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GEORGE C. MARSHALL SPACE FLIGHT CENTER

SATURN V FLIGHT MANUAL

SA 504

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SATURN V FLIGHT MANUAL
SA-504

FORWARD

This manual was prepared to provide the astronaut with a single source reference as to the characteristics and functions of the SA-504 launch vehicle and AS-504 flight mission. A revision to the manual, incorporating the latest released data on the vehicle and mission, will be released approximately 30 days prior to the scheduled launch date.

The manual provides general mission and performance data, emergency detection system information, a description of each stage and the IU, and a general discussion of ground support facilities, equipment, and mission control. A bibliography identifies additional references if a more comprehensive study is desired.

Major hardware associated differences between Saturn V launch vehicles SA-503 and SA-504 have been annotated in the manual. They are identified by reference numbers in the margin adjacent to the new information. These reference numbers refer to footnotes which are located at the end of each section.

This manual is for information only and is not a control document. If a conflict should be discovered between the manual and a control document the control document will rule.

Recommended changes or corrections to this manual should be forwarded, in writing, to the Saturn V Systems Engineering Management Office (I-V-E), MSFC, Attention: Mr. H.P. Lloyd; or to the Crew Safety and Procedures Branch (CF-24), MSC, Attention: Mr. D.K. Warren.

SECTION I

GENERAL DESCRIPTION

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SATURN V SYSTEM DESCRIPTION

The Saturn V system in its broadest scope includes conceptual development, design, manufacture, transportation, assembly, test and launch. The primary mission of the Saturn V launch vehicle, three-stage-to-escape boost launch of an Apollo Spacecraft, established the basic concept. This mission includes a suborbital start of the third stage engine (S-IVB) for final boost into earth orbit and subsequent reignition to provide sufficient velocity for escape missions including the lunar missions.

LAUNCH VEHICLE DEVELOPMENT

The Saturn launch vehicles are the product of a long evolutionary process stemming from initial studies in 1957 of the Redstone and Jupiter missiles. Early conceptual studies included other proven missiles such as Thor and Titan and considered pay loads ranging from earth orbiting satellites to manned spacecraft such as Dynasoar, Mercury, Gemini, and eventually Apollo.

The Saturn V launch vehicle evolved from the earlier Saturn vehicles as a result of the decision in 1961 to proceed with the Apollo manned lunar mission. As the Apollo mission definition became clear conceptual design studies were made considering such parameters as structural dynamics, staging dynamics, and propulsion dynamics.

Design trade-offs were made in certain areas to optimize the launch vehicle design based on mission requirements. The best combination of design parameters for liquid propellant vehicles result in low accelerations and low dynamic loads. Reliability, performance and weight were among primary factors considered in optimizing the design.

Although the Saturn/Apollo vehicle is approximately the same length and weight as a modern Navy Destroyer its structural design carefully considered the weight factor. Structural rigidity requirements are dictated largely by two general considerations: flight control dynamics and propellant slosh problems. Gross dimensions (diameter & length) are dictated generally by propellant tankage size.

As propulsion requirements were identified system characteristics emerged: thrust levels, burning times, propellant types, and quantities. From these data engine requirements and characteristics were identified and the design and development of the total launch vehicle continued centered around the propulsion systems.

Some of the principal design ground rules developed during the conceptual phase, which were applied in the final design, are discussed in the following paragraphs.

VEHICLE DESIGN GROUND RULES

Safety

Safety criteria are identified by Air Force Eastern Test Range (AFETR) Safety Manual 127-1 and AFETR Regulation 127-9.

Crew safety considerations required the development of an Emergency Detection System (EDS) with equipment located throughout the launch vehicle to detect emergency conditions as they develop. If an emergency condition is detected this system will either initiate an automatic abort sequence or display critical data to the flight crew for their analysis and reaction.

Each powered stage is designed with dual redundant range safety equipment which will effect engine cutoff and propellant dispersion in the event of a launch abort after liftoff. Engine cutoff results from closing valves and terminating the flow of fuel and oxidizer. Propellant dispersion is accomplished by detonating linear shaped charges and longitudinally opening the propellant tanks.

Stage Separation

The separation of the launch vehicle stages in flight required design studies involving consideration of many parameters such as time of separation, vehicle position, vehicle attitude, single or dual plane separation; and the type, quantity, and location of ordnance.

Separation of the launch vehicle stages in flight is accomplished by explosively severing a circumferential separation joint and firing retrorocket motors to decelerate the spent stage. Stage separation is initiated when stage thrust decays to a value equal to or less than 10% of rated thrust. A short coast mode is used to allow separation of the spent stage and to effect ullage settling of the successive stage prior to engine ignition.

A delayed dual plane separation is employed between the S-IC and S-II stages while a single plane separation is adequate between the S-II and S-IVB stages.

Umbilicals

In the design and placement of vehicle plates consideration was given to such things as size, locations, methods of attachment, release, and retraction.

The number of umbilicals is minimized by the combining of electrical connectors and pneumatic and propellant couplings into common umbilical carriers. Location of the umbilicals depended upon the location of the vehicle plates which were limited somewhat by the propellant tanking, plumbing, and wiring runs inside the vehicle structure. Umbilical disconnect and retraction systems are redundant for reasons of reliability and safety.

Electrical Systems

An electrical load analysis of the launch vehicle provided the basic data (voltage, frequency, and power requirements) for design of the electrical system.

Such factors as reliability, weight limitations, and weight distributions dictated the requirements to minimize electrical wiring yet distribute the electrical loads and power sources throughout the launch vehicle. Each stage of the vehicle has its own independant electrical system. No electrical power is transferred between stages; only control signals are routed between stages.

Primary flight power is supplied by wet cell batteries in each stage. The sizes, types, and characteristics are discussed in subsequent sections of this manual. Where alternating current or direct current with a higher voltage than the batteries is required inverters and/or converters convert the battery power to the voltages and frequencies needed.

All stages of the launch vehicle are electrically bonded together to provide a unipotential structure and minimize current transfer problems in the common side of the power systems.

MANUFACTURE AND LAUNCH CONCEPT

The development of the vehicle concept required concurrent efforts in the areas of design, manufacture, transportation, assembly, check-out, and launch.

The size and complexity of the vehicle resulted in the decision to have detail design and manufacture of each of the three stages, the IU, and the engines accomplished by separate contractors under the direction of MSFC.

This design/manufacturing approach required the development of production plans and controls and transportation and handling systems capable of handling the massive sections. This is described in more detail in following paragraphs.

The assembly, checkout, and launch of the vehicle required the development of an extensive industrial complex at KSC. Some of the basic ground rules which resulted in the KSC complex described in section VIII are:

1. The vehicle would be assembled and checked out in a protected environment before being moved to the launch site.
2. A final checkout will be performed at the launch site prior to launch.
3. Once the assembly is complete the vehicle will be transported in the erect position without disconnecting the umbilicals.
4. Automatic checkout equipment would be required.
5. The control center and checkout equipment would be located away from the launch area.

LAUNCH REQUIREMENTS

Some of the launch requirements which have developed from the application of these ground rules are:

1. Several days prior to the actual launch time the vehicle is moved to the launch area for prelaunch servicing and checkout. During most of this time the vehicle systems are sustained by ground support equipment. However, at T-50 seconds power is transferred to the launch vehicle batteries and final vehicle systems monitoring is accomplished. In the event of a hold, the launch vehicle can operate on internal power for up to 12 hours before a recycle for batteries would be required.
2. While in the launch area environmental control within the launch vehicle is provided by environmental control systems in the mobile launcher and on the pad. The IU also utilizes an equipment cooling system in which heat is removed by circulation of a methanol/water coolant. During preflight heat is removed from the coolant by a GSE cooling system located on the ML. During flight heat is removed from the coolant by a water sublimator system.
3. While in transit between assembly area and launch area, or while in the launch area for launch preparations, the assembled launch vehicle must withstand the natural environment. The launch vehicle is designed to withstand 99.9% winds dur-

ing the strongest wind month while either free standing or under transport, with the damper system attached. In the event of nearby explosion of a facility or launch vehicle the Saturn V will also withstand a peak overpressure of 0.4 psi.

4. To more smoothly control the engine ignition, thrust buildup and liftoff of the vehicle, restraining arms provide support and hold down at four points around the base of the S-IC stage. A gradual controlled release is accomplished during the first six inches of vertical motion.

RELIABILITY AND QUALITY ASSURANCE

The Apollo Program Office, MA, has the overall responsibility for development and implementation of the Apollo reliability and quality assurance (R & QA) program. NASA Centers have the responsibility for identifying and establishing R & QA requirements and implementing an R & QA program to the extent necessary to assure the satisfactory performance of the hardware for which they are responsible. The Apollo R & QA program is defined by the Apollo Program Development Plan, M-D MA 500 and Apollo R & QA Program Plan, NHB 5300-1A.

Crew safety and mission success are the main elements around which the R & QA program is built. The primary criterion governing the design of the Apollo system is that of achieving mission success without unacceptable risk of life or permanent physical disablement to the crew.

It is Apollo program policy to use all applicable methods and disciplines which can contribute to the making of sound evaluations and decisions to ensure the reliability and quality of Apollo/Saturn systems. Some of these methods and disciplines are discussed in subsequent paragraphs.

Analysis of Mission Profiles

The mission profile is analyzed to determine the type and scope of demands made on equipment and flight crew during each phase of the mission. This has resulted in the incorporation of design features which will enable the flight crew to detect and react effectively to abnormal circumstances. This permits the flight crew to abort safely if the condition is dangerous or continue the normal mission in an alternate mode if crew safety is not involved but equipment is not operating properly.

Failure Effects and Criticality Analyses

The modes of failure for every critical component of each system are identified. The effect of each failure mode on the operation of the system is analyzed and those parts contributing most to unreliability are identified. These analyses have resulted in the identification of mission compromising single point failures and aided in the determination of redundancy requirements and/or design changes.

Design Reviews

A systematic design review of every part, component, subsystem, and system has been performed using comprehensive check lists, failure effects analysis, criticality ratings, and reliability predictions. These tools have enabled the designer to review the design approach for problems not uncovered in previous analyses. In the R & QA area the preliminary design review (PDR) and critical design review (CDR) required by the Apollo Program Directive No. 6 represents specialized application of this discipline.

VEHICLE DEVELOPMENT FLOW

Principal milestones in the hardware and mission phases of the Apollo program are shown in figure 1-1.

Certification and Review Schedules

Certificates of Flight Worthiness (COFW) function as a certification and review instrument. A COFW is generated for each major piece of flight hardware. The certificate originates at the manufacturing facility and is shipped with the hardware wherever it goes to provide a time phased historical record of the items test results, modifications, failures, and repairs.

The program managers pre-flight review (PMPFR) and the program directors flight readiness review (PDFRR) provides a final assessment of launch vehicle, spacecraft, and launch facility readiness at the launch site. During the final reviews the decision is made as to when deployment of the world wide mission support forces should begin.

TRANSPORTATION

The Saturn stage transportation system provides reliable and economical transportation for stages and special payloads between manufacturing and test areas and KSC. The various modes of transportation encompass land, water, and air routes.

Each stage in the Saturn V system requires a specially designed transporter for accomplishing short distance land moves at manufacturing, test, and launch facilities. These transporters have been designed to be compatible with manufacturing areas, dock facility roll-on/roll-off requirements and to satisfy stage protection requirements.

Long distance water transportation for the Saturn V stages is by converted Navy barges and landing ship dock type ocean vessels. Stage restraint during transit is provided by tie-down

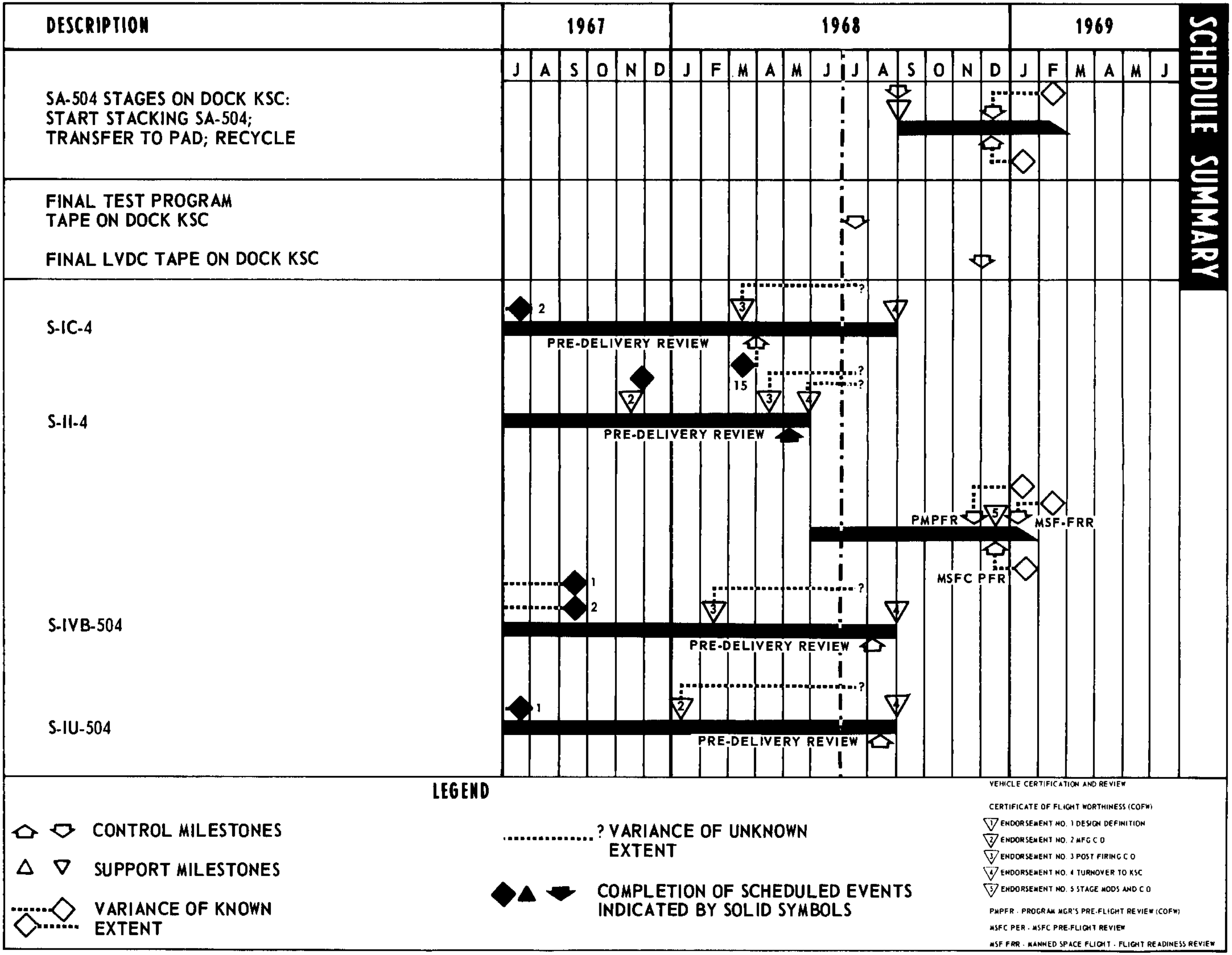

SCHEDULE SUMMARY
DESCRIPTION
1967
1968
1969
J A S O N D J F M A M J J A S O N D J F M A M J
SA-504 STAGES ON DOCK KSC:
START STACKING SA-504;
TRANSFER TO PAD; RECYCLE
FINAL TEST PROGRAM
TAPE ON DOCK KSC
FINAL LVDC TAPE ON DOCK KSC
S-IC-4
PRE-DELIVERY REVIEW
S-II-4
PRE-DELIVERY REVIEW
PMPFR
MSF-FRR
MSFC PFR
S-IVB-504
PRE-DELIVERY REVIEW
S-IU-504
PRE-DELIVERY REVIEW
LEGEND
CONTROL MILESTONES
SUPPORT MILESTONES
VARIANCE OF KNOWN EXTENT
? VARIANCE OF UNKNOWN EXTENT
COMPLETION OF SCHEDULED EVENTS INDICATED BY SOLID SYMBOLS
VEHICLE CERTIFICATION AND REVIEW
CERTIFICATE OF FLIGHT WORTHINESS (COFW)
ENDORSEMENT NO. 1 DESIGN DEFINITION
ENDORSEMENT NO. 2 MFG C O
ENDORSEMENT NO. 3 POST FIRING C O
ENDORSEMENT NO. 4 TURNOVER TO KSC
ENDORSEMENT NO. 5 STAGE MODS AND C O
PMPFR - PROGRAM MGR'S PRE-FLIGHT REVIEW (COFW)
MSFC PER - MSFC PRE-FLIGHT REVIEW
MSF FRR - MANNED SPACE FLIGHT - FLIGHT READINESS REVIEW

Figure 1-1

systems. Ocean vessels have the capability of ballasting to mate with barges and dock facilities for roll-on/roll-off loading. Docks are located at MSFC, KSC, Michoud, MTF, and Seal Beach, California (near Los Angeles).

Air transportation is effected by using a modified Boeing B-377 (Super Guppy) aircraft. This system provides quick reaction time for suitable cargo requiring transcontinental shipments. For ease in loading and unloading the aircraft, compatible ground support lift trailers are utilized.

A Saturn transportation summary is presented in figure 1-2.

LAUNCH VEHICLE DESCRIPTION

GENERAL ARRANGEMENT

The Saturn V Apollo general configuration is illustrated in figure 1-3. Also included are tables of engine data, gross vehicle dimensions, ullage and retrorocket data, and stage contractors.

INTERSTAGE DATA FLOW

In order for the Saturn V Launch Vehicles and Apollo Spacecraft to accomplish their objectives a continuous flow of data is necessary throughout the vehicle. Data flow is in both directions, from spacecraft to stages and from stages to the spacecraft. The IU serves as a central data processor and nearly all data flows through the IU.

Specific data has been categorized and tabulated to reflect, in figure 1-4, the type of data generated, its source and flow. Each stage interface also includes a confidence loop, wired in series through interstage electrical connectors, which assures the LVDC in the IU that these connectors are mated satisfactorily.

RANGE SAFETY AND INSTRUMENTATION

GENERAL

In view of the hazards inherent in missile/space vehicle programs certain stringent safety requirements have been established for the Air Force Eastern Test Range (AFETR). Figure 1-5

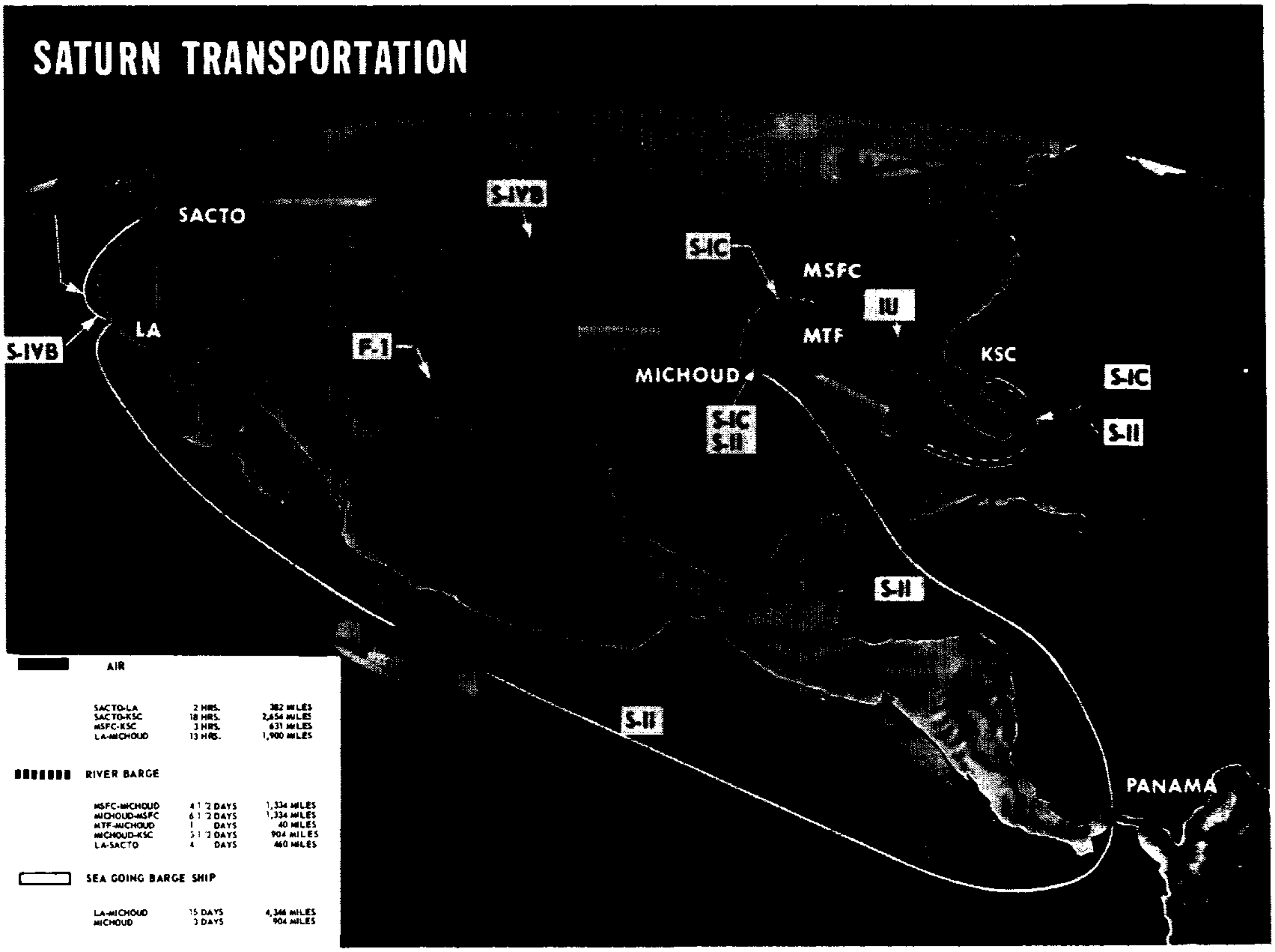


Figure 1-2

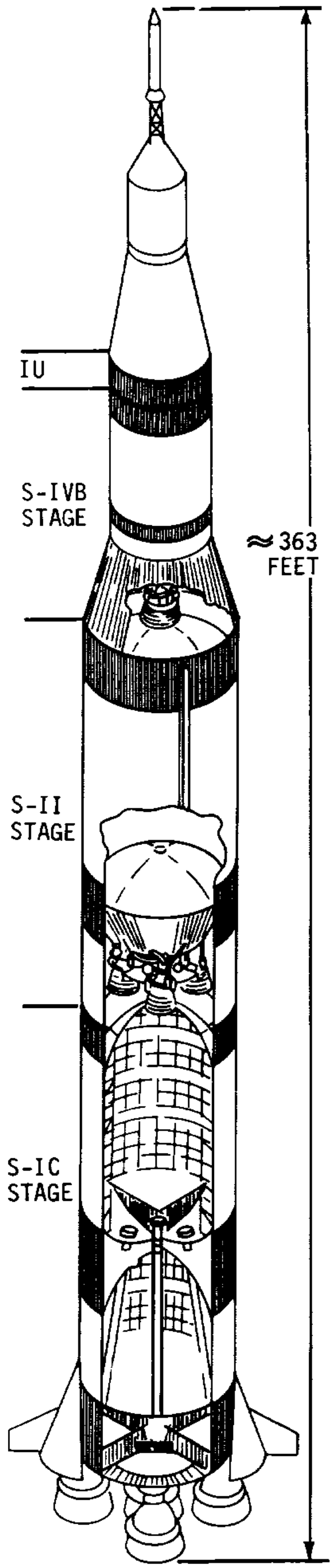
SATURN V LAUNCH VEHICLE

SOLID ULLAGE ROCKET AND RETROROCKET SUMMARY				
STAGE	TYPE	QUANTITY	NOMINAL THRUST AND DURATION	PROPELLANT GRAIN WEIGHT
S-IC	RETROROCKET	8	87,913 POUNDS 0.633 SECONDS	278.0 POUNDS
S-II	ULLAGE	8	22,700 POUNDS 3.71 SECONDS	336.4 POUNDS
	RETROROCKET	4	34,810 POUNDS 1.52 SECONDS	268.2 POUNDS
S-IVB	ULLAGE	2	3,390 POUNDS 3.87 SECONDS	58.8 POUNDS

	ENGINE DATA				
	STAGE	QTY	ENGINE MODEL	THRUST AT IGNITION EACH	THRUST AT IGNITION TOTAL
1	S-IC	5	F-1	1,526,500	7,632,500
2	S-II	5	J-2	230,000	1,150,000
	S-IVB	1	J-2	232,000	232,000

STAGE DIMENSIONS		
	DIAMETER	LENGTH
S-IC (Base including fins)	63.0 FEET	138 FEET
Mid-stage	33.0 FEET	
S-II Stage	33.0 FEET	81.5 FEET
S-IVB Stage	21.7 FEET	59.3 FEET
Instrument Unit	21.7 FEET	3.0 FEET

SATURN V STAGE MANUFACTURERS	
STAGE	MANUFACTURER
S-IC	THE BOEING COMPANY
S-II	NORTH AMERICAN-ROCKWELL
S-IVB	McDONNELL - DOUGLAS CORP.
S-IU	INTERNATIONAL BUSINESS MACHINE CORP.



3 PRE-LAUNCH LAUNCH VEHICLE GROSS WEIGHT
≈6,370,000 POUNDS

Figure 1-3

illustrates the launch azimuth limits and destruct azimuth limits for the Atlantic Missile Range (AMR).

Prime responsibility and authority for overall range safety is vested in the Commander, AFETR, Patrick AFB, Florida. However, under a joint agreement between DOD and NASA, ground safety within the confines of the Kennedy Space Center will by managed by NASA.

To minimize the inherent hazards of the Saturn/Apollo program a number of safety plans have been developed and implemented in accordance with AFETR regulations.

These plans cover all phases of the Saturn Apollo program from design through launch of the vehicle into orbit.

To enhance the development and implementation of the range safety program two general categories have been established: ground safety and flight safety.

GROUND SAFETY

The ground safety program includes a ground safety plan which calls for the development of safety packages. The major categories covered by these packages are:

1. Vehicle Destruct System. This package includes a system description, circuit descriptions, schematics, ordnance system description, specifications, RF system description, installation, and checkout procedures.
2. Ordnance Devices. This package includes descriptive information on chemical composition and characteristics, mechanical and electrical specifications and drawings, and electrical bridgewire data.
3. Propellants. This package includes descriptive data on chemical composition, quantities of each type, locations in the vehicle, handling procedures, and hazards.
4. High Pressure Systems. This package includes types of gases, vehicle storage locations, pressures, and hazards.
5. Special precautionary procedures. This package covers possible unsafe conditions and includes lightning safeguards, use of complex test equipment, and radiological testing.

Also included under ground safety are provisions for launch area surveillance during launch activities. Surveillance methods include helicopters, search radars, and range security personnel. Automatic plotting boards keep the range safety officer (RSO) informed of any intrusion into the launch danger zones by boats or aircraft.

To further assist the RSO in monitoring launch safety a considerable amount of ground instrumentation is used. A vertical wire sky screen provides a visual reference used during the initial phase of the launch to monitor vehicle attitude and position. Television systems photographing the launch vehicle from different angles also provide visual reference. Pulsed and CW tracking radars and real time telemetry data provide an electronic sky screen which displays on automatic plotting boards and charts the critical flight trajectory parameters.

In the event that the launch vehicle deviates from its planned trajectory to the degree that it will endanger life or property the RSO must command destruct by means of the range safety command system. The range safety system is active until the vehicle has achieved earth orbit, after which the destruct system is deactivated (safed) by command from the ground.

FLIGHT SAFETY

Flight safety planning began during the conceptual phases of the program. One of the requirements of the range safety program is that during these early phases basic flight plans be outlined and discussed and prior to launch a final flight plan be submitted and approved. As the program develops the flight planning is modified to meet mission requirements. The flight plan is finalized as soon as mission requirements become firm.

In addition to the normal trajectory data given in the flight plan other trajectory data is required by the AFETR. This data defines the limits of normality; maximum turning capability of the vehicle velocity vector; instant impact point data; drag data for expended stages and for pieces resulting from destruct action; location and dispersion characteristics of impacting stages.

In the event the RSO is required to command destruct the launch vehicle he will do so by manually initiating two separate command messages. These messages are transmitted to the launch vehicle over a UHF radio link. The first message shuts off propellant flow and results in all engines off. As the loss of thrust is monitored by the EDS the ABORT light is turned on in the CM. Upon monitoring a second abort cue the flight crew will initiate the abort sequence. The second command from the RSO is for propellant dispersion and explosively opens all propellant tanks.

Each powered stage of the launch vehicle is equipped with dual redundant command destruct antenna, receivers, decoders, and ordnance to ensure positive reaction to the destruct commands.

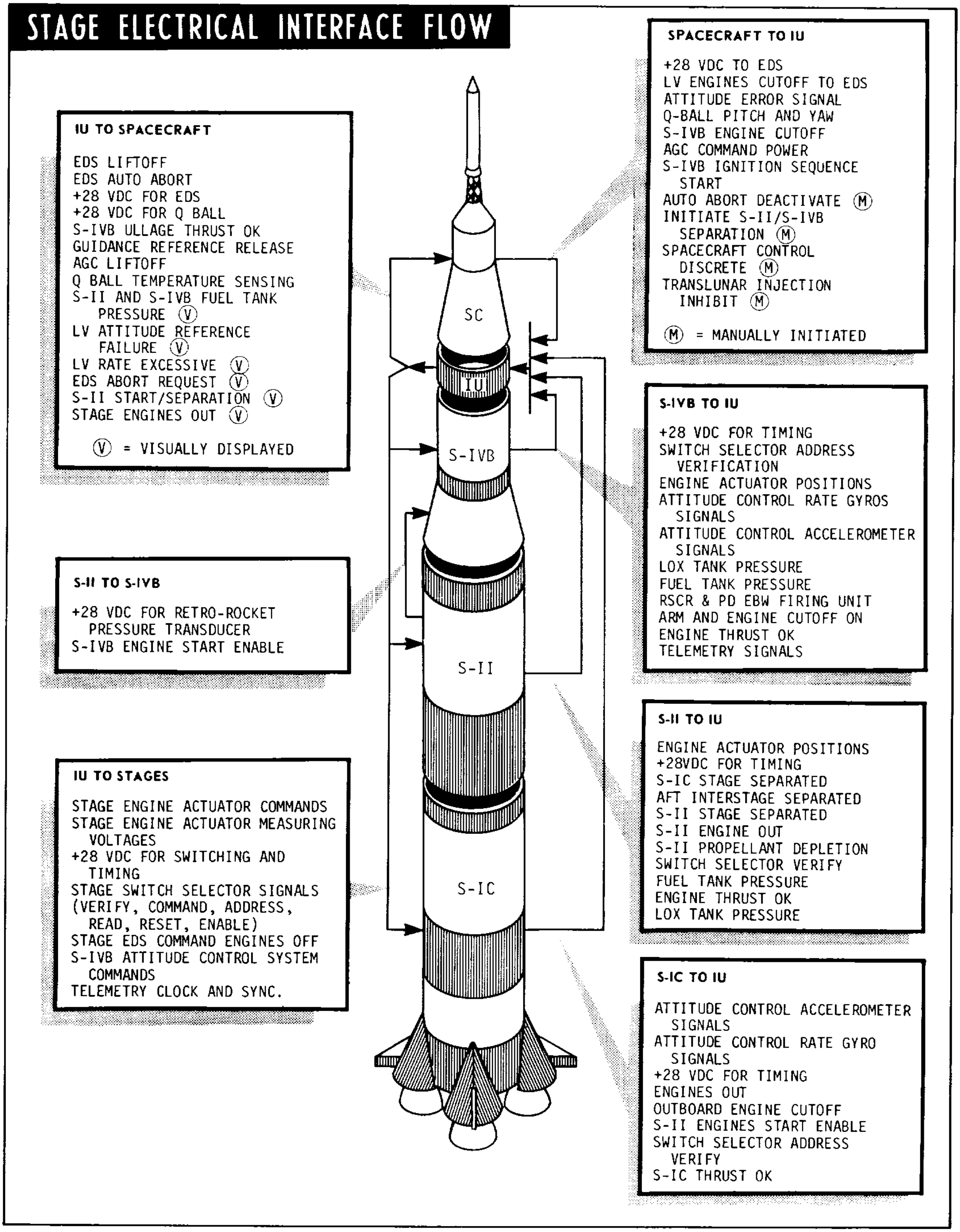

STAGE ELECTRICAL INTERFACE FLOW
IU TO SPACECRAFT
EDS LIFTOFF
EDS AUTO ABORT
+28 VDC FOR EDS
+28 VDC FOR Q BALL
S-IVB ULLAGE THRUST OK
GUIDANCE REFERENCE RELEASE
AGC LIFTOFF
Q BALL TEMPERATURE SENSING
S-II AND S-IVB FUEL TANK PRESSURE (V)
LV ATTITUDE REFERENCE FAILURE (V)
LV RATE EXCESSIVE (V)
EDS ABORT REQUEST (V)
S-II START/SEPARATION (V)
STAGE ENGINES OUT (V)
(V) = VISUALLY DISPLAYED
S-II TO S-IVB
+28 VDC FOR RETRO-ROCKET PRESSURE TRANSDUCER
S-IVB ENGINE START ENABLE
IU TO STAGES
STAGE ENGINE ACTUATOR COMMANDS
STAGE ENGINE ACTUATOR MEASURING VOLTAGES
+28 VDC FOR SWITCHING AND TIMING
STAGE SWITCH SELECTOR SIGNALS (VERIFY, COMMAND, ADDRESS, READ, RESET, ENABLE)
STAGE EDS COMMAND ENGINES OFF
S-IVB ATTITUDE CONTROL SYSTEM COMMANDS
TELEMETRY CLOCK AND SYNC.
SC
IU
S-IVB
S-II
S-IC
SPACECRAFT TO IU
+28 VDC TO EDS
LV ENGINES CUTOFF TO EDS
ATTITUDE ERROR SIGNAL
Q-BALL PITCH AND YAW
S-IVB ENGINE CUTOFF
AGC COMMAND POWER
S-IVB IGNITION SEQUENCE START
AUTO ABORT DEACTIVATE (M)
INITIATE S-II/S-IVB SEPARATION (M)
SPACECRAFT CONTROL DISCRETE (M)
TRANSLUNAR INJECTION INHIBIT (M)
(M) = MANUALLY INITIATED
S-IVB TO IU
+28 VDC FOR TIMING
SWITCH SELECTOR ADDRESS VERIFICATION
ENGINE ACTUATOR POSITIONS
ATTITUDE CONTROL RATE GYROS SIGNALS
ATTITUDE CONTROL ACCELEROMETER SIGNALS
LOX TANK PRESSURE
FUEL TANK PRESSURE
RSCR & PD EBW FIRING UNIT ARM AND ENGINE CUTOFF ON
ENGINE THRUST OK
TELEMETRY SIGNALS
S-II TO IU
ENGINE ACTUATOR POSITIONS
+28VDC FOR TIMING
S-IC STAGE SEPARATED
AFT INTERSTAGE SEPARATED
S-II STAGE SEPARATED
S-II ENGINE OUT
S-II PROPELLANT DEPLETION
SWITCH SELECTOR VERIFY
FUEL TANK PRESSURE
ENGINE THRUST OK
LOX TANK PRESSURE
S-IC TO IU
ATTITUDE CONTROL ACCELEROMETER SIGNALS
ATTITUDE CONTROL RATE GYRO SIGNALS
+28 VDC FOR TIMING
ENGINES OUT
OUTBOARD ENGINE CUTOFF
S-II ENGINES START ENABLE
SWITCH SELECTOR ADDRESS VERIFY
S-IC THRUST OK

Figure 1-4

RANGE SAFETY AZIMUTH LIMITS

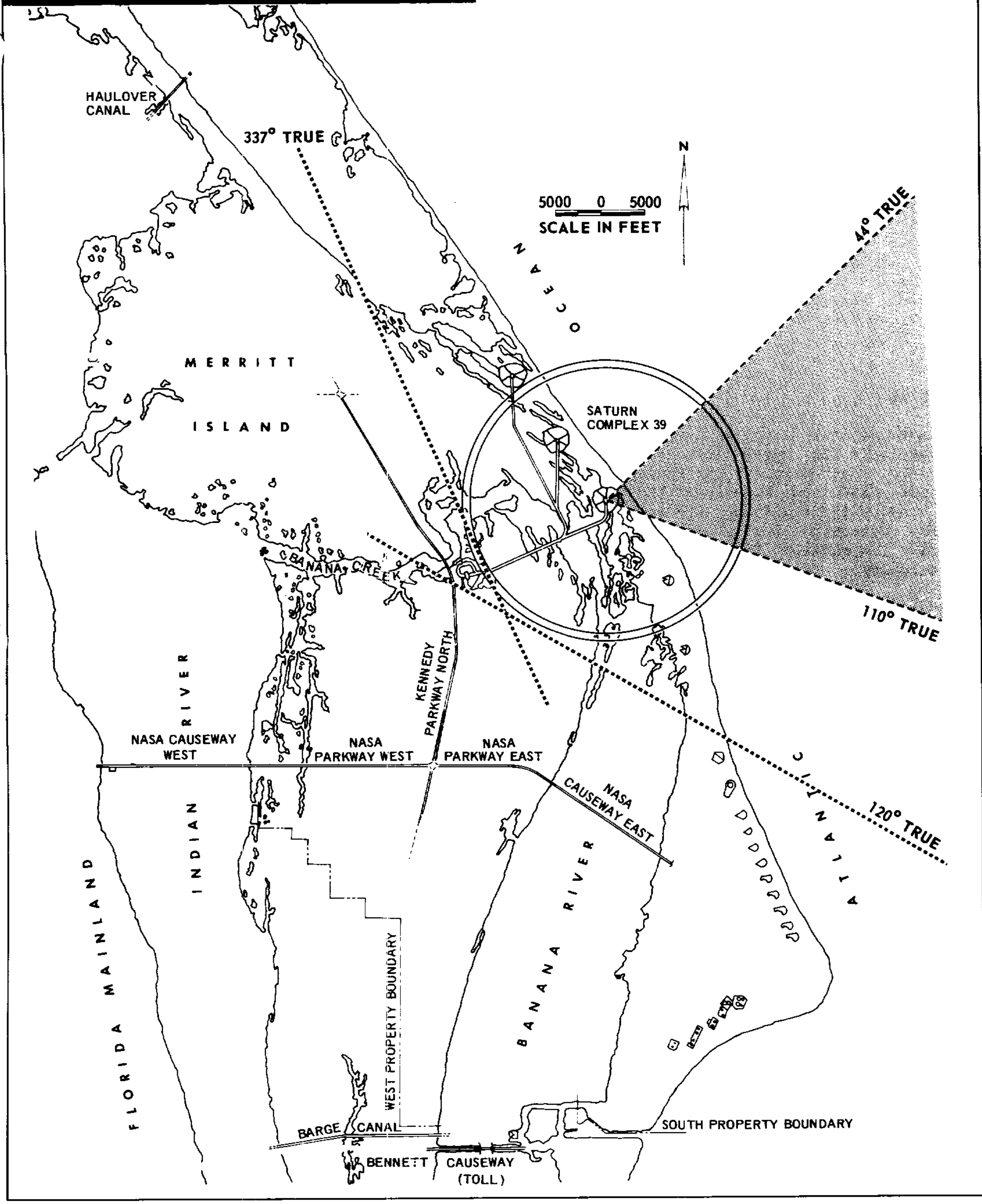


Figure 1-5

To augment flight crew safety an Emergency Detection System (EDS) monitors critical flight parameters. Section III provides a more detailed discussion of the EDS.

OPERATIONAL SEQUENCES

The majority of operational sequences on the Saturn V launch vehicle are automatic, but a few may be monitored and/or controlled from the Command Module (CM). The operational sequences are controlled from or observed in the CM through the controls and indicators shown at the top of figure 1-6. The launch vehicle components to which these controls and indicators relate are shown in figure 1-6 arranged below the CM equipment in the same order as the launch vehicle is stacked. Refer to figure 3-1 to locate those components to be found on the CM's main display console (MDC).

MONITORING

Displays to monitor the launch vehicle (LV) are illustrated in the upper left hand portion of figure 1-6. Each display, its sensors and controls are discussed briefly in the following text. Many of these displays are related to the Emergency Detection System and are discussed more fully in Section III, Emergency Detection and Procedures.

LV Engine Status Display

The LV engine status display consists of five lights numbered sequentially from 1 to 5. Each of these lights represent a specific launch vehicle engine on the operating stage. The lights are controlled by thrust OK pressure switches contained in each of the three stages. The controlling set of pressure switches is selected (figure 1-7) by the program in the LVDC.

In the S-IC there are three switches per engine operating on two out of three logic. In the other two stages there are two thrust OK pressure switches per engine. These, unlike the switches on the S-IC, indicate thrust is OK when either of the two switches is actuated. The thrust pressure level to actuate the pressure switches is 90 percent of maximum in the S-IC and 65 percent of maximum in the S-II and S-IVB stages. This lower actuation level on the S-II and S-IVB stages is necessitated by the variable thrust pressure caused by the changing mixture ratio of the J-2 engine. These lights also indicate when stage separation takes place. Separation is indicated by the lights going out after being illuminated by stage engine cutoff.

LV RATE Light

The LV RATE light (figure 1-6) is illuminated any time the LV experiences an excessive pitch, yaw or roll rate. It remains energized as long as the excessive rate condition exists. Those excessive rates are sensed by rate gyros in the Instrument Unit. (See Section III).

S-II SEP Light

The S-II SEP light (figure 1-6) indicates both S-II second plane separation and beginning of S-IVB restart preparations. (See Section III).

LV GUID Light

This light indicates when the LV attitude reference (ST-124M-3 stable platform) fails. It will remain illuminated as long as the failed condition exists. (See Section III.)

LV TANK PRESS Gauges

The four LV TANK PRESS gauges indicate ullage pressure in the S-II and S-IVB fuel tanks and S-IVB oxidizer tank. Prior to S-II/S-IVB separation the two left hand pointers (dual redundancy) indicate the pressure in the S-II fuel tank. These same two left hand pointers indicate S-IVB oxidizer pressure after S-II/S-IVB separation. The two right hand pointers indicate S-IVB fuel tank pressure until spacecraft/launch vehicle separation.

The LV tank ullage pressure is sensed by pressure transducers on each of the three tanks. Switching of the reading from the S-II fuel tank to the S-IVB oxidizer tank is controlled and accomplished by relay switchover logic in the IU.

Abort Lights

The abort light may be manually illuminated as stated in Section III or automatically by a launch vehicle initiated automatic abort signal (S-IC two engine out, range safety command or excessive launch vehicle rates automatic abort signal). The S-IC two engine out automatic abort is inhibited from liftoff until T_1 + 14.0 seconds.

CONTROL OF THE LAUNCH VEHICLE

Controls used in controlling the launch vehicle consist of switches, the T-handle on the translation controller, and the hand grip on the pilots hand controller. All of these controls are illustrated in the upper right hand portion of figure 1-6. Refer to figure 3-1 for switch location on the CM's main display console. Each control is discussed separately in the following paragraphs. Many of these controls are related to the Emergency Detection System and are discussed more completely in Section III Emergency Detection and Procedures.

Translational Controller

The operation of the translational controller is explained in Section III.

GUIDANCE Switch

The normal position of the GUIDANCE switch is IU. In this position guidance of the launch vehicle is controlled by the program in the LVDC and the flight control computer. (See Section VII for a description of the launch vehicle guidance and navigation system). When the GUIDANCE switch is placed in the CMC position launch vehicle guidance is transferred from the launch vehicle to the spacecraft. This is accomplished by blocking the launch vehicle flight control signal path and providing flight control commands from the command module computer (CMC). These commands from the CMC may be modified by hand controller operation.

NOTE

> The CMC position will be used only during orbital flight for control of the S-IVB with the hand controller.

S-II/S-IVB-LV STAGE Switch

See Section III for a description of this switch.

XLUNAR INJECT Inhibit Switch

When the XLUNAR INJECT inhibit switch is placed in the SAFE position, a signal to the LVDC in the IU inhibits the restart of the S-IVB engine. When placed in the INJECT position this switch allows the automatic sequence to direct injection of the spacecraft into the waiting orbit.

UP TLM-IU Switch

The UP TLM-IU switch normally allows the LVDC to be updated via the update link. Moving this switch to its BLOCK position prevents updating the LVDC from the updata link. See Section VII for additional information concerning the command communications system.

ABORT SYSTEM-2 ENG OUT Switch

The OFF position of the ABORT SYSTEM-LV RATES switch disables the LV excessive rate automatic abort circuitry. Positioning this switch to the AUTO position enables the LV excessive rate automatic abort circuitry.

ABORT SYSTEM-LV RATES Switch

The ABORT SYSTEM - 2 ENG OUT switch disables the S-IC two engine out automatic abort circuitry when in the OFF position. When this switch is in the AUTO position the S-IC two engine out automatic abort circuitry is activated.

LIFT OFF/NO AUTO ABORT Lights

See Section III for a description of these lights.

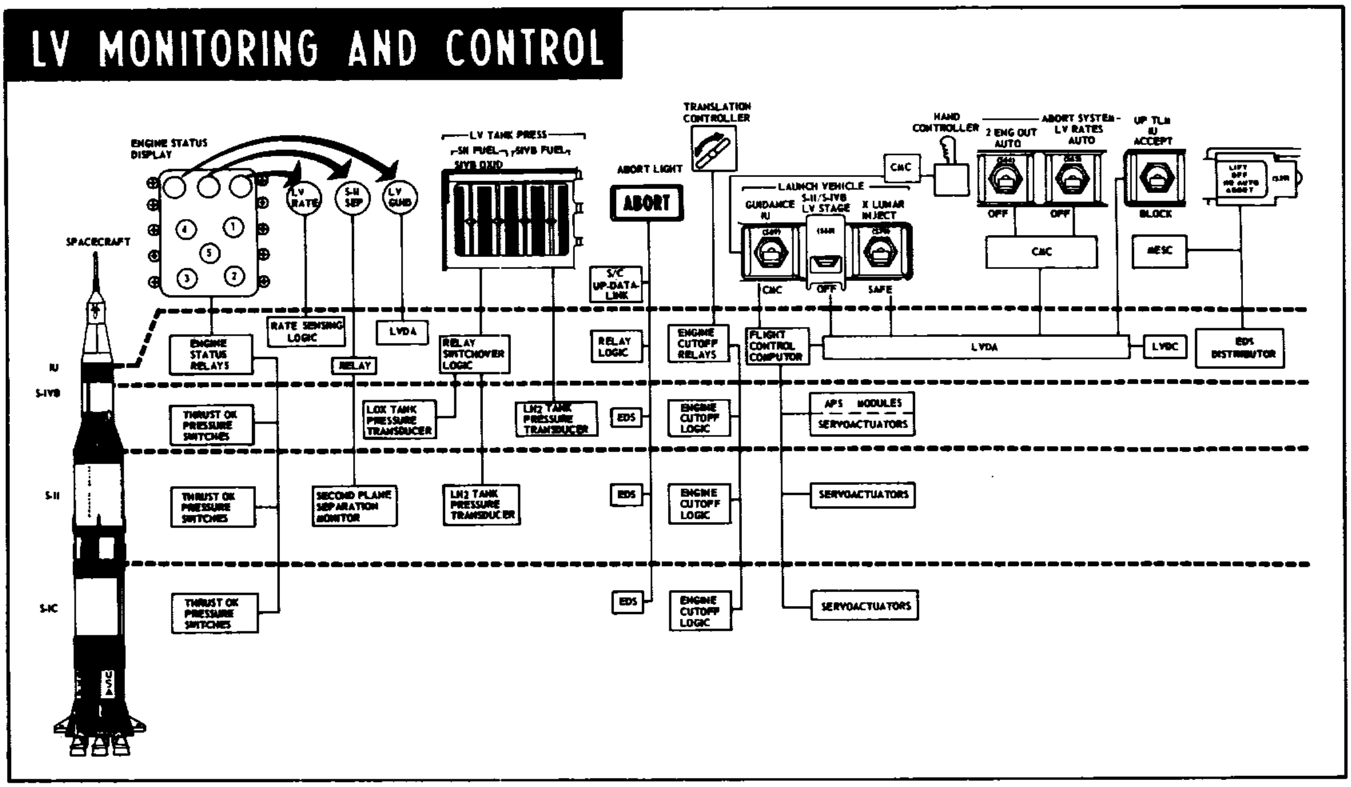


Figure 1-6

NORMAL LV ENGINE STATUS DISPLAY LIGHTS OPERATION

DISPLAY LIGHT	STAGE	ENABLED BY LVDC (LIGHT ON)	ENGINE START (LIGHT OFF)	ENGINE CUTOFF (LIGHT ON)	DISABLED BY LVDC (LIGHT OFF)	STAGE SEPARATION (LIGHT OFF)
NO. 1	S-IC	T_1 - 00:08.0	T_1 - 00:01.3	T_3 + 00:00.0 (02:38.4)	N/A	S-IC/S-II T_3 + 00:00.7 (02:39.1)
	S-II	T_3 + 00:01.5 (02:39.9)	T_3 + 00:05.2 (02:43.4)	T_4 + 00:00.0 (09:02.2)	N/A	S-II/S-IVB T_4 00:00.8 (09:03.0)
	1st Burn S-IVB	T_4 + 00:01.8 (09:04.0)	T_4 + 00:06.2 (09:08:4)	T_5 + 00:00.0 (11:33.6)	T_5 + 00:10.0 (21:33.6)	N/A
	2nd Burn S-IVB	T_6 + 07:37.4 (TBD)	T_6 + 07:40.2 (TBD)	T_7 + 00:00.0 (TBD)	TURNED OFF BY SPACECRAFT SEPARATION	N/A
NO. 2	S-IC	*	*	*	N/A	*
	S-II	*	*	*	N/A	*
NO. 3	S-IC	*	*	*	N/A	*
	S-II	*	*	*	N/A	*
NO. 4	S-IC	*	*	*	N/A	*
	S-II	*	*	*	N/A	*
NO. 5	S-IC	*	*	T_2 + 00:00.0 (02:26.2)	N/A	*
	S-II	*	*	*	N/A	*

* SAME AS LIGHT NO. 1 FOR SAME STAGES.

NOTE: T-TIMES (TIME BASE TIMES) ARE SHOWN IN MINUTES AND SECONDS, TO THE NEAREST TENTH OF A SECOND.

NOMINAL FLIGHT TIMES ARE SHOWN IN BRACKETS IN MINUTES AND SECONDS, TO THE NEAREST TENTH OF A SECOND.

Figure 1-7

PERCEPTIBLE PRELAUNCH EVENTS

Prelaunch events which occur subsequent to astronaut loading (T-3 hours 40 minutes), and which may be felt or heard by the flight crew inside the spacecraft, are identified in figure 1-8. Other events, not shown, combine to create a relatively low and constant background. This background noise includes the sounds of environmental control, propellant replenishment, control pressure gas supplies, propellant boiloff and low pressure, low volume purges.

Significant noises and vibrations may be caused by the starting or stopping of an operation or they may result from turbulent flow of gases or liquids. Figure 1-8 illustrates those events most likely to be heard or felt above the background noise or vibration. Spacecraft close out (1) includes crew entrance into the spacecraft, closing of the access door, and the exit of ground personnel from the pad area. Significant noise experienced during this period will generally be recognized or expected. One possible exception could be the pressurization of the cold helium spheres (2) in the S-IVB. This will start about 25 minutes after the cabin leak check has been performed.

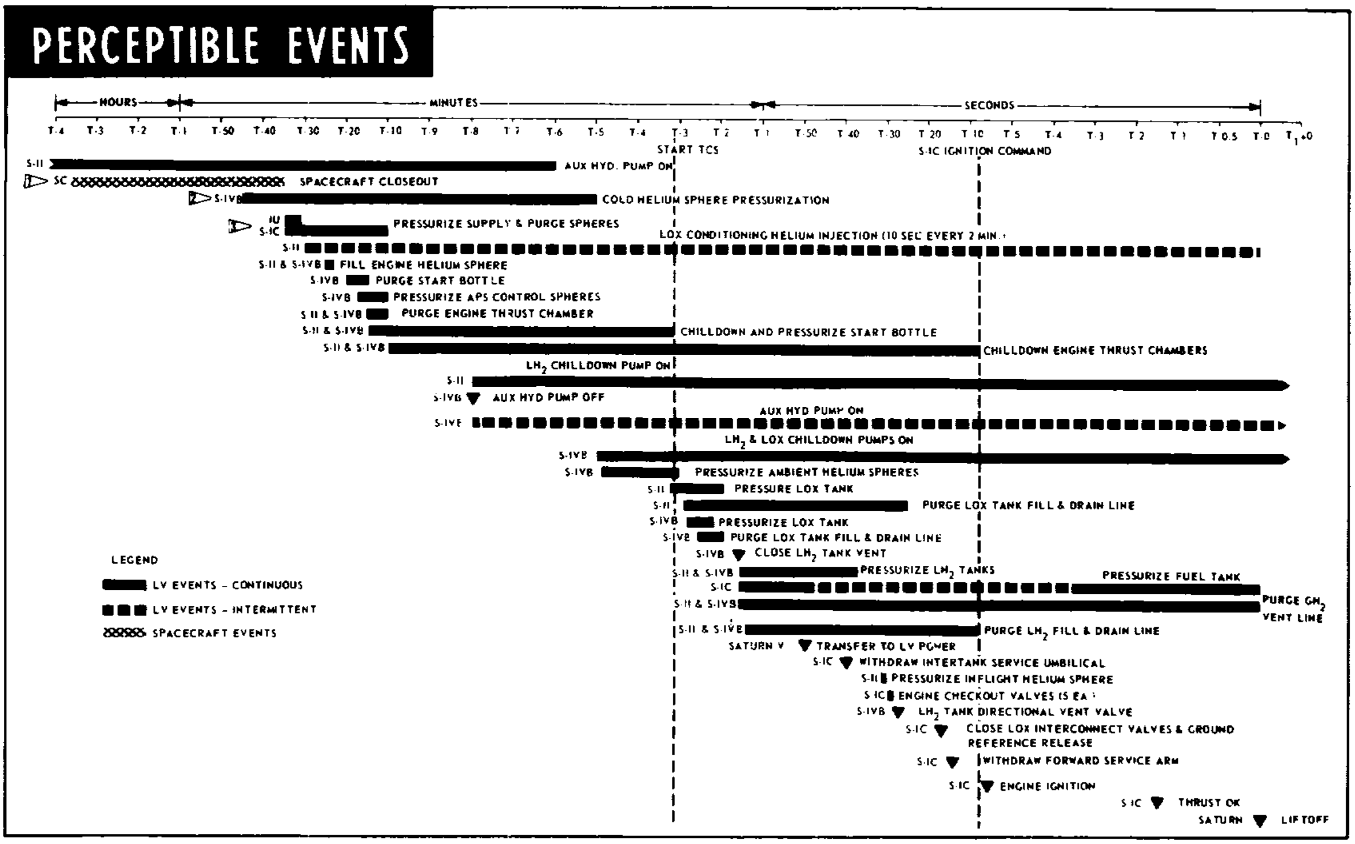


Figure 1-8

Nearly all items shown in figure 1-8 are noise producers and each will have individual characteristics resulting from such things as proximity, volume, and timing. For example pressurizing the supply spheres (3) in the IU will be more noticeable than pressurizing the supply and purge spheres (3) on the S-IC. Yet with both items starting at the same time, each sound will add to the other to make the total sound. When pressurization of the IU supply sphere (3) ends at T-31 minutes, it is likely that the sound of the pressurization of the S-IC supply and purge will blend with the cold helium sphere pressurization (2) sound and be indiscernible as such in the command module. In this manner, the sounds of the remainder of the items illustrated will rise and fall, join and separate, to form the sounds of the Saturn V. At approximately T-61 seconds all sounds are hidden by the ignition of the engines on the S-IC.

POGO

During the flight of AS-502, a closed loop system instability of POGO type resulted from the dynamic coupling of the structure, propellant feed and engine thrust systems. This was indicated by a buildup in acceleration, fluid pressure and thrust oscillation which reached a peak at approximately 125 seconds in the flight. The frequency of this oscillation was approximately 5.3 cycles per second, which is close to the first longitudinal vehicle frequency for this period of the flight. At approximately 130 seconds the system began to decouple and the POGO phenomena ceased.

To eliminate the problem it is planned to use the cavity between the lox flow tube and the lox prevalve housing of each engine as an accumulator. Each of these cavities will be precharged with helium at T-10 minutes from ground supply and be maintained following liftoff from the onboard helium supply. The helium in these accumulators interfaces with the lox in the ducts and acts as a spring thereby reducing the modules of elasticity of the lox liquid columns and lowering their natural frequency.

MAJOR DIFFERENCES BETWEEN SATURN V LAUNCH VEHICLES SA-503 & SA-504

1 *Thrust at ignition for S-IC stage engines increased from 1,522,000 pounds each and 7,610,000 pounds total to 1,526,500 pounds each and 7,632,500 pounds total.*

2 *Thrust at ignition for S-II stage engines increased from 228,000 pounds each and 1,140,000 pounds total to 230,000 pounds each and 1,150,000 pounds total.*

3 *Launch vehicle prelaunch gross weight increased from approximately 6,000,000 pounds to 6,370,000 pounds.*

SECTION II

PERFORMANCE

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INTRODUCTION

Saturn V launch vehicle performance is described in terms of payload, structural, guidance, control, flight and operational capabilities under a set of constraints established by environment and mission requirements. The SA-504 mission capabilities, are described in this section. Mission profile, variables, constraints and requirements are discussed in Section X.

FLIGHT SEQUENCE

LAUNCH AND BOOST TO EARTH PARKING ORBIT

The SA-504 vehicle, will be launched at the Kennedy Space Center from Complex 39. A time correlated sequence of the significant events is listed in figure 2-1 and shown graphically in figure 2-2. These events occur as described in the following paragraphs.

The vehicle rises vertically to an altitude of 450 feet to assure vertical clearance of the launch tower. (A yaw maneuver introduced right after liftoff provides tower clearance under adverse conditions of wind, deviation from nominal flight and/or engine failure during initial liftoff). Launch vehicle guidance then initiates a maneuver (pitch and roll) to the desired flight attitude and azimuth. Launch azimuth is 72 degrees. From the end of the short tilt maneuver for a period of about 2 minutes and 15 seconds, the vehicle flies an approximate zero-lift (gravity-turn) trajectory, achieved by open-loop pitch-attitude commands biased for winds of the launch month. At this point vehicle pitch-attitude commands are arrested (tilt arrest), and the attitude remains constant until after launch escape tower (LET) jettison during S-II stage flight. At about 1 minute and 17 seconds after liftoff the maximum dynamic pressure is encountered. The S-IC center engine is cut off at 2 minutes and 26 seconds by propellant level sensors in the lox and fuel tank standpipes. The S-IC outboard engines are cutoff about 12 seconds after center engine cutoff.

The S-II stage reaches 90 percent thrust about 5 seconds after the signal for the S-IC outboard engine shutdown is received. (Threshold for engine status light OFF is 65 percent thrust.) During this period, S-IC/S-II separation occurs, retrorockets slow the S-IC stage, and ullage motors seat the S-II stage propellant. The S-II aft interstage is jettisoned about 30 seconds after the S-IC stage outboard engines cutoff signal. Five seconds later, the Launch Escape Tower is jettisoned.

During S-II burn, two fuel mixture-ratio (MR) shifts occur. The initial MR is 5.0. The second mixture ratio, 5.5, is initiated 2 seconds after the S-II stage reaches 90 percent thrust. The higher MR produces increased thrust at lowered specific impulse. The shift to a third MR of 4.7 is initiated by a signal from the S-II propellant sensors when the remaining oxidizer-to-fuel-weight ratio is 4.7. The 4.7 mixture ratio yields a reduced thrust, at increased specific impulse. The S-II engines are cut off simultaneously by lox and LH_2 tank sensors. An iterative guidance mode (IGM) controls the vehicle during S-II stage flight following LET jettison. This mode provides guidance commands during vacuum flight.

A 7 second time interval occurs between the S-II stage engine shutdown signal and 90 percent thrust on the S-IVB engine. (Threshold for engine status light OFF is 65 percent thrust.) During the staging coast period, S-II/S-IVB separation occurs. Retrorockets slow the S-II stage, and ullage rockets seat the S-IVB stage propellant. The S-IVB stage first burn then inserts the vehicle into a 100 nautical mile (NMI) altitude circular earth parking orbit.

CIRCULAR EARTH PARKING ORBIT

The iterative guidance mode (IGM) steers the vehicle into a nearly circular earth parking orbit with a mean altitude of approximately 100 NMI at the equator. At insertion, the parking orbit inclination is 32.57 degrees. Ullage, O_2/H_2 burner, and venting thrust, together with mass losses of gaseous hydrogen, helium, oxygen, and Auxiliary Propulstion System (APS) propellant, occur during parking orbit. After two circular orbits the vehicle is boosted into a high apogee elliptical waiting orbit.

WAITING ORBIT INSERTION BOOST

S-IVB stage reignition for the boost to waiting orbit insertion occurs shortly after the start of the third revolution of the parking orbit.

The boost to waiting orbit insertion lasts for 152 seconds, and insertion occurs at an altitude of 111.77 NMI, an inertial velocity of 29,679.1 fps, and an inertial flight-path angle of 1.593

degrees. The vehicle weight at insertion is 207,001 pounds, which includes 79,329 pounds of Flight Performance Reserves, Flight Geometry Reserves, and usable mainstage propellant.

FLIGHT PERFORMANCE

FLIGHT PERFORMANCE PARAMETER

The flight performance parameters for a typical AS-504 flight profile are described graphically in figures 2-3 through 2-17. These parameters are shown for the nominal case and for maximum and minimum 3-sigma dispersions from the nominal for the Launch to Earth Parking Orbit, Earth Parking Orbit and Insertion Into Waiting Orbit phases. The parameters shown are; pitch attitude, weight, axial force, altitude, inertial velocity, acceleration, dynamic pressure, azimuth, latitude, longitude and range.

OFF NOMINAL PERFORMANCE

Off-nominal performance is evaluated in terms of the following basic factors:

1. Flight parameter dispersions during boost.
2. Flight performance reserve propellant based upon a statistical summation of the propellant dispersion values presented in (1) above. (See figure 2-18.)
3. Composite trajectory profiles for maximum dynamic pressure, maximum acceleration and maximum heating conditions.
4. Payload performance.

FLIGHT PARAMETER DISPERSIONS

Typical S-IVB propellant weight dispersions and 3-sigma vehicle systems and environmental perturbation sources are tabulated in figure 2-19. These dispersions are provided at parking orbit insertion and waiting orbit insertion. They are based on nominal parking orbit and waiting orbit insertion weight of 282,588 pounds and 207,001 pounds respectively. Other flight parameter dispersions which occur are shown as maximum and minimum 3-sigma dispersion curves in figure 2-3 through 2-17.

Flight Performance and Flight Geometry Reserves

The launch vehicle nominally delivers 95,000 pounds of payload into the high apogee elliptical waiting orbit. An S-IVB propellant margin of 72,152 pounds remains at thrust termination. This propellant margin is in addition to the 7,177 pounds required for vehicle performance and flight geometry reserves shown in figure 2-18.

NOMINAL SEQUENCE OF EVENTS		
TIME FROM LIFTOFF	TIME FROM REFERENCE (MIN:SEC)	EVENT
0:00.00	t_1 + 0:00	Lift off
1:17	t_1 + 1:17	Maximum Dynamic Pressure
2:26	t_2 + 0:00	S-IC Center-Engine Cutoff
2:38	t_3 + 0:00	S-IC Outboard Engines Cutoff
2:39	t_3 + 0:01	S-IC/S-II Separation
2:41	t_3 + 0:03	S-II J-2 Engine Ignition
2:43	t_3 + 0:05	S-II J-2 Engines at 90 Percent of Operating Thrust
2:45	t_3 + 0:07	First S-II MR shift
3:09	t_3 + 0:31	S-IC/S-II Interstage Drop
3:14	t_3 + 0:36	LET Jettison
7:00	t_3 + 4:22	Second S-II MR shift
9:02	t_4 + 0:00	S-II J-2 Engines Cutoff
9:03	t_4 + 0.01	S-II/S-IVB Separation
9:07	t_4 + 0:05	S-IVB J-2 Engine Ignition
9:09	t_4 + 0:07	S-IVB J-2 Engine at 90 Percent of Operating Thrust
11:34	t_5 + 0	S-IVB J-2 Engine Cutoff-Parking Orbit Insertion-Ullage Thrust Start
12:33	t_5 + 0:59	Continuous Vent Start
13:02	t_5 + 1:28	Ullage Thrust Cutoff
TBD	t_6 + 0:00	Begin Restart Preparations
TBD	t_6 + 1:25	O_2/H_2 Burner Start
TBD	t_6 + 1:26	Continuous Vent Cutoff
TBD	t_6 + 5:46	Ullage Thrust Start
TBD	t_6 + 5:47	O_2/H_2 Burner Cutoff
TBD	t_6 + 7:30	End Parking Orbit-Reignite S-IVB
TBD	t_6 + 7:33	Ullage Thrust Cutoff
TBD	t_7 + 0:00	S-IVB J-2 Engine Cutoff-Waiting Orbit Insertion

Figure 2-1

MAJOR EVENTS - LAUNCH TO WAITING ORBIT INSERTION

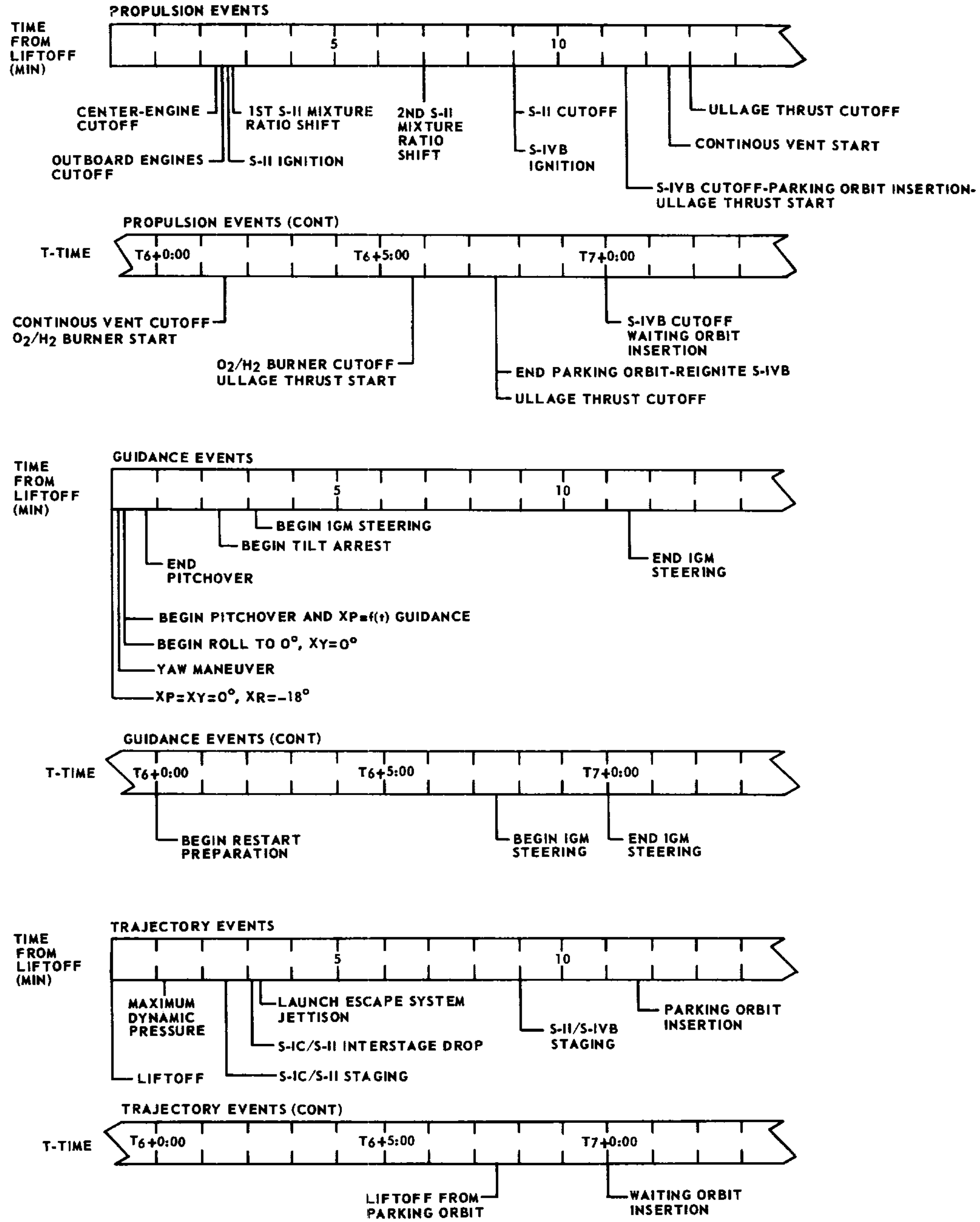
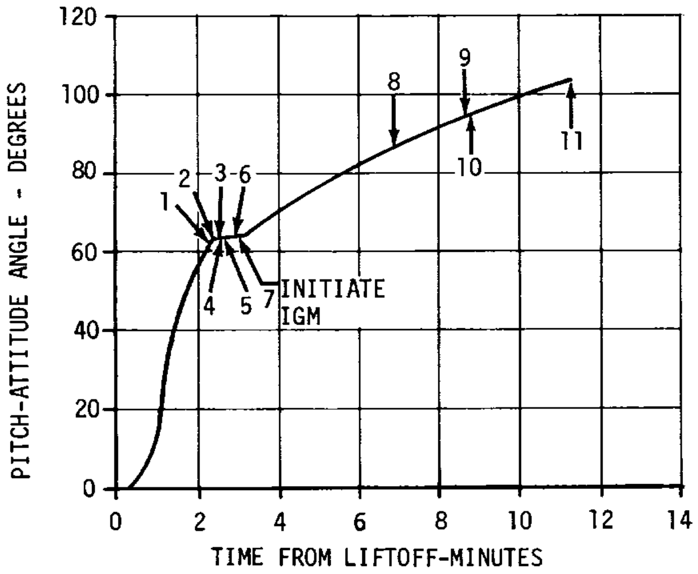


Figure 2-2

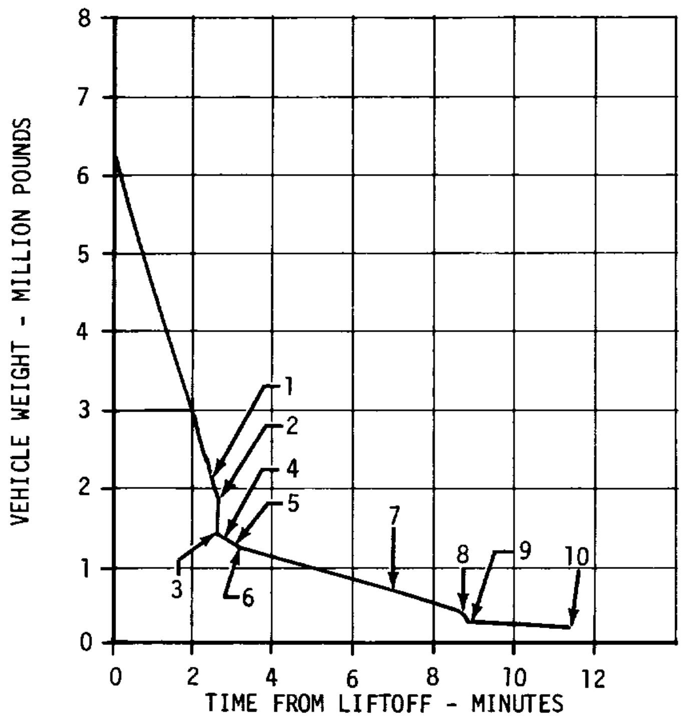
TYPICAL PITCH ATTITUDE ANGLE DURING BOOST TO PARKING ORBIT



EVENTS
1. TILT ARREST
2. CENTER-ENGINE CUTOFF
3. OUTBOARD ENGINES CUTOFF
4. S-II IGNITION
5. FIRST S-II MIXTURE-RATIO SHIFT
6. S-IC/S-II INTERSTAGE DROP
7. LAUNCH ESCAPE SYSTEM JETTISON
8. SECOND S-II MIXTURE-RATIO SHIFT
9. S-II CUTOFF
10. S-IVB IGNITION
11. S-IVB CUTOFF-PARKING ORBIT INSERTION

Figure 2-3

TYPICAL VEHICLE WEIGHT DURING BOOST TO PARKING ORBIT



EVENTS
1. S-IC CENTER-ENGINE CUTOFF
2. S-IC OUTBOARD ENGINES CUTOFF
3. S-II IGNITION
4. FIRST S-II MIXTURE-RATIO SHIFT
5. S-IC/S-II INTERSTAGE DROP
6. LAUNCH ESCAPE SYSTEM JETTISON
7. SECOND S-II MIXTURE-RATIO SHIFT
8. S-II CUTOFF
9. S-IVB IGNITION
10. S-IVB CUTOFF-PARKING ORBIT INSERTION

Figure 2-4

TYPICAL AXIAL FORCE DURING BOOST TO PARKING ORBIT

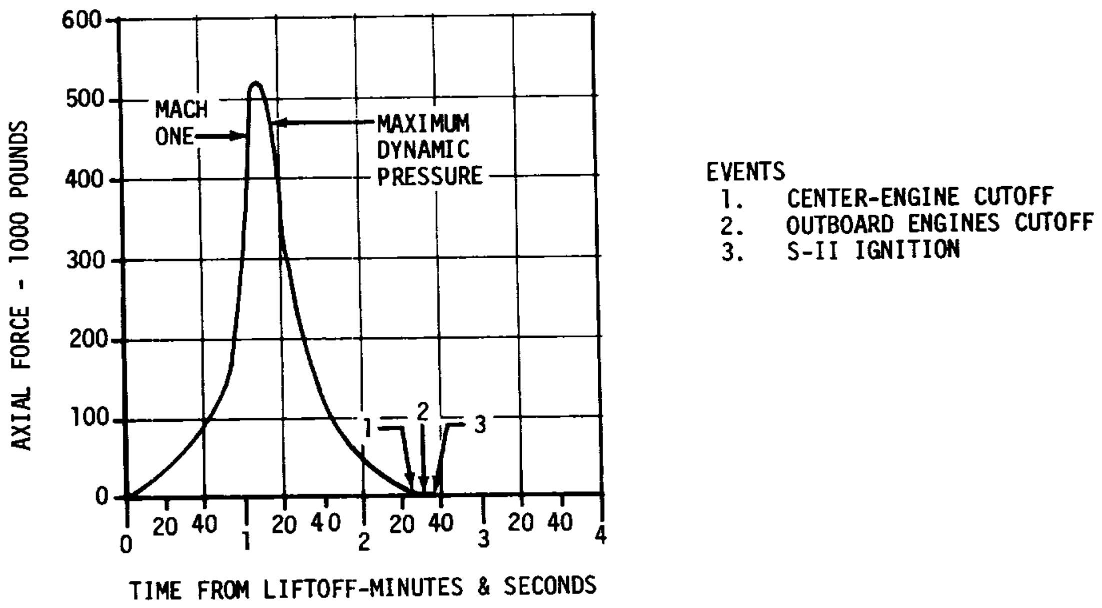


Figure 2-5

ALTITUDE CORRIDOR DURING BOOST TO PARKING ORBIT

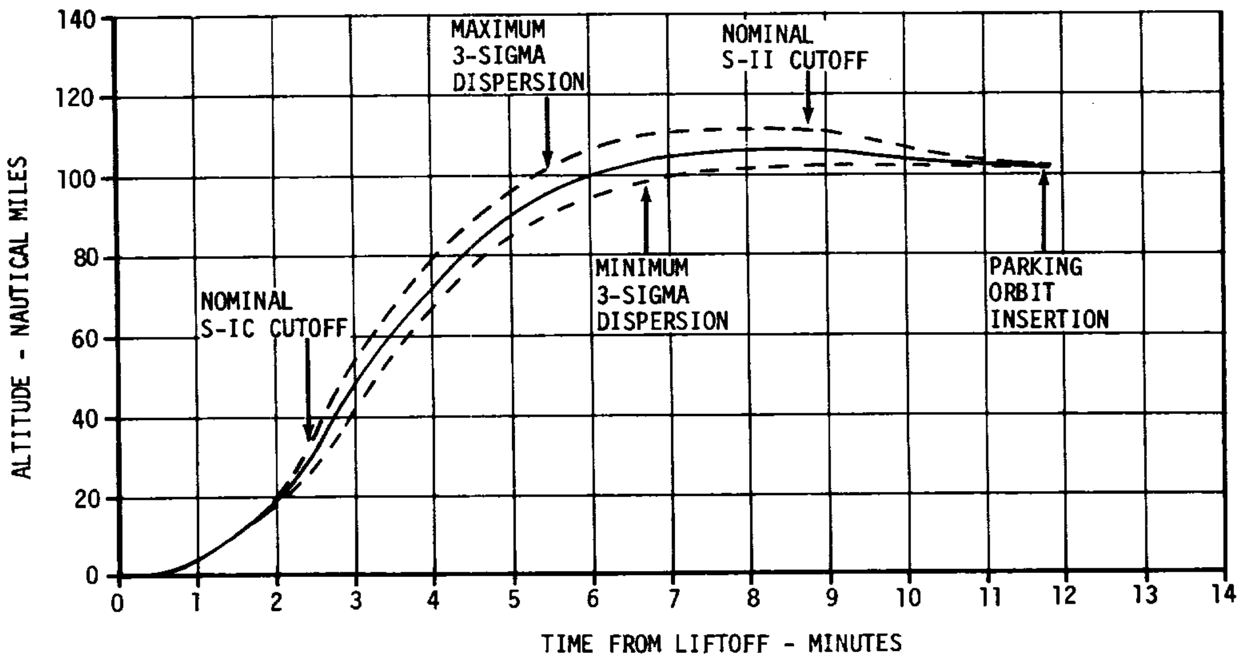


Figure 2-6

TYPICAL INERTIAL VELOCITY CORRIDOR DURING BOOST TO PARKING ORBIT

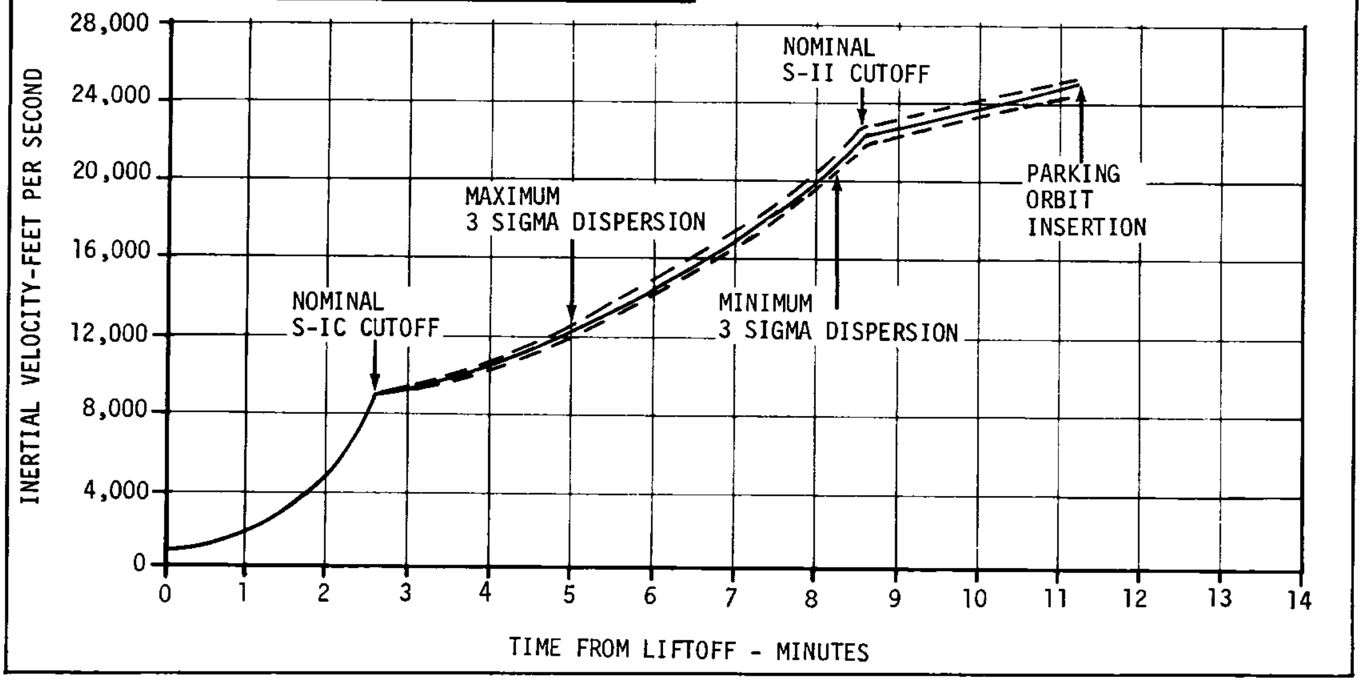


Figure 2-7

TYPICAL LONGITUDINAL ACCELERATION CORRIDOR DURING BOOST TO PARKING ORBIT

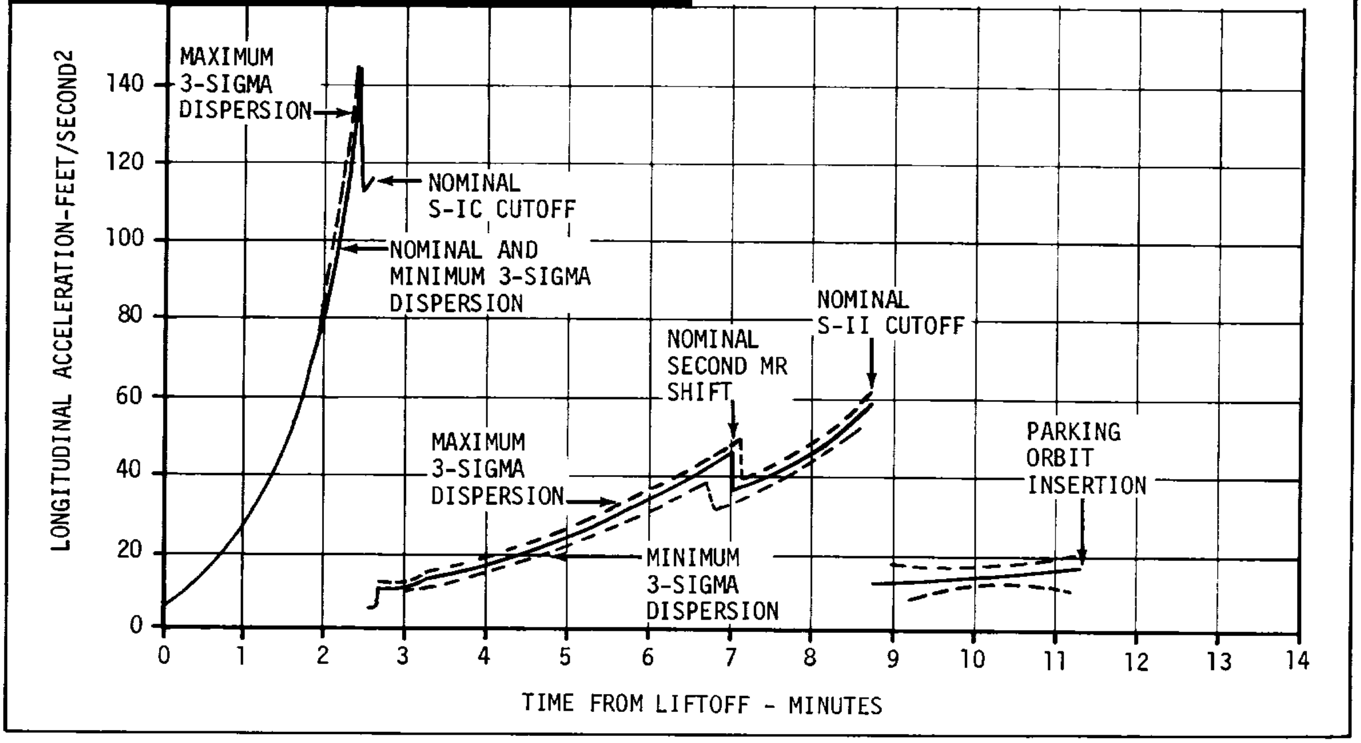


Figure 2-8

TYPICAL DYNAMIC PRESSURE CORRIDOR DURING BOOST TO PARKING ORBIT

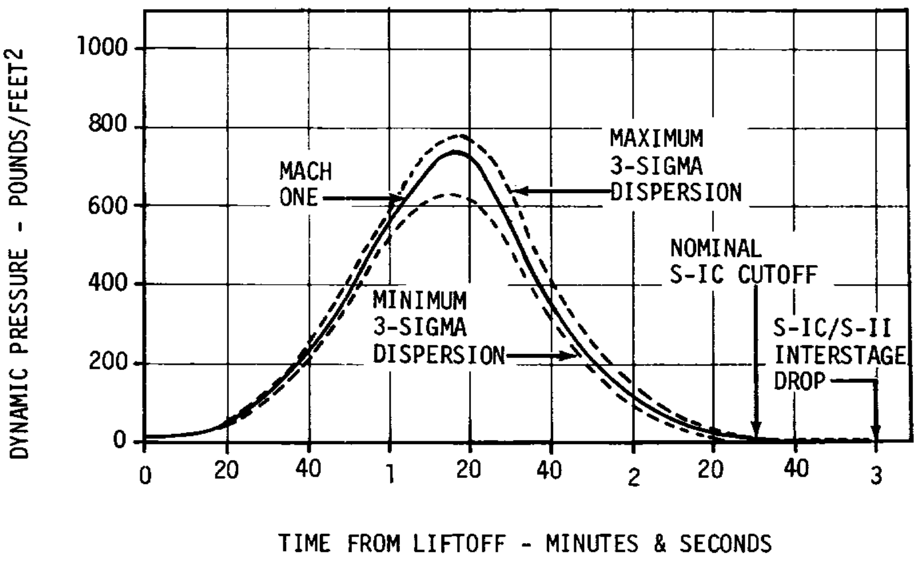


Figure 2-9

TYPICAL INERTIAL AZIMUTH CORRIDOR DURING BOOST TO PARKING ORBIT

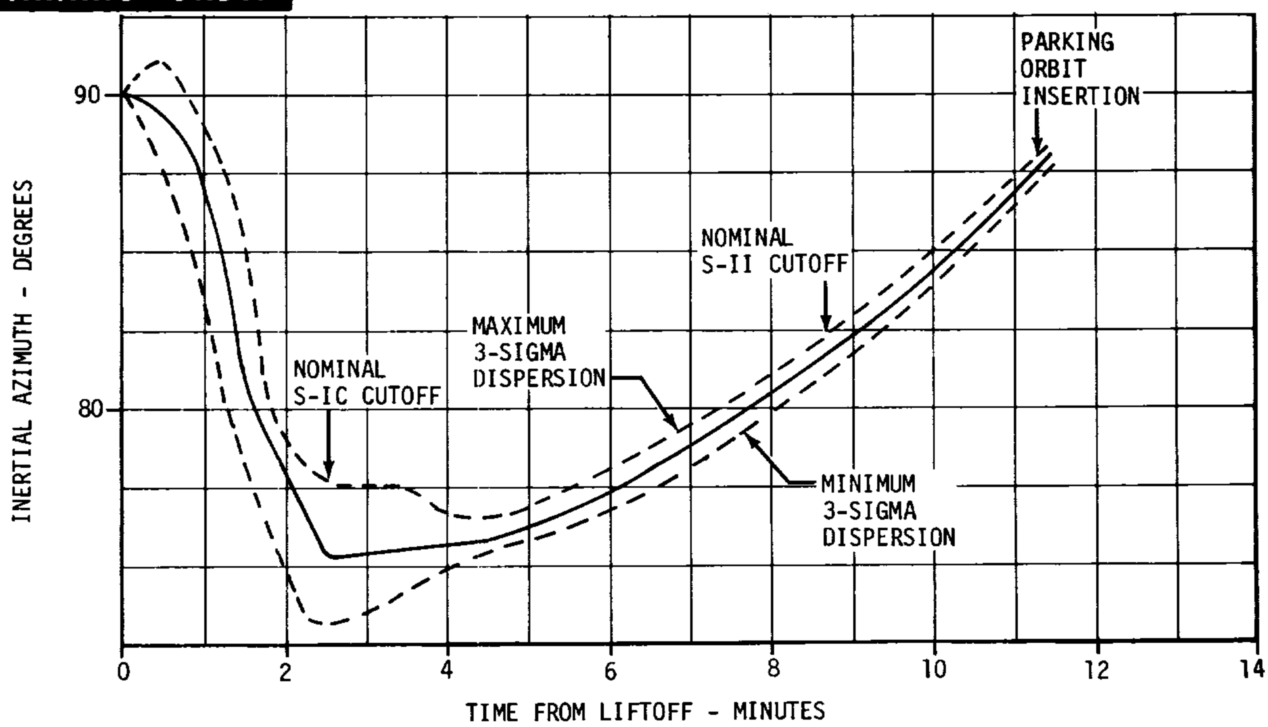


Figure 2-10

TYPICAL GEODETIC LATITUDE CORRIDOR DURING BOOST TO PARKING ORBIT

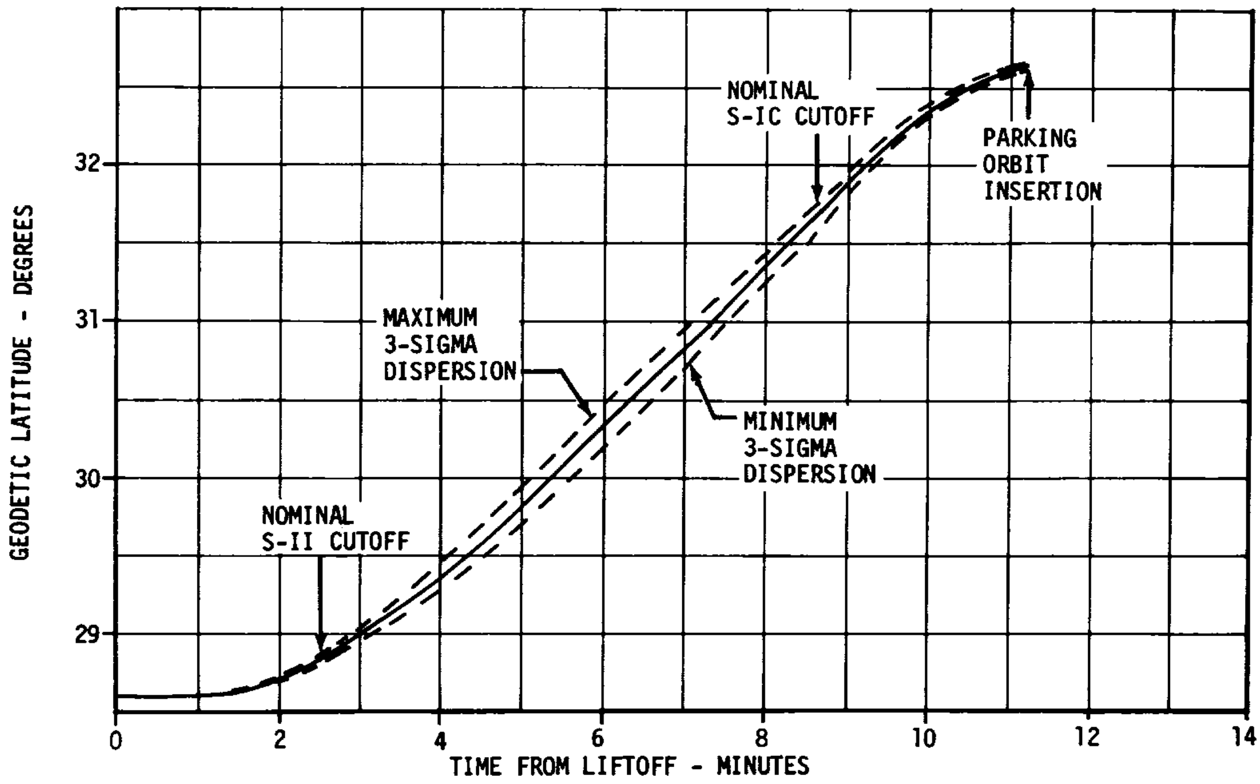


Figure 2-11

TYPICAL LONGITUDE CORRIDOR DURING BOOST TO PARKING ORBIT

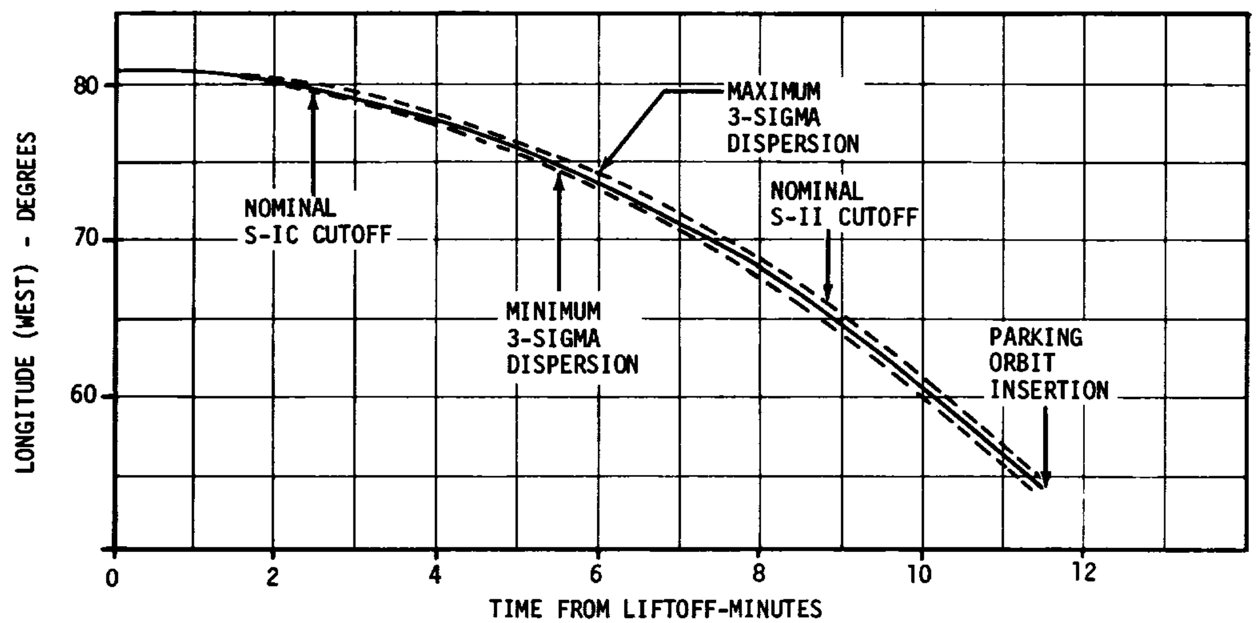


Figure 2-12

TYPICAL RANGE CORRIDOR DURING BOOST TO PARKING ORBIT

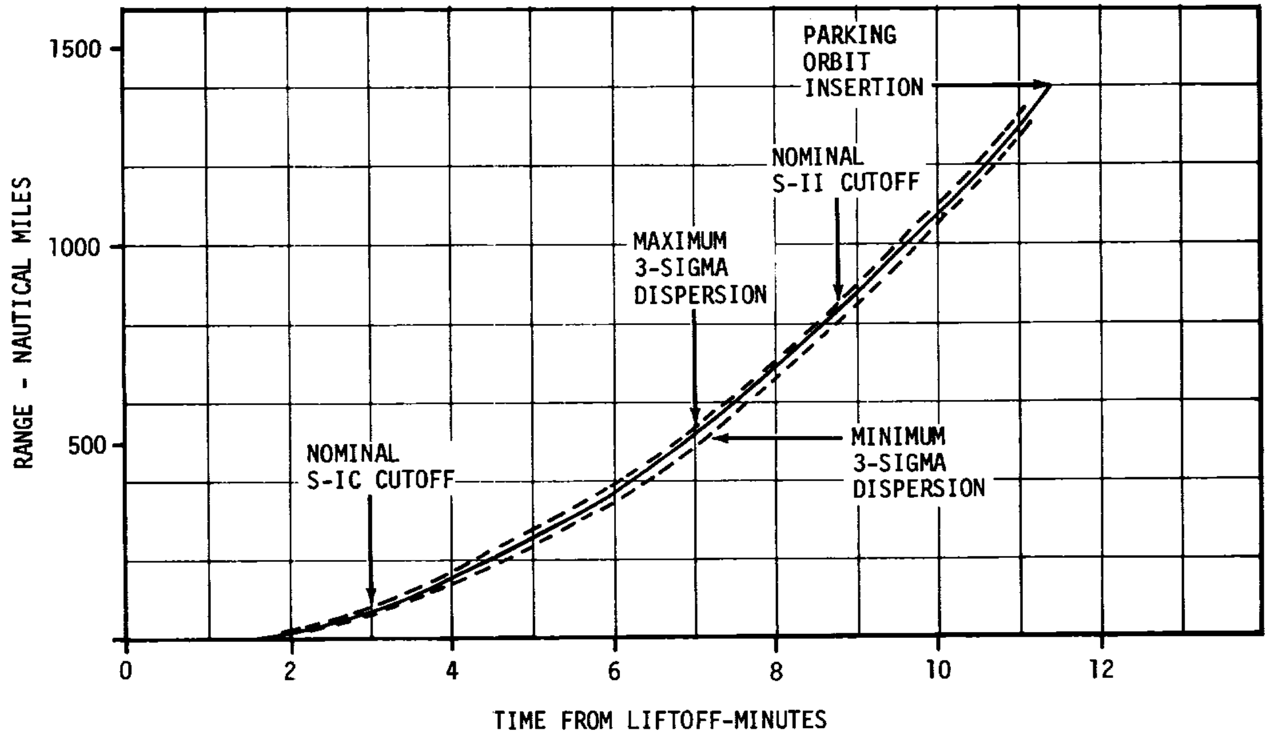


Figure 2-13

TYPICAL ALTITUDE CORRIDOR IN PARKING ORBIT

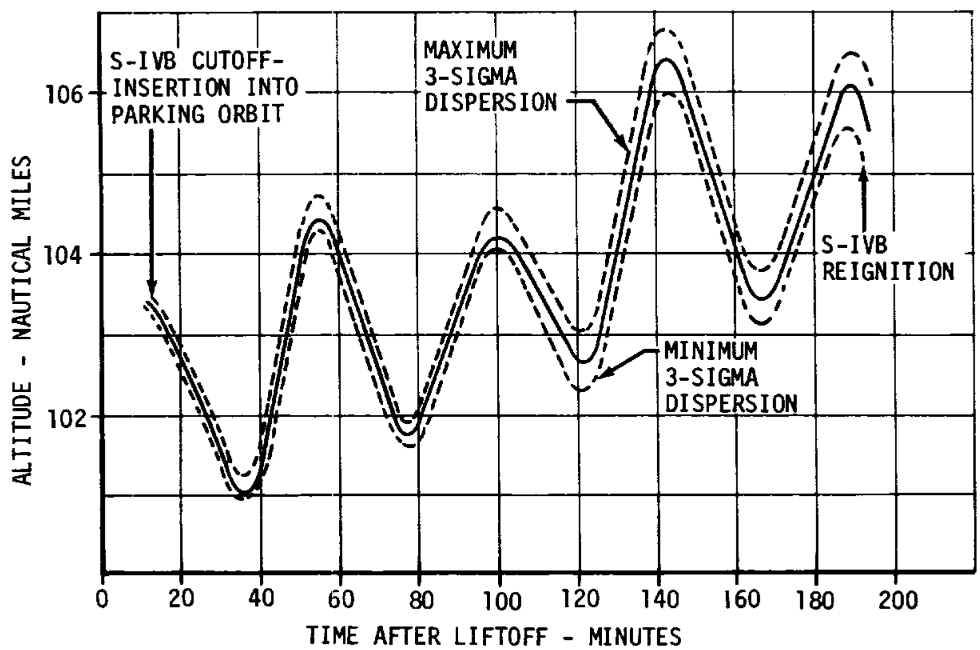


Figure 2-14

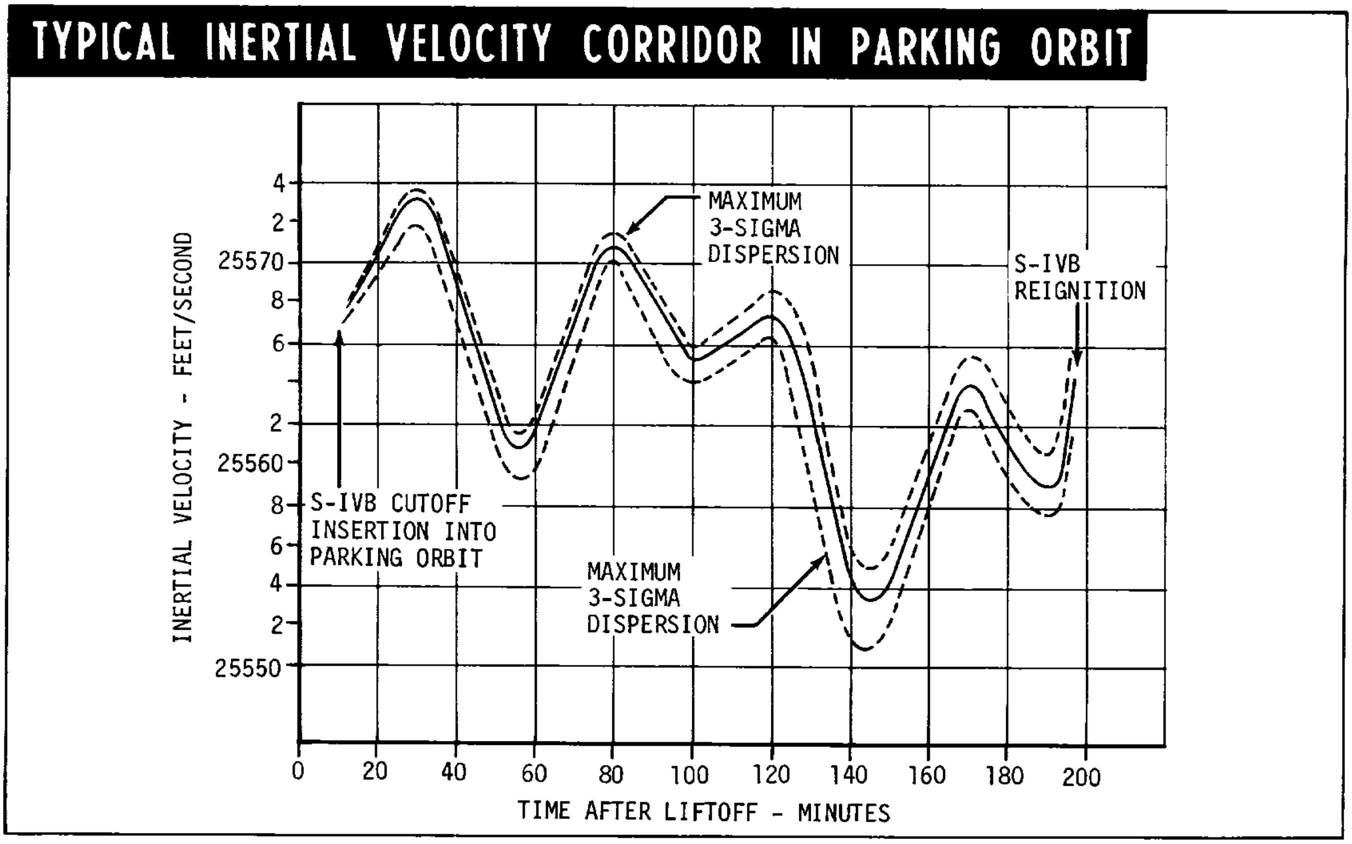


Figure 2-15

Composite Trajectory Parameters

Three parameters significant to flight loads are the aerodynamic heating indicator, dynamic pressure and acceleration. These values are developed around an off-nominal trajectory which would occur if weighted functions of the more significant system dispersions all contributed to increasing the subject flight loads parameters. Typical composite flight data are shown in figures 2-20 through 2-22.

FLIGHT LOADS

Flight loads are dependent on the flight trajectory associated flight parameters and wind conditions. Detailed discussions are contained in the following sections.

WIND CRITERIA

Winds have a significant effect on Saturn V launch vehicle flight loads. Wind criteria used in defining design flight loads for the Saturn V launch vehicle was a scalar wind profile constructed from 95 percentile windiest month speed with 99 percentile shear and a 29.53 feet per second gust. A criteria revision reduces the criteria conservatism for a gust in conjunction with wind shear. Trajectory wind biasing reduces flight loads. Wind biased trajectories are used for launch months with predictable wind speed magnitude and direction.

ENGINE OUT CONDITIONS

Engine out conditions if they should occur will effect the vehicle loads. The time at which the malfunction occurs, which engine malfunctions, peak wind speed and azimuth orientation of the wind are all independent variables which combine to produce load conditions. Each combination of engine-out time, peak wind velocity, wind azimuth, and altitude at which the maximum wind shear occurs produces a unique trajectory. Vehicle responses such as dynamic pressure, altitude, Mach number, angle-of-attack, engine gimbal angles, yaw and attitude angle time histories vary with the prime conditions.

Studies indicate that the immediate structural dynamic transients at engine out will not cause structural failure. However, certain combinations of engine failure and wind direction and magnitude may result in a divergent control condition which could cause loss of the vehicle.

The "Chi-Freeze" schedule is incorporated into the vehicle guidance program as an alternate to reduce the effect of loss in thrust from an S-IC engine. Freeze initiation and freeze duration are dependent upon the time at which the loss in thrust occurred. This schedule holds the pitch attitude command constant. Thereby providing a higher altitude trajectory for minimizing payload losses into orbit and improving engine-out dynamic response by a lower-velocity entry into the maximum aerodynamic region.

A single control engine-out during S-II powered flight does not produce load conditions which are critical.

PROPULSION PERFORMANCE

Propulsion system performance is described in terms of thrust, nominal propellant weights, venting of gaseous propellants, mixture ratio shifts and propellant residuals in the following paragraphs.

PROPELLANT WEIGHTS

Nominal propellant weights for each stage are listed in figures 2-23, 2-24 and 2-25. These figures describe the propellant utilization including the residual and trapped propellants as well as the propellant used for such things as

TYPICAL ALTITUDE CORRIDOR DURING BOOST TO WAITING ORBIT INSERTION

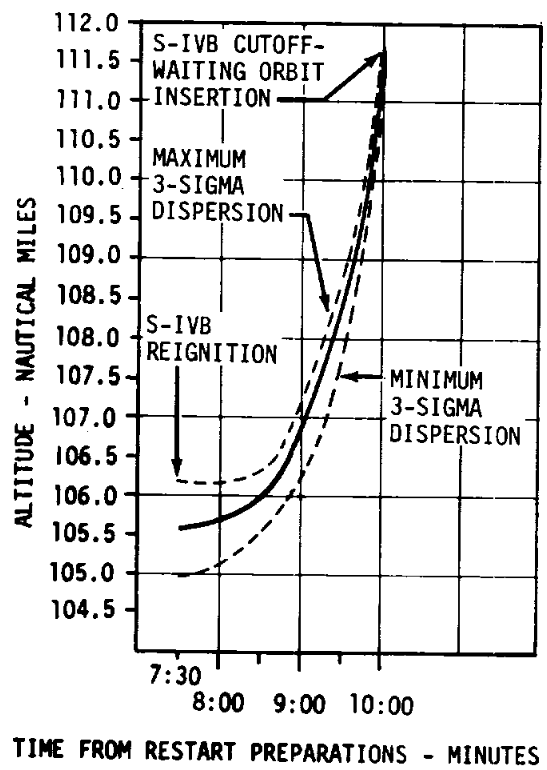


Figure 2-16

TYPICAL INERTIAL VELOCITY CORRIDOR DURING BOOST TO WAITING ORBIT INSERTION

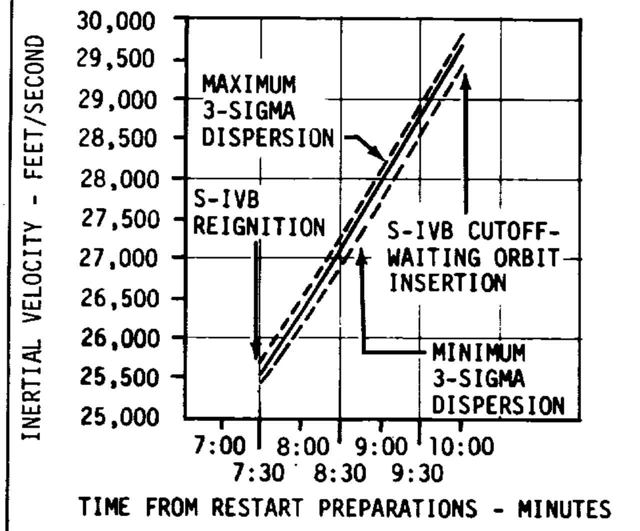


Figure 2-17

S-IVB PROPELLANT MARGIN AT WAITING ORBIT INSERTION

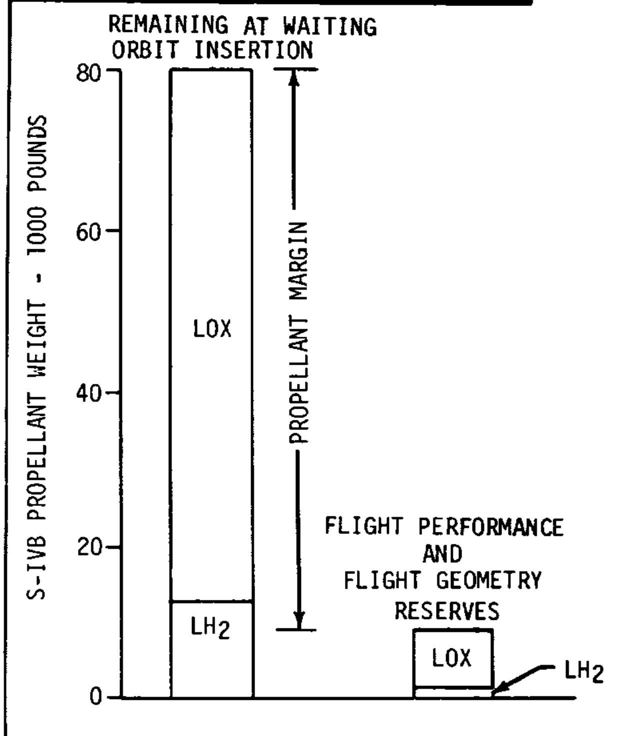


Figure 2-18

S-IVB PROPELLANT-WEIGHT DISPERSIONS AT PARKING ORBIT AND WAITING ORBIT			
PARAMETER	PERTURBATION	WEIGHT CHANGE AT PARKING ORBIT (LBS)	WEIGHT CHANGE AT PARKING ORBIT (LBS)
S-IC STAGE			
Thrust	+ 0.716 pct - 0.716 pct	+ 73 - 127	+ 54 - 94
Specific Impulse	+ 1.118 sec - 1.118 sec	+1277 -1299	+ 945 - 958
Propellant Weight	+ 0.5 pct - 0.5 pct	+ 755 - 765	+ 558 - 569
Inert Weight (Small Interstage & S-IC/S-II Ullage Rocket Propellant)	+ 2,003 lbs - 2,003 lbs	- 219 + 215	- 162 + 157
Large Interstage Weight	+ 350 lbs - 350 lbs	- 43 + 43	- 34 + 29
Mixture Ratio	+ 0.895 pct - 0.895 pct	-2350 -1456	-1736 -1072
Engines Misalign Pitch (down) (up)	+ 0.7433 deg - 0.7433 deg	+1404 -2105	+1035 -1558
Yaw (left) (right)	+ 0.7433 deg - 0.7433 deg	- 461 - 658	- 341 - 485
Axial-Force Coefficient	+ 10 percent - 10 percent	- 509 + 505	- 380 + 371
Radial CG Offset Pitch	+ 2 inches - 2 inches	+ 307 - 322	+ 226 - 241
Yaw	+ 2 inches - 2 inches	- 34 + 5	- 25 + 5
Atmospheric Density	$+\Delta\rho = f(h)$ $-\Delta\rho = f(h)$	- 285 + 338	- 209 + 247
October Headwind (Az = 72°)	3-sigma	- 724	- 538
October Tailwind (Az = 252°)	3-sigma	+1982	+1463
October Right Cross-wind (Az = 162°)	3-sigma	- 38	- 29
October Left Cross-wind (Az = 342°)	3-sigma	- 120	- 92

Figure 2-19 (Sheet 1 of 2)

S-IVB PROPELLANT-WEIGHT DISPERSIONS AT PARKING ORBIT AND WAITING ORBIT			
PARAMETER	PERTURBATION	WEIGHT CHANGE AT PARKING ORBIT (LBS)	WEIGHT CHANGE AT WAITING ORBIT (LBS)
S-II STAGE			
Thrust	+ 1.34 pct - 1.34	+ 579 - 890	+ 426 - 660
Specific Impulse	+ 1.789 sec - 1.789 sec	+1532 -1527	+1128 -1129
Propellant Weight	+ 0.5 pct - 0.5 pct	+ 268 - 325	+ 201 - 239
Inert & S-II/S-IVB Interstage Weight	+ 1581 lbs - 1581 lbs	- 896 + 901	- 661 + 661
Mixture Ratio	+ 0.895 pct - 0.895 pct	- 76 - 282	- 57 - 210
Engine Misalign Pitch (down) (up)	+ 0.7433 deg - 0.7433 deg	+ 53 - 432	+ 39 - 317
Yaw (left) (right)	+ 0.7433 deg - 0.7433 deg	- 173 - 183	- 130 - 135
S-IVB STAGE			
Thrust	+ 3.0 pct - 3.0 pct	+ 70 - 82	+ 51 - 68
Specific Impulse	+ 4.0 sec - 4.0 sec	+ 663 - 677	+1076 -1090
Propellant Weight	+ 1.0 pct - 1.0 pct	+ 473 - 482	+ 383 - 357
Inert & Vehicle IU Weight	+ 583 lbs - 583 lbs	- 465 + 460	- 494 + 494
Mixture Ratio	+ 2.0 pct - 2.0 pct	- 26 + 23	- 95 + 93
Engines Misalign Pitch (down) (up)	+ 2.1 deg - 2.1 deg	- 53 - 82	- 96 - 120
Yaw (left) (right)	+ 2.1 deg - 2.1 deg	- 63 - 67	- 106 - 111
Ullage Thrust	+ 3.536 pct - 3.536 pct	N/A N/A	- 3 + 3
Venting in Parking Orbit	+ 3-sigma - 3-sigma	N/A N/A	- 280 + 263

Figure 2-19 (Sheet 2 of 2)

TYPICAL COMPOSITE AERODYNAMIC HEATING INDICATOR TIME HISTORY

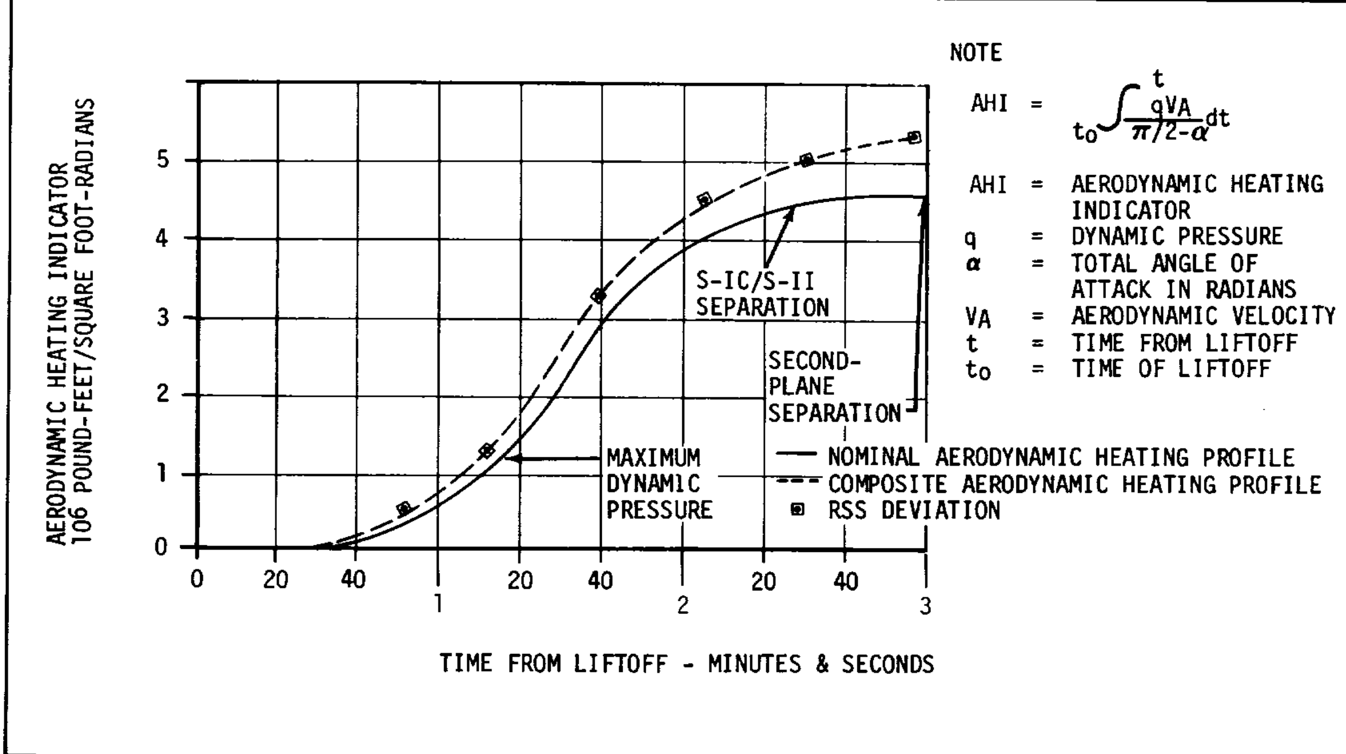


Figure 2-20

TYPICAL LONGITUDINAL ACCELERATION INCREASE DUE TO S-IC COMPOSITE ACCELERATION TRAJECTORY

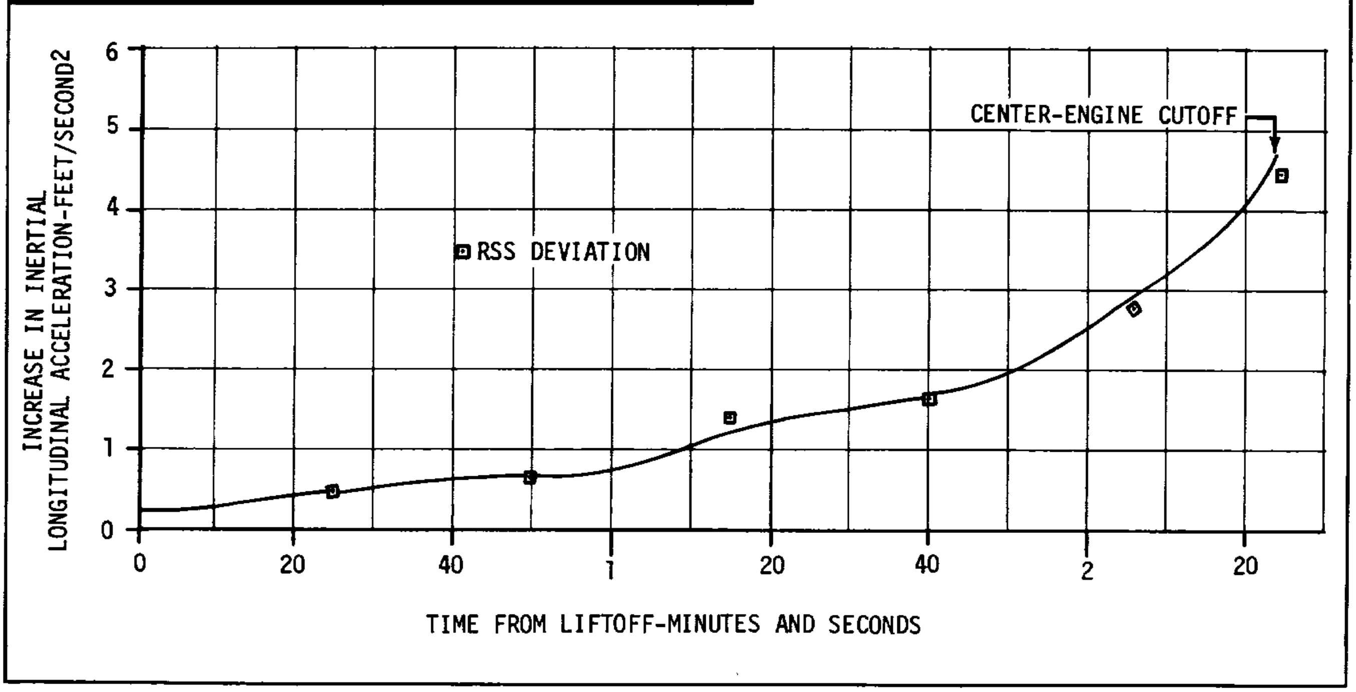


Figure 2-21

hold down, thrust buildup, and decay, and main-stage thrust.

CUTOFF AND MIXTURE RATIO SHIFT TIME

S-IC Stage

The S-IC stage inboard engine is cutoff at 2 minutes 26 seconds after vehicle first motion by propellant level sensors in the lox tank stand pipe. The outboard engines cutoff about 12 seconds later.

S-II Stage

A propellant utilization (PU) system is used to minimize propellant residuals in the S-II stage. During the 2 seconds after 90% thrust is attained the propellant utilization valve is locked in the nominal position (5.0 mixture ratio). After 2 seconds the P.U. system controls the engines to a mixture ratio of 5.5 which lasts for the next 4 minutes and 15 seconds and then to a 4.7 mixture ratio for the remainder of the S-II flight. During nominal operation all five engines are cut off simultaneously by the propellant height sensors located in the propellant tanks. Cutoff is due to lox depletion and occurs at approximately 6 minutes and 19 seconds after attainment of 90% thrust.

S-IVB Stage

S-IVB nominal stage operation is at a 5.5 mixture ratio during first burn. Engine cutoff into orbit is controlled by the guidance system and occurs at about 2 minutes and 27 seconds after 90% thrust is attained. For second burn the engine is started with the PU valve on the lower stop (mixture ratio approximately 4.5). The valve is released to respond to the propellant error signal in an effort to reach near simultaneous depletion of lox and LH_2, 2.5 seconds after 90% thrust is attained and then operates at a mixture ratio of approximately 5.0 throughout the second burn. Engine cutoff is guidance controlled for the second burn, if allowed to operate until propellant depletion the cutoff would be initiated by the height sensors in the lox tank at approximately 5 minutes and 12 seconds after 90% thrust.

THRUST VS TIME HISTORY

S-IC Stage

The S-IC stage thrust history shown in figure 2-26 illustrates the thrust reduction (approximately 1.9 million pounds) which occurs when the center engine shuts down at 2 minutes, 26 seconds. Total thrust termination occurs 12 seconds later.

S-II Stage

The S-II stage thrust history shown in figure 2-27 illustrates the thrust reduction which occurs due to the programmed mixture ratio shift which occurs at 4 minutes, 15 seconds.

S-IVB Stage

The S-IVB stage thrust history is illustrated in figure 2-28. The first burn at a 5.5 mixture ratio provides approximately 232,000 pounds of thrust for 2 minutes 27 seconds. The second burn performance is influenced by a 5.0 mixture ratio which provides approximately 211,000 pounds of thrust.

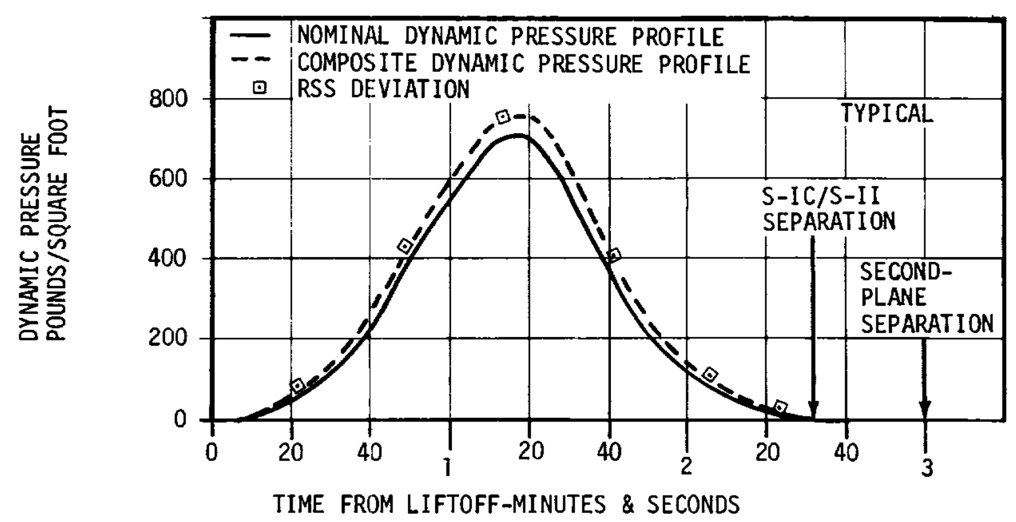


Figure 2-22

S-IC STAGE PROPELLANT WEIGHTS				
	RP-1 (LB_M)		LOX (LB_M)	
Usable Propellants				
Buildup and Holddown	20,874		80,899	
Mainstage	1,377,372		3,192,358	
Thrust Decay	3,434		5,331	
Pressurization	---------		5,404	
Expended	415		1,625	
Total		1,402,095		3,285,617
Fuel Bias	5,694		---------	
Unusable Propellant	2,768		2,100	
Tanks	6,449		15,100	
Suction Lines	---------		330	
Interconnect Lines	6,585		2,165	
Engines				
Total		15,802		19,695
Total Stage Fill Weight		1,423,591		3,305,312

Figure 2-23

S-II STAGE PROPELLANT WEIGHTS				
	LH_2 (LB_M)		LOX (LB_M)	
Usable Propellant				
Mainstage	152,434		817,562	
Thrust Buildup Propellant	598		1,409	
Thrust Decay Propellant	108		332	
Total		153,140		819,303
Fuel Bias		360		0
Unusable Propellant	48		490	
Engine	186		843	
Line	10		230	
Recirculation	54		216	
Initial Ullage Mass	2,682		2,878	
Tank & Sump	1,189		3,045	
Pressurization Gas				
Total		4,169		7,702
Total Stage Fill Weight		157,669		827,005

Figure 2-24

S-IVB STAGE PROPELLANT WEIGHTS (Based on 2-1/2 Orbit Coast Period with a 5.0 MR Second Burn)	LH_2 (LB_M)		LOX (LB_M)	
Usable Propellant (includes Mainstage Flight Performance Reserves and Flight Geometry Reserves)		37,081		191,723
Fuel Bias		126		0
Unusable Propellant				
Orbital Boiloff	3,055		250	
Flight Boiloff	85		100	
Fuel Lead	49			
Subsystems*	302		10	
Engine Trapped	10		108	
Lines Trapped	38		259	
Tank Unavailable	685		40	
Buildup Transients (1st & 2nd Burn)	222		644	
Decay Transients (1st & 2nd Burn)	62		139	
Total		4,508		1,550
Total Stage Fill Weight		41,715		193,273

*Includes 12 LB_M Fuel and 10 LB_M Lox for O_2/H_2 Burner

Figure 2-25

TYPICAL S-IC VEHICLE THRUST VS TIME HISTORY

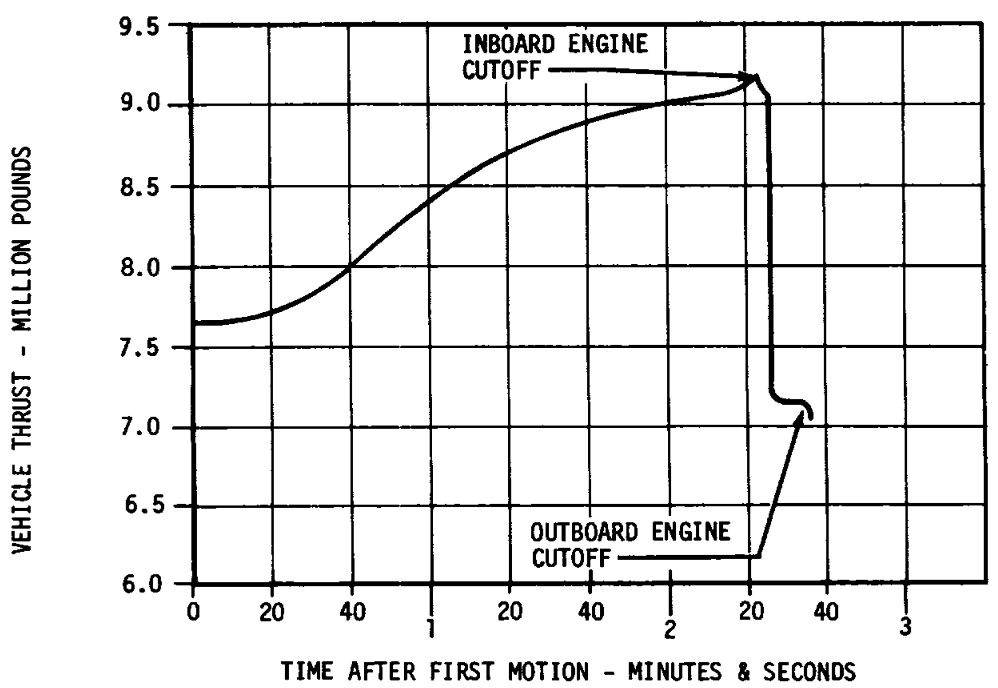


Figure 2-26

TYPICAL S-II VEHICLE THRUST VS TIME HISTORY

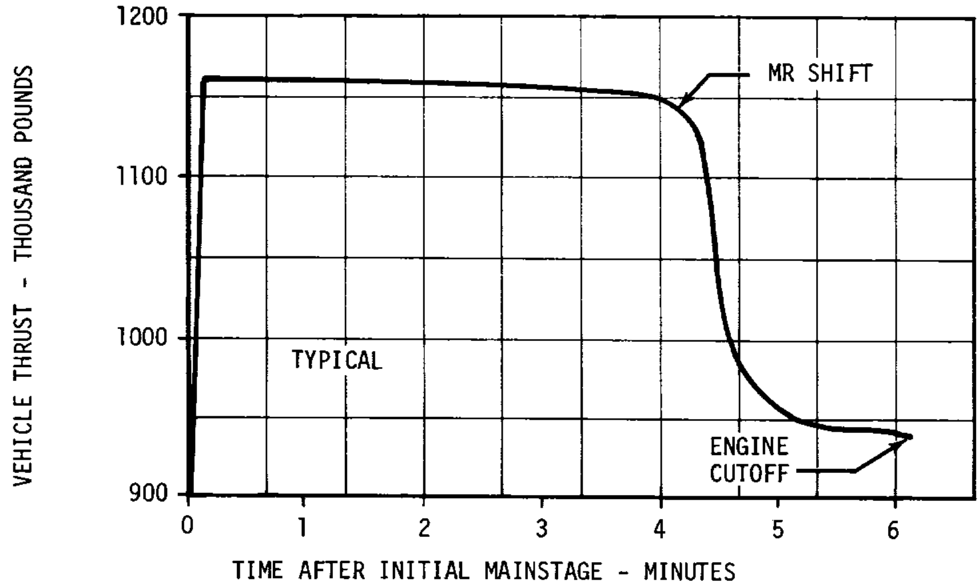


Figure 2-27

TYPICAL S-IVB VEHICLE THRUST VS TIME HISTORY

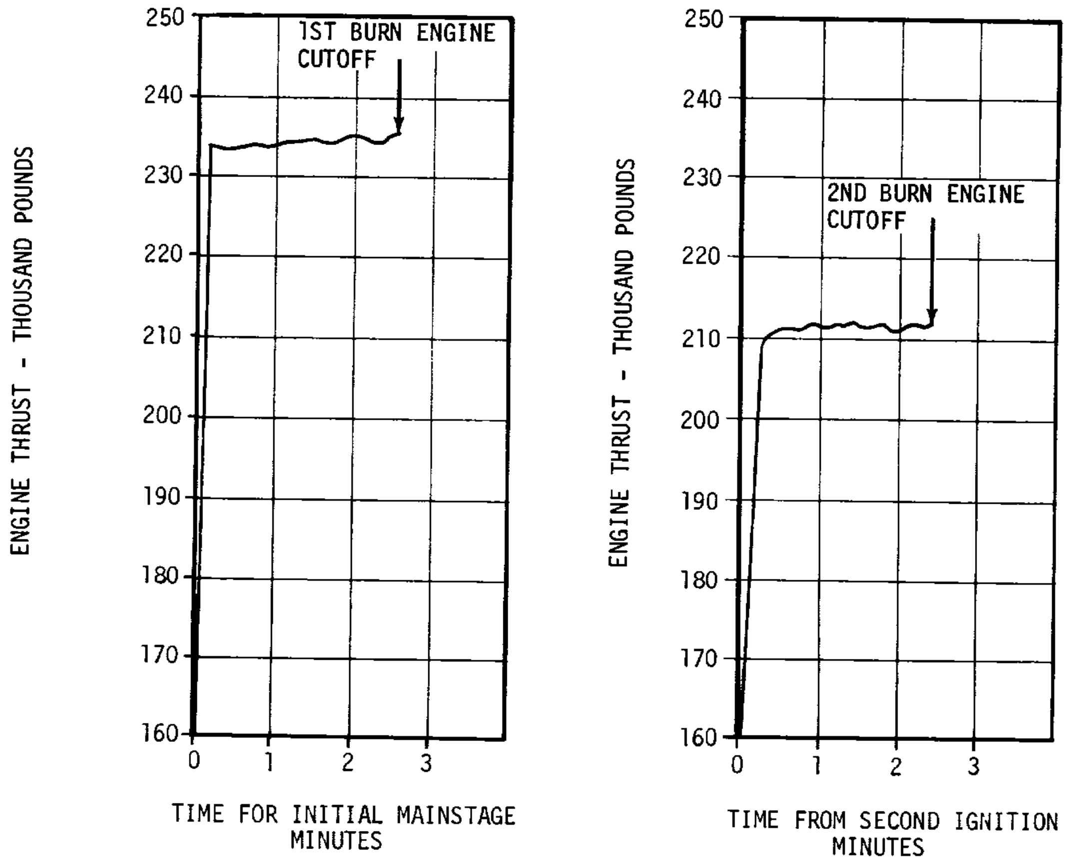


Figure 2-28

SECTION III

EMERGENCY DETECTION AND PROCEDURES

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EMERGENCY MODES 3-13

EMERGENCY DETECTION SYSTEM

The displays implemented for Emergency Detection System (EDS) monitoring (figure 3-1) were selected to present as near as possible those parameters which represent the failures leading to vehicle abort. Whenever possible the parameter was selected so that it would display total subsystem operation. Manual abort parameters have been implemented with redundant sensing and display to provide highly reliable indications to the crewmen. Automatic abort parameters have been implemented triple redundant, voted two-out-of-three to protect against single point hardware or sensing failures and inadvertent abort.

The types of displays have been designed to provide onboard detection capability for rapid rate malfunctions which may require abort. Pilot abort action must, in all cases, be based on two separate but related abort cues. These cues may be derived from the EDS displays, ground information, physiological cues, or any combination of two valid cues. In the event of a discrepancy between onboard and ground based instrumentation, onboard data will be used.

EDS DISPLAYS

The flight director attitude indicator (FDAI) (figure 3-2) provides indications of launch vehicle attitude, attitude rates, and attitude errors, except that attitude errors will not be displayed during S-IVB flight.

Excessive pitch, roll, or yaw indications provide a single cue that an abort is required. Additional abort cues will be provided by the FDAI combining rates, error, or total attitude. Second cues will also be provided by the LV RATE light, LV GUID light, physiological, and MCC ground reports.

The FDAI will be used to monitor normal launch vehicle guidance and control events. The roll and pitch programs are initiated simultaneously at + 10 seconds. The roll program is terminated at 72 degrees azimuth (18 seconds) and the pitch program continues throughout first stage flight. Guidance initiate will occur at approximately 3 minutes 8 seconds.

LV ENGINE LIGHTS

The five launch vehicle engine lights ON indicate that each corresponding S-IC or S-II engine is below nominal thrust levels. (F-1 engine is 90% and J-2 is 65%). The engine light cluster also provides indications of the LV staging sequence (figure 3-1, sheet 1).

Physical separation of the S-IC and S-II is indicated by all lights off. (After normal S-IC cutoff).

The lights all come on again at S-II ignition and go off again at 65% engine thrust. A similar sequence occurs during S-II and S-IVB separation with the exception that the No. 1 light is used to represent the S-IVB stage.

LV RATE LIGHT

The LV RATE light (figure 3-1, sheet 2), when ON, is the primary cue from the launch vehicle that preset overrate settings have been exceeded. It is a single cue for abort, while secondary cues will be provided by FDAI indications, physiological cues, or ground information.

Automatic L/V rate aborts are enabled automatically at lift-off (with EDS AUTO and LV RATES AUTO switches enabled in S/C) and are active until deactivated by the crew. EDS auto abort deactivation times will be governed by mission rules. The automatic L/V rate abort capability is also deactivated by the launch vehicle sequencer prior to center engine cutoff and is not active during S-II or S-IVB flight.

The automatic abort overrate settings are constant throughout first stage flight. The overrate settings are:

Pitch and Yaw	$4 \pm 0.5°$/sec.	Lift-off to S-II ignition
	$9 \pm 0.5°$/sec.	S-II ignition to S-IVB cutoff
Roll	$20 \pm 0.5°$/sec.	Lift-off to S-IVB cutoff

The LV RATE light will illuminate at any time during first, second, or third stage flight if the L/V rates exceed these values.

The LV RATE light will also light in response to a separate signal to provide a second cue for a LV platform failure while in the "Max q" region. This circuit is only active during the time the overrate portion of the auto abort system is armed (Lift Off to approximately T+2 minutes).

NOTE

The LV RATE light may blink ON and OFF during normal staging.

LV GUID Light

The L/V platform (ST-124M-3) is interrogated every 25 milliseconds for the correct attitude. If an excessive attitude discrepancy is found during three consecutive checks on the fine resolvers, three more checks will be performed on the coarse resolvers. (The L/V can continue the mission on the coarse resolvers alone). If the coarse resolvers fail to control within 15 derees per second rate of attitude change in any plane, commands sent to the control system to change L/V attitude will be inhibited and the control system will hold the last acceptable control command.

A signal is sent from the LVDA to activate the LV GUID light (figure 3-1, sheet 2) in the Command Module (CM) at the same time the control commands are inhibited. It is a single cue for abort. Second cues will be provided by the LV RATE Light (only when the Auto Abort system is on) and by the FDAI, angle of attack ($q\alpha$) meter and/or ground information.

LIFT OFF/NO AUTO ABORT LIGHTS

The LIFT OFF and NO AUTO ABORT lights (figure 3-1, sheet 2) are independent indications contained in one switch/light assembly.

The LIFT OFF light ON indicates that vehicle release has been commanded and that the IU umbilical has ejected. The S/C digital event timer is started by the same function. The LIFT OFF light is turned OFF at S-IC center engine cutoff.

The NO AUTO ABORT light ON indicates that one or both of the spacecraft sequencers did not enable automatic abort capability at liftoff. Automatic abort capability can be enabled by pressing the switch/light pushbutton. If the light remains ON, then the crew must be prepared to back up the automatic abort manually. The NO AUTO ABORT light is also turned OFF at S-IC inboard engine cutoff.

WARNING

If the NO AUTO ABORT pushbutton is depressed at T-0 and a pad shutdown should occur, a pad abort will result.

ABORT LIGHT

The ABORT light (figure 3-1, sheet 2) may be illuminated by ground command from the flight director, the Mission Control Center (MCC) booster systems engineer, the Flight dynamics officer, the complex 39 launch operations manager (until tower clearance at +10 seconds), or in conjunction with range safety booster engine cutoff and destruct action. The ABORT light ON constitutes one abort cue. An RF voice abort request constitutes one abort cue.

NOTE

Pilot abort action is required prior to receipt of an ABORT light or a voice command for a large percentage of the time critical launch vehicle malfunctions, particularly at liftoff and staging.

ANGLE OF ATTACK

The angle of attack ($q\alpha$) meter (figure 3-1, sheet 2) is time shared with service propulsion system (SPS) chamber pressure. The $q\alpha$ display is a pitch and yaw vector summed angle of attack/dynamic pressure product ($q\alpha$). It is expressed in percentage of total pressure for predicted launch vehicle breakup (abort limit equals 100%). It is effective as an abort parameter only during the high q flight region from +50 seconds to +1 minute 40 seconds.

Except as stated above, during ascent, the $q\alpha$ meter provides trend information on launch vehicle flight performance and provides a secondary cue for slow rate guidance and control malfunctions. Primary cues for guidance and control malfunctions will be provided by the FDAI, physiological cues, and/or Mission Control Center (MCC) callout.

Nominal angle of attack meter indications should not exceed 25%. Expected values based on actual winds aloft will be provided by MCC prior to launch.

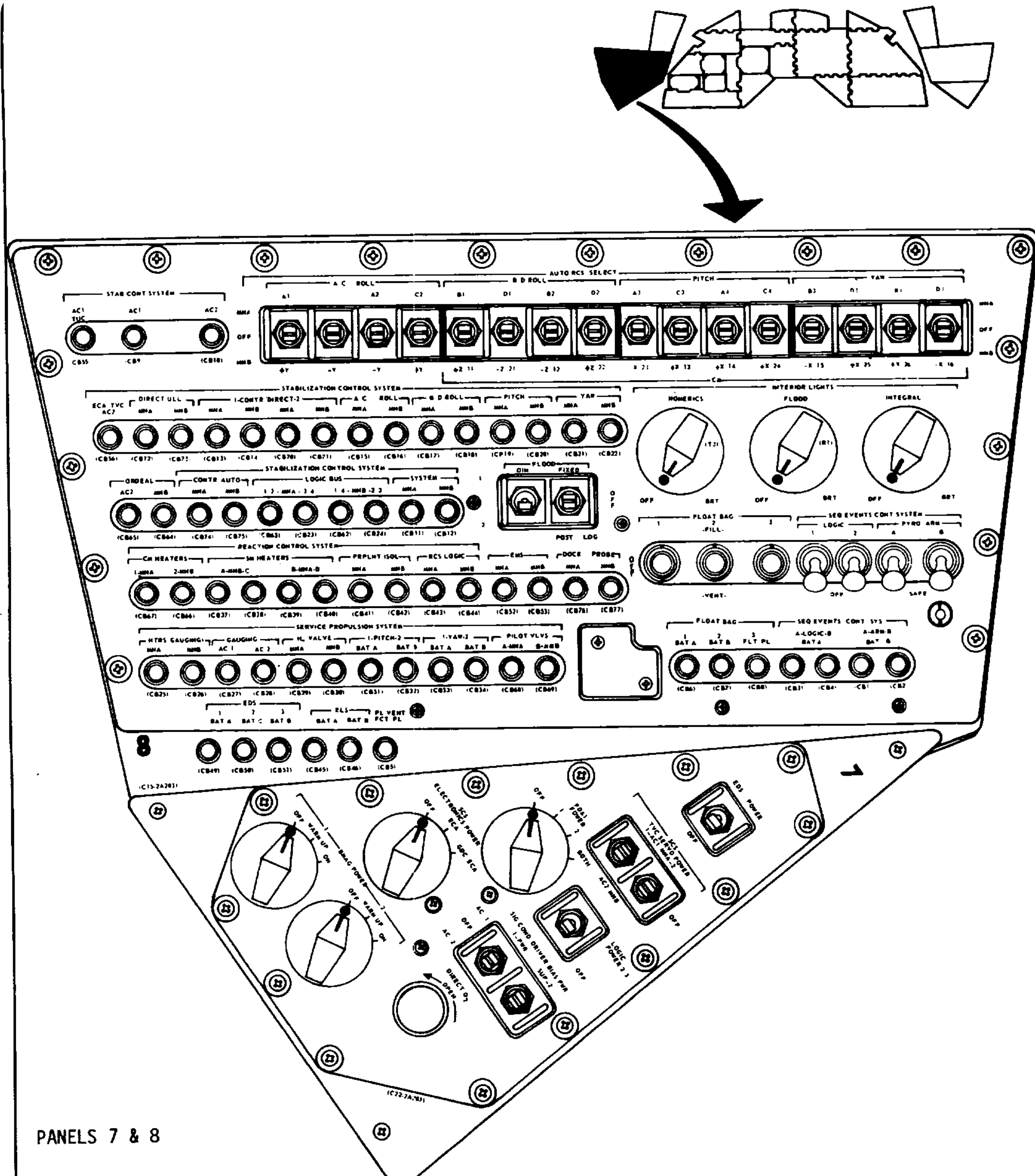
ACCELEROMETER

The accelerometer (figure 3-1, sheet 2) indicates longitudinal acceleration/deceleration. It provides a secondary cue for certain engine failures and is a gross indication of launch vehicle performance. The accelerometer also provides a readout of G-forces during re-entry.

S-II SEP LIGHT

With S-IC jettison (staging), the S-II SEP light will illuminate. The light will go out

MAIN DISPLAY CONSOLE EDS PANELS



PANELS 7 & 8

Figure 3-1 (Sheet 1 of 3)

MAIN DISPLAY CONSOLE EDS PANELS

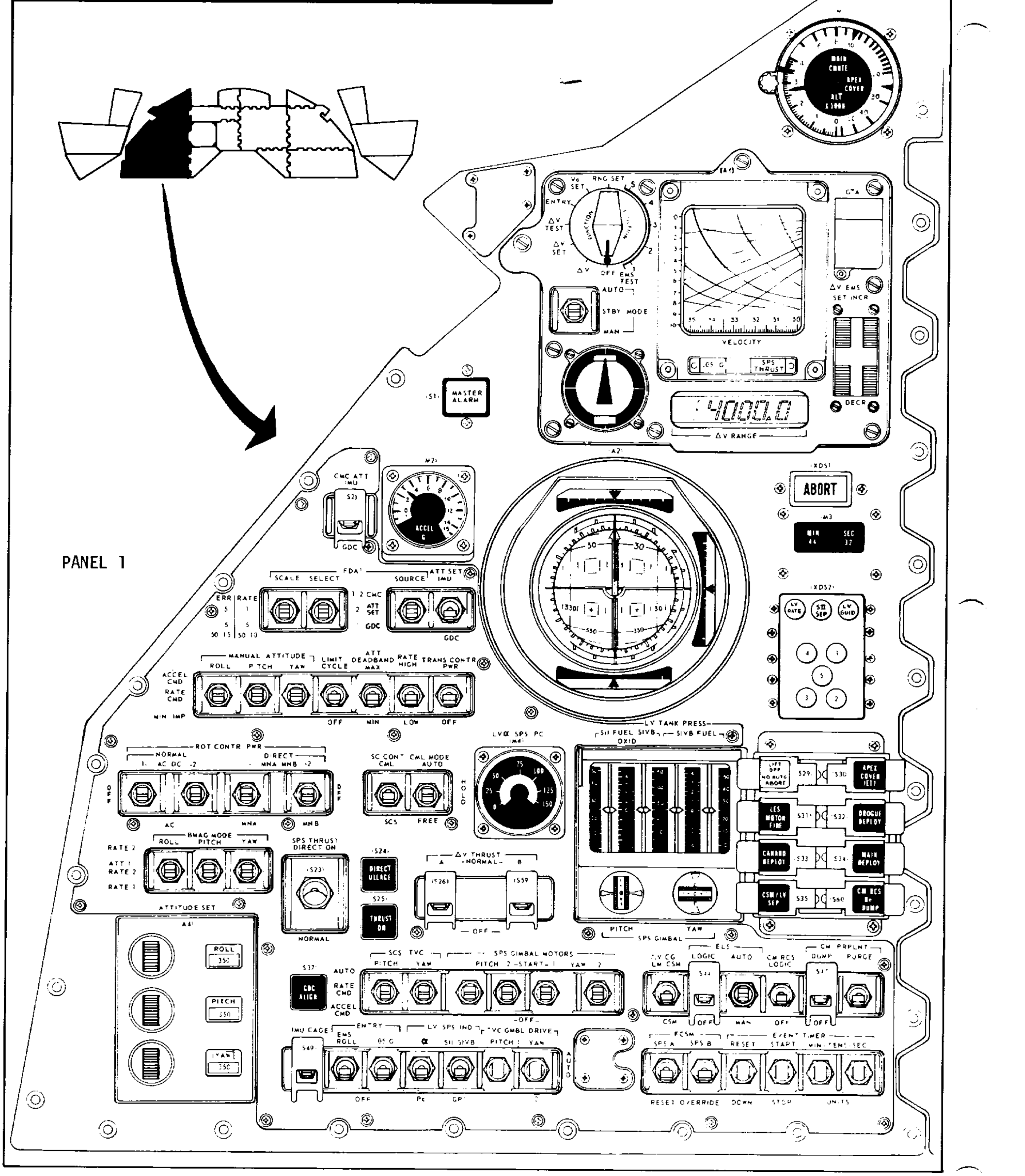


Figure 3-1 (Sheet 2 of 3)

MAIN DISPLAY CONSOLE EDS PANELS

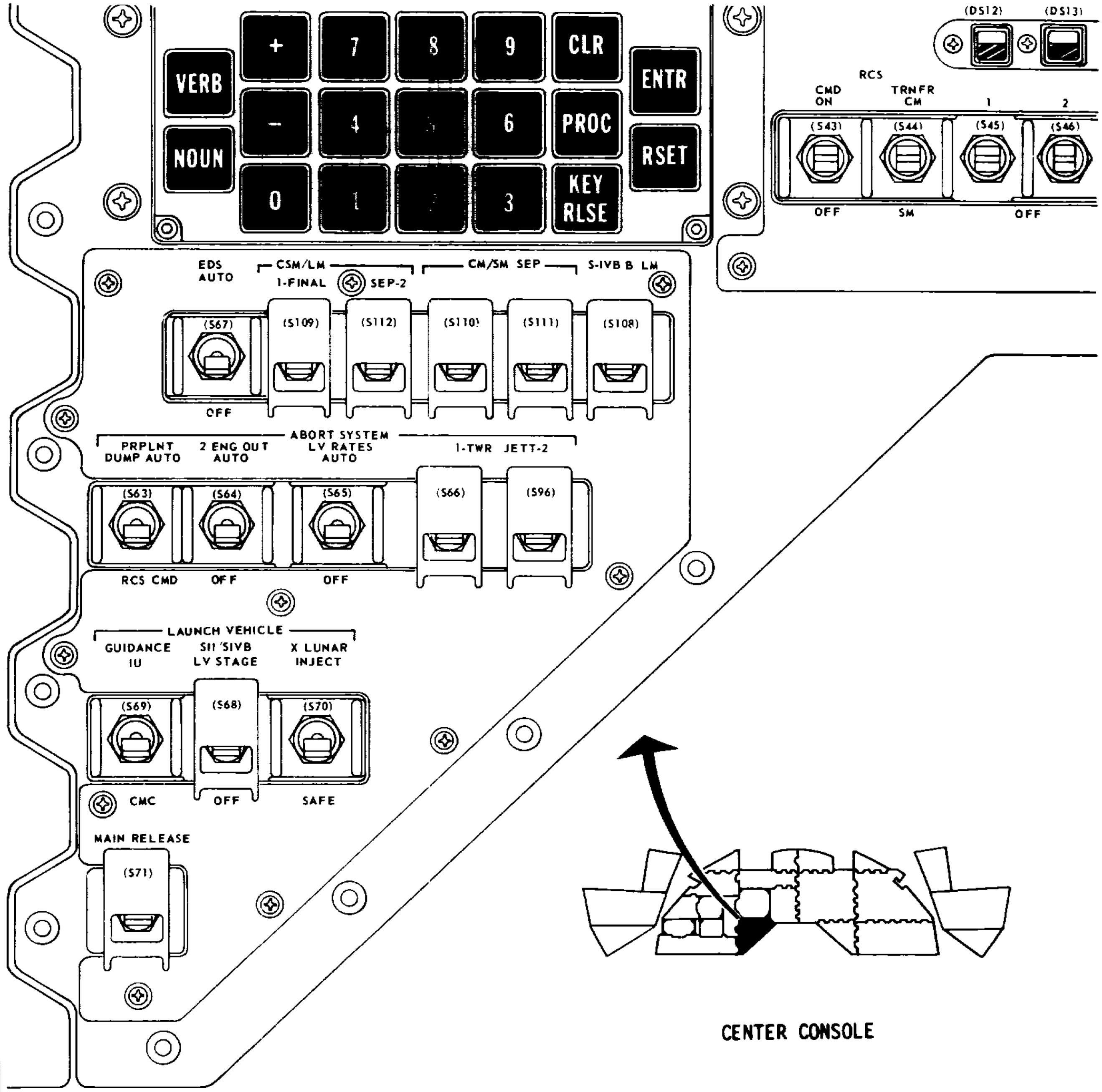


Figure 3-1 (Sheet 3 of 3)

FLIGHT DIRECTOR ATTITUDE INDICATOR

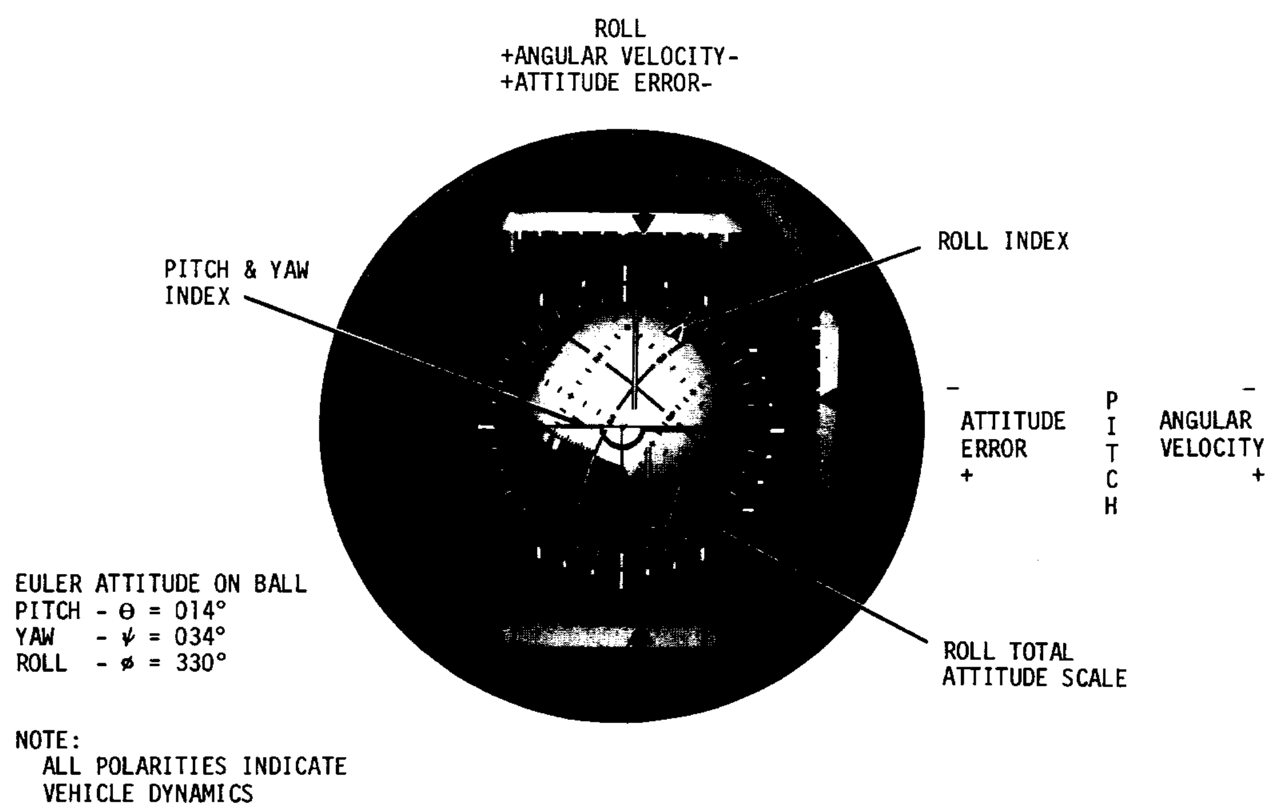


Figure 3-2

approximately 30 seconds later when the interstage structure is jettisoned. A severe overheating problem will occur if the structure is not jettisoned at the nominal time. Under the worst conditions, abort limits will be reached within 25 seconds from nominal jettison time. Confirmation from Mission Control of interstage failure to jettison serves as the second abort cue.

The S-II SEP light also indicates during S-IVB restart as follows:

Light ON from T_6 + 0:00 to T_6 + 1:25.
Light OFF from T_6 + 1:25 to T_6 + 6:20.
Light ON from T_6 + 6:20 to T_6 + 7:20.
Light OFF at T_6 + 7:20.

ALTIMETER

Due to dynamic pressure, static source location, and instrument error the altimeter is not considered to be an accurate instrument during the launch phase.

The primary function of the altimeter is to provide an adjustable (set for barometric pressure on launch date) reference for parachute deployment for pad/near pad launch escape system (LES) aborts. However, the aerodynamic shape of the CM coupled with the static source location produces errors up to 1300 feet. Therefore, the main parachutes must be deployed at an indicated 3800 feet (depends on launch day setting) to ensure deployment of 2500 feet true altitude.

EVENT TIMER

The event timer is critical because it is the primary cue for the transition of abort modes, manual sequenced events, monitoring roll and pitch program, staging, and S-IVB insertion cutoff. The event timer is started by the lift-off command which enables automatic aborts. The command pilot should be prepared to manually back up its start to assure timer operation.

The event timer is reset to zero automatically with abort initiation.

MASTER ALARM LIGHT

The three MASTER ALARM Lights ON alert the flight crew to critical spacecraft failures or out-of-tolerance conditions identified in the caution and warning light array. After extinguishing the alarm lights, action should be initiated to correct the failed or out-of-tolerance subsystem. If crew remedial action does not correct the affected subsystem, then an abort decision must be made based upon contingencies. In many cases remedial action will correct the malfunctioned or out-of-tolerance subsystem. Secondary abort cues will come from subsystem displays, ground verification, and physiological indications.

NOTE

The commander's MASTER ALARM light will not illuminate during the launch phase but the other two MASTER ALARM lights will illuminate and the alarm tone will sound.

EMERGENCY DETECTION SYSTEM CONTROL

EDS AUTO SWITCH

The EDS AUTO switch (figure 3-1, sheet 3) is the master switch for EDS initiated automatic abort. When placed in the AUTO position (normally prior to liftoff) an automatic abort will be initiated if:

1. A L/V structural failure occurs between the IU and the CSM.
2. Two or more S-IC engines drop below 90% of rated thrust.
3. L/V rates exceed 4 degrees per second in pitch or yaw or 20 degrees per second in roll.

The two engine out and L/V rate portions of the auto abort system can be manually disabled individually by the crew. However, they are automatically disabled by the L/V sequencer prior to inboard cutoff.

PRPLNT SWITCH

The PRPLNT switch is normally in the DUMP AUTO position prior to liftoff in order to automatically dump the CM reaction control system (RCS) propellants and fire the pitch control (PC) motor if an abort is initiated during the first 42 seconds of the mission. The propellant dump and PC motor are inhibited by the S/C sequencer at 42 seconds. The switch in the RCS CMD position will inhibit propellant dump and PC motor firing at any time.

TOWER JETTISON SWITCHES

Either of two redundant switches can be used to fire the explosive bolts and the tower jettison motor. The appropriate relays are also de-energized so that if an abort is commanded, the SPS abort sequence and not the LES sequence will occur. The switches are momentary to the TWR JETT position. Both switches should be activated to ensure that redundant signals are initiated.

No other automatic functions will occur upon activation of the TOWER JETTISON switches.

CM/SM SEP SWITCHES

Redundant momentary ON guarded switches, spring loaded to the OFF position, are used by the command pilot to accomplish Command Module/Service Module (CM/SM) separation when required. Both switches should be activated to ensure that redundant signals are initiated.

These switches can also be used to initiate an LES abort in case of a failure in either the EDS or the translational controller. All normal post abort events will then proceed automatically. However, the CANARD DEPLOY pushbutton should be depressed 11 seconds after abort initiation because canard deployment and subsequent events will not occur if the failure was in the EDS instead of the translational controller. If the CANARD DEPLOY pushbutton is depressed, all automatic functions from that point on will proceed normally.

S-II/S-IVB LV STAGE

Switch activation applies power to the LVDA to initiate the S-II/S-IVB staging sequence. This capability is provided to allow the crew to manually upstage from a slowly diverging failure of the S-II stage. If the S-IVB does not have a sufficient velocity increment (ΔV) available for orbit insertion, an abort can be performed at predetermined locations.

S-IVB SEP

Switch activation will sever the tension ties and start two parallel time delays of 0.03 sec. each. At expiration of the time delay, the LM tension tie firing circuit is deadfaced and the guillotine is fired.

CSM/LM FINAL SEP

The LM docking ring is simultaneously jettisoned with the TWR JETT command during an LES abort. During a normal entry or an SPS abort, the ring must be jettisoned by the CSM/LM FINAL SEP switch located on panel 1. Failure to jettison the ring could possibly hamper normal earth landing system (ELS) functions.

ELS AUTO /MANUAL Switch

The ELS AUTO/MANUAL switch (figure 3-1, sheet 2) is used in conjunction with either the automatically initiated ELS sequence (post abort) or with the manually initiated sequence (ELS LOGIC ON). It is normally in the AUTO position as it will inhibit all ELS function when in the MAN position.

ELS LOGIC SWITCH

The ELS LOGIC switch (figure 3-1, sheet 2) is a guarded switch which should only be activated during normal re-entry or following an SPS abort; and then only below 45,000 feet altitude. If activated at any time below 30,000 feet (pressure altitude) the landing sequence will commence; i.e., LES and apex cover jettison and drogue deployment. If activated below 10,000 feet altitude the main chutes will also deploy.

ELS LOGIC is automatically enabled following any manual or auto EDS initiated LES abort. It should be manually backed up if time permits.

WARNING

Activation of ELS LOGIC switch below 30,000 feet altitude will initiate landing sequence; i.e., LES and apex cover jett, and drogue deploy.

CM RCS PRESS

Any time the CM is to be separated from the SM, the CM RCS must be pressurized. The normal sequence of events for an abort or normal CM/SM SEP is to automatically deadface the umbilicals, pressurize the CM RCS, and then separate the CM/SM. However, if the automatic pressurization fails, the CM RCS can be pressurized by the use of the guarded switch located on panel 2.

MAIN CHUTE RELEASE

The MAIN CHUTE RELEASE switch is a guarded switch. It is used to manually release the main chutes after the Command Module has landed. No automatic backup is provided. This switch is armed by the ELS LOGIC switch ON and the 10k barometric switches closed (below 10,000 feet altitude).

NOTE

The ELS AUTO switch must be in the AUTO position to allow the 14 second timer to expire before the MAIN CHUTE RELEASE switch will operate.

ABORT CONTROLS

TRANSLATIONAL CONTROLLER

Manual aborts will be commanded by counterclockwise (CCW) rotation of the translational controller T-handle. Clockwise (CW) rotation will transfer S/C control from the command module computer (CMC) to the stability control system (SCS).

For LES aborts, the CCW position sends redundant engine cutoff commands (engine cutoff from the S/C is inhibited for the first 30 seconds of flight) to the launch vehicle, initiates CM/SM separation, fires the LES motors, resets the sequencer, and initiates the post abort sequence.

For Service Propulsion System (SPS) aborts, the CCW rotation commands L/V engine cutoff, resets the spacecraft sequencer and initiates the CSM/LV separation sequence.

The T-handle also provides CSM translation control along one or more axes. The control is mounted approximately parallel to the S/C axis; therefore, T-handle movement will cause corresponding S/C translation. Translation in the +X axis can also be accomplished by use of the direct ullage pushbutton; however, rate damping is not available when using this method.

SEQUENCER EVENT MANUAL PUSHBUTTONS

The LES MOTOR FIRE, CANARD DEPLOY, CSM/LV SEP, APEX COVER JETT, DROGUE DEPLOY, MAIN DEPLOY, and CM RCS He DUMP, pushbuttons (figure 3-1, sheet 2) provide backup of sequenced events for both abort and normal re-entry situations.

The MAIN DEPLOY pushbutton is the primary method of deploying the main parachutes for pad/near pad aborts to assure deceleration to terminal velocity at touchdown (downrange tipover).

SPS SWITCHES AND DISPLAYS

Direct Ullage Pushbutton

When the DIRECT ULLAGE pushbutton (figure 3-1, sheet 2) is depressed, a +X translation utilizing all four quads results. It is the backup method for ullage maneuvers prior to an SPS burn (the prime method for ullage is the translational controller). The DIRECT ULLAGE switch is momentary and must be held until the ullage maneuver is complete. It will not provide rate damping.

THRUST ON Pushbutton

The THRUST ON pushbutton (figure 3-1, sheet 2) can be used to start the SPS engine under the following conditions:

1. S/C control is in the SCS mode.
2. Ullage is provided.
3. ΔV THRUST switches (either of two) are in the NORMAL position.

NOTE

Both switches must be OFF to shut off the engine.

4. Backup for guidance and navigation (G&N) start command.

SPS Engine Shutdown

The SPS engine can be shut off (when fired as described above) in the following manner:

1. Flight combustion stability monitor (FCSM) shuts it down automatically.
2. $\Delta V = 0$ (SCS or MTVC).
3. ΔV THRUST switches (both) OFF.

NOTE

The SPS thrust light, located on the EMS panel, will illuminate when the engine is firing.

SPS THRUST DIRECT SWITCH

The switch is a two position lever lock toggle switch. The ON position provides a ground for the solenoid valve power and all of the SCS logic. The engine must be turned off manually by placing the SPS THRUST DIRECT switch in NORMAL. The ΔV THRUST switches must be in the NORMAL position (at least one) to apply power to the solenoids for the SPS THRUST DIRECT switch to operate.

WARNING

The SPS THRUST DIRECT switch is a single point failure when the ΔV THRUST switches are in the NORMAL position.

SPS Gimbal Motors/Indicators

Four gimbal motors control the SPS engine position in the pitch and yaw planes (two in each plane). These motors are activated by four switches located on the panel. The motors should be activated one at a time due to excessive current drain during the start process.

The gimbal thumbwheels can be used to position the gimbals to the desired attitude as shown on the indicators. The indicators are analog displays time shared with the booster fuel and oxidizer pressure readings.

The other methods of controlling the gimbal movement is through the hand controller in the MTVC mode or by automatic SCS logic.

ΔV THRUST (Prevalves and Logic)

The two guarded switches apply power to the SPS solenoid prevalves and to the SCS logic for SPS ignition. These switches must be on NORMAL position before the SPS engine can be started; even by the SPS THRUST DIRECT switch.

WARNING

Both switches must be OFF to stop the engine.

SCS Thrust Vector Control (TVC)

These switches are active only in the SCS mode. The pitch and yaw channels can be used independently; i.e., pitch control could be in SCS AUTO and yaw in either RATE CMD or ACCEL CMD position. The three available modes are:

1. AUTO: The TVC is directed by the SCS electronics.
2. RATE CMD: MTV with rate damping included,
3. ACCEL CMD: MTVC without rate damping.

ΔV and ΔV SET Switches

In order for the ΔV counter to operate during an SPS burn, the switches located on the EMS panel must be in the following positions:

1. EMS MODE - AUTO
2. EMS FUNCTION - ΔV

To set the ΔV counter for a desired ΔV burn the switches would be as follows:

1. EMS MODE - AUTO
2. EMS FUNCTION - ΔV SET

The five position ΔV SET SLEW switch is then used to place the desired quantity on the ΔV display.

SCS SYSTEM SWITCHES

AUTO RCS SELECT Switches

Power to the RCS Control Box Assembly is controlled by 16 switches located on panel eight (figure 3-1, sheet 1). Individual RCS engines may be enabled or disabled as required. Power to the Attitude Control logic is also controlled in this manner, which thereby controls all attitude hold and/or maneuvering capability using SCS electronics (automatic coils). The direct solenoids are not affected as all SCS electronics are bypassed by activation of the Direct RCS switches.

NOTE

The automatic coils cannot be activated until the "RCS enable" is activated either by the MESC or manually.

DIRECT RCS Switches

The two DIRECT RCS switches provide manual control of the SM RCS engines. The control is achieved by positioning the rotation control hardover to engage the direct solenoids for the desired axis change.

All SCS electronics are bypassed when this switch is activated.

ATT SET Switch

The ATT SET switch selects the source of total attitude for the ATT SET resolvers as outlined below.

Position	Function	Description
UP	IMU	Applies inertial measurement unit (IMU) gimbal resolver signal to ATT SET resolvers. FDAI error needles display differences. Needles are zeroed by maneuvering S/C or by moving the ATT SET dials.
DOWN	GDC	Applies GDC resolver signal to ATT SET resolvers. FDAI error needles display differences resolved into body coordinates. Needles zeroed by moving S/C or ATT SET dials. New attitude reference is established by depressing GDC ALIGN button. This causes GDC to drive to null the error; hence, the GDC and ball go to ATT SET dial value.

MANUAL ATTITUDE Switches

These three switches (ROLL, PITCH, and YAW) are only operative when the S/C is in the SCS mode of operation.

Position	Description
ACCEL CMD	Provides direct RCS firing as a result of moving the rotational controller out of detent (2.5°) to apply direct inputs to the solenoid driver amplifiers.
RATE CMD	Provides proportional rate command from rotational controller with inputs from the BMAG's in a rate configuration.
MIN IMP	Provides minimum impulse capability through the rotational controller.

LIMIT CYCLE Switches

The pseudo-rate function provides the capability of maintaining low S/C rates while holding the S/C attitude within the selected deadband limits (limit cycling). This is accomplished by pulse-width modulation of the switching amplifier outputs. Instead of driving the S/C from limit-to-limit with high rates by firing the RCS engines all the time, the engines are fired in "spurts" proportional in length and repetition rate to the switching amplifier outputs.

Extremely small attitude corrections could be commanded which would cause the pulse-width of the resulting output command to be of too short a duration to activate the RCS solenoids. A "one-shot" multivibrator is connected in parallel to ensure a long enough pulse to fire the engines.

RATE and ATT DEADBAND Switches

The switching amplifier deadband can be interpreted as a rate or an attitude (minimum) deadband. The deadband limits are a function of the RATE switch. An additional deadband can be enabled in the attitude control loop with the ATT DEADBAND switch (see figure 3-3).

ATTITUDE DEADBAND SWITCH POSITION			
RATE SWITCH POSITION	RATE DEADBAND °/SEC	ATT DEADBAND SWITCH POSITION	
		MINIMUM	MAXIMUM
LOW	± 0.2	$\pm 0.2°$	$\pm 4.2°$
HIGH	± 2.0	$\pm 2.0°$	$\pm 8.0°$

Figure 3-3

The rate commanded by a constant stick deflection (Proportional Rate Mode only) is a function of the RATE switch position. The rates commanded at maximum stick deflection (soft stop) are shown in figure 3-4.

MAXIMUM PROPORTIONAL RATE COMMAND		
RATE SWITCH POSITION	MAXIMUM PROPORTIONAL RATE COMMAND	
	PITCH AND YAW	ROLL
LOW	0.65°/sec	0.65°/sec
HIGH	7.0 °/sec	20.0 °/sec

Figure 3-4

SC CONT Switch

This switch selects the spacecraft control as listed below:

Position	Description
CMC	Selects the G & N system Computer controlled S/C attitude and TVC through the digital autopilot. An autopilot control discrete is also applied to CMC.

SCS The SCS system controls the S/C attitude and TVC.

MAG MODE Switches

These switches (ROLL, PITCH, YAW) (figure 3-1, sheet 2) select displays for the FDAI using SCS inputs.

Position	Description
RATE 2	BMAG Set No. 2 provides the rate displays on the FDAI. There is no Body Mounted Attitude Gyro (BMAG) attitude reference available.
ATT 1 RATE 2	BMAG Set No. 1 provides attitude reference on the FDAI, while Set No. 2 provides the rate display.
RATE 1	BMAG Set No. 1 provides the rate displays on the FDAI. There is no BMAG attitude reference available.

ENTRY EMS ROLL Switch

This switch enables the EMS roll display for earth re-entry phase of the flight.

ENTRY, .05 G Switch

Illumination of the .05 G light located on the EMS panel is the cue for the crew to actuate the .05 G switch. During atmospheric re-entry (after .05 G), the S/C is maneuvered about the stability roll axis rather than the body roll axis. Consequently, the yaw rate gyro generates an undesirable signal. By coupling a component of the roll signal into the yaw channel, the undesirable signal is cancelled. The .05 G switch performs this coupling function.

EMS DISPLAYS

Threshold Indicator (.05 G Light)

This indicator provides the first visual indication of total acceleration sensed at the reentry threshold (approximately 290,000 feet). Accelerometer output is fed to a comparison network and will illuminate the .05 G lamp when the acceleration reaches .05 G. The light will come on not less than 0.5 seconds or more than 1.5 seconds after the acceleration reaches .05 G and turns off when it falls below .02 G (skipout).

Corridor Indicators

By sensing the total acceleration buildup over a given period of time, the re-entry flight path angle can be evaluated. This data is essential to determine whether or not the entry angle is steep enough to prevent superorbital "skipout."

The two corridor indicator lights are located on the face of the roll stability indicator (bottom left-hand corner of the EMS panel).

If the acceleration level is greater than 0.2 G at the end of a ten second period after threshold (.05 G light ON), the upper light will be illuminated. It remains on until the G-level reaches 2 G's and then goes OFF. The lower light illuminates if the acceleration is equal to or less than 0.2 G at the end of a ten second period after threshold. This indicates a shallow entry angle and that the lift vector should be down for controlled entry; i.e., skipout will occur.

Roll Stability Indicator

This indicator provides a visual indication of the roll attitude of the CM about the stability axis. Each revolution of the indicator represents 360 degrees of vehicle rotation. The display is capable of continuous rotation in either direction. The pointer up position (0°) indicates maximum liftup vector (positive lift) and pointer down (180°) indicates maximum lift-down vector (negative lift).

G-V PLOTTER

The G-V plotter assembly consists of a scroll of mylar tape and a G-indicating stylus. The tape is driven from right to left by pulses which are proportional to the acceleration along the velocity vector. The stylus which scribes a coating on the back of the mylar scroll, is driven in the vertical direction in proportion to the total acceleration.

The front surface of the mylar scroll is imprinted with patterns consisting of "high-G-rays" and "exit rays." The "high-G-rays" must be monitored from initial entry velocity down to 4,000 feet per second. The "exit rays" are significant only between the entry velocity and circular orbit velocity and are, therefore, only displayed on that portion of the pattern.

The imprinted "high-G-rays" and "exit rays" enable detection of primary guidance failures of the type that would result in either atmospheric exits at supercircular speeds or excessive load factors at any speed. The slope of the G-V trace is visually compared with these rays. If the trace becomes tangent to any of these rays, it indicates a guidance malfunction and the need for manual takeover.

EMS FUNCTION Switch

This is a 12 position selector mode switch located in the upper lefthand area of the EMS panel used as outlined in figure 3-5.

EMS MODE Switch

This switch, located below the EMS FUNCTION switch, performs the following functions in the positions indicated:

AUTO

1. Acts as a backup display for G & N entry
2. Initiates the function selected by the EMS FUNCTION switch.

STBY

1. Resets circuits following tests
2. Removes power if EMS FUNCTION switch is OFF.

MAN

1. Position for manual entry and TVC modes, or auto entry back up display
2. Does not permit negative acceleration spikes into countdown circuits

EMS FUNCTION SWITCH OPERATION			
Operational Mode	Switch Selection	Switch Position	Description
ΔV Mode	Start at ΔV and rotate clockwise	ΔV	Operational mode for monitoring ΔV maneuvers
		ΔV Set	Establish circuitry for slewing ΔV counter for self test or as operational
		ΔV Test	Operational mode for self test of the ΔV subsystem
Self Test and Entry Mode	Start at No. 1 and rotate counter-clockwise	No. 1	Tests lower trip point of .05 G threshold comparator
		No. 2	Tests higher trip point of .05 G threshold comparator
		No. 3	Tests lower trip point of corridor verification comparator
		No. 4	Tests velocity integration circuitry, g-servo circuitry, G-V plotter, and the range-to-go subsystem
		No. 5	Tests higher trip point of corridor verification comparator
		RNG Set	Establish circuitry for slewing range-to-go counter for operational and test modes
		Vo Set	Establish circuitry for slewing G-V plotter scroll for operational mode
		Entry	Operational mode for monitoring entry mode
		OFF	Turns OFF all power except to the SPS thrust light and switch lighting.

Figure 3-5

ABORT MODES AND LIMITS

The abort mode and limits listed in figures 3-6 and 3-7 are based on a nominal launch trajectory. More specific times can be obtained from current mission documentation.

NOMINAL LAUNCH CALLOUTS

The nominal launch callouts are listed in figure 3-8 for the boost phase only.

EMERGENCY MODES

Aborts performed during the ascent phase of the mission will be performed by either of the two following methods:

LAUNCH ESCAPE SYSTEM

The Launch Escape System (LES) consists of a solid propellant launch escape (LE) motor used to propel the CM a safe distance from the launch vehicle, a tower jettison motor, and a canard subsystem. A complete description on use of the system can be found in the specific mission Abort Summary Document (ASD). A brief description is as follows:

Mode IA Low Altitude Mode

In Mode IA a pitch control (PC) motor is mounted normal to the LE motor to propel the vehicle downrange to enhance water landing and escape the "fireball." The CM RCS propellants are dumped through the aft heat shield during this mode to prevent a possible fire source at landing.

The automatic sequence of major events from abort initiation is:

Time	Event
00:00	Abort Ox Rapid Dump LE and PC Motor Fire
00:05	Fuel Rapid Dump
00:11	Canards Deploy
00:14	ELS Arm
00:14.4	Apex Cover Jett
00:16	Drogue Deploy
00:18	He Purge
00:28	Main Deploy

The automatic sequence can be prevented, interrupted, or replaced by crew action.

Mode IB Medium Altitude

Mode IB is essentially the same as Mode IA with the exception of deleting the rapid propellant dump and PC motor features. The canard subsystem was designed specifically for this altitude region to initiate a tumble in the pitch plane. The CM/tower combination CG is located such that the vehicle will stabilize (oscillations of $\pm$ 30 degrees) in the blunt-end-forward (BEF) configuration. Upon closure of barometric switches, the tower would be jettisoned and the parachutes automatically deployed.

As in Mode IA, the crew intervention can alter the sequence of events if desired.

APOLLO ABORT MODES		
Time Period	Mode	Description
Pad to 42 sec	Mode IA	LET Low Alt
42 Sec to 1 min 50 sec	Mode IB	LET MED Alt
1 min 50 sec to 3 min 07 sec	Mode IC	LET High Alt
3 min 07 sec to 10 min 30 sec	Mode II	Full Lift
10 min 30 sec to insertion	Mode III for CSM No Go	SPS Retro Half Lift
10 min 7 sec to insertion	Mode IV for CSM Go	SPS to Orbit

Figure 3-6

ABORT LIMITS

RATES

1. Pitch and Yaw

L/O to S-IC/S-II Staging	4° per second
Staging to SECO (Excluding staging)	10° per second

2. Roll

L/O to SECO	20° per second

MAX Q REGION

NOTE: The following limits represent single cues and are restricted to the time period from 50 seconds to 1 minute 40 seconds. Abort action should be taken only after both have reached threshold.

1. Angle of Attack = 100 percent
2. Roll Error = 6 degrees

AUTOMATIC ABORT LIMITS
L/O until deactivate (time to be determined)

1.	Pitch and Yaw	4.0° $\pm$ 0.5° per second
	Roll	20.0° $\pm$ 0.5° per second
2.	Any Two Engines Out	

NOTE: Between L/O and 2 + 0, switch TWO ENG OUT AUTO to OFF following confirmation of ONE ENG OUT.

3. CM IU breakup

ENGINE FAILURE
(L/O to CECO)

1.	One Engine Out	Continue Mission
2.	Simultaneous Loss two or more engines	Abort
3.	Second engine loss following confirmation of one engine out	Continue Mission

S-IVB TANK PRESSURE LIMITS
(L/O to CSM/LV SEP)

ΔP LH_2 LO_2 = 26 PSID

LO_2 LH_2 = 36 PSID

Figure 3-7

NOMINAL LAUNCH PHASE VOICE CALLOUTS (BOOST ONLY)			
TIME	STATION	REPORT	EVENT
-0:09	LCC	IGNITION	S-IC IGNITION
0:00	LCC	LIFT-OFF	UMBILICAL DISCONNECT
0:01	CDR	LIFT-OFF	CMD TO P11 DET START
0:11	CDR	ROLL COMMENCE	ROLL PROGRAM STARTS
0:21	CDR	PITCH TRACKING	PITCH RATE DETECTION
0:29	CDR	ROLL COMPLETE	180° ROLL ATTITUDE
0:42	MCC	MARK, MODE IB	PRPLNT DUMP - RCS CMD
1:50	MCC	MARK, MODE IC	h = 100,000 FT, 16.5 NM
2:00	CDR	EDS MANUAL	EDS RATES - OFF
			EDS ENG - OFF
			EDS AOA - Pc
2:10	MCC	GO/NO GO FOR STAGING	STAGING STATUS-TWR JETT STATUS IF REQUIRED
2:12	CDR	GO/NO GO FOR STAGING	
2:25	CDR	INBOARD OFF	S-IC INBOARD ENG - OFF
2:29	CDR	OUTBOARD OFF	S-IC OUTBOARD ENG - OFF
2:33	CDR	STAGING	S-II LIGHTS OFF
3:01	CDR	SECOND PLANE SEP	GIMBAL MOTOR (4) ON
3:07	CDR	TOWER JETT	TOWER JETTISONED
		MARK, MODE II	MAN ATT (P) - RATE CMD
3:12	CDR	GUID INITIATE	IGM STARTS
4:00	CDR	S/C GO/NO GO	
	MCC	GUIDANCE GO/NO GO	IGM LOOKS GOOD
4:30	MCC	TRAJECTOR GO/NO GO	TRAJECTORY STATUS
5:00	CDR	S/C GO/NO GO	
5:19	MCC	S-IVB TO ORBIT CAPABILITY	
6:00	CDR	S/C GO/NO GO	
7:00	CDR	S/C GO/NO GO	
8:00	CDR	S/C GO/NO GO	
8:20	MCC	GO/NO GO FOR STAGING	STAGING STATUS
8:41	CDR	S-II OFF	S-II LIGHTS - ON
8:45	CDR	STAGING	S-IVB LIGHT - OFF
9:00	CDR	S/C GO/NO GO	
9:30	MCC	TRAJECTORY STATUS	
10:00	MCC CDR	S/C GO/NO GO	
10:06	MCC	MARK, MODE IV	TRAJECTORY WITHIN MODE IV CAPABILITY
11:00	MCC	TRAJECTORY STATUS	
11:29	CDR	SECO	S-IVB LIGHT ON
12:00	MCC	ORBITAL GO/NO GO	hp>75 NM (FDO)

Figure 3-8

Mode IC High Altitude

During Mode IC the L/V is above the atmosphere. Therefore, the canard subsystem cannot be used to induce a pitch rate to the vehicle. The crew will, therefore, introduce a five degree per second pitch rate into the system. The CM/tower combination will then stabilize BEF as in Mode IB. The ELS would likewise deploy the parachutes at the proper altitudes.

An alternate method (if the L/V is stable at abort) is to jettison the tower manually and orient the CM to the re-entry attitude. This method provides a more stable entry, but requires a functioning attitude reference.

SERVICE PROPULSION SYSTEM

The Service Propulsion System (SPS) Aborts utilize the Service Module SPS engine to propel the CSM combination away from the L/V, maneuver to a planned landing area, or boost into a contingency orbit. The SPS abort modes are:

Mode II

The SM RCS engines are used to propel the CSM away from the L/V unless the vehicle is in danger of exploding or excessive tumble rates are present at LV/CSM separation. In these two cases the SPS engine would be used due to greater ΔV and attitude control authority. When the CSM is a safe distance and stable, the CM is separated from the SM and maneuvered to a re-entry attitude. A normal entry procedure is followed from there.

Mode III

The SPS engine is used to slow the CSM combination (retrograde maneuver) so as to land at a predetermined point in the Atlantic Ocean. The length of the SPS burn is dependent upon the time of abort initiation. Upon completion of the retro maneuver the CM will separate from the SM, assume the re-entry attitude, and follow normal entry procedures.

Mode IV

The SPS engine can be used to make up for a deficiency in insertion velocity up to approximately 3,000 feet per second. This is accomplished by holding the CSM in an inertial attitude and applying the needed ΔV with the SPS to acquire the acceptable orbital velocity. If the inertial attitude hold mode is inoperative, the crew can take over manual control and maneuver the vehicle using onboard data.

SECTION IV

S-IC STAGE

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STRUCTURE 4-1
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INTRODUCTION

The S-IC stage (figure 4-1) is a large cylindrical booster, 138 feet long and 33 feet in diameter, powered by five liquid propellant F-1 rocket engines. These engines develop a nominal sea level thrust of 1,526,500 pounds each, approximately 7,632,500 pounds total, and have a burn time of 150.5 seconds. The stage dry weight is approximately 295,300 pounds and the total stage weight is approximately 5,030,300 pounds.

The S-IC stage provides first stage boost of the Saturn V launch vehicle to an altitude of about 200,000 feet (approximately 38 miles), and provides acceleration to increase the vehicle velocity to 7,700 feet per second (approximately 4,560 knots). It then separates from the S-II stage and falls to earth about 320 nautical miles downrange.

The stage interfaces structurally and electrically with the S-II stage. It also interfaces structurally, electrically, and pneumatically with two umbilical service arms, three tail service masts, and certain electronic systems by antennae.

The stage consists of: the structural airframe (figure 4-1); five F-1 engines; 648 vehicle monitoring points; electrical, pneumatic control, and emergency flight termination equipment; and eight retrorockets. The major systems of the stage are: structures, propulsion, environmental control, fluid power, pneumatic control, propellants, electrical, instrumentation, and ordnance.

STRUCTURE

The S-IC structure design reflects the requirements of F-1 engines, propellants, control, instrumentation and interfacing systems. The structure maintains an ultimate factor of safety of at least 1.40 applied to limit load and a yield factor of safety of 1.10 on limit load. Aluminum alloy is the primary structural material. The major components, shown in figure 4-2, are the forward skirt, oxidizer tank, intertank section, fuel tank, and thrust structure.

FORWARD SKIRT

The aft end of the forward skirt (figure 4-2) is attached to the oxidizer (lox) tank and the forward end interfaces with the S-II stage. The forward skirt has accommodations for the forward umbilical plate, electrical and electronic canisters, and the venting of the lox tank and interstage cavity. The skin panels, fabricated from 7075-T6 aluminum, are stiffened and strengthened by ring frames and stringers.

IV

OXIDIZER TANK

The 345,000 gallon lox tank is the structural link between the forward skirt and the intertank section. The cylindrical tank skin is stiffened by "integrally machined" T stiffeners. Ring baffles (figure 4-1) attached to the skin stiffeners stabilize the tank wall and serve to reduce lox sloshing. A cruciform baffle at the base of the tank serves to reduce both slosh and vortex action. Support for four helium bottles is provided by the ring baffles. The tank is a 2219-T87 aluminum alloy cylinder with ellipsoidal upper and lower bulkheads. The skin thickness is decreased in eight steps from .254 inches at the aft section to .190 inches at the forward section.

INTERTANK SECTION

The intertank structure provides structural continuity between the lox and fuel tanks. This structure provides a lox fill and drain interface to the intertank umbilical. One opening vents the fuel tank. The corragated skin panels and circumferential ring frames are fabricated from 7075-T6 aluminum.

FUEL TANK

The 216,000 gallon fuel tank (figure 4-2) provides the load carrying structural link between the thrust and intertank structures. It is cylindrical with ellipsoidal upper and lower bulkheads. Antislosh ring baffles are located on the inside wall of the tank and antivortex cruciform baffles are located in the lower bulkhead area (figure 4-1). Five lox ducts (figure 4-1) run from the lox tank, through the RP-1

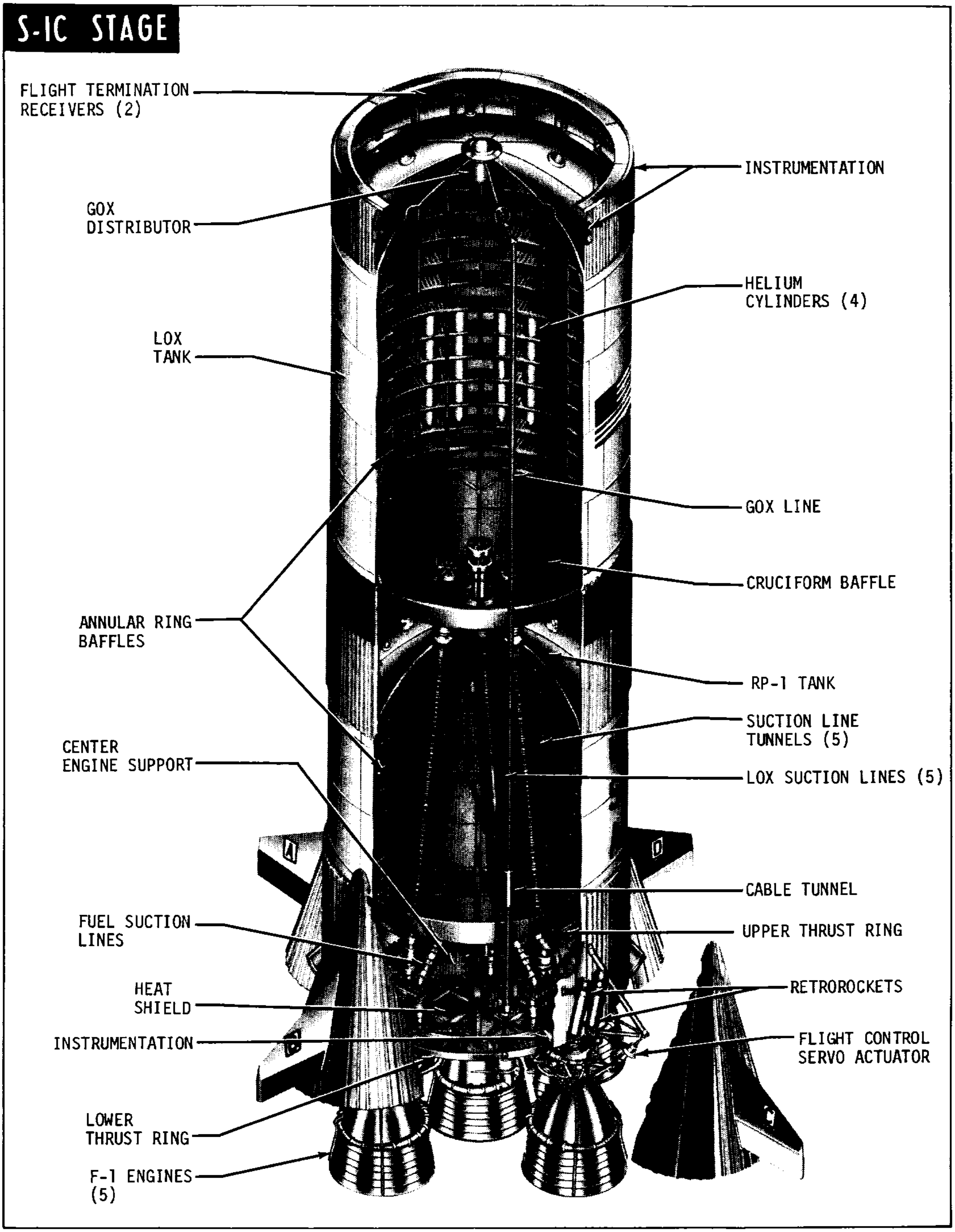

S-IC STAGE
FLIGHT TERMINATION RECEIVERS (2)
INSTRUMENTATION
GOX DISTRIBUTOR
HELIUM CYLINDERS (4)
LOX TANK
GOX LINE
CRUCIFORM BAFFLE
ANNULAR RING BAFFLES
RP-1 TANK
SUCTION LINE TUNNELS (5)
CENTER ENGINE SUPPORT
LOX SUCTION LINES (5)
CABLE TUNNEL
UPPER THRUST RING
FUEL SUCTION LINES
HEAT SHIELD
RETROROCKETS
INSTRUMENTATION
FLIGHT CONTROL SERVO ACTUATOR
LOWER THRUST RING
F-1 ENGINES (5)

Figure 4-1

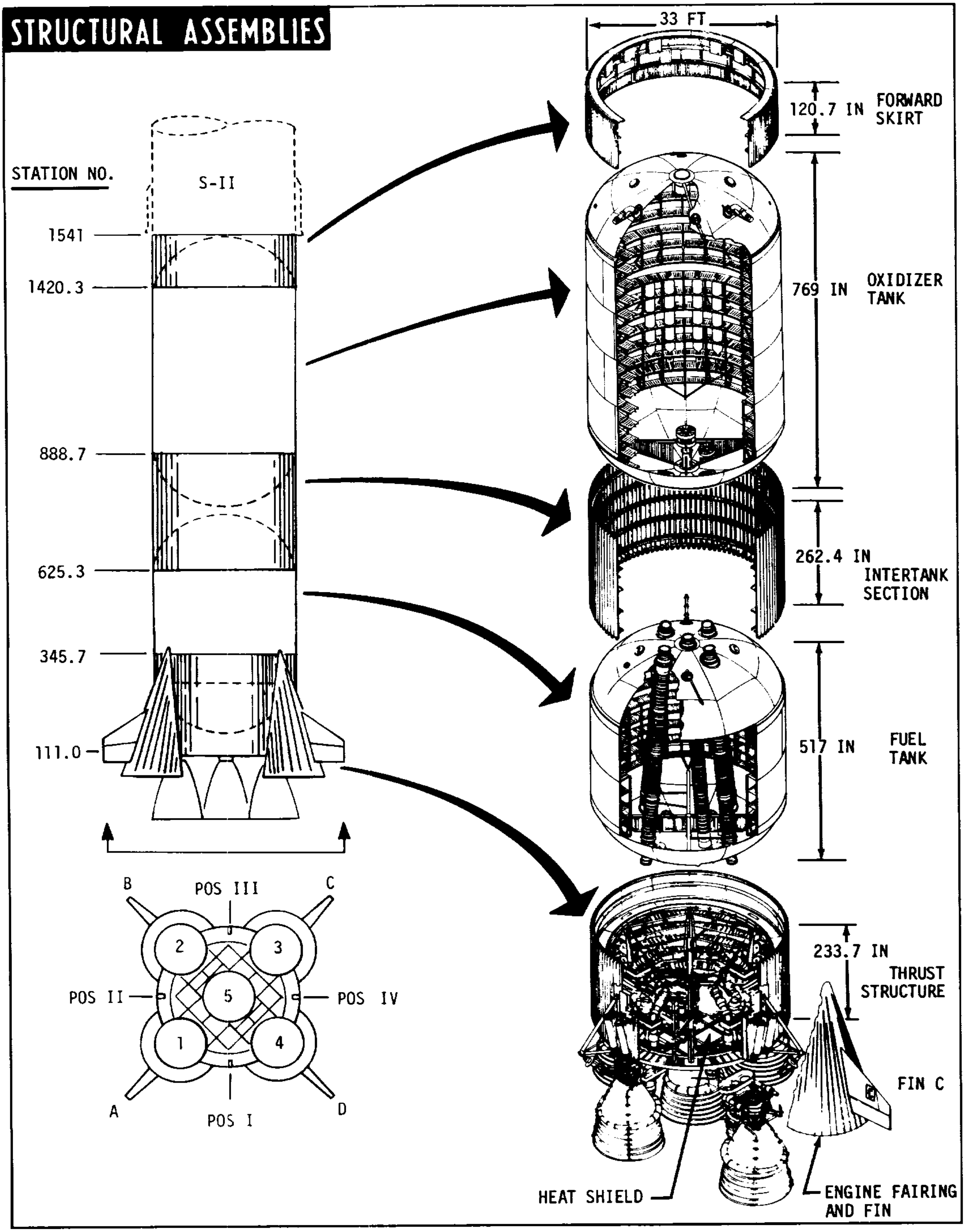

STRUCTURAL ASSEMBLIES
STATION NO.
S-II
1541
1420.3
888.7
625.3
345.7
111.0
B
POS III
C
2
3
POS II
5
POS IV
1
4
A
POS I
D
33 FT
120.7 IN
FORWARD SKIRT
769 IN
OXIDIZER TANK
262.4 IN
INTERTANK SECTION
517 IN
FUEL TANK
233.7 IN
THRUST STRUCTURE
FIN C
HEAT SHIELD
ENGINE FAIRING AND FIN

Figure 4-2

IV

tank, and terminate at the F-1 engines. The fuel tank has an exclusion riser, made of a lightweight foam material, bonded to the lower bulkhead of the tank to minimize unusable residual fuel. The 2219-T87 aluminum skin thickness is decreased in four steps from .193 inches at the aft section to .170 inches at the forward section.

THRUST STRUCTURE

The thrust structure assembly (figure 4-1) redistributes locally applied loads of the five F-1 engines into nearly uniform loading about the periphery of the fuel tank. Also, it provides support for the five F-1 engines, engine accessories, base heat shield, engine fairings and fins, propellant lines, retrorockets, and environmental control ducts. The lower thrust ring has four holddown points which support the fully loaded Saturn V/Apollo (approximately 6,000,000 pounds) and also, as necessary, restrains the vehicle from lifting off at full F-1 engine thrust. The skin segments are fabricated from 7075-T6 aluminum alloy.

The base heat shield is located at the base of the S-IC stage, forward of the engine gimbal plane. The heat shield provides thermal shielding for critical engine components and base region structural components for the duration of the flight. The heat shield panels are constructed of 15-7 PH stainless steel honeycomb 1.00-inch thick, brazed to .010-inch steel face sheets. Each outboard F-1 engine is protected from aerodynamic loading by a conically shaped engine fairing (figure 4-1). The fairings also house the retrorockets and the engine actuator supports. The fairing components are primarily titanium alloy below station 115.5 and aluminum alloy above this station. Four fixed titanium covered stabilizing fins augment the stability of the Saturn V vehicle.

ENVIRONMENT CONTROL

During launch preparations the environmental control system (ECS) protects the S-IC stage and stage equipment from temperature extremes, excessive humidity, and hazardous gases. Conditioned air, provided by the ground support equipment environmental control unit (GSE-ECU), is forced into the forward skirt and thrust structure where it is used as a temperature and humidity control medium. Approximately 20 minutes before the two upper stages are loaded with cryogenic fluids gaseous nitrogen (GN_2) replaces conditioned air and is introduced into the S-IC as the conditioning medium. The GN_2 flow terminates at umbilical disconnect since the system is not needed in flight.

FORWARD SKIRT COMPARTMENT - ECS

The environmental control system distributes air or GN_2 to 10 canisters in the forward skirt. These canisters house electrical or electronic equipment modules. The temperature of the flow medium is controlled by onboard probes to maintain canister temperature at 80 (+20/-20)°F. The input flow compensates for external ambient air change, internal heat generated in the canisters, and the chilling effect of lox. Three phases of conditioning-purge flow are involved during normal launch preparation. The first phase supplies cool, conditioned air to the canister whenever any onboard electrical systems are energized before cryogenic loading. The second phase substitutes relatively warm GN_2 for the cool air at the beginning of lox loading. the third phase continues the GN_2 flow, but at a still warmer temperature, when second stage J-2 engine thrust chamber childown occurs. GN_2, an inert purge gas, reduces concentrations of gaseous oxygen or hydrogen in addition to its temperature control function. The air or GN_2 is vented from the canisters to the compartment where it escapes overboard through various vent openings in the forward skirt of the S-IC stage.

THRUST STRUCTURE COMPARTMENT - ECS

The environmental control system, in the thrust structure, distributes air or GN_2 to 22 orificed duct outlets which discharge directly into the upper structure interior. The GSE-ECU supplies conditioned air at two umbilical couplings during launch preparations until 20 minutes before cryogenic loading. At this time the flow medium is switched to GN_2. Temperature of the flow medium is varied as required to keep the compartment temperature at 80 (+10/-10)°F. The temperature of the flow medium compensates for external ambient air change, and the chilling effect of lox in the suction ducts, prevalves, and interconnect ducts. The GN_2 prevents the oxygen concentration in the compartment from going above 6 percent.

HAZARDOUS GAS DETECTION

The hazardous gas detection system monitors the atmosphere in the forward skirt and the thrust structure compartment of the S-IC (figures 4-3) and 4-4). Samples from these areas are analyzed for dangerous levels of oxygen or hydrogen by ground support equipment. Sampling stops at umbilical disconnect and there are no provisions for hazardous gas detection during flight. This system is not redundant; however, large leaks may be detected by propellant pressure indications displayed in the Launch Control Center.

PROPULSION

The F-1 engine is a single start, 1,526,500-pound fixed thrust, calibrated, bipropellant engine which uses liquid oxygen as the oxidizer and RP-1 as the fuel. Engine features include a bell shaped thrust chamber with a 10:1 expansion ratio, and detachable, conical nozzle extension

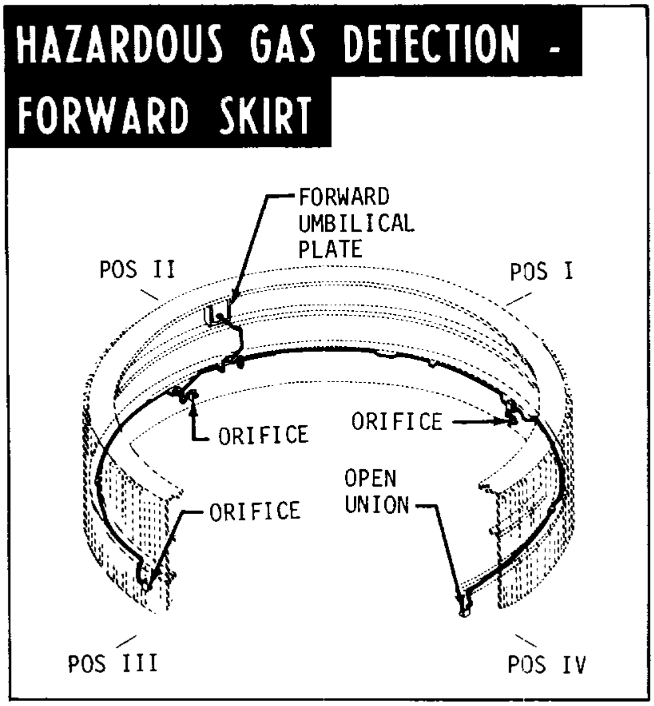


Figure 4-3

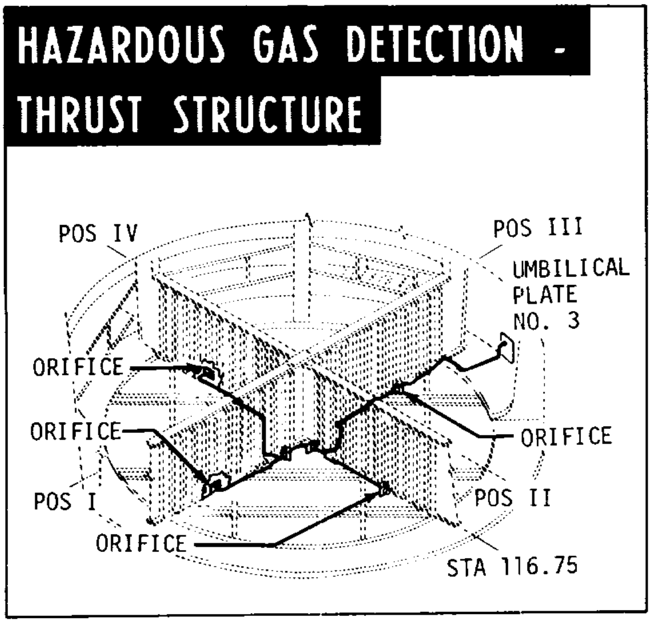


Figure 4-4

which increases the thrust chamber expansion ratio to 16:1. The thrust chamber is cooled regeneratively by fuel, and the nozzle extension is cooled by gas generator exhaust gases. Liquid oxygen and RP-1 fuel are supplied to the thrust chamber by a single turbopump powered by a gas generator which uses the same propellant combination. RP-1 fuel is also used as the turbopump lubricant and as the working fluid for the engine hydraulic control system. The four outboard engines are capable of gimbaling and have provisions for supply and return of RP-1 fuel as the working fluid for a thrust vector control system. The engine contains a heat exchanger system to condition engine supplied liquid oxygen and externally supplied helium for stage propellant tank pressurization. An instrumentation system monitors engine performance and operation. External thermal insulation provides an allowable engine environment during flight operation.

ENGINE REQUIREMENTS

The engine requires a source of pneumatic pressure, electrical power, and propellants for sustained engine operation. A ground hydraulic pressure source, an inert thrust chamber prefill solution, gas generator igniters, gas generator exhaust igniters, and hypergolic fluid are required during the engine start sequence. The engine is started by ground support equipment (GSE) and is capable of only one start before reservicing.

PURGE, PREFILL, AND THERMAL CONDITIONING

A gaseous nitrogen purge is applied for thermal conditioning and elimination of explosive hazard under each engine cocoon. Because of the possibility of low temperatures existing in the space between the engine and its cocoon of thermal insulation, heated nitrogen is applied to this area. This purge is manually operated, at the discretion of launch operations, whenever there is a prolonged hold of the countdown with lox onboard and with an ambient temperature below approximately 55°F. In any case, the purge will be turned on five minutes prior to ignition command and continue until umbilical disconnect.

A continuous nitrogen purge is required to expel propellant leakage from the turbopump lox seal housing and the gas generator lox injector. The purge pressure also improves the sealing characteristic of the lox seal. The purge is required from the time propellants are loaded and is continuous throughout flight.

A nitrogen purge prevents contaminants from accumulating on the radiation calorimeter viewing surfaces. The purge is started at T-52 seconds and is continued during flight.

A gaseous nitrogen purge is required to prevent contaminants from entering the lox system through the engine lox injector or the gas generator lox injector. The purge system is activated prior to engine operation and is continued until umbilical disconnect.

The thrust chamber tubes and fuel manifold are filled with an ethylene glycol/water solution prior to engine start. This inert fluid smoothes out the start sequence.

Ethylene glycol solution is supplied to the thrust chambers of all five engines. At approximately T-6 hours, 550 to 600 gallons of ethylene glycol solution is supplied to fill the five engines, with about five gallons overflow. Flow is terminated by a signal from an observer at the engines. At approximately T-5 minutes, 50 gallons are supplied to top off the system to compensate for liquid loss during engine gimbaling.

ENGINE SUBSYSTEMS

The subsystems of the F-1 engine shown in figure 4-5 are the turbopump, checkout valve, hypergol manifold, heat exchanger, main fuel valve and main lox valve. Subsystems not shown are the gas generator, 4-way control valve, and pyrotechnic igniters.

Hypergol Manifold

The hypergol manifold consists of a hypergol container, an ignition monitor valve (IMV), and an igniter fuel valve (IFV). The hypergol solution is forced into the thrust chamber by the fuel where combustion is initiated upon mixing with the lox. The IFV prevents thrust chamber ignition until the turbopump pressure has reached 375 psi. The IMV prevents opening of the main fuel valves prior to hypergolic ignition. Position sensors indicate if the hypergol cartridge has been installed. The loaded indication is a prerequisite to the firing command.

Control Valve-4 Way

The 4-way control valve directs hydraulic fluid to open and close the fuel, lox, and gas generator valves. It consists of a filter manifold, a start and stop solenoid valve, and two check valves.

Turbopump

The turbopump is a combined lox and fuel pump

F-1 ENGINE MAJOR COMPONENTS

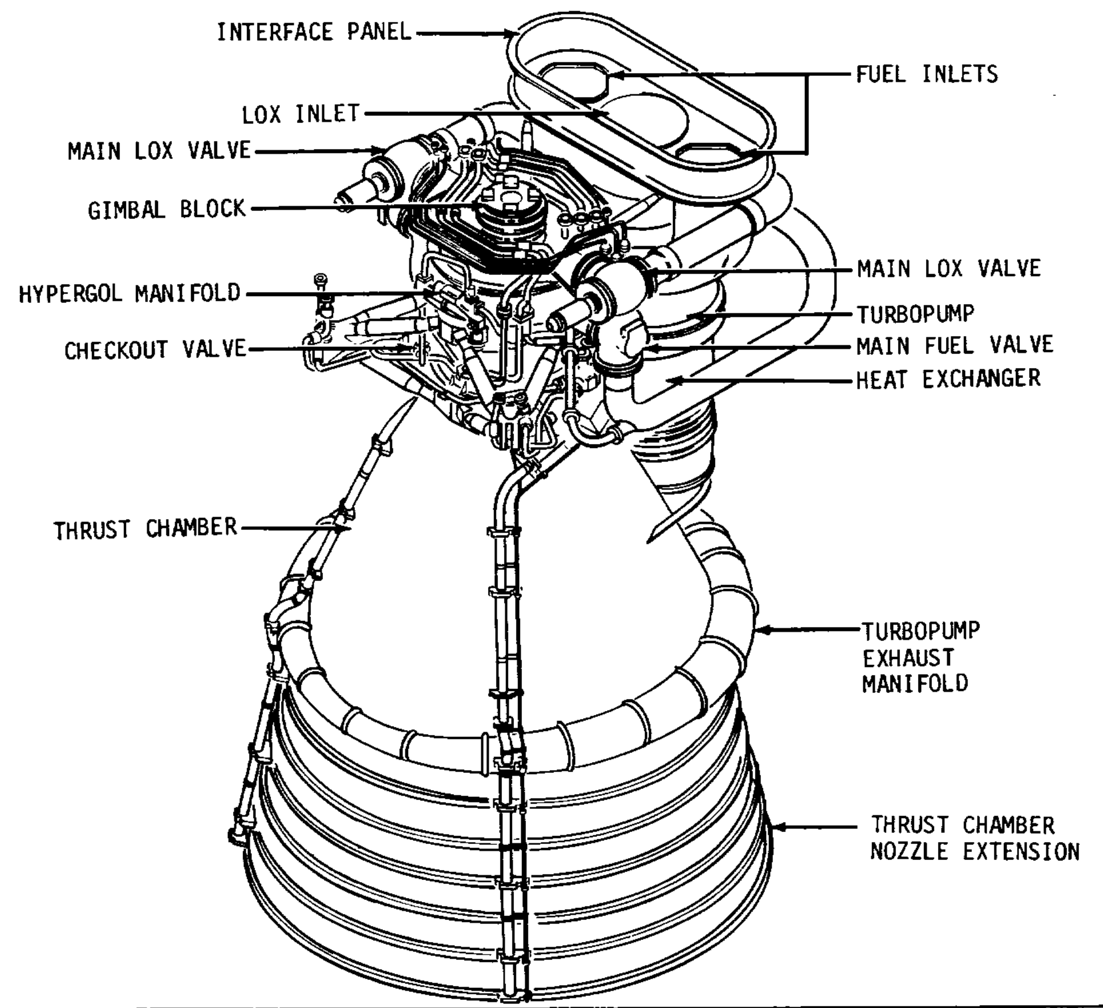


Figure 4-5

driven through a common shaft by a single gas turbine.

Gas Generator

The gas generator (GG) provides the gases for driving the turbopump. Its power output is controlled by orifices in its propellant feed lines. The gas generator system consists of a dual ball valve, an injector, and a combustor body. Combustion is initiated by two pyrotechnic igniters. Total propellant flow rate is approximately 170 lb/sec at a lox/RP-1 mixture ratio of 0.42:1. (Note that these propellants are the same propellants used in the F-1 engines). The dual ball valve must be closed prior to fuel loading and must remain closed to meet an interlock requirement for engine start.

Heat Exchanger

The heat exchanger expands lox and cold helium for propellant tank pressurization. The cold fluids, flowing through separate heating coils, are heated by the turbopump exhaust. The warm expanded gases are then routed from the heating coils to the propellant tanks.

Main Fuel Valve

There are two main fuel valves per engine. They control flow of fuel to the thrust chamber. The main fuel valve is a fast acting, pressure balanced, hydraulically operated, poppet type valve. Movement of the poppet actuates a switch which furnishes valve position signals to the telemetry system. This valve is designed to remain open, at rated engine pressures and flowrates, if the opening control pressure is lost. Both valves must be in the closed position prior to fuel loading or engine start.

Main Lox Valve

The two main lox valves on each engine control flow to the thrust chamber. These valves are fast acting, pressure balanced, poppet type, hydraulically operated valves. A sequence valve operated by the poppet allows opening pressure to be applied to the GG valve only after both main lox valve poppets have moved to a partially open position. This valve is designed to remain open, at rated engine pressures and flowrates, if the opening control pressure is lost. Both main lox valves must be in the closed position prior to lox loading or engine start.

Checkout Valve

The checkout valve directs ground supplied control fluid from the engine back to ground during engine checkout. Approximately 23 seconds prior to the firing command the valve is actuated to the engine position. In this position it directs control fluid to the No. 2 turbopump inlet. An ENGINE POSITION indication is required from this valve prior to, and is interlocked with, forward umbilical disconnect command.

High Voltage Igniters

Four high voltage igniters, two in the gas generator (GG) body and two in the engine thrust chamber nozzle extension ignite the GG and the fuel rich turbopump exhaust gases. They are ignited during the F-1 engine start sequence by application of a nominal 500 volts to the igniter squibs.

ENGINE OPERATION

Engine operation is illustrated in figure 4-6 in terms of engine start sequence and transition to mainstage for a typical single engine.

IV

Figure 4-7 lists the times the start signal is initiated to approximate a 1-2-2 engine start sequence. This sequence allows the engines to attain 100 psig thrust chamber pressure at the times shown.

ENGINE CUTOFF

The normal inflight cutoff sequence is center engine first, followed by the four outboard engines. The center engine optical type lox depletion sensor initiates the center engine cutoff sequence. This command also initiates time base No. 2 (T_2 + 0.0). The outboard engines are also cutoff by optical type lox depletion sensors. The outboard engine cutoff signal initiates time base No. 3 (T_3 + 0.0). The center engine uses an optical type lox depletion sensor as backup while the outboard engines use an optical type fuel sensor as backup. 6

At T_2 + 9.8 seconds, the LVDC supplies a command to the switch selector to enable outboard engines cutoff circuitry. When two or more of the four lox level sensors are energized a timer is started. Expiration of the timer energizes the 4-way control valve stop solenoid on each outboard engine. The remaining shutdown sequence of the outboard engines is the same as for the center engine which is explained in figure 4-8.

EMERGENCY ENGINE CUTOFF

In an emergency, the engine will be cut off by any of the following methods: Ground Support Equipment (GSE) Command Cutoff, Range Safety Command Cutoff, Thrust Not OK Cutoff, Emergency Detection System, Outboard Cutoff System.

GSE has the capability of initiating engine cutoff anytime until umbilical disconnect. Separate command lines are supplied through the aft umbilicals to the engine cutoff relays and prevalve close relays.

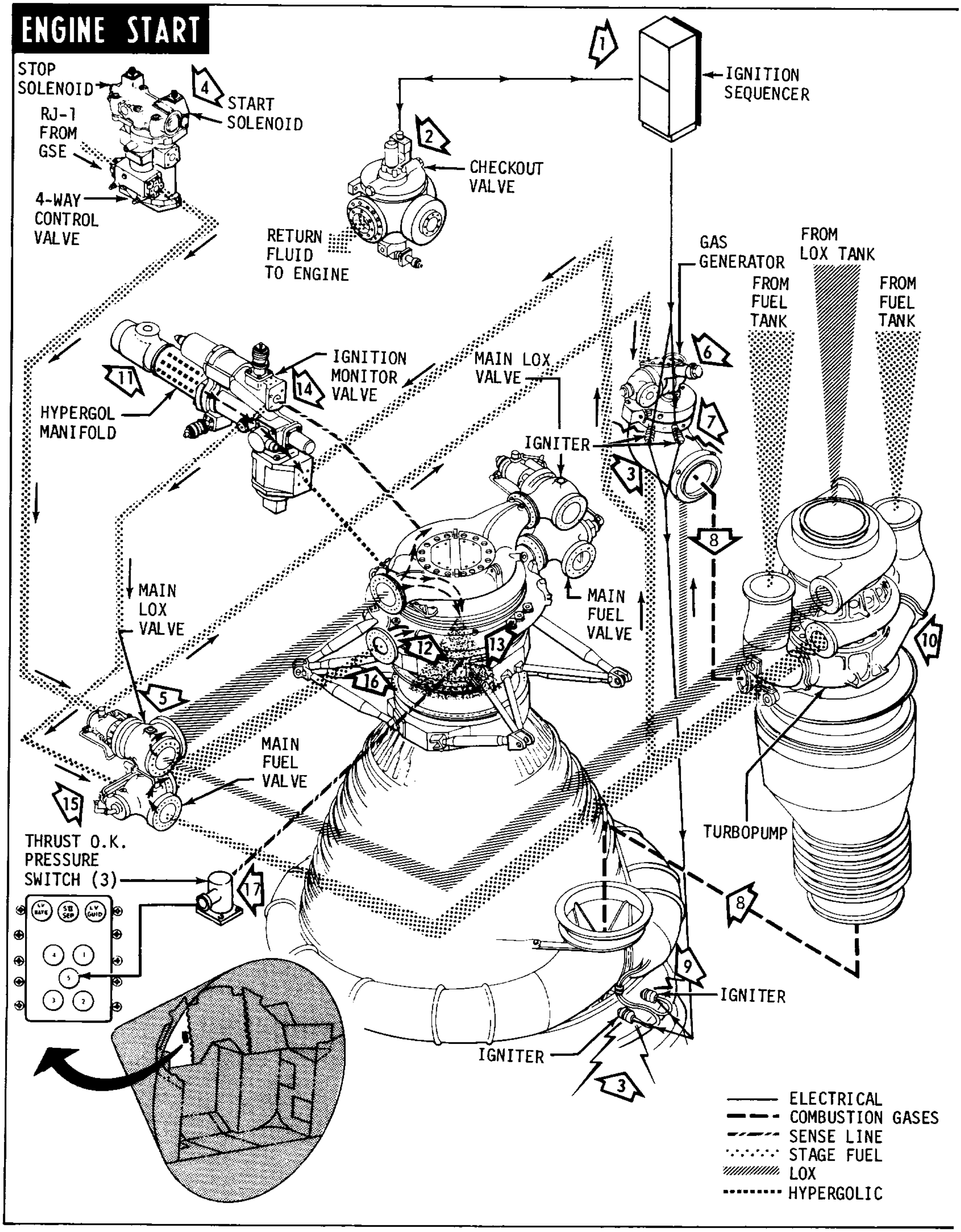


Figure 4-6 (Sheet 1 of 2)

1 Engine start is part of the terminal countdown sequence. When this point in the countdown is reached, the ignition sequencer controls starting of all five engines.

2 Checkout valve moves to engine return position.

3 Electrical signal fires igniters (4 each engine).

a) Gas generator combustor and turbine exhaust igniters burn igniter links to trigger electrical signal to start solenoid of 4-way control valve.

b) Igniters burn approximately six seconds.

4 Start solenoid of 4-way control valve directs GSE hydraulic pressure to main lox valves.

5 Main lox valves allow lox to flow to thrust chamber and GSE hydraulic pressure to flow through sequence valve to open gas generator ball valve.

6 Propellants, under tank pressure, flow into gas generator combustor.

7 Propellants are ignited by flame of igniters.

8 Combustion gas passes through turbopump, heat exchanger, exhaust manifold and nozzle extension.

9 Fuel rich turbine combustion gas is ignited by flame from igniters.

a) Ignition of this gas prevents backfiring and burping.

b) This relatively cool gas (approximately 1,000°F) is the coolant for the nozzle extension.

10 Combustion gas accelerates the turbopump, causing the pump discharge pressure to increase.

11 As fuel pressure increases to approximately 375 psig, it ruptures the hypergol cartridge.

12 The hypergolic fluid and fuel are forced into the thrust chamber where they mix with the lox to cause ignition.

TRANSITION TO MAINSTAGE

13 Ignition causes the combustion zone pressure to increase.

14 As pressure reaches 20 psig, the ignition monitor valve directs fluid pressure to the main fuel valves.

15 Fluid pressure opens main fuel valves.

16 Fuel enters thrust chamber. As pressure increases the transition to mainstage is accomplished.

17 The thrust OK pressure switch (which senses fuel injection pressure) picks up at approximately 1060 psi and provides a THRUST OK signal to the IU.

Figure 4-6 (Sheet 2 of 2)

APPROXIMATE F-1 ENGINE START TIME			
ENGINE NUMBER	SERIAL NUMBER	INITIATE START (SECONDS)	100 PSIG THRUST CHAMBER PRESSURE ATTAINED (SECONDS)
1	F-5029	T-6.0	T-2.7
2	F-5032	T-5.7	T-2.4
3	F-5031	T-6.1	T-2.7
4	F-5033	T-6.1	T-2.4
5	F-5030	T-6.3	T-3.0
NOTE: "T" TIMES ARE BASED ON T-0 AT LAUNCH COMMIT.			

Figure 4-7

Range Safety Cutoff has the capability of engine cutoff anytime after liftoff. If it is determined during flight that the vehicle has gone outside the established corridor, the Range Safety Officer will send commands to effect engine cutoff and propellant dispersion.

Three thrust OK pressure switches are located on each F-1 engine thrust chamber fuel manifold and sense main fuel injection pressure. If the pressure level drops below the deactivation level of two of the three pressure switches, an engine cutoff signal is initiated. The circuitry is enabled at T_1 + 14.0 seconds to allow the vehicle to clear the launch pad.

Prior to its own deactivation, the Emergency Detection System (EDS) initiates engine cutoff when it is determined that two or more engines have shutdown prematurely. When the IU receives signals from the thrust OK logic relays that two or more engines have shutdown, the IU initiates a signal to relays in the S-IC stage to shutdown the remaining engines. S-IC engine EDS cutoff is enabled at T_1 + 30.0 seconds and continues until 0.8 second before center engine cutoff or until deactivated.

Following EDS deactivation, the outboard cutoff system is activated 0.7 second before center engine cutoff, and is similar in function to the EDS. Whereas the EDS initiates emergency engine cutoff when any two engines are shutdown, the outboard cutoff system monitors only the outboard engines and provides outboard engine cutoff if the thrust OK pressure switches cause shutdown of two adjacent outboard engines.

NOTE

Loss of two adjacent outboard engines, after center engine cutoff, could cause stage breakup.

ENGINE CUTOFF

1. The 4-Way Control Valve Stop Solenoid is energized, which routes closing pressure to the following valves.
2. Gas Generator Ball Valve closes.
3. Main Lox Valves (2 ea) close.
4. Main Fuel Valves (2 ea) close.
5. The Thrust Chamber pressure decay causes the thrust OK pressure switch to drop out (3 ea).
6. Ignition Fuel Valve and Ignition Monitor Valve closed.

Figure 4-8 (Sheet 1 of 2)

FLUID POWER

The S-IC fluid power system (figure 4-9) supplies power to operate the engine valves and the thrust vector control system. The fluid power system consists of two parts: the ground hydraulic supply system using RJ-1 fluid, and the stage hydraulic supply system using RP-1 fuel.

Each supply also provides a warming flow through drilled passages in each of the fluid system valves.

The ground hydraulic supply system provides hydraulic pressure to gimbal the engines during test and pre-launch checkout and engine start. The flight hydraulic supply system provides hydraulic pressure for flight control and S-IC engine valve actuation and also prevents main lox valves from freezing.

FLIGHT CONTROL SYSTEM

The S-IC thrust vector control consists of four outboard F-1 engines, gimbal blocks to attach these engines to the thrust ring, engine hydraulic servoactuators (two per engine), and an engine hydraulic power supply.

Engine thrust is transmitted to the thrust structure through the engine gimbal block. Direction of the engine is changed by applying force to the servoactuator attach points. There are two servoactuator attach points per engine, located 90 degrees from each other. This gimbaling of the four outboard engines changes the

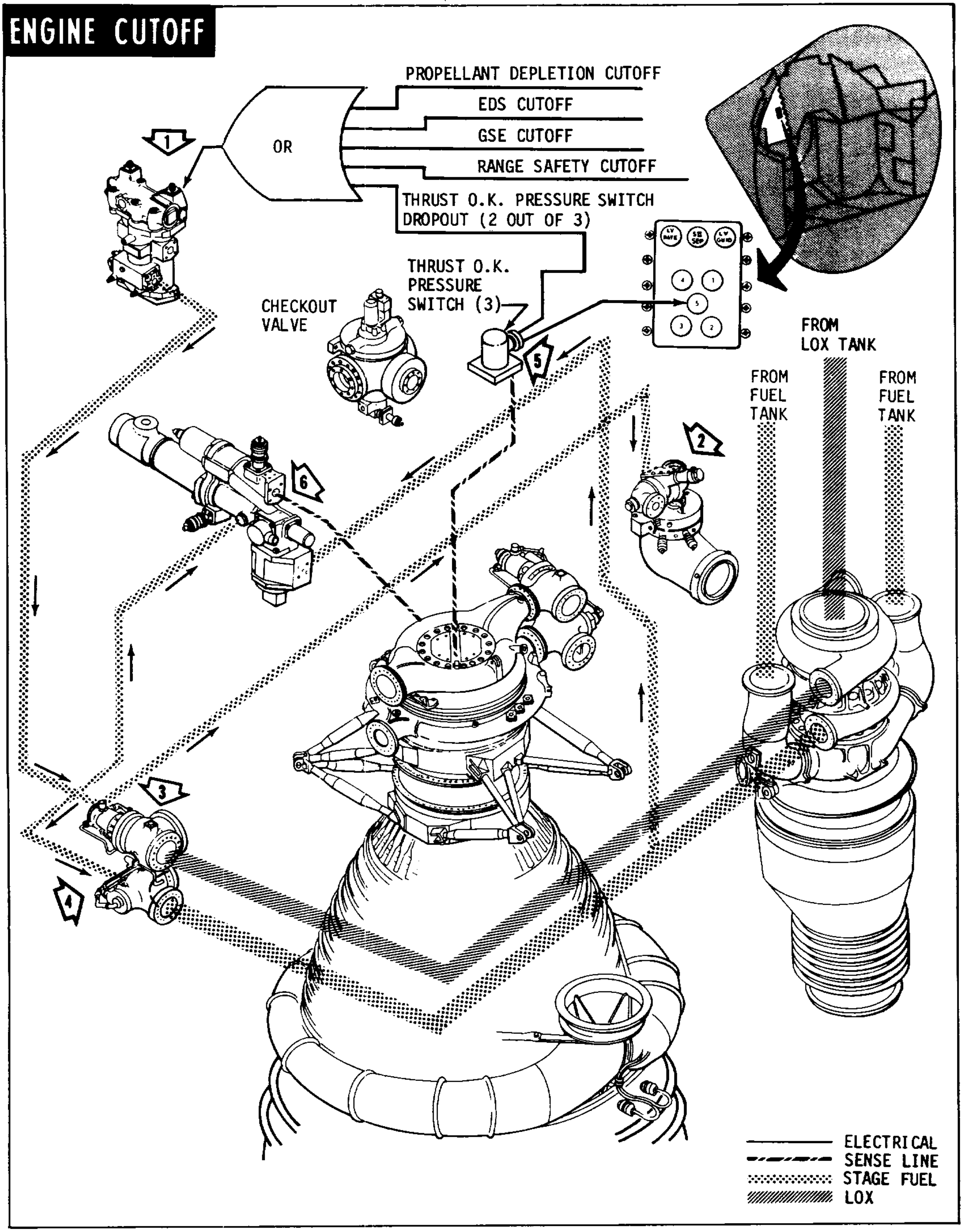


Figure 4-8 (Sheet 2 of 2)

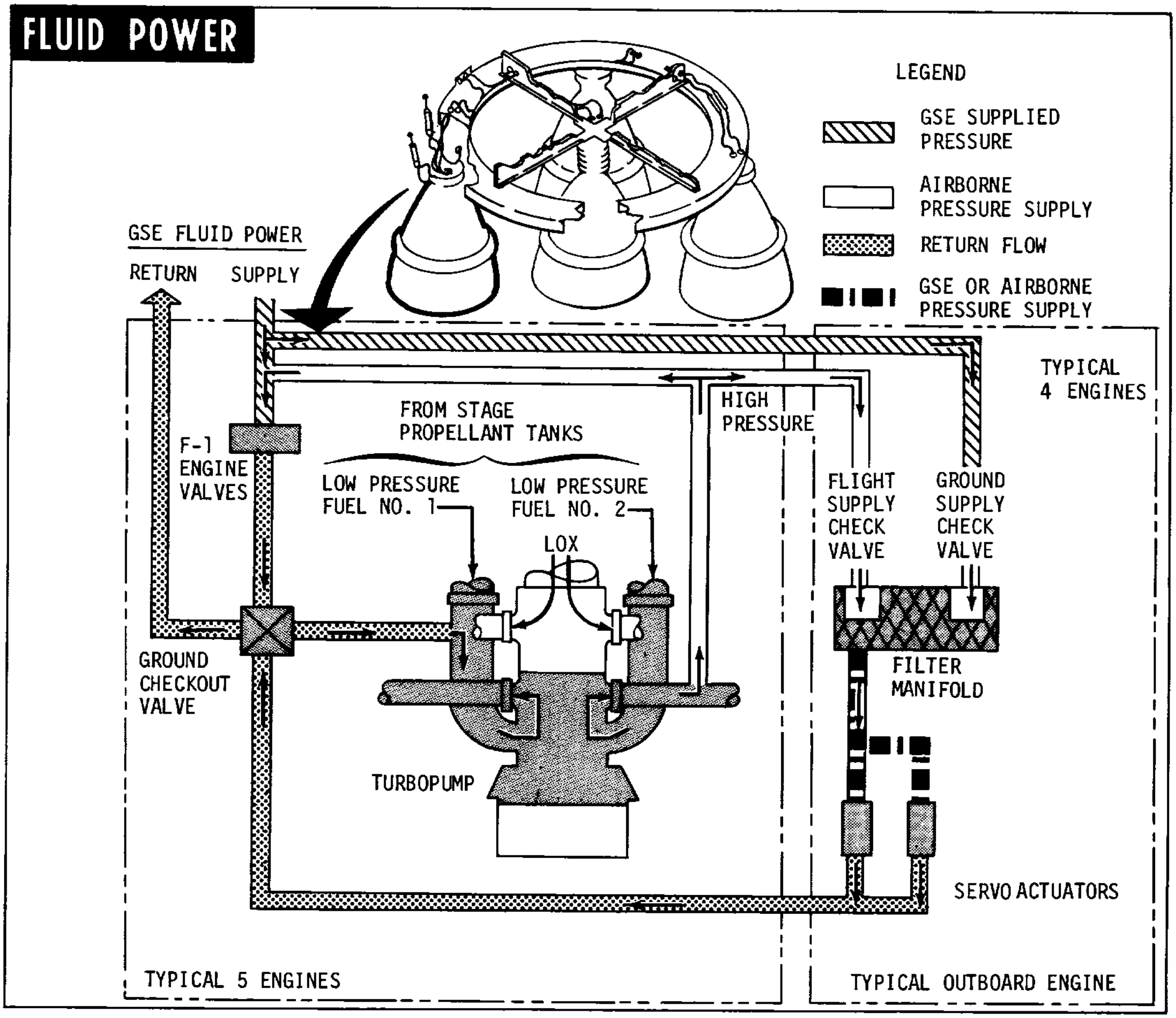


Figure 4-9

direction of thrust and as a result corrects the attitude of the vehicle to correspond to the desired trajectory.

HYDRAULIC SERVOACTUATOR

The servoactuator (figure 4-10) is the power control unit for converting electrical command signals and hydraulic power into mechanical outputs to gimbal the engines on the S-IC stage. It is controlled by the flight control computer in the instrument unit (IU). The flight control computer receives inputs from the guidance system in the IU and then sends signals (1) to the servoactuators to gimbal the outboard engines in the direction and magnitude required. An integral mechanical feedback (3), varied by piston (2) position, modifies the effect of the IU control signal. A built in potentiometer, not part of the actuator control circuit, senses the servoactuator position and continuously sends this information to the IU for transmission back to ground via telemetry.

CONTROL PRESSURE

The control pressure system, shown in figure 4-11, provides a pressurized nitrogen supply for command operation of various pneumatic valves.

Pneumatic control of the fuel fill and drain valve (2), the lox fill and drain valve (1), and the No. 2 lox interconnect valve (3), is provided directly from GSE. The onboard system, for these valves, consists only of the pressure lines from the stage umbilicals and the valves. The onboard pressure system controls the lox interconnect valves No. 1, 3, and 4 (13) during prelaunch operations.

HYDRAULIC SERVOACTUATOR

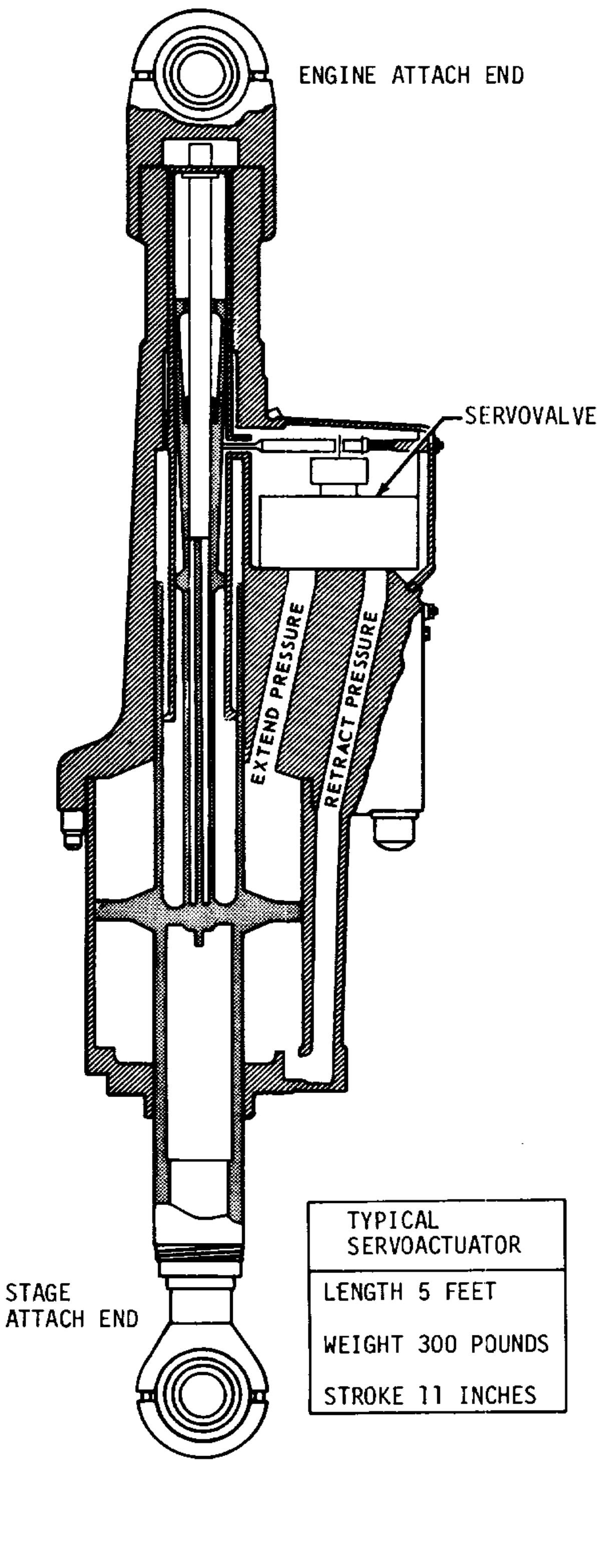


Figure 4-10

The control pressure system, for those valves which must be controllable during flight [fuel prevalves (4), lox prevalves (5), fuel vent valve (6), and lox vent valves (7), (8)], is charged by a high pressure (3200 psi) GSE nitrogen source. The system is charged through onboard filters (9), solenoid valve (10), storage bottle (11), and a pressure regulator (12) which reduces system pressure to 750 psi. For emergency engine shutdown, prior to liftoff, there are direct lines from the GSE to the prevalve solenoid valves. Orifices in the fuel prevalve lines and in the lox prevalves control closing times of the prevalves.

ATTITUDE CONTROL

Refer to Section VII for attitude control of the S-IC and the launch vehicle.

PROPELLANTS

The S-IC stage fuel system delivers RP-1 to the five F-1 engines. The fuel system includes hardware for fuel fill and drain operations, tank pressurization prior to and during flight, and delivery of fuel to the engines. Figure 4-12 is a simplified block diagram of the system and figure 4-13 shows the functional relationship of the system components. The propellant loading sequence is shown in figure 4-14.

RP-1 FILL AND DRAIN

Fuel loading (figure 4-13) starts approximately 84 hours before liftoff. RP-1 is loaded through a 6-inch fill and drain valve (2). The total mass loaded is controlled by the fuel loading probe (1) mounted in the fuel tank. At T-20 minutes the propellant management GSE gives the command to begin fuel level adjust to the predetermined flight level. The RP-1 tank has been purposely overfilled to approximately 102%. Therefore, this command will initiate a limited drain. When it is necessary to drain RP-1 the fuel vent and relief valve (3) will be opened for gravity drain or closed for pressurized drain.

RP-1 Pressurization

Fuel tank pressurization (figure 4-13) is required during engine starting and flight to establish and maintain a net positive suction head at the fuel inlet to the engine turbopumps. Ground supplied helium (5) for prepressurization is introduced into the cold helium line downstream from the flow controller (6), resulting in helium flow through the engine heat exchanger and the hot helium line to the fuel tank distributor (9). During flight, the source of fuel tank pressurization is helium from storage bottles (7) mounted inside the lox tank. The high pressure helium is routed through the flow controller (6), cold helium supply line, engine

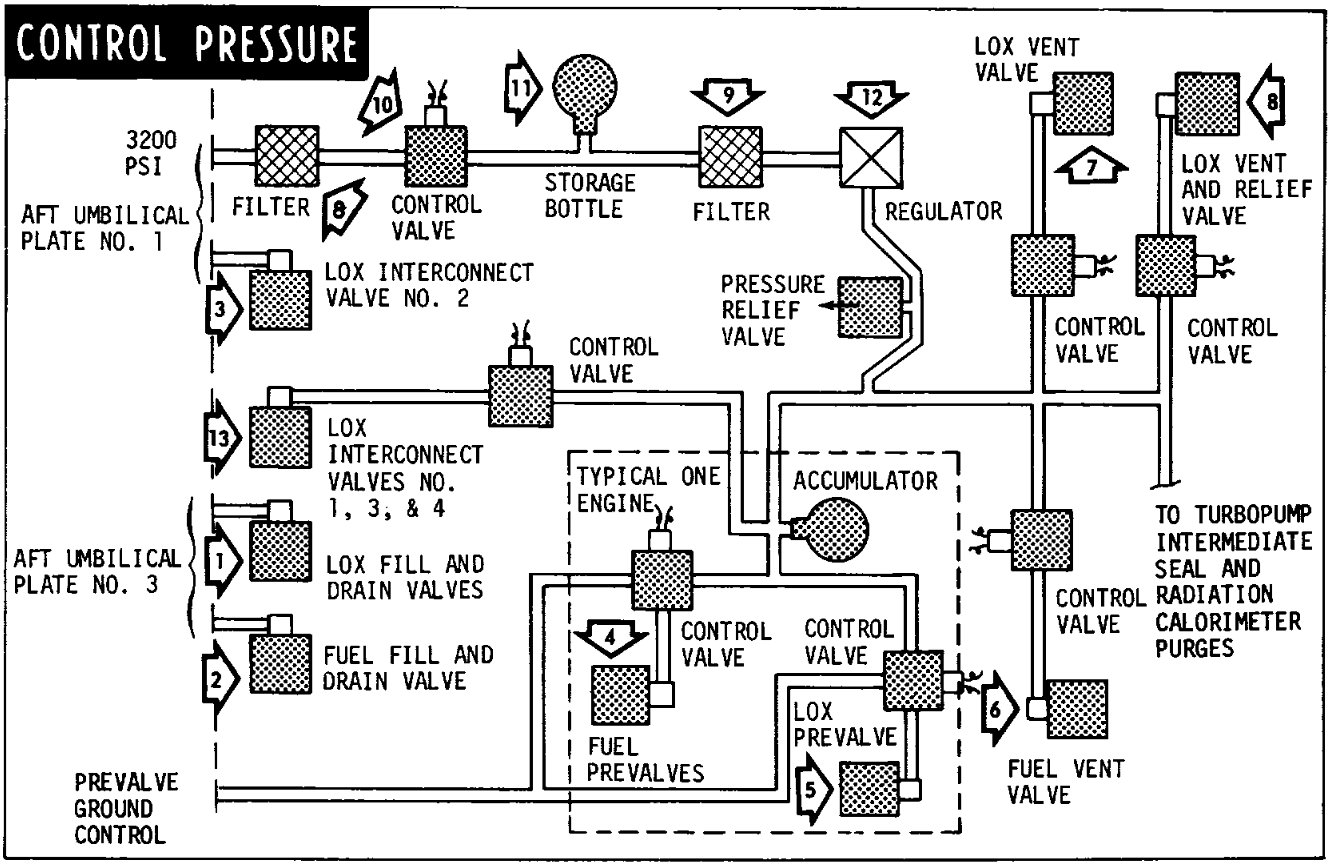


Figure 4-11

heat exchanger (8) and hot helium line to the fuel tank distributor (9).

Fuel tank pressure switches (10) control the fuel vent and relief valve, the GSE pressure supply during filling operations, prepressurization before engine ignition, and pressurization during flight. The flight pressurization pressure switch (10) actuates one of the five control valves (11) in the flow controller to ensure a minimum pressure of 24.2 psia during flight. The other four valves (11) are sequenced by the IU to establish an adequate helium flow rate with decreasing storage bottle pressure.

The onboard helium storage bottles are filled through a filtered fill and drain line (12) upstream from the flow controller. The storage bottles are filled to a pressure of 1600 psi prior to lox loading. Fill is completed to 3150 psi after lox loading when the bottles are cold.

RP-1 Delivery

Fuel feed (figure 4-13) is accomplished through two 12-inch ducts which connect the fuel tank to each F-1 engine. The ducts are equipped with gimbaling and sliding joints to compensate for motions from engine gimbaling and stage stresses. Prevalves (4), one in each fuel line, serve as an emergency backup to the main engine fuel shutoff valves. The prevalves also house flowmeters which provide flowrate data via telemetry to GSE. A fuel level engine cutoff sensor (15), in the bottom of the fuel tank, initiates engine shutdown when fuel is depleted if the lox sensors have failed to cut the engines off first.

LOX SYSTEM

Lox is used as the oxidizer in the S-IC stage bipropellant propulsion system. The lox system includes hardware for lox fill and drain, lox conditioning, and tank pressurization, prior to and during flight, and delivery of lox to the engines. Figure 4-15 is a simplified block diagram of the system and figure 4-16 shows the functional relationships of the system components.

Lox Fill and Drain

The 345,000-gallon lox tank (figure 4-16) is filled through two 6-inch fill and drain lines (3) connected to the bottom of the tank. Three different fill rates are used; a 300-gallon per minute rate to chilldown the tank, a 1500-gallon per minute slow fill rate to establish a stable liquid level and thus prevent structural damage, and a fast fill rate of 10,000 gallons per minute. When the tank is nearly full, the fill rate is again reduced to 1500 gallons per minute until the lox loading level sensor auto-

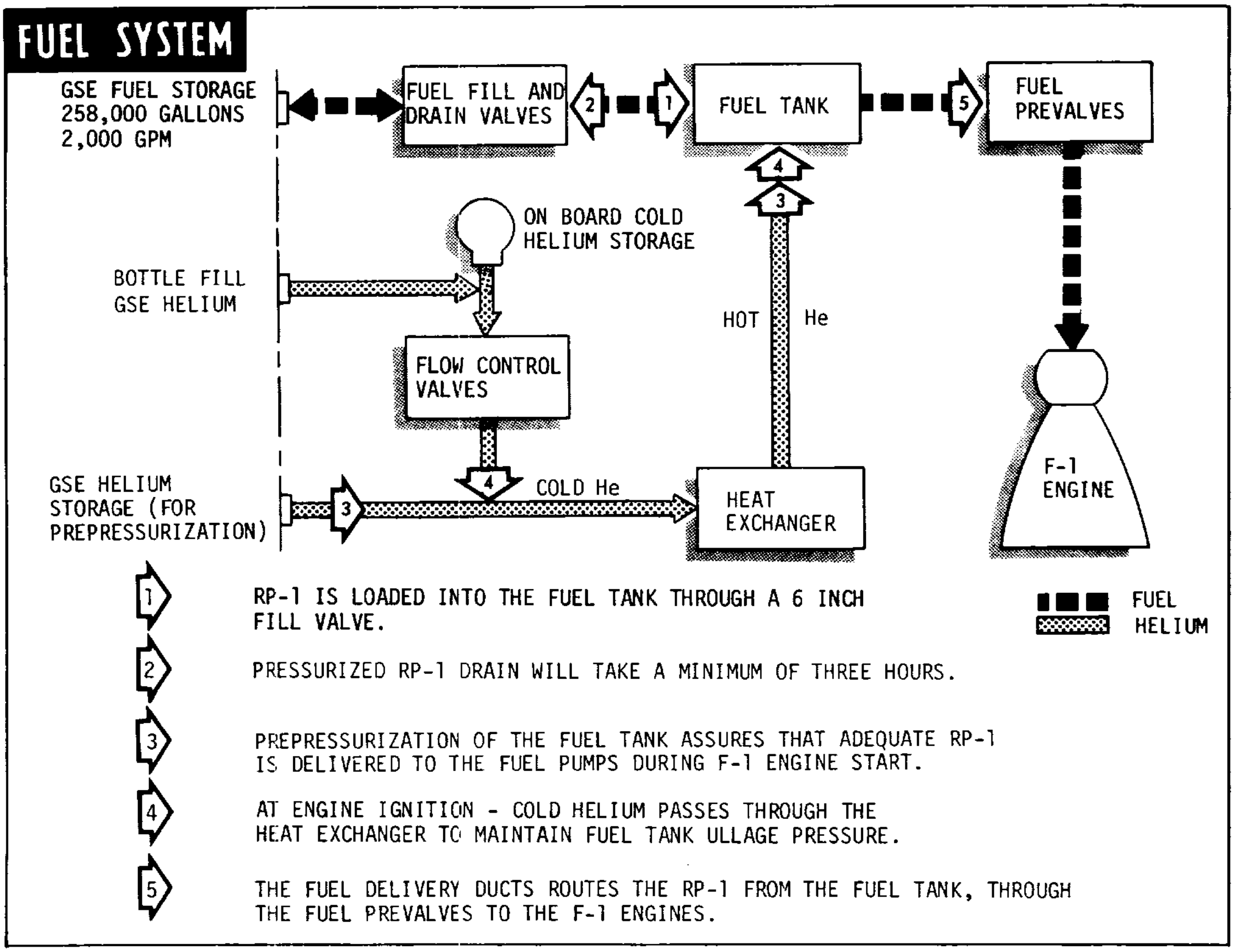


Figure 4-12

matically stops the fill mode. Lox boiloff is replenished at 500 gpm.

The lox is drained through the two fill and drain lines and a lox suction duct drain line (4) in the thrust structure. The total drain capability is 7,500 gpm. During lox drain, positive ullage pressure is maintained by a GSE pressure source and two vent valves which are kept closed except when overpressure occurs.

Prior to launch, boil off in the lox tank may be harmlessly vented overboard; however, excessive geysering from boiling in the lox suction ducts can cause structural damage, and high lox temperatures near the engine inlets may prevent normal engine start. The lox bubbling system (5) eliminates geysering and maintains low pump inlet temperatures. The helium induced convection currents circulate lox through the suction ducts and back into the tank. Once established, thermal pumping is self sustaining and continues until the interconnect valves (13) are closed just prior to launch.

Lox Pressurization System

Lox tank pressurization (figure 4-16) is required to ensure proper engine turbopump pressure during engine start, thrust buildup, and fuel burn. The pressurization gas, prior to flight (prepressurization), is helium. It is supplied at 240 lbs/min for forty seconds to the gox distributor (6) in the lox tank. Lox tank pressure is monitored by three pressure switches (7). They control the GSE pressure source, the lox vent valve (2), and the lox vent and relief valve (1) to maintain a maximum of 25 psig ullage pressure. Prepressurization is maintained until T-3 seconds and gox is used for pressurizing the lox tank during flight. A portion of the lox supplied to each engine is diverted from the lox dome into the engine heat exchanger (8) where the hot turbine exhaust transforms lox into gox. The heated gox is delivered through the gox pressurization line and a flow control valve (9) to the gox distributor (6) in the lox tank.

FUEL SUBSYSTEMS

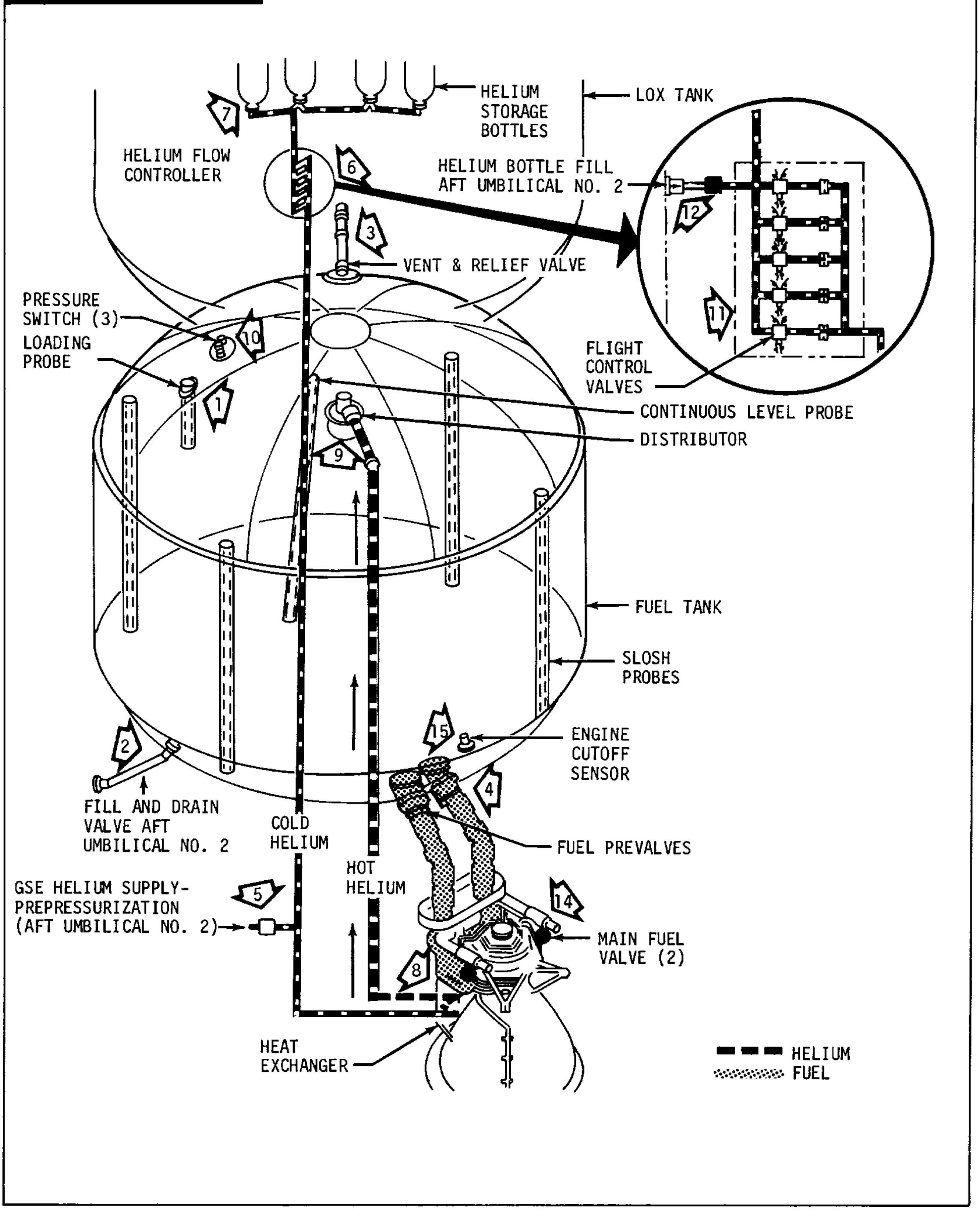


Figure 4-13

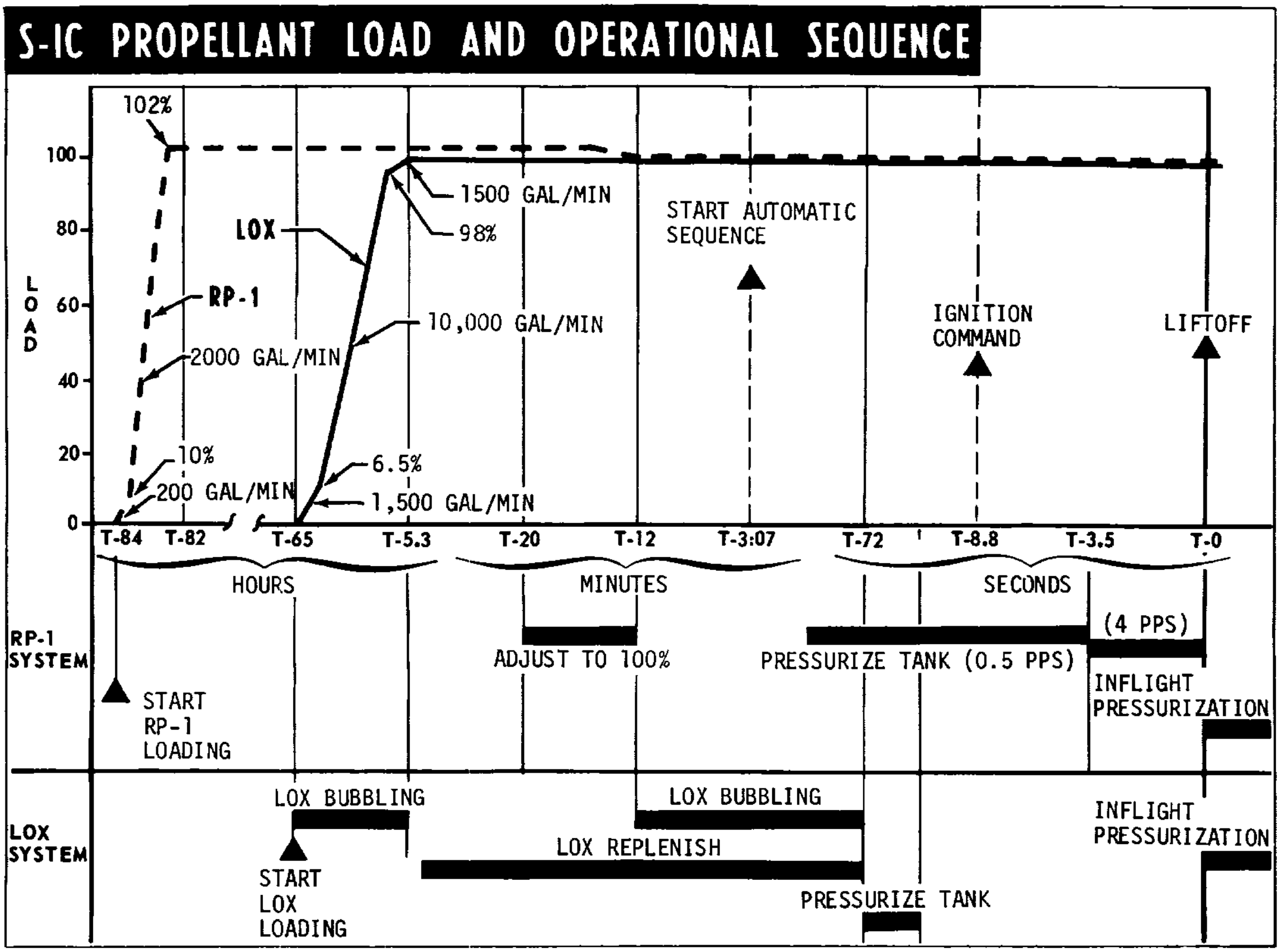


Figure 4-14

A sensing line provides pressure feedback to the flow control valve to regulate the gox flow rate and maintain ullage pressure between 18 and 20 psia.

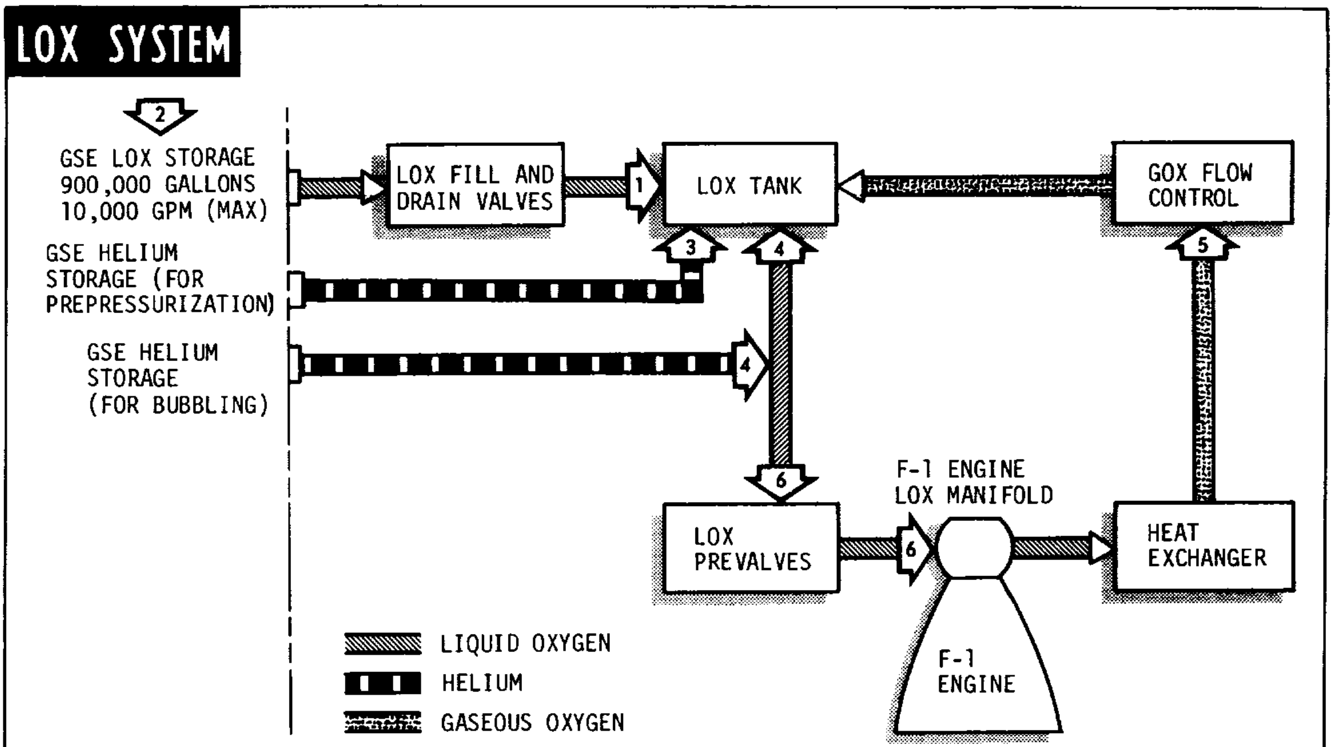
Lox Delivery

Lox is delivered to the engines through five suction lines (figure 4-16). The ducts are equipped with gimbals and sliding joints (10) to compensate for motions from engine gimbaling and stage stresses. Pressure volume compensating ducts (10) ensure constant lox flowrate regardless of the gimbaled position of the engine. Lox level engine cutoff sensors (11), mounted in the upper portion of each suction line, initiate engine shutdown at lox depletion, which normally occurs before fuel depletion. It goes from wet to dry first and initiates a center engine (No. 5) shutdown command as well as enabling a two out of four voting logic which activates the outboard engine shutdown sequence. When the lox level passes any two of the four outboard sensors, the outboard engines are shutdown. The shutdown events are set so that a safe residual of lox remains in the suction lines at engine shutdown. Each suction line has a lox prevalve (12) which is a backup to the engine lox valve (14).

ELECTRICAL

The electrical power system of the S-IC stage is made up of two basic subsystems: the operational power subsystem and the measurements power subsystem. Onboard power is supplied by two 28 volt batteries located as shown in figure 4-17. Battery characteristics are listed in figure 4-18.

In figure 4-19, power distribution diagram, battery number 1 is identified as the operational power system battery. It supplies power to operational loads such as valve controls, purge and venting systems, pressurization systems, and sequencing and flight control. Battery number 2 is identified as the measurement power system battery. It supplies power to measurements loads such as telemetry systems, transducers, multiplexers, and transmitters. Both batteries supply power to their loads through a common main power distributor but each system is completely isolated from the other.



1. LOX IS LOADED INTO THE LOX TANK THROUGH TWO 6 INCH FILL VALVES
2. THE THREE DRAIN VALVES HAVE A COMBINED MAXIMUM DRAIN RATE OF 7,500 GAL/MIN
3. PREPRESSURIZATION OF THE LOX TANK ASSURES THAT ADEQUATE LOX IS DELIVERED TO THE LOX PUMPS DURING F-1 ENGINE START.
4. THE HELIUM BUBBLING SYSTEM INDUCES GOX AND/OR LOX CIRCULATION THROUGH THE SUCTION DUCTS AND BACK INTO THE TANK.
5. LOX TANK PRESSURE IS MAINTAINED BY GOX THAT HAS BEEN GENERATED IN THE HEAT EXCHANGER FROM LOX.
6. THE LOX DELIVERY DUCTS ROUTES THE LOX FROM THE LOX TANK, THROUGH THE LOX VALVES TO THE F-1 ENGINES.

Figure 4-15

During the prelaunch checkout period power for all electrical loads, except range safety receivers, is supplied from GSE. The range safety receivers are hardwired to batteries 1 and 2 in order to enhance the safety and reliability of the range safety system. At T-50 seconds a ground command causes the power transfer switch to transfer the S-IC electrical loads to onboard battery power. However, power for engine ignition and for equipment heaters (turbopump and lox valves) continues to come from the GSE until umbilical disconnect.

DISTRIBUTORS

There are six power distributors on the S-IC stage. They facilitate the routing and distribution of power and also serve as junction boxes and as housing for relays, diodes, switches and other electrical equipment.

There are no provisions for switching or transferring power between the operational power distribution system and the measurement power system. Because of this isolation, no failure of any kind in one system can cause equipment failure in the other system.

Main Power Distributor

The main power distributor contains a 26 pole power transfer switch, relays, and the electrical distribution busses. It serves as a common distributor for both operational and mea-

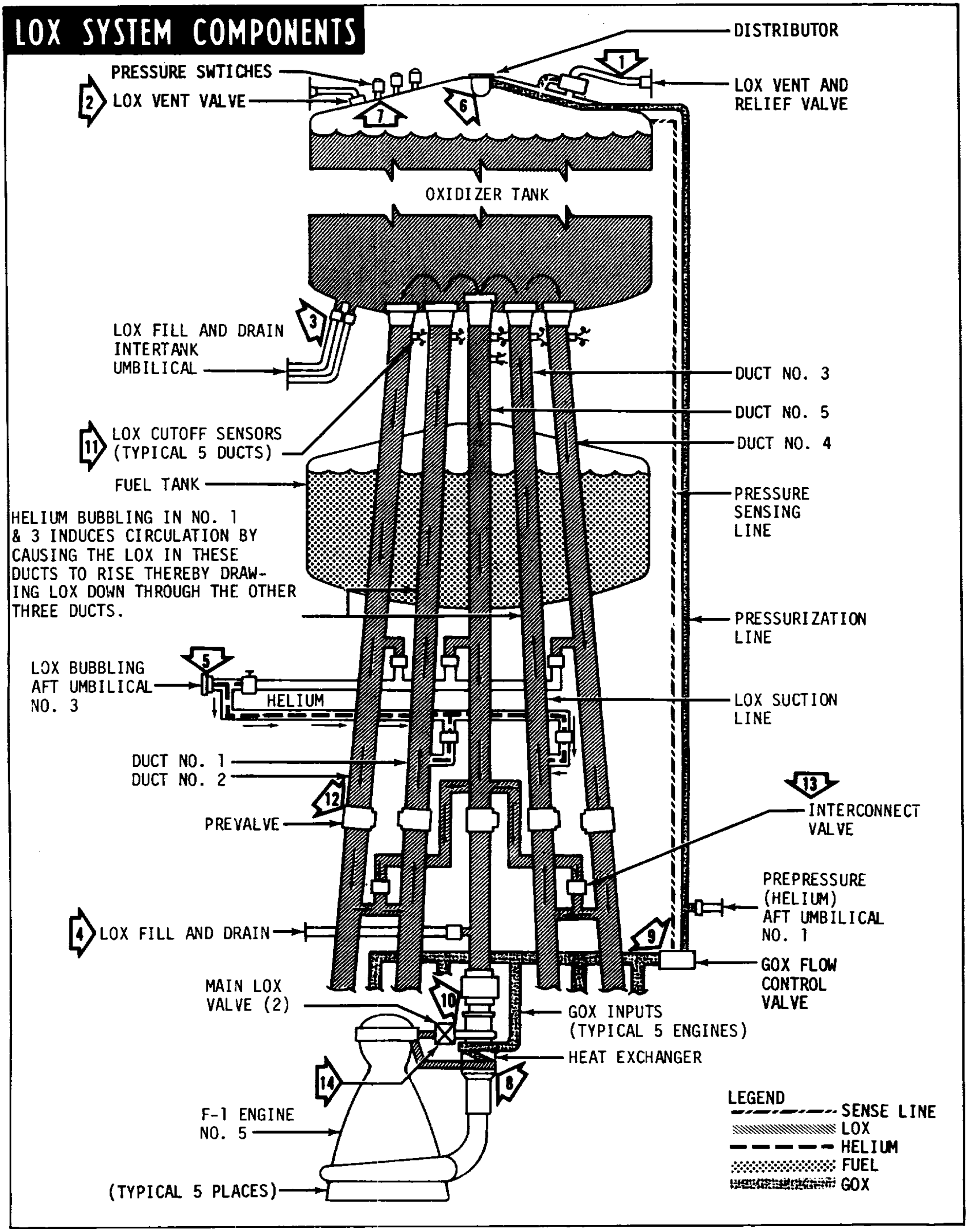

LOX SYSTEM COMPONENTS
DISTRIBUTOR
PRESSURE SWTICHES
2 LOX VENT VALVE
1
LOX VENT AND RELIEF VALVE
6
7
OXIDIZER TANK
3
LOX FILL AND DRAIN INTERTANK UMBILICAL
DUCT NO. 3
DUCT NO. 5
11 LOX CUTOFF SENSORS (TYPICAL 5 DUCTS)
DUCT NO. 4
FUEL TANK
PRESSURE SENSING LINE
HELIUM BUBBLING IN NO. 1 & 3 INDUCES CIRCULATION BY CAUSING THE LOX IN THESE DUCTS TO RISE THEREBY DRAWING LOX DOWN THROUGH THE OTHER THREE DUCTS.
PRESSURIZATION LINE
5
LOX BUBBLING AFT UMBILICAL NO. 3
HELIUM
LOX SUCTION LINE
DUCT NO. 1
DUCT NO. 2
12
13
PREVALVE
INTERCONNECT VALVE
PREPRESSURE (HELIUM) AFT UMBILICAL NO. 1
4 LOX FILL AND DRAIN
9
GOX FLOW CONTROL VALVE
MAIN LOX VALVE (2)
10
GOX INPUTS (TYPICAL 5 ENGINES)
HEAT EXCHANGER
14
8
F-1 ENGINE NO. 5
(TYPICAL 5 PLACES)
LEGEND
SENSE LINE
LOX
HELIUM
FUEL
GOX

Figure 4-16

S-IC ELECTRICAL POWER EQUIPMENT LOCATIONS

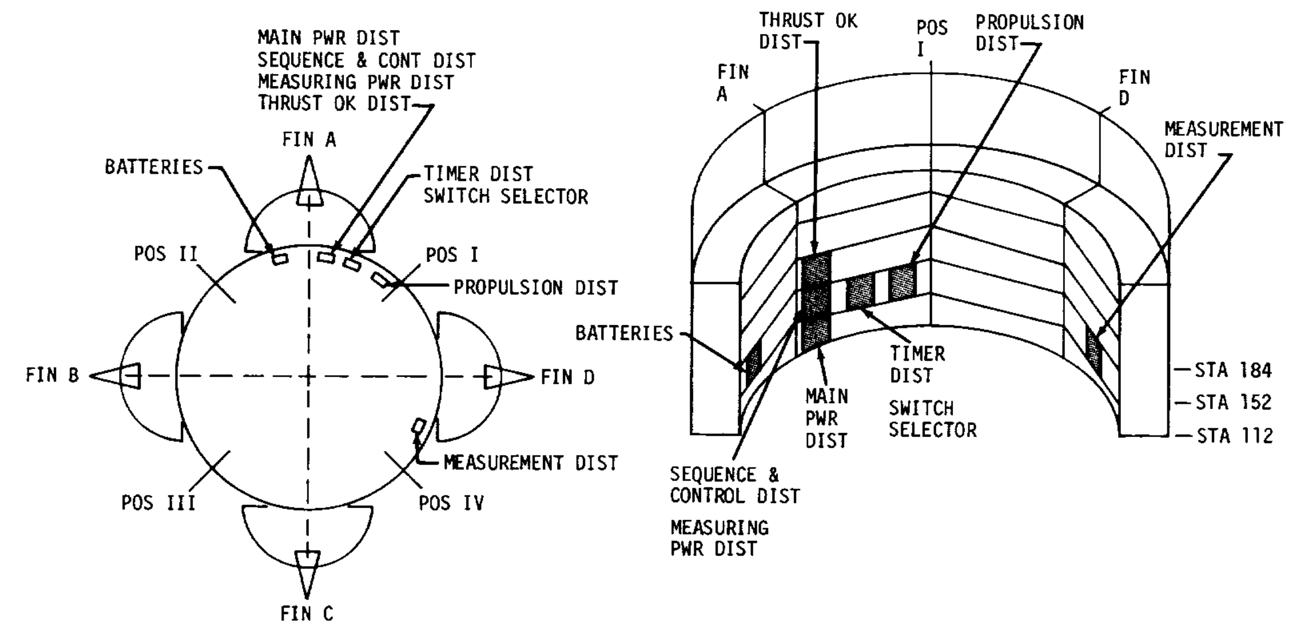


Figure 4-17

S-IC BATTERY CHARACTERISTICS	
TYPE	DRY CHARGE
MATERIAL	ZINC-SILVER OXIDE
ELECTROLYTE	POTASSIUM HYDROXIDE (KOH) IN PURE WATER
CELLS	20 WITH TAPS FOR SELECTING 18 OR 19 TO REDUCE OUTPUT VOLTAGE AS REQUIRED
NOMINAL VOLTAGE	1.5 VDC PER CELL: 28 $\pm$ 2 VDC PER 18 TO 20 CELL GROUP
CURRENT RATINGS	BATTERY NO. 1 - OPERATIONAL LOADS = 640 AMPERE/MINUTE BATTERY NO. 2 - MEASUREMENT LOADS = 1250 AMPERE/MINUTE
GROSS WEIGHT	BATTERY NO. 1 = 22 LBS. BATTERY NO. 2 = 55 LBS.

Figure 4-18

surement power subsystems. However, each of these systems is completely independent of the other. The power load is transferred from the ground source to the flight batteries at T-50 seconds. Inflight operation of the multicontact make-before-break power transfer switch is prevented by a brake, by mechanical construction, and by electrical circuitry. Operation of the switch several times during countdown verifies performance of the brake, motor, contacts, and mechanical components.

Sequence and Control Distributor

The sequence and control distributor accepts command signals from the switch selector and through a series of magnetically latching relays provides a 28-volt dc command to initiate or terminate the appropriate stage function. The input from the switch selector latches a relay corresponding to the particular command. A 28-volt dc signal is routed through the closed contacts of the relay to the stage component being commanded. The relays, one for each command function, may be unlatched by a signal from GSE. The normally closed contacts of the relays are connected in series. A 28-volt dc signal is routed through the series connected relay contacts to indicate to GSE when all sequence and control relays are in the reset state.

Propulsion Distributor

The propulsion distributor contains relays, diodes, and printed circuit boards for switching and distributing propulsion signals during launch preparation and flight.

THRUST OK Distributor

The THRUST OK distributor contains relays and printed circuit assemblies which make up the THRUST OK logic networks and timers required to monitor engine THRUST OK pressure switches and

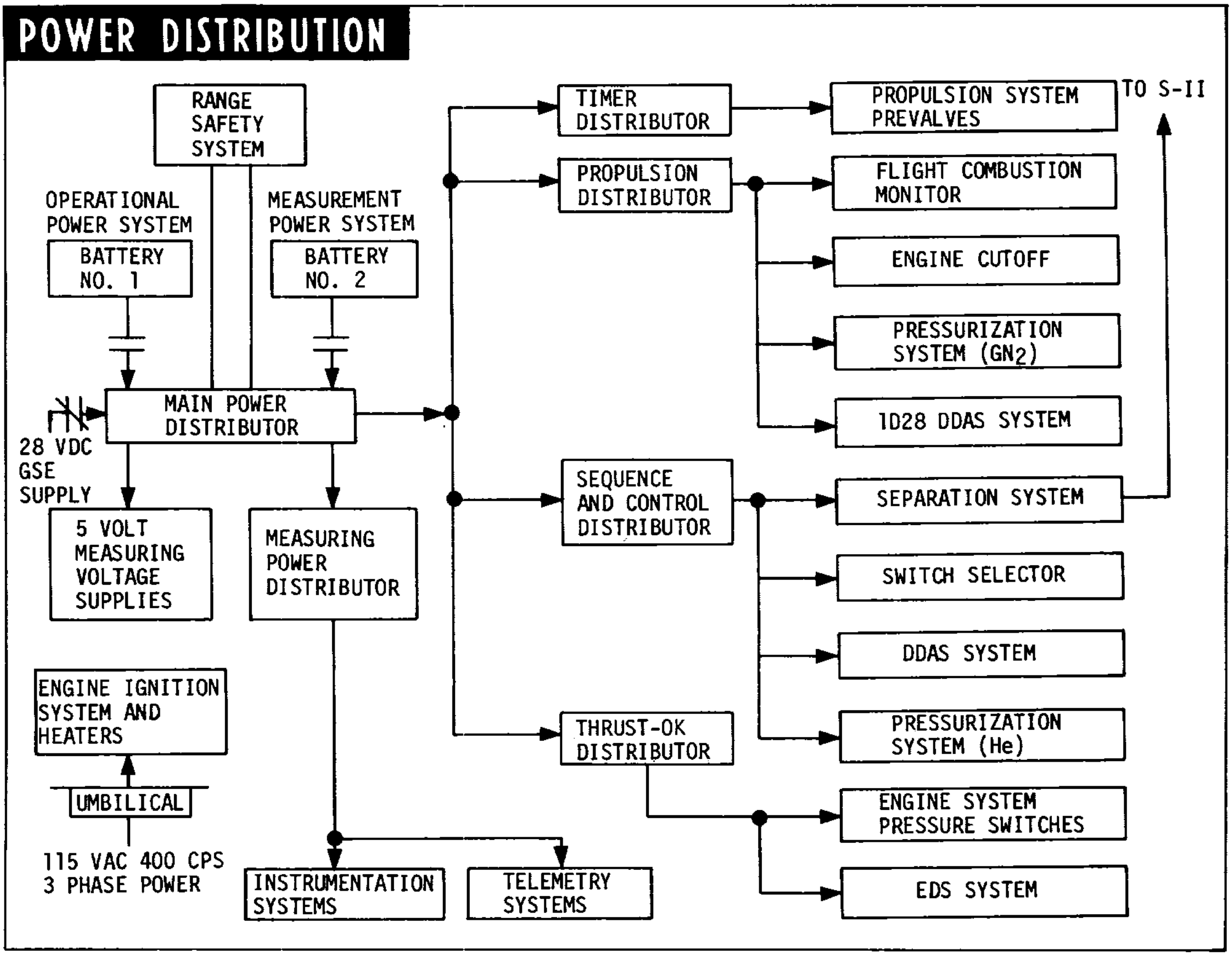


Figure 4-19

initiate engine shutdown. Signals from two of the three THRUST OK pressure switches on a particular engine will result in an output from a two-out-of-three voting network. This output activates a 0.044 second timer. If the THRUST OK condition is missing longer than 0.044 seconds the timer output sends a signal to initiate engine shutdown.

Timer Distributor

Circuits to time the operation of relays, valves, and other electromechanical devices are mounted in the timer distributor.

Measuring Power Distributor

Each regulated 5-volt dc output from the seven measuring power supplies is brought to an individual bus in the measuring power distributor and then routed to the measuring and telemetry systems.

SWITCH SELECTOR

The S-IC stage switch selector is the interface between the LVDC in the IU and the S-IC stage electrical circuits. Its function is to sequence and control various flight activities such as TM calibration, retro rocket initiation, and pressurization as shown in figure 4-20.

A switch selector is basically a series of low power transistor switches individually selected and controlled by an eight-bit binary coded signal from the LVDC in the IU. A coded word, when addressed to the S-IC switch selector, is accepted and stored in a register by means of magnetically latching relays. The coded transmission is verified by sending the complement of the stored word back to the LVDC in the IU. At the proper time an output signal is initiated via the selected switch selector channel to the appropriate stage operational circuit. The switch selector can control 112 circuits.

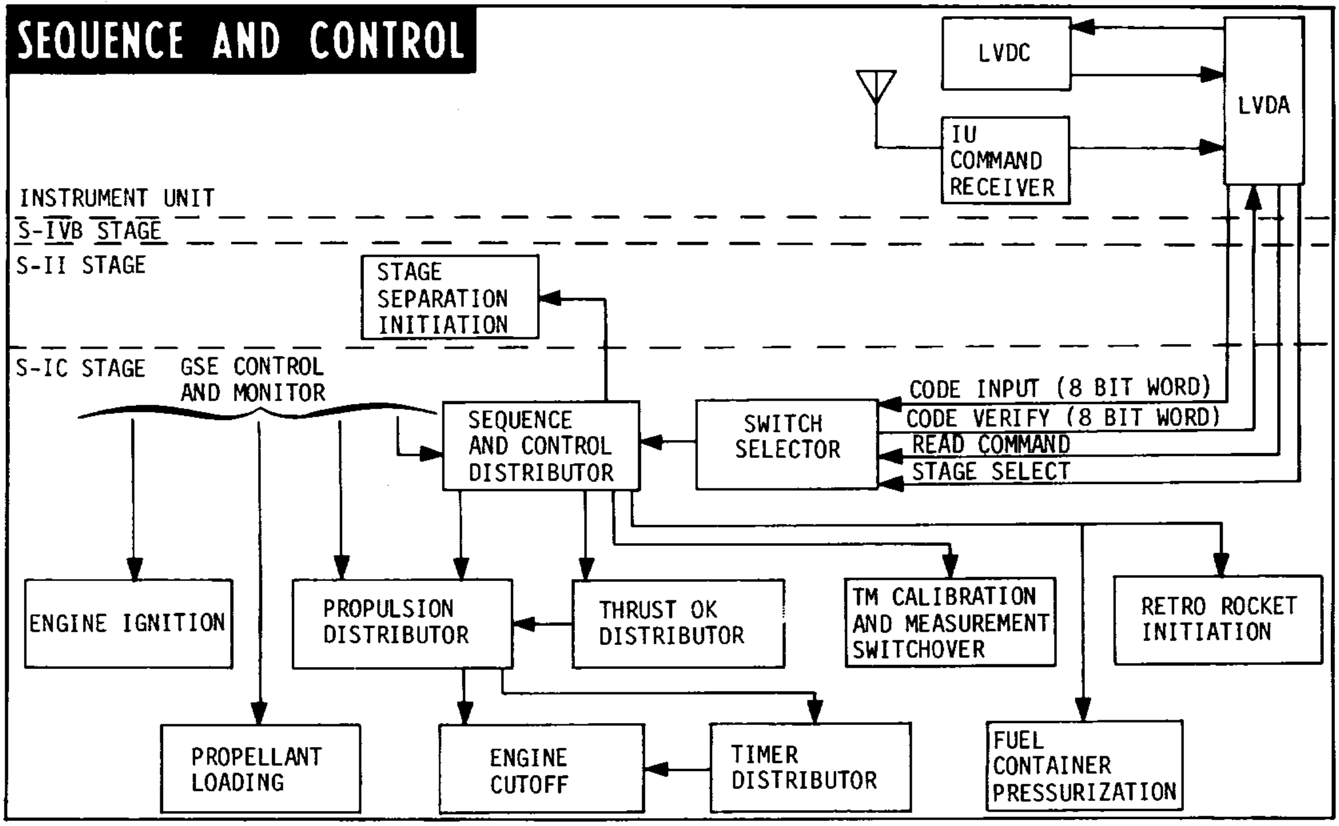


Figure 4-20

LVDC commands activate, enable, or switch stage electrical circuits as a function of elapsed flight time. Computer commands include:

1. Telemetry calibration.
2. Remove telemetry calibration.
3. Open helium flow control valve No. 2.
4. Open helium flow control valve No. 3.
5. Open helium flow control valve No. 4.
6. Enable center engine cutoff.
7. Enable outboard engine cutoff.
8. Arm EBW firing unit, retrorockets, and separation system.
9. Fire EBW firing unit, retrorockets, and separation system.
10. Measurement switchover.

In addition, a command from the emergency detection system in the IU can shut down all S-IC stage engines.

INSTRUMENTATION

The S-IC stage instrumentation system monitors functional operations of stage systems and provides signals for vehicle tracking during the S-IC burn. Prior to liftoff, measurements are telemetered by coaxial cable to ground support equipment. During flight, date is transmitted to ground stations over RF links. The ODOP system uses the doppler principle to provide vehicle position and acceleration data during flight.

MEASUREMENT SYSTEM

The measurement system senses performance parameters and feeds signals to the telemetry system. It includes transducers, signal conditioning, and distribution equipment necessary to provide the required measurement ranges and suitably scaled voltage signals to the inputs of the telemetry system.

The S-IC measuring system performs three main functions:

1. Detection of the physical phenomena to be measured and transformation of these phenomena into electrical signals.
2. Process and condition the measured signals into the proper form for telemetering.
3. Distribution of the data to the proper channel of the telemetry system.

Measurements fall into a number of basic categories depending upon the type of measured variable, the variable rate of change with time, and other considerations. Since the vehicle is subjected to maximum buffeting forces during the S-IC boost phase, a large number of strain and vibration measurements are required. Figure 4-21 summarizes the measurements to be taken on the S-IC stage.

TELEMETRY SYSTEM

The telemetry system accepts the 648 signals produced by the measuring portion of the instrumentation system and transmits them to ground stations. The telemetry equipment includes multiplexers, subcarrier oscillators, amplifiers, modulators, transmitters, and an omnidirectional system of four antennae. The telemetry subsystems use multiplex techniques (time sharing) to transmit large quantities of measurement data over a relatively small number of basic RF links. This equipment also includes tape recorders for recording critical data which would otherwise be lost due to telemetry blackout during S-II ullage and S-IC retrorocket firing. The recorders, which have a 3 minute record capability, will playback the critical data during stage free fall.

Several telemetry subsystems are provided in the S-IC stage. There are three basic types of telemetry subsystems (figure 4-22). The high frequency data such as vibration and acoustics measurements are transmitted via two independent single sideband (SSB) FM telemetry links

STAGE MEASUREMENTS	
TYPE	QTY
Acceleration	3
Acoustic	4
Temperature	166
Pressure	177
Vibration	67
Flowrate	35
Position	1
Discrete Signals	142
Liquid Level	22
Voltage, Current, Frequency	11
Miscellaneous	12
Angular Velocity	3
RPM	5
TOTAL	648

Figure 4-21

(3). Three pulse amplitude modulated/frequency modulated/frequency modulated (PAM/FM/FM) links (1) are used for telemetering low-to-medium frequency data such as pressure, temperature, or strain indications. Time multiplexed data from the PAM links are also routed through the PCM link at one third sampling rate for DDAS transmission during preflight testing and for redundant RF transmission during flight. A pulse code modulated/digital data acquisition system (PCM/DDAS) link (2) provides for acquisition of analog and digital flight data, provides a hardwire link for obtaining PCM data and PAM time multiplexed data during test and checkout, and permits the redundant monitoring of PAM data during flight.

The PCM/DDAS system assembles and formats PCM/FM time shared data so it can be sent over coaxial cables for automatic ground checkout or over an RF link during flight.

Remote Automatic Calibration System (RACS)

IV

The RACS is used to verify measurement circuit operation and continuity by stimulating the transducer directly, or by inserting a simulated transducer signal in the signal conditioner circuit. Measurement operation is verified at 80 percent of the maximum transducer range (high level), at 20 percent of the maximum range (low level), and at the normal run level. Approximately 415 of the 648 measurements can be addressed by the RACS.

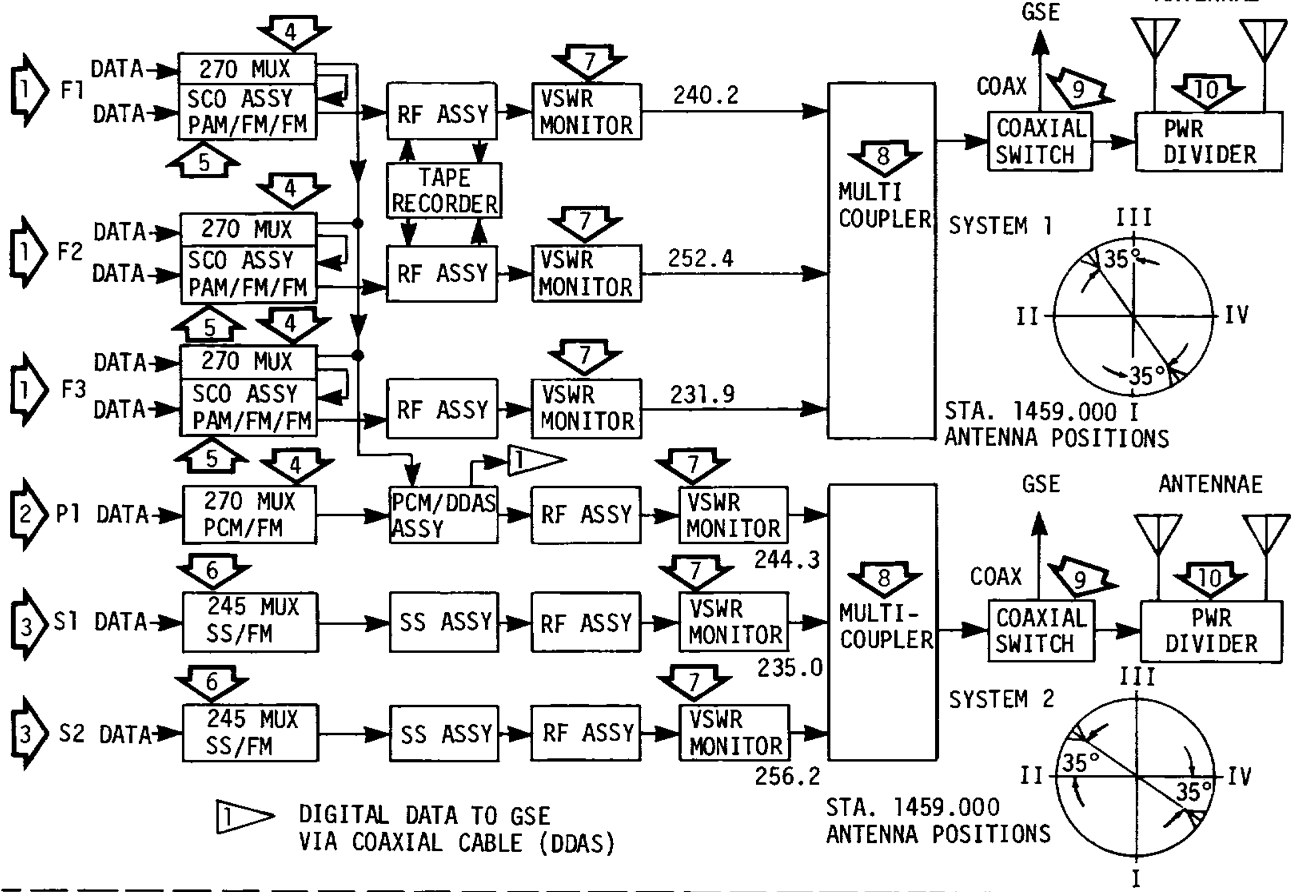
Antennae

The S-IC stage telemetry system utilizes a total of four shunt fed stub antennae operating in pairs as two independent antenna systems shown as system I and system II in figure 4-22. Information from telemetry links operating at 240.2, 252.4 and 231.9 MHz is transmitted through system I, and the information from telemetry links operating at 235.0, 244.3, and 256.2 MHz is transmitted through system II.

ODOP

An offset doppler (ODOP) frequency measurement system (figure 4-23) is an elliptical tracking system which measures the total doppler shift in a ultra high frequency (UHF) continuous wave (CW) signal transmitted to the S-IC stage. The ODOP system uses a fixed station transmitter, a vehicle borne transponder,and three or more fixed station receivers to determine the vehicle position. In this system the transmitter, transponder, and one receiver describe an ellipsoid where the transponder is at a point on the ellipsoid surface. The second receiver describes an ellipsoid whose intersection with the first ellipsoid is a line. The addition of the third receiver produces a third ellipsoid whose intersection with the line of intersection of the first and second ellipsoids is a point. This point is the transponder.

TELEMETRY SYSTEMS



1. The PAM/FM/FM system is used to transmit data in the frequency range below 1000 hertz. Each PAM/FM/FM link provides 14 inter-range instrumentation group (IRIG) continuous data channels with a maximum frequency response of 450 hertz and 27 time-sharing channels capable of transmitting a maximum of 230 measurements (10 per channel) with a sampling rate of 12 samples per second and 4 channels at 120 samples per second.

2. The PCM/FM system provides a primary data acquisition link for analog and digital data and a redundant means of monitoring data transmitted by the three PAM links. This system assemblies time-shared data into a serial PCM format which is transmitted by a RF assembly and a 600 kc voltage controlled oscillator (VCO) in the PCM/DDAS assembly.

3. SS/FM system links S1 and S2 transmit acoustical and vibration data in the frequency range of 50 to 3000 hertz. Total bandwidth of the system is 76 kHz. Each multiplexer can handle up to 80 measurements: each single sideband assembly handles 15 continuous channels. The SS/FM system provides frequency division multiplexing of fifteen AM data channels on a FM carrier.

4. The model 270 time division multiplexer accepts voltage inputs from 0 to 5 volts. It provides outputs to the PCM/DDAS assembly and the PAM Links. Of 30 channels, 3 channels (28, 39, and 30) are used for voltage reference and synchronization, 4 channels (24, 25, 26, and 27) are sampled at 120 samples per second the remaining 23 channels are sub-multiplexed and sampled at 12 samples per second to accomodate 10 measurements each.

Figure 4-22 (Sheet 1 of 2)

TELEMETRY SYSTEMS

5. A SUB-CARRIER OSCILLATOR ASSEMBLY (SCO) provides the frequency division multiplexing for the PAM/FM/FM links. Channels 2 through 15 provide 14 continuous data channels and band X handles the time-division multiplexed data from the model 270 multiplexer. All oscillator outputs are mixed in an amplifier and routed to the FM transmitter in the RF assembly.

6. A model 245 multiplexer increases the number of measurements which may be transmitted over one RF link. The unit permits 80 data measurements to be time division multiplexed into 16 output channels. An auxiliary output is also provided for channel identification.

7. The voltage standing wave ratio (VSWR) monitor is used to monitor the performance of the telemetry antenna system and the output of the telemetry transmitter.

8. The multicoupler couples three simultaneous radio frequency (RF) signals into a common output without mutal interference and with minimum insertion loss (0.15 db or less).

9. The Coaxial Switch provides a means of connecting to the GSE coaxial transmission line or to the vehicle antennae. Switching between the two lines is accomplished by application or removal of 28-volts DC to the switch.

10. The Power Divider splits RF power between the two antennae while handling an average power level of 100 watts over a range of 215 to 260 MHz.

Figure 4-22 (Sheet 2 of 2)

A fixed frequency continuous wave carrier of 890 MHz is transmitted from the ground transmitter. This signal is received by the vehicle transponder which amplifies, processes, and retransmits the signal at 960 MHz. At the ground station, the signal from the vehicle transponder is heterodyned with a reference 960 MHz signal derived from the 53.3 MHz signal of the ground transmitter station. The resultant beat frequency is proportional to the vehicle velocity components radial to the transmitter and receiver.

Range and position data are derived by integration of the doppler frequency record from a known reference point. The distance from three stations are combined geometrically to give the position of the vehicle in space.

There are two non-interchangeable antennae. One antenna is for transmitting and the other is for receiving. The mounting positions of the antennae on the vehicle are shown in figure 4-23.

Figure 4-24 summarizes the S-IC stage instrumentation.

ORDNANCE

The S-IC ordnance systems include the propellant dispersion (flight termination system (figure 4-25) and the retrorocket system (figure 4-26).

PROPELLANT DISPERSION SYSTEM

The S-IC propellant dispersion system (PDS) provides the means of terminating the flight of the Saturn V if it varies beyond the prescribed limits of its flight path or if it becomes a safety hazard during the S-IC boost phase. The system is installed on the stage in compliance with Air Force Eastern Test Range (AFETR) Regulation 127-9 and AFETR Safety Manual 127-1.

The PDS is a dual channel, parallel redundant system composed of two segments. The radio frequency segment receives, decodes, and controls the propellant dispersion commands. The ordnance train segment consists of two exploding bridgewire (EBW) firing units, two EBW detonators, one safety and arming (S&A) device (shared by both channels), six confined detonating fuse (CDF) assemblies, two CDF tees, two CDF/flexible linear shaped charge (FLSC) connectors, and two FLSC assemblies.

The S&A device (figure 4-27) is a remotely controlled electro-mechanical ordnance device that is used to make safe and to arm the S-IC, S-II, and S-IVB stage PDS's. The device can complete and interrupt the explosive train by remote control, provide position indications to remote monitoring equipment, and provide a visual position indication. It also has a manual operation capability. The S&A device consists of a rotary solenoid assembly, a metal rotor shaft with two explosive inserts, and position sensing and command switches that operate from a rotor shaft cam. In the safe mode, the longitudinal axis of the explosive inserts are perpendicular to the detonating wave path, thus forming a barrier to the explosive train. To arm the device, the shaft is rotated 90° to align the inserts between the EBW detonators and the CDF adapters to form the initial part of the explosive train.

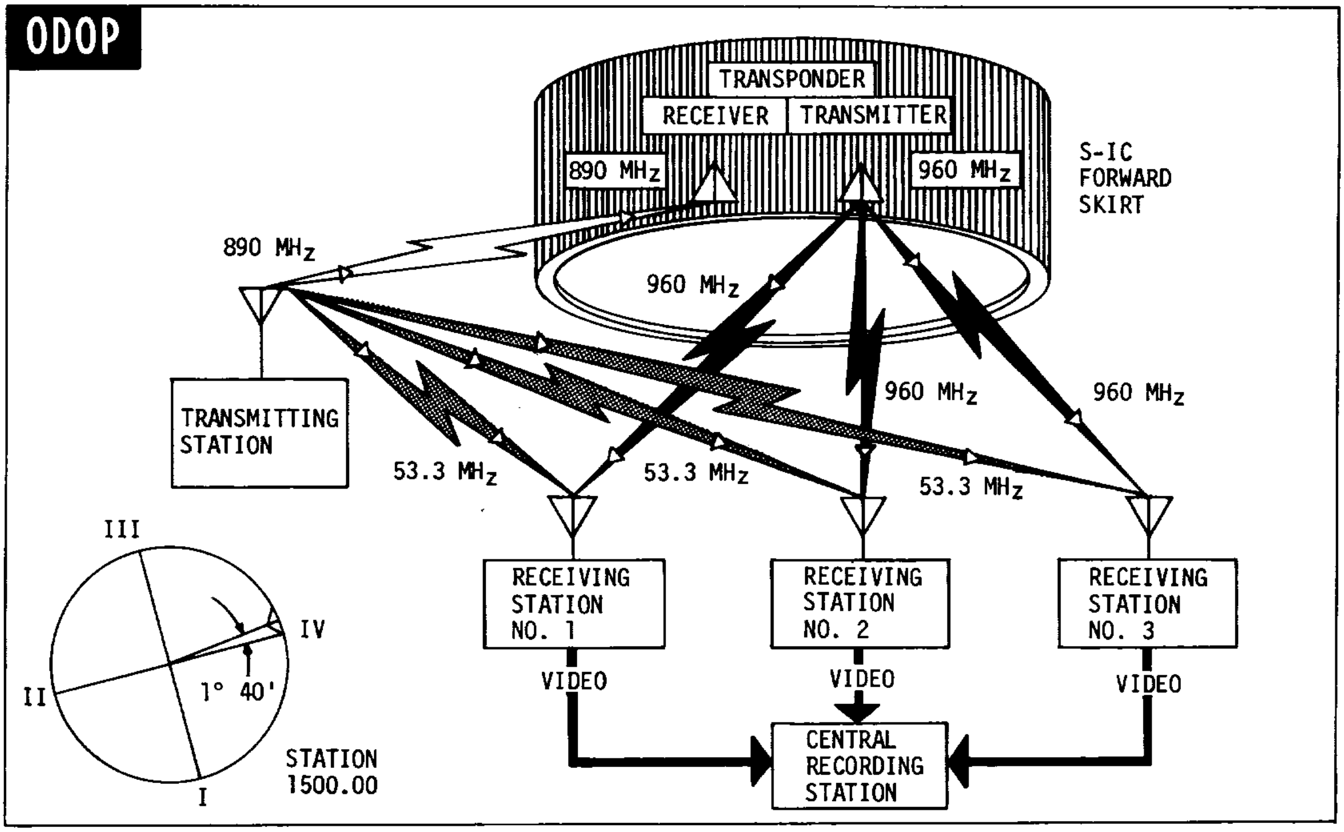


Figure 4-23

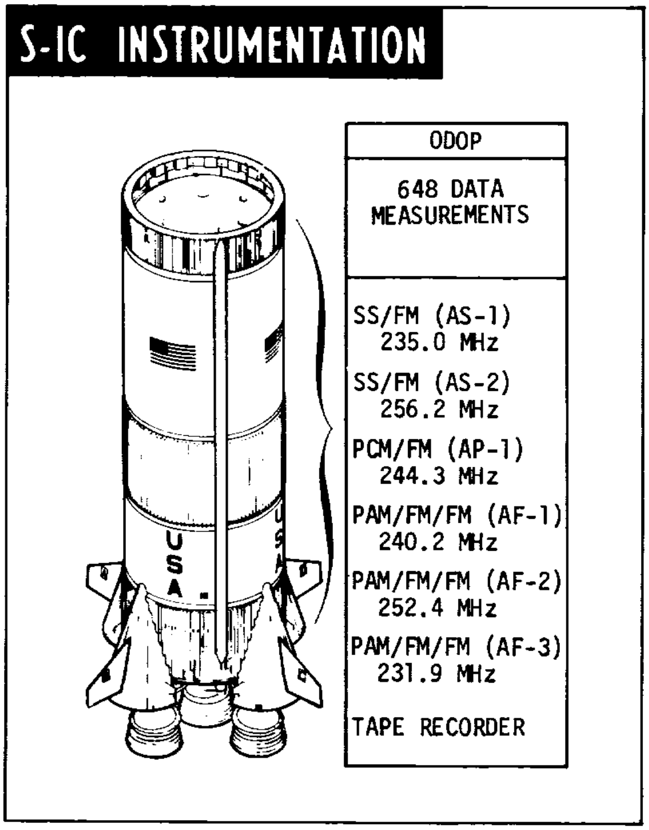


Figure 4-24

Should emergency flight termination become necessary, two coded radio frequency commands are transmitted to the launch vehicle by the range safety officer. The first command arms the EBW firing units and initiates S-IC engine cutoff. (See figure 4-25 for block diagram of the PDS and location of PDS components.) The second command, which is delayed to permit charging of the EBW firing units, discharges the EBW firing units across the exploding bridgewire in the EBW detonators mounted on the S&A device (see figure 4-27). The resulting explosive wave propagates through the S&A device inserts to the CDF assemblies and to the CDF tees. The CDF tees propagate the wave through insulated CDF assemblies to the FLSC assemblies mounted on the lox and RP-1 tanks. The FLSC's provide the explosive force to longitudinally sever the propellant tanks and disperse the propellants. There are six 88-inch FLSC sections mounted on the lox tank and three 88-inch sections on the fuel tank. These sections are positioned on the propellant tanks to minimize mixing of the propellants after the tanks are severed.

RETROROCKETS

The eight retrorockets (figure 4-26), that provide separation thrust after S-IC burnout, are attached externally to the thrust structure inside the four outboard engine fairings. The firing command originates in the Instrument Unit and activates redundant firing systems. Ad-

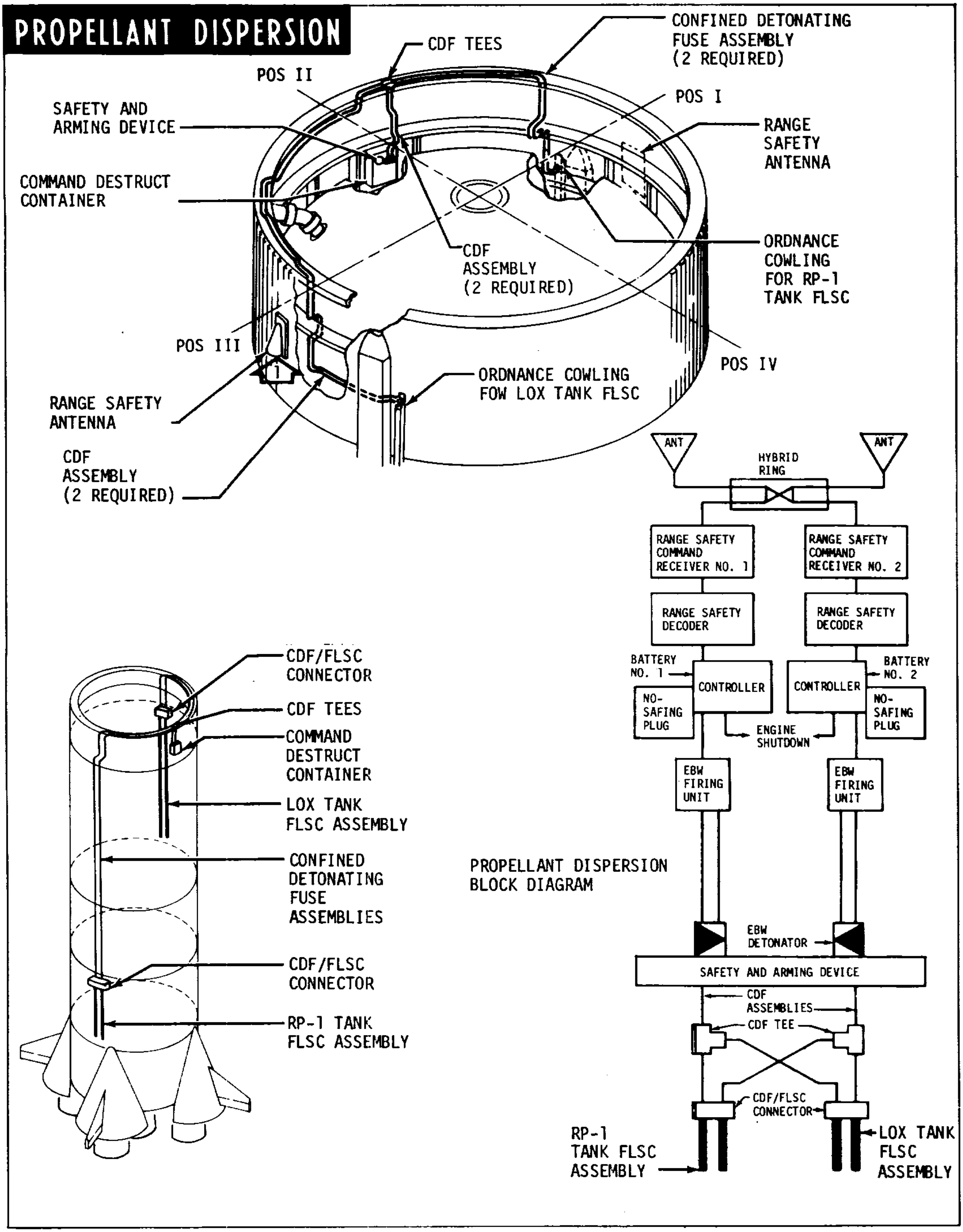


Figure 4-25

IV

RETROROCKETS

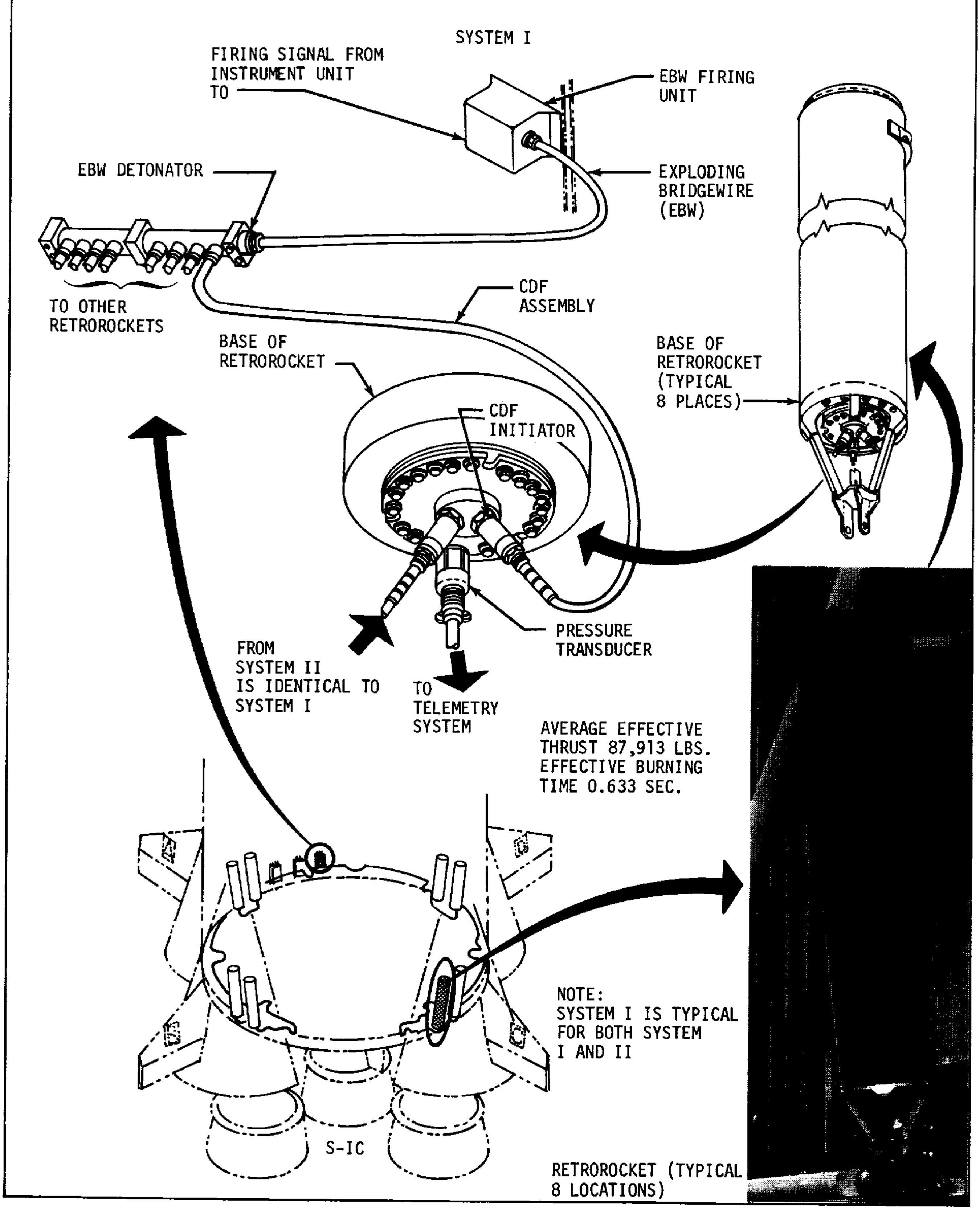


Figure 4-26

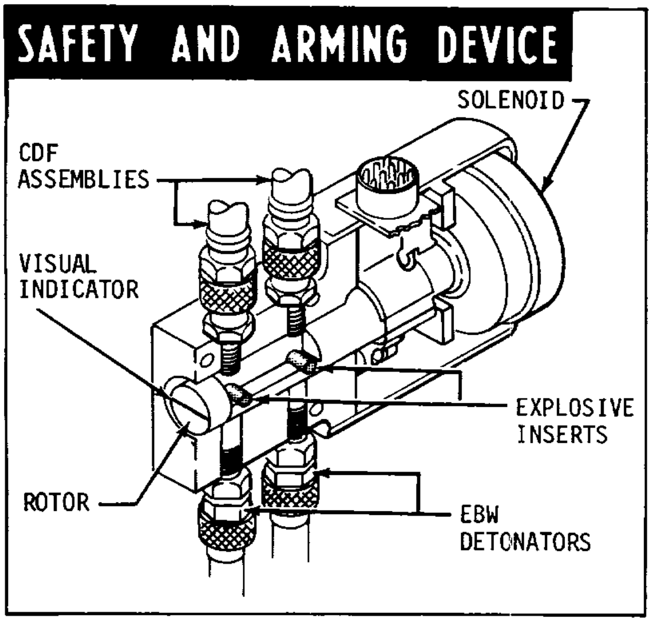


Figure 4-27

ditional redundancy is provided by interconnection of the two confined detonating fuse (CDF) manifolds with CDF assemblies. The exploding bridgewire (EBW) firing unit circuits are grounded by a normally closed relay until the firing command is initiated by the Instrument Unit. High voltage electrical signals are released from the two EBW firing units to the EBW detonators upon vehicle deceleration to 0.5g. The signals cause the detonator bridgewires to explode, thereby detonating the surrounding explosives. The explosion then propagates through the CDF manifold explosive and CDF assemblies into the igniter assembly. The igniter assembly located within the base of each retrorocket is then ignited, causing a buildup and release of the gases into the main grain of the retrorocket. Each retrorocket is ignited by either of two CDF initiators mounted on its aft structure. Operational ground check of the system through the firing unit is accomplished through use of pulse sensors which absorb the high voltage impulse from the firing unit and transmit a signal through the telemetry system. The pulse sensors are removed prior to launch.

The S-IC retrorockets are mounted, in pairs, (figure 4-26 in the fairings of the F-1 engine. At retrorocket ignition the forward end of the fairing is burned and blown through by the exhausting gases. Each retrorocket is pinned securely to the vehicle support and pivot support fittings at an angle of 7.5° from center line. The thrust level developed by seven retrorockets (one retrorocket out) is adequate to separate the S-IC stage a minimum of six feet from the vehicle in less than one second.

Each retrorocket is a solid propellant rocket with a case bonded, twelve-point star, internal burning composite propellant cast directly into the case and cured. The propellant is basically ammonium perchlorate oxidizer in a polysulfide fuel binder. The motor is 86 inches long by 15-1/4 inches diameter and weighs 504 pounds, nominal, of which 278 pounds is propellant.

MAJOR DIFFERENCES BETWEEN SATURN V S-IC-3 & S-IC-4 STAGES

1▶ *Engine nominal thrust at sea level increased from 1,522,000 pounds to 1,526,500 pounds each and from 7,610,000 pounds to 7,632,500 pounds total.*

2▶ *Dry weight reduced from 304,000 pounds to 295,300 pounds.*

3▶ *Weight at ground ignition increased from 4,800,000 pounds to 5,030,300 pounds.*

4▶ *Instrumentation measurements reduced from 891 to 648.*

5▶ *Lox tank center standpipe extended.*

6▶ *Engine cutoff control changed from IU command to lox depletion.*

7▶ *Visual instrumentation electrical power system not installed on S-IC-4.*

8▶ *TV camera system not installed on S-IC-4.*

9▶ *Film camera system not installed on S-IC-4.*

IV S-IC

SECTION V

S-II STAGE

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INTRODUCTION

The S-II stage (figure 5-1) is a large cylindrical booster, 81.5 feet long and 33 feet in diameter, powered by five liquid propellant J-2 rocket engines which develop a nominal vacuum 1▶ thrust of 230,000 pounds each for a total of 1,150,000 pounds. Its purpose is to provide second stage boost for the Saturn V launch vehicle. At engine cutoff the S-II stage separates from the S-IVB and, following a suborbital path, reenters the atmosphere where it disintegrates due to reentry environment.

Dry weight of the S-II stage is approximately 2▶ 85,522 pounds (95,779 pounds including the S-IC/S-II interstage). The stage approximate gross 3▶ weight is 1,080,500 pounds. The four outer J-2 engines are equally spaced on a 17.5-foot diameter circle and are capable of being gimbaled through a plus or minus 7.0 degree square pattern for thrust vector control. The fifth engine is mounted on the stage centerline and is fixed.

The stage (figure 5-1) consists of the structural airframe, the J-2 engines, piping, valves, wiring, instrumentation, electrical and electronic equipment, ordnance devices, and four solid propellant ullage rockets. These are collected into the following major systems: structural, environmental control, propulsion, flight control, pneumatic, propellant, electrical, instrumentation, and ordnance. The stage has structural and electrical interfaces with the S-IC and S-IVB stages; and electric, pneumatic, and fluid interfaces with ground support equipment through its umbilicals and antennae.

STRUCTURE

The S-II airframe (figure 5-2) consists of a body shell structure (forward and aft skirts and interstage), a propellant tank structure (liquid hydrogen and liquid oxygen tanks), and a thrust structure. The body shell structure transmits first and second stage boost loads (axial, shear, and bending moment) and stage body bending and longitudinal forces between the adjacent stages, the propellant tank structure, and the thrust structure. The propellant tank structure holds the propellants, liquid hydrogen (LH_2) and liquid oxygen (lox), and provides structural support between the aft and forward skirts. The thrust structure transmits the thrust of the five J-2 engines to the body shell structure; compression loads from engine thrust; tension loads from idle engine weight; and cantilever loads from engine weight during S-II boost.

BODY SHELL STRUCTURE

The body shell structure units, the forward skirt, aft skirt, and interstage (figures 5-3 and 5-4), are of the same basic design except that the aft skirt and interstage are of generally heavier construction because of higher structural loads.

Each unit is a cylindrical shell of semimonocoque construction, built of 7075 aluminum alloy material, stiffened by external hat-section stringers and stabilized internally by circumferential ring frames. The forward skirt has a basic skin thickness of 0.040 inch and the aft skirt and interstage both have basic skin thicknesses of 0.071 inch.

THRUST STRUCTURE

The thrust structure (figure 5-4), like the body shell structure, is of semimonocoque construction but in the form of a truncated cone increasing in size from approximately 18 feet in diameter to the 33 foot outside diameter of the airframe. It is stiffened by circumferential ring frames and hat-section stringers. Four pairs of thrust longerons (two at each outboard engine location) and a center engine support beam cruciform assembly accept and distribute the thrust loads of the J-2 engines. The shell structure is of 7075 aluminum alloy. A fiberglass honeycomb heat shield, supported from the lower portion of the thrust structure, protects the stage base area from excessive temperatures during S-II boost.

PROPELLANT TANK STRUCTURE

The propellant tank structures (figure 5-2) are of 2014 aluminum alloy material. The liquid hydrogen tank consists of six cylindrical sections, an ellipsoidal forward bulkhead, and the LH_2/lox tank common bulkhead all joined to form one assembly comprised of two tanks. The sidewall contains integral circumferential and longitudinal stiffeners on the internal surface

S-II STAGE

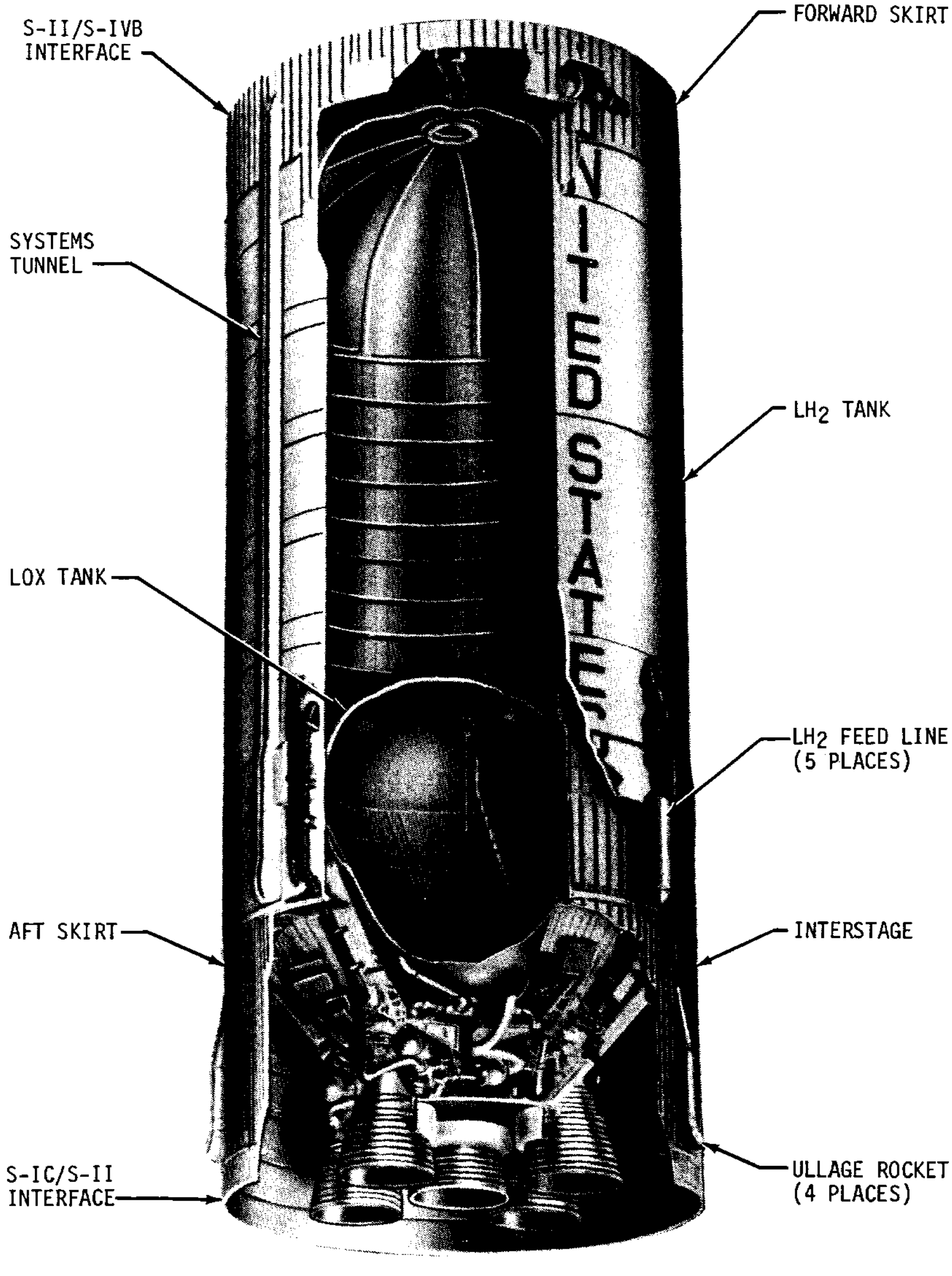


Figure 5-1

STAGE STRUCTURE

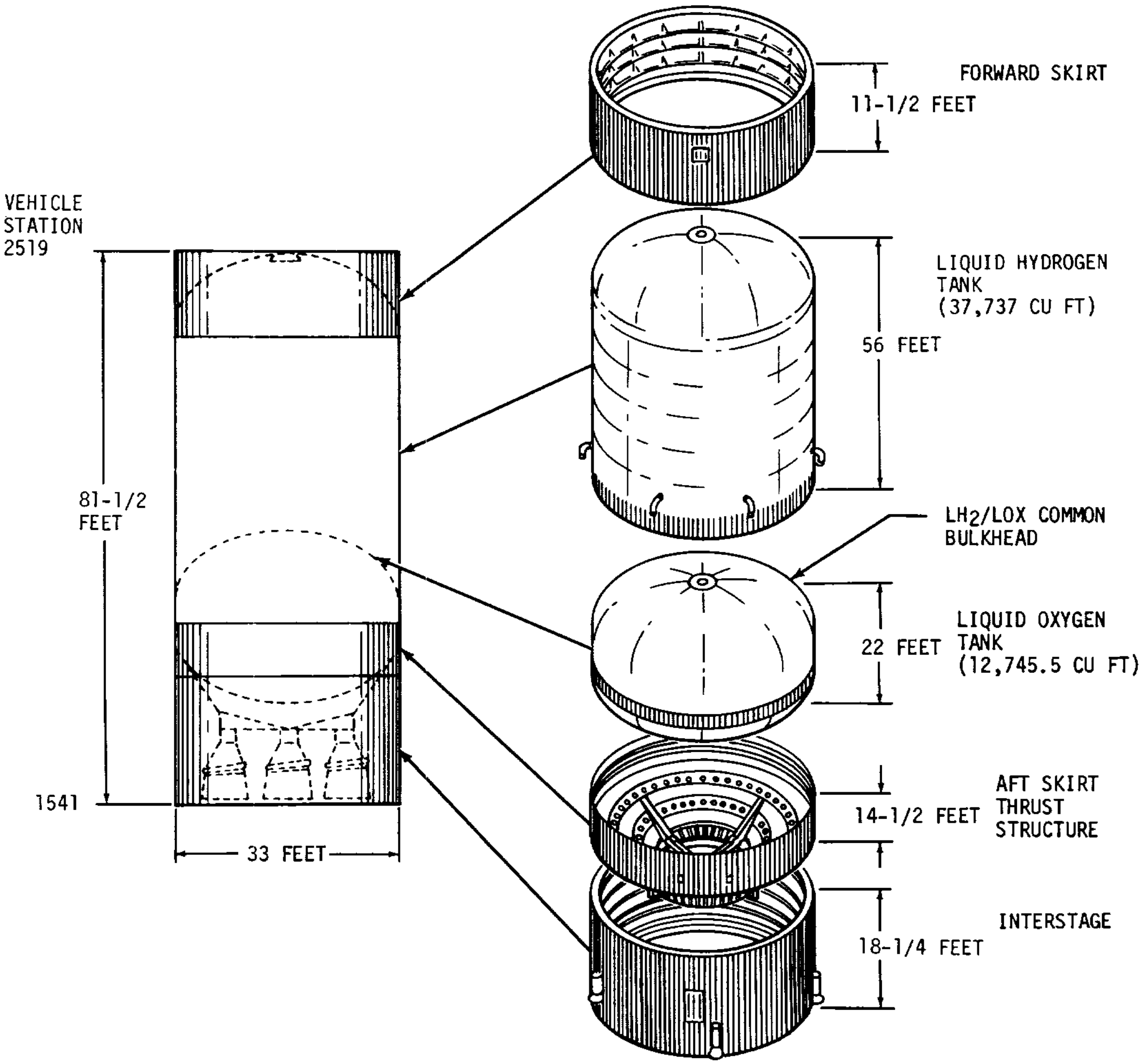


Figure 5-2

while the lower portion of the tank has external stringers. The LH_2 tank is insulated on the outside with a layer of foam-filled honeycomb to prevent excessive boil-off of the propellant during prelaunch operations and flight.

The common bulkhead is a sandwich structure forming the aft closure of the LH_2 tank and the forward half-section of the lox tank (figure 5-5). It is an adhesive-bonded sandwich assembly employing facing sheets of 2014 aluminum alloy and fiberglass/phenolic honeycomb core to prevent heat transfer and retain the cryogenic properties of the two fluids to which it is exposed. Fiberglass core insulation thickness varies from approximately 5 inches at the apex to 0.080 inch at the outer extremity. No connections or lines pass through the common bulkhead. The forward skin has a "J" section return at the outer edge to permit peripheral attachment to the LH_2 tank while the lower facing is carried through to provide structural continuity with the lox tank aft bulkhead.

The liquid oxygen tank (figure 5-5) consists of ellipsoidal fore and aft halves with waffle-stiffened gore segments. The tank is fitted with three ring-type slosh baffles to control propellant sloshing and minimize surface disturbances and cruciform baffles to prevent the generation of vortices at the tank outlet ducts and to minimize residuals. A six-port sump assembly located at the lowest point of the lox tank provides a fill and drain opening and openings for five engine feed lines.

SYSTEM TUNNEL

A systems tunnel, housing electrical cables, pressurization lines, and the tank propellant dispersion ordnance, is attached externally from the S-II stage aft skirt area to the forward skirt. It has a semicircular shape 22 inches wide and about 60 feet long. Cabling which connects the S-IC stage to the instrument unit also runs through this tunnel (see figure 5-1).

ENVIRONMENTAL CONTROL

The environmental control system is supplied dehumidified, thermally-conditioned air and nitrogen from a ground source for temperature control and purging of the compartments during prelaunch operations only.

THERMAL CONTROL

The thermal control system (figure 5-6) provides temperature control to forward and aft skirt mounted equipment containers. The system is put into operation shortly after vehicle is mated to pad facilities. Air is used as the conditioning medium until approximately twenty minutes prior to liquid hydrogen (LH_2) loading. At this time gaseous nitrogen (GN_2) is used, until umbilical disconnect, to preclude the possibility of an explosion in the event of LH_2 leakage. The change to GN_2 is made before propellant loading to ensure that all oxygen is expelled and dissipated before a hazard can arise. The

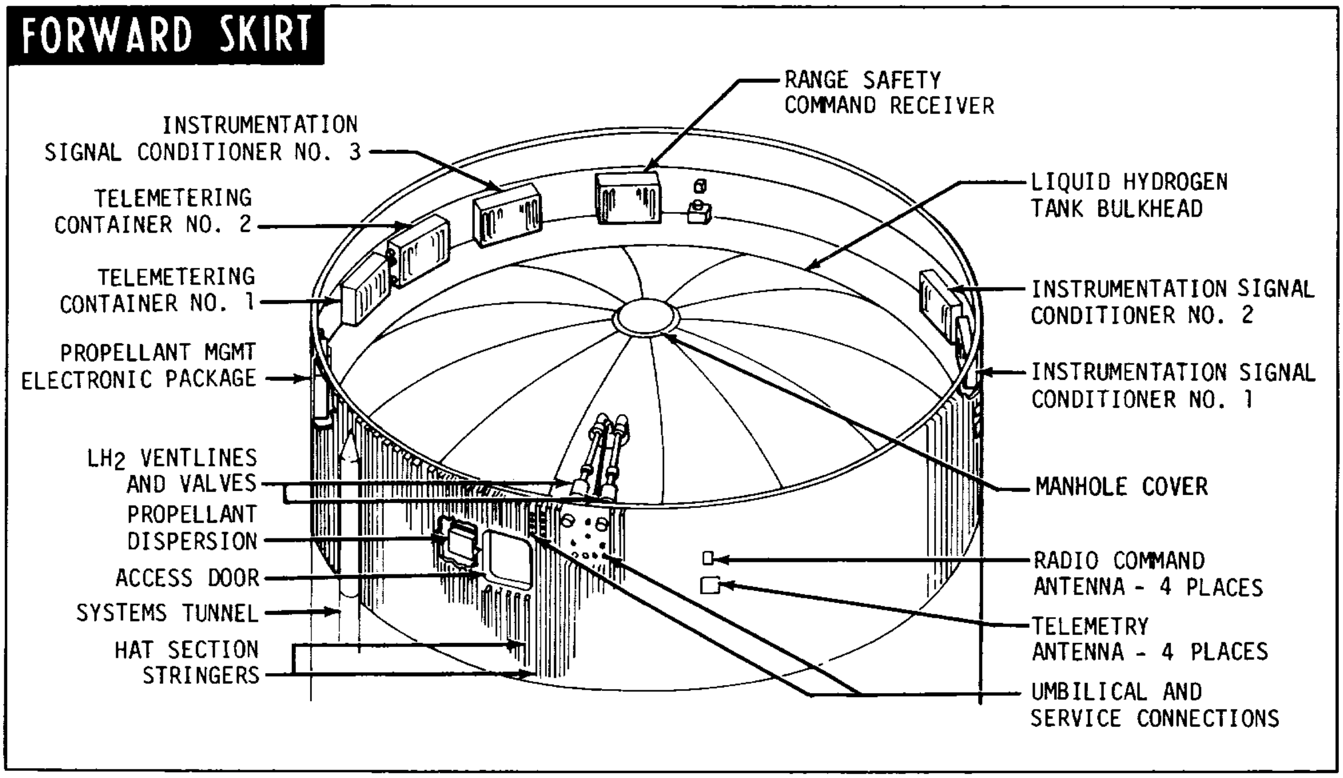


Figure 5-3

nitrogen flow is terminated at liftoff, and no flow is provided during boost, as the container insulation maintains equipment temperatures throughout the S-II flight trajectory.

ENGINE COMPARTMENT CONDITIONING

The engine compartment conditioning system purges the engine and interstage areas of explosive mixtures and maintains a proper temperature. Purging the compartment is accomplished prior to propellant tanking and whenever propellants are on board. A 98 percent GN_2 atmosphere within the compartment maintains desired temperature while the danger of fire or explosion resulting from propellant leakage is minimized.

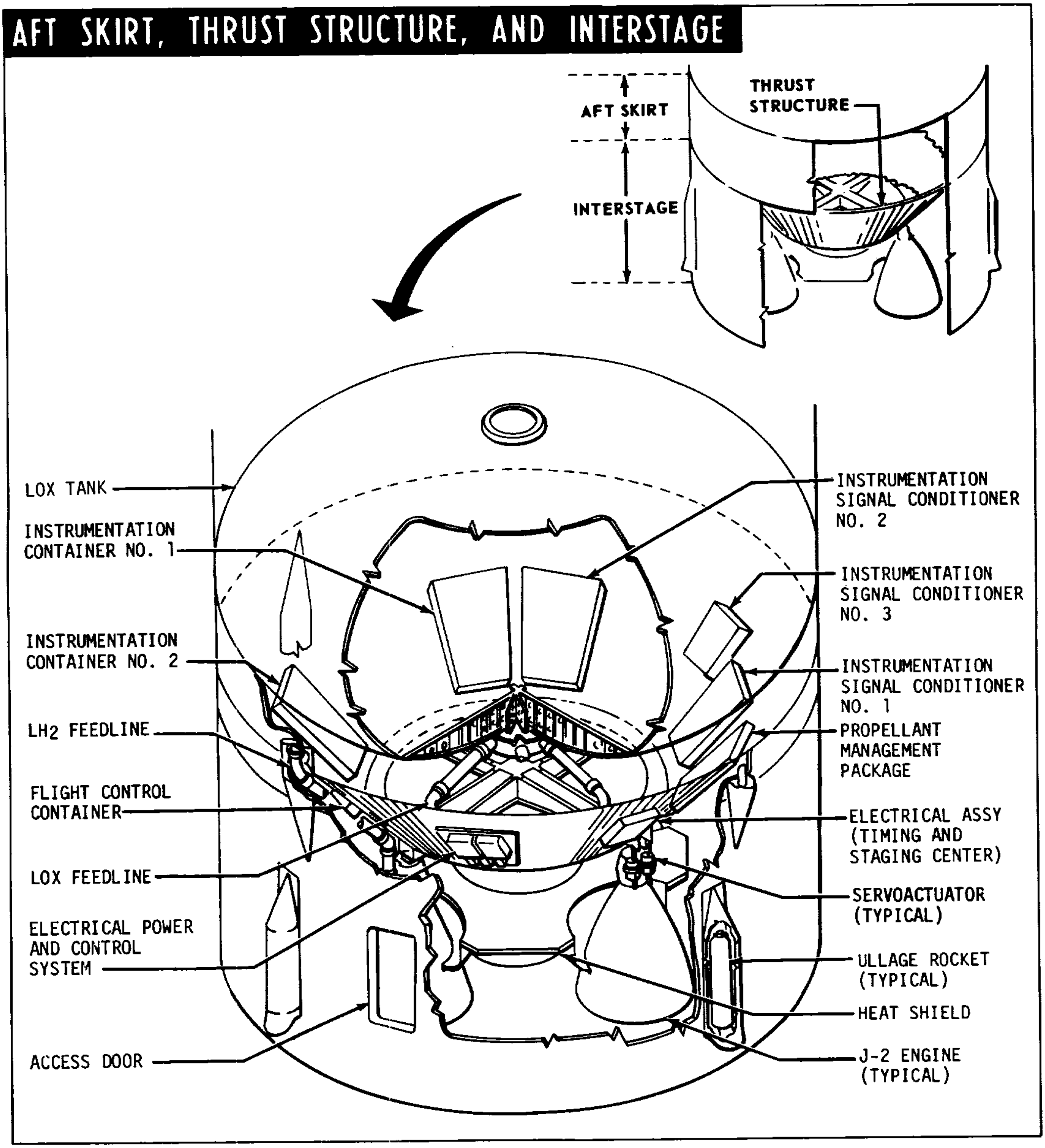


Figure 5-4

INSULATION PURGE LEAK DETECTION

All exposed surfaces of the LH_2 tank require insulation to prevent condensation and to reduce temperature rise during cryogenic operations.

Insulation/Purge

The insulation material is a foam-filled honeycomb, (figure 5-7) approximately 1.6 inches thick on the LH_2 tank sidewalls, and 0.5 inch thick on the forward bulkhead. The insulation has a network of passages through which helium gas is drawn (see flow arrows in figure 5-7). The areas to be purged are divided into several circuits: the LH_2 tank sidewalls, forward bulkhead, the common bulkhead, the LH_2 tank/forward skirt junction, and the lower LH_2 tank/bolting ring areas.

Leak Detection

The purge system is used in conjunction with the leak detection system (figure 5-8) in the LH_2 tank sidewall, forward bulkhead, and common bulkhead areas to provide a means of detecting any hydrogen, oxygen, or air leaks while diluting or removing the leaking gases. From initiation of propellant loading until launch, the insulation is continuously purged of hazardous gases and a gas analyzer determines any leakage in the purging gas.

PROPULSION

The S-II stage engine system consists of five single-start J-2 rocket engines utilizing liquid oxygen and liquid hydrogen for propellants. The four outer J-2 engines are mounted parallel to the stage centerline. These engines are suspended by gimbal bearings to allow thrust vector control. The fifth engine is fixed and is mounted on the centerline of the stage.

J-2 ROCKET ENGINE

The J-2 rocket engine (figure 5-9) is a high performance, high altitude, engine utilizing liquid oxygen and liquid hydrogen as propellants. The only substances used in the engine are the propellants and helium gas. The extremely low operating temperature of the engine prohibits the use of lubricants or other fluids. The engine features a single tubular walled bell shaped thrust chamber and two independently driven direct drive turbopumps for liquid oxygen and liquid hydrogen. Both turbopumps are powered in series by a single gas generator, which utilizes the same propellants as the thrust chamber. The main hydraulic pump is driven by the oxidizer turbopump turbine. The ratio of

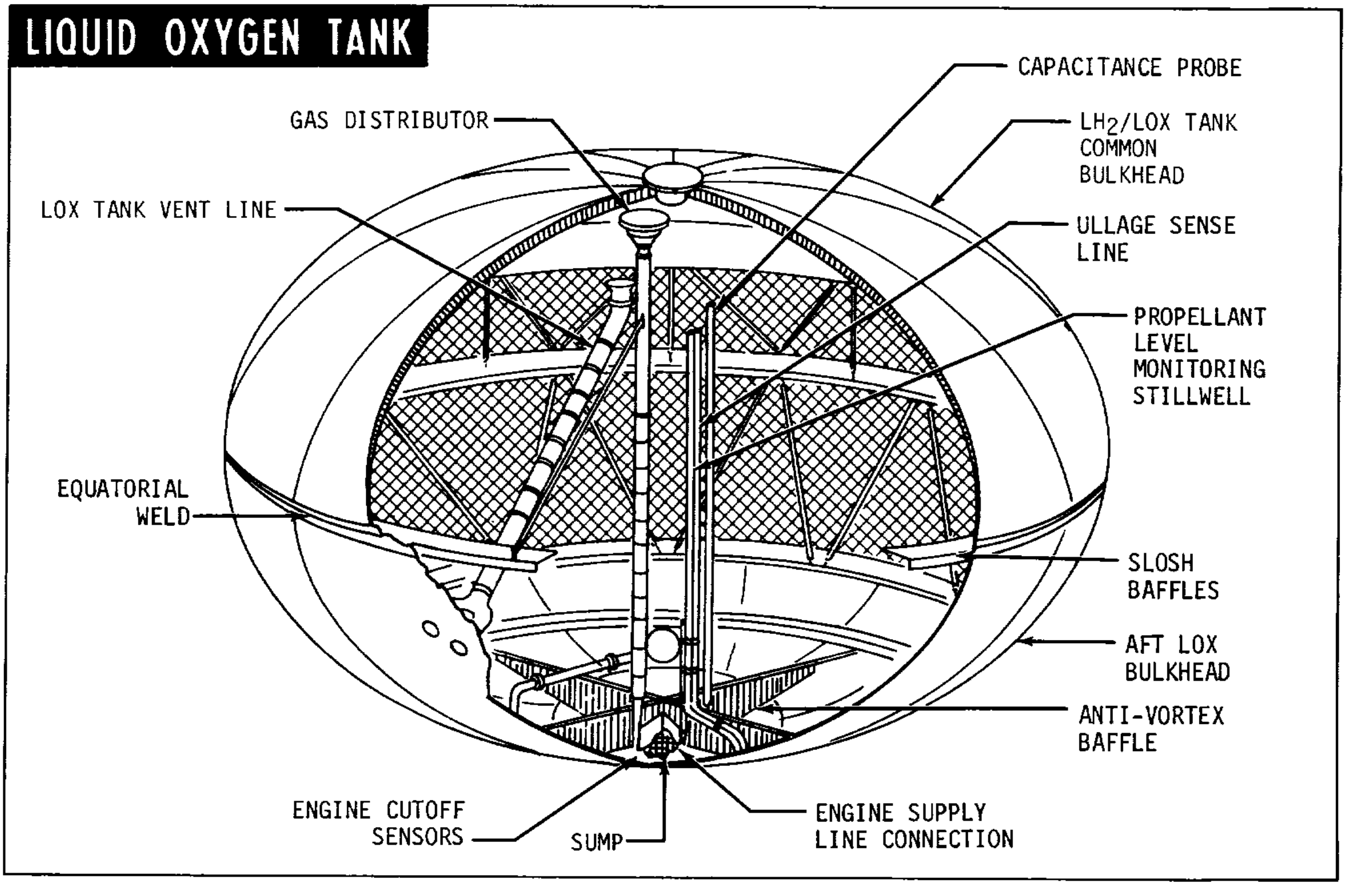


Figure 5-5

THERMAL CONTROL

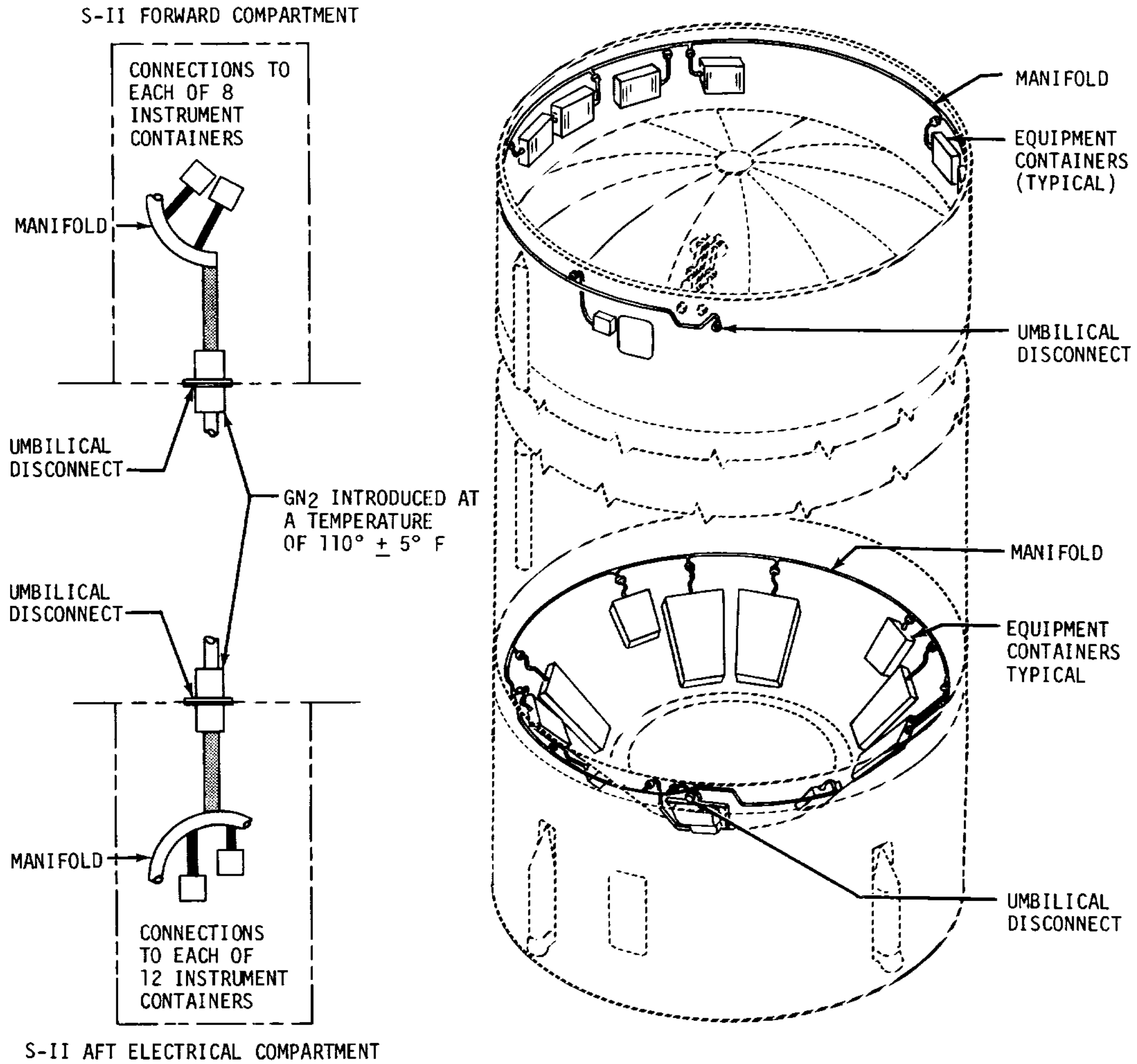


Figure 5-6

fuel to oxidizer is controlled by bypassing liquid oxygen from the discharge side of the oxidizer turbopump to the inlet side through a servovalve.

The engine valves are controlled by a pneumatic system powered by gaseous helium which is stored in a sphere inside the start tank. An electrical control system, which uses solid state logic elements, is used to sequence the start and shutdown operations of the engine. Electrical power is stage supplied.

During the burn periods, the lox tank is pressurized by flowing lox through the heat exchanger in the oxidizer turbine exhaust duct. The heat exchanger heats the lox causing it to expand. The LH_2 tank is pressurized during burn periods by GH_2 from the thrust chamber fuel manifold.

Thrust vector control is achieved by gimbaling the engine. Hydraulic pressure for gimbal actuation is provided by the hydraulic system which receives pressure from a hydraulic pump.

Start Preparations

Preparations for an engine start include ascertaining the positions and status of various engine and stage systems and components. The J-2

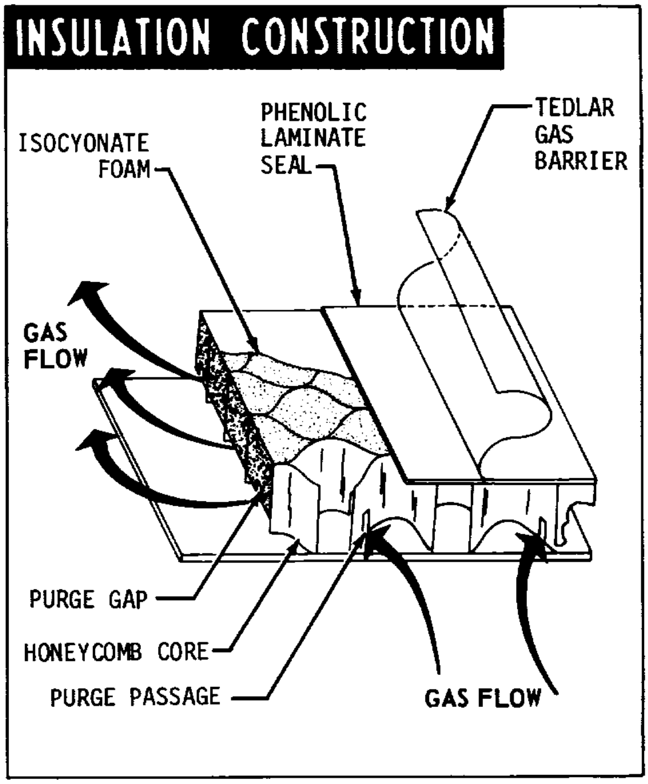


Figure 5-7

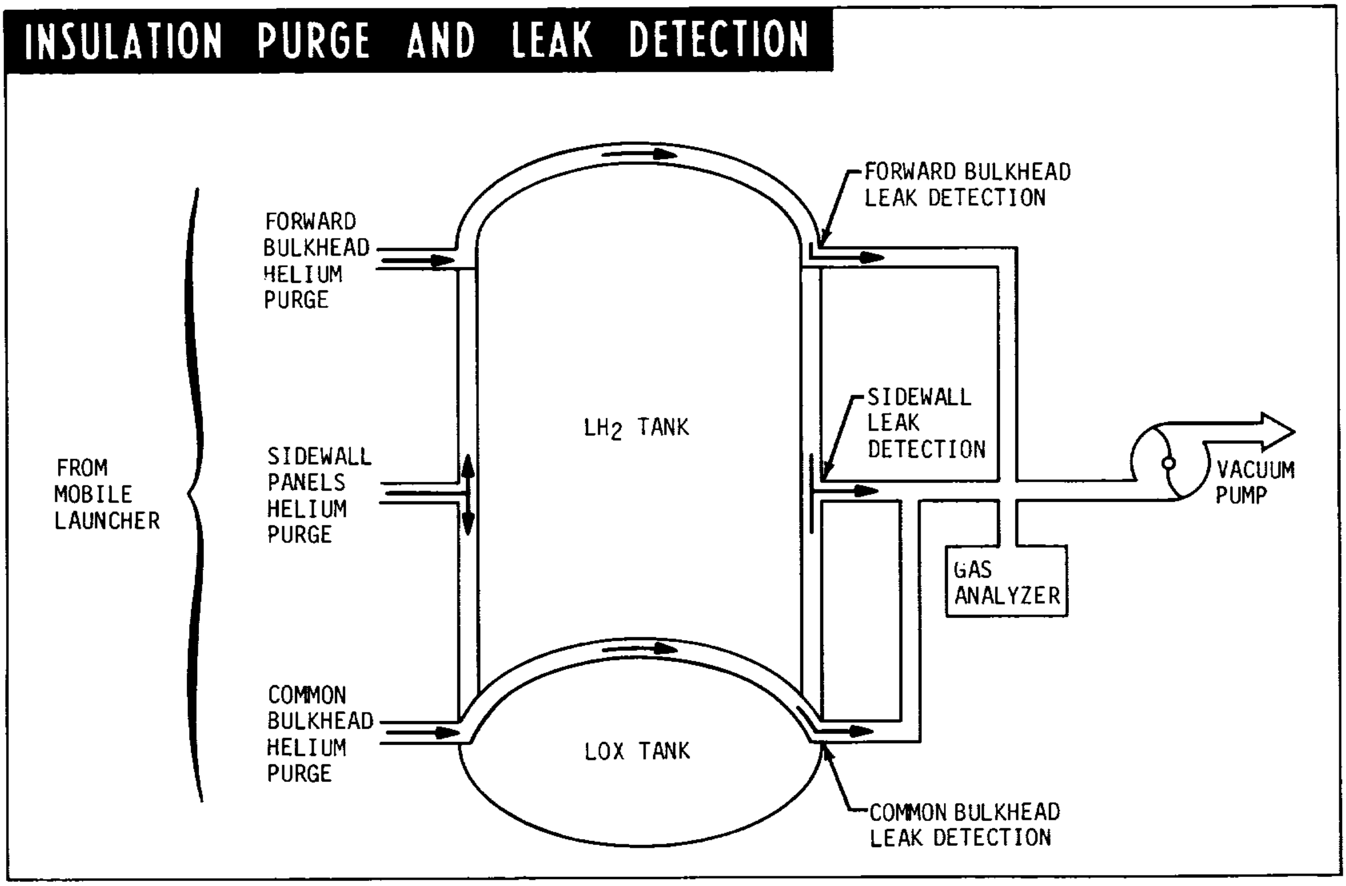


Figure 5-8

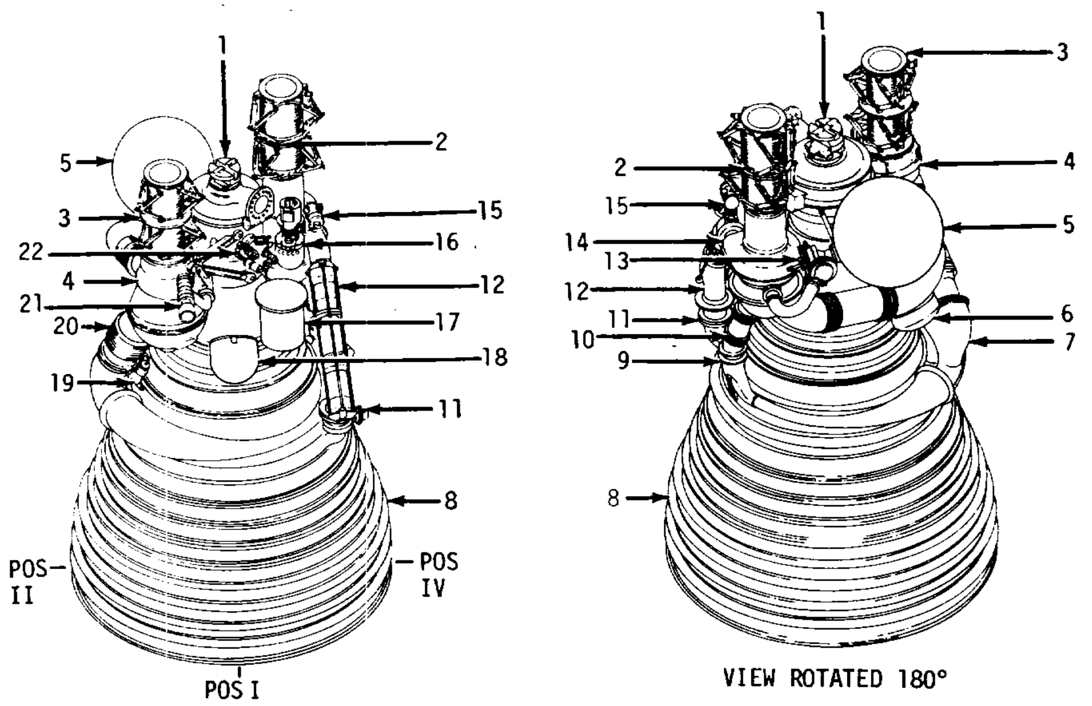
engine electrical control system controls engine operation by means of electrical signals. The heart of the engine electrical control system is the electrical control package (17, figure 5-9) which sequences and times the engine start or cutoff functions.

Engine cutoff automatically causes the electrical control package circuitry to reset itself ready for restart providing all reset conditions are met. The LVDC issues an engine ready bypass signal just prior to an engine start attempt. This bypass signal acts in the same manner as a cutoff would act. The reset signals engine ready and this allows the LVDC to send its start command. Receipt of the start command initiates the engine start sequence.

ENGINE START SEQUENCE

When engine start is initiated (3, figure 5-10) the spark exciters in the electrical control package provide energy for the gas generator (GG) and augmented spark igniter (ASI) spark plugs (4). The helium control and ignition phase control valves, in the pneumatic control package (1), are simultaneously energized allowing helium from the helium tank (2) to flow through the pneumatic regulator to the pneumatic control system. The helium is routed through the internal check valve in the pneumatic control package (1) to ensure continued pressure to the engine valves in the event of helium supply failure. The regulated helium fills a pneumatic accumulator, closes the propellant bleed valves, (5) and purges (6) the oxidizer dome and gas generator oxidizer injector manifold. The oxidizer turbopump (12) intermediate seal cavity is continuously purged. The mainstage control valve closes the main oxidizer valve and opens the purge control valve which allows the oxidizer dome and gas generator oxidizer injector to be purged (6). The mainstage control valve also supplies opening control pressure to the

J-2 ROCKET ENGINE



LEGEND

1. GIMBAL
2. FUEL INLET DUCT
3. OXIDIZER INLET DUCT
4. OXIDIZER TURBOPUMP
5. START TANK
6. AUXILIARY FLIGHT INSTRUMENTATION PACKAGE
7. EXHAUST MANIFOLD
8. THRUST CHAMBER
9. OXIDIZER TURBINE BYPASS VALVE
10. TURBINE BYPASS DUCT
11. MAIN FUEL VALVE
12. HIGH PRESSURE FUEL DUCT
13. START TANK DISCHARGE VALVE
14. FUEL TURBOPUMP
15. FUEL BLEED VALVE
16. GAS GENERATOR
17. ELECTRICAL CONTROL PACKAGE
18. PRIMARY FLIGHT INSTR. PACKAGE
19. ANTI-FLOOD CHECK VALVE
20. HEAT EXCHANGER
21. PROPELLANT UTILIZATION VALVE
22. PNEUMATIC CONTROL PACKAGE

Figure 5-9

V

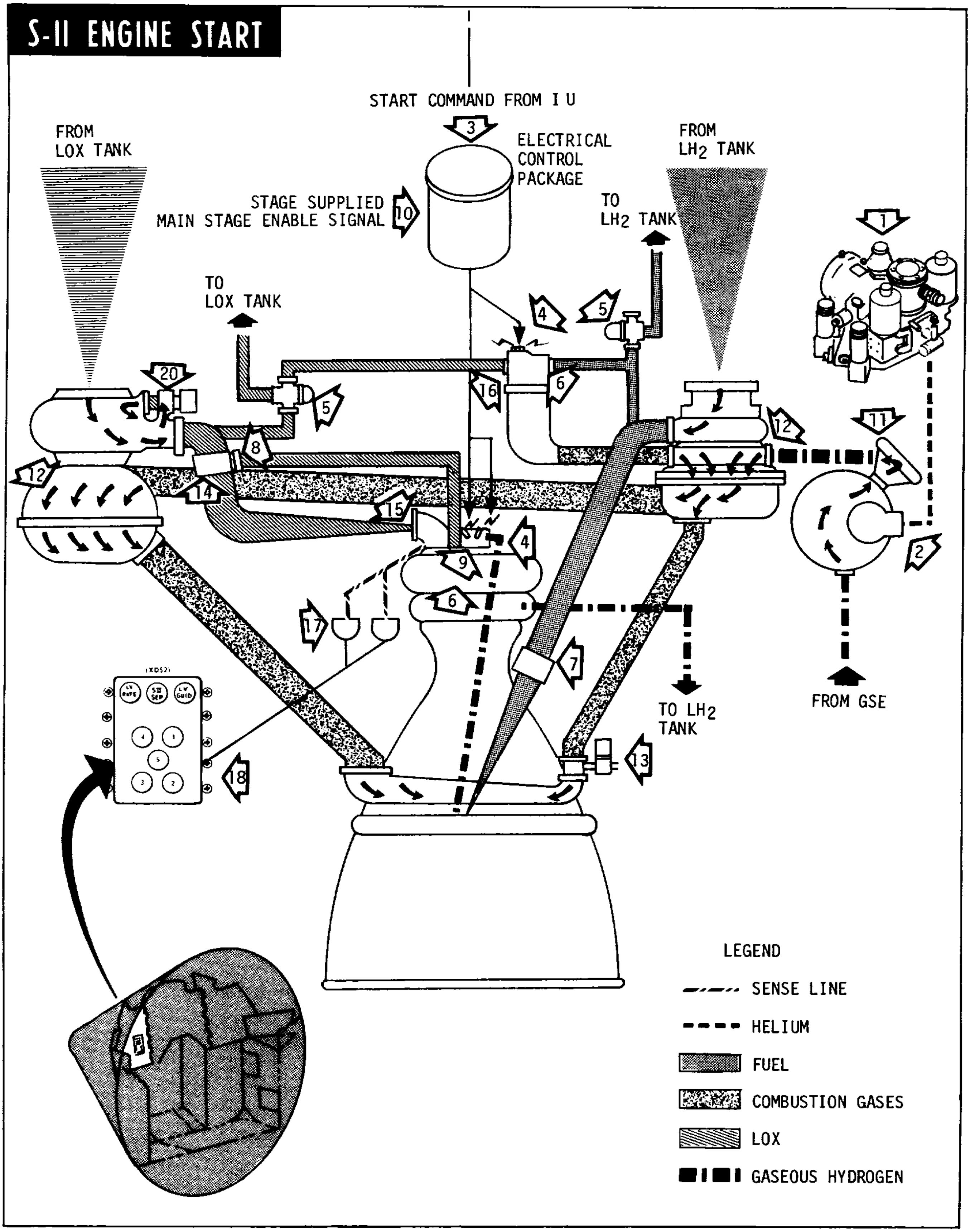


Figure 5-10 (Sheet 1 of 3)

S-II ENGINE START

SEQUENCE	EVENT	TIME IN SECONDS (T3 + 1.0, 3.0, 5.0, 7.0, 9.0, 11.0, 13.0)
	PRE-IGNITION NOTE ALL PNEUMATIC VALVES ARE CONTROLLED BY THE PNEUMATIC CONTROL PACKAGE 1 WHICH IS LOCATED ON THE ENGINE. HELIUM FOR THE PNEUMATIC SYSTEM IS SUPPLIED BY THE CONTROL SPHERE INSIDE THE GH_2 START TANK 2.	
3	Start Command (Occurs at T_3 + 1.4) Engine Ready → Engine Start Sequence	
4	Augmented Spark Igniter (ASI) Spark Plugs and gas generator (GG) spark plugs fire.	
5	Bleed valves stop return flow to propellant tanks.	
6	Oxidizer dome and gas generator oxidizer injector are purged	
7	Main fuel valve allows LH_2 to flow into engine thrust chamber and into ASI.	
8	ASI oxidizer valve allows lox to flow to ASI	
9	Sparks ignite the propellants in the ASI	
10	Stage supplied mainstage enable signal	

Figure 5-10 (Sheet 2 of 3)

V S-II

S-II ENGINE START

SEQUENCE	EVENT
11	Start tank discharges GH_2 causing
12	the LH_2 and lox turbopumps turbines buildup propellant pressure
13	Lox turbopump bypass valve opens to control lox pump speed.
14	Main oxidizer valve opens allowing
15	lox to be injected into thrust chamber
16	G G valves admit propellants. (Spark ignites propellants causing pressure build up.)
	MAIN STAGE
17	OK pressure switches send mainstage OK signal to CM.
18	Engine out lights go out.
19	Engine reaches and maintains 90% thrust or more.
20	P.U. valve controls mass ratio by returning lox from pump discharge to pump inlet.

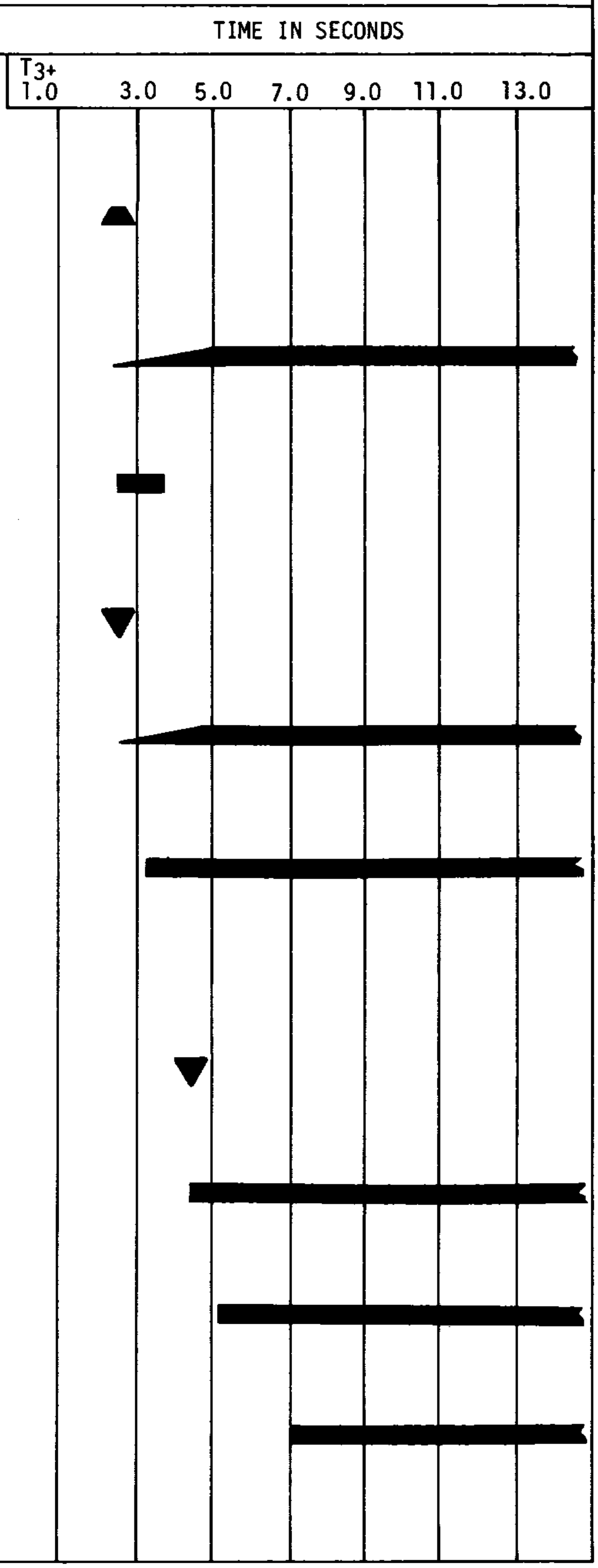


Figure 5-10 (Sheet 3 of 3)

oxidizer turbine bypass valve (13). The ignition phase control valve, when actuated, opens the main fuel valve (7) and the ASI oxidizer valve (8) and supplies pressure to the sequence valve located within the main oxidizer valve (14). Fuel is tapped from downstream of the main fuel valve for use in the ASI (4). Both propellants, under tank pressure, flow through the stationary turbopumps (12).

The sequence valve, in the main fuel valve (12), opens when the fuel valve reaches approximately 90 percent open and routes helium to the start tank discharge valve (STDV) (11) control valve. Simultaneously with engine start, the STDV delay timer is energized. Upon expiration of the STDV timer and the receipt of a stage supplied mainstage enable signal the STDV control valve and ignition phase timer are energized. As the STDV control valve energizes the discharge valve opens allowing gaseous hydrogen under pressure to flow through the series turbine drive system. This accelerates both turbopumps (12) to the proper operating levels to allow subsequent ignition and power build up of the gas generator (16). The relationship of fuel to lox turbopump speed buildup is controlled by an orifice in the oxidizer turbine bypass valve (13). During the start sequence the normally open oxidizer bypass valve (13) permits a percentage of the gas to bypass the oxidizer turbine.

During this period, ASI combustion is detected by the ASI ignition monitor. (Absence of the ignition detection signal or a start tank depressurized signal will cause cutoff at the expiration of the ignition phase timer.) With both signals present at ignition phase timer expiration, the mainstage control valve energizes. Simultaneously, the sparks deenergize timer is energized and the STDV control valve is deenergized, causing the STDV to close. Helium pressure is vented from the main oxidizer valve (14) and from the purge control valve through the mainstage control valve. The purge control valve closes, terminating the oxidizer dome and gas generator oxidizer injector manifold purges (6). Pressure from the mainstage control valve is routed to open the main oxidizer valve (14).

A sequence valve operated by the main oxidizer valve (14) permits gaseous helium to open the gas generator control valve (4) and to close the oxidizer turbine bypass valve (13). (Flow to close the oxidizer turbine bypass valve (13) is restricted as it passes through an orifice. The orifice controls the closing speed of this valve.)

Propellants flowing into the gas generator (16) are ignited by the sparkplugs (4). Combustion of the propellants cause the hot gases to drive the turbopumps (12). The turbopumps rotation causes propellant pressure to buildup. The propellant flow increases and is ignited in the thrust chamber by the torch from the ASI.

Transition into mainstage occurs as the turbopumps (12) accelerate to steadystate speeds. As oxidizer injection pressure increases a thrust OK signal is generated by either of the two thrust OK pressure switches (17). (Cutoff occurs if no signal is received before expiration of the sparks deenergized timer). The ASI and GG sparks exciters are deenergized at expiration of the sparks deenergized timer. Cutoff occurs if both pressure switch actuated signals (thrust OK) are lost during mainstage operation.

Steadystate operation is maintained until a cutoff signal is initiated. During this period, gaseous hydrogen is tapped from the fuel injection manifold to pressurize the LH_2 tank. The lox tank is pressurized by lox heated by the heat exchanger in the turbine exhaust duct.

Propellant utilization is provided by bypassing oxidizer from the oxidizer turbopump outlet back to the inlet. The propellant utilization valve is controlled by signals from the propellant utilization system. The engine mixture ratio varies from about 4.7:1 to 5.5:1.

Nominal engine thrust and specific impulse as a function of mixture ratio for the engines are shown in figure 5-11.

ENGINE CUTOFF

The S-II J-2 engine may receive cutoff signals from several different sources. These sources include engine interlock deviations, EDS automatic and manual abort cutoffs and propellant depletion cutoff. Each of these sources signal the LVDC in the instrument unit (IU). The LVDC sends the engine cutoff signal to the S-II switch selector. The switch selector in turn signals the electrical control package. The electrical control package controls all the local signals for the cutoff sequence.

Cutoff Sequence

The engine cutoff sequence is shown graphically in figure 5-12. When the electrical control package receives the cutoff signal (1), it deenergizes the mainstage and ignition phase control valves in the pneumatic control package (2) while energizing the helium control deenergize timer. The mainstage control valve closes the main oxidizer valve (3) and opens the purge control valve and the oxidizer turbine bypass valve (5). The purge control valve directs a helium purge (11) to the oxidizer dome and GG oxidizer injector. The ignition phase control valve closes the ASI oxidizer valve (4) and the main fuel valve (5) while opening the fast shutdown valve. The fast shutdown valve now rapidly vents the return flow from the GG control valve. All valves except the ASI oxidizer valve (4) and oxidizer turbine bypass valve (8), are spring loaded closed. This causes the valves to start moving closed as soon as the pressure to open them is released. GG combustion pressure aids closing of the GG control valve.

Expiration of the helium control deenergize timer causes the helium control valve to close. When the helium control valve closes it causes the oxidizer dome and GG oxidizer injector purges (11) to stop. An orifice in the locked up lines bleeds off pressure from the propellant bleed valves (13). This loss of pressure allows springs to open the valves. When open, the propellant bleed valves allow propellants to flow back to the propellant tanks.

MALFUNCTION DETECTION

Each engine is provided with a system to detect malfunctions and to effect a safe shutdown. If neither mainstage OK pressure switch has indicated sufficient thrust for mainstage operation of the ignition phase timer, a shutdown of the particular engine is initiated. Once an engine attains mainstage operation, it is shut down if both mainstage OK pressure switches deactuate due to low level thrust.

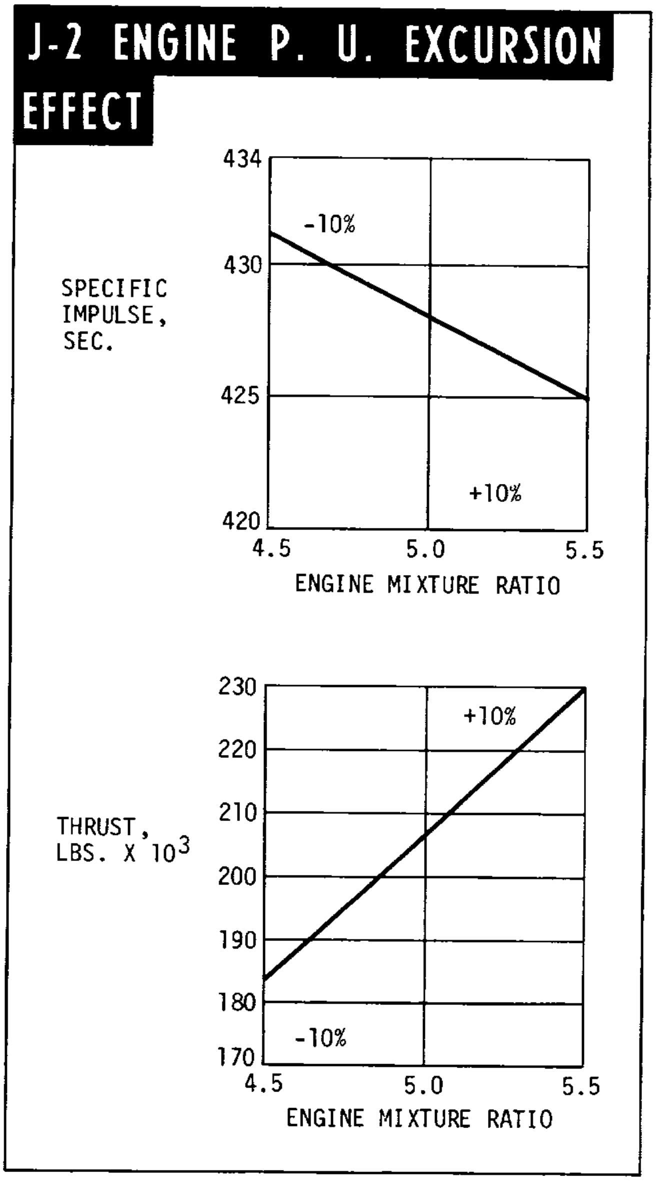


Figure 5-11

FLIGHT CONTROL

The center engine is fixed in place while the four outer engines are gimbaled in accordance with electrical signals from the flight control computer in the instrument unit for thrust vector (flight attitude) control. Each outboard engine is equipped with a separate, independent, closed-loop, hydraulic control system (figure 5-13) that includes two servoactuators mounted in perpendicular planes to provide control over the vehicle pitch, roll and yaw axes. The servoactuators are capable of deflecting the engine $\pm$ 7 degrees in the pitch and yaw planes, $\pm$ 10 degrees diagonally, at the rate of 8 degrees per second.

The primary components of the hydraulic subsystem are an auxiliary pump, a main pump, an accumulator/reservoir manifold assembly, and two servoactuators (figures 5-13 and 5-14). The auxiliary pump is used prior to launch to maintain hydraulic fluid (MIL-H-5606) temperature control between 65 and 105 degrees F. The pump delivers two gallons per minute at 3650 psig and is driven by a 400-cycle ac motor on GSE power.

The main pump is mounted to and driven by the engine lox turbopump. It is used during stage powered flight and delivers hydraulic fluid at 8 gallons per minute at 3500 psig. Prior to launch, the accumulator is pressurized with GN_2 and filled with hydraulic fluid from the pressurized auxiliary pump flow. The reservoir is in turn pressurized by the accumulator through a piston-type linkage. The accumulator/reservoir manifold assembly consists of a high-pressure (3500 psig) accumulator which receives high pressure fluid from the pumps and a low pressure (88 psig) reservoir which receives return fluid from the servoactuators. During engine firing, hydraulic fluid is routed under pressure from the main pump to the pressure manifold of the reservoir/accumulator.

Hydraulic fluid under pressure in the accumulator furnishes high pressure fluid for sudden demands and smooths out pump pulsations. It directs pressure to the two identical electrically controlled, hydraulically powered, servoactuators. The servoactuators have a nominal operation pressure of 3500 psig and provide the necessary forces and support to accurately position the engine in response to flight control system signals. The servoactuator is a power control unit that converts electrical command signals and hydraulic power into mechanical outputs that gimbal the engine. The developed force, direction, and velocity are determined by an electrohydraulic servovalve.

Command signals received from the guidance system are interpreted as linear variations from a known piston position. Hydraulic fluid is then directed to either side of the actuator piston as required to satisfy the guidance command.

Actuator return fluid is routed to the reservoir which stores hydraulic fluid at sufficient pressure to supply a positive pressure at the main pump inlet.

PREFLIGHT OPERATION

During and following propellant loading, the hydraulic system fluid is intermittently recirculated by the electrically driven auxiliary pump in order to prevent the fluid from freezing. Recirculation is terminated just prior to S-IC ignition command. Recirculation is not necessary during S-IC burn, due to the short duration of the burn.

At approximately T-5 minutes, fluid is stored under high pressure in the accumulator by closing both hydraulic lockup valves which are contained in the accumulator/reservoir manifold assembly (figure 5-13).

INFLIGHT OPERATION

After S-IC/S-II stage separation, an S-II switch selector command unlocks the accumulator lockup valves, releasing high pressure fluid to each of the two servoactuators. The accumulator stored fluid provides gimbaling power prior to main hydraulic pump operation. During S-II mainstage operation the main hydraulic pump supplies high pressure fluid to the servoactuators for gimbaling.

PNEUMATICS

The pneumatic control system consists of the ground pneumatic control system and the onboard pneumatic control system. The ground system utilizes helium supplied directly from a ground source. The onboard system utilizes helium from onboard storage spheres.

GROUND PNEUMATIC CONTROL SYSTEM

Ground supplied helium (figure 5-15) controls and actuates various valves during preflight operations. These include the vent valve, fill and drain valves, recirculation return line valve, and main propellant line prevalves. The gas is also used for all engine purges.

ONBOARD PNEUMATIC CONTROL SYSTEMS

The onboard pneumatic control systems consist of a stage propellant valve control system and an engine pneumatic actuation and purge system.

Stage Propellant Valve Control System

The stage onboard pneumatic control system (figure 5-15) is supplied from the helium storage sphere. It is pressurized to 3000 psig at approximately T-4:40.00. Pneumatic pressure from the storage sphere is regulated to 750 psig by the pneumatic control regulator and is utilized during flight to actuate the prevalve and recirculation valves.

PROPELLANTS

The propellant systems supply fuel and oxidizer to the five engines. This is accomplished by the propellant management components and the servicing, conditioning, and engine delivery subsystems.

PROPELLANT SERVICING SYSTEM

Pad servicing operations include the filling, and draining if necessary, of both propellant tanks and the purging of the tank fill lines.

Ground interface is through the umbilicals, to the fill and drain valves, and into the propellant tanks. The propellants then enter the engine feed lines, stopping at the closed main valves. Refer to figure 5-16 for propellant loading procedure. The tanks are vented by pneumatically opening the tank vent valves, two per propellant tank, to allow ullage gas to pass from the tanks. Actuation pressure for the propellant tank vent valves is provided by two separate 750-psig ground-supplied helium systems. One system actuates the liquid oxygen tank vent valves, and the other system actuates the liquid hydrogen tank vent valves. The vent valves are open during propellant loading operations and closed for tank pressurization.

If the launch is aborted, draining of the propellant tanks can be accomplished by pressurizing the tanks, opening the fill valves, and reversing the fill operation.

PROPELLANT CONDITIONING SYSTEMS

Propellant conditioning is required to prevent geysering and stratification in the vehicle storage tanks. It provides propellants of uniform temperature and density.

Lox Recirculation

Lox conditioning by natural convection (figure 5-17) is initiated shortly after start of lox fill and continues until T-30 minutes. At that time, helium is injected into the lox recirculation return line to boost recirculation and continues until approximately 10 seconds prior to S-II ignition. During recirculation, lox prevalves and recirculation return valves remain open. Return line valves are closed at termination of recirculation.

Lox conditioning is accomplished by recirculating the lox down the engine feed ducts (1), through the prevalves (2), the lox turbopump

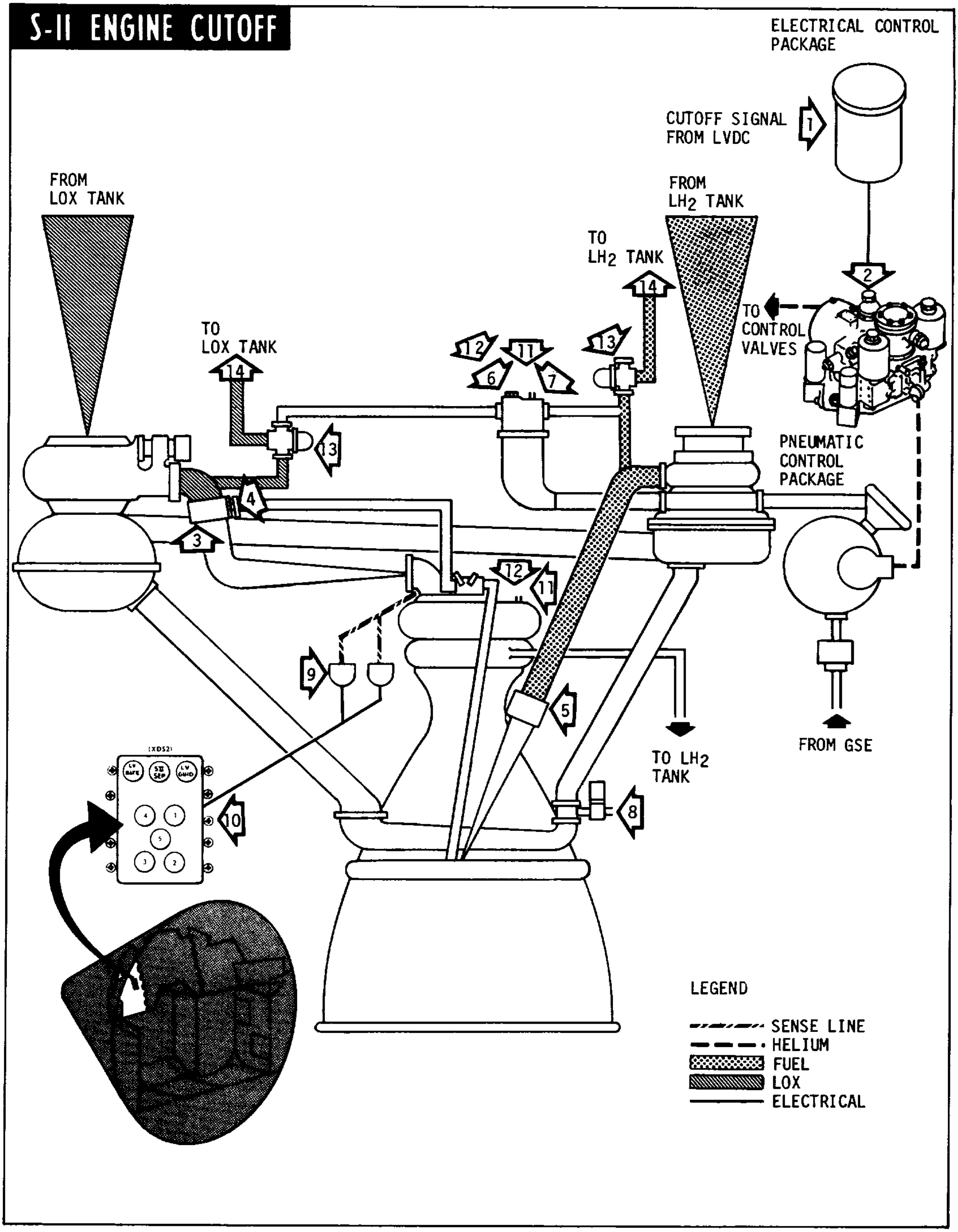


Figure 5-12 (Sheet 1 of 2)

S-II ENGINE CUTOFF

TIME IN SECONDS FROM CUTOFF SIGNAL: 0.0 0.1 0.2 0.3 0.4 0.5 0.6 0.7 0.8 0.9 1.0 1.1 1.2

SEQUENCE	EVENTS
1	Cutoff signal from LVDC triggers
2	valves in pneumatic control package.
3	Pneumatic pressure closes main oxidizer valve;
4	ASI oxidizer valve;
5	main fuel valve;
6	GG oxidizer valve;
7	GG fuel valve and
8	Opens oxidizer turbine bypass valve. (Within 10 seconds.)
9	Thrust OK pressure switches sense loss of thrust. (This event starts time base No. 4)
	NOTE Loss of Thrust could have been the cause of the cutoff signal 1 and have preceeded previous events:
10	Engine out lights illuminate.
11	Helium flow purges oxidizer dome and GG oxidizer injector.
12	Helium control deenergize timer stops oxidizer dome and GG oxidizer purge.
13	Propellant bleed valves open allowing
14	propellant flow.

Figure 5-12 (Sheet 2 of 2)

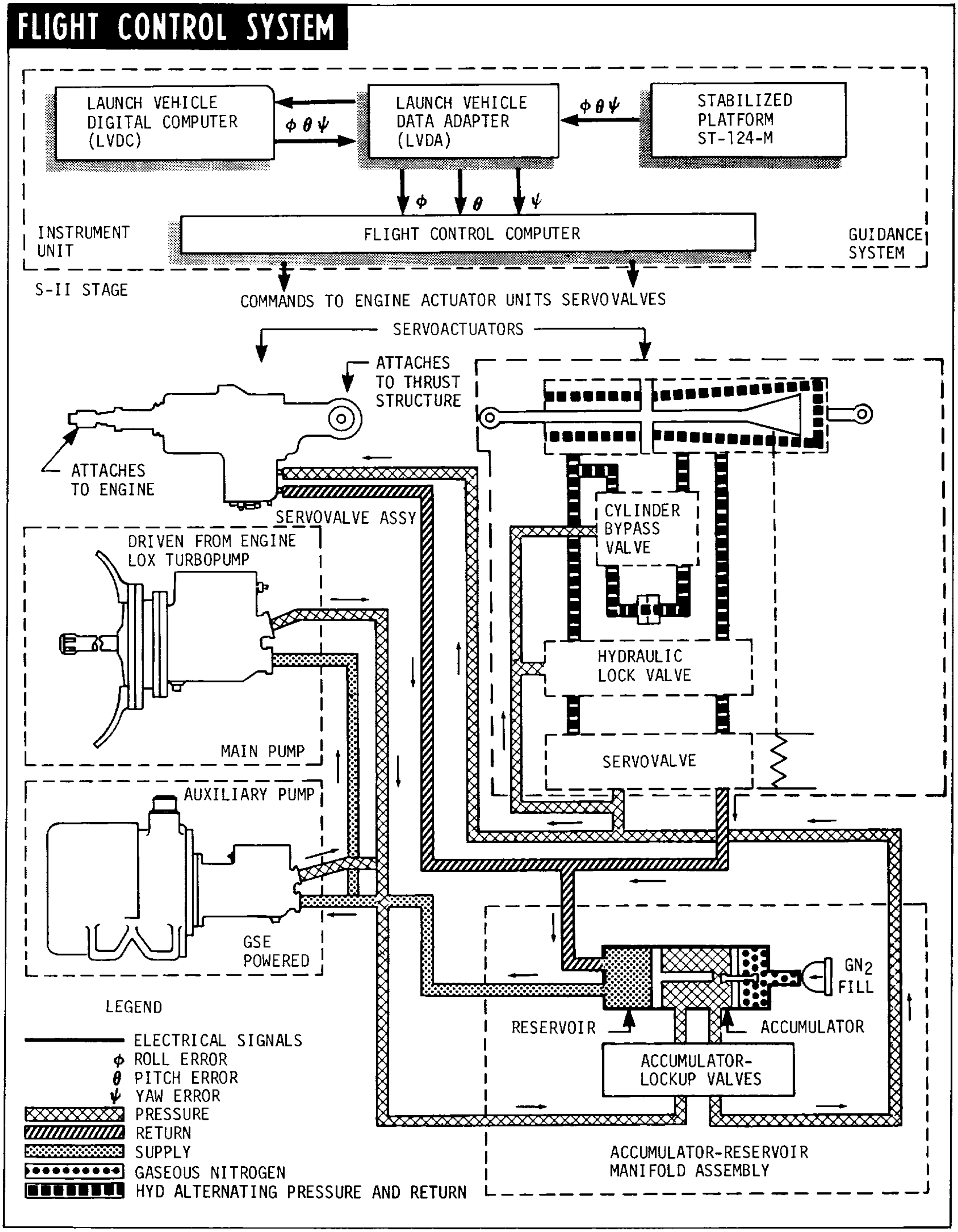

FLIGHT CONTROL SYSTEM
LAUNCH VEHICLE DIGITAL COMPUTER (LVDC)
LAUNCH VEHICLE DATA ADAPTER (LVDA)
STABILIZED PLATFORM ST-124-M
$\phi\ \theta\ \psi$
FLIGHT CONTROL COMPUTER
INSTRUMENT UNIT
GUIDANCE SYSTEM
S-II STAGE
COMMANDS TO ENGINE ACTUATOR UNITS SERVOVALVES
SERVOACTUATORS
ATTACHES TO THRUST STRUCTURE
ATTACHES TO ENGINE
SERVOVALVE ASSY
CYLINDER BYPASS VALVE
HYDRAULIC LOCK VALVE
SERVOVALVE
DRIVEN FROM ENGINE LOX TURBOPUMP
MAIN PUMP
AUXILIARY PUMP
GSE POWERED
RESERVOIR
ACCUMULATOR
GN_2 FILL
ACCUMULATOR-LOCKUP VALVES
ACCUMULATOR-RESERVOIR MANIFOLD ASSEMBLY
LEGEND
ELECTRICAL SIGNALS
ϕ ROLL ERROR
θ PITCH ERROR
ψ YAW ERROR
PRESSURE
RETURN
SUPPLY
GASEOUS NITROGEN
HYD ALTERNATING PRESSURE AND RETURN

Figure 5-13

(3) into the return lines (5) through the engine bleed valves (4) and back into the lox tank.

LH_2 RECIRCULATION

LH_2 recirculation (figure 5-18) is initiated at approximately T-20 minutes and is terminated about 3 seconds prior to S-II ignition. Forced circulation during launch and S-IC boost consists of closing the LH_2 feed line prevalves and starting the LH_2 recirculation pumps (1). A separate recirculation pump is provided for each of the five feed ducts.

LH_2 conditioning is accomplished by pumping the fuel through the recirculation bypass valve (2), into the LH_2 feed ducts downstream of the prevalves, through the LH_2 turbopump (3), into the return lines (5), through the LH_2 bleed valve (4), and back into the fuel tank.

Recirculation is terminated by opening the prevalves, stopping the pumps, and closing the recirculation return valves.

PREPRESSURIZATION

After loading has been completed, and shortly before liftoff, the vent valves are closed and the propellant tanks are pressurized to their required levels by helium from ground supplies. Pressurization of the propellant tanks (lox and LH_2) is required prior to liftoff to provide the required net positive suction head (NPSH) at the turbopump inlets for engine start. It is accomplished from a ground regulated helium source. Pressurization is initiated by the terminal countdown sequencer at approximately T-186 seconds for the lox tank and T-97 seconds for the LH_2 tank. Pressurization is terminated at T-127 for the lox tank and T-37 for the LH_2 tank.

Both propellant tanks are prepressurized in the same manner by separate systems (figures 5-17 and 5-18). At initiation of prepressurization, the tank vent valves are closed and the disconnect valve and ground prepressurization valves are opened to allow GHe at minus 275°F to flow from the ground source through the prepressurization disconnect coupling (6), through the prepressurization solenoid valve (7), and into the tank pressurization line (8). This line carries helium into the propellant tank through the tank gas distributor.

Each propellant tank has a fill overpressure switch (9) for personnel safety. The switch sends a signal to the GSE and is used only during loading.

V
S-II

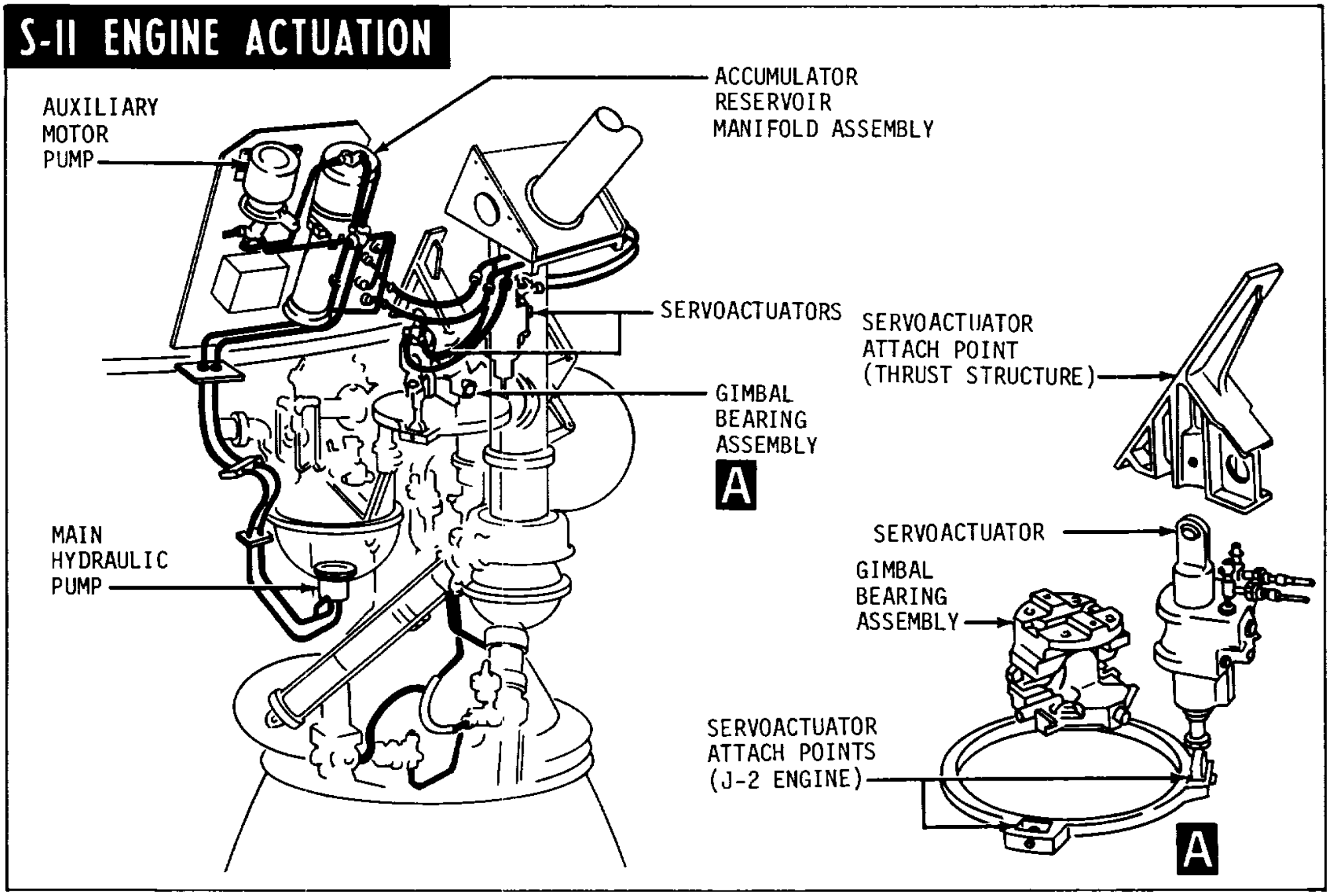


Figure 5-14

The vent valves (11) act as relief valves allowing ullage gas to be vented directly overboard during flight. The LH_2 vent valves operate between 27.5 and 29.5 psig prior to S-II ignition and at 30.5 to 33 psig during S-II burn. The lox vent valves crack at 42 psia and reseat at 39.5 psia.

PROPELLANT DELIVERY SUBSYSTEMS

The function of the engine feed systems (figure 5-19) is to transfer the liquid propellants from the appropriate tanks to the J-2 rocket engines. Each propellant tank is provided with five prevalves which provide open/close control of the flow of propellants through separate feedlines to each engine.

The prevalve is a normally open, pneumatically actuated, electrically controlled, butterfly-gate type valve. A built-in four-way pneumatic control solenoid permits 750 $\pm$ 50 psig helium pressure to actuate the prevalve. Should a loss

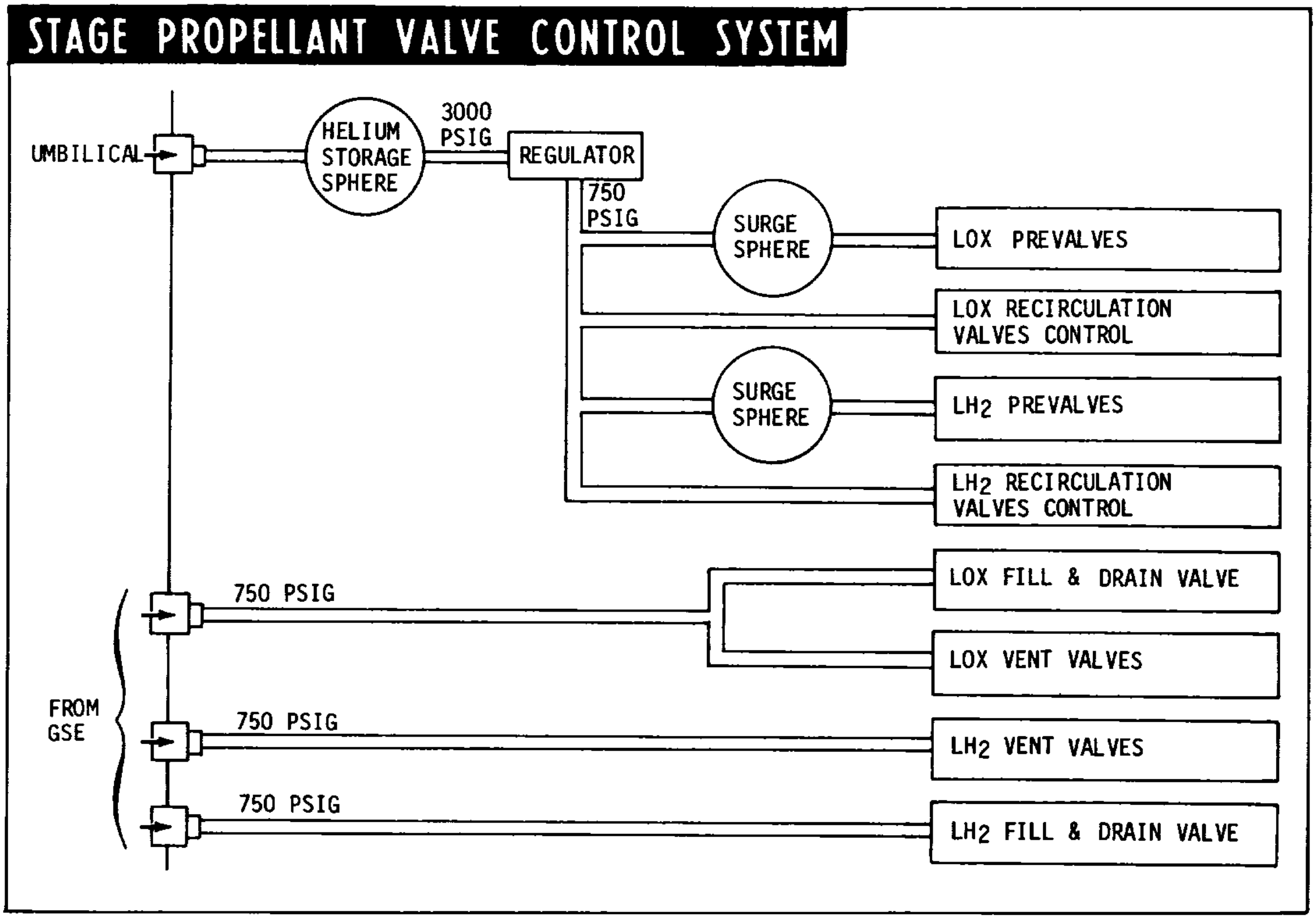


Figure 5-15

PROPELLANT LOADING				
PROPELLANT	TYPE FILL	RATE (GALLONS/MINUTE)	LEVEL (TANK % LEVEL)	COUNT TIME (HR: MIN: SEC)
LOX	PRECOOL FAST SLOW REPLENISH	500 1,200 500 0 to 500	0 to 5 5 to 96 96 to 100	T-7:24:00 TO T-7:17:00 TO T-6:59:00 TO T-6:20:00 TO 0:02:52
LH_2	PRECOOL FAST SLOW REPLENISH	1,000 10,000 1,000 0 to 500	0 to 5 5 to 98 98 to 100	T-5:41:00 TO T-5:11:00 TO T-4:30:00 TO T-3:55:00 TO 0:01:22

Figure 5-16

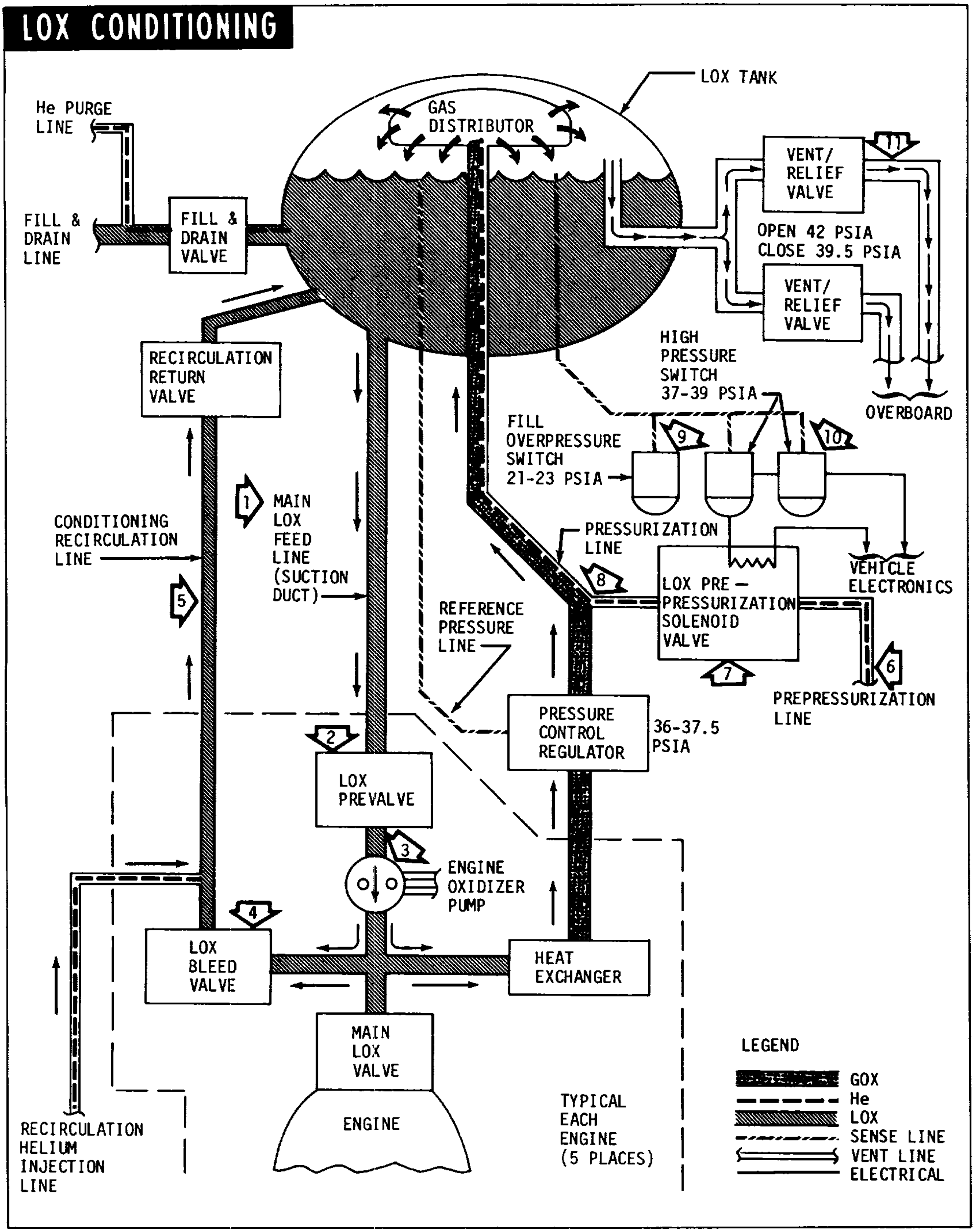

LOX CONDITIONING
LOX TANK
GAS DISTRIBUTOR
He PURGE LINE
FILL & DRAIN LINE
FILL & DRAIN VALVE
VENT/ RELIEF VALVE
OPEN 42 PSIA
CLOSE 39.5 PSIA
VENT/ RELIEF VALVE
HIGH PRESSURE SWITCH 37-39 PSIA
OVERBOARD
RECIRCULATION RETURN VALVE
FILL OVERPRESSURE SWITCH 21-23 PSIA
MAIN LOX FEED LINE (SUCTION DUCT)
CONDITIONING RECIRCULATION LINE
PRESSURIZATION LINE
LOX PRE-PRESSURIZATION SOLENOID VALVE
VEHICLE ELECTRONICS
REFERENCE PRESSURE LINE
PREPRESSURIZATION LINE
PRESSURE CONTROL REGULATOR
36-37.5 PSIA
LOX PREVALVE
ENGINE OXIDIZER PUMP
LOX BLEED VALVE
HEAT EXCHANGER
MAIN LOX VALVE
ENGINE
TYPICAL EACH ENGINE (5 PLACES)
RECIRCULATION HELIUM INJECTION LINE
LEGEND
GOX
He
LOX
SENSE LINE
VENT LINE
ELECTRICAL
1
2
3
4
5
6
7
8
9
10
11

Figure 5-17

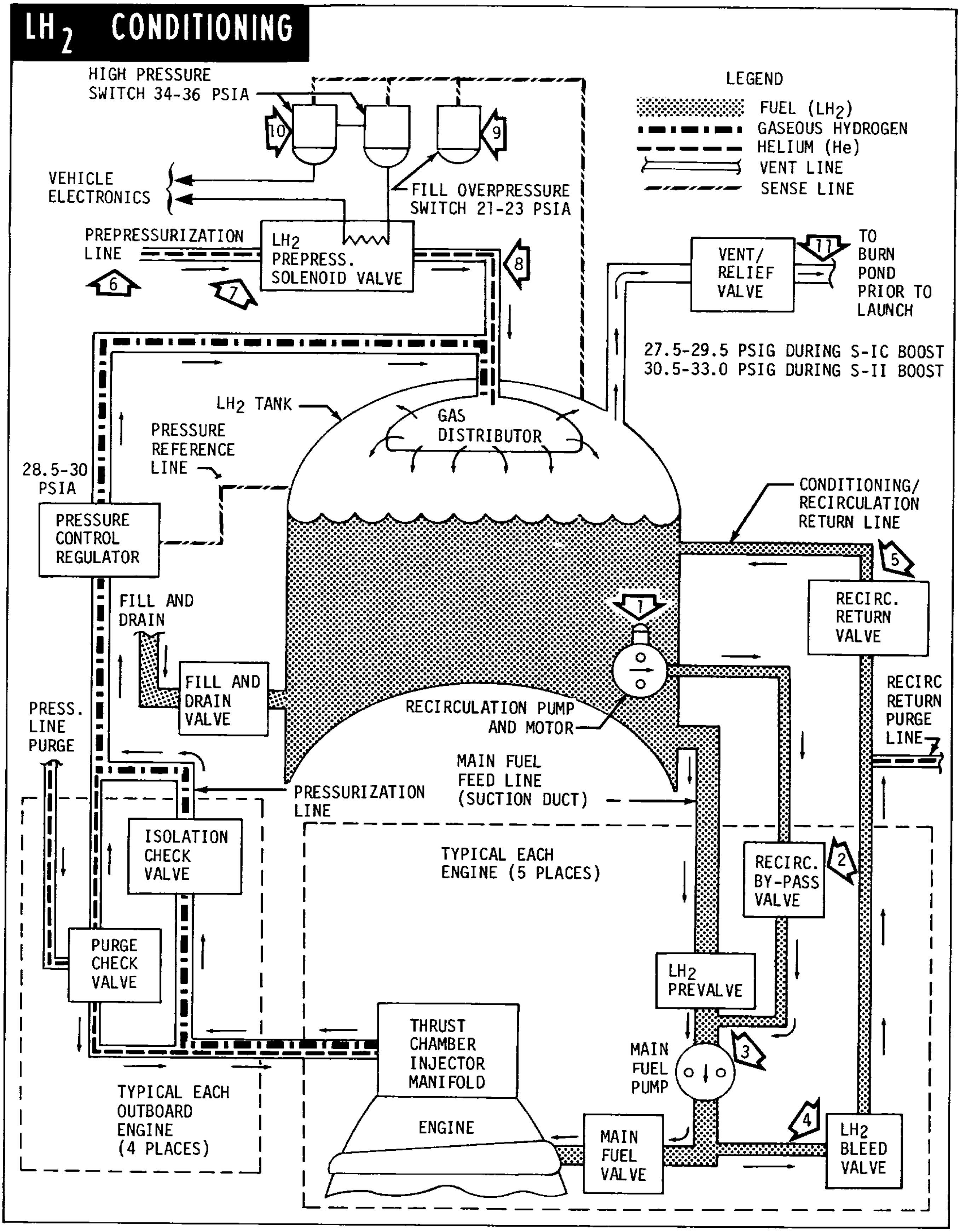

LH2 CONDITIONING
HIGH PRESSURE SWITCH 34-36 PSIA
LEGEND
FUEL (LH2)
GASEOUS HYDROGEN
HELIUM (He)
VENT LINE
SENSE LINE
VEHICLE ELECTRONICS
FILL OVERPRESSURE SWITCH 21-23 PSIA
PREPRESSURIZATION LINE
LH2 PREPRESS. SOLENOID VALVE
VENT/ RELIEF VALVE
TO BURN POND PRIOR TO LAUNCH
27.5-29.5 PSIG DURING S-IC BOOST
30.5-33.0 PSIG DURING S-II BOOST
LH2 TANK
GAS DISTRIBUTOR
PRESSURE REFERENCE LINE
28.5-30 PSIA
PRESSURE CONTROL REGULATOR
CONDITIONING/ RECIRCULATION RETURN LINE
RECIRC. RETURN VALVE
FILL AND DRAIN
FILL AND DRAIN VALVE
RECIRCULATION PUMP AND MOTOR
RECIRC RETURN PURGE LINE
PRESS. LINE PURGE
MAIN FUEL FEED LINE (SUCTION DUCT)
PRESSURIZATION LINE
ISOLATION CHECK VALVE
TYPICAL EACH ENGINE (5 PLACES)
RECIRC. BY-PASS VALVE
PURGE CHECK VALVE
LH2 PREVALVE
THRUST CHAMBER INJECTOR MANIFOLD
MAIN FUEL PUMP
TYPICAL EACH OUTBOARD ENGINE (4 PLACES)
ENGINE
MAIN FUEL VALVE
LH2 BLEED VALVE
1
2
3
4
5
6
7
8
9
10
11

Figure 5-18

of pneumatic or electrical power occur, the prevalve is spring actuated to the open position. The prevalve remains open during S-II powered flight unless a signal is received from the engine shutdown system.

LOX FEED SYSTEM

The lox feed system furnishes lox to the five engines. This system includes four 8-inch, vacuum jacketed feed ducts, one uninsulated feed duct, and five normally open prevalves. At engine start, lox flows from the tank, through the prevalves and feed lines, to each engine. Approximately 300 milliseconds after main valve closure, the lox prevalves are closed, providing a redundant shutoff for the lox feed system.

LH_2 FEED SYSTEM

The LH_2 feed system furnishes LH_2 to the five engines. This system includes five 8-inch vacuum-jacketed feed ducts and five normally open prevalves. The prevalves are closed following tank loading and remain closed until

PROPELLANT FEED

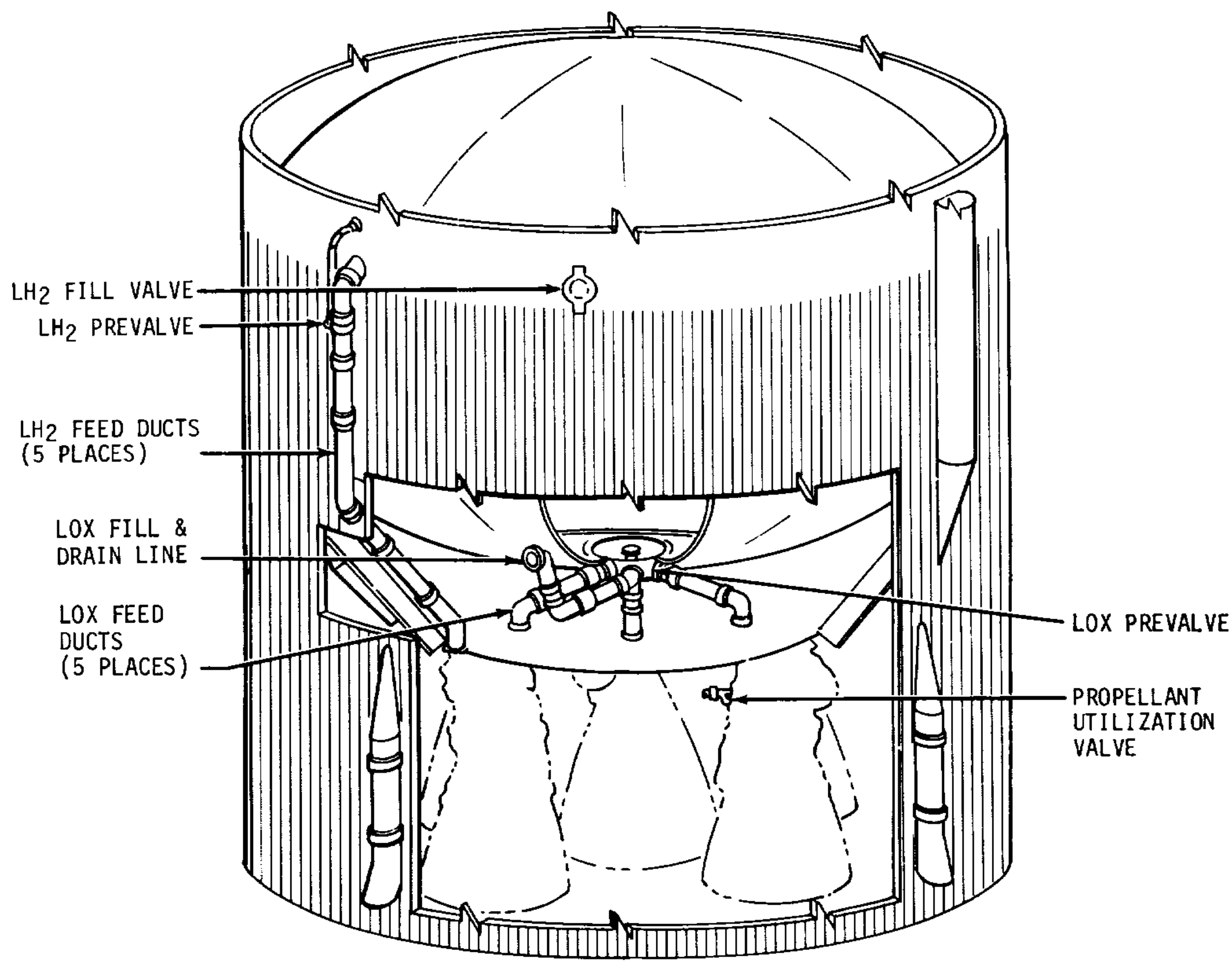


Figure 5-19

approximately 3 seconds prior to S-II ignition command. At engine start, LH_2 flows from the tank, through the prevalves and feed lines, to each engine. Approximately 425 milliseconds after main valve closure, the prevalves are closed, providing a redundant shutoff for the LH_2 feed system.

LOX TANK PRESSURIZATION

Lox tank pressurization (figure 5-20) is initiated at S-II ignition and continues until engine cutoff. Pressurization is accomplished with gaseous oxygen obtained by heating lox bled from the lox turbopump outlet.

When the turbine discharge pressure reaches a pressure differential of 100 psi, a portion of the lox (1) supplied to the engine is diverted into the heat exchanger (2) where it is turned into gox. The gox flows from each heat exchanger (3) into a common pressurization duct (4), through the tank pressurization regulator (5), and into the tank through the gas distributor (6). The flowrate is varied according to the tank ullage pressure, which is sensed by the reference pressure line connecting the tank and the tank pressurization regulator (5). The tank pressurization regulator provides continuous regulation of the tank pressure throughout S-II powered flight.

LH_2 TANK PRESSURIZATION

During S-II powered flight gaseous hydrogen (GH_2) for LH_2 tank pressurization (figure 5-20) is bled from the thrust chamber hydrogen injector manifold (7) of each of the four outboard engines. After S-II engine ignition, liquid hydrogen is preheated in the regenerative cooling tubes of the engine and tapped off from the thrust chamber injector manifold in the form of GH_2 to serve as a pressurizing medium.

The GH_2 passes from each injector manifold into a stage manifold (8), through the pressurization line and tank pressurization regulator (9), and into the tank through the LH_2 tank gas distributor (10). The flowrate is varied according to the LH_2 tank ullage pressure, which is sensed by the reference pressure line (11) connecting the LH_2 tank and the tank pressurization regulator (9).

At approximately 300 seconds after S-II engine ignition a step pressurization command from the stage switch selector activates the regulator to a fully open position, where it remains the rest of S-II boost. When the regulator is in the full open position, LH_2 tank pressure increases to a nominal 33 psia. Pressure in excess of 33 psia is prevented by the LH_2 tank vent valve. This is to compensate for the loss of head pressure caused by the lowering of the fuel level in the tank.

PROPELLANT MANAGEMENT

The propellant management system (figure 5-21) monitors propellant mass for control of propellant loading, utilization, and depletion. Components of the system include continuous capacitance probes, propellant utilization valves, liquid level sensors, and electronic equipment.

Propellant Loading

The main function of the propellant loading system is to control loading and maintain propellant at a predetermined level. The control of propellant loading and replenishing is performed by a ground-based computer in conjunction with stage-mounted electronic equipment. This equipment continuously monitors the output of both full length capacitance probes which sense the liquid mass in the LH_2 and lox tanks. Discrete liquid level sensors mounted on the LH_2 and lox capacitance probes monitor propellant liquid levels as discrete functions.

Propellant Utilization

The propellant utilization system is integrated with the propellant loading system, sharing the full length tank probes.

During flight, the signals from the tank continuous capacitance probes are monitored and compared to provide an error signal to the propellant utilization valve on each lox pump. Based on this error signal the propellant utilization valves are positioned to minimize residual propellants and assure a fuel-rich cutoff by varying the amount of lox delivered to the engines.

Propellant level monitoring is not a control type system but does monitor propellant levels during S-II firing and provides calibration accuracy for the continuous capacitance probes. The point sensors are mounted on a continuous stillwell adjacent to and parallel with the capacitance probe in each tank.

Propellant Depletion

Five discrete liquid level sensors per propellant tank provide initiation of engine cutoff upon detection of propellant depletion. The LH_2 tank sensors are located above each feedline outlet while the lox tank sensors are located directly above the sump. The cutoff sensors will initiate a signal to shutdown the engines when two out of five engine cutoff signals from the same tank are received.

PROPELLANT TANK PRESSURIZATION

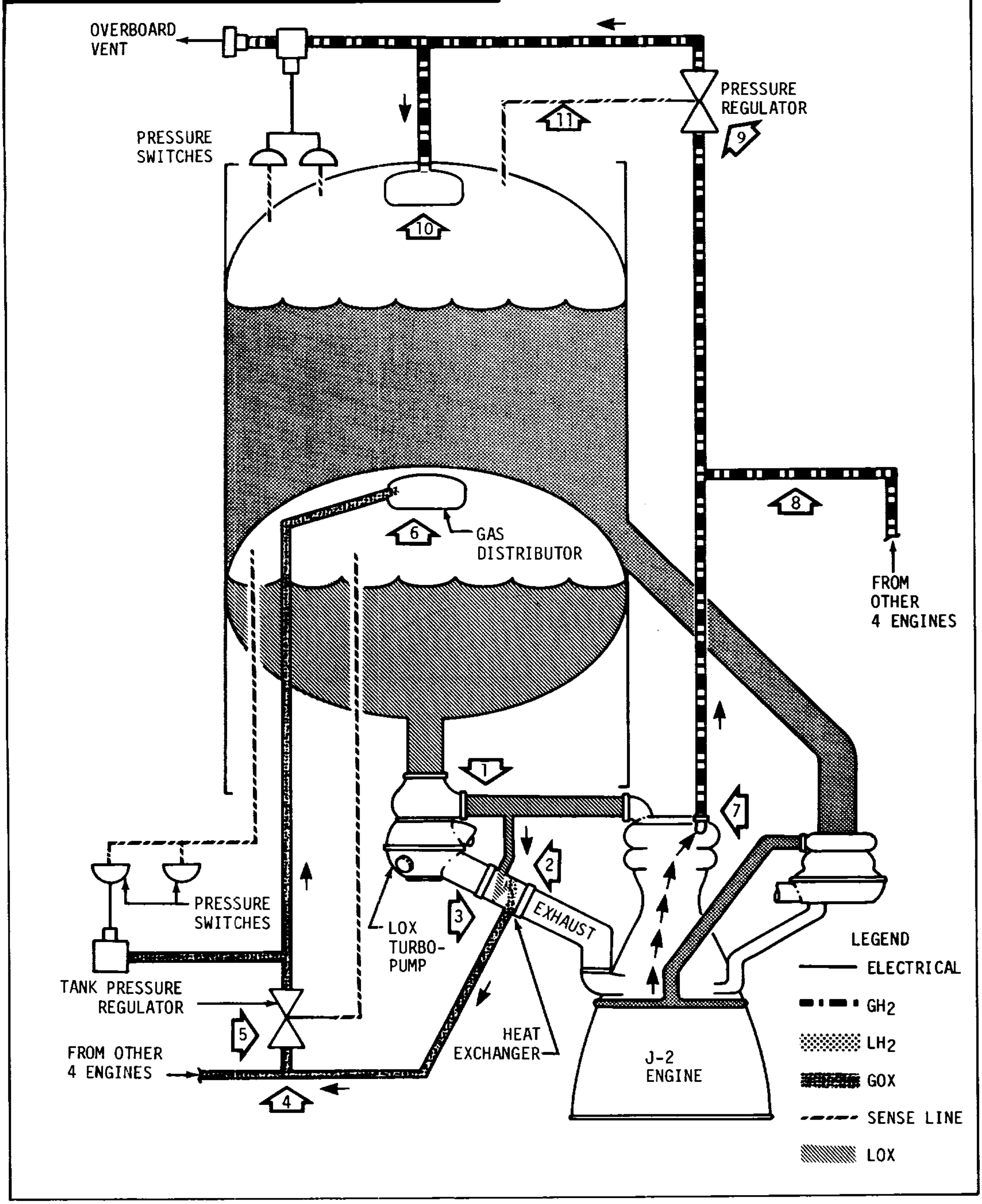


Figure 5-20

V

PROPELLANT MANAGEMENT

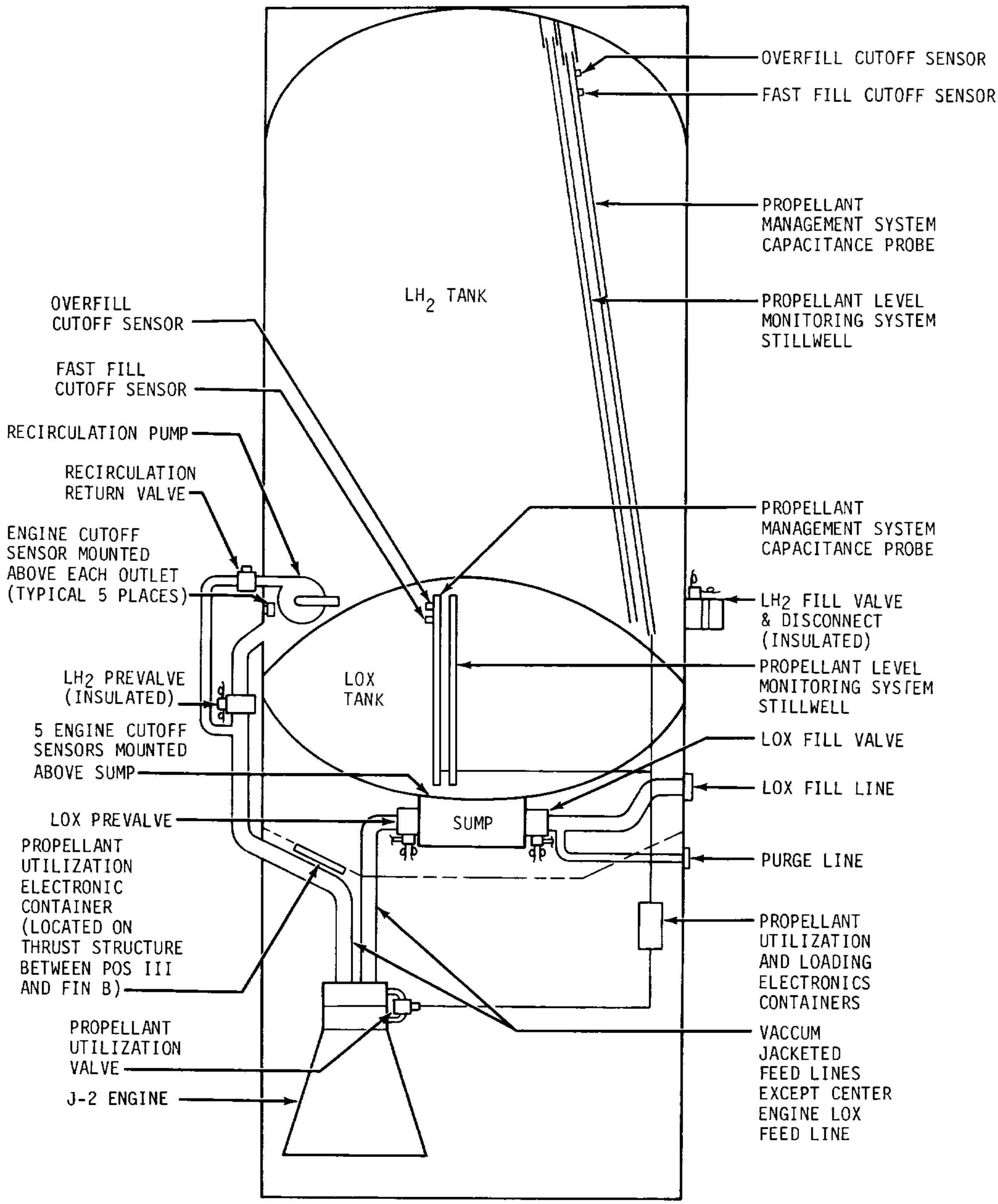


Figure 5-21

ELECTRICAL

The electrical system is comprised of the elec-ical power and electrical control subsystems. The electrical power system provides the S-II stage with the electrical power source and distribution. The electrical control system interfaces with the IU to accomplish the mission requirements of the stage. The LVDC in the IU controls inflight sequencing of stage functions through the stage switch selector. The stage switch selector can provide up to 112 individual outputs in response to the appropriate commands. These outputs are routed through the stage electrical sequence controller or the separation controller to accomplish the directed operation. These units are basically a network of low power transistorized switches that can be controlled individually and, upon command from the switch selector, provide properly sequenced electrical signals to control the stage functions.

Figure 5-22 shows the location of the stage electrical system equipment with relation to the engines.

ELECTRICAL POWER SYSTEM

The electrical power system consists of six dc bus systems and a ground supplied ac bus system. In flight the electrical power system busses are energized by four zinc-silver oxide batteries. See figure 5-23 for battery characteristics. An integral heater and temperature probe are included in each battery. Power for battery heaters and for auxiliary hydraulic pump motors is supplied by GSE and is available only during prelaunch operations. Stage-mounted motor driven power transfer switches are employed to remotely disconnect all batteries from busses until just before launch. Approximately 30 seconds prior to liftoff, a power transfer sequence is initiated which changes the source power over to the stage mounted batteries. During the prelaunch checkout period all electrical power is supplied from GSE.

The motorized power transfer switches have a make-before-break (MBB) action to prevent power interruption during transfer from ground power to onboard battery power.

Each power source has an independant distribution system. There are no provisions for switching between the primary power sources or their associated distribution systems. No electrical failure of any type in one system can cause a failure in the other systems.

Distribution

Figure 5-24 illustrates the electrical system distribution. The loads have been distributed between the various busses in accordance with the following criteria:

1. Inflight loads, critical to mission continuance without performance degradation, are supplied by the main dc bus.
2. All instrumentation loads are supplied by the instrumentation dc bus.
3. All loads operational only on the ground are isolated from flight loads and supplied from ground power.
4. Two independent power sources supply the propellant dispersion and emergency detection systems.
5. The recirculation pump motor system is supplied from a 56 volt dc system (two 28 volt batteries in series).

The division of loads between main dc bus and instrumentation dc bus leads to several advantages: closer voltage regulation and freedom from voltage variations is obtained; the number of loads on the main dc bus may be minimized and thus potential failure modes for the bus system minimized; instrumentation of most stage systems is still maintained after partial or total failure of the main dc bus system so that failure analysis capability is maintained.

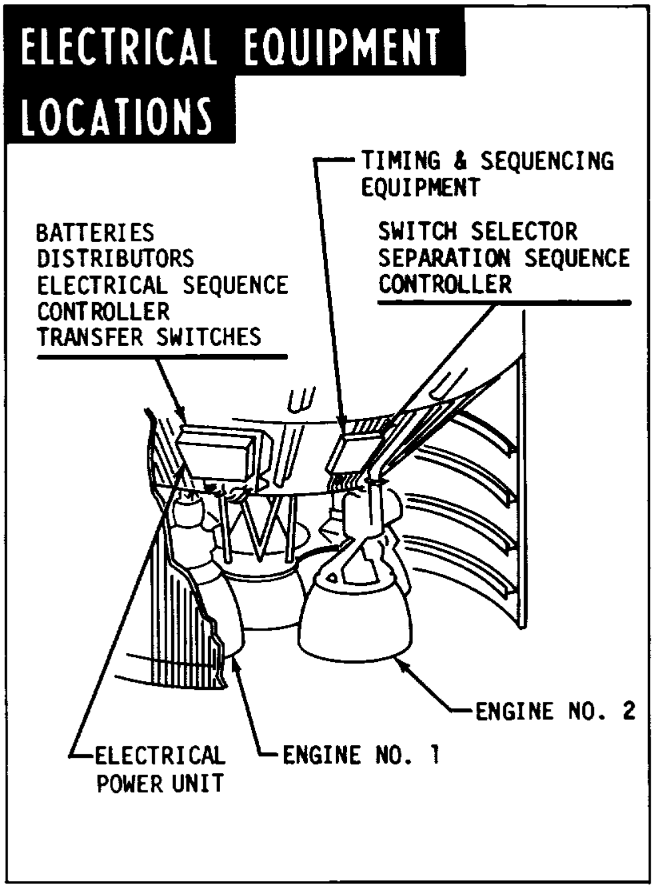


Figure 5-22

V S-II

S-II BATTERY CHARACTERISTICS		
Type	Dry Charge	
Material	Alkaline Silver-zinc	
Electrolyte	Potassium Hydroxide (KOH)	
Cells	20 with taps to reduce voltage as required	
Nominal Voltage	28 ± 2 vdc	
	TYPE I	TYPE II
Nominal Rating	25 Amp Hours	35 Amp Hours
Gross Weight	49.5 POUNDS (each)	41.0 POUNDS

Figure 5-23

Primary power is fed to high current capacity busses in the power distributor. Power is then routed to auxiliary and control distributors, or to measuring distributors for instrumentation power. Components which require high current levels are supplied directly from the main power distributor busses.

Main DC Bus

The main battery provides 28 volts dc to the main dc bus. Power during the prelaunch period is supplied from a regulated dc ground power supply. Transfer from ground power to onboard battery power is accomplished with a motorized power transfer switch.

Instrumentation DC Bus

The instrumentation battery provides 28 volts dc to the instrumentation bus. Power during the prelaunch period is supplied from a regulated dc ground power supply.

Recirculation DC Bus

Two 28 volt batteries connected in series supply 56 vdc to the recirculation bus. During the prelaunch period power is supplied from the ground support equipment (GSE) by a regulated dc power supply. The recirculation bus supplies power to five inverters which convert the 56 vdc to 42 volts rms, three phase, 400 Hz. The inverters are turned on and off individually by five magnetically latching relays under control of the switch selector. The inverters supply power to five LH_2 recirculation pump motors.

Ignition DC Bus

A center tap from the recirculation dc battery system supplies 28 volts dc to the ignition bus. The ignition bus supplies power to the five J-2 engine ignition busses. Power during the prelaunch period is supplied from the main dc bus regulated ground power supply.

Heater DC Bus

The heater dc bus receives 28 volts dc from a regulated ground power supply. It supplies power to the J-2 engine instrumentation package heaters and the battery heaters prior to liftoff.

Ground DC Busses

The ground dc busses receive 28 vdc from regulated GSE power supplies. They provide, prior to liftoff, a means of monitoring stage systems such as propellant loading, pressurization, J-2 engines, flight control system, and power transfer switches.

ELECTRICAL CONTROL SYSTEM

The electrical control system provides electrical control for the various stage-mounted systems to implement normal flight operations, or prelaunch operations and checkout functions. The electrical control system contains most of the electrical and electronic components that are installed on the stage and required by the various mechanical systems for normal stage operation.

The stimuli for the electrical control system are provided by sources such as: IU, switch selector, propellant management system, flight termination system, EDS, and GSE. The electrical control system is arranged to perform the following functions.

1. Control propellant fill valve solenoids for ground filling and detanking operations.
2. Sense tanked propellant mass and control the propellant utilization valves.
3. Sense engine compartment temperature and provide a control signal that defines the purge gas temperature.
4. Control propellant tank pressurization and vent valve solenoids for ground or flight operation.
5. Provide control wiring from the flight control engine actuators to the instrument unit.
6. Control the hydraulic accumulator lockup solenoids for ground and flight operation, and control of the hydraulic circulation motors for ground operation.
7. Control the propellant chilldown system

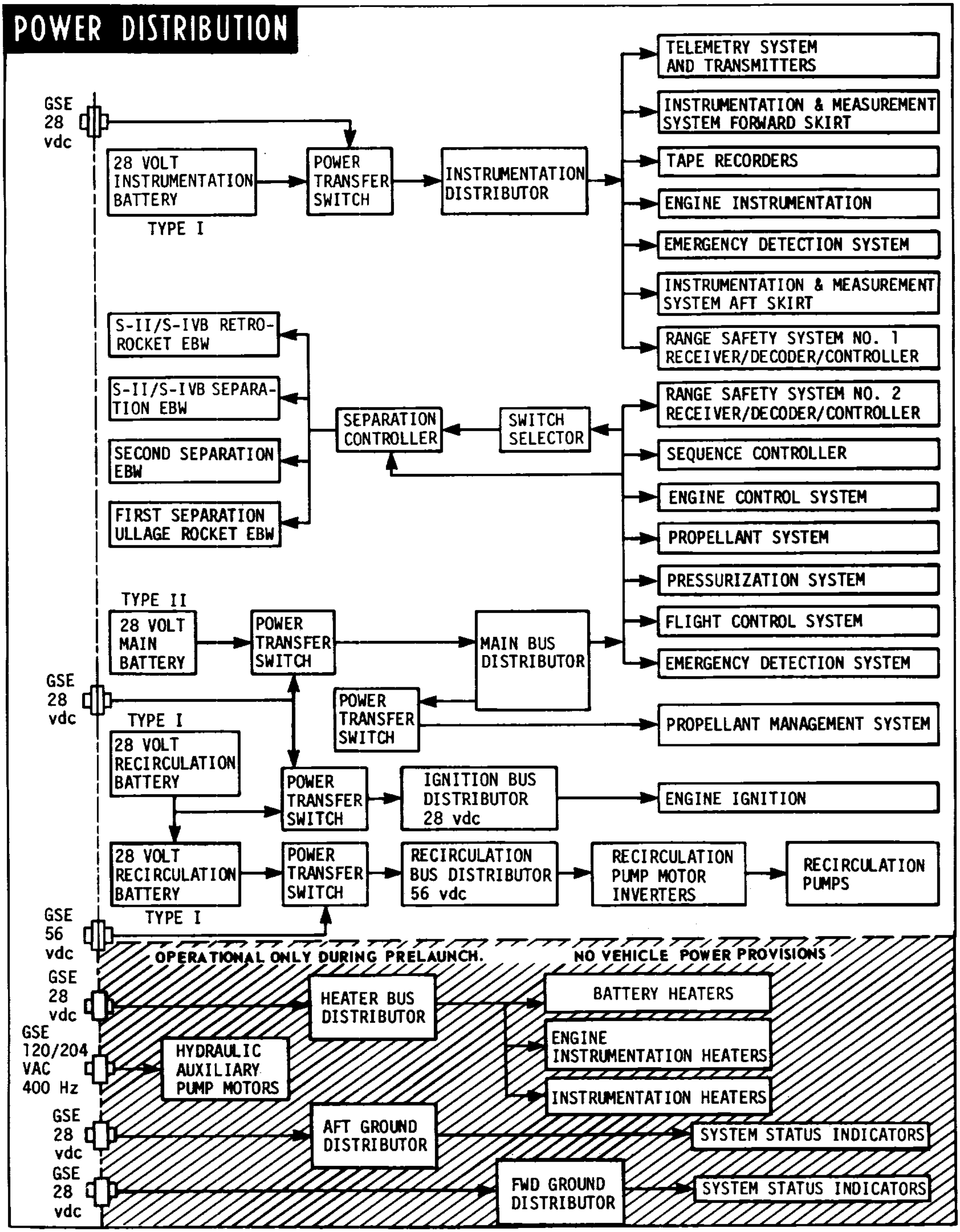

POWER DISTRIBUTION
GSE 28 vdc
28 VOLT INSTRUMENTATION BATTERY
TYPE I
POWER TRANSFER SWITCH
INSTRUMENTATION DISTRIBUTOR
TELEMETRY SYSTEM AND TRANSMITTERS
INSTRUMENTATION & MEASUREMENT SYSTEM FORWARD SKIRT
TAPE RECORDERS
ENGINE INSTRUMENTATION
EMERGENCY DETECTION SYSTEM
INSTRUMENTATION & MEASUREMENT SYSTEM AFT SKIRT
RANGE SAFETY SYSTEM NO. 1 RECEIVER/DECODER/CONTROLLER
S-II/S-IVB RETRO-ROCKET EBW
S-II/S-IVB SEPARATION EBW
SECOND SEPARATION EBW
FIRST SEPARATION ULLAGE ROCKET EBW
SEPARATION CONTROLLER
SWITCH SELECTOR
RANGE SAFETY SYSTEM NO. 2 RECEIVER/DECODER/CONTROLLER
SEQUENCE CONTROLLER
ENGINE CONTROL SYSTEM
PROPELLANT SYSTEM
PRESSURIZATION SYSTEM
FLIGHT CONTROL SYSTEM
EMERGENCY DETECTION SYSTEM
TYPE II
28 VOLT MAIN BATTERY
POWER TRANSFER SWITCH
MAIN BUS DISTRIBUTOR
GSE 28 vdc
POWER TRANSFER SWITCH
PROPELLANT MANAGEMENT SYSTEM
TYPE I
28 VOLT RECIRCULATION BATTERY
POWER TRANSFER SWITCH
IGNITION BUS DISTRIBUTOR 28 vdc
ENGINE IGNITION
28 VOLT RECIRCULATION BATTERY
TYPE I
POWER TRANSFER SWITCH
RECIRCULATION BUS DISTRIBUTOR 56 vdc
RECIRCULATION PUMP MOTOR INVERTERS
RECIRCULATION PUMPS
GSE 56 vdc
OPERATIONAL ONLY DURING PRELAUNCH.
NO VEHICLE POWER PROVISIONS
GSE 28 vdc
HEATER BUS DISTRIBUTOR
BATTERY HEATERS
ENGINE INSTRUMENTATION HEATERS
INSTRUMENTATION HEATERS
GSE 120/204 VAC 400 Hz
HYDRAULIC AUXILIARY PUMP MOTORS
GSE 28 vdc
AFT GROUND DISTRIBUTOR
SYSTEM STATUS INDICATORS
GSE 28 vdc
FWD GROUND DISTRIBUTOR
SYSTEM STATUS INDICATORS

Figure 5-24

V

for ground and flight operations.

8. Starting and cutoff of the J-2 engine systems; monitoring of certain engine systems conditions and either act on these conditions or transmit this information to other systems for action as required.

9. Operation of the propellant prevalves for propellant chilldown and after engine shutdown in flight.

INSTRUMENTATION

The S-II instrumentation system consists of both operational and R & D measurement and telemetry systems. The measurement system detects and measures the condition of the S-II systems while the telemetry system transmits the information gathered by the measurement system to ground stations.

MEASUREMENT SYSTEM

The measurement system senses performance parameters and feeds signals to the telemetry system. It includes transducers, signal conditioners, and distribution equipment necessary to provide the required measurement ranges and to present suitably scaled voltage signals to the inputs of the telemetry system (figure 5-25).

The S-II measuring system detects the physical phenomena to be measured and transforms these phenomena into electrical signals. It also processes and conditions the measured signals into the proper form for telemetering, and distributes these signals to the proper channels of the telemetry system.

Measurements fall into a number of basic categories depending upon the type of measured variable, the variable rate of change with time, and other considerations. Because the stage engines are ignited in flight, a large number of engine and environmental control measurements are required (figure 5-26).

TELEMETRY SYSTEM

The telemetry system accepts the signals produced by the measuring portion of the instrumentation system and transmits them to the ground

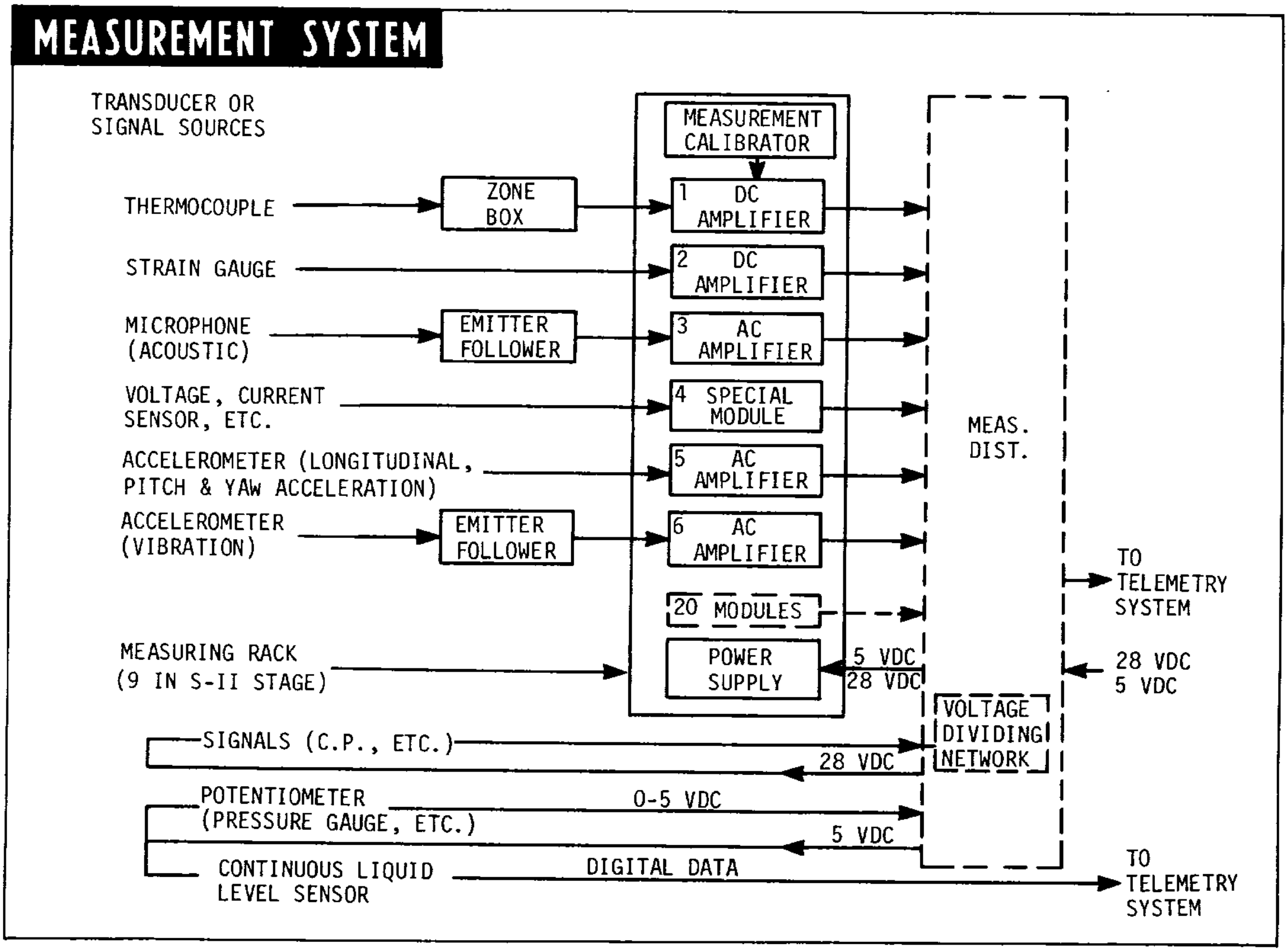


Figure 5-25

STAGE MEASUREMENTS	
TYPE	QTY.
Acceleration	11
Acoustic	5
Temperature	310
Pressure	176
Vibration	60
Flowrate	10
Position	36
Discrete Signals	225
Liquid Level	4
Voltage, Current, Frequency	60
Strain	16
RPM	10
Miscellaneous	4
TOTAL	927

Figure 5-26

stations. Telemetry equipment includes signal multiplexers, subcarrier oscillators, amplifiers, modulators, transmitters, RF power amplifiers, RF multiplexers (mixers) and an omnidirectional system of four antennae. This equipment also includes tape recorders for recording critical data during separation telemetry blackout (separation period) for later play back during stage free fall. The telemetry subsystems use multiplex techniques (signal mixing and time sharing) to transmit large quantities of measurement data over a relatively small number of basic RF links.

Inflight data is transmitted in the form of frequency-modulated RF carriers in the 225 to 260-MHz band (figure 5-27) through the common omnidirectional antenna system.

Several telemetry subsystems are provided in the S-II stage. Telemetry data is grouped in three general catagories: low frequency data, medium frequency data, and high frequency data. Several different modulation techniques are employed in the telemetry systems to facilitate both quality and quantity of measured parameters. These modulation techniques include: Pulse amplitude modulation/frequency modulation/frequency modulation (PAM/FM/FM) and pulse code modulation/frequency modulation (PCM/FM) for low-frequency data; frequency modulation/frequency modulation (FM/FM) for medium-frequency data; and single sideband/frequency modulation (SS/FM) for high-frequency information. A pulse code modulation/digital data acquisition system (PCM/DDAS) transmits measurements by coaxial cable for automatic ground checkout of the stage (figure 5-28).

The PCM/DDAS assembly converts analog transducer signals into digital representations, combines these representations with direct inputs, such as those from the guidance computer, and arranges this information into a format for transmission to the ground station on a 600 kHz carrier signal by means of coaxial cable for ground checkout of the stage.

ANTENNAE

The telemetry antennae have been successfully used by vehicle measurement systems on several programs, including Thor and Saturn I. Four antennae, installed at 90 degree intervals, are employed to provide omnidirectional coverage.

They are located around the S-II forward skirt at vehicle azimuths of 80.5, 170.5, 260.5, and 350.5 degrees.

The antennae are linear cavity-backed slot antennae which are fed from a hybrid junction ring and power dividers.

ORDNANCE

The S-II ordnance systems include the separation, ullage rocket, retrorocket, and propellant dispersion (flight termination) systems.

SEPARATION SYSTEM

The Saturn V launch vehicle system provides for separation of an expended stage from the remainder of the vehicle. For S-IC/S-II separation, a dual plane separation technique is used wherein the structure between the two stages is severed at two different planes (figure 5-29). The S-II/S-IVB separation occurs at a single plane (figure 5-29). All separations are controlled by the launch vehicle digital computer (LVDC) located in the instrument unit (IU).

A sequence of events for S-IC/S-II/S-IVB separations and a block diagram of the separation systems is contained on figure 5-30.

Ordnance for first plane separation consists of two exploding bridgewire (EBW) firing units, two EBW detonators, and one linear shaped charge (LSC) assembly which includes the LSC with a detonator block on each end (figures 5-29 and 5-30). The EBW firing units are installed on the S-IC/S-II interstage slightly below the S-II first separation plane. The leads of the EBW firing units are attached to the EBW detonators which are installed in the detonator blocks of the LSC assembly. The LSC detonator blocks are

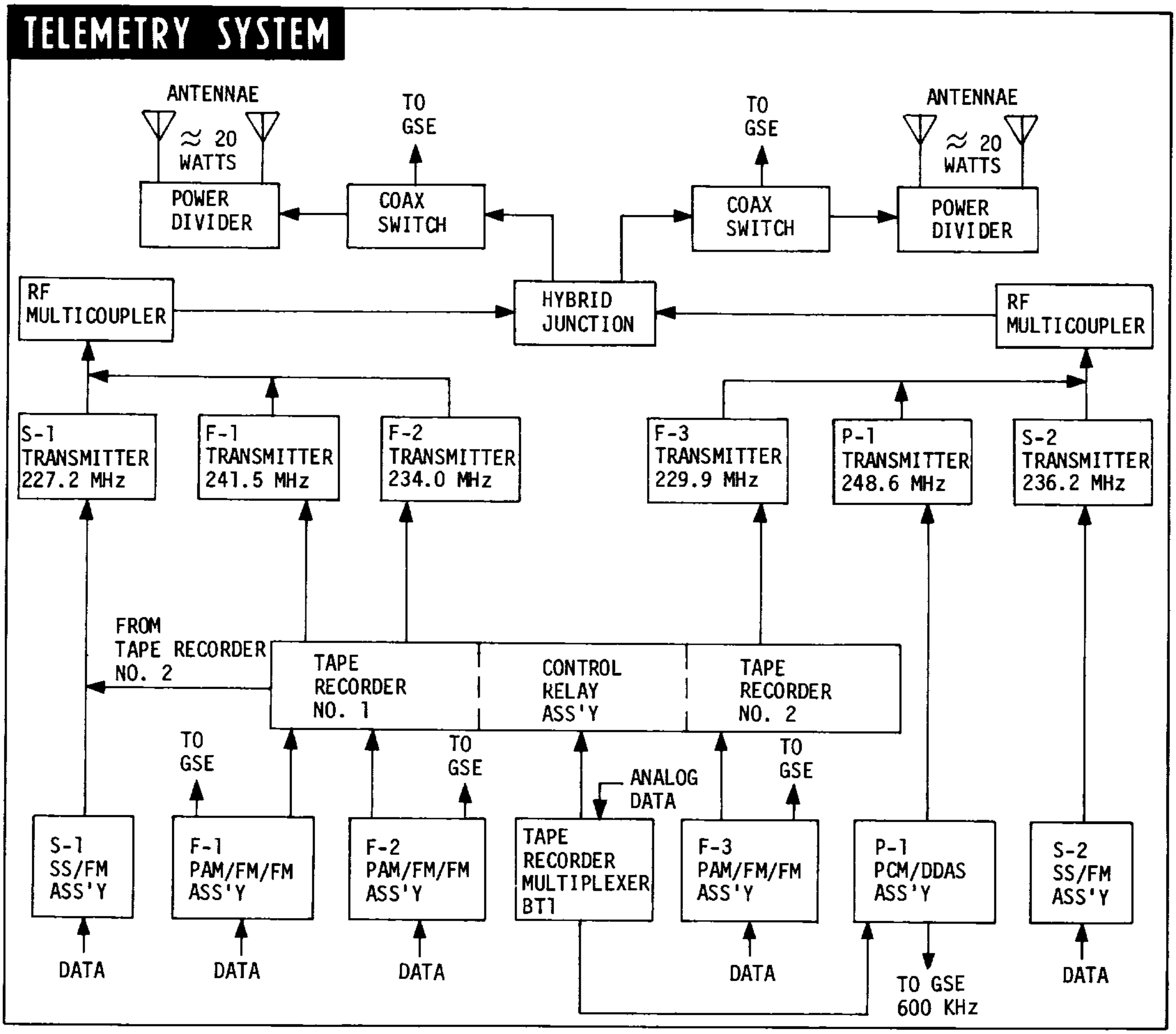


Figure 5-27

installed on adjustable mounts to provide for length variations of the LSC assembly and the circumference tolerances of the interstage. The LSC is routed from the detonator blocks around the periphery of the interstage.

The LSC is held in place by retaining clips and encased by covers which are secured by clips and sealed to environmentally protect the LSC. The two EBW firing units provide redundant signal paths for initiation of the LSC assembly. The storage capacitor in each of the EBW firing units is charged by 28 vdc power during the latter part of S-IC boost. The trigger signal causes the storage capacitor to discharge into an EBW detonator which explodes the bridgewire to release energy to detonate the explosive charge in the detonator. The output of the detonators initiates each end of the LSC assembly.

MODULATION TECHNIQUE AND FREQUENCY		
LINK NO.	MODULATION	FREQUENCY MHz
BF-1	PAM/FM/FM	241.5
BF-2	PAM/FM/FM	234.0
BF-3	PAM/FM/FM	229.9
BP-1	PCM/FM	248.6
BS-1	SS/FM	227.2
BS-2	SS/FM	236.2

Figure 5-28

STAGE SEPARATION SYSTEMS

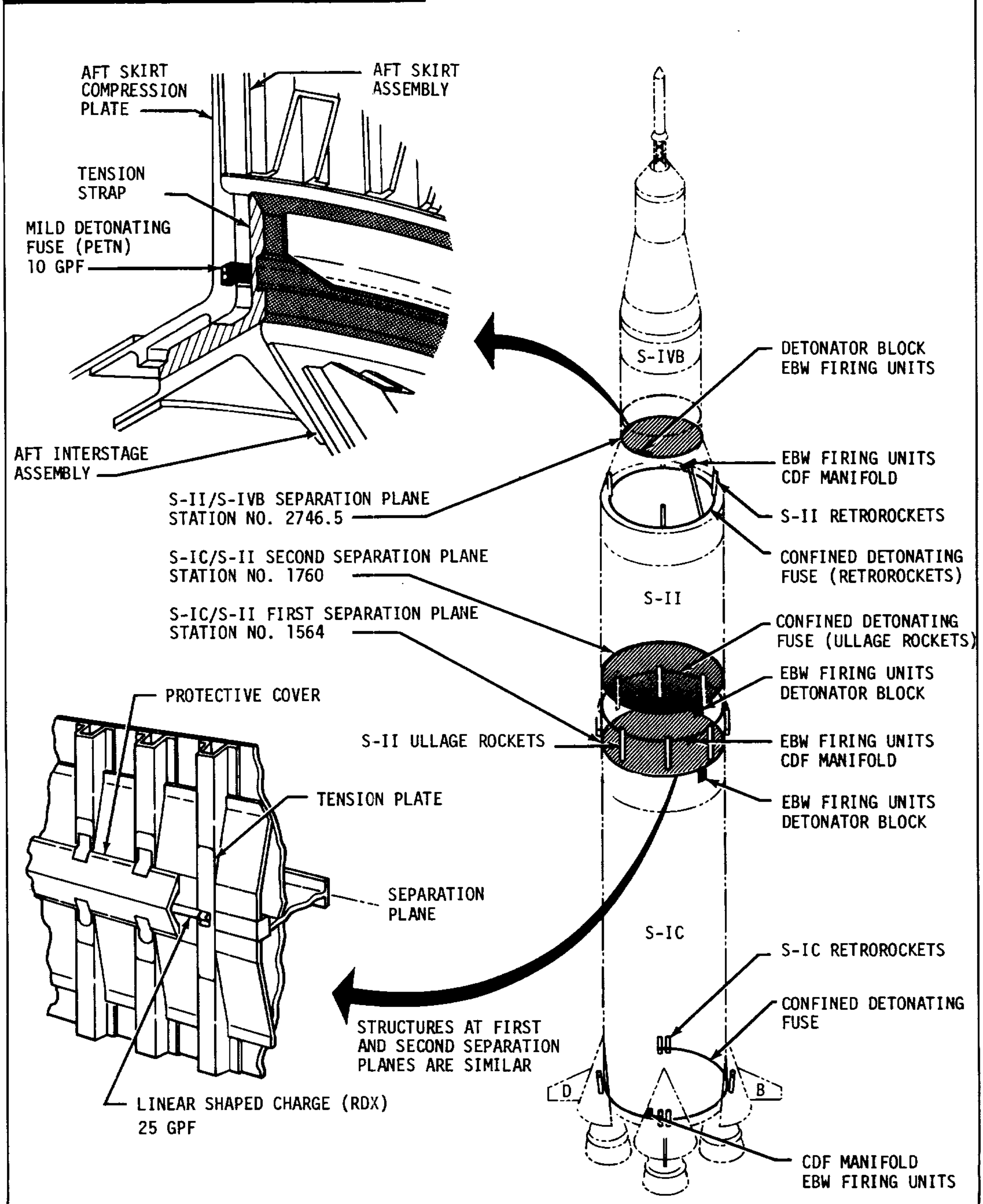


Figure 5-29

S-II SEPARATION

Flight Time (Min:Sec)

Liftoff (T_1 + 00:00) - - - - EBW firing units armed

1 A ground-latched interlock renders all the EBW firing units on the Saturn V inoperative while the vehicle is on the launch pad. The interlock is released with umbilical disconnect during liftoff, and the subsystem is reset to flight conditions.

T_2 + 00:00 - - - - - - - - - S-IC center engine cutoff

T_2 + 00:07.2 - - - - - - - - S-IC/S-II separation ordnance arm

2 The ordnance-arm command is routed through the S-II switch selector to both the S-IC stage electrical circuitry to supply 28 vdc to the EBW units for first-plane separation and retrorocket ignition, and to the S-II stage electrical circuitry to supply 28 vdc to the EBW units for ullage rocket ignition and second-plane separation.

T_3 + 00:00 - - - - - - - - - S-IC outboard engine cutoff

T_3 + 00:00.5 - - - - - - - - S-II ullage rocket ignition

3 Four solid-propellant S-II ullage rockets (figure 5-31) are mounted at equal intervals around the periphery of the S-II interstage. Each ullage rocket burns for 3.71 seconds and develops a thrust of 22,700 lbs (at 175,000 feet 70°F). With any one ullage rocket out, the remaining rockets are capable of maintaining a minimum vehicle acceleration of 0.1 g during the coast portion of S-IC/S-II separation.

T_3 + 00:00.7 - - - - - - - - First plane separation

4 The LSC assembly consists of 25 grains per foot of RDX (cyclotrimethylenetrinitramine) and forms a complete loop around the outer periphery of the vehicle at the separation plane (figure 5-29).

Second plane separation is enabled by the removal of an electrical interlock during first plane separation, when the structures are a minimum of six feet apart.

T_3 + 00:01.4 - - - - - - - J-2 engine ignition

T_3 + 00:30.7 - - - - - - - - Second plane separation

5 The second plane separation command is generated by the IU approximately thirty seconds after first plane separation. This delay permits the transient vehicle motion, associated with first plane separation, to dampen out.

The separation command is routed to the S-II switch selector to trigger the ordnance train and ignite the LSC for second plane separation. The LSC detonates, severing the S-II interstage from the S-II stage. The combined effect of vehicle acceleration and the reaction caused by the J-2 engine exhaust plume impingement, retards the interstage.

S-II/S-IVB Separation

Physical separation is initiated by the IU at the end of the S-II boost phase, following shutdown of the five J-2 engines. Separation requires the performance of the following major functions in the sequence described:

Liftoff (T_1 + 00:00) - - - - EBW firing units armed

T_3 + 05:48.0 - - - - - - - - S-II/S-IVB separation ordnance arm

6 The ordnance-arm command is routed through the S-II switch selector to both the S-II and S-IVB stage electrical circuitry and carries 28 vdc to the EBW firing units for S-II/S-IVB separation and retrorocket ignition.

T_4 + 00:00 - - - - - - - - - S-II engine cutoff

T_4 + 00:00.8 - - - - - - - - S-II/S-IVB separation

7 Four solid propellant S-II retrorockets (figure 5-31) are mounted at equal intervals on the periphery of the S-II/S-IVB interstage structure and are used to retard the S-II stage after separation. Each retrorocket has a burning time of 1.52 seconds and develops a thrust of 34,810 pounds (vacuum thrust at 60°F).

Figure 5-30 (Sheet 1 of 2)

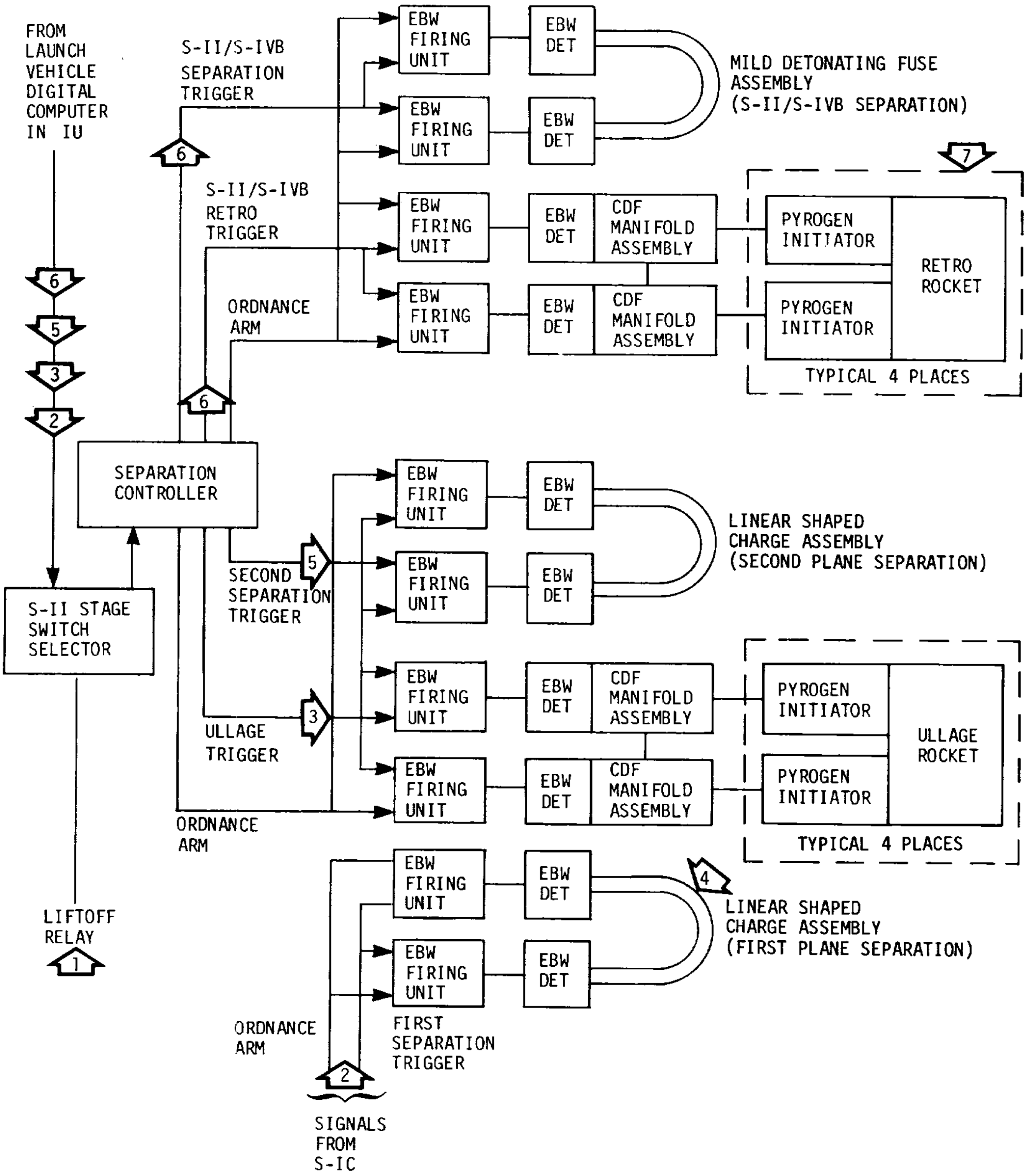


Figure 5-30 (Sheet 2 of 2)

V
S-II

Detonation of the LSC assembly severs the tension members attaching the S-IC/S-II interstage at station 1564.

The second plane separation ordnance is similar in composition and function to that of the first plane separation. The EBW firing units are installed on the S-IC/S-II interstage slightly below the separation plane. Detonation of the LSC assembly severs the tension members attaching the S-IC/S-II interstage at station 1760.

No heat-sensitive primary explosives are used and the detonators are not sensitive to accidental application of vehicle or ground power, static discharge, or RF energy. A spark gap in one pin of the firing circuitry prevents burnout of the bridgewire if power is accidentally applied.

S-II/S-IVB third plane separation is discussed in Section VI.

ULLAGE ROCKET SYSTEM

To ensure stable flow of propellants into the J-2 engines, a small forward acceleration is required to settle the propellants in their tanks. This acceleration is provided by ullage rockets (figure 5-31).

The S-II ullage rocket system consists of two EBW firing units, two EBW detonators, two CDF manifolds, nine CDF assemblies, eight CDF initiators and four ullage rockets. CDF assemblies connect the two CDF manifolds together and both manifolds to each of the four ullage rockets (see block diagram on figure 5-30). The ullage rockets are mounted parallel to vehicle centerline 90° apart on the periphery of the S-IC/S-II interstage at its aft end (figure 5-29). The rocket nozzles are just above the first separation plane and are canted outward 10° to reduce the moment that would result from one or more rockets malfunctioning and to reduce exhaust plume impingement. With any one ullage rocket inoperative, the remaining rockets are capable of maintaining a minimum vehicle acceleration necessary for proper S-II engine ignition.

Figure 5-30 (item 3) relates ullage rocket performance to S-IC/S-II separation.

Each ullage rocket contains approximately 336 pounds of solid propellant, cast-in-place, in a four point star configuration. Ammonium perchlorate composes 82 percent of the propellant weight. The case is 4130 steel. The rocket is approximately 89 inches long by 12-1/2 inches in diameter.

RETROROCKET SYSTEM

To separate and retard the S-II stage, a deceleration is provided by the retrorocket system.

The system consists of two EBW firing units, two EBW detonators, two CDF manifolds, nine CDF assemblies, eight pyrogen initiators, and four retrorockets (figure 5-31). The components are connected to each other in a manner similar to that of the ullage rocket system (see block diagram on figure 5-30). The retrockets are mounted 90° apart in the aft end of S-II/S-IVB interstage between stations 2519 and 2633 (figure 5-29). The retrorockets are canted out from the vehicle centerline approximately three degrees with the nozzles canted out nine and one-half degrees from the centerline.

Figure 5-30 (item 7) relates retrorocket performance to S-II/S-IVB separation.

Each retrorocket contains approximately 268.2 pounds of case-bonded, single-grain, solid propellant with a papered, five-point star configuration. The 4130 steel case is 9 inches in diameter and 90.68 inches long. The approximate length and weight of the rocket are 104.68 inches and 377.5 pounds, respectively.

PROPELLANT DISPERSION SYSTEM

The S-II propellant dispersion system (PDS) provides for termination of vehicle flight during the S-II boost phase if the vehicle flight path varies beyond its prescribed limits or if continuation of vehicle flight creates a safety hazard. The S-II PDS may be safed after the launch escape tower is jettisoned. The system is installed in compliance with Air Force Eastern Test Range (AFETR) Regulation 127-9 and AFETR Safety Manual 127-1.

The S-II PDS is a dual channel, redundant system composed of two segments (figure 5-32). The radio frequency segment receives, decodes, and controls the propellant dispersion commands. The ordnance train segment consists of two EBW firing units, two EBW detonators, one safety and arming (S&A) device (shared by both channels), six CDF assemblies, two CDF tees, one LH_2 tank LSC assembly, two lox tank destruct charge adapters and one lox tank destruct charge assembly.

Should emergency flight termination become necessary, two coded radio frequency commands are transmitted to the launch vehicle by the range safety officer. The first command arms the EBW firing units (figure 5-32) and initiates S-II stage engine cutoff. The second command, which is delayed to permit charging of the EBW firing units, discharges the storage capacitors in the EBW firing units across the exploding bridgewire in the EBW detonators mounted on the S&A device. The resulting explosive wave propagates through the S&A device inserts to the CDF assemblies and

S-II ULLAGE AND RETROROCKETS

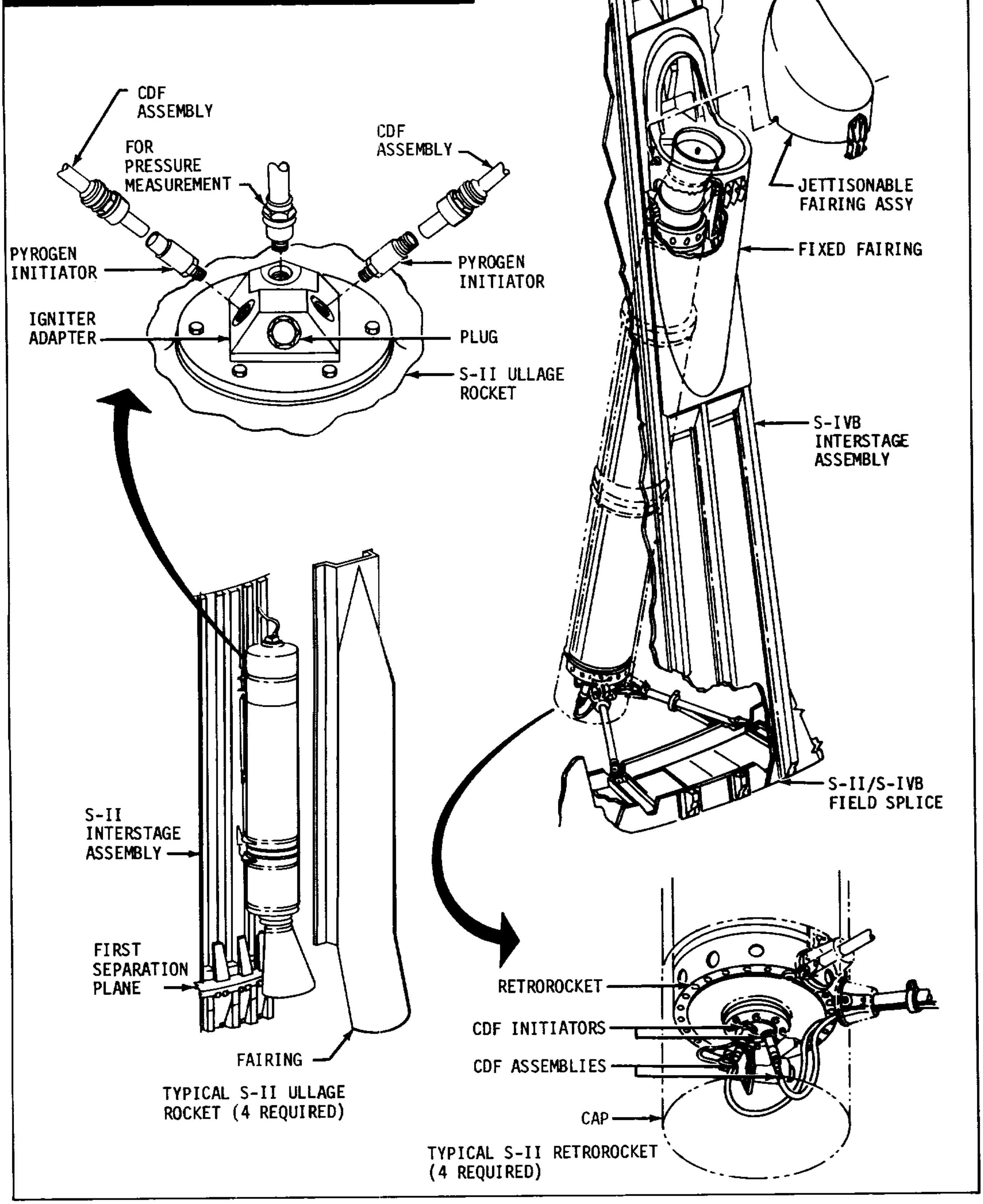


Figure 5-31

PROPELLANT DISPERSION

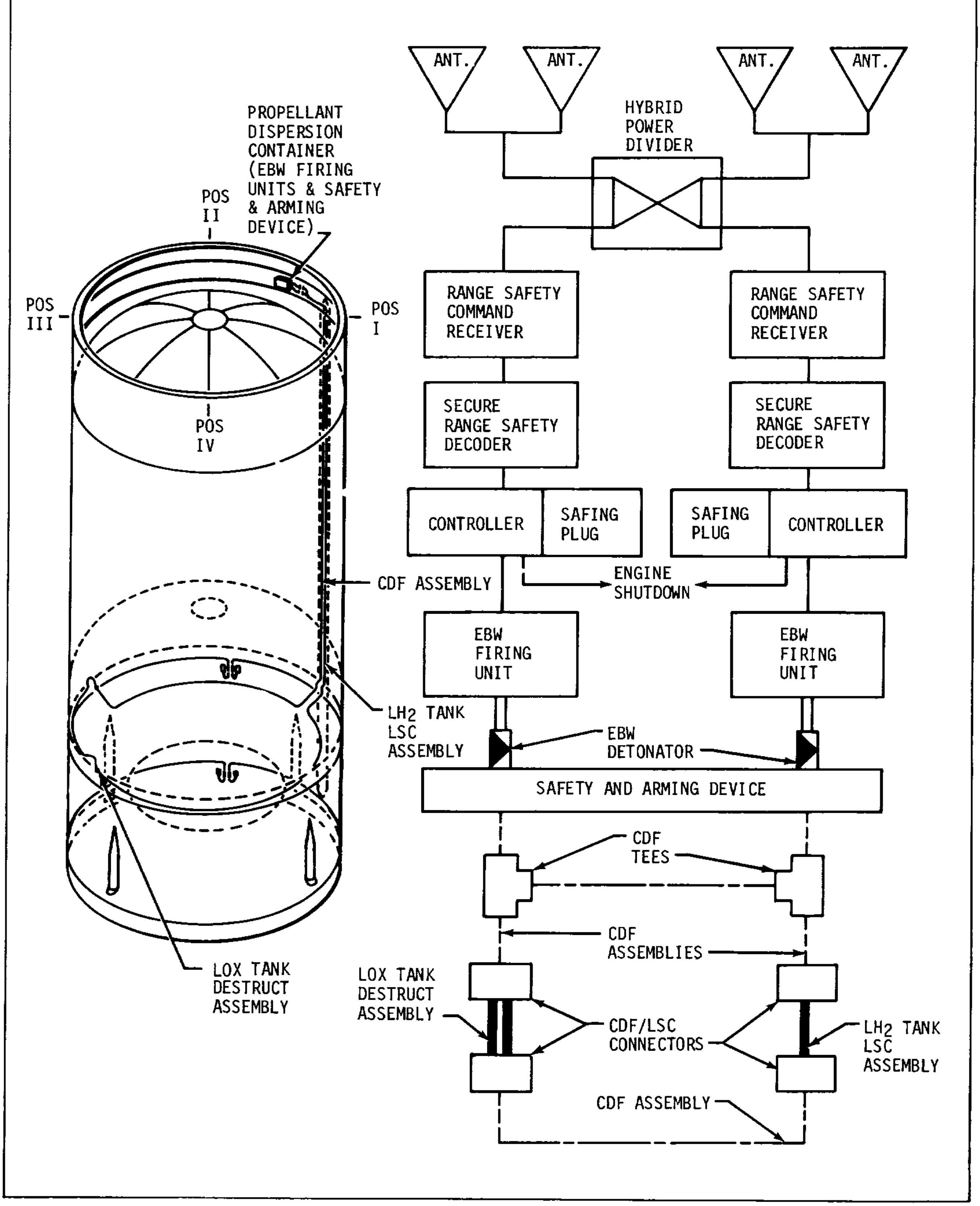


Figure 5-32

to the CDF tees. The CDF tees, installed on the S-IC forward skirt, propagate the wave to two CDF assemblies which detonate to their respective destruct assemblies. The destruct assemblies are connected by a CDF assembly to provide redundancy to the system.

A description of the S&A device is included in the PDS discussion in Section IV.

The LH_2 tank linear shaped charge when detonated cuts a 30-foot vertical opening in the tank. The LSC assembly consists of two 15-foot sections of RDX loaded at 600 grains per foot.

The lox tank destruct charges cut 13-foot lateral openings in the lox tank and the S-II aft skirt simultaneously. The destruct assembly consists of two linear explosive charges of RDX loaded at 800 grains per foot. The destruct charges are installed in a figure-eight tube mounted on the inside of the aft skirt structure near station number 1831.0.

MAJOR DIFFERENCES BETWEEN SATURN V S-II-3 & S-II-4 STAGES

1 *Nominal vacuum thrust for J-2 engines increased from 228,000 pounds each and 1,140,000 pounds total to 230,000 pounds each and 1,150,000 pounds total.*

2 *Approximate empty S-II stage weight reduced from 88,000 to 85,522 pounds. S-IC/S-II interstage weight reduced from 11,800 to 10,257 pounds.*

3 *Approximate stage gross liftoff weight increased from 1,035,000 pounds to 1,080,500 pounds.*

4 *S-II instrumentation system changed from R&D to combination of R&D and operational.*

5 *Instrumentation measurements decreased from 1,226 to 927.*

SECTION VI
S-IVB STAGE

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INTRODUCTION

The Saturn S-IVB (figure 6-1) is the third booster stage. Its single J-2 engine is designed to boost the payload into a circular orbit on the first burn then boost the payload to a proper position and velocity for lunar intercept with a second burn. The subsystems to accomplish this mission are described in the following section.

STRUCTURAL ARRANGEMENT

The S-IVB stage is a bi-propellant tank structure, designed to withstand the normal loads and stresses incurred on the ground and during launch, pre-ignition boost, ignition, and all flight phases.

The basic S-IVB stage airframe, illustrated in figure 6-2, consists of the following structural assemblies: the forward skirt, propellant tanks, aft skirt, thrust structure, and aft interstage. These assemblies, with the exception of the propellant tanks, are all of a skin/stringer type aluminum alloy airframe construction. In addition, there are two longitudinal tunnels which house wiring, pressurization lines, and propellant dispersion systems. The tunnel covers are made of aluminum stiffened by internal ribs. These structures do not transmit primary shell loads but act only as fairings.

FORWARD SKIRT ASSEMBLY

Cylindrical in shape, the forward skirt (figure 6-2) extends forward from the intersection of the liquid hydrogen (LH_2) tank sidewall and the forward dome, providing a hard attach point for the instrument unit (IU). It is the load supporting member between the LH_2 tank and the IU. An access door in the IU allows servicing of the equipment in the forward skirt. The five

S-IVB STAGE

CONTRACTOR:	McDONNELL-DOUGLAS	
ENGINE:	J-2	
PROPELLANT:	LOX/LH_2	
THRUST:	1st BURN	232,000 LBS
	2nd BURN	211,000 LBS
WEIGHT:	DRY	31,571 LBS
	AT IGNITION	265,559 LBS

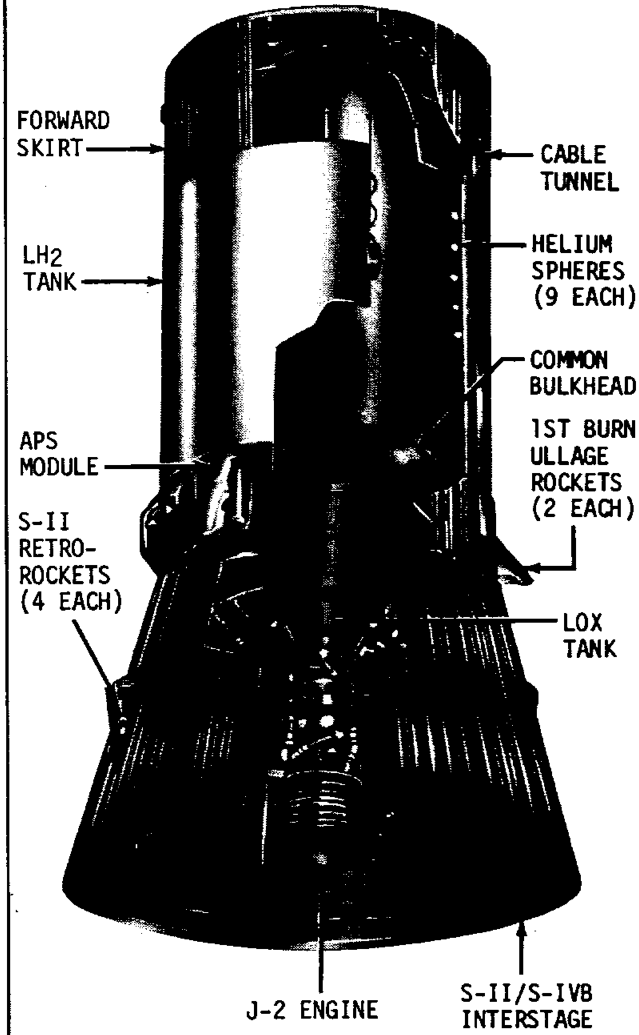

Figure 6-1

VI

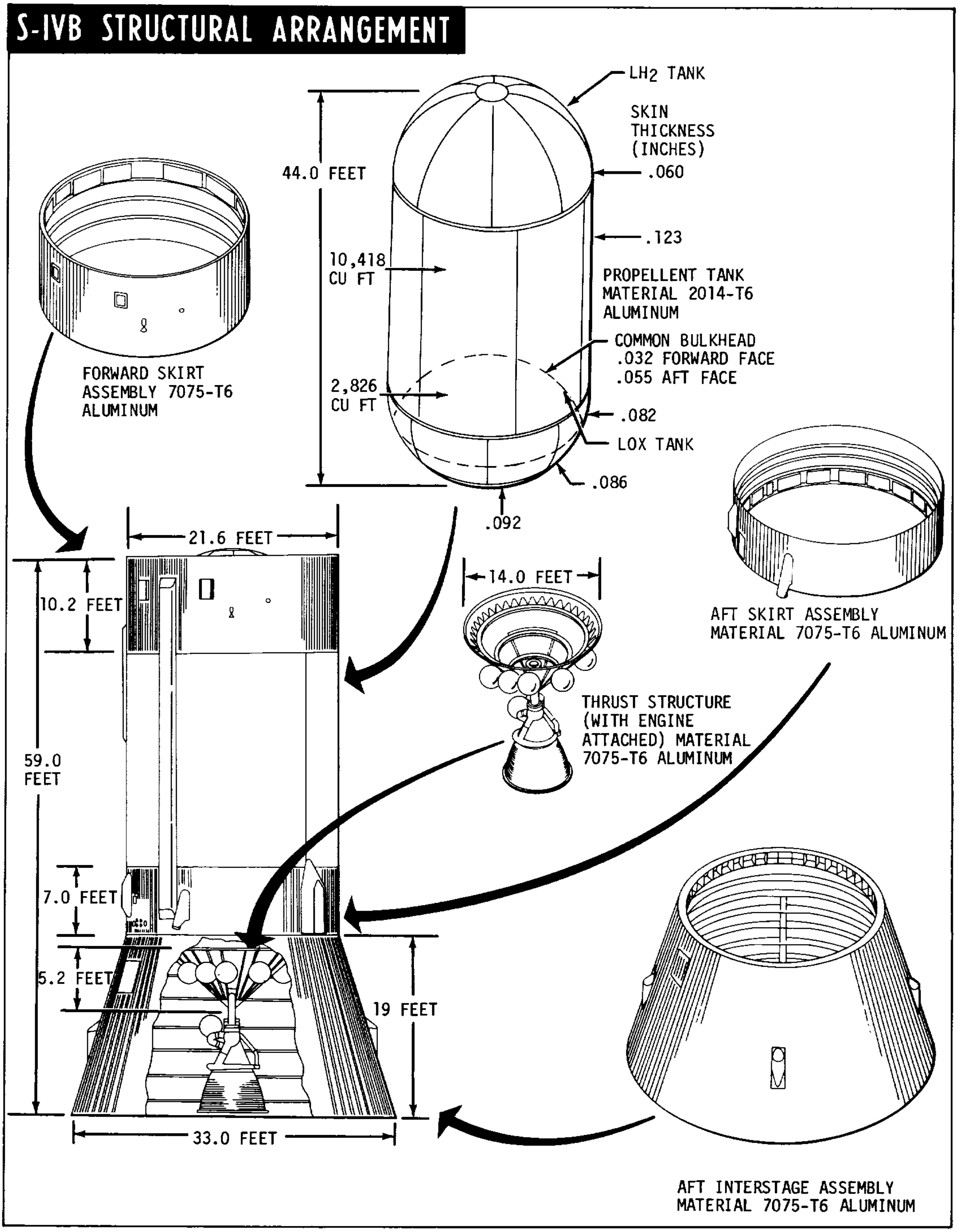
S-IVB STRUCTURAL ARRANGEMENT
LH2 TANK
SKIN
THICKNESS
(INCHES)
.060
.123
44.0 FEET
10,418
CU FT
PROPELLENT TANK
MATERIAL 2014-T6
ALUMINUM
COMMON BULKHEAD
.032 FORWARD FACE
.055 AFT FACE
2,826
CU FT
.082
LOX TANK
.086
.092
FORWARD SKIRT
ASSEMBLY 7075-T6
ALUMINUM
21.6 FEET
14.0 FEET
AFT SKIRT ASSEMBLY
MATERIAL 7075-T6 ALUMINUM
10.2 FEET
THRUST STRUCTURE
(WITH ENGINE
ATTACHED) MATERIAL
7075-T6 ALUMINUM
59.0
FEET
7.0 FEET
5.2 FEET
19 FEET
33.0 FEET
AFT INTERSTAGE ASSEMBLY
MATERIAL 7075-T6 ALUMINUM

Figure 6-2

environmental plates which support and thermally condition various electronic components, such as the transmitters and signal conditioning modules, are attached to the inside of this skirt. The forward umbilical plate, antennae, LH_2 tank flight vents and the tunnel fairings are attached externally to this skirt.

PROPELLANT TANK ASSEMBLY

The propellant tank assembly (figure 6-2) consists of a cylindrical tank with a hemispherical shaped dome at each end, and a common bulkhead to separate the lox from the LH_2. This bulkhead is of sandwich type construction consisting of two parallel hemispherical shaped aluminum alloy (2014-T6) domes bonded to and separated by a fiberglass-phenolic honeycomb core. This effectively insulates the LH_2 (-423°F) from the lox (-297°F). The internal surface of the LH_2 tank is machine milled in a waffle pattern to obtain required tank stiffness with minimum structural weight. To minimize LH_2 boil off polyurethane insulation blocks, covered with a fiberglass sheet and coated with a sealant, are bonded into the milled areas of the waffle patterns.

The walls of the tank support all loads forward of the forward skirt attach point and transmit the thrust to the payload. Attached to the inside of the LH_2 tank are a 34 foot propellant utilization (PU) probe, nine cold helium spheres, brackets with temperature and level sensors, a chilldown pump, a slosh baffle, a slosh deflector, and fill, pressurization and vent pipes. Attached to the inside of the lox tank are slosh baffles, a chilldown pump, a 13.5 foot PU probe, temperature and level sensors, and fill, pressurization and vent pipes. Attached externally to the propellant tank are helium pipes, propellant dispersion components, and wiring which passes through two tunnel fairings. The forward edge of the thrust structure is attached to the lox tank portion of the propellant tank.

THRUST STRUCTURE

The thrust structure assembly (figure 6-2) is an inverted, truncated cone attached at its large end to the aft dome of the lox tank and attached at its small end to the engine mount. It provides the attach point for the J-2 engine and distributes the engine thrust over the entire tank circumference. Attached external to the thrust structure are the engine piping, wiring and interface panels, eight ambient helium spheres, hydraulic system, oxygen/hydrogen burner, and some of the engine and lox tank instrumentation.

AFT SKIRT ASSEMBLY

The cylindrical shaped aft skirt assembly is the load bearing structure between the LH_2 tank and aft interstage. The aft skirt assembly is bolted to the tank assembly at its forward edge and connected to the aft interstage. A frangible tension tie separates it from the aft interstage at S-II separation.

AFT INTERSTAGE ASSEMBLY

The aft interstage is a truncated cone that provides the load supporting structure between the S-IVB stage and the S-II stage (figure 6-2). The interstage also provides the focal point for the required electrical and mechanical interface between the S-II and S-IVB stages. The S-II retrorockets motors are attached to this interstage and at separation the interstage remains attached to the S-II stage.

ENVIRONMENTAL CONTROL

There are three general requirements for environmental control during checkout and flight operations of the S-IVB stage. The first is associated with ground checkout and prelaunch operations and involves thermal conditioning of the environments around the electrical equipment, auxiliary propulsion system (APS), and hydraulic accumulator reservoir. In addition, there is a requirement for aft skirt and interstage purging. The second involves forward skirt area purging, while the third concerns inflight heat dissipation for the electrical-electronic equipment.

AFT SKIRT AND INTERSTAGE THERMOCONDITIONING

During countdown, air/GN_2 is supplied by the environmental control system which is capable of switching from air to GN_2 purge. Air or GN_2 is supplied at the rate of approximately 3500 scfm. The air purge is initiated when electrical power is applied to the vehicle. GN_2 flow is initiated 20 minutes prior to lox chilldown (at T-5 hrs 20 min) and continued until liftoff. During periods of hold, GN_2 purge is continued. The aft skirt and interstage thermoconditioning and purge subsystem provides the following:

1. Thermal conditioning of the atmosphere around electrical equipment in the aft skirt during ground operations.
2. Thermal conditioning of the APS, hydraulic accumulator reservoir, and ambient helium bottle.
3. Purging of the aft skirt, aft interstage and thrust structure, and the forward skirt of the S-II stage of oxygen, moisture and combustible gases.

The subsystem consists of a temperature-controlled air or GN_2 distribution system (figure 6-3). The main manifold of the distribution system is formed between the ring frames at stations 2786.603 and 2802.353, the lox dome and the aft skirt.

The ring frame at station 2786.603 also has uniformly spaced holes around the stage circumference to distribute purging gas evenly around the stage. The purging gas passes over electrical equipment below the ring frame and flows into the interstage. A duct from the skirt manifold directs air or GN_2 to a thrust structure manifold. Another duct directs the gas to a shroud covering the ambient helium bottle used to purge lox and LH_2 pump shaft seal cavities. From the thrust structure manifold supply duct a portion of air or GN_2 is directed to a shroud covering the hydraulic accumulator reservoir.

Temperature control is accomplished by two dual-element thermistor assemblies located in the gaseous exhaust stream of each of the APS modules. One element from each thermistor assembly is wired in series to sense average temperature. One series is used for temperature control, the other for temperature recording.

FORWARD SKIRT THERMOCONDITIONING

The electrical/electronic equipment in the S-IVB forward skirt area is thermally conditioned by a heat transfer subsystem using a circulating coolant for the medium. Principal components of the sys[illegible]n, located in the S-IVB stage forward skirt a[illegible]a, are a fluid distribution subsystem and col[illegible] plates. The coolant is supplied to the S-IVB b[illegible] the IU thermoconditioning system starting wh[illegible] electrical power is applied to the vehicl[illegible]nd continuing throughout the mission. The ele[illegible]rical/electronic equipment is attached to the [illegible]d plates and dissipates heat by conductio[illegible]rough its mounting feet to the plates and flu[illegible] The coolant consists of a 60 percent by we[illegible] methyl alcohol and a 40 percent by weight [illegible]illed water solution that contains a sodium benzoate corrosion inhibitor. The coolant is supplied through quick disconnect fittings at the IU/S-IVB interface at a flow rate of 3500 $\pm$ 175 pounds/hour and maintained within temperature limits of +40° to 60°F.

FORWARD SKIRT AREA PURGE

The forward skirt area is purged with GN_2 to minimize the danger of fire and explosion while propellants are being loaded or stored in the stage, or during other hazardous conditions. The purge is supplied by the IU purge system which purges the entire forward skirt/IU/adapter area. The total flow rate into this area is approximately 3500 scfm.

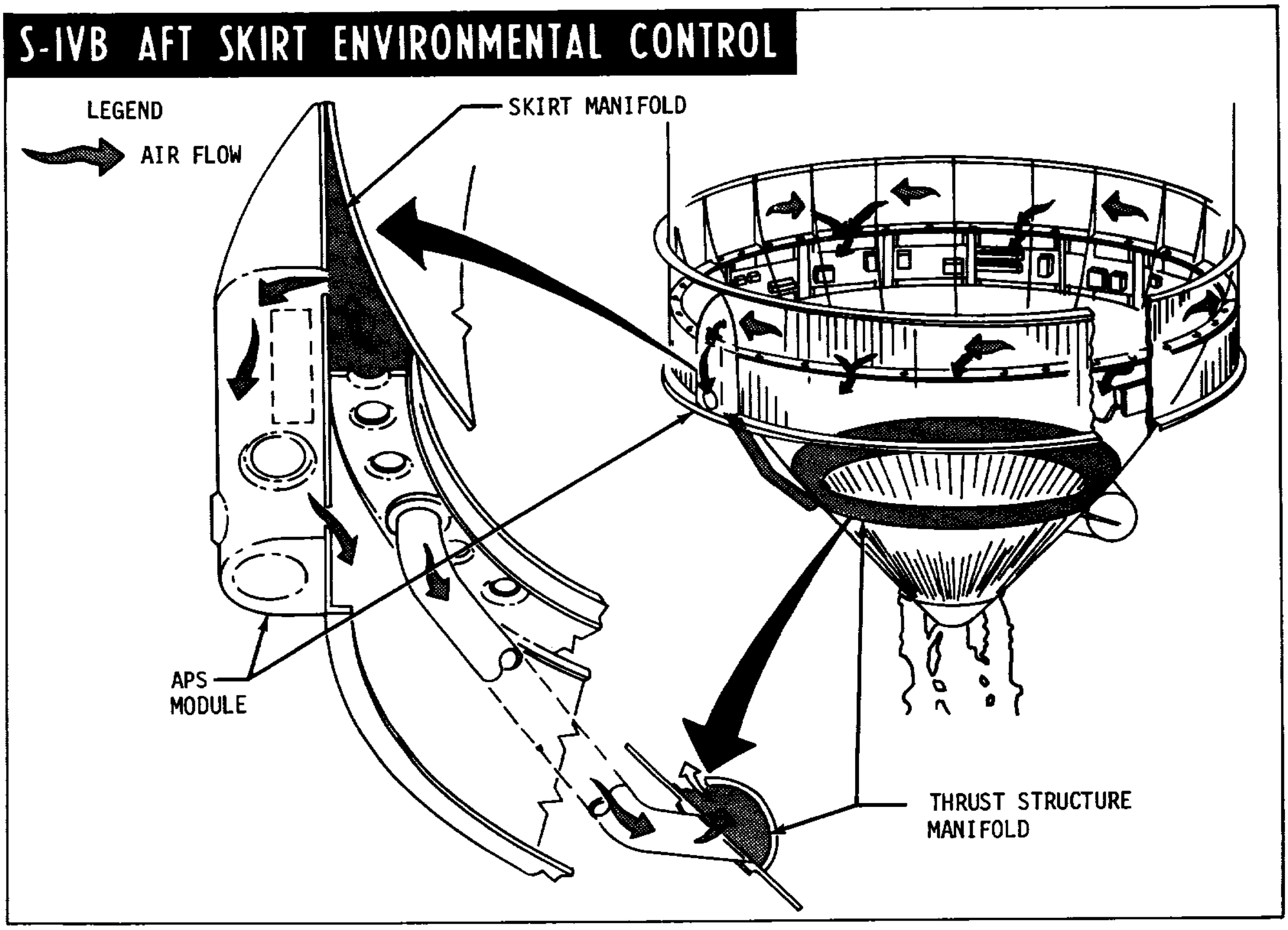

Figure 6-3

PROPULSION

This stage provides vehicle propulsion twice during this mission. The first burn occurs immediately after S-II/S-IVB separation. It lasts long enough to insert the vehicle into earth orbit. Second burn injects the spacecraft into a high apogee elliptical orbit.

At the end of J-2 engine first burn, the APS ullage engines are fired to stabilize and settle the propellants. At this time, the APS yaw and pitch control modes are enabled (roll already active) for the required attitude control of the stage and payload during coast. At J-2 engine first burn cutoff, the APS ullage engines are ignited and burn until T_5 + 1 minute 28 seconds. LH_2 continuous venting is activated at T_5 + 59.0 seconds and continues until just prior to the second engine burn period.

Prior to second burn, the systems are again readied for an engine start. Approximately 7 minutes before restart, the chilldown systems are reactivated to remove gases collected in the supply systems and to condition the lines and engine pumps. The oxygen/hydrogen (O_2/H_2) burner is started about 6 minutes prior to second burn. It pressurizes the propellant tanks ullage space while its thrust maintains the propellants in a settled state.

After the O_2/H_2 burner has started, the LH_2 continuous venting is terminated. Then, approximately one minute before engine start, the APS ullage engines are fired and the O_2/H_2 burner is shut down. The recirculation system is deactivated followed by initiation of engine restart. The APS ullage engines are then shut off and the APS yaw and pitch control modes are de-energized. The roll control mode remains active throughout second burn.

At the end of J-2 engine second burn, the pitch and yaw modes of the APS are again enabled for the required attitude control of the stage and payload during terminal coast. Following J-2 engine second burn cutoff, the lox and LH_2 tank pressure are vented through the stage nonpropulsive vent subsystem.

J-2 ROCKET ENGINE

The J-2 rocket engine (figure 5-11) is a high performance multiple restart engine utilizing liquid oxygen and liquid hydrogen as propellants. The only substances used in the engine are the propellants and helium gas. The extremely low operating temperature of the engine prohibits the use of lubricants or other fluids. The engine features a single tubular walled bell shaped thrust chamber and two independently driven direct drive turbopumps for liquid oxygen and liquid hydrogen. Both turbopumps are powered in series by a single gas generator, which utilizes the same propellants as the thrust chamber. The main hydraulic pump is driven by the oxidizer turbopump turbine. The ratio of fuel to oxidizer is controlled by bypassing liquid oxygen from the discharge side of the oxidizer turbopump to the inlet side through a servovalve.

The engine valves are controlled by a pneumatic system powered by gaseous helium which is stored in a sphere inside the start bottle. An electrical control system, which uses solid state logic elements, is used to sequence the start and shutdown operations of the engine. Electrical power is supplied from aft battery No. 1.

During the burn periods, the lox tank is pressurized by flowing cold helium through the heat exchanger in the oxidizer turbine exhaust duct. The heat exchanger heats the cold helium causing it to expand. The LH_2 tank is pressurized during burn periods by GH_2 from the thrust chamber fuel manifold.

Thrust vector control, in the pitch and yaw planes, during burn periods, is achieved by gimbaling the entire engine. Hydraulic pressure for gimbal actuation is provided by the hydraulic system which receives pressure from a hydraulic pump mounted on the lox pump. Thrust vector control in all planes during coast, and the roll plane during burn periods, is achieved by firing the auxiliary propulsion system engines.

Start Preparations

Preparations for an engine start include ascertaining the positions and status of various engine and stage systems and components. The J-2 engine electrical control system controls engine operation by means of electrical signals. The heart of the engine electrical control system is the electrical control package (17, figure 5-9). It sequences and times the functions required during engine start or cutoff.

Each cutoff automatically causes the electrical control package circuitry to reset itself ready for restart providing that all reset conditions are met. The LVDC issues an engine ready bypass signal just prior to each engine start attempt. This bypass signal acts in the same manner as a cutoff would act. The reset signals engine ready and this allows the LVDC to send its start command. Receipt of the start command initiates the engine start sequence.

ENGINE START SEQUENCE

When engine start is initiated (3, figure 6-4) the spark exciters in the electrical control package provide energy for the gas generator (GG) and augmented spark igniter (ASI) spark plugs (4). The helium control and ignition phase control valves, in the pneumatic control package (1), are simultaneously energized allowing helium from the helium tank (2) to flow through the pneumatic regulator to the pneumatic

S-IVB ENGINE START

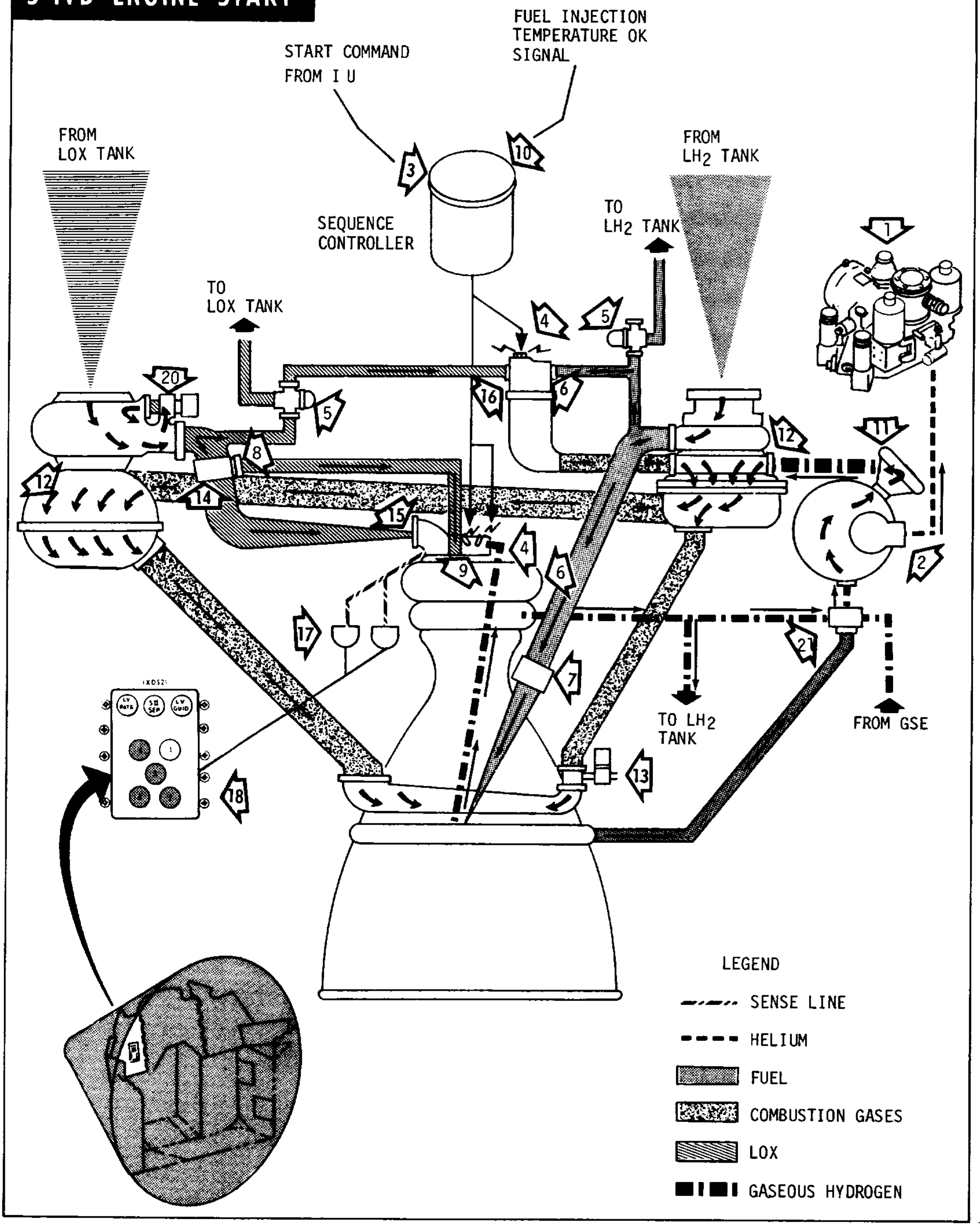

Figure 6-4 (Sheet 1 of 3)

S-IVB ENGINE START

SEQUENCE	EVENT	TIME IN SECONDS
	S-IVB 1ST BURN	T4 + 1.0 3.0 5.0 7.0 9.0 11.0 13.0 15.0
	S-IVB 2ND BURN	T6 + 450.0 452.0 454.0 456.0 458.0 460.0 462.0 464.0
	PRE-IGNITION NOTE ALL PNEUMATIC VALVES ARE CONTROLLED BY THE PNEUMATIC CONTROL PACKAGE 1 WHICH IS LOCATED ON THE ENGINE. HELIUM FOR THE PNEUMATIC SYSTEM IS SUPPLIED BY THE CONTROL SPHERE INSIDE THE GH_2 START TANK 2.	
3	Start Command (Occurs at T_4 + 1.0 and T_6 + 450.0) → Engine Ready → Engine Start Sequence	
4	Augumented Spark Igniter (ASI) Spark Plugs and gas generator (GG) spark plugs fire.	
5	Bleed valves stop return flow to propellant tanks.	
6	Oxidizer dome and gas generator oxidizer injector are purged.	
7	Main fuel valve allows LH_2 to flow into engine thrust chamber and into ASI.	
8	ASI oxidizer valve allows lox flow to ASI.	
9	Sparks ignite the propellants in the ASI.	
10	Fuel injection temperature OK signal received causing the	

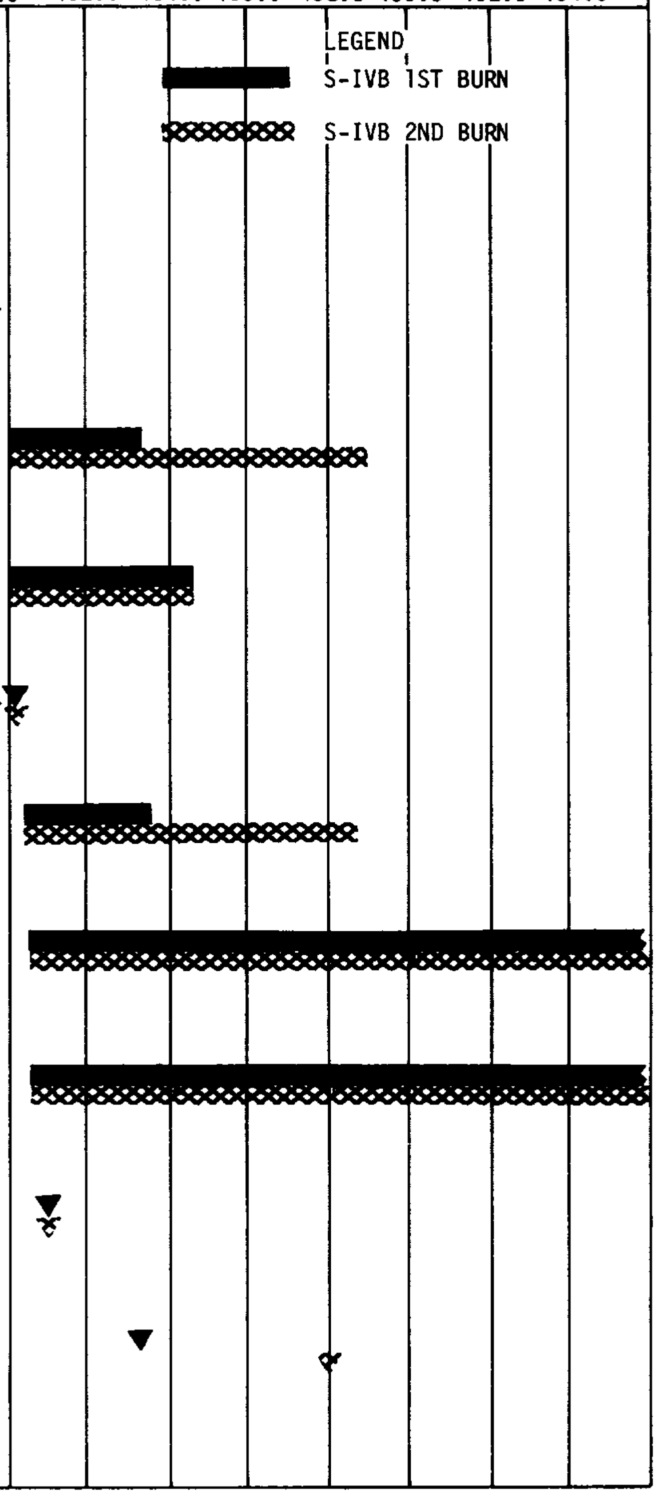

Figure 6-4 (Sheet 2 of 3)

S-IVB ENGINE START

SEQUENCE	EVENT
	TIME IN SECONDS — S-IVB 1ST BURN T_4 + 1.0, 3.0, 5.0, 7.0, 9.0, 11.0, 13.0, 15.0
	TIME IN SECONDS — S-IVB 2ND BURN T_6 + 450.0, 452.0, 454.0, 456.0, 458.0, 460.0, 462.0, 464.0
11	start tank discharge valve to open.
12	GH_2 spins LH_2 and lox turbopumps causing propellant pressure to buildup.
13	Lox turbopump bypass valve open to control lox pump speed.
14	Main oxidizer valve opens allowing,
15	lox to be injected into thrust chamber.
16	G G valves admit propellants. (Spark ignites propellants causing pressure build up.)
	MAIN STAGE
17	OK pressure switches send mainstage OK signal to CM.
18	No. 1 engine out light goes out. (Remaining 4 lights not operative).
19	Engine reaches and maintains 90% thrust or more.
20	P.U. valve controls mass ratio by returning lox from pump discharge to pump inlet.
21	GH_2 Start tank is refilled with GH_2 and LH_2.

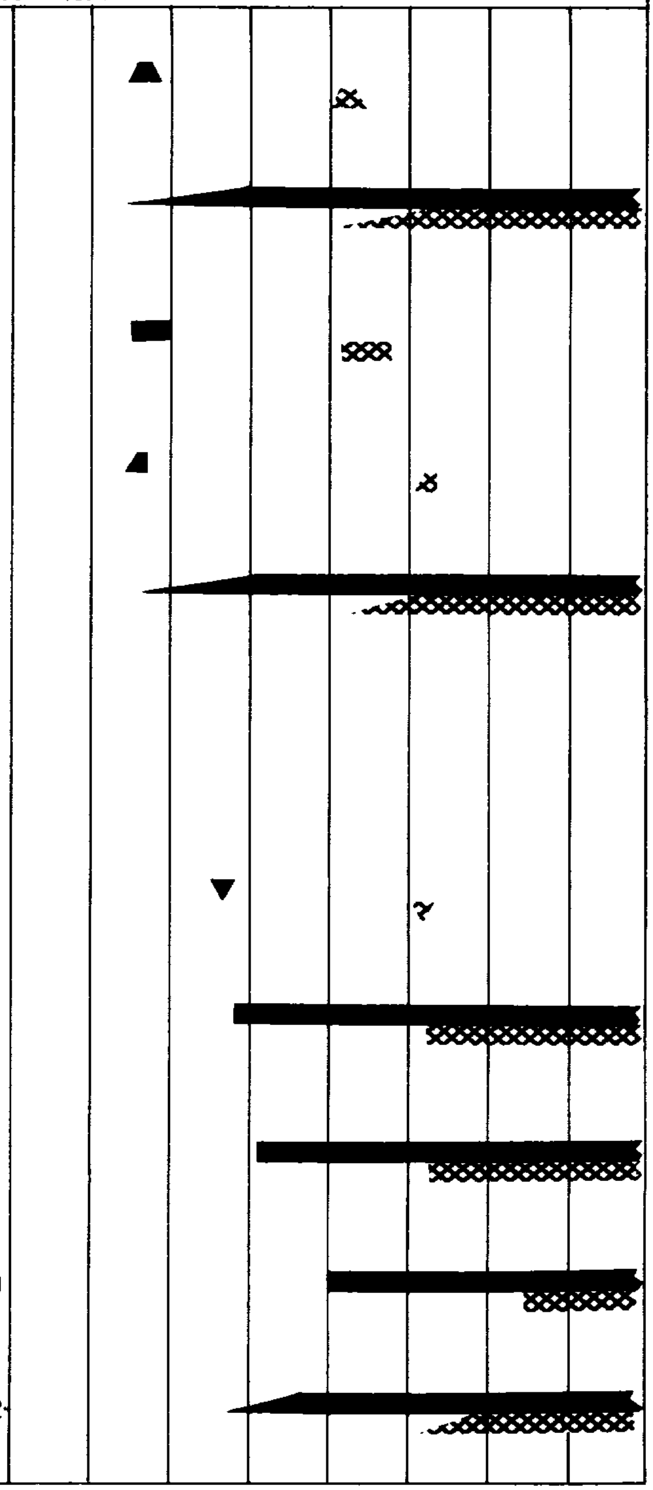

Figure 6-4 (Sheet 3 of 3)

control system. The helium is routed through the internal check valve in the pneumatic control package (1) to ensure continued pressure to the engine valves in the event of helium supply failure. The regulated helium fills a pneumatic accumulator, closes the propellant bleed valves, (5) and purges (6) the oxidizer dome and gas generator oxidizer injector manifold. The oxidizer turbopump (12) intermediate seal cavity is continuously purged. The mainstage control valve closes the main oxidizer valve and opens the purge control valve which allows the oxidizer dome and gas generator oxidizer injector to be purged (6). The mainstage control valve also supplies opening control pressure to the oxidizer turbine bypass valve (13). The ignition phase control valve, when actuated, opens the main fuel valve (7) and the ASI oxidizer valve (8) and supplies pressure to the sequence valve located within the main oxidizer valve (14). Fuel is tapped from downstream of the main fuel valve for use in the ASI (4). Both propellants, under tank pressure, flow through the stationary turbopumps (12).

The sequence valve, in the main fuel valve (12), opens when the fuel valve reaches approximately 90 percent open and routes helium to the start tank discharge valve (STDV) (11) control valve. Simultaneously with engine start, the STDV delay timer is energized. Upon expiration of the STDV timer and the receipt of a fuel injection temperature OK signal the STDV control valve and ignition phase timer are energized. As the STDV control valve energizes the discharge valve opens allowing gaseous hydrogen under pressure to flow through the series turbine drive system. This accelerates both turbopumps (12) to the proper operating levels to allow subsequent ignition and power build up of the gas generator (16). The relationship of fuel to lox turbopump speed buildup is controlled by an orifice in the oxidizer turbine bypass valve (13). During the start sequence the normally open oxidizer bypass valve (13) permits a percentage of the gas to bypass the oxidizer turbine.

During this period, ASI combustion is detected by the ASI ignition monitor. (Absence of the ignition detection signal or a start tank depressurized signal will cause cutoff at the expiration of the ignition phase timer.) With both signals present at ignition phase timer expiration, the mainstage control valve energizes. Simultaneously, the sparks deenergize timer is energized and the STDV control valve is deenergized, causing the STDV to close. Helium pressure is vented from the main oxidizer valve (14) and from the purge control valve through the mainstage control valve. The purge control valve closes, terminating the oxidizer dome and gas generator oxidizer injector manifold purges (6). Pressure from the mainstage control valve is routed to open the main oxidizer valve (14). A sequence valve operated by the main oxidizer valve (14) permits gaseous helium to open the gas generator control valve (4) and to close the oxidizer turbine bypass valve (13). (Flow to close the oxidizer turbine bypass valve (13) is restricted as it passes through an orifice. The orifice controls the closing speed of this valve.)

Propellants flowing into the gas generator (16) are ignited by the sparkplugs (4). Combustion of the propellants cause the hot gases to drive the turbopumps (12). The turbopumps rotation causes propellant pressure to buildup. The propellant flow increases and is ignited in the thrust chamber by the torch from the ASI.

Transition into mainstage occurs as the turbopumps (12) accelerate to steadystate speeds. As oxidizer injection pressure increases a thrust OK signal is generated by either of the two thrust OK pressure switches (17). (Cutoff occurs if no signal is received before expiration of the sparks deenergized timer). The ASI and GG sparks exciters are deenergized at expiration of the sparks deenergized timer. Cutoff occurs if both pressure switch actuated signals (thrust OK) are lost during mainstage operation.

Steadystate operation is maintained until a cutoff signal is initiated. During this period, gaseous hydrogen is tapped from the fuel injection manifold to pressurize the LH_2 tank. The lox tank is pressurized by gaseous helium heated by the heat exchanger in the turbine exhaust duct. Gaseous hydrogen is bled from the thrust chamber fuel injection manifold and liquid hydrogen is bled from the ASI fuel line to refill start tank for engine restart. (Approximately 50 seconds of mainstage engine operation is required to recharge the start tank.)

Propellant utilization is provided by bypassing oxidizer from the oxidizer turbopump outlet back to the inlet. The propellant utilization valve is controlled by signals from the propellant utilization system. The engine mixture ratio varies from 4.5:1 to 5.5:1.

Engine Cutoff

The J-2 engine may receive cutoff signals from the following sources: EDS No.'s 1 and 2, range safety systems No.'s 1 and 2, thrust OK pressure switches, propellant depletion sensors and an IU programmed command (velocity or timed) via the switch selector.

The switch selector, range safety system No. 2, EDS No. 2, and the propellant depletion sensors cutoff commands are tied together (but diode isolated) and sent to the electrical control package cutoff circuit. The dropout of the thrust OK pressure switches removes a cutoff inhibit function in the electrical control package cutoff circuit. EDS No. 1 and range safety system No. 1 cutoff commands will indirectly transfer the engine control power switch to the OFF position causing the engine to shutdown due to power loss.

S-IVB ENGINE CUTOFF

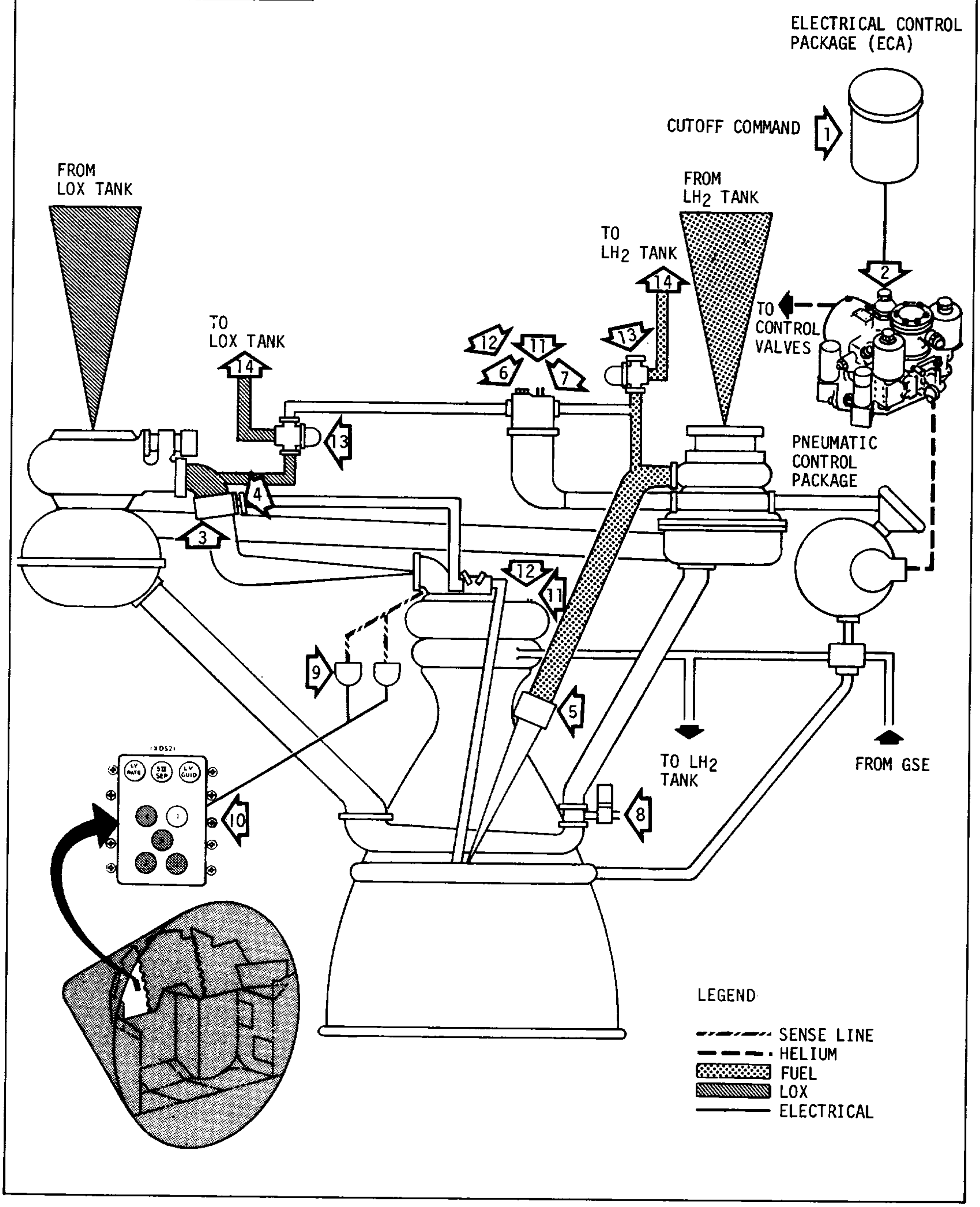

Figure 6-5 (Sheet 1 of 2)

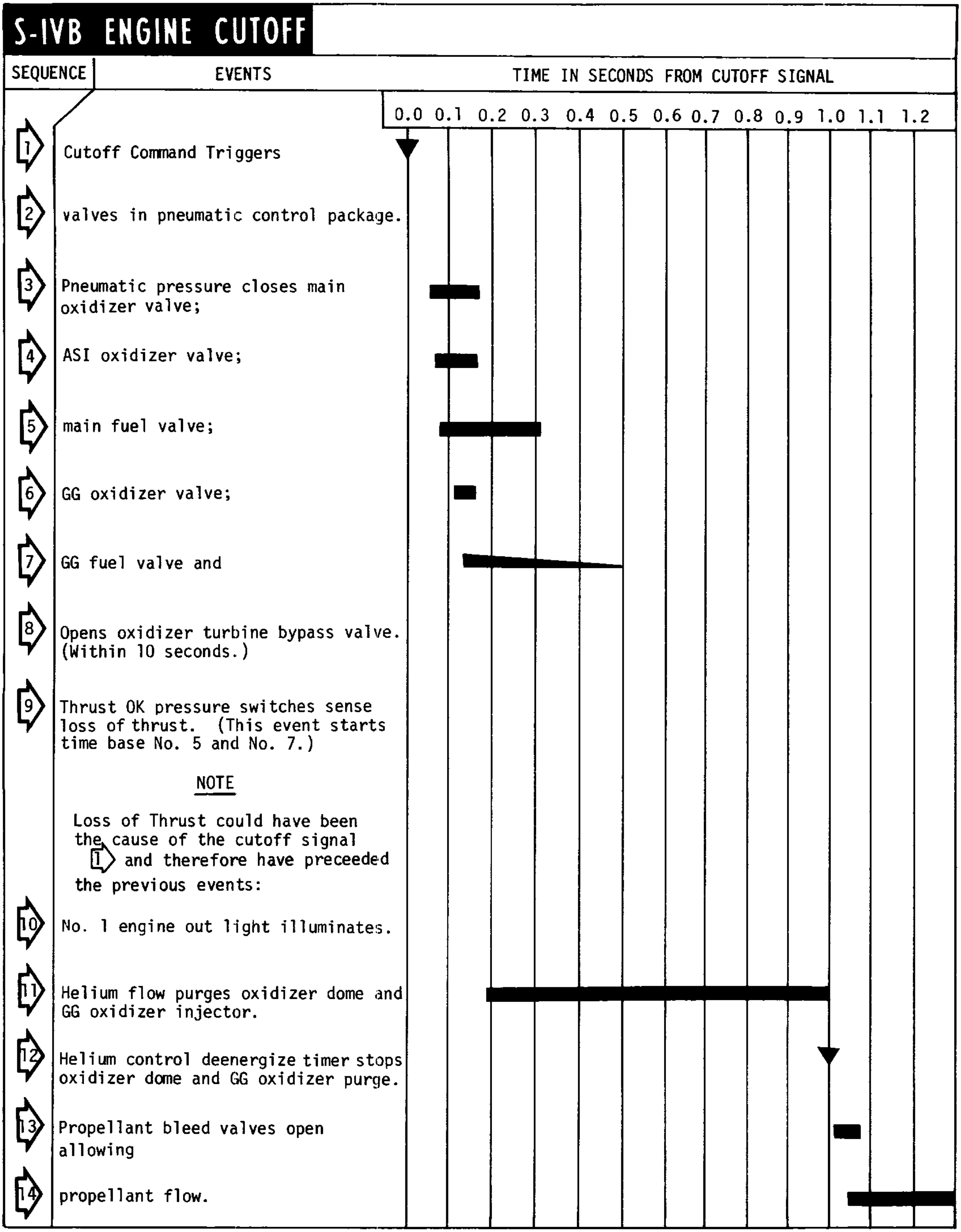

Figure 6-5 (Sheet 2 of 2)

Cutoff Sequence

The engine cutoff sequence is shown graphically in figure 6-5. When the electrical control package receives the cutoff signal (1), it deenergizes the mainstage and ignition phase control valves in the pneumatic control package (2) while energizing the helium control deenergize timer. The mainstage control valve closes the main oxidizer valve (3) and opens the purge control valve and the oxidizer turbine bypass valve (5). The purge control valve directs a helium purge (11) to the oxidizer dome and GG oxidizer injector. The ignition phase control valve closes the ASI oxidizer valve (4) and the main fuel valve (5) while opening the fast shutdown valve. The fast shutdown valve now rapidly vents the return flow from the GG control valve. All valves except the ASI oxidizer valve (4) and oxidizer turbine bypass valve (8), are spring loaded closed. This causes the valves to start moving closed as soon as the pressure to open them is released. GG combustion pressure aids closing of the GG control valve.

Expiration of the helium control deenergize timer causes the helium control valve to close. When the helium control valve closes it causes the oxidizer dome and GG oxidizer injector purges (11) to stop. An orifice in the locked up lines bleeds off pressure from the propellant bleed valves (13). This loss of pressure allows springs to open the valves. When open, the propellant bleed valves allow propellants to flow back to the propellant tanks.

Restart

The restart of the J-2 engine is identical to the initial start except for the fill procedure of the start tank. The start tank is filled with LH_2 and GH_2 during the first burn period.

To insure that sufficient energy will be available for spinning the LH_2 and lox pump turbines, a waiting period of between approximately 90 minutes to 6 hours is permitted. The minimum time is required to build sufficient pressure by warming the start tank through natural means. The minimum wait is also needed to allow the hot gas turbine exhaust system to cool. Prolonged heating will cause a loss of energy in the start tank. This loss occurs when the LH_2 and GH_2 warms and raises the gas pressure to the relief valve setting. If this venting continues over a prolonged period the total stored energy will be depleted. This limits the waiting period prior to a restart attempt to 6 hours.

PROPELLANTS

The propellant tank assembly accepts, stores and supplies the lox and LH_2 used for J-2 engine operation.

Separate pressurization and repressurization systems (figure 6-6) are provided for each propellant tank to monitor and control tank pressures in order to assure the engine propellant turbopump a net positive suction head (NPSH) of 42 feet for lox and 150 feet for LH_2 (minimum) during engine 1st and 2nd burn. Each tank is equipped with a pressure sensing switch which commands the opening or closing of the respective tank pressurization or repressurization control module valves. The control modules, in turn, control the passage of pressurizing gases to the tanks as required. Prior to flight the ambient and cold helium storage bottles are

Pressurization Systems		
Time	Tank	Pressurization Source
Prior to liftoff	LOX	Helium ground source
	LH_2	Helium ground source
1st & 2nd stage boost	LOX	Cold helium*
	LH_2	None added
During S-IVB 1st burn	LOX	Cold helium heated in Heat Exchanger
	LH_2	GH_2 bled from J-2 engine
Prior to 2nd burn	LOX	Ambient helium** or cold helium heated by oxygen/hydrogen burner
	LH_2	
During 2nd burn	LOX	Cold helium heated in Heat Exchanger
	LH_2	GH_2 bled from J-2 engine

*Cold helium is stored in 9 spheres in the LH_2 tank.
**Ambient helium is stored in 8 spheres on the Thrust Structure.

Figure 6-6

filled and the propellant tanks are prepressurized utilizing a common ground source of cold gaseous helium. Control is maintained by the GSE in conjunction with the pressure switch in each tank.

Oxygen/Hydrogen Burner

The oxygen/hydrogen (O_2/H_2) burner (figure 6-7) uses stage onboard propellants as an energy source to heat cold helium. Propellants are fed through vacuum jacketed, low pressure ducts to the heater regenerative coils where they are vaporized and injected into the combustion chamber. The propellants are mixed and burned in the combustion chamber. Control of the propellant shutoff valves in the propellant feed ducts is maintained through separate actuation control modules.

In addition to the propellant regenerative coils, there are helium repressurization coils wherein the helium is heated before being inducted to the propellant tanks. Helium flow to the repressurization coils is controlled by the LH_2/lox tank repressurization control module (dual valves). The burner produces 15 to 35 pounds thrust through the center of gravity as shown in figure 6-7.

Propellant Conditioning

During filling operations, the prevalves are allowed to stay in the open position to provide a deadhead type chilldown of the feed system hardware (low pressure feed duct and engine pump) allowing temperature stabilization of this hardware prior to activation of the recirculation chilldown system. At approximately five minutes before liftoff, the prevalves are closed and recirculating chilldown flow is initiated and continues for approximately 7-1/2 minutes until J-2 engine prestart (figure 6-8). Since lox is already a sub-cooled liquid (no two phase flow in the return line), prepressurization has negligible effect on the flowrate. The LH_2, however, becomes a subcooled liquid at prepressurization (eliminating two phase flow in the return line) resulting in an increased LH_2 chilldown flow.

Chilldown conditioning of the engine pumps, inlet ducting and the engine hardware for both engine starts is accomplished by separate lox and LH_2 chilldown systems. The purpose of the chilldown is to condition the ducting and engine to the proper temperature level and to eliminate bubbles (two phase flow) prior to pressurization. The chilldown, along with the net positive suction head which is obtained by the proper pressure levels, provides the proper starting conditions.

Propellants from each tank are recirculated through the feed systems and return bleed lines by chilldown pumps. Check valves, prevalves, shutoff valves, and ducts control and route the fluids to perform the chilldown. Pneumatic pressure for operating the shutoff valves and prevalves is supplied by the stage pneumatic helium control bottle.

LOX SYSTEM

Lox is stored in the aft tank of the propellant tank structure (figure 6-9) at a temperature of -297°F. Total volume of the tank is approximately 2826 cubic feet with an ullage volume of approximately 108 cubic feet. The tank is prepressurized between 38 and 41 psia and is maintained at that pressure during boost and engine operation. Gaseous helium is used as the pressurizing agent.

Lox Fill and Drain

The lox fill and drain valve is capable of allowing flow in either direction for fill or drain operations. During tank fill, the valve is capable of flowing 1000 gpm of lox at -297°F at an inlet pressure of 51 psia. Pneumatic pressure for operating the fill and drain valve is supplied by the stage pneumatic control bottle. Before loading begins, the ground controlled combination vent and relief valve is pneumatically opened and the lox tank is purged with gaseous helium.

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S-IVB

Loading begins with a precooling flowrate of 500 gpm. When the 5 percent load level is reached, fast fill (1000 gpm) is initiated. At the 98 percent load level, fast fill stops and a slow fill at 300 gpm begins. A fast fill emergency cutoff sensor has been provided to compensate for a primary control cutoff failure. Slow fill is terminated at the 100 percent load level and this level is then maintained by a replenish flowrate of 0 to 30 gpm, as required. The replenish flow is maintained through the complete lox tank prepressurization operation and the 100 percent lox load operation. Liquid level during fill is monitored by means of the PU mass probes.

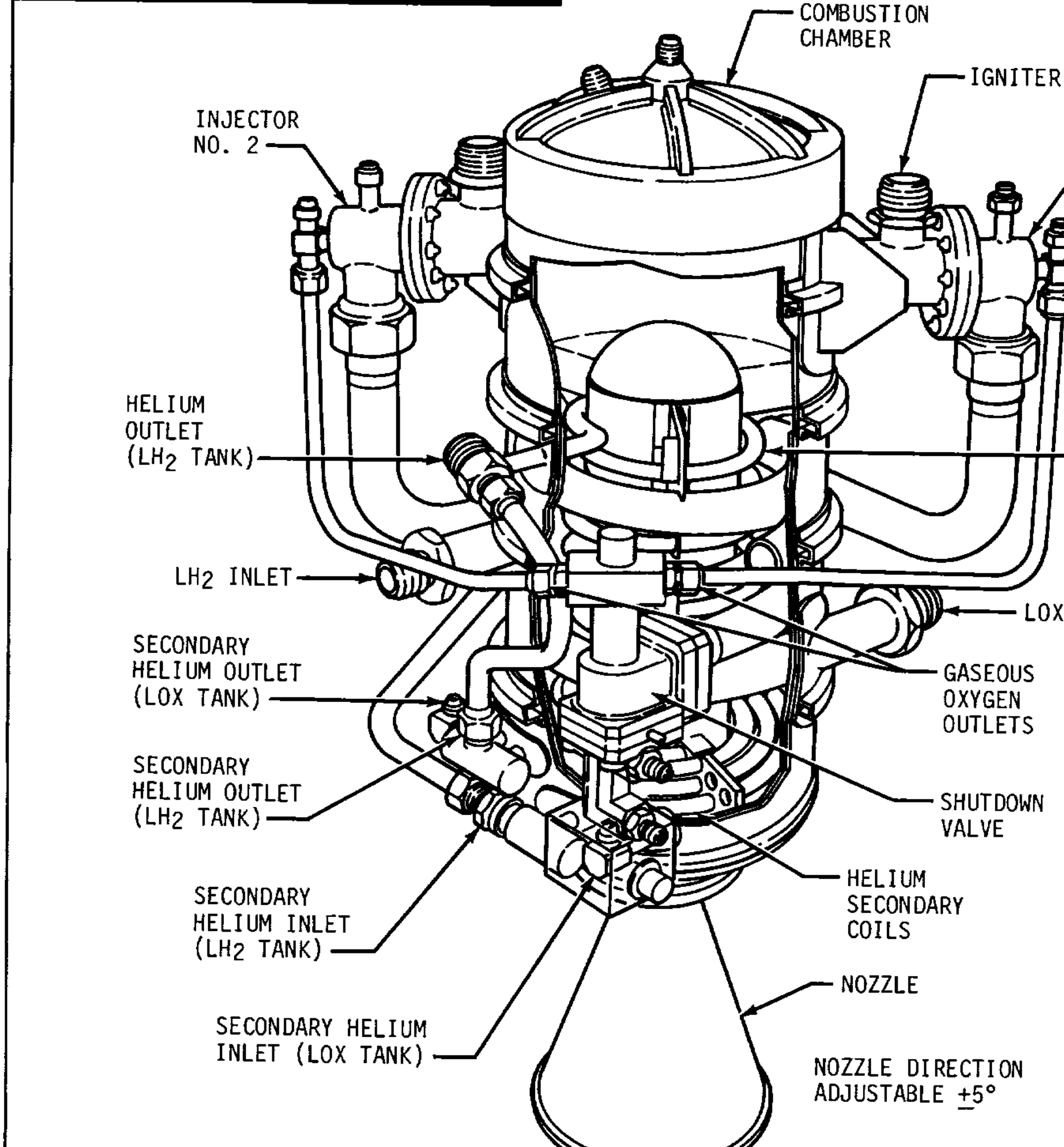
A lox vent system provides command venting of the lox tank, plus overpressure relief capability. Pressure-sensing switches are used to control the tank pressure during fill and flight.

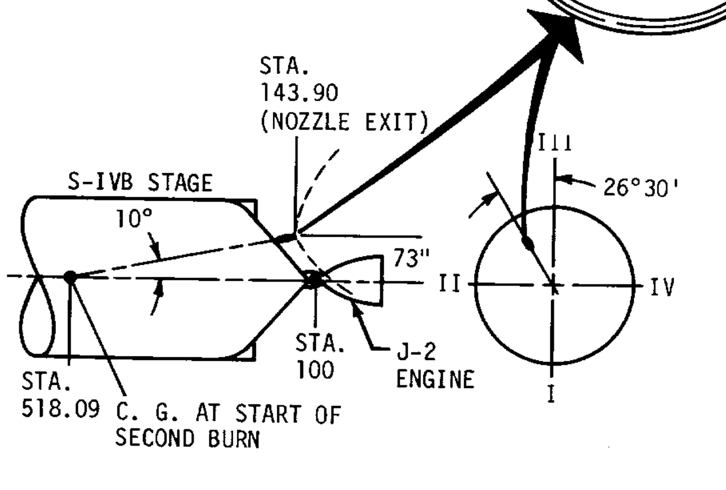
In the event of lox tank overpressurization (41 $\pm$ 0.5 psia) the pressure switch for the lox tank ground fill valve control signals the ground automatic sequencer to close the lox ground fill valve.

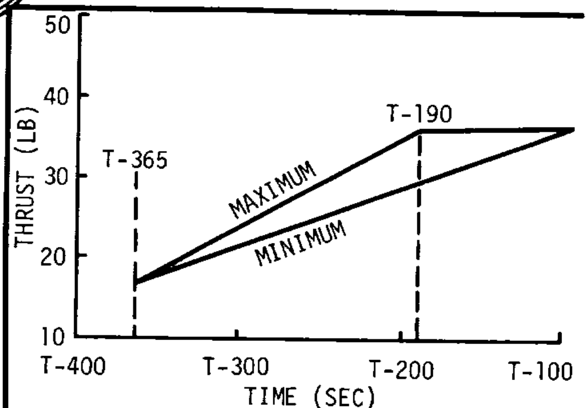
Lox Engine Supply

A six inch low pressure supply duct supplies lox from the tank to the engine. During engine burn, lox is supplied at a nominal flow rate of 392 pounds per second, and at a transfer pressure above 25 psia. The supply duct is equipped with bellows to provide compensating flexibility for engine gimbaling, manufacturing tolerances,

OXYGEN/HYDROGEN BURNER







NOTE: AT TIME (T) EQUAL TO ZERO, THE SECOND IGNITION OF THE S-IVB STAGE OCCURS

Figure 6-7

RECIRCULATION SYSTEMS

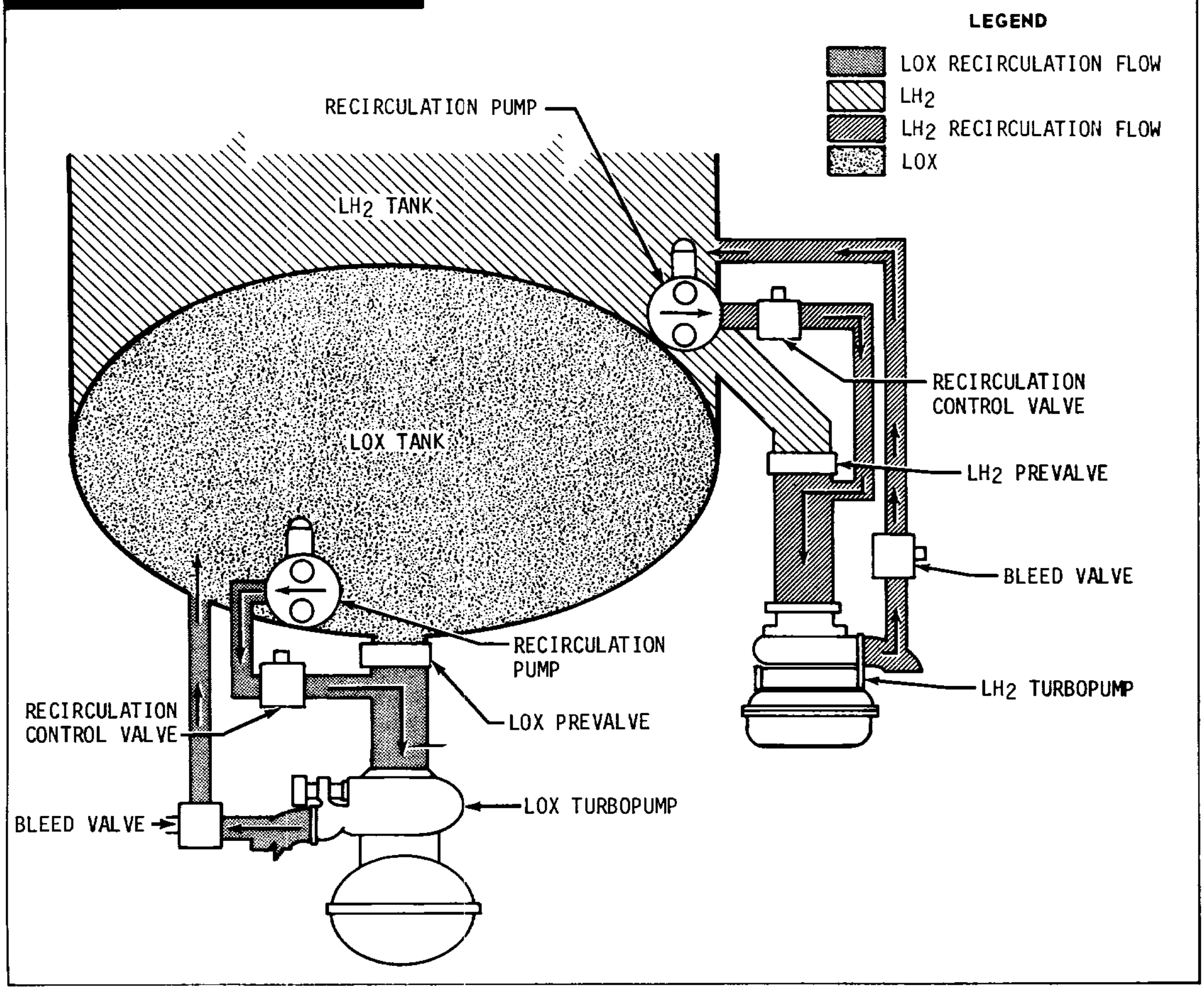


Figure 6-8

and thermal movement of structural connections.

LOX TANK PRESSURIZATION

Lox tank pressurization is divided into three basic procedures. These procedures are called prepressurization, pressurization, and repressurization. The term prepressurization is used for that portion of the pressurization performed on the ground prior to liftoff. The term pressurization is used to indicate pressurization during engine burn periods, and lastly, repressurization indicates pressurization just before a burn period.

The pressurant used during the three lox tank pressurization procedures is gaseous helium. Cold helium from a ground source (1, figure 6-10) is used during the prepressurization period. This ground source of cold helium also is used to charge the nine cold helium storage spheres (2). The cold helium storage spheres (2), located in the LH_2 tank, supply cold helium for both the pressurization and repressurization periods. The ambient helium storage spheres (3), filled by ground support equipment (26), supply an alternate source of helium for use during the repressurization period.

The lox tank pressure is controlled by the flight control pressure switches (4) (dual redundant) regardless of the pressurization procedure used. These switches control solenoid shutoff valves in each of the supply subsystems.

Prepressurization

At T-167 seconds the lox tank is pressurized (figure 6-10) by ground support equipment. Pressure regulated cold helium (1) passes through

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S-IVB

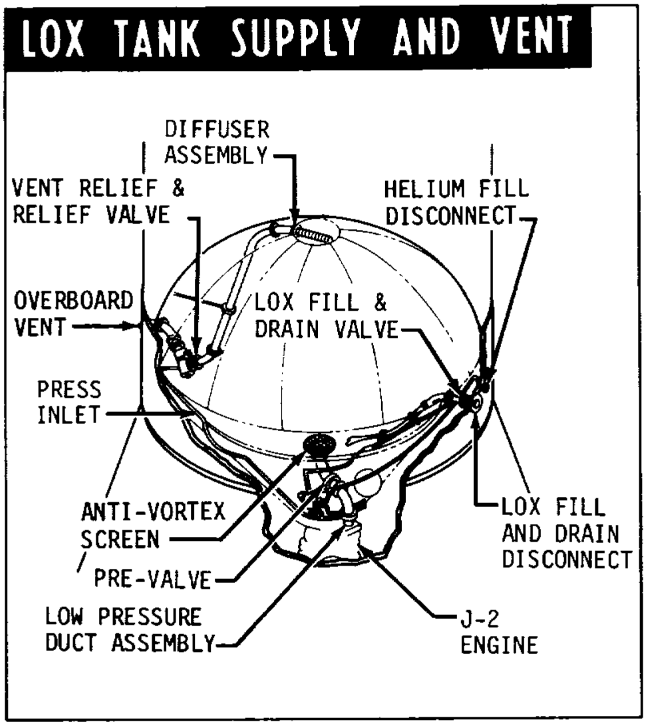

Figure 6-9

the lox tank pressure control module (5) and flows (6) into the lox tank. When the lox tank pressure increases to 41 psia the pressurization is completed and the flight control pressure switch shuts off the ground supply of cold helium.

Pressurization

After S-II/S-IVB separation at T_4 + 3.8, the lox tank flight pressure system is activated. When this system is activated cold helium from cold helium storage spheres (2, figure 6-10) is routed through the lox tank pressure control module (5) to the J-2 engine heat exchanger (7). When the cold helium passes through the lox tank pressure control module (5) its pressure is reduced to approximately 385 psia. As it passes through the heat exchanger it is expanded and routed to the lox tank. A small portion of cold helium bypasses the heat exchanger through a control orifice and mixes with the hot gas prior to entering the lox tank. The flight control pressure switch (4) controls the operation of a solenoid valve in the lox tank pressure control module (5) to control lox tank pressure to 38-41 psia. The flow (7) of the helium to the lox tank is monitored by a backup pressure switch (8). If normal pressure regulation fails, this switch (8) will control lox tank pressure through operation of solenoid valves in the lox tank pressure control module. In this manner lox tank pressurization is maintained during all engine burn periods. A S-IVB lox tank pressure reading becomes available in the command module (CM) at S-II/S-IVB separation. This pressure is sensed by the pressure transducer (12) and is relayed to the S-II FUEL/S-IVB OXID gauges (13) in the CM and, via telemetry, to the ground.

Repressurization

The normal repressurization procedure is initiated at T_6 + 96.3 seconds. It uses cold helium from the cold helium storage spheres (2, figure 6-10). The cold helium pressure is reduced to approximately 385 psia as it flows through the lox tank pressure control module (5). It next flows through the lox tank repressurization control (10), and into the O_2/H_2 burner (11). Should the regulator in the lox tank pressure control module (5) fail, the backup pressure switch (9) will maintain a pressure of 350-465 psia at the O_2/H_2 burner. The backup pressure switch controls the pressure by opening or closing valves in the lox tank repressurization module (10). As the cold helium is heated in the O_2/H_2 burner it expands and is routed to the lox tank. Pressure in the lox tank increases and is sensed by the flight control pressure switch (4) and the pressure transducer (12). The pressure switch (4) controls lox tank pressure between 38-41 psia by opening and closing solenoid shutoff valves in the lox tank repressurization control module (10). The pressure transducer (12) transmits a continuous pressure reading to telemetry and to the LV TANK PRESS gauges (13) in the CM. At T_6 + 346.4 seconds cryogenic repressurization is switched off. Ambient repressurization is turned on at T_6 + 380.0 seconds. Ambient helium from the ambient helium storage spheres (3) flows through the lox tank ambient helium repressurization control module (14) to the lox tank. Here the pressure is sensed by the flight control pressure switches (4). The pressure switches (14) control lox tank pressure by opening or closing the control valves in the lox tank ambient helium repressurization control module (14). Just before the engines second burn, ambient repressurization is terminated. (T_6 + 448.8 seconds).

NOTE

> The ambient repressurization portion of the repressurization procedure is in reality a backup procedure. Should the O_2/H_2 burner fail, ambient repressurization insures lox tank pressure for engine start requirements.

LH_2 SYSTEM

The LH_2 is stored in an insulated tank at less than -423°F. Total volume of the tank is approximately 10,400 cubic feet with an ullage volume of approximately 400 cubic feet. The LH_2 tank is prepressurized to 28 psia minimum and 31 psia maximum.

LH_2 Low Pressure Fuel Duct

LH_2 from the tank is supplied to the J-2 engine

PROPELLANT TANKS PRESSURIZATION

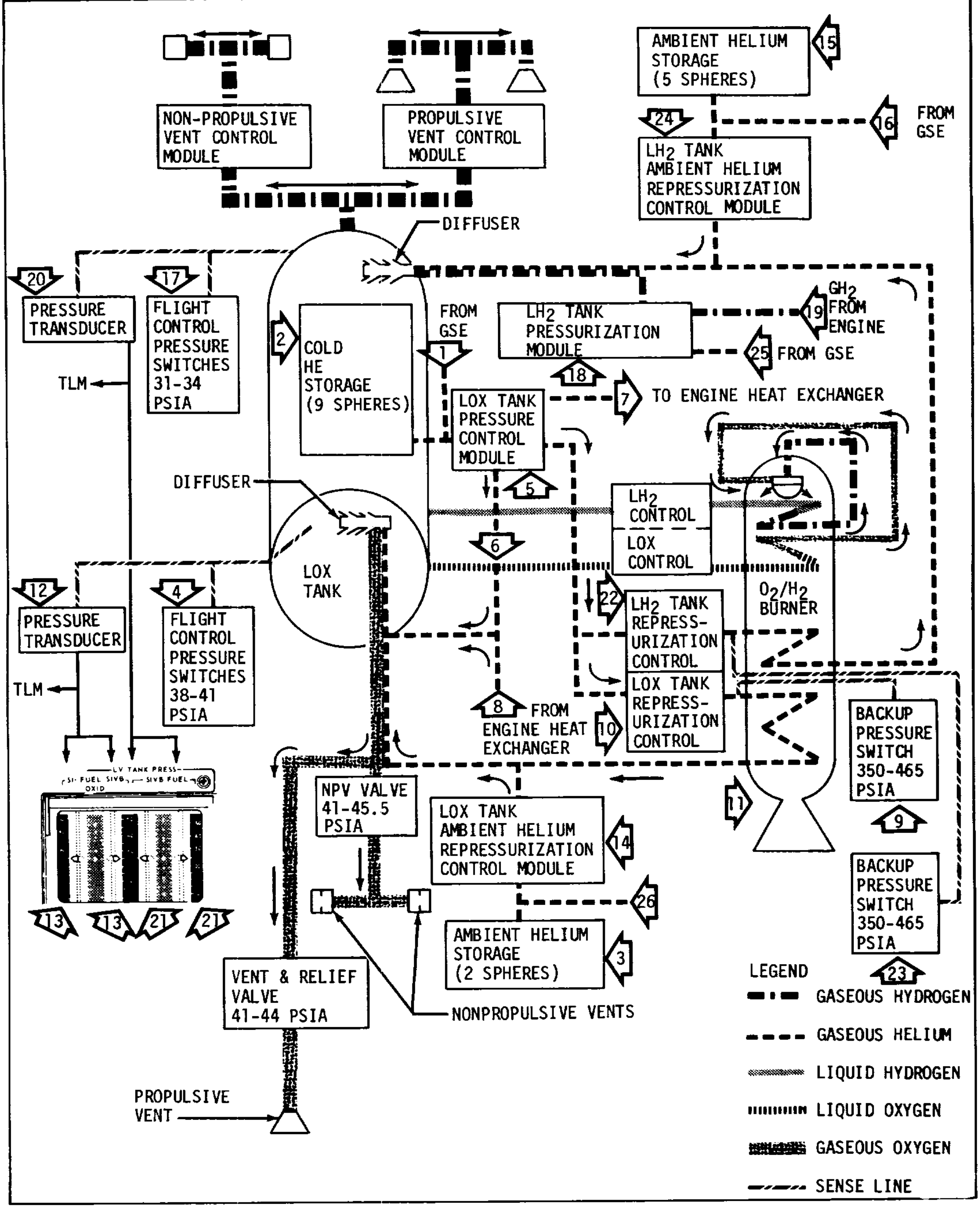

Figure 6-10

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turbopump through a vacuum jacketed low pressure 10-inch duct (figure 6-11). This duct is capable of flowing 80-pounds per second at -423°F and at a transfer pressure of 28 psia. The duct is located in the aft tank side wall above the common bulkhead joint. Bellows in this duct compensate for engine gimbaling, manufacturing tolerances, and thermal motion.

LH2 Fill and Drain

Prior to loading, the LH_2 tank is purged with helium gas. At the initiation of loading, the ground controlled combination vent and relief valve is pneumatically opened. The directional control valve is also positioned at this time to route GH_2 overboard to the burn pond.

Loading begins with precool at a flow of 500 gpm. When the 5 percent load level is reached fast fill is initiated at a flow of 3000 gpm. At the 98 percent load level, fast fill stops and a slow fill at 500 gpm begins. A fast fill emergency cutoff sensor has been provided to compensate for a primary control cutoff failure. Slow fill is terminated at the 100 percent load level and this level is then maintained by a replenish flowrate of 0 to 300 gpm, as required. The replenish flow is maintained through the complete LH_2 tank prepressurization operation.

Liquid level during fill is monitored by means of the PU mass probes. A backup overfill sensor is provided to terminate flow in the event of a 100 percent load cutoff failure.

An LH_2 vent system provides command venting of the LH_2 tank plus overpressure reelief capability. Pressure sensing switches are provided to control tank pressure during fill and flight.

LH2 TANK PRESSURIZATION

LH_2 tank pressurization is divided into three basic procedures. These procedures are called prepressurization, pressurization, and repressurization. The term prepressurization is used for that portion of the pressurization performed on the ground prior to liftoff. The term pressurization is used to indicate pressurization during engine burn periods, and lastly, repressurization indicates pressurization just before a burn period.

The pressurants used during the three LH_2 tank pressurization procedures are gaseous hydrogen (GH_2) and gaseous helium. Cold helium from a ground source (25, figure 6-10) is used during the prepressurization period. The cold helium storage spheres (2), located in the LH_2 tank, supply cold helium for use during the repressurization period. The five ambient helium storage spheres (15), filled by ground support equipment (16), supply an alternate source of helium for use during repressurization.

The LH_2 tank pressure is controlled by the flight control pressure switches (17) (dual redundant) regardless of the pressurization procedure used. These switches control solenoid shutoff valves in each of the supply subsystems.

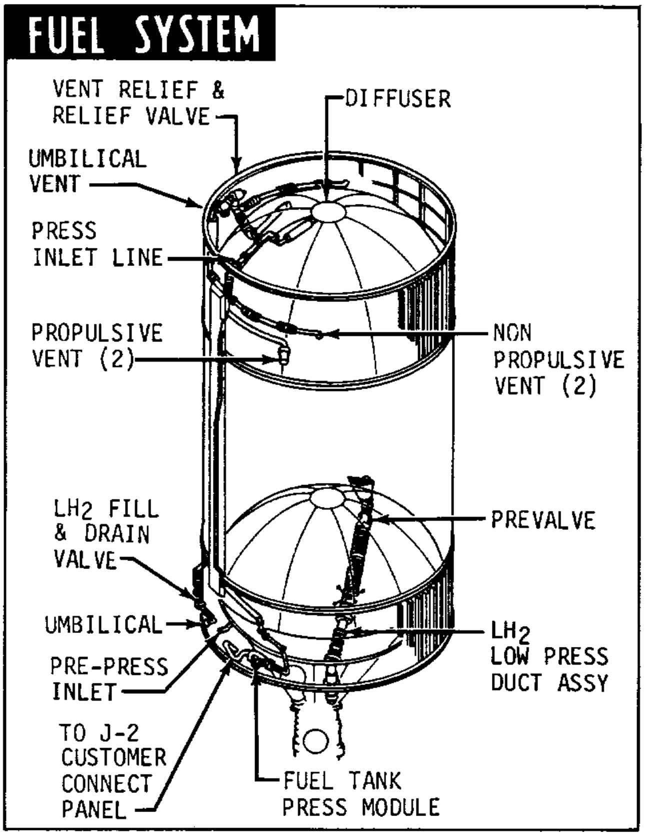


Figure 6-11

Prepressurization

At T_1 -187 seconds the LH_2 tank is prepressurized (figure 6-10) by ground support equipment. Cold helium (25) flows through the LH_2 tank pressurization module (18) and into the LH_2 tank. When the LH_2 tank pressure increases to 31 psia (T_1 -43 seconds) the flight control pressure switch (17) shuts off the ground supply of cold helium (25) to complete prepressurization.

Pressurization

Pressurization is controlled by the flight control pressure switches (17, figure 6-10) which open or close solenoid valves in the LH_2 tank pressurization module (18). Gaseous hydrogen (GH_2) (19) bled from the J-2 engine flows through the LH_2 tank pressurization module (18) to the LH_2 tank. As pressure in the LH_2 tank increases to 31 psia the flight control pressure switches (17) close valves in the LH_2 tank pressurization module to control tank pressure at 28-31 psia. This pressure is sensed by the pressure transducer (20) and is relayed to the S-IVB fuel gages (21) in the CM and, via telemetry, to the ground. In this manner LH_2 tank pressurization is maintained during engine burn periods.

Repressurization

The normal repressurization procedure is initiated at T_6 + 96.1 seconds. It uses cold helium from the cold helium storage spheres (2, figure 6-10). The cold helium pressure is reduced to approximately 385 psia as it flows through the lox tank pressure control module (5). The cold helium next flows through the LH_2 tank repressurization control (22), and into the O_2/H_2 burner (11). Should the regulator in the lox tank pressure control module (5) fail, the backup pressure switch (23) will maintain a pressure of 350-465 psia at the O_2/H_2 burner. The backup pressure switch controls the pressure by opening or closing valves in the LH_2 tank repressurization module (22). As the cold helium is heated in the O_2/H_2 burner (11) it expands and is routed to the LH_2 tank. Pressure in the LH_2 tank increases and is sensed by the flight control pressure switch (17) and the pressure transducer (20). The pressure switch (17) controls LH_2 tank pressure between 28-31 psia by opening and closing solenoid shutoff valves in the LH_2 tank repressurization control module (22). The pressure transducer (21) transmits a continuous pressure reading to telemetry and to the LV TANK PRESS gauges (21) in the CM. At T_6 + 346.6 seconds cryogenic repressurization is switched off. Ambient repressurization is turned on a T_6 + 379.8 seconds. Ambient helium from the ambient helium storage spheres (15) flows through the LH_2 tank ambient helium repressurization control module (24) to the LH_2 tank. Here the pressure is sensed by the flight control pressure switches (17). The pressure switches (17) control LH_2 tank pressure by opening or closing the control valves in the LH_2 tank ambient helium repressurization control module (24). Just before the engines second burn, ambient repressurization is terminated. (T_6 + 449.0 seconds).

NOTE

The ambient repressurization portion of the repressurization procedure is in reality a backup procedure. Should the O_2/H_2 burner fail, ambient repressurization insures LH_2 tank pressure for engine start requirements.

Directional Control Valve

The directional control valve is a two position valve provided to enable command routing of gaseous hydrogen through the ground vent line or through the flight nonpropulsive vents. During LH_2 loading the subsystem provides overboard venting of boiloff vapors through the ground line to the gaseous hydrogen burn pond. The products of vaporization are vented through a disconnect located in the forward skirt.

At T-27 seconds, the directional control valve is positioned for flight venting to direct GH_2 through the nonpropulsive vents. The nonpropulsive vent assembly consists of a hydrogen vent line originating at one outlet of the directional control valve. The main line branches into two feeder lines that terminate in two nonpropulsive vent orifices located 180 degrees apart in the forward skirt area. The ports direct the exhausts for total thrust cancellation.

PNEUMATIC CONTROL

The pneumatic control system provides supply pressure for all stage pneumatically operated valves with the exception of J-2 engine valving but including the engine start tank vent valve. The pressure source is ambient helium stored at 3100 ± 100 psia at 70°F in the ambient helium sphere. A pneumatic power control module reduces and regulates the pressure to 490 ± 25 psia for operation of the stage pneumatic valves.

Prior to loading propellant tanks, the ambient helium control bottle is pressurized to a minimum of 750 psia. This provides sufficient control pressure to operate the stage valves until the ground helium supply pressure is switched to 3100 psia. The cold helium bottles in the LH_2 tank are chilled prior to pressurizing to 3100 psia. All helium bottles are, therefore, pressurized to flight pressure when the LH_2 load level reaches 92 percent.

During the propellant loading operations regulated control pressure is supplied by the control module to maintain the lox and LH_2 chilldown shutoff valves in the closed position and to open and close the lox and LH_2 fill and drain valves as required.

It should be noted that the pneumatic regulator is set at 475 psig which is equivalent to 490 psia on the ground and 475 psia in orbit. During stage operations the system also supplies pressure to the solenoid valves to open and close the lox and LH_2 tank vent/relief valves and the LH_2 directional control valve.

At the initiation of GH_2 start tank fill, the pneumatic control system supplies helium to operate the start tank vent/relief valve. Prior to loading propellants, the system also supplies purge helium gas to the J-2 turbopumps and the lox chill pump.

Generally speaking, each pneumatically operated component has a separate actuation control module (ACM). Two solenoids on each module enable command on-off control of each pneumatically operated valve. However, the propellant prevalves and the propellant chilldown shutoff valves share one ACM.

The pneumatic control subsystem is protected from overpressure by a normally open solenoid valve controlled by a downstream pressure switch. At pressures greater than 600 ± 15 psia, the pressure switch actuates and closes the valve. At pressure below 490 ± 25 psia, the

VI S-IVB

pressure switch deactuates and the valve opens. In this manner the pressure switch acts as a backup to the regulator.

FLIGHT CONTROL

The flight control system incorporates two systems for flight and attitude control. During powered flight thrust vector steering is accomplished by gimbaling the J-2 engine for pitch and yaw control and by operating the APS engines for roll control. Steering during coast flight is by use of the APS engines alone. (See Auxiliary Propulsion Systems subsection for coast flight steering).

ENGINE GIMBALING

During the boost and separation phase the J-2 engine is commanded to the null position to prevent damage by shifting. The engine is also nulled before engine restart to minimize the possibility of contact between the engine bell and the interstage at S-II/S-IVB separation and to minimize inertial effects at ignition. The engine is gimbaled (figure 6-12) in a $\pm$ 7.5 degrees square pattern by a closed loop hydraulic system. Mechanical feedback from the actuator to the servovalve provides the closed engine position loop.

When a steering command is received from the flight control computer, a torque motor in the servovalve shifts a control flapper to direct the fluid flow through one of two nozzles. The direction of the flapper is dependent upon signal polarity.

Two actuators are used to translate the steering signals into vector forces to position the engine. The deflection rates are proportional to the pitch and yaw steering signals from the flight control computer.

HYDRAULIC SYSTEM

Major components of the hydraulic system (figure 6-13) are an engine driven hydraulic pump, an electrically driven auxiliary hydraulic pump, two hydraulic actuator assemblies, and an accumulator/reservoir assembly.

Hydraulic Pumps

The engine/driven hydraulic pump is a variable displacement type driven directly from the engine oxidizer turbopump. In normal operation, the pump delivers up to 8 gpm under continuous working pressure.

The auxiliary hydraulic pump is an electrically driven pump which is capable of supplying a minimum of 1.5 gpm of fluid to the system. This pump supplies pressure for preflight checkout, to lock the J-2 engine in the null position during boost and separation phase and as emergency backup. During orbit the auxiliary pump, controlled by a thermal switch, circulates the hydraulic fluid to maintain it between +10°F and +40°F. The auxiliary pump is started before liftoff and during coast periods.

Accumulator/Reservoir Assembly

The accumulator/reservoir assembly (figure 6-13) is an integral unit mounted on the thrust structure. The reservoir section is a storage area for hydraulic fluid having a maximum volume of 167 cubic inches.

During system operation, between 60 and 170 psig is maintained in the reservoir (figure 6-12) by two pressure operated pistons contained in the accumulator section. In addition to maintaining pressure in the reservoir, the system accumulator supplies peak system demands and dampens high pressure surging.

Hydraulic Actuators - Pitch and Yaw

The pitch and yaw actuators and servovalve (figure 6-13) are integrally mounted and are interchangeable. The actuators are linear and double acting. During powered flight, pitch and yaw control is provided by gimbaling the main engine, the two actuator assemblies providing deflection rates proportional to pitch and yaw steering signals from the flight control computer.

AUXILIARY PROPULSION SYSTEM

The S-IVB auxiliary propulsion system provides three axis stage attitude control (figure 6-14) and main stage propellant control during coast flight.

APS CONSTRUCTION

The APS engines are located in two modules 180° apart on the aft skirt of the S-IVB stage (see figure 6-15). The modules are detachable and are easily checked or replaced. Each module contains four engines; three 150-pound thrust control engines (figure 6-16) and one 70-pound thrust ullage engine (figure 6-17). Each module contains its own oxidizer, fuel, and pressurization systems. A positive expulsion propellant feed subsystem is used to assure that hypergolic propellants are supplied to the engines under "zero G" or random gravity conditions. This subsystem consists of separate fuel and oxidizer propellant tank assemblies (figure 6-18) each containing a bladder for propellant expulsion, individual propellant control modules that filter the propellants and provide auxiliary ports for subsystem servicing operations, and a propellant manifold for distribution of propellants to the engines.

Nitrogen tetroxide (N_2O_4), MIL-P-28539A, is used as an oxidizer and monomethyl hydrazine (MMH), MIL-P-27404, is used as fuel for these engines. The 150-pound thrust engines utilize eight con-

THRUST VECTOR CONTROL SYSTEM

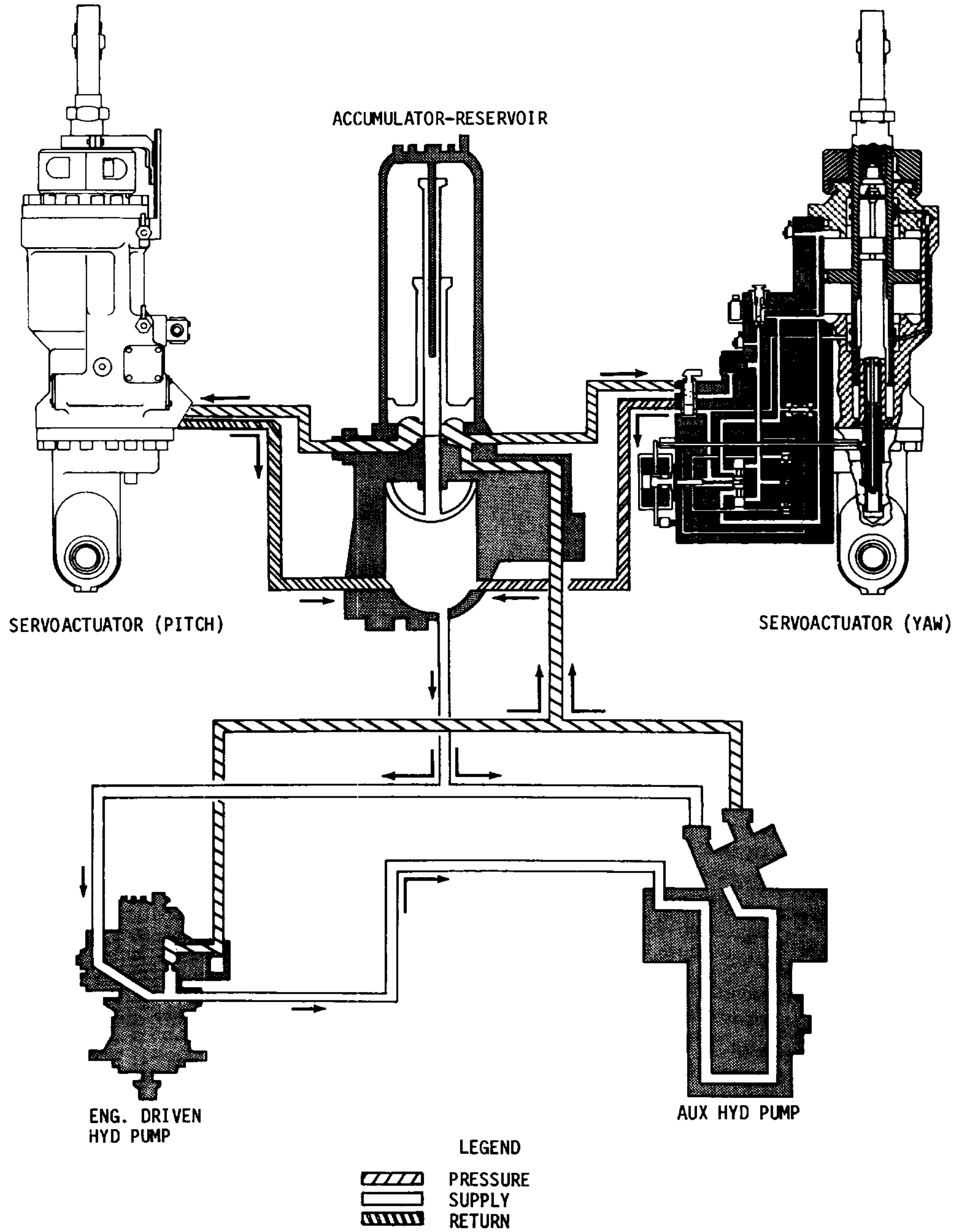


Figure 6-12

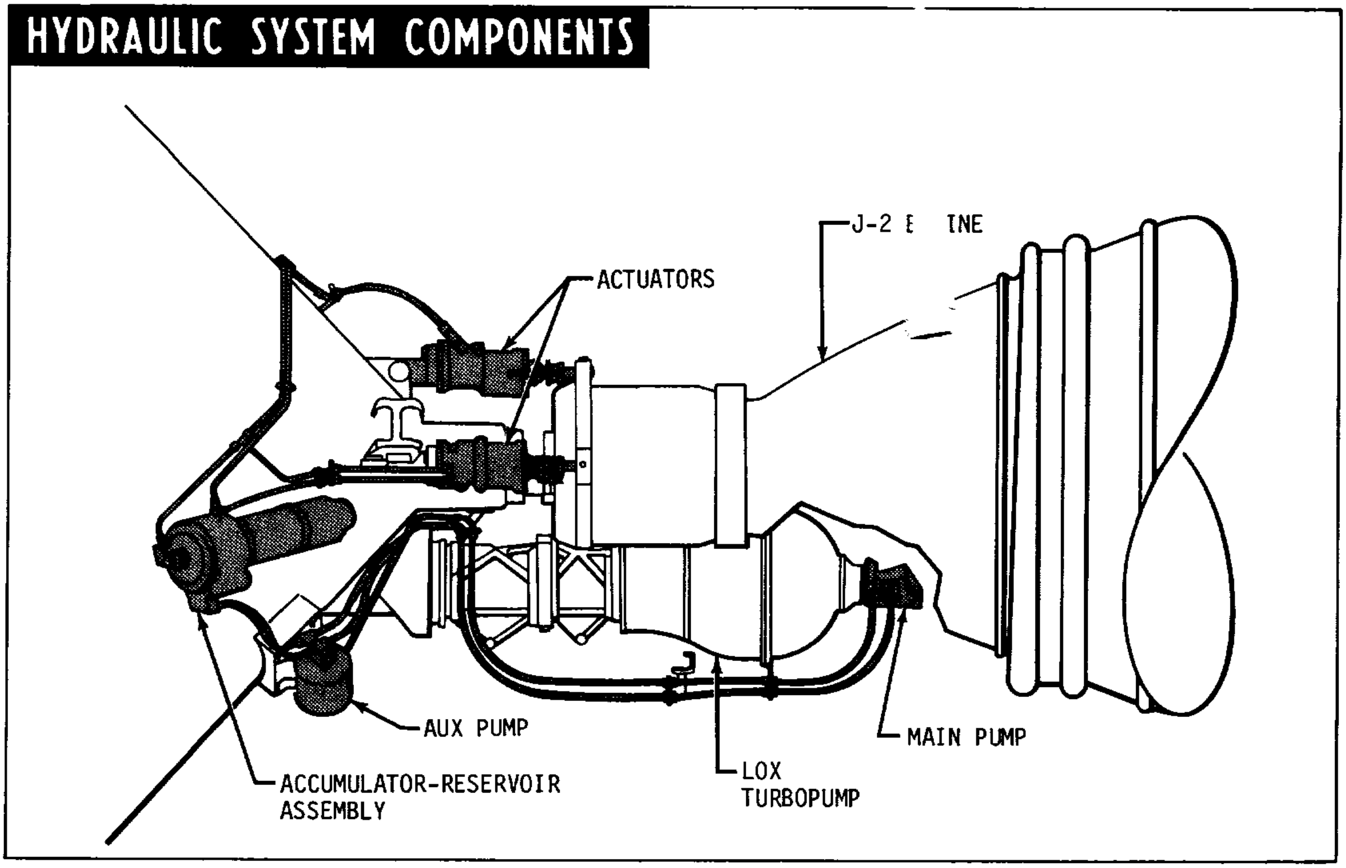


Figure 6-13

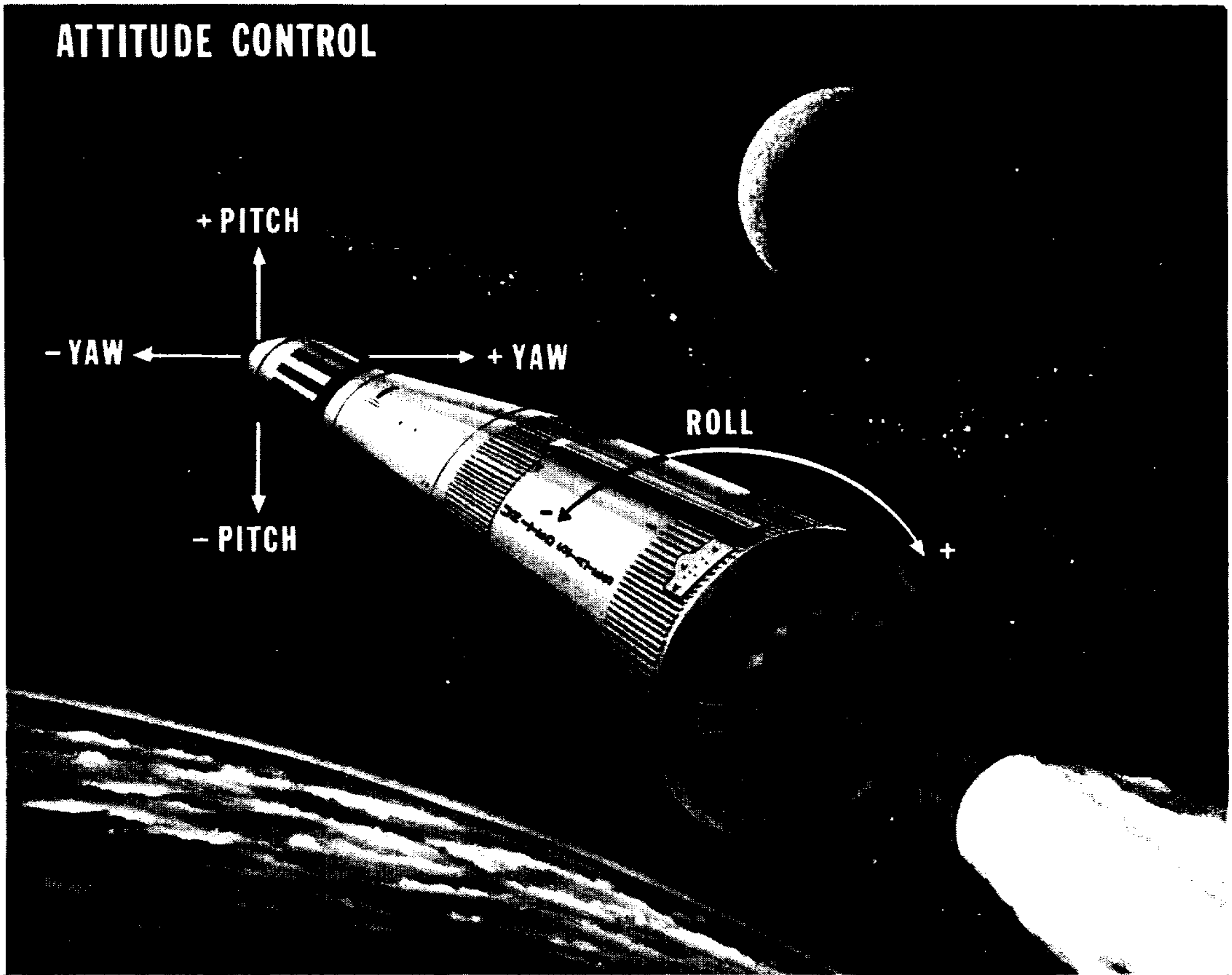
trol valves (four for fuel, four for oxidizer) in a fail safe series parallel arrangement. The 70-pound ullage engine uses only single valves on both the fuel and oxidizer lines.

APS OPERATION

Two of the 150-pound thrust engines in each module control roll and yaw, while one controls pitch. The 70-pound thrust engine in each module is used to settle the propellants in the propellant tanks. Just prior to J-2 engine restart, propellant control is accomplished by firing the 70-pound thrust engines. The attitude control engines are fired for attitude correction. The minimum engine firing pulse duration is approximately 70 milliseconds. Typical operation (figure 6-19) involves a signal from the instrument unit to energize the pitch control relays which open the eight normally closed quad redundant propellant valves (4 oxidizer, 4 fuel) arranged in two series parallel circuits. Valve assembly failure cannot occur unless two valves fail open in series (propellant drain) or valves fail closed in parallel (oxidizer or fuel starvation). Single valve failures does not affect engine proficiency. The attitude deviation dead band for roll, pitch, and yaw is $\pm1°$.

The attitude engine control signal is composed of an attitude error signal and a vehicle turning rate signal. The body mounted control EDS rate gyros, located in the Instrument Unit, supply the vehicle turning rate information. Attitude error information may come from two sources. When the Instrument Unit is controlling the vehicle attitude, error information is supplied by the LVDC through the flight control computer. When the spacecraft is controlling the vehicle, attitude error information is obtained from the spacecraft. Limits are imposed by the LVDC for IU generated attitude error signals and by the IU flight control computer for spacecraft generated attitude error signals. It should be noted that attitude error limits of $\pm2.5°$ in pitch and yaw and $\pm3.5°$ in roll are imposed on the attitude error signals. These error signals are used only by the auxiliary propulsion system. The attitude error signals from the spacecraft may originate in the Apollo navigation, guidance and control system or may be generated by the Astronaut through manual control. In any case, the limiters in the IU flight control computer will limit the angular rate. These limits prevent excessive propellant usage which would result from large angular rate commands while driving the vehicle to the desired attitude.

The Apollo spacecraft attitude reference system can follow the instantaneous vehicle attitude. This is accomplished by driving the command display unit servo motor with an error signal which


ATTITUDE CONTROL
+ PITCH
- YAW
+ YAW
- PITCH
ROLL
-
+

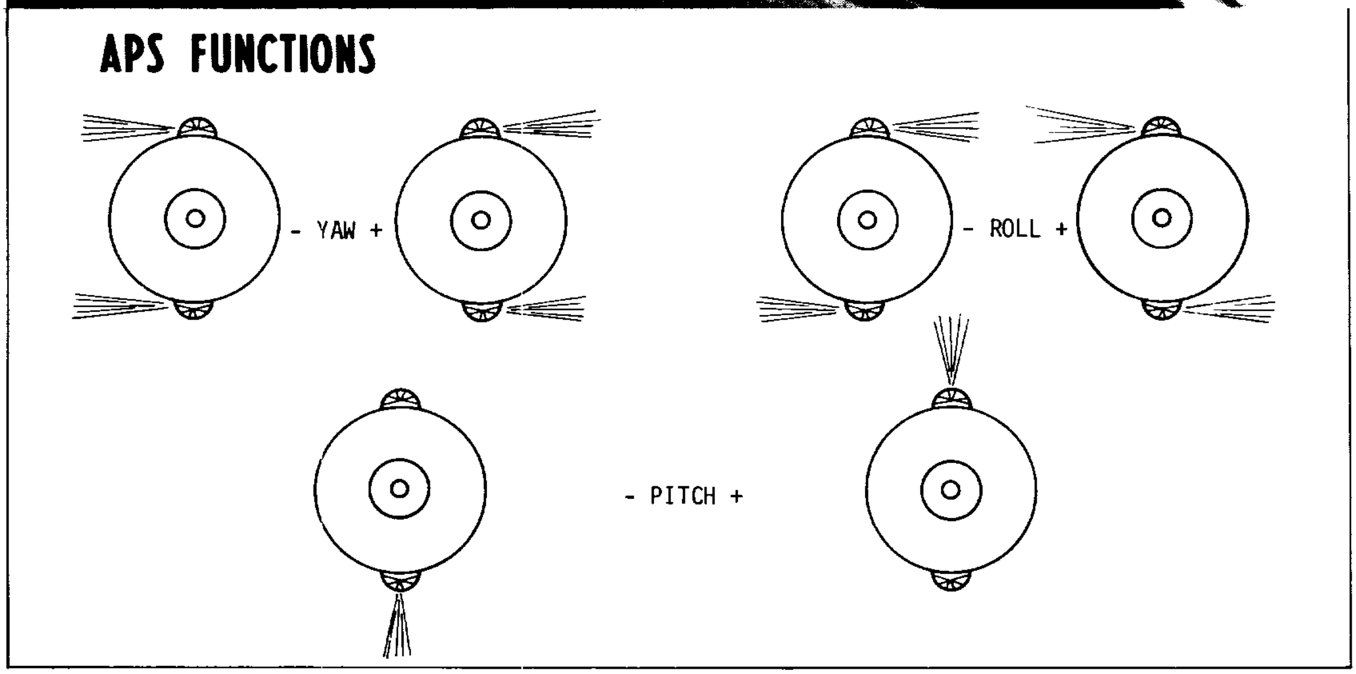

APS FUNCTIONS
- YAW +
- ROLL +
- PITCH +

Figure 6-14

AUXILIARY PROPULSION SYSTEM CONTROL MODULE

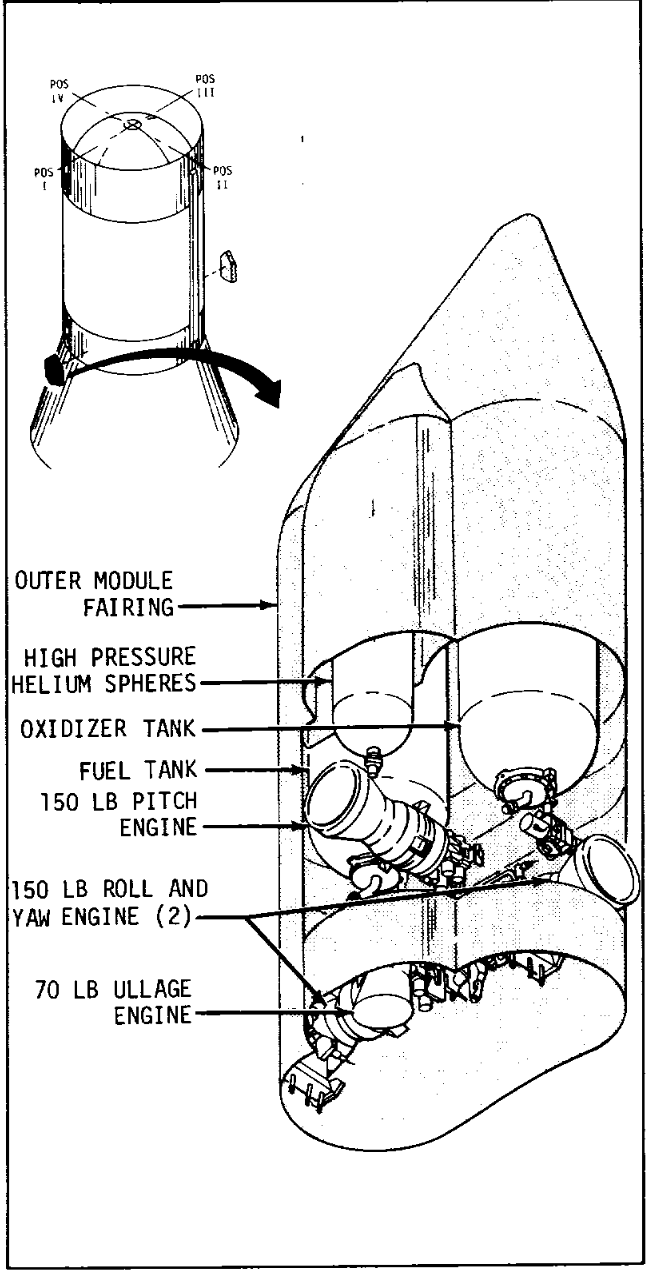


Figure 6-15

is formed by differencing the commanded and actual gimbal angles. When the Astronaut wishes to maintain a particular attitude orientation, he can use the computer to set the command display unit command resolver to the desired gimbal value. The difference between the commanded and actual gimbal angles results in an error signal which is resolved into vehicle coordinates and is given to the IU flight control computer as an attitude error signal. The S-IVB attitude control system then operates in the limit cycle mode about this command attitude.

ELECTRICAL

The electrical system of the S-IVB stage is comprised of two major subsystems: the electrical power subsystem which consists of all the power sources on the stage; and the electrical control subsystem which distributes power and control signals to various loads throughout the stage.

BATTERIES

On board power is supplied by four zinc silver-oxide batteries. Two are located in the forward equipment area and two in the aft equipment area. These batteries are activated and installed in the stage during the final prelaunch preparations. Heaters and instrumentation probes are an integral part of each battery. See figure 6-20 for a table of battery characteristics and figure 6-21 for relative location of batteries.

POWER DISTRIBUTION

Two main forward busses and two main aft busses supply electrical power to all stage systems. Busses are electrically isolated from each other with each main bus utilizing a power transfer switch to switch from GSE power to stage mounted batteries.

Figure 6-22 illustrates the electrical power distribution for the S-IVB stage. The four main busses are configured to perform the following functions:

1. The function of forward bus No. 1 is to provide 28 vdc power to the telemetry transmitters, data acquisition system, transducers, level sensors, switch selector, range safety system No. 1, checkout measurement group, 5-vdc excitation modules, and the forward battery No. 1 and No. 2 heaters.

2. Forward bus No. 2 supplies 28 vdc power to the propellant utilization electronic assembly, the propellant utilization static inverter-converter, and range safety system No. 2. The requirements of redundant range safety and emergency detection systems dictated the use of a second battery in the forward skirt.

3. Aft bus No. 1 supplies 28 vdc power to the J-2 engine, sequencer, pressure switches, all propulsion system valves which are flight operational, attitude control system modules No. 1 and No. 2, ullage rocket motor ignition and jettison system, aft battery No. 1 and No. 2 heaters.

4. The aft bus No. 2 provides 56 vdc power to the LH_2 and lox chilldown inverters and the auxiliary hydraulic pump motor. The high power requirements of the chilldown

150-POUND THRUST APS ENGINE

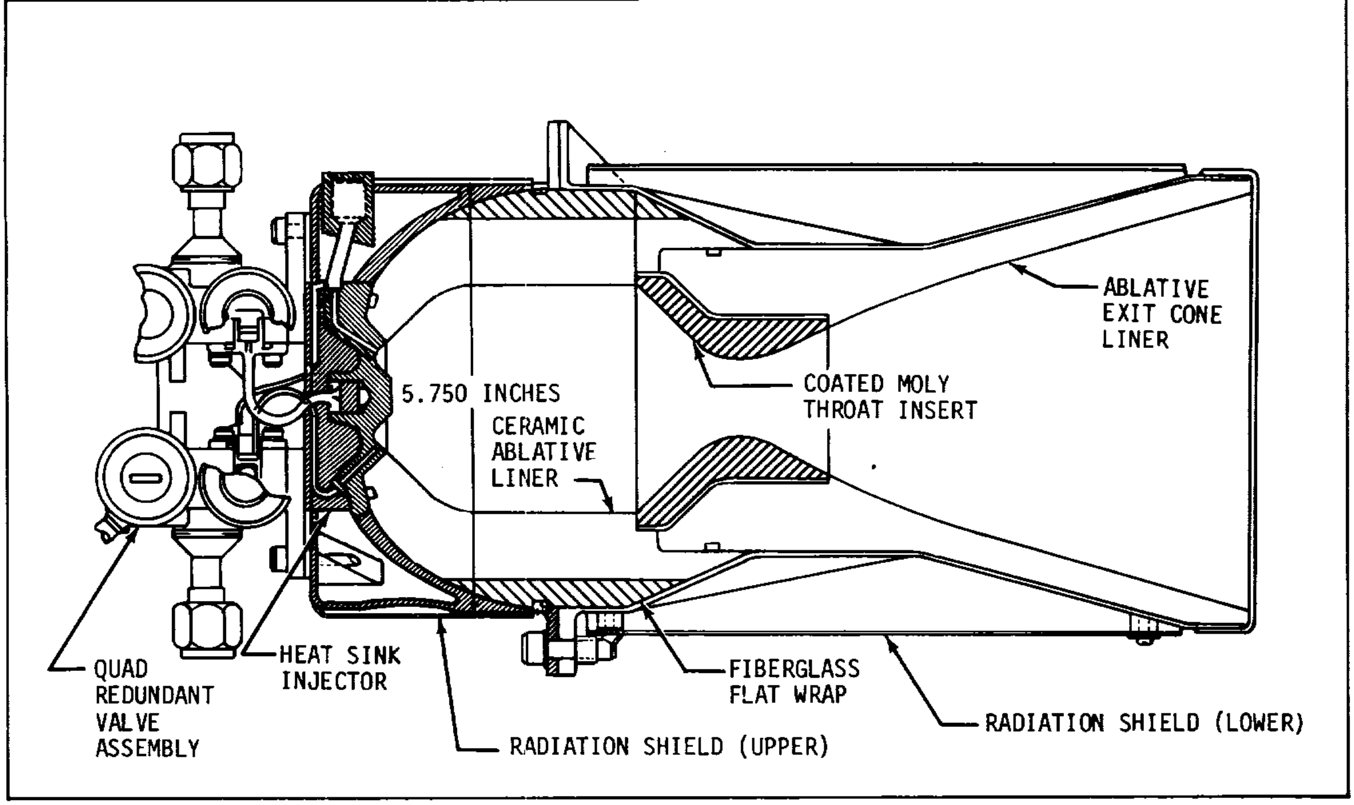


Figure 6-16

70-POUND THRUST APS ENGINE

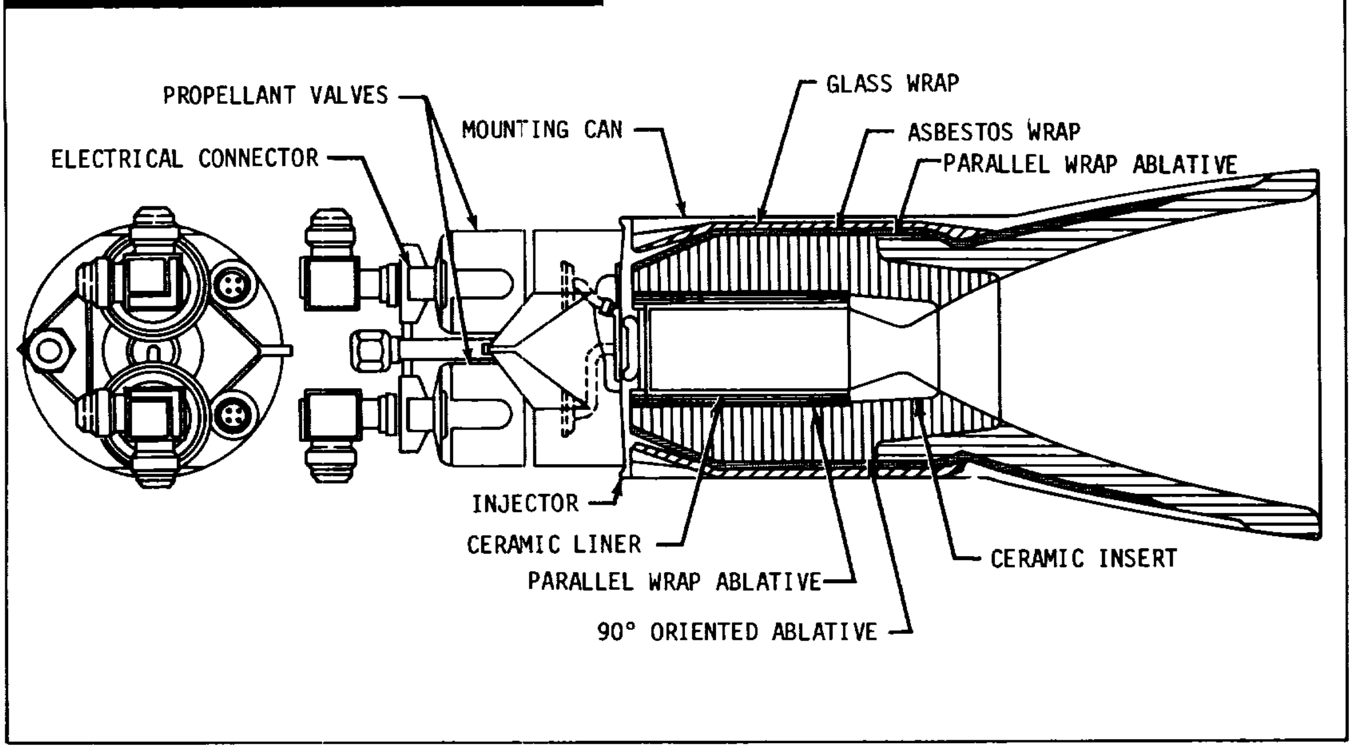


Figure 6-17

VI
S-IVB

motors and hydraulic pump motor necessitated the 56 volt battery.

CHILLDOWN INVERTERS

The chilldown inverters are three-phase, 1500 volt-amp, solid-state power conversion devices. The purpose of the chilldown inverters is to provide electrical power to the motor driven pumps for circulation of the lox and LH_2 to ensure propellant temperature stabilization at the J-2 engine inlets.

PROPELLANT UTILIZATION (PU) STATIC INVERTER-CONVERTER

The static inverter converter is a solid state power supply which provides all the regulated ac and dc voltages necessary to operate the S-IVB stage propellant utilization electronics assembly.

The static inverter converter converts a 28 vdc input into the following outputs:

1. 115 vac, 400 Hz required to drive propellant utilization valve positioning motors and bridge rebalancing servo motors.
2. 2 vac, peak-to-peak square wave required to convert propellant utilization error signal to alternating current.
3. 5 vdc required to provide excitation for propellant utilization fine and coarse mass potentiometers.
4. Regulated 22 vdc required for propellant utilization bridges.
5. 117.5 vdc floating supply used to provide propellant utilization summing potentiometer excitation.
6. 49 vdc floating supply required to provide excitation for valve feedback potentiometers.

EXCITATION MODULES

The 5 volt and 20 volt excitation modules are transistorized power conversion devices which are used to convert the 28 vdc to the various regulated voltages required by the instrumentation, signal conditioning, and emergency detection system transducers.

ELECTRICAL CONTROL SUBSYSTEM

The electrical control subsystem function is to distribute the command signals required to control the electrical components of the stage. The major components of the electrical control subsystem are the power and control distributors, the sequencer assemblies, and the pressure sensing and control devices.

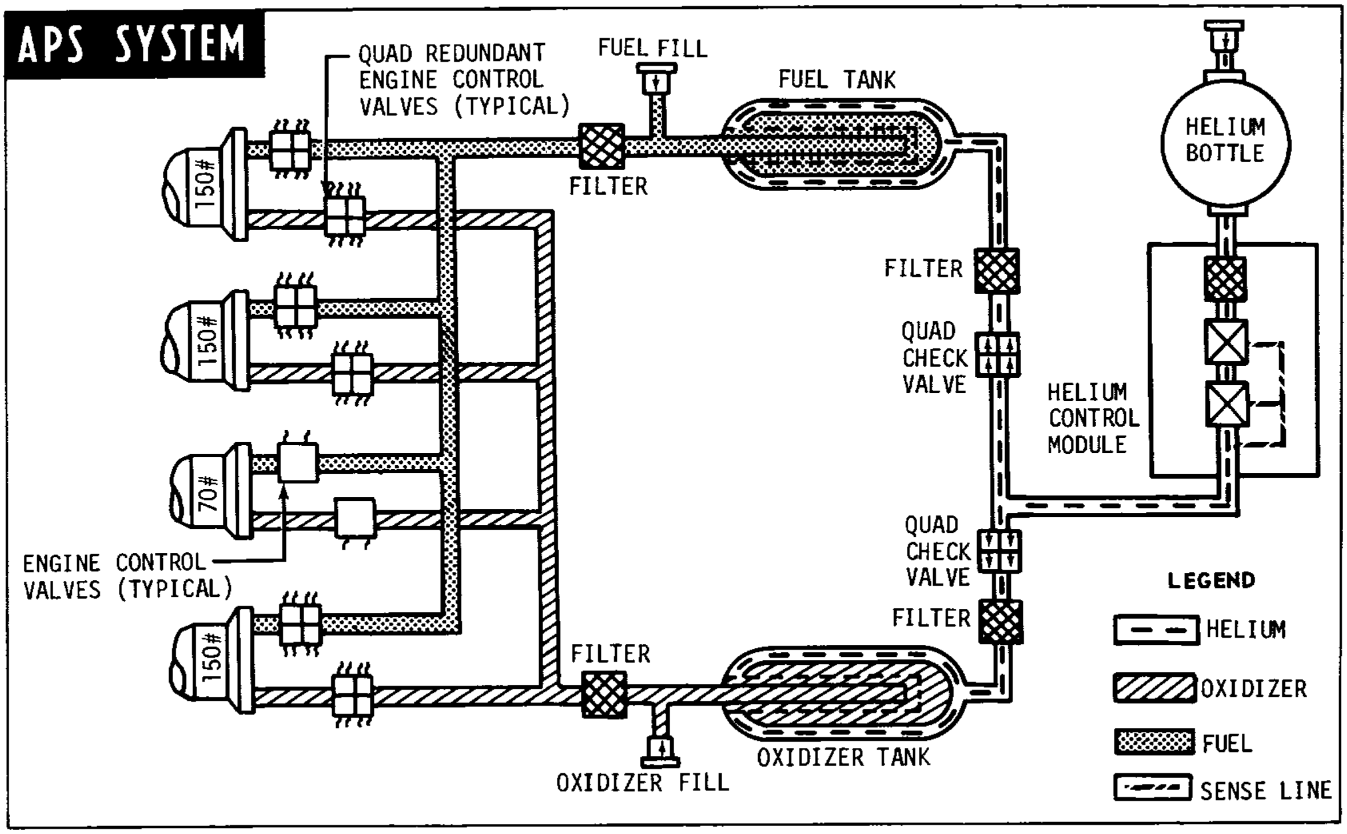


Figure 6-18

SEQUENCER ASSEMBLIES

The two major elements in the stage sequencing system are the switch selector and the stage sequencer. During flight, sequencing commands are received from the IU. Each command is in digital form and consists of an 8-bit word accompanied by a "read" pulse. The commands are interpreted by the S-IVB stage at the switch selector.

Switch Selector

The switch selector is an electronic assembly utilized as the primary device for controlling the inflight sequencing of the stage. A switch selector is utilized in each stage of the launch vehicle.

The switch selector consists of relays, a diode matrix, and low-power transistor switches used as relay drivers controlled by binary-coded signals from the LVDC in the IU. The function of the switch selector is to operate magnetically latching relays in the sequencer and power distribution assemblies.

The switch selector provides electrical isola-

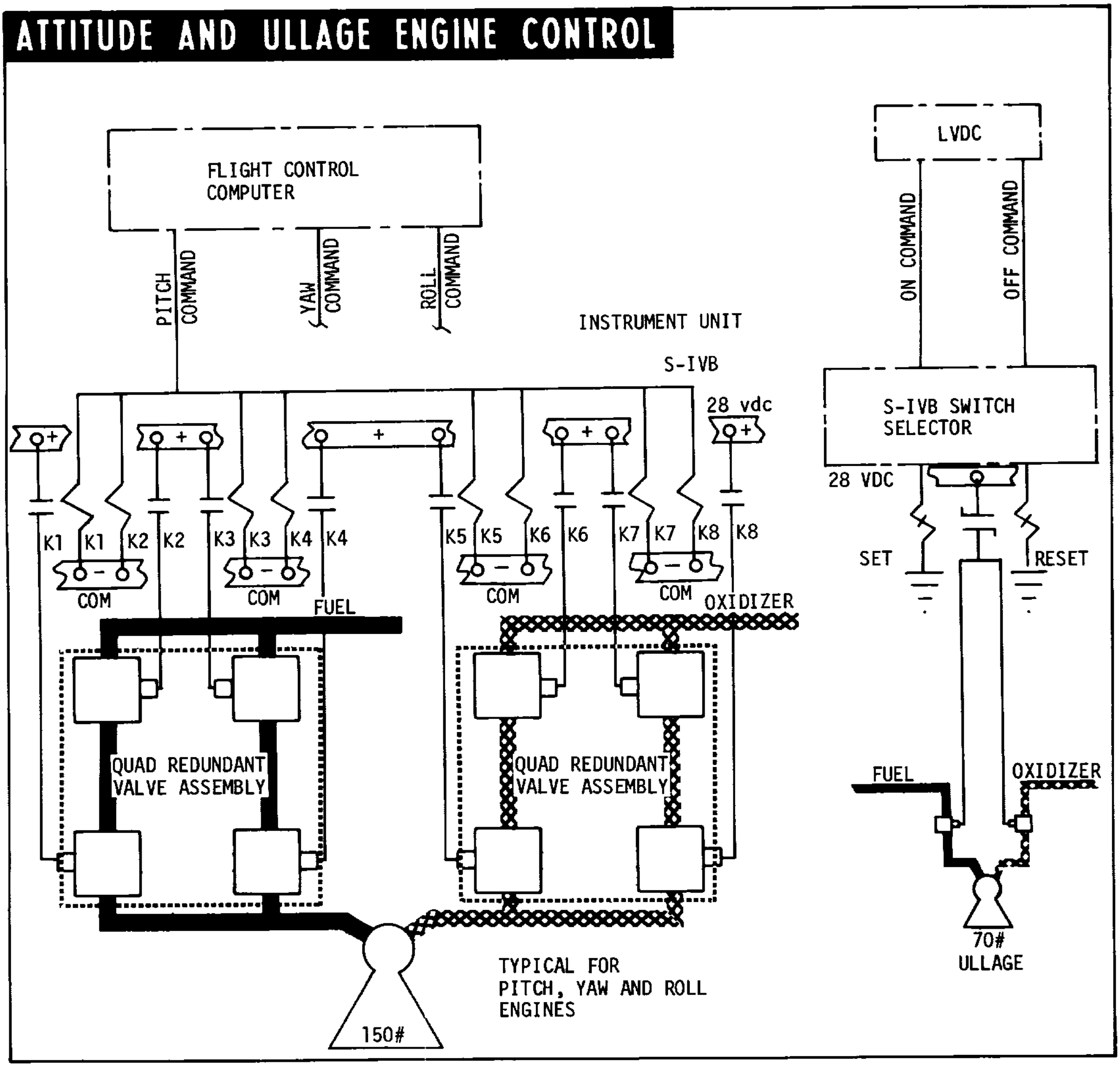


Figure 6-19

tion between the IU and the S-IVB stage systems and decodes digital information from the IU into discrete output commands. Capability exists to provide 112 discrete commands to the stage sequencer.

Stage Sequencer

The stage sequencer operates upon receipt of discrete inputs from the switch selector (and other S-IVB stage subsystems) and initiates S-IVB flight functions by supplying or removing power from the appropriate equipment. Sequence circuits perform logical gating of inputs necessary for sequencing control with as few timed commands from the IU as possible. It controls only those functions established as sequencing events.

PRESSURE SWITCHES

Calibratable pressure switches (calips) are used on the S-IVB stage to perform various control functions. As examples:

1. LH_2 Tank System
   a. Pressurization, ground fill, valve control.
   b. LH_2 tank pressure control backup.
2. Lox Tank System
   a. Ground fill, valve control, pressurization.
   b. Lox tank pressure control backup.
3. Pneumatic Power System
   a. Regulator backup.
   b. Engine purge.
   c. Lox chilldown pump container purge.

These pressure switches are located in the aft thrust structure area and in the forward interstage as requirements dictate.

Pressure switches are located in either the aft section (thrust structure) or the forward section (interstage) depending upon the pressure control system in which the switch is used.

The calips pressure switch employs two pressure ports, each isolated from the other. The test port provides for remote checkout without disconnecting or contaminating the primary pressure system. The test pressure settings are calibrated during manufacture of the switch to provide an accurate indication of the system pressure settings.

Calips pressure switches utilize a single Belle-

S-IVB BATTERY CHARACTERISTICS				
TYPE	Dry Charge			
MATERIAL	Zinc-silver oxide			
ELECTROLYTE	Potassium Hydroxide (KOH) in pure water.			
CELLS	20 with taps for selecting 18 or 19 to reduce output voltage as required.			
NOMINAL VOLTAGE OUTPUT	1.5 vdc per cell 28 ± 2 vdc per 18 to 20 cell group Aft Battery No. 2 is made up of two regular 28 ± 2 vdc batteries and has an output of 56 ± 4 vdc			
	FOREWARD NO. 1	FOREWARD NO. 2	AFT NO. 1	AFT NO. 2
CURRENT RATING	179 AH	12.2 AH	179 AH	49.6 AH
Gross Weight (Design target weight)	90 lbs	Two Units 20 lbs ea.	90 lbs	75 lbs

Figure 6-20

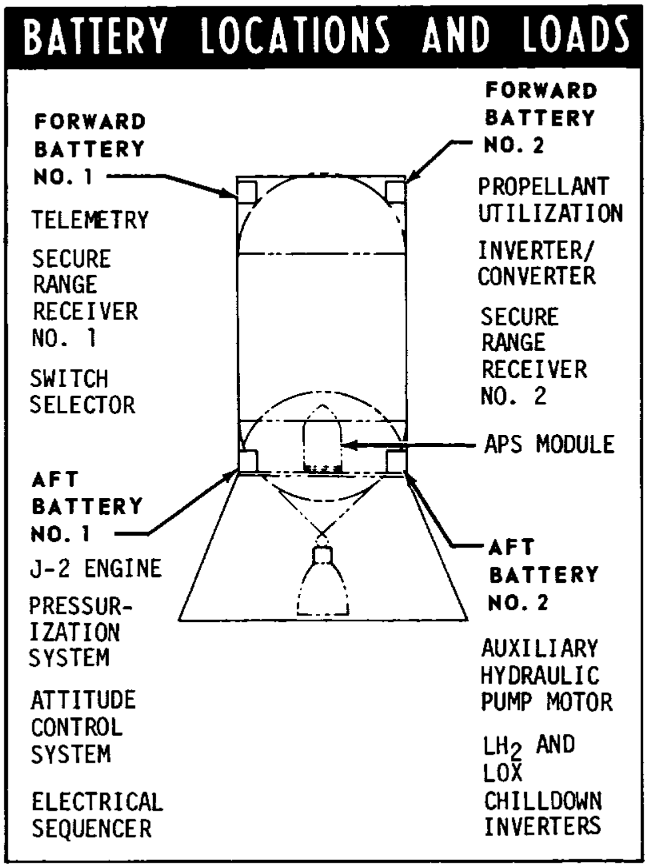


Figure 6-21

ELECTRICAL POWER DISTRIBUTION

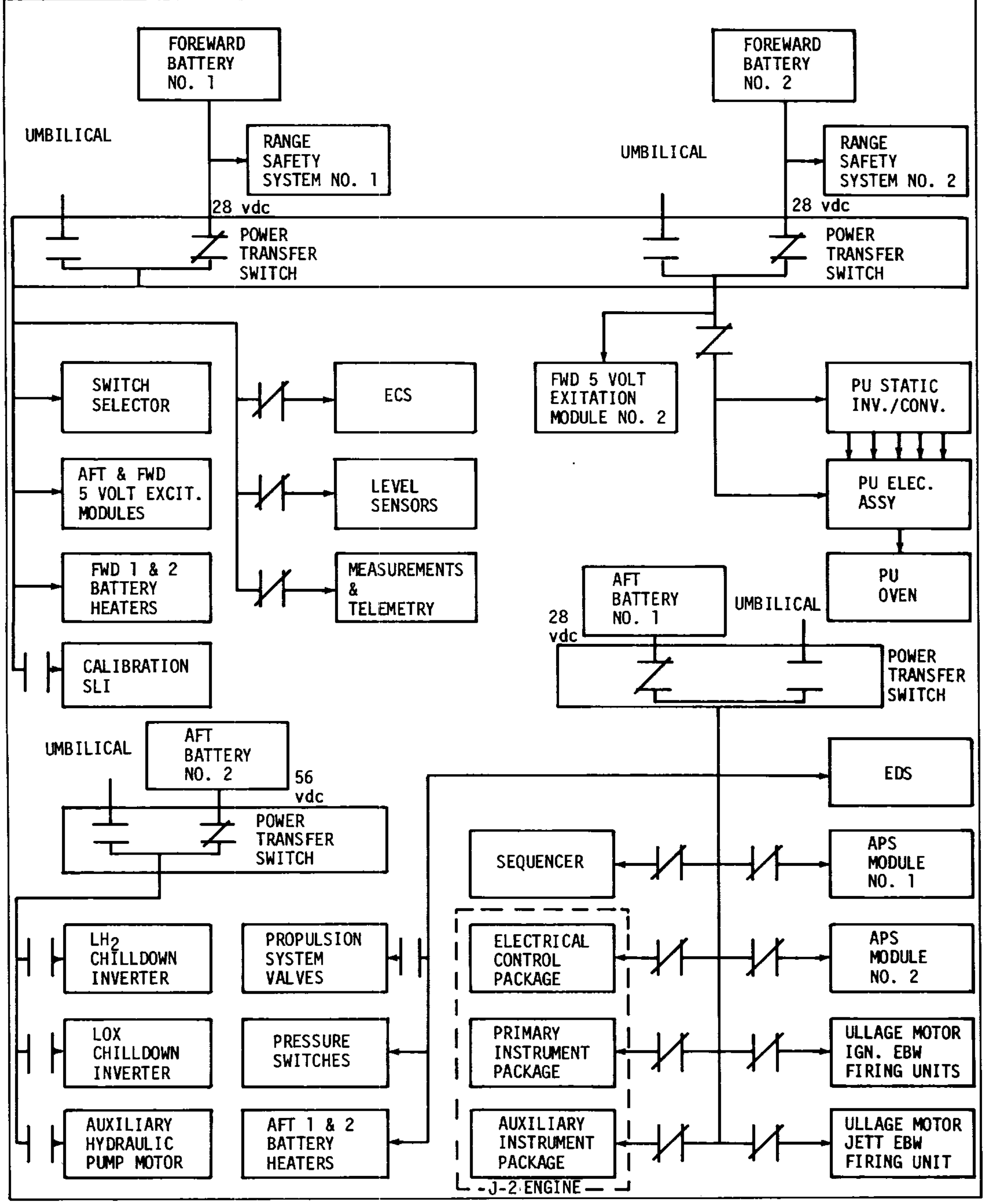


Figure 6-22

VI

ville spring which provides "snap" response to actuation or deactuation pressures. This response provides switching of 28 vdc power to relays in the stage sequencer for operation and control of propulsion system solenoid valves.

PROPELLANT UTILIZATION

The propellant utilization subsystem (figure 6-23) provides a means of controlling the propellant mass ratio. It consists of lox and LH_2 tank mass probes, a propellant utilization valve, and an electronic assembly. These components monitor the propellant and maintain command control. In addition, propellant depletion sensors are provided in each tank to initiate engine cutoff when liquid level is at the minimum allowable for engine operation. Overfill and fast-fill emergency point level sensors are also provided in each tank to help meet stage safety requirements during loading.

The propellant utilization subsystem is additionally employed during propellant loading to provide an accurate indication of the propellant masses loaded. The electronic assembly can be biased to compensate for LH_2 boiloff during flight and orbit. During powered flight the propellant utilization subsystem provides PU valve position and onboard propellant data for telemetry.

The propellant utilization valve is a rotary valve which controls the quantity of lox flowing to the engine. Operation of this valve varies the amount of lox bypassed from the lox pump discharge back to the lox pump inlet. The excursion effect caused by varying the mixture ratio is shown graphically in figure 5-11.

PROPELLANT UTILIZATION SUBSYSTEM OPERATION

The subsystem provides for control of propellant consumption during engine burn periods by controlling the usage ratio of the lox to assure minimum propellant residuals at termination of flight. The subsystem is capable of reducing the stage propellant residual to less than 0.25 percent of the total propellant mass loaded (or a three sigma of 575 lbs), under the assumption that the engine burns until a propellant depletion signal is received. The propellant utilization subsystem is activated during flight for first and second burn, beginning shortly after engine start, and is deactivated after engine cutoff. During second burn, normal cutoff will be initiated by either the IU or by a signal from the tank depletion sensors. During first burn the desired propellant consumption is maintained by controlling the engine mixture ratio to approximately 5.5 to 1.0. Measured lox mass, as provided by the lox mass probe, is compared with the product of 5.5 times the measured LH_2 mass. The difference between lox mass and 5.0 times the LH_2 mass then generates a corrective command to proportionately control the lox pump bypass flowrate, and the engine mixture ratio. A change in the mixture ratio during engine operation produces a change in the lox tank mass with respect to the LH_2 tank mass.

For second burn engine start the PU valve is commanded to the full open position for 12 seconds. At that time the system is enabled to operate around a referenced mixture ratio of 5.0 to 1 for the remaining powered flight.

The propellant utilization subsystem provides an accurate indication of propellant mass for loading. Loading commands are initiated and terminated by monitoring the propellant utilization output signals.

INSTRUMENTATION

The S-IVB stage instrumentation monitors functional operations of stage systems. Before liftoff, measurements are telemetered by coaxial cable to ground support equipment. During flight, radio frequency antennae convey data to ground stations. See figure 6-24 for a block diagram of the S-IVB measurement and telemetry systems.

MEASUREMENT SYSTEM

Monitoring functional operations of stage systems is the purpose of the measurement subsystem. It acquires functional data; conditions it; and supplies it to the telemetry system for transmission to the ground stations.

Various parameters, as listed in figure 6-25, are measured by the types of transducers described in the following paragraphs:

1. Temperature transducers are of two types; the platinum wire whose resistance changes with a change in temperature, and the thermocouple which shows an output voltage increase with an increase in temperature. Individual bridges and dc voltage amplifiers are employed for each temperature measurement. In cases where the temperature bridge connects directly to the remote analog submultiplexer (RASM), the external dc voltage amplifier is omitted.

2. Pressure Transducers. Two basic types of pressure transducers are used, the conventional potentiometer (pot) type (5000 ohms) and the strain gage type. The pot type transducers are used on 12 samples per second (S/S) measurements and where the temperature environment is no colder than -100°F. The strain gage transducers are used on all 120 S/S measurements and where it is established that temperatures at the transducer will be colder than -100°F. All strain gage types have a high and low simulated calibration capability.

3. Flowmeters. Conventional turbine type flowmeters with frequency to direct current (dc) (0 to 5 volts) converters are supplied for flow measurements.

4. Positions of actuators and metering valves are measured by conventional potentiometers.

5. Events are points in time where a switch opens or closes and are not conditioned but fed directly to multiplexers.

6. Liquid level sensors use a capacitive bridge type control unit to signal and condition the measurement for each data channel.

7. Voltages, currents, and frequencies, are conditioned to 0 to 5 volt analog signals.

8. Miscellaneous measurements are conditioned to 0 to 5 volts.

9. Speed. Magnetic pickups with frequency to dc converters condition the lox and LH_2 turbopump speeds to 0 to 5 volt analog signals.

An S-IVB stage measurement summary is given in figure 6-24.

TELEMETRY SYSTEM

The telemetry system, installed in the S-IVB stage, is a pulse-code-modulated (PCM) digital data acquisition system (DDAS). The data system consists of a basic PCM frequency-modulated (FM) system operating at 72 kilobits per second and utilizing a 10-bit encoded word.

The PCM/FM system consists of two model 270

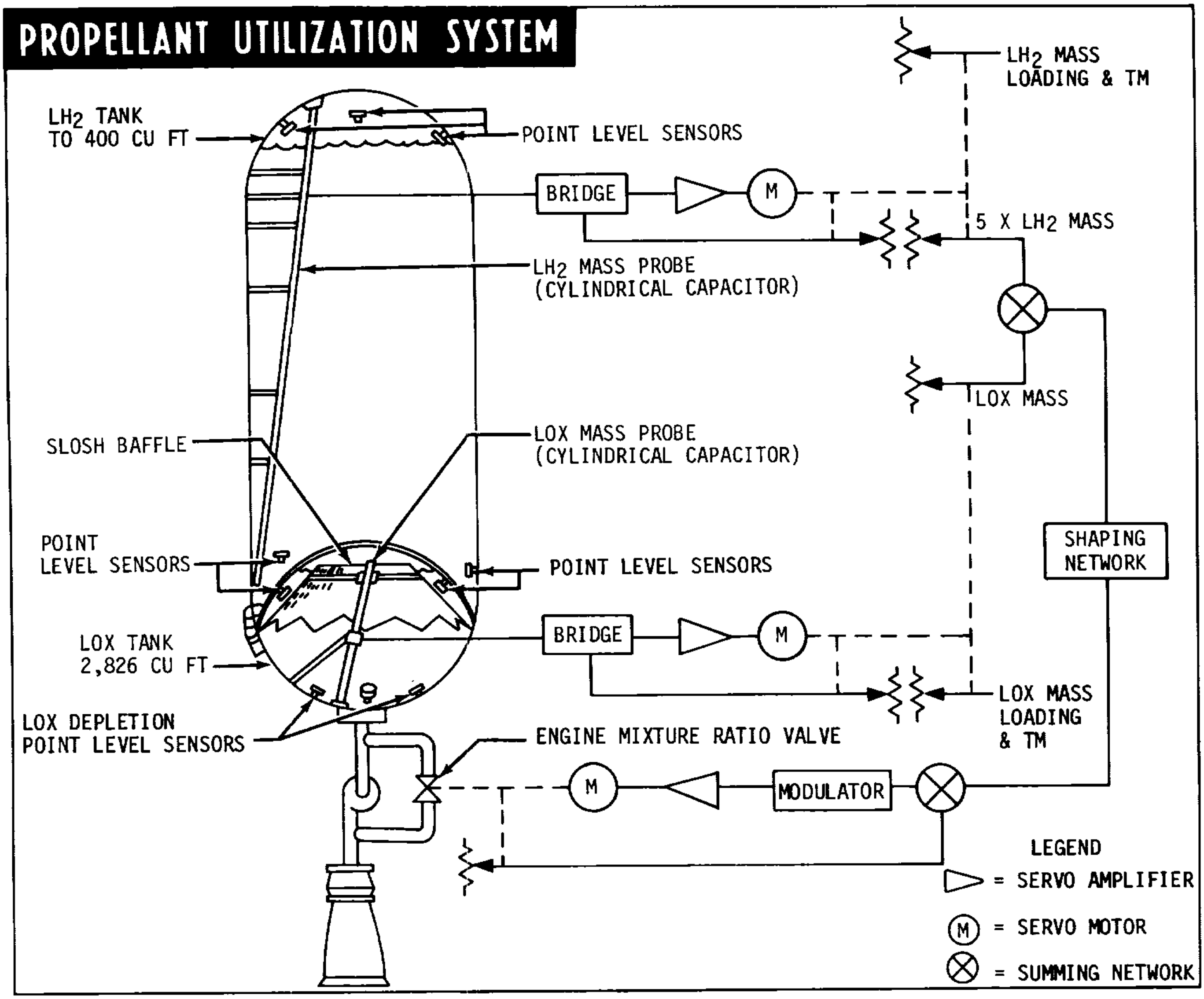


Figure 6-23

VI S-IVB

multiplexers, one remote analog submultiplexer (RASM), and one remote digital submultiplexer (RDSM) operating into a digital data acquisition subsystem (PCM/DDAS), model 301. The output of the DDAS is directly inputed to the model II PCM RF assembly. The DDAS output is indirectly routed to the IU by way of the PCM RF assembly for redundant transmission at S-band frequencies.

One Model 270 multiplexer will receive synchronization from, and input data to, the S-IVB stage DDAS. The second Model 270 multiplexer will receive synchronization from, and input data to, the IU DDAS. The Model 270 multiplexer receiving synchronization from the IU DDAS will have up to six of its prime channels allocated to a remote analog submultiplexer.

Analog parameters designated as "Flight Control Data" (FCD) are fed in parallel to both Model 270 multiplexers. FCD bi-level or discrete functions are placed on that Model 270 multiplexer receiving synchronization from the IU DDAS (via summing network) and also input to the remote digital submultiplexer or DDAS directly. Analog functions may be placed upon either Model 270 multiplexer. "Low Level" measurements located aft are carried by the remote analog multiplexer while those forward require individual amplifiers. Bi-level measurements will normally use either the RDSM aft or the DDAS forward.

RADIO-FREQUENCY SYSTEM

The RF subsystem consists of a PCM RF assembly, bi-directional coupler, RF detectors, dc amplifiers, coaxial switch, dummy load, RF power divider, and associated cabling (figure 6-26).

Omnidirectional antenna pattern coverage is provided by the folded-sleeve dipoles. One is located 11° from fin plane 2 toward fin plane 1; the other is located 11° from fin plane 4 toward fin plane 3. The electrical phasing between antennae is 180° thereby reducing nulls off the nose and tail. The effective radiating

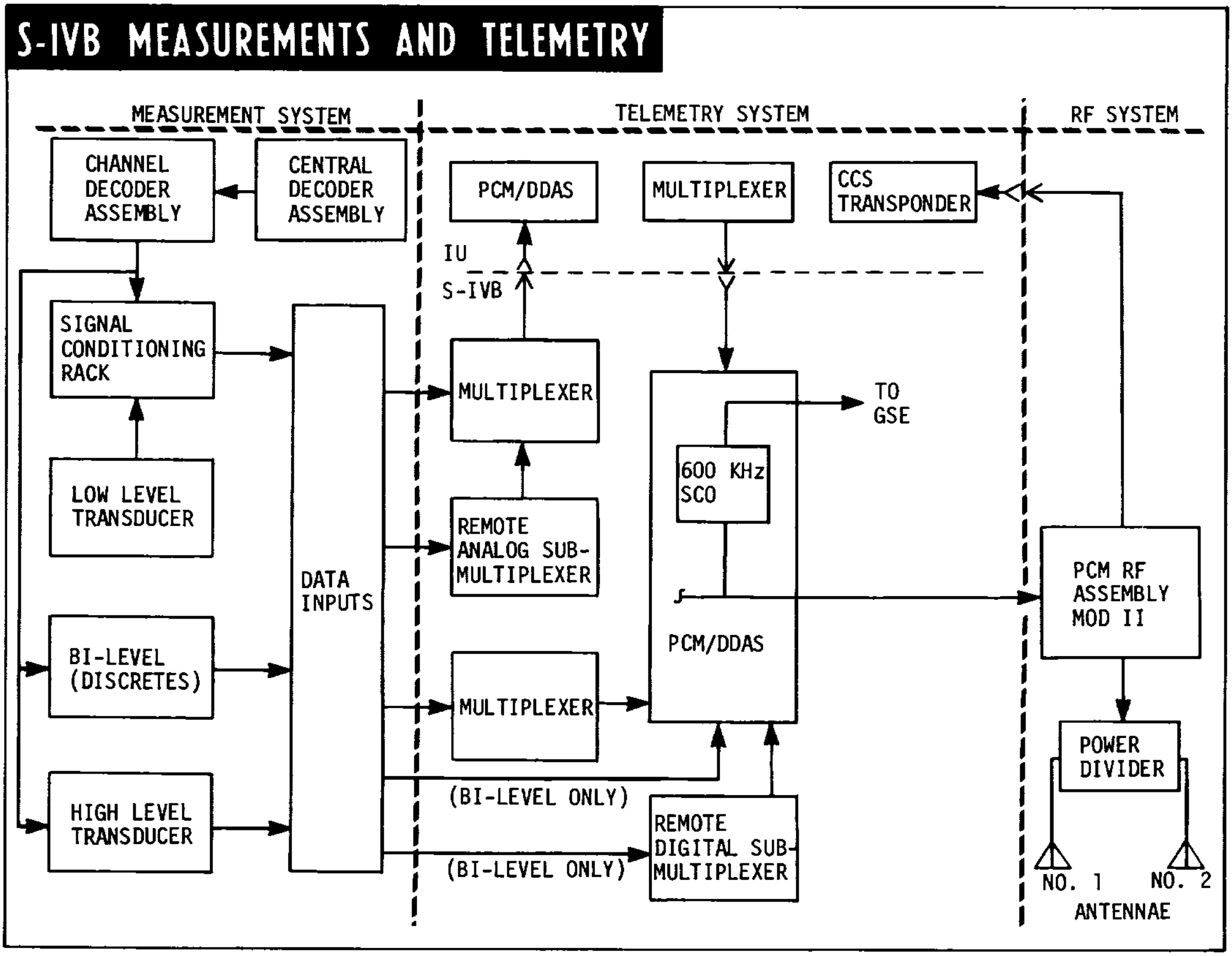


Figure 6-24

power of the system is 20 watts nominal (+43 dbm) and 16 watts minimum.

The antennae are designed to operate with a 50-ohm impedance and a voltage standing wave ratio (VSWR) no greater than 1.5:1 over the 225 to 260 MHz band. Transmitted and reflected power measurements are routed through the DDAS for ground checkout, by use of the bi-directional coupler to provide RF outputs without interferences. The coaxial switch provides a means of connecting a dummy load to the RF assembly for ground testing purpose.

The Model II PCM RF assembly uses the PCM pulse train to modulate the VHF carrier (258.5 MHz) in the transmitter. The transmitter output is amplified in the RF power amplifier. The radiation pattern coverage of the two antenna is designed to provide coverage to at least two ground stations simultaneously during flight.

ORDNANCE

The S-IVB ordnance systems include the separation, ullage rocket, ullage rocket jettison, and propellant dispersion (flight termination) systems.

SEPARATION SYSTEM

The third plane separation system for S-II/S-IVB is located at the top of the S-II/S-IVB interstage (figure 5-29). The separation plane is at station 2746.5. Ordnance for the third plane separation consists of two exploding bridgewire (EBW) firing units, two EBW detonators, one detonator block assembly and a detonating fuse assembly (figure 6-27). The EBW firing units are on the S-II/S-IVB interstage slightly below the third separation plane. The leads of the EBW firing units are attached to the EBW detonators which are installed in the detonator block assembly. The detonator block assembly is mounted just inside the skin of the vehicle and the ends of the detonating fuse assembly are installed within the detonator block assembly. The detonating fuse assembly is mounted around the periphery of the vehicle beneath the tension strap.

The two EBW firing units for third plane separation provide redundant signal paths for initiation of the detonating fuse assembly. The function of the ordnance train is similar to that described in the separation system discussion in

STAGE MEASUREMENTS

TYPE	QTY
Temperature	72
Pressure	70
Flowrate	4
Position	8
Discrete Signals	70
Liquid Level	7
Voltage, Current, Frequency	38
RPM	2
Miscellaneous	9
TOTAL	280

Figure 6-25

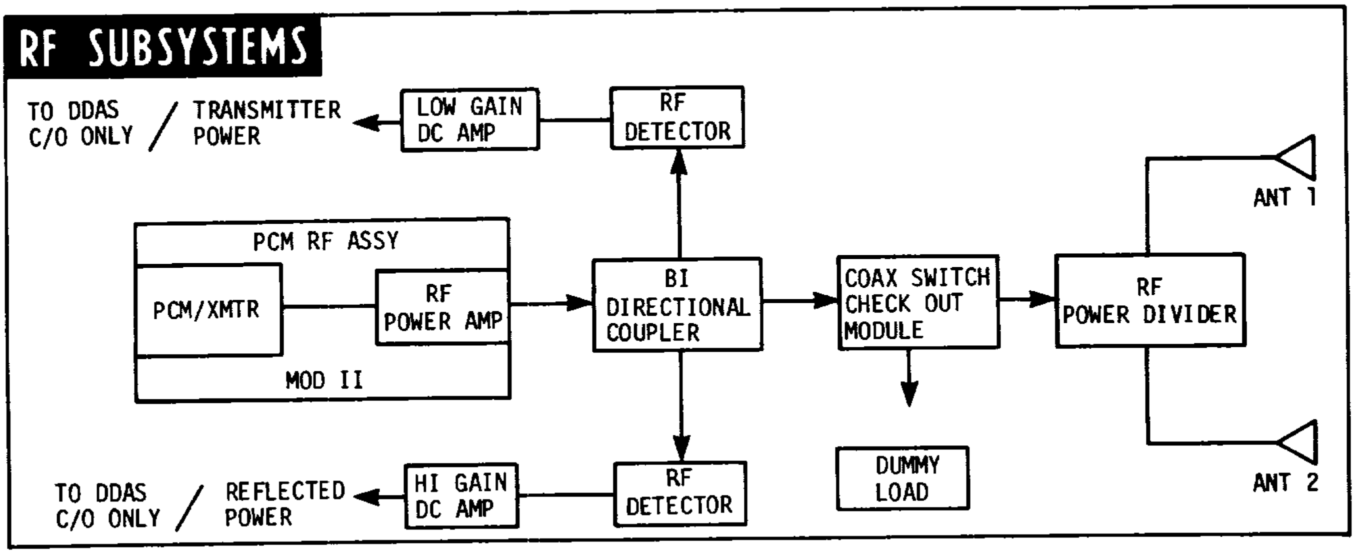


Figure 6-26

Section V. Detonation of the detonating fuse assembly severs the tension strap attaching the S-II/S-IVB interstage at station 2746.5 (figure 5-29).

A sequence of events for S-IC/S-II/S-IVB separations and a block diagram of the separation systems is contained on figure 5-30.

At the time of separation, four retrorocket motors mounted on the interstage structure below the separation plane fire to decelerate the S-II stage. For information on the S-II retrorocket system, refer to Section V.

ULLAGE ROCKET SYSTEM

To provide propellant settling and thus ensure stable flow of lox and LH_2 during J-2 engine start, the S-IVB stage requires a small acceleration. This acceleration is provided by two ullage rockets.

The S-IVB ullage rocket system (figure 6-27) consists of two EBW firing units, two EBW detonators, two confined detonating fuse (CDF) manifolds, nine CDF assemblies, two separation blocks, four CDF initiators, and two ullage rockets. The EBW firing units, EBW detonators, and CDF manifolds are mounted on the S-IVB aft skirt. The CDF assemblies connect the manifolds to the separation blocks and then to the CDF initiators. The rockets are within fairings mounted diametrically opposite each other on the S-IVB aft skirt. The rockets are canted outward from the vehicle to reduce effects of exhaust impingement and to reduce the resulting moment if one rocket fails.

A separation block is used between the stage and each ullage rocket to allow jettison and maintain CDF continuity. The separation block, an inert item, is located on the skin of the S-IVB aft skirt under the ullage rocket fairing. Each block consists of two machined pieces of aluminum. The upper piece holds the ends of the CDF assemblies to the initiators while the lower piece holds the CDF assemblies from the manifolds. The separation block forms a housing or connector that holds the CDF assembly ends together to ensure propagation and to contain the detonation of the connection. At jettison the block slips apart with the lower portion remaining on the stage and the upper portion falling away with the rocket and fairing.

Each ullage rocket has a single grain, five point star configuration, internal burning, polymerized solid propellant that is case bonded in a 4135 steel case. The propellant weighs approximately 58.8 pounds and burns for 3.87 seconds, developing a thrust of 3,390 pounds (175,000 feet, 70°F).

The firing sequence begins with the arming of the EBW firing units by charging the storage capacitors to 2300 volts. As the S-II engine shutdown, the EBW units receive a trigger signal which discharges the storage capacitors, releasing high energy pulses to the EBW detonators and thereby exploding the bridgewires. The resulting detonations propagate through the CDF manifolds, CDF assemblies, separation blocks and to the CDF initiators which cause the ullage rockets to ignite. A crossover CDF assembly between CDF manifolds provides redundancy and added system reliability.

ULLAGE ROCKET JETTISON SYSTEM

To reduce weight, the ullage rockets and their fairings are jettisoned after J-2 engine start. The system, located on the S-IVB aft skirt, uses two EBW firing units, two EBW detonators, one detonator block, two CDF assemblies, four frangible nuts, and two spring-loaded jettison assemblies (figure 6-27).

The EBW firing units are armed by charging their storage capacitors to 2300 volts about five seconds after the S-IVB ullage rockets have stopped firing. A trigger signal releases the high voltage pulse to explode the bridgewire in the EBW detonator. Either detonator will detonate both CDF assemblies (figure 6-27) through the detonator block. The detonation propagates through the CDF assemblies to detonate and fracture the frangible nuts. This frees the bolts that secure the ullage rocket and fairing assemblies to the aft skirt. The spring-loaded jettison assemblies propel the spent rocket and fairing assemblies away from the vehicle.

PROPELLANT DISPERSION SYSTEM

The S-IVB propellant dispersion system (PDS) provides for termination of vehicle flight during the S-IVB first engine firing boost phase if the vehicle flight path varies beyond its prescribed limits or if continuation of vehicle flight creates a safety hazard. The S-IVB PDS may be safed after the launch escape tower is jettisoned. The system is installed in compliance with Air Force Eastern Test Range (AFETR) Regulation 127-9 and AFETR Safety Manual 127-1.

The S-IVB PDS is a dual channel, parallel redundant system composed of two segments (figure 6-27). The radio frequency segment receives, decodes and controls the propellant dispersion commands. The ordnance train segment consists of two EBW firing units, two EBW detonators, one safety and arming (S&A) device (shared by both channels), seven CDF assemblies, two CDF tees, and three linear shaped charge (LSC) assemblies.

Should emergency termination become necessary, two coded messages are transmitted to the launch vehicle by the range safety officer. The first command arms the EBW firing units and initiates S-IVB stage engine cutoff. The second command, which is delayed to permit charging of the EBW firing units, discharges the storage capacitors across the exploding bridgewires in the EBW detonators mounted on the S&A device. The resulting explosive wave propagates through the S&A device inserts and through the remainder of the ordnance train to sever the LH_2 and lox tanks.

A description of the S&A device is included in the PDS discussion in Section IV.

The linear shaped charges for the LH_2 and lox tanks are RDX loaded at 150 grains per foot. Two assemblies are used to cut two 20.2-foot long parallel openings in the side of the LH_2 tank. One assembly is used to cut a 47-inch diameter hole in the bottom of the lox tank.

Following S-IVB engine cutoff at orbit insertion, the PDS is electrically safed by ground command.

MAJOR DIFFERENCES BETWEEN SATURN V S-IVB-503N & S-IVB-504N STAGES

1 ▶ *S-IVB dry stage weight increased from 26,421 to 31,571 pounds.*

2 ▶ *S-IVB gross stage weight at liftoff increased from 263,204 to 265,559 pounds.*

3 ▶ *Stage measurements evolve from R&D/Operational to operational status.*

4 ▶ *Instrumentation measurement reduced from 342 to 280.*

MAJOR ORDNANCE COMPONENTS

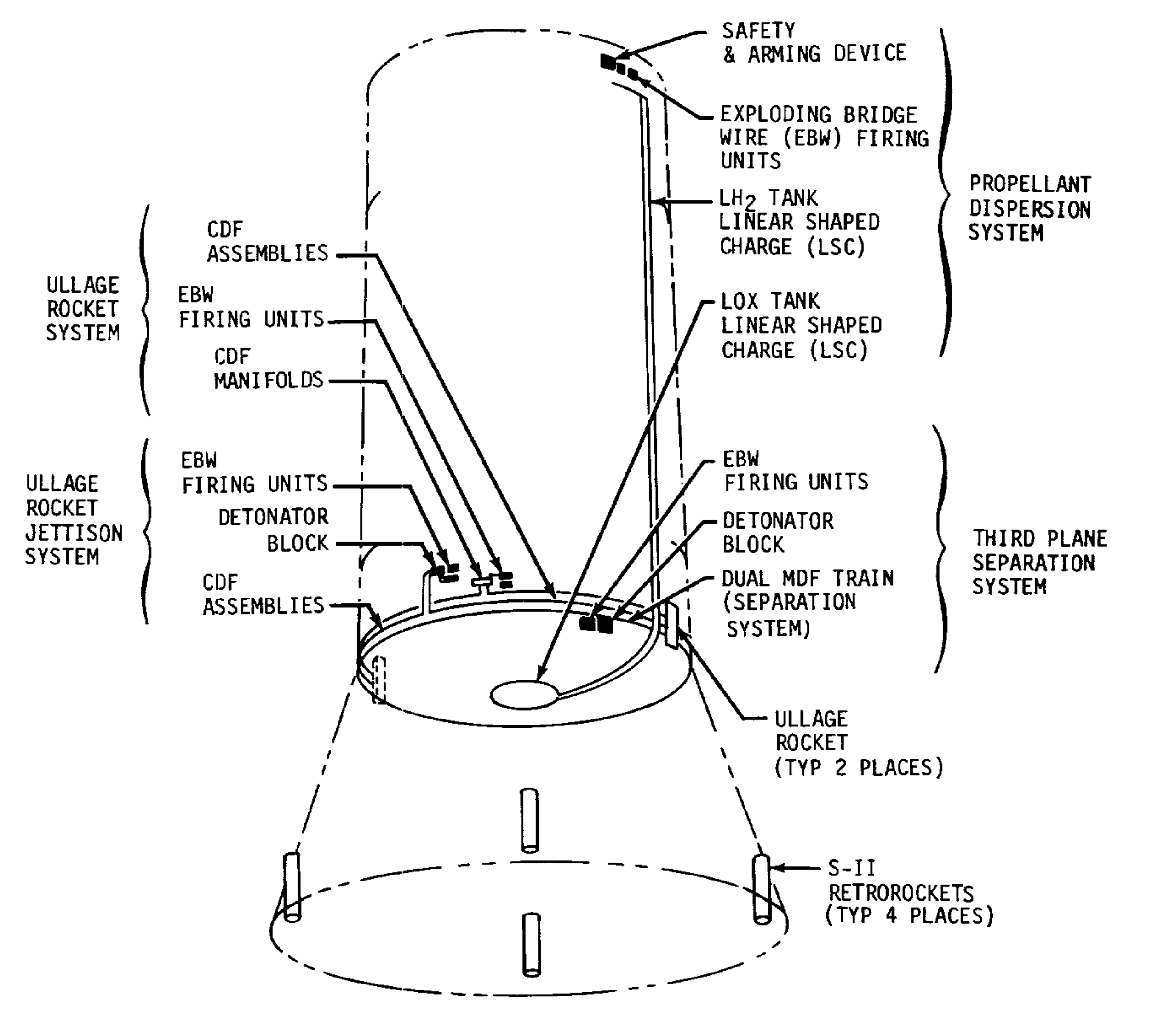


Figure 6-27 (Sheet 1 of 2)

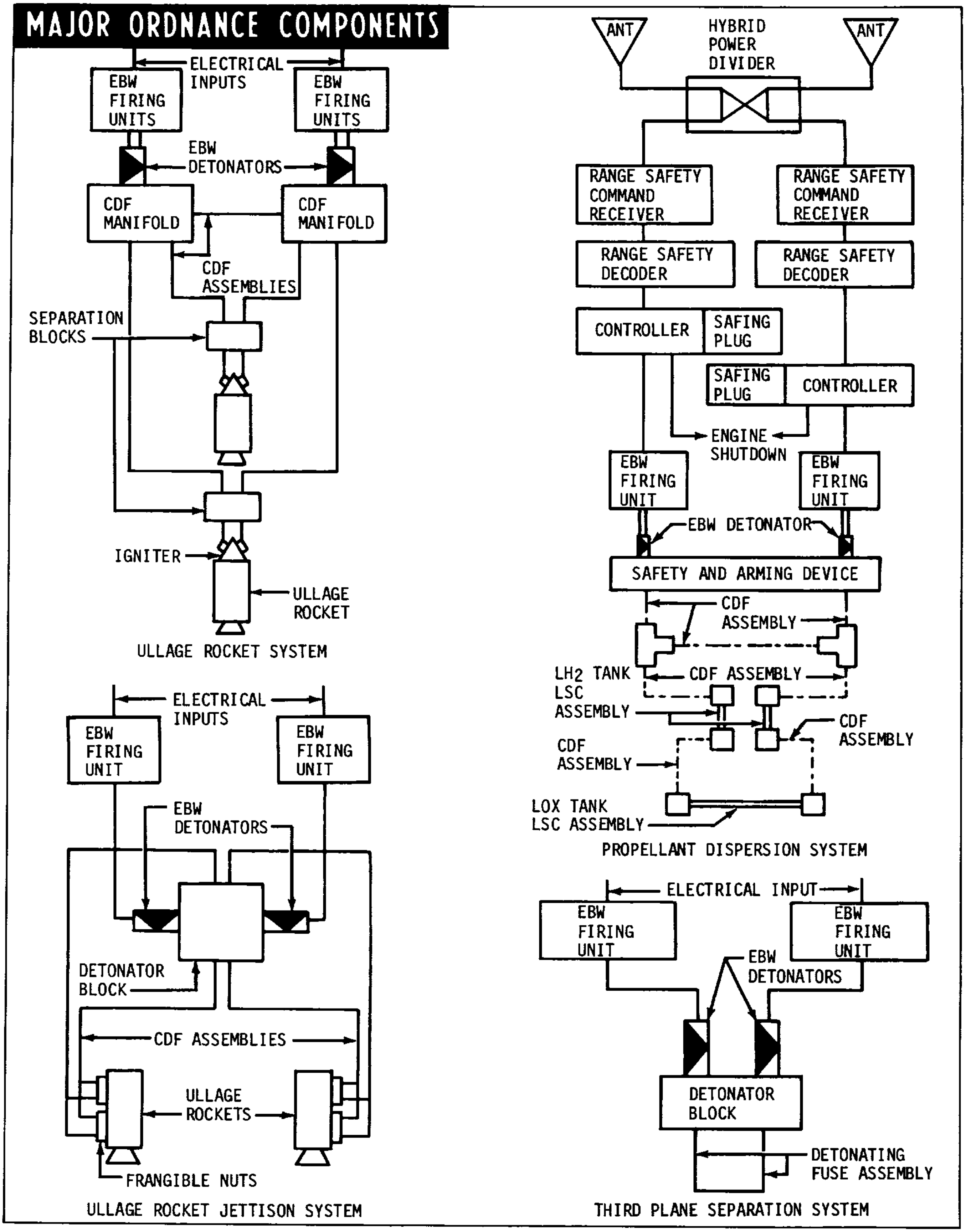


Figure 6-27 (Sheet 2 of 2)

SECTION VII

INSTRUMENT UNIT

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INTRODUCTION

The Instrument Unit (IU) is a cylindrical structure installed on top of the S-IVB stage (see figure 7-1). The IU contains the guidance, navigation and control equipment which will guide the vehicle through its earth orbits and subsequently into its mission trajectory. In addition, it contains telemetry, communications, tracking, and crew safety systems along with their supporting electrical power and environmental control systems.

This section of the Flight Manual contains a description of the physical characteristics and functional operation for the equipment installed in the IU.

SATURN INSTRUMENT UNIT

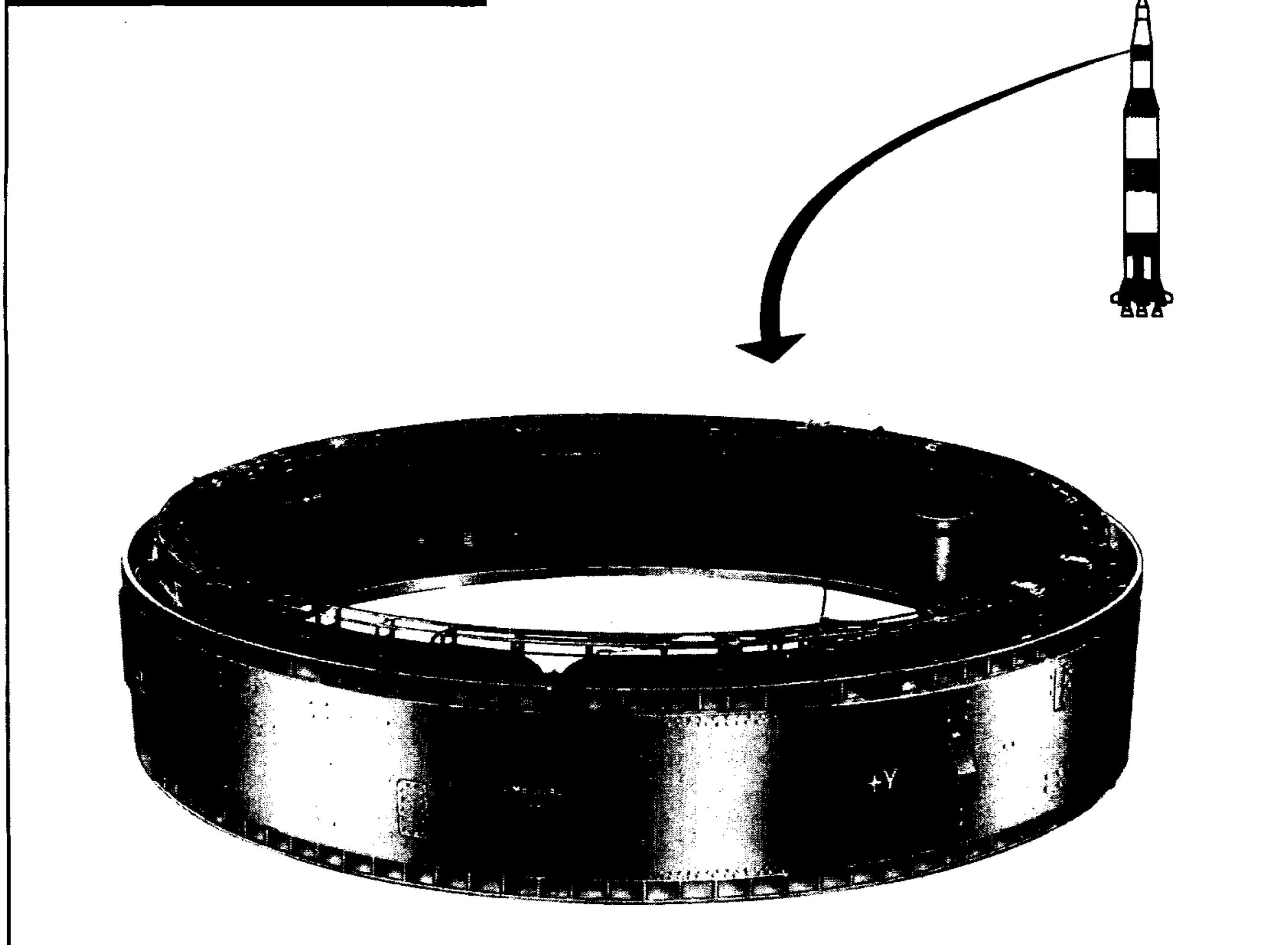


Figure 7-1

STRUCTURE

The basic IU structure is a short cylinder fabricated of an aluminum alloy honeycomb sandwich material (see figure 7-2). The structure is fabricated from three honeycomb sandwich segments of equal length. The top and bottom edges are made from extruded aluminum channels bonded to the honeycomb sandwich. This type of construction was selected for its high strength to weight ratio, acoustical insulation and thermal conductivity properties. The cylinder is manufactured in three 120 degree segments (figure 7-4) which are joined by splice plates into an integral structure. The three segments are the access door segment, the flight control computer segment and the ST-124-M3 segment. The access door segment has an umbilical door frame and plate, and an umbilical door in addition to the access door. The access door has the requirement to carry flight loads and still be removable at any time prior to flight.

Attached to the inner surface of the cylinder are cold plates which serve both as mounting structure and thermal conditioning units for the electrical/electronic equipment. Mounting the electrical/electronic equipment around the inner circumference of the IU leaves the center of the unit open to accommodate the convex upper tank bulkhead of the S-IVB stage and the landing gear of the Lunear Excursion Module.

Cross section "A" of figure 7-3 shows equipment mounting pads bolted and bonded to the honeycomb structure. This method is used when equipment is not mounted on thermal conditioning cold plates. The bolts are inserted through the honeycomb core and the bolt ends and nuts protrude through the outside surface. Cross section "B" shows a thermal conditioning cold plate mounting panel bolted to brackets which, in turn, are bolted on the honeycomb structure. The bolts go through the honeycomb core with the bolt heads protruding through the outer surface. Cross section "C" shows the cable tray supports bolted to inserts which are potted in the honeycomb core at the upper and lower edges of the structure.

Figure 7-4 shows the relative locations of all equipment installed in the IU.

LAMINATED HONEYCOMB SANDWICH MATERIAL

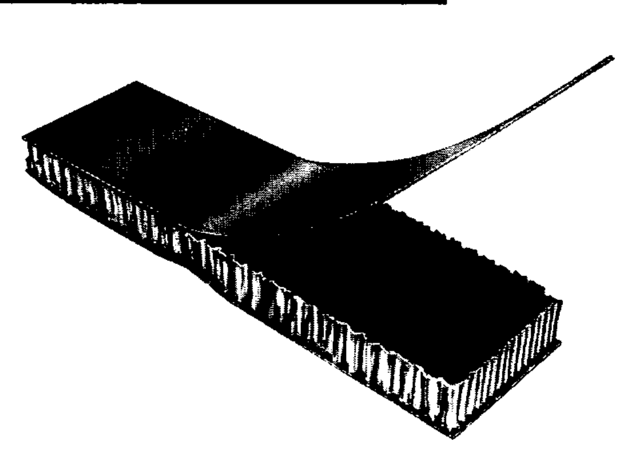


Figure 7-2

ENVIRONMENTAL CONTROL SYSTEM

The environmental control system (ECS) maintains an acceptable operating environment for the IU equipment during preflight and flight operations. The ECS is composed of the following:

1. The thermal conditioning system (TCS) which maintains a circulating coolant temperature to the electronic equipment of 59° $\pm$ 1°F.
2. Preflight purging system which maintains a supply of temperature and pressure regulated air/GN_2 in the IU/S-IVB equipment area.
3. Gas bearing supply system which furnishes GN_2 to the ST-124-M3 inertial platform gas bearings.
4. Hazardous gas detection sampling equipment which monitors the IU/S-IVB forward interstage area for the presence of hazardous vapors.

THERMAL CONDITIONING SYSTEM

Thermal conditioning panels, also called cold plates, are located in both the IU and S-IVB stage; sixteen in each stage. Each cold plate contains tapped bolt holes in a grid pattern which provides flexibility of component mounting.

The cooling fluid circulated through the TCS is a mixture of 60% methanol and 40% demineralized water by weight. Each cold plate is capable of dissipating at least 420 watts.

A functional flow diagram is shown in figure 7-5. The main coolant loop is the methanol-water mixture. Two heat exchangers are employed in the system. One is used during the preflight mode and uses a demineralized methanol-water solution as the coolant. The other is the flight mode unit and it uses ground supplied, chilled, methanol-water as the coolant and the principle of sublimation to effect the cooling.

INSTRUMENT UNIT STRUCTURAL DETAILS

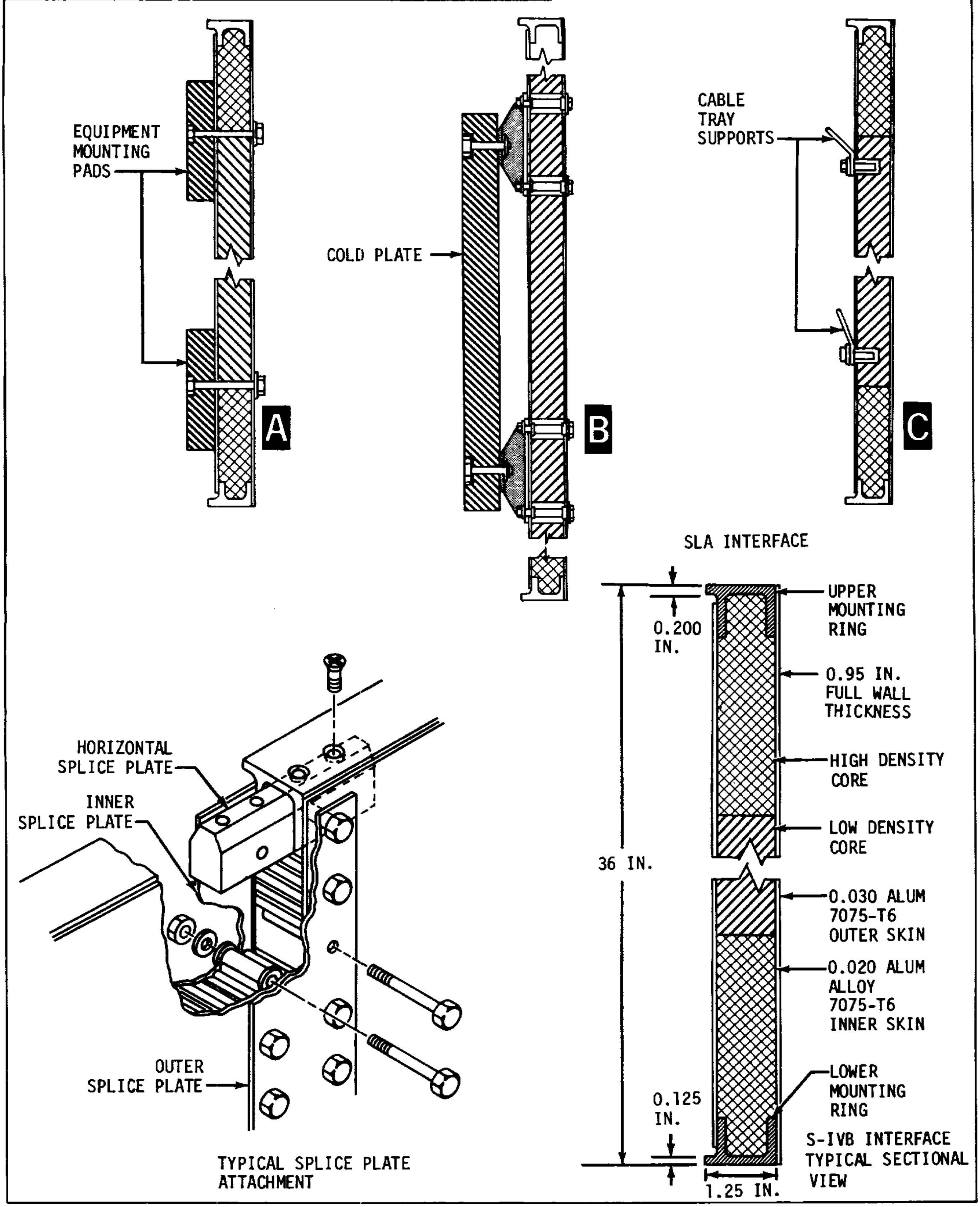


Figure 7-3

The manifold and methanol-water accumulator are filled with coolant through quick-disconnects from the GSE. The accumulator serves as a damper for pressure fluctuations and thermal expansion and also as a reservoir to replace any coolant which may be lost. There is a flexible diaphragm in the accumulator which is backed by low pressure GN_2.

During operation of the TCS the methanol-water coolant is circulated through a closed loop by electrically driven redundant pumps. The flow is from the heat exchanger past the accumulator, through the pumps, through a temperature sensor to an orifice assembly which diverts part of the coolant to the 16 cold plates in the S-IVB stage and part to the 16 cold plates, gas bearing heat exchanger, inertial platform, LVDC/LVDA and flight control computer in the IU. Return flow is through a modulating flow control assembly which regulates the amount of coolant flowing into the heat exchangers or around them. Operation of this valve is based on fluid temperature.

During the preflight mode the sublimator heat exchanger is inactive due to the high ambient pressure (one atmosphere) and a solenoid valve blocking the water flow. The preflight heat exchanger is operated from the GSE and cools the closed loop fluid.

Approximately 180 seconds after launch the water solenoid valve is opened and the sublimator heat exchanger becomes active. During the period between GSE disconnect and sublimator activation the residual cooling in the system is sufficient to preclude equipment overheating.

The water for the sublimation heat exchanger is supplied by a water accumulator which is filled manually during prelaunch preparations. The water accumulator, like the methanol-water accumulator, has a flexible diaphragm backed by low pressure GN_2. Thus, when the water control solenoid is opened water is forced from the accumulator into the sublimator. The sublimator element is a porous plate. Since the sublimator is not activated until approximately 180 seconds after launch the ambient temperature and pressure outside the porous plates is quite low. Water flows readily into the porous plates and attempts to flow through the pores. However, the water freezes when it meets the low temperature of the space environment and the resulting ice blocks the pores (see figure 7-6).

As heat is generated by equipment the temperature in the methanol-water solution increases. This heat is transferred within the sublimator to the demineralized water. As the water temperature rises it causes the ice in the pores to sublime. The vapor is vented overboard. As the heat flow decreases ice plugs are formed in the pores decreasing the water flow. Thus, the sublimator is a self regulating system.

INSTRUMENT UNIT EQUIPMENT LOCATIONS

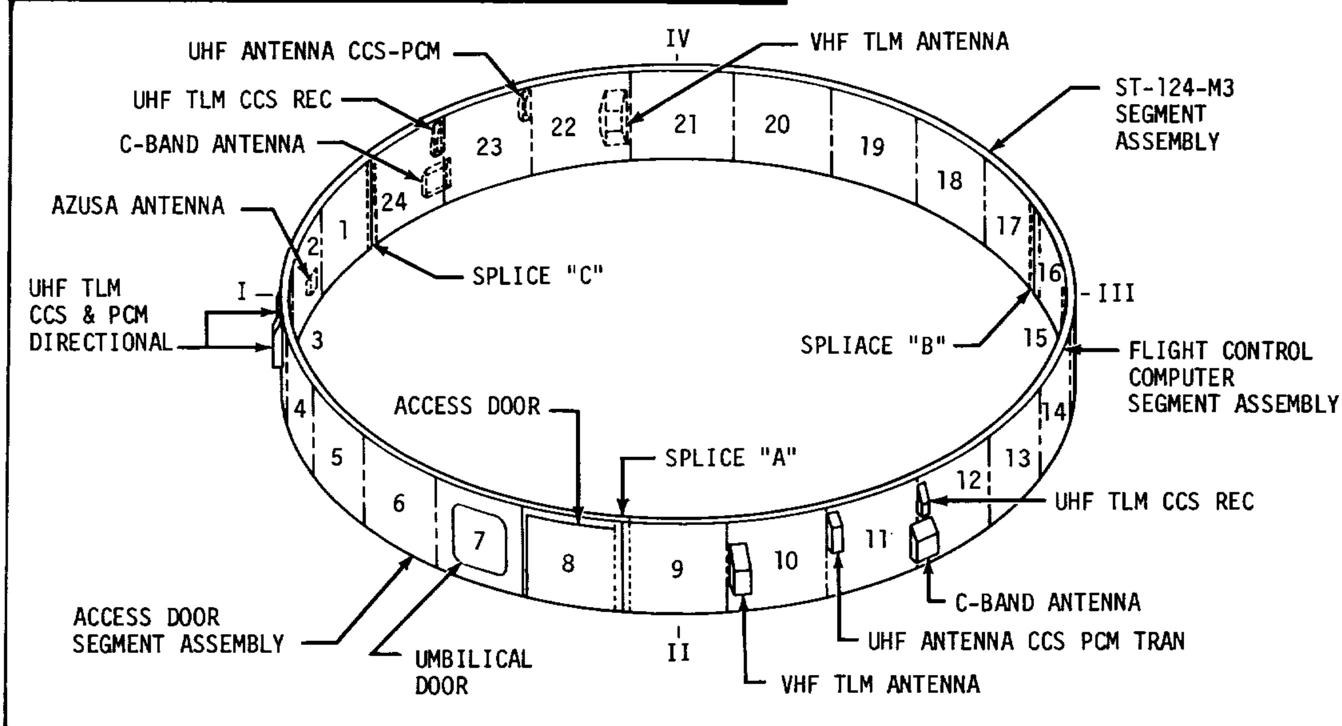


Figure 7-4 (Sheet 1 of 5)

INSTRUMENT UNIT EQUIPMENT LOCATIONS

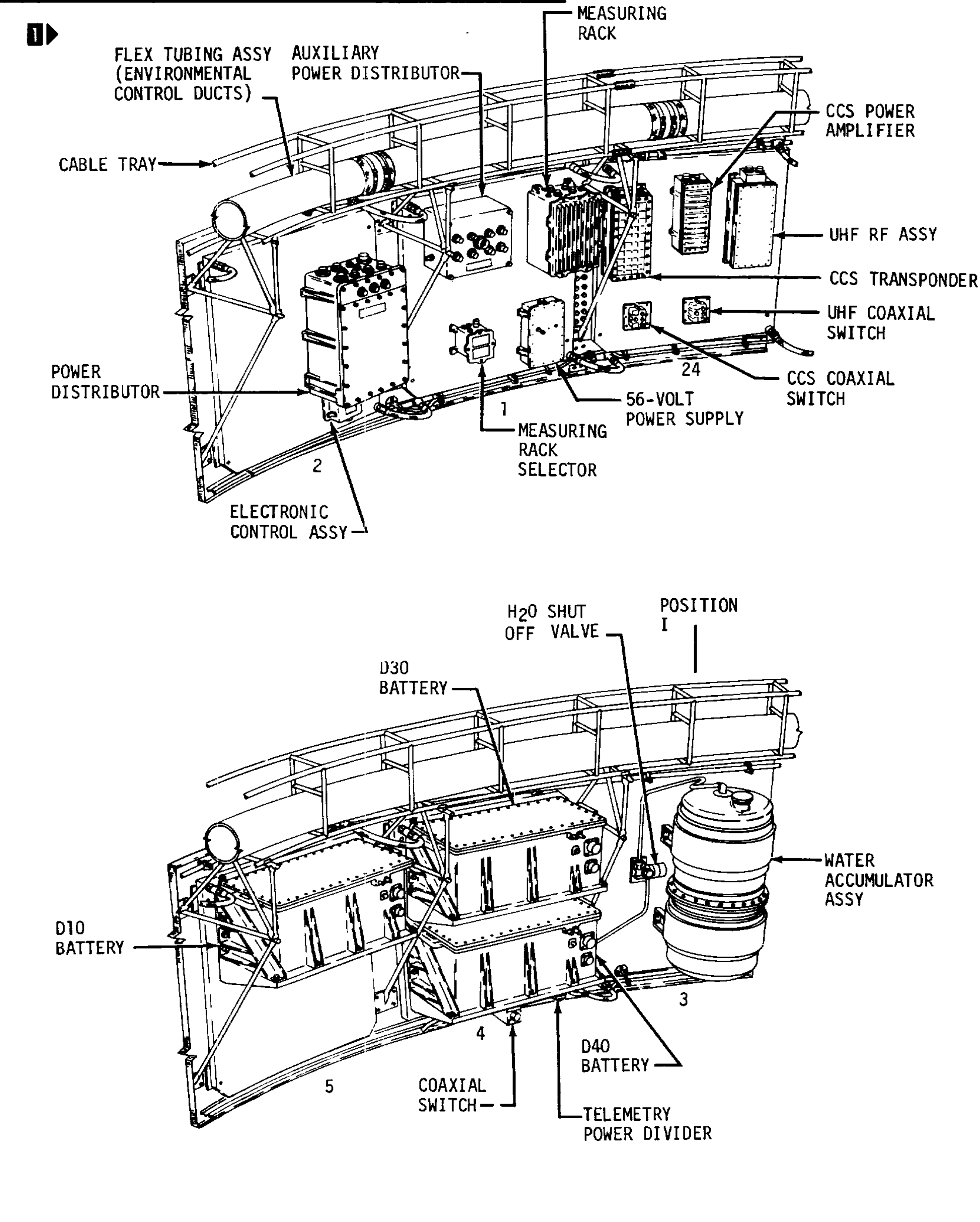
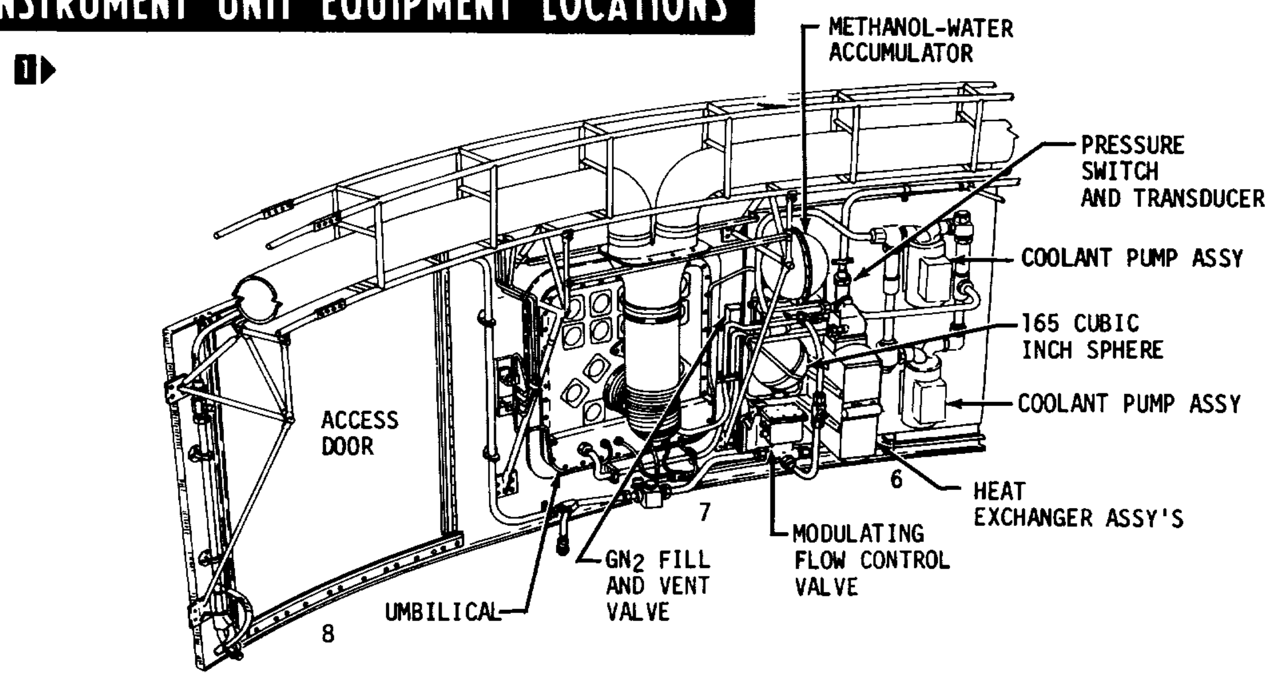


Figure 7-4 (Sheet 2 of 5)

VII
IU

INSTRUMENT UNIT EQUIPMENT LOCATIONS



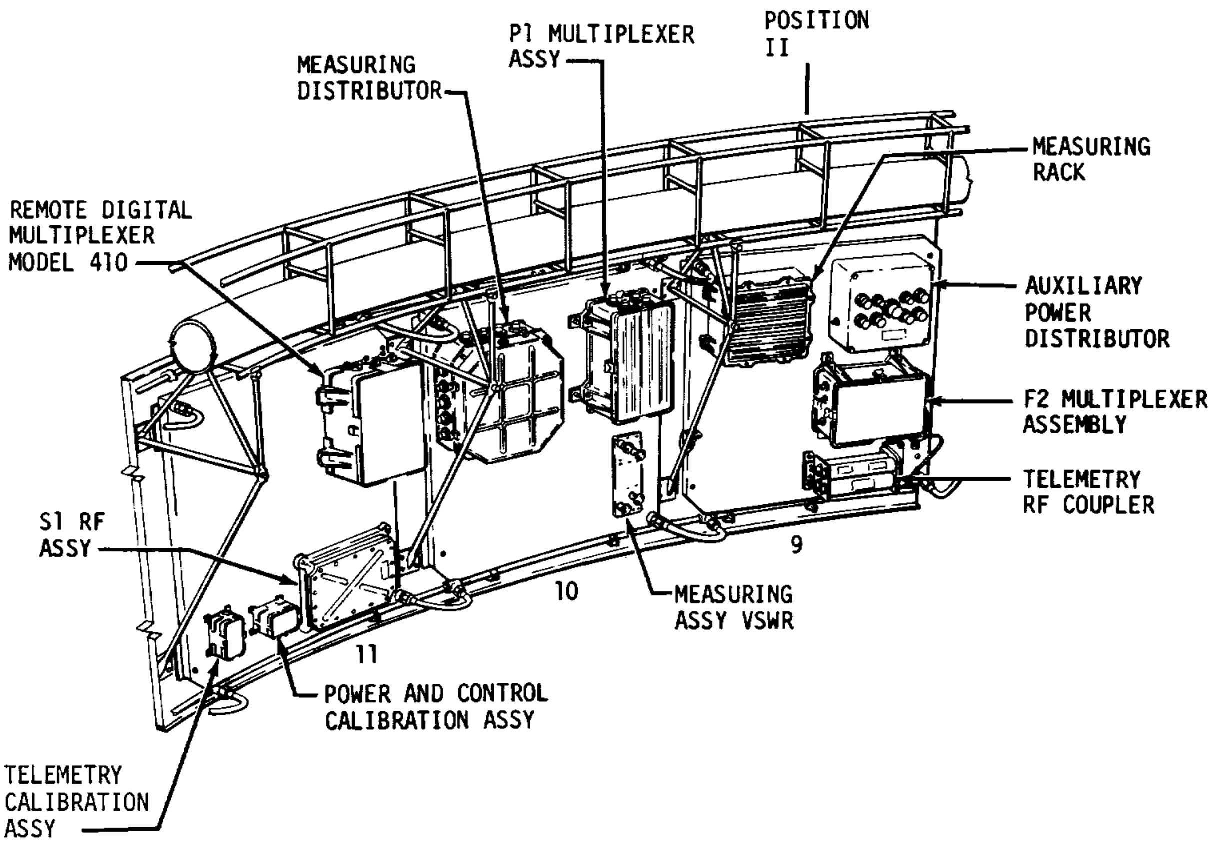
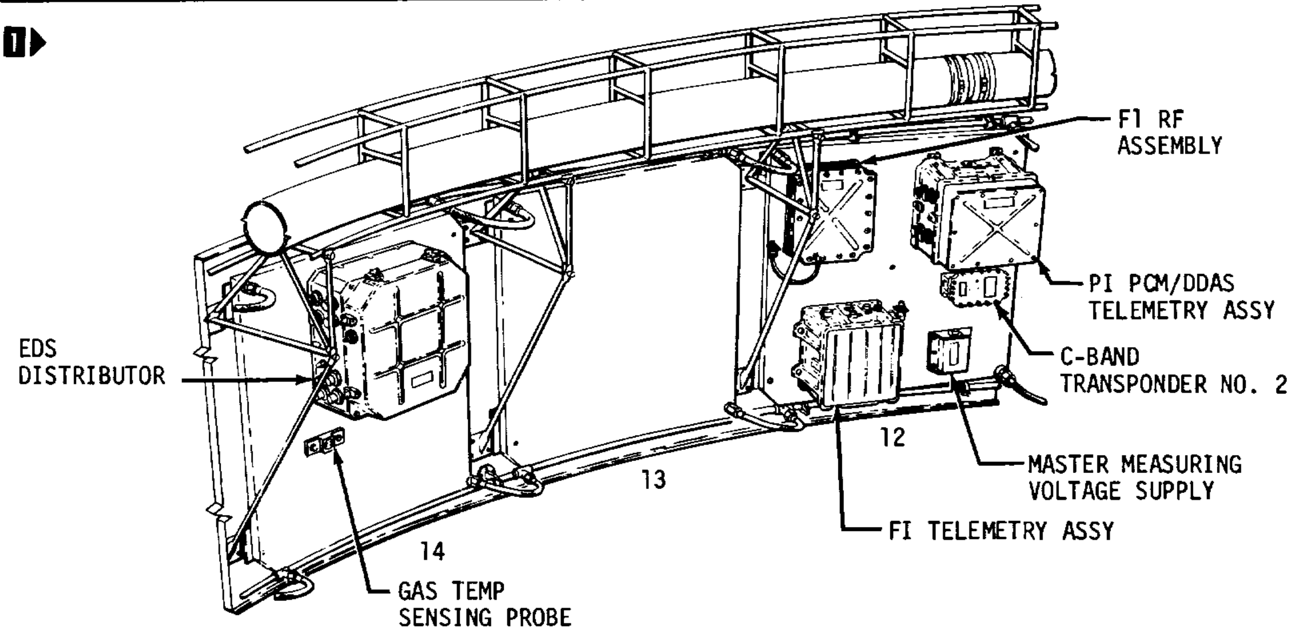


Figure 7-4 (Sheet 3 of 5)

INSTRUMENT UNIT EQUIPMENT LOCATIONS



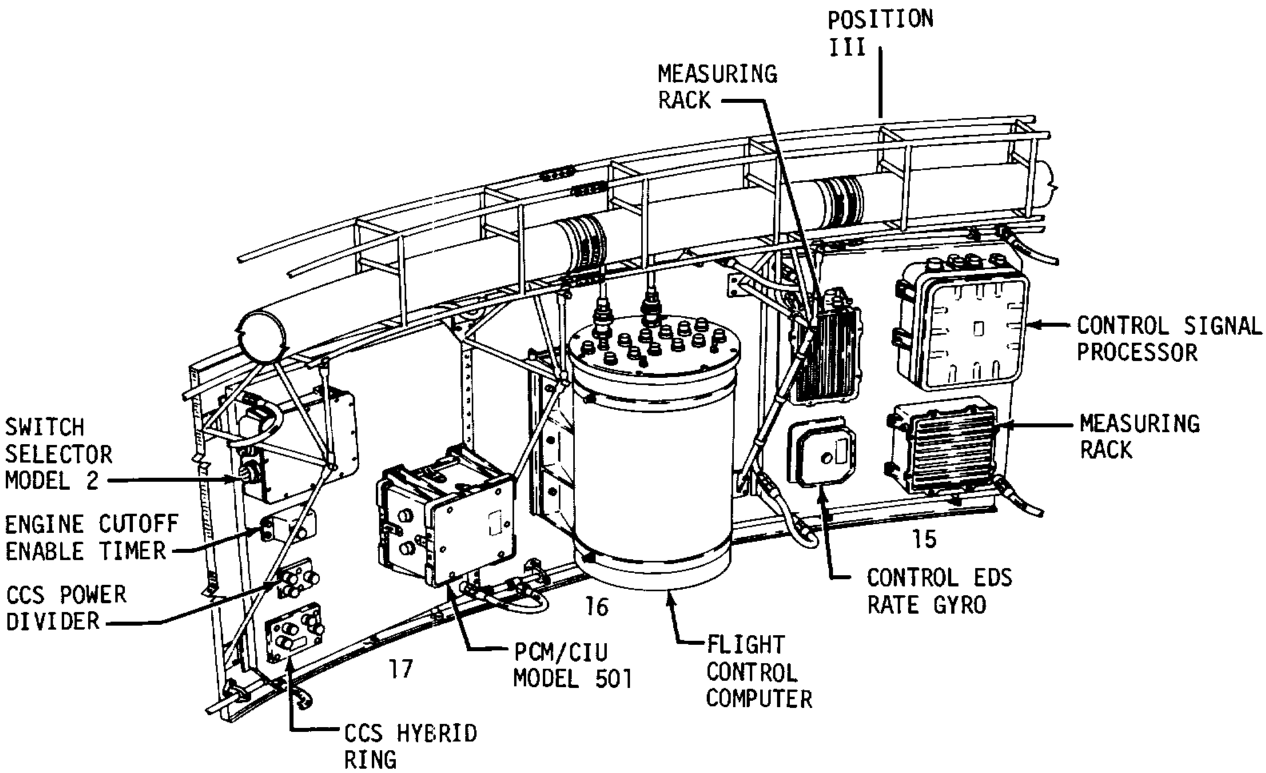
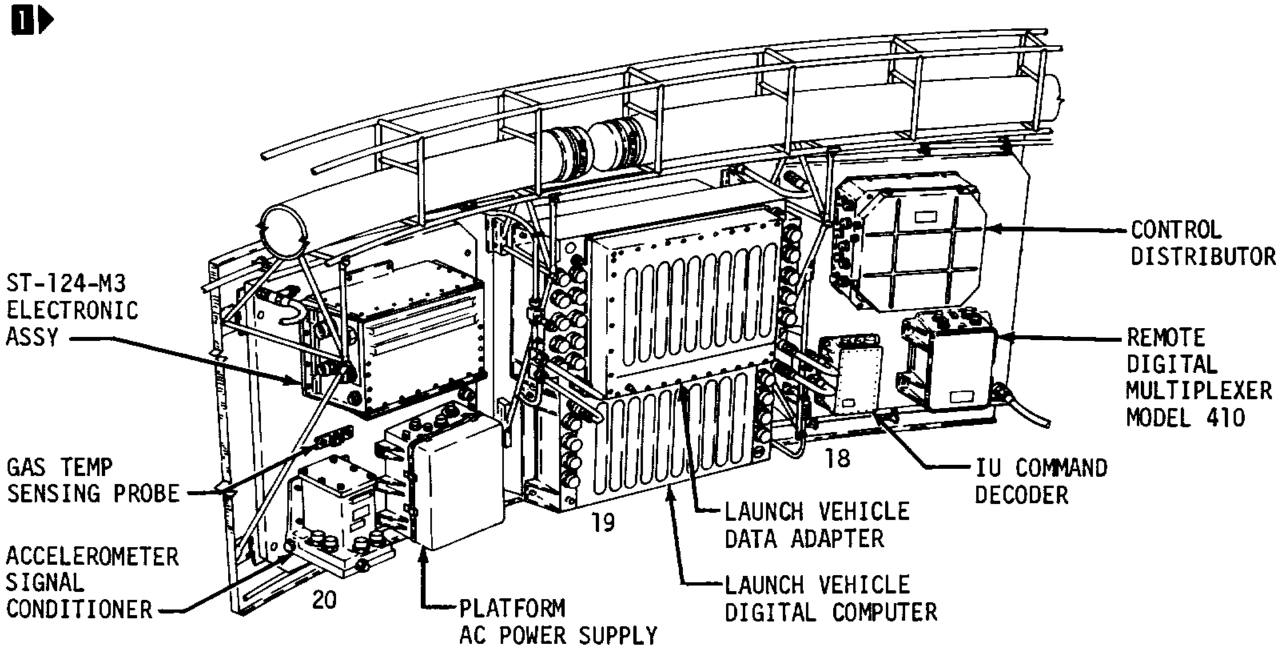


Figure 7-4 (Sheet 4 of 5)

INSTRUMENT UNIT EQUIPMENT LOCATIONS



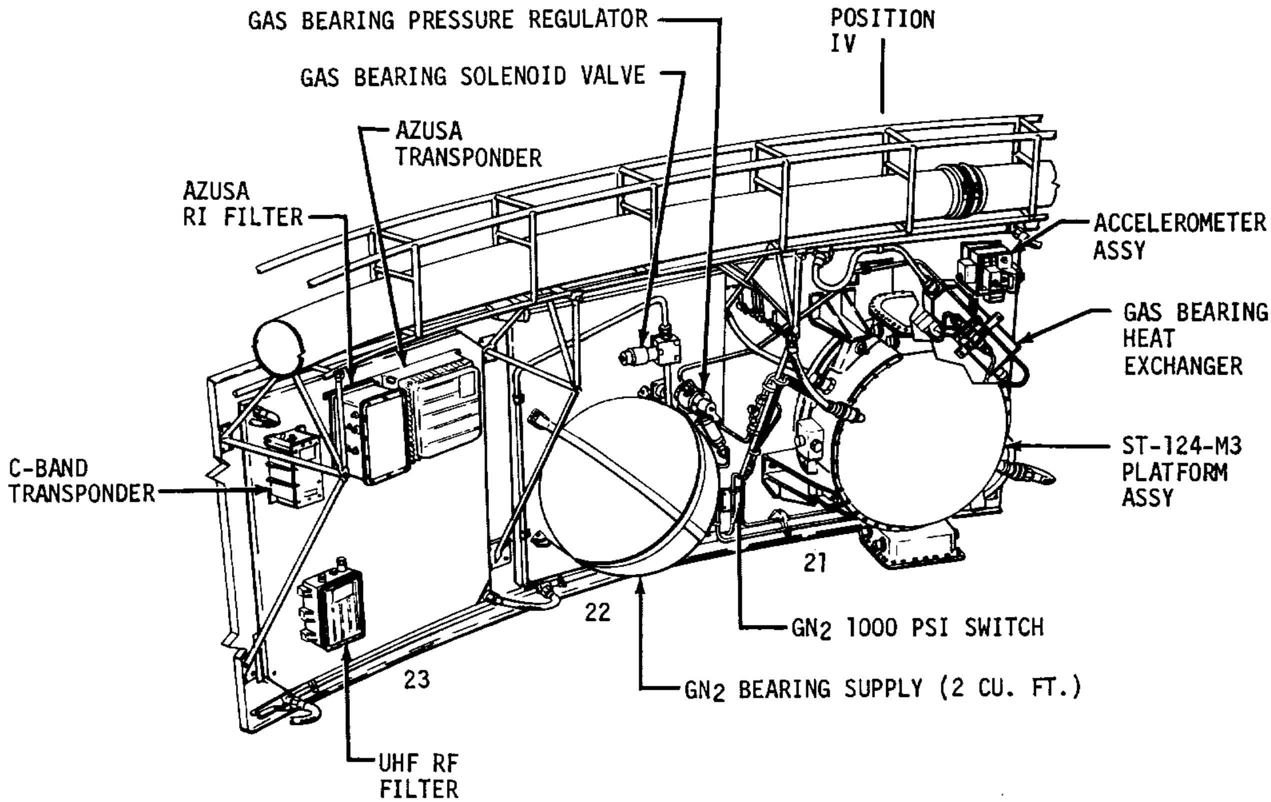


Figure 7-4 (Sheet 5 of 5)

THERMAL CONDITIONING SYSTEM FLOW DIAGRAM

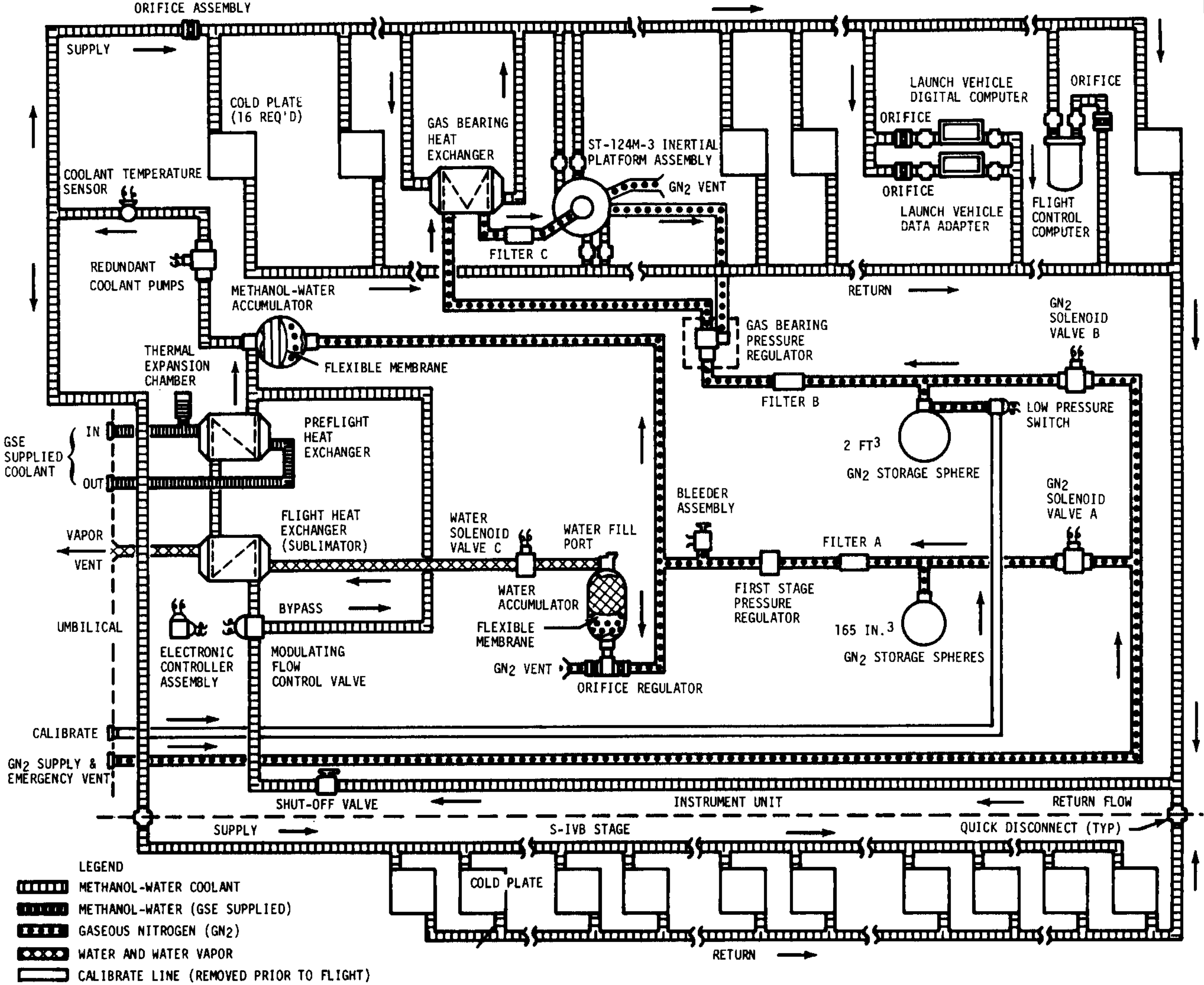


Figure 7-5

VII

Gaseous nitrogen (GN_2) for the methanol-water and water accumulators is stored in a 165 cubic inch sphere in the IU at a pressure of 3,000 psig. Filling of the sphere is accomplished by applying high pressure GN_2 through the umbilical. A solenoid valve controls the flow into the sphere and a pressure transducer indicates to the GSE when the sphere is pressurized. The output of the sphere is applied to the accumulators through a filter and a pressure regulator which reduces the 3,000 psig to 15 psia. An orifice regulator further reduces the pressure at the accumulator to 5 psia; the differential of 10 psia being vented into the IU.

PREFLIGHT AIR/GN2 PURGE SYSTEM

The preflight air/GN_2 purge system (see figure 7-7) consists primarily of flexible ducting located above the IU payload interface. The system distributes ground supplied, temperature and pressure regulated, filtered air or GN_2 through openings in the ducting. During preflight phases ventilating air is furnished. During fueling, inert GN_2 is furnished to prevent accumulation of a hazardous and corrosive atmosphere.

GAS BEARING SUPPLY

Gaseous nitrogen for the ST-124-M3 stable platform is stored in a 2 cubic foot sphere in the IU at a pressure of 3,000 psig (see figure 7-5). Filling of the sphere is accomplished by applying high pressure GN_2 through the umbilical under control of the IU pneumatic console. A low pressure switch monitors the sphere and if the pressure falls below 1,000 psig the ST-124-M3 stable platform is shut down to preclude damage to the gas bearing.

Output of the sphere is through a filter and a pressure regulator. The regulator reduces the sphere pressure to a level suitable for gas bearing lubrication. This pressure is controlled by a pressure feedback loop. Gas bearing pressure is sampled and applied as a control pressure to the pressure regulator. This provides for a constant pressure across the gas bearing. From the main regulator the gas flows through a heat exchanger where its temperature is stabilized, then through another filter and on to the gas bearings. Spent gas is then vented to the IU.

Hazardous Gas Detection

The hazardous gas detection system is used to monitor for the presence of hazardous gases in the IU and S-IVB stage forward compartments during vehicle fueling. The monitoring operation

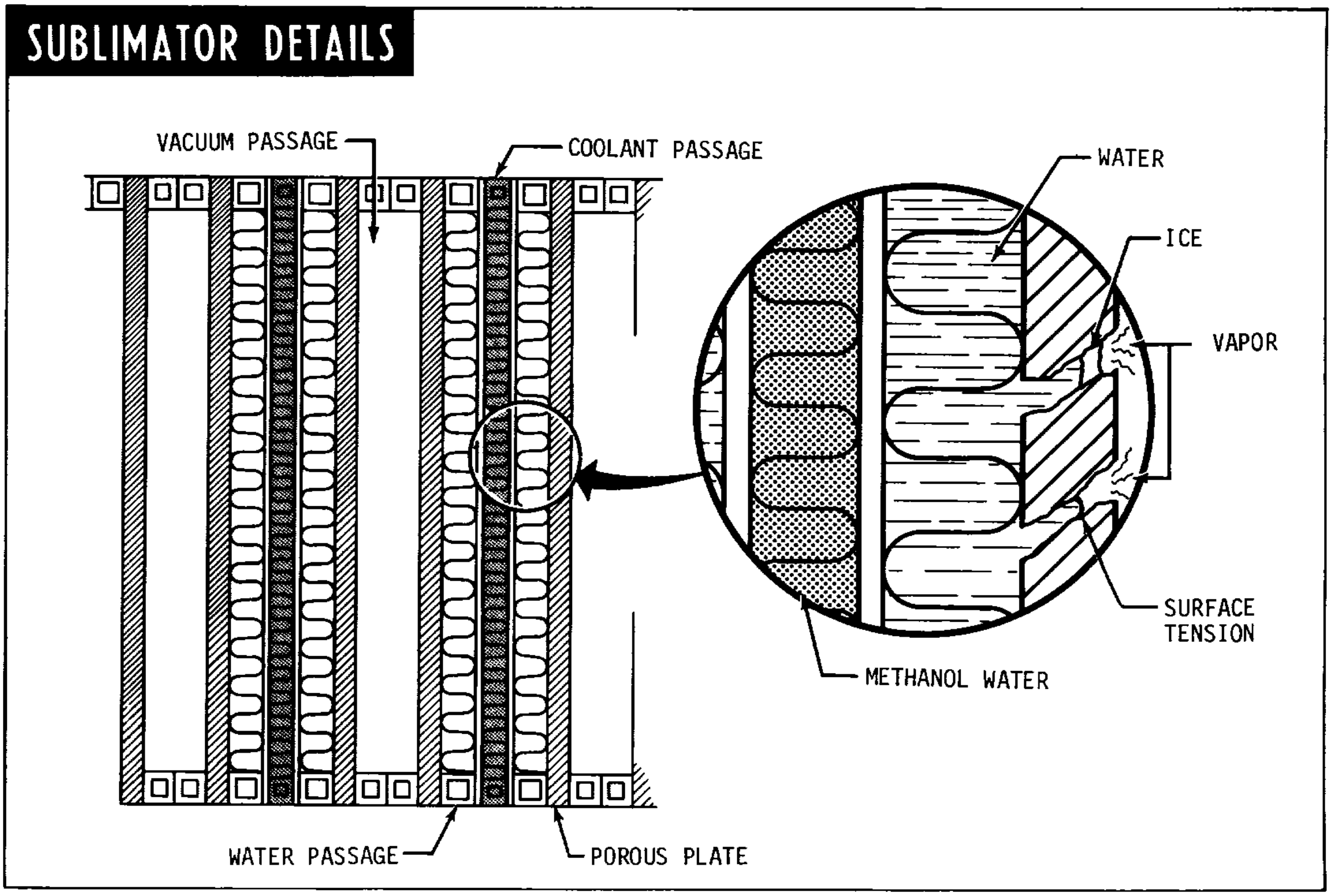


Figure 7-6

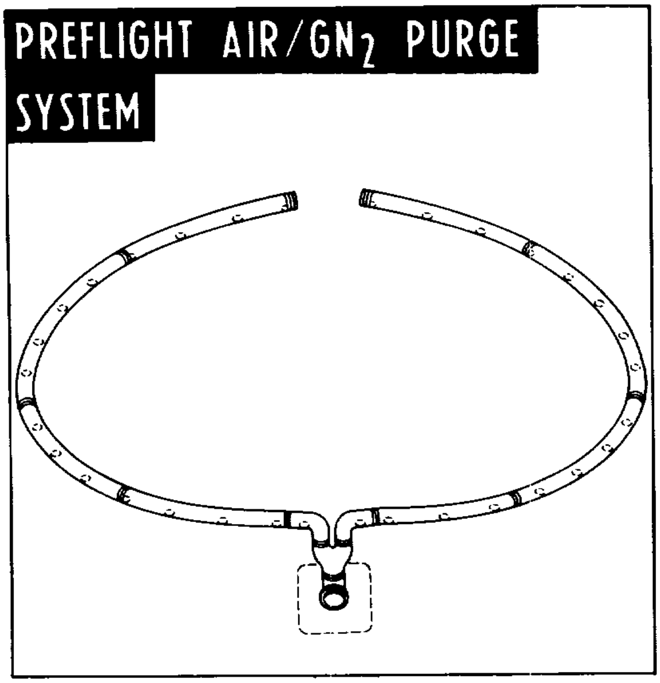


Figure 7-7

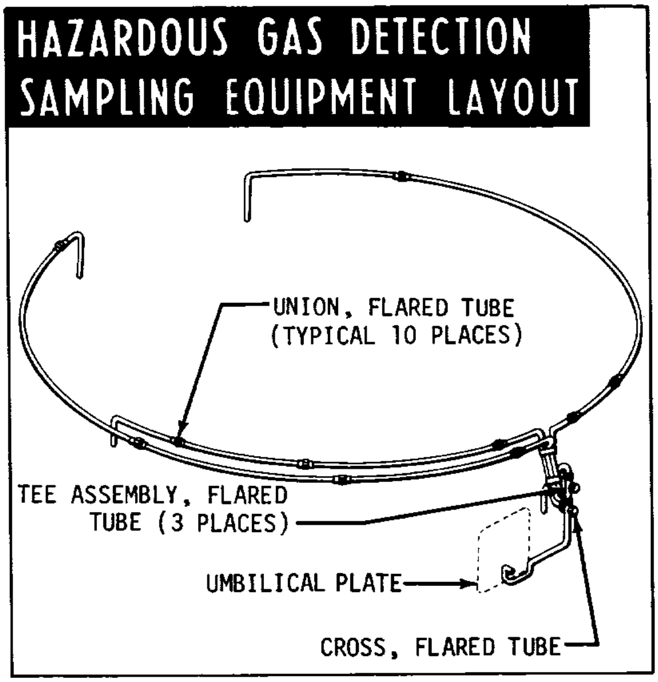


Figure 7-8

is continuous from the start of vehicle fueling to umbilical disconnect at liftoff.

The hazardous gas detection sampling equipment consists of 4 tubes which are open ended between panels 1 and 2, 7 and 8, 13 and 14, and 19 and 20. The tubes are connected to a quick disconnect coupling on a single tube (see figure 7-8).

The hazardous gas detection equipment (GSE) extracts samples through the 4 tubes and monitors the samples for the presence of hazardous gases.

ELECTRICAL POWER SYSTEMS

Primary flight power for the IU equipment is supplied by silver zinc batteries at a nominal voltage level of 28 $\pm$ 2 vdc. During prelaunch operations, primary power is supplied by the GSE. Where ac power is required within the IU it is developed by solid state dc to ac inverters. Power distribution within the IU is accomplished through power distributors which are essentially junction boxes and switching circuits.

BATTERIES

Silver-zinc primary flight batteries are installed in the IU during prelaunch preparations. These batteries are physically and electrically identical but each is connected to a separate bus in a power distributor. Flight components are connected to these busses in such a manner as to distribute the electrical load evenly between the batteries.

IU BATTERY CHARACTERISTICS	
Type	Map 4240-Dry charges
Material	Alkaline silver-zinc
Cells	20 (with taps for selecting 18 or 19 cells if required to reduce high voltage)
Nominal Voltage	1.5 per cell
Electrolyte	Potassium hydroxide (KOH)
Output Voltage	+28 $\pm$ 2 vdc
Output Current	35 amperes for a 10 hour load period (if used within 72 hours of activation)
Gross Weight	165 pounds
Volume	2450 in^3

Figure 7-9

An attractive feature of the silver-zinc batteries is their high efficiency. Their ampere-hour rating is about four times as great as that of a lead-acid or nickle-cadmium battery of the same weight. The low temperature performance of the silver-zinc batteries is also substantially better than the others.

The battery characteristics are listed in figure 7-9.

POWER CONVERTERS

The IU electrical power systems contains a 56 volt power supply and a 5 volt measuring voltage supply.

56 Volt Power Supply

The 56 volt power supply furnishes the power required by the ST-124-M3 platform electronic assembly and the accelerometer signal conditioner. It is basically a dc to dc converter that uses a magnetic amplifier as a control unit. It converts the unregulated 28 vdc from the batteries to a regulated 56 vdc. The 56 volt power supply is connected to the platform electronic assembly through the power and control distributors.

5 Volt Measuring Voltage Supply

The 5 volt measuring voltage supply converts unregulated 28 vdc to a closely regulated 5 $\pm$ .005 vdc for use throughout the IU measuring system. This regulated voltage is used primarily as excitation for measurement sensors (transducers) and as a reference voltage for inflight calibration of certain telemetry channels. Like the 56 volt supply it is basically a dc to dc converter.

DISTRIBUTORS

The distribution system within the IU is comprised of the following:

- 1 Measuring distributor
- 1 Control distributor
- 1 Emergency detection system distributor
- 1 Power distributor
- 2 Auxiliary power distributors

Measuring Distributor

The primary function of the measuring distributor is to collect all measurements that are transmitted by the IU telemetry system and to direct them to their proper telemetry channels. These measurements are obtained from instrumentation transducers, functional components, and various signal and control lines. Also, the measuring distributor distributes the 5 volt output of the 5 volt measuring voltage supply throughout the measuring system.

Through its switching capabilities the measuring distributor can change the selection of measurements monitored by the telemetry system. The switching function transfers certain measurements to channels which had been allotted to expended functions. If it were not for this switching, these channels would be wasted for the remainder of the flight.

Control Distributor

The control distributor serves as an auxiliary power distributor for IU segment 603. It provides distribution of 28 volt power to small current loads and distributes 56 vdc from the 56 volt power supply to the ST-124-M3 inertial platform assembly. The control distributor provides power and signal switching during prelaunch checkout for testing various guidance, control, and emergency detection system functions requested by the launch vehicle data adapter through the switch selector.

Emergency Detection System Distributor

The EDS distributor provides the only electrical link between the spacecraft and the Saturn vehicle. All EDS signals from the Saturn vehicle are routed to the logic circuits in the EDS Distributor. Output EDS signals from these logic circuits are then fed to the spacecraft and to the IU telemetry. Also, EDS signals from the spacecraft are routed back through the EDS logic circuits before being sent to the S-IVB, S-II and S-IC vehicle stages.

Power Distributor

The power distributor provides primary distribution for all 28 volt power required by IU components. Inflight 28 volt battery power, or prelaunch electrical support equipment 28 volt power, is distributed by the power distributor.

The distributor also provides paths for command and measurement signals between the ESE and IU components. The distributor connects the IU component power return and signal return lines to the IU single point ground and to the umbilical supply return bus. These return lines are connected to the common bus in the distributor, directly or indirectly, through one of the other distributors.

Silver-zinc batteries supply the inflight 28 vdc power. See figure 7-10 for a typical power distribution block diagram. Each battery is connected to a separate bus in the distributor and the flight components are connected to the busses in such a manner as to distribute the load evenly across the batteries.

Auxiliary Power Distributors

Two auxiliary power distributors supply 28 vdc power to small current loads. Both auxiliary power distributors receive 28 vdc from each of the battery busses in the power distributor so that current loads on each of the batteries may be evenly distributed. Relays in the auxiliary power distributors provide power ON/OFF control for IU components during prelaunch checkout and also while in flight. These relays are controlled by the electrical support equipment and the switch selector.

IU GROUNDING

All IU grounding is referenced to the outer skin

TYPICAL IU POWER DISTRIBUTION

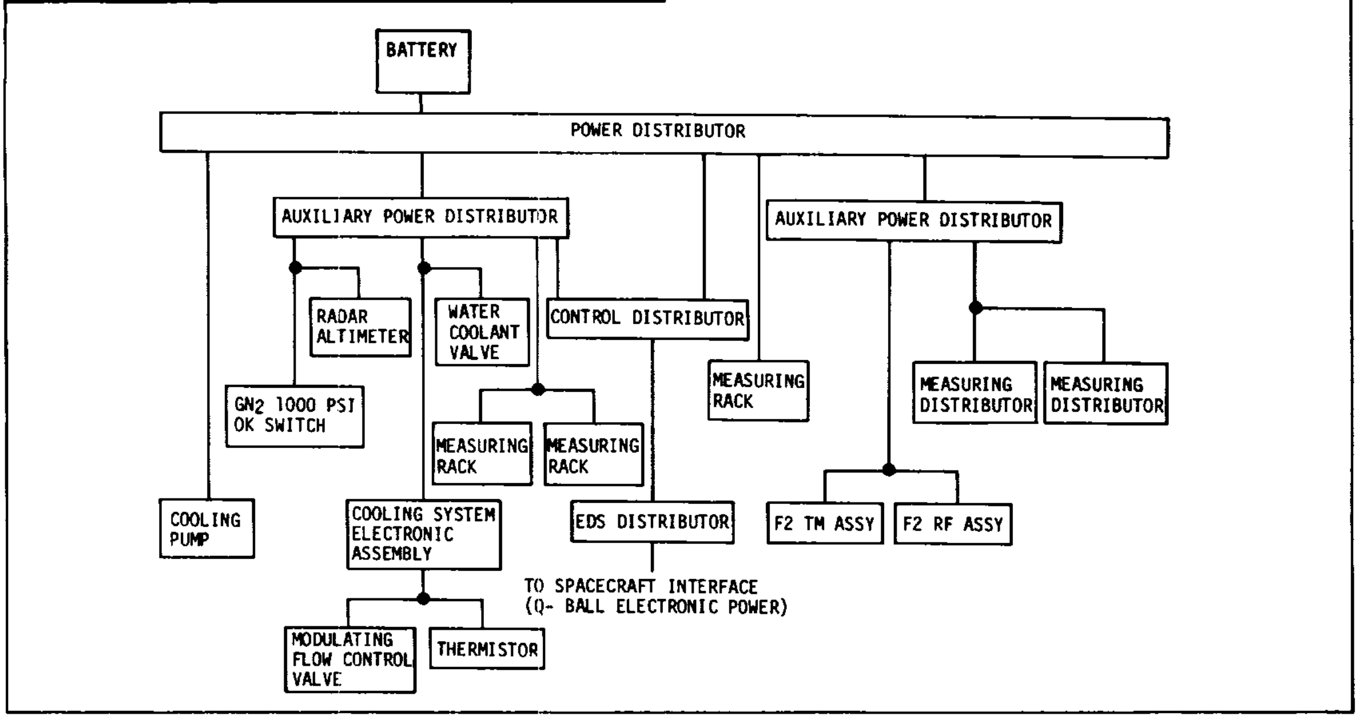


Figure 7-10

of the stage. The power system is grounded by means of hardwires routed from the power distributor COM bus to a grounding stud attached to the stage skin. All COM busses in the various other distributors are wired back to the COM bus in the power distributor. This provides for a single point ground.

Grounding of equipment boxes is accomplished by direct metal to metal contact with cold plates or other mounting surfaces which are common to the stage skin. Most cabling shields are grounded to a COM bus in one of the distributors. However, where shielded cables run between equipment boxes and not through a distributor only one end of the shield would be grounded.

During prelaunch operations the IU and GSE COM busses are referenced to earth ground. To assure the earth ground reference until after all umbilicals are ejected two single wire grounding cables are connected to the IU below the umbilical plates. These are the final conductors to be disconnected from the IU stage.

EMERGENCY DETECTION SYSTEM

The emergency detection system (EDS) is one element of several crew safety systems. EDS design is a coordinated effort of crew safety personnel from several NASA centers.

The EDS will sense initial development of conditions in the flight vehicle during the boost phases of flight which could cause vehicle failure. The EDS will react to these emergency situations in either of two ways. If breakup of the vehicle is imminent an automatic abort sequence will be initiated. If, however, the emergency condition is developing slowly enough or is of such a nature that the flight crew can evaluate it and take action, only visual indications will be provided to the flight crew. Once an abort sequence has been initiated, either automatically or manually, it is irrevocable and runs to completion.

The EDS is comprised of sensing elements such as signal processing and switching circuitry, relay and diode logic circuitry, electronic timers and display equipment, all located in various places on the flight vehicle. Only that part of the EDS equipment located in the IU will be discussed here.

There are nine EDS rate gyros installed in the IU. Three gyros monitor each of the 3 axes, pitch, roll, and yaw, thus providing triple redundancy.

The control signal processor provides power to and receives inputs from the nine EDS rate gyros. These inputs are processed and sent on to the EDS distributor and to the flight control computer.

The EDS distributor serves as a junction box and switching device to furnish the spacecraft display panels with emergency signals if emergency conditions exist. It also contains relay and diode logic for the automatic abort sequence.

There is an electronic timer which is activated at liftoff and produces an output 30 seconds later. This output energizes relays in the EDS distributor which allow multiple engine shutdown. This function is inhibited during the first 30 seconds of launch.

Inhibiting of automatic abort circuitry is also provided by the vehicle flight sequencing circuits through the IU switch selector. This inhibiting is required prior to normal S-IC engine cutoff and other normal vehicle sequencing. While the automatic abort is inhibited, the flight crew must initiate a manual abort if an angular-overrate or two-engine-out condition arises.

See Section III for a more complete discussion of the overall EDS. Section III includes abort limits, displays, controls, diagrams, and a description of the voting logic.

NAVIGATION, GUIDANCE AND CONTROL

The Saturn V launch vehicle is guided from its launch pad into earth orbit by navigation, guidance, and control equipment located in the IU. An all inertial system using a space stabilized platform for acceleration and attitude measurements is utilized. A launch vehicle digital computer (LVDC) is used to solve guidance equations and a flight control computer (analog) is used for the flight control functions.

In the following discussions the terms navigation, guidance, and control are used according to these definitions:

> Navigation is the determination of the flight vehicles present position and velocity from measurements made on board the vehicle.
>
> Guidance is the computations of maneuvers necessary to achieve the desired flight path.
>
> Control is the execution of the guidance maneuver by controlling the proper hardware.

Consider the block diagram of the overall Saturn V guidance and control subsystem shown in figure 7-11. The three-gimbal stabilized platform (ST-124-M3) provides a space-fixed coordinate reference frame for attitude control and for navigation (acceleration) measurements. Three

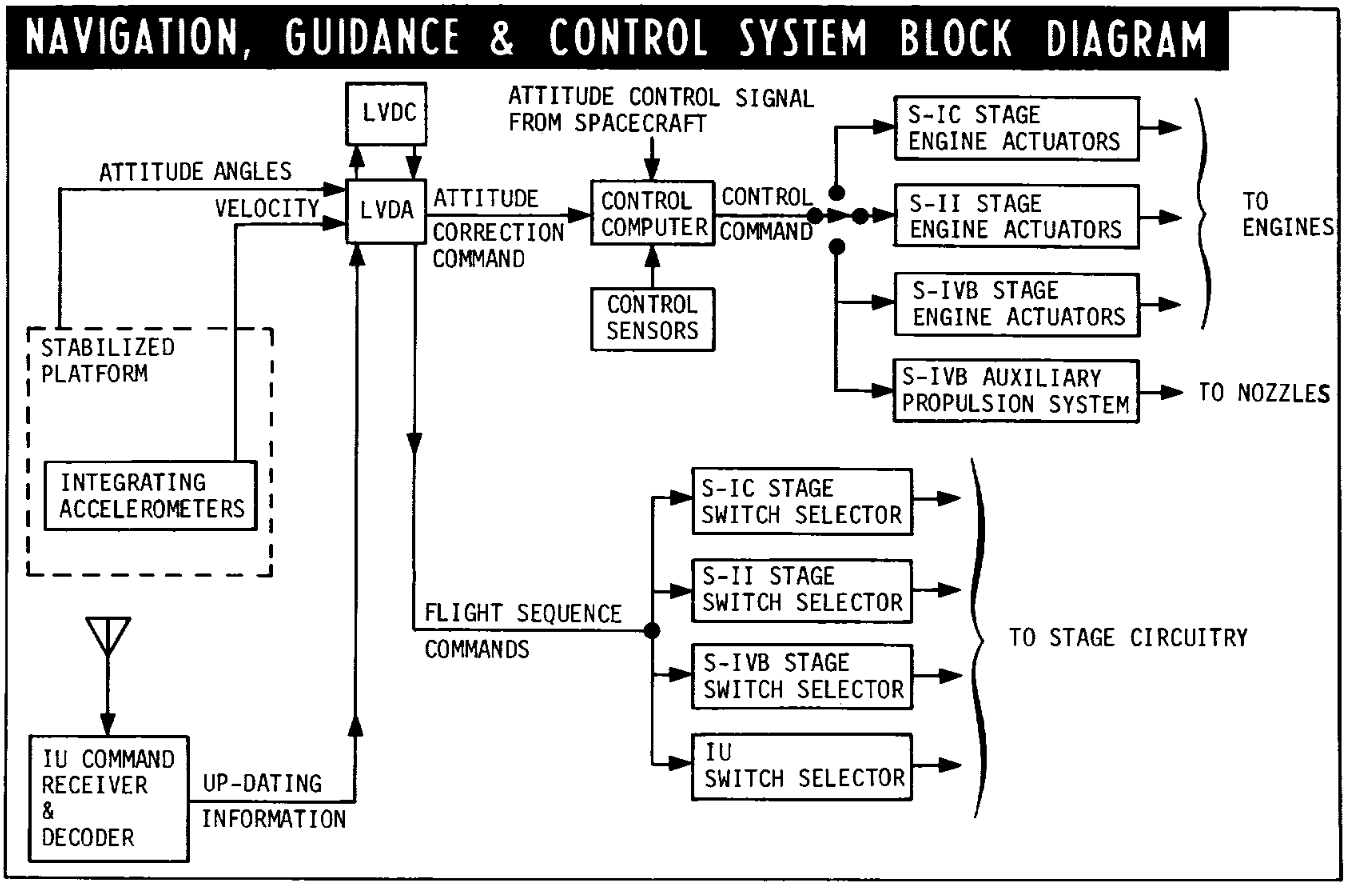


Figure 7-11

integrating accelerometers, mounted on the gyro-stabilized inner gimbal of the platform, measure the three components of velocity resulting from vehicle propulsion. The accelerometer measurements are sent through the launch vehicle data adapter (LVDA) to the LVDC. In the LVDC, the accelerometer measurements are combined with the computed gravitational acceleration to obtain velocity and position of the vehicle.

The LVDA is the input/output device for the LVDC. It performs the necessary processing of signals to make these signals acceptable to the LVDC.

According to the guidance scheme (programmed into the computer), the maneuvers required to achieve the desired end conditions are determined by the LVDC. The instantaneous position and velocity of the vehicle are used as inputs. The result is the required thrust direction (guidance command) and the time of engine cutoff.

Control of the launch vehicle can be divided into attitude control and discrete control functions. For attitude control, the instantaneous attitude of the vehicle is compared with the desired vehicle attitude (computed according to the guidance scheme). This comparison is performed in the LVDC. Attitude correction signals are derived from the difference between the existing attitude angles (platform gimbal angles) and the desired attitude angles. In the flight control computer, these attitude correction signals are combined with signals from control sensors to generate the control commands for the engine actuators. The required thrust direction is obtained by gimbaling the engines in the propelling stage to change the thrust direction of the vehicle. In the S-IC and S-II, the four outboard engines are gimbaled to control roll, pitch, and yaw. Since the S-IVB stage has only one engine, an auxiliary propulsion system is used for roll control during powered flight. The auxiliary propulsion system provides complete attitude control during coast flight of the S-IVB/IU stage.

Guidance information stored in the LVDC (e.g., position, velocity) can be updated through the IU command system by data transmission from ground stations. The IU command system provides the general capability of changing or inserting information into the LVDC.

NAVIGATION SCHEME

Powered Flight

The basic navigation scheme is shown in figure 7-12. Gimbal resolvers supply platform position in analog form to the LVDA. An analog-to-digital converter in the LVDA converts the signal to the digital format required by the LVDC.

Platform integrating accelerometers sense acceleration components and mechanically integrate them into velocity. The LVDA provides signal conditioning. Within the LVDC, initial velocity imparted by the spinning earth, gravitational velocity and the platform velocities are algebracially summed together. This vehicle velocity is integrated by the LVDC to determine vehicle position.

Acceleration can be defined as the rate-of-change (derivative) of velocity. Velocity is

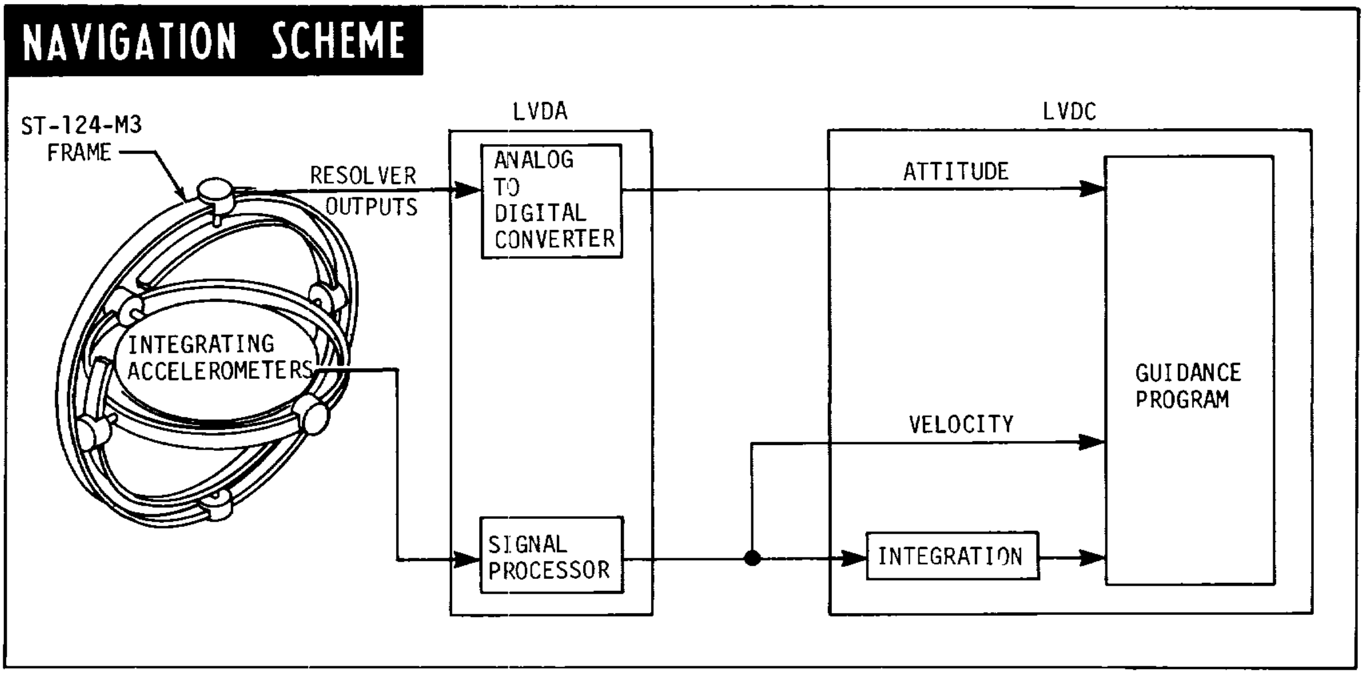


Figure 7-12

VII IU

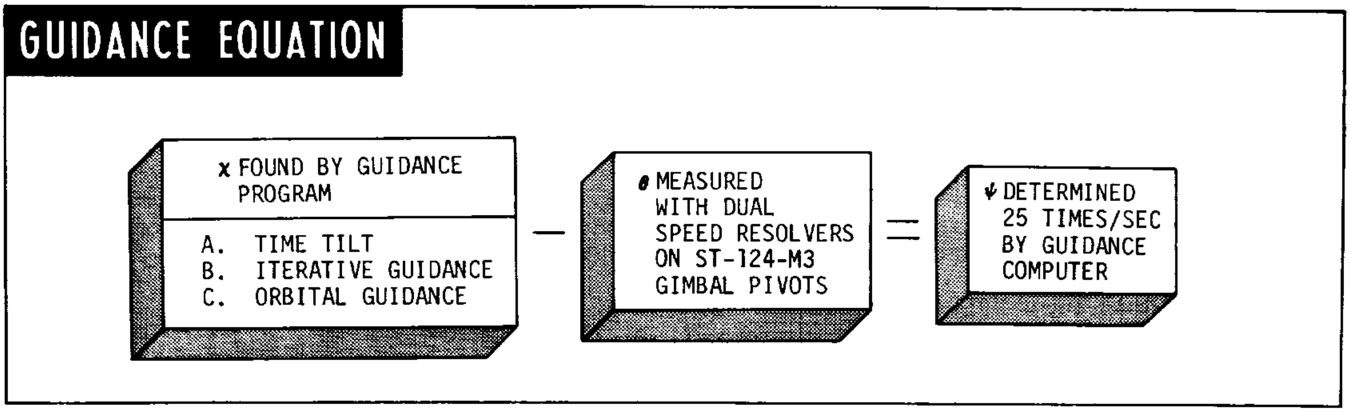


Figure 7-13

the rate-of-change of position (distance). Velocity is the integral of acceleration, and position is the integral of velocity. Therefore, position is obtained by integrating acceleration twice.

Orbital Flight

During orbital coast flight, the navigational program continually computes the vehicle position, velocity, and acceleration from equations of motion which are based on vehicle conditions at the time of orbital insertion. In orbit, navigation and guidance information in the LVDC can be updated by digital data transmission through the command and communications system.

Additional navigational computations are used in maintaining vehicle attitude during orbit. These computations establish a local vertical which is used as a reference for attitude control. The attitude of the vehicle roll axis will be maintained at 90 degrees with respect to the local vertical.

GUIDANCE COMPUTATIONS

The guidance function of the launch vehicle is accomplished by computing the necessary flight maneuvers to meet the desired end conditions of the flight (e.g., inserting the payload into earth orbit). Guidance computations are performed within the LVDC by programmed guidance equations which use navigation data and mission requirements as their inputs. These computations are actually a logical progression of computed formulas which have a guidance command as their solution. After the desired attitude has been determined by the "best path" program, the guidance computation might be reduced into the following equation: $\chi - \Phi = \psi$ (See figure 7-13).

Where:

χ is the desired attitude
Φ is the vehicle attitude
ψ is the attitude error command

CONTROL SUBSYSTEM

The control subsystem (figure 7-14) is designed to maintain and control vehicle attitude by forming the steering commands to be used by the controlling engines of the active stage.

Vehicle attitude is achieved by gimbaling the four outboard engines of the S-IC stage, the four outboard engines of the S-II stage, or the single engine of the S-IVB stage. Gimbaling of these engines is accomplished through hydraulic actuators. Roll attitude control on the S-IVB stage cannot, of course, be controlled with a single thrust engine. Therefore, roll control

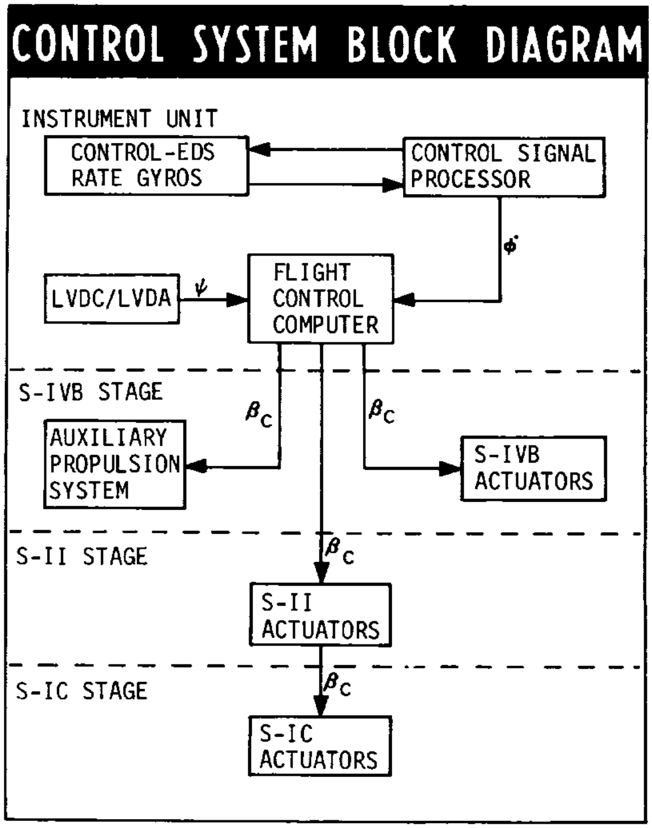


Figure 7-14

of the S-IVB stage is accomplished by the auxiliary propulsion system (APS) (see figure 7-15). During the coast period of the mission, the S-IVB APS will be used to control the vehicle attitude in all three axes.

The control system accepts guidance computations from the LVDC/LVDA Guidance System. These guidance commands, which are actually attitude error signals, are then combined with measured data from the various control sensors. The resultant output is the command signal to the various engine actuators and APS nozzles.

The final computations (analog) are performed within the flight control computer. This computer is also the central switching point for command signals. From this point, the signals are routed to their associated active stages and to the appropriate attitude control devices.

CONTROL SYSTEM COMPONENTS

Control Signal Processor

The control signal processor demodulates the ac signals from the control-EDS rate gyros into dc analog signals required by the flight control computer. The control signal processor compares the output signals from the triple redundant gyros and selects one each of the pitch, yaw, and roll signals for the flight control computer. The control signal processor supplies the control-EDS rate gyro package with the necessary control and reference voltages. EDS and DDAS rate gyro monitoring signals also originate within the control signal processor thus accounting for the EDS portion of the control-EDS rate gyro name.

Flight Control Computer

The flight control computer is an analog computer which develops attitude correction commands (ψ) and angular change rates ($\dot{\phi}$) into APS thruster nozzle and/or engine actuator positioning commands.

Input signals to the flight control computer include:

1. Attitude correction commands (ψ) from the LVDC/LVDA or spacecraft
2. Angular change rates ($\dot{\phi}$) from the control-EDS rate gyro package via the control signal processor

Since one of the inputs to the flight control computer is from the control-EDS rate gyros, an excessive attitude correction command from the LVDC is limited within the flight control computer to protect the vehicle structure.

Output signals from the flight control computer include:

1. Command signals to the engine actuators (βc)
2. Command signals to the APS thruster nozzles
3. Telemetry outputs which monitor internal operations and functions

ST-124-M3 Inertial Platform Assembly

The inertial platform assembly used in the Saturn V vehicles is designated ST-124-M3 and is a three gimbal system.

The gimbal configuration of the ST-124-M3 offers unlimited freedom about the X & Y axes, but is limited to ± 45 degrees about its Z axis (vehicle yaw at launch). See figure 7-16.

The gimbal system allows the inertial gimbal rotational freedom. Three single-degree-of-freedom gyroscopes have their input axes aligned along an orthogonal inertial coordinates system X_I, Y_I, and Z_I of the inertial gimbal. The signal generator, which is fixed to the output axis of each gyro, generates electrical signals proportional to torque disturbances. These signals are transmitted through the servo electronics which terminate in the gimbal pivot servotorque motors. The servoloops maintain the inner gimbal rotationally fixed in inertial space.

The inner gimbal has three pendulous integrating gyroscopic accelerometers oriented along the inertial coordinates X_I, Y_I, and Z_I. Each accelerometer measuring head contains a pendulous single-degree-of-freedom gyro; the rotation of the measuring head is a measure of acceleration along the input axis of the accelerometer. Since acceleration causes the accelerometer shaft to be displaced as a function of time, the shaft position (with respect to a zero reference) is proportional to velocity, and the accelerometer is referred to as an integrating accelerometer.

Vehicle attitude is measured with respect to the inertial platform using dual speed (32:1) resolvers located at the gimbal pivot points. The outputs of these angle encoders are converted into a digital count in the LVDA.

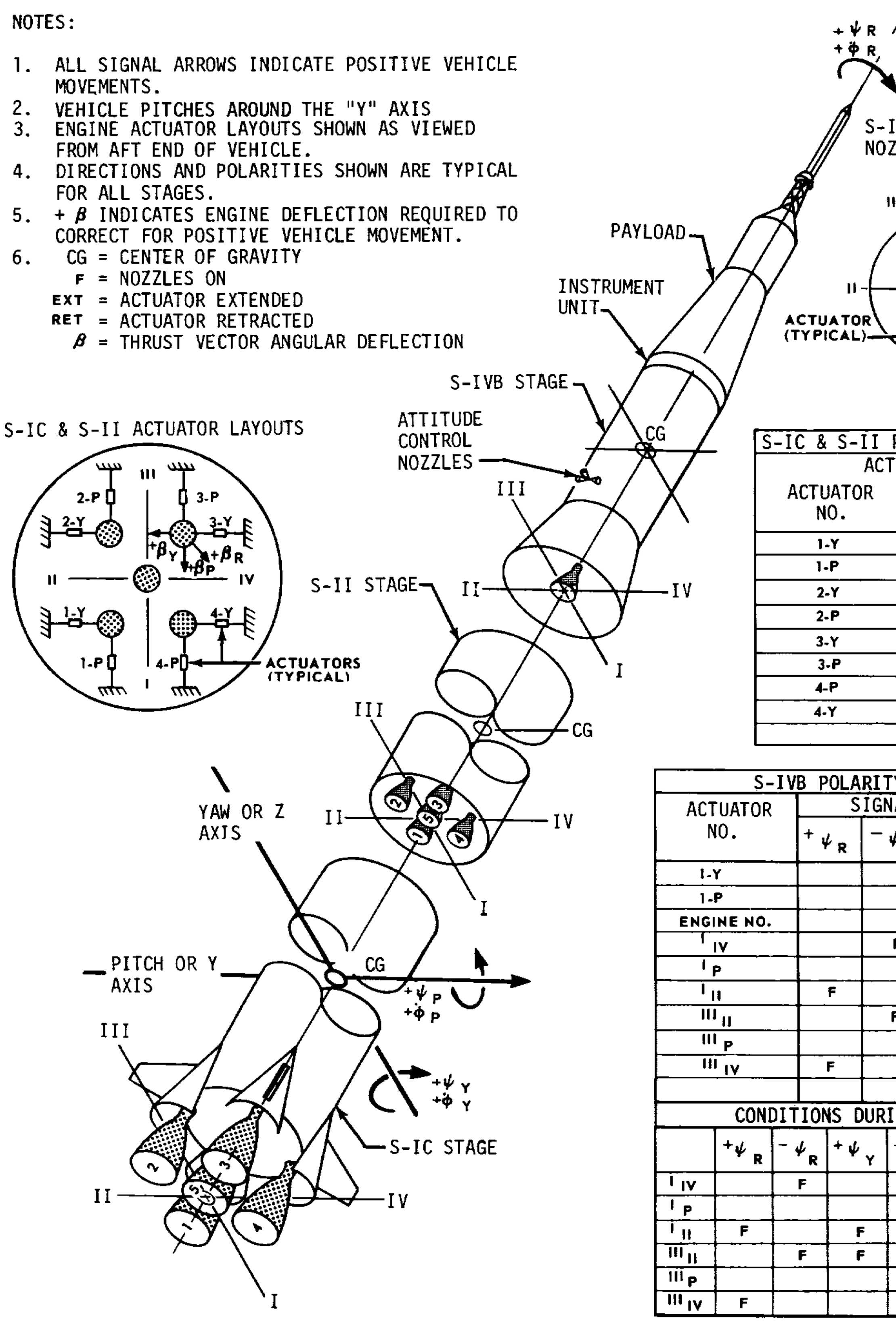
During prelaunch, the ST-124-M3 platform is held aligned to the local vertical by a set of gas bearing leveling pendulums. The pendulum output is amplified in the platform, and then transmitted to the ground equipment alignment amplifier. The alignment amplifier provides a signal to the torque drive amplifier and then to the platform gyro torque generator. The vertical alignment system will level the platform to an accuracy of ± 2.5 arc seconds.

VII

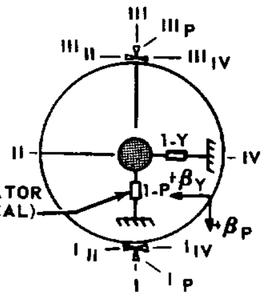
SATURN V ENGINES, ACTUATORS AND NOZZLE ARRANGEMENT

NOTES:

1. ALL SIGNAL ARROWS INDICATE POSITIVE VEHICLE MOVEMENTS.
2. VEHICLE PITCHES AROUND THE "Y" AXIS
3. ENGINE ACTUATOR LAYOUTS SHOWN AS VIEWED FROM AFT END OF VEHICLE.
4. DIRECTIONS AND POLARITIES SHOWN ARE TYPICAL FOR ALL STAGES.
5. + β INDICATES ENGINE DEFLECTION REQUIRED TO CORRECT FOR POSITIVE VEHICLE MOVEMENT.
6. CG = CENTER OF GRAVITY
   F = NOZZLES ON
   EXT = ACTUATOR EXTENDED
   RET = ACTUATOR RETRACTED
   β = THRUST VECTOR ANGULAR DEFLECTION



S-IVB ACTUATOR AND NOZZLE LAYOUT



S-IC & S-II ACTUATOR LAYOUTS

S-IC & S-II POLARITY TABLE

ACTUATOR NO.	ACTUATOR MOVEMENT $+\phi_R$	$+\phi_Y$	$+\phi_P$
1-Y	RET	RET	
1-P	EXT		RET
2-Y	EXT	RET	
2-P	RET		EXT
3-Y	RET	EXT	
3-P	EXT		EXT
4-P	EXT	EXT	
4-Y	RET		RET

S-IVB POLARITY TABLE

ACTUATOR NO.	SIGNAL & ACTION $+\psi_R$	$-\psi_R$	$+\psi_Y$	$+\psi_P$
1-Y			EXT	
1-P				RET
ENGINE NO.				
I_{IV}		F		
I_P				
I_{II}	F			
III_{II}		F		
III_P				
III_{IV}	F			

CONDITIONS DURING COAST

	$+\psi_R$	$-\psi_R$	$+\psi_Y$	$-\psi_Y$	$+\psi_P$	$-\psi_P$
I_{IV}		F		F		
I_P					F	
I_{II}	F		F			
III_{II}		F	F			
III_P						F
III_{IV}	F			F		

Figure 7-15

The azimuth alignment is accomplished by means of a theodolite on the ground and two prisms on the platform; one fixed and one servo driven. The theodolite maintains the azimuth orientation of the moveable prism and the ground based digital computer computes a mission azimuth and programs the inner gimbal to its mission azimuth. The laying system has an accuracy of ± 5 arc seconds.

At approximately liftoff minus seventeen seconds, the platform is released to maintain an inertial reference initiated at the launch point. At this point, the LVDC begins navigation using velocity accumulations derived from the ST-124-M3 inertial platform.

Platform Electronic Assembly

The ST-124-M3 platform electronic assembly (PEA) contains the electronics, other than those located in the platform, required for the inertial gimbal and the accelerometer stabilization. Switching electronics for controlling platform system power and checkout functions are also located in the ST-124-M3 platform electronic assembly.

The PEA includes the following circuitry:

1. Amplifiers, modulators, and stabilization networks for the platform gimbal and accelerometer servo loops.
2. Relay logic for signal and power control.
3. Amplifiers for the gyro and accelerometer pick-off coil excitation.

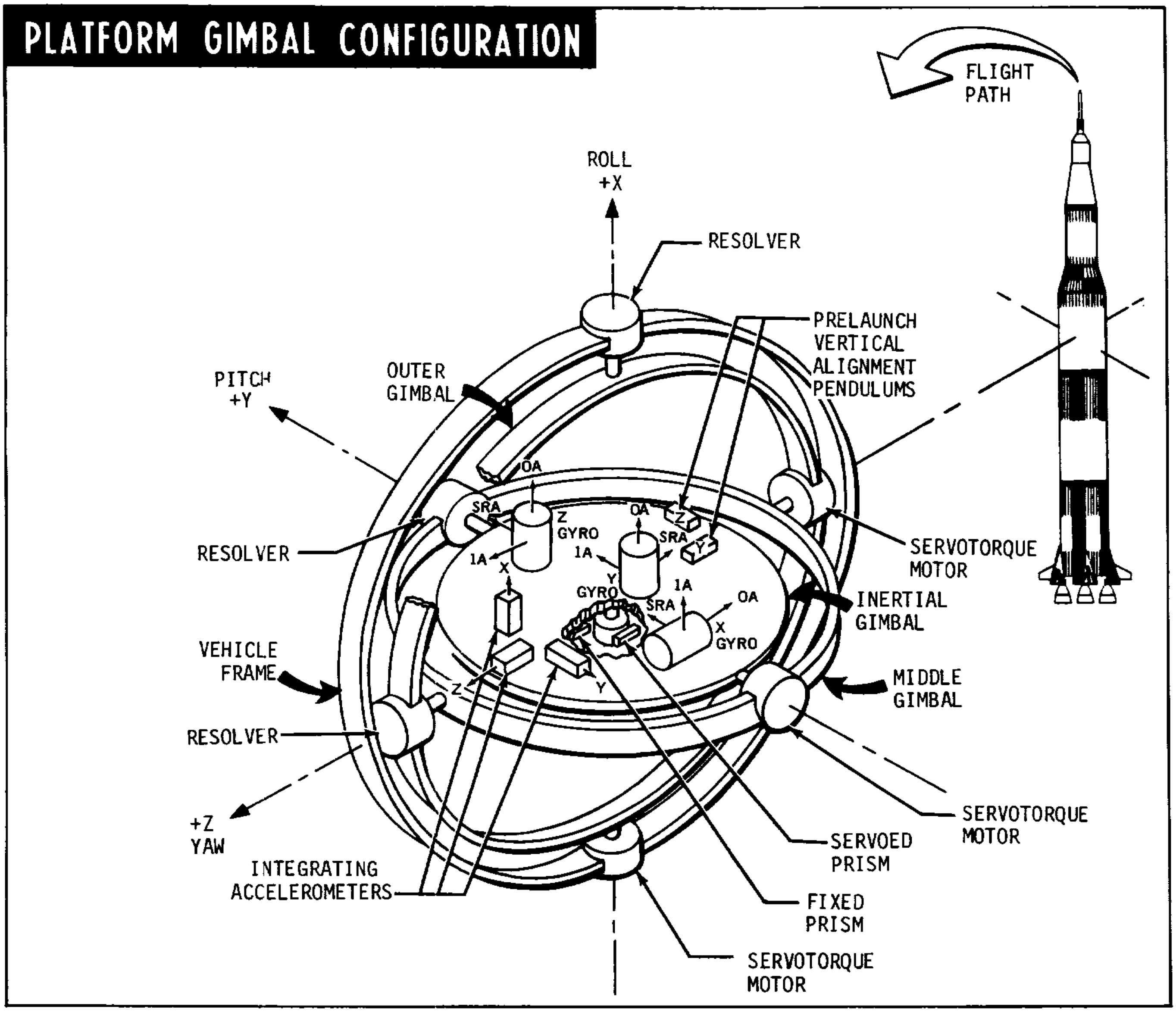


Figure 7-16

4. Automatic checkout selection and test circuitry for servo loops.

5. Control circuitry for the heaters and gas supply.

ST-124-M3 AC Power Supply

The ST-124-M3 platform ac power supply furnishes the power required to run the gyro rotors and provides excitation for the platform gimbal synchros. It also is the frequency source for the resolver chain references and for gyro and accelerometer servo systems carrier.

The supply produces a three-phase, sine wave output which is fixed at 26 volts (rms) line-to-line at a frequency of 400 $\pm$ 0.01 Hertz. Three singlephase, 20 volt reference outputs (square wave) of 4.8 kHz, 1.92 kHz, and 1.6 kHz are also provided. With a normal input voltage of 28 vdc, the supply is capable of producing a continuous 250 va output.

Accelerometer Signal Conditioner

The accelerometer signal conditioner accepts the velocity signals from the accelerometer optical encoders and shapes them before they are passed on to the LVDA/LVDC.

Each accelerometer requires four shapers; a sine shaper and cosine shaper for the active channel and a sine shaper and cosine shaper for the redundant channels. Also included are four buffer amplifiers for each accelerometer; one for each sine and consine output.

Accelerometer outputs are provided for telemetry and ground checkout in addition to the outputs to the LVDA.

LV Digital Computer and LV Data Adapter

The LVDC and LVDA comprise a modern electronic digital computer system. The LVDC is a relatively high-speed computer with the LVDA serving as its input/output device. Any signal to or from the computer is routed through the LVDA. The LVDA serves as central equipment for interconnection and signal flow between the various systems in the Instrument Unit. See figures 7-17 and 7-18 for LVDC and LVDA characteristics.

LAUNCH VEHICLE DIGITAL COMPUTER CHARACTERISTICS

ITEM	DESCRIPTION
Type	General Purpose, Digital, Stored Program
Memory	Random Access, Ferrite (Torodial) Core, with a Capacity of 32,768 words of 28 Bits each.
Speed	Serial Processing at 512,000 Bits Per Seconds
Word Make-Up	Memory = 28 Bits Data = 26 Bits Plus 2 Parity Bits Instruction = 13 Bits Plus 1 Parity Bit
Programming	18 Instruction Codes 10 Arithmetic 6 Program Control 1 Input/Output 1 Store
Timing	Computer Cycle = 82.03 μ sec. Bit Time = 1.95 μ sec. Clock Time = 0.49 μ sec.
Input/Output	External, Program Controlled

Figure 7-17

LAUNCH VEHICLE DATA ADAPTER CHARACTERISTICS

ITEM	DESCRIPTION
Input/Output Rate	Serial Processing at 512,000 Bits Per Second
Switch Selector	8 Bit Input 15 Bit Output
Telemetry	
Command Receiver	14 Bits for Inout Data
Data Transmitter	38 Data and Identification Bits Plus Validity Bit and Parity Bit
Computer Interface Unit	15 Bits Address Plus 1 Data Request Bit 10 Bits for Input Data Plus 1 Bit for Data Ready Interrupt
Delay Lines	3 Four-Channel Delay Lines for Normal Operation 1 Four-Channel Delay Line for Telemetry Operations
Output to Launch Computer	41 Data and Identification Bits Plus Discrete Outputs
Input From RCA-110 GCC	14 Bits for Data Plus Interrupt

Figure 7-18

The LVDA and LVDC are involved in the following main operations:

1. Prelaunch checkout
2. Navigation and guidance computations
3. Vehicle sequencing
4. Orbital checkout

The LVDC is a general purpose computer which processes data under control of a stored program. Data is processed serially in two arithmetic section which can, if so programmed, operate concurrently. Addition, subtraction, and logical extractions are performed in one arithmetic section while multiplication and division are performed in the other.

The principal storage device is a random access, ferrite-core memory with separate controls for data and instruction addressing. The memory can be operated in either a simplex or duplex mode. In duplex operation, memory modules are operated in pairs with the same data being stored in each module. Readout errors in one module are corrected by using data from its mate to restore the defective location. In simplex operation, each module contains different data which doubles the capacity of the memory. However, simplex operation decreases the reliability of the LVDC because the ability to correct readout errors is sacrificed. The memory operation mode is program controlled. Temporary storage is provided by static registers composed of latches and by shift registers composed of delay lines and latches.

Computer reliability is increased within the logic sections by the use of triple modular redundancy. Within this redundancy scheme, three separate logic paths are voted upon by voter logic in order to correct any errors which might develop.

FLIGHT PROGRAM

A flight program is defined as a set of instructions which controls the launch vehicle digital computer (LVDC) operation from seconds before liftoff until the end of the launch vehicle mission. These instructions are stored in memory within the LVDC.

The flight program performs many functions during the launch vehicle mission. These functions include: navigation, guidance, attitude control, event sequencing, data management, ground command processing, and hardware evaluation. Specific definition of these functions depends on mission objectives.

For purposes of discussion the flight program is divided into five subelements. These are the powered flight major loop, the orbital flight program, the minor loop, interrupts, and telemetry.

The powered flight major loop contains guidance and navigation calculations, timekeeping, and all repetitive functions which do not occur on an interrupt basis. The orbital flight program consists of an executive routine concerned with IU equipment evaluation during orbit and a telemetry time-sharing routine to be employed while the vehicle is over receiving stations. In addition, in the orbital flight program, all navigation, guidance, and timekeeping computations are carried out on an interrupt basis keyed to the minor loop. The minor loop contains the platform gimble angle and accelerometer sampling routines and control system computations. Since the minor loop is involved with vehicle control, minor loop computations are executed at the rate of 25 times per second during the powered phase of flight. However, in earth orbit, a rate of only 10 executions per second is required for satisfactory vehicle control.

PRELAUNCH AND INITILIZATION

Until T-20 minutes, the LVDC is under control of the ground control computer (GCC). At T-10 the GCC issues a prepare-to-launch (PTL) command to the LVDC. The PTL routine performs the following functions:

1. Executes an LVDC/LVDA self-test program and telemeters the results.
2. Monitors accelerometer inputs and calculates the platform-off-level indicators. Telemeters accelerometer outputs and time.
3. Performs reasonableness checks on particular discrete inputs and alerts. These discretes and alerts include PTL, guidance reference release (GRR), liftoff, S-IC fuel depletion, S-II propellant depletion, and S-IVB engine cutoff.
4. Interrogates the error monitor register. The purpose of the error monitor register is to detect any errors in the operation of the LVDC and telemeter the results.
5. Keeps all ladder outputs zeroed which keeps the engines in a neutral position for launch.
6. Processes the GRR interrupt and transfers LVDC control to the flight program.
7. Samples platform-gimbal angles.

At T-22 seconds, the launch sequencer issues a GRR alert signal to the LVDC and GCC. At T-17 seconds, a GRR interrupt signal is sent to the LVDC and GCC. With the receipt of this signal, the PTL routine transfers control of the LVDC to the flight program.

When the GRR interrupt is received by the LVDC, the following events take place:

1. The LVDC sets time base zero (T_0).
2. Gimbal angles and accelerometers are sampled and stored for use by flight program routines.
3. Time and accelerometer readings are telemetered.
4. All flight variables are initialized.
5. The GCC is signaled that LVDC is under control of the flight program.

During the time period between GRR and liftoff, the LVDC begins to perform navigational calculations and processes the minor loops. At T-8.8 seconds, the engines are ignited. At T-0 liftoff occurs and a new time base (T_1) is initiated.

POWERED FLIGHT MAJOR LOOP

The major loop contains the navigation and guidance calculations, timekeeping, and other repetitive operations of the flight program. Its various routines are subdivided by function. Depending upon mode of operation and time of flight, the program will follow the appropriate sequence of routines.

The accelerometer processing routine accomplishes two main objectives: it accumulates velocities as measured by the platform, and tries to detect velocity measurement errors through "reasonableness" tests.

The boost navigation routine combines gravitational acceleration with measured platform data to compute position and velocity.

The "pre-iterative" guidance mode, or "time-tilt" guidance program, is that part of the flight program which is performed from GRR until the end of the S-IC burn. The guidance commands issued during the time-tilt phase are functions of time only. This phase of the program is referred to as open loop guidance since vehicle dynamics do not affect or influence the guidance commands. When the launch vehicle has cleared the mobile launcher, the time-tilt program first initiates a roll maneuver to align the vehicle with the proper azimuth. After this command, roll and yaw commands remain at zero and the vehicle is gradually pitched about the vehicle's Y azis to its predetermined boost heading. Rate limiting of the output commands prevents the angles (desired flight attitude angles) from exceeding 1° per second.

The iterative guidance mode (IGM) routine, or "path adaptive" guidance, commences after second-stage ignition and continues until the end of S-IVB first burn. Cutoff occurs when the velocity required for earth orbit has been reached. IGM is used again during S-IVB second burn. IGM is based on optimizing techniques using the calculus of variations to determine a minimum propellant flight path which satisfies mission requirements. Since the IGM considers vehicle dynamics, it is referred to as closed loop guidance.

INTERRUPTS

An interrupt routine permits interruption of the normal program operation to free the LVDC for priority work and may occur at any time within the program. When an interrupt occurs, the interrupt transfers LVDC control to a special subroutine which identifies the interrupt source, performs the necessary subroutines, and then returns to the point in the program where the interruption occurred. Figure 7-19 is a list of interrupts in the order of decreasing priority.

TELEMETRY ROUTINE

A programmed telemetry feature is also provided as a method of monitoring LVDC and LVDA operations. The telemetry routine transmits specified information and data to the ground via IU telemetry equipment. In orbit, telemetry data must be stored at times when the vehicle is not within range of a ground receiving station. This operation is referred to as data compression. The stored data is transmitted on a time-shared basis with real-time telemetry when range conditions are favorable.

DISCRETE BACKUPS

Certain discrete events are particularly important to the flight program since they periodically reset the computer time base which is the

INTERRUPTS	
Decreasing Priority	Function
1	Minor Loop Interrupt
2	Switch Selector Interrupt
3	Computer Interface Unit Interrupt
4	Temporary Loss of Control
5	Command Receiver Interrupt
6	Guidance Reference Release
7	S-II Propellant Depletion/Engine Cutoff
8	S-IC Propellant Depletion/Engine Cutoff "A"
9	S-IVB Engine Out "B"
10	Program Re-Cycle (RCA-110A Interrupt)
11	S-IC Inboard Engine Out "A"
12	Command LVDA/RCA-110A Interrupt

Figure 7-19

reference for all sequential events. For Saturn V vehicles, these significant events (time bases) are:

1. T_1 Liftoff (LO)
2. T_2 S-IC center engine cutoff (CECO)
3. T_3 S-IC outboard engine cutoff (OECO)
4. T_4 S-II cutoff
5. T_5 S-IVB cutoff (boost phase)
6. T_6 S-IVB restart
7. T_7 S-IVB cutoff (orbital phase)

Since switch selector outputs are a function of time (relative to one of the time bases), no switch selector output could be generated if one of the discrete signals were missed. A backup routine is provided to circumvent such a failure. The discrete backup routine will simulate these critical signals if they do not occur when expected.

In the cases of the backup routines for LO and CECO, special routines are established as a double safety check. In both cases, motion as well as time are confirmed before a backup discrete is used. For LO, the backup routine is entered 17.5 seconds after GRR. If the vertical acceleration exceeds 6.544 ft/sec^2 for four computation cycles, the vehicle is assumed to be airborne and the liftoff discrete is issued. For IECO, an assurance is made that an on-the-pad firing of the S-II stage cannot occur if T_1 is accidentally set. Before T_2 can be initiated, velocity along the downrange axis is tested for a minimum of 500 m/sec.

Refer to MODE AND SEQUENCE CONTROL, Section VII, for a discussion of the discrete time bases T_1 through T_7.

The execution time for any given major loop, complete with minor loop computations and interrupts, is not fixed. The average execution time for any given major loop in powered flight, complete with minor loop computation and interrupt processing, is called the normal computation cycle for that mode. The computation cycle is not fixed for two reasons. First, the various flight modes of the program have different computation cycle lengths. Second, even in a given flight mode, the uncertainties of discrete and interrupt processing and the variety of possible paths in the loop preclude a fixed computation cycle length.

MODE AND SEQUENCE CONTROL

Mode and sequence control involves most of the electrical/electronic systems in the Saturn V launch vehicle. However, in this section the discussion will deal mainly with the switch selector and associated circuitry.

The launch vehicle digital computer (LVDC) memory contains a predetermined number of sets of instructions which, when initiated, induce portions of the launch vehicle electrical/electronics systems to operate in a particular mode. Each mode consists of a predetermined sequence of events. The LVDC also generates appropriate discrete signals such as engine ignition, engine cutoff, and stage separation.

Mode selection and initiation is accomplished through either an automatic LVDC internal command or through an external command from ground checkout equipment, IU command system, or from the flight crew in the spacecraft.

The flexibility of the mode and sequence control scheme is such that no hardware modification is required for mode and flight sequence changes. The changes are accomplished by changing the instructions and programs in the LVDC memory.

SWITCH SELECTOR

Many of the sequential operations in the launch vehicle that are controlled by the LVDC are performed through a switch selector located in each stage. The switch selector decodes digital flight sequence commands from the LVDA/LVDC and activates the proper stage circuits to execute the commands. The outputs of the switch selector drive relays either in the units effected or in the stage sequencer.

Each switch selector can activate, one at a time, 112 different circuits in its stage. The selection of a particular stage switch selector is accomplished through the command code. Coding of flight sequence commands and decoding by the stage switch selectors reduces the number of interface lines between stages and increases the flexibility of the system with respect to timing and sequence. In the Saturn V launch vehicles, which contain 4 switch selectors, 448 different functions can be controlled using only 28 lines from the LVDA. Flight sequence commands may be issued at time intervals of 100 milli-seconds.

To maintain power isolation between vehicle stages, the switch selectors are divided into sections: the input sections (relay circuits) of each switch selector receive their power from the IU; the output sections (decoding circuitry and drivers) receive their power from the stage in which the switch selector is located. The inputs and outputs are coupled together through a diode matrix. This matrix decodes the 8 bit input code and activates a transistorized output driver, thus producing a switch selector output.

The output signals of the LVDA switch selector register, with the exception of the 8 bit command, are sampled at the control distributor in the IU and sent to IU PCM telemetry. Each switch selector also provides 3 outputs to the telemetry system within its stage.

The switch selector is designed to execute flight sequence commands given by the 8 bit code or by its complement. This feature increases reliability and permits operation of the system despite certain failures in the LVDA switch selector register, line drivers, interface cabling, or switch selector relays.

The flight sequence commands are stored in the LVDC memory and are issued according to the flight program. When a programmed input/output instruction is given, the LVDC loads the 15 bit switch selector register with the computer data.

The switch selector register bits 1 through 8 represent the flight sequence command. Bits 9 through 13 select the switch selector to be activated. Bit 14 resets all the relays in the switch selectors in the event data transfer is incorrect as indicated by faulty verification information received by the LVDA. Bit 15 activates the addressed switch selector for execution of the command. The switch selector register is loaded in two passes by the LVDC: Bits 1 through 13 on the first pass and either bit 14 or bit 15 on the second pass, depending on the feedback code. The LVDA/LVDC receives the complement of the code after the flight sequence command (bits 1 through 8) has been picked up by the input relays of the switch selector. The feedback (verification information) is returned to the LVDA and compared with the original code in the LVDC. If the feedback agrees the LVDC/LVDA sends a read command to the switch selector. If the verification is not correct a reset command is given (forced reset) and the LVDC/LVDA reissues the 8 bit command in complement form.

Figure 7-20 illustrates the Saturn V switch selector functional configuration. All switch selector control lines are connected through the control distributor in the IU to the LVDC and the electrical support equipment.

The LVDC switch selector interconnection diagram is shown in figure 7-21. All connections between the LVDA and the switch selectors, with the exception of the stage select inputs, are connected in parallel.

OPERATION SEQUENCE

The Saturn V operation sequence starts during the prelaunch phase at approximately 22 hours when the electrical power from the ground support equipment is applied to all stages of the launch vehicle. During this time the sequencing is controlled from the launch control center/mobile launcher complex utilizing both manual and automatic control to checkout the functions of the entire launch vehicle. After the umbilicals are disconnected, the sequencing is primarily controlled by the flight program within the LVDC.

Since flight sequencing is time phased, the sequencing operation is divided into seven primary time bases. Each time base is related to a particular flight event. Four alternate time bases have also been provided for. These time bases are defined as follows:

Time Base No. 1 (T_1)

T_1 is initiated by a liftoff signal provided by deactuation of the liftoff relay in the IU at the umbilical disconnect. However, as a safety measure, the LVDC will not recognize the liftoff signal and start T_1 prior to receiving guidance reference release (GRR) plus 16.0 seconds.

A backup method for starting T_1 is provided should the LVDC fail to receive or recognize the liftoff signal. If T_1 is not initiated within 17.5 seconds after guidance reference release, the LVDC will monitor the vertical accelerometer. If a significant positive acceleration (in excess of 1 g) exists, the LVDC assumes liftoff has occurred and begins T_1. A compensating time adjustment is made by the LVDC.

No "negative backup" (i.e., provisions for the LVDC to return to prelaunch conditions) is provided because, in the event T_1 began by error, the Saturn V vehicle could safely complete T_1 on the pad without catastrophic results.

Time Base No. 2 (T_2)

The S-IC center engine will be cutoff by the LVDC through the S-IC switch selector at a predetermined time. At this time the LVDC will monitor the downrange accelerometer. If sufficient downrange velocity exists, the LVDC will start T_2.

Use of the downrange velocity reading provides a safeguard against starting T_2 on the pad should T_1 be started without liftoff. Furthermore, if T_2 is not established, no subsequent time bases can be started. This insures a safe vehicle requiring at least one additional failure to render the vehicle unsafe on the pad.

Time Base No. 3 (T_3)

T_3 is initiated at S-IC outboard engine cutoff by either of two redundant outboard engines cutoff signals. However, the LVDC must arm outboard engines propellant depletion cutoff prior to starting T_3. Outboard engines propellant depletion cutoff is armed 1 second prior to calculated outboard engines cutoff.

Time Base No. 4 (T_4)

After arming S-II lox depletion cutoff, the LVDC will initiate T_4 upon receiving either of two signals: S-II engines cutoff, or S-II engines out. The S-II engine cutoff signal is the primary signal for starting T_4. The S-II engines out signal from the thrust OK circuitry is a backup.

As a safeguard against trying to separate the S-II stage with the thrust of the engines present, a redundant S-II engines cutoff command is

SWITCH SELECTOR FUNCTIONAL CONFIGURATION

NOTES: SIGNAL RETURN LINES FROM THE SWITCH SELECTORS, THROUGH THE CONTROL DISTRIBUTOR, TO THE LVDA ARE NOT SHOWN IN THIS FIGURE.
THE LETTERS USED TO LABEL INTERSTAGE CONNECTIONS BETWEEN UNITS ARE NOT ACTUAL PIN OR CABLE CONNECTORS. THE LETTER CODE IS DENOTED BELOW:

a = 8-DIGIT COMMAND (8 LINES)
b = FORCE RESET (REGISTER) (1 LINE + 1 REDUNDANT LINE)
c = REGISTER VERIFICATION (8 LINES)
d = READ COMMAND (1 LINE + 1 REDUNDANT LINE)
e =, f =, g =, h = STAGE SELECT LINES (1 LINE + 1 REDUNDANT LINE)
j = b, c, d, e, f, g, AND h TO IU TELEMETRY
k = REGISTER TEST, ZERO INDICATE, SW SEL OUTPUT TO STAGE TELEMETRY (1 LINE EACH)
m = +28 VDC FROM THE INSTRUMENT UNIT

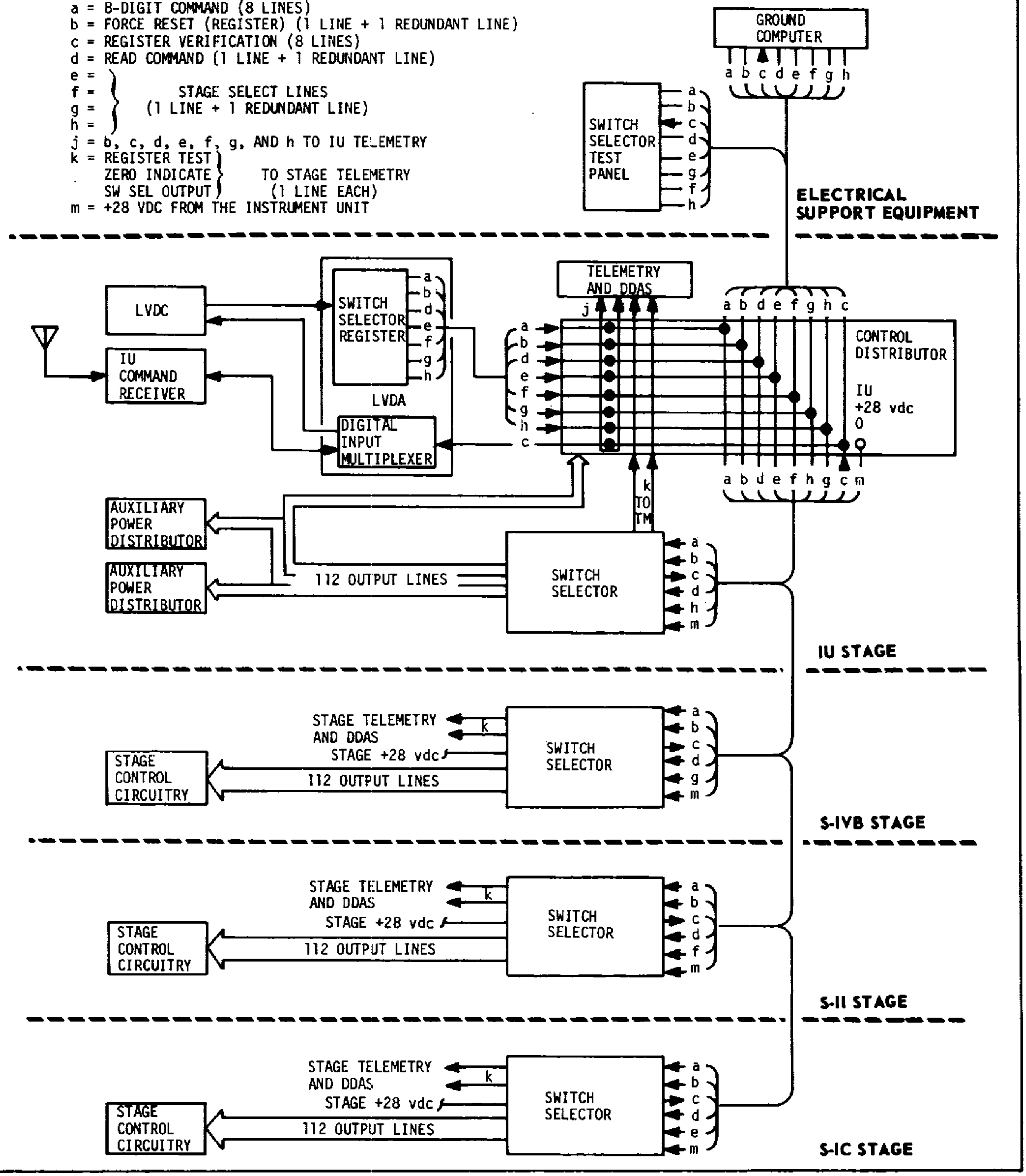


Figure 7-20

LVDC-SWITCH SELECTOR INTERCONNECTION DIAGRAM

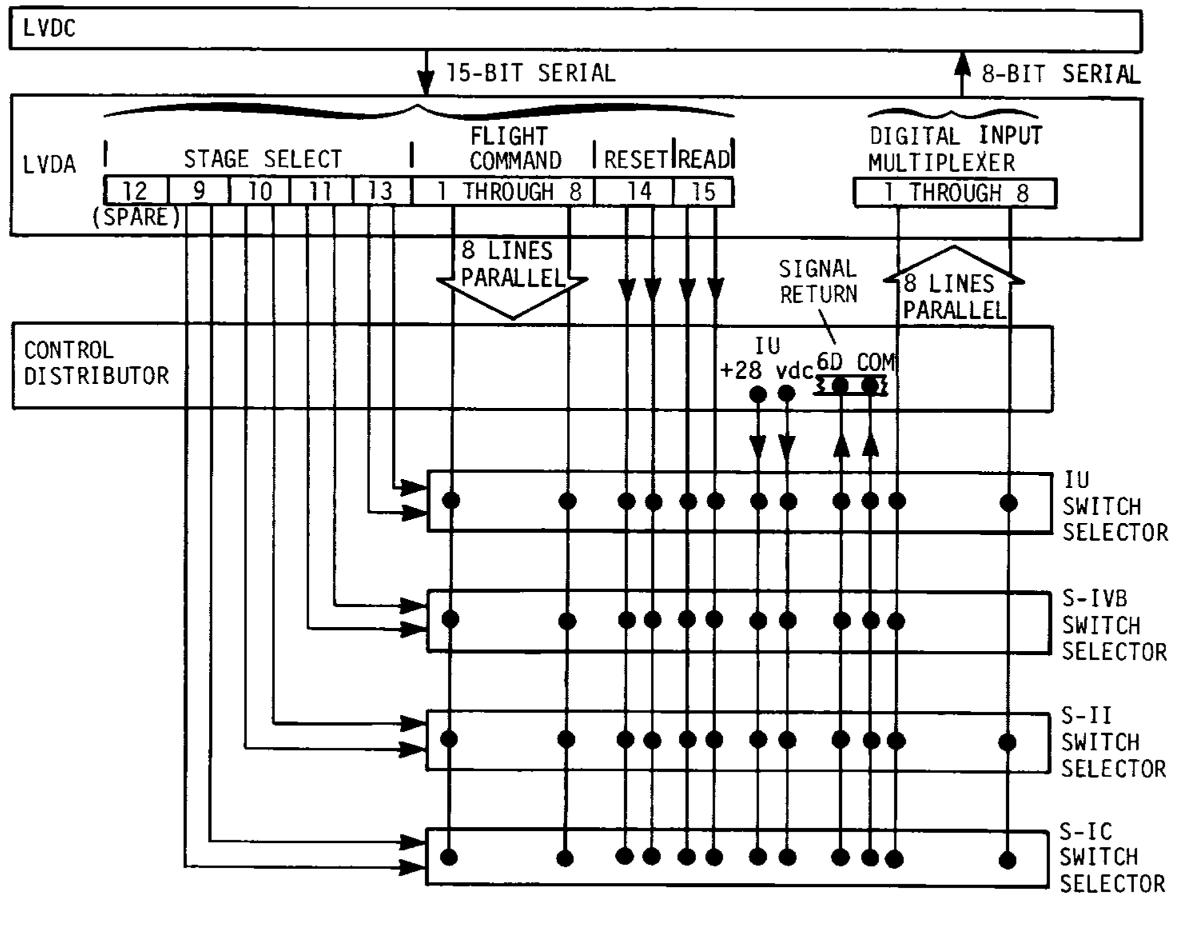


Figure 7-21

issued by the LVDC at the start of T4.

Alternate Time Base No. 4a (T_{4a})

T_{4a} is programmed for use in early staging of the S-IVB stage. This time base will be initiated by the LVDC upon receiving either of two signals: Spacecraft initiation of S-II/S-IVB separation "A", or spacecraft initiation of S-II/S-IVB separation "B". Starting of T_{4a} will be inhibited until T_3 + 1.4 seconds.

Time Base No. 5 (T_5)

T_5 is initiated by the deactuation of the S-IVB thrust OK pressure switches at S-IVB cutoff. The LVDC will start T_5 after receiving any two of four functions monitored by the LVDC. The functions are: S-IVB engine out "A", S-IVB engine out "B", S-IVB velocity cutoff which is issued by the LVDC, and/or loss of thrust determined by LVDC using accelerometer readings.

A redundant S-IVB cutoff command is issued at the start of T_5 as a safeguard against having started time base 5 with the thrust of the S-IVB engine present.

Time Base No. 6 (T_6)

T_6 will be initiated by the LVDC upon solving the restart equation.

With spacecraft control of the Saturn (guidance switchover), T_6 will be initiated by the LVDC upon solving the restart equation or it can be initiated by the spacecraft S-IVB IGNITION SEQUENCE START signal.

Alternate Time Base No. 6a (T_{6a})

T_{6a} is programmed for use should the oxygen-hydrogen burner malfunction between the times T_6 + 1 minute and 31.7 seconds and T_6 + 3 minutes and 55 seconds. This alternate time base will be initiated by the LVDC upon receiving an oxygen-hydrogen burner malfunction signal from the S-IVB stage.

Alternate Time Base No. 6b (T_{6b})

T_{6b} is programmed for use should the oxygen-hydrogen burner malfunction between the times T_6 + 3 minutes and 55 seconds and T_6 + 5 minutes and 46.6 seconds. This alternate time base will be initiated by the LVDC upon receiving an oxygen-hydrogen burner malfunction signal from the S-IVB.

Alternate Time Base No. 6c (T_{6c})

T_{6c} is programmed for use should a failure occur which would require a delay in the S-IVB restart attempt. The TRANSLUNAR INJECTION INHIBIT signal from the spacecraft will be required by the LVDC before this alternate time base will be initiated.

Time Base No. 7 (T_7)

After a predetermined time, sufficient to allow the S-IVB engine to establish thrust OK, the LVDC will start T_7 after receiving any two of four functions monitored by the LVDC. The functions are: S-IVB engine out "A", S-IVB engine out "B", S-IVB velocity cutoff which is issued by the LVDC, and/or loss of thrust determined by LVDC using accelerometer readings.

As a safeguard against starting T_7 with the thrust of the S-IVB engine present, a redundant S-IVB engine cutoff command is issued at the start of T_7.

MEASUREMENTS AND TELEMETRY

The instrumentation within the IU consists of a measuring subsystem, a telemetry subsystem and an antenna subsystem, as illustrated in figure 7-22. This instrumentation is for the purpose of monitoring certain conditions and events which take place within the IU and for transmitting monitored signals to ground receiving stations. Telemetry data is used on the ground for the following purposes:

1. Prior to launch to assist in the checkout of the launch vehicle.

2. During vehicle flight for immediate determination of vehicle condition and for verification of commands received by the IU command system

3. Postflight scientific analysis of the mission

MEASUREMENTS

Some of the typical conditions and events which are monitored within the IU are: temperatures, pressures, flow rates, vibration, accelerations, angular rates, voltages, frequencies, current flow.

The requirement for measurements of great variety and quantity has dictated the use of transducers of many types and at many locations. A discussion of the transducers used requires a level of detail beyond the scope of this manual.

Signal conditioning is accomplished by amplifiers or converters located in measuring racks. There are 10 measuring racks in the IU and 20 signal conditioning modules in each. Each signal conditioning module contains, in addition to its conditioning circuitry, two relays and circuitry to simulate its transducers at both their high range and low range extremities. These relays and transducer simulation circuitry are used for pre-launch calibration of the signal conditioners.

Conditioned signals are routed to their assigned telemetry channel by the measuring distributors. Switching functions connect different sets of measurements to the same telemetry channels during different flight periods. These switching functions, controlled from the ground through the umbilical, connect measurements not required during flight to digital data acquisition system channels for ground checkout and return the channels to flight measurements after checkout.

TELEMETRY

The function of the telemetry system is to format and transmit measurement signals received from the measurement distributors.

VII

The approximately 200 measurements made on the IU are transmitted via five telemetry links. The three modulation techniques used are:

2

1. Pulse Code Modulation/Frequency Modulation (PCM/FM)

2. Frequency Modulation/Frequency Modulation (FM/FM)

3. Single Sideband/Frequency Modulation (SS/FM)

The PCM/FM data is transmitted over VHF and UHF links to provide redundancy for the command and communications system (CCS) and for certain critical measurements.

Multiplexing

In order for the five IU telemetry links to handle approximately 300 separate measurements these links must be "shared". By proper multiplexing it is possible to transmit several different signals simultaneously from one telemetry system. Both frequency sharing and time sharing multiplexing techniques are used to accomplish this. Refer to figure 7-23 for a block diagram of the IU telemetry system.

Several Model 270 multiplexers (MUX-270) are used in the IU telemetry system. The MUX-270 is a time sharing multiplexer. Each one operates as a 30 x 120 (30 primary channels, each sampled

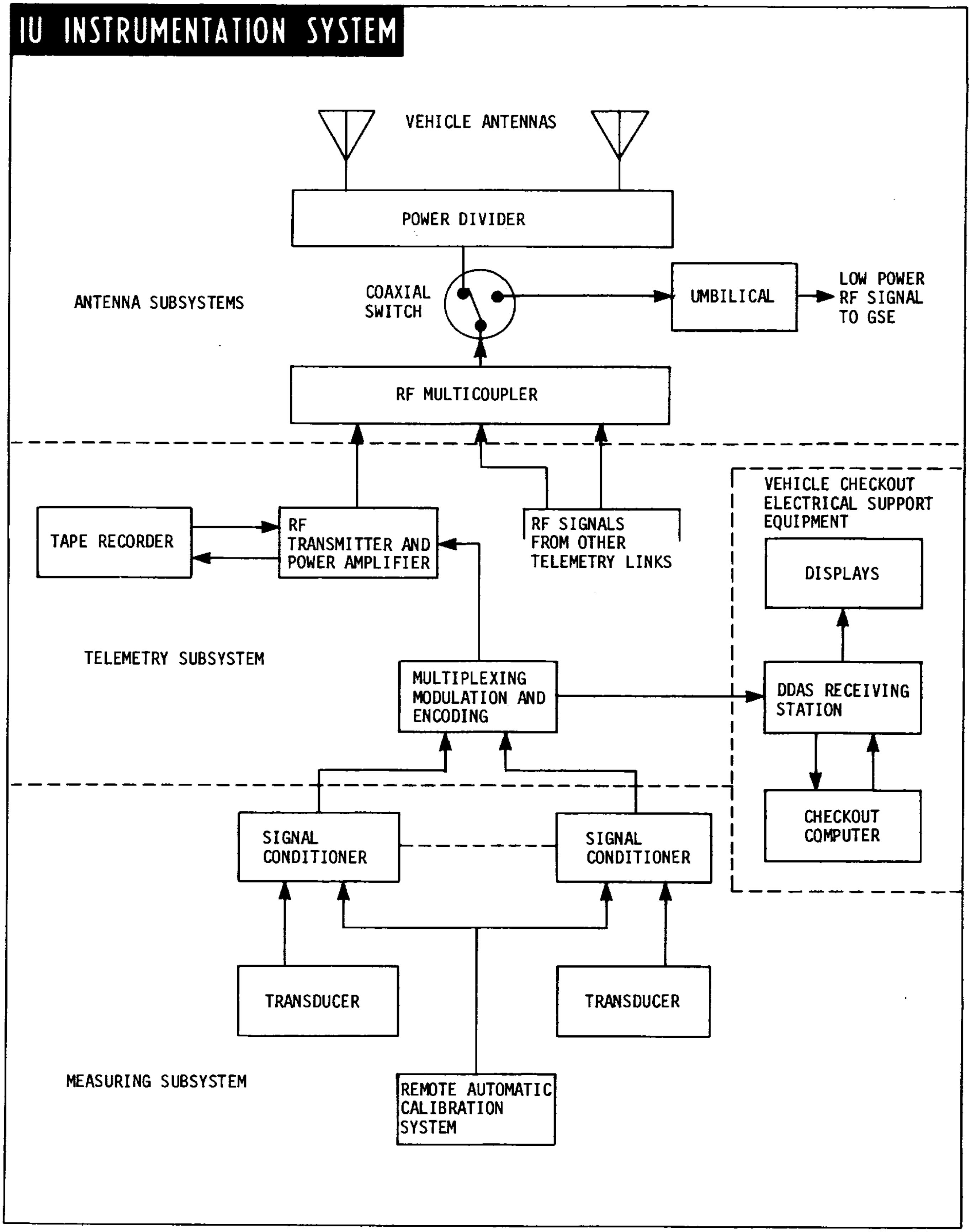
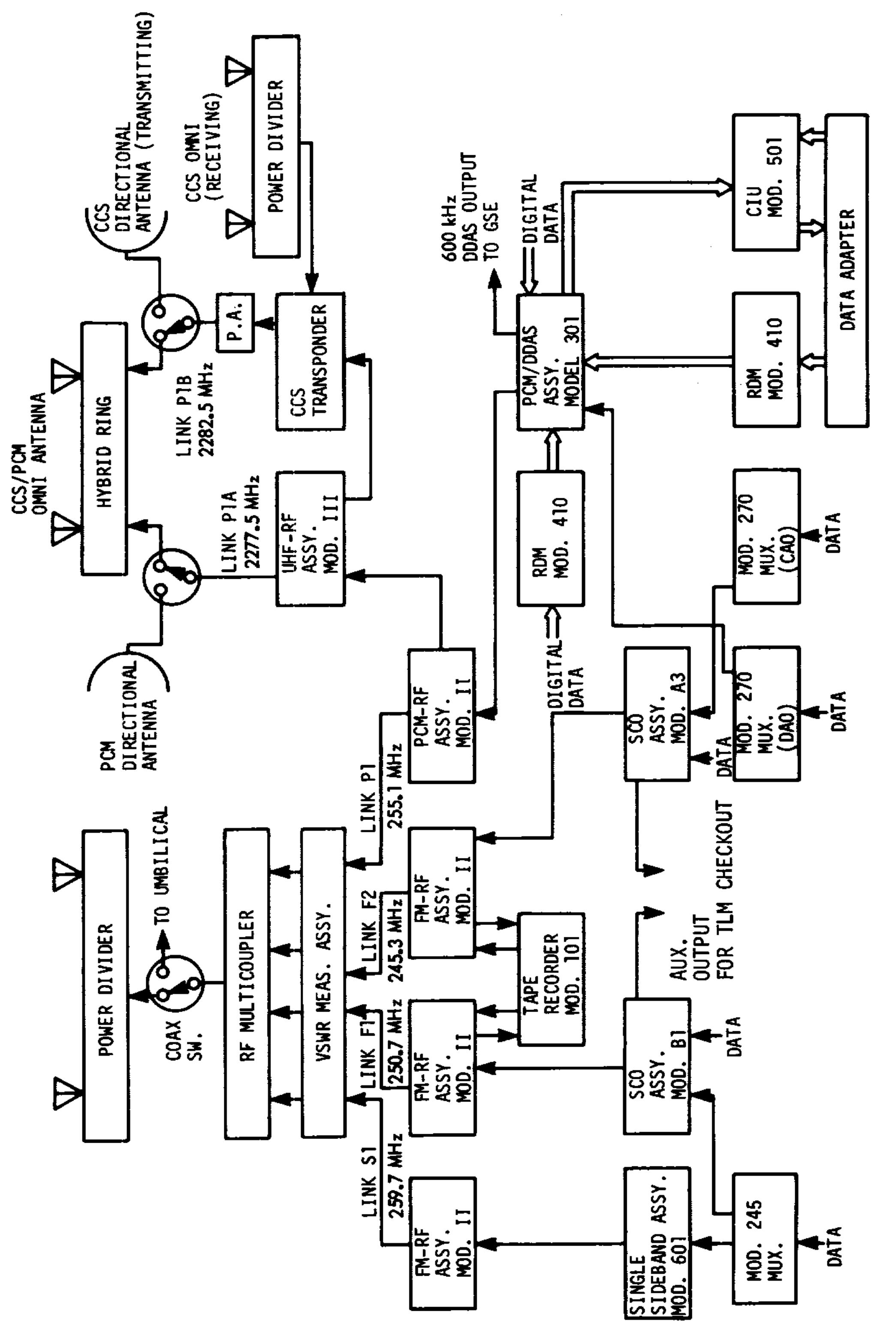

IU INSTRUMENTATION SYSTEM
VEHICLE ANTENNAS
POWER DIVIDER
ANTENNA SUBSYSTEMS
COAXIAL SWITCH
UMBILICAL
LOW POWER RF SIGNAL TO GSE
RF MULTICOUPLER
VEHICLE CHECKOUT ELECTRICAL SUPPORT EQUIPMENT
TAPE RECORDER
RF TRANSMITTER AND POWER AMPLIFIER
RF SIGNALS FROM OTHER TELEMETRY LINKS
DISPLAYS
TELEMETRY SUBSYSTEM
MULTIPLEXING MODULATION AND ENCODING
DDAS RECEIVING STATION
CHECKOUT COMPUTER
SIGNAL CONDITIONER
SIGNAL CONDITIONER
TRANSDUCER
TRANSDUCER
MEASURING SUBSYSTEM
REMOTE AUTOMATIC CALIBRATION SYSTEM

Figure 7-22

IU TELEMETER SYSTEM DATA FLOW



VII

Figure 7-23

120 times per second) multiplexer with provisions for submultiplexing individual primary channels to form 10 subchannels each sampled at 12 times per second. Twenty-seven of the 30 primary channels are used for analog data, while the remaining three are used for references. Ten-channel submultiplexer modules, which plug into the MUX-270, can be used to submultiplex any primary data channel, providing a 10 to 1 increase in the quantity of channels in exchange for a 10 to 1 decrease in sampling rates. Any proportion of the 23 data channels can be submultiplexed or sampled at the 120 per second rate.

The MUX-270 also has an integral calibration generator for in-flight calibration capability. Upon command the calibration generator seeks the next available master frame and applies a sequence of five calibration voltages to all data channels. Each level is sustained for one master frame and approximately 400 milliseconds are required for the full sequence.

The Model 245 multiplexer (MUX-245) is also a time sharing type. Each MUX-245 has provisions for up to 80 analog data inputs. Outputs are read out on 16 channels. Each of these 16 channels can provide a continuous single data signal or a time shared output of two, four or five data signals on a twelve second repetition cycle. The number of time shared outputs is dependant upon plug-in modules of which there are four types: the dummy module connects an input directly to an output channel; the two-channel, four-channel and five-channel modules provide time sharing as indicated by their numbers.

The Model 410 remote digital multiplexer (RDM-410) will accept ten 10-bit parallel words and transfer this data to the Model 301 PCM/DDAS according to a programmed format.

During launch vehicle staging, retrorockets create patches of ionized atmosphere which interfere with reliable transmission of telemetry data. To preclude loss of data or transmission of incorrect data telemetry transmission is stopped at these times and the data is put on magnetic recording tape for transmission at a more favorable time. The tape recorder which accomplishes this is the Model 101, a two channel unit. Tape speed is 60 inches per second with a recording time of 180 seconds.

Low level conditioned signals (0-5 volts) from the MUX-245 and MUX-270 being routed to the FM/FM telemetry links are fed to subcarrier oscillators (SCO). The Model A3 SCO has a capacity of 22 continuous data channel inputs. The Model B1 SCO has a capacity of 28 continuous data channel inputs. Each input is applied to a separate channel within the SCO and each channel produces a different output frequency. These output frequencies are combined within each SCO assembly and the composite signal is used to frequency modulate an FM-RF assembly producing the FM/FM telemetry signals.

The SS/FM telemetry link is made up of an FM-RF unit (amplitude) modulated by SS subcarriers. The Model 601 single sideband unit (SS-601) has a capacity of 15 channels. A MUX-245 furnishes inputs to the SS-601 which in turn modulates the FM-RF unit to produce the SS/FM telemetry link. The MUX-245 sequentially samples 75 signals and provides time sharing for the 15 channel SS-601 unit.

The PCM/FM system performs a dual function. During flight it serves as a telemetry link and during prelaunch checkout it serves as an IU interface with the digital GSE. PCM techniques provide the high degree of accuracy required for telemetering certain signal types. The PCM-301 unit accepts analog inputs from MUX-270 or RDM-410 units or direct inputs in digital form. All inputs are digitized and encoded. Output of the PCM-601 unit is a serial train of digital data which modulates the PCM-RF transmitter.

All of the RF assemblies are essentially the same. All use combinations of solid state and vacuum tube electronics. Frequency outputs of each unit are, of course, different and are applied to the antenna subsystem.

ANTENNA SUBSYSTEM

The antenna subsystem includes that equipment from the output of the RF units through the radiating or receiving elements.

VSWR MONITOR

Voltage standing wave ratio (VSWR) between the antenna and RF unit is checked and telemetered to the ground. The VSWR is a ratio of RF power being transmitted to RF power being reflected and is an indication of the efficiency of the RF unit, transmission coupling and antenna elements.

Multicoupler

The multicoupler provides a means of simultaneously coupling two, three or four RF signals into a common antenna without mutual interference and with maximum efficiency.

Coaxial Switch

The coaxial switch provides a means of connecting a coaxial transmission line to either of two other coaxial transmission lines.

Power Driver

The power divider provides a means for splitting RF power from one transmission line equally into two other transmission lines.

RADIO COMMAND SYSTEM

COMMAND COMMUNICATIONS SYSTEM (CCS)

The CCS provides for digital data transmission from ground stations to the LVDC. This communications link is used to up-date guidance information or command certain other functions through the LVDC. Command data originates in the Mission Control Center and is sent to remote stations of the MSFN for transmission to the launch vehicle.

At the time of spacecraft separation from the IU/S-IVB the IU CCS transmitter will be commanded off for a short period of time to preclude interference with the spacecraft S-band transponder. After adequate separation of the spacecraft and IU/S-IVB the IU CCS transmitter will be commanded on again to provide for psuedo-random noise (PRN) turnaround ranging to facilitate IU/S-IVB tracking.

The CCS equipment located in the IU consists of:

1. Antenna systems
   a. transmitting and receiving
   b. directional and omni
2. Antenna switching elements
   a. coaxial switches
   b. hybrid rings
   c. power divider
3. S-band transponder;
   a phase coherent receiver-transmitter
4. Command decoder, which precludes unauthorized command data entry

Figure 7-24 is a block diagram of the overall CCS. Command messages are transmitted from the unified S-band ground stations on a carrier frequency of 2101.8 MHz, modulated by a subcarrier of 70kHz which is modulated by a digital message. The transmitted message is received by

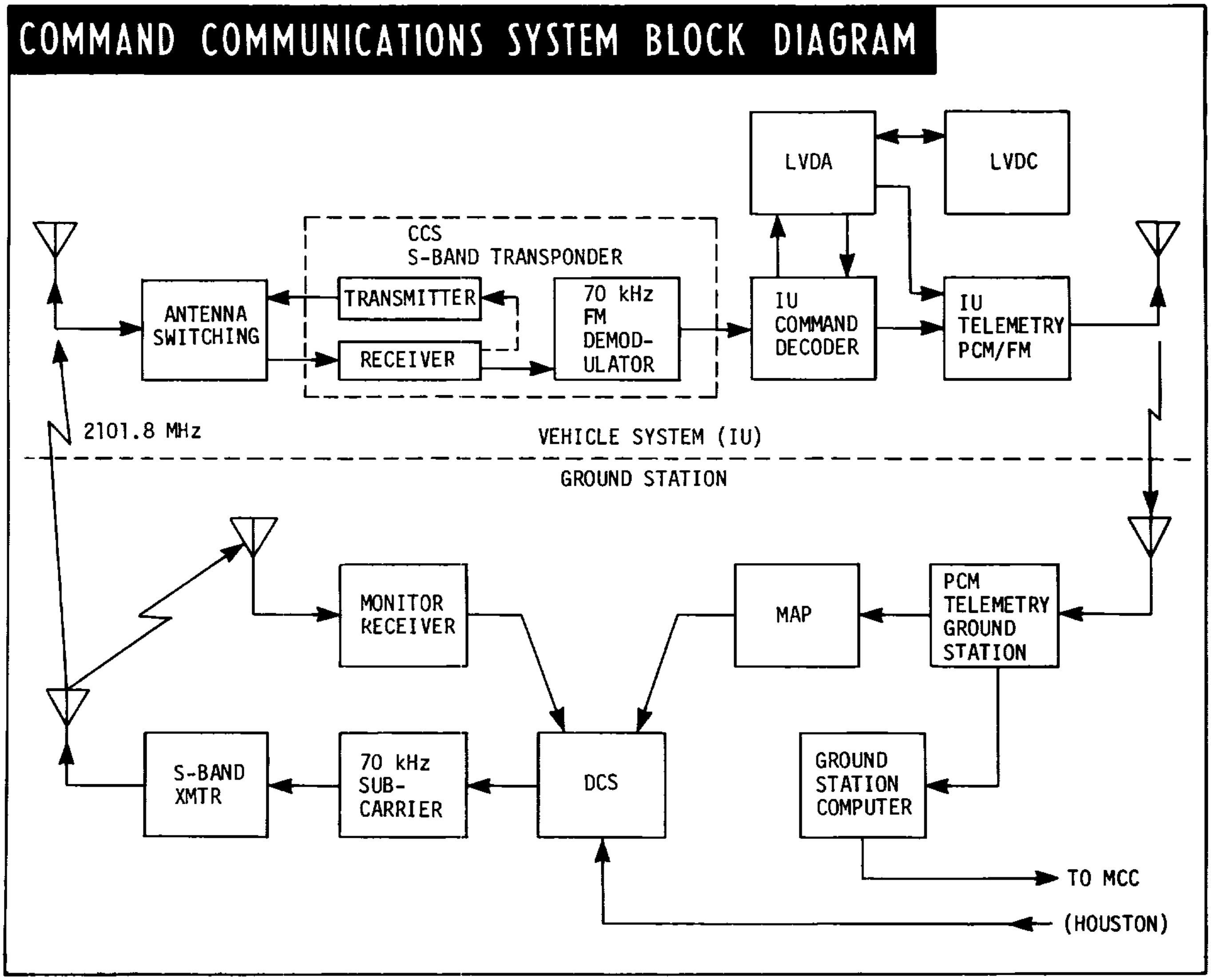


Figure 7-24

the airborne transponder where demodulation is accomplished. The resulting digital message is passed on to the command decoder where it is checked for authenticity before being passed to the LVDC. Verification of message receipt is accomplished through the IU PCM telemetry system.

The command decoder and the LVDC are programmed for the acceptance of seven different message types. These seven message types are:

1. Update LVDC
2. Execute update
3. Enter switch selector mode
4. Enter closed-loop test
5. Execute subroutine command (e.g., telemeter flight control measurements)
6. Memory sector dump
7. Telemeter single memory address

SATURN TRACKING INSTRUMENTATION

The purpose of radio tracking is the determination of the vehicle's trajectory. Tracking data is used for mission control, range safety, and post-flight evaluation of vehicle performance.

The Saturn V instrumentation unit carries two C-band radar transponders and an Azusa/GLOTRAC tracking transponder. A combination of tracking data from different tracking systems provides the best possible trajectory information and increased reliability through redundant data. The tracking of the Saturn launch vehicle may be divided into 4 phases: powered flight into earth orbit, orbital flight, injection into mission trajectory, and coast flight after injection.

Continuous tracking is required during powered flight into earth orbit. Because of the long burning time (11 minutes and 40 seconds) of the 3-stage Saturn V launch vehicle, the end of the

C-BAND TRANSPONDER CHARACTERISTICS	
Receiver Characteristics	
Frequency (Tunable externally)	5400 to 5900 MHz (set to 5690 ±8 MHz)
Frequency stability	±2.0 MHz
Bandwidth (3 db)	10 MHz
Off-frequency rejection	50 db image; 80 db minimum, 0.15 to 10,000 MHz
Sensitivity (99% reply)	-65 dbm over entire frequency range and all environments
Maximum input signal	-20 dbm
Interrogation code	Single or double pulse
Pulse width	0.2 to 5.0 μsec (single phase), 0.2 to 1.0 μsec (double pulse)
Pulse spacing	Continuously settable between 5 and 12 μsec (set to 8 ±0.05 μsec)
Decoder limits	±0.25 μsec accept, ± 0.85 μsec reject (5 to 12 μsec)
Transmitter Characteristics	
Frequency (Tunable externally)	5400 to 5900 MHz (set to 5765 ±2 MHz)
Peak power output	400 watts minimum, 700 watts nominal
Pulse width	1.0 ± 0.1 μsec
Pulse jitter	0.020 μsec maximum for signals above -55 dbm
Pulse rise time (10% to 90%)	0.1 μsec maximum
Duty cycle	0.002 maximum
VSWR of load	1.5:1 maximum
Pulse repetition rate	10 to 2000 pps; overinterrogation protection allows interrogation at much higher rates with count-down; replies during overinterrogation meet all requirements
Transponder Characteristics	
Recover time	50 μsec single pulse, 62 μsec double pulse maximum for input signal levels differing by up to 65 db (recovers to full sensitivity with no change in transmitter reply power or frequency with multiple radars interrogating simultaneously)
Fixed delay	Settable 2 ± 0.1 and 3.0 to 0.01 μsec (set to 3.0 ± 0.01 μsec)
Delay variation with signal level	50 nanoseconds maximum from -65 dbm to 0 dbm
Power requirements	24 to 30 volts
Primary current drain	0.7 ampere standby; 0.9 ampere at 1000 pps
Weight	5.5 lbs

Figure 7-25

powered flight phase cannot be covered sufficiently from land-based tracking stations. Therefore, tracking ships will be located in the Atlantic to obtain the tracking data during insertion which is required for orbit determination. The number of stations which can "see" the vehicle depends on the launch azimuth.

In addition, the Saturn launch vehicle will be tracked from S-band stations at Cape Kennedy and on the Atlantic tracking ships. These stations have dual tracking capability: i.e., they can simultaneously track the two S-band transponders on the vehicle; one in the IU and the other in the Apollo Spacecraft. The S-band station on Bermuda has only a single capability and will track the Apollo Spacecraft transponder. Refer to radio command systems for additional information on the S-band equipment.

During orbital flight, tracking is accomplished by S-band stations of the Manned Space Flight Network (MSFN) and by C-band radar stations. The S-band stations, including the Deep Space Instrumentation Facility, can track the Apollo Spacecraft to the moon and will also be involved in tracking after injection. Tracking information collected during orbital flight may be used to update the Saturn guidance before injection.

C-BAND RADAR

The function of the C-band radar transponder is to increase the range and accuracy of the radar ground stations equipped with AN/FPS-16, and AN/FPQ-6 radar systems. C-band radar stations at the Kennedy Space Center, along the Atlantic Missile Range, and at many other locations around the world, provide global tracking capabilities. Two C-band radar transponders are carried in the IU to provide radar tracking capabilities independent of the vehicle attitude. This arrangement is more reliable than the antenna switching circuits necessary if only one transponder would be used.

The transponder consists of a single compact package. Major elements include: an integrated RF head, an IF amplifier, a decoder, overinterrogation protection circuitry, a fast recovery solid-state modulator, a magnetron, a secondary power supply, and transducers for telemetry channels. The complete unit weighs 5.5 pounds and has a volume of only 100 cubic inches.

The transponder receives coded or single pulse interrogation from ground stations and transmits a single-pulse reply in the same frequency band.

Six conditioned telemetry outputs are provided: input signal level, input PRF, temperature, incident power, reflected power, and reply PRF.

The characteristics of the C-band radar transponder are given in figure 7-25.

AZUSA/GLOTRAC

The Azusa is an automatic, high-precision, real-time, trajectory-measuring electronic system that operates at microwave frequencies. The system consists of a vehicle-borne transponder and a ground-based tracking station. Two Azusa ground stations, designated Azusa Mark I and Azusa Mark II, are installed in the Atlantic Missile Range at Grand Bahama Island and at Cape Kennedy, respectively. Both systems are essentially the same. Therefore, only the Mark II will be discussed in this manual.

The Azusa ground station sends continuous trajectory data in digital form to the range computer. The computer derives the position coordinates of the trajectory. This data is presented to the range safety officer on automatic plotting boards. The trace appearing in real-time indicates the locus of points at which impact would occur should thrust be terminated at any time.

The position of the vehicle (transponder) is determined at the Azusa ground stations by measuring range (R) and two direction cosines (l and m) with respect to the antenna baselines. The antenna layout of the Azusa MK II station consists of two mutually perpendicular baselines as shown in figure 7-26. Antennas arranged along these baselines form X and Y coordinate axes. The transmitter antenna (T) radiates a CW signal at 5 GHz to the vehicle. This signal is offset by 60 MHz in the transponder and retransmitted to the ground station receiving antennas. The direction cosine, with respect to a baseline, is obtained from the measurement of the phase difference between transmitted and received signal. For range ambiguity resolution, the transmitted carrier is modulated with several low frequencies. See figure 7-27 for Azusa parameters and characteristics and figure 7-28 for definition of terms.

VII

The signals received at the two spaced antennas of an antenna pair have a phase difference caused by the difference in range between the transponder and each of the antennas. This phase difference is measured and used to compute the direction cosine. The accuracy of the mesurement increases with increasing baseline length, but data becomes ambiguous and coarse measurements are necessary for ambiguity resolution.

Direction cosine measurement is accomplished by using the antennas in pairs to provide baselines of 16.36 feet, 163.6 feet, and 1,636.0 feet as indicated in the table of figure 7-28. A conical scan antenna (DF) yields unambiguous direction measurement and furnishes ambiguity resolution for the 16.36 feet baselines. The 16.36 feet baselines resolve ambiguity for the more accurate 163.6 feet baselines, and the 1,636.0 feet baselines supply information for computing cosine rate data. The conical scan direction finder antenna (DF) provides pointing information for all other antennas.

Range is determined at the ground station by comparing the instantaneous phase of the transmitted modulation signal to that of the received modulation signal. The resulting phase difference is directly proportional to the propagation time from the ground station to the transponder and back to the ground station. In other words, phase difference is proportional to range. Using a high range modulation frequency to obtain fine resolution, results in output data that are ambiguous, the number of ambiguities is proportional to the modulation frequency.

The ambiguities in range measurement are resolved electronically by using three modulation frequencies that are obtained by the frequency-division of a single, precise, frequency source at the ground station. The phase shift is measured for each of these harmonically related signal pairs with the lower frequency phase data being used to resolve the ambiguities in the next higher frequency phase data. The modulation signals used for range measurement are 157.4 Hz, 3.934 kHz, and 98.356 kHz. The fine range modulation signal, 98.356 kHz, remains on at all times so the transponder can "lock-on" with the ground station.

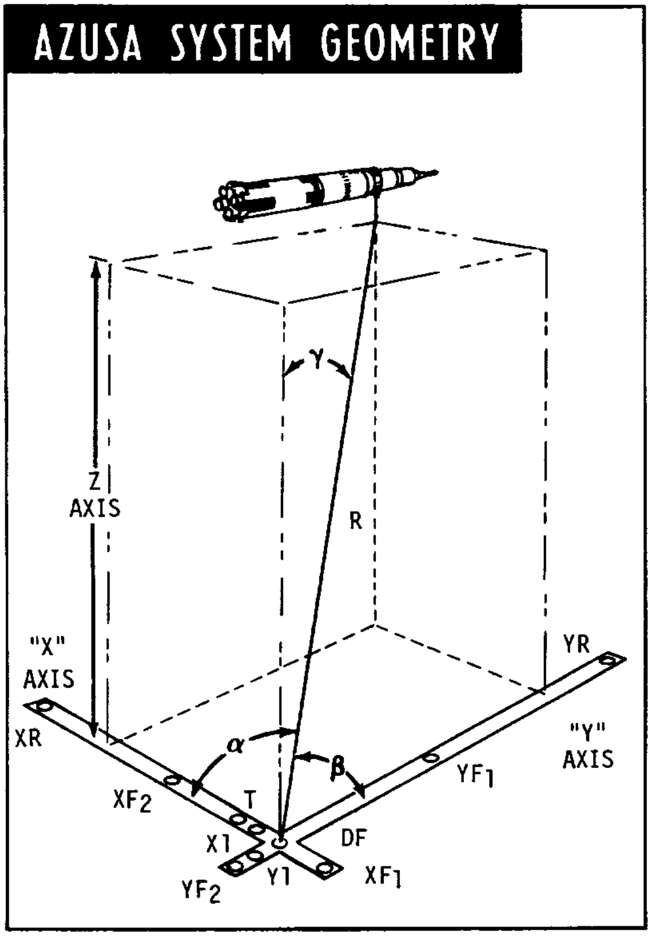


Figure 7-26

Data and reference signals are fed into a discriminator which provides one pulse (count) output for each 360-degree phase difference (1 cycle) in the input circuit. Plus and minus pulses are fed on separate lines to a bidirectional counter. The coherent carrier range data is then fed to an IBM 7090 computer with the direction cosine data (1, m) and modulation-derived slant range data. The computer, using 20 input samples per second, solves the equations for the position of the vehicle.

TRANSPONDER

The type-C transponder uses transistors in all circuitry except for the klystron. The type-C transponder operates also with GLOTRAC stations described briefly in the following paragraphs.

GLOTRAC

GLOTRAC ground stations are equipped with either a transmitter or a receiver or both. Both existing Azusa stations may be considered as part of GLOTRAC. The transponder in the vehicle is interrogated by an Azusa ground station or by a GLOTRAC transmitter site. The transponder offsets the received frequency and retransmits the signal to GLOTRAC receiving sites where the doppler shift is measured by comparing the received signal with the transmitter signal (if the receiver is located near the transmitter) or with a local frequency source. At GLOTRAC stations equipped with both a transmitter and receiver, the range to the transponder is measured by phase comparison between the transmitted and received signals. The Azusa type-C transponder can also be interrogated by C-band radars for range and angle determination.

The range rates measured at three receiving stations yield the vehicle velocity, and by integrating this velocity, the position is obtained. Initial conditions for integration are obtained from radar range measurements. Data measured at all stations is transmitted to the computer at Cape Kennedy. Accuracy of GLOTRAC measurements are approximately 98.4 feet in position and 0.5 feet/second in velocity.

GROUND SUPPORT EQUIPMENT

The IU, because of its complex nature, requires the services of many types of ground support equipment (GSE); mechanical, pneudraulic, electrical, electronic, and personnel. This section of the manual is limited to a very brief description of the IU Ground Support Equipment.

There are three primary interfaces between the IU and its GSE. One is the IU access door used during prelaunch preparations for battery installation, ordnance servicing, servicing IU equipment, S-IVB forward dome and LEM servic-

AZUSA TERMS AND DEFINITIONS	
TERM	DEFINITION
BASELINE	The line defined by two antennas supplying data for phase comparison. There are three colinear baselines (intermediate, 16.36 feet; fine, 163.6 feet; rate or incremental, 1,636.0 feet) on each axis.
COHERENCE, CARRIER	Term applied to the condition wherein the instantaneous phase of the output carrier of the Azusa transponder is directly related to the instantaneous phase of the input carrier and its fine modulation signal.
COSINE, DIRECTION	The cosine of the included angle between a line starting at the transponder and passing through the intersection of the two axes, and the extension of the positive X-asis, is called X direction cosine. Y direction cosine is determined in a similar manner by using an extension of the Y-axis.
DATA, AMBIGUOUS	Azusa data without a point of reference. A mechanical analogy would be the measurements made on a micrometer where the divisions on the vernier scale of the thimble are not valid without the index provided by the scale on the sleeve.
DATA, FINE COSINE	Digital or analog information representing the cosine of the direction angle or as determined at the163.6 foot baseline antennas. To qualify as "fine data," ambiguities, due to the large number of cycles at the measurement frequency, must have been resolved to provide accurate data with high resolution.
DATA, FINE RANGE	Digital or analog information representing the distance from the baseline intersection to the transponders as determined by using the 98.356 KHz modulation frequency. To qualify as "fine data," ambiguities, due to the large number of cycles at the fine range modulation frequency, must have been resolved to provide accurate data with high resolution.
DATA, INCREMENTAL COSINE	Similar to fine cosine data except that resolution obtained by using 1,636.0 foot baselines is higher and ambiguities are not resolved. The computer operates on this data to derive a measure of the rate of change or direction cosines.
DATA, INCREMENTAL RANGE	Similar to fine range data except that the carrier frequency is used to provide increased resolution. Ambiguities are not resolved. The computer operates on this data to derive a measure of missile radial velocity.
DATA, INTERMEDIATE	Cosine or range --Provides resolution, better than that obtained with coarse data, but below that obtained with fine data. Data is unambiguous and is used to resolve fine data ambiguities.
DATA, PHASE	Data obtained by measuring the phase difference between two signals having a common frequency.
RESOLUTION, AMBIGUITY	The determination of the exact cycle, at the measurement frequency, at which the equipment is comparing phase angle. Ambiguities are resolved by comparing the phase of a data signal with that of a reference signal in a "three speed" servo system. Starting with unambiguous data, successive comparisons are made to resolve ambiguities in the next "higher speed" data. The process is analogous to mechanical vernier measurements.
SIGNAL, INTERROGATION	Ground-transmitted signal.
SIGNAL, RESPONSE	Transponder-transmitted signal, offset 60.2 MHz from the transponder-received signal.

Figure 7-27

AZUSA PARAMETERS AND CHARACTERISTICS

SYSTEM GEOMETRY

R = slant range (best value from refraction corrected r_m or r_{cc} axis)

α = direction angle with respect to X-axis

β = direction angle with respect to Y-axis

γ = direction angle with respect to Z-axis

MEASURED PARAMETERS

$l = \cos\alpha = \frac{x}{R}$ - Data from 163.6 foot x-baseline

$m = \cos\beta = \frac{y}{R}$ - Data from 163.6 foot Y-baseline

r = uncorrected slant range (either r_m or r_{cc})

r_m = modulation slant range (nonambiguous)

r_{cc} = incrementally derived range

l_e = incremental cosine from extended 1,636.0 foot X-baseline

m_e = incremental cosine from extended 1,636.0 foot Y-baseline

Δt = time increment between data samples

COMPUTED PARAMETERS

$$n = \cos\gamma = \sqrt{1 - l^2 - m^2}$$

$$\dot{r}_{cc} = \frac{1}{2\Delta t}\left(r_{cc_{i+1}} + r_{cc_{i-1}}\right) = \text{radial rate}$$

$$\dot{l}_e = \frac{1}{2\Delta t}\left(l_{e_{i+1}} - l_{e_{i-1}}\right) = \text{cosine rate}$$

$$\dot{m}_e = \frac{1}{2\Delta t}\left(m_{e_{i+1}} - m_{e_{i-1}}\right) = \text{m cosine rate}$$

$$\dot{n} = -\frac{l\dot{l}_e + m\dot{m}_e}{n}$$

$$X = Rl \qquad \dot{X} = R\dot{l}_e + l\dot{r}_{cc}$$

$$Y = Rm \qquad \dot{Y} = R\dot{m}_e + m\dot{r}_{cc}$$

$$Z = Rn \qquad \dot{Z} = R\dot{n} + n\dot{r}_{cc}$$

Figure 7-28 (Sheet 1 of 2)

AZUSA PARAMETERS AND CHARACTERISTICS

Transponder Type-C

Receiver frequency........	5060.194 MHz
Transmitter frequency.....	5000.000 MHz
RF power output...........	2.5 watts
Input voltage.............	28 vdc
Input current.............	5 amperes
Receiver input signal.....	-12 to -90 dbm
Weight....................	19.3 lbs
Size......................	372 inches^3

AZUSA Ground Station Mark II

Transmitted power.........	2 kw
Transmitter frequency.....	5060.2 MHz $\pm$0.75 MHz
Ranging modulation........	157.4 Hz, 3.934 kHz, 98.351 kHz
Receiver frequency........	5000 MHz
Receiver sensitivity......	-145 to -147 dbm
Receiver antenna gain.....	33 db (MK II)
Accuracies:	
Range..................	9.98 feet
Angle..................	1 x 10^{-5} in cosine data

AZUSA Characteristics

$$l = \cos = \frac{L_x}{R}$$

$$m = \cos = \frac{L_y}{R}$$

T: Transmitter Antenna
DF: Direction Finder Antenna (Receiving)
XF, XR, YF, YR: Receiving Antennas
L: Distance between antennas

Antennas	Baseline Length L_x, L_y	Measurements
DF - X1	16.36 ft.	Intermediate cos
XF1 - XF2	163.6 ft.	Fine cos
XF1 - XR	1,636.0 ft.	Rate cos
DF - Y1	16.36 ft.	Intermediate cos
YF2 - YF1	163.6 ft.	Fine cos
YF2 - YR	1,636.0 ft.	Rate cos
Geometrical Definitions		

Figure 7-28 (Sheet 2 of 2)

ing. The second interface is the umbilical through which the IU is furnished with ground power, purging air/GN_2, methanol-water for environmental control, and hardwire links with electrical/electronic checkout equipment. The third interface is the optical window through which the guidance system ST-124-M3 stable platform is aligned.

IU ACCESS DOOR

The structure of the IU consists of three 120-degree segments of aluminum honeycomb sandwich joined to form a cylindrical ring. After assembly of the IU, a door assembly provides access to the electronic equipment inside the structure. This access door has been designed to act as a load supporting part of the structure in flight.

Work platforms, lights, and air-conditioning are used inside the IU to facilitate servicing operations. When the spacecraft is being fueled through the IU access door, a special protective cover is installed inside the IU to protect components from any possible volatile fuel spillage.

Approximately 20 hours before launch the IU flight batteries, each weighing 165 pounds, are activated in the battery shop and installed in the IU through the access door.

At approximately T-6 hours, the service equipment is removed and the access door is secured.

IU UMBILICAL

The physical link between the IU and the GSE is through the umbilical connection located adjacent to the access door. The umbilical is made up of eighteen electrical connectors, two pneudraulic couplings and an air conditioning duct. The electrical connectors provide ground power and the electrical/electronic signals necessary for prelaunch checkout of the IU equipment. The pneudraulic couplings provide for circulation of GSE supplied methanol-water coolant fluid for the IU/S-IVB ECS. The air conditioning duct provides for compartment cooling air or purging GN_2.

The umbilical is retracted at liftoff and a spring loaded door on the IU closes to cover the connectors.

OPTICAL ALIGNMENT

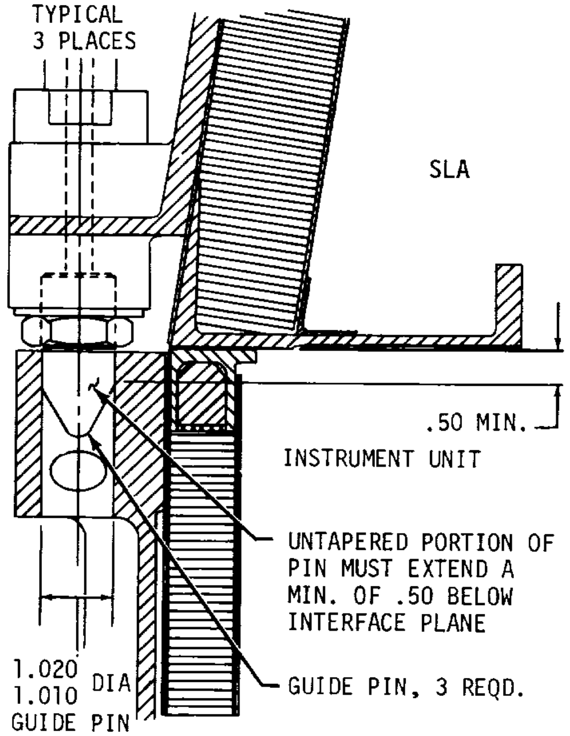
The IU contains a window through which the ST-124-M3 stable platform has its alignment checked and corrected by a theodolite located in a hut on the ground and a computer feedback loop. By means of this loop the launch azimuth can be monitored, updated and verified to a high degree of accuracy.

IU/SLA INTERFACE

MECHANICAL INTERFACE

The IU and spacecraft-LM adapter (SLA) are mechanically aligned with three guide pins and brackets as shown in figure 7-29. These pins facilitate the alignment of the close tolerance interface bolt holes as the two units are joined during vehicle assembly. Six bolts are installed around the circumference of the interface and sequentially torqued using a special MSFC designed wrench assembly. These six bolts secure the IU/SLA mechanical interface. (See figure 7-30).

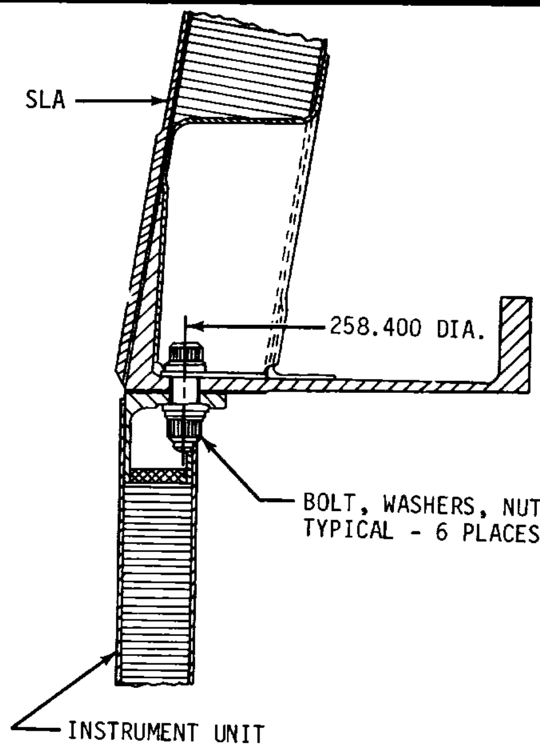
IU/SLA ALIGNMENT



ALL DIMENSIONS ARE IN INCHES

Figure 7-29

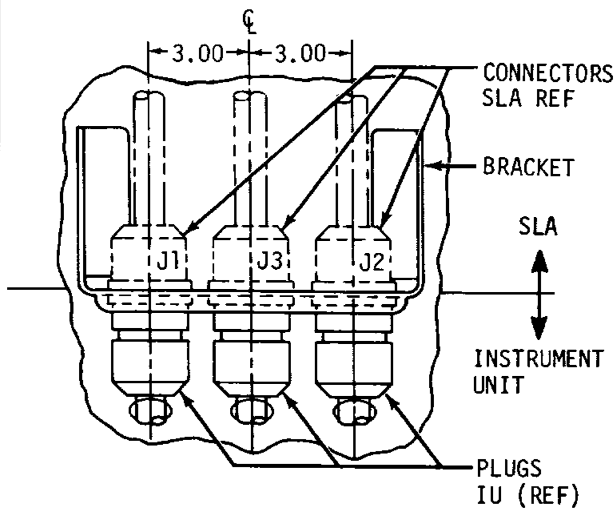
IU/SLA MECHANICAL ATTACHMENT



ALL DIMENSIONS ARE IN INCHES

Figure 7-30

IU/SLA INTERFACE ELECTRICAL CONNECTORS



ALL DIMENSIONS ARE IN INCHES

Figure 7-31

ELECTRICAL INTERFACE

The electrical interface between the IU and spacecraft consists of three 61 pin connectors (see figure 7-31). The definition and function of each connector is presented in the following paragraphs.

IU/Spacecraft Interface Connector J-1

This connector provides lines for power, control, indication circuitry and emergency detection system (EDS) circuitry working off the EDS bus No. 1 and IU bus +6D91.

IU/Spacecraft Interface Connector J-2

This connector provides lines for power, control, and indications for the Q-ball circuitry and the EDS circuitry working off the EDS bus No. 2 and IU bus +6D02.

IU/Spacecraft Interface Connector J-3

This connector provides lines for power, control, and indication circuitry and EDS circuitry working off the EDS bus No. 3 and IU bus +6D03.

MAJOR DIFFERENCES BETWEEN SATURN V IU-3 AND IU-4

1 ▶ *IU equipment changes consist of deletions as follows:*

Measuring Rack, 6 places
Source Follower
D20 Battery
F2 RF Ass'y.
F2 TLM Ass'y.
Slow Speed MUX
Tape Recorder
Measuring Distributor
P1 RF Ass'y.
Thermal Probe
Rate Gyro Timer

2 ▶ *Quantity of instrumentation measurements reduced from 339 to 221. Instrumentation measurements converted from R&D to Operational.*

VII

SECTION VIII

GROUND SUPPORT INTERFACE

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LAUNCH COMPLEX 39

Launch Complex 39 (LC-39), located at Kennedy Space Center, Florida, is the facility provided for the assembly, checkout and launch of the Apollo/Saturn V space vehicle. Assembly and checkout of the vehicle is accomplished on a mobile launcher, in the controlled environment of the vehicle assembly building. The space vehicle and the launch structure are then moved as a unit by the crawler-transporter to the launch site, where vehicle launch is accomplished after propellant loading and final checkout. The major elements of the launch complex shown in figure 8-1, are the vehicle assembly building (VAB), the launch control center (LCC), the mobile launcher (ML), the crawler-transporter (C-T), the crawlerway, the mobile service structure (MSS), and the launch pad.

LC-39 FACILITIES AND EQUIPMENT

VEHICLE ASSEMBLY BUILDING

The VAB is located adjacent to Kennedy Parkway about five miles north of the KSC industrial area. Its purpose is to provide a protected environment for receipt and checkout of the propulsion stages and instrument unit (IU), erection of the vehicle stages and spacecraft in a vertical position on the ML, and integrated checkout of the assembled space vehicle.

The VAB, as shown in figure 8-2 is a totally enclosed structure covering eight acres of ground. It is a structural steel building approximately 525 feet high, 518 feet wide, and 716 feet long. The siding is insulated aluminum except where translucent fiberglass sandwich panels are used in part of the north and south walls.

The principal operational elements of the VAB are the low bay area and high bay area. A 92 foot wide transfer aisle extends through the length of the VAB and divides the low and high bay areas into equal segments. (See figure 8-3.)

Low Bay Area

The low bay area provides the facilities for receiving, uncrating, checkout, and preparation of the S-II stage, S-IVB stage, and the IU. The low bay area, located in the southern section of the VAB, is approximately 210 feet high, 442 feet wide, and 274 feet long. There are eight stage preparation and checkout cells, four of which are equipped with systems to simulate interface operations between stages and the IU.

Work platforms, made up of fixed and folded sections, fit the various stages as required. The platforms are bolted to the low bay structure to permit vertical repositioning for changes in stage configuration. Access from fixed floor levels to the work platforms is provided by stairs.

Transfer Aisle

The transfer aisle provides the means for unloading the stages and movement of the stages between the high and low bay areas. A 175-ton bridge crane, with a hook height of 166 feet above ground floor level, serves the transfer aisle.

The motor driven transfer aisle doors are of the sliding type. The doors provide an opening 55 feet wide and 96 feet high on the south end of the low bay area and 55 feet wide and 57 feet high on the north end of the high bay area.

High Bay Area

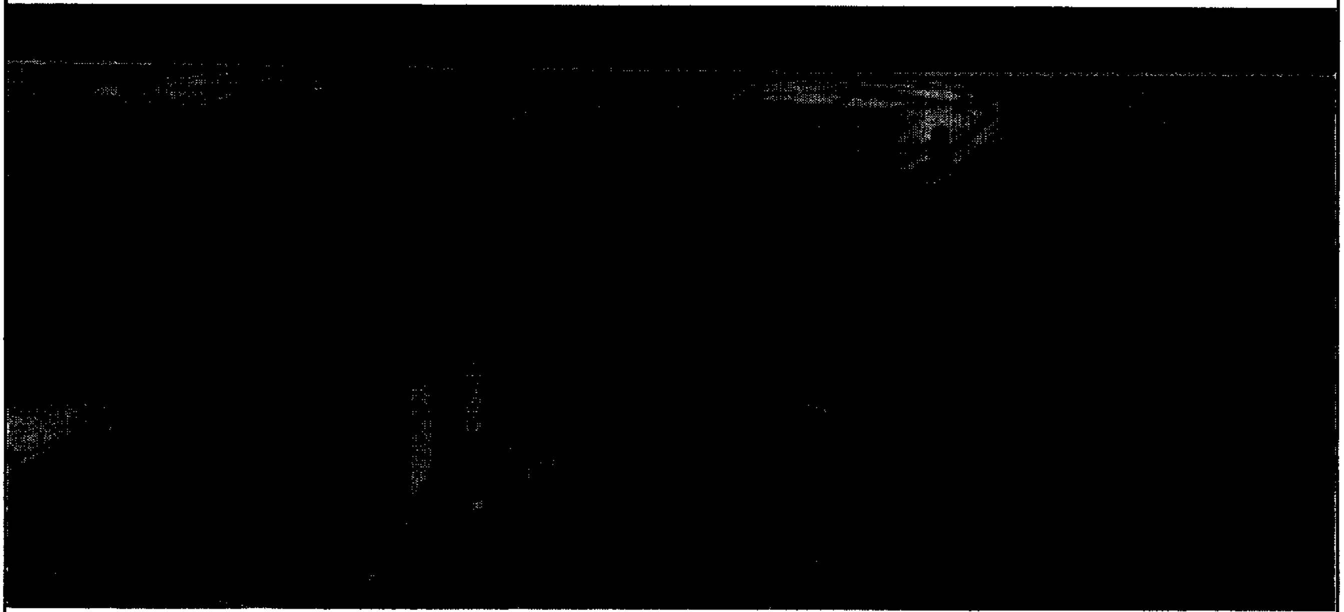
The high bay area provides the facilities for erection and checkout of the S-IC stage; mating and erection operations of the S-II stage, S-IVB stage, IU, and spacecraft; and integrated checkout of the assembled space vehicle. The high bay area is located in the northern section of the building.

The high bay area is approximately 525 feet high, 518 feet wide, and 442 feet long. It contains four checkout bays, each capable of accommodating a fully assembled, Saturn V space vehicle.

Access to the vehicle at various levels is provided from air-conditioned work platforms that extend from either side of the bay to completely surround the launch vehicle. Each platform is composed of two biparting sections which can be positioned in the vertical plane. The floor and roof of each section conform to and surround the vehicle. Hollow seals on the floor and roof of the section provide an environmental seal between the vehicle and the platform.

Each pair of opposite checkout bays is served by a 250-ton bridge crane with a hook height of 462

LAUNCH COMPLEX 39



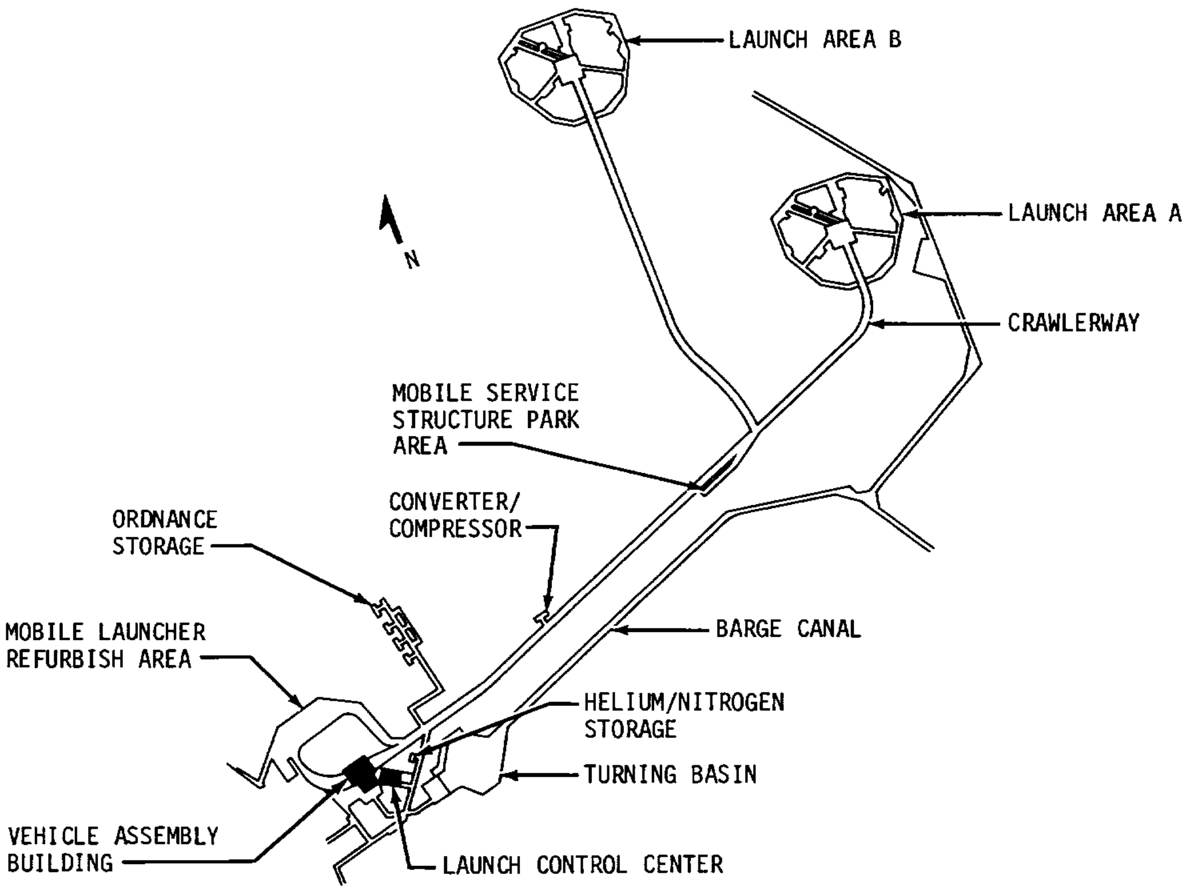


Figure 8-1

VEHICLE ASSEMBLY BUILDING

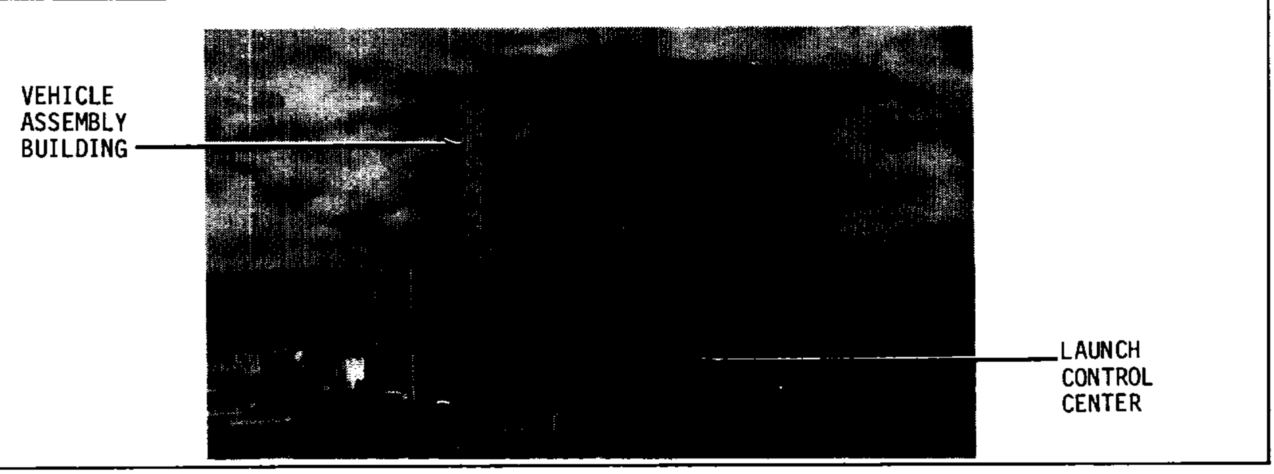


Figure 8-2

VAB INTERIOR

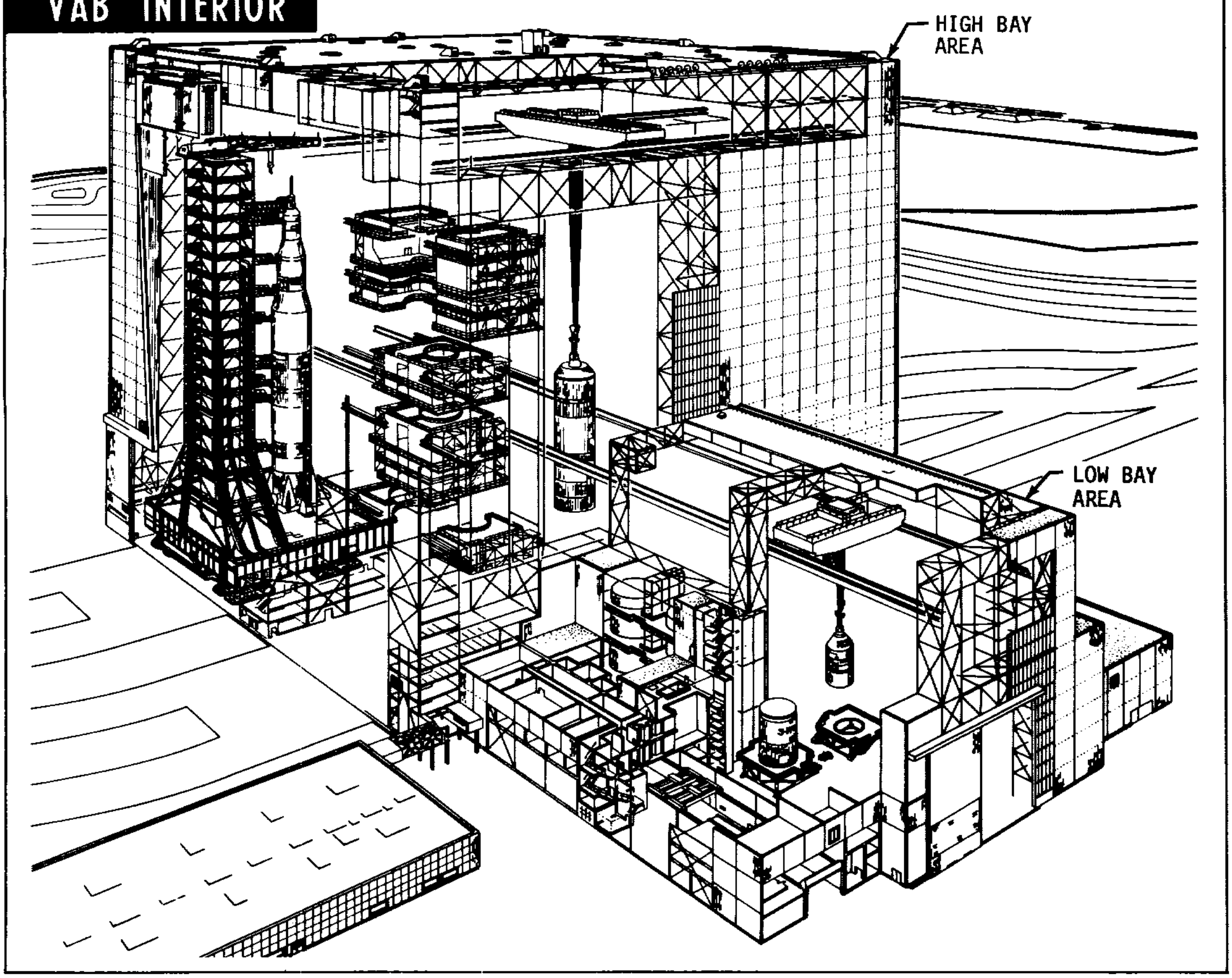


Figure 8-3

feet. The wall framing between the bays and the transfer aisle is open above the 190-foot elevation to permit movement of components from the transfer aisle to their assembly position in the checkout bay. The bay areas are ventilated and have nitrogen and helium gas concentration detection systems.

The high bay doors provide an inverted T-shaped opening 456 feet in height. The lower portion of the opening is closed by doors which move horizontally on tracks. The upper portion of the opening is closed by seven vertically moving doors.

Utility Annex

The utility annex, located on the west side of the VAB, supports the VAB, LCC and other facilities in the VAB area. It provides air conditioning, hot water, compressed air, water for fire protection and emergency electrical power.

Helium/Nitrogen Storage-VAB Area

The gas storage facility at the VAB provides high-pressure gaseous helium and nitrogen. It is located east of the VAB and south of the crawlerway. The roof deck of the building is removable to permit installation and removal of pressure vessels through the roof. This facility is serviced from the converter/compressor facility by a 6,000 psig gaseous helium line and a 6,000 psig gaseous nitrogen line.

LAUNCH CONTROL CENTER

The LCC, figure 8-4, serves as the focal point for overall direction, control, and surveillance of space vehicle checkout and launch. The LCC is located adjacent to the VAB and at a sufficient distance from the launch pad (three miles) to permit the safe viewing of liftoff without requiring site hardening. An enclosed personnel and cabling bridge connects the VAB and LCC at the third floor level.

The LCC is a four-story structure approximately 380 by 180 feet. The ground floor is devoted to service and support functions such as cafeteria, offices, shops, laboratories, the communications control room, and the complex control center. The second floor houses telemetry, RF and tracking equipment, in addition to instrumentation and data reduction facilities.

The third floor is divided into four separate but similar control areas, each containing a firing room, computer room, mission control room, test conductor platform area, visitor gallery, offices and frame rooms. The four firing rooms, one for each high bay in the VAB, contain control, monitoring and display equipment for automatic vehicle checkout and launch.

Direct viewing of the firing rooms and the launch area is possible from the mezzanine level through specially designed, laminated, and tinted glass windows. Electrically controlled sun louvers are positioned outside the windows.

The display rooms, offices, launch information exchange facility (LIEF) rooms, and mechanical equipment are located on the fourth floor.

The electronic equipment areas of the second and third floors have raised false floors to accommodate interconnecting cables and air-conditioning ducts.

The power demands in this area are large and are supplied by two separate systems, industrial and instrumentation. The industrial power system supplies electric power for lighting, general use receptacles, and industrial units such as air conditioning, elevators, pumps and compressors. The instrumentation power system supplies power to the electronic equipment, com-

LCC EXTERIOR

Figure 8-4

puters, and related checkout equipment. This division between power systems is designed to protect the instrumentation power system from the adverse effects of switching transients, large cycling loads, and intermittent motor starting loads. Communication and signal cable provisions have been incorporated into the design of the facility. Cable troughs extend from the LCC via the enclosed bridge to each ML location in the VAB high bay area. The LCC is also connected by buried cableways to the ML refurbishing area and to the pad terminal connection room (PTCR) at the launch pad. Antennas on the roof provide an RF link to the launch pads and other facilities at KSC.

MOBILE LAUNCHER

The mobile launcher (figure 8-5) is a transportable steel structure which, with the crawler-transporter, provides the capability to move the erected vehicle to the launch pad. The ML is divided into two functional areas, the launcher base and the umbilical tower. The launcher base is the platform on which a Saturn V vehicle is assembled in the vertical position, transported to a launch site, and launched. The umbilical tower, permanently erected on the base, is the means of ready access to all important levels of the vehicle during the assembly, checkout, and servicing periods prior to launch. The equipment used in the servicing, checkout, and launch is installed throughout both the base and tower sections of the ML. The intricate vehicle-to-ground interfaces are established and debugged in the convenient and protected environment of the VAB, and move undisturbed aboard the ML to the pad.

Launcher Base

The launcher base (figure 8-6) is a two story steel structure 25 feet high, 160 feet long, and 135 feet wide. Each of the three levels provides approximately 12,000 square feet of floor space. The upper deck, designated level 0, contains, in addition to the umbilical tower, the four holddown arms and the three tail service masts. Level A, the upper of the two internal levels, contains 21 compartments and level B has 22 compartments. There is a 45-foot square opening through the ML base for first stage exhaust.

Access to the base interior is provided by personnel/equipment access doors opening into levels A and B and equipment access hatches located on levels 0 and A.

The base has provisions for attachment to the crawler-transporter, six launcher-to-ground mount mechanisms, and four extensible support columns.

All electrical/mechanical interfaces between vehicle systems and the VAB or the launch site are located through or adjacent to the base structure. A number of permanent pedestals at the launch site provide support for the interface plates and servicing lines.

The base houses such items as the computer systems test sets, digital propellant loading equipment, hydraulic test sets, propellant and pneumatic lines, air-conditioning and ventilating systems, electrical power systems, and water systems. Shock-mounted floors and spring supports are provided so that critical equipment receives less than $\pm$ 0.5 g mechanically-induced vibrations. Electronic compartments within the ML base are provided with acoustical isolation to reduce the overall rocket engine noise level.

The air-conditioning and ventilating system for the base provides environmental protection for the equipment during operations and standby. One packaged air-conditioner provides minimal environmental conditioning and humidity control during transit. Fueling operations at the launch area require that the compartments within the structure be pressurized to a pressure of three inches of water above atmospheric pressure and that the air supply originate from a remote area free from contamination.

The primary electrical power supplied to the ML is divided into four separate services: instrumentation, industrial, in-transit and emergency. Instrumentation and industrial power systems are separate and distinct. During transit, power from the crawler-transporter is used for the water/glycol systems, computer air-conditioning, threshold lighting, and obstruction lights. Emergency power for the ML is supplied by a diesel-driven generator located in the ground facilities. It is used for obstruction lights, emergency lighting, and for one tower elevator. Water is supplied to the ML at the VAB and at the pad for fire, industrial and domestic purposes and at the refurbishment area for domestic purposes.

Umbilical Tower

The umbilical tower is an open steel structure 380 feet high which provides the support for eight umbilical service arms, one access arm, 18 work and access platforms, distribution equipment for the propellant, pneumatic, electrical and instrumentation subsystems, and other ground support equipment. The distance from the vertical centerline of the tower to the vertical centerline of the vehicle is approximately 80 feet. The distance from the nearest vertical column of the tower to the vertical centerline of the vehicle is approximately 60 feet. Two

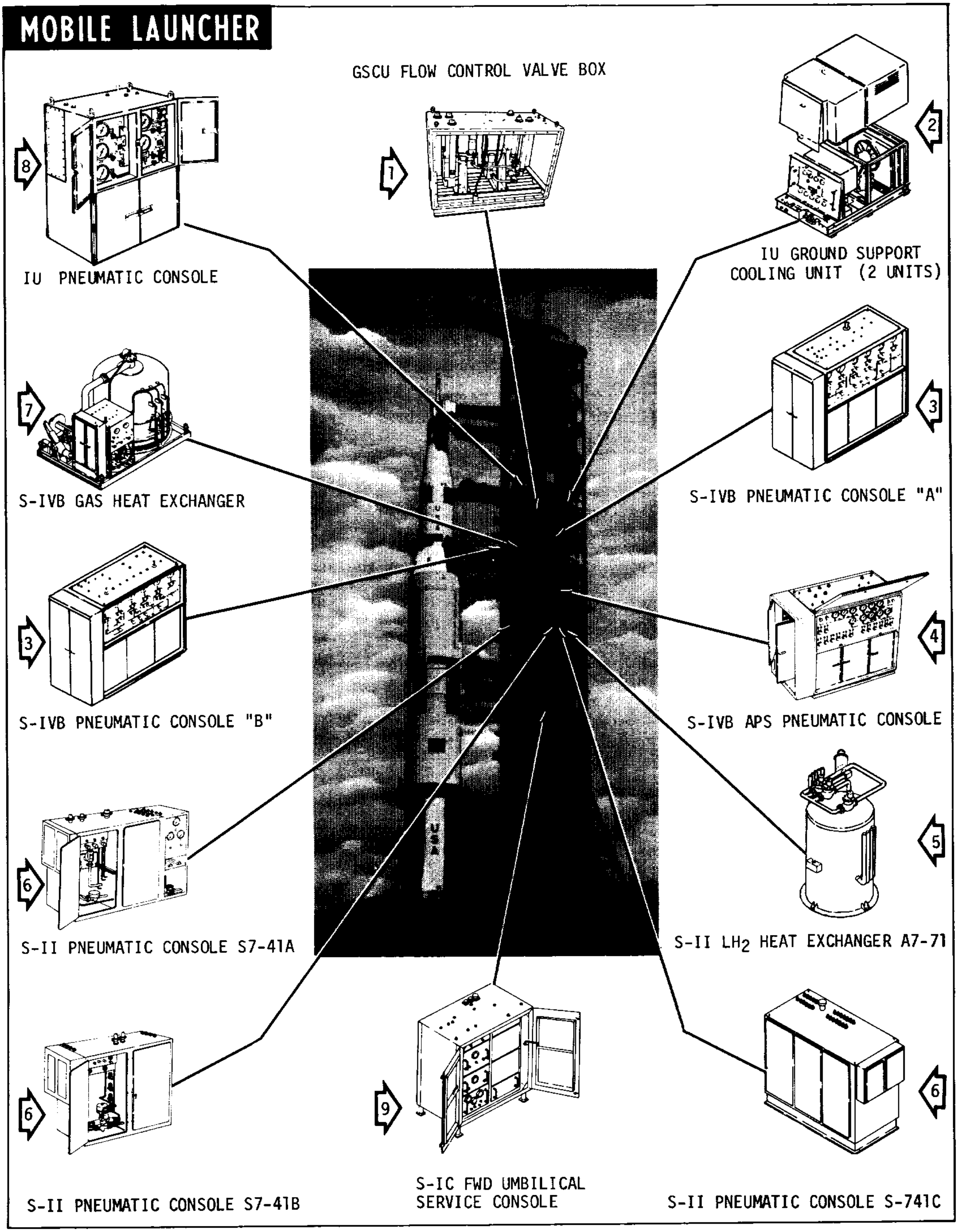


Figure 8-5 (Sheet 1 of 2)

MOBILE LAUNCHER

1 GSCU Flow Control Valve Box
Selects either GSCU for operation of one unit while the other recirculates.

2 Ground Support Cooling Unit
Supplies water-methanol to the heat exchanger in the IU thermal conditioning system to absorb heat in the IU generated by electronic equipment.

3 S-IVB Pneumatic Console A&B
Regulates and controls helium and nitrogen gases for leak testing, functional checkout, propellant loading, purge, and propellant unloading.

4 S-IVB APS Pneumatic Console
Regulate and distribute helium and nitrogen gases during checkout and propellant loading.

5 S-II LH_2 Heat Exchanger A7-71
Provides gases to the S-IC stage for the following:
1. Fuel tank pressurization
2. LOX tank pre-pressurization
3. Thrust Chamber jacket childown

6 S-II Pneumatic Consoles S7-41A, B, & C
Regulate, control, and monitor gases for S-II stage during standby, prelaunch, and launch.

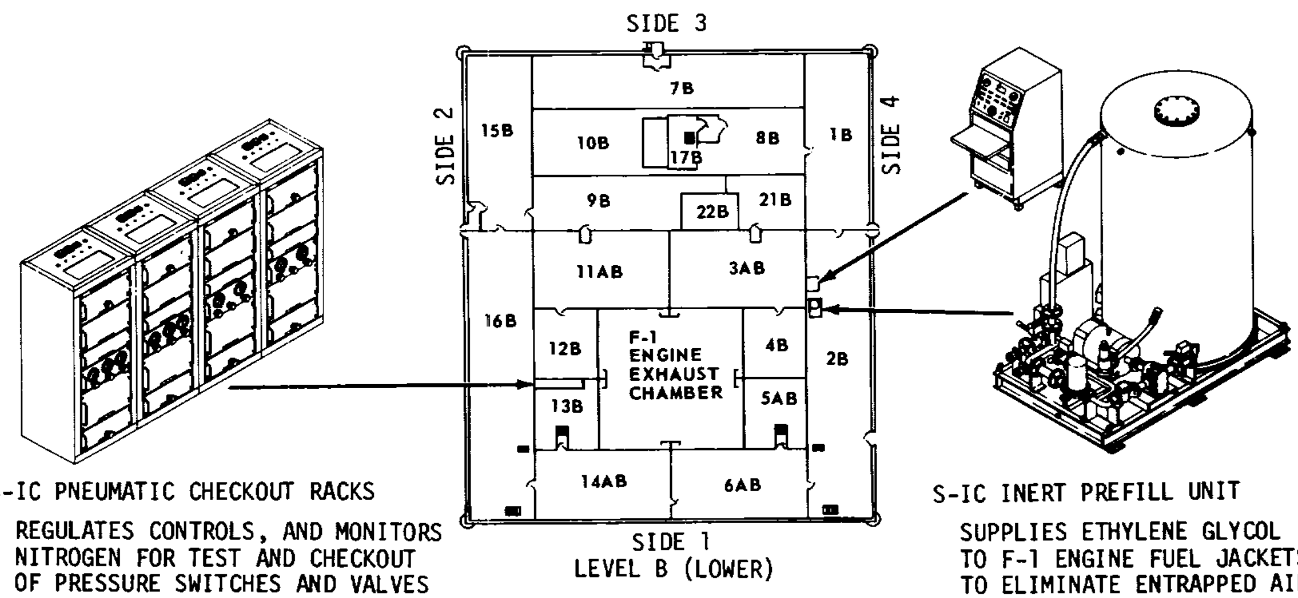
7 S-IVB Gas Heat Exchanger
Supplies cold helium or hydrogen for the following:
1. Lox and Fuel Tank Pre-Pressurization
2. Thrust chamber jacket childown
3. Pressurize engine turbine start bottle

8 IU Pneumatic Console
Regulates, monitors, and controls pneumatic pressure to pressurize, checkout, and test the air bearing spheres and related pneumatic and electro-mechanical circuitry.

9 S-IC Forward Umbilical Service Console
Supplies nitrogen from three regulation modules to S-IC stage pneumatic systems (camera lens purge, camera eject, and lox and fuel tank preservation) through the forward umbilical plate.

Figure 8-5 (Sheet 2 of 2)

MOBILE LAUNCHER LEVEL A AND B



S-IC PNEUMATIC CHECKOUT RACKS

REGULATES CONTROLS, AND MONITORS NITROGEN FOR TEST AND CHECKOUT OF PRESSURE SWITCHES AND VALVES IN STAGE PROPULSION SYSTEM.

S-IC INERT PREFILL UNIT

SUPPLIES ETHYLENE GLYCOL TO F-1 ENGINE FUEL JACKETS TO ELIMINATE ENTRAPPED AIR.

Figure 8-6 (Sheet 1 of 2)

VIII GRND. SUPP.

MOBILE LAUNCHER LEVEL A AND B

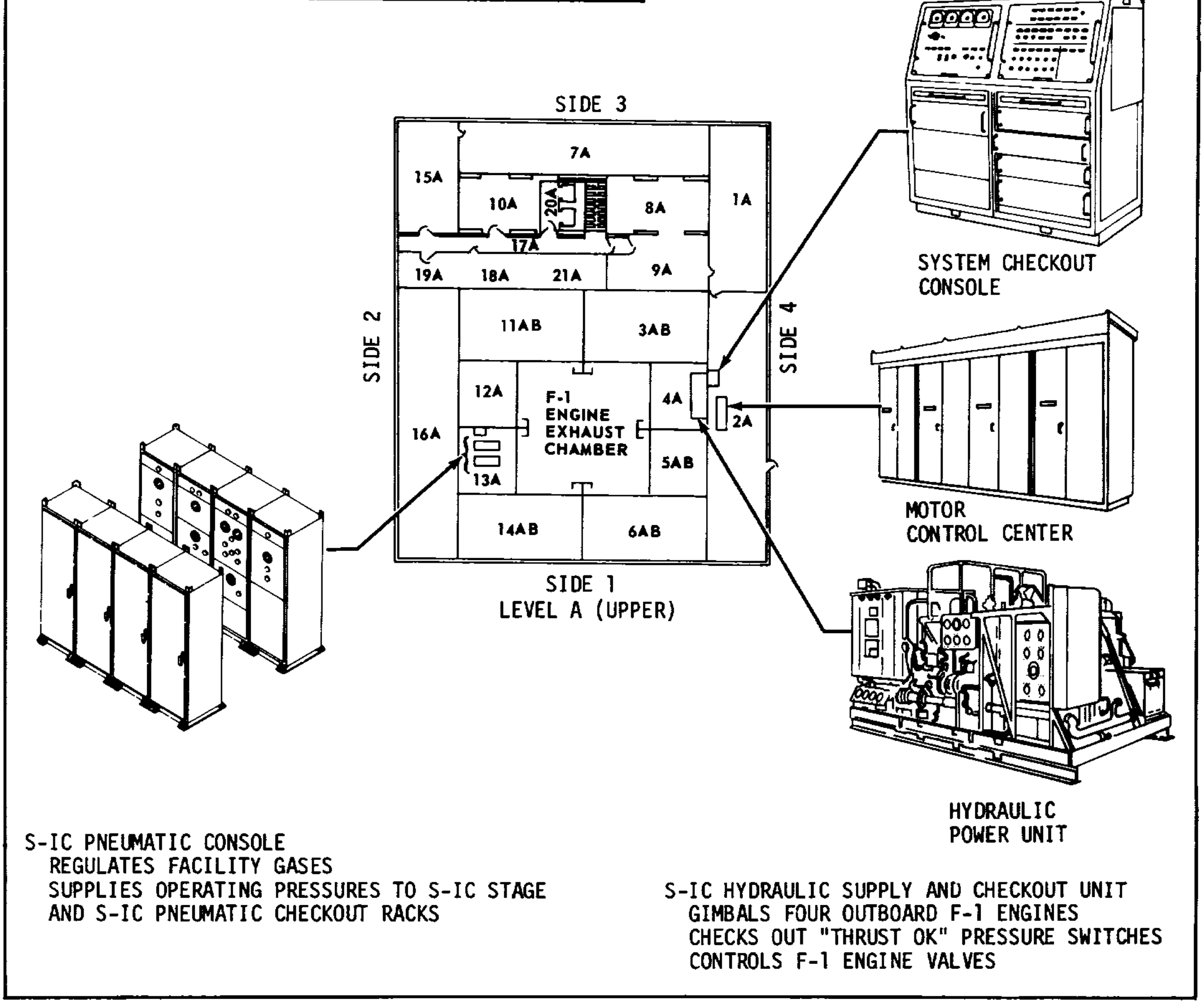


Figure 8-6 (Sheet 2 of 2)

high speed elevators service 18 landings, from level A of the base to the 340-foot tower level.

The hammerhead crane, located at the top of the umbilical tower, has a load capacity of 10 to 25 tons depending on hook distance from tower vertical centerline. The crane is capable of 360 degrees of rotation and one complete revolution per minute in either direction. The hook can be extended 468 feet from the 380-foot level of the tower. Remote control of the crane from the ground and from levels 0 through 360 feet of the tower is provided by portable plug-in type control consoles.

Holddown Arms

The four holddown arms (figure 8-7) are mounted on the ML deck 90° apart around the vehicle base. They position and hold the vehicle on the ML during the VAB checkout, movement to the pad, and pad checkout. The arm bases have sufficient strength to support the vehicle before launch and to withstand the dynamic loads caused by engine cutoff in an abort situation. The vehicle base is held with a preloaded force of 700,000 pounds at each arm.

At engine ignition, the vehicle is restrained until proper engine thrust is achieved, at which time a signal from the countdown sequencer causes each of two identical pneumatic systems to release high-pressure helium to a separator mechanism in each holddown arm. The unlatching interval for the four arms should not exceed 0.050 seconds. If any of the separators fails to operate in 0.180 seconds, release is effected by detonating an explosive nut link.

HOLDDOWN ARMS/TAIL SERVICE MAST

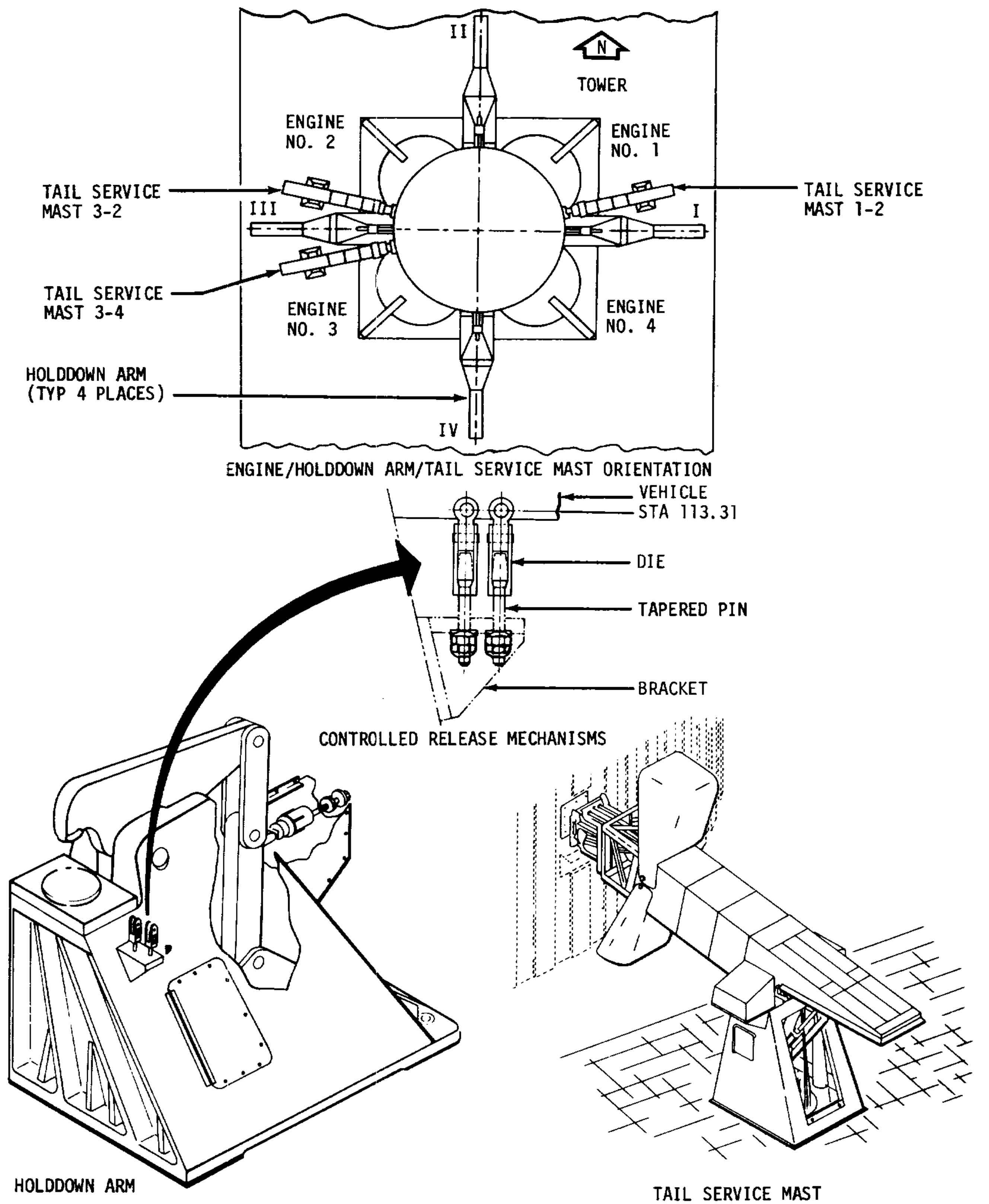


Figure 8-7

At launch, the holddown arms quickly release the vehicle so that the vehicle may accelerate too rapidly and create unallowable stresses in its structure. To provide smooth vehicle release, controlled release mechanisms (figure 8-7) are used to further restrain the vehicle during the first few inches of travel. Each controlled release mechanism basically consists of a bracket bolted to the holddown arm base, a tapered pin fastened to the bracket, and a die coupled to the vehicle. Upon vehicle release, the tapered pin is drawn through the die during the first six inches of vehicle travel. This reduces the diameter of the pin from its maximum to the diameter of the die. The force required to draw the pins through the dies decreases linearly from maximum restraint at holddown release to zero restraint when the pins are free of the dies. The vehicle is then free with the pins remaining in the bracket and the dies travelling with the vehicle. There are provisions for as many as 16 mechanisms per vehicle. The precise number is determined on a mission basis.

Service Arms

The nine service arms provide access to the launch vehicle and support the service lines that are required to sustain the vehicle as described in figure 8-8. The service arms are designated as either preflight or in-flight arms. The preflight arms are retracted and locked against the umbilical tower prior to liftoff. The in-flight arms retract at vehicle liftoff, after receiving a command signal from the service arm control switches located in the holddown arms.

The in-flight service arm launch retract sequence typically consists of the four following operations: arm unlock, umbilical carrier release, carrier withdrawal, and arm retraction and latchback. At T-15 seconds the service arms are unlocked by a signal from the terminal countdown sequencer. When the vehicle rises 3/4-inch, the primary liftoff switches on the holddown arms activate a pneumatic system which unlocks the umbilical carriers and pushes each carrier from the vehicle. If this system fails, the secondary mechanical release mechanism will be actuated when the vehicle rises approximately two inches. If both systems fail the carrier is cammed from the vehicle when it rises approximately 15 inches. Upon carrier ejection, a double-pole switch activates both the carrier withdrawal and arm retraction systems. If this switch fails, it will be by-passed by a signal from the secondary liftoff switches when the vehicle rises 18 inches. Line handling devices on the S-IVB forward and aft arms are also activated on carrier ejection. Carrier withdrawal and arm retraction is accomplished by pneumatic and/or hydraulic systems.

Tail Service Masts

The three tail service mast (TSM) assemblies, figure 8-7, support service lines to the S-IC stage and provide a means for rapid retraction at vehicle liftoff. The TSM assemblies are located on level 0 of the ML base. Each TSM is a counter-balanced structure which is pneumatically/electrically controlled and hydraulically operated. Retraction of the umbilical carrier and vertical rotation of the mast is accomplished simultaneously to ensure no physical contact between the vehicle and mast. The carrier is protected by a clam-shell hood which is closed by a separate hydraulic system as the mast rotates.

LAUNCH PAD

The launch pad, figure 8-9, provides a stable foundation for the ML during Apollo/Saturn V launch and prelaunch operations and an interface to the ML for ML and vehicle systems. There are presently two pads at LC-39 located approximately three miles from the VAB area. Each launch site is an eight-sided polygon measuring approximately 3,000 feet across.

Launch Pad Structure

The launch pad is a cellular, reinforced concrete structure with a top elevation of 48 feet above sea level (42 feet above grade elevation). The longitudinal axis of the pad is oriented north-south, with the crawlerway and ramp approach from the south.

Located within the fill under the west side of the structure (figure 8-10) is a two-story concrete building to house environmental control and pad terminal connection equipment. On the east side of the structure, within the fill, is a one-story concrete building to house the high-pressure gas storage battery. On the pad surface are elevators, staircases, and interface structures to provide service to the ML and the mobile service structure (MSS). A ramp, with a five percent grade, provides access from the crawlerway. This is used by the C-T to position the ML/Saturn V and the MSS on the support pedestals. The azimuth alignment building is located on the approach ramp in the crawlerway median strip. A flame trench 58 feet wide by 450 feet long, bisects the pad. This trench opens to grade at the north end. The 700,000-pound mobile wedge-type flame deflector is mounted on rails in the trench.

An escape chute is provided to connect the ML to an underground hard room. This room is located in the fill area west of the support structure. This is used by astronauts and service crews in the event of a malfunction during the final phase of the countdown.

Pad Terminal Connection Room

The pad terminal connection room (PTCR) (figure 8-10) provides the terminals for communication and data link transmission connections between

MOBILE LAUNCHER SERVICE ARMS

1. S-IC Intertank (preflight). Provides lox fill and drain interfaces. Umbilical withdrawal by pneumatically driven compound parallel linkage device. Arm may be reconnected to vehicle from LCC. Retract time is 8 seconds. Reconnect time is approximately 5 minutes.

2. S-IC Forward (preflight). Provides pneumatic, electrical, and air-conditioning interfaces. Umbilical withdrawal by pneumatic disconnect in conjunction with pneumatically driven block and tackle/lanyard device. Secondary mechanical system. Retracted at T-20 seconds. Retract time is 8 seconds.

3. S-II Aft (preflight). Provides access to vehicle. Arm retracted prior to liftoff as required.

4. S-II Intermediate (in-flight). Provides LH_2 and lox transfer, vent line, pneumatic, instrument cooling, electrical, and air-conditioning interfaces. Umbilical withdrawal systems same as S-IVB Forward with addition of a pneumatic cylinder actuated lanyard system. This system operates if primary withdrawal system fails. Retract time is 6.4 seconds (max).

5. S-II Forward (in-flight). Provides GH_2 vent, electrical, and pneumatic interfaces. Umbilical withdrawal systems same as S-IVB Forward. Retract time is 7.4 seconds (max).

6. S-IVB Aft (in-flight). Provides LH_2 and lox transfer, electrical, pneumatic, and air-conditioning interfaces. Umbilical withdrawal systems same as S-IVB Forward. Also equipped with line handling device. Retract time is 7.7 seconds (max).

7. S-IVB Forward (in-flight). Provides fuel tank vent, electrical, pneumatic, air-conditioning, and preflight conditioning interfaces. Umbilical withdrawal by pneumatic disconnect in conjunction with pneumatic/hydraulic redundant dual cylinder system. Secondary mechanical system. Arm also equipped with line handling device to protect lines during withdrawal. Retract time is 8.4 seconds (max).

8. Service Module (in-flight). Provides air-conditioning, vent line, coolant, electrical, and pneumatic interfaces. Umbilical withdrawal by pneumatic/mechanical lanyard system with secondary mechanical system. Retract time is 9.0 seconds (max).

9. Command Module Access Arm (preflight). Provides access to spacecraft through environmental chamber. Arm may be retracted or extended from LCC. Retracted 12° park position until T-4 minutes. Extend time is 12 seconds from this position.

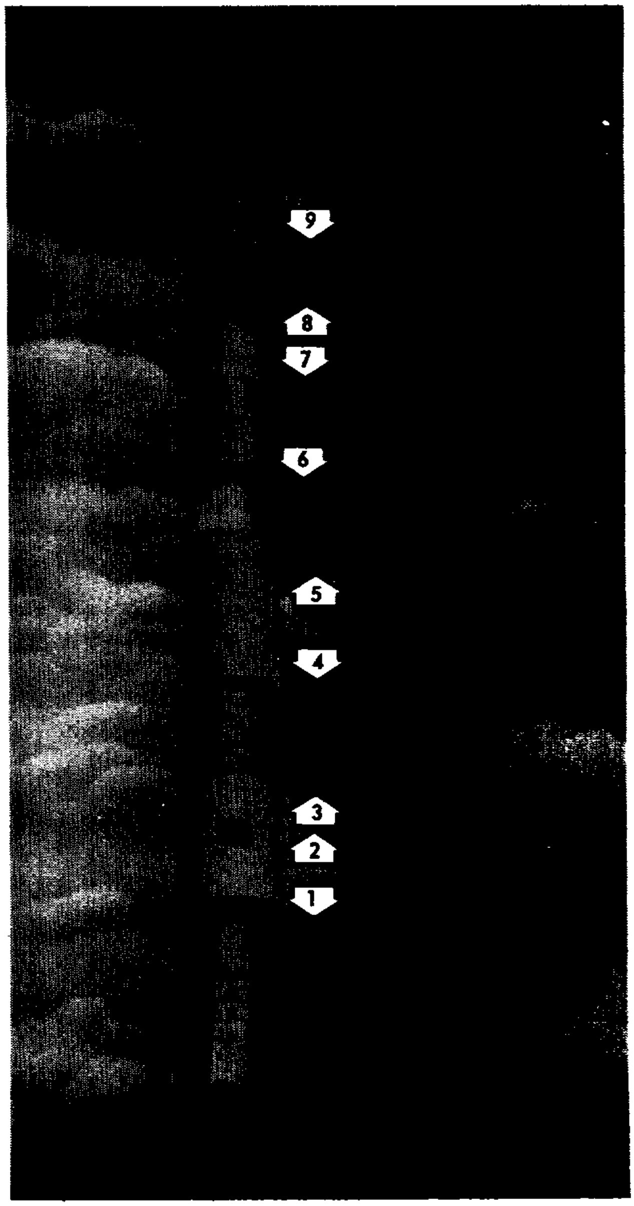


Figure 8-8

VIII
GRND SUPP.

LAUNCH PAD A, LC-39

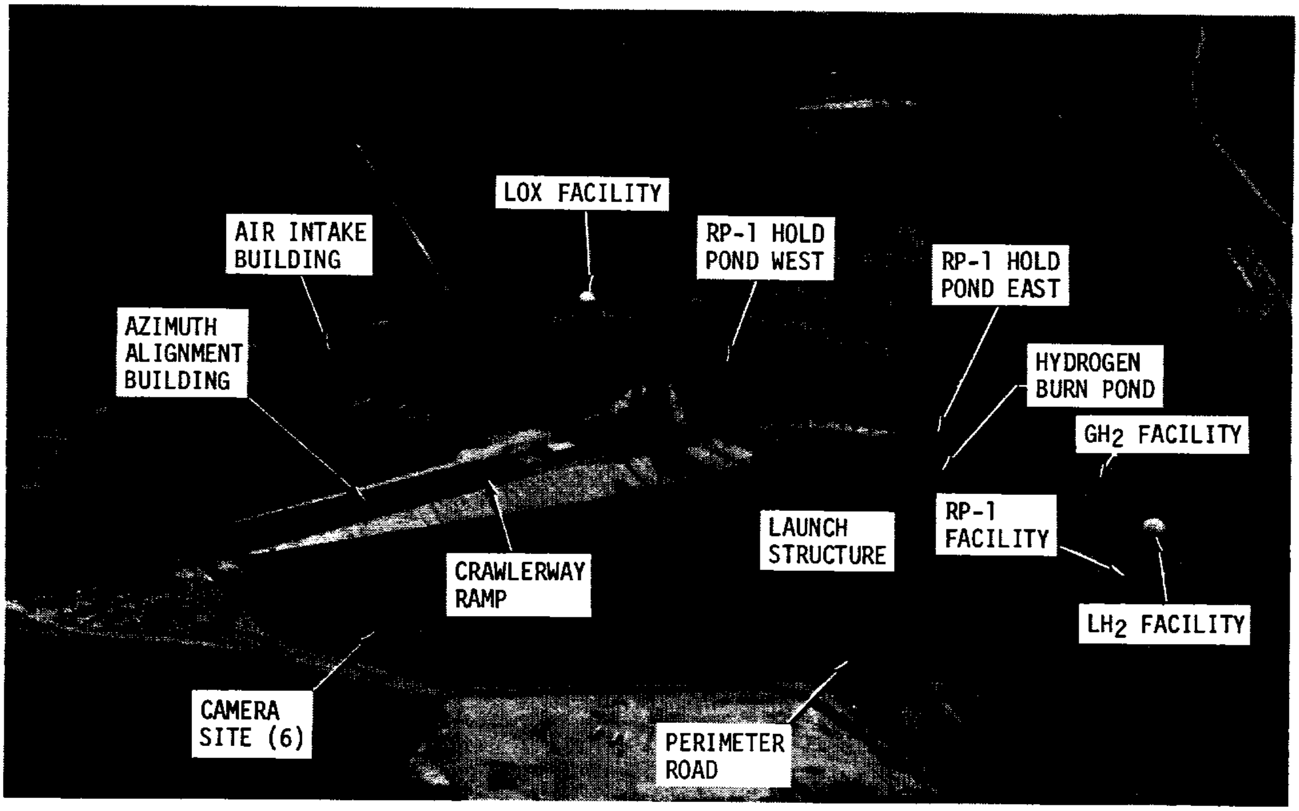


Figure 8-9

LAUNCH STRUCTURE EXPLODED VIEW

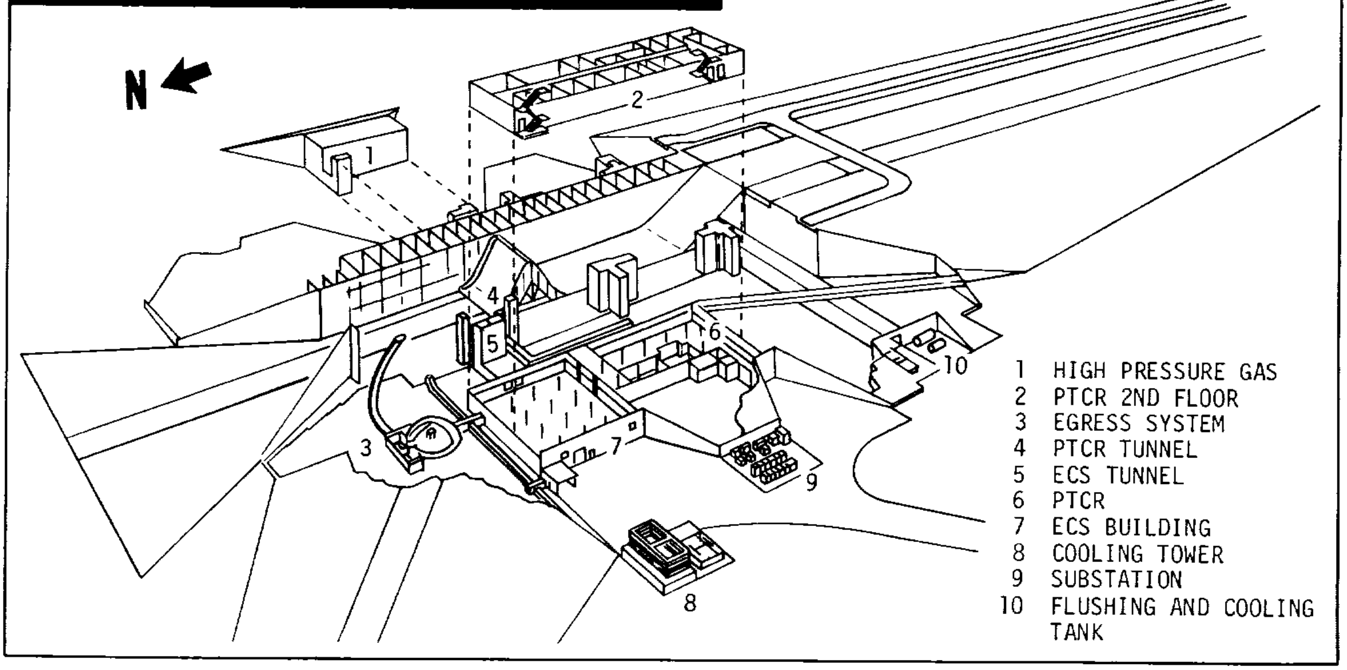


Figure 8-10

the ML or MSS and the launch area facilities and between the ML or MSS and the LCC. This facility also accommodates the electronic equipment that simulates the vehicle and the functions for checkout of the facilities during the absence of the launcher and vehicle.

The PTCR is a two-story hardened structure within the fill on the west side of the launch support structure. The launch pedestal and the deflector area are located immediately adjacent to this structure. Each of the floors of this structure measures approximately 136 feet by 56 feet. Entry is made from the west side of the launch support structure at ground level into the first floor area. Instrumentation cabling from the PTCR extends to the ML, MSS, high-pressure gas storage battery area, lox facility, RP-1 facility, LH_2 facility, and azimuth alignment building. The equipment areas of this building have elevated false floors to accommodate the instrumentation and communication cables used for interconnecting instrumentation racks and terminal distributors.

The air-conditioning system, located on the PTCR ground floor, provides a controlled environment for personnel and equipment. The air-conditioning system is controlled remotely from the LCC when personnel are evacuated for launch. This system provides chilled water for the air handling units located in the equipment compartments of the ML. A hydraulic elevator serves the two floors and the pad level.

Industrial and instrumentation power is supplied from a nearby substation.

Environmental Control System

The ECS room located in the pad fill west of the pad structure and north of the PTCR (figure 8-10) houses the equipment which furnishes temperature and/or humidity controlled air or nitrogen for space vehicle cooling at the pad. The ECS room is 96 feet wide by 112 feet long and houses air and nitrogen handling units, liquid chillers, air compressors, a 3000-gallon water/glycol storage tank, and other auxiliary electrical and mechanical equipment.

High Pressure Gas System

The high-pressure gas storage facility at the pad provides the launch vehicle with high-pressure helium and nitrogen. This facility is an integral part of the east portion of the launch support structure. It is entered from ground elevation on the east side of the pad. The high pressure (6,000 psig) facilities at the pad are

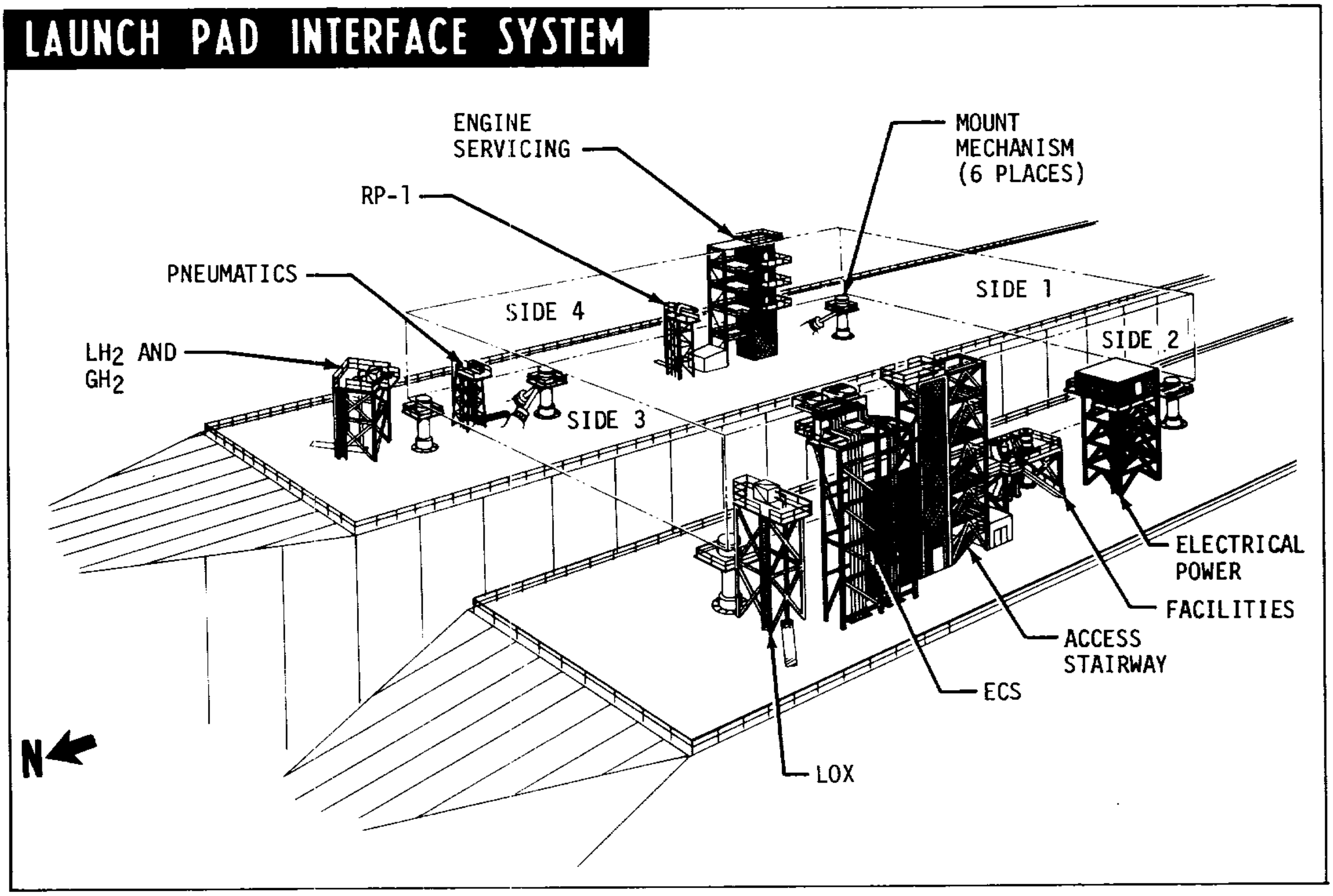


Figure 8-11

VIII GRND. SUPP.

provided for high pressure storage of 3,000 cubic feet of gaseous nitrogen and 9,000 cubic feet of gaseous helium.

Launch Pad Interface Structure

The launch pad interface structure (figure 8-11) provides mounting support pedestals for the ML and MSS, an engine access platform, and support structures for fueling, pneumatic, electric power, and environmental control interfaces.

The ML at the launch pad (as well as at the VAB and refurbish area) is supported by six mount mechanisms which are designed to carry vertical and horizontal loading. Four extensible columns, located near each corner of the launcher base exhaust chamber also support the ML at the launch site. The columns are designed to prevent excessive deflection of the launcher base when the vehicle is fueled and from load reversal in case of an abort between engine ignition and vehicle liftoff.

The MSS is supported on the launch pad by four mounting mechanisms similar to those used to support the ML.

The engine servicing structure provides access to the ML deck for servicing of the S-IC engines and ML deck equipment.

Interface structures are provided on the east and west portions of the pad structure (figure 8-11) for propellant, pneumatic, power, facilities, environmental control, communications, control, and instrumentation systems.

Apollo Emergency Ingress/Egress and Escape System

The Apollo emergency ingress/egress and escape system provides access to and from the Command Module (CM) plus an escape route and safe quarters for the astronauts and service personnel in the event of a serious malfunction prior to launch. The system includes the CM access arm, two 600 feet per minute elevators from the 340-foot level to level A of the ML, pad elevator No. 2, personnel carriers located adjacent to the exit of pad elevator No. 2, the escape tube, and the blast room.

The CM access arm provides a passage for the astronauts and service personnel from the spacecraft to the 320-foot level of the tower. Egressing personnel take the high speed elevators to level A of the ML, proceed through the elevator vestibule and corridor to pad elevator No. 2, move down this elevator to the bottom of the pad, and mount personnel carriers which remove them from the pad area.

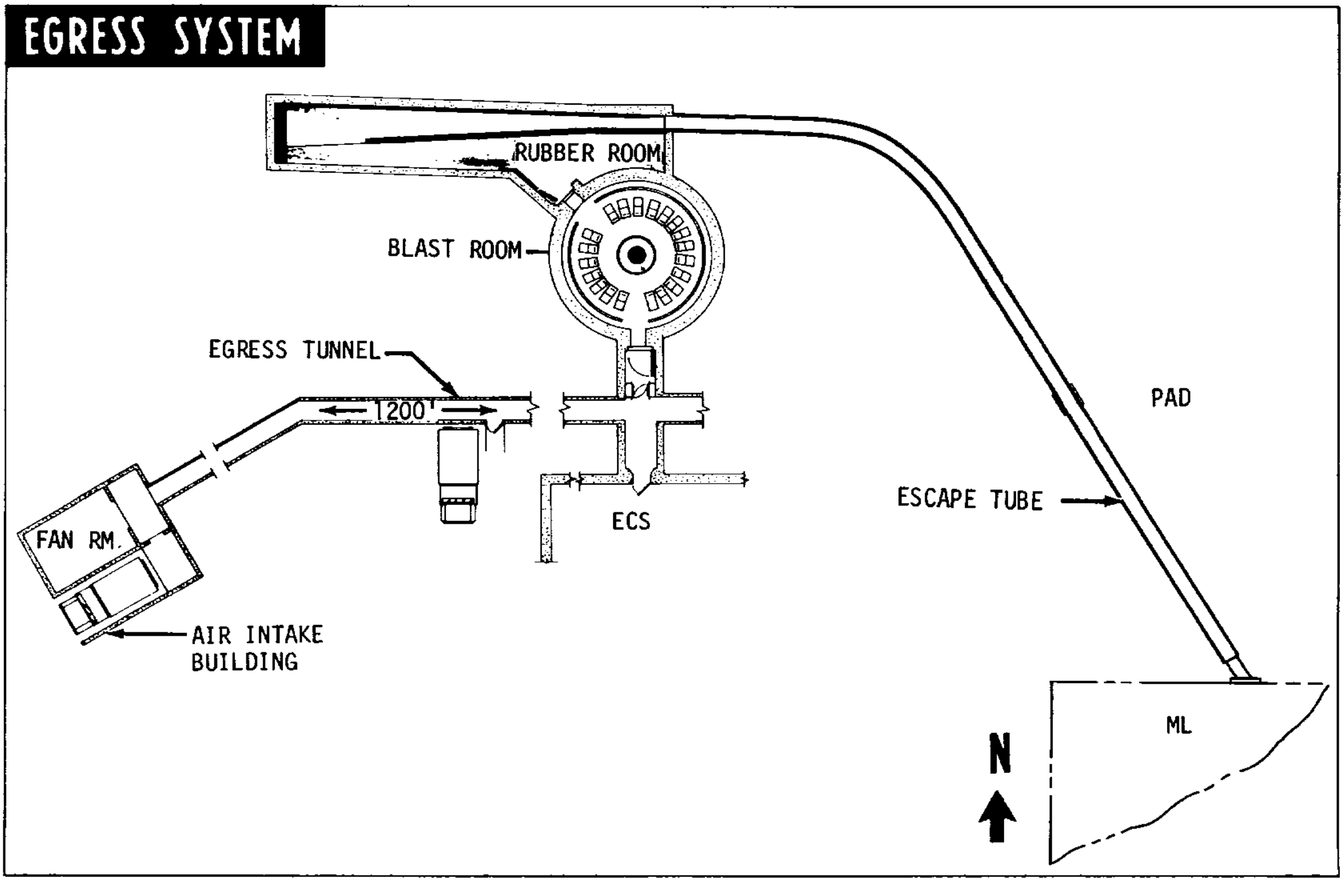


Figure 8-12

When the state of the emergency allows no time for retreat by motor vehicle, egressing personnel upon reaching level A of the ML slide down the escape tube into the blast room vestibule, commonly called the rubber room (see figure 8-12). The escape tube consists of a short section which extends from the elevator vestibule at ML level A to side 3 of the ML base where it interfaces with a fixed portion that penetrates the pad at an elevation of 48 feet. At the lower extremity of the illuminated escape tube, a deceleration ramp is provided to reduce exit velocity, permitting safe exit for the user.

Entrance to the blast room is gained through blast-proof doors controllable from either side. The blast room floor is mounted on coil springs to reduce outside acceleration forces to 3 to 5 g's. Twenty people may be accommodated for 24 hours. Communication facilities are provided in the room including an emergency RF link in which the receiving antenna is built into the ceiling. In the event that escape via the blast-proof doors is not possible, a hatch in the top of the blast room is accessible to rescue crews.

An underground air duct from the vicinity of the blast room to the remote air intake facility permits egress from the pad structure to the pad perimeter. Provision is made to decrease air velocity in the duct to allow personnel movement through the duct.

An alternate emergency egress system employs a slide wire from the vicinity of the 320-foot level of the ML to a 30-foot tail tower on the ground approximately 2,500 feet west of the launcher. Egressing personnel slide down the wire on individual seat assemblies suspended from the wire. Speeds of approximately 50 miles per hour are attained at the low point approximately 20 feet above ground level. A ferrule at the low point activates a braking mechanism which causes a controlled deceleration of each seat assembly to a safe stop. Up to 11 persons may be accommodated by the system.

Fuel System Facilities

The fuel facilities, located in the northeast quadrant of the pad approximately 1,450 feet from pad center, store RP-1 and liquid hydrogen.

The RP-1 facility consists of three 86,000 gallon (577,000 pound) steel storage tanks, a pump house, a circulating pump, a transfer pump, two filter-separators, an 8-inch stainless steel transfer line, RP-1 foam generating building, and necessary valves, piping, and controls. Two RP-1 holding ponds, 150 feet by 250 feet with a water depth of two feet are located north of the launch pad, one on each side of the north-south axis. The ponds retain spilled RP-1 and discharge water to drainage ditches.

The LH_2 facility consists of one 850,000 gallon spherical storage tank, a vaporizer/heat exchanger which is used to pressurize the storage tank to 65 psig, a vacuum-jacketed, 10-inch, Invar transfer line, and a burn pond venting system. The internal tank pressure, maintained by circulating LH_2 from the tank through the vaporizer and back into the tank, is sufficient to provide the proper flow of LH_2 from the storage tank to the vehicle without using a transfer pump. Liquid hydrogen boil-off from the storage and ML areas is directed through vent-piping to bubblecapped headers submerged in the burn pond. The hydrogen is bubbled to the surface of the 100 foot square pond where a hot wire ignition system maintains the burning process.

LOX System Facility

The lox facility is located in the northwest quadrant of the pad area, approximately 1,450 feet from the center of the pad. The facility consists of one 900,000 gallon spherical storage tank, a lox vaporizer to pressurize the storage tank, main fill and replenish pumps, a drain basin for venting and dumping of lox, and two transfer lines.

Gaseous Hydrogen Facility

This facility is located on the pad perimeter road northwest of the liquid hydrogen facility. The facility provides GH_2 at 6,000 psig to charge storage bottles mounted in the S-II and S-IVB stages where the gaseous hydrogen is used to start turbines within these stages. The facility consists of four storage tanks having a total capacity of 800 cubic feet, a flatbed trailer on which are mounted liquid hydrogen tanks and a liquid-to-gas converter, a transfer line, and necessary valves and piping.

Azimuth Alignment Building

The azimuth alignment building is located in the approach ramp to the launch structure in the median of the crawlerway about 700 feet from the ML positioning pedestals. The building houses the auto-collimator theodolite which senses, by a light source, the rotational output of the stable platform. A short pedestal, with a spread footing isolated from the building, provides the mounting surface for the theodolite.

Photography Facilities

These facilities support photographic camera and closed circuit television equipment to provide real-time viewing and photographic documentation coverage. There are six camera sites in the launch pad area, each site containing an access road, five concrete camera pads, a target pole, communication boxes, and a power transformer with a distribution panel and power boxes. These sites cover prelaunch activities and launch operations from six different angles at a radial distance of approximately 1,300 feet from the launch vehicle. Each site has four engineering sequential cameras and one fixed high speed metric camera (CZR). A target pole for optical alignment of the CZR camera is located approximately 225 feet from the CZR pad and is approximately 86 feet high.

Pad Water System Facilities

The pad water system facilities furnish water to the launch pad area for fire protection, cooling, and quenching. Specifically, the system furnishes water for the industrial water system, flame deflector cooling and quench, ML deck cooling and quench, ML tower fogging and service arm quench, sewage treatment plant, Firex water system, lox and fuel facilities, ML and MSS fire protection, and all fire hydrants in the pad area. The water is supplied from three 6-inch wells, each 275 feet deep. The water is pumped from the wells through a desanding filter and into a 1,000,000 gallon reservoir.

Air Intake Building

This building houses fans and filters for the air supply to the PTCR, pad cellular structure, and the ML base. The building is located west of the pad adjacent to the perimeter road.

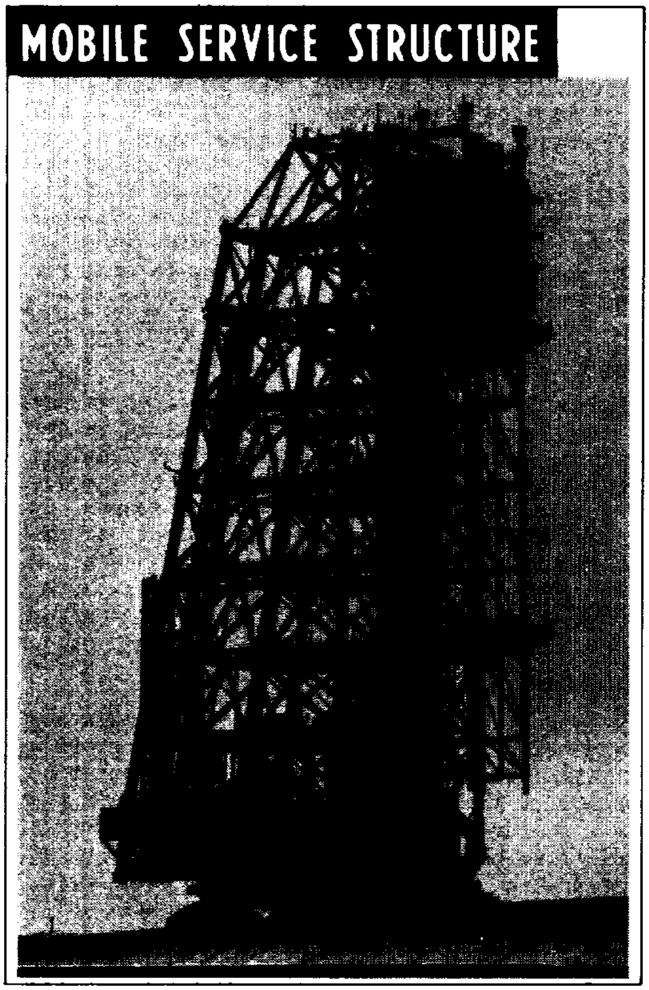


Figure 8-13

Flame Deflector

There are two flame deflectors provided at each pad; one for use and the other held in reserve. Their normal parking position is north of the launch support structure within the launch pad area. The flame deflector protects the boattail section of the Saturn V launch vehicle and the launch stand from hot gases, high pressures, and flame generated by the launch vehicle during the period of engine ignition and liftoff.

MOBILE SERVICE STRUCTURE

The mobile service structure (figure 8-13) provides access to those portions of the space vehicle which cannot be serviced from the ML while at the launch pad. During nonlaunch periods, the MSS is located in a parked position along side of the Crawlerway, 7,000 feet from the nearest launch pad. The MSS is transported to the launch site by the C-T where it is used during launch pad operations. It is removed from the pad a few hours prior to launch and returned to its parking area.

The MSS is approximately 402 feet high, measured from ground level, and weighs 12 million pounds. The tower structure rests on a base 135 feet by 135 feet. The top of the MSS base is 47 feet above grade. At the top, the tower is 87 feet by 113 feet.

The structure contains five work platforms which provide access to the space vehicle. The outboard sections of the platforms open to accept the vehicle and close around it to provide access to the launch vehicle and spacecraft via these platforms. The movable sections of the platforms are actuated by hydraulic cylinders. The lower two platforms are vertically adjustable to serve different parts of the launch vehicle. The upper three platforms are fixed but can be disconnected from the tower and relocated as a unit to serve different vehicle configurations. The two lower platforms and the uppermost platform are open and bordered by a chain-link fence. The remaining two platforms are enclosed and provide environmental control for the spacecraft.

The MSS is equipped with the following systems: air-conditioning, electrical power, various communication networks, fire protection, compressed air, nitrogen pressurization, hydraulic pressure, potable water, and spacecraft fueling.

CRAWLER-TRANSPORTER

The crawler-transporter (figure 8-14) is used to transport the mobile launcher and the mobile service structure. The ML, including the space vehicle is transported from the vehicle assembly building to the launch pad. The MSS is transported from its parking area to and from the launch pad. After launch, the ML is transported to the refurbishment area and subsequently back

CRAWLER TRANSPORTER

Figure 8-14

to the VAB. The C-T is capable of lifting, transporting, and lowering the ML or the MSS, as required, without the aid of auxiliary equipment. The C-T supplies limited electric power to the ML and the MSS during transit.

The C-T consists of a rectangular chassis which is supported through a suspension system by four dual-tread crawler-trucks. The overall length is 131 feet and the overall width is 114 feet. The unit weighs approximately 6 million pounds. The C-T is powered by self-contained, diesel-electric generator units. Electric motors in the crawler-trucks propel the vehicle. Electric motor-driven pumps provide hydraulic power for steering and suspension control. Air conditioning and ventilation are provided where required.

The C-T can be operated with equal facility in either direction. Control cabs are located at each end and their control function depends on the direction of travel. The leading cab, in the direction of travel, will have complete control of the vehicle. The rear cab will, however, have override controls for the rear trucks only.

Maximum C-T unloaded speed is 2 mph, 1 mph with full load on level grade, and 0.5 mph with full load on a five percent grade. It has a 500-foot minimum turning radius and can position the ML or the MSS on the facility support pedestals within ± two inches.

CONVERTER/COMPRESSOR FACILITY

The converter/compressor facility (CCF) converts liquid nitrogen to low pressure and high pressure gaseous nitrogen and compresses gaseous helium to 6,000 psig. The gaseous nitrogen and helium are then supplied to the storage facilities at the launch pad and at the VAB. The CCF is located on the north side of the crawlerway, approximately at the mid-point between the VAB and the main crawlerway junction to launch pads A and B.

The facility includes a 500,000 gallon liquid nitrogen Dewar storage tank, tank vaporizers, high pressure liquid nitrogen pump and vaporizer units, high pressure helium compressor units, helium and nitrogen gas driver/purifiers, rail and truck transfer facilities, and a data link transmission cable tunnel.

The 500,000-gallon storage tank for the liquid nitrogen is located adjacent to the equipment building that houses the evaporators for conversion of the LN_2 to high-pressure gas. The liquid nitrogen is transferred to the vaporizer-compressors by pressurizing the storage tank. After vaporizing and compressing to 150 psig or 6,000 psig, the gaseous nitrogen is piped to the distribution lines supplying the VAB area (6,000 psig) and the pad (150 psig and 6,000 psig).

The gaseous helium is stored in tube-bank rail cars. These are then connected to the facility via a common manifold and a flexible one-inch inside diameter high-pressure line. The helium passes through the CCF helium compressors which boost its pressure from the tube-bank storage pressure to 6,000 psig after which it is piped to the VAB and pad high-pressure storage batteries.

Controls and displays are located in the CCF Mass flow rates of high-pressure helium, high-pressure nitrogen, and low-pressure nitrogen gases leaving the CCF are monitored on panels located in the CCF via cableway ducts running between the CCF and the VAB, LCC, and launch pad.

ORDNANCE STORAGE AREA

The ordnance storage area serves LC-39 in the capacity of laboratory test area and storage area for ordnance items. This facility is located on the north side of the crawlerway and approximately 2,500 feet north-east of the VAB, This remote site was selected for maximum safety.

The ordnance storage installation, enclosed by a perimeter fence, is comprised of three arch-type magazines, two storage buildings, one ready-storage building, an ordnance test building and a guard service building. These structures are constructed of reinforced concrete, concrete blocks, and over-burdened where required. This facility contains approximately 10,000 square feet of environmentally controlled space. It provides for storage and maintenance of retrorockets, ullage rockets, explosive separation devices, escape rockets, and destruct packages. It also includes an area to test the electro-explosive devices that are used to initiate or detonate ordnance items. A service road from this facility connects to Saturn Causeway.

VEHICLE ASSEMBLY AND CHECKOUT

The vehicle stages and the instrument unit (IU) are, upon arrival at KSC, transported to the VAB by special carriers. The S-IC stage is erected on a previously positioned mobile launcher (ML) in one of the checkout bays in the high bay area. Concurrently, the S-II and S-IVB stages and the IU are delivered to preparation and checkout cells in the low bay area for inspection, checkout, and pre-erection preparations. All components of the launch vehicle, including the Apollo spacecraft and launch escape system, are then assembled vertically on the ML in the high bay area.

Following assembly, the space vehicle is connected to the LCC via a high speed data link for integrated checkout and a simulated flight test. When checkout is completed, the crawler-transporter (C-T) picks up the ML with the assembled space vehicle and moves it to the launch site over the crawlerway.

At the launch site, the ML is emplaced and connected to system interfaces for final vehicle checkout and launch monitoring. The mobile service structure (MSS) is transported from its parking area by the C-T and positioned on the side of the vehicle opposite the ML. A flame deflector is moved on its track to its position beneath the blast opening of the ML to deflect the blast from the S-IC stage engines. During the prelaunch checkout, the final system checks are completed, the MSS is removed to the parking area, propellants are loaded and various items of support equipment are removed from the ML and the vehicle is readied for launch. After vehicle launch, the C-T transports the ML to the parking area near the VAB for refurbishment.

TEST SYSTEM

A computer controlled automatic checkout system is used to accomplish the VAB (high bay) and pad testing. An RCA-110A computer and the equipment necessary to service and check out the launch vehicle are installed on the ML. Also an RCA-110A computer and the display and control equipment necessary to monitor and control the service and checkout operations are installed in the LCC. The computers operate in tandem through a data link with the computer in the ML receiving commands from and transmitting data to the computer in the LCC. The physical arrangement of the LCC and the ML are illustrated in figures 8-15 and 8-6 respectively.

Test System Operation

Test system operation for Saturn V launch vehicle checkout is conducted from the firing room (see figure 8-16). During prelaunch operations, each stage is checked out utilizing the stage control and display console, is processed through the computer complex, and is sent to the vehicle. The response signal is sent from the vehicle, through the computer complex, and the result is monitored on the display console. The basic elements of the test system and their functional relationship are shown in figure 8-17.

A switch on the control console can initiate individual operation of a system component or call up a complete test routine from the computer. A CRT is also provided for test conduction and evaluation.

The insertion of a plastic coded card key, prior to console operation, is a required precaution against improper program callup. Instructions, interruptions, and requests for displays are entered into the system by keying in proper commands at the console keyboards.

A complete test routine is called up by initiating a signal at the control panel. The signal is sent to the patch distributor located in the LCC and is routed to the appropriate signal conditioning equipment where the signal is prepared for acceptance by the LCC computer complex. The LCC computer commmunicates with the ML computer to call up the test routine. The ML computer complex sends the signal to the ML signal conditioning equipment and then to the stage relay rack equipment. The signal is then routed to the terminal distribution equipment and through the crossover distributor to interrogate the vehicle sensors. The sensor outputs are sent back to the ML computer complex for evaluation. The result is then sent to the LCC computer complex which routes the result to the stage console for display. Manual control of vehicle functions is provided at the control consoles. This control bypasses the computers and is sent to the vehicle by means of hardwire. The result is also sent back to the display console by hardwire.

The digital data acquisition system (DDAS) collects the vehicle and support equipment responses to test commands, formats the test data for transmission to the ML and LCC, and decommutates the data for display in the ML and LCC. Decommutated test data is also fed to ML and LCC computer for processing and display, and for computer control of vehicle checkout. The DDAS consists of telemetry equipment, data transmission equipment, and ground receiving stations to perform data commutation, data transmission, and data decommutation.

The digital event evaluators are used to monitor the status of input lines and generate a time tagged printout for each detected change in input status. High speed printers in the LCC are connected to each DEE to provide a means for real time or post-test evaluation of discrete data. There are two systems (DEE-3 and DEE-6) utilized to monitor discrete events.

The DEE-3 is located in the PTCR with a printer located in the LCC. It monitors 768 inputs associated with propellant loading, environmental

LCC FACILITY LAYOUT

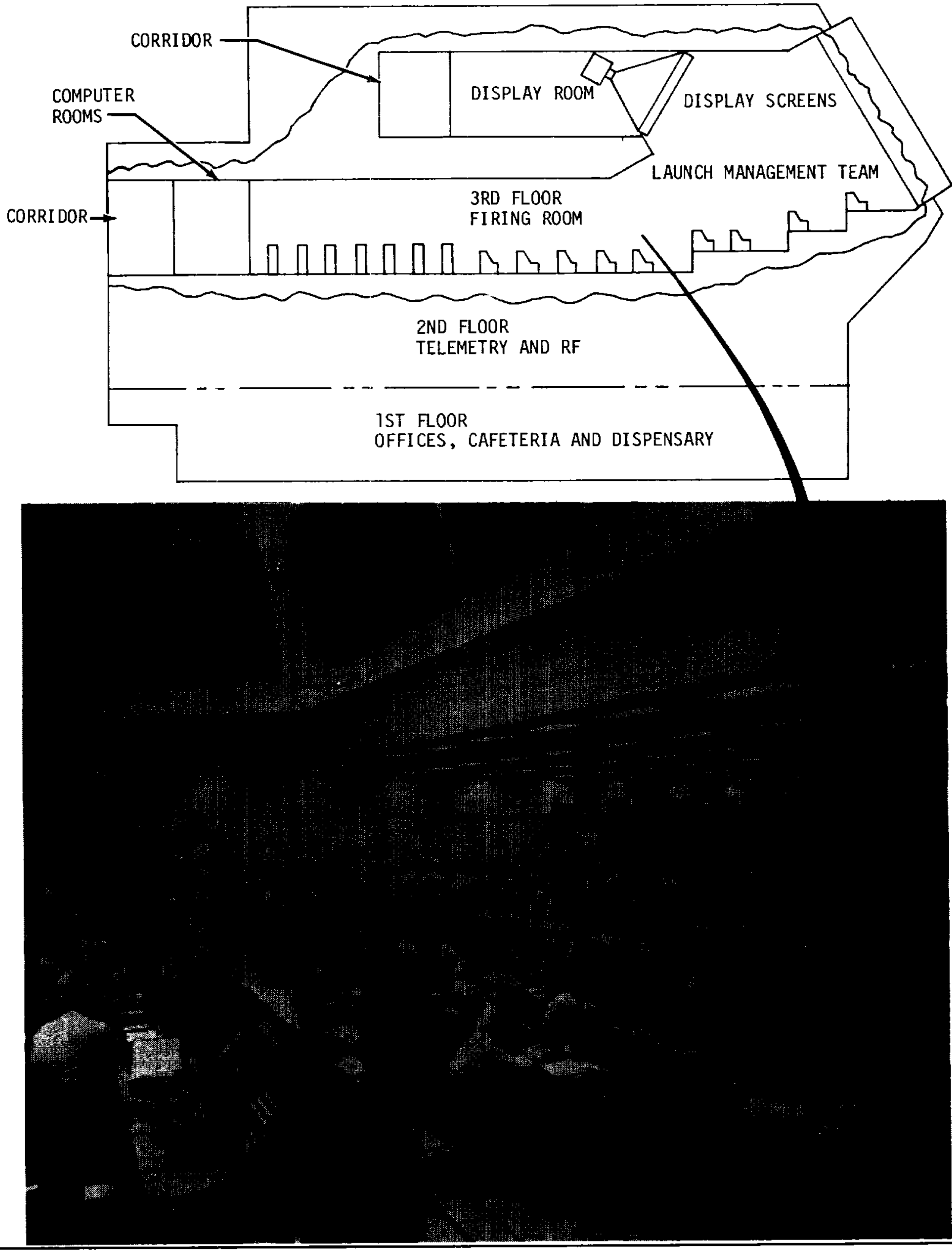


Figure 8-15

FIRING ROOM (TYPICAL)

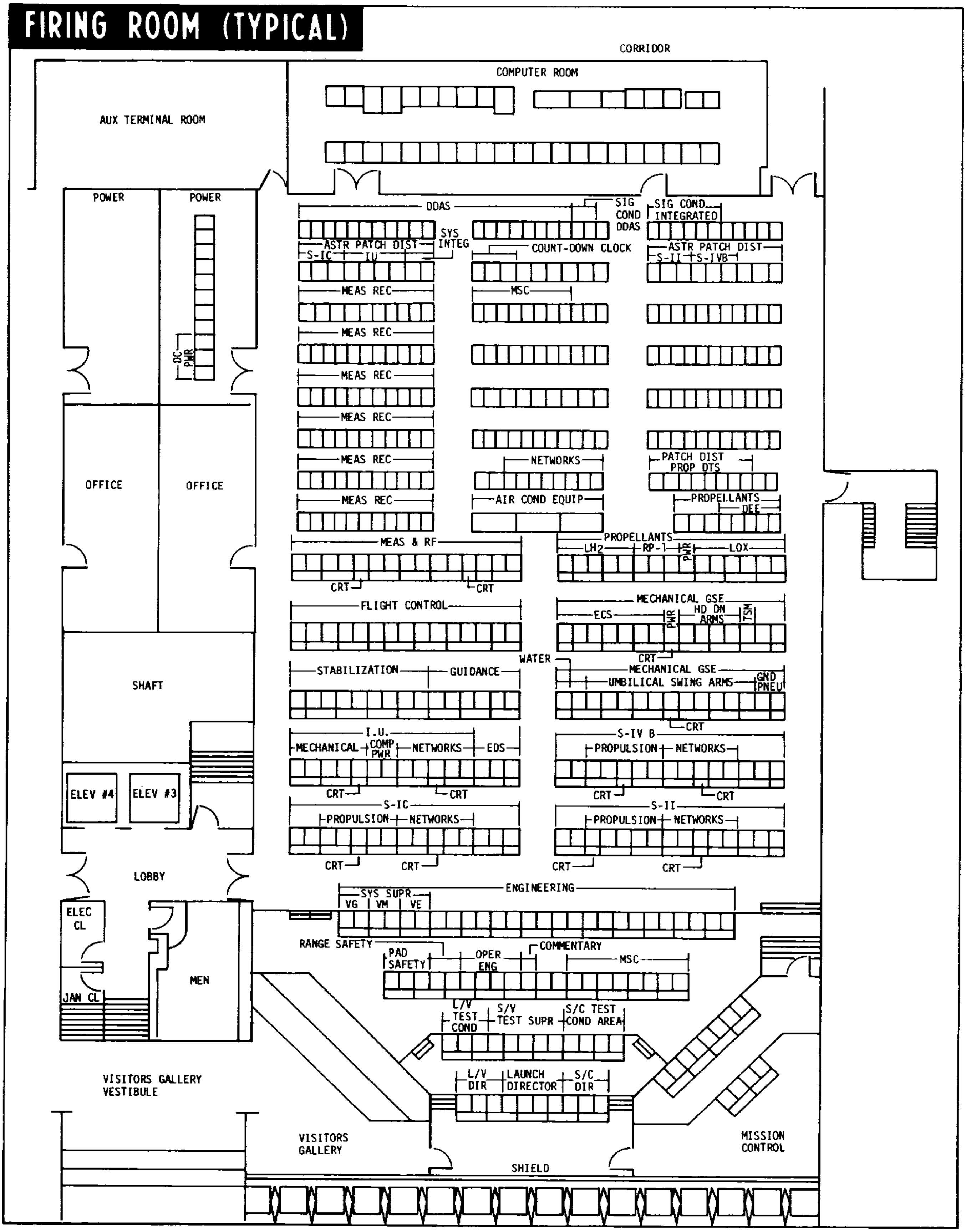


Figure 8-16

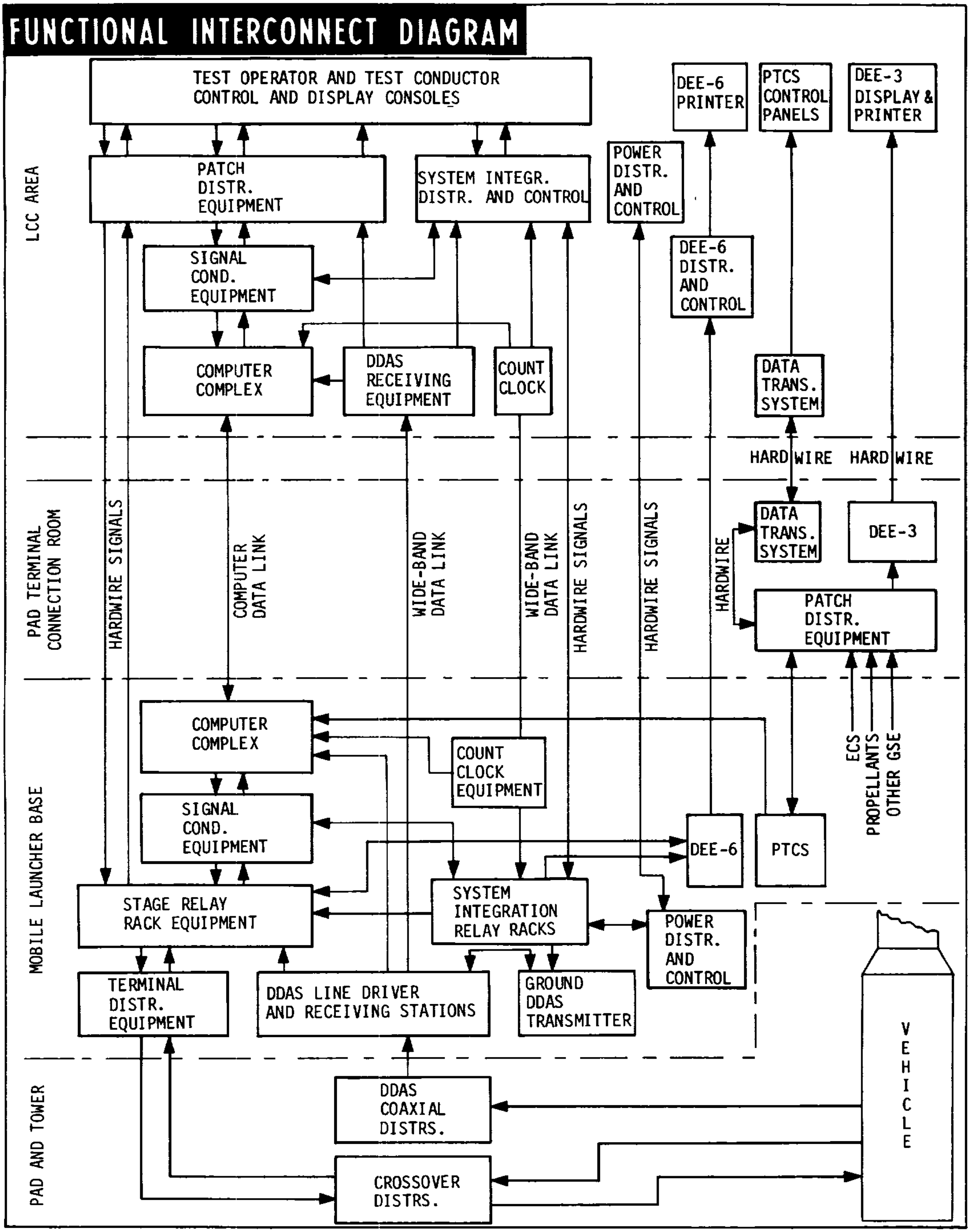

FUNCTIONAL INTERCONNECT DIAGRAM
LCC AREA
PAD TERMINAL CONNECTION ROOM
MOBILE LAUNCHER BASE
PAD AND TOWER
TEST OPERATOR AND TEST CONDUCTOR CONTROL AND DISPLAY CONSOLES
PATCH DISTR. EQUIPMENT
SYSTEM INTEGR. DISTR. AND CONTROL
POWER DISTR. AND CONTROL
DEE-6 PRINTER
PTCS CONTROL PANELS
DEE-3 DISPLAY & PRINTER
SIGNAL COND. EQUIPMENT
DEE-6 DISTR. AND CONTROL
COMPUTER COMPLEX
DDAS RECEIVING EQUIPMENT
COUNT CLOCK
DATA TRANS. SYSTEM
HARD WIRE
HARD WIRE
HARDWIRE SIGNALS
COMPUTER DATA LINK
WIDE-BAND DATA LINK
WIDE-BAND DATA LINK
HARDWIRE SIGNALS
HARDWIRE SIGNALS
HARDWIRE
DATA TRANS. SYSTEM
DEE-3
PATCH DISTR. EQUIPMENT
COMPUTER COMPLEX
COUNT CLOCK EQUIPMENT
ECS
PROPELLANTS
OTHER GSE
SIGNAL COND. EQUIPMENT
DEE-6
PTCS
STAGE RELAY RACK EQUIPMENT
SYSTEM INTEGRATION RELAY RACKS
POWER DISTR. AND CONTROL
TERMINAL DISTR. EQUIPMENT
DDAS LINE DRIVER AND RECEIVING STATIONS
GROUND DDAS TRANSMITTER
VEHICLE
DDAS COAXIAL DISTRS.
CROSSOVER DISTRS.

Figure 8-17

control, water control, and DDAS.

The DEE-6 is located in the ML base with a printer and remote control panel in the LCC. It monitors up to 4320 discrete signals from the vehicle stage umbilicals, pad and tower ground support equipment, and the DDAS.

The computer complex consists of two RCA-110A general purpose computers and peripheral equipment. This equipment includes a line printer, card reader, card punch, paper tape reader, and magnetic tape transports. The peripheral equipment provides additional bulk storage for the computer, acts as an input device for loading test routines into the computer memory, and as an output device to record processed data. One computer is located in the ML base and the other in the LCC behind the firing room. The computers are connected by underground hardwire. The computer system uses a tandem philosophy of checkout and control. The LCC computer is the main control for the system. It accepts control inputs from test personnel at the consoles in the firing room as well as inputs from tape storage and transmits them as test commands to the ML computer. The ML computer has the test routines stored in its memory banks. These routines are called into working memory and sent as discrete signals to the launch vehicle in response to the commands received from the LCC computer. The ML computer reports test routine status, data responses and results of test to the LCC computer. It is through this link that the control equipment and personnel in the firing room are informed of the test progress.

The propellant tanking computer system (PTCS) determines and controls the quantities of fuel and oxidizer on board each stage. Optimum propellant levels are maintained and lox and LH_2 are replenished as boiloff occurs during the countdown. The propellant tanking operation is monitored on the PTCS control panel.

Visual surveillance of launch vehicle checkout is provided to the launch management team and for distribution to MSC and MSFC through the operational television system (OTV). Sixty cameras provide this capability, 27 of which are located on the ML, 15 in the pad area, 12 on the MSS and 6 in the LCC. Any camera may be requested for viewing on the 10 x 10 foot screens in the firing room.

Certain major events may be observed by members of the launch management team who occupy the first four rows in the firing room. The significant launch vehicle events which are displayed on the 10 x 10 foot screen are shown in figure 8-18.

PRELAUNCH SEQUENCE

The prelaunch sequence of events (see figure 8-19) take place first in the VAB and then at the pad. The VAB events are V-times and are referenced to completion of VAB activities. The pad events are T-times and are referenced to liftoff.

Prior to VAB activities, the stages and components are received at KSC. The stages and components are unloaded, transported to the VAB, inspected, and erected in the applicable checkout bay.

VAB Activities

The VAB activities are the assembly and checkout activities which are completed in two major areas of the VAB - the high bay and the low bay. These activities require approximately 250 clock hours and include phases A and B as illustrated in figure 8-19.

Phase A includes the time period V-250 to V-115 and encompasses the vehicle assembly and checkout activities accomplished prior to spacecraft assembly and installation. Phase B includes the remaining activities which are completed in the VAB.

Low Bay Activities. The low bay activities include receiving and inspection of the S-II stage, S-IVB stage and IU, and the assembly and checkout of the S-II and S-IVB stages.

The S-II stage is brought into the low bay area and positioned on the checkout dolly and access platforms are installed. An insulation leak check, J-2 engine leak check, and propellant level probes electrical checks are made.

The S-IVB stage is brought into the low bay area and positioned on the checkout dolly and access platforms are installed. A fuel tank inspection, J-2 engine leak test, hydraulic system leak check, and propellant level sensor electrical checks are made.

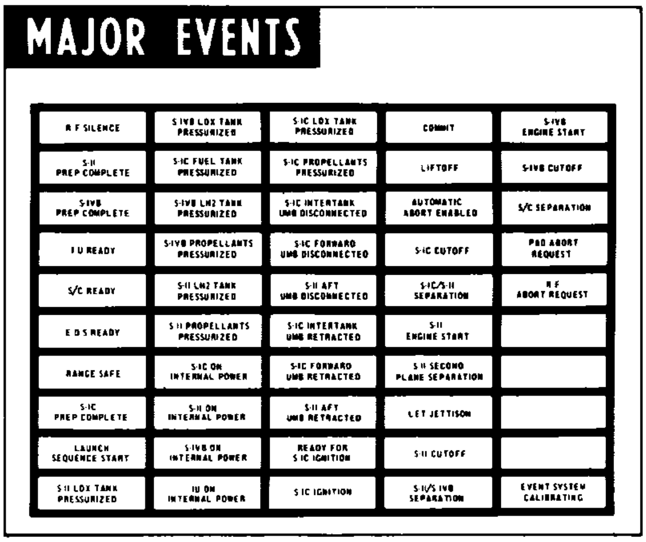


Figure 8-18

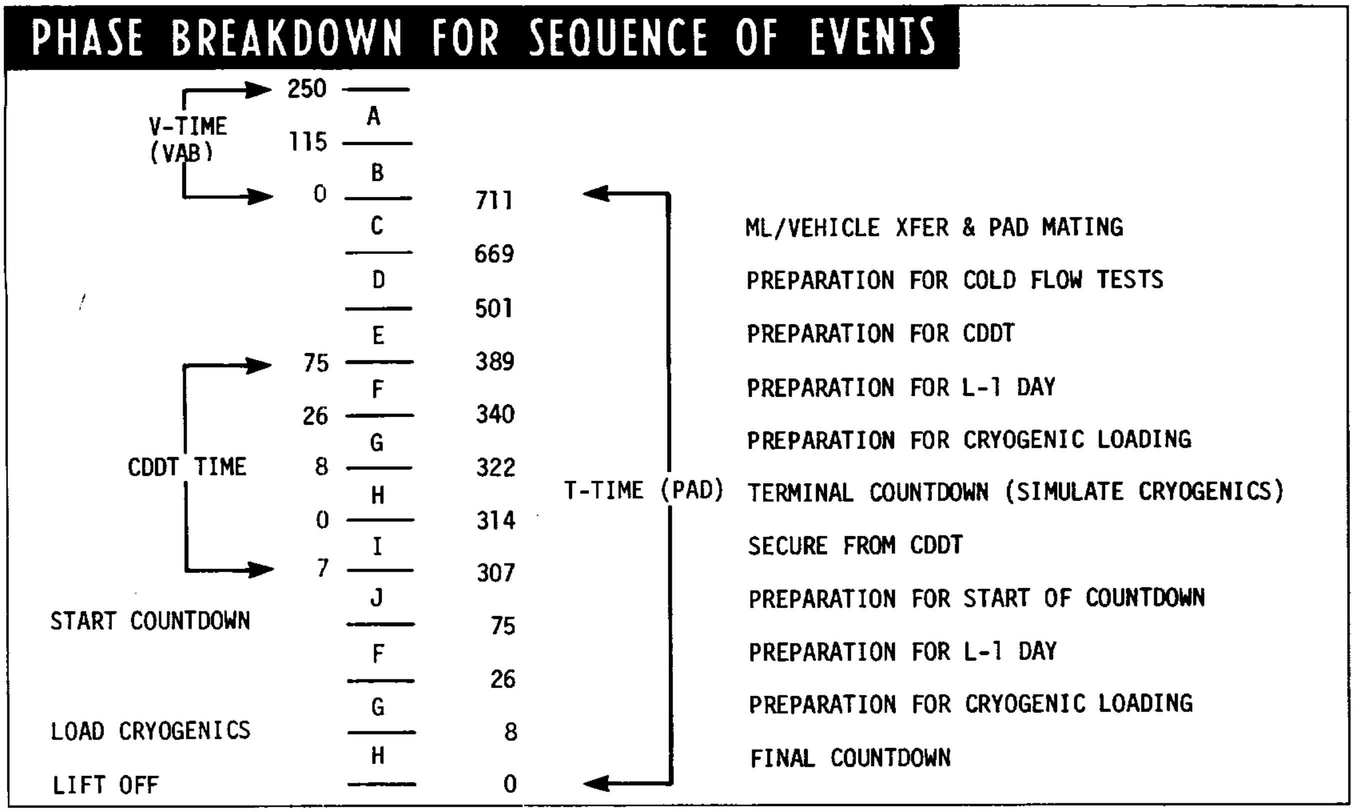


Figure 8-19

High Bay Activities. High bay activities include S-IC stage checkout, stage mating, stage systems tests, launch vehicle integrated tests, space vehicle overall tests, and a simulated flight test. High bay checkout activities are accomplished using the consoles in the firing room, the computer complex, and display equipment.

The S-IC stage is positioned and secured to the ML and access platforms are installed. The umbilicals are secured to the vehicle plates. Prepower and power-on checks are made to ensure electrical continuity. Pneumatic, fuel, lox, and F-1 engine leak checks are made. Instrumentation, ODOP, and range safety system checks are made.

The S-II stage is mated to the S-IC stage. The umbilicals are secured to the vehicle plates. Pre-power and power-on checks are made to ensure electrical continuity. Engine hydraulic and S-II pressurization system checks are made. Instrumentation, propulsion, propellant, and range safety system checks are made.

The S-IVB stage is mated to the S-II stage and the IU is mated to the S-IVB. The S-IVB and IU umbilicals are secured to the vehicle plates. Pre-power and power-on checks are made to ensure electrical continuity. S-IVB engine hydraulic, pressurization, and auxiliary propulsion system leak checks are made. S-IVB propellant, propulsion, pressurization, and range safety system checks are made. IU S-band, C-band, AZUSA, and guidance and navigation system checks are made.

Following completion of the stage system tests, launch vehicle integrated checks are accomplished. Vehicle separation, flight control, sequence malfunction, and emergency detection system checks are made. The spacecraft is then mated to the launch vehicle.

After the spacecraft is mated, space vehicle overall checks are made. Two overall tests are performed. Test number 1 is performed to verify RF, ordnance, pressurization, propulsion, guidance and control, propellant, and emergency detection system proper operation. Test number 2 is performed to verify proper operation of all systems during an automatic firing sequence and flight sequence. This includes a simulated holddown arm release, electrical umbilical ejection, swing arm retraction, and firing of live ordnance in test chambers. Flight type batteries will be used to check out internal power.

A simulated flight test is run when the overall tests are completed. The simulated flight test verifies proper operation of the space vehicle during a normal minus count and an accelerated plus count. A normal mission profile is followed during this time. The simulated flight test ensures that the space vehicle is ready for transfer to the pad. The launch escape system is installed on the command module of the spacecraft. The ML and space vehicle are now ready for transport to the pad.

Vehicle Transfer and Pad Mating Activities

As illustrated in figure 8-19, phase C consumes approximately 42 clock hours. Approximately eight hours are required for the mobile launcher and space vehicle to move from the VAB to the launch area. The remaining time is consumed as shown in figure 8-20.

Pad Activities

In general, once the vehicle and ML have been mated to the pad facility, two major operations must be performed. The first is to verify the readiness of the launch vehicle, spacecraft, and launch facility to perform the launch sequence and the second is to complete the final launch preparations and launch. These two operations, barring any situation which will result in a hold of more than five hours, take approximately 673 hours to complete.

The functions performed to complete the checkout are grouped in phases as shown in figure 8-19. The major functions in each of these phases are shown in relation to each other and are referenced from time to liftoff.

At approximately T-673 hours, the connection of the ML to the launch pad facility is completed. The various support systems (power, ECS, and pneumatics) are active and are being monitored by DDAS/computer systems. These systems remain active throughout the entire time period since they are required for test, checkout, and launch support.

During phase D preparations are performed on the Saturn V system for cold flow checks. The purpose of these checks is to verify that the cryogenic fluids can be made available to the loading equipment when required later in the countdown. Propellant loading is simulated to verify that system controls are functional.

During phase E preparations are performed on the Saturn V system to support start of the countdown demonstration test (CDDT). These preparations verify that the vehicle is ready for propellant loading. The hazardous gas analyzer system is activated prior to RP-1 loading to protect the Saturn V system from leakage induced hazards.

At the completion of phase E the CDDT is begun. The CDDT duplicates very closely the actual final countdown. For this reason phases F, G, and H are duplicated in figure 8-19 but are illustrated only once in figure 8-20. The CDDT differs from the final countdown in that the test is stopped at the point where the S-IC ignition command would normally be given. It differs also in the fact that hypergolics are not loaded and installed ordnance is not armed.

The 7 hour period of phase I is consumed by the activities of securing from the CDDT. These activities involve draining vehicle propellants, purging tanks, venting pressure spheres, and generally deactivating the space vehicle.

With the completion of the CDDT, preparations for starting the actual countdown are initiated. These preparations include some items which would either compromise the safety of the vehicle if accomplished later in the countdown or impose additional constraints on pad access during the final phases of the countdown.

At T-75 hours, the final countdown begins. Phase F generally prepares the Saturn V system for final checkout and servicing operations required for flight.

Phase G ends at the start of cryogenic loading. During this time period, most of the mechanical tasks required for the launch are completed.

The final phase of the countdown, phase H, normally covers a time span of eight hours. During this period of time cryogenics are loaded, conditioned, and pressurized. Final checks are performed on all subsystems. The propulsion systems are serviced and prepared for launch. All onboard spheres are brought up to flight pressure and the crew mans the Command Module.

By the time spacecraft closeout is complete, most major operations have been completed. Propellants are being replenished as required to supplement cryogens lost due to boiloff. Boiloff will continue until the various stage vent valves are closed for tank prepressurization and some vapor may be noticeable.

With the start of the automatic sequence at T-187 seconds, the final operations required for launch begin. All pneumatic and propellant supply lines are vented and purged to prevent damage to the vehicle at umbilical release. The vehicle is switched to internal power, necessary purges are put in launch mode and some service arms are retracted.

At T-8.8 seconds, the S-IC ignition command is given. This signal, in addition to starting engine No. 5 also causes the venting of remaining high pressure pneumatic lines.

At T-0 seconds, the launch commit signal is given, causing the holddown arms to retract hydraulically. These four arms restrain the launch vehicle until a satisfactory thrust level is achieved after which the controlled release assemblies provide for gradual release of the vehicle during liftoff.

SEQUENCE OF EVENTS (TYPICAL)

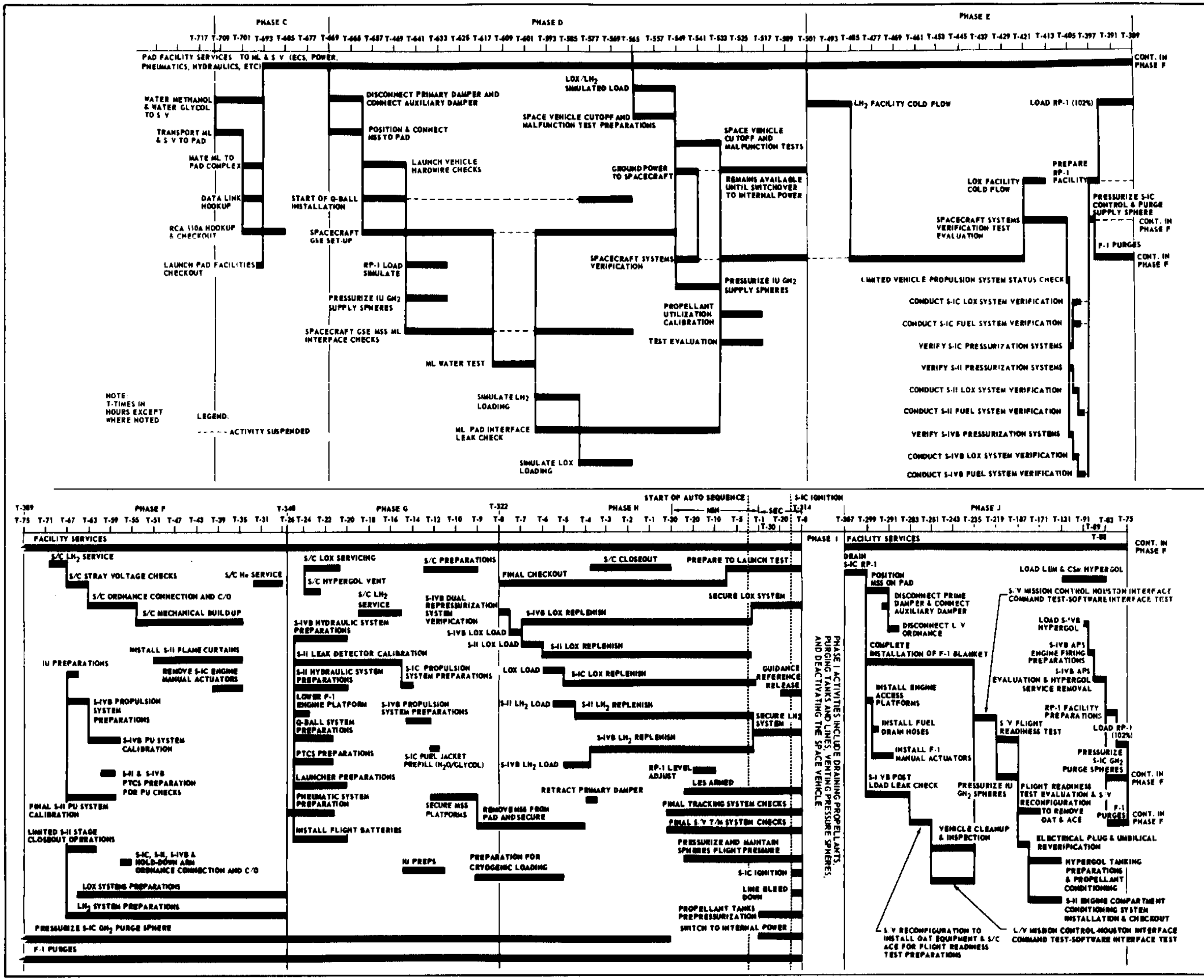


Figure 8-20

VIII GRND SUPP.

Hold and Recycle

A recycle is required any time there is a scrub during the countdown. The recycle activities are based on a normal launch countdown attempt followed by a five hour hold, a scrub and recycle, and a second countdown attempt. Since recycle sequences are established as a result of conditions and/or malfunctions prevailing at the time in the launch sequence the decision to recycle is made, there is no universal recycle which will apply to all conditions at any point in the launch sequence. The examples given below are typical and are presented to show the major milestones of activity and the length of time to accomplish each recycle.

The recycle points within the countdown are considered at T-20 seconds (retraction of S-IC Forward Service Arm), at T-6 hours (near end of lox loading and before LH_2 loading), and T-7:50 hours (just prior to LV cryogenic loading). In each case, the count is recycled to T-13 hours and held until the required reservicing and retest is accomplished.

T-20 Second Scrub. The elapsed time is approximately 52 hours, thus delaying a second launch attempt until 3 days after T-0. The launch vehicle lox and LH_2 are drained. After the drain operation is completed, the tanks are purged, the pad is opened and ordnance is safed (propellant safe & arm devices, ullage and retro rocket initiators, and separation system detonators). The MSS is moved in to provide access to the vehicle. S-IVB propellant and J-2 engine system leak checks are made and the propellant utilization system is calibrated. The countdown is started again at T-13 hours (see figure 8-20).

T-6 Hour Scrub. The elapsed time is approximately 47 hours, thus delaying a second launch attempt until three or two days after T-0. The major events would be the same as those described for a T-20 second scrub except that the LH_2 would not be on board as yet.

T-7:50 Hour Scrub. The elapsed time is approximately 41 hours, thus delaying a second launch attempt until 2 days after T-0. Ordnance is safed (propellant safe & arm devices, ullage and retro rocket initiators, and separation system detonators). The MSS is moved in to provide access to the vehicle. The countdown is started again at T-13 hours (see figure 8-20).

LAUNCH CUTOFF INTERLOCKS

The Saturn V vehicle countdown is automatically controlled from T-187 seconds to T+6 seconds by the terminal countdown sequencer (TCS), located in the mobile launcher (ML). This portion of the countdown is referred to as the automatic sequence.

Three interlocks must be activated prior to T-187 seconds to enable sequencer start: the TCS must be manually armed; the firing command preparation complete summation interlock must be activated; and the firing command enable switch on the S-IC launch control panel must be depressed. With these three interlocks activated, the T-187 second pulse from the countclock will start the TCS. The firing command preparation complete summation interlock includes many critical system parameters, events, and conditions, e.g., gas generator valves closed, all main fuel valves closed, EDS ready, exhaust igniters installed, ordnance preparation complete, spacecraft ready for launch, hydraulic pressure OK, ignition source voltage OK, launch support preparation complete, all checkout valves in ground return position, IU ready for firing, and many similar items. Improper operation or failure of any of these items will inhibit activation of the firing command preparation complete summation interlock and consequently prevent the start of the TCS.

Once the automatic sequence has been initiated, it can be stopped only by a cutoff signal; there are no provisions for holding.

Manual Cutoff Capability

Manual closure of the cutoff switch directly initiates a cutoff command. This capability may be used if the test conductor observes a condition serious enough to warrant cutoff. Manual cutoff is available at any time from TCS start to umbilical separation.

Instant Cutoff Interlocks

The TCS may be terminated automatically by any one of the following six cutoff interlocks:

1. S-IC stage logic cutoff. The cutoff signal is caused by (a) the presence of an engine shutdown signal on the stop solenoids from internal sources; or (b) a malfunction in either of the redundant range safety command destruct systems. The range safety command receiver system check is enabled from T-17 seconds to T-50 milliseconds (ms).

2. S-IC main fuel valve failure. The cutoff signal is caused by (a) both main fuel valves on any engine open prior to hypergol rupture or (b) on any engine, one main fuel valve open while the other valve remains closed (to prevent lox-rich condition in the engines). For main fuel valve failure, engines will shut down in a 3-2 sequence with the first three engines shutting down at cutoff and the other two 100 ms later. If the failure occurs in engine 2 or 4, a preferred engine shutdown sequence will occur, i.e., 2, 4, and 5 followed by 1 and 3. If the failure occurs in engine 1, 3, or 5, the normal engine shutdown sequence will occur, i.e., 1, 3, and 5 followed by

2 and 4. If cutoff is caused by other than main fuel valve failure, the normal shutdown sequence will occur.

3. Sequencer power supply failure. The cutoff signal is caused by an out-of-tolerance value of the voltage supply. A new TCS under development having a battery backup will eventually eliminate this interlock.

4. S-IC voltage failure. The cutoff signal is caused by improper voltage output from either the stage main bus (+1D11) or the stage instrumentation bus (+1D21).

5. Emergency detection system (EDS) failure. The EDS failure interlock is enabled from T-8.9 seconds to T-50 ms. A cutoff signal is caused by one of the three manual cutoff commands from the spacecraft or loss of one of the three EDS voting logic buses.

6. IU failure cutoff. This interlock is enabled from T-8.9 seconds to T-50 ms. During this period, a loss of IU ready to launch will initiate cutoff. The IU ready to launch interlock monitors the IU power systems, the flight computer, and the presence of the S-IC ignition command.

SECTION IX
MISSION CONTROL MONITORING

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INTRODUCTION

Mission control monitoring involves the following functions: prelaunch checkout and launch operations; tracking the space vehicle to determine its present and future position; securing information on the status of the flight crew and launch vehicle systems (via telemetry); evaluation of telemetry information; commanding the space vehicle by transmitting real-time and update commands to the onboard computer; voice communication between flight and ground crews; and recovery operations.

These functions require the use of a facility to assemble and launch the space vehicle (described in Section VIII of this manual); a central flight control facility; a network of remote stations located strategically around the world; a method of rapidly transmitting and receiving information between the space vehicle and the central flight control facility; a real-time data display system in which the data is made available and presented in usable form at essentially the same time that the data event occurred; and ships/aircraft to recover the spacecraft on return to earth.

Associated with the flight crew in mission control operations are the following organizations and facilities:

1. Mission Control Center (MCC), Manned Spacecraft Center, Houston, Texas. The MCC contains the communication, computer, display, and command systems to enable the flight controllers to effectively monitor and control the space vehicle.

2. Kennedy Space Center, Cape Kennedy, Florida. The space vehicle is launched from the facility. Prelaunch, launch, and powered flight data are collected at the Central Instrumentation Facility (CIF) at KSC from the launch pads, CIF receivers, Merritt Island Launch Area (MILA), and the downrange Air Force Eastern Test Range (AFETR) stations. This data is transmitted to MCC via the Apollo Launch Data System (ALDS). Also located at KSC is the Impact Predictor (IP).

3. Goddard Space Flight Center (GSFC), Greenbelt, Maryland, GSFC manages and operates the Manned Space Flight Network (MSFN) and the NASA Communications (NASCOM) networks. During flight, the MSFN is under operational control of the MCC.

4. George C. Marshall Space Flight Center (MSFC), Huntsville, Alabama. MSFC by means of the Launch Information Exchange Facility (LIEF) and the Huntsville Operations Support Center (HOSC) provides launch vehicle systems real-time support to KSC and MCC for preflight, launch, and flight operations.

A block diagram of the basic flight control interfaces is shown on figure 9-1.

VEHICLE FLIGHT CONTROL CAPABILITY

Flight operations are controlled from the MCC. The MCC is staffed by flight control personnel who are trained and oriented on one program and mission at a time. The flight control team members perform mission planning functions and monitor flight preparations during preflight periods. Each member becomes and operates as a specialist on some aspect of the mission.

MCC ORGANIZATION

The MCC has two control rooms for flight control of manned space flight missions. Each control room, called a Mission Operations Control Room (MOCR), is used independently of the other and is capable of controlling individual missions. The control of one mission involves one MOCR and a designated team of flight controllers. Staff Support Rooms (SSR's) located adjacent to the MOCR are manned by flight control specialists who provide detailed support to the MOCR. Figure 9-2 outlines the organization of the MCC for flight control and briefly describes key responsibilities. Information flow within the MOCR is shown in figure 9-3.

The consoles within the MOCR and SSR's permit the necessary interface between the flight controllers and the spacecraft. The displays and controls on these consoles and other group displays provide the capability to monitor and evaluate data concerning the mission and based on these evaluations, to recommend or take appropriate action on matters concerning the flight crew and spacecraft.

Problems concerning crew safety and mission success are identified to flight control personnel in the following ways:

1. Flight crew observations;
2. Flight controller real-time observations;
3. Review of telemetry data received from tape recorder playback;
4. Trend analysis of actual and predicted values;
5. Review of collected data by systems specialists;
6. Correlation and comparision with previous mission data;
7. Analysis of recorded data from launch complex testing.

The facilities at the MCC include an input/output processor designated as the Command, Communications and Telemetry System (CCATS) and a computational facility, the Real-Time Computer Complex (RTCC). Figure 9-4 shows the MCC functional configuration.

The CCATS consists of three Univac 494 general purpose computers. Two of the computers are configured so that either may handle all of the input/output communications for two complete missions. One of the computers acts as a dynamic standby. The third computer is used for nonmission activities.

The RTCC is a group of five IBM 360 large scale, general purpose computers. Any of the five computers may be designated as the mission operations computer (MOC). The MOC performs all the required computations and display formatting for a mission. One of the remaining computers will be a dynamic standby. Another pair of computers may be used for a second mission or simulation.

SPACE VEHICLE TRACKING

From liftoff of the launch vehicle to insertion into orbit, accurate position data are required to allow the Impact Predictor (IP) and the RTCC to compute a trajectory and an orbit. These computations are required by the flight controllers to evaluate the trajectory, the orbit, and/or any abnormal situations to insure safe recovery of the astronauts. The launch tracking data are transmitted from the AFETR sites to the IP and thence to the RTCC via high-speed data communications circuits at the rate of ten samples per second (s/s). The IP also generates a state vector smooth sample which is transmitted to the RTCC at a rate of two s/s. (A state vector is defined as spacecraft inertial position and inertial rate of motion at an instant of time.) The message from the IP to the RTCC alternately contains one smoothed vector, then five samples of best radar data. Low speed

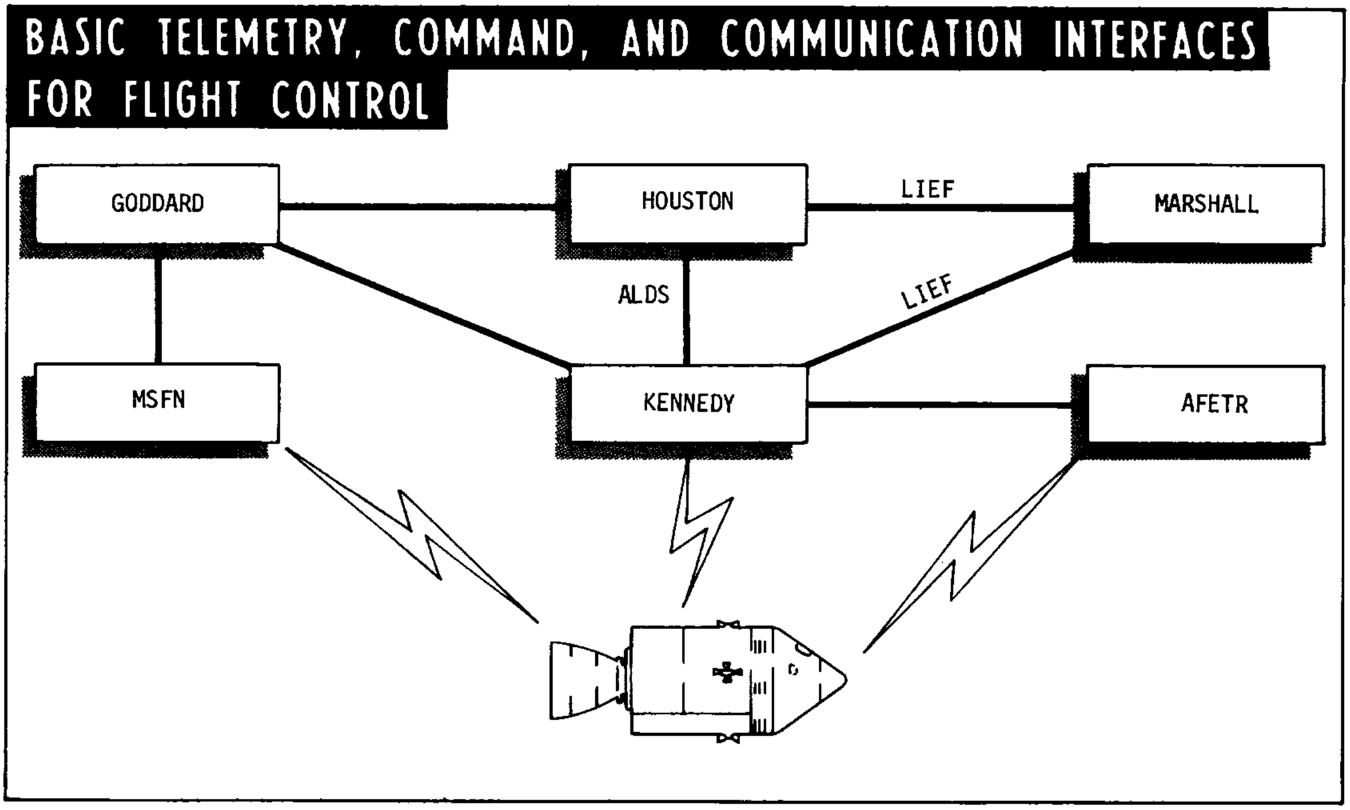


Figure 9-1

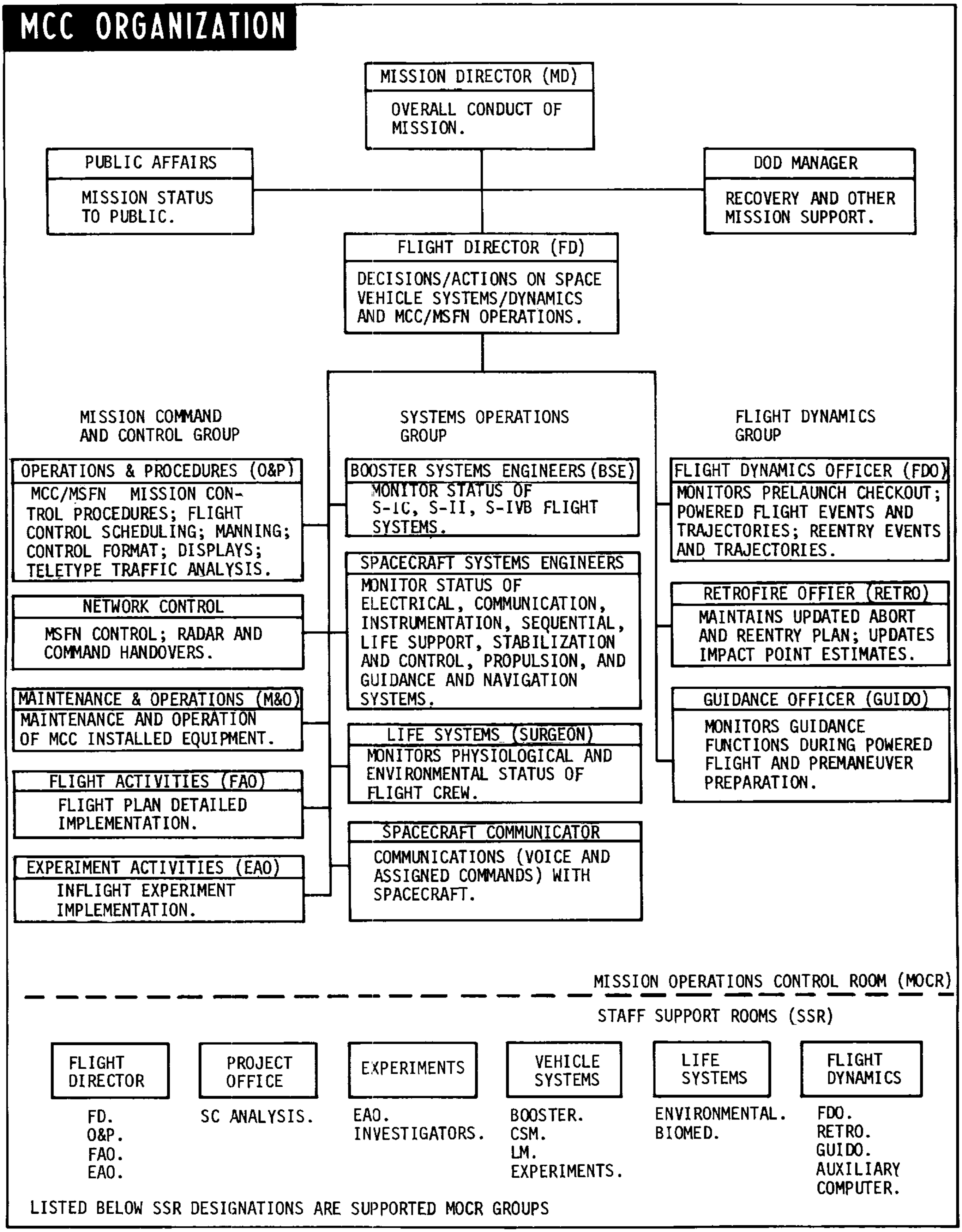

MCC ORGANIZATION
MISSION DIRECTOR (MD)
OVERALL CONDUCT OF MISSION.
PUBLIC AFFAIRS
MISSION STATUS TO PUBLIC.
DOD MANAGER
RECOVERY AND OTHER MISSION SUPPORT.
FLIGHT DIRECTOR (FD)
DECISIONS/ACTIONS ON SPACE VEHICLE SYSTEMS/DYNAMICS AND MCC/MSFN OPERATIONS.
MISSION COMMAND AND CONTROL GROUP
SYSTEMS OPERATIONS GROUP
FLIGHT DYNAMICS GROUP
OPERATIONS & PROCEDURES (O&P)
MCC/MSFN MISSION CONTROL PROCEDURES; FLIGHT CONTROL SCHEDULING; MANNING; CONTROL FORMAT; DISPLAYS; TELETYPE TRAFFIC ANALYSIS.
NETWORK CONTROL
MSFN CONTROL; RADAR AND COMMAND HANDOVERS.
MAINTENANCE & OPERATIONS (M&O)
MAINTENANCE AND OPERATION OF MCC INSTALLED EQUIPMENT.
FLIGHT ACTIVITIES (FAO)
FLIGHT PLAN DETAILED IMPLEMENTATION.
EXPERIMENT ACTIVITIES (EAO)
INFLIGHT EXPERIMENT IMPLEMENTATION.
BOOSTER SYSTEMS ENGINEERS (BSE)
MONITOR STATUS OF S-IC, S-II, S-IVB FLIGHT SYSTEMS.
SPACECRAFT SYSTEMS ENGINEERS
MONITOR STATUS OF ELECTRICAL, COMMUNICATION, INSTRUMENTATION, SEQUENTIAL, LIFE SUPPORT, STABILIZATION AND CONTROL, PROPULSION, AND GUIDANCE AND NAVIGATION SYSTEMS.
LIFE SYSTEMS (SURGEON)
MONITORS PHYSIOLOGICAL AND ENVIRONMENTAL STATUS OF FLIGHT CREW.
SPACECRAFT COMMUNICATOR
COMMUNICATIONS (VOICE AND ASSIGNED COMMANDS) WITH SPACECRAFT.
FLIGHT DYNAMICS OFFICER (FDO)
MONITORS PRELAUNCH CHECKOUT; POWERED FLIGHT EVENTS AND TRAJECTORIES; REENTRY EVENTS AND TRAJECTORIES.
RETROFIRE OFFIER (RETRO)
MAINTAINS UPDATED ABORT AND REENTRY PLAN; UPDATES IMPACT POINT ESTIMATES.
GUIDANCE OFFICER (GUIDO)
MONITORS GUIDANCE FUNCTIONS DURING POWERED FLIGHT AND PREMANEUVER PREPARATION.
MISSION OPERATIONS CONTROL ROOM (MOCR)
STAFF SUPPORT ROOMS (SSR)
FLIGHT DIRECTOR
FD. O&P. FAO. EAO.
PROJECT OFFICE
SC ANALYSIS.
EXPERIMENTS
EAO. INVESTIGATORS.
VEHICLE SYSTEMS
BOOSTER. CSM. LM. EXPERIMENTS.
LIFE SYSTEMS
ENVIRONMENTAL. BIOMED.
FLIGHT DYNAMICS
FDO. RETRO. GUIDO. AUXILIARY COMPUTER.
LISTED BELOW SSR DESIGNATIONS ARE SUPPORTED MOCR GROUPS

Figure 9-2

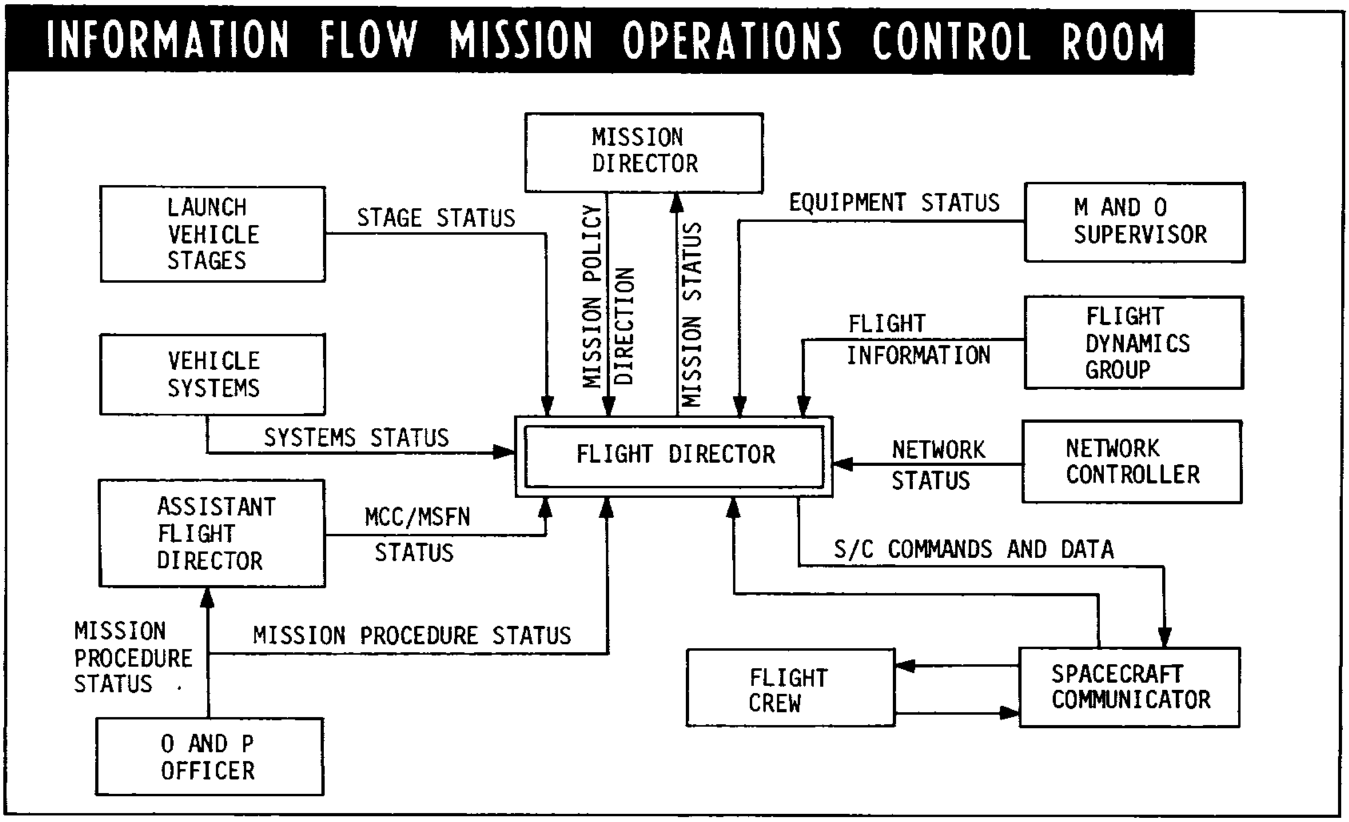


Figure 9-3

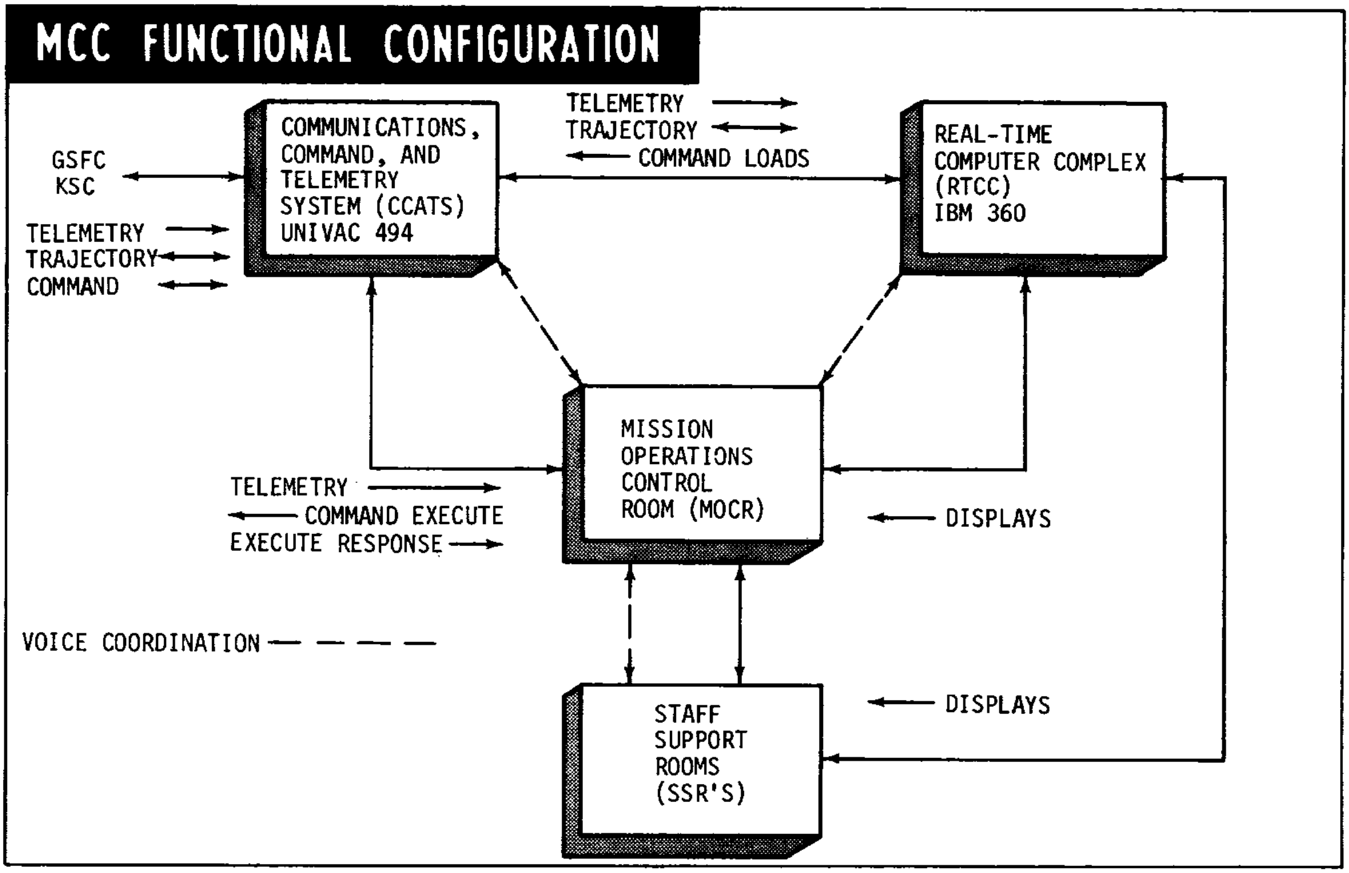


Figure 9-4

tracking data are also transmitted via teletype (TTY) to MCC at a rate of one sample per six seconds from all stations actively tracking the spacecraft. Figure 9-5 shows data flow from liftoff to orbital insertion.

As the launch vehicle is boosting the spacecraft to an altitude and velocity that will allow the spacecraft to attain earth orbit, the trajectory is calculated and displayed on consoles and plotboards in the MOCR and SSR's. Also displayed are telemetry data concerning status of launch vehicle and spacecraft systems. If the space vehicle deviates excessively from the nominal flight path, or if any critical vehicle condition exceeds tolerance limits, or the safety of the astronauts or range personnel is endangered, a decision is made to abort the mission.

During the orbit phase of a mission, all stations that are actively tracking the spacecraft will transmit the tracking data through GSFC to the RTCC by teletype at a frequency of one sample every six seconds. If a thrusting maneuver is performed by the spacecraft, high-speed tracking data at the rate of five s/s is transmitted in addition to the teletype data.

Any major maneuver during a mission is planned to occur during or just prior to acquisition by a tracking station that can relay high-speed tracking data to the MCC. This is to insure that data is available for the calculation of the new spacecraft orbit and ephemeris.

Approximately 25 minutes prior to anticipated spacecraft acquisition by a tracking station, a message giving time, antenna position coordinates, and range is dispatched to that station. This information is computed from the ephermeris and is used by station personnel to preposition the antenna and enable spacecraft acquisition with minimum delay.

COMMAND SYSTEM

The Apollo ground command systems have been designed to work closely with the telemetry and trajectory systems to provide flight controllers with a method of "closed-loop" commanding. This does not exclude the "man in the loop." The astronauts and flight controllers act as vital links in the commanding operation. For example, analysis of spacecraft data by flight controllers results in a command to alter an observed condition. The effects of the command will be observed in subsequent data presented to the flight control team. This verifies the correct execution of the command and closes the loop. In some cases, such as maneuvering the spacecraft, the command may not be immediately executed, but instead, the astronaut will maneuver the spacecraft at the optimum time specified by the command. The resulting telemetry and trajectory will reflect the maneuver and close the commanding loop.

To prevent spurious commands from reaching the space vehicle, switches on the Command Module console block uplink data from the onboard computers. At the appropriate times the flight crew will move the switches from the BLOCK to ACCEPT position and thus permit the flow of uplink data.

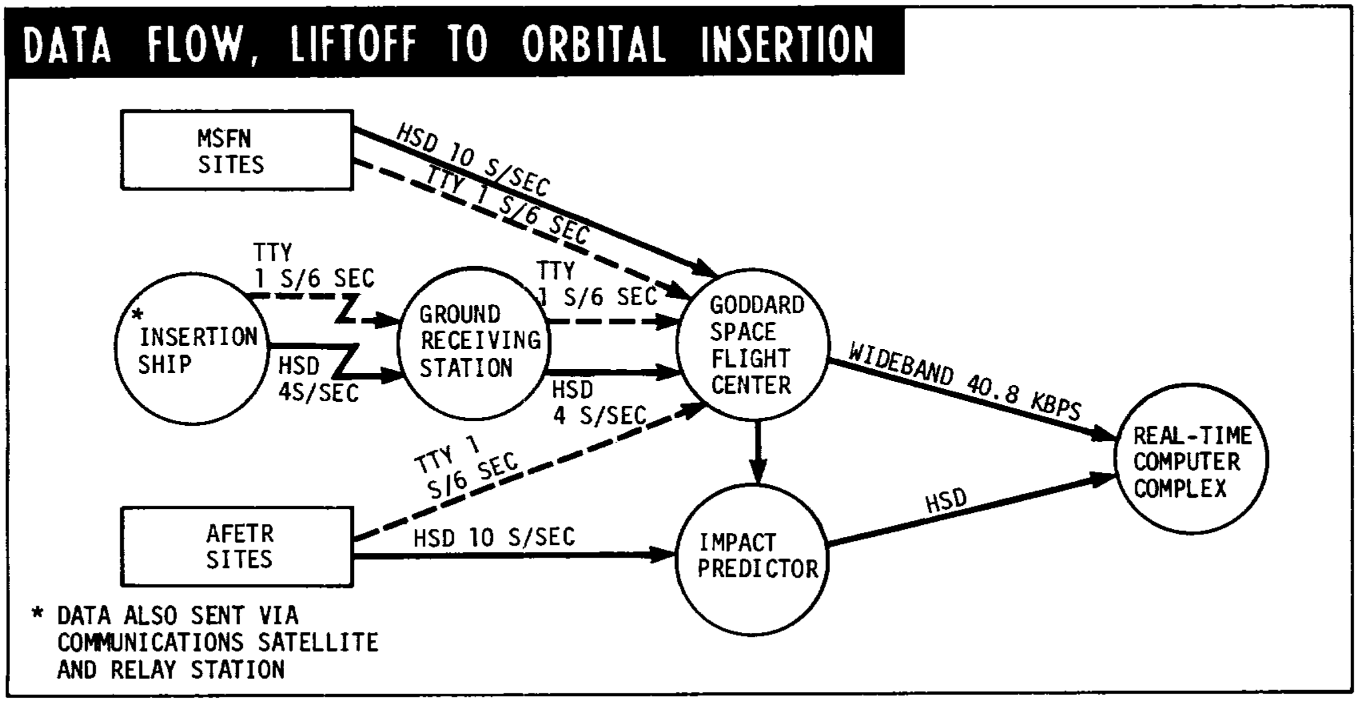


Figure 9-5

With a few exceptions commands to the space vehicle fall into two categories, real-time commands and command loads (also called computer loads, computer update, loads, or update). Among the exceptions is the "clock word" command. This command is addressed to the onboard timing system and is used when the downlink (telemetry) time word and the ground timing system are out of tolerance.

Real-Time Commands

Real-time commands are used to control space vehicle systems or subsystems from the ground. The execution of a real-time command results in immediate reaction by the affected system. Real-time commands correspond to unique space vehicle hardware and therefore require careful premission planning to yield commands which provide alternate systems operation in the event of an anticipated failure. Premission planning also includes commands necessary to initiate mission contingency plans. Descriptions of several real-time commands used by the Booster System Engineer follow.

The ALTERNATE SEQUENCE real-time commands permit the onboard Launch Vehicle Digital Computer (LVDC) to deviate from its normal program and enter a predefined alternate sequence of program steps. For example, should the S-II engines fail to ignite or the engines shut down prematurely, an alternate switch selector sequence would jettison the S-II stage and the S-IVB stage would be used to attain the normal parking orbit.

The SEQUENCE INHIBIT real-time command provides the capability to inhibit a programed sequence, usually a maneuver. Each sequence must be separately inhibited with the command being processed immediately after LVDC acceptance. The maneuvers may be inhibited in any random order required during the mission. If an update for a particular command is received after the inhibit for that command, the inhibit is removed and the maneuver will occur at the update time specified.

Other examples of real-time commands are: LH_2 VENT CLOSED, LH_2 VENT OPEN, LOX VENT OPEN, TERMINATE, SET ANTENNA OMNI, SET ANTENNA LO-GAIN, and SET ANTENNA HI-GAIN.

Real-time commands are stored prior to the mission in the Command Data Processor (CDP) at the applicable command site. The CDP, a Univac 642B general purpose digital computer, is programed to format, encode, and output commands when a request for uplink is generated.

Command Loads

Command loads are generated by the real-time computer complex on request of flight controllers. Command loads are based on the latest available telemetry and/or trajectory data. Due to the nature of these commands, the data structure cannot be determined prior to the mission but must be calculated as a result of real-time data. A command load, for example, may define the exact conditions under which a thrust may be applied that will change a faulty orbit to the desired orbit.

The RTCC operating personnel take data supplied by the flight controllers requesting the command load and by selecting the appropriate computer program cause the computer to "make up" a command load. When the load is "ready", it is reviewed by the responsible flight controller via the display system. When the load is approved it is transferred via NASCOM in the form of high-speed data and/or teletype messages to the appropriate site and stored in its command data processor. The CCATS will retain the load in memory where it is available for retransmission should difficulties be encountered in the transfer procedure. When the command load is properly stored in the site's command data processor, a load validation message is sent to the CCATS and to the flight controller.

Flight controllers typically required to generate a command load include the Booster Systems Engineer (BSE), the Flight Dynamics Officer (FDO), the Guidance Offier (GUIDO), and the Retrofire Officer (RETRO).

Prior to the acquistion of the space vehicle by a site, the flight controllers requiring command capability during the pass indicate their requirements to the Real-Time Command Controller (R/T CMD) in the CCATS area. The R/T CMD will enable the circuitry to permit the command function for console/site combinations.

When the space vehicle has been acquired by the site, it will be announced over one of the voice coordination loops and each flight controller will execute his commands according to the priority assigned by real-time decision. Disposition of each command will be indicated by the light logic of the command panel. These indicator lights are operated by CCATS from the verification and/or reject messages from the site. Typical Command loads (BSE) are described below.

The SECTOR DUMP command causes the LVDC to telemeter the entire contents of one LVDC memory sector, or a series of memory sectors within the same memory module. For example, this command is used to telemeter the memory sector in which the navigation update parameters are stored. The real-time TERMINATE command may be used to halt a sector dump before the last block of data is telemetered. The SECTOR DUMP command applies to the orbital phase.

The NAVIGATION UPDATE command permits loading of six navigation parameters and an execution time into the LVDC.

The SEQUENCE INITIATE UPDATE command permits update of stored values for the time of initiation of each of the preprogramed maneuvers specified for the mission. The prestored values will be adjusted immediately upon the receipt of the update command.

Other examples of command loads are: TIME BASE UPDATE, SLV PRELAUNCH TARGET UPDATE, and SLV ORBIT TARGET UPDATE.

DISPLAY AND CONTROL SYSTEM

MCC is equipped with facilities which provide for the input of data from the MSFN and KSC over a combination of high-speed data, low-speed data, wide-band data, teletype, and television channels. This data is computer processed for display to the flight controllers. With this displayed data, detailed mission control by the MOCR and detailed support in the various specialty areas by the SSR's are made possible.

Display System

Several methods of displaying data are used including television (projection TV, group displays, close circuit TV, and TV monitors), console digital readouts, and event lights. The display and control system interfaces with the RTCC and includes computer request, encoder multiplexer, plotting display, slide file, digital-to-TV converter, and telemetry event driver equipments (see figure 9-6).

The encoder multiplexer receives the display request from the console keyboard and encodes it into digital format for transmission to the RTCC.

The converter slide file data distributor routes slide selection data from the RTCC to reference slide files and converter slide files; receives RTCC control data signals required to generate individual console television displays and large scale projection displays; and distributes control signals to a video switching matrix to connect an input video channel with an output television viewer or projector channel.

The digital-to-television conversion is accomplished by processing the digital display data into alphanumeric, special symbol, and vector displays for conversion into video signals. This process produces analog voltages which are applied to the appropriate element of a character-shaped beam cathode ray tube. The resultant display image on the face of the cathode ray

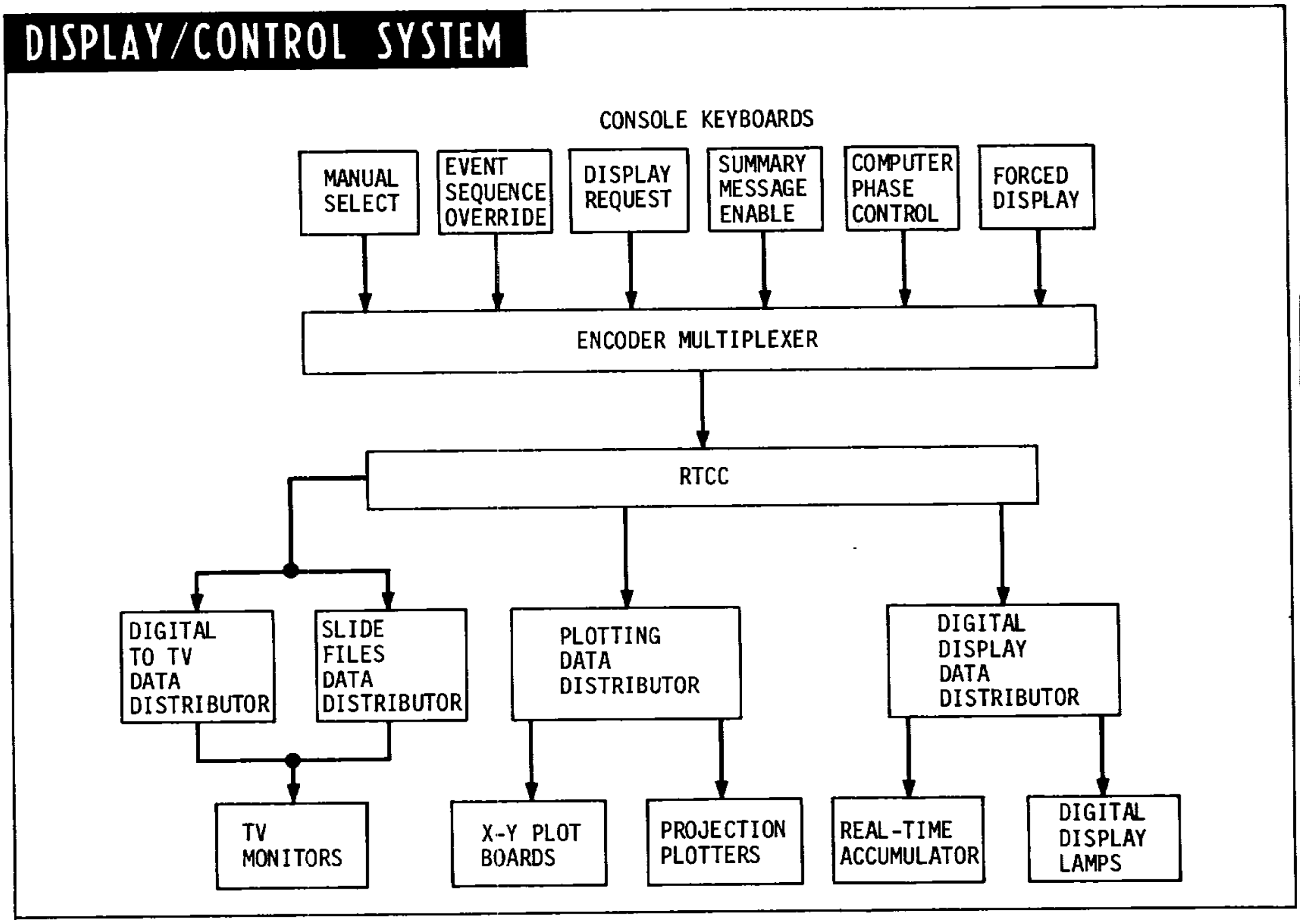


Figure 9-6

IX MISS. CONT.

tube is optically mixed with the slide file image and viewed by a television camera, which transmits the mixed images to the TV monitors and projectors. The digital-to-TV data consists of preprogramed computer-generated dynamic data formats which are processed and combined with background data on film sides.

Control System

A control system is provided for flight controllers to exercise their respective functions for mission control and technical management. This system is comprised of different groups of consoles with television monitors, request keyboards, communications equipment, and assorted modules added as required to provide each operational position in the MOCR with the control and display capabilities required for the particular mission. The console components are arranged to provide efficient operation and convenience for the flight controller. The console configuration for the Booster System Engineer is shown in figure 9-7. Brief descriptions of the console modules are contained in the following paragraphs.

The event indicator modules display discrete mission events, system modes, equipment modes, and vehicle system status. Each BSE event module consists of 18 bilevel indicators which permit a total of 36 event positions per module. Another type of event module has the capability to display up to 72 events. The signals affecting the lights are telemetry inputs from the space vehicle.

The ground elapsed time module shows the elapsed time from liftoff.

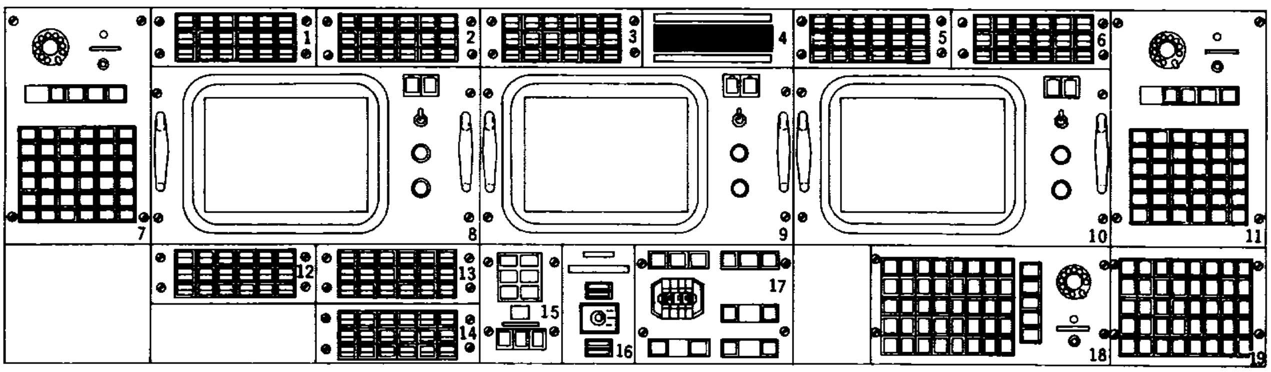
The communications module provides rapid access to internal, external, and commercial voice communications circuits. Flight controllers may monitor as well as talk over these circuits.

The command module provides the flight controller with the means to select and initiate realtime commands and command loads for transmission to the space vehicle. The module also indicates receipt/rejection of commands to the space vehicle and verification of proper storage of command loads in a site command data processor. The modules are made up of pushbutton indicators (PBI's) which are labeled according to their function.

The status report module (SRM) provides flight controllers in the MOCR with the means to report mission status to the flight director and assistant flight director and to review systems status in the SSR's.

The abort request indicator provides the capability by toggle switch action to indicate an

TYPICAL CONFIGURATION - BOOSTER SYSTEMS ENGINEERS CONSOLE



1. EVENT INDICATORS (S-IC/S-II)
2. EVENT INDICATOR (VEHICLE TELEMETRY STATUS, EDS, COMMAND, RANGE SAFETY)
3. EVENT INDICATORS (S-IVB)
4. GROUND ELAPSED TIME MODULE
5. EVENT INDICATORS (GUIDANCE AND NAVIGATION, ATTITUDE CONTROL)
6. EVENT INDICATORS (GUIDANCE AND NAVIGATION, ATTITUDE CONTROL)
7. COMMUNICATIONS MODULE
8. TV MONITOR
9. TV MONITOR
10. TV MONITOR
11. COMMUNICATIONS MODULE
12. COMMAND MODULE
13. COMMAND MODULE
14. COMMAND MODULE
15. STATUS REPORT MODULE
16. ABORT REQUEST MODULE
17. MANUAL SELECT KEYBOARD
18. COMMUNICATIONS MODULE
19. SUMMARY MESSAGE ENABLE KEYBOARD

Figure 9-7

abort condition. This produces a priority command to the spacecraft.

The manual select keyboard (MSK) permits the flight controller to select a TV channel, a computer-generated display format, or a reference file for viewing on the TV monitor. The desired item is selected by use of a PBI three-mode switch (TV channel, reference file, a display request) and a select number thumbwheel encoder. BSE consoles may obtain a hardcopy of a display by use of a PBI. The hardcopy is delivered to the console via a pneumatic tube system.

The summary message enable keyboard (SMEK) permits the flight controller to instruct the RTCC to strip out selected data from telemetry inputs and to format this data into digital-to-TV summary displays. The SMEK is also used to instruct the RTCC to convert data into specific teletype messages to designated MSFN sites. The module contains appropriately labeled PBI's.

The TV monitor module provides viewing of computer-generated displays, reference file data, closed circuit TV within the MCC and KSC, and commercial TV.

Console modules not illustrated in figure 9-7 but used in MOCR and SSR consoles are described below.

The analog meter module displays parameters in engineering units (voltage, degrees, etc). Up to 15 measurements, determined prior to the mission can be displayed. There are movable markers on each meter which are manually set to show the nominal value of the parameter as well as the upper and lower limits. When a parameter exceeds the established tolerance, a red warning light at the bottom of the meter is lit. The light extinguishes when the parameter returns to tolerance or the exceeded limit is manually extended.

The forced display module (FDK) indicates to the flight controller a violation of preprogramed limits of specific analog parameters as determined from the incoming data. When an out of tolerance parameter exists, a warning will be displayed by illuminating the appropriate PBI in the module. When the flight controller acknowledges the lit PBI by depressing it, a readout indicator on the FDK will display a four digit code which identifies the display format on which the out of tolerance parameter appears.

The display request keyboard (DRK) provides a fast means of requesting the RTCC for a specific display format. The display is called up by depressing the appropriately labeled PBI. This keyboard provides the same capability as the MSK in the "display request mode" except that the callup is faster in that thumbwheel selection is not required.

CONTINGENCY PLANNING AND EXECUTION

Planning for a mission begins with the receipt of mission requirements and objectives. The planning activity results in specific plans for prelaunch and launch operations, preflight training and simulation, flight control procedures, flight crew activities, MSFN and MCC support, recovery operations, data acquisition and flow, and other mission related operations. Simulations are planned and performed to test procedures and train flight control and flight crew teams in normal and contingency operations. The simulation and training exercises result in a state of readiness for the mission. Mission documentation covering all aspects of the mission is developed and tested during the planning and training period. Included in this documentation are the mission rules.

MISSION RULES

Mission rules are a compilation of rules governing the treatment of contingency situations. The purpose of the mission rules is to outline preplanned actions to assist in making rapid real-time decisions during prelaunch, flight, and recovery operations. The mission rules are based upon the mission objectives and on the objective of maintaining a high degree of confidence in crew safety during mission implementation. The mission rules categorize the degree of importance assigned to space vehicle/operational support elements as follows:

A mandatory item (M) is a space vehicle element or operational support element that is essential for accomplishment of the primary mission, which includes prelaunch, flight, and recovery operations that ensure crew safety and effective operational control as well as attainment of the primary mission objectives.

A highly desirable item (HD) is a space vehicle or operational support element that supports and enhances the accomplishment of the primary mission or is essential for accomplishment of the secondary mission objectives.

Redline values are the maximum and/or minimum limits of a critical parameter (redline function) necessary to describe vehicle, system, and component performance and operation. Redline functions are mandatory items.

Launch Mission Rules

Launch mission rules cover the following listed items and other information as appropriate:

1. Mandatory and highly desirable onboard instrumentation required to collect data for flight control purposes or postflight evaluation;
2. Mandatory and highly desirable onboard instrumentation required to verify that

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the space vehicle is ready for launch;

3. Redline values defining upper and lower limits of pressure, temperature, voltage, current, operating time, etc., for any system/subsystem essential to mission success;

4. Mandatory and highly desirable range and instrumentation support required to prepare and launch the space vehicle and for postflight analysis of launch;

5. Range safety requirements and instrumentation established by the Air Force Eastern Test Range;

6. Wind and weather restrictions for launch;

7. Long-range camera coverage required for launch;

8. Launch window definition and launch window rules pertaining to launch operations;

9. Space vehicle functional sequence;

10. Time span before launch during which manual cutoff will not be attempted.

Flight Mission Rules

Flight mission rules cover the following listed items and also medical decision rules for appropriate mission go/no-go points:

1. Mandatory and highly desirable instrumentation for control of the space vehicle after liftoff;

2. Space vehicle nominal and non-nominal subsystem performance in accordance with alternate mission capability;

3. Trajectory and guidance;

4. Mandatory and highly desirable items of mission support in the MCC;

5. Mandatory and highly desirable range and MSFN support required to support the mission after liftoff, and for subsequent analysis and evaluation;

6. Rules relating to human or medical aspects of manned flight;

7. Recovery restrictions;

8. Launch window rules pertaining to items such as time of liftoff, launch azimuth, recovery, spacecraft performance limitations.

VEHICLE MISSION ABORT CAPABILITY

Section III of this manual, dealing with the emergency detection system, describes the manual and automatic capabilities for mission abort designed into the Saturn-Apollo system. Also described in Section III are the abort modes and limits, and the emergency procedures related to mission abort.

Time critical aborts must be initiated onboard the spacecraft because sufficient time is not available for response by the MCC ground based flight controllers.

The detection of slowly diverging conditions which may result in an abort is the prime responsibility of MCC. In the event such conditions are discovered MCC requests abort of the mission or, circumstances permitting, sends corrective commands to the vehicle or requests corrective flight crew action.

In the event of a non-catastrophic contingency, MCC recommends alternate flight procedures, and mission events are rescheduled to derive maximum benefit from the modified mission.

ABORT GROUND RULES

Flight crew safety shall take precedence over the accomplishment of mission objectives.

The Command Pilot of a manned mission may initiate such inflight action as he deems necessary for crew safety.

The Launch Operations Manager may send an abort request signal from the time the launch escape system (LES) is armed until the space vehicle reaches sufficient altitude to clear the top of the umbilical tower.

From liftoff to tower clear, the Launch Director and Flight Director have concurrent responsibility for sending an abort request.

Control of the space vehicle passes from the Launch Director to the Flight Director when the space vehicle clears the top of the tower.

In the Mission Control Center, the Flight Director, Flight Dynamics Officer, and Booster Systems Engineer have the capability to send an abort request signal.

Where possible, all manual abort requests from the ground during flight will be based on two independent indications of the failure.

LAUNCH VEHICLE CONTINGENCIES/REACTIONS

Malfunctions which could result in loss of the space vehicle are analyzed and a mission rule is developed to respond to the malfunction. The contingencies and corresponding reactions are incorporated into the premission simulations and the mission rules are refined as required. Typical contingencies/reactions on which mission rules are based are listed in figure 9-8.

TYPICAL INFLIGHT CONTINGENCIES/REACTIONS

CONTINGENCY	FLIGHT TIME MIN:SEC	EFFECT	SENSORS OR DISPLAYS	MISSION RULE DATA
Loss of thrust for any single engine	0:00-0:00.25	Pad fallback will occur prior to 2.63 seconds	1. Thrust OK Light 2. Loss of vertical velocity	Manual abort by LES with two cues 1. Thrust OK Light 2. Indication of loss of vertical velocity
Loss of thrust for single control engine	0:00-0:01	Collision with hold-down post will occur prior to 3.3 seconds	1. Thrust OK Light 2. Visual	Manual abort by LES with two cues 1. Thrust OK Light 2. Visual
Loss of thrust for single engine (#1 or #2)	0:01-0:04	Tower collision will occur prior to 10.7 seconds	1. Thrust OK Light 2. Attitude error more negative than 3 degress	Manual abort by LES with two cues 1. Thrust OK Light 2. Attitude error limits exceeded
Loss of thrust for single control engine	0:00-0:10	Possible false abort prior to 50 seconds if not covered by above contingencies	1. Thrust OK Light 2. Q-Ball may exceed 3.2 psi (prior to 50 seconds)	Recommendation required to assure no false abort
Loss of thrust any single engine	0:10-0:55	No effects	1. Thrust OK Light	No abort required
Loss of thrust any single control engine	0:55-1:25	Possible loss of control of vehicle resulting in structural breakup within 0.6 seconds after malfunction	1. Attitude rate exceeds 4°/second 2. Thrust OK Light 3. Q-Ball exceeds 3.2 psi	Automatic abort when attitude rate is exceeded. Manual abort by LES with two cues 1. Thrust OK Light 2. Q-Ball limits exceeded
Loss of thrust any single engine	1:25-2:31	No effect	1. Thrust OK Light	No abort required
	2:31-8:34	No effect	1. Thrust OK Light	No abort required
	S-IVB Burn	Complete loss of thrust	1. Thrust OK Light	Abort by SPS
Loss of thrust any two engines	0:00-2:31	Loss of control resulting in structural breakup	1. Two thrust OK switches result in automatic abort	Loss of thrust for any two engines results in automatic abort with LES
Loss of thrust 2 adjacent control engines	2:31-6:55	Possible loss of control	1. Two thrust OK switches	Manual abort by LES prior to tower jettison and by SPS after tower jettison
	6:55-8:30	Loss of IGM in S-IVB flight	1. Two thrust OK switches	Direct staging
	8:30 8:45	No effect	1. Two thrust OK switches	Continue mission
Loss of thrust 2 opposite control or center engines and one control engine	2:31-8:30	Possible loss of IGM in S-IVB flight	1. Two thrust OK switches	Direct staging
	8:30 8:45	No effect	1. Two thrust OK switches	Continue mission

Figure 9-8 (Sheet 1 of 5)

TYPICAL INFLIGHT CONTINGENCIES/REACTIONS (CONTINUED)				
CONTINGENCY	FLIGHT TIME MIN:SEC	EFFECT	SENSORS OR DISPLAYS	MISSION RULE DATA
One actuator inoperative (null position)	0:00 8:45	No effect for S-IC or S-II as other control engines will control flight.	1. No sensor (none required)	No abort required
One actuator inoperative (null position)	S-IVB boost	Loss of control	1. Deviation from trajectory	1. Manual abort by SPS.
Any single actuator hardover (pitch or yaw)	0:00-0:00.2	Collision with holddown post will occur.	1. Visual	Manual abort upon contact
Any single actuator hardover(yaw negative only)	0:00-0:01.5	Tower collision will occur prior to 9.4 seconds	1. Roll attitude error will exceed 5° 2. Yaw attitude error exceeds 3° 3. Actuator position	Manual abort by LES with two cues 1. Roll attitude error limits exceeded 2. Yaw attitude error limits exceeded 3. Engine position
Any single actuator hardover (pitch or yaw)	0:00-0:55	No effect unless covered by above	1. Actuator position	No abort required.
Any single actuator hardover (pitch or yaw)	0:55-1:25	Possible loss of control resulting in structural breakup	1. Attitude rate exceeds 4°/second 2. Roll attitude error exceeds 5° 3. Q-Ball exceeds 3.2 psi 4. Actuator position	Automatic abort by LES when attitude rate limit is exceeded. Manual abort by LES with two cues 1. Roll attitude limit exceeded 2. Q-Ball limits exceeded 3. Engine position
Any single actuator hardover	1:25-2:31	No effect	1. Actuator position	No abort required.
	2:31-8:45	No effect	1. Actuator position	No abort required.
	S-IVB Burn Phase	Loss of control	1. Attitude rate exceeds 10°/second 2. L/V rate light 3. Acutator position	Manual abort by SPS with two cues 1. Attitude rate limit is exceeded 2. Attitude error limit is exceeded 3. Engine position
Engine Hardover	S-IVB Restart	Loss of control if restarted	1. Engine position (ground confirmation) 2. Pressure (ground) 3. Temperature of hydraulic fluid (ground) 4. Lead thrust would cause vehicle rotation (by crew)	Do not attempt restart. Manual abort by SPS.

Figure 9-8 (Sheet 2 of 5)

TYPICAL INFLIGHT CONTINGENCIES/REACTIONS (CONTINUED)				
CONTINGENCY	FLIGHT TIME MIN:SEC	EFFECT	SENSORS OR DISPLAYS	MISSION RULE DATA
Loss of both actuators for same engine	0:00 2:31	Loss of control resulting in structural breakup if engine goes to roll corner	1. Roll rate exceeds 20° 2. Q-Ball exceeds 3.2 psi 3. Actuator position	Automatic abort by LES when roll rate limit is exceeded. Manual abort by LES with two cues 1. Q-Ball limit exceeded 2. Engine position
	2:31-8:45	No effect	1. Actuator Position	No abort required.
	S-IVB Burn Phase	Loss of control	1. Attitude rate exceeds 10°/second 2. L/V rate light 3. Actuator position	Manual abort by SPS 1. Attitude rate limit is exceeded 2. Attitude error limit is exceeded 3. Engine position
Loss of Inertial attitude	0:00-1:30	Loss of guidance control resulting in structural breakup (backup system to be incorporated to prevent loss of control)	1. Attitude rate exceeds 4°/second 2. Guidance failure light 3. Q-Ball exceeds 3.2 psi	Automatic abort by LES when attitude rate limit is exceeded. Manual abort with two cues (without backup control) 1. Guidance failure light 2. Q-Ball limit exceeded
	1:30-2:31	Loss of guidance with no structural failures (backup system to be incorporated to prevent loss of control)	1. Attitude rate exceeds 4°/second 2. Guidance failure light 3. Q-Ball exceeds 3.2 psi	Automatic abort by LES when attitude rate limit is exceeded. Manual abort by LES with two cues (without backup control) 1. Guidance failure light 2. Q-Ball limits exceeded
	2:31 8:45	Loss of guidance	1. Attitude deviation exceeds 20° 2. Guidance failure light	Manual abort by LES prior to tower jettison or by SPS after tower jettison 1. Attitude deviation is exceeded 2. Guidance failure light
Loss of inertial attitude	S-IVB Burn Phase	Loss of guidance	1. Guidance failure light 2. Attitude deviation exceeds 20° first burn, 7° second burn	Manual abort by SPS with single cue 1. Guidance failure light 2. Attitude deviation
Loss of inertial velocity	All Stages	No effect		No abort required

Figure 9-8 (Sheet 3 of 5)

IX

TYPICAL INFLIGHT CONTINGENCIES/REACTIONS (CONTINUED)				
CONTINGENCY	FLIGHT TIME MIN:SEC	EFFECT	SENSORS OR DISPLAYS	MISSION RULE DATA
Loss of attitude error commands	All Stages	Loss of control	1. Attitude rate exceeds 4°/second for S-IC 2. Attitude rate exceeds 10°/second for S-II and S-IVB 3. Attitude error exceeds 5° for S-IC or 20° for S-II and S-IVB 4. Ground confirmation	Automatic abort by LES during S-IC required when attitude rate limit is exceeded. Manual abort by LES prior to tower jettison and SPS after tower jettison with a single cue 1. Attitude error limits exceeded 2. Ground confirmation
Attitude error command saturated	All Stages	Loss of control	1. Attitude rate exceeds 4°/second in pitch or yaw or 20°/second in roll for S-IC 2. Attitude rate exceeds 10°/second for S-II and S-IVB 3. Attitude error exceeds 5° for S-IC 4. L/V rate light	Automatic abort by LES during S-IC required when attitude rate limit is exceeded. Manual abort by LES prior to tower jettison and SPS after tower jettison with a single cue 1. Attitude error limits exceeded
Loss of attitude command signal	All stages	Loss of guidance	1. Attitude rate exceeds 4°/second for S-IC 2. Attitude rate exceeds 10°/second for S-II and S-IVB 3. Attitude error exceeds 5° for S-IC or 20° for S-II or S-IVB 4. Ground confirmation	Automatic abort by LES during S-IC required when attitude rate limit is exceeded. Manual abort by LES prior to tower jettison and SPS after tower jettison with a single cue 1. Attitude error limits exceeded 2. Ground confirmation
Attitude rate signal saturated	All stages	Loss of control resulting in loss of vehicle	1. Attitude rate exceeds 4°/second in pitch or yaw or 20°/second in roll during S-IC 2. Attitude limit and attitude rate limit during S-II and S-IVB	Automatic abort by LES during S-IC and manual abort during S-II and S-IVB required when attitude and rate limits are exceeded
Loss of attitude rate signal	All stages	Loss of control resulting in loss of vehicle	1. Attitude rate exceeds 4°/second during S-IC 2. Attitude limit and attitude rate limit during S-II and S-IVB 3. Attitude oscillatory	Automatic abort by LES during S-IC and manual abort during S-II and S-IVB required when attitude and rate limits are exceeded
Lack of S-IC first plane separation	2:31	No S-II ignition possible	Lack of S-II thrust	Early S-IVB staging by crew (orbit not possible)

Figure 9-8 (Sheet 4 of 5)

TYPICAL INFLIGHT CONTINGENCIES/REACTIONS (CONTINUED)

CONTINGENCY	FLIGHT TIME MIN:SEC	EFFECT	SENSORS OR DISPLAYS	MISSION RULE DATA
Lack of S-II second plane separation	2:57	Probable subsequent loss of vehicle due to excessive temperature	Second plane separation indicator (redundant indication)	Shutdown prior to overheat and early S-IVB staging required by crew only, limited time between normal jettison time and excessive temperature may preclude ground information backup
Partial S-II second plane separation	2:57	Probable subsequent loss of vehicle due to excessive temperature	Second plane separation indicator (redundant indication)	Shutdown prior to overheat and early S-IVB staging required by crew only, limited time between normal jettison time and excessive temperature may preclude ground information backup
Lack of S-II/S-IVB plane separation	8:34	No S-IVB ignition	Lack of S-IVB thrust	SPS abort required
Partial S-II/S-IVB plane separation	8:34	No S-IVB ignition	Lack of S-IVB thrust	SPS abort required
Loss of both APS modules, S-IVB	S-IVB burn phases	Lack of roll control	1. Roll attitude may be excessive	Manual control by crew with RCS when roll attitude is excessive
Loss of one APS module, S-IVB	S-IVB burn phases	Negligible	1. No indication.	Manual control by crew with RCS when roll attitude is excessive
LET fails to jettison	3:03	Vehicle may go unstable during S-IVB flight	1. Visual by crew 2. Attitude rate 3. Vehicle oscillatory	Shutdown S-IVB where vehicle becomes oscillatory. Crew to use emergency procedures for removal of LET if orbit was achieved.

Figure 9-8 (Sheet 5 of 5)

VEHICLE FLIGHT CONTROL PARAMETERS

In order to perform flight control monitoring functions, essential data must be collected, transmitted, processed, displayed, and evaluated to determine the space vehicle's capability to start or continue the mission. Representative parameters included in this essential data will be briefly described in the following paragraphs.

PARAMETERS MONITORED BY LCC

The launch vehicle checkout and prelaunch operations monitored by the Launch Control Center (LCC) were briefly discussed in Section VIII of this manual. These operations determine the state of readiness of the launch vehicle, ground support, telemetry, range safety, and other operational support systems. During the final countdown hundreds of parameters are monitored to ascertain vehicle, system, and component performance capability. Among these parameters are the "redlines". The redline values must be within the predetermined limits or the countdown will be halted. Typical redlines are fuel and oxidizer tank ullage pressure, GN_2 and helium storage sphere pressure, hydraulic supply pressures, thrust chamber jacket temperatures, bus voltages, IU guidance computer operations, H_2 and O_2 concentrations, and S-IVB oxidizer and fuel recirculation pump flow. In addition to the redlines, there are a number of operational support elements such as ALDS, range instrumentation, ground tracking and telemetry stations, ground communications, and other ground support facilities which must be operational at specified times in the countdown.

PARAMETERS MONITORED BY BOOSTER SYSTEMS GROUP

The Booster Systems Group monitors launch vehicle systems (S-IC, S-II, S-IVB, and IU) and advises the flight director and flight crew of any system anomalies. It is responsible for abort actions due to failure or loss of thrust and overrate conditions; for confirming inflight power, stage ignition, holddown release, all engines go, roll and pitch initiate, engine cutoffs, etc; for monitoring attitude control and stage separations; and for digital commanding of LV systems.

Specific responsibilities in the group are allocated as follows:

1. BSE No. 1 has overall responsibility for the group, for commands to the launch vehicle, and for monitoring and evaluating the S-IC and S-II flight performance. Typical flight control parameters monitored include engine combustion chamber pressure, engine gimbal system supply pressure, fuel and oxidizer tank ullage pressure, helium storage tank pressure, engine actuator (yaw/pitch/roll) position, THRUST OK pressure switches, logitudinal acceleration, vent valve positions, engine ignition/cutoff, and various bus voltages.

2. BSE No. 2 supports BSE No. 1 in monitoring the S-II flight and assumes responsibility for monitoring the S-IVB burns. Parameters monitored are similar to those monitored by BSE No. 1.

3. BSE No. 3 monitors the attitude control, electrical, guidance and navigation, and IU systems. Typical parameters monitored include roll/pitch/yaw guidance and gimbal angles; angular rates; ST-124 gimbal temperature and bearing pressure; LVDC temperature; and various bus voltages.

The preceding flight controllers are located in the MOCR. The following are located in the vehicle system SSR.

4. Guidance and Navigation Systems (GND) engineer monitors the guidance, navigation, and digital (sequential) systems. The GND provides detailed support to BSE No. 3. Typical parameters monitored include ST-124 accelerometer and gyro pickups (X,Y,Z axes); and fixed position and fixed velocity X,Y,Z components).

5. Attitude Control and Stabilization Systems (ACS) engineer monitors the attitude control system, the S-IVB hydraulic and auxiliary propulsion systems, and the emergency detection system. The ACS provides detailed support to BSE's No. 2 and 3. Typical parameters monitored include hydraulic accumulator pressures, hydraulic reservoir piston position, attitude control fuel and oxidizer module temperatures, and rate excesses (pitch/roll/yaw).

6. Engine Systems Engineer monitors the S-II and S-IVB engine systems and O_2/H_2 burner. He provides detailed support to BSE No. 2. Among the parameters monitored are thrust chamber pressure, engine inlet lox and LH_2 pressure, pre-valve position, and O_2/H_2 burner chamber dome temperature.

7. Stage Systems Engineer monitors the pressurization, repressurization, bulkhead pressure differential and chilldown. He provides detailed support to the BSE No.2. Typical parameters monitored include helium tank pressure, start tank pressure/temperature, and common bulkhead pressure.

8. Electrical Network and Systems (ENS) engineer monitors electrical systems (all LV stages), IU environmental control system, and range safety systems (safing at orbital insertion). The ENS provides detailed support to BSE No. 3. Typical parameters monitored include exploding bridgewire voltages, sublimator inlet temperature, GN_2 regulator inlet temperature, and various bus voltages and currents.

9. Command Systems Engineer monitors the commands sent to the launch vehicle and advises BSE No. 1 on their status. In the event of rejection of a command by the onboard computer, he determines the cause of the rejection, i.e., improperly coded command, malfunction of the command system, malfunction of the computer, etc.

10. Consumables Engineer monitors status at all times of launch vehicle consumables including all high pressure spheres, APS propellants, and main stage propellants. He advises the BSE No. 2 of mission impact when consumables are depleted beyond predicted limits.

PARAMETERS MONITORED BY FLIGHT DYNAMICS GROUP

The Flight Dynamics Group monitors and evaluates the powered flight trajectory and makes the abort decisions based on trajectory violations. It is responsible for abort planning, entry time and orbital maneuver determinations, rendezvous planning, inertial alignment correlation, landing point prediction, and digital commanding of the guidance systems.

The MOCR positions of the Flight Dynamics Group include the Flight Dynamics Officer (FDO), the Guidance Officer (GUIDO), and the Retrofire Officer (RETRO). The MOCR positions are given detailed specialized support by the flight dynamics SSR.

The surveillance parameters measured by the ground tracking stations and transmitted to the MCC are computer processed into plotboard and digital displays. The flight dynamics group compares the actual data with premission calculated nominal data and is able to determine mission status. The surveillance parameters include slant range, azimuth and elevation angles, antenna polarization angle, and other data. From these measurements, space vehicle position, velocity, flight path angle, trajectory, ephemeris, etc., may be calculated. Typical plotboard displays generated from the surveillance parameters are altitude versus downrange distance, latitude versus longitude, flight path angle versus inertial velocity, and latitude versus flight path angle.

PARAMETERS MONITORED BY SPACECRAFT SYSTEMS GROUP

The Spacecraft Systems Group monitors and evaluates the performance of spacecraft electrical, optical, mechanical, and life support systems; maintains and analyzes consumables status; prepares mission log; coordinates telemetry playback; determines spacecraft weight and center of gravity; and executes digital commanding of spacecraft systems.

The MOCR positions of this group include the Command and Service Module Electrical, Environmental, and Communications Engineer (CSM EECOM), the CSM Guidance, Navigation, and Control Engineer (CSM GNC), the Lunar Module Electrical, Environmental, and Communications Engineer (LM EECOM), and the LM Guidance, Navigation, and Control Engineer (LM GNC). These positions are backed up with detailed support from the vehicle systems SSR.

Typical parameters monitored by this group include fuel cell skin and condenser temperatures, fuel cell current, various battery and bus voltages, launch escape tower and motor discretes, AGCU drift, SPS helium tank pressure, SPS fuel and oxidizer tank pressure, and fuel and oxidizer inlet pressure differential.

PARAMETERS MONITORED BY LIFE SYSTEMS GROUP

The life systems group is responsible for the well being of the flight crew. The group is headed by the Flight Surgeon in the MOCR. Aeromedical and environmental control specialists in the life systems SSR provide detailed support to the Flight Surgeon. The group monitors the flight crew health status and environmental/biomedical parameters.

MANNED SPACE FLIGHT NETWORK

The Manned Space Flight Network (MSFN) is a global network of ground stations, ships, and aircraft designed to support manned and unmanned space flight. The network provides tracking, telemetry, voice and teletype communications, command, recording, and television capabilities. The network is specifically configured to meet the requirements of each mission.

MSFN stations are categorized as lunar support stations (deep-space tracking in excess of 15,000 miles), near-space support stations with Unified S-Band (USB) equipment, and near-space support stations without USB equipment. Figure 9-9 is a matrix listing the stations, designators, and capabilities. Figure 9-10 shows the geographical location of each station.

MSFN stations include facilities operated by NASA, the United States Department of Defense (DOD), and the Australian Department of Supply (DOS).

The DOD facilities include the Eastern Test Range (ETR), Western Test Range (WTR), White Sands Missile Range (WSMR), Range Instrumentation Ships (RIS), and Apollo Range Instrumentation Aircraft (ARIA). Recovery forces under DOD are not considered to be part of the MSFN.

NASA COMMUNICATION NETWORK

The NASA Communications (NASCOM) network is a point-to-point communications systems connecting the MSFN stations to the MCC. NASCOM is managed by the Goddard Space Flight Center, where the primary communications switching center is located. Three smaller NASCOM switching centers are located at London, Honolulu, and Canberra. Patrick AFB, Florida and Wheeler AFB, Hawaii serve as switching centers for the DOD eastern and western test ranges, respectively. The MSFN stations throughout the world are interconnected by landline, undersea cable, radio and communications satellite circuits. These circuits carry teletype, voice, and data in real-time support of the missions. Figure 9-11 depicts a typical NASCOM configuration.

Each MSFN USB land station has a minimum of five voice/data circuits and two teletype circuits. The Apollo insertion and injection ships have a similar capability through the communications satellites.

The Apollo Launch Data System (ALDS) between KSC and MSC is controlled by MSC and is not routed though GSFC. The ALDS consists of wide-band telemetry, voice coordination circuits, and a high speed circuit for the Countdown and Status Transmission System (CASTS). In addition, other circuits are provided for launch coordination, tracking data, simulations, public information, television, and recovery.

MSFC SUPPORT OF LAUNCH AND FLIGHT OPERATIONS

The Marshall Space Flight Center (MSFC) by means of the Launch Information Exchange Facility

(LIEF) and the Huntsville Operations Support (HOSC) provides real-time support of launch vehicle prelaunch, launch, and flight operations. MSFC also provides support via LIEF for post-flight data delivery and evaluation.

LAUNCH INFORMATION EXCHANGE FACILITY

The LIEF encompasses those personnel, communications, data processing, display, and related facilities used by the MSFC launch vehicle design team to support Apollo-Saturn mission operations. The LIEF operations support organization is shown in figure 9-12.

In-depth real-time support is provided for prelaunch, launch, and flight operations from HOSC consoles manned by engineers who perform detailed system data monitoring and analysis.

System support engineers from MSFC and stage contractors are organized into preselected subsystem problem groups (approximately 160 engineers in 55 groups) to support KSC and MSC in launch vehicle areas which may be the subject of a request for analysis. The capabilities of MSFC laboratories and the System Development Facility (SDF) are also available.

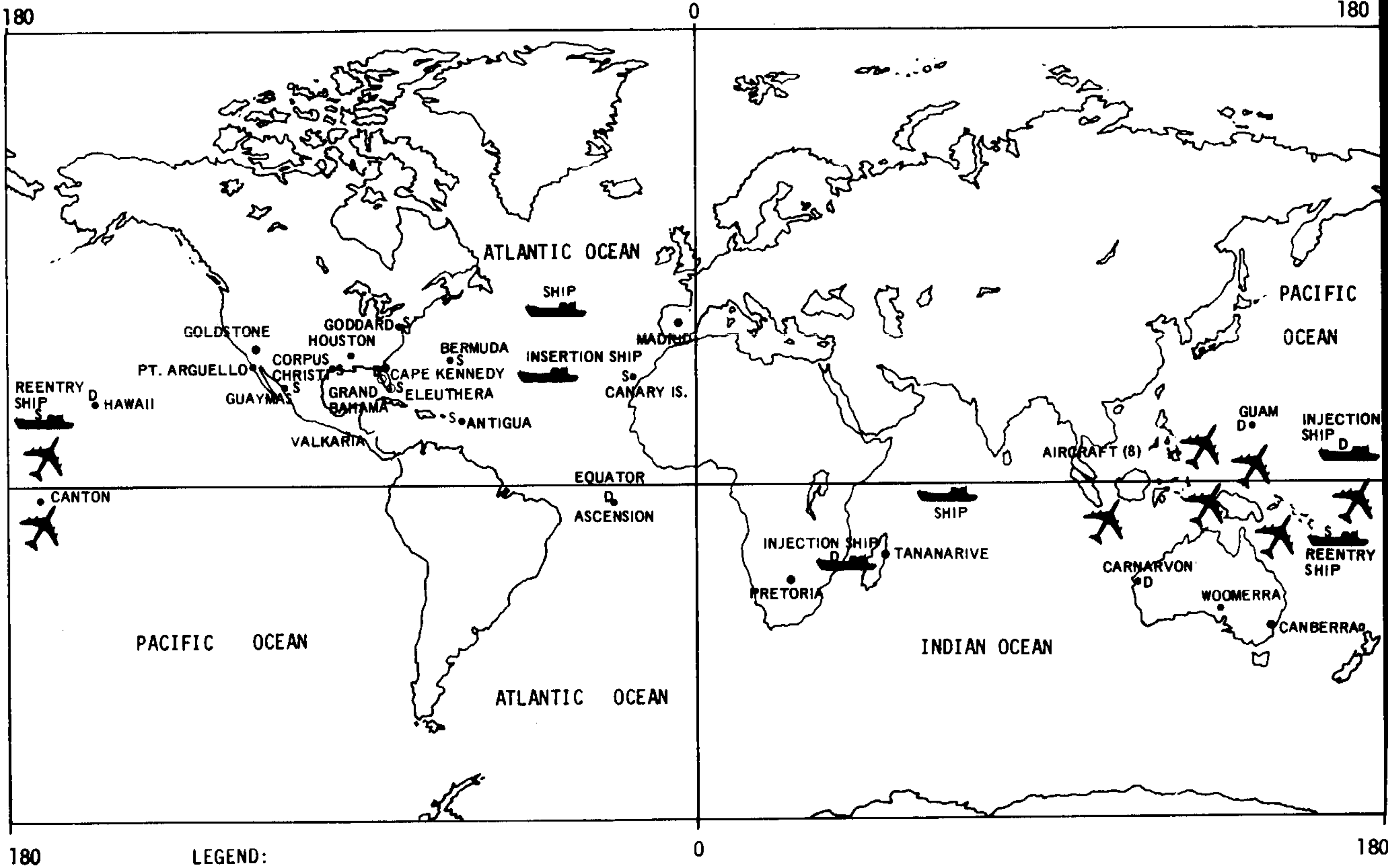
MSFN STATIONS/EQUIPMENT MATRIX

	Station \ Systems	USB - 85' Antenna	USB - 30' Ant (Single)	USB - 30' Ant (Dual)	TLM - Ant (AGAVE)	TLM - Ant (Teltrac)	TLM - Ant (TLM-18)	TLM - Ant (TAA-3A)	C-band Radar	C-band Hi-speed Radar Data	FC A/G Comm.	A/G Comm. (Remote)	ALDS	Hi-speed TLM Data	RF Command	RSDP SUM	Gemini Consoles & Displays	Apollo Consoles & Displays	Tone Command	Dig. Comm. System	Downrange Up-link	TLM R&R	TLM D/T	USB - 12' Ant (Single)	FC Manned	Saturn Booster TLM	Voice (SRO)	Voice (SCAMA)	TTY	Apollo Data Processors	Apollo TV Monitor	TLM Flat Broadside Ant.	S-band Tel R&R	RSO Displays
Mission Control Center	MCC									X	X	X				X	X	X		X			X		X				X					
MSFN Operations Center	MSFNOC											X	X	X		X				X							X	X	X	X	X			
Central Instrumentation Facility	CIF												X																					
	TEL IV							X														X	X			X							X	
Ascension Island, U.K. (NASA)	ACN #			X								X										X	X					X	X	X	X		X	
Antigua, U.K. (NASA)	ANG #		X									X		X								X	X			X		X	X	X	X		X	
Antigua (DOD-ETR)	ANT #					X	X	X	X	X		X		X	X				X		X	X	X		X	X	X		X				X	
Ascension Island (DOD-ETR)	ASC #					X	X		X										X		X	X	X				X		X				X	
Bermuda, U.K. (NASA)	BDA #		X			X			X	X		X		X	X				X		X	X	X					X	X	X	X		X	X
Canberra, Australia (WRE)	CNB *	X																										X	X	X	X		X	
Vandenberg, California (DOD-WTR)	CAL #				X				X	X		X										X	X					X	X					
Cape Kennedy (DOD-ETR)	CNV #						X		X						X				X		X	X	X				X		X					
Carnarvon, Australia (WRE)	CRO #			X		X			X	X	X				X	X	X	X		X		X	X		X			X	X	X	X		X	
USNS Coastal Sentry (DOD-ETR)	CBQ #				X						X	X			X	X	X		X	X		X	X											
Canton Island (NASA)	CTN #				X							X										X	X					X	X					
Canary Islands, Spain (NASA)	CYI #		X		X	X			X		X	X			X	X	X			X		X	X					X	X	X	X		X	
Grand Bahama Island (DOD-ETR)	GBI #					X			X	X		X		X	X						X	X	X			X	X		X				X	
Grand Bahamas, U.K. (NASA)	GBM #		X									X		X								X	X			X		X	X	X			X	
Goldstone, California (NASA)	GDS *	X																		X						X		X		X	X		X	
Grand Turk Island (DOD-ETR)	GTK #					X			X	X									X			X					X		X					
Guam, U.S. (NASA)	GWM #			X		X						X								X								X		X	X		X	
Guaymas, Mexico (NASA)	GYM #		X		X	X					X	X					X	X		X		X	X		X			X	X	X	X		X	
Kokee Park, Hawaii (NASA)	HAW #			X.	X	X			X		X	X			X	X	X		X	X		X	X					X	X	X	X		X	
Madrid, Spain (NASA)	MAD *	X																		X								X		X	X		X	
Merritt Island, Florida (NASA)	MIL #			X		X						X						X		X		X	X			X		X	X	X			X	
Merritt Island (DOD-ETR)	MLA #								X	X																	X		X		X			
Patrick AFB, Florida (DOD-ETR)	PAT #								X	X																	X		X					
Pretoria, So. Africa (DOD-ETR)	PRE #								X			X															X		X					
USNS Rose Knot (DOD-ETR)	RKV #				X						X	X			X	X	X		X	X		X	X			X								
USNS Range Tracker (DOD-WTR)	RTK #				X				X			X										X	X					X	X					
Tananarive, Malagasy Rep. (NASA)	TAN #				X							X										X	X					X	X					
Corpus Christi, Texas (NASA)	TEX #		X		X	X					X	X		X	X	X	X			X		X	X					X	X	X	X		X	
White Sands, N.M. (DOD-WSMR)	WHS #				X				X	X																		X	X					
Woomera, Australia (WRE)	WOM #				X				X																				X					
Apollo Range Inst. Aircraft	ARIA 1-8											X										X	X				X							
USNS Huntsville (DOD - WTR)	HTV #								X													X	X	X					X			X	X	
USNS Mercury (DOD - WTR)	MER #		X						X			X			X			X		X		X	X		X				X		X	X	X	
USNS Redstone (DOD - WTR)	RED #		X						X			X			X			X		X		X	X		X				X		X	X	X	
USNS Vanguard (DOD - WTR)	VAN #		X						X			X			X			X		X		X	X		X				X		X	X	X	
USNS Watertown (DOD - WTR)	WTN #								X													X	X	X					X			X	X	

* Lunar Support Stations
# Near Space Support Stations
Supplemental Stations

Figure 9-9

MANNED SPACE FLIGHT NETWORK APOLLO STATIONS

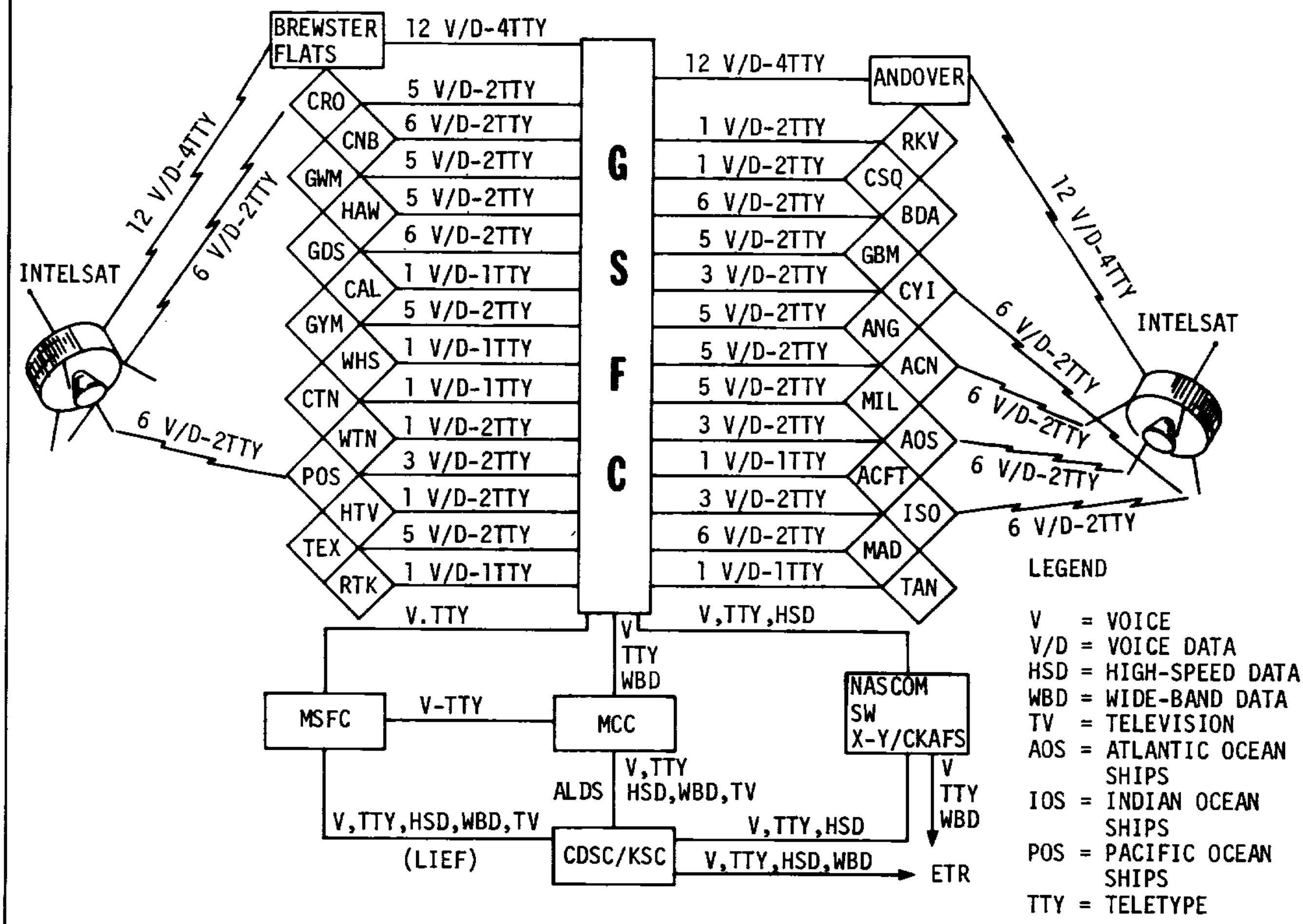


LEGEND:

- ○ Communications Only
- D• 30' Dual USBS
- S• 30' Single USBS
- ● 85' Antenna

Figure 9-10

TYPICAL APOLLO COMMUNICATIONS NETWORK CONFIGURATION



NOTES:

1. NASCOM interface points for ships and aircraft are mission dependent.
2. Additional V/D and TTY circuits between GSFC and centers are augmented as required.
3. Communications circuits as shown exist at this time or will be available for missions as required.
4. Circuit routing is not necessarily as indicated.
5. Voice Data Network to and from the Apollo USB stations will be configured with usage as indicated.
   a. Net #1 Mission Conference
   b. Net #2 Biomed Data
   c. Net #3 Network M & O
   d. Net #4 Telemetry Data/Command Data (USB)
   e. Net #5 Tracking Data (USB)
6. One additional voice data circuit will be available to each Apollo station equipped with an 85' antenna and to Bermuda for high-speed tracking data.
7. Apollo stations and ships connected to GSFC via Intelsat can receive at all times, regardless of number of stations or ships. Only two stations or ships can transmit at a given time via each Intelsat.
8. Apollo ship may be assigned to any of these areas.

Figure 9-11

LIEF OPERATIONS SUPPORT ORGANIZATION

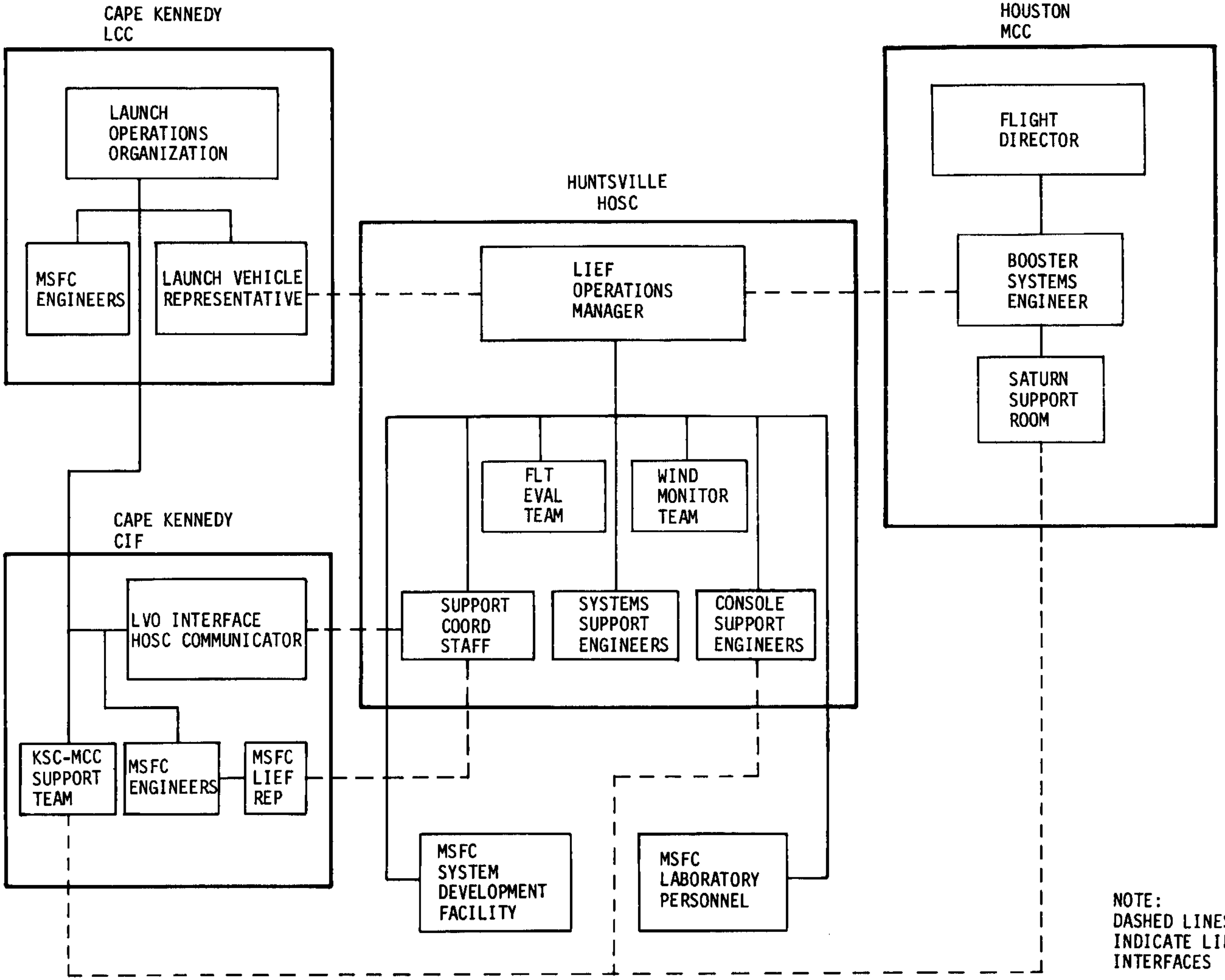


Figure 9-12

PRELAUNCH WIND MONITORING

Prelaunch flight wind monitoring analyses and trajectory simulations are jointly performed by MSFC and MSC personnel located at MSFC during the terminal countdown. Beginning at T-24 hours, actual wind data is transmitted periodically from KSC to the HOSC. These data are used by the MSFC/MSC wind monitoring team in vehicle flight digital simulations to verify the capability of the vehicle with these winds. Angle of attack, engine deflections, and structural loads are calculated and compared against vehicle limits. Simulations are made on either the IBM 7094 or B5500 computer and results are reported to the Launch Control Center (LCC) within 60 minutes after wind data transmission. At T-2 1/4 hours a go no-go recommendation is transmitted to KSC by the wind monitoring team. A go no-go condition is also relayed to the LCC for lox and LH_2 loading.

In the event of marginal wind conditions, contingency wind data balloon releases are made by KSC on an hourly basis after T-2 1/2 hours and a go no-go recommendation is transmitted to KSC for each contingency release. This contingency data is provided MSFC in real-time via CIF/DATA-CORE and trajectory simulations are preformed on-line to expedite reporting to KSC.

Ground wind monitoring activities are also performed by MSFC laboratory personnel for developmental tests of displays. Wind anemometer and strain gauge data are received in real-time and bending moments are computed and compared with similar bending moment displays in CIF.

LAUNCH AND FLIGHT OPERATIONS SUPPORT

During the prelaunch period, primary support is directed to KSC. Voice communications are also maintained with the Booster Systems Group at MCC and the KSC/MCC support engineers in CIF to coordinate preparations for the flight phase and answer any support request.

At liftoff primary support transfers from KSC to the MCC. The HOSC engineering consoles provide support as required to the Booster Systems Group for S-IVB/IU orbital operations by monitoring detailed instrumentation for the evaluation of system inflight and dynamic trends, assisting in the detection and isolation of vehicle malfunctions, and providing advisory contact with vehicle design specialist. This support is normally provided from liftoff through the active launch vehicle post-spacecraft separation phase approximately T + 6 hours, or until LIEF mission support termination.

SECTION X
MISSION VARIABLES AND CONSTRAINTS

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LAUNCH VEHICLE MISSION PROFILE 10-1
LAUNCH VEHICLE CONSTRAINTS 10-1

INTRODUCTION

This section describes the major objective, profile and constraints defined for Mission E (Lunar Development Mission).

LAUNCH VEHICLE MISSION OBJECTIVE

Demonstrate the capability of the launch vehicle to perform the lunar mission by simulation of lunar mission operations in near earth space.

LAUNCH VEHICLE MISSION PROFILE

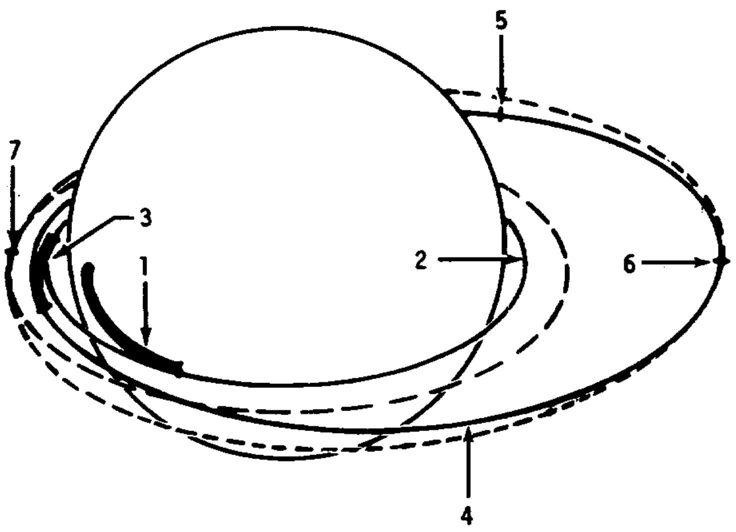
The flight profile into the high-apogee elliptical waiting orbit is shown in figure 10-1 and described in detail as to performance characteristics in Section II. A typical ground track through the fourth elliptical orbit is shown in figure 10-2.

LAUNCH VEHICLE CONSTRAINTS

The launch vehicle system requirements impose the following mission constraints:

1. Guidance command angle rate shall not exceed one degree per second in pitch, yaw, and roll (first-stage tilt program and upper-stage guidance program).
2. Maximum command attitude in the yaw plane shall not exceed 45 degrees.

MISSION E LAUNCH VEHICLE REFERENCE TRAJECTORY PROFILE



1. S-IC, S-II, AND S-IVB STAGE BOOST TO EARTH PARKING ORBIT.
2. COAST IN CIRCULAR PARKING ORBIT.
3. S-IVB STAGE BOOST FROM PARKING ORBIT TO HIGH-APOGEE ELLIPSE.
4. COAST IN HIGH-APOGEE ELLIPSE.
5. TRANSPOSITION, DOCKING, & L/V - S/C SEPARATION IS COMPLETED.
6. S/C BOOST RAISES PERIGEE ALTITUDE OF S/C ELLIPSE. S-IVB STAGE CONTINUES IN ORIGINAL ELLIPSE UNTIL ORBIT DECAY CAUSES REENTRY.
7. AFTER A MAXIMUM OF FOUR REVOLUTIONS IN HIGH-APOGEE ELLIPSE, S/C MANEUVERS INTO CIRCULAR ORBIT AND CONTINUES LLM MANEUVERS.

Figure 10-1

MISSION E LAUNCH VEHICLE REFERENCE TRAJECTORY GROUND TRACE

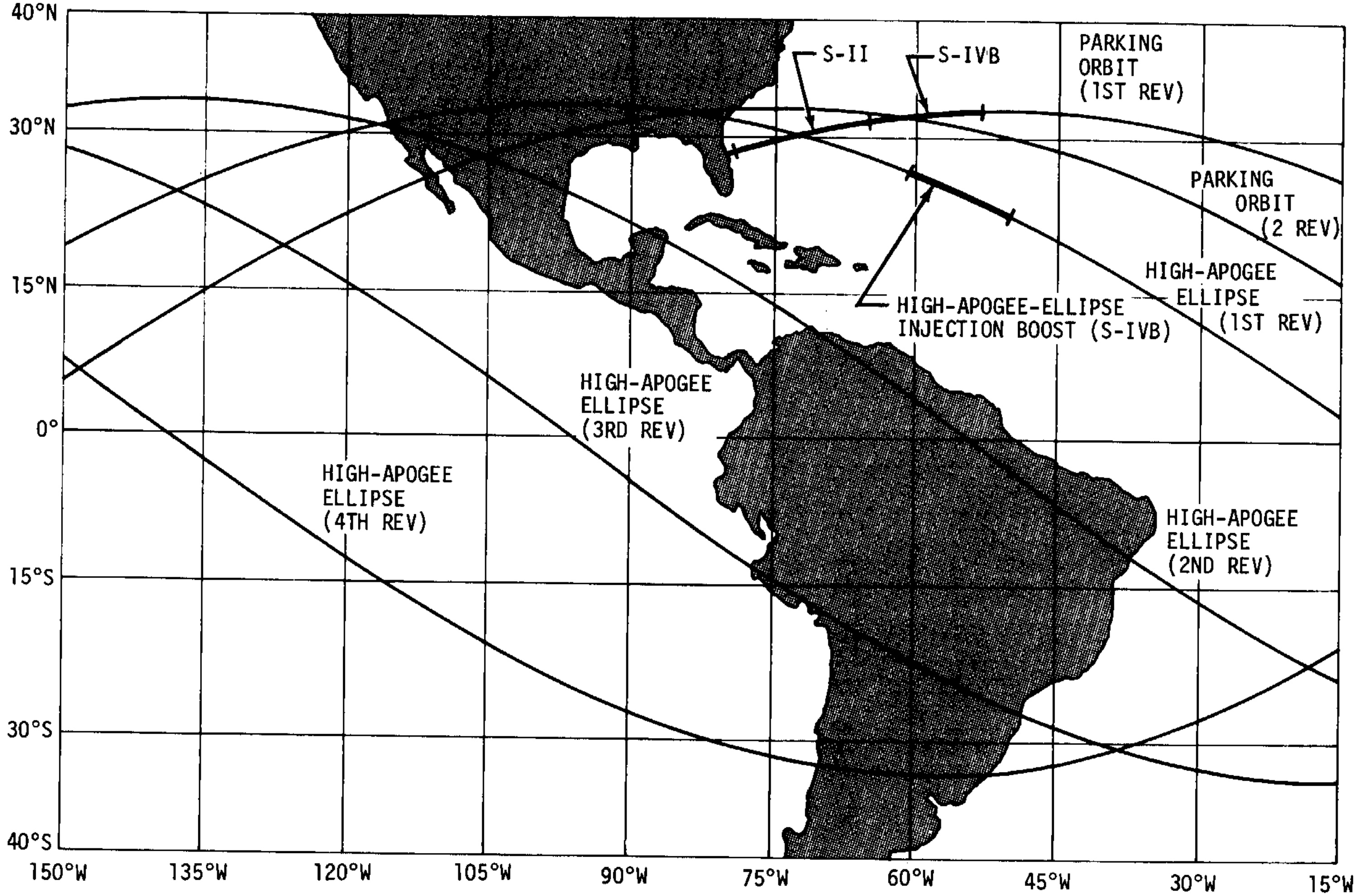


Figure 10-2 (Sheet 1 of 3)

MISSION E LAUNCH VEHICLE REFERENCE TRAJECTORY GROUND TRACE

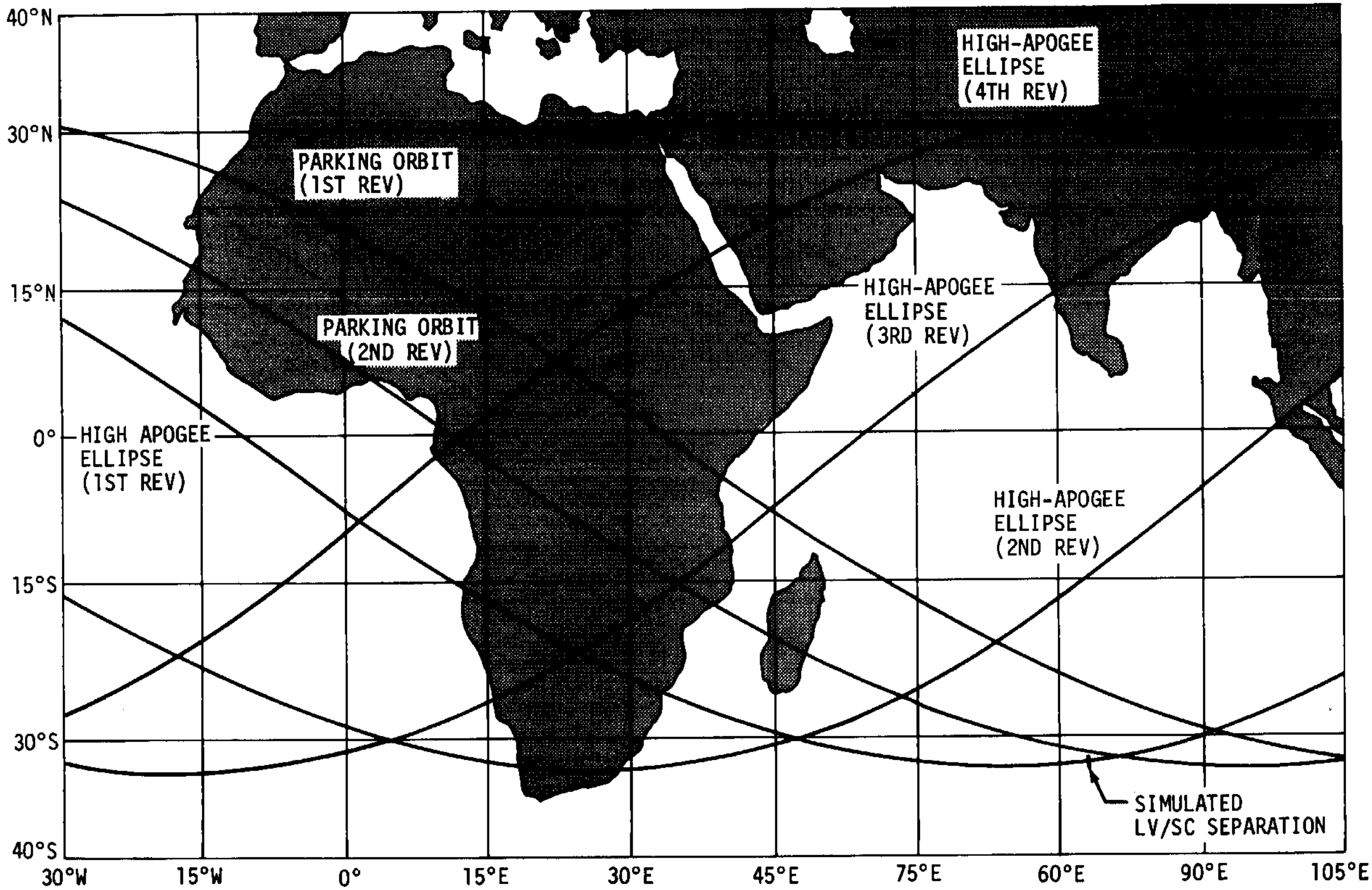


Figure 10-2 (Sheet 2 of 3)

MISSION E LAUNCH VEHICLE REFERENCE TRAJECTORY GROUND TRACE

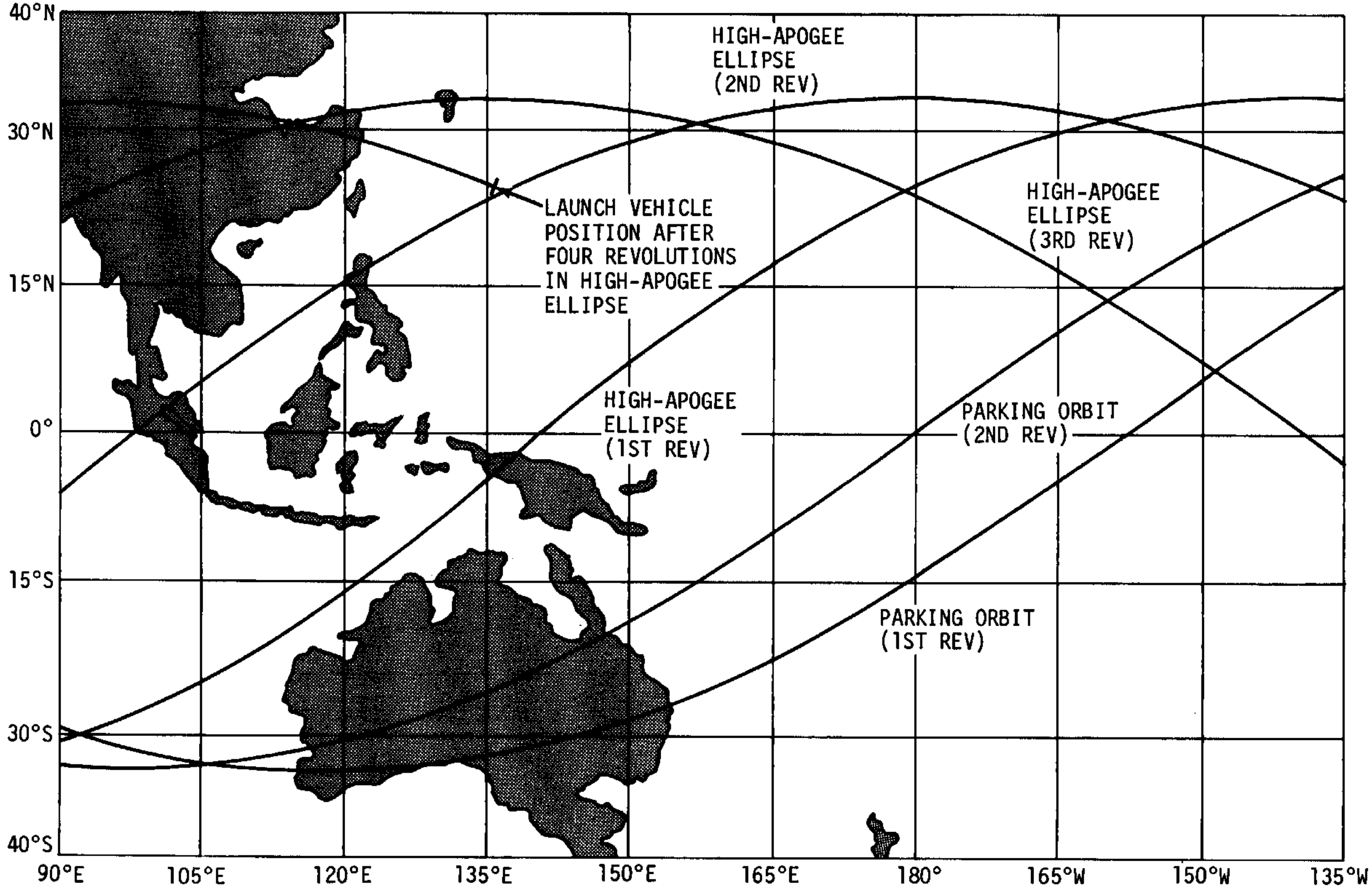


Figure 10-2 (Sheet 3 of 3)

APPENDIX A

ABBREVIATIONS AND ACRONYMS

A

A.........Astronaut
ac........Alternating Current
ACCEL.....Acceleration
ACE.......Acceptance Checkout Equipment
ACM.......Actuation Control Module
ACN.......NASA MSFN Station, Ascension Island
ACS.......Attitude Control and Stabilization Systems Engineer (Booster Systems)
ACT'R.....Actuator
AF........Air Force
AFB.......Air Force Base
AFD.......Assistant Flight Director
AFETR.....Air Force Eastern Test Range
AGAVE.....Automatic Gimbaled Antenna Vectoring Equipment
AGC.......Apollo Guidance Computer
A/G COMM..Air to Ground and Ground to Air Communications
AGCU......Apollo Guidance Control Unit
ALDS......Apollo Launch Data System
Alt.......Altitude
AM........Amplitude Modulation
amp.......Ampere
amp.......Amplifier
AMR.......Atlantic Missile Range
ANG.......NASA MSFN Station, Antigua
Ant.......Antenna
ANT.......DOD-ETR MSFN Station, Antigua
AOA.......Angle of Attack
AOS.......Acquisition of Spacecraft (by a site)
AOS.......Atlantic Ocean Ships
APS.......Auxiliary Propulsion System
ARIA......Apollo Range Instrumentation Aircraft
ASC.......DOD-ETR MSFN Station, Ascension Island
ASD.......Abort Summary Document
ASI.......Augmented Spark Igniter
ATT.......Attitude
AUTO......Automatic

B

BDA.......NASA MSFN Station, Bermuda
BIOMED....Biomedical
BMAG......Body Mounted Attitude Gyro
BEF.......Blunt End Forward
BSE.......Booster Systems Engineer

C

CAL.......DOD-WTR MSFN Station, Vandenberg AFB, California
calips....Calibrationable Pressure Switch
CAPCOM....Spacecraft Communicator
CASTS.....Countdown and Status Transmission System
CBQ.......DOD-ETR MSFN Station, USNS Coastal Sentry
CCATS.....Command, Communications and Telemetry System
CCF.......Converter/Compressor Facility
CCS.......Command Communications System
CCW.......Counterclockwise
CDDT......Countdown Demonstration Test
CDF.......Confined Detonating Fuse
CDP.......Command Data Processor, MSFN Site
CDR.......Critical Design Review
CDR.......Spacecraft Commander
CECO......Center Engine Cutoff
CG........Center of Gravity
CIF.......Central Instrumentation Facility (Located at Kennedy Space Center)
CIU.......Computer Interface Unit
CKAFS.....Cape Kennedy Air Force Stations
CM........Command Module
CMC.......Command Module Computer
CMD.......Command
CNB.......WRE MSFN Deep-Space Station, Canberra
CNV.......DOD-ETR MSFN Station, Cape Kennedy
CO........Checkout
C/O.......Checkout
COFW......Certificate of Flight Worthiness
COM.......Common
cps.......Cycles per Second (Hertz)
CRO.......WRE MSFN Station, Carnarvon
CRT.......Cathode Ray Tube
CSM.......Command and Service Module
C-T.......Crawler-Transporter
CTN.......NASA MSFN Station, Canton Island
CW........Clockwise
CW........Continuous Wave
CYI.......NASA MSFN Station, Canary Islands
CZR.......High Speed Metric Camera

D

DATA-CORE.CIF Telemetry Conversion System

A

db.......Decibel
dbm.......Decibels Referenced to One Milliwatt
dc........Direct Current
DCR.......Destruct Command Receiver
DCR.......Design Certification Review
DCS.......Digital Computer System
DDAS......Digital Data Acquisition System
DEE.......Digital Event Evaluator
deg.......Degree
Dig. Comm.Digital Command
DOD.......Department of Defense
DOS.......Department of Supply (Australia)
DRK.......Display Request Keyboard
DSKY......Display and Keyboard (Spacecraft Guidance and Control)
DTS.......Data Transmission System

E

EAO.......Experiment Activities Officer
EBW.......Exploding Bridgewire
ECA.......Electrical Control Assembly
ECS.......Environmental Control System
ECU.......Environmental Control Unit
EDS.......Emergency Detection System
EECOM.....Electrical, Environmental, and Communications Engineer
ELS.......Earth Landing System
EMS.......Entry Monitor System
ENG.......Engine
ENS.......Electrical Network and Systems Engineer (Booster Systems)
ESE.......Electrical Support Equipment
ETR.......Eastern Test Range

F

F.........Fahrenheit
F.........Force
FACI......First Article Configuration Inspection
FAO.......Flight Activities Officer
FC........Flight Controller
FCO.......Flight Control Office
FCSM......Flight Combustion Stability Monitor
FD........Flight Director
FDAI......Flight Director Attitude Indicator
FDK.......Forced Display Module
FDO.......Flight Dynamics Officer
FET.......Flight Evaluation Team
FEWG......Flight Evaluation Working Group
FLSC......Flexible Linear Shaped Charge
FLTR......Filter
FM........Frequency Modulation
FRT.......Flight Readiness Test
FT........Foot or Feet
FTS.......Flight Termination System

G

G.........Acceleration of Gravity
GAL.......Gallon(s)
GBI.......DOD-ETR MSFN Station, Grand Bahama Island
GBM.......NASA MSFN Station, Grand Bahama Island
GCC.......Ground Control Computer
GDC.......Gyro Display Coupler
GDS.......NASA MSFN Deep Space Station, Goldstone, California
GET.......Ground Elapsed Time
GG........Gas Generator
GH_2.......Gaseous Hydrogen
GHz.......Gigihertz (One Billion Hertz)
GLOTRAC...Global Tracking System
GN_2.......Gaseous Nitrogen
G&N.......Guidance and Navigation
GNC.......Guidance, Navigation, and Control Engineer
GND.......Guidance and Navigation Systems Engineer (Booster Systems)
GOX.......Gaseous Oxygen
gpm.......Gallons per Minute
GRR.......Guidance Reference Release
GSE.......Ground Support Equipment
GSE-ECU...Ground Support Equipment-Environmental Control Unit
GSFC......Goddard Space Flight Center
GTK.......DOD-ETR MSFN Station, Grand Turk Island
GUID......Guidance
GUIDO.....Guidance Officer
G-V.......Gravity versus Velocity
GWM.......NASA MSFN Station, Guam

H

H_2........Hydrogen
HAW.......NASA MSFN Station, Hawaii
HD........Highly Desireable Mission Rule Item
He........Helium
HF........High Frequency (3-30 MHz)
HOSC......Huntsville Operations Support Center
HSD.......High-Speed Data
HTV.......DOD-WTR MSFN Station, USNS Huntsville
Hz........Hertz (one cycle per second)

I

IA........Input Axis
IBM.......International Business Machines Corporation
ICD.......Interface Control Document
ICO.......Inboard Cutoff
IECO......Inboard Engine Cutoff
IF........Intermediate Frequency
IFV.......Igniter Fuel Valve
IGM.......Iterative Guidance Mode
IMP.......Impulse
IMU.......Inertial Measurement Unit
IMV.......Ignition Monitor Valve
in........Inch
INTELSAT..Communications Satellite
IO........Industrial Operations

IOS.......Indian Ocean Ships
IP........Impact Predictor (Located at Kennedy Space Center)
IRIG......Inter-Range Instrumentation Group
IU........Instrument Unit

J-K

kbps......Kilobits per Second
KHz.......Kilohertz (One Thousand Hertz)
KM........Kilometer
KOH.......Potassium Hydroxide
KSC.......Kennedy Space Center

L

lb(s).....Pound(s)
LCC.......Launch Control Center
LCR.......LIEF Control Room
LE........Launch Escape
LEM.......Lunar Excursion Module
LES.......Launch Escape System
LET.......Launch Escape Tower
LOS.......Loss of Spacecraft (by a site)
LH_2.......Liquid Hydrogen
LIEF......Launch Information Exchange Facility
LM........Lunar Module
LO........Liftoff
L/O.......Liftoff
lox.......Liquid Oxygen
LSC.......Linear Shaped Charge
LSD.......Low-Speed Data
LV........Launch Vehicle
L/V.......Launch Vehicle
LVDA......Launch Vehicle Data Adapter
LVDC......Launch Vehicle Digital Computer

M

M.........Mandatory Mission Rule Item
M.........Mass
MA........Apollo Program Office (Symbol)
MAD.......NASA MSFN Deep Space Station, Madrid, Spain
MAP.......Message Acceptance Pulse
MAX.......Maximum
MCC.......Mission Control Center
MCP.......Mission Control Programmer
MCR.......Main Conference Room
MD........Mission Director
MDC.......McDonnell-Douglas Corporation
MDF.......Mild Detonating Fuse
MED.......Medium
MER.......DOD-WTR MSFN Station, USNS Mercury
MESC......Master Event Sequence Controller
MFCO......Manual Fuel Cutoff
MHz.......Megahertz (One Million Hertz)
MIL.......NASA MSFN Station, Merritt Island, Florida
MILA......Merritt Island Launch Area
MIN.......Minimum
Min.......Minute
ML........Mobile Launcher
MLA.......DOD-ETR MSFN Station, Merritt Island, Florida
MMH.......Monomethyl Hydrazine
M&O.......Maintenance and Operations
MOC.......Mission Operations Computer
MOCR......Mission Operations Control Room
MOD.......Model
MOD.......Modification
MR........Mixture Ratio
MSC.......Manned Spacecraft Center
m/sec.....Millisecond (1/1000 Second)
MSFC......Marshall Space Flight Center
MSFN......Manned Space Flight Network
MSFNOC....Manned Space Flight Network Operations Center
MSK.......Manual Select Keyboard
MSR.......Mission Support Room
MSS.......Mobile Service Structure
MTF.......Mississippi Test Facility
MTVC......Manual Thrust Vector Control
MUX.......Multiplexer

N

N/A.......Not Applicable
NASA......National Aeronautics and Space Administration
NASCOM....NASA Communications Network
No........Number
NPSH......Net Positive Suction Head
NMI.......Nautical Mile

O

OA........Output Axis
OAT.......Overall Acceptance Test
OBECO.....Outboard Engine Cutoff
OCO.......Outboard Cutoff
ODOP......Offset Doppler
OECO......Outboard Engine Cutoff
O_2........Oxygen
O&P.......Operations and Procedures
OSC.......Oscillator
OSR.......Operations Support Room
OTV.......Operational Television System
OX........Oxidizer

P

P.........Pitch
PAM.......Pulse Amplitude Modulation
PAT.......DOD-ETR MSFN Station, Patrick AFB, Florida
PBI.......Push-Button Indicator
PC........Pitch Control
PCM.......Pulse Code Modulation
pct.......Percent
PDFRR.....Program Director's Flight Readiness Review
PDR.......Preliminary Design Review
PEA.......Platform Electronics Assembly
PETN......Pentaerythrite Tetranitrate
PMPFR.....Program Manager's Preflight Review
POS.......Pacific Ocean Ships
POS.......Position

A

POT.......Potentiometer
pps.......Pulses per Second
PRE.......DOD-ETR MSFN Station, Pretoria, So. Africa
PRF.......Pulse Repatition Frequency
PRN.......Psuedo-Random Noise
PRPLNT....Propellant
psi.......Pounds per Square Inch
psia......Pounds per Square Inch Absolute
psid......Pounds per Square Inch, Differential
psig......Pounds per Square Inch Gauge
PTCR......Pad Terminal Connection Room
PTCS......Propellant Tanking Computer System
PTL.......Prepare to Launch
PU........Propellant Utilization

Q

QLDS......Quick Look Data Station

R

R.........Roll
RACS......Remote Automatic Calibration System
RASM......Remote Analog Submultiplexer
RCR.......Recovery Control Room
RCS.......Reaction Control System
RDM.......Remote Digital Multiplexer
RDX.......Cyclotrimethylene-trinitramine
R&DO......Research and Development Operations
RDSM......Remote Digital Submultiplexer
RED.......DOD-WTR MSFN Station, USNS Redstone
RETRO.....Retrofire Officer
RF........Radio Frequency
RI........Radio Interference
RIS.......Range Instrumentation Ship
RKV.......DOD-ETR MSFN Station, USNS Rose Knot
RMS.......Root Mean Square
RNG.......Range
RP-1......Rocket Propellant
R&R.......Receive and Record
RSCR......Range Safety Command Receiver
RSDP......Remote Site Data Processor
RSO.......Range Safety Officer
R&QA......Reliability & Quality Assurance
RSS.......Root Sum Square
RTC.......Real Time Command
RTCC......Real Time Computer Complex
R/T CMD...Real-Time Command Controller
RTK.......DOD-WTR MSFN Station, USNS Range Tracker

S

S&A.......Safety and Arming
SACTO.....Sacramento Test Operations
SC........Spacecraft
S/C.......Spacecraft
SCAMA.....Switching, Conferencing, and Monitoring Arrangement
SCO.......Subcarrier Oscillator
SCS.......Stability Control System
SDF.......Systems Development Facility
sec.......Second
SECO......Single Engine Cutoff
SEP.......Separation
SIT.......Systems Interface Test
SLA.......Spacecraft-LM Adapter
SLV.......Saturn Launch Vehicle
SM........Service Module
SMEK......Summary Message Enable Keyboard
S/N.......Serial Number
SPS.......Service Propulsion System
SRA.......Spin Reference Axis
SRM.......Status Report Module
SRO.......Superintendent of Range Operations
s/s.......Samples per Second
SSB.......Single Sideband
SSR.......Staff Support Room
ST-124-M3.Saturn V Stable Platform
STDV......Start Tank Discharge Valve
SW........Switch(ing)
SYNC......Synchronize

T

TAN.......NASA MSFN Station, Tananarive, Malagasy
TBD.......To Be Determined
TCS.......Thermal Conditioning System
TEL IV....AFETR Telemetry Station at Cape Kennedy
Teltrac...Telemetry Tracking
TEX.......NASA MSFN Station, Corpus Christi, Texas
TLI.......Translunar Injection
TLM.......Telemetry
TM........Telemetry
TSM.......Tail Service Mast
TTY.......Teletype
TV........Television
TVC.......Thrust Vector Control
TWR JETT..Tower Jettison

U

UDL.......Up-Data-Link
UHF.......Ultra High Frequency (300-3000 MHz)
USB.......Unified S-Band
USBS......Unified S-Band Station
USNS......United States Navy Ship

V

V.........Velocity
V.........Voice
v.........Volts
v-a.......Volt-ampere
VAB.......Vehicle Assembly Building
VAN.......DOD-WTR MSFN Station, USNS Vanguard
V/D.......Voice and Data
vdc.......Volts, Direct Current
VHF.......Very High Frequency (30-300 MHz)
vswr......Voltage Standing Wave Ratio

W

WBD.......Wide Band Data
W/G.......Water/Glycol
WHS.......DOD MSFN Station, White Sands Missile Range, N. Mexico
WOM.......WRE MSFN Station, Woomera, Australia
WRE.......Weapons Research Establishment, Australian Department of Supply
WSMR......White Sands Missile Range
WTN.......DOD-WTR MSFN Station, USNS Watertown
WTR.......Western Test Range

X-Z

XLUNAR....Translunar
XTAL......Crystal
Y.........Yaw

GREEK SYMBOLS

β.........Thrust Vector Angular Deflection
Δp ....Differential Pressure
ΔV........Velocity Increment
$\Delta \rho$ ....Atmospheric Density Increment
μsec......Microsecond
ϕ.........Vehicle Attitude
$\dot{\phi}$.........Vehicle Attitude Angular Change Rate
ψ.........Attitude Error Command
χ.........Desired Attitude

A

APPENDIX B

BIBLIOGRAPHY

Apollo Design Certification Review; April 21, 1966; Apollo Program Directive No. 7.

Apollo Flight Mission Assignments (U), (Confidential); January 1968; SE 010-000-1.

Apollo Flight Readiness Reviews, Part I; November 8, 1965; Apollo Program Directive No. 8.

Apollo Mission Rules; April 12, 1967; NASA Management Instruction 8020.9.

Apollo/Saturn V Facility Description; October 1, 1966; K-V-012.

Apollo Test Requirements; March 1967; NHB 8080.1

Astrionics System Handbook, Saturn Launch Vehicle; August 15, 1966; Not numbered.

Automatic Terminal Countdown Sequence Interlocks SA-502; February 16, 1968; 40M50547

Azusa an Engineering Brochure Describing in General Terms; Not dated; Not numbered.

Battery Assembly; May 19, 1965; 40M30780.

Battery, S-IVB Aft Bus No. 2; Rev. J, February 13, 1967; 1A68317.

Battery, S-IVB Forward Bus No. 1 and Aft Bus No. 1; Rev. L; March 29, 1967, 1A59741.

Battery, S-IVB Forward Bus No. 2; Rev. K; February 3, 1967; 1A68316.

Battery Technical Manual; June 1, 1967; IBM 66-966-0024.

C-Band Radar Transponder Set SST-135C; March 1, 1967; IBM 66-966-0008.

Cable Interconnection Diagram, Instrument Unit, S-IU-503; June 27, 1966; 7909181.

Clearance Envelope, LEM/S-IVB/IU, Physical; Rev. B; Not dated; 13M50123.

Command Decoder Technical Manual; March 1, 1967; IBM 66-966-0019.

Components Clearance Envelope S-IVB to IU; May 27, 1966; 13M50117.

Composite Mechanical Schematic; Rev. F; November 1, 1967; 10M30531.

Design Layout-Main Display Console, Command Module 103; V36-976020B; (Redline Copy from MSFC, Received 11-13-67).

Dynamic Launch Decision Criteria Feasibility Study Report Saturn V; January 31, 1967; D5-15685.

EDS Checkout Schematics, SA-503, (Unreleased); August 21, 1967; 40M37543.

Electrical Interface, Instrument Unit/ESE, Definition of Saturn V Vehicle SA-504; January 24, 1967; 40M35564.

Electrical Interface, SA-504/SC-104, ICD Definition of Saturn; Rev. B; January 19, 1968; 40M37520.

Electrical Interface, S-II/S-IVB, Definition of Saturn V Vehicle; Rev. B; April 19, 1965; 40M30593.

Electrical Interface, S-IVB/Instrument Unit, Definition of Saturn V Vehicle; Rev.D; December 1, 1967; 40M30597.

Electromagnetic Compatibility, Specification, MSFC/NASA; June 1, 1964; MSFC-SPEC-279.

Emergency Detection and Procedures (Draft Copy) Received from MSFC November 7, 1967; Not numbered.

Emergency Detection and Procedures, Addendum, (Draft copy); Received from MSFC November 17, 1967; Not numbered.

Emergency Detection System Analysis, Saturn V Launch Vehicle, SA-503; Rev. B; April 19, 1968; D5-15555-3B.

Emergency Detection System Description, Apollo-Saturn IB; February 1, 1967; IBM 66-966-0010.

Emergency Detection System Description, Apollo-Saturn V; July 1, 1967; IBM 66-966-0022.

Emergency Detection System, Saturn SA-504 and Apollo SC-103, ICD Description of; Rev. A; September 15, 1966; 40M37558.

Engine Actuation, S-II, FISR; June 30, 1966; D5-15657.

Engine Compartment Conditioning Subsystem Report, Saturn S-II; November 15, 1963; SID 62-142

B

Engine-Out and Malfunction Propulsion System Performance Prediction, AS-504; January 22,1968; D5-15704-4.

Environmental Control System, Operational Description of IU and S-IVB; Rev. B; April 28, 1965; 20M97007.

Environmental Control System Saturn V Vehicle/ Launch Complex 39, FISR; June 30, 1966; D5-15662

F-1 Engine Allocation Chart, S-IC; Rev. F; June 13, 1967; 60B37303.

Firing Report, Saturn S-IVB-503N Stage Acceptance; June 1967; DAC-56358.

Flight Control Data Requirements, Saturn Launch Vehicle, AS-503; January 15, 1968; Not numbered.

Flight Control Data Requirements, Saturn Launch Vehicle, Standard; August 1, 1967; Not numbered.

Flight Control, Introduction to; June 16, 1967; SCD No. T.01.

Flight Control System Saturn S-II; Feburary 1, 1965; SID 62-140.

Flight Mission Rules, Apollo 6; December 12, 1967; FC021.

Flight Operations Plan, AS-501; July 8, 1966; Not numbered.

Flight Operations Planning and Preparation for Manned Orbital Missions; (John H. Boynton and Christopher C. Kraft, Jr.) AIAA Paper No. 66-904

Flight Performance Handbook for Powered Flight Operations; (Space Technology Laboratories, Inc.) Not dated; Not numbered.

Flight Sequence Program, Definition of Saturn SA-504; Rev. A; July 26, 1967; 40M33624.

Fluids Requirements, Saturn V Vehicle Instrument Unit, ICD; Rev. A; March 29, 1967; 13M50099.

Fluids Requirements, Saturn V Launch Vehicle, S-II Stage; Rev. C; March 9, 1967; 13M50097.

Fluids Requirements, Saturn ICD, S-IVB Stage; Rev. B; February 3, 1966; 13M50098.

Frequency Plan, Saturn Apollo, Instrumentation and Communications; Rev. B; May 9, 1966; 13M60003.

Fuel Tank Pressurization System, Advanced Functional Schematic; Rev. L; November 21, 1967; 1B62829.

Gaseous Helium System, Functional Interface Fluid Requirements; Rev. A; February 28, 1967; 65ICD9401.

Gaseous Nitrogen System, Functional Interface Fluid Requirements; Rev. A; February 28, 1967; 65ICD9402.

Ground Safety Plan, Apollo/Saturn IB; April 1, 1966; K-IB-023.

Ground Support Equipment Fact Booklet, Saturn V Launch Vehicle; Chg. August 25, 1967; MSFC-MAN-100.

Hazardous Gas Detection System; June 19, 1967; MSFC-MAN-234.

Inboard Profile, Saturn V; Rev. M; February 25, 1966; 10M03369.

Instrument Program and Components, S-IC-4; Rev. B; April 11, 1967; 60B57414.

Instrument Program and Components, S-II-4; Rev. C; November 8, 1967; V7-976404.

Instrument Program and Components, S-IVB-504 New (DAC S/N 1009); Rev. V; November 7, 1967; 1B43570.

Instrument Unit Assembly; Rev. E; August 18, 1967; 10Z22501-1.

Instrument Unit Environmental Control System Description, Saturn IB/V; March 15, 1966; IBM 66-966-0009.

Instrument Unit-Saturn V, Functional Interlock Requirement; Rev. A; May 16, 1966; 10M30554.

Instrument Unit System Description and Component Data (S-IU-201-212/501-515), Saturn IB/V; Chg. June 6, 1967; IBM 66-966-0006.

Instrument Unit to Spacecraft Physical Requirements; August 1, 1965; 13M50303.

Instrumentation and Communications Interfaces, Saturn Launch Vehicle SA-504 to Apollo Spacecraft CSM-103 and LEM-4, Apollo Inter-Center ICD; Rev. A; January 25, 1968; 13M63041.

Instrumentation Program and Components, S-IU-5-4; Rev. A; February 23, 1966; 6009048.

Instrumentation Subsystem Report; July 1, 1967; SID 62-136.

Instrumentation System Description; October 1, 1966; IBM 65-966-0021.

Instrumentation System Description-Launch Vehicle AS-503; September 1, 1966; D5-15338-3.

J-2 Rocket Engine Data Manual; Chg. 3; November 29, 1967; R-3825-1.

Launch Mission Rules Handbook, Apollo/Saturn; July 26, 1967; 630-23-0002.

Launch Mission Rules, Apollo 6 (AS-502); February 20, 1968; K-V-05.10/2.

Launch Operations Plan, Apollo/Saturn V AS-501/502; January 6, 1967; K-V-051/1/2.

Launch Vehicle Ground Support Plan, Issue 3 (SA-501/CSM-17); July 31, 1967; Report No. I-MO-4-67

Launch Vehicle to Spacecraft Procedural Requirements AS-503 and Subsequent, Apollo Inter-Center ICD; Rev. A; December 7, 1966; 13M03083.

LIEF Operations Plan, AS-501; August 23, 1967; Not numbered.

Lox Tank Pressurization System, Advanced Functional Schematic; Rev. C; August 1, 1966; 1B62828.

Major Structural Test Requirements, Saturn S-II; April 1965; SID 65-730.

Manned Spacecraft: Engineering Design & Operation (Purser, Faget & Smith); Not numbered.

Measuring Rack and Measuring Rack Selector (S-IU-204 thru 212/501 thru 515); October 1, 1966; IBM 66-966-0017.

Mission Program Plan, Saturn Instrument Unit; July 5, 1966; IBM 66-966-0023.

Mobile Launcher, Vehicle Holddown Arms and Service Platforms, Apollo Saturn V, Operations and Maintenance; September 15, 1967; TM 487-MD.

Model Specification for Saturn V S-II Stage; April 1, 1964; SID 61-361.

Navigation, Guidance, and Control System Description; November 1, 1966; IBM 66-966-0003.

Network Operations Document, All Flights, Goddard Space Flight Center; December 15, 1967; Not numbered.

Onboard EDS, Design Criteria, Saturn V; Rev. C; May 5, 1966; 13M65001.

Operational Data Acquisition System Design Requirements Drawing, Saturn S-IVB; Rev. F; May 23, 1966; 1B38225.

Ordnance Systems, Saturn V Launch Vehicle; April 1, 1968; SE 005-003-2H.

Overall Test (Plugs Out), Apollo/Saturn Launch Vehicle Operations for Support of Space Vehicle; Draft 1; May 11, 1967; V-20012-SA-502.

Performance Analysis, Saturn V AS-503 Launch Vehicle; September 30, 1966; D5-15519-3.

Performance/Design and Qualification Requirements, Part I, Contract End Item Detail Specification; August 4, 1965; CP621M0001A.

Prelaunch Checkout and Launch Operations Requirements, Saturn V Launch Vehicle; July 18, 1967; I-V-TR-501-10.

Prelaunch Sequence of Events, Saturn V; Rev. E; January 15, 1967; 10M30520.

Program Operations Plan, MSFC, Apollo/Saturn; January 31, 1966; Not numbered.

Program Orientation, Saturn S-IVB; August 1964; SM-46688.

Program Schedules, Status and Analysis, Saturn V; March 28, 1968; Issure 21, I-V-P.

Propulsion Systems Handbook (Preliminary); August 4, 1967; D5-15660.

Propulsion System Performance Prediction with Dispersions, AS-504; Volumes 1 and 2 of 3, Rev. B, October 12, 1967; Volume 3 of 3, June 30, 1967; D5-15506-4.

Range Safety Analysis, (Preliminary), Saturn V AS-503; November 10, 1966; D5-15523-3.

Range Safety Manual; September 10, 1965; AFETRM 127-1.

Recycle to Next Launch Window, Sequence of Events - Time Bars - Saturn V Vehicle - KSC; October 19, 1967; D5-16004-913.

Redlines Digest, Countdown Observer, As-502; March 5, 1968; I-V-RLD-502.

Reference Trajectory, AS-503 Launch Vehicle; August 29, 1966; D5-15480-1.

S-IC Flight Readiness Report; Not dated; Not numbered.

S-IVB Stage Electrical Subsystem, Fragmented Design Certification Review, SA-501; March 9; 1967; Not numbered.

S-IVB Stage Project Review, Saturn, (AS-204 thru AS-515); September 12, 1967; Not numbered.

S-IVB to Instrument Unit Functional Requirements, ICD; November 18, 1967; 13M07002.

S-IVB to Instrument Unit Physical Requirements; April 14, 1965; 13M50302.

S-IVB to Instrument Unit Procedural Requirements, Saturn ICD; April 12, 1966; 13M03002.

S-IVB-V Stage End Item Test Plan; Rev. B; April 13, 1967; 1B66684.

S-IVB-V-503 Stage End Item Test Plan; Rev. B; December 29, 1966; 1B66170.

Safety; September 10, 1965; AFETRR 127-9.

Saturn Data Summary Handbook; Douglas Logistics Support Department; October 1, 1965; Not numbered.

Saturn S-II General Manual; August 1, 1963; SID 63-1.

Secure Range Safety Command System for Saturn; November 9, 1964; NASA TM X-53162.

Sequence and Flow of Hardware Development and Key Inspection, Review, and Certification Checkpoints; August 30, 1966; Apollo Program Directive No. 6A.

Sequence of Events-Time Bar-Assembly, Checkout, and Launch, Saturn V SA-504 Vehicle - KSC; Rev. B; September 25, 1967; D5-16004-911.

Sequence of Events-Time Line Analysis-Assembly, Checkout, and Launch, Saturn V SA-504 Vehicle-KSC-Pad, Volume 3 of 3; Rev. B; September 25, 1967; D5-16004-910.

Simulated Flight Test, Apollo/Saturn Launch Vehicle Operations for Support of Space Vehicle; Rev. B; August 9, 1967; V-20013-SA-501.

Spacecraft (Block II) Description, Apollo Operations Handbook; March 1, 1967 (Revised May 28, 1968); SM2A-03-Block II-(1).

Specification Baseline, Saturn V Program (SA-504 through SA-515); September 23, 1966 (Revised September 15, 1967); RS02W-1000004A.

Stage Flight Measurements S-IC (504 and 505); May 15, 1967; MSFC-MAN-042-4-5.

Stage System Description S-IC-4; May 15, 1967; MSFC-MAN-040-4.

Start Times - F-1 Engine, S-IC; Rev E; August 2, 1967; 60B37301.

Stress Analysis of the Saturn S-II Stage Airframe (S-II-S, S-II-T, and S-II-F); January 1, 1966; SID 65-1676.

Stress Analysis of the Saturn S-II Stage Airframe (S-II-1, S-II-2, and S-II-3); February 1, 1967; SID 66-322.

Structure Subsystem Report S-II-1, Design Certification Review SA-501; March 10, 1967; SID 67-32.

Switch Selector, Model II Saturn IB/V Instrument Unit; Chg. March 1, 1967; IBM 66-966-0001.

Systems and Performance Briefing, MSFC, April 27-28, 1967; Not numbered.

Telemetry System Manual, Saturn V; Rev. B; October 31, 1966; 50M71535.

Test and Checkout Requirements Document, Launch Vehicle, SA-502 through SA-510; September 29, 1967; I-V-TR-502-10.

Test Specifications and Criteria, KSC Prelaunch Checkout and Launch Operations, S-IVB-V; Rev. A; August 18, 1966; 1B66262.

Thermal Control, Subsystem Report for Saturn S-II; April 15, 1965; SID 62-137.

Turnaround Plan, Apollo/Saturn V Space Vehicle; September 29, 1967; 630-39-0004.

Turnaround Time for Launch, Following a Decision to Recycle, Saturn V Launch Vehicle, Memorandum To: Mr. Bell, I-V-E; From: Chief, Vehicle Systems Group, R-TO-V; April 27, 1967; R-TO-V-67-66.

Weight and Balance Status Report, Saturn S-IVB Stage, Model No. DSV-4B; July 1967; DAC-56576.

Weight and Balance Status Report, Saturn S-IVB Stage; Monthly; DAC-56627.

5 Volt Measuring Voltage Supply, Tech Manual; October 1, 1965; IBM 65-966-0019H.

56 Volt Power Supply (S-IU-201-212/501-515); Chg. November 1, 1966; IBM 66-966-0018H.

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