provide the connection for electrical circuits between the spacecraft and the TDA so that hardline commands may be transmitted from the spacecraft to the TDA.

The main umbilical plug assembly (Figure 14-2) is mounted on one of the cone structural members which supports one of the latch assemblies. In the rigidizing operation, the mooring drive mechanism extends the plug out of its casing so that it mates with the receptacle on the spacecraft. The plug remains





[1] NOT OPERATIONAL ON ATDA.

Figure 14-3 Status Display Panel

extended and mated until the cone is unrigidized.

Two rendezvous umbilical lever assemblies (Figure 14-1) are mounted on the cone assembly and are mated with an electrical contact pad on the spacecraft when the latch hooks engage the latch receptacles.

DOCKING ADAPTER

The following docking units are mounted on the docking adapter (Figure 14-1): the mooring drive system, the target vehicle status display panel, the acquisition lights, the radar transponder, three radar antennas and two approach lights.

The mooring drive (rigidizing) system draws the docking cone tightly against the adapter structure and extends the umbilical plug. The cone is drawn into the adapter structure by the rigidizing linkages. The linkages are driven by three gear boxes. The gear boxes are driven by a single electric motor, through flexible drive cables and an H-drive gear box. The motor is started automatically and the cone rigidized when the spacecraft is latched into the cone. The docking adapter is unrigidized (motor reversed) and the spacecraft released by a single pilot command through the umbilical connection. Prior to docking, the unrigidizing process is initiated by rf command (radar-transponder Command Link or uhf ground radio link).

The target vehicle status display panel (Figure 14-3) is mounted on the docking adapter structure and on one cone damper, just above the V-notch in the cone. It visually displays information to the pilot on twelve functions when docking with the Agena Target Vehicle (ATV). The display consists of nine lights,

two clocks and one synchro indicator. All power required by the panel is supplied from the Agena power system. The ATV displays are as follows: DOCK (green light), RIGID (green light), PWR (green light), MAIN (red light), MAIN (green light), ARMED (amber light), SEC HI (green light), SEC LO (green light), ATT (green light), MAIN TIME (clock display), SEC TIME (clock display) and ATT GAS (synchro display).

The status display panel visually displays information to the pilot on four functions when docking with the Augmented Target Docking Adapter (ATDA). The display consists of four lights: DOCK (green light), RIGID (green light), ARMED (amber light) and ATT (green light). Power required by the panel is supplied from the equipment section of the ATDA.

The two acquisition lights aid the pilots in visual tracking of the ATV. They produce a flashing light that can be seen for approximately 20 nautical miles. They are mounted at the outer edges of the adapter structure so that they are visible to the pilots around the outer edge of the cone. The lights are mounted so that they rotate outward as the cone is unrigidized. Once the cone is unrigidized the lights are spring held in the outward position.

The radar transponder and the radar antennas (2 spiral and 1 dipole) are used for pilot command and tracking of the target vehicle. The radar transponder is energized and the dipole antenna (boom) is extended by uhf ground command.

The two approach lights are mounted on the lower inside structure of the adapter. They are arranged to shine through the rear of the cone and on the upper inside

surface near the V-notch. The lights are turned off and on by pilot or ground command. When on, they aid the pilot in attaining proper alignment of the spacecraft as they approach the docking cone.

SYSTEM UNITS

The TDA system units are those units which are concerned with mooring, rigidizing and unrigidizing the docking cone to the docking adapter. Units which are part of the other systems (Radar, Command Link and Docking) are not discussed in detail in this section.

DOCKING CONE

The docking cone (Figure 14-1 and 14-2) is designed to mate with the spacecraft R & R section. In the moored condition, the spacecraft R & R section fits tight against the cone bottom and is held there by the mooring latch hooks. The cone shape tends to guide the spacecraft to the proper mooring position. This action is aided by a V-notch and its terminating slot. When the spacecraft index bar is aligned with the slot, the spacecraft and target vehicle are properly oriented in roll for mooring. The cone is connected to the docking adapter by three cone damper assemblies. A rigidizing linkage links the cone to the mooring drive (rigidizing) system mounted on the docking adapter main structure.

CONE DAMPER ASSEMBLIES

Two sets of damper assemblies are composed of one lateral and one longitudinal assembly and one set is composed of one lateral and two longitudinal assemblies (7 dampers in all). When subjected to an impact, the dampers (Figure 14-1) compress slowly, absorbing energy as they move. In this way, most of the

energy of impact is absorbed by the fluid in the dampers and very little transmitted to the target vehicle. Springs and gas pressure return the dampers to their extended positions when the impact is dissipated.

APPROACH LIGHTS

Two approach lights are mounted on the docking adapter (Figure 14-1). They are positioned so that they shine through the rear opening in the cone to illuminate the notch during final approach of the spacecraft to the target docking vehicle. Some light, however, reaches the entire inner surface of the cone. The pilots turn the lights off and on by using the Command Link. Electric power is supplied by the target vehicle power system.

MOORING LATCH ASSEMBLIES

There are three mooring latch assemblies (Figure 14-2). Each assembly is mounted to the cone as shown in the illustration. Each assembly contains a bellcrank which actuates the mechanism. The bellcranks of the three assemblies are connected together by a cable assembly. One bellcrank is connected to the latching actuator by a linkage. When the actuator is extended, the bellcranks are rotated so that the latch assemblies are in the reset condition. This is the condition which must exist just prior to mooring. As the spacecraft slides into the docking cone and contacts the latch hooks, the hooks are pushed aside to allow the latch catches to get in front of the hooks. As this happens, the hooks drop in behind the catches (elevated part of the receptacle) and hold the spacecraft in place. The umbilical plug is mounted on one of the cone structure members which supports a latch assembly. The latch actuator is electrically driven and either retracts or resets the latch hooks. Power

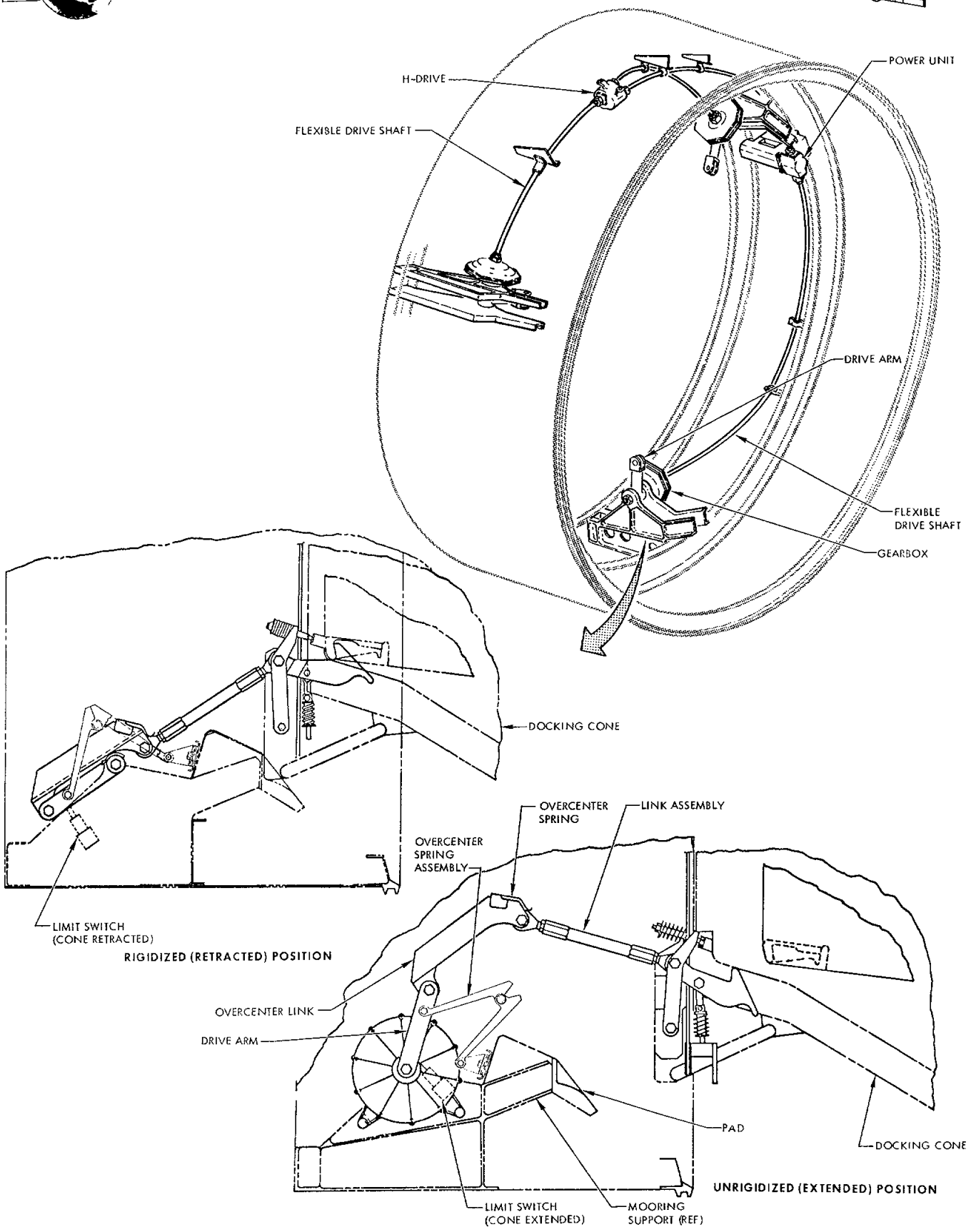


Figure 14-4 Mooring Drive System

to the latch actuator is switched by the latched sensing switch, the unlatched limit switch and the unrigidized limit switch. During the unrigidize sequence, the actuator is energized automatically to rotate the bellcranks and retract the latch hooks to the disengaged condition. When the latches are disengaged, there is a 30-second delay before the latch hooks are automatically reset. The latched sensing switch is operated only when the latch receptacle is present. If the latch receptacle is present, the latched sensing switch will initiate the rigidize sequence when the latch hooks are reset.

TARGET VEHICLE STATUS DISPLAY PANEL

The status display panel (Figure 14-3) is located on the docking adapter just back of the V-notch in the docking cone. The notch gives the pilots a full view of the docking panel during near approach to the docking cone. The displays on the panel and their functions are presented in detail in the Docking System Section.

MOORING DRIVE SYSTEM

The mooring drive system (Figure 14-4) is composed of a power unit (DC motor and gears), four flexible drive shafts, an H-drive, three gear boxes, six limit switches and a mooring drive linkage. A drive arm is attached to the output shaft of each gear box.

The power unit supplies power to the gear boxes through the flexible drive shafts and the H-drive unit. The gear boxes rotate the drive arms which in turn, retract or extend the rigidizing linkage. When the linkage is retracted, the docking cone is rigidized to the docking adapter and the spacecraft/target vehicle combination can be operated as a single unit. The linkage also moves an

actuator which extends the umbilical plug out of its casing. When the linkage is extended, the umbilical plug is retracted into its casing and the docking cone is moved away from the adapter so that it is completely supported by the dampers. The limits switches sense when the linkage is either fully extended or fully retracted and in each case remove power from the drive motor when the limit is reached.

The power unit is operated to retract (rigidize) the system by the actuation of three sensing switches on the mooring latch assemblies. The sensing switches energize the mooring drive power unit when the spacecraft is latched to the docking cone. The linkage can also be retracted by the RIGIDIZE command to the target vehicle command programmer. Limit switches stop the power unit when the linkage is fully extended. The same limit switches also apply power to the unlatch side of the latch release actuator of the mooring latch assemblies.

ACQUISITION LIGHTS

The acquisition lights (Figure 14-1) are used for visual guidance and tracking of the target vehicle when the vehicles are 20 nautical miles or less apart.

Two lights are provided. They are mounted on the docking adapter and are held in the retracted position during the boost and insertion phases of a mission. The docking cone holds the lights in the retracted position until the cone is unrigidized by the pilots via the Command Link.

Each light consists of a capacitor discharge flashing light system. The lamp flashes at a rate of 65 flashes per minute and has a minimum of 100 candles

effective intensity through an included angle of ± 90 degrees from the lamp longitudinal axis. A reflector increases the intensity so that the lamp is visible from 20 nautical miles with the intensity of a third magnitude star. The pilots turn the lights off and on via the Command Link.

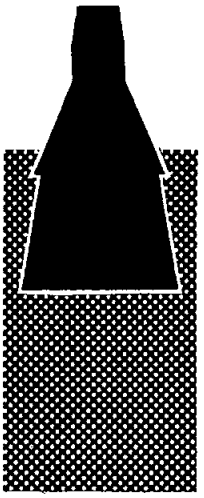
RADAR TRANSPONDER AND ANTENNAS

The radar transponder and antennas are considered part of the Rendezvous Radar System, the details are presented in Section VIII.

STATIC DISCHARGE DEVICE

A discharge device (Figure 14-1) is mounted on the docking cone to neutralize the electrostatic potential between the spacecraft and the target vehicle.

This device consists of three flexible metal fingers that protrude from the docking cone. They make the first contact between the spacecraft and target vehicle. The device is so wired that any static charge between the two orbiting vehicles will be dissipated at a controlled rate.



AUGMENTED TARGET DOCKING ADAPTER

Section XV

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AUGMENTED TARGET DOCKING ADAPTER-GENERAL

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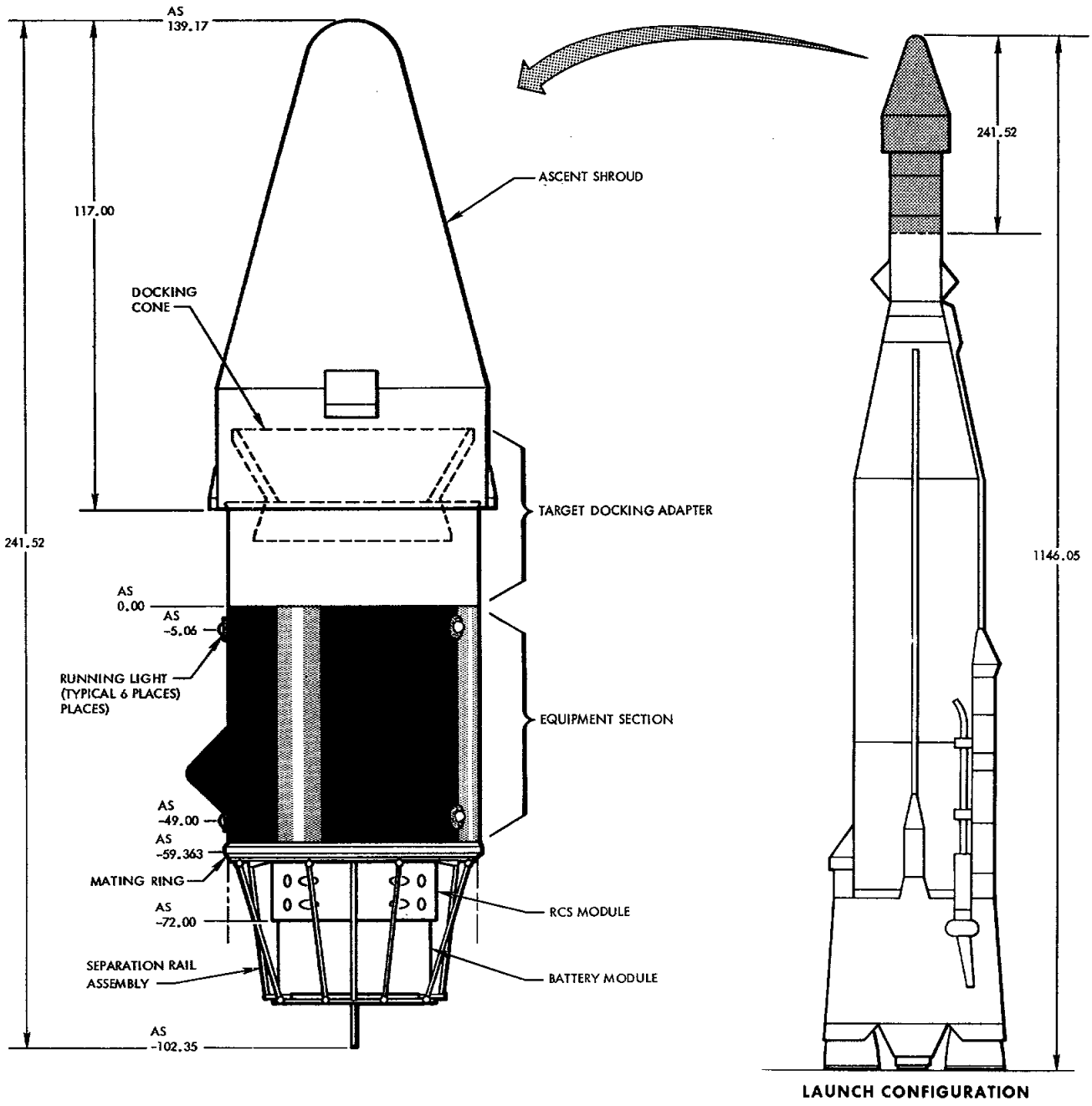

Figure 15-1 Augmented Target Docking Adapter

AUGMENTED TARGET DOCKING ADAPTER - GENERAL

ATDA DESCRIPTION

The Augmented Target Docking Adapter (ATDA) is an alternate target vehicle for the Gemini Spacecraft. The ATDA may be substituted for the Agena Target Vehicle provided sufficient mission objectives can be accomplished. The ATDA is capable of performing all but one of the major functions of the Agena. The ATDA is not capable of a translational maneuver. It does, however, have small thrusters to control attitude rates.

The ATDA (Figure 15-1) consists of a Target Docking Adapter (TDA) augmented with power, communication, instrumentation and stabilization systems. It is made up of five major sections or modules. They are: the ascent shroud, the TDA, the equipment section, the Reaction Control System (RCS) module, and the battery module. Only the equipment section, part of the TDA, and the ascent shroud are visible when the ATDA is mated with the launch vehicle. A mating ring attached to the equipment section, mates the ATDA to the launch vehicle.

The ATDA is cylindrical in shape; approximately 11 feet long, 5 feet in diameter at the greatest cross section and 3 feet at the smallest. Although constructed for the most part of aluminum alloy, the ATDA weighs a little over 2000 pounds with all equipment aboard. To aid in locating the ATDA during daylight hours, the equipment section is painted black with white longitudinal stripes. The other sections and modules are aluminum color.

ASCENT SHROUD

The ascent shroud is a weather tight jettisonable fairing, constructed of phenolic fiberglass. It is 9 feet 9 inches high and 5 feet 5 inches in diameter. The shroud is used to protect the TDA from aerodynamic pressures and thermal damage during launch. At the end of the boost phase, the shroud is jettisoned by a ground command to expose the docking cone for mission operation.

TARGET DOCKING ADAPTER

The TDA is an electromechanical unit constructed to receive the Gemini Spacecraft and to rigidly dock with it. The TDA is 5 feet in diameter and approximately 4 feet long. The TDA consists of two major assemblies; a docking cone and an adapter. The docking cone has a V-shaped notch to provide a roll alignment path for the spacecraft during the docking maneuver. The adapter assembly mates the TDA to the equipment section and is the mounting structure for all TDA components. A complete description of the TDA is contained in Section XIV of this manual.

EQUIPMENT SECTION

The equipment section is 60 inches in diameter and 54 inches long. This section contains the largest concentration of the ATDA electronic components. The components are mounted on internal cross beams. Entry into the equipment section is provided by four access doors.

REACTION CONTROL SYSTEM MODULE

The RCS module is approximately 40 inches in diameter and 18 inches long. The module contains two side-by-side rings of eight thrust chamber assemblies and duplicate fuel, oxidizer, and pressurant supplies. The RCS supply tanks are mounted on the RCS module but extend into the equipment section.

BATTERY MODULE

The battery module contains the three main batteries and two squib batteries. The module is 36 inches in diameter and approximately 18 inches long.

ATDA OPERATION

Before the ascent shroud is installed on the ATDA, the docking cone is rigidized. During pre-launch operations, ground personnel check out the ATDA using on-board equipment in conjunction with Aerospace Ground Equipment (AGE). AGE power cables are used to supply power to the ATDA systems during pre-launch to prevent depletion of the batteries. The cables are removed and internal power selected just prior to lift-off.

The ATDA is boosted by an Atlas Launch Vehicle into a circular orbit. (See Figure 15-2). At the end of the boost phase, the ATDA is separated from the launch vehicle; the ascent shroud that covers the TDA is jettisoned; and all systems that are used in orbit are turned on or put in readiness to operate.

As the ATDA passes over the launch facility, a Gemini Spacecraft is launched into an elliptical coplanar orbit. A catch-up maneuver is performed by the spacecraft. When the spacecraft gets within radar range (approximately 250 miles), rendezvous computations are performed and the spacecraft is maneuvered within 100 feet of the ATDA. At this time relative velocities and attitudes are adjusted in preparation for docking.

While the docking maneuver is being performed, ATDA systems stabilize the rotation of the ATDA in all three axes. When the spacecraft enters the docking cone, the docking latches engage and the rigidize motors are actuated. The spacecraft is docked and rigidly connected to the ATDA. Commands to unrigidize and unlatch

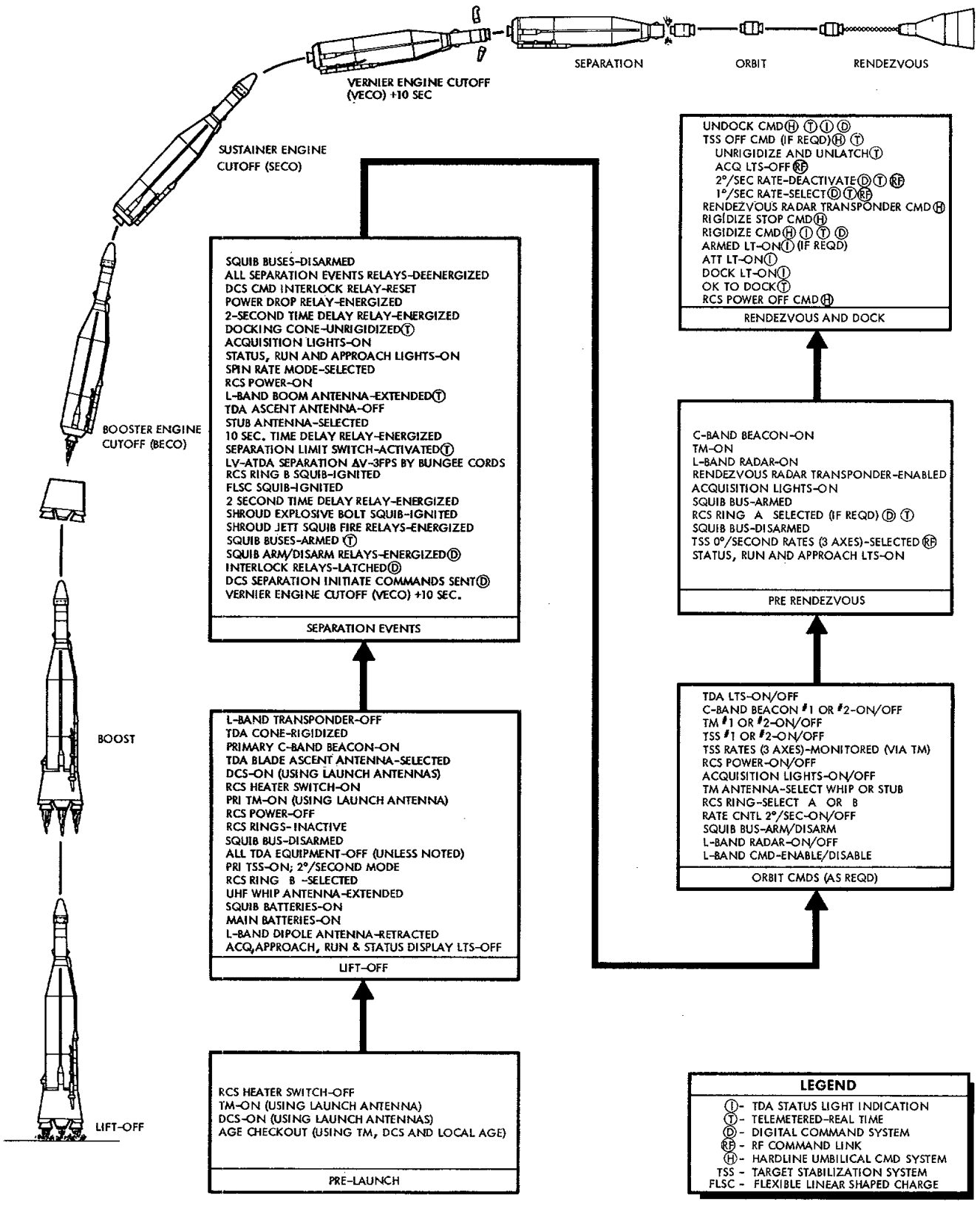

Figure 15-2 Sequence Of Events

can be given via the hardline umbilical which mates during the docking sequence or by ground commands.

ATDA SYSTEMS

Eight systems establish the rendezvous and docking capability of the ATDA. These systems are: Docking, Sequential, Electrical Power, Communication, Instrumentation, Target Stabilization, Reaction Control and Lighting. Sequential functions, such as ATDA/launch vehicle separation, ascent shroud separation, and RCS activation, are performed by pyrotechnic devices and mechanical separation assemblies. These are described in a separate subsection.

DOCKING SYSTEM

The Docking System is utilized to mate, secure, and rigidize the spacecraft with the ATDA. A detailed description of the Docking System is contained in Section XIII.

SEQUENTIAL SYSTEM

The Sequential System prepares the orbiting ATDA for rendezvous and docking with the Gemini Spacecraft. The system controls the sequence and timing of events which occur at ATDA/launch vehicle separation.

ELECTRICAL POWER SYSTEM

The Electrical Power System consists of the batteries and buses which supply and distribute dc power to all the systems of the ATDA. The system includes a means of substituting ground power for the on-board batteries in order to checkout the ATDA systems before launch.

COMMUNICATION SYSTEM

The Communication System provides the means of tracking the ATDA, transmitting information as to the condition and progress of the ATDA. The Communications System also accepts switching commands from the ground station.

INSTRUMENTATION SYSTEM

The Instrumentation System monitors and collects information on the environmental conditions of the ATDA and on the operational conditions of its electronic and electromechanical equipment. This information is assembled in the proper format and signal conditioned (if necessary) by the Instrumentation System, for transmission to the ground station by telemetry.

TARGET STABILIZATION SYSTEM

The Target Stabilization System (TSS) monitors the attitude rates of the ATDA and generates thrusting commands to control it at preselected rates.

REACTION CONTROL SYSTEM

The RCS is the dual rings of eight small thrusters each. These thrusters are fired by TSS commands. They produce the yaw, pitch, and roll thrust to control the ATDA at the selected attitude rate.

LIGHTING SYSTEM

The Lighting System consists of the acquisition, approach and status display panel lights on the TDA, and the running lights on the equipment section. A description of the TDA lights is contained in Sections XIII and XIV

Description of Running Lights

The six running lights are mounted on the surface of the ATDA equipment section.

Two are green, two red, and two amber. Viewing the space vehicle from the TDA end, the green lights are on the top left, the red on the top right, and the amber at the bottom. Lights of the same color are arranged on a longitudinal line with each other. The unfiltered intensity of each lamp is approximately 21 candlepower. The dome-shaped color filters which cover each lamp reduce the light transmission less than 15 percent. Each lamp and color filter is covered with a clear protective dome.

Operation of Running Lights

The operational readiness of the running lights is checked during pre-launch. At lift-off, all lights on the ATDA are off. They remain off during the boost phase. At the end of the separation events, all TDA lights and the runnings lights are turned on together. The lights may be left on until the Gemini Spacecraft and ATDA rendezvous. If the rendezvous exercise is not carried out as scheduled or if there is some reason for conserving electrical power, all ATDA lights can be turned off by a ground command. Ground station digital command can turn these lights on again. After docking, the acquisition lamps can be turned off by hardline umbilical commands.

SEQUENTIAL SYSTEM

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latch coils of the squib bus arm relays and the separation command initiate relays to the DCS pulse bus. An open set of contacts on both separation command initiate relays is also ready to connect the latch coils of the squib bus arm relays to the common control bus.

When the primary DCS execute on command (channel 18) is sent at VECO plus 10 seconds, the DCS pulse bus is armed. The DCS pulse bus and the separate command initiate relays redundantly latch the squib bus arm relays. (See Figure 15-4.)

SEPARATION INITIATE COMMAND

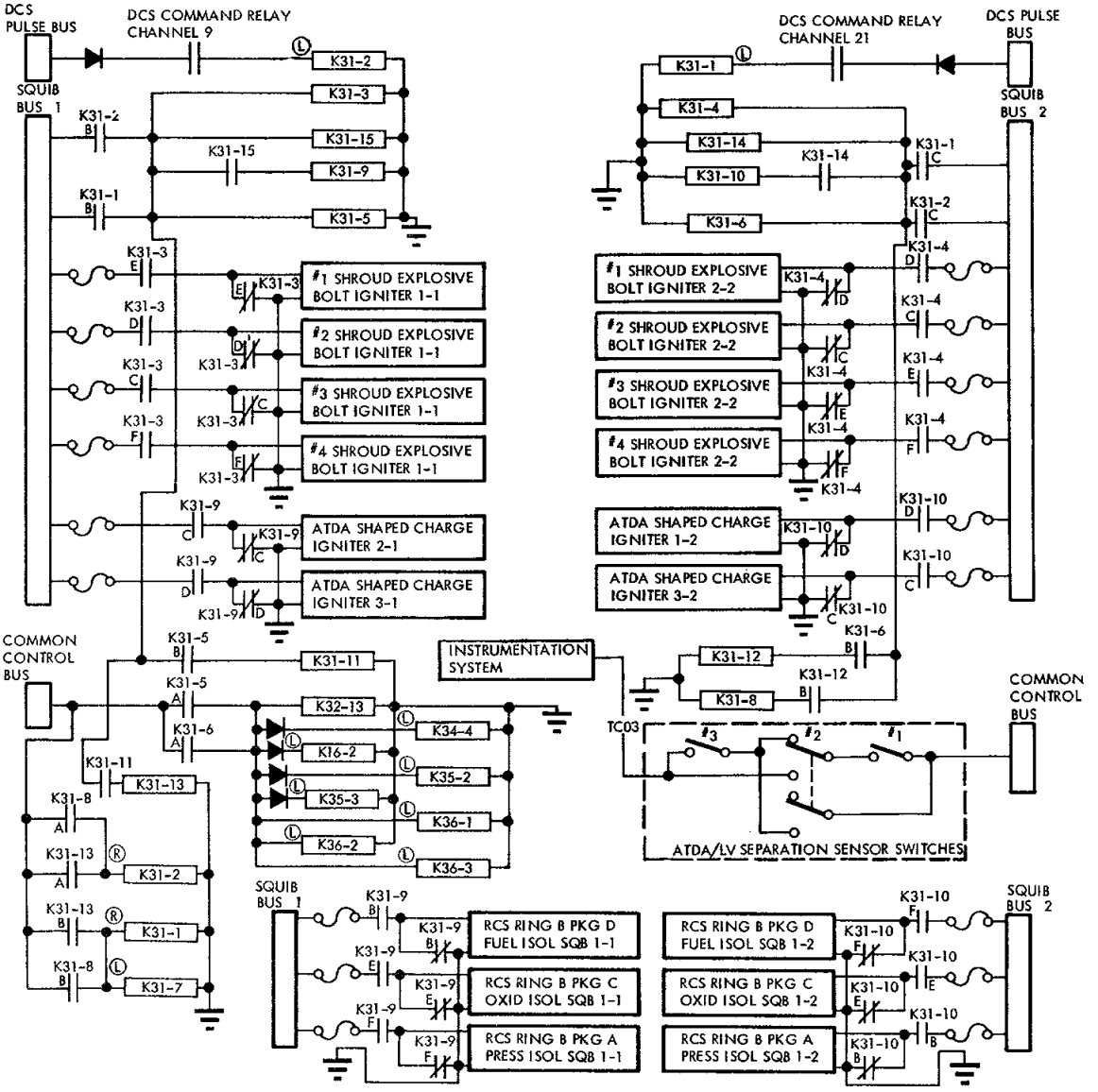
The primary DCS execute on command latches the 1 and 2 DCS command separation interlock relays, K31-2 and K31-1. K31-2 and K31-1 interlock the Sequential System in the on condition until a cycle of operation has been completed. K31-2 and K31-1 energize shroud jettison relays K31-3 and K31-4, separation 2-second timer relays K31-15 and K31-14, and separation events 10-second time delay relays K31-5 and K31-6. The ATDA separation events begin to happen with the energizing of the shroud jettison relays.

ASCENT SHROUD JETTISON

The shroud jettison squib fire relays, K31-3 and K31-4, energize immediately. Their C, D, E and F contacts connect squib bus power to redundant igniters of four shroud explosive bolts. Explosive charges cause the bolts to break as if sheared off. This allows the shroud separation assembly, which is described in the Pyrotechnics and Separation Assemblies subsection, to function. The shroud separates and falls away from the target vehicle.

FIRST 2-SECOND TIME DELAY

Three time delays are built into the ATDA Sequential System. The time delays



NOTE

(L) LATCH COIL OF LATCHING RELAY

(R) RESET COIL OF LATCHING RELAY

RELAY	REDUNDANT RELAY	RELAY NOMENCLATURE	RELAY PANEL
K31-2		#1 DCS COMMAND SEPARATION EVENTS INTERLOCK	ATDA POWER & SEQUENTIAL
	K31-1	#2 DCS COMMAND SEPARATION EVENTS INTERLOCK	ATDA PWR & SQL
K31-3		#1 SHROUD JETTISON SQUIB FIRE	ATDA PWR & SQL
	K31-4	#2 SHROUD JETTISON SQUIB FIRE	ATDA PWR & SQL
K31-5		#1 SEPARATION EVENTS 10-SECOND TIME DELAY	ATDA PWR & SQL
	K31-6	#2 SEPARATION EVENTS 10-SECOND TIME DELAY	ATDA PWR & SQL
K31-7		SEPARATION SENSOR POWER DROP	ATDA PWR & SQL
K31-9		#1 ATDA SHAPED CHARGE & RCS RING B SQUIB FIRE	ATDA PWR & SQL
	K31-10	#2 ATDA SHAPED CHARGE & RCS RING B SQUIB FIRE	ATDA PWR & SQL
K31-11		#1 SEPARATION POWER DROP 2-SECOND TIME DELAY	ATDA PWR & SQL
	K31-12	#2 SEPARATION POWER DROP 2-SECOND TIME DELAY	ATDA PWR & SQL
K31-13		#1 SEPARATION EVENTS POWER DROP SLOW RELEASE	ATDA PWR & SQL
	K31-8	#2 SEPARATION EVENTS POWER DROP SLOW RELEASE	ATDA PWR & SQL
K31-15		#1 SEPARATION 2-SECOND TIMER	ATDA PWR & SQL
	K31-14	#2 SEPARATION 2-SECOND TIMER	ATDA PWR & SQL
K32-13		ANTENNA CONTROL	ATDA SYSTEMS CONTROL
K34-4		RCS RING "B" CONTROL	ATDA SYSTEMS CONTROL
K16-2	K16-17	UNRIGIDIZE COMMAND	MOORING SQL CONTROL
K35-2		ACQUISITION LIGHTS POWER	ATDA SYSTEMS CONTROL
K35-3		BOOM ANTENNA EXTEND	ATDA SYSTEMS CONTROL
K36-1		STATUS DISPLAY & RUN LIGHTS POWER	ATDA SYSTEMS CONTROL
K36-2		RUN LIGHTS POWER	ATDA SYSTEMS CONTROL
K36-3		APPROACH LIGHTS POWER	ATDA SYSTEMS CONTROL

Figure 15-4 Sequential System Schematic

prevent the violence or rapid occurrence of one event from disrupting any other event. The first time delay is inserted between ascent shroud jettison and mating ring separation. This delay is controlled by the separation 2-second timer relays, K31-14 and K31-15.

ATDA/LAUNCH VEHICLE SEPARATION

Two seconds after ascent shroud jettison, K31-9 and K31-10 are energized. These are the ATDA shaped charge and the RCS ring B squib fire relays. Their C and D contacts connect the squib buses to the shaped charge igniters (or squibs). The shaped charges explode, severing the mating ring. The mechanical separation device now performs like a catapult. The bungee cords contract and shove the ATDA out of the launch vehicle with a separation velocity of 3 feet per second.

SEPARATION SENSING

As the ATDA starts its movement away from the Atlas Launch Vehicle, the blast shield surface that held the plungers of the separation sensors in the preloaded position is left behind. The compressed springs in the sensors thrust the plungers downward and the linkages pull the toggles down to the closed positions. Only two of the three sensor switches need close to complete the sensing circuit. The closed switches connect common control bus power to the Instrumentation System Programmer. This bi-level signal is telemetered to the ground monitoring station as the ATDA separation parameter.

RCS RING B ACTIVATION

At lift-off, the RCS fuel and oxidizer tanks are isolated from the thrusters, and the nitrogen pressurant is isolated from the fuel and oxidizer tanks by

pyrotechnic valves. The B, E and F contacts of relays K31-9 and K31-10 connect squib bus power to six RCS ring B pyrotechnic valves. The pyrotechnic valves fire, activating ring B of the RCS.

TEN-SECOND TIME DELAY

The second time delay begins when the execute command to initiate separation is given. It is timed redundantly by the separation events 10-second time delay relays, K31-5 and K31-6. When these relays energize, they connect the common control bus to the separation events relays. The separation events relays apply RCS power, select the uhf antennas, extend the L-band boom antenna, turn on the ATDA lights, and unrigidize the docking cone. The B contacts of K31-5 and K31-6 control the last 2-second time delay.

RCS POWER APPLICATION

The RCS latching control relay, K34-4, connects common control bus power to all RCS solenoid valves. RCS ring A/ring B select relays were set at lift-off, by DCS command, to select ring B. The Target Stabilization System has been operating since lift-off, generating rate stabilization commands. These commands now switch the various RCS ring B thrusters on and off to stabilize the ATDA.

UHF ANTENNA SELECTION

The antenna control relay, K32-13, connects the common control bus to the position 1 input on coax switch 3. This switches the primary telemetry transmitter and one of the DCS receivers from the uhf ascent blade antenna to the uhf stub antenna.

TRANSPONDER ANTENNA EXTENSION

The L-band transponder antenna extend relay, K35-3, connects the common control bus to the extend terminals of the antenna drive motor. Within 30 seconds, the

transponder antenna rises to a height of approximately 85 inches and closes the extend limit switches. One of the switches removes the extension voltage from the drive motor. The other switch connects the common control bus to the Instrumentation System programmer. This parameter is telemetered to the ground monitoring station, confirming that the L-band transponder antenna has been fully extended.

DOCKING CONE UNRIGIDIZING

The unrigidize command relays, K16-2 and K16-17, are latched, energizing the unrigidize winding of the mooring drive motor and the unlatch winding of the latch release motor. The mooring drive motor runs, activating three sets of unrigidized limit switches (one open and two closed). The latch release motor runs, closing the latch actuator limit switch. The opened set of limit switches causes power to be removed from the drive motor unrigidize winding. One closed set of limit switches resets the undock circuit. The latch actuator limit switch and the other set of closed limit switches completes a circuit between contacts of K36-1 and the DOCK light on the status display panel. This circuit also causes the unrigidize parameter to be applied, through contacts of K16-20, to the Instrumentation System programmer. This parameter is telemetered to the ground station for telemetry confirmation of cone position. One set of rigidized limit switches opens to prevent the RIGID light on the status display panel from illuminating.

ACQUISITION LIGHTS ON

The acquisition lights are located on the top left and bottom right of the docking cone. The acquisition lights, like the docking cone behind which they are mounted, are held in a rigid retracted position during ascent. When the mooring drive motor

unrigidizes the docking cone, the movement causes the lights to move outward from behind the cone into their operating positions. The acquisition lights power relay, K35-2, applies common control bus power to the lights.

RUNNING LIGHTS ON

The running lights power relay, K36-2, and status display and running lights power relay, K36-1, are latched. K36-2 connects main bus power to the red, green and amber running lights at station -5.00 which is near the Target Docking Adapter. K36-1 applies the same power to the three running lights at station -49.00 near the RCS module. The running lights can be turned off by DCS command.

STATUS DISPLAY LIGHTS ON

The status display panel on the ATDA is the same panel as the one used on the Agena Target Vehicle. However, the ATDA uses only four of the status display lights: DOCK, RIGID, ARMED and ATT (attitude).

The status display and running lights power relay, K36-1, which powered up three of the running lights, also connects power to the circuits which control the status display lights. At this time, only the DOCK light illuminates. The DOCK light illuminates green to indicate that the TDA cone is unrigidized and the docking latches are ready to receive an entering spacecraft. The green RIGID light which indicates a rigidized docking cone, the amber ARMED light which indicates RCS ring A has been activated, and the green ATT light which indicates that zero degree per second rates have been selected cannot illuminate now. These indicators will illuminate when the appropriate function has been selected by DCS command.

APPROACH LIGHTS ON

The approach light power relay, K36-3, connects main bus power to the two series-wired floodlights inside the TDA. These floodlights, called approach lights, are located near the bottom of the TDA and illuminate the space behind the docking cone for a visual approach. The two lights are turned on and off by the commands which control the running lights.

LAST 2-SECOND TIME DELAY

The third time delay is a 2-second interval between the start of the separation events and the removal of Sequential System relay power. This delay is provided by the separation power drop time delay relays, K31-11 and K31-12. The B contacts of K31-11 and K31-12 connect squib bus power, switched by K31-1 and K31-2, to the power drop slow release relays.

SEQUENTIAL POWER DROP OUT

Many relays, energized by the ten second time delay relays, are latching relays. Such relays are latched by permanent magnets in the positions to which they have been switched. Hence, they no longer need electrical power to maintain their selected positions. Relays K32-13, K34-4, K35-1, K35-2, K35-3, K36-1, K36-2 and K36-3 are of this kind. Power which was applied to their coils can be removed without changing their contact positions.

Besides the latching relays, there are twelve relays, the contacts of which change when the relays are deenergized. These relays are used to fire the squibs and to produce time delays. After their functions are completed, power to their coils also may be removed.

The power drop sequence is accomplished in seven steps: First, a 2-second time delay is placed between the application of the separation events signal and the start of the power drop sequence. The delay permits the separation events relays to reach a steady state. Second, power drop 2-second time delay relays K31-11 and K31-12 close, connecting squib bus power to power drop slow release relays K31-13 and K31-8. Third, the power drop relays connect the common control bus to the reset coils of K31-1 and K31-2. Fourth, K31-1 and K31-2 reset, opening the circuit from the squib buses to K31-3, K31-4, K31-5, K31-6, K31-9, K31-10, K31-14 and K31-15. Fifth, K31-5 and K31-6 deenergize, disconnecting squib bus power from eight separation events latching relays and from K31-11 and K31-12. Sixth, K31-11 and K31-12 deenergize, disconnecting squib bus power from K31-13 and K31-8, the slow release relays. Seventh, in approximately 40 to 70 milliseconds, K31-13 and K31-8 deenergize, disconnecting the common control bus from the reset coils of K31-1 and K31-2. This step completes the power drop-out sequence.

SYSTEM UNITS

The units of the Sequential System are the relays, the relay panels and sensor switches which control or monitor the separation events. All of the Sequential System relays are located on the power and sequential relay panel and the systems control panel.

Although some unrigidizing relays are operated by the separation events signal, the relays and their relay panels are properly part of the Docking System. The sensor switches which monitor separation are discussed here, the pyrotechnic devices which cause separation and some of the separation events are discussed in the Pyrotechnics and Separation Assemblies subsection.

POWER AND SEQUENTIAL RELAY PANEL

The power and sequential relay panel (Figure 15-3) is approximately 8.7 inches high by 9.8 inches wide by 3.5 inches thick. This panel is located in the equipment section of the ATDA and provides a mounting structure for fifteen sequential relays. These relays and their functions are identified in Figure 15-4. Seven connectors provide electrical access to the relay solenoids and contacts. Relays mounted on this panel are identified by a K31 series reference designation.

SYSTEMS CONTROL RELAY PANEL

The systems control relay panel (Figure 15-3) is approximately 7.5 inches high by 11.5 inches wide by 3.5 inches thick. This panel is located in the equipment section of the ATDA and provides a mounting structure for eight sequential relays. Seven of these relays and their functions are identified in Figure 15-4. The remaining relay, K35-1 is used to complete a telemetry path, confirming when the docking cone is unrigidized. In addition to the sequential relays, eleven system control relays are located on this panel. These relays are used to select secondary systems or different modes for the various ATDA systems. Six connectors provide electrical access to relay solenoids and contacts. Relays mounted on this panel are identified by a K32 through K36 series reference designation.

SEPARATION SENSORS

The separation sensors are electromechanical devices used to detect the separation of the ATDA from the launch vehicle. The electrical part of the sensor is a double-pole double-throw toggle switch. Only switch 2 uses both poles. (See Figure 15-4.)

The mechanical part of the sensor is a bracket, on which the switch is mounted, and a linkage designed to operate the toggle. (See Figure 15-3.) The linkage is basically a plunger or piston confined in a housing. Shafts attached to both faces of the plunger extend out the top and bottom of the housing. The bottom shaft, which is narrow, controls the position of the plunger. The top shaft, which is thicker, has an oblong hole, somewhat larger than a toggle switch handle, machined into it. A compression spring fits over the top shaft inside the housing and holds the plunger against the bottom of the housing. The toggle of the switch extends into the top shaft hole which pulls the toggle into the down or closed position.

Three separation sensors are mounted in the end of the equipment adapter where it interfaces with the launch vehicle mating ring. Each sensor fits through a hole cut in the end of the equipment section, so that the control shaft extends into the mating ring. A heat and shrapnel protective shield surrounds the inside of the mating ring and forms a trough. The sensor shafts rest against the bottom of this trough. When the vehicles are mated, the trough pushes the shafts and plungers upward in the housing, compressing the springs and moving the toggles to the up position. These are the launch positions of the separation sensors. When the shaped charge severs the mating ring, the trough stays with the launch vehicle and the sensors leave with the ATDA. The spring loaded sensor switches now complete the path from the common control bus to the Instrumentation System.

ELECTRICAL POWER SYSTEM

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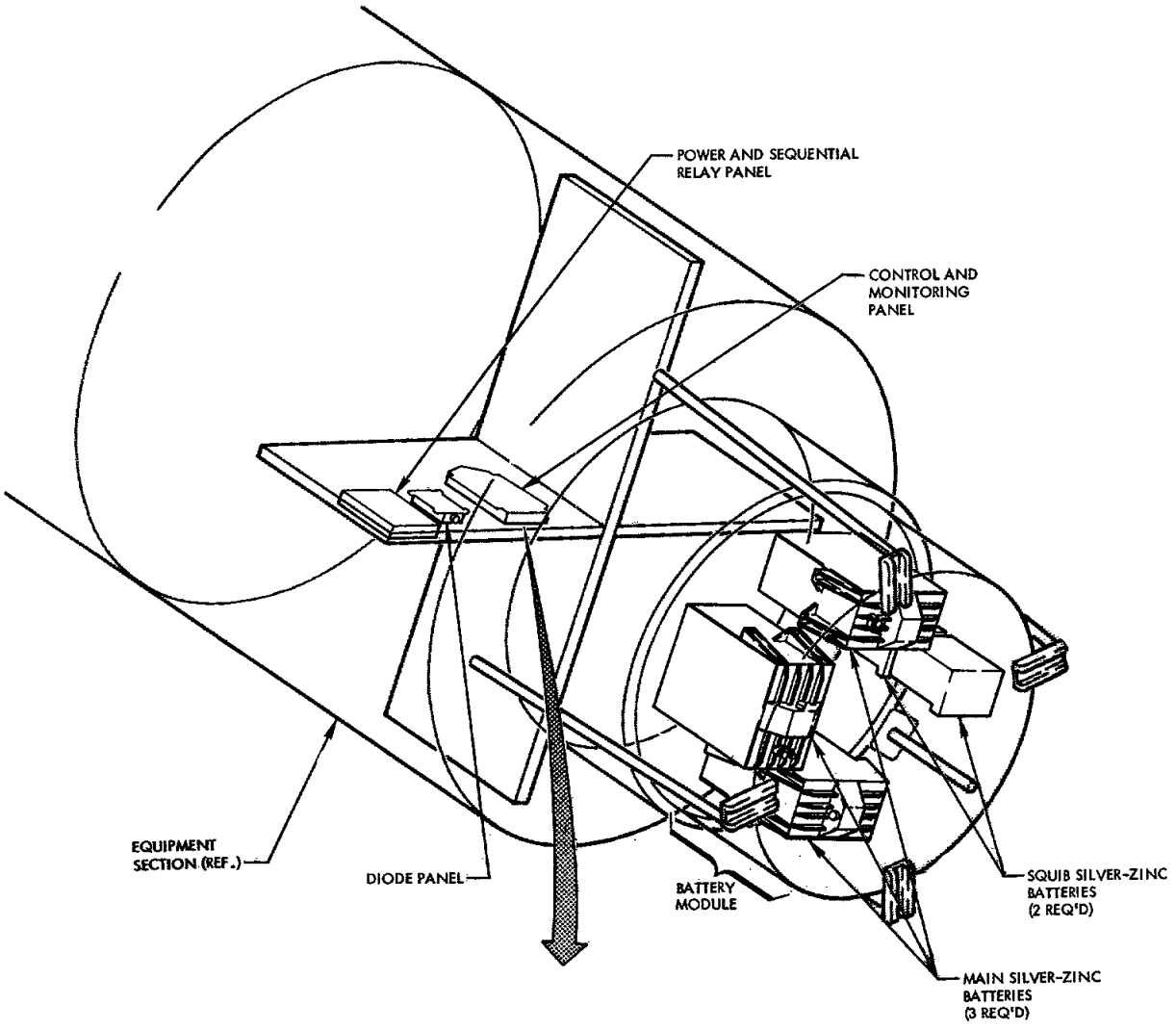

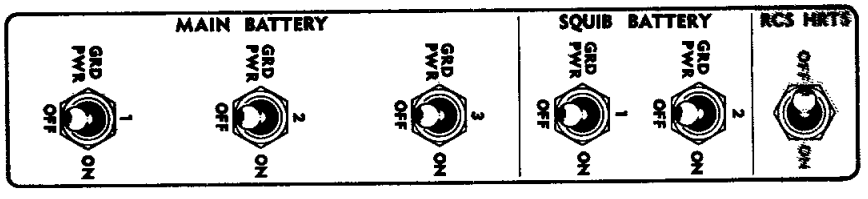

POWER CONTROL SWITCHES

Figure 15-5 Electrical Power System Installation

ELECTRICAL POWER SYSTEM

SYSTEM DESCRIPTION

The Electrical Power System for the Augmented Target Docking Adapter (ATDA) basically consists of three silver-zinc main batteries, two silver-zinc squib batteries, a control and monitoring panel, a diode panel and relays for controlling squib bus power. Refer to Figures 15-5 and 15-6.

The three main batteries provide dc power to the ATDA main power bus. The two squib batteries provide dc power to the common control bus and the two squib power buses which are isolated from the main and common control bus. The control and monitoring panel provides switching for the power system and the capability for utilizing Aerospace Ground Equipment (AGE) external dc power and remote monitoring of the power system parameters.

The diode panel provides electrical isolation between the two squib batteries for individual battery fault protection. All of the power system circuits are of a redundant nature. No primary ac electrical power system is provided for the ATDA. Devices requiring ac power will obtain this power from self-contained inverters within the individual systems.

Both squib batteries supply power to the common control bus. Squib battery 1 and squib battery 2 are separately connected to squib bus 1 and 2 respectively via the squib bus arm relays. These relays are controlled by the Digital Command System (DCS).

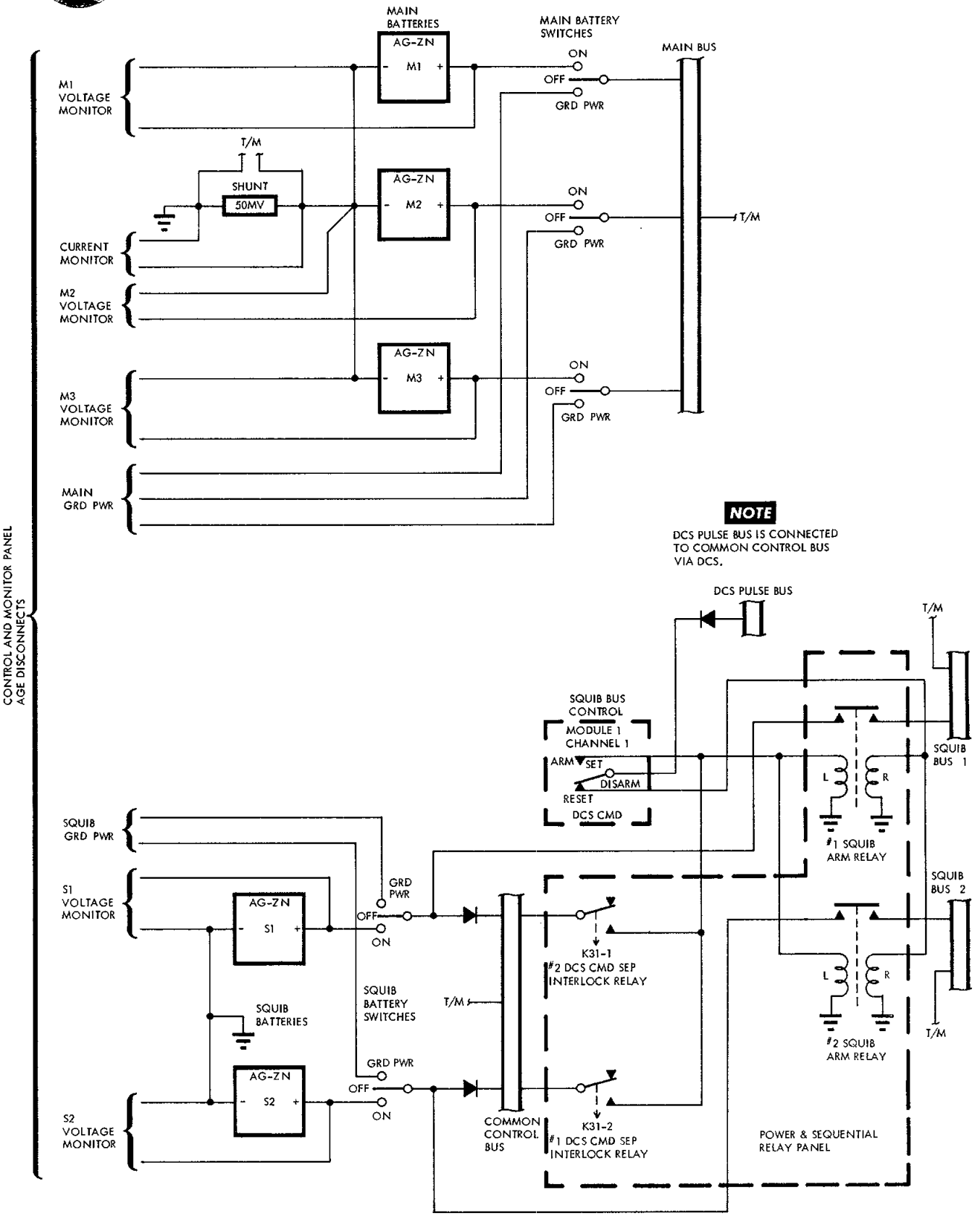

Figure 15-6 Electrical Power System Schematic

The three main batteries and two squib batteries are installed in the battery module, which is located in the opposite end of the ATDA from the docking adapter. The control and monitoring panel, diode panel and power and sequential relay panel (containing the squib bus arm relays) are located in the equipment section of the ATDA.

SYSTEM OPERATION

PRE-LAUNCH

In order to conserve the ATDA batteries, AGE external dc electrical power is utilized during pre-launch checkout of the ATDA systems. External power is supplied to the ATDA through AGE cables connected to the control and monitoring panel receptacles. Remote monitoring of main bus current, individual main battery voltage and individual squib battery voltage is also accomplished through the AGE cables.

External power is applied to the main power bus and common control bus by setting the MAIN BATTERY and SQUIB BATTERY switches to the GRD PWR position. Just prior to launch, all battery switches are set to ON position and the AGE cables are removed from the ATDA. The squib power buses are not armed prior to launch.

ORBIT

The squib power buses are armed via the two squib arm relays just prior to orbital insertion. The relays are energized to latch position by a command from the DCS. Common control bus voltage, squib bus 1 and 2 voltage and main bus voltage and current are monitored by the Instrumentation System.

After the required squib functions for orbital insertion are accomplished, the

squib buses are disarmed by a command from the DCS and may be rearmed if required. The squib buses are disarmed prior to rendezvous of the spacecraft with the ATDA.

SYSTEM UNITS

SILVER-ZINC BATTERIES

The three main batteries are 400 ampere/hour, 16 cell, silver-zinc batteries and are identical to the adapter power supply batteries used on spacecraft 6. The two squib batteries are 15 ampere/hour, 16 cell, silver-zinc batteries and are identical to the squib batteries used in the spacecraft.

The main battery cases are constructed of magnesium. The approximate activated (wet) weight of each main battery is 118 lbs. The squib battery cases are constructed of titanium. The approximate wet weight of each squib battery is 8 lbs. The squib batteries are special high-discharge-rate batteries which will maintain a terminal voltage of 18 volts for one second under a 75 ampere load. All of the silver-zinc batteries have an open circuit terminal voltage of 28.8 to 29.9 volts.

The battery electrolyte consists of a 70 percent solution of reagent grade potassium hydroxide and distilled water. The squib batteries have a vent valve in each cell designed to prevent electrolyte loss. The valve will vent the cell to atmospheric pressure in the event that a pressure in excess of 40 psig builds up within the cell.

All of the silver-zinc batteries are equipped with relief valves which maintain a

tolerable interior to exterior differential pressure in the battery cases. The batteries are capable of operating in any attitude in a weightless state. Prior to installation into the battery module, the batteries are activated and sealed at sea level pressure.

CONTROL AND MONITORING PANEL

The control and monitoring panel contains the main and squib battery switches and the Reaction Control System (RCS) heater switch. This panel also provides receptacles for connecting AGE external power and test cables to the ATDA.

DIODE PANEL

The diode panel contains diodes required to provide individual fault protection for the squib batteries. The squib batteries are connected to the common control bus via these blocking diodes to provide battery isolation.

POWER AND SEQUENTIAL RELAY PANEL

The power and sequential relay panel, which is essentially part of the Sequential System, contains the squib bus arm relays in addition to the other sequencing and control relays. The squib arm relays are controlled by the DCS and redundantly by the Sequential System.

COMMUNICATION SYSTEM

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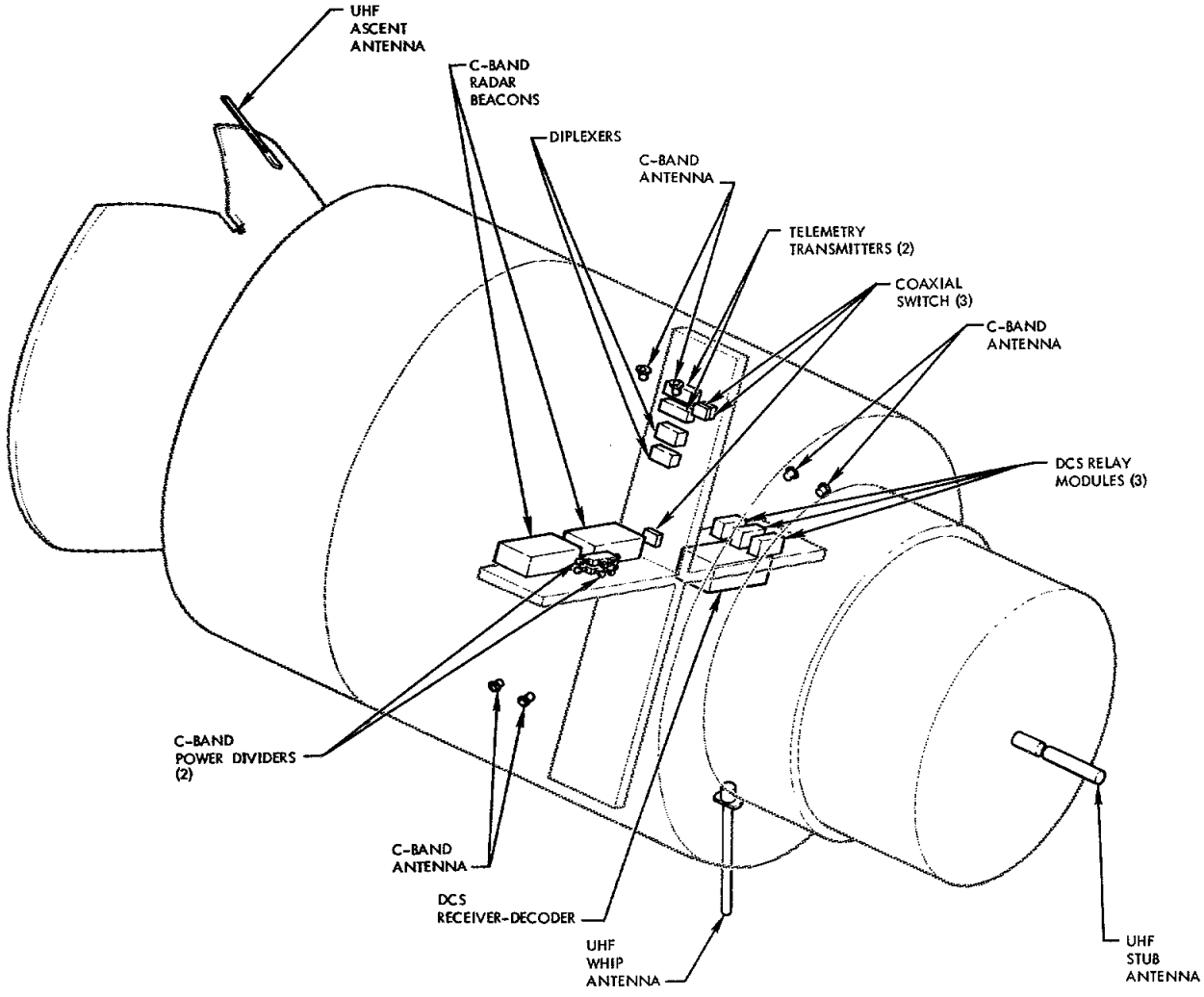


Figure 15-7 Communication System

COMMUNICATION SYSTEM

SYSTEM DESCRIPTION

The Communication System is the only communication link between the ground and the Augmented Target Docking Adapter (ATDA). The Communication System provides the following capabilities: radar tracking of the ATDA, ground command to the ATDA, and telemetry transmission. To make possible these various capabilities, the Communication System components may be divided into the following categories: antennas, including diplexers and coaxial switches; beacons; telemetry transmitters; and a Digital Command System (DCS).

The Communication System components are located throughout the ATDA with the largest concentration being in the equipment section. The location of the Communication System components is illustrated in Figure 15-7.

ANTENNAS

Three uhf antennas and two sets of C-band helical antennas provide transmission and/or reception capabilities for the various Communication System components. The Communication System (Figure 15-8) contains the following antennas: uhf stub, uhf whip, uhf ascent, and two sets of C-band helical antennas. Each set of C-band antennas consist of a power divider and three helical antennas. Three coaxial switches permit antenna and transmitter/receiver switching for best DCS and telemetry communication with the ground stations during the various phases of the mission (pre-launch, launch, orbit, and docking).

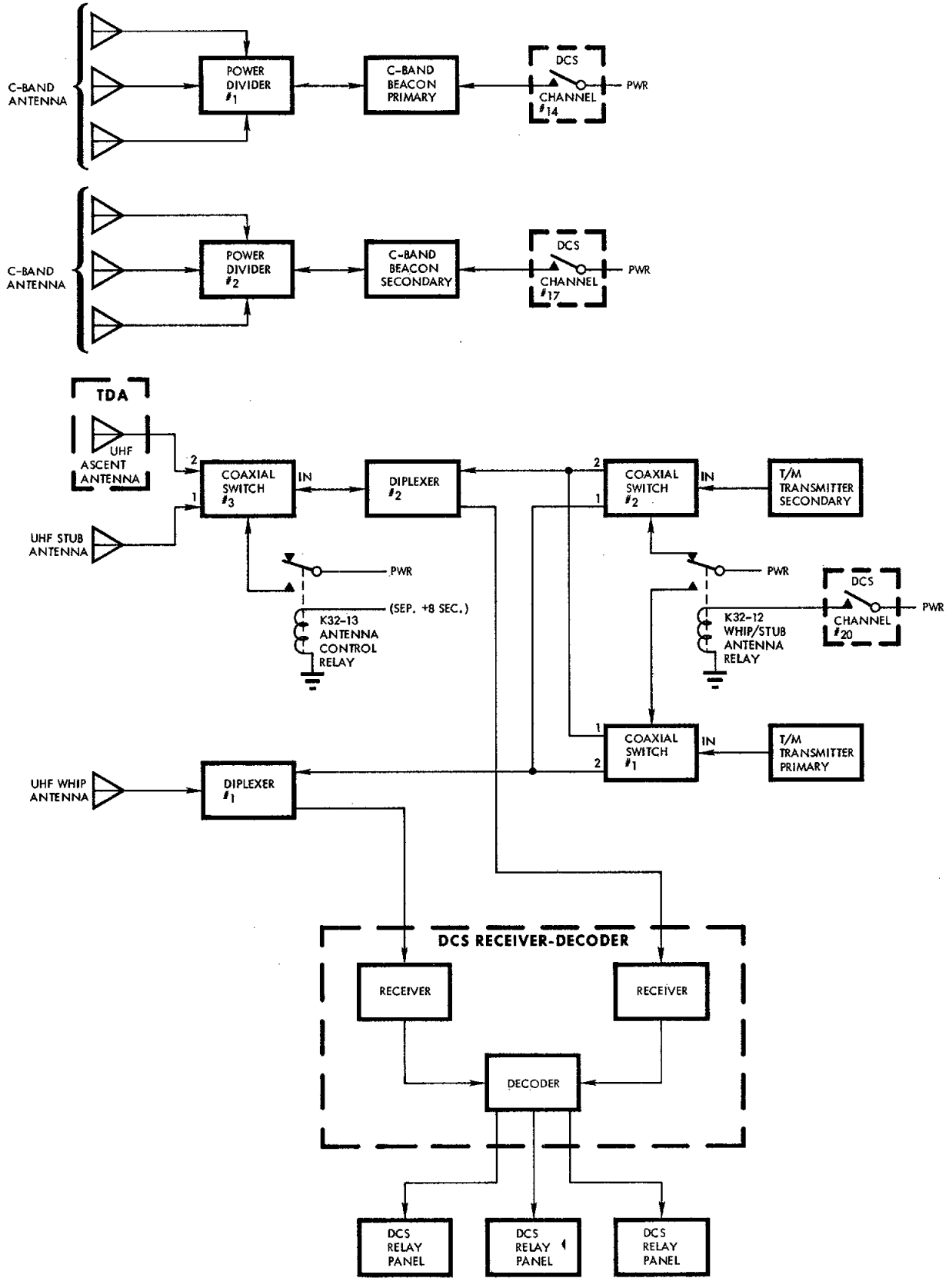


Figure 15-8 Communication System Block Diagram

BEACONS

Two C-band radar beacons establish the capability of tracking the ATDA during the mission. The beacons are transponders which, when properly interrogated by the ground station, transmit signals for accurate ATDA tracking. The radar beacons are controlled by DCS commands from the ground stations.

TELEMETRY TRANSMITTERS

Two identical transmitters supply the radio frequency link from the ATDA to the ground for transmission of instrumentation data. During pre-launch the telemetry transmitters are used for pad checkout of the ATDA. During orbit, telemetry transmission is made while the ATDA is in range of a ground station.

DIGITAL COMMAND SYSTEM

The DCS is the communication link for ground commands to the ATDA. The DCS receives and decodes command transmissions from the ground stations. These commands are used to operate relays which control the operation of various ATDA equipment.

The DCS consists of a receiver-decoder and three relay units located in the equipment section. The DCS operates from pre-launch throughout the mission.

SYSTEM OPERATION

The ATDA Communication System operates from pre-launch throughout the mission. The sequence and theory of operation of the Communication System is as described in the following paragraphs and as illustrated in Figures 15-7 and 15-8.

RADAR TRACKING

Radar tracking of the ATDA is accomplished by the use of two C-band radar beacons. Complete redundancy of the tracking capability is available by using

two beacons, each controlled by a DCS channel and operating with its own antennas.

The radar beacons are transponders which, upon reception of a properly coded interrogation signal from the ground station, transmit a pulse modulated return signal. The location of the ATDA is determined by measuring the elapsed time between transmission and reception at the tracking station, compensating for the known time delay of the beacon.

At lift-off, the primary C-band beacon is activated via channel 14 of the DCS. The primary beacon is used for tracking during the launch phase and orbital insertion. The DCS is used to control the operation of the radar beacons during the remainder of the mission. The secondary beacon can be selected via DCS channel 17 should it be needed as a result of primary beacon malfunction.

The ground interrogation signal is coupled from the antenna to the receiver via a ferrite circulator (Figure 15-9). The circulator isolates the receiver from the transmitter to permit the use of a common antenna for reception and transmission. The rf filter is a three stage preselector, employing three separately tuned resonator cavities which gives adequate rf selectivity and protects the mixer crystal from damage by the reflected transmitter power from the antenna.

The output of the preselector is combined with the local oscillator output in the mixer. The mixer consists of a coaxial directional coupler and a crystal. The directional coupler isolates the local oscillator output from the antenna and directs it to the mixer crystal. The local oscillator is a re-entrant cavity type which uses a planar triode to generate the signal required to operate the mixer.

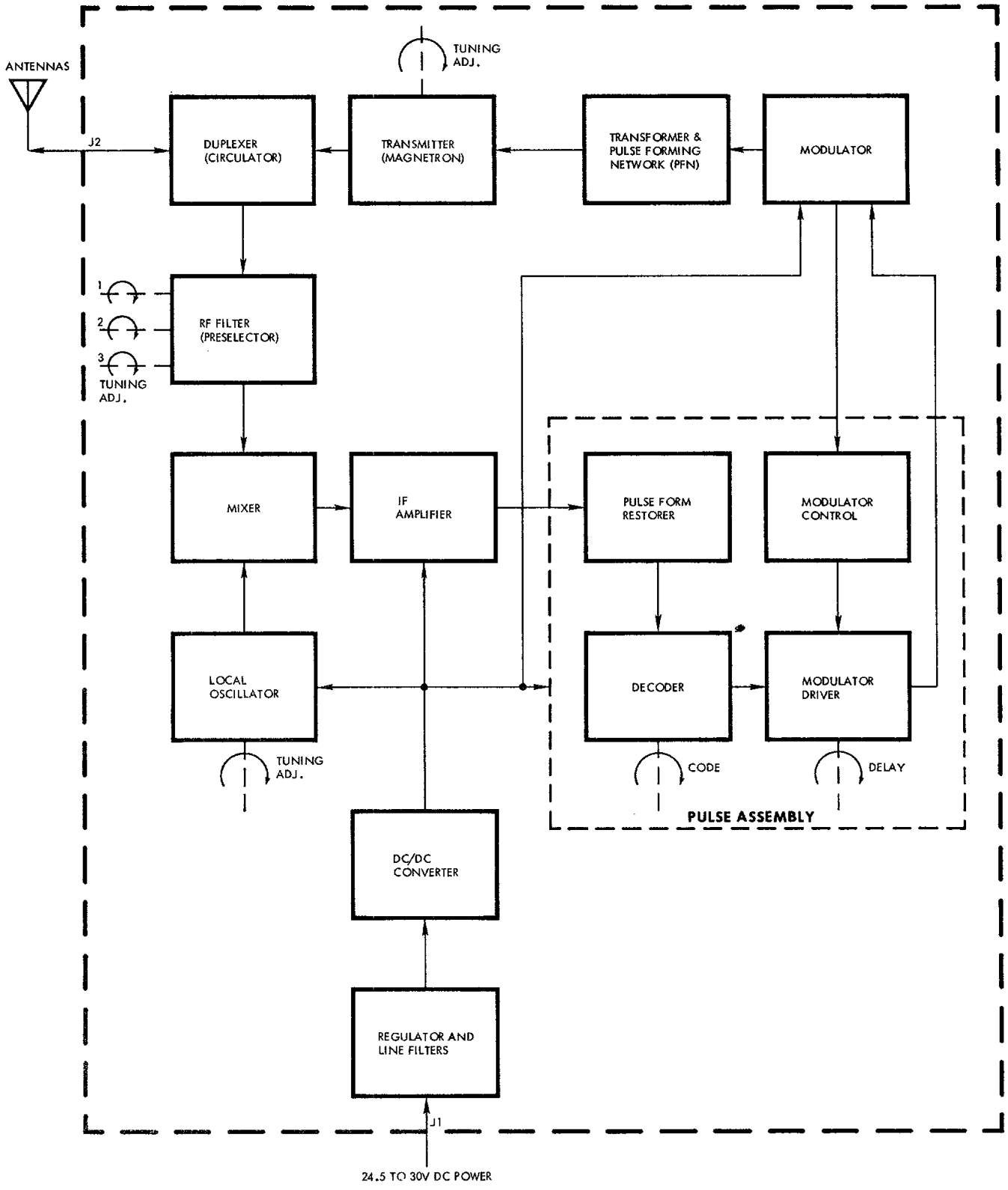


Figure 15-9 C-Band Radar Transponder Block Diagram

The intermediate frequency amplifier is a high gain amplifier composed of an input stage, five amplifier stages, and a video amplifier. The amplified video output is fed to the pulse form restorer circuits which prevent a ranging error due to variations in receiver input signal levels, and also provides a standard amplitude pulse to the decoder for each input signal exceeding its triggering threshold. The decoder determines when a correctly coded signal is received and supplies an output to the modulator driver. The type code to be accepted is selected by the CODE switch. Single pulse, two pulse, or three pulse codes may be selected. The modulator driver and control circuits initiate and control triggering of the transmitter modulator. The modulator driver supplies two fixed values of overall system delay. The desired delay is selected by the position of the DLY switch. The modulator control furnishes the trigger and turn-off pulse for the modulator and limits modulator triggers to prevent the magnetron duty cycle from being exceeded, regardless of the interrogating signal frequency. The modulator circuit employs silicon controlled rectifiers which function similar to a thyratron, but require a much shorter recovery time.

The associated modulator pulse forming network and transformer provide the necessary pulse to drive the transmitter magnetron. The desired pulse width is selected by the internal connections made to the pulse forming network.

The transponder power supply consists of input line filters, a series regulator, and a dc-dc converter. The power supply furnishes the required regulated output voltages with the unregulated input voltage between 21 and 30 vdc. The converter employs a multivibrator and full wave rectifier circuits.

The output of the transmitter is applied to the power divider via the circulator. The power divider gives equal transmission power to the three helical antenna radiating elements.

DIGITAL COMMAND

The DCS provides a discrete link between ground command and the ATDA. The discrete link enables the ground command to control the operation of the various ATDA systems. The command transmissions from the ground are received and decoded by the DCS and are used to operate DCS relays that control power directly or energize relays in the ATDA that determine equipment usage. During pre-launch the DCS is used for pad checkout of the ATDA systems. Discrete commands are sent to the ATDA via the DCS and their execution is verified from instrumentation data.

The uhf ascent and the uhf whip antennas are used for the reception of DCS ground commands during pre-launch and lift-off through ATDA separation. At separation coaxial switch 3 changes DCS reception from the uhf ascent antenna to the uhf stub antenna. The two diplexers permit the use of a common antenna for reception of DCS commands and telemetry transmissions.

The DCS consists of a receiver-decoder and three relay units. The receiver-decoder contains two uhf receivers and the decoder. The two receivers are redundant, and the system will operate properly if only one receiver is functioning. Each of the three DCS relay units contains eight magnetically latched relays. Each relay is set or reset by a DCS command.

The DCS receives phase shift keyed frequency modulated signals composed of a reference and an information signal. The information signal is in phase with

the reference for a logical 1 and 180 degrees out of phase with the reference for a logical 0, thus establishing the necessary requirements for digital data.

The ground command to the DCS is a 12-bit message. Each bit consists of five sub-bits. The five sub-bits are coded to represent a logical 1 or 0. The first three bits of each message is the vehicle address. The second three bits of the message is the system address specifying real time command. The last six bits in the message contain a five-bit relay number and a one-bit relay set-reset discrete command. Table 15-1 lists the DCS channels and contains a brief description of their function.

The DCS uses two types of commands. One type command is executed the moment it is received. This type command operates DCS channel relays which directly control power to the selected ATDA system. Power is applied in the set condition and removed in the reset condition.

The other type command is received but not immediately executed. These commands operate DCS channel relays that connect the control circuits of selected ATDA systems to a pulse bus. The commands are executed when the pulse bus is energized by an execution command. The channels marked by an asterisk in Table 15-1 indicate the DCS commands which require an execution command.

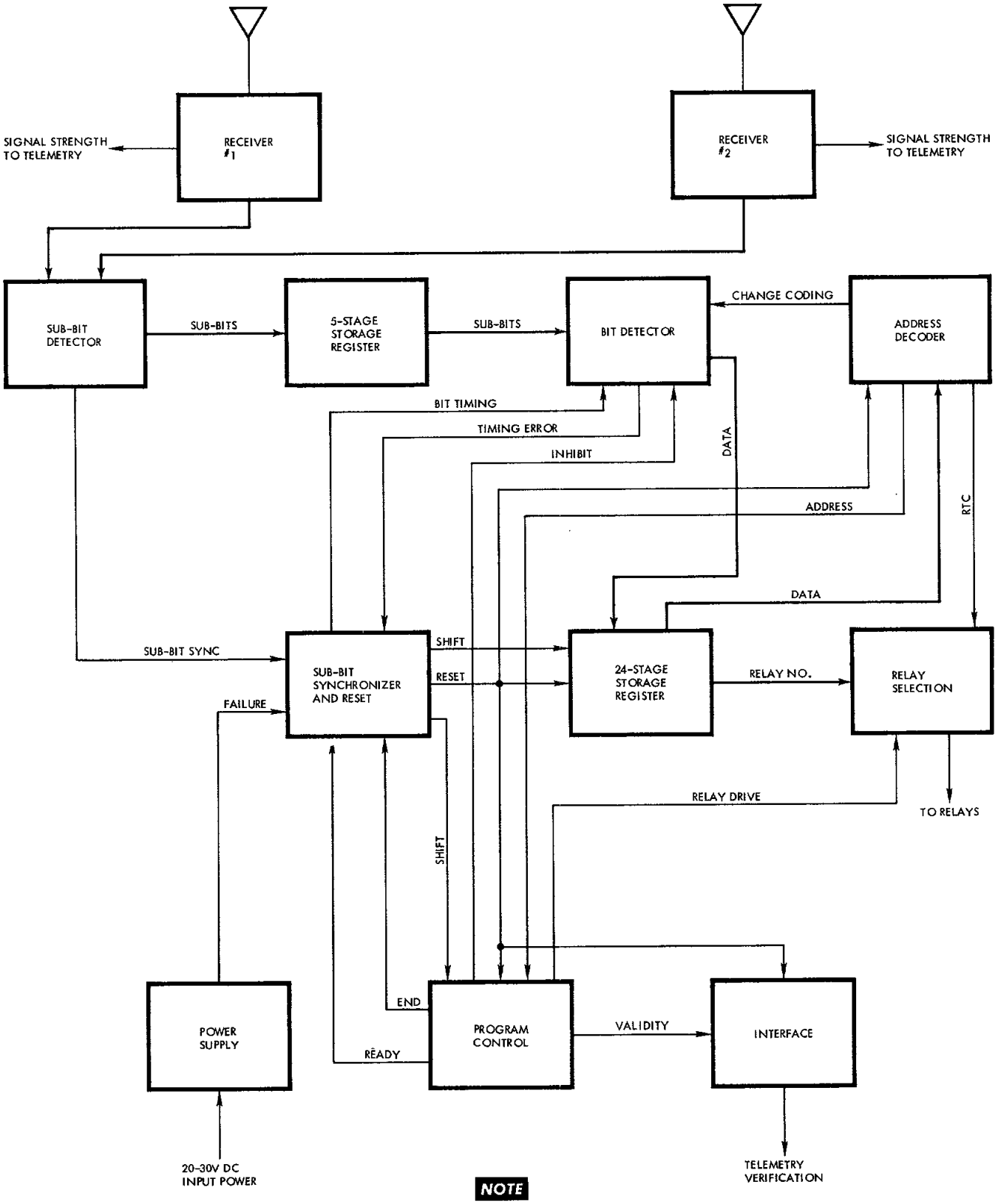
A block diagram of the DCS receiver-decoder is shown in Figure 15-10. Basically, the block diagram consists of a receiver, a decoder, and a power supply common to both sections.

DCS COMMANDS	
CHANNEL	FUNCTION
*1	ARM AND DISARM BUS FOR SQUIB POWER
2	SELECT SPIRAL ANTENNA FOR L-BAND TRANSPONDER
*3	COMMAND UNRIGIDIZE AND UNLATCH
*4	COMMAND RIGIDIZATION OF TDA CONE WHEN LIMIT SWITCHES CLOSE
*5	TURN ACQUISITION LIGHTS ON OR OFF
*6	TURN PRIMARY CONVERTER AND TRANSMITTER ON OR OFF
*7	TURN PRIMARY STABILIZATION SYSTEM ON OR OFF
*8	SELECT RCS RING A OR B
*9	COMMAND SEPARATION SEQUENCE
*10	TURN SECONDARY CONVERTER AND TRANSMITTER ON OR OFF
*11	TURN SECONDARY STABILIZATION SYSTEM ON OR OFF
*12	TURN RCS RING POWER ON OR OFF
*13	SELECT BIASED RATE DAMPING OR NORMAL RATE DAMPING TSS MODE
14	TURN PRIMARY C-BAND BEACON ON OR OFF
**15	EXECUTE PRESELECTED DCS COMMANDS
16	PRIMARY OR SECONDARY DCS EXECUTE SELECT
17	TURN SECONDARY C-BAND BEACON ON OR OFF
**18	EXECUTE PRESELECTED DCS COMMANDS
*19	TURN L-BAND TRANSPONDER ON OR OFF
*20	CONTROL SELECTION OF WHIP OR STUB ANTENNA
*21	COMMAND SEPARATION SEQUENCE
22	ENABLE OR DISABLE L-BAND TRANSPONDER
23	TURN OFF TDA LIGHTS WITH THE EXCEPTION OF ACQUISITION LIGHTS
24	NOT ASSIGNED

* INDICATES DCS COMMANDS THAT REQUIRE AN EXECUTION COMMAND

** INDICATES DCS EXECUTION COMMAND

Table 15-1



NOTE

1. HEAVY LINES DENOTE DATA FLOW.

Figure 15-10 Digital Command System Block Diagram

The audio outputs of the two receivers are linearly summed in an emitter follower of the sub-bit detector module. The sub-bit detector converts the audio to sub-bits. The 5-stage shift register provides buffer storage for the output of the sub-bit detector. The states of the five stages of the shift register represent the sub-bit code. When a proper sub-bit code exists in the shift register, the bit detector produces a corresponding 1 or 0 bit. The output of the bit detector is applied to the 24-stage shift register.

The sub-bit synchronizer counter produces a synchronizing bit output for every five sub-bits. The synchronizing bit is used to gate the 24-stage shift register.

When a message is received, the vehicle address is inserted into the first three stages of the 24-stage shift register. If the vehicle address is correct, the vehicle address decoder circuit will produce an output to the bit detector which changes the acceptable sub-bit code for the remainder of the message. The next three bits of the message, the system address, are inserted into the first three stages of the 24-stage shift register, displacing the vehicle address to the next three stages. The system address decoder circuit identifies the specific address and sets up the DCS to handle the remainder of the message.

When the system address is a real time command, the message is inserted into the first six stages of the 24-stage shift register and the system address and vehicle address are shifted into the next six stages. The real time command selection circuit recognizes the first stage of the 24-stage shift register to be a relay set or reset function and will apply a positive voltage to set or reset all relay coils, as applicable. The real time command selection gates select the proper relay from the relay number stored in the 24-stage shift register and

provides an output which applies power to the coil of the selected relay.

Upon completion of data transfer or if the system to which the data was transferred fails to respond within 100 milliseconds, the DCS will reset in preparation for the next message. The DCS will also reset in the event of a timing error in transmission of data, or if the DCS power supply voltages become out of tolerance.

The DCS power supply operates from the ATDA main power bus and supplies the receivers and decoder with regulated dc voltages.

A verification signal is supplied by the decoder for telemetry transmission when a ground command has been received.

TELEMETRY

Transmission of instrumentation data is accomplished by the telemetry transmitters. Operation of the transmitters is controlled by the ground station via DCS channels 6 and 10. An interlock in the ATDA prevents both transmitters from operating at the same time.

Coaxial switches select the antenna used for telemetry transmission. The uhf ascent and uhf whip antennas are used during pre-launch checkout and lift-off through ATDA separation. After separation the output connection of coaxial switch 3 is switched from the uhf ascent antenna to the uhf stub antenna.

Data from the Instrumentation System programmer is supplied to the telemetry transmitters in non-return-to-zero pulse code modulated pulse trains. The transmitters relay the information to the ground stations in a digital format composed of 1's and 0's at a rate of 51.2 kilobits per second. The carrier

frequency is deviated to the higher frequency deviation limit in order to transmit a 1 and to the lower deviation limit to transmit a 0. Figure 15-11 is a block diagram of the telemetry transmitters. The transmitters operate on the same frequency and are identical in operation.

The input signal is amplified by the video amplifier and is used to modulate the output of the oscillator. The oscillator is crystal controlled for good frequency stability. The modulated signal is passed through a series of buffer amplifiers and two phase shifters. The buffer amplifiers increase the signal level and isolate the crystal circuit from the frequency multipliers. The phase shifters provide impedance matching of the crystal oscillator to improve signal linearity for large deviations in frequency.

A times-four multiplier, a power amplifier, and a times-three multiplier increase the carrier frequency and power to the desired output values. The bandpass filter minimizes spurious radiations at the output of the transmitter.

The line filter prevents noise on the input power bus from affecting transmitter operation and prevents transients generated within the transmitter from feeding back to the input power bus.

The transmitter dc-dc converter is a completely encapsulated unit employing transistors, diodes and a transformer to provide regulated outputs of 30 vdc and 70 vdc from unregulated input voltage of 18 to 30.5 vdc. The converter is a constant power input type, thus minimizing the heat dissipation caused by high voltage inputs.

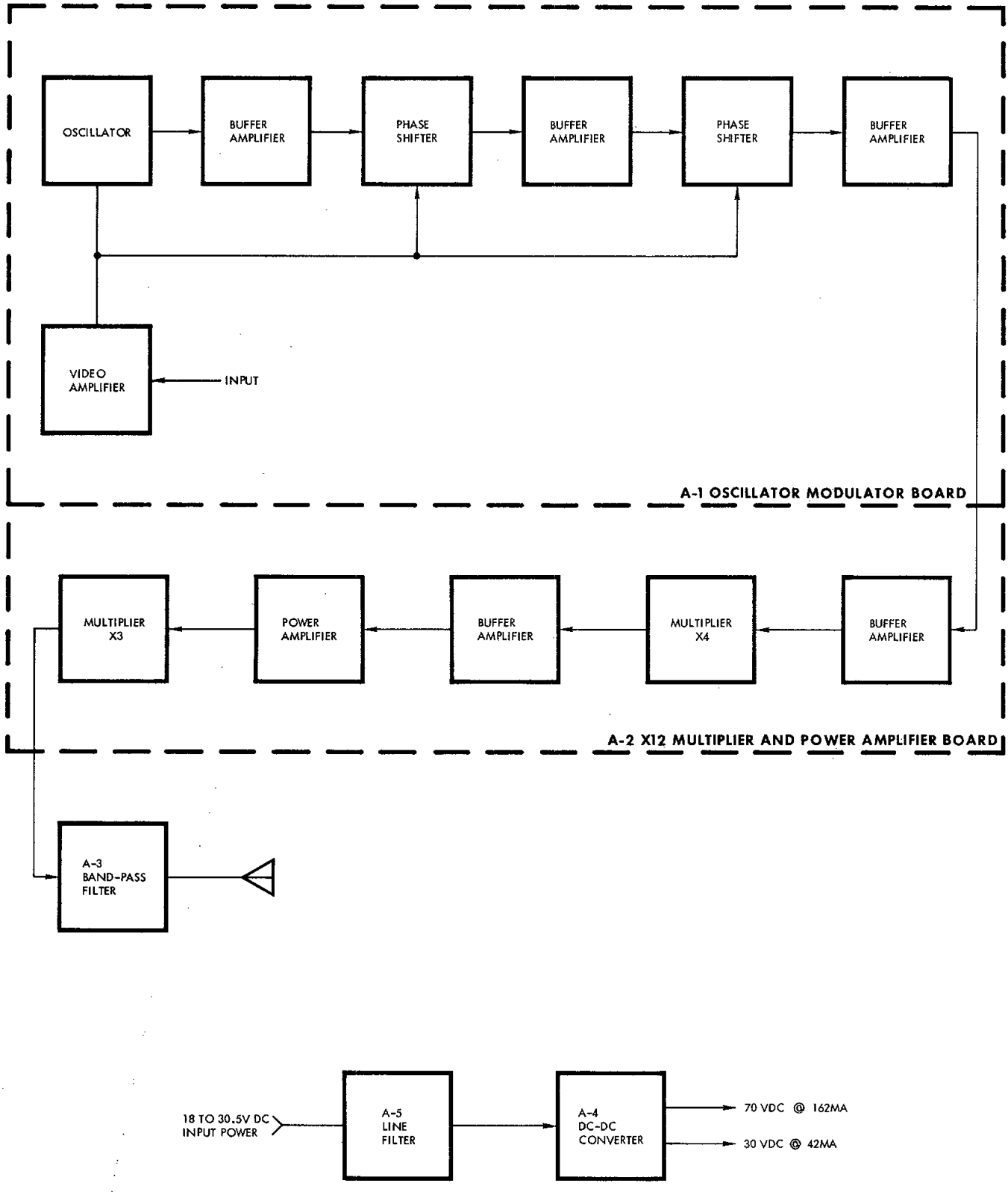


Figure 15-11 Telemetry Transmitter Block Diagram

SYSTEM UNITS

ANTENNAS

UHF Antennas

The three uhf antennas (ascent, stub and whip) provide simultaneous transmission for the telemetry transmitters and reception for the DCS receivers. The uhf ascent antenna is used during pre-launch for pad checkout and from lift-off until ATDA separation. The uhf stub antenna gives omnidirectional roll coverage after ATDA separation. Switching from the ascent antenna to the stub antenna is accomplished by coaxial switch 3 at ATDA separation. The uhf whip antenna provides omnidirectional yaw coverage and is used from pre-launch throughout the mission. The antennas have a quarter-wavelength radiation pattern.

UHF Ascent Antenna

The uhf ascent antenna (Figure 15-12) is mounted on the Target Docking Adapter cone where it is protected from wind blast and launch temperatures during lift-off by the ascent shroud.

The uhf ascent antenna is approximately 16 inches long. The antenna element consists of two 1/2-inch wide gold-plated steel blades which are bolted together. For rigidity, the antenna element is shaped in a 0.5 inch wide arc having a radius of 1.5 inches. The two laminations of steel blades, compounding a single antenna element, are rigidly secured together with nuts and bolts at the top and bottom.

UHF Stub Antenna

The uhf stub antenna, physically constructed as illustrated in Figure 15-12, is mounted on the battery module. The antenna protrudes aft from the battery module

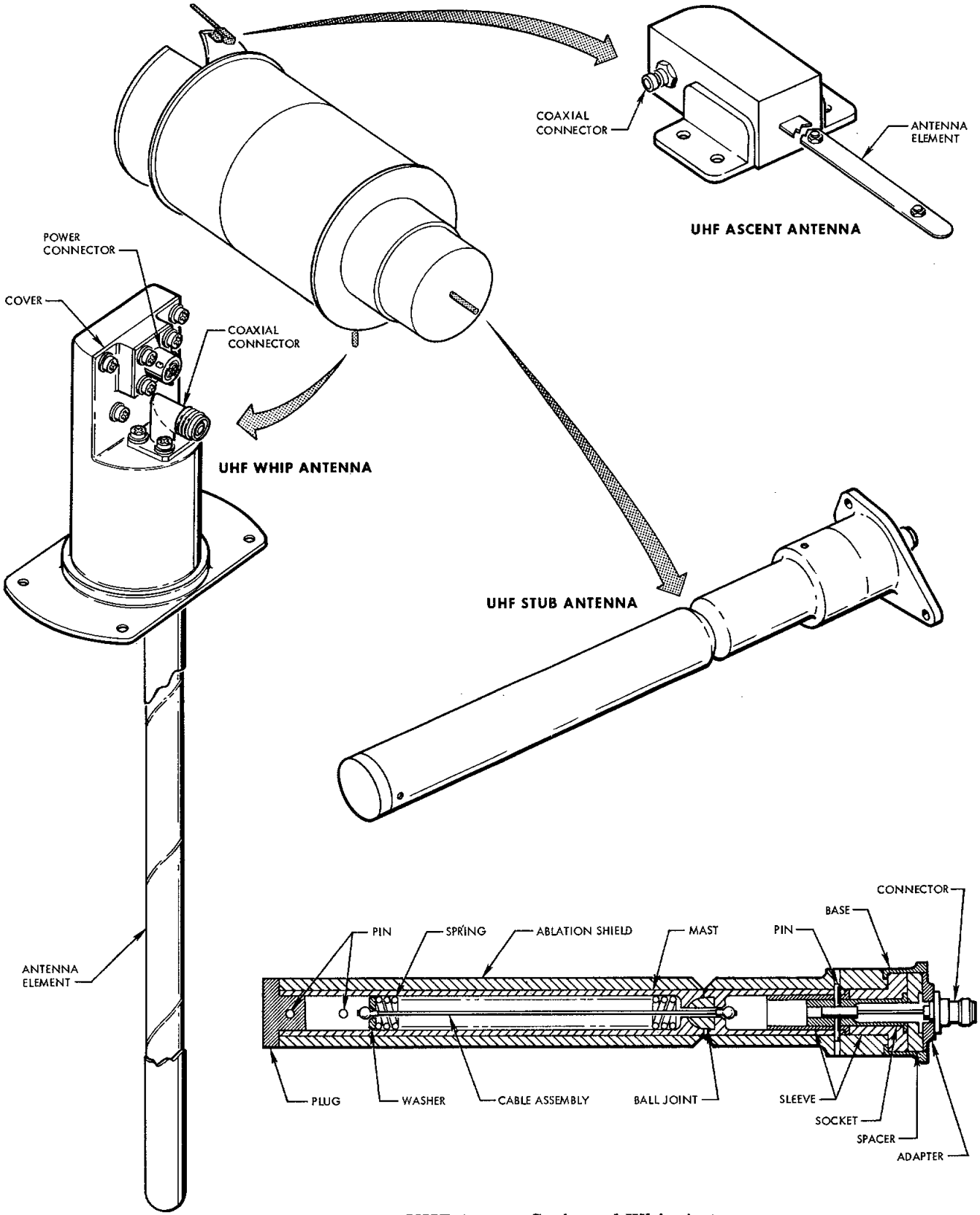


Figure 15-12 UHF Ascent, Stub, and Whip Antennas

into the forward section of the launch vehicle. This protects it during the boost phase of the mission.

The antenna consists of a mast and base which weighs approximately 1.1 pounds. The mast is constructed of 3/4-inch cobalt steel, machined to tubular form, and covered by a Teflon ablation shield. The antenna is approximately 13.5 inches long including the connector, and 1.25 inches in diameter over the ablation material. The radiating length of the antenna is approximately 11.2 inches long. The mast consists of two sections. The front section is mounted on a cobalt steel ball joint and retained to the rear section by a spring loaded cable. Electrical contact between the mast sections is made through the ball joint and the spring loaded cable assembly. The ball joint allows the front section of mast to be deflected to approximately 90 degrees in any direction around the antenna axis. The spring of the cable assembly is preloaded to approximately 45 pounds to cause the front section, when deflected, to return to the erected position.

The rf connector is press fitted into a socket and makes contact to the mast through the socket and sleeve, which are the same material as the mast. The shell of the rf connector is mounted to the base which is isolated from the mast by a Teflon spacer and sleeve.

UHF Whip Antenna

The uhf whip antenna is mounted on the equipment section of the ATDA. The whip antenna is extended during pre-launch and is used throughout the mission. A fiberglass fairing protects the antenna from damage during launch.

The antenna element (Figure 15-12) is a tubular device made from a 2-inch wide beryllium copper strip processed in the form of a tube. The antenna forms an

element that is approximately 9.2 inches long and 1/2 inch in diameter.

C-Band Antennas and Power Dividers

Two sets of C-band helical antennas supply the transmission and reception capability for the C-band radar beacons. Each set of helical antennas consists of three radiating elements (antennas) and a power divider. The three antennas have a radiation pattern with three symmetrically located lobes. Two of the lobes are oriented toward the earth when the ATDA is roll-stabilized in flight.

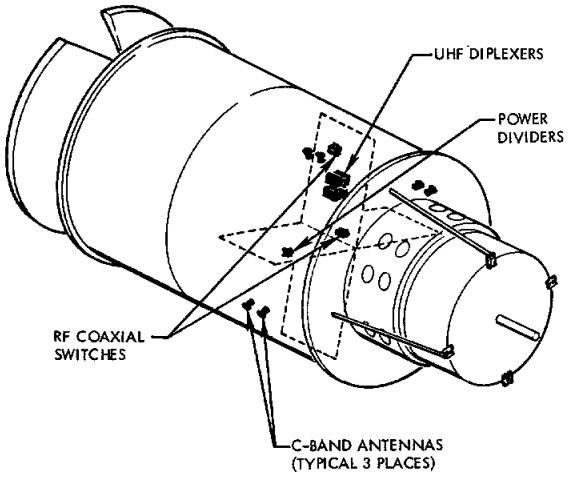
The C-band helical antennas (Figure 15-13) are mounted on the equipment section flush with the outside skin of the ATDA and spaced 120 degrees apart. Each antenna unit is approximately 3.4 inches long, 1.8 inches wide, has a depth of 2.21 inches over the connector, and weighs approximately 3.5 ounces.

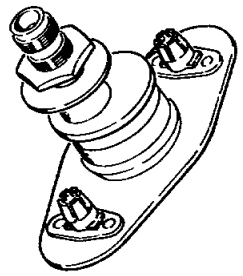
The power divider (Figure 15-13) measures approximately 3.86 inches over the connectors, 4.0 inches over the tuning knobs, and weighs about 6.5 ounces.

The power divider is basically a cavity type power splitter. During beacon transmission, power is delivered to the power divider where it is divided equally among the C-band radiating elements. The power divider contains a double stub tuner to compensate for mismatch between the C-band beacon and the C-band radiating elements. Tuning is accomplished by means of a self-locking tuning shell located underneath each tuning stub cap.

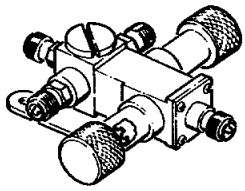
Diplexers

The two uhf diplexers provide isolation between the DCS receivers and the telemetry transmitters to permit the use of a common antenna for both transmission and reception. The diplexers are located in the ATDA equipment section. The

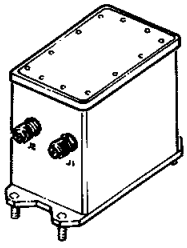




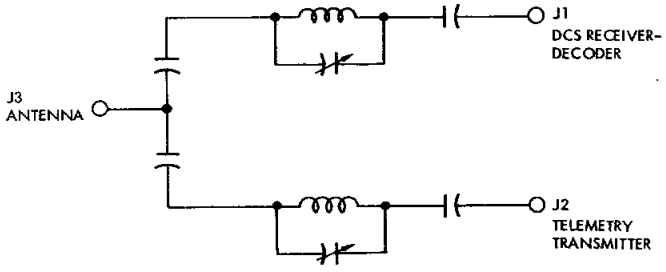
C-BAND HELICAL ANTENNA

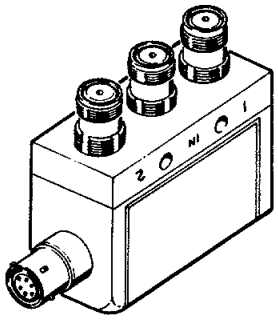


POWER DIVIDER



UHF DIPLEXER





RF COAXIAL SWITCHES

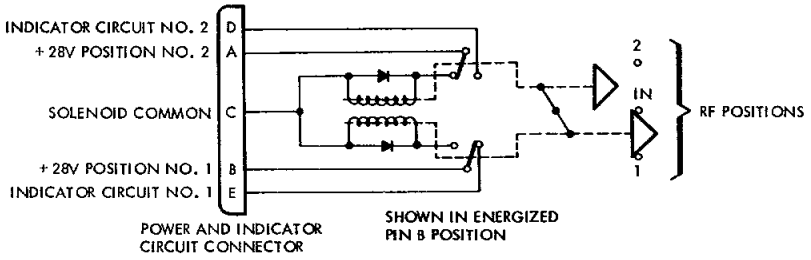


Figure 15-13 C-Band Antennas, Diplexers, And Coaxial Switches

physical representation and a schematic of the diplexers is illustrated in Figure 15-13.

The diplexer is approximately 4.5 inches wide, 4 inches high, and 2.7 inches deep, contains two input and one output connectors, and weighs approximately 1.25 pounds. Each channel consists of a high Q cavity, tuned to the corresponding operating frequency. All channels are isolated from each other without appreciably attenuating the rf signals passing through it. Each channel can be re-tuned if the assigned operating frequency is changed.

Coaxial Switches

The three coaxial switches permit antenna and transmitter/receiver selection. The coaxial switches are located in the ATDA equipment section. The physical construction and approximate location of the coaxial switches is illustrated in Figure 15-13.

Coaxial switch 1 is used for antenna selection for the primary telemetry transmitter.

Coaxial switch 2 is for antenna selection for the secondary telemetry transmitter.

Coaxial switch 3 provides antenna selection for one of the DCS receivers and either of the telemetry transmitters.

Each switch contains a power connector, an input connector, two output connectors, and weighs approximately 0.5 pounds. The dimensions of each switch are approximately 2.65 inches long, 1.82 inches high, and 1 inch wide. The three coaxial switches are identical and may be used interchangeably. Basically, the coaxial

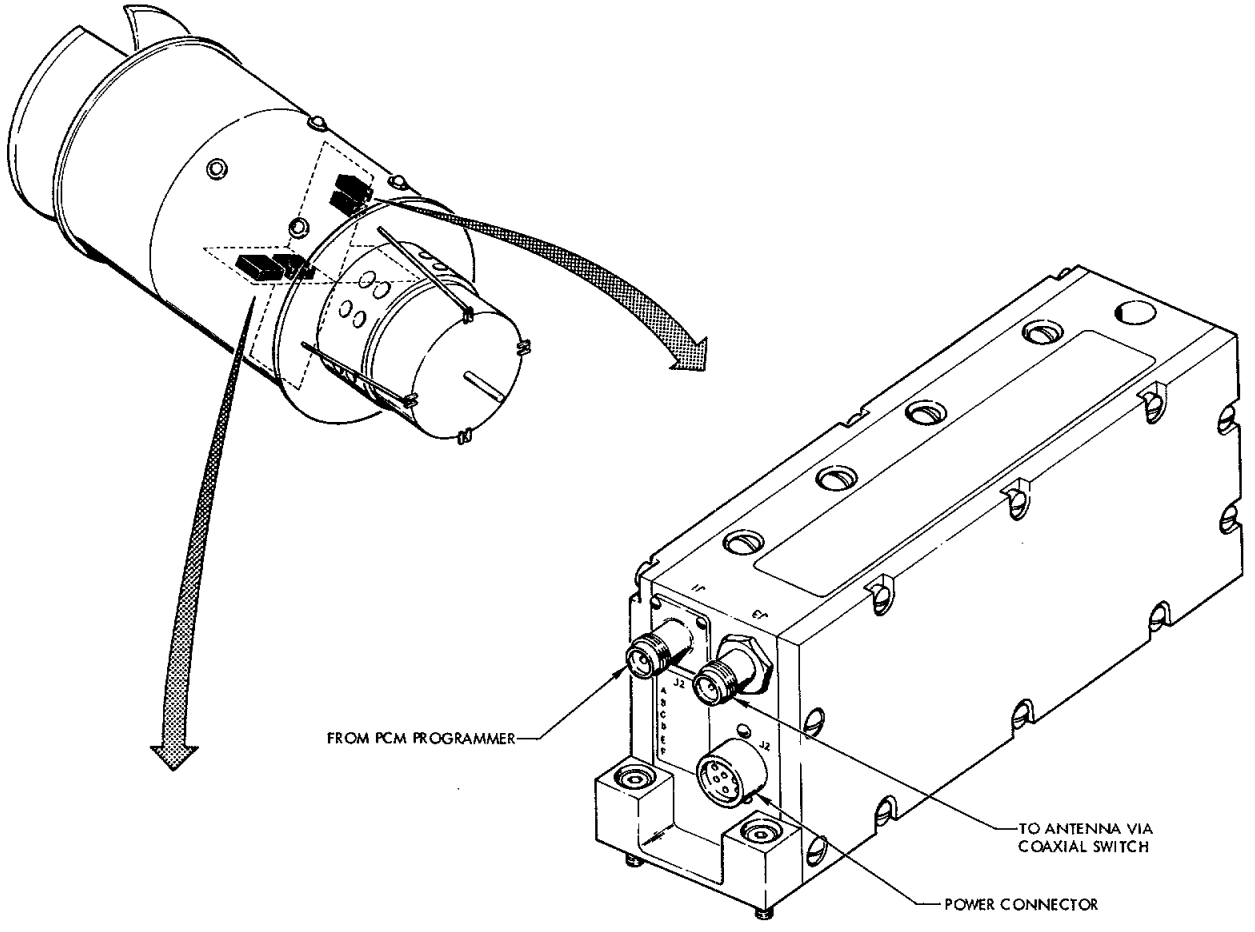
switches supply single-pole double-throw switching action as illustrated in Figure 15-13. The switch, having a 20 millisecond maximum operation time, operates on 3 amperes at 28 vdc and uses a latching solenoid break-before-make switching action. The coaxial switches are designed to operate from 15 mc to 500 mc, and from 5500 mc to 5900 mc. Pins A and B of each switch are utilized to accomplished the switching action.

BEACONS

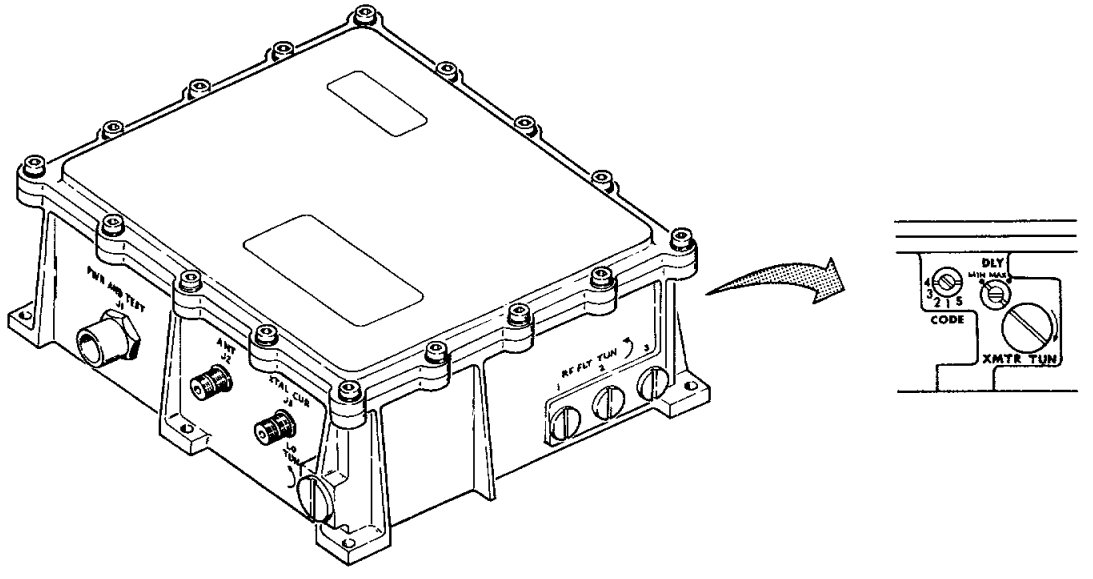
The two C-band radar beacons supply the tracking capability for the ATDA. The two radar beacons are transponders which transmit a reply when properly interrogated. The beacons are located in the equipment section of the ATDA. Each beacon consists of a superheterodyne receiver, a decoder section, and a transmitter. The receivers are tuned to a center frequency of 5690 megacycles and the transmitters to 5765 megacycles. The transmitter peak power output is 500 watts minimum to the antennas. Code spacing for the beacons is 3.0 microseconds. The C-band beacon is a sealed unit and measures approximately 9.34 inches by 8.03 inches by 3.26 inches. As illustrated in Figure 15-14, the beacon has a power and test connector, an antenna connector, and a crystal current test point connector. The beacon contains external adjustments for local oscillator, preselector (rf filter), and transmitter tuning; switches for selecting the desired interrogation code; and one of two preset transponder fixed delay times. These adjustments and switches are accessible by removing pressure sealing screws. The beacon employs solid-state circuitry, except for the transmitter magnetron and receiver local oscillator.

TELEMETRY TRANSMITTERS

The telemetry transmitters provide the instrumentation data transmission capabil-



TELEMETRY TRANSMITTER



C-BAND RADAR BEACON

Figure 15-14 C-Band Beacon and Telemetry Transmitters

ity. The telemetry transmitters are located in the ATDA equipment section. The physical construction and approximate location of the telemetry transmitters is illustrated in Figure 15-14.

The two telemetry transmitters are identical. They weigh about 41 ounces and are 2.75 inches high, 2.25 inches wide and are 6.5 inches long. Each transmitter contains a dc power connector, an rf output power connector, and a video input connector.

The telemetry transmitters are solid-state frequency modulated transmitters. Each transmitter consists of an oscillator-modulator, a times-12 (x12) multiplier and power amplifier, a bandpass output filter, a line filter and a dc-dc converter. After a 30-second warm-up, the transmitters are capable of continuous uninterrupted operation for 500 hours. The transmitters operate at 246.3 megacycles with 2.0 watts minimum power output. Peak carrier frequency deviation is $\pm$ 150 kilocycles.

DIGITAL COMMAND SYSTEM

The DCS consists of a receiver-decoder and three relays and supplies the link for ground commands to the ATDA. The DCS components (Figure 15-15) are located in the equipment section of the ATDA.

The two DCS receivers operate on a fixed frequency in the 406-450 megacycle range. Each receiver consists of a preselector, local oscillator and multiplier, two intermediate frequency strips, a discriminator, and an output audio amplifier.

The decoder contains a sub-bit detector, a bit detector, an address decoder, a 5-stage and a 24-stage storage register, relay selection circuits, and a synchronizing and reset circuit.

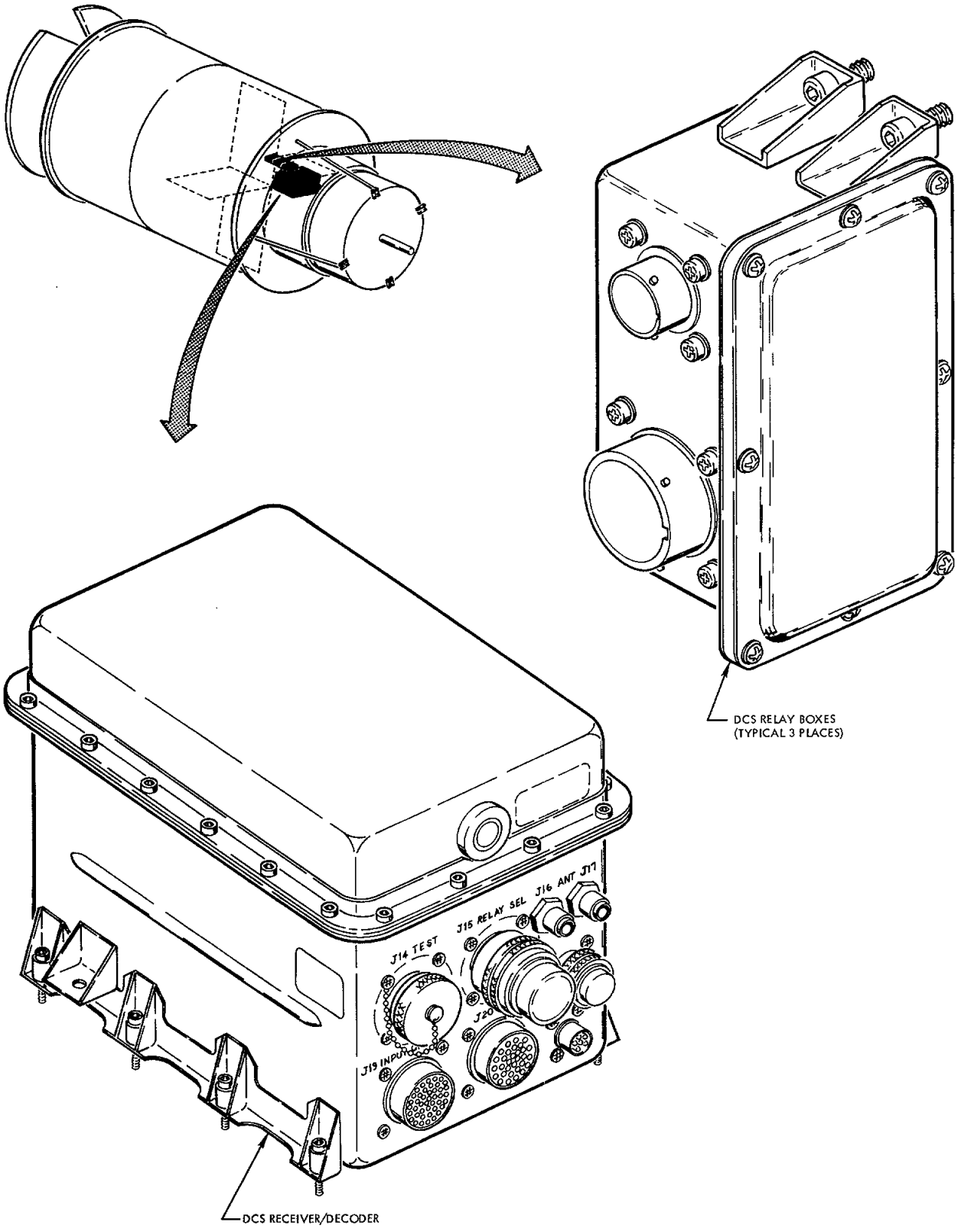


Figure 15-15 DCS Receiver/Decoder and Relay Boxes

The receiver-decoder package is approximately 8 inches high, 8 inches wide, and 12 inches long. The relay boxes are identical. Each relay box is approximately 2.25 inches wide, 5 inches high, and 3 inches deep. The combined weight of the receiver-decoder package and the relay boxes is approximately 23 pounds. The receiver-decoder package contains two uhf receivers and a decoder while each of the relay boxes contain eight relays.

INSTRUMENTATION SYSTEM

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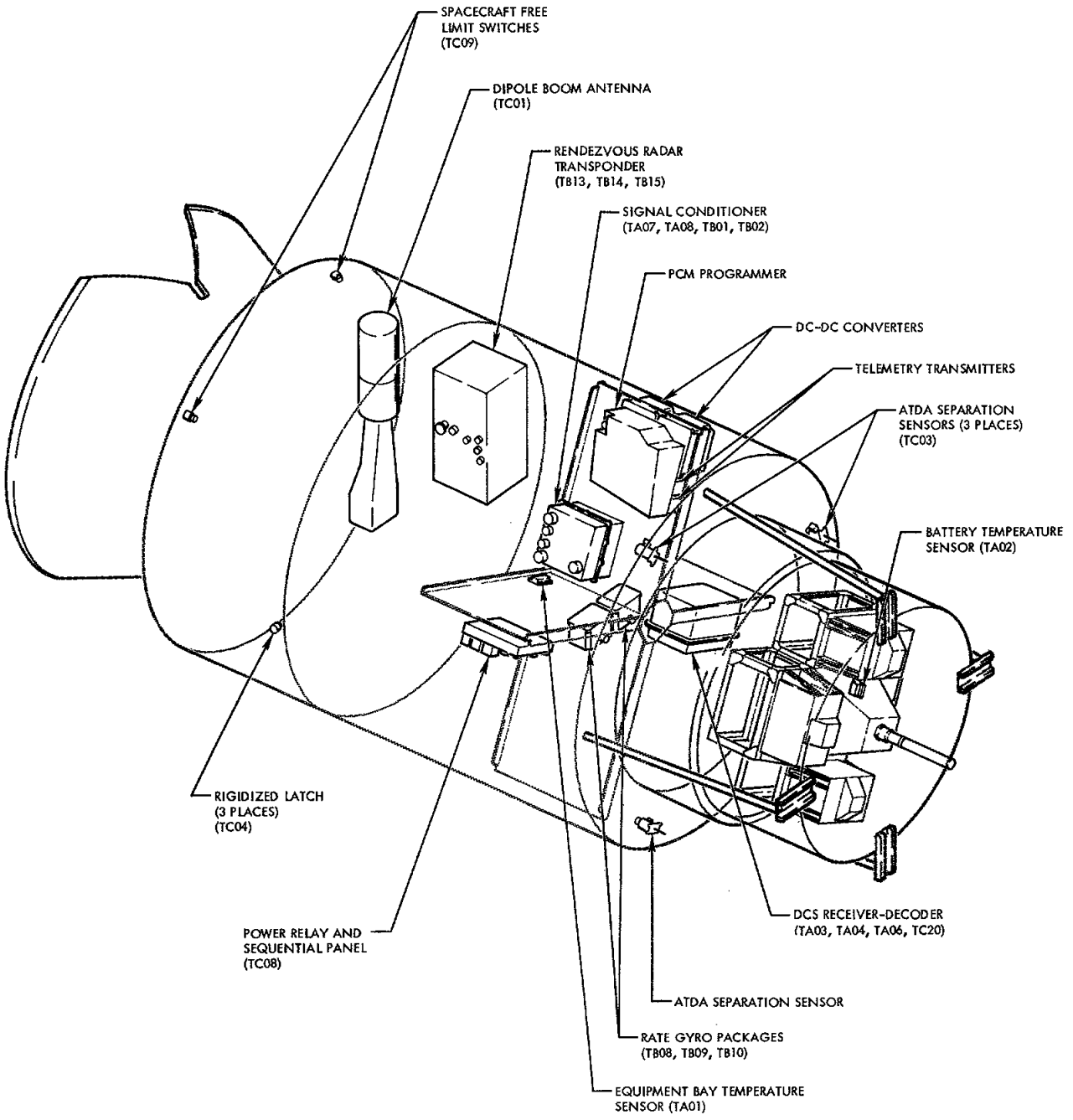


Figure 15-16 Instrumentation System

INSTRUMENTATION SYSTEM

SYSTEM DESCRIPTION

The Instrumentation System provides a means of data acquisition with respect to the progress and condition of the ATDA. Data acquisition in the ATDA is the sensing of specific conditions or events on board the ATDA and displaying the data derived from these inputs to ground operation personnel. In this respect the data acquisition is shared by all ATDA systems and the ground operation support system.

The basic components of the Instrumentation System are: temperature sensors, signal conditioners, dc-dc converters, and a programmer. The location of the Instrumentation System components and various signal monitoring points are illustrated in Figure 15-16.

The Instrumentation System supplies 43 parameters (measurements) to the ground station. The system primarily monitors ATDA system parameters. These parameters are used for determining the progress of the ATDA and the performance of the various systems, and for making decisions concerning mission safety and success. During pre-launch checkout of the ATDA, ground commands are given and their execution is verified from instrumentation data. Equipment status and sequential switch positions are monitored to insure all ATDA systems are in lift-off configuration. During the orbital phase of the mission, the Instrumentation System supplies real-time telemetry information while the ATDA is within range of the ground station.

SYSTEM OPERATION

The Instrumentation System is controlled by ground commands via the Digital Command System (DCS). DCS channel 6 is used to apply main bus power to the primary dc-dc converter and telemetry transmitter; while channel 10 controls power to the secondary converter and telemetry transmitter. An interlock in the wiring arrangement of the power control relays (K33-1 and K33-2) prevent application of power to the primary and secondary units simultaneously. A block diagram of the ATDA Instrumentation System is illustrated in Figure 15-17.

The Instrumentation System obtains data from temperature sensors, pressure transducers, switch and relay actuations, and various monitoring points in the ATDA systems. The majority of the signals obtained are compatible with the multiplexing and encoding equipment without alteration. Some of the signals, however, are routed to a signal conditioner where their characteristics and/or amplitudes are changed. The resulting signals, as well as those from other sensors, are of three basic types: low-level (0 to 20 millivolts dc), high-level (0 to 5 vdc), and bi-level (0 or 28 vdc). The programmer combines and converts these inputs into a serial binary-coded digital signal and supplies it to the telemetry transmitters.

PCM PROGRAMMER OPERATION

The basic functions of the Pulse Code Modulated (PCM) programmer are: data multiplexing; timing, to support the multiplexing functions; and analog to digital conversion. The output of the programmer to the transmitter is a pulse code modulated pulse train. A block diagram of the PCM programmer is illustrated in Figure 15-18.

The basic components of the programmer are a high-level analog subcommutator, prime subcommutator, master commutator, analog to digital converter, output

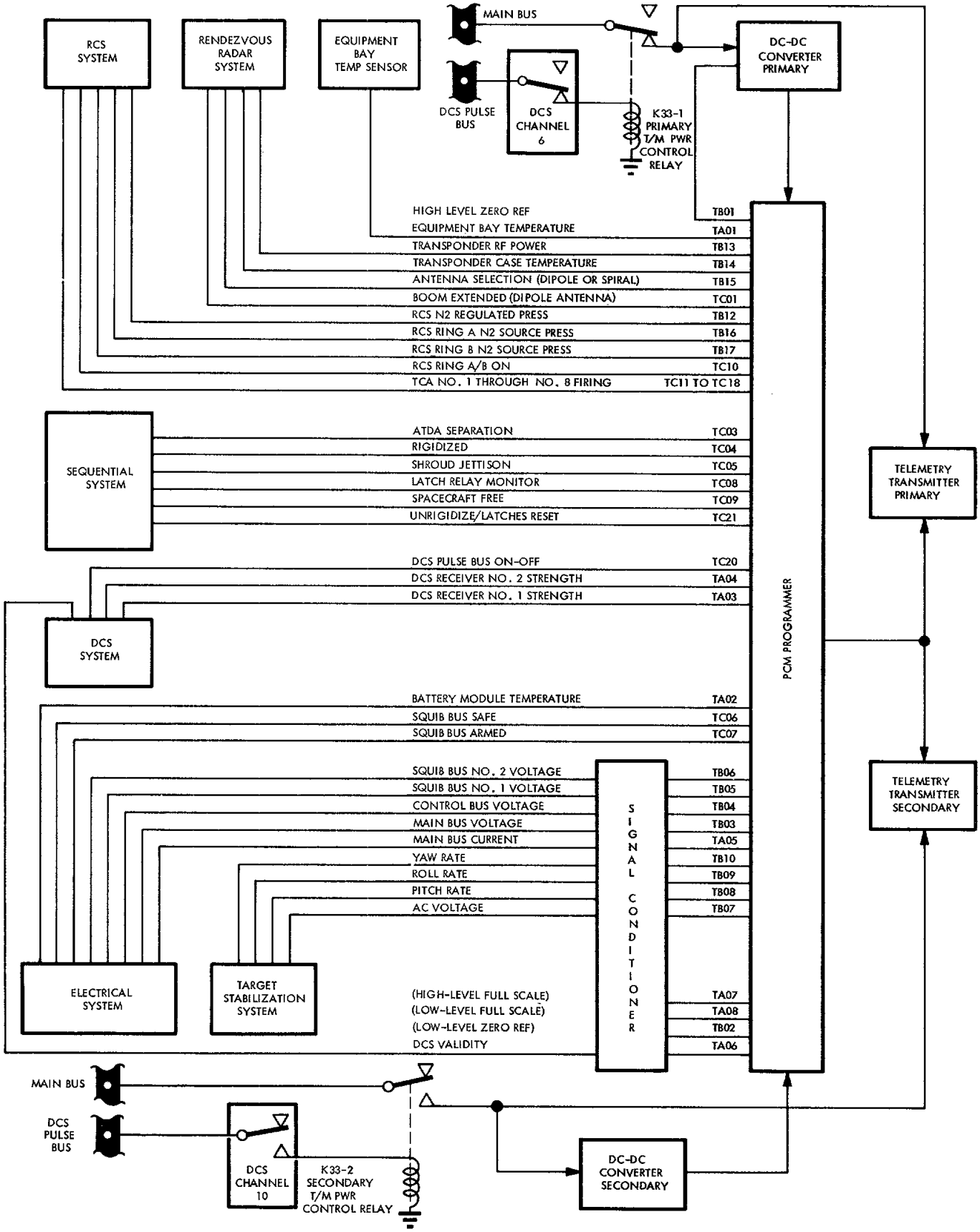


Figure 15-17 Instrumentation System Block Diagram

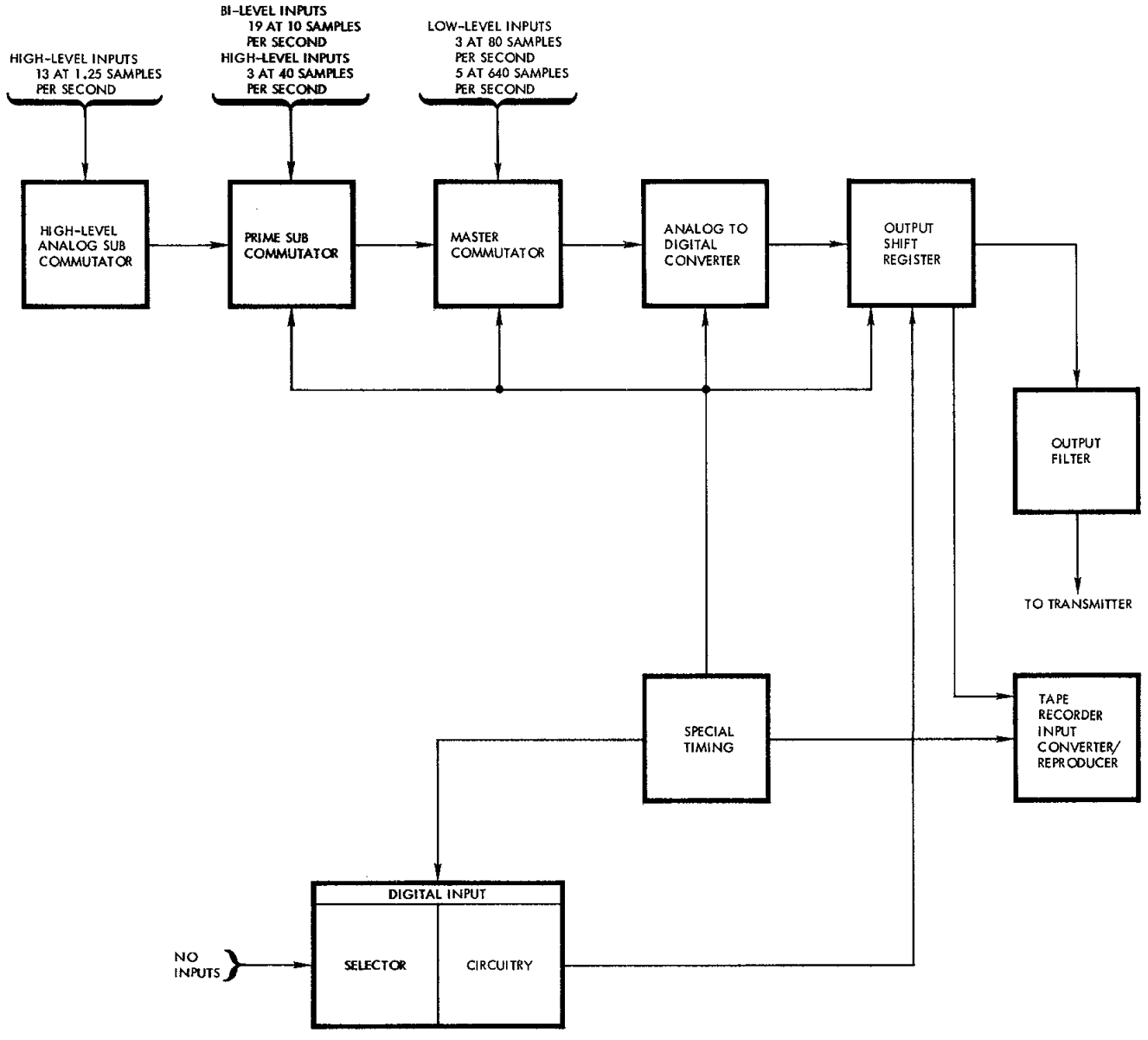


Figure 15-18 PCM Programmer Block Diagram

shift register, digital input selector and input circuitry, special timing, output filter, and tape recorder input converter/reproducer.

The programmer high-level analog subcommutator samples 13 high-level inputs at 1.25 samples per second. The analog subcommutator receives its inputs directly from the signal sources, or from the signal conditioner. The output of the analog subcommutator is applied to the prime subcommutator for further multiplexing.

The prime subcommutator, in addition to accepting the sampled high-level analog subcommutator data output, samples 19 bi-level signals at 10 samples per second, and 3 high-level signals at 40 samples per second. The prime subcommutator supplies its output to the master commutator.

In addition to accepting the inputs from the prime subcommutator, the master commutator samples 5 low-level signals at 640 samples per second and 3 low-level signals at 80 samples per second. The output of the master commutator is applied to the analog to digital converter where the analog output from the master commutator is converted to digital data.

The digital data from the output of the analog to digital converter is applied to the output shift register which provides a continuous non-return-to-zero PCM pulse train of 51.2 kilobits per second. The output of the output shift register is applied to the telemetry transmitter through the output filter.

Timing of all operations in the PCM programmer is provided by the special timing circuits. Two crystal controlled oscillators and a series of counters are used to supply the gate pulses which determine the sampling rates of the programmer.

The digital input selector and input circuitry and the tape recorder input converter/reproducer components are not utilized during ATDA operation. The Instrumentation System does not sample digital data or record data for data-dump transmission.

The output signal from the PCM programmer is a 51.2 kilobit non-return-to-zero signal with a voltage that is adjustable between 0.1 volt and 1.0 volt peak. The serial output has positive voltage for a 1 and zero or negative voltage for a 0.

A brief description of all ATDA instrumentation parameters is contained in the following paragraphs. The parameters are described in groups identified by their applicable data source system. Data flow from the signal source to the programmer is illustrated in Figure 15-17.

SEQUENTIAL SYSTEM PARAMETERS

The Instrumentation System monitors six Sequential System parameters. The sequential parameters are bi-level signals originating from switch or relay actuations and are supplied directly to the PCM programmer.

During pre-launch checkout of the ATDA, latch relay monitor (TC08) indicates that all control latch relays are in the sequential mode required for launch condition. When all control relays are latched or reset in the proper position a 28 vdc signal is applied to the PCM programmer. One or more relays in an abnormal mode reduces this signal to 0 vdc.

With the ascent shroud installed, a 24 vdc signal is applied to the PCM programmer. At shroud jettison (TC05), the 24 vdc circuit is broken thus reducing the voltage applied to the programmer to 0 vdc. A 24 vdc signal is supplied to the PCM programmer to indicate ATDA separation (TC03). This signal results when any two of the three separation sensor switches close.

Unrigidize/latches reset (TC21) indicates the Target Docking Adapter cone is unrigidized and all latches are reset. When this condition exists a 24 vdc signal is supplied to the PCM programmer. Rigidized (TC04) is obtained from three rigidized limit switches which close when the rigidized limit is reached. When all three limit switches are closed a 28 vdc signal is applied to the PCM programmer. Spacecraft free (TC09) indicates whether or not physical attachment exists between the ATDA and the spacecraft. A 24 vdc signal is fed through one or both of two limit switches to the PCM programmer when complete attachment does not exist.

ELECTRICAL SYSTEM PARAMETERS

The Electrical System supplies the ground station with 8 parameters which indicate the condition of the ATDA batteries. Five of the signals are routed through the signal conditioner to make them compatible with the PCM programmer circuits.

Battery temperature (TA02) indicates the battery case temperature. A resistive element temperature sensor mounted on the battery supplies the low-level signal to the PCM programmer.

Squib bus safe (TC06) and squib bus armed (TC07) are bi-level signals applied to the PCM programmer which indicate the position of the squib control relays. When all relays that control power to squib igniters are deenergized (TC06) a 24 vdc signal is obtained. When any of these relays are energized it removes the 24 vdc signal from the PCM programmer. When one or more of the squib control relays are energized (TC07) a 24 vdc signal is applied to another channel of the PCM programmer.

Main bus current (TA05) is used to monitor the total current being drawn from

the batteries. The signal originates from a 50 millivolt shunt installed at the main bus. The signal is routed to the signal conditioner which provides a low-level signal output to the PCM programmer.

Main bus voltage (TB03), control bus voltage (TB04), squib bus 1 voltage (TB05), and squib bus 2 voltage (TB06) are measured to supply the ground station with information relative to battery condition. The high-level signals are routed through the signal conditioner before being applied to the PCM programmer.

TARGET STABILIZATION SYSTEM PARAMETERS

The Instrumentation System monitors four parameters from the Target Stabilization System. These signals are high-level signals conditioned by the signal conditioner prior to being applied to the PCM programmer.

AC voltage (TB07) gives an indication of the output of the inverter that is in operation. Pitch, roll, and yaw rates (TB08, TB09, TB10) are monitored to allow evaluation of the rate control portion of the Target Stabilization System. Primary and secondary rate package signals are monitored on the same telemetry channel, depending on which system has been selected by the DCS.

DIGITAL COMMAND SYSTEM PARAMETERS

The DCS supplies four parameters for ground monitoring. These signals provide information relative to the operation of the DCS. Three of the four signals are low-level signals.

Receiver 1 signal strength (TA03) and receiver 2 signal strength (TA04) give an indication of the strength of the signals received by the receivers in the DCS.

DCS validity (TA06) is monitored to indicate that a command signal was received by the DCS receiver-decoder.

DCS pulse bus (ON-OFF) (TC20) indicates that power is applied to the DCS execute circuitry. A 28 vdc bi-level signal from the DCS pulse bus is applied to the PCM programmer each time the pulse bus is energized by a ground command.

REACTION CONTROL SYSTEM PARAMETERS

Five parameters from the Reaction Control System are monitored to indicate the condition of the system. Reaction Control System ring A/B (TC10) is a bi-level signal that indicates which thruster ring is energized. A 28 vdc signal to PCM programmer indicates ring A operation; 0 vdc indicates ring B operation. Thruster firing, TCA 1 through 8 (TC11 through TC18), are bi-level signals are are monitored to indicate proper operation of the Reaction Control System. Bi-level pulse signals of 0 or 28 vdc are derived from the solenoid command signals. A 0 vdc signal indicates thruster firing. Regulated nitrogen pressure (TB12) is a high-level signal obtained from a potentiometer-type transducer to indicate the pressure (0-500 psi) of the Reaction Control System regulated nitrogen.

Ring A and B nitrogen source pressure (TB16, TB17) parameters indicate the source pressure in each nitrogen supply. The pressure is measured by a potentiometer-type pressure transducer which is a part of the Reaction Control System.

RENDEZVOUS RADAR TRANSPONDER PARAMETERS

To monitor the operation and environmental condition of the rendezvous radar transponder, four parameters are relayed to the ground station. A temperature transducer installed in the transponder supplies a high-level signal output for transponder case temperature (TB14).

Boom (dipole) antenna extended (TC01) is monitored for operational reliability. An output of 28 vdc from the boom actuator indicates the antenna is extended. An output of 0 vdc indicates the antenna is not fully extended. Antenna selection (TB15) indicates spiral or dipole antenna usage. Selection of the antenna is governed by the strength of the interrogating signal. An output of 1.0 to 5.0 vdc indicates dipole antenna usage and 0 to 0.5 vdc indicates spiral antenna usage.

Transponder rf power (TB13) is a high-level signal that indicates when the high voltage power supply in the radar transponder is energized. As long as the transponder receives interrogation signals the high-voltage power supply is energized. Loss of the interrogating signals deenergizes the power supply. An output of 2 to 5 vdc indicates an on condition, and an output of 0 to 0.5 vdc indicates an off condition.

INSTRUMENTATION SYSTEM PARAMETERS

To insure proper scaling is being employed by the Instrumentation System, four parameters are monitored. These reference signals, low-level full-scale (TA07), low-level zero (TA08), high-level full-scale (TB02), and high-level zero (TB01) are outputs of the dc-dc converter that have been conditioned by the signal conditioner. The data received from these parameters provide the ground station verification that the Instrumentation System is functioning properly.

STRUCTURAL PARAMETERS

Equipment bay temperature (TA01) is the only structural parameter monitored and indicates the temperature that exists within the equipment bay. The temperature is measured by a resistive-element temperature sensor which supplies a low-level signal to the PCM programmer.

SYSTEM UNITS

TEMPERATURE SENSORS

The two temperature sensors are resistive-element, surface-mounted, sensors that convert the temperature into directly proportional electrical signals. The sensing elements are fully-annealed pure-platinum wire encased in ceramic insulation in a strain-free manner to provide maximum stability.

The physical construction of the sensors and a typical schematic is illustrated in Figure 15-19. The sensors are approximately 0.4 inches by 0.75 inches by 2.0 inches and weighs a maximum of 0.075 pounds. They have a temperature range of 0 to 400 degrees Fahrenheit and provide a 0 to 20 millivolt dc output.

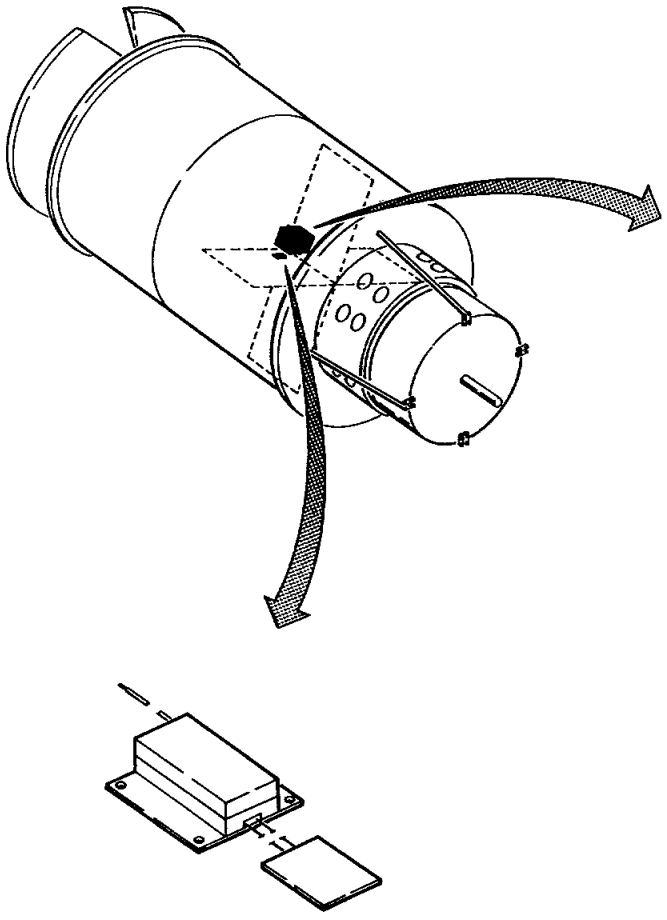
One of the temperature sensors is mounted on the battery case and the other in the equipment section at the intersection of the equipment support assemblies.

SIGNAL CONDITIONER

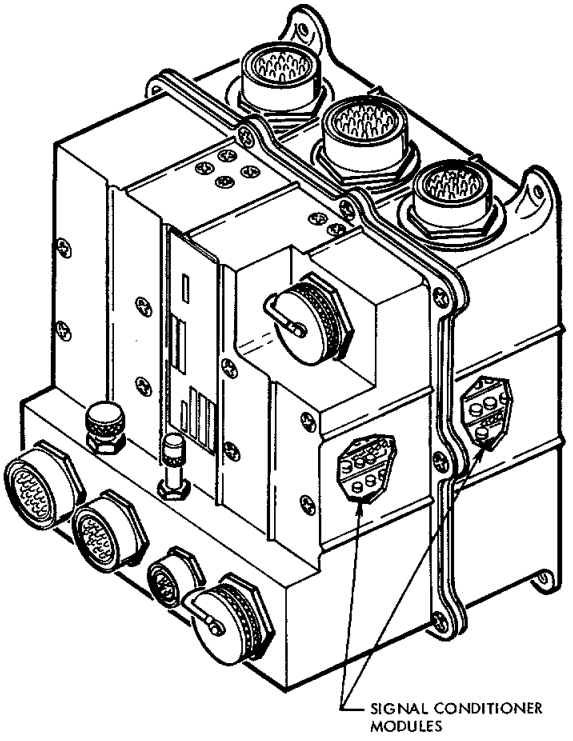
The signal conditioner is approximately 10 inches by 10 inches by 8 inches and weighs about eight pounds. The signal conditioner is located in the equipment section of the ATDA. Figure 15-19 illustrates the physical construction of the signal conditioner.

The signal conditioner contains 12 signal conditioning modules: one ac voltage monitor; four dc voltage monitors; one dc millivolt monitor; three attenuators; and three phase sensitive demodulators. The modules are constructed on circuit boards using printed wiring techniques.

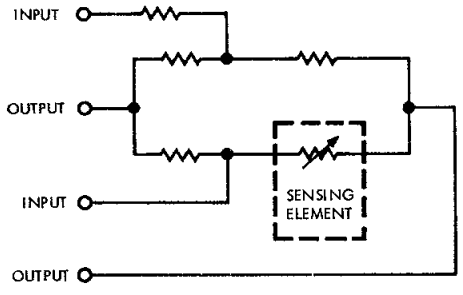
The ac voltage monitor accepts a signal ranging from 23 to 29 volts rms over a



TEMPERATURE SENSOR AND BRIDGE



SIGNAL CONDITIONER



TYPICAL SCHEMATIC TEMPERATURE SENSOR

Figure 15-19 Signal Conditioner & Temperature Sensor

frequency range of 380 to 420 cycles. The output is from 0 to 5 vdc, varying only with the input voltage.

The four types of dc voltage monitors are designed to accept various positive and negative dc voltage inputs and provide outputs of 0 to 5 vdc.

The dc millivolt monitor accepts an input of 0 to 50 millivolts and provides a proportional output of 0 to 20 millivolts.

There are three types of attenuator modules. These modules have various dc inputs which are attenuated to the 0 to 20 millivolt range or the 0 to 5 vdc range. Some attenuator modules contain two data channels.

There are three variations of the phase sensitive demodulators. Basically, the phase sensitive demodulator accepts two input voltages: one signal voltage and one reference. It provides a dc output of five volts for a full scale input signal that is in phase with the reference and an output of 0 volts for a full scale signal that is 180 degrees out of phase with the reference.

DC-DC CONVERTERS

The two dc-dc converters supply the Instrumentation System with regulated dc voltages of +5 and $\pm$ 24 vdc. The converters are essentially voltage regulators which operate on 18 to 30.5 vdc and supply regulated outputs.

The dc-dc converters (Figure 15-20) are approximately 5.5 inches by 5.5 inches by 7 inches, and weigh about seven pounds each. The converters are located in the ATDA equipment section.

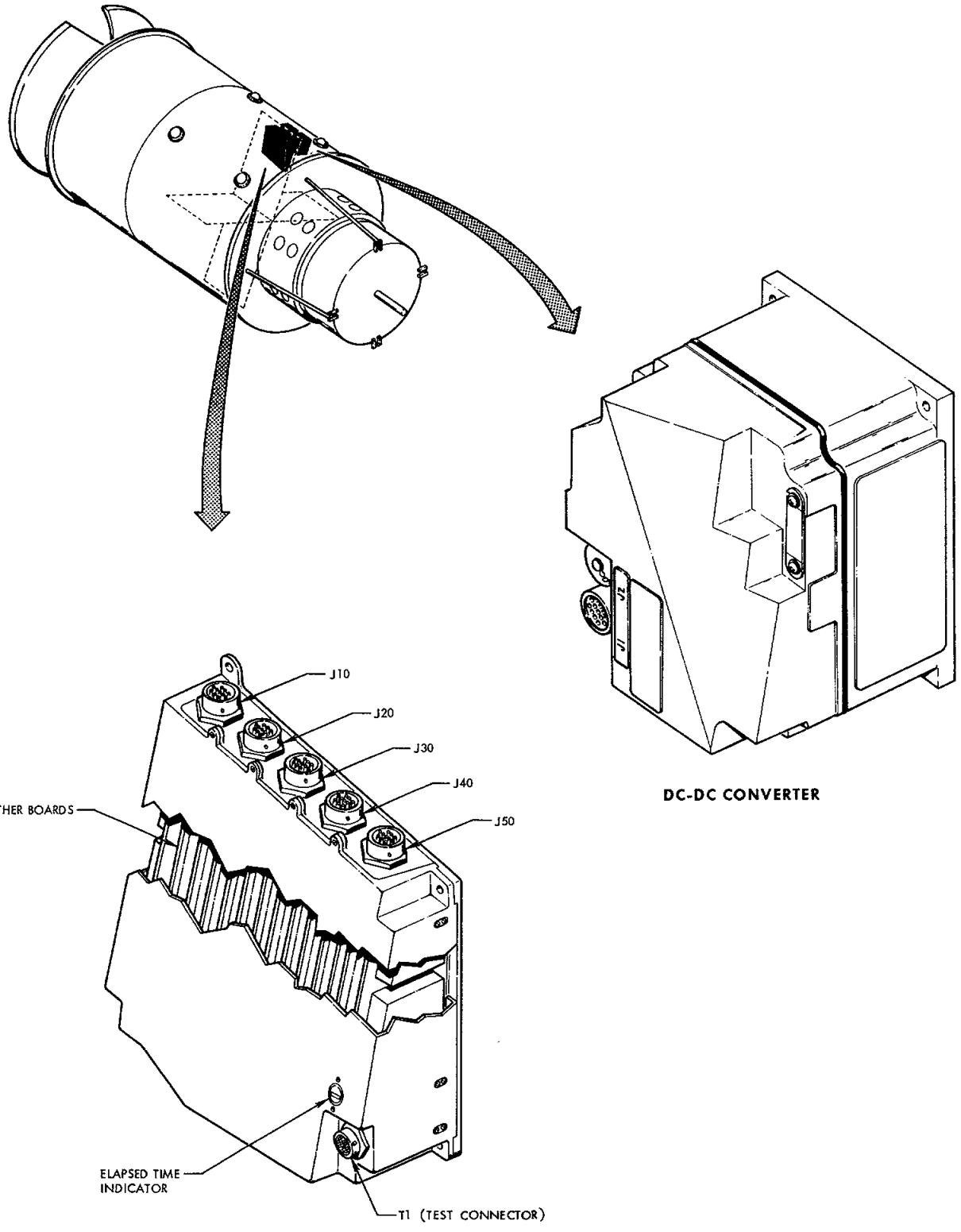


Figure 15-20 DC-DC Converter and PCM Programmer

PCM PROGRAMMER

The PCM programmer is located in the equipment section of the ATDA and encodes all instrumentation data for transmission. The programmer weighs approximately 20 pounds and is 11 inches by 11 inches by 4.5 inches. (Figure 15-20).

The programmer is composed of 16 subassemblies. Thirteen of the subassemblies are multilayer printed-circuit boards (mother-boards) constructed of a glass-epoxy bass material laminated at high temperatures. Most of the programmer subassemblies are embedded in polyurethane foam to provide rigidity and damping. An elapsed time meter indicates total operating time.

TELEMETRY TRANSMITTERS

Two telemetry transmitters transmit the Instrumentation System data to the ground stations. Although the transmitters serve the Instrumentation System, the antennas and associated switching are part of the Communication System; therefore, the transmitters are described in detail in Section XV, Communication System.

PYROTECHNIC DEVICES AND SEPARATION ASSEMBLIES

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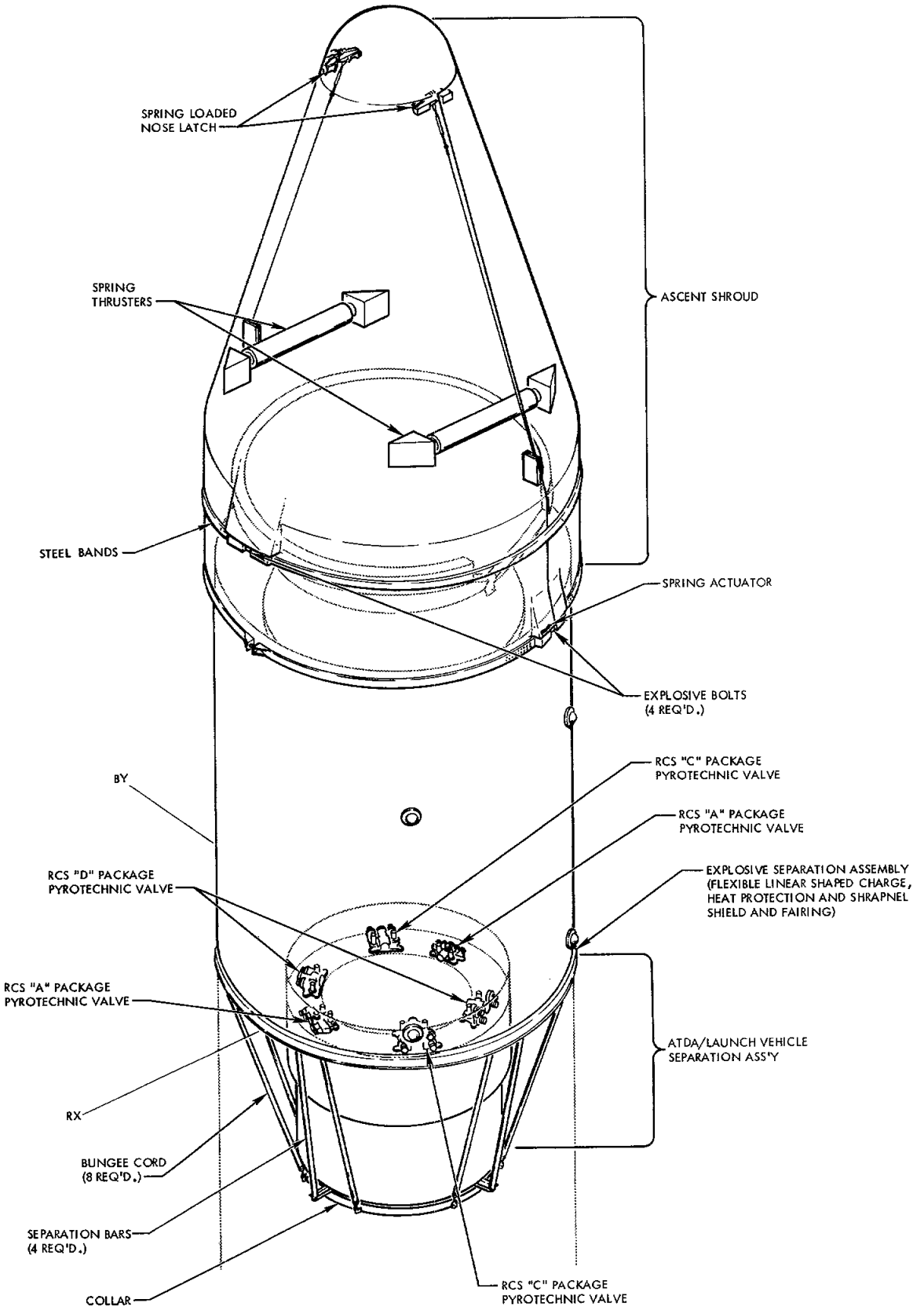


Figure 15-21 Pyrotechnic Devices and Separation Assemblies

PYROTECHNIC DEVICES AND SEPARATION ASSEMBLIES

INTRODUCTION

The pyrotechnic devices and separation assemblies (Figure 15-21), installed in the Augmented Target Docking Adapter (ATDA), are used to execute events initiated by the Sequential System. The events executed by these devices, in sequential order, are: release the shroud retaining bands; jettison the ascent shroud, sever the ATDA/launch vehicle mating ring, separate the ATDA from the launch vehicle and activate the Reaction Control System (RCS).

The pyrotechnic devices and separation assemblies which perform these functions and the events executed by them are as follows:

Explosive bolts - release shroud retaining bands.

Shroud separation assembly - separate and jettison the shroud.

Flexible Linear Shaped Charge - sever the ATDA/launch vehicle mating ring.

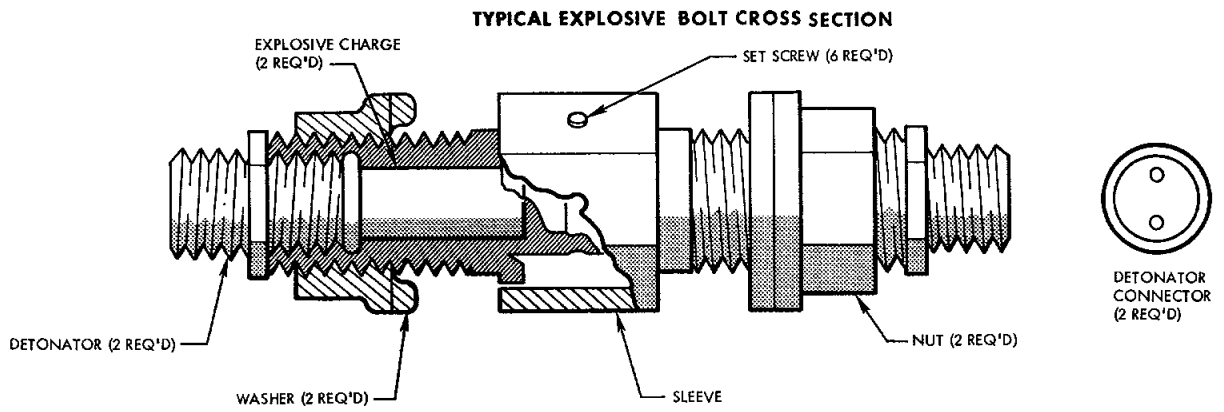
ATDA/launch vehicle separation assembly - provide a separation velocity between the launch vehicle and the ATDA.

Pyrotechnic valves - open lines and activate the RCS.

EXPLOSIVE BOLTS

Explosive bolts (Figure 15-22) are located at four points on the ATDA. Two bolts secure the steel retaining bands around the shroud. The other two bolts secure the base of the shroud, at the separation line. When initiated by the Sequential System, the bolts explode, releasing their respective clamps and allowing the separation assembly to jettison the shroud.

The bolts are electrically connected to the Sequential System via two cables. The cables are located 90 degrees from the shroud separation line, near the base of each segment.



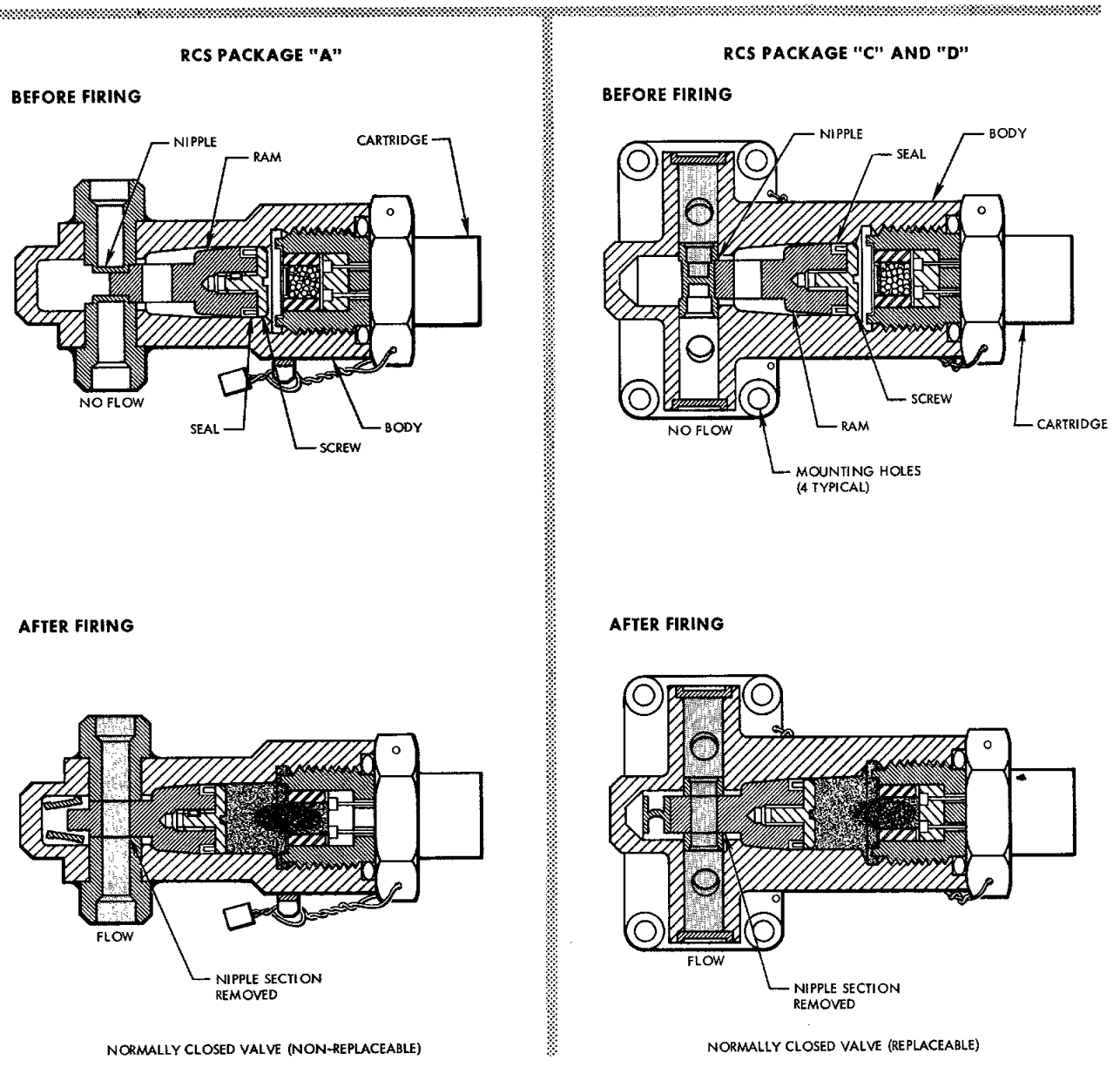


Figure 15-22 Pyrotechnic Valves and Explosive Bolts

DESCRIPTION

The four identical explosive bolts are five inches long, 5/8 of an inch in diameter and are made of stainless steel. A deep cylindrical cavity is machined into each end of the bolts. The cavities extend in approximately two inches from each end of the bolt. A solid part in the center is not drilled out. The end of each cavity is tapped to receive the threaded electrical detonator. The bottom of each cavity is so designed that the explosive blast will cause the bolt to break or fail at the central solid part as though sheared off there.

The detonators used are of the bridge wire type. Redundancy is maintained by using two detonators per bolt. A sleeve is slipped over the outside of the bolt and held in a center position with set screws. The sleeve maintains the parting planes of each bolt outside the clamps which it holds in place, and prevents any bolt fragmentation from piercing the shroud.

OPERATION

When the shroud jettison squib fire relays connect squib bus power to the eight shroud explosive bolt detonators, a current sufficient to set off the ignition mix, flows in the bridge wire. The energy from the ignition mix detonates the Penta Erythritol Tetra Nitrate (PETN) explosive charge next to it. The stress which the explosion puts upon the bolt structure causes the hollow parts on each side to break away from the central solid part.

PYROTECHNIC VALVES

Six pyrotechnic valves (Figure 15-22) are installed in the RCS. Three valves are used in each ring, one valve for each package. Each ring has a fuel package (called the "D" package), an oxidizer or "C" package, and a pressurant or "A" package. The valves isolate the fuel, oxidizer and pressurant from the rest of

the RCS by keeping them in their respective supply tanks. When the valves of either A or B ring are actuated, pressurant is released into the fuel and oxidizer tanks, and fuel and oxidizer are forced into lines to the eight activated thrusters. The pyrotechnic valves are one-time actuating devices which control the flow of these fluids.

Two types of valves are installed in the RCS. Both are normally closed. One type however is replaceable, the other is not. The RCS package "A" uses the nonreplaceable type. The RCS packages "C" and "D" use the replaceable type. If a nonreplaceable valve is defective, or if the squib has been fired, the package must be changed. Replaceable valves can be changed individually without changing the whole package.

DESCRIPTION

A pyrotechnic valve consists of a cartridge, a valve body, one or two nipples, a ram, a seal and a screw. The replaceable valve uses one closed-end nipple which is installed within the valve body and welded into place. The nipple is in the line from the supply. As long as the nipple is intact, it blocks the fluid from the supply. The nonreplaceable valve has two closed-end nipples butted together; one is the inlet, the other is the outlet port. The ram is in a slot or channel in the valve body, indexed directly above the centers of the nipples. The seal and screw are in the head of the ram. The cartridge is in the valve body at the top of the ram head. The function of the seal is to prevent blow-by of the actuation gas. The function of the cartridge is to generate the energy to drive the ram. The ram has two functions. One is to shear off the nipples and open the lines. The other is to complete the line by aligning the hole in its body with both sides of the line.

OPERATION

At Vernier Engine Cut-Off (VECO) plus 12 seconds, the squib buses are connected to six squibs on the three pyrotechnic valves of RCS ring B. Each valve operates in an identical manner. The squib sets off the cartridge which generates sufficient gas pressure to drive the ram. The ram is propelled along the channel, and the ram head shears off the nipples in its path. As the ram comes to rest at the opposite end of the channel, a hole through its body is aligned with the lines once blocked by the nipples. Fluid flows freely from the supply tank through the lines.

This same process is repeated on RCS ring A the first time this ring is selected by a Digital Command System (DCS) command.

FLEXIBLE LINEAR SHAPED CHARGE

The Flexible Linear Shaped Charge (FLSC) is the pyrotechnic device which severs the ATDA/launch vehicle mating ring. The FLSC is shown in cross section in Figure 15-23. The FLSC is a V-shaped flexible lead sheathing containing a core of cyclonite (RDX) high explosive. The specific type, shape and thickness of the material to be severed by a FLSC dictates the amount of explosive to be used. In this explosive separation assembly, dual strands of FLSC are used.

DESCRIPTION

The FLSC assembly consists of three detonator blocks, three crossovers, three dual boosters, a molded backup retainer and a back blast shield. Single bridge wire detonators are used in two detonator blocks, a double bridge wire in the third. In each detonator block, the crossovers are next to the detonators, and the boosters are next to the crossovers. The crossovers apply the shock wave

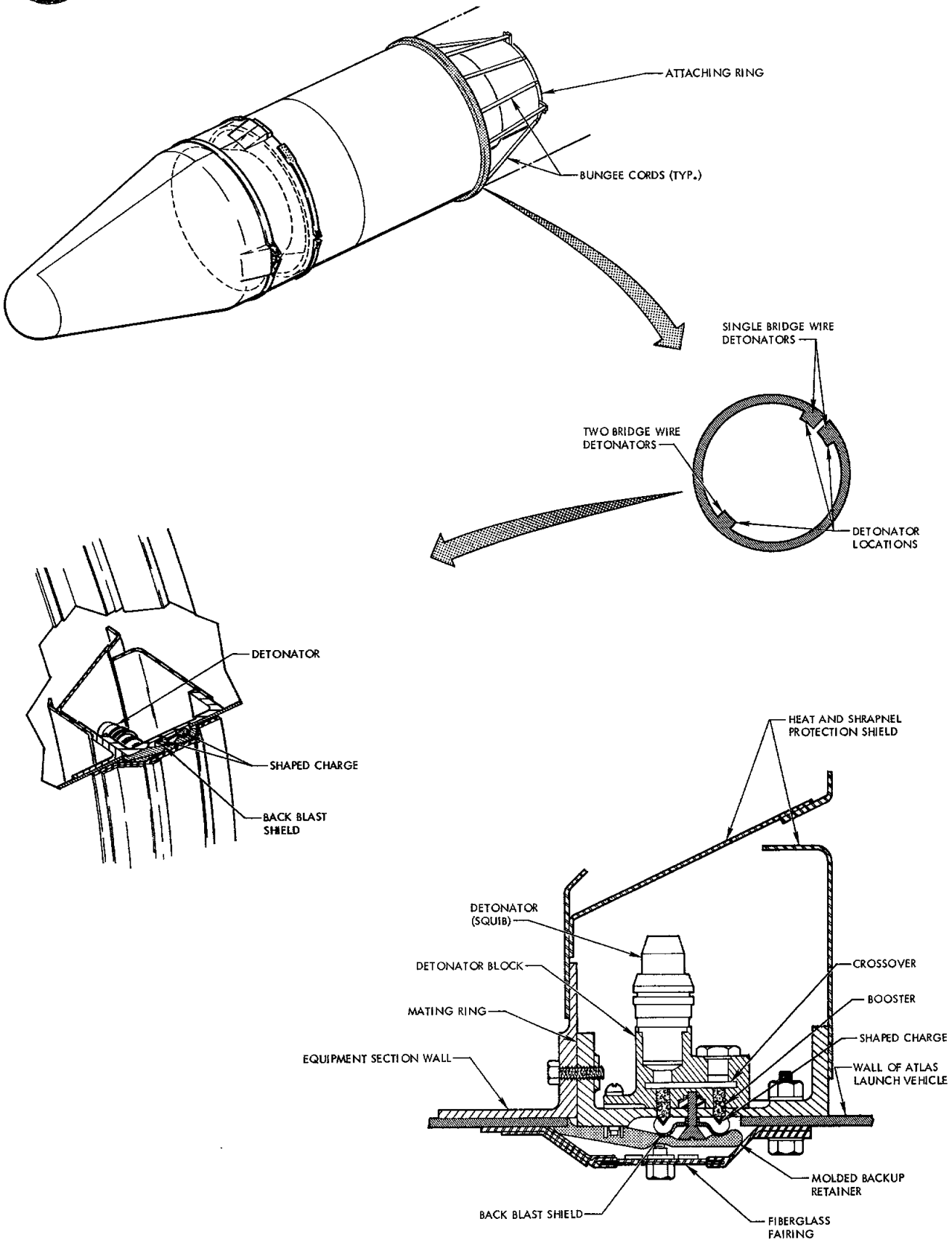


Figure 15-23 FLSC Mechanical Separation Assembly

to the booster. The booster propagates the shock wave to the FLSC to insure detonation of both strands.

When the explosive separation assembly is installed, the dual FLSC is attached to the periphery of the ATDA/launch vehicle mating ring. The open portion of the V-shaped FLSC is placed toward the surface to be cut.

The detonator blocks are installed on the inside of the mating ring and extend through the skin into the FLSC molded backup retainer. The detonators are wired to the ATDA shaped charge and RCS ring B squib fire relays. The back blast shield surrounds the molded backup retainer and is attached to it. A fiberglass fairing covers the retainer, shield and FLSC to protect the explosive separation assembly from aerodynamic heating.

OPERATION

At VECO plus 12 seconds, the Sequential System connects the squib buses to four ATDA shaped charge igniters, setting them off. The igniters detonate the crossover, the crossover detonates the boosters, and the boosters detonate the redundant strands of FLSC.

As the FLSC detonates, the concussion collapses the sheathing in each V-groove. This directed action produces a cutting jet of explosive products and minute metal particles. The jet in turn produces extremely high localized pressures. The resulting stress is far above the yield strength of the ring material. The skin is severed and the ATDA is detached, permitting the mechanical separation assembly to operate.

SHROUD SEPARATION ASSEMBLY

The shroud separation assembly is designed to jettison the ascent shroud at

the completion of the launch phase. The shroud is held together by steel bands until after VECO, then explosive bolts are fired, releasing the steel bands and allowing the separation assembly to function. The shroud is constructed in two segments, one long segment (including the nose cone) and one short segment. When jettisoned, the shroud separates and falls away to the rear, one segment on each side.

DESCRIPTION

The shroud separation assembly (Figure 15-24) consists of two preloaded spring actuators; two preloaded spring thrusters; two pivots; two spring loaded nose latches; two elastic latch cords with ball detents; two pair of split anchor blocks; and two pair of steel bands and clamps (held together by explosive bolts). The spring actuators are mounted at the base of the shroud at the separation line. The spring thrusters are mounted just below the shroud center of gravity. The nose latches hold the upper end of the shroud together before jettison, and unlatch to permit the segments to part at the nose. The elastic cords apply tension to the latch handles to keep the latches engaged. The ball detents on the cords are clamped in the anchor blocks to hold tension on the latches. The anchor blocks, which are compressed and trapped in the parting line by the tension of the steel bands, maintain tension on the latch cords. The base clamps, when pulled together by explosive bolts, compress the spring actuators and hold the shroud base in firm contact with the Target Docking Adapter (TDA). The entire assembly is held together by the steel bands, until the explosive bolts are fired by the Sequential System.

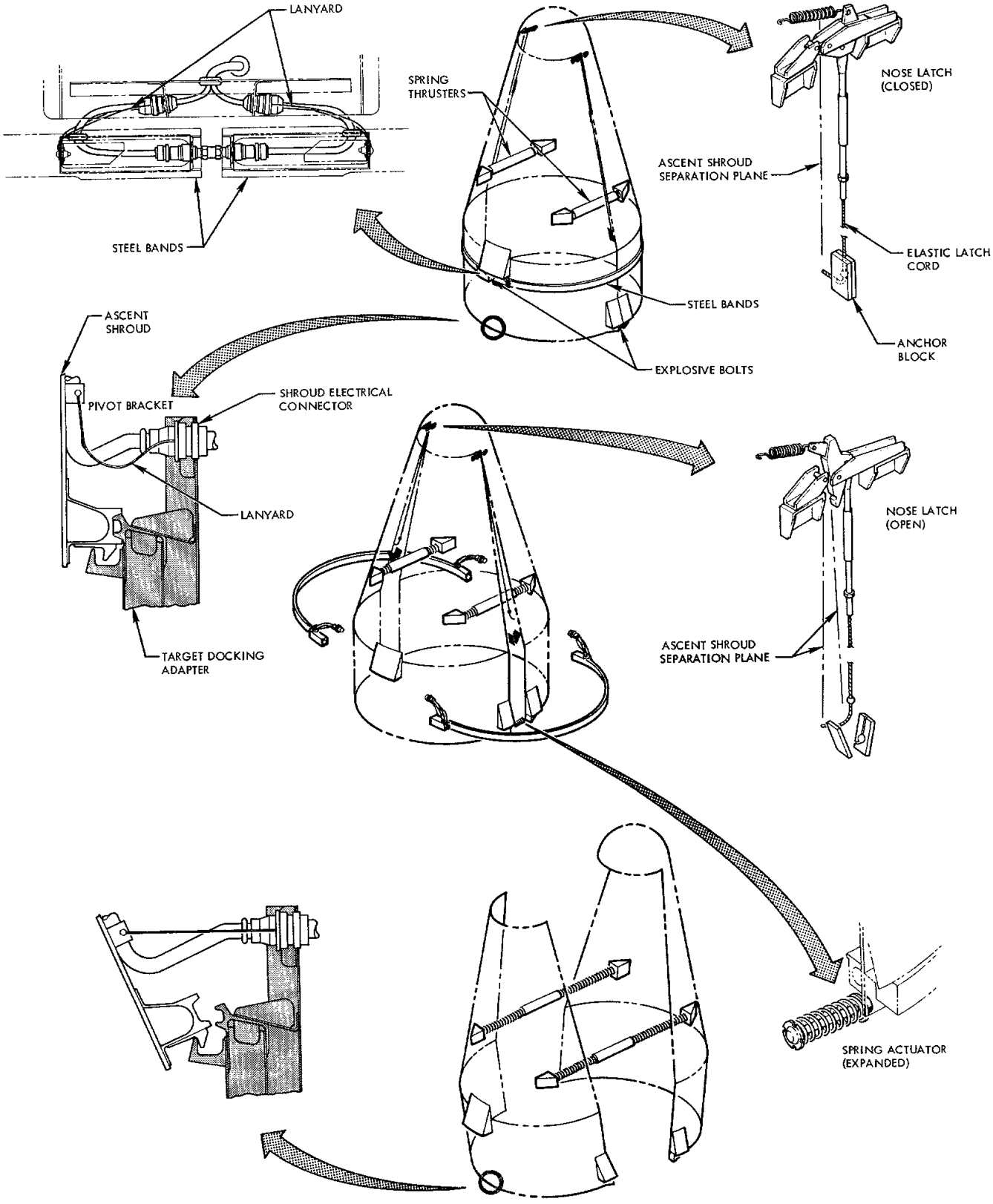


Figure 15-24 Shroud Separation Assembly

OPERATION

Operation of the shroud separation assembly (Figure 15-24) is initiated by the Sequential System signal which fires the explosive bolts. When the bolts fire, they release clamps in the steel retaining bands and in the base of the shroud. Four lanyards are used to disconnect the explosive bolt cables as the retaining bands spread. Spring actuators spread the shroud base apart, driving pivots against brackets on the TDA. When the base has spread far enough, the anchor blocks separate and release the elastic latch cords. When tension is removed from the latch cords, the spring loaded nose latches open and allow the shroud to separate at the nose. The spring loaded thrusters now drive the two segments apart, nose first. As the shroud segments separate, lanyards are pulled tight on two electrical connectors near the base of each segment. The lanyards disconnect the electrical connectors when the shroud segments are separated by approximately 50 degrees. The shroud segments continue to swivel on the pivots until separated by approximately 180 degrees. At this point the pivots allow the shroud to separate from the ATDA. The two shroud segments then fall behind and away from the ATDA.

ATDA/LAUNCH VEHICLE SEPARATION ASSEMBLY

The ATDA/launch vehicle separation assembly (Figure 15-23) is used to release the ATDA from the launch vehicle and provide it with a separation velocity.

DESCRIPTION

The separation assembly consists of the mating ring; a catapult, made up of eight bungee cords and attaching ring; and four separation rails. The mating ring structurally bonds the ATDA to the launch vehicle during ascent and provides a base from which the bungee cords are suspended. The mating ring also provides

a separation line where the two vehicles are severed pyrotechnically when orbit is achieved.

The catapult is elastically stressed to approximately 500 pounds and provides a separation velocity when the mating ring is severed. An attaching ring supplies a common surface for attaching the eight bungee cords which provide the thrust for the catapult. The attaching ring also provides a surface on which the battery module rests during launch. The bungee cord catapult is suspended from hangers on the underside of the mating ring.

Four separation rails span the RCS and battery module to guide the ATDA during the separation sequence. The rails protect the RCS and battery module from damaging contact with the mating ring or launch vehicle during separation.

OPERATION

At VECO plus 12 seconds, the Sequential System initiates the separation sequence. The pyrotechnics sever the mating ring and allow the separation assembly to catapult the ATDA away from the launch vehicle. The ATDA, which weighs approximately 2000 pounds, separates from the launch vehicle at a velocity of approximately three fps (two mph).

TARGET STABILIZATION SYSTEM

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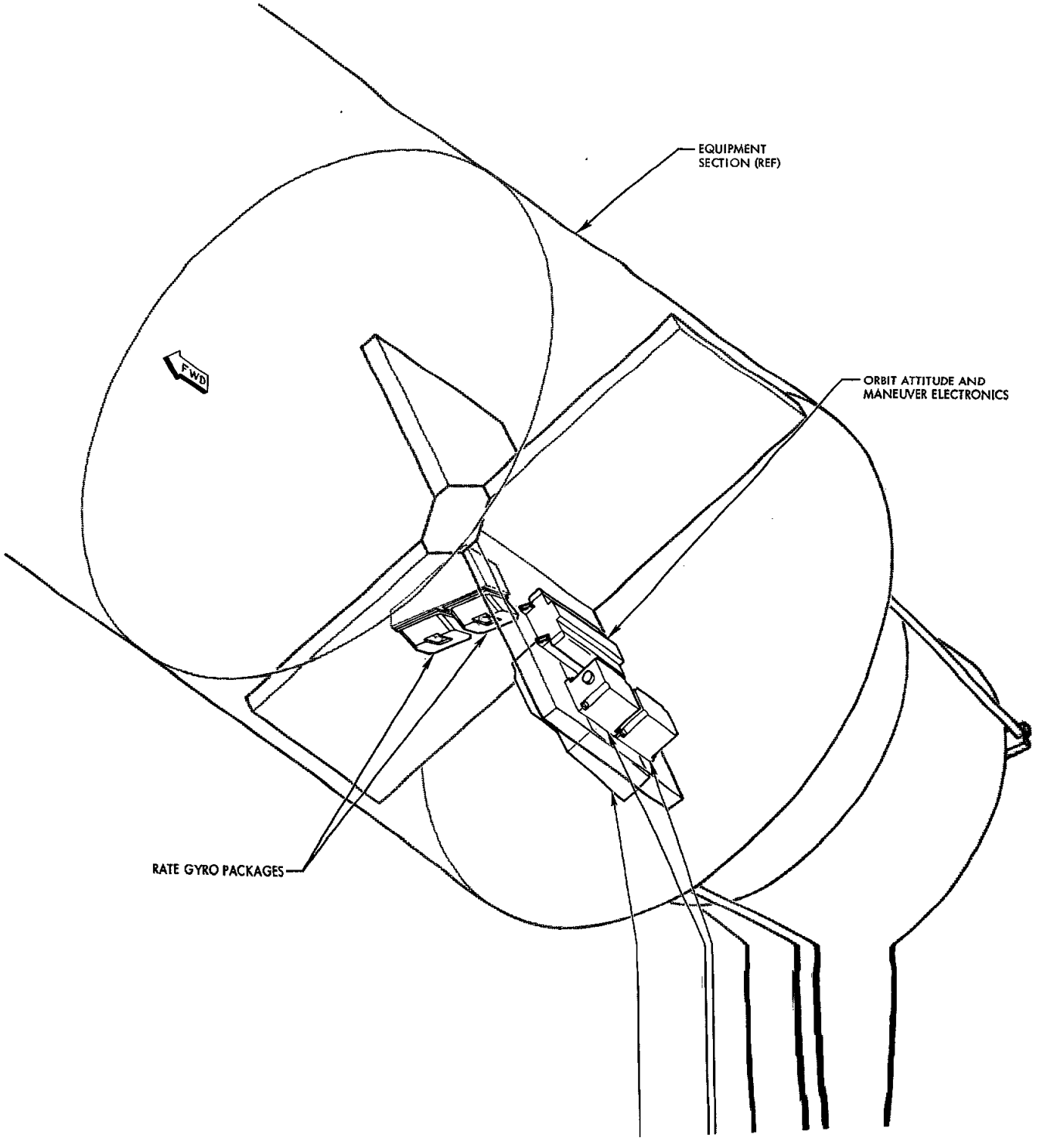
EQUIPMENT SECTION (REF)
FWD
ORBIT ATTITUDE AND MANEUVER ELECTRONICS
RATE GYRO PACKAGES

TARGET STABILIZATION SYSTEM

SYSTEM DESCRIPTION

The Target Stabilization System (TSS) is used in conjunction with the Reaction Control System (RCS) to control the rotational rates of the ATDA in the pitch, roll and yaw axes. It provides two modes of control which are selectable from a ground station via the Digital Command System (DCS) or from the spacecraft via the Command Link.

Prior to the docking phase of the mission, the TSS operates in the Biased Rate Damping (BRD) mode. In this mode, the TSS establishes nominal ATDA rotational rates of 1.8 degrees per second in the pitch axis, 2.2 degrees per second in the roll axis, and 0.25 degrees per second in the yaw axis. These rates are established to provide uniform solar heating, minimum radar ellipticity errors, and maximum DCS and Rendezvous Radar reception. The BRD mode is selected prior to launch and is initiated by the Digital Command System after the ATDA is inserted in its orbit.

For the docking phase of the mission, the TSS is placed in the Normal Rate Damping (NRD) mode. In this mode, the ATDA rotational rates are stabilized at less than 0.5 degrees per second in all three axes to faciliate spacecraft docking.

The TSS incorporates basically the same equipment as the Attitude Control and Maneuver Electroincs in the Gemini Spacecraft. It consists of an Attitude Control Electronics (ACE) package, an Orbit Attitude and Maneuver Electronics (OAME) package, primary and secondary Rate Gyro Packages (RGP), and primary and secondary

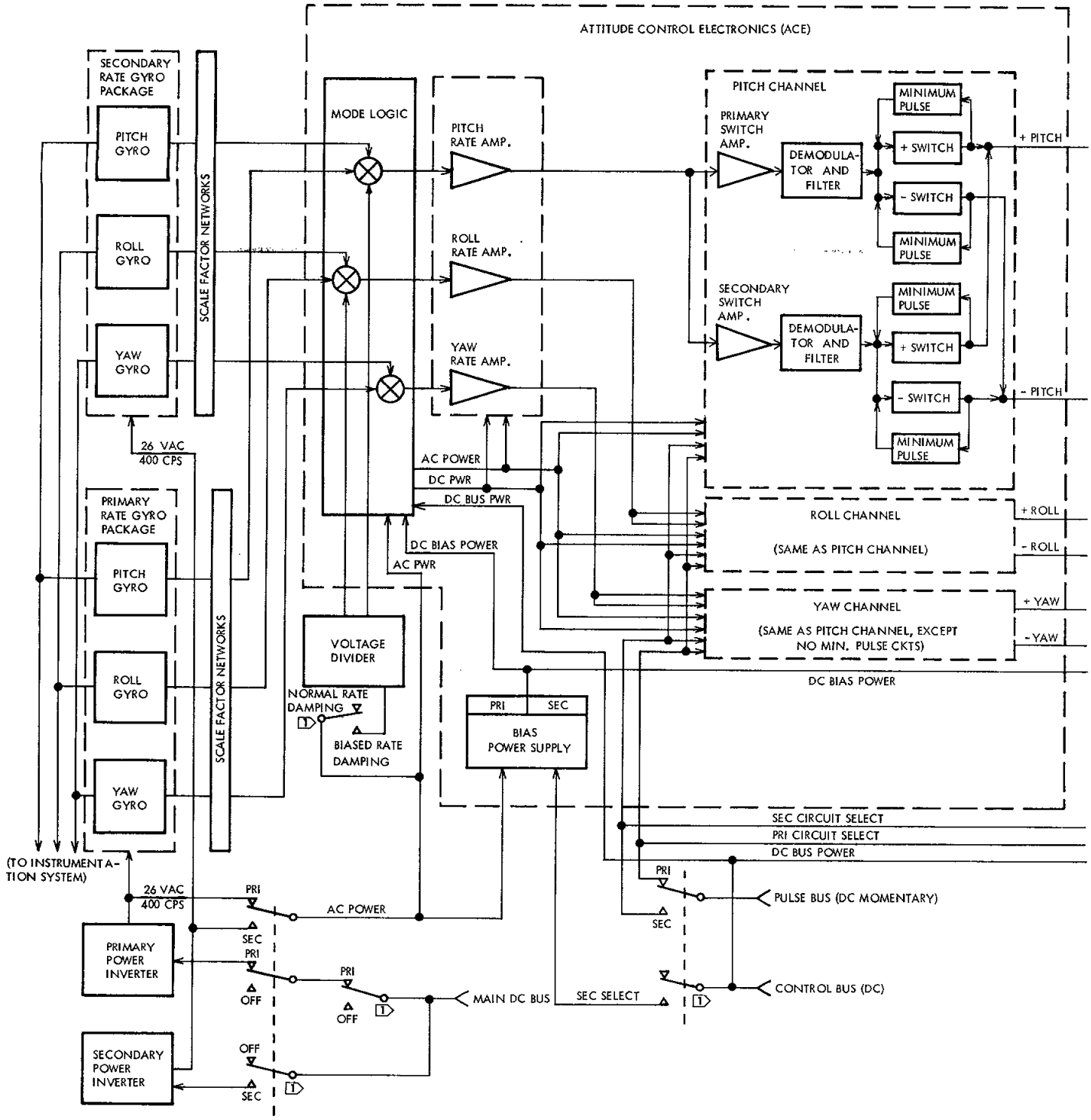

Figure 15-26 TSS Functional Block Diagram (Sheet 1 of 2)

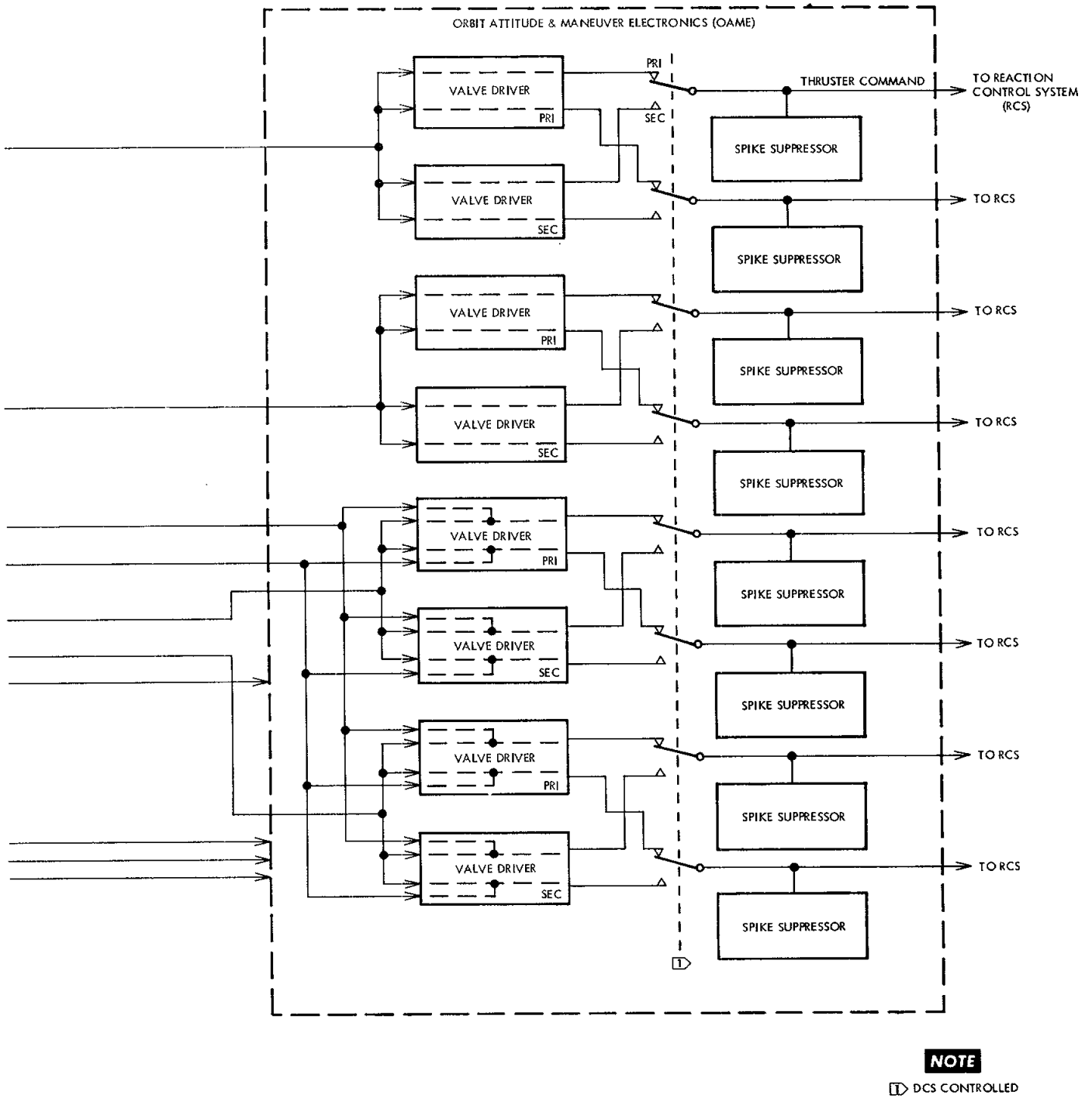

Figure 15-26 TSS Functional Block Diagram (Sheet 2 of 2)

Power Inverters (PI). All equipment is located in the equipment section of the ATDA (Figure 15-25).

SYSTEM OPERATION

The TSS controls the firing of the RCS thrusters to establish the rotational rates commanded by a selected control mode. The existing rates in the three ATDA axes are sensed by three rate gyros in the operating RGP. Signals representing these rates are supplied to the ACE package where they are compared with signals representing the commanded rates for each axis (Figure 15-26). When a difference exists between an actual rate and a commanded one, the appropriate rotational command (i.e., pitch up, pitch down, etc.) is developed to effect a rate change. This command is routed from the ACE to the OAME package where it is used to develop individual thruster firing commands. The firing commands, in turn, are transmitted to the RCS to fire the thrusters. The thrusters continue to fire until the rate gyro and rate command signals are of equal magnitude. At this time, the thruster firing commands are terminated.

RATE COMMANDS

The rate commands, in the BRD control mode, are generated by a voltage divider within the TSS. The voltage divider generates commands only for the pitch and roll axes since the yaw axis is rate damped in both control modes.

When the BRD mode is selected, the rate control relays in the system control relay panel are latched. This causes a 26 vac, 400 cps supply voltage to be applied to the voltage divider. The outputs of the voltage divider are used as commands to establish the pitch and roll rotational rates for the BRD mode. These commands are always in phase with the basic 400 cps supply voltage.

Selection of the NRD control mode causes the rate control relays in the system control relay panel to reset. Thus, the excitation voltage is removed from the voltage divider, and its outputs drop to zero. This results in the ATDA being rate damped about all three axes.

SECONDARY OPERATION

The TSS has provision for either primary or secondary system operation. Selection can be made from the ground, via the DCS. For each type of operation, different RGP and power inverters are utilized. In addition, certain selections of the ACE and OAME packages are redundant to provide for primary and secondary operation.

ELECTRICAL POWER

The basic operating power for the TSS is 28 vdc from the ATDA main bus. This power is used by the power inverters to produce the 26 vac, 400 cps power for the system components.

SYSTEM UNITS

RATE GYRO PACKAGE

The two rate gyro packages in the TSS are identical. Each is approximately 3.5 inches by 4 inches by 5.6 inches and weighs about 4 pounds. Each unit contains three individual rate gyros which sense ATDA rotational rates in the pitch, roll and yaw axes. The gyros are hermetically-sealed, single-degree-of-freedom sensors. They receive 26 vac, 400 cps power from the power inverter and provide outputs to the ACE package and the Instrumentation System. Each one also has a torquing capability for use during ground checkout. The output of each gyro is a 400 cps signal whose amplitude is proportional to the angular

rate about the sensitive axis. The signal can be either in phase or out of phase with the supply voltage, depending on the direction of rotation. Upon leaving the RGP, the rate signal is routed through a scale factor network which attenuates it to the proper level for use by the ACE.

ATTITUDE CONTROL ELECTRONICS

The ACE package is approximately 16.3 inches by 8.4 inches by 5.1 inches and weighs about 17 pounds. It contains mode logic circuits, rate amplifiers (preamps), switching amplifiers, demodulators, positive and negative switches, minimum pulse circuits, and a bias power supply. All sections, except the mode logic and rate amplifiers, are redundant to provide for primary or secondary operation.

The ACE package generates the rotational commands to effect necessary changes in ATDA rotational rates. The logic for producing specific commands is developed by summing the rate gyro and rate command input signals together. This is accomplished by routing the two signals to a common summing point. If the signals are of equal magnitude and are directly out of phase, it is indicative that the actual rate is the same as the commanded one. In this instance, the signals null each other out, and a zero-level resultant signal is obtained. In cases where the rate gyro signal has a different magnitude or phasing with respect to the rate command, either an additive or difference signal will result. If the resultant signal is greater than a certain level (deadband threshold), it causes a specific rotational command to be developed. A resultant signal which is in phase with the 400 cps supply voltage will cause a negative rotational command (pitch down, roll left or yaw left) to be produced. An out-of-phase resultant signal will cause a positive command (pitch up, roll right or yaw right) to be produced.

Mode Logic

The mode logic section performs several functions within the ACE. First, it establishes and maintains the conditions for effecting the rate command operational mode in the TSS. This basically consists of setting up the proper servo loop gain and turning on the transistor switches which route the rate signals from the rate preamplifiers to the switching amplifiers. In addition, the mode logic controls the operation of the primary and secondary circuits within the ACE (except for the bias power supply). This is accomplished by supplying operating voltages only to the selected circuits. Primary and secondary operation of the bias power supply is controlled directly by a relay in the system control relay panel.

Amplifier Sections

Each of the three signals resulting from the summing process is routed to a rate preamplifier. The rate preamplifier amplifies and reverses the phase of the input signal and supplies it to either the primary or secondary switching amplifier. The switching amplifier further amplifies the signal to a usable level for the demodulator.

Demodulators

The demodulators discriminate between in-phase and out-of-phase ac signals. If the input signal is in phase, the demodulator transmits a positive dc signal to the positive switch. If the input signal is out of phase, the demodulator transmits a negative dc signal to the negative switch.

Positive and Negative Switches

The positive switches, when turned on by the demodulator output, provide a path to ground through transistors in the positive switch circuits. This connection to ground acts as a logic command to the OAME where it causes the positive-torquing thrusters to fire. In the same manner, the negative switches control the firing of the negative-torquing thrusters.

Minimum Pulse Circuits

The outputs of the positive and negative switches for the pitch and roll channels are also used to trigger minimum pulse circuits. Each of these circuits generates a pulse which is fed back to the input of its respective switch circuit. The pulse causes the switch circuit to be clamped on for the duration of the pulse (nominally, 18 milliseconds). Thus, when a switch circuit is turned on, it is held on for at least this duration.

ORBIT ATTITUDE AND MANEUVER ELECTRONICS

The OAME package is approximately 8 inches by 10.2 inches by 3 inches and weighs about 8 pounds. It has a removable cover and contains three replaceable module boards (two relay boards and one component board) as well as fixed components. Electrically, the OAME consists of primary and secondary valve driver circuits and a spike suppression circuit for each of the eight thruster solenoid valves.

The OAME converts the logic commands from the ACE into individual firing commands for the RCS thrusters. When turned on, the valve driver provides a circuit ground to actuate a thruster solenoid valve in the RCS. A back-to-back diode/ zener diode spike suppressor for each thruster channel is provided in the OAME

to minimize voltage transients when a solenoid valve is actuated or de-actuated.

POWER INVERTER

The power inverters convert main bus dc power to 26 vac, 400 cps, single-phase power for use by the ACE, RGP and system control relay panel. The two units are designated primary and secondary and operate as such in the system. Each package is approximately 5 inches by 5.3 inches by 4 inches and weighs about 7 pounds.

REACTION CONTROL SYSTEM

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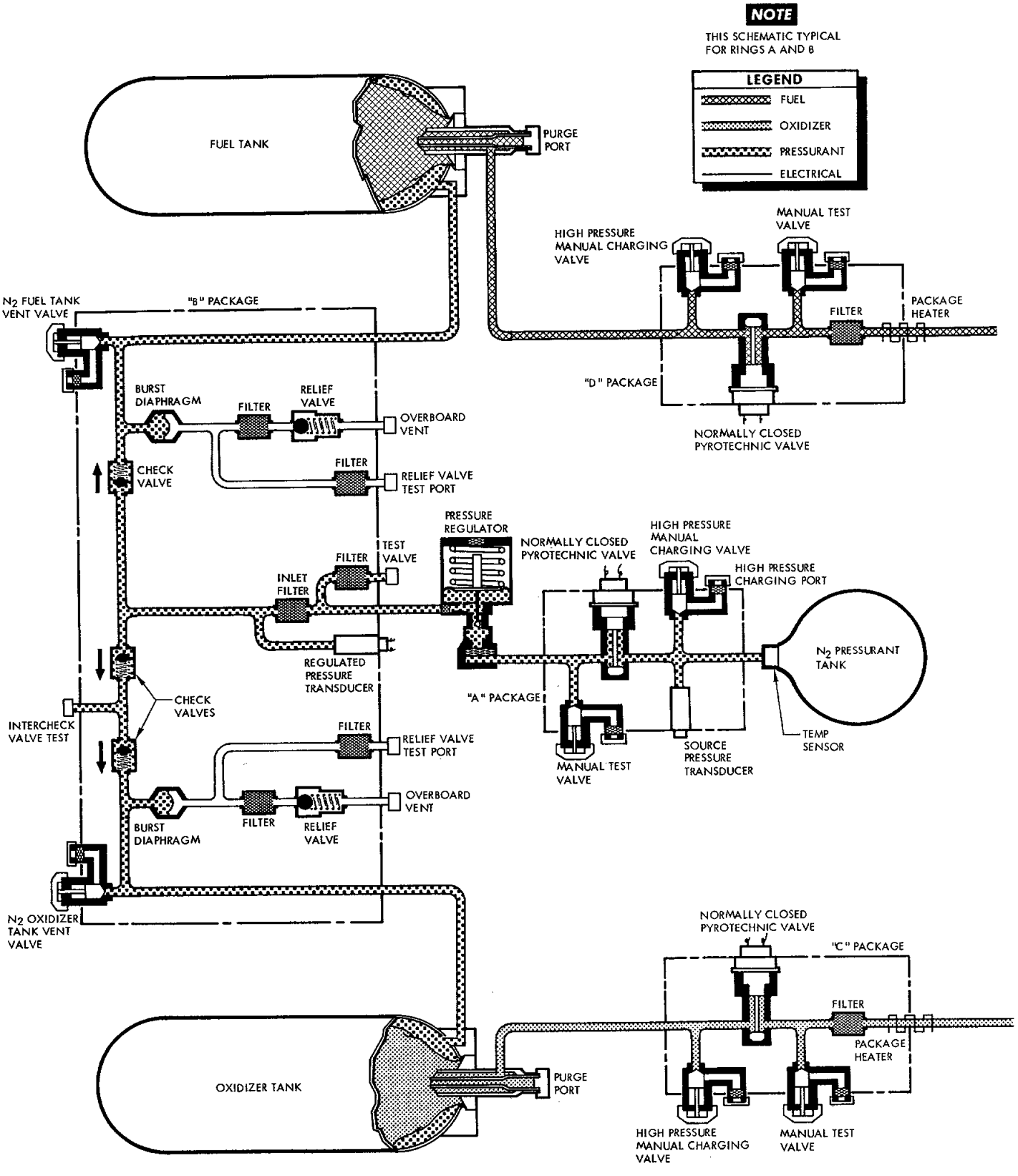

Figure 15-28 Reaction Control System Schematic (Sheet 1 of 2)

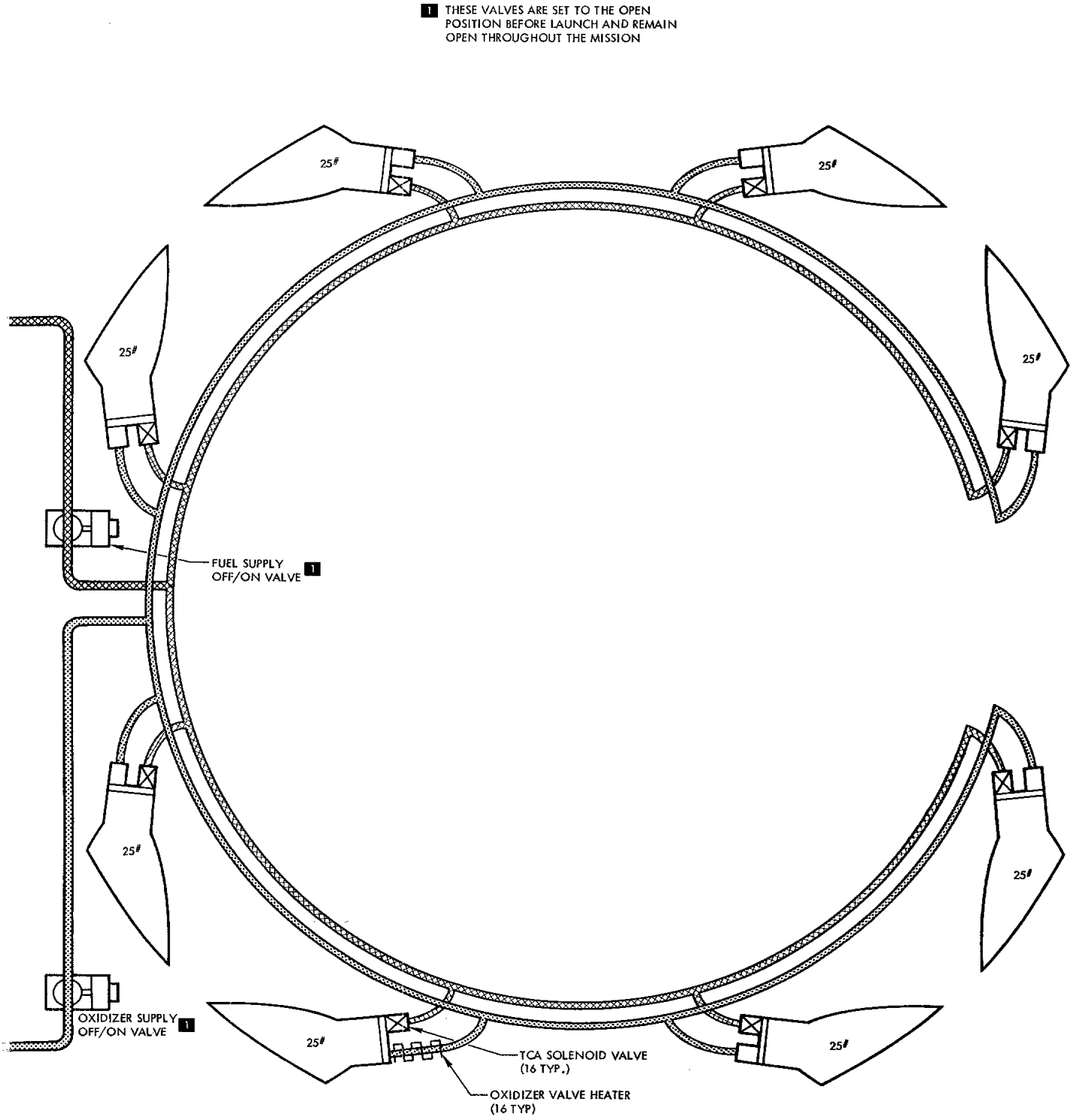

Figure 15-28 Reaction Control System Schematic (Sheet 2 of 2)

signal from the Sequential System. Ring B is thus activated and it begins immediately to execute TSS commands. Ring A is not activated until it is selected. For explanation purposes, the RCS is divided into three groups: the Pressurant, Fuel/Oxidizer (or propellant) and Thrust Chamber Assembly (TCA).

PRESSURANT GROUP

The pressurant group (Figure 15-28) consists of a pressurant tank, "A" package, pressure regulator and "B" package. Valves and test ports are provided at accessible points to permit servicing, venting, purging and testing. Filters are provided throughout the system to prevent contamination. The pressurant is stored and isolated from the remainder of the system by a normally-closed pyrotechnic actuated valve, located in the "A" package.

FUEL/OXIDIZER GROUP

The fuel/oxidizer (propellant) group (Figure 15-28) consists of expulsion bladder storage tanks, "C" (oxidizer) and "D" (fuel) packages. Valves, ports and test ports are provided at accessible areas to permit servicing, venting, purging and testing. Filters are provided throughout the system to prevent contamination. The propellants are isolated in the storage tanks from the remainder of the system by normally-closed pyrotechnic actuated valves in the "C" and "D" packages. Heaters are installed on the packages to maintain an operating temperature. The propellants used are:

Oxidizer - Nitrogen Tetroxide ($N_2 9 4$) conforming to

Specification MIL - P - 26539A

FUEL - Monomethyl Hydrazine ($N_2H_3CH_3$) confroming to Specification MIL - P - 27403

THRUST CHAMBER ASSEMBLY GROUP

The TCA group (Figure 15-28) consists of eight 25-pound TCA's used for attitude (roll, pitch and yaw) control of the ATDA. Each TCA is equipped with two solenoid valves which open simultaneously. Heaters, installed on the oxidizer solenoid valves, maintain the oxidizer at an operating temperature.

SYSTEM OPERATION

RCS thruster operation is controlled by signals from the TSS. The RCS (Figure 15-28) is activated by a signal from the Sequential System. The RCS, in conjunction with the TSS, provides attitude control of the target vehicle in the roll, pitch and yaw axes.

Ring B of the RCS module is selected during pre-launch by a DCS command to provide attitude control and is activated when power is applied at ATDA/Launch Vehicle separation plus eight seconds. Selection of ring A, in the event of a malfunction in ring B, may be accomplished by a DCS command.

The operation of the RCS is described in three groups, Pressurant, Fuel/Oxidizer and Thrust Chamber Assembly.

PRESSURANT GROUP

(Figure 15-28) High pressure nitrogen (N_2) pressurant is stored at 3000 psi in the pressurant tank. The tank is serviced through the "A" package high pressure gas charging port. Pressure from the pressurant tank is isolated from the

remainder of the system by a normally-closed pyrotechnic actuated valve located in the "A" package. Stored nitrogen pressure is monitored by the source pressure transducer located in the "A" package and relayed to the ground station as instrumentation data. Upon command, the "A" package pyrotechnic actuated valve is opened and nitrogen flows to the pressure regulator and "B" package. The "B" package routes the pressure to both propellant tanks. The regulated pressure is sensed by the pressure transducer ("B" package) and supplies an output to the Instrumentation System indicating pressure downstream of the regulator. The check valves prevent backflow of propellant vapors into the pressurant system. The "B" package has a safety feature to prevent over pressure of the fuel and oxidizer tank bladders. Should the system be over pressurized downstream of the regulator, the pressure would first rupture the burst diaphragms, then be vented overboard through the relief valves.

FUEL/OXIDIZER GROUP

Fuel and oxidizer propellants are stored in their respective tanks, and are serviced through the high pressure charging ports in the "C" and "D" packages. The propellants are isolated from the remainder of the system by a normally closed pyrotechnic valves in the "C" and "D" packages. Upon command, the "C" (oxidizer) and "D" (fuel) package pyrotechnic actuated valves are opened and propellants are distributed through their separate tubing manifold system to the thrust chamber inlet solenoid valves.

Two motor operated valves are located in the propellant feed lines, upstream of the TCA's. These valves are opened prior to launch and will remain open throughout the mission. The valves are not connected electrically. Heaters are

connected to the output lines of the "C" and "D" packages to prevent freeze-up and are turned on prior to launch by the RCS HRTS switch.

THRUST CHAMBER ASSEMBLY GROUP

The TCA fuel and oxidizer solenoid valves are controlled by signals from the TSS. In response to the signals, the valves open and propellants are directed through small injector jets into the thrust chamber. The fuel and oxidizer impinge on one another, where they ignite hypergolically to burn and create thrust. Heaters are connected to each TCA oxidizer solenoid valve to prevent freeze-up and are turned on at launch by the RCS HRTS switch.

SYSTEM UNITS

PRESSURANT STORAGE TANK

The nitrogen (N_2) pressurant is stored in a welded, titanium spherical tank. The tank is 7.25 inches in diameter and has an internal volume of 185.0 cubic inches. Nitrogen gas is stored at 3000 psi and held therein by the "A" package pyrotechnic valve until released by a sequential signal. This nitrogen under pressure is used to expel the fuel and oxidizer from their respective tanks.

"A" PACKAGE

The "A" package consists of a pressure transducer, isolation valve, filters and two high pressure gas charging valves. The pressure transducer monitors the source pressure and supplies an output to the Instrumentation System, indicating the pressure of the stored gas. The normally closed isolation valve is used to isolate the pressure from the remainder of the system.

The valve is pyrotechnically opened to activate the system for operation. Two dual seal, high pressure gas charging valves and ports are provided, on on each side of the isolation valve. The upstream valve is used for servicing, venting and purging the pressurant tank, while the downstream valve is used to test downstream components. Filters are installed to prevent contaminants from entering the system.

PRESSURE REGULATOR

The pressure regulator is conventional, mechanical-pneumatic type. The regulator reduces the source pressure to system operating pressure. An inlet filter reduces any contamination in the gas to an acceptable level.

"B" PACKAGE

The "B" package consists of filters, pressure transducer, three check valves, two burst diaphragms, two relief valves, regulator output test port, fuel tank vent valve, oxidizer vent valve, check valve test port and two relief valve test ports. The inlet filter reduces contamination in the gas to an acceptable level. The relief valve filters prevent particals from a ruptured burst diaphragm from entering the relief valves. Test port filters prevent contaminants from entering the system through the test ports. The pressure transducer monitors the regulated pressure and supplies an output to the Instrumentation System. Check valves prevent backflow of fuel or oxidizer into the pressurant system. The burst diaphragms are safety devices that repture when the pressure reaches the design failure pressure, thus, prevents imposing excessive pressure on the propellant bladders.

The two relief valves are conventioan mechanical-pneumatic type with the pre-

set opening pressure. In the event of burst diaphragm rupture, the relief valve opens to vent excess pressure overboard. The valve reseats when a safe level is reached. This prevents venting the entire gas source. Manual valves and ports are provided to vent, purge and test the regulated system.

FUEL TANK

The fuel tank is a welded, titanium cylindrical tank which contains an internal bladder and purge port. The tank is 5.10 inches outside diameter, 30.7 inches in length and has a fluid volume capacity of 546.0 cubic inches. The nitrogen pressurant is imposed on the exterior of the bladder to expel fuel through the "D" package to the TCA solenoid valves. The purge port is proveded to purge and vent the fuel tank bladder.

OXIDIZER TANK

The oxidizer tank is a welded, titanium cylindrical tank which contains a bladder and purge port. The tank is 5.10 inches outside diameter, 25.2 inches in length and has a fluid volume capacity of 439.0 cubic inches. The bladder is a double layered Teflon, positive expulsion type. The nitrogen pressurant is imposed on the exterior of the bladder to expel the oxidizer through the "C" package to the TCA solenoid valve. The purge port is provided for purging and venting the oxidizer tank bladder.

"C" AND "D" PACKAGES

The "C" and "D" packages are identical in operation and are located downstream of the tanks of their respective system. Each package consists of filters, an isolation valve, propellant charging valve and test valve. The in line filter reduces

partical contamination to an acceptable level. The valve filters prevent contaminants from entering the system. The normally closed isolation valve is used to isolate propellants from the remainder of the system until ready for use. The isolation valve is opened pyrotechnically to activate the system. The propellant charging valve is located upstream of the isolation valve and is used for servicing and venting the system. A test valve is used to test the system, downstream of the isolation valve.

PROPELLANT SUPPLY SHUTOFF VALVES

Motor operated propellant supply shutoff valves are located downstream of the "C" and "D" packages. The valves are opened during pre-launch and are not connected electrically.

THRUST CHAMBER ASSEMBLY GROUP

Each TCA consists of two propellant valves, injection system, calibrated orifices, combustion chamber and expansion nozzle. The fuel and oxidizer solenoid valves are the normally-closed quick acting type, which open simultaneously upon command. The action permits fuel and oxidizer flow into the injector system. The injectors use precise jets to impinge fuel and oxidizer streams on one another for controlled mixing and combustion. The calibrated orifices are fixed devices used to control propellant flow. Hypergolic ignition occurs in the combustion chamber. The combustion chamber and expansion nozzle are lined with ablative materials and insulation to absorb and dissipate heat and control external wall temperature. TCA's are installed within the RCS module mold line, with the nozzles terminating flush with the outer mold line. TCA's are located at fixed points on the RCS module in a location suitable for attitude control.

Thermostatically controlled electric heaters, located at various points in the system, are used to prevent propellant from freezing.

EXTRA-VEHICULAR ACTIVITY

Section XVI

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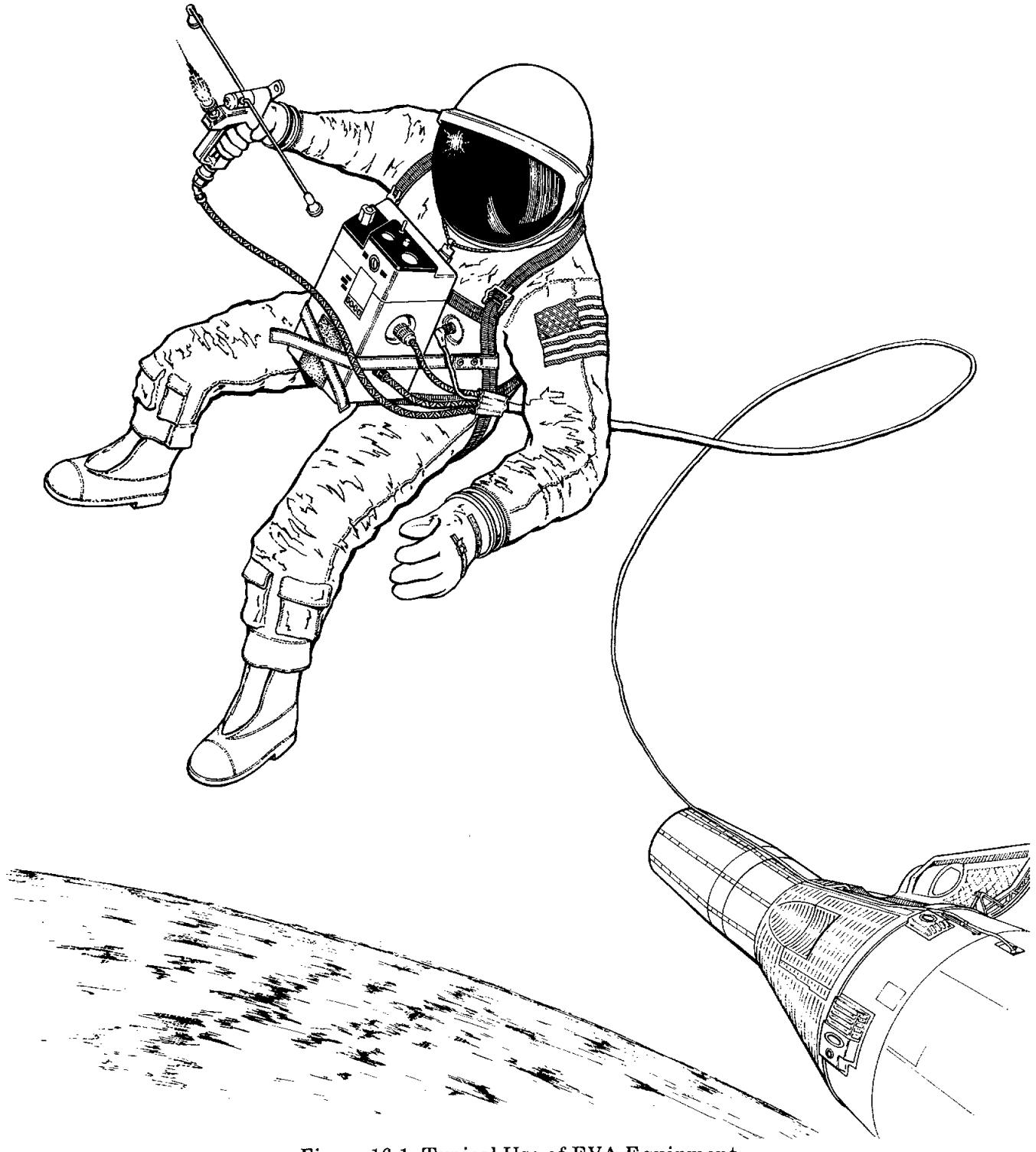


Figure 16-1 Typical Use of EVA Equipment

SECTION XVI EXTRA-VEHICULAR ACTIVITY

SYSTEM DESCRIPTION

Extra-Vehicular Activity (EVA) is that portion of the mission during which the pilot egresses from the cabin to conduct experiments and perform maneuvers. The purpose of EVA is to: determine man's performance capabilities wearing a pressurized suit in a free space environment, demonstrate the ability to perform controlled maneuvers in space independent of the spacecraft, and conduct experiments outside the spacecraft. EVA is planned for spacecraft 8 through 12 missions.

EVA is accomplished using the Extra-vehicular Life Support System in conjunction with: the Extra-vehicular Life Support Package on spacecraft 8; the Hand Held Maneuvering Unit on spacecraft 10 and 11; and the Modular Maneuvering Unit on spacecraft 9 and 12. The Modular Maneuvering Unit is an experiment; therefore it is not included in the EVA System description. No EVA was accomplished on spacecraft 8 which utilized the Extra-vehicular Life Support Package; therefore, information pertaining to that configuration is not included.

The Extra-vehicular Life Support System (ELSS) is the primary component of the EVA System on each spacecraft mission. The ELSS provides environmental control and supplies the necessary electrical and tether connections to support the pilot during the entire EVA. The ELSS consists basically of a chest pack, umbilical, and electrical jumper. The ELSS umbilical is modified for spacecraft 10 and 11 to include a nitrogen hose to supply propellant to the Hand Held Maneuvering Unit (HHMU). The HHMU is used to maneuver through space. It provides forward or reverse thrust as required. Typical use of EVA equipment is illustrated in Figure 16-1.

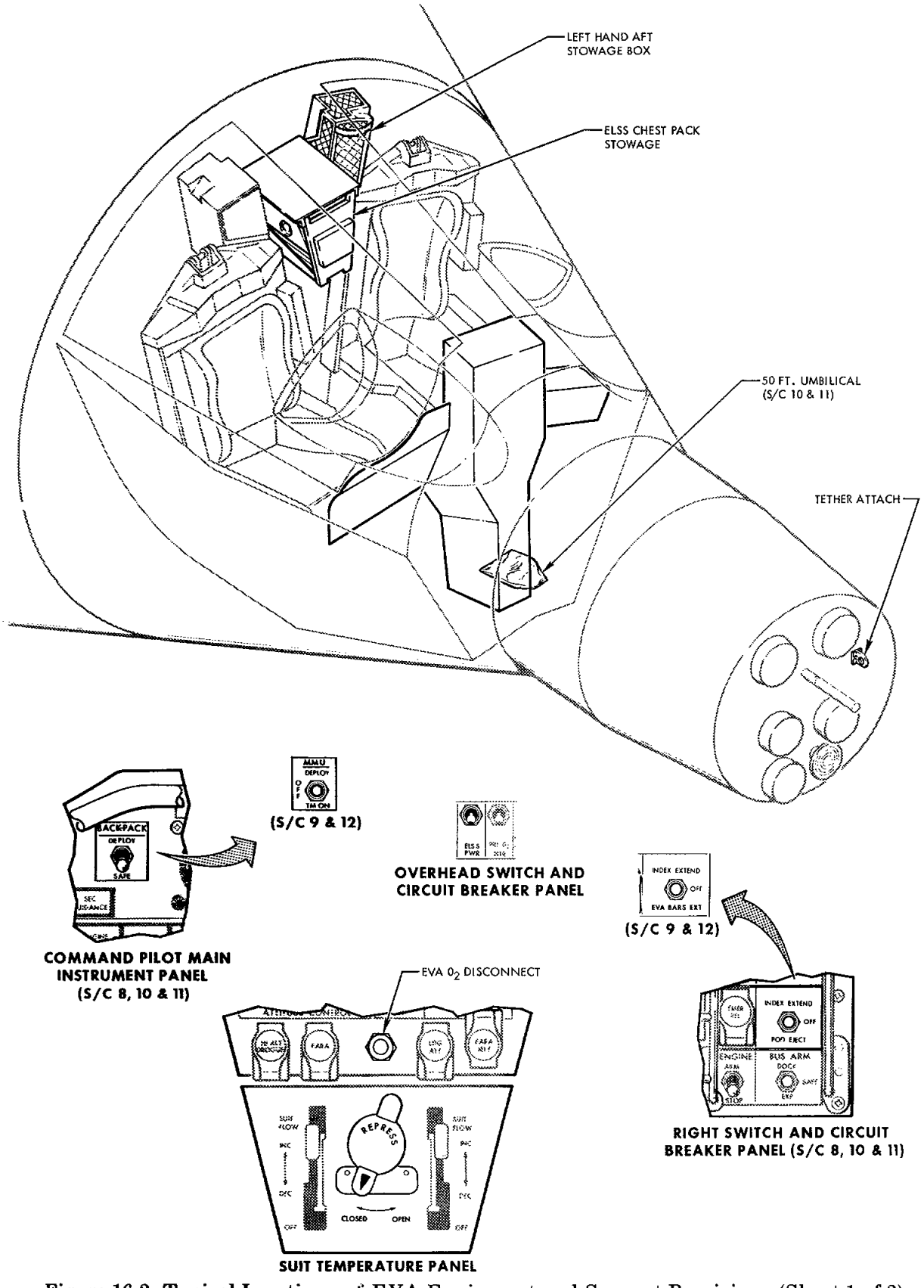


Figure 16-2 Typical Locations of EVA Equipment and Support Provisions (Sheet 1 of 2)

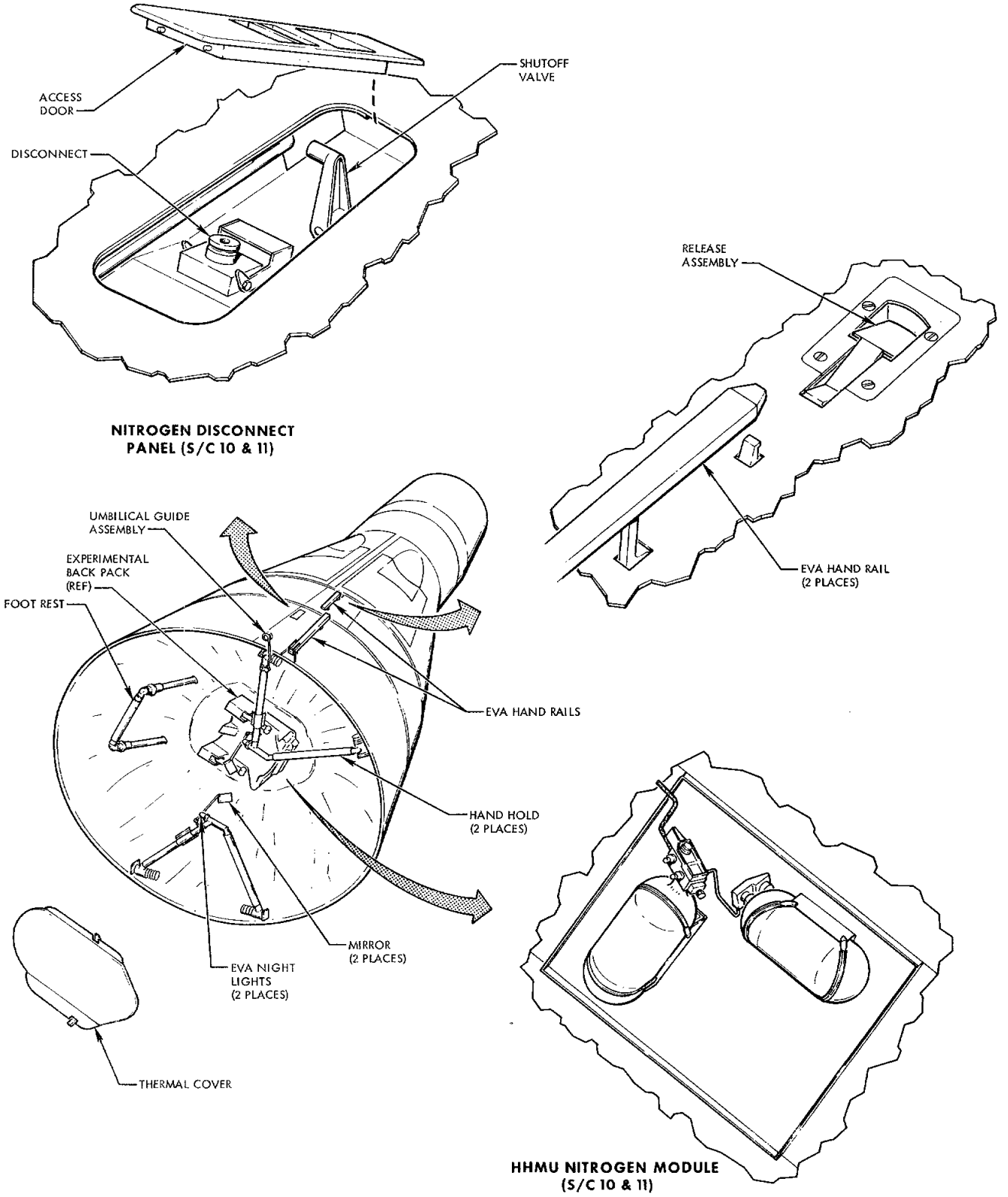


Figure 16-2 Typical Locations of EVA Equipment and Support Provisions (Sheet 2 of 2)

EVA support provisions consist of handrails, handholds, footrest, lights and tether attach points. These aid the pilot in maneuvering to the rear of the adapter and in donning the back pack or performing experiments in that area. Figure 16-2 illustrates the typical location of EVA equipment and support provisions.

SYSTEM OPERATION

The ELSS provides suit pressurization and environmental control during EVA. Oxygen, from the spacecraft primary oxygen supply, is supplied to the ELSS chest pack via the umbilical. The umbilical oxygen attach point is located on the center pedestal in the spacecraft cabin. Oxygen flow to the umbilical is controlled by the REPRESS valve on the suit temperature panel. On spacecraft missions utilizing a back pack, a limited oxygen supply in the back pack replaces the spacecraft oxygen supply to permit farther excursions from the spacecraft.

In addition to supplying oxygen to the chest pack, the umbilical provides voice communications and telemetry information, as well as a tether restraint. The modified ELSS umbilical used on spacecraft 10 and 11 also includes a propellant hose for the HHMU. The electrical jumper provides a single connecting point for the chest pack and pressure suit to the umbilical.

ELSS CHEST PACK OPERATION

Oxygen Loop - Normal Operation

Oxygen enters the chest pack at the spacecraft UMBILICAL O_2 CONNECTOR or the MMU O_2 CONNECTOR. The connectors are self-sealing quick disconnects. In either case the oxygen is supplied through check valves to a flow control

valve. Figure 16-3 is a functional diagram of the ELSS chest pack and Figure 16-4 illustrates its physical construction and external controls and indicators.

Oxygen flow to the suit loop is manually controlled by the O_2 FLOW SELECTOR valve. The selector valve has three positions: OFF, MEDIUM (restricted flow), and HIGH (full open). A push-to-turn motion is required when changing the valve position. The valve handle and valve position indications are illuminated for use during darkness. The selector valve is adjusted to give maximum comfort.

Oxygen enters the suit loop at the ejector. The ejector supplies the means of circulating the flow through the suit loop. The energy of the expanding high-pressure oxygen (92 psia) from the selector valve provides the required energy for circulation of the low-pressure (3.7 psia) gas from the evaporator-condenser.

The oxygen entering the ejector is heated by two 10 watt heaters connected in series. One of the heaters is wrapped around the ejector throat and the other is wrapped around the inlet line to the ejector. The heaters are energized when power is applied to the chest pack.

Suit pressure is maintained at 3.7 psia by the suit outflow valve which vents a portion of the exhausted suit gas overboard. In addition to maintaining suit pressure, the suit outflow valve accomplishes two additional functions: sufficient carbon dioxide is vented to maintain a safe level within the suit loop and a portion of the heat load is removed with the vented exhausted suit gas. The remaining exhausted suit gas enters the evaporator-condenser to be cooled.

The evaporator-condenser is a temperature-controlled heat exchanger. Cooling is accomplished by transferring heat to water stored in the integral metal

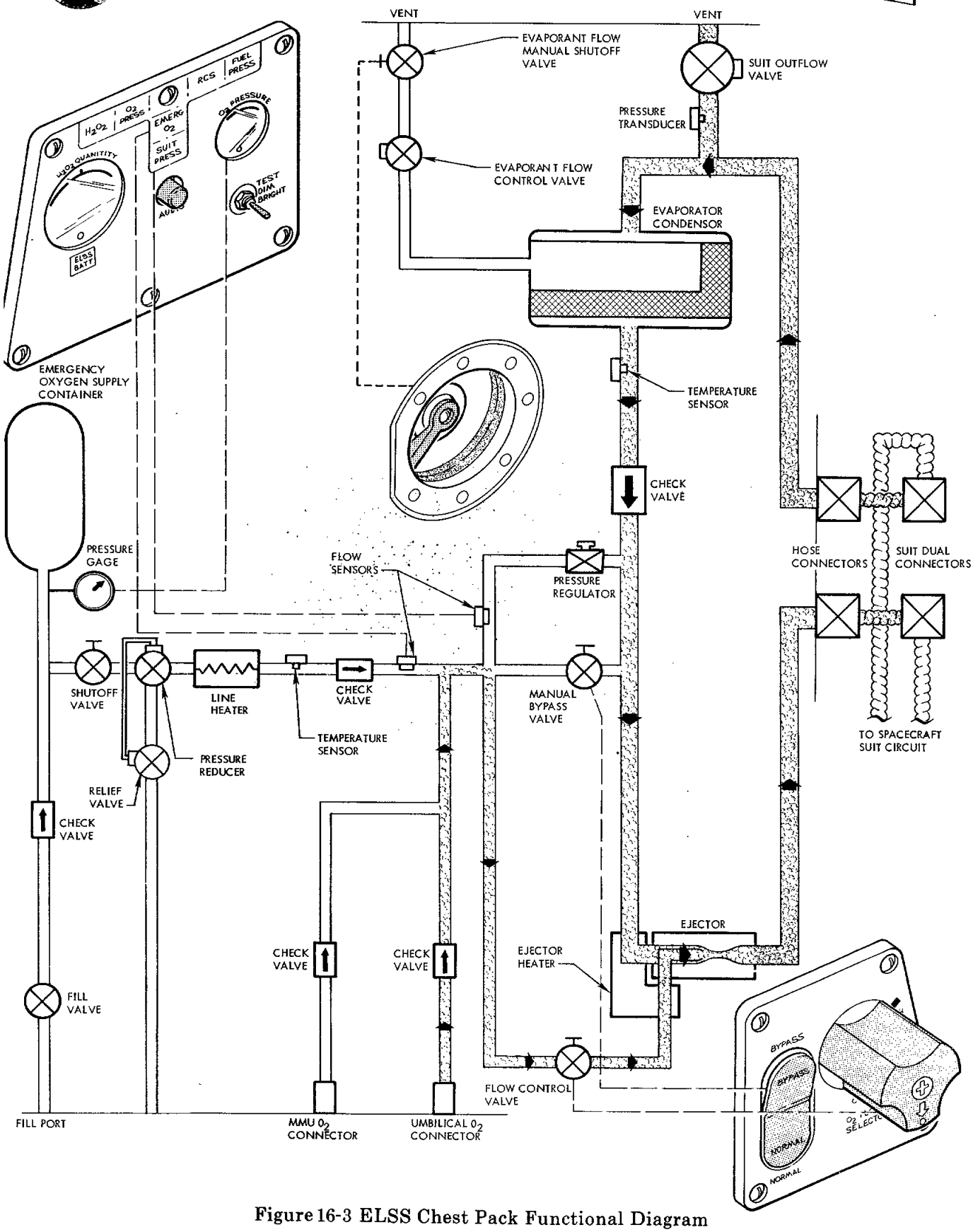
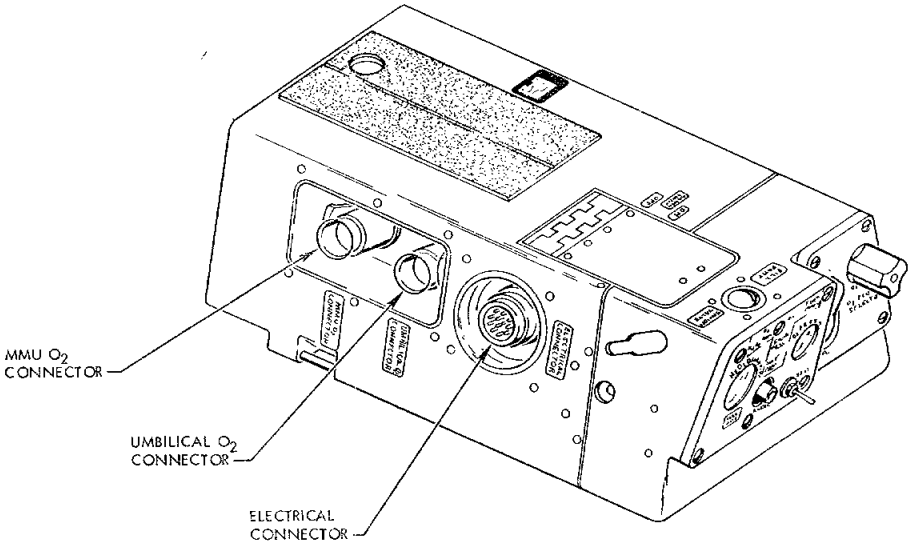
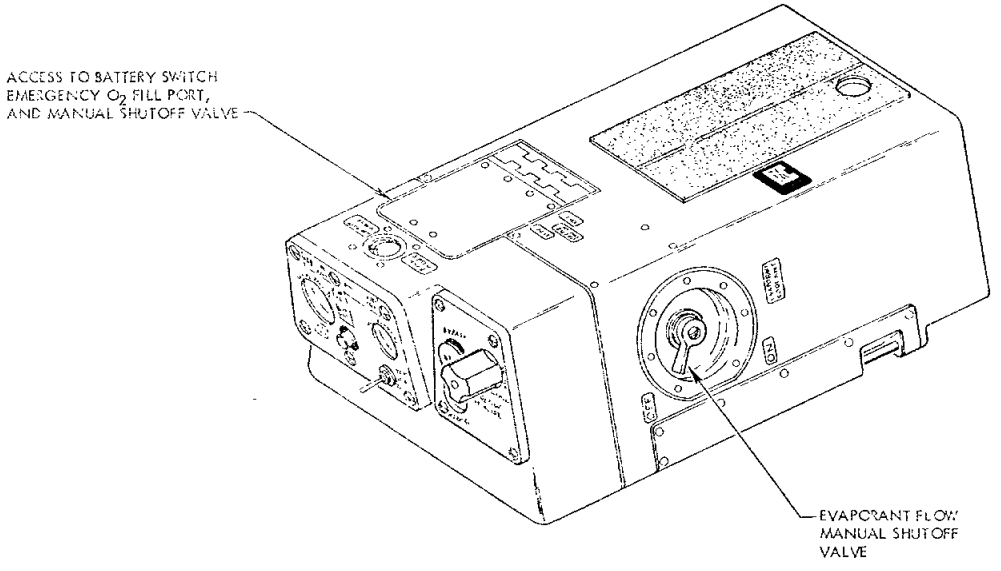


Figure 16-3 ELSS Chest Pack Functional Diagram





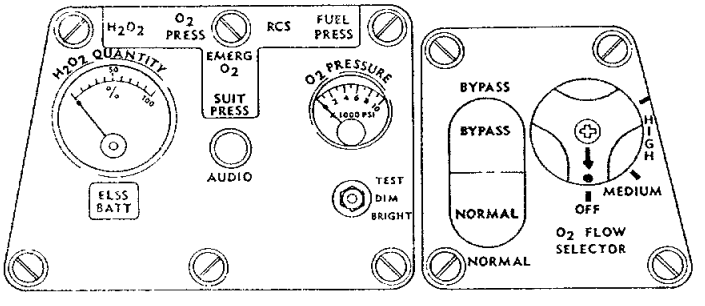


Figure 16-4 ELSS Chest Pack

wicks of the heat exchanger. As the gas is cooled the water vapor in the exhausted suit gas condenses and is separated and removed with evaporant wicks by capillary action. The condensed water vapor is passed through the evaporant wicks to the evaporative heat sink. The evaporative heat sink uses the condensed water vapor as a coolant, thereby reducing the initial amount of water which must be stored in the heat sink.

The outlet temperature of the gas leaving the evaporator-condenser is controlled by the EVAPORANT FLOW CONTROL valve. A temperature sensor at the evaporator-condenser outlet controls the operation of the EVAPORANT FLOW CONTROL valve to maintain the temperature at a nominal 45 degrees Fahrenheit. An increase in outlet gas temperature opens the flow control valve, lowering the boiling point of the stored water, thus allowing greater heat dissipation in the heat exchanger. The EVAPORANT FLOW CONTROL valve is shut-off until the ELSS is in operation to prevent loss of the water in the heat sink. If suit temperature falls to an uncomfortable level during operation, the valve may be closed.

Oxygen Loop - Emergency Operation

There are three conditions that are considered as emergencies. These conditions are: loss of external oxygen supply pressure, low suit pressure, and heat exchanger failure. Each of these three conditions is cause for aborting the EVA. The first two conditions are automatically compensated for; while the third, which is not as critical, requires manual action by the pilot.

In the event the external pressure falls below 67 $\pm$ 10 psig (reference to ambient conditions), the emergency pressure reducer automatically actuates

to supply emergency oxygen to the ejector. The emergency oxygen supply has the capacity of delivering oxygen to the ejector for a period of 16 to 27 minutes, depending upon the setting of the O_2 FLOW SELECTOR valve. The pressure reducer has a relief feature which will vent the emergency oxygen supply overboard should the pressure reducer fail in the open position. Sufficient oxygen will be vented to avoid pressurizing the suit loop above 5.0 psig. A pressure gage on the instrument panel allows monitoring of the emergency oxygen supply pressure. The emergency oxygen, upon leaving the storage tank, passes through a manual shutoff valve, a pressure reducer, and a line heater. A temperature sensor located downstream of the heater is used in conjunction with a temperature control circuit to maintain the emergency oxygen at a temperature of 45 degrees Fahrenheit. A flow sensor in the emergency oxygen line initiates an audio and visual alarm when the emergency oxygen supply is being used.

A demand type suit-pressure regulator in parallel with the BYPASS-NORMAL and O_2 FLOW SELECTOR valves opens in the event suit pressure decays below 3.3 psia. Oxygen flow through the suit-pressure regulator is from the same supply which is supplying the suit loop via the ejector. The suit-pressure regulator will maintain suit pressure at 3.3 psia. A flow sensor in series with the regulator initiates an audio and visual alarm when flow through the regulator is sensed.

A failure of the evaporator-condenser will cause a considerable rise in suit inlet temperature. If, after increasing the O_2 FLOW SELECTOR valve to HIGH, sufficient cooling in the suit loop is not obtained, the BYPASS-NORMAL valve is placed in the BYPASS position. This increases the flow of oxygen to the

suit loop. The BYPASS-NORMAL valve is the push-to-open and push-to-close type. Valve position is illuminated for use during darkness.

Electrical Operation

The electrical portion of chest pack consists of a temperature control and oscillator module, oscillator and light controller module, battery, relay, heaters, and associated switches and indicators. Figure 16-5 is a schematic of the chest pack electrical components.

The chest pack operates from battery power or external power from the spacecraft supplied through the umbilical. During normal operation external power is utilized. A control relay in the chest pack is used to select internal or external power. The relay is energized through the umbilical and supplies external power to the chest pack when the spacecraft ELSS PWR circuit breaker is closed. When the relay is de-energized battery power is utilized and the ELSS PWR indicator on the chest pack instrument panel illuminates.

The battery is the silver-zinc alkaline type using potassium hydroxide as the electrolyte. The battery consists of a group of series-parallel cells connected to provide a nominal load voltage of 24 vdc. Regardless of the power used, the chest pack circuits are not powered until the battery power switch, located underneath the access door on the chest pack, is placed in the ON position.

The oscillator and light controller module consists of a voltage regulator and a logic control circuit. The voltage regulator supplies the necessary 28 vdc and 14 vdc regulated voltages. The logic control circuit consists of a

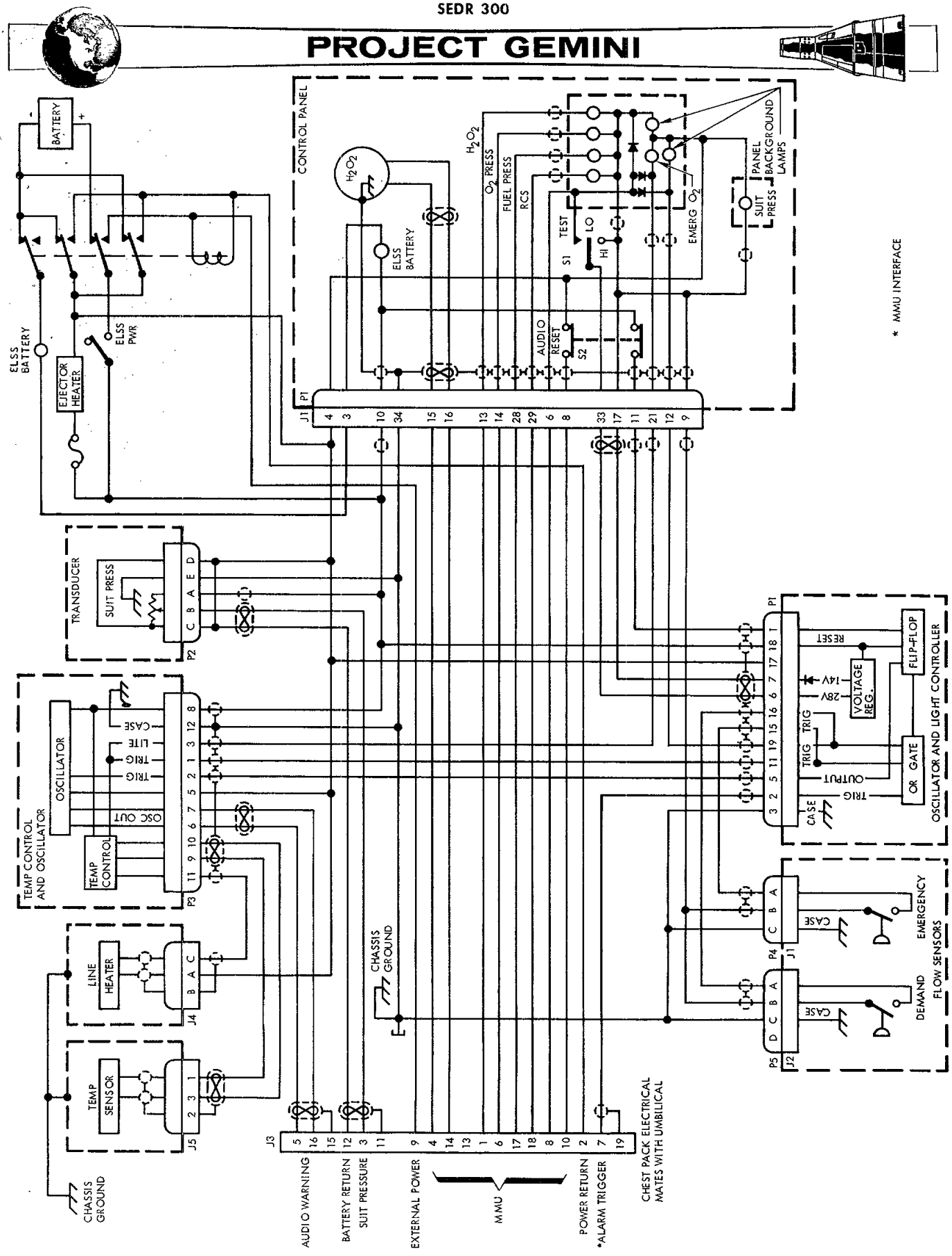


Figure 16-5 Chest Pack Electrical Schematic

flip-flop and an OR gate. The OR gate obtains trigger inputs from the suit pressure regulator and emergency oxygen flow sensors. Upon receipt of one of the trigger inputs, the OR gate causes the flip-flop to change states and provide an output to the audio oscillator in the temperature control and oscillator module. The alarm can be reset by depressing the AUDIO pushbutton on the control panel which causes the flip-flop to reset.

The temperature control and oscillator module provides control for the emergency oxygen line heater and the audible and visible warning devices. The emergency oxygen line heater is controlled by the temperature control circuitry obtaining inputs from the emergency oxygen temperature sensor and emergency oxygen flow sensor. When the temperature sensor indicates a temperature below 45 degrees Fahrenheit, and a trigger signal from the emergency oxygen flow sensor exists, the temperature control circuit allows power to be supplied to the line heater. When the temperature at the sensor exceeds 45 degrees Fahrenheit, the line heater is deactivated. A signal representing a temperature in excess of 45 degrees Fahrenheit and an emergency oxygen flow signal have to be present before the line heater is activated. The EMERG O_2 light is also illuminated by the emergency oxygen flow signal. The oscillator is activated by the output from the flip-flop of the oscillator and light controller and will supply an audible alarm to the pilots headset and to the spacecraft via the umbilical.

The suit-pressure regulator flow sensor signal, in addition to triggering the OR gate of the oscillator and light controller, is used to illuminate the SUIT PRESS indicator light.

The pressure transducer in the suit loop monitors suit pressure. The output is

returned to the spacecraft for transmission as telemetry information. The pressure transducer is a variable reluctance transducer with an output of 0 to 5 vdc with a sensed pressure of 2.5 to 5 psig.

The wrap-on heaters at the ejector provides 20 watts of power to maintain the suit loop inlet temperature at 45 ± 10 degrees Fahrenheit. When the battery switch is placed in the ON position, power is applied to the wrap-on heaters.

The panel illumination lights are turned on and off by the battery switch. A three position switch (TEST, DIM, BRIGHT) controls the intensity of the panel lights and the emergency indicator lights. In the TEST position, the panel lights, emergency indicator lights, and the warning alarm audio oscillator are tested. The voltage regulator provides the reduced voltage for the low-intensity position of the light switch.

The following light indicators on the instrument panel are not operational unless the experimental back pack is connected: H_2O_2, RCS, O_2 PRESS, and FUEL PRESS. The H_2O_2 pressure indicator is operational when either of the back packs are connected.

HAND HELD MANEUVERING UNIT OPERATION

The HHMU (Figure 16-6) provides the necessary propulsion required to maneuver in space. The HHMU has two tractor nozzles and a larger pusher nozzle. The HHMU provides two pounds of thrust in the positive or braking direction. The pusher nozzle is attached to the major assembly and is used to supply the braking thrust. The two tractor nozzles are on arms which extend out on each side. This positions the nozzles for thrusting in the positive or forward direction.

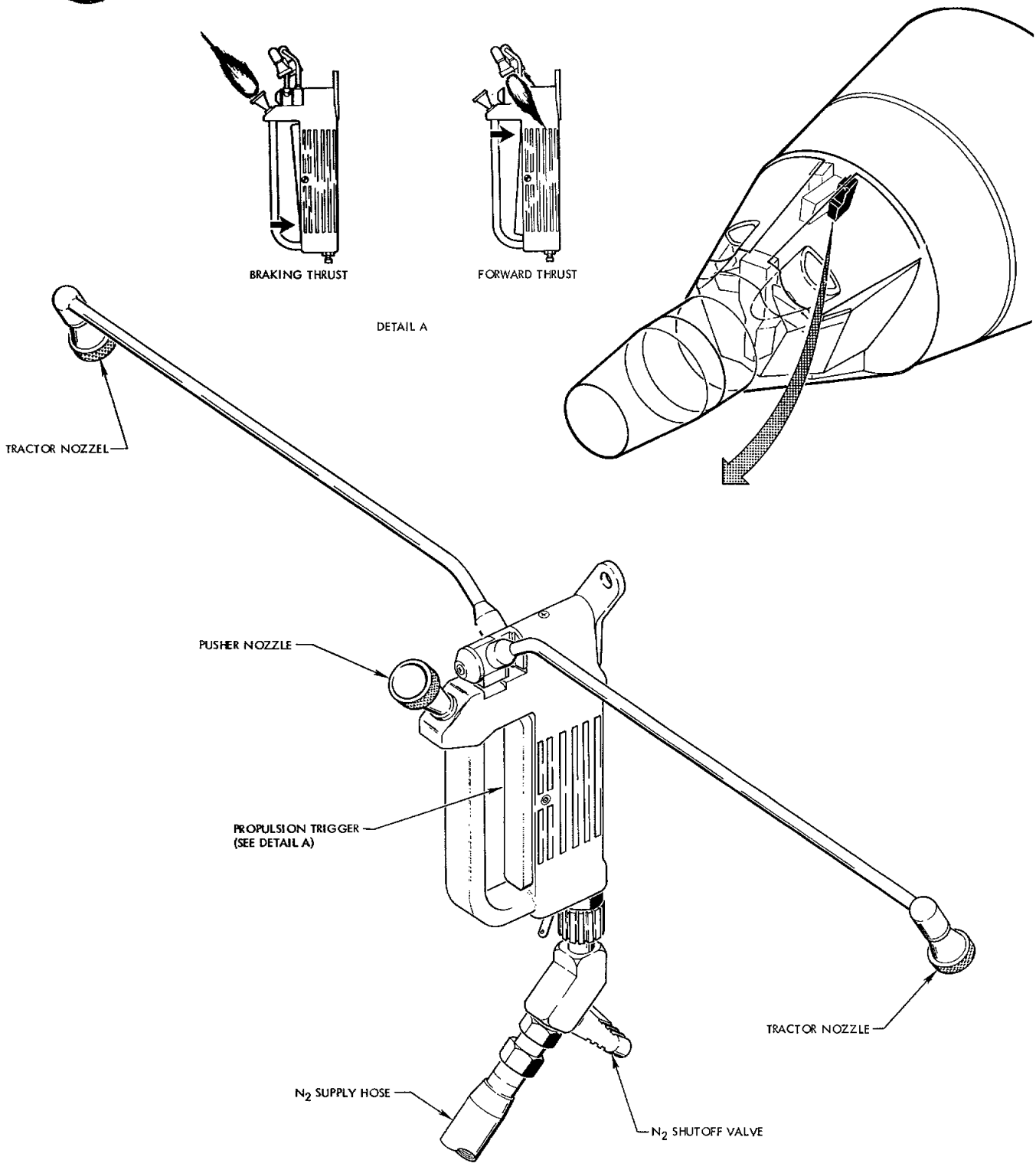


Figure 16-6 Hand Held Maneuvering Unit

The trigger is pivoted in the center so the tractor nozzles or the pusher nozzle can be controlled with one hand. Squeezing the lower end of the trigger supplies nitrogen to the pusher nozzle and squeezing the upper end of the trigger supplies nitrogen to the tractor nozzles.

The propulsion gas for the HHMU is provided by two bottles of gaseous nitrogen that are mounted in the adapter equipment section. The gas is routed to the adapter skin through metal tubing. A quick disconnect connector and a lever shutoff valve are located on the outside of the adapter and can be reached by opening a small door on the spacecraft skin.

SYSTEM UNITS

ELSS CHEST PACK

The ELSS chest pack is 18 by 10 by 6 inches and weighs 42 pounds when fully charged. Operating devices on the chest pack used by the pilot include four valves, three switches, two meters, seven indicators, and five connectors. The chest pack housing is molded from fiberglass with a volar finish.

When in use, the chest pack is positioned on the pilots chest (occupying an area from above the thighs to below the chin), and the control panels are readily visible and accessible. The chest pack is held in place by a self-adhering web-belt attached to the pilots personal harness. The belt ends are laid across strips of the same self-adhering, web-belt material affixed to the chest pack.

Two dual oxygen connectors are provided with the chest pack to permit continuous oxygen flow to the pilots pressure suit during the egress and ingress preparation procedures. The dual connectors are stowed in the left hand aft stowage container.

ELSS 25 FOOT UMBILICAL AND ELECTRICAL JUMPER

The 25 foot umbilical (Figure 16-7) is an aluminized mylar and nylon covered assembly that contains the pilots oxygen supply hose, electrical leads and restraint tether.

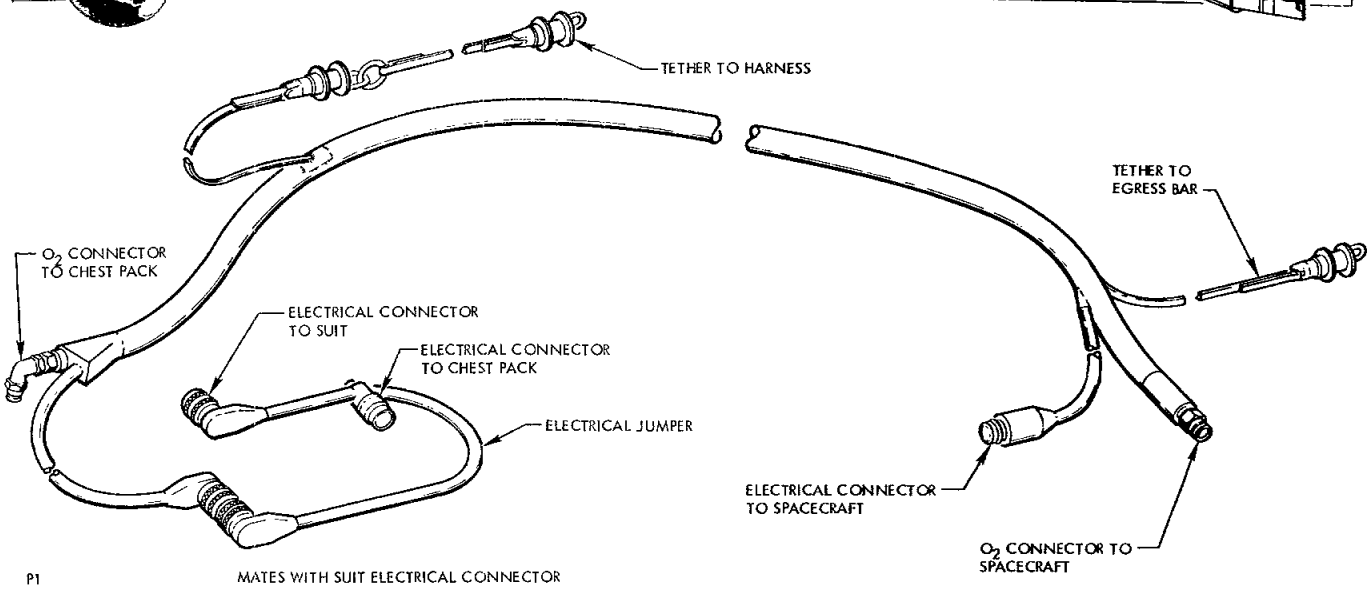
The restraint tether consists of a flat nylon ribbon with a minimum breaking strength of 900 pounds. The tether is connected by snap rings to the right egress bar, located at the forward end of the right hatch opening, and to the D-ring on the pilots harness.

Electrical leads through the umbilical are wrapped around the oxygen hose and carry the bio-med instrumentation and communication signals, and spacecraft power to the chest pack. The umbilical is protected in passing over the hatch opening by an umbilical cord guard.

The oxygen supply hose routes spacecraft primary oxygen to the chest pack. It connects to the disconnect on the suit temperature panel in the spacecraft cabin. The oxygen supply hose is disconnected when the back pack is connected. Oxygen is supplied from the supply tank contained in the back pack.

The electrical jumper provides a single connecting point between the pressure suit and chest pack to the umbilical. The electrical jumper interconnects the signals passing between the chest pack, pressure suit, and spacecraft.

The 25 foot umbilical, electrical jumper, and umbilical cord guard are stowed in the left hand aft stowage container. The 25 foot umbilical, electrical jumper, and umbilical cord guard are used on spacecraft 8, 9 and 12 missions.



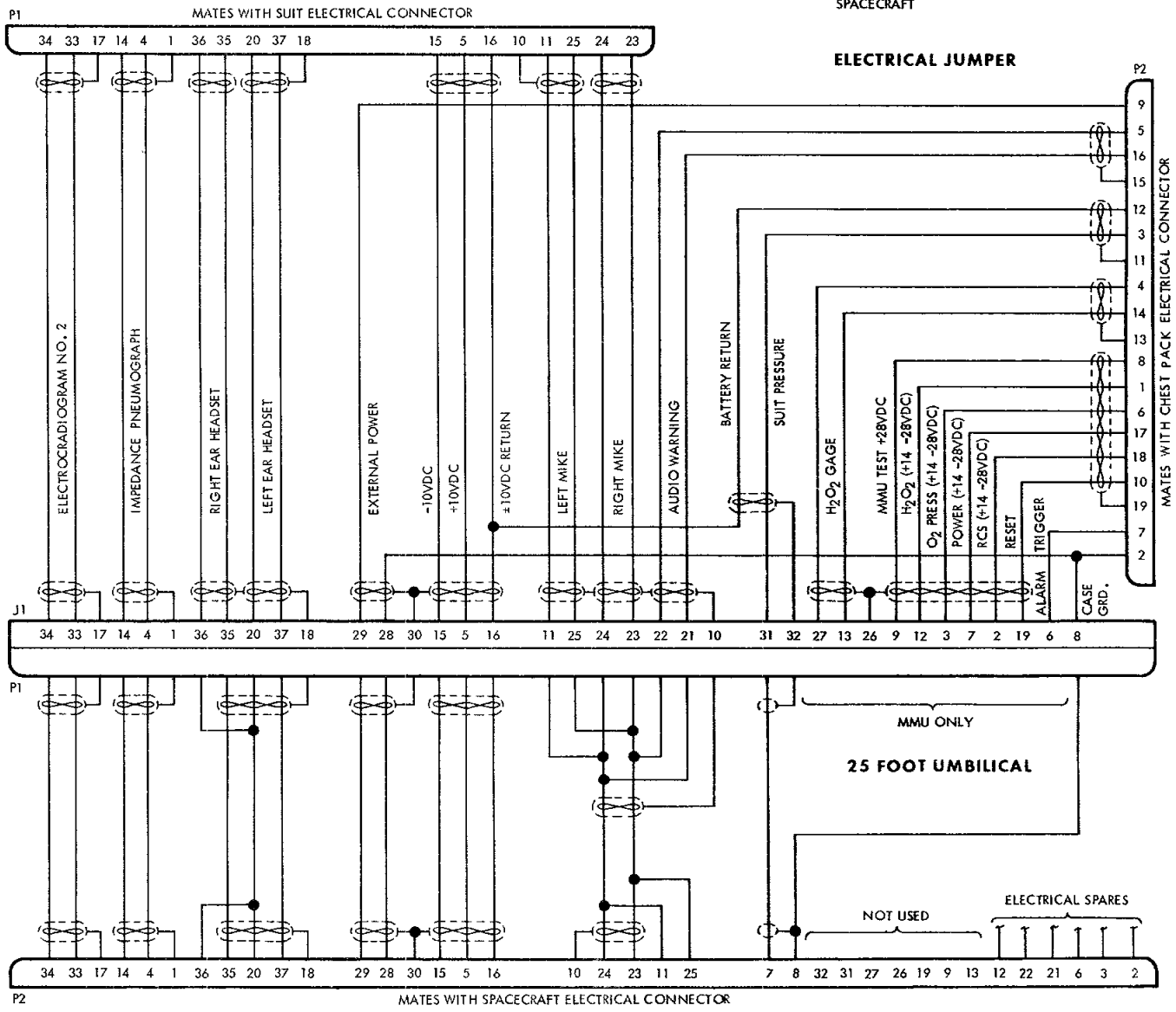


Figure 16-7 ELSS 25 Foot Umbilical and Electrical Jumper

ELSS 50 FOOT UMBILICAL

The 50 foot umbilical (Figure 16-8) is an aluminized mylar and nylon covered assembly that contains the pilots oxygen supply hose, HHMU nitrogen hose, electrical leads and restraint tether.

The restraint tether consists of a flat nylon ribbon with a minimum breaking strength of 900 pounds. The tether is connected by a snap ring to the egress bar in front of the right seat and by a harness clamp to the pilots harness.

Electrical leads through the umbilical are wrapped around the hoses and carry the bio-med instrumentation and communication signals. The cabling monitors an electrocardiogram and impedance pneumograph. Communications are established through a left and right microphone and a left and right earphone. Electrical power is supplied in + 10 vdc and -10 vdc along with power for chest pack operation. The umbilical is protected in passing over the hatch opening by a nylon chaffing guard. The oxygen supply hose connects to the oxygen disconnect on the suit temperature panel in the spacecraft and to the UMBILICAL O_2 CONNECTOR on the chest pack. The nitrogen hose connections are the quick disconnect aft of the pilots hatch and the HHMU.

The 50 foot umbilical is stored in the left hand footwell stowage pouch. The 50 foot umbilical is used on spacecraft 10 and 11 missions.

HAND HELD MANEUVERING UNIT

The HHMU provides the necessary propulsion required to maneuver in space. The HHMU is stowed in the left hand aft stowage container. The HHMU is used with

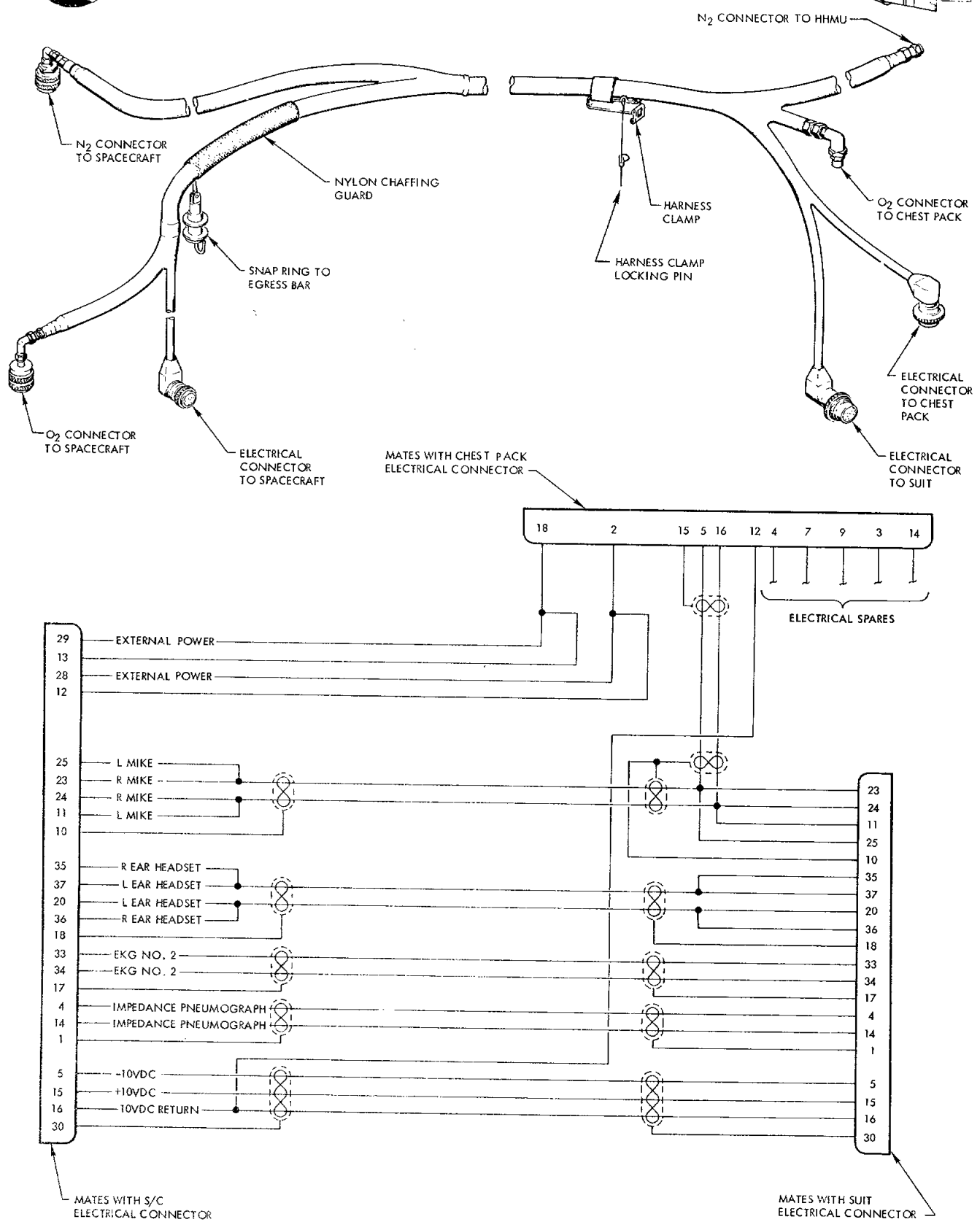


Figure 16-8 ELSS 50 Foot Umbilical

the 50 foot umbilical on spacecraft 10 and 11.

The HHMU is constructed of fiberglass reinforced with polyester resin. The HHMU is approximately 11 inches long, 6 inches wide, and 1 inch thick. A pivoted trigger controls the operation of the two valves. The pusher nozzle is mounted on the main body of the HHMU. The two tractor nozzles are on 14 inch arm assemblies which swivel away from the forward end of the HHMU.

GASEOUS NITROGEN MODULE

The gaseous nitrogen module (Figure 16-2) is bracket mounted in the spacecraft adapter section. The module consists of two bottles, a pressure regulator, and fill valve. Nitrogen from the bottles is routed to a quick disconnect and shutoff valve located aft of the right hatch by a quarter inch metal tubing.

An access door (Figure 16-2) protects the disconnect and valve during the launch phase. The door is opened by compressing the spring latch assembly. Movement of the door past the vertical position disengages the hinges for removal.

Each nitrogen bottle will hold up to 438 cubic inches of gaseous nitrogen, providing a system capacity of 879 cubic inches, which represent 11.5 pounds of nitrogen (10.75 pounds of the nitrogen is usable for the pilot). The nitrogen is stored at a pressure 5000 to 5500 psig and is regulated to 120 $^{+10}_{-15}$ psig by a built-in regulator on the bottle. The pressure at the HHMU will be approximately 100 psig. The system has sufficient capacity for a little over five minutes of continuous usage.

EVA HANDRAILS

The two EVA handrails are located along the top of the adapter and aid the pilot in maneuvering from the spacecraft cabin to the rear of the adapter. The handrails are illustrated in Figure 16-9.

Launch vehicle-spacecraft separation releases the handrail on the adapter equipment section. The handrail is held in the stowed position by a release pin. The release pin is physically attached to the launch vehicle. At separation, the release pin is pulled away with the launch vehicle allowing the handrail to move slightly aft. This allows the handrail to clear the spacecraft retaining hooks and self-extend by spring force. Upon full extension the handrail is locked in the extended position.

The forward handrail is extended manually by the pilot when he depresses the release button. Depressing the release button releases the stop and allows the handrail to move slightly forward. This forward movement allows the handrail to clear the retaining hooks and self-extend by spring force.

An umbilical guide (Figure 16-2) is located at the rear of the handrail on the adapter equipment section. The umbilical guide assembly prevents damage to the umbilical in passing over the edge of the adapter. The umbilical guide assembly is extended when the handholds and footrest in the adapter are extended.

HANDHOLDS AND FOOTREST

The handholds and a footrest are located in the adapter on spacecraft 8, 9, 11, and 12 missions. These provide a means for the pilot to hold on and remain in

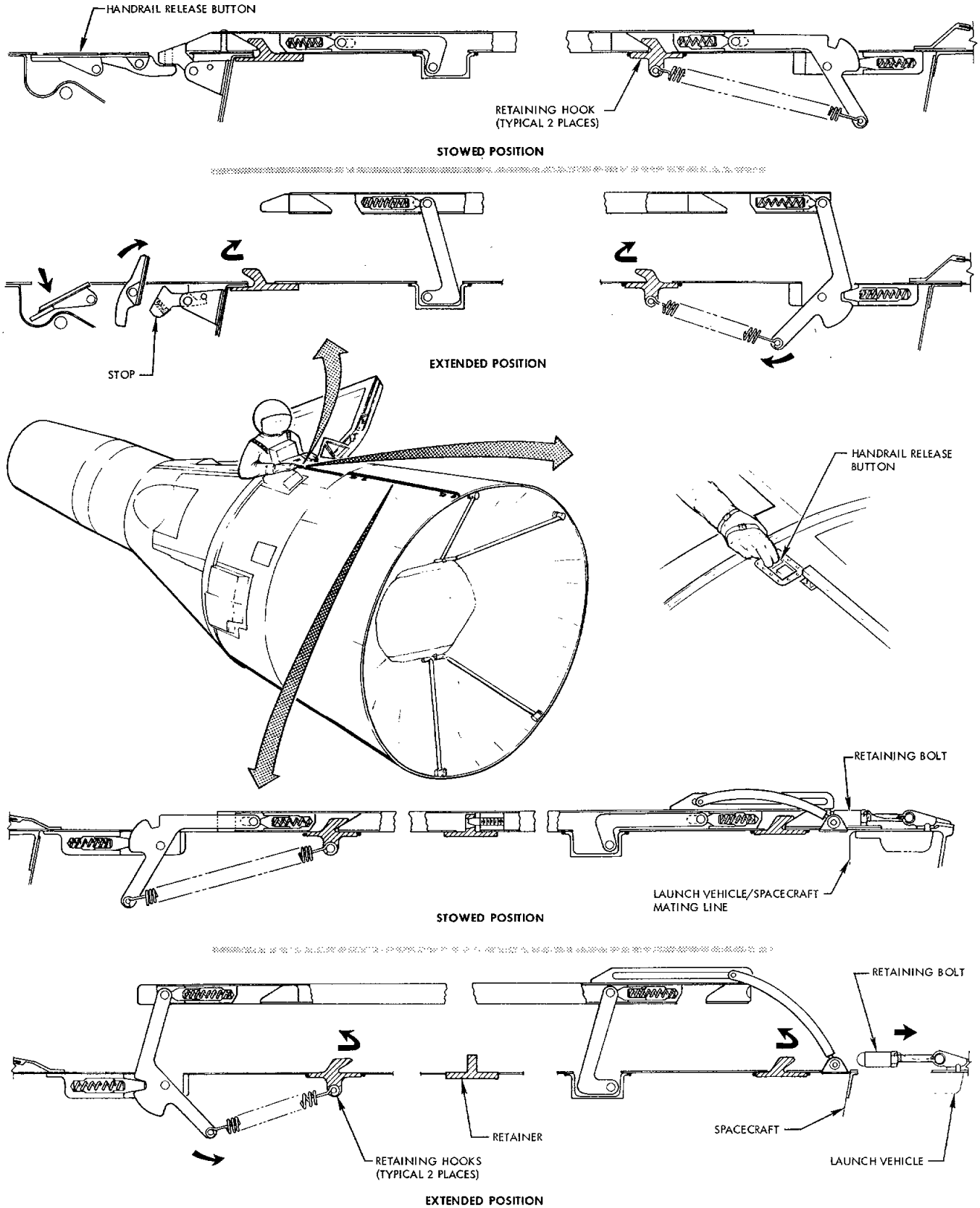


Figure 16-9 EVA Handrails

the adapter to don the back pack (on spacecraft mission 8, 9 and 12) or perform experiments. Figure 16-2 illustrates the handholds and foot rest.

The handholds and footrest are retracted prior to use and are self-extended by spring action. On missions utilizing a back pack, the handholds are held in the retracted position by the thermal covering which protects the back pack. On missions without a back pack, a retaining bar is used.

The handholds and footrest are extended when the INDEX EXTEND-POD EJECT switch in the cabin is positioned to POD EJECT. The switch is located on the right switch and circuit breaker panel (Figure 16-2). The switch controls two guillotines which sever the cables holding the handholds and footrest in the retracted position. A mechanical latch locks them in position. On spacecraft 9 and 12 the control switch is labeled INDEX EXTEND-EVA BARS EXT.

A light is installed on each handhold to illuminate the adapter area during the dark side of the revolution. The lights are controlled by the EXT LTS switch on the right switch and circuit breaker panel. The lights are illustrated in Figure 16-2.

A mirror is attached to each handhold to allow the pilot to observe the back pack during the donning procedure. The mirrors may be positioned as required by the pilot. Figure 16-2 illustrates the physical location of the mirrors.

The back packs are attached to the spacecraft by means of a retaining bolt which passes through a guillotine. After the back pack is donned by the pilot, the command pilot positions the BACK PACK-DEPLOY switch (located on the main instrument panel) to the DEPLOY position. This actuates the guillotine and severs the retaining bolt releasing the back pack.