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APOLLO SYSTEMS DESCRIPTION

VOLUME II

SATURN LAUNCH VEHICLES

MARSHALL SPACE FLIGHT CENTER

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SECTION I

GENERAL

1-1. DEFINITION AND SCOPE.

The Apollo system consists of the Apollo space vehicle, the flight crew, the earth-based support systems and the ground crews to be employed in manned lunar exploration missions. The Apollo space vehicle is made up of a Saturn V launch vehicle and the Apollo spacecraft. The Saturn V launch vehicle in turn consists of an S-IC first stage, an S-II second stage, an S-IVB third stage and an instrument unit. The Apollo system depends on the development of the Saturn I and Saturn IB vehicles.

This volume contains a description of the Saturn I, IB and V launch vehicles. The volume is divided into chapters, the contents of which are described below:

Chapter 1 describes the scope and coverage of this volume, and contains a history of the Apollo Project, an introduction to the Saturn-Apollo vehicle configuration, and the program plan.

Chapters 2, 3 and 4 contain respectively a description of the functional systems of the Saturn I, IB and V launch vehicles. Each chapter is divided into sections, one for each launch vehicle system.

Chapter 5 contains a description of the Saturn launch vehicle facilities. The chapter is divided into two sections; one contains a description of the facilities, the other, logistics.

1-2. METHOD OF COVERAGE.

This document is a condensed version of a complete description of the Saturn systems. The material is arranged so that an aerospace engineer can understand the functional operation of the many systems that make up the Saturn System.

Coverage of functions and systems is limited to those under the jurisdiction of the Marshall Space Flight Center except for any related areas that are necessary to understand the operation of a Saturn system.

The general mode of system description is to relate each system for a Saturn launch vehicle configuration to its basic flight mission for the reader to understand the requirements, operations, and interfaces. This "why" and "how" becomes the introduction to the hardware description.

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SECTION II
HISTORY OF SATURN PROGRAM

2-1. MANNED FLIGHT PROGRAM.

The exploration of space is the dominant mission in our space program. Within the framework of the broad national space-research program, manned flight is just coming of age. The need for participation of human pilots in the space-flight program was recognized from the outset and was provided for by the organization of the Space Task Group concurrently with the establishment of the civilian National Aeronautics and Space Administration.

The cumulative technology of Mercury, Gemini, Apollo and space-station operations will establish a sound base for manned interplanetary flight. The initial experience of manned spaceflight has been successfully obtained in the Mercury Project. This experience is important not only to flight and ground-operations crews but also in all phases of design engineering and management.

Gemini provides the first attempts at maneuvering in space in which the magnitude and direction of the velocity changes made will be computed during the flight in response to the situation created during the mission. Similarly, the capability is being developed to land at a predetermined point by guiding the spacecraft in reentry and descent attitudes. Gemini also allows longer flights and more complex experiments than were possible with the Mercury spacecraft. It is a major introductory step to manned lunar landing.

The manned segment of the lunar-landing program was named Project Apollo in July 1960. In the months since President Kennedy made the lunar landing timetable decision in May 1961 it has rapidly unfolded into a program which measures the total technical competence of the nation, through the engineering and scientific advances it requires and the industrial and management capabilities that must be marshalled to carry it out.

2-2. MARSHALL SPACE FLIGHT CENTER DEVELOPMENT.

The Saturn launch vehicles that are described in this volume stem from the studies of large boosters that were conducted at Huntsville in 1957 by the Army Ballistic Missile Agency (ABMA) the pioneering organization which later provided the nucleus for the present Marshall Space Flight Center.

The studies were begun after ABMA had concluded that the United States would need a launch vehicle larger than any then under development, if this country were to be able to engage effectively in space exploration projects. In February, 1958, the Advanced Research Projects Agency (ARPA), responsible for the nation's outer space program, was established by the Department of Defense. Discussions followed between ARPA and ABMA concerning the development of a suitable vehicle, and in August, 1958, ARPA issued Order No. 14-59 to the Army Ordnance Missile Command authorizing ABMA to develop a 1.5-million pound thrust, clustered-engine booster for the multi-stage vehicle program. This booster became the first in the series of launch vehicles for the Saturn-Apollo program.

In October and November of 1959 President Eisenhower announced his decisions to transfer part of ABMA's personnel, facilities and missions, and responsibility for the Saturn program, from Army monitorship to the National Aeronautics and Space Administration (NASA). The technical direction of Saturn was assumed by NASA in November, 1959, pending formal transfer of the program from the Army. The NASA Huntsville facility was named the George C. Marshall Space Flight Center in March, 1960, with formal transfer ceremonies at Redstone Arsenal. It was formally dedicated by President Eisenhower and Mrs. George C. Marshall in September of that year.

2-3. PLANNED DEVELOPMENT.

A large number of participating organizations throughout the United States are working toward the accomplishment of the Apollo objectives. These include not only various parts of NASA and the Department of Defense, but also many universities and industrial contractors.

The NASA organization is structured to integrate the many areas of effort. Major responsibilities which must be integrated into the whole include flight missions and their analyses, the design, development, and fabrication of launch vehicles, spacecraft, ground based mission support equipment, and launch facilities, and all other direct and indirect activities and equipment.

The Office of Manned Space Flight (OMSF) provides program management, planning and coordination of the effort. The Manned Spacecraft Center (MSC) at Houston is charged with spacecraft development and support of manned space flight missions. The Manned Spacecraft Center also provides a training center for the Apollo flight crews. The Launch Operation Center (LOC) is responsible for developing launch facilities and for conducting the launch of Apollo program space vehicles. The Marshall Space Flight Center (MSFC) is responsible for providing the launch vehicles needed for the Apollo program, together with associated support equipment.

The final objectives of the Apollo program will be achieved as the culmination of a logical and carefully planned development and flight test program. This development and test program is structured to develop the launch vehicle, spacecraft, ground equipment and techniques in "buildup" missions which progress in a reasonable and expeditious manner to the final Apollo lunar landing mission. First flights in the program have already been accomplished in the early Saturn launches and spacecraft tests.

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3-1. MISSIONS.

The mission of Apollo is threefold. First, there will be extended-duration earth-orbital flights; then circumlunar exploratory flights; and finally lunar landing and return. The manned lunar landing missions will be accomplished using the Saturn V launch vehicle and a lunar orbit-rendezvous mode.

The plans for the Saturn-Apollo missions are based on an orderly progression of accomplishments that culminate in man landing on the moon and his safe return to earth. The manned lunar landing, which is to be accomplished in this decade, is preceded by development flights that prove the space vehicles, and permit practice of flight techniques and the accumulation of operational experience.

Three Saturn configurations are being used in the Saturn-Apollo missions; the Saturn I, IB and V launch vehicles. The missions for Saturn I are development flights for the launch vehicle systems that are used for the larger Saturn IB and Saturn V boosters. Two of the ten Saturn I flights are scheduled to place micrometeroid satellities in eccentric earth orbits. The nominal capability of the Saturn I is to place a 22,500-pound payload into a 100-nautical mile circular earth orbit.

The Saturn IB missions will develop the launch vehicle and spacecraft systems and operations to the point where extended-duration earth-orbital flights are successful. Nominal payload capability is 32,500 lb. in a 105-nautical mile circular earth orbit.

The Saturn V missions will build up vehicle operation through Command Module (CM) ultravelocity re-entry flights and then circumlunar flights prior to the ultimate mission. Individual missions for each of the Saturn-Apollo vehicles are described in the introductory section of each chapter.

3-2. SATURN LAUNCH VEHICLE CONFIGURATION.

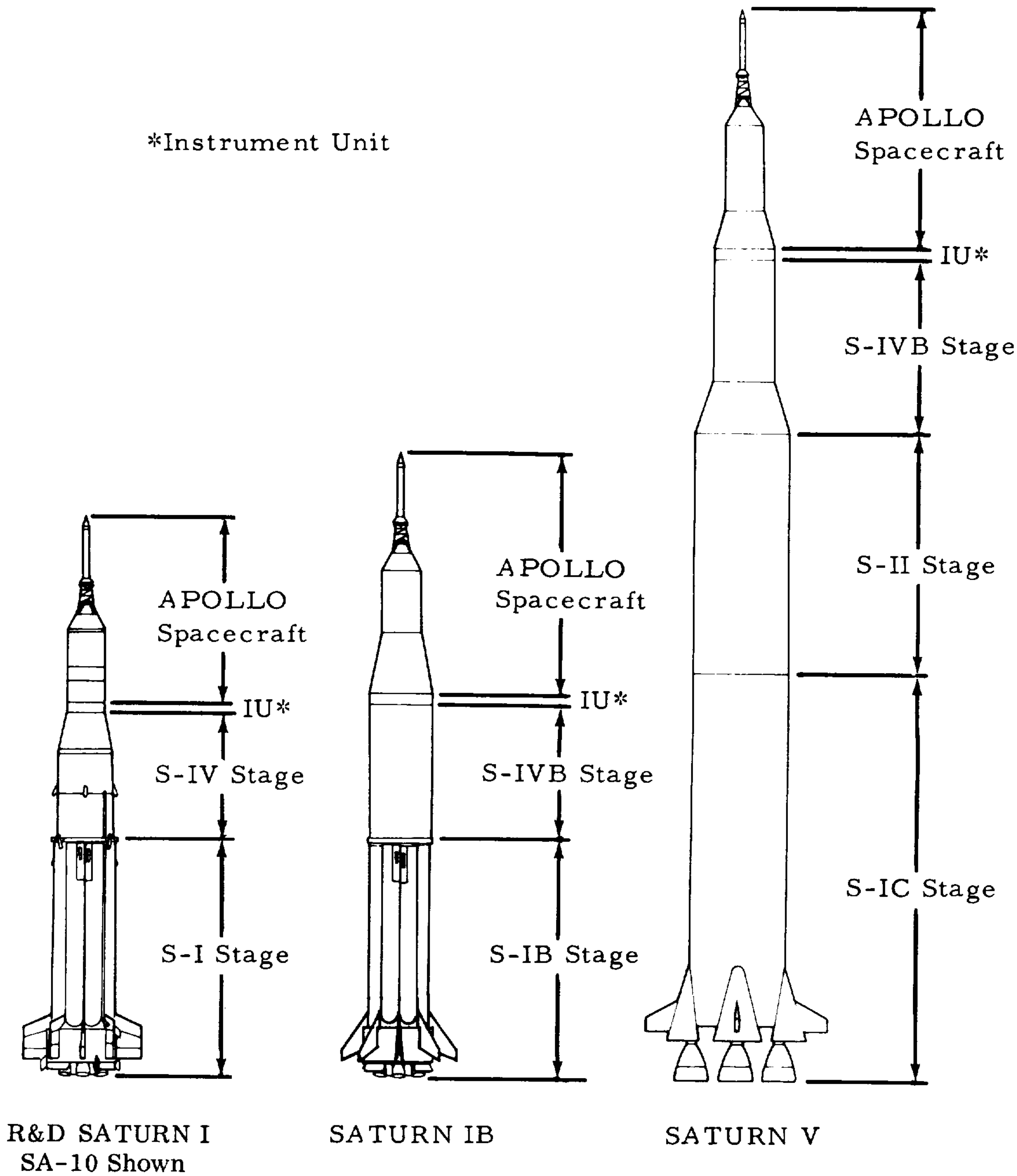
The systems descriptions in subsequent chapters of this volume cover the Saturn I, Saturn IB and Saturn V launch vehicles. An Apollo payload is termed a spacecraft. A spacecraft and a launch vehicle in combination are collectively termed a space vehicle. The configurations of the Saturn I, Saturn IB and Saturn V launch vehicles are shown in Figure 3-1. The salient features of these vehicles are noted in the paragraphs below. Detailed descriptions, including dimensions, are given in Chapters 2, 3 and 4.

3-3. SATURN I CONFIGURATION.

The Saturn I launch vehicle, Figure 3-1, consists of two propulsion stages and an instrument unit. The first stage is an S-I stage, with eight H-1 rocket engines which have a combined thrust of approximately 1,500,000 pounds. The four outboard engines are mounted in gimbals which permit them to be pivoted. A guidance and control system gimbals the engines as required to steer the space vehicle along a desired flight path. For aerodynamic stability, the first stage is fitted with eight fixed fins (four stub fins and four larger fins). The second stage of the launch vehicle is an S-IV stage, with six gimballed RL10A-3 engines which have a combined thrust of 90,000 pounds.

Ten research and development (R&D) Saturn I launch vehicles are scheduled for flight-testing the various vehicle components to be flown. The first four of these have a configuration designated as the Saturn I Block I launch vehicle. Each consists of an S-I first stage without fins, a dummy S-IV second stage, a dummy S-V third stage and an R&D payload. The other six R&D vehicles are Saturn I Block II launch launch vehicles. Each consists of a finned S-I first stage, a live S-IV stage, an instrument unit and a payload.

The numbering system for the Saturn I launch vehicles and their individual stages is included in Table 3-1.



3-2B

Figure 3-1. Configurations of Saturn-Apollo Space Vehicles

Table 3-1. Numbering System for Saturn Launch Vehicles and Stages

Launch Vehicle	First Stage	Second Stage	Third Stage	Instrument Unit
SATURN I BLOCK I Nos. SA-1 through SA-4	S-I Stage Nos. S-I-1 through S-I-4	S-IV Stage Nos. S-IV-1D through S-IV-4D	S-V Stage Nos. S-V-1D through S-V-4D	None
SATURN I BLOCK II Nos. SA-5 through SA-10	S-I Stage Nos. S-I-5 through S-I-10	S-IV Stage Nos. S-IV-5 through S-IV-10	None	Nos. S-IU-5 through S-IU-10
SATURN IB Nos. SA-201, SA-202, etc.	S-IB Stage Nos. S-IB-1 S-IB-2, etc.	*S-IVB Stage Nos. S-IVB/IB-1 S-IVB/IB-2, etc.	None	Nos. S-IU-201, S-IU-202, etc.
SATURN V Nos. SA-501, SA-502, etc.	S-IC Stage Nos. S-IC-1, S-IC-2, etc.	S-II Stage Nos. S-II-1, S-II-2, etc.	*S-IVB Stage Nos. S-IVB/V-1, S-IVB/V-2, etc.	Nos. S-IU-501, S-IU-502, etc.

*S-IVB stages are used in both the Saturn IB and the Saturn V launch vehicles.

3-4. SATURN IB CONFIGURATION.

The Saturn IB launch vehicle, Figure 3-1, consists of two propulsion stages and an instrument unit. The first stage is an S-IB stage, with eight H-1 engines which have a combined thrust of approximately 1,600,000 pounds. Four of the engines are gimballed for directional control. Eight fixed fins of equal size are fitted to the first stage to provide aerodynamic stability. The second stage is an S-IVB stage, with a single J-2 engine of 200,000 pounds thrust that is gimballed for directional control.

The numbering system for the Saturn IB launch vehicles and their individual stages is included in Table 3-1. The first Saturn IB is No. SA-201.

3-5. SATURN V CONFIGURATION.

The Saturn V launch vehicle, Figure 3-1, consists of three propulsion stages and an instrument unit. The first stage is an S-IC stage, with five F-1 engines which have a combined thrust of 7,500,000 pounds. The four outboard engines are gimballed for directional control. Four fixed fins of equal size are fitted to the first stage for aerodynamic stability. The second stage is an S-II stage, with five J-2 engines which have a combined thrust of 1,000,000 pounds. Four of these engines are gimballed. The third stage is an S-IVB stage with one gimballed J-2 engine of 200,000 pounds thrust.

The numbering system for the Saturn V launch vehicles and theri individual stages is included in Table 3-1. The first Saturn V is No. SA-501.

3-6. LAUNCH VEHICLE AXES.

The system of body axes used to described the attitude and motion of a launch vehicle about its center of gravity (CG) is shown in Figure 3-2. As is common in aerodynamic practice, the rotational motions of the vehicle are termed pitch, yaw and roll.

3-7. APOLLO SPACE CRAFT CONFIGURATION.

The launch configuration of the Apollo spacecraft is shown in Figure 3-3. In its complete form, this spacecraft is a payload for the Saturn V launch vehicle, and is capable of accomplishing a manned lunar landing mission, including the safe return of the crew to earth. In some Saturn-Apollo missions, as described in Chapters 2, 3 and 4, the payloads are spacecraft which are incomplete in varying degrees, consistent

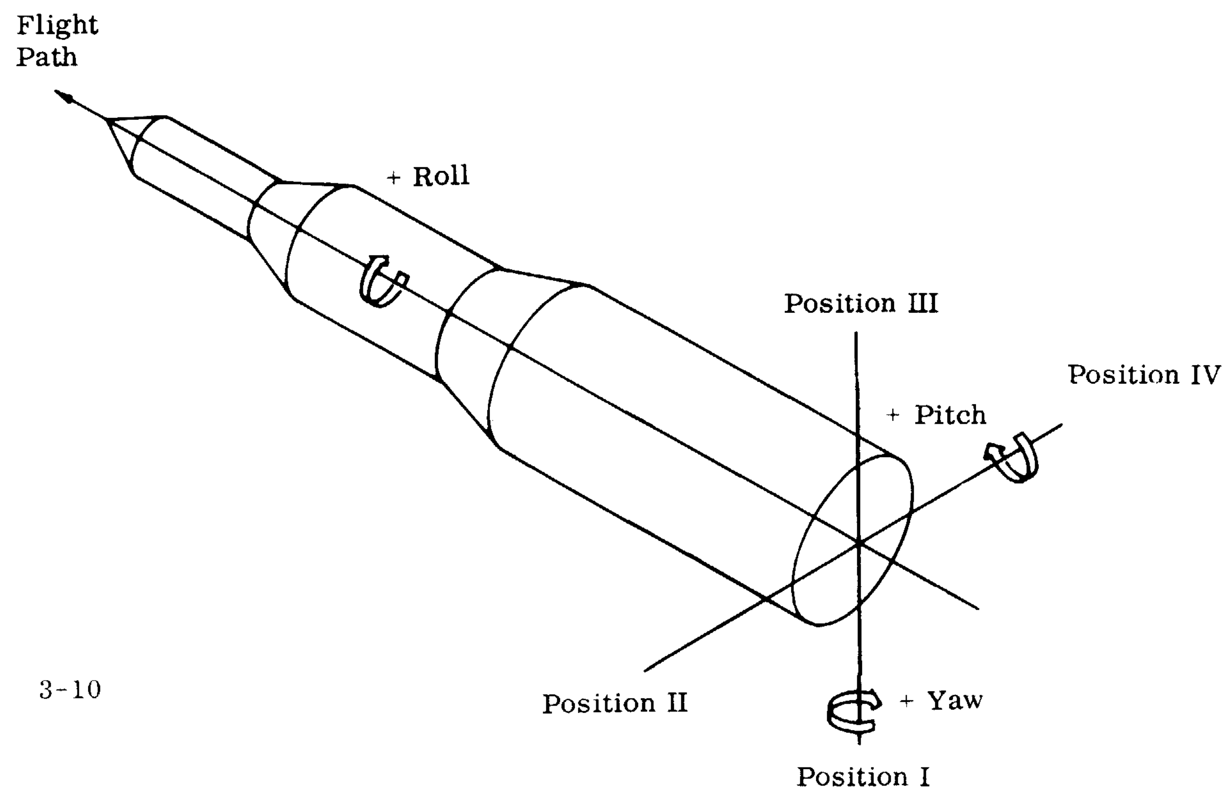


Figure 3-2. Launch Vehicle Axes

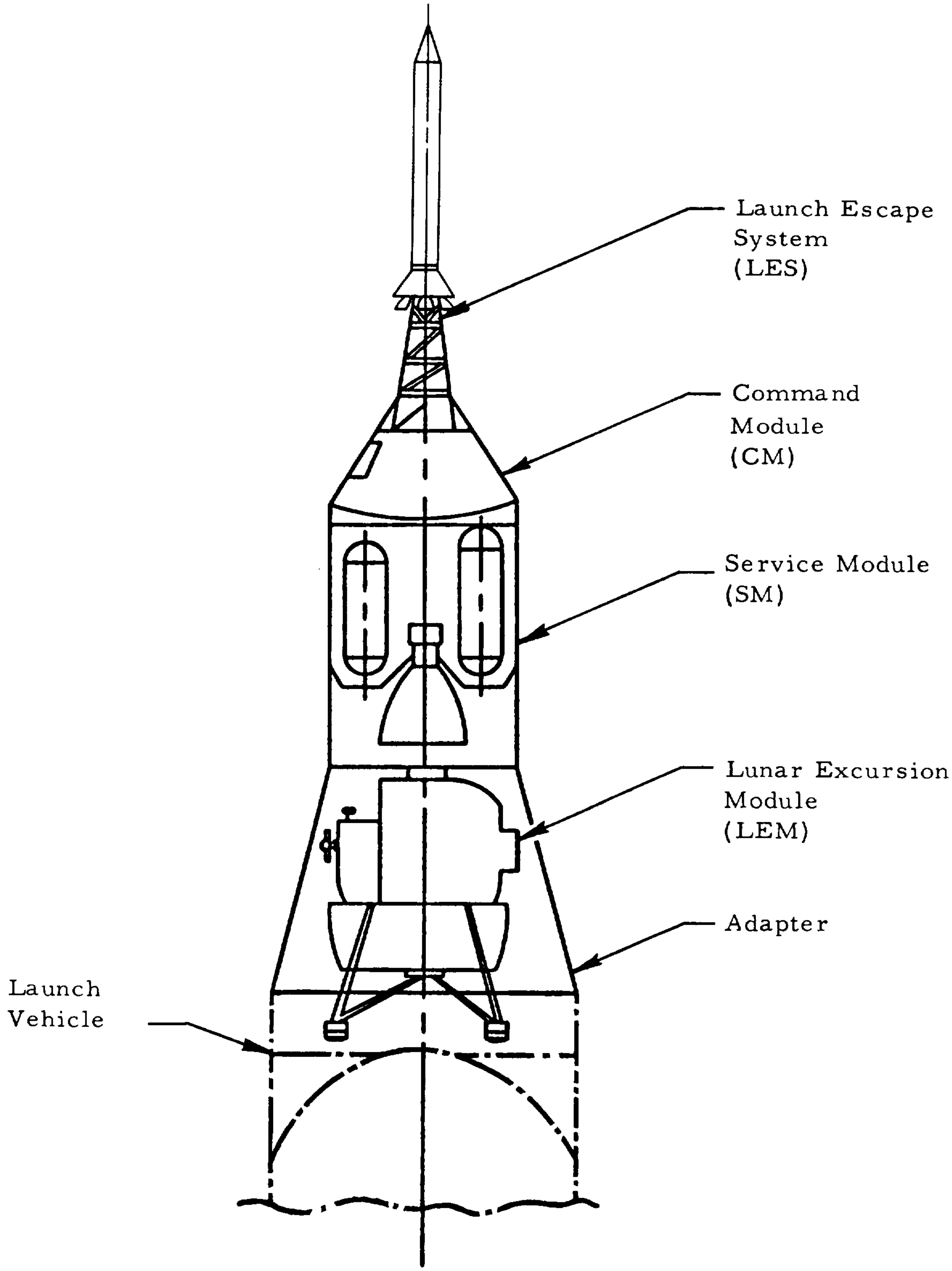


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Figure 3-3. Launch Configuration of Apollo Spacecraft with the mission objectives and the payload-carrying capacities of the launch vehicles.

The spacecraft (SC) is composed of the launch escape system (LES), the command module (CM), the service module (SM), the lunar excursion module (LEM), and the spacecraft adapter (Figure 3-3). The concept of individual functional units or modules is employed so that systems peculiar to a specfic mission can be modified without substantially affecting the design of systems common to general or ultimate missions. In a given mission, optimum weight is attained for each phase of flight by jettisoning of expendable units.

The LES, which is part of the CM, contains a launch escape rocket motor capable of lifting the CM free of the rest of the space vehicle. The purpose of the LES is the removal of the crew from the vehicle in the event of a serious emergency on the pad or during the early part of a mission. The forward section of the LES contains a smaller rocket motor which is capable of lifting the LES, alone, free of the CM. During a normal mission this motor is fired shortly after the second-stage launch vehicle engines are started, to jettison the LES.

The CM of the Apollo spacecraft, Figure 3-3, provides the three-man crew with a command center in which crew-initiated in-flight control functions are exercised. The CM provides the crew with living quarters also, and protects them from the space environment. The CM is the only part of the space vehicle that re-enters the earth's atmosphere under control, and the only part that is recovered after flight. The CM carries a thermal shield that protects it against aerodynamic heating during re-entry, a reaction control system, and parachutes that slow it to a safe speed for impact on land or on water. The earth landing is the only landing of the CM during a mission; the CM does not land on the moon, but remains in lunar orbit during lunar landing operations.

The SM contains the service propulsion system plus selected equipment and stores which service the equipment and crew of the CM. It is unmanned, does not require in-flight crew access and remains with the CM during lunar operations. It is separated from the CM prior to re-entry and is nonrecoverable. The SM provides propulsion capability for the CSM (the CM and SM combination) and its reaction control supplements that of the CM. Its structure provides a mounting surface and environmental protection for all SM systems, carries all ground and flight loads, and is compatible with the over-all spacecraft structure.

The LEM serves as a vehicle for carring two of the three-man crew and a development and scientific payload from the CSM in luncar orbit to the lunar surface and back. The LEM also provides a base for lunar operations and crew exploration in the vicinity of the lunar touchdown point. The LEM is fitted with a multi-strut, wheelless landing gear that helps to absorb the landing shock after the speed of descent has been slowed by the reverse thrust of a rocket engine. At liftoff from the moon, the LEM separates into two sections. The lower section, which includes the landing gear, serves as a launch platform for the upper section, or ascent stage, and remains in place on the moon. The spacecraft adapter provides the physical bond which mates the launch vehicle to the SM. For the lunar landing mission the spacecraft adapter houses the LEM.

To prepare the spacecraft for deployment of the LEM, the configuration shown in Figure 3-3 (less the jettisoned LES) is altered in flight. This alteration is effected after the last stage (the S-IVB stage) of the launch vehicle has propelled the configuration of Figure 3-3 (less the jettisoned LES) into the translunar trajectory, a flight course that will transfer the spacecraft from earth orbit to lunar orbit. The CSM separates from the LEM, instrument unit and S-IVB stage (collectively designated LEM/IU/S-IVB) and the adapter is jettisoned. While the S-IVB stage of the launch vehicle stabilizes the LEM/IU/S-IVB, the CSM turns end for end, lines up with the LEM/IU/S-IVB and rejoins the LEM/IU/S-IVB, so that the nose of the CM is coupled to the LEM. These evolutions are termed turn-around docking.

The S-IVB stage and instrument unit (collectively designated S-IVB/IU) are then jettisoned. At this point the launch vehicle completes its part in the Saturn-Apollo mission The spacecraft, which now consists of the CSM and the LEM, continues along the translunar trajectory, executing one or more midcourse corrections. As the spacecraft approaches the moon, the propulsion engine in the SM (at the forward end of the altered configuration) is fired to decrease the speed of the spacecraft permitting it to enter the lunar orbit. While the spacecraft coasts in lunar orbit, two crew members transfer from the CM to the LEM through connecting hatches. The LEM then separates from the CSM and descends to the moon, while the CSM continues in lunar orbit with the third crew member on board in the CM. On completion of the lunar exploration, the ascent stage of the LEM rises on a course that intersects the orbital path of the CSM, and the two are rejoined. This technique is termed Lunar-Orbit Rendezvous (LOR). The LEM crew then returns to the CM, and the ascent stage of the LEM is jettisoned, remaining in lunar orbit. For the return of the CM to earth, the propulsion engine of the SM is fired to place the CSM on an earth transfer trajectory. Later, after one or more midcourse corrections, and before re-entry, the SM is jettisoned. The CM is maneuvered by its reaction control system, so that its heat shield faces forward, and the CM re-enters the earth's atmosphere. After re-entry a drogue parachute is deployed to stabilize the CM and slow it further, and main parachutes are deployed for the final descent to an earth landing.

CHAPTER 1

SECTION IV

PROGRAM PLAN

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PROGRAM PLAN

The Marshall Space Flight Center is responsible for providing the launch vehicles needed for the Apollo program, together with the associated support equipment. To discharge these responsibilities MSFC performs the functions of project management, engineering design and development, fabrication and assembly, procurement of subcontracted items, modifications and construction of facilities, and qualification, checkout and flight testing.

4-1. SCHEDULES.

Presidential and Congressional authorization for a National Space Exploration Program calls for a manned lunar landing within this decade as one of the major program milestones. The Saturn project is organized to meet a schedule which will provide a launch vehicle capable of performing this mission within the prescribed time, while also permitting the early testing of components and methods. This schedule is shown in Table 4-1.

4-2. MANAGEMENT PLAN.

The organization of the Marshall Space Flight Center is illustrated in Figure 4-1. The present organization is the result of revisions effective August 26, 1963, which streamlined the Center, made it stronger, more dynamic, and more flexible, the better to meet the challenges of the Manned Lunar Landing Program. It will also be noted that both the Michoud Operations and the Mississippi Test Operations have now completed their resources buildup.

To complete the scope of work of the Saturn launch vehicle project in accordance with the established schedules, the Marshall Space Flight Center is drawing upon the resources of industrial contractors. The procurement of the industrial support is so organized as to require a minimum number of individual negotiations conducted by MSFC. The instrument units for all of the Saturn launch vehicles are designed and manufactured at MSFC. The first stages of the operational Saturn I, IB and V launch vehicles are produced at MSFC's Michoud Operations (New Orleans, Louisiana).

Table 4-1. Saturn I, IB and V Delivery and Launch Schedule

Year	1963		1964				1965				1966				1967				1968				1969			
Quarter	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4	1	2	3	4
Saturn I																										
Delivery	1		1	1	1	1	1																			
Launch			*	1	1	1	1																			
Saturn IB																										
Delivery									1	1	1	1	1	1	1	1	1	1	1							
Launch											**	1	1	1	1	1	1	1	1	1	1	1				
Saturn V																										
Delivery													1	1	1	1	1	2	1	2	1	2	1	1		
Launch															***	1	1	1	2	1	2	1	2	1	2	

* SA-5
** SA-201
*** SA-501

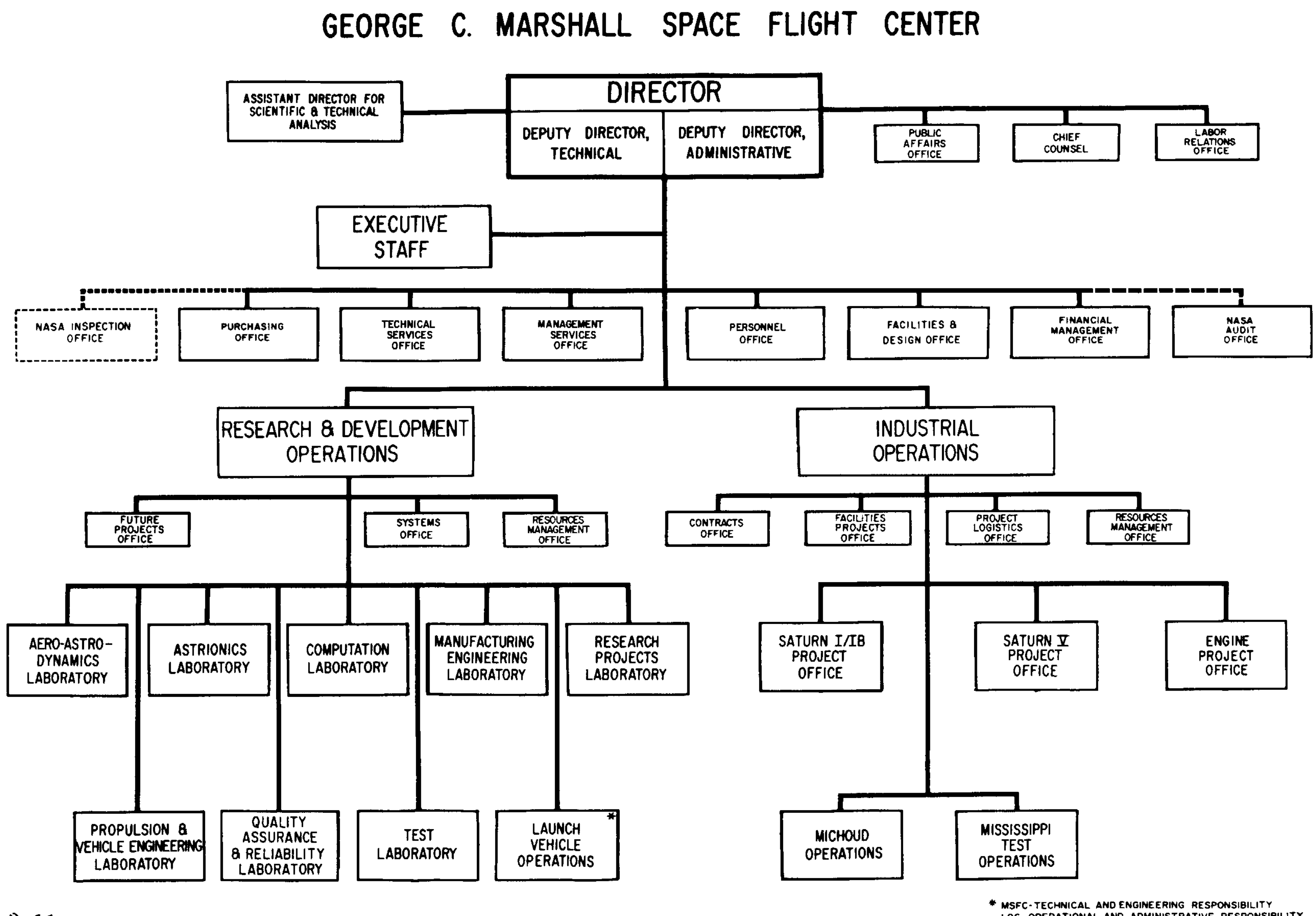

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Figure 4-1. Marshall Space Flight Center Organization

It will be noted the other stages of the Saturn launch vehicles are produced at contractor plants. The responsibilities of the major contractors are indicated in Figure 4-2.

Industrial participation in MSFC programs accounts for more than ninety percent of the total budget. The Industrial Operations consolidates all industrial project management activities, while the Research and Development Operations carry out the Huntsville based research and development work and provides the knowledge and penetration-in-depth to assist in, monitor, and, influence the technical effort at the many contractor organizations. Two organizations, the NASA Audit Office and the NASA Inspection Office, reporting to NASA Headquarters, provide a review capability for NASA Headquarters at this Center.

The interdependence of the Saturn launch vehicles, the Apollo spacecraft and the launch facilities necessitates effective coordination among the Marshall Space Flight Center (MSFC), the Manned Spacecraft Center (MSC), and the Launch Operations Center (LOC). This coordination is accomplished by a formal organization as shown in Figure 4-3.

4-3. PANEL REVIEW BOARD.

The Panel Review Board supervises the activities of, and acts as an appeal board for, the inter-Center Panels. The members of the PRB are as follows:

- OMSF: The Deputy Director (Systems) and the Deputy Director (Programs).
- MSFC: The Director and Deputy Director for Research and Development Operations and Director for Industrial Operations.
- MSC: The Deputy Director for Development and Programs and the Deputy Director for Mission Requirements and Flight Operations.
- LOC: The Assistant Director for Plans and Project Management.

The OMSF Deputy Director (Systems) serves as Chairman. The Executive Secretariat consists of a member from each Center, and supports the PRB.

4-4. INTER-CENTER PANELS.

The panels are formed to make available the technical competence of OMSF, MSFC, LOC and MSC, and their contractors for the solution of the interrelated problems of the launch vehicle, the spacecraft, support facilities, and associated equipment. The panels are responsible to the Panel Review Board. Each Panel has the authority

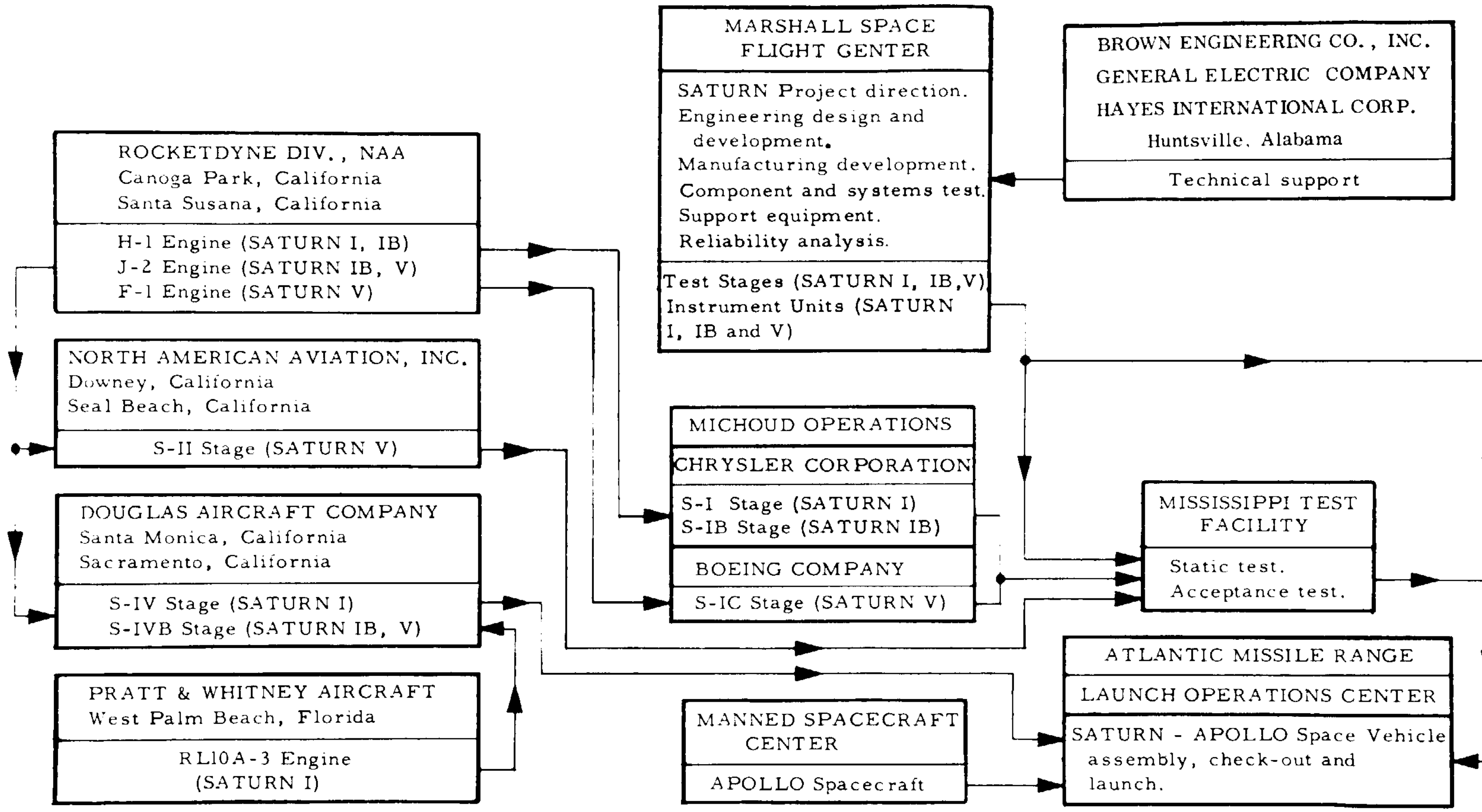

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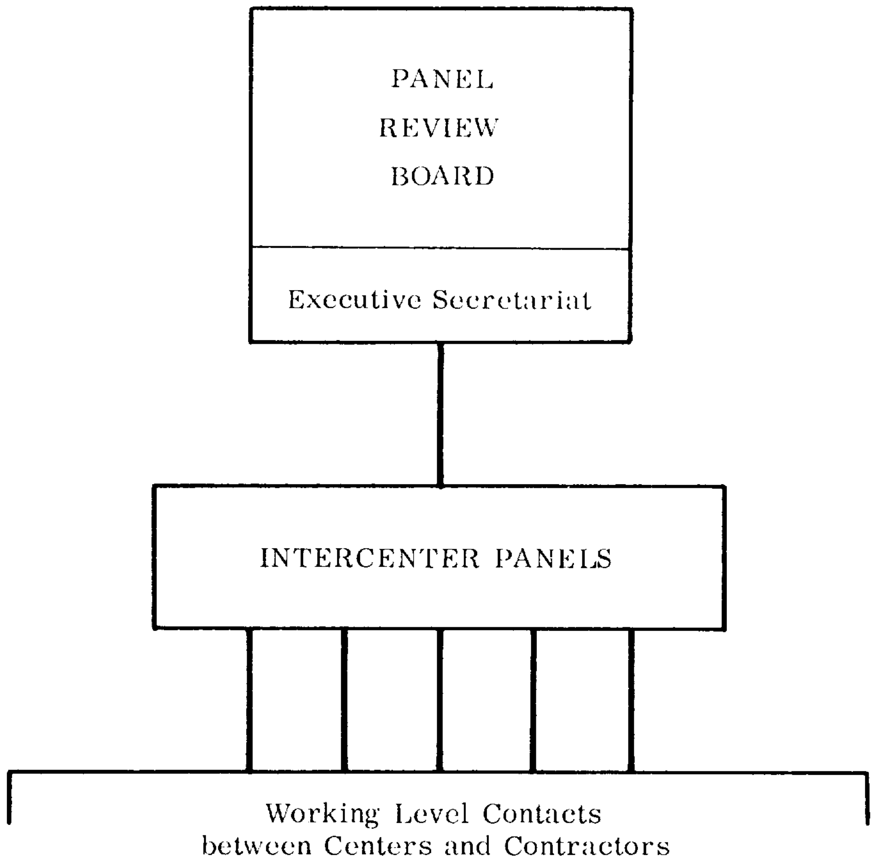
Figure 4-2. Major Contractor Responsibilities in Saturn Launch Vehicle Project



3-12

Figure 4-3. Apollo Program Coordination

for its defined area and may initiate action and resolve problems of design, analysis, test and operations.

Panels presently constituted are as follows:

- Launch Operations Coordination Panel
- Mechanical Integration Panel
- Electrical Systems Integration Panel
- Instrumentation and Communication Panel
- Flight Mechanics, Dynamics and Control Panel
- Flight Evaluation Coordination Panel
- Crew Safety Panel
- Mission Control Operations Panel
- Documentation Panel

4-5. WORKING GROUPS.

The purpose of the working groups is to initiate technical direction to the prime contractor through Industrial Operations and to validate the prime contractor activity in matters of stage design, development, manufacture, checkout, test, launch preparation, and flight evaluation.

The present working groups are:

- Electrical Systems Design Integration Working Group
- Vehicle Mechanical Design Integration Working Group
- Vehicle Instrumentation Working Group
- Vehicle Dynamics and Control Working Group
- Launch Operations Working Group
- Flight Evaluation Working Group
- Systems Checkout Working Group
- Manufacturing Engineering Working Group
- Static Firing Working Group

4-6. RELIABILITY.

The reliability goals for the Saturn project are consistent with the requirement that the space vehicle be suitable for manned use. MSFC is responsible for the reliability of all systems of the launch vehicles and the associated support equipment.

The reliability effort for the Saturn systems is directed toward achieving design maturity early in the development periods, so that the reliability inherent in the design concepts for the systems can be approached as the ultimate objectives. The reliability goals are expressed where possible in terms of mean-times-to failure or safety margins, for given phases of the project.

The activities that are undertaken to achieve the reliability goals include mission profile examinations, design reviews, failure analyses, component verification and system verification. Disciplines, facilities and controls for the rapid collection and dissemination of reliability data are established as a continuing effort. Reliability estimation models are developed to indicate the level of reliability that can be achieved within the current state-of-the-art. Information is obtained both from laboratory test results and from flight test results to determine the actual reliability that is being achieved and to evaluate each equipment's performance in terms of over-all mission success.

Other areas of activity in the reliability program are concerned with achieving equipment maturity as early in the program as possible. The design review system is employed to provide for each design which is produced by MSFC or one of its subcontractors a review in detail by the most mature and experienced engineers and scientists of the MSFC rocket team. The intent of this activity is to ensure that each design is of the same quality which would be achieved if our most mature scientists participated in each detailed design activity.

The failure analysis activity is directed at the detailed analysis of each failure that may occur in any portion of the testing program, to correct deficiencies as early in the program as possible. A concentrated effort is made to correct any deficiency the first time it is detected.

These proven reliability techniques are carried out as an integral part of the design and manufacturing activities at Marshall and in the plants of each subcontractor to ensure the achievement of the Saturn project reliability goals.

4-7. TEST PLANS.

Mission success and personnel safety are being ensured by a test program so comprehensive that all launch vehicle hardware, from the smallest part to the largest assembly is covered. Assurance of proper operation and adequate reliability is accomplished through implementation, in proper combination, of the concepts described below.

Hardware criticality is determined by a failure effect analysis on each individual item. Qualification testing and reliability demonstration testing have a mandatory dependency on the hardware criticality. In addition, all other test planning must be cognizant of and keyed to hardware criticality.

Design development tests are performed to establish the engineering design verification or provide design change information. Where the design status is sufficiently advanced, the test is devised to serve also as a qualification test.

As a general requirement, all flight hardware must be qualified by ground qualification test prior to unmanned flight, and by flight qualification test prior to manned flight. Similarly, ground support systems hardware must be qualified prior to use with flight hardware.

Another major objective of the testing program is the acquisition of information and data for evaluation of hardware reliability. Hardware in the most severe criticality categories is subjected to reliability demonstration tests.

Production hardware testing ensures acceptance for fabrication and assembly of hardware with satisfactory and uniform quality. This is accomplished by a production test program covering all testing phases of manufacturing, and quality control activities from receiving tests to final acceptance tests. Tests are performed at all hardware generation levels, from materials and piece parts to complete stages and instrument untis. Premating checkout tests are conducted on each stage and instrument unit as they are progressively prepared for assembly into a launch vehicle, and on the launch vehicle prior to assembly with the spacecraft.

CHAPTER 2

SECTION V

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SECTION V.

INTRODUCTION

5-1. SATURN I LAUNCH VEHICLE

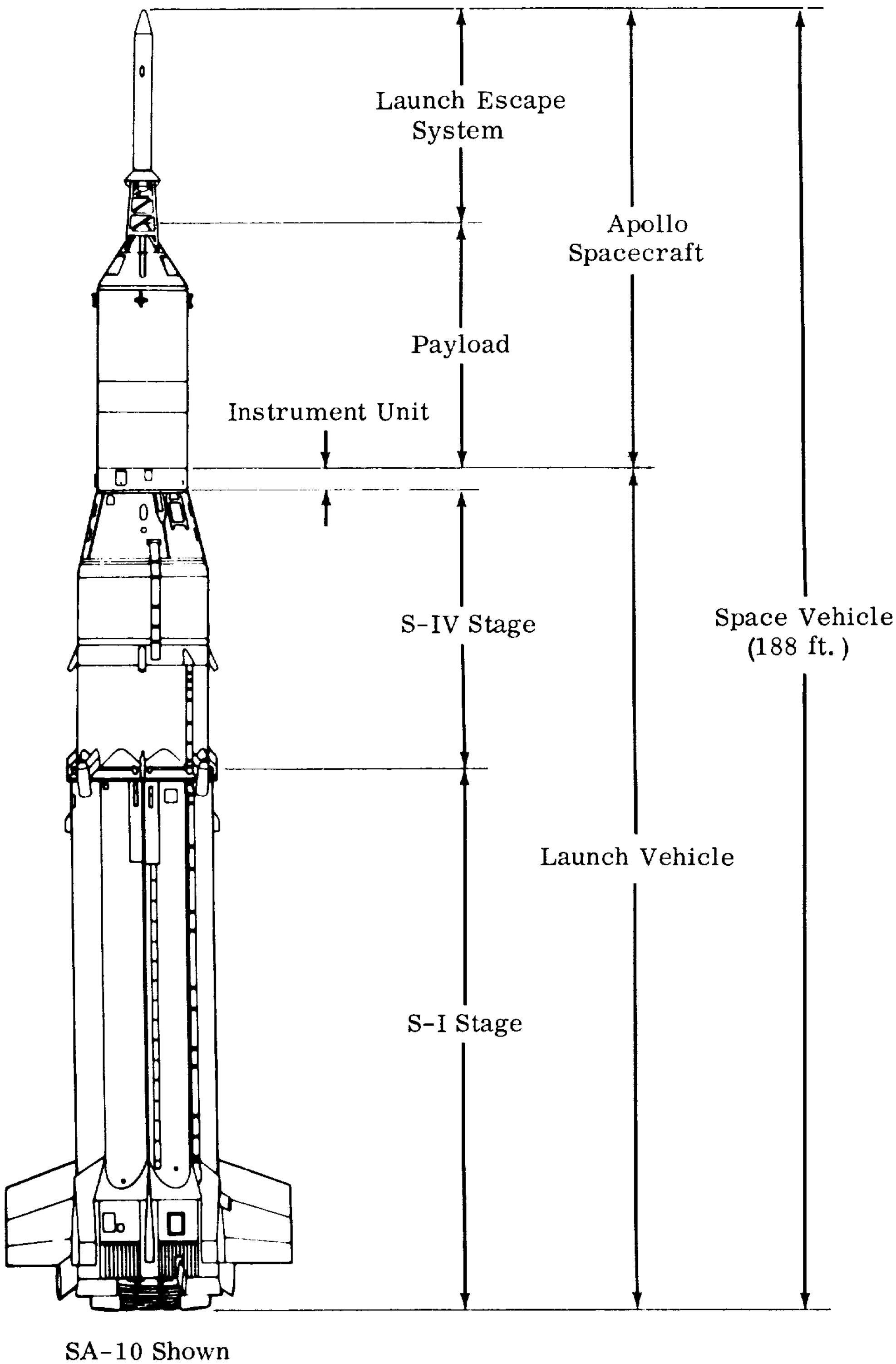
NOTE

The material in this chapter was prepared when vehicle SA-10 had the space vehicle qualification mission in the Saturn I program. This mission no longer exists and at this time a new mission has not been defined for the vehicle. Since the Saturn I vehicle has the capability for the original mission, the decriptive material on this mission has not been deleted. Redefinition of SA-10 objectives will be covered by revision of this document.

The Saturn I is the first generation of the Saturn launch vehicle family. Intended first for research and development of a multi-engine, multi-stage booster and secondly, for development of the Apollo spacecraft, much of the first stage design is based on components used in the earlier Redstone and Jupiter programs. The four Saturn I - Block I vehicles, already flown, consisted of an S-I first stage, dummy S-IV second stage, dummy S-V third stage, and a modified Jupiter nose cone as payload. The Saturn I - Block II launch vehicle, Figure 5-1, is composed of an S-I first stage, an S-IV second stage and an instrument unit mounted above the second stage. The payload varies from a modified Jupiter nose cone for SA-5 to an Apollo payload with attached LES for SA-6 through SA-10. The Apollo payload consists of a CM, an SM, and an adapter section. SA-8 and SA-9 carry a micrometeroid detection capsule within the SM as a secondary payload. Operational data for launch vehicle SA-10 are listed in Table 5-1.

5-2. SATURN I-APOLLO MISSION OBJECTIVES

The ultimate mission of the Saturn I launch vehicle was the placing of an Apollo spacecraft into earth orbit for manned flight tests. This mission was to have been acomplished by four operational Saturn I manned flights preceded by a series of four Saturn I - Block I and six Saturn I - Block II R&D flights. To reduce program costs and eliminate schedule conflicts between the Saturn I and Saturn IB programs, all manned flights of the Saturn I vehicles have been cancelled. The primary mission



3-13

Figure 5-1. Saturn I Launch Vehicle

[redacted]

Table 5-1. Saturn I, SA-10 Vehicle, Data

Item	Data
VEHICLE	
Number of stages	2
Length - without spacecraft	124.5 feet
Maximum diameter - without fins	22.8 feet
- with fins	40.7 feet
[1]Launch vehicle weight - at ground ignition	1,165,000 pounds
Payload Type	Apollo Spacecraft
[2]Payload weight - at ground ignition	29,100 pounds
[3]Injection weight - Earth orbit	22,500 pounds
S-I STAGE	
Prime contractor	Chrysler Corporation
Length	80.2 feet
Maximum diameter - without fins (across thrust structure)	22.8 feet
- with fins	40.7 feet
Stage weight - at ground ignition	1,016,000 pounds
Dry weight	103,000 pounds
Engines	Rocketdyne H-1 (8)
Total nominal thrust (sea level)	1,500,000 pounds
Propellants	LOX and RP-1
Mainstage propellant weight	880,000 pounds
Mixture ratio (oxidizer to fuel)	2.26:1
Specific impulse (sea level)	256 seconds
S-IV STAGE	
Prime contractor	Douglas Aircraft Co.
Length	41.4 feet
Diameter	18.3 feet
[4]Stage weight - at ground ignition	114,000 pounds
[4]Dry weight	13,000 pounds
Engines	Pratt and Whitney RL10A-3 (6)

[redacted]

Table 5-1. Saturn I, SA-10 Vehicle Data (Cont'd)

[1]Includes two stages, instrument unit, payload and LES.

[2]Includes 6600 pounds for the LES.

[3]100-nautical mile circular orbit, payload only.

[4]Excludes 2100 pounds for the S-I/S-IV interstage

Note: Weights in this table are specification weights from Memorandum No. M-P&VE-V-33, "Saturn I, IB and V Launch Vehicle Specification Weights and Compatible Trajectories," dated May 13, 1963.

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objectives remaining are: development of the launch vehicle systems required for a 1,500,000-pound thrust booster which remains virtually unchanged in the Saturn IB, and development of liquid hydrogen - liquid oxygen propulsion for the second stage.

Secondary objectives are: determination of launch and exit environmental parameters using Apollo boilerplate spacecraft, and micrometeriod experiments on SA-8 and SA-9. Detailed information about the Saturn I Apollo mission objectives and flight data is summarized in Table 5-2.

5-3. MISSION PROFILE

The SA-10 vehicle mission profile, through which a Saturn I launch vehicle lifts an R&D Apollo spacecraft (less LEM) into a 100-nautical mile circular earth orbit is illustrated in Figure 5-2. The mission events occurring along the profile are listed in Table 5-3. The mission profile for SA-10 is chosen as being the most representative Saturn/Apollo flight of the Saturn I vehicles. Similar or lesser requirements are placed on the launch vehicle in missions SA-5 through SA-9.

The mission of the launch vehicle ends with the separation of the Apollo spacecraft from the instrument unit, Event No. 6 of the mission profile. The launch vehicle mission can be divided into prelaunch, launch, ascent, and orbital phases. For the purpose of this description these phases are defined by the following limits:

a. Prelaunch- Beginning with stage testing and ending with start of countdown.

b. Launch - Beginning with start of countdown and ending with liftoff.

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LAUNCH VEHICLE NO.	SA-1	SA-2	SA-3	SA-4
MANNED/UNMANNED MISSION	←			
MISSION OBJECTIVES A. SATURN LAUNCH VEHICLE (LV) OBJECTIVES	P 1. Structures. 2. Propulsion (165 K engines).			
B. APOLLO SPACECRAFT (SC) OBJECTIVES	None ———			
C. SPACE VEHICLE (SV) OBJECTIVES	None			
D. OTHER OBJECTIVES				
LAUNCH VEHICLE PAYLOAD CAPACITY (LBS.)	———			
SPACECRAFT ORBITAL WEIGHT (lbs.)*	———			
FLIGHT DATA A. FLIGHT AZIMUTH B. FLIGHT PROFILE C. NOMINAL ORBIT ALTITUDE (naut. mi.) D. MINIMUM STAY TIME IN ORBIT	 100° Ballistic ———			
AMR LAUNCH COMPLEX NO.	34			
RECOVERY	None			

P Primary objective of SV mission; S Secondary objectiv

*Does not include 6,600 lb. weight of Launch Escape Syst

FOLDOUT FRAME 1

[redacted]

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Tab

SA-5	SA-6	SA-7	SA-9
		Unmanned	
P 1. Structures. 2. Propulsion (188 K engines). 3. Guidance (Passengers). 4. S-I/S-IV Stage separation.	P 1. Structures. 2. Propulsion. 3. Guidance (active). 4. S-I/S-IV Stage separation.		P 1. Structures. 2. Propulsion. 3. Guidance (active). 4. S-I/S-IV Stage separation.
None ———	S 1. Launch and exit environment. 2. LES structural characteristics. 3. LES jettison. SC No. BP-13	S SC No. BP-15	S SC No. BP-16
None	S 1. Physical and flight compatibility of LV and SC. 2. Compatibility of R&D communications and instrumentation between SV and ground stations.	S	S
			S 1. Micrometeroid experime
18,500	18,600	18,600	16,600
———	12,360 · Ballast to 18,600		12,360
105^{o} Elliptical Orbit 100/>100 1 Day	105^{o} Orbit 1 Day	105^{o} Orbit 1 Day	105^{o} Elliptical Orbit 255/675 1 Year
37B	37B	34	37B
None	None	None	None

e of SV mission

m (jettisoned after second-stage ignition).

FOLDOUT FRAME 2

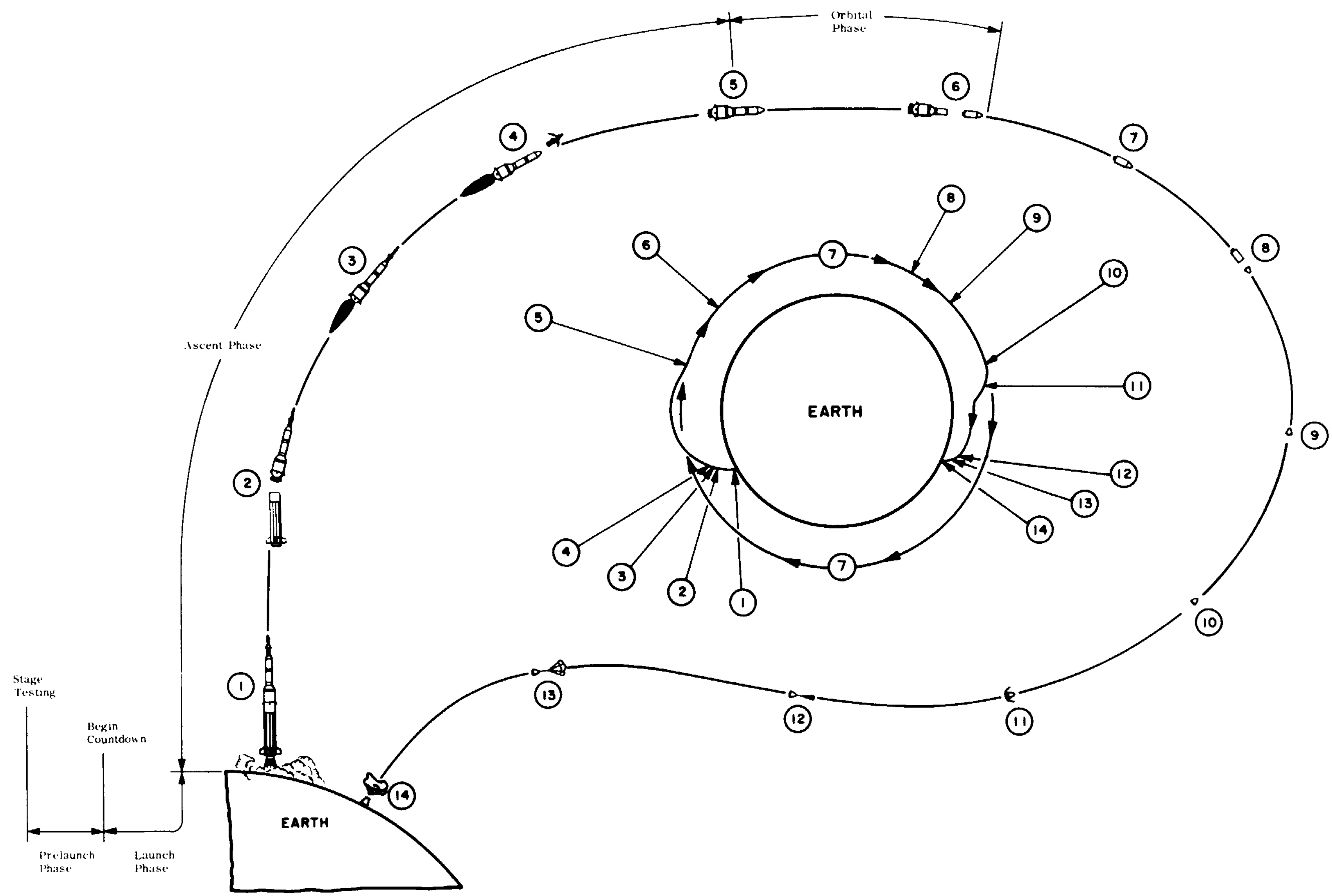
[redacted]

le 5-2. Saturn I-Apollo Mission Objectives and Flight Data

	SA-8	SA-10
		→
	P 1. Structures. 2. Propulsion. 3. Guidance (active). 4. S-I/S-IV Stage separation.	S LV QUALIFICATION 1. Structures. 2. Propulsion. 3. Guidance (active). 4. S-I/S-IV Stage separation.
	S Alternate Mission (with BP-18) 1. SM and Adapter structural integrity 2. CSM/Adapter separation. 3. Crew safety. SC No. BP-26	S SC QUALIFICATION 1. Structures. 2. SC systems. 3. SM propulsion (w re-start) 4. CM re-entry. 5. LES jettison. 6. SM-CM separation. 7. Crew safety. 8. Recovery system. 9. Guidance and Navigation. SC No. AFRM-009
	S Alternate Mission (with BP-18) 1. CSM/LV separation.	P SV QUALIFICATION 1. LV-SC compatibility. 2. LV-SC separation. 3. Instrumentation, Communications, Tracking.
nt.	S 1. Micrometeroid experiment.	
	16,600	22,000
	12,360	22,000
	105^{o} Elliptical Orbit 255/675 1 Year	72^{o} 105^{o} Orbit or Sub-orbit
	37B	34
	None	Water

FOLDOUT FRAME 3 [redacted]

[redacted]



3-7A

Figure 5-2. Saturn I-Apollo Mission Profile

[redacted]

c. Ascent - Beginning with liftoff and ending with orbit injection.

d. Orbital - Beginning with orbit injection and ending with payload separation.

Table 5-3. Description of Saturn I-Apollo Mission, Vehicle SA-10

Event No. *	Approx. Time After Liftoff (Sec.)	Event
1	0	Liftoff of Saturn I-Apollo space vehicle (SV) from AMR launch complex No. 34.
	8	Start roll to align SV pitch plane with flight azimuth. Start time tilt. (By launch vehicle (LV) systems.)
	18	Arrest roll (SV correctly aligned with flight azimuth).
	20	Activate accelerometer control of LV guidance and control system.
	90	Deactivate accelerometer control of LV guidance and control system.
	143	Arrest time tilt.
	150	Shut down inboard first-stage (S-I stage) engines.
	156	Shut down outboard first-stage engines. Start timing for stage separation sequence.
	156.3	Ignite second-stage (S-IV stage) ullage motors (3-second minimum duration of burning).
2	156.4	Separate first stage from second stage. Transfer control functions from first to second stage. Ignite first-stage retromotors.
3	158	Start second-stage engines.
4	168	Jettison Launch Escape System (LES) from Apollo spacecraft (SC).
	176.4	Jettison second-stage ullage motors.
	179	Start path guidance mode.

*No. Refers to Figure 5-2. (Major events indicated only)

[redacted]

Table 5-3. Description of Saturn I-Apollo Mission, Vehicle SA-10 (Cont'd)

Event No. *	Approx. Time After Liftoff (Sec.)	Event
	550	Reach path angle parallel to local horizontal, at altitude of approximately 112 naut. mi. (207 km); continue to pitch down.
5	630	Inject SC into 100-naut. mi. (185-km) circular earth orbit. Shut down second-stage engines.
6		Separate second stage and instrument unit from SC, ending LV mission.
7		Continue orbital coast of SC. Perform scheduled mission exercises.
8		Jettison Service Module (SM) of SC from Command Module (CM).
9		Orient CM in re-entry attitude.
10		Initiate CM re-entry.
11		Re-enter earth's atmosphere.
12		Deploy drogue parachute.
13		Jettison drogue parachute and deploy main parachutes.
14		Alight on water or on land.

*No. Refers to Figure 5-2. (Major events indicated only)

5-4. LAUNCH VEHICLE REQUIREMENTS

The SA-10 vehicle is required to inject a payload of 22,500 pounds into a 100-nautical mile circular earth orbit. To accomplish this, the launch vehicle must boost the payload to altitude, guide it so that the final flight-path angle is 90 degrees (with respect to local vertical) and impart to it a final velocity of 25,581 ft/sec. Its R&D mission requires that information on vehicle performance be returned to earth. The vehicle is subject to the following constraints:

a. Launch site (Cape Kennedy) latitude of 28 degrees, 30 minutes which intro-

[redacted]

duces a minimum orbital inclination of the same degree.

b. Launch Facility, VLF 34, requires a launch azimuth of 100 degrees.

c. Vehicle visibility requirement for tracking and telemetry networks restricts azimuth path to a sector from 70 degrees to 110 degrees.

d. Range Safety limits flight azimuths to a sector from 45 degrees to 110 degrees. Flights outside this sector endanger populated areas.

To optimize vehicle performance and increase range safety, a minimum vehicle lift-off thrust to weight ratio of 1.25:1 is specified. Higher mission reliability is achieved by a single engine out capability in either stage provided that the other stage functions properly.

The primary vehicle requirements are accomplished by systems described in this chapter as Astrionics, Structures, Propulsion, Mechanical, and Ground Support Equipment. Tables 5-4 through 5-7 list the basic requirements of each of these systems for the four phases of the launch vehicle mission.

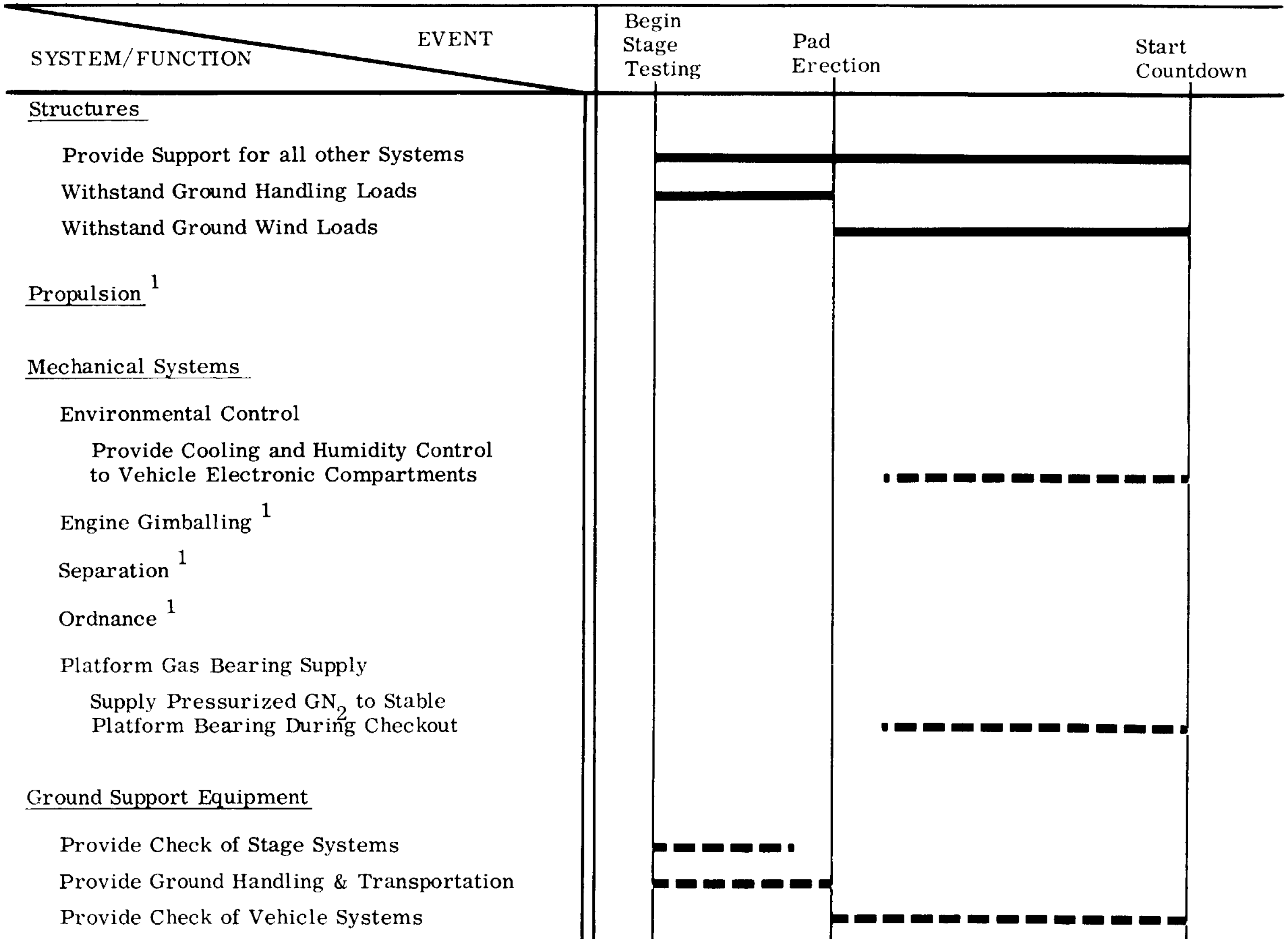
The time function in the table is not to scale as it is intended to indicate only relative phasing of requirements. Although the table is primarily a listing of system requirements, certain major events are listed to show their relationship to the requirements.

Detailed information on the systems is presented in Sections VI through X. Inboard profiles of each stage are included in Section XI.

Table 5-4. Saturn I Requirements, Prelaunch Phase

SYSTEM/FUNCTION \ EVENT	Begin Stage Testing	Pad Erection	Start Countdown
Astrionics			
Command			
Control Checkout Sequence		– – – – – –	
Control Ground Support Activity		– – – – – –	
Communications			
Transmit Information for Checkout		– – – – – –	
Transmit Mission Planning Information		– – – – – –	
Instrumentation			
Provide Vehicle System Information in Support of Checkout		– – – – – –	
Checkout			
Provide Stimuli and Comparison Networks for Checkout		– – – – – –	
Guidance [1]			
Attitude Control and Stabilization [1]			
Tracking [1]			
Range Safety [1]			
Electric Power			
Provide Ground Power for System Checkout		– – – – – –	

Table 5-4. Saturn I Requirements. Prelaunch Phase (Cont'd)



Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬▬ Operating; ▬ ▬ Intermittent Operation.

Table 5-5. Saturn I Requirements, Launch Phase

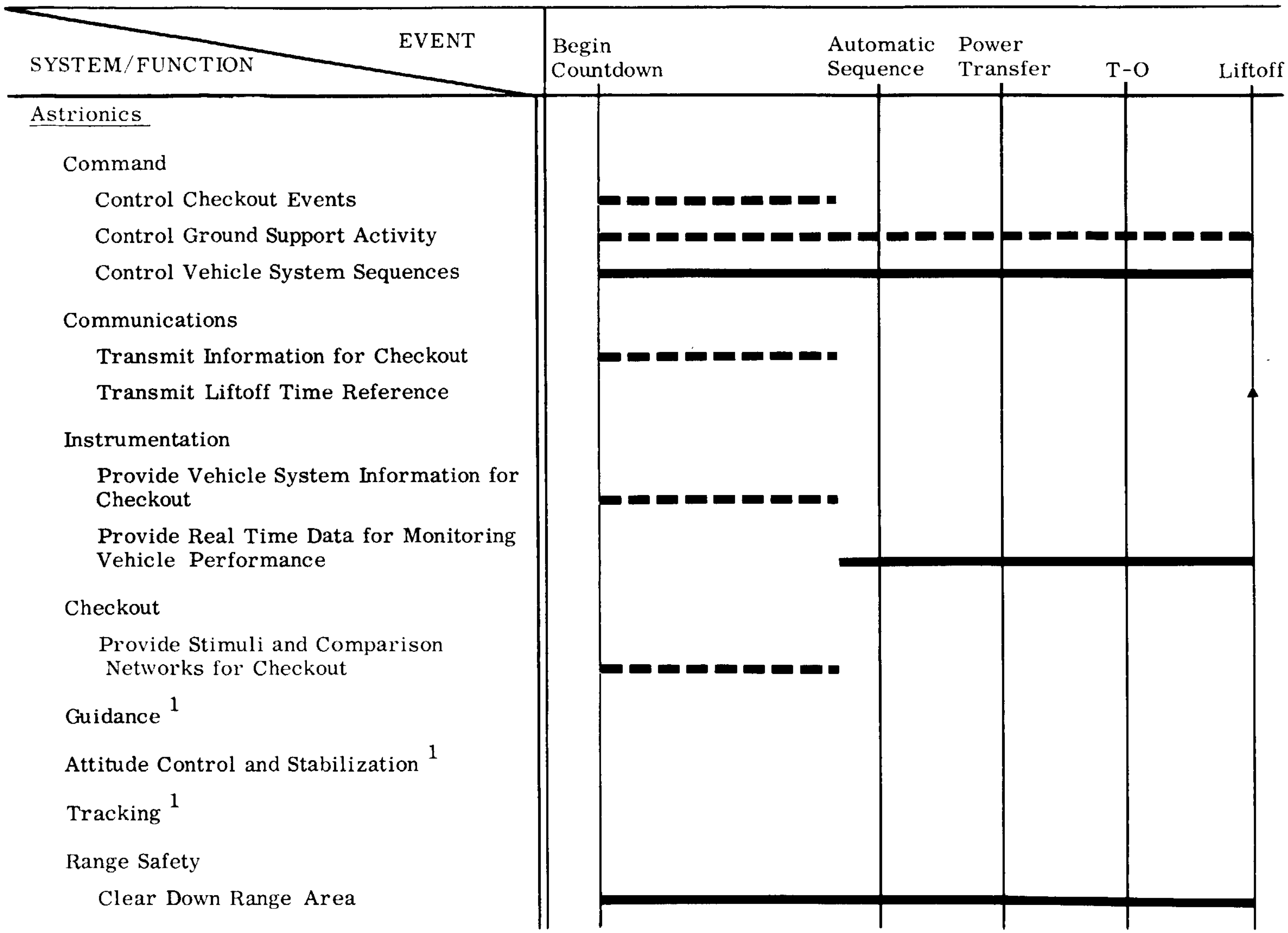


Table 5-5. Saturn I Requirements, Launch Phase (Cont'd)

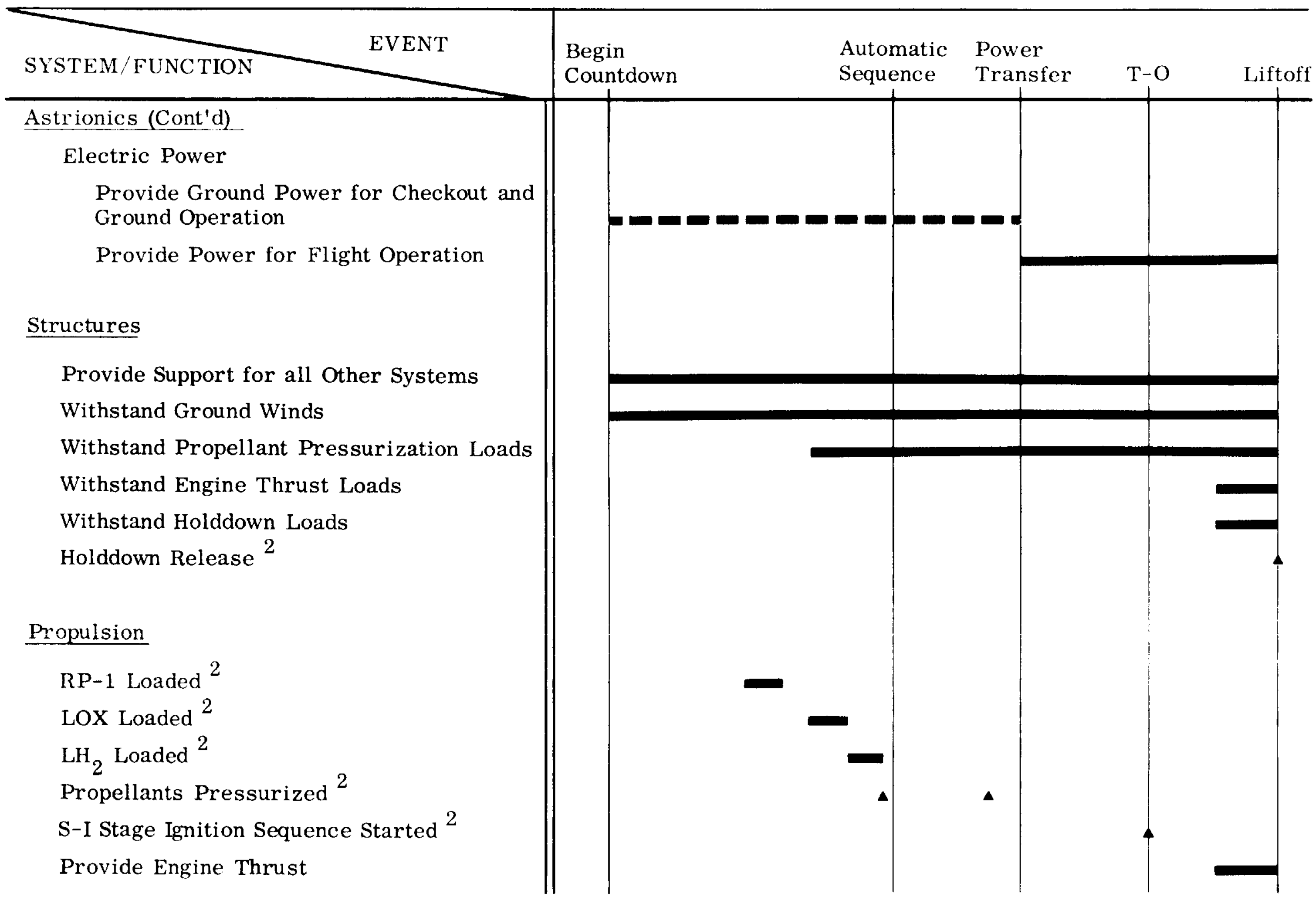


Table 5-5. Saturn I Requirements. Launch Phase (Cont'd)

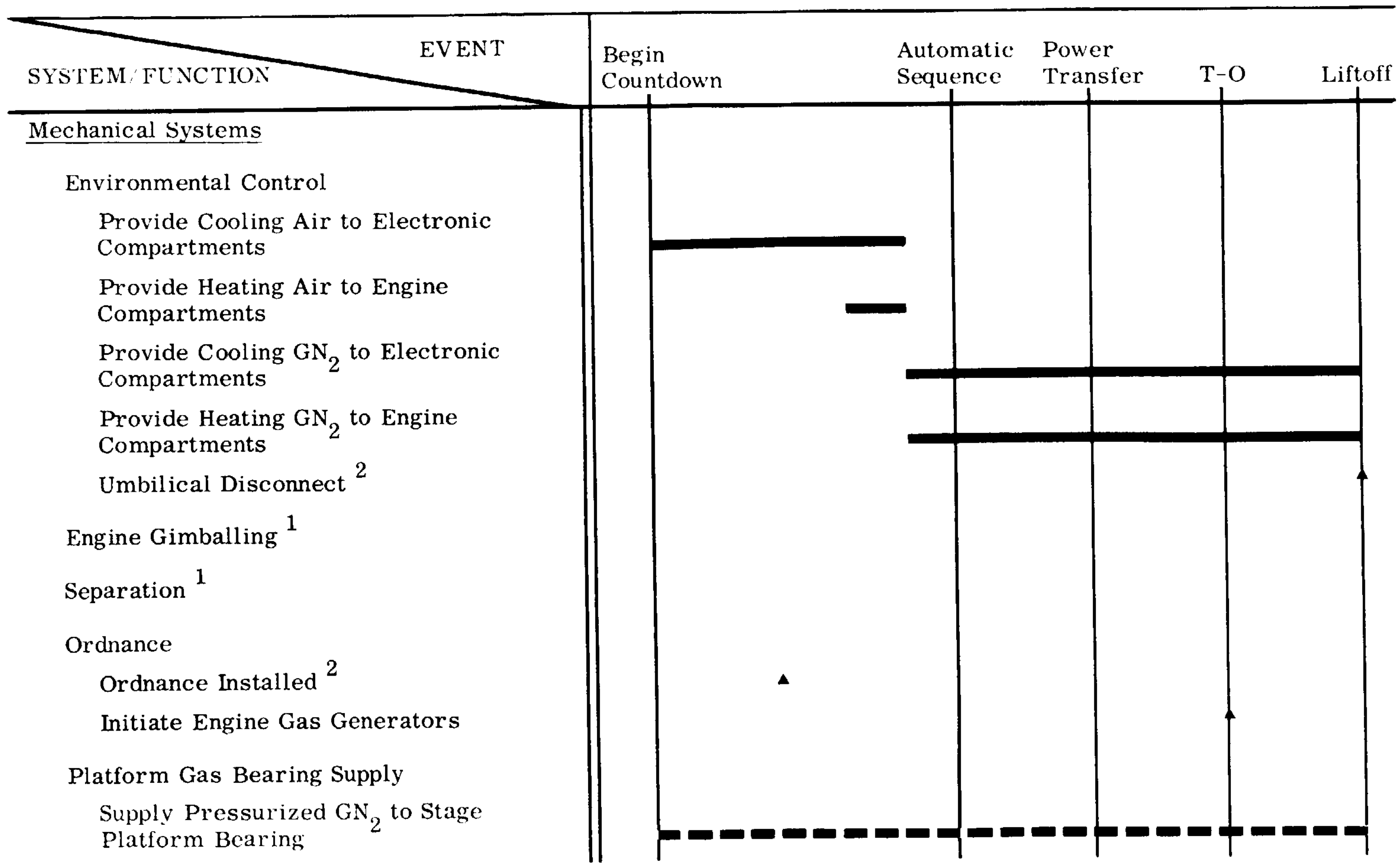


Table 5-5. Saturn I Requirements, Launch Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT	Begin Countdown	Automatic Sequence	Power Transfer	T-O	Liftoff
Ground Support Equipment					
Provide Propellants from Ground Supply -					
RP-1	▬				
LOX	▬				
LH_2	▬				
Provide Propellant Pressurization -					
RP-1	▲				
LOX & LH_2		▲			

Legend: [1] Inactive; [2] Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation

Table 5-6. Saturn I Requirements. Ascent Phase

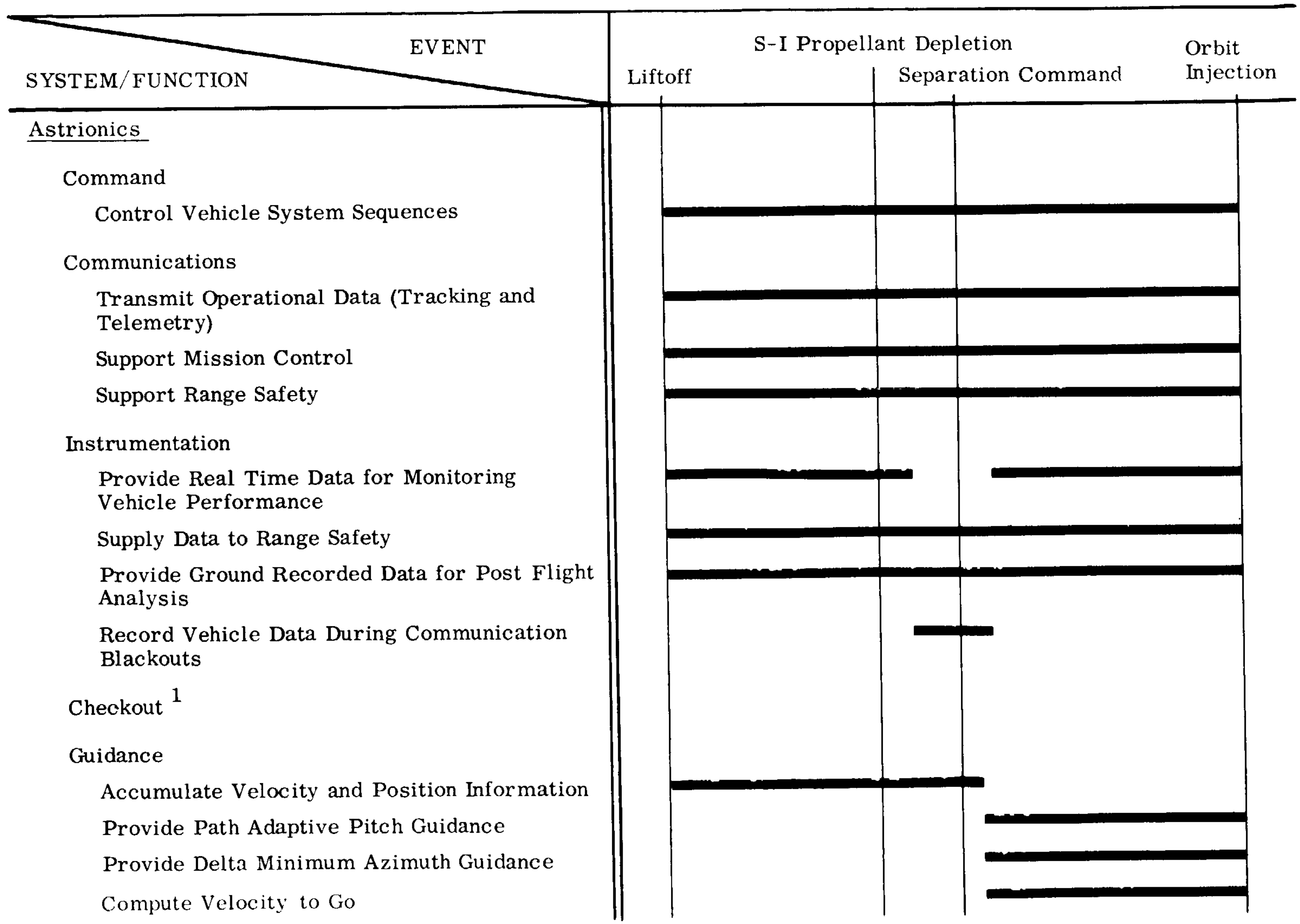


Table 5-6. Saturn I Requirements. Ascent Phase (Cont'd)

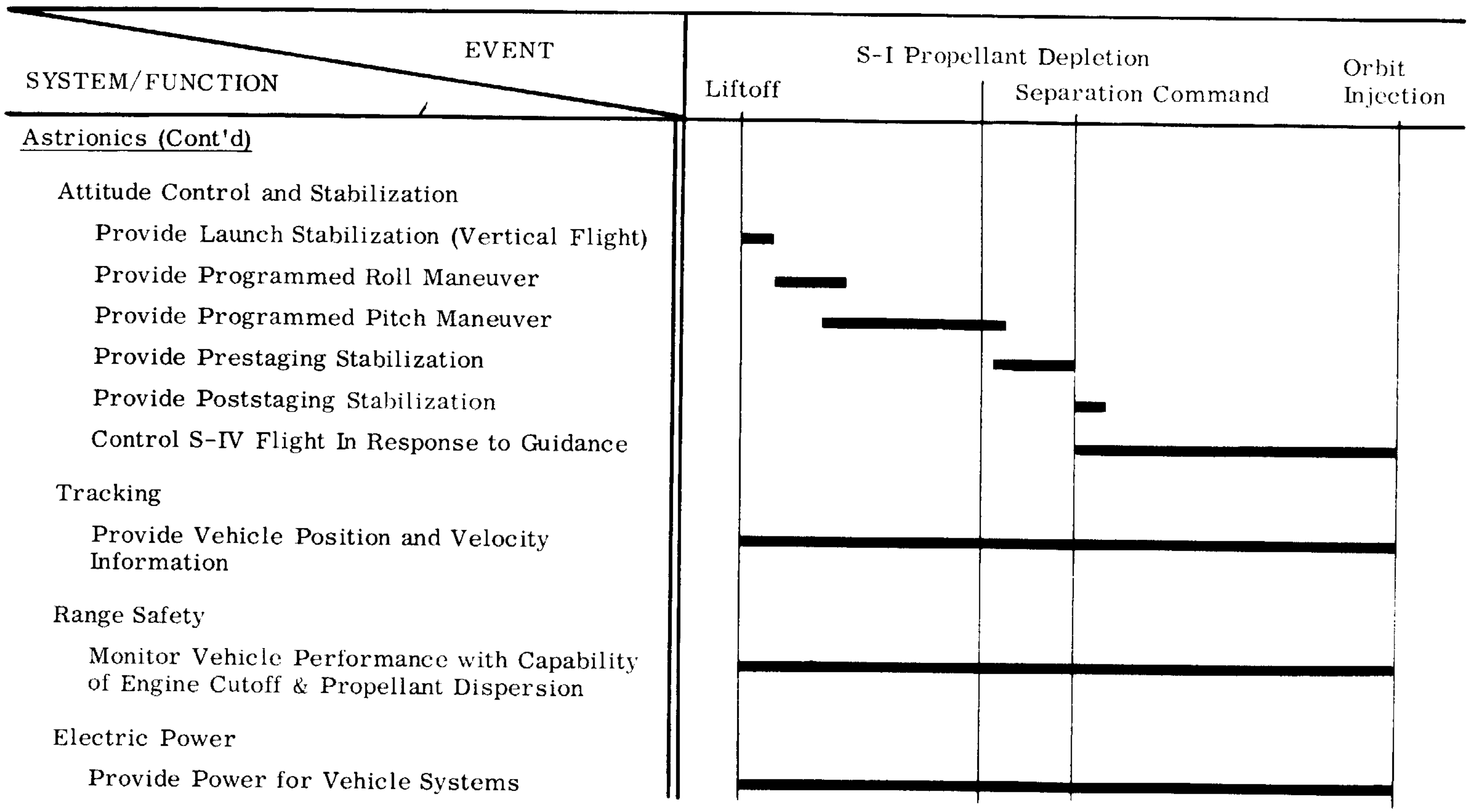


Table 5-6. Saturn I Requirements, Ascent Phase (Cont'd)

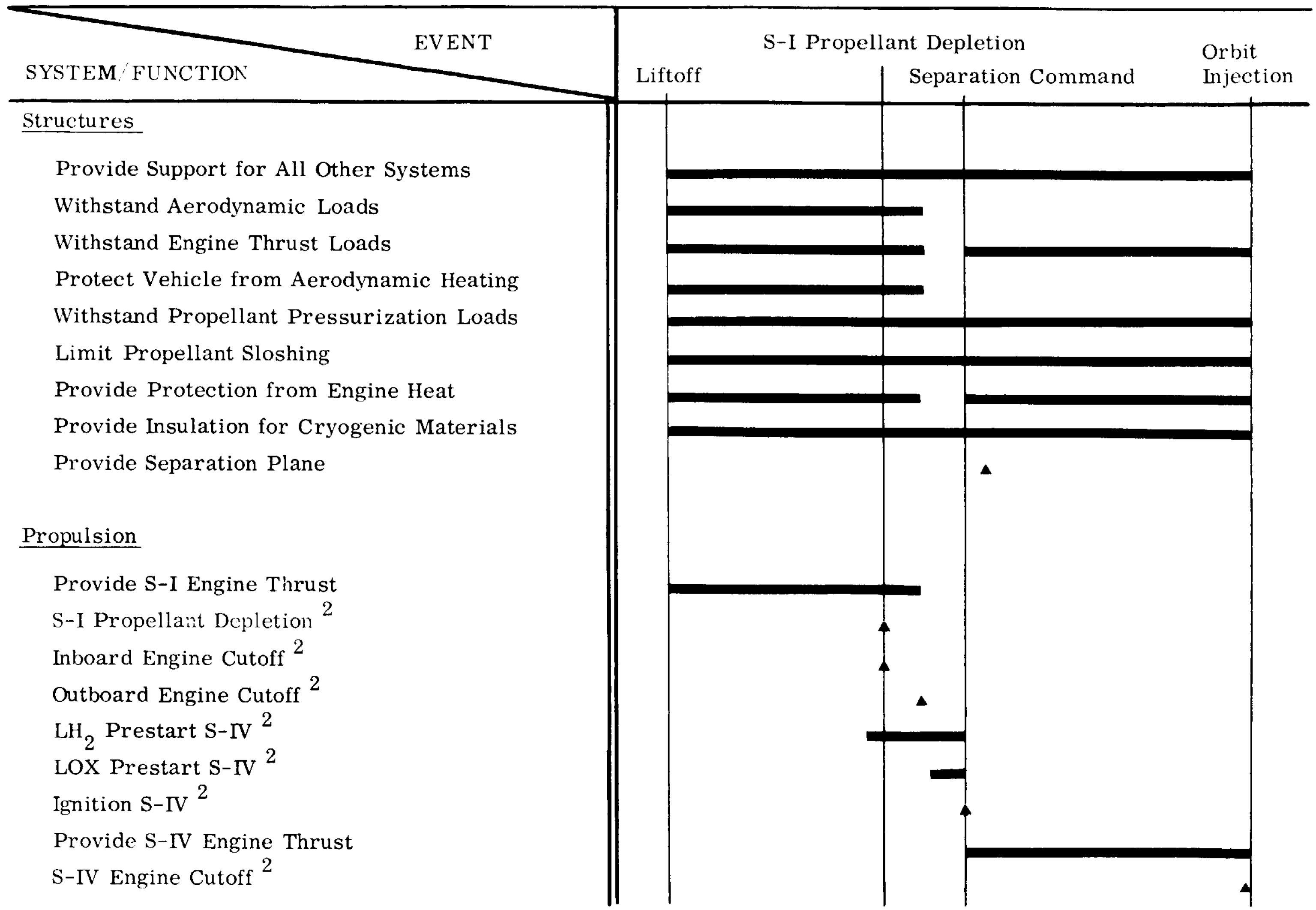
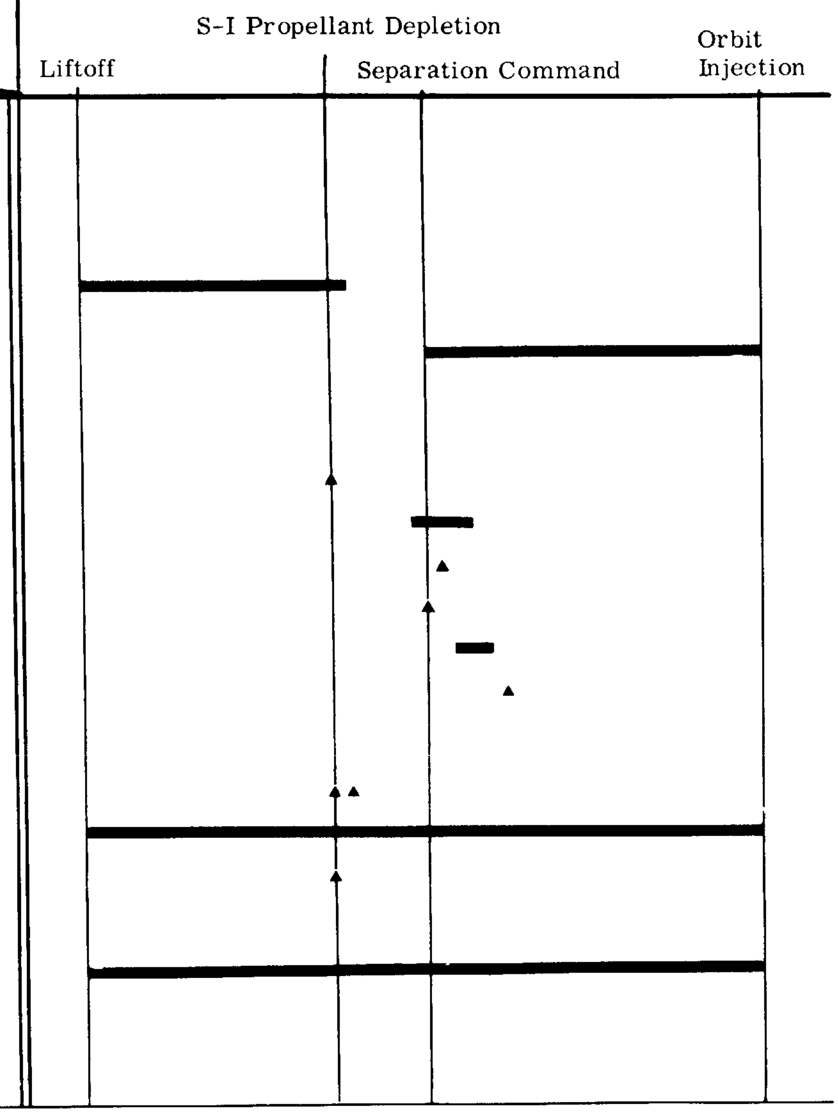


Table 5-6. Saturn I Requirements, Ascent Phase (Cont'd)

SYSTEM/FUNCTION (EVENT →)
Mechanical Systems
Environmental Control [1]
Engine Gimballing
Position S-I Outboard Engines in Response to Control Signals
Position S-IV Engines in Response to Control Signals
Separation System
Initiate LOX/SOX Disposal
Initiate S-IV Ullage Rocket Burn
Actuate Separation Nuts
Transfer Engine Gimbal Control
Initiate S-I Retromotor Burn
Jettison Ullage Rocket
Ordnance
Actuate Conax Valves
Propellant Dispersion Capability Active
Actuate S-IV Blowout Panels
Platform Gas Bearing Supply
Supply Pressurized GN_2 to Stable Platform Bearing
Ground Support Equipment [1]



Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 5-7. Saturn I Requirements, Orbital Phase

SYSTEM/FUNCTION \ EVENT	Orbit Injection	Payload Separation
During the Orbital Phase, all systems [1] are passive with the following exceptions.		
Astrionics		
Command		
Control Vehicle System Sequences		
Communications		
Transmit Operational Data		
Support Mission Control		
Instrumentation		
Provide Data on Vehicle Environment		
Playback Recorded Data		
Tracking		
Determine Orbital Parameters		

Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

CHAPTER 2

SECTION VI

ASTRIONICS

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SECTION VI.

ASTRIONICS

6-1. GENERAL.

The Astrionics system provides the electrical and electronic functions required for Saturn I. The functions, listed below and described in detail in the following paragraphs, are accomplished utilizing both vehicle and ground based subsystems.

a. Command - Performs management of Saturn systems by initiating all operational events and sequences. The issuance of commands is dependent on time and events.

b. Communication - Transfers intelligence within and among the Saturn systems. This intelligence is in four forms: voice, digital, discrete and analog signals.

c. Instrumentation - Monitors the performance of launch vehicle systems to acquire operational and engineering appraisal data.

d. Checkout - Provides assurance during the launch phase that the launch vehicle is capable of performing its assigned mission.

e. Guidance - Provides steering and thrust cutoff commands to adjust the vehicle motion in a manner leading to mission accomplishment.

f. Attitude Control and Stabilization - Provides signals to the engine gimballing system to maintain a stable launch vehicle motion and adjusts this motion in accordance with guidance commands.

g. Tracking - Obtains and records the launch vehicle's position and velocity during flight.

h. Range Safety - Insures that life and private property are not endangered in the event of a vehicle malfunction during the ascent and orbital phase.

i. Electrical System - Supplies and distributes the electrical power required for vehicle operation.

6-2. COMMAND FUNCTION.

The Saturn I command function performs the operational management of the propulsion, astrionics, structures, mechanical and ground support systems. Because of the quantity, priority and degree of decision involved in controlling the vehicle operations, the command function is accomplished in levels. During the mission, the number of levels in the command function and the relative responsibility of each level varies to satisfy the command requirements peculiar to the mission phases.

The launch phase performance of ground support, launch vehicle and payload systems is coordinated to meet a launch time parameter. This performance includes launch vehicle checkout, alignment and physical preparations such as the loading of pressurized gases and propellants.

During the launch phase, the vehicle systems are checked out and aligned. To accomplish this in a reasonable time requires the rapid generation of a large volume of system stimuli. The application of these stimuli excites the systems resulting in the acquisition of performance data which is assimulated and evaluated. If a system malfunction is detected, decisions are made for corrective action. When the operation of each system is validated, the automatic launch sequence is initiated respecting the mission time requirements.

A number of significant commands are necessary during the performance of the automatic launch sequence. The launch vehicle systems are switched from the checkout and alignment modes of operation to the flight modes. Various events and sequences are initiated with the systems performance of the flight mode being evaluated prior to the vehicle being committed to flight. A launch commit command causes the vehicle to be released to begin the ascent phase.

A source of commands is provided the vehicle propulsion, astrionics and mechanical systems during the ascent phase. These commands switch the systems to various modes of operation and initiate events such as staging, engine starting and engine cutoff. This phase of the mission requires the availability of a range safety command to ensure the safety of life and private property. (Refer to Paragraph 6-64.)

6-3. THEORY OF OPERATION

The Saturn command function initiates launch vehicle operational events and sequences from the beginning of the launch phase until termination of the orbital phase with the jettisoning of the S-IV/IU stage. Launch phase command is accomplished with four levels; mission control, launch control manned, launch control computer, and vehicle computer, Figure 6-1.

The mission control level imposes ready-to-launch time requirements on the ground support, payload and launch vehicle. If a mission launch hold becomes necessary for any reason, new ready-to-launch time requirements are imposed by mission control. These time requirements ensure that all systems necessary for mission completion are operating properly at launch time.

The launch control manned level directs operation of the launch complex, payload and launch vehicle systems. This command level can control the launch vehicle by direct issuance of commands or by selecting the mode of operation of the launch control computer level. Mode of operation is defined as a sequence of instructions leading to the accomplishment of a particular systems performance. The launch control manned level, comprised of the test supervisor, test conductor and systems personnel, monitors the launch control computer level and the launch vehicle. System personnel monitor data displayed by consoles arranged in launch vehicle systems oriented groups. These personnel are connected by communications with the test conductor and test supervisor resulting in the coordinated operation of launch vehicle systems. To maintain manned command responsibility, the systems personnel have the capability of issuing commands through the system consoles to the vehicle.

The launch control computer level provides the means for generating a magnitude of commands and permits the assimulation and evaluation of a large amount of performance data in a limited time. This automated command level issues launch vehicle stimuli and performs operational evaluation of performance data within the parameters of the mode of operation selected by the launch control manned level. The computer level filters out high priority data and displays this data to the manned levels. The combination of manned levels of command and a computer level of command permits manned responsibility while providing the large amount of stimuli required by the vehicle systems.

In performing the selected mode of operation, the launch control computer level interacts with the vehicle computer level to control the vehicle systems. These levels issue stimuli to the vehicle systems to accomplish checkout and alignment. These stimuli are in the forms of discrete (on-off) commands and analog signals.

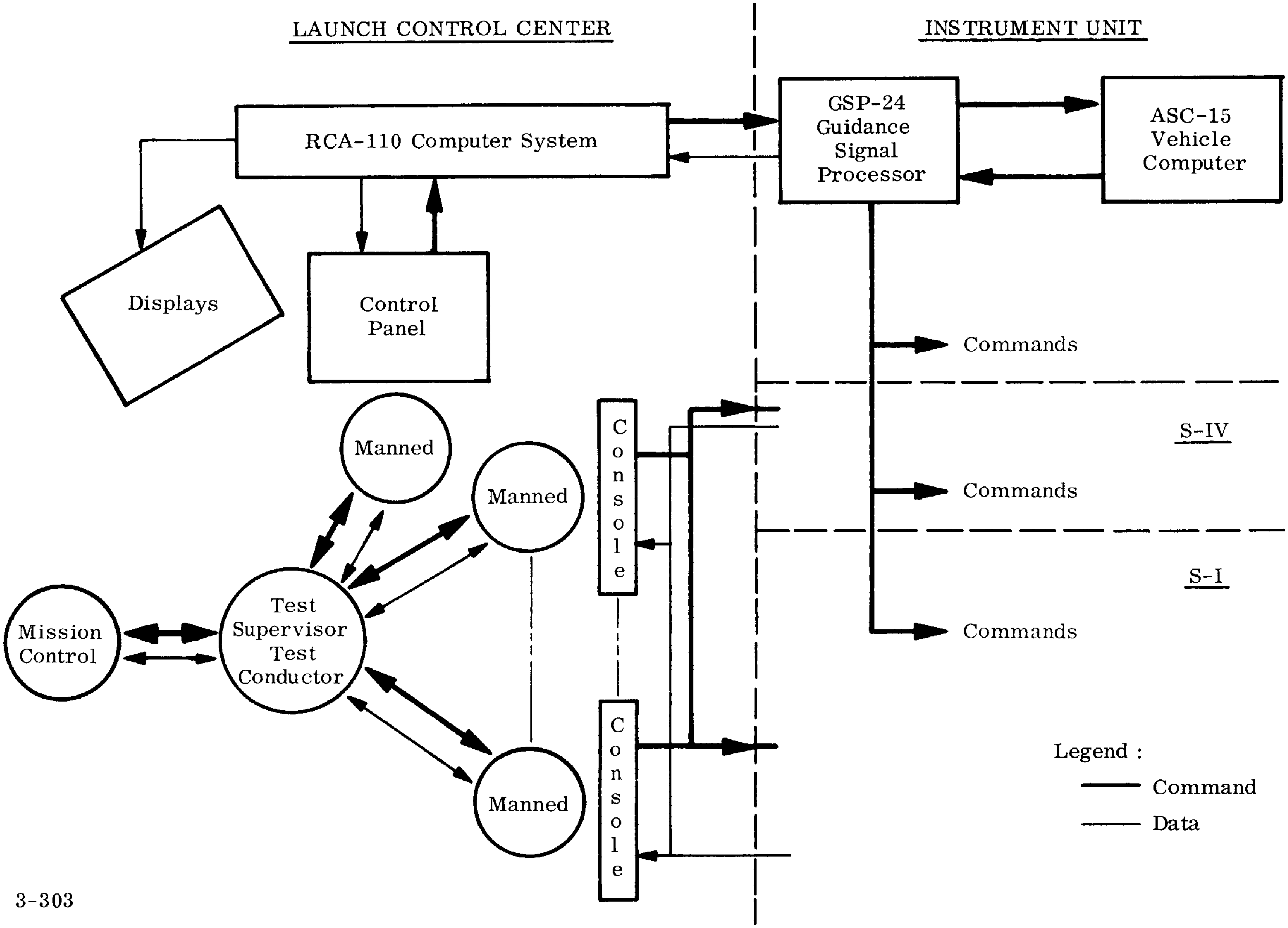

Figure 6-1. Launch Phase Command Configuration, Saturn I

During the ascent phase, launch vehicle operational command is provided by the vehicle computer level, Figure 6-2. This level monitors systems performance and issues systems commands dependent on flight variables and/or time. These commands are in the form of discrete impulses which are applied to the stage flight sequencer.

The flight sequencers are composed of stepping switches which contain a program of vehicle operation. Application of discrete signals to the flight sequencer steps the switches to initiate programmed events such as engine start, engine cutoff, and stage separation.

6-4. IMPLEMENTATION.

During the launch phase, the command function is accomplished with manned levels and hardware systems. The mission control level is manned for command responsibility and is implemented with hardware systems for data acquisition and display. For Saturn I missions, the mission control level is located at the John F. Kennedy Space Center. This level is tied to the various mission elements by a communication network.

The launch control manned level and the launch control computer level of command are located in the block house at the launch site. The launch control manned level performance is accomplished with systems personnel and hardware for data display and command acceptance.

The launch control computer level is implemented with an RCA 110 computer system. During launch operations, the process control capability of the computer is utilized to accomplish systems checkout and alignment. The computer has an automatic priority interrupt which permits control of several vehicle systems because their needs for control inputs can be satisfied on a demand basis. Additional information concerning the RCA 110 computer is presented in Paragraph 20-4.

The vehicle computer level of command is implemented with an ASC-15 digital computer. Information concerning this unit may be found in Paragraph 6-46.

Ascent phase command is accomplished with the vehicle computer and the availability of a range safety command. The vehicle computer is an ASC-15 digital computer. The range safety command is provided by the range safety function. (Refer to Para-

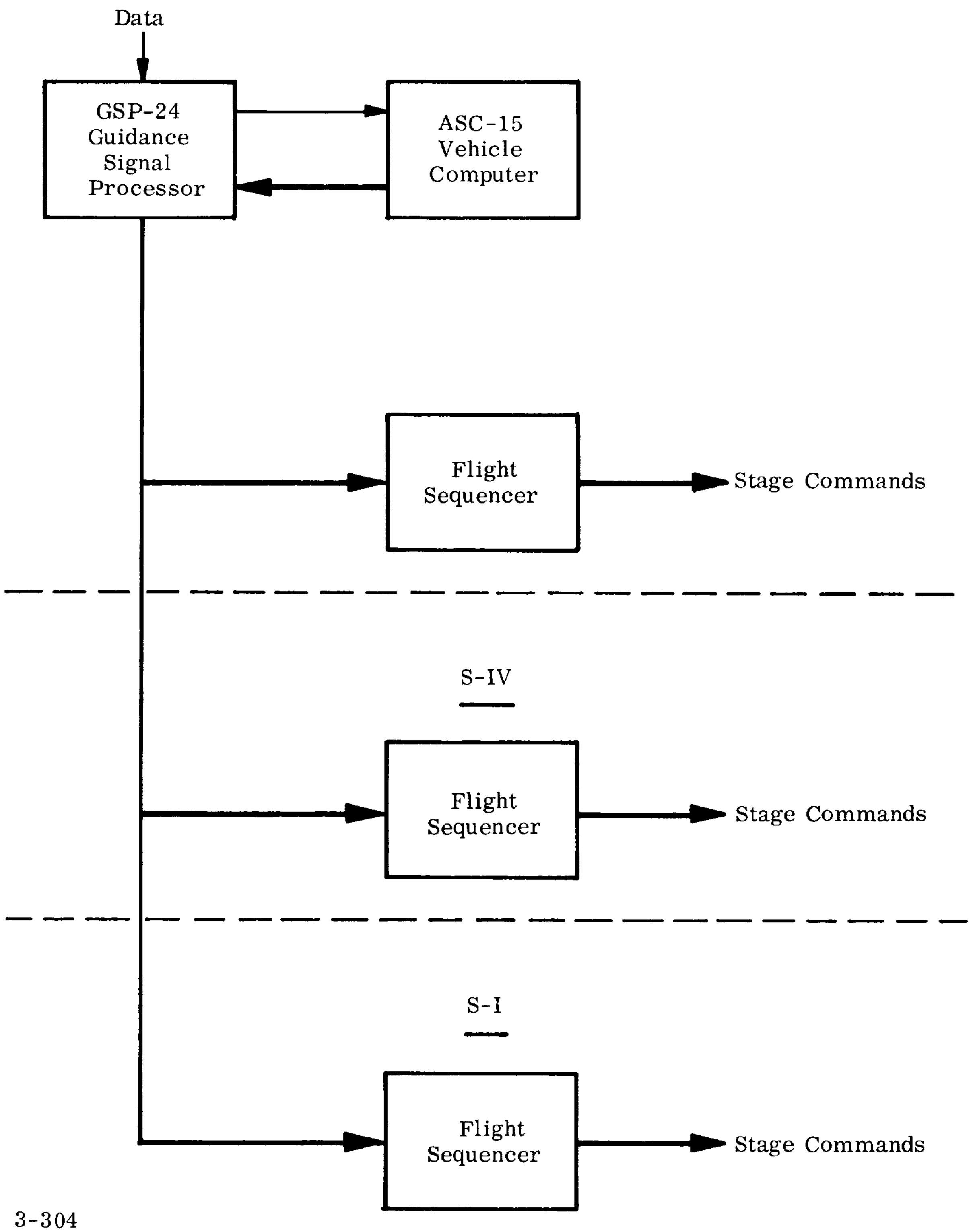

Figure 6-2. Ascent Phase Command Configuration, Saturn I

graph 6-64.)

6-5. COMMUNICATION FUNCTION.

Successful completion of the Saturn I mission is dependent not only on the proper performance of the vehicle but also on the coordination and operation of all its supporting functions. This coordination requires a communication function to provide flow of administrative and operational control information to world-wide stations monitoring the mission, and flow of data from those stations to control locations. Additionally, communication links must exist between earth and the vehicle for operational control during its flight.

The communication function is active throughout all phases of the mission (prelaunch, launch, ascent and orbital). During the prelaunch and launch phases, operational readiness of all supporting functions must be made known to the launch control center, and count-down information supplied to the supporting functions. Operational readiness information includes status of telemetry-reception stations and stations participating in the world-wide tracking network, as well as the integrity of the communications network.

During and after launch, the communication function actively supports the command, tracking, instrumentation and range safety functions. At lift-off, transmission of a zero-time reference ensures the synchronization of mission events at all participating locations. In support of the command function, data must be rapidly delivered to the four command levels for evaluation and decisions. Decisions in turn, must be delivered rapidly from one command level to the next as required.

6-6. OPERATION.

To accomplish the transfer of intelligence, ground-based command levels are interconnected with each other and with all other functions by a network of hard-wire and radio frequency links, including voice, teletype and data transmission channels. The launch vehicle computer is integrated with the network through a command receiver on board the vehicle which is linked to rf transmitters at the command transmitter sites on earth.

The communication function supports instrumentation and tracking functions through transmission of data from telemetry receiving and tracking stations to locations where the data is recorded, used in real-time computation, and disseminated further (e.g., for command function inputs, range safety information, etc.). Tracking information from each station is transmitted to Goddard Space Flight Center for trajectory computation. Predicted positions and times are then transmitted from Goddard to each of the tracking stations in order, to enable acquisition of the vehicle by their narrow beamwidth (high gain) antennas as it comes into range.

The range safety function is dependent on the communication function for tracking and telemetry data. This data is delivered through the communications network for display to the range safety officer. Other stations monitoring performance of the launch vehicle during its ascent are tied in by telephone with the range safety officer. In the event of flight termination for range safety purposes, the communication function provides transmission of the termination signal from the range safety officer's control box to the command transmitter, and from there to the vehicle.

6-7. IMPLEMENTATION.

The Saturn I communication function is implemented with both vehicle and earth-based systems. The major systems are described in the following paragraphs.

6-8. Earth-Vehicle Communications. For the Saturn I, communications between earth and vehicle consist of radio frequency systems involved in tracking, instrumentation (telemetry) and range safety functions. These systems are described in the sections covering those functions.

A digital command receiver and decoder system will be flown as passenger or developmental equipment on the vehicles SA 8, 9, and 10. This will permit communication of additional commands, such as trajectory corrections, and operation of on-board functions from the command transmitters on earth.

6-9. Point-to-Point Communications (Earth). Stations interconnected in the communications network are listed in Table 6-1. Indicated in the table are types of communications facilities existing at each station for transmission of information.

The facilities existing on the Atlantic Missile Range (AMR) for point-to-point communications are typical of the communications implementation for Saturn I. A wide variety of equipment is used including submarine cable, high frequency radio, troposcatter, microwave and wire.

Submarine Cable. The AMR submarine cable, Figure 6-3, extends from Cape Kennedy, Florida to Grand Turk Island, with communications circuits available at the Cape and Point Jupiter, Florida, at Grand Bahama Island, Eleuthera Island, San Salvador, Mayeguana and Grand Turk Island. A single coaxial cable links all stations.

The band width of the submarine cable is 150 kc: it accommodates twelve duplex telephone circuits of 250-3100 cycles and a band of 10,515 kc for transmission of telemetry data up-range to Cape Kennedy. When telemetry transmission is required, three channels of telephone circuits up range are disconnected.

High Frequency Radio. High-frequency, single-sideband radio is used for long range communications over water. These radio systems interconnect Cape Kennedy, Antigua Island, Ascension Island, and Pretoria, South Africa. Each link can accomodate voice, teletype or high bit-rate data. The associated transmitters operate in the 2 to 30 mc range with an output power of 45 KW. Cape Kennedy has three transmitters of this type, Antigua has four, Ascension five, and Pretoria two.

A low-power (2.5 KW) high-frequency, single-sideband transmitter provides communications from Trinidad to Cape Kennedy.

Troposcatter. A quadruple-diversity tropospheric scatter system, AN/MRC-85, exists between Grand Turk Island and East Island, Puerto Rico. Phase-locked multiplex equipment provides communications of twenty-three 3-kc voice channels, 16 full-duplex teletype channels and three 48-kc wideband channels. The rf equipment operates in the 1000 mc spectrum at 10 kw.

Microwave. Three microwave links are used at AMR: one for operation and data transmission for the MISTRAM at Valkaria, Florida; one for inter-island communications in the area around Grand Bahama Island; and one for tying the

Table 6-1. Communications Stations

Location	Tracking	Telemetry	Capsule Communications	Command Control	Voice	TTY	SSB
Cape Kennedy	X	X	X	X	X	X	
Grand Bahama	X	X	X	*	X	X	
Grand Turk	X	X	X	*	X	X	
Bermuda	X	X	X	X	X	X	
Atlantic Ocean Ship	X	X	X			X	X
Grand Canary Island	X	X	X			X	X
Kano, Nigeria	X	X	X			X	X
Zanzibar	X	X	X			X	X
Indian Ocean Ship	X	X	X			X	X
Muchea, Australia	X	X	X	X	X	X	X
Woomera, Australia	X	X	X		X	X	
Canton Island	X	X	X			X	
Kauai Island, Hawaii	X	X	X	X	X	X	
Point Arguello, California	X	X	X	X	X	X	
Guaymas, Mexico	X	X	X	X	X	X	
White Sands, New Mexico	X				X	X	
Corpus Christi, Texas	X	X	X		X	X	
Eglin, Florida	X				X	X	
Goddard Space Flight Center					X	X	
Eleuthera Island	X	X					
San Salvador	X	X		*	X	X	X

Table 6-1. Communications Stations (Cont'd)

Location	Tracking	Telemetry	Capsule Communications	Command Control	Voice	TTY	SSB
Antigua	X	X			X	X	X
Ascension	X	X			X	X	X
Pretoria	X	X			X	X	X

*Remote Command Transmitters for Cape Kennedy Command Control

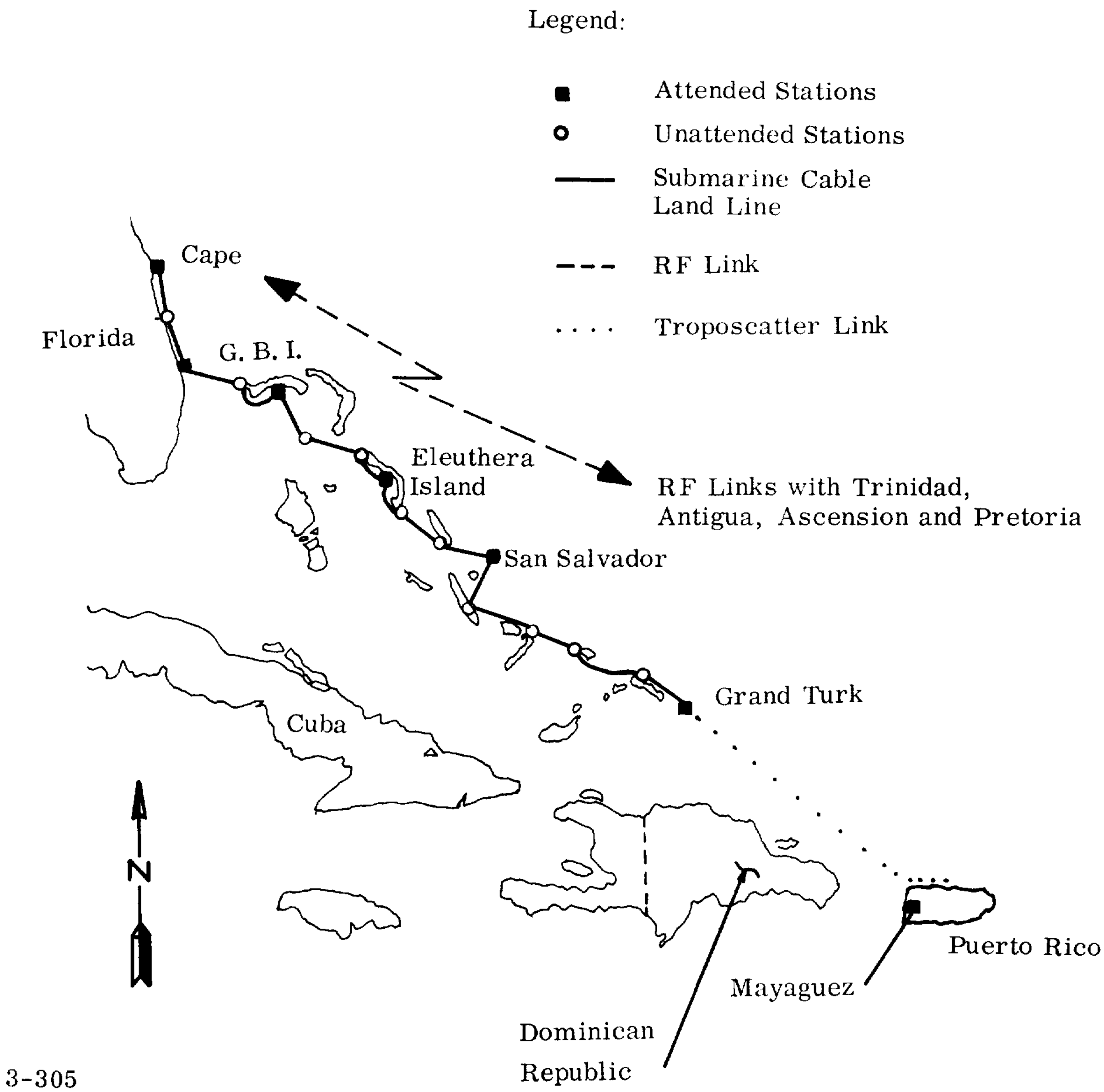

Figure 6-3. AMR Submarine Cable

range communications into Ramey AFB at Puerto Rico.

The MISTRAM system extends from Valkaria, Florida, to Cape Kennedy, with a relay station at Patrick AFB. Two carrier frequencies are assigned on each leg. Vertical polarization is used in both directions on the Valkaria - Patrick AFB links, and horizontal polarization on the Patrick AFB - Cape Kennedy links.

The Cape Kennedy to Valkaria link uses the following:

a. Order wire: 0 to 4 kc

b. Multiplex carrier: 60 kc

c. Timing carrier: 100 kc

d. 24 voice channels: 312 to 552 kc

e. Timing synchronization: 1.2, 1.4 and 1.6 mc bursts

The Valkaria to Cape Kennedy link does not use the 1.2- and 1.4-mc timing synchronization bursts and the 100-kc timing signal; the 4 to 16 kc band is used for data transmission.

The system at Grand Bahama Island uses frequency diversity with two carrier frequencies on each leg.

The Puerto Rico system, which is operated by Ramey AFB, supplies 5 channels to the AMR between Ramey AFB and Fort Buchanan. A cable then extends these channels to East Island.

6-10. Air-Ground/Ship-Shore Communications. Communications to ships and aircraft are available at the major communications stations using HF-SSB, VHF and UFH. The transmitters and receivers used are listed by type number, and location in Table 6-2.

Table 6-2. Communication Transmitters and Receivers

Location	HF/SSB		VHF		UHF	
	No.	PEP (kw)	No.	Power (w)	No.	Power (w)
Cape Kennedy	4	10	8	50	12	50
	3	2.5				
Grand Bahama Island	2	2.5	4	50	6	50
San Salvador Island	2	2.5	4	50	6	50
Grand Turk Island	2	2.5	4	50	6	50
Antigua Island	1	10	4	50	6	50
	2	2.5				

Table 6-2. Communication Transmitters and Receivers (Cont'd)

Location	HF/SSB		VHF		UHF	
	No.	PEP (kw)	No.	Power (w)	No.	Power (w)
Ascension Island	2	10	2	50	4	
	2	2.5				
Pretoria	2	45				

For ship-to-shore communication, the entire range is separated into three areas for communication control, including assignment of frequencies, status maintenance, and distribution of range test information. The three control points are Cape Kennedy, Antigua and Ascension.

6-11. INSTRUMENTATION.

Saturn I instrumentation collects vehicle status and operational data, and transmits, records or displays this information in accordance with specific requirements in each phase of launch vehicle operation. This data is made available as required for display in real time to other functions in the Saturn system, to aid them in carrying out their role in the mission.

Instrumentation is initially activated during checkout in the prelaunch phase and remains active until end of mission for the launch vehicle. The many tasks assigned to instrumentation can be grouped in three major areas; checkout support, in-flight data collection, and data recording for post-flight analysis.

During the prelaunch phase, instrumentation forms a highly important data link in the checkout of the vehicle. The checkout can be performed either manually or automatically (digital computer controlled). Instrumentation is designed to be compatible with the checkout system, and as such is capable of presenting all major data channels in digital format.

From liftoff, when all physical connections between the vehicle and ground are severed, until the end of the mission, instrumentation provides the vehicle-to-ground data link. Since this is the only means by which operational information can be obtained from the vehicle, a highly reliable telemetry system is required. The primary Saturn I telemetry system is the pulse amplitude modulation/frequency modulation/frequency multiplexing system which has been proven very reliable in previous launch vehicle programs.

Vehicle performance data falls into two categories; engineering data and operational data. Engineering data includes parameters such as temperature, acceleration, vibration, and stress; operational data includes vehicle computer commands and event sequences such as those associated with first stage cutoff, stage separation or second stage ignition. Examples of instrumentation measurements acquired during a mission are listed in Table 6-3.

Table 6-3. Typical Instrumentation Measurements

Measurement	S-I	S-IV	Instrument Unit
Propulsion	26	12	
Temperature	119	94	22
Pressure	95	118	9
Strain and Vibration	52	32	26
Flight Mechanics	13	11	17
Steering Control	4		40
RF and Telemetry	1		14
Discrete Signals	38	3	22
Voltage, Current and Frequency	7	16	32
Miscellaneous	1	11	1

6-12. OPERATION.

Saturn I instrumentation is comprised of ground instrumentation stations and vehicle instrumentation systems. The ground instrumentation stations form a global network, essentially the old Mercury network, which is being expanded to meet the requirements for the Apollo program. A discussion of the ground stations is presented in Paragraph 6-63. Tracking, and a discussion of the transfer of data from the ground stations to the Mission Control Center is contained in Paragraph 6-51, Tracking. Saturn I instrumentation is stage oriented. The two stages and the instrument unit each contain separate, independent instrumentation which is comprised of the following systems (Figure 6-4).

a. Measuring

b. Telemetry

c. Antenna

On some missions these systems are augmented with a recording system.

The systems are described in the following paragraphs, and where there is a difference in the implementation between stages this difference is noted.

6-13. Measuring System. The measuring system, Figure 6-5, is composed of transducers, signal conditioners, and a measuring distributor. It senses vehicle operational parameters and transforms this information to signals compatible with the telemetry subsystem requirements.

Transducers fall into two main groups depending on their output characteristics. The outputs of the first group of transducers consist of signals in the 0 to 5-volt dc range and excite the telemeters directly without any form of signal conditioning. Into this group fall pressure transducers, ON/OFF indicators and position indicators. These transducers are excited from the measuring voltage supplies. The second group contains transducers which require modification or amplification of their outputs. Examples of these transducers are thermocouples, strain gauges and vibration pick-ups. These outputs consist of a millivoltage or resistance changes which must be modified and amplified before being applied to the telemeters.

Transducers are calibrated prior to installation in the vehicle and the calibration data is recorded on IBM cards. A card system has been established to facilitate data processing of these cards and support the automatic checkout system for the Saturn vehicle. The IBM card is initiated in the Astrionics Laboratory and delivered to the Quality Laboratory with its associated component where it is utilized in an automatic checkout system for component test. Upon completion of tests a duplicate card is sent to the computer lab where it is used for data reduction of static test data. A copy of the card will follow the vehicle to the launching site.

The signal conditioners convert the transducer outputs into signals compatible with the telemetry subsystem. The conditioners are plug-in modules of standard config-

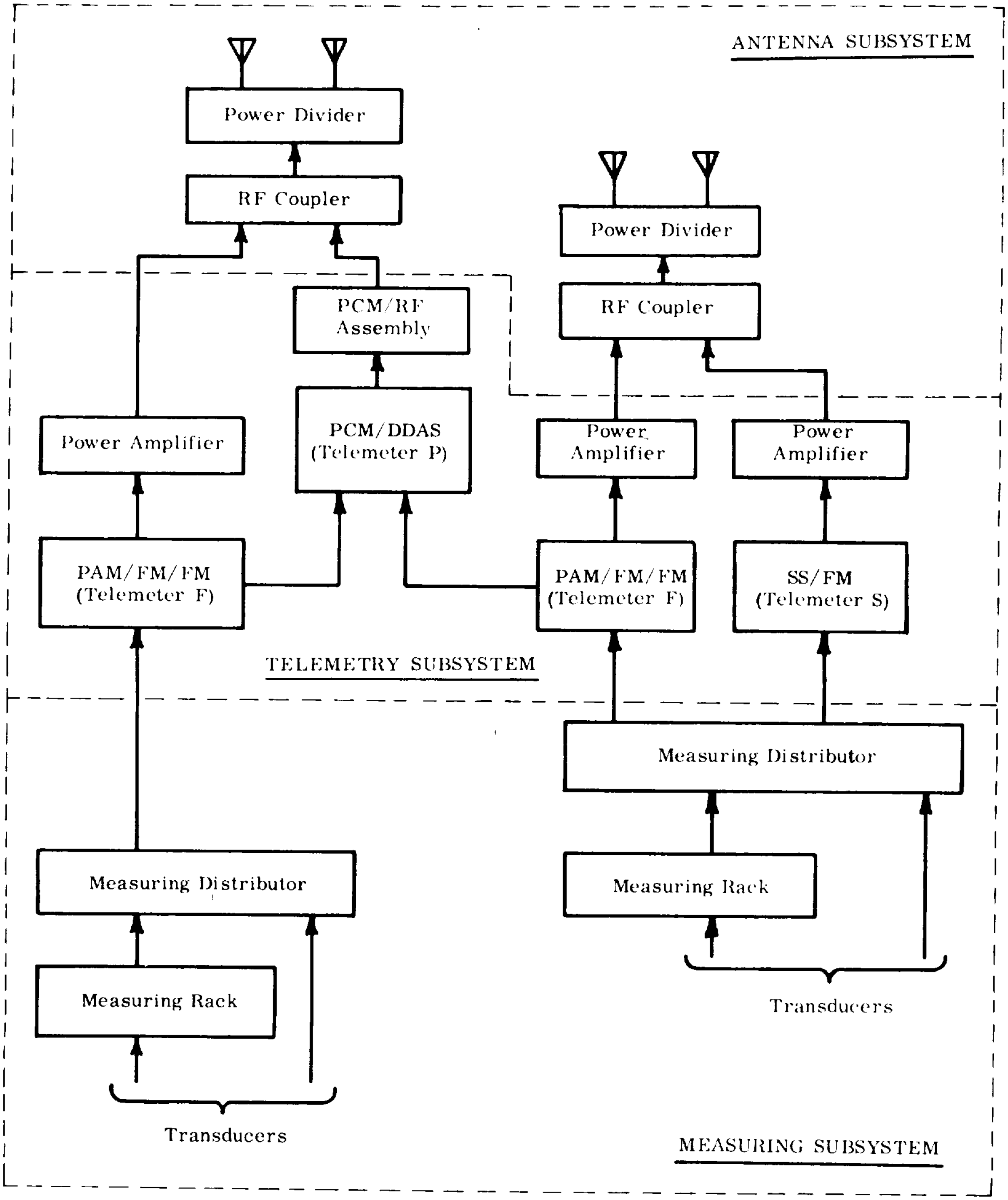

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Figure 6-4. Typical Stage Instrumentation System

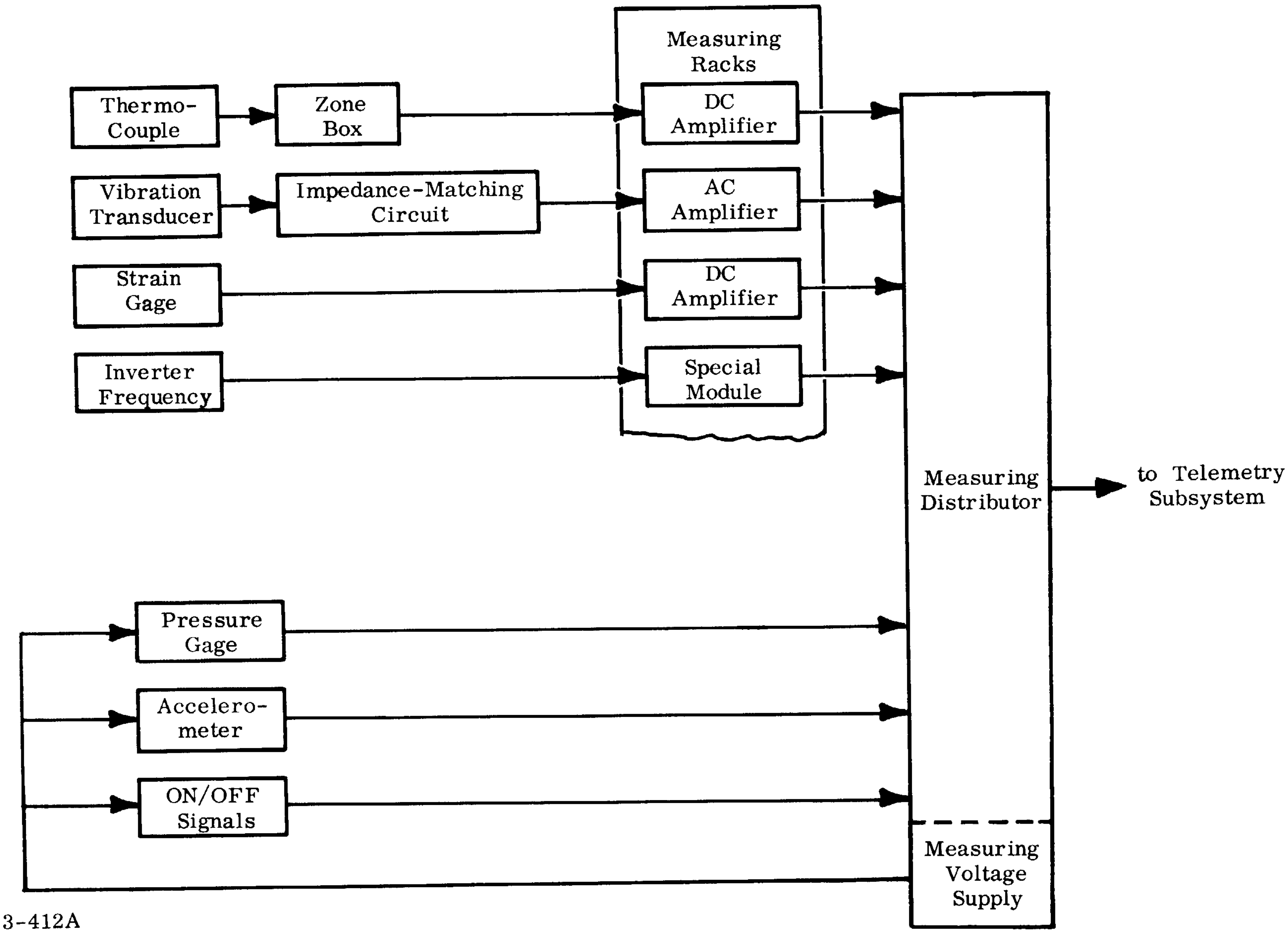


Figure 6-5. Measuring Subsystem

uration and are adapted to specific applications by plug-in range cards.

A typical module is provided with both local and remote calibration control. During calibration a simulated transducer signal is placed on the input terminals of the module (instead of the transducer) in steps of 0, 20, 40, 60 and 100 percent of full scale value. The output of the module is read out through the telemeters.

There are four standard modules used in addition to the regulated power supply. These are, ac amplifiers, carrier amplifiers, dc narrow-band amplifiers and dc wide-band amplifiers.

The ac amplifier, used to amplify the signals sensed by vibration transducers, has a bandwidth of 10 to 3000 hz (cps). The output of the amplifier is a linear 0 to 6 volt peak-to-peak voltage biased at 2.5 volts. A zener diode limits the output, preventing cross-talk or other interference which might result from overdriving the telemetry subcarrier oscillators.

Signals from control accelerometers and servos are amplified by the carrier amplifier, which has an output level within the range of 0 to 5 volts. A zener diode similar to that in the ac amplifier limits the output.

The narrow-band dc amplifier accommodates low-level signals (in the millivolt range) derived from thermocouples, resistance thermometers, thermistor bridges, and similar transducers. The module contains a 10-volt dc regulated power supply used for energizing a thermistor or strain gage bridge when required. The amplifier has a nominal gain of 1000, and is adapted to a specific transducer by a signal conditioning plug-in module.

A wide-band dc amplifier is used in applications requiring amplification of slowly varying dc signals, such as those emanating from strain gages. Power for the associated signal conditioners is supplied by a dc power supply which is an integral part of the amplifier module. The amplifier has a nominal gain of 1000.

The signal conditioner modules are assembled into measuring racks, each rack being able to accommodate approximately 20 modules. Each measuring rack is provided with a regulated power supply which provides power for all the modules in it.

The measuring distributor is the central distribution point for all signals in the measuring subsystem. The collection of all distribution functions in one component has the advantage that if changes to the instrumentation program are required only this component need be altered.

6-14. Telemetry System. The telemetry system receives data signals from the measuring system and encodes these signals on an rf carrier frequency for transmission to the ground instrumentation systems. In order to fulfill the requirements of the wide range of measurements in the Saturn launch vehicles it has been necessary to employ four different types of telemetry. These are:

a. Pulse amplitude modulation/frequency modulation/frequency multiplexing (PAM/FM/FM)

b. Pulse duration modulation/frequency modulation/frequency multiplexing (PDM/FM/FM)

c. Pulse coded modulation/frequency modulation (PCM/FM)

d. Single sideband/frequency modulation (SS/FM)

Two types of multiplexing are used: frequency-division and time-division.

In frequency-division multiplexing (FM), each data channel is allocated a separate subcarrier frequency. Several subcarrier frequencies are then combined into a composite signal which modulates the rf carrier frequency. Subcarriers can be further subdivided by the same method. However, the increase in channel capacity results in a decrease in frequency response of the multiplexed data. The data content is conveyed by modulation of the subcarrier frequency which is generated by a voltage-controlled subcarrier oscillator (SCO). The SCO output frequency will vary from minus to plus 7.5 percent deviation about nominal for input voltage variations from 0 to 5 volts dc. The SCO can be zero offset 2.5 volts, resulting in an input range from -2.5 to +2.5 volts dc. Frequency modulation/frequency division multiplexing is denoted by FM/FM. A subcarrier frequency that is further frequency-division submultiplexed is denoted by FM/FM/FM or FM^3.

In time-division multiplexing, each data channel is sampled in a fixed sequence. The information on a channel is represented by a series of discrete samples of the original signal. To obtain adequate transmission of a signal, the sampling rate must be at least several times the signal frequency. This multiplexing method has been utilized in the Model 270 and in the vibration multiplexer which will be described later.

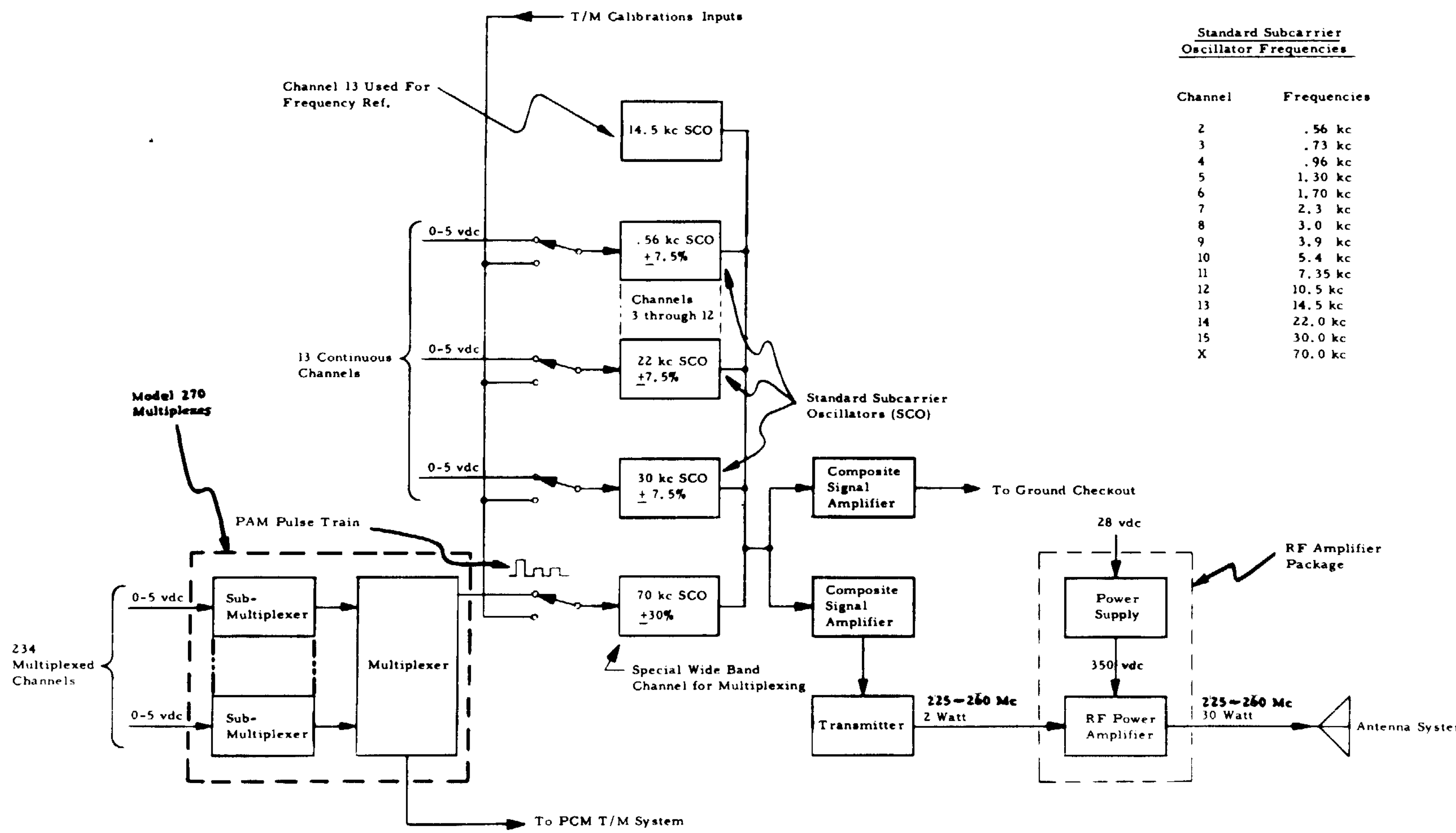
PAM/FM/FM Telemetry. The PAM/FM/FM telemetry system, Figure 6-6, is contained in two packages, a telemetry package and an RF amplifier. The telemetry package consists of a Model 270 multiplexer, several subcarrier oscillators, a composite signal amplifier, and an RF transmitter.

The Model 270 multiplexer, which utilizes time-division multiplexing, consists of 30 primary channels of which 27 are for data and three for frame identification. Each primary channel may be further subcommutated by 10, resulting in a total capability of 270 data channels. Twenty-three submultiplexers are located in the Model 270 package, providing a total package capability of 230 subcommutated and four primary multiplexed channels. The multiplexer is capable of controlling four remote submultiplexers, the outputs of which are applied to the remaining four primary channels.

The package contains 13 voltage-controlled oscillators (subcarrier oscillators) operating on the standard subcarrier oscillator frequencies. (The 400 cps subcarrier frequency is not used in the Saturn program as it is too susceptible to noise from the 400 cps power supplies in the vehicle.) The continuous channel oscillators use ± 7.5 percent deviation for input signals of 0 to 5 volts dc. The multiplexed oscillator is deviated ± 30 percent for inputs of 0 to 5 volts dc.

The composite signal amplifier mixes the outputs of the subcarrier oscillators and amplifies the signal to a level usable by the rf transmitter. In the rf transmitter, the composite subcarrier signal frequency modulates a radio frequency carrier in the 225 to 260 mc range. The nominal output of two watts is applied to an RF amplifier, which boosts the signal to thirty watts for transmission.

PDM/FM/FM Telemetry. The PDM/FM/FM telemetry system is used only on the second stage of Saturn I and IB. In pulse duration modulation, each sample in the output of the multiplexer generates a pulse duration proportional to the numerical value of the sample. The pulses, of varying duration, are then used to modulate the carrier.



Standard Subcarrier Oscillator Frequencies

Channel	Frequencies
2	.56 kc
3	.73 kc
4	.96 kc
5	1.30 kc
6	1.70 kc
7	2.3 kc
8	3.0 kc
9	3.9 kc
10	5.4 kc
11	7.35 kc
12	10.5 kc
13	14.5 kc
14	22.0 kc
15	30.0 kc
X	70.0 kc

3-400

Figure 6-6. Typical PAM/FM/FM Telemetry Link

The PDM/FM/FM system, Figure 6-7, uses two multiplexers. A 90 x 10 (90 samples 10 times a second) accommodates high-level data in the 0 to 5 volt dc range, and a 45 x 2.5 multiplexer accommodates low-level data. The output from the multiplexer is amplified to the 0 to 5 volt dc level before being applied to the subcarrier oscillator.

PCM/FM Telemetry. The PCM telemetry utilizes time-division multiplexing. Each channel is sampled in a fixed sequence and the information on the channel is converted into a binary coded digital word. The words are then converted into a serial pulse train which, in turn, modulates the rf carrier.

The PCM/FM/FM telemetry system, Figure 6-8, provides a transmission mode which can be adapted both to in-flight data transmission and to ground checkout with computer controlled checkout equipment. It consists of two assemblies, a PCM/DDAS assembly and a PCM/radio frequency amplifier.

The PCM/DDAS assembly (Pulse Coded Modulation/Digital Data Acquisition System) accepts data from standard Model 270 multiplexers and converts it into a digital coded pulse train. The assembly has three outputs; one output is used to modulate the PCM/RF assembly, a second output modulates a 600 kc carrier with the digital pulse train for input to the ground DDAS equipment and on the third output the digital information is presented in parallel form. The assembly is capable of receiving data from a total of six Model 270 multiplexers by commands to the PAM scanner. The design makes it possible to read out any Model 270 multiplexer through the PCM/DDAS assembly during the checkout phase.

The PCM/RF assembly accepts a digital-coded pulse train and transmits it to the ground station on a carrier in the 225 to 260 mc range with an output power of 15 watts. The assembly is powered by 28-volt dc. High voltage is supplied from a power supply contained in the assembly.

SS/FM Telemetry. The SS/FM telemetry system, Figure 6-9, is used for the transmission of wide-band data such as vibration measurements. The basic unit can transmit 15 continuous channels. A vibration multiplexer provides the capability of time-sharing each channel between two or four data

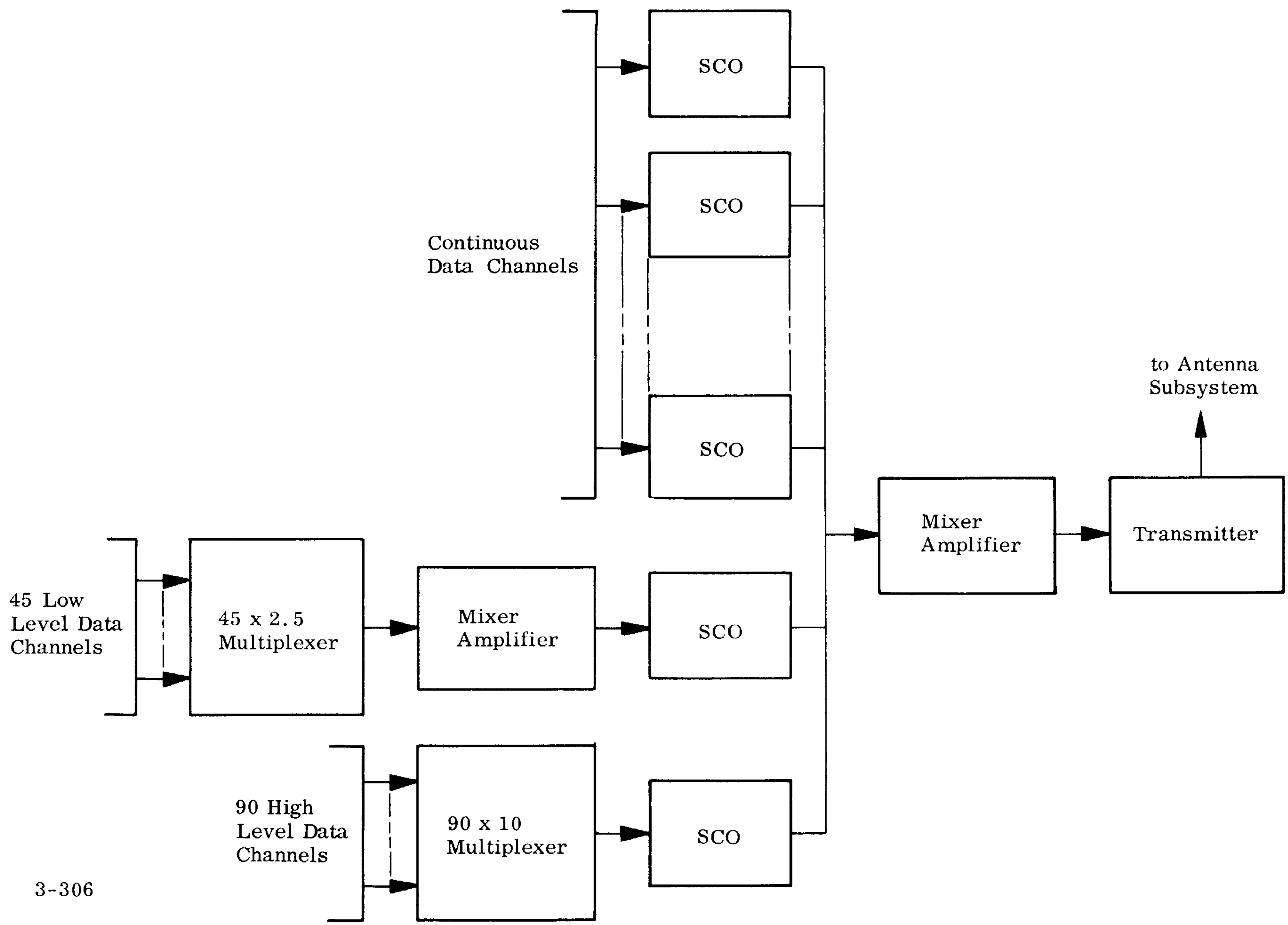


Figure 6-7. PDM/FM Telemetry Link

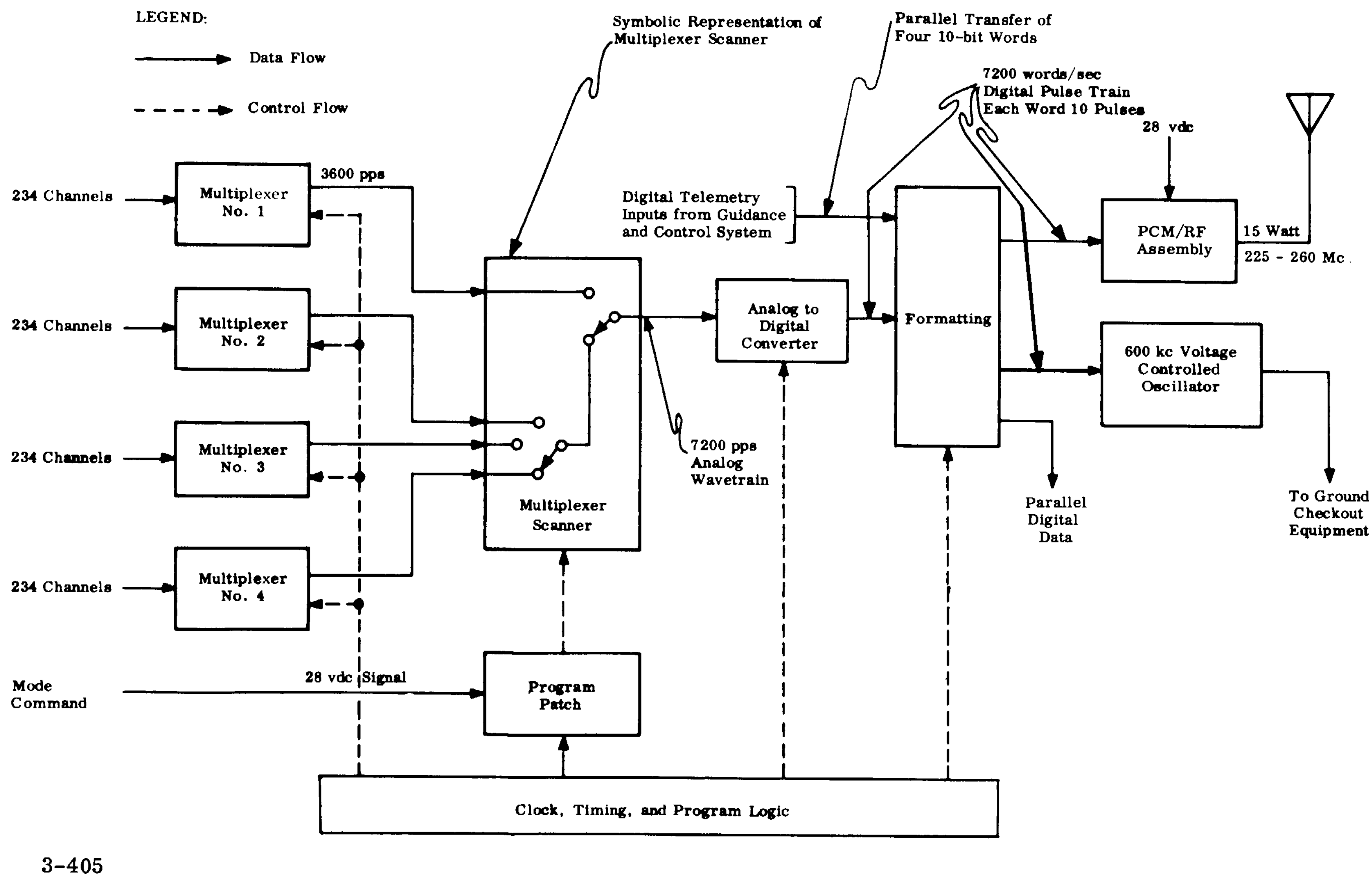


Figure 6-8. PCM/FM/FM Telemetry Link

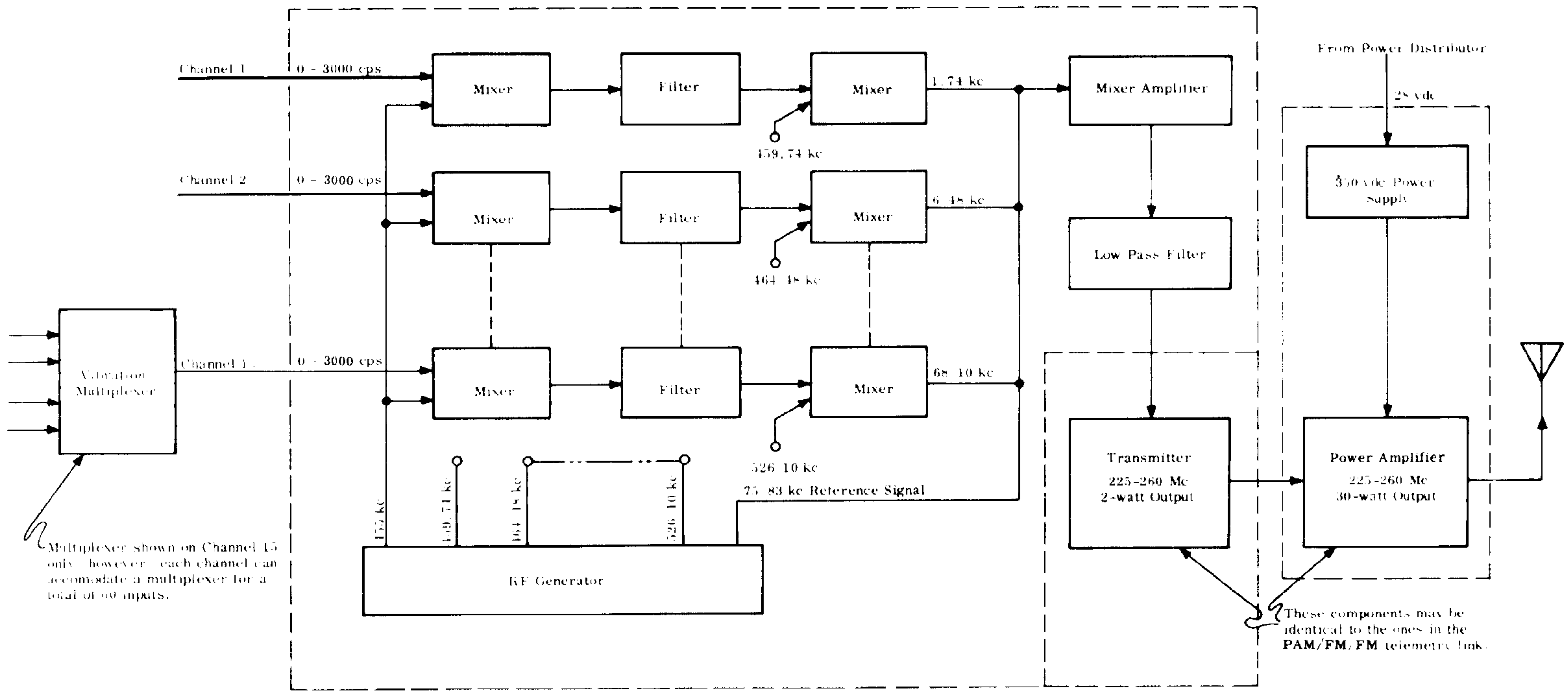


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Figure 6-9. SS/FM Telemetry Link

inputs to give a total capability of as many as 60 data inputs. The multiplexer samples 4 vibration measurements for approximately 3 seconds each, once each 13 seconds, or 2 vibration measurements for approximately 6 seconds each, once each 13 seconds.

The transmitter and power amplifier may be identical to the ones in the PAM/FM/FM telemetry system.

6-15. Recording System. During the periods when radio communication is blocked out, as during the firing of ullage rockets prior to separation of the first stage, the vehicle instrumentation system uses a tape recorder for recording significant separation data. The recorder has a rapid play-back capability, to reduce the loss of real time data during subsequent transmission from the stage.

6-16. Antenna System. Two separate, but identical telemetry antenna systems are used. Each system consists of an RF coupler, a power divider and two antennas. The RF coupler provides impedance matching between telemeters and antennas, and isolation between transmitters where several telemeters are feeding the same antenna system. The power divider, an impedance matching device, is used to connect two or more antennas to a single RF source. The antennas are slot-type radiators phased to provide maximum radiation in the aft direction.

6-17. IMPLEMENTATION.

The Saturn I stages and instrument unit are implemented with various telemetry systems. The allocation of the systems is presented in Table 6-4, below.

Table 6-4. Telemetry System Allocations

System	S-I	S-IV	Instrument Unit
PAM/FM/FM Telemetry	2		2
SS/FM Telemetry	1		1
PCM/DDAS Telemetry	1		1
PDM/FM/FM Telemetry		3	
(Tape Recorder)	1		

6-18. CHECKOUT.

Checkout is the process of verifying that the launch vehicle is capable of performing its mission. This process consists of a series of tests that start at the component level during manufacturing and end during the prelaunch phase with a simulated flight test involving the complete vehicle.

The Saturn I Block II checkout begins a transition in checkout philosophy from manual to fully automatic. (Ultimately, it is desired to have the checkout of Saturn V fully automated.) This change in philosophy is necessitated by the increased complexity of the Saturn generation of launch vehicles. As the vehicles increase in complexity they approach a state where a checkout within reasonable manpower and time limitations is not possible unless the checkout is fully automated. For example, the checkout of a Redstone required 12 man days, a Jupiter 16 to 20 man days, and Saturn I Block I 144 to 170 man days. During this transitional period it is planned to develop not only the equipment necessary to perform the fully automated checkout, but also a pool of personnel who have experience in the use of the equipment.

The transition will progress in an orderly manner with more tests being automated with each successive vehicle. Initially, when a test is first automated, the manual test equipment will be retained as a backup to be used primarily for debugging the automatic test program. When a high degree of reliability has been established in the automatic test program, the manual test backup will be abandoned. In the description that follows, the checkout philosophy is first discussed; then the checkout organization; and finally the equipment and methods used in the checkout of the S-I stage and instrument unit at MSFC, the S-IV stage at Douglas, and the Saturn I launch vehicle at VLF 34/37.

6-19. CHECKOUT PHILOSOPHY.

(To be supplied at a later date.)

6-20. CHECKOUT FLOW ORGANIZATION.

(To be supplied at a later date.)

6-21. STAGE CHECKOUT.

The two stages and instrument unit of the Saturn I are checked out either directly by MSFC or under the cognizance of MSFC. For this description, the checkout is confined to the tests that are performed on the composite stage after final assembly and inspection.

6-22. S-I STAGE CHECKOUT.

The checkout of the S-I stage is performed by MSFC after its arrival from the Chrysler Corporation manufacturing facility (Michoud, Louisiana). The checkout consists of a series of tests divided into the following major categories.

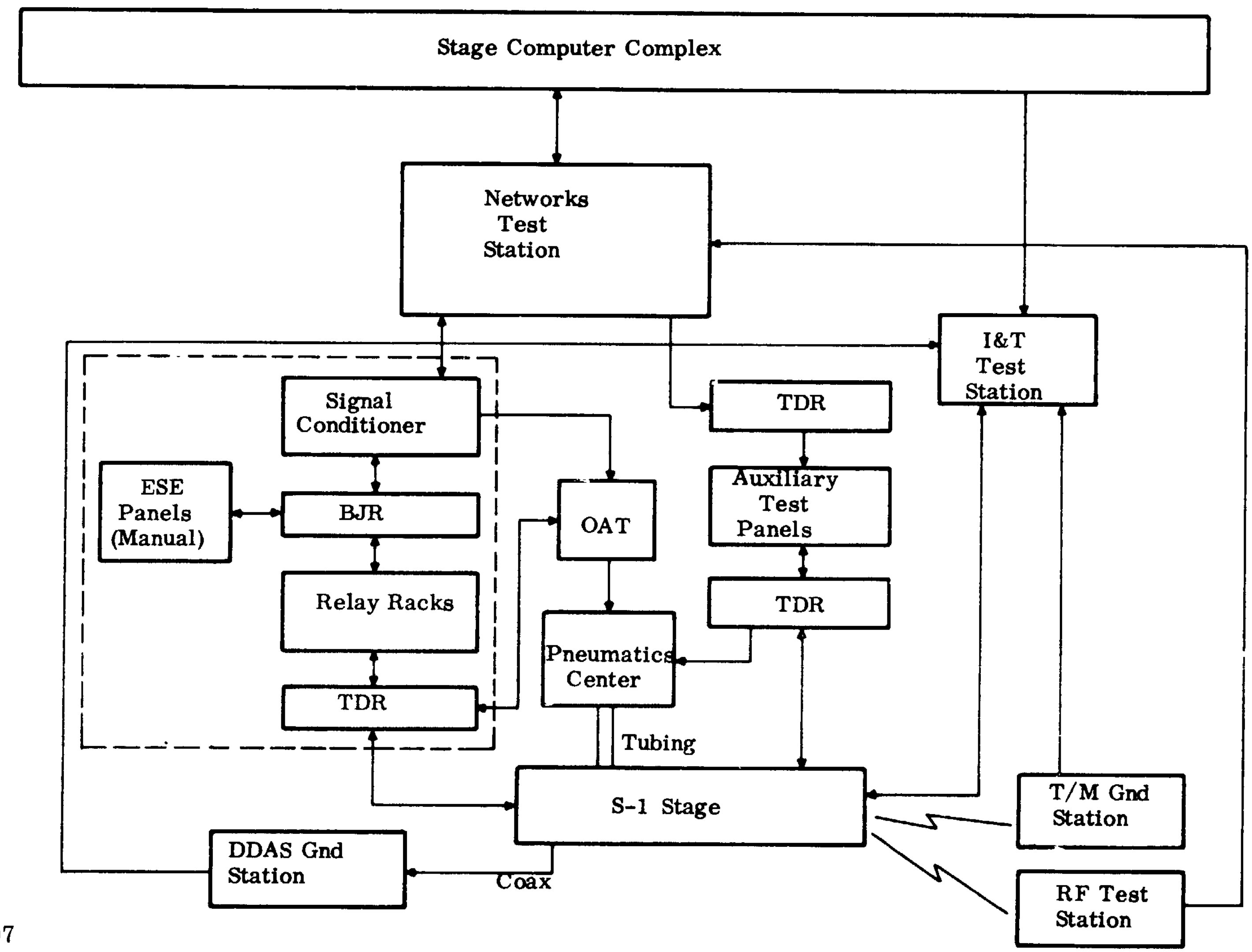
a. Electrical networks
b. Measuring, rough combustion cutoff, and fire detection.
c. Telemetry
d. Radio frequency systems
e. Guidance and control systems
f. Mechanical systems
g. Vehicle systems

Automated testing and checkout of the Saturn launch vehicle propulsion stage, as performed at MSFC, centers about the use of one or more general purpose digital computers and associated peripheral equipment the purpose of which is to provide computer programmed control of remotely located test stations, Figure 6-10. Stage interface substitutes are used as a necessary adjunct to this individual stage checkout.

The test stations, also referred to as satellite stations, contain a modular assembly of operator switches and indicator lights, buffers and logic circuitry, switching and selection circuitry, stimuli generators, analog-to-digital (A/D) and digital-to-analog (D/A) data conversion equipment, and power supplies to allow stimulation of and measurements on those stage subsystems or systems to be associated with a particular test station. This checkout is accomplished under programmed control from the Stage Computer Complex or from manual control by the test station operator.

The Stage Computer Complex at MSFC employs three Packard Bell-250 digital computers in a master-slave arrangement. The ability to expand the number and scope of computer programmed stage checkout operations (by the addition of test stations and more PB 250 computers as slave computers) allows this computer-automated system to progressively replace or complement in an orderly manner

TDR - Test Distributor Rack
BJR - Blockhouse Junction Rack



3-307

Figure 6-10. Over-all Test Setup for S-1 Stage the manual checkout console and procedures used to date, Figure 6-11.

There are six test stations that are an integral part of the total automated checkout system and are tied into the Stage Computer Complex.

a. Electrical Systems Test Station
b. Networks Test Station
c. Mechanical Assembly Test Station
d. Vehicle Test Station
e. RF Test Station
f. Instrumentation and Telemetry Test Station

The specific checkout functions and equipment in each test station are described in the following paragraphs.

6-23. Electrical Systems Test Station. The electrical system test station tests electrical ground support equipment and vehicle cables associated with the S-I stage and instrument unit. The electrical ground support equipment is tested in subunits. When connected to the test station, these subunits are automatically tested and the test results are analyzed and presented in a convenient format.

The test station can be divided functionally into three distinct stations, which are satellite station buffer and control, cable analyzer, and functional analyzer.

The satellite station buffer is functionally an extension of the Stage Computer Complex, and is used in controlling the cable and functional analyzers. It is also common to all test stations within the automatic checkout system.

The buffer and control section is composed of four rack-mounted chassis, a display and control panel, and a Flexowriter. The primary functions of the satellite station buffer and control section are (a) communication with the Stage Computer Complex, (b) limited manual control of the test station operation, and (c) visual display of test data and program status.

The cable analyzer is designed to automatically perform two basic tests, a high-voltage dc leakage test and a continuity test. It is provided with sixty 50-pin connectors (3000 test terminals), and is controlled by a 38-bit buffer register which can be fully loaded by four separate data words from the satellite station buffer. Each

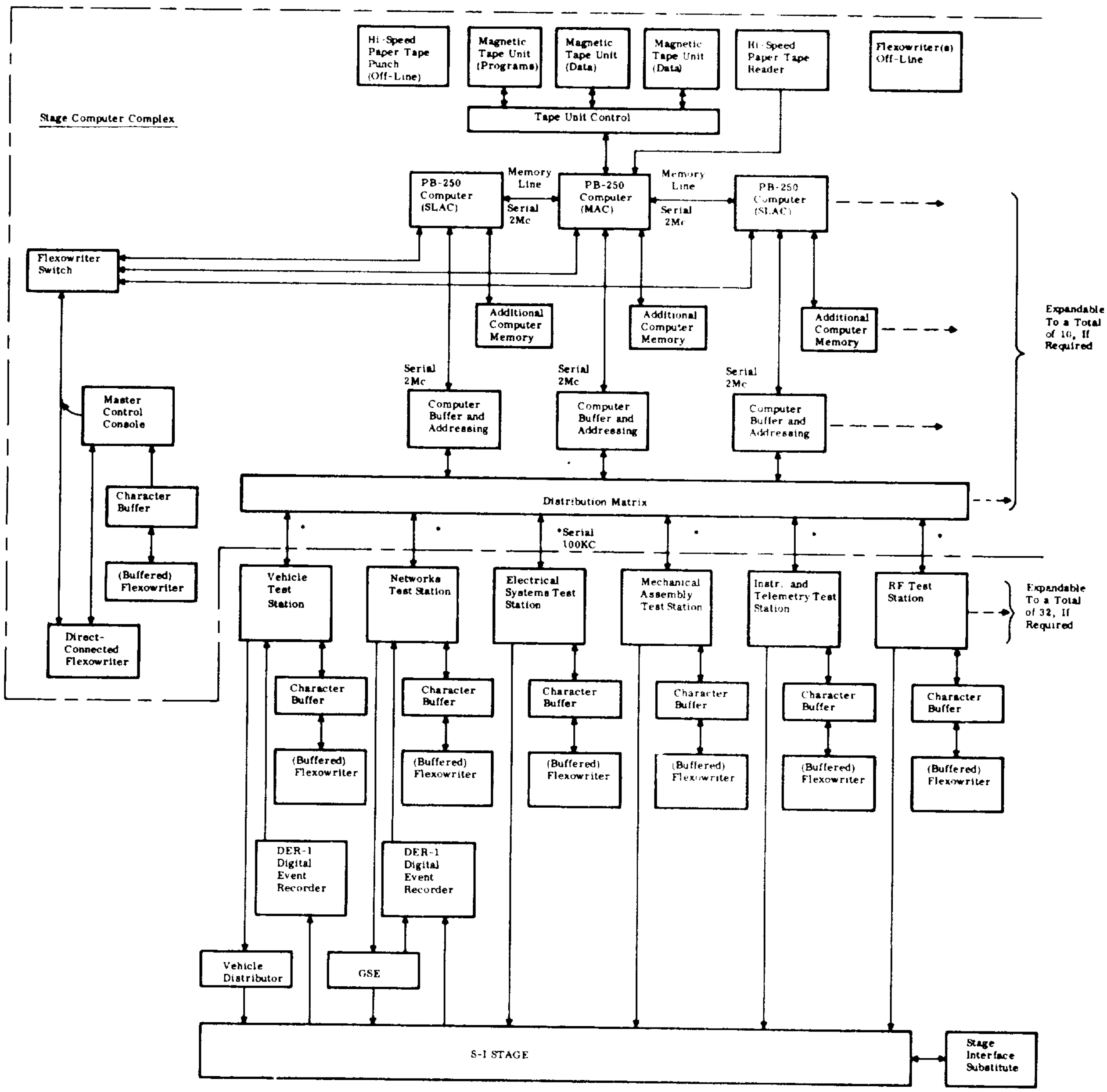


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Figure 6-11. Quality Assurance Laboratory Automated S-I Stage Checkout Facility data word requires a distinct address.

The functional analyzer is designed to provide all the stimuli, switching, and response measuring capabilities required for automatic testing of the rack-mounted units and panels in the Saturn electrical ground support equipment.

The analyzer requires programmable stimuli generators and sufficient switching capabilities so that any one of the several stimuli generators may be connected to any one of a combination of test terminals. In addition, switching capabilities are provided which select a combination of test terminals where the desired response to stimuli may be present. The response terminals are connected to program-selected response conditioning and measuring equipment. The analog test measurement data is then converted to the required digital form for transmittal to the Stage Computer Complex. The Stage Computer Complex then analyzes, evaluates, and records the test data, as well as initiates data display at the satellite station buffer.

The electrical system test station is contained in five mobile carts. One cart housed the satellite station and buffer and control unit. The cable analyzer and functional analyzer are housed in two carts each. The contents of an entire station may be used together, or either the cable analyzer or the functional analyzer may be used alone with the satellite station buffer. The carts may be moved individually or coupled and moved as a group.

6-24. Networks Test Station. The networks test station is used to perform general networks evaluation and over-all acceptance testing. General networks evaluation is accomplished by providing stimuli and response monitoring for vehicle electrical systems and vehicle simulation for checkout of the GSE compatibility. The vehicle simulator performs the same function as the ground equipment test set (GETS). For over-all acceptance testing, the networks test station provides control and monitoring of the facility and vehicle power sources, switching necessary for launch preparations and firing sequence, functions for checkout of guidance and control components, pneumatic supplies, and simulation and substitution. Test procedures are written and stored on magnetic tape at the Stage Computer Complex. In a fully automatic mode, a test request is made to the master computer. A slave computer is loaded with proper tape and the switching sequence is begun. Communication with the Stage Computer Complex utilizes the standard station buffer and the 22-bit word. Patch-boards are built up for a particular test set-up and are plugged in when that test set-up is required.

In the present configuration, the over-all tests can be performed by any one of three modes: automatic, computer controlled manual, or fully manual utilizing the manual ESE. The interface between the networks test station and the manual ESE is illustrated in Figure 6-12. When a relay is energized in the stimulus selection matrix, representing a particular discrete function to be performed on the vehicle, it does not directly provide the stimuli, but energizes a relay in the relay racks, via the signal conditioner, which normally would be operated manually from the ESE. This relay provides the stimulus to the proper line. The equipment within the dotted lines is supplied by the Astrionics Laboratory and is identical at the factory and launch site. Thus, the ESE/automatic interface remains the same regardless of the control computer configuration. The stimuli, both ac and dc, are provided from the digital-to-analog converters through the stimulus selection matrix in the same manner as the discrete switching signals described above. All responses from the system under test enter through the patchboards and are either measured or monitored. Frequency measurements are performed by a frequency counter. All other measurements are sent through the analog-to-digital converter to the appropriate measuring device in the networks test station, and then to the computer for comparison. Discrete events are monitored by the matrix gate for continued responses and sent through the buffer to the stage computer. Responses requiring a hard-copy output are sent back through the patchboard to the digital event recorder. Responses for which timing is critical to the program underway can be directed back to the controlling computer through the digital event recorder buffer.

Over-all tests for which this station is used are: power distribution and pneumatics components test, general networks and malfunction cutoff test, control over-all test, simulated plug drop test, and simulated flight test.

The station is also used to support the control subsystem, RF subsystem, and telemetry calibration tests.

6-25. Mechanical Assembly Test Station. The mechanical assembly test station is used to optically align critical surfaces and centerlines, measure stage weight, and determine the center of gravity and mass moments of inertia.

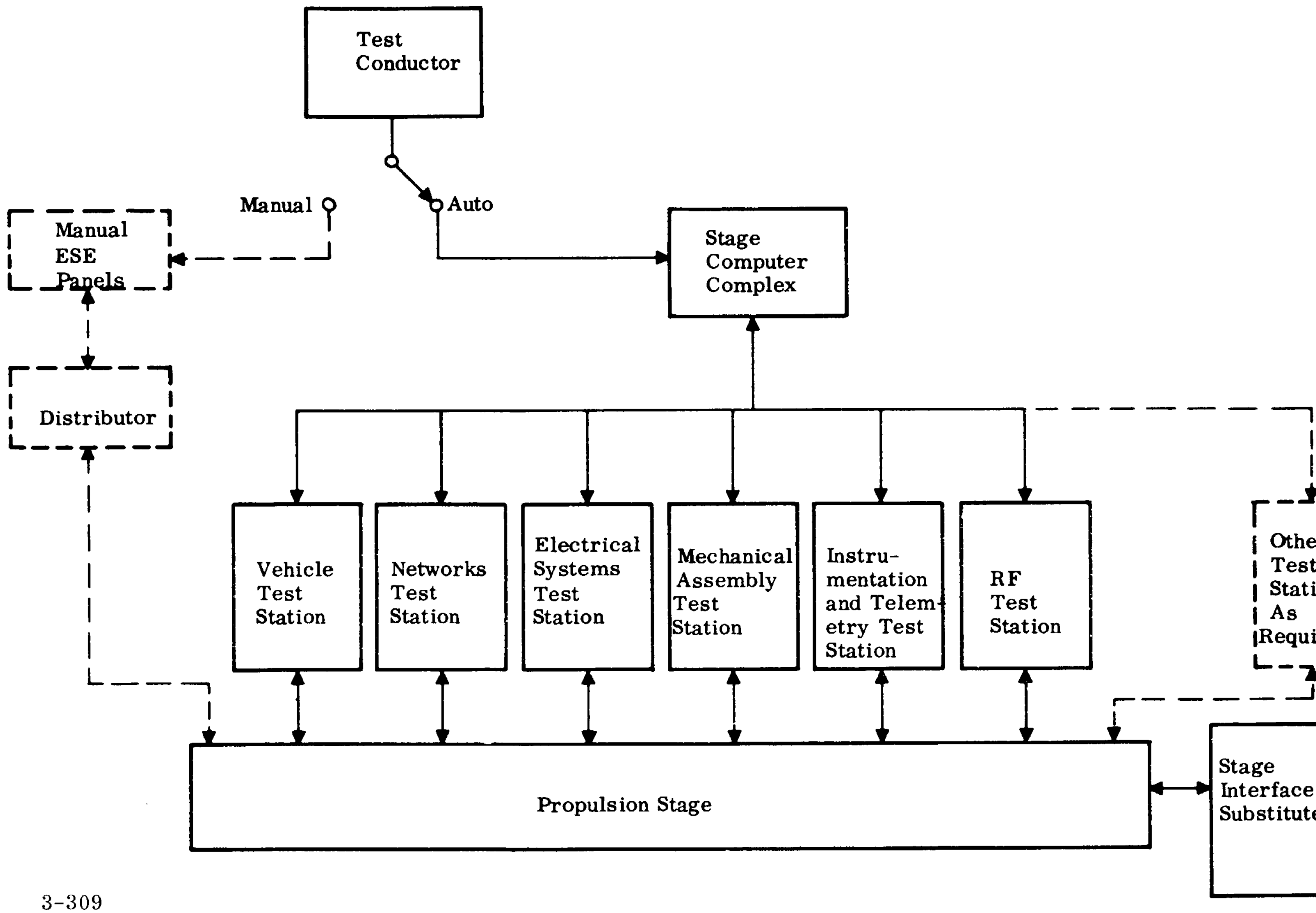

Figure 6-12. Quality Assurance Laboratory Propulsion Stage Checkout

The station can be operated either automatically through the Stage Computer Complex or manually by an operator. When the test station is computer controlled, a choice of automatic or single-step operation is available. In the manual mode, the test station is programmed and controlled by the operator.

The station, Figure 6-13, is used under computer control to generate stimuli, perform necessary switching and measuring, and transmit data to the Stage Computer Complex for the following measurements of the S-I stage:

a. Planer quality of the fuel tank manhole cover sealing flanges.

b. Geometric thrust vector, area, and centroid of the rocket engine thrust chamber.

c. Optical alignment with predetermined set points.

d. Weight and mass moments of inertia and center of gravity.

e. Level sensing.

The station consists primarily of three racks containing as major components a low-level multiplexer, analog-to-digital converter, satellite station buffer, operator console, flexowriter, displacement indicators, demodulators and amplifiers, stimuli generation matrices, response selection matrices, and power supplies.

6-26. Vehicle Test Station. The vehicle test station is used to perform mechanical systems pressure and functional tests. The station consists of computer controlled test equipment designed to measure the condition and flight readiness of the S-I stage mechanical system "critical items." The capabilities of this equipment include generating stimuli, maintaining both electrical and mechanical control over the hydraulic and pneumatic systems being tested, performing the necessary functions to accomplish the tests, and the transmission of response data to the Stage Computer Complex.

The vehicle test station performs such tests as:

a. Verifying heater operation and thermostat settings for all ac heaters.

b. Verifying actuation and deactuation settings and leak checking all pressure switches.

c. Verifying operation of the control pressure system components and leak checking all tubing and connections.

d. Verifying operation and leak checking the gas generator and LOX control valve assemblies.

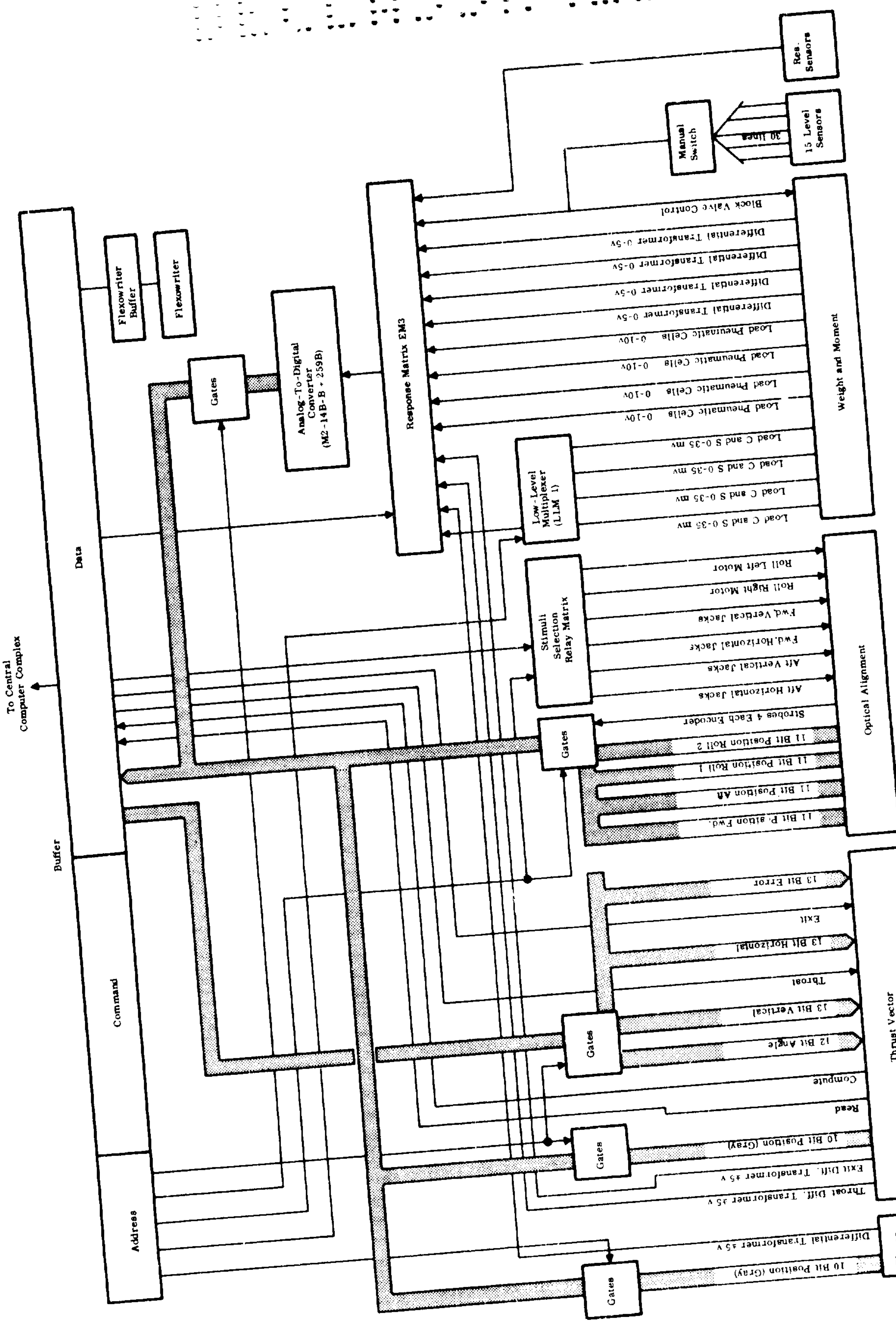

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Figure 6-13. Mechanical Assembly Test Station Block Diagram

e. Leak checking the gas generator, gas turbine, turbine exhaust and turbopump gearcase.

f. Functionally testing and leak checking the fuel additive mixer unit and the H-I engine control system.

g. Leak checking the GOX line assembly, purge system, air bearing system, helium system, LOX containers and combustion chamber, GN_2 fuel high-pressure spheres and fuel containers.

h. Pressure testing the instrument containers and cooling system.

i. Functionally testing the hydraulic system.

The transition from manual checkout to fully automated checkout in the area of the vehicle test station is progressing through three phases:

Phase I - manual hookup procedures, phasing into automated instrumentation readout.

Phase II - Semi-automated operation utilizing the available computer programming capability in conjunction with the automated instrumentation readout, but continuing manual hookup.

Phase III - achieve significant automation by implementing vehicle design changes to include built-in monitoring instrumentation and centralized hookup points.

6-27. <u>RF Systems Test Station.</u> The only piece of RF equipment which is a part of the stage checkout and which is system checked with the stage is the command receiver.

In the stage checkout phase, the command receiver is checked out utilizing a PB-250 computer. On the other hand, the automated checkout for the instrument unit, which contains the remainder of the RF system components, uses the RCA-110 computer as the control and comparison device.

6-28. <u>Instrumentation and Telemetry (I&T) Test Station.</u> The I&T test station is used to stimulate and calibrate transducers and signal conditioning equipment aboard the Saturn vehicle, in performing these functions, the test station monitors the output signals, and compares them to the same signals after transmission over the telemetry links. The I&T test station can be operated either through the Stage Computer Complex (PB-250), or manually by the operator. Data transmission between the test station and the central computer is by means of a 22-bit word.

The Stage Computer Complex controls the various tests in accordance with a predetermined, stored program. The computer controls the generation and distribution of the stimuli. When the data is received from the I&T test station, the computer activates switches or displays and prepares a final permanent record of the test results. When the test process requires operator participation for a scheduled break in the program or for out-of-tolerance data, the computer is programmed to issue instructions to the operator and to halt the automatic process until the operator provides the necessary stimulus or reply. A Flexowriter provides communication between the operator and the computer.

The test station has three major functions: to initiate, monitor, and interpret calibration and test results of the universal measuring adapter (UMA) calibration system, pneumatic pressure distribution system and miscellaneous sensor response system.

High-pressure and low-pressure digital-to-pressure generators are used for stimulating pressure transducers. Twenty-two relay closures are provided to select the pressure and 145 closures are provided to switch manifold configurations for directing the proper pressure to the transducer under computer control. Stimulation of other transducer types is primarily manual or mechanical, but under computer control.

The telemetry substation, which is under computer control, has 128 contact closures for setting up the proper receiver-discriminator combination required by the various telemetry channels. It has circuitry for decoding and measuring decommutated PAM data and also has the ability to drive calibration devices for calibrating the telemetry equipment itself. All measured data are returned to the I&T test station for processing.

6-29. INSTRUMENT UNIT CHECKOUT.

Automated testing and checkout of the Saturn instrument unit at MSFC follows the same general plan as the S-I stage test and checkout. The components and subsystems are thoroughly evaluated prior to assembly into the instrument unit. After assembly, testing of individual components within the instrument unit is performed only to the extent that in-line equipment designs permit identification within each subsystem.

For systems tests and over-all tests, instrument unit checkout automatic support equipment performs the following functions:

a. Over-all computer program control
b. Electrical network control and checkout
c. Over-all test control and checkout
d. Guidance and control checkout
e. Instrumentation calibration
f. PCM telemetry checkout
g. RF checks
h. Digital data acquisition subsystem checks
i. Sequence event recorder
j. Test program and data storage

The main constituent of the automatic support equipment is the RCA-110 computer which directs the associated input/output equipment. In performing a test, the computer directs program information to the selected test station via a buffer. This information, in message or word form, is translated by the test station and used to perform an operation (switching, stimuli, and measuring) on the instrument unit. The results of this operation are converted to computer language by the test station under computer control. Test results are then evaluated by the computer and stored in a standard format.

A parallel monitoring system, the digital data acquisition system (DDAS), is used to validate the digital data approach. The DDAS/PCM ground station consists of digital decoders which present data to the RCA-110 computer for evaluation. Consequently, there are three methods of collecting test data during the instrument unit checkout.

a. Hardwire by the test stations and computer
b. Coax by the DDAS/PCM ground station and computer
c. Manually by electrical support equipment.

The manual electrical support equipment operation is independent of the automatic equipment and provides a backup control of the instrument unit test.

After validation of the DDAS, the hardware monitoring method of data acquisition will be minimized.

A general block diagram of the instrument unit checkout equipment is illustrated in Figure 6-14.

The instrument unit test operation contains the following features:

a. Test stations may request, via operator action, a specific component test to be performed and evaluated by the computer, or evaluated by the test station display console.

b. The computer can directly insert exercise problems into the flight guidance computer where results are automatically or visually evaluated.

c. Test results from the digital data acquisition subsystem coax link can be compared with results obtained through test stations by hardware.

d. A complete dynamic test can be performed by commanding a motion simulator to exercise the guidance system through a pre-determined test regime and recording control responses.

e. A complete manual test can be performed by using GSE. Even in automatic mode, the GSE has a passive monitoring function.

Checkout equipment for the instrument unit encompasses both conventional manual control and automatic computer control, with a gradual phaseover to the automatic mode. The automatic support equipment consists of the following major elements:

a. RCA-110 computer with associated input/output units
b. Computer external buffers
c. Test stations
d. Digital data acquisition subsystem/PCM ground station
e. Guidance monitor
f. Analog and discrete signal conditioning racks.

The mechanical support equipment consists of the following major elements:

a. Saturn instrument unit motion simulator (SIUMS).
b. S-I dynamic substitute (SIDS).
c. Air conditioning unit.
d. LN_2 trailer.

As an integral part of the automated checkout system, and tied into the computer complex above, are three test stations, which are:

a. RF systems test station.
b. Guidance and control (G&C) test station.
c. Instrumentation and telemetry (I&T) test station.

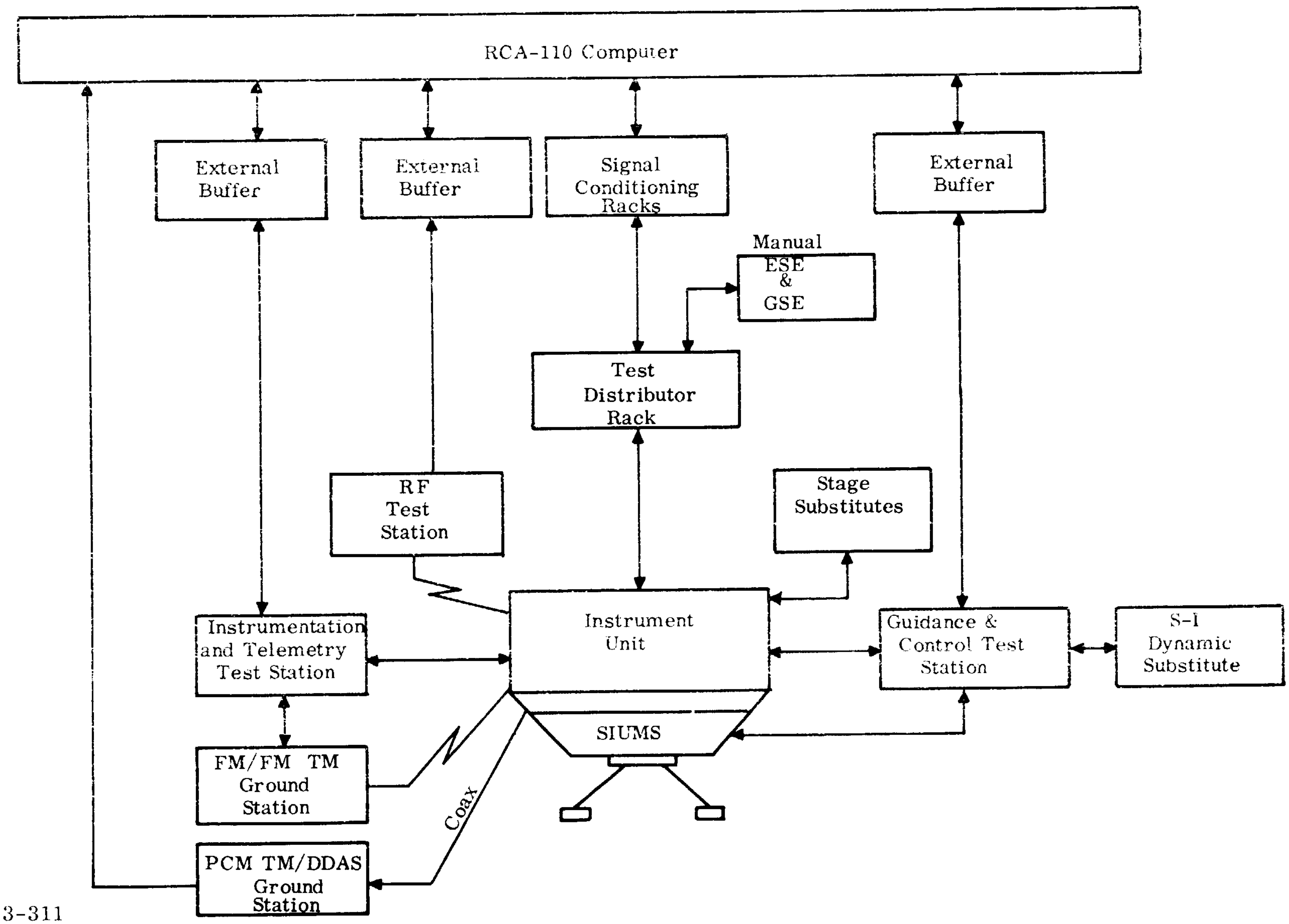


Figure 6-14. Computer Complex for Instrument Unit Test

6-30. RF Systems Test Station. The RF systems test station is designed for operation by the PB-250 computer in checking out the command receivers located in the S-I booster stages. With a change in external buffering to match it to the RCA-110 computer, it operates from the RCA-110 to checkout command receivers and other RF equipment located in the instrument unit.

For each RF subsystem, the following test modes are used:

a. A component functional test in which all important component performance parameters are measured.

b. An on-vehicle functional test consisting of system level tests, phasing checks, RF loss measurements, and antenna.

c. A compatibility test in which all on-board RF systems are operated in conjunction with the telemetry system to check for random triggering and intersystem interference.

d. Over-all tests and simulated flight test in which all vehicle/instrument unit equipment is operated in the flight mode.

6-31. Guidance and Control Test Station. The guidance and control test station is used under program control to generate stimuli, perform switching and measuring, and transmit data to the RCA-110 computer for evaluation and storage. The test station performs two major functions:

a. Monitoring all input and output conditions.

b. Controlling all input and output conditions.

The test station is capable of performing component level tests as well as tests involving the entire guidance and control system in over-all test and flight simulation test. All testing involving the guidance and control system is performed by hardware through the test station. Monitoring of guidance and control testing is performed by RF systems and the digital data acquisition system. It is anticipated that eventually the test station will be removed either in part or in its entirety and all testing will be accomplished by direct communication between the guidance and control system and the computer by the use of digital data techniques (digital data acquisition system).

The test station performs two levels of tests:

a. Various tests to validate proper operation of the stabilized platform, control computer, and their associated electronics for individual and integrated performance.

b. Integrated test of the guidance and control system and associated networks by a program sequence through power transfer, lift-off, S-I and S-IV cutoff and payload separation.

Testing of the guidance and control system is performed in four modes:

a. Fully automatic

b. Single-step automatic

c. Manual programming

d. Manual electrical support equipment control

The fully automatic mode allows a complete test to be performed automatically without any manual assistance. The single-step automatic mode allows a test to be performed in single-step entities with the test station or the RCA-110 computer manually advancing the program in a step-by-step fashion. The manual programming mode allows a test to be performed by the test station or the RCA-110 computer manually entering each program step into the computer for effecting the step operation. The manual electrical support equipment mode allows a test to be performed by the conventional manual electrical support equipment controls.

6-32. Instrumentation and Telemetry (I&T) Test Station. The checkout of the instrument unit I&T follows the same pattern as the checkout of the S-I stage I&T, only the test sequence is simpler for the instrument unit I&T. The RCA-110 computer is used with the instrument unit I&T rather than the PB-250 computer which is used with the stage checkout.

6-33. S-IV STAGE CHECKOUT.

The S-IV stage is manufactured and tested by the Douglas Aircraft Company and shipped directly to AMR. There are two Douglas facilities instrumental in the testing. After final assembly and inspection at Santa Monica, California, the stage is calibrated and given a functional checkout to qualify it for shipment to the static test facility at Sacramento, California. Here, the stage is given a checkout to verify that no degradation has occurred during shipping. A static firing test is then performed. After the firing test, the stage is removed from the static test firing stand, given another checkout, and is then shipped to AMR. This test program is performed by Douglas Aircraft personnel under the cognizance of MSFC.

6-34. SATURN I CHECKOUT AT VLF 34/37.

(To be supplied at a later date.)

6-35. ATTITUDE CONTROL AND STABILIZATION.

6-36. REQUIREMENTS.

The Saturn I attitude control and stabilization function maintains a stable vehicle motion (through the engine gimballing system) and adjusts this motion in accordance with guidance commands. During the ascent phase this function directs the vehicle orientation about its axes, maintains the angular rate of vehicle movement about the axes within allowable limits and damps any first bending mode oscillation of the vehicle structure.

The attitude control and stabilization function performance is limited by various constraints. During S-I stage flight, the high aerodynamic pressures encountered by the launch vehicle result in structural constraints and related control problems. The launch vehicle is aerodynamically unstable, therefore, a minimum angle-of-attack flight prevents excessive structural loading from aerodynamic forces and large gimbal deflections of the control engines.

A constraint exists because of the natural bending of the vehicle structure. During S-I stage powered flight, any oscillations occurring in the first bending mode of the structure must be actively damped by thrust vectoring.

The Saturn vehicle is required to maintain the launch orientation for several seconds after liftoff, permitting it to rise above the launch facilities to gain maneuvering clearance. The size and complexity of the launch vehicle and launch facilities constrain the launch vehicle to a specific launch orientation.

Immediately prior to vehicle staging the attitude control and stabilization function must restrain the launch vehicle to a constant attitude orientation to prevent excessive rotational rates during the separation process. After S-I stage separation and S-IV stage engine ignition, any separation transients must be damped.

For S-IV stage flight, the attitude control and stabilization function is required to accept guidance-steering commands and direct the launch vehicle motion to meet the requirements of these commands.

6-37. OPERATION.

Due to the various launch vehicle and control constraints, a programmed attitude control, without active guidance, is used for S-I stage powered flight. The programmed attitude control is accomplished in three phases; launch stabilization, maneuvering, and prestaging stabilization.

The launch stabilization period begins with liftoff and terminates after several seconds during which time the launch vehicle rises vertically to attain a physical clearance with the launch facilities.

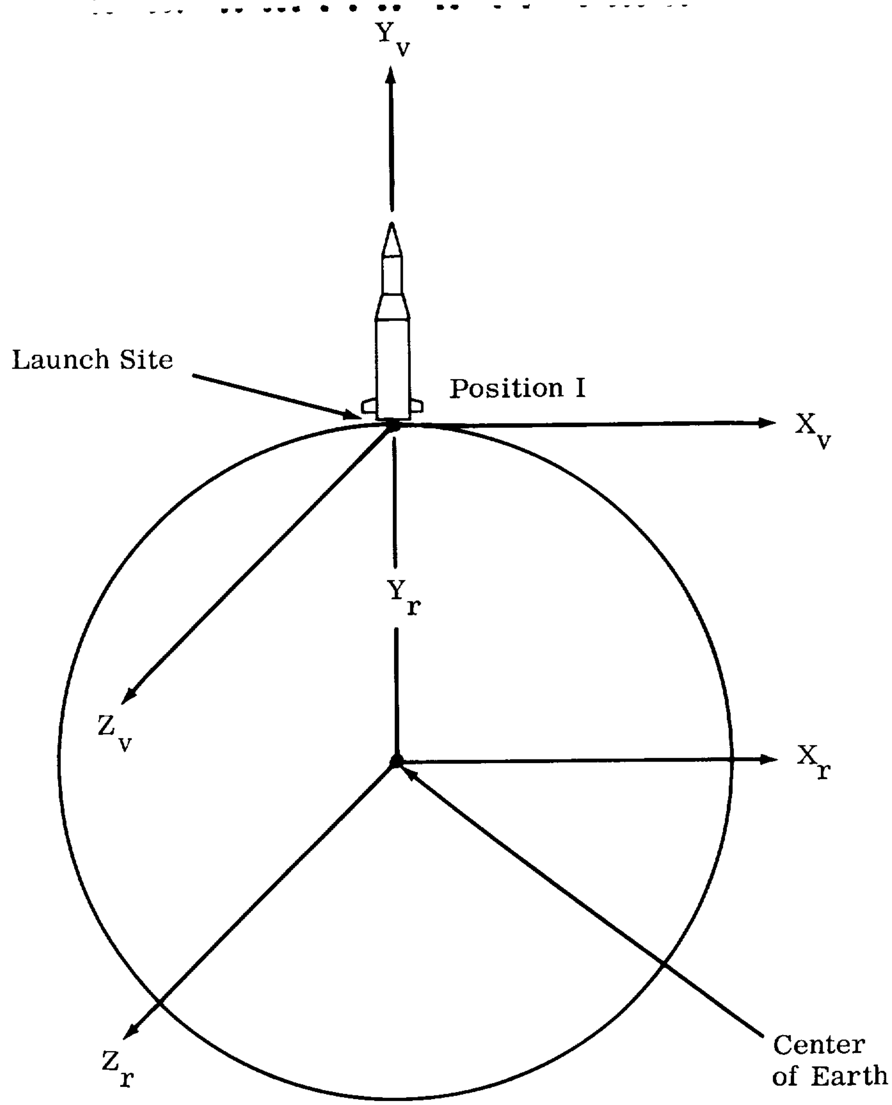
Upon termination of the launch stabilization period, the launch vehicle begins the maneuvering phase with a programmed roll maneuver. This maneuver consists of the launch vehicle maintaining a constant rate of roll until such time as its pitch plane coincides with the flight azimuth. Coincident with initiation of the roll maneuver, the launch vehicle starts a gravity-turn, time-tilt pitch maneuver. This maneuver rotates the longitudinal axis of the launch vehicle in the pitch plane toward the flight azimuth. A few seconds prior to vehicle staging, the time-tilt maneuver is terminated.

The prestaging-stabilization phase begins with termination of the time-tilt maneuver and ends with S-I stage outboard engine cutoff. The prestaging stabilization maintains the launch vehicle in a fixed attitude orientation.

During S-IV stage flight, the attitude control and stabilization function maintains a stable vehicle motion and orients this motion as directed by guidance steering commands.

The Saturn I attitude control and stabilization function utilizes two reference systems, the measuring coordinate system and the vehicle axes coordinate system, Figure 6-15.

The measuring coordinate system (X_m, Y_m, Z_m) has its origin at the launch site. The Y_m axis of the measuring coordinate system passes through the center of the



(X_r, Z_r, Y_r) Space-fixed reference coordinate system

(X_v, Z_v, Y_v) Vehicle coordinate system

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Figure 6-15. Coordinate Systems, Saturn I

earth parallel to the direction of gravity and is positive outward from the earth's surface at the launch site. The X_m axis is oriented perpendicular to the Y_m axis and lies along the flight azimuth. The Z_m axis is orthogonal to the other two axes, Figure 6-16.

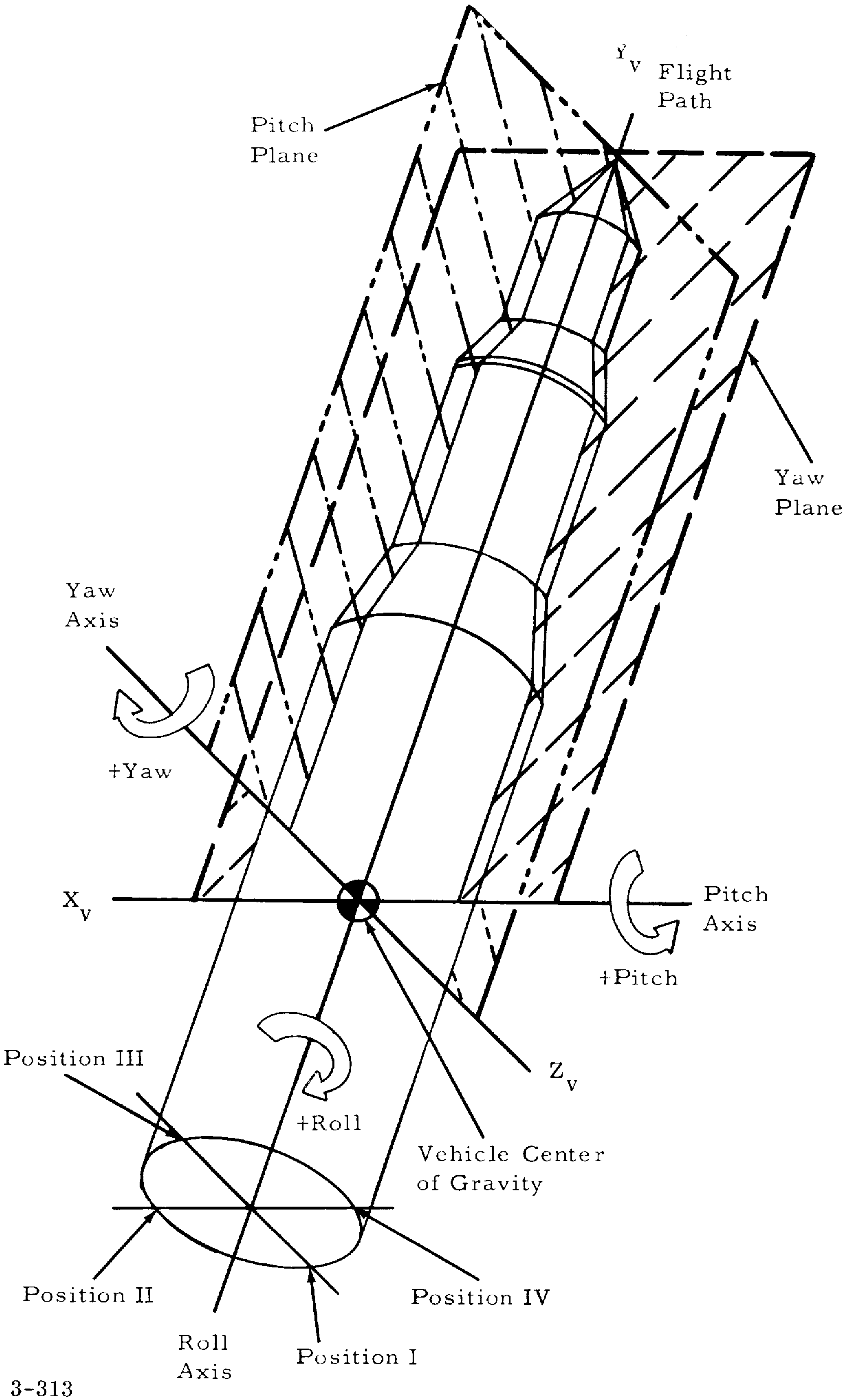


Figure 6-16. Vehicle Axes, Saturn I

The roll maneuver performed during ascent orients the vehicle X_v and Z_v axes to correspond to the measuring coordinates X_m and Z_m, respectively. Upon completion of this maneuver, the vehicle coordinate system and the measuring coordinate system are considered to be coincident, therefore, any movement of the launch vehicle can be sensed against the measuring coordinate system.

The attitude control and stabilization function is accomplished in three operations; sensing, error detection and engine actuation, Figure 6-17. During S-I stage flight vehicle lateral acceleration and body bending is sensed by two body mounted accelerometers. This information is used in limiting the vehicle angle of attack and in damping the oscillations of the vehicle first bending mode. The accelerometer location on the vehicle structure is based on vehicle dynamic properties (bending and angular rotation).

Vehicle angular rate is derived from attitude orientation information. The attitude

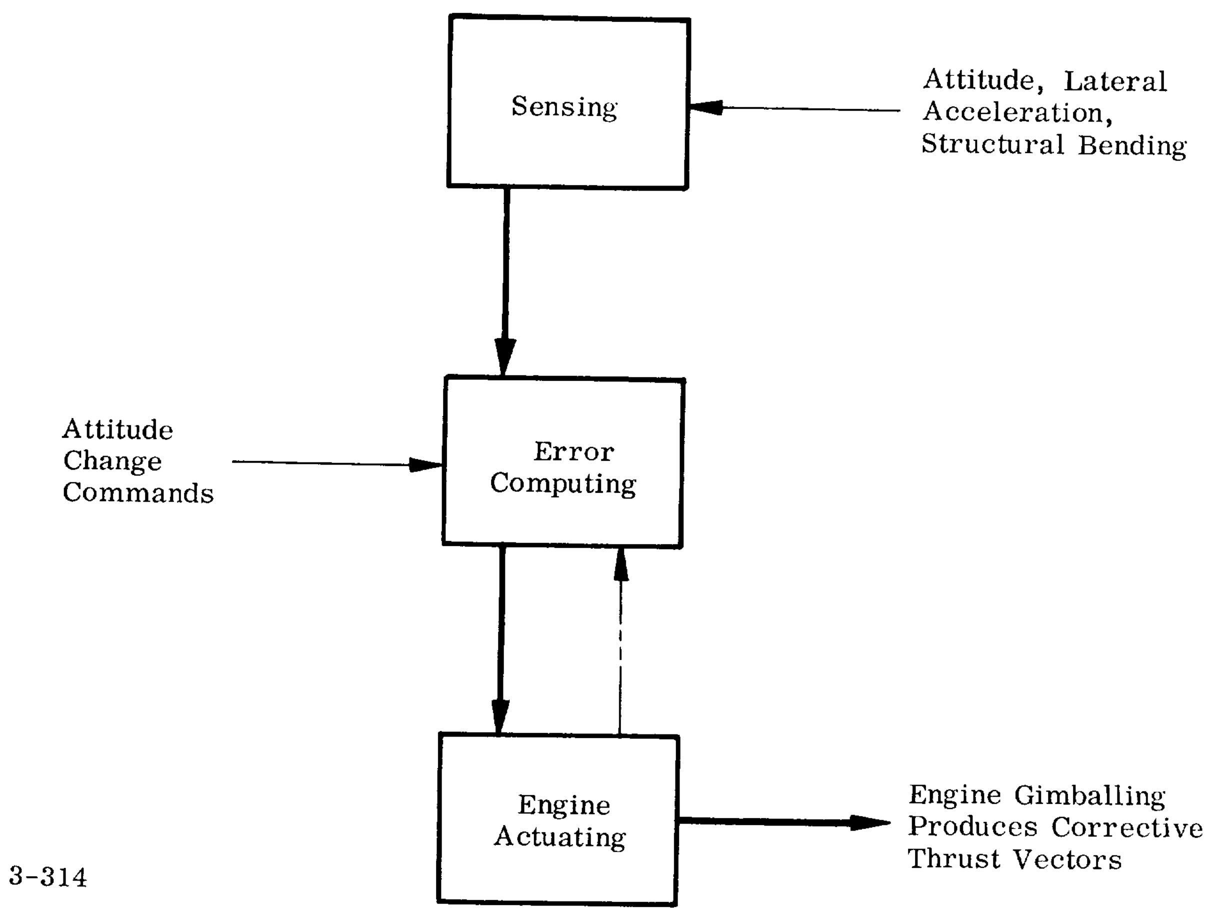


Figure 6-17. Attitude Control and Stabilization Operation, Saturn I

orientation is measured against an inertial reference (stabilized platform) which is space fixed a few seconds before liftoff. The attitude orientation of the vehicle with respect to the stable element is measured by resolvers on the roll, pitch and yaw gimbals of the platform. These resolvers are part of a resolver chain comprised of the platform resolvers and command resolvers.

The error detection is performed in the resolver chain by comparing the present vehicle attitude orientation with that specified by the command resolvers. These command resolvers receive programmed attitude commands during S-I stage flight and guidance steering commands during S-IV stage flight. If there are attitude errors, the resolvers generate corrective signals which are applied to the control computer.

Engine actuation signals are generated by the control computer. The control computer receives attitude errors from the resolvers, engine position feedback signals, and during S-I stage flight receives accelerometer outputs. These signals are filtered, scaled in amplitude and mixed in the right phase relationship to produce the engine gimbal signals which are applied to the hydraulic actuators.

The resultant mechanical gimballing of control engines produces corrective thrust vectors that change the orientation and/or angular rate of the vehicle. The engine position feedback information used in loop stabilization is generated by actuator position transducers.

6-38. GUIDANCE.

The Saturn I guidance function generates and applies commands to correct the motion of the launch vehicle toward a path that produces success in its assigned mission. The guidance process involves steering the vehicle in the pitch and azimuth planes with the generation of engine-cutoff commands upon attainment of proper vehicle velocity in relation to its position in space.

6-39. REQUIREMENTS.

During S-I stage flight the guidance function senses and accumulates velocity information to be used in guidance computations. Vehicle control during this portion of the flight is supplied by the attitude control and stabilization function. (Refer to Paragraph 6-35.)

During S-IV stage flight, the Saturn guidance generates steering and engine cutoff commands. This guidance is path adaptive in the pitch plane and delta minimum in the yaw plane with velocity-to-go computations used to generate the S-IV stage engine cutoff command.

The path-adaptive guidance steers the vehicle in the pitch plane along a constantly optimized trajectory to meet the mission requirements. This guidance does not adhere to a specific reference trajectory but adapts to the immediate flight situation by taking into account vehicle state variables and selecting new trajectories which are shaped to optimize desired features such as minimum fuel consumption or flight time. The selection of new trajectories or the optimization process is accomplished respecting the end or cutoff parameters of the mission. Utilizing path adaptive guidance, the launch vehicle performance is maximum even though perturbations, such as thrust deviation from normal, occur.

A delta-minimum guidance is utilized in the azimuth plane since accuracy requirements for this plane are not as stringent as those of the pitch plane. The delta-minimum guidance restrains the vehicle causing it to fly a pre-determined path in the azimuth plane. This guidance minimizes the vehicle displacement from a reference azimuth trajectory.

The velocity-to-go computation compares the vehicle present velocity with that velocity required to meet mission parameters. The difference between the two velocities represents the velocity to go. When the mission trajectory is correct and the velocity to go reaches zero, the S-IV stage engines are cut off.

6-40. OPERATION.

The guidance generation of steering and engine cutoff signals is accomplished in three operations; sensing, position computing and signal computing. The sensing and position computing operations are performed using separate, but related coordinate systems. These coordinate systems are the measuring coordinate system and the reference coordinate system. (Refer to Paragraph 6-37.)

The reference coordinate system is that coordinate system with axes oriented parallel to those of the measuring coordinate system at T_o. This coordinate system is inertially fixed with the center being the center of the earth.

6-41. Sensing. The sensing operation detects the launch vehicle apparent velocity in the measuring coordinate system. The apparent velocity is comprised of vehicle actual velocity plus components of gravitational velocity, due to the effects of gravity on the sensors. This velocity information is used as a basis for determining the launch vehicle position in space.

The sensing operation is accomplished by a stabilized platform system which is aligned prior to launch and maintains the orientation of the measuring coordinate system during the ascent of the launch vehicle.

6-42. Position Computing. The position computing operation locates the launch vehicle in the reference or space fixed coordinate system. To accomplish this operation, position computing obtains apparent velocity as sensed in the measuring coordinate system and subtracts the components of gravity velocity to yield components of the launch vehicle velocity. The components of velocity are then integrated from T_o and a transformation is made to the reference coordinate system to obtain the vehicle position in space.

6-43. Signal Computing. The signal computing operation performs three major operations during the S-IV stage flight:

a. Determines the optimum pitch path that the vehicle must follow to perform the mission. This determination is accomplished by combining the vehicle position, velocity and other state information with stored guidance constants to select a trajectory. The guidance constants are coefficients of precomputed approximating polynomials representing a class of trajectories which will lead to mission accomplishment. Signal computing selects the present optimum trajectory from this class and generates steering signals which are applied to the attitude control and stabilization function to direct the vehicle along the selected trajectory.

b. Determines the vehicle displacement from the referenced azimuth path and issues corrective steering signals to minimize the vehicle deviation from this reference.

c. Determines the velocity to go and evaluates it with the mission injection pitch-path angle to derive S-IV stage engine cutoff time. Signal computing then initiates the engine cutoff signal at the correct time.

6-44. GUIDANCE, ATTITUDE CONTROL AND STABILIZATION IMPLEMENTATION.

The guidance, and the attitude control and stabilization functions are jointly implemented in the launch vehicle as the Guidance and Control System. This hardware system is comprised of the ST-124 stabilized platform, ASC-15 digital computer, GSP-24 guidance signal processors, control computer, control sensors and servo actuators. With the exception of the control sensors and servo actuators, the units are located in the launch vehicle instrument unit. The control sensors are located in the S-I stage and the servo actuators are located in both the S-I stage and S-IV stage.

The attitude control and stabilization function, Figure 6-18, is implemented with the ST-124 stabilized platform, control computer, ASC-15 digital computer, GSP-24 guidance signal processor, servo actuators, and control sensors.

The guidance function, Figure 6-19, is implemented with an ST-124 stabilized platform, ASC-15 digital computer and GSP-24 guidance signal processor.

6-45. ST-124 STABILIZED PLATFORM.

The ST-124 platform has a four-gimbal configuration which provides the guidance function with vehicle velocity information and the attitude control and stabilization function with an attitude and angular rate reference.

The inner gimbal or stable element of the platform is maintained in a space fixed orientation utilizing three platform mounted single-degree of freedom gyroscopes as inertial sensors to drive servo systems which position the platform yaw, pitch and roll gimbals. The power to orient the gimbals is provided by dc direct drive servo motors attached to the gimbals.

The stable element also carries three orthogonally-mounted integrating gyro accelerometers which provide inertial velocity information for use by the digital computer. Additional units mounted on the stable element include three gas bearing pendulums for preflight platform alignment and accelerometer checkout, and one poroprism synchro and digital encoder assembly for preflight azimuth alignment.

Platform gimbal angles relative to vehicle attitude are measured by four pancake type resolvers. The platform gimbal resolvers are electrically connected in a

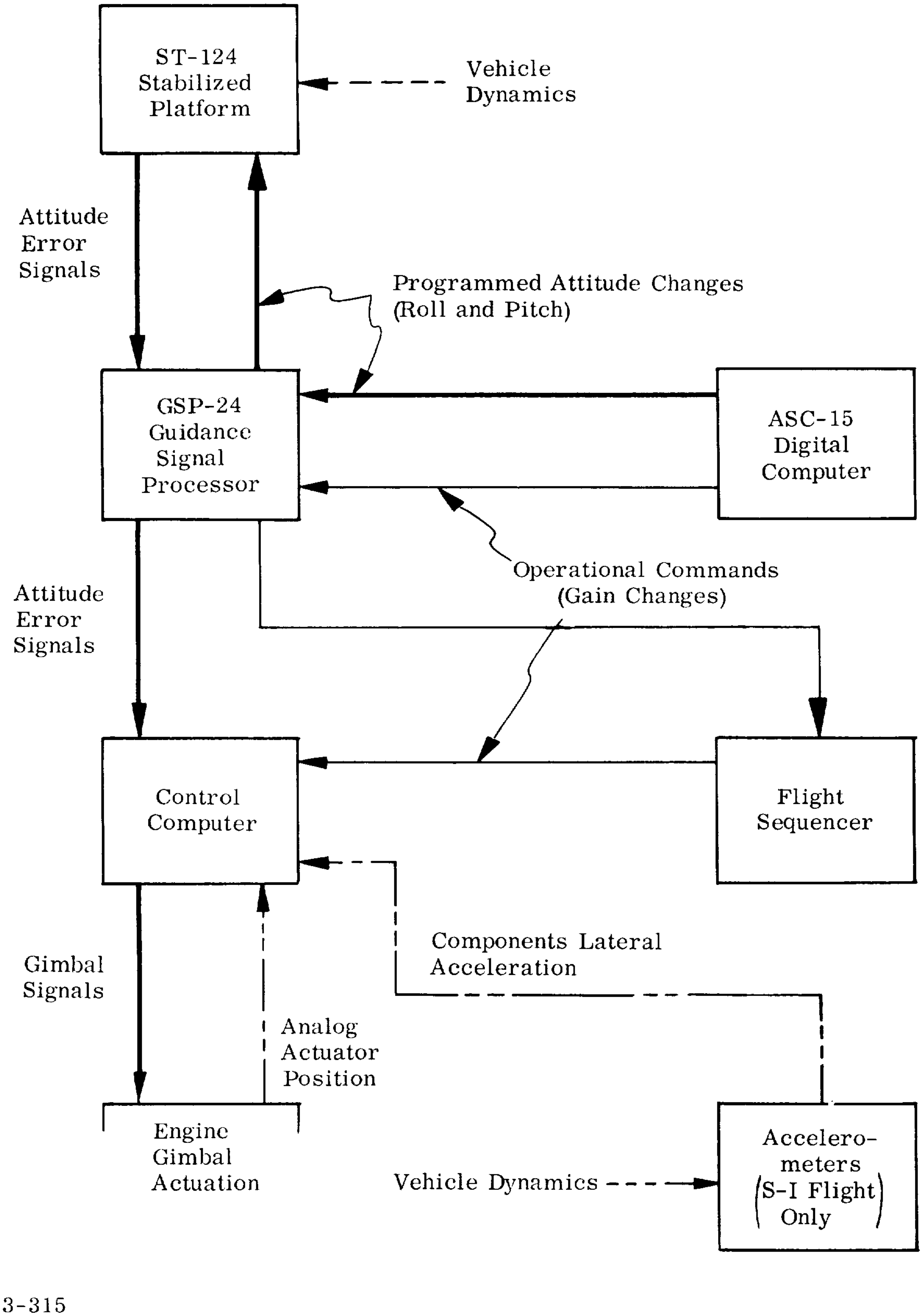


Figure 6-18. Attitude Control and Stabilization Implementation

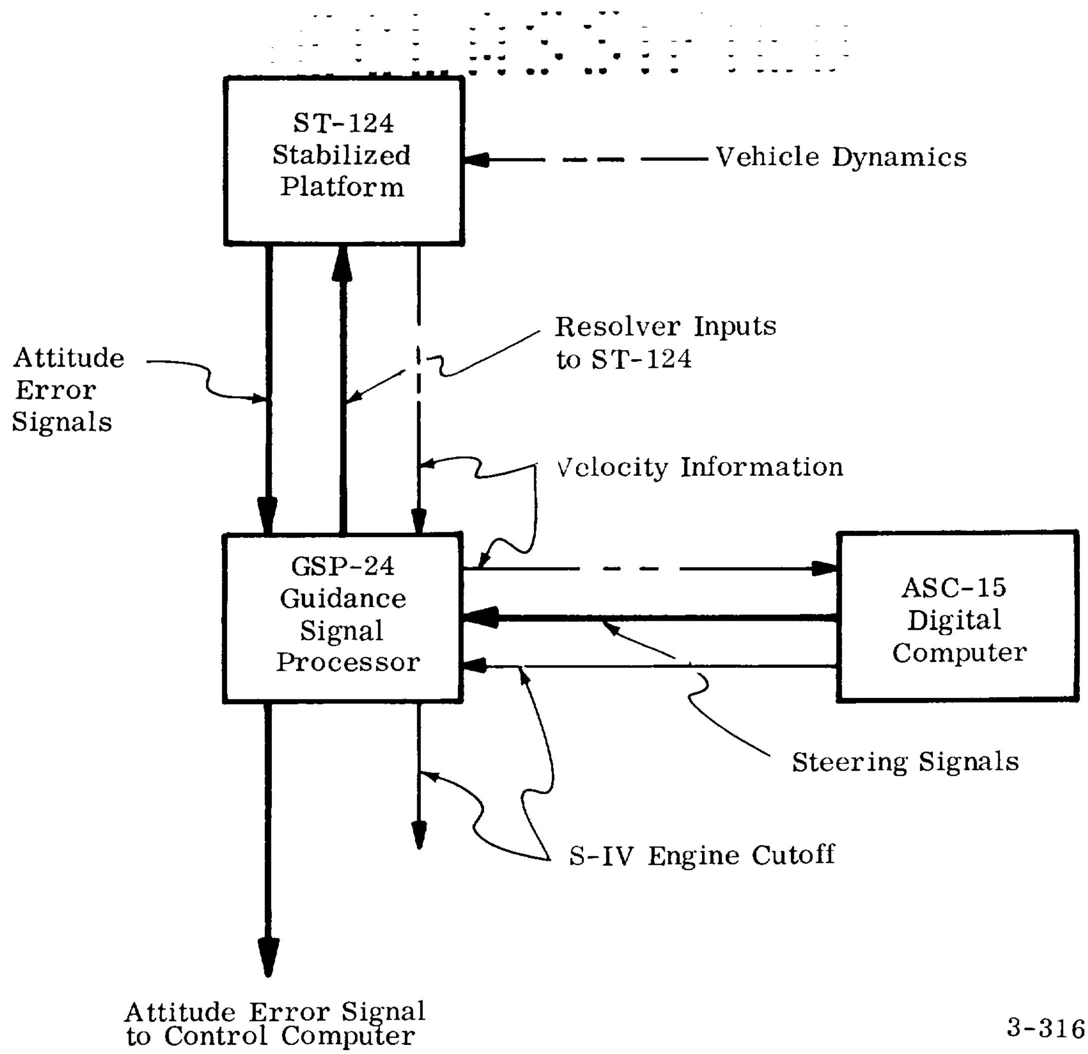


Figure 6-19. Guidance Implementation, Saturn I

chain with three command resolvers located in the guidance signal processor. This resolver chain converts space reference guidance commands to vehicle referenced steering commands, which are applied to the flight control computer.

During the alignment, the guidance and control system operates with ground equipment (RCA-110 computer) to accomplish platform alignment in the azimuth plane. The platform checkout module is contained in an electronics box which contains the circuitry essential for operation of the stabilized platform.

The ST-124 stabilized platform system is sealed so that operation in a near vacuum is possible. Auxiliary heaters in the platform system external cover provide preheating and inflight temperature control.

The gas bearings for platform gyroscopes and accelerometers are supplied with nitrogen by the gas bearing supply. This supply conditions the gas by controlling temperature, pressure and impurities.

The ST-124 characteristics are listed in Table 6-5.

Table 6-5. ST-124 Stabilized Platform Characteristics

Item	Characteristic
Physical Data	
Total weight of platform	147 lbs.
Size of platform	19-inch dia. sphere with a mounting ring
Number of gimbals	Four
Gimbal order (vehicle ref. to inner gimbal) and programming freedom	Pitch redundant (Z_r) (360°)
	Outer yaw (X) (360°)
	Middle pitch limited (Z) (20°)
	Inner Roll (Y) (360°)
Gimbal resolvers	Four, pancake type
Resolver chain accuracy	± 6 minutes of arc
Gimbal torquer voltage	56 vdc (quiescent - 8W) (Flt Oper. - 110W)
Maximum torque available	170 oz. in.
AC power	115 (± 1) vac, 3 phase, 400 (± 0.025) cps, 62W (sync)
DC power	± 28 vdc, 30W
Environmental Data	
Vibration	5g, 20-2000 cps
Shock	15g, 15 msec rise
Linear acceleration	10g
Warmup time from room temperature	30 minutes
Temperature limits for optimum accuracy	70 (± 10) $^\circ$F (ambient) 104 (± 10) $^\circ$F (mass)
Temperature degredation of accuracy	0° to 120° F ambient

6-46. ASC-15 DIGITAL COMPUTER.

During S-I stage flight the computer provides the source of programmed attitude changes for the attitude control and stabilization function and generates guidance steering signals during S-IV stage flight. The computer is a serial, binary, special purpose computer composed of five functional sections: storage, input, control, arithmetic, and output. The computer contains a magnetic storage drum which performs the following functions:

a. Supplies timing signals to all timing circuits
b. Supplies instructions to the control section
c. Supplies data to the arithmetic, control and output sections
d. Forms part of the arithmetic section shift registers
e. Stores input data and the results of arithmetic computations

Timing circuits, using the storage section as a reference, generate all the timing signals for the five sections of the digital guidance computer.

The input section accepts the following types of signals: inertial velocity components; launch constants; launch constant modifiers; program control signals; and discrete signals. The inertial velocity inputs are incremental and are continuously sampled and automatically accumulated. Attitude inputs are applied to the computer in serial form. The remaining inputs to the computer are applied to the input section in parallel form, converted to serial form by the input section, and stored in the storage section by a command from the control section.

The arithmetic section receives data from either the input section or the storage section, and after performing any mathematical operation defined by the control section, stores the result in the storage section. The arithmetic section performs five mathematical operations: addition of one number to another; subtraction of one number from another; comparison of one number with another; multiplication of one number by another; and conversion of gray code inputs (vehicle rates) to binary numbers.

The output section receives data from the storage section on command from the control section; this data is either converted to a proportional analog voltage and applied to the command resolvers in the guidance signal processor or transmitted to the ground equipment. Discrete commands are applied to the output section directly from the control section. The control section determines, by monitoring data from the storage section, when a discrete command is to be issued.

6-47. GSP-24 GUIDANCE SIGNAL PROCESSOR.

The guidance signal processor provides the interface between the digital computer and other guidance and control system components. The guidance signal processor is composed of:

a. Attitude command resolvers (including frequency sources, servos and demodulators

b. Telemetry register

c. Accelerometer signal shaper

d. Command and GSE switching networks

e. Accelerometer telemetry shaper

f. Power sequencing circuitry and power supply

The attitude command resolver chain is comprised of command resolvers located in the guidance signal processor and resolvers mounted on the stabilized platform. The command resolvers accept space-referenced steering commands from the digital computer, and through interaction with the platform mounted resolvers convert these commands into vehicle referenced attitude error signals. When the digital computer commands a change in command resolver positioning, an analog of the resolver rotor shaft position is fed back to the computer through an incremental encoder preventing the accumulation of long-term rate errors. Large surges of command values to the vehicle control system are restrained by the resolvers within a speed limitation of approximately one degree per second.

The majority of the auxiliary equipment for the resolver chain is located in the processor. Two frequency sources 1500 cps and 1800 cps are included. These are derived from the basic 400 cps voltage. The voltage is controlled because any error becomes a direct gain error in the over-all vehicle control loop.

The demodulators are phase and frequency sensitive, using the 1500 and 1800 cps sources as references. In one case a resolver output is demodulated in two demodulators: one demodulator, using the 1500 cps reference, demodulates this output to give the roll attitude errors; the other demodulator, using 1800 cps reference, gives the yaw attitude error. A third demodulator, using the 1500-cps reference, demodulates the output of another resolver to give the pitch output. All demodulators have a 3-volt/degree output which is accurate to within a small percentage over a range of $\pm$ 15 degrees.

Another resolver is mounted on the shaft of the pitch module to position the outer gimbal of the platform.

Telemetry of guidance functions is performed with the telemetry register. In the Saturn I launch vehicle, computer words are buffered and fed at 100 words/second to the processor. A command is applied to the telemeter each time one of the desired words passes through the register. When the telemeter command reaches the processor, the telemeter gate opens and the next word from the accumulator enters the shift register. The data is then available in parallel form to the PCM telemeter system during flight, and the GSE during prelaunch.

The accelerometer signal shapers convert sine-wave acceleration information received from optisyns into square-wave information for sampling by the digital guidance computer. The signal outputs from the accelerometer encoders (platform mounted) are sine-wave and cosine-wave signals which are applied to the accelerometer signal shapers. The signal shapers condition the signals into square waves of voltage displaced 90 degrees. The square waves are applied to the digital computer which processes the information contained in these signals to obtain steering signals.

The switching network selects the GSE or the command system as a source for loading the computer. This provides the capability of loading either while on the ground or while in a coast condition. In addition, it allows the ground or command system to control various modes of computer operation.

The accelerometer telemetry shapers receive signals from the accelerometer encoder shaper and condition them for telemetry. The square wave from the accelerometer shapers are given specific dc levels, added together and sent to the telemetry system as discrete levels between 0 and +5 volts.

The guidance signal processor power supply supplies all power required in the processor and supplies power for the encoders on the platform. In addition, all power to the computer power supply passes through the processor.

Since the drum of the computer utilizes 400-cps, two-phase power, it is necessary to convert the three-phase power available from the vehicle inverter to two-phase. This is done by a Scott connected transformer or similar device in the processor. Approximately 70 watts of 81.5 $\pm$2.5-volt, 400-cps, two-phase power is required by the drum. In addition, approximately 240 watts of 28 $\pm$ 2.0-volt dc passes through the processor to the dc-to-dc converter in the computer where the various levels necessary for computer operation are developed and regulated.

The guidance signal processor requires approximately 65 watts of 115-volt, 400-cps, single-phase power and 215 watts of 28-volt dc power.

6-48. FLIGHT CONTROL COMPUTER.

The analog flight control computer accepts signals from the stabilized platform, control accelerometers and actuator position feedback potentiometers. After performing signal filtering, shaping and mixing, the computer provides steering and control signals to the engine gimbal actuators. The major modules of the flight control computer are the servo amplifiers, filtering and shaping networks and the gain programmers.

The servo amplifier is a magnetic amplifier plug-in module used for signal mixing, scaling and polarity selection. The signal filtering and shaping networks provide signal conditioning based on the dynamic qualities of the vehicle. The gain programmer is a motor driven cam which positions a potentiometer to adjust the gain in each channel.

6-49. CONTROL SENSORS.

Two control accelerometers are used in the launch vehicle to measure lateral acceleration (perpendicular to the longitudinal axis) in the vehicle pitch and yaw planes. The outputs of the instruments are used by the control system to reduce structural loading and engine gimbal angle. The control accelerometer is a spring mass, fluid-damped accelerometer with an inductive pickoff. The range of the instrument is $\pm$10 meters per second per second.

6-50. ENGINE GIMBAL ACTUATORS.

Two linear, double-acting, equal area, electro-hydraulic servo actuators gimbal the engine in response to commands from the flight control computer. A feedback transducer mounted on each actuator transmits an electrical signal to the flight control computer which is proportional to the actuator position. (Refer to Paragraph 9-9.)

6-51. TRACKING

The Saturn I tracking function integrates vehicle-borne equipment with earth-based tracking facilities to obtain position and velocity information from Saturn I missions. Some of this information is analyzed and used in real-time decisions for mission control. However most of the information is recorded for post-flight evaluation of the mission. It will also be used to evaluate the operation of specific tracking systems and to improve tracking techniques, contributing to the ultimate goal of perfecting Apollo GOSS (Ground Operational Support System) to support the lunar mission.

The tracking function is active during launch, ascent and orbital phases of the Saturn I mission. Pulse radar, continuous-wave radio frequency, optical and infrared tracking systems located at earth-based tracking stations acquire information during these phases. Vehicle-borne transponders and a high-altitude radar altimeter aid in the tracking.

During launch phase, operational readiness of all tracking systems is determined by checkout. Reference data for each tracking system is obtained in the period just prior to liftoff. The position and velocity information obtained during the ascent phase is used to determine if the vehicle has proper trajectory and velocity. In addition, presentations based on tracking are monitored by the range safety officer to aid him in deciding whether to terminate vehicle flight to eliminate danger to personnel and property. Continuous tracking is also required during this phase to accurately determine the first and second stage engine cutoffs and separations, confirm that orbital conditions can be reached, and predict future positions of the vehicle. The future position information is used in transferring the tracking assignment from one tracking station to another so that acquisition can readily be obtained. Continuous tracking is also required for a short period after injection into orbit, to verify the orbit conditions. Thereafter, periodic tracking observations are required to confirm and refine the predicted positions and velocities.

To satisfy these requirements, tracking stations have been established at selected locations around the earth to ensure that vehicles can be tracked continuously by at least two stations or systems from launch to orbital injection and that orbiting vehicles will pass within line-of-sight of at least one of the stations on each revolution. In addition, several different tracking systems are used, to provide redundant tracking data.

For post-flight evaluation of the vehicle performance, the tracking information is compared with theoretically calculated information. From this comparison, and subsequent analysis, an insight is gained into the actual functioning of the vehicle systems in flight, and corrections may be determined for future missions.

6-52. OPERATION

6-53. Launch Phase. During the launch phase, all vehicle-borne and earth-based tracking systems and tracking support systems (computers, data links, data cabling, and relay networks) are checked out both statically and dynamically. Static testing performed includes checking out system assemblies (transmitters, receivers, etc.) and subassemblies (master oscillators, local oscillators, automatic frequency control loops, etc.) with portable or fixed test sets. A static test may also be a complete system test of interconnected system assemblies and subassemblies. The digital computers associated with the tracking systems are statically tested with test program tapes or a test program previously entered in the computer memory. Static test problems, target position and velocity analogs are used to test analog computers.

Dynamic testing, or system testing, consists of checking out complete tracking systems, including associated support systems. For example, a dynamic test of the AN/FPS-16 (pulse) radar requires that the radar lock on a distant fixed-radar reflector or prominent land target (hill, water tower, etc.) the exact location of which is known from survey data. The information obtained from the fixed target by the radar is in polar-coordinate form (target slant range, azimuth angle, and elevation angle). This information is converted to rectangular coordinates (with target distances in the form X_t, Y_t. H_t) by analog or digital computer, converted to digital form by an analog-to-digital converter, and then applied to a data link transmitter which forwards the information to one of the network control centers. The control center equipment demodulates the FM carrier to recover the digital information. The digital information is then converted to analog form by a digital-to-analog converter, and the resulting rectangular-coordinate information is applied to a horizontal and a vertical plotting board. The horizontal plotting board plots the X_t and Y_t information (earth plane), and the vertical plotting boards plots the H_t information (vertical plane).

By comparing the AN/FPS-16 target position information, presented as inked marks on the plotting boards, with the known position of the fixed target, presented on map overlays on the plotting boards, the control center personnel can ascertain the operational status of the AN/FPS-16 and its support systems.

The continuous wave (cw) tracking systems are dynamically tested in a similar manner, using a fixed-position test set to simulate a target. The test set responds to a tracking transmitter interrogation by transmitting a simulated doppler-frequency modulated carrier.

Aircraft equipped with transponders similar to the type utilized on the launch vehicle simulate the vehicle for more accurate tracking systems testing. Since the majority of the pulse radar systems (land- and sea-based) are of the monopulse type using MTI (Moving Target Indicator) tracking, the moving aircraft can more realistically simulate the characteristics of a moving vehicle than can a fixed test set. The moving aircraft is also a more realistic target for the cw tracking systems, since the cw systems acquire tracking information from the doppler resulting from moving targets.

When a network tracking station has completed static and dynamic testing of the tracking and tracking support systems, its systems are conditioned to standby (transmitters to passive-radiate). In the final minutes of the countdown, the systems that will track the initial portion of the trajectory, from liftoff through the ascent phase, are conditioned to activate (transmitters to active-radiate.) In this condition, the high-powered pulse radars are operated in the beacon mode at reduced power to interrogate associated vehicle-borne transponders. The radars lock on the resulting responses. The continuous-wave tracking systems (AZUSA, ODOP, and MISTRAM) lock on associated vehicle-borne transponder responses which also result from ground-based transmitter interrogations. After the tracking systems have attained lock-on they are ready to track the vehicle.

All launch phase initial tracking information (data for T = O) is acquired near the end of the phase, between engine ignition and vehicle liftoff. Cameras mounted on the launch pad photograph the vehicle engine area near the fire wall. Other cameras located around the pad photograph the exhaust flame. Additional flame information is obtained by cinespectrographs that photographically record the vehicle plume color spectrum. All of the launch phase information is employed in post-flight evaluation.

6-54. Ascent Phase. During the first few thousand feet of the phase, the most accurate data is provided by optical systems, including theodolites and camera systems. Cameras mounted on the launch pad monitor the launch vehicle engines. Around the pad, high-speed cameras (800-1000 frames per second) and still cameras film the exhaust flame of the vehicle for post-flight analysis. Ballistic cameras located along the Florida mainland and down range monitor the vehicle above 50 meters. Tracking theodolites located at the Cape and down range are operated in pairs to provide accurate information of vehicle position in space, changes in position with respect to time (velocity), and acceleration information. Optical trackers are used at many tracking stations to aid the narrow-beam pulse radars in obtaining the proper azimuth and elevation for acquiring the vehicle during the ascent and orbital phases. Infrared tracking systems at the Cape track the radiation of the vehicle plume. Radio frequency transmission is also employed. Vehicle-borne transponders reply to pulsed and continuous wave transmissions of ground rf systems to provide tracking information during ascent.

6-55. Orbital Phase. Tracking information during the orbital phase is obtained primarily through the radio frequency systems, utilizing earth-based tracking stations and vehicle-borne transponders. Tracking data is also obtained during orbit by photographic techniques, in which the orbiting vehicle's position is located relative to a known time base and to a background of stars whose angular positions from the recording station are accurately known. A radar altimeter aboard Saturn I vehicles provides supplemental data, primarily for orbital passes over areas where ground tracking facilities are not available.

6-56. IMPLEMENTATION

Transponders are carried aboard the Saturn I instrument unit and interface with earth-based radio frequency tracking systems which provide the position and velocity data for mission control and post-flight mission evaluation. Some of these systems are developmental or passenger items, proposed for operational status in the Saturn IB and V programs. Others are operational throughout the Saturn I program or will become operational at some point during the program. These systems include:

a. AZUSA
b. UDOP/ODOP
c. MISTRAM
d. Minitrack
e. C-Band Radar
f. Vehicle Radar Altimeter

The role of each system having components aboard the launch vehicle and its operational status, is described below. Earth-based components are also covered to clarify over-all system operation.

6-57. AZUSA Tracking System. This system provides real-time position and velocity information by determining successive trajectory positions of the vehicle through continuous comparison of phase differences between microwave signals transmitted to and received from a vehicle-borne transponder. These phase differences are a measure of two-direction cosines and a slant range to the vehicle. Frequency-controlled signals are transmitted from a ground transmitter to the transponder, which retransmits them to ground receivers where they are converted to a form usable for phase comparison. Two AZUSA tracking systems are located on the Atlantic Missile Range, Mark II at Cape Canaveral and Mark I at Grand Bahama Island. The AZUSA data are presented in Table 6-6.

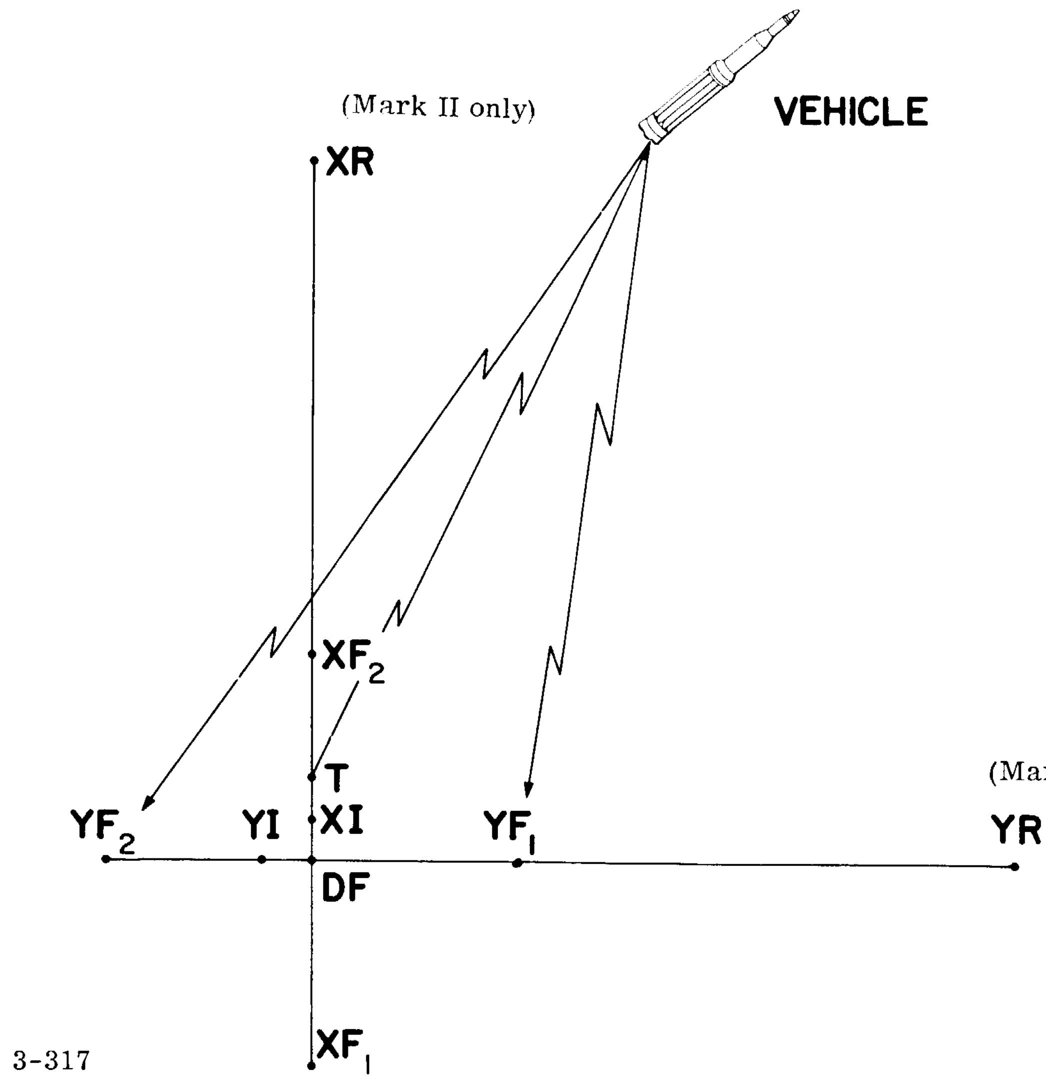
AZUSA Transponder. The AZUSA transponder is operational on the Saturn I program. It is a single-container unit, the air to ground link in the AZUSA tracking system. The primary function of the transponder is to accept an rf signal from the ground station, accurately reduce the signal's frequency by a fixed amount, and retransmit the reduced frequency signal to the ground station. The transponder also functions in a servo control loop which shifts the transmitter frequency to compensate for doppler shift, so that a constant

Table 6-6. AZUSA Data

Item	Data
Transponder (vehicle)	Receiver frequency: 5060.2 $\pm$0.75 mc Transmitter frequency: 5000 $\pm$0.75 mc Input signal level: -90 dbm to -12 dbm Power output: 2.5 watts
Ground Transmitter	Frequency: 5060.2 $\pm$0.75 mc Power: 2,000 watts Antenna type: parabolic Antenna polarization: adjustable vertical to horizontal Coverage: hemispherical to 2 deg elevation
Ground Receiver	Frequency: 5000 $\pm$0.75 mc Sensitivity: -125 dbw for MK I, -135 dbw for MK II Antenna type: parabolic Antenna gain: 33 db for MK I, 35 db for MK II Coverage: hemispherical to 2 deg elevation
Modulation	FM: 98.36 kc, 3.93 kc, 157 cps
Tracking Rates	Range: 30,000 fps Angles: 0.1 cos/sec

frequency difference is maintained between transmitted and received signals at the ground station. The nominal signal input frequency to the transponder is 5060.194 mc; the output frequency is 5000 mc. The AZUSA antenna is located on the exterior of the instrument unit, so oriented that the radiation pattern is predominantly in the aft direction. The single antenna unit is used for receiving and transmitting.

AZUSA Mark I. AZUSA MK I is a single-site, short-baseline tracking system with two baselines perpendicular at their midpoints, Figure 6-20. Each baseline has two cosine antenna pairs (fine and coarse) including a reference antenna common to both baselines. The direction from which a signal arrives is determined by comparing the phases of signals received at each antenna of a pair. The spacing of the cosine-antenna pairs is 80 wave-lengths for the coarse-



DF	Direction Finder Receiving
DF-XI	Intermediate Cosine (X) Receiving
$XF_1 - XF_2$	Fine Cosine (X) Receiving
$XF_1 - X$	Rate Cosine (X) Receiving (Mark II only)
T	Transmitter
DF-YI	Intermediate Cosine (Y) Receiving
YF_2-YF_1	Fine Cosine (Y) Receiving
YF_2-YR	Rate Cosine (Y) Receiving (Mark II only)

Figure 6-20. Azusa Antenna Baselines

cosine pair and 800 wave-lengths for the fine-cosine pair. This allows coarse-cosine data to resolve phase-counting ambiguities in the fine-cosine data.

AZUSA Mark II. The AZUSA MK II system is similar to the MK I system in operation. The main differences between the two systems are the refinement of circuitry design in the MK II and the addition of cosine-rate baselines which give more realistic direction-cosine data. Transmitter output is the same, but the antenna configuration is modified.

The MK II antenna configuration, Figure 6-20, consists of three antenna pairs on each of the two intersecting baselines. These pairs are spaced at 5, 50 and 500 meters. The 50-meter and 500-meter pairs of each baseline have one antenna in common. A conical scan antenna acts as reference for the 5-meter pairs on both baselines. In this configuration, the system incorporates nine receiving antennas and one transmitting antenna. The conical scan antenna, with a tracking capability of 360 degrees in azimuth and 85 degrees in elevation, furnishes ambiguity resolution for the 5-meter baselines. In turn the 5-meter baselines resolve ambiguity for the precision 50-meter baselines. The 500-meter baselines supply information for computing cosine rate data. The conical scan antenna, located at the intersection of the baselines, resolves the coarse ambiguities of the baselines and provides pointing information for the other antennas.

Slant range is determined from the energy received from the transmission of a C-band carrier modulated by a set of ranging frequencies. The carrier is shifted 60 mc by the vehicle transponder and reradiated to the ground. Specifically, the transponder receives and demodulates the frequency-modulated carrier transmitted from the ground station. The resulting signal modulates the transmitter portion of the transponder. The separation frequency between the transponder receiver and transmitter in noncoherent models is about 60 mc. In coherent models, the separation frequency is approximately 60.2 mc. The 0.2 mc difference between the two models exists because, in the coherent model, the frequency difference between input and output RF is phase locked to a multiple of the fine modulation frequency. This frequency difference can be duplicated by the ground station. Since the frequency received at the ground station is held constant by varying the transmitter frequency to compensate for doppler effect, the frequency difference between ground station transmitter and receivers can be measured and compared to the same multiple of modulation used in the transponder to measure range. Slant range is determined at the ground station from comparison of the phases of transmitted modulating frequencies and frequencies received from the transponder. Several modulation frequencies are used, thereby limiting phase-counting ambiguities.

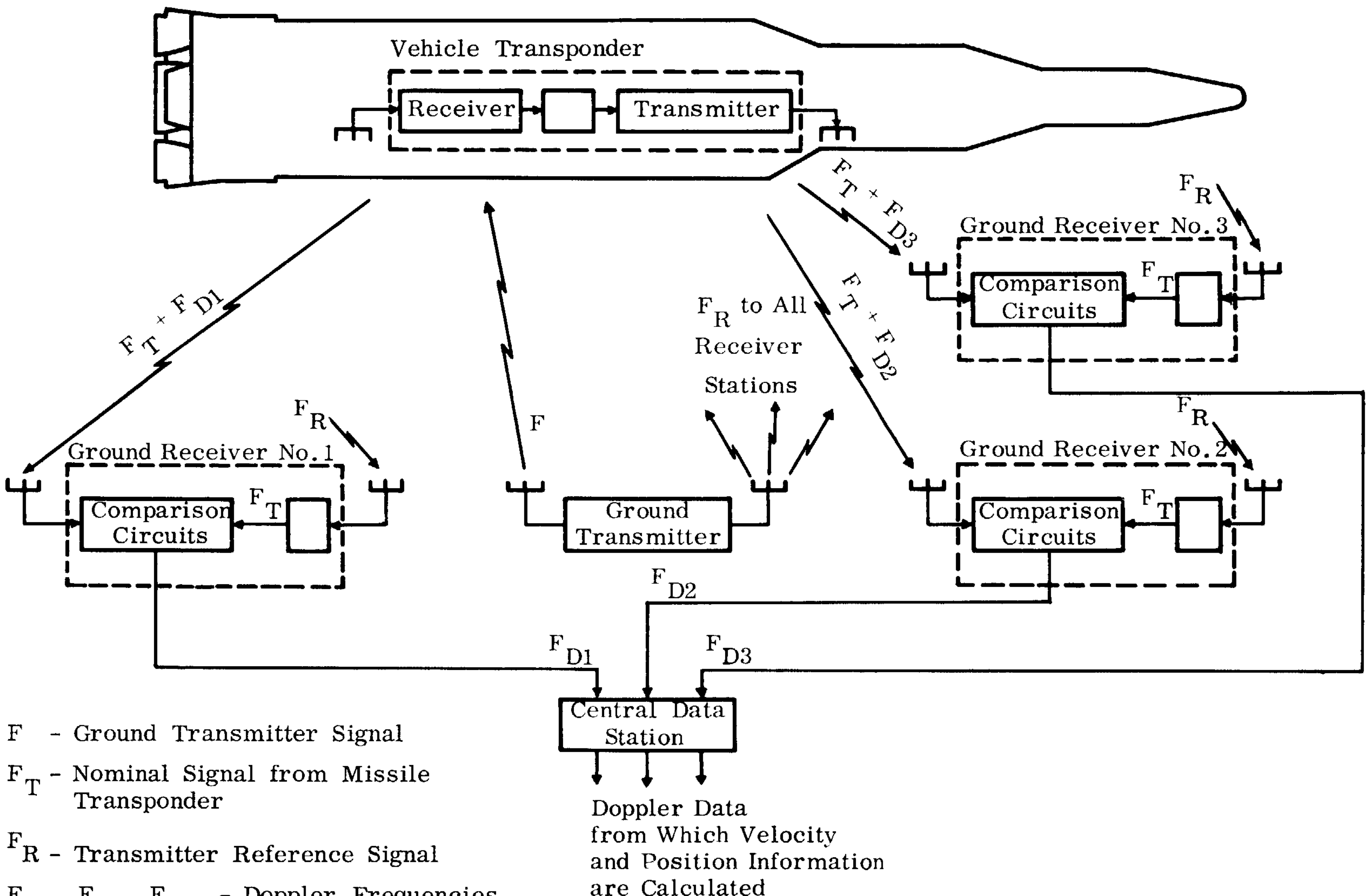
Incremental range is measured by "beating" the transmitted rf carrier with the receiver carrier to acquire a vernier-range reading. This is accomplished by phase-locking the vehicle-borne transponder transmitter to the ground transmitter. The incremental range reading is used as a vernier on the non-ambigious range data as well as to compute the radial velocity.

The system supplies continuous trajectory data in digital form to an IBM 709 Computer. The computer solves equations to derive the position coordinates of the trajectory. These data are presented as continuous plots for range safety purposes.

6-58. UDOP and ODOP. UDOP (ultra high frequency doppler velocity and position) is an operational tracking system on vehicles SA 5, 6 and 7. ODOP (offset doppler) is a passenger system on vehicles SA 6 and 7, replacing UDOP as an operational system on vehicles SA 8, 9 and 10. Primary difference in the two systems is that the UDOP is capable of transmitting a continuous-wave (cw) frequency double that received, while the ODOP transponder offsets the received frequency a fixed amount for retransmission. The UDOP nominal input - output frequencies are 450 and 900 mc and the ODOP nominal input - output frequencies are 890 and 960 mc. The increased ground to vehicle transmission frequency in ODOP results in a reduction of range error in the system. Because the two systems are similar, only the ODOP is discussed here.

The ODOP (Offset doppler) tracking system, Figure 6-21, uses frequency comparison techniques to determine velocity and position of the launch vehicle. The tracking data is recorded for subsequent analysis.

The system consists of a ground-based transmitter, a vehicle transponder, four or more ground receiving stations and a central recording station. The transmitter



F - Ground Transmitter Signal

F_T - Nominal Signal from Missile Transponder

F_R - Transmitter Reference Signal

F_{D1}, F_{D2}, F_{D3} - Doppler Frequencies

3-409A

Figure 6-21. ODOP Tracking System

sends an 890 mc signal to the transponder and a phase-coherent reference frequency signal to the ground receivers. The vehicle transponder receives the 890 mc signal, which, due to doppler effect, has been shifted in frequency an amount proportional to the radial velocity of the vehicle with respect to the transmitter. The transponder shifts the frequency of the signal and transmits the resulting signal to the ground stations. The frequency detected by each ground receiver has been subjected to an additional doppler shift (return trip doppler) which is proportional to the radial velocity of the vehicle with respect to the receiving site.

The received signal is compared with a reference and a difference frequency is produced as an output signal. This signal is sent to the central recording station via a data transmission link. At the central recording station, the doppler frequency is converted into a cycle count. The cycle counts from at least four receiver sites are translated into a data transmission format for recording and for transmission to the data handling center. The position of the vehicle is determined from a combination of a known initial position and the range sum. Range sum is defined as the total distance from transmitter to vehicle to receiver and is obtained from the accumulated cycle count. The known initial position is determined from a survey of the launch site or from a position pin-pointed by other range instrumentation systems after launch. Each range sum describes an ellipsoid, the focal points of which are represented by the transmitter and one of the receivers. The vehicle location is at the intersection of at least three such ellipsoids, as determined from the data received from three ground receiving stations. Data received at the remaining ground receiving station is used to validate the tracking measurements.

6-59. MISTRAM. The MISTRAM system uses continuous wave (cw) phase comparison techniques to measure range from a central station, and range difference across orthogonal baselines. Range is measured by counting the number of wavelengths traveled by the signal to the vehicle and back to the central station. Range difference is measured by counting the difference of the number of wavelengths traveled by the signals from the vehicle to each end of the baselines. The final data available from MISTRAM are range and range differences. Vehicle position is then fixed by the range and range differences. An external computer is used to compute trajectory and the rates at which the range and range differences are varying to determine velocity.

MISTRAM Airborne Transponder. The MISTRAM Airborne Transponder is carried as passenger equipment aboard Saturn I vehicles SA 5 through SA 10. It receives two continuous wave X-band signals (range and calibration channels) from the ground-based antenna. These signals are amplified, frequency shifted and retransmitted to the ground where they are used in obtaining vehicle position and velocity data. The retransmitted signals are phase-locked to the signals received by the transponder.

MISTRAM Earth Stations. MISTRAM I central and remote stations, Figure 6-22, are arranged in an L-configuration. The installation, located at Valkaria,

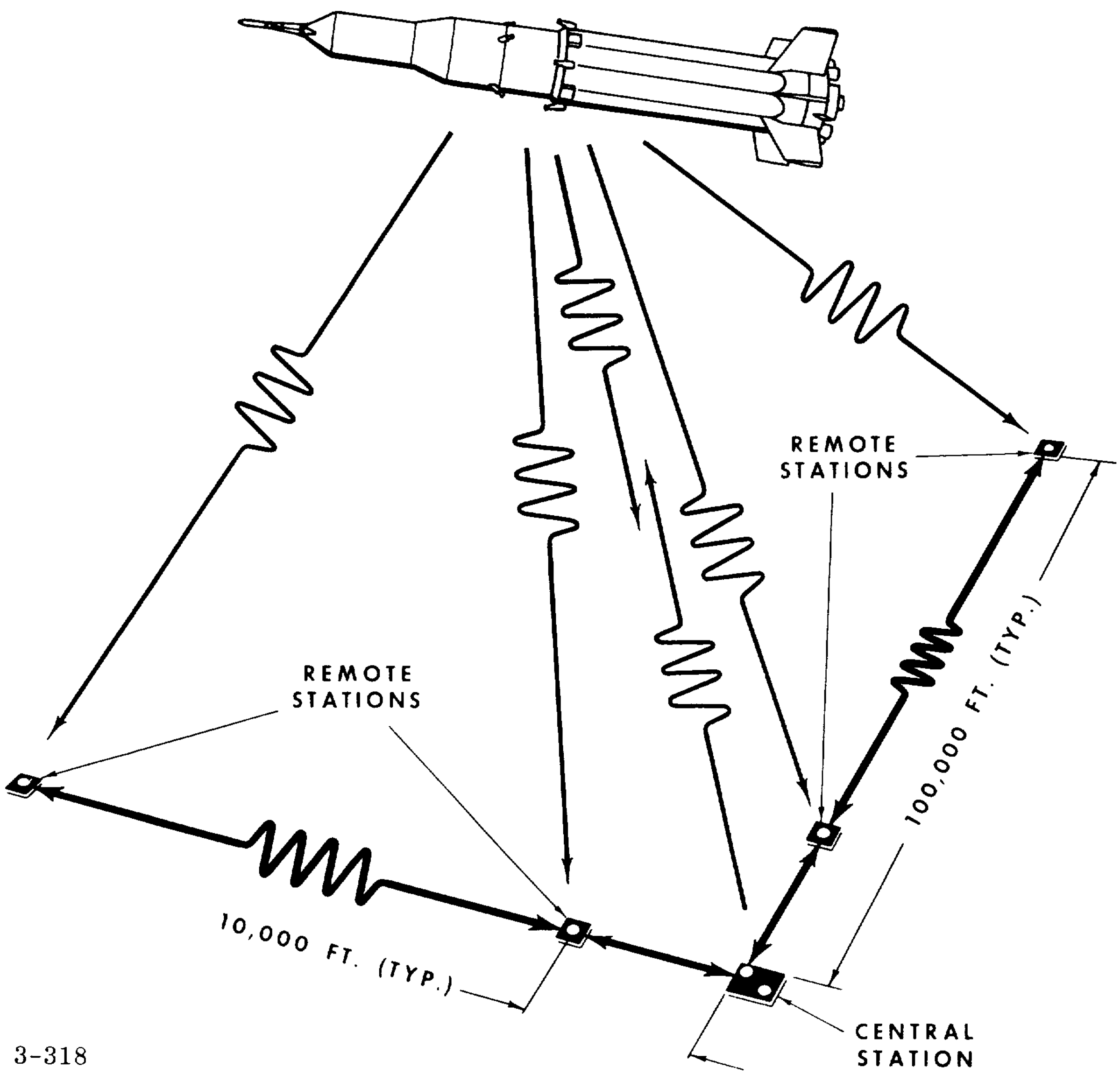


Figure 6-22. MISTRAM Ground Station Configuration

Florida, consists of a central station at the vertex of the L and four remote stations spaced along the baselines of the L at 10,000-ft. and 100,000-ft. distances. The 10,000-ft. stations are connected with the central station by 3-in. diameter circular waveguides and the 100,000-ft. stations by airlink transmissions. MISTRAM II, Eleuthera, is essentially the same as MISTRAM I, but does not have the two 10,000-ft. stations. Both systems have microwave antenna towers located at the vertex and the two extremities of the long baselines. MISTRAM data are listed in Table 6-7.

Table 6-7. MISTRAM Data

Item	Data
Vehicle-borne Transponder	(RT612/DRS-3)
Size	5.4 x 8.9 x 12.1 inches
Weight	16.5 lbs.
Power Consumption	5.3 Amps max. at 25.2 - 32.2 VDC
RF Power Output	200 - 500 milliwatts per channel
Operating Frequencies (nominal)	
Received	Range Channel - 8148 mc Calibration Channel - 7884 to 7892 mc (Swept)
Transmitted	Range Channel - 8216 mc Calibration Channel - 7952 to 7960 mc (Swept)
Phase Coherence	Less than 45° error between the transmitted and received 256 mc difference frequencies Less than 2° error between the end frequencies of the calibration channel sweep
Dynamic Range	Minus 30 to minus 105 dbm
System Coverage	
Azimuth	360 deg
Elevation	5 to 85 deg full accuracy 0 to 85 deg decreased accuracy (The full accuracy coverage is limited by the elevation angle from any one antenna.)
Range	20 to 600 nm full accuracy 20 to 1000 + nm decreased accuracy
Range Velocity	0 to 50,000 fps
Range Acceleration	0 to 750 fps^2

Table 6-7. MISTRAM Data (Cont'd)

Item	Data
Rate of Change of Range Acceleration	0 to 50 fps^3
Range Difference Velocity	0 to 3,000 fps
Azimuth and Elevation Tracking Rate	0 to 45 deg/sec
Azimuth and Elevation Acceleration	0 to 250 deg/sec^2
System Accuracy	Maximum Error: Range - 0.40 ft Range Difference - 0.03 ft Range Rate - 0.02 fps Range Rate Difference - 0.002 fps

6-60. Minitrack. Minitrack is a continuous-wave radio frequency system which determines angular direction to the vehicle by interferometer techniques. It consists of a vehicle-borne beacon, tracked by a world-wide network of stations arranged such that at least one station is in line-of-sight of the vehicle on each orbit. The stations are listed below.

Fairbanks, Alaska	Lima, Peru
Goldstone, California	Antofagasta, Chile
San Diego, California	Santiago, Chile
East Grand Forks, Minn.	Antigua Island, British West Indies
Blossom Point, Maryland	St. Johns, Newfoundland
Rosman, North Carolina	Winkfield, England
Fort Myers, Florida	Johannesburg, South Africa
Quito, Ecuador	Woomera, Australia

The Minitrack beacon, carried aboard the Saturn I instrument unit, radiates at a frequency of 139.995 mc, with an output power of 20 milliwatts. The beacon may be modulated for telemetry purposes.

Each Minitrack station has an antenna pattern on crossed baselines (similar to AZUSA). A direction cosine with respect to each baseline is computed from measurement of phase-difference in the reception of radio frequency energy at separated antennas along the baseline. Each station computes two direction cosines, with respect to its space-fixed antenna baselines, as a function of time. The vehicle orbit is computed from angle measurements made at a series of ground stations.

6-61. C-Band Radar Tracking. A C-band transponder (SST-102A) is operational on Saturn vehicles SA 5 through 10. The SST-102A transponder functions with earth-based AN/FPS-16 radar sets to provide accurate tracking data on the vehicle trajectory.

The AN/FPS-16 is a high-precision, C-band, monopulse tracking radar designed specifically for long-range tracking. The monopulse radar derives target position information from each returned signal, instead of using several pulses as is necessary in lobing-type radars.

C-Band Transponder. The transponder (or radar beacon) provides transmission of high-energy radar pulses in response to uncoded or coded (single or double) pulse interrogations from the earth-based radar. Its use ensures a point-source of return energy to the radar set, thus increasing tracking accuracy by eliminating the uncertainty of point on vehicle being tracked. The transponder is a compact single-package receiver-transmitter and power supply operating in the 5400-5900 mc range. Its receiver sensitivity is minus 70 dbm. Output power of the transponder is a minimum of 500 watts (peak).

C-Band Radar. On the AN/FPS-16, a fixed-frequency magnetron transmitter produces a peak power out of one megawatt. (This power may be reduced for tracking close-in targets.) The transmitted energy is radiated by a four-feedhorn array which feeds a parabolic reflector to produce a very narrow beam. The transmitted signal may be either single pulse for skin track or coded pulse for beacon track. Target return rf energy is received by the four-feedhorn array and applied to an rf comparator which, by addition of the energy received from selected pairs of feedhorns (horizontal and vertical), develops azimuth and elevation error signals. These error signals represent target displacement from the beam centerline. In addition, the outputs from all four feed horns are summed, deriving a reference signal. Each signal is applied to a separate tracking section where it is converted to a 30 mc IF signal, amplified, and compared with the reference signal. The phase relationship represents the error direction and the amplitude represents the error magnitude. The resulting error direction and magnitude signals are detected and commutated, and in turn, used to control the antenna positioning servos.

One reference signal is applied to the range tracking section where it is used in generating the ranging voltages. The ranging voltages are ultimately used to gate the receiver channels so that they are receptive only to targets being tracked. The range section provides the slant range analogs for the digital section and the video presentation console.

The outputs from the AN/FPS-16 (polar coordinates) are in gray-code serial-binary form.

The data for the AN/FPS-16 radar system and the SST-102A transponder are listed in Tables 6-8 and 6-9, respectively.

Table 6-8. AN/FPS-16 Data

Item	Data
Transmitter	Frequency Fixed - 5480 $\pm$30 mc Tunable - 5400 to 5900 mc
	Peak Power Fixed frequency - 0.7 to 1.3 Mw Tunable frequency - 0.2 to 0.4 Mw
	Pulse Width - 0.25, 0.5, 1.0 usec Pulse Rate - 71, 80, 142, 160, 285, 320, 341, 366, 640 (for XN-1 delete 80 add 233)
Receiver	Frequency - 5400 to 5900 mc Noise figure - 11 db.
Antenna	Type - 13 ft. parabolic reflector Gain - 43.5 db (nominal) Scan - Circle and sector Polarization - vertical Beam Width - 1.2°

Table 6-8. AN/FPS-16 Data (Cont'd)

Item	Data
Coverage	Azimuth - 360^{o} Elevation - minus 10^{o} to 190^{o} Range - 1000 nm Accuracy - ± 5 yards range ± 0.2 milliradian angle
Tracking rates and accelerations	Azimuth - 750 mil/sec 550 mil/sec^2 Elevation - 400 mil/sec 350 mil/sec^2 Range - 8000 yd/sec 2000 yd/sec^2

Table 6-9. SST-102A C-Band Transponder Data

Item	Data
Frequency Range	5400 to 5900 mc.
Frequency Stability	± 2.0 mc.
IF Frequency	60 mc.
Receiver Sensitivity	-70 dbm over entire frequency range
Receiver Bandwidth	10 mc.
Transmitter Pulse Width (50% Amplitude)	0.25 ± 0.05 0.75 ± 0.05 (selectable)
Interrogation Rate	0 to 4000 pps.
Pulse Rise Time (10% to 90%)	0.10 sec. max.
Pulse Delay	2.0 ± 0.1 sec.
Peak Power Output	500 watts, min.
Supply Voltage	28 v.d.c. nominal
Operating Range	5395 to 5905 mc.
Supply Current	1.9 amps.

6-62. Radar Altimeter. The Saturn high-altitude altimeter, Figure 6-23, has been developed for onboard instrumentation to supply tracking data for vehicle trajectories not completely covered by earth-based tracking stations (e.g. over long stretches of ocean). The altimeter determines range from vehicle to earth by accurate measurement of the time interval between its transmitted pulse and the return echo. This range information is digitally encoded and transmitted through the vehicle telemetry link to ground receiving stations for support of the tracking function.

The heat of the altimeter is a stable crystal oscillator which controls the radar pulse repetition rate and supplies timing intervals for the counting circuit. Transmission of the radar pulse gates the counter "on;" reception of the return pulse gates the counter "off." The number of counts between each pulse and its return represents a number of timing intervals which is analogous to vehicle altitude.

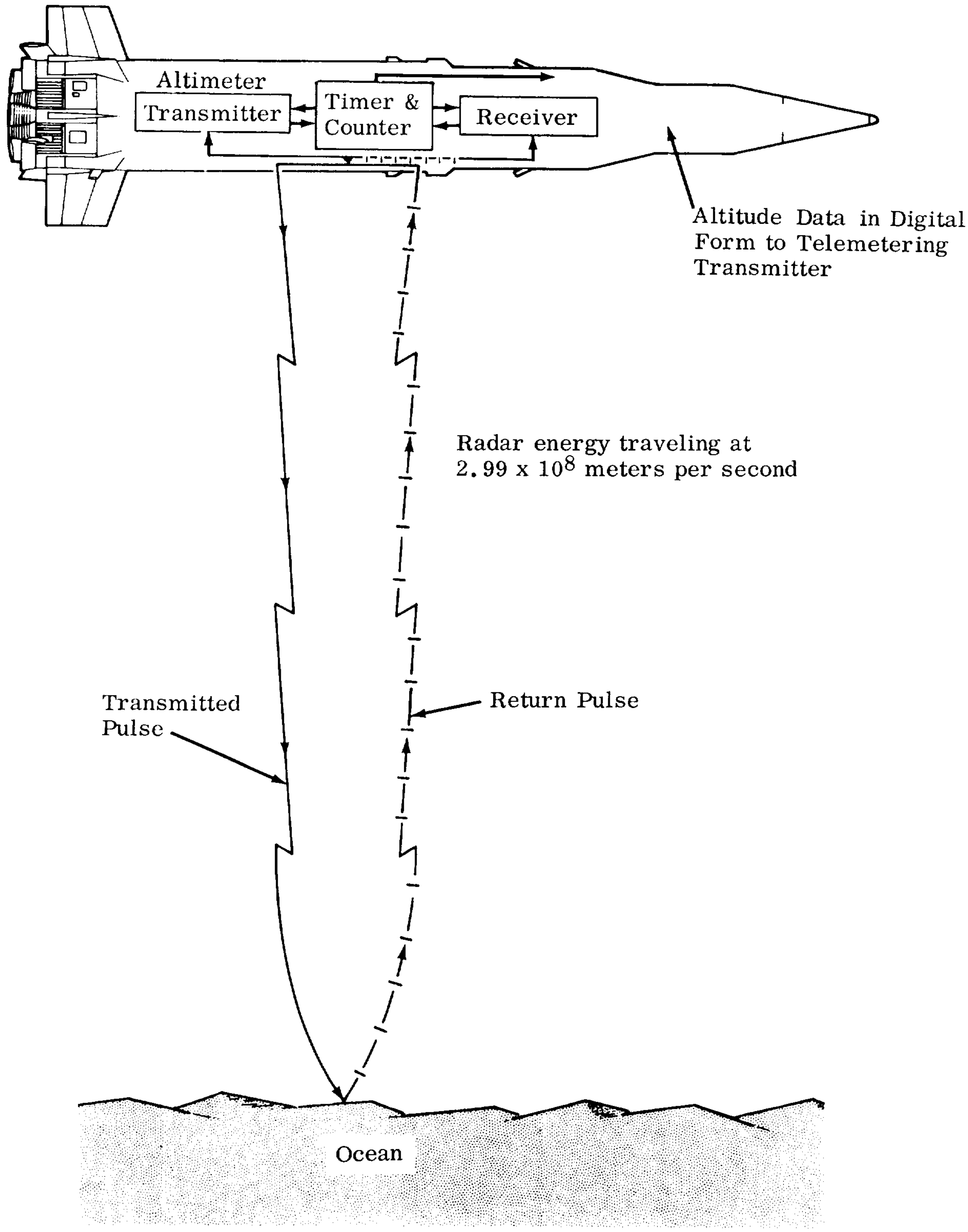
The altimeter operates at a frequency of 1610 mc. A single antenna (Model 502) serves both transmitting and receiving functions and is mounted on the exterior of the instrument unit.

The altimeter is passenger equipment for Saturn I vehicles SA 5, 6 and 7 and operational for SA 8, 9, and 10.

6-63. Tracking Network. The Saturn I tracking function is controlled by a network developed from the Mercury network and Atlantic Missile Range facilities. The network encompasses tracking stations located around the earth between 35 degree North latitude and 35 degree South latitude. This tracking network, providing integrated coverage for flight azimuths of 72 to 105 degrees, includes control centers, fixed land-based tracking stations, and tracking ships which fill the gaps between the land-based stations.

Tracking Stations. The network is implemented with radio frequency, optical and infrared tracking systems. The types and locations of the tracking systems are listed in Table 6-10.

Located along the 72-degree azimuth launch orbit are tracking stations at Cape Kennedy, Bermuda Island, Grand Canary Island, Kano (Nigeria), and Zanzibar Island (off the east coast of Tanganyika, Africa). To obtain a continuous track,



3-319

Figure 6-23. Radar Altimeter

Table 6-10. Tracking Stations and Systems

Stations		Radar				Continuous Wave	
No.	Name	L-Band	S-Band	C-Band	X-Band	Model	Freq. (mc)
	72-Degree Azimuth						
M-1	Cape Kennedy	AN/FPS-8	MOD II	AN/FPS-16	Mod IV	AZUSA (MK II)	C-Band
						GLOTRAC	C-Band
						ODOP	900
M-2	Bermuda			AN/FPS-16		GLOTRAC	C-Band
M-3	(Mid-Atlantic Ship)						
M-4	Grand Canary Is.						
M-5	Kano, Nigeria						
M-6	Zanzibar, E. Africa						
M-7	Indian Ocean Ship[1]						
M-8	Muchea, Australia						
M-9	Woomera, Australia			AN/FPS-16			
M-10							
M-11	Canton Is.						
M-12	Kauai Is., Hawaii			AN/FPS-16			

Table 6-10. Tracking Stations and Systems (Cont'd)

Stations		Radar				Continuous Wave	
No.	Name	L-Band	S-Band	C-Band	X-Band	Model	Freq. (mc)
	72-Degree Azimuth						
M-13	Point Arguello, California			AN/FPS-16			
M-14	Guaymas, Mex.						
M-15	White Sands, New Mexico			AN/FPS-16			
M-16	Corpus Christi, Texas						
M-17	Eglin, Florida			AN/FPS-16	AN/MPQ-31		
3	Downrange						
3	Grand Bahama Is.		Mod II	AN/FPS-16		AZUSA (MK II)	C-Band
						GLOTRAC	C-Band
						ODOP	900
4	Eleuthera Is.					MISTRAM	X-Band
5	San Salvador Is.		Mod II	AN/FPS-16		GLOTRAC	C-Band
7	Grand Turk Is.		Mod II			GLOTRAC	C-Band

Table 6-10. Tracking Stations and Systems (Cont'd)

Stations		Radar				Continuous Wave	
No.	Name	L-Band	S-Band	C-Band	X-Band	Model	Freq. (mc)
9.1	Antigua Is.			AN/TPQ-18		GLOTRAC	C-Band
12	Ascension Is.		Mod II	AN/FPS-16			
13	Pretoria, South Africa			AN/MPS-25			
	Advanced Research Instrumentation Ships (ARIS)						
	Gen. H. H. Arnold	(L-Band)		(C-Band)	(X-Band)		
	Gen. Hoyt Vanderberg	(L-Band)		(C-Band)	(X-Band)		
	Range Instrumentation Ship						
	Twin Falls Victory			AN/FPS-16			

a tracking ship is stationed between Bermuda Island and Grand Canary Island, and two tracking ships (Advanced Range Instrumentation Ships - ARIS) are stationed in the Indian Ocean.

Down range from Cape Kennedy, tracking stations are located at Grand Bahama Island, Eleuthera Island, San Salvador Island and Grand Turk Island all of which are in the Bahama Islands; Antigua Island in the Leeward Islands; Ascension Island off the west coast of Africa; and Pretoria, South Africa. The downrange stations are primarily used in tracking the 105-degree azimuth launch orbit, and provide additional tracking for the ascent phase of the 72-degree launches. Each station provides tracking whenever the vehicle is within its area of coverage.

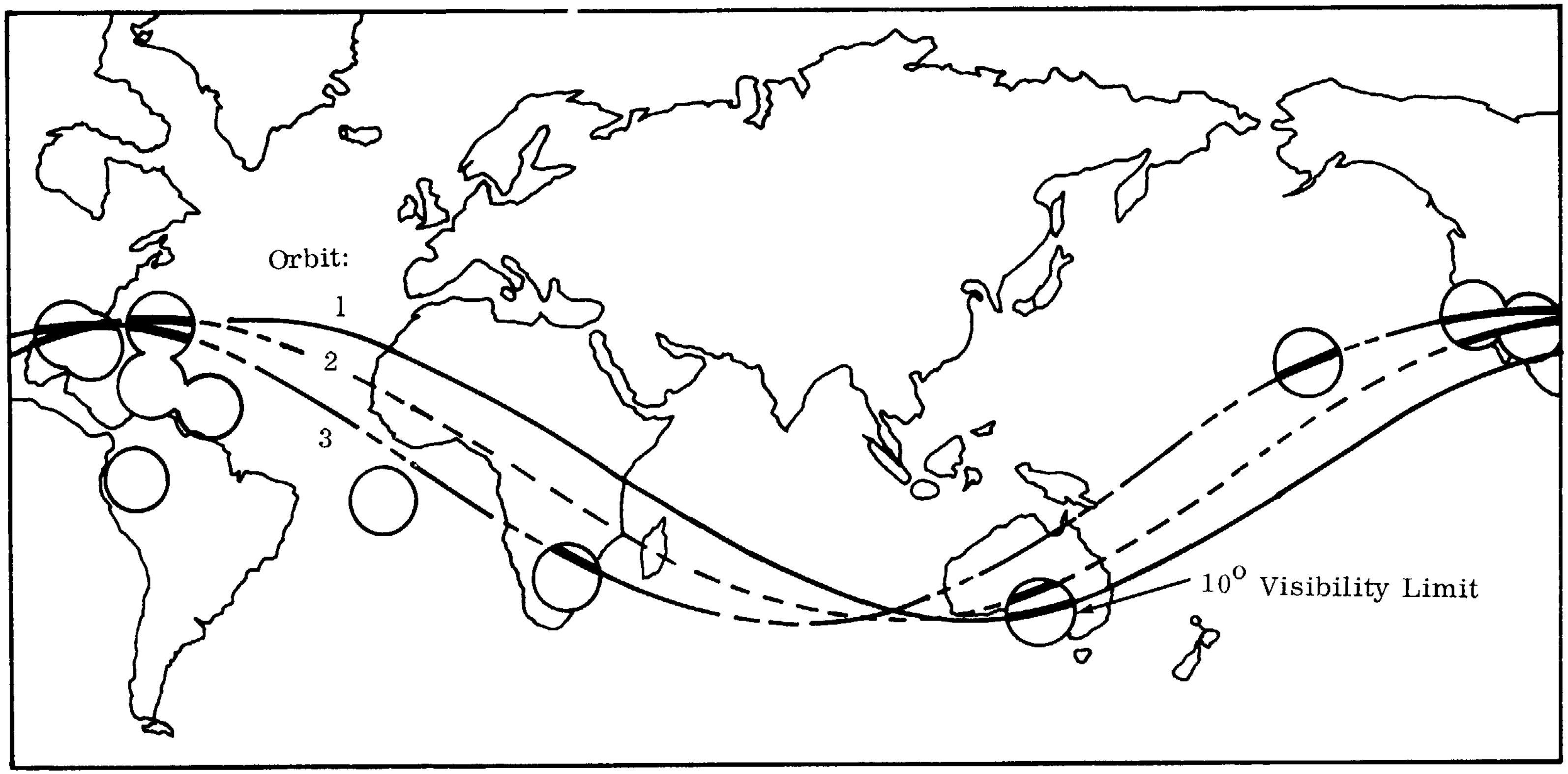
From 90-degree East longitude, the network extends through the Pacific area to the western United States mainland, and across the southern part of the United States to Cape Kennedy. For this portion of the network, tracking stations are located at Muchea and Woomera, Australia; Canton Island; Kauai Island, Hawaii; Point Arguello, California; Guaymas, Mexico; Corpus Christi, Texas; and Eglin, Florida. Tracking coverages of three orbits for 72-degree and 105-degree flight azimuths are illustrated in Figures 6-24 and 6-25, respectively.

6-64. RANGE SAFETY.

The range safety function ensures safety of the launch range (AMR) and adjacent areas against malfunction of vehicles launched on the range.

The function is of extreme importance during the early part of the flight, diminishing in importance (or criticality) as a function of flight time (or distance traveled down range) until, on attainment of orbital conditions, the range safety function can be commanded "safe." Thus, the function is used only during the ascent phase, although operational readiness is determined by checkout of the function during the prelaunch phase.

When applied, the range safety function results in termination of power (thrust) and, by an additional command, dispersion of propellants to preclude explosion and fire damage upon impact of the vehicle.



3-320

Figure 6-24. Orbital Path, 72 Degree Azimuth

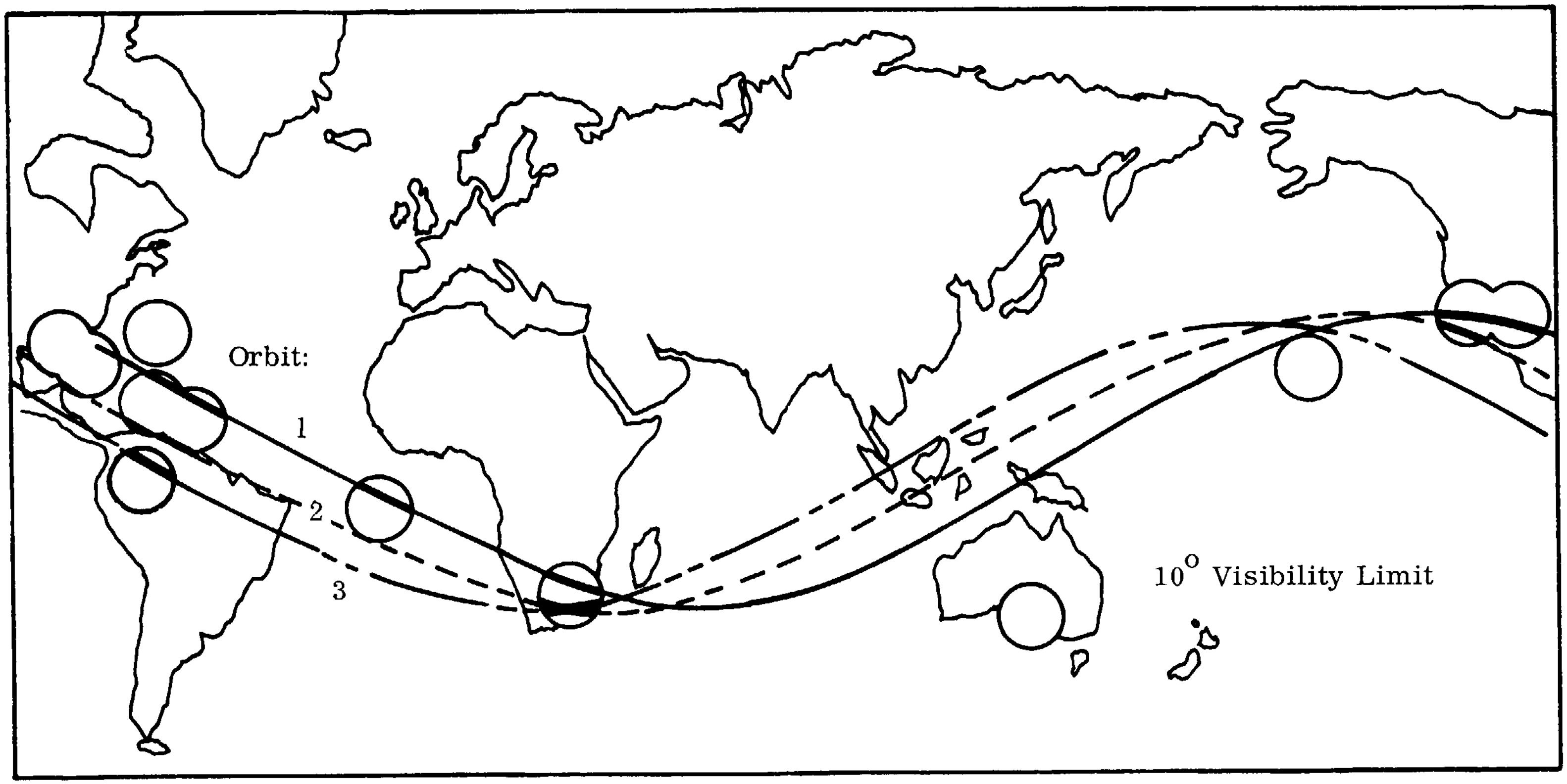


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Figure 6-25. Orbital Path, 105 Degree Azimuth

6-65. OPERATION.

Range safety is accomplished by integrating related functions, including tracking, instrumentation, command, communications, and range surveillance. The range safety officer has control of switches which, through radio transmission, command (1) vehicle engine cutoff and (2) initiation of ordnance elements aboard the vehicle to release propellants, after fuel flow to the engines has been cutoff. He initiates the first, and, if necessary, the second of these commands whenever, in his opinion, further flight of the vehicle constitutes a danger to life or property on or adjacent to the range. A time delay in vehicle-borne equipment delays arming of ordnance-initiation circuits for a short time after receipt of the engine cutoff command.

To aid the range safety officer in making his decision, presentations of information from the tracking, instrumentation and range surveillance functions are displayed on his control console. This information includes: Traces of vehicle present position in three coordinate planes; a trace showing ballistic impact point if thrust were terminated at that instant; selected telemetry data of vehicle performance; a manual plot showing locations of air and ship traffic in the range area; and a television presentation of the vehicle while in visual range.

6-66. Present Position Displays. Present position of the vehicle as obtained from tracking information, is resolved into three coordinate planes. Charts of the vehicle trajectory projection are plotted mechanically on these planes. The charts for trajectory plotting show the nominal, or expected trajectory and a family, or families of range safety curves.

The range safety curves in each plane are developed by determining a direction for each of many points on the plane which would permit the vehicle to impact on an established limit line if thrust were terminated at that instant, Figure 6-26. Limit lines are established to ensure that a vehicle does not impact on inhabited areas, and flight termination is indicated when the vehicle trajectory (in a given plane) parallels an adjacent range safety curve. Figure 6-27 is a representation of a nominal trajectory and its projections on the three planes.

6-67. Ballistic Impact Point. From present vehicle position and velocity information, a computation is made of ballistic impact point (or instantaneous impact point), if thrust were terminated at that instant. This information is presented

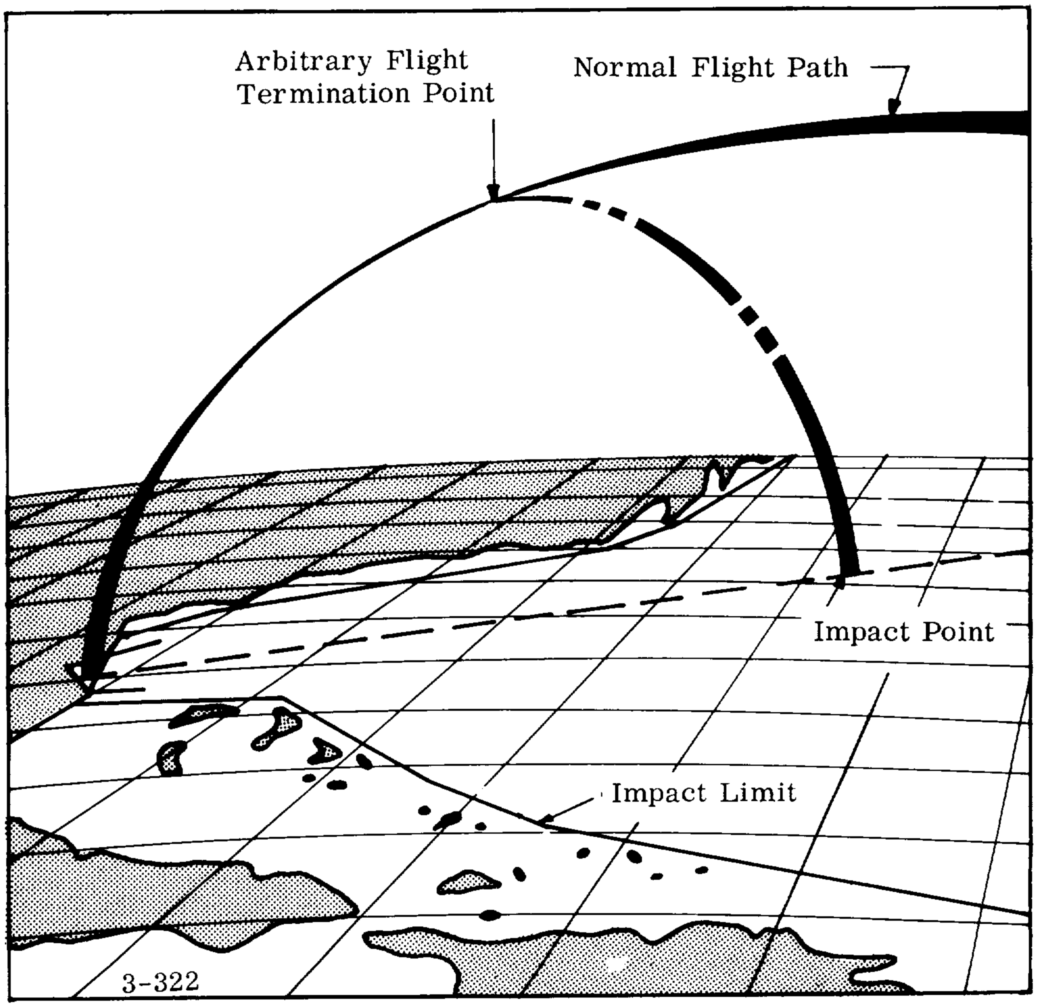


Figure 6-26. Range Safety Limits

continuously, as a trace on a chart which shows the corridor through which the instantaneous impact points must pass. An indication that the vehicle would violate the boundary of this corridor if powered flight were continued is the basis for a flight termination decision. Figure 6-28 illustrates plots of the type used.

6-68. Range Surveillance Data. Position of all ships and aircraft in the range area are plotted manually on a plexiglass plotting board that can be seen by the range safety officer. Plotted data is derived from air and surface surveillance radar information and, for the nearby sea area, from visual surveillance by observers in the lighthouse at Cape Kennedy.

6-69. Television Presentation. During the early moments of the vehicle flight, it

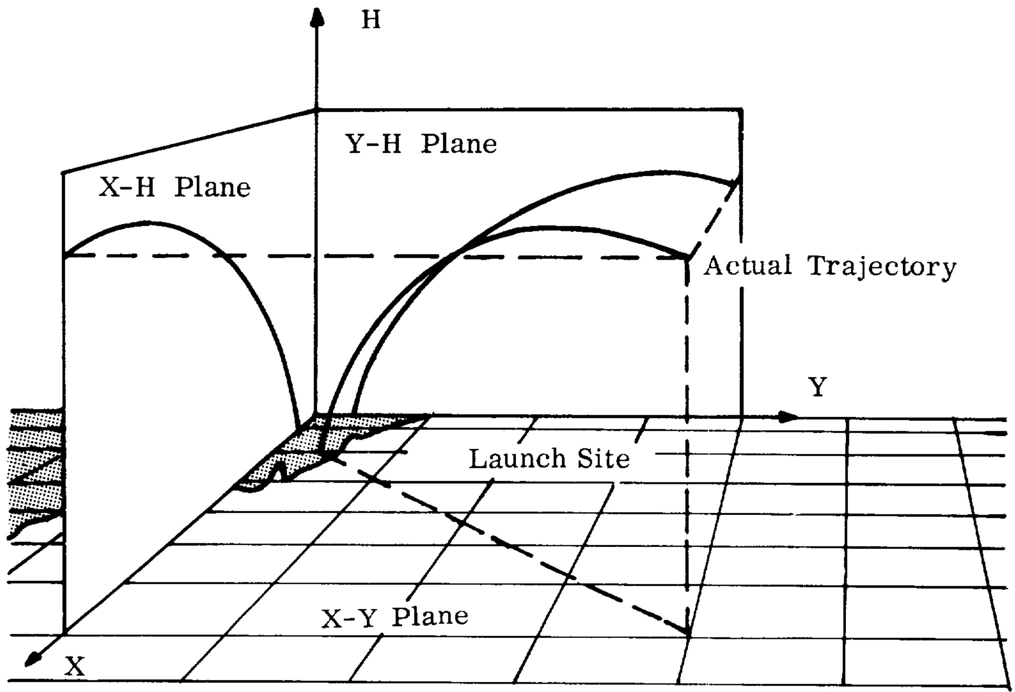


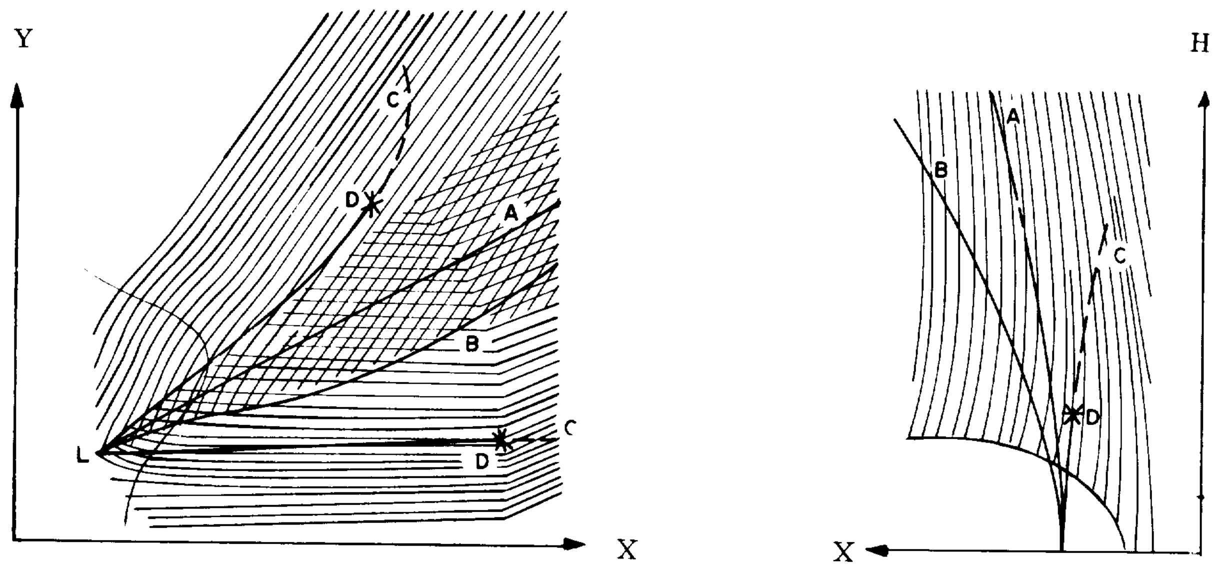
Figure 6-27. Three Coordinate Projection of Saturn Trajectory

is tracked by television cameras. This visual information of vehicle action is presented on a closed-circuit television monitor available to the range safety officer.

6-70. IMPLEMENTATION.

Command transmitters, located at Cape Kennedy, Grand Bahama Island, San Salvador, Grand Turk Island, and Ascension Island transmit the coded signals which initiate engine cutoff and propellant dispersion. Two transmitters at each site, one operating and one standing by, ensure reliability of communications. In the event of failure of the operating transmitter, automatic equipment switches the standby transmitter into service.

Two pairs of command receivers and decoders located aboard each stage of the vehicle receive and decode the transmitted commands. The required action order is transferred through command destruct controllers to other equipment aboard the vehicle. Each command receiver is served by two antennas, located 180 degrees apart on the periphery of the stage to ensure that one is always in line of sight of the transmitter. The two antenna pairs on each stage are located 90 degrees apart, to further enhance reliability of reception. Figure 6-29 illustrates the typical



X-Y Plot

X-H Plot
(Y-H Plot is similar)

X and Y = horizontal coordinates
H = vertical coordinate
A = projection of nominal trajectory
B = permissible trajectory
C = nonpermissible trajectory; as soon as the projection of the trajectory parallels neighboring range safety lines (at D), flight termination action is taken.

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Figure 6-28. Range Safety Plots

mechanization of the range safety function on a vehicle stage.

Operational range safety command equipment, which is normally employed on Saturn I vehicles includes the AN/DRW-13 command receiver. For the AN/DRW-13, commands are transmitted to the vehicle through frequency modulation of the command transmitter's carrier signal by coded combinations of audio tones. The carrier signal is received and demodulated by the command receiver. The recovered audio tones are sorted by frequency into the proper channels to energize a combination of relays and execute the transmitted command, Figure 6-30.

The range safety operation is detailed in the following paragraphs which describe a

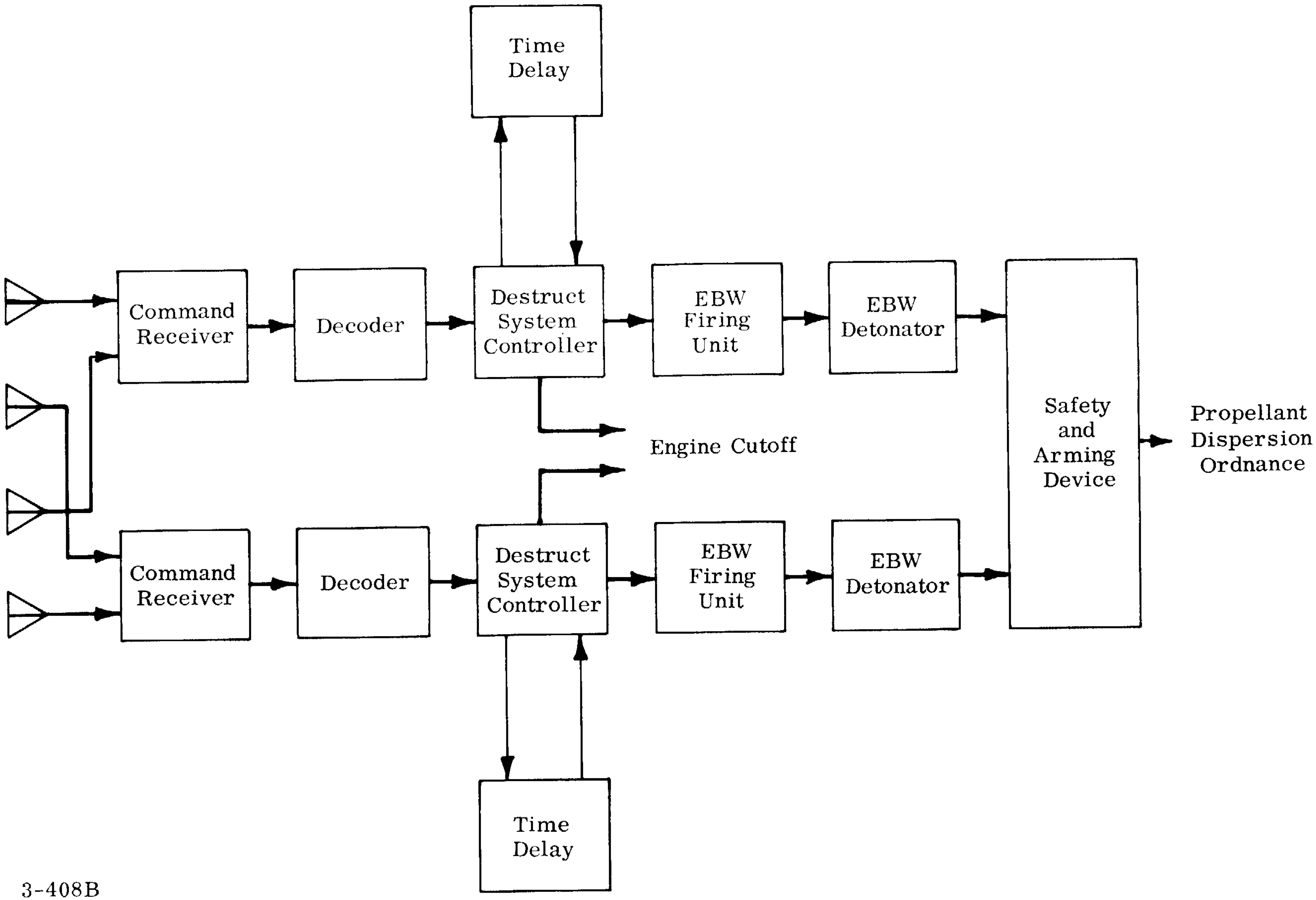
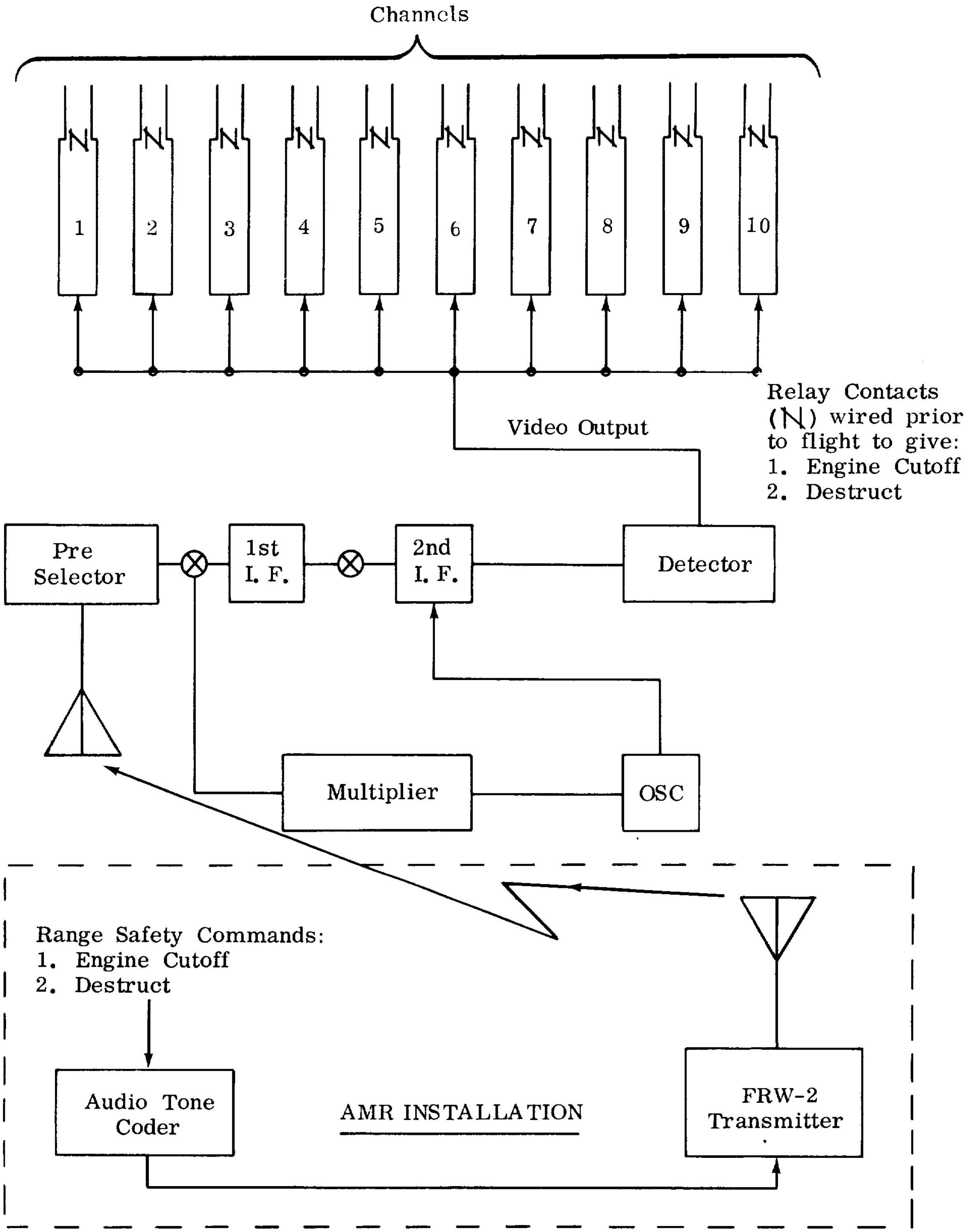


Figure 6-29. Range Safety Command System



3-325

Figure 6-30. AN/DRW-13 Command Receiver

hypothetical situation. (This description involves the digital command system, Figure 6-31, which is carried as passenger, or developmental, equipment on Saturn I vehicles, rather than the AN/DRW-13 command receiver.)

The range safety officer decides that the vehicle constitutes a danger and must have its thrust terminated and propellants dispersed. He actuates, in sequence, the engine cutoff and destruct switches. The resulting signals are digitally encoded, and, together with a digital address for the vehicle, are delivered to the command transmitter, which is rf-linked with the vehicle. The command transmitter sends the command to the vehicle, where it is received by the command receiver, its address compared with vehicle coding, and accepted. The "message" is then decoded, and translated into relay closures which deliver "engine cutoff" and "destruct" command signals to the command destruct controller.

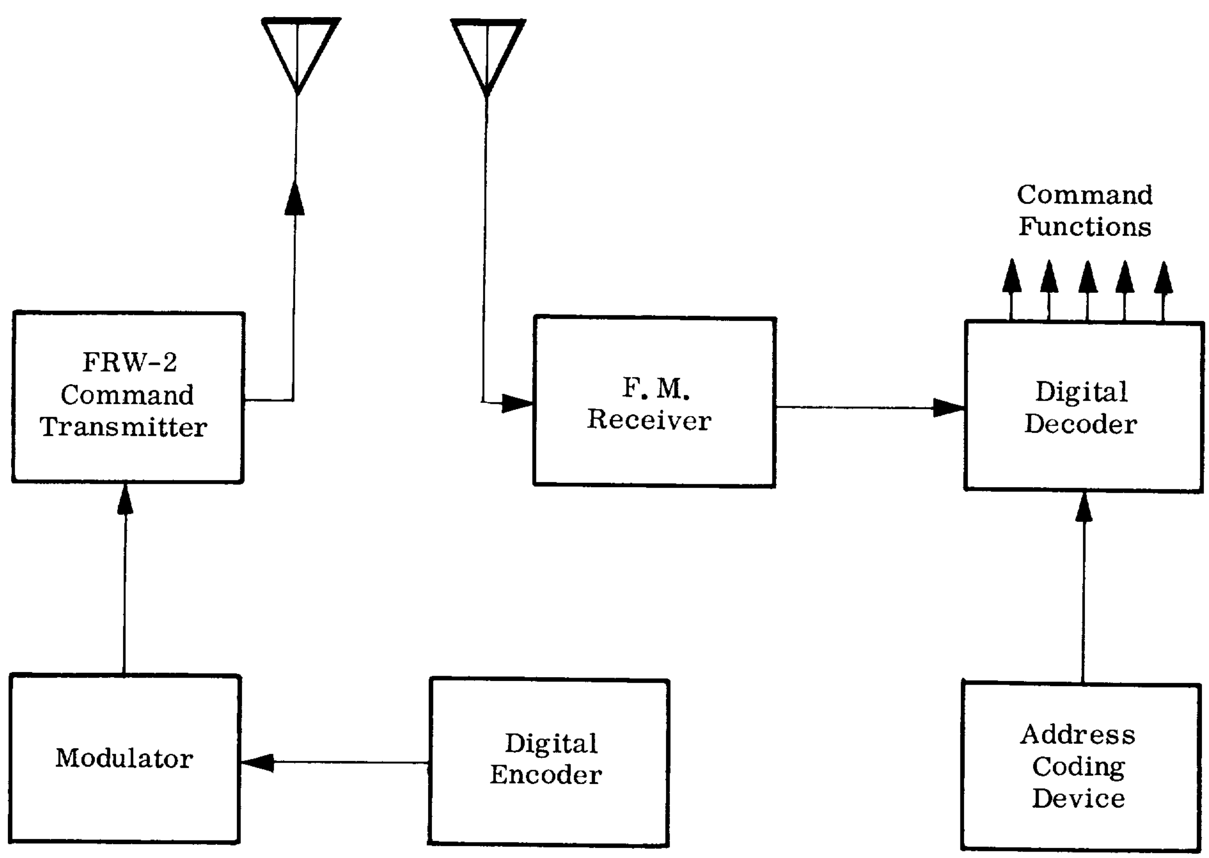


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Figure 6-31. Digital Command System

The "engine cutoff" signal sets up switching in the controller to initiate the cutoff of propellants to the engines through other vehicle systems. The "engine cutoff" signal also starts a delay timer, which, after the desired time delay, relays the "destruct" input through the command destruct controller to trigger an EBW firing unit and set off the propellant dispersion ordnance (described in Paragraph 9-26).

6-71. ELECTRICAL SYSTEM.

The two stages of the Saturn I launch vehicle and the instrument unit are electrically independent. Each contains a complete electrical system which supplies all of its power requirements.

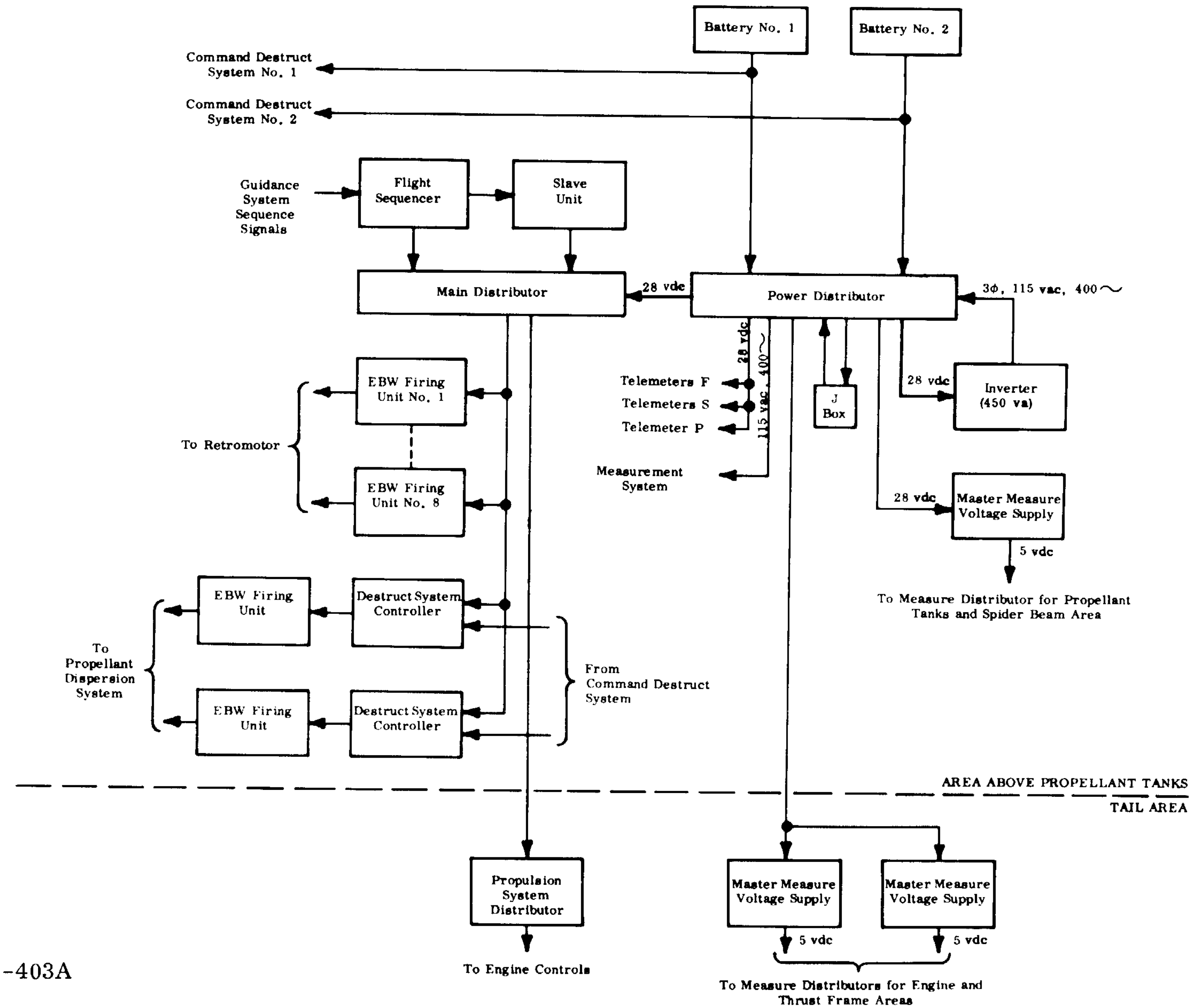
The Saturn I electrical systems are active throughout all mission phases. During the prelaunch phase and the majority of the launch phase, primary power (28-volt dc) for the systems is supplied by generators located at the Automatic Ground Control Station (AGCS). The generator source of primary power is maintained until near the end of the launch phase (approximately T minus 35 seconds) at which time the primary power is switched without interruption from the ground generator source to stage batteries by a network of relays. At launch, explosive switches, connected in parallel with the relay contacts, are fired to permanently lock the power transfer functions thus preventing power interruptions that could occur due to relay failure or contact bounce.

Throughout the mission, the ac power 115 volt (three phase 400 cps) for each electrical system is supplied by stage electrical distributors and networks.

6-72. OPERATION.

The operation of each electrical system is similar. Therefore, only that of the S-I stage is described. The S-I stage electrical system, Figure 6-32, is comprised of two 28-volt batteries, dc power supplies, a dc-to-ac inverter, distributors, a flight sequencer, a slave unit, and several types of J-boxes.

6-73. Batteries. Inflight power for the stage is supplied by two 28-volt batteries. The battery cells are constructed of zinc-silver oxide using potassium hydroxide as electrolyte. Each is rated at 1650 ampere-minutes and is provided with taps which are used to adjust the output voltage to 28-volt dc (nominal) under load.



3-403A

Figure 6-32. Electrical System, S-I

6-74. Inverter. A 450 volt-ampere solid-state inverter is used to convert 28-volt battery power to 115-volt, 400 cps, three phase power. The output voltage is used to power various components in the measuring system.

6-75. Master Measuring Voltage Supply. The master measuring voltage supply is a solid-state dc-to-dc converter. It converts 28-volt dc inputs into 5-volt dc outputs (one amp capacity), controlled to within 0.25 percent. The reference voltage for measurement transducers and signal conditioners is supplied by this unit.

6-76. Distributors. The distributors are the switching and distribution centers for all of the electrical circuits in the stage. They contain relays, buses, and current limiting components. The stage switching and distribution functions are assembled into groups of identical or similar functions (measuring, power distribution, etc.) and a distributor is furnished for each group. A brief description of each type of distributor follows:

Power Distributor. Prior to primary power transfer, the stage primary power is supplied to the power distributor from ground generators. After the power transfer and during vehicle flight, primary power is supplied to the distributor from the 28 volt batteries. The power distributor contains two separate dc output buses, one for steady loads, and one for varying loads. The steady-load bus supplies power to the measurement system components; the varying load bus supplies power to relays, valves, and other control equipment. A third bus system, supplied from the inverter, serves all the ac power loads.

Main Distributor. Vehicle functions that are initiated or controlled by the flight sequencer are distributed by this unit. The main distributor dc power is supplied by the power distributor.

Propulsion System Distributor. This component receives 28-volt dc power from the power distributor and distributes it to the circuits that control the engine functions. Thrust OK pressure switches and relays used for the operation of fuel and LOX fill and drain, and replenishing valves are contained in this unit.

6-77. Flight Sequencer and Slave Unit. The flight sequencer, a relay device, distributes 28-volt dc power to stage relays and control devices. The capacity of the basic unit is a 10-step program. Each step in the program is expandable in multiples of 10 steps by the addition of slave units. The timing pulses for driving the flight sequencer originate in the guidance computer (part of the guidance and control system).

6-78. J-Box. The J-box is a standard connector. Outer terminals of the connector may be soldered together to form junction points, or used to connect simple circuit elements into the circuits of the distributors. The J-box functions as a small remote distributor and signal conditioner.

6-79. IMPLEMENTATION.

(To be supplied at a later date.)

CHAPTER 2

SECTION VII

STRUCTURES

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SECTION VII.

STRUCTURES

7-1. STRUCTURAL REQUIREMENTS.

The Saturn I launch vehicle structure is designed to withstand all loads that can be expected to occur during ground handling, prelaunch, launch and flight operations. The structure also contains the propellant for the stages. The design requirements for the vehicle structure are determined after a careful analysis of the conditions that will be encountered during all operations.

7-2. GROUND HANDLING CONDITIONS.

Handling procedures and equipment are designed so that loads imposed on the structure during fabrication, transportation, and erection do not exceed flight loads and thus do not impose any flight performance penalty.

7-3. PRELAUNCH CONDITIONS.

The vehicle, empty or fueled, pressurized or unpressurized and free-standing (attached to the launcher only) is structurally capable of withstanding loads resulting from winds having a 99.9 percent probability of occurrence during the strongest wind month of the year. The bending moments (Figure 7-1) and shears resulting from the wind are combined with the longitudinal force due to the weight of the vehicle in defining the worst prelaunch loading condition.

7-4. LAUNCH CONDITIONS.

At launch the vehicle structure is capable of withstanding loads from two conditions, holddown and rebound. The holddown condition is imposed on the structure after engine ignition, but before the launcher releases the vehicle. The holddown loads result from wind (bending moments and shears), engine thrust (forward axial load), vehicle inertia (aft axial load) and vibration transients due to initial engine combustion. The rebound condition occurs when the engines are cut off before the launcher releases the vehicle. Axial loads result from deceleration of the vehicle which suddenly reverses the direction of the load at the holddown points. Combined with the axial

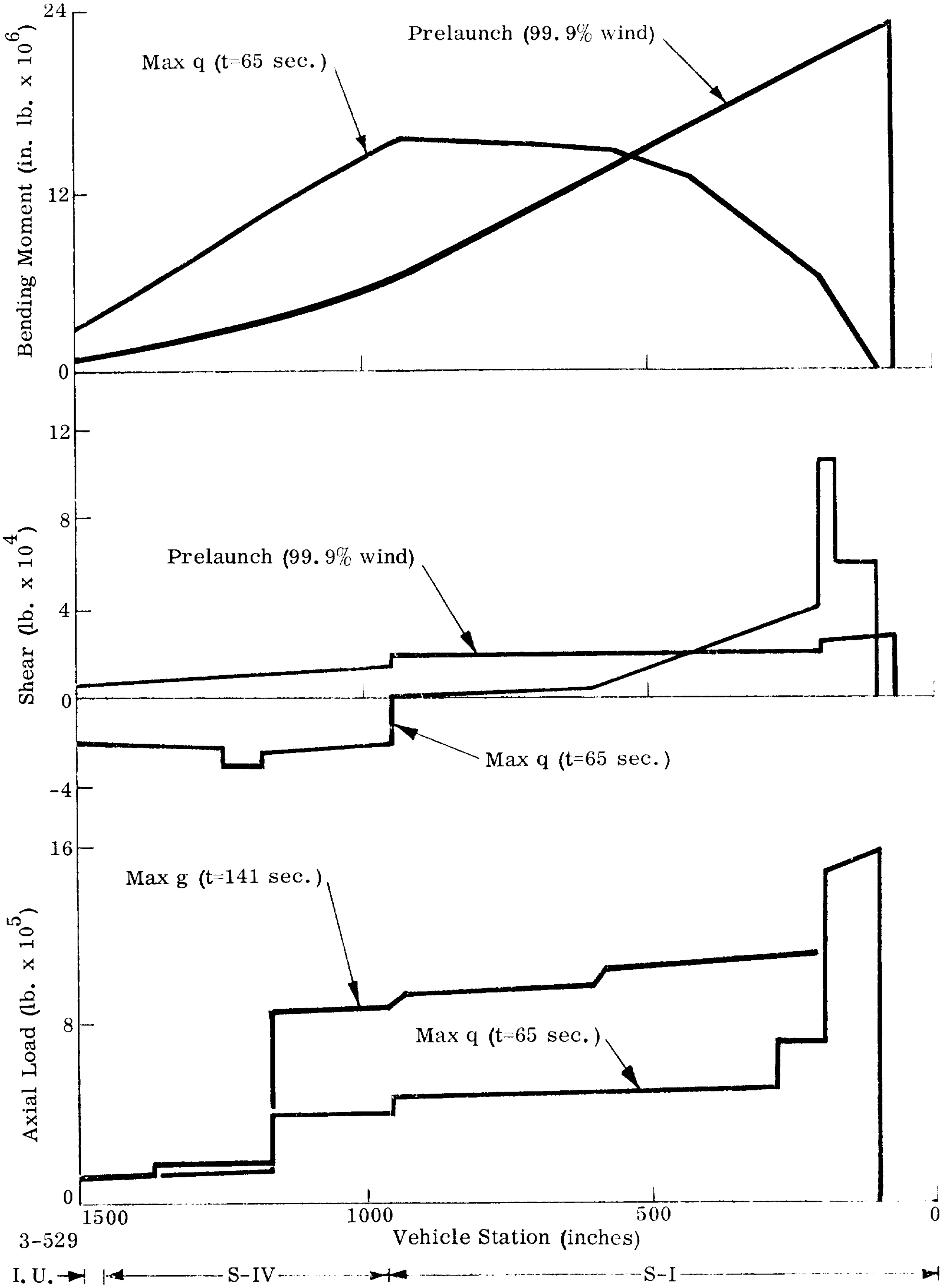

Figure 7-1. Saturn I Loads

loads are wind loads (bending moments and shears) and vibration transients resulting from engine cutoff.

7-5. FLIGHT CONDITIONS.

During flight the structure is subjected to engine thrust and heat, dynamic, aerodynamic, inertia and propellant loads.

7-6. Engine Thrust and Heat Loads. The first stage thrust (Figure 7-2) increases as the vehicle gains altitude, reaches a maximum at approximately 110 seconds after liftoff, and then decreases slightly prior to first stage engine cutoff. After first stage separation, the second stage engines impose thrust loads, which are relatively constant, on the remainder of the vehicle. The thrust produces axial loads, shears and bending moments on the vehicle. The moments and shears are a result of the engines' gimballing.

The first stage engines impose a heat load on the base of the vehicle through radiation and circulation of the exhaust gases. After separation the second stage engines impose a heat load on the base of the second stage.

7-7. Dynamic Loads. Vehicle dynamic loads result from external and internal disturbances. Three main sources of excitation - mechanical, acoustical and aerodynamic produce the vehicle vibration environment. The mechanical source begins at engine ignition and remains relatively constant until engine cutoff. The acoustical source begins with the sound field generated at engine ignition. It is maximum at vehicle liftoff and becomes negligible after Mach 1 (approximately 58 seconds after liftoff). The aerodynamic source begins as the vehicle velocity increases and is most influential during transition at Mach 1 and at maximum dynamic pressure. Transient vibrations, which are relatively high in magnitude and present only for short periods of time, occur during engine ignition, vehicle liftoff, Mach 1, region of maximum dynamic pressure, engine cutoff, and stage separation.

Propellant sloshing, another type of dynamic loading, results from a relative motion between the container and the center of gravity of the fluid mass and is generally caused by gust loads, control modes and vehicle bending modes. Reaction of the control system (gimballing engines) to gust loads produces considerable bending deflection in the vehicle structure. Since the structure and propellant are not

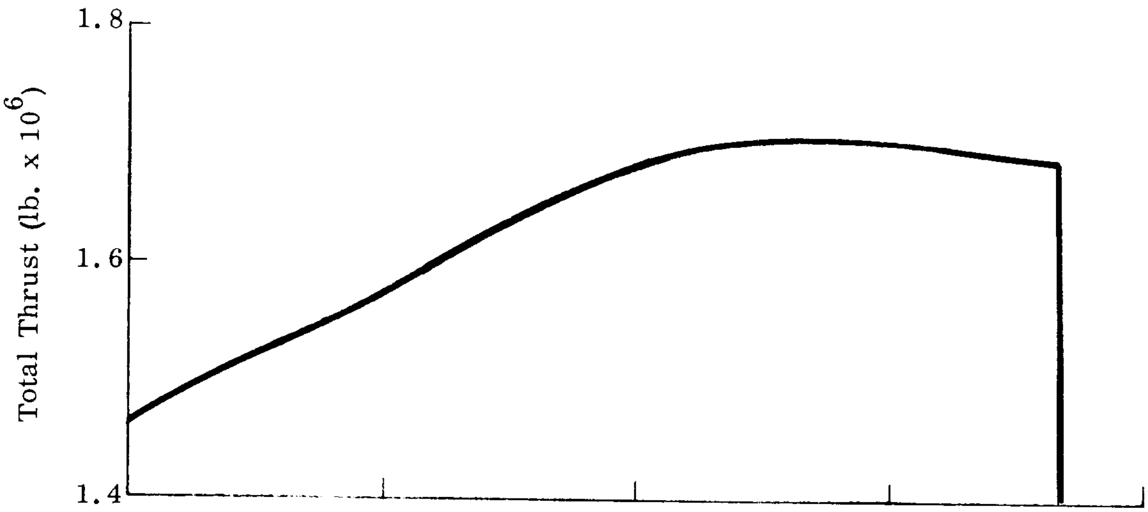

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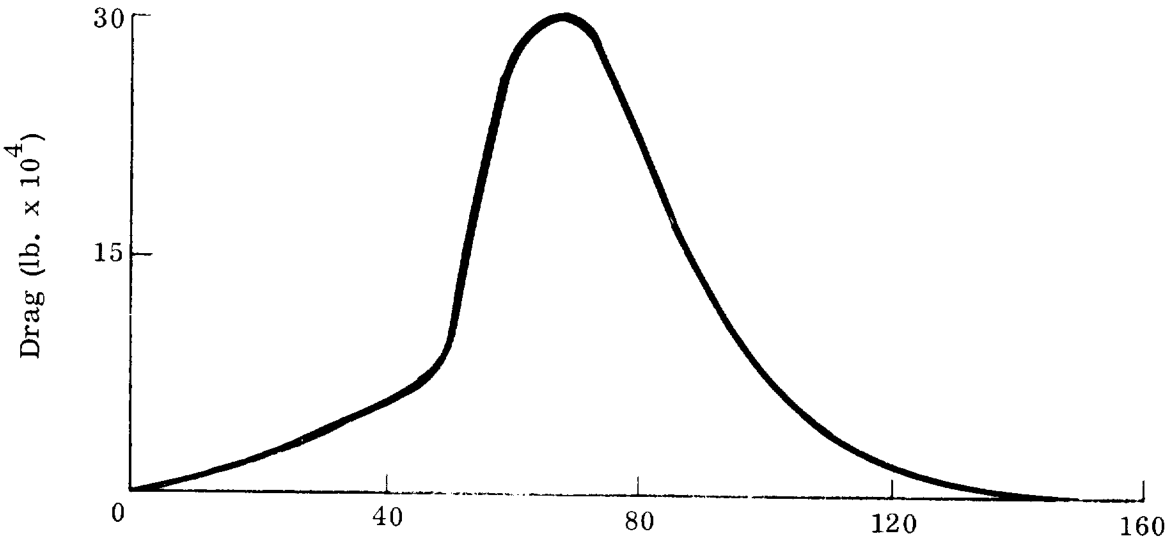
Figure 7-2. S-I Thrust



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Flight Burning Time (sec.)

Figure 7-3. Saturn I Drag

integral and do not deflect together, sloshing results. If the propellant sloshing is not damped, compensation for the resulting perturbations must be provided by the control system.

7-8. Aerodynamic Loads. Aerodynamic loading is a result of drag, angle of attack and wind gusts. Aerodynamic drag (Figure 7-3) increases to a maximum approximately 65 seconds after liftoff (max q condition) and then decreases to nearly zero before first stage burnout. Aerodynamic drag imposes an axial load on the structure and when combined with an angle of attack results in bending moments and shears. When the vehicle is in the region of high drag, structural bending moments are minimized by the control system which reduces the vehicle angle of attack.

Aerodynamic heating on the vehicle is a result of friction caused by the vehicle moving through the atmosphere. The heating increases until first stage burnout and then decreases. Vehicle surfaces which are not parallel to the vehicle centerline have the greatest temperature increase during flight.

7-9. Inertia Loads. Inertia loads result from the vehicle acceleration due to an increase in the thrust/weight ratio during flight. Peak acceleration is at first stage cutoff (max g condition). The acceleration decreases at separation and then increases during second stage burning, but never reaches the peak achieved at first stage cutoff.

7-10. Propellant Loads. The loads imposed on the structure by the propellant are due to a combination of hydrostatic head and ullage and ambient pressures. The hydrostatic head, varying during flight, is a function of the density of the fluid, height of the fluid in the container and the acceleration of the vehicle. The ullage pressure is supplied by the pressurization system and is limited by relief valves. As the altitude of the vehicle increases during flight, the ambient pressure decreases. At any time during flight (at any location in the container) the maximum pressure differential across the container wall is equal to the ullage pressure plus the hydrostatic head minus the ambient pressure.

7-11. STRUCTURAL DESIGN.

The Saturn I launch vehicle consists of two stages joined by an interstage. An instrument unit mounted forward of the second stage provides the support for the spacecraft. Critical loading conditions for various portions of the vehicle occur at different times. The critical conditions occur on the S-I structure during prelaunch (ground wind), launch (holddown and rebound) and flight (max q and max g). They occur on the S-IV structure during prelaunch (ground wind) and flight (max g and after separation), and on the instrument unit during flight (max q). For the propellant containers, critical external loads are combined with the internal gas pressure and hydrostatic head to obtain the structural design loads.

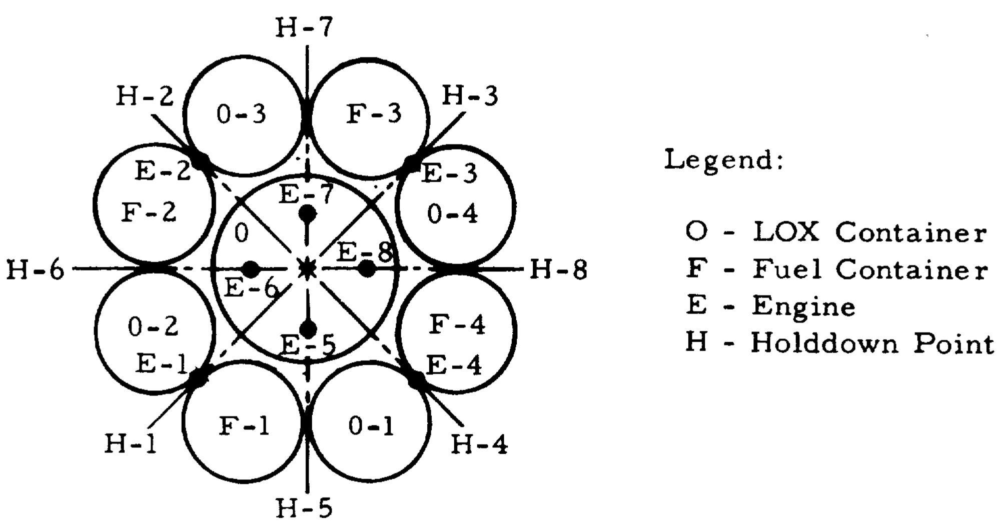
Slosh baffles are installed in the S-I fuel and LOX containers and in the S-IV LOX container. The baffles dampen the sloshing propellant and transfer absorbed slosh forces to the container walls. Slosh baffles are not required in the S-IV LH_2 container because of the low density of the LH_2.

7-12. S-I STAGE.

The S-I structure is an assembly of nine propellant containers (five LOX and four fuel) supported at the forward end by the second stage adapter and at the aft end by the tail section. Eight fins are attached to the tail section. A 105-inch diameter LOX container is located on the stage centerline. Alternately spaced around the center container (Figure 7-4) are four LOX and four fuel containers; each is 70 inches in diameter. The containers are structurally independent of one another. The nine container configuration was selected because manufacturing techniques for these size containers had been previously established, thus the fabrication time could be shortened.

The second stage adapter (spider beam), five LOX containers and tail section resist the loads encountered during all vehicle operations through first stage burnout. The LOX containers carry axial load in both directions; the fuel containers carry axial load only in the aft direction. The fuel containers are supported at the forward end by a sliding pin connection which permits relative movement between the spider beam and thrust structure due to the contraction of the LOX containers as the containers are being filled.

Several conditions produce critical loads on the thrust structure. The maximum loads on the thrust structure outriggers are produced by the holddown, rebound and max q conditions. For the thrust structure barrel assembly the max q and



3-523

Figure 7-4. Container, Engine, Holddown Schematic, S-I

max g (engine thrust) conditions produce the maximum axial loads, bending moments and shears. The aft end of the thrust structure is protected from the hot engine exhaust gases by the heat shield and flame shield.
Eight aerodynamic fins aid in stabilization during flight. The maximum loading condition on the fins occurs at max q. Incorporated in each fin is a holddown fitting for attachment to the launcher. A local critical loading condition on the fins is produced by the rebound condition.

The critical loading on the center LOX container and container skirts is a result of the prelaunch (container full and unpressurized) and max q conditions. For the outboard LOX and fuel containers, the critical loading conditions occur during prelaunch (containers empty and unpressurized) and max q. The skirts for the outboard LOX and fuel containers are critically loaded during max q condition. The critical load on the spider beam occurs at max q.

In addition to the external loads carried by the LOX containers, all the containers must withstand propellant and internal pressurization loads. Each container consists of a forward and aft bulkhead joined by a cylindrical section. The maximum pressure differential on the container forward bulkheads occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical sections and aft bulkheads varies during flight because the propellant level and ambient pressure decrease while the acceleration of the vehicle increases.

7-13. S-IV STAGE.

The S-IV structure is an assembly of an aft interstage, an aft skirt, a thrust structure, a base heat shield, an integral propellant container, and a forward skirt. To reduce the length of the vehicle and thus reduce external loading, the propellants are contained in an integral container. Located within the container is the common bulkhead which separates the fuel (LH_2) from the oxidizer (LOX). To reduce the loads on the vehicle the LOX which weighs five times as much as the LH_2 is located aft.

The aft interstage, aft skirt, cylindrical section of the propellant container, and forward skirt withstand the loads encountered during all vehicle operations through first stage burnout. Following stage separation and until spacecraft separation, the thrust structure, LOX container aft bulkhead, cylindrical section of the LH_2 container, and forward skirt resist all loads encountered as a result of S-IV engine operation.

The critical design condition for the aft interstage and aft skirt occurs at max g. This condition produces the largest compressive buckling load on the structure. For the cylindrical section of the LH_2 container, the prelaunch condition (container full and unpressurized) is most critical. Maximum loading on the forward skirt occurs at max q, but because of allowable stress reduction due to aerodynamic heating the max g condition is more critical.

Engine thrust, the principal load during S-IV engine operation, produces a critical loading condition only in the thrust structure. The base heat shield, which is attached to the thrust structure, is designed to protect the aft end of the S-IV from engine heat.

In addition to the external loads carried by the cylindrical section, the propellant container must resist propellant and pressurization loads. The container consists of a forward bulkhead, a cylindrical section, an aft bulkhead and a common bulkhead. The maximum pressure differential on the container forward bulkhead occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical section and the aft bulkhead is at first stage cutoff. At this time the vehicle acceleration is greatest and the ambient pressure is zero. The common bulkhead is designed for both bursting and collapsing pressure conditions. The critical conditions are based on combinations of LH_2 and LOX pressures and temperatures.

7-14. INSTRUMENT UNIT.

The instrument unit structure resists the loads encountered during all vehicle operations through payload separation. The critical design condition occurs during flight at max q when a combination of bending moment and axial force produces the largest compressive buckling load on the structure.

7-15. S-I STRUCTURAL CONFIGURATION.

The S-I stage structure is 962 inches (80.2 feet) long, 257 inches (21.4 feet) in diameter across the containers, 274 inches (22.8 feet) in diameter across the thrust structure, and has a span of 488 inches (40.7 feet) across the fins. A tail section, nine propellant containers (five LOX and four fuel) and a second stage adapter are structurally joined together to make up the stage. Eight aerodynamic fins (four large and four stub) are attached to the tail section.

7-16. TAIL SECTION.

The tail section supports the eight H-1 engines and transmits thrust loads to the five LOX containers. In addition, the tail section supports the four fuel containers and protects the engines and associated installations from aerodynamic loads and engine heating. Holddown loads are transmitted to the tail section through the fins. A thrust structure, shrouds and heat shielding are structurally joined to make up the tail section.

Thrust loads are transmitted to the LOX containers through the aluminum-alloy thrust structure (Figure 7-5). The four inboard engines, equally spaced on a 64-inch diameter, are mounted in a fixed position and are canted 3 degrees from the vehicle centerline. The four outboard engines gimbal and are equally spaced between the inboard engines. The outboard engines are mounted on a 190-inch diameter and are canted 6 degrees from the vehicle centerline.

Thrust loads from the inboard engines are transmitted to the thrust-structure barrel assembly which is 105 inches in diameter and approximately 75 inches long. Lateral loads (resulting from the engines being canted) and axial loads are transmitted to the barrel assembly aft ring through the engine mounting pads. The aft ring is a built-up box section. A cross beam structure is attached to the inside of the aft ring. This structure supports the fixed link actuators which support the inboard engines. Axial loads are transmitted to tapered longerons by the aft ring. In turn the longerons transmit the axial loads to the skin and the four fin support outriggers. The four fin-support outriggers and four thrust-support outriggers are supported by the aft and forward rings of the barrel assembly. The forward ring is a built-up box section. An internal ring located between the aft and forward rings supports the barrel skin. Cutouts in the skin are provided for routing propellant lines through the barrel assembly.

The forward ring of the barrel assembly is attached to the center LOX container. Part of the thrust load from the four inboard engines is transmitted to the center

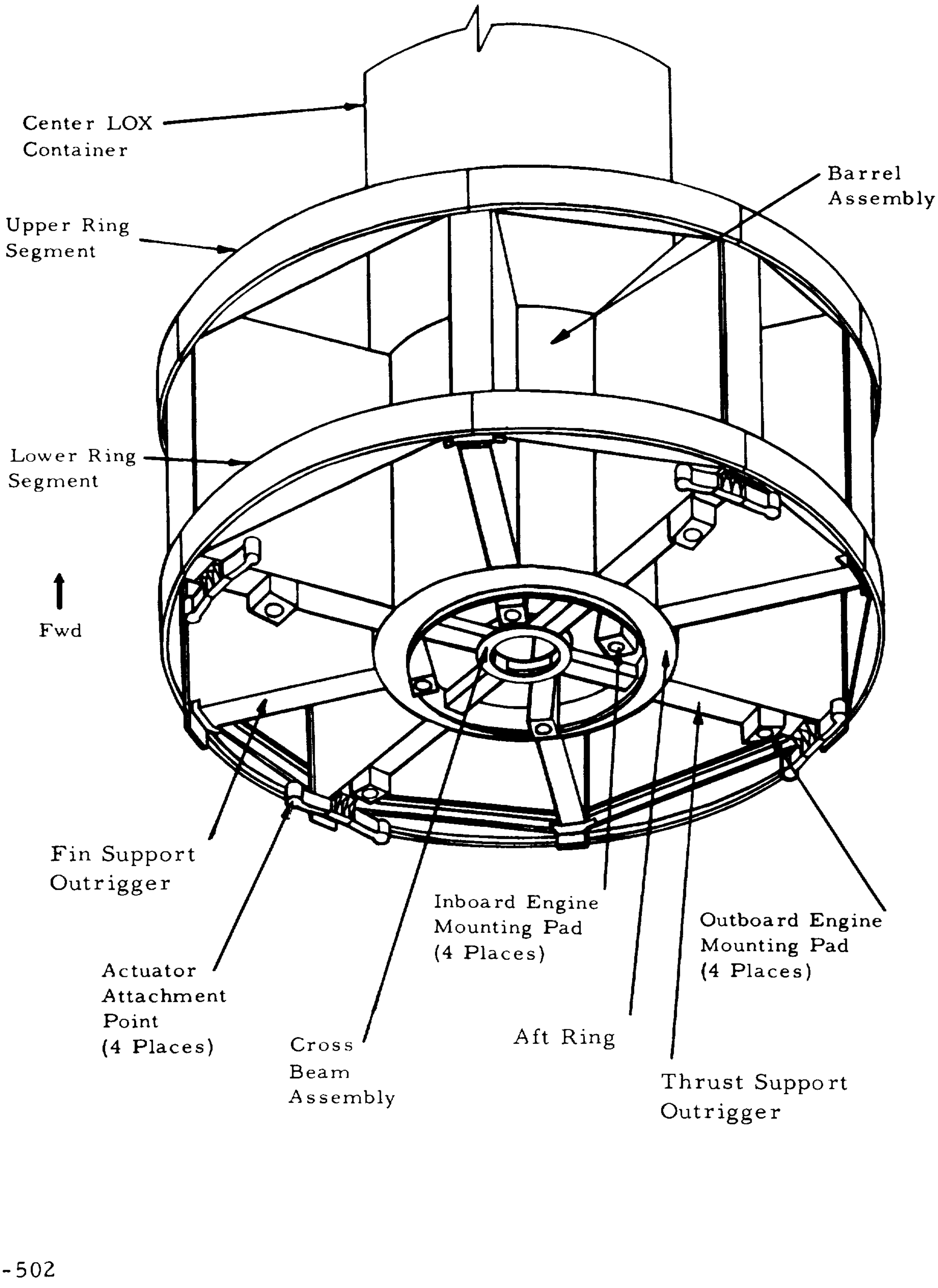


Figure 7-5. Thrust Structure, S-I

container. The remainder of the thrust load is transmitted to the four fin-support outriggers.

The fin-support and thrust-support outriggers are attached to the barrel assembly. The four fin-support outriggers receive inboard engine thrust load from the barrel assembly. The four thrust-support outriggers support the outboard engines. Two mounting points on each of the outriggers support the outboard propellant containers which are on a 187-inch diameter. Each outrigger has a support point for a fuel and a LOX container. Thrust loads are transmitted from the outriggers to the outboard LOX containers. (The fuel containers do not carry thrust load.) All support points are capable of carrying lateral loads.

Each outrigger consists of two plates stiffened with horizontal and vertical members. Thrust loads from the outboard engines are transmitted to the plates through thrust beams. The thrust beams are located between the plates and backup the outboard engine mounting pads. Actuators for the outboard engines are attached to a beam assembly mounted on the thrust support outriggers. Upper and lower ring segments each with a radius of 135 inches join the outboard ends of the outriggers.

Attached to the ring segments and the outrigger shroud support plates are eight forward shroud panels. The shroud panels protect the compartment between the propellant containers and the engines from aerodynamic pressure and thermal loads. Each panel is stiffened with internal longitudinal and circumferential members and has a door for access to the compartment.

Located at the aft end of the thrust structure are firewall panels attached to the aft ends of the outriggers and lower ring segments. The firewall panels form a fire barrier between the forward (propellant container) compartment and the aft (engine) compartment.

The aft compartment is protected from aerodynamic pressure and thermal loads by the aft shroud which is attached to the lower ring segments. The shroud, 270 inches in diameter and 60 inches long, is a continuous corrugation supported by internal rings. The corrugated skin exposes the maximum amount of surface area to the engine compartment permitting maximum heat dissipation.

The lower end of the aft compartment is closed by the heat shield (Figure 7-6) which provides protection from engine heat. Constructed of stainless steel stiffened panels, the heat shield is covered on the aft face with an ablative insulation. The panels are supported by a complex of cross beams which are attached to the aft end of the aft shroud. Cutouts are provided in the shield for gimballing the outboard engines. These cutouts and the cutouts for the inboard engines are sealed with flexible curtains that are attached to the engines and heat shield. The curtains are constructed of fiberglass cloth and refrasil. Access to the compartment is provided by eight doors in the heat shield.

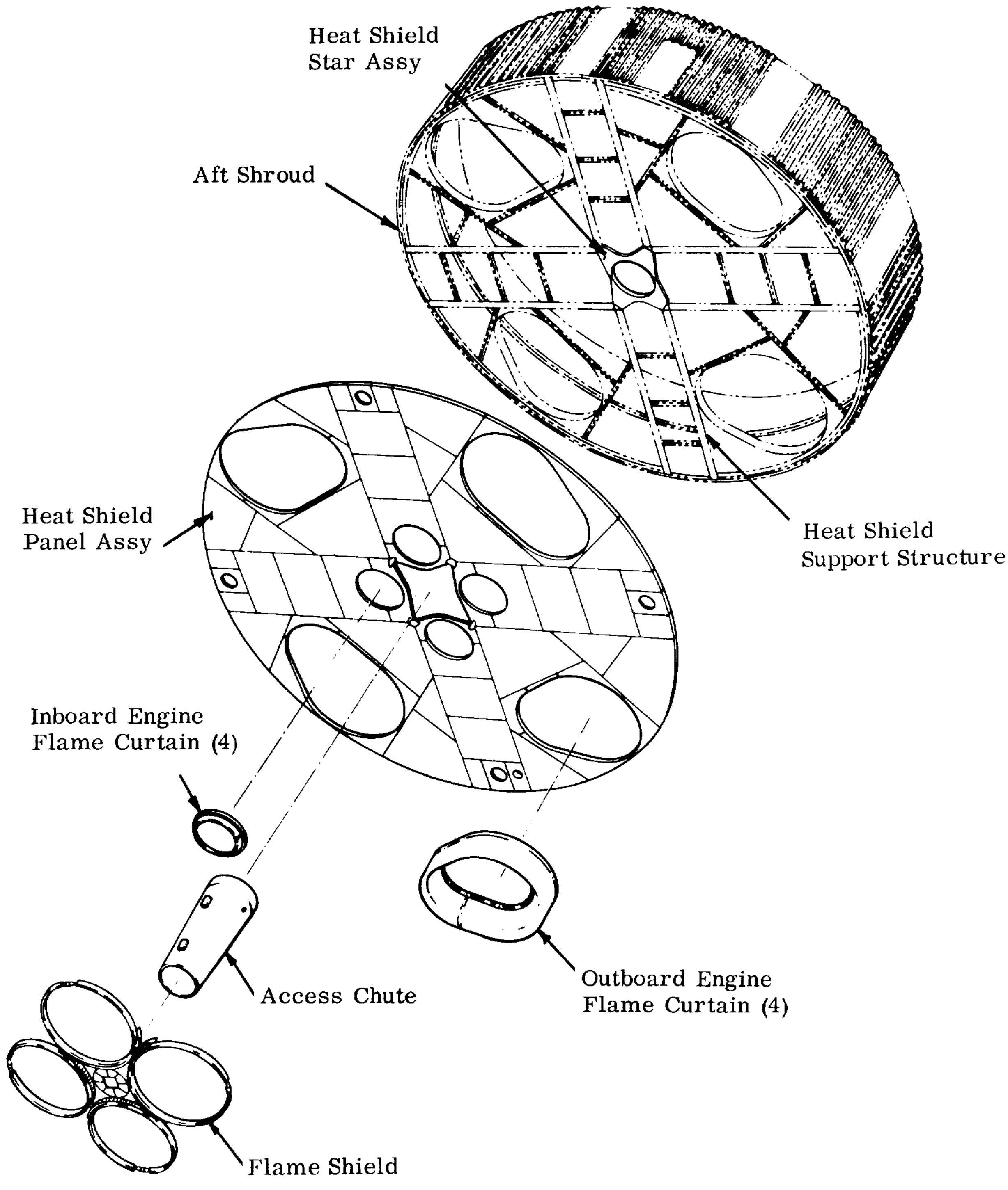
The flame shield is supported from the heat shield by the conical frustum access chute. At the forward end, the access chute is attached to the heat shield star assembly (center portion of the heat shield). The flame shield is located between the four inboard engines at the thrust chamber outlets. It is constructed of stainless steel and is attached to the inboard engine thrust chambers with steel bands insulated with fiberglass cloth.

The four engine skirts attached to the heat shield protect the engines from aerodynamic forces that would produce excessive loads on the control actuators. The engine skirts are conical segments 32 inches long. The inside surface of the skirts below the heat shield is protected from engine heat by a layer of ablative insulation.

7-17. FINS.

Four large fins and four stub fins, attached to the tail section, aid in maintaining vehicle aerodynamic stability. The fins are also the holddown and launch pad support points for the vehicle. Holddown and support loads are transmitted to the thrust structure outriggers. The support points are located on the aft face of the fins and are on a 344-inch diameter.

The stub fins are located at the outboard engine positions and the large fins are equally spaced between. Both types of fins have trapezoidal planforms. The large fins have an area of 128 square feet; the stub fins have an area of 52 square feet. The leading edges which are steel, are swept back 20 degrees. The remainder of the fin structure is aluminum alloy with an ablative insulation on the exterior surface.



3-532

Figure 7-6. Flame and Heat Protection, S-I

7-18. LIQUID OXYGEN CENTER CONTAINER.

Approximately 36 percent of the LOX for the S-I stage is contained in the center container, Figure 7-7. The container, a cylinder with torispherical bulkheads, is 105 inches in diameter and 678 inches long. Designed to carry flight pressurization and propellant loads due to acceleration, the center container also transmits part of the thrust load from the thrust structure to the second stage adapter. At the aft end the container is attached to the thrust structure barrel assembly and at the forward end to the spider beam in the second stage adapter.

The cylindrical section, fabricated of 5456 aluminum alloy, is 749 inches long. Recessed into the forward and aft ends of the cylinder are torispherical bulkheads fabricated of 5086 aluminum alloy. The bulkheads are joined to the cylinder by circumferential welds. The aft bulkhead has a sump with four outlets for connection to the LOX manifold. The forward bulkhead has four outlets for connection to the pressure manifold and three outlets for connections to vent lines. A pressure diffuser is mounted to the forward bulkhead. In the area above and below the container (forward and aft container skirts), longitudinal stringers are attached to the cylindrical skin. The stringers distribute the loads received at the container support points. Cutouts for pressurization and vent lines are provided in the skin forward of the container. Cutouts in the skin aft of the container are for the LOX and fuel manifolds (interconnect lines). Circular rings welded to the interior of the cylindrical section support the slosh baffles which are arranged in eight vertical rows equally spaced around the cylinder periphery.

7-19. LIQUID OXYGEN OUTBOARD CONTAINERS.

Approximately 16 percent of the LOX for the S-I stage is contained in each of the four outboard containers. Each container (Figure 7-8) is a cylinder with hemispherical bulkheads, a diameter of 70 inches, and a length of 678 inches. Designed to carry flight pressurization and propellant loads due to acceleration, each of the outboard LOX containers transmits thrust load from the tail section to the second stage adapter. The containers are supported at the aft end by the thrust structure outriggers and at the forward end by the spider beam in the second stage adapter. On the outriggers there are two diametrically opposed support points for each container. Each support point transfers axial and lateral loads. On the spider beam there are also two diametrically opposed support points for each container.

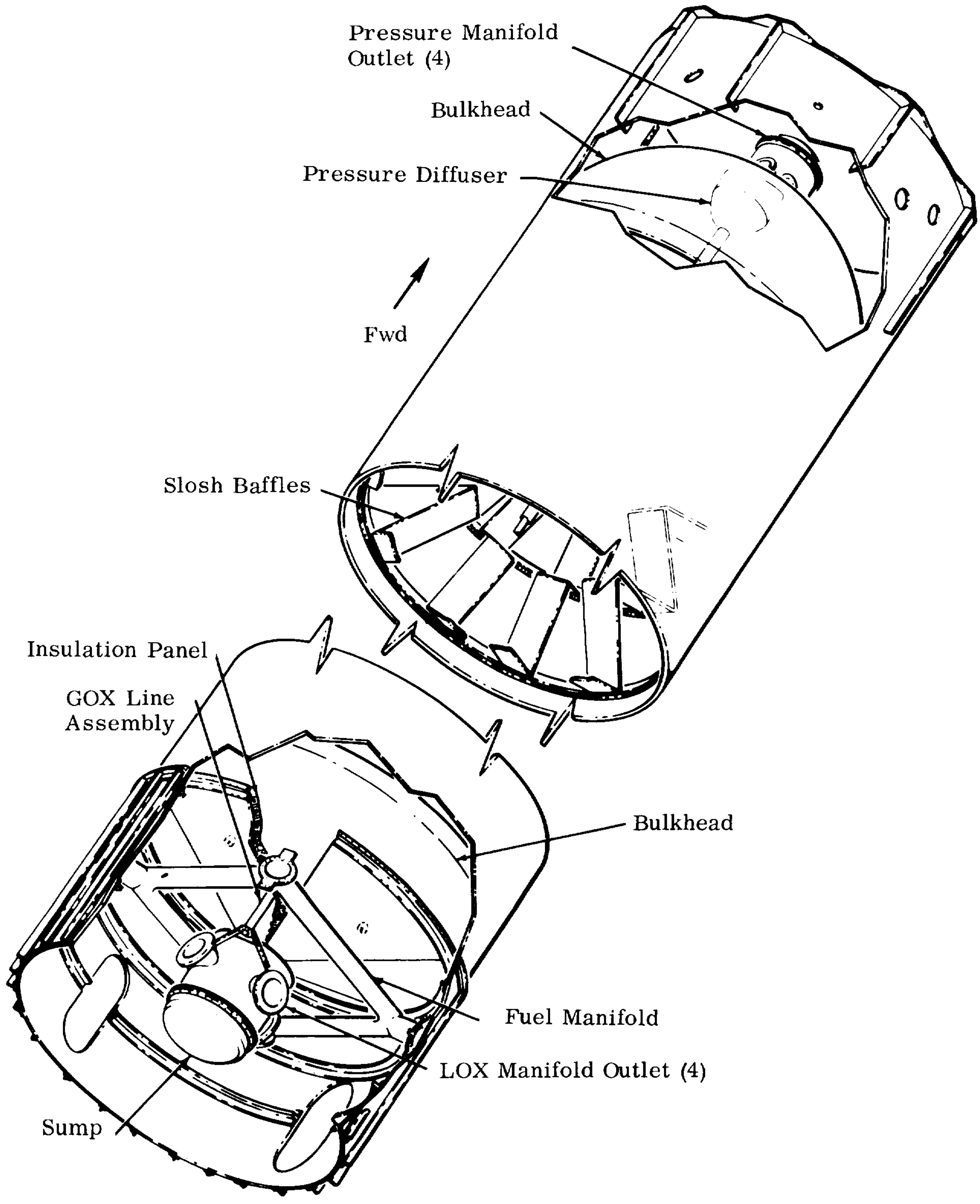


Figure 7-7. Center LOX Container, S-I

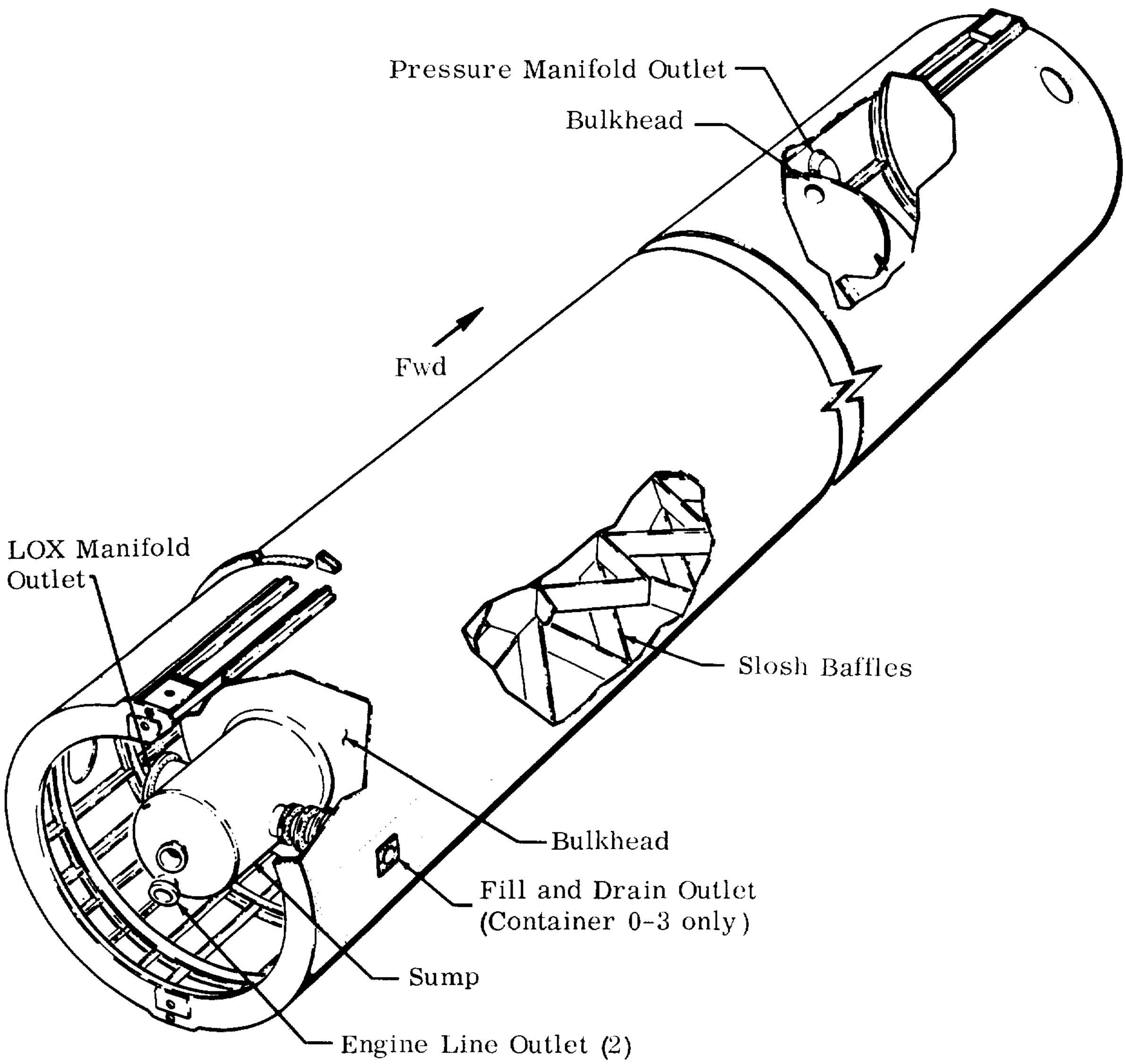


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Figure 7-8. Outboard LOX Container (0-3), S-I

Each support point consists of an adjustable mounting stud which transmits axial and lateral loads.

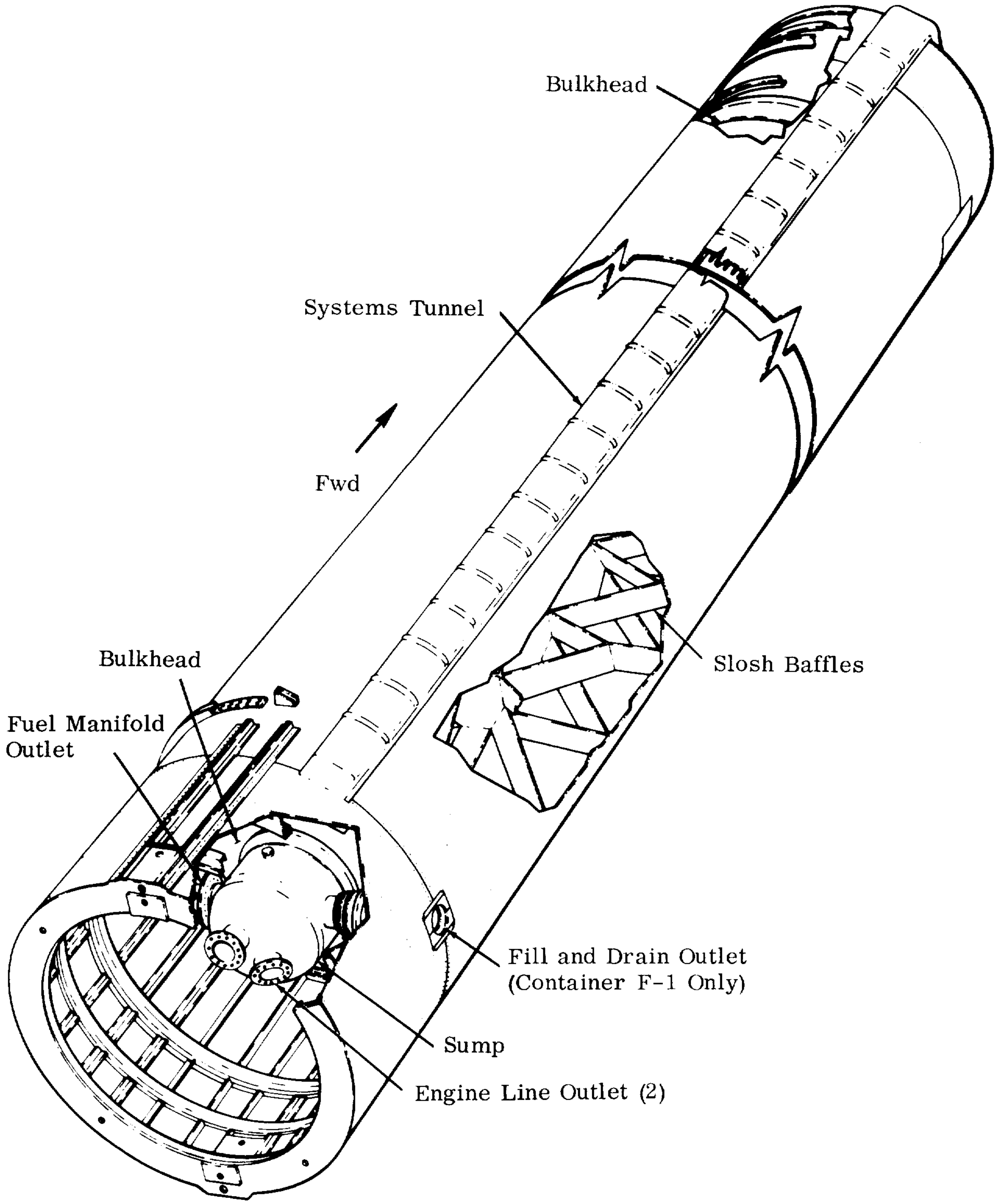
The cylindrical section, fabricated of 5486 aluminum alloy, is 746 inches long. Recessed into the forward and aft ends of the cylinder are hemispherical bulkheads fabricated of 5086 aluminum alloy. The bulkheads are joined to the cylinder by circumferential welds. The aft bulkhead has a sump with three outlets, two for the engine lines and another for the LOX manifold (interconnect line). Container 0-3 has an additional outlet that is used for fill and drain. The forward bulkhead has an outlet for a pressure manifold connection.

In the area above and below the container (forward and aft container skirts), there are longitudinal stringers attached to the cylindrical skin. The stringers distribute the concentrated loads received at the two container support points. The skin above and below the container has cutouts for the lines connecting to the various outlets. Circular rings welded to the interior of the cylindrical section support the slosh baffles which are arranged in six vertical rows equally spaced around the cylinder periphery.

7-20. FUEL CONTAINERS.

Approximately 25 percent of the fuel for the S-I stage is contained in each of the four fuel containers. The containers (Figure 7-9) are cylinders with hemispherical aft bulkheads and torispherical forward bulkheads. The containers have a diameter of 70 inches and a length of 652 inches. The containers are designed to carry flight pressurization and propellant loads due to acceleration. The containers are supported at the aft end by the thrust structure outriggers and at the forward end by the spider beam in the second stage adapter. On the outriggers there are two diametrically opposed support points for each container. Each support point transfers axial and lateral loads. On the adapter spider beam there are also two diametrically opposed support points for each container. Each support point consists of a sliding pin joint. The pin joint resists lateral loading but allows for differential expansion between the fuel and LOX containers in the longitudinal direction.

The cylindrical section, fabricated of 5486 aluminum alloy, is 743 inches long. Recessed into the forward and aft ends of the cylinder are bulkheads fabricated of 5086 aluminum alloy. The bulkheads are joined to the cylinder by circumferential welds. The aft bulkhead has three outlets, two for engine lines and another for the fuel manifold. Container F-1 has an additional outlet for fill and drain. The forward bulkhead has an outlet for the pressure manifold.



3-500A

Figure 7-9. Fuel Container (F-1), S-I

In the area above and below the container (forward and aft container skirts), longitudinal stringers are attached to the cylindrical skin. The stringers distribute the concentrated loads received at the two container support points. The skin above and below the container has cutouts for the lines connecting to the various outlets. Above containers F-1 and F-2 are compartments for mounting electronic equipment. Circular rings welded to the interior of the cylindrical section support the slosh baffles which are arranged in six vertical rows equally spaced around the cylinder periphery.

7-21. SECOND STAGE ADAPTER.

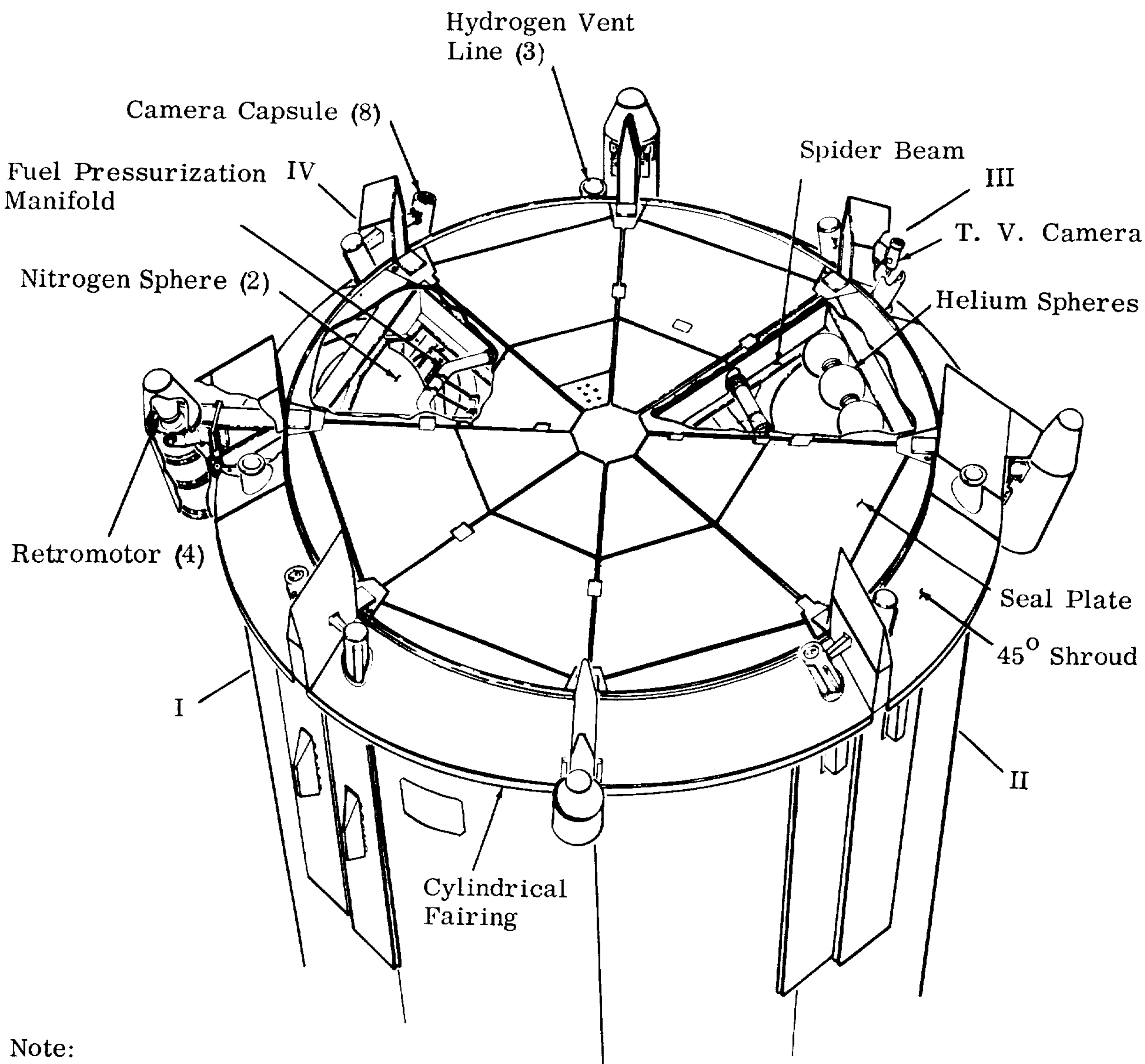
Loads are transmitted to the second stage through the second stage adapter (Figure 7-10) composed of a spider beam, seal plate panels, a 45 degree shroud assembly and a cylindrical fairing. The spider beam (Figure 7-11) supports the propellant containers at the forward end. Fabricated from 7075 aluminum alloy, the spider beam is composed of an octagonal ring and eight radial beams which extend inward from the points of the octagon and are joined at the center with plate gussets. The octagonal ring and radial beams are 20-inch deep I-sections. To absorb vertical loads, the radial beams are stiffened at the propellant container support points. The spider beam is bolted to the S-IV stage at eight points at MSFC station 962.

Mounted on the forward side of the spider beam are seal plate panels. These panels are of honeycomb sandwich construction. Sections of the seal plate may be removed for access to the forward propellant container area. The 45-degree shroud assembly is attached to the periphery of the seal plates and to the ends of the radial beams. Attached to the lower end of the shroud is a cylindrical fairing. The shroud and fairing protect the forward container area from aerodynamic loads. Helium spheres and retromotors are mounted on the spider beam.

7-22. MISCELLANEOUS.

A systems tunnel is attached externally to each of the four fuel containers. Each tunnel joins the tail section and second stage adapter. Three of the tunnels shield electrical cables; the other is for routing tubing. The tunnels are constructed in sections to permit easy removal for maintenance and repair.

A conical shaped fairing extends forward from the aft ends of the propellant containers. It fairs the area between the containers and the 270-inch diameter forward



Note:
LOX/SOX Disposal System
Omitted for Clarity

3-533

Figure 7-10. Second Stage Adapter, S-I

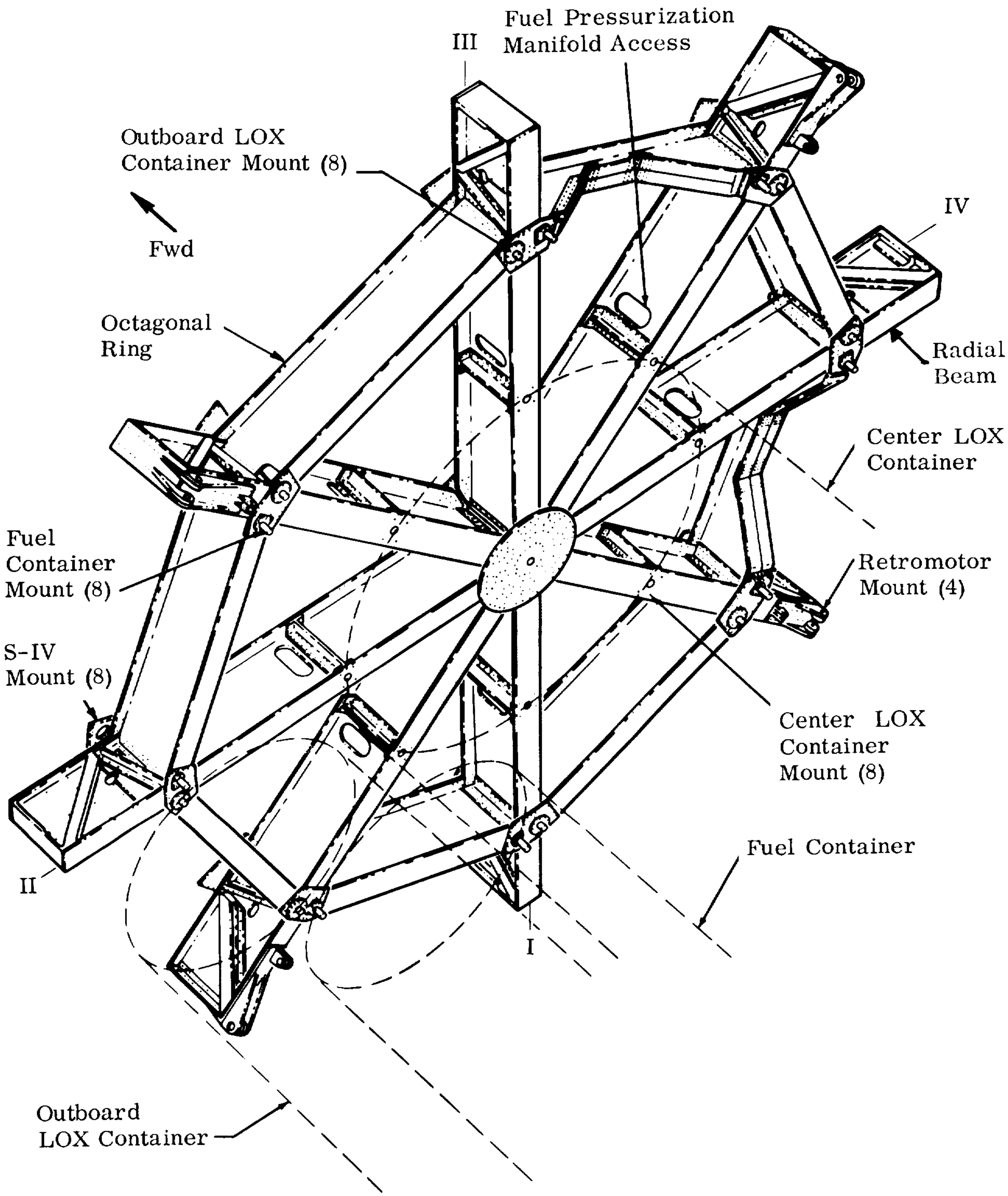


Figure 7-11. Spider Beam, S-I

shroud panels. The exterior of the fairing is coated with an ablative insulation.

Three LH_2 chilldown vent lines are located on the exterior of the vehicle. These lines connect to the LH_2 chill-down vent lines on the S-IV aft interstage. The lines run aft and are ducted through three of the stub fins.

7-23. S-IV STRUCTURAL CONFIGURATION.

The S-IV stage structure, Figure 7-12, is approximately 497 inches (41.4 feet) long and 220 inches (18.3 feet) in diameter. An aft interstage, an aft skirt, a thrust structure, a base heat shield, two propellant containers, and a forward skirt are structurally joined to make up the stage.

7-24. AFT INTERSTAGE.

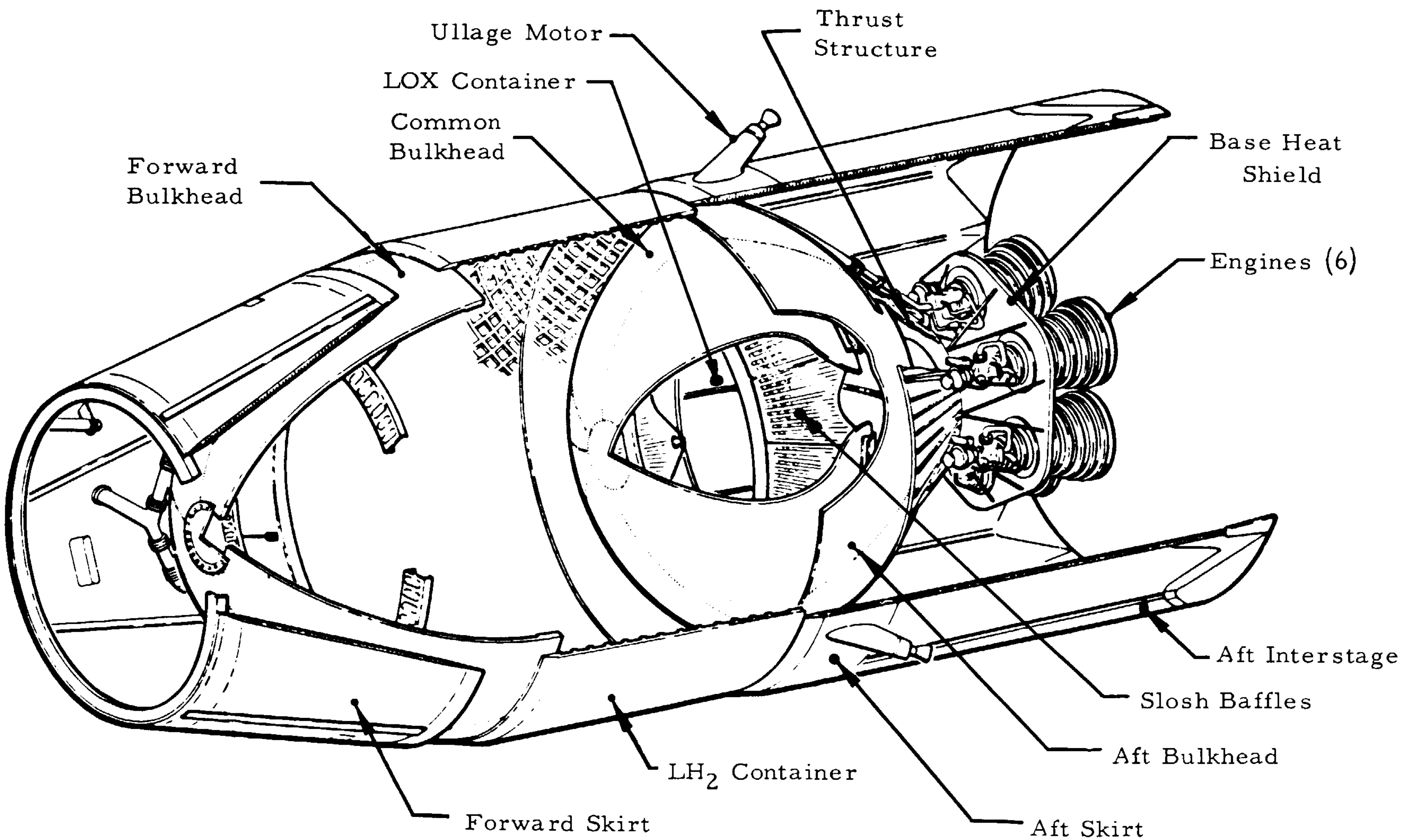
Loads from the first stage are transmitted to the S-IV stage through the aft (S-I/S-IV) interstage. The interstage, a cylinder approximately 184 inches long, is constructed of eight 45-degree cylindrical segment panels joined by longitudinal splices. The panels are of honeycomb sandwich construction consisting of 7075 aluminum alloy faces bonded to a 5052 aluminum alloy core.

Loads are introduced to the interstage at eight points through a field splice with the S-I stage at MSFC station 962. The loads are carried forward by tapered longerons which shear the concentrated loads into the sandwich panels. The panels distribute the loads uniformly to the aft skirt.

Between the longerons, at the aft end of the structure, are triangular vent ports covered with fabric blowout panels. The panels are removable to permit servicing of equipment within the structure. Three hydrogen chill-down vent lines are mounted on the exterior of the structure.

7-25. AFT SKIRT.

Loads from the S-I stage are transmitted to the LH_2 container through the aft skirt. The skirt is approximately 48 inches long and is constructed of eight 45-degree cylindrical segment panels joined by longitudinal splices. The panels are of honeycomb sandwich construction consisting of 7075 aluminum alloy faces bonded to a 5052 aluminum alloy core. The skirt and aft interstage are attached by explosive



3-517

Figure 7-12. S-IV Stage Structure

bolts. When fired, the bolts allow the S-IV stage to separate from the first stage. (The separation occurs at MSFC station 1147.) The skirt is welded to the LH_2 container at the tangent point of the aft bulkhead.

Four ullage motors and fairings are mounted on the exterior of the aft skirt. Cutouts are provided in the aft skirt for the umbilical plate, propellant fill and topping lines, oxygen vent line and ground air conditioning line.

7-26. THRUST STRUCTURE.

The thrust structure transmits engine thrust loads to the LOX container. The 7075 aluminum-alloy structure is a conical frustum with the following approximate dimensions: an aft diameter of 98 inches, a forward diameter of 170 inches and a length of 60 inches. The skin slope is tangent to the LOX container aft bulkhead at the interface. The six engines, mounted on a 92-inch diameter, are canted 6 degrees from the vehicle centerline. Two control actuators for each engine are also supported by the thrust structure. Lateral loads (resulting from engine gimballing and cant angle) and axial loads are transmitted from the gimbal bearing joints to the LOX container aft bulkhead through the thrust structure thrust beams, skin and stringers.

The skin and stringers are supported by an aft ring, two internal intermediate rings, and a forward ring. Lateral loads are sheared by the aft ring into the thrust structure skin. Axial loads are transmitted from the aft ring through the thrust beams and external longitudinal hat section stringers to the forward ring. The forward ring is attached to a milled land on the LOX container aft bulkhead. Loads transmitted from the forward ring are distributed to the LOX container aft bulkhead.

7-27. BASE HEAT SHIELD.

The base heat shield protects the forward propulsion area from engine heat. The heat shield is located approximately 48 inches aft of the engine gimbal plane and is supported from the thrust structure. The heat shield is an insulated honeycomb sandwich panel. Cutouts in the panel, sealed with flexible curtains attached to the engines and heat shield, provide clearance for the engine gimballing action.

7-28. LIQUID OXYGEN CONTAINER.

The LOX for the S-IV stage is contained in a 2014 aluminum-alloy container. Two bulkheads, an aft and a common, are attached through two rings to form the container. The aft bulkhead, a hemisphere with a spherical radius of 110 inches, is constructed of six gores and a circular center piece welded together. It is designed to support flight pressurization and propellant loads resulting from acceleration.

The other bulkhead, termed a common bulkhead because it is common to both the LOX and LH_2 containers, is a spherical segment with a spherical radius of 110 inches. The common bulkhead is of honeycomb sandwich construction consisting of 2014 aluminum alloy faces bonded to a fiberglass core. The common bulkhead has sufficient insulating properties to prevent the LOX from freezing during a 12-hour ground-hold period. Two compression rings are welded to the periphery of the bulkhead. These rings are attached to the aft bulkhead by welds and mechanical fasteners.

A milled land on the aft bulkhead provides a mounting surface for the engine thrust structure. Engine thrust loads are transmitted through the land, to the aft bulkhead, and are then carried into the LH_2 container cylindrical section.

Ring baffles of aluminum alloy are installed in the container to prevent sloshing of the LOX. The baffles are supported by a sheetmetal conical frustum which is attached to the aft bulkhead at the common bulkhead joint. A manhole in the center of the aft bulkhead provides access to the container. Outlet fittings in the sump at the bottom of the bulkhead are provided for six LOX engine lines and for two vent lines. A screen in the aft bulkhead over the engine line outlets retards formation of vortices during draining.

7-29. LIQUID HYDROGEN CONTAINER.

The LH_2 for the S-IV stage is contained in a 2014 aluminum-alloy container 257 inches long. The container is composed of a cylindrical section closed at the forward end by a hemispherical bulkhead, and closed at the aft end by the LOX container (discussed above). The forward bulkhead and LOX container aft bulkhead are welded to the cylindrical section.

Designed to support flight pressurization loads, the forward bulkhead is constructed of six gores and a circular center piece welded together to form a hemisphere. The bulkhead has a spherical radius of 110 inches. Three openings are provided in the bulkhead; one for container access and two for hydrogen vent lines.

The LH_2 cylindrical section is designed to carry pressurization, propellant loads due to acceleration, and external flight loads. It is composed of three 120-degree cylindrical segments each 110 inches long. Each segment is machine milled on the internal surface to a square waffle pattern with a 45-degree skew angle. The segments are welded into a cylinder. The waffle stiffeners provide sufficient buckling strength to give the structure a free-standing capability when the container is unpressurized. First stage loads are introduced into the LH_2 container through a weld joint connecting the container to the aft skirt. The LH_2 container transmits loads to the forward skirt through a weld joint on the forward bulkhead. Six LH_2 engine line-outlet fittings covered with antivortex screens are located just forward of the aft bulkhead-common bulkhead joint.

With the exception of the common bulkhead, all inside surfaces of the liquid hydrogen container are insulated with polyurethane foam. Bonded to the container walls, the insulation limits hydrogen boiloff during launch operations and flight.

7-30. FORWARD SKIRT.

The forward skirt (forward interstage) transmits the loads from the LH_2 container to the instrument unit. The skirt is a conical frustum approximately 130 inches long with an aft diameter of approximately 214 inches, and a forward diameter of 154 inches. The slope of the forward skirt is tangent to the LH_2 container bulkhead at the aft interface. The skirt is constructed of eight 45-degree conical segment panels joined by longitudinal splices. The panels are of honeycomb sandwich construction consisting of 7075 aluminum alloy faces bonded to a 5052 aluminum alloy core. Loads are transmitted to the panels through a weld joint at the LH_2 forward bulkhead. From the panels, the loads are transmitted to the forward ring which provides an interchangeable mating face for the attachment of the instrument unit (a field splice at MSFC station 1460).

A door in the forward skirt provides access to the equipment installations, and cutouts are provided for the hydrogen vent line, telemetry antennas and range safety antennas. Mounting provisions for two retromotors (which may not be installed) are located on the forward skirt.

7-31. SYSTEMS TUNNEL AND EXTERNAL FAIRINGS.

The systems tunnel, designed to accommodate cables and tubing, is located externally on the S-IV stage body and extends from the aft skirt to the forward skirt. The fairings are designed to carry aerodynamic pressure and thermal loads.

7-32. INSTRUMENT UNIT STRUCTURAL CONFIGURATION.

The instrument unit, Figure 7-13, structure transmits loads from the S-IV stage to the payload. The aluminum-alloy structure is 154 inches (12.83 feet) in diameter and 34 inches long.

Axial load and bending moment are carried by internal longitudinal hat-section stringers and the shear load is carried by the skin. The aft and forward rings provide mating faces for attachment to the adjacent structures. Loads are transmitted to the aft ring by the S-IV stage through a field splice at MSFC station 1460. The aft ring transmits axial and shear load to the stringers and skin. Loads are transmitted by the forward ring to the payload at MSFC station 1494. Internal longitudinal stringers, attached to the skin and rings, provide support for the equipment mounting plates.

Access to the instrument unit is through the S-IV stage forward skirt. Cutouts are provided in the skin for the umbilical plate, stabilized platform window, antennas, and vents. Four equally spaced vents, located at the forward end of the instrument unit, provide a common environment for the S-IV forward skirt, the instrument unit, and the spacecraft adapter.

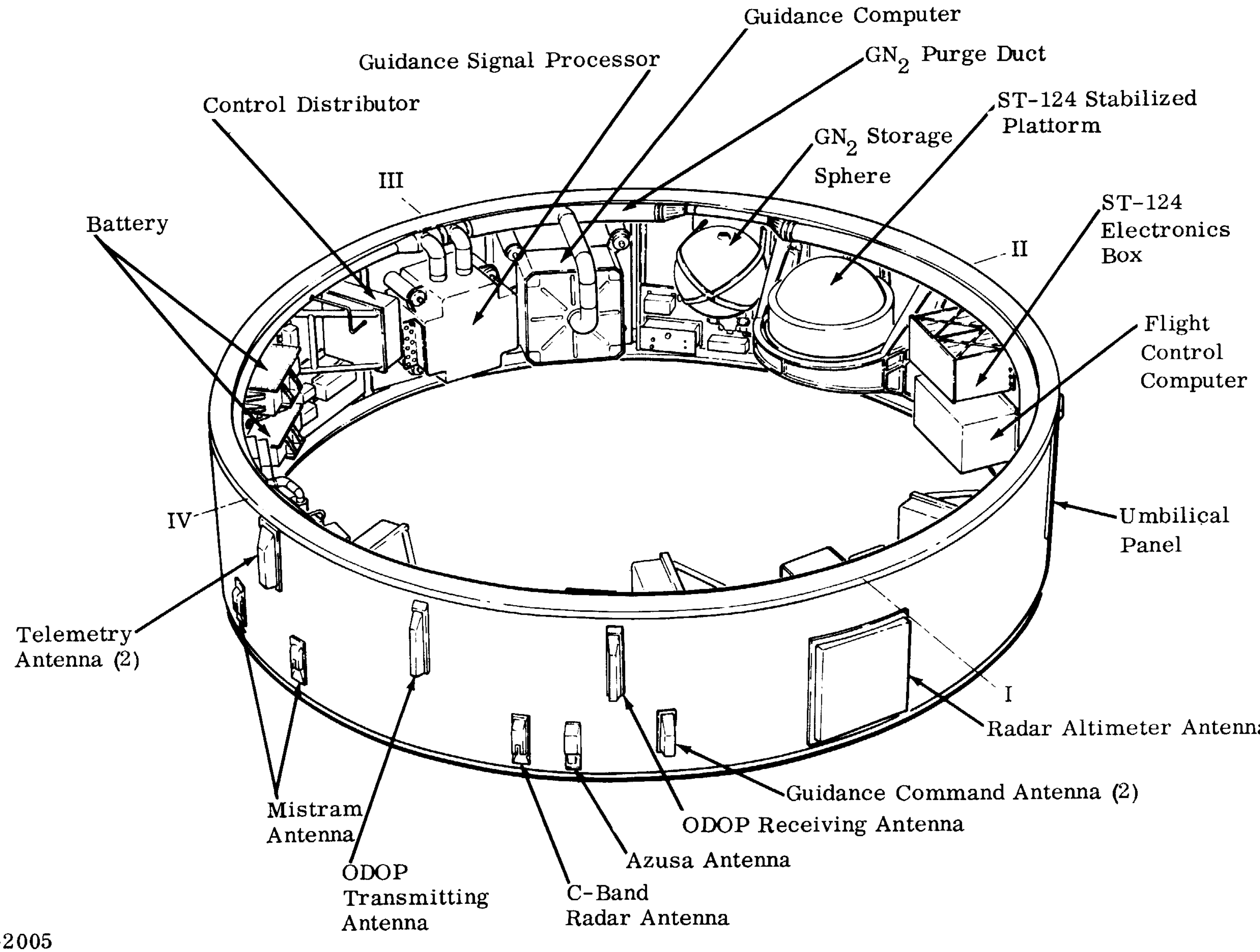


Figure 7-13. Instrument Unit, Saturn I

CHAPTER 2

SECTION VIII
PROPULSION

VIII

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SECTION VIII.
PROPULSION

8-1. REQUIREMENTS.

The Saturn I propulsion system is required to launch and insert a 22, 500 pound Apollo spacecraft into a nominal 100-nautical mile circular earth orbit or to perform other launch and insertion missions with an equivalent energy envelope. The system is required to function during both the launch and ascent phases of the mission. Propellant systems and propulsion devices (engines) constitute the propulsion system.

A two-stage launch vehicle provides the necessary impulse. First stage cutoff occurs at an altitude of 38-nautical miles and a velocity of approximately 6000 miles per hour. Second stage cutoff occurs at a nominal altitude of 100 nautical miles at a velocity of approximately 17, 000 miles per hour. Thrust vector control is required to maintain vehicle attitude orientation and angular rates as defined by the control system and, in addition, to damp the amplitude of the first bending mode oscillations of the structure during first stage operation.

An additional series of impulses are required to ensure successful staging. Both retrothrust to decelerate the first stage and ullage thrust to accelerate the second stage are necessary to aid separation. The ullage thrust also settles the propellants in the aft end of the containers insuring a sufficient suction head to prevent propellant pump cavitation at engine start. (Refer to Paragraph 9-13.)

During the launch phase a rapid fill and drain capability is required of the propellant storage and feed system due to the highly volatile properties of the cryogenic propellants (LH_2 and LOX). Provisions for the purging of the containers and feed lines before filling or after draining operations are required as part of the propellant storage and feed system. During the ascent phase the system must be capable of storing the propellants and delivering them as required to the engines.

8-2. OPERATION.

After the propellant containers have been loaded and pressurized (during the countdown), the eight S-I stage engines are started. The starting occurs in a pre-determined sequence a few seconds prior to liftoff. A total thrust of 1,500,000 pounds is provided at liftoff resulting in a thrust-weight ratio in excess of 1.25:1.00. The overall propulsion sequence is presented in Table 8-1.

As a result of decreasing ambient pressure as the vehicle ascends, the S-I stage thrust increases to 1,705,000 pounds at an altitude of 13.5 miles and due to under expansion decreases to 1,687,000 pounds prior to cutoff. Thrust vector and attitude control are provided by the four outboard gimballed engines (Figure 8-1) in response to commands from the control system. Engine cutoff results from a propellant depletion signal (level), cutting off the inboard, fixed, engines first, and later the outboard engines.

Prior to staging, the propellant pumps and engine feed lines of the S-IV stage are cooled down to prevent pump cavitation at engine start up. Cooldown is accomplished by venting propellants overboard through the feed lines and pumps.

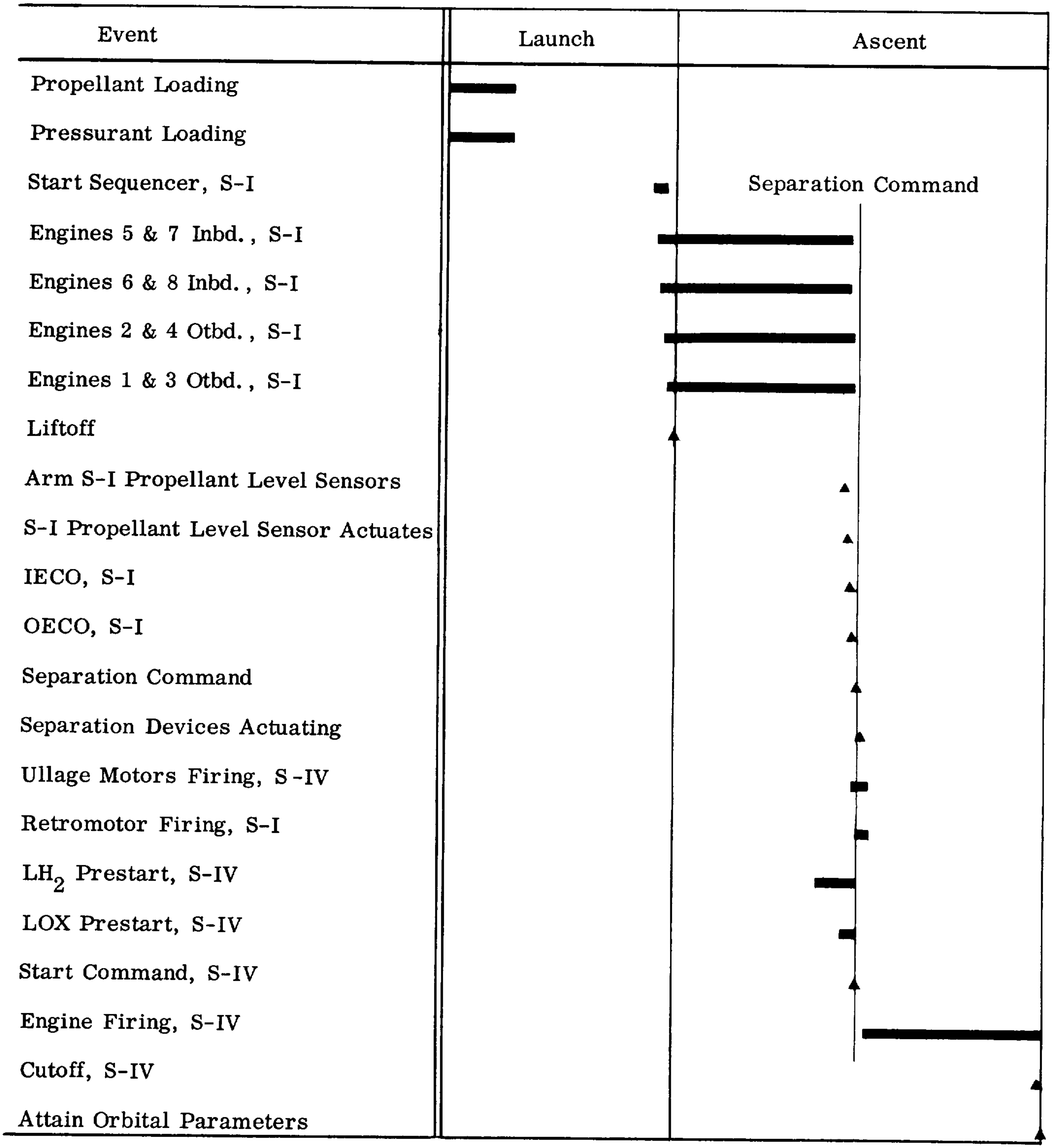
The S-IV engine start command is initiated in coincidence with the separation command. Several seconds later the six RL10A-3 engines reach a total rated thrust of 90,000 pounds. All engines gimbal to provide thrust vector control in response to commands from the control system. Roll control is provided by gimballing only four engines, Figure 8-2.

Engine cutoff occurs as a result of the termination of an electrical signal from the stage sequencer in response to a signal from the vehicle computer. The vehicle computer signal is applied prior to cutoff such that the total impulse delivered by the S-IV engines subsequent to the cutoff signal results in a velocity-to-go requirement of zero at thrust termination. This results in the attainment of proper orbital parameters.

8-3. S-I PROPULSION SYSTEM.

Two stages, the S-I and S-IV, and an instrument unit comprise the launch vehicle (Figure 5-1). The instrument unit provides initiation and control commands for the

Table 8-1. Saturn I Propulsion Sequence



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Operation

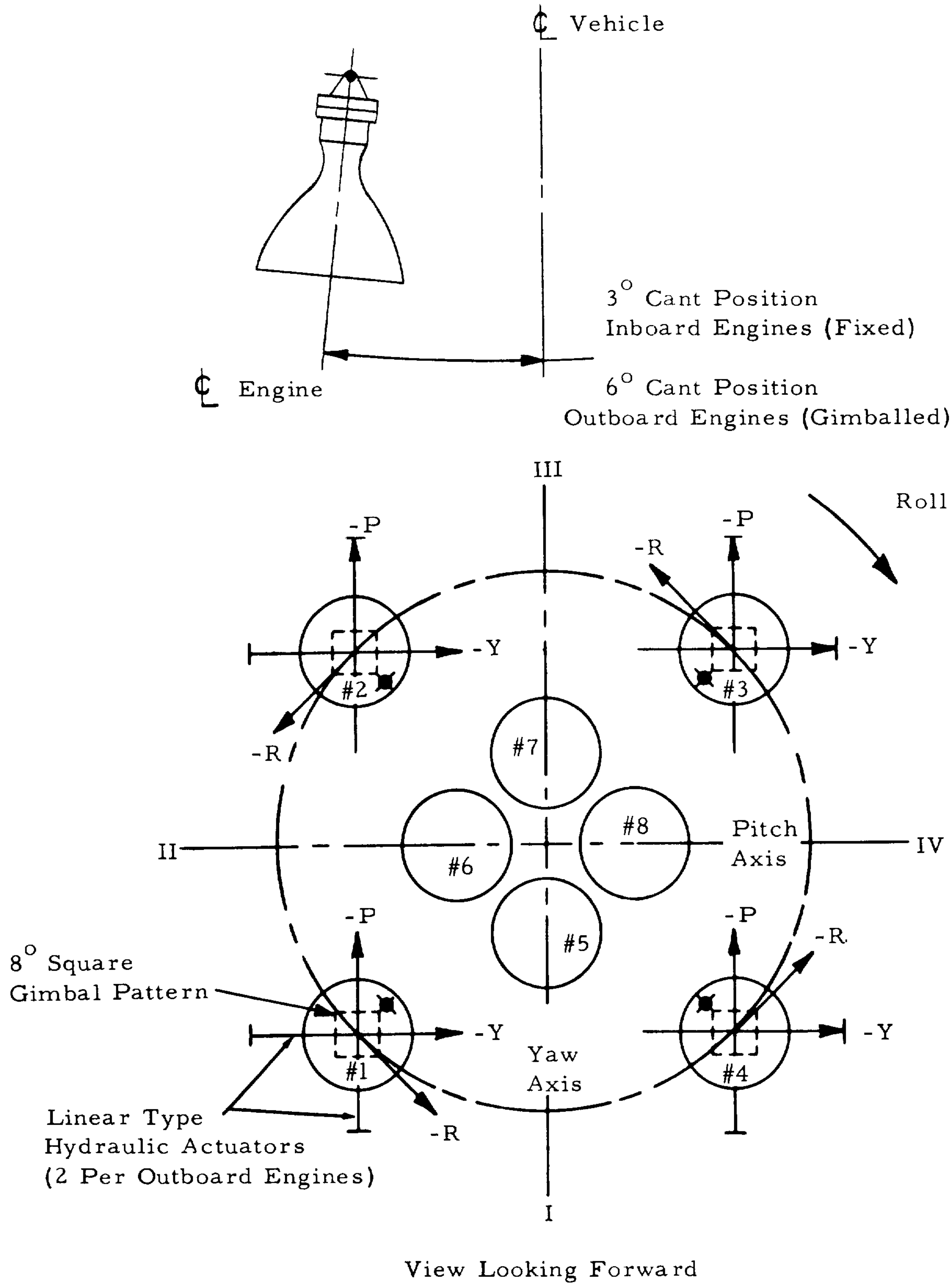

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Figure 8-1. Engine Location and Gimbal Pattern, S-I

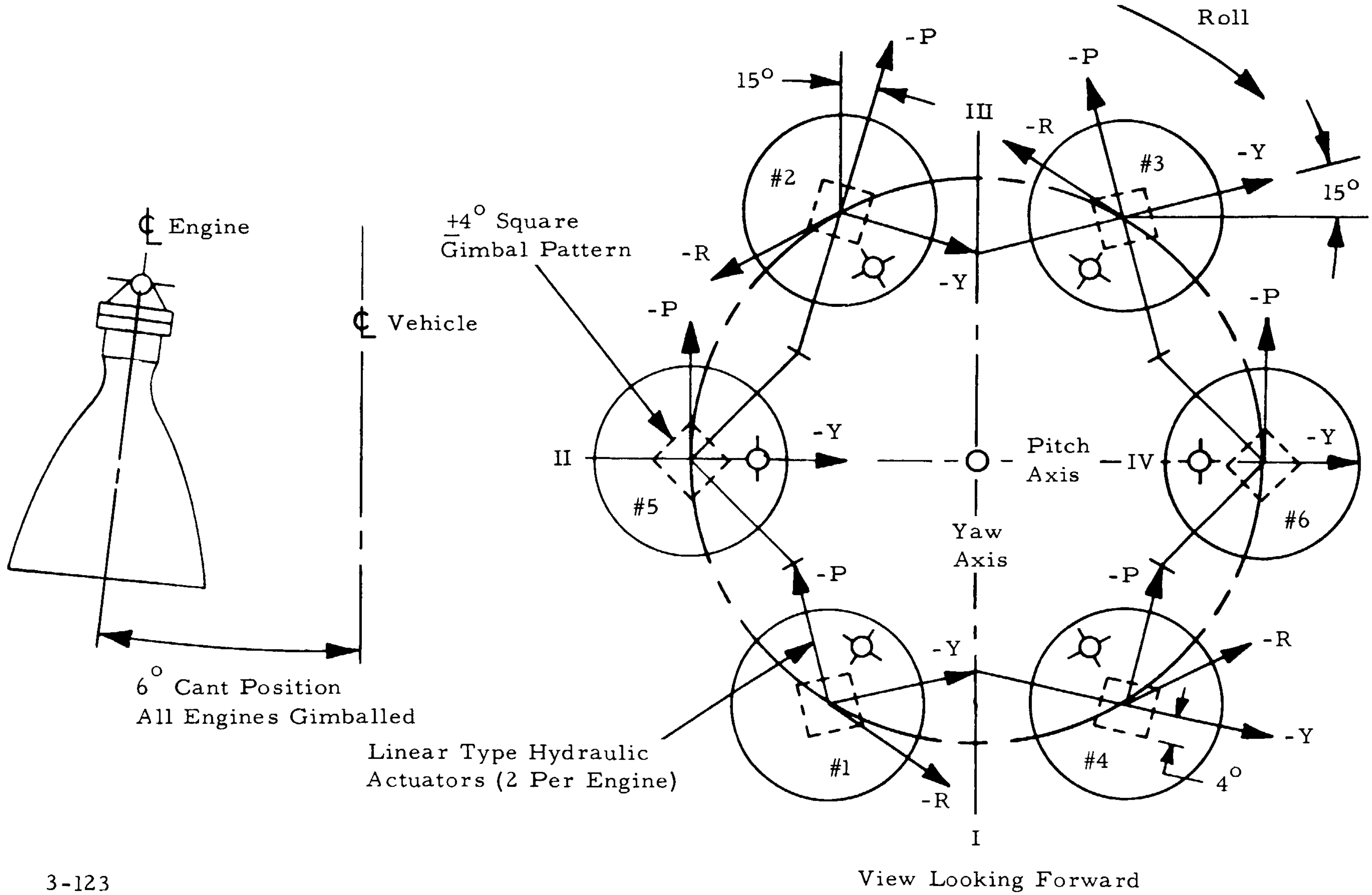

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Figure 8-2. Engine Gimbal Pattern and Cant Angles, S-IV propulsion system. (Refer to Paragraph 6-1.) Functionally, the S-I propulsion system is composed of eight Rocketdyne H-1 liquid-rocket engines and a propellant system.

8-4. ENGINE.

The H-1 engine, Figure 8-3, is a single start, fixed thrust, bi-propellant engine using LOX as oxidizer and RP-1 as fuel. The RP-1 is also used as turbopump lubricant (with additive) and propellant valve control fluid. A hypergolic mixture is used for the primary ignition of the propellants. Performance parameters and physical characteristics of the H-1 engine are given in Tables 8-2 and 8-3.

Table 8-2. H-1 Engine Performance Parameters

Item	Parameter
Nominal engine thrust (sea level)	188,000 ± 3 percent pounds
Nominal specific impulse (sea level)	256.2 seconds
Engine mixture ratio	2.23
Oxidizer flow	505.5 pounds per second
Fuel flow	226.7 pounds per second
LOX pump NPSH* (minimum)	35.0 feet
Fuel pump NPSH* (minimum)	35.0 feet

*Net Positive Suction Head

Table 8-3. H-1 Engine Physical Characteristics

Item	Characteristic
Weight, dry (outboard)	1959 pounds
Weight, wet (outboard)	2199 pounds
Over-all engine length (outboard)	104 inches
Over-all engine length (inboard)	101 inches
Throat diameter	16.2 inches
Nozzle exit diameter	47.6 inches
Expansion ratio	8:1

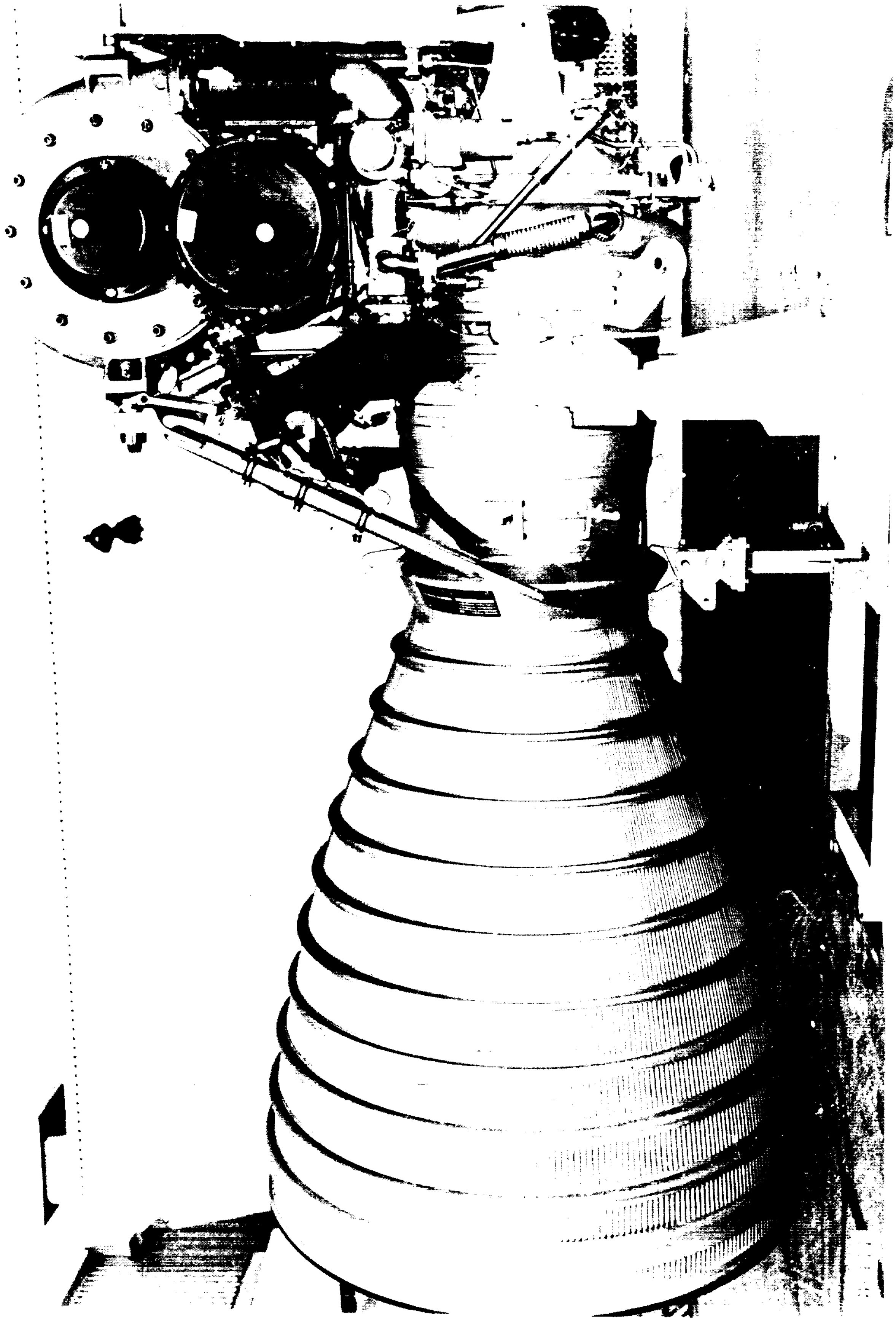

Figure 8-3. H-1 Engine

The four inboard engines are equally spaced on a 64-inch diameter and are canted 3 degrees from the vehicle roll axis. The four outboard engines are oriented 45 degrees from the inner engines and located on a 190-inch diameter. Each outboard engine is gimbal mounted to permit a ±8-degree pattern from the null position (Figure 8-1). Gimbal control is accomplished by two hydraulic actuators mounted on the engine circumference 90 degrees apart. These engines are canted 6 degrees from the vehicle roll axis at the null position of the two actuators. This minimizes pitch and yaw disturbances that may result from thrust variation or total loss of an engine prior to stage separation.

The primary subassemblies of the engine are the thrust chamber, gas generator, turbopump, propellant valves, and ignition subsystem. A brief description of each follows (refer to Figure 8-4).

8-5. Thrust Chamber. The thrust chamber (23) receives propellants under turbopump pressure. The propellants are then burned and expelled through a supersonic nozzle which is designed to provide a high time-weighted average specific impulse. This is achieved with a nozzle which is over expanded at sea level and under expanded at burnout altitude. The thrust chamber propellant flow rate is nominally 732.2 pounds per second. The out-board engine thrust chambers ar equipped with aspirators (22) for control of turbine exhaust gases. The thrust chamber consists of a LOX dome, gimbal, propellant injector, thrust chamber body, bleed valve, and drain plugs.

LOX Dome. The LOX dome distributes LOX to ring orifices and provides mounting for the engine gimbal.

Gimbal. The gimbal secures the thrust chamber to the stage thrust ring and permits thrust chamber pivotal movement. The gimbal is a universal joint mounted on perpendicular thrust alignment slides.

Propellant Injector. The propellant injector meters the propellants into a prescribed pattern to ensure efficient combustion. The injector incorporates 21 rings of propellant nozzles; the outer ring and alternate inner rings are fuel nozzles. The orifices are angled to produce a like-on-like (fuel-on-fuel and LOX-on-LOX) impingement. The injector is also the primary thrust-bearing component. Thrust chamber combustion pressure acts on the face of the injector, which transmits the thrust to the LOX dome, the gimbal bearing assembly, and subsequently to the vehicle structure.

Thrust Chamber Body. The thrust chamber body is a cylindrical venturi (convergent-divergent) unit with a 205.5-square inch throat and an expansion ratio of 8:1. The chamber body wall is constructed of longitudinal nickel tubes joined by silver brazing and retained by external stiffening rings and tension bands. The tubes are of a variable rectangular cross-section and are shaped to conform to the longitudinal thrust chamber contour. This method of construction permits circulation of fuel through the chamber walls, providing thrust-chamber cooling and fuel preheating.

Bleed Valve and Drain Plugs. A bleed valve, located on the fuel injector manifold, provides venting during fuel jacket wet-start filling and draining. Four drain plugs provide fuel-jacket draining.

8-6. Gas Generator. A liquid-propellant gas generator produces hot gases at the rate of 17 pounds per second during rated operation for driving the turbine (14). The gas generator operates on LOX and RP-1 fuel bootstrap flow ignited by the turbine spinner (15) hot gases. A fuel-rich mixture ratio is used to prevent excessive temperature within the gas turbine. The gas generator assembly consists of a control valve assembly, two auto igniters, and a gas generator combustion chamber. During engine starting the solid-propellant turbine spinner (15) supplies power to the turbine for starting. The turbine spinner contains two initiators which are in parallel to provide redundancy for igniting the solid-propellant charge used to supply hot gas at the rate of 4.7 pounds per second for approximately one second.

The control valve assembly (19), actuated by thrust chamber injector-manifold fuel pressure, controls the flow of bootstrap propellants into the gas generator. A leakage line, from the control-valve assembly, vents opening-port fuel leakage overboard. The two auto igniters are used to ensure ignition of the bootstrap propellants if not ignited by hot gases from the solid-propellant gas generator charge.

8-7. Turbopump. A turbopump assembly supplies LOX and fuel to the thrust chamber at the required pressure and flow rates to maintain engine operation at rated thrust.

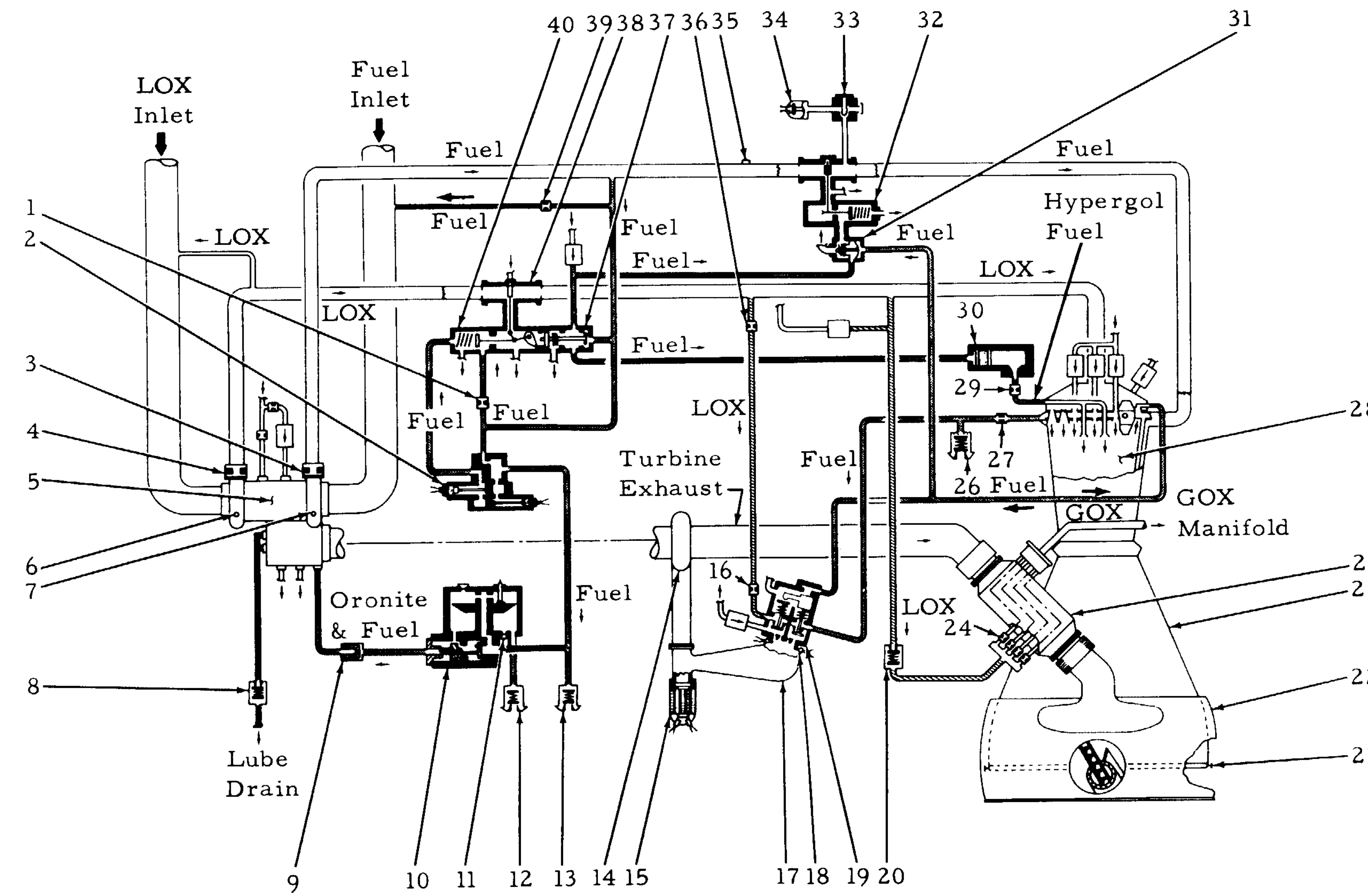

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Figure 8-4. H-1 Engine Schematic

1 Orifice
2 Conax Valve
3 Fuel Discharge Orifice
4 Oxidizer Discharge Orifice
5 Fuel and LOX Turbopump
6 Drain Plug
7 Drain Plug
8 Check Valve
9 Filter
10 Fuel Additive Blender Unit
11 Drain Plug
12 Coupling Half
13 Coupling Half
14 Turbine
15 Turbine Spinner
16 Orifice
17 Gas Generator
18 Auto Ignitors
19 Control Valve Assembly
20 Check Valve
21 Drain Plug
22 Aspirator
23 Thrust Chamber
24 Orifice
25 Heat Exchanger
26 Coupling Half
27 Orifice
28 Combustion Chamber
29 Orifice
30 Hypergol Container
31 Ignition Monitor Valve
32 Main Fuel Valve
33 Three-way Needle Valve
34 Thrust OK Pressure Switch
35 Drain Plug
36 Orifice
37 Sequence Valve
38 Main LOX Valve
39 Orifice
40 Main LOX Valve Control

Figure 8-4. H-1 Engine Schematic (Cont'd)

The turbopump also supplies the gas generator with the required propellants. The turbopump assembly consists of a turbine, a gearbox, two propellant pumps, a blender, and a heat exchanger.

Turbine. A two-stage pressure compound impulse turbine (14) drives the turbopump through a reduction gear train; the turbine operates at 66-percent efficiency at 1200 degrees F; and develops 3800 shaft horsepower at 31,900 rpm.

Gearbox. A gearbox containing gear-train reduction (approximately 4.9:1) drives both propellant pumps from a common shaft. The gearbox also contains an accessory drive pad and a turbopump over-speed trip. Lubrication is provided by a fuel and additive (Oronite 262) mixture. The gearbox is pressurized with GN_2 to prevent rapid lube draining and lube vaporization at high altitude. Three drain lines (a lube drain, a LOX-seal drain, and a lube-seal drain) pass leakage lubricant overboard at the engine exhaust plane.

Propellant Pumps. The centrifugal propellant pumps (fuel and LOX) (5) are mounted back-to-back on either side of the gearbox, and operate nominally at 6540 rpm. The fuel pump requires 1480 bhp and the LOX pump requires 1970 bhp for nominal operation.

Blender. A fuel-additive blender (10) unit provides a mixture of fuel and additive (Oronite 262) for turbopump gearbox lubrication and cooling. The blender unit, operated by fuel pump discharge pressure acting through a fuel feeder line, is provided with: a storage cylinder used to store the additive, metering orifices and injectors used to control the flow of additive (2.75 $\pm$0.75 percent Oronite) and fuel to the gearbox, and a drain plug used to drain additive from the storage cylinder.

Heat Exchanger. The heat exchanger (25), located in the turbine exhaust duct, utilizes the hot exhaust gases to convert LOX to GOX for LOX container in-flight pressurization. Heat-exchanger exhaust is ducted through the vehicle tail shield on inboard engines and through the aspirators on the outboard engines.

8-8. Propellant Valves. There are five valves which control the propellants: a main fuel valve, a main LOX valve, a sequencer valve, a Conax valve, and an ignition monitor valve. The function of each is discussed below.

Main Fuel Valve. The normally closed main fuel valve (32) is installed in the high-pressure fuel line between the fuel pump and the thrust chamber. The main fuel valve has a 4.25-inch unbalanced butterfly gate and is spring loaded to the closed position. The valve is initially opened by turbopump fuel discharge pressure acting through the ignition fuel line and the ignition monitor. Three drain lines, one from the valve body and two from the actuator, transfer leakage fuel to a manifold which drains overboard.

Main LOX Valve. The normally closed main LOX valve (38) is installed in the high-pressure LOX line between the LOX pump and the LOX dome. The main LOX valve is of the same basic type as the main fuel valve. The LOX valve is initially opened by turbopump fuel discharge pressure acting through a control line. The actuation cylinder on the main LOX valve is equipped with a heater blanket to prevent seals from freezing due to the extreme low temperature. Two LOX-leakage drain lines, one from the main valve body and one from the actuator, vent LOX overboard. A fuel overboard drain line is provided from the closing port of the main LOX-valve actuator cylinder.

Sequencer Valve. The sequencer valve (37), attached to and operated by a cam located on the main LOX-valve actuator shaft, controls ignition fuel flow sequencing during engine start. The sequence valve opens when the main LOX valve is approximately 80 percent open, and closes when the main LOX valve is approximately 20 percent closed. The sequence valve is equipped with a heater to prevent seal freezing. A fuel leakage drain line from the sequence valve drains fuel overboard.

Conax Valve. A Conax valve (2) closes the main LOX valve for H-1 engine cut-off. The Conax valve is located in a fuel control line which leads to the closing port of the main LOX valve actuator. The Conax valve consists of two redundant self-contained, pyrotechnic actuated, two-way normally closed control valves. Firing of either or both pyrotechnic charges moves a piston which bursts a metallic membrane and allows the control line fuel to flow to the closing port of the main LOX valve actuator, equalizing the pressure and permitting the valve to close under spring tension.

Ignition Monitor Valve. The normally closed ignition monitor valve (31) is operated by 28 psig pressure from the fuel injector during primary ignition. The valve opens to allow fuel igniter line pressure to energize the main fuel valve actuator. A fuel leakage drain line leads from the ignition monitor valve to drain overboard.

8-9. Ignition Subsystem. The ignition subsystem consists of the hypergol assembly and ignition fuel ducting. The hypergol container (30) contains a hypergolic fluid which ignites the main propellants when they reach the thrust chamber. The hypergol container is located in the fuel igniter lines between the sequence valve and thrust chamber. It contains burst diaphragms which rupture when fuel igniter line pressure, reaches approximately 300 psig, and hypergolic fluid (triethyl aluminum) which ignites spontaneously upon contact with LOX.

8-10. ENGINE OPERATION.

For structural considerations the H-1 engines are started in pairs: inboard engines 5 and 7; inboard engines 6 and 8; outboard engines 2 and 4, and outboard engines 1 and 3. The engine starting and cutoff sequences are described below.

8-11. Engine Starting Sequence. The ignition sequence for an engine is shown in Figure 8-5. During ignition, the following occurs:

a. The turbine spinner (15) receives an electrical start signal and two initiators ignite the solid-propellant charge.

b. Hot, high-pressure gases formed by the burning of solid propellant are forced through the gas turbine (14) which, in turn, drives the fuel (7) and LOX (6) pumps.

c. Fuel from the pump volute is forced through the discharge line to the inlet side of the normally closed main fuel valve (12). Fuel from the discharge line is also directed to a fuel control line which branches into:

(1) The normally-closed sequence valve (37).

(2) The main LOX valve control (40) by way of an orifice.

(3) The Conax valve (2).

(4) The fuel-additive blender unit (10).

(5) A bleed line, containing an orifice leading to the fuel pump suction line.

d. LOX from the pump volute is forced through an orifice into the LOX discharge line and the inlet side of the normally closed main LOX valve (38). A bleed line exists between the LOX discharge line and the container pump suction line (some LOX recirculation occurs).

e. Turbopump acceleration produces pressure build-up in the fuel control line. The increasing pressure is applied to the normally closed sequence valve (37) and to the main LOX valve control (40) by way of an orifice. Spring-closing pressure in the main LOX valve control is overcome when the control line fuel pressure reaches approximately 230 psig, and the valve begins to open, allowing LOX to flow through the supply line, LOX dome, and LOX injector nozzles into the thrust chamber. LOX also flows from the supply line through a four coil heat exchanger installation (25) containing a check valve and four orifices. The vaporized LOX from the heat exchanger pressurizes the vehicle LOX containers.

f. A mechanical linkage opens the sequence valve when the main LOX valve is approximately 80 percent open, and allows control line fuel pressure to flow into the hypergol container (30) and the inlet port of the normally closed ignition monitor valve (31).

g. Hypergol container burst diaphragms rupture as control line fuel pressure increases to approximately 300 psig. This allows hypergolic fluid, followed by igniter

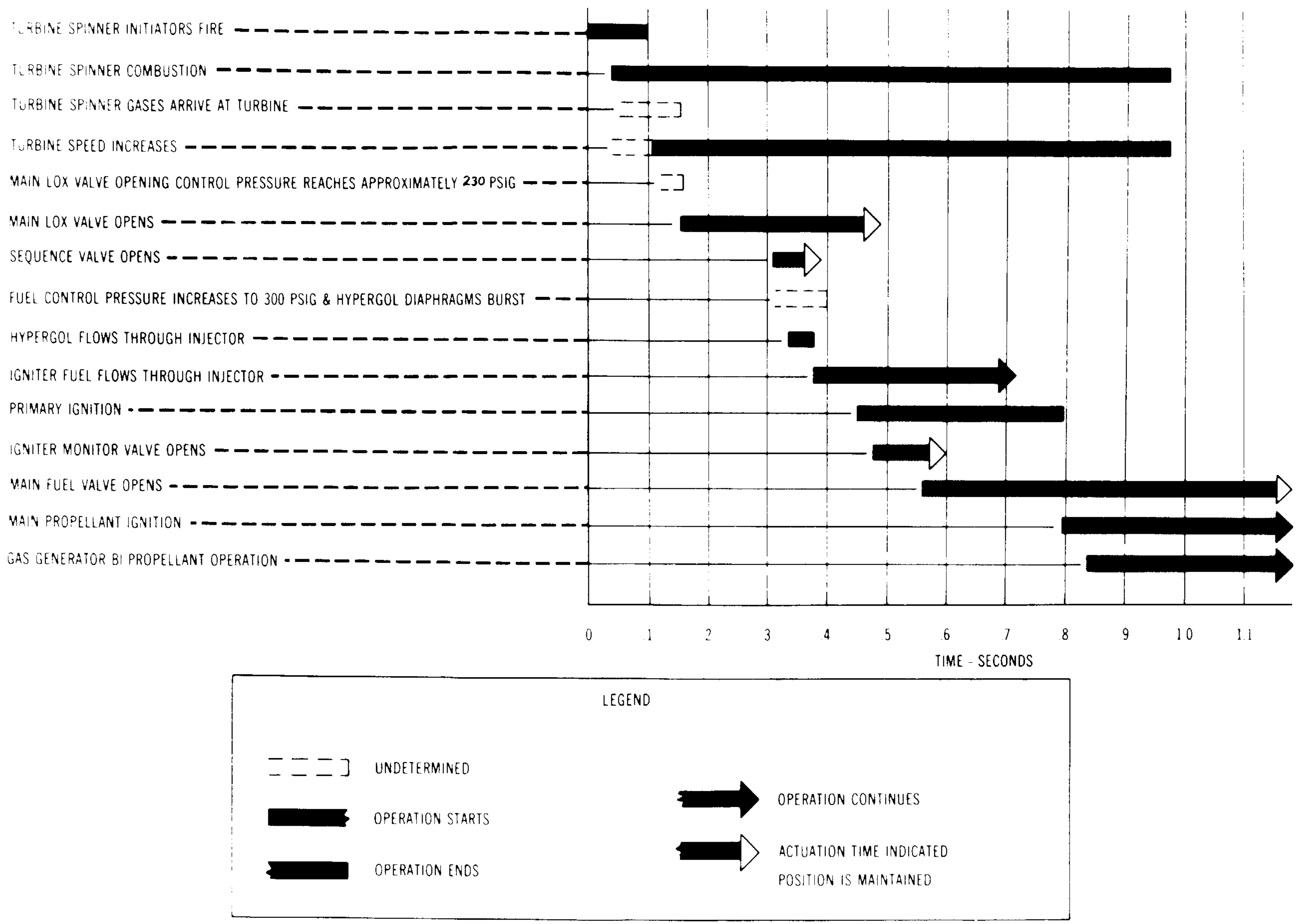

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Figure 8-5. H-1 Engine Ignition Sequence fuel, to flow through an orifice (29) to the injector ignition fuel spray nozzles and into the thrust chamber. The hypergolic fluid and igniter fuel mixture ignite upon contact with the previously injected LOX, resulting in primary ignition.

During the transition period, the following events occur:

a. The ignition-monitor valve opens when the fuel injector manifold pressure reaches approximately 28 psig and allows fuel igniter line pressure to overcome spring-closing pressure in the main fuel valve.

b. The main fuel valve (32) opens and allows fuel to flow through the thrust chamber fuel jacket into the fuel injector manifold and into the thrust chamber. The fuel then combines with the previously ignited LOX and igniter fuel, and main propellant ignition occurs. A thrust-OK pressure switch, located on the main fuel valve and calibrated by a hand valve is used to monitor fuel injector manifold pressure.

c. Main propellant ignition results in thrust chamber pressure buildup.

d. Fuel pressure in the fuel injector manifold becomes sufficient to open the port on the gas generator control valve assembly (19) allowing the following:

(1) Bootstrap fuel under turbopump pressure flows from the fuel injector manifold through the fuel bootstrap line, into the gas generator (17) by way of the control valve assembly fuel valve.

(2) Bootstrap LOX under turbopump pressure flows from the main LOX valve through the LOX bootstrap line, containing orifices, and into the gas generator by way of the LOX valve. (LOX leads the fuel into the gas generator to prevent detonation.)

e. The bootstrap propellants are ignited by the turbine spinner hot gases. Auto igniters, located in the gas generator combustion area, provide a secondary ignition system for the bootstrap propellants to ensure continuous operation of the gas turbine.

f. The gas turbine operates on combined turbine spinner and gas generator hot gasses for approximately 200 milliseconds. The turbine spinner, its solid propellant spent, then ceases operation, and the gas generator continues to power the gas turbine for the remainder of the engine operation.

8-12. Engine Cutoff Sequence. (Figure 8-6). Engine cutoff is accomplished by the pyrotechnically energized Conax valve (2), which may be actuated by various means prior to or during vehicle flight. During cutoff, the following events occur:

a. Engine cutoff may be initiated by means of automatic or manual ground controls and by automatic vehicle controls which actuate Conax valves in case of fire or equipment malfunctions.

(1) Any thrust-OK pressure switch on a failing engine may initiate an actuation signal to cutoff all engines from approximately 3.3 seconds after ignition until launch commit.

(2) During the period from launch commit until 10 seconds after liftoff a single engine cutoff may be initiated by the thrust-OK pressure switch.

(3) Any failing engine may be cut off by a thrust-OK pressure switch.

(4) After inboard engine cutoff, a thrust-OK cutoff signal will cut off all remaining outboard engines.

(5) The command system may signal engine cutoff any time after liftoff.

b. Normal engine cutoff is initiated by an electrical cutoff signal from any one of the five propellant container cutoff switches. The cutoff switches actuate and signal the Conax valves when one of the propellants is depleted to the cutoff switch actuation level.

c. An explosive charge within the Conax valve ignites, actuating the valve.

d. The main LOX valve closes under spring pressure and LOX ceases to flow to the engine thrust chamber and gas generator. As a result, ignition is terminated, causing thrust, turbopump speed, and discharge pressure to decay.

e. Spring-closing pressure in the main fuel control valve overcomes the decreasing fuel pressure (the fuel pressure drops to approximately 200 psig). The main fuel valve closes, shutting off the fuel supply to the thrust chamber and to the gas generator. (A fuel rich cutoff prevents excessive combustion temperatures which would damage the gas generator and turbine and also results in a relatively small, predictable cutoff impulse.) Within 150 milliseconds after the engine cutoff signal is received, engine cutoff operations are completed. Within 400 milliseconds, the engine thrust decays to less than 10 percent.

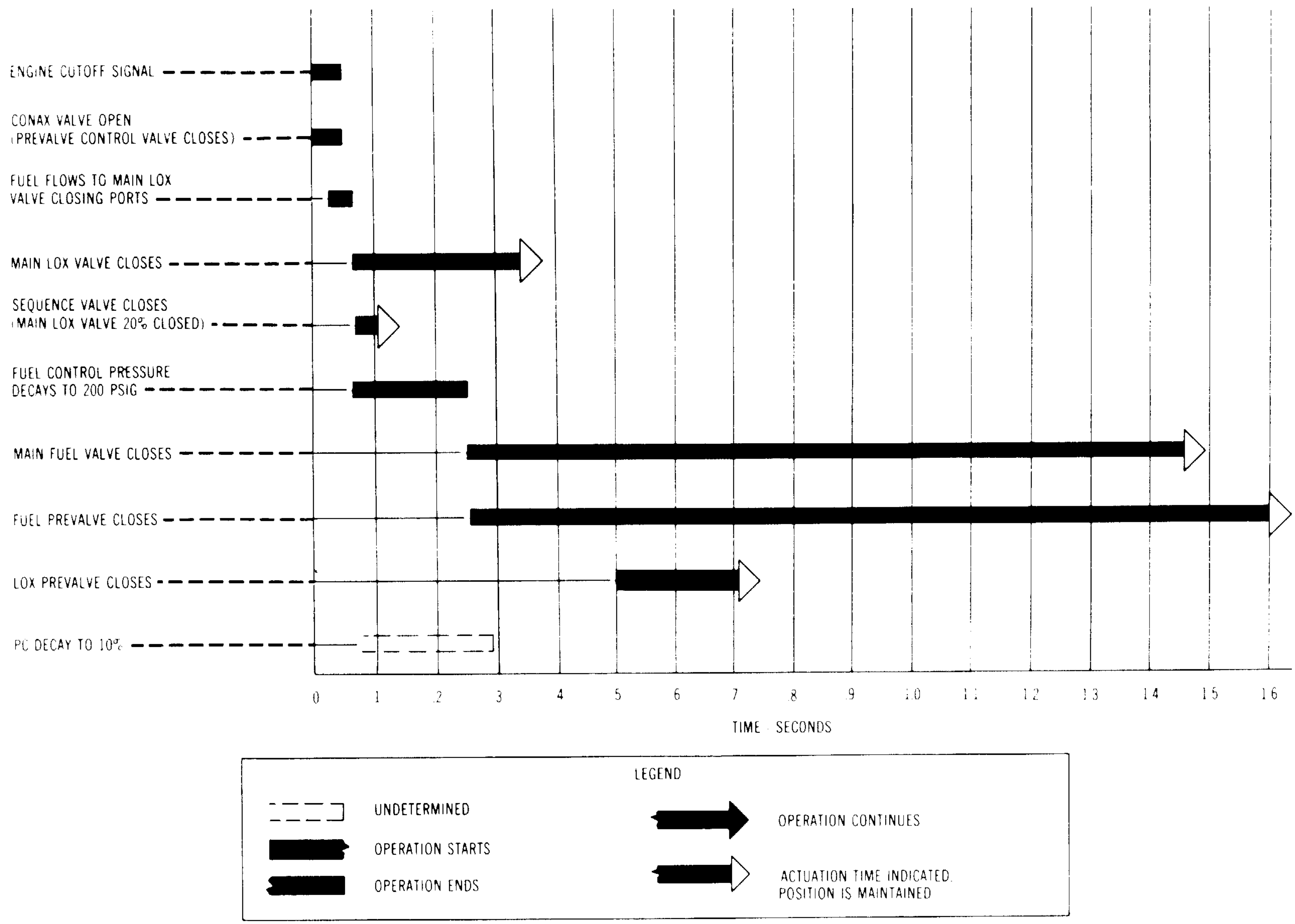

Figure 8-6. H-1 Engine Cutoff Sequence

f. Under normal cutoff conditions, the four inboard engines are simultaneously cut off, followed by cutoff of the four outboard engines upon LOX depletion.

8-13. PROPELLANT SYSTEM.

The propellant system consists of the following systems:

a. Fuel Storage and Feed
b. Oxidizer Storage and Feed
c. NPSH Pressurization
d. Control Pressurization
e. Propellant Conditioning
f. Propellant Loading
g. Purging

8-14. FUEL STORAGE AND FEED SYSTEM (FIGURE 8-7).

This system includes four fuel containers, upper and lower manifolds, and suction lines.

8-15. Fuel Containers. The fuel containers are mounted alternately with outer LOX containers around the central LOX container. Each container supplies one inboard and one outboard engine and has a capacity of 1419 cubic feet. An ullage volume is provided for expansion and pressurization reducing the actual fuel capacity of the container. Internal baffles are constructed in the containers to prevent fuel sloshing. Screens above the container sump filter the fuel and straighten the flow. Fuel level sensors located near the bottom of container F-2 and F-4 initiate inboard engine cutoff when the fuel reaches a predetermined level. A liquid level switch is located in the entrance to the suction line to indicate outboard engine cutoff should fuel deplete prior to LOX depletion cutoff of the outboard engines.

8-16. Upper Manifold. The upper manifold connects the tops of the four fuel containers and maintains pressure equalization between containers. Two vent valves contained in the manifold, pressure operated by a 750 psig nitrogen control line, open during container filling and draining. The valves also vent the fuel containers at 19 psig if overpressurization occurs. If either of the vent valves fails, an associated safety relief valve opens at 23 psig to release the pressure. When the engines are firing, the containers are pressurized through three GN_2 pressurization inlets.

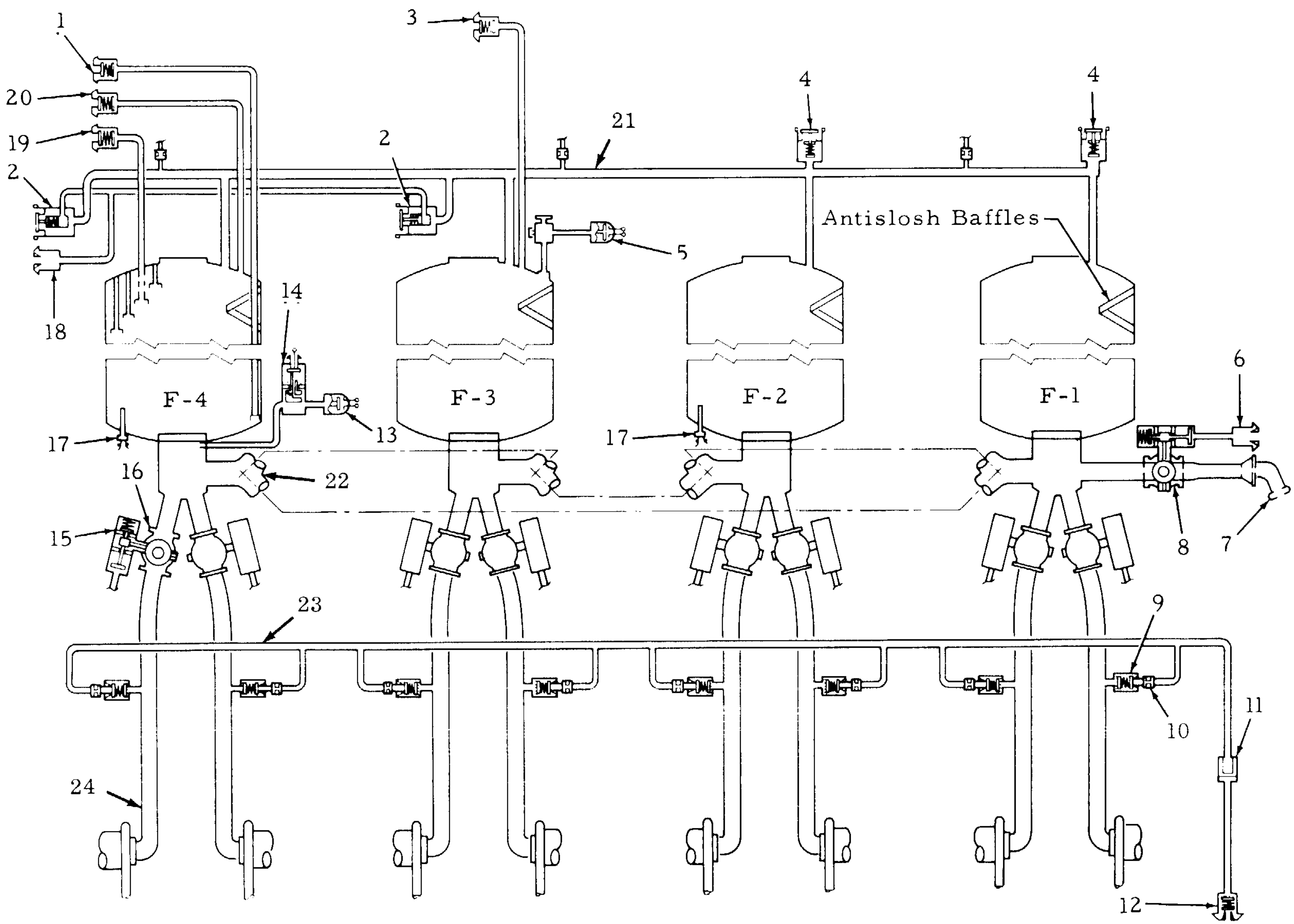

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Figure 8-7. Fuel Storage and Feed System, S-I

1 Quick-Disconnect Coupling
2 Vent Valve (2)
3 Quick-Disconnect Couplings
4 Safety Relief Valve
5 Pressure Switch
6 Quick-Disconnect Coupling
7 Fuel Quick-Disconnect Coupling Nozzle
8 Fuel Fill and Drain Valve
9 Check Valve (8)
10 Orifice Assembly (8)
11 Filter Assembly
12 Quick-Disconnect Coupling
13 Fuel-Step Pressure Switch
14 Calibration Valve
15 Control Valve (8)
16 Prevalve (8)
17 Fuel Level Sensor (2)
18 Quick-Disconnect Coupling
19 Quick-Disconnect Coupling
20 Quick-Disconnect Coupling
21 Upper Manifold
22 Lower Manifold
23 Manifold Ring Line
24 Suction Line (8)

Figure 8-7. Fuel Storage and Feed System, S-I (Cont'd)

8-17. Lower Manifold. The lower manifold interconnects the four fuel container sumps to maintain approximate uniform fuel level in the containers. In the event of engine failure, the manifold distributes most of the dead engine fuel to the other engines. A normally closed fuel fill and drain valve, and associated line provides a filling connection in the manifold.

8-18. Suction Lines. Eight-inch diameter suction lines supply fuel at a nominal rate of 227 pounds per second from the containers to the engine pumps. Two suction lines are connected to each fuel pump, one to an inboard engine line and one to an outboard engine line. Normally closed prevalves, located near the top of each fuel suction line, are actuated by GN_2 control pressure. The prevalves are opened prior to fueling and remain open except in case of emergency, such as engine failure or a broken line. The fuel containers are loaded from the launch complex storage containers in the following manner:

a. The normally closed vent valves in the upper manifold are pneumatically opened by GN_2 ground control pressure.

b. The normally closed fuel fill and drain valve is pneumatically opened by GN_2 ground control pressure.

c. Fuel is pumped under pressure from the ground storage containers through the fill and drain valve. The lower manifold distributes fuel to the four containers.

In the event of a cancelled launch, the containers are drained in a manner similar to the filling operation.

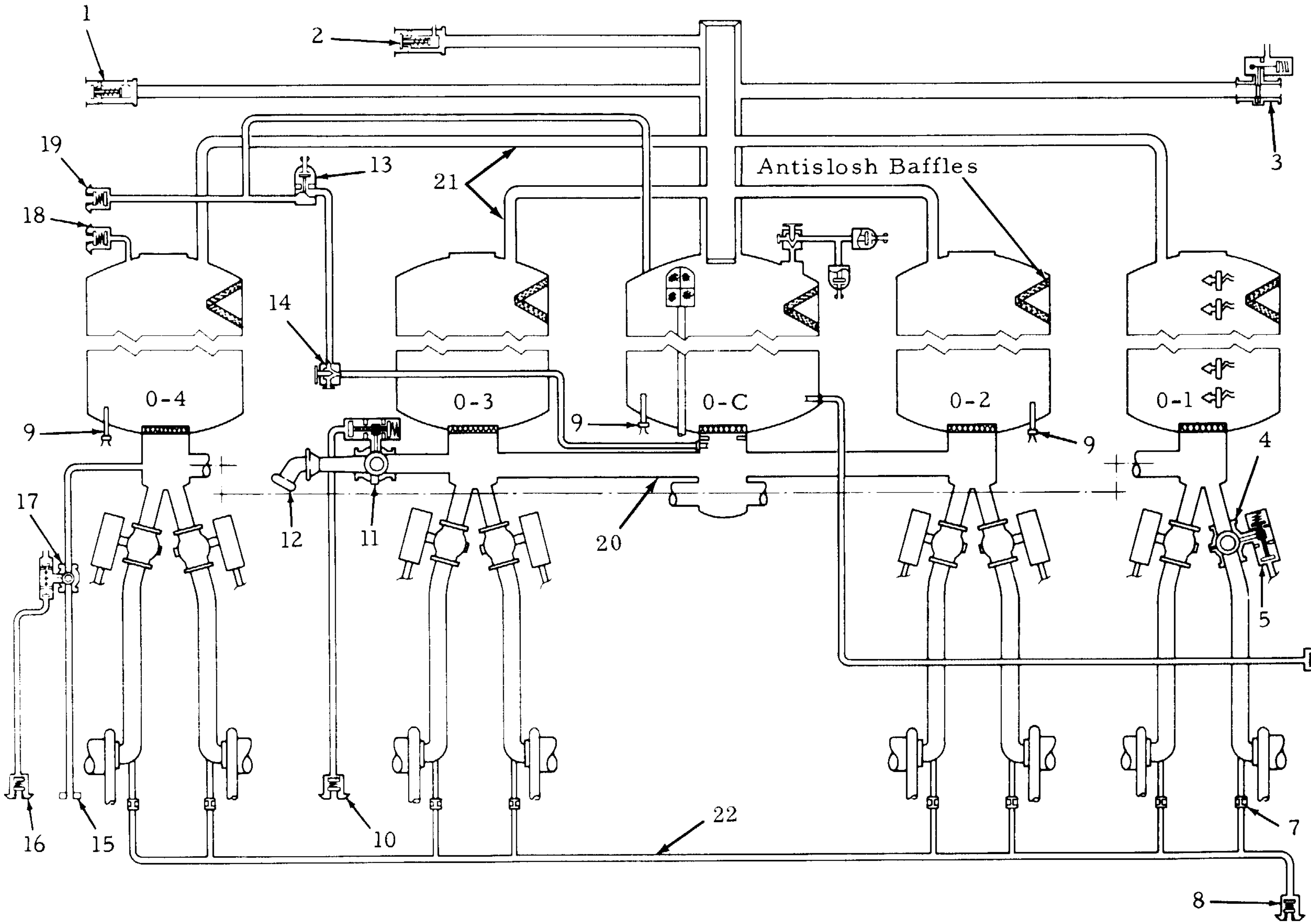

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Figure 8-8. Oxidizer Feed and Storage System, S-I

1 Relief Valve No. 1
2 Relief Valve No. 2
3 Vent Valve
4 Prevalve (8)
5 Prevalve Control Valve (8)
6 Quick-Disconnect Coupling
7 Orifice (8)
8 Quick-Disconnect Coupling
9 Level Sensor
10 Quick-Disconnect Coupling
11 LOX Fill and Drain Valve
12 LOX Quick-Disconnect Coupling Nozzle
13 LOX Step Differential Pressure Switch
14 Calibration Valve
15 LOX Replenishing Coupling
16 Quick-Disconnect
17 LOX Replenishing Valve
18 Quick-Disconnect Coupling (LOX Pressure Monitoring Line)
19 Quick-Disconnect Coupling
20 Lower Manifold
21 Upper Manifold
22 Manifold Ring Line
23 Suction Line (8)

Figure 8-8. Oxidizer Storage and Feed System, S-I (Cont'd)

8-19. OXIDIZER STORAGE AND FEED SYSTEM (FIGURE 8-8).

This system includes the LOX containers, upper and lower manifolds, and suction lines.

8-20. LOX Container. The LOX container system consists of a central 105-inch diameter container (designated O-C) surrounded by four 70-inch diameter containers (designated O-1, O-2, O-3, O-4). The four outboard LOX containers are mounted alternately between the fuel containers and each container supplies one inboard and one outboard engine. The capacity of each outboard container is 1459 cubic feet and that of the center container is 3244 cubic feet (volume at ambient temperature, not LOX temperature). Vertical rows of radial mounted baffles are installed in the containers to screen out impurities. Located near the bottom of containers O-2 and O-4 are level sensors which initiate engine cutoff when the LOX reaches a predetermined level. Near the bottom of container O-C and O-2 are four slosh measuring probes used to indicate differential pressure. An emergency LOX vent switch and a LOX pressurization switch are located in container O-C.

8-21. Upper Manifold. The upper manifold interconnects the tops of the five LOX containers to provide pressure equalization. The manifold contains a normally closed vent valve, operated by GN_2 control pressure. The valve opens during container filling and draining. The valve is also opened by the emergency vent switch assembly whenever container pressure exceeds 65 psig. The manifold also contains two pressure relief valves which mechanically open between 57 and 62 psig. For redundancy, one of these valves is also opened by a command from the emergency vent switch when the container pressure exceeds 65 psig.

8-22. Lower Manifold. The lower manifold consists of four interconnecting lines connected from the sump of the center container to the sumps of the outboard containers. This manifold maintains an approximate uniform LOX level in the containers. In the event of an engine failure, the manifold distributes most of the dead engine LOX to the other engines.

8-23. Suction Lines. Eight-inch diameter suction lines supply LOX at a nominal flow rate of 505 pounds per second. Pneumatically operated prevalves located near the container end of the suction lines are normally open, except in case of engine failure or broken line. The LOX containers are loaded from the launch complex storage containers in the following manner:

a. The normally closed vent valve and relief valves are opened.

b. The normally closed LOX fill and drain valve on container O-3 is opened.

c. Liquid oxygen pumped into container O-3 flows through the lower manifold into the other containers.

8-24. NPSH PRESSURIZATION SYSTEM.

This system provides the propellant pressurization required to maintain the net positive suction head (NPSH) at the inlet of the turbopumps.

8-25. Fuel Container Pressurization System. This system maintains a constant pressure in the fuel containers during flight. The components of the system are: two nitrogen pressure spheres, three pressurizing control valves, a pressure switch, vent valves, filters, orifices and associated ducting (Figure 8-9). The two 20-cubic foot high-pressure spheres are pressurized to 3000 psi from a ground source. As fuel is consumed during flight, the pressure switch senses the drop in the fuel container pressure and signals the pressurizing control valves to open. When the container pressure exceeds 17 psig the control valves close. Vehicle acceleration and pressure decay in the high-pressure spheres cause varying GN_2 flow rates. The flow is controlled by sequencing the three pressurizing control valves. A programmed tape removes pressure switch control of one valve each at launch +39, launch +54 and launch +70 seconds. Any over pressurization is normally controlled by the fuel vent valves which open at 19 psig. The fuel container safety valves open whenever the pressure exceeds 23 psig.

8-26. Oxidizer Container Pressurization System (Figure 8-10). Preflight pressurization of the LOX container is supplied by ground source helium through the upper manifold. After engine start, container pressure is maintained by transforming LOX to GOX in the engine heat exchangers and by ground source helium from start to first motion by opening the normally closed bypass solenoid valve. A portion of the LOX passing through the main oxidizer valve of each engine is diverted and passed through the heat exchanger mounted in the gas turbine exhaust duct. The GOX then enters the upper manifold to maintain LOX container pressure. Over pressurization is prevented by the LOX container relief and vent valves.

8-27. CONTROL PRESSURIZATION SYSTEM.

The control pressurization system, Figure 8-11, stores GN_2 at 3000 psig. It supplies pressure upon command to the pneumatically actuated valves in the propulsion system and nitrogen for LOX pump gearbox pressurization. Major components of the system are as follows:

a. Two high-pressure spheres, 1.5 cubic feet and 1.0 cubic feet.

b. A filter to keep impurities from the control system.

c. A pressure regulator to reduce container pressure from 3000 psig to 750 psig.

d. A manifold to supply 750 psig GN_2 to the various control valves.

e. A relief valve to protect manifold and valves against over pressurization in event of regulator failure.

f. A pressure switch to monitor manifold pressure for ground control.

g. Electrically actuated control valves, which upon receipt of an electrical command, open (or close) to permit passage of GN_2 to the proper pneumatic valve (i.e., relief valves, prevalves, etc.) in the propulsion system.

8-28. PROPELLANT CONDITIONING SYSTEM.

The propellant conditioning system is composed of the fuel and LOX conditioning systems described below.

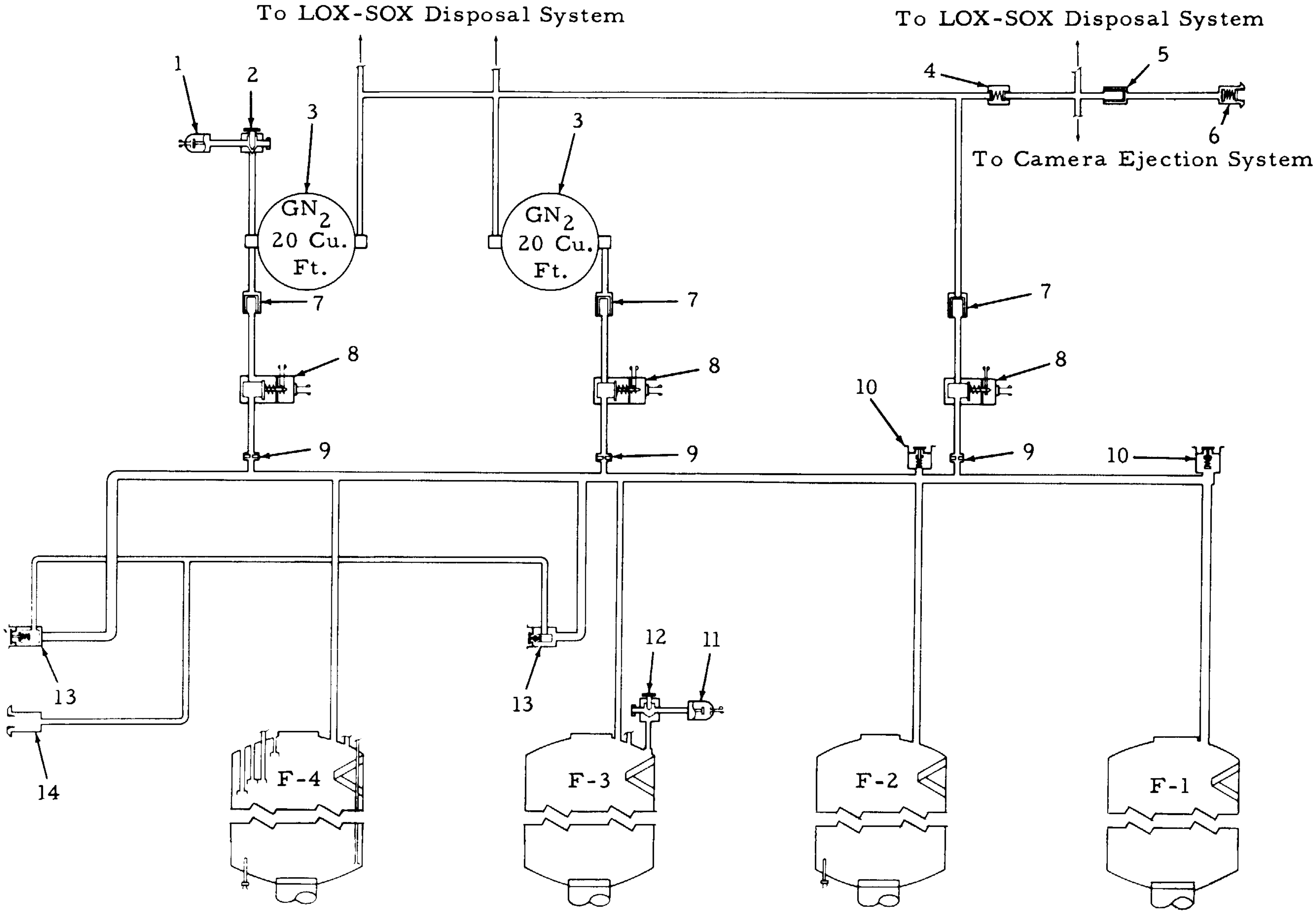

Figure 8-9. Fuel Container Pressurization System, S-I

1 Pressure OK Switch
2 Calibration Valve
3 Nitrogen Pressure Sphere (2)
4 Check Valve
5 Filter
6 Quick-Disconnect Coupling
7 Filter
8 Pressurizing Control Valve (3)
9 Orifice (3)
10 Fuel Container Safety Valve (2)
11 Pressure Switch
12 Calibration Valve
13 Fuel Vent Valve (2)
14 Quick-Disconnect Coupling

Figure 8-9. Fuel Container Pressurization System (Cont'd)

8-29. Fuel Conditioning System (Figure 8-7). This system provides a flow of ground source GN_2 to the fuel suction lines during final count down. A manifold ring line (24) distributes the GN_2. The GN_2 circulates the fuel maintaining a homogeneous temperature within each suction line.

Fuel bubbling is initiated prior to LOX loading and continues until fuel container pressurization. The GN_2 is vented through open vent valves in the fuel container. A filter prevents impurities in the gaseous nitrogen from entering the fuel container system and check valves permit the nitrogen to enter the fuel suction lines. (The check valves also prevent fuel from flowing back into the nitrogen line.)

8-30. Oxidizer Conditioning System (Figure 8-8). To create LOX circulation and maintain a suitable temperature at the pump inlets during final countdown, helium from the ground source is bubbled into the LOX suction lines. The helium, distributed by a manifold ring line (22), passes into the LOX containers and is vented through the LOX vent and relief valves.

8-31. PROPELLANT LOADING SYSTEM.

Pressure taps, located near the bottom of the containers, supply information to the ground support equipment ground computer used to monitor and control propellant loading. The pressure taps contain check valves which provide sealing after the service lines are disconnected.

8-32. PURGING SYSTEMS.

Purging of propulsion components is required at various times prior to launch and during flight. The purging systems consist of tubing, restricting orifices and check valves which permit the passage of GN_2 to the component being purged.

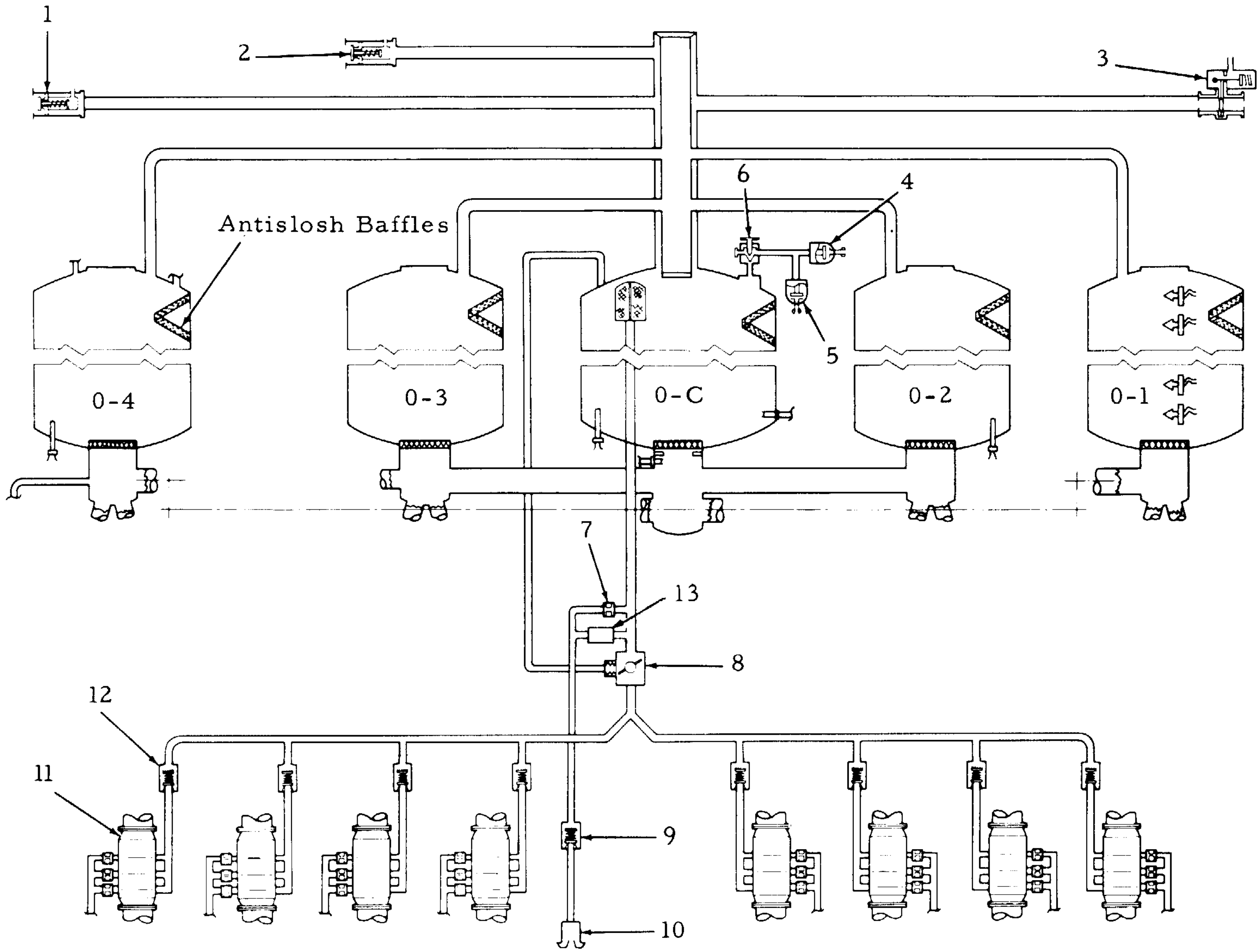

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Figure 8-10. Oxidizer Container Pressurization System, S-I

1 Relief Valve
2 Relief Valve
3 Vent Valve
4 Emergency LOX Vent Switch Assembly
5 LOX Pressurizing and Relief Switch Assembly
6 Calibration Valve
7 Ground LOX Pressurizing Orifice
8 GOX Flow Control Valve
9 Check Valve
10 Quick-Disconnect Coupling
11 Heat Exchanger (8)
12 Check Valve (8)
13 Bypass Solenoid Valve

Figure 8-10. Oxidizer Container Pressurization System (Cont'd)

8-33. Oxidizer Pump Seal Purge and Gearbox Pressurization. These two operations commence with control pressure system pressurization (occurs prior to propellant filling) and continue throughout preparation for launch, engine starting, and flight. If a launch is aborted, continuous purging is required until the turbopumps return to ambient temperature.

Gaseous nitrogen for this purging is furnished by the control pressurization spheres.

The oxidizer pump seal purge isolates LOX and lubricant leakage in the seal cavity and prevents gearbox contaminants from passing through the seal into the LOX pump. The gearbox pressurization improves the quality of lubrication and allows detection of any fuel leakage past the fuel pump seal by forcing it out the lubricant drain line. A check valve in the drain line maintains the desired pressure in the gearbox by venting the excess nitrogen out the drain line.

8-34. Oxidizer Dome Purge. This purge removes oxidizer dome contaminants. Ground source GN_2 flows to branch lines for each engine. The nitrogen passes into the LOX discharge duct, oxidizer dome, and out the thrust chamber. When aborting, this purge operation is also required at engine cutoff to prevent contamination of the LOX system by combustion by-products.

8-35. Gas Generator Oxidizer-Injector Manifold Purge. , This purge removes any fuel vapor from the LOX injector manifold and prevents by-products from the burning solid propellant in the turbine spinner from contaminating the manifold prior to arrival of oxidizer. The purge is initiated at the firing command and is terminated by pressure build up in the manifold due to oxidizer and fuel ignition. In the event of an aborted launch, this purge is required immediately following engine cutoff and again following removal of the turbine spinner. The GN_2 is received from the ground

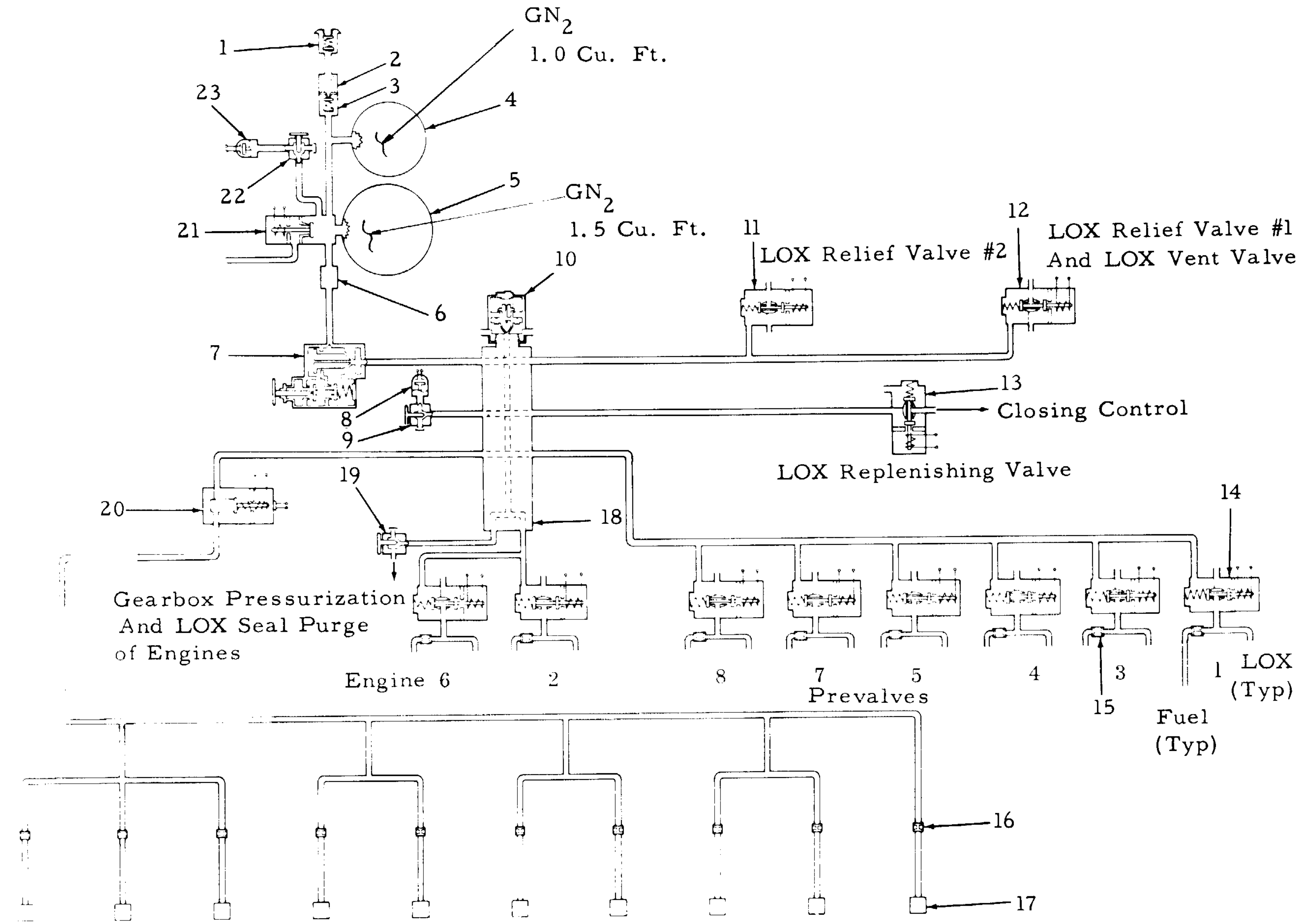

Figure 8-11. Control Pressure System, S-I

1 Quick-Disconnect Coupling
2 Filter
3 Check Valve
4 High-Pressure Sphere
5 High-Pressure Sphere
6 Control Pressure Filter
7 Pressure Regulator
8 Pressure Switch
9 Calibration Valve
10 Relief Valve
11 Control Valve
12 Control Valve
13 Control Valve
14 Control Valve (8)
15 Orifice (8)
16 Orifice (10)
17 Calorimeter (10)
18 Manifold
19 Hand Valve
20 Solenoid Valve
21 Bottle Fill and Vent Valve
22 Calibration Valve
23 High-Pressure Switch

Figure 8-11. Control Pressure System (Cont'd)

source.

8-36. Thrust Chamber Fuel Injector Manifold Purge. This purge prevents contamination of the fuel injector manifold and the fuel jacket from blow back of oxidizer rich combustion byproducts. Gaseous nitrogen from the ground source passes through the fuel injector manifold and is vented out of the thrust chamber. After engine start, pressure build-up in the manifold closes a check valve, thereby terminating the purge.

8-37. Deluge Purge System. A deluge purge system is used in event of a launch abort. Gaseous nitrogen from the ground source is ducted into the engine compartment area at a maximum flow rate of 420 pounds per minute at 3 psig pressure. The deluge purge system utilizes the onboard plumbing of the water quench system. Also included in the purge system is a prelaunch purge, utilizing preheated GN_2, that commences five minutes prior to liftoff. The flow rate is 140 pounds per minute at a pressure of 1.5 psig. During prelaunch checkout, conditioned ground source air is supplied to the engine compartment through the deluge purge system.

8-38. Water Quench System. During launch operations and static testing, a fire detection and water quench system is used in the event of a fire in the engine compartment. The water quench system, Figure 8-12, mounted in the engine compartment area, consists of four independent pipe arrangements, each protecting one inboard and one outboard engine. Four couplings located on the tail shroud engage with water supply lines from the launcher. The couplings disconnect at liftoff. The water is pumped under 100 psig pressure at a flow rate of 2000 gallons per minute per line.

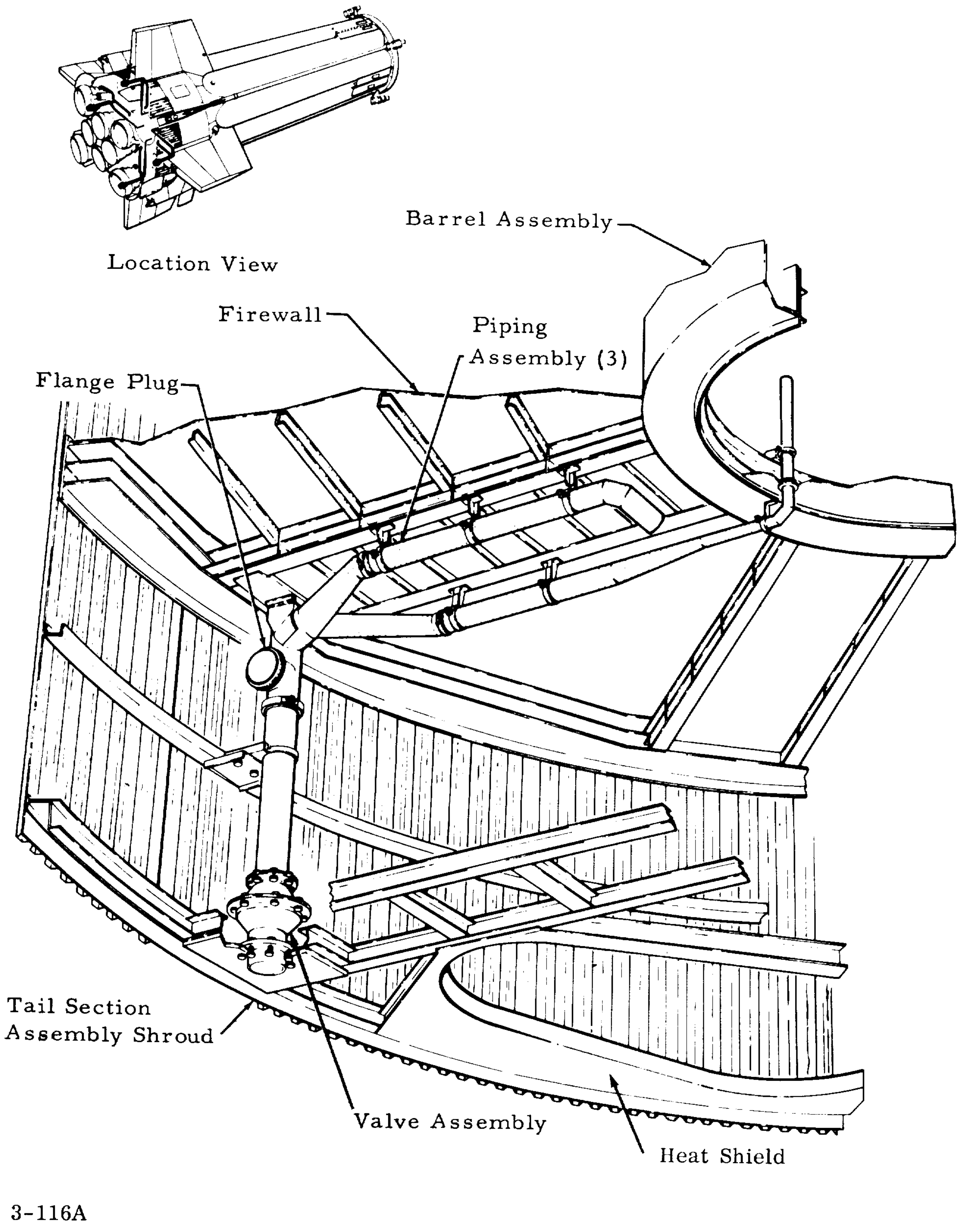

Figure 8-12. Water Quench System, S-I

8-39. S-IV STAGE PROPULSION SYSTEM

After S-I staging, the S-IV stage propulsion system injects the space vehicle into earth orbit. Functionally, the propulsion system is composed of a cluster of six RL10A-3 liquid-rocket engines and a propellant system.

8-40. ENGINE.

The S-IV stage is powered by six RL10A-3 liquid-propellant rocket engines one of which is illustrated in Figure 8-13. The engine incorporates a regeneratively cooled thrust chamber and a turbopump-fed propellant system. Heat absorbed by the fuel in cooling the thrust chamber provides power for a hydrogen turbine that drives the propellant pumps.

A nominal propellant consumption rate of 35.2 pounds per second (5:1 nominal LOX-to-fuel ratio) enables each engine to develop a nominal thrust of 15,000 pounds (200,000-foot altitude rating) at a nominal specific impulse of 427 seconds and absolute thrust chamber pressure of 300 psia. The firing duration of each engine is 470 seconds. Each engine has a dry weight of approximately 290 pounds. The RL10A-3 engine performance parameters are summarized in Table 8-4.

The engines, arranged in a circular pattern, are gimbal mounted to provide a ±4 degree thrust vector for vehicle attitude control. The engine gimbal pattern and cant angles are shown on Figure 8-2. All six engines are used for pitch and yaw control. Engines 1, 2, 3 and 4 provide roll control. The engine subassemblies are described below.

8-41. Thrust Chamber. The thrust chamber provides injection and combustion of 35.2 pounds of propellant per second and exhaust of the burned gases. A nominal thrust of 15,000 pounds (at an altitude of 200,000 feet) is achieved. The thrust chamber consists of a thrust-chamber body, a propellant injector, and a spark igniter.

Thrust-Chamber Body. The thrust-chamber body is a brazed assembly consisting of an inlet manifold, 180 short single-tapered tubes, turnaround or rear manifold, 180 full-length double-tapered tubes, exit or front manifold, and external stiffeners. The full-length tubes lead axially rearward from the hydrogen exit manifold and for the full periphery of the combustion

Table 8-4. RL10A-3 Engine Performance Parameters

Item	Parameter
Nominal engine thrust (vacuum)	15,000 pounds
Thrust stability (vacuum)	$\pm$300 pounds
Nominal specific impulse (vacuum)	427 seconds
Rated duration	470 seconds
Maximum time from ignition to 90 percent thrust	2 seconds
Maximum thrust (start transient)	17,250 pounds
Engine mixture ratio	5.0:1 $\pm$2 percent
Cutoff impulse (vacuum)	1300 pounds per second
Cutoff impulse variation (vacuum)	$\pm$250 pounds per second
LOX pump inlet nominals	48.5 psia at 163.5°R
Fuel pump inlet nominals	33 psia at 38.5°R
Rate of thrust increase (maximum)	250 pounds per millisecond
Nozzle area expansion ratio	40:1
Nominal chamber pressure	300 psia
LOX pump NPSP* (minimum required)	15 psi
Hydrogen pump NPSP (minimum required)	8 psi

*Net positive suction pressure.

chamber, throat, and forward part of the expansion chamber. The short tubes lead rearward from the hydrogen inlet manifold and interweave between the full-length tubes to form the remainder of the expansion chamber. The turnaround manifold at the aft end of the expansion chamber nozzle interconnects the short tubes to the long tubes. Brazing between the tubes serves mainly as a seal. Inlet and exit manifolds provide entrance of unheated fuel and exit of the regeneratively heated fuel, respectively. The chamber hoop loads are carried by reinforcing rings. The nominal combustion chamber pressure is 300 psia with a nominal LOX-to-fuel mixture ratio of 5-to-1 and a 35.2 pps flow rate. The thrust-chamber body, designed with a 40-to-1 expansion ratio, employs a truncated nozzle to minimize weight.

Propellant Injector. The propellant injector, located on the thrust chamber, atomizes and promotes mixing of the LH_2 and LOX to provide the correct conditions for ignition and efficient combustion. The propellant injector

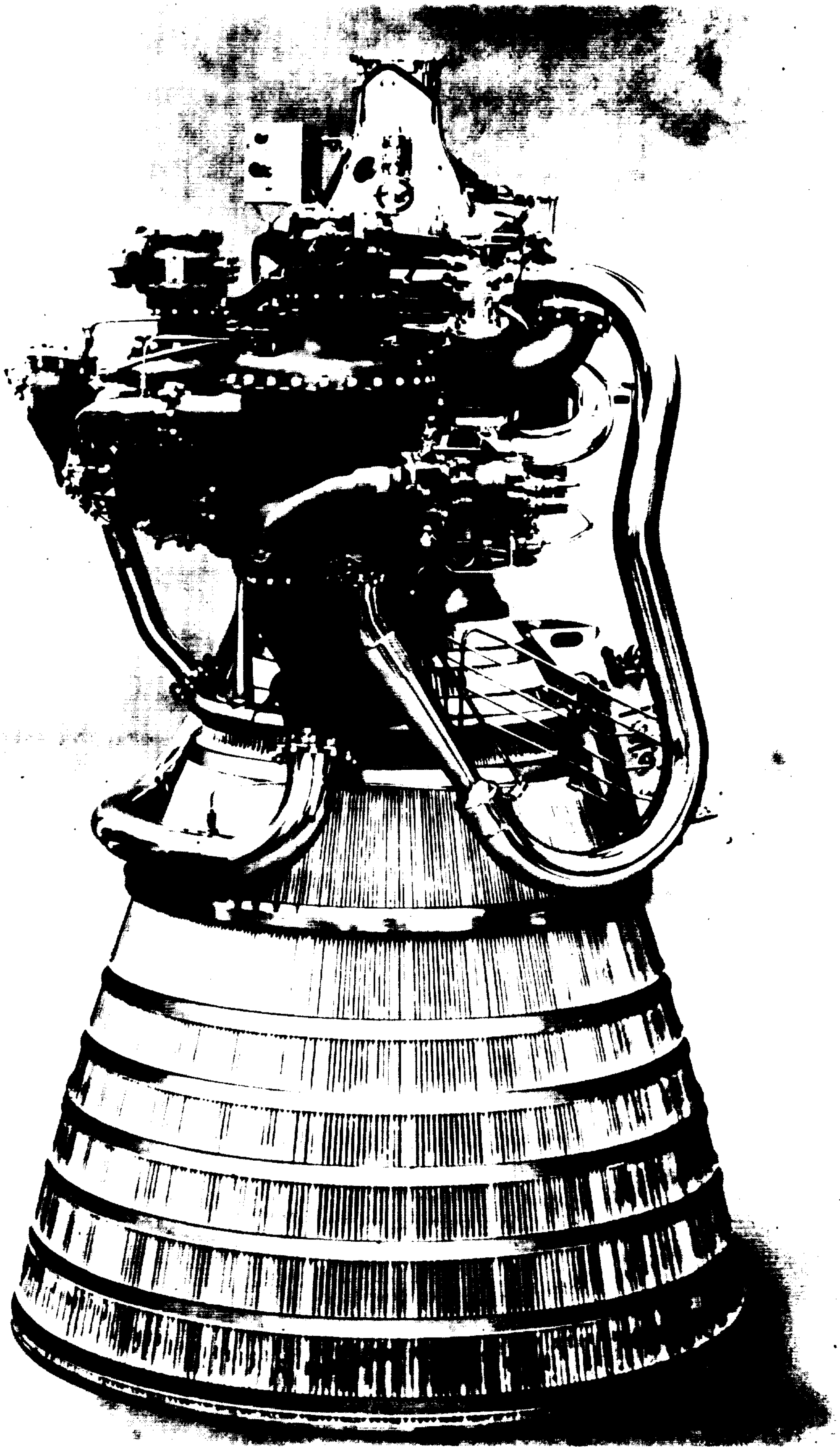

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Figure 8-13. RL10A-3 Engine

consists of 216 elements arranged in eight equally spaced concentric circles. Each element is composed of a LOX nozzle and a concentric fuel annulus. With the exception of those in the inner and outer rows, all nozzles use swirlers to produce efficient propellant mixing. The LOX nozzles are fed from a conical LOX chamber, within which is a conical fuel chamber that feeds the fuel annulus. The fuel chamber wall facing the combustion chamber is formed of porous welded steel mesh to provide transpiration cooling of the injector face. This cooling is accomplished with a fuel flow of 0.56 pounds per second or 10.4 percent of the total fuel flow.

Spark Igniter. The igniter is a recessed center electrode, air-gap type that ignites the propellants by a high-voltage capacitor discharge at a rate of 20 sparks per second. The igniter is recessed in the injector face to form a chamber that keeps the combustible mixture near the spark. Because of the spark concentration in the vacuum conditions, the proximity of the propellant mixture to the spark is critical.

8-42. Turbopump Assembly. The turbopump assembly consists of a two-stage hydrogen turbine, gear box, two-stage fuel pump and single-stage LOX pump. The turbopump is an integral unit which pumps pressurized propellants from the vehicle containers to the engine thrust chamber.

Turbine. The two-stage, partial-admission, impulse-type turbine is driven by expanding hydrogen gas flowing from the jacket and through a venturi. Both blade stages are mounted on a single rotor and are fully shrouded to minimize blade tip leakage. A rated turbine speed of 28,400 rpm develops 592 horsepower from a hydrogen flow rate of 5.56 pounds per second (approximately 95 percent of the total rated flow) working between inlet conditions of 331 degrees R. and 649 psia total pressure, and exit conditions of 312 degrees R. and 436 psia.

Gearbox. The turbopump gearbox transmits power from the main turbine driveshaft to the LOX pump shaft through a 2.5-to-1 reduction geartrain. Gearbox and oxidizer shaft cooling is provided by a 0.01 pound-per-second LH_2 coolant flow from the first-stage pump volute. The main drive shaft provides LH_2 coolant bleed flow from the second-stage fuel pump inlet to the support bearings at the turbine drive end. Gearbox pressurization is maintained at 18 to 25 psi above ambient, and excess gas is vented into a cooldown vent manifold.

Fuel Pump. The fuel pump consists of two stages mounted back-to-back to minimize axial thrust. A common shaft drives the fuel pumps directly from the turbine. The pump has a constant velocity collecting volute for equal circumferential pressure distribution, and a straight-tangential nozzle diffuser for velocity-head recover. A power requirement of 509 horsepower is necessary to drive the fuel pump at a rated operating speed of 28,400 rpm and a flow rate of 602 gpm (5.85 pounds per second).

The first-stage fuel pump is preceded by a three-bladed axial flow inducer which operates at the same speed as the aluminum-alloy impeller. A 50-degree exit angle backswept blade design is incorporated into the back-shrouded impeller to provide a suitable low-flow allowable stress characteristic.

The second-stage fuel pump impeller is also of aluminum alloy and incorporates a back-shrouded radial blade design with a 90-degree exit angle.

LOX Pump. The LOX pump is mounted on the turbopump gearbox beside the fuel pump and is driven through the 2.5-to-1 reduction geartrain located within the gearbox. A three-bladed axial flow fully-shrouded stainless steel inducer increases impeller inlet pressure above the vehicle supply pressure to prevent impeller cavitation. The centrifugal pump has a single-stage fully shrouded stainless steel impeller.

A constant velocity collecting volute designed for equal circumferential pressure distribution and a straight tangential discharge nozzle diffuser for velocity-head recovery are employed within the oxidizer pump housing. An accessory drive pad, located on the aft end of the oxidizer pump shaft, provides a mounting for the main hydraulic pump.

The oxidizer pump operates at a nominal speed of 11,350 rpm with a nominal flow rate of 1847 gpm (29.3 pounds per second) when operating at inlet and discharge pressures of 48.5 psia and 464 psia, respectively. A pump efficiency of 59.7 percent at the rated conditions results in a power requirement of 78.2 horsepower.

8-43. Propellant Inlet Shutoff Valves. The fuel pump and oxidizer pump inlet shutoff valves control the flow of the propellant form the vehicle containers to the engine pumps. Both valves are similar and are normally closed, two-position rotating ball-type valves. The valves are opened by a 450 ± 50 psia control helium actuator piston and are spring closed. The fuel pump inlet shutoff valve moves from closed to fully open in approximately 30 milliseconds and moves from fully open to closed in approximately 389 milliseconds. The oxidizer pump inlet shutoff valve moves from closed to fully open in approximately 17 milliseconds, and moves from fully open to closed in approximately 158 milliseconds.

8-44. Solenoid Valves. The prestart and start solenoid valves control the flow of helium pressure from the stage storage tank to the engine system propellant-control valves. The prestart and start solenoid valves are identical in design, operation and construction. The solenoids when energized operate a two-way poppet. The poppet, in turn, controls the flow (450 ± 50 psia helium pressure) to the propellant control valves. In this manner the helium actuator flow is controlled.

8-45. Prestart Solenoid Valves. The prestart solenoid valves control the helium pressure which opens the fuel and oxidizer pump inlet shutoff valves. The prestart solenoid valves remain in the open position as long as the solenoid remains energized. The prestart solenoid valves are closed by a spring at engine shutdown.

8-46. Start Solenoid Valve. The start solenoid valve controls the helium pressure which initiates the opening of the main fuel pump inlet shutoff valve, and the closing of the interstage and downstream cooldown and bleed valves.

The start solenoid valve, is opened by a start signal, which occurs 41.6 seconds after the prestart signal. The solenoid remains energized throughout engine operation and holds the start valve in the open position. The valve is closed by a spring when the engine is cut off.

8-47. Fuel Pump Cooldown, Bleed, and Pressure Relief Valves. The cooldown and bleed valves provide overboard venting of fuel to cooldown both fuel pump stages during engine prestart. The valve also allows fuel bleed during pump acceleration to provide transient stability and pressure relief when the engine is shutdown. The valves are pressure-boosted, three-position, sleeve-type valves, spring-loaded open to vent fuel overboard during non-running and cooldown periods. Helium pressure from the start solenoid valve partially closes the cooldown and bleed valves. The cooldown and bleed valves are designed so that a partial closing operates a sleeve valve within the cooldown and bleed valve. The cooldown and bleed valves are opened in approximately 15 milliseconds by spring compression boosted by trapped helium pressure and fuel-discharge pressure (routed to an opening booster piston during engine shutdown). This procedure alleviates high fuel system pressure.

8-48. Thrust Control Valve. The thrust control valve, a servo-operated, variable-position valve, controls engine thrust by regulating the amount of fuel bypassing the turbine as a function of thrust chamber combustion pressure (300 psia). This, in turn, controls the speed of the turbopump. The thrust control valve is located in a bypass line between the turbine inlet and exit. Thrust chamber combustion pressure operates the motor bellows which is referenced to a spring and to a vacuum reference bellows. The motor bellows actuates a carriage which, in turn, operates a servo-lever regulating the vent area of the servo-pressure supply port. The supply pressure of GH_2 from the thrust chamber heat exchanger discharge line is approximately 672 psia. Bypassed hydrogen is returned to the turbine exhaust line. The pressure difference between servo pressure and combustion pressure exerts a force on the resisting spring to produce the corrective motion in the turbine bypass flow-regulating sleeve valve. The thrust control valve is designed so that motion of the bypass sleeve valve is transmitted to the valve carriage by a low-rate feedback spring, which begins correcting the servo pressure before a new chamber pressure is achieved.

8-49. Main Fuel Shutoff Valve. The normally closed main fuel shutoff valve controls the flow of fuel to the thrust chamber. The bullet shaped valve (tapered inlet and exit cones) is located within the turbine discharge lines just upstream of the thrust chamber fuel-manifold. During the cooldown period, the shutoff valve prevents the control pressure from working against a shutoff spring. It controls the fuel flow by opening or closing an annular valve housing area about the exit cone. Turbine discharge pressure keeps the shutoff valve open during rated engine operation. A delay in main fuel shutoff valve closing occurs during engine shut-down until after the bleed valves are opened. The delay allows fuel to flow out through the thrust chamber heat exchanger and prevents fuel pump housing rupture that would result from increased pressure of overheated trapped fuel.

8-50. Oxidizer Flow Control Valve. The oxidizer flow control valve is located in the LOX pump discharge line upstream of the igniter oxidizer supply control valve; it performs the following functions:

a. Maintains a constant LOX flow during engine cooldown.

b. Controls the oxidizer-to-fuel ratio during the start period within the rich and lean blowout limits for proper ignition.

c. Controls the consumption of LOX to minimize residual propellants in the vehicle containers at burnout.

d. Permits ground trim of the mixture ratio.

Three orifices in the control valve, one of which has a variable area, are operated by the resultant force of reference spring pressure and inlet LOX pressure. Both provide uniform LOX cooldown flow of approximately 2.2 pounds per second for the full range of inlet conditions. The oxidizer flow control valve contains a spring-loaded inlet piston which senses LOX pump inlet pressure on its back face, and LOX pump discharge pressure on its upstream face. This inlet piston controls the size of an annular LOX inlet orifice (closed during the initial portion of the start cycle) which opens to allow a nominal flow of 29.3 pounds per second when the piston pressure differential across the oxidizer control valve reaches approximately 109.3 $\pm$ 16 psi. The oxidizer flow control valve has provisions to mount a drive motor that is controlled by the vehicle propellant-utilization system. The motor controls the position of a variable-area piston within a discharge orifice. In turn, the piston controls the consumption of LOX to minimize residual propellants on board at burnout. A vehicle supplied nitrogen atmosphere purge prevents ice from forming on the propellant utilization adjustment assembly during cooldown. Various adjustment hardware provides for ground trim of nominal mixture ratio setting.

8-51. Igniter Oxidizer Supply Control Valve. The igniter oxidizer supply control valve regulates gaseous oxygen flow to the spark igniter to insure ignition within the thrust chamber. An igniter oxidizer valve poppet is opened by LOX pressure from the oxidizer pump inlet when the prestart valve is actuated. The poppet controls the LOX, which is bled from the supply lines to the injector entering the combustion chamber at the spark igniter tip during engine starting. The poppet is closed by LOX pump discharge pressure.

8-52. Fuel Container Pressurizing Valve. The LH_2 container is pressurized by means of a fuel bleed from the engine injector to the container through a sealed pressurizing valve. The pressurizing valve is a single position poppet valve referenced to gear box pressure sensing fuel injector manifold pressure. It provides a seal between the fuel container and the combustion chamber when the engine is not operating.

8-53. Engine Operation. Two independent prestart, or cooling sequences, one for the LH_2 system and the other for the LOX system, are initiated by electrical signals from the vehicle. The first energizes the fuel prestart solenoid valve which permits control helium (445 ± 25 psia) to pressurize the actuator of the fuel pump inlet shutoff valve opening the shutoff valve. Liquid hydrogen flows through the two pump stages and is discharged overboard through the cooldown valves. A signal, approximately 32 seconds later energizes the oxidizer prestart solenoid valve which admits control helium (455 ± 25 psi) to the actuator of the oxidizer pump inlet shutoff valve. Liquid oxygen then flows through the LOX pump discharging through the propellant injector. At the end of the prestart sequence, the propellant pumps have cooled down to a temperature which will prevent cavitation during pump acceleration.

8-54. Prestart Sequence. The six engines must pass through a prestart cooldown sequence because of the low temperature characteristics of their propellants. The engines are started in unison a minimum of 41.6 seconds after the pre-start signal has been initiated.

The engine schematic is illustrated in Figure 8-14. The engine operating sequence, illustrated in Figure 8-15, is described below.

8-55. Start Sequence. An electrical signal from the vehicle initiates the start sequence by energizing the start solenoid valve. An interval of at least 20 seconds must exist between the first prestart signal and the start signal. The start signal also energizes the ignition system. Pressurized helium flowing through the

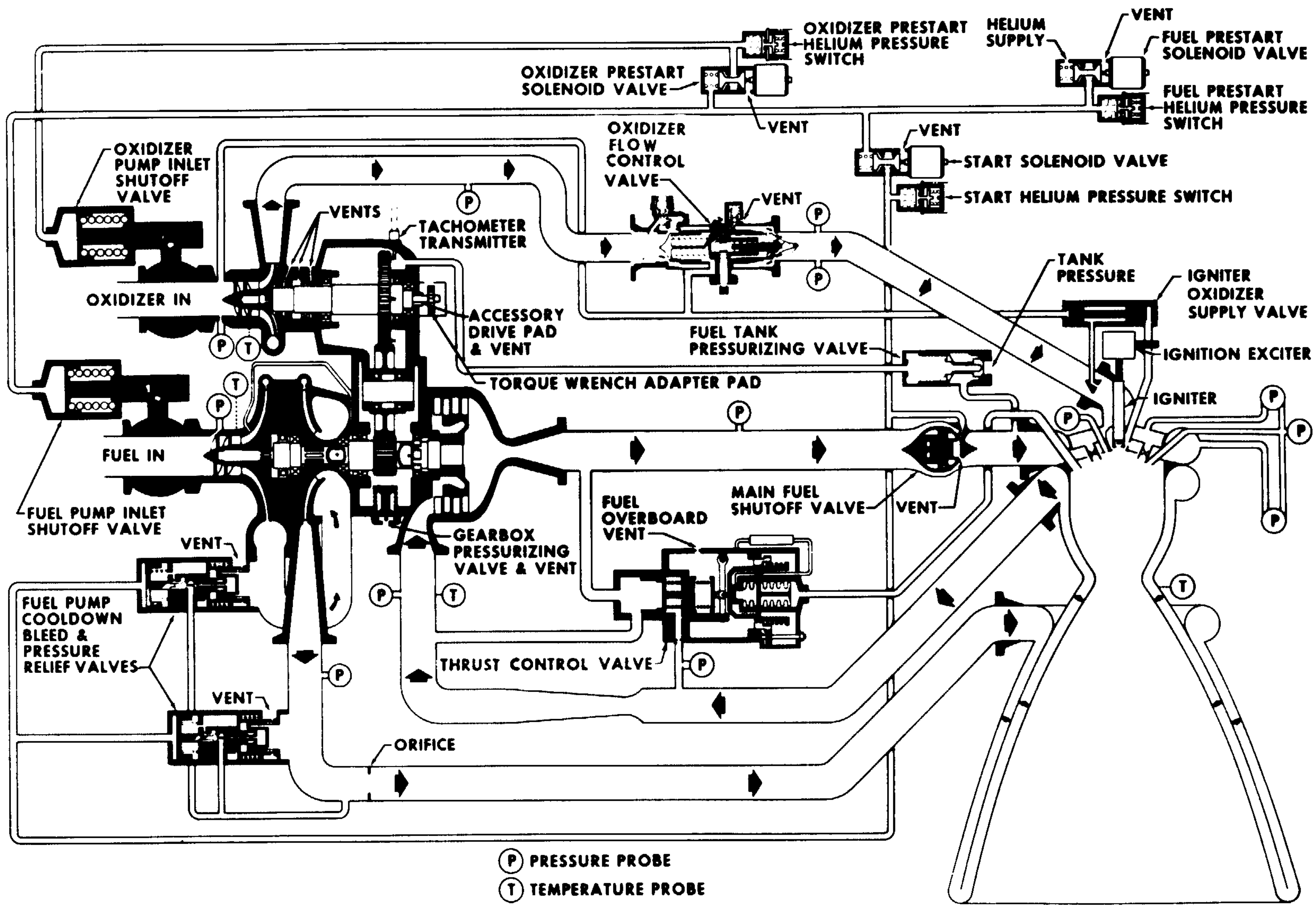

Figure 8-14. RL10A-3 Engine Schematic

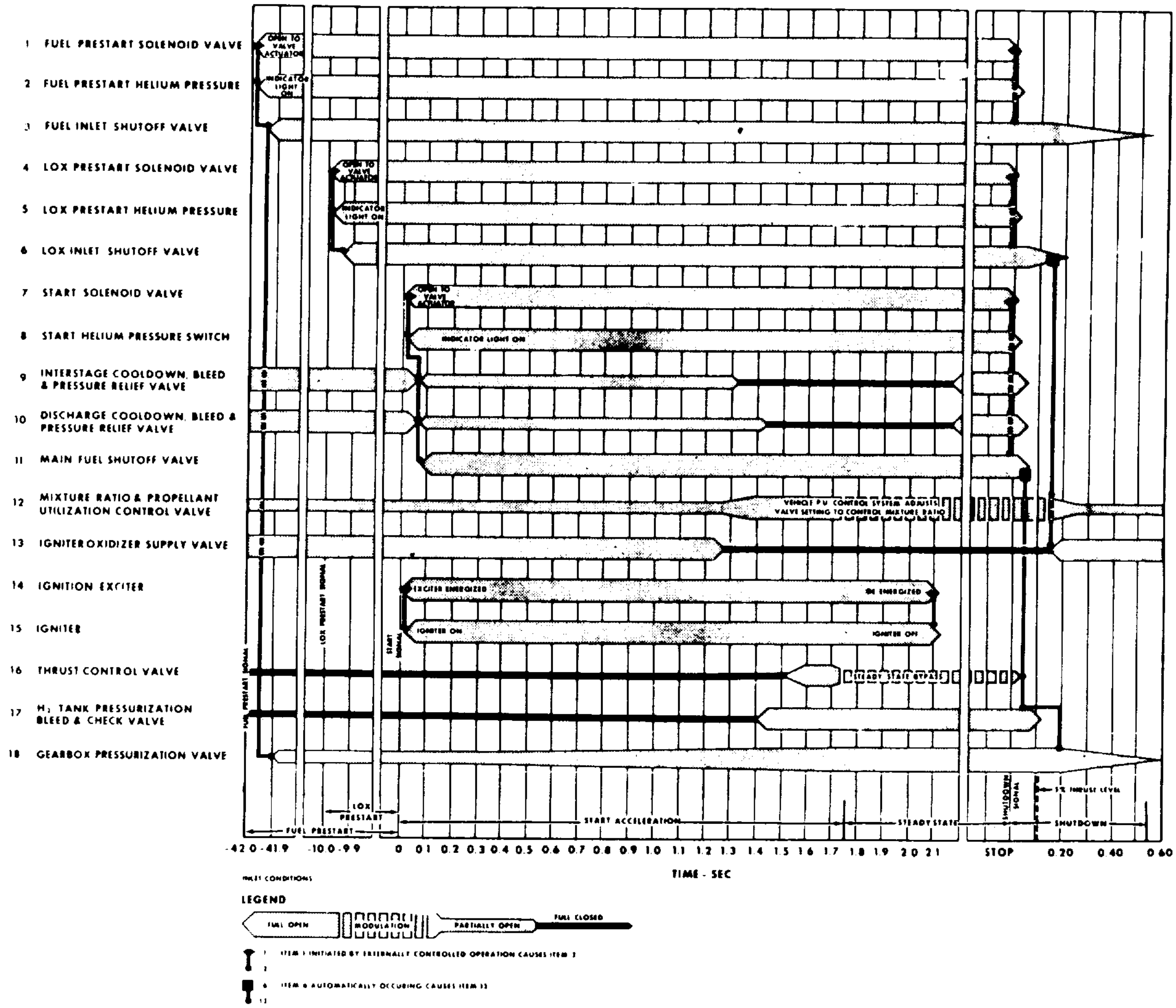

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Figure 8-15. RL10A-3 Engine Operating Sequence

energized start solenoid valve opens the main fuel pump inlet shutoff valve and partially closes the fuel pump cooldown and bleed valve. This permits fuel from the pump discharge to flow through the thrust chamber tubes, absorbed heat providing the energy for the turbine to overcome the static friction of the turbo-pump assembly and start turbopump rotation. The partially closed fuel pump cool-down bleed and pressure relief valve acts as a bleed during acceleration to provide fuel pump transient stability. In the start position, the oxidizer flow control valve controls LOX flow as a function of inlet pressure. When a combustible mixture is developed in the thrust chamber, the propellants are ignited by the spark igniters and the engine accelerates to rated thrust. The fuel pump cooldown bleed and pressure relief valve closes as the fuel pump discharge pressure increases. The oxidizer flow control valve opens as a function of LOX pump pressure rise to provide the proper mixture ratio for engine acceleration.

8-56. Steady-State Operation. During steady-state operation the metering orifice area in the oxidizer flow control valve is varied for propellant utilization control. Thrust is controlled by the thrust control valve which regulates turbine bypass flow as a function of chamber pressure.

8-57. Shutdown Sequence. Termination of the electrical signal from the stage sequencer initiates shutdown. The solenoids return to their normally closed position shutting off the helium supply and venting helium from all valve actuators. The fuel pump cooldown, bleed and pressure relief valves open, draining fuel from the system to prevent a pressure buildup caused by closing the main fuel pump inlet shutoff valve. This stops the flow through the turbine thus stopping the pump rotation causing the system to come to rest. The oxidizer pump inlet shutoff valve closes, stopping the LOX flow into the engine. The remaining oxidizer in the engine vents through the injector into the thrust chamber. The fuel pump inlet shutoff valve closes preventing fuel from entering the system.

8-58. Cooldown and Leakage Venting. A collection system is employed whereby combustible waste fuel is directed to a vent manifold which discharges the waste fuel overboard. Fuel cooldown and bleed flow from the cooldown and bleed valve is directed through a vent line which also collects discharge from the gearbox check valve vent, a vent on the fuel side of the LOX pump seal, the gearbox accessory pad seal vent and the main fuel shutoff valve vent. The thrust control valve is not vented into the vent collector manifold because the performance of the thrust control valve would be affected by the manifold back pressure. The thrust control is vented to the vehicle interface connection at the collector manifold.

8-59. Propellant Utilization System. Capacitor-type sensors located in each propellant container supply information to the propellant utilization system which by varying the LOX flow rate causes simultaneous depletion of both propellants.

8-60. PROPELLANT SYSTEM.

The propellant system consists of the following systems:

a. Fuel Storage and Feed

b. Oxidizer Storage and Feed

c. NPSH Pressurization

d. Propellant Sensing

e. Control Pressurization

f. Chill-Down Purge

The fuel (LH_2) and oxidizer (LOX) are delivered by separate feed systems, Figure 8-16. The propellant feed system furnishes LH_2 and LOX under pressure to the six engines during operation, but may be isolated from any or all of the engines in an emergency. The propellant container capacity is approximately 100,000 pounds of usable propellants.

8-61. FUEL STORAGE AND FEED SYSTEM.

The LH_2 container has an approximate volume of 4274 cubic feet, including 4-percent ullage. Helium spheres, installed within the LH_2 container store 3000 psig cold helium for the LOX container pressurization. A separate LH_2 suction line is installed from the fuel container to each of the engines. Liquid hydrogen consumption is initiated by a signal which opens the fuel inlet shutoff valve. The signal occurs during the S-I/S-IV stage separation sequence (initiation of S-IV stage cooldown). (The LH_2 flows from the LH_2 container to the inlet side of the turbopump through and LH_2 suction line and LH_2 inlet shutoff valve.) The mass flow rate of LH_2 to the engine is 585 pounds per second at a nominal LOX-to-LH_2 mixture ratio of 5:1.

8-62. OXIDIZER STORAGE AND FEED SYSTEM.

The LOX container has an approximate volume of 1262 cubic feet, including 4-percent ullage. The six engines are supplied LOX from separate suction lines equally spaced around the bottom of the LOX container. Each suction line includes flexible bellows which allow sufficient freedom for engine gimballing.

Liquid oxygen consumption is initiated by a signal which opens the LOX inlet shutoff valve. The signal occurs during S-I/S-IV separation. LOX flows from the LOX container to the inlet side of the turbopump through a LOX suction line and the LOX inlet shutoff valve. The mass flow rate of LOX to each engine is 29.3 pounds per

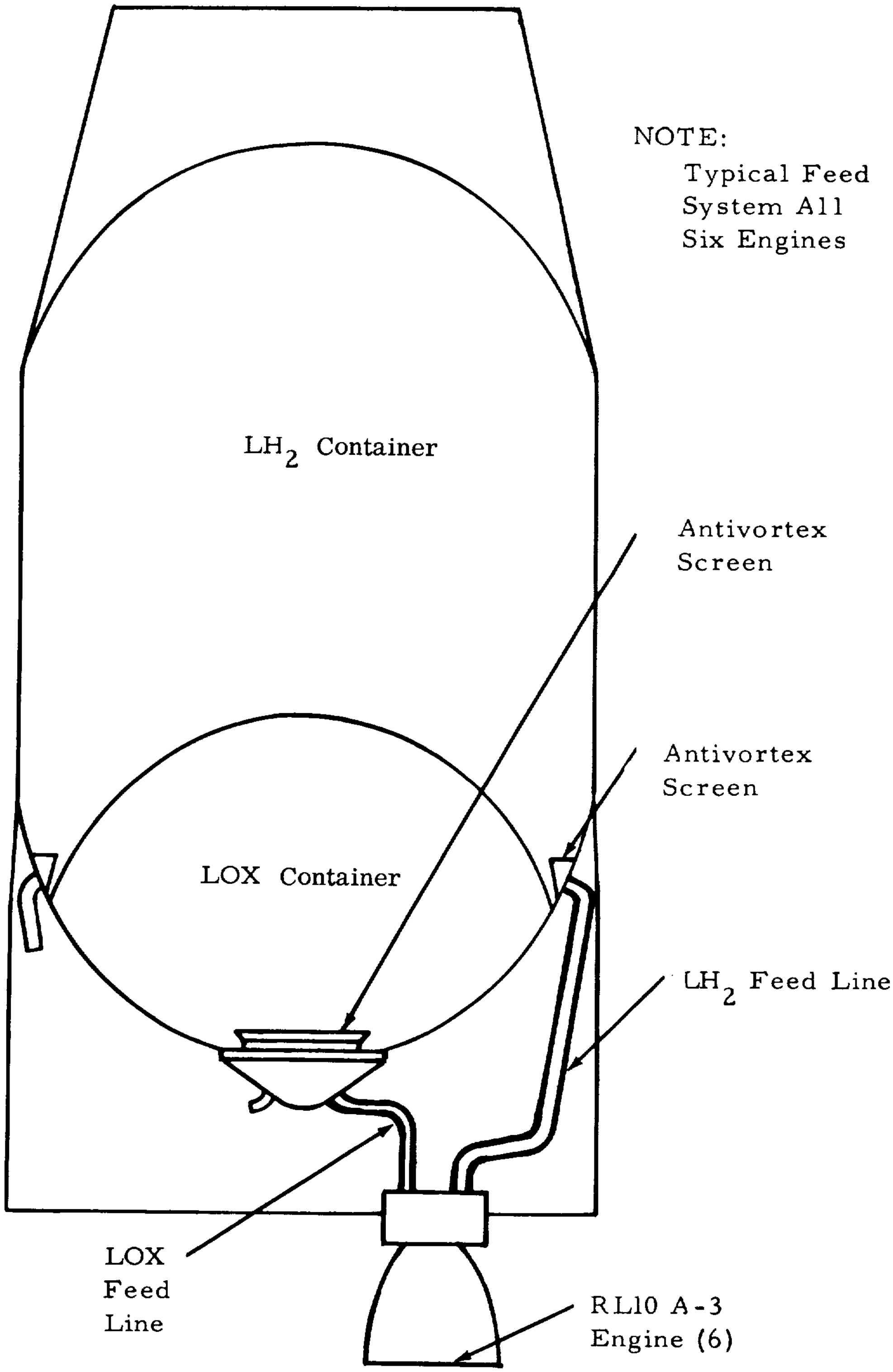

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Figure 8-16. Propellant System, S-IV

second at a nominal LOX-to-LH_2 mixture ratio of 5:1.

8-63. NPSH PRESSURIZATION SYSTEM.

This system provides the propellant pressurization which maintains a net positive suction head (NPSH) at the inlet of the LOX and LH_2 pumps.

8-64. Fuel Container Pressurization System. The LH_2 container is blanket pressurized with 2.0 ± 1.5 psig GH_2 from a service line prior to filling and replenishing. After the container has been filled and replenished, but prior to launch, the LH_2 container is pre-pressurized to 36.0 ± 1.5 psig with cold helium from a service line. The pressurization is maintained by ambient helium (contained in a sphere mounted on the vehicle thrust structure) during operation of the S-I stage to a value of 31.0 ± 1.0 psig. After the S-IV stage engines are ignited, pressurization is maintained with GH_2 (31 psia) from engine bleed lines.

8-65. Oxidizer Container Pressurization System. The LOX container is purged and blanket pressurized with 4.0 ± 0.5 psig GN_2 from a service line prior to filling and replenishing. After the container has been filled and replenished, it is pre-pressurized to a value of 46.5 ± 1.5 psig by a cold helium bottle fill service line. If during the S-I boost phase the LOX container ullage pressure drops below 45.5 ± 0.5 psia, the container ullage pressure switch opens the primary cold helium valve and permits cold helium from the 3.5 cubic foot spheres located in the LH_2 container to maintain the LOX container pressure at 46.5 ± 1.5 psia. The cold helium stage-stored pressure prior to liftoff is 3000 psig. After S-IV stage ignition the helium is routed to the LOX container through a helium heater that burns LH_2 and LOX. The combustion gases from the helium heater are exhausted through the vehicle heat shield.

8-66. PROPELLANT SENSING SYSTEM (PROPELLANT LOADING)

The capacitor type sensors which supply information to the propellant utilization system also supply information to the ground support equipment. This information is used to monitor and control propellant loading.

8-67. CONTROL PRESSURIZATION SYSTEM.

A high-pressure helium sphere, located in the engine section, provides ambient temperature GHe for engine requirements and vehicle pneumatic control. The sphere contains 1.5 cubic feet of helium at 3000 psi. It is pressurized from a service line which remains connected until vehicle first motion.

8-68. CHILL-DOWN PURGE SYSTEM.

The chill-down purge system removes contaminants from the chilldown system prior to the introduction of LH_2 and LOX. The purge system uses helium stored at 3000 psia in three spheres mounted on the S-I stage spider beam. The helium is routed through the chill-down system prior to S-IV engine chill-down.

CHAPTER 2

SECTION IX
MECHANICAL SYSTEMS

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SECTION IX.
MECHANICAL SYSTEMS

9-1. GENERAL.

The mechanical systems of the Saturn I launch vehicle include environmental control, engine gimballing, separation, ordnance, and platform gas-bearing supply.

9-2. ENVIRONMENTAL CONTROL SYSTEM.

The Saturn I environmental control system controls the environment in certain compartments of the launch vehicle and Apollo payload. The system protects electrical and mechanical equipment from thermal extremes, controls humidity, and provides an inert atmosphere for the vehicle compartments. Operation of the system is controlled by ground based equipment.

The environmental control system allows the use of "off the shelf" electrical components on the launch vehicle which otherwise could not be used without elaborate provisions for heat dissipation.

Active environmental conditioning begins during prelaunch upon the application of electrical power to the launch vehicle and ends when the vehicle umbilicals are disconnected at liftoff. During the remainder of the mission thermal inertia and component insulation maintain temperatures within the design ranges.

9-3. OPERATION.

The various operations of the environmental control system are controlled from the launch control center and the automatic ground control station. The ground equipment used to control and supply the conditioning mediums is located within six different facilities (Figure 9-1):

a. Converter compressor facility (GN_2)

b. Remote fresh-air intake facility (air)

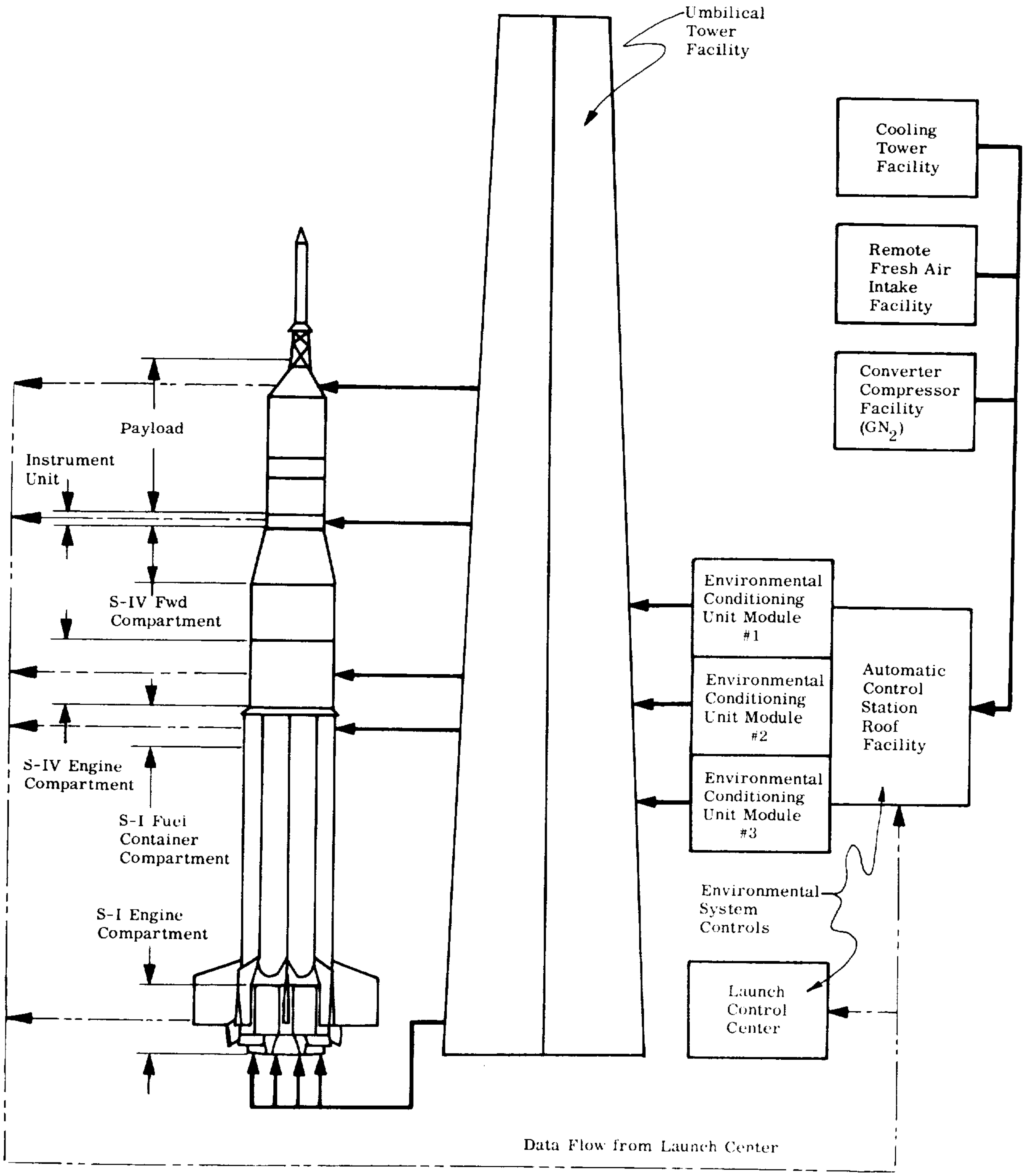

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Figure 9-1. Environmental Control System, Saturn I

c. Cooling tower facility
d. Automatic ground control station roof facility
e. Umbilical tower facility
f. Launch control center

The ground equipment conditions the following vehicle and payload areas:

a. S-I stage engine compartment
b. S-I stage fuel container instrument compartments
c. S-IV stage engine compartment
d. S-IV stage forward compartment
e. Instrument unit
f. Apollo payload

Three environmental conditioning modules, located in the umbilical tower facility, provide filtered and conditioned air or GN_2, or both simultaneously, through the vehicle umbilical to the vehicle. The maximum flow rate of conditioned gas from each module is 300 pounds per minute (maximum) at a pressure of 48 inches of water. The gas temperature can be controlled from 35 to 250 degrees F.

Nozzles or orifices, which are part of the vehicle plumbing, provide each of the compartments being conditioned with constant gas-flow rates. Strategically located temperature probes supply area temperature information to the ground control stations for temperature control.

At the start of the launch vehicle electrical equipment checkout during prelaunch, the environmental control system supplies cool dry air to the two fuel container instrument compartments of the S-I stage, the S-IV stage forward compartment, and the instrument unit. The cool air maintains the electrical components located in these compartments within design temperature limits. The compartments receive cool air until 15 minutes before the start of LH_2 loading in the S-IV stage.

Prior to loading LOX in the S-IV stage, warm air is delivered to the S-IV stage engine compartment to prevent supercooling of equipment located in this area. For the same reason, the S-I stage engine compartment receives heated air prior to loading LOX in the S-I stage. Air is supplied to the two engine compartments until 15 minutes before LH_2 loading begins in the S-IV stage.

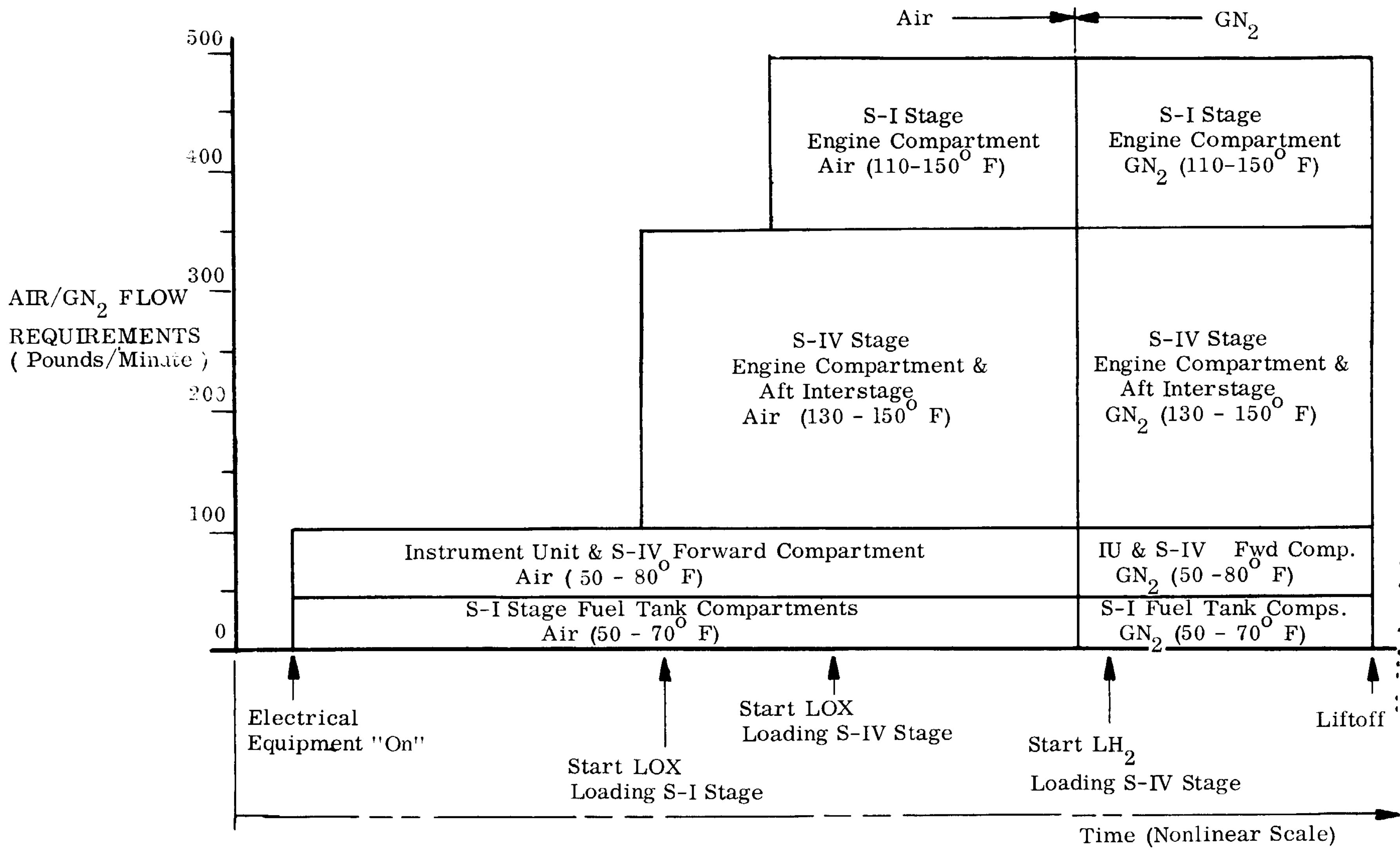
The environmental control system medium is changed from air to GN_2 for all compartments a minimum of 15 minutes before the start of LH_2 loading in the S-IV stage. This prevents possible fire or explosion by maintaining the O_2 content below the level which will support combustion and by preventing any significant accumulation of GH_2. The flow rates and temperatures remain unchanged, Figure 9-2.

The Apollo payload is also conditioned by the environmental control system. The medium, flow rate, temperature, and delivery schedules are determined by MSC.

9-4. S-I STAGE IMPLEMENTATION.

The S-I stage environmental control system maintains a predetermined temperature and humidity level in the engine compartment and in the two instrument compartments located in the forward end of the fuel containers F-1 and F-2. The engine compartment, located between the heat shield and the firewall, and the area under the center LOX container are serviced through the same piping that is used for the water quench system. The piping consists of four independent assemblies each of which is connected through quick-disconnect couplings to a separate line from the environmental control system ground facilities. The vehicle plumbing is disconnected from the ground lines at liftoff. One of the four pipe assemblies is shown in Figure 8-12. Warm air (110 to 150 degrees F) at a flow rate of approximately 147 pounds per minute and at 20 to 30 inches water pressure, is delivered to the engine compartment before LOX is loaded in the S-I stage. A minimum of 15 minutes prior to the start of LH_2 loading in the S-IV stage, the air is replaced with GN_2 at the same temperature and flow rate. The temperature within the compartment is monitored by two probes which supply temperature data to the environmental system ground control stations.

The two instrument compartments located in the forward portion of fuel containers F-1 and F-2 are serviced from the ground system through a common umbilical duct connected to a manifold. The manifold distributes the conditioning medium to each compartment. During prelaunch checkout, cool, dry air (50 to 70 degrees F) at a flow rate of 45 pounds per minute and a pressure of 12 inches of water is supplied as soon as compartment electrical equipment operation begins. Gaseous nitrogen at the same temperature and flow rate replaces the air 15 minutes before loading LH_2 in the S-IV stage. The temperature of the inlet air or GN_2, sensed by a thermistor probe, is monitored by the ground system.



3-228

Figure 9-2. Environmental Control System, Air/GN_2 Requirements

9-5. S-IV STAGE IMPLEMENTATION.

Electrical and mechanical components located in the engine compartment and in the forward compartment of the S-IV stage are also protected from environmental extremes by the environmental control system. The conditioning medium is supplied to the engine compartment distribution manifold through an umbilical duct connection at the vehicle skin. The manifold is formed by enclosing the area between the LH_2 container wall and the skirt structure with a flexible membrane located forward of the separation plane. Reaching the aft interstage area through orifices spaced around the circumference of the membrane, the conditioning medium is directed into the area between the engine thrust structure and the propellant container through nozzles fed by ducts connected to the manifold, Figure 9-3. Warm, dry air (130 to 150 degrees F) at a flow rate of 204 to 240 pounds per minute and a pressure of 15 inches of water is supplied to the engine compartment prior to loading LOX on the S-IV stage. A minimum of 15 minutes before LH_2 loading, the air is replaced by GN_2 at the same temperature and flow rate. The temperature in the compartment is measured by a thermistor probe and monitored by the environmental system ground control stations.

The forward compartment of the S-IV stage receives conditoned air or GN_2 that is exhausted from the instrument unit compartment. The air or GN_2 is vented from the vehicle through two vent holes located in the instrument unit skin. One of the vent holes contains a thermistor probe that senses the exhaust temperature. Dry air (73 to 80 degrees F) at a flow rate of 59 pounds per minute is supplied to the compartment when the instrument unit electrical equipment checkout begins. The air is replaced with GN_2 at the same temperature and flow rate a minimum of 15 minutes before LH_2 loading.

9-6. INSTRUMENT UNIT IMPLEMENTATION.

The instrument unit electrical equipment is prevented from overheating by the vehicle environmental control system. During prelaunch checkout cool, dry air (60 to 80 degrees F) at a flow rate of 59 pounds per minute and a pressure of 29 inches of water flows to the instrument unit when the electrical equipment in the compartment is first energized. Gaseous nitrogen at the same temperature and flow rate replaces the air approximately 15 minutes before LH_2 loading. The conditioning medium is delivered from the ground system through an umbilical duct.

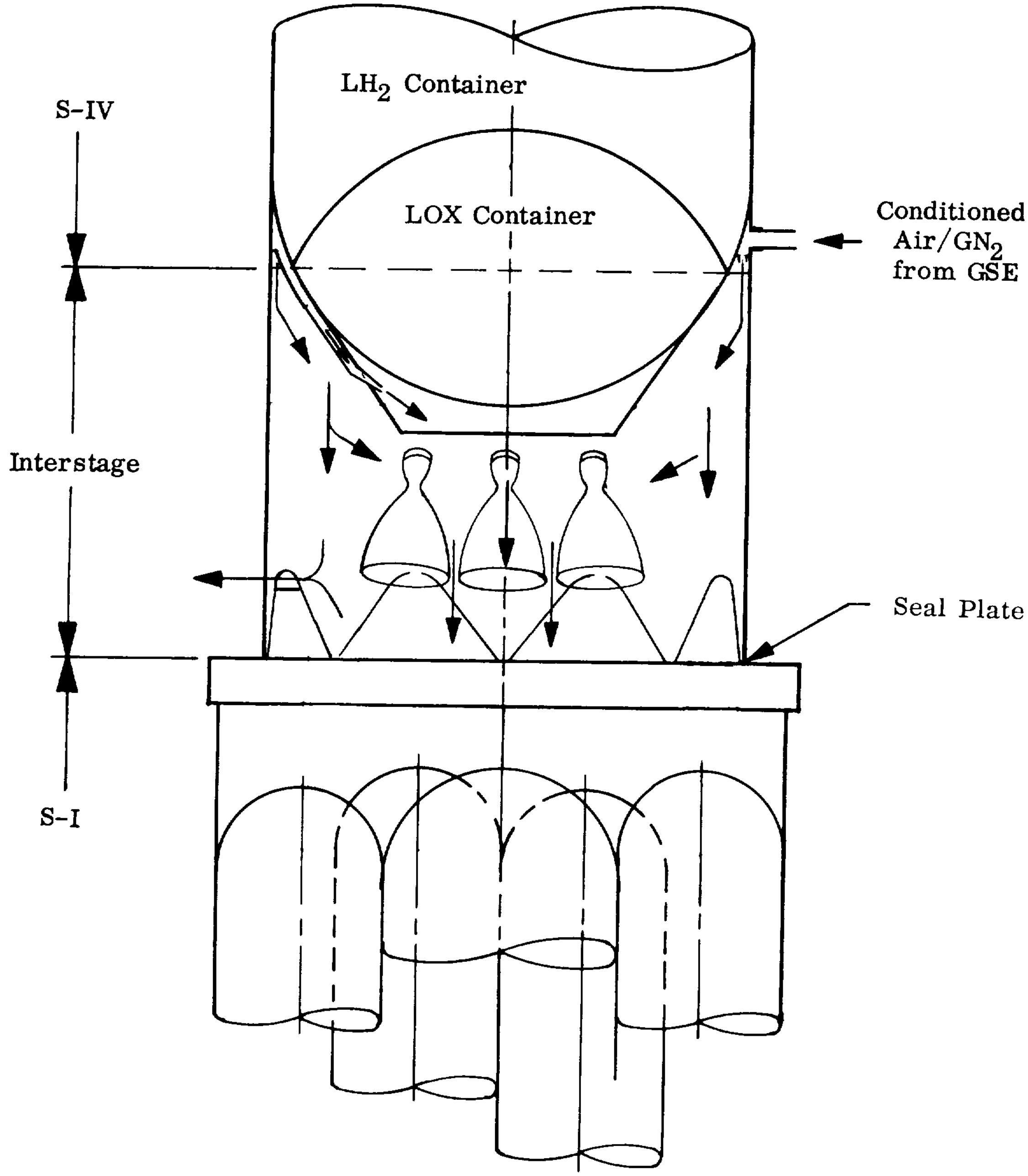


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Figure 9-3. Interstage Compartment Environmental Control, S-I/S-IV

It is exhausted into the S-IV stage forward compartment and then overboard through vent valves in the instrument unit skin. A thermistor probe located in the exhaust flow of one of the valves supplies temperature data to the ground control stations for temperature regulation of the air or GN_2.

9-7. ENGINE GIMBALLING SYSTEM.

The Saturn I engine gimballing system positions the gimballed engines of the active stage to provide the thrust vectors required for vehicle control. In performing these functions, the gimballing system is controlled by commands initiated by the attitude control and stabilization function. (Refer to Paragraph 6-30).

The engine gimballing system steers the vehicle along its trajectory by providing engine thrust vectors for control of pitch, yaw and roll. The system is active during the ascent phase of the mission (throughout S-I stage and S-IV stage powered flight.) As the vehicle ascends, in addition to the region of high aerodynamic pressure (35,000 to 50,000 feet), it may encounter other disturbances such as thrust misalignments and winds. The forces produced on the vehicle by such disturbances are counteracted by gimballing the engines of the active stage providing thrust vectors which minimize vehicle structural loading and maintain the vehicle on trajectory.

9-8. OPERATION.

The gimballed engines of the two Saturn I stages are positioned by independent, electro-hydraulic servo loops, which are similar in operation. Each of the four outboard engines of the S-I stage is gimballed. The associated servo loop is capable of gimballing the engine in a ± 8-degree square pattern (Figure 8-1) for pitch, yaw or roll control. All six of the S-IV stage engines are gimballed in a ± 4-degree pattern, Figure 8-2, to provide pitch and yaw control. Engines 1, 2, 3 and 4 are utilized for roll control.

9-9. STAGE IMPLEMENTATION.

The typical hydraulic actuation system, Figure 9-4, is composed of an accumulator and manifold assembly, a main hydraulic pump and associated lines and valves, and two servo actuators. These components are described in the following paragraphs. The location of components on the four outboard H-1 engines of the S-I stage is shown in Figure 9-5.

1 Main Hydraulic Pump
2 Seepage Plug (2)
3 Auxiliary Pump
4 Auxiliary Pump, Electrical Motor
5 Case Drain Filter Element
6 Check Valve
7 Check Valve
8 Quick Disconnect High Pressure Nipple
9 Filter Element
10 Accumulator Reservoir Manifold Assembly
11 GN_2 Charging Valve
12 High Pressure Relief Valve
13 Quick-Disconnect Low-Pressure Nipple
14 Low-Pressure Relief Valve
15 Pressure Transducer
16 Thermal Switch
17 Actuator Arms (4)
18 Servo Valves (2)
19 Servoactuators (2)

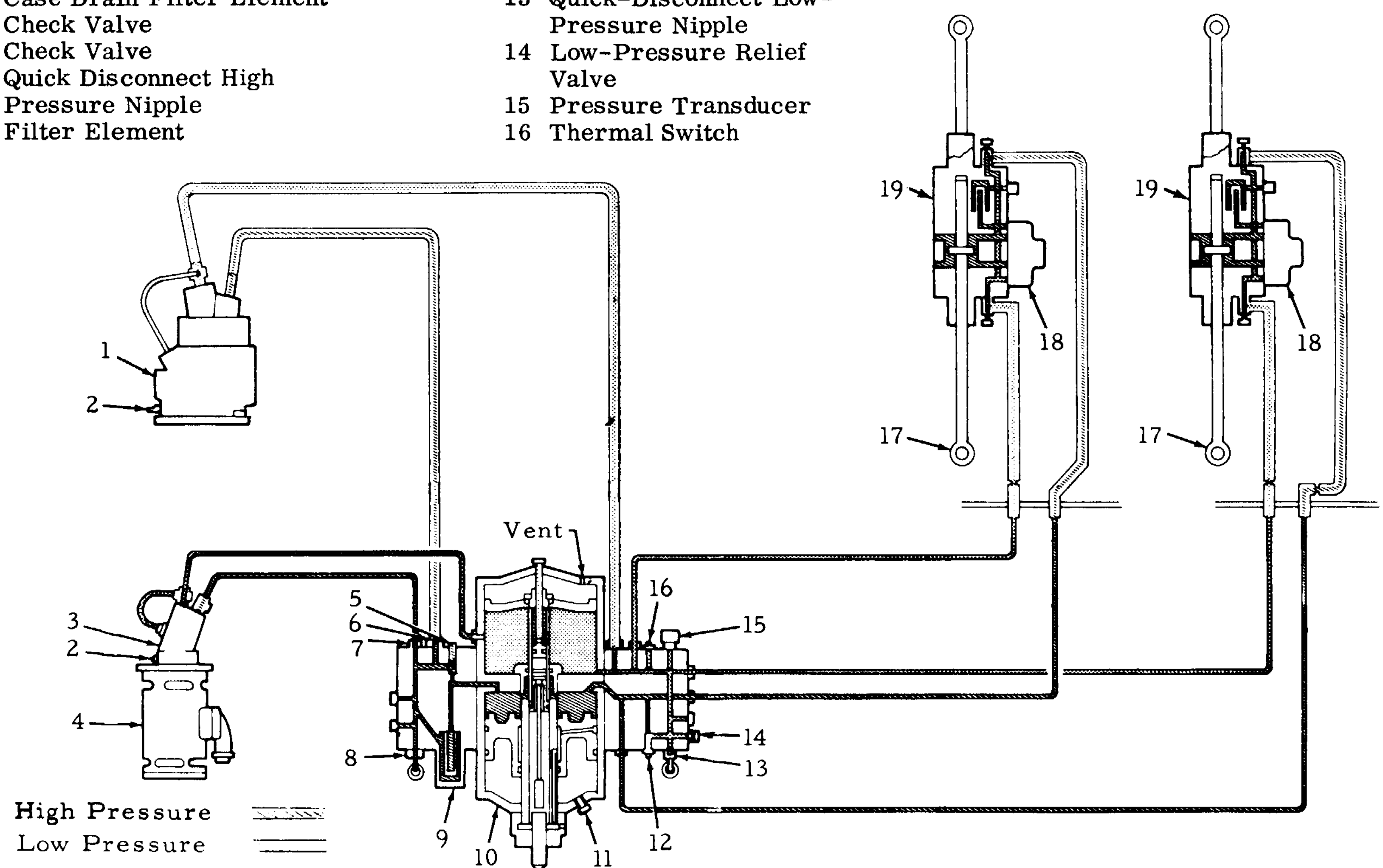

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Figure 9-4. Engine-Gimbal Hydraulic System

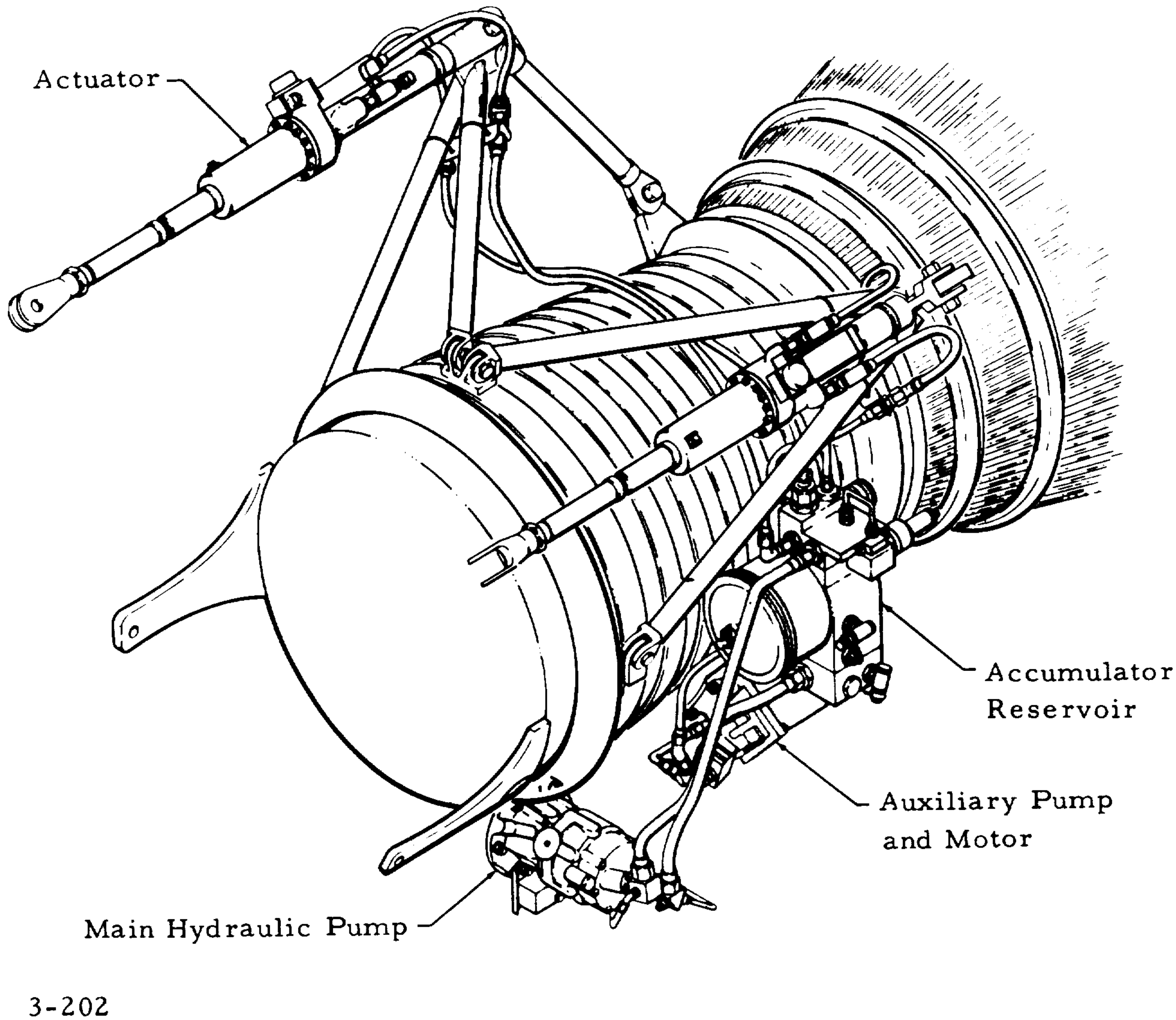

Figure 9-5. Engine Gimballing System Components

9-10. Accumulator Reservoir and Manifold Assembly. This assembly is composed of a high-pressure piston-type accumulator, a low-pressure piston-type reservoir, and a manifold assembly.

The accumulator functions as a secondary source of fluid power and supplies instantaneous actuator demand flow in excess of pump capacity. In addition, the accumulator functions as a pressure surge suppressor and pump ripple eliminator

Fluid within the accumulator is maintained at a pressure of 3200 psig nominal. The reservoir stores the hydraulic fluid for the system. A low-pressure piston unit located inside the reservoir compensates for fluid expansion caused by temperature variations. The reservoir is bootstrapped to the accumulator to maintain return line pressurization and prevent pump inlet cavitation. Reservoir fluid is pressurized at 53.3 psig.

Prior to being filled, the accumulator is charged with GN_2 from a ground source through a charging valve (11). Figure 9-4. The system is filled with hydraulic fluid through a quick disconnect high-pressure nipple (8) and then purged and bled. All hydraulic fluid pumped into the system flows through the filter element (9) into the accumulator reservoir and manifold assembly (10). The functions of the assembly components are as follows:

a. a differential pressure indicator indicates the pressure drop across the filter element,

b. a thermal switch (16) transmits a signal if fluid temperature exceeds a predetermined level,

c. a pressure transducer (15) monitors fluid pressure in the high-pressure accumulator,

d. a potentiometer continuously monitors the fluid level in the reservoir,

e. a high-pressure relief valve (12) protects the high-pressure side of the system by allowing excessive pressure to vent into the low-pressure side of the system,

f. a low pressure relief valve (14) protects the low-pressure side of the system.

A quick disconnect low-pressure nipple (13) is used to drain the system. The system may also be drained by removing plugs located in the servoactuator housings (19). After the system is drained the filter element can be removed for cleaning. Gaseous nitrogen pressure in the accumulator reservoir and manifold assembly (10) can be released through the GN_2 charging valve (11). Both the auxiliary (3) and the main (1) hydraulic pump are provided with seepage plugs (2). Bleed valves for the high- and low-pressure sides of the system are contained on both pumps. Fluid from the auxiliary pump is filtered by the case drain filter element (5) before entering the accumulator.

9-11. Main Hydraulic Pump. The main hydraulic pump (1), a variable displacement type, is driven by the H-1 engine turbopump. Hydraulic fluid, drawn from the low-pressure reservoir, is pumped through the check valve (6) and the filter element (9) into the high-pressure accumulator where it is distributed to the two servo actuators (19) through a filter within each actuator. Fluid pressure on the high-pressure side of the pump is approximately 3200 psig.

9-12. Servo Actuators. Two linear, equal area, double acting, electro-hydraulic servo actuators located 90 degrees apart on each engine gimbal the engine. The electro-hydraulic servo valve (18) on each actuator is controlled by a command from the control computer located in the instrument unit. The servo valve directs high-pressure fluid against the actuator piston moving the actuator arms (17). A feedback transducer (potentiometer) is mounted on each actuator which transmits an electrical feedback signal to the control computer indicating actuator position. Hydraulic fluid from the actuators is returned to the low pressure reservoir. A manually operated bypass valve interconnects the two sides of the actuator cylinder to provide manual movement of the actuator.

9-13. Auxiliary Pump. The auxiliary pump (3) is a single stage, fixed angle, variable delivery, nine cylinder unit driven by an electric motor (4). The pump supplies hydraulic pressure to the system for ground operation. During auxiliary pump operation, the main hydraulic pump (1) is protected from high pressure fluid by the check valve (6). After engine ignition, a check valve (7) protects the auxiliary pump (3) from high-pressure fluid. Excessive motor temperature is indicated by a thermal switch on the electric motor.

9-14. SEPARATION SYSTEM.

The primary function of the Saturn I separation system is to provide positive separation of the S-I stage from the S-IV stage during vehicle flight. (The following description does not include an explanation of the separation of the S-IV stage/instrument unit from the Apollo payload occurring after the payload is injected into earth orbit.)

To lift a given payload into orbit, it is desirable to use a launch vehicle of minimum weight. The design of a minimum-weight vehicle capable of lifting the payload required for the Apollo program necessitates the use of more than one propulsion stage when restricted to present space vehicle technology. During the flight of a multistage vehicle, as a stage is expended it is discarded and the next stage forward provides the thrust for continued payload boost.

9-15. OPERATION.

In separating the two stages of the Saturn I launch vehicle, the following principal functions occur:

a. Purging and ventilating of the S-IV stage engine compartment during prestart chilldown.

b. Cutoff of engines of the S-I stage.

c. Acceleration of the S-IV stage.

d. Physical separation of the S-I stage from the vehicle.

e. Deceleration of the S-I stage.

f. Ignition of the S-IV stage engines.

Prior to separating the stages and starting the cryogenic-propellant S-IV stage engines, it is necessary to cool down the propellant feed system so that propellants do not vaporize within the pump or feed lines during engine starting. Prestart chilldown is accomplished by circulating the fuel (LH_2) and the oxidizer (LOX) through the engine feed system. The fuel is then vented overboard; the oxidizer flows out through the thrust chamber of the engines into the interstage. Purging of the area beneath each S-IV stage engine, during chilldown and venting of the engine compartment, is required to maintain an inert atmosphere.

The separation operation is initiated approximately 148 seconds after liftoff when a low-level sensor in one of the S-I stage propellant containers indicates that the propellants are near depletion. When this occurs control circuits within the vehicle initiate engine cutoff. A controlled thrust termination is necessary to prevent attitude deviations which could occur from unsymmetrical booster burnout. Burnout, as opposed to controlled cutoff, occurs when engines stop burning as a result of propellant depletion. A controlled cutoff is important because during the separation sequence there is a period of approximately four seconds, between S-I stage engine cutoff and S-IV stage engine ignition and thrust buildup when the vehicle coasts in uncontrolled flight. In terminating the S-I stage thrust, the inboard engines are cut off first.

Following the controlled cutoff of the inboard engines, and then the outboard engines, the ullage motors are ignited to provide acceleration of the S-IV stage. The acceleration provides sufficient propellant pressure at the inlet of each engine pump for reliable starting. The propellant pressure at the pump inlet is maintained above the design NPSH (net positive suction head) to prevent cavitation.

Adequate clearance (10 feet minimum) between the separating stages must be achieved prior to S-IV stage engine ignition to minimize stage interactions. The signal that activates the frangible nuts to detach the S-I stage from the vehicle also ignites the retromotors. A circuit time delay of 0.05 seconds nominal ensures that the frangible nuts actuate before the retromotors ignite. This prevents retro thrust from acting on the vehicle before physical separation occurs and eliminates the possibility of unseating S-IV stage propellants. The retromotor thrust decelerates the S-I stage providing rapid and complete physical separation of the stages.

Upon completion of the physical separation, the S-IV stage engines are started. The final function of the separation system is to jettison the burned-out ullage motors from the S-IV stage minimizing the vehicle weight. The complete staging sequence is tabulated in Table 9-1.

9-16. S-I STAGE IMPLEMENTATION

The separation system components associated with the S-I stage include retromotors and the LOX-SOX disposal system.

Four solid-propellant retromotors are mounted 90 degrees apart on the spider beam located at the forward end of the S-I stage. The thrust vectors of the motors are directed aft and 11 degrees, 6 minutes radially inward, Figure 9-6. The motors provide deceleration of the stage to aid in the complete and expeditious separation of the S-I stage from the vehicle.

The LOX-SOX disposal system (Figures 9-7 and 9-8) supplies GN_2 for purging of the area beneath each S-IV stage engine during prestart cooldown. The disposal system is mounted on the forward end of the S-I stage. Beneath each S-IV stage engine there is a dispersal manifold ring which has a row of holes around its inner circumference. The inert gaseous nitrogen, flowing out of the holes, saturates the area beneath each S-IV engine. The GN_2 is supplied from four high-pressure triplex spheres and two single high-pressure spheres located in the forward section of the S-I stage.

9-17. S-IV STAGE IMPLEMENTATION.

The separation system components associated with the S-IV stage include blowout panels, frangible nuts and ullage motors.

Table 9-1. S-I/S-IV Staging Sequence

Item	*Approximate Time	Remarks
S-I level sensor signal to open S-I/S-IV interstage ports initiate S-IV LOX prestart	L. O +148 sec. (nominal) -8.4 sec.	The interstage ports are opened at this time to provide a path for venting the LOX prestart product. LOX prestart duration is 10 sec.
Signal to open LOX/SOX disposal valves 2, 3, 5 and 6.	-7.3 sec.	
Signal to cutoff inboard engines	-6.4 sec.	Conax valves are fired. Thrust decay to 10% estimated at 0.4 sec.
Signal to open LOX/SOX disposal system valve 4	-5 sec.	
Signal to open LOX/SOX disposal system valves 1 and 7	-3.5 sec.	
Signal to cutoff outboard engines.	-0.4 sec.	Thrust decay to 10% is estimated at 0.4 sec.
Signal to ignite S-IV ullage motors	-.111 sec.	Ullage motor thrust buildup to 50% of maximum occurs in a maximum time of 0.060 seconds (nominal 0.030 seconds at 70° F.). Ullage motors are required to burn a minimum of three seconds.

Table 9-1. S-I/S-IV Staging Sequence (Cont'd)

Item	*Approximate Time	Remarks
Signal to activate separation nuts, retromotors, actuate control switch	-0	The S-IV control switch, a motor driven rotary switch, transfers control and telemetry signals from the S-I stage to the S-IV stage at separation.
Explosive nuts actuate, separation bolts start to retract	+.015 sec.	First possible S-IV stage axial motion
Control switch completes operation	+.147 sec.	
Signal to open S-IV hydraulic accumulator valves	+.85 sec.	
Signal to open helium heater LOX valve	+1.4 sec.	
Signal to open start valves on S-IV overboard LH_2 valve, open helium heater LH_2 valve, engine and helium heater igniters on.	+1.7 sec.	Minimum of 10 feet clearance between stages (S-IV stage exit plane to separation plane) at S-IV stage engine ignition.
Ullage motors burn out	+3.5 sec.	RL10A-3, S-IV stage engines attain mainstage thrust before ullage motors burn out.

Table 9-1. S-I/S-IV Staging Sequence (Cont'd)

Item	*Approximate Time	Remarks
Signal to: arm S-IV engine out cutoff capability, de-energize hydraulic accumulator valves (stay open) de-energize engines and helium heater igniters	+4.6 sec.	
Signal to activate propellant utilization system	+6.6 sec.	
Signal to jettison ullage motors	+20 sec.	

*NOTE: Time values are based on preliminary information.

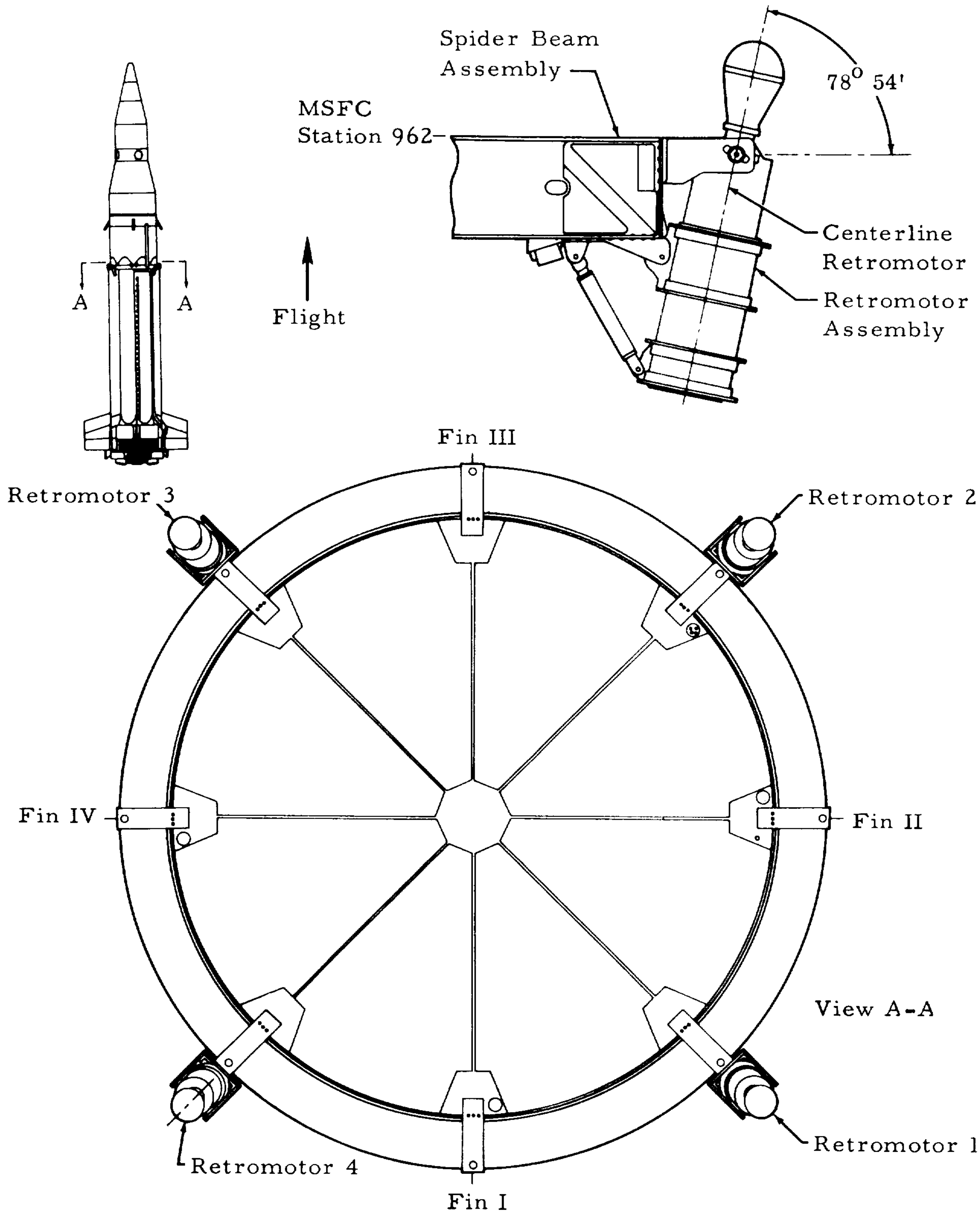

Figure 9-6. Retromotor Installation

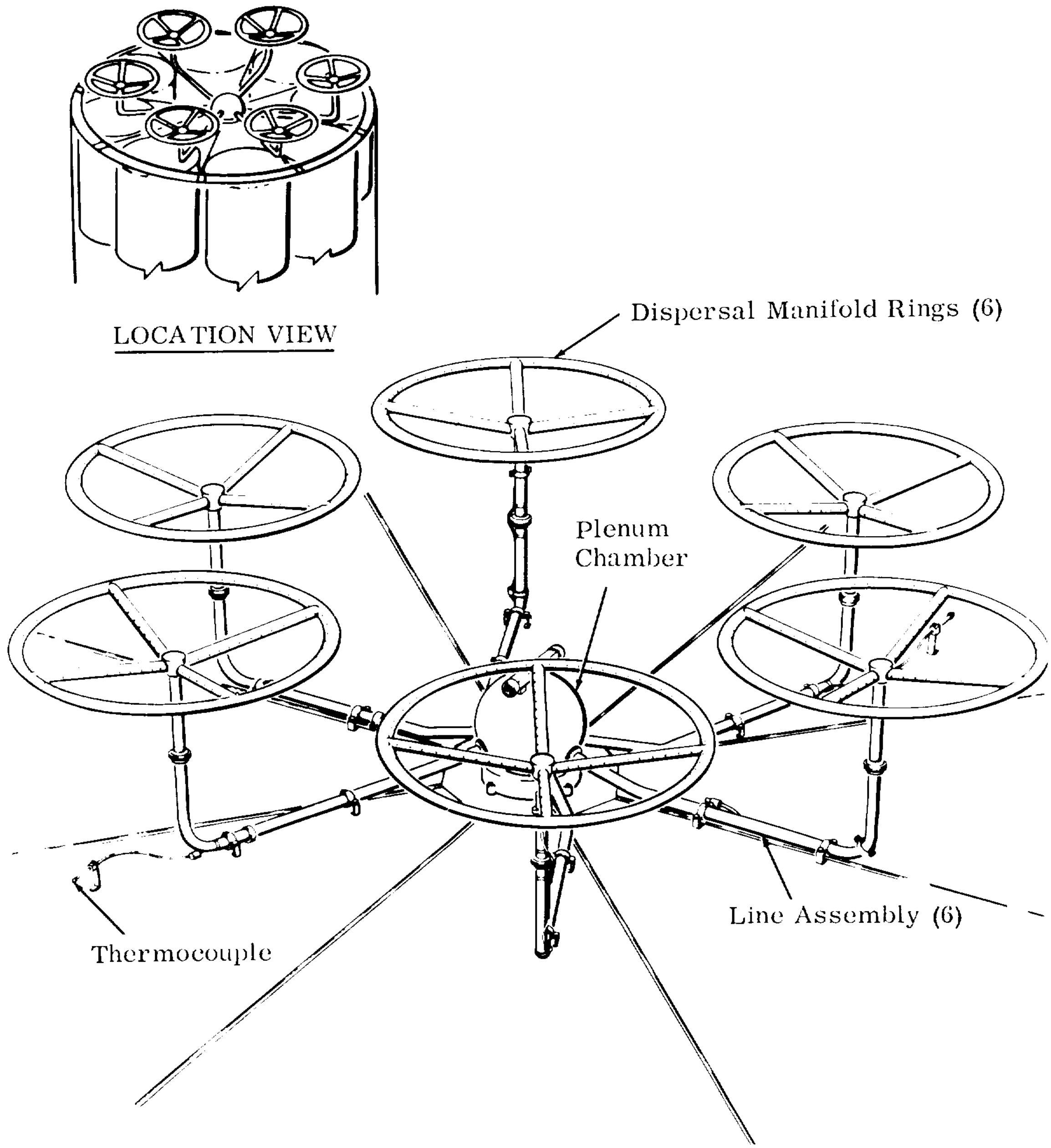

Figure 9-7. LOX-SOX Disposal System

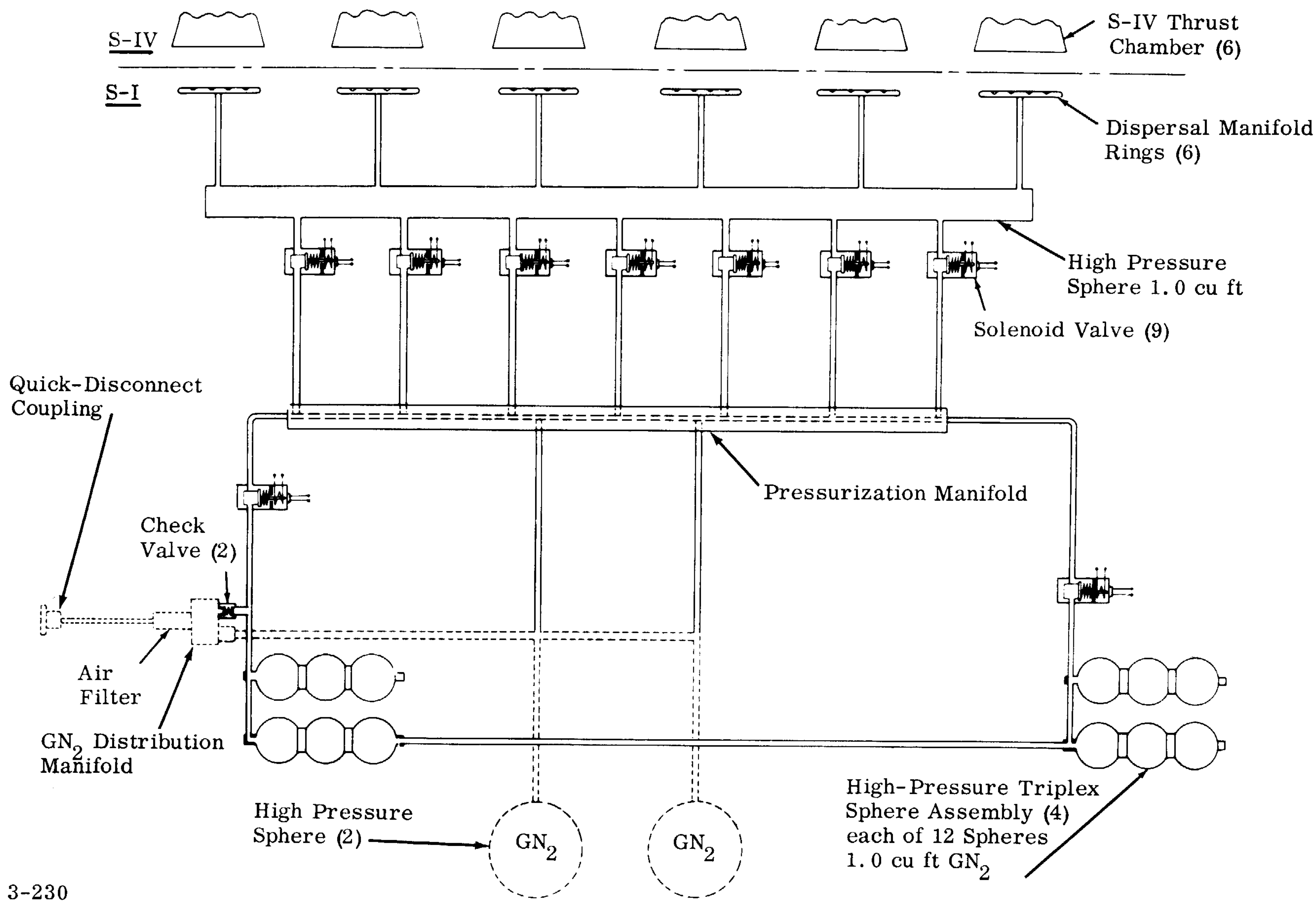

Figure 9-8. LOX-SOX Disposal System Schematic

The eight blowout panels are evenly spaced around the aft end of the S-I/S-IV interstage. Each panel covers a triangular vent port which is opened at the beginning of the S-IV stage engine-prestart cooldown to vent LOX from the interstage. The panels are removable for servicing and maintenance of equipment.

Four separation frangible nut and bolt assemblies join the S-IV stage to the S-I stage (at MSFC station 1147). During S-I/S-IV staging the explosive frangible nuts are broken by means of two explosive charges within each nut. Physical separation of the stages occurs after the nuts are fractured.

Four solid-propellant ullage motors are used to accelerate the S-IV stage providing propellant positioning and sufficient turbopump inlet pressure for engine starting. Each ullage motor is mounted in a fairing which is bolted to the aft skirt of the S-IV stage at two points using frangible nuts, Figure 9-22. The motors, located 90 degrees apart around the skirt, are canted 35 degrees from the vehicle centerline to minimize the effect of exhaust gases on the vehicle hardware. Each ullage motor provides a nominal average thrust of 3460 pounds at 70°F to position S-IV propellants for RL10A-3 engine ignition and to aid in separation during S-I/S-IV staging. After the motors are expended, the four fairings are jettisoned by breaking the frangible nuts. This occurs approximately 20 seconds after the separation signal is initiated.

(Retromotors are not required on the S-IV stage for separation of the S-IV stage and instrument unit from the Apollo payload. However, the vehicle is designed with a capability for inclusion of two TX-280 solid-propellant retromotors on the stage.)

9-18. ORDNANCE SYSTEMS.

Many of the mechanical operations performed during a Saturn I mission require reliable, short time, high energy, concentrated forces. These forces are provided by the ordnance system components. High reliability is achieved by providing redundant components throughout the system.

During launch, the S-I stage engines are started by ordnance components which provide the forces required for initial turbopump operation and ignition of propellants used to continue the operation. At lift-off, the ground-to-vehicle electrical power transfer is made positive and permanent by ordnance components. During S-I/S-IV staging, blowout panels are released, the individual engine thrusts are terminated in symmetrical unison, ullage and retromotors are fired to provide auxiliary propulsion, vehicle structural connections are severed, and spent ullage motors are jettisoned. These operations are also accomplished by components of the ordnance systems. For range safety, ordnance components are used to terminate engine thrust and disperse vehicle propellants.

9-19. OPERATION.

Ordnance components used on the Saturn I launch vehicle are operational during the launch and ascent phases of the mission. Because of the potential hazard involved, the explosive initiators of components are not installed, and the electrical circuits of the ordnance system are not completed until all personnel except the ordnance crew are clear of the launch pad.

9-20. Launch Phase. During launch H-1 engine starting is initiated by ignition of a solid-propellant gas generator (SPGG). The SPGG produces gas for the initial acceleration of the high-speed turbine which drives the LOX-fuel turbopump and provides primary ignition of the liquid-propellant gas generator (LPGG). Secondary ignition of the LPGG is supplied by LPGG igniters. The LPGG produces the gas for continued operation of the high-speed turbine.

At liftoff, explosive switches are fired to provide positive and permanent connections between the launch vehicle electrical system and its internal power supply.

9-21. Ascent Phase. During ascent of the launch vehicle, when a low-level sensor in one of the S-I stage propellant containers indicates that propellants are near depletion, the S-I/S-IV separation sequence is initiated. Ordnance components play a major role during separation. Detonating cord cuts blowout panels to open vent ports in the S-I/S-IV interstage at the beginning of the RL10A-3 engine prestart sequence to vent LOX from the interstage area. An explosively actuated Conax valve on each H-1 engine provides for the controlled cutoff of first the four inboard engines and then the four outboard engines. Ullage motors provide vehicle acceleration for propellant positioning and to ensure sufficient turbopump inlet pressure for S-IV stage engines ignition. Retromotors decelerate the S-I stage providing rapid and complete physical separation of the stages. Physical separation of the S-I stage from the S-IV stage is accomplished by breaking the frangible nuts which are used to join the stages. Explosive charges within each nut are ignited to fracture the nuts. Frangible nuts are also used to attach the ullage motor fairings to the S-IV aft skirt. The nuts are broken to jettison the ullage motors after they have finished burning.

Throughout the ascent phase of the mission the range safety officer can terminate the flight at any time by means of the propellant dispersion system. When the system is actuated the active stage engines are shut down and detonating cord is ignited to cut open the propellant containers. To attain high reliability each stage (S-I and S-IV) has a separate dispersion system.

9-22. S-I STAGE IMPLEMENTATION.

Ordnance on the S-I stage includes components used for permanent transfer of power and engine starting and cutoff, retromotors used during stage separation, and components used for propellant dispersion.

9-23. Explosive Liftoff Switches. Approximately 35 seconds before liftoff the vehicle is switched from ground power to internal power. Transfer of power is accomplished by a network of relays. At launch, explosive switches connected in parallel with the relay contacts are fired to form positive and permanent circuits which eliminate possible power interruptions caused by relay failure or relay contact chatter.

9-24. H-1 Engine Ordnance. Ordnance components are used both in starting and cutting off the H-1 engines. For starting, each engine is equipped with a solid-propellant gas generator, two solid-propellant gas generator initiators, and two liquid-propellant gas generator igniters. A Conax valve initiates engine cutoff. The components are described below.

Solid-Propellant Gas Generator. The solid-propellant gas generator (SPGG), mounted on each engine as illustrated in Figure 9-9, is a solid-propellant disposable cartridge which cannot be reloaded or reused. During engine starting the SPGG, Figure 9-10, produces gas at a rate of 4.8 pounds per second for approximately 1.0 second to accelerate a high-speed turbine which drives the LOX-fuel turbopump. The solid-propellant grain continues to burn

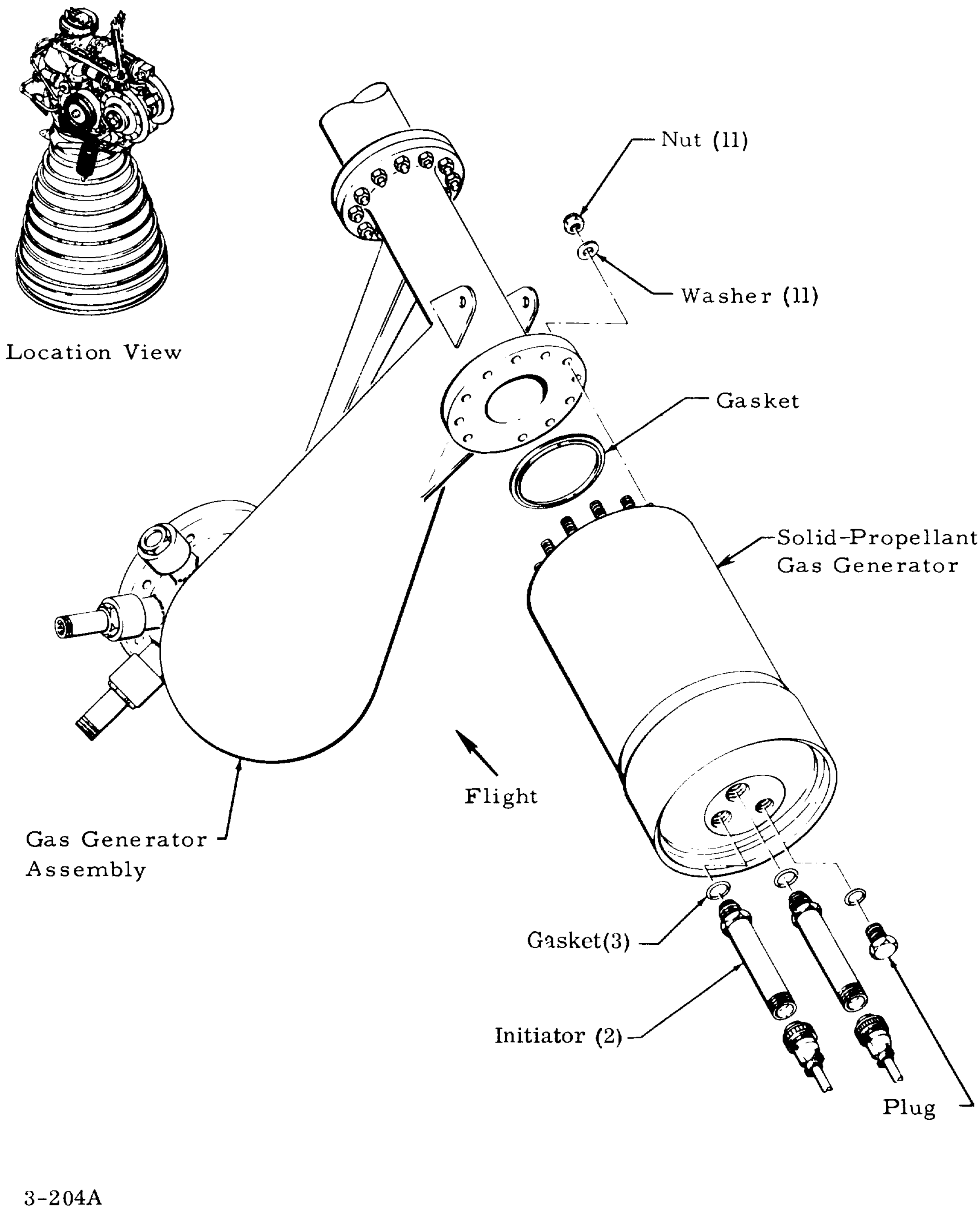

Figure 9-9. Solid-Propellant Gas Generator and Initiator Assembly

100 to 200 milliseconds after LOX and fuel enter the combustion chamber of the liquid-propellant gas generator (LPGG) providing primary ignition of the LPGG propellants.

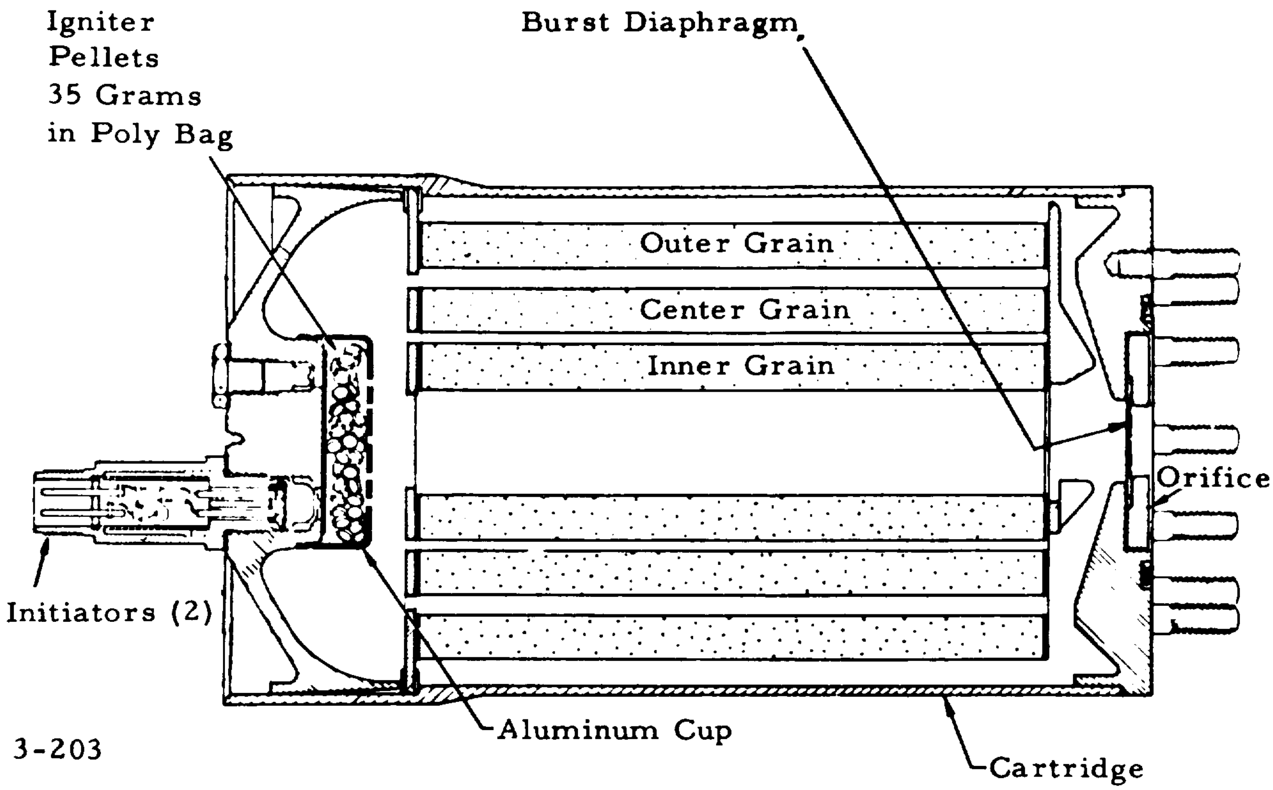

Figure 9-10. Solid-Propellant Gas Generator

Solid Propellant Gas Generator Initiators (Figure 9-11). The burning of the solid propellants in the SPGG of each engine is started by two initiators. The initiators are pyrotechnic devices consisting of a two-pin electrical receptacle and a moisture sealed cartridge assembly containing a pyrotechnic "match-head mix" material. An electrical impulse of 500-volt ac, 1.5 amps (minimum) closes the circuit in the initiator causing a nichrome wire to glow, igniting the pyrotechnic material.

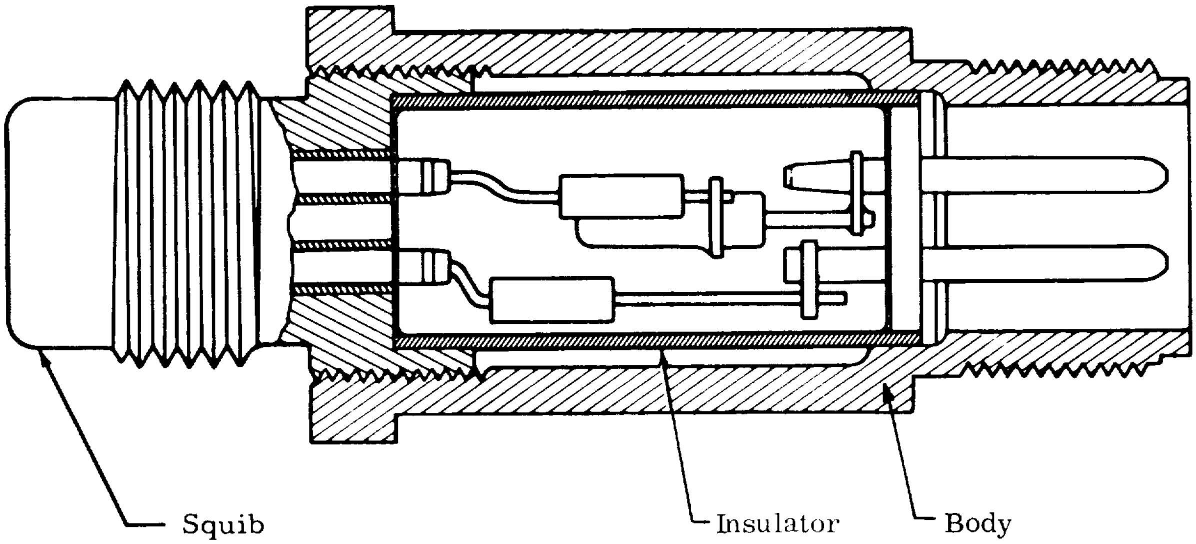

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Figure 9-11. Solid-Propellant Gas Generator Initiator

Liquid-Propellant Gas Generator Igniters (Figure 9-12). The auto-ignition igniters, installed on the engine as illustrated in Figure 9-13, are pyrotechnic devices that provide secondary ignition of the LOX-fuel mixture in

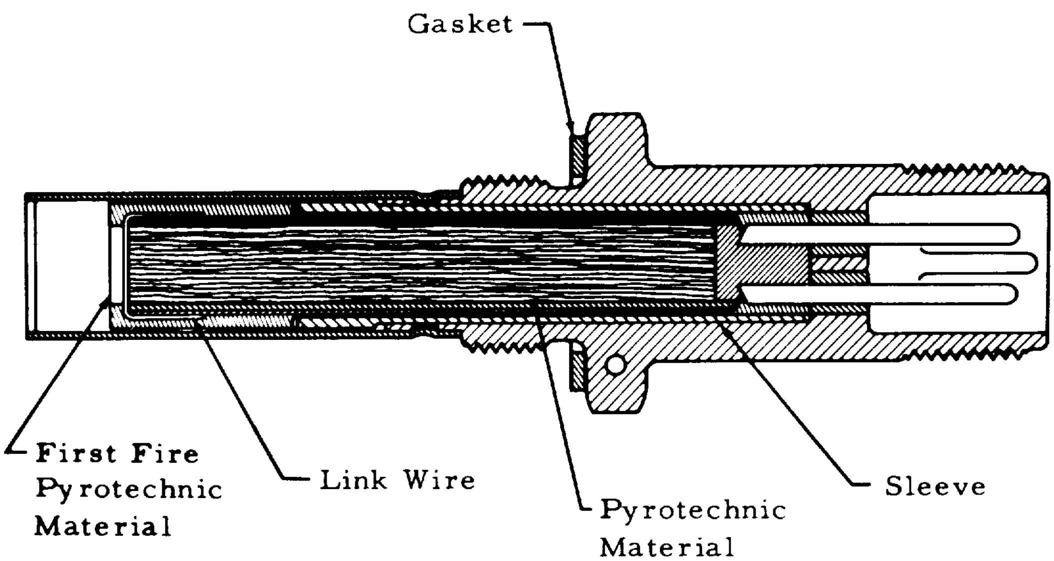

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Figure 9-12. Liquid-Propellant Gas Generator Igniter

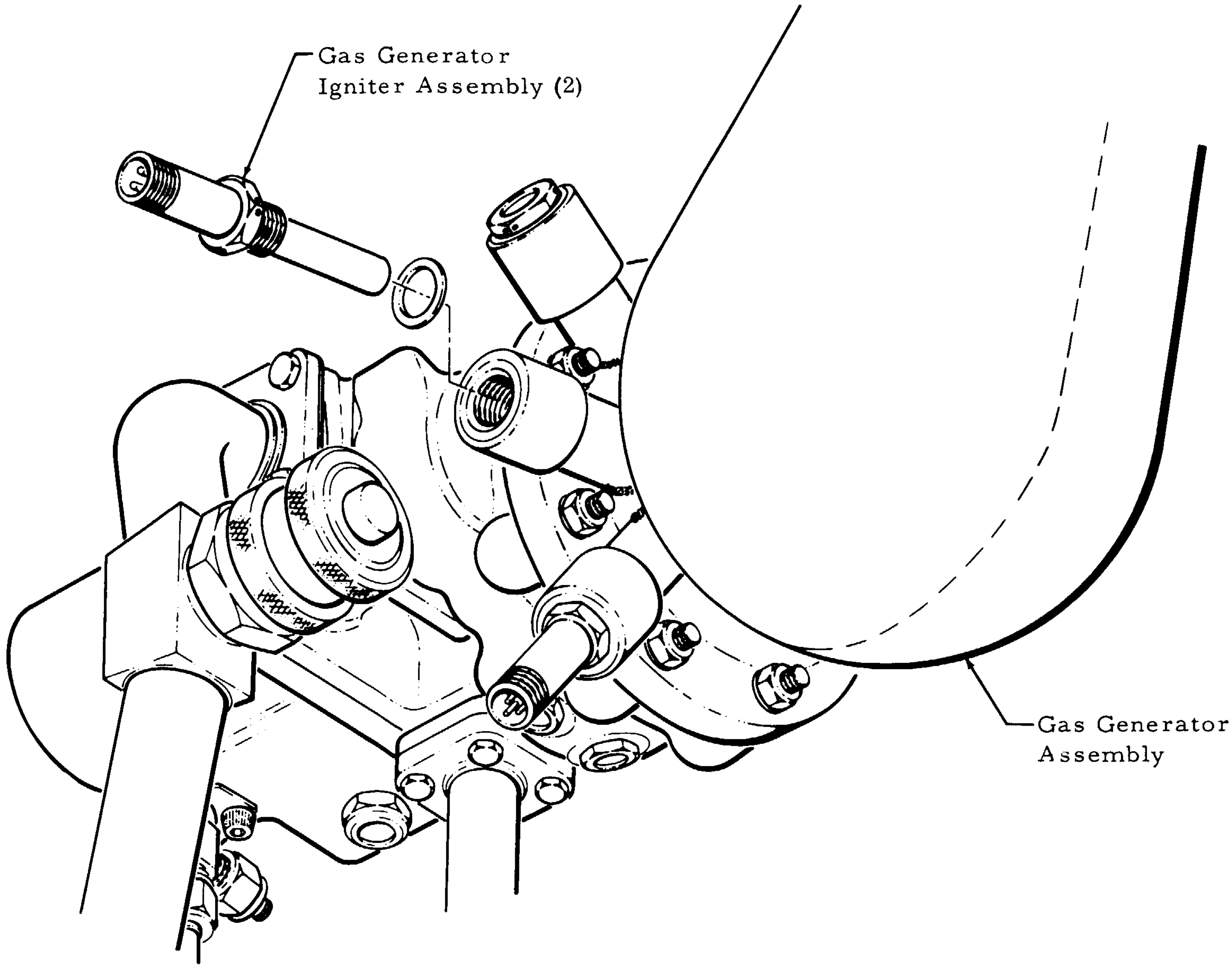

Figure 9-13. Liquid-Propellant Gas Generator Igniter Installation

the LPGG combustion chamber. Each igniter (two per engine) consists of an electrical receptacle and a cartridge assembly. The cartridge is housed in a tube assembly utilizing two inner sleeve assemblies containing the main pyrotechnic charge and a first-fire pyrotechnic charge. A two-amp fuseable-link wire housed in the cartridge assembly is used in indicating that the device has fired. The igniters, which are sensitive to heat and impact, are ignited by hot gases produced by the SPGG.

Main LOX Valve Closing Control Valve (Conax Valve). Each H-1 engine is equipped with one normally closed Conax valve (Figure 9-14). When the valve is open, fuel from the pump outlet flows through the valve to the closing port of the main LOX valve initiating engine shutdown. The Conax valve is opened when an electrical signal ignites the explosive charge in one or both

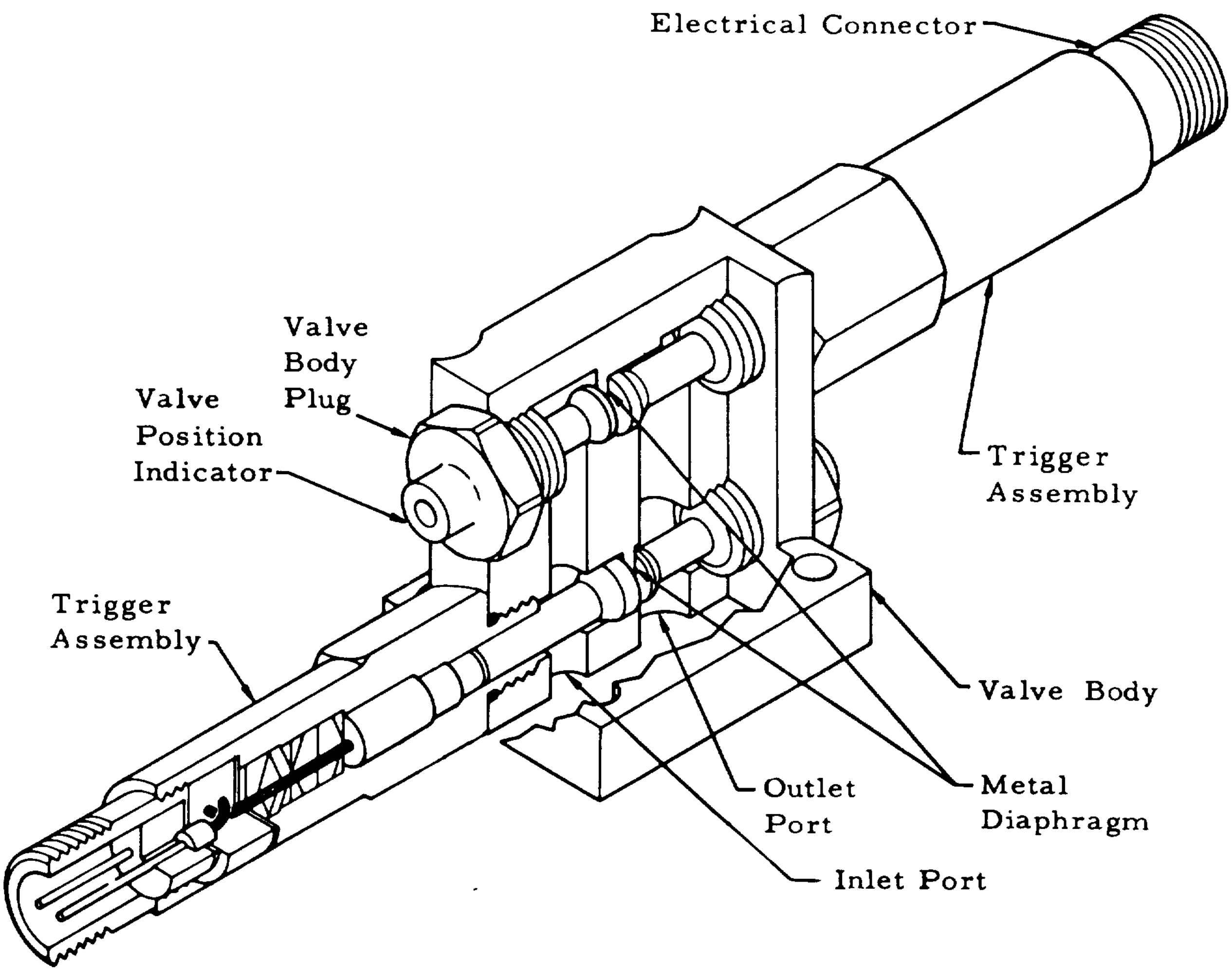

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Figure 9-14. Main LOX Valve Closing Control Valve (Conax Valve)

[redacted]

trigger assemblies. The resulting force shears one or both of the metal diaphragms within the valve body. A valve position indicator opposite each trigger assembly indicates whether or not the valve is open.

9-25. Retromotors. Four solid-propellant Aerojet 2KS 36,250 retromotors provide deceleration to the S-I stage during S-I/S-IV staging to prevent stage interaction. The retromotors are mounted 90 degrees apart on the spider beam located at the forward end of the S-I stage, Figure 9-6. The motor thrust vectors are directed aft and 11 degrees, 6 minutes radially inward. The ignition system for each retromotor is illustrated in Figure 9-15. Two electronic bridge wire firing units furnish the electric firing charge to two EBW initiators mounted in the motor igniter. When fired the exhaust gases produced by the motor igniter ignite the retromotor solid propellant. A pressure gage connected at the base of each retromotor by a pressure tube indicates whether or not the motor is firing. Pressure calibration valves are installed adjacent to each pressure gage. The performance parameters for the retromotors are given in Table 9-2.

Table 9-2. Performance Parameters, 2KS-36, 250 Retromotor

Item	Parameter
Length (over-all)	64.28 inches
Total weight (maximum)	500 pounds
Total weight (nominal)	481 pounds
Propellant weight (nominal)	327 pounds
Time of burning (t_b at 60° F)	2.15 seconds
Thrust (average during t_b at 250,000 feet)	37,000 pounds
Total impulse	74,500 pounds per second
Propellant disignation	ANP-512DS Mod. 3
Flame temperature (adiabatic)	4600° F
Ignition	Exploding bridgewire
Experimental specific impulse	224 seconds
Theoretical specific impulse	232 seconds

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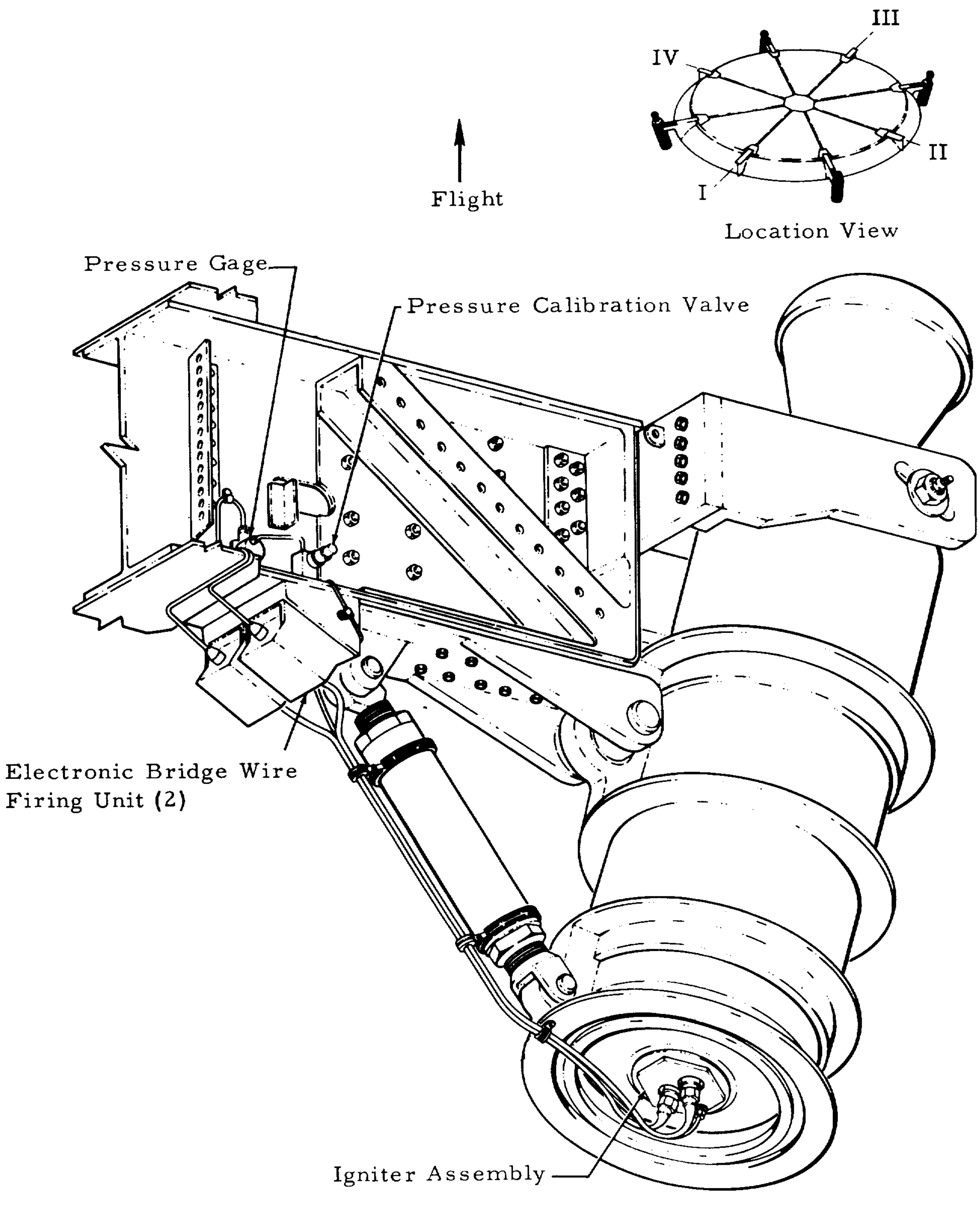

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Figure 9-15. Retromotor Ignition System

9-26. Propellant Dispersion System Ordnance. If the vehicle becomes a hazard during flight, the range safety officer can terminate the flight by means of the propellant dispersion system. The system ordnance consists of two electronic bridge wire firing units, a safety and arming (S&A) device into which are assembled two EBW detonators and two Primacord initiators, Primacord trains, and linear shaped charges. When the system is activated, shutdown of the active stage engines is initiated.

Electronic Bridge Wire Firing Units (Figure 9-16). The firing units consist of a high-voltage supply, a capacitor, and an arc-gap switch closed by means of a trigger circuit. The unit furnishes the high-voltage power for ignition of an EBW detonator. When the switch is closed by a trigger signal from the destruct system controller the capacitor, charged to 2300 $\pm$ 100 volts dc, discharges, firing the EBW detonator to which the unit is connected. Two firing units are used to increase the reliability of the system.

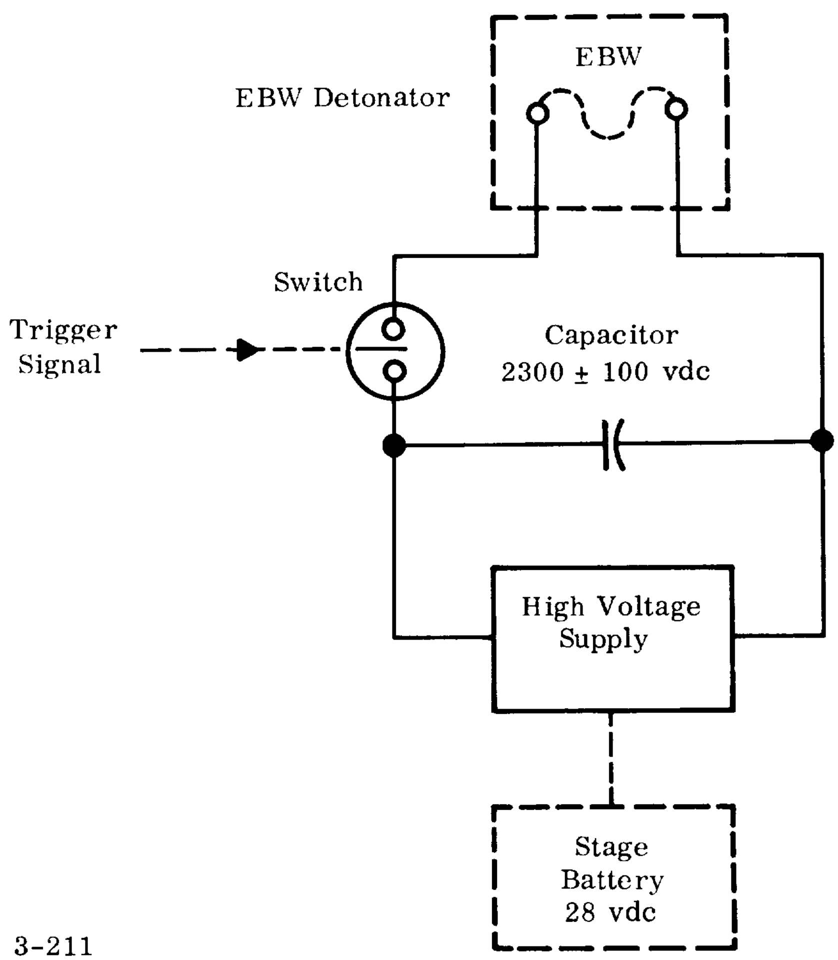

Figure 9-16. Electronic Bridge Wire Firing Unit

EBW Detonators. The EBW detonators are electrically activated devices which rapidly and reliably initiate the explosive leads in the rotor of the S&A device. Each detonator is fitted with two-pin contacts which serve as mounting posts for the bridgewire assembly inside the detonator. The pins are connected externally to the lead-wire cable of the electronic bridge wire firing unit. When the firing unit is triggered, a high-energy pulse of 2300 $\pm$ 100 volts dc is applied to the bridgewire. The wire explodes with the rapid release of a large amount of energy which ignites a train of chemical explosives. A gap in the bridgewire circuit prevents the wire from burning out if power from a source other than the firing unit capacitor is accidentally applied to the detonator. The detonator is hermetically sealed.

Safety and Arming Device (Figures 9-17 and 9-18). The S&A device provides safety for personnel during installation of EBW detonators. The device is an electromechanical unit used as a switch to connect or interrupt the explosive train. The unit includes a rotary solenoid and a rotor containing two explosive leads. Two EBW detonators and two Primacord initiators are installed on opposite sides of the device. In the safe position, the rotor, which is mounted on the solenoid shaft, is positioned such that the explosive leads are perpendicular to and therefore isolated from the EBW detonators and the Primacord initiators. When the solenoid is energized by a signal from the blockhouse prior to liftoff, the rotor is turned 90 degrees to the armed position. The explosive leads are then in line with the explosive train. Firing of the EBW detonators produces a shock wave which is transferred through the rotor explosive leads to Primacord initiators. A visual indicator and monitoring switches indicate whether the device is in the safe or armed position. The housing is pressurized with GN_2.

Primacord Initiators. Two Primacord initiators transfer the firing charge from the S&A device to the Primacord.

Primacord. Primacord is an explosive cord capable of propagating a detonation along any desired path at a speed of approximately 21,000 feet per second. Primacord trains carry the firing charge from the S&A device to the linear shaped charges. The Primacord trains consist of two lengths of

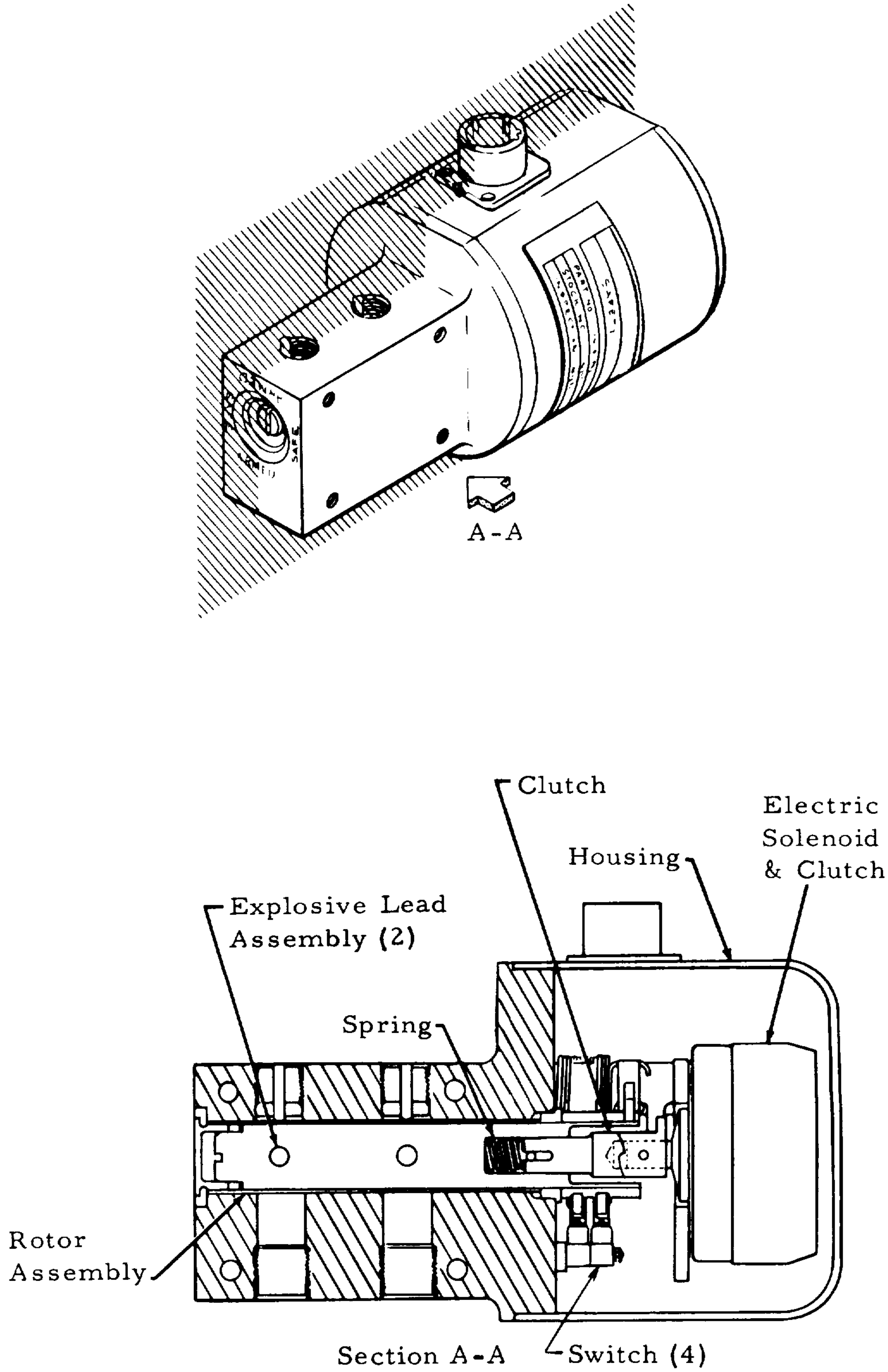

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Figure 9-17. Safety and Arming (S&A) Device

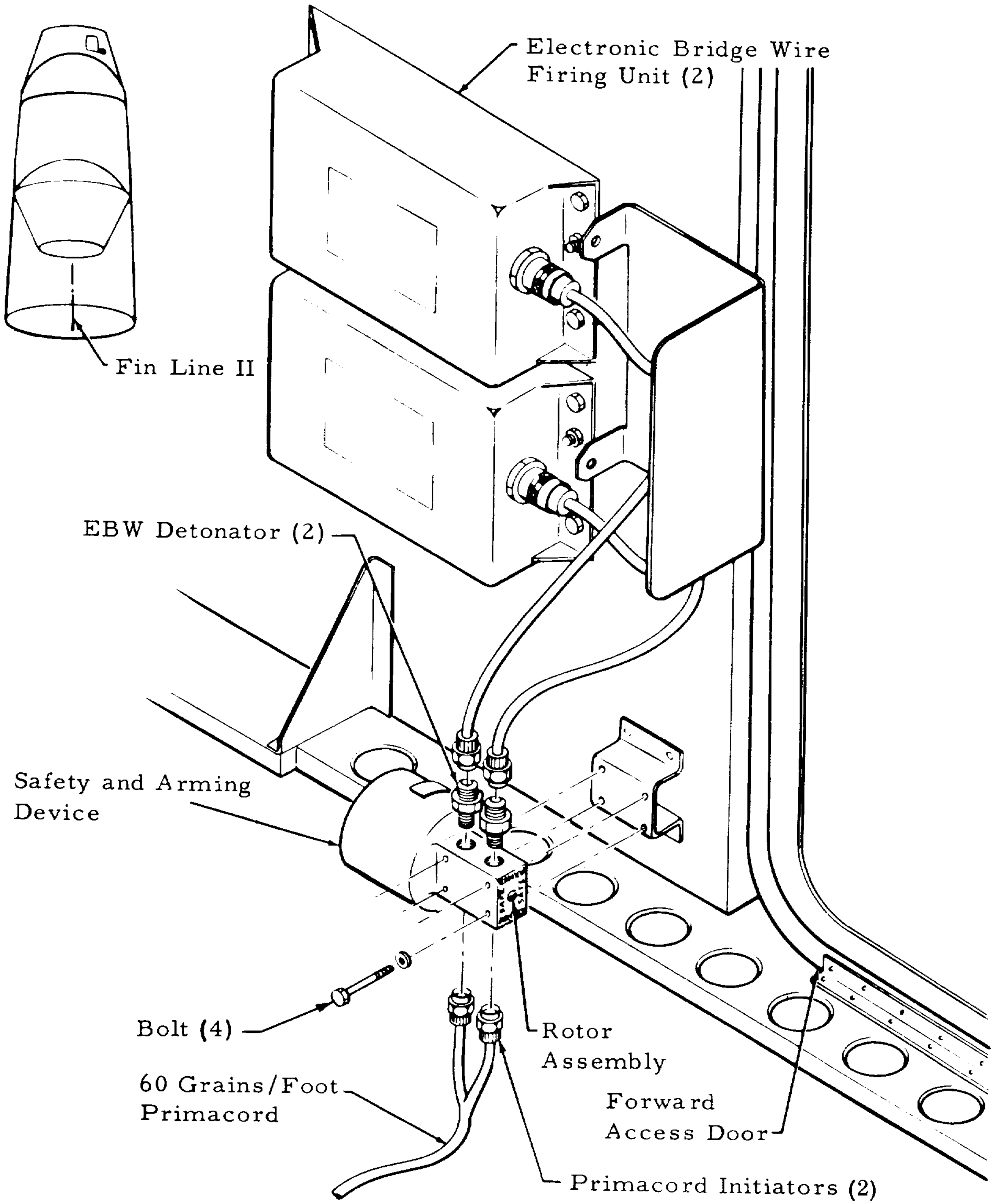

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Figure 9-18. Safety and Arming (S&A) Device Installation

50 grains per foot Primacord. approximately 34-feet and 30-feet long, and two lengths of 60 grains per foot Primacord 5 feet long, Figure 9-19. The two pieces of 60 grains per foot Primacord are connected at one end to the S&A device with the other ends connected to the 50 grains per foot Primacord. The other ends of the two 50 grains per foot Primacord leads are connected together on the side of the stage opposite the S&A device to make a closed circuit. Firing of either or both firing units will ignite the entire train.

Linear Shaped Charges. The explosive charges, consisting of 100 grains per foot lead-sheathed flexible linear shaped charges (FLSC) bonded to a silicon rubber insulation, concentrate the explosive force to provide a cutting action on the surface to which the charges are attached. They are installed on the outside of the propellant containers along the full length of the eight outboard containers and for a distance of 20 feet from the forward end of the center LOX container, Figure 9-19. The FLSC is ignited by primacord spliced to the primacord train.

9-27. S-IV STAGE IMPLEMENTATION.

Ordnance on the S-IV stage includes explosive liftoff switches (refer to Paragraph 9-23), ullage motors, frangible nuts and blowout panels used during separation, and components associated with the propellant dispersion system.

9-28. Ullage Motors. Four GFE solid-propellant Thiokol TX-280 rocket motors are used to position S-IV propellants for RL10A-3 engine ignition and to aid in separation during S-I/S-IV staging. The ullage motors are mounted in fairings on the aft skirt of the S-IV stage and are located at 90-degree intervals around the skirt and are canted at 35 degrees from the vehicle centerline to minimize the effect of exhaust gases on the vehicle hardware (Figure 7-14). Each motor has a nominal burning time of 3.87 seconds and develops a nominal average thrust of 3460 pounds at 70°F under vacuum conditions. Two electronic bridge wire firing units in conjunction with two EBW initiators and a motor igniter provide ignition of each ullage motor, Figure 9-20. A pressure transducer connected by tubing from the igniter of each ullage motor detects ullage motor firing.

9-29. Frangible Nuts. Frangible nuts, Figure 9-21, are used to join the S-IV

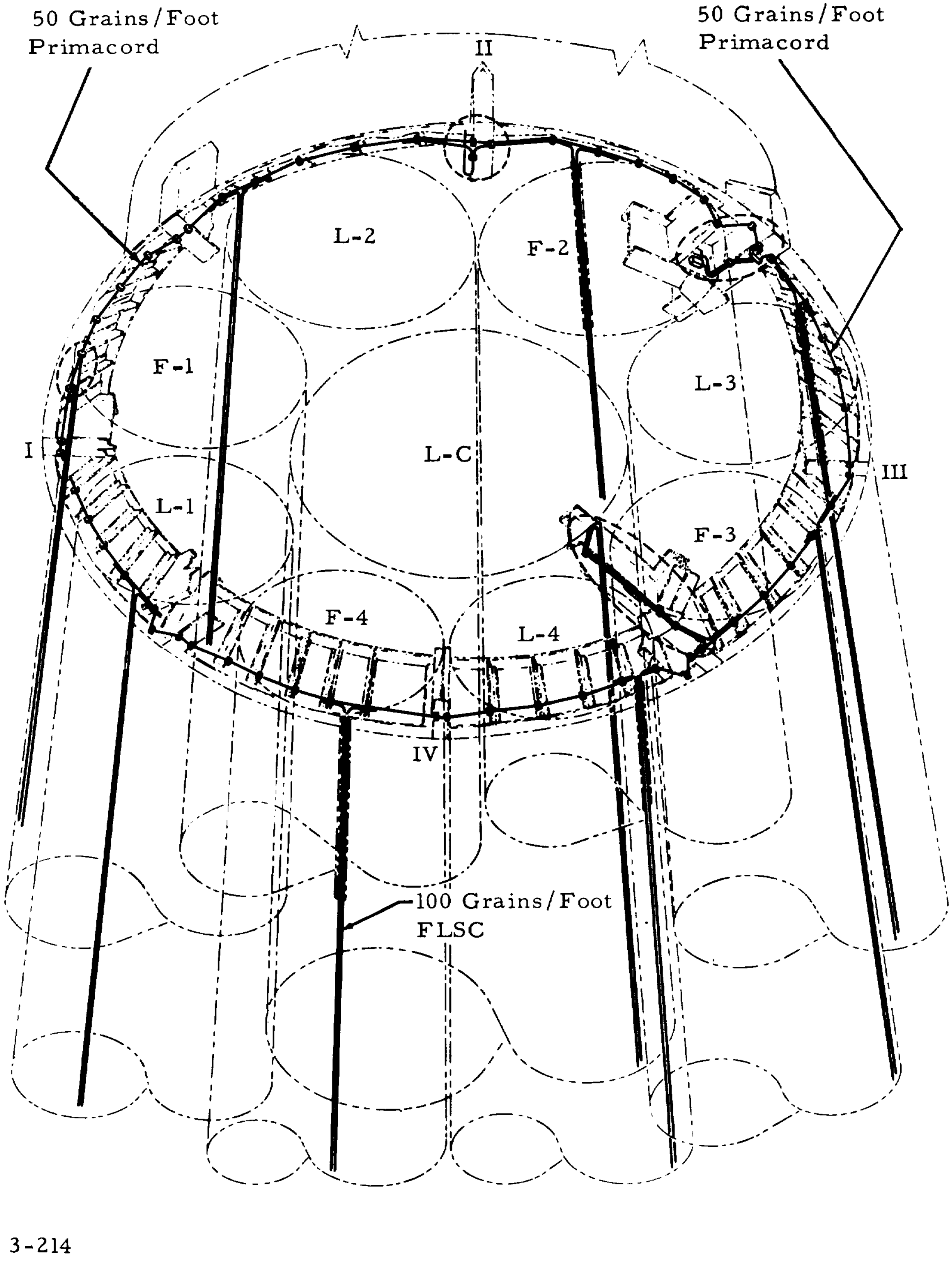

Figure 9-19. Primacord and FLSC Installation, S-I

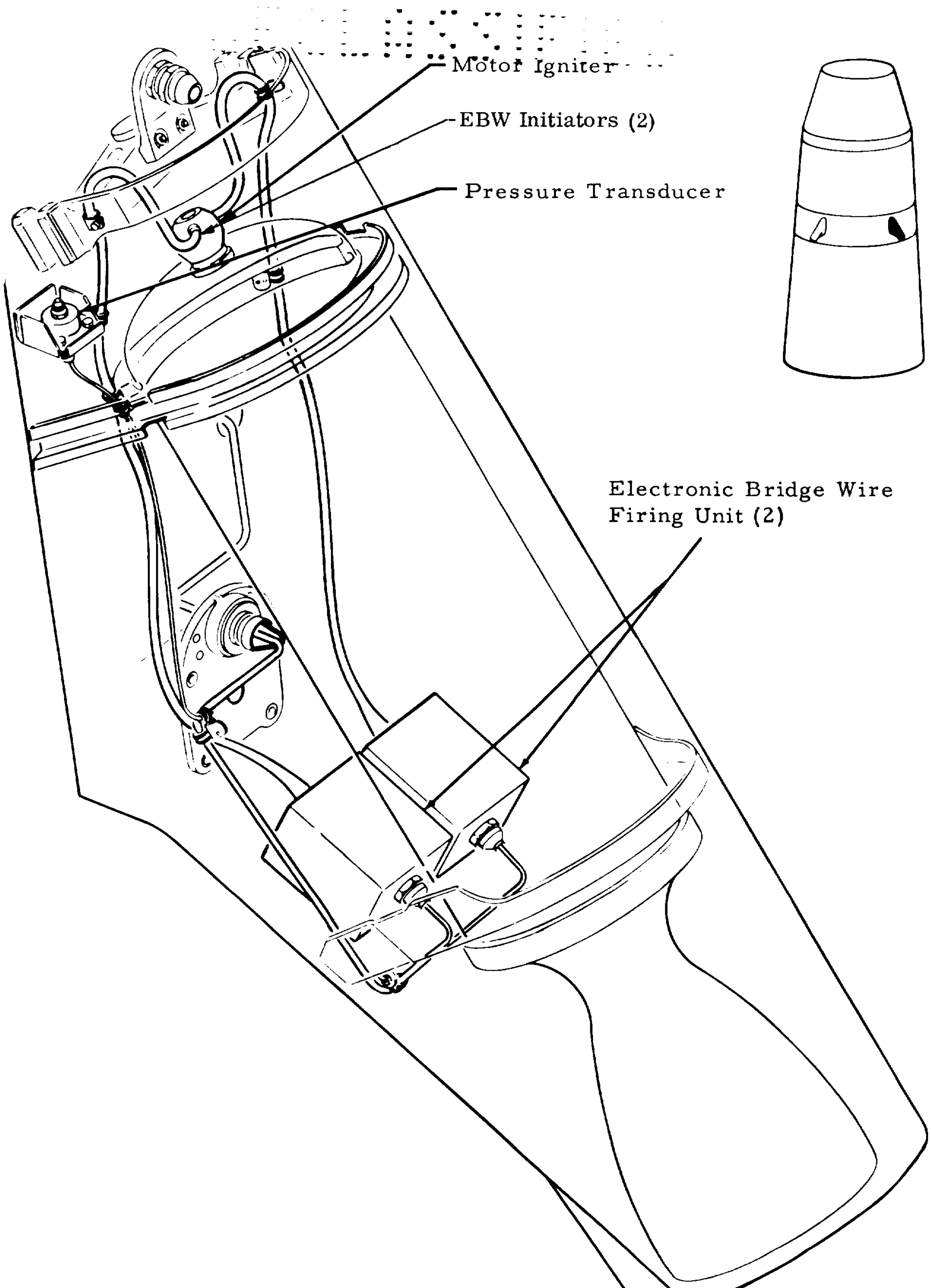

Figure 9-20. Ullage Motor Ignition System, S-IV

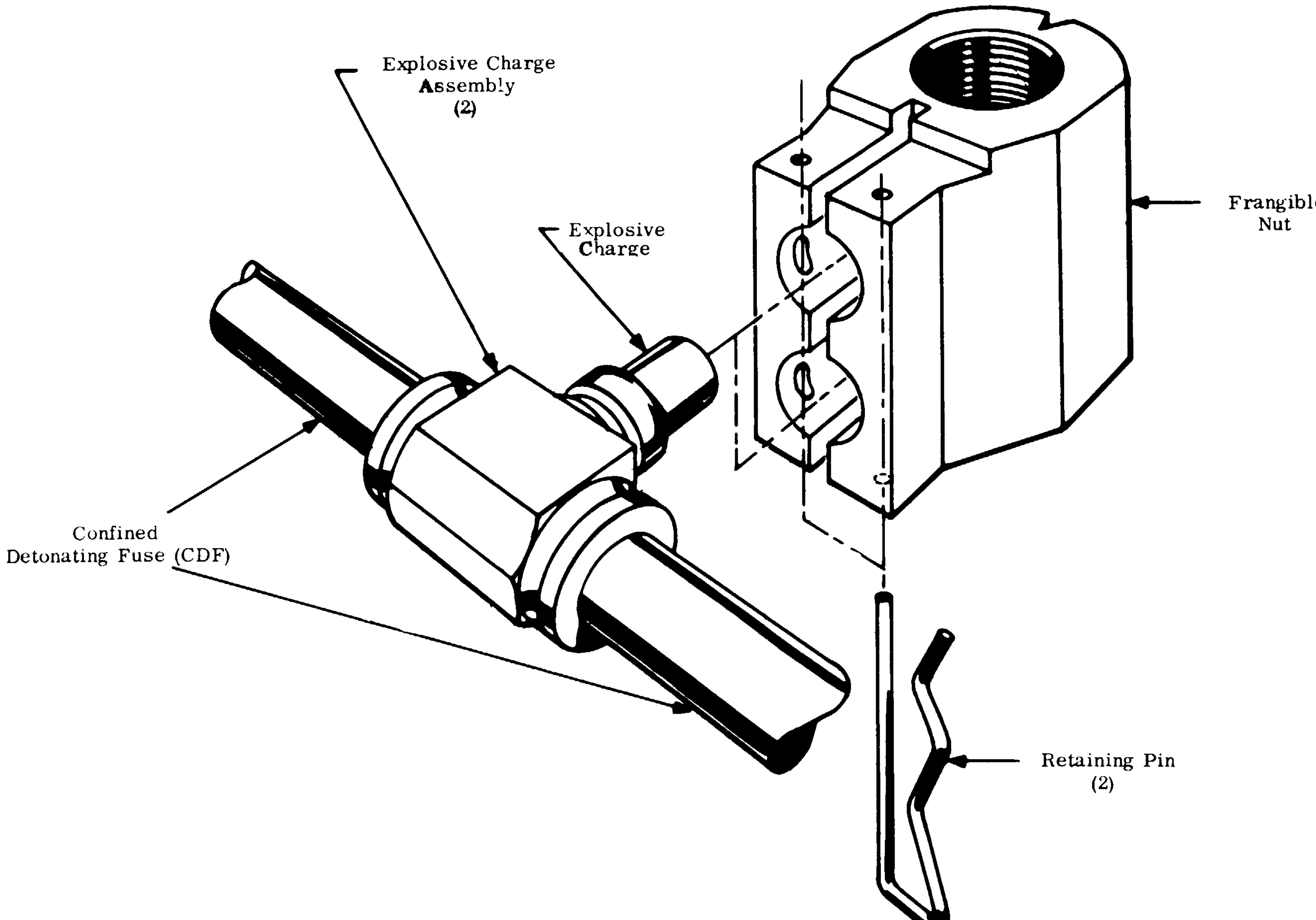

Figure 9-21. Frangible Nut and Explosive Charge Assembly

stage to the S-I stage, and to attach the ullage motor fairings to the S-IV aft skirt. Each frangible nut contains two explosive charges which when ignited, fracture the nut.

Four separation frangible nut and bolt assemblies are used to join the S-IV stage to the S-I stage at MSFC station 1147. During S-I/S-IV staging, the frangible nuts are broken by means of the two internally installed explosive charges. Each charge is detonated by a mild detonating fuse (MDF) train connected to a detonator block. Two independent electronic bridge wire firing units connected to two EBW detonators installed in the detonator block ignite the MDF train when triggered by a signal from the control computer. (The ignition system is similar to the one illustrated in Figure 9-22.) After the nuts are fractured, the spring-loaded bolts retract. The thrust of the S-I retromotors and the S-IV ullage motors provides the ultimate separation of the S-I stage from the S-IV stage. Each ullage motor is mounted in a fairing which is bolted to the aft skirt of the S-IV stage at two points by frangible nuts. The four fairings that contain the spent ullage motors are jettisoned by breaking the frangible nuts 20 seconds after the separation command is initiated. Upon receipt of a signal from the flight sequencer the electronic bridge wire firing units apply 2300 $\pm$ 100 volts dc to detonators installed in the detonator block. The block distributes the charge to the MDF harness which ignites the explosive charges in each frangible nut. A compression spring located between each fairing and the vehicle skin provides the required thrust to jettison each ullage motor/fairing unit. The jettison system is illustrated in Figure 9-22.

9-30. Retromotors. Retromotors are not required on the S-IV stage for the separation of the S-IV stage/instrument unit from the payload. However, the stage is designed with a capability for the inclusion of two TX-280 solid-propellant retromotors.

9-31. Blowout Panels. Eight blowout panels are evenly spaced around the aft end of the S-I/S-IV interstage, Figure 11-2. The panels cover triangular vent ports which are opened to vent LOX from the interstage area at the beginning of the prestart sequence for the RL10A-3 engines. The panels are removable for servicing and maintenance of equipment. Upon initiation of the prestart chilldown process, a five grain mild detonating fuse (MDF) is detonated cutting the fabric

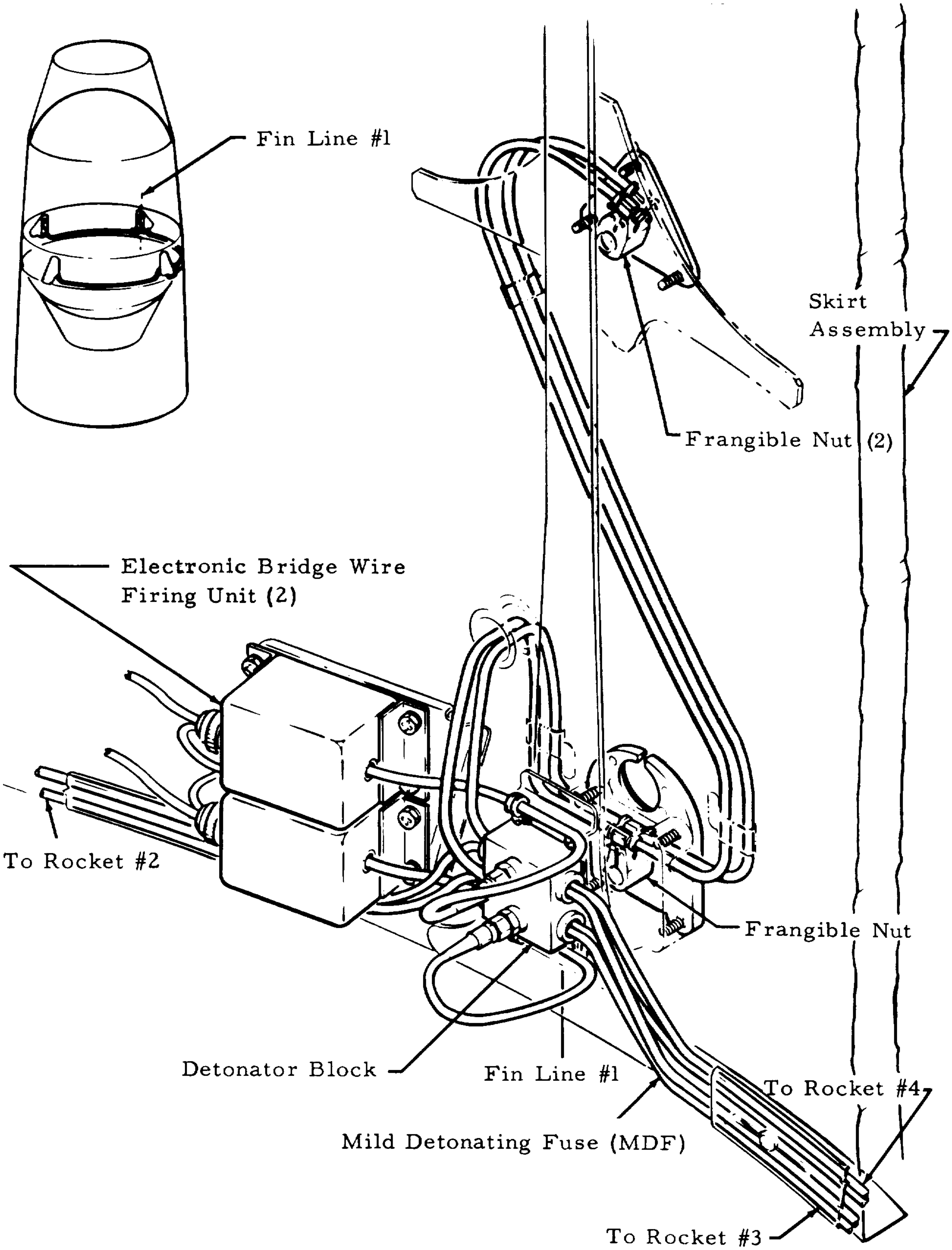

Figure 9-22. Ullage Motor Jettison System, S-IV

panels to open the vent ports. The detonating fuse or cord is connected to two EBW detonators in a detonator block. Two electronic bridge wire firing units trigger the system. (A basic firing unit is illustrated in Figure 9-16.) A 28-volt battery, located in the S-IV stage, supplies power for the system.

9-32. Propellant Dispersion System Ordnance. The propellant dispersion system ordnance for the S-IV stage consists of two electronic bridge wire firing units, a safety and arming (S&A) device, a 60 grains per foot Primacord lead, and a 100 grains per foot linear shaped charge (LSC). The firing units and detonators, and the S&A device are similar to those used on the S-I stage (Refer to Paragraph 9-26). Two strands of 100 grains per foot LSC are installed approximately 1/2-inch on center, longitudinally along the outside of the LH_2 container. The LSC is ignited by a 60 grains per foot Primacord lead extending from the S&A device. The LOX container is ruptured by cutting out a portion of the bottom bulkhead with 100 grains per foot LSC which is interconnected to the LH_2 container LSC by 60 grains per foot Primacord.

9-33. PLATFORM GAS-BEARING SUPPLY SYSTEM.

The Saturn I platform gas-bearing supply system furnishes filtered GN_2 at a regulated pressure, temperature, and flow rate to the gas bearings of the ST-124-M stabilized platform. The GN_2 is supplied to the stabilized platform from the start of checkout during prelaunch until separation of the S-IV stage and instrument unit from the Apollo payload during the orbital phase of the mission.

9-34. OPERATION.

The gas-bearing supply system receives GN_2 from a ground source during prelaunch and launch until liftoff. During the ascent and orbital phase of the mission until payload separation a high-pressure sphere which is charged during prelaunch and launch supplies GN_2 to the system. If the supply pressure falls below the minimum required for safe operation of the ST-124-M platform during standby operation, a pressure switch actuates to shut down the stabilized platform.

9-35. IMPLEMENTATION.

The gas-bearing supply system, Figure 9-23, is composed of a high-pressure storage sphere, a regulator and heater assembly (containing a solenoid valve,

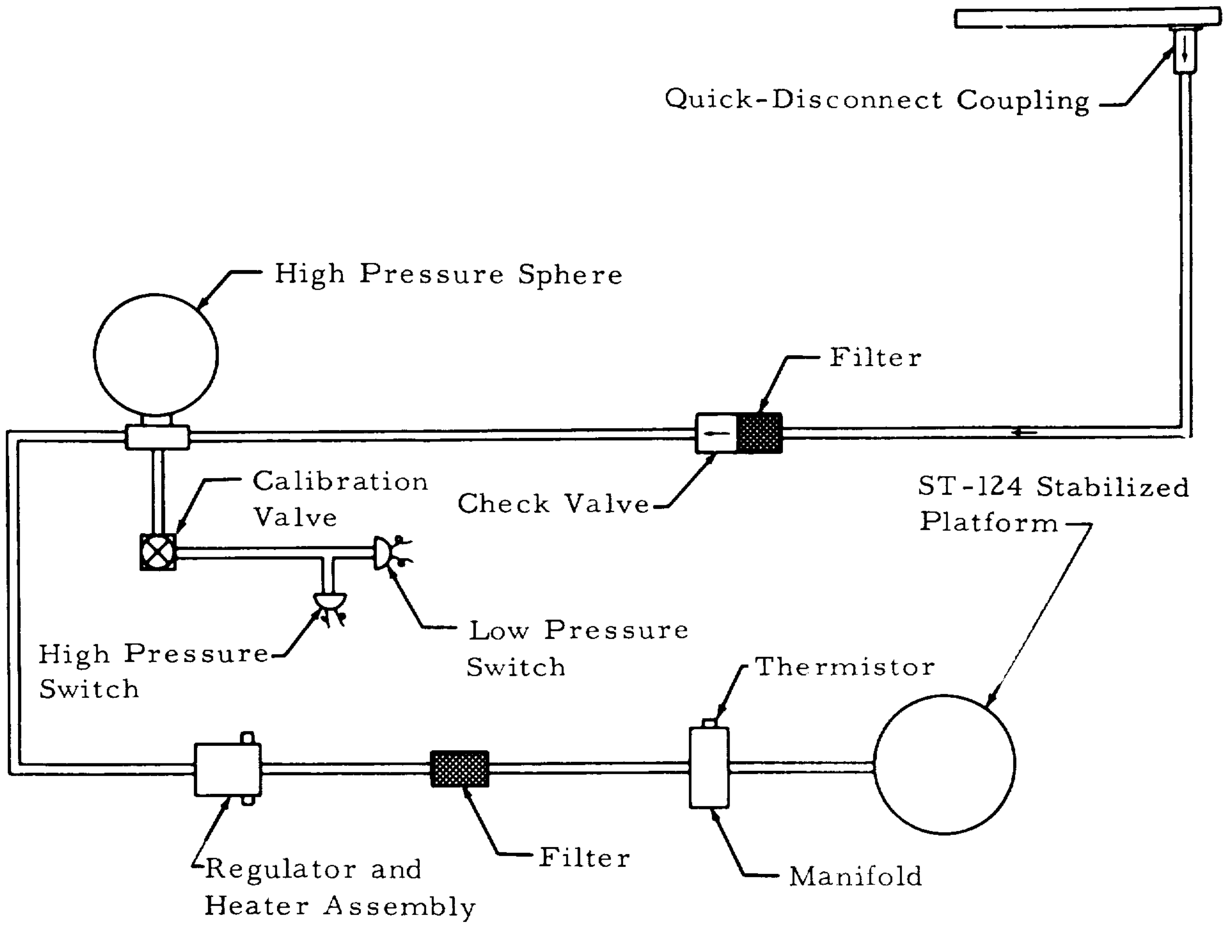

Figure 9-23. Platform Gas-Bearing Supply System

a bypass orifice, a heater and filters), a check valve, pressure switches, and associated tubing. The system is mounted adjacent to the ST-124-M stabilized platform on the inside of the instrument unit structure, Figure 11-3.

The GN_2 is supplied from a ground source at approximately 3000 psig through a quick-disconnect coupling, filter and check valve to a high-pressure sphere where it is stored until needed. Two pressure switches (high-and low-pressure), pneumatically connected to a calibration valve, are used in indicating when the supply pressure within the sphere is within the operating range.

High-pressure GN_2 flows from the high-pressure storage sphere through a regulator and heater assembly, where it is reduced from 3000 psig to operating pressure, and heated to the required temperature. (The regulator and heater assembly contains a filter, a solenoid shutoff valve and a bypass orifice.) The GN_2 then flows through a manifold assembly to the stabilized platform. The manifold assembly contains a filter and a thermistor which monitors the gas temperature.

If GN_2 pressure within the storage sphere decreases below 1200 psig during stand-by operation, the low-pressure switch actuates and initiates the removal of electrical power to the stabilized platform. In addition, the switch removes power from the solenoid shutoff valve located within the regulator and heater assembly. The GN_2 then bypasses the shutoff valve and flows through the bypass orifice at a reduced rate to allow safe bearing runout as the speeds of the platform gyros decay.

CHAPTER 2

SECTION X

GROUND SUPPORT EQUIPMENT

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X

SECTION X.

GROUND SUPPORT EQUIPMENT

10-1. GENERAL

The Saturn I ground support equipment (GSE) includes all of the ground equipment required to support the fabrication, checkout, transportation, static testing, and launch operations related to the S-I stage, S-IV stage and instrument unit. The GSE in this section excludes launch-peculiar GSE which is described in Volume I. In supporting the above operations, the GSE is formed into functional ground system, subsystem, and unit configurations. The various configurations are employed as required at all locations involved in the research and development of the vehicle and its stages. Since the operation of each configuration may vary depending on the location where used, an operational description is not contained in this document. Instead, the major GSE is listed and primary functions described.

10-2. GROUND SUPPORT EQUIPMENT, S-I STAGE.

In general, the S-I stage GSE is classified as test, checkout and monitoring; servicing; handling; and transportation. Tables 10-1 through 10-4 list the equipment and functions of each classification.

Table 10-1. Test, Checkout, and Monitoring Equipment, S-I

Equipment	Function
Instrumentation Equipment	Consists of pressure gages and panels used for transducer checkout and calibration.
Safety Monitor and Action Equipment	a. Used when the S-I stage is undergoing tests and during prelaunch operations. b. Provides shutdown capability in the event that a dangerous condition develops.
Central Control Equipment	Provides a central control console for use during checkout and launch having a capability of directing the program to start, stop, or hold any system test sequence.

Table 10-1. Test, Checkout, and Monitoring Equipment, S-I (Cont'd)

Equipment	Function
Stage Propulsion Equipment	Used to energize, control, monitor and test the electrical components associated with the stage electrical power supplies, pneumatic systems, and pyrotechnics, and the electro-mechanical components associated with the propellant containers and rocket engines.
Ground Power System	a. Supplies electrical power (28-volt dc, 115/208-volt, 400 cps) to ground support equipment. b. Used to control and monitor the electrical power that is applied from other power sources to S-I stage components and test site systems during test, checkout and static firing.
Ground Equipment Test Set (GETS)	Used to validate the operation of electrical circuits of ground support equipment prior to mating the S-I stage and ground support equipment.
Ground Support Equipment Testing	Used in vehicle component and subsystem verification testing of propellant system and engine heaters, hydraulic control system, cooling system, stage destruct firing circuits, engine Conax valve firing circuits and instrument canisters.
Ground Telemetry Station	a. Used to test the S-I stage telemetry system. b. Used to check the operation of various transducers in the instrumentation system.
Upper Stage Simulator	a. Presents the proper impedances to circuitry which normally terminates in an upper stage. b. Contains equipment with test point facilities for use in troubleshooting and for insertion of stimuli.
S-I Stage Simulator	a. Used to checkout ground support equipment. b. Presents the proper impedances and sufficient typical stage outputs to establish confidence in the ground support equipment.

Table 10-1. Test, Checkout, and Monitoring Equipment, S-I (Cont'd)

Equipment	Function
	c. Contains equipment with test point facilities for use in troubleshooting and for insertion of stimuli.
Fuel Tanking Simulator	Supplies calibration signals.
Fuel Density Simulator	Supplies calibration signals to the fuel density monitor panel.
Liquid Oxygen Tanking Simulator	Supplies calibration signals to the liquid oxygen tanking control panel.
Engine Simulator	a. Simulates electrical network of the engine and verifies operation of the ground support equipment. b. Simulates the electrical responses of an engine during stage testing.
Radio Frequency (RF) Test Bench	Provides a central source of equipment and power to calibrate, troubleshoot, and repair RF equipment of the S-I stage and ground support equipment.

Table 10-2. Servicing Equipment, S-I

Equipment	Function
RP-1 Fuel Filling	Controls the transfer of RP-1 from the facility storage tanks to the S-I stage fuel containers.
Fuel Replenishing	Provides the control for adjusting and loading fuel weight to the S-I stage.
Liquid Oxygen Filling	Controls the transfer of LOX from the storage tanks to the S-I stage LOX containers.
Liquid Oxygen Replenishing	Provides the LOX replenishing to compensate for boiloff.
Pneumatic Control System	Supplies GN_2 and helium from the storage facility to stage. The GN_2 and helium are used for stage pressurization and purging, LOX

Table 10-2. Servicing Equipment, S-I (Cont'd)

Equipment	Function
	and fuel bubbling, and fuel container pre-pressurization. In addition, the gases are used to support the operation of the launcher and tower equipment, and pneumatically controlled devices in the stage and test complex.
Environmental Control System	a. Supplies humidity and temperature controlled air or GN_2 to the S-I stage and test complex. b. Supplies air conditioning for S-I stage and provides inert gas for purging stage compartments.
Hydraulic Servicer System	Supplies the S-I stage with hydraulic fluid used for cleaning and checkout operations of the engine gimbal system.

Table 10-3. Handling Equipment, S-I

Equipment	Function
Stage Handling Equipment	Used for handling and loading the S-I stage, assemblies, components, and certain items of ground support equipment. The equipment consists of a set of slings and handling rings.
Engine Handling Equipment	Used on S-I stage to support the installation, removal, servicing, and maintenance of an H-1 engine.

Table 10-4. Transportation Equipment, S-I

Equipment	Function
Transporter	Used for horizontal support and transportation of the assembled S-I stage during all phases of factory and field operations.
Transporter Dolly	Composed of a frame and running gear assembly, towbar, steering and braking system, and operator controls. (A fore and aft transporter dolly connected by a structural frame forms a complete transporter.)

Table 10-4. Transportation Equipment, S-I (Cont'd)

Equipment	Function
Transportation Accessories Kit	a. Provides the equipment required to prepare the stage for transportation, protect small parts during transportation, and to tie down, block, and shore the stage transporter on the barge. b. Includes environmental control equipment which controls the temperature and humidity of environmental sensitive items (such as those of instrumentation), during extended barge transportation.

10-3. GROUND SUPPORT EQUIPMENT, S-IV STAGE.

The S-IV stage GSE is classified as test, checkout and monitoring; transportation, protection and handling; stage subsystem testing; instrumentation; and propellant and gas servicing. Tables 10-5 through 10-9 list the equipment and functions of each classification.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV

Figure	Equipment	Function
10-1 (Sheet 1)	Ground Support Equipment Test Set	a. Electrically simulates components and circuits of the S-IV stage to verify proper operation of the GSE. b. Allows operation of the GSE and the stage functions without the stage being present.
10-1 (Sheet 1)	S-IV Stage Substitute	a. Simulates components and circuits of the stage in order that associate contractors can check out adjacent stages when the S-IV stage is not available.
10-1 (Sheet 1)	Stage Power Control and Monitor Panel	a. Provides remote control facilities for transferring ground power between the generator room and the stage, and between the generator room and the ground equipment.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

Figure	Equipment	Function
		b. Used for monitoring facilities and power supply buses, vehicle dc buses, ground and stage 400-cycle power, battery temperatures, and emergency battery and inverter outputs.
10-1 (Sheet 2)	Instrumentation Power Control and Monitor Panel	a. Provides remote control facilities for transferring ground power from the utility room to the facilities equipment. b. Used to monitor the external 28-volt dc bus, and ground 5-volt dc bus.
10-1 (Sheet 2)	Propulsion System Preparation and Control Panel	Used to control and monitor the control helium pressure; monitor LH_2 and LOX container ullage pressures; energize prestart, start, and helium heater valves; energize engine and helium heater igniter components; and indicate propulsion system status.
10-1 (Sheet 2)	Stage Power Control and Monitor Chassis	Used to control external electrical power distribution to the S-IV stage.
10-1 (Sheet 2)	Propulsion System Preparation and Control Chassis	Used to form the terminal electrical switching for the propulsion system preparation and control panel.
10-1 (Sheet 3)	Hydraulic Control and Monitor Panel	a. Provides control for the stage electric auxiliary pump motors and accumulator valves. b. Used to monitor hydraulic fluid levels, accumulator pressures, and fluid temperatures.
10-1 (Sheet 3)	Hydraulic System Control Chassis	Provides the control circuit that controls the stage hydraulic system, and monitoring functions for the control circuitry.
10-1 (Sheet 3)	Gimbal Control Panel	a. Provides the slewing controls for single or multiple engines.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

Figure	Equipment	Function
		b. Displays (on panel-mounted meters) the slew command and direction for the yaw, pitch, and roll planes.
10-1 (Sheet 3)	Gimbal Monitor Panel	Provides the indicators used for monitoring hydraulic valve excitation unbalance, and for monitoring each engine position during testing.
10-1 (Sheet 3)	Flight Sequence Control Panel	Tests the propulsion system logic circuits by controlling inputs supplied to the logic circuits from an external programmer, and monitoring outputs of the propulsion system logic circuits.
10-1 (Sheet 3)	Flight Sequence Control Chassis Nos. 1 and 2 (Typical)	Contain the logic circuits used with the flight sequence control panel for monitoring inputs from the S-IV stage propulsion system logic circuits, command circuit (S-IV stage prestart), and talkbacks from the stages.
10-1 (Sheet 4)	Propellant Utilization Checkout and Control Panel	a. Provides the controls and indicators used for partial checkout of the S-IV stage closed loop propellant utilization system.
		b. Contains the control panel indicators used to monitor positions of mixture-ratio valves, and the operation of the propellant utilization sequence switch.
10-1 (Sheet 4)	Pneumatic System Control Panel	a. Provides facilities for manual and remote control of the stage cold helium loading, propellant container pressurization, engine section purge, and nozzle purge.
		b. Contains remote temperature and pressure indicators, and the controls used to check out pneumatic consoles A and B and the helium precool heat exchanger.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

Figure	Equipment	Function
10-1 (Sheet 4)	Propulsion System Test Set, Launch Complex	Used in monitoring and testing the S-IV stage propulsion system and pneumatic consoles A and B.
10-1 (Sheet 4)	Propulsion System Test Set, Hangar	Used to test and monitor the stage propulsion system while the stage is in the hangar.
10-1 (Sheet 5)	Flight Sequence Recorder Chassis	Provides the hard-wire recorder used to record engine sequence and other pertinent flight sequence events.
10-1 (Sheet 5)	Recorder Isolation Amplifier Chassis	Amplifies low-level electrical signals that originate in the instrumentation isolation circuits.
10-1 (Sheet 5)	Recorder System Test Panel	a. Used to test the flight sequence recorder chassis and the recorder isolation amplifier chassis. b. Supplies signals to other GSE items that indicate when specific channels are activated.
10-1 (Sheet 5)	Propellant Loading Control and Monitor Panel	Used to control solenoid-actuated control valves in the LOX and LH_2 fill and topping control systems for loading propellants into the S-IV stage.
10-1 (Sheet 6)	Propellant Loading Computer Control Panel	Used to control the fuel and oxidizer loading computer, and the propellant loading computers.
10-1 (Sheet 6)	Fuel Loading Computer Chassis	Controls the propellant valves used for attaining and maintaining the fuel at a predetermined mass level.
10-1 (Sheet 6)	Fuel Loading Computer Relay Chassis	a. Contains the circuitry used for for computer checkout. b. Uses the fuel loading computer signals to control and maintain the correct amount of LH_2 for a given mission.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

Figure	Equipment	Function
10-1 (Sheet 6)	LOX Loading Computer Chassis	Controls the propellant valves used for attaining and maintaining the oxidizer at a predetermined mass level.
10-1 (Sheet 7)	LOX Loading Computer Relay Chassis	a. Contains the circuitry used for computer checkout.
		b. Uses the oxidizer loading computer signals to control and maintain the correct amount of LOX for a given mission.
10-1 (Sheet 7)	Test Conductor Monitor Panel	Uses lamps to indicate the readiness of the S-IV stage for specific use of the test conductor.
10-1 (Sheet 7)	Hangar Umbilicals Junction	a. Contains relays and contactors for operation of solenoids, valves, and relays in the stage, and for disconnecting all electrical connections between the stage and the GSE.
		b. Provides a convenient point for troubleshooting the umbilicals and the GSE.
10-1 (Sheet 8)	Operational and Test Stand Checkout Pneumatic Console A-Checkout Accessories	Used to supply the S-IV stage propulsion system with helium gas at the pneumatic pressures required for loading, unloading, and purging.
10-1 (Sheet 8)	Stage System Status Panel	Used to control and monitor the automated countdown from T minus 100 seconds until launch.
10-1 (Sheet 8)	Operational and Test Stand Checkout Pneumatic Console B	a. Used to supply the stage propulsion system with GH_2 at the pressures required for loading, unloading, and purging.
		b. Used for prepressurization of the stage LH_2 containers, and for the GN_2 pressurization of the LOX and LH_2 main fill and topping control systems.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

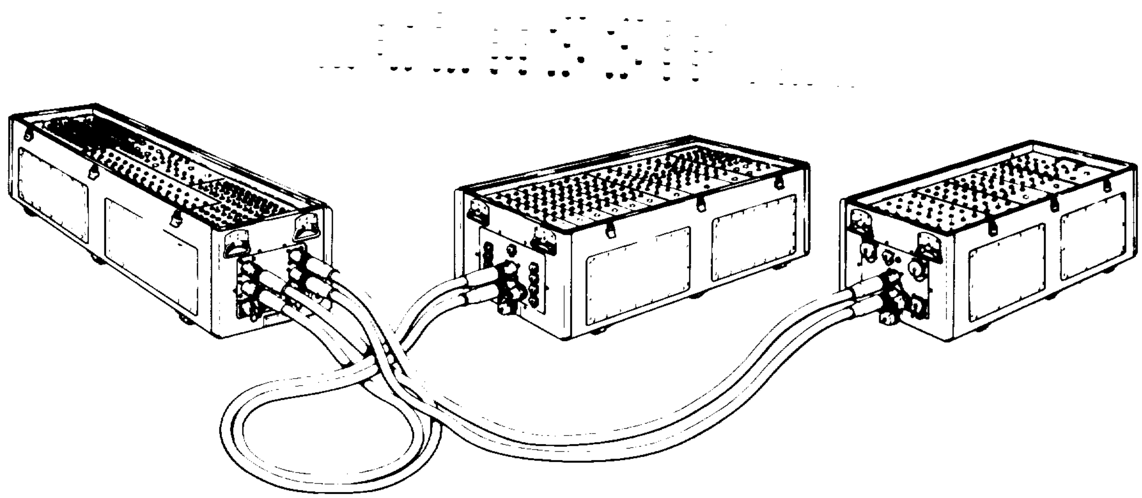
Figure	Equipment	Function
10-1 (Sheet 8)	Stage Checkout Area Pneumatic Console - Checkout Accessories	Used in the hangar to supply the pneumatic pressures used for leak and functional checkout of the S-IV stage propulsion system.
10-1 (Sheet 9)	Ordnance Monitor Panel	Used to monitor the voltage across the S-IV stage electronic bridge wire (EBW) firing unit capacitor and the response of the EBW firing unit to the trigger unit firing pulse.
10-1 (Sheet 9)	Ordnance Monitor-Control Chassis	Contains the logic circuits required for the operation of the ordnance monitor panel.
10-1 (Sheet 9)	EBW Firing Unit Test Set	a. Contains the circuitry required to test the firing unit prior to its installation in the S-IV stage. b. Used to perform quantitative checks on firing units when the units are initially received by the Douglas Aircraft Company.
10-1 (Sheet 9)	EBW Initiator Test Set	a. Used to determine if the electrical characteristics of the initiator are within tolerance. b. Used to perform quantitative checks on initiators when they are initially received by the Douglas Aircraft Company, and prior to their installation in the S-IV stage.
10-1 (Sheet 9)	EBW System Pulse Test Set	Contains the circuits used during system tests to determine the energy level output of the firing unit.
	Pressure Plug Kit	Contains the plugs used in performing propellant line leak checks.
	EBW System Checkout Power Supply	Supplies 28-volt dc power to the EBW pulse sensor during S-IV stage checkout.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

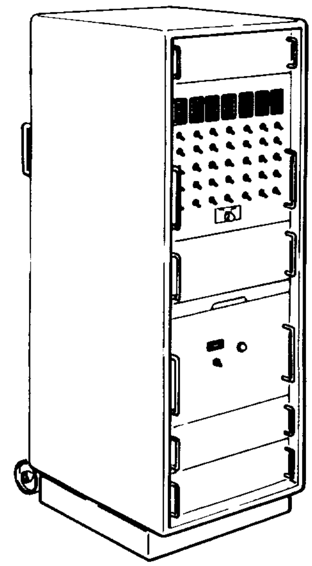
Figure	Equipment	Function
	EBW System Checkout Recorder	Records EBW system test results during S-IV stage checkout.
	EBW System Checkout Molded Junction Box	Used as a junction between the EBW system power supply and the eight EBW pulse sensors.
	Flight Sequence Monitor Chassis Nos. 2, 3, 4	Contains the circuitry used in monitoring the condition of the S-IV stage sequencer and related systems.
	EBW System Checkout Recorder Power Distribution	a. Supplies controls for the EBW system checkout recorder. b. Monitors output from the EBW system checkout pulse sensor.
	Propulsion System Maintenance Tool	Contains the tools used in the installation and checkout of the propulsion system.
	Checkout Equipment Kit	Contains the propulsion section equipment used in the checkout of the S-IV stage.
	Hangar Circuit Protection Junction Box	Provides overload protection for electrical circuits of the cable assembly.
	S-IV Explosive Initiator Test Kit	(To be supplied at a later date.)
	Checkout Accessories Kit	Contains the quick-disconnect fittings, flexible hoses, filters, fluid line fittings, thermocouple vacuum gages, and flowmeters used by the vehicle checkout area pneumatic console in performing leak and functional checkout of the S-IV stage propulsion systems and components.
	System Signal Conditioning Console	Conditions signals from the S-IV stage instrumentation for transmittal to the remote sequence recorders, panel lights, and amplifiers for monitoring meters.

Table 10-5. Test, Checkout, and Monitoring Equipment, S-IV (Cont'd)

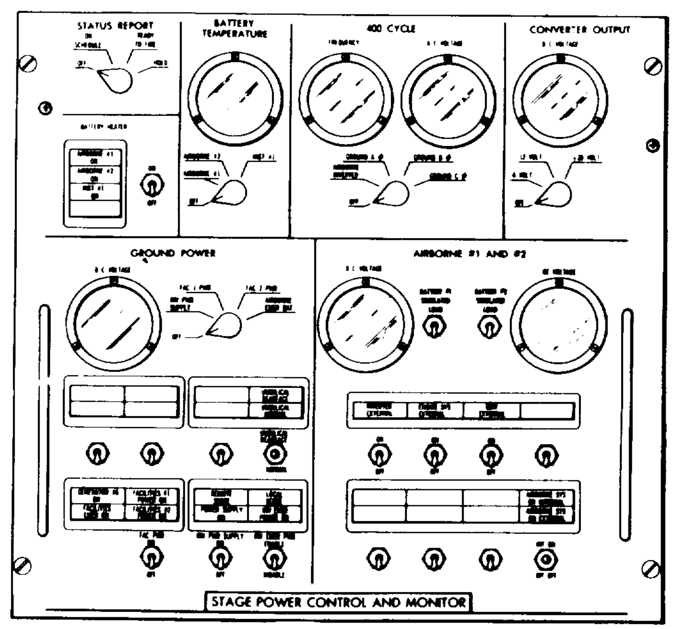
Figure	Equipment	Function
	Stand Circuit Protection Junction Box	Provides overload protection for electrical circuits of the cable assembly.
	Stage Facilities Control Chassis	Provides the circuits for use in controlling and monitoring of miscellaneous facility items.
	S-IV Engine Deflection Panel	Monitors each engine position in response to manual or programmed signal inputs during S-IV stage checkout.
	Patch Junction Box Panel	a. Contains facilities for interconnecting the electrical GSE. b. Provides the interface between GSE and the AMR blockhouse equipment. c. Provides an interface between the GSE and the automatic ground control station equipment.
	System Signal Conditioning Console	Accepts and conditions instrumentation signals from the S-IV stage for remote monitoring meters, sequence recorders, and panel lights.
	Hangar Patch Panel Junction Box.	Used to interconnect the GSE.
	Launcher Umbilical Distribution Box	Contains facilities used for troubleshooting and revising of umbilical wiring.
	Hangar Umbilicals Junction Box	Used to interconnect the S-IV stage umbilicals and the patch panel during checkout.
	EBW Initiator Simulator Assembly	Simulates EBW initiators for testing S-IV stage systems.
	Pneumatic System Control Chassis	Provides terminal switching circuits for the pneumatic system control panel.



Ground Support Equipment Test Set



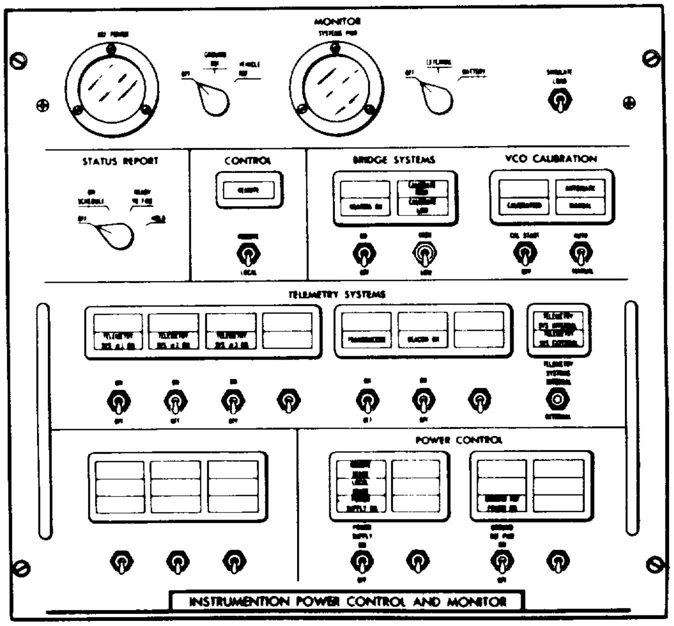
S-IV Stage Substitute



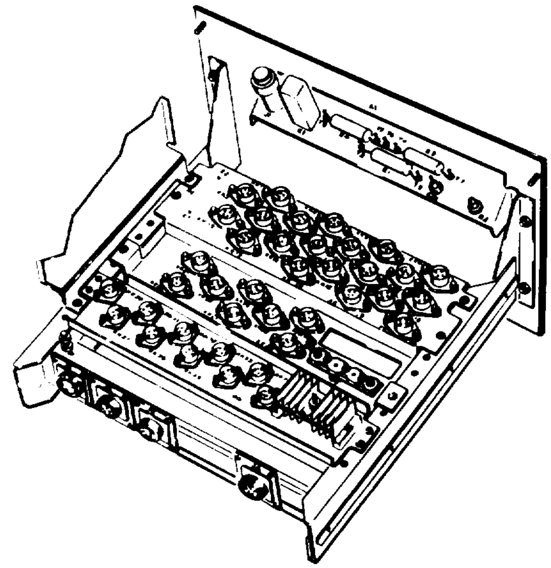
Stage Power Control and Monitor Panel

3-804

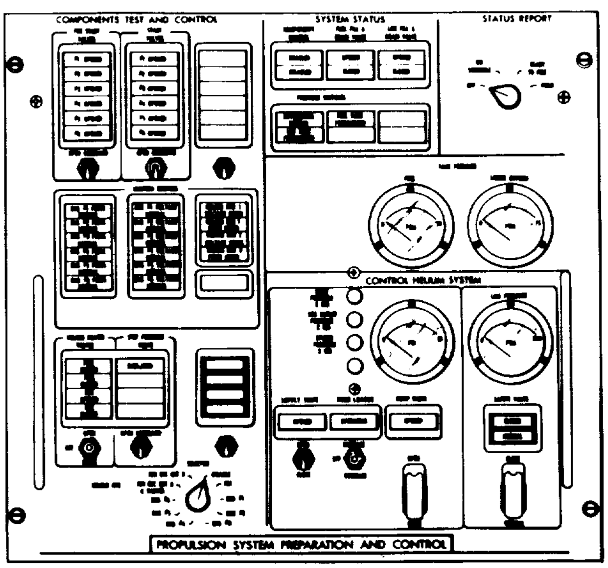
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (1 of 9)



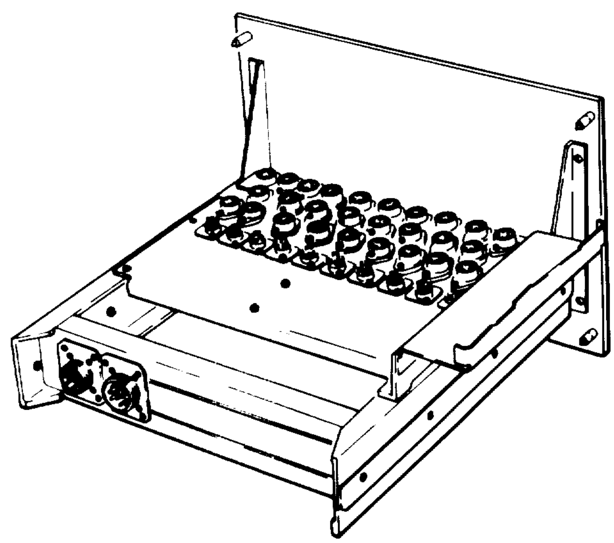
Instrumentation Power Control and Monitor Panel



Stage Power Control and Monitor Chassis



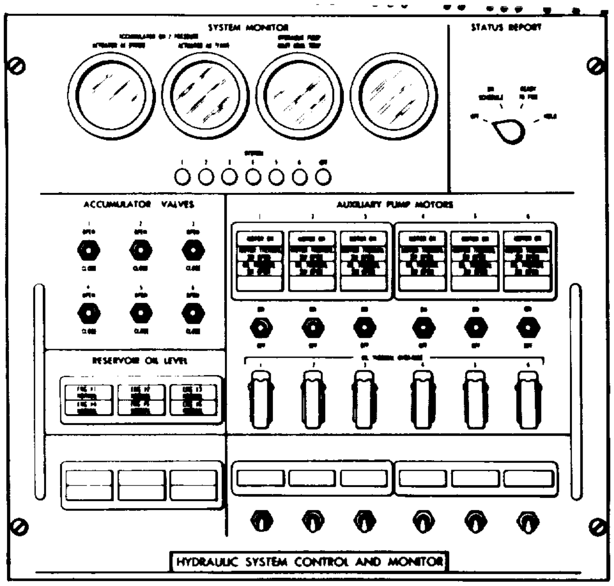
Propulsion System Preparation and Control Panel



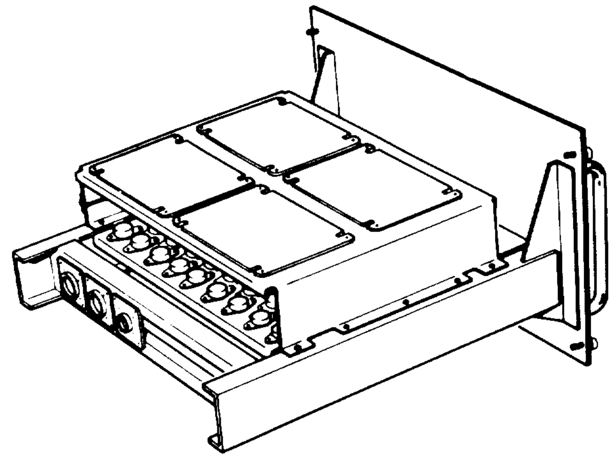
Propulsion System Preparation and Control Chassis

3-805

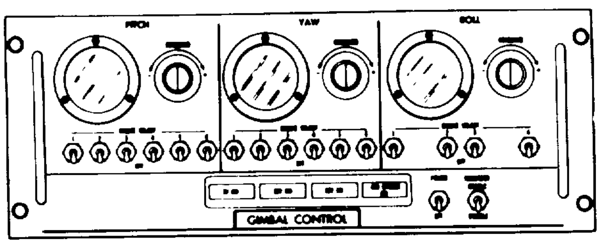
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (2 of 9)



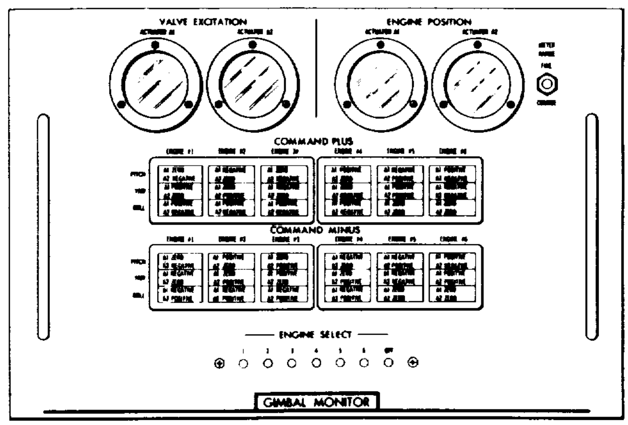
Hydraulic Control and Monitor Panel



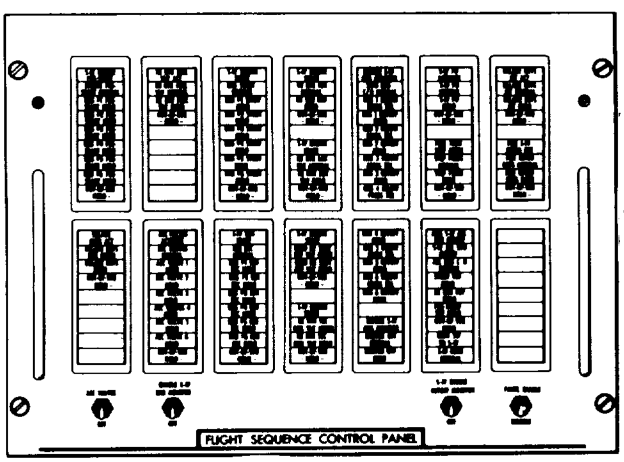
Hydraulic System Control Chassis No. 1



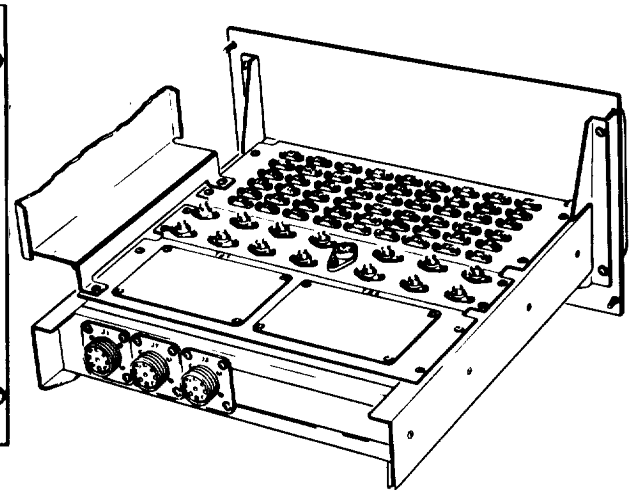
Gimbal Control Panel



Gimbal Monitor Panel



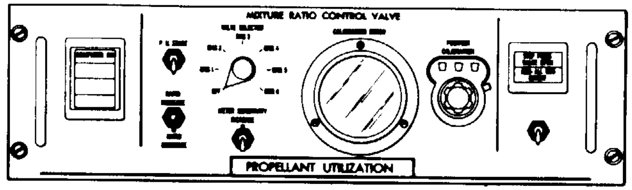
Flight Sequence Control Panel



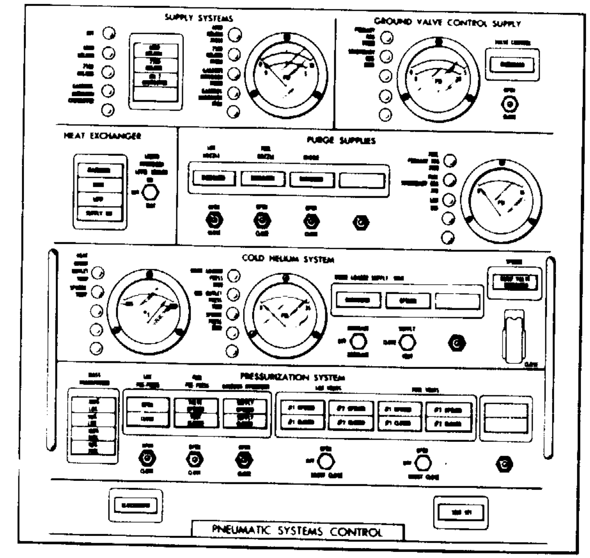
Flight Sequence Control Chassis Nos. 1 and 2 (Typical)

3-806

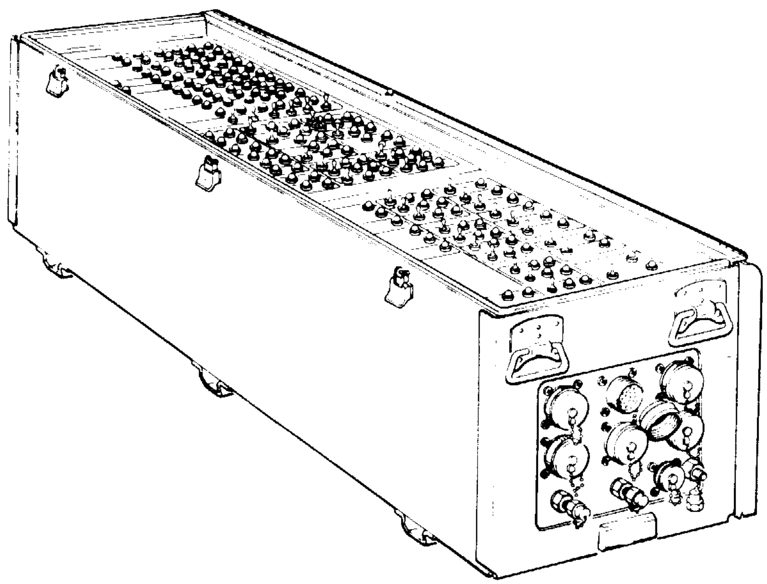
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (3 of 9)



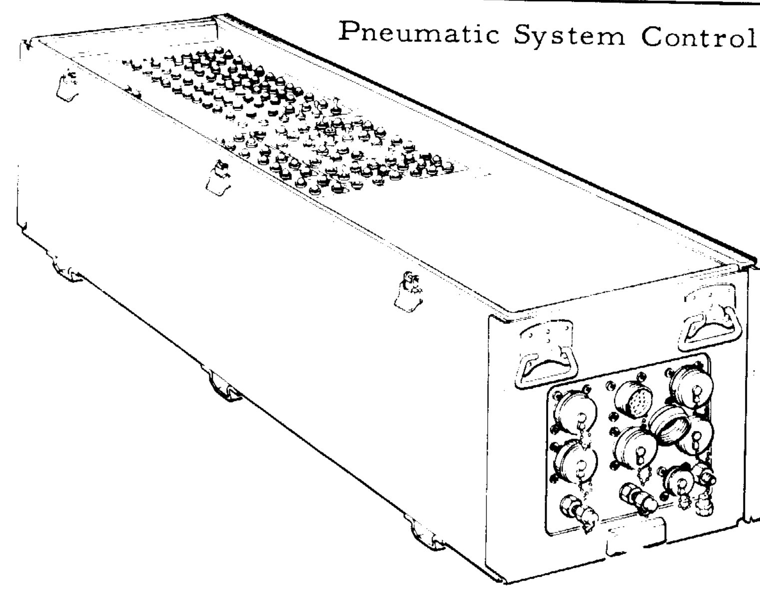
Propellant Utilization Checkout
and Control Panel



Pneumatic System Control Panel



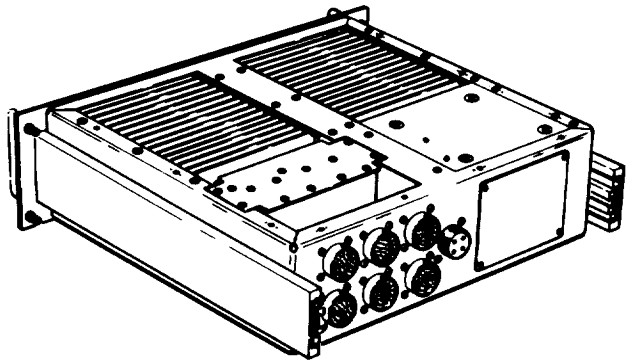
Propulsion System Test Set, Launch Complex



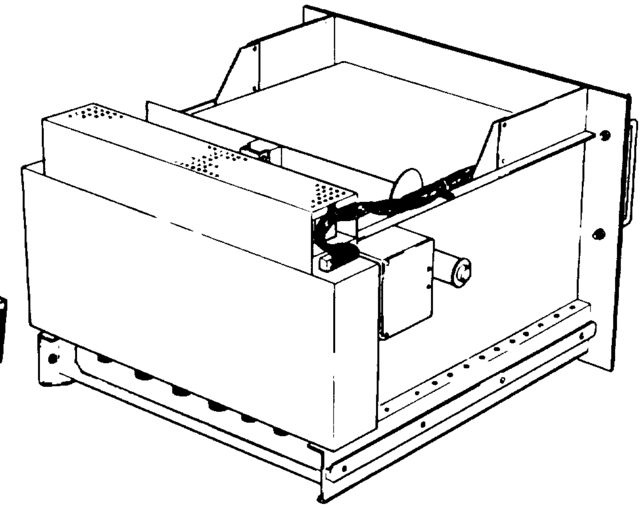
Propulsion System Test Set, Hangar

3-807

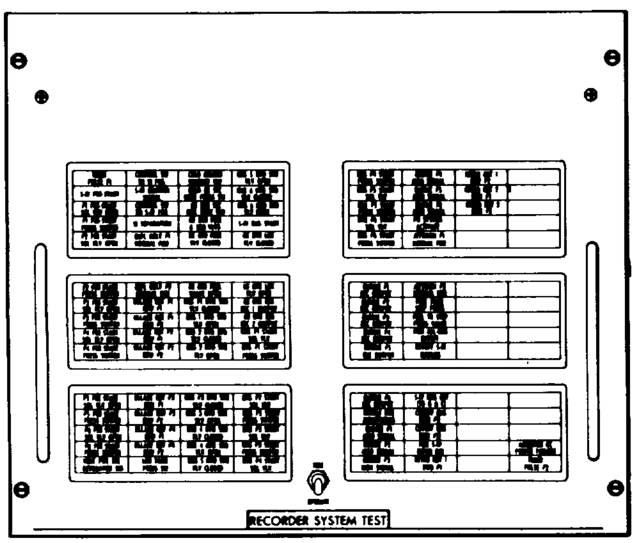
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (4 of 9)



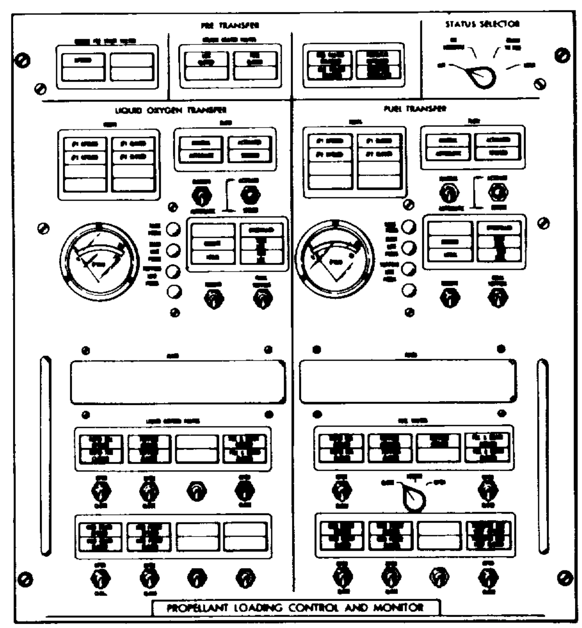
Flight Sequence Recorder Chassis



Recorder Isolation Amplifier Chassis



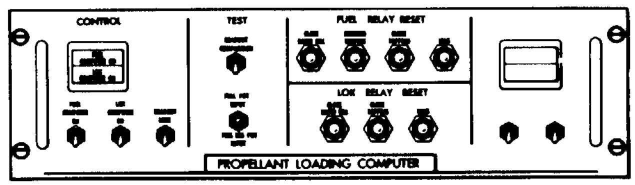
Recorder System Test Panel



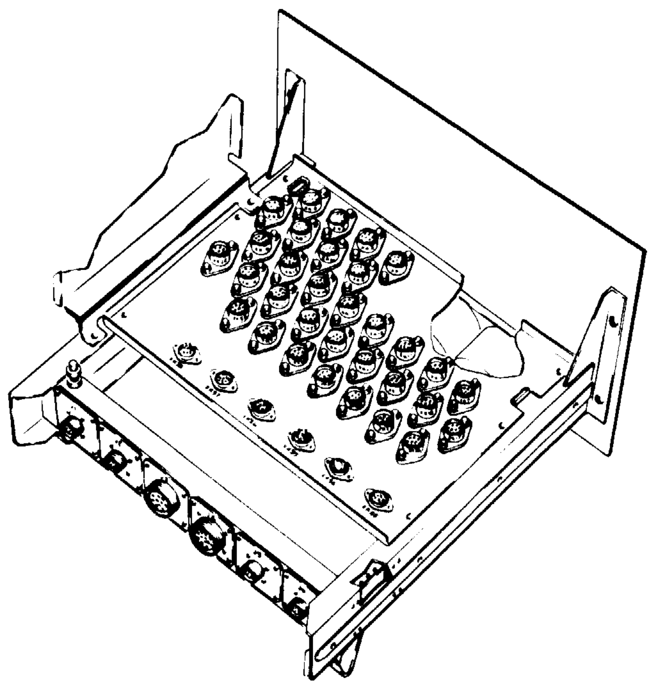
Propellant Loading Control and Monitor Panel

3-808

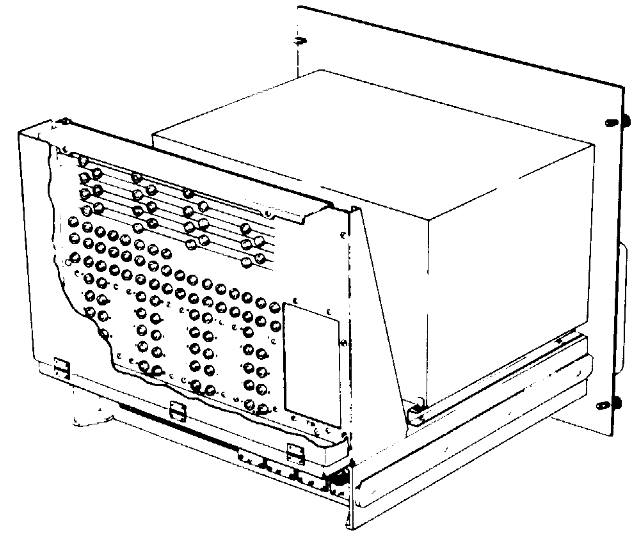
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (5 of 9)



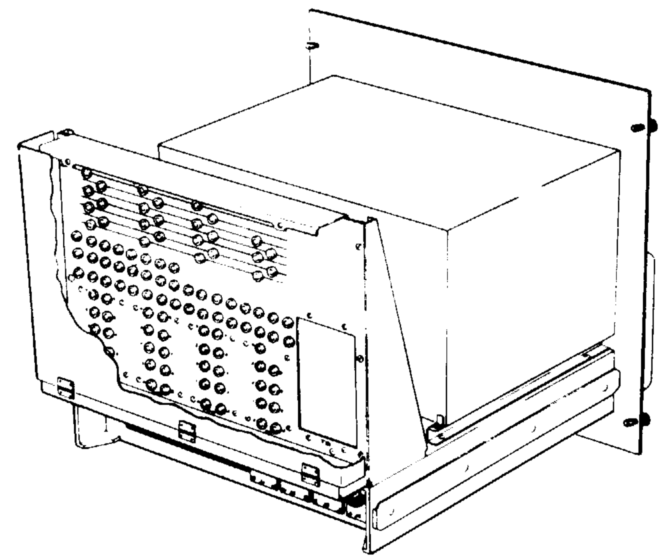
Propellant Loading Computer
Control Panel



Fuel Loading Computer
Relay Chassis



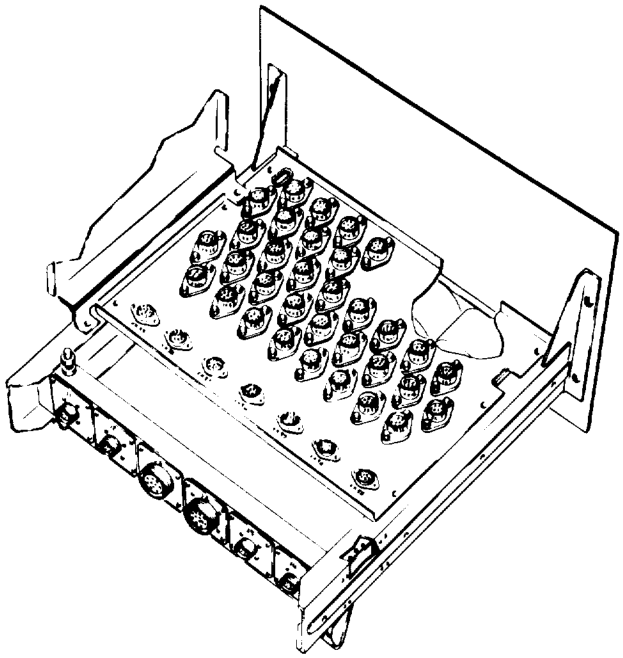
Fuel Loading Computer Chassis



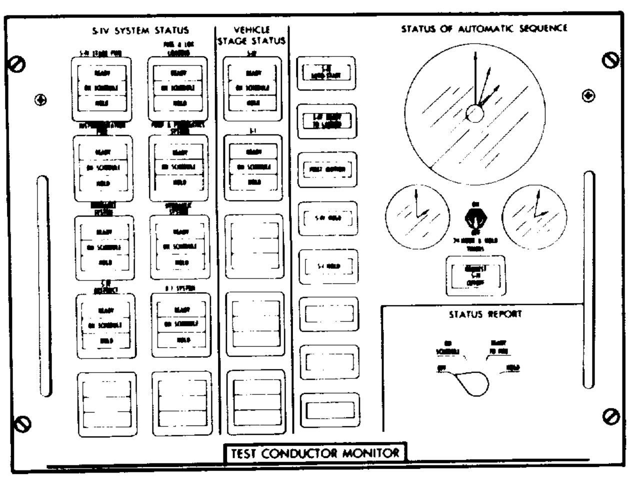
LOX Loading Computer Chassis

3-809

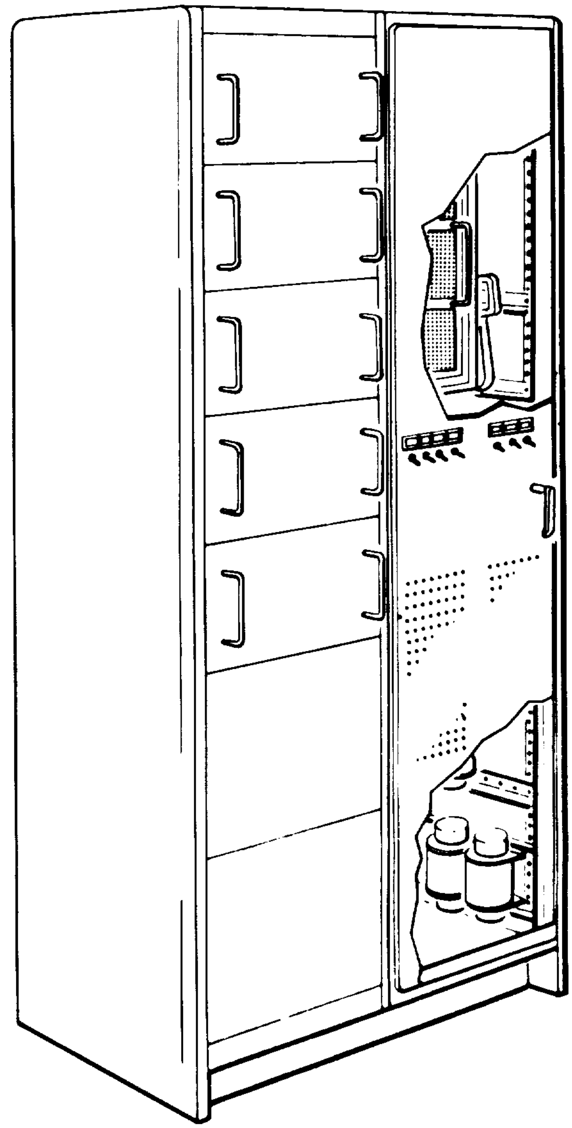
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (6 of 9)



LOX Loading Computer
Relay Chassis



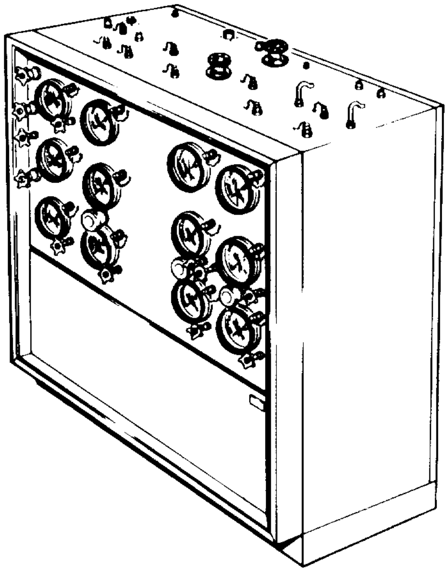
Test Conductor Monitor Panel



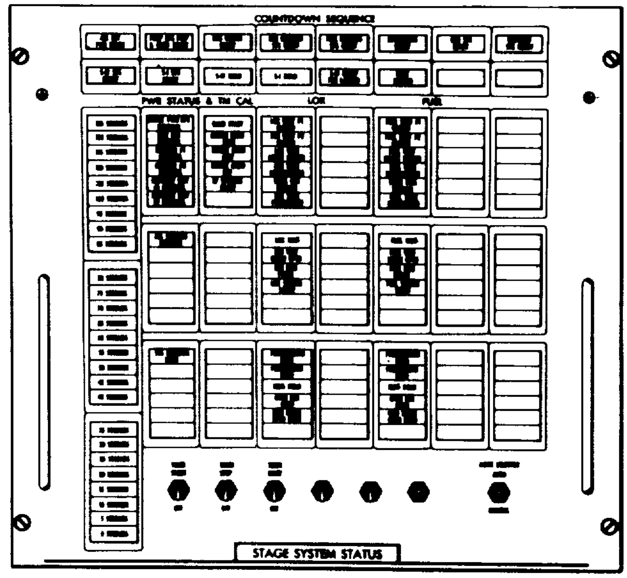
Hangar Umbilicals Junction Box

3-810

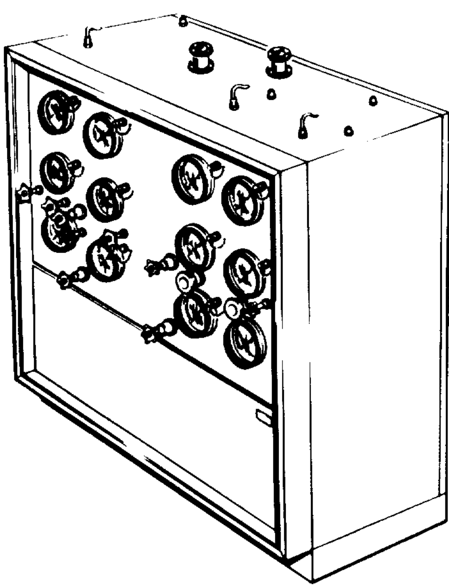
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (7 of 9)



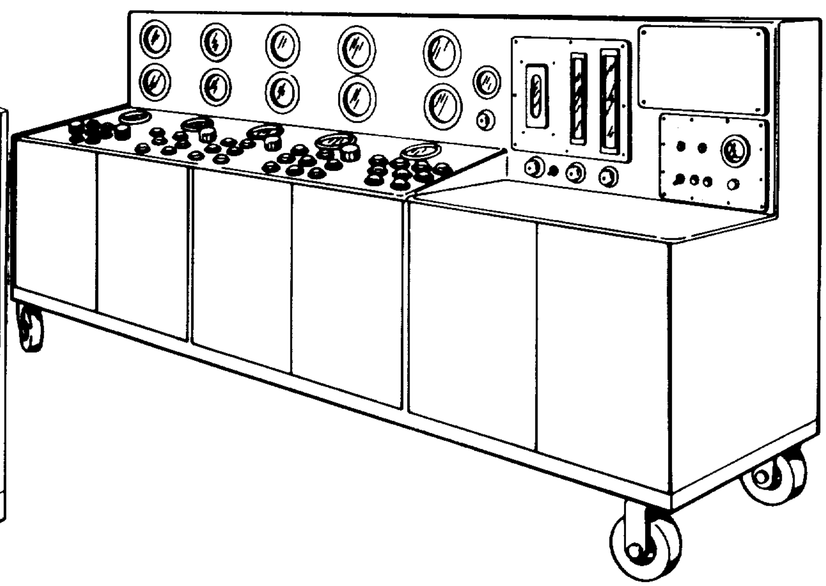
Operational and Test Stand Checkout Pneumatic Console A -Checkout Accessories



Stage Systems Status Panel



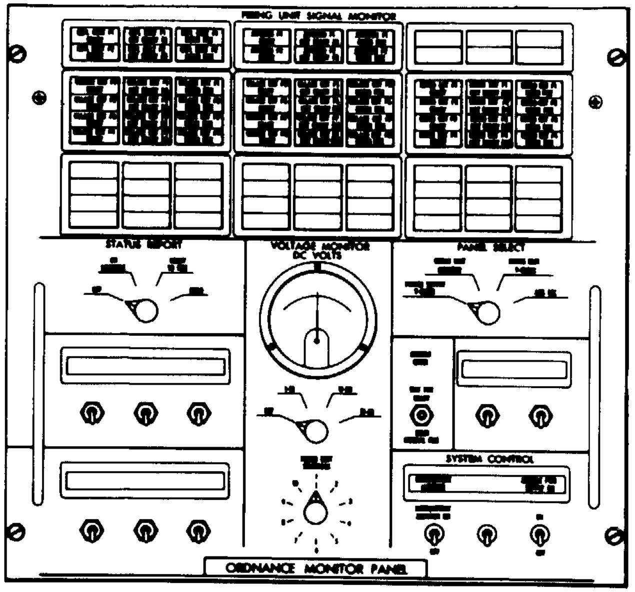
Operational and Test Stand Checkout Pneumatic Console B



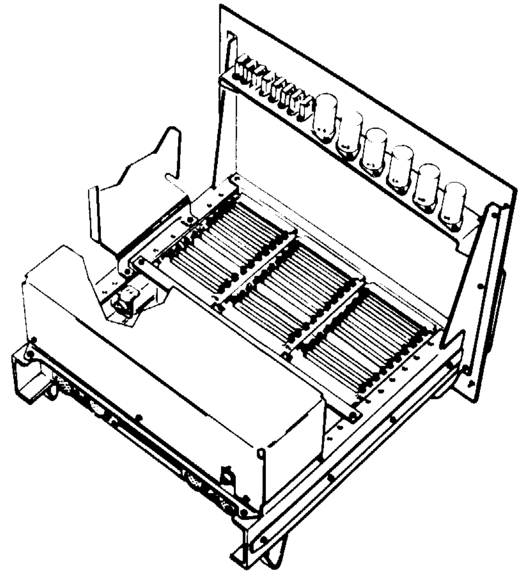
Stage Checkout Area Pneumatic Console - Checkout Accessories

3-811

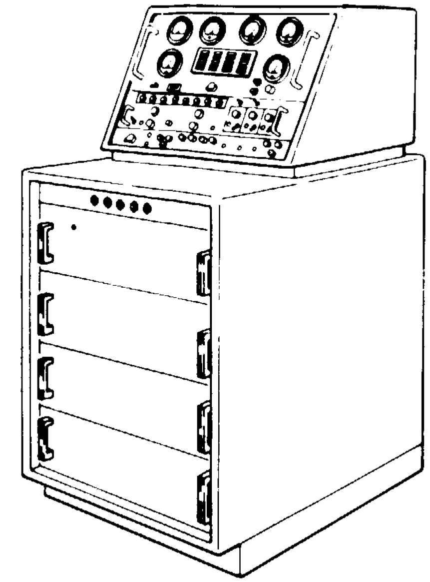
Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (8 of 9)



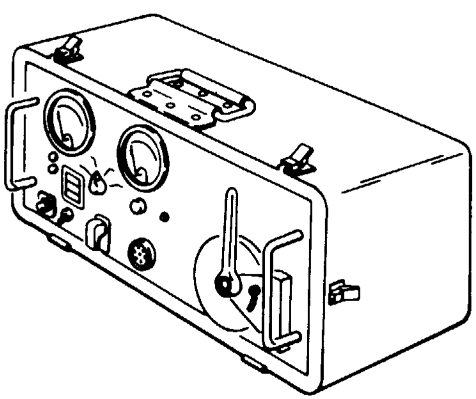
Ordnance Monitor Panel



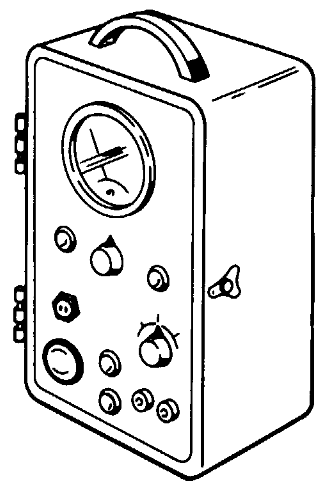
Ordnance Monitor-Control Chassis



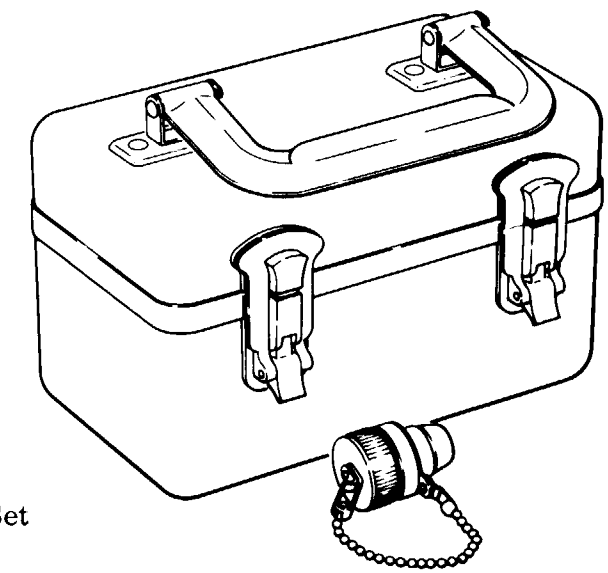
EBW Firing Unit Test Set



EBW Initiator Test Set



EBW System Pulse Test Set



EBW Initiator Simulator

3-812

Figure 10-1. Test, Checkout, and Monitoring Equipment, S-IV (9 of 9)

Table 10-6. Transportation, Protection and Handling Equipment, S-IV

Figure	Equipment	Function
10-2 (Sheet 1)	S-IV Hydraulic Servicer	Supplies hydraulic fluid to the stage engine hydraulic systems for filling, flushing, cleaning, leak checking, air purging, and checking the operation of subsystem components.
10-2 (Sheet 1)	Helium Precool Heat Exchanger	Cools and transforms helium gas to pneumatic console B for subsequent charging of the cold helium storage bottles.
10-2 (Sheet 1)	LOX Main Fill and Topping Control System	Controls the transfer of LOX from the ground storage facilities until the LOX container is filled and topped to a desired weight load.
10-2 (Sheet 1)	LH_2 Main Fill and Topping Control System	a. Controls the transfer of LH_2 from the ground storage facilities until the stage LH_2 container is filled and topped to the desired weight load. b. Controls the transfer of LH_2 to the helium precool heat exchanger.
10-2 (Sheet 2)	Transporter Assembly	Provides support, mobility, and shock isolation for the S-IV stage except when the stage is in a test stand.
10-2 (Sheet 2)	Transport Handling Kit	Used on the transporter for mounting and handling the S-IV stage during ground and water transportation.
10-2 (Sheet 2)	Transport Protective and Tiedown Kit	Provides environmental protection during all phases of transportation, and devices for shipboard tiedown during water transportation.
10-2 (Sheet 2)	Forward Interstage End Protective Cover	Protects the forward interstage area of the S-IV stage from the elements while the stage is in the test stand without the upper stages.

Table 10-6. Transportation, Protection and Handling Equipment, S-IV (Cont'd)

Figure	Equipment	Function
10-2 (Sheet 3)	Horizontal Engine Handling Fixture	Used to remove and replace the RL10A-3 engine while the stage is horizontal in the transporter.
10-2 (Sheet 3)	Forward Section Access Kit	Provides access and protection to the forward section of the stage during maintenance.
10-2 (Sheet 3)	Aft Interstage Access Kit	Provides access and protection to the aft interstage during maintenance.
10-2 (Sheet 3)	Container Interior Access Kit	Provides access, support, and lighting in the interior of the LH_2 container while the stage is in a vertical position.
	Umbilical Checkout Stand	Supports the checkout lines and maintains their attachment to the stage during checkout.
	Special Tools Kit	Provides the special tools required for maintaining and handling the S-IV stage.
	Stage Support Fixture	Used to support the stage horizontally during hangar storage.
10-2 (Sheet 4)	Liquid Hydrogen Vent Line Separation and Retraction Kit	a. Provides facilities used for transferring boil-off gaseous hydrogen from the S-IV stage to the test stand vent stack. b. Provides facilities for separating and retracting the vent line.
10-2 (Sheet 4)	GH_2 Vent Line Installation	Used in transferring GH_2 from the stage to hydrogen disposal area.
10-2 (Sheet 4)	Service Line Umbilical Installation	a. Provides the controls used for transferring propellant and pressurized gases from the facility propellant and pneumatic supply lines to the stage. b. Provides support for the umbilical carrier and the umbilical connecting and disconnecting hardware.

Table 10-6. Transportation, Protection and Handling Equipment, S-IV (Cont'd)

Figure	Equipment	Function
	Engine Alignment Kit	Contains the equipment used for aligning the S-IV stage engines at the required outboard cant angles.
10-2	Nitrogen Fill Truck	a. Used to pressurize the pneumatic side of each of the 12 stage hydraulic accumulators. b. Used to purge the stage electronic equipment containers and fill the hydraulic servicer GN_2 bottle.
	Vacuum Pumping Unit	a. Used to evacuate the annuluses of vacuum-jacketed propellant transfer lines, engine feed lines, LH_2 supply line (connected to the helium precool heat exchanger), and gas generator helium heater lines. b. Used (at Sacramento, California) to evacuate a vacuum tank that simulates altitude conditions for the engine thrust control valves during static firing.
	Propellant Valve Positioning Alignment Fixture	Used to mechanically align the propellant valve in the S-IV stage for electrical null check.
	Service Line Umbilical Kit	a. Contains the electrical cables, air conditioning lines, propellant lines, and pneumatic lines used to connect the propellant and pneumatic lines kit to the S-IV stage. b. Provides the facilities used for attaching the umbilicals to the S-IV stage and for disconnecting the umbilical carrier from the S-IV stage.
	Propellant and Pneumatic Lines Kit	Contains the lines, fittings, brackets, and hardware used to transfer propellants and gases from the GSE to the service line umbilical kit.

Table 10-6. Transportation, Protection and Handling Equipment, S-IV (Cont'd)

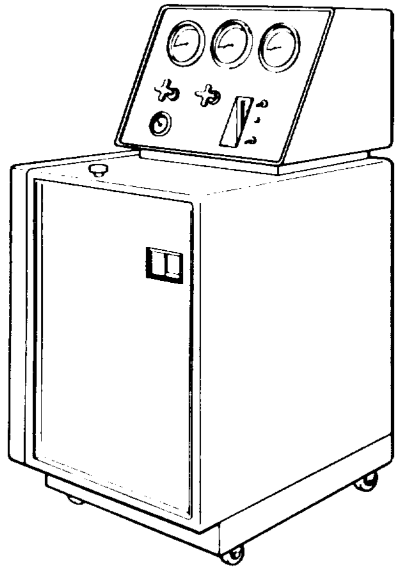
Figure	Equipment	Function
	Ullage Rocket Fairing Handling and Storage Container Fixture	Protects the ullage rocket and fairing kit during handling and storage.
	Retrorocket and Ullage Rocket Handling Sling Kit	Supports the ullage rockets and retrorockets during removal and installations.
	Weight and Balance Kit	Used to determine the dry weight of the S-IV stage and/or aft interstage.
	Hangar Cable Network Kit	Contains the cables used for connecting electrical GSE to the S-IV stage for checkout.
	Cable Network Kit	Contains the cables used for connecting electrical GSE to contractor-furnished terminal distributors.
	Liquid Hydrogen Vent Line	Contains the equipment used in transferring gaseous hydrogen from the S-IV stage to the umbilical tower vent stack.
	Vehicle Mounting Alignment Kit	Contains the alignment pins used in aligning the aft skirt to the aft interstage.
	Liquid Hydrogen Main Fill	a. Used in controlling the transfer of LH_2 from the ground storage facilities into the LH_2 container in the S-IV stage until filled and topped to the desired mass load during countdown. b. Controls the transfer of LH_2 to the helium pre-cool heat-exchanger.
	Engine Turbine Torque Wrench Adapter	Used to adapt the torque wrench to the engine turbine gear box for determining gear torque.
	Weight and Balance Kit	Contains the equipment used to mechanically weigh the S-IV stage and aft interstage to determine the center of gravity.

Table 10-6. Transportation, Protection and Handling Equipment, S-IV (Cont'd)

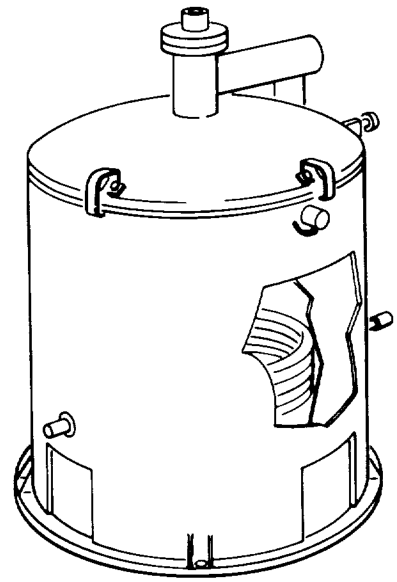
Figure	Equipment	Function
	Vehicle Mounting Alignment	Contains the equipment used in aligning the aft skirt with the aft interstage and the S-IV stage with the test stand.

Table 10-7. Stage Subsystem Test Equipment, S-IV

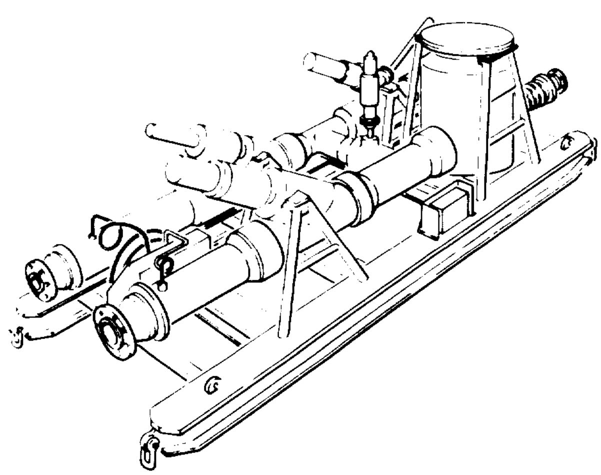
Figure	Equipment	Function
10-3 (Sheet 1)	Valve Actuator Test Set	Used to test the S-IV stage hydraulic valve actuator assembly, engine hydraulic system, and valve actuator potentiometers in both the stage checkout area and the component laboratory.
10-3 (Sheet 1)	S-IV Battery Test Set and Charger	Used to check the S-IV stage batteries and the heater blanket circuits.
10-3 (Sheet 2)	S-IV Sequencer Subsystem Test Set	a. Utilizes simulated flight inputs to check out the engines, payload, safety, and stage sequence circuits. b. Used to detail troubleshoot the stage sequencer.
10-3 (Sheet 2)	Inverter Test Set	Used in the bench maintenance area to test the S-IV stage static inverter-converter.
10-3 (Sheet 2)	Inverter Ground Power Supply	Supplies 28-volt and 32-volt dc power to the S-IV stage static inverter.
	Propellant Utilization Electronics System Test Set	a. Used to check out the propellant utilization electronics assembly. b. Used to perform operational checks on the stage valve positioner assembly while the stage is in the assembly area.



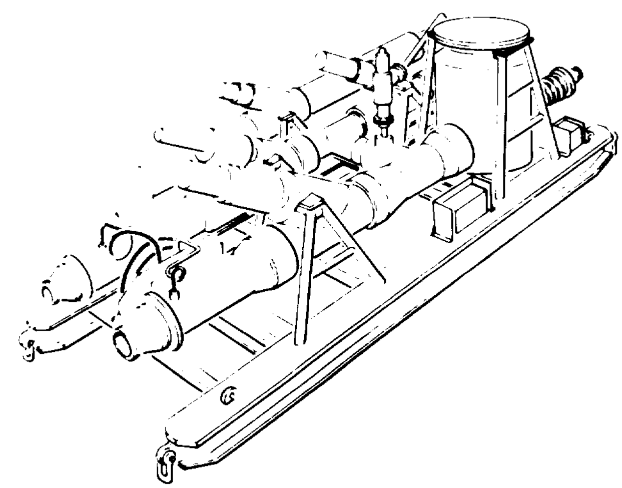
S-IV Hydraulic Servicer



Helium Precool Heat Exchanger



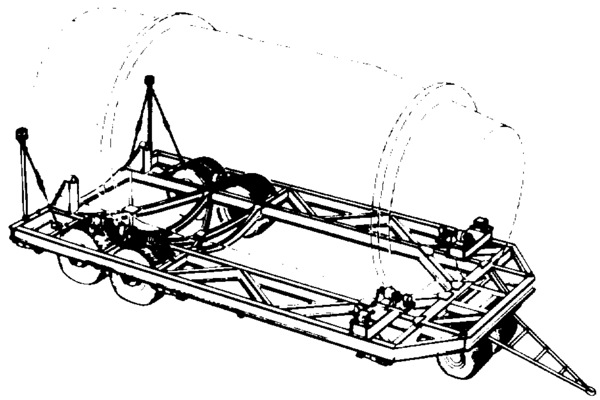
LOX Main Fill and Topping Control System



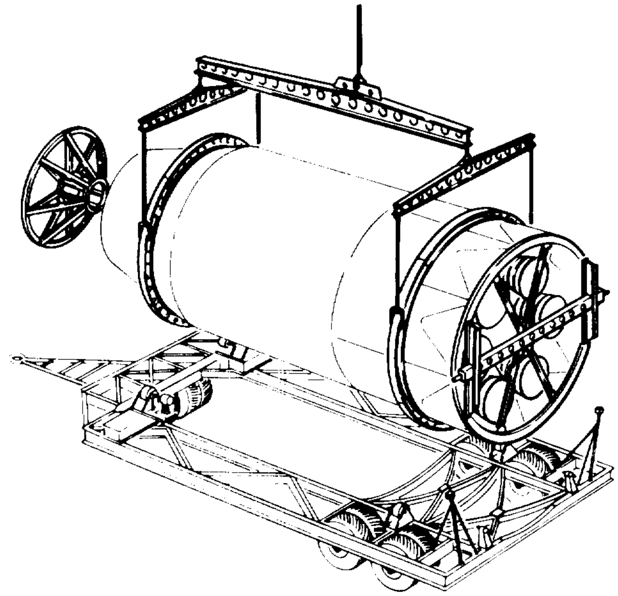
LH_2 Main Fill and Topping Control System

3-800

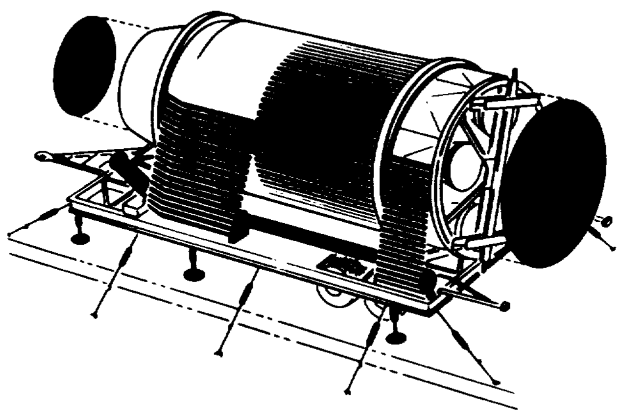
Figure 10-2. Transportation, Protection, and Handling Equipment, S-IV (1 of 4)



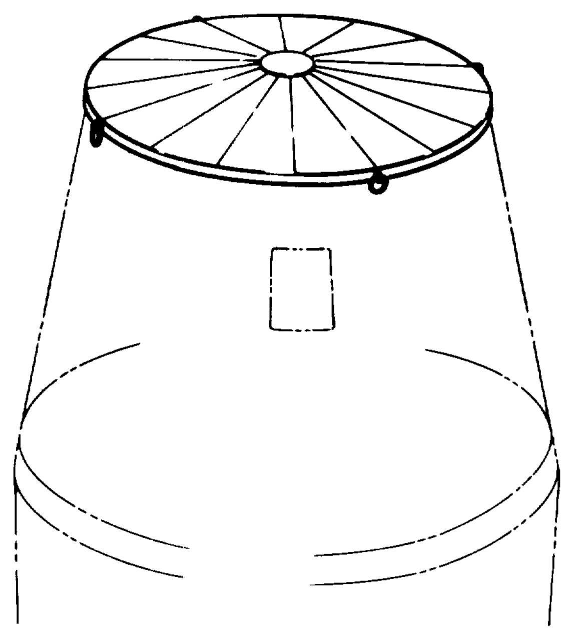
Transporter Assembly



Transport Handling Kit



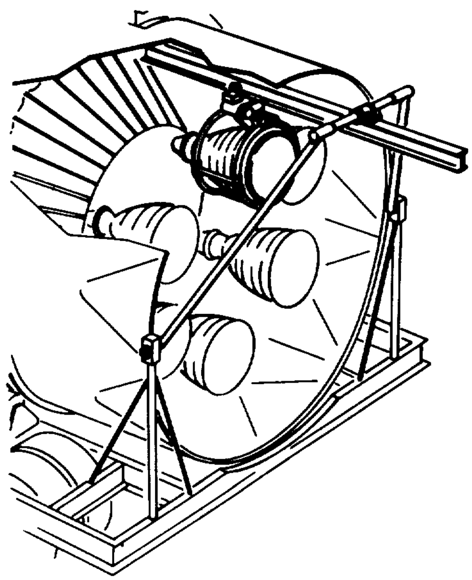
Transport Protective and
Tiedown Kit



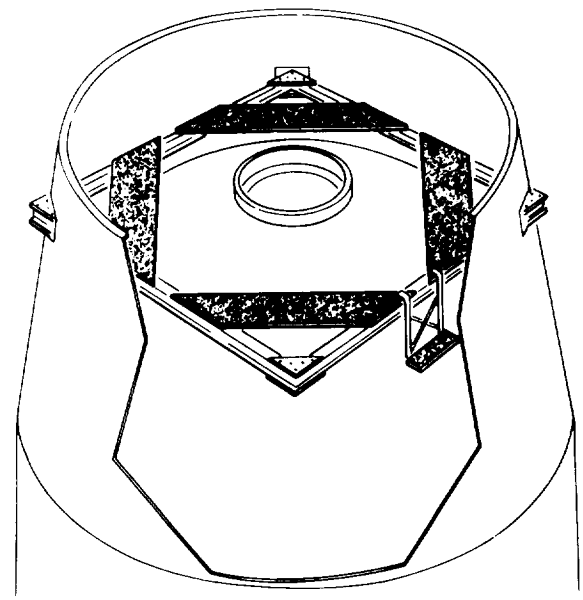
Forward Interstage End Protective
Cover (Tentative)

3-801

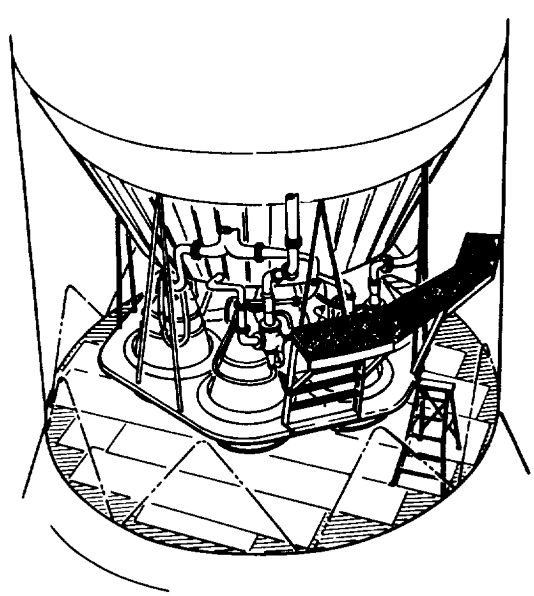
Figure 10-2. Transportation, Protection, and Handling Equipment, S-IV (2 of 4)



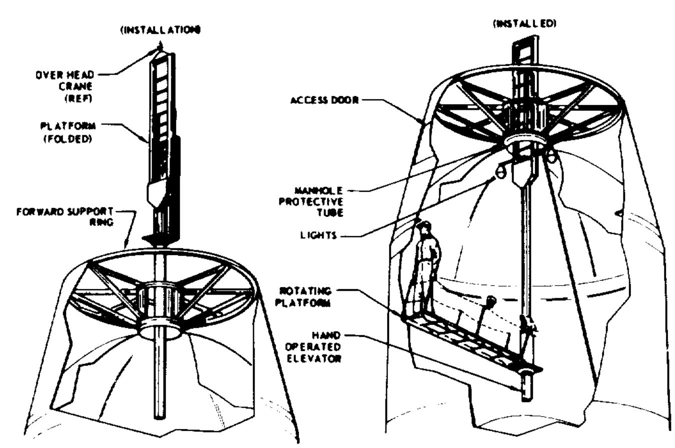
Horizontal Engine Handling Fixture



Forward Section Access Kit



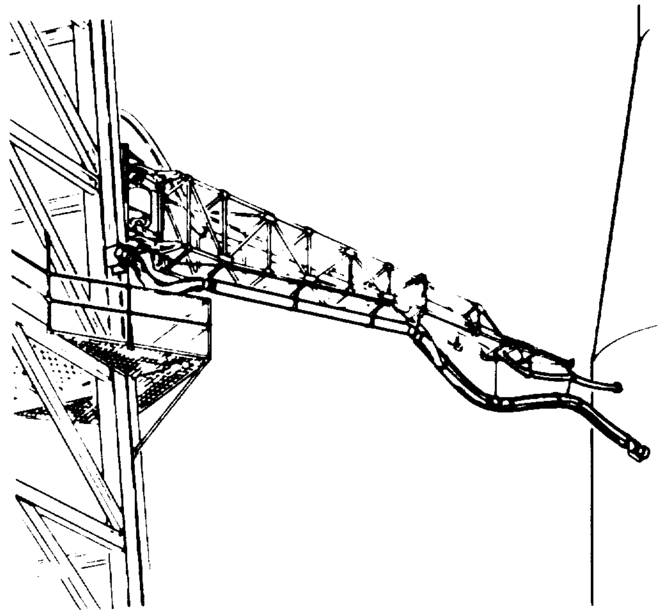
Aft Section Access Kit



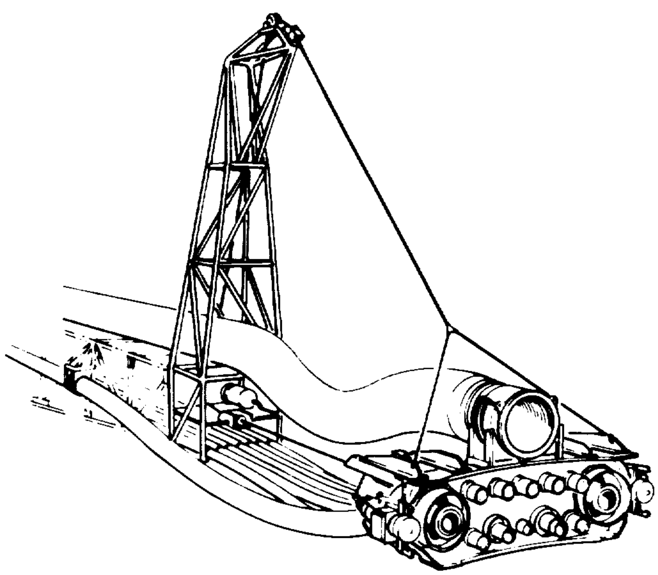
Container Interior Access Kit

3-802

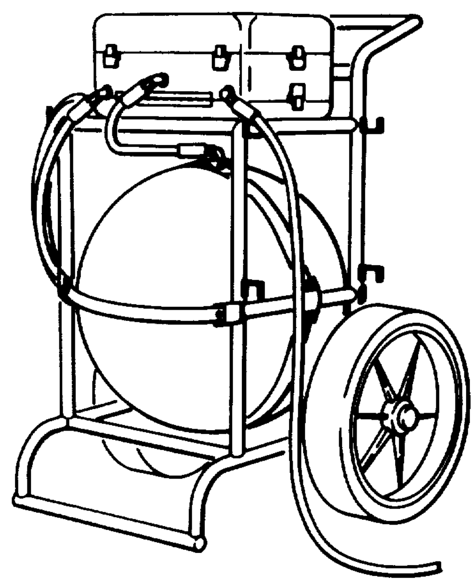
Figure 10-2. Transportation, Protection, and Handling Equipment, S-IV (3 of 4)



GH_2 Vent Line Installation



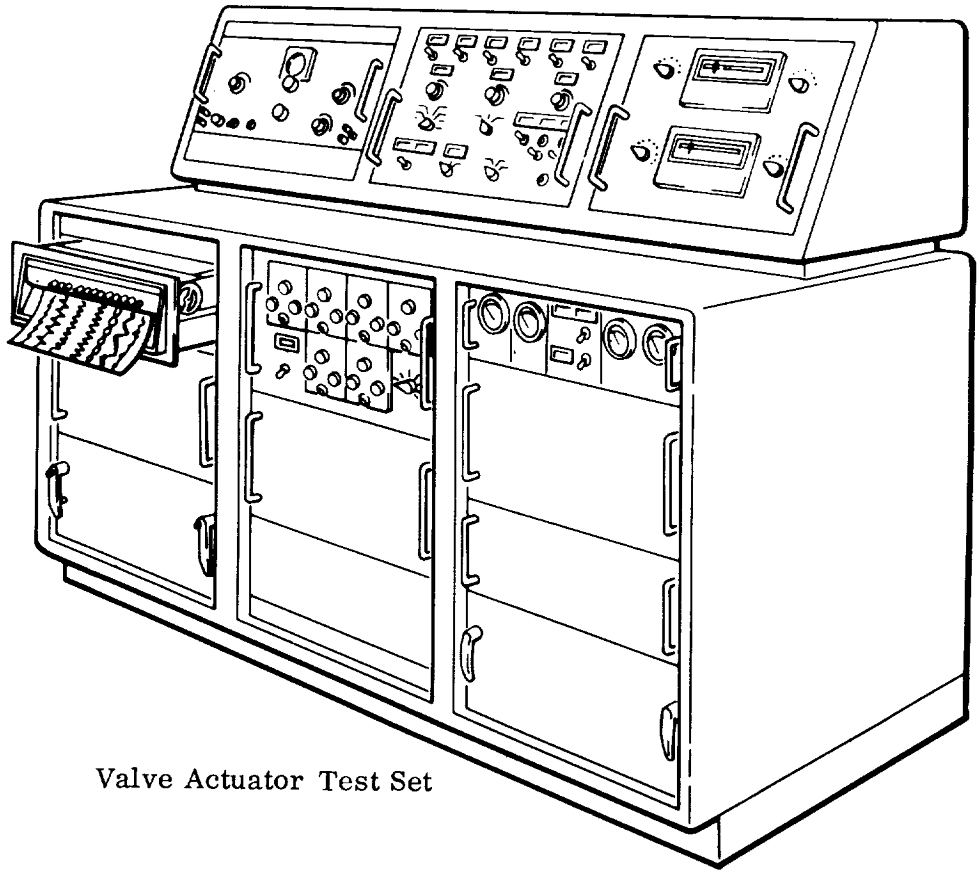
Service Line Umbilical Installation

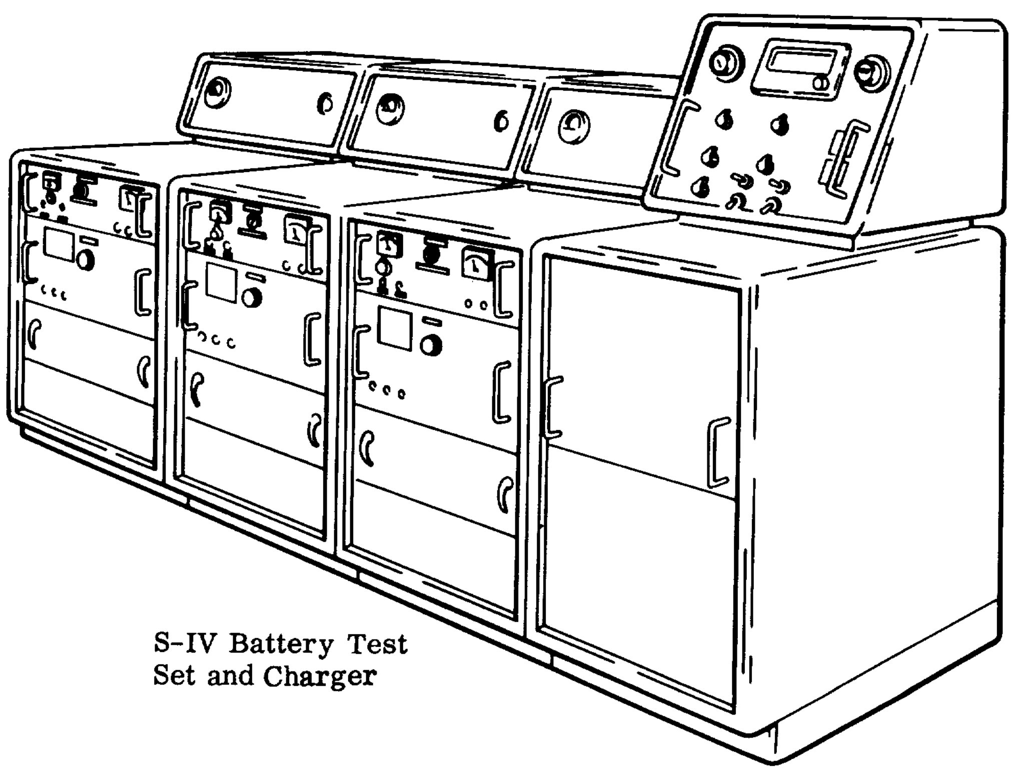


Nitrogen Fill Truck

3-803

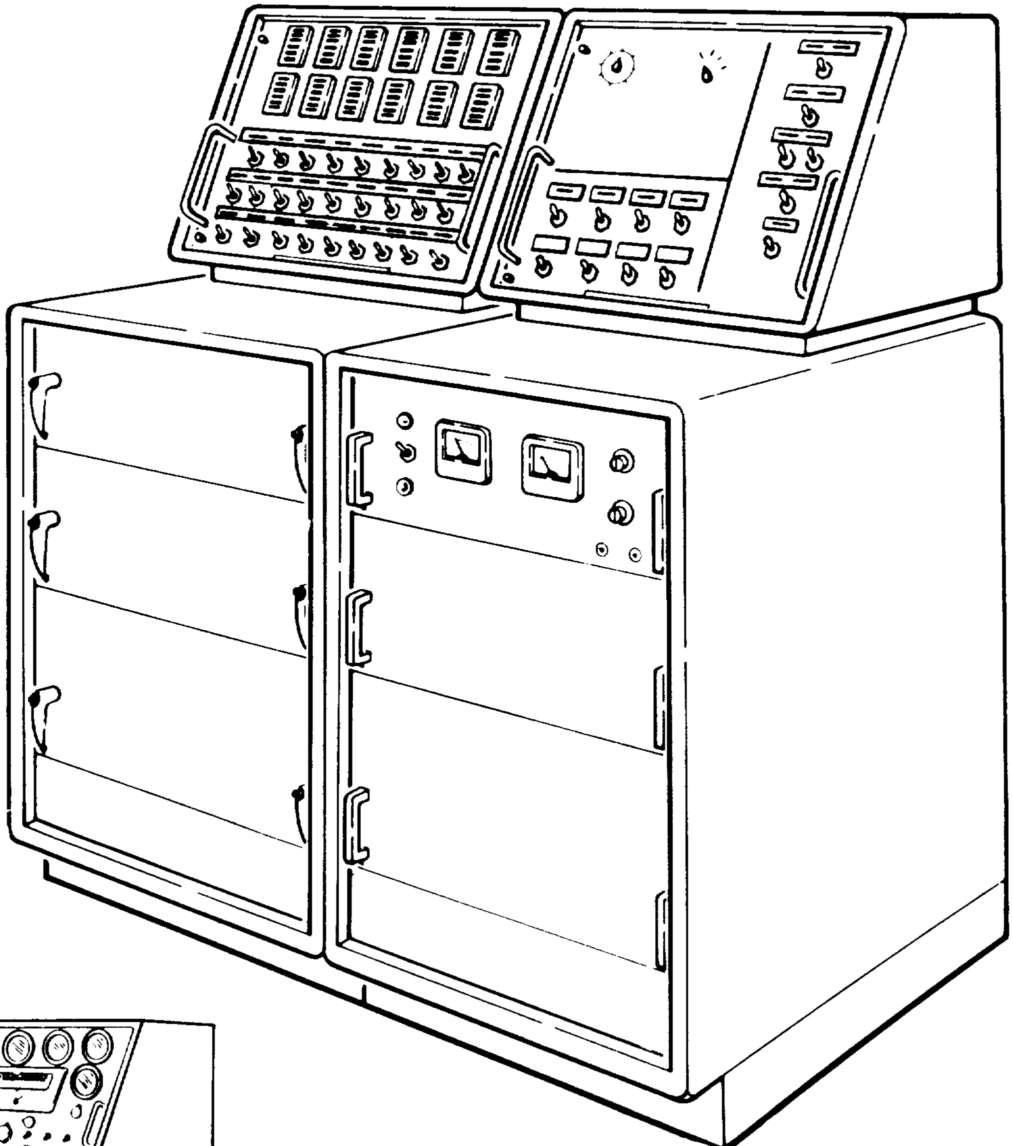
Figure 10-2. Transportation, Protection, and Handling Equipment, S-IV (4 of 4)



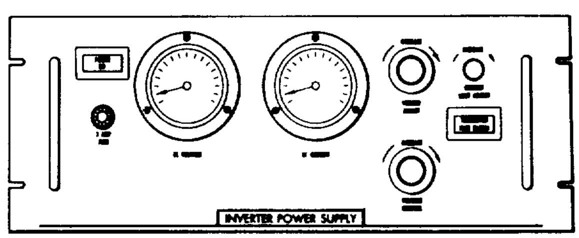


3-813

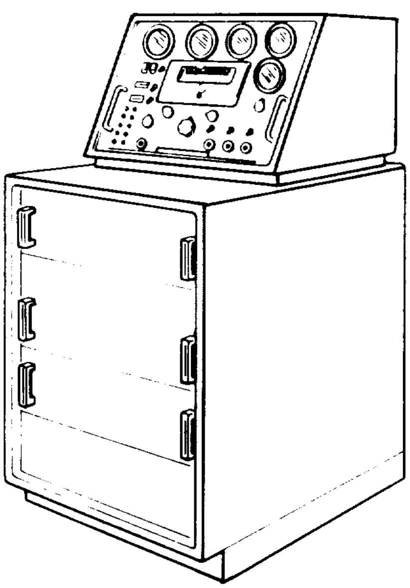
Figure 10-3. Stage Subsystem Test Equipment, S-IV (1 of 2)



S-IV Sequencer Subsystem Test Set



Inverter Ground Power Supply



Inverter Test Set

3-814

Figure 10-3. Stage Subsystem Test Equipment, S-IV (2 of 2)

Table 10-8. Instrumentation Equipment, S-IV

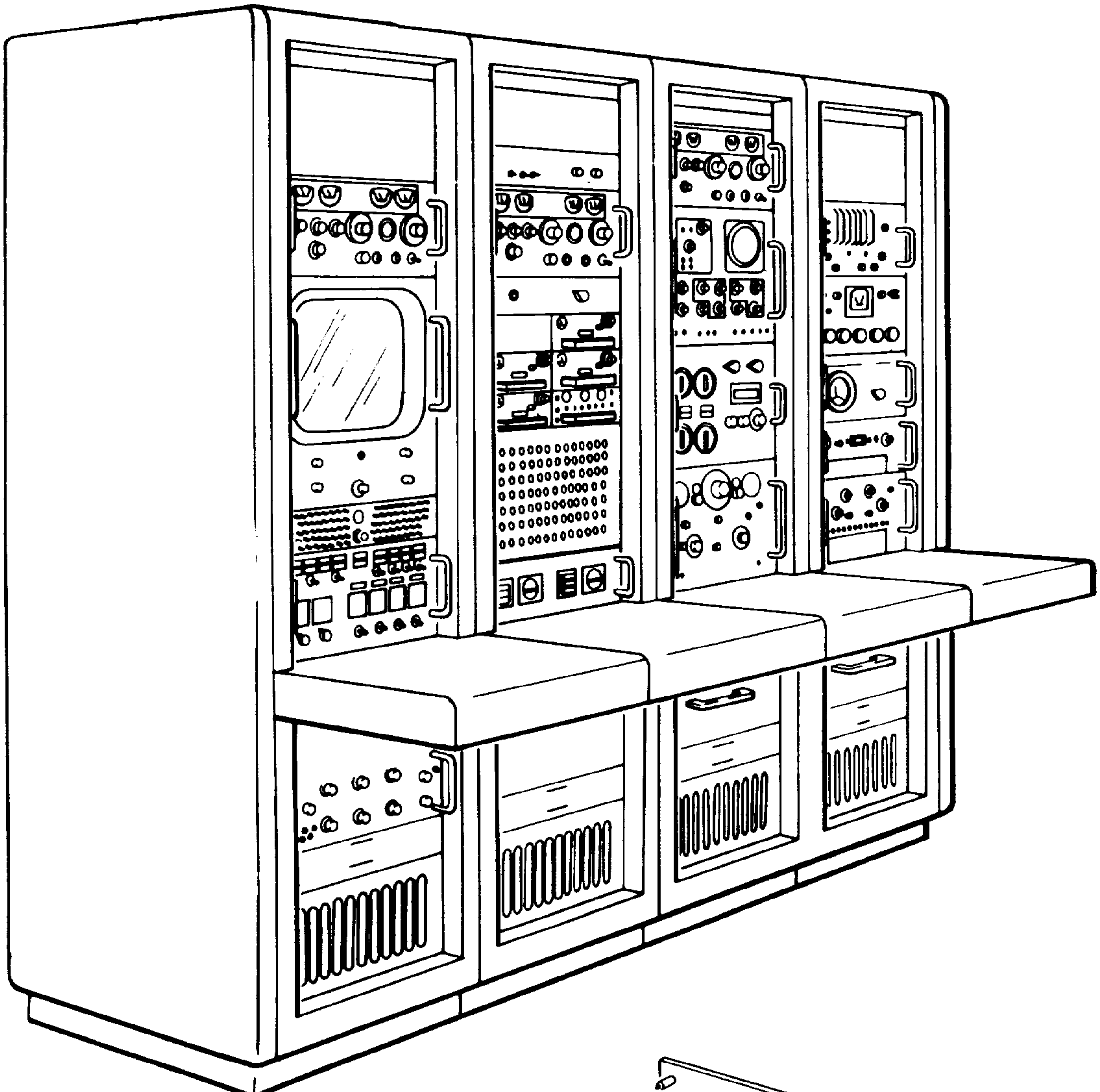
Figure	Equipment	Function
10-4 (Sheet 1)	PDM/FM/FM Checkout Monitor Consoles	a. Used to checkout the stage instrumentation. b. Contains the circuits used to monitor and check out the composite stage telemetry signal.
10-4 (Sheet 1)	Stage System Status Relay Assemblies Nos. 1, 2 and 3 (Typical)	a. Provide the automatic logic circuitry from initiation of terminal countdown to launch. b. Provide logic circuits for instrumentation, calibrating, LOX and LH_2 loading, stage readiness monitoring, and for the transfer of all stage power to internal power.
10-4 (Sheet 2)	PDM/FM/FM Component Test Console	Used for testing the PDM/FM/FM telemetry system and its components prior to stage installation.
10-4 (Sheet 2)	Signal Conditioning Console	a. Receives instrumentation signals from the S-IV stage. b. Conditions the signals to the proper level and format for transmittal to remote sequence recorders, panel lights, and amplifiers.
10-4 (Sheet 3)	Command Destruct Receiver Component Test Set	Used for testing the command destruct system components prior to stage installation.
10-4 (Sheet 3)	Command Destruct Receiver Simulator	Used to supply the RF carrier and audio signal tones (via closed loop) to check out the complete destruct system after its installation in the stage.
10-4 (Sheet 3)	S-IV Destruct Panel	a. Used to remote control and monitor the stage receiver functions. b. Used to control the destruct EBW firing unit monitoring system.

Table 10-8. Instrumentation Equipment, S-IV (Cont'd)

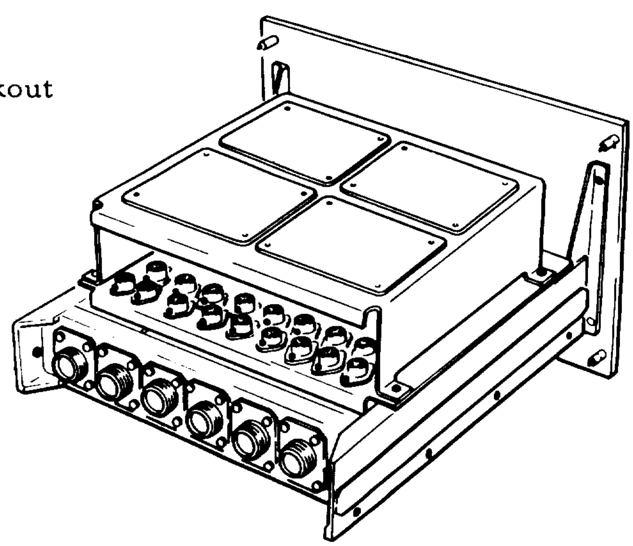
Figure	Equipment	Function
10-4 (Sheet 3)	Aft Interface Junction Box	Provides a convenient and flexible means of performing the following interconnections: a. GSE to the S-IV aft interface b. GSE to the GSE test set c. S-I stage substitute to the S-IV stage aft interface d. S-IV stage substitute to the GSE e. S-I stage substitute to the GSE
	Telemetry Power Supply	Supplies external power to the instrumentation telemetry systems.
	Stage Instrumentation Simulator	Used to check the interfaces between the telemetry systems and the sensing devices.
	Helium Heater and Engine Exciter Test Set	Used in performing qualitative and analytical tests on the helium heater and engine exciter.
	Telemetry Test Evaluation Consoles	Used for recording and reproducing the telemetry system data during checkout.

Table 10-9. Propellant and Gas Servicing Equipment, S-IV

Figure	Equipment	Function
10-5	Remote Propellant Loading Relay Assembly	Provides the remote controls used for stage propellant loading and monitoring.
10-5	Propellant Utilization Calibration and Checkout Test Set	a. Used in calibrating the LOX and LH_2 container full and empty bridge circuits located in the stage electronics assembly.



PDM/FM/FM Checkout
Monitor Consoles



Stage System Status Relay Assemblies
Nos. 1, 2, 3 (Typical)

3-815

Figure 10-4. Instrumentation, S-IV (1 of 3)

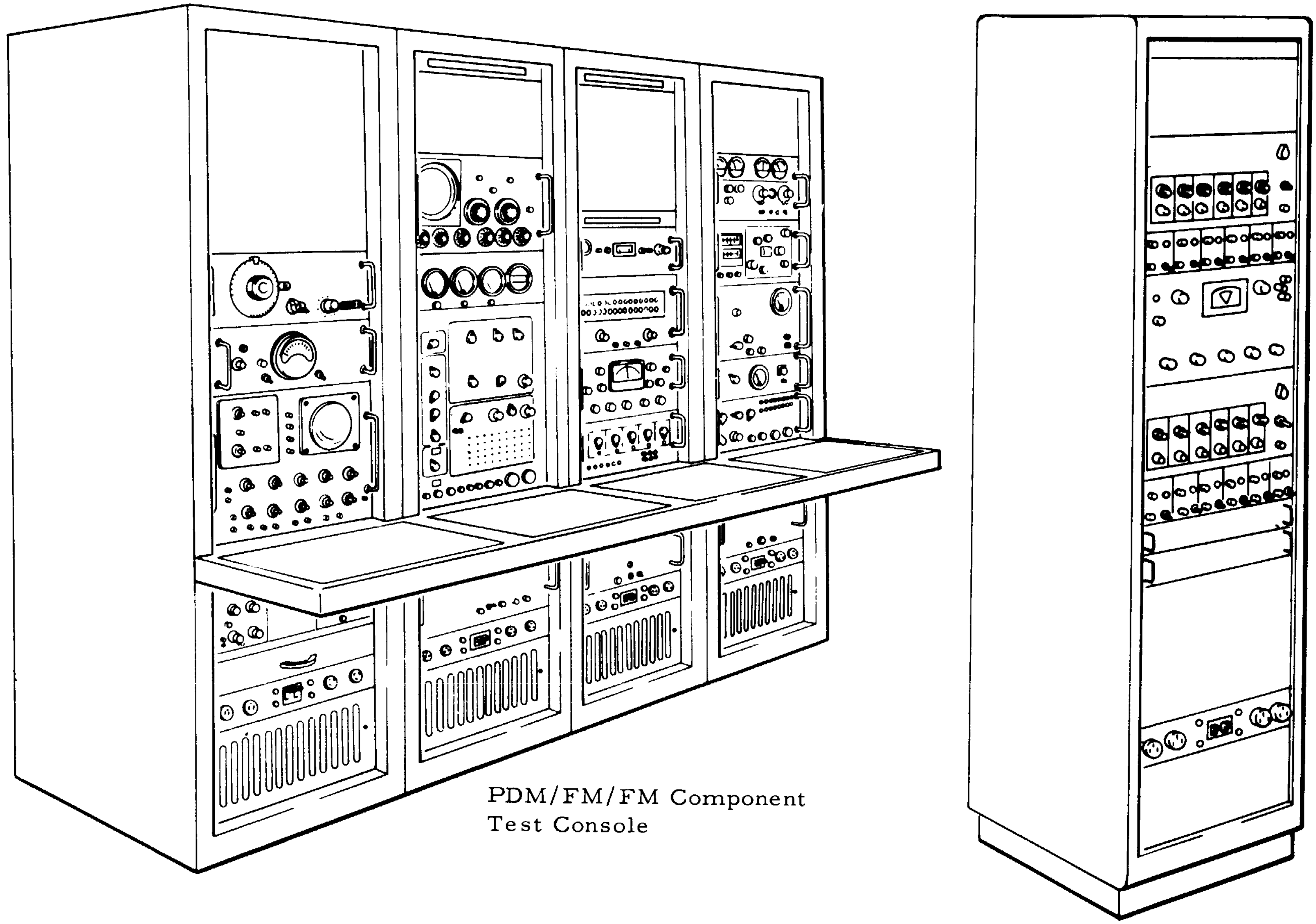


Figure 10-4. Instrumentation, S-IV (2 of 3)

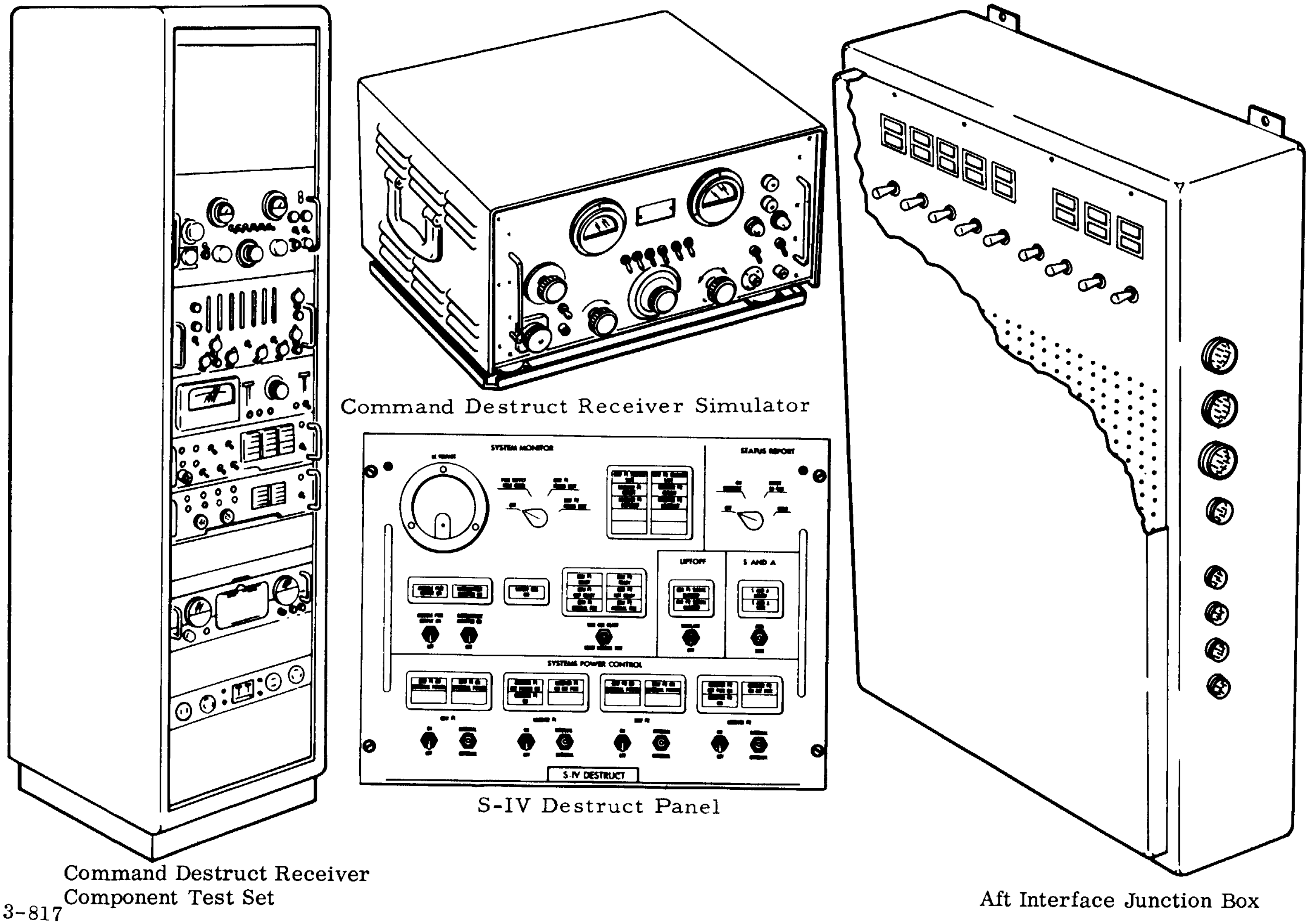
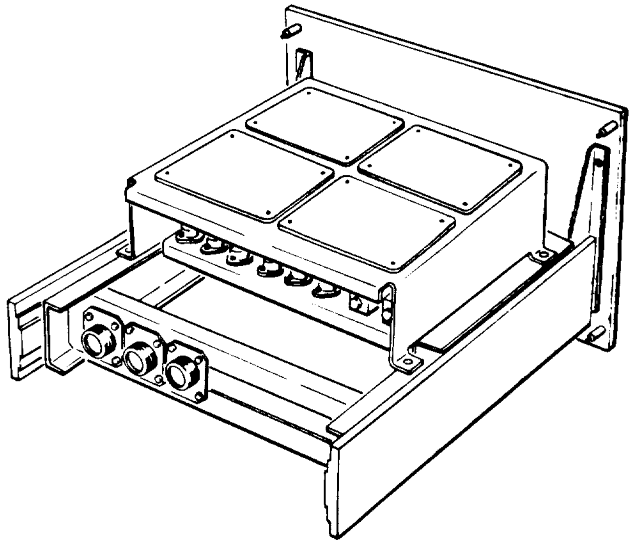


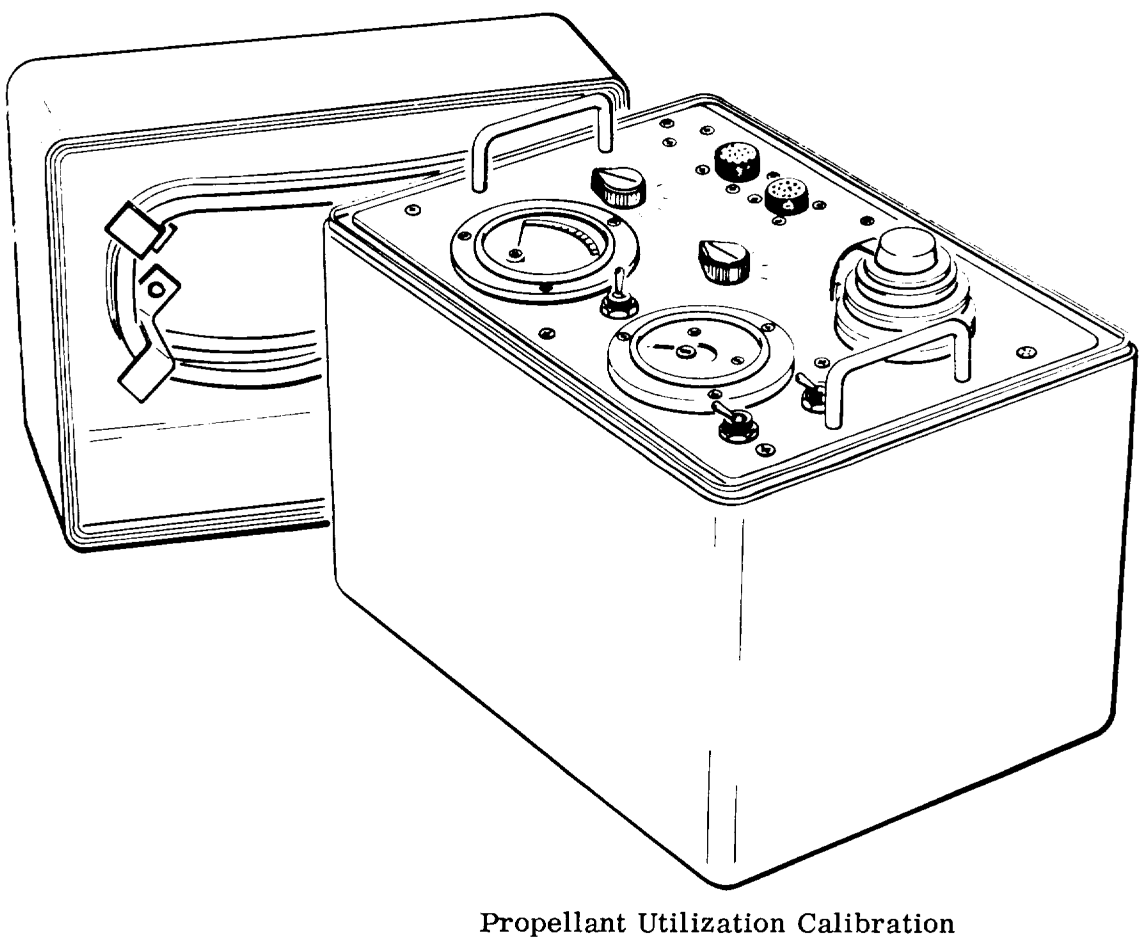
Figure 10-4. Instrumentation, S-IV (3 of 3)

Table 10-9. Propellant and Gas Servicing Equipment, S-IV (Cont'd)

Figure	Equipment	Function
		b. Used in checking the output of the valve controller amplifiers.
	Propellant Utilization System Test Set	Used in testing the propellant utilization system.



Remote Propellant Loading Relay Assembly



Propellant Utilization Calibration
and Checkout Test Set

3-818

Figure 10-5. Propellant and Gas Servicing Equipment, S-IV

CHAPTER 2

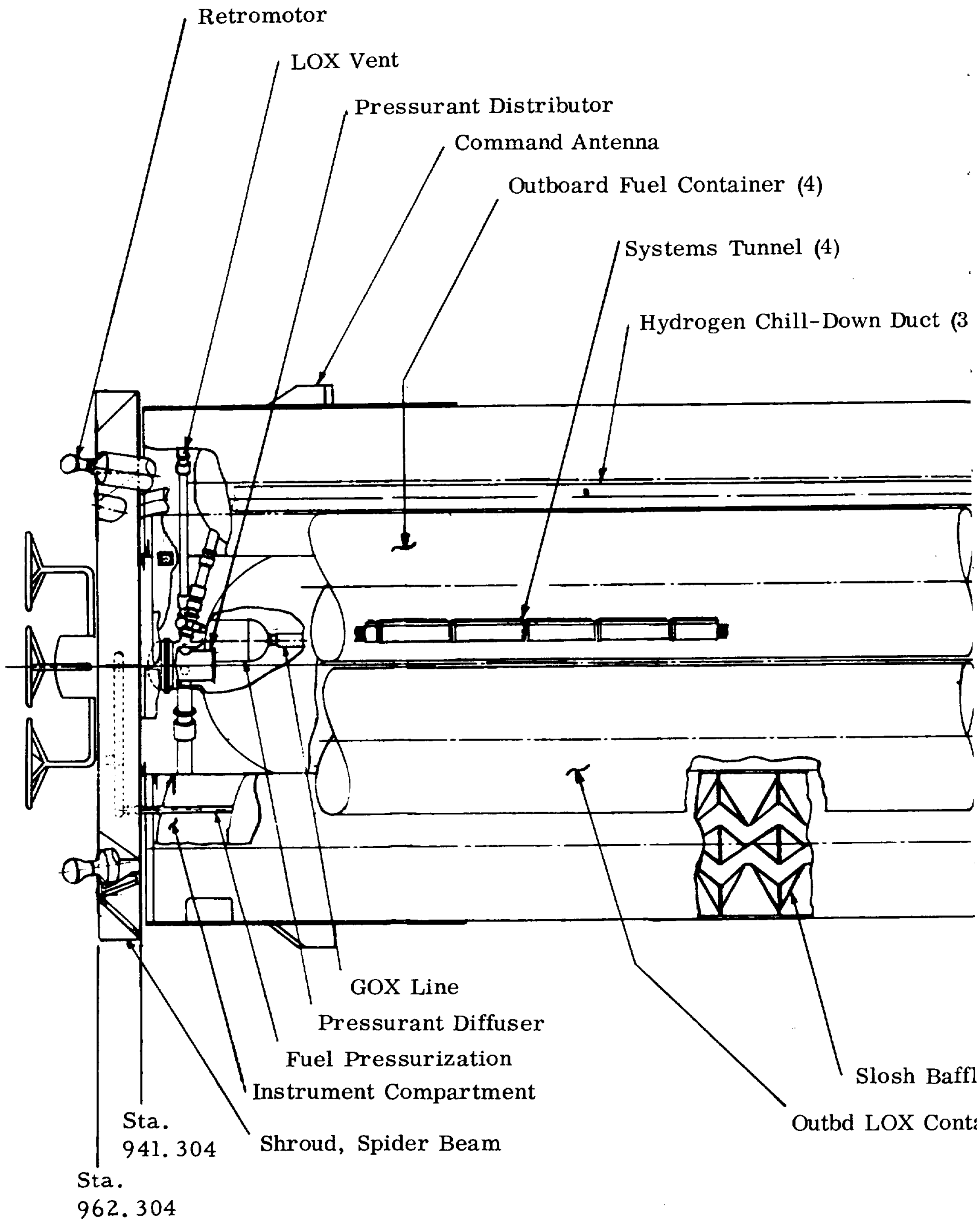
SECTION XI

STAGE CONFIGURATIONS, SATURN I

XI

LIST OF ILLUSTRATIONS

XI



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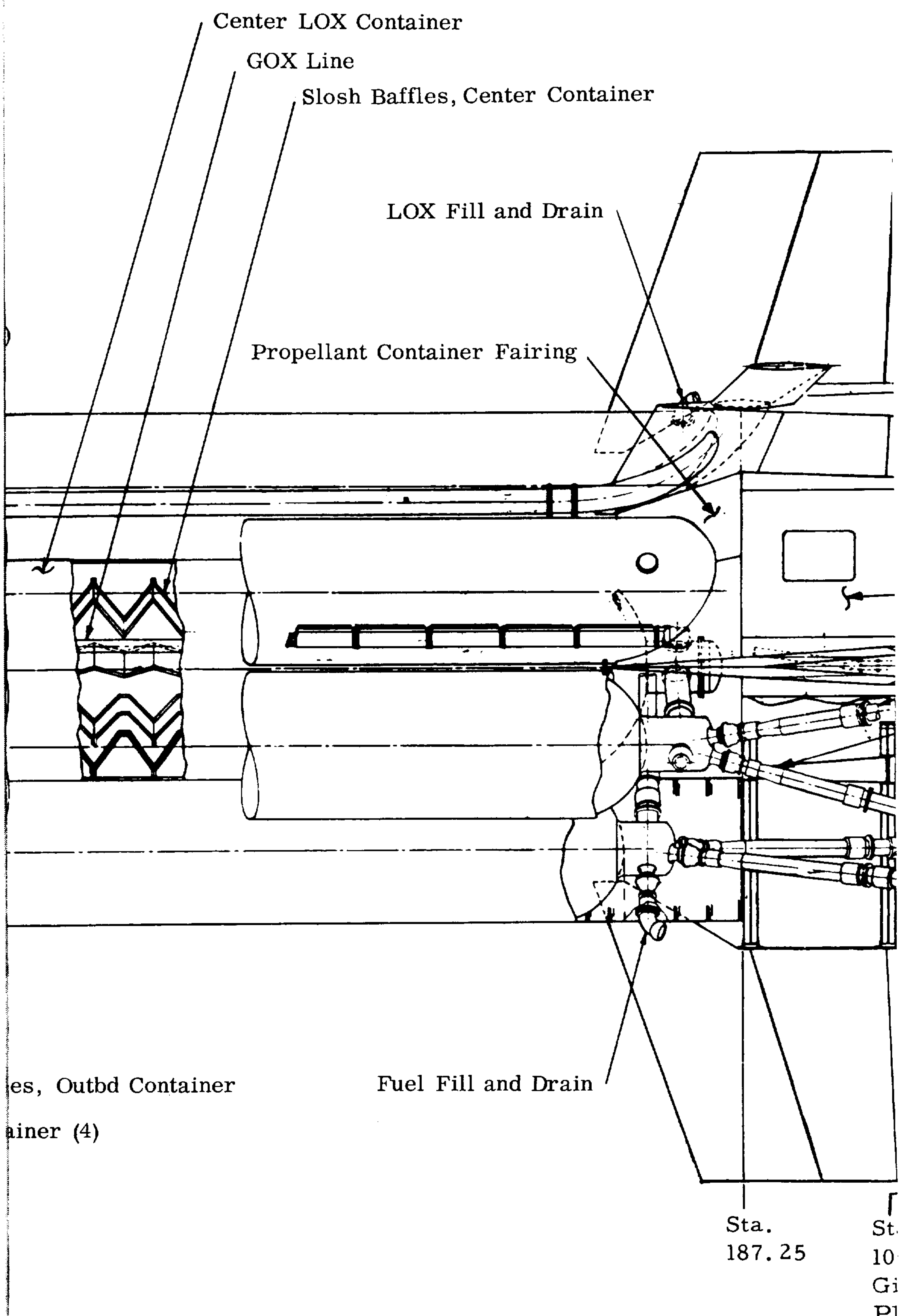
FOLDOUT FRAME 1



FOLDOUT FRAME 2

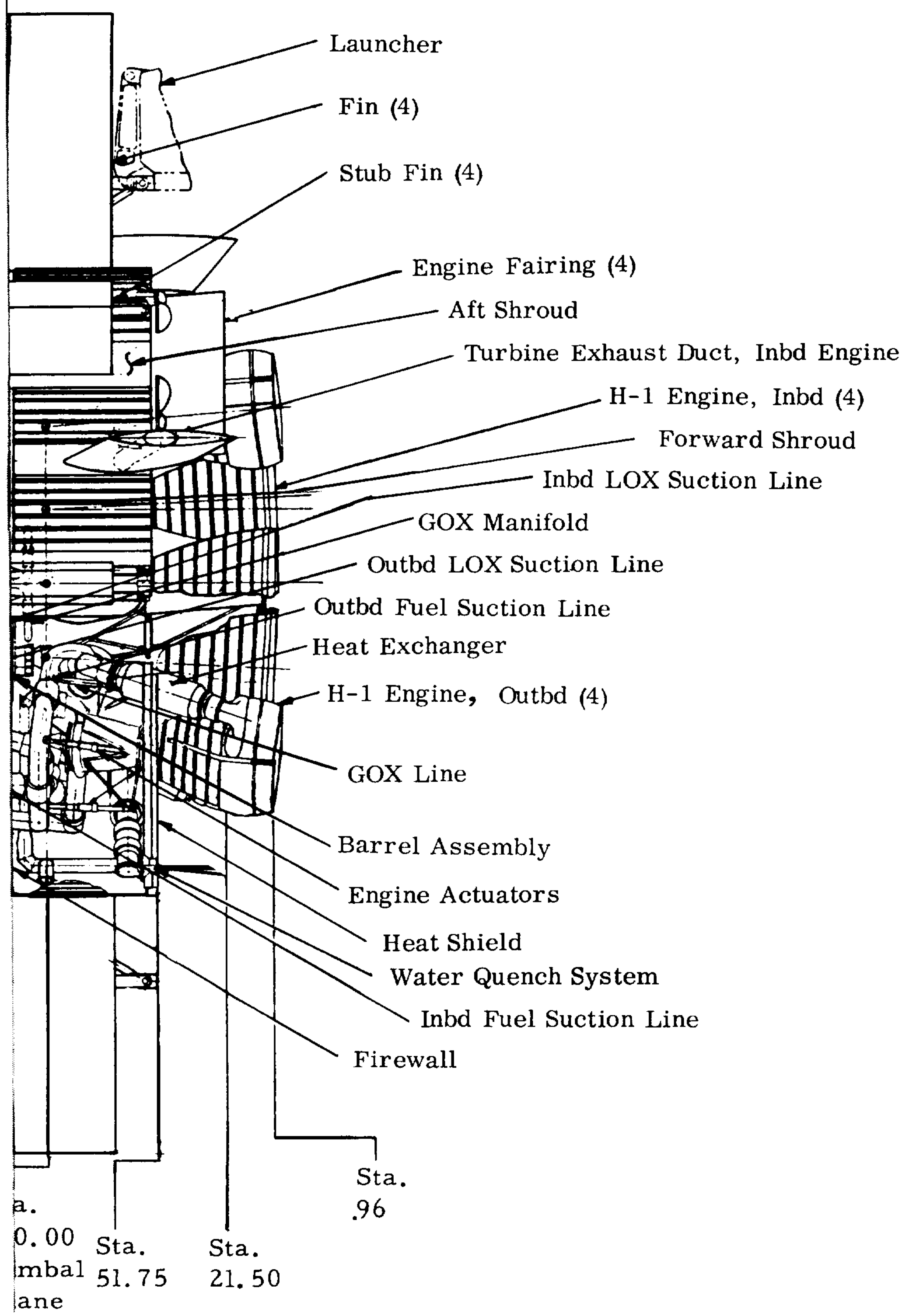


Figure 11-1. S-I Inboard Profile

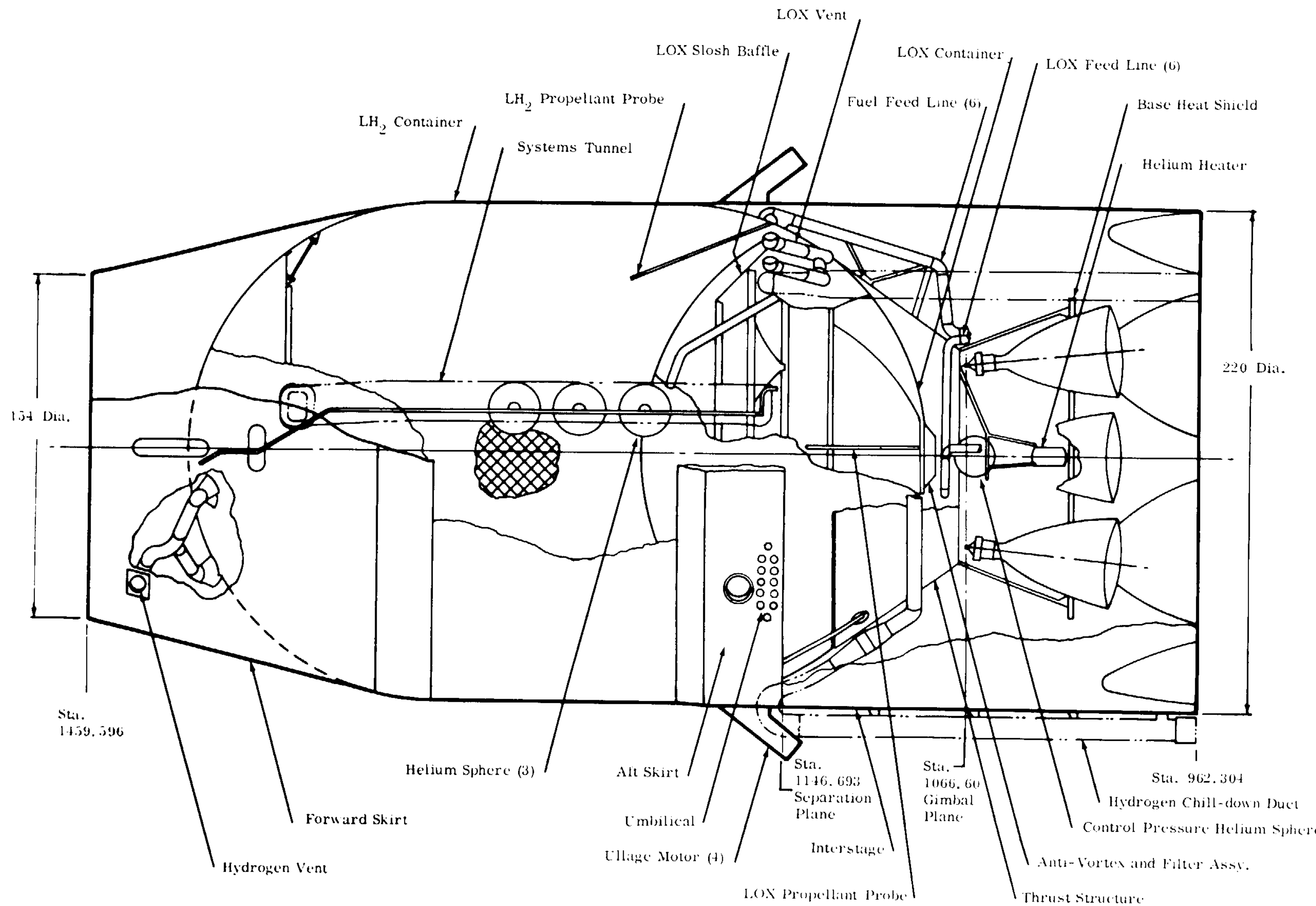


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Figure 11-2. S-IV Inboard Profile

CHAPTER 3

SECTION XII

INTRODUCTION

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SECTION XII.
INTRODUCTION

12-1. SATURN IB LAUNCH VEHICLE.

The Saturn IB launch vehicle, Figure 12-1, consists of an S-IB first stage, an S-IVB second stage and an instrument unit mounted above the second stage. Operational data for the vehicle are listed in Table 12-1.

12-2. SATURN IB - APOLLO MISSION OBJECTIVES.

The principal objective of the Saturn IB - Apollo space vehicle program is manned Apollo flight operations in extended earth orbit. Ten Saturn IB -- Apollo flights are planned, utilizing launch vehicles SA-201 through SA-210. Two additional launch vehicles, SA-211 and SA-212, are designated as spares.

In the first two Saturn IB - Apollo flights (SA-201 and SA-202) the primary mission objective is flight testing of the launch vehicle. Flight testing of the unmanned spacecraft and compatibility testing of the space vehicle are secondary mission objectives. The flight testing of the S-IVB stage of the launch vehicle supports also the Saturn V project. (The S-IVB stage is used in both Saturn IB and Saturn V.)

The third though sixth Saturn IB flights (SA-203 through SA-206) will be used as man-rating flights, resulting in qualification of both the launch vehicle and the Apollo spacecraft. Consideration will be given to manning some of these flights in the event of successful early flights.

Vehicles SA-207 through SA-210 are planned as manned flights with extended duration earth orbital operation as the primary objective. Operational experience with the launch vehicle is a secondary mission objective.

Detailed information about the Saturn IB - Apollo mission objectives and flight data is summarized in Table 12-2.

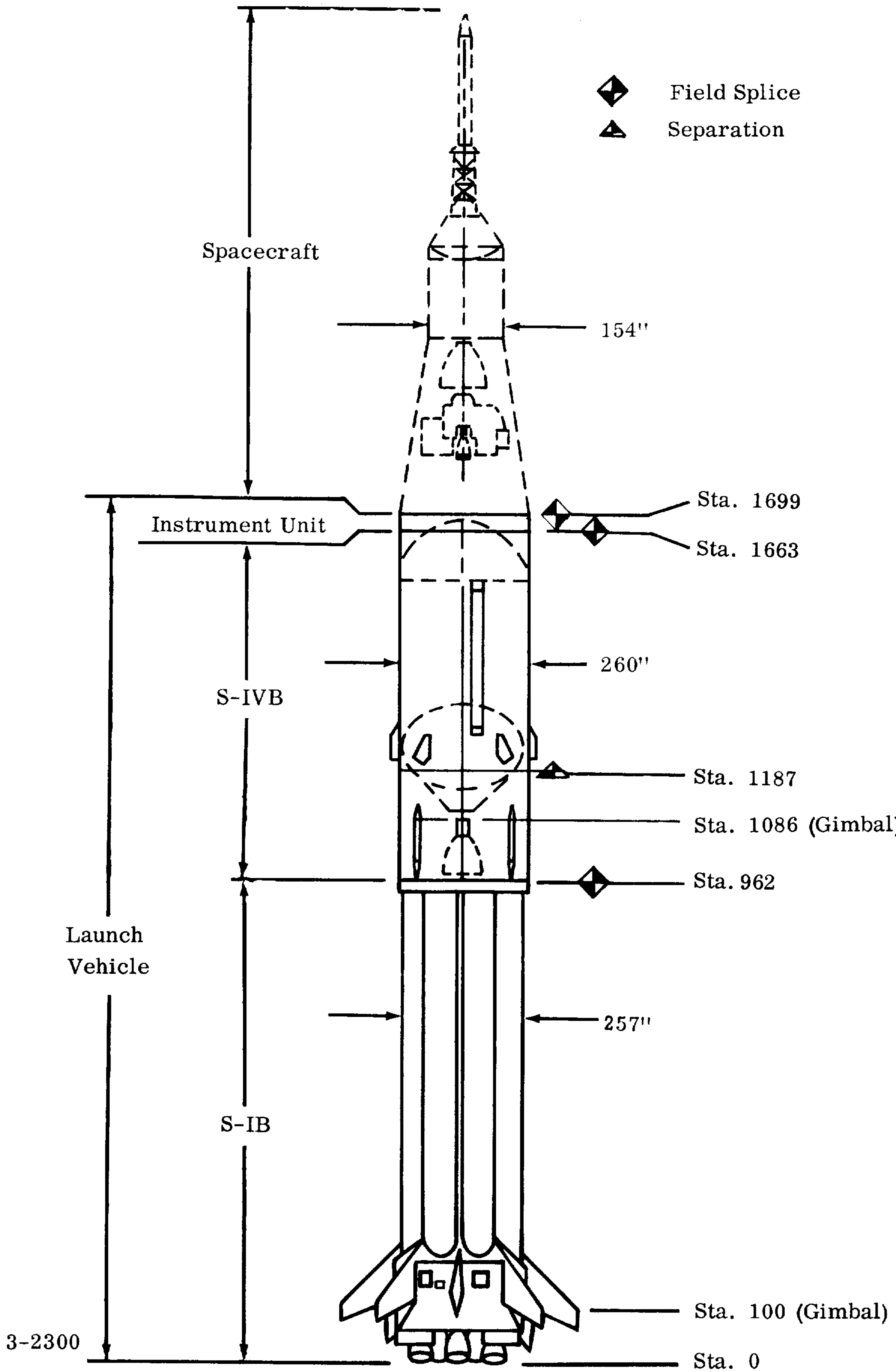


Figure 12-1. Saturn IB Launch Vehicle

[redacted]

Table 12-1. Saturn IB Operational Data

Item	Data
VEHICLE	
Number of stages	2
Length - Without spacecraft	141.6 feet
Maximum diameter - without fins	22.8 feet
- with fins	40.7 feet
[1]Launch vehicle weight - at ground ignition	1,294,000 pounds
Payload type	Apollo Spacecraft
[2]Payload weight - at ground ignition	40,600 pounds
[3]Injection weight - Earth orbit	34,000 pounds
S-IB STAGE	
Prime contractor	Chrysler Corporation
Length	80.2 feet
Maximum diameter - without fins (across thrust structure)	22.8 feet
- with fins	40.7 feet
Stage weight - at ground ignition	1,003,000 pounds
Dry weight	91,000 pounds
Engines	Rocketdyne H-1 (8)
Total nominal thrust (sea level)	1,600,000 pounds
Propellants	LOX and RP-1
Mainstage propellant weight	882,000 pounds
Mixture ratio (oxidizer to fuel)	2.26:1
Specific impulse (sea level)	256 seconds
S-IVB STAGE	
Prime contractor	Douglas Aircraft Co.
Length	59.1 feet
Diameter	21.7 feet
[4]Stage weight - at ground ignition	243,000 pounds
[4]Dry weight	20,000 pounds
Engine	Rocketdyne J-2 (1)
Total nominal thrust (vacuum)	200,000 pounds
Propellants	LOX and LH_2

[redacted]

Table 12-1. Saturn IB Operational Data (Cont'd)

Item	Data
[5] Mainstage propellant weight	219,000 pounds
Mixture ratio (oxidizer to fuel)	5:1
Specific impulse (vacuum)	426 seconds
INSTRUMENT UNIT	
Prime contractor	MSFC
Length	3.0 feet
Diameter	21.7 feet
[5] Weight - at ground ignition	2,600 pounds

[1] Includes two stages, instrument unit, payload and LES.

[2] Includes 6600 pounds for the LES, no coast mission.

[3] 105-nautical mile circular orbit, payload only, no coast mission.

[4] Excludes 5600 pounds for the S-IB/S-IVB interstage and retromotors, no coast mission.

[5] No coast mission.

In all ten planned Saturn IB - Apollo flights, the Apollo spacecraft configuration includes a CM, an SM, an adapter and an LES that is jettisoned after second-stage ignition.

Vehicles SA-203 through SA-210 will also have the ascent stage of a LEM.

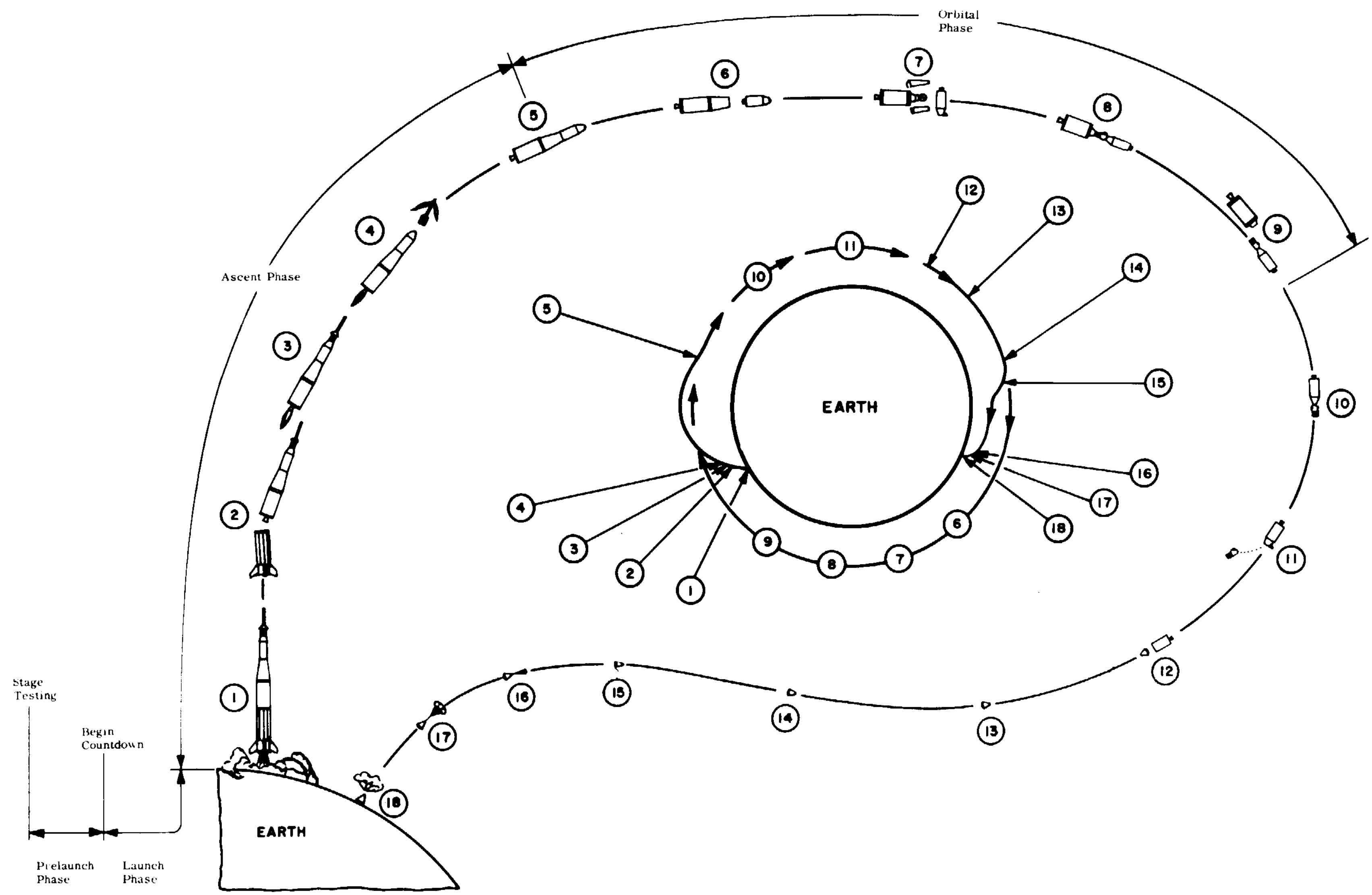
12-3. MISSION PROFILE.

A typical Saturn IB - Apollo mission profile, through which a Saturn IB launch vehicle lifts a manned R&D spacecraft into a 105-nautical mile circular earth orbit, is illustrated in Figure 12-2. The launch vehicle, by means of first stage and second stage burn, injects the payload into the circular orbit. The S-IVB stage then stabilizes the LEM while the remainder of the spacecraft (CM and SM) separates from the LEM, turns around and docks, nose to nose, with the LEM. At this point the spacecraft

[redacted]

Table 12-2. Saturn IB-Apollo Mission Objectives and Flight Data

(To be supplied at a later date.)



3-8A

Figure 12-2. Typical Saturn IB-Apollo Mission Profile separates from the S-IVB/IU and the planned mission exercies are performed by the crew. Upon completion of the mission exercises, the LEM and SM are jettisoned and the CM re-enters the earth atmosphere and is recovered. For a detailed listing of mission events refer to Table 12-3.

Table 12-3. Description of Typical Saturn IB - Apollo Mission

*Event No.	Approx. Time After Liftoff (Sec.)	Event
1	0	Liftoff of Saturn IB - Apollo space vehicle (SV) from AMR launch complex No. 37A or 37B.
		Start roll to align SV pitch plane with flight azimuth. Start time tilt. (By launch vehicle (LV) systems.)
		Arrest roll (SV correctly aligned with flight azimuth).
		Activate accelerometer control of LV guidance and control system.
		Deactivate accelerometer control of LV guidance and control system.
		Arrest time tilt.
	144.8	Shut down inboard first-stage (S-IB stage) engines.
	150.8	Shut down outboard first-stage engines, beginning staging period. Start timing for stage separation sequence.
		Ignite second-stage (S-IVB stage) ullage motors.
2		Separate first stage from second stage. Transfer control functions from first to second stage. Ignite first-stage retromotors.
3		Start second-stage engine, ending staging period.
4		Jettison Launch Escape System from Apollo spacecraft (SC).
		Jettison second-stage ullage motors.
		Start Path Guidance Mode.

*No. Refers to Figure 12-2. (Major events indicated only)

Table 12-3 Description of Typical Saturn IB - Apollo Mission (Cont'd)

*Event No.	Approx. Time After Liftoff (Sec.)	Event
5	620.8	Inject SC into 105-naut. mi. (194-km) circular earth orbit. Shut down second-stage engine.
		Continue orbital coast of SC. Perform scheduled mission exercises. For example:
		Check out crew and equipment.
6		Separate spacecraft CSM from spacecraft LEM, instrument unit and second stage (LEM/IU/S-IVB).
7		Jettison spacecraft Adapter and initiate turnaround of CSM.
8		Dock CSM to LEM/IU/S-IVB.
9		Jettison instrument unit and second stage, ending LV mission.
10		Transfer two members of SC crew to LEM ascent stage. (Third man remains in CM.)
		Check out LEM crew and equipment. Perform planned mission exercises.
		Return LEM crew to CM.
11		Jettison LEM ascent stage from CSM.
12		Jettison SM from CM.
13		Orient CM in re-entry attitude (heat shield forward).
14		Initiate CM re-entry.
15		Re-enter earth's atmosphere.
16		Deploy drogue parachute.
17		Jettison drogue parachute and deploy main parachutes.
18		Alight on surface of earth (on land).

*No. Refers to Figure 12-2. (Major events indicated only)

The mission of the launch vehicle ends with the final separation of the Apollo spacecraft from the S-IVB/IU, event number 9 of the mission profile. The launch vehicle mission can be divided into prelaunch, launch, ascent and orbital phases. For this description these phases are defined by the following limits:

a. Prelaunch - From start of stage testing to start of countdown.

b. Launch - From start of countdown to liftoff.

c. Ascent - From liftoff to orbit injection.

d. Orbital - From orbit injection to final payload separation.

12-4. LAUNCH VEHICLE REQUIREMENTS.

The Saturn IB launch vehicle is required to inject an Apollo spacecraft payload of 34,000 pounds into a 105-nautical mile circular earth orbit. To accomplish this, the launch vehicle must boost the payload to altitude, guide it so that the final flight path is 90 degrees (with respect to local vertical) and impart to it a final velocity of 25,563 ft/sec.

After injection into circular orbit, the launch vehicle is required to stabilize the LEM during the CSM turnaround and docking maneuver. Performance of the orbital coast mission requires a total life time of 4.5 hours for the S-IVB/IU systems.

The vehicle is subject to the following constraints:

a. Launch site (Cape Kennedy) latitude of 28 degrees, 30 minutes which introduces a minimum orbital inclination of the same degree.

b. Launch facility, VLF 37, requires a launch azimuth of 90 degrees.

c. Vehicle visibility for tracking and telemetry networks restricts azimuth path to a sector from 70 degrees to 110 degrees.

d. Range safety limits flight azimuth to a sector from 45 degrees to 110 degrees.

The primary vehicle requirements are accomplished by systems described in this chapter as astrionics, structures, propulsion, mechanical, and ground support equipment. Tables 12-4 through 12-7 list the basic requirements of each of these systems for the four phases of the launch vehicle mission. The time function indicated in the table is not to scale as it is intended to indicate only relative phasing of the requirements. Although the table is primarily a listing of system requirements, specific major events are included to show their relationship to the requirement.

Detailed information on the systems is presented in sections XIII through XVII. Inboard profiles of each stage are included in section XVIII.

Table 12-4. Saturn IB Requirements, Prelaunch Phase

SYSTEM/FUNCTION \ EVENT	Begin Stage Testing	Pad Erection	Start Countdown
Astrionics			
Command			
Control Checkout Sequence		- - - - - - - -	
Control Ground Support Activity		- - - - - - - -	
Communications			
Transmit Information for Checkout		- - - - - - - -	
Transmit Mission Planning Information		- - - - - - - -	
Instrumentation			
Provide Vehicle System Information in Support of Checkout		- - - - - - - -	
Checkout			
Provide Stimuli and Comparison Networks for Checkout		- - - - - - - -	
Guidance [1]			
Attitude Control and Stabilization [1]			
Tracking [1]			
Range Safety [1]			
Crew Safety [1]			
Electric Power			
Provide Ground Power for System Checkout		- - - - - - - -	

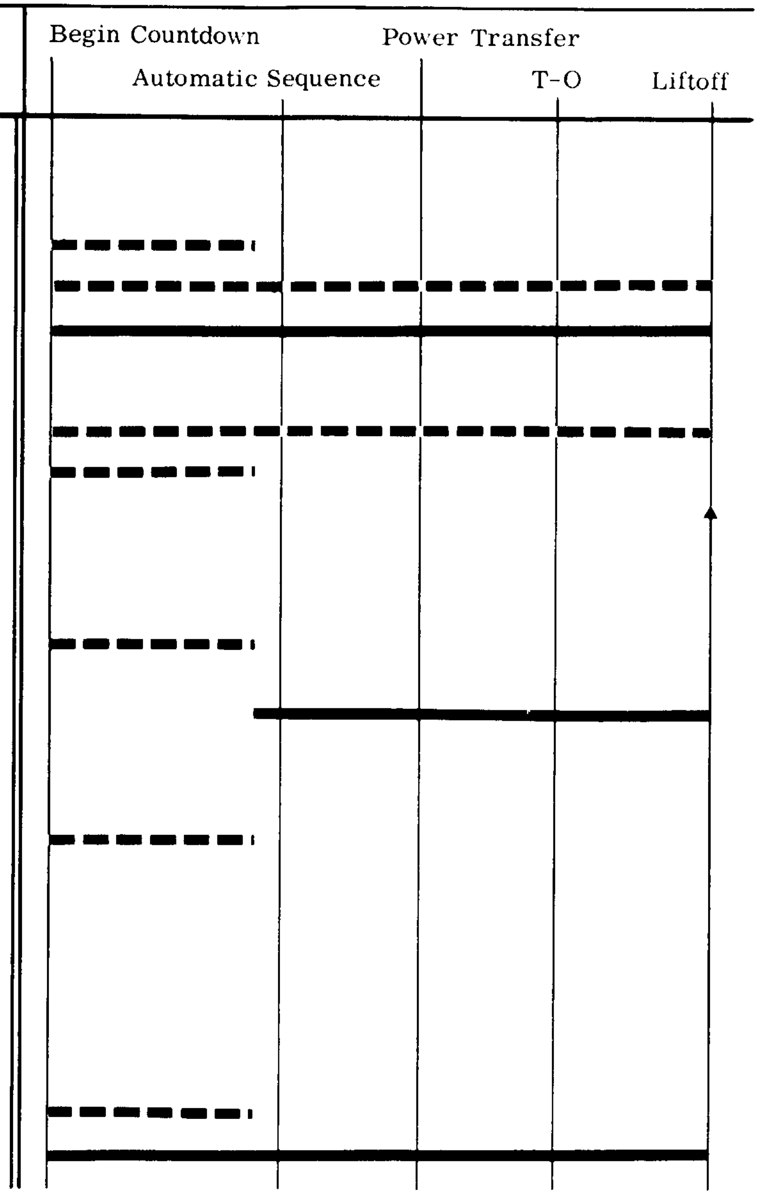
Table 12-4. Saturn IB Requirements, Prelaunch Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT	Begin Stage Testing – Pad Erection	Pad Erection – Start Countdown
Structures		
Provide Support for all other Systems	Operating	Operating
Withstand Ground Handling Loads	Operating	
Withstand Ground Wind Loads		Operating
Propulsion [1]		
Mechanical Systems		
Environmental Control		
Provide Cooling and Humidity Control to Vehicle Electronic Compartments		Intermittent Operation
Provide Supplemental Cooling to IU/S-IVB		Intermittent Operation
Engine Gimballing [1]		
Separation [1]		
Ordnance [1]		
Platform Gas Bearing Supply		
Supply Pressurized GN_2 to Stable Platform Bearing During Checkout		Intermittent Operation
Ground Support Equipment		
Provide Check of Stage Systems	Intermittent Operation	
Provide Ground Handling and Transportation	Intermittent Operation	
Provide Check of Vehicle Systems		Intermittent Operation

Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 12-5. Saturn IB Requirements, Launch Phase

SYSTEM/FUNCTION — EVENT



Astrionics

- Command
  - Control Checkouts Events
  - Control Ground Support Activity
  - Control Vehicle System Sequences
- Communications
  - Transmit Mission Information
  - Transmit Information for Checkout
  - Transmit Liftoff Time Reference
- Instrumentation
  - Provide Vehicle System Information for Checkout
  - Provide Real Time Data for Monitoring Vehicle Performance
- Checkout
  - Provide Stimuli and Comparison Networks for Checkout
- Guidance [1]
- Attitude Control and Stabilization [1]
- Tracking [1]
- Range Safety
  - Control Hazardous Operations
  - Clear Down Range Area

Table 12-5. Saturn IB Requirements, Launch Phase (Cont'd)

SYSTEM/FUNCTION — EVENT

<u>Astrionics (Cont'd)</u>

Crew Safety

Monitor Vehicle Conditions with Capability for Abort in Emergency

Electric Power

Provide Ground Power for Checkout and Ground Operations

Provide Power for Flight Operations

<u>Structures</u>

Provide Support for all Other Systems

Withstand Ground Winds

Withstand Propellant Pressurization Loads

Withstand Engine Thrust Loads

Withstand Holddown Loads

Holddown Release [2]

<u>Propulsion</u>

RP-1 Loaded [2]

LOX Loaded [2]

LH_2 Loaded [2]

Propellants Pressurized [2]

S-IB Stage Ignition Sequence Started [2]

Provide Engine Thrust

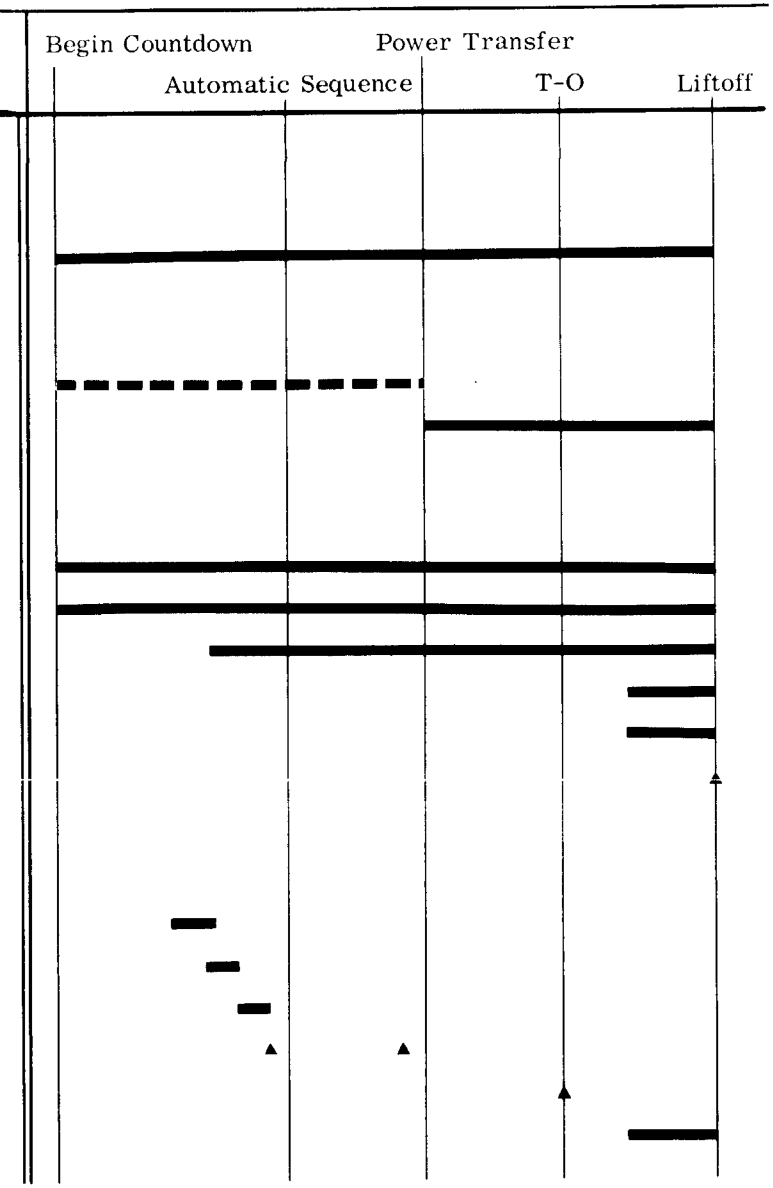
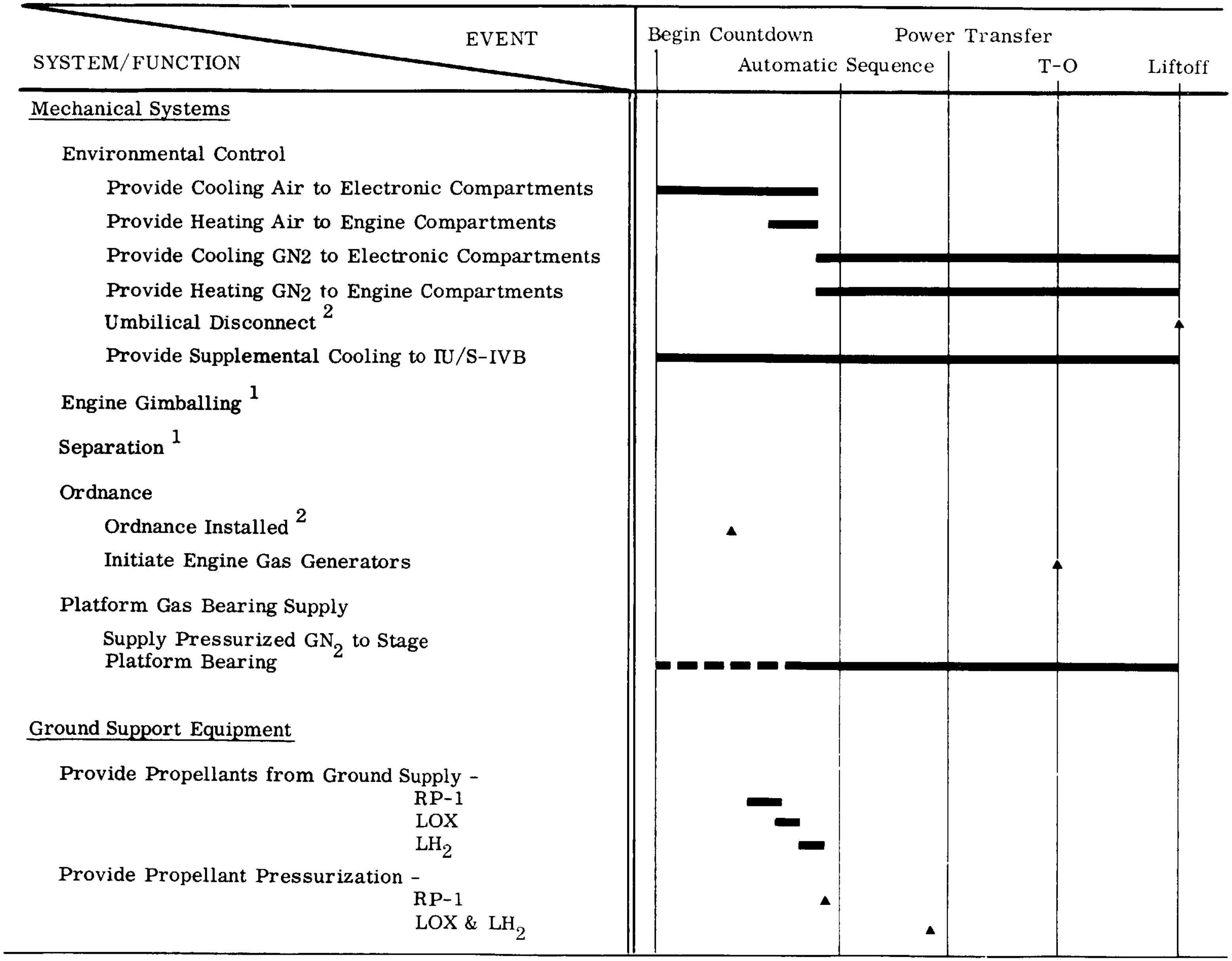


Table 12-5. Saturn IB Requirements, Launch Phase (Cont'd)



Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 12-6. Saturn IB Requirements, Ascent Phase

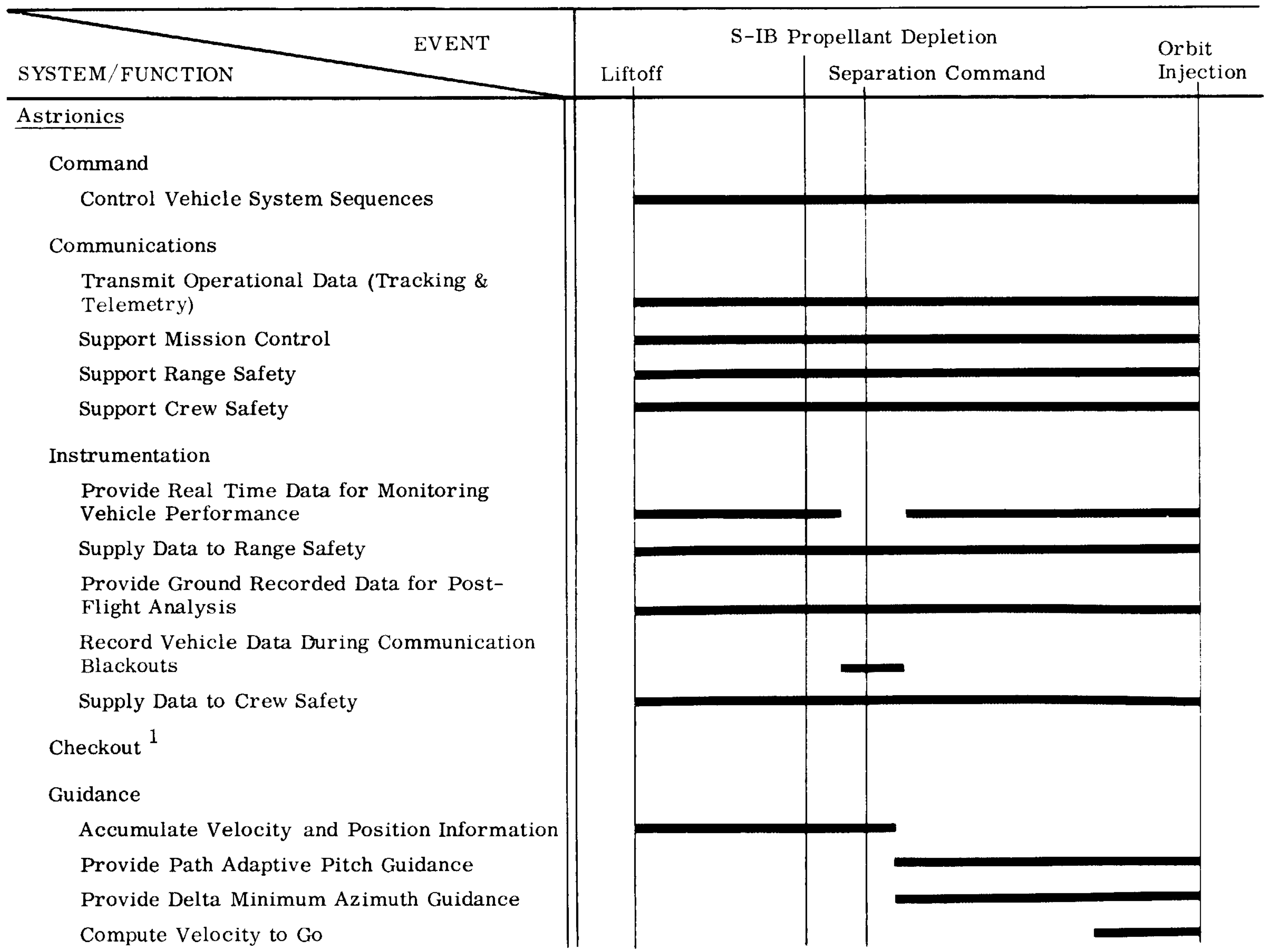


Table 12-6. Saturn IB Requirements, Ascent Phase (Cont'd)

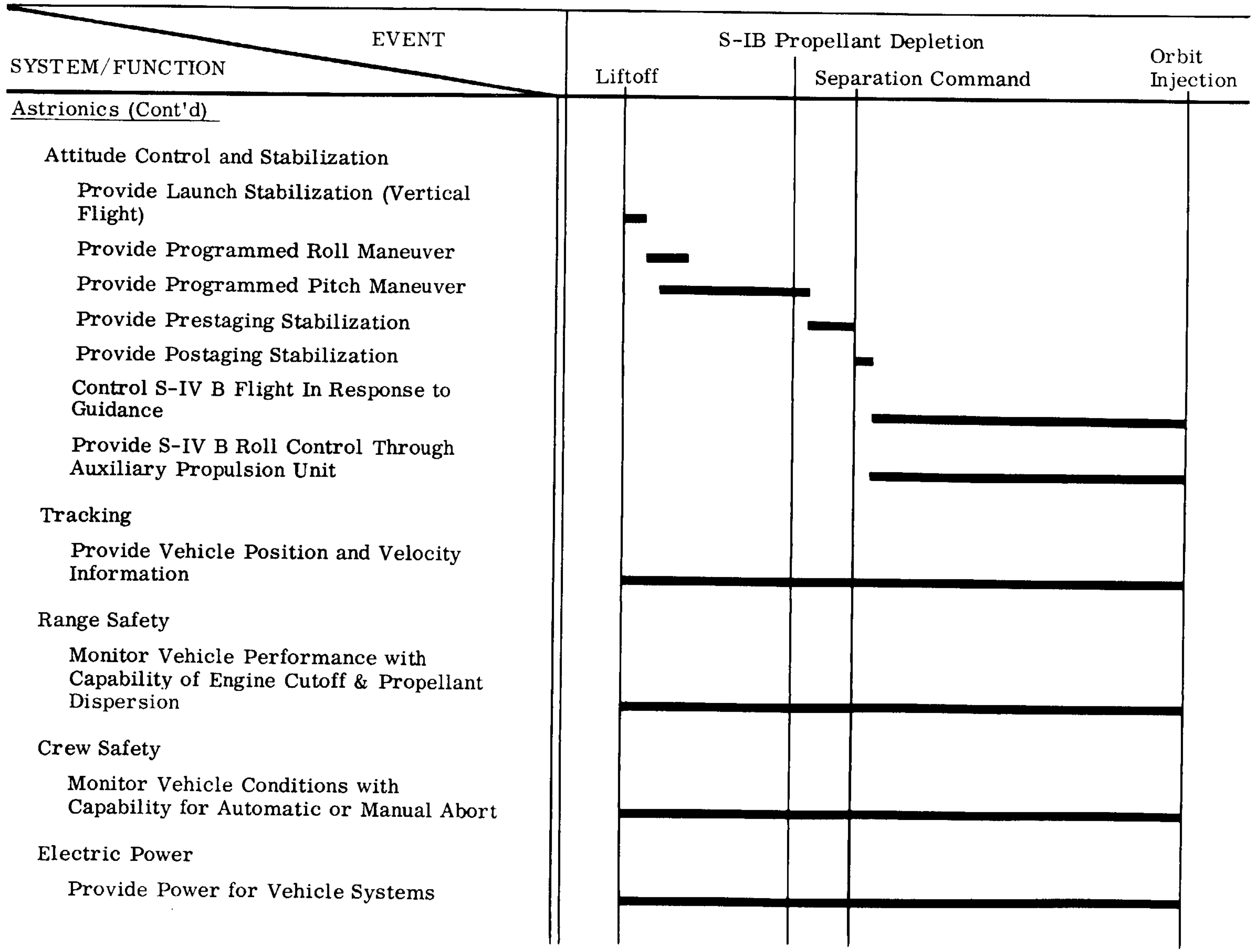


Table 12-6. Saturn IB Requirements, Ascent Phase (Cont'd)

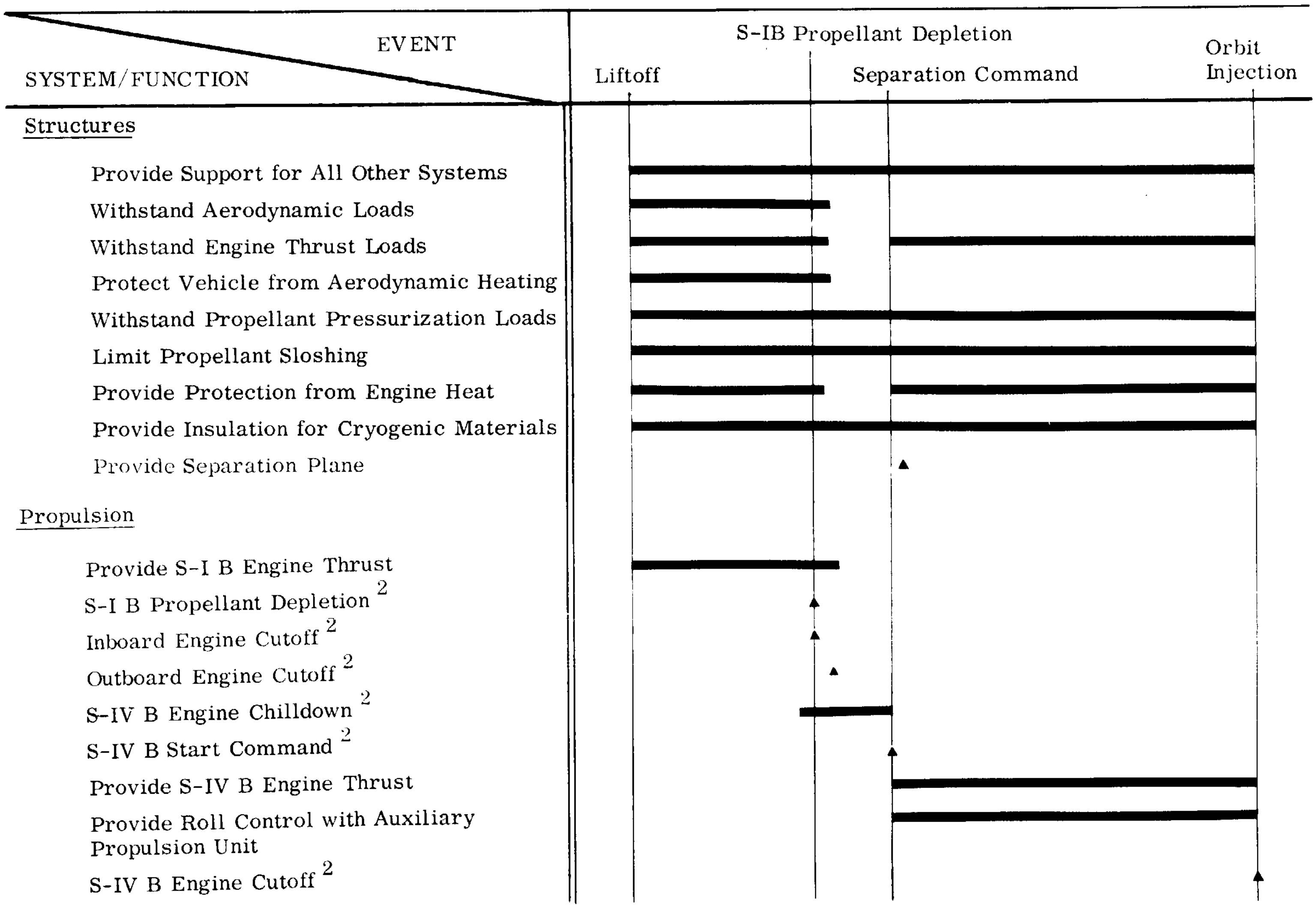
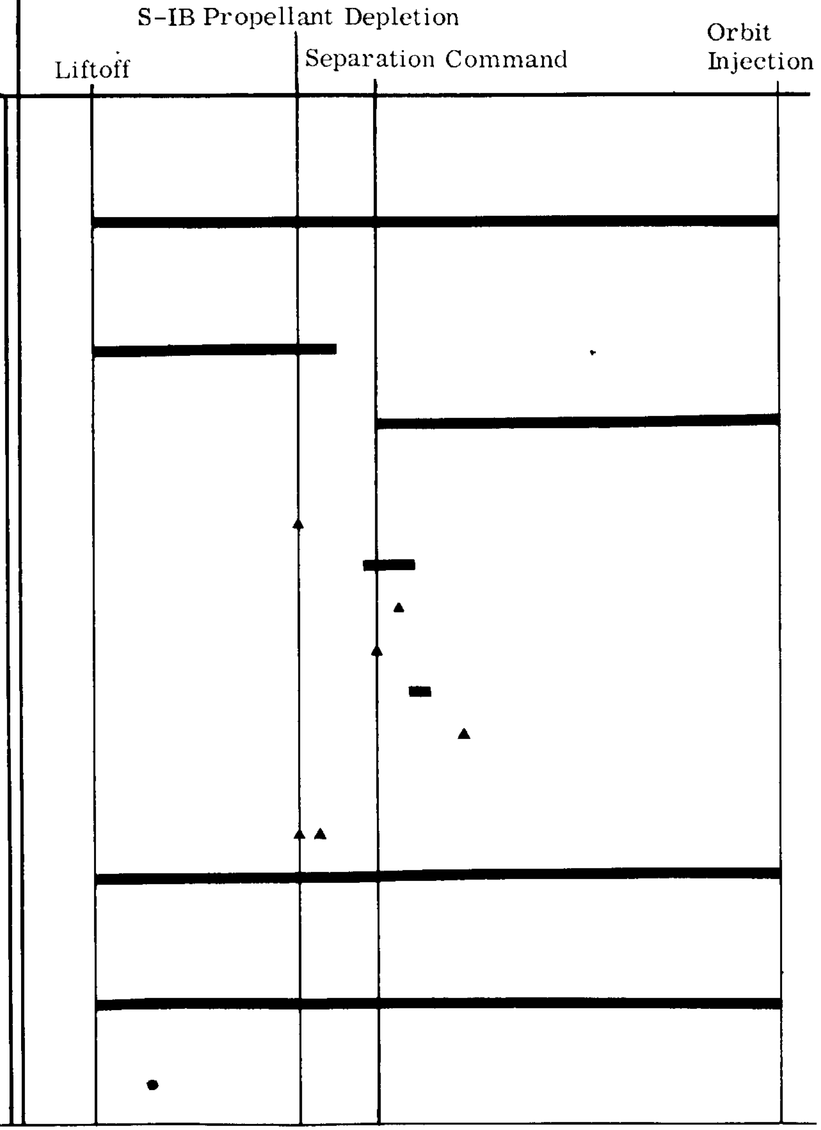


Table 12-6. Saturn IB Requirements, Ascent Phase (Cont'd)



SYSTEM/FUNCTION — EVENT
Mechanical Systems
Environmental Control
Provide Cooling to IU/S-IVB
Engine Gimballing
Position S-IB Outboard Engines in Response to Control Signals
Position S-IV B Engine in Response to Control Signals
Separation System
Initiate LOX/SOX Disposal
Initate S-IVB Ullage Rocket Burn
Actuate Separation Nuts
Transfer Engine Gimbal Control
Initate S-I Retromotor Burn
Jettison Ullage Rocket
Ordnance
Actuate Conax Valves
Propellant Dispersion Capability Active
Platform Gas Bearing Supply
Supply Pressurized GN_2 to Stable Platform Bearing
Ground Support Equipment[1]

Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 12-7. Saturn IB Requirements, Orbital Phase

SYSTEM/FUNCTION
Astrionics
Command
Control Vehicle System Sequences
Communications
Transmit Operational Data
Support Mission Control
Support Crew Safety
Instrumentation
Provide Real Time Data for Monitoring Vehicle Performance
Supply Data to Crew Safety
Provide Ground Recorded Data for Post Flight Analysis
Record Vehicle Data During Communication Blackouts
Checkout [1]
Guidance [1]
Attitude Control & Stabilization
Provide Stabilization in Response to Stable Platform, Horizon Sensor, or Apollo Spacecraft
Tracking
Provide Vehicle Position and Velocity Information
Range Safety [1]

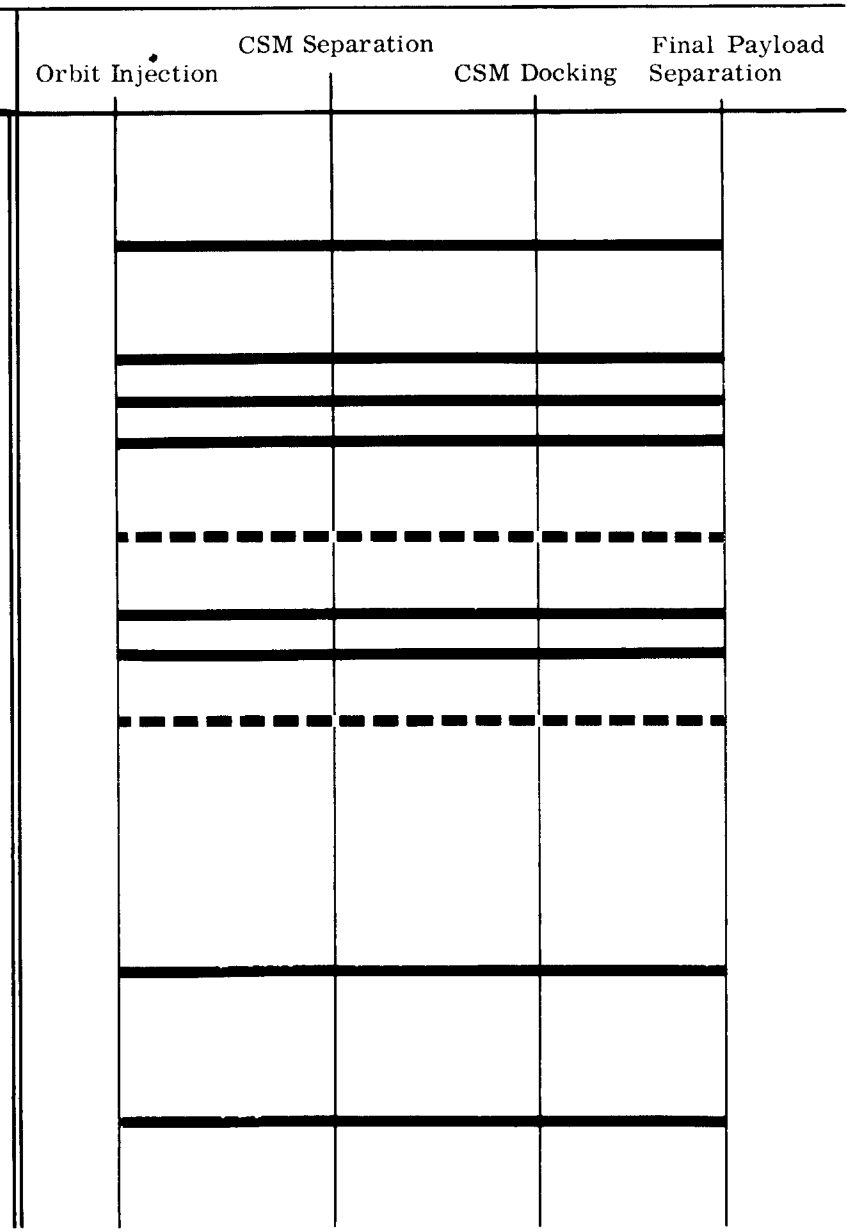


Table 12-7. Saturn IB Requirements, Orbital Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT	Orbit Injection	CSM Separation	CSM Docking	Final Payload Separation
<u>Astrionics (Cont'd)</u>				
Crew Safety				
Monitor Vehicle Conditions with Capability for Manual Abort				
Electric Power				
Provide Power for Vehicle Systems				
<u>Structures</u>				
Provide Support for all other Systems				
<u>Propulsion</u>				
Provide Stabilizing Thrust in Response to Attitude Control Signals				
<u>Mechanical Systems</u>				
Environmental Cooling				
Provide Cooling to IU/S-IVB				
Engine Gimballing [1]				
Separation System [1]				
Ordnance [1]				

Table 12-7. Saturn IB Requirements, Orbital Phase (Cont'd)

SYSTEM/FUNCTION / EVENT	Orbit Injection	CSM Separation	CSM Docking	Final Payload Separation
Mechanical Systems (Cont'd)				
Platform Gas Bearing Supply				
Supply Pressurized GN_2 to Stable Platform Bearing				
Ground Support Equipment [1]				

Legend: [1]Inactive: [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

CHAPTER 3

SECTION XIII

ASTRIONICS

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SECTION XIII.

ASTRIONICS

13-1. GENERAL.

The Astrionics system provides the electrical and electronic functions required for Saturn IB. The functions, listed below and described in the following paragraphs, are accomplished utilizing both vehicle and ground based subsystems.

a. Command - Performs management of Saturn systems by initiating all operational events and sequences. The issuance of commands is dependent on time and events.

b. Communication - Transfers intelligence within and among the Saturn systems. This intelligence is in four forms: voice, digital, discrete, and analog signals.

c. Instrumentation - Monitors the performance of launch vehicle systems to acquire operational and engineering appraisal data.

d. Checkout - Provides assurance during the launch phase that the launch vehicle is capable of performing its assigned mission.

e. Guidance - Provides steering and thrust cutoff commands to adjust the vehicle motion in a manner leading to mission accomplishment.

f. Attitude Control and Stabilization - Provides signals to the engine gimballing system to maintain a stable launch vehicle motion and adjusts this motion in accordance with guidance commands.

g. Tracking - Obtains and records the launch vehicle's position and velocity during flight.

h. Crew Safety - Ensures safety of the astronauts in the event of a malfunction in the Saturn/Apollo vehicle.

i. Range Safety - Ensures that life and private property are not endangered in the event of a vehicle malfunction during the ascent and orbital phase.

j. Electrical System - Supplies and distributes the electrical power required for vehicle operation.

13-2. COMMAND.

The Saturn IB command function is similar to that of Saturn V. (Refer to Paragraph 20-2.)

13-3. COMMUNICATIONS.

The Saturn IB communication function is similar to that of Saturn V. (Refer to Paragraph 20-11.)

Additionally, the Saturn IB/Apollo mission requires voice communications between earth and the CM. (Stations having this capability are listed in the "capsule communications" column of Table 6-1.)

13-4. INSTRUMENTATION.

Saturn IB instrumentation collects status and operational data from the launch vehicle and makes this data available to other functions of the Saturn system to aid them in carrying out their part in the mission.

Instrumentation is initially activated during checkout in the prelaunch phase and remains active until end of mission. The many tasks assigned to instrumentation can be grouped in three major areas: checkout support, in-flight data collection, and data recording for post-flight analysis.

During the prelaunch phase, instrumentation is used in checking out the complete launch vehicle and its stages. The checkout is performed utilizing automatic systems controlled by digital computers. Instrumentation supplies all significant vehicle data in the format which is compatible with that of the checkout systems.

From liftoff, when all physical connections between the vehicle and ground are severed, until the end of the mission, instrumentation provides the vehicle-to-ground data link. Since this is the only means of obtaining vehicle operational information, the instrumentation must be highly reliable. All data received during this portion of the mission is recorded for post-flight analysis.

Vehicle performance data falls into two categories; engineering data and operational data. Engineering data includes parameters such as temperature, acceleration, vibration, and stress; operational data includes vehicle computer commands and event sequences such as those associated with first stage cutoff, stage separation or second stage ignition.

The tentative parameters and number of measurements to be obtained for each stage of the SA-202 launch vehicle are listed in Table 13-1. Requirements for measurements are expected to decrease on subsequent flights.

Table 13-1. Measuring Program for SA-202

Parameters	S-IB	S-IVB	Instrument Unit
Temperature	76	104	60
Pressure	73	54	15
Strain and Vibration	118	48	29
Flight Mechanics	9	70	19
Discrete Signals	31	26	7
Voltage, Current and Frequeny	10	30	19
Miscellaneous	32	34	12
Guidance and Control	-	-	65
RF and Telemetry	-	-	55

13-5. OPERATION

The Saturn IB instrumentation is comprised of measuring, telemetry, antenna, and ground recording systems. The operation of these systems is similar to that of the Saturn I Block II vehicle. (Refer to Paragraph 6-12).

13-6. IMPLEMENTATION

The Saturn IB stages (S-IB and S-IVB) and the instrument unit contain independent instrumentation systems. The configuration and number of system components vary depending on the objective of the mission. Complexity of the launch vehicle and its missions requires a large number of measurements, particularly in the early flights of the program. The requirements decrease on later flights.

The Saturn IB launch vehicle utilizes the following types of telemetry systems.

a. PCM/FM/FM

b. PAM/FM/FM

c. SS/FM

13-7. CHECKOUT.

The Saturn IB checkout function is similar to that of Saturn V. (Refer to Paragraph 20-28.)

13-8. ATTITUDE CONTROL AND STABILIZATION.

The Saturn IB attitude control and stabilization function is similar to that of Saturn V. (Refer to Paragraph 20-35.) The Saturn IB, S-IB stage utilizes electrical feedback in the engine gimballing system. This requires a minor change in the Saturn IB control computer.

13-9. GUIDANCE.

The Saturn IB guidance function is similar to that of Saturn V. (Refer to Paragraph 20-41.)

13-10. TRACKING.

The tracking function obtains vehicle position and velocity information from Saturn IB missions. As an extension of the development program of Saturn I, the Saturn IB tracking function contributes toward the goal of perfecting the Apollo Ground Operational Support System (GOSS) to support the ultimate manned lunar mission.

13-11. OPERATION.

The operation of the Saturn IB tracking function is similar to that of Saturn I. (Refer to Paragraph 6-51.) The tracking systems used in the Saturn I missions are used for tracking the Saturn IB vehicles. An additional system, the airborne range and orbit determination (AROD) system, is implemented with airborne and earth-based equipment for Saturn IB tracking.

13-12. IMPLEMENTATION.

Radio frequency equipment carried aboard the Saturn IB instrument unit is integrated with earth-based equipment to provide the position and velocity data for mission control and post-flight evaluation of the mission. The radio frequency tracking systems include:

a. AZUSA
b. ODOP
c. MISTRAM
d. Minitrack
e. C-Band Radar
f. Radar Altimeter
g. AROD

All of these systems except AROD are operational for the Saturn IB program. The AROD system is a developmental system. The systems are described below.

13-13. AZUSA. This system is the same as used for Saturn I. (Refer to Paragraph 6-52.)

13-14. ODOP. The offset doppler (ODOP) system became operational during the Saturn I program. A description of ODOP is presented in Paragraph 6-53.)

13-15. MISTRAM. The missile trajectory measurement (MISTRAM) system is operational on Saturn IB. The description of MISTRAM (passenger equipment on Saturn I) is given in Paragraph 6-54.

13-16. Minitrack. A Minitrack beacon is carried aboard the Saturn IB instrument unit. The beacon is a self-contained transmitter radiating a continuous-wave signal at a frequency of 139.65 mc. Earth-based stations determine direction to the vehicle as a function of time through comparison of phases of the beacon signals received at antenna pairs on crossed baselines. Refer to Paragraph 6-55 for a more detailed description of the Minitrack system.

13-17. C-Band Radar. The SST-102A C-band radar transponder aboard the Saturn IB instrument unit functions with earth-based radar installations to provide position and velocity information on the Saturn IB vehicles. C-Band tracking, described in Paragraph 6-55 for the Saturn I, is applicable to the Saturn IB.

13-18. Vehicle Radar Altimeter. The high altitude radar altimeter, used on Saturn I missions, is also operational on the Saturn IB. Refer to Paragraph 6-56 for a description of the radar altimeter.

13-19. AROD. The airborne range and orbital determination (AROD) system is being developed on the Saturn IB program. It is expected to solve the problems of tracking vehicles over long expanses of water and provide a more economical means of establishing additional ground stations to provide greater tracking coverage of orbiting vehicles.

AROD is similar in principle to ODOP, but is inverted in the sense that the transmitter is carried on the Saturn IB instrument unit, with transponders located at ground stations. The transmitter radiates a continuous-wave radio frequency signal, modulated to provide resolution of ambiguity in range measurement. Transponders located on the ground receive the transmitted signal, offset it in frequency and re-transmit it to the vehicle. Vehicle-borne equipment measures the phase delay between transmitted and received signals to determine range between a ground station and the vehicle. Radial velocity of the vehicle with respect to the ground station is determined by the doppler shift in the received signal.

Computation of vehicle position and velocity requires simultaneous measurements to at least three ground stations. The on-board equipment is capable of tracking four ground stations simultaneously.

Figures 13-1 and 13-2 illustrate the AROD components on board the vehicle and at ground stations, respectively.

Unmanned transponder stations can be used for the AROD tracking system. A VHF command transmitter on the vehicle turns ground stations on and off as the vehicle passes over. Each ground station transmits an identification code, enabling the system to select station location data stored in the vehicle computer. Each station transponder transmits at a frequency matching one of the four channels of the AROD on-board tracking receiver.

Outputs of the on-board AROD system are in digital form. They may be either transmitted by telemetry to ground stations for trajectory computation or delivered

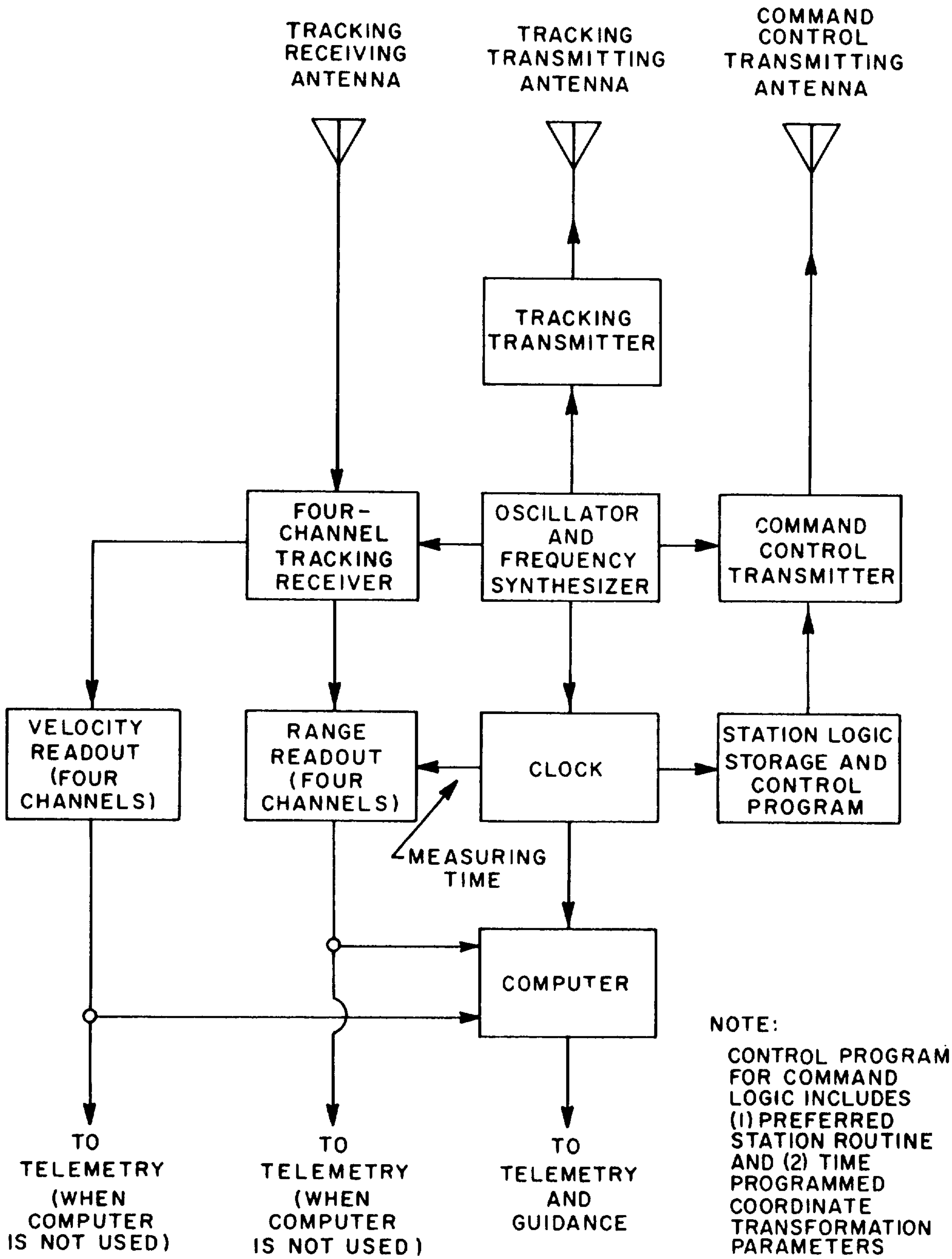

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Figure 13-1. AROD Onboard Equipment

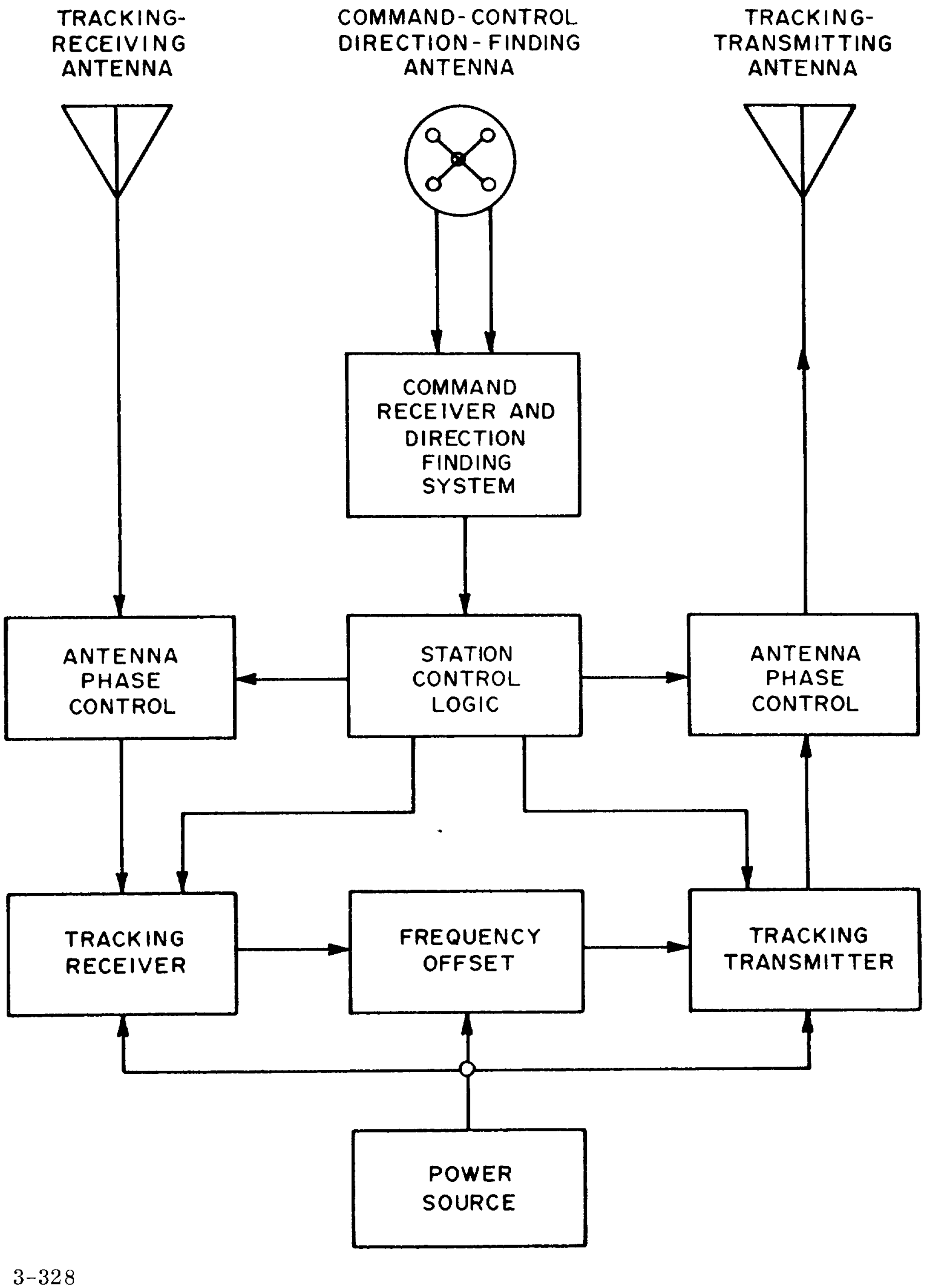

Figure 13-2. AROD Transponder Ground Station

to the vehicle guidance computer for navigational use. Nominal characteristics of the AROD systems are listed in Table 13-2.

Table 13-2. Characteristics of the AROD System

Item	Characteristic
Vehicle Equipment	
Transmitter Frequency	2276 mc
Power Output	20 watts
Ground Station	
Transponder Frequency	2214 mc
Power Output	100 watts
Accuracies	
Range	10 ft
Velocity	0.2 ft/sec

13-20. CREW SAFETY (VEHICLE EMERGENCY DETECTION SYSTEM)

The crew safety function ensures safety of the spacecraft crew in event of malfunction of the Saturn IB launch vehicle.

Requirements of the function are generally the same as for crew safety on the Saturn V launch vehicle. (Refer to Paragraph 20-94.) The Saturn IB vehicle emergency detection system provides signals for automatically initiating the escape sequence for:

a. Structural failure
b. Excessive turning rate in roll, pitch or yaw
c. Loss of thrust of two or more engines on S-IB stage

Performance parameters which are sensed and displayed for crew decision for manual initiation of the escape sequence are:

a. Thrust status of engines on active stage
b. Staging sequence
c. Status of vehicle digital computer and data adapter
d. Angle-of-attack
e. Three-axis angular rates of the spacecraft
f. Excessive turning rate in roll, pitch or yaw.
g. Spacecraft attitude error
h. Engine cut-off for range safety purposes

13-21. OPERATION.

The Saturn IB crew safety operational philosophy is similar to that of Saturn V. (Refer to Paragraph 20-95.)

13-22. IMPLEMENTATION.

The Saturn IB vehicle emergency detection system is illustrated in Figure 13-3. The VEDS consists of sensors in the stages and instrument unit and a distributor in the instrument unit which transfers vehicle performance information to display equipment in the CM. Implementation of the system is described in relation to the parameters sensed for automatic and manual initiations of the escape sequence.

13-23. <u>Structural Failure.</u> Structural integrity of the Saturn IB launch vehicle is monitored by "hot wire" circuits installed in three geographical paths from the instrument unit down the S-IVB and S-IB stages. Three circuits are installed in each geographical path. Loss of power in two of the three circuits in any geographical path causes an abort signal output from the VEDS distributor to the CM.

13-24. <u>Excessive Turning Rate.</u> Vehicle turning rates in roll, pitch and yaw are sensed by a rate gyro package in the instrument unit. The package contains three gyros which sense rates in each plane. When an individual gyro senses a rate in excess of a predetermined limit, a rate switch is closed, actuating a relay. Actuation of any two of the three relays associated with an axis provides an output to the VEDS distributor. The VEDS distributor transfers a signal to the CM, where it actuates an over-rate light on the display panel. The signal also initiates an

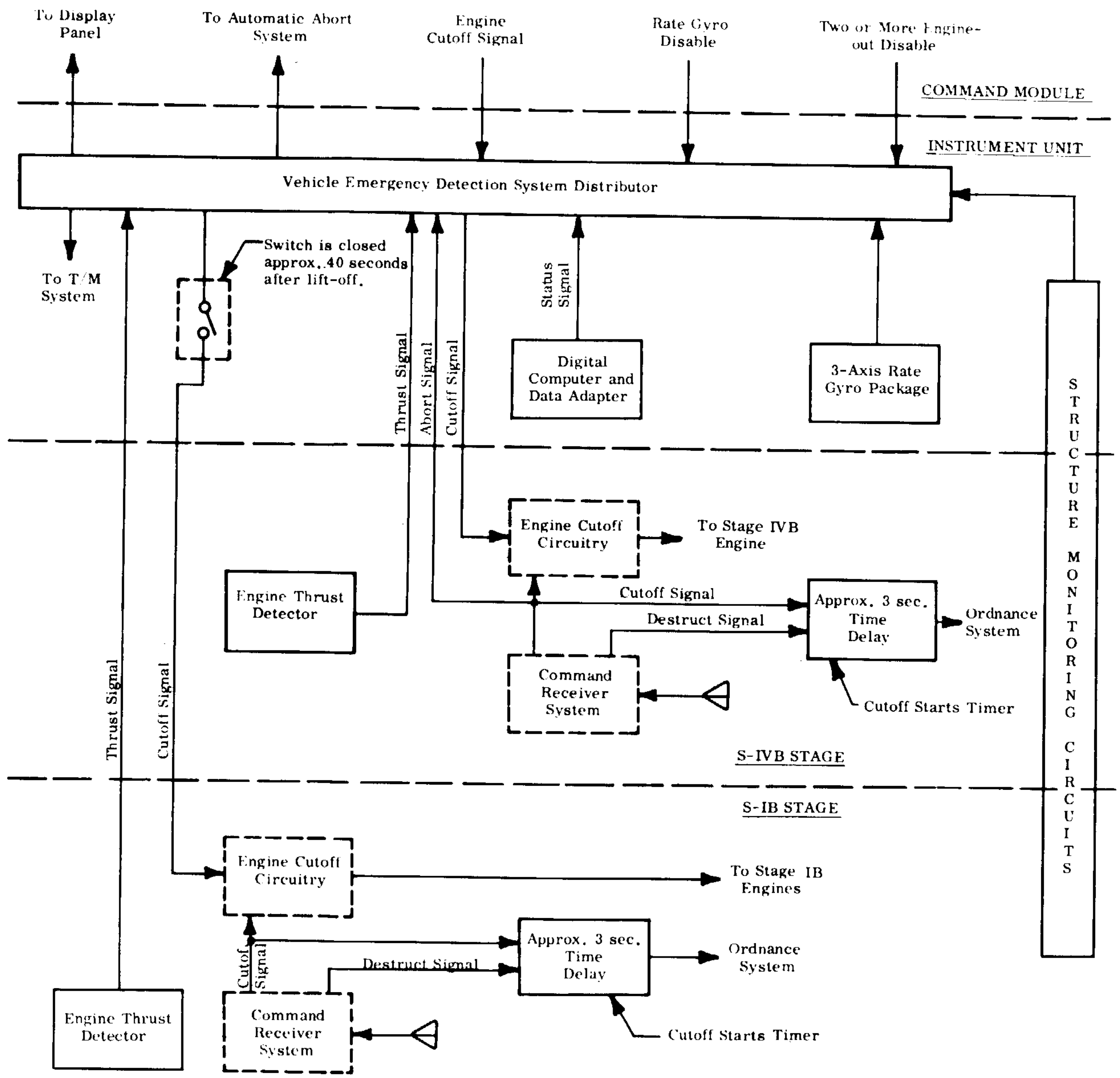

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Figure 13-3. Vehicle Emergency Detection System automatic abort if the excessive rate occurs before the automatic feature is disabled. Disabling of the automatic abort feature can be controlled by the crew through a single switch in the spacecraft. Disabling is also controlled separately for roll and pitch (yaw combined) through event sequencing by the vehicle digital computer. Disabling times are established in planning for the mission.

After the automatic abort feature is disabled, excessive rate becomes a parameter for manual abort procedures.

A thrust detector, generating a discrete signal on loss of thrust, is installed on each engine of S-IB stage. Outputs of the thrust detectors are routed to the VEDS distributor in the instrument unit. From the distributor, the thrust information is sent to the CM for display by engine status lights and to a logic circuit which has an output if thrust is lost by two or more engines. The logic circuit output is delivered through the distributor to the CM for automatic activation of the LES during the early moments of flight. This automatic feature can be disabled by the crew by a switch in the spacecraft. Disabling of this automatic feature is also accomplished by event sequencing command of the vehicle digital computer, at a time established in planning of the mission.

Engine status (both stages) is also a parameter for manual abort. The manual abort for loss of thrust is governed by rules established for the individual mission.

13-25. <u>Staging Sequence.</u> Failure of S-IB/S-IVB separation is a basis for crew decision to initiate the escape sequence.

Separation of the stages will be indicated by the S-IB stage engine status lights.

13-26. <u>Digital Computer and Data Adapter Status.</u> A signal from the data adapter in the instrument unit is delivered to the VEDS distributor when the digital computer and data adapter are operating improperly. The distributor delivers a signal to the command module to trigger a light indicating this malfunction, a basis for crew decision to initiate the abort procedure.

13-27. <u>Angle of Attack.</u> Angle-of-attack is displayed in analog form in the spacecraft as information for manual abort decision.

13-28. Spacecraft Angular Rates. Analogs of spacecraft angular rates about three axes are presented on the CM flight director attitude indicator as an aid to decision for manual abort.

13-29. Spacecraft Attitude Error. Errors in spacecraft attitude will be displayed on the flight director attitude indicator. During S-IB stage flight, the attitude display will be compared with the vehicle tilt program for crew information and decision on abort.

13-30. Engine Cutoff for Range Safety Purposes. Whenever engine cutoff is commanded for range safety purposes, a signal is delivered to the VEDS distributor from the command receivers on S-IVB stage. The distributor, in turn, transfers the engine cutoff signal to the CM to warn the crew of possible initiation of propellant dispersion ordnance after a three second time delay. The crew initiates abort manually (unless the range safety command occurs during the time when loss of engine thrust causes abort automatically.)

13-31. RANGE SAFETY.

The Saturn IB range safety function requirements are similar to those of Saturn I. (Refer to Paragraph 6-58.) The primary differences between the Saturn IB and Saturn I range safety are in implementation. These differences are described below.

The command receivers of the S-IVB stage supply an engine cutoff signal to the vehicle emergency detection system distributor if flight termination is commanded. The signal is used for crew safety which is not implemented on Saturn I.

In addition, an ordnance interface is provided between stages of the Saturn IB to ensure that initiation of propellant dispersion ordnance of one stage is transmitted to the other, increasing the reliability of the system. (Refer to Paragraph 16-23 for a description of the propellant dispersion ordnance.)

13-32. ELECTRICAL SYSTEM.

The two stages and instrument unit of the Saturn IB have independent electrical systems.

Except for number of components and power distribution differences, the Saturn IB systems are similar to those of Saturn I. (Refer to Paragraph 6-65.) Primary differences are:

a. The Saturn IB stages do not have a central source of 400 cps ac power.

b. Sequencing functions for the Saturn IB are performed by a switch selector and control distributor on each stage in response to digitally encoded commands from the digital computer. (This mechanization eliminates the flight sequencer and slave unit used on Saturn I.)

CHAPTER 3

SECTION XIV

STRUCTURES

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STRUCTURES

14-1. STRUCTURAL REQUIREMENTS.

The Saturn IB launch vehicle structure is designed to withstand all loads that can be expected to occur during ground handling, prelaunch, launch and flight operations. The structure also contains the propellant for the stages. The design requirements for the vehicle structure are determined after a careful analysis of the conditions that will be encountered during all operations.

14-2. GROUND HANDLING CONDITIONS.

Handling procedures and equipment are designed so that loads imposed on the structure during fabrication, transportation, and erection do not exceed flight loads and thus do not impose any flight performance penalty.

14-3. PRELAUNCH CONDITIONS.

The vehicle, empty or fueled, pressurized or unpressurized and free-standing (attached to the launcher only) is structurally capable of withstanding loads resulting from winds having a 99.9 percent probability of occurrence during the strongest wind month of the year. The bending moments (Figure 14-1) and shears resulting from the wind are combined with the longitudinal force due to the weight of the vehicle in defining the worst prelaunch loading condition.

14-4. LAUNCH CONDITIONS.

At launch the vehicle structure is capable of withstanding loads from two conditions, holddown and rebound. The holddown condition is imposed on the structure after engine ignition but before the launcher releases the vehicle. The holddown loads result from wind (bending moments and shears), engine thrust (forward axial load), vehicle inertia (aft axial load) and vibration transients due to initial engine combustion.

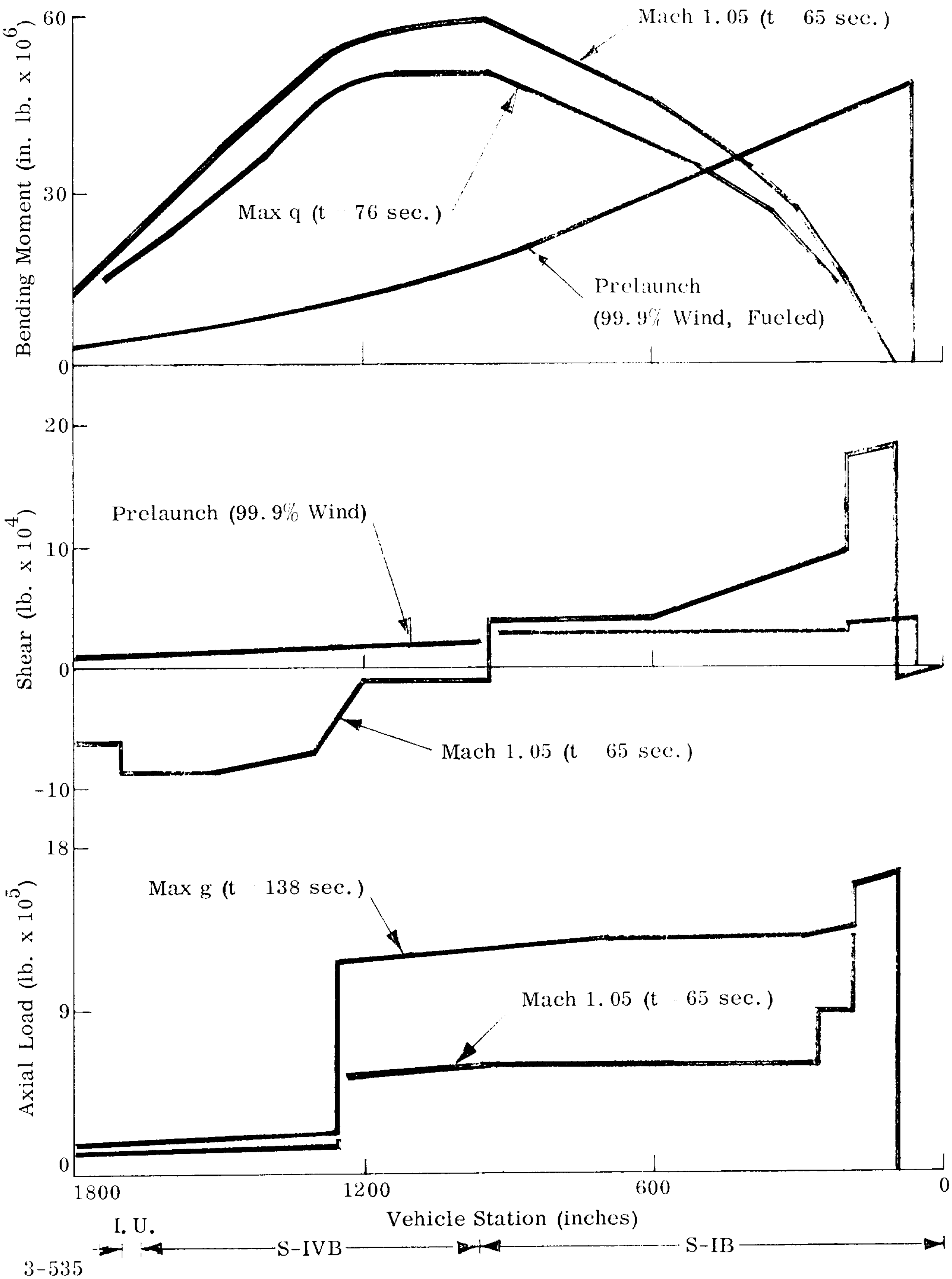

Figure 14-1. Saturn IB Loads

The rebound condition occurs when the engines are cut off before the launcher releases the vehicle. Axial loads result from deceleration of the vehicle which suddenly reverses the direction of the load at the holddown points. Combined with the axial loads are wind loads (bending moments) and vibration transients resulting from engine cutoff.

14-5. FLIGHT CONDITIONS.

During flight the structure is subjected to engine thrust and heat, dynamic, aerodynamic, inertia and propellant loads.

14-6. Engine Thrust and Heat Loads. The first stage thrust increases as the vehicle gains altitude, reaches a maximum at approximately 106 seconds after liftoff, and then decreases slightly prior to first stage engine cutoff. After stage separation, the second stage engines impose relatively constant thrust loads on the remainder of the vehicle. The thrust produces axial loads, shears and bending moments on the vehicle. The moments and shears are a result of the engines gimballing.

The first stage engines impose a heat load on the base of the vehicle through radiation and circulation of the exhaust gases. After separation the second stage engines impose a heat load on the base of the second stage.

14-7. Dynamic Loads. Vehicle dynamic loads result from external and internal disturbances. Three main sources of excitation - mechanical, acoustical and aerodynamic produce the vehicle vibration environment. The mechanical source begins at engine ignition and remains relatively constant until engine cutoff. The acoustical source begins with the sound field generated at engine ignition. It is maximum at vehicle liftoff and becomes negligible after Mach 1 (approximately 64 seconds after liftoff).

The aerodynamic loading begins as the vehicle velocity increases and is most influential during transition at Mach 1 and at maximum dynamic pressure. Transient vibrations, which are relatively high in magnitude and present only for short periods of time, occur during engine ignition, vehicle liftoff, Mach 1, region of maximum dynamic pressure, engine cutoff, and stage separation.

Propellant sloshing, another type of dynamic loading, results from a relative motion between the container and the center of gravity of the fluid mass and is generally caused by gust loads, control modes and vehicle bending modes. Reaction of the control system (gimballing engines) to gust loads produces considerable bending deflection in the vehicle structure. Since the structure and propellant are not integral and do not deflect together, sloshing results. If the propellant sloshing is not damped, compensation for the resulting perturbations must be provided by the control system.

14-8. Aerodynamic Loads. Aerodynamic loading is a result of drag, angle of attack and wind gusts. Aerodynamic drag increases to a maximum approximately 76 seconds after liftoff (max q condition) and then decreases to nearly zero before first stage burnout. Aerodynamic drag imposes an axial load on the structure and when combined with an angle of attack results in bending moments and shears. Two critical conditions result from aerodynamic loading, Mach 1.05 (approximately 66 seconds) and max q. When the vehicle is in the region of high drag, structural bending moments are minimized by the control system which reduces the vehicle angle of attack.

Aerodynamic heating on the vehicle is a result of friction caused by the vehicle moving through the atmosphere. The heating increases until first stage burnout and then decreases. Vehicle surfaces which are not parallel to the vehicle centerline have the greatest temperature increase during flight.

14-9. Inertia Loads. Inertia loads result from the vehicle acceleration due to an increase in the thrust/weight ratio during flight. Peak acceleration is at first stage cutoff (max g condition). The acceleration decreases at first and second stage separation and then increases during second stage burning, but never reaches the peak achieved at first stage cutoff.

14-10. Propellant Loads. The loads imposed on the structure by the propellant are due to a combination of hydrostatic head, and ullage and ambient pressures. The hydrostatic head, varying during flight, is a function of the density of the fluid, height of the fluid in the container and the acceleration of the vehicle. The ullage pressure is supplied by the pressurization system and is limited by relief valves.

As the altitude of the vehicle increases during flight, the ambient pressure decreases. At any time during flight (at any location in the container) the maximum pressure differential across the container wall is equal to the ullage pressure plus the hydrostatic head minus the ambient pressure.

14-11. STRUCTURAL DESIGN.

The Saturn IB launch vehicle consists of two stages joined by an interstage. An instrument unit mounted forward of the second stage provides the support for the spacecraft. Critical loading conditions for various portions of the vehicle occur at different times. The critical conditions occur on the S-IB structure during prelaunch (ground wind), launch (holddown and rebound) and flight (Mach 1.05, max q and max g). On the S-IVB structure the critical conditions occur during prelaunch (ground wind) and flight (max q, max g, and after separation) and on the instrument unit during flight (max q). For the propellant containers, critical external loads are combined with the internal gas pressure and hydrostatic head to obtain the structural design loads.

Slosh baffles are installed in the S-IB RP-1 and LOX containers and in the S-IVB LOX container. The baffles dampen the sloshing propellant and transfer absorbed slosh forces to the container walls. Slosh baffles are not required in the S-IVB LH_2 container because of the low density of the LH_2.

14-12. S-IB STAGE.

The S-IB structure is as assembly of nine propellant containers (five LOX and four RP-1) supported at the forward end by the second stage adapter and at the aft end by the tail section. Eight fins are attached to the tail section. A 105-inch diameter LOX container is located on the stage centerline. Alternately spaced around the center container are four LOX and four RP-1 containers; each is 70 inches in diameter. The containers are structurally independent of one another.

The second stage adapter (spider beam), five LOX containers and tail section resist the loads encountered during all vehicle operations through first stage burnout. The LOX containers carry axial load in both directions; the RP-1 containers carry axial load only in the aft direction. The RP-1 containers are supported at the forward end by a sliding pin connection which permits relative movement between the spider beam and thrust structure due to the contraction of the LOX containers as the containers are being filled.

Several conditions produce critical loads on the thrust structure. The maximum loads on the thrust structure outriggers are produced by the holddown, rebound and max q conditions. For the thrust structure barrel assembly the max q and max g (engine thrust) conditions produce the maximum axial loads, bending moments and shears. The aft end of the thrust structure is protected from the hot engine exhaust gases by the heat shield and flame shield.

Eight aerodynamic fins aid in stabilizing the vehicle during flight. The maximum loading condition on the fins occurs at Mach 1.05. Incorporated in each fin is a holddown fitting for attachment to the launcher. Local critical loading conditions on the fins are produced by the rebound condition.

The critical loading on the center LOX container is a result of the prelaunch condition (container empty and unpressurized). This condition and max q produce the critical loads on the center LOX container skirts. For the outboard LOX containers and container skirts the critical loading conditions occur at Mach 1.05 and max q respectively. The critical loading on the RP-1 containers occurs during prelaunch (containers empty and unpressurized) and at Mach 1.05. For the RP-1 container skirts, the loads that occur at Mach 1.05 are the most critical. The critical load on the spider beam occurs at max q.

In addition to the external loads carried by the LOX containers, all the containers must withstand propellant and internal pressurization loads. Each container consists of a forward and aft bulkhead joined by a cylindrical section. The maximum pressure differential on the container forward bulkheads occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical sections and aft bulkheads varies during flight because the propellant level and ambient pressure decrease while the acceleration of the vehicle increases.

14-13. S-IVB STAGE.

The S-IVB structure is an assembly of an aft interstage, an aft skirt, a thrust structure, an integral propellant container, and a forward skirt. To reduce the length of the vehicle and thus reduce external loading, the propellants are contained in an integral container. Located within the container is the common bulkhead which separates the LH_2 from the LOX. To reduce the loads on the vehicle, the LOX which weighs five times as much as the LH_2 is located aft.

The aft interstage, aft skirt, cylindrical section of the propellant container, and forward skirt withstand the loads encountered during all vehicle operations through first stage burnout. Following stage separation and until spacecraft separation, the thrust structure, LOX container aft bulkhead, cylindrical section of the LH_2 container, and forward skirt resist all loads encountered as a result of S-IVB engine operation.

The critical design condition for the aft interstage and forward skirt is max q. For the aft skirt the critical loads are produced by the max g condition. Critical loading on the cylindrical section of the LH_2 container occurs during prelaunch (container full and unpressurized). Engine thrust, the principal load during S-IVB engine operation, produces a critical loading condition only in the thrust structure.

In addition to the external loads carried by the cylindrical section, the propellant container must resist propellant and pressurization loads. The container consists of a forward bulkhead, a cylindrical section, an aft bulkhead and a common bulkhead. The maximum pressure differential on the container forward bulkhead occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical section and the aft bulkhead is at first stage cutoff. At this time the vehicle acceleration is greatest and the ambient pressure is zero. The common bulkhead is designed to resist both bursting and collapsing pressure conditions. The critical conditions are based on combinations of LH_2 and LOX pressures and temperatures.

14-14. INSTRUMENT UNIT.

The instrument unit structure resists the loads encountered during all vehicle operations through payload separation. The critical design condition occurs during flight at max q which results in a combination of bending moment and axial force producing the largest compressive buckling load on the structure.

14-15. S-IB STRUCTURAL CONFIGURATION

The S-IB stage structure is 962 inches (80.2 feet) long, 257 inches (21.4 feet) in diameter across the containers, 274 inches (22.8 feet) in diameter across the thrust structure, and has a span of 488 inches (40.7 feet) across the fins. A tail section, nine propellant containers (five LOX and four RP-1) and a second stage adapter are structurally joined together to make up the stage. Eight aerodynamic fins are attached to the tail section.

There are only minor configuration differences between the S-IB stage for Saturn IB and the S-I stage for Saturn I (see Section VII). The most significant differences are: the eight equal-size fins on the S-IB stage (the S-I stage has four large and four stub fins), the elimination of the LOX-SOX disposal system and hydrogen vent lines, the moving of the retromotors from the second stage adapter (spider beam) to the S-IVB aft interstage, redesign of the second stage adapter, and less weight.

The fins are equally spaced around the periphery of the tail section. Each fin has an area of approximately 54 square feet. The leading and trailing edges are swept back 45 and 25 degrees respectively. With the exception of the leading edge which is steel, the fins are constructed of aluminum alloy. The exterior of the fins is coated with an ablative insulation.

The second stage adapter is similar to that for the S-I stage except for the deletion of the 45-degree fairing and the cantilevered ends of the spider beam radial members.

More specific payload and mission definition has resulted in less severe desigh loading conditions on the S-IB stage than on the S-I stage. The result is a lighter weight structure with the principal reductions being in the spider beam, propellant container skirts and thrust structure.

14-16. S-IVB STRUCTURAL CONFIGURATION

The S-IVB stage structure (Figure 14-2) is 260 inches (21.7 feet) in diameter and 709 inches (59.1 feet) long. An aft interstage, an aft skirt, a thrust structure, two propellant containers and a forward skirt are structurally joined to make up the stage. The thrust structure and propellant containers are identical to those of the S-IVB stage for Saturn V (see Section XXI). The aft and forward skirts are similar but have been modified because of lower design loads. The aft interstage is a com-

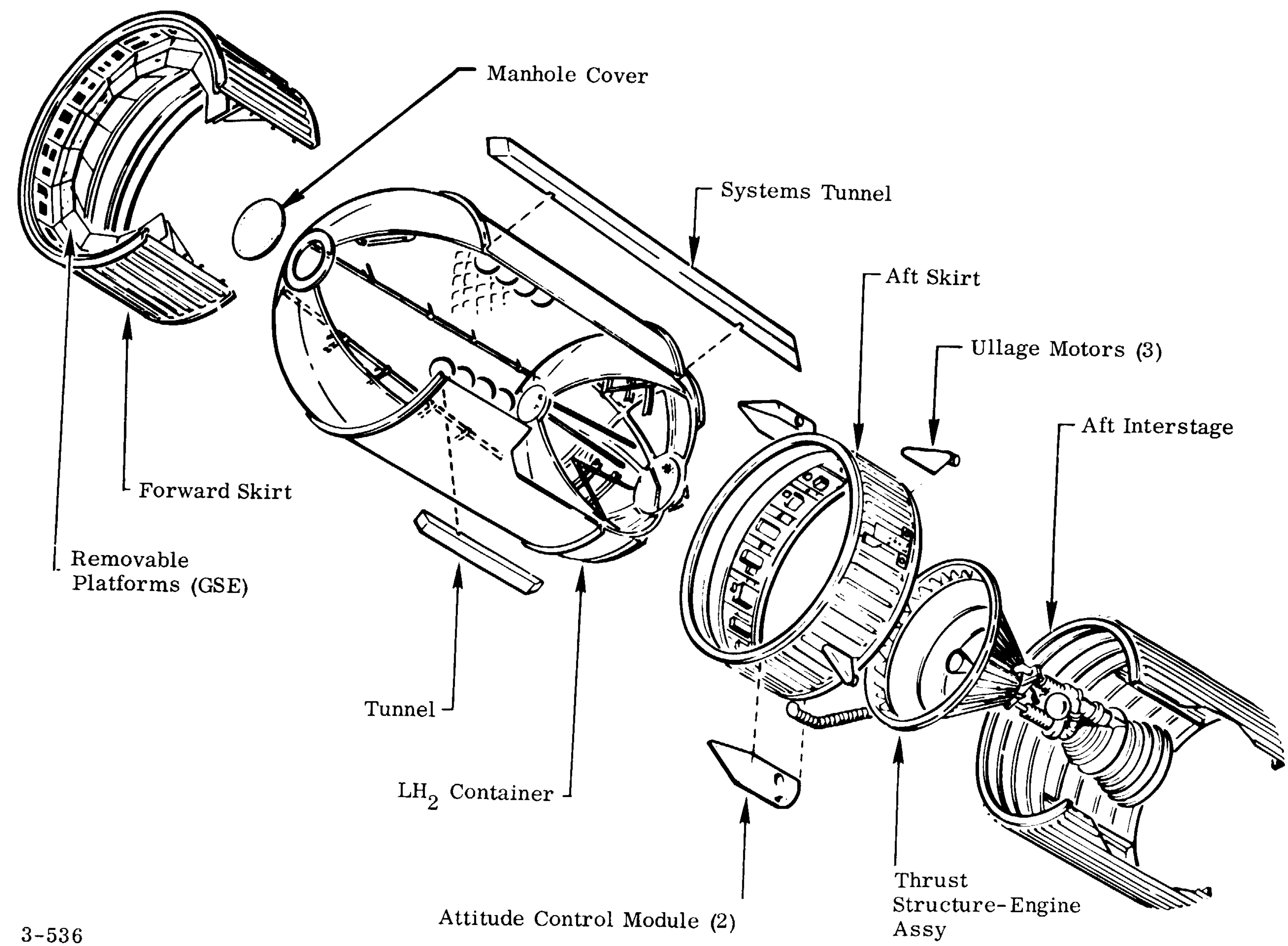

Figure 14-2. S-IV B Stage Structure, Saturn IB

pletely different design.

The loads from the first stage are transmitted to the S-IVB stage through the aft (S-IB/S-IVB) interstage. The aluminum-alloy interstage is a cylinder with a diameter of 260 inches and a length of 224.5 inches. External longitudinal hat section stringers carry the axial load and bending moment and the skin carries the shear load. The interstage skin and stringers are supported by an aft ring, seven internal intermediate rings, and a forward ring. Mating surfaces for the first stage and aft skirt are provided by the aft and forward rings, respectively. The aft interstage is attached by a field splice to the first stage of the launch vehicle (at MSFC station 962). The interstage aft ring, attached to the first stage at eight places on a 220-inch diameter bolt circle, transmits concentrated loads to eight longerons. The longerons shear the load into the skin. The load is uniformly distributed to the forward ring by the stringers. Loads are transmitted to the aft skirt through the forward ring. Four retromotors are mounted on the interstage aft of the separation plane. Attached to the aft end of the interstage is a 260 inch diameter, 27 inch long skirt which shrouds the S-IB stage spider beam.

14-17. INSTRUMENT UNIT CONFIGURATION

The Saturn IB structure for the instrument unit is similar to that of the Saturn V (refer to Section XXI). The major difference is the location of cutouts in the sandwich panels.

The instrument unit is attached to the S-IVB stage and payload in field splices located at MSFC stations 1663 and 1699, respectively.

CHAPTER 3

SECTION XV

PROPULSION

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SECTION XV.
PROPULSION

15-1. REQUIREMENTS.

The Saturn IB propulsion system is required to launch and insert a 34,000 pound Apollo spacecraft into a nominal 105-nautical mile circular earth orbit and to provide attitude stabilization during the first 4.5 hours of orbit. The system is required to function during the launch, ascent, and orbital phases of the mission. Propellant storage and feed systems and propulsion devices (engines) constitute the propulsion system.

A two-stage launch vehicle provides the necessary impulse. First stage cutoff occurs at an altitude of 35.6-nautical miles and a velocity of approximately 3600 knots. Second stage cutoff occurs at a nominal altitude of 105-nautical miles at a velocity of approximately 15,100 knots. Thrust vector control is required to maintain vehicle attitude orientation and angular rates as defined by the control system and, in addition, to damp the amplitude of the first bending mode oscillations of the structure during first stage operation.

A series of impulses is required to ensure successful staging. Both retrothrust to decelerate the first stage and ullage thrust to accelerate the second stage are necessary to aid separation. The ullage thrust also settles the propellants in the aft end of the containers insuring a sufficient suction head to prevent propellant pump cavitation at engine start. (Refer to Paragraph 16-18.)

During the launch phase, a rapid fill and drain capability is required of the propellant storage and feed systems due to the highly volatile properties of the cryogenic propellants (LH_2 and LOX). Provisions for the purging of the propellant containers and feed lines are required before filling or after draining operations as part of the propellant storage and feed system. During the ascent and orbital phases the system must be capable of storing the propellants, and delivering them as required to the engines.

15-2. OPERATION.

The propulsion system operation begins in the launch phase and ends after 4.5 hours of the orbital phase. Table 15-1 lists the major events of the propulsion sequence.

15-3. LAUNCH PHASE.

During the countdown, the propellant containers are purged, loaded, pressurized and conditioned; the pressure storage spheres are purged and charged; and the main stage engines are purged and conditioned prior to being started. A few seconds prior to liftoff, the eight S-IB stage engines are started in a predetermined sequence as commanded by a start sequence initiated by a ground command. The launch phase ends at liftoff.

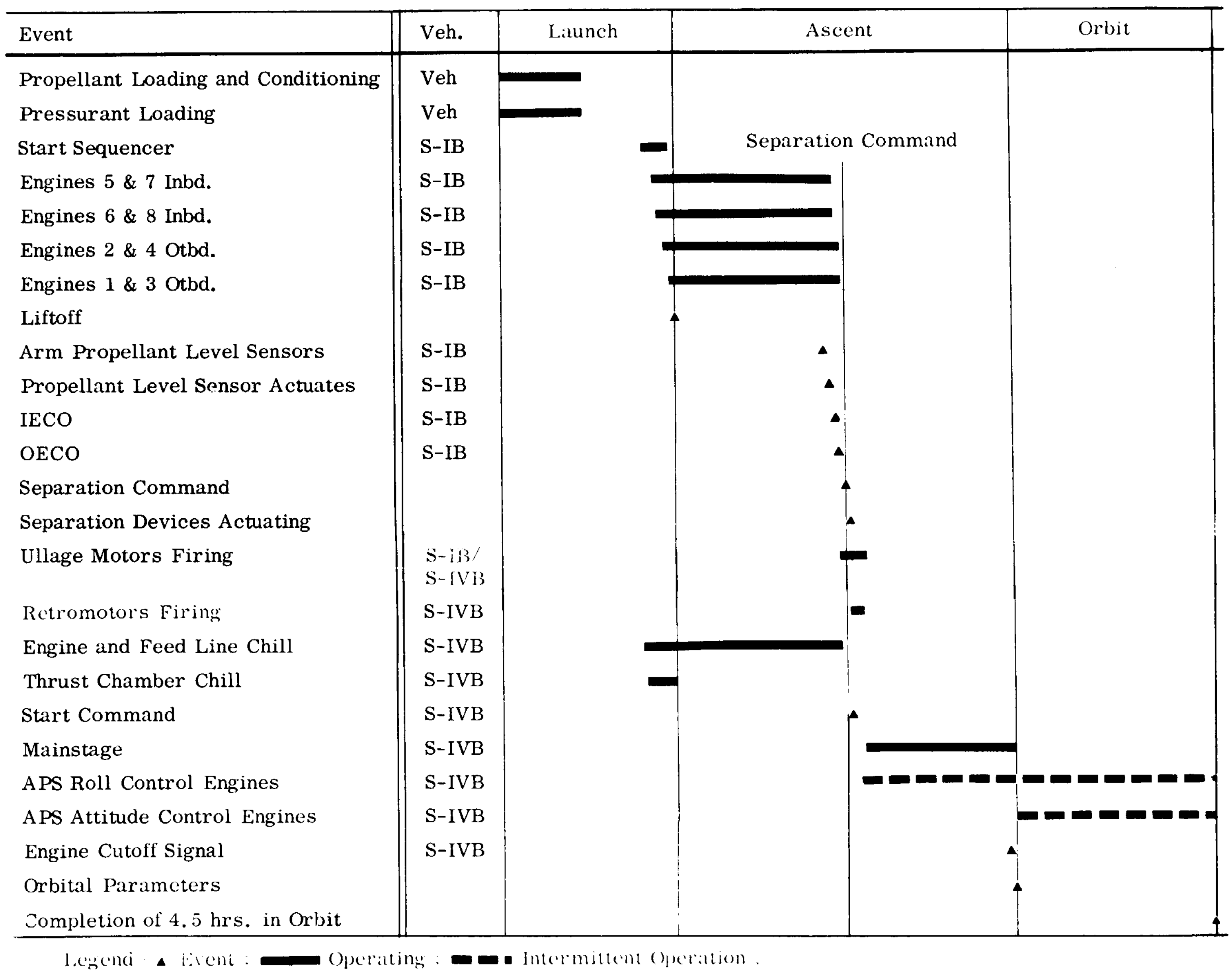
15-4. ASCENT PHASE.

A total nominal thrust of 1,600,000 pounds is provided at liftoff. As a result of decreasing ambient pressure as the vehicle ascends, the stage thrust increases to 1,786,000 pounds at an altitude of 16.3 nautical miles and as a result of under expansion decreases to 1,754,000 pounds prior to engine cutoff. While the vehicle is ascending, thrust-vector and attitude control are provided by the four outboard gimballed engines (Refer to Figure 8-1), in response to commands from the control system. Engine cutoff results from a propellant depletion (level) signal, cutting off the inboard engines a few seconds before the outboard engines.

Prior to staging, a cool down of the single S-IVB stage engine is accomplished by the circulation of propellants through the pumps and feed lines. The chilldown of the thrust chambers is completed after separation and prior to ignition of the engine.

The engine, providing a nominal thrust of 200,000 pounds, is ignited in response to a start command from the instrument unit. Thrust-vector control for the stage is provided by gimballing the main engine; roll control is provided by firing the roll control engines of the auxiliary propulsion system. Both occur in response to the commands of the control system. Engine cutoff occurs as the result of the termination of an electrical signal from the instrument unit. The signal is terminated such that the

Table 15-1. Saturn IB Propulsion Sequence



total impulse delivered by the engine subsequent to the signal results in a velocity to go requirement of zero at thrust termination. The ascent phase ends with the attainment of proper orbital parameters.

15-5. ORBITAL PHASE.

During the orbital phase, the auxiliary propulsion system provides attitude stabilization by firing the attitude and roll control engines in response to commands from the control system. After 4.5 hours of the orbital phase, the propulsion system operations are complete.

15-6. S-IB STAGE IMPLEMENTATION.

The S-IB stage propulsion system provides the 1,600,000 pounds of thrust (nominal at sea level) which accelerates the space vehicle to a sufficient velocity such that after staging the S-IVB stage can subsequently inject the spacecraft into earth orbit. Eight H-1 engines, operating on LOX and RP-1 supplied by the propellant feed and storage system, power the stage. The propulsion system of the S-IB stage is similar to that of the S-I stage (refer to Paragraph 8-3).

15-7. S-IVB STAGE IMPLEMENTATION.

The S-IVB stage is provided with a main propulsion system and an auxiliary propulsion system. After S-IB stage separation, the 200,000-pound thrust of the S-IVB stage main propulsion system injects the space vehicle into earth orbit. The auxiliary propulsion system supplies thrust for roll control during powered flight and attitude stabilization during orbit coast. Ullage thrust for S-IB/S-IVB separation and J-2 engine start is provided by three Thiokol TX-280 rocket motors.

15-8. MAIN PROPULSION SYSTEM.

This system, with the exception of the restart fuel pressurization helium bottle, is basically similar to that described in Paragraph 22-51. The bottles are not provided in this system.

15-9. AUXILIARY PROPULSION SYSTEM.

The auxiliary propulsion system provides roll control during powered flight and attitude stabilization during orbital coast. (During powered flight, pitch and yaw control are provided by gimballing the main engine.) Two auxiliary propulsion system modules are mounted 180 degrees apart on the aft skirt. Three TAPCO hypergolic engines, propellant and pressurant containers and valves are mounted in each module, Figure 15-1. Each module has a propellant capacity of 60 pounds. The basic design of the module is similar to the auxiliary propulsion module of the Saturn V, S-IVB stage (refer to Paragraph 22-58).

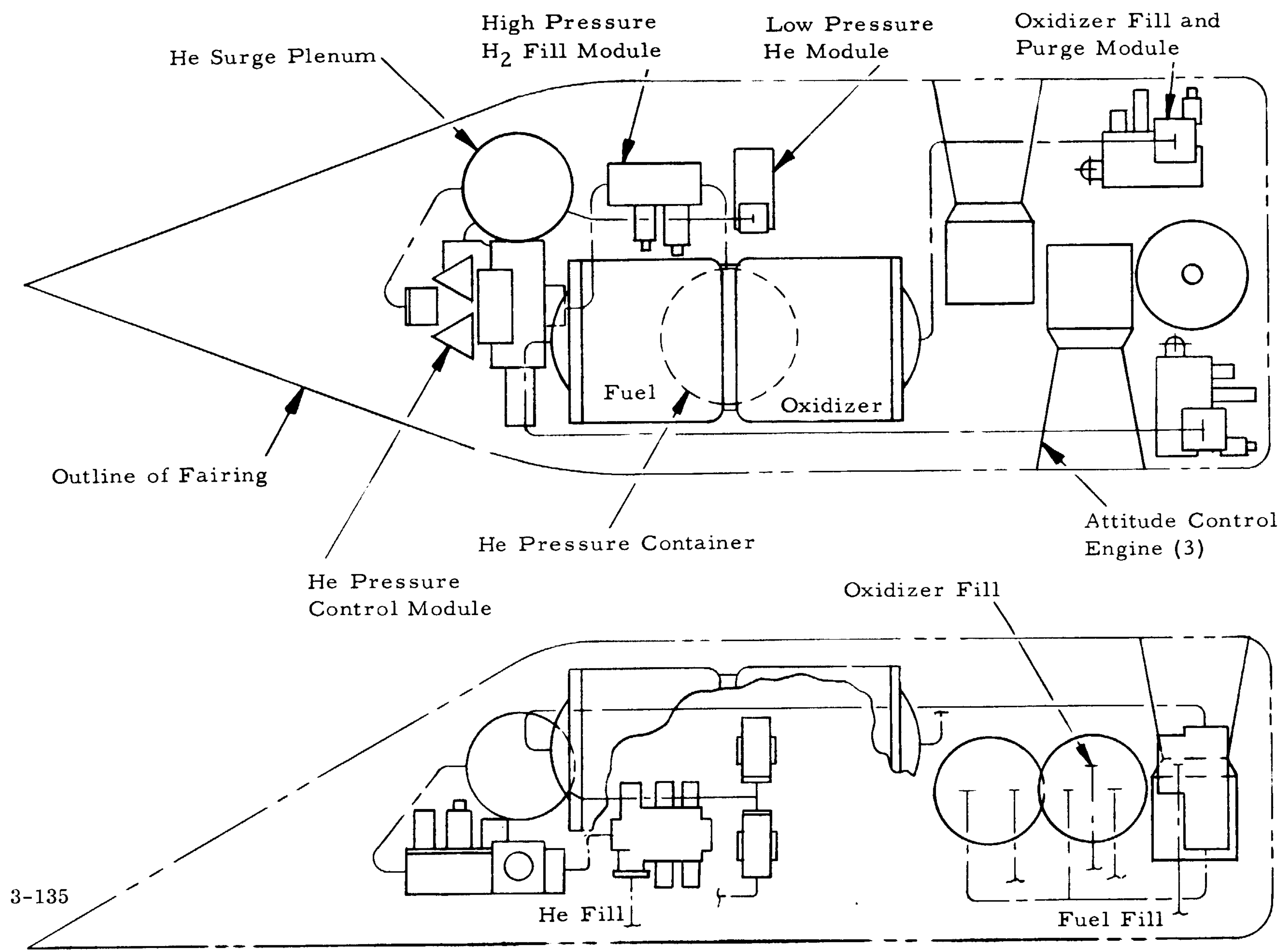


Figure 15-1. Auxiliary Propulsion Module, S-IVB/Saturn IB

CHAPTER 3

SECTION XVI

MECHANICAL SYSTEMS

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SECTION XVI.
MECHANICAL SYSTEMS

16-1. GENERAL.

The mechanical systems of the Saturn IB launch vehicle include environmental control, engine gimballing, separation, ordnance, and platform gas-bearing supply. All of the systems are similar in some degree to the respective systems of the Saturn I and the Saturn V launch vehicles.

16-2. ENVIRONMENTAL CONTROL SYSTEM

The Saturn IB environmental control system controls the environment in certain compartments of the launch vehicle and Apollo payload. The system protects electrical and mechanical equipment from thermal extremes, controls humidity and provides an inert atmosphere for the vehicle compartments. Operation of the system is controlled by ground-based equipment.

The environmental control system allows the use of "off the shelf" electrical components on board the vehicle which otherwise could not be used without elaborate provision for heat dissipation. The system includes a thermoconditioning unit for the cooling of instrumentation located in the instrument unit and in the S-IVB forward compartment.

Environmental conditioning begins during the prelaunch phase upon the application of electrical power to the launch vehicle. Active conditioning of the vehicle compartment ends when the vehicle umbilicals are disconnected at lift off. The thermoconditioning unit continues to provide thermal protection to instrumentation mounted in the instrument unit and the S-IVB forward stage throughout the ascent, and the earth orbital phases of the mission. Thermoconditioning ends when the S-IVB/instrument unit is separated from the Apollo payload.

16-3. OPERATION

The following vehicle and payload areas are conditioned by filtered and thermally controlled dry air or GN_2 supplied by ground equipment.

a. S-IB stage engine compartment
b. S-IB stage fuel container instrument compartments
c. S-IVB stage engine compartment
d. Instrument unit including S-IVB stage forward compartment

The ground facilities also supply a thermally conditioned fluid to the thermoconditioning unit in the instrument unit throughout the prelaunch and launch phases of the mission. At the start of the launch vehicle electrical equipment checkout during prelaunch, the environmental control system supplies cool air to the S-IB stage engine compartment, to two fuel container instrument compartments located forward in the S-IB stage, and to the instrument unit and S-IVB stage forward compartment. The cool air maintains electrical components in these compartments within design temperature limits. When loading of the hypergolic fuel for the auxiliary propulsion system (APS) of the S-IVB stage begins, conditioned air is supplied to the S-IB/S-IVB interstage. The temperature conditioned air circulates through the APS modules maintaining the temperature critical fuel in a liquid state. Prior to loading LOX in the S-IVB stage, warm air is delivered to the S-IB/S-IVB interstage. Warm air is next supplied to the S-IB stage engine compartment prior to loading LOX in the S-IB stage. The warm air flow continues until 30 minutes before the start of LH_2 loading in the S-IVB stage.

The environmental control system medium is changed from air to GN_2 for all compartments and instrument containers a minimum of 30 minutes before the start of LH_2 loading in the S-IVB stage. This prevents possible fire or explosion by maintaining the O_2 content below the level which will support combustion and by preventing any significant accumulations of GH_2. The flow rates and temperature remain unchanged.

The Apollo payload is also conditioned by the environmental control system. The media, flow rate, temperature, and delivery schedules are determined by MSC.

The vehicle thermoconditioning unit provides additional thermal conditioning for instrumentation mounted in the instrument unit and in the S-IVB stage forward compartment. Operation of the thermoconditioning unit begins at the start of the launch vehicle electrical checkout during prelaunch and continues until separation of the Apollo payload.

16-4. S-IB STAGE IMPLEMENTATION

The environmental control system for the S-IB stage maintains the necessary temperature and humidity levels for the protection of instruments, electrical components and ordnance devices in the stage during the prelaunch and launch phase of the mission. The system is similar to that used on the S-I stage of the Saturn I launch vehicle. (Refer to Paragraph 9-4.)

16-5. S-IVB STAGE AND INSTRUMENT UNIT IMPLEMENTATION.

The environmental control system implementation for the S-IVB stage and the instrument unit is similar to that for the systems used on the S-IVB stage and the instrument unit of the Saturn V launch vehicle. (Refer to Paragraphs 23-6 and 23-7.) The instrument unit contains a thermoconditioning unit which provides additional temperature control for temperature sensitive equipment and instrumentation located in the S-IVB forward compartment and the instrument unit.

16-6. ENGINE GIMBALLING SYSTEM.

The Saturn IB engine gimballing system positions the gimballed engines of the active stage to provide the thrust vectors required for vehicle control. In performing this function, the gimballing system is controlled by commands initiated by the attitude control and stabilization function. (Refer to Paragraph 13-8.)

The engine gimballing system steers the vehicle along its trajectory by providing engine thrust vectors for pitch, yaw and (except for the S-IVB) roll control. The system is active during the ascent phase of the mission during S-IB stage, and S-IVB stage powered flight. As the vehicle ascends, in addition to the region of high aerodynamic pressure (35, 000 to 50, 000 feet), it may encounter other disturbances such as thrust misalignments and winds. The external forces produced on the vehicle by such disturbances are counteracted by gimballing the engines of the active stage providing thrust vectors which minimize vehicle structural loading and maintain the vehicle on trajectory.

16-7. OPERATION.

The gimballed engines of the two Saturn IB stages are positioned by means of similar servo actuator systems. Each of the four outboard H-1 engines of the S-IB stage are gimballed through a ± 8-degree square pattern for pitch, yaw and roll control. The single J-2 engine of the S-IVB stage is gimballed to provide pitch and yaw control of the vehicle. Roll control during S-IVB stage powered flight is accomplished by means of the roll control engines of the auxiliary propulsion system.

16-8. STAGE IMPLEMENTATION.

The gimballing systems used on the S-IB and S-IVB stages are similar to the system employed on the Saturn I, S-I stage H-1 outboard engines. (Refer to Paragraph 9-9.)

16-9. SEPARATION SYSTEM.

The primary function of the Saturn IB separation system is to provide positive separation of the S-IB stage from the S-IVB stage during vehicle flight. (The following description does not include an explanation of the separation of the S-IVB stage/instrument unit from the Apollo payload occurring after the payload is injected into earth orbit.)

To lift a given payload into orbit, it is desirable to use a launch vehicle of minimum weight. The design of a minimum-weight vehicle capable of lifting the payload required for the Apollo program necessitates the use of more than one propulsion stage when restricted to present space vehicle technology. During the flight of a multistage vehicle, as a stage is expended it is discarded and the next stage forward provides the thrust for continued payload boost.

16-10. OPERATION.

In separating the two stages of the Saturn IB launch vehicle, the following principal functions occur:

a. Cutoff of engines of the S-IB stage.
b. Acceleration of the S-IVB stage.
c. Physical separation of the S-IB stage from the vehicle.
d. Deceleration of the S-IB stage.
e. Ignition of the S-IVB stage.

The separation operation is initiated approximately 145 seconds after liftoff when a low-level sensor in one of the S-IB stage propellant containers indicates that the propellants are near depletion. When this occurs, control circuits within the vehicle initiate engine cutoff. A controlled thrust termination is necessary to prevent attitude deviations which could occur from unsymmetrical booster burnout. Burnout, as opposed to controlled cutoff, occurs when engines stop burning as a result of propellant depletion. A controlled cutoff is important because during the separation sequence there is a period of approximately 4 seconds, between S-IB stage engine cutoff and S-IVB stage engine ignition and thrust buildup, when the vehicle coasts in uncontrolled flight. In terminating the S-IB stage thrust, the inboard engines are cutoff first.

Following the controlled cutoff of the inboard engines, and then the outboard engines, the ullage motors are ignited to provide acceleration of the S-IVB stage. The acceleration provides sufficient propellant pressure at the inlet of the engine pump for reliable starting. The propellant pressure at the pump inlet is maintained above the design NPSH (Net Positive Suction Head) to prevent cavitation.

Adequate clearance between the separating stages must be achieved prior to S-IVB stage engine ignition to minimize stage interactions. The signal that activates the mild detonating fuse (MDF) which physically severs the S-IB stage from the vehicle is concurrent with the signal that ignites the retromotors. Separation occurs in a single plane located at the forward end of the S-IVB aft interstage at MSFC station 1187. The retromotor thrust decelerates the S-IB stage providing rapid and complete physical separation of the stages. The S-IB/S-IVB interstage remains with the S-IB stage after separation.

Upon completion of the physical separation, the S-IVB stage engine is started. The final function of the separation system is to jettison the burned-out ullage motors from the S-IVB stage minimizing the vehicle weight. The complete staging sequence is tabulated in Table 16-1.

16-11. S-IB STAGE IMPLEMENTATION.

Four solid-propellant retromotors are mounted 90 degrees apart on the S-IB/S-IVB interstage. The thrust vectors of the motors are directed aft and radially inward. The motors provide deceleration of the stage to aid in the complete and expeditious separation of the S-IB stage from the vehicle.

16-12. S-IVB STAGE IMPLEMENTATION.

The S-IVB stage separation system components include three ullage motors and a mild detonating fuse (MDF).

Table 16-1. S-IB/S-IVB Staging Sequence

Item	*Approximate Time	Remarks
Signal from computer to arm level sensors	L. O. +139 sec.	Four level sensors in S-IB containers F-2, F-4, 0-2 and 0-4 are armed at this time.
S-IB level sensor actuates.	L. O. +145 sec. -8.4 sec. (nominal)	Two seconds after a level sensor actuates, the computer sends a command to cut off the inboard engines.
Signal from computer to cutoff inboard engines	-6.4 sec.	Conax valves are fired.
Signal from computer to: Shutdown, S-IVB oxygen chilldown pump Shutdown, S-IVB fuel chilldown pump close S-IVB bleed valves open S-IVB prevalves arm outboard engines cutoff close S-IVB recirculation pumps discharge valves.	-1.4 sec.	Outboard engines thrust OK switches are grouped together so that any one deactuating will give outboard engines cutoff (LOX depletion). Two level probes in S-IB fuel container sumps F-1 and F-3 are also armed so that any one deactuating will give outboard engine cutoff (fuel depletion).
Signal to cutoff outboard engines.	-0.4 sec. (nominal)	Conax valves are fired.
Signal to initiate ullage rockets (from accelerometer or timer).	-0.1 sec.	Ullage rockets burn for approximately 4.0 seconds.

Table 16-1. S-IB/S-IVB Staging Sequence (Cont'd)

Item	*Approximate Time	Remarks
Signal from computer to: activate separation devices initiate retro rockets.	0	
Separation devices actuate.	+0.02 sec.	First possible S-IVB stage axial motion.
Retromotor thrust buildup begins.	+0.03 sec.	One retromotor out capability for successful separation. Takes approximately 0.018 seconds to reach 90 percent thrust.
Signal from computer to activate S-IVB roll control system	+0.2 sec.	
Signal from computer to switch engine position control from S-IB actuators to S-IVB actuators	+1.0 sec.	
S-IVB engine bell clears interstage	+1.64 sec.	Minimum of 10 feet clearance between stages before S-IVB engine ignition command.
Signal from computer to Initiate S-IVB engine start	+1.84 sec.	J-2 engine chilldown commences. The J-2 engine contains a sequencer which sequences the necessary commands to the engine for starting.

[redacted]

Table 16-1. S-IB/S-IVB Staging Sequence (Cont'd)

Item	*Approximate Time	Remarks
Signal from engine sequencer to open start tank discharge valve	+2.34 sec.	J-2 engine chilldown complete. Thrust buildup begins.
Signal from computer to initiate engine gimballing		
Signal from computer to activate S-IVB P.U. system	+3.84 sec.	
J-2 engine at 90 percent thrust	+5.14 sec.	Maximum of 2.80 seconds for the engine to reach 90 percent thrust.
Signal from computer to jettison ullage motors	+17.0 sec.	

*Note: Time values are based on preliminary information.

[redacted]

Three solid-propellant ullage motors radially mounted at 120 degree intervals on the S-IVB aft skirt are used to accelerate the S-IVB stage during S-IB/S-IVB stage separation.

An MDF is used to physically sever the S-IB stage from the S-IVB stage during separation.

Retromotors are not required on the S-IVB stage for S-IVB/instrument unit separation from the Apollo payload. However, the Saturn IB vehicle is designed with a capability for inclusion of two TX-280 solid-propellant retromotors on the S-IVB stage.

16-13. ORDNANCE SYSTEMS.

Many of the mechanical operations performed during a Saturn IB mission require reliable, short time, high energy, concentrated forces. These forces are provided by the ordnance system components. High reliability is achieved by providing redundant components throughout the system.

During launch, the S-IB stage engines are started by ordnance components which provide the forces required for initial turbopump operation and ignition of propellants used to continue the operation. At lift-off, the ground-to-vehicle electrical power transfer is made positive and permanent by ordnance components. During S-IB/S-IVB staging, the individual engine thrusts are terminated in symmetrical unison, ullage and retro motors are fired to provide auxiliary propulsion, vehicle structural connections are severed, and spent ullage motors are jettisoned. These operations are also accomplished by components of the ordnance systems. For range safety, ordnance devices are used to terminate engine thrust and disperse vehicle propellants.

16-14. OPERATION.

Ordnance devices used on the Saturn IB launch vehicle are operational during the launch and ascent phases of the mission. Because of the potential hazard involved, the explosive initiators of ordnance devices are not installed, and the electrical circuits of the ordnance system are not completed until all personnel except the ordnance crew are clear of the launch pad.

16-15. Launch Phase. During launch, H-1 engine starting is initiated by ignition of a solid-propellant gas generator (SPGG). The SPGG produces gas for the initial acceleration of the high-speed turbine which drives the LOX fuel turbopump and provides primary ignition of the liquid-propellant gas generator (LPGG). Secondary ignition of the LPGG is supplied by LPGG igniters. The LPGG produces the gas for continued operation of the high-speed turbine.

At liftoff explosive switches are fired to provide positive and permanent connections between the launch vehicle electrical system and its internal power supply.

16-16. Ascent Phase. During ascent of the launch vehicle when a low-level sensor in one of the S-IB stage propellant containers indicates that propellants are near depletion, the S-IB/S-IVB separation sequence is initiated. Ordnance devices play a major role during separation. An explosively actuated Conax valve on each H-1 engine provides for the controlled cutoff of first the four inboard engines and then the four outboard engines. Ullage motors provide vehicle acceleration for propellant positioning and to ensure sufficient turbopump inlet pressure for S-IVB stage engine ignition. Retromotors decelerate the S-IB stage providing rapid and complete physical separation of the stages. Physical separation of the S-IB stage from the S-IVB stage is accomplished by means of a mild detonating fuse which severs the vehicle structure at the separation plane. Frangible nuts are used to attach the ullage motor fairings to the S-IVB aft skirt. Explosive charges within each nut are ignited, after separation of the stages, to fracture the nuts in order to jettison the spent ullage motors.

Throughout the ascent phase of the mission the range safety officer can terminate the flight at any time by means of the propellant dispersion system. When the system is actuated the active stage engines are shut down and detonating cord is ignited to cut open the propellant containers. To attain high reliability each stage (S-IB and S-IVB) has a separate dispersion system.

16-17. S-IB STAGE IMPLEMENTATION.

Ordnance devices on the S-IB stage include components used for transfer of electrical power, engine starting and cutoff, and propellant dispersion system ordnance. These components are similar to those used on the S-I stage of the Saturn I launch vehicle. (Refer to Paragraph 9-22.)

16-18. S-IVB STAGE IMPLEMENTATION.

Ordnance on the S-IVB stage includes explosive liftoff switches used during launch (Refer to Paragraph 9-23), ullage motors, retromotors, a mild detonating fuse (MDF), and frangible nuts used during separation, and components associated with the propellant dispersion system.

16-19. Ullage Motors. Three solid-propellant Thiokol TX-280 rocket motors provide an acceleration of 0.01 -g to the S-IVB stage to position propellants for J-2 engine ignition and to aid in separation during S-IB/S-IVB staging. The ullage motors are mounted in fairings on the aft skirt of the S-IVB stage and are located at 120 degree intervals around the skirt and are canted at 35 degrees from the vehicle centerline to minimize the effect of exhaust gases on the vehicle hardware (Figure 7-14). Each motor burns for 3.0 seconds (minimum) and develops a nominal average vacuum thrust of 3390 pounds at 70 degrees F. Two electronic bridge wire firing units supply a 2300 $\pm$ 100- volt dc pulse to two EBW initiators installed in the igniter of each ullage motor. A pressure transducer connected by tubing from the igniter of each motor detects ullage motor firing.

16-20. Retromotors. Four, TE-29-1B solid propellant retromotors are used to decelerate the S-IB stage during separation. The motors are mounted at 90 degree intervals around the S-IB/S-IVB interstage. Ignition of each motor is accomplished in the same manner as the ullage motors described in paragraph 16-19.

Retromotors are not required on the S-IVB stage for separation of the S-IVB/instrument unit from the Apollo payload. However, the vehicle is designed with structural capability for the inclusion of two Thiokol TX-280 solid-propellant retromotors on the S-IVB stage.

16-21. Mild Detonating Fuse (MDF). An MDF is used to physically sever the S-IB stage from the S-IVB. Installation and operational details are the same as for the MDF used to separate the S-II stage from the S-IVB stage on the Saturn V launch vehicle. (Refer to Paragraph 23-31.)

16-22. Frangible Nuts. Frangible nuts, Figure 9-21, are used to attach ullage motor fairings to the S-IVB aft skirt. The nuts are fractured by means of two explosive charges in order to jettison the spent ullage motors after the separation of the S-IB stage and the S-IVB stage. The frangible nuts are the same as those used on the S-IV stage of the Saturn I launch vehicle. (Refer to Paragraph 9-29.)

16-23. Propellant Dispersion System Ordnance. The propellant dispersion system ordnance for the S-IVB stage consists of two electronic bridge wire firing units, two EBW detonators, a safety and arming (S&A) device, detonating cord and linear shaped charges. The system is similar to that used on the S-IVB stage of the Saturn V launch vehicle. (Refer to Paragraph 23-32.)

16-24. PLATFORM GAS-BEARING SUPPLY SYSTEM.

The Saturn IB platform gas-bearing supply system furnishes filtered GN_2 at a regulated pressure, temperature, and flow rate to the gas bearings of the ST-124-M stabilized platform. The GN_2 is supplied to the stabilized platform from the start of checkout during prelaunch until separation of the S-IVB stage and instrument unit from the Apollo payload during the orbital phase of the mission.

The system is similar to the platform gas-bearing supply system used on the Saturn I launch vehicle. (Refer to Paragraphs 9-33 and 9-34.)

CHAPTER 3

SECTION XVII

GROUND SUPPORT EQUIPMENT

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SECTION XVII.
GROUND SUPPORT EQUIPMENT

17-1. GENERAL.

The Saturn IB ground support equipment (GSE) includes all of the ground equipment required to support the fabrication, checkout, transportation, servicing, handling, static testing, and launch operations related to the S-IB stage, S-IVB stage and instrument unit. The GSE in this section excludes launch-peculiar GSE which is described in Volume III. In supporting the above operations, the GSE is formed into functional ground system, subsystem, and unit configurations. The various configurations are employed as required at all locations involved in the research and development of the vehicle and its stages. Since the operation of each configuration may vary depending on the location where used, an operational description is not contained in this document. Instead, the major GSE is listed and primary functions described.

17-2. ELECTRICAL SUPPORT EQUIPMENT, SATURN IB.

The Saturn IB ESE is used during the checkout, static testing, and launch of the vehicle. The majority of this equipment is located at the Automatic Ground Checkout Station (AGCS). This ESE is classified as follows.

a. Monitoring and Control Equipment
b. System Integration Equipment
c. Networks, Distribution and Control Equipment
d. Simulation Equipment
e. Ground Equipment Test Sets
f. Recording Group Equipment
g. Peripheral Equipment
h. Overall Test Equipment (OAT)
i. Systems Integration Sets

With the exception of the monitoring and control equipment and recording group equipment, MSFC is responsible for fabrication of all of the above. For these two classifications, MSFC has partial fabrication responsibility. A summary of the Saturn IB ESE functions is given in Table 17-1.

Table 17-1. Electrical Support Equipment, Saturn IB

Equipment	Function
Monitoring and Control Equipment	a. Provides monitoring and control of systems under test by means of panel meters, switches, light banks, and displays. b. Control and display equipment is provided for the following systems: emergency detection, mechanical, propellant loading, ordnance, measuring and RF, navigation, propulsion, networks, and computer display.
Systems Integration Equipment	Used for signal distribution to the stage GSE from the computer and from the computer to the monitoring and control consoles.
Networks, Distribution, and Control Equipment	a. Provides proper distribution and sequencing of the control signals and power to the particular stage under test. b. Contains switches for relay control and meters on the front panels.
Ground Equipment Test Set (GETS)	Provides signals for checking out GSE prior to connecting it to the integrated vehicle or stage simulators.
Recording Group Equipment	Records all vehicle discrete outputs and inputs during the checkout sequence.
Peripheral Equipment	a. The countdown clock provides the time base for all functions during countdown. The clock, syncronized with WWV, converts the output to real time readout and supplies real time commands to the instrument unit guidance programmer from the RCA-110 computer. b. The signal conditioning equipment reduces the inputs from 28-volt dc to standard 5-volt dc acceptable to the computer.
Overall Test Equipment (OAT)	Simulates functions which cannot be actually performed by the systems under test because of the resulting hazardous conditions.
Systems Integration Sets (SIS)	Simulate interface signals between stages.

17-3. GROUND SUPPORT EQUIPMENT, S-IB STAGE.

In general, the S-IB stage GSE is classified as test, checkout, and monitoring; transportation, protection and handling; and servicing. Tables 17-2 through 17-4 list the equipment and functions of each classification.

Table 17-2. Test, Checkout and Monitoring Equipment, S-IB

Equipment	Function
Instrumentation Equipment	Supplies switching signals to the various conditioners used in the instrumentation system.
Safety Monitor Equipment	a. Used when the S-IB stage is undergoing tests and during prelaunch operations. b. Provides necessary interface requirements with the stage when less than a complete test complex is attached. c. Provides shutdown capability in the event that a dangerous condition develops.
Central Control Equipment	Provides a central control console for use during checkout and launch having a capability of directing the program to start, stop, hold, modify, or rerun any system test sequence.
Stage Propulsion Equipment	Provides capability of energizing, controlling, monitoring, and testing the electrical components associated with the stage electrical power supplies, pneumatic systems, pyrotechnics, and the electromechanical components associated with the propellant containers and rocket engines.
Ground Power System	Provides electrical power (28-volt dc, 115/208-volt, 400 cps ac) to the applicable GSE, and controls, monitors, and relays electrical power to the S-IB stage components and other test site systems during test, checkout, static fire and launch countdown operations.
Ground Equipment Test Station (GETS)	Verifies and validates the electrical circuits of GSE prior to the mating of the S-IB stage and GSE.
Tower Test Monitor System	Simulates those functions in the stage and its support mechanism which cannot feasibly be performed during a sequential final checkout or compatibility test of the S-IB stage and GSE.

Table 17-2. Test, Checkout and Monitoring Equipment, S-IB (Cont'd)

Equipment	Function
Ground Support Equipment Testing	Accomplishes vehicle component and subsystem verification testing of engine heaters, the hydraulic control system, propellant system heaters, instrument canisters, the cooling system, stage destruct firing circuits, and the engine Conax valve firing circuits.
FM/FM Ground Telemetry Station	Checks the proper operation of various transducers in the instrumentation system and tests the stage FM/FM telemetry system.
SS/FM Ground Telemetry Station	Checks the proper operation of various transducers in the instrumentation system and tests the stage SS/FM telemetry system.
Upper Stage Simulator	a. Provides proper loading of circuitry which normally terminates in an upper stage. b. Contains equipment with test point facilities for use in troubleshooting and for insertion of stimulus if required.
S-IB Stage Simulator	a. Designed to checkout GSE. b. Presents the proper impedances and sufficient typical stage outputs to establish confidence in GSE. c. Contains equipment with test point facilities for use in troubleshooting and for insertion of stimulus if required.
Fuel Tanking Simulator	Supplies calibration signals to the fuel control panel.
Fuel Density Simulator	Supplies calibration signals to the fuel density monitor panel.
Liquid Oxygen Tanking Simulator	Supplies calibration signals to the LOX tanking control panel.
Engine Simulator	a. Simulates the electrical network of the engine and verifies the operation of the GSE. b. Used during stage testing when the electrical responses of an engine are required but the actual engine has not been installed.

Table 17-2. Test, Checkout and Monitoring Equipment, S-IB (Cont'd)

Equipment	Function
Command Destruct System Test Set	a. Verifies proper operation of the stage command destruct (propellant dispersion) subsystem. b. Generates coded RF signals, and monitors the command destruct subsystem ability to receive, decode, and generate an appropriate response to the input stimuli.
Radio Frequency Test Bench	Provides a central source of equipment and necessary power to calibrate, troubleshoot, and repair radio frequency equipment of the S-IB stage and GSE.
Exploding Bridge Wire Test Set	a. Provides stimuli to check out the exploding bridge wire unit and firing units. b. Sensors monitor the firing units, and the test set ascertains if the sensor response code is compatible with the stimuli output code.

Table 17-3. Transportation, Protection, and Handling Equipment, S-IB

Equipment	Function
Stage Handling Equipment	Consists of a set of slings that are used for handling and loading the S-IB stage, assemblies, components, and certain items of GSE.
Fin Sling	Used to lift and handle the S-IB stage fins during installation or removal operations.
Engine Handling Equipment	Provided in support of the S-IB stage for installation, removal, servicing, and maintaining the H-1 engine.
Transporter	Used in the horizontal support and transportation of the assembled S-IB stage during all phases of mobility, in factory and field operations.

Table 17-3. Transportation, Protection, and Handling Equipment, S-IB (Cont'd)

Equipment	Function
Transporter Dolly	Consists of a frame and running gear assembly and provides a towbar, steering system, braking system, and operator controls. (A fore and aft transporter dolly connected by a structural frame provides a complete transporter.)
Transportation Accessories Kit	Provides the necessary equipment to prepare the stage for transportation, protection of small parts during transportation, and to tie-down, block, and shore the stage transporter on the barge.

Table 17-4. Servicing Equipment, S-IB

Equipment	Function
RP-1 Fuel Filling	Controls the transfer of RP-1 from the facility storage tanks to the S-IB stage fuel containers either manually or automatically.
Fuel Replenishing	Provides the necessary control for adjusting fuel weight to the S-IB stage requirements and holding for a minimum pad standby time of 12 hours.
Liquid Oxygen Filling	Controls the transfer of LOX from the storage tanks to the S-IB stage LOX containers either manually or automatically.
Liquid Oxygen Replenishing	Provides the necessary LOX replenishing to compensate for boiloff for a minimum holding time of 12 hours.
Pneumatic Control System	Supplies GN_2 and helium from the high pressure GN_2 storage facility for stage pressurization, purges, operation of launcher and tower equipment, LOX bubbling, LOX container prepressurization, and operation of pneumatically controlled devices in the stage and launch complex.

Table 17-4. Servicing Equipment, S-IB (Cont'd)

Equipment	Function
Environmental Control System	a. Provides air or GN_2 at the required humidity and temperature to the S-IB stage and launcher. b. Satisfies all S-IB stage air conditioning requirements, and provides inert gas purging for stage compartments.
Swing Arm System	Supports the service lines that link the S-IB stage to the ground supply systems.
Holddown Arm System	Secures the stage to the launcher until all engines reach satisfactory operating conditions and all hydraulic systems reach operational pressures.

17-4. GROUND SUPPORT EQUIPMENT, S-IVB STAGE.

The S-IVB stage GSE is classified as test, checkout, and monitoring; transportation, protection and handling; servicing; and auxiliary. Tables 17-5 through 17-8 list the equipment and functions of each classification.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB

Figure	Equipment	Function
17-1 (Sheet 1)	EBW Initiator Test Set	a. Performs qualitative checks of the initiator in an explosion-proof container. b. Performs quantitative checks on initiators. c. Determines if the electrical characteristics of the initiator are within tolerance.
17-1 (Sheet 1)	EBW Firing Unit Component Test Set	a. Provides the circuitry required to test the firing unit as a component. b. Performs the quantitative checks on firing units.
17-1 (Sheet 1)	Destruct System Component Test Set	Used for testing the command destruct system components prior to installation in the stage.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
17-1 (Sheet 1)	EBW Pulse Checker	Determines if electronic bridge wire units deliver sufficient current through an initiator simulator to retain a GO or NO-GO decision.
17-1 (Sheet 2)	Battery Charger Component Test Set	Used to charge silver-zinc batteries at rates up to 10 amperes per minute.
17- (Sheet 2)	Battery Discharger Component Test Set	Used to discharge silver-zinc batteries, check the batteries, and check the heater blanket circuitry and heater blanket thermostat.
17-1 (Sheet 2)	Printed Circuit Card Test Set	a. Checks the printed circuit cards used as a component or module of the S-IVB GSE. b. Accomplishes fault isolation down to a particular part or group of parts. c. Provides all necessary voltage levels, input stimuli, loads, and output monitoring.
17-1 (Sheet 3)	Digital Magnetic Tape Unit	a. Records responses and decisions of the computer. b. Recompiles computer programs and updates stage-peculiar data in the computer. c. Records test results.
17-1 (Sheet 3)	Checkout Computer	a. Used to execute stored program instructions to control the automatic complex and to control the input/output equipment associated with the computer and the operator displays. b. Evaluates S-IVB stage responses and makes decisions where required. c. Performs self-test routines and tests on computer controlled equipment.
17-1 (Sheet 3)	Patch Panel Distribution Box	a. Provides a convenient and flexible means of interconnecting, by patch cords, the various units of GSE.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
		b. Provides an interface between the GSE and facility items.
17-1 (Sheet 3)	Telemetry Tape Unit	Receives and stores telemetered signal data for eventual playback and data analysis.
17-1 (Sheet 4)	Signal Distribution Unit	a. Provides an end distribution point between the GSE and the stage. b. Performs the switching required for control of the stage and facilities. c. Performs switching and distributing functions for fault-isolation and calibration routines.
17-1 (Sheet 4)	Destruct System Test Set	Provides RF stimulation to the stage destruct system so that the system receivers and controllers can be tested.
17-1 (Sheet 4)	Automatic Typewriter	a. Used to introduce information into the computer to effect changes in checkout or trouble-shooting programs and in stored data in the field. b. Provides a hardcopy output of detailed information from the computer concerning test results which indicate component failure.
17-1 (Sheet 4)	Portable Display	a. Provides personnel at remote locations with access to information from the check-out computer. b. Displays four digits, alphabetical or numerical, representing the value or state of a selected parameter.
	Propellant Utilization System Test Set	Used for making adjustments to and testing of the propellant utilization electronic assembly and valve positioner.
	Sequencer Test Set	Tests the stage sequencer and isolates malfunctions down to a module such as a relay, resistor or diode.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
	Power Systems Electrical Component Test Set	Used for testing the stage inverter.
	PCM/FM Telemetry Test Set	a. Used to adjust, calibrate, and evaluate all components of the PCM telemetry system. b. Used to test the complete system from multiplexer inputs to output data.
	PAM/FM/FM Component Test Set	a. Used for testing, calibrating, adjusting, and monitoring the signal conditioning units, slow-speed commutators, calibration units, voltage-controlled oscillators, and summing amplifier of the PAM/FM/FM system. b. Tests the components when assembled in a system, and performs fault isolation tests down to printed-circuit card level.
	SS/FM Telemetry Test Set	a. Used for calibrating, adjusting, and checking out the single sideband translator assembly. b. Conducts tests on the entire system when assembled. c. Used for fault isolation down to the printed circuit card level.
	FM Transmitter Test Set	a. Used to check the FM transmitters for proper operation, both dynamic and static. b. Isolates malfunctions down to a part or group of parts. c. Provides the necessary operating voltages, input stimuli, and output monitoring.
	Magnetic Tape Recorder	a. Used to check the stage tape recorder. b. Provides the operating voltages, input stimuli, and output monitoring facilities required to isolate faults. c. Tests the recorder for data transfer accuracy.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
	Propellant Utilization System Calibration Unit	Simulates LOX and fuel container propellant loads from 0- to 100- percent.
	Input/Output Console	a. Provides the necessary indicators, projection displays, and switches to operate the computer.
		b. Provides a continuous display of computer events and permits independent computer operation.
	Computer Interface Unit	a. Performs conversion of waveforms and information formats required for communication between the computer and end items such as the test station console, stimuli and response conditioners, etc.
		b. Accepts or generates parallel information for intercommunication with the checkout computer.
		c. Accepts and generates special control signals as required.
	Display Buffer	a. Provides temporary storage of digital quantities and conversion to analog voltages for display.
		b. Selects appropriate words from the PCM data train or computer output in accordance with operator display controls.
		c. Provides routing of analog voltages to displays as determined by display operators.
	Special Purpose Display Console	a. Provides real time display of analog signals for operator monitoring.
		b. Provides analog display of information that is processed in digital form.
		c. Provides a recorded output of test results in analog form for future reference.

Table 17-5. Test, Checkout and Monitoring Equipment, S-IVB (Cont'd)

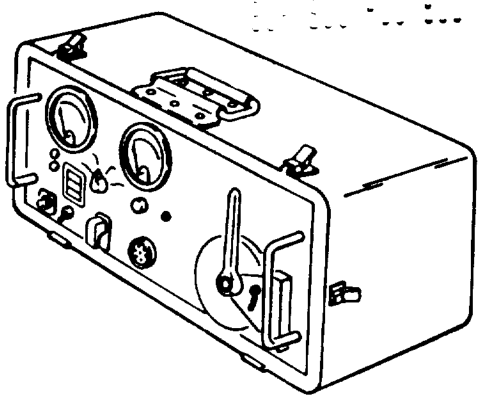
Figure	Equipment	Function
	System Status Display Console	a. Used to display the parameters of of any part of the system on a television-type display unit. b. Presents symbolic and alphabetical or numerical information displayed as an overlay on a slide-supplied background.
	Test Operator Console	a. Acts as the primary master control station for all integrated tests. b. Provides the operator with a means of monitoring and controlling the automatic system during subsystem tests.
	Frequency Calibration Unit	a. Measures and displays the frequency of the received telemetry signals from each telemetry ground station. b. Supplies frequencies as a secondary transfer standard for the purpose of calibrating the telemetry signals.
	PAM/FM/FM Telemetry Ground Station	a. Acts as a monitoring and receiving station for FM data from the PAM/FM/FM and FM/FM stage telemetry transmission systems. b. Displays individual channels locally on a raster monitor or sent to external areas for recording and display purposes.
	PCM/FM Telemetry Ground Station	a. Receives PCM data from a stage telemetry system and demodulates the data for individual channel analysis. b. Regenerates incoming data and sends it to external areas for computer storage and analysis. c. Converts PCM data to analog for transfer to external display units.
	SS/FM Telemetry Ground Station	a. Acts as a receiving station for SS-multiplexed signals from the stage SS/FM telemetry transmission subsystem. b. Demultiplexes and demodulates the incoming data into individual channels.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB (Cont'd)

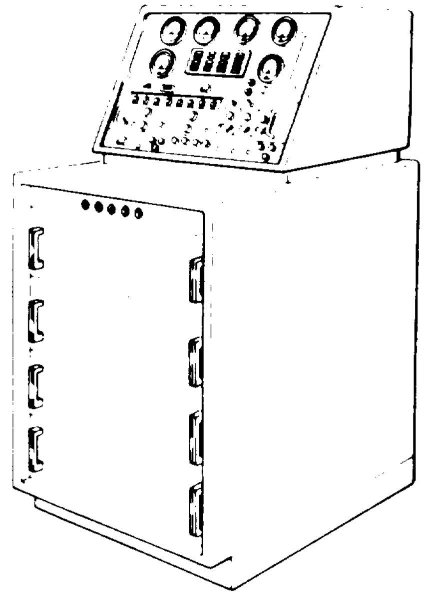
Figure	Equipment	Function
		c. Displays individual channels locally on a monitor and routes them to external areas for further processing.
	Range Time Generator	Provides a reference time for use during stage checkout at Huntington Beach and Sacramento, California.
	Leak Detection Equipment	a. Detects leakage in stage component mounting boxes.
		b. Isolates and determines quantities by gas types, and provides analog voltage signals indicative of the quantity of a particular gas type.
	Stimuli Signal Conditioner	a. Generates hardline stimuli to test stage hardware.
		b. Simulates signals normally received from the stage instrument package.
		c. Supplies control signals to test standard facilities units to effect automatic test of the stage system.
	Response Signal Conditioner	a. Provides the signal isolation and buffering necessary to condition stage and facility signals.
		b. Digitizes conditioned signals for automatic control and response evaluation during the test.
		c. Selects the appropriate analog signal or group of signals as requested by the stage checkout computer buffer.
	Ground Support Equipment (GSE) Test Set	a. Used for overall checks of the GSE system when the stage is not connected.
		b. Verifies the satisfactory operation of that portion of the GSE not checked or verified by the self-test programs and routines of the automatic system.

Table 17-5. Test, Checkout, and Monitoring Equipment, S-IVB (Cont'd)

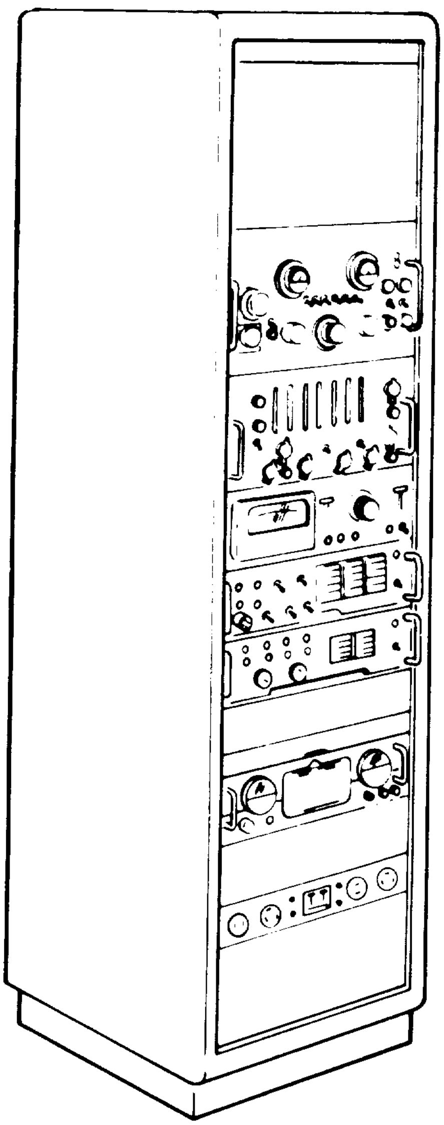
Figure	Equipment	Function
	Stage External Power Racks	a. Provides the ground power source for stage systems. b. Used as a simulated stage internal power source. c. Contains an emergency power chassis to sense the dc level of the power source and switches the output to an emergency battery if necessary.
	Safety Item Monitor	a. Provides isolation and buffering between the stage and computer portion of the GSE system. b. Provides level detection of analog safety items and memory elements. c. Provides independent high-speed scan of elements and generation of a computer interrupt when a failure is indicated. d. Provides identification of the element that indicated a malfunction.
	Cable Network	Provides electrical interconnection between the GSE and the stage or unit under test.



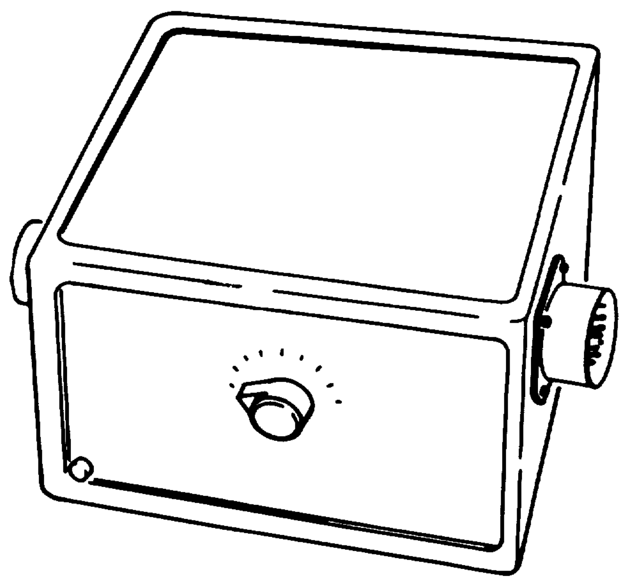
Test Set, EBW Initiator



EBW Firing Unit Component Test Set



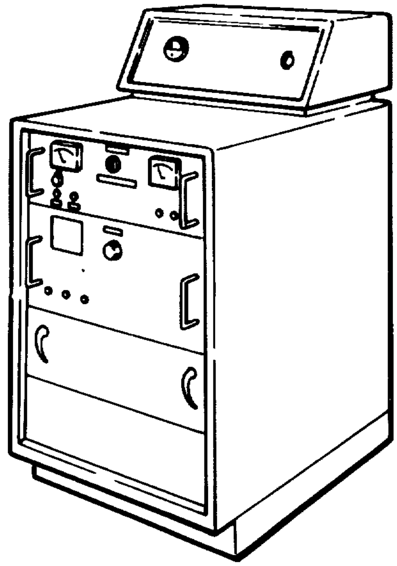
Destruct System Component Test Set



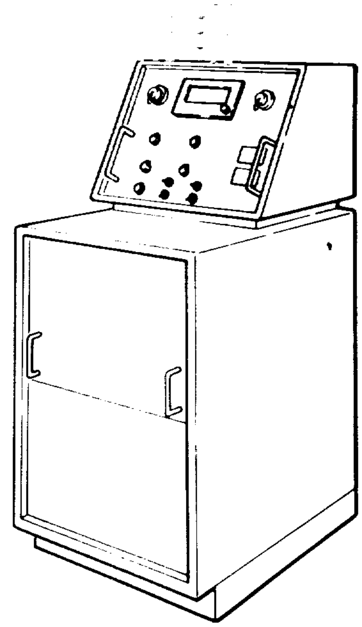
EBW Pulse Checker

3-825

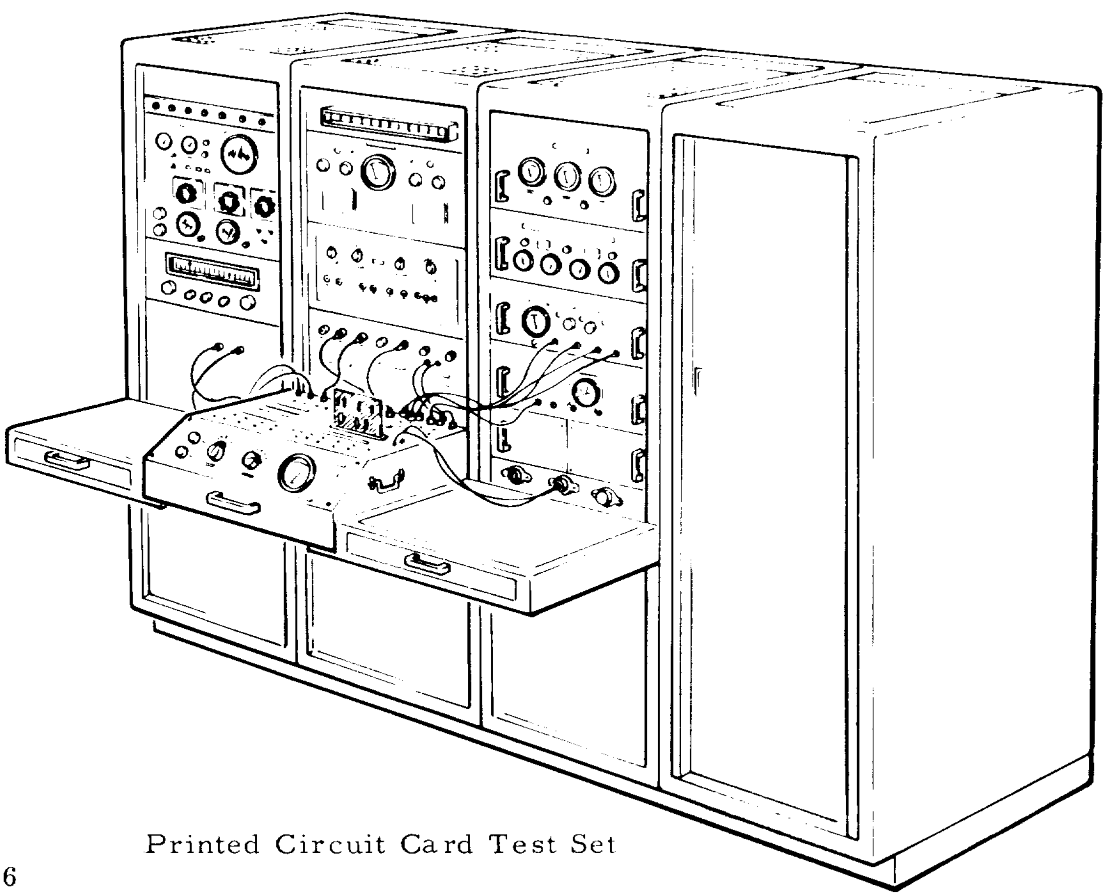
Figure 17-1. Test, Checkout, and Monitoring Equipment, S-IVB (1 of 4)



Battery Charger Component Test Set



Battery Discharger Component Test Set



Printed Circuit Card Test Set

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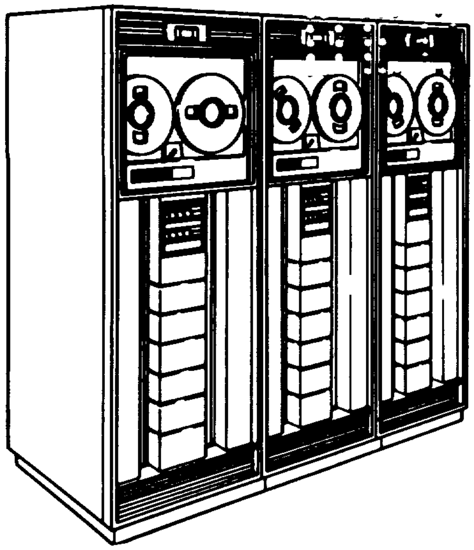
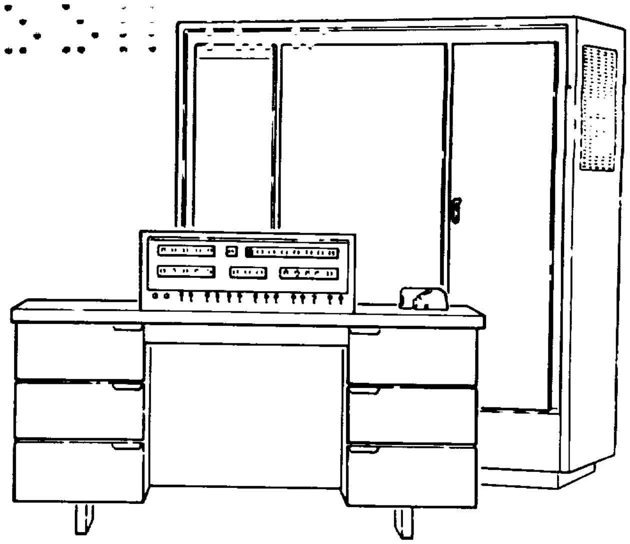
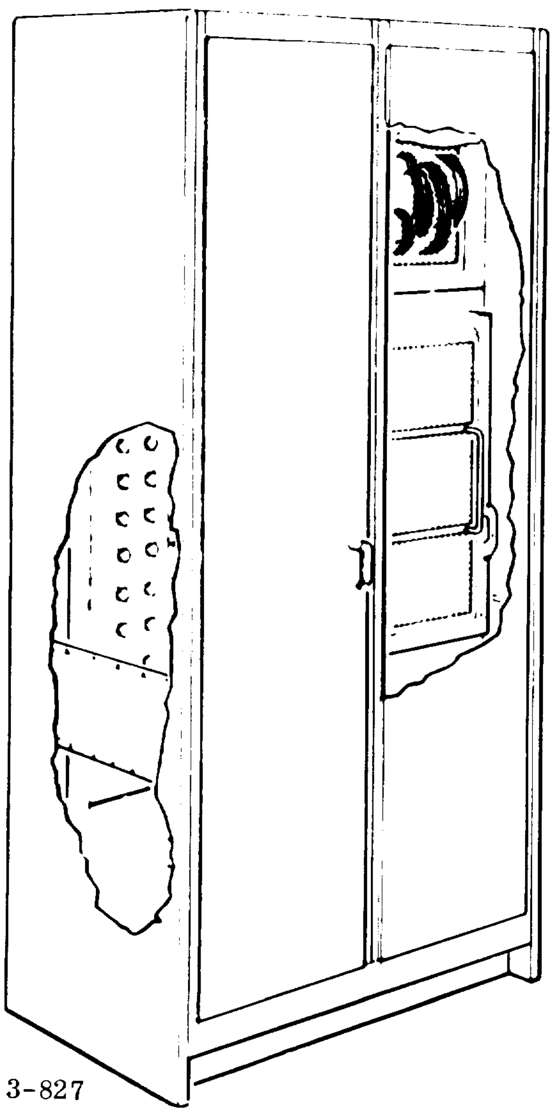
Figure 17-1. Test, Checkout, and Monitoring Equipment, S-IVB (2 of 4)



Digital Magnetic Tape Unit

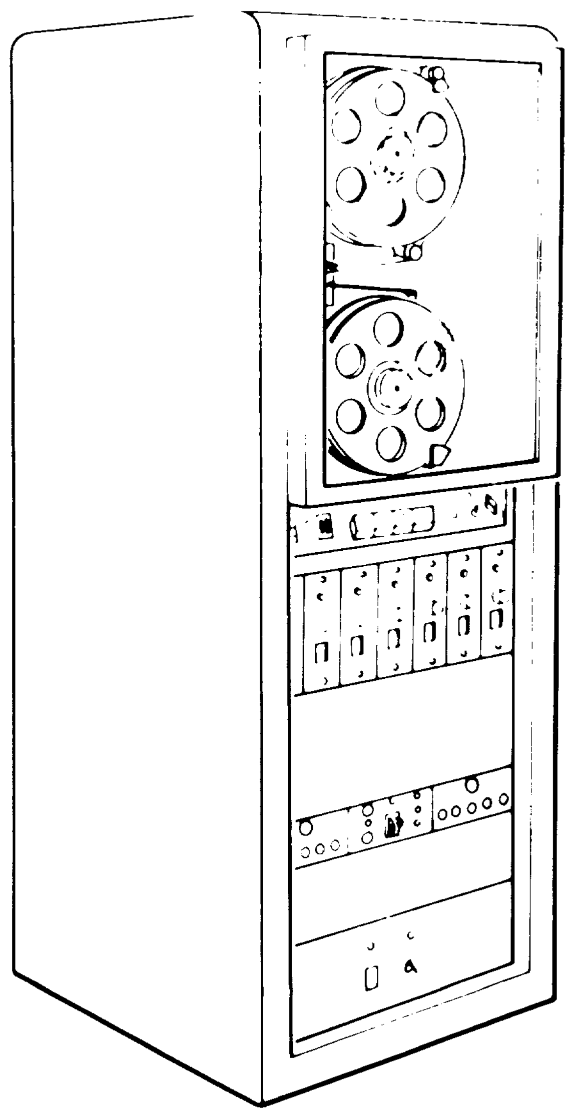


Checkout Computer



3-827

Patch Panel Distribution Box



Telemetry Tape Unit

Figure 17-1. Test, Checkout, and Monitoring Equipment, S-IVB (3 of 4)

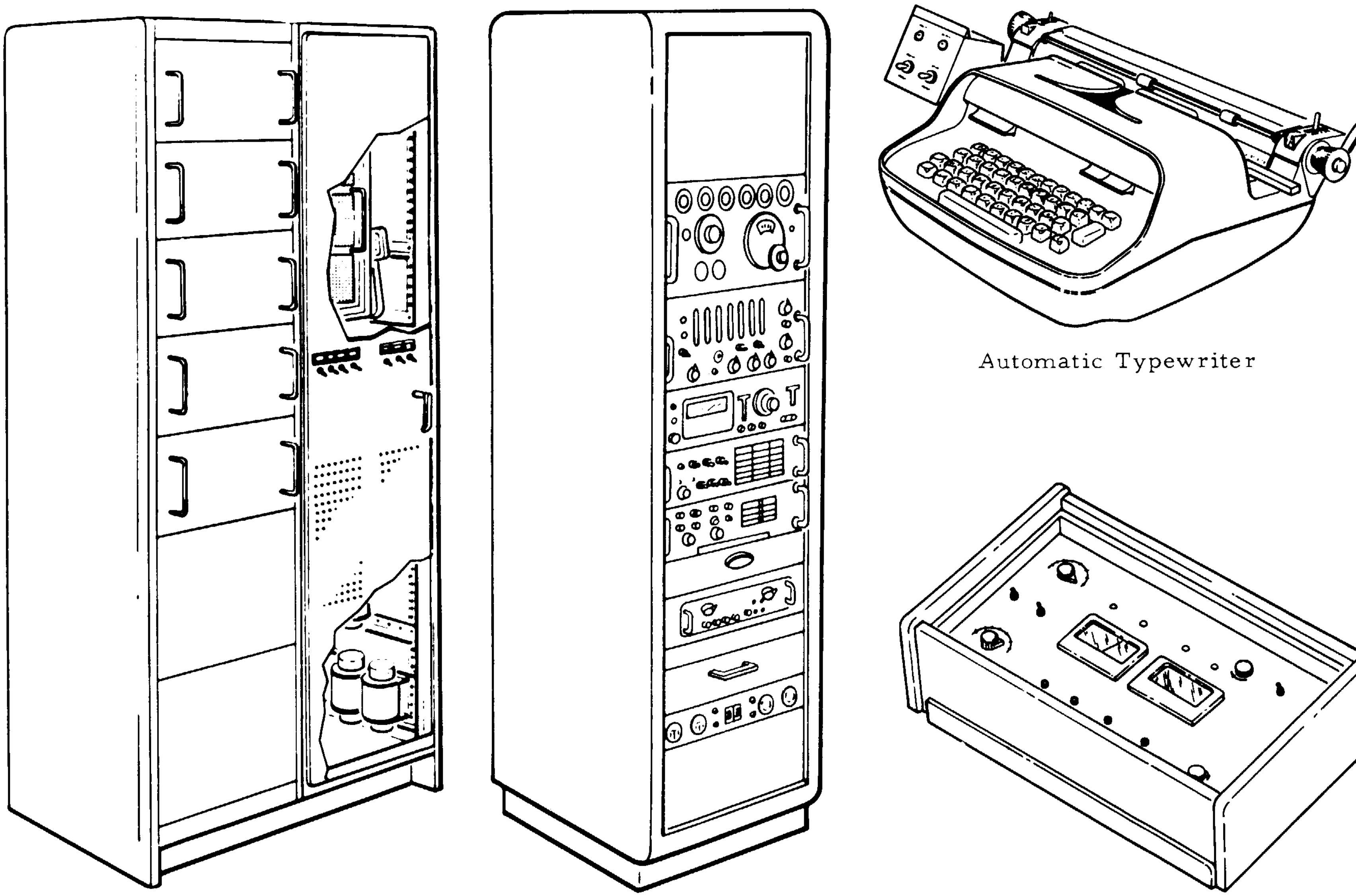


Figure 17-1. Test, Checkout, and Monitoring Equipment, S-IVB (4 of 4)

Table 17-6. Transportation, Protection, and Handling Equipment, S-IVB

Figure	Equipment	Function
17-2 (Sheet 1)	Transportation Cradles, Dollies, and Handling Kits	a. Provides support for the S-IVB stage during all phases of land and water transportation. b. Provides overland mobility for the S-IVB stage between manufacturing, dock facilities, static test, and launch sites. c. Provides rings for mounting and hoisting the S-IVB stage so that the induced loads are transmitted safely to the stage structure.
17-2 (Sheet 1)	Transport Kit, Protective and Tiedown	Provides environmental protection during all phases of transport.
17-2 (Sheet 2)	Container Interior Access Kit	a. Provides access while the stage is in the vertical position. b. Facilitates interior maintenance and checkout operations. c. Provides interior lighting in the container.
17-2 (Sheet 2)	Hoist Kit	Provides hardware for lifting the S-IVB stage to and from the dollies to all ground and water carriers, and vertical assembly and staging.
17-2 (Sheet 3)	Shipping and Handling Equipment, Flared Aft Interstage	a. Provides hardware for transporting and handling the aft interstage in two sections. b. Maintains the interstage section shape and environmentally protects the interstage during transportation.
17-2 (Sheet 3)	Forward Section Vertical Access Kit	Provides access to the forward section of the stage for maintenance while the stage is in the vertical position.
17-2 (Sheet 3)	Small Arms Protective Cover	Provides protection for the S-IVB stage from small arms fire during barge transportation.
17-2 (Sheet 4)	Weight and Balance Kit, Stage and Aft Interstage	Used to determine the weight and center of gravity of the S-IVB stage and aft interstage (at Huntington Beach, California).

Table 17-6. Transportation, Protection and Handling Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
17-2 (Sheet 4)	Aft Section Vertical Access Kit	Provides access to the aft section of the stage for maintenance while the stage is in the vertical position.
17-2 (Sheet 4)	Forward Skirt End Protective Cover	Provides protection to the forward area of the S-IVB stage from rain and other elements while the stage is in the test stand.
	Handling Kit, Retromotor	Provides hardware for storing, lifting, assembling, and installing and retromotors.
	Aft Umbilical Kit, Static Test Stand	Provides hardware for supporting pressurized gas lines, attaching the umbilical connections to the S-IVB stage, and separating the umbilical carrier from the stage. (Sacramento, California.)
	Forward Umbilical Kit, Static Test Stand	Provides hardware for supporting electrical cables, pneumatic lines and a GH_2 vent line, attaching the umbilical connection to the S-IVB stage, and separating the umbilical carrier from the stage.
	Forward Umbilical Kit, Checkout Stand	Provides hardware for supporting electrical cables and pressure lines while maintaining their attachment to the S-IVB stage. (Huntington Beach, California.)
	Aft Umbilical Kit, Checkout Stand	Provides means of supporting electrical cables and the air conditioning duct, and of maintaining their attachment to the S-IVB stage. (Huntington Beach, California.)
	Fixture, Engine Actuator Adjustment	Provides hardware for removal and replacement of the engine actuator without changing the length setting of the removed actuator.
	Alignment Kit, Vehicle Mounting	Provides hardware for aligning and installing the stage in the test stand. (Sacramento, California.)
	Alignment Kit, Engine	Provides hardware for aligning the J-2 engine with the S-IVB stage.
	Special Tool Kit	Provides all tools required for adequate maintenance and handling of the S-IVB stage.

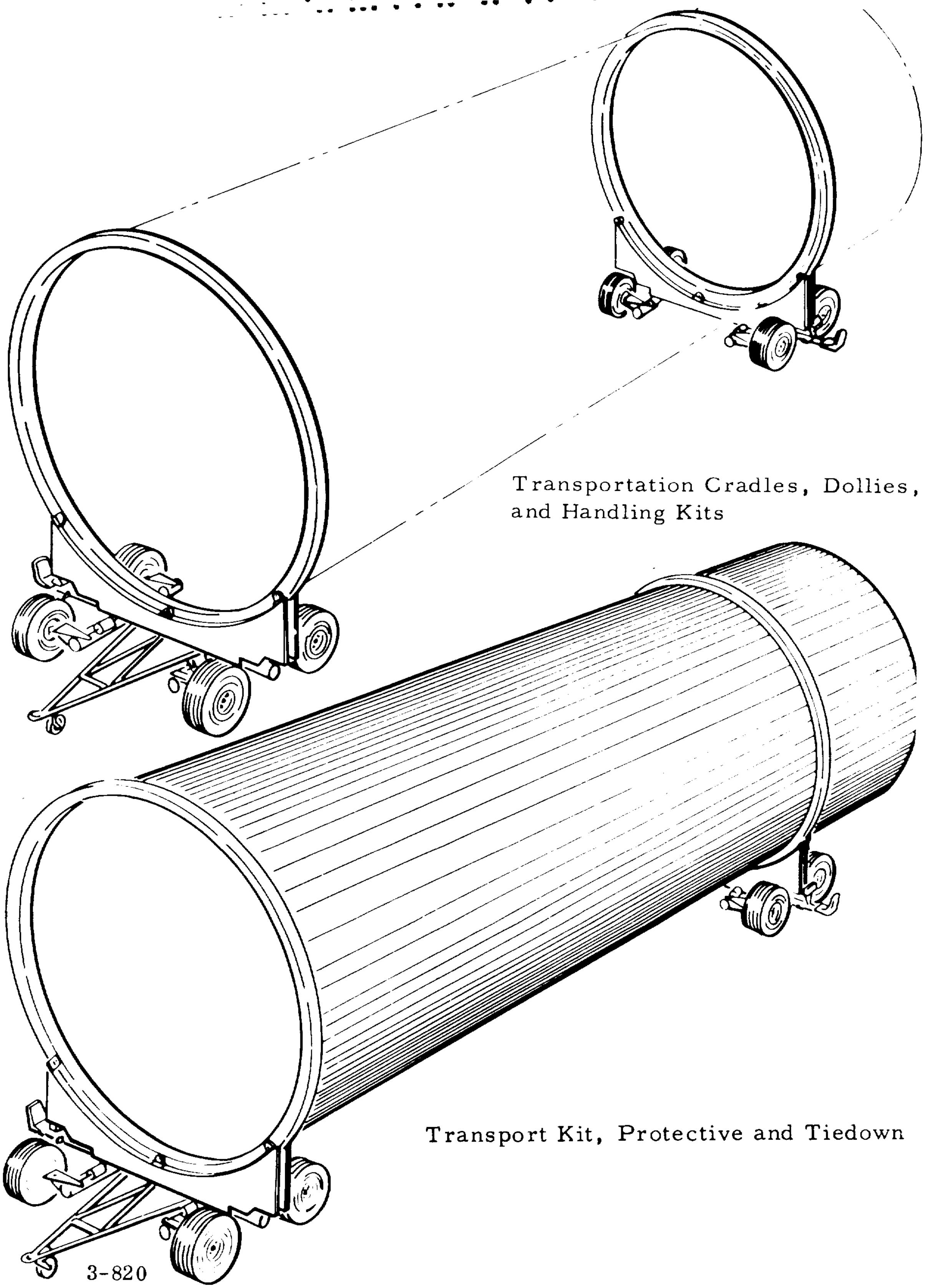
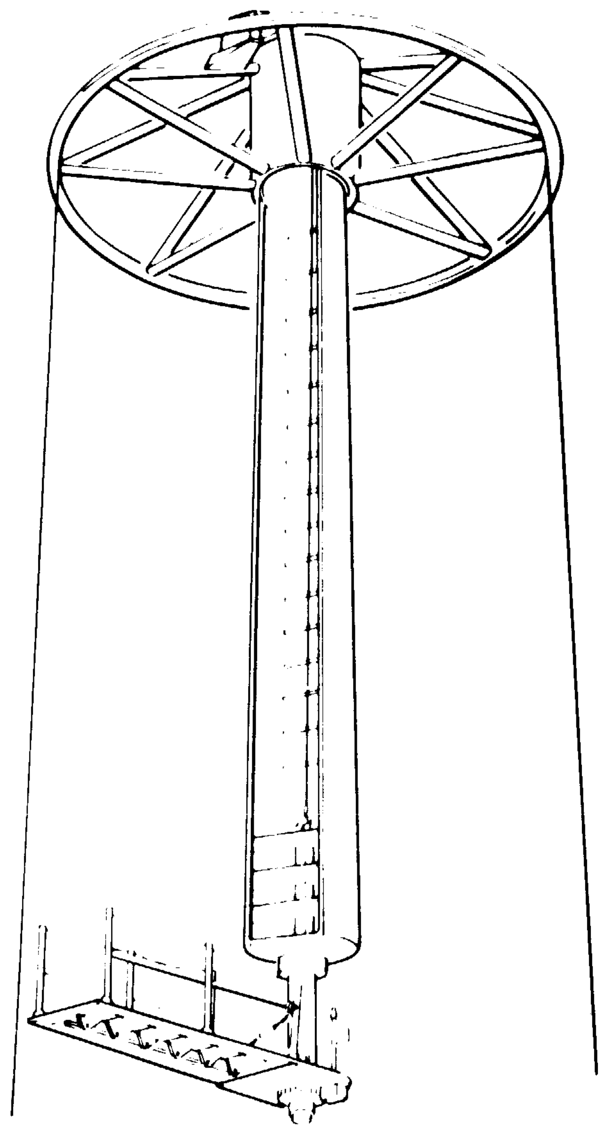
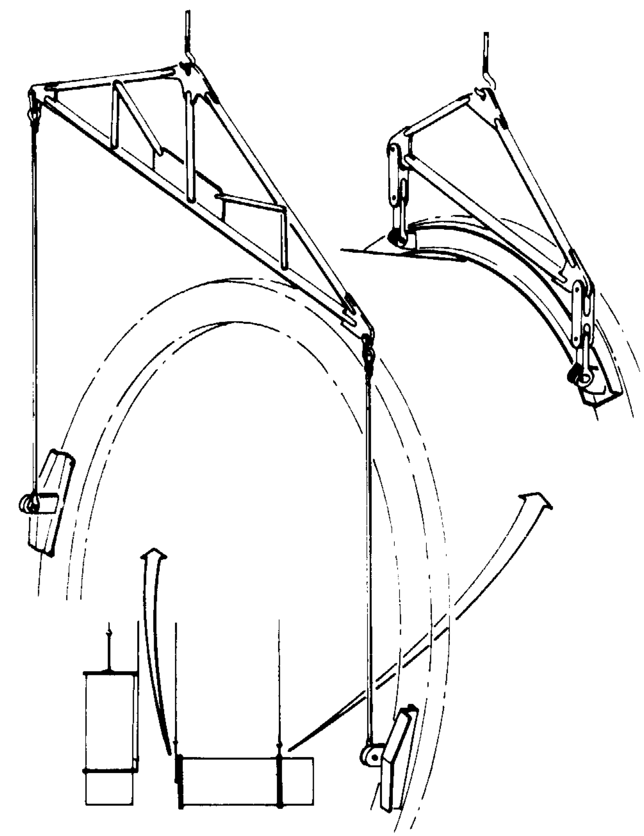


Figure 17-2. Transportation, Protection, and Handling Equipment,S-IVB (1 of 4)



Container Interior Access Kit



Hoist Kit

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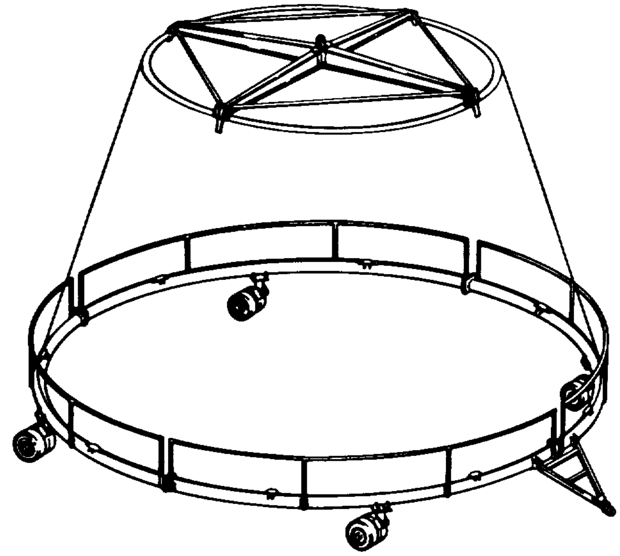
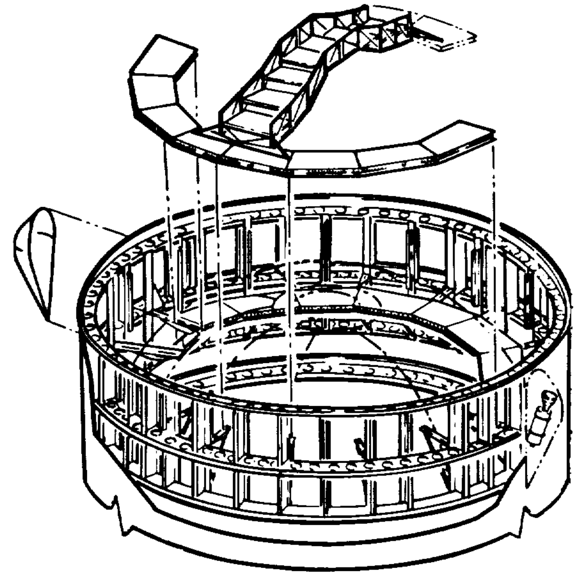
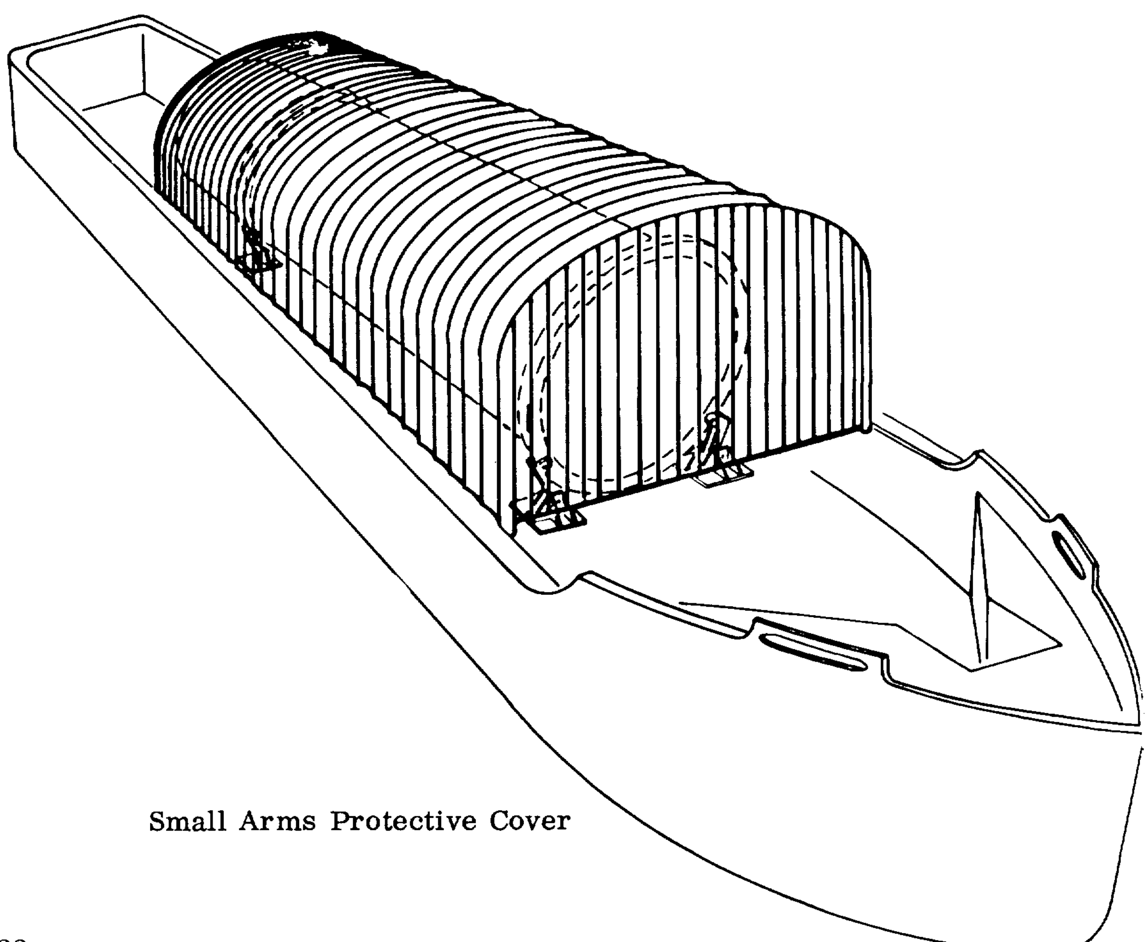
Figure 17-2. Transportation, Protection, and Handling Equipment, S-IVB (2 of 4)



Shipping and Handling Equipment,
Flared Aft Interstage



Forward Section Vertical
Access Kit



Small Arms Protective Cover

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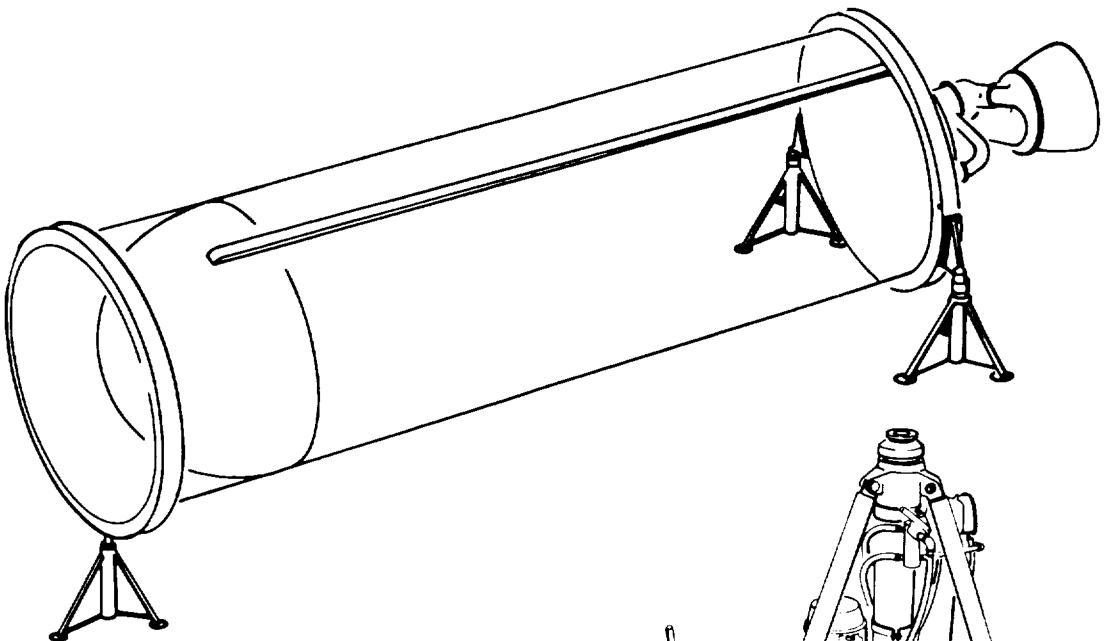
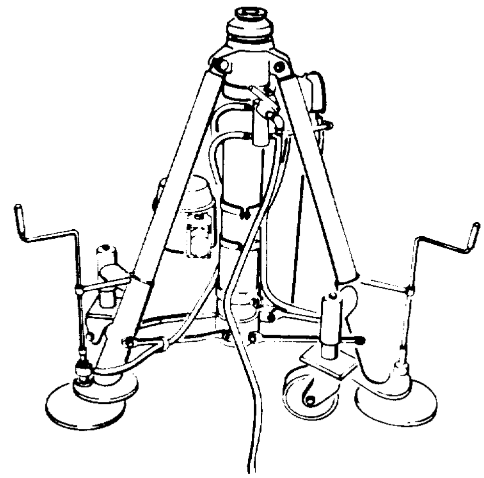
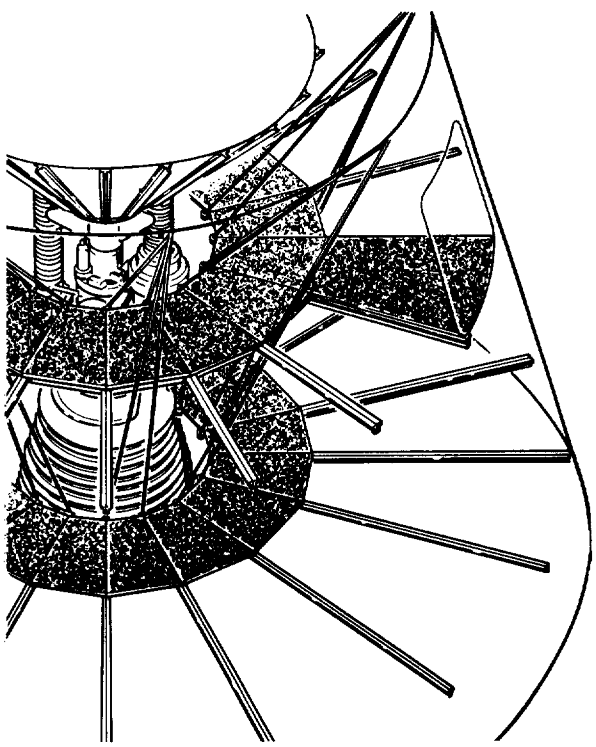
Figure 17-2. Transportation, Protection, and Handling Equipment,S-IVB (3 of 4)



Stage and Aft Interstage
Weight and Balance Kit





Aft Section Vertical Access Kit

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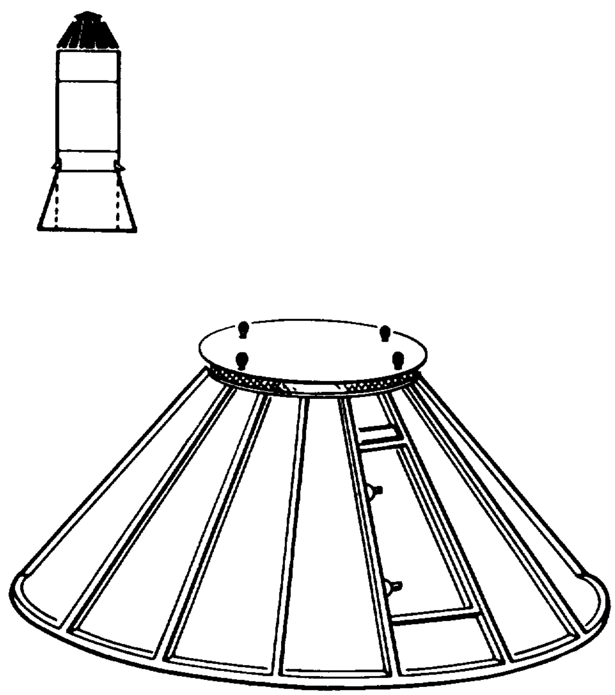


Forward Skirt End Protective Cover

Figure 17-2. Transportation, Protection, and Handling Equipment, S-IVB (4 of 4)

Table 17-7. Servicing Equipment, S-IVB

Figure	Equipment	Function
17-3	Liquid Oxygen Valve Control Complex	Controls the transfer of the LOX from the ground storage facilities into the stage until the stage LOX container is filled and topped.
17-3	Liquid Hydrogen Valve Control Complex	a. Controls the transfer of LH_2 from the ground storage facilities to the stage until the stage LH_2 container is filled and topped.
17-3	Gas Heat Exchanger	Receives regulated, ambient gaseous helium and hydrogen from the automatic stage servicing console "A", subcools these gases to the proper temperature, and returns them to console "B" and thence to the stage during countdown.
17-3	Vacuum Pumping Unit	Used in periodically evacuating, to required values, the individual vacuum jackets of various S-IVB stage and GSE components before countdown.
17-3	Auxiliary Propulsion System Mobile Servicer	Transports nitrogen tetroxide (oxidizer) from the facility storage area, and transfers it to the S-IVB stage auxiliary propulsion modules.
17-3	Automatic Stage Servicing Pneumatic Console A (DSV-4B-319)	Provides ambient gaseous hydrogen, nitrogen, and helium to meet the S-IVB stage propulsion system requirements during checkout operations and for propellant loading, unloading, purging, etc., during countdown.
17-3	Automatic Stage Servicing Pneumatic Console B (DSV-4B-320)	Provides ambient and cold gaseous hydrogen and helium for the S-IVB stage propulsion system requirements during checkout, and for pressurization and propellant-loading operations during countdown.
17-3	Stage Checkout Pneumatic Console (DSV-4B-321)	Provides ambient GN_2 and helium to meet the S-IVB stage propulsion system requirements for leak and functional checkouts.
	Automatic Checkout Accessories Kit	Provides the necessary flexible hoses, fittings, disconnects, etc., to make the connections between the S-IVB stage instrumentation taps and the stage servicing and checkout pneumatic consoles for automatic leak and functional checkout of the propulsion system.

Table 17-7. Servicing Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
	Hydraulic Servicer	Supplies hydraulic fluid to the engine hydraulic system of the S-IVB stage for filling, flushing, cleaning, leak checking, air purging, and checking the operation of certain subsystem components.
	Nitrogen Fill Truck	a. Used to pressurize the pneumatic side of the stage hydraulic accumulator. b. Used to purge the stage electronic equipment containers and to fill the hydraulic accumulator.
	Adapter, Turbine Torque Wrench	Used with the J-2 LOX pump to determine if excessive torque loads exist prior to actual firing.
	Aft Interstage Environmental Control System	a. Purges the aft interstage area to minimize fire and explosion hazards during the period that propellants are being loaded or stored in the stage and during test firings. (Sacramento, California) b. Provides a temperature-controlled environment in the aft interstage at proper operating temperatures.
	Forward Skirt Environmental Control System	Supplies coolant to the forward skirt area for environment control. Equipment mounting panels are used as cold plates for heat transfer.

Table 17-8. Auxiliary Equipment, S-IVB

Figure	Equipment	Function
	Propulsion System Preparation Panel	Controls and monitors propellant container pre-pressurization, container and line purges, and engine chilldown.
	Hydraulic and Gimbal Control Panel	a. Provides control for the stage electric auxiliary pump motor.

Table 17-8. Auxiliary Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
		b. Monitors system pressures and fluid temperatures. c. Provides slewing control and displays the slew command and direction on meters for the yaw and pitch planes.
	Pneumatic Consoles Control Panel	Provides manually operated electrical control for regulation and monitoring of temperatures and pressures of the pneumatic systems checkout consoles and the helium precool heat exchanger.
	Propellant Utilization Checkout and Control Panel	Provides the controls and indicators used for partial checkout of the closed loop propellant utilization system.
	Propellant-Loading Control Panel	Controls and monitors the solenoid-actuated valves in the loading systems for LOX and LH_2 during tests at Sacramento, or switches to the electronic computer for automatic loading.
	Propellant-Loading Computer Control Panel	Provides the ON-OFF control for the loading computer and the other controls required for checkout and operation of the propellant-loading computer and its associated circuitry.
	Stage Pneumatic Bottles Control Panel	Provides electrical controls and indications for filling the S-IVB stage pneumatic bottles.
	Control Switching Rack	a. Provides a convenient and flexible means of interconnecting the electrical ground support equipment. b. Provides an interface for the umbilical J-box, facilities, and control and monitor panels and chassis.
	Umbilical Junction Box	a. Provides a transition point between the battleship firing-stand equipment and the battleship stage. b. Provides control relays and contactors required to reduce voltage drop in high current circuits.

Table 17-8. Auxiliary Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
	Patch Panel Junction Box	Provides a convenient and flexible means of interconnecting the battleship firing GSE for checkout and control of the battleship firing stand.
	Stage Systems Power Panel	a. Provides remote control for activating power for the battleship firing equipment and the GSE.
		b. Provides meter indication of facilities and power supply busses, inverter and ground 400-cycle power, and battleship dc buses.
	Engine-Firing Control Panel	a. Provides the necessary circuitry to control and monitor, through the automatic engine-firing system, engine firing on the battleship stand.
		b. Provides manual controls for engine cutoff, ignition detectors, and firing control power.
	Gimbal Power Supply	Provides 60-volt power to the feedback potentiometers located on the engine actuators.
	Test Conductor Panel	a. Displays system readiness and safety conditions.
		b. Provides control for emergency stop.
	Cable Network	Used to interconnect the GSE through the patch panel junction box at the blockhouse and through the control switching rack in the terminal room.
	Inverter Power Supply	Provides regulated 28-volt dc power to the stage inverter.
	Test Stand Cable Network	Provides for the interconnection of electrical and electronic end items at the battleship test stand and the connection of the stand to contractor furnished terminal distributors.

Table 17-8. Auxiliary Equipment, S-IVB (Cont'd)

Figure	Equipment	Function
	External Power Rack	a. Provides regulated 28-volt dc to the sequencer, solenoid-operated valves, and certain stage systems during test stand checkout and firing. b. Provides circuitry to switch automatically to emergency power in case of malfunction.
17-4	Pneumatic Console A (DSV-4B-327)	Provides ambient GN_2 and helium at the proper pressures to meet the S-IVB stage battleship container propulsion system requirements during checkout and countdown at Sacramento.
17-4	Pneumatic Console B (DSV-4B-333)	Provides ambient GN_2, ambient helium and cold helium at the proper pressures to meet the S-IVB stage battleship container propulsion system requirements during checkout and countdown at Sacramento.
17-4	Pneumatic Console C (DSV-4B-328)	a. Provides ambient and cold gaseous hydrogen and helium at the proper pressures to meet the S-IVB stage battleship container propulsion system requirements during checkout and countdown at Sacramento. b. Receives GN_2 for pneumatic valve actuation within the console.
	Aft Interstage Environmental Control System - Battleship	a. Purges the aft interstage area to minimize fire and explosion hazards when propellants are being loaded or stored and during test firings. b. Used to perform the environmental control test utilizing a dummy aft interstage.

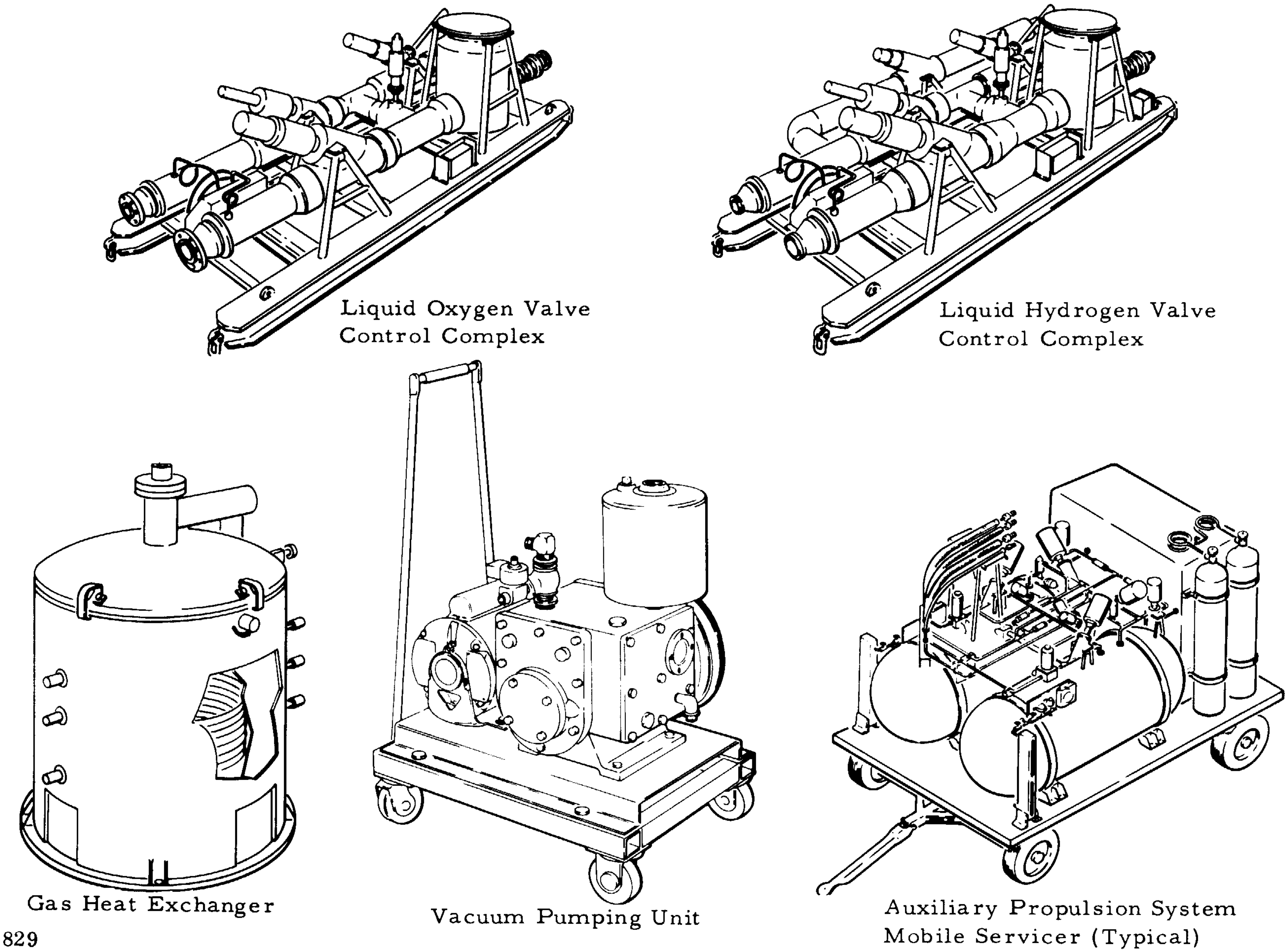
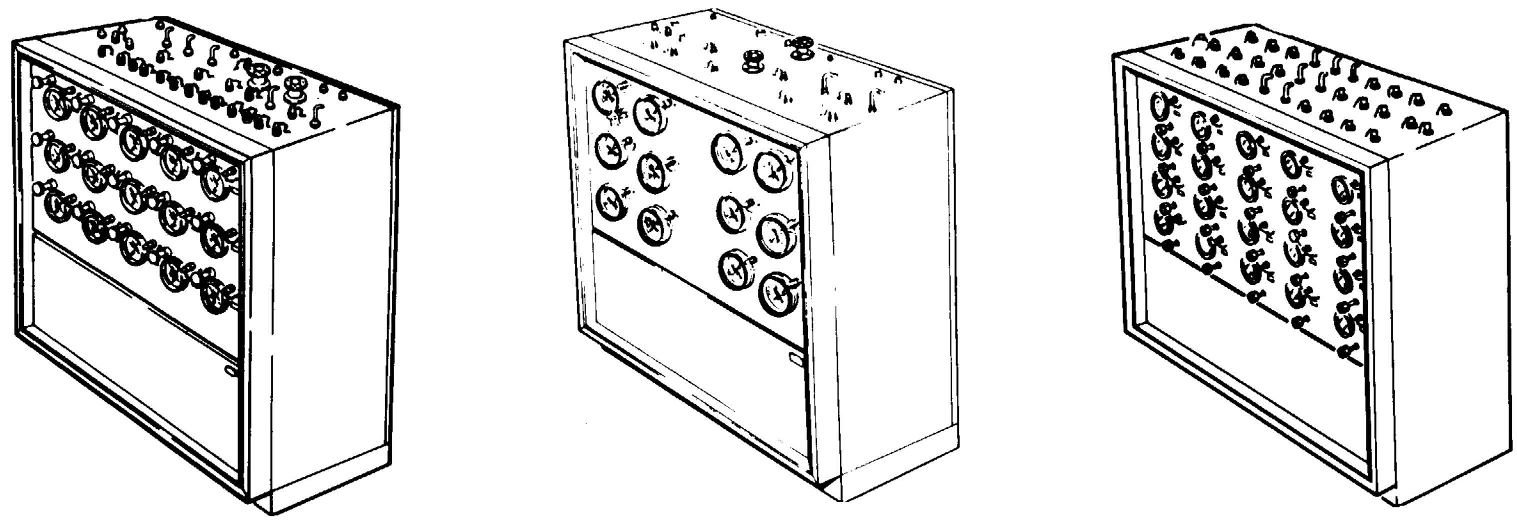


Figure 17-3. Servicing Equipment, S-IVB

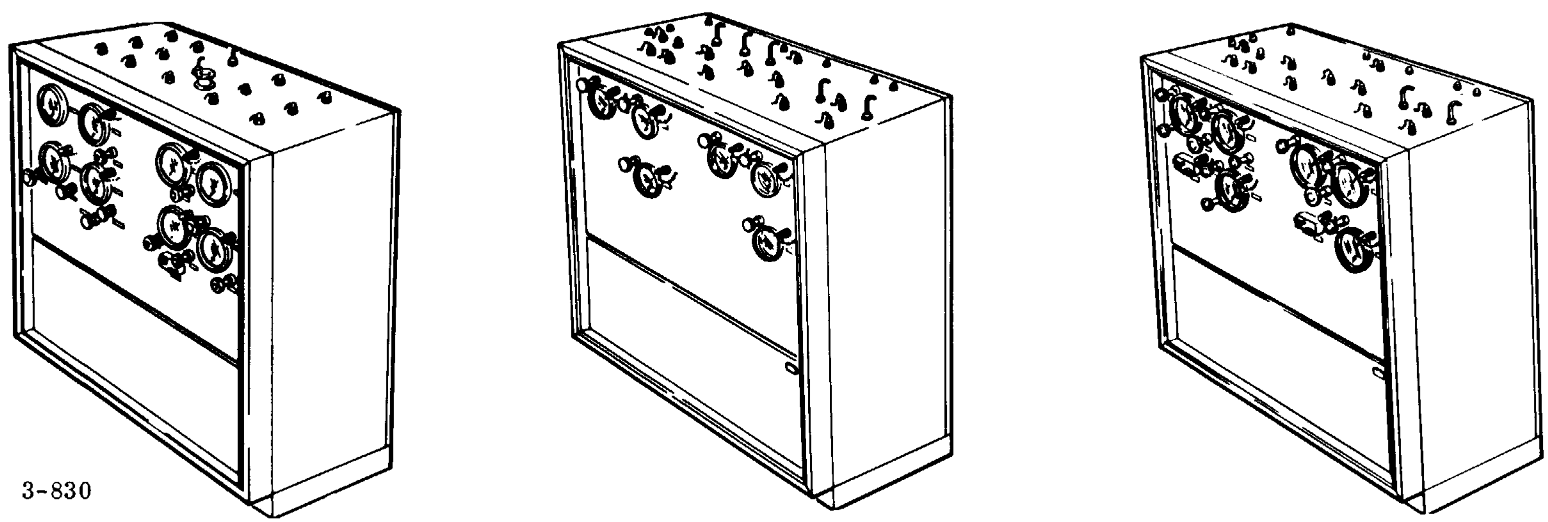


Automatic Stage Servicing Pneumatic Console A

Automatic Stage Servicing Pneumatic Console B

Stage Checkout Pneumatic Console

Figure 17-3. Servicing Equipment, S-IVB



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Pneumatic Console A

Pneumatic Console B

Pneumatic Console C

Figure 17-4. Auxiliary Equipment, S-IVB

CHAPTER 3

SECTION XVIII

STAGE CONFIGURATIONS, SATURN IB

LIST OF ILLUSTRATIONS

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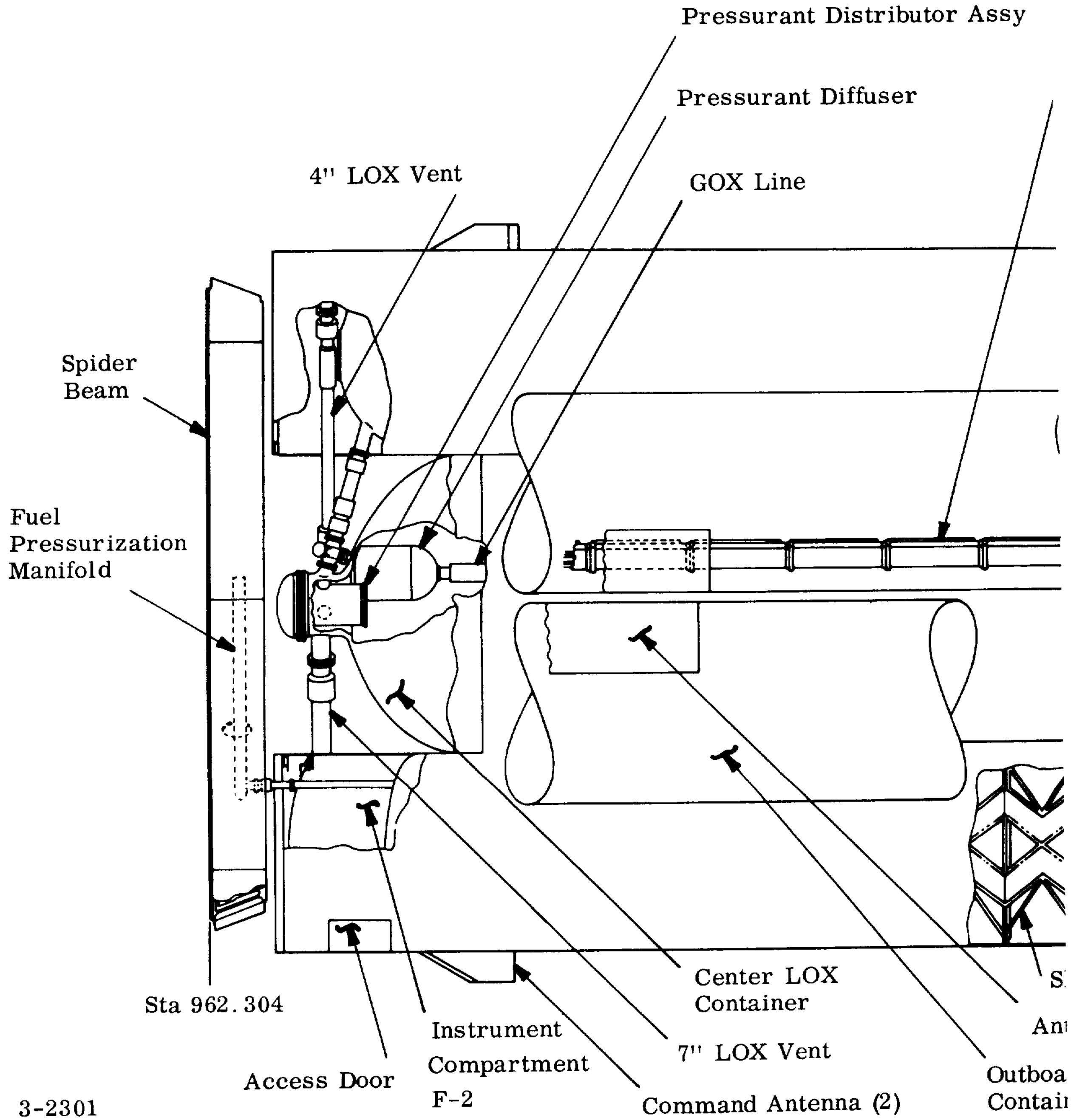
XVIII



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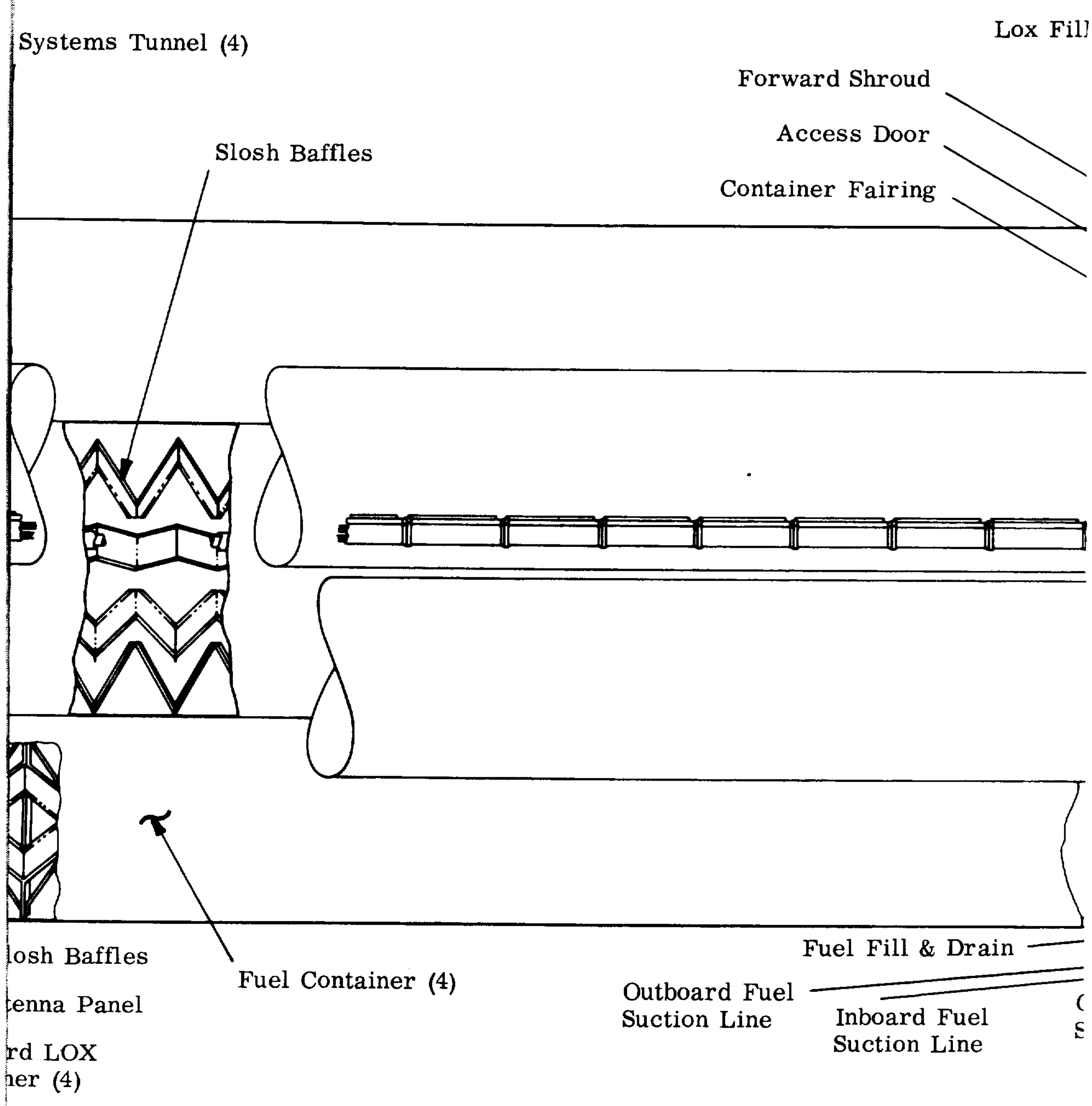
FOLDOUT FRAME



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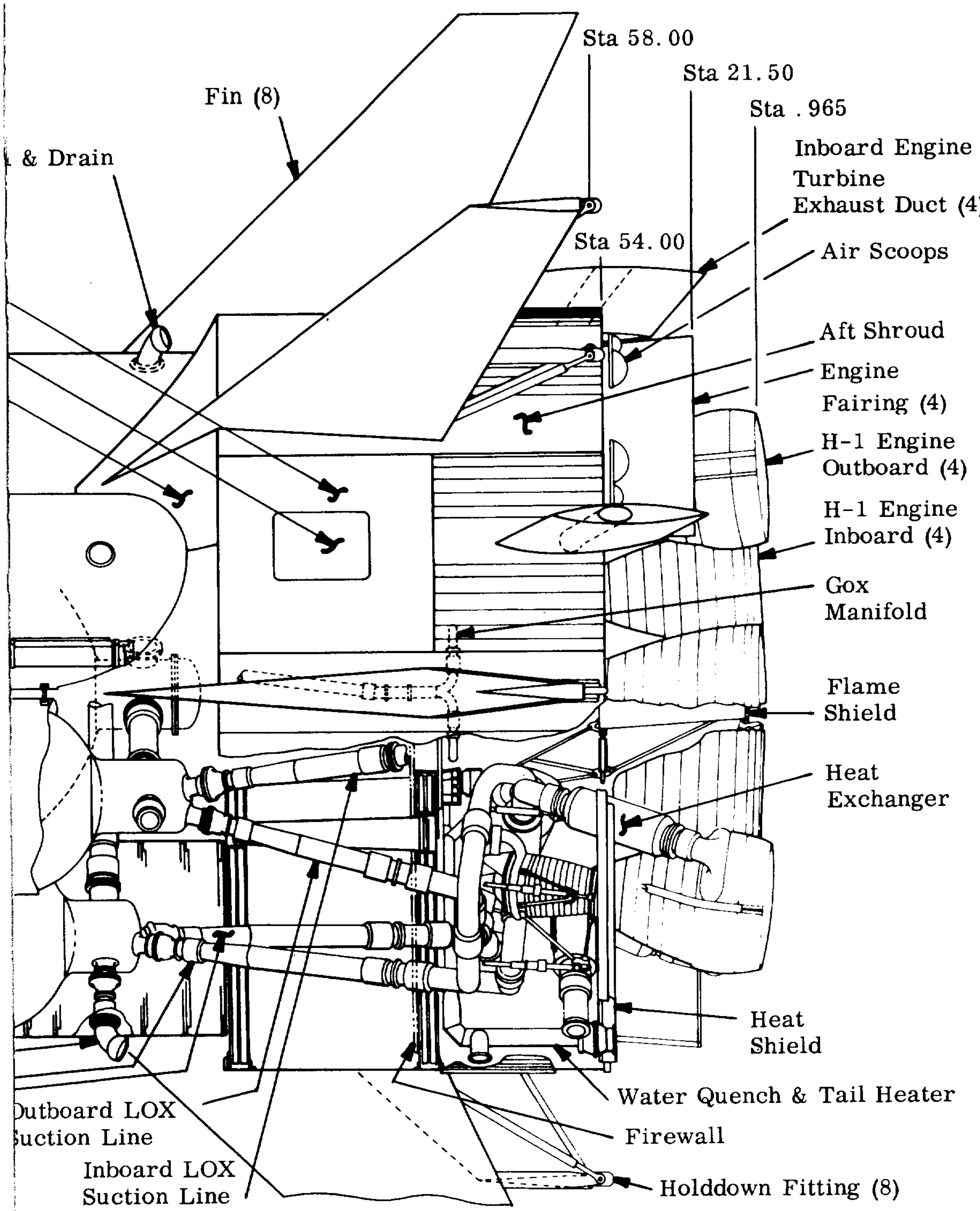
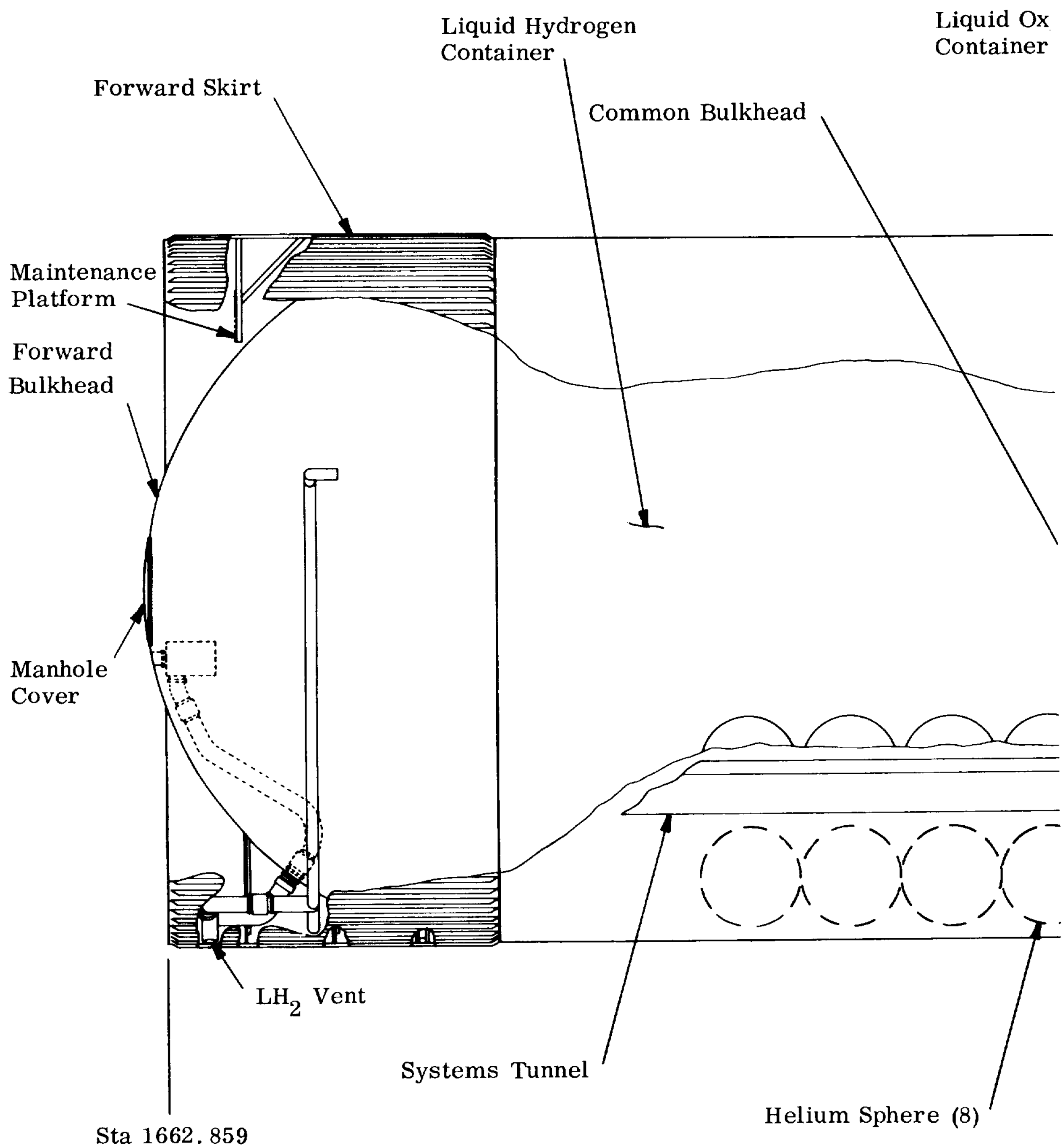


Figure 18-1. S-IB Inboard Profile

FOLDOUT FRAME 3



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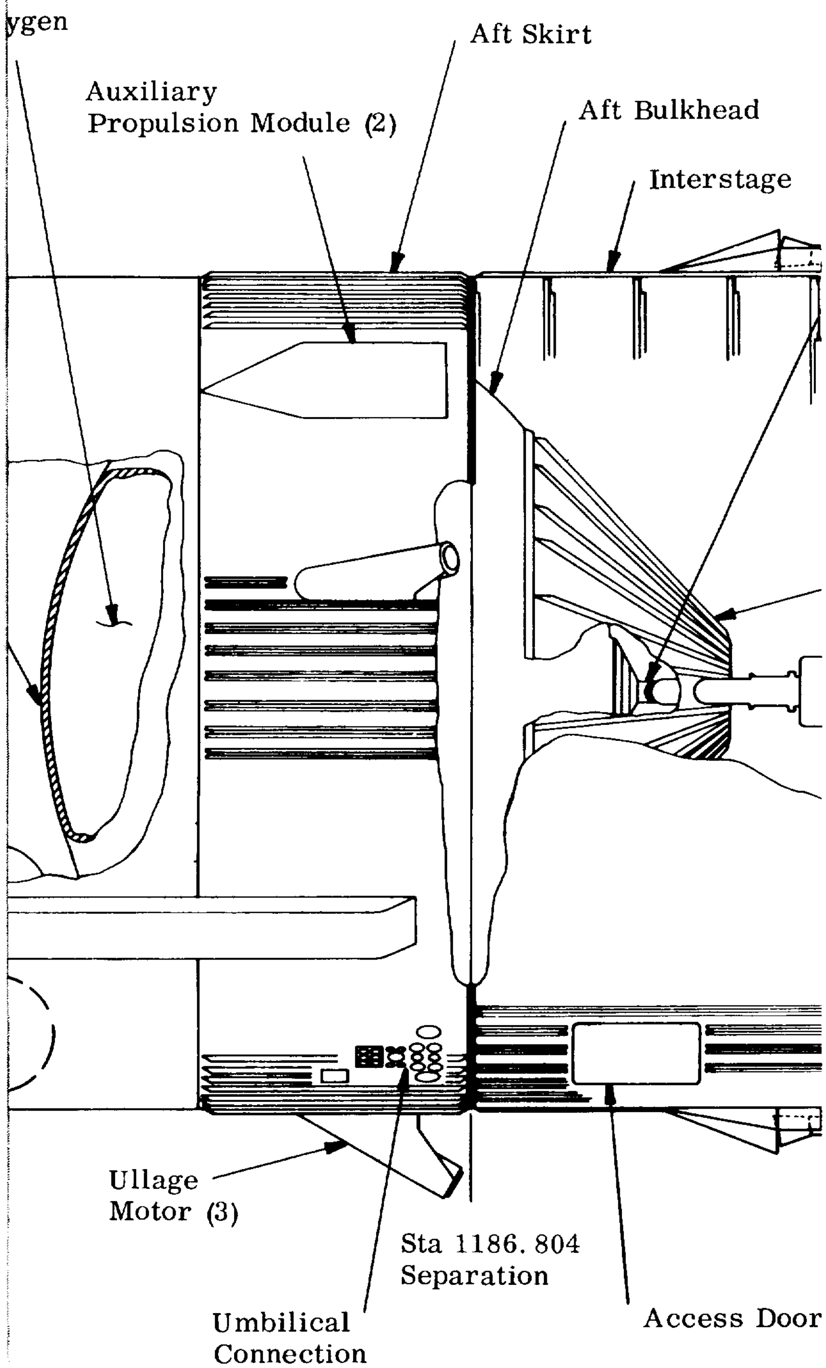
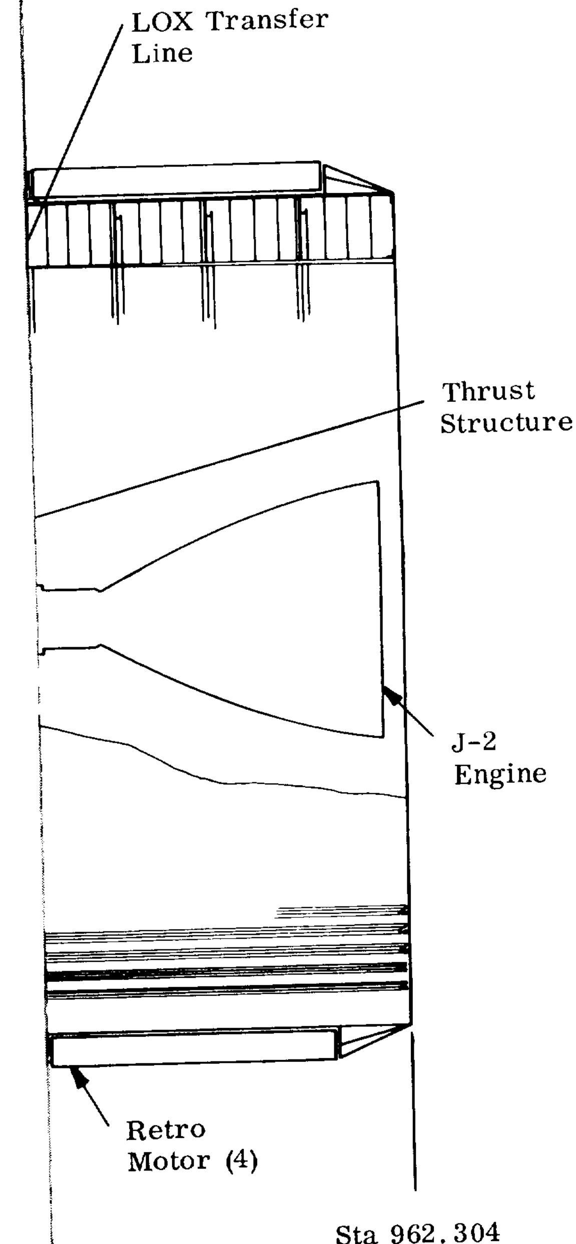
FOLDOUT FRAME 1



Figure

FOLDOUT FRAME 2



18-2. S-IVB Inboard Profile, Saturn IB

FOLDOUT FRAME 3

CHAPTER 4

SECTION XIX

INTRODUCTION

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SECTION XIX.
INTRODUCTION

19-1. SATURN V LAUNCH VEHICLE

The Saturn V launch vehicle, Figure 19-1, consists of an S-IC first stage, an S-II second stage, an S-IVB third stage, and an instrument unit mounted above the third stage. Operational data for the vehicle are listed in Table 19-1.

19-2. SATURN V - APOLLO MISSION OBJECTIVES

The ultimate objective of the Saturn V - Apollo program is manned lunar landing within the present decade. Fifteen space vehicles are planned for attaining this objective.

In the first two Saturn V - Apollo flights, SA-501 and SA-502, the mission objectives are flight testing the launch vehicle and testing of the CM heat shield under lunar re-entry velocity conditions.

The objective of SA-503, through SA-506 flights is qualification or man-rating of the space vehicle.

The seventh Saturn V - Apollo Vehicle (SA-507) is targeted for achievement of manned flight with a potential of a lunar mission.

Mission objectives for the eighth and subsequent Saturn V - Apollo flights will be defined later in the program. However, it is anticipated that the mission objectives for the initial flights will follow a sequence similar to the following:

a. Circular earth orbit with validation of orbital checkout procedures.

b. Circumlunar orbit with manned observation of potential lunar landing areas.

c. Lunar landing - two astronauts and a minimum of 215 pounds of scientific equipment shall be landed on the moon for purposes of exploration of the lunar surface to distances of approximately one-half mile from the landing site. The astronauts and a minimum of 80 pounds of scientific payload shall be returned to earth and safely re-

[redacted]

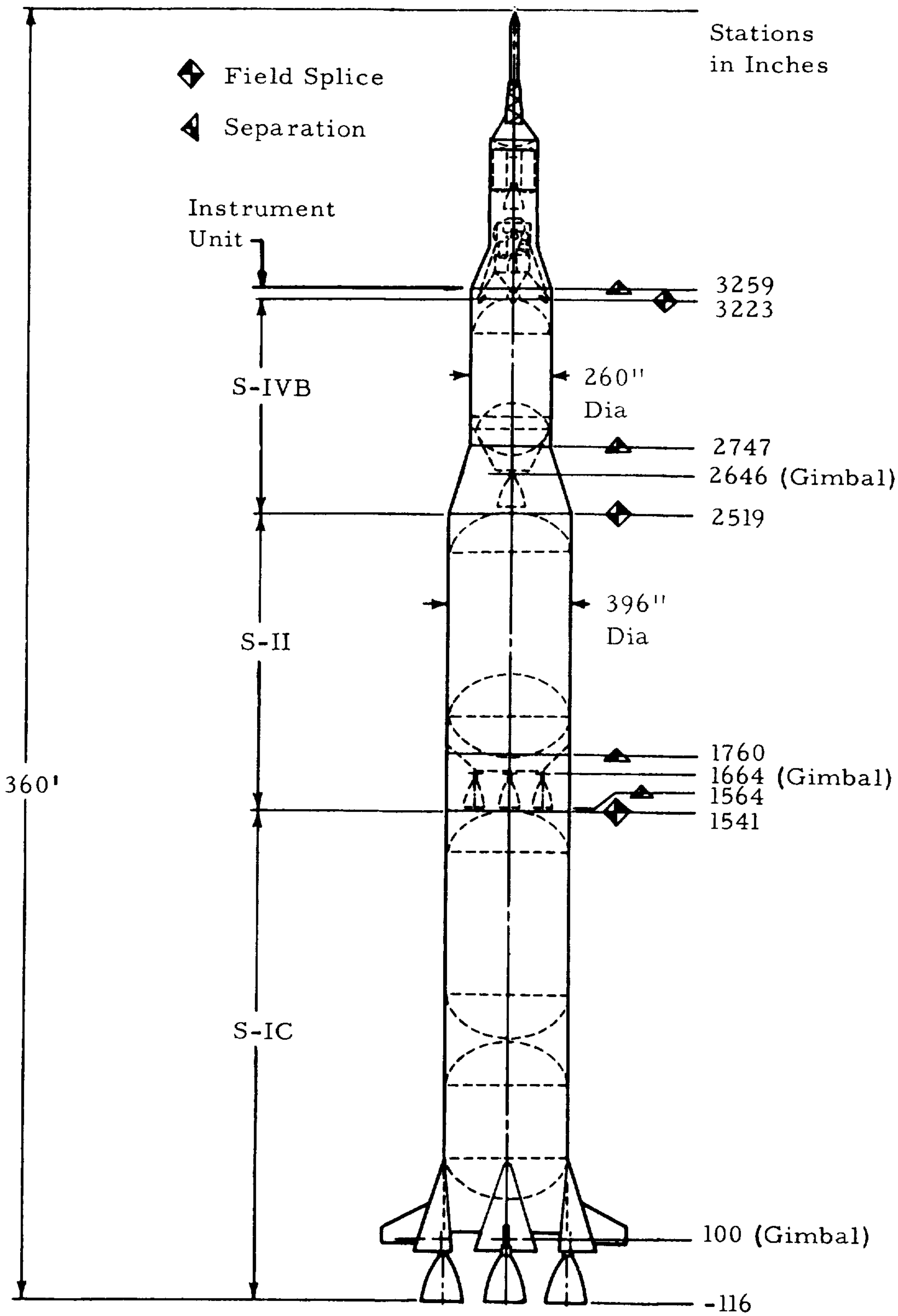

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Figure 19-1. Saturn V Launch Vehicle

[redacted]

[redacted]

Table 19-1. Saturn V Operational Data

Item	Data
VEHICLE	
Number of stages	3
Length - without spacecraft	281.2 feet
Maximum diameter - without fins	
- with fins	63.0 feet
[1]Launch vehicle weight - at ground ignition	6,102,000 pounds
Payload type	Apollo Spacecraft
[2]Payload weight - at ground ignition	96,600 pounds
[3]Injection weight - lunar transfer orbit	90,000 pounds
S-IC STAGE	
Prime contractor	Boeing Aircraft Co.
Length	138.1 feet
Diameter - without fins	33.0 feet
- with fins	63.0 feet
Stage weight - at ground ignition	4,711,000 pounds
Dry weight	287,000 pounds
Engines	Rocketdyne F-1 (5)
Total nominal thrust (sea level)	7,500,000 pounds
Propellants	LOX and RP-1
Mainstage propellant weight	4,245,000 pounds
Mixture ratio (oxidizer to fuel)	2.25:1
Specific impulse (sea level)	265 seconds
S-II STAGE	
Prime contractor	North American Aviation, Inc.
Length	81.5 feet
Diameter	33.0 feet
[4]Stage weight - at ground ignition	1,002,000 pounds
[4]Dry weight	75,000 pounds
Engines	Rocketdyne J-2 (5)
Total nominal thrust (vacuum)	1,000,000 pounds

[redacted]

Table 19-1. Saturn V Operational Data (Cont'd)

Item	Data
Propellants	LOX and LH_2
Mainstage propellant weight	913,000 pounds
Mixture ratio (oxidizer to fuel)	5:1
Specific impulse (vacuum)	426 seconds
S-IV STAGE	
Prime contractor	Douglas Aircraft Co.
Length	59.3 feet
Diameter (forward of interstage)	21.7 feet
[5]Stage weight - at ground ignition	262,000 pounds
[5]Dry weight	22,000 pounds
Engine	Rocketdyne J-2 (1)
Total nominal thrust (vacuum)	200,000 pounds
Propellants	LOX and LH_2
[6]Mainstage propellant capacity	230,000 pounds
Mixture ratio (oxidizer to fuel)	5:1
Specific impulse (vacuum)	426 seconds
INSTRUMENT UNIT	
Prime contractor	MSFC
Length	3.0 feet
Diameter	21.7 feet
Weight - at ground ignition	3500 pounds

[1]Includes three stages, instrument unit, payload and LES.
[2]Includes 6600 pounds for the LES.
[3]72 hour lunar transfer orbit, payload only.
[4]Excludes 13,800 pounds for S-IC/S-II interstage and ullage motors.
[5]Excludes 7400 pounds for S-II/S-IVB interstage and retromotors.
[6]Includes orbital launch window propellants and flight performance reserve propellants.

Note: Weights in this table are specification weights from Memorandum No. M-P&VE-V-33, "Saturn I, IB and V Launch Vehicle Specification, Weights and Compatible Trajectories, dated May 13, 1963.

covered from land or water impact.

Detailed information about the Saturn V - Apollo mission objectives, as far as defined, and flight data is listed in Table 19-2.

19-3. MISSION PROFILE

The Saturn V - Apollo mission profile for the lunar landing mission is illustrated in Figure 19-2. The mission is achieved by the lunar-orbit rendezvous (LOR) mode. In this mode, the launch vehicle, by means of S-IC boost, S-II boost, and first burn of the S-IVB stage propel the spacecraft (consisting of a CM, SM and LEM) into a 100-nautical mile earth parking orbit. After checkout of crew and space vehicle, a second burn of the S-IVB stage injects the spacecraft into a lunar transfer trajectory. After engine cutoff, the S-IVB maintains the attitude of the LEM while the CSM (CM and SM combination) separates, turns around and docks, nose to nose, with the LEM. At this point, the spacecraft separates from the S-IVB/IU and the SM provides the propulsion for midcourse corrections and injection into a lunar parking orbit. Two of the three crew members transfer to the LEM which separates form the CSM and decends to the lunar surface. The third crew member remains in the CSM orbiting aroung the moon. After lunar exploration, the two crew members ascend in the LEM on a trajectory that permits rendezvous with the orbiting CSM. After the LEM crew has transferred to the CM, the LEM is jettisoned. The SM provides propulsion for return to the vicinity of the earth including midcourse corrections. Before re-entry into the earth's atmosphere the SM is jettisoned and the CM reoriented with the heat shield pointed forward. The CM module is slowed to a safe landing speed by aerodynamic braking and parachute deployment. For a detailed listing of mission events refer to Table 19-3.

The mission of the launch vehicle ends with the final separation of the Apollo spacecraft from the S-IVB/IU, event number 15 of the mission profile.

The launch vehicle mission is divided into prelaunch, launch, ascent, earth orbital and translunar trajectory phases. These phases are defined by the following limits:

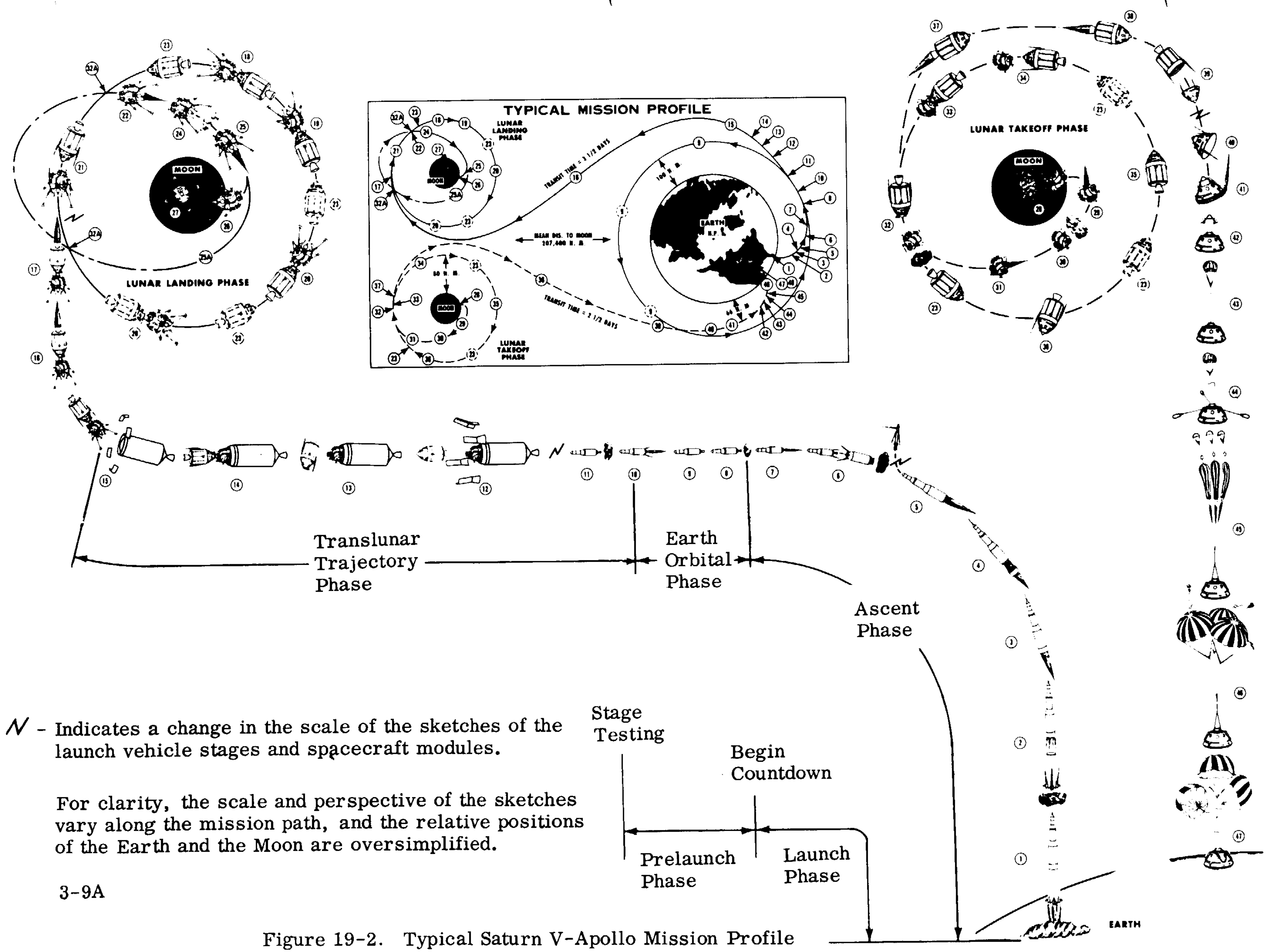
Prelaunch - From start of stage testing to start of countdown.

Launch - From start of countdown to liftoff

Ascent - From liftoff to earth orbit injection.

Table 19-2. Saturn V-Apollo Mission Objectives and Flight Data

(To be supplied at a later date.)



$\mathcal{N}$ - Indicates a change in the scale of the sketches of the launch vehicle stages and spacecraft modules.

For clarity, the scale and perspective of the sketches vary along the mission path, and the relative positions of the Earth and the Moon are oversimplified.

3-9A

Figure 19-2. Typical Saturn V-Apollo Mission Profile

Table 19-3. Description of Typical SATURN V-APOLLO Mission

*Event No.	Approx. Time After Liftoff (Sec.)	Event
1	0	Liftoff of SATURN V-APOLLO Space Vehicle (SV) from AMR Launch Complex No. 39.
		Start roll to align SV pitch plane with flight azimuth. Start time tilt. (By launch vehicle (LV) systems.)
		Arrest roll (SV correctly aligned with flight azimuth).
		Activate accelerometer control of LV guidance and control system.
		Deactivate accelerometer control of LV guidance and control system.
		Arrest time tilt.
		Shut down center first-stage (S-IC Stage) engine.
		Shut down outboard first-stage engines, beginning staging period. Start timing for stage separation sequence.
		Ignite second-stage (S-II Stage) ullage motors.
2		Separate first stage from second stage. Transfer control functions from first to second stage. Ignite first-stage retromotors.
3		Start second-stage engines, ending staging period.
4		Jettison S-II aft interstage at approximately full second-stage thrust.
5		Jettison Launch Escape System from APOLLO Spacecraft (SC).
		Start Path Guidance Mode.
		Shut down all five second-stage engines, beginning staging period. Start timing for stage separation sequence.
		Ignite third-stage (S-IVB Stage) ullage motors

*No. Refers to Figure 19-2. (Major events indicated only)

Table 19-3. Description of Typical SATURN V-APOLLO MISSION (Cont'd)

*Event No.	Approx. Time After Liftoff (Sec.)	Event
6		Separate second stage from third stage. Transfer control functions from second to third stage. Ignite second-stage retromotors.
7		Ignite third-stage engine, ending staging period.
		Resume Path Guidance Mode.
8		Inject SC into 100-naut. mi. (185-km) circular Earth parking orbit. Shut down third-stage engine.
		Receive confirmation from Integrated Mission Control Center (IMCC) regarding acceptability of parking orbit. Check out SC systems.
		Compute initial conditions for achieving lunar transfer orbit from Earth parking orbit (by both SC guidance system computer and ground-based support system).
		Ignite third-stage hydrogen venting ullage motors for brief burn. (Repeat at intervals).
9		Check out crew and equipment.
		Receive command generated by IMCC for continuing mission.
		Correct attitude of SC (by LV attitude control system) for injection of SC into lunar transfer trajectory.
		Ignite third-stage ullage motors.
10		Ignite third-stage engine to inject SC into lunar transfer trajectory. Control powered flight by LV or spacecraft Command Module (CM) guidance system.
11		Shut down third-stage engine (by LV or CM guidance system).
		Receive confirmation from IMCC regarding acceptability of lunar transfer trajectory.

*No. Refers to Figure 19-2. (Major events indicated only)

Table 19-3 Description of Typical SATURN V - APOLLO Mission (Cont'd)

*Event No.	Approx. Time After Liftoff (Sec.)	Event
		Check out crew and equipment.
12		Jettison forward section of spacecraft Adapter. Separate spacecraft Command and Service Modules (CM/SM) from spacecraft Lunar Excursion Module, LV Instrument Unit and third stage (LEM/IU/S-IVB).
13		Initiate turnaround of CM/SM
14		Dock CM/SM to LEM/IU/S-IVB.
15		Jettison aft section of spacecraft Adapter, Instrument Unit and third stage, ending LV mission.
16		Execute midcourse correction of lunar transfer trajectory. (Repeat as necessary).
17		Ignite SM engine for transfer of SC into approximately circular 80-naut. mi. (148-km) lunar orbit.
18		Coast in lunar orbit. Check out crew and equipment.
19		Transfer two members of crew from CM to LEM. (Third man remains in CM.)
20		Check out LEM crew and equipment. Reconnoiter lunar surface.
21		Separate LEM from CM/SM. Correct LEM attitude for descent to lunar surface.
22		Ignite LEM landing stage engine; initiate descent.
23		Continue CM/SM lunar-orbital coast.
24		Cut off LEM engine. Coast in elliptical orbit to vicinity of lunar surface.
25		Re-start LEM engine; brake LEM out of elliptical orbit.
25A		(If lunar landing is not possible, omit Events Nos. 25 through 32 and go to Event No. 32A.)

*No. Refers to Figure 19-2. (Major events indicated only)

Table 19-3. Description of Typical SATURN V-APOLLO Mission (Cont'd)

*Event No.	Approx. Time After Liftoff (Sec.)	Event
26		Land LEM on lunar surface, after hover and translation maneuvers.
27		Explore lunar surface. Perform experiments. Collect specimens.
28		Launch manned ascent stage of LEM. (Landing stage of LEM remains on Moon.)
29		Lift LEM ascent stage into Hohmann transfer ellipse.
30		Cut off LEM engine. Coast in Hohmann transfer ellipse.
31		Re-start and cut off LEM engine as required to correct course.
32		Execute Lunar-Orbit Rendezvous between LEM ascent stage and orbiting CM/SM.
32A		(If lunar landing was omitted, rendezvous LEM with CM/SM as their orbits intersect.)
33		Return LEM crew and lunar specimens to CM.
34		Jettison LEM ascent stage from CM/SM, leaving it in lunar orbit.
35		Check out crew and equipment.
36		Correct CM/SM attitude.
37		Ignite SM engine; inject CM/SM into Earth transfer trajectory. Cut off SM engine.
38		Execute midcourse correction of Earth transfer trajectory. (Repeat as necessary.)
39		Jettison SM from CM.
40		Orient CM in re-entry attitude (heat shield forward).
41		Re-enter Earth's atmosphere.

*No. Refers to Figure 19-2 (Major events indicated only)

Table 19-3. Description of Typical SATURN V-APOLLO Mission (Cont'd)

*Event No.	Approx. Time After Liftoff (Sec.)	Event
42		Jettison CM heat shield (at 50,000-ft. altitude).
43		Deploy drogue parachute (at 25,000-ft. altitude).
44		Jettison drogue parachute and deploy pilot parachutes (at 15,000-ft. altitude).
45		Deploy reefed main parachutes.
46		Deploy main parachutes fully.
47		Alight on surface of Earth (on land).

*No. Refers to Figure 19-2 (Major events indicated only)

Earth orbital - From orbit injection to S-IVB restart.

Translunar Trajectory - From S-IVB restart to final payload separation.

19-4. LAUNCH VEHICLE REQUIREMENTS

For the lunar landing mission, the Saturn V launch vehicle is required to inject an Apollo spacecraft payload of 90,000 pounds into a 72 hour translunar trajectory. To accomplish this, this launch vehicle first injects the payload into a 100-nautical mile earth parking orbit by means of successive burns and separation of the S-IC stage, S-II stage and a first burn of the S-IVB stage. After a final checkout of the Apollo spacecraft, the S-IVB stage engine is re-ignited at the proper position in the parking orbit to inject the payload into the translunar trajectory. Final cutoff of the launch vehicle propulsion occurs with the following nominal parameters:

a. Altitude - 155-nautical miles

b. Inertial Velocity - 35,650 Ft./sec.

c. Angle between velocity vector and local horizon - 6.3° degrees

d. Latitude - 31.4 degrees

e. Longitude - 55.4 degrees east of Cape Kennedy

After injection into translunar orbit, the launch vehicle is required to stabilize the LEM while the CSM separates, turns around and docks. At the conclusion of this maneuver the S-IVB/IU completes its mission by separating from the spacecraft and propelling itself into a separation trajectory. Performance of the translunar injection mission requires a total life time of 6.5 hours for the S-IVB/IU systems.

The launch vehicle is subject to the following constraints:

a. Launch site (Cape Kennedy) latitude of 28 degrees, 30 minutes which introduces a minimum orbital inclination of the same degree. This constraint can be overcome by a "dogleg" maneuver in the trajectory.

b. Launch facility, VLF 39, requires a launch azimuth of 90 degrees.

c. Tracking, telemetry and communication networks restrict the vehicle to an azimuth path of 72 degrees to 105 degrees, depending on the network used.

d. Range safety limits flight azimuths to a sector of 45 degrees to 110 degrees.

To optimize vehicle performance and increase crew safety, a minimum vehicle liftoff thrust to weight ratio of 1.25: 1 is specified.

The primary vehicle requirements are accomplished by systems described in this chapter as astrionics, structures, propulsion, mechanical and ground support equipment. Tables 19-4 through 19-8 illustrate the basic requirements of each of these systems for the five phases of the launch vehicle mission. The time function indicated in the table is not to scale as it is intended to indicate only relative phasing of requirements. Although the table is primarily a listing of system requirements, specific major events are included to show their relationship to the requirements.

Detailed information on the systems is presented in sections XX through XXIV. Inboard profiles of each stage are included in section XXV.

Table 19-4. Saturn V Requirements, Prelaunch Phase

SYSTEM/FUNCTION \ EVENT	Begin Stage Testing	LUT in place	Begin Countdown
Astrionics			
Command			
Control Checkout Sequence			
Control Ground Support Activity			
Communications			
Transmit Information for Checkout			
Transmit Mission Planning Information			
Instrumentation			
Provide Vehicle Syst. Information in Support of Checkout			
Checkout			
Provide Stimuli and Comparison Networks for Checkout			
Guidance[1]			
Attitude Control and Stabilization[1]			
Tracking[1]			
Range Safety			
Develop Range Safety Boundary Charts			
Control Hazardous Operations			

Table 19-4. Saturn V Requirements, Prelaunch Phase (Cont'd)

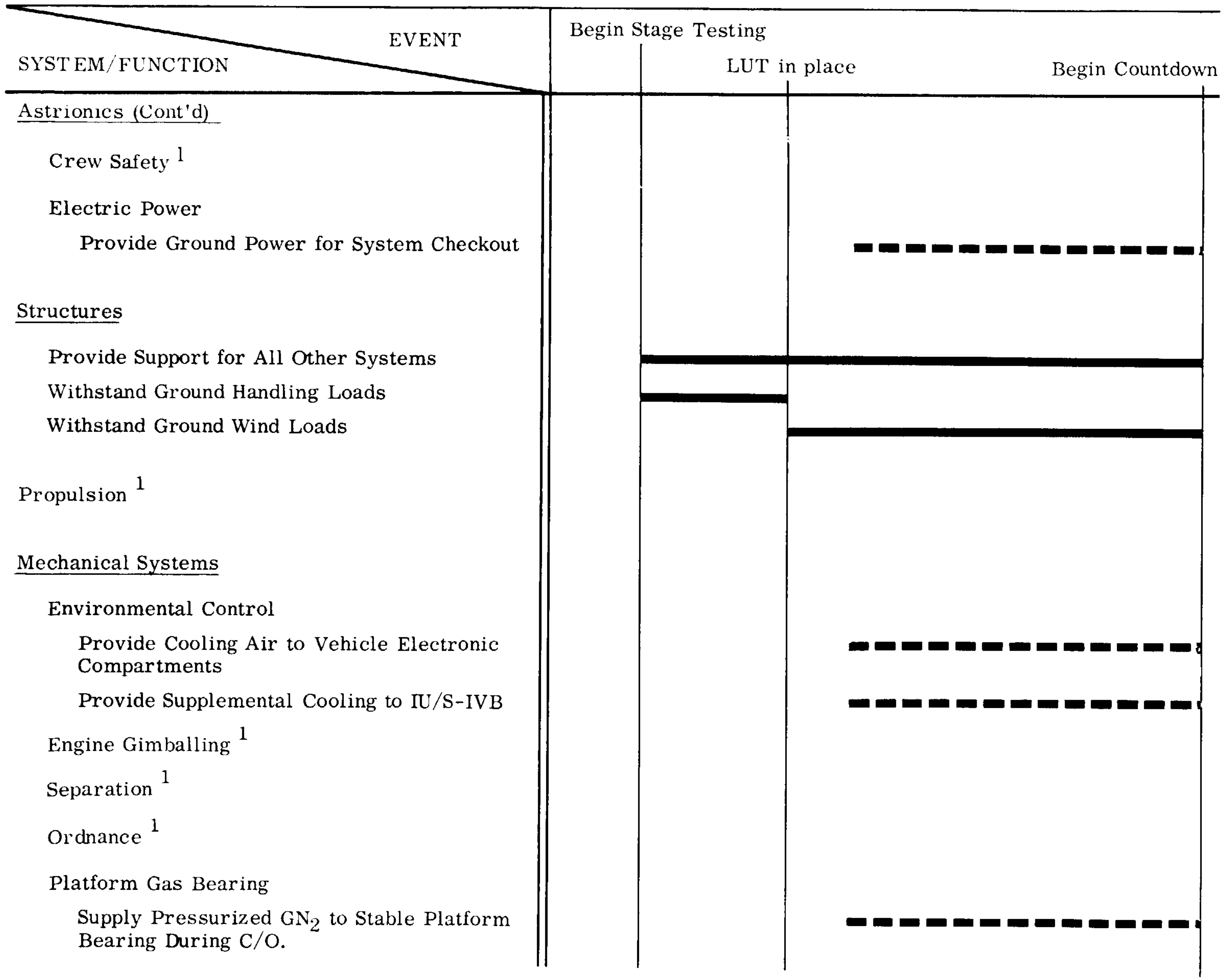


Table 19-4. Saturn V Requirements, Prelaunch Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT		Begin Stage Testing	LUT in place	Begin Countdown
Ground Support Equipment				
Provide Check of Stage Systems				
Provide Ground Handling and Transportation				
Provide Check of Vehicle Systems.				

Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 19-5. Saturn V Requirements, Launch Phase

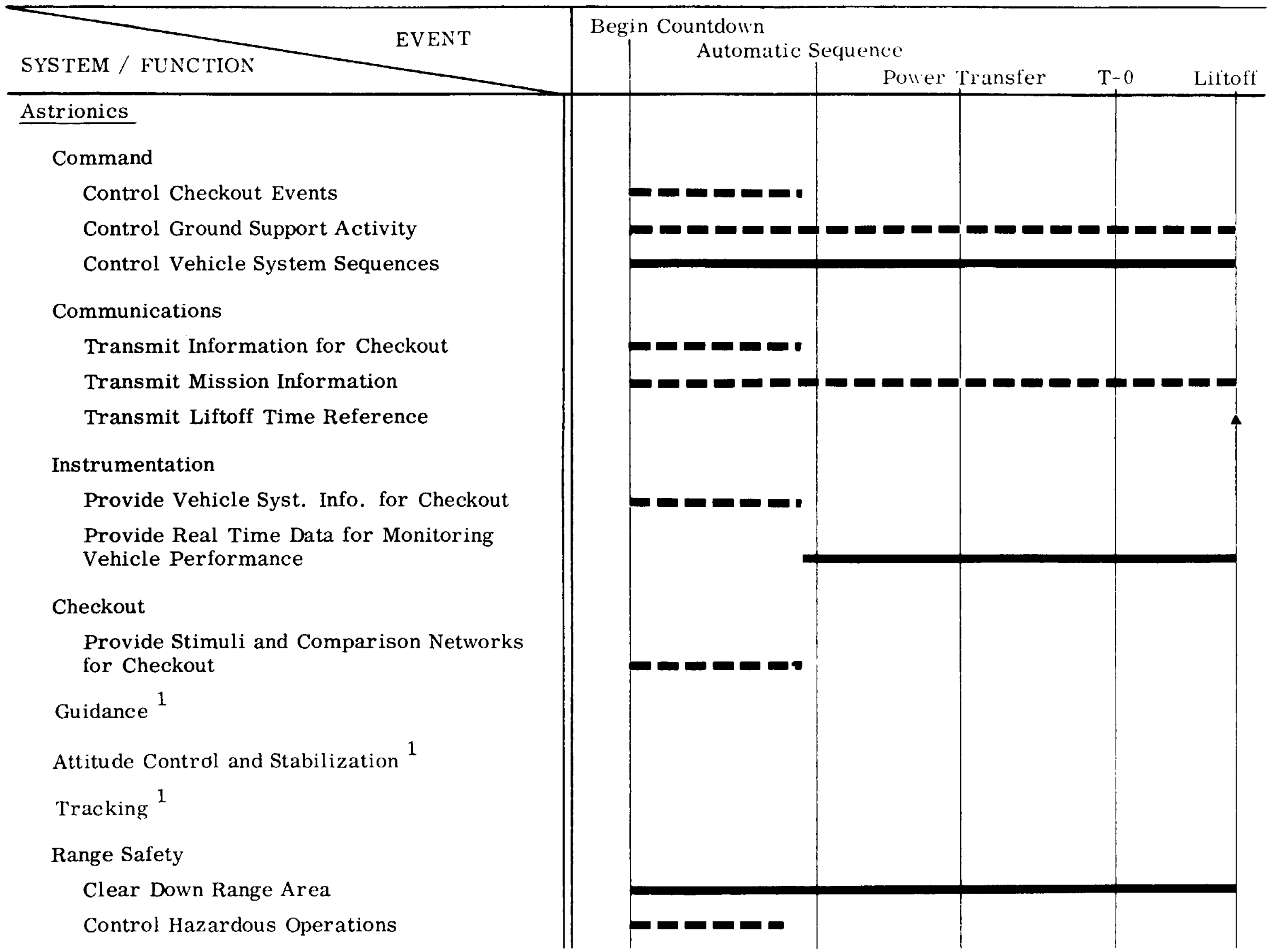


Table 19-5. Saturn V Requirements, Launch Phase (Cont'd)

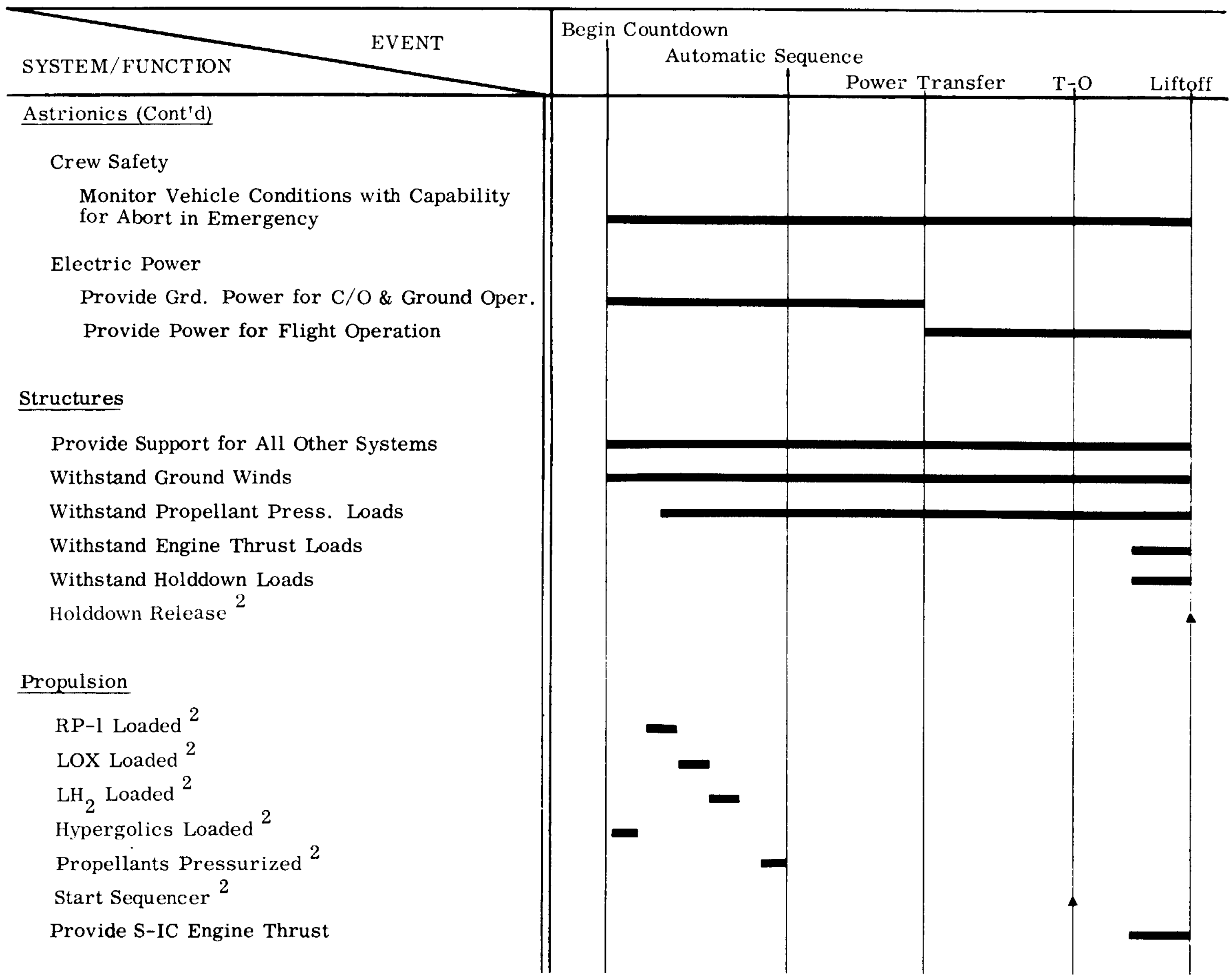
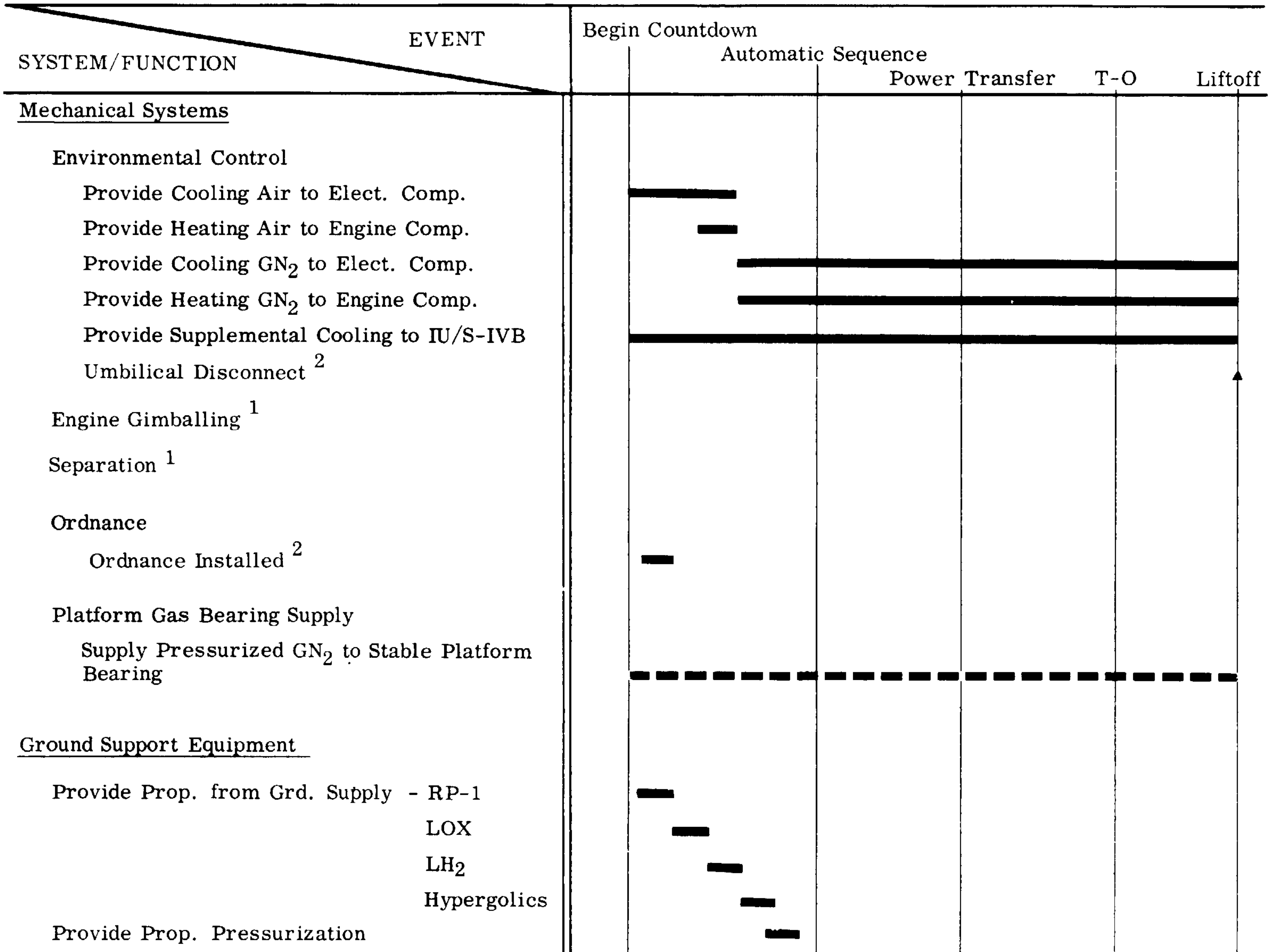


Table 19-5. Saturn V Requirements, Launch Phase (Cont'd)



Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 19-6. Saturn V Requirements, Ascent Phase

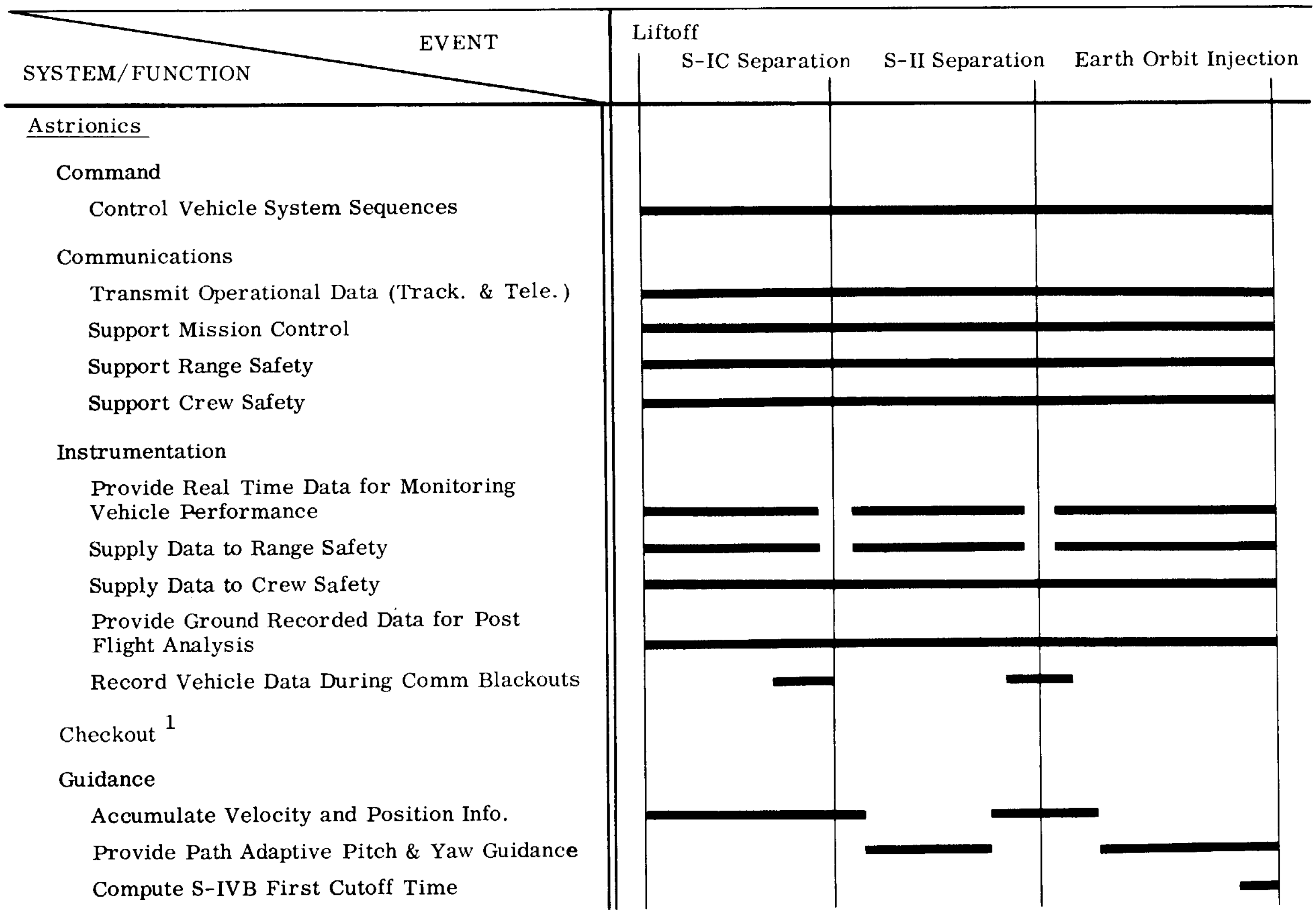


Table 19-6. Saturn V Requirements, Ascent Phase (Cont'd)

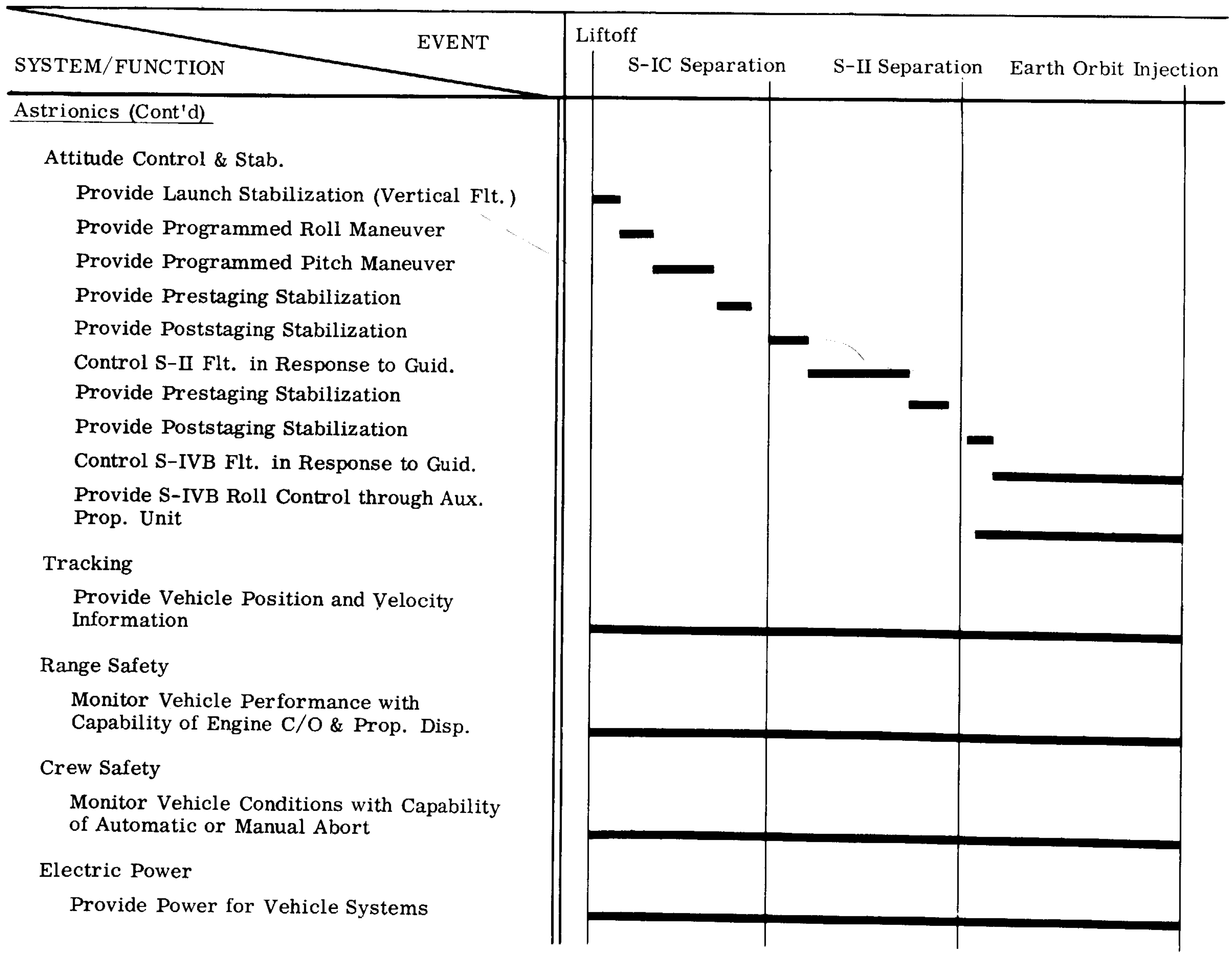


Table 19-6. Saturn V Requirements, Ascent Phase (Cont'd)

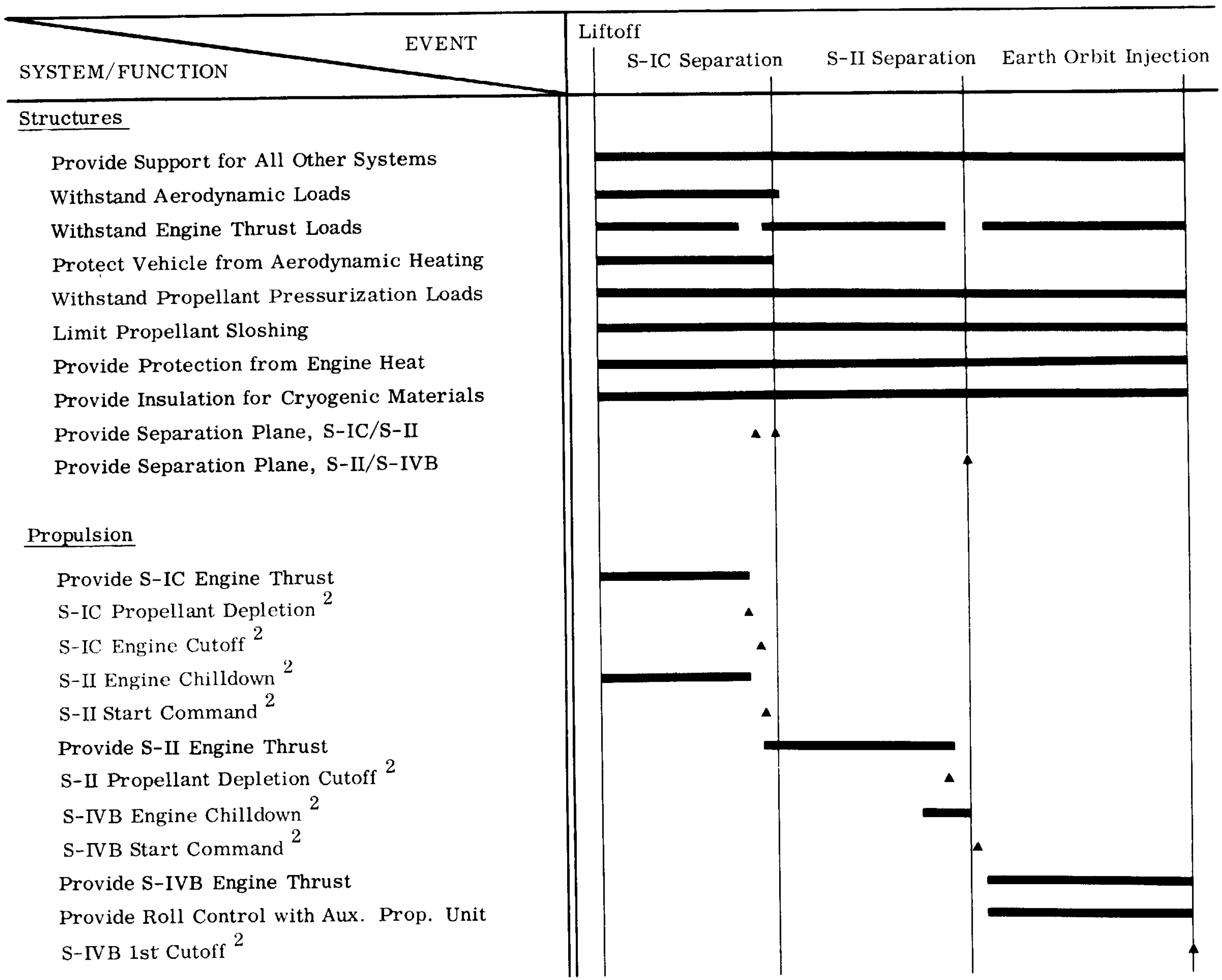


Table 19-6. Saturn V Requirements, Ascent Phase (Cont'd)

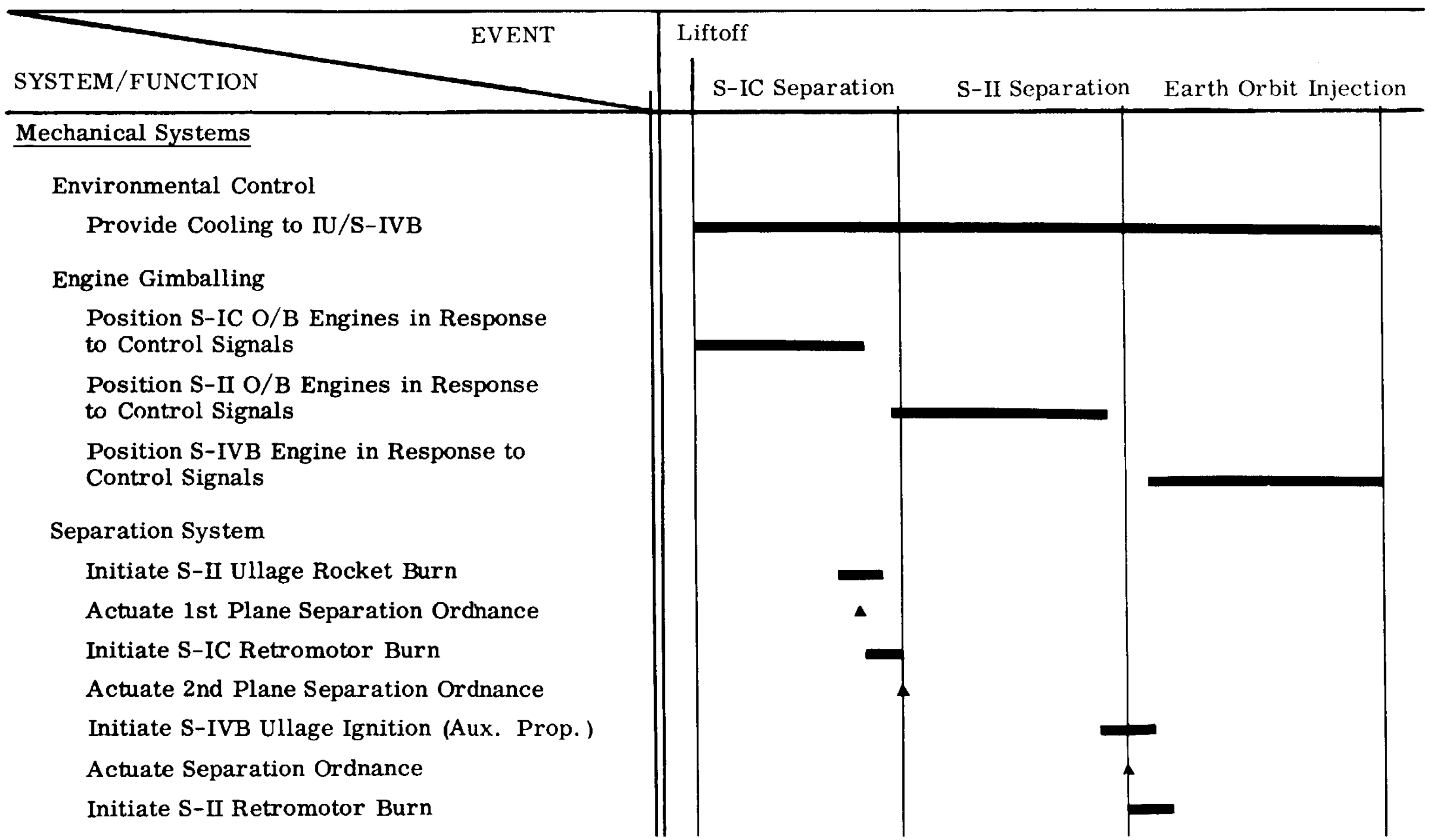


Table 19-6. Saturn V Requirements, Ascent Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT	Liftoff – S-IC Separation	S-IC Separation – S-II Separation	S-II Separation – Earth Orbit Injection
Mechanical Systems (Cont'd)			
Ordnance			
Blow Protective Cover on Horizon Sensor		▲	
Propellant Dispersion Capability Active			
Platform Gas Bearing Supply			
Supply Pressurized GN_2 to Stable Platform Bearing			
Ground Support Equipment [1]			

Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 19-7. Saturn V Requirements, Orbital Phase

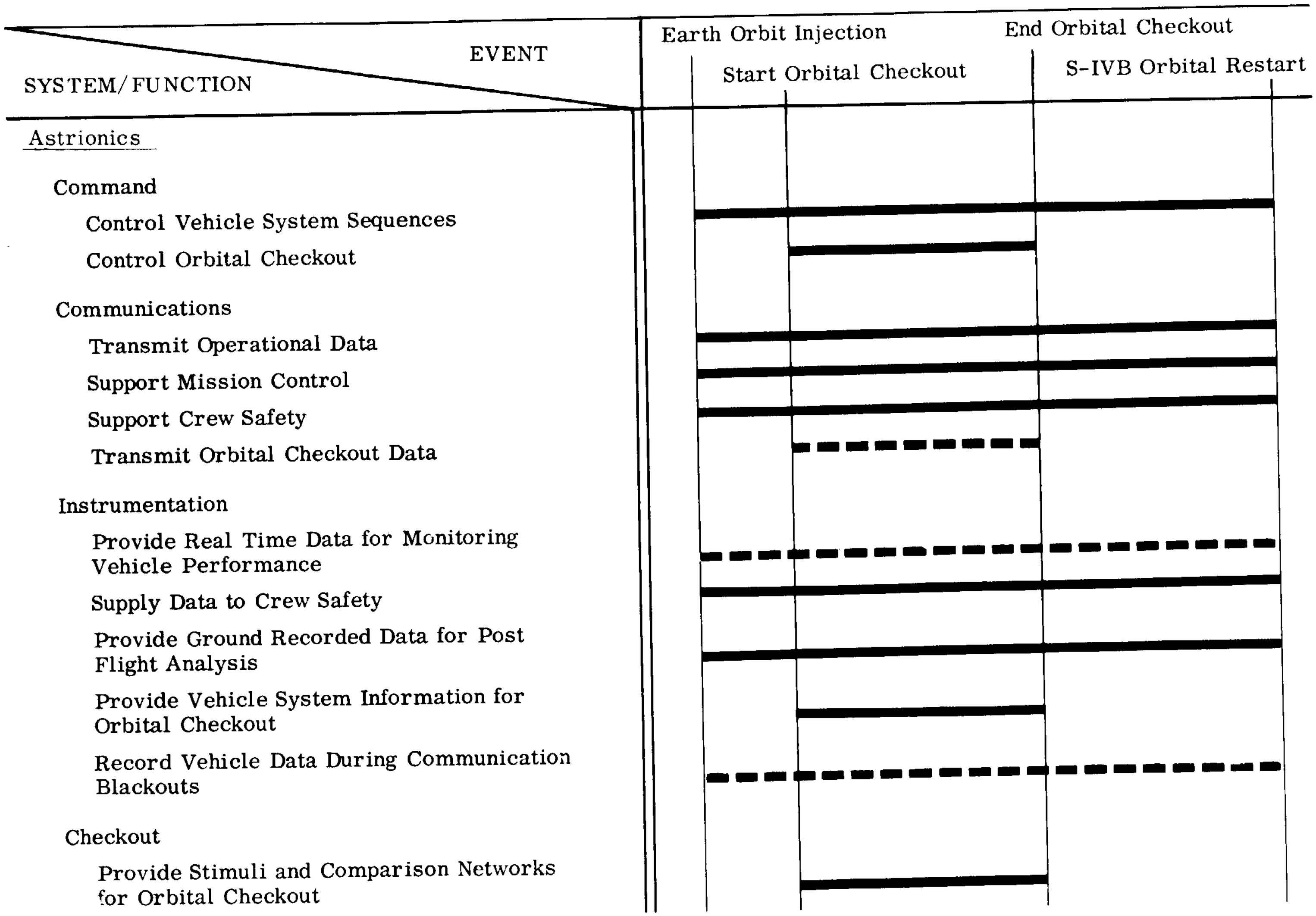


Table 19-7. Saturn V Requirements, Orbital Phase (Cont'd)

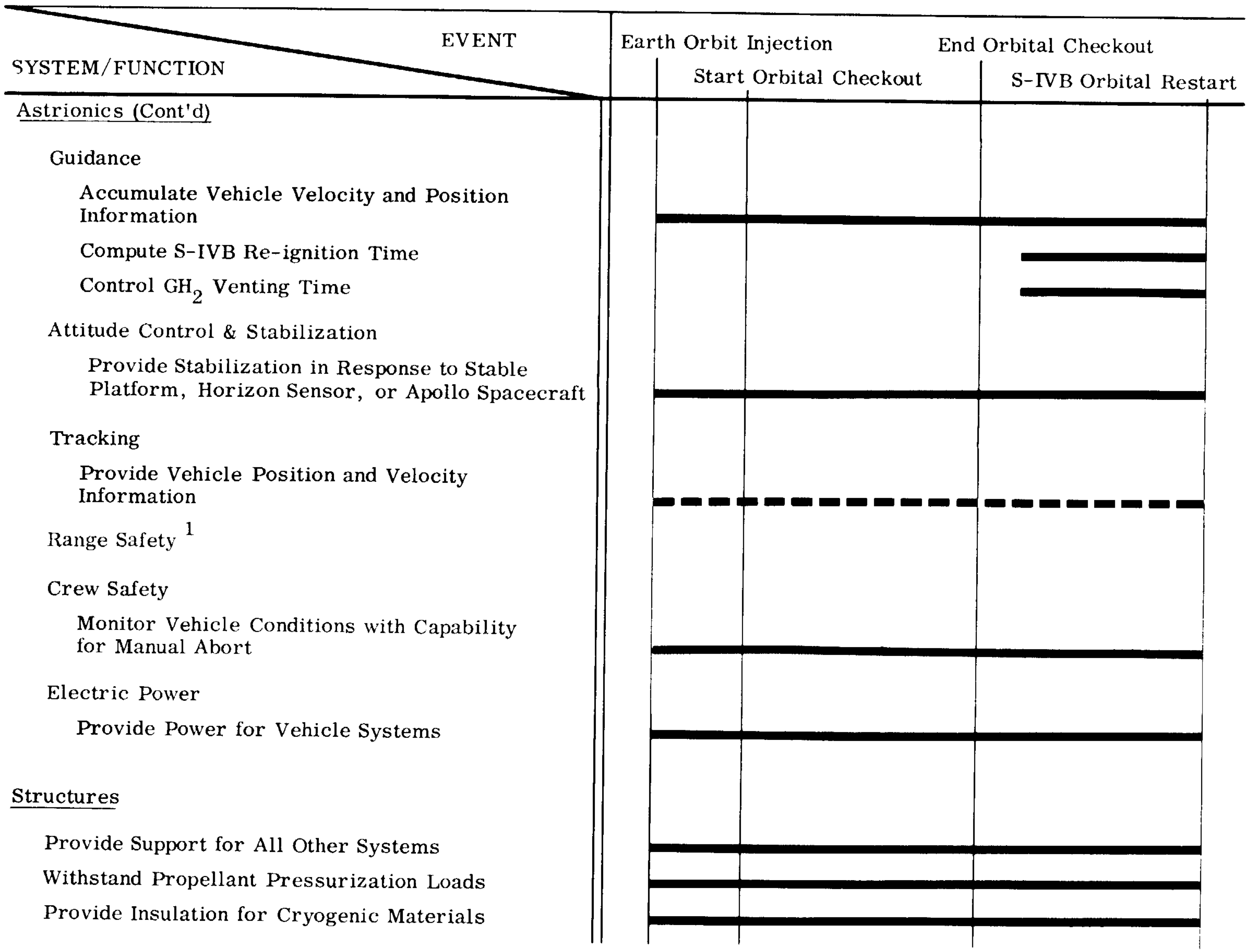
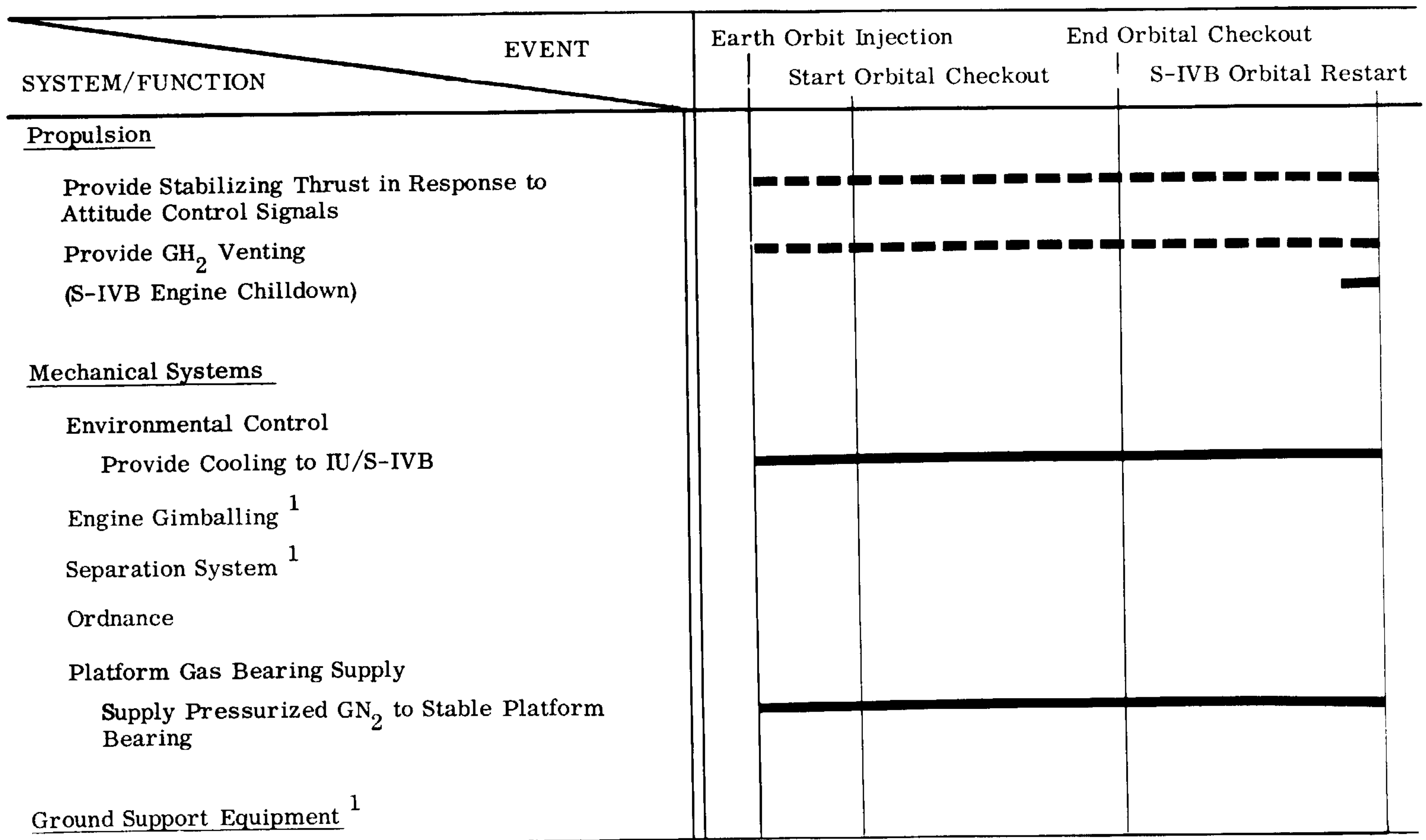


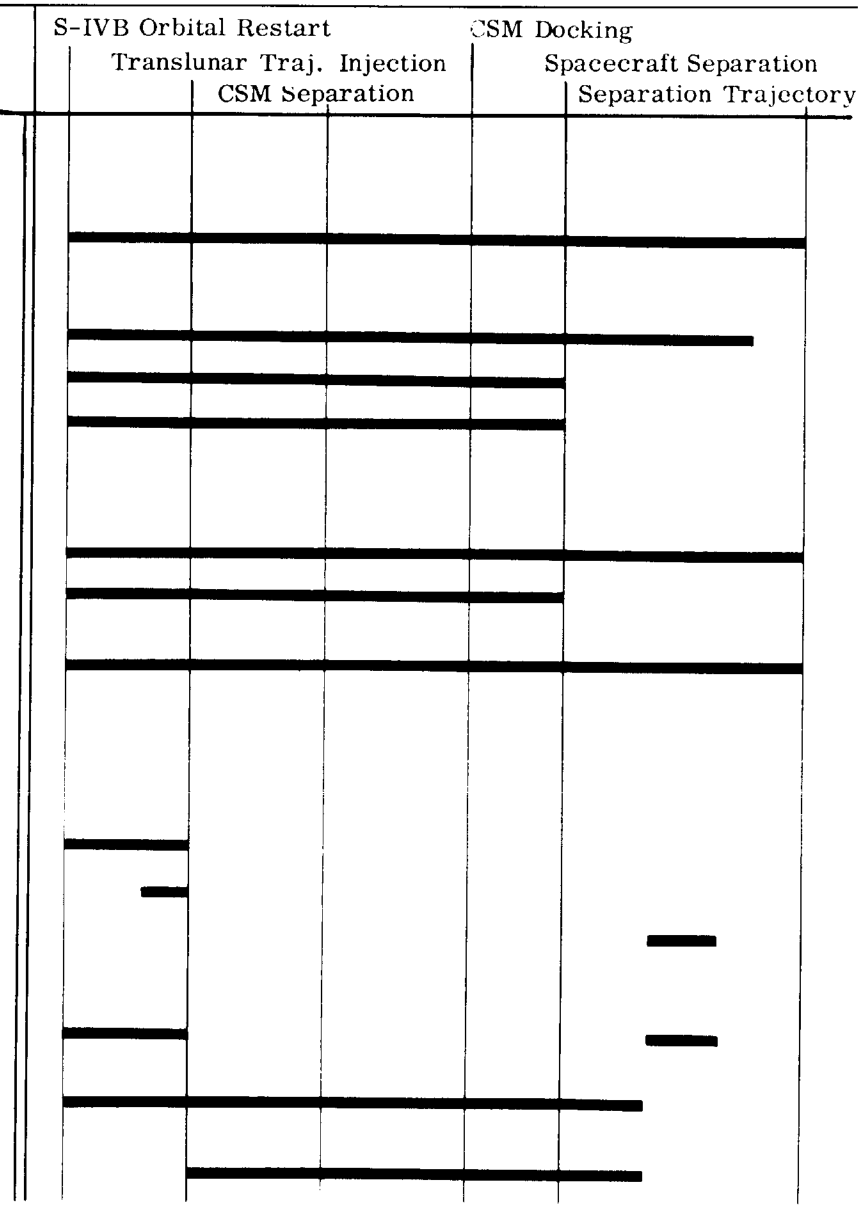
Table 19-7. Saturn V Requirements, Orbital Phase (Cont'd)



Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

Table 19-8. Saturn V Requirements, Translunar Phase

SYSTEM/FUNCTION — EVENT



Astrionics

- Command
  - Control Vehicle System Sequences
- Communications
  - Transmit Operational Data
  - Support Mission Control
  - Support Crew Safety
- Instrumentation
  - Provide Real Time Data for Monitoring Vehicle Performance
  - Supply Data to Crew Safety
  - Provide Ground Recorded Data for Post Flight Analysis.
- Checkout [1]
- Guidance
  - Provide Path Adaptive Pitch and Yaw Guidance
  - Compute S-IVB 2nd Cutoff Time
  - Provide Separation Guidance
- Attitude Control & Stabilization
  - Control S-IVB in Response to Guidance
  - Provide S-IVB Roll Control through Aux. Propulsion Unit
  - Provide Stabilization in Response to Stable Platform or Apollo Spacecraft

Table 19-8. Saturn V Requirements, Translunar Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT
<u>Astrionics (Cont'd)</u>
Tracking
Provide Vehicle Position and Velocity Information
Range Safety [1]
Crew Safety
Monitor Vehicle Conditions with Capability for Manual Abort
Electric Power
Provide Power for Vehicle Systems
<u>Structures</u>
Provide Support for All Other Systems
Withstand Propellant Pressurization Loads
Provide Insulation for Cryogenic Materials
Withstand Engine Thrust Loads
<u>Propulsion</u>
(S-IVB Start Command) [2]
Provide S-IVB Engine Thrust
Provide Roll Control with Aux. Prop.
(S-IVB 2nd Cutoff) [2]
Provide Stabilization Thrust in Response to Attitude Control Signals
Provide Thrust for Separation

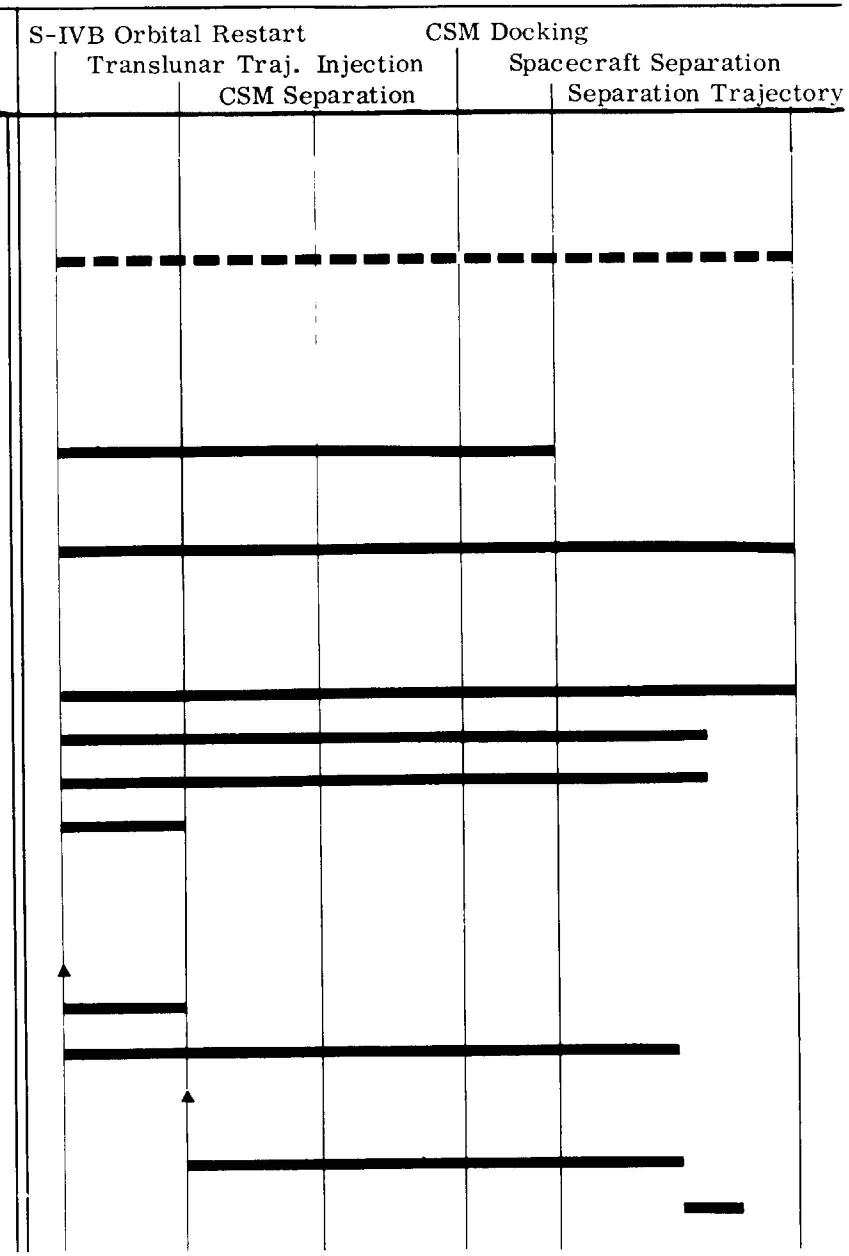


Table 19-8. Saturn V Requirements, Translunar Phase (Cont'd)

SYSTEM/FUNCTION \ EVENT	S-IVB Orbital Restart – Translunar Traj. Injection	Translunar Traj. Injection – CSM Separation	CSM Separation – CSM Docking	CSM Docking – Spacecraft Separation	Spacecraft Separation – Separation Trajectory
Mechanical Systems					
Environmental Control					
Provide Cooling to IU/S-IVB	▬	▬	▬	▬	▬
Engine Gimballing					
Position S-IVB Engine in Response to Attitude Control Signals	▬				
Separation					
Initiate Separation Ordnance S-IVB/IU from Spacecraft					▲
Ordnance [1]					
Platform Gas Bearing Supply					
Supply Pressurized GN_2 to Stable Platform Bearing	▬	▬	▬	▬	▬
Ground Support Equipment [1]					

Legend: [1]Inactive; [2]Key Event; ▲ Event; ▬ Operating; ▬ ▬ Intermittent Operation.

CHAPTER 4

SECTION XX

ASTRIONICS

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SECTION XX.
ASTRIONICS

20-1. GENERAL.

The Astrionics system provides the electrical and electronic functions required for Saturn V. The functions, listed below and described in the following paragraphs, are accomplished utilizing both vehicle and ground based subsystems.

a. Command - Performs management of Saturn systems by initiating all operational events and sequences. The issuance of commands is dependent on time and events.

b. Communication - Transfers intelligence within and among the Saturn systems. This intelligence is in four forms: voice, digital, discrete, and analog signals.

c. Instrumentation - Monitors the performance of launch vehicle systems to acquire operational and engineering appraisal data.

d. Checkout - Provides assurance during the prelaunch and launch phases that the launch vehicle is capable of performing its assigned mission.

e. Guidance - Provides steering, thrust cutoff and engine restart commands to adjust the vehicle motion in a manner leading to mission accomplishment.

f. Attitude Control and Stabilization - Provides signals to the engine gimballing system to maintain a stable launch vehicle motion and adjusts this motion in accordance with guidance commands.

g. Tracking - Obtains and records the launch vehicle position and velocity during flight.

h. Crew Safety - Ensures safety of the astronauts in the event of a malfunction in the Saturn/Apollo vehicle.

i. Range Safety - Ensures that life and private property are not endangered in the event of a vehicle malfunction during the ascent and orbital phase.

j. Electrical System - Supplies and distributes the electrical power required for vehicle operation.

20-2. COMMAND.

The Saturn V command function performs the operational management of astrionics, propulsion, structure, mechanical and ground operating support systems. These systems have an inherent requirement for a number of command stimuli of various priorities. The command function is accomplished with a chain of command to satisfy the priorities and to generate the many commands within a single priority. During the mission, the number of levels in the command function and the relative responsibility of each level varies to satisfy the command requirements peculiar to the individual mission phases.

The launch phase performances of the Apollo spacecraft, ground operating support system, and launch vehicle are coordinated to meet the mission launch time parameter. This operation includes launch vehicle checkout, alignment, and physical preparation such as the loading of fuel and cryogenics.

Due to the complexity of the ground operating support system and the launch vehicle, a volume of stimuli must be rapidly generated to accomplish launch phase performance in a reasonable time. The application of these stimuli causes system operation resulting in the generation of performance data which is assimilated and evaluated. If a systems malfunction occurs, decisions and commands are required to initiate corrective action. A manned critical decision and command capability is maintained for the launch phase. This capability exists should a situation develop whereby the astronaut's safety or the launch vehicle is jeopardized. An example of a critical situation might be the improper venting of a LOX container. In this case the critical command would initiate action to limit the progression of hazardous conditions. Final countdown events, including the switching of launch vehicle systems from the checkout and alignment modes of operation to the flight mode and system performance are monitored and evaluated. The vehicle flight then is initiated by a launch commit command which causes holddown release.

The launch vehicle operational commands for the ascent phase are supplied by an internal source. This source supplies the command stimuli to control the vehicle and stage events such as engine cutoff, separation of an expended stage and actuation of the succeeding stage.

A range safety command is available should the vehicle deviate from the planned flight pattern. This command capability can cutoff the launch vehicle engines and may initiate propellant dispersion if the vehicle becomes a hazard to private life and property.

The astronauts can command launch vehicle engine cutoff to permit their escape from the proximity of the vehicle in the event of a malfunction necessitating mission abort.

During the orbital phase the command function provides stimuli to checkout and evaluate the operation of the S-IVB/instrument unit (S-IVB/IU) prior to translunar injection. This provides assurance that the launch vehicle can accomplish its objective of placing the Apollo spacecraft in a translunar trajectory with the correct velocity.

The mission orbital phase is a significant evaluation period. During this time the parameters of the orbit are confirmed by ground system. The Apollo guidance system is aligned using celestial references, and the orbital parameters as determined by the Apollo system are compared with those determined by ground means. The Apollo guidance system is compared with the Saturn guidance system to give assurance that guidance operation is proper. Should these evaluations indicate an out-of-tolerance condition then corrective measures are taken and the mission continued or an alternate mission selected. The mission is aborted and the astronauts are returned to earth, if corrective measures cannot alleviate the problem or an alternate mission cannot be accomplished.

To provide maximum flexibility and reliability the S-IVB/IU events and sequences can be initiated from internal command or from the ground. This capability permits the selection of a system mode of operation to fit the particular orbital situation and provides a certain amount of redundancy in the source of system stimuli.

Prior to translunar injection the guidance system of the Saturn vehicle can be aligned utilizing ground command.

Alignment of the Saturn guidance and control reference during the orbital phase permits a more accurate injection into the translunar orbit.

Stored guidance constants for the Saturn system are updated to permit S-IVB re-ignition at the most opportune time for the mission. The updating of the guidance constants results in an optimum translunar trajectory considering the orbital parameters and other state conditions.

During the translunar phase an internal source provides command stimuli to initiate the events and sequences within the S-IVB/IU stage. This command is active in the mission until the final jettison of the S-IVB/IU stage.

20-3. OPERATION.

Saturn V launch phase command is accomplished in five levels, Integrated Mission Control Center (IMCC), Launch Control Center (LCC) manned, Launch Control Center computer, Launch Umbilical Tower (LUT) and vehicle levels. The IMCC maintains overall mission responsibilities and coordinates the operation of the ground operating support systems, payload and launch vehicle. This level imposes a ready-to-launch time requirement on the launch vehicle and the payload. If for any reason during the launch phase a hold is required, then the IMCC imposes a new time to launch requirement on the various portions of the Apollo system. The decisions made by IMCC have a mission level priority.

The LCC manned level of command assures that the launch phase performance of the launch complex, payload, and launch vehicle meets the time requirements imposed by IMCC. This performance includes the physical preparation of the payload and launch vehicle and the checkout and alignment of both the payload and launch vehicle. The LCC manned level has the highest level of responsibility in the launch area. This level controls the LCC computer and various launch complex subordinate levels of command. The LCC manned level is responsible for critical decisions. A critical decision is one that involves the astronaut's safety and the integrity of launch vehicle operation. Data monitored by the LCC manned level has been previously filtered so that only the highest priority data is presented. This filtering of data prior to presentation to the manned level permits secure control of operation and does not impose an overwhelming monitoring on this level.

The LCC computer level of command is the first or highest level of automated data monitoring and decision selection. This level performs the management for lower levels of automated command. The LCC computer level scans previously filtered data and selects non-critical decisions for the lower levels, and then filters data scanned and displays the critical data for LCC manned decision. The LCC computer level is the first level of command capable of generating a volume of stimuli in a limited time. These stimili select the mode of operation for lower levels of command and can-excite some vehicle systems. The LCC computer level realm of responsibility includes the launch complex, LUT and launch vehicle.

The LUT command level is an automated level with direct control of the launch vehicle. The LUT command level monitors data requiring a fast scan rate, selects decisions and applies stimili directly to the vehicle. The LUT command level filters data and presents high priority data to the LCC computer level.

Systems operation within the mode selected by higher command levels is performed by interaction between the LUT command level and the launch vehicle. These operations include checkout and alignment, and the switching of vehicle systems from the launch modes of operation to the flight modes. The forms of commands issued by the LUT command level are discrete (on off) commands, digital encoded commands and analog stimuli.

The vehicle level of command controls the mode of vehicle systems operations and issues stimuli in the correct sequences to accomplish systems operation.

The terms, mode and sequence, are defined at this time to explain their relationship with the system. The digital computer memory contains a predetermined number of sets of instructions and, when initiated, induces the whole or portions of the system to operate in a particular manner. The instructions represent a predetermined sequence of operations which occur at any time the computer is directed to work with that particular set of instructions. The term "mode selection" means the selecting or commanding of a particular set of instructions in the computer which then defines a certain type of system operation. An example of a mode of operation is the solving of guidance equations with interlaced attitude control and sequencing commands during first stage launch. Here the system continuously solves equations based on transducer inputs from vehicle systems and computes the guidance angles which are used ultimately to control the engine thrust vector. It also computes thrust cutoff where necessary and initiates other discrete operations. A predetermined sequence of events is initiated when the particular mode is chosen, and it continues until completed or until another mode is selected.

Mode selection and initiation is accomplished through one of several sources. A new mode results from one of three actions: (1) the successful completion of a previous mode or (2) computer switching to a new mode based on real time, or (3) an event occurrence. In addition to the normal internal mode selections which the system makes, mode switching is accomplished by commands from the LCC computer (prelaunch), the instrument unit command system, or the Apollo spacecraft. There is a built-in safeguard feature that gives the system the capability of refusing conflicting commands or commands that would be detrimental to vehicle safety.

The hardware interfaces which implement the mode and sequence control are described in the following paragraphs. The LCC computer, as ground checkout equipment, is required to completely check out the Astrionics system by exercising all modes of the system. This includes simulated launch and orbit programs as well as functional operation of all system parameters to ensure satisfactory operation prior to launch.

The LCC computer commands a particular mode of system operation by sending a coded command to the data adapter, which reads it into the digital computer. In here, the mode command is decoded, and the set of instructions or program is selected in the computer memory, which is defined by the decoded mode command. The digital computer then begins accomplishing, either internally, or initiating action elsewhere in the system, the instructions in that mode program that are required to integrate the system operation with that of the LCC computer.

A stage-located switch selector allows the digital computer to control 111 different events in each stage, this being accomplished through an 8-bit coded command to the stage currently under control. Before a "read" command is given to execute the coded command, an "echo-check" (return) signal verifies that the correct switch selector has received the command. The functions in each stage are controlled according to real time or as a result of the solution of equations for a given set of conditions from the data adapter and digital computer in combination. The functions to be performed include engine cutoff, telemetry calibration, stage separation, and any additional functions in the selected program. An additional capability is provided by a special mode wherein all stage functions controlled through the switch selector are commanded directly from the LCC computer. The LCC computer modes discussed here are applicable to the instrument unit command system operations after launch. The remote automatic calibration system (RACS), used to check out telemetry transducer inputs, is controlled from the ground support equipment prior to launch. Each RACS telemetry input channel can be calibrated upon command.

It is essential that the astronauts have a method of controlling the mode of operation from the spacecraft as it is necessary for them to be able to select system modes so that spacecraft control can be exercised over the S-IVB/IU during certain phases of the mission. To do this, the spacecraft issues a mode command to the data adapter which, along with the digital computer, processes and decodes the command. The digital computer then performs the necessary sequencing to allow guidance signals from the spacecraft to feed directly into the control computer. The digital computer has the capability of making the necessary operational changes so that the Astrionics system may follow the vehicle attitude, as commanded by the spacecraft, and be ready to assume control of the S-IVB/IU when the spacecraft has completed the special tasks. Control is returned to the Astrionics system by issuance of mode command from the spacecraft.

20-4. IMPLEMENTATION.

The launch phase command function is implemented in the launch complex with the RCA-110 computer and manned consoles.

The vehicle implementation, common for all flight phases, consists of the data adapter, vehicle computer and the switch selectors. The data adapter and vehicle computer are described in Paragraphs 20-45 and 20-64 , respectively.

20-5. RCA-110 COMPUTER.

The RCA-110 Computer is a general-purpose digital computer capable of automatic monitoring and control. This computer is comprised of five major sections. The major sections of the computer are control, input, output, storage and arithmetic. The computer data are presented in Table 20-1.

20-6. Input/Output Section. Information is transferred into, and out of, the computer by means of input/output devices. Data to be processed, or programs to be performed, are "read" into the machine by paper tape, or by magnetic-tape readers, or by other peripheral equipment. Information is returned from the computer by a paper-tape punch, a magnetic-tape recorder, typewriter, or other type of visual display. Within the computer are several registers that sense, select, and control the information to and from the input/output equipment.

20-7. Control Section. The control section is the command unit. It governs all operations in the machine such as information transfers, arithmetic performance, and the sequence of instructions. The control section may be a complete unit consisting of several registers, such as the program computer, the instruction register, and the timer.

20-8. Arithmetic Section. This section of a computer performs mathematical operations: addition, subtraction, multiplication, and division. It also performs "logical" operations. The arithmetic section will contain such units as the left and right accumulators, the adder, and the counter.

20-9. Storage Section. The storage, or memory unit is used to store information (in machine language) until it is required for use during program execution. The term, memory, is usually referred to as the storage within the computer. Information is retained in units such as a coincident core or a magnetic drum. Storage outside the computer is generally on paper or magnetic tape.

Table 20-1. RCA-110 Computer Data

Item	Data
General.	
Type of logic	Serial
Internal clock rate	936 kc
Word size	24 bits
Arithmetic	Fixed point
Instructions	Single address (72)
Index registers	7 (stored in memory)
Accumulators	Left and right
Priority interrupt	4 levels (2 programs per level)
Basic Timing.	
Word time	28.85 usec

Table 20-1. RCA-110 Computer Data (Cont'd)

Item	Data
Add/subtract	57.7 usec
Multiply	799 usec
Divide	865 usec
<u>Data and Instruction Storage.</u>	
High-speed coincidence-current core memory (HSM)	
Memory access time	3.5 and 10.25 usec
Number of words (storage)	512 to 4096
Word size	24 bits, plus parity bit
Bulk storage - magnetic drum	
Drum speed	3600 rps
Access time	8.2 msec (avg.)
	17.0 msec (max.)
Word size	24 bits, plus parity bit
Main storage capacity	4096 to 32,768 words
Number of tracks	32 to 256 (128 words each)
Buffer tracks	up to 16
<u>Input/Output Capabilities.</u>	
Magnetic-tape stations	1 to 10 stations (15,000 characters/sec.)
Paper-tape reader	60 characters/sec.
Paper-tape punch	60 characters/sec.
Monitor typewriter	10 characters/sec.
Input/output buffer registers	1 to 8
I/O sense lines	24 lines/set (1 to 8 sets)
I/O address lines	24 lines/set (1 to 8 sets)

20-10. SWITCH SELECTOR.

The Saturn V system utilizes the digital computer in the instrument unit for control of mode and sequence of functions in all stages. The switch selector provides the communications link between the computer-data adapter and the control distributor in the instrument unit and each stage.

Redundancy is used to increase reliability within the equipment. The reset, stage select, and read command relays are redundant, offering improved reliability in relay coil operation and its associated contacts. The register is protected from failure by the fact that either the code or its complement will operate a specific driver.

The switch selector is an individual stage device and has control of the computer on a particular stage. There are five switch selectors in Saturn V, one in each of the launch vehicle stages, one in the instrument unit, and one spare. All lines to the switch selectors except the stage select lines are paralleled to all stages; thus, the five devices require 32 input lines from the data adapter and one from 28-volt dc instrument unit power (refer to Figure 20-1). A list later in this section indicates individual line usage and will substantiate the required lines when stage select is multiplied by the number of switch selectors used.

The switch selector is divided into two sections; the input or register section, composed of latch relays, which are powered from the data adapter; the output relay drivers, which are powered from stage supplies and maintain stage isolation. The input and output are coupled together through a diode matrix which decodes the 8-bit input code and furnishes an output from one of the relay driver outputs. The output of the switch selector is composed of 114 possible relay drivers but, since zeros and ones are used for test purposes there are 112 possible functional outputs. The zero indication line, (0000 0000) consisting of eight zeros, is carried to the ESE through the umbilical so that it may be interlocked with firing command. The eight one's line (1111 1111) is not used for a timed output but as a register test.

The input code of the switch selector is positive logic; the "one's" are 28 volts dc and the "zero's" are 0 volts dc or open. The outputs are also positive logic, giving a positive output voltage pulse upon read command. This output pulse is a square wave, duration not less than 25 milliseconds, and the voltage not less than two volts below the stage input voltage. Loading current must not exceed 100 ma at 26 volts dc.

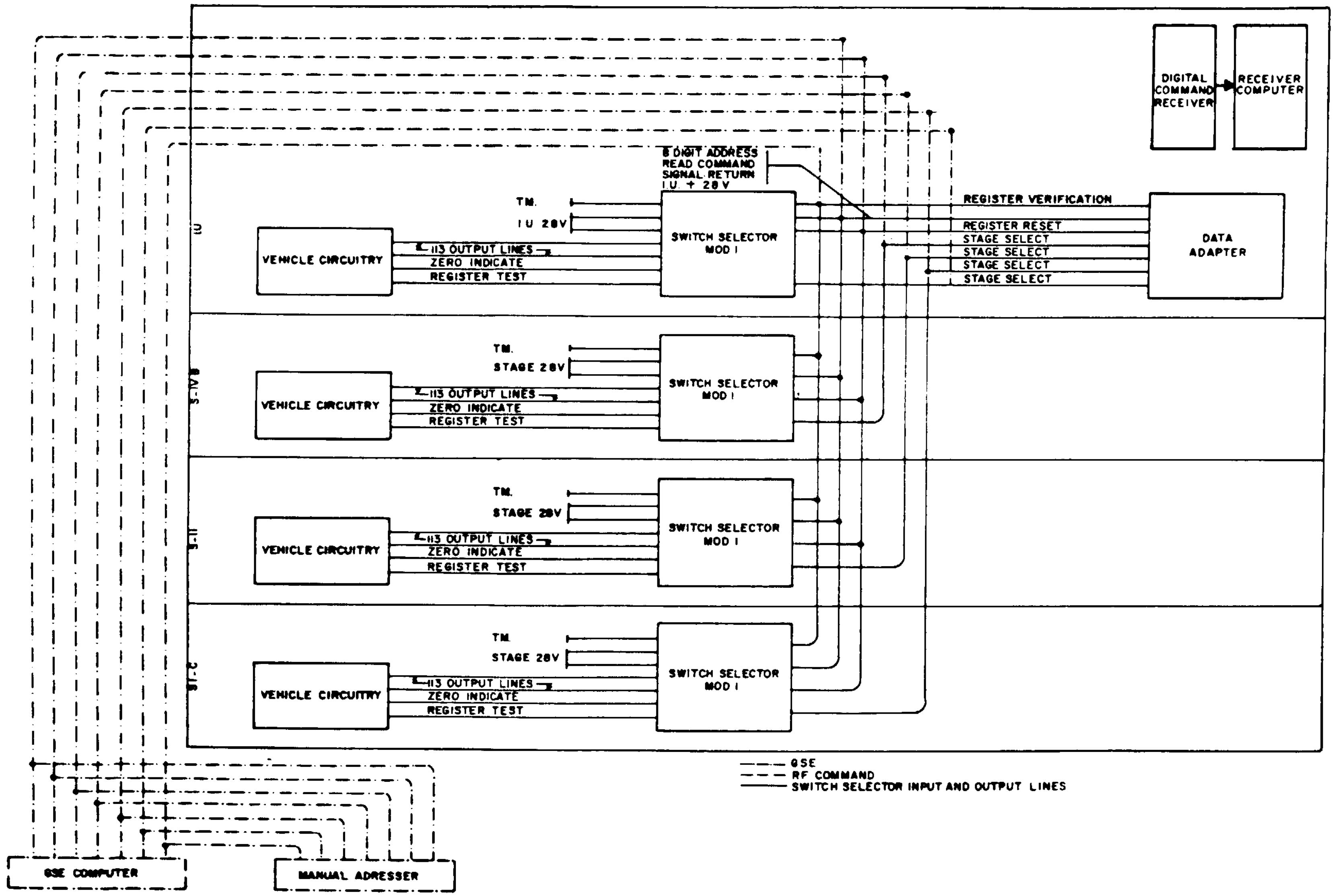

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Figure 20-1. Switch Selectors, Block Diagram, Saturn V

The requirements and characteristics of the switch selector are:

a. The bit-coded signal address to the switch selector and the stage select bit is 28-volt dc and should be not less than 20 milliseconds duration. The signal address lines should be back to 0 volts dc before the read command is given.

b. The read command pulse to the switch selector should not be less than 28 milliseconds, or greater than 50 milliseconds.

c. The minimum time between sequential outputs from the switch selector is 112 milliseconds. The output pulse from the switch selector is a minimum of 25 milliseconds, and a maximum of 47 milliseconds, depending on the length of the read command and read command relay drop time.

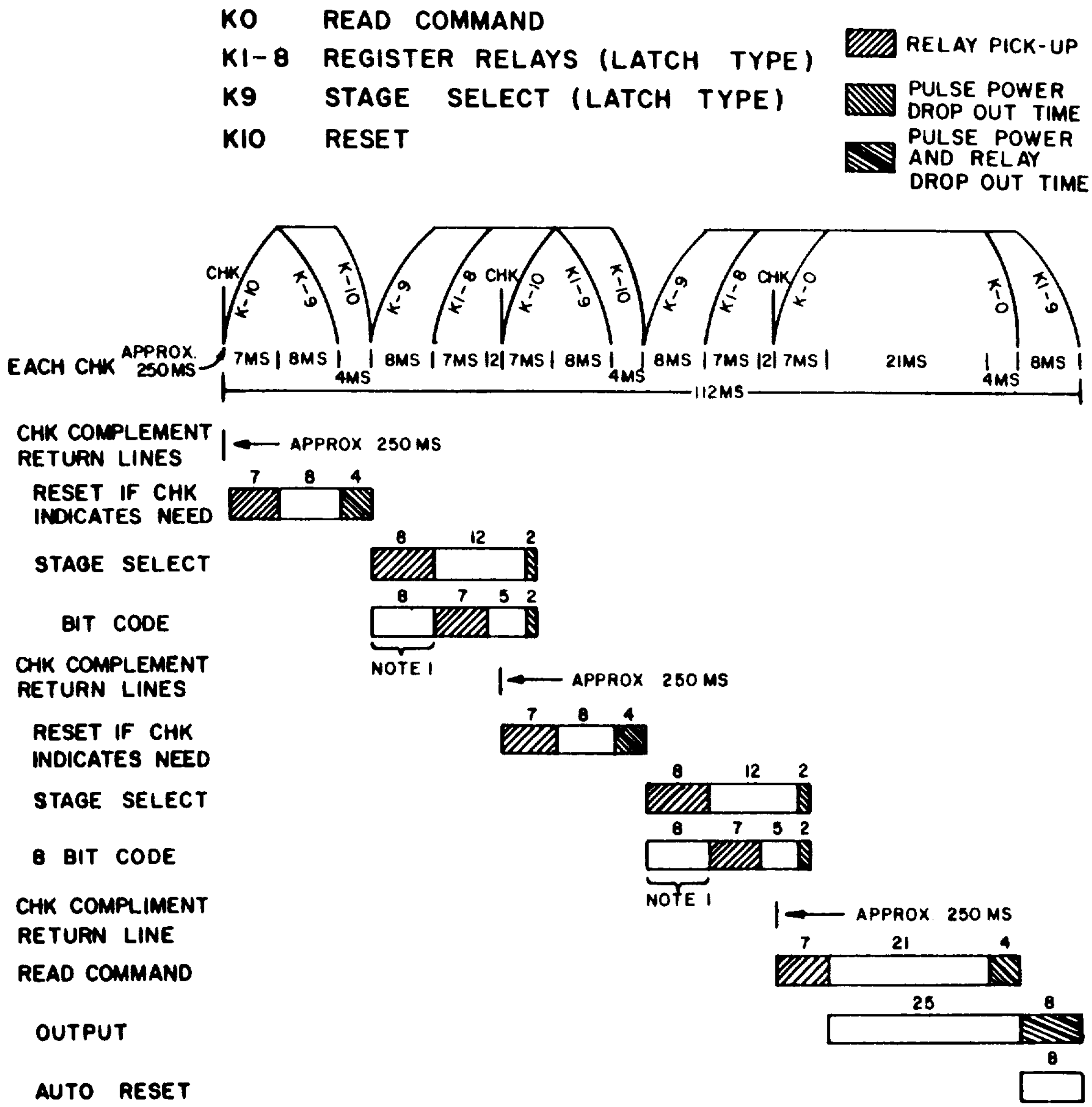
There are 24 lines between the switch selector and the data adapter. The complement code comes from the instrument unit 28-volt dc power. These lines serve the following functions:

a. 8 Signal code digits
b. 8 Complement of signal code digits
c. 2 Signal return (one redundant)
d. 2 Read command (one redundant)
e. 2 Stage select (one redundant)
f. 2 Reset register (one redundant)
g. 1 28-volt dc line

All but stage select are parallel lines to all switch selectors from the data adapter. Therefore, an 8-bit code for a particular output set by the data adapter appears at each switch selector. The stage select is a specific line to a specific switch selector and its presence is necessary to operate a particular register. Prior to operating any switch selector, a check is made of the complement code return lines and the absence of 28-volt dc on all of the lines indicates that all stage select relays were properly reset on the previous switch selector operation. The computer addresses the switch selector from which an output is desired with the stage select line. The 8-bit code is then set into the proper register only. The eight complement lines return to the computer via the data adapter and the transmitted code is checked. In the event of an error detection, the computer pulses the reset line, resetting all registers to all zeros, and then transmits the complement code. Using either the code or its complement to operate the same relay driver gives the switch selector the ability to work around an inoperative relay in the register. With the complement check passed, the computer gives the "clock" signal or read command to all selectors at the desired time. This read command allows the switch selector, or selectors that have been given a stage select, to drive the addressed output. Addresses in switch selector registers are automatically reset to zero after the read command. The register may also be reset over the reset line without giving an output. (Refer to timing chart Figure 20-2.) (Figure 20-2 is intended to show the sequence of events in terms of relay pick up and drop out times and pulse lengths involved. The total process involves three checks, and it is possible through positive check results to shorten the total time considerably.)

Register reset is normally used as a manual interrupt when reset is required, and an undesirable command in the switch selector register must not be executed. Normal reset is automatic with the read command but this action forces execution of the command. Register reset is also used to unlatch a register relay that remains latched after automatic reset.

Prior to the operation of any switch selector, a check is made of the complement return lines, and if 28-volt dc (logic 1) is not found, it can be assumed that no stage select relay remained latched because it would have switched the 28 volts dc to the complement code lines. If this check proves that a stage select K9 remains latched after auto reset, manual reset K10 is pulsed in an effort to unlatch the K9. The proper stage select line is then pulsed. Stage select K9 provides a signal return for the register and closes a set of contacts necessary to turn on the proper relay driver and one necessary to furnish 28-volts dc to the complement code lines. The code lines K1-K8 are then pulsed to set the proper code in the relay register. Another set of K1-K8 contacts returns the complement of the code to the computer which checks for proper relay action. If an incorrect complement is received, the computer pulses the reset line and sends the complement of the original code to the switch selector. This is again verified but the results are used only for telemetry recording. The code now established in the register works through a diode matrix to bias off (with 28-volt dc) the base of all relay drivers except the one coded for an output. The relay driver chosen for an output has the normal off bias on it, but when the read command is given, K0 is energized and the base is brought near ground thru a 7.5K resistor to turn on the relay driver. K0, the read command relay, also applies stage voltage to condenser "C." Then K0 is de-energized at the end of the read command pulse and the energy in the condenser



NOTE I. CODE LINES ARE ON BUT REGISTER CAN NOT REACT UNTIL STAGE SELECT IS UP THUS PROVIDING A SIGNAL RETURN PATH.

3-330

Figure 20-2. Switch Selector Sequence and Timing Chart, Saturn V is applied to the reset coils of the relay register, resetting the relay and giving an output on the zero indicate, 0000 0000.

A telemetry output called a confidence line is provided which will indicate if no output, one output, or more than one output occurs from the switch selector. A properly operating selector produces only one output in response to each read command.

The instrument unit telemetry monitors the total action of the switch selectors. Present planning calls for the following to be monitored:

a. Code and complement code
b. Stage select
c. Reset
d. Read command
e. All zeros
f. Confidence lines

20-11. COMMUNICATIONS.

The communications function actively supports the command, tracking, instrumentation, range safety and crew safety functions through transfer of information between points participating in the mission.

Requirements for a communications function to support the Saturn V missions are basically the same as for the Saturn I. (Refer to Paragraph 6-5.) An additional requirement, peculiar to the Saturn V/Apollo mission, is the need to extend the earth-vehicle communications link to deep space.

20-12. OPERATION.

To coordinate the various operations involved in the Saturn V mission, earth-based command levels and other support functions are interconnected by a network of wired and radio frequency (RF) links, which include channels of voice, teletype and telemetry data. The space vehicle is integrated with this communications network through links which connect vehicle-borne systems with command transmitter sites, tracking stations and telemetry reception stations on earth. The communications network also provides voice communications between the spacecraft crew and ground support personnel. This network relays control information and operational orders between the space vehicle and earth-based installations during all phases of the mission.

Tracking and instrumentation function support is also provided by communications. It transmits tracking and telemetry data from the receiving stations to data recording and evaluation centers for real-time computation and evaluation and predicted trajectory information from Goddard Space Flight Center to the tracking stations for vehicle acquisition.

Communications provides the range safety function with tracking and telemetry data. The communications function also provides for transmission of range safety commands from the range safety officer to command transmitters and to the vehicle.

The crew safety function is also supported through communications. Tracking and telemetry data, delivered through the communications network, are monitored by ground operational personnel. Their evaluation of vehicle conditions and crew safety is relayed to the spacecraft crew by voice transmission.

20-13. IMPLEMENTATION.

The communications function is implemented with vehicle and earth communication links.

20-14. Earth -to-Vehicle Communications. Communications between earth and the Saturn V launch vehicle include the radio frequency systems used in tracking, instrumentation and range safety functions. These systems are included in the sections describing those functions. In addition, radio frequency voice links are provided between earth and the spacecraft, and a guidance command system on board the Saturn V instrument unit links earth-based mission control and the vehicle control systems. Each of the Deep Space Instrumentation Facility stations (Goldstone, California; Johannesburg, South Africa; Woomera, Australia) have voice links with the spacecraft as do those stations listed for "capsule communications" in Table 6-1.

The vehicle guidance command system consists of an MCR-503 receiver and a digital decoder. Digitally-encoded commands transmitted from command transmitters on earth are received and translated into signals which control on-board events or provide inputs to the vehicle guidance computer for trajectory correction.

20-15. Point-to-Point Communications (Earth). Stations interconnected to form the Saturn V communications network include those listed in Table 6-10, which were derived from the Mercury Network and Atlantic Missile Range facilities; the Minitrack network, shown on Figure 20-3; and the Deep Space Instrumentation Facilities, under operational control of Jet Propulsion Laboratory's Space Flight Operations Facility at Pasadena, California. (Stations at Goldstone, California; Johannesburg, South Africa; and Woomera, Australia).

The generalized communications network for Saturn V-Apollo missions is illustrated in Figure 20-3. Aircraft may participate in the communications links to relay data from ships to land-based stations in the network. The Minitrack network can also be used as a communications backup.

Addition of the Deep Space Instrumentation Facility (DSIF) stations is the major difference between Saturn V and Saturn I communications. The projected DSIF communications network is shown in Figure 20-4. Note that this network interfaces with other sub-networks of Saturn V-Apollo through Goddard Space Flight Center and the Integrated Mission Control Center.

20-16. INSTRUMENTATION.

Saturn V instrumentation collects status and operational data from the launch vehicle for use by the other functions. The system to accomplish this is composed of a measuring system to gather the data on the physical quantities and signals onboard the vehicle, and a telemetry system to transmit the data to ground stations. Optical systems used to provide performance data are included in this description. Instrumentation data is required to supply information for the following:

a. Automatic preflight checkout of the vehicle.
b. Monitoring of vehicle performance during powered flight.
c. Monitoring and checkout of the vehicle during orbital flight.
d. Verification of commands received in the vehicle from ground stations.
e. Preflight and Inflight telemetry calibrations.

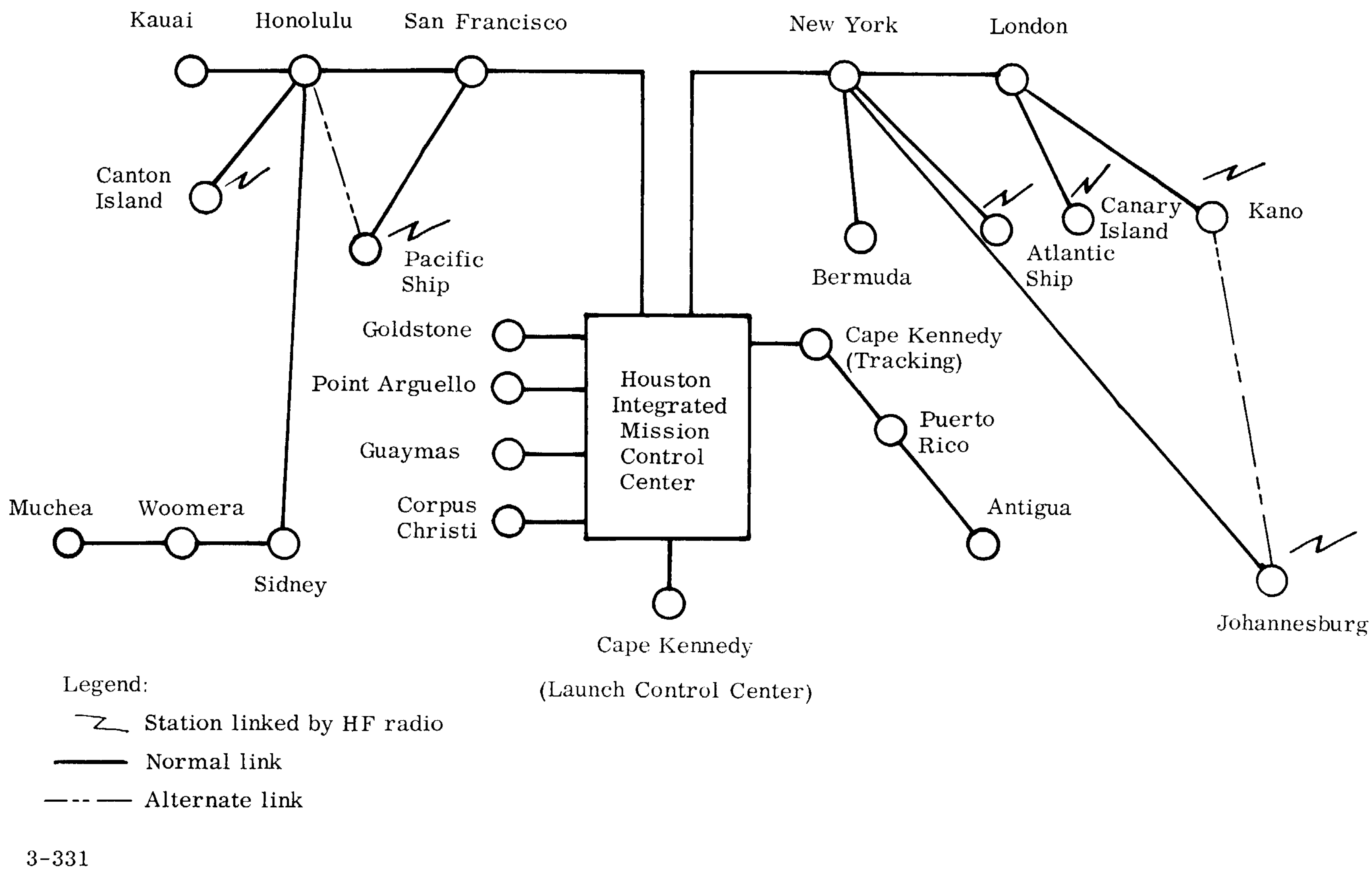
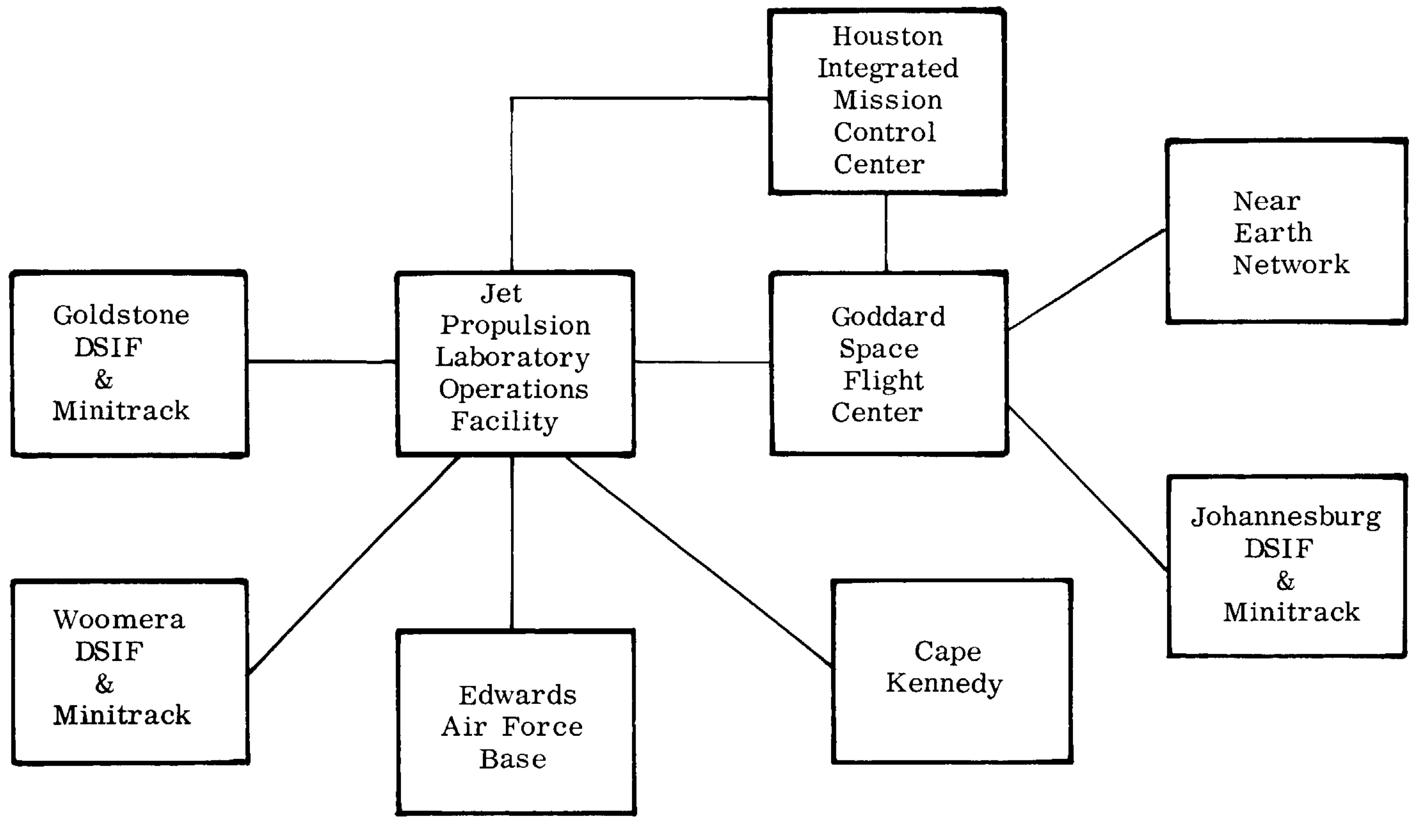


Figure 20-3. Communications Network, Saturn V



3-332

Figure 20-4. DSIF Communications Network

During the prelaunch phase, the instrumentation system is activated to provide the data link between the vehicle and the checkout equipment. The systems information is provided through the vehicle digital data acquisition system (DDAS) to the checkout facility by coaxial cable from each stage.

During the flight phases of the mission, instrumentation provides the vehicle performance data required by the range and crew safety systems, and the checkout and command verification information needed to direct the mission.

The complexity of the launch vehicle and its missions require a large number of measurements. The measuring program estimates at this time are listed in Table 20-2.

Table 20-2. Measuring Program Estimates

Stage	Measurements
S-IC	875
S-II	930
S-IVB	350
Instrument Unit	350
	Total 2555

This large number of system parameter measurements is obtained by several types of transducers. A typical list of transducers employed and the type of measurement obtained is provided in Table 20-3.

Table 20-3. Typical Transducers and Measurements

Transducers	Measurements
Vibration pressure transducer	Engine combustion chamber pressure
Force balance accelerometer	Lateral acceleration (pitch and yaw axes)
Rate gyro	Angular velocity of the vehicle
Tachometers	RPM of turbopumps

Table 20-3. Typical Transducers and Measurements (Cont'd)

Transducers	Measurements
Flowmeter	Rate of propellant flow
Resistance thermometer	Cryogenic measurements
Calorimeter	Thermal flux
Thermocouple	Temperature
Piezoelectric accelerometer	Vibration
Microphone	Acoustic energy

20-17. OPERATION.

To retrieve the required data, instrumentation system elements are located both in the launch vehicle and on the ground. Each stage has an independent instrumentation system. Figure 20-5 illustrates the signal flow through the system. The transducers convert the physical quantities to be measured (e.g. pressure, temperature, etc.) into electrical signals. These transducer signals are modified by signal-conditioning devices into voltages suitable as inputs to the telemetry system. The measuring distributor feeds the conditioned transducer signals to the telemetry system. In the telemetry system, the signals are modulated on RF carriers and transmitted to the telemetry ground stations.

20-18. Measuring System. The measuring system includes transducers, signal conditioners, and measuring distributors. Figure 20-6 illustrates typical components of the measuring system. The following description of the measuring system is for the S-IC stage. Measuring systems in the other stages and the instrument unit are similar but not identical to the S-IC stage.

The measurements are divided into two groups. In the first group, physical quantities such as pressure, temperature, and vibrations are transformed by transducers into electrical signals suitable for transmission. The second group of measurements are signals (voltages, currents, and frequencies) which are used for monitoring the performance of onboard equipment and the sequence of flight events (e.g., stage separation, engine cutoff, and others). The signals to be measured exist in analog and digital form.

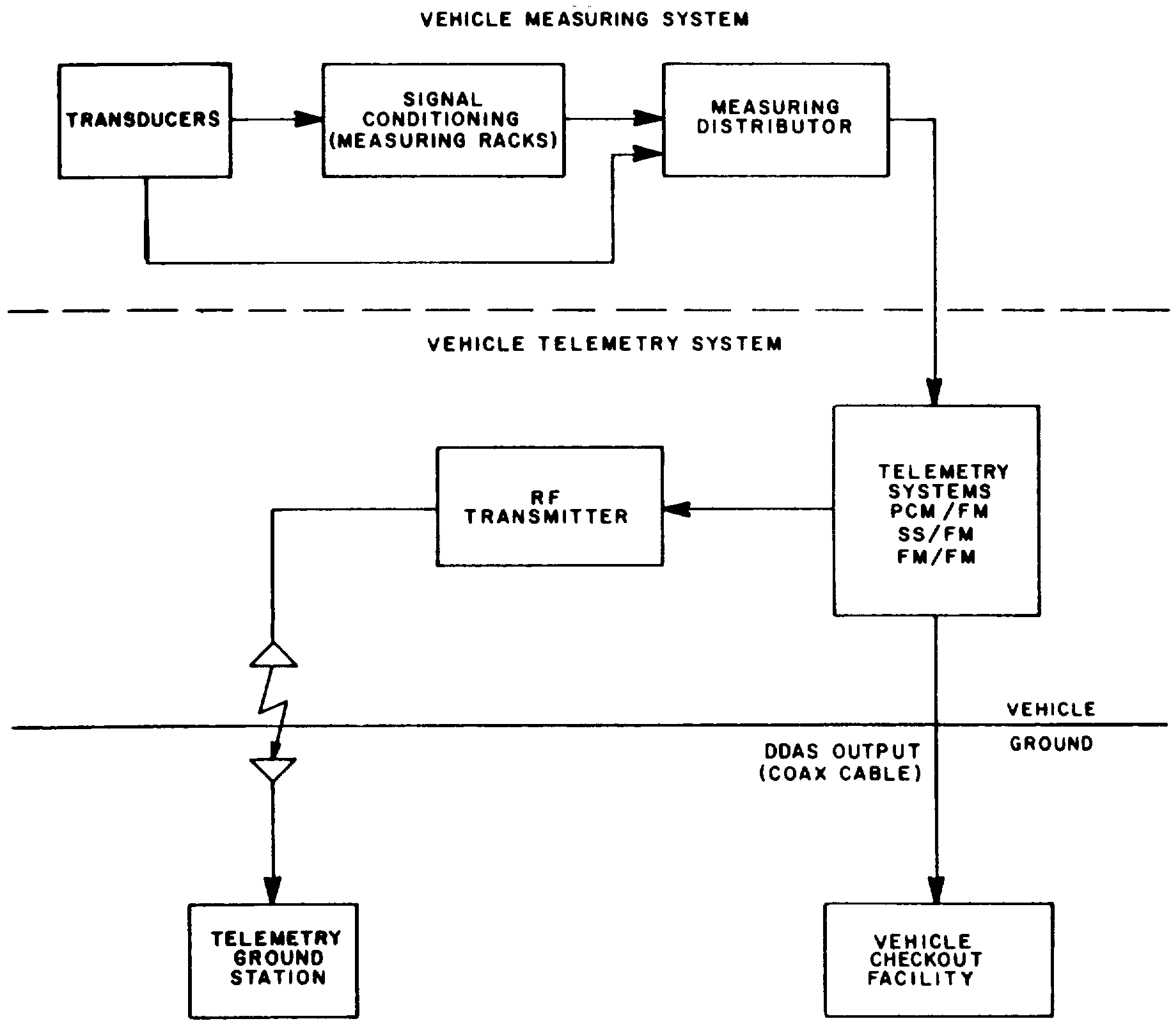


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Figure 20-5. Instrumentation System, Saturn V

Transducers. The transducers are precision electro-mechanical measuring instruments containing sensing devices carefully designed for accuracy, reliability, and resistance to unfavorable environment. Evaluation of vehicle performance and in-flight monitoring requires the measurement of a large variety of physical quantities onboard the vehicle. Therefore, many different types of transducers are used.

Signal Conditioning. Signal-conditioning modules are employed to adapt the outputs of the transducers to the electrical input requirements of the telemetry system. The modules are mounted in measuring racks which provide flexibility and ease of maintenance. Certain transducers have output signals which do not require signal conditioning. These signals are

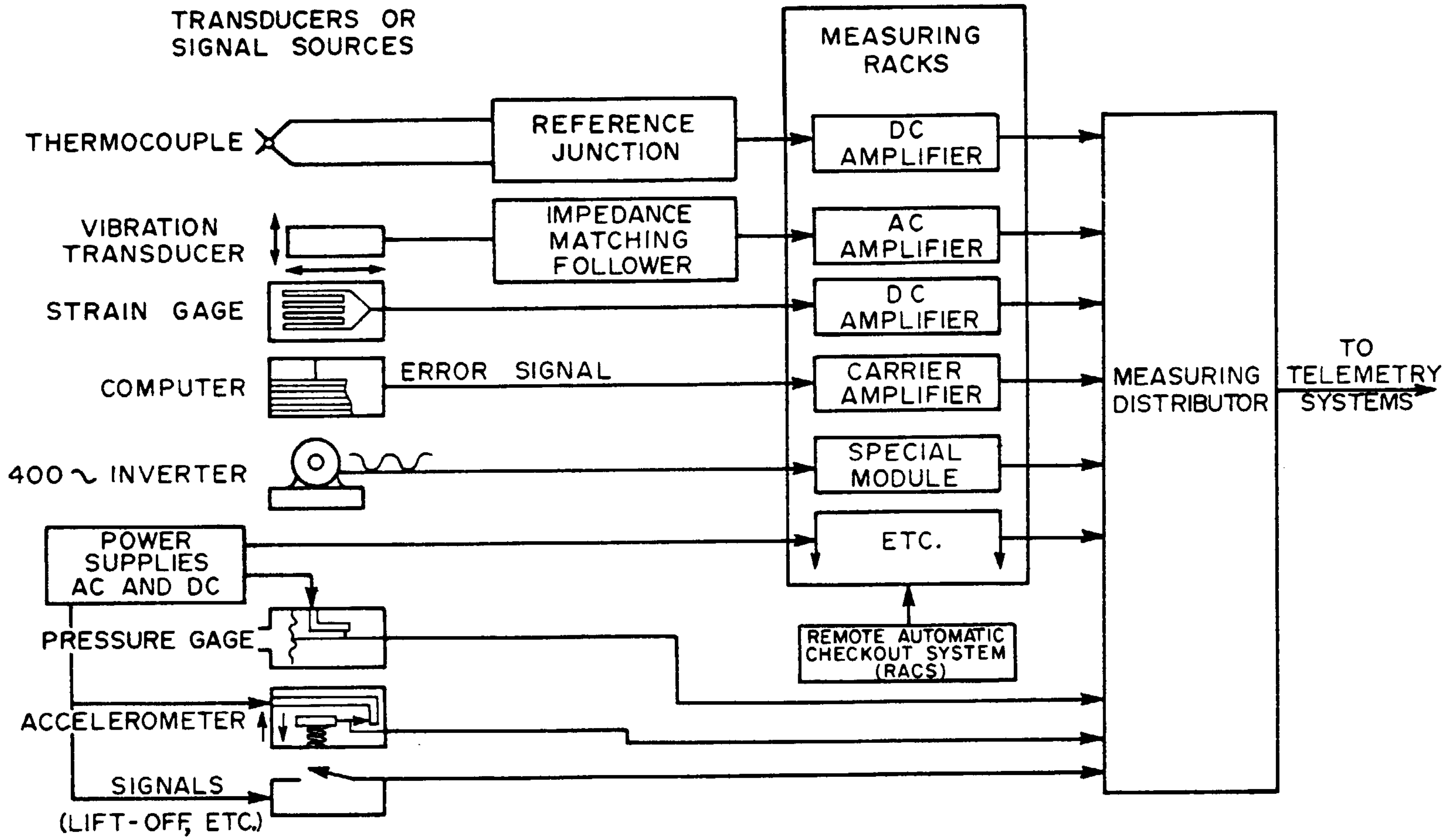

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Figure 20-6. Measurement System, Saturn V fed directly to the measuring distributor.

The power module input is 28 volts dc. Most modules contain isolated regulated power supplies for transducer excitation. The design of the plug-in printed circuit board enables amplifier adaptation to several different types of measurements, and changes in the range of measurements. The printed circuit board also includes transducer-simulating circuits for calibration purposes. There are four standard modules which are used in addition to the regulated power supply and non-standard modules. These are:

a. AC amplifier
b. Carrier amplifier
c. Narrow band dc amplifier
d. Wide band dc amplifier

The ac amplifier is a relatively wide-band ac amplifier with a frequency response of 10 Hz to 3100 Hz. The amplifier input impedance is 10,000 ohms, which is compatible with standard sensing devices in common use. The output signal is a waveform that is linear 0 to 5 volts, peak to peak. A bias voltage, applied at the output of the amplifier, provides a zero offset of 2.5 volts at the center frequency. The output signal is then applied to the 0-to-5 volt, voltage-controlled, subcarrier oscillator (SCO) or to the SS/FM. A signal-limiting device, at the output of the amplifier, prevents crosstalk or interference with other channels which could result from overdriving the subcarrier oscillator. Two types of gain control are provided in this unit: a step type and a continuous control. These are connected in series and may vary the gain from 1 to 240.

The carrier amplifier is primarily used to amplify signals from strain gages and similar pick-offs such as rate gyros. This amplifier is similar to the vibration amplifier, but has a balanced ring demodulator and a highly selective low-pass LC filter at the output. The gain control is the same as for the ac amplifier.

The narrow-band ac amplifier is primarily used to amplify low-level signals (in the millivolt range) which may be derived from thermocouples, resistance thermometers, thermistor bridges, or similar transducers. Solid-state devices are used to solve the drift and low reliability problems normally associated with amplification of low-level dc signals. A 10-volt regulated independent bridge supply is provided for use when thermistor, resistance thermometer, and straingage bridges require energizing. This voltage may also be used in thermocouples for the artificial reference junction. The bridge is located on the signal-conditioning plug-in-board. (Nominal gain for this narrow-band dc amplifier is 1000.)

The wide-band dc amplifier is energized by a 28-volt dc source and operates in essentially the same manner as the carrier and ac amplifiers. The frequency response is zero to 3 Hz.

Measuring Distributor. The measuring distributor is similar to a junction box. All measurements in the measuring system are connected to the distributor and are directed to their pre-assigned channel. The distributor provides versatility in changing channel assignments, with the changes being made by physically re-arranging jumper wires within the measuring distributor. This versatility eliminates extensive cable changes and allows channel changes to be made just prior to launch.

A remote automatic calibration system (RACS) (Figure 20-7) enables a remote calibration of the flight instrumentation system and equipment used for maintaining the functional readiness of the vehicle, thus affording a great savings in time during launch preparations.

Each signal-conditioning module contains two relays and the necessary circuitry required to simulate the transducer as well as the upper (hi) end and the lower (lo) end of the calibrated range for the measurement. The transducer is connected to the module only in the run mode.

A control panel in the Launch Control Center (LCC) allows selection of the desired measurement module in the vehicle and the calibration mode (hi, lo, and run). This is accomplished by sending a binary-coded signal from the LCC through the umbilical cable to the vehicle. Any number of channels can be selected and energized in any of the three modes, either individually, or in a random sequence.

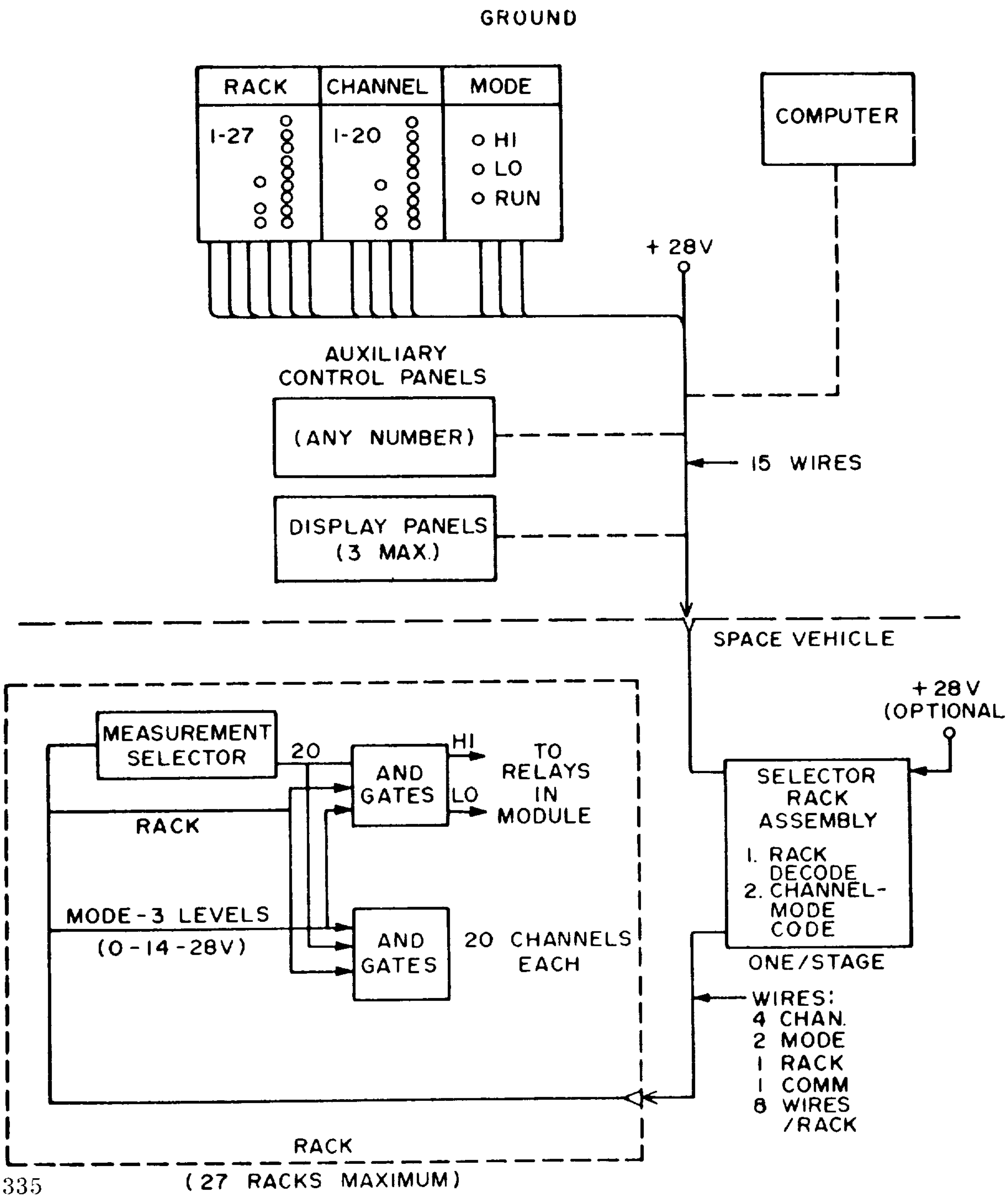

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Figure 20-7. Remote Automatic Calibration System (RACS) Block Diagram

Each of the signal-conditioning amplifiers has push-buttons on the front of the module for manual operation of the calibration inside the vehicle. The system is operated from the LCC computer, or other programming device. Data readout and display equipment is provided in the LCC.

20-19. Telemetry System. Each stage of the launch vehicle has an independent measuring and telemetry system with very little interfacing of measuring lines between stages (Figure 20-8). Before launch, coaxial cables from each stage-telemetry system supply digital data to the checkout facility through the Digital Data Acquisition System (DDAS). During flight, the telemetry data are radiated from separate antenna systems on each stage. The data adapter in the instrument unit has access to telemetry data from both the instrument unit and S-IVB stage.

In the telemetry system, the conditioned measuring signals are modulated on radio frequency carriers. Some measuring signals (e.g., vibration measurements) require wide bandwidths while other measurements which change very slowly require narrow bandwidths. The measurements, when grouped according to frequency and accuracy requirements, can be most effectively transmitted by employing different types of modulation techniques. Table 20-4 lists the Saturn telemetry systems for each stage.

a. PAM/FM/FM - Pulse Amplitude Modulation/Frequency Modulation/Frequency Modulation

b. FM/FM - Frequency Modulation/Frequency Modulation

c. SS/FM - Single Sideband/Frequency Modulation

d. PCM/FM - Pulse Code Modulation/Frequency Modulation

The standard inter-range instrumentation group (IRIG) telemetry channels are listed in Table 20-5.

20-20. Types of Multiplexing. Each stage data system utilizes three telemetry multiplexing techniques on multiple RF carriers:

a. FM/FM, with PAM and Triple FM as auxiliary techniques;

b. SS/FM;

c. PCM/FM.

The number of R-F carriers utilizing each technique is chosen to provide a balanced-data transmission capability for the variety of data types originating on the stage. A typical stage of the R&D vehicle requires 500 to 800 measurements varying in frequency response requirements from very low to 3000 Hz per channel.

The telemetry equipment associated with a Saturn V stage consists of a "building-block" arrangement, which may be connected in numerous combinations to satisfy

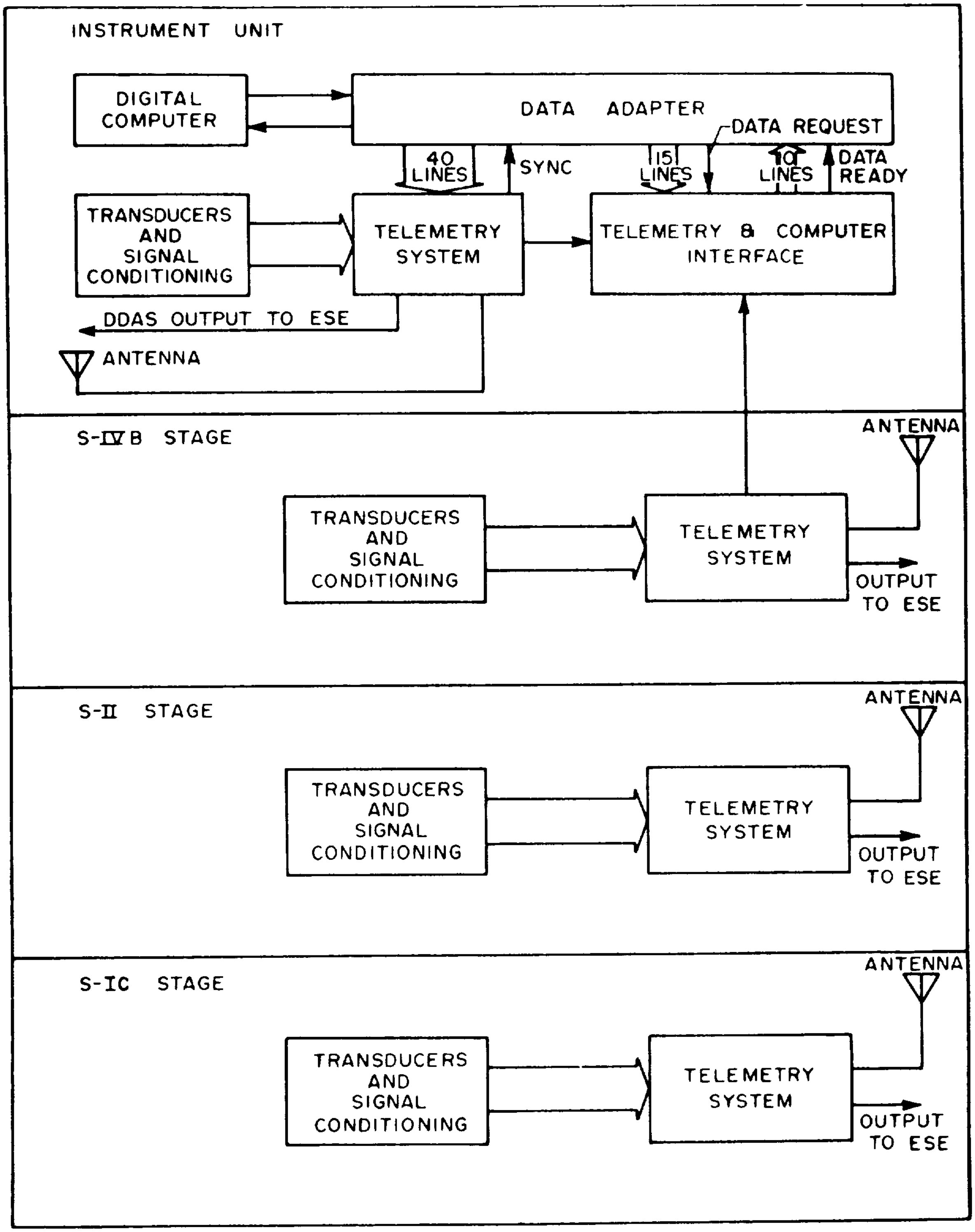

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Figure 20-8. Stage Instrumentation, Saturn V

Table 20-4. Saturn V Telemetry Systems

Stage	Telemetry System	No. of RF Links	Channels Available	Transmitter Frequency	Transmitter Power, Watts
IU	PAM/FM/FM	1			
	FM/FM	1		225-260 MHz	
	SS/FM	1	500		20
	PCM/FM	1			
S-IVB	FM/FM	3		225-260 MHz	
	SS/FM	1	1000		20
	PCM/FM	1			
S-II	PAM/FM/FM	3		225-260 MHz	
	SS/FM	2	1000		20
	PCM/FM	1			
S-IC	PAM/FM/FM	3		225-260 MHz	
	SS/FM	2	1000		20
	PCM/FM	1			

specific requirements. A typical stage telemetry system is illustrated in block diagram form in Figure 20-9.

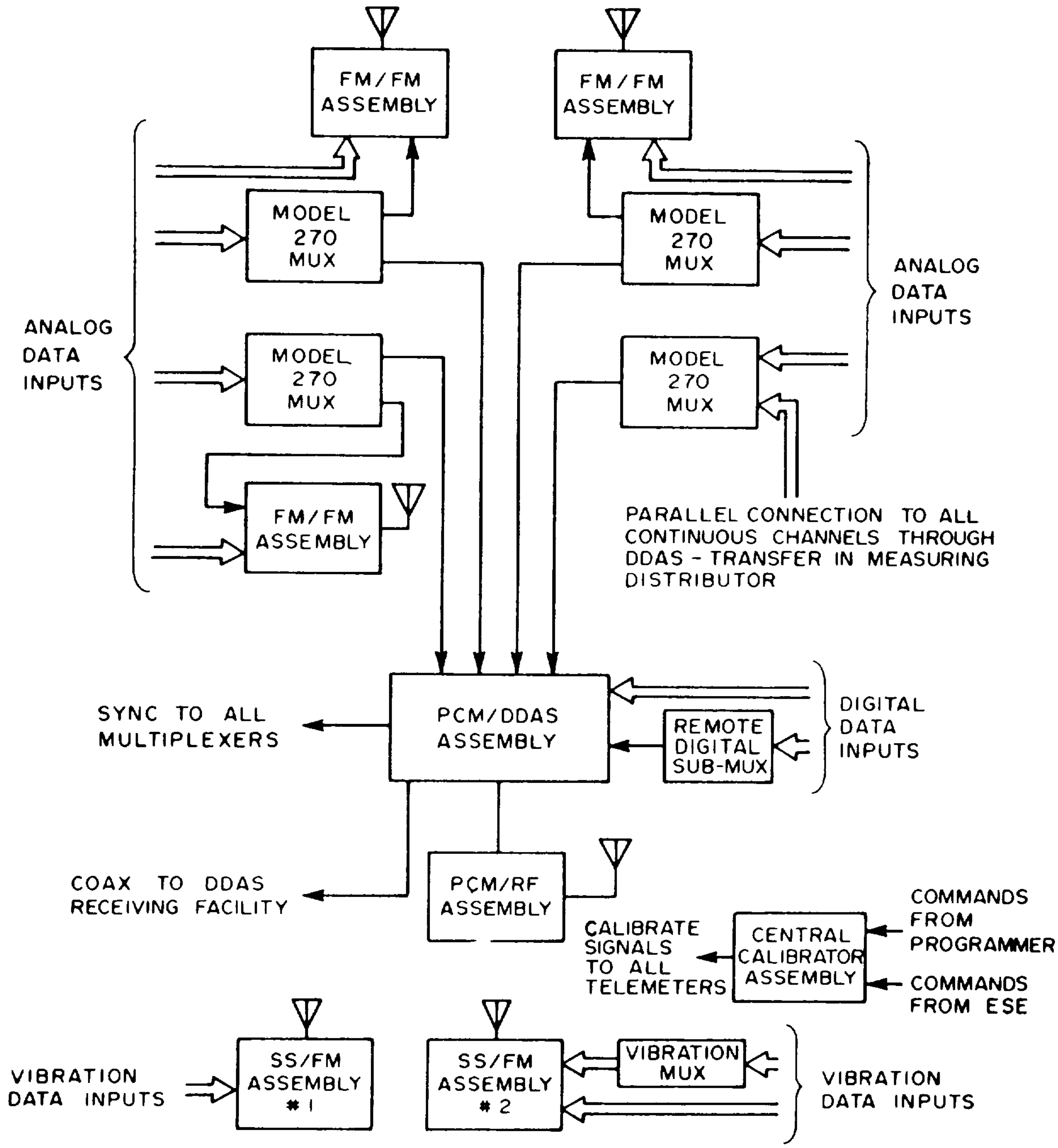
From one to six time-division multiplexers are synchronized from a central timing source located in the PCM/DDAS assembly. Each time-division multiplexer provides an output to the PCM/DDAS assembly which combines the outputs into a single serial wavetrain. The individual analog samples are digitized and combined into a digital format which is transmitted via coaxial cable to the ground checkout equipment. This data is also transmitted via a PCM/FM carrier for in-flight monitoring.

Table 20-5. Standard IRIG FM Subcarrier Bands

Frequency (Hz)				Maximum Intelligence Frequency (Hz)
Band	Lower Limit	Center Freq.	Upper Limit	
1	370	400	430	6
2	518	560	602	8.4
3	675	730	785	11
4	888	960	1032	14
5	1202	1300	1398	20
6	1572	1700	1828	25
7	2127	2300	2473	35
8	2775	3000	3225	45
9	4607	3900	4193	59
10	4995	5400	5805	81
11	6795	7350	7901	110
12	9712	10,500	11,288	160
13	13,412	14,500	15,588	220
14	20,350	22,000	23,650	330
15	27,750	30,000	32,250	450
16	37,000	40,000	43,000	600
17	46,560	52,000	56,440	790
18	64,750	70,000	75,250	1050

Each of the time-division multiplexers has a second data output which is identical to the output provided to the PCM/DDAS assembly except that it is conditioned for PAM transmission. These outputs may modulate a 70-KHz voltage-controlled oscillator (VCO) in FM/ FM telemeter assemblies. This arrangement provides redundant transmission of some multiplexer outputs using both PAM and PCM techniques.

Data with medium frequency response characteristics (50 to 1000 Hz) are applied to VCO's of the FM/FM assemblies. In some cases, lower frequency VCO's are modulated onto higher frequency VCO's to increase the number of available VCO



3-337

Figure 20-9. Typical Stage Telemetry System, Saturn V

data channels. This technique is referred to as triple FM (FM^3).

Vibration and acoustic data channels are typically applied to channels of the SS/FM assembly. These channels transmit a data spectrum from 30 to 3000 Hz. The number of SS/FM channels available is expanded by time-sharing specific channels through a slow time-division multiplexer (three or six seconds per contact).

Data that originates in digital form is inserted into the PCM/FM and DDAS outputs of the telemetry system. Typical sources of data in this category are the guidance system, the horizon sensor system, the command system, and discrete (off-on) measurements. These data channels are programmed into selected time slots of the digital format in the PCM/DDAS assembly. The number of digital input channels available in the PCM/DDAS assembly is expandable by adding remotely located digital submultiplexers.

The central calibrator assembly provides calibration commands and calibration reference signals to all assemblies. The reference signals are derived from the stage measuring supply. Calibration sequences are of two types: preflight, initiated from ESE; and in-flight, which may be initiated either from ESE or the vehicle programmer.

20-21. Telemetry and Computer Interface. The telemetry system in the S-IVB/IU functions during launch, earth orbit, and lunar-injection phases of the mission. During these phases, periodic checks are required of the vehicle's performance or operating status. This is accomplished by inserting specific segments of the telemetered information into the computer. (To accomplish the necessary interchange of information between telemetry and computer, the system operates as shown in Figure 20-8.)

During orbital checkout, which is initiated by a command signal to the digital computer via the instrument unit command receiver, the digital computer requires a real-time value of measurements, which are part of the total measurements being telemetered by the S-IVB/IU stage telemetry system. The computer provides a 15-bit address identifying the specific measurement value required by the instrument unit telemetry system. The computer also supplies a data-request signal.

Upon receipt of the address and data request, the instrument unit telemetry scans its stored addresses until a correct comparison is obtained. The telemetry then seeks the required data which is normally being transmitted at a rate of either 120, 40, 12, or 4 times per second. When the telemetry system obtains the correct data, it puts the data, a 10-bit word, into an output register, then provides a "data-ready" signal. It then branches to a sub-routine which operates to transfer the data from the telemetry output register to the data adapter. Synchronization between the telemetry system and the data adapter is accomplished in the following manner. Each time the telemetry receives an address from the data adapter followed by a valid data-request signal, it recognizes this input as the initiation of a new data-seeking cycle as well as a signal to read in the data. Upon this recognition, the telemetry first resets its output data register and then begins seeking the data requested by the data adapter. The data adapter and digital computer ensures that a new address with a valid read bit is not generated until data from the telemetry output register has been received in response to the previous address.

During the launch, earth orbital, and lunar-injection phases, there are times when information processed by the computer is desired at the ground station. Also, during periods when specific commands are being given through the instrument unit command to the digital computer, it will be necessary to transmit to ground the particular command prior to processing by the digital computer. Since the information to be telemetered is dependent on particular missions and has a random characteristic, provision will be made in the telemetry to accommodate these outputs. Specific PCM telemetry system channels are assigned to accommodate the 40-bit data adapter outputs. The assigned channels are sampled at a rate of 240 times per second.

The data adapter identifies valid data by the presence of a validity bit which has no significance to the telemetry, but is transmitted as part of the data telemetered to the ground. The ground computer automatically determines the existence of valid data by recognizing the validity bit in a data word. The validity bit is present with the valid data for at least 4.5 milliseconds to ensure at least one transmission of the valid data.

20-22. Digital Data Acquisition System. The digital data acquisition system (DDAS) is a function associated with Saturn V PCM telemetry and is utilized in both preflight and flight phases.

During preflight checkout, the telemetry system presents data over coaxial cables to one or more locations remote from the vehicle. These measurements are available to digital computers in real time through a special data-receiving facility interfaced with the computers. The data-receiving facility also provides outputs for display of selected channels in either digital or analog form for visually determining the status and readiness of vehicle subsystems and tape records the DDAS inputs for analysis at a later time. During flight, the DDAS function is performed between the telemetry system, data adapter, and digital computer. Upon request, data in digital form is made available to the digital computer during flight and is used by the digital computer to perform vehicle checkout.

20-23. SS/FM and FM/FM Modulation Systems. The SS/FM telemetry system, Figure 20-10, is designed specifically for transmission of the large volume of vibration data from the Saturn vehicle. This system can transmit 15 channels, each having a response of 30 to 3000 Hz, for a total data bandwidth of approximately 45 KHz within the standard telemetry RF carrier bandwidth.

Each of the 15 data inputs is fed to a balanced modulator and heterodyned with a 455-KHz carrier. The output of the modulator is fed to a mechanical bandpass filter (455 to 458 KHz) which passes the upper sideband. The output of the filter is fed to a second balanced modulator where it is translated to the proper baseband frequency. The baseband position is determined by the carrier supplied from the frequency synthesizer. The two balanced modulators and the mechanical bandpass filter for each data channel make up the channel units, which are identical for all channels. The outputs of the 15-channel units are mixed and amplified to the proper level to modulate the FM transmitter.

The frequency synthesizer generates the 15 carriers for the second modulator and a 75.83-KHz pilot tone for the ground equipment. To provide a 3-KHz information bandwidth and allow sufficient guardband, a channel spacing of 4.74 KHz is used. This spacing is convenient to generate in the synthesizer and allows adequate guardband of 1.74 KHz. The 75.83-KHz pilot tone falls just above the highest baseband frequency. It is used as a reference in the ground demodulation equipment to regenerate the basic 455 KHz and 4.74 KHz. Since the amplitude of the transmitted 75.83-KHz pilot is regulated, it is also used as an automatic gain control (AGC).

The SS/FM is used in conjunction with a vibration multiplexer to expand its data-handling capability by time-sharing specific data channels.

The FM/FM system configuration for each vehicle stage is selected to accommodate

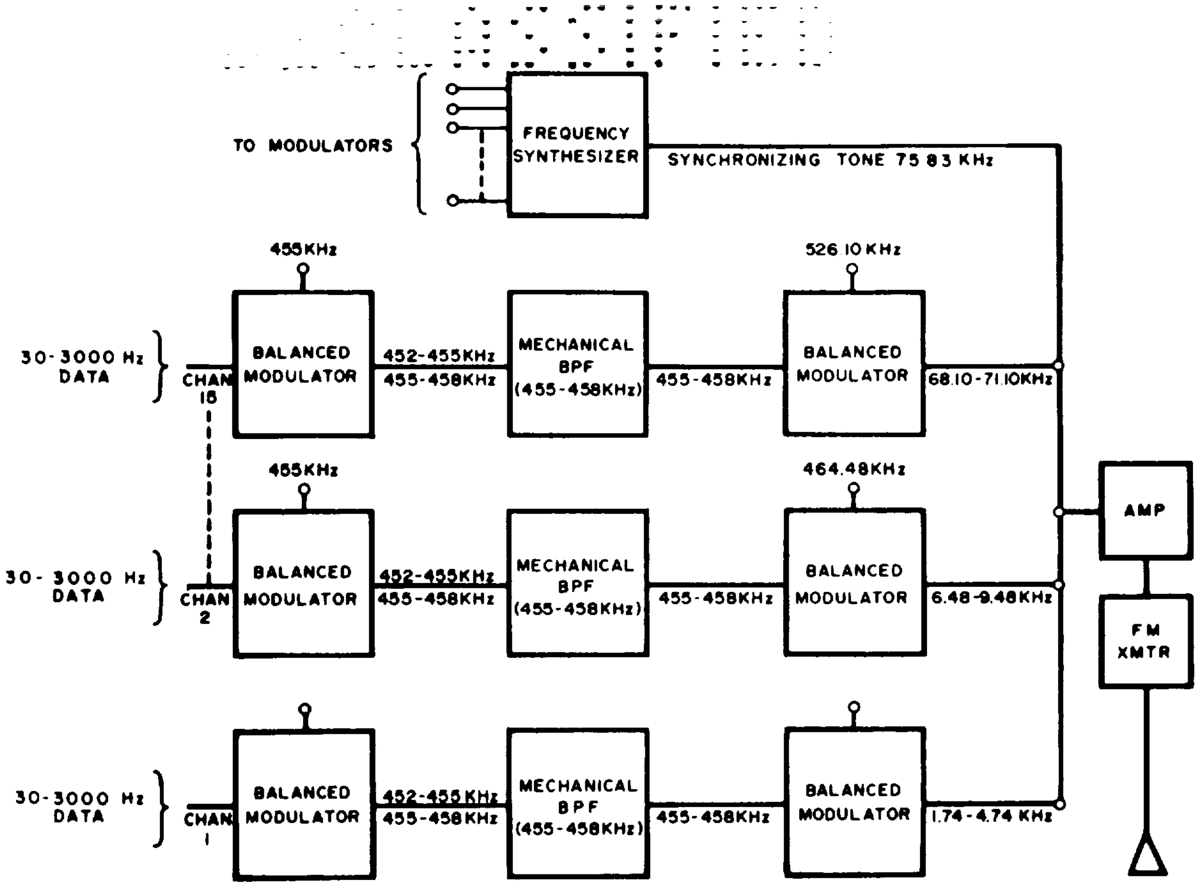


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Figure 20-10. SS/FM Telemetry System, Saturn V

the particular types and amounts of measurements unique to a stage. The basic modulation scheme and principal components used (subcarrier oscillators, mixer, power amplifier, and transmitter) are essentially the same for each stage FM/FM system. Figure 20-11 illustrates a typical Saturn stage FM/FM system.

Pulse amplitude modulation (PAM) and triple FM techniques are applied to specific subcarriers to expand channel capacity when required. Pulse amplitude modulation when used, is at a pulse rate of 3600 samples per second and is modulated onto a 70-KHz wideband VCO deviated $\pm$ 30 per cent. All IRIG channels above 30 KHz must be eliminated when this technique is used. When PAM is not utilized on a specific FM/FM link, IRIG channels 2 through 18 are used. Triple FM modulation is typically applied on any IRIG channel above 13.

The signal flow through the system is essentially the same for each channel. The channel receives a signal from the measurement system. When the measurement source signals are unsuitable for direct input to the FM/FM telemetry, signal-

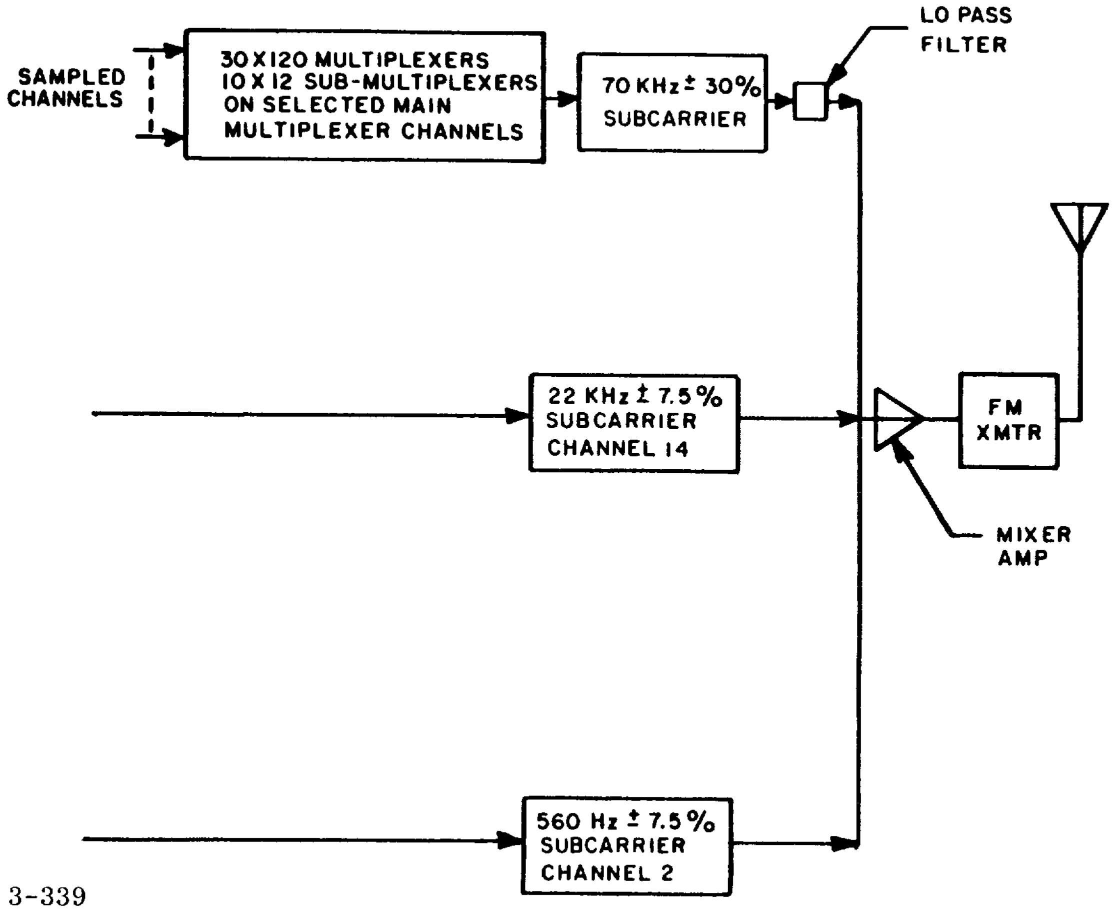

Figure 20-11. Typical Stage FM/FM Telemetry System, Saturn V

conditioning devices are used. The input signal modulates a voltage-controlled subcarrier oscillator which modulates the FM transmitter. The RF power amplifier amplifies the FM/FM output signal to a 20-watt level. The frequency of transmission in the VHF band is from 225 to 260 MHz.

20-24. Digital Telemetry System. Digital telemetry techniques are utilized on the launch vehicle for the following functions:

a. Monitoring data sources that originate data in digital form
b. Monitoring of data required for real time evaluation
c. Monitoring of analog data sources requiring accuracy, but which are not compatible with analog telemetry techniques
d. Primary transmission (without back-up) of up to 20 per cent of the sampled data originating on a stage
e. Redundant transmission of sampled data which is also transmitted by PAM techniques.

Some of the digital data sources that are monitored are: a digital computer, a horizon sensor, a radar altimeter, an instrument unit command system, liquid level sensors, a fire detection system, the AROD tracking system, and numerous sources of discrete (off-on) functions. The data required for real-time monitoring for determination of vehicle readiness is provided in digital form on a 600-KHz carrier transmitted from the vehicle via coaxial cable.

A central telemetry assembly, the PCM/DDAS assembly, (Figure 20-12), provides the following functions:

a. Scans the PAM wavetrains of several PAM multiplexers in a programmed sequence and combines these wavetrains into a single PAM wavetrain
b. Encodes into 10-bit digital form the PAM samples in the wavetrain
c. Accepts data in digital form and programs it into selected time slots in the output serial format
d. Provides a 600-KHz FM modulated carrier as the DDAS output and an NRZ modulating output for the (PCM/RF) assembly;
e. Provides the synchronization outputs necessary to synchronize multiplexers and remote digital submultiplexers.

The PCM/DDAS assembly contains the six functional subsystems listed below:

a. PAM scanner (an associated program patch)
b. Analog-to-digital converter (ADC)
c. Digital multiplexing and formating logic
d. Clock timing and programming logic
e. DDAS voltage controlled oscillator (VCO)
f. Power supplies.

20-25. Calibration. The central calibrator is used in conjunction with FM/FM and PCM/FM telemetry. In each stage, it functions as calibration control and a reference signal source for up to six telemetry units. In addition to in-flight and preflight calibration, this calibrator provides input calibrations for all continuous channels. There are five steps (dc voltage levels) applied to each telemetry link.

The calibrator provides up to six outputs to energize the calibrate relays in each telemetry link at the appropriate time.

In-flight calibration is initiated by command from a program device or the

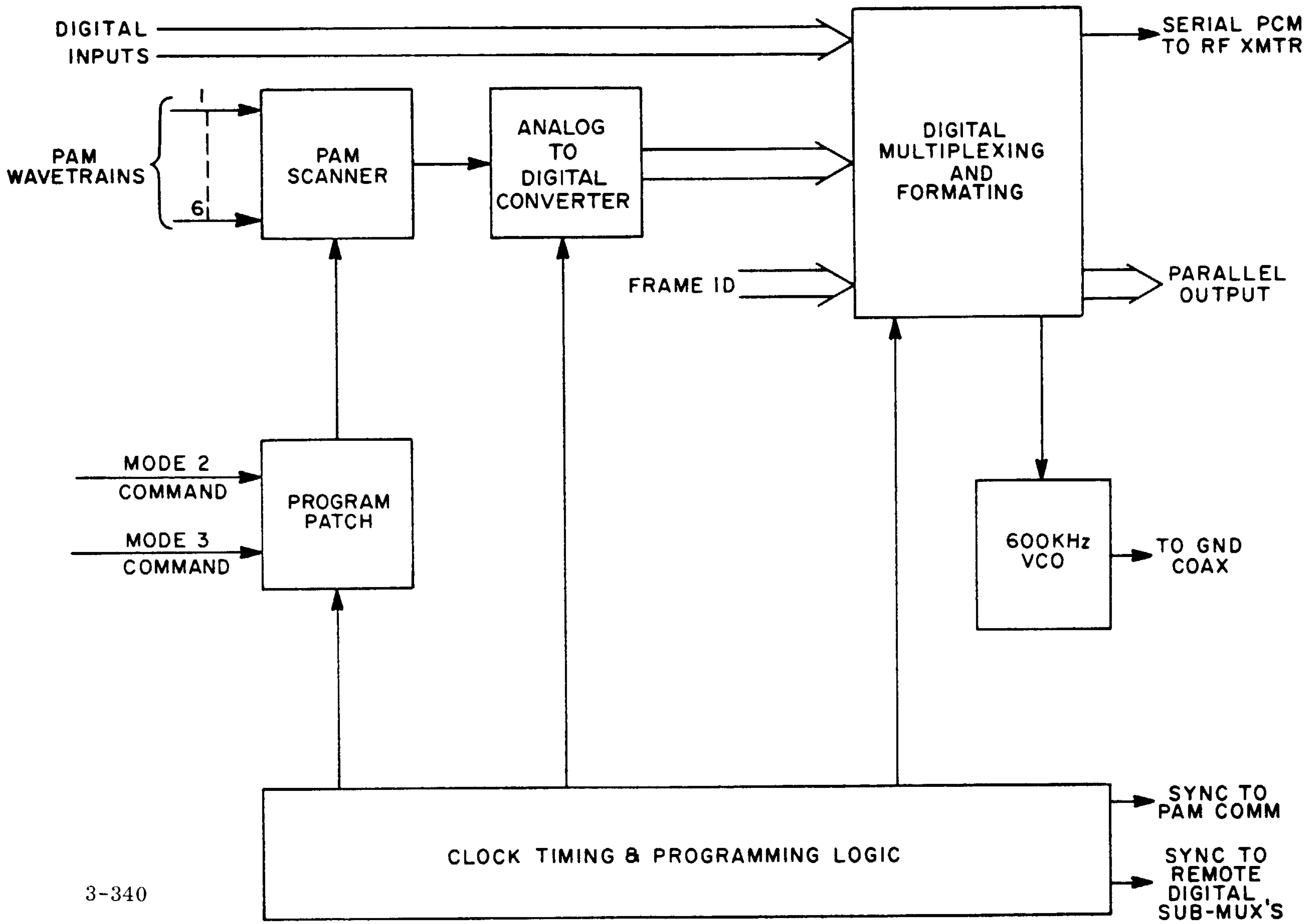

Figure 20-12. PCM/DDAS Assembly, Block Diagram, Saturn V

computer. Upon command, the calibrator supplies a control signal to a telemetry link which, in turn, transfers its measurement inputs to a calibration bus; simultaneously, the calibrator begins a five-step sequence, which appears on the calibration bus. When the step-sequence is completed, the calibrator transfers the control signal to another link and the calibration process is repeated. After all links have been calibrated, the calibrator assumes a quiescent state until the next command is received.

Control console switching, in the launch control center, sets the central calibrator to a preflight mode; this switching also sets all FM/FM telemetry equipment to a preflight mode. In the preflight mode, the inputs of all units are switched to the calibration bus so any signal appearing on this bus is applied to all telemetry channels. The calibrator supplies a signal to the calibration bus that may be a 0, 25, 50, 75, or 100 per cent level, or it may be a continuous step sequence of these levels. The calibrator preflight output may be selected from the control console in the launch control center.

20-26. Airborne Tape Recorder. The primary use of airborne tape recorders in the launch vehicles is for data storage during periods of flight which are not covered by ground stations. The stored data is transmitted upon command when ground station coverage is available.

The tape recorder is also used for critical environmental events occurring during vehicle flight. For example, pertinent R-F modulation data may be paralleled into the tape recorder during retromotor firing when resulting flame attentuation may be significantly affecting the RF signal transmission. At a later convenient time during flight, the tape recorder playback is used to modulate an RF transmitter and the delayed transmission of data during the retro fire periods is accomplished without the effects of RF flame attenuation.

20-27. Optical Systems. In addition to the conventional measuring system, a system consisting of motion picture film and television cameras is used to provide real-time data and a permanent record of vehicle systems operation where action like stage separation, retromotor firing, and propellant motion can best be visually observed.

Film Camera System. Recoverable film cameras are used in the S-IC and S-II stages. The cameras view liquid motion in the LOX container and, with two externally mounted cameras, view S-IC/S-II first plane separation forward. Two cameras mounted on the S-II stage, looking aft, view first and second plane separation between the S-IC and S-II stages.

The advantages of the film camera system are: high picture resolution (in color or black and white) and slow motion (high-speed photography) studies can be performed for analysis of performance.

Some disadvantages of the film camera system are:
Action cannot be viewed in real time, filming is limited to a few seconds (at high speeds), and the camera capsules must be ejected from the vehicle and recovered by ship or paradiver. The camera assembly contains a radio beacon which is active during the recovery phase and provides location vectoring.

In support of the film cameras, a system of lenses, fiber-optical devices, light sources and a precision timing system are used.

Television Systems. The Saturn V launch vehicle television system is used to provide both real-time and permanent visual data on the performance of certain vehicle functions.

A block diagram of the vehicle and ground equipment is shown in Figure 20-13. Table 20-6 lists the television characteristics. Up to four cameras may be used with a single sequence switcher to make observations at different locations in the vehicle. The sequence switcher selects the output of one to four cameras. A separate programmer is used to change the rate of switching or the number of cameras being switched. The camera control unit provides all scanning signals to the camera and also provides video amplification from the camera. The cameras may be placed up to 30 meters away from the control unit. The cameras are small, having a maximum outside diameter of seven centimeters and a length (excluding the lens system) of 35 centimeters. From one to seven cameras, with

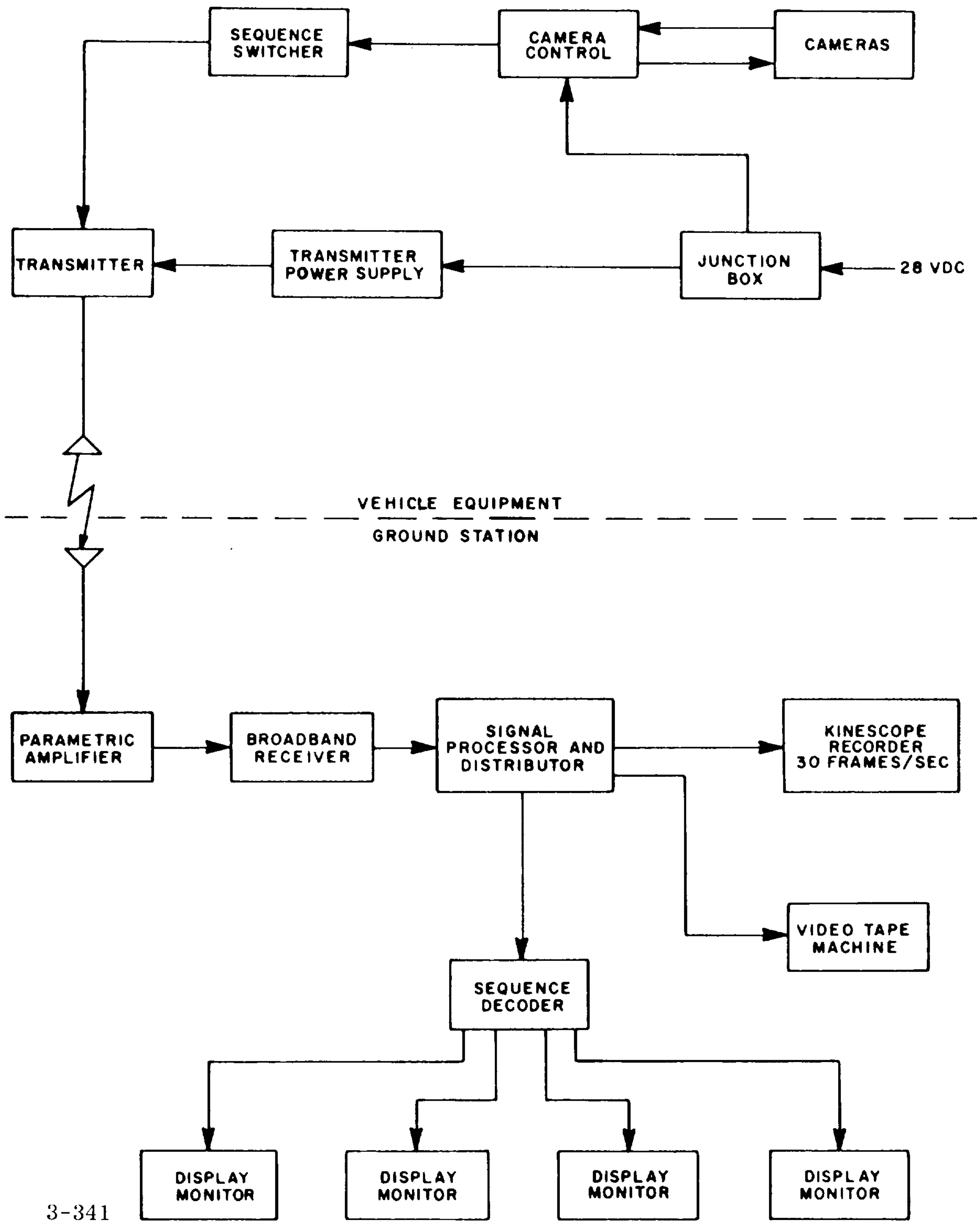

Figure 20-13. Vehicle/Ground Television System, Saturn V

Table 20-6. Saturn V Launch Vehicle Television Data

Item	Data
Transmitter	
Video bandwidth	8MHz
Modulation	FM
Deviation	16 MHz (for composite video)
Output power	2.5 watts min.
Unmodulated frequency	1700 MHz $\pm$ 0.20 %
Video resolution (horizontal) of received picture	600 lines
Closed Circuit Camera System	
Camera light sensitivity	1.0 foot candle
Video bandwidth	8 MHz
Frame rate	30 per sec.
Scanning	2:1 interlace
Specifications of Television Ground Station for Support of Saturn Television System	
Parametric Amplifier:	
Gain	20db
Noise figure	1.35 db
Frequency range	1700 to 1720 MHz
Receiver	
Frequency range	1700 to 1720 MHz
Gain	90db
Noise figures	12 db

Table 20-6. Saturn V Launch Vehicle Television Data (Cont'd)

Item	Data
Signal Processing and Distributing Amplifier	
Video bandwidth	8 MHz
Number of outputs	4
Sequence Decoder	
Video bandwidth each output	8 MHz
Number of outputs selectable	1 to 16
Switching time	0.1 usec
Video Tape Recorder	
Video bandwidth	5.5 MHz
Tape speed	15 in. per sec.
Recording time	96 min.
Kinescope recorder	
Camera frame rate	30 per sec.
Kine-monitor tube	White face, type P-4 phosphor
Film capacity	1200 ft.
Viewing monitor	
Video bandwidth	8 MHz
Video resolution (horizontal)	600 lines

control units, are used with a single transmitter. Television signals are transmitted to ground stations by frequency modulation.

A frequency modulated signal from the vehicle to the ground station receiver is decoded into separate signals representing the number of onboard cameras in use. A storage tube with continuous readout is used for each camera channel to provide continuous viewing or conventional monitoring.

The received signal is also recorded on video tape for post-flight analysis. The tape has frame code numbering, and when used with the storage tube, the system provides automatic selection and storage of any one frame of any camera.

In addition to the video tape recording, a kinescope recorder is used to make a 16-millimeter film recording of the intramixed camera signal transmission. The camera photographs one picture for each frame from each TV camera (30 pictures per second). These pictures are used to make single-frame enlargements for study purposes.

The ground monitoring and recording station consists of the following:

a. Parametric amplifier
b. Wide-band superheterodyne receiver;
c. Signal processing and distributing amplifier;
d. Sequence decoder;
e. A continuous readout storage tube;
f. Video tape recorder;
g. Kinescope recorder;
h. Storage tube for automatic frame selection from any camera.

20-28. IMPLEMENTATION.

(To be supplied at a later date.)

20-29. CHECKOUT

Checkout is the process of verifying that the launch vehicle is capable of performing its mission. This process consists of a series of tests that start at the component level during manufacturing and end during the prelaunch phase with a simulated flight test involving the complete vehicle.

In this description the checkout is confined to the tests that are performed on the composite stage after final assembly and inspection.

Checkout is performed on three levels, qualification, prelaunch, and launch site level. Qualification checkout is performed on an individual stage, prelaunch checkout during the assembly of the stages into a launch vehicle, and launch site checkout on the complete S/V. These checkouts are performed at various NASA facilities throughout the country and it is the intent that they shall be performed with test equipment and test procedures which are similar from facility to facility. This checkout philosophy will make it possible to assemble a history of the performance of the many subsystems and systems comprising the vehicle and on the basis of this history to make an accurate prediction of the probability for a successful mission prior to the launching.

20-30. CHECKOUT FLOW.

Each stage and the instrument unit of the launch vehicle will be individually qualified for flight through a series of tests consisting of: a prestatic checkout, static firing test and a post static checkout.

The three stages and the instrument unit are then shipped to the vertical assembly building (VAB) of launch complex 39 at the Merritt Island Launch Area. The VAB consists of two major areas - high bay, and low bay. Upon arrival at VAB, the S-II and S-IVB stages undergo visual and mechanical checks in the low-bay area. The S-IC stage is erected on the launcher-umbilical tower (LUT) in the high-bay area and mated with the integrated launch control checkout system (ILCCS). The instrument unit is assembled and taken to the high-bay area for connection to ILCCS. In this configuration, they are both checked out as separate stages. When the checkouts of the S-II and S-IVB stages have been completed, the stages are properly positioned on the S-IC stage. The instrument unit is then placed on the S-IVB stage to form a complete launch vehicle on the LUT. System tests of the complete launch vehicle are then performed using the ILCCS. The Saturn V checkout flow is illustrated in Figure 20-14.

20-31. IMPLEMENTATION.

The checkout of the Saturn V will be performed using computer controlled automatic checkout systems. These systems consist in general of a digital computer which controls a number of substations; each substation is designed to accomplish a separate category of tests. The major categories of tests are:

a. Electrical networks
b. Measuring, rough combustion cutoff, and fire detection
c. Telemetry
d. Radio frequency systems
e. Guidance and control systems
f. Mechanical systems
g. Vehicle systems

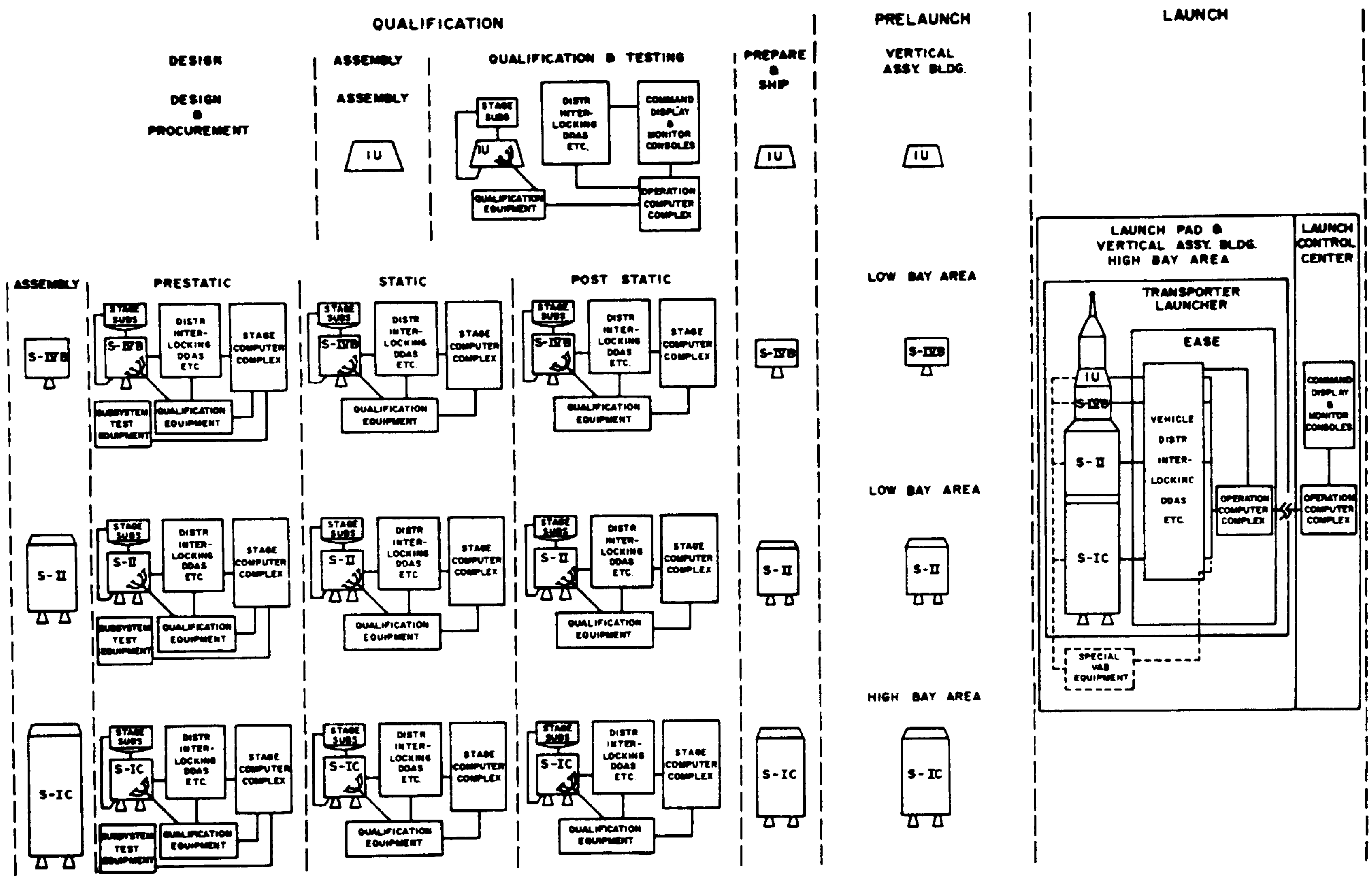

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Figure 20-14. Saturn V Vehicle Flow Diagram

The significant tests performed on a stage at the three checkout levels (qualification, prelaunch, and launch pad) are described in the following paragraphs.

20-32. Qualification. Qualification checkout is performed on individual stages and is the first checkout of an assembled stage. The purpose of this checkout is to qualify the stage for flight. Qualification is performed in three steps:

a. Prestatic checkout

b. Static firing test

c. Poststatic checkout

S-IC Stage. The S-IC stage will be manufactured by the Boeing Company at their Michoud facility. The prestatic checkout and the poststatic test of this stage for the first two vehicles (501 and 502) will be performed at MSFC in the Quality Laboratory using checkout equipment developed there. For succeeding vehicles the stage will be checked out at Michoud. (A detailed description of the checkout configuration and objectives will be supplied at a later date.)

S-II Stage. (To be supplied at a later date.)

S-IVB Stage. (To be supplied at a later date.)

Instrument Unit. The instrument unit will be manufactured at MSFC and all qualification tests will be performed there. The qualification program will include detailed tests and calibration of individual subsystems, plus a series of simulated flight tests culminating in a simulation of a complete mission using stage substitutes for the propulsion stages of the vehicle. (A detailed description of the checkout configuration and objectives will be supplied at a later date.)

20-33. Prelaunch Checkout at VAB. After qualification for flight, the individual stages are shipped to AMR. Upon arrival at AMR the stages are taken to the VAB.

The S-IC stage is inspected for shipping damage and erected in the LUT in the high-bay area and integrated with the launch checkout and control subsystem. The instrument unit is assembled and taken to the high-bay area for connection to the

S-IC and launch vehicle checkout computer system. In this configuration, the S-IC and instrument unit are checked out both as separate stages and as integrated parts of the launch vehicle. When the checks of the S-II and S-IVB stages have been completed, these stages are properly positioned on the S-IC stage and the instrument unit is then placed on the S-IVB stage to form a complete launch vehicle on the LUT. System tests of the complete launch vehicle are then performed using the ILCCS.

The ILCCS is composed of two main equipment groups, one group located in the Launch Control Center (LCC) and the other in the LUT.

The LCC contains the central checkout computer complex and all the control consoles and overall operation of the ILCCS is controlled from here.

The LUT equipment consists of the computer complex, the digital data acquisition system (DDAS), remote automatic calibration system (RACS), computer-launch vehicle communications lines, electrical support equipment (ESE), equipment to mate with facilities located in the pad interface and distribution equipment, and communication equipment required for operation under control of the LCC.

After all launch vehicle system tests have been completed, the spacecraft is mounted upon the instrument unit in the VAB, and the final VAB system tests are performed to verify that the space vehicle is ready for launch.

After all system tests have been satisfactorily performed, the LUT is moved by a crawler-transporter to one of the three launch pads servicing launch complex 39. The mobile arming tower for servicing the vehicle is also transported to the same pad by the crawler-transporter.

20-34. Launch Pad Checkout. Upon arrival at the launch pad, the utilities, service units, and data links at the pad are connected to the LUT system. The launch vehicle is then subjected to a pre-countdown verification of its subsystems to verify the new interfaces and to reaffirm system integrity after transfer of the vehicle from the VAB.

After final connections and tests, the mobile arming tower is withdrawn by the crawler-transporter, and the vehicle is ready for fueling and launch countdown. The entire launch countdown is controlled by the remote LCC computer complex and other associated equipment, using the LUT computer complex in the same manner as is used in the VAB.

20-35. ATTITUDE CONTROL AND STABILIZATION

The Saturn V attitude control and stabilization function maintains a stable vehicle motion and adjusts this motion in accordance with programmed attitude change, guidance or Apollo spacecraft commands.

20-36. REQUIREMENTS

During the ascent phase, the function directs the vehicle attitude orientation about its axes, maintains the angular rate of vehicle motion about its axes within allowable limits, and damps any first and second bending mode oscillations of the vehicle structure.

The ascent performance of the attitude and stabilization function is limited by various constraints.

The size and complexity of the launch vehicle and facility imposes the constraint of a specific launch orientation. The Saturn vehicle is required to maintain this launch orientation for several seconds after liftoff, permitting it to rise above the launch facilities to gain maneuvering room. During the S-IC stage flight, the high aerodynamic pressures encountered by the launch vehicle results in the requirement that the control system limit the angle of attack. A further constraint exists because of the natural bending of the vehicle structure, necessitating damping of first and second bending mode oscillations.

Immediately prior to vehicle staging, the attitude control and stabilization function restrains the vehicle to a constant attitude orientation to prevent excessive rotational rates during the separation. Following separation of the depleted stage and ignition of the succeeding stage, separation transients must be damped.

For S-II and S-IVB stage powered flight, the attitude control and stabilization function accepts steering commands and directs the vehicle motion accordingly. Orbital phase performance of the attitude control and stabilization function includes maintaining the vehicle attitude orientation constant in respect to the earth or producing vehicle attitude changes in obeyance to programmed commands or the Apollo spacecraft.

Prior to re-ignition of the S-IVB stage, the vehicle is oriented to the translunar injection orientation. This orientation is accomplished in response to Saturn guidance or Apollo spacecraft commands.

During the powered flight portion of the translunar phase, the attitude control and stabilization function accepts steering commands from either the Saturn guidance function or the Apollo spacecraft. After termination of powered flight, the attitude control and stabilization function maintains a stable orientation for the combination S-IVB stage/instrument unit and LEM while the remaining Apollo spacecraft separates from the LEM, performs a turn-around maneuver, and then reconnects to the LEM.

After final separation of the Apollo from the S-IVB stage/IU, the S-IVB/IU is propelled to a different trajectory utilizing the auxiliary propulsion system.

The attitude control system for the Saturn V vehicle is required to operate during powered flight of all stages and during the coast phase of the S-IVB/IU stage for a maximum total time of 6.5 hours.

20-37. OPERATION

Due to the various launch vehicle constraints, a programmed attitude control, without active guidance, is used for S-IC stage flight. The programmed attitude control is accomplished in three periods; launch stabilization, maneuvering, and prestaging stabilization. The launch stabilization begins with liftoff and terminates after several seconds during which time the vehicle rises vertically to attain a physical clearance with the launch facilities.

Upon termination of the launch stabilization period, the launch vehicle begins the maneuvering phase with a programmed roll maneuver. This maneuver consists of the launch vehicle maintaining a constant rate of roll until such time as its pitch plane coincides with the flight azimuth. Several seconds after initiation of the roll maneuver, the launch vehicle starts a gravity-turn, time-tilt maneuver. This maneuver rotates the longitudinal axis of the launch vehicle in the pitch plane toward the flight azimuth. A few seconds prior to vehicle staging, the time-tilt maneuver is terminated.

Prestaging stabilization is accomplished for several seconds prior to stage separation. During this period, the launch vehicle is restrained to a constant attitude orientation.

Control of the launch vehicle is achieved by gimbaling the outboard engines for thrust vector control. Figure 20-15 shows the basic equipment configuration. The desired vehicle attitude for the S-IC stage flight is programmed in the digital computer as a function of time.

Present attitude is measured by the inertial platform in the form of three angular measurements and is transmitted to the data adapter in analog form. After analog-to-digital conversion in the data adapter, the angular measurements are available to the digital computer. In the digital computer the angles are compared with the

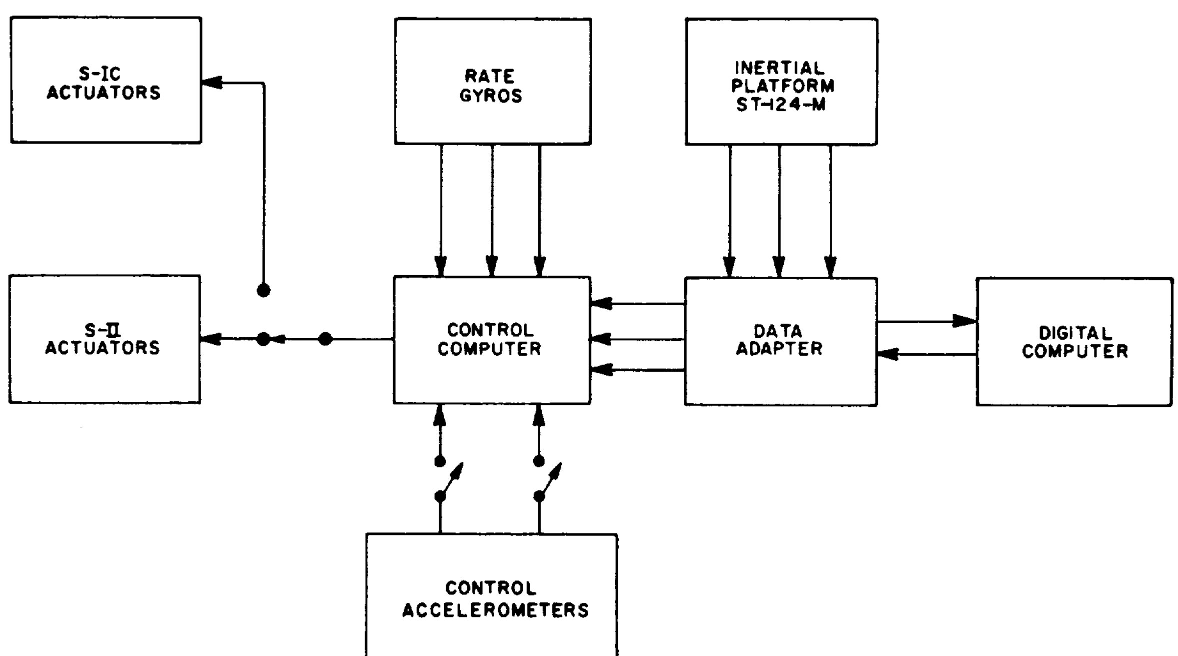

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Figure 20-15. Thrust Vector Control System for S-IC and S-II Stages desired attitude angles and the errors are resolved into the vehicle fixed coordinate frame. These error signals then go to the data adapter where digital-to-analog conversion is accomplished. This digital portion of the control loop has a recurrent rate of 25 to 50 per second.

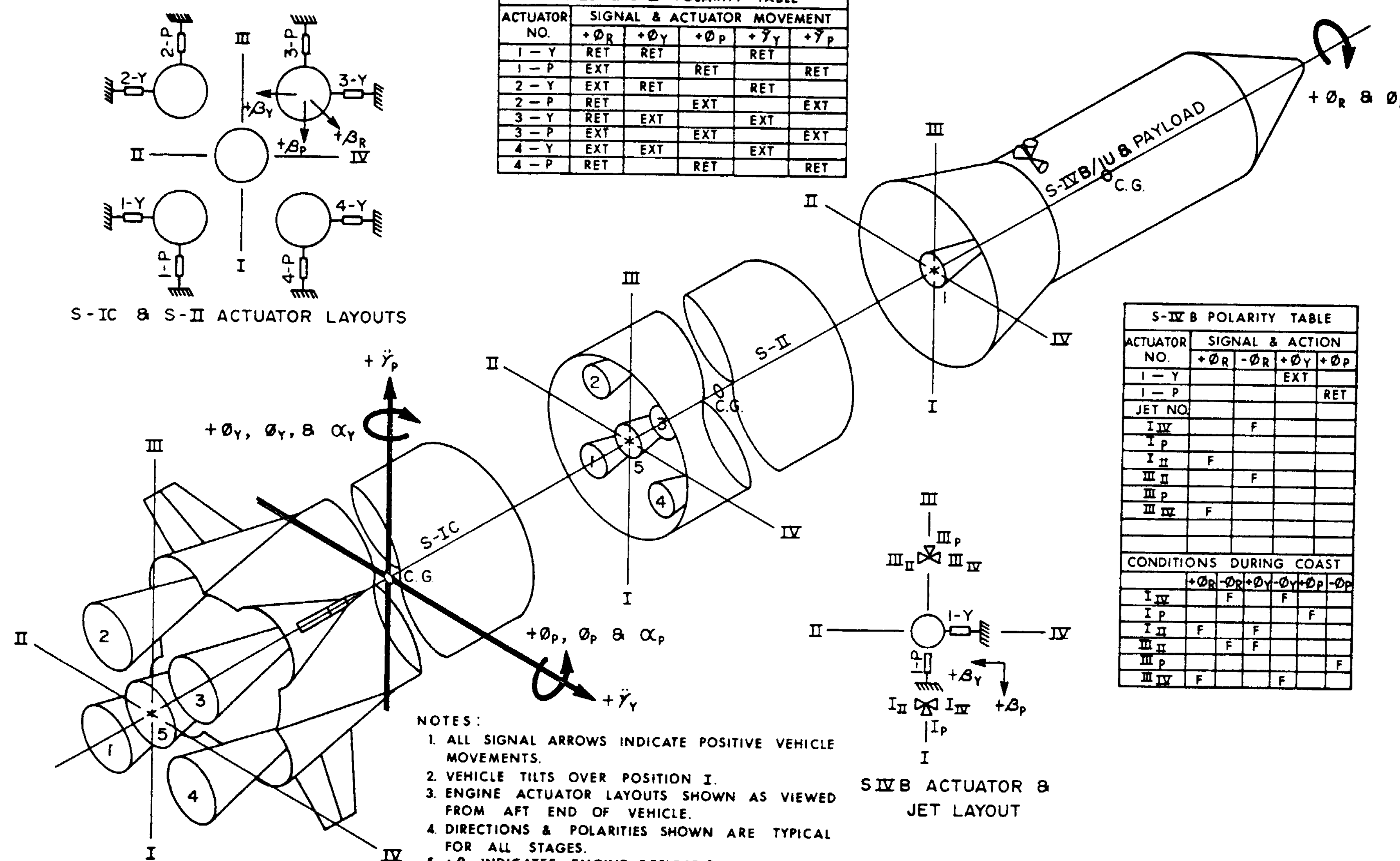
The analog outputs from the data adapter are transmitted to the control computer where the angular error signals are mixed with angular rate signals from a set of three rate gyros, along with two lateral acceleration signals from control accelerometers that are mounted along the pitch and yaw axes. Lateral accelerometers are required during the S-IC stage burn to provide angle-of-attack control. Rate gyros mounted in the S-IC stage are utilized during S-IC powered flight phase. The control computer filters all input voltages to remove local effects and provide gain and phase requirements to ensure stability of the vehicle in the presence of structural bending and propellant sloshing. Stabilization is accomplished in the first two bending modes utilizing phase stabilization and in higher modes utilizing gain stabilization.

In addition to doing signal mixing and filtering, the control computer contains the logic required to select the proper engine actuators for gimbaling and contains magnetic amplifiers to drive the torque motors which control the servo actuator valves. The engine actuators use mechanical feedback, thus requiring no electrical feedback to the magnetic amplifier. Figure 20-16 shows the gimbaling arrangement for four F-1 engines in the S-IC stage, and four J-2 engines in the S-II stage.

The operation of the attitude control system during S-II flight is similar to operation during S-IC flight. The lateral accelerometers for angle of attack control are not required because the vehicle is through the area of high dynamic pressures. The angular rate information is provided by rate gyros in the instrument unit. Desired vehicle attitude angles are calculated by the guidance system.

Attitude control for the S-IVB flight has several different modes. Figure 20-17 shows the data flow and switching for controlling these modes.

During S-IVB first burn, switches S-1 and S-2 are in the powered position and the



S-IC & S-II POLARITY TABLE

ACTUATOR NO.	SIGNAL & ACTUATOR MOVEMENT				
	$+\emptyset_R$	$+\emptyset_Y$	$+\emptyset_P$	$+\ddot{\gamma}_Y$	$+\ddot{\gamma}_P$
1 – Y	RET	RET		RET	
1 – P	EXT		RET		RET
2 – Y	EXT	RET		RET	
2 – P	RET		EXT		EXT
3 – Y	RET	EXT		EXT	
3 – P	EXT		EXT		EXT
4 – Y	EXT	EXT		EXT	
4 – P	RET		RET		RET

S-IVB POLARITY TABLE

ACTUATOR NO.	SIGNAL & ACTION			
	$+\emptyset_R$	$-\emptyset_R$	$+\emptyset_Y$	$+\emptyset_P$
1 – Y			EXT	
1 – P				RET
JET NO.				
I_{IV}		F		
I_P				
I_{II}	F			
III_{II}		F		
III_P				
III_{IV}	F			

CONDITIONS DURING COAST

	$+\emptyset_R$	$-\emptyset_R$	$+\emptyset_Y$	$-\emptyset_Y$	$+\emptyset_P$	$-\emptyset_P$
I_{IV}		F		F		
I_P					F	
I_{II}	F		F			
III_{II}		F	F			
III_P						F
III_{IV}	F			F		

NOTES:
1. ALL SIGNAL ARROWS INDICATE POSITIVE VEHICLE MOVEMENTS.
2. VEHICLE TILTS OVER POSITION I.
3. ENGINE ACTUATOR LAYOUTS SHOWN AS VIEWED FROM AFT END OF VEHICLE.
4. DIRECTIONS & POLARITIES SHOWN ARE TYPICAL FOR ALL STAGES.
5. $+\beta$ INDICATES ENGINE DEFLECTION REQUIRED TO CORRECT FOR POSITIVE VEHICLE MOVEMENT.

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Figure 20-16. Saturn V Astrionics Polarity Chart

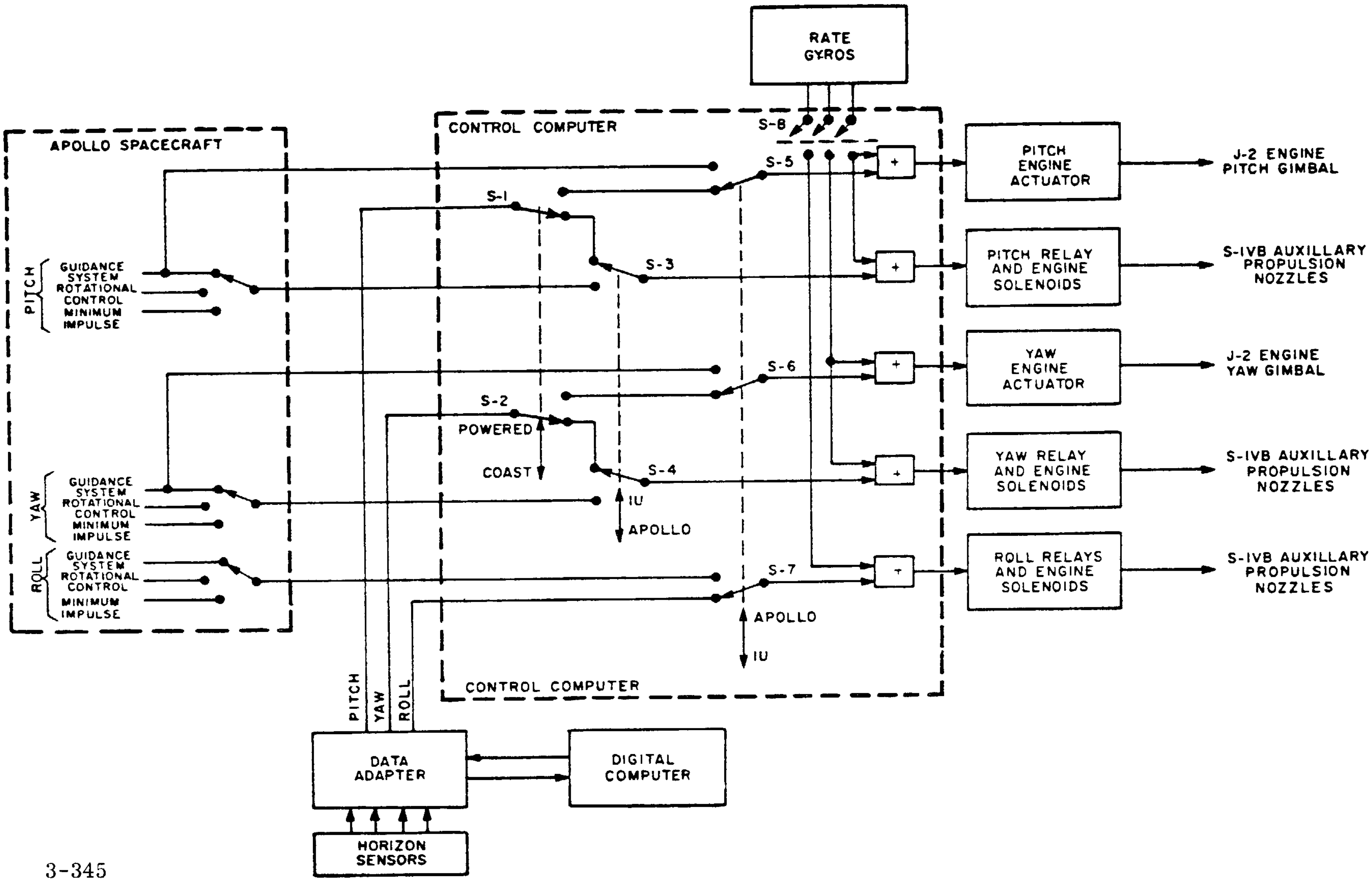

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Figure 20-17. S-IVB/IU Control Switching System pitch and yaw signals from the data adapter control the inputs to the control computer and result in gimbaling the S-IVB engine in a manner similar to S-II stage operation. Roll control cannot be provided by the single S-IVB main propulsion engine and must be accomplished by the stage auxiliary propulsion unit. Figure 20-16 shows the auxiliary propulsion nozzle configuration. Four of the six nozzles are used for roll control. The roll attitude signal from the data adapter goes to the roll channel in the auxiliary propulsion section of the control computer and results in operation of the proper pair of roll nozzles.

During the S-IVB orbital (coast) phase, switches S-1 and S-2 are in the coast position and all attitude control is performed by the auxiliary propulsion unit. All six nozzles are used; two for pitch and combinations of the other four for roll and yaw. Control of the vehicle can be provided by either the instrument unit or the Apollo spacecraft.

In the instrument unit mode of operation the control system has a limit cycle operation, which is intended to conserve propellants. This is accomplished by introducing an attitude error deadband of ± 1 degree, as well as by limiting maximum attitude rates to 0.3 degrees per second for yaw and pitch and 1.1 degrees per second for roll. Because of limit cycle operation, a certain amount of angular drift occurs. When this control system configuration is used, the actual attitude commands are derived either from the inertial platform or the horizon sensor. Each type of command is discussed in the following paragraphs.

20-38. Inertial Platform Control. The gimbal angles of the inertial platform are available to the digital computer in the same way as in the thrust vectoring control modes. Thus, the digital computer commands attitudes relative to the inertial platform orientation. This gives three-axis control which holds any space-fixed attitude. The rate gyros are used as inputs to the control computer for rate feedback in this mode. This configuration is as shown in Figure 20-17, with the switches S-1 and S-2 in the coast position. This is the basic mode of operation when attitude control, during coast, is provided totally by the Astrionics system. When the control is taken by the astronaut, the Astrionics system monitors the vehicle position. When control is released by the astronaut, the system is ready to hold the attitude fixed relative to the inertial reference, unless directed to the horizon sensor mode.

20-39. Horizon Sensor Control. Another mode of operation has been made available during coast phases of the S-IVB/IU stage. An automatic leveling loop, which keeps the vehicle longitudinal axis perpendicular to the radius vector from earth center to the vehicle, is obtained by using inputs from four horizon sensors. Angles measured by the horizon sensors are converted to digital signals by the data adapter for processing by the digital computer. Error signals are derived by the digital computer and transmitted to the control computer through the data adapter. The rate gyro inputs are also used by the control computer in this mode to enhance stability. In addition to the level orientation, the digital computer can command any fixed angle relative to the radius vector which does not exceed the limits of the horizon sensor scan angle. It should be noted that fixing the angle of the vehicle longitudinal axis, with respect to the radius vector to earth center, does not keep the vehicle from rotating about the radius vector. For three-axis stabilization, an additional reference about the yaw axis is obtained by using the inertial platform in a gyro compassing mode; however, for Apollo missions the inertial data from the platform is used to provide this third reference.

20-40. Apollo Spacecraft Control. In the Apollo spacecraft mode of control, the control system characteristics change. The attitude deadband of the control system can be selected for either ± 0.5 degree or ± 5 degrees, when command signals originate in the spacecraft guidance system. This change is effected by switches S-3, S-4, and S-7, Figure 20-7, being transferred to the spacecraft position in the control computer. Use of two other spacecraft signal sources, rotational command control and minimum impulse, causes the control system attitude deadband to become 0.1 degree. When rotational control inputs are used, motion about only one axis is commanded at any given time. This is rate control type command wherein the maximum rates are 0.3 degree in pitch and yaw and 1.1 degrees in roll. During this coast phase mode the rate gyros of the instrument unit are switched off (S-8) to prevent interference with minimum impulse operations during navigational sightings.

In the rotational command control mode, the astronaut controls the vehicle's attitude rate by positioning a hand control, which produces a rate proportional voltage. This signal turns on the S-IVB/IU stage attitude control nozzles through the control computer. The nozzles are turned off when the proper attitude rate is obtained, through signals fed back by the rate gyros to the control computer. During the time the hand control is commanding the control system, the spacecraft attitude reference system can follow the present vehicle attitude. This "attitude-follow" capability is instrumented by driving the command display unit servo motor with an error signal 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 commanded and actual gimbal angles are then differenced. This error signal is resolved into vehicle coordinates, and given to the control computer as an attitude error signal. The control system stabilizes and limit-cycles about this command attitude. Figure 20-18 shows single-axis information flow, when in this rotational command control mode.

The minimum impulse mode is an attitude position control scheme for introducing small changes in vehicle attitude. To effect an attitude correction, the astronaut manually introduces pulses, in the desired number, into the control system.

During S-IVB second burn either the spacecraft or the launch vehicle guidance system can command the attitude control system. The mode change is accomplished by switches S-5, S-6, and S-7, and occurs when a spacecraft mode control command is received by the data adapter. Operation of the control system is identical to operation during S-IVB first burn.

After injection into translunar trajectory (S-IVB second cutoff) the attitude control system is used to stabilize the S-IVB/IU/LEM while the CSM separates, turns around and docks. The operation of the control system during this period is similar to that during earth orbit.

(The attitude control and stabilization function shares hardware systems with the guidance function. Refer to Paragraph 20-44 for a description of the joint implementation.)

20-41. GUIDANCE.

The Saturn V guidance function generates and applies steering commands to correct the motion of the launch vehicle toward a path that produces success in its assigned mission.

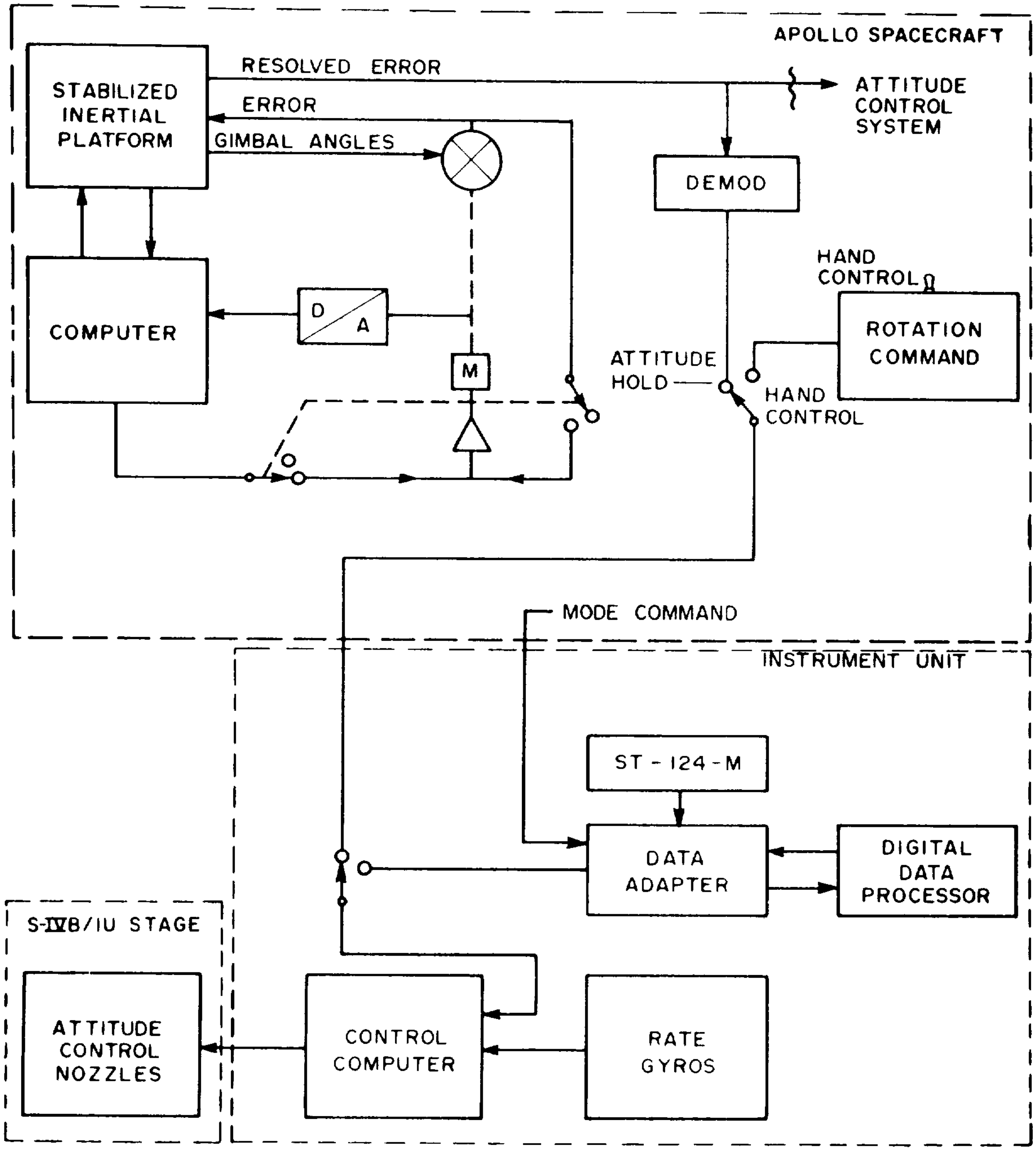

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Figure 20-18. Rotational Command Control Mode

20-42. REQUIREMENTS.

The function, active during the ascent, translunar injection and translunar flight phases, steers the vehicle in the pitch and azimuth planes and generates engine cutoff signals when the vehicle attains the proper velocity in relation to its position in space.

During S-IC stage flight, active guidance is not utilized due to the structural and related control constraints imposed on the launch vehicle while it is passing through the region of high aerodynamic pressure. This stage of the flight is accomplished utilizing an autopilot type attitude control which results in the exact position of the vehicle at staging time being unknown prior to the event. The guidance function must recognize this staging location and after S-II stage actuation must steer the vehicle along an optimum trajectory to accomplish insertion into the parking orbit.

For the ascent phase, the Saturn vehicle has an engine out capability for S-IC and S-II stages. An engine out situation causes the vehicle thrust and fuel mass flow rates to vary over a considerable range. Though these perturbations exists, the guidance function steers the vehicle, during S-II and S-IVB stage flight, along a constantly optimized trajectory. The optimization factor in the Saturn V is minimum fuel consumption or the shortest powered flight time.

The launch azimuth for an Apollo mission is 90 $\pm$18 degrees, however, the orientation of the orbital plane has not been defined at this time. This plane is a single earth orbital plane or a variable-inclination earth orbital plane with an orientation constantly changing as a function of time and is determined by the location of the center of the earth, the center of the moon and the location of the launch site. (Figure 20-19.) Regardless of the orientation of this plane at launch time, the guidance function is capable of steering the vehicle into the proper earth orbit. This is accomplished by selecting an optimum azimuth for vehicle ascent and direct orbit insertion or by performing a yaw maneuver to place the vehicle into the orbital plane.

After the orbital plane is defined, a volume of trajectories are calculated to insert the vehicle into the orbital plane during the launch window. A launch window exists, because the precise time of launch cannot be predicted. It is estimated that the length of this launch window will be approximately 3 hours. The magnitude of the volume of trajectories is determined by the desired degree of probability that it contains the vehicle's actual inflight trajectory. Each trajectory is optimized using calculus of variations techniques and represents an optimum solution to the guidance problem as defined by the mission criteria, trajectory boundary conditions and particular trajectory variations.

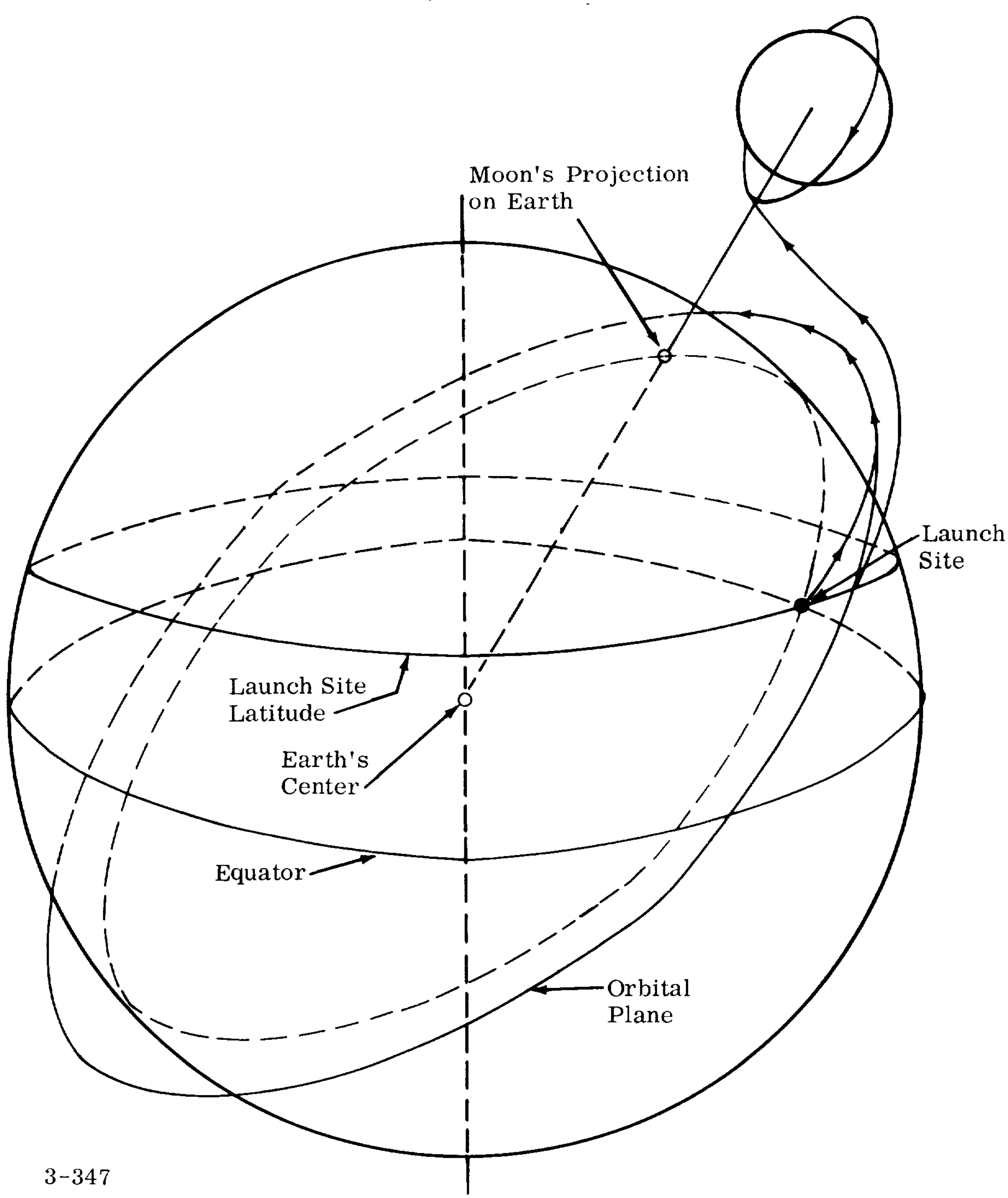


Figure 20-19. Variable-Inclination Earth-Orbital Plane

The ground operational support system (GOSS) is used for tracking and performing the Apollo mission communications. Due to the fixed location of some GOSS stations in the network, a constraint is imposed on the trajectory of the Saturn launch vehicle.

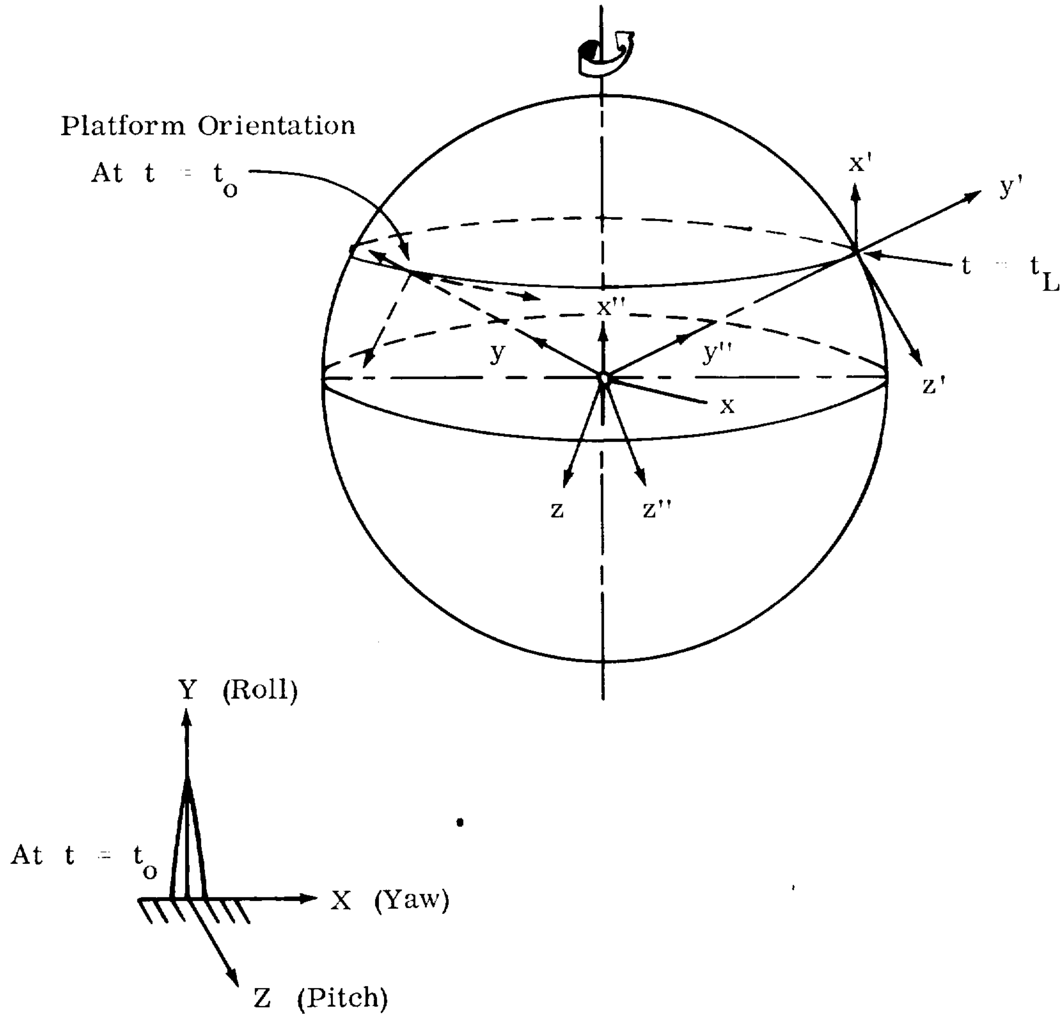
During earth orbit, the guidance function performs a series of time-to-fire computations to accomplish re-ignition of the S-IVB stage engine. These computations are also used to synchronize the S-IVB venting cycles to prevent venting during the reignition and translunar injection phase. The synchronizing process forces a vent cycle to occur just prior to start of the reignition sequence. The re-ignition sequence starts at approximately 575 seconds before the thrust buildup of the J-2 engine and consist of a period equal to one vent cycle.

The guidance system has a minimum mission operational lifetime of 6.5 hours. A maximum duration of orbital flight for the combination S-IVB stage/instrument unit and Apollo is 4.5 hours or approximately three orbits. The elapsed time from insertion of the vehicle into a translunar trajectory and separation of the S-IVB stage/instrument unit from the Apollo spacecraft is approximately one hour. An additional guidance system lifetime of one hour is provided.

20-43. OPERATION.

The Saturn V guidance function utilizes three coordinate systems; the measuring coordinate, reference coordinate and the earth centered coordinate system (Figure 20-20). The measuring coordinate system (X') is defined as that coordinate system in which the stage platform (guidance sensor) outputs are measured. Its origin is at the launch site and it is inertially fixed at approximately 10 minutes before launch. The reference coordinate system (X) is that coordinate system whose axes are oriented parallel to those of the measuring coordinate system at the beginning of the launch window, t_o. Its origin is at the center of the earth. This coordinate system is inertially-fixed and will be defined by the mission itself, i.e., by the location of the moon (both with respect to the earth and to the spacecraft) at the time of the moon's encounter with the spacecraft. The earth-centered coordinate system (X'') is that system which has its origin at the earth's center and whose axes always remain parallel to the axes of the measuring coordinate system.

The guidance equations are expressed as steering polynomials (X's), time of engine cutoff polynomials (t_c's), and engine re-ignition polynomials (t_i's) as functions of time (t), vehicle velocity (v), displacement (r) and a performance parameter (F/M). X_z, X_x, and X_y are the three eulerian angles taken successively about the three body fixed coordinate axes (Z, X, Y). Thus, they define the vehicle orientation in the reference coordinate system. A typical steering polynomial (desired direction



X - Y : Reference Flight Plane

X : Reference coordinate system (X, Y, Z) : inertially-fixed, earth-centered, parallel to platform orientation at $t = t_o$.

X' : Measuring coordinate system (X', Y', Z') : Platform orientation--inertially-fixed $[(\xi, \eta, \zeta)]$ at $t = t_A \approx t_L - 10$ minutes

X'' : Earth-centered coordinate system (X'', Y'', Z'') : inertially-fixed, parallel to X'

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Figure 20-20. Coordinate Systems

of the thrust vector) is of the form:

$$\chi = a_0 + a_1 X + a_2 Y + a_3 Z + a_4 \dot{X} + a_5 \dot{Y} + a_6 \dot{Z} \; 2 \; a_7 t + a_8 (F/m) + a_9 X^2 +$$

$$a_{10} Y^2 + \ldots + a_{16} (F/m)^2 + a_{17} XY + a_{18} XZ + a_{19} X\dot{X} + a_{20} X\dot{Y} + \ldots + a_{42} \dot{Z} t$$

$$+ a_{43} \dot{Z} (F/m) + a_{44} t (F/m) + \text{selected 3rd order terms,}$$

where the a_i's are stored constants and probably will differ for each stage. While this form of polynomial will surely be used for the pitch steering (χ_p), the possibility of using a simpler form (such as delta-minimum) of yaw steering (χ_y) is being studied. The state variables are in the reference coordinate system, where R becomes X, Y, and Z, and V becomes $\dot{X}$, $\dot{Y}$, and $\dot{Z}$. F/m is determined in the computer from the accelerometer outputs ($\delta\xi, \delta\eta$, and $\delta\zeta$), where the ξ, η, and ζ directions are parallel to the X', Y', and Z' directions, respectively.

$$F/m = \sqrt{(d\xi/dt)^2 + (d\eta/dt)^2 + (d\zeta/dt)^2}$$

The time of cutoff polynomials (for each cutoff of the S-IVB stage) and the time of reignition of the S-IVB stage (to initiate injection into the translunar trajectory from earth-orbit) are of similar form to the steering polynomial, depending on the same variables.

The guidance function is implemented with a stabilized platform, inertial data box, data adapter and digital computer, Figure 20-21.

Vehicle position and velocity determination in all three stages are accomplished as follows: The accelerometers, located on the inertial platform, supply signals representing incremental vehicle thrust velocities. These analog signals are converted to binary numbers in the data adapter. Approximately once per second, the velocities in the data adapter are sampled by the onboard digital computer. Each velocity sample is accompanied by a sample from the clock, also located in the data adapter. Successive values of velocities are differenced by the digital computer to obtain incremental velocities and times. The incremental velocities are then accumulated and transformed into the reference coordinate system. The inertial velocities are obtained by correcting for gravitational effects. A subsequent integration gives the inertial position coordinate, Figure 20-22.

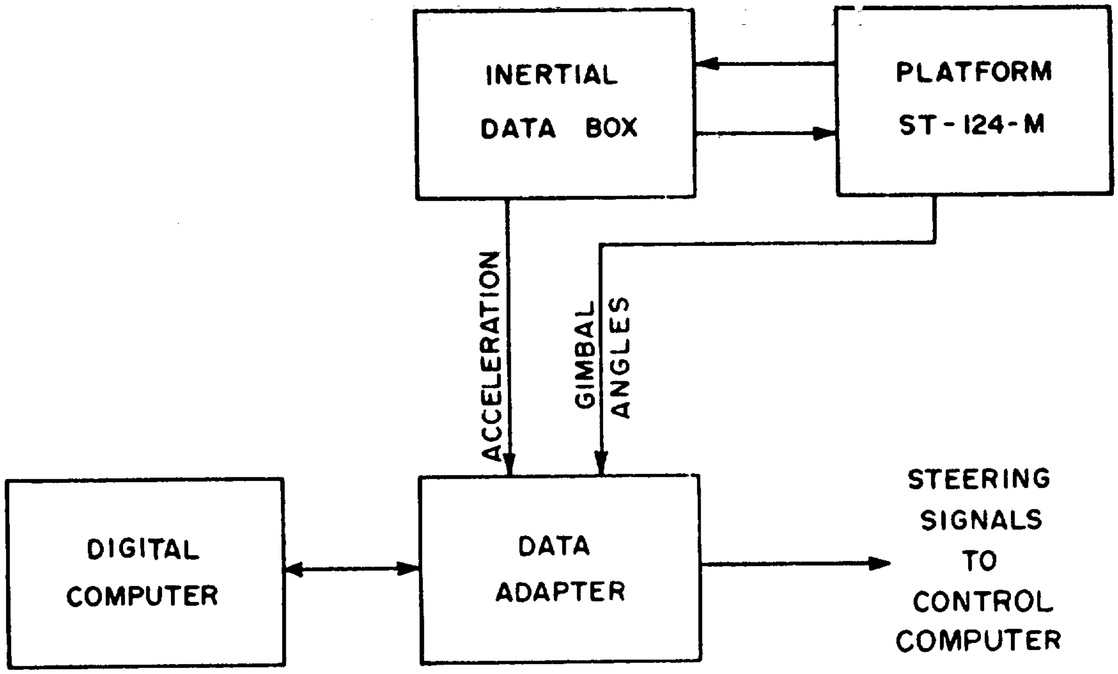
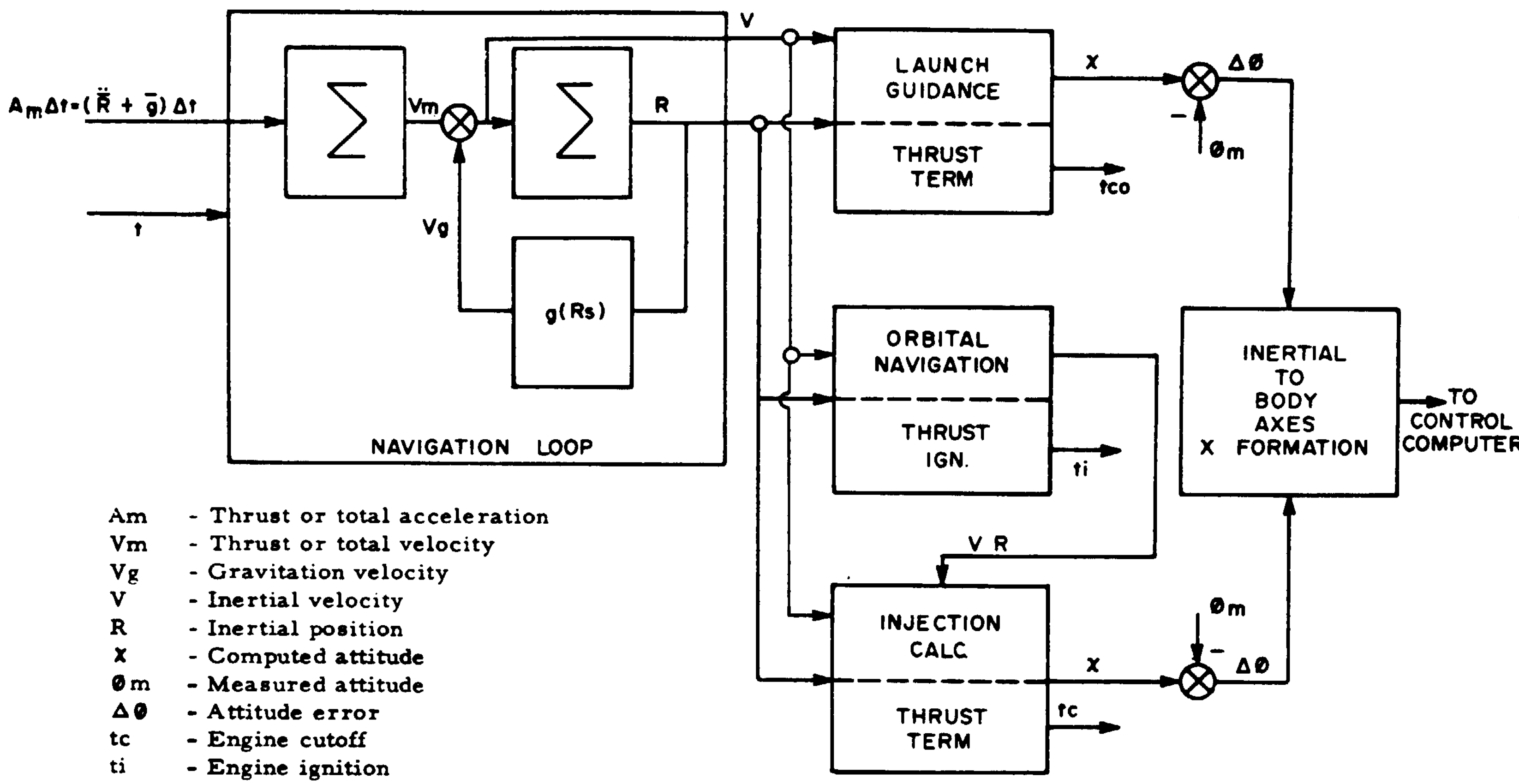


Figure 20-21. Guidance System Steering Signal Generation

Desired vehicle attitude (X) is calculated from the guidance equations and is compared with measured vehicle attitude (ϕ) from the stable platform. The angular errors are transformed into vehicle-body axes and sent to the control computer as analog steering signals. Engine cutoff and reignition times are computed in the same manner.

The approach to the generation of steering equations which operate according to the adaptive principle has also been studied with the intent to produce "closed form" steering expressions. The result of this study has been an "iterative" guidance mode which is also based on the calculus of variations. The resulting steering equations are functions of the same variables and utilized in the overall system in the same manner as those previously described. The method used will be determined when all of the variations and computational requirements have been considered. Both approaches result in about the same impact on the hardware involved.

An alternate hardware steering scheme using a resolver chain is included in the launch vehicle hardware. This method is similar to the approach used on the Saturn I vehicles and has been a development "backup" to the scheme previously



Am	- Thrust or total acceleration
Vm	- Thrust or total velocity
Vg	- Gravitation velocity
V	- Inertial velocity
R	- Inertial position
X	- Computed attitude
Øm	- Measured attitude
ΔØ	- Attitude error
tc	- Engine cutoff
ti	- Engine ignition

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Figure 20-22. Guidance Computer Data Flow described. The alternate method operates as follows: The digital computer determines the desired inertial position of the vehicle by solving the guidance equations. The inertial position is defined by three angles related to the inertial measuring axes. These three angles (X_x, X_y, X_z) are outputs from the data adapter to the inertial data box. The inertial data box contains three command servo modules which convert the computer angle outputs into three analog shaft positions representing the three angles. Each command module has a resolver attached to it which is positioned to represent X_x, X_y, and X_z respectively. These three resolvers are excited by 1.8 KHz and 1.5 KHz signals and are connected in a chain with three similar resolvers in the ST-124-M platform. The three resolvers in the platform measure the actual vehicle attitude relative to the inertial measuring platform. The chain of resolvers (three in the inertial data box and three or four in the platform) establish the required vehicle attitude, compare it with actual vehicle attitude and transform the resultant differences into the vehicle frame of reference such that roll, pitch, and yaw difference signals are defined. These difference signals are sent to the inertial data box for demodulation to dc voltages. The resultant dc (roll, pitch and yaw) voltages are sent to the control computer to steer the vehicle. The alternate steering method is shown in Figure 20-23.

The primary steering method has certain operational advantages over the alternate method. It allows constant monitoring of vehicle attitude versus the inertial reference and reduces hardware requirements. If at any time the vehicle fails to respond in the manner directed by the outputs to the control computer, this failure is sensed by gimbal angle monitoring and appropriate action is taken. In the alternate method, the control is essentially an open loop. (Refer to Figure 20-24.) That is, the command angles are transmitted to the inertial data box and there is no way to verify actual vehicle reaction to the commands except through the relatively slow response accelerometer input. This advantage of the primary scheme over the alternate scheme has significant impact on total system operational techniques.

During component checkout of the platform system and during simulated flight tests involving the entire control system, the alternate loop through the platform system is utilized with the ground equipment. This arrangement is used to position the platform and manipulate it in a manner which simulates the changes in its configuration during flight. This allows the active flight system to operate throughout the

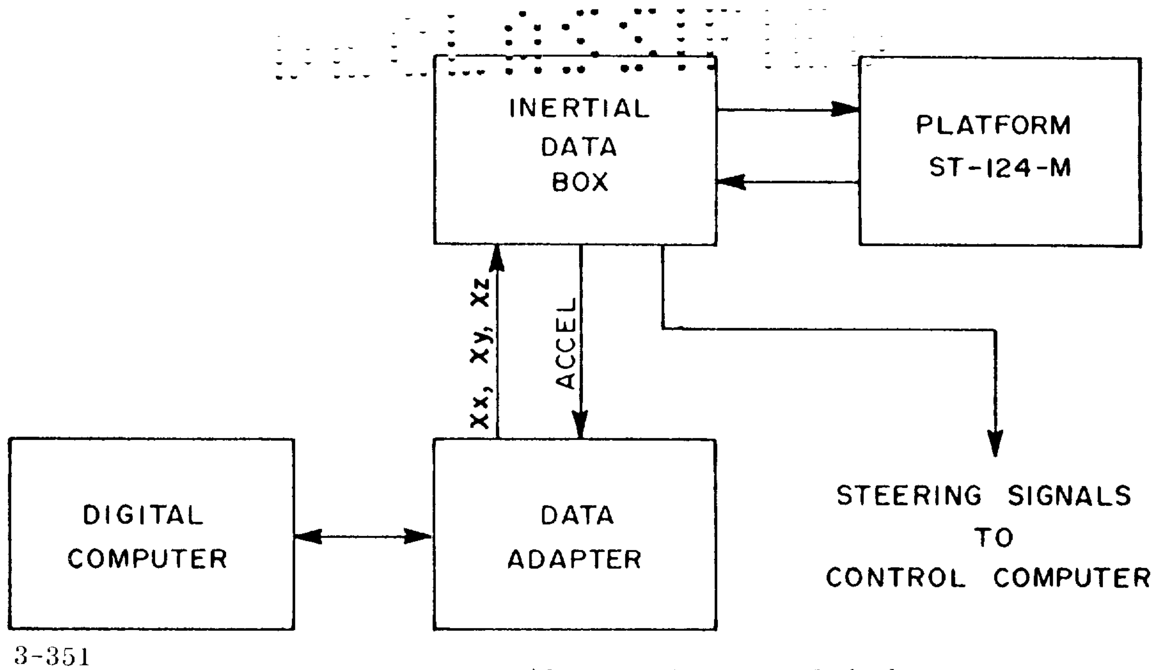


Figure 20-23. Alternate Steering Method

test without saturation of the steering error signals.

Before the actual launch can occur, certain guidance operations must be performed at the launch site. Prior to the time that the launch window begins, the stable platform is erected and torqued so that it is aligned with the desired coordinate system. The platform is rotated continually in azimuth, during the launch window, so that the vehicle (which rolls to align itself to this azimuth after launch) is oriented within the orbital plane. Computation of the launch azimuth is made by the ground computer. During the launch window, the platform will have been rotated so that, after the vehicle performs its initial roll maneuver (to achieve the desired launch azimuth) the vehicle's motion will lie in that direction which coincides with the desired platform orientation.

Approximately one minute (or less) before expected time of launch, the platform is released and becomes inertially-fixed. The real-time clock in the vehicle digital computer is also released at this time.

During the burning of the S-IC stage, the vehicle is rolled to the correct azimuth (so that it coincides with the platform orientation in azimuth). It is then pitched by a time-tilt program, X_p (t_L), without an attendant yaw maneuver. During S-IC stage flight the guidance system continuously calculates vehicle velocity and position

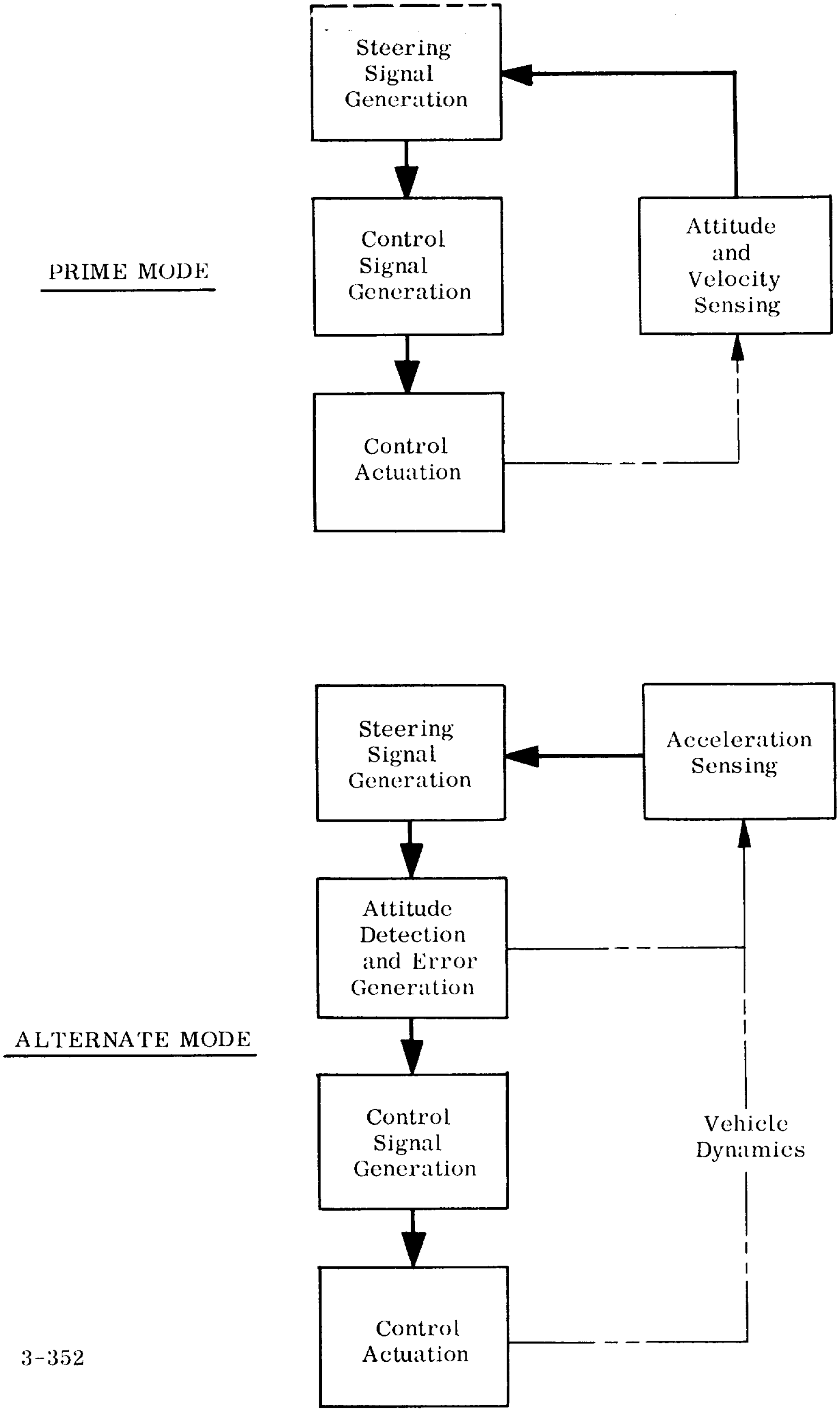


Figure 20-24. Saturn V Guidance Modes

by integration of the three-axis accelerometer outputs of the stable platform, Figure 20-22. This velocity and position information provides reference information for the path adaptive guidance used during S-II and S-IVB flight.

During S-II stage burn and first S-IVB stage burn, the steering polynomials (X_p and X_y) are calculated continually. The time of first S-IVB stage cutoff (t_c) will be calculated during the latter portion of the third stage of flight. At this cutoff time, the vehicle will have the proper velocity and altitude for injection into a circular earth orbit.

During earth orbit, the position and velocity determinations are based upon orbit insertion conditions and the equations of motion. It is necessary to readjust the solution to account for perturbations introduced by vending cycles during orbit. In order to do this, accelerometer monitoring is used to measure velocity changes. The digital computer is capable of forcing a vent cycle to prevent interference with a possible injection opportunity. Changes noted in velocity due to venting are also telemetered to ground stations to aid in ground orbital determination. Likewise, velocity and displacement may be updated via the command link to eliminate effects of injection errors. Re-ignition times for the S-IVB/IU are computed for each orbit and the countdown sequence is initiated at the appropriate time. The injection guidance equations are designed to place the S-IVB/IU and spacecraft into a free-return trajectory which meets spacecraft "aimpoint" requirements. It is necessary for the guidance equations to have the capability of performing injection guidance on any one of the three orbits.

As the S-IVB stage re-ignites, injection into the translunar trajectory begins. During this phase, X_p, X_y, and t_{c2} (second S-IVB cutoff) must be calculated. Attitude stabilization of the combination S-IVB stage/instrument unit/lunar excursion module is provided while the combination command module/service module separates, turns around, and docks with LEM. After this docking operation has been completed, the S-IVB stage/instrument unit combination is disengaged from the LEM, and the S-IVB stage auxiliary propulsion system is used to propel it to a different trajectory.

20-44. IMPLEMENTATION.

The guidance, and the attitude control and stabilization functions are jointly implemented in the launch vehicle as the guidance and control system. This hardware system is comprised of the data adapter, digital computer, ST-124-M stabilized platform system, control computer, control sensors (rate gyros and control accelerometers) and horizon sensor. These units are described in the following paragraphs.

20-45. DATA ADAPTER.

The data adapter is the input-output unit that accompanies the Saturn V digital computer. Its function can be broken down into three main categories:

a. Control data flow; such as the storage of telemetry data from the computer and data adapter in the buffer registers; the temporary storage of telemetry scanner addresses during orbital checkout; and the transmission of guidance data from the computer to the analog control computer.

b. Transform data into compatible format; such as digital-to-analog, analog-to-digital, and signal level conversions; the formation of 40-bit launch computer and telemetry words from 26-bit computer words; and buffering of communications between the computer and the ground-based launch computer to reconcile the difference in clock rates.

c. Perform simple computational operations; such as keeping track of real time, and decoding of operand addresses in process input-output operations.

Communication with the computer is carried out through 512-kilobit-per-second serial transmission. The process input-output instruction permits the specification of either input or output operations, and addresses the device to be affected. A single 26-bit word is transferred to the computer accumulator or from the accumulator or memory.

The data adapter employs unit logic device circuit modules and multilayer interconnection boards for circuit interconnections. Where low-power logic circuits are used, leadless semiconductors are mounted on unit logic devices. For those applications where high power dissipation is required, where precision components are needed, or where leadless devices are not available, standard discrete components packaged in encapsulated modules are used. This applies particularly in the case of power supplies, ladder networks, and cross-over detectors.

A complete listing of data adapter characteristics is presented in Table 20-7.

Table 20-7. Data Adapter Data

Item	Data
Computer Input-Output Rate	512-KHz serial
Power Supplies	6 pairs of duplexed supplies
Switch Selector	8-bit switch-selector input 15-bit switch-selector output
Discretes	13 discrete outputs 32 discrete inputs
Buffer Register Tag Register Mode Register	26-bit 8-bit 6-bit Provides communication with the launch computer, telemetry transmitter, and the computer interface unit.
Digital-to-Analog Converter	8-bit plus sign, 2-msec operation 3 attitude commands, 2 spare outputs
Analog-to-Digital Converter	18 resolver inputs, equivalent of 16 bits from a 2-speed resolver
Platform	4, 2-speed gimbal angle resolver inputs
Horizon Scanner	4 single-speed resolver inputs
Spares	6 resolver inputs
Delay Lines	3, 4-channel delay lines for normal input-output operations 1, 4-channel delay line for telemetry operations
Telemetry	
Command Receiver	13 bits for input data, 3 bits for sync and mode, 2 bits for priority interrupt
Data Transmitter	38 data and identification bits plus validity bit and parity bit
DDAS Computer Interface Unit	15 bits address plus validity bit for output data, 10 bits for input data
Launch Computer	39 data and identification bits plus validity bit for output data, 14 bits for input data plus interrupt
Reliability	0.99 probability of success for 250 hrs;

20-46. Angle Measurement. The data adapter is capable of measuring shaft angles in digital form to an accuracy of approximately one part in two thousand. This requires an analog-to-digital conversion of 11 bits. To accomplish measurements and conversion of the shaft angles, a time-duration measurement is used. This is accomplished by connecting a resolver in such a manner that two signals are generated to switch a high-frequency counter on and off. The resultant count, after a measurement cycle, represents the angle in binary form.

The basic principle of operation can be seen by considering the sum of the constant amplitude and frequency sine wave modulated by the cosine and sine respectively of the variable of interest. This is:

$$e_r = (E \sin \omega t) \cos \theta + (E \sin \omega t) \sin \theta.$$

The sum can be modified to a useful trigonometric form by shifting one of the waves of 90 deg. Thus

$$e_r = (E \sin \omega t) \sin \theta + E \sin (\omega t + \frac{\pi}{2}) \cos \theta$$

where, by a standard identity,

$$e_r = E \cos (\omega t - \theta).$$

These operations are carried out by the resolver and its associated circuitry by feeding an excitation signal to the resolver as shown in Figure 20-25. The input sinusoid is multiplied by the sine and cosine of the angular rotation of the rotor with respect to the stator of the resolver. A phase shifting network connected to the two rotor windings causes their outputs to differ in phase by 90 degrees.

Addition takes place within the network and the resultant output is a sinusoid, shifted by an amount proportional to θ plus a constant shift which can be calibrated out. Thus, the ratio of the amount of phase shift due to the rotation of the resolver relative to 2π gives a direct measure of the angles.

Assuming a 2.048 MHz clock for the counter and a 1016 Hz reference supply, the resolution of a single phase-shifted input is

$$\frac{1016 \times 360}{2.048 \times 10^6} = 0.178 \text{ deg/binary bit.}$$

System requirements dictate angle measurement accuracies of one minute of arc. This is achieved by using two-speed resolvers with a coarse-to-fine ratio of 32:1. Coarse and fine inputs are each measured to a resolution of 11 bits. The combined resolution is 16 bits.

The sine wave inputs are accepted from the resolvers by 38 cross-over detectors. Each cross-over detector (COD) has an output to each of two multiplexers, which in turn select the cross-over detectors to start and stop the two 11-bit counters. Selection of the cross-over detector (and ultimately the resolver) is under the control of a process input-output instruction.

When two RC networks are placed across the outputs of a resolver as shown in Figure 20-25, it is shown that cross-over detectors detect a phase shift which is twice that for a configuration with a single RC network. This fact is used in the data adapter to effectively double the "speed" of each resolver. In the case of the coarse gimbal angles, the data adapter normally reads the angles in the manner illustrated in Figure 20-25, i.e., as resolvers having 1 degree of electrical phase shift for every degree of shaft rotation. But if the program should detect a failure in the fine resolver, it can turn on a bit in the internal control discrete register which replaces the inputs of the fine resolver with the inputs derived from the appropriate coarse resolver. Thus, in this way the coarse resolver is used to back up the fine resolver. The counter then for the coarse gimbal angles has a resolution of 0.0893 degrees per binary bit.

The fine resolvers are always used with the double RC network. The 32:1 resolvers for the gimbal angles therefore have a resolution of 0.00279 degrees/bit. The single speed resolvers for the horizon sensors have a ratio of 4:1, but including the effect of the double RC network, the resolution provided by the counter is 0.0446 degrees per bit.

Execution of the process input-output instructions to read any angle through the cross-over detector-counter hardware involves two important steps. A single instruction first transfers the contents of both counters to the accumulator of

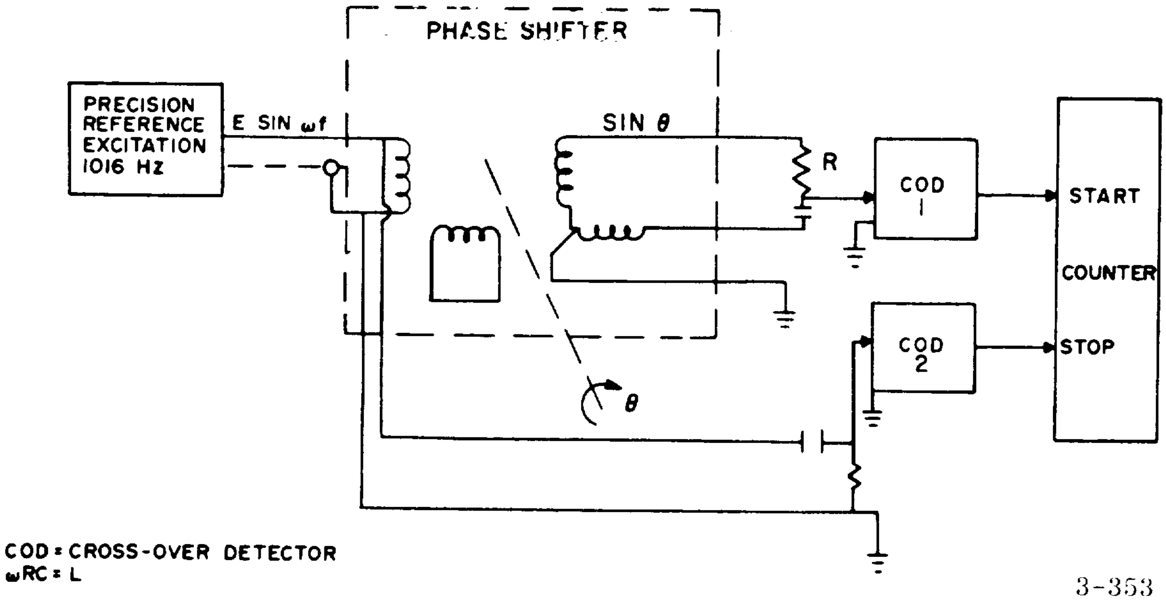


Figure 20-25. Angle Digitizer

the computer and then selects the pair of cross-over detectors to start and stop the counters for the next reading. The program must wait a minimum of 2 milliseconds before issuing another process input-output to the cross-over detectors in order to ensure that the counter has completed its cycle for the last pair of cross-over detectors. For the special case of a minor loop interrupt, the hardware always selects automatically the cross-over detectors for the fine resolvers of the yaw gimbal angles. Hence, 2 milliseconds after the interrupt, the first process input-output to the cross-over detectors always transfers a counter reading for the yaw fine resolver.

20-47. Digital to Analog Conversion. Attitude commands are analog signals which are generated from digital information by means of ladder circuits and sample-and-hold circuits. Digital information is simultaneously placed in three registers for redundancy purposes. One of the registers serves as a reference against which signals derived from the other two are compared. Conversion to analog signals is implemented by the ladder circuits and their use is time shared with each of the five output channels. Time sharing is made possible by means of the multiplexers, which direct the analog signal to the appropriate sample-and-hold circuit. This circuit in turn holds the signal by means of a capacitor for 40 milliseconds for its particular output channel. The signal to each output is renewed cyclically by reloading the ladder registers from the computer with the same or a new output word.

Comparator circuits monitor the output signals and compare them with the reference signal. Should the comparison show an out-of-tolerance condition, a signal is sent to the error monitor register. The computer can then change the ladder networks by means of a signal from the internal control discrete register to the channel selector.

20-48. Digital Data Monitoring. The PCM telemetry system used for monitoring digital data accepts 40-bit words at an asynchronous rate of 240 words per second. Special buffering and control logic is provided in the data adapter to temporarily store input and output data. A four channel delay line provides the buffering capability for data gathering. A sequence counter which is stepped by synchronizing pulses from telemetry every 4.17 milliseconds selects the channel to be read into the telemetry system. Serial information in the delay line channel is assembled in parallel form in the buffer, mode and tag registers, which along with the parity and validity bit generators, present 40-bits of information to the telemetry-computer interface unit.

The delay line buffer is loaded automatically by the computer as a by-product of normal process input-output operation to and from various data adapter registers if, and only if, there is an empty channel on the delay line. The information is lost if all channels are filled. Data adapter hardware automatically selects with each process input-output an empty channel, and during Phase "A" time, reads 5-bits of real time and 7-bits of tag address (word identification). For the case of cross-over detector count and ladder register, a single channel of the delay line contains both kinds of information. Phase "B" always contains ladder register information along with 3 bits of ladder identification; phase "C" always contains the cross-over detector counter reading. For this case one bit is reserved to indicate to the hardware when both phases are filled; in the situation where one of the phases is left empty, this bit is forced to a "full" condition by a programmed process input-output at the conclusion of the analog processing cycle.

For digital outputs, data are monitored at the interface of the data adapter to determine whether the correct signals were sent to external equipment. This applies to the switch selector register and the discrete output register. In addition to loading these registers, the computer must wait a suitable time interval for the signals to stabilize and issue a special process input-output to load the delay line storage with the output data at the interface; the input and output data are stored with separate tags and therefore in separate delay line channels. The internal control discrete register is monitored similar to the discrete output register except that a single process input-output loads, in a single channel of the delay line, both the input and output data of the register.

All discrete inputs are monitored by the delay line when the appropriate process input-output is given by the computer. Besides the discrete inputs this group also contains data from the accelerometers, telemetry, scanning, ground computer, command receiver, error monitor register, switch selector feedback and the interrupt register (for other than timed interrupts).

The computer can also load the buffer, mode and tag registers independently of the delay line. When this happens, any word in the delay line is prevented from entering the register until the word has been successfully accepted by either the telemetry or the ground computer. An internal control discrete is set by the program to inhibit advancing the sequence counter for the delay line or transferring data from any of the channels.

A special circuit monitors the constant-amplitude, phase-shifted input to the cross-over detectors. If, due to a malfunction, the signal level exceeds established limits in the positive or negative direction, the output of this circuit is a logical "1." There are presently 19 pairs of cross-over detectors that must be monitored by these circuits, and their outputs must be telemetered. This is done by one telemetry word. The best means of controlling the data output multiplexer to provide this function is to provide an additional process input-output address to serialize the parallel outputs of these circuits so they are read by the computer and stored in the data output multiplexer. As a program consideration, this is probably done once during the major loop, i.e., once or twice a second. There is no storage capability in the individual monitoring circuits, so if an intermittent malfunction occurs and clears between process input-output samples, it is not detected. Tables 20-8 and 20-9 present details on process input-output and tag bit coding.

Table 20-8. Definition of Use of Address Line Bits to the Data Adapter for Process Input-Output Operations

Group	A8	A2	A1	Function
1	See Below	0	0	Input to data adapter (computer telemetry operations)
2	See Below	1	0	Input to data adapter (load registers and delay lines)
3	See Below	0	1	Input to data adapter (computer telemetry operations)
4	0	1	1	Output from data adapter (register and delay line read)
5	1	1	1	Output from data adapter (COD CTR READ and set up new COD using address lines)

	Group 1	Group 2	Group 3	Group 4	Group 5
A3	Address	Address	Address	Address	Address
A4	Address	Address	Address	Address	Address
A5	Address	Address	Address	Address	Address
A6	Address	Address	Address	Address	Address
A7	Address	Address	Address	Address	Address
A8	See Below	See Below	See Below	0	1
A9	See Below	See Below	See Below	Not Used	Not Used

Group 1	Group 2	Group 3	Groups 4 and 5
A8 A9	A8 A9	A8 A9	Bit A8 is used to recognize COD group.
0 0 ACC	0 0 M MEM	0 0 ACC	
0 1 ACC	0 1 ACC	0 1 ACC	
1 0 M MEM	1 0 M MEM	1 0 M MEM	
1 1 R MEM	1 1 R MEM	1 1 RES MEM	
64 ACC	64 ACC	96 ACC	
32 RES MEM	32 RES MEM	32 RES MEM	
32 MAIN MEM	32 MAIN MEM	32 MAIN MEM	

Table 20-9. Definition of Tag Code to be Used with Telemetry

		Process Input-Output Group				
Tag Bit A9		1	2	3	4	5
	1	A8	Forced 0	A8	Forced 1	Forced 0
	2	A9	Forced 1	A9	Forced 0	Forced 0
	3	A3	A3	A3	A3	A3
	4	A4	A4	A4	A4	A4
	5	A5	A5	A5	A5	A5
	6	A6	A6	A6	A6	A6
	7	A7	A7	A7	A7	A7
	8	Forced 1 (TM)	Forced 0 (DOM)	Forced 1 (TM)	Forced 0 (DOM)	Forced 0 (DOM)
Real Time	1	Forced 0	Real Time 1	Forced 1	Real Time 1	Real Time 1
	2	Mode Reg	Real Time 2	Mode Reg	Real Time 2	Real Time 2
	3	Mode Reg	Real Time 3	Mode Reg	Real Time 3	Real Time 3
	4	Mode Reg	Real Time 4	Mode Reg	Real Time 4	Real Time 4
	5	Validity Bit	Real Time 5	Validity Bit	Real Time 5	Real Time 5

20-49. Analog Data Monitoring. Certain data in the data adapter is monitored by the analog input channels to the PCM telemetry system. These include the following.

a. Unfiltered 28-volt dc input to the data adapter

b. Filtered 28-volt dc output from the data adapter

c. 6 volts dc from power supplies 1 and 2

d. 12, 20, -3, and -20-volt dc

e. Attitude commands A, B and C

f. Spare ladder outputs A and B

g. Computer thermistor output

h. Data adapter thermistor outputs A and B

i. Resolver excitation

For a. through f. above, the signals are scaled down in most cases to be compatible with the full scale range of 0 to 5-volt dc for the telemetry system inputs. No interface circuits are required to connect the thermistor outputs into the telemetry system. Each thermistor provides two output lines to telemetry. In addition, all computer thermistor circuits are routed to telemetry through the data adapter.

A functional description of the data adapter assemblies and special circuit designs is presented in the following paragraphs.

20-50. Address Generator and Tag Register. The address generator and tag register decodes the computer instruction words. During input-output operations, the computer selects a register in the data adapter which contains or receives the input-output data. The address of the selected register and the correct data are determined by the operand bits of the instruction word along with the process input-output lines from the computer.

20-51. Switch Selector Register. The switch selector register controls the outputs of five switch selectors located within the stages of the vehicle. The register is loaded by the computer whenever the computer wishes to give commands to specific vehicle devices such as fuel valve controls. The register has a 15-bit storage capacity and is loaded by a process input-output instruction. The 15 bits are used as follows:

a. Eight bits make up a relay code which is distributed in parallel to each of the five switch selectors

b. Five bits determine which switch selector will be activated (No more than two selectors may be addressed at one time.)

c. One bit commands the assigned switch selector to activate the device selected by the relay code

d. One bit resets all switch selector relays which were turned on by the previously described bits.

20-52. Discrete Output Register. Certain functions within the vehicle, excluding those controlled by the switch selector register, are controlled by a 13-bit discrete output register. Discrete inputs are signals which do not require storage within the data adapter. The data adapter is designed to handle 32 of these inputs. Groups of these discrete inputs are treated as words by the computer - one 26-bit word and one 8-bit word. Each word is read by the computer as requested by a process input-output address. The computer reads the discretes periodically and performs the necessary program steps. Examples of discretes are:

a. A signal from the control distributor indicating vehicle stage separation;

b. A signal from the spacecraft indicating a command to start the thrust sequence of the S-IVB stage.

To ease programming requirements for changing specific discretes while not affecting others, the discrete output register is not loaded in the same way as the other registers. If certain discretes are to be activated, a process input-output is set up to address the "set" side of all latches in the register. Conversely, if certain discretes are to be deactivated, another process input-output selects the opposite or "reset" side of all latches in the register. The desired bits in the register are changed by placing "ones" in the corresponding bit locations of the data word transferred to the register from the computer, while the unchanged bit positions have "zeros" in the data word.

When the switch selectors are operated as previously described, relay tree feedback lines are tested to assure that the code was set properly by the data adapter. Eight lines from a separate set of contacts on the code relays contain the complement of the data word used to set the code relays. These lines are inputs to the data adapter which do not require storage and are addressed by the data adapter in the same manner as other discrete inputs. This feedback word is separated from the other discrete inputs so that the word may be processed more easily in the computer when comparing it with the word used to "set" the relay code.

20-53. Interrupt Register. As a means of notifying the computer that immediate attention be given to an external operation, an interrupt line is wired from the data adapter to the computer. The interrupt register (delay line) is capable of accepting 13 different signals and storing them until the computer has acted upon them. Presently, there are requirements for only eight interrupt signals. The signals are OR'ed together so that only one interrupt line to the computer is required. After an interrupt, the computer branches to a subroutine to read the interrupt register using a process input-output operation. A computer analysis is then made, testing the highest priority bit positions first in case more than one interrupt signal is stored in the register. During this testing, the computer stores the contents of the memory address register and the instruction counter and branches to an interrupt subroutine. While in this subroutine, the computer does not recognize further interrupts. The next to last instruction of the interrupt subroutine is a process input-output addressed to the interrupt register to reset the particular bit causing the interrupt. The hardware provides a time delay to prevent further immediate interrupts from the same source. The source must disappear and return before another interrupt is honored from that source. This prevents slow-acting devices such as relays from regenerating interrupts while they are being activated by discrete outputs which occur during the interrupt subroutine. Each interrupt signal must be at least 84 usec duration to assure storage in the delay line.

The computer is also capable of inhibiting the interrupt, as commanded by the program, with process input-output instructions whenever the function of the subroutine warrants this precaution. However, a few of the inputs bypass this inhibit control; these latter inputs are caused by functions which require the highest priority of attention. Examples of interrupts are:

a. An interrupt which is timed to ensure regular processing of guidance data

b. An interrupt from the DDAS interface unit indicating that requested data are available.

20-54. Buffer Register. The buffer register provides storage for a 26-bit word and is loaded by process input-output operations or the data output multiplexer for data adapter telemetry operations. It provides part of the interface required for transferring data to the telemetry transmitter and/or the LCC computer. It also stores addresses to be compared in the telemetry scanner address comparator during orbital or ground checkout. It provides parallel outputs to all of these external systems simultaneously. These systems read data from this register asynchronously with respect to computer timing.

20-55. Mode Register. The mode register is similar to the buffer register and other one-word registers loaded by the computer. It provides storage for a 5-bit computer word which defines the computer mode of operation. While communicating with the LCC computer, these five outputs are read in parallel by the launch computer. The telemetry data multiplexer reads three of these outputs when transmitting computer telemetry words, but real-time information data is substituted for these bits when data adapter data is transmitted.

20-56. Validity Bit Generator. Since the telemetry data multiplexer addresses the data adapter asynchronously with respect to computer timing, it is possible for telemetry words to be read while they are being changed by process input-output operations. However, data read at this time are invalid. Also, since the buffer register is used to store addresses of other telemetry system parameters during orbital checkout, these data are invalid as telemetry outputs from the data adapter.

Therefore, a signal must be included in the telemetry word which indicates the validity of the word. The validity bit generator performs this function. Data are invalid any time the computer mode register, tag register and buffer register are being loaded. It is also invalid when the buffer register contains orbital checkout address information.

20-57. Ready-Bit Generator. During orbital checkout, the computer examines various parameters which are monitored by the telemetry system. The computer obtains one of these inputs by sending a telemetry scanner address to the buffer register. This 15-bit address is compared in the telemetry scanner address comparator. When comparison occurs, the telemetry word is stored in a 10-bit register which is read by the data adapter. Another line interrupts the computer to notify it that data are available.

Since the buffer register is continuously connected to the telemetry scanner address comparator, as well as the telemetry data multiplexer and the launch computer interface, it is necessary to indicate to the address comparator when the buffer register data are ready for comparison. This is the function of the ready bit generator. The "ready" bit is turned on after the 15-bit address is loaded into the buffer register, under control of a special process input-output instruction. The bit remains on until after the address has been compared and the 10-bit data word has been stored; it is turned off by the line causing the computer interrupt.

20-58. Parity Generator. To ensure that computer data sent out over the RF telemetry link to ground equipment is received without error, a parity bit is included in each 40 bit data word send out by the data adapter.

The telemetry data word is formed from three subwords plus a validity bit. The validity bit however, is not included in the parity check. Odd parity is used. This means that, excluding the validity bit, all the "ones" in the three subwords, plus the parity bit, add up to an odd number. The easiest way to generate total parity is to generate an individual parity bit for each subword. The three parity bits are then checked for total parity and a resultant parity bit is generated.

20-59. Internal Control Discrete Register. Certain functions within the data adapter must be controlled by the computer. A 13-bit register, very similar to the discrete output register, is included to provide these controls. Some of the functions of these discretes are:

a. Control switching of duplex delay line channels
b. Selection of the duplex analog output channels to be used
c. Selection of coarse resolvers as backup of fine resolvers

20-60. Process Input-Output Digital Input Multiplexer and Serializer. Excluding the computer, all digital input words except accelerometer inputs occur in parallel form. Since the computer can read only one group of inputs (one word) at a time, the group of inputs selected by the process input-output request is switched to a single serializer which converts the parallel inputs to the 512-KHz serial bit rate. The output of this serializer is applied to the accumulator input data bus. The process input-output multiplexer provides the necessary switching for the input word selected by the computer.

Data from the LCC computer and the command receiver have the same address. If the LCC computer is connected to the system, a discrete input indicates this and provides a control gate to inhibit inputs from the command receiver while allowing inputs to come from the LCC computer. The converse of this is true if the LCC computer is not connected into the system.

20-61. Triple Modular Redundancy Delay Line. The use of glass delay lines in a triple modular redundancy configuration has effected significant component savings and resultant reliability improvements in the data adapter. They replace several latch registers that would otherwise be required for the functions being implemented.

This triple modular redundancy delay line has been organized around computer timing such that the information it contains remains synchronized with the computer operation cycle. The total circulation time of the delay line and its associated electronics is equal to the basic computer instruction cycle time of 82.03 microseconds (42 bit times). The delay line is divided into three 14-bit word times corresponding to the three computer phase times. Furthermore, the four clock times into which each computer bit time is divided is used to time-share the delay line among four channels of 512-KHz serial information. Hence, a total of twelve 14-bit words can be stored in a single delay line by operating the line at 2.048 MHz per second. The word locations relative to the four channels are presented in Table 20-10.

In performing a process input-output operation, the computer sends out or looks for information only during phase-times "B" and "C." Real time has been assigned to a phase "A" word time. This is done to facilitate the use of real-time information in the data output multiplexer. However, real time is made available to the computer during phase "B" via the multiplexer register and the serializer latch.

The velocity accumulations, which are the processed outputs of the accelerometer optisyns, are arranged in such a manner as to provide duplex redundancy, matching the duplexed optisyns, in the triple modular redundancy delay line. One line contains outputs X_1 and Y_2, another accumulates Y_1 and Z_2, while still another processes Z_1 and X_2. When the computer calls for a given velocity accumulation, it receives the processed output of one of the optisyns on the selected accelerometer in phase "B" and the output of other optisyns on the same accelerometer during phase "C." These two values are processed separately in the computer such that any one of the delay lines or any optisyn could fail without failing the system.

Table 20-10. Word Locations

Channel	Phase Times		
	Phase A	Phase B	Phase C
W Clock (Read)	Spare	Spare	Interrupt Storage
X Clock (Write)	Spare	Switch Selector Interrupt Countdown	Interrupt Limiting
Y Clock (Write)	Millisecond Countdown	Minor Loop Interrupt Countdown	Interrupt Inhibit
Z Clock (Read)	Real Time Accumulation	Velocity Accumulation X_1 (Y_1, Z_1)	Velocity Accumulation Y_2 (Z_2, X_2)

No initialization has been provided for this delay line. The real-time accumulation is voted upon in triple modular redundancy voters during every circulation, so the values in all three lines will always agree. The duplex operation of the accelerometer processors does not allow voting, so there is no guarantee that the absolute value of the two readings will agree. Real time is accumulated in 246.1 microsecond increments, while the least significant bit in the velocity measurement has a weight of 0.05 meters per second.

The delay channel, in which bits are written at Y time, is used to time three functions in the data adapter/computer system. In phase "A," a time delay of approximately one millisecond duration for use in the resolver frequency source is generated by counting 12 circulations of the delay line. In phase "B," time-to-go until the next computer interrupt for the minor loop function is counted down, while the bits for interrupt inhibit are stored during phase "C." These two count-downs occur at the rate of one count every 0.4922 millisecond, and they generate an interrupt when the count passes through zero. The length of the count is determined by the computer, which loads a value of time-to-go to indicate each count. Switch selector interrupts in phase "B" of channel "X" are handled similar to the minor loop interrupt.

Computer interrupts are stored in phase "C" of channel "W." Once the computer recognizes an interrupt, it sets the corresponding bit in phase "C" of channel "X" and resets this bit in channel "W." The associated circuitry prevents a new interrupt from being recognized in this bit position until the previous interrupt has disappeared. The only constraint, therefore, on the length of the interrupt signal is that it lasts for at least 82.03 microseconds. Should the computer wish to inhibit certain interrupts, it can do so by writing corresponding bits in phase "C" of channel "Y." The inhibit bits do not erase or prevent writing in the storage channel; when the computer erases the inhibit bits any corresponding bits that may exist in the storage channel become effective.

In Table 20-10, it can be seen that three spare words are left in channels "W" and "X." Channel "W" may be conveniently read by the computer, while channel "X" may be conveniently written into from the computer. As many as three of the normal 14 bits may be sacrificed if it is desired to use either of these two channels in the opposite manner.

20-62. Power Supplies. The power supplies which serve the data adapter and computer are contained in the data adapter. These power supplies, composed of modules, are duplexed for reliability; thus, each supply is capable of supplying the full current load for that voltage. Voltage sequencing is provided where required, and power supply lines can be switched to permit single channel computer operation.

The Saturn digital computer and data adapter require five dc supply voltages. To handle the large current requirements of one of these supplies (6-volt dc) with available high-quality components, this load is split and is furnished by two independent sources. The power supply subsystem consists of 12 power converter modules and 24 feedback amplifiers arranged to furnish six highly reliable power sources.

The power subsystem is isolated from the vehicle 28-volt fuel cell supply by a dc-to-dc static converter. The dc output voltages are determined by the circuit requirements of the data adapter and the computer. The power supplies contain relays which operate the computer on the ground permitting ground checkout of the redundant circuits to verify that all redundant functions are operating.

The efficiency of the complete dc power system is approximately 60 percent. The efficiency of a comparable dual series-regulator power supply is estimated to be about 30 percent. The better efficiency, resulting from use of pulse-width-modulated power supplies, is due primarily to the absence of any linear elements in series with the power source. A block diagram of the pulse-width-modulated power supply module is shown in Figure 20-26.

The timing oscillator provides an unregulated dc voltage for the driver stages to ensure power ground isolation. It also provides a square-wave output which determines the switching rate of the power inverter. Integrators in the predriver stage convert this square wave into a triangular drive signal whose average dc value is a function of the control input from the dc feedback amplifier. The biased triangular signal determines the degree of modulation. The shaped output from the driver stage is transformer-coupled to the power inverter to main ground isolation.

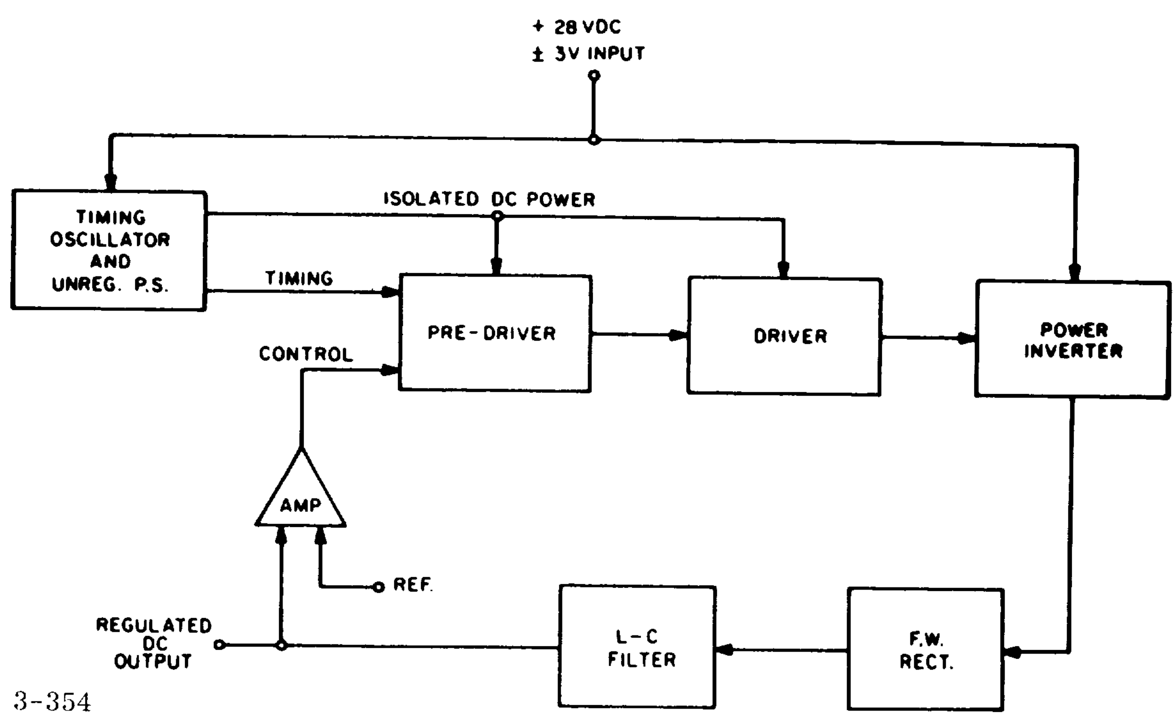


Figure 20-26. Pulse-Width-Modulated Power Supply Module Block Diagram

The push-pull power inverters switch the 28-volt dc source to the primary of the power transformer. The full-wave rectified output of the transformer constitutes a unipolar pulse train whose on-off ratio is proportional to the circuit losses and inversely proportional to the 28-volt dc line voltage variations. The single section LC filter smooths the modulated pulses into a low-ripple, regulated dc voltage. Any variation in the average value of the output voltage is sensed by the feedback amplifier, and the error signal is used to control the power inverter pulse width.

20-63. Special Circuit Design. Most of the digital circuits used in the data adapter are identical to those used in the computer. Some special circuits are needed to accommodate the interfaces to external equipment. Two special circuit designs are discussed in the following paragraphs.

A buffer circuit is used to convert the 28-volt digital input signals to 6-volt ground reference signals, compatible with the data adapter logic circuitry. Since an input noise of 4 volts is expected, an inverter with input noise rejection of at least 7 or 8 volts is used. Either component redundant or triple modulator redundancy techniques are used to obtain reliability.

The 1016 Hz frequency needed to drive the resolvers is obtained by counting down from computer timing pulses. This is accomplished with a three-stage ring counter followed by a latch. A variable clipper controls the amplitude of the 1016 Hz square wave obtained from the counter. The clipping level is set by level-sensing detector-amplifier circuitry. The fundamental component of the square wave is obtained by filtering, and is amplified to a 26-volt level which is adequate to drive the resolvers. The 26-volt level is maintained by an amplitude sensitive feedback circuit. The harmonic content is reduced by filtering. This filtering is accomplished by incorporating frequency selective feedback techniques in the amplifier circuitry. The resolver frequency source is duplexed in a sense, i.e., each source supplies power for half of the resolver inputs in such a manner that fine and coarse resolver excitation for any input parameter is not supplied by the same source. Since fine and coarse inputs serve as a backup for each other (under the proper program control), duplex redundancy is used for the excitation source.

20-64 DIGITAL COMPUTER.

The Saturn V digital computer is a serial machine using a random access magnetic core memory. It uses microminiature packaging techniques (developed under the Advanced Saturn Technology program), triple modular redundancy in the central computer, and multiple duplex memory modules for high reliability. Glass delay lines are used for the serial arithmetic registers and for the storage of the instruction counter. The characteristics of the computer are summarized in Table 20-11.

The computer provides general purpose computing capability characterized by high

Table 20-11. Saturn V Computer Data

Item	Data
Type of Computer	Stored program, general purpose, serial point, binary
Clock rate	512 kilobits per second, 2048 MHz clock
Speed	Add-subtract and multiply-divide simultaneously:
Add Time, Accuracy Multiply Time, Accuracy Divide Time, Accuracy	82 usec, 26 bits 328 usec, 24 bits 656 usec, 24 bits
Storage Capacity (4 memory modules simplex, or two pair duplex)	16,384 words (each 26 bits) plus two parity bits expandable in 4096-word modules to 32,768 words total (simplex). The memory modules may be used in simplex or duplex operation. Memory can be divided between program and data as desired, typically: 2000 data words (25 bits and sign) 28,768 instructions (each 13 bits)
Input-Output	External - computer programmed input-output control
Component Count (including 4 memory modules)	40,800 silicon semiconductors and cermet resistors; 458,752 ferrite cores
Temperature	60° F inlet coolant temperature, 100° C maximum junction temperature allowable
Reliability	0.990 probability of success for 250-hour mission using triple modular redundacy logic and multiple duples memory modules.
Packaging	78 electronic page assemblies, four 4096-word (28 plane) memory assemblies. Integral liquid cooling.

internal computing speed and variable capacity random access core memory. The internal arithmetic structure employs both adder and multiplier units which may operate concurrently with a single program control unit.

Memory words are 28 bits in length, (including two parity bits). The memory is arranged so that one data word or two instructions may occupy one 28-bit memory word. The memory uses fourteen 64 by 128 (4096 words) magnetic core planes plus the required drive and sensing circuits. From one to eight memory modules may be used in the computer, providing flexibility in memory size for different Saturn missions. Independent memory modules may be used in duplex fashion for high reliability on long missions. This report assumes the use of 32,768 instruction words, or four modules.

The triple modular redundancy system uses three identical simplex computer logic channels and subdivides each channel into seven functional modules. The outputs from each channel are voted upon in voter circuits before the signal is sent to another module. The output of the voter circuit is equal to the majority of the inputs to the circuit. Thus, even if one of the three inputs is incorrect, the output to the next module will be correct. Figure 20-27 is an example of triple modular redundancy voter signal outputs. An average of 13 output signals from each module are voted on. The voter circuit outputs may go to any of the other subdivided modules of the computer.

The computer data flow is illustrated in Figure 20-28. This simplified block diagram depicts the major data flow paths and associated register level logic. The timing logic and input-output section are not shown, but, are described in this section under the instruction sequencing and computer input-output capability portions.

The computer is a serial, fixed point, stored program, general purpose machine which processes data using two's complement arithmetic. Two's complement arithmetic obviates the recomplementation cycle required when using sign plus magnitude arithmetic. Special algorithms have been developed and implemented for multiplication and division of two's complement numbers. Multiplication is done 4 bits at a time and division 2 bits at a time. These algorithms are treated separately in the arithmetic portion of this section.

A random access magnetic core memory is used as the computer storage unit. A

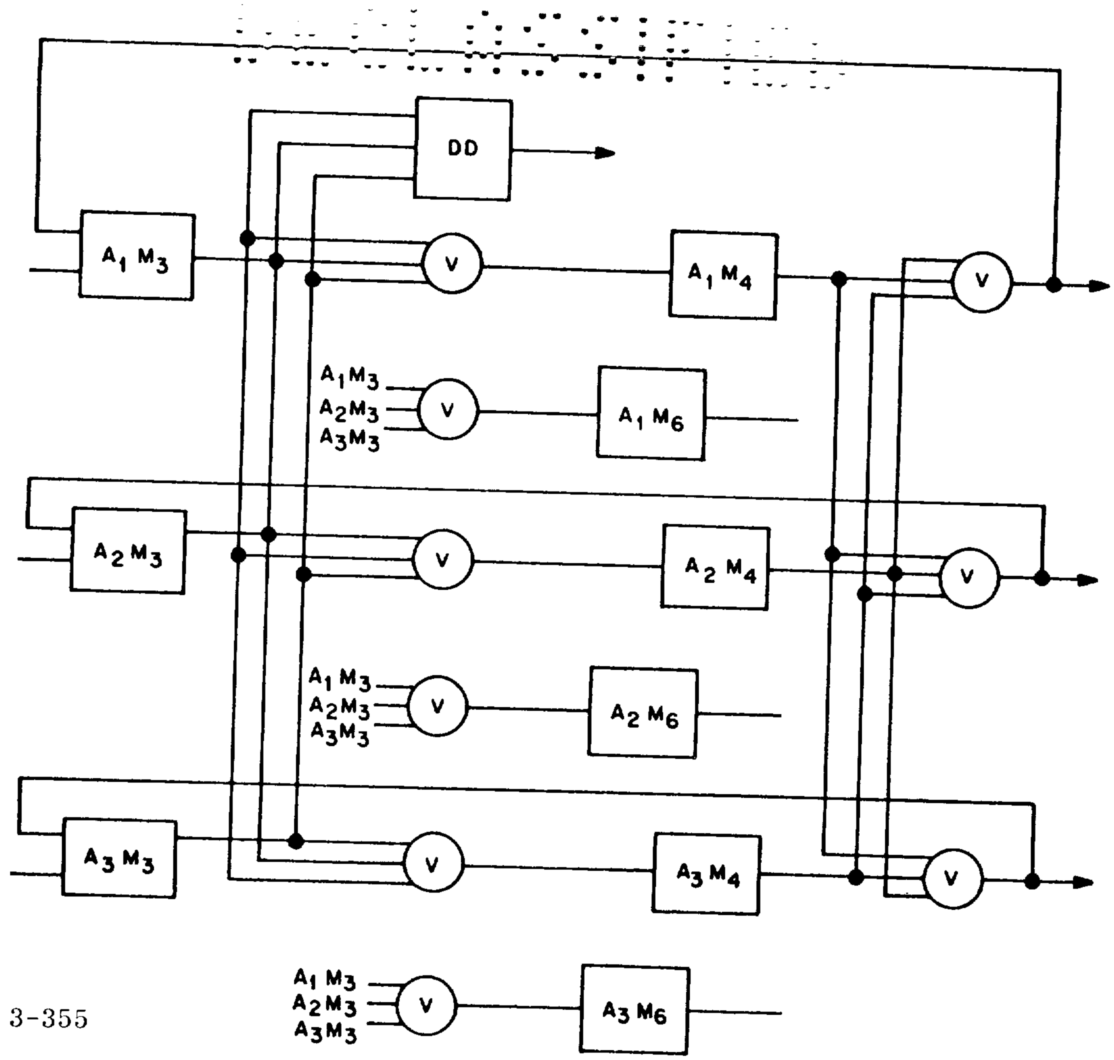
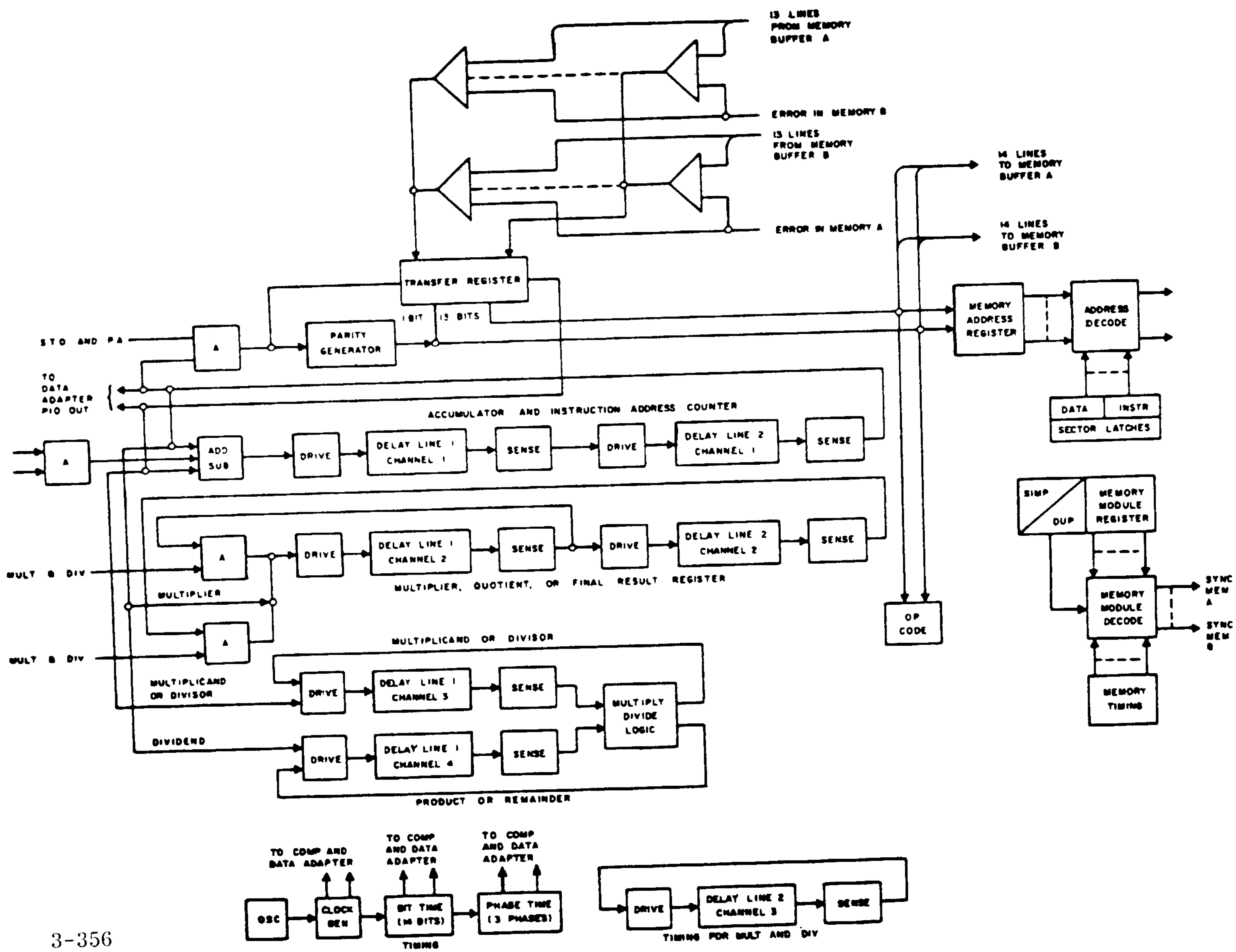


Figure 20-27. Triple Modular Redundancy Voter Signal Outputs

serial data rate of 512 kilobits per second is maintained by operating the memory units in a serial by byte, parallel-by-bit operating mode. This allows the memory to work with a serial arithmetic unit. The parallel read-write work length of 14 bits includes one parity bit to allow checking of the memory operations.

Storage external to the memory is located predominantly in the shift register area. High reliability in this area is achieved by using glass delay lines for arithmetic registers and counters. Delay lines are the best choice when the number of transistors which would be required for the various registers is considered.

20-65. Word Format and Addressing. Each computer instruction word is comprised of a 4-bit operation code and a 9-bit operand address. The 9-bit address allows 512



3-356

Figure 20-28. Guidance Computer Data Flow. Block Diagram, Saturn V locations to be directly addressed. The total memory is divided into sectors of 256 words, and contains a residual memory of 256 words. The 9-bit address specifies a location in either the previously selected sector (data sector latches) or in the residual memory. If the operand address bit (R) is a binary 0, then the data comes from the section specified by the sector register. If R is a 1 the data comes from residual memory.

Instructions are addressed from an 8-bit instruction counter which is augmented by a 4-bit instruction sector register. Sector memory selection is changed by special instructions which change the contents of the sector register. Sector size is large enough so that this is not a frequent operation.

Data words consist of 26 bits. Instruction words consist of 13 bits and are stored in memory two instructions per data word. Hence, instructions are described as being stored in syllable one or syllable two of a memory word. Two additional bits are used in the memory to provide parity checking for each of the two syllables. (Refer to Table 20-12.)

Table 20-12. Digital Computer Data and Instruction Word Format

Memory	Syllable 2	1	2	- - - - - - - - - - - - 13	14
Plane	Syllable 1	15	16	- - - - - - - - - - - - 27	28
Data	Syllable 2	S	2^{-1}	- - - - - - - - - - - - 2^{-12}	P
Word	Syllable 1	2^{-13}	2^{-14}	- - - - - - - - - - - - 2^{-25}	P
Instruction Word	Syllable 1 or 2	A8	A7	- - A1 R OP4 OP3 OP2 OP1	P

S . . . . Sign Position

A8, A7, etc . . . . Operand Address

R . . . . Residual Bit

OP1, OP2, etc . . . . . Operation Codes

P . . . . Parity Bit

The computer is programmed by means of single-address instructions. Each instruction specifies an operation and an operand address. Instructions are addressed sequentially from memory under control of the instruction counter. Each time the instruction counter is used, it is incremented by one to develop the address of the next instruction. After the instruction is read from memory and parity checked, the operation code is sent from the transfer register to the operation code register, a static register which stores the operation code for the duration of the execution cycle.

The operand address portion of the instruction is transferred in parallel (9 bits) from the transfer register to the memory address register. The transfer register is then cleared.

If the operation code requires reading the memory, the contents of the operand address are read 14 bits at a time (including parity) from the memory into the buffers register where a parity check is made. Data bits are then sent in parallel to the transfer register. This information is then serially transfered to the arithmetic section of the computer. If the operation code is a store (STO), the contents of the accumulator are transferred serially into the transfer register and stored in two 14-bit bytes. A parity bit is generated for each byte.

Upon completion of the arithmetic operation, the contents of the instruction counter are transferred serially into the transfer register. This information is then transferred in parallel (just as the operand address has previously been transferred) into the memory address register. The transfer register is then cleared and the next instruction is read, thus completing one computer cycle.

The data word is read from the memory address specified by the memory address register and from the sector specified by the sector register. Data from the memory goes directly to the arithmetic section of the computer where it is operated on as directed by the operation code.

The arithmetic section contains an add-subtract element, a multiply-divide element, and storage registers for the operands. Registers are required for the accumulator, product, quotient, multiplicand, multiplier, positive remainder and negative remainder. The add-subtract and the multiply-divide elements operate independently of each other. Therefore, they can be programmed to operate concurrently if desired; i.e., the add-subtract element can do several short operations while the multiply-divide element can do several short operations while the multiply-divide element is in operation.

No dividend register is shown in Figure 2-28 because it is considered to be the first remainder. The divisor is read from the accumulator during the first cycle time and can be regenerated from the two remainders on subsequent cycles. As indicated, both multiply and divide require more time for execution than the rest of the computer operations. A special counter is used to keep track of the multiply-divide progress and to stop the operation when completed. The product-quotient (PQ) register has been assigned an address and is addressable from the operand address of any instruction. The answer remains in the product-quotient register until another multiply-divide is initiated.

20-66. Timing. The three levels of computer timing are illustrated in Figure 20-29. Basically, the computer is organized around a four clock system. The width of each clock is approximately 0.4 microseconds and the pulse repetition frequency is 512 KHz. The bit time (four clock pulses) is 1.95 microseconds. Fourteen bit times occur in one phase time, resulting in a phase time of 27.3 microseconds. Three-phase times, P_A, P_B, and P_C are required to perform a complete computer operation cycle. Phase A (P_A) makes up the instruction cycle and phases B and C (P_B and P_C) make up the data cycle.

20-67. Computer Control. An instruction list for computer operation is presented in Table 20-13.

All operations except MPY, MPH and DIV require one operational cycle (82 microseconds) for execution. The MPY and DIV instructions must be executed concurrently with any of the other instructions (except MPH). Three instructions can be executed between the start on the MPY and the time when the product is available; similarly, seven instructions can be executed between the start and finish of DIV.

More one-word-time instructions can be inserted before the product or quotient is addressed if maximum efficiency is not required since multiplication or division is stopped automatically and the result retained until addressed. Figure 20-30 illustrates the timing of the MPY and DIV operations.

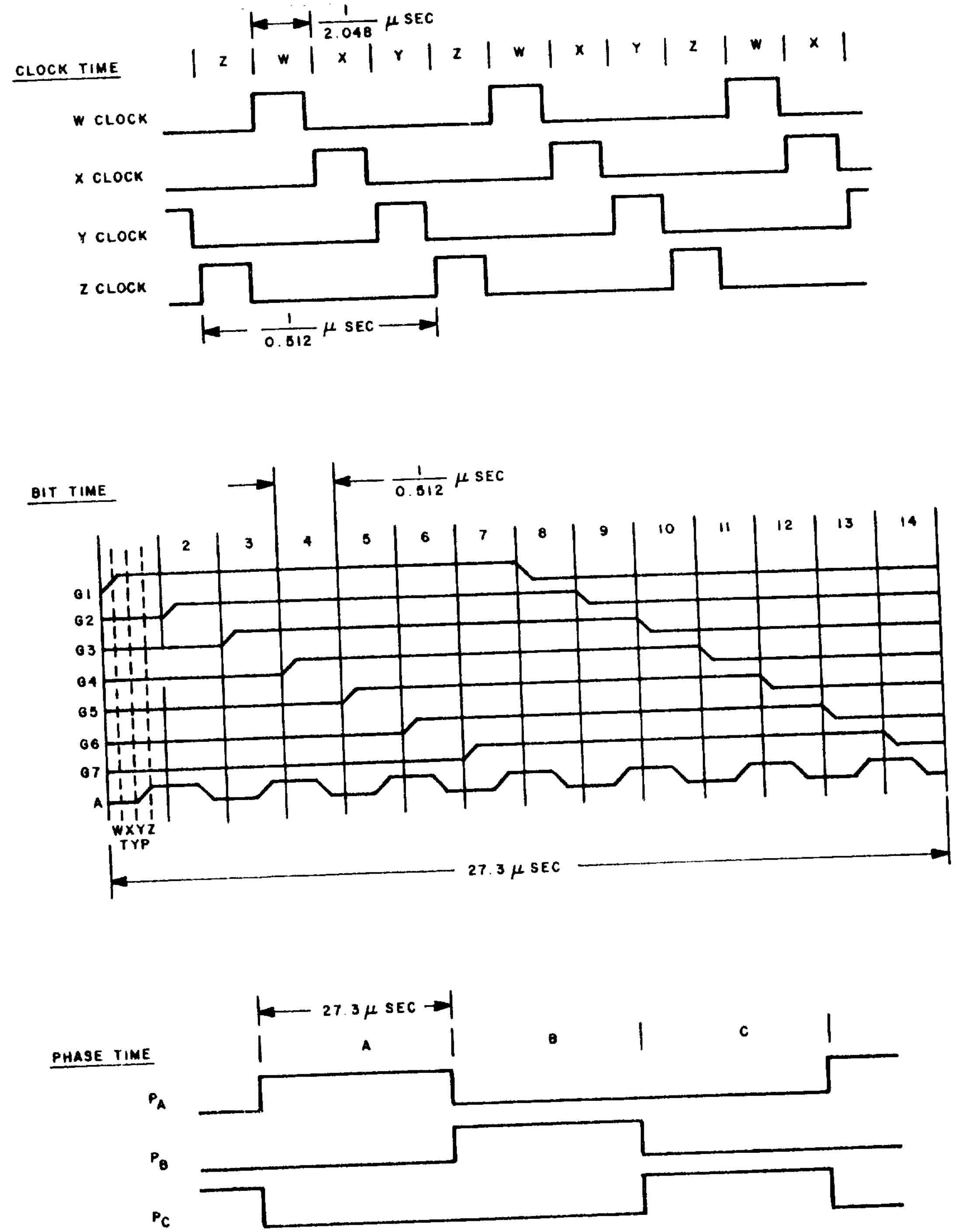


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Figure 20-29. Guidance Computer Timing Charts

Table 20-13. Operation Code Map

Code	Operation
HOP (82 usec) 0000	The contents of the memory address specified by the operand address specify the next instruction address and data sector. Four bits identify the next instruction sector, 8 bits are transferred to the instruction address counter, 1 bit conditions the syllable control, 4 bits identify the next data sector, 3 bits identify the next memory module, 1 bit defines either simplex or duplex memory operation, and 1 bit resets the memory error latch when specifying a new memory module.
TRA (82 usec) 1000	The 8-bit operand address is transferred to the instruction counter. The residual bit in the operand address is used to specify the instruction syllable latch. The sector register remains unchanged.
TMI (82 usec) 1100	A transfer occurs on the minus accumulator sign. If the sign is positive (zero is considered positive), the next instruction in sequence is chosen (no branch); if the sign is negative, the 8 bits of operand address become the next instruction address (perform branch), and a TRA operation is executed.
TNZ (82 usec) 0100	A transfer occurs when the accumulator contains a nonzero number. If the accumulator is zero, the next instruction in sequence is chosen; if the accumulator is not zero (either negative or positive), the 8 bits of the operand address become the next instruction address, and a TRA operation is executed.
SHF (82 usec) 1110	The SHF instruction shifts the accumulator contents right or left one or two places as specified by the operand address. A1 Right Shift 1 A5 Left Shift 1 A2 Right Shift 2 A6 Left Shift 2
AND (82 usec) 0110	The contents of the memory location specified by the operand address are logically AND'ed, bit-by-bit, with the accumulator contents. The result is retained in the accumulator.
CLA (82 usec) 1111	The contents of the location specified by the operand address are transferred to the accumulator.
ADD (82 usec) 0111	The contents of the location specified by the operand address are added to the accumulator contents. The result is retained in the accumulator.
SUB (82 usec) 0010	The contents of the location specified by the operand address are subtracted from the accumulator contents. The result is retained in the accumulator.

Table 20-13. Operation Code Map (Cont'd)

Code	Operation
STO (82 usec) 1011	The contents of the accumulator are stored in the location specified by the operand address. The contents of the accumulator are retained.
DIV (656 usec) 0011	The contents of the accumulator are divided by the contents of the register specified by the operand address. The 24-bit quotient is in the product-quotient delay line. Concurrent use of the adder-subtracter element is required.
MPY (328 usec) 0001	The contents of the memory location specified by the operand address are multiplied by the accumulator contents. The 24 high-order bits of the multiplier and multiplicand are multiplied together to form a 24-bit product. Concurrent use of the add-subtract element is required. The product is stored in the product-quotient register.
MPH (410 usec) 0101	This is the multiply and hold operation. It is the same as the MPY operation except concurrent use of the add-subtract element is not permitted and the product is stored in the accumulator.
XOR (82 usec) 1101	The contents of the memory location specified by the operand address are exclusively OR'd, bit-by-bit, with the contents of the accumulator. The result is retained in the accumulator.
PIO (82 usec) 1010	The low order address bits, A1 and A2, determine whether the operation is an input or output instruction. The high order address bits, A8 and A9, determine whether the data contents are transferred from the main memory, residual memory or accumulator.

The MPH instruction inhibits further access to memory until completed, and cannot be operated concurrently with other operations.

A limited program interrupt feature is provided to aid the input-output processing. An external signal can interrupt the computer program and cause a transfer to a subprogram. Interrupt occurs when the instruction in progress is completed. The instruction counter, sector and module registers, and syllable latch are stored automatically in a reserved residual memory location (octal address 777). A HOP constant is retrieved from a second reserved residual memory location (octal address 706). The HOP constant designates the start of the subprogram. Automatic storage of the accumulator and product-quotient registers is not provided. This must be accomplished by the subprogram. Protection against multiple interrupts and interrupts

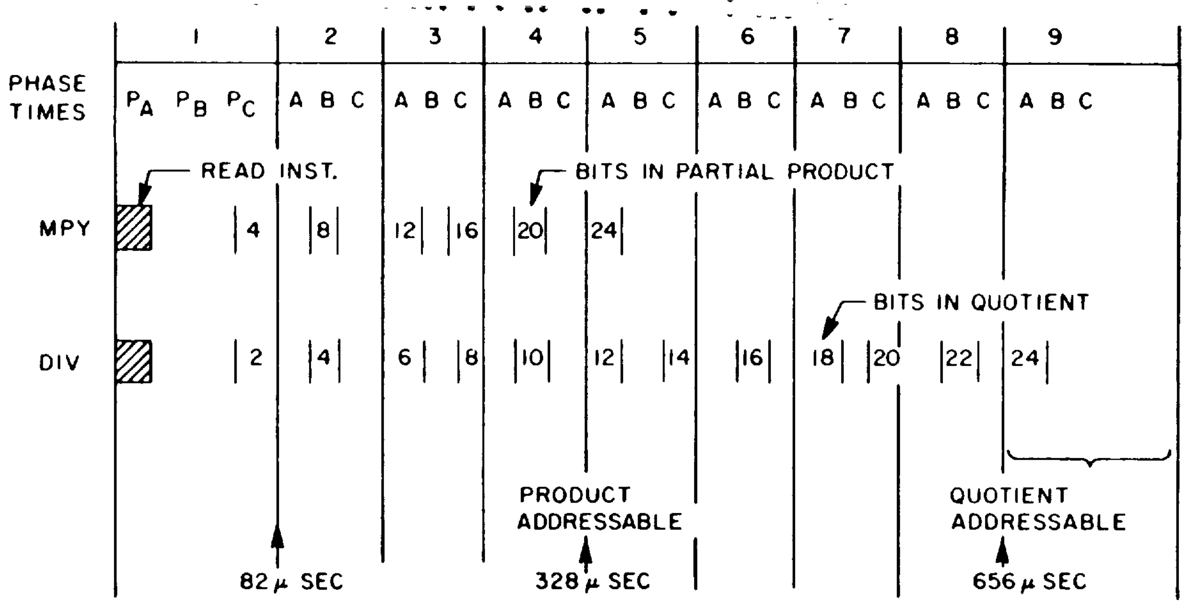

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Figure 20-30. MPY-DIV Timing Chart

during MPY and DIV operations is provided.

The interrupt signal may be generated by a timed source. The rate at which it is generated is controlled by changing the magnitude of a number which is being continually summed. When the summed number reaches a predetermined value, the interrupt signal is generated. This is accomplished in the data adapter equipment.

The main program can be resumed by addressing the contents of residual memory word 777 with a HOP instruction.

Certain discrete input signals are allowed to cause interrupt. These are useful in causing the input-output subprogram to give immediate attention to an input or output operation.

The digital computer uses a conventional complement of arithmetic instructions including add, subtract, multiply, and divide. Two multiply instructions are included. MPY requires that one-word-time operations be performed in the adder unit during the multiplication process because the instruction counter advances each word-time. This procedure speeds up the computer operation by permitting simultaneous multiplication and one-word operations. Trial programming has shown a speed increase of up to 40 percent over a conventional sequential computer.

When the program is multiply-limited, and a sufficient number of useful one-word operations cannot be located in the portion of the flow diagram being executed, the MPH instruction is used. This instruction inhibits advance of the instruction counter so no new instructions are read from memory until the operation is completed. This feature conserves program steps. Having both types of multiply instructions permits the increased speed of concurrent operation without sacrifice in the number of program steps required, and permits a programming tradeoff of speed and number of instructions required.

Instructions, TRA, TMI, and TNZ, provide flexibility in programming unconditional transfers in branch instructions, through transfer of the contents of the accumulator, and in easy handling of discrete inputs, which are obtained in the accumulator through masking with an AND instruction.

The HOP instruction is used for transfers outside of the sector currently being used. HOP permits jumping to another portion of the flow diagram and to subroutines. To return from a subroutine, the last instruction in the routine is a HOP. The HOP constant causes a return to the original program sequence. Since each use of a subroutine in the program results in return to a different place in the flow diagram, the HOP constant is loaded prior to entering the subroutine. An automatic program compiler is used to generate the correct HOP constants.

An exclusive OR operation, XOR, is provided to permit the rapid checking of changes in discrete inputs, which are grouped into data-word inputs. Discrete output words may be generated by masking out the bit to be changed with an AND instruction and adding the discrete output into the selected position.

The product-quotient register can be addressed (by Octal 775) with the operations CLA, ADD, SUB, STO, AND and XOR.

The interrupt feature in the computer facilitates the timing of input-output operations by causing a transfer to an input-output subprogram. The interrupt signal is generated in the data adapter and may be set to interrupt at the highest rate at which any input-output quantity must be handled. This method avoids the necessity of keeping track of time expired since last entering the input-output subprogram.

The automatic interrupt also makes it possible to permit certain discrete inputs to cause interrupt. Allowing discrete inputs to interrupt makes it possible to demand that the program give attention to an important discrete input. Communications between the computer and the vehicle telemetry monitoring system are thus facilitated.

The vehicle monitor system is selected by an address code from the computer. The definition of the vehicle parameter to be monitored is given over the output lines to the data adapter and stored in a buffer register. When the monitor has acquired the desired parameter, an interrupt is generated causing the computer input-output subprogram to read the value of the parameter as an input. This scheme permits computing to continue while waiting for the monitor system to acquire the parameter.

The data sector register permits considerable flexibility in the handling of data and constants. Instructions indicate whether data are located in the residual sector or the sector referred to by the data sector register. By confining data to the residual register and a limited nember of memory sectors, the changing of the data sector register can be minimized. The residual sector is then made more readily usable for data referred to by instructions stored in many sectors. The small size of each sector, achieved by concentrating instructions rather than both data and instructions in each sector, reduces the size of the instruction word and conserves memory core planes. The programmer is free to move between separate parts of the program without frequently changing instruction or data sector registers.

The data sector register is also useful in addressing sets of constants stored for use with polynomial injection guidance equations. The instructions necessary to compute the polynomials are stored once. Sets of coefficients for the many different polynomials are each stored in different memory sectors. The coefficients can be readily retrieved by use of the data address register, which is set to select a given set of coefficients in the evaluation of the polynomial. Thus, the location of the polynomial number is set in the sector register and the coefficients are selected.

The separate instructions and data sector register feature eliminates the need for indexing, since it accomplishes the same end result in polynomial evaluation, the chief application of indexing. Hardware and instruction bits are saved by omitting indexing.

Upper and lower limits for orbital checkout parameters are stored in the two halves of a data word. Addressing of the parameter through the monitoring system is related to the storage location of the limits in memory. A simple, regular sequence of addresses makes programming easy by the use of address modification techniques.

20-68. Computer Arithmetic

The Saturn V computer has two independent arithmetic elements, the add-subtract element and the multiply-divide element. Although both operate independently, they are serviced by the same program control circuits and may be operated concurrently. During each program cycle-time, the add-substract element can perform any one of the computer instructions, except MPY, MPH, and DIV. Also during each program cycle-time, the results of the simple arithmetic operations are circulated through the accumulator delay line and through the accumulator sync delay line channel to prevent processing of the results.

The multiply-divide element uses three channels of a delay line as shown in Figure 20-30. One channel of the instruction counter delay line is used as a counter to stop the multiply or divide operations. Another channel of the instruction counter delay line is used to synchronize the product or quotient when the operation is completed. This controlled automatically by the counter.

The product-quotient register is addressable as a residual memory word and has the octal address 775. The product or quotient can be obtained on any subsequent operation, but must be used before initiation of another multiply or divide operation. The product of the MPH operation is stored in the accumulator.

The recursion formulas for implementing multiply and divide instructions with two's complement numbers are explained in the following paragraphs.

Multiply. The multiply element operates in a two-phase cycle, serial-by-four parallel, and requires 15 phase times, including instruction access time. The program initiates a multiply by placing the 24 high-order bits of the contents of the memory location specified by the operand address into the multiplicand delay line. The multiplier delay line contains the 24 high-order bits of the contents of the accumulator. The phase counter terminates a multiply instruction.

The instrumentation of the multiply algorithm requires three delay line channels. Two of the channels contain the partial product and the multiplier. These channels shift both the partial product and the multiplier four places to the right every two-phase cycle. The third channel contains the multiplicand. The accumulator portion (fourth channel) of this delay line is not involved in the multiply operation and can be used concurrently with the multiply operation.

Upon initiation of a multiply and during every other phase time thereafter, the five low-order bits of the multiplier (MR_1, MR_2, MR_3, MR_4, and MR_5) are placed in latches or tratches and are used to condition addition or subtraction of multiples of the multiplicand to the partial product. The following algorithm is utilized for multiply:

$$P_i = 1/16 \left[P_{(i-1)} + \Delta 1 + \Delta 2 \right]$$

P_i is the new partial product, and $\Delta 1$ and $\Delta 2$ are formed according to the rules:

MR_1	MR_2	MR_3	$\Delta 1$	
MR_3	MR_4	MR_5		$\Delta 2$
0	0	0	0	0
1	0	0	+2M	+8M
0	1	0	+2M	+8M
1	1	0	+4M	+16M
0	0	1	-4M	-16M
1	0	1	-2M	-8M
0	1	1	-2M	-8M
1	1	1	0	0

M represents the multiplicand. For the first multiplication cycle $P_{(i-1)}$ and MR_1 are made zero.

<u>Divide.</u> The divide element operates in a two-phase cycle, serial-by-two-parallel, and requires 27 phase times per divide, including instructions access time. The program initiates a divide by transferring the 26 bits of the addressed memory location (divisor) and the 26 bits of the accumulator (dividend) to the divide element. The phase counter terminates a divide operation. The following algorithm is instrumented to execute divide:

$$Q_i = R_{is} \cdot DV_s + \overline{R_{is}} \cdot \overline{DV_s} \qquad (1)$$

and

$$R_{i+1} = 2R_i + (1 - 2Q_i)\ DV \qquad (2)$$

where:

i = 1, 2, 3, . . . 24

Q_i = The i^{th} quotient bit

R_{is} = The sign of the i^{th} remainder

DV_s = The sign of the divisor

R_i = The i^{th} remainder

R_1 = The dividend

DV = The divisor

Equation (1) states that the i^{th} quotient bit is equal to a "1" if the sign of the i^{th} remainder is identical to the sign of the divisor. The high-order quotient bit (sign bit) is the only exception to this rule. Q_i as determined by equation (1) is used to solve equation (2) but must be complemented before it is stored as the sign bit of the quotient.

The instrumentation of the divide algorithm requires three channels of a delay line. One channel contains the quotient, one the divisor, and one the dividend. These three channels are used during multiply to contain the multiplier, the multiplicand, and the partial product respectively. The quotient and the remainder channels of the delay line have been lengthened by latches to shift two places to the left each two-phase cycle. The divisor circulates once each two-phase cycle.

In the two's complement number system, the high-order bit determines the sign of the number. Since this is the last bit read from memory, it is impossible to solve equations (1) or (2) until the entire divisor has been read from memory. However, equations (1) and (2) can have only two possible solutions.

Either,

Q_i = 1

and,

$$R_{(i+1)} = 2R^{i} - DV$$

or,

$$Q_i = 0$$

and,

$$R_{(i+1)} = 2R_i + DV$$

Both the borrow of $2R_i - DV$ and the carry of $2R_i + DV$ are generated as the dividend and divisor registers are loaded. When the sign bits of these quantities are finally entered into their respective registers, equation (1) is solved for the first quotient bit. If this quotient bit is a one, the borrow is examined to determine the second quotient bit. If the first quotient bit is a zero, the carry is examined to determine the second quotient bit. The following truth table is solved to determine the second quotient bit if the first quotient bit is a one.

R_i	DV_s	B	$R_{(i+1)s}$	Q
0	0	0	0	1
0	0	1	1	0
0	1	0	1	1
0	1	1	0	0
1	0	0	1	0
1	0	1	0	1
1	1	0	0	0
1	1	1	1	1

Where

R_i = The first remainder bit to the right of the sign bit

DV_s = The divisor sign

B = The borrow into the R_i, DV_s position

$R_{(i+1)s}$ = The sign of the new remainder

Q = The quotient bit as determined by comparing DV_s with $R_{(i+1)s}$ according to Equation (2).

$$Q = \overline{R_i}\,\overline{DV_s} \cdot \overline{B} + \overline{R_i}\, DV_s\, \overline{B} + R_i \cdot \overline{DV_s} \cdot B + R_i \cdot DV_s \cdot B$$

$$= \overline{R_i} \cdot \overline{B}\,(\overline{DV_s} + DV_s) + R_i \cdot B\,(\overline{DV_s} + DV_s)$$

$$= \overline{R_i} \cdot \overline{B} + R_i \cdot B$$

The equation used in generating the new remainder, R_{i+2}, is obtained by expanding equation (2).

$$R_{(i+2)} = 2R_{(i+1)} + (1 - 2Q_{(i+1)})\, DV$$

$$R_{(i+2)} = 2\left[R_i + (1 - 2Q_i)\, DV\right] + (1 - 2Q_{(i+1)})\, DV$$

$$R_{(i+2)} = 4R_i + 2\,(1 - 2Q_i)\, DV + (1 - 2Q_{(i+1)})\, DV$$

As $R_{(i+2)}$ is being generated, the next iteration of divide is started by generating, as already described, the borrow and carry for $2R_{i+2} = DV$.

20-69. Computer Memory Section. The digital computer uses conventional toroidal cores in a unique self-correcting duplex system for achieving a memory reliability of 0.990 for 250 hours of duplex operation or 0.958 for 250 hours when operating simplex (for 8000 words of memory). The memory consists of four identical 4096-word memory modules which may be operated in simplex for increased storage capability or in duplex pairs for high reliability. The basic computer program is loaded at electronic speeds into the instruction and constants sectors of the memory on the ground just prior to launch. Thereafter, the information content of constants and data can be electrically altered, but only under control of the computer program.

The self-correcting duplex system uses an odd-even parity bit detection scheme in conjunction with memory drive current error detection circuitry for malfunction indication and correction. Unlike conventional toroid random access memories, the self-correcting extension of the basic duplex approach permits regeneration of correct information after transients or intermittent failures which otherwise would result in destructive read-out of the memory.

The configuration, Figure 20-31, consists of a pair of memories providing storage for 8192 14-bit memory words when operating duplex, or 16,384 14-bit memory words when simplex operation is desired. Each of the simplex memories includes independent peripheral instrumentation consisting of timing, control, address drivers, inhibit drivers, sense amplifiers, error detection circuitry and input-output connections to facilitate failure isolation. Computer functions common to these simplex units consist of the following:

a. Memory address register outputs;
b. Memory transfer register input-output;
c. Store gate command;
d. Read gate command;
e. Syllable control gates.

Computer functions, which are separate for each simplex memory, consist of the synchronizing gates, which provide conversion of the serial data rate of 512 kilobits per second. These gates also provide selection of multiple simplex memory units for storage flexibility and permit partial or total duplex operation through the mission profile for purposes of extending the mean-time-before-failure for long mission times. Each of the simplex units can operate independently of the others or in a duplex manner.

Memory modules are divided into two groups, one group consisting of even numbered modules (0-6), the other consisting of odd numbered modules (1-7). A buffer register associated with each group is set by the selected modules.

For duplex operation, as shown in Figure 20-31, each memory is under control of independent buffer registers when both memories are operating without failure. Both memories are simultaneously read and updated in parallel (14 bits). A single cycle is required for reading instructions (13 bits plus 1 party bit per instruction word). Two memory cycles are required for reading and updating data (26 bits plus 2 parity bits).

The parallel outputs of the memory buffer registers are serialized at a 512-kilobit rate at the memory transfer register under control of the memory select logic.

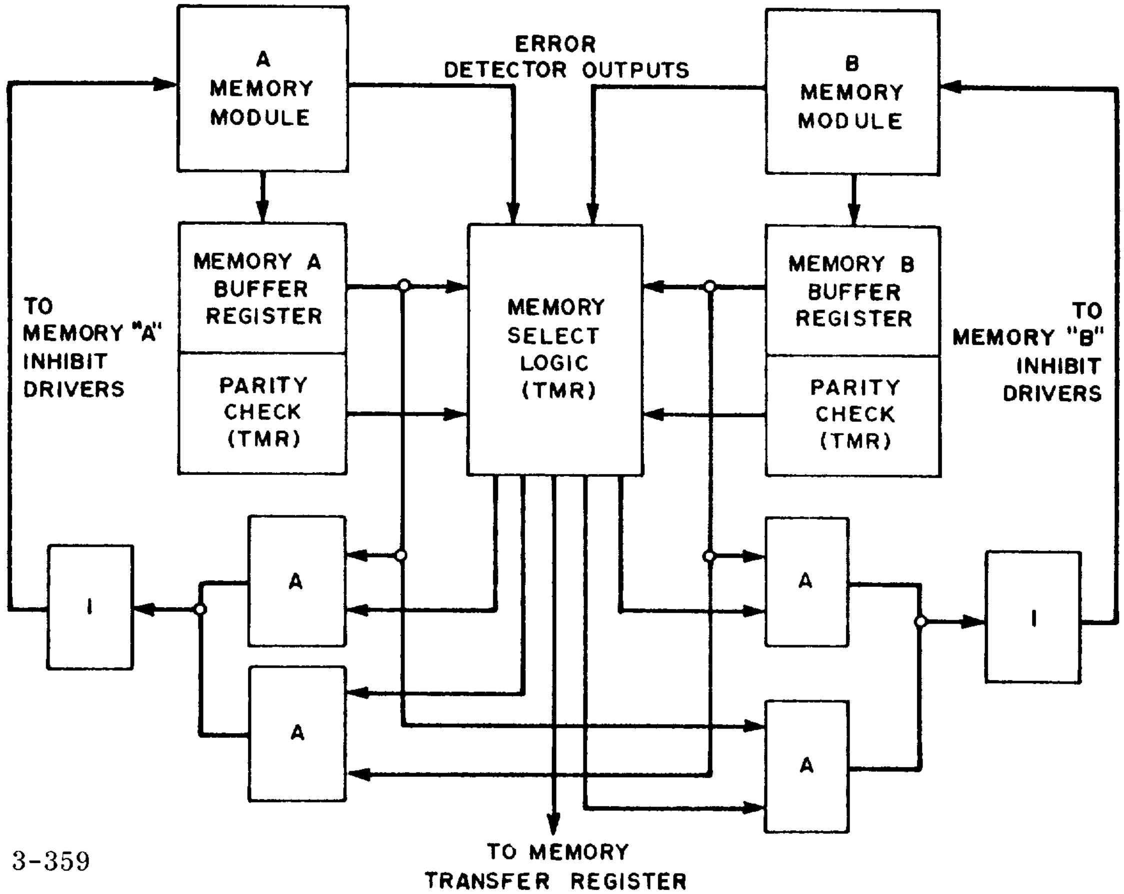

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Figure 20-31. Self-Correcting Duplex-Toroid Computer Memory System

Initially, the outputs of only one buffer register are being used with simultaneous parallel parity checking being performed on both register outputs. When an error is detected in the memory being used, operation immediately transfers to the other memory. Both memories are then regenerated by the buffer register of the "good" memory, thus correcting transient errors.

After the parity-checking and error detection circuits have verified that the erroneous memory has been corrected, operation returns to the condition where each memory is under control of its own buffer register. Operation is not transferred to the previously erroneous memory until the "good" memory develops its first error. Consequently, instantaneous switching from one memory output to another permits uninterrupted computer operation until simultaneous failures at the same location in both memories causes complete system failure.

Proper operation of the memory system during read cycles is indicated by each 14-bit word containing an odd number of "one's" and a logical "one" output of the error detecting circuitry. If either or both of these conditions are violated, operation is transferred to the other memory.

During regenerate or store cycles, parity checking cannot be performed. Failure detection is accomplished by the error detection circuitry only. Parity checking is performed during subsequent read cycles.

Intermittent addressing of memory between normal cycles is detected by the error detecting circuitry producing a logical "one" output at the improper time. Figure 20-32 indicates the system connection of the error detector circuits for a simplex memory.

The control latch circuits are packaged with the buffer register circuitry in the computer. The output latch is in a logical "zero" state for normal operation. If the error detector output is a logical "zero" at normal cycle times, or a logical "one" at the improper time, the output latch is set to the "one" state indicating an

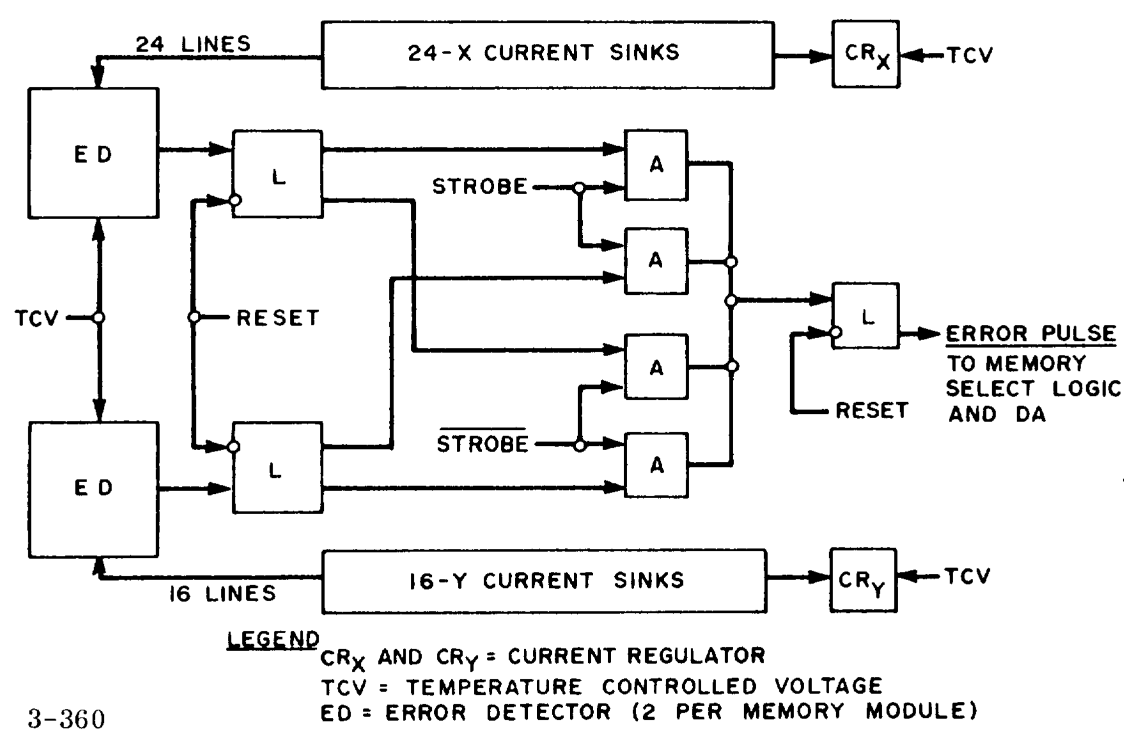

Figure 20-32. Error Detection Circuit Connection for Simplex Computer Memory

error. Conditions which will result in an error output are as follows:

a. Address without voltage source

b. Address without current sink

c. No address

d. Dual source-single sink address

e. Single source-dual sink address

20-70 Computer Input-Output Section. The computer input-output section is characterized by the type of input-output instructionused and its interrupt feature. The process input-output instruction provides for transferring of a single word into or out of the accumulator or out of the memory.

The process input-output instruction transfers data between the accumulator or memory and one-word registers and delay lines located in the data adapter or other subsystem. The operand address is used to select the desired register.

Discrete inputs and outputs can be processed by this instruction. It is possible to pack 26 discrete signals into one word. The XOR instruction will determine if any of the 26 discrete inputs have changed state. The AND instruction is used to set or reset any of the discrete outputs.

Interrupt signals can be generated within the data adapter. These signals will stop the computer program and cause a branch to a subprogram. The location of the subprogram is program-controlled and is dependent upon the HOP constant stored in a specific memory location. This subprogram is normally used to process a block of input-output data on a periodic basis. The rate at which the timed interrupt occurs is also program-controlled and can be adjusted as dictated by the various modes of operation during a given mission.

The main program can be resumed after completion of the subprogram by executing a HOP operation from another specified memory location. This location contains the contents of the instruction counter, sector register, and syllable latch, which were stored there when the interrupt occurred.

20-71. ST-124-M INERTIAL PLATFORM SYSTEM*

The ST-124-M inertial platform system provides the inertial reference for the Saturn V vehicle guidance. Ths system also furnishes the mechanics for thrust vector attitude programming, steering error signals, platform gimbal positions for attitude computation, and velocity information for computation of vehicle position and velocity. A block diagram of the guidance system interconnection is in Figure 20-33. Figure

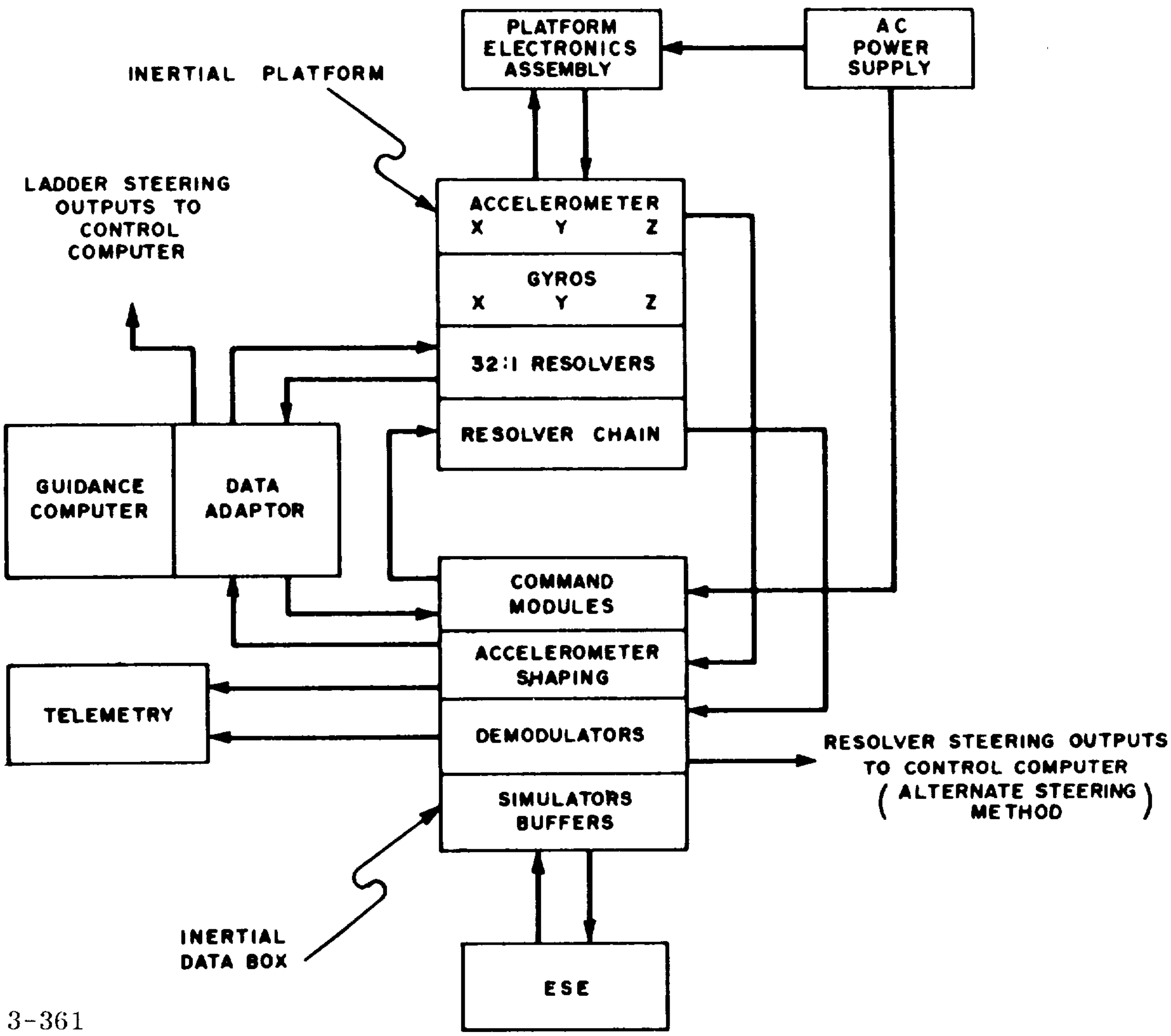

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Figure 20-33. Guidance System Interconnection Block Diagram

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*A general Description of the ST-124-M Inertial Platform System (Report No. M-ASTR-IN-63-27)
H. E. Thomason and J. G. Rowell, Gyro and Stability Branch, Astrionics Division, September 23, 1963.

20-22 shows two sets of outputs to the control computer. Only one set will be used. As defined in guidance operation the control computer receives outputs from the data adapter in the primary guidance method. The alternate method uses the resolver chain approach with the outputs from the demods in the inertial data box being supplied as inputs to the control computer. The blocks that comprise the ST-124-M inertial system are the inertial platform, the platform electronic assembly, the inertial data box, and the platform ac power supply.

The ST-124-M system is a modification of the ST-124-2 system developed for the Saturn I vehicle. The major differences between the systems are the additions of the inertial data box, the platform ac power supply, and multi-speed resolvers as digital shaft encoders to measure platform gimbal angles. A description of the major assemblies of the system is presented in the following paragraphs.

20-72. ST-124-M Inertial Platform. The St-124-M inertial platform is designed for a three or four gimbal configuration. The four gimbal configuration, Figure 20-34, is designated the ST-124-M MOD IV inertial platform. The three gimbal configuration, the ST-124-M MOD III inertia platform, has identical outer, middle and inner gimbals, but no redundant gimbal. The vehicle wiring is also identical so that only the platform vehicle mounting frame is affected by a configuration change.

The ST-124-M Mod IV offers unlimited freedom about all three inertial reference axes while the ST-124-M Mod III is limited to $\pm$ 45 degrees about its X axis (vehicle yaw at launch). The vehicle mission dictates which of the configurations is required.

The location of the major platform components is illustrated in Figure 20-35. A brief discussion of the function and characteristics of each follows.

The AB-5K8 stabilizing gyroscope, Figure 20-36, is used to maintain the inner gimbal fixed in inertial space. Its data are listed in Table 20-14.

The AMAB-3K8 pendulous integrating accelerometer, Figure 20-37, provides the vehicle acceleration information to the guidance computer. The accelerometer data are listed in Table 20-15.

The gas bearing erection pendulum, Figure 20- 38, is used for erection of the inertial

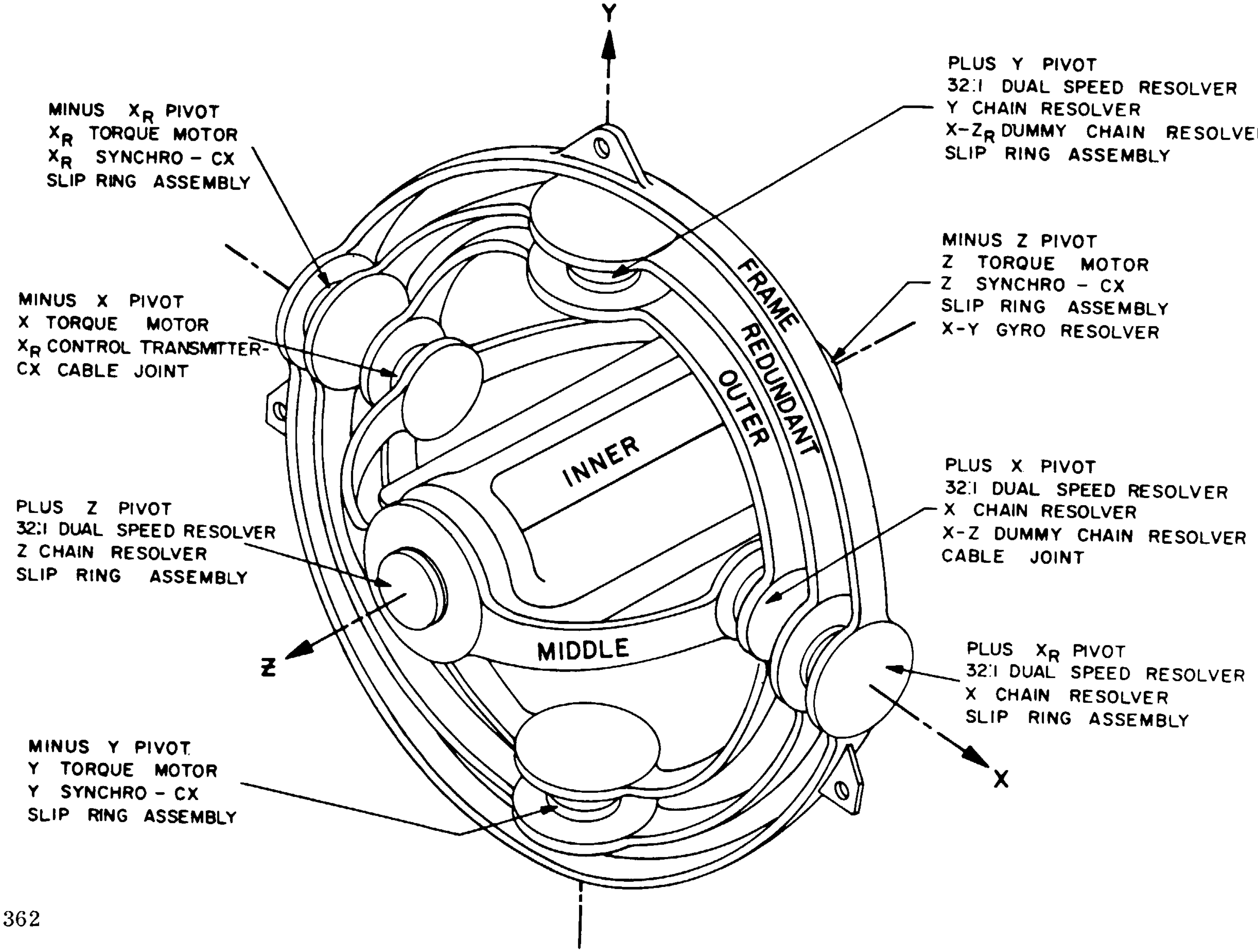

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Figure 20-34. Four-Gimbal Configuration

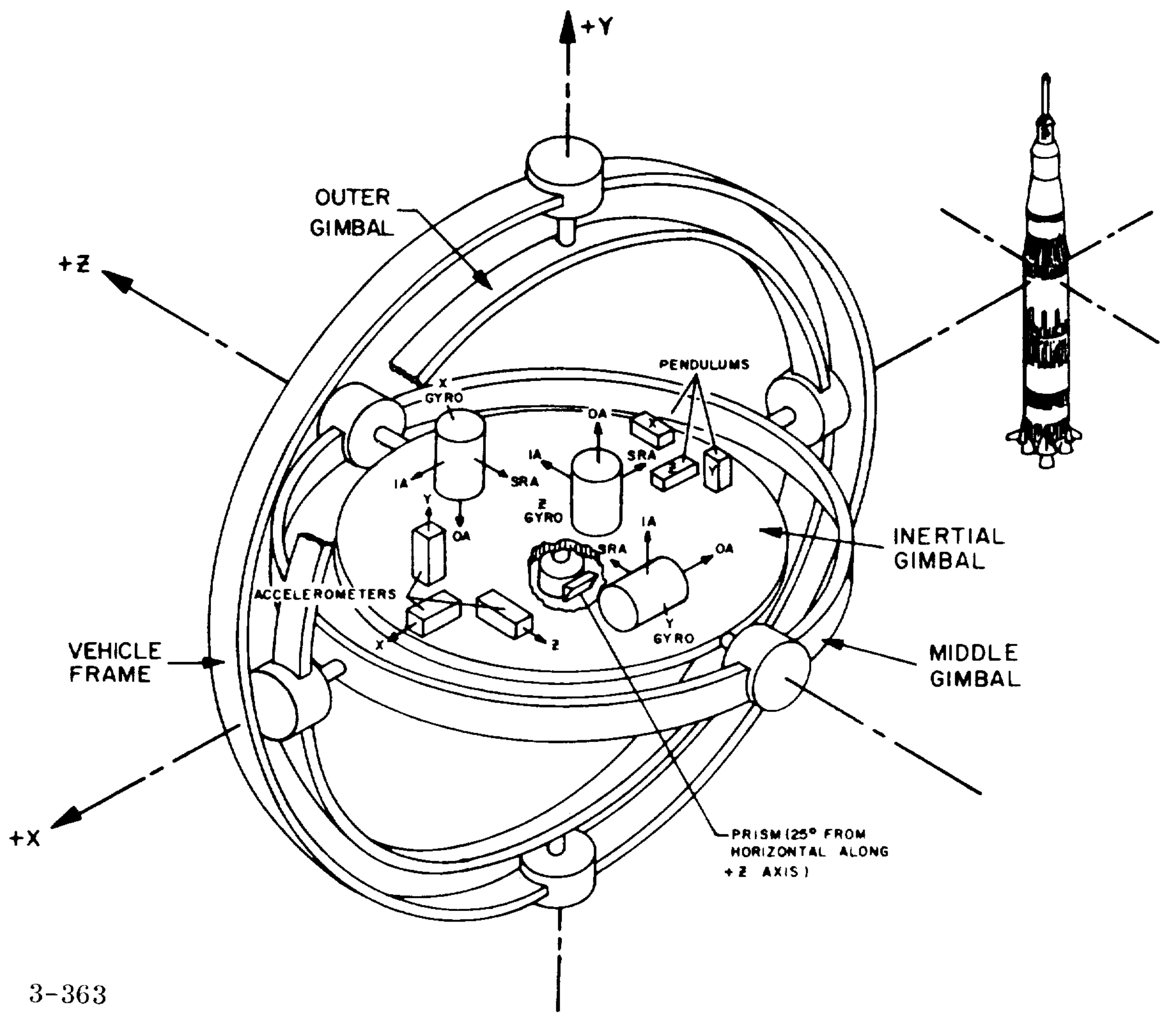

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Figure 20-35. ST-124 M Gimbal Configuration

platform. Three gas bearing pendulums are mounted on the platform inertial gimbal. Two are used for platform erection to the local vertical. The third is oriented 90 degrees to the others and is used to erect the platform in other than normal positions for accelerometer testing. The pendulum data are listed in Table 20-16.

The resolver chain is used in the alternate guidance mode. The ST-124-M inertial platform has, fixed to each gimbal, a resolver which is electrically connected in series with three program or command resolvers to form a chain. The chain performs the coordinate transformation computations. The output signals are furnished in the form of steering signals to the control computer.

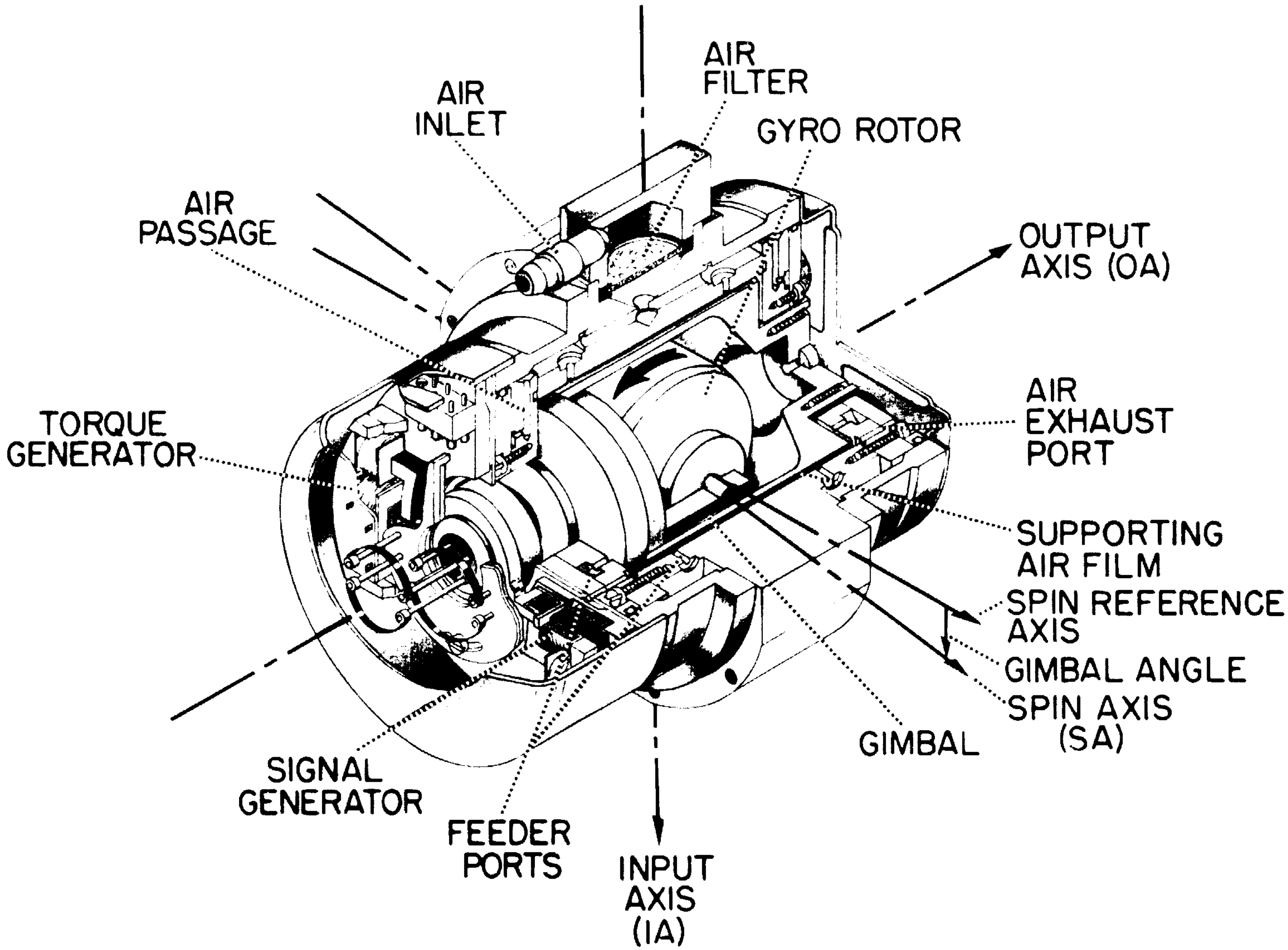

Figure 20-36. Single Axis Integrating Gyro

Table 20-14. AB-5K8 Stabilizing Gyroscope Data

Item	Data
Gyro Wheel	
Type	Synch. hys.
Angular momentum	2×10^6 g cm^2/s
Wheel speed	24,000 rpm
Wheel excitation	26 volt, 3 phase, 400 Hz
Wheel bearing preload	3.4 kg operating
Wheel power at sync	10 watts
Wheel life	2000 hrs. min.
Wheel mount	Sym.
Wheel sync time	90 sec.
Gas Bearing	
Gas Pressure	1.03 bars diff.
Gas flow rate	2000 cc/min.
Air gap	.0015 cm to .002 cm
Orifice restrictors	Millipore discs
Signal Generator	
Type	Shorted turn reluctance
Excitation	10 volts, 4.8 KHz
Sensitivity	420 millivolts/degree with 10 K load
Float freedom	$\pm$ 3 degrees
Torquer	
Type	Shorted turn reluctance
Normal erection rate	6 degrees /min
Fixed coil excitation	26 volts 400 Hz - 45 ma
Maximum variable coil excitation	30 volt 400 Hz - 50 ma
Physical Characteristics	
Size	3 in. dia. by 4 in. length
Weight	900 gm
Mounting	Three point flange
Temperature Characteristics	
Calibration temperature	$40^{\circ}C$ (gyro housing)
Drift vs. temperature gradient	$.008^{\circ}/h/^{\circ}C$

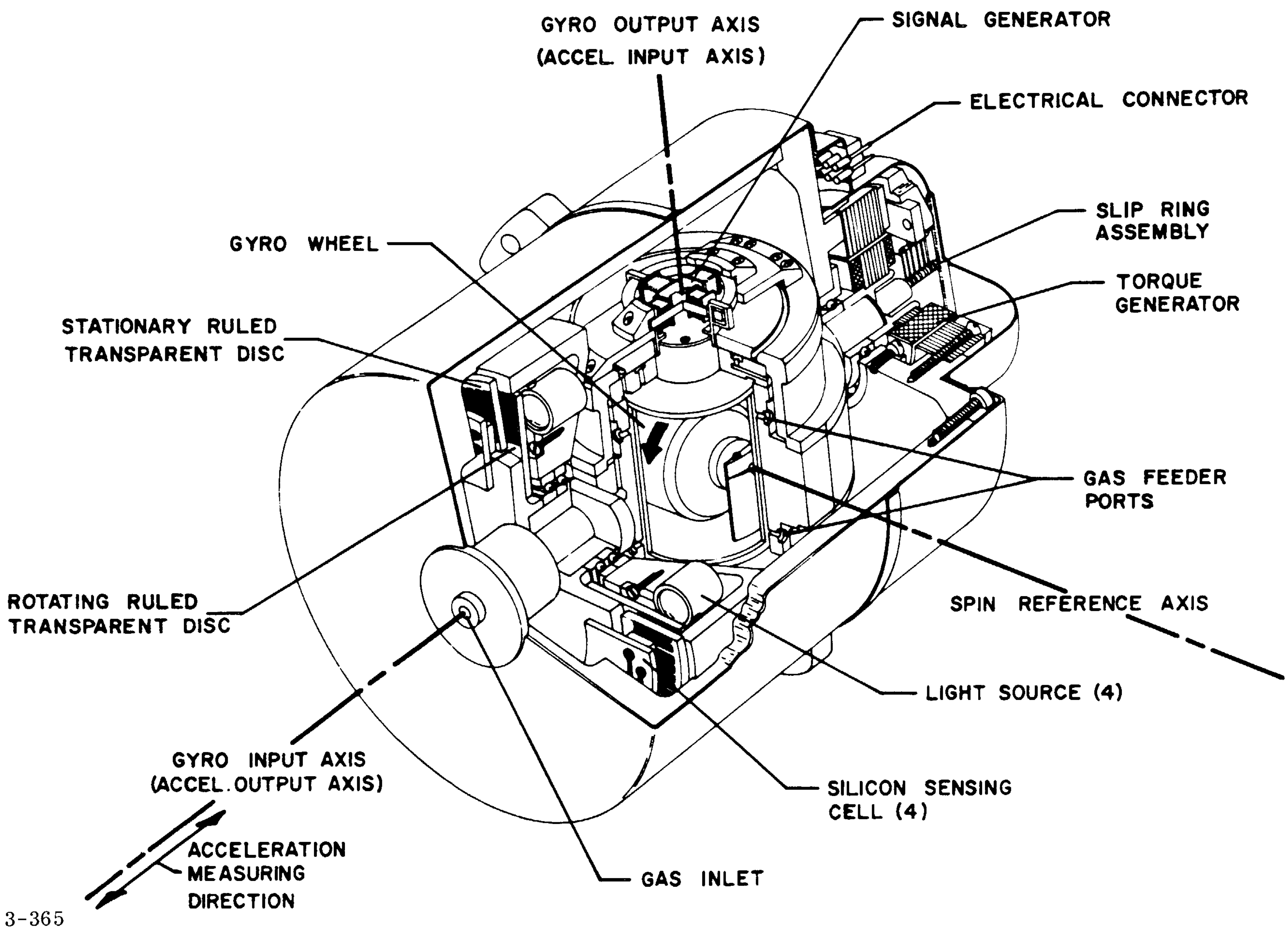

Figure 20-37. Pendulous Integrating Gyro Accelerometer

Table 20-15. AMAB-3K8 Pendulous Integrating Accelerometer Data

Item	Data
Gyro Wheel	
Type	Synch. hys.
Angular momentum	1×10^5 g cm^2/g
Wheel speed	12,000 rpm
Wheel excitation	26 volts, 3 phase, 400 Hz
Wheel sync time	90 sec.
Wheel power at sync	4.5 watts
Wheel life	2000 hrs. min.
Wheel mount	Sym.
Wheel bearing preload	907.2 gm.
Gas Bearing	
Gas pressure	1.03 bars diff.
Gas flow rate	4800 cc/min.
Air gap	.0015 cm to .002 cm
Orifice restrictors	Millipore discs
Signal Generator	
Type	Four pole shorted turn reluctance
Excitation	10 volts, 4.8 KHz
Sensitivity	285 millivolts/degree with 10 K load
Float freedom	$\pm$ 3 degrees
Torque Motor	
Type	Direct axis dc torquer
Maximum torque	1.440 kg cm at 1.1 A
Incremental Digital Encoder	
Type	Optical grid with redundancy
Counts	6000 counts per revolution
Physcial Characteristics	
Size	3.25 in. dia. by 5 in. length
Weight	900 gm
Mounting	Three point flange mounting
Temperature Characteristics	
Calibration temperature	40°C amb.

Table 20-15. AMAB-3K8 Pendulous Integrating Accelerometer Data (Cont'd)

Item	Data
Ambient temperature range for accuracies stated.	$40^{o}C \pm 5^{o}C$

Table 20-16. Gas Bearing Erection Pendulum Bearing Data

Item	Data
<u>Physical Characteristics</u>	
Size	2.25 in. by 1.5 in. by 1.25 in.
Weight	92 gm
<u>Gas Bearing</u>	
Gas pressure	1.03 bars diff.
Gas flow	100 cc/min
Air Gap	.0016 cm to .0018 cm
<u>Signal Generator</u>	
Type	Inductive
Excitation	4 volts, 400 Hz
Sensitivity	300 millivolts/degree
<u>Performance</u>	
Leveling accuracy	$\pm$ 5 arc sec
Input range	$\pm$ .5 degree (signal saturation)
Time constant	10 sec.

The command modules in the inertial data box receives commands from the guidance computer and generate analog signals through the chain for vehicle attitude control. The analog signals generated are conditioned by the command voltage demodulators into a dc voltage whose polarity and amplitude represent the vehicle displacement from the desired attitude.

The resolver chain data are listed in Table 20-17.

The gimbal angle multi-speed resolvers, one on each gimbal, are used as start

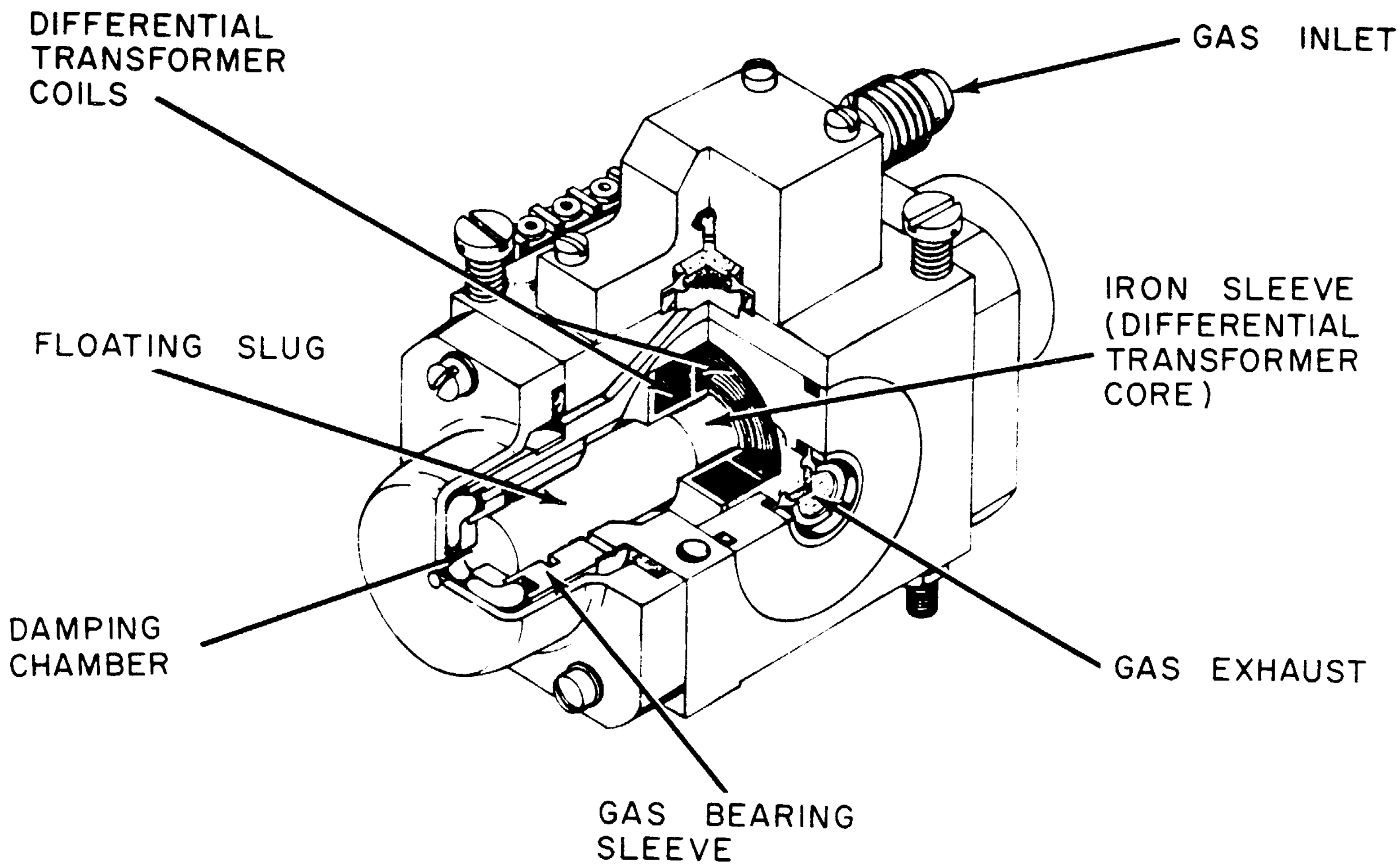

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Figure 20-38. Gas Bearing Erection Pendulum position angle encoders. The output of each resolver is sent to the guidance computer and the gimbal angles from launch are measured and stored. This measurement actually closes the guidance loop around the platform. The resolver data are listed in Table 20-18. A schematic diagram of the multi-speed resolver and bridge connection is shown in Figure 20-39.

Table 20-17. Resolver Chain Data

Item	Data
Excitation	
f_1	26 volts, 1.6.KHz
f_2	26 volts, 1.92 KHz
Demodulator Output	
To control computer	3 volts dc/degree
To telemetry (fine)	$\pm$ 2.5 volts dc/ $\pm$ 3^{o}
(coarse)	$\pm$ 2.5 volts dc/ $\pm$ 15^{o}
Linear range	$\pm$ 15 degrees

Table 20-18. Resolver Data

Item	Data	
Resolver Characteristics	32 Speed	Single Speed
Excitation voltage	26 volts $\pm$ 5%	26 volts $\pm$ 5%
Excitation frequency	1000 Hz $\pm$ 0.01%	1000 Hz $\pm$ 0.01%
Excitation power	1.15 watts	0.05 watts
Mechanical accuracy	$\pm$ 10 arc sec	$\pm$ 30 arc min
System Characteristics		
System hi-speed	64:1	
System lo-speed	1:1	
Static accuracy	$\pm$ 30 arc sec	
Dynamic accuracy (error is proportional to input rate)	20 arc sec at 0.2 rad/sec	
Computer clock frequency	2×10^6 Hz $\pm$ 0.01%	
Temperature range for optimum accuracy.	$\pm$ 30^{o} C	

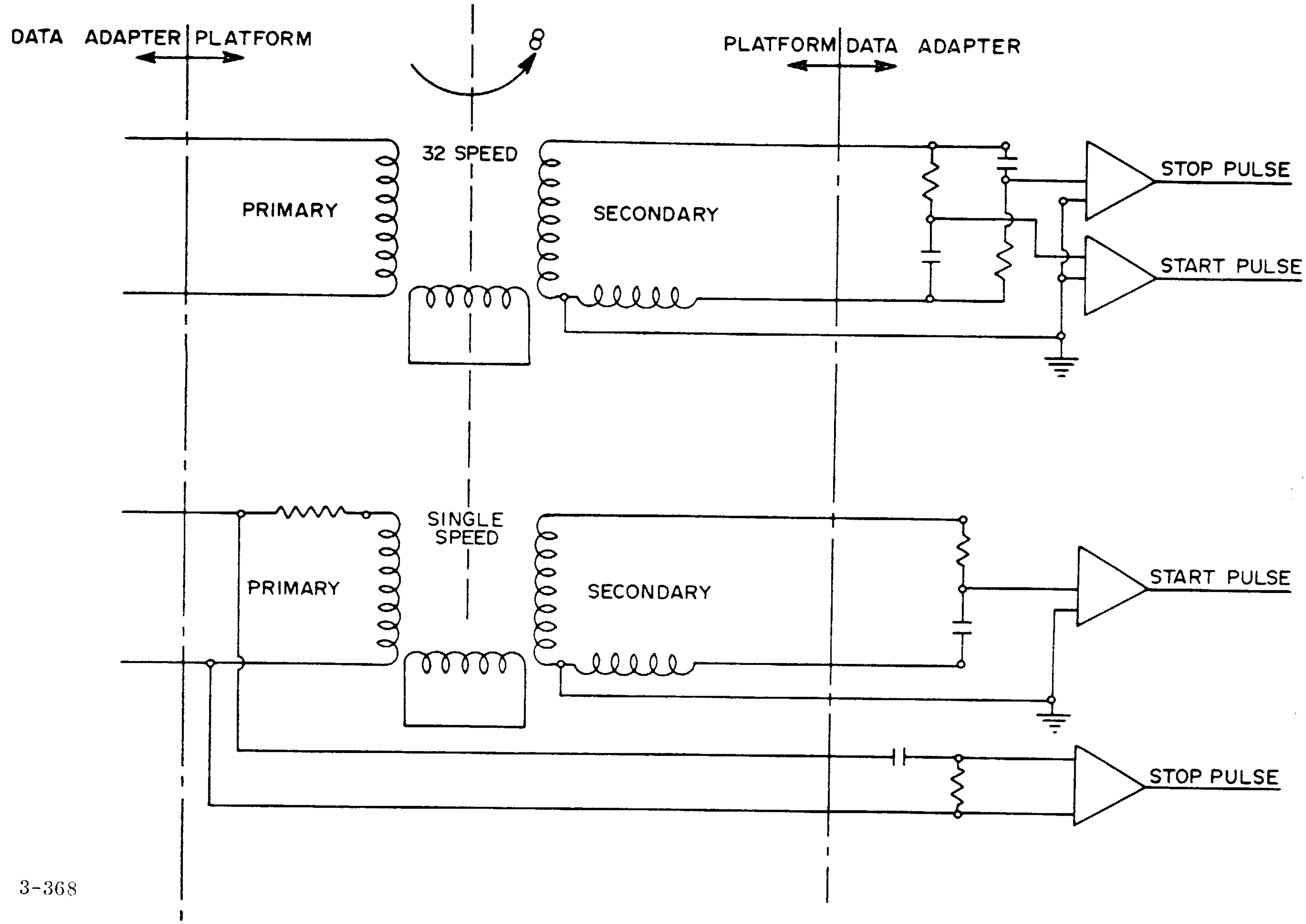

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Figure 20-39. Two-Speed Resolver Schematic

20-73. Platform Electronics Assembly. The platform electronics assembly contains the electronics, other than those located in the platform, required for platform axis stabilization. The following is a list of components for the ST-124-M Mod IV platform electronics assembly:

a. Three gyro servo amplifier cards
b. Three gimbal torquer power stages
c. One redundant gimbal servo amplifier card
e. One redundant gimbal torquer power stage
f. Three accelerometer servo amplifier cards
g. Three accelerometer torquer power stages
h. One 4.8 KHz voltage amplifier card
i. One automatic checkout selector switch
j. One gyro wheel current transformer assembly
k. One relay card assembly
l. One elapsed time indicator
m. Four power switching relays
n. Eight electrical connectors
o. One 400 Hz keying transformer
p. One temperature sensor
q. Elapse time indicator

The majority of the items listed are plug-in modules. The assembly for the MOD III is identical except items (c) and (d) are deleted.

Modules requiring pressurization are hermetically sealed. Internal heat sources are heat-sinked to the main casting and cooling realized by conduction into the temperature-controlled mounting panels of the instrument unit.

20-74. Inertial Data Box Assembly. The inertial data box is a conditioner for signals between the platform system and the remainder of the guidance system including the telemetry and ESE. Its primary functions ar to: *(1) accept attitude programming from the guidance computer and convert it to analog inputs to the resolver chain, *(2) convert the output of the resolver chain into steering signals for the control computer, (3) condition the accelerometer digital encoder outputs for use by the guidance com-

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*Alternate steering scheme only.

puter and ESE monitor, (4) accept command from the ESE for control system check-out, (5) condition the attitude and acceleration outputs for telemetry, and *(6) furnish the excitation voltages for the resolver chain.

The assembly is constructed similar to the platform electronic assembly and is hermetically sealed. The expected weight of the assembly is 22.5 kilograms (50 pounds).

The following is a list of modules located in this assembly:

a. Three command voltage demodulators (steering signals)
b. Three attitude command programming modules
c. One 1.6 KHz voltage amplifier
d. One 1.92 KHz voltage amplifier
e. Three accelerometer output shaping modules
f. Three accelerometer output buffer amplifiers
g. Three accelerometer telemetry conditioners
h. Three ESE simulated attitude command modules
i. Temperature sensor
j. Elapse time indicator

20-75. <u>Platform ac Power Supply.</u> This assembly furnishes the power required to run the gyro wheels, excitation for the platform gimbal synchros, and frequency sources for the resolver chain excitation and servo carrier. All frequencies are derived from a crystal and are accurate to = .01 Hz. The assembly is constructed similar to the other electronic assemblies and is hermetically sealed. The following outputs are generated in the ac power supply:

a. 26 volt, 3 phase, 400 = .01 Hz
b. 20 volt, 4.8 KHz;
c. 20 volt, 1.6 KHz;
d. 20 volt, 1.92 KHz.

The platform ac power supply contains the following:

a. Electronic modules
b. Frequency standard;
c. Three electrical connectors;
d. Pressure sensor;
e. Temperature sensor;

f. Elapsed time indicator.

20-76. Platform Erection and Azimuth Alignment. The erection and alignment of the stabilized platform is described in the following paragraphs.

Platform Erection. The erection of the platform gimbals is accomplished by use of gas bearing pendulums. Three pendulums are mounted on the platform inertial gimbal with their input axes normal to each other and parallel to the accelerometer measuring axes. The inertial gimbal is erected in any of six positions by applying the proper pendulum signal to the proper gyro torquer. This enables the laboratory or prelaunch check of each accelerometer in plus-minus orientations in the earth's gravitational field. One of the six positions is the normal erected position of the platform gimbals, and the ground support equipment is designed to accomplish this erection automatically.

A typical erection servo loop is shown in Figure 20-40. The pendulum output signal is preamplified in the platform and transmitted to the alignment panel located in the ground support equipment. The electromechanical integrator eliminates any standing error caused by earth's rotation. The integrator output and its derivative are fed through the torquer amplifier to the electromagnetic torquer on the output axis of the platform stabilization gyro. The precession rate caused by this applied torque positions the inertial gimbal until the pendulum is null. The normal erection rate is limited by the gyro electromagnetic torquer to six degrees per minute. By applying biasing signals into the servo loop, slewing rates up to 45 degrees per minute can be obtained.

Azimuth Alignment. For preflight alignment, the azimuth heading of the inertial gimbal must be held to a close tolerance. On Saturn class vehicles, this alignment must be accomplished with relatively large vehicle sway and twist present in the area of the platform compartment. The alignment of the inertial gimbal to any azimuth heading, regardless of vehicle heading, is also required. These are accomplished by use of a prism ring mounted to the inertial gimbal and an automatic, sway-compensating, long range theodolite located on the ground.

The prism ring is a motor driven gimbal containing a porro prism and the sta-

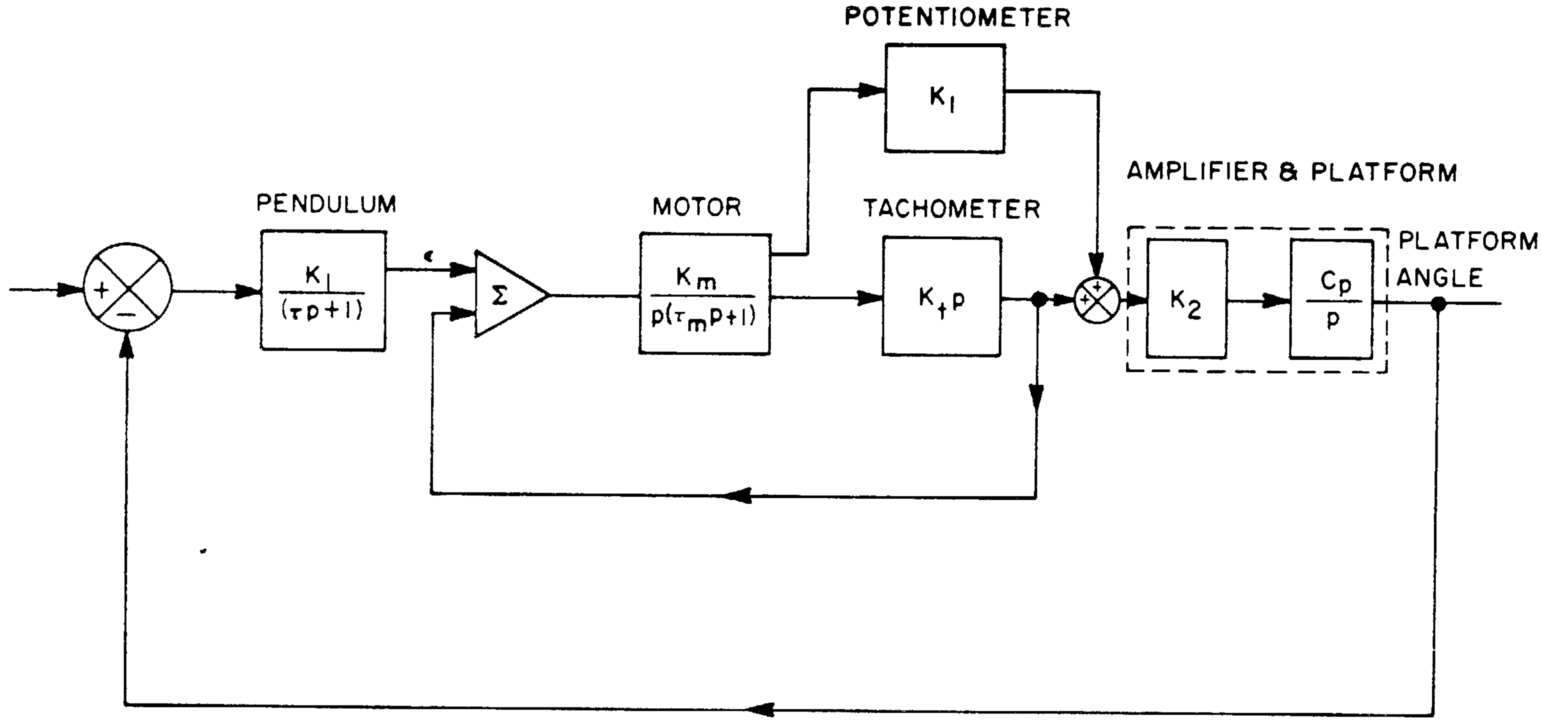

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Figure 20-40. Gas Pendulum Erection Servo Loop tor of a multi-speed synchro. It is capable of being driven about the vertical axis determined by the pendulums in the normal erected platform position.

The initial alignment is accomplished by electrically driving the prism ring to the inertial gimbal by using the azimuth pickup. (See Figure 20-41) This

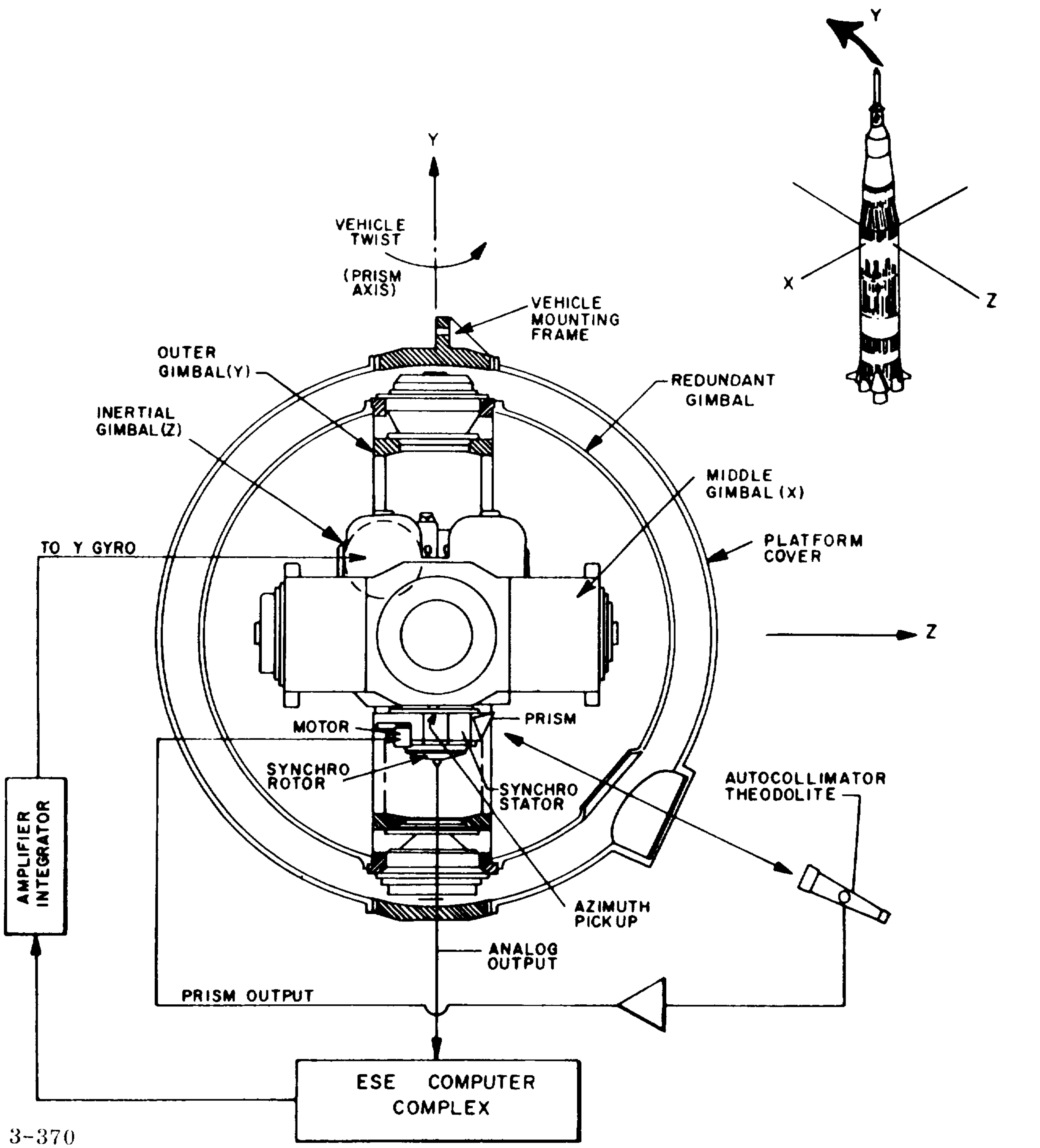

Figure 20-41. Automatic Azimuth Alignment

brings the accelerometer measuring axis into alignment with the prism. The next step is to acquire the prism with the theodolite and maintain it on a known azimuth line by means of a nulling servo loop feeding the vertical axis stabilization gyro torquer. The multi-speed synchro output is used to slave a ground-based digital encoder so that it reads the angle between the inertial gimbal and the prism ring. The digital reading, when the prism ring is driven to the inertial gimbal and the prism maintained on a known azimuth line, is a reading of that azimuth.

The prism is then released from the inertial gimbal and maintained on the known heading by controlling it directly with the theodolite. The inertial gimbal may now be rotated. The digital encoder will track it and measure its deviation from the known heading. The launch control computer can now compute the required launch azimuth and supply the signal to rotate the inertial gimbal to and maintain it on the launch azimuth.

Figure 20-42 shows the alternate steering scheme used to control the alignment of the ST-124-M.

20-77. CONTROL COMPUTER (FIGURE 20-43).

The control computer is an analog device that instruments and solves the vehicle thrust vector equation $B = a_0(x - \phi) + a_1\dot{\phi} - g_2\dot{\gamma}$, where B is the thrust vector required to attain a given angular attitude, $(x - \phi)$ is the attitude error input, i.e., the angular difference between the commanded vehicle angle and the present vehicle angle, $\dot{\phi}$ is the rate at which the vehicle's angular attitude is changing and $\dot{\gamma}$ is the lateral acceleration or drift away from the desired flight path. The coefficients a_0, a_1, and g_2 are time-varying functions that determine which of the three terms of the equation are dominant in determining the value of B during first stage flight of the S-IC. For the S-II and S-IVB stages, the a_0 and a_1 coefficients are constant and g_2 is zero.

Since the vehicle must be commanded in three axes (pitch, yaw and roll), the three terms on the right side of the equation are made up of three attitude error imputs and three attitude rate signals, one for each axis and two lateral acceleration inputs, one each for pitch and yaw. The latter term requires only two axes of motion since any lateral movement away from the desired flight path can be completely described by a lateral

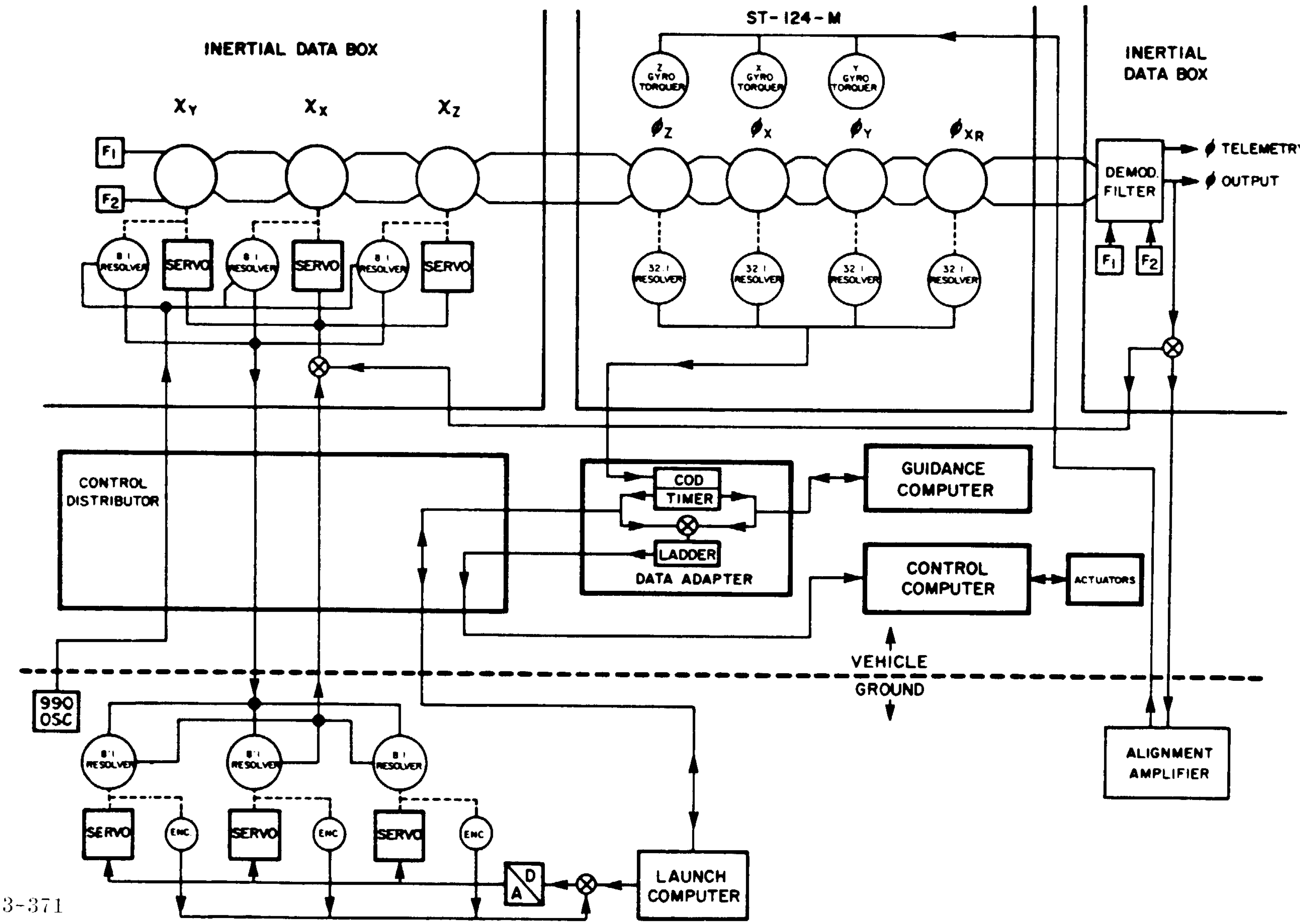

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Figure 20-42. Prelaunch Test Configuration

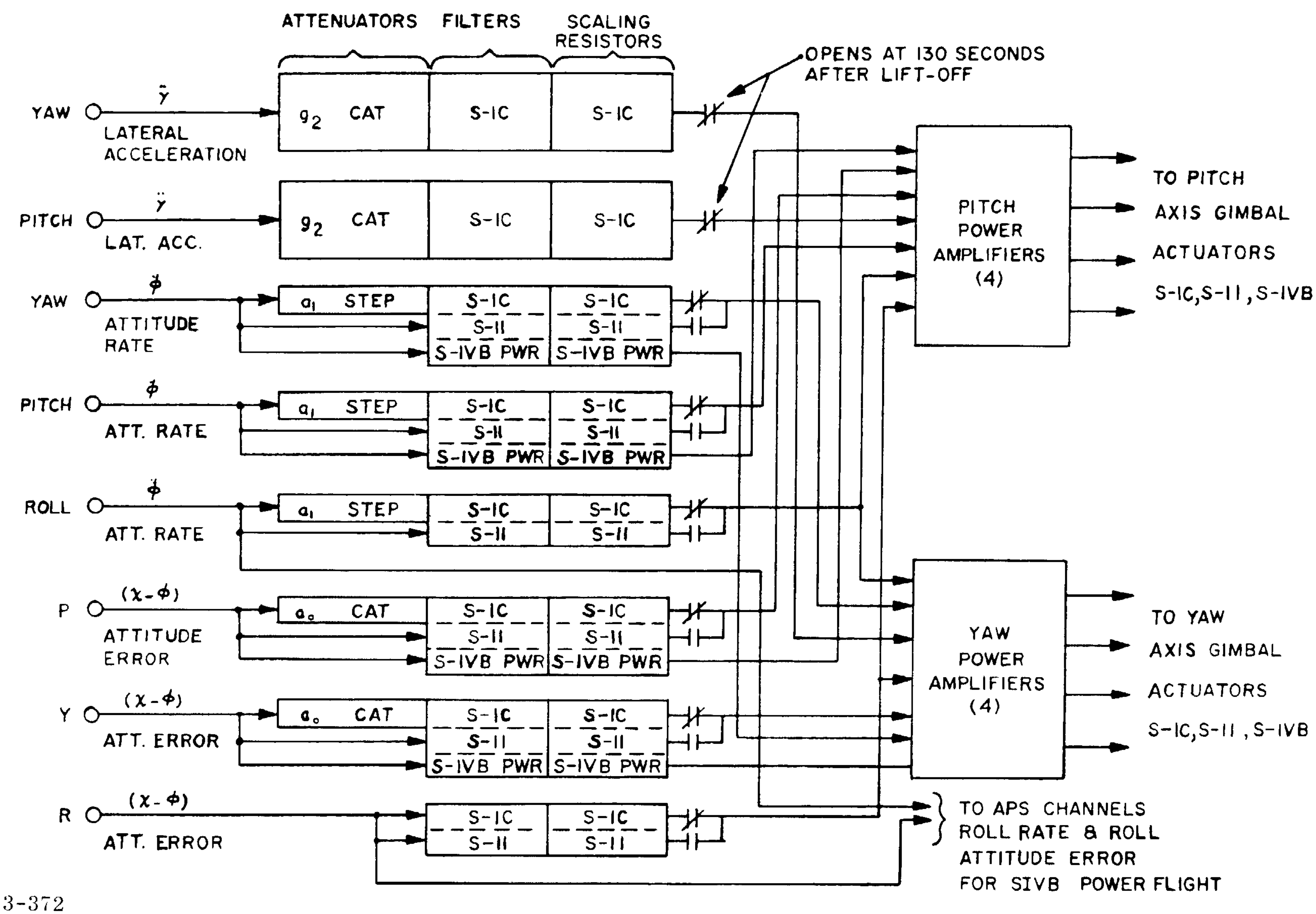

Figure 20-43. Control Computer. Engine Control Channels

motion in the pitch or yaw axes or by a vector sum of the two.

The magnitude of g_2 (see Figure 20-44) is programmed by the control attenuation timers device to increase during the period in which the first stage is passing through the area of maximum dynamic pressure and is at a maximum when these dynamic pressures are maximum. At the same time, the value of the a_0 coefficient is decreased toward a predetermined minimum. This area of maximum dynamic pressures is called "Q maximum," and maximizing the g_2 coefficient here allows the lateral acceleration signals, $\ddot{Y}$ to be dominant factor in determining the B thrust vector. Since the lateral accelerations experienced by the vehicle when passing through a "Q maximum" are primarily due to angle-of-attack changes, this will produce dominant angle-of-attack feedback in the vehicle attitude control system. The decrease of a_0 during this time is consistent with a "minimum load" program, although some attitude feedback, $a_0 (x-\phi)$, is required to maintain adequate low frequency response in the control system.

After passing through the "Q maximum" area, the g_2 coefficient is decreased as the a_0 coefficient is increased and a_0 increase in dominance. Eventually, as shown in Figure 20-44, the g_2 term is completely eliminated from entering into the equation solution by opening of a relay contact in series with this signal at 130 seconds after lift-off.

The gain program for a_1, the attitude rate coefficient, is not cam-programmed by the control attenuation timers device but is a discrete step attenuation performed by relay switching. The time-based command for controlling the relay, however, may be issued from the control attenuation timers or as a discrete output from the data adapter. Figure 20-44 shows that the discrete step change in a_1 occurs at 95 seconds after lift-off, and reduces the value of a_1 to a value which is only slightly larger than a_0 at 125 seconds.

The filters, Figure 20-43, are used to decouple the control frequency from the undesirable bending mode, elastic deformation and propellant sloshing frequencies transmitted by the control sensors. The control sensors have been placed at vehicle stations in an attempt to decouple them from these disturbances. However, any one location cannot be optimum for all bending modes.

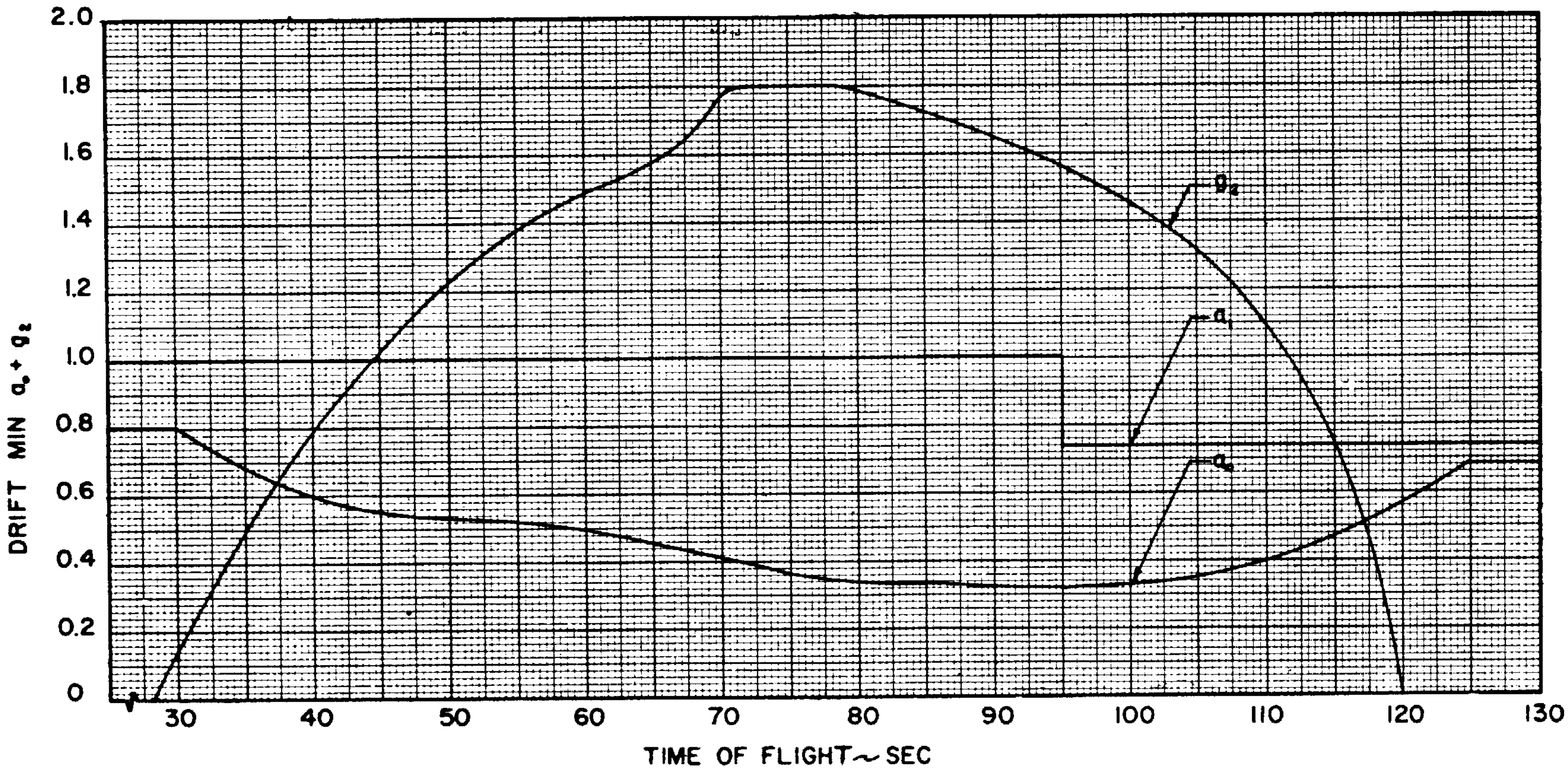

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Figure 20-44. Typical Gain Program

Thus the complete equation for any given value of B requires eight terms, three for attitude error, three for attitude rate and two for lateral acceleration. The inputs to the control computer, attitude error, rate and lateral acceleration signals, are supplied by the digital computer, rate gyros and body-fixed control accelerometers. Each of these devices supplies signals which are analogs of the physical values they represent. The digital computer, rate gyros and control accelerometers are discussed in detail in other sections.

The control computer must exercise attitude control through two types of propulsion systems. Engine gimbal control is required during all powered flight and auxiliary propulsion control is required during S-IVB stage flight.

20-78. Engine Gimbal Control. The control computer uses eight channels to implement the eight required control terms for engine gimballing. The channels are basically identical in that each channel may be divided into four sections which are designated according to the process being performed on the input signal as it proceeds through the channel. These sections are first stage attenuators, filters, scaling resistors, and power amplifiers.

First Stage Attenuators (Figure 20-43). The first stage attenuators are of two types, (1) the control attenuation time (CAT) and (2) the relay-switched discrete step type. The attenuators are used only during the S-IC stage to provide the gain co-efficients a_0, a_1 and g_2. For the S-II and S-IVB stages the attenuators are either bypassed, as in the case of a_0 and a_1, or their outputs are unused as in the case of g_2. For all stages after the first, the values of a_0 and a_1 are unity and the value of g_2 is zero.

The CAT is a mechanical device that is used to vary the gain co-efficeint, a_0, of the attitude error pitch and yaw channels and the gain co-efficient, g_2, of the lateral acceleration, pitch and yaw channels. The a_0 program is inscribed on one side of a motor driven cam and the g_2 program upon the other side. For each program, a rocker arm drives two ganged potentiometers which are electrically connected to the appropriate channel. The output from the potentiometers are thus a time-varying function of the input signal, i.e., the $(x - \phi)$ and $\dot{\gamma}$ signal inputs to the attentuator appear at the attenuator output modified by a_0 and g_2.

Therefore, the filters are introduced into the system to provide an operation which "phase stabilizes" the first and second bending modes, and "gain stabilizes" higher frequencies.

Gain stabilization rests on the ability of amplitude attentuation to keep the bending mode lobes small and adequately removed, regardless of phase, from the minus one (instability) point on a Nyquist plot. Gain stabilization is instrumented by low pass filters which decouples bending (and other disturbance) frequencies from the control frequency. In the Saturn V launch vehicle, however the frequencies of the first and second bending modes are sufficiently near the 0.15 Hz control frequency that this low pass filtering would constrain the control frequency bandwidth and reduce the systems transient response to wind gusts, which in itself poses serious structural problems. Lowering the control frequency reduces the low frequency response of the control system so that this method cannot be used to discriminate further between the control frequency and bending mode frequencies. The problem is solved by phase-stabilizing the first and second bending modes and gain-stabilizing the higher bending modes and other disturbances with frequencies greater than the second bending mode frequency.

Phase stabilization shifts the lobes of the bending mode lobes such that the resulting phase margin from the minus one point on a Nyquist plot is independent of the bending magnitude. Thus, phase stabilization allows an increase in control bandwidth without inducing instability, which increases the system's transient response. It also results in increased closed loop first mode damping, which permits the system to follow first mode oscillations and provide increased damping with properly phased thrust actuation. Proper placement of the rate gyros can be useful in providing phase stabilization regardless of the amplitude of the first mode.

As shown in Figure 20-43, each filter section of the eight channels in the control computer are sub-divided into unique filters for the S-IC, S-II and S-IVB stages. This is necessary since the undesirable signals caused by bending modes, elastic deformation and propellant sloshing change in frequency as the vehicle becomes shorter due to stage separation. In addition, the external forces of aerodynamic pressure and wind gusts, which tend to aggravate the undesirable bending and deformation characteristics, are absent after the vehicle has departed the earth's atmosphere.

The filter required for each stage is selected by relay switching at the output of the scaling resistor following each filter. In addition to the relay switching used for filter selection during staging, there is also relay switching employed at the input of the S-IVB power filter in the pitch and yaw attitude error channels. This additional relay switching allows selection of the pitch and yaw attitude rates from either the digital computer in the IU via the data adapter or from the Apollo spacecraft. This provides a back-up capability for controlling the S-IVB powered flight.

The filters are packaged in removable modules which facilitate changing of filtering network when different filtering characteristics are required for different type missions.

Scaling Resistors (Figure 20-43). The scaling resistors are relay switched, voltage-divider networks which adjust the signal outputs of the filter networks to some predetermined scale factor. The desired scale factors for the signals vary from channel to channel and also within any given channel as staging of the S-IC, S-II and S-IVB vehicles occurs.

Only one scaling network is required for the two lateral acceleration channels since these channels are operative only during the first 130 seconds of S-IC flight.

The attitude rate and attitude error channels have a scaling resistor corresponding to each stage.

Power Amplifier (Figure 20-45). The power amplifiers are magnetic servo amplifiers which receive, sum and amplify the attenuated, filtered, and scaled inputs and feed them to transistorized differential integrating amplifiers which provide high dc gain and attenuation of 800 Hz ripple. The output stage of the integrating amplifier is a low impedance differential driver which can provide up to 50 ma of current to the torque motor operated control valve of each hydraulic servo actuator of the gimbaled engines to provide the necessary thrust

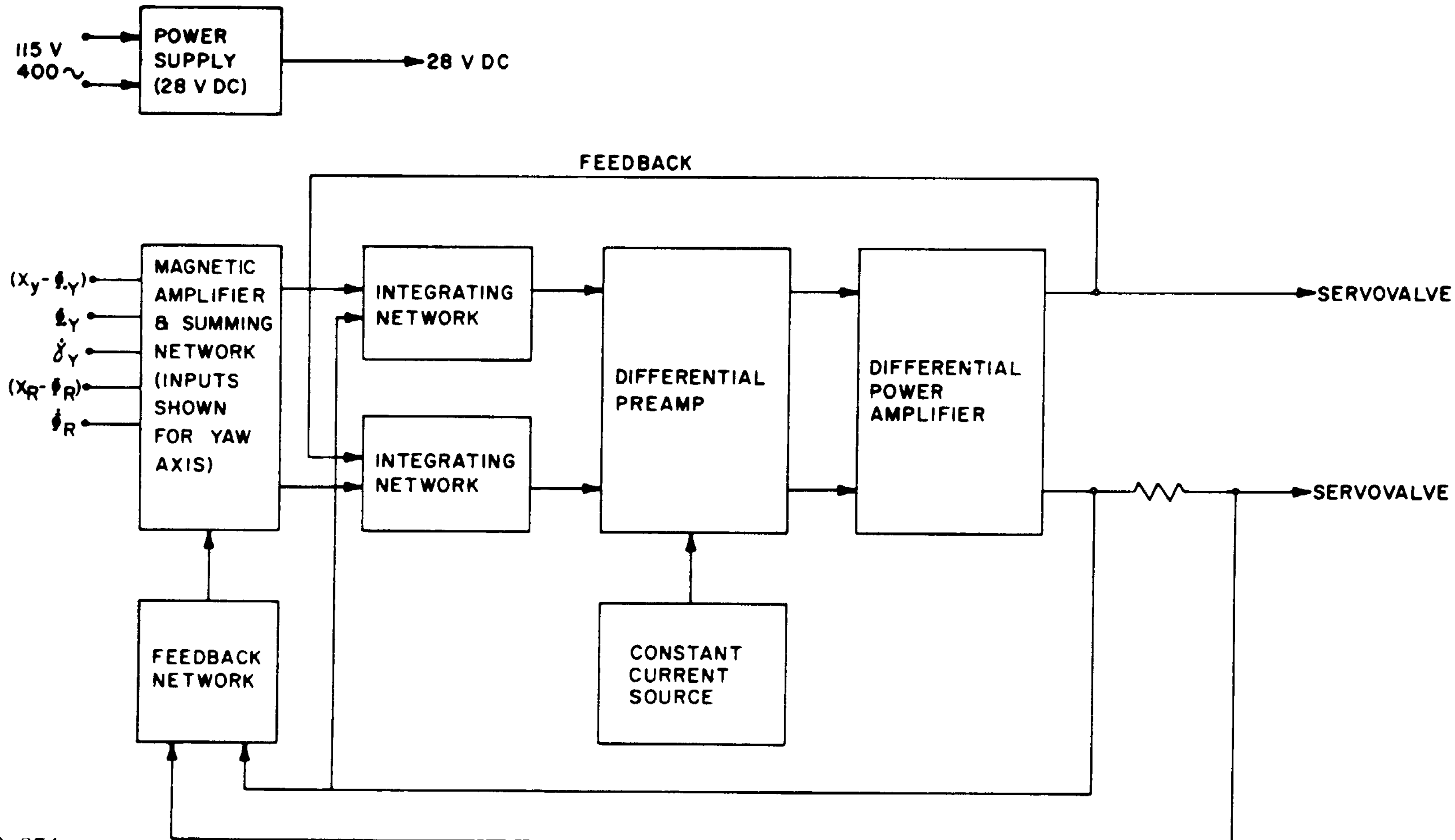

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Figure 20-45. Control Computer, Power Amplifier Block Diagram vector control.

A portion of this output current is returned to a feedback network to provide the proper closed loop gain and linearity accuracy.

There are eight power amplifiers in the control computer. However, it is not to be construed that there is one power amplifier associated with each of the eight signal channels. Eight power amplifiers are required because there are eight hydraulic servo actuators, two for each of the four gimbaled engines. Since each gimbaled engine has an actuator associated with the pitch and yaw axes, the eight power amplifiers are consequently divided into four pitch amplifiers and four yaw amplifiers. Since there are three channels at the control computer which process yaw signals (yaw attitude error, yaw attitude rate and yaw lateral acceleration) all three of these channels are sent to each of the four yaw amplifiers. A similar situation exists for pitch signals and the three pitch channels are sent to each of the four pitch amplifiers. There are only two channels that process roll signals, the roll attitude error channel and the roll attitude rate channel. To accomplish a given roll maneuver, it is necessary to gimbal any given engine in both the yaw and pitch axes. Thus, the two roll channels are sent to all eight power amplifiers. The summing networks at the input of the magnetic servo amplifiers must properly sum these various pitch, yaw and roll terms of the B thrust vector equation so that the output currents from the current drivers cause the solenoid valves to be actuated in the proper dirction or allow them to remain neutral. This allows the hydraulic servo actuators to gimbal the engines in the proper direction to obtain the desired thrust vector.

All eight power amplifiers are required for thrust vecor control for S-IC and S-II stages. As separation of the S-IC stage occurs, the outputs of the eight power amplifiers are relay-switched from the eight gimbal actuators of S-IC to the eight gimbal actuators of S-II. The S-IVB stage has one gimballed engine that requires two power amplifiers. Since eight power amplifiers are available, a triple redundancy scheme is used for control of the S-IVB stage. When separation of the S-II and S-IVB stage occurs, relays switch three pitch and three yaw power amplifiers into a triple redundancy and comparator configuration. The remaining two power amplifiers are not used during this phase.

20-79. Auxiliary Propulsion Control. In addition tc the eight channels that process the control signals for the thrust vector control of the gimbaled propulsion engine, the control computer also has six channels which provide on-off control of the S-IVB auxiliary propulsion system nozzles. The auxiliary propulsion system nozzles provide attitude control for the S-IVB/Apollo vehicle when in the powerless or coast phase, and roll-attitude control when in S-IVB/Apollo powered phases. The system consists of six nozzles mounted on the periphery of the S-IVB in two three-nozzle clusters. Two of the three nozzles of each cluster are used for both roll and yaw maneuvers and the other nozzle is used for pitch maneuvers only. (See Figure 20-46)

The auxiliary propulsion system (APS) channels of the control computer implement the equations

$$APS = a_0(x - \phi) + a_1\dot{\phi} \qquad \text{threshold level,}$$

and

$$\overline{APS} = a_0(x - \phi) + a_1\dot{\phi} \qquad \text{threshold level,}$$

where APS and $\overline{APS}$ are the on-off states, respectively, of the six APS noxxles. $(x - \phi)$, $\dot{\phi}$, a_0, and a_1 are as previously explained in part 1, and the threshold level is a value that is determined by special circuits within the attitude error $(x - \phi)$ channels of the auxiliary propulsion system section of the control computer. As previously explained, there are three $(x - \phi)$ and three $\dot{\phi}$ terms, one for each axis, so that the complete equation for APS or $\overline{APS}$ requires proper summation of six terms.

As shown in Figure 20-47, the six channels required for auxiliary propulsion system control are divided into two groups of three channels each. The first group contains channels for attitude error signal processing and the second group contains channels for attitude rate signal processing.

The attitude error channels consist of five sections, (1) attitude deadband, (2) relay switching, (3) limiter, (4) dc amplifier and (5) spatial amplifier. The attitude rate channels consist of three sections, (1) scaling resistor, (2) dc amplifier and (3) spatial amplifier. There are no filters associated with the six auxiliary propulsion system channels since bending moments and elastic deformations are negligible during the S-IVB/Apollo coast phases.

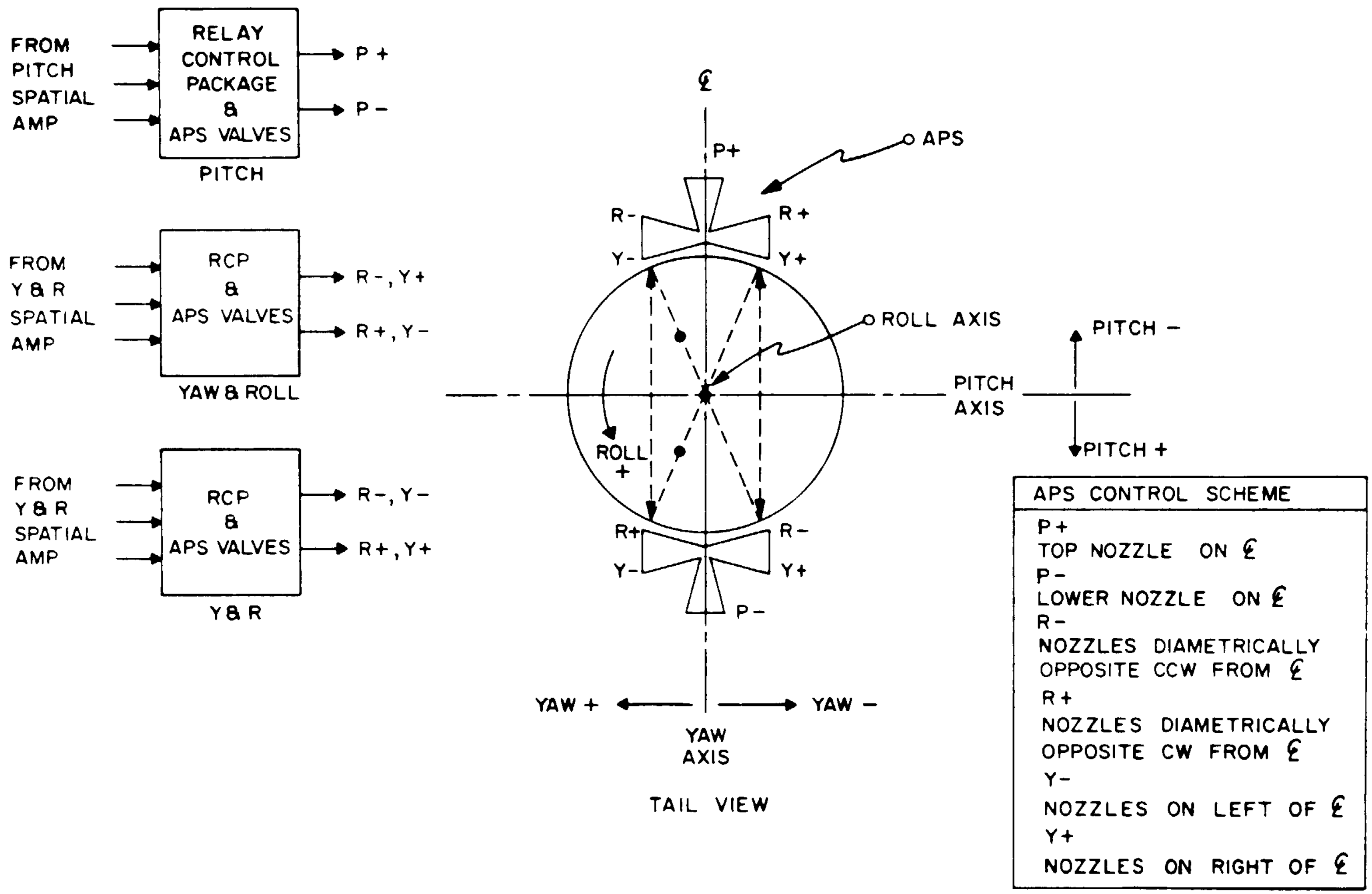

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Figure 20-46. S-IVB Auxiliary Propulsion System

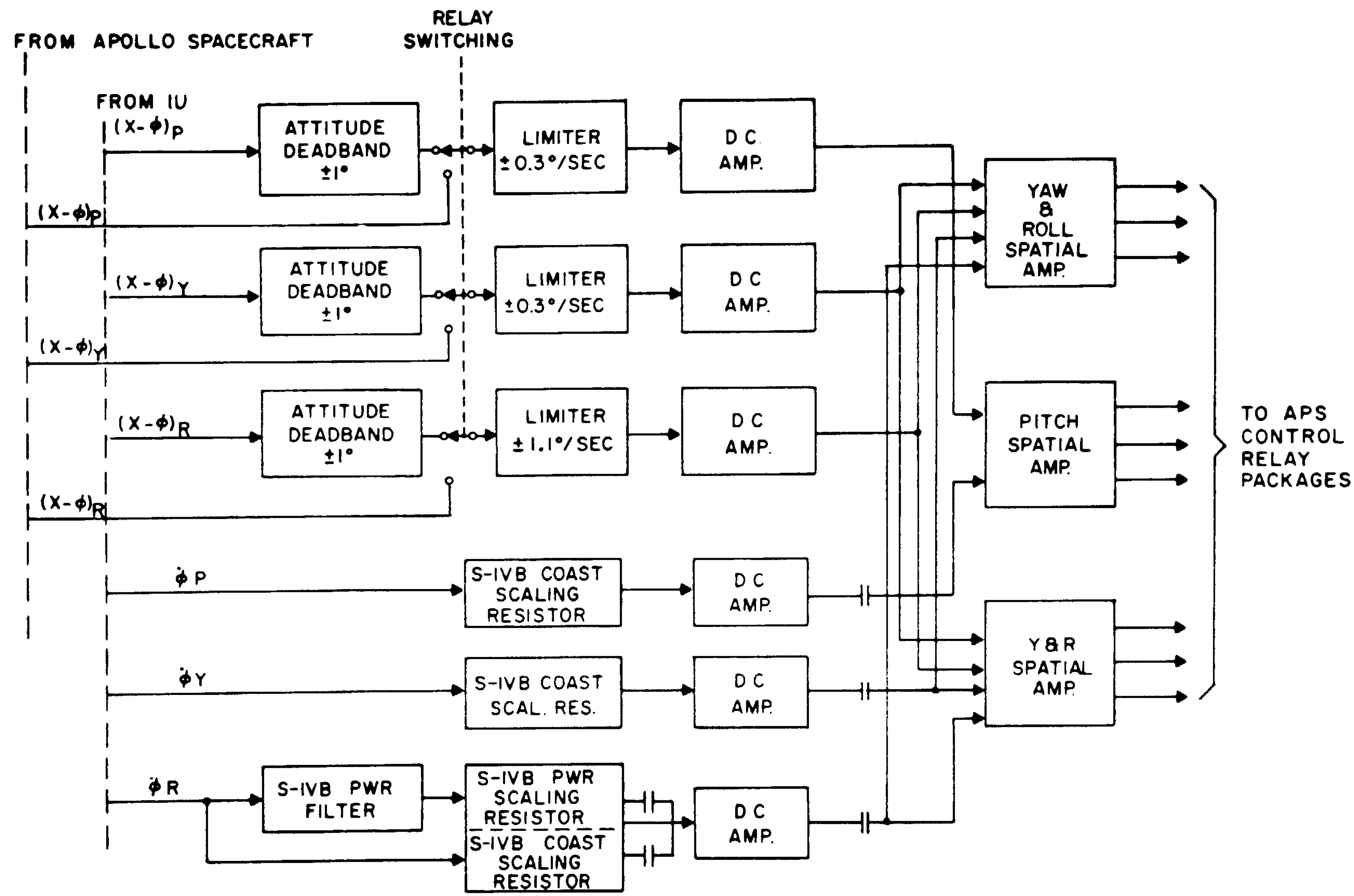

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Figure 20-47. Control Computer, Auxiliary Propulsion

The auxiliary propulsion system channels used for roll attitude error and roll attitude rate processing serve a dual function in that these are the only auxiliary propulsion system channels used during both the powered and coast phase of the S-IVB/ Apollo vehicle. Although the roll attitude rate channel requires both filter and scaling resistor for use in S-IVB powered phases, the roll attitude error channel does not require either, since the roll attitude error signal is always subject to attitude deadband processing. This is detailed below under Attitude Deadband.

The six channels of the auxiliary propulsion system section of the control computer process the six terms of the given equation and when their summation exceeds the threshold level, the appropriate auxiliary propulsion system nozzles are turned on so as to cause the error signal to be reduced. When the error signals fall below the threshold level the nozzles are turned off and the system awaits the next command.

Attitude Deadband. The attitude deadbands of the attitude error channels prevent attitude error signals, which originate in the IU digital computer, from reaching the limiters until these error signals have increased beyond a predetermined threshold level. Since the attitude error signal arrives at the deadband with a predetermined scale factor, the threshold level of the attitude deadband is set to the voltage that corresponds to an attitude error of ± 1 degree and no additional scaling is required.

The effect of this deadband is such that the vehicle attitude is corrected to with ± 1 degree of the commanded angle.

Actually, the attitude deadband is only one part of a composite deadband formed by the attitude deadband, limiter and additional deadbands within the spatial amplifiers. This composite deadband greatly reduces the fuel comsumption of the auxiliary propulsion system by eliminating fuel expenditures that would be required to correct for small attitude errors less than ± 1 degree. The composite deadband is discussed in greater detail under Spatial Amplifiers.

Relay Switching (Figure 20-47.) Relay switches at the input of the limiters provide for selection of an attitude error signal from either the digital computer in the IU or from one of the three attitude control systems within the Apollo capsule. When the relay is in the Apollo input position, the attitude deadband within the control computer is bypassed and the signal is presented directly to the limiter. Separate relay switching within the Apollo spacecraft inserts one of two deadbands which are part of the Apollo attitude reference systems. These deadbands are similar to those in the control computer with the exception that the deadband widths are ±0.5 degree for one and ±5 degrees for the other. The three sources of attitude control, available from the Apollo, are the Apollo digital guidance computer, the minimum impulse system and the rotational command system. The latter is a manual attitude control system which is discussed briefly in connection with the composite deadband under the discussion of the spatial amplifier.

Limiters (Figure 20-47.) The function of the limiter circuits is to limit the maximum rate at which the S-IVB auxiliary propulsion system may be commanded by a manually introduced error signal from the astronaut's hand control while in the Apollo rotational-control mode of operation. The limiters are designed to limit the amplitude of any error signal in the pitch and yaw channels to a voltage corresponding to a correction rate of 0.3 degree per second and to limit the roll channel to a voltage corresponding to a correction rate of 1.1 degrees per second.

When the S-IVB is in the Apollo rotational-command mode, a voltage from the hand control is applied to the spatial amplifiers. This turns on the appropriate auxiliary propulsion system nozzles, which remain on until the Apollo rate gyro feedback signals null out the hand control voltage to a value within the spatial amplifier deadband. Thus, the S-IVB is in a rate-controlled mode.

The limiters thus serve to conserve auxiliary propulsion system fuel by precluding the introduction of large angular rates which would require expenditure of excessive amounts of fuel.

DC Amplifiers. Each of the six auxiliary propulsion system channels contains a dc amplifier which receives the amplitude limited signals of the attitude error channels and the scaled signals of the attitude rate channels, amplifies them and sends the amplified signals to the spatial amplifiers. The dc amplifiers supply the signal power needed to drive the magnetic amplifiers within the spatial amplifiers. Scaling resistors are used in the attitude rate channels to scale the rate signals to values that allow proper summation of of these signals with the attitude error signals. Relay contacts in series with the dc amplifiers in the attitude rate channels prevent signals in the yaw and pitch channels from reaching the spatial amplifiers during S-IC, S-II and S-IVB powered stages. The relay contact in the roll channel is opened during S-IC and S-II and closed during S-IVB power or coast.

Spatial Amplifiers (Figure 20-48). There are nine spatial amplifiers associated with the six auxiliary propulsion system channels but only three amplifiers need be considered for a functional description of the auxiliary propulsion system. The remainder of the spatial amplifiers are used in a triple redundancy and comparator network for increased reliability.

The three spatial amplifiers receive the precessed attitude error and attitude rate signals, sum and amplify them in a magnetic amplifier, compare them to a threshold level in a deadband circuit and when the amplified summation exceeds the threshold, a dc switching amplifier energizes relays which operate the solenoid valves in the hypergolic propellant supply lines to the auxiliary propulsion system nozzles. A feedback network provides negative feedback to the mag-amp for additional damping to the control system.

The auxiliary propulsion system nozzles are either full-on or full-off, depending on whether the spatial amplifier is on or off. One spatial amplifier is used to control the two pitch nozzles, receiving as inputs the pitch attitude error and the pitch attitude rate signals. The remaining two spatial amplifiers control roll and yaw attitude and each amplifier receives roll and yaw attitude error, and roll and yaw attitude rate. This is necessary since the same nozzles are used for roll and yaw maneuvers. When roll maneuvers are required, two nozzles which are diametrically opposite are energized and for yaw maneuvers, two nozzles on the same side of the center line are energized.

The deadband within the spatial amplifier, along with the attitude deadband and limiter of the attitude error channels, form a composite deadband which is shown in Figure 20-49. The deadband is shown for pitch and yaw axes. The roll deadband differs from this diagram in that it has a maximum maneuver rate of 1.1 degrees per second. Point Po in the diagram represents an initial condition such as would be present at earth orbit insertion. This point is shown

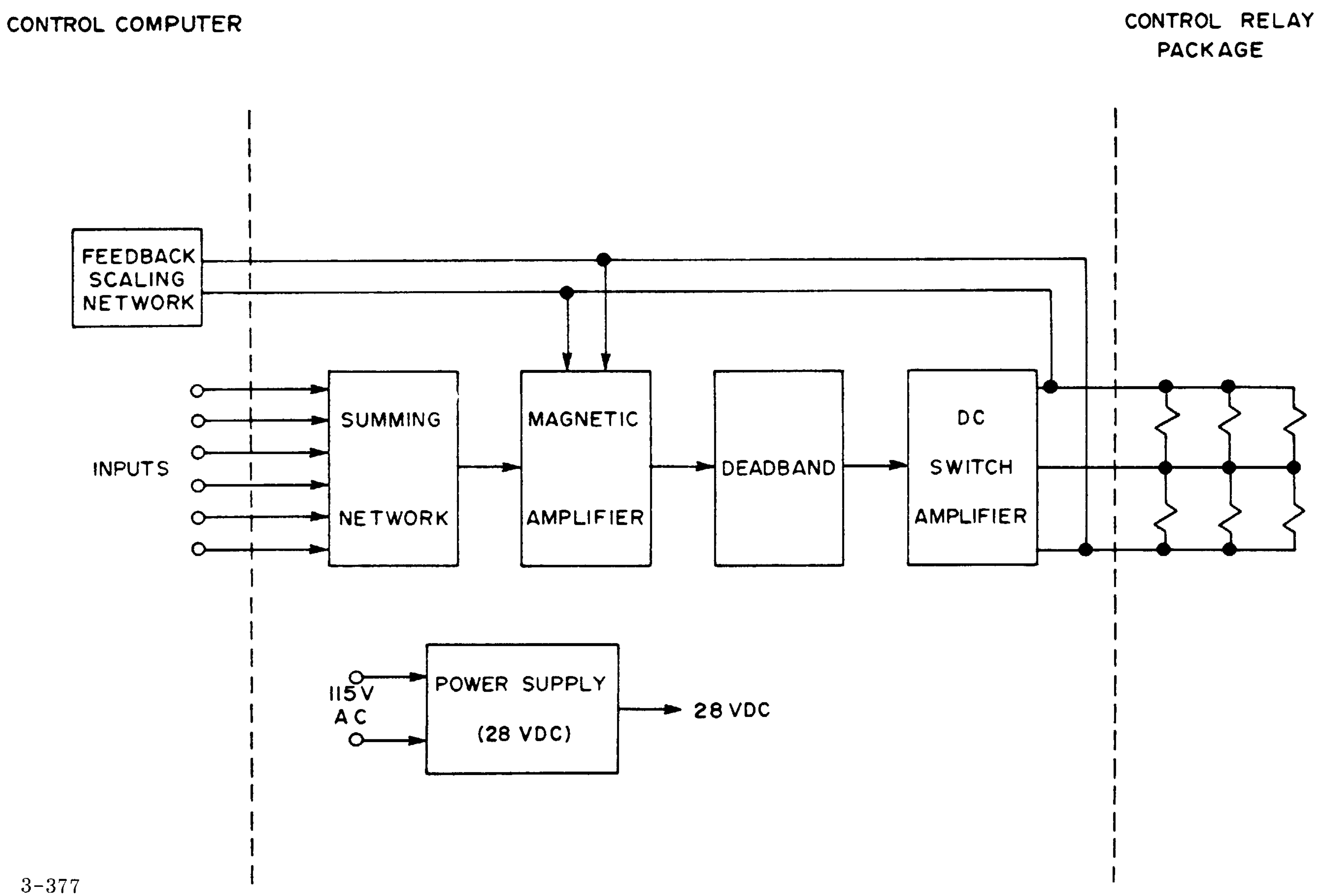

Figure 20-48. Spatial Amplifier, Block Diagram

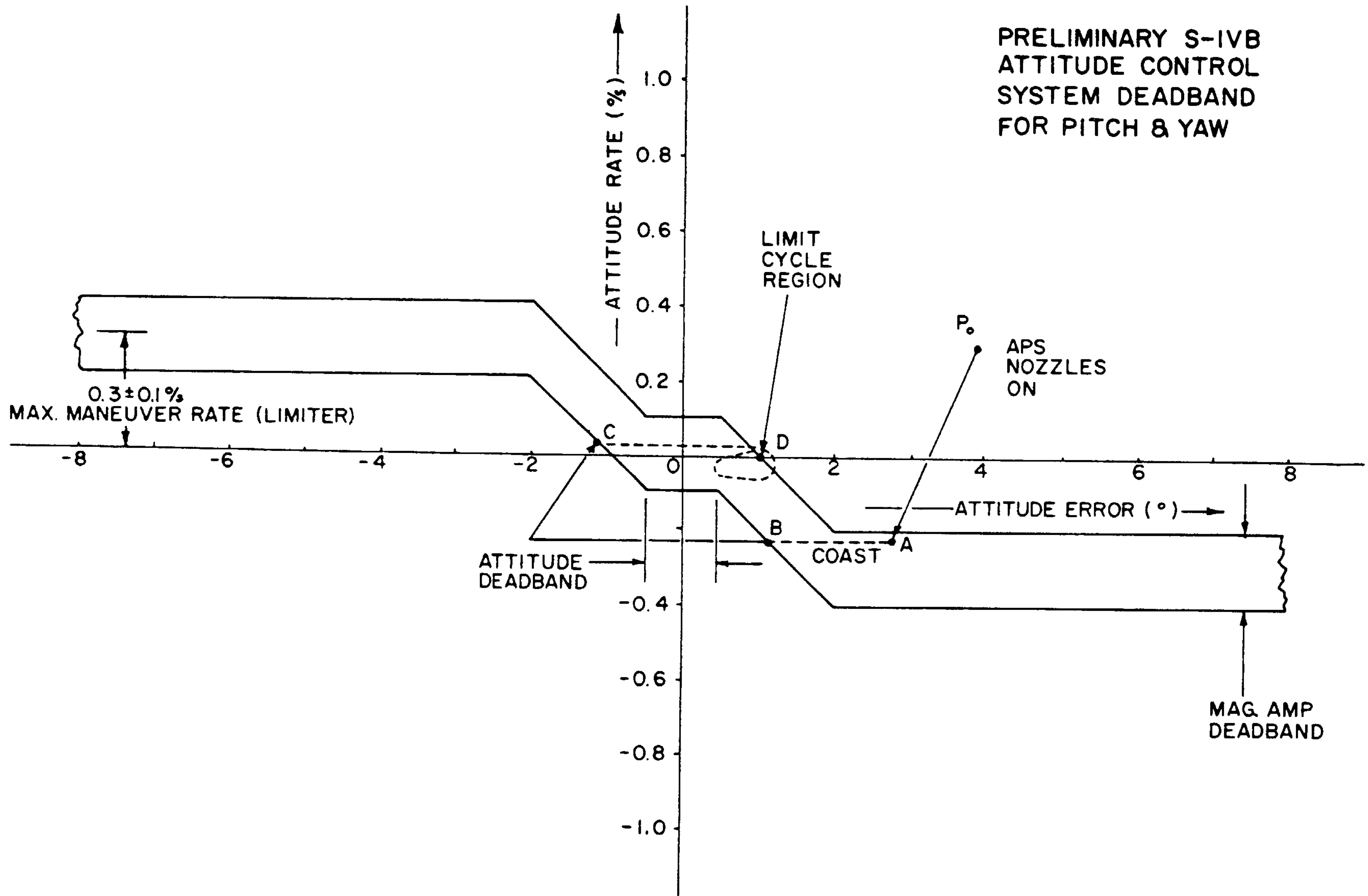

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Figure 20-49. Composite Deadband, Auxiliary Propulsion Control outside the deadband, that is, the attitude error signals would exceed the deadband thresholds; therefore, the spatial amplifiers turn on the proper auxiliary propulsion system nozzles which correct the vehicle attitude toward point A, where the nozzles are turned off (since the signals have fallen below the deadband thresholds). The vehicle continues to coast with a constant rate to point B where the error signals once again exceed the deadband thresholds and turn on the proper nozzles driving the vehicle toward point C. At point C the deadband is entered and the vehicle coasts at a constant rate to point D. The system limit-cycles about point D as shown, which is within one degree of the desired attitude.

When the auxiliary propulsion system attitude error channels are switched to the Apollo and the manual rotational command mode is selected, the attitude error signals are replaced by a voltage which is set by the astronaut's hand control. The auxiliary propulsion system channels turn on the control nozzles when this voltage exceeds the spatial amplifier deadband. The nozzles remain on until the attitude rate signal feedback reduces the input voltage below the deadband. The vehicle coasts at the rate established at cut-off until a new voltage is sent in by the hand control. In this manual control system, the auxiliary propulsion system limiter circuit-limits the voltage input to the spatial amplifiers to the equivalent of a correction rate of 0.3 degree per second in pitch and yaw, and 1.1 degrees per second in roll. When the hand control is returned to neutral, the system reverts to the limit cycle operation about the vehicle's attitude at that time. That is, the vehicle maintains within ± 1 degree of any attitude that is commanded by the astronaut's hand control. This attitude is maintained by one of the other two attitude control systems within the Apollo. When these systems are employed, attitude deadbands of either 0.5 degree or ± 5 degrees within the Apollo reference systems are inserted to replace the ± 1 degree attitude deadband in the auxiliary propulsion system channels of the control computer.

20-80. RATE GYROS

Attitude rate feedback, used in all phases of powered and coast flight of the Saturn V vehicle, is instrumented by single degree of freedom rate gyros which sense the angular velocity of the vehicle about the pitch, yaw, and roll axes. The rate gyro consists of a high inertia gyroscopic torque wheel, that is torqued by a motor stator mounted on the wheel. The precessional torque observed at the output axis in response to a torque on the input axis is balanced by the restraining force of a torsion wire. A microsyn differential transformer senses the torsion wire angular displacement which is proportional to the angular velocity about the input axis and therefore provides a voltage output signal proportional to angular velocity. The gyro is fluid damped. The full scale response of the gyro is 10 degrees/second and it uses a scale factor of 1-volt per degree per second. A heater is incorporated into each rate gyro because observed data shows that the mean time to failure of the rate gyro decreases for temperature less than -20 degrees F.

A built-in test capability exists within the rate gyro for testing spin motor rotational polarity and synchronization, and the output scaling factor. The rate gyro outputs are also used to detect excessive vehicle angular rates for the emergency detection system (EDS).

The use of rate gyros is necessitated by the significant bending modes that render differentiation of the angular displacement from the ST-124-M stabilized platform undesirable during S-IC and S-II powered flight. The elastic dynamics of the S-IVB stage do permit this differentiation but for functional redundancy purposes a source of angular rate independent of the stabilized platform was desired, and rate gyros were selected.

Multiple rate gyro packages fly in the Saturn V. Nonredundant packages are instrumented to sense angular rates about the pitch and yaw axes for use during S-IC and S-II powered flight. The locations and number used (two, three, or four) is determined after the vehicle bending characteristics are more fully defined. A triple redundant arrangement of nine rate gyros in the IU is instrumented to sense angular rates about all three axes. A triple redundant configuration was selected for the IU so that the S-IVB attitude control system would be ensured of increased reliability to compensate for the reliability degradation suffered with the relatively long operating times of orbital and translunar injection. The relatively shorter times during which the S-IC and S-II rate gyros are used did not dictate a redundant configuration for these stages.

The nonredundant rate gyro packages, consist of one regulator and inverter and one power supply for all three rate gyros. The rate gyro outputs are directed to amplifier and demodulation circuits and the demodulated output is sent to the control computer pitch and yaw attitude rate channels. The roll attitude rates, $\dot{\phi}_R$, from these rate gyro packages are not used. The $\dot{\phi}_R$ signals required by the control computer are obtained at all times from the triple redundant rate gyros in the IU.

The redundant rate gyro (control rate gyro processor), Figure 20-50, is packaged in two units. One contains the nine rate gyros and the other contains the associated electronics. The outputs of two of the three gyros, along each axis, are compared, and if a disagreement (failure) occurs, the active output of the redundant configuration is switched to the third (stand by) rate gyro.

An emergency detection system (EDS) has also been incorporated into the redundant rate gyro package to detect excessive rates All nine of the rate gyro signals are fed to electronic EDS rate switches which have been designed to energize relays in the emergency detection distributor when rates exceed preset values in the EDS rate switch. (Preliminary estimates of abort thresholds are ± 5 degrees/second in yaw and pitch and ± 10 degrees/second in roll.) The contacts of these relays for each axis are wired such that it requires two relays be energized in order that the EDS circuit be completed. Referring to the processor portion of the pitch axis on Figure 20-50, it can be seen that two relays 1 and 2, 2 and 3, or 1 and 3 are required to complete the circuit. K 3 contacts alone are prevented from completing the circuit by the addition of a diode between its contacts.

20-81. CONTROL ACCELEROMETERS

Angle-of-attack control in the launch vehicles is instrumented by two body-fixed accelerometers that sense translational accelerations ($\ddot{Y}_P$ and $\ddot{Y}_Y$) normal to the longitudinal axis of the vehicle. Q-ball, angle-of-attack transducers will continue to be flown in the Saturn V until the control accelerometers are fully proven in flight.

Angle-of-attack control has two purposes, one of which is to reduce the steady state drift normal to the nominal vehicle trajectory, which is "drift minimum" control. The second purpose is to minimize the bending moments on the vehicle structure by reducing the angle of attack and lateral component of thrust through dominant angle-of-attack feedback. This function is critical during the high dynamic pressure of the powered trajectory, when both dynamic pressure and high wind induced angles of attack are creating large aerodynamic forces in the vehicle structure. In the Saturn V vehicle, maximum aerodynamic pressure typically is reached at lift-off plus 77

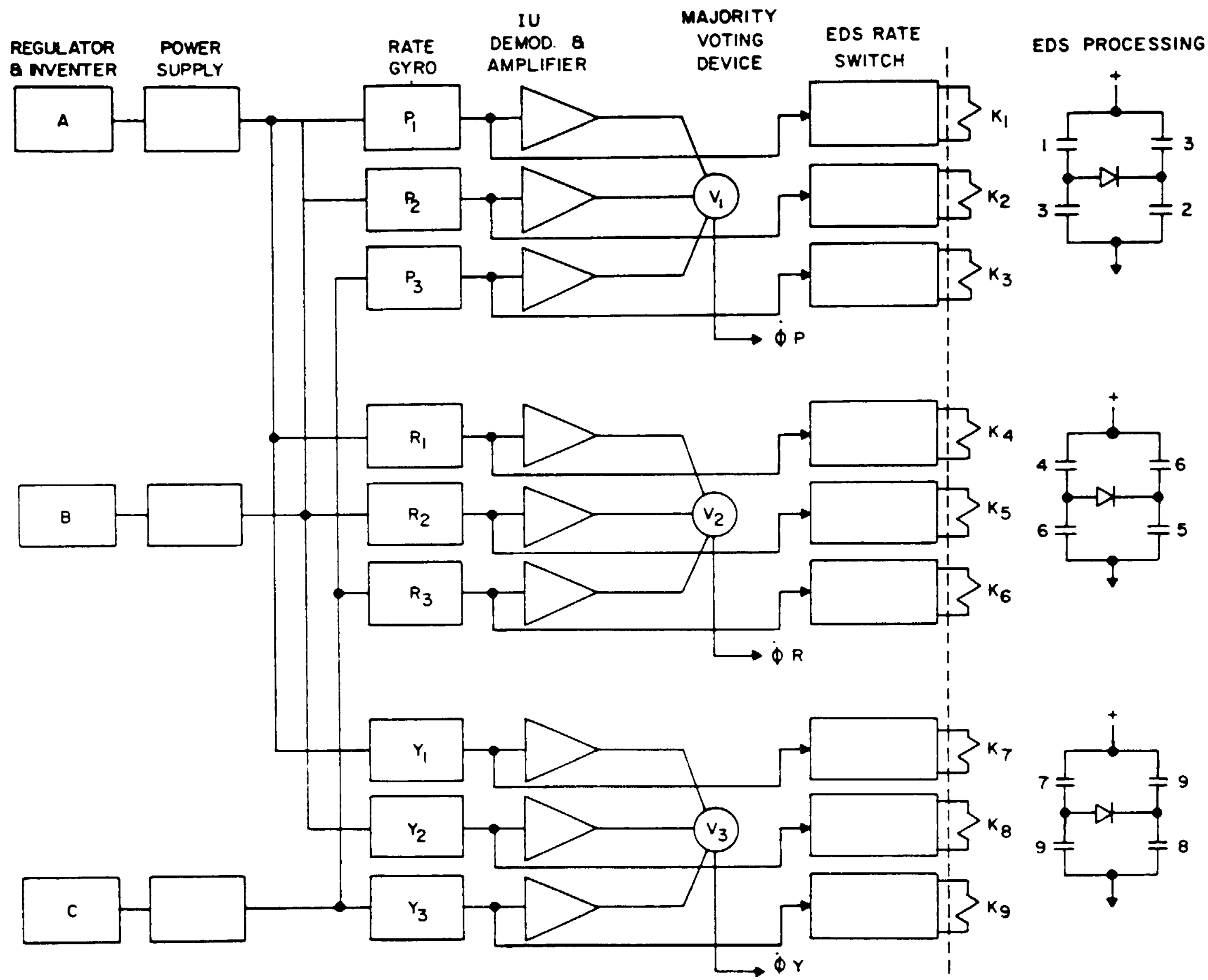

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Figure 20-50. Redundant Rate Gyro Package seconds, corresponding to an altitude of approximately 12,600 meters. A typical trajectory program employs the control accelerometers for 30 to 121 seconds after lift-off in the S-IC stage flight. Angle-of-attack control is not used after this time.

The control accelerometer designated for use in the launch vehicle is a linear, fluid-damped device with an inductive pickoff. The output of the accelerometer is a 400-Hz voltage that is proportional to sensed acceleration by a scale factor of 0.5 volts vrms per meter per second per second. This signal is amplified and demodulated by a circuit contained within the accelerometer package, producing an output voltage to the control system of 1 vdc per meter per second per second (or equivalently 2 volts dc per volt rms). The accelerometer package also contains a static inverter which converts 28 volts dc to 400 Hz 115 volts.

The control accelerometer contains an internal electromagnetic force coil which permits the displacement of the seismic mass by an amount proportional to the differential current applied to the force coil. The proposed design goal is an apparent acceleration of 1 meter per second per second for an input direct current of 20 ma. This force coil is used primarily for checking correct polarities in the thrust vector control system in the last minutes before launch. During prelaunch simulated flights, the control accelerometer channel accuracies are evaluated by tilting the control accelerometers at known angles to obtain fractions of 1g on the sensitive axis, and measuring the resulting angles of the gimbaled engines through readouts from the B feedback potentiometers in the servo actuators. The force coil offers potential automatic checkout capability, utilizing calibrated analog current inputs.

Figure 20-51 is a block diagram of the accelerometer and its associated electronics. Accelerometer specifications are given in Table 20-19.

20-82. HORIZON SENSOR

The purpose of the horizon sensor system is to provide onboard information in the form of pitch and roll attitude data for the S-IVB stage. The sensor system consists of four infrared sensors mounted in a base plane, parallel to the vehicle yaw plane and tangential to the skin of the vehicle IU at position I. The sensors are located in the base plane such that two sensors lie along the roll axis and the remaining two lie along the pitch axis. Each sensor may scan in a plane perpendicular to the base plane through an arc reaching from the base plane to 90 degrees away.

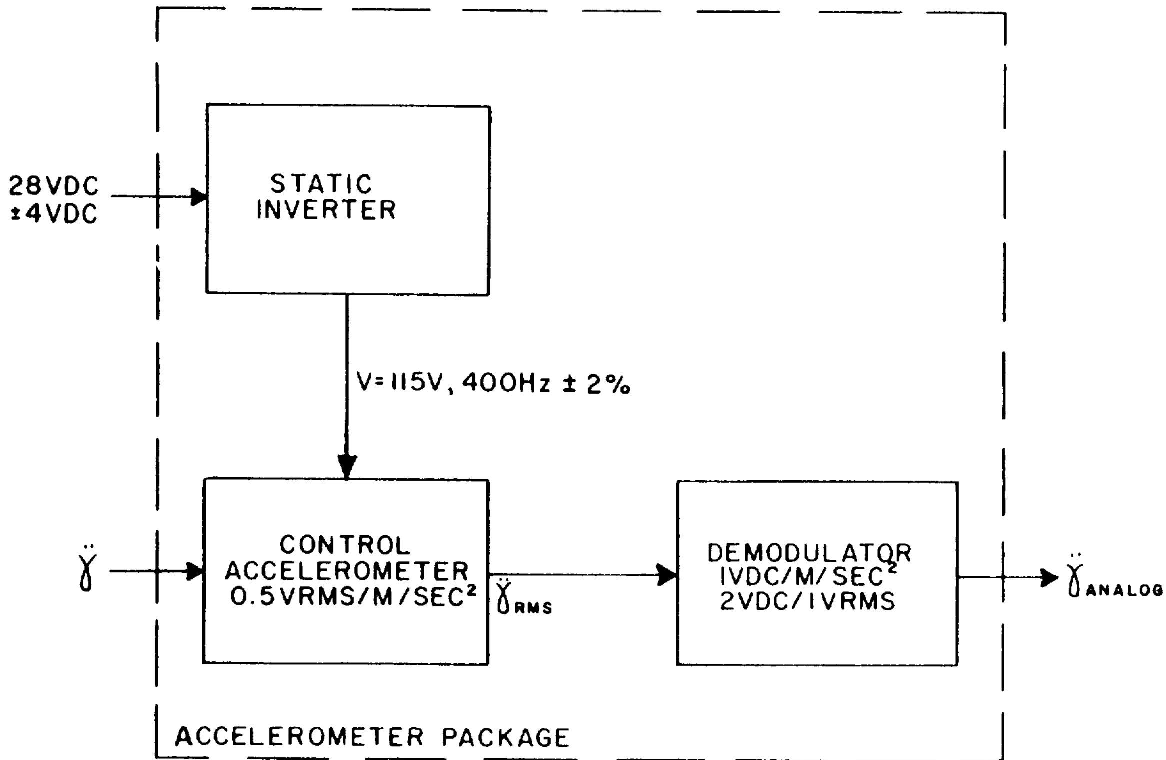

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Figure 20-51. Demodulator Block Diagram (Electronics)

The instantaneous field of view of each sensor is approximately 3 degrees wide by 0.5 degrees long. This is determined by the size and shape of the thermistor flake and the focal length of the optical system. The thermistor flake is cemented to the rear surface of the hemispherical lens giving the effect of immersion in the lens and thereby preventing energy losses which are inherent in an air spaced system. The thermistors operate in an oven at a controlled temperature of 60 degrees Centigrade. The oven is energized by the application of 28 vdc power, 20 minutes before launch of the vehicle. During preheat, the system is considered to be in the "stand-by" mode of operation.

Sensor scanning is accomplished by the rotation of a gold plated aluminum tracking mirror through a 45 degree arc, giving an effective scanning angle of 90 degrees. The tracking mirror is rotated by a dc torquer which is limited to 45 degrees of rotation. Tachometer feedback is used to regulate the rate of torquer drive.

The image reflected from the tracking mirror is reflected into the objective lens by a second mirror which oscillates through a three degree arc while "searching"

Table 20-19. Control Accelerometer Data

Item	Data
Accelerometer Sensor	
Range	$\pm$10M/sec/sec, stops at $\pm$12.5M/sec/sec
Natural Frequency	9Hz $\pm$1Hz
Cross Axis Sensitivity	0.002 g per g
Scale Factor	0.5 vrm/M/sec/sec
Force Coil Scale Factor	1M/sec/sec/20 ma
Voltage and Power	
Voltage Input to Inverter	28 vdc $\pm$4 vdc
Voltage Input to Accelerometer Sensor	115 vrms, 400 Hz $\pm$2%
Heater Power	50 watts (cyclic)
Accelerometer	10 watts
Inverter	
Demodulator	
Output from Accelerometer Package	
Scale Factor	1 vdc/M/sec/sec
Null	80 mv
Ripple	40 mv
Output Impedance	500 ohms
Physical Characteristics	
Weight	Approx 6 lbs.
Size	5 in. x 5 in. x 4 in.
Environmental Specifications	
Operating Temp.	-20^{o} F to 160^{o} F
Storage Temp.	-265^{o} F to 185^{o} F
Vibration	20_gss (20 to 2000 Hz)
Shock	50_g for 10 msec
Constant Acceleration	50_g

and a one degree arc while "tracking". The ocillating mirror is supported by flex pivots which give the assembly a natural resonant frequency of approximately 40 Hz. This assembly is driven electromagnetically by an oscillator which uses the mechanically resonant mirror assembly as its tuned element. A feedback coil provides a signal to the oscillator to maintain a constant oscillation amplitude. When a temperature gradient is present in the field of view it is swept back and forth across the thermistor at the 40 Hz rate, generating an ac signal.

During ascent through the atmosphere, the sensor system is protected from aerodynamic heating by a laminated fiberglass hemispherical dome approximately 14 inches in diameter at the base with a wall thickness of 3/8 inch. This dome contains an electrically heated artificial target which is used during ground checkout to test the ability of the sensors to detect a temperature gradient and lock onto it. The angular accuracy of this test is limited because of the possible variation in dome postion, and particularly because of the close proximity of the target and the sensors and the resulting poor focus of the optical system.

Approximately four to ten seconds after separation of the first stage, the protective dome is ejected to enable the sensors to receive radiant energy. At the same time the sensors are caused to enter the "search" mode of operation by application of power to the horizon sensor system. During the "search" mode, the tracking mirror on each sensor independently rotates at a rate of approximately five degrees per second, causing each sensor to scan through the 90 degree field in approximately nine seconds. The direction of scan is from space toward the earth to assure that the first appreciable gradient sensed will be the horizon. If the tracking mirror completes its 45 degrees of travel without bringing a gradient of sufficient magnitude into the field of view, the mirror is driven back to the initial position in approximately three seconds. During flyback, the sensor is prevented from backing on any temperature gradient that appear in the field of view. The "search" cycle of operation is continued until a gradient is detected. When a gradient of at least 175 degrees absolute remains in the oscillating field of view long enough to generate two or three cycles of the 40 Hz signal in the detector, the sensor switches to "track" mode and "locks" onto the temperature gradient.

Normally, once a gradient is detected, the sensor will continue to track it as long as the gradient remains within the 90 degree scanning arc. However, for the Saturn application, the sensor may be forced to recycle periodically regardless of whether or not it is tracking the horizon. This is to prevent the possibility of a sensor continuously tracking a false horizon and supplying incorrect data throughout the entire flight. The recycle signal of approximately 0.5 second duration is provided from the data adapter and is applied to only one sensor at a time so that three sensors may continue to track during this time.

The angle of the tracking mirror on each sensor must be transmitted accurately to the data adapter. This is accomplished by the use of an eight pole resolver driven by the mirror. Each resolver has a maximum electrical rotation of 180 degrees, corresponding to 45 degrees of tracking mirror rotation. The electrical signals from each resolver are carried by a calibrated cable, approximately 30 to 35 feet long, to the data adapter, where they are combined so as to multiply each electrical degree of resolver rotation by two. Thus, for each degree of mirror rotation there are eight degrees of electrical phase shift in the signal used in the adapter. The electrical phase shift of the signal from the resolver is converted in the data adapter to a digital number. This digital number defines the mirror position as an angle. This angle is the angle between the base plane and the temperature gradient which the sensor is tracking. A signal, indicating whether the sensor is in "search" or "track", also is sent to the data adapter. Theoretical analysis has indicated that roll and pitch attitudes can be computed with an error of less than 0.1 degree if a minimum of three sensors is tracking the horizon.

The horizon sensor system is packaged in a hermetically sealed unit, pressurized with dry nitrogen to an absolute pressure of approximately 25 psi. Four germanium windows, 4mm thick, pass the energy into the optical system. These windows have a transmission efficiency of approximately 80 per cent for energy in the 13 to 17 micron region. A special filter behind the objective lens limits the transmission of energy to wavelengths longer than 13 microns.

Cooling of the horizon sensor is accomplished by circulating coolant fluid through stainless steel coils cast in the base of the unit.

20-83. TRACKING

The tracking function provides accurate position and velocity information on the Saturn V launch vehicle from launch until injection of the Apollo spacecraft into its translunar trajectory. The function is then continued for the spacecraft, following its separation from the S-IVB stage/instrument unit.

Information derived by the tracking function is required for real time decisions supporting mission control and for postflight evaluation of the mission.

Real-time decisions requiring tracking data include:

a. Abort of mission for either range safety or crew safety
b. Selection of alternate mission for the flight
c. Updating of the vehicle guidance system prior to translunar injection
d. Over-riding of vehicle event sequencing (such as S-IVB engine cut-off)

For post-flight evaluation of the vehicle's performance, the tracking information is compared with computed data for the planned mission. From this comparison, and analysis of the differences, an insight is gained into the actual functioning of the vehicle systems in flight. Corrections may then be determined for future missions.

During the launch phase of the Saturn V mission, the tracking function is active for purposes of determining operational status of tracking systems, including both earth-based and vehicle-borne equipment. Additionally, reference data for each tracking system is obtained just prior to lift-off.

Continuous tracking information is required during the ascent phase for mission control Presentations based on tracking data are monitored by the range safety officer to aid him in deciding whether to terminate the vehicle flight to eliminate danger to personnel and property. Data available through continuous tracking are:

a. Accurate position and velocity at stage engine cutoffs and stage separations.

b. Confirmation that continued vehicle performance will permit accomplishment of the assigned mission, or that an alternate mission must be chosen.

c. Prediction of future positions of the vehicle to aid in transferring the tracking assignment from one station to another.

In the orbital phase of the mission, continuous tracking is required for a short period after injection into orbit, to verify that orbit conditions have been reached. Thereafter, periodic tracking observations are required to confirm or refine the predicted positions and velocities.

During the translunar phase, position and velocity information is required for real-time and post-flight evaluation of the vehicle trajectory. The real-time data is monitored by the Mission Control Center (MCC) to determine progress of the mission and may be used as a basis for initiating trajectory corrections through the command function.

To satisfy the tracking requirements, tracking stations have been established at selected locations around the earth to ensure that Saturn V vehicles can be tracked continuously from launch to orbital injection and that tracking data can be obtained periodically during the vehicle's orbits. Additionally, tracking stations have been established at locations such that at least one maintains tracking of the vehicle and spacecraft during transfer from earth parking orbit to lunar trajectory.

20-84. OPERATION.

Operation of the Saturn V tracking function is similar to Saturn I for launch, ascent and orbital phases of the mission. (Refer to Paragraph 6-47) During the translunar phase of the mission the tracking responsibility is transferred to deep space stations, which track the vehicle with radio frequency systems associated with equipment on the vehicle and spacecraft.

20-85. IMPLEMENTATION.

The tracking function is implemented with vehicle-borne equipment and a network of tracking stations interconnected with the MCC through high and low-speed communications systems. Radio frequency equipment at the tracking stations operates with vehicle-borne equipment to determine continuously the position and velocity of the vehicle. This position and velocity information is converted to a data format compatible with communications and computing facilities and then transmitted to the MCC.

20-86. VEHICLE IMPLEMENTATION.

Vehicle-borne equipment on the Saturn V is integrated with ground-based facilities to implement the tracking function. Tracking systems used with the S-IC and S-II stages and the instrument unit are:

a. ODOP

b. MISTRAM

c. Vehicle Radar Altimeter

d. C-Band Radar

e. AZUSA

f. MINITRACK

These systems are described below.

20-87. S-IC Stage - ODOP System. The offset doppler system (ODOP) is used for tracking the S-IC stage of Saturn V, beginning with vehicle SA-504. (The system will have been flown on the instrument unit of vehicles SA-501 through 503.) This system, a continuous-wave radio frequency system, consists of the ODOP transponder aboard the vehicle and ground facilities comprising a transmitter site and at least three receiving sites. Vehicle position and velocity are computed from doppler measurements performed at the three receivers.

The system employed for Saturn V is similar to that used on Saturn I. The operating frequency on the Saturn V is to be in the S-band (2200 to 2400 mc) rather than the 900-mc band used on the Saturn I ODOP. Refer to paragraph 6-53 for a more detailed description of ODOP.

20-88. S-II Stage - MISTRAM System. Tracking of the S-II stage of Saturn V vehicles will be accomplished by the missile trajectory measurement system (MISTRAM). This continuous wave system comprises ground facilities operating in conjunction with the vehicle-borne MISTRAM transponder to determine position and velocity of the vehicle in real time.

The description of MISTRAM for Saturn I vehicles is applicable to the S-II stage tracking system. Refer to paragraph 6-54.

20-89. S-IVB Stage. Since the S-IVB stage and instrument unit do not separate from each other, tracking systems used with the instrument unit also determine position and velocity of the stage. No tracking equipment is carried by the S-IVB stage.

20-90. Instrument Unit. Tracking systems implemented on the Saturn V instrument unit are: radar altimeter, C-Band radar, AZUSA/Glotrac and Minitrack. Additionally, the ODOP system is being used in the instrument unit of developmental vehicles SA 501, 502 and 503. The instrument unit tracking systems are described below:

Radar Altimeter - The high-altitude altimeter, a pulsed radar system, determines distance of the vehicle above the earth by measuring the time for a pulse to travel from the vehicle to earth and back to the vehicle. It is designed to provide the altitude element of tracking information for portions of the vehicle orbit when it is not visible to tracking stations, such as during passage over ocean areas. Data obtained by the altimeter is digitally encoded for telemetering to the ground, and may be tape-recorded for playback when the vehicle is in sight of a ground station. The radar altimeter is described in more detail in the tracking function description for Saturn I. Refer to Paragraph 6-56.

C-Band Radar - To support ground-based AN-FPS-16 radar tracking systems, an SST-102A transponder is carried on the Saturn V instrument unit. This transponder supplies a high-power (500 watt) radar return in response to pulse interrogations from the groun-based radar. A further description of the transponder and C-Band radar tracking is given in Paragraph 6-55. (Saturn I tracking function)

AZUSA/Glotrac - The AZUSA transponder carried aboard the Saturn V instrument unit aids tracking the AZUSA/Glotrac system. A description of the transponder and AZUSA tracking systems is given in the discussion of Saturn I tracking function. See Paragraph 6-52. The Glotrac application is described below.

20-91. GLOTRAC Tracking System. Glotrac uses information from existing continuous wave (AZUSA MK II) and pulse radar systems, as well as newly developed range and range rate equipment, to make high-accuracy measurements of target velocity and position. It was originally planned as a global instrumentation system and derives its name from "global tracking." Range rate measurements are performed by a ground-based transmitter which interrogates a vehicle-borne transponder. The transponder offsets the received signal and re-transmits it to ground-based receiving stations so that a range rate sum is obtained by comparing the frequency of the received signal with a local frequency. Range is obtained at the transmitting site by counting the total cycle difference between the transmitted and received signals.

Tracking begins after the stations receive antenna-pointing information from radars located on the Cape, optical trackers, or other external sources. The modified AZUSA MK II measures range, angle, angle rate and range rate. Remote stations at Cherry Point, Bermuda, Grand Turk, and San Salvador measure range rate. The San Salvador and Bermuda stations have transmitters, and a Bermuda receiver measures non-ambigious continuous wave (cw) range information. Pulse radar systems located at San Salvador and Antigua measure radar range, azimuth, and elevation which are also used in acquiring Glotrac tracking information. At launch, a vehicle is tracked by the modified AZUSA MK II at Cherry Point, Bermuda, and San Salvador as line-of-sight permits. As the vehicle moves downrange, the AZUSA MK II transmitter is shut down and the Bermuda transmitter is activated. The three range rates, measured by the rate stations at Grand Turk, Antigua, and Bermuda, yield the velocity solution, while the radar ranges measured at Grand Turk and Antigua, plus the cw range measured at Bermuda, yield the initial condition information for integration of range rate information. The information measured at all stations is transmitted to the computer at Cape Kennedy for use in updating the pointing information, and to Goddard Space Flight Center for trajectory computation. The instruments used for Glotrac include the AXUSA MK II at Cape Kennedy and pulse radars at San Salvador and Antigua as well as the following:

Transmitter and Range Rate Station (San Salvador). This equipment, which is used only with Glotrac, consists of a 5-kw transmitter and a doppler receiver. The transmitter operates on either 5060.194 mc or 5052.0833 mc and feeds a 5-foot antenna, which can be slaved to a local tracking instrument. The transmitter frequency is controlled by an atomic frequency reference. This frequency is also used for a coherent reference in the range rate receiving section. The range rate equipment is identical to that described in the next paragraph.

Range Rate Station (Cherry Point, North Carolina: Antigua, and Grand Turk). The 5000 mc$\pm f_d$ (doppler frequency shift) is received on a 5-foot antenna and fed to a crystal mixer through a parametric preamplifier. These signals are mixed with 5040 mc signals from the frequency synthesizer (frequency controlled by an atomic clock) to produce a 40 mc IF output. A 34 mc variable frequency oscillator is phase-locked to the 40 mc IF output. The output of the 35 mc variable frequency oscillator contains the phase rate information. The variable frequency osicillator is part of the correlator which includes a servo composed of a mixer, a 5 mc IF amplifier with crystal filter, and a phase detector. The output of the 35 mc variable frequency ocillator contains the doppler tracking data in a 100 cycle bandwidth. The 35 mc data output is heterodyned to 5 mc and applied to a quadrature phase detector which provides four-fold multiplication of data resolution. The multipled phase data is applied through a bidirectional counter which reads out to a magnetic tape recorder at a rate of 10/sec and to a digital-to-teletype converter for real-time transmission at a rate of 1 sample every 6 sec. The range rate data is transmitted at 24 bits/sample and the range data at 10 bits/sample.

The antenna can be used for conical scan and can be slaved to azimuth and elevation acquisition data received by teletype or synchro.

Range Rate Station with Range Module (Bermuda). At the Bermuda station a range module is added to provide circuitry for detecting 98.3569, 4 and 0.160 kc range modulation signals that measure unambiguous range as well as range rate.

The frequency-modulated 5000 mc signals received from the transponder are heterodyned successively to 40 mc and to 5 mc. The FM signals are amplified and applied to a coherent demodulator. The other input to the demodulator is a phase-adjustable 5 mc reference signal. The phase of the reference signal is adjusted to compensate for phase errors arising from transmission through the 5 mc amplifier. When the input signals are properly phased, the demodulator output is the 98.3569 kc range modulation signal.

Since phase-shift errors can be introduced into the range data from various components in the range channel, it is desirable to detect the range data as close to the antennas as possible. This is achieved by an autocorrelation wipe-off technique. The microwave local-oscillator mixing signal is frequency modulated in the frequency synthesizer at a modulation index almost equal to the modulation index of the 98.3569 kc signal on the 5000 mc carrier. This effectively detects the range data at the microwave crystal mixer.

20-92. Minitrack. The minitrack beacon on the Saturn V instrument unit transmits a low-power continuous wave signal. This transmission from the vehicle provides a point source of energy for earth-based tracking stations which determine direction cosines of the vehicle with respect to their antenna baselines as a function of time.

Position and velocity data is computed from angle and time information derived from a sequence of tracking stations.

20-93. GROUND STATION IMPLEMENTATION

Station and facilities of the network used for tracking of the Saturn I vehicles are a part of the tracking network for Saturn V. (Refer to paragraph 6-57.) A world-wide network of stations, listed in Table 20-20, track the vehicle Minitrack beacon. Locations of stations in this network ensure that at least one station is in line of sight of the vehicle on each orbit. In addition to those ground stations, used primarily for launch, ascent and orbital phases of the Saturn V mission, deep space stations perform the tracking function during the translunar phase. The deep-space stations are located at Goldstone, California; Johannesburg, South Africa; and Woomera, Australia. Coverage provided by these stations, as a function of vehicle altitude is shown in Figure 20-52. The deep-space stations interface with the Saturn V Minitrack system and with radio frequency equipment aboard the Apollo spacecraft.

Table 20-20. Minitrack Stations and Locations

Stations	Longitude	Latitude
Antofagasta, Chile	$289^{\circ}43'$ 36.838"E	$23^{\circ}37'$ 15.993"S
Fairbanks, Alaska	212-09-47.387E	64-52-18.591N
Blossom Point, Md.	282-54-48.170E	38-25-49.718N
East Grand Forks, Minn.	262-59-21.556E	48-01-20.668N
Johannesburg, S. Africa	027-42-27.931E	25-5258.862S
Lima, Peru	282-50-58.184E	11-46-36.492S
San Diego, Calif.	243-01-43.707E	32-34-47.701N
St. John's, Nfld.	307-16-43.240E	47-44-29.049N
Woomera, Australia	136-46-59.52E	31-06-09.49S
Antigua Island, BWI	298-13-16.536E	17-08-32.586N
Quito, Ecuador	281-25-14.770E	00-37-21.751S
Santiago, Chile	289-19-51.283E	33-08-58.106S
Winkfield, England	359-18-14.615E	51-26-44.122N
Fort Myers, Fla.	278-08-03.887E	26-32-53.516N
Goldstone Lake, Calif.	243-06-02.776E	35-19-48.525N

To be supplied at a later date.

Figure 20-52. Deep Space Tracking Network, Saturn V

20-94. CREW SAFETY (VEHICLE EMERGENCY DETECTION SYSTEM).

The crew safety function ensures safety of the spacecraft crew in the event of malfunction of the Saturn V/Apollo space vehicle. The function provides for sensing and display of performance parameters, to enable the crew to initiate an escape sequence if an emergency occurs. It also provides automatic initiation of the crew escape sequence for emergencies not permitting time for initiation of the escape sequence by the crew.

Safety of the crew aboard the Apollo spacecraft is of major concern in the planning of Apollo/Saturn missions as well as the design of both the launch vehicle and the spacecraft. It has been recognized in planning for earth-orbital and lunar missions that contingencies may arise requiring either pursuit of alternate missions or abandonment of the mission in process. The choice of action depends on the nature of the contingency.

In planning for contingencies, possible malfunctions have been examined in relation to their effect on the vehicle and mission. Failures leading to loss of the vehicle are classified as "catastrophic" if they would result in vehicle breakup in a time (from first indication of failure) insufficient for crew initiation of the escape sequence. Failures leading to loss of the vehicle are classified as "critical" if the time-to-vehicle breakup permits crew initiation of the escape sequence. Failures which may not result in vehicle loss, but may result in alteration or delay of the mission are classified as "non-critical".

Contingencies which involve either catastrophic or critical malfunctions in the spacecraft/launch vehicle must be detected as early as possible and announced to the crew to provide them with adequate warning if it becomes necessary to abort a mission. Catastrophic malfunctions require automatic initiation of the escape sequence to ensure survival of the crew. Malfunctions considered to be catastrophic include: excessive turning rate in yaw, pitch or roll; structural failure and multiple engine failure in the early moments of flight. Parameters for which information is displayed to permit crew decision for manual initiation of the escape sequence are: Status of engine thrust on stages, staging operation, digital computer status, angle-of-attack, fuel container pressure (S-II and S-IVB stages), spacecraft attitude error and angular rates, and range safety engine cutoff.

In this discussion "abort" is defined as the sequence of separating the command module (CM) from the space vehicle and bringing it safely back to ground. Constraints imposed on the abort procedures for various stages of flight are illustrated in Figure 20-53. During the first 10 seconds (approximately) of flight, an abort can be initiated either manually or automatically. In this time interval the failure of two or more engines makes abort mandatory. After the first 10 seconds, an abort due to engine failure will be manual or automatic at the option of the crew. During the first 40 seconds (approximately) of flight, an abort will not initiate engine shutdown. After this period, engine shutdown will be initiated as part of the abort sequence.

20-95. OPERATION.

The crew safety function is accomplished by integration of several functions, including range safety, communications, command, the spacecraft emergency detection system (VEDS), the launch escape system (LES), and the crew.

When the vehicle flight is terminated for range safety purposes, an engine cutoff warning is displayed in the crew compartment at least five seconds prior to initiation of ordnance which ruptures propellant containers on the vehicle stages. (Refer to Range Safety, Paragraph 20-99.). This delay gives the crew sufficient time to initiate the escape sequence.

The crew receives continuous information on performance of the launch vehicle and the spacecraft through displays in the CM and communications from earth. The crew monitors and assesses this information, and decides whether to abort the mission. They are aided in decision-making by the displays and by operational personnel on earth, who continuously monitor vehicle performance data through tracking and instrumentation functions and maintain voice communications with the crew.

In the event of an abort decision on the launch pad or during first stage boost the launch escape system (LES) is activated by either the crew or the Mission Control Center. The LES is automatically programmed to place the CM on a safe escape trajectory. For abort decisions after jettison of the LES, the abort maneuver may require use of service module (SM) propulsion, and action similar to the re-entry procedure normally used after completion of a mission.

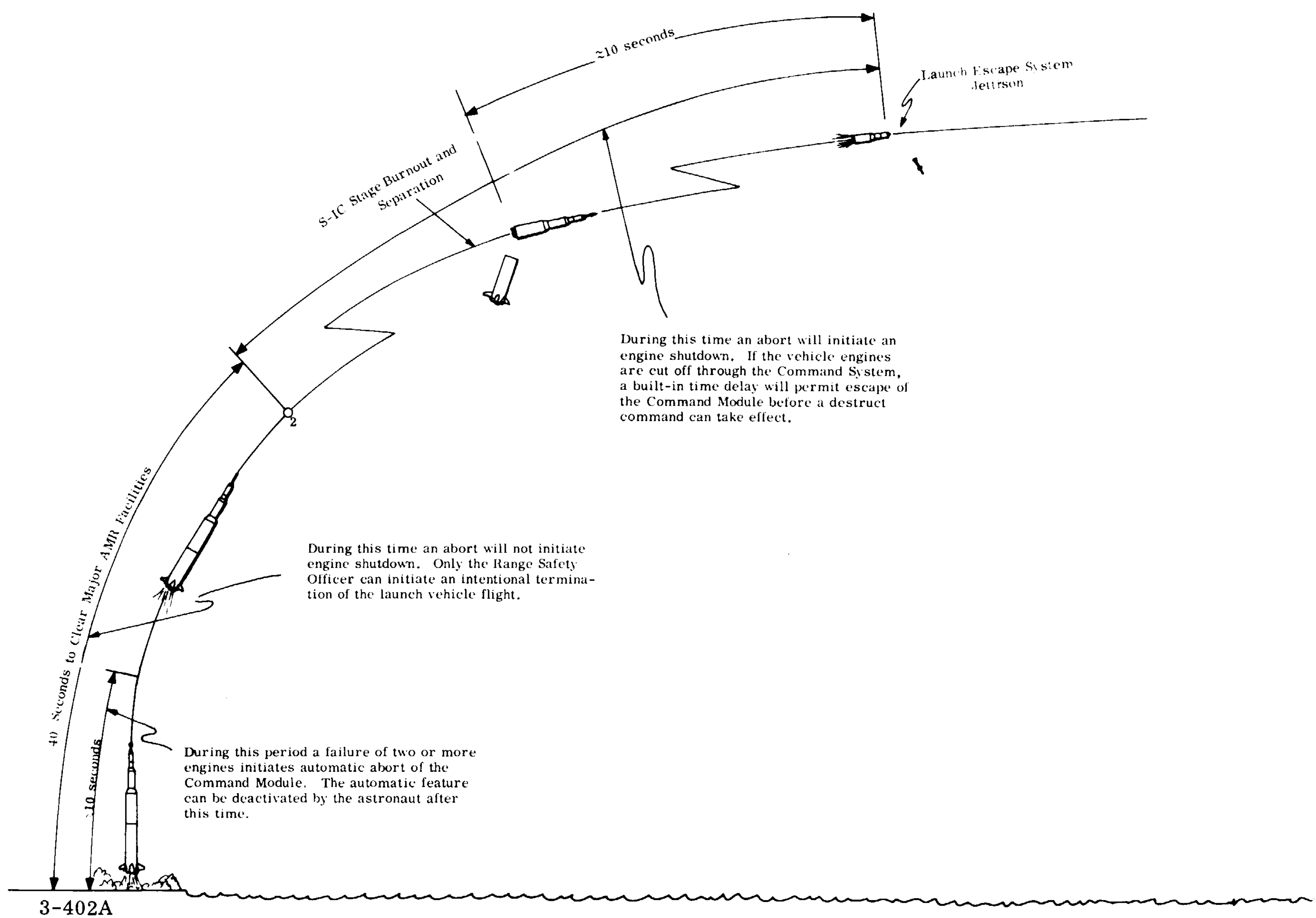

Figure 20-53. Abort Procedure Constraints, Saturn V

An abort sequence is initiated by one of several methods:

a. By wire link prior to liftoff
b. By radio link prior to and after liftoff
c. By the crew after a certain point in the countdown
d. By the vehicle emergency detection system (VEDS) after liftoff.

The crew safety system aboard the space vehicle consists of sensors and test equipment to detect and diagnose malfunctions, and displays to permit the crew to make a reasonable assessment of contingencies. The escape sequence is automatically initiated by the VEDS for certain emergencies, such as those described in Paragraph 20-94.

20-96. IMPLEMENTATION

The launch vehicle portion of the crew safety system is shown in Figure 20-54. This illustration and the discussion following are based on the VEDS concept, since the design is in its beginning stages.

The VEDS consists of stage-mounted sensors and a distributor in the instrument unit, which transfers vehicle performance information to display equipment in the CM. The distributor delivers an abort signal to the CM if a catastrophic malfunction occurs, and transfers engine cutoff signals to the active stage when an abort is initiated. To enhance reliability of the VEDS, power requirements for the system are supplied from two primary sources on each stage and the instrument unit. Isolation of sources is provided by diodes.

20-97. Automatic Abort. Parameters for automatic abort of Saturn V mission are structural failure, excessive vehicle turning rate in yaw, pitch or roll, (vehicle overrate), and loss of thrust of two or more engines on the S-IC stage in the early moments of flight. These parameters are sensed by hot wire structural monitors throughout the vehicle, rate gyros in the instrument unit, and thrust OK switches on the S-IC engines.

Structural monitoring is accomplished by installation of "hot" wires from logic circuitry in the instrument unit down through the S-IC stage and up through the LEM and SM to the CM. Three circuits in each of three separate geographical paths are

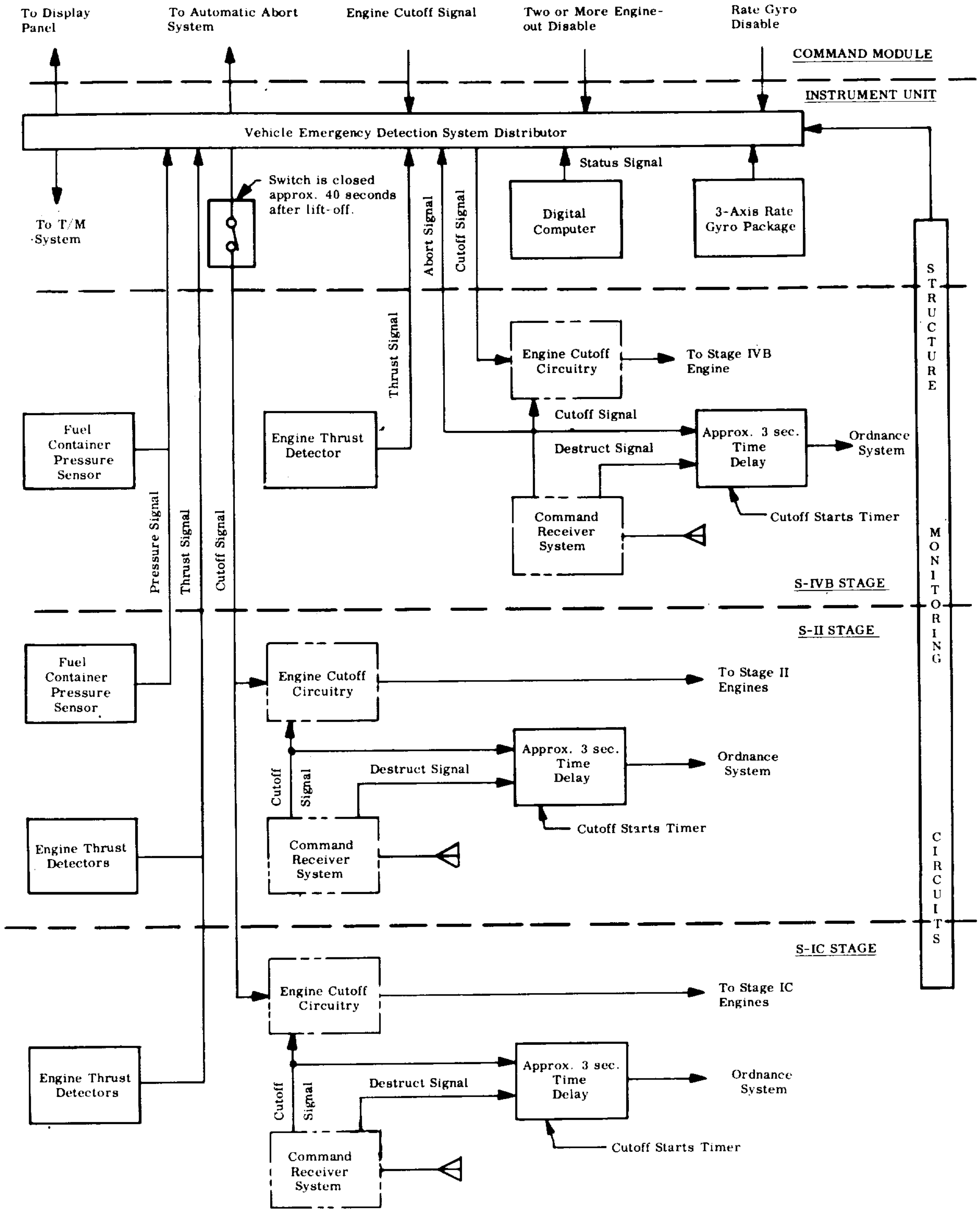

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Figure 20-54. Vehicle Emergency Detection System, Saturn V monitored for electrical continuity to indicate structural integrity of the vehicle. Loss of electrical continuity in two out of three circuits in any geographical path automatically initiates an abort.

Vehicle rates are sensed by a gyro package containing three rate gyroscopes for each plane (pitch, yaw and roll). Abort is automatically initiated and the overrate indicating light in the CM is energized when two out of three gyros in a plane indicate that the limit is being exceeded. Limit settings are: five degrees per second, pitch and yaw rates; 40 degrees per second, roll rate. A switch in the CM permits crew deactivation of the overrate automatic abort signal, after which the overrate indicating light provides information as a basis for manual abort. Deactivation of the automatic abort circuitry also is accomplished within the Saturn V launch vehicle. The vehicle computer provides the stimulus for deactivation of the automatic abort circuits.

Thrust OK switches on the five engines of the S-IC stage provide inputs to logic circuitry which give an automatic abort signal when thrust is lost on any two of the five engines. Outputs of the thrust OK switches are also used to display engine status on indicating lights in the command module. A light is energized when thrust is lost by its associated engine.

20-98. Manual Abort

Parameters monitored for the manual abort decision are vehicle and spacecraft rates, engine status, staging sequence, computer status, angle of attack, fuel container pressure, spacecraft attitude, and range safety engine cutoff.

Vehicle angular rates, when they exceed pre-set limits in any plane, cause an indicator to light in the command module. After deactivation of the overrate automatic abort circuitry, this indication and the analog display of spacecraft angular rates on the CM flight director attitude indicator are monitored to determine whether to initiate abort manually.

Thrust OK switches on each engine of the S-IC, S-II and S-IVB stages, provide engine status information which is displayed by lights in the spacecraft. Loss of thrust by an engine causes the associated light to be energized. The crew initiates an abort manually in accordance with rules established for the mission in event of engine failure.

Engine status lights also are used to provide information on the staging sequence. Since engine starting of S-II and S-IV-B stages is interlocked with physical separation of the previous stage, failure to separate gives the same end result (engine status lights on) as the engine out condition.

A status light in the spacecraft indicates improper operation of the launch vehicle digital computer. Additionally, excessive deviation of vehicle attitude from its required (or computed) attitude energizes a light to indicate improper operation of the vehicle control computer.

Angle of attack is displayed in analog form as a parameter for manual abort. The decision to abort depends on magnitude of the angle of attack and other existing aerodynamic considerations (e.g. altitude, air speed).

Fuel container pressures in the S-II and S-IVB stages are also presented as analog displays in the spacecraft.

Attitude error, as determined by the spacecraft guidance and navigation system is displayed on the flight director attitude indicator in the spacecraft. During S-IC flight, the indicator is programmed with the launch vehicle tilt program.

When engine cutoff is commanded for range safety purposes, a warning signal is delivered to the spacecraft. This signal is provided by the engine cutoff channel of the two command receivers aboard the S-IVB stage and is transmitted to the spacecraft through the distributor in the instrument unit.

20-99. RANGE SAFETY.

The range safety function ensures safety of the launch range and adjacent areas against malfunction of Saturn V vehicles launched on the range.

Requirements and operation of the function are essentially the same as for Saturn I vehicles. (See Paragraph 6-58.)

Implementation of the Saturn V range safety function is similar to the Saturn I. Because of the longer powered flight of Saturn V, it is necessary to extend the command transmitter coverage to enable transmission of engine cutoff and propellant dispersion signals. Command transmitters are located at Cape Kennedy, Grand Bahama Island, San Salvador, Grand Turk Island, Bermuda, Atlantic Tracking Ship, Grand Canary Island and Ascension Island to ensure that range safety commands can be transmitted to the vehicle prior to earth orbit injection.

A range safety command system (Figure 6-29) is carried on each stage of the Saturn V vehicle to execute range safety commands. The AN/DRW-13 system is operational on vehicles SA-501, SA-502, SA-503. The digital command system is operational on vehicles SA-504 and subsequent. A description of the digital system is given in paragraph 6-64. (Saturn I range safety)

The command system on the S-IVB stage provides a 28-volt signal to the emergency detection system when engine cutoff is commanded. A five-second time delay (mechanized by a unit plugged into the command destruct controller) prevents initiation of propellant dispersion ordnance until five seconds after engine cutoff. This delay permits escape of the spacecraft crew prior to rupturing of propellant containers.

20-100. ELECTRICAL SYSTEM

The three stages of the Saturn V launch vehicle and the instrument unit are independent of each other for electrical power. Each stage has a complete electrical system which supplies all of its flight power requirements.

The Saturn V electrical systems are active throughout all mission phases. During pre-launch and most of the launch phase, generators located at the Automatic Ground Control Station supply the primary power (28-volt dc) through umbilical connections. Switching from ground power to stage batteries is accomplished without interruption by stage relay networks just prior to lift-off (T minus 35 seconds).

20-101. OPERATION

The operation of each stage and instrument unit electrical system is similar. These systems are similar also to that described for Saturn I (see paragraph 6-65) with the exception that there is no central supply of alternating current. Any equipment requiring ac power has its own inverter.

Two buses in each stage distribute primary power (28 volt dc) to the individual equipment. The main power distribution is through the power distributor, which is directly controlled by the control distributor, which is in turn controlled by the switch selector. The switch selector exercises control of stage sequencing in response to sequencing commands from the vehicle guidance computer. Figure 20-55 illustrates the Saturn V power distribution and sequencing scheme.

Current return for all power and signal circuits is through hardwire ground system. Each return wire is selected such that the total voltage drop from source to equipment is not more than two volts. Ground side of power supplies are tied to the vehicle frame. The on-pad grounding system is illustrated in Figure 20-56.

Separate, isolated power supplies are located in the instrument unit and in each stage to provide 5 volts dc for instrumentation purposes. This voltage is used as a source for some types of sensors. The outputs of the sensors then become 0-5V analogs of the parameter being sensed and are used as input signals to the telemetry links with ground stations.

20-102. IMPLEMENTATION -

Typical components of each stage and instrument unit are described below.

20-103. Batteries - Inflight power for each stage is supplied by two(or more) 28-volt batteries. The cells use potassium hydroxide as electrolyte and have zinc-silver oxide electrodes. Each battery is sized for its application requirements and is provided with taps for adjustment of its output voltage to the nominal 28-volts under load conditions.

20-104. Measuring Voltage Supply. The measuring voltage supply is a solid state dc-to-dc converter. It converts the stage 28 volt dc power into a closely-controlled 5 volt dc output. The 5 volt output is used as reference voltage for measurement transducers and signal conditioners in the instrumentation system.

20-105. Switch Selector - The switch selector is the interface device between the vehicle guidance computer and equipment controlled by the guidance computer in each stage. It is capable of controlling 112 operations. Each output of the switch selector is commanded by an 8-bit code supplied by the guidance computer through the data adapter in the instrument unit. This code, together with a stage select signal sets up

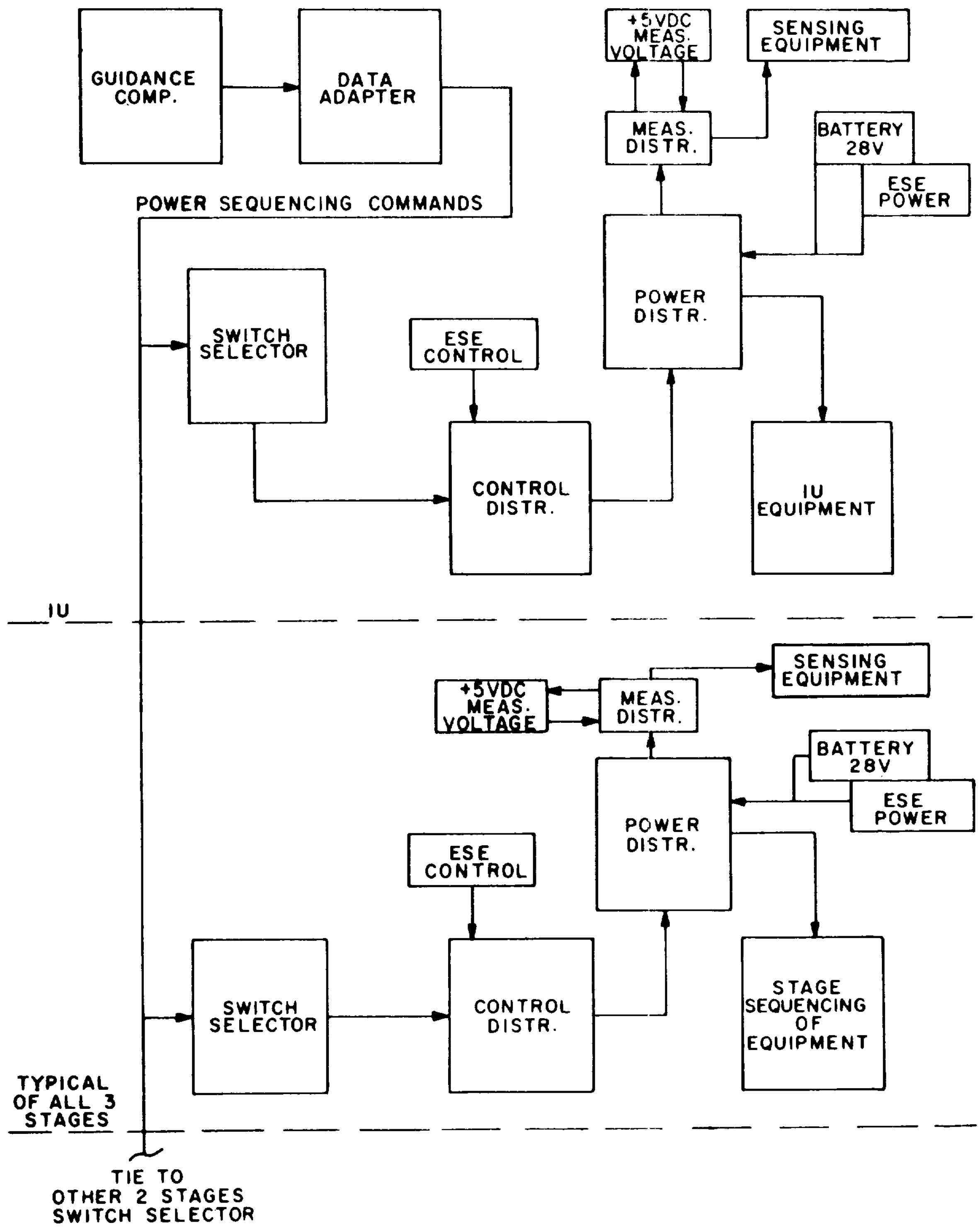

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Figure 20-55. Power Distribution and Sequencing

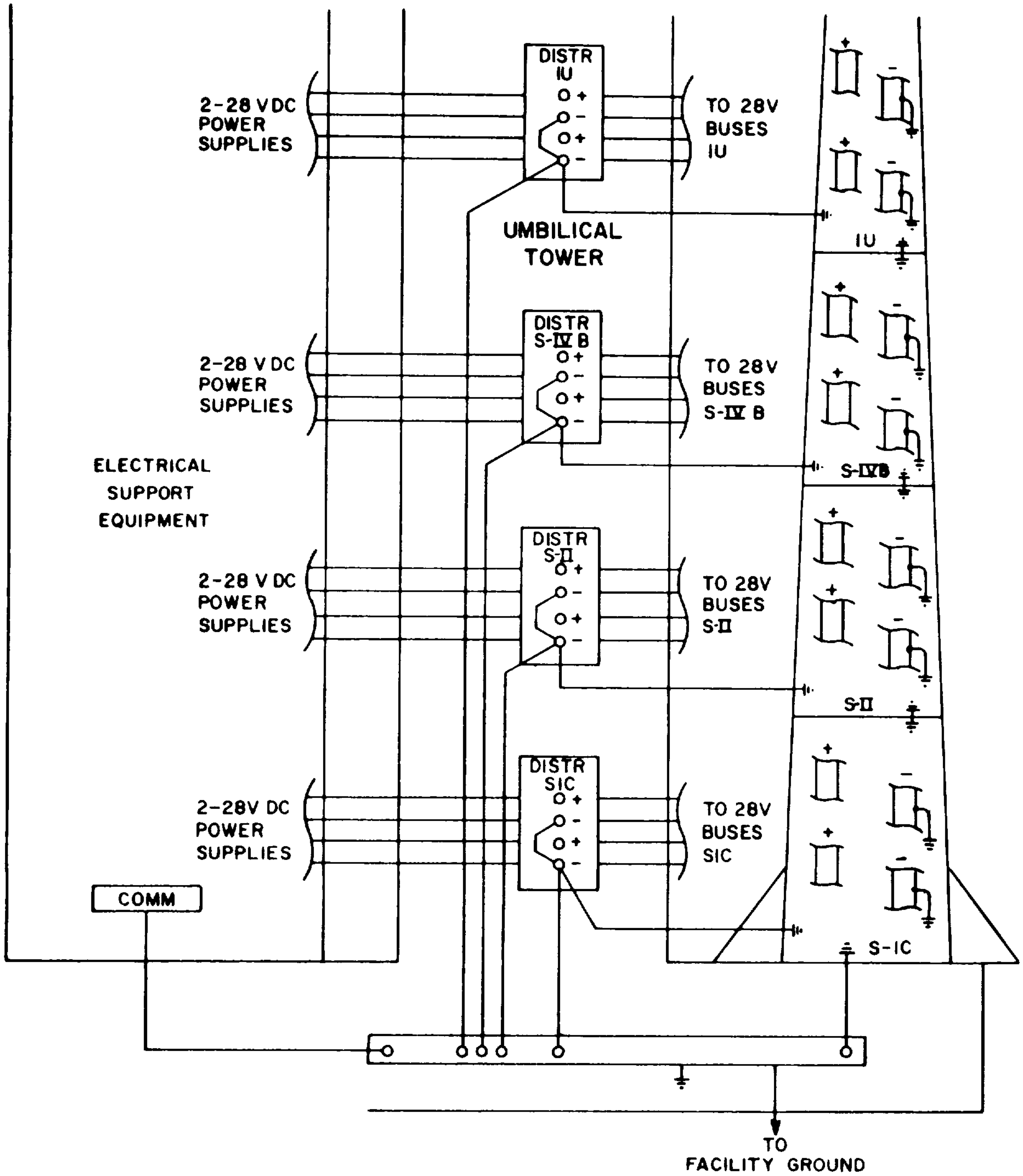

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Figure 20-56. On Pad Grounding, Saturn V relay logic in the control distributor, which in turn controls the sequencing of equipment on the stage.

20-106 Control Distributor. The control distributor contains relay logic which provides control of equipment sequencing on the stage. This distributor is controlled by the switch selector. Power distribution is indirectly controlled by the control distributor, which controls the contactors in the power distributor.

20-107. Power Distributor. Distribution of vehicle primary power (28-volt dc) is accomplished through the power distributor. This distributor contains the heavy current carriers and contactors, which transfer power from stage batteries on ground sources to the two stage distribution buses. Transfer of power from ground source to stage batteries is accomplished by the power distributor contactors under control of the control distributor.

CHAPTER 4

SECTION XXI
STRUCTURES

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SECTION XXI.
STRUCTURES

21-1. STRUCTURAL REQUIREMENTS

The Saturn V launch vehicle structure is designed to withstand all loads that can be expected to occur during ground handling, prelaunch, launch and flight operations. The structure also contains the propellant for the stages. The design requirements for the vehicle structure are determined after a careful analysis of the conditions that will be encountered during all operations.

21-2. GROUND HANDLING CONDITIONS

Handling procedures and equipment are designed so that loads imposed on the structure during fabrication, transportation, and erection do not exceed flight loads and thus do not impose any flight performance penalty.

21-3. PRELAUNCH CONDITIONS

The vehicle, empty or fueled, pressurized or unpressurized and free-standing (attached to the launcher only) is structurally capable of withstanding loads resulting from winds having a 99.9 percent probability of occurence during the strongest wind month of the year. The bending moments (Figure 21-1) and shears resulting from the wind are combined with the longitudinal force due to the weight of the vehicle in defining the worst prelaunch loading condition.

21-4. LAUNCH CONDITIONS

At launch the vehicle structure is capable of withstanding loads from two conditions, holddown and rebound. The holddown condition is imposed on the structure after engine ignition but before the launcher releases the vehicle. The holddown loads result from wind (bending moments and shears), engine thrust (forward axial load), vehicle inertia (aft axial load) and vibration transients due to initial engine combustion.

The rebound condition occurs when the engines are cut off before the launcher re-

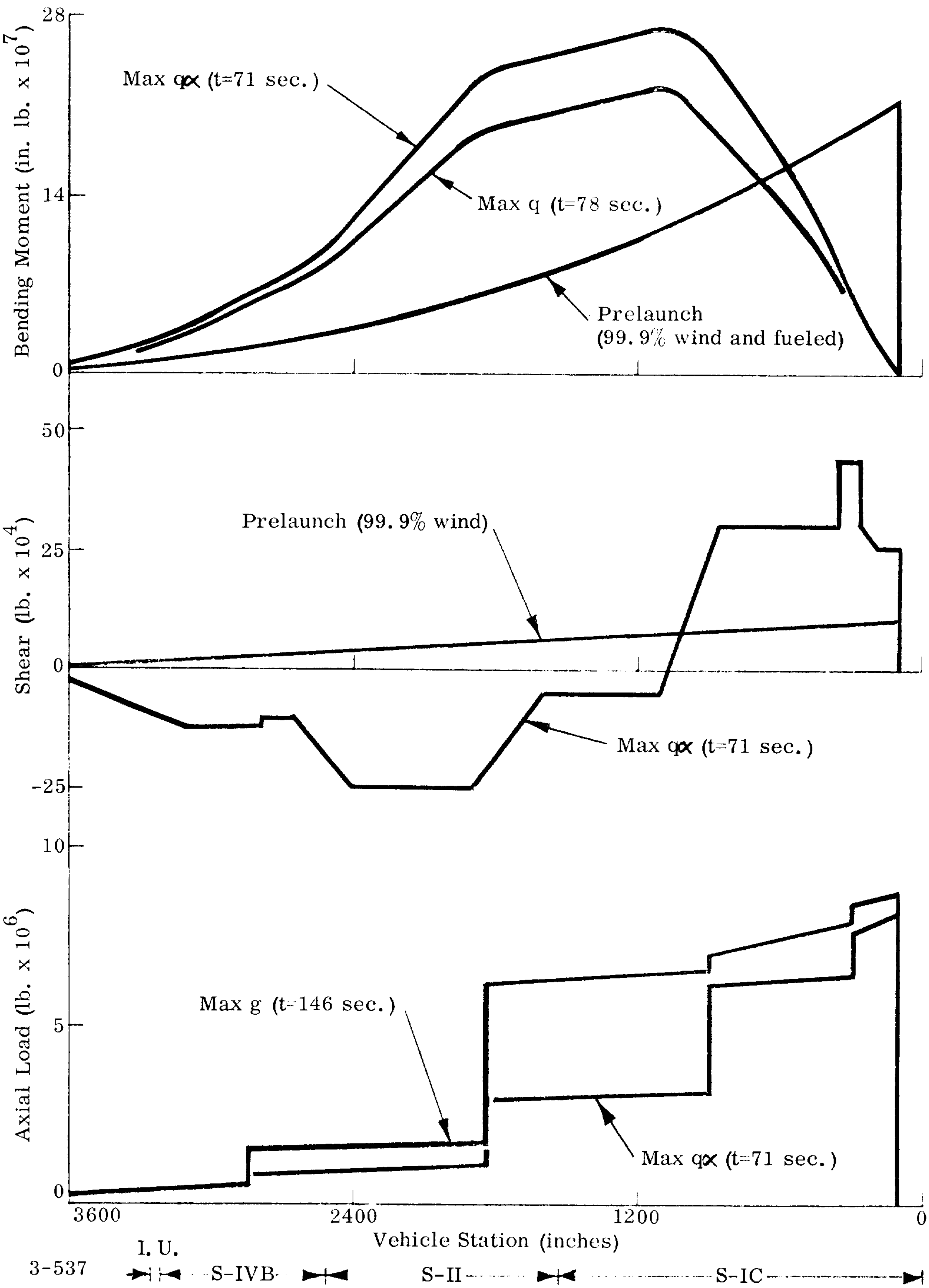

Figure 21-1. Saturn V Loads

leases the vehicle. Axial loads result from deceleration of the vehicle which suddenly reverses the direction of the load at the holddown points. Combined with the axial loads are wind loads (bending moments and shears) and vibration transients resulting from engine cutoff.

21-5. FLIGHT CONDITIONS

During flight the structure is subjected to engine thrust and heat, dynamic, aerodynamic, inertia and propellant loads.

21-6. Engine Thrust and Heat Loads. The first stage thrust (Figure 21-2) increases as the vehicle gains altitude and reaches a maximum at engine cutoff. After first stage separation, the second stage engines impose relatively constant thrust loads on the remainder of the vehicle. After second stage separation, the third stage engine imposes a relatively constant thrust load on the vehicle. The thrust produces axial loads, shears and bending moments on the vehicle. The moments and shears are a result of the engines gimballing.

The first stage engines impose a heat load on the base of the vehicle through radiation and circulation of the exhaust gases. After separation the second stage engines impose a heat load on the base of the second stage.

21-7. Dynamic Loads. Vehicle dynamic loads result from external and internal disturbances. Three main sources of excitation - mechanical, acoustical and aerodynamic produce the vehicle vibration environment. The mechanical source begins at engine ignition and remains relatively constant until engine cutoff. The acoustical source begins with the sound field generated at engine ignition. It is maximum at vehicle liftoff and becomes negligible after Mach 1 (approximately 61 seconds after liftoff). The aerodynamic source begins as the vehicle velocity increases and is most influential during transition at Mach 1 and at maximum dynamic pressure. Transient vibrations, which are relatively high in magnitude and present only for short periods of time, occur during engine ignition, vehicle liftoff, Mach 1, region of maximum dynamic pressure, engine cutoff, and stage separation.

Propellant sloshing, another type of dynamic loading, results from a relative motion between the container and the center of gravity of the fluid mass and is generally caused by gust loads, control modes and vehicle bending modes. Reaction of the

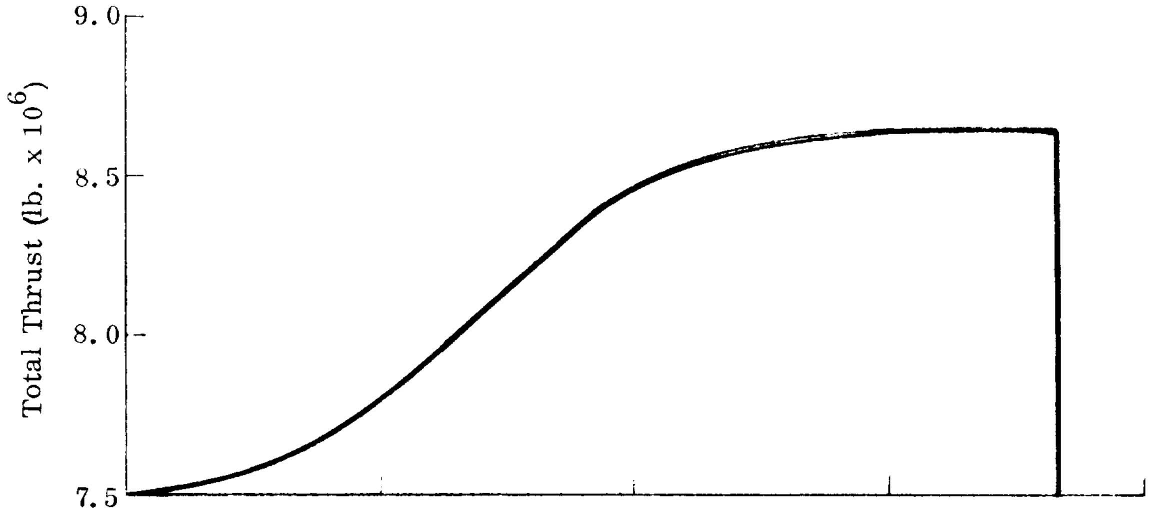

Figure 21-2. S-IC Thrust

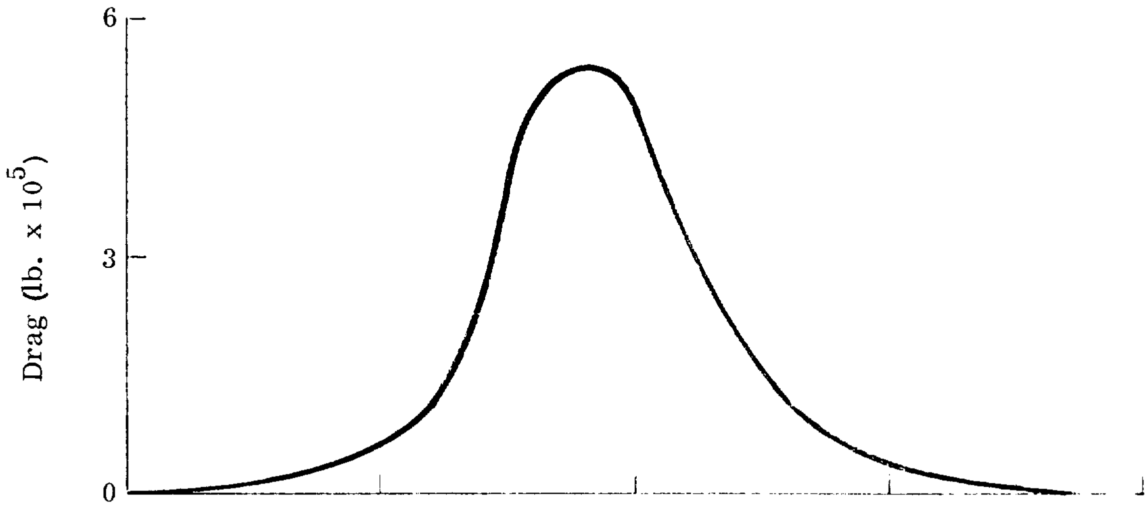

Figure 21-3. Saturn V Drag

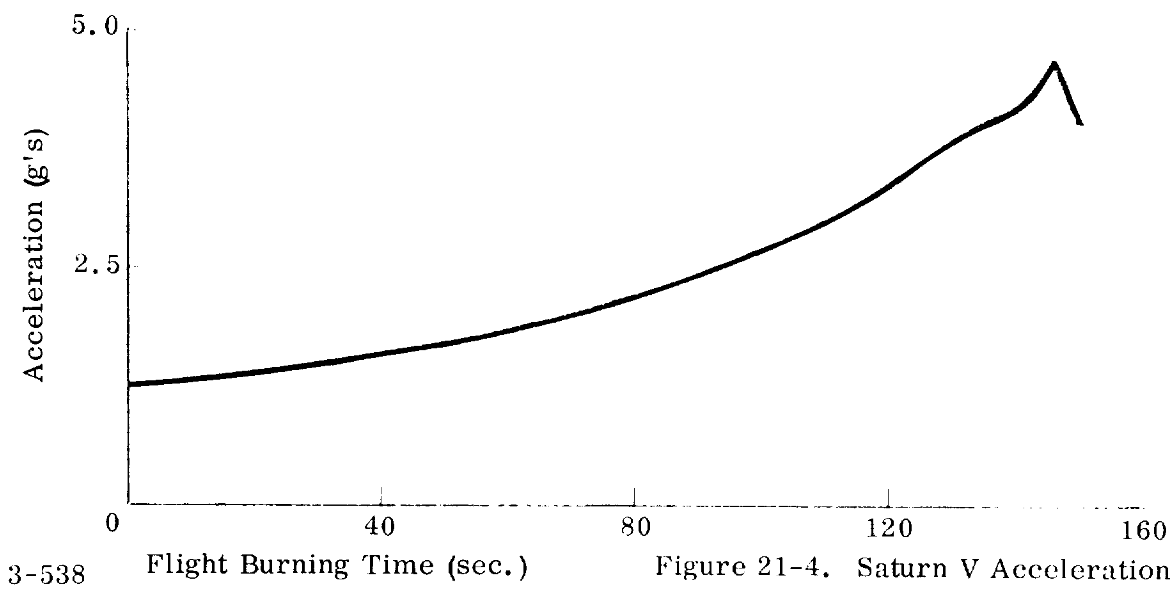

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Figure 21-4. Saturn V Acceleration

control system (gimballing engines) to gust loads produces considerable bending deflection in the vehicle structure. Since the structure and propellant are not integral and do not deflect together, sloshing results. If the propellant sloshing is not damped, compensation for the resulting perturbations must be provided by the control system.

21-8. Aerodynamic Loads. Aerodynamic loading is a result of drag, angle of attack and wind gusts. Aerodynamic drag (Figure 21-3) increases to a maximum approximately 78 seconds after liftoff (max q condition) and then decreases to nearly zero before first stage burnout. Aerodynamic drag imposes an axial load on the structure and when combined with an angle of attack results in bending moments and shears which are maximum approximately 71 seconds after liftoff ($q\alpha$ max condition). When the vehicle is in the region of high drag, structural bending moments are minimized by the control system which reduces the vehicle angle of attack.

Aerodynamic heating on the vehicle is a result of friction caused by the vehicle moving through the atmosphere. The heating increases until first stage burnout and then decreases. Vehicle surfaces which are not parallel to the vehicle centerline have the greatest temperature increase during flight.

21-9. Inertia Loads. Inertia loads (Figure 21-4) result from the vehicle acceleration due to an increase in the thrust/weight ratio during flight. Peak acceleration is at first stage cutoff (max g condition). The acceleration decreases at first and second stage separation and then increases during second stage burning, but never reaches the peak achieved at first stage cutoff. This is also true after second stage separation.

21-10. Propellant Loads. The loads imposed on the structure by the propellant are due to a combination of hydrostatic head, and ullage and ambient pressures. The hydrostatic head, varying during flight, is a function of the density of the fluid, height of the fluid in the container and the acceleration of the vehicle. The ullage pressure is supplied by the pressurization system and is limited by relief valves. As the altitude of the vehicle increases during flight, the ambient pressure decreases. At any time during flight (at any location in the container) the maximum pressure differential across the container wall is equal to the ullage pressure plus the hydrostatic head minus the ambient pressure.

21-11. STRUCTURAL DESIGN

The Saturn V launch vehicle consists of three stages joined by interstage structures. An instrument unit mounted forward of the third stage provides the support for the spacecraft. Critical loading conditions for various protions of the vehicle occur at different times. The critical conditions occur on the S-IC structure during prelaunch (ground wind), launch (rebound), and flight ($q\alpha$ max and max g). On the S-II structure the critical conditions occur during prelaunch (ground wind) and flight ($q\alpha$ max and max g). They occur on the S-IVB structure during prelaunch (ground wind) and flight ($q\alpha$ max) and on the instrument unit during flight ($q\alpha$ max). For the propellant containers, critical external loads are combined with the internal gas pressure and hydrostatic head to obtain the structural design loads.

Slosh baffles are installed in the S-IC RP-1 and LOX containers and in the S-II and S-IVB LOX containers. The baffles dampen the sloshing propellant and transfer absorbed slosh forces to the container walls. Slosh baffles are not required in the S-II and S-IVB LH_2 containers because of the low density of the LH_2.

21-12. S-IC STAGE

The S-IC structure is an assembly of a thrust structure, a RP-1 container, an intertank section, a LOX container and a forward skirt. Attached to the thrust structure are a base heat shield, four aerodynamic fins and four engine fairings. Since both propellants are relatively dense a separate rather than integral container configuration is used.

Several conditions produce critical loads on the thrust structure. The rebound and max g conditions produce the maximum axial loads, bending moments and shears in the cylindrical section of the thrust structure. Axial load resulting from the max g condition (engine thrust) is critical for the center engine support. The aft end of the thrust structure is protected from the hot engine exhaust gases by the base heat shield.

Four aerodynamic fins aid in stabilizing the vehicle during flight. Maximum loading on the fins results from the $q\alpha$ max condition.

The maximum compressive buckling load in the RP-1 container is produced by a combination of bending moment and axial load resulting from the prelaunch and $q\alpha$

max conditions. The lower protion of the container is critical during prelaunch (container full and unpressurized) and the upper portion at $q\alpha$ max. The critical load on the intertank section occurs at $q\alpha$ max. For the LOX container and forward skirt, the maximum compressive buckling load is produced by the max g condition.

In addition to the external loads carried by the container cylindrical sections, both containers must withstand propellant and internal pressurization loads. Each container consists of a forward and aft bulkhead joined by a cylindrical section. The maximum pressure differential on the container forward bulkheads occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical sections and aft bulkheads varies during flight because the propellant level and ambient pressure decrease while the acceleration of the vehicle increases.

21-13. S-II STAGE

The S-II structure is an assembly of an aft interstage, an aft skirt, a thrust structure, a heat shield, an integral propellant container and a forward skirt. To reduce the length of the vehicle and thus reduce external loading, the propellants are contained in an integral container. Located within the container is the common bulkhead which separates the LH_2 from the LOX. To reduce the loads on the vehicle, the LOX, which weighs five times as much as the LH_2, is located aft.

The aft interstage, aft skirt, cylindrical section of the propellant container, and forward skirt withstand the loads encountered during all vehicle operations through first stage burnout. Following stage separation and until second stage burnout, the thrust structure, aft skirt, cylindrical section of the LH_2 container, and forward skirt resist all loads encountered as result of S-II engine operation.

The critical design condition for the aft interstage and aft skirt occurs at max g at which time the largest compressive buckling load is produced on the structure. For the cylindrical section of the LH_2 container two conditions govern. The critical load on the lower portion of the container occurs at $q\alpha$ max and for the upper portion occurs during prelaunch (container full and unpressurized). The $q\alpha$ max condition produces the most critical loads on the forward skirt.

Engine thrust, the principal load during S-II engine operation, produces a critical loading condition only in the thrust structure. The heat shield, which is attached to the thrust structure, is designed to protect the aft end of the S-II from engine heat.

In addition to the external loads carried by the cylindrical section, the propellant container must resist propellant and pressurization loads. The container consists of a forward bulkhead, a cylindrical section, an aft bulkhead and a common bulkhead. The maximum pressure differential on the container forward bulkhead occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical section and the aft bulkhead occurs at first stage cutoff. At this time the vehicle acceleration is greatest and the ambient pressure is zero. The common bulkhead is designed to resist both bursting and collapsing pressure conditions. The critical conditions are based on combinations of LH_2 and LOX pressures and temperature.

21-14. S-IVB STAGE

The S-IVB structure is an assembly of an aft interstage, an aft skirt, a thrust structure, an integral propellant container, and a forward skirt. To reduce the length of the vehicle and thus reduce external loading, the propellants are contained in an integral container. Located within the container is the common bulkhead which separates the LH_2 from the LOX. To reduce the loads on the vehicle the LOX, which weighs five times as much as the LH_2, is located aft.

The aft interstage, aft skirt, cylindrical section of the propellant container, and forward skirt withstand the loads encountered during all vehicle operations through second stage burnout. Following separation from the second stage, the thrust structure, LOX container aft bulkhead, cylindrical section of the LH_2 container, and forward skirt resist all loads encountered as a result of S-IVB engine operation.

The critical design condition for the aft interstage, aft skirt and forward skirt occurs at $q\alpha$ max and produces the largest compressive buckling load on the structure. For the cylindrical section of the LH_2 container two conditions govern. The critical load on the lower portion of the container is produced at $q\alpha$ max and for the upper portion during prelaunch (container full and unpressurized). Engine thrust, the principal load during S-IVB engine operation, produces a critical loading condition only in the thrust structure.

In addition to the external loads carried by the cylindrical section, the propellant container must resist propellant and pressurization loads. The container consists of a forward bulkhead, a cylindrical section, an aft bulkhead and a common bulkhead. The maximum pressure differential on the container forward bulkhead occurs when the vehicle reaches the altitude where the ambient pressure is zero. The maximum pressure differential on the cylindrical section and the aft bulkhead occurs at first stage cutoff. At this time the vehicle acceleration is greatest and the ambient pressure is zero. The common bulkhead is designed to resist both bursting and collapsing pressure conditions. The critical conditions are based on combinations of LH_2 and LOX pressures and temperatures.

21-15. INSTRUMENT UNIT

The instrument unit structure resists the loads encountered during all vehicle operations through payload separation. The critical design condition occurs during flight at $q\alpha$max when a combination of bending moment and axial force produces the largest compressive buckling load on the structure.

21-16. S-IC CONFIGURATION

The S-IC stage structure is 1492 inches (124.4 feet) long and 396 inches (33.0 feet) in diameter. It has a 756-inch (63.0 feet) span across the fins. The stage structure includes: a thrust structure, engine fairings, fins, a base heat shield, a RP-1 container, an intertank section, a LOX container, and a forward skirt.

21-17. THRUST STRUCTURE

The thrust structure, Figure 21-5, is designed to distribute the thrust loads from the five F-1 engines to the fuel container during flight and to provide hold-down points for the vehicle during static test and launch. The thrust structure, constructed primarily of 7075 and 7079 aluminum alloy, is approximately 230 inches long. In addition to the engines, the thrust structure supports the base heat shield, engine fairings and fins.

The outboard engines are mounted on a 364-inch diameter, 90 degrees apart. Clear-

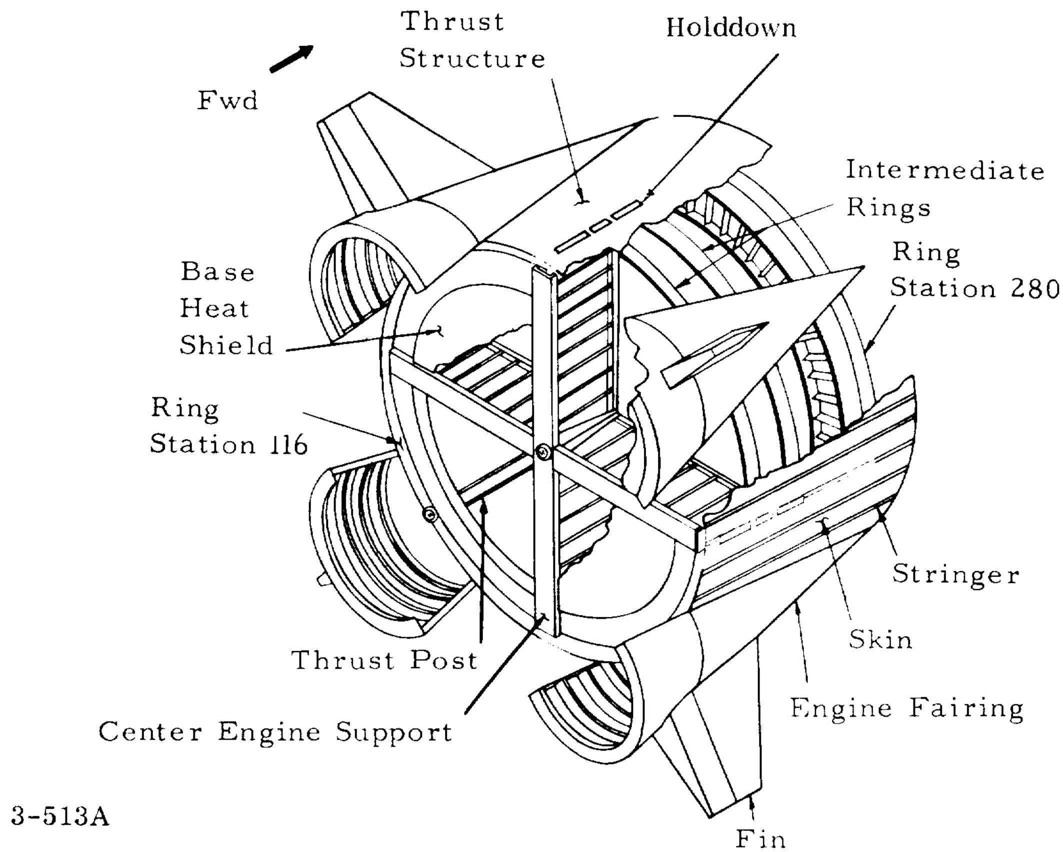

Figure 21-5. Thrust Structure, Fins, Engine Fairings, S-IC

ance between the engines and structure is based on a 7-degree square gimbal pattern. Lateral loads (resulting from engine gimballing) and axial loads are transmitted from the gimbal bearing joints through the ring at MSFC station 116 to the thrust posts. The thrust posts shear the axial load into the thrust structure cylindrical skin. Moments produced by the lateral loads are reacted by the rings at MSFC stations 116 and 280. Loads reacted by the two rings are sheared into the thrust structure skin. Both rings are heavy built-up sections.

In addition to supporting the thrust posts, the rings support the four hold-down posts. The hold-down posts, which are equally spaced between the thrust posts, transmit hold-down loads from the launcher to the thrust structure skin. The hold-down posts also transmit thrust load from the center engine support to the skin.

The center engine support is constructed of four 80-inch deep built-up beams arranged in a cruciform. The beams are joined at the center by a post. The outboard ends of the beams are attached to the hold-down posts.

Loads sheared into the thrust structure skin by the thrust posts and hold-down posts are distributed by longitudinal stringers. The skin varies in thickness, being greater adjacent to the thrust posts and hold-down posts where the loads are higher. Loads are transmitted from the skin and stringers to the fuel container by fittings which bolt to the container aft Y-ring. In addition to the rings at MSFC stations 116 and 280, the skin and stringers are supported by four intermediate rings. These rings are also built-up sections.

Supports for the engine gimballing actuators are located in the engine fairings. For each outboard engine there are two actuator attachment points. The engine actuator supports are attached to the rings at MSFC station 116 and 280. These two rings and the actuator supports are designed with a stiffness to prevent the natural frequency of the engines from becoming a problem in the control of the vehicle.

Aft supports for first stage ground transportation are located at the hold-down positions. Supports for propellant lines are built-up oval type brackets which attach to the LOX and RP-1 lines. The brackets are supported from the thrust structure. The thrust structure is vented into the engine fairings to limit the pressure differential across the base heat shield. Cutouts are provided in the thrust structure for the emergency RP-1 drain line, RP-1 fill and drain line, and LOX fill and drain line.

21-18. ENGINE FAIRINGS

To prevent excessive loads in the control actuators, the outboard engines are protected from aerodynamic loads by the four engine fairings, Figure 21-6. The fairings, constructed of 2024 and 7075 aluminum alloy, are conical in shape with a 15-degree side slope and a 100-inch radius at the aft end. The fairings are approximately 300 inches long with the aft end located 48 inches aft of the gimbal plane. Aerodynamic loads are transmitted to the rings through the skin and longitudinal external stringers. The rings transmit the aerodynamic loads to the thrust structure. Each fairing has four air scoops.

21-19. FINS

Four fins (Figure 21-7), located outboard of the engines, augment vehicle aerodynamic stability. The fins are rigidly attached to the thrust structure at each engine fairing and are designed to withstand aerodynamic heating and pressure. Each fin has a 75 square foot trapezoidal planform. The leading edge of each fin is swept back 30-

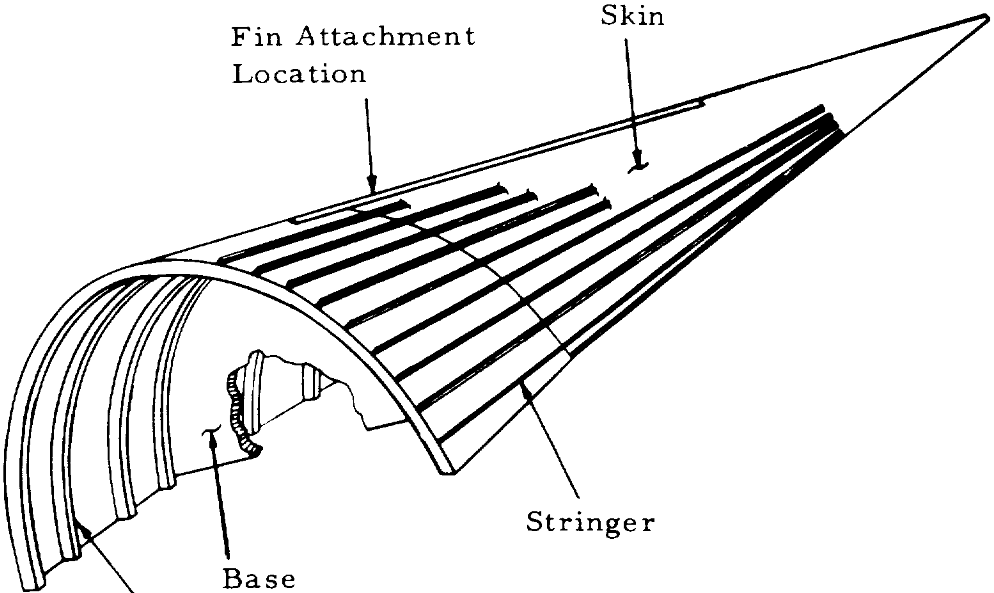

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Figure 21-6. Engine Fairing, S-IC

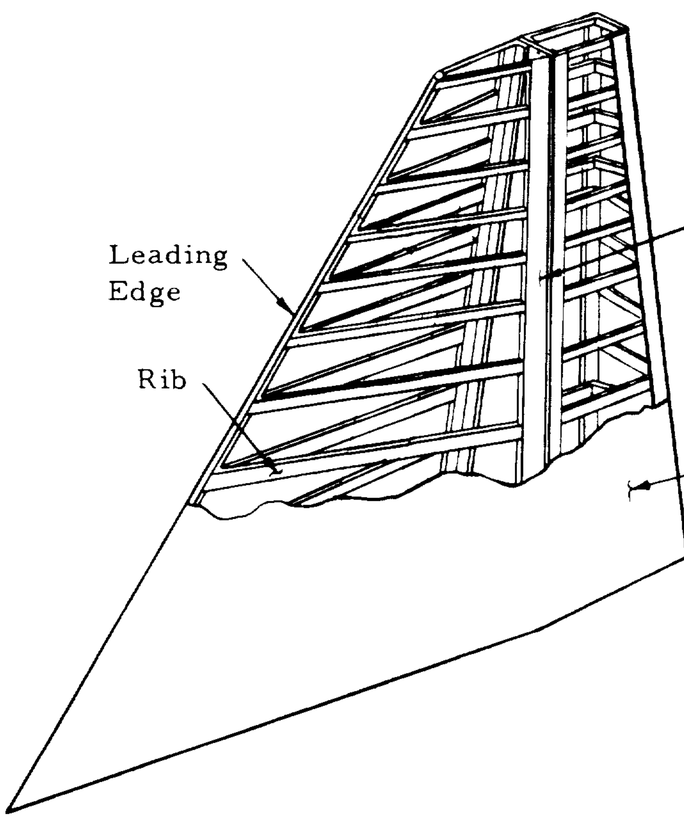

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Figure 21-7. Fin, S-IC

degrees and has a 10-degree wedge angle. The fin leading edge is constructed of 7178 aluminum alloy with a steel tip. The remainder of the fin is constructed of 2024 and 7079 aluminum alloy. Aerodynamic loads are transmitted through the fin skin and ribs to the main spar which in turn transmits the loads to the thrust structure.

21-20. BASE HEAT SHIELD

The base heat shield protects the thrust structure and components mounted within from engine heat. Insulation between the hot and cold sides of the heat shield is such that with a maximum hot side temperature of 2000 degrees F. the cold side temperature is less than 300 degrees F. The heat shield is designed so that the deflection due to differential pressure loading is limited to two inches.

The shield consists of honeycomb sandwich panels with steel faces and core, covered on the hot side with a layer of ablative insulation. Support for the heat shield is provided by a complex of beams attached to the center engine support and aft frame. Openings are provided in the base heat shield for LOX and RP-1 lines. The openings are sealed with flexible curtains which are attached to the lines and heat shield. The curtains are constructed of fiberglass cloth sandwiched between an inconel wire mesh. Removable panels in the heat shield provide access to the tail section. In addition to the base heat shield, there is a heat shield for each engine fairing. These shields are of the same type construction as the base heat shield and are located in the same plane. Each shield is supported by the engine fairing actuator support structure and thrust structure ring at MSFC station 116. Openings are provided in each shield for the engine actuators.

21-21. FUEL CONTAINER

Fuel for the S-IC stage is contained in a 517-inch long all-welded 2219 aluminum-alloy container (Figure 21-8). The container is a cylindrical section closed at both ends with ellipsoidal bulkheads. The bulkheads are welded to Y-rings which are in turn welded to the cylindrical section.

The aft bulkhead, designed to withstand flight pressurization and propellant loads due to acceleration, is constructed of eight 45-degree gores and a circular center piece which are welded together. The bulkhead is attached to the cylindrical section with a Y-ring.

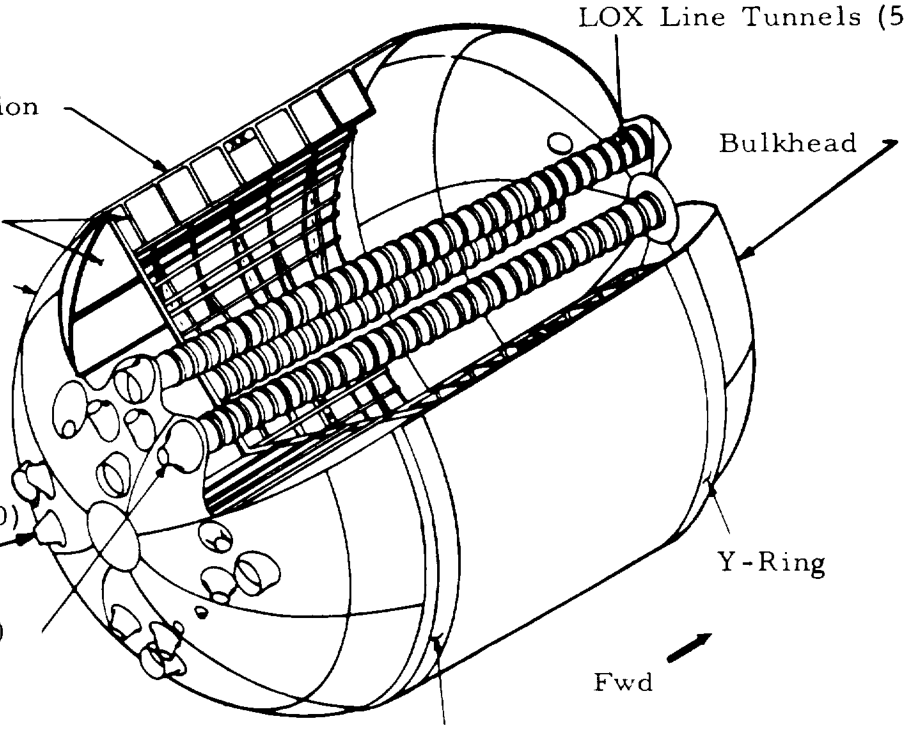

Figure 21-8. Fuel Container, S-IC

The fuel container cylindrical section is 243 inches long and is designed to withstand flight pressurization, flight loads, and propellant loads due to acceleration. The cylindrical section skin is supported by internal rings and internal integral stiffeners. The rings, constructed of 7178 aluminum alloy, are mechanically attached to the inboard flanges of the stiffeners. The stiffener-skin combination, designed to withstand bending moment and axial load, gives the structure a free-standing capability on the launch pad even though the container is unpressurized.

Loads are transmitted from the thrust structure to the cylindrical section through the aft Y-ring. The loads are carried forward to the intertank section through the cylindrical section. A forward Y-ring joins the cylindrical section to the forward bulkhead and intertank section. The loads are transmitted to the intertank section from the forward Y-ring.

The forward bulkhead of the RP-1 container is similar in construction and contour to the aft bulkhead. The skin is thinner since the bulkhead carries only flight pressurization loads.

The forward bulkhead has four access manholes and five outlet fittings for the LOX line tunnels. Outlet fittings are also provided for the RP-1 vent line and the RP-1 pressurization line. The aft bulkhead has five outlet fittings for the LOX line tunnels and ten outlet fittings for the RP-1 lines. Outlets are also provided for emergency RP-1 drain line and RP-1 fill and drain line.

The five LOX lines are routed through tunnels in the RP-1 container. The tunnels, constructed of 2219 aluminum alloy and stiffened with external rings, are attached rigidly to the aft bulkhead. Attachment to the forward bulkhead is with a seal joint that compensates for vertical and rotational deflections.

Ring-type slosh baffles are attached to each of the internal rings. The baffles are supported at their inboard flanges by longitudinal stringers. A cruciform slosh baffle is located in the bottom of the RP-1 container. Each panel of the baffle consists of a continuous corrugation supported by a truss structure.

21-22. INTERTANK SECTION

Structural continuity between the LOX and RP-1 containers is provided by the intertank section (Figure 21-9). The 7075 aluminum-alloy intertank section is a cylinder

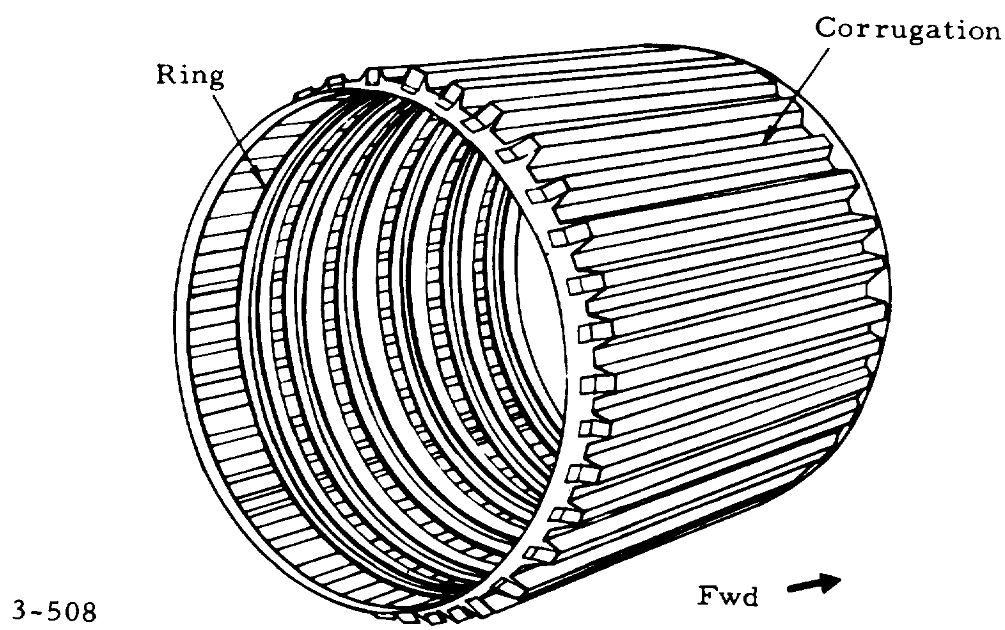

Figure 21-9. Intertank Section, S-IC

263 inches long. Axial loads and bending moment are carried by a continuous corrugation. Internal built-up rings support the corrugation. Loads are transmitted from the RP-1 container to the intertank section through fittings which are bolted to the RP-1 container forward Y-ring. The loads are transmitted from the intertank section to the LOX container through a similar type of joint.

Two doors provide access to the intertank section. Cutouts are provided in the intertank section for the RP-1 pressurization line, RP-1 vent line, and LOX emergency drain line. The cutout for the LOX emergency drain line is also used as an access door.

21-23. OXIDIZER CONTAINER

Liquid oxygen for the S-IC Stage is contained in a 769-inch long all-welded 2219 aluminum-alloy container (Figure 21-10). The container is a cylindrical section closed at both ends with ellipsoidal bulkheads. The bulkheads are welded to Y-rings which are in turn welded to the cylindrical section.

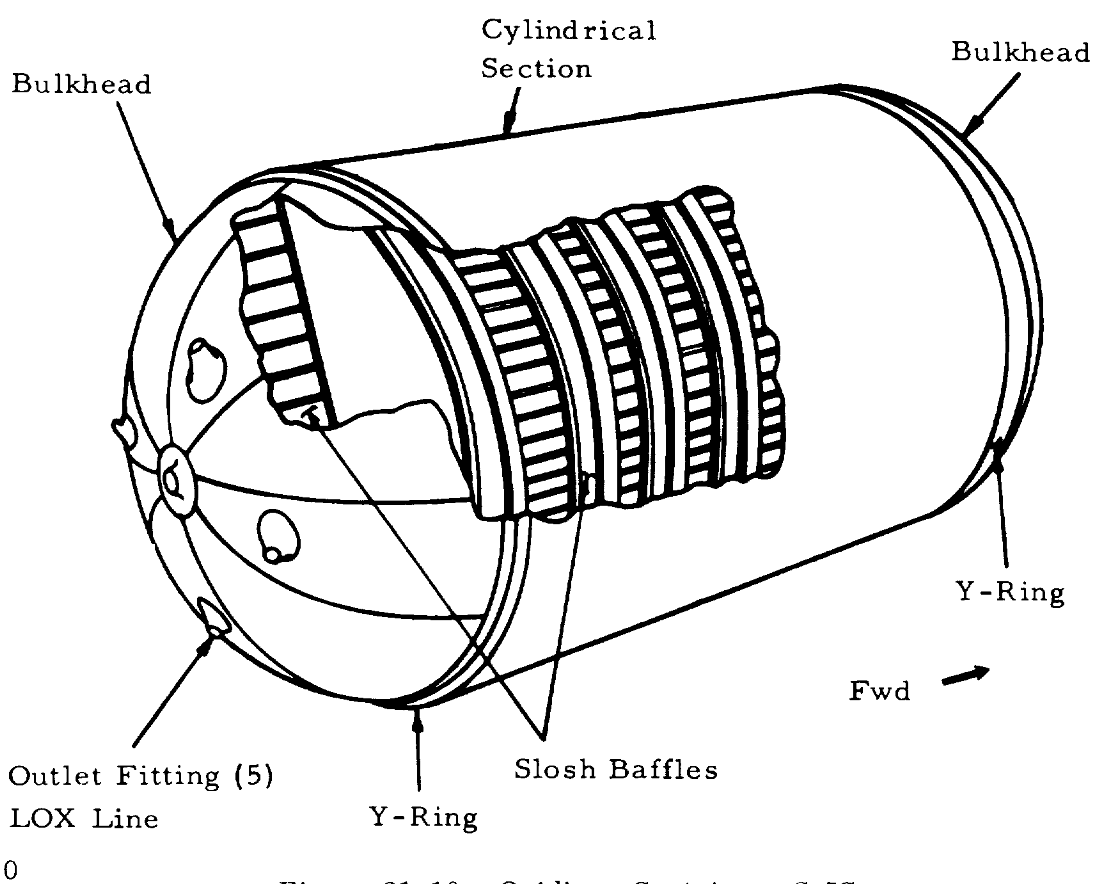

Figure 21-10. Oxidizer Container, S-IC

The aft bulkhead is designed to withstand flight pressurization and propellant loads due to acceleration. The bulkhead is constructed of eight 45-degree gores and a circular center piece welded together. The aft bulkhead is joined to the cylindrical skin section with a Y-ring.

The cylindrical section is 489 inches long. It is designed to withstand flight pressurization, flight loads, and propellant loads due to acceleration. The cylindrical section has internal integral stiffeners. Rings are mechanically attached to the inboard flanges of the longitudinal stiffeners. The stiffeners and rings give the structure a free-standing capability on the launch pad even though the container may be unpressurized. The stiffener skin combination is designed to withstand bending moment and axial load.

Loads are transmitted from the intertank section to the cylindrical section through the aft Y-ring. The loads are carried forward by the cylindrical section and are transmitted to the forward skirt through the forward Y-ring. The forward Y-ring also joins the cylindrical section to the forward bulkhead.

The forward bulkhead is similar in construction and contour to the aft bulkhead. The skin is thinner since the bulkhead carries only flight pressurization loads.

The forward bulkhead of the LOX container has one access manhole and two vent outlet fittings. The aft bulkhead has five LOX line outlet fittings and one outlet fitting for the emergency LOX drain.

The cylindrical section has internal support brackets for mounting four helium cylinders. The bracketry is supported by ring-type slosh baffles. The slosh baffles, attached to each of the internal rings, are joined together at their inboard flanges by longitudinal stringers. A cruciform slosh baffle is located in the bottom of the LOX container. Each panel of the cruciform baffle consists of a continuous corrugation supported by a truss structure.

21-24. FORWARD SKIRT

Structural continuity between the S-IC and S-II is provided by the forward skirt (Figure 21-11). The forward skirt, constructed of 7075 aluminum alloy, is a cylinder 120 inches long. Axial loads and bending moment are carried by a

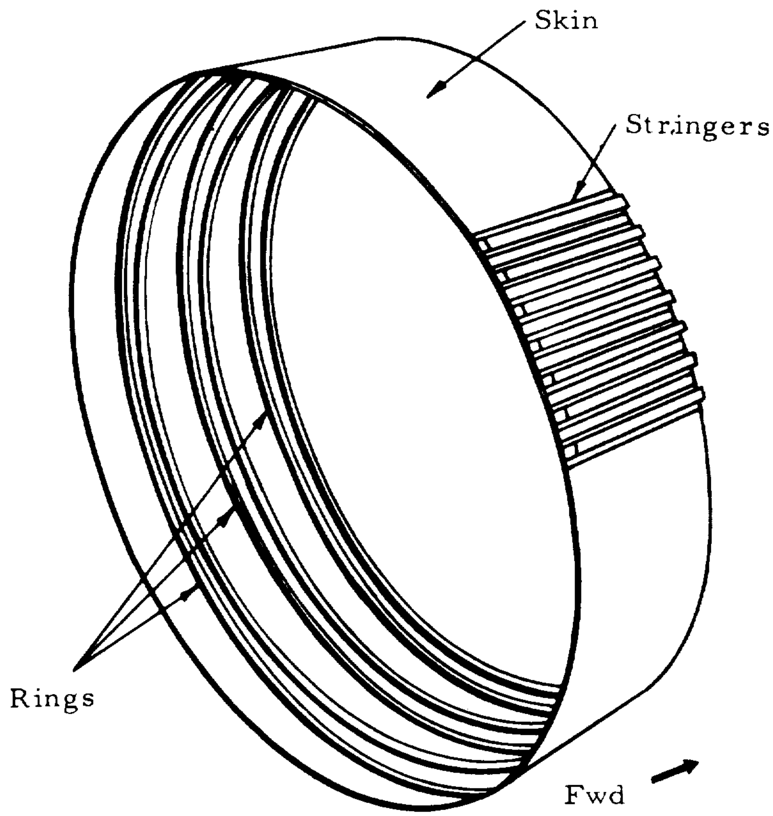

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Figure 21-11. Forward Skirt, S-IC

combination of skin and external longitudinal hat section stringers; shear load is carried by the skin. Loads are transmitted to the skin and stringers through a bolt attachment to the LOX container forward Y-ring. Three internal rings support the skin and stringers. The forward ring provides a mating face for attachment to the S-II aft interstage in a field splice (at MSFC station 1541).

A door provides access to the forward skirt. Cutouts are provided in the forward skirt for vent lines, command and telemetry antennas, and the umbilical plate.

21-25. SYSTEMS TUNNEL

Two tunnels mounted on the exterior surface of the stage contain cable, tubing and linear shaped charge runs between the thrust structure, the intertank section and the forward skirt. The tunnels are constructed in sections to permit easy removal for maintenance and repair.

21-26. S-II CONFIGURATION

The S-II Stage structure, Figure 21-12 is 978 inches (81.5 feet) long and 396 inches (33 feet) in diameter. An aft interstage, an aft skirt and thrust structure, and a heat shield, two propellant containers, and a forward skirt are structurally joined to make up the stage.

21-27. AFT INTERSTAGE

Loads from the first stage are transmitted to the S-II Stage through the aft (S-IC/ S-II) interstage. The 7075 aluminum-alloy interstage is a cylinder 219 inches long. A combination of external longitudinal hat section stringers and skin carry the axial load and bending moment. The skin also carries the shear load. The interstage skin and stringers are supported by internal rings.

The aft interstage is attached to the first stage of the launch vehicle (at MSFC station 1541) by a field splice. The aft ring, providing a mating face for the attachment, transmits loads to the stringers and skin. Intermediate rings support the skin and stringers. Loads are then transmitted to the forward ring of the interstage which provides a mating face for attachment to the aft skirt.

The separation plane of the S-IC/S-II stages is at MSFC station 1564. The aft interstage separates from the S-II stage at MSFC station 1760. At the separation planes, tension straps splice each stringer. An aft interstage door provides access to the propulsion installation. Cutouts are provided in the structure for electrical and fluid umbilical connectors. Eight ullage motors are mounted on the exterior of the interstage. The interstage structure contains provisions for installation of work platforms for maintenance while the stage is on the launch pad.

21-28. AFT SKIRT AND THRUST STRUCTURE

The aft skirt and thrust structure are an integral 7075 aluminum-alloy assembly. The thrust structure transmits second stage engine thrust loads to the aft skirt. First stage loads and second stage engine thrust loads are transmitted to the LH_2 container through the aft skirt.

The conical frustum thrust structure transmits the engine thrust loads to the aft

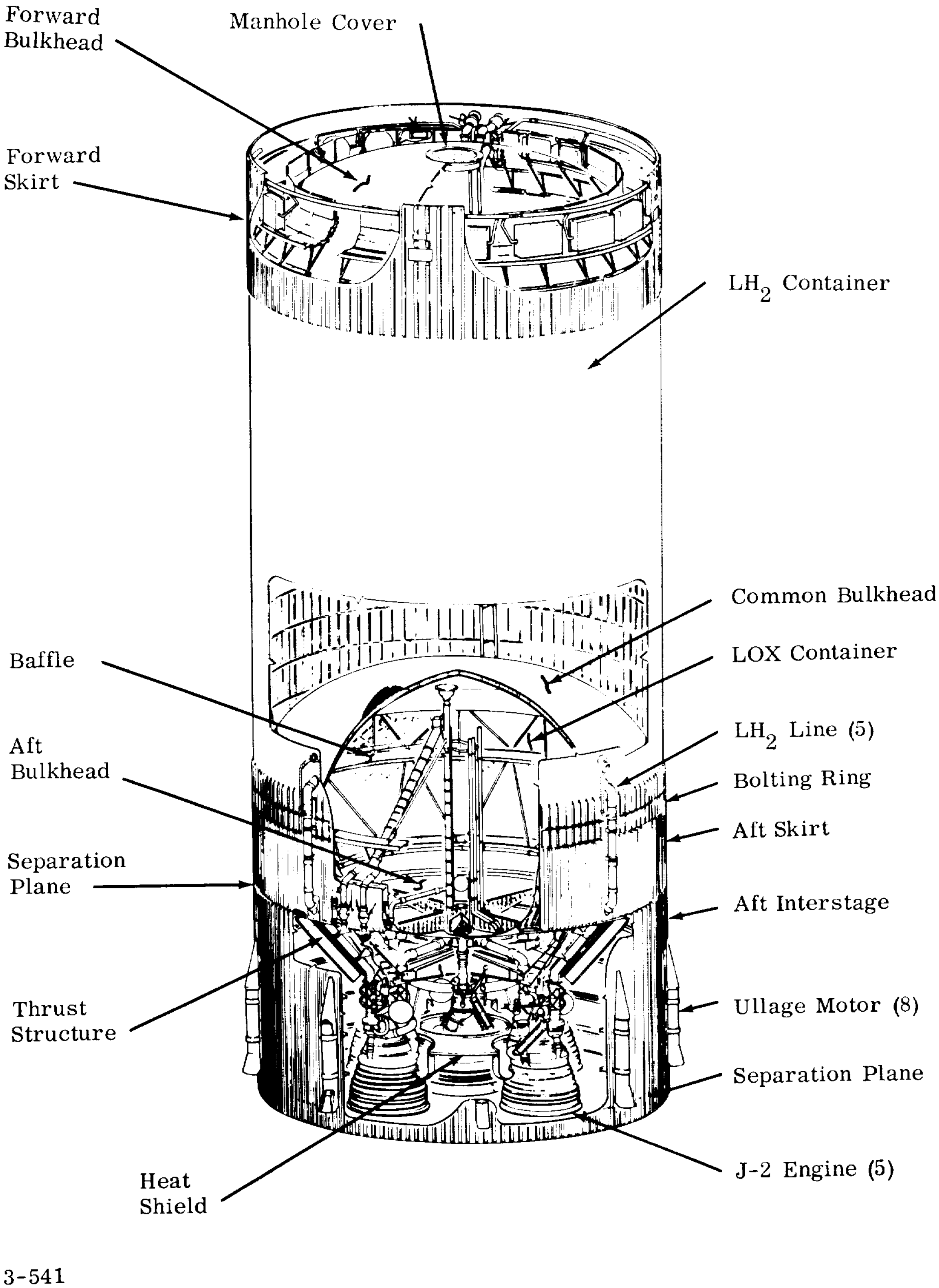

Figure 21-12. S-II Stage Structure

skirt. The thrust structure has an aft diameter of 210 inches, a forward diameter of 396 inches and a length of 111 inches. The four outboard engines are equally spaced on a 210-inch diameter. Longitudinal loads and lateral loads (resulting from outboard engines gimballing) are transmitted by the gimbal bearing joints to the thrust structure engine mounting ring. The center engine support assembly transmits thrust load from the center engine to the thrust structure and stiffens the engine mounting ring. The assembly consists of four honeycomb sandwich type beams arranged in cruciform and joined at the stage centerline by a machined fitting. The outboard ends of the beams, machined fittings, attach the support assembly to the thrust structure. Loads are transmitted from the engine mounting ring to four pairs of thrust longerons which back up the outboard engine attachment fittings. The thrust longerons shear the concentrated loads into the conical frustum which in turn distributes them uniformly into the aft skirt. Longitudinal hat section stringers distribute the load which is sheared into the skin by the longerons. Supporting the skin and stringers are internal built-up rings. Loads are transmitted from the skin and stringers of the conical frustum to the forward ring which is attached to the aft skirt. This attachment, 61 inches aft of the skirt forward interface, changes the load path from a cone to a cylinder creating a lateral load. The lateral load is sheared into the aft skirt skin by the forward ring. The axial load is transmitted directly from the forward ring to the aft skirt stringers.

The aft skirt transmits thrust loads to the LH_2 container. In addition, the skirt transmits first stage loads from the aft interstage to the LH_2 container. The skirt is 87 inches long. External longitudinal hat section stringers carry the axial load and bending moment; the skin carries the shear. The skirt skin and stringers are supported by internal rings. The aft skirt is bolted to the aft interstage through a ring which transmits the load to the skin and stringers. Loads are transmitted from the skin and stringers to the LH_2 container through a circumferential splice. Five LH_2 lines are attached externally to the aft skirt and are routed through the skirt to the engines. Support for the lines is provided by the thrust structure. The LOX fill and drain line is routed through and supported by the aft skirt and thrust structure.

21-29. HEAT SHIELD

The heat shield protects the stage base area from recirculation of engine exhaust gases. The shield, 210 inches in diameter, is of lightweight construction protected by low density ablative material. It is located 54 inches aft of the engine gimbal plane. The heat shield support structure is attached to the engine mounting ring. An opening in the heat shield is provided for each engine. A flexible curtain protects the area between each engine and its associated opening.

21-30. LIQUID OXYGEN CONTAINER

The LOX for the S-II stage is contained in a 2014 aluminum-alloy ellipsoidal container. Two bulkheads, an aft and a common, are joined by a circumferential weld to form the container. Incorporated in the joint is a ring for attaching the LOX container to the LH_2 container. The aft bulkhead, an ellipsoid constructed of welded gores, is designed to carry flight pressurization and propellant loads due to acceleration. Access to the container is through the sump located in the center of the bulkhead. The other bulkhead, termed a common bulkhead because it is common to both the LOX and LH_2 containers, is also ellipsoidal. This bulkhead is of honeycomb sandwich construction.

Five engine line outlet fittings and one fill and drain line fitting are located in the aft bulkhead sump. A cruciform baffle in the bottom of the LOX container limits formation of vortices at the engine line outlets. Installed inside the container are three slosh baffle rings which are tied together and supported by a series of struts. A mast, supported at the bottom of the container and located near its center, supports the pressurization distributor and vent line in the container ullage space.

21-31. LIQUID HYDROGEN CONTAINER

The LH_2 for the S-II stage is contained in a 2014 aluminum alloy container 671 inches long. The container is composed of a cylindrical section closed at the forward end by an ellipsoidal bulkhead and at the aft end by the common bulkhead (discussed above). The forward bulkhead and the common bulkhead are welded to the cylindrical section.

The forward bulkhead, designed to support flight pressurization loads, is constructed of gores, joined by welds. A manhole in the center of the bulkhead provides access to the container.

The cylindrical section, 539 inches long, is designed to carry flight pressurization, flight loads, and propellant loads due to acceleration. It is constructed of cylindrical segment panels. Located on the inside surface of each panel are integral longitudinal and circumferential stiffeners which form a rectangular grid pattern. The long side of the grid is in the longitudinal direction. The panels are welded together to form the cylinder. Bending moment and axial load are carried by the stiffeners. The longitudinal stiffeners are supported by internal rings constructed of 2024 aluminum alloy which are mechanically attached to the integral circumferential stiffeners. The internal stiffeners and rings provide the structure with a free-standing capability even though the container may be unpressurized.

First and second stage loads are transmitted from the aft skirt to the LH_2 container through a 15-inch long load transition cylinder (bolting ring). The load transition cylinder is bolted to the aft skirt, LOX and LH_2 containers.

The aft 25 inches of the LH_2 cylindrical section has tapered-integral longitudinal stiffeners on the exterior surface. These tapered stiffeners provide a load path transition from the load transition cylinder to the internal longitudinal stiffeners of the cylindrical section. Loads are carried by the LH_2 container to the forward skirt. The forward skirt is bolted to an internal boss at the forward end of the LH_2 cylindrical section.

Five outlet fittings for LH_2 engine lines, and one outlet fitting for fill and drain are located in the cylindrical section just forward of the joint with the LOX container. Two outlet fittings for LH_2 venting are provided in the forward bulkhead. A GH_2 diffuser is located at the top of the container.

The cylindrical section and forward bulkhead of the container are insulated externally. The insulation which limits the propellant boil-off rate to approximately 6-per cent per hour during launch operations, is bonded to the container walls. The insulation consists of a glass-phenolic honeycomb core filled with isocynate foam and covered externally with a nylon-phenolic skin sealed with Tedlar.

21-32. FORWARD SKIRT

A 7075 aluminum-alloy forward skirt transmits loads from the LH_2 container to the S-IVB stage. The forward skirt is 137 inches long. External longitudinal hat section stringers carry axial load and bending moment, and the skin carries shear load. Four internal rings support the skirt stringers and skin. The aft end of the skirt bolts to the cylindrical section of the LH_2 container. Loads, transmitted to the stringers and skin, are carried forward to the forward ring which transmits the loads to the S-IVB stage. The forward ring provides a mating face for attachment of the S-IVB stage in a field splice at MSFC station 2519.

A door is provided for access to the forward skirt. In addition, cutouts are provided for the umbilical plate and hydrogen vent. Antennas for the range safety and telemetry sets are mounted on the exterior of the skirt. There are also provisions for mounting a work platform.

Because of the decrease in diameter forward of the S-II stage, high aerodynamic heating is experienced on the forward skirt. To protect the structure, the forward 130 inches of the skirt is insulated with a bonded sandwich external insulation.

21-33. SYSTEM TUNNEL

The systems tunnel, located externally on the stage, contains cable, tubing and linear shaped charge runs between the aft skirt and the forward skirt. The tunnel fabricated from fiberglass, is constructed in sections to allow for thermal expansion and contraction and to provide easy removal for repair and maintenance. The tunnel sidewalls are interconnected by supports for the cables and tubing. The inner surface of the tunnel is insulated to protect electrical cabling from extreme temperatures.

21-34. S-IVB CONFIGURATION.

The S-IVB stage structure, Figure 21-13, is 712 inches (59.3 feet) long and 260 inches (21.7 feet) in diameter. An aft interstage, an aft skirt, a thrust structure, two propellant containers, and a forward skirt are structurally joined to make up the stage.

21-35. AFT INTERSTAGE.

Loads from the first and second stages are transmitted to the S-IVB stage through the 7075 aluminum alloy aft (S-II/S-IVB) interstage. The interstage is a conical frustum with an aft diameter of 396 inches, a forward diameter of 260 inches, and a length of 227.5 inches. External longitudinal hat section stringers carry the

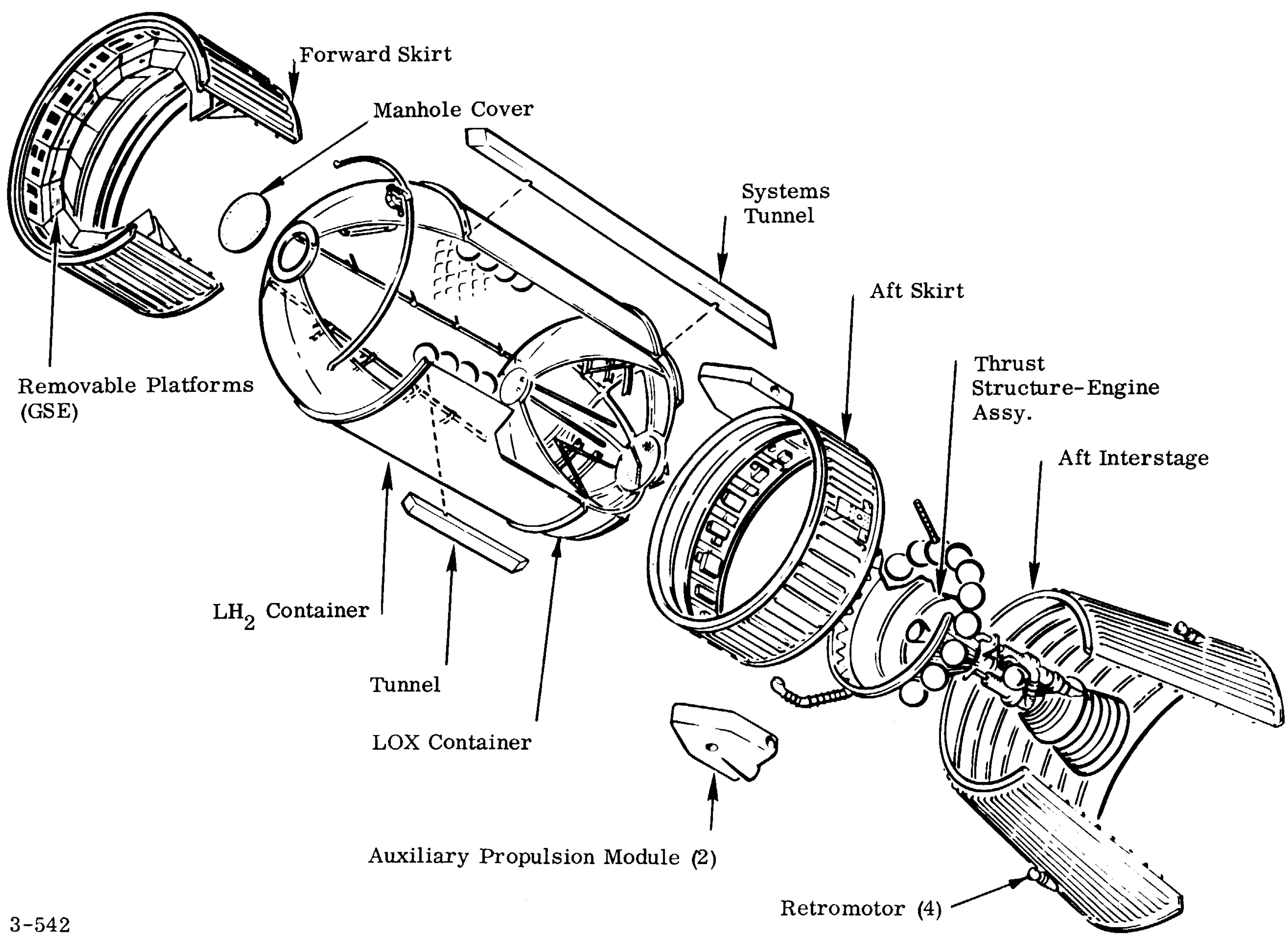

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Figure 21-13. S-IVB Stage Structure, Saturn V axial load and bending moment and the skin carries the shear load. The interstage skin and stringers are supported by an aft ring, five internal intermediate rings, and a forward ring. The aft and forward rings also provide mating surfaces for the second stage and aft skirt, respectively. The aft interstage is attached to the second stage of the launch vehicle (at MSFC station 2519) by a field splice. The interstage aft ring provides the mating face for the attachment. This attachment changes the loadpath from a cylinder to a cone creating a lateral load. The lateral load is sheared into the interstage skin by the aft ring; the axial load is transmitted directly from the aft ring to the stringers of the interstage. Loads from the stringers and skin are transmitted to the aft skirt through the forward ring of the interstage. The lateral component of the load, created by changing the loadpath from a cone to a cylinder is sheared into the interstage skin by the forward ring. An aft interstage door provides access to the propulsion installation. Retromotors are supported from the interior of the interstage aft of the separation plane.

21-36. AFT SKIRT

The loads from the aft interstage are transmitted to the LH_2 container through the aft skirt. The 7075 aluminum alloy skirt is 85.5 inches long. External longitudinal hat section stringers carry the axial load and bending moment; the skin carries the shear load. The skirt skin and stringers are supported by four rings; an aft ring, two internal intermediate rings and forward ring.

The aft skirt is bolted to the aft interstage through the aft ring which transmits the load to the skin and stringers. Loads are transmitted from the stringers and skin to the LH_2 container by the forward ring. The forward ring bolts to the container.

There are three cutouts in the aft skirt; two for routing the engine line from the LH_2 container to the engine, and one for the aft umbilical plate. Supports for the auxiliary propulsion modules are located on the exterior of the aft skirt.

21-37. THRUST STRUCTURE

The thrust structure transmits engine thrust loads to the LOX container. The 7075 aluminum-alloy structure is a conical frustum with an aft diameter of 34 inches, a forward diameter of 168 inches, and a length of 63 inches. The skin slope is tangent to the LOX container bulkhead at the attachment point. Lateral loads (resulting from engine gimballing) and axial loads are transmitted from the gimbal bearing joint to the LOX container aft bulkhead through the thrust structure skin and stringers.

The skin and stringers are supported by an aft ring, two internal intermediate rings, and a forward ring. Lateral loads are sheared from the aft ring into the thrust structure skin. Axial loads are transmitted from the aft ring though external longitudinal hat section stringers to the forward ring. The forward ring bolts to a ring on the LOX container aft bulkhead.

Cutouts are provided in the thrust structure to accommodate the LOX and LH_2 engine lines; two doors are provided for structure access. Four helium bottles are mounted externally on the thrust structure.

21-38. LIQUID OXYGEN CONTAINER

The LOX for the S-IVB stage is contained in a 2014 aluminum-alloy container. Two hemispherically shaped bulkheads, an aft and a common, are welded together through two angle section compression rings to form the container.

The aft bulkhead is constructed of nine gores and a circular center piece fusion welded together. The resulting hemisphere, designed to withstand flight pressurization and propellant loads due to acceleration, has a spherical radius of 130 inches.

The other bulkhead, termed a common bulkhead because it is common to both the LOX and LH_2 containers, is a spherical segment (less than a hemisphere) with a radius of 130 inches. This bulkhead is of honeycomb sandwich construction with 2014 aluminum alloy facing sheets bonded to a fiberglass core. The common bulkhead has sufficient insulating properties to keep the LOX from freezing during a 12 hour ground hold or a 4-1/2 hour orbit.

A ring attached to the aft bulkhead provides a mounting surface for the engine thrust structure. Engine thrust loads are transmitted through the ring to the aft bulkhead, and then into the LH_2 container cylindrical section.

Liquid oxygen sloshing is controlled by internal ring baffles supported by a sheetmetal conical frustum attached to the aft bulkhead. A screen, also attached to the aft bulkhead, provides vortex suppression. Container access is provided by removal of the engine outlet fitting.

21-39. LIQUID HYDROGEN CONTAINER

The LH_2 for the S-IVB stage is contained in a 2014 aluminum-alloy container 513 inch long. The container is composed of a cylindrical section closed at the forward end by a hemispherical bulkhead, and closed at the aft end by the LOX container (discussed above). The forward bulkhead and LOX container aft bulkhead are fusion welded to the cylindrical section. The cylindrical section and forward bulkhead are internally insulated with polyurethane foam bonded to the container walls. The insulation limits hydrogen boiloff during launch operations and flight.

The forward bulkhead, designed to withstand flight pressurization loads, is constructed of nine gores and a circular centerpiece welded together. The bulkhead has a spherical radius of 130 inches. Two openings are provided in the bulkhead; one for container access and the other for the hydrogen flight vent line.

The LH_2 cylindrical section, 268 inches long, is designed to carry flight pressurization, flight loading, and propellant loads due to acceleration. The section is composed of seven panels. Each panel is milled to a square waffle pattern with a 45-degree skew angle. The panels are welded into a cylinder. The internal waffle stiffeners provide sufficient buckling strength to give the structure a free-standing capability when the container is unpressurized. An external ring is welded to the cylindrical section at the tangent point of the aft bulkhead. First and second stage loads are transmitted through the ring to the LH_2 container cylindrical section by the aft skirt. The loads are then transmitted from the cylindrical section to the forward skirt through a second external ring. This ring is welded to the cylindrical section at the tangent point of the forward bulkhead.

A LH_2 line outlet is provided just below the weld that joins the cylindrical section to the aft bulkhead of the LOX container. A box shaped screen covers the outlet in order to suppress the vortex created by LH_2 flow.

21-40. FORWARD SKIRT

The cylindrical forward skirt transmits the loads from the LH_2 container to the instrument unit. The skirt, 122 inches long, is fabricated of 7075 aluminum alloy. Five rings support the forward skirt skin and external longitudinal stringers; an aft ring, three internal intermediate rings, and a forward ring.

The aft ring bolts to the LH_2 container. From this ring loads are transmitted to the stringers and skin. Axial load and bending moment are carried by the skin. Loads are transmitted from the skin and stringers to the instrument unit by the forward ring. The forward ring provides an interchangeable mating face for the attachment of the instrument unit (a field splice at MSFC station 3223).

The forward skirt has provision for a removable service platform. In addition, the skirt has cutouts for an umbilical plate, and ground and flight hydrogen vents. A door in the instrument unit provides access to the forward skirt.

21-41. SYSTEMS TUNNEL AND EXTERNAL FAIRINGS

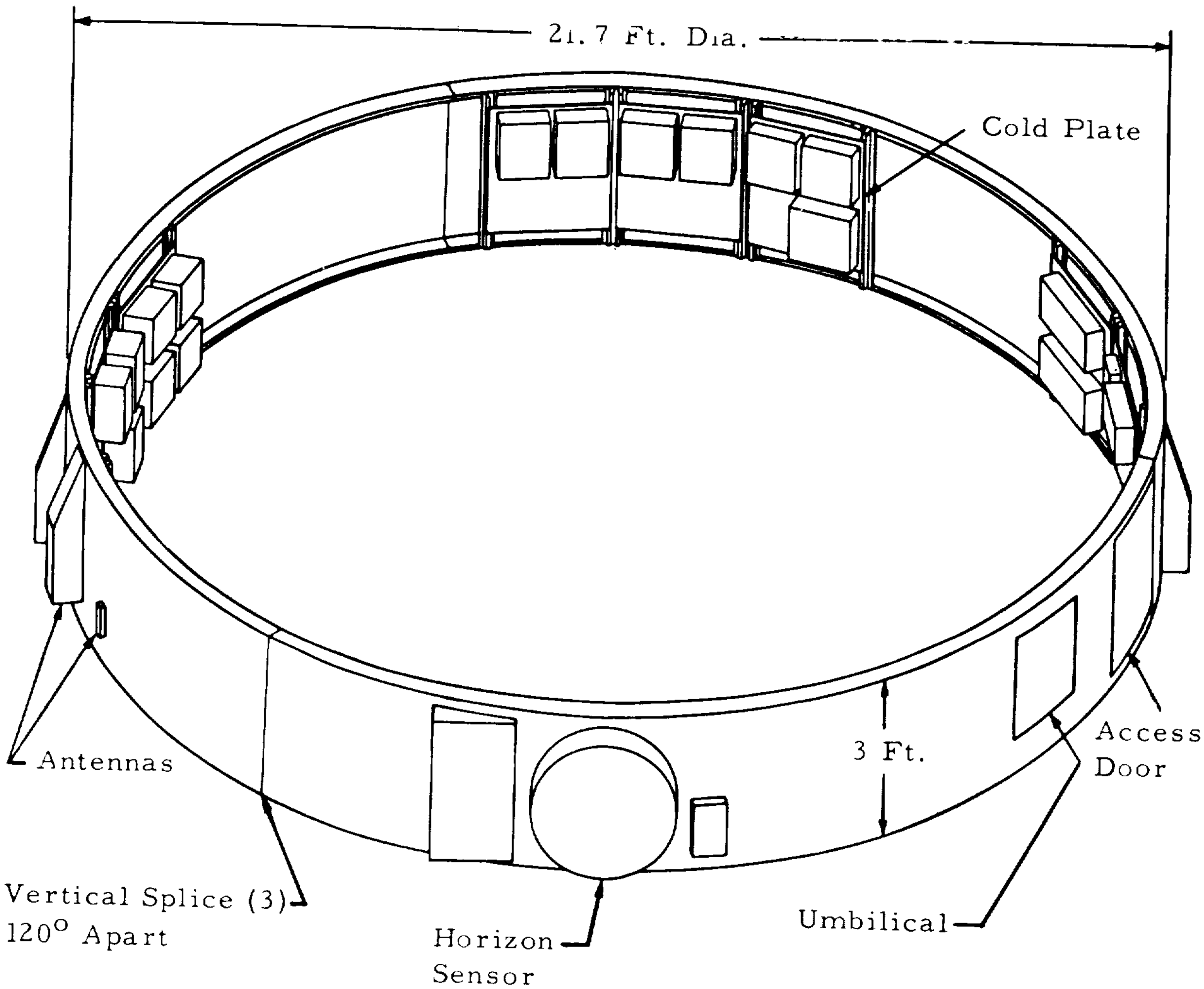
The systems tunnel is located externally on the third stage body. The tunnel extends from the aft skirt to the forward skirt and accommodates cable, tubing and linear shaped charge runs.

The third stage has several fairings which are designed to carry aerodynamic pressure and thermal loads. An LH_2 engine line fairing is located on the aft skirt. Fairings for the two auxiliary propulsion modules are also located on the aft skirt and have cutouts in each side and top for the attitude control nozzles.

21-42. INSTRUMENT UNIT CONFIGURATION.

The instrument unit structure (Figure 21-14) transmits loads from the S-IVB stage to the payload. The structure is 260 inches (21.7 feet) in diameter and 36 inches long. It is constructed of three, 120 degree cylindrical panels joined with longitudinal field splices.

The instrument unit is of honeycomb sandwich type construction. Loads are transmitted to the sandwich panels through the aft ring which is attached to the S-IVB stage in a field splice at MSFC station 3223. The loads are transmitted by the



3-521

Figure 21-14. Instrument Unit, Saturn V

panels to the forward ring which attaches to the payload in a field splice at MSFC station 3259. Brackets bonded to the sandwich panels provide support for the electrical and electronic equipment mounting plates. The equipment is grouped so that clearance is provided for the LEM landing gear which extends into the instrument unit. A load carrying door provides access to the instrument unit and cutouts are provided for the umbilical plate and horizon sensor.

CHAPTER 4

SECTION XXII

PROPULSION

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SECTION XXII.
PROPULSION

22-1. REQUIREMENTS.

The Saturn V propulsion system, in the normal operational mode, is required to launch and inject a 90,000 pound net payload into the 72 hour lunar transfer trajectory. The normal operational mode of Saturn V is defined as: suborbital start of the S-IVB stage followed by coast in earth parking orbit, and then restart of the S-IVB to provide the velocity required for injection into lunar transfer trajectory. The net payload consists of all weight at the time of injection forward of the instrument unit. In addition, attitude control of the launch vehicle and payload is provided during earth orbit and for two hours after insertion into lunar transfer trajectory, including the period of spacecraft reorientation.

A three-stage launch vehicle provides the necessary impulse. Table 22-1 contains the nominal staging parameters.

Table 22-1. Nominal Staging Parameters

Stage	Altitude	Velocity
S-IC	34.3 nm	5300 kts
S-II	99 nm	13,200 kts
S-IVB First burn (Earth orbit injection)	100 nm	15,100 kts
S-IVB Second burn (Lunar transfer injection)	155 nm	21,200 kts

Thrust-vector control is required to maintain vehicle attitude orientation and angular rates as defined by the control system during main stage.

An additional series of impulses are required to ensure successful staging and stage operation. Both retro thrust to decelerate the spent stage, and ullage thrust to accelerate forward stages and spacecraft are necessary to aid separation.

The ullage thrust also settles the propellants in the aft end of the containers ensuring a sufficient suction head to prevent pump cavitation at engine start.

During the launch phase a rapid fill and drain capability is required of the propellant systems due to the highly volatile properties of the cyrogenic propellants (LH_2 and LOX). Provisions for the purging of the propellant containers and feed lines before filling or after draining operations are required as part of the propellant storage and feed system. During the ascent, orbit and lunar injection phases, the system must be capable of storing the propellants, minimizing boiloff, and delivering them as required to the engines.

22-2. OPERATION

The operation of the propulsion system begins with the launch phase and concludes with the separation of the spacecraft from the launch vehicle. The events of the propulsion sequence is presented in Table 22-2.

22-3. LAUNCH PHASE.

During the count down, the propellant containers are purged, loaded, pressurized and conditioned; pressure storage spheres are purged and charged; and the main stage engines are purged and conditioned prior to starting. A few seconds prior to lift off, the five S-IC stage engines are started in a predetermined sequence. The center engine is started followed by diametrically opposite engines in pairs. The engines are started in response to a ground command. The launch phase ends at lift off.

22-4. ASCENT PHASE.

A total nominal thrust of 7,500,000 pounds is provided at lift off resulting in a thrust-weight ratio of 1.25:1.

As a result of decreasing ambient pressure as the vehicle ascends, the stage thrust increases to 8,635,000 pounds prior to center engine cut off. Thrust vector and attitude control are provided by the four outboard gimballed engines (Figure 22-1) in response to commands from the control systems. Engine cut-off results from a propellant depletion (level) signal cutting off the center engine a few seconds before the outboard engines. Planned cutoff is utilized rather than

[redacted]ASSIFIED

Event	Veh.	Launch	
Propellant Loading and Conditioning			
Pressurant Loading			Separation
Start Sequencer	S-IC		
Mainstage	S-IC		
Liftoff			
IECO	S-IC		
OECO	S-IC		
Separation Command			
Ullage Motors Firing	S-II		
Retromotor Firing	S-IC		
First Plane Separation	S-IC/S-II		
Second Plane Separation	S-II		
Thrust Chamber Chill	S-II		
Engine and Feed Line Chill	S-II		
Start Command	S-II		
Mainstage	S-II		
Propellant Depletion Cutoff Signal	S-II		
Separation Command			
Separation	S-II/S-IVB		
Retromotor Firing	S-II		
Thrust Chamber Chill	S-IVB		
Engine and Feed Line Chill	S-IVB		
Start Command	S-IVB		
Mainstage	S-IVB		
APS Ullage Engines	S-IVB		
APS Roll Control Engines	S-IVB		
APS Attitude Control Engines	S-IVB		
APS GH_2 Venting Ullage Engines	S-IVB		
Engine Cutoff Signal	S-IVB		
Orbital Parameters			
Translunar Parameters			
Separation Command			
Separation	IU/PL		

Legend : ▲ Event ; ▬ Operating ; ▬ ▬ ▬ Intermit

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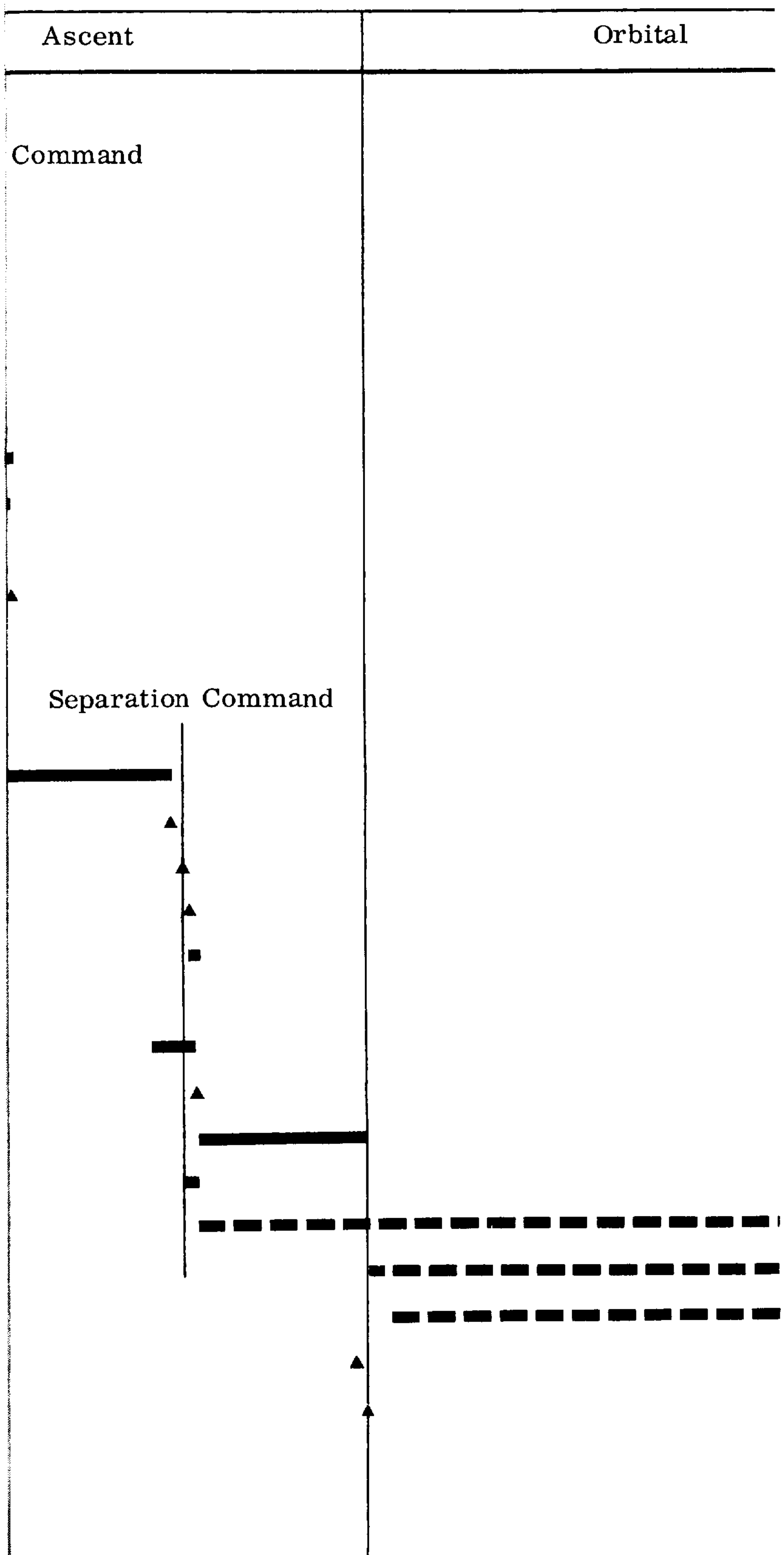
FOLDOUT FRAME 1



tent Operation.

FOLDOUT FRAME 2

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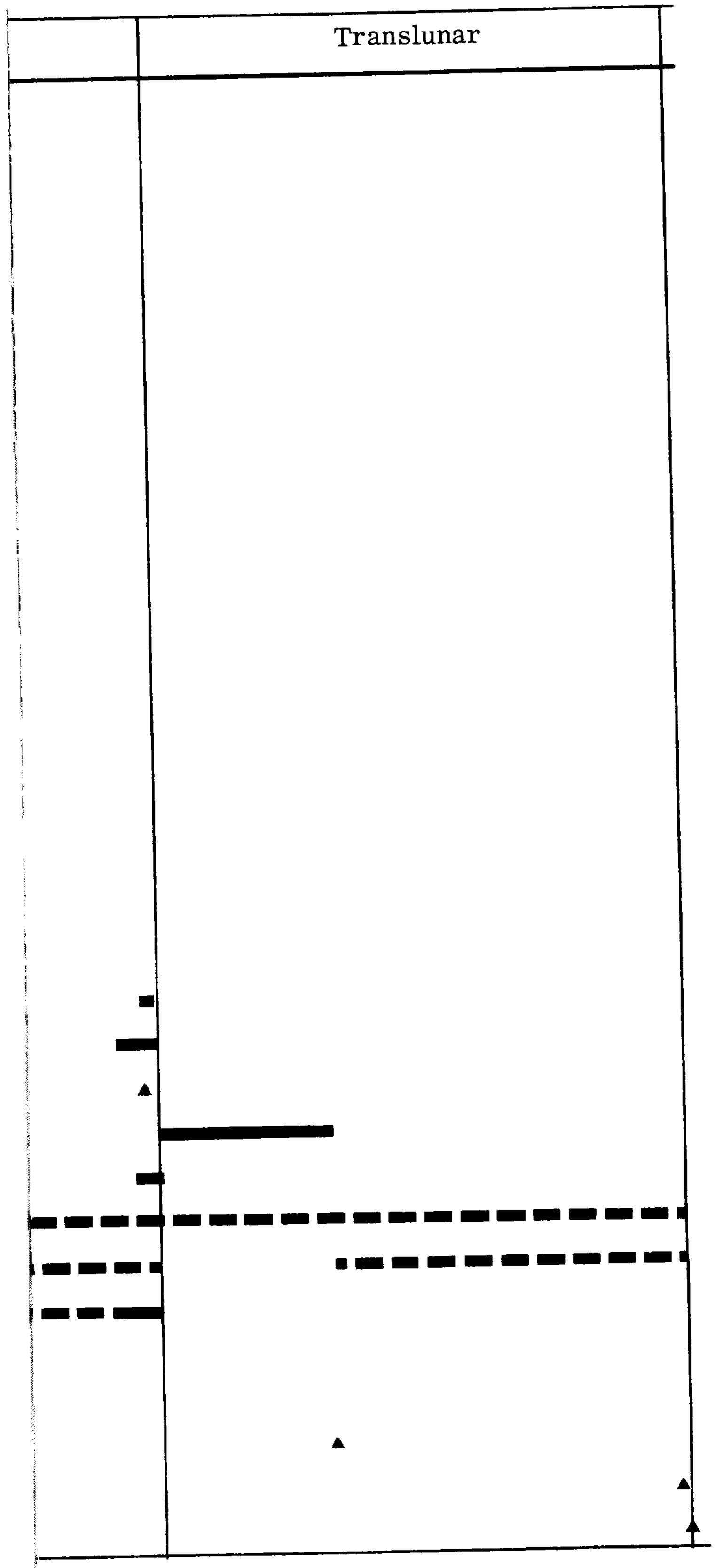
Table 22-2. Saturn V Propulsion Sequence



FOLDOUT FRAME 3

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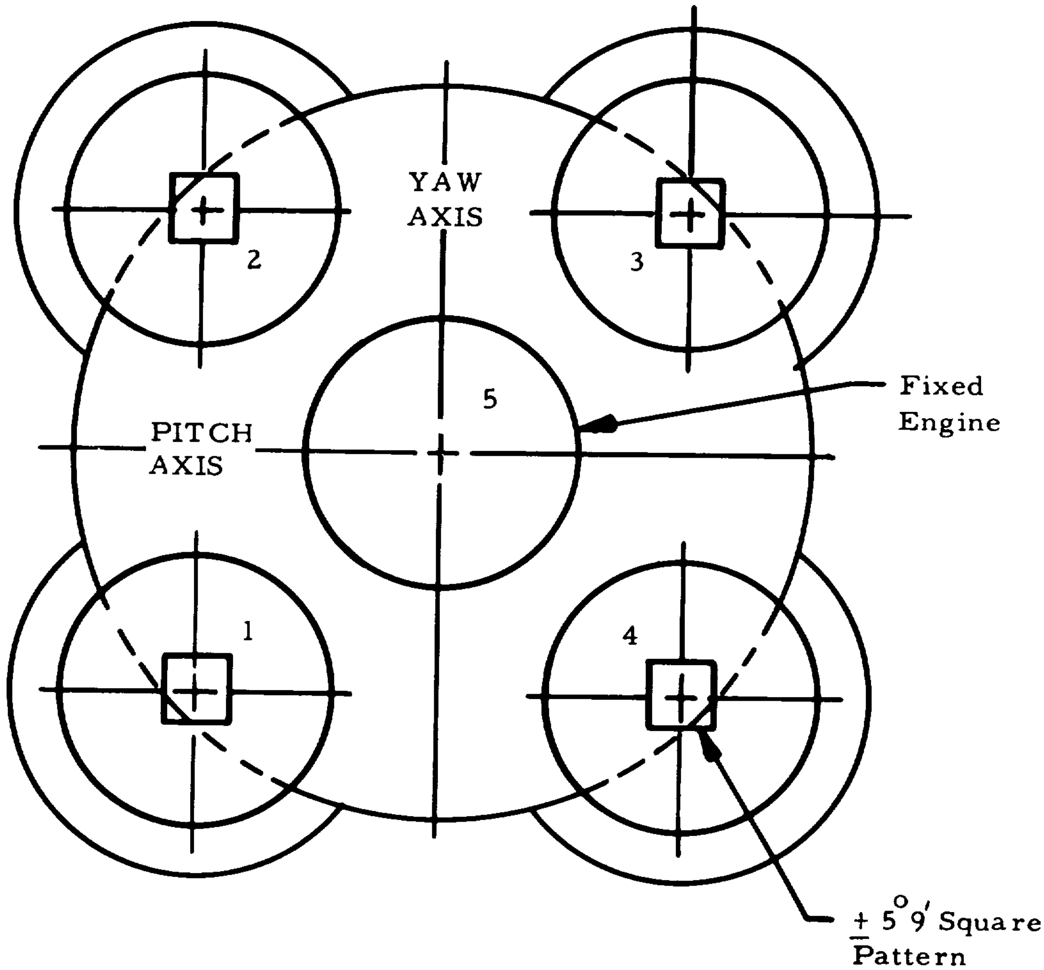
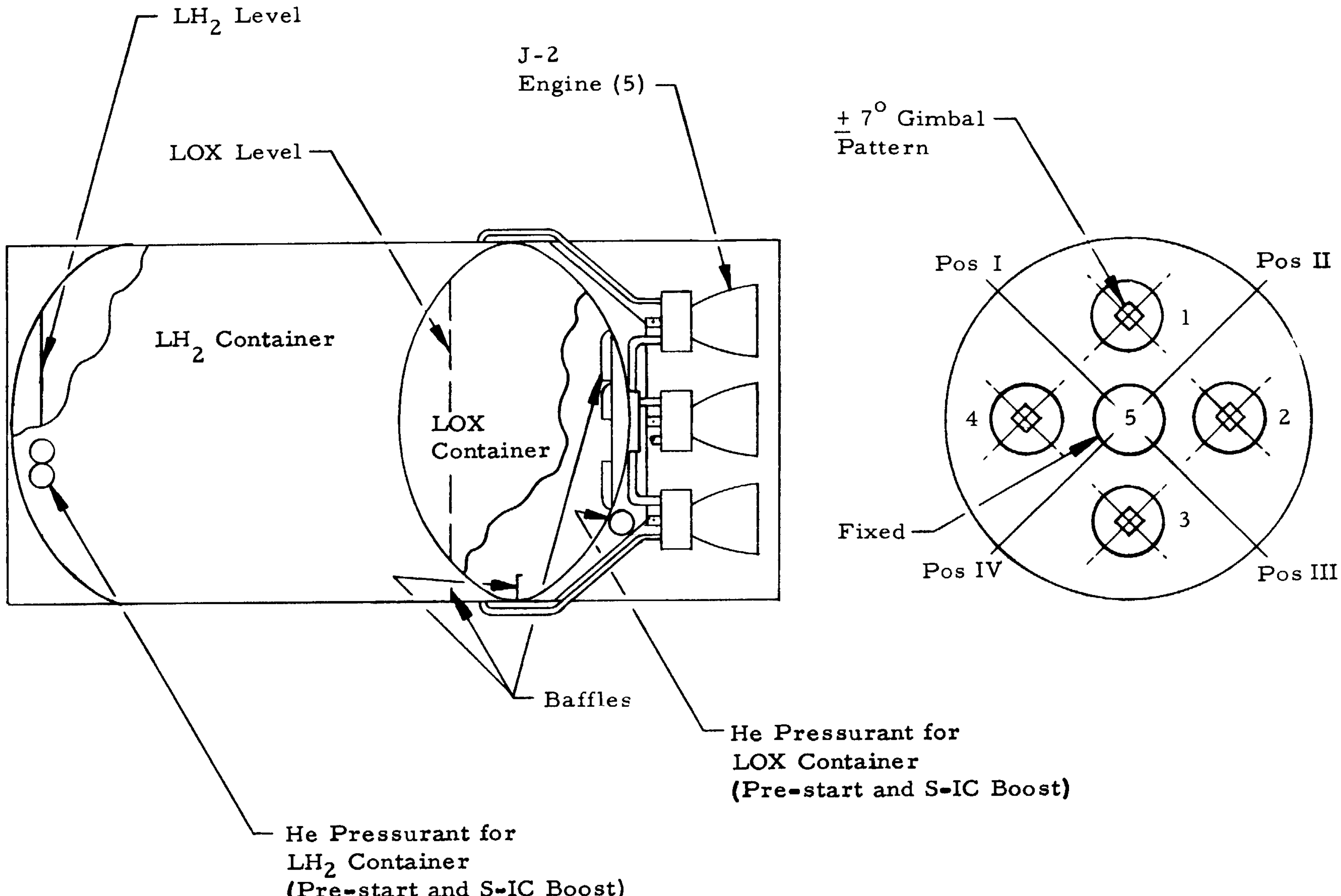
burn out to achieve predictable cutoff impulses and thrust vectors. This minimizes the possiblity of tumbling and chugging at separation.



3-143

Figure 22-1. Engine Location and Gimbal Pattern, S-IC

A chill down of the S-II stage engines begins prior to liftoff with the chill down of the thrust chambers. Propellants are circulated through the pump and feed lines during first stage operation until a few seconds before first plane separation. The five engines of the S-II stage which provide a total thrust of 1,000,000 pounds, are started in unison in response to a command from the instrument unit after first plane separation. Second plane separation and the jettisoning of the S-IC/S-II interstage, occurs about 30 seconds later. Thrust vector and roll control are provided by gimballing the four outboard engines (Figure 22-2) in response to com-



3-150

Figure 22-2. Propulsion System and Gimbal Pattern, S-II mands of the control system. Engine cutoff results from a propellant depletion (level) signal cutting off the five engines simultaneously.

The S-IVB engine is chilled down by circulating propellants through the pumps and feedlines prior to staging. The chill down of the thrust chamber is completed after separation and prior to ignition of the engine.

The S-IVB stage engine, providing a thrust of 200,000 pounds, is ignited in response to a start command from the instrument unit. Thrust vector control is provided by gimballing the main engine and roll control is provided by firing the roll control engines of the auxiliary propulsion system in response to the commands of the control system. Engine cutoff occurs as the result of termination of an electrical signal from the instrument unit. The signal is terminated such that the total impulse delivered by the engine subsequent to the signal results in a velocity-to-go requirement of zero at thrust termination. The ascent phase ends upon attainment of proper orbital parameters.

22-5. ORBITAL PHASE.

During the orbital phase, the auxiliary propulsion system provides attitude stabilization and GH_2 venting ullage. Attitude stabilization is provided by firing the attitude and roll control engines in response to the commands of the control system, and GH_2 venting ullage is provided by firing the ullage engines in response to a stage command. Prior to the restart of the main engine, the GH_2 venting ullage engines and later the main ullage engine are fired to settle the propellants during circulation for pump and feed line chilldown and during restart.

Commands from the control system also fire the attitude control engines to provide the proper attitude before restarting the main engine. The orbital phase ends with the achievement of mainstage.

22-6. TRANSLUNAR PHASE.

During mainstage, in response to the commands of the control system, thrust vector control is provided by gimballing the main engine, and roll control is provided by firing the roll control engines of the auxiliary propulsion system. Thrust termination occurs upon attainment of the 72-hour lunar transfer trajectory. Until separation from the spacecraft, attitude stabilization is provided by firing the attitude and roll control engines in response to the commands of the control system.

The translunar phase for the propulsion system ends with separation from the spacecraft.

22-7. S-IC STAGE PROPULSION SYSTEM.

Three stages, the S-IC, S-II and SIVB, and an instrument unit comprise the launch vehicle, Figure 19-1. The instrument unit provides initiation and control commands for the propulsion system. (Refer to Paragraph 20-1.) Functionally, the S-IC propulsion system is composed of five Rocketdyne F-1 liquid-rocket engines and a propellant system.

22-8. ENGINE.

The F-1 engine, Figure 22-3, is a single start, fixed thrust, bi-propellant system, using LOX as oxidizer and RP-1 as fuel, turbopump lubricant and control system working fluid. Four engines of the S-IC Stage, equally spaced on a 364-inch diameter, are gimbal mounted for flight control. The maximum gimbal angle is a $\pm$ 5-degree, 9-minute square pattern, Figure 22-1. The fifth engine is fixed on the centerline of the stage. Nominal engine rated thrust at sea level is 1,500,000 pounds. Engine performance parameters are given in Table 22-3.

Table 22-3. Performance Parameters, F-1 Engine

Item	Parameter
Oxidizer	Liquid Oxygen
Fuel	RP-1
Number of thrust chambers	1
Number of turbopumps	1
Number of gas generators	1
Sea-level thrust (total)	1,500,000 lbs.
Dry weight including accessories	16,825 lbs.
Thrust duration	150 sec.
Sea-level specific impulse	265.4 sec.

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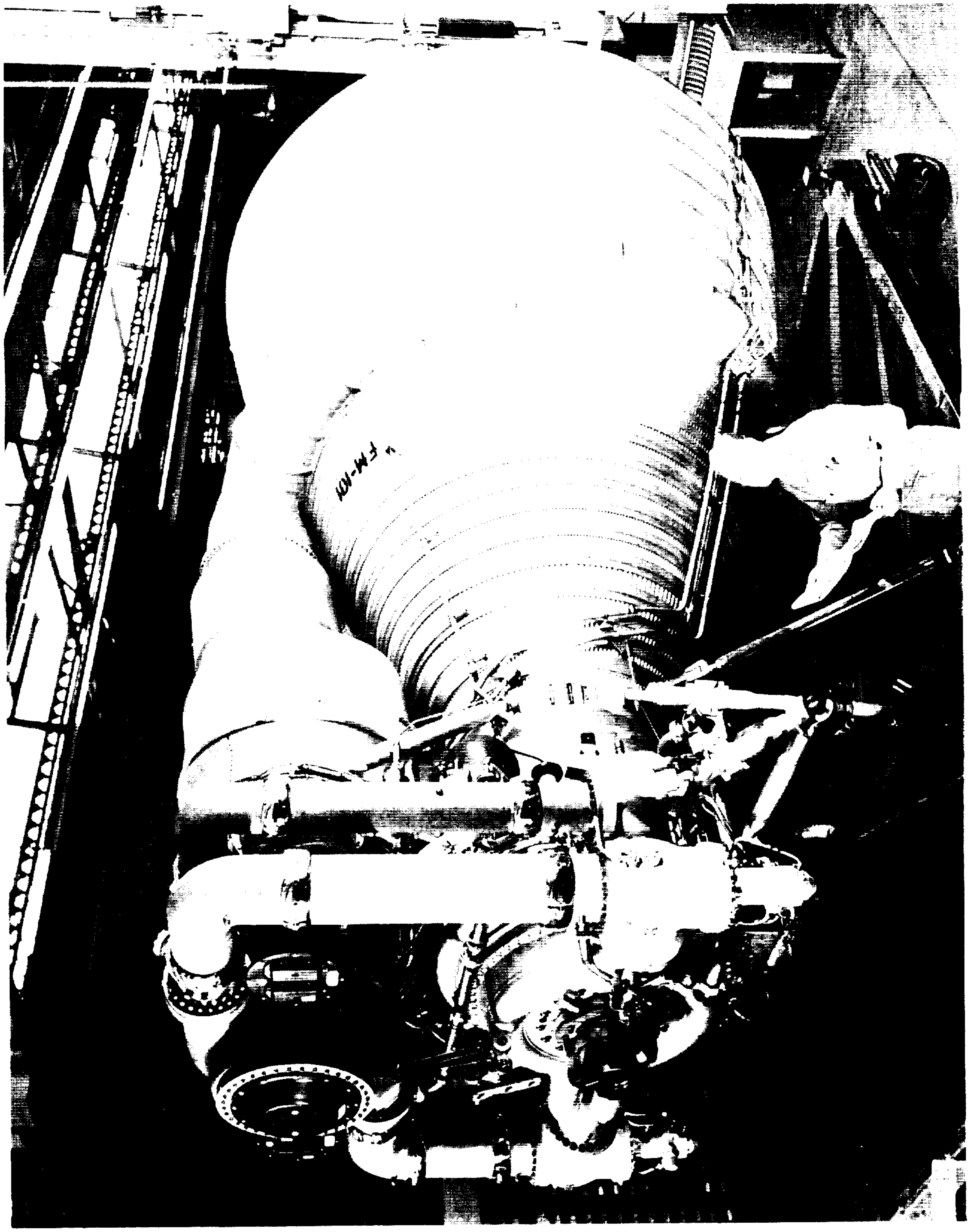


Figure 22-3. F-1 Engine

[redacted]

Table 22-3. Performance Parameters, F-1 Engine (Cont'd)

Item	Parameter
Total propellant flowrate (sea level)	
Thrust chamber and gas generator	5,685 lb./sec.
Mixture ratio, W_o/W_F	2.77:1
Flowrates (thrust chamber and gas generator)	
Oxidizer	4064 lb./sec.
Fuel	1790 lb./sec.
Diameter	148 in.
Length	222 in.

The primary components of the engine are the thrust chamber, gas generator, turbopump, propellant valves, ignition system, engine control system and electrical system, Figure 22-4. Each engine is attached to the thrust structure with a gimballed bearing joint. A brief description of each major component of the engine follows.

22-9. Thrust Chamber. In the thrust chamber the propellants are mixed, burned, and expelled through a nozzle. The thrust chamber includes the following major components:

a. A LOX dome which distributes LOX to the propellant injector and provides mounting for the gimbal bearing.

b. A fuel manifold which distributes fuel to the propellant injector.

c. A propellant injector which meters the propellants and injects them into the combustion area.

d. A thrust chamber body composed of tubular walls through which fuel is circulated to provide regenerative cooling and fuel pre-heating.

e. A turbine exhaust cooled thrust chamber extension.

22-10. Gas Generator. The gas generator produces hot gases to drive the turbopump. The generator operates on LOX and fuel, which are bootstrap-fed from the high pressure side of the propellant pump. The assembly consists of

[redacted]

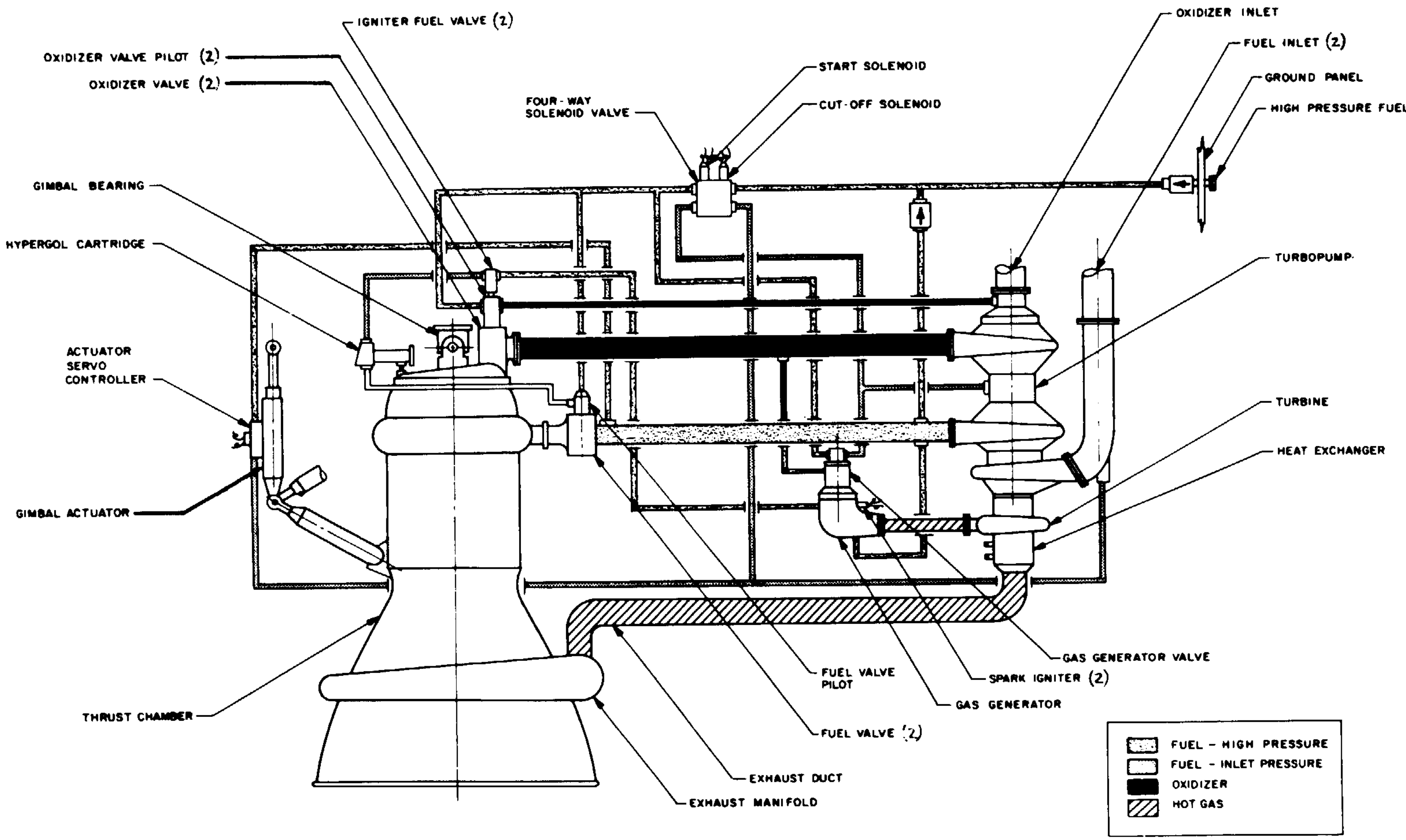


Figure 22-4. F-1 Engine Schematic

[redacted]

the following components:

a. Gas generator propellant control valve which controls the flow of propellants into the generator.

b. An auto ignitor used to ensure ignition of the propellants.

c. A combustion chamber which provides space for burning the propellants.

22-11. Turbopump. The turbopump assembly supplies LOX and fuel to the engine thrust chamber at the proper pressure and flow rates to maintain engine rated thrust. In addition, the fuel pump supplies high-pressure fuel to the engine control system. The turbopump consists of a centrifugal fuel pump, centrifugal LOX pump and a turbine mounted on a common shaft.

22-12. Propellant Valves. Propellants are admitted to the thrust chamber through four normally-closed valves, two in parallel for LOX and two in parallel for fuel. The oxidizer valves are initially opened by ground supplied control fluid pressure. The fuel valves are operated by fuel pressure acting through the ignition monitor valve.

22-13. Ignition System. The thrust-chamber ignition system consists of the hypergol cartridge and the ignition fuel duction. Ignition occurs when fuel from the fuel pump outlet enters the hypergol manifold assembly, ruptures the burst diaphragm and forces the hypergol fluid through the injector and into the combustion chamber.

22-14. Engine Control System. The engine control system hydraulically operates the fuel, oxidizer and gas generator valves in the proper sequence to start the engine, maintain rated thrust during powered flight and shut down the engine at cutoff. The working fluid for this system is high-pressure fuel tapped from the inboard discharge line of the fuel pump. A ground system, connected through the customer connect panel, provides high-pressure fuel to the engine control system for opening the valves prior to buildup of fuel pump discharge pressure. The system is composed of the following components:

a. A four-way solenoid valve which transfers high-pressure control fuel to the proper valve actuators in response to an electrical command.

b. Two sequence valves which are mechanically actuated by the position of the main oxidizer valve. The sequence valve opens when the oxidizer valve is approximately 80 percent open and closes when the oxidizer valve is 20 percent closed.

c. Filters, check valves and interconnecting tubing.

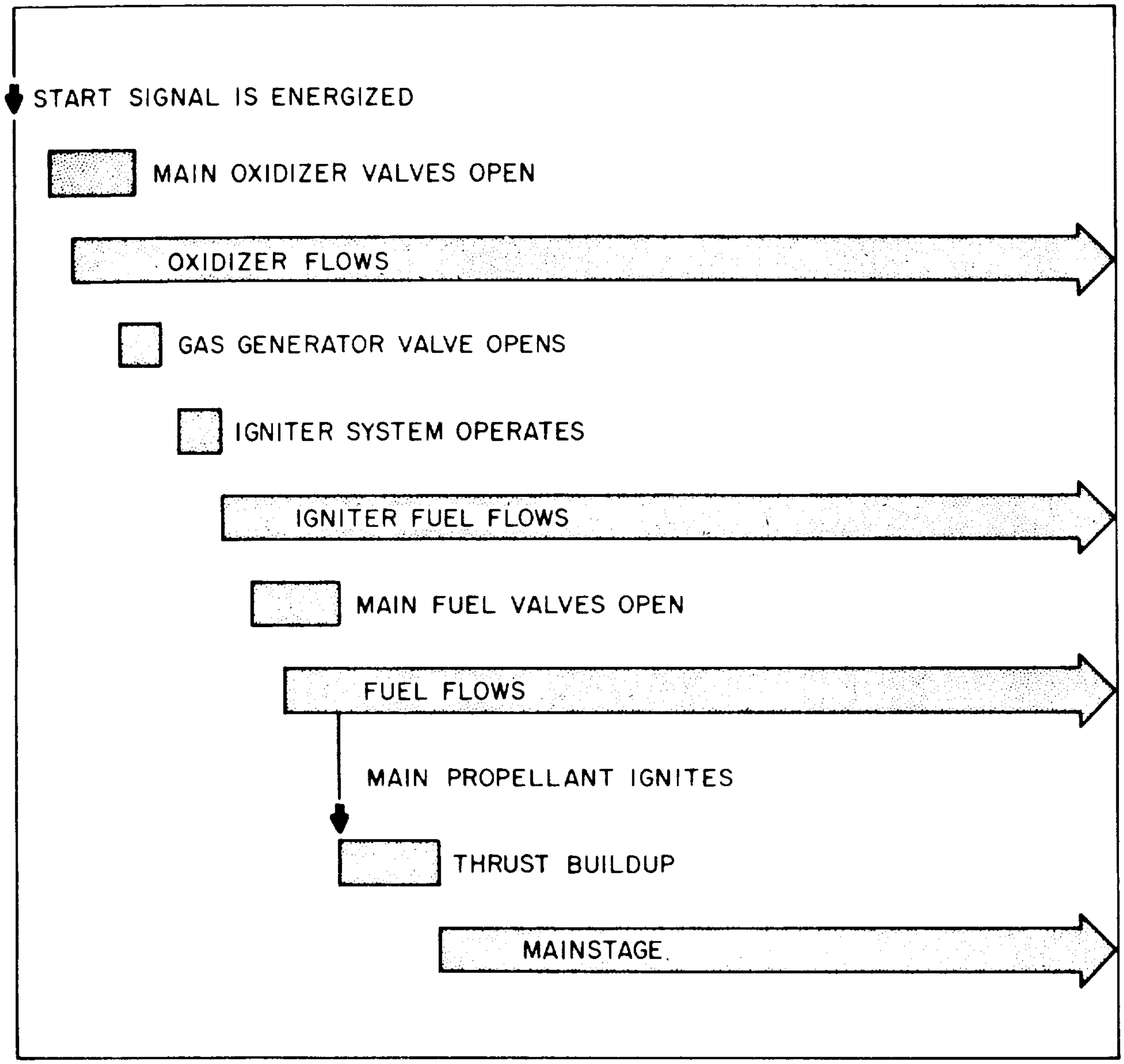
22-15. Electrical System. Electrical connections are required for start-stop sequencing, operating the turbopump heating element, and transmitting engine instrumentation and control information. Electrical power for engine actuation and controls is obtained from the main distribution system.

22-16. Gimbal Bearing Joint. A gimbal bearing joint attaches each engine to the vehicle thrust structure, absorbs thrust loads, and permits each outboard engine to be moved for thrust vector control. An additional structural interface exists where outrigger arms on the thrust chamber are attached to the hydraulic thrust vector actuators. (The center engine employs fixed links instead of actuators.)

22-17. ENGINE OPERATION.

The engines are started in a predetermined sequence. The center engine is started first followed by diametrically opposite engines in pairs as commanded by the start sequencer. During standby, control pressure fluid is supplied from the ground through a quick disconnect coupling to the gimbal hydraulic supply manifold and to the four-way solenoid valve. With the checkout valve in the ground position, the control pressure fluid is returned to the ground supply through a quick disconnect coupling.

22-18. Engine Start Sequence. (Figure 22-5) Prior to start the checkout valve is actuated to the vehicle position which allows the control pressure fluid to return to the fuel pump inlet. An electrical signal from the ground energizes the gas generator spark ignitor and when it is operating properly, as determined by a self-monitoring circuit, an electrical signal is impressed on the start solenoid of the four-way solenoid valve. Upon actuation of the start solenoid, control fluid is vented from the actuator closing ports of the main fuel valves, main oxidizer valves and the gas generator valve. Control fluid is directed from the four-way solenoid valve to the opening ports of the main oxidizer valves. Opening of the oxidizer valves admits LOX under container head pressure to the thrust chamber and actuates mechanical linked sequence valves allowing control fluid to be admitted to the opening port of the gas generator valve. Opening of the gas generator valve allows LOX and fuel under container head pressure to enter the gas generator combustion chamber. LOX flows from downstream of the turbopump to the gas generator



3-175

Figure 22-5. Engine Start Sequence

combustion chamber. Fuel flows from downstream of the turbopump to the gas generator combustion chamber. The propellant mixture is ignited in the gas generator combustion chamber by the spark ignitor and the hot gases are directed to the turbine which drives the propellant turbopump. The expended gases are discharged through the heat exchanger into the thrust chamber extension where they are ignited by the thrust chamber extension ignitors. With turbopump acceleration, the LOX and fuel outlet pressures increase resulting in an increased flow rate of propellants to the gas generator and LOX to the thrust chamber.

The gas generator combustion chamber is cooled by fuel flowing from the fuel turbopump outlet around the chamber and returning to the fuel turbopump inlet. The fuel ball valve mechanism of the gas generator valve is protected from the cold environment of the LOX by circulating fuel through the assembly. The flow is from the four-way solenoid valve to the gas generator valve, and returning to the four-way solenoid valve. The flow rate is controlled by an orifice.

When the discharge fuel pressure of the turbopump reaches the burst pressure in the hypergol cartridge, diaphragms rupture. This permits fuel and hypergol to flow to the thrust chamber mixing with LOX, and primary ignition is established. Rupturing of the hypergol container diaphragms actuates a mechanical safety device which allows the ignition monitor valve to be opened by thrust chamber pressure buildup as sensed through the checkout valve. Opening of the ignition monitor valve directs fuel pressure from the four-way solenoid valve to the opening ports of the fuel valves. Opening of the main fuel valve permits fuel to enter the thrust chamber mixing with the LOX and transition from primary ignition to mainstage ignition occurs.

The fuel cavity between the turbopump fuel impeller and volute is maintained at the controlled pressure for impeller thrust balancing by leakage past the volute ring seal. Liquid oxygen line surge is controlled by orifices located between the LOX turbopump inlet and outlets.

The turbine and turbopump lubrication system begins operation when the fuel pump outlet pressure exceeds the vehicle container head fuel pressure. A bearing coolant valve, consisting of a filter, check valve and orifice, opens with the increase of pressure, and circulates fuel through the bearing to an overboard drain. The lubrication ceases with fuel pressure decay at engine shutdown.

The single pass, dual-medium heat exchanger receives the turbine exhaust gases which are used to heat and vaporize the LOX and to heat and expand the helium that is used to pressurize the propellant container ullage. A portion of the LOX supplied to the LOX dome is diverted to the heat exchanger. The GOX formed in the heat exchangers of the individual engines is routed into a common pressurization duct containing a GOX flow control valve which regulates the flow of GOX from the LOX container. Helium supplied from the helium cylinders is routed through the valve and manifold assembly to the individual heat exchangers where the helium is expanded and then routed through a common pressurization duct into the fuel container.

22-19. Engine Stop Sequence. (Figure 22-6) Engine cutoff is initiated by an electrical signal which energizes the stop solenoid on the four-way solenoid valve. The stop solenoid closes the pressurizing port venting the entrapped fluid, and directs closing pressure to the gas generator valve, the oxidizer valves and

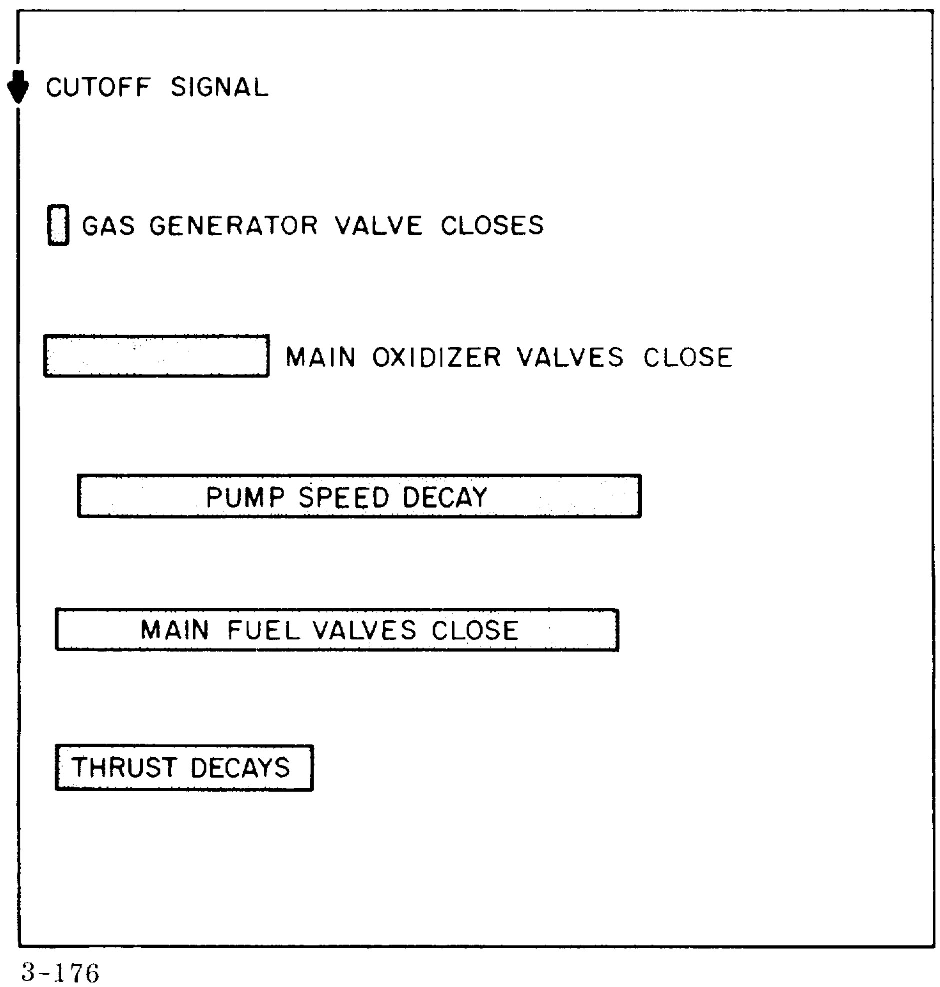


Figure 22-6. Engine Cutoff Sequence

the fuel valves. To maintain a fuel-rich engine cutoff the main LOX valves close first. The rate of closure is determined by orifices in the vent ports of the main LOX valves.

22-20. PROPELLANT SYSTEM.

The propellant systems, Figure 22-7, consist of the containers, ducts, valves, and flexible joints required to deliver propellants to the engine turbopumps. The maximum usable propellant capacity from liftoff to cutoff signal, including propulsion performance reserve, is 4,400,000 pounds.

22-21. FUEL FEED SYSTEM.

The fuel container is connected to the engine turbopumps through ten, 12-inch suction lines, two for each engine. A combination of fluid head, ullage gas pressure, and vehicle acceleration forces the fuel to the engine turbopump inlets at a nominal flowrate of 7800 gpm per line. Each line is installed with gimbal and expansion joints to allow for alignment tolerances, thermal expansion and engine gimballing.

A pneumatic piston-operated ball valve is located in each line at the container outlet to function as a prevalve. The prevalves serve as an emergency control to stop fuel flow in the event of engine failure. A single solenoid valve using 750 psig nitrogen from the control pressure system operates these prevalves. The RP-1 line inlets are equipped with anti-vortexing devices, and the turbopump inlets are equipped with pressure volume compensating ducts to allow line movement while maintaining constant fluid volume. A pressure balancing feature minimizes pressure loads transmitted to the turbopump assembly. When the fuel depletes to a pre-determined level on the container, propellant level sensors will shut down the inboard engine. Approximately six seconds later, a timer or level sensor will initiate outboard engine cutoff. During this interval, most of the remaining propellants are used.

22-22. OXIDIZER FEED SYSTEM.

The oxidizer container is mounted above the fuel container. Five 20-inch LOX feed lines pass through the fuel container in separate 25-inch tunnels. At the tunnel outlet the line size is reduced to 17 inches. A combination of fluid head, ullage gas pressure and vehicle acceleration forces the LOX through the five suction lines to the engine turbopump inlets at a nominal flow rate of 24,630 gpm per line. Each line is installed with gimbal and expansion joints to allow for alignment tolerances, thermal expansion, and engine gimballing. A pneumatic piston-operated ball

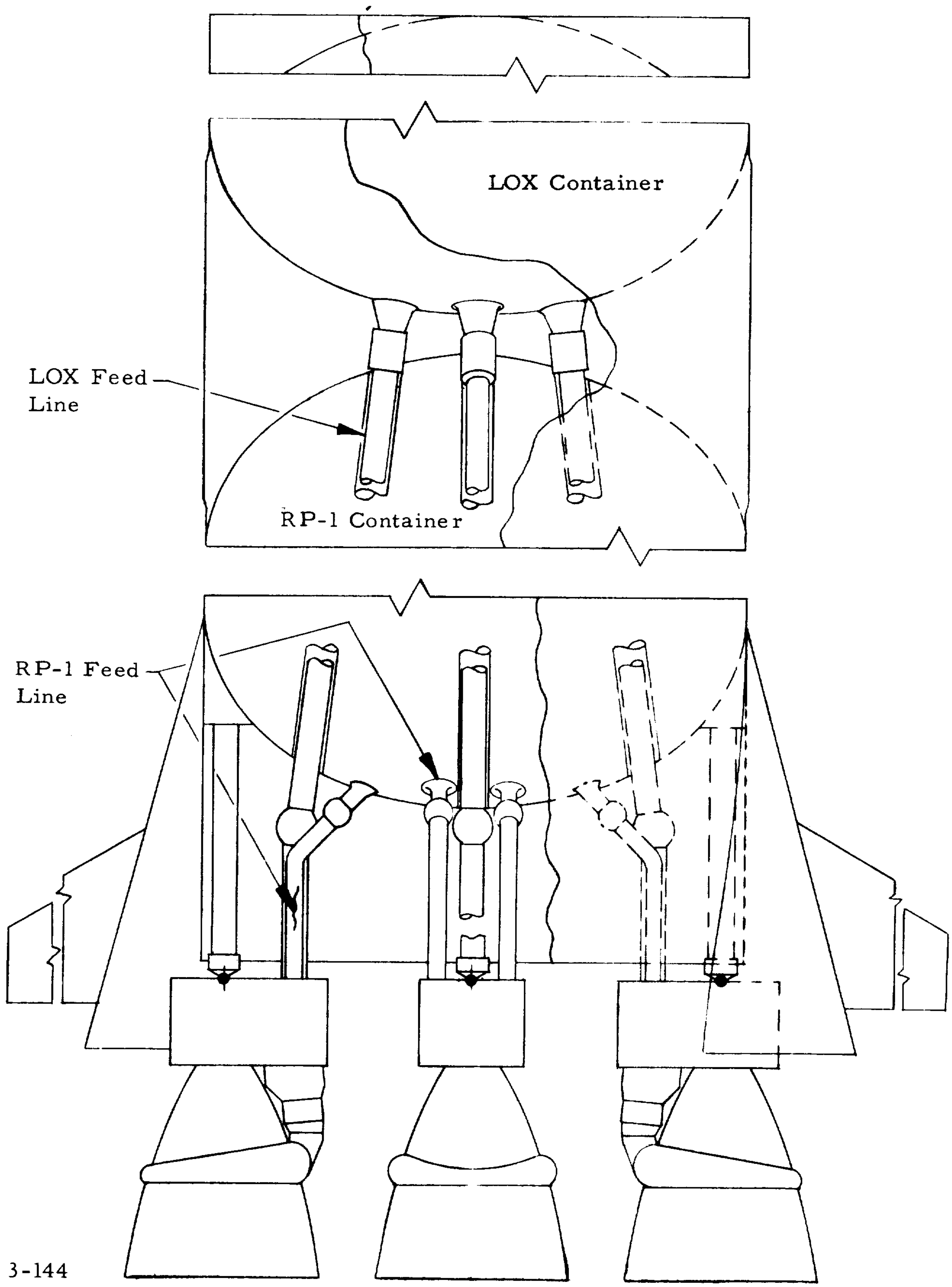


Figure 22-7. Propellant System, S-IC

valve is located in each of the five lines at the point where they emerge from the fuel container tunnels. These valves function as prevalves, serving as controls to stop LOX flow to the engines in the event of engine failure. A single solenoid valve, using 750 psig nitrogen from the control pressure system, operates the three prevalves for each engine (one LOX and two fuel prevalves) simultaneously. The LOX container is equipped with a propellant level sensor which will initiate an inboard engine cutoff signal. A level sensor or timer cuts off the outboard engines about six seconds later. The LOX suction line inlets are equipped with anti-vortexing devices. The turbopump inlets have pressure-volume compensating ducts allowing for engine movement while maintaining a constant fluid volume. This pressure balancing feature prevents excessive pressure loads from being transferred to the turbopump assembly.

22-23. PROPELLANT PRESSURIZATION SYSTEM.

This system provides the pressurization to the propellants so that the net positive suction head (NPSH) requirement at the inlet of the turbopumps is maintained.

22-24. Fuel Container Pressurization System. Heated helium gas is used to pressurize the RP-1 container. The helium is stored in four high pressure bottles located in the LOX container. Helium is piped from the bottles through pressure control valves to a heat exchanger located on each engine. The heated helium leaving the heat exchangers is manifolded and piped to the top of the RP-1 container. The RP-1 container is prepressurized with helium from a ground source approximately 90 seconds before liftoff.

22-25. Oxidizer Container Pressurization System. This system pressurizes the LOX container with gaseous oxygen (GOX) obtained by bleeding a flow of high-pressure LOX from each of the five engines into a heat exchanger located on each engine. The GOX is piped from the five heat exchangers through a flow control valve to the top of the container. The pressurization system is designed to provide sufficient pressure to prevent flash boiling and the flow rates are sized with zero venting as a design objective.

The LOX container is pre-pressurized with helium from a ground source approximately 90 seconds before liftoff. The helium is fed into the container through a liftoff disconnect valve in the GOX piping system. Provision is made for supplementing and/or replacing the flight vehicle pressurant with GN_2 during static firing of either the flight vehicle or the static test vehicle.

A GOX flow control valve modulates the GOX flow between flow rates of 30 to 50 pounds per second in response to a pressure signal from the LOX container. The valve is designed to maintain the LOX container pressure to 20.5 $\pm$2.5 psia.

22-26. PROPELLANT CONDITIONING SYSTEM.

The propellant conditioning system provides fuel bubbling and LOX conditioning. These operations are described below.

22-27. Fuel Bubbling. In order to prevent extreme fuel temperature stratification in the fuel container and suction lines, GN_2 is bubbled through the fuel. Gaseous nitrogen is supplied to a vehicle manifold from a ground source through a coupling and filter. The manifold routes the nitrogen to the fuel suction lines through branch lines containing an orifice and check valves.

22-28. LOX Conditioning. The LOX temperature in the suction lines must be maintained below saturation temperature during countdown to prevent LOX from geysering in the suction lines. Geysering is prevented by thermal pumping, which is accomplished by interconnecting the LOX suction lines below the prevalves. These interconnect lines contain normally-open interconnect valves. Three of the LOX suction lines are insulated in the fuel container tunnel. LOX flows down the insulated lines and returns to the container through the two uninsulated lines. If an emergency requires that the prevalves be closed, helium is supplied from a ground source into the LOX suction lines just above the prevalves. This helium bubbling maintains the required low temperature in the suction line.

22-29. PROPELLANT LOADING SYSTEM.

The propellant loading system includes all ducts, valves and flexible joints required to fill and drain the vehicle propellants during static tests or prior to launch.

22-30. Fuel Loading. Fuel is loaded at 2000 gpm. A 6-inch pneumatic piston-operated ball valve is used to close off the container upon completion of fuel loading. The working fluid (750 psig nitrogen from the control pressure system) is passed to the ball valve by a solenoid-actuated pilot valve. A loss of pneumatic pressure or electrical power closes the ball valve. Fuel loading is controlled by a capacitance-type gage mounted inside the fuel container. When fuel reaches a predetermined level, a signal is generated closing the fuel fill and drain valve. The fill line is attached to the ground systems with a quick disconnect coupling. Any additional adjustment of the fuel level is accomplished using the same fill and drain line. An emergency drain is provided during static firings to permit a rapid drain rate. This consists of a 12-inch gimbal joint and prevalve attached to a special drain nozzle on the fuel container. Upon completion of static testing the line is removed and the container nozzle is capped off.

22-31. LOX Loading. LOX is loaded at 10,000 gpm through two 6-inch fill lines that tie into the inboard engine suction line at a point just above the prevalve. Each line contains a 6-inch pneumatic piston-operated ball valve located at the point where the fill line connects to the suction line. Each valve is controlled by a separate solenoid valve using 750 psig nitrogen from the control pressure system. A loss of pneumatic pressure or electrical power causes the fill and drain valve to fall in the closed position. LOX loading is controlled by a capacitance type gage mounted inside the LOX container. When LOX reaches a predetermined level, a signal is generated to close the LOX fill and drain valves permitting the fill lines to be drained. The fill lines attach to the ground system with quick disconnect couplings. During loading operations, the prevalves open permitting LOX chill-down to the engine main LOX valves. An emergency drain is provided during static firings to permit a more rapid drain rate. This consists of a 17-inch gimbal joint and prevalve attached to a special drain nozzle on the LOX container. This line is removed from the flight vehicle and the container nozzle is capped off upon completion of static tests.

During a hold, the LOX container is continually replenished using the same LOX fill and drain line.

22-32. PROPELLANT UTILIZATION SYSTEM.

An active closed loop propellant utilization system is provided. Five continuous level sensors are installed in each propellant container. These provide data for a continuous propellant profile, permitting sloshing and consumption to be determined as a function of flight time. The total main loading is controlled by propellant loading sensors mounted in the containers.

22-33. S-II STAGE PROPULSION SYSTEM.

After S-IC staging, the S-II stage propulsion system, Figure 22-2, provides the thrust which accelerates the space vehicle to a sufficient velocity whereby the S-IVB stage can subsequently complete the injection of the Apollo Spacecraft into the planned earth parking orbit and after a coast period into the translunar trajectory. Functionally, the propulsion system is composed of a cluster of five Rocketdyne J-2 engines and a propellant system.

22-34. ENGINE.

The engine cluster consists of one engine mounted on the stage longitudinal axis with the four remaining engines mounted outboard 90 degrees apart. The outboard engines are gimballed for pitch, yaw, and roll control.

The J-2 engine, Figure 22- 8, is an advanced, high-performance design utilizing LOX and LH_2 as propellants. The engine envelope is 80 inches in diameter and 116 inches long. Nominal thrust, specific impulse and weight are 200, 000 pounds, 426 seconds and 2190 pounds, respectively.

The J-2 engine, Figures 22- 9 and 22- 10, features a single-tubular wall, bell-shaped thrust chamber, and independently driven, direct drive turbopumps for LOX and LH_2. Each turbopump utilizes the same propellants as the main thrust chamber. A brief description of the engine components follows. Table 22- 4 lists the engine performance parameters and mechanical characteristics. A schematic diagram of the engine is illustrated in Figure 22- 11.

22-35. Thrust Chamber. The thrust-chamber body consists of a cylindrical section, a narrowing throat section and an expansion section. The body is constructed of longitudinal brazed stainless steel tubes. An intake manifold routes fuel through the tubing, cooling the thrust chamber and converting the fuel to a gaseous state before injection into the combustion chamber.

22-36. Fuel Turbopump. The fuel turbopump, Figure 22-12, is an axial flow pump consisting of seven stages in addition to an inducer. It is a direct turbine drive, self-lubricated, high-speed pump driven by the exhaust gases from the gas generator. The

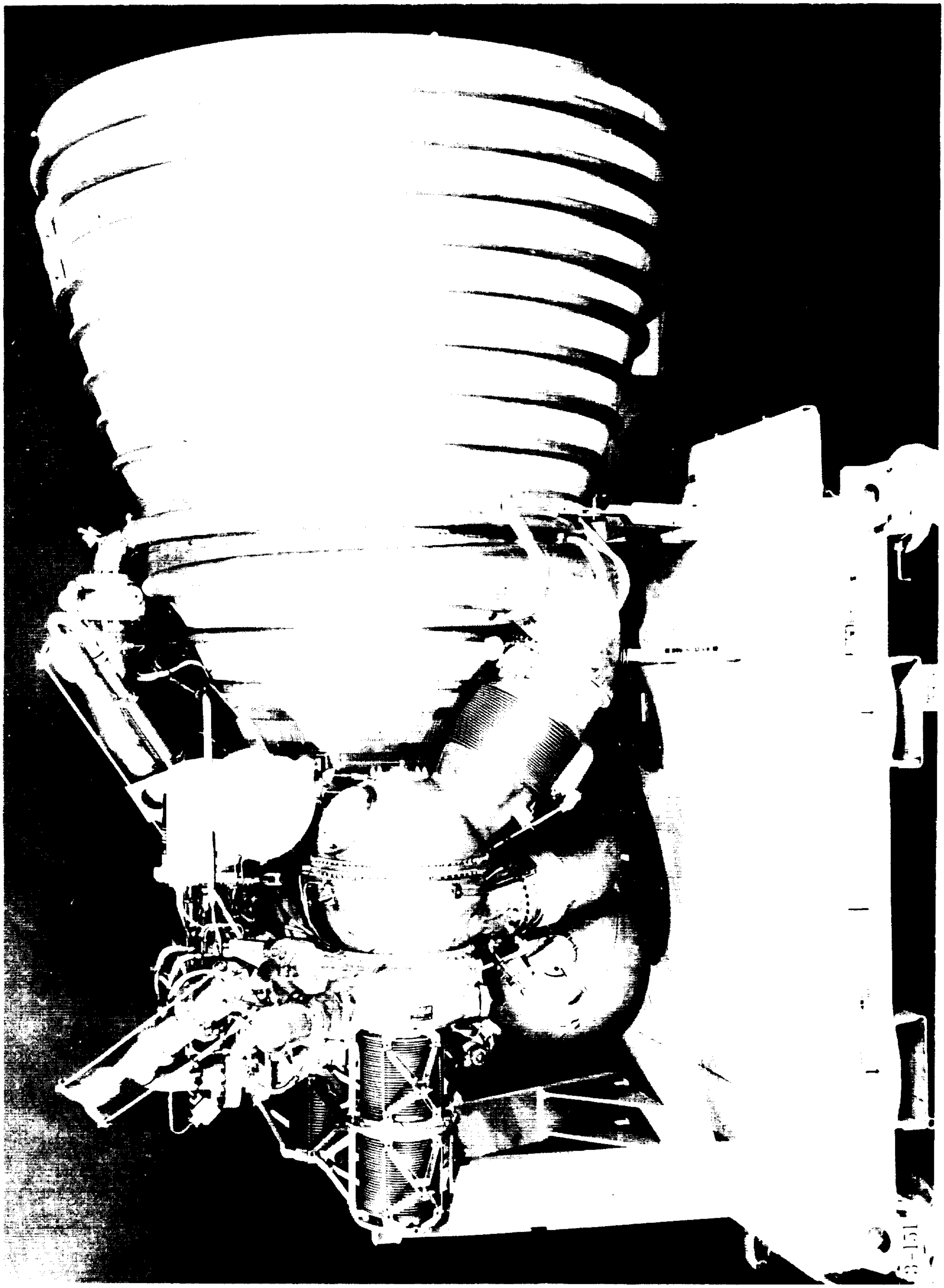


Figure 22-8. J-2 Engine

1 Gimbal

2 Main Oxidizer Valve

3 Gas Generator Control Valve

4 Fuel Turbopump

5 Gas Generator

6 Main Fuel Duct

7 Turbine Bypass Duct

8 Oxidizer Turbine Bypass Valve

9 Main Fuel Valve

10 Thrust Chamber

11 Fuel Manifold

12 Exhaust Manifold

13 Container Pressurization Supply (Oxidizer)

14 Electrical Control Package

15 Helium Regulator

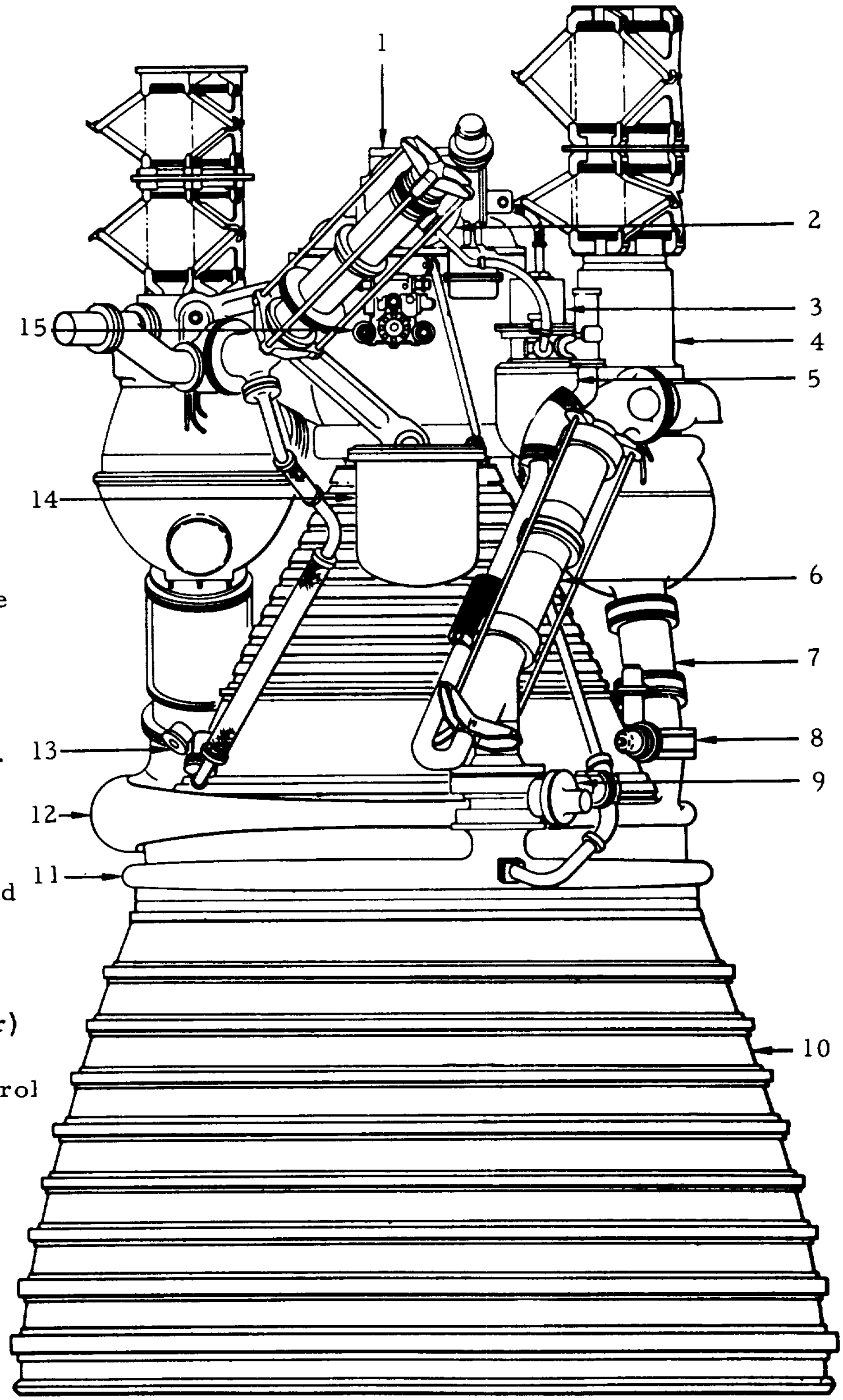


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Figure 22-9. J-2 Engine Component Locations

1 Gimbal

2 Oxidizer Turbopump

3 Turbine Exhaust Duct

4 Heat Exchanger

5 Exhaust Manifold

6 Fuel Manifold

7 Oxidizer Turbine Bypass Valve

8 Turbine Bypass Duct

9 Fuel Turbopump

10 Start Tank

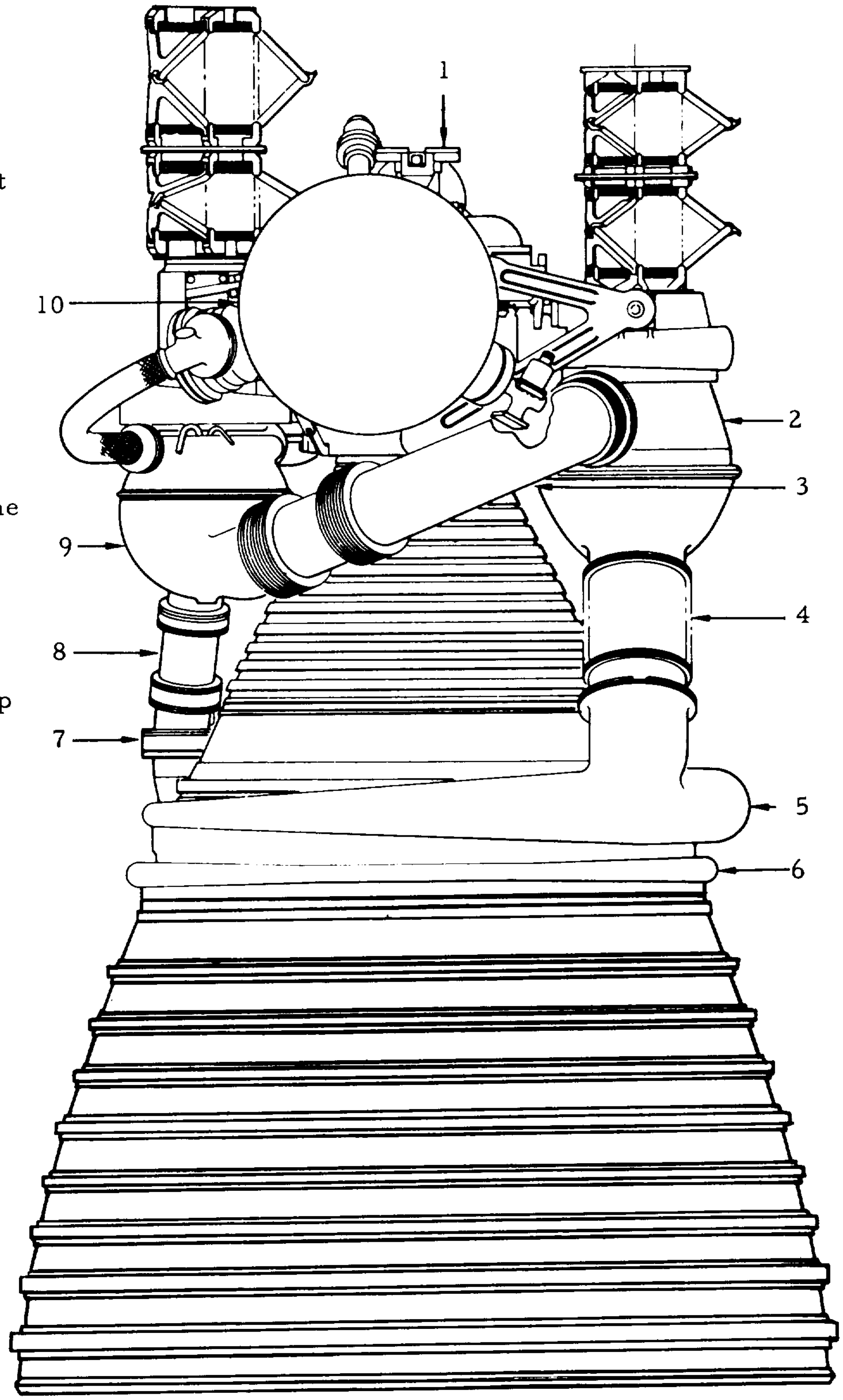

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Figure 22-10. J-2 Engine Component Locations

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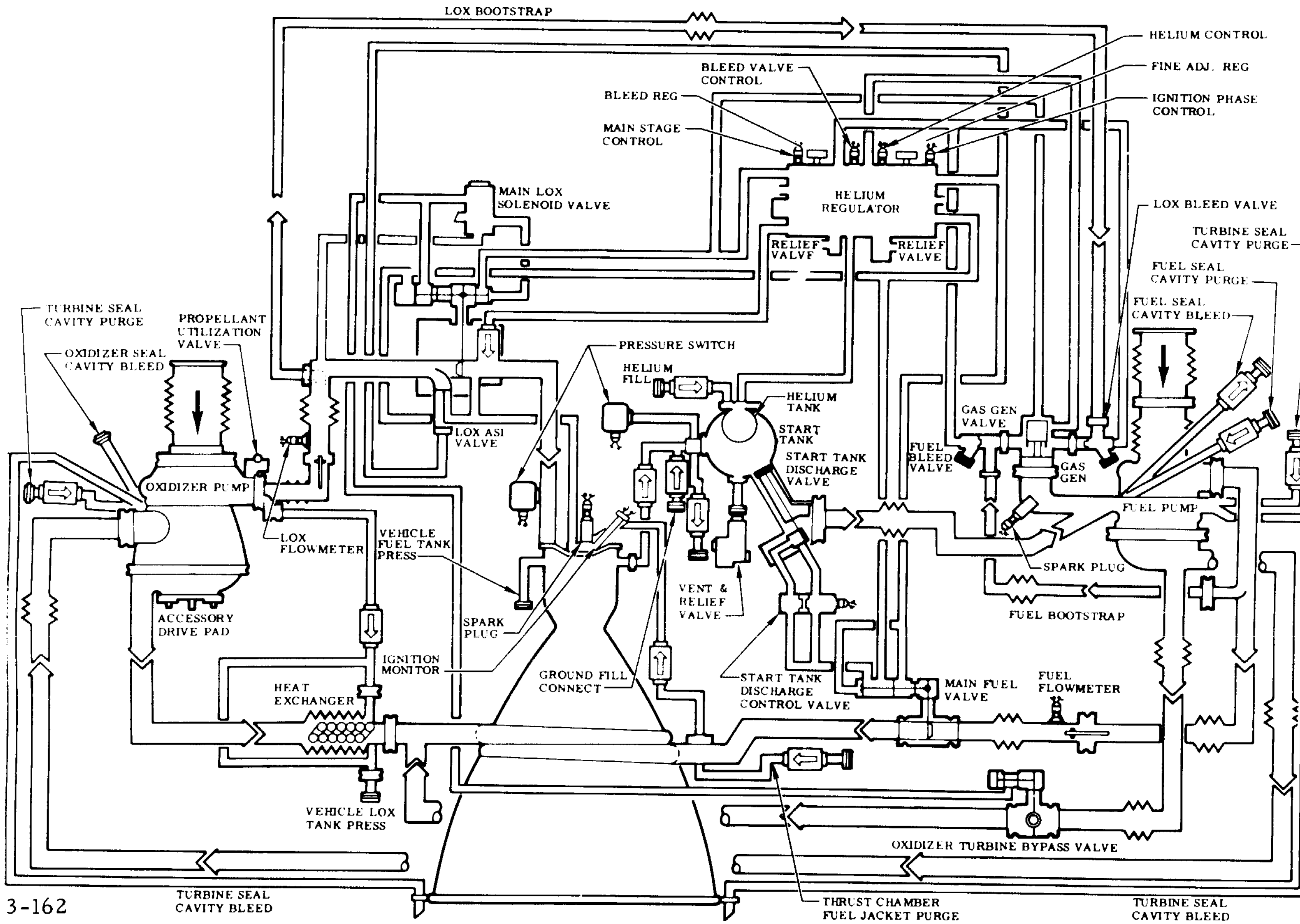

Figure 22-11. J-2 Engine Schematic

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[redacted]

Table 22-4. Performance Parameters and Mechanical Characteristics, J-2 Engine

Item	Characteristic
Oxidizer	Liquid Oxygen
Fuel	Liquid Hydrogen
Thrust (Altitude)	200, 000 pounds
Specific Impulse	426 seconds
Mixture Ratio O/F	5.00
Rated Duration	250 seconds
Oxidizer Flowrate	291.30 pounds per second
Fuel Flowrate	78.26 pounds per second
Chamber Pressure, psia	682.5
Expansion Ratio	27.5:1
Diameter	80 inches
Length	116 inches
Weight, Dry	3028 pounds
Weight, Wet	3188 pounds

turbine shaft turns the inducer, forcing LH_2 through a series of seven stages.

22-37. Oxidizer Turbopump. The oxidizer turbopump, Figure 22-13, is a single stage, centrifugal pump, self-lubricated and self-cooled with direct turbine drive. Exhaust gases from the fuel turbopump drive the turbine.

22-38. Gas Generator. A gas generator supplies the hot gases that drive the turbopump turbines. The gas generator consists of a combustor, an injector, oxidizer and fuel poppets, and two spark igniters. The gas generator supplies sufficient energy to operate the fuel and oxidizer turbopumps. (Together they require 8500 horsepower.)

22-39. Propellant Utilization Valve. An electrically operated, motor driven, propellant utilization valve provides for simultaneous depletion of the propellants. During engine operation, propellant level sensing devices in the propellant containers control the position of the valve. Oxidizer modulation is accomplished by bypassing LOX back into the pump inlet.

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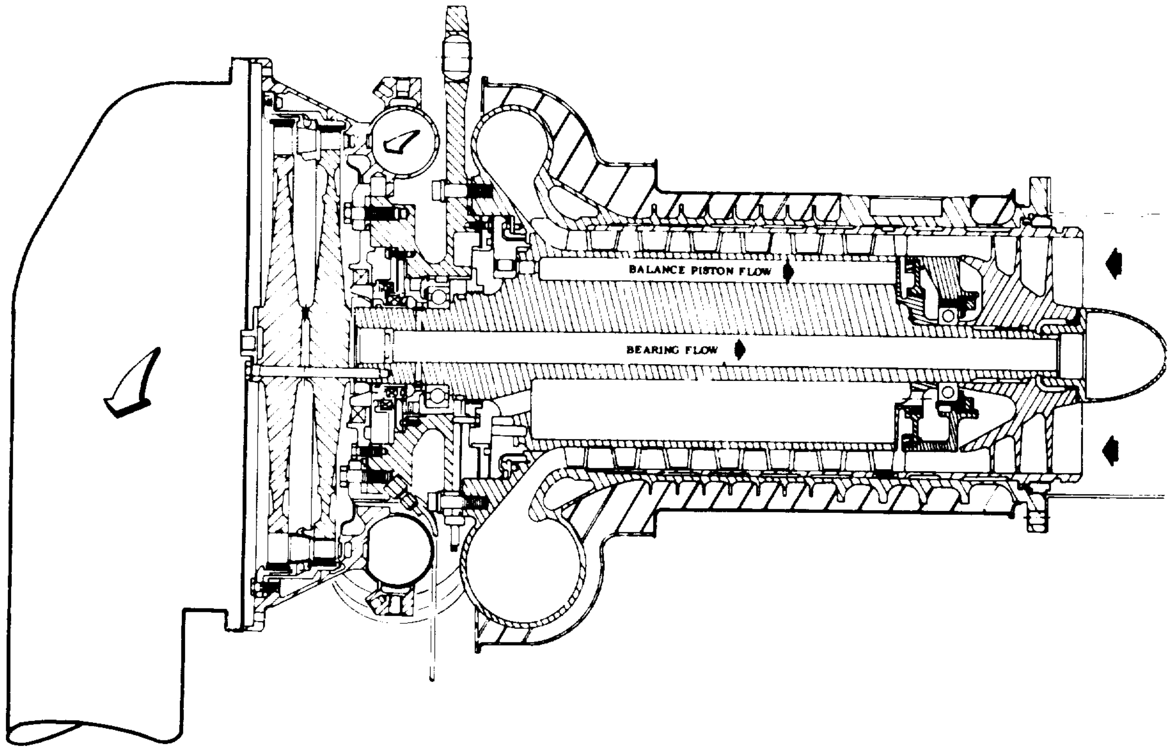

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Figure 22-12. J-2 Fuel Turbopump

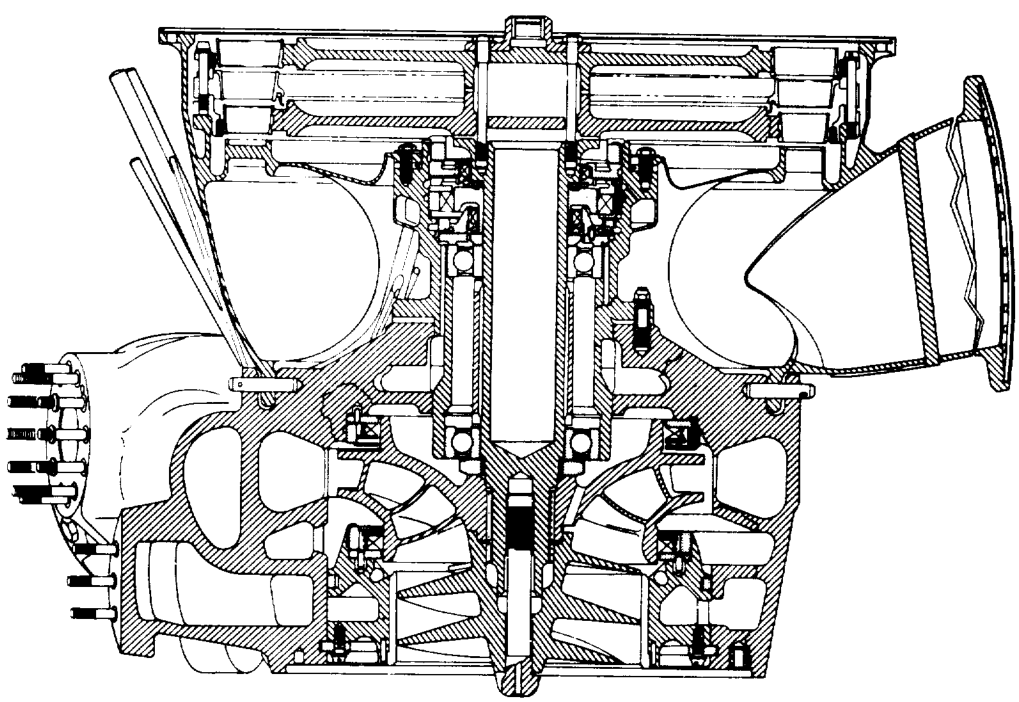

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Figure 22-13. J-2 Oxidizer Turbopump

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22-40. Electrical Control Package. The electrical control package contains spark exciters and a sequence controller which control the engine system. The package receives 28-volt dc signals from the stage sequencer which initiate engine start or cutoff. The sequence controller performs the necessary sequencing and timing functions for proper operation of the engine system. The electrical control package automatically resets for restart capability.

22-41. ENGINE OPERATION.

The five J-2 engines of the S-II stage are started in unison after first plane separation. The typical operation of an engine is described below.

22-42. Engine Start Sequence (Figure 22- 14). When engine start is initiated, after chilldown of the propellant pumps and feed lines, the spark exciters in the sequence controller are energized and provide energy to spark plugs located in the gas generator combustor, and the augmented spark igniter located in the thrust chamber. Simultaneously, the helium control and the ignition phase control solenoids in the pneumatic control package are energized, allowing helium flow through the helium regulator to the control system. Helium is internally routed through a check valve in the regulator to assure holding propellant valves open in the event that the gas supply system fails. Helium flows through a check valve in the main oxidizer valve, and purges the thrust chamber oxidizer dome until oxidizer injection pressure closes the check valve. The ignition phase control solenoid valve in the helium regulator is initially energized allowing control helium to open the oxidizer augmented spark igniter (ASI) valve and main fuel valves. Igniter fuel is tapped off the high-pressure fuel duct downstream of the main fuel valve, and routed to the ASI for ignition with the oxidizer.

A sequence valve located within the main fuel valve assembly is opened when the fuel valve reaches approximately 90-percent open. Simultaneously, along with engine start, the start tank discharge valve delay timer in the sequence controller is energized. This delay permits fuel to be bled overboard through the thrust chamber to prechill the system. The delay time setting depends upon initial hardware temperature. When the start tank discharge valve delay timer expires and if the temperature sensed in the thrust chamber fuel manifold meets requirements, the start tank discharge valve solenoid and the ignition phase timer are energized. When the start tank discharge valve opens, gaseous hydrogen (GH_2), stored under pressure in the start tank, flows through the series turbine drive system accelerating both turbopumps. The relationship of

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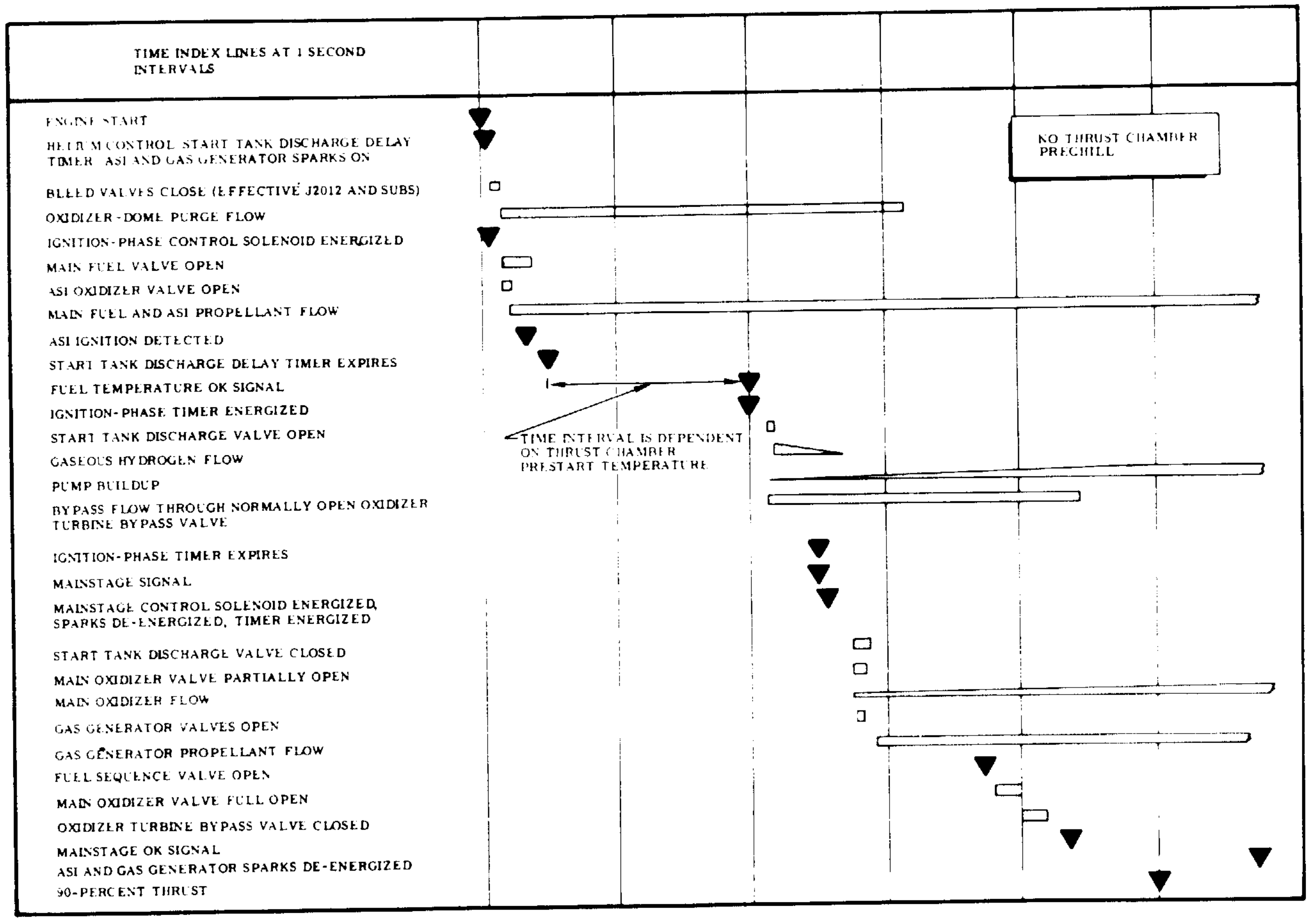

Figure 22-14. J-2 Engine Starting Sequence

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the fuel turbopump to oxidizer turbopump buildup is controlled by the oxidizer turbine bypass valve, which is in the normally-open position, permitting a percentage of the gas to bypass the oxidizer turbine and vent overboard through the thrust chamber. During this period, ASI combustion should have been detected by the ASI ignition monitor. (Absence of ignition would cause cutoff at expiration of the ignition phase timer.) If ignition is detected, the timer expiration energizes the mainstage control solenoid in the pneumatic control package.

Simultaneously, the sparks de-energize timer in the sequence controller is energized and the start tank discharge valve control solenoid is de-energized closing the valve. Opening pressure from the mainstage control solenoid is ported to the main oxidizer valve first-position actuator which opens the main oxidizer valve approximately 25-percent, and to the opening control port of the gas generator control valve.

The propellants flowing into the gas generator are ignited by the spark plugs; the main duct oxidizer flow increases, primes the oxidizer dome, and ignites in the thrust chamber. When the main oxidizer turbopump outlet pressure has reached a predetermined value, oxidizer pressure actuates the main oxidizer valve pressure actuated control valve. Control pressure is directed by the control valve to complete the opening of the main oxidizer valve (second stage) and to close the oxidizer turbine bypass valve.

Transition into mainstage occurs as the turbopumps accelerate to steady state speeds. As oxidizer injection pressure increases toward the steady state level, a mainstage-OK signal is generated by the oxidizer injection pressure switch. (Cutoff results if no signal occurs before expiration of the sparks de-energize timer.) The oxidizer injection pressure also overcomes the oxidizer dome purge pressure, closing the purge check valve. The augment spark igniter and gas generator spark exciters are de-energized by expiration of the sparks de-energize timer.

22-43. Steady-State Operation. Steady-state operation is maintained until a cutoff signal is initiated. During this period GH_2 is tapped off the fuel injection manifold to pressurize the stage fuel container. The stage oxidizer container is pressurized by GOX diverted from the oxidizer high-pressure duct through a heat exchanger located in the oxidizer turbine exhaust duct.

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Propellant utilization control is provided by bypassing oxidizer from the oxidizer turbopump discharge back to the turbopump inlet. The propellant utilization valve is positioned by electrical input from level-sensing devices in each propellant container. The engine mixture ratio may be varied ±0.4 mixture ratio units.

22-44. Cutoff Sequence (Figure 22-15). The cutoff signal is received by the sequence controller which simultaneously de-energizes the mainstage and ignition phase solenoid valves and energizes the helium control solenoid de-energize timer. Opening control pressure to the main fuel valve, main oxidizer valve, oxidizer ASI valve and gas generator control valve, and closing control pressure to the oxidizer turbine bypass valve, propellant bleed valves and start tank-discharge valve, are vented.

Closing control pressure is routed to the oxidizer ASI valve, main fuel valve, and main oxidizer valve and opening control pressure is routed to the oxidizer turbine by pass valve. As chamber pressure decays below oxidizer dome purge pressure, the check valve opens, allowing helium gas to purge the residual oxidizer from the thrust chamber oxidizer dome. The helium control solenoid de-energize timer expires, causing all valve-closing control pressure to vent, and the purge flow to subside.

22-45. PROPELLANT SYSTEM.

The propellant system consists of a 36,883 cubic foot fuel container and a 11,108 cubic foot LOX container described in Paragraphs 21-30 and 21-31, respectively. The main stage propellant capacity is 930,000 pounds.

The feed systems (Figure 22-16) for both propellants are similar in design and operation employing emergency shutoff valves, engine mounted turbopumps and main valves. A propellant utilization valve is provided on each LOX turbopump on each of the five J-2 engines. The ducts for both feed systems, mounted in the lowest possible position on each container, are insulated and have vacuum-jacketed flexible joints to permit engine gimballing.

The propellants pass through a screen in each fuel line and through the engine main turbopumps, the engine main shutoff valves and into the engine. Each engine flow rate is measured by volumetric flow meters. An exclusion riser in the bottom of the LOX container assures a minimum of residual propellant. Anti-vortex and slosh baffles are provided.

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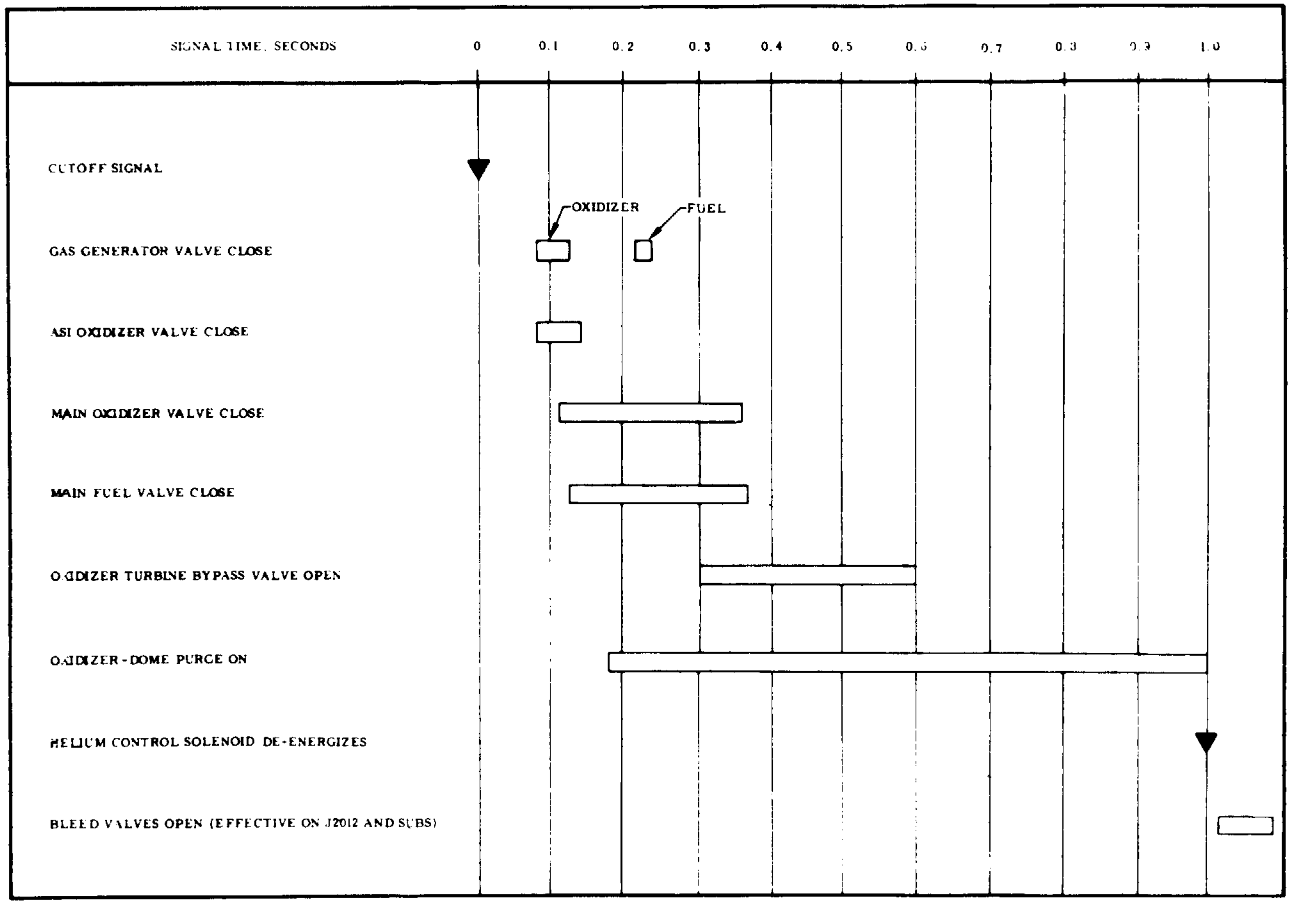

Figure 22-15. J-2 Engine Cutoff Sequence

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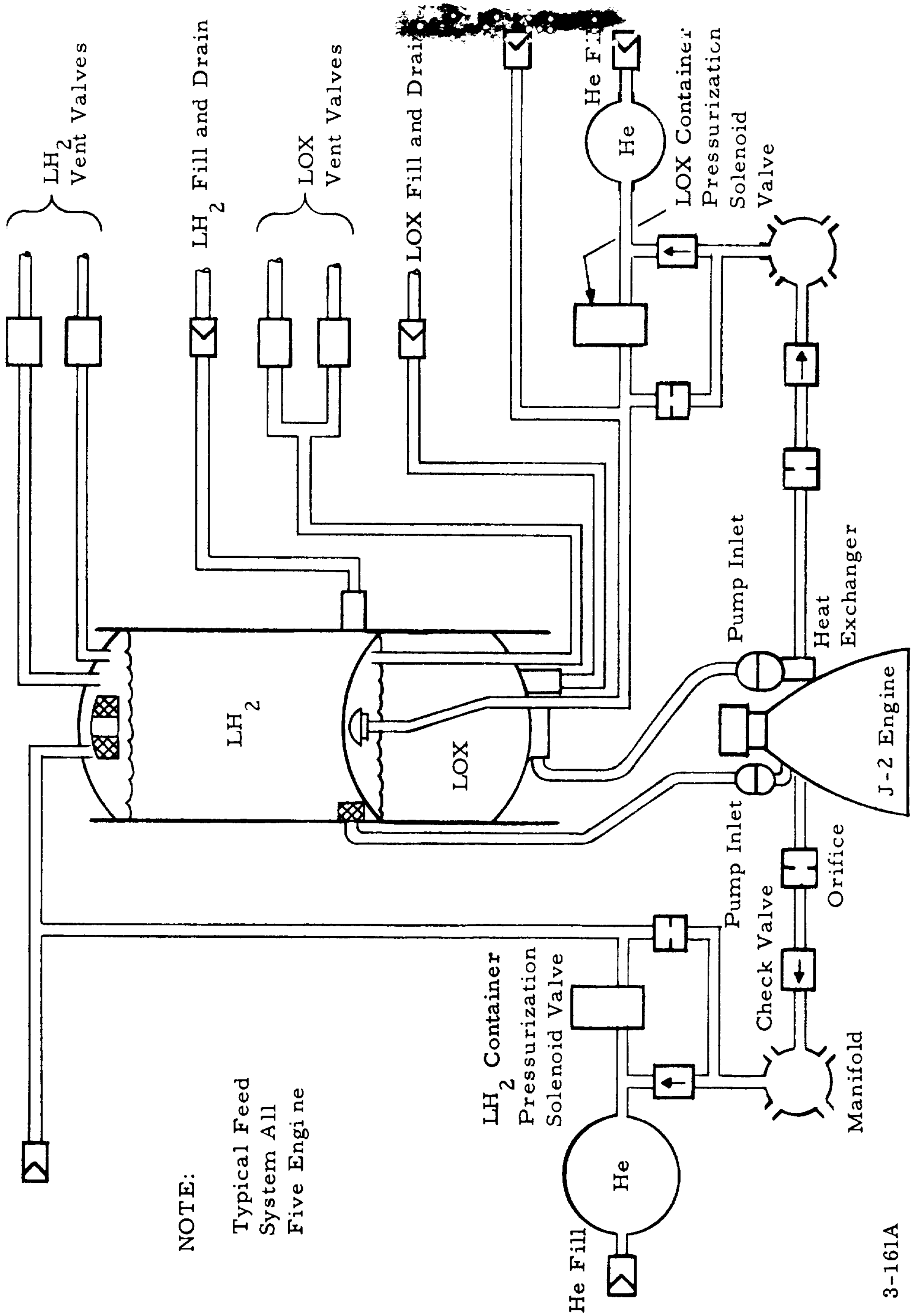

Figure 22-16. Propellant Feed System S-II

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22-46. PROPELLANT PRESSURIZATION SYSTEM.

Pressurization of the propellant containers is required to provide a net positive suction head for the J-2 turbopumps. Initial pressurization with gaseous helium prior to launch is obtained from a ground source. During the S-IC boost phase and J-2 engine start, separate high-pressure stage-stored helium supplies maintain pressurization. During the S-II boost phase, the LOX container is pressurized by gaseous oxygen obtained by passing LOX through the heat exchanger located in the LOX turbine pump exhaust system of each engine. The LH_2 container is pressurized by bleeding off gaseous hydrogen from each engine at a point between the thrust chamber cooling system and the injector.

A 1.5 cubic foot 3000 psi helium sphere mounted on the thrust cone provides the prestart flight pressurization for the LOX container and two 6.0 cubic foot 3000 psi helium spheres mounted on the forward skirt provide the prestart flight pressurization for the LH_2 container.

22-47. PROPELLANT MANAGEMENT SYSTEM.

Operation of the propellant management system is governed by the amount of propellant mass in each container. Control, monitoring and checkout is provided for:

a. Propellant loading
b. Propellant quantity indication
c. Propellant utilization
d. Propellant depletion cut off signal

The propellant loading and quantity indication systems control and monitor the propellant flow rates, and maintain the proper mass ratio remaining in the containers. Propellant quantity is measured and telemetered for check out and monitoring purposes. The propellant utilization system provides closed-loop control of the engine mixture ratio for minimizing residuals at propellant depletion. The propellant depletion engine cut off system provides a signal to indicate when the level of either propellant reaches the depletion point.

Full length and vernier capacitance sensing probes are used separately and in various combinations. These provide the data necessary for the propellant management system.

22-48. CONTROL PRESSURE SYSTEM.

A stage mounted control pressure system provides regulated operating pressure for the electro-pneumatic valves. Each engine is equipped with a self contained control pressure system.

22-49. RECIRCULATION CHILLDOWN SYSTEM.

The engine propellant pumps and gas generators must be chilled prior to start. This is accomplished during S-IC boost phase. LH_2 is circulated, Figure 22-17, by means of stage mounted pumps through the engine LH_2 feed lines, engine LH_2 pumps, and gas generator LH_2 bleed valves and then returned to the container.

LOX is circulated, Figure 22-18, by means of thermal convection through the engine LOX feed lines, engine LOX pumps and gas generator LOX bleed valves and returned to the LOX container.

22-50. S-IVB STAGE PROPULSION SYSTEMS.

The S-IVB stage is provided with both a main propulsion system and an auxiliary propulsion system. After separation from the S-II, the thrust of the main propulsion system completes the injection of the space vehicle into the earth parking orbit, and later after a coast period injects the space vehicle into a lunar transfer trajectory. The auxiliary propulsion system provides thrust for roll control during powered flight, ullage thrust during S-II/S-IVB separation, orbit coast, and engine start and attitude control during the coast periods, Figure 22-19.

22-51. MAIN PROPULSION SYSTEM.

The main propulsion system is composed of a single Rocketdyne J-2 engine and associated propellant system.

22-52. ENGINE.

The J-2 engine, also used on the S-II stage, is described in detail in Paragraph 22-35. Engine restart capability is obtained by refilling the start tank with gaseous hydrogen from the engine cycle after initially starting the engine. A minimum of seven seconds of mainstage is required to recharge the start tank. Ignition for the starts is provided by an electrical spark system located within the gas generator and thrust chamber.

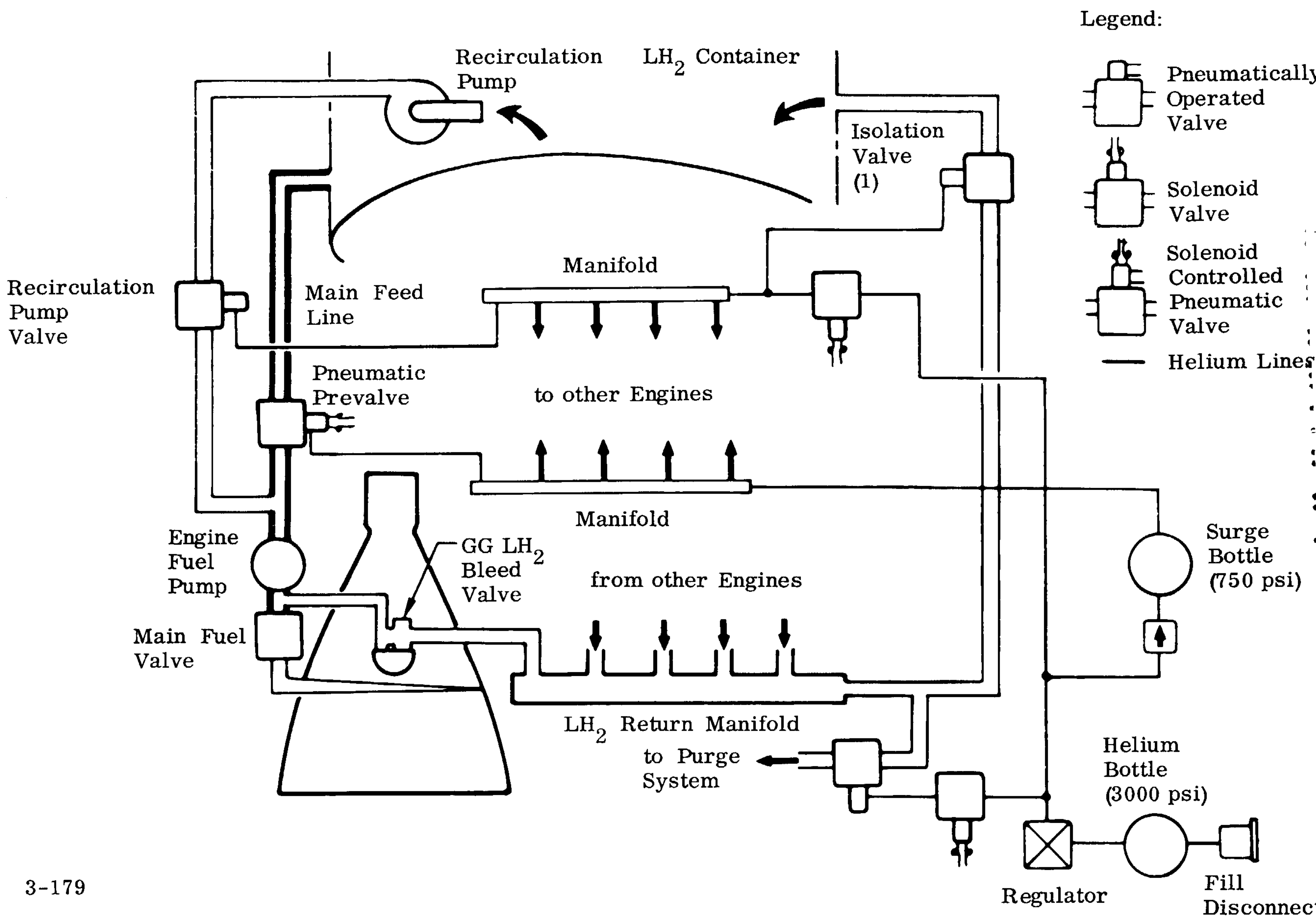

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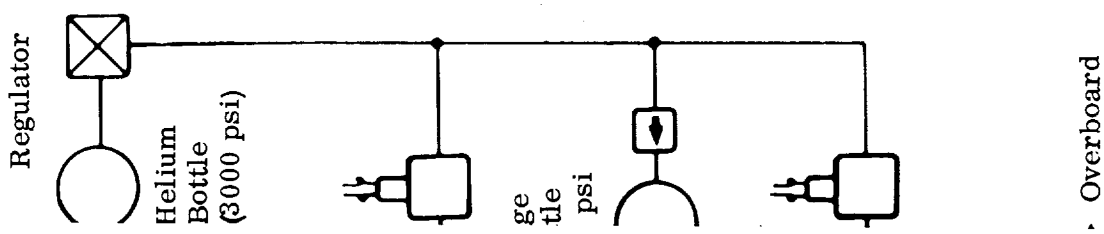
Figure 22-17. LH_2 Recirculation Chilldown System, S-II



22-53. MAIN PROPELLANT SYSTEM (FIGURE 22-2C).

The main propellant system consists of the propellant containers, fuel and oxidizer feed systems, and recirculation chilldown system.

22-54. Propellant Containers. Propellants for the J-2 engine are supplied from LH_2 and LOX containers which form an integral part of the S-IVB stage structure. These have a total volume of 13,250 cubic feet with a resultant main stage propellant capacity of 230,000 pounds from 90 percent of full thrust to cutoff signal.

22-55. Fuel Feed System. To induce fuel feed, the fuel container is initially pressurized from a ground source of cold helium. After main engine ignition, the container pressure is maintained with GH_2 bled from the engine during the first and second burns. Pre-pressurization for second burn is provided by 3000 psia helium bottles.

The single vacuum jacketed fuel line to the J-2 engine is connected to a fuel container outlet located forward of the common bulkhead joint. To ensure sufficient freedom for misalignments due to tolerance buildup and structural deflections, the fuel line includes a flexible bellows. The fuel line is designed to withstand surge pressures experienced during test and in-flight.

22-56. LOX Feed System. To induce LOX feed, the LOX container is pressurized by stage stored helium (3000 psia bottles located in the LH_2 container) heated by the heat exchanger in the LOX turbine exhaust duct.

The single LOX feed line for the J-2 engine is vacuum jacketed. It includes a flexible bellows to ensure sufficient freedom for misalignments due to tolerance buildup and structural deflections.

22-57. Recirculation Chilldown System. Propellants from each container are circulated prior to engine start by means of stage mounted pumps through the engine feed lines, engine pumps and gas generator bleed valves and are then returned to their respective containers. This system is similar to the S-II LH_2 recirculation system.

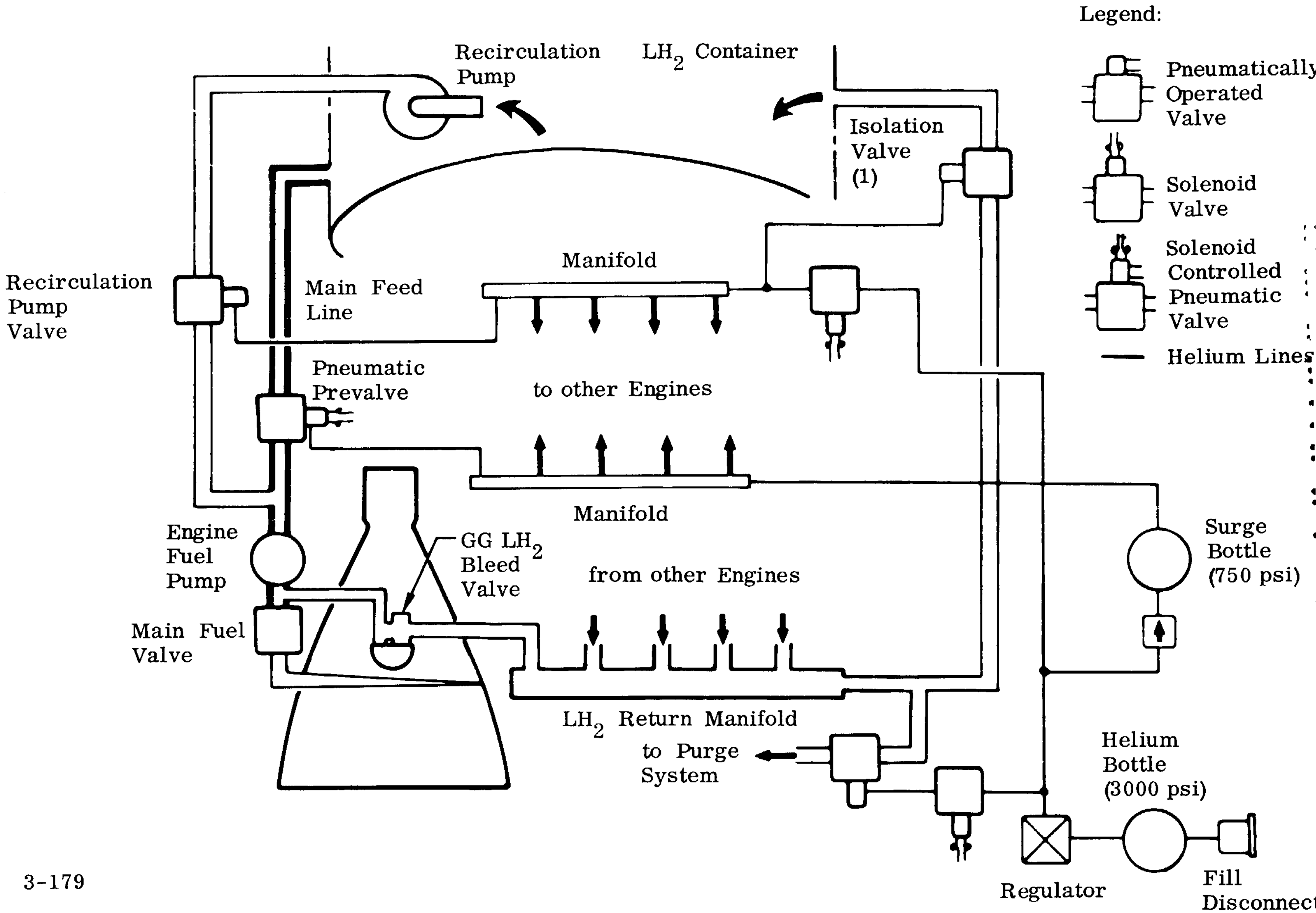


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Figure 22-17. LH_2 Recirculation Chilldown System, S-II

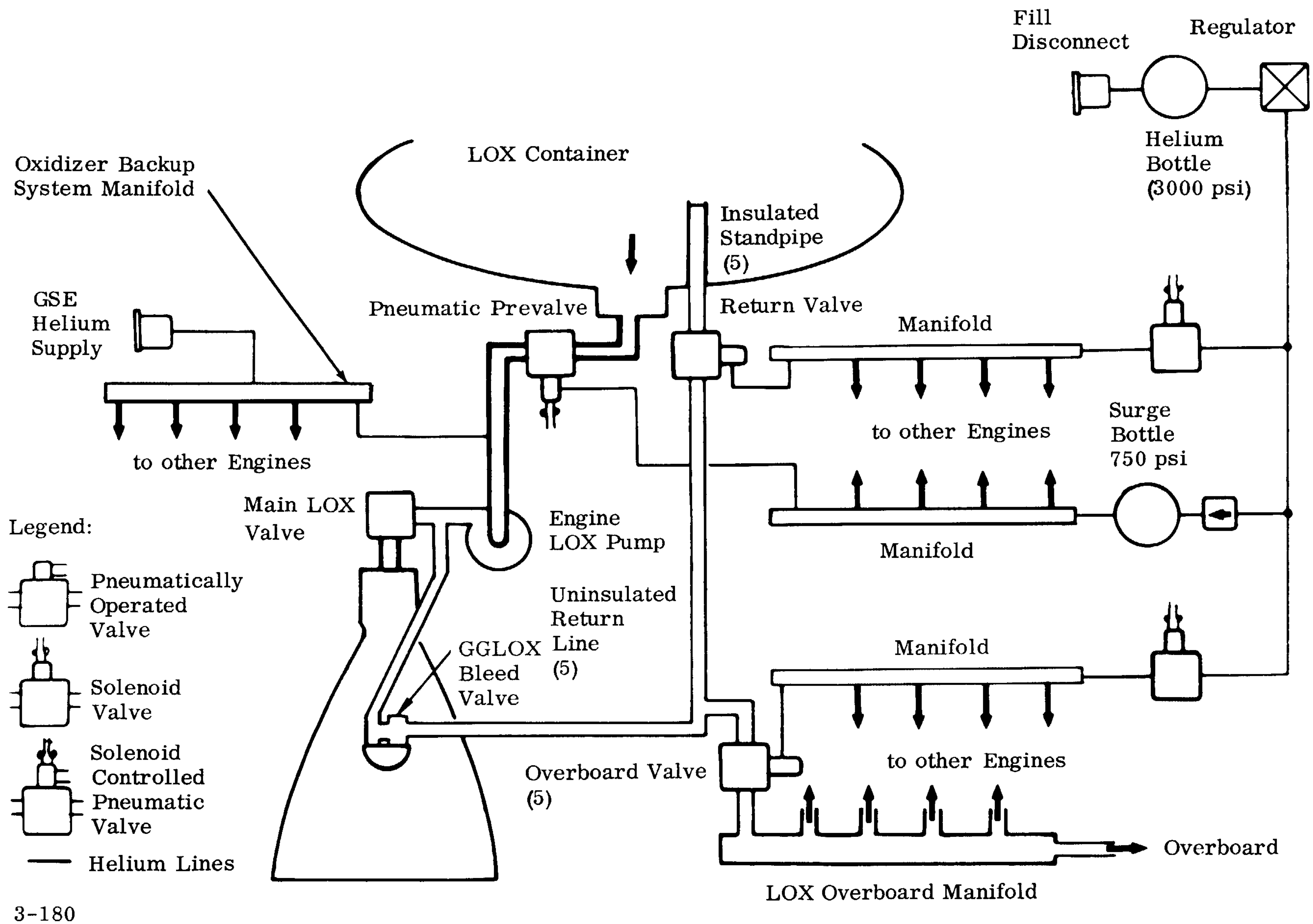

Figure 22-18. LOX Recirculation Childown System, S-II

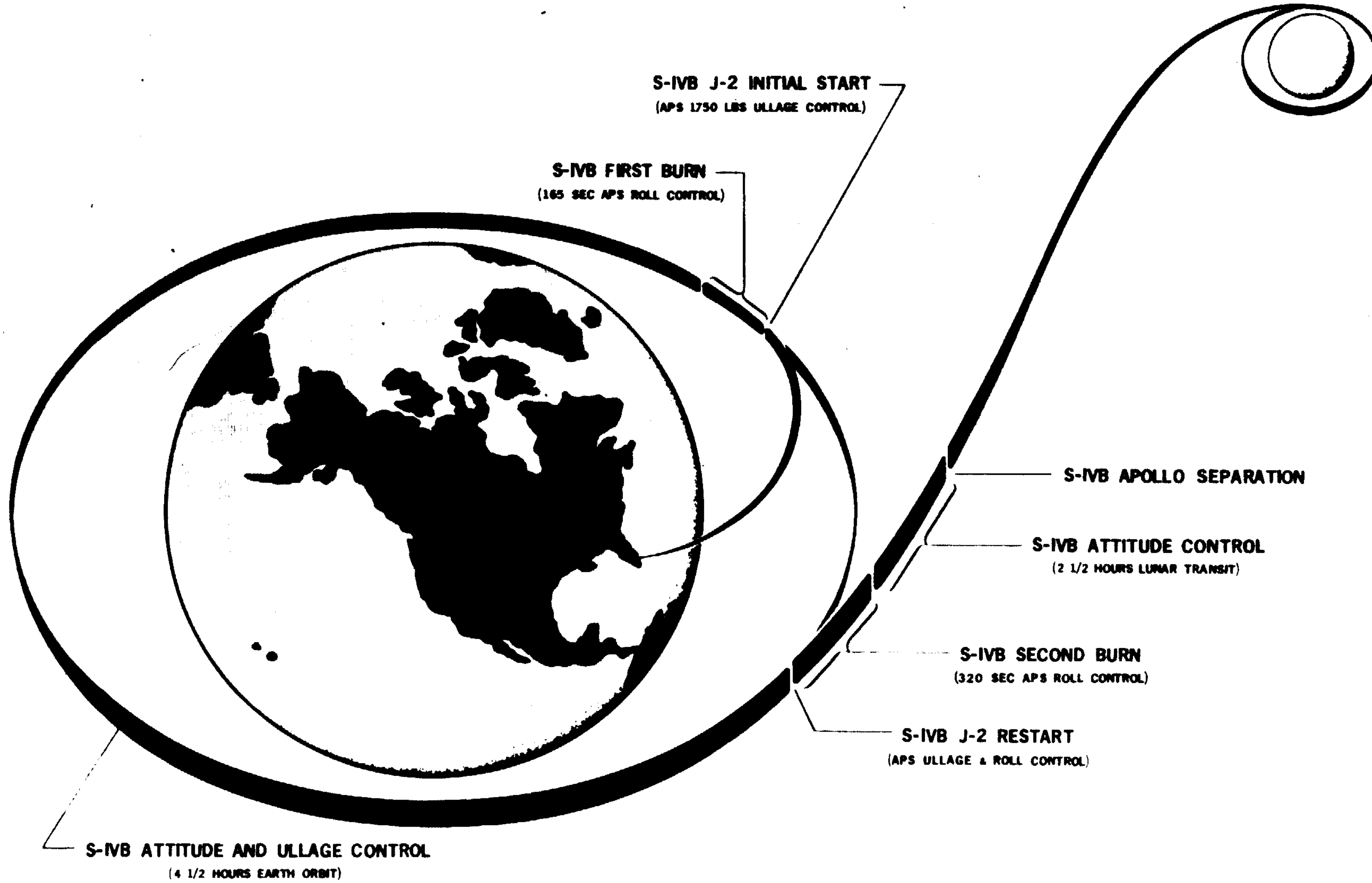

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Figure 22-19. Auxiliary Propulsion System Operation

22-53. MAIN PROPELLANT SYSTEM (FIGURE 22-2C).

The main propellant system consists of the propellant containers, fuel and oxidizer feed systems, and recirculation chilldown system.

22-54. Propellant Containers. Propellants for the J-2 engine are supplied from LH_2 and LOX containers which form an integral part of the S-IVB stage structure. These have a total volume of 13,250 cubic feet with a resultant main stage propellant capacity of 230,000 pounds from 90 percent of full thrust to cutoff signal.

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The single LOX feed line for the J-2 engine is vacuum jacketed. It includes a flexible bellows to ensure sufficient freedom for misalignments due to tolerance buildup and structural deflections.

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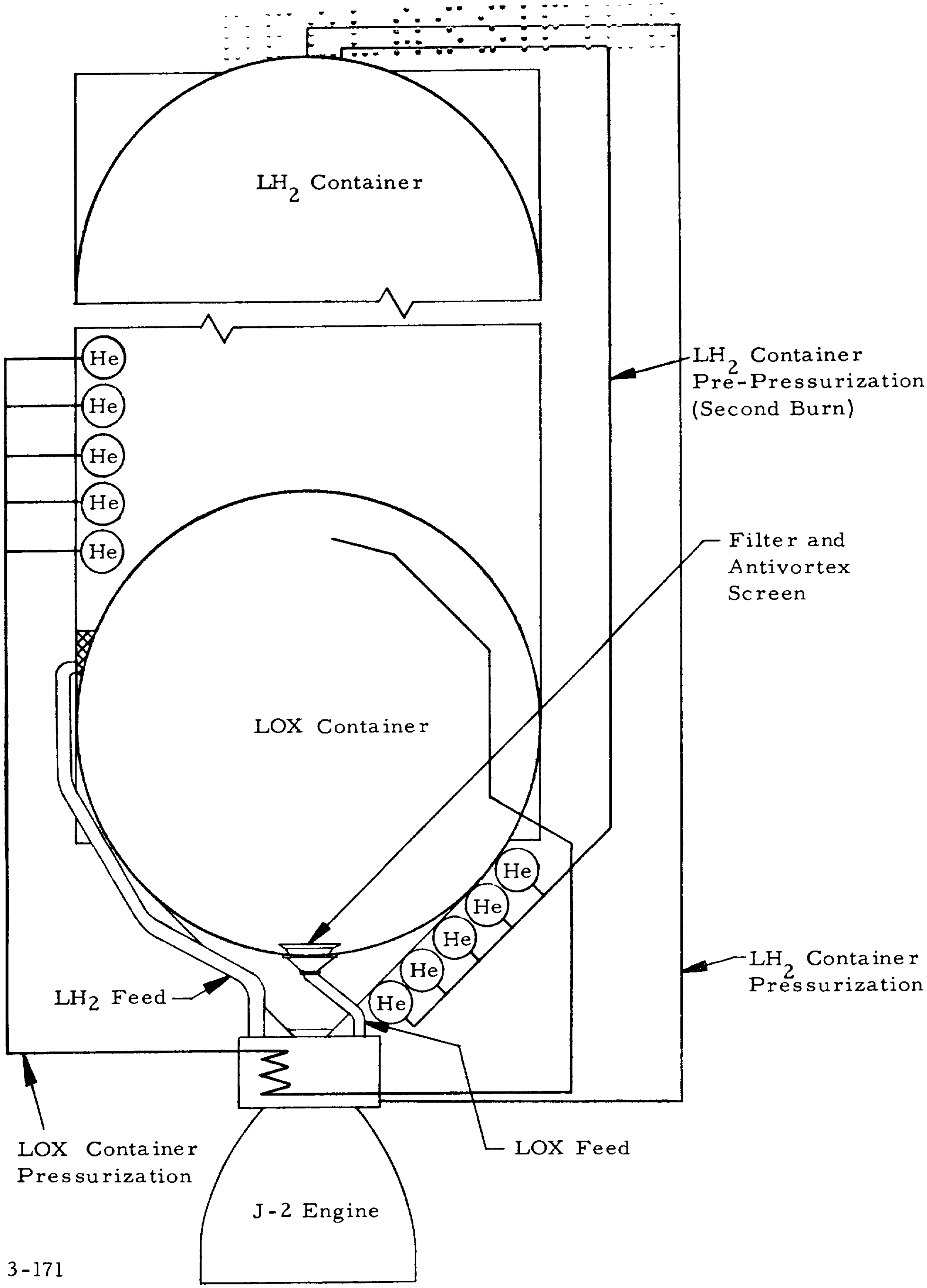

Figure 22-20. Main Propellant System, S-IV B

22-58. AUXILIARY PROPULSION SYSTEM

The auxiliary propulsion system is comprised of two 1630-pound modules mounted 180 degrees apart on the stage aft skirt. Each module contains one main ullage engine, one GH_2 venting ullage engine, three attitude control engines, and a propellant container and feed system (Figures 22-21 and 22-22).

22-59. AUXILIARY ENGINES.

Two types of hypergolic fueled engines are utilized in the auxiliary propulsion system, ullage and attitude control.

22-60. Ullage Engines. Ullage acceleration for S-II/S-IVB separation and the J-2 engine restart is supplied by a 1750-pound thrust Marquardt engine in each module. The engine nozzle assembly consists of a thrust chamber and related fuel and oxidizer pilot valves.

22-61. Attitude Control Engines. Four TAPCO (Thompson Aeronautics Products Company) 150-pound thrust engines provide thrust for attitude control, roll control, and, during orbit coast, ullage for hydrogen venting. The engine nozzle assembly consists of a thrust chamber and two sets of redundant solenoid-operated poppet valves, one for fuel and one for oxidizer.

22-62. AUXILIARY PROPELLANT SYSTEM.

Each module stores 820 pounds of hypergolic propellants in two identical positive expulsion containers, one for the fuel and the other for oxidizer. The propellant system is comprised of these containers, a pressure system and control valves. Since the auxiliary propulsion system must function without the aid of ullage acceleration, the necessary positive expulsion of the propellants is provided by utilizing a dual container system with a collapsible container inside a pressurized container. The collapsible container, storing the propellant, is a thin wall stainless steel bellows. The propellants are delivered to the appropriate engines upon command of the guidance and control system, located in the instrument unit, or upon command of the hydrogen venting system.

Sufficient propellant is provided for:

a. S-II/S-IVB separation thrust

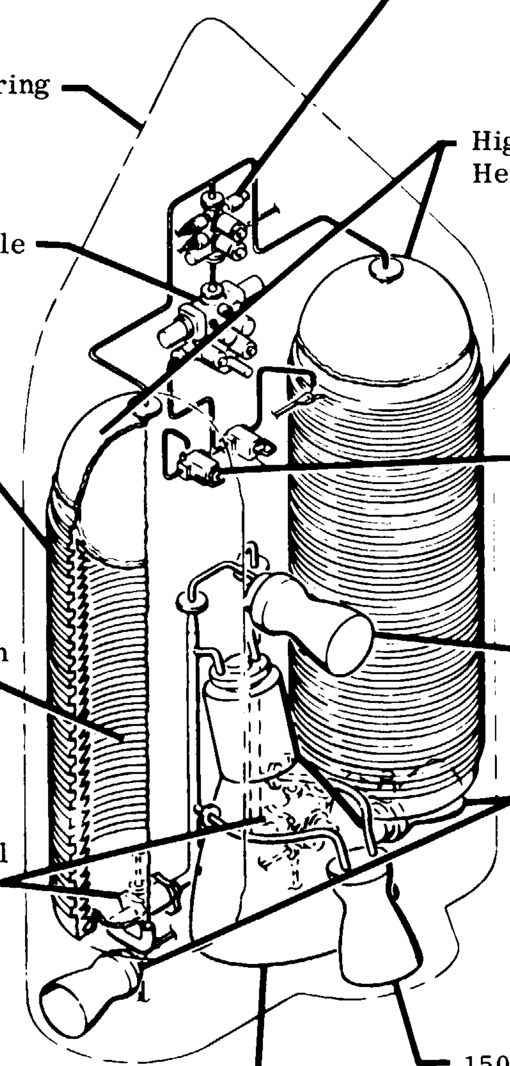

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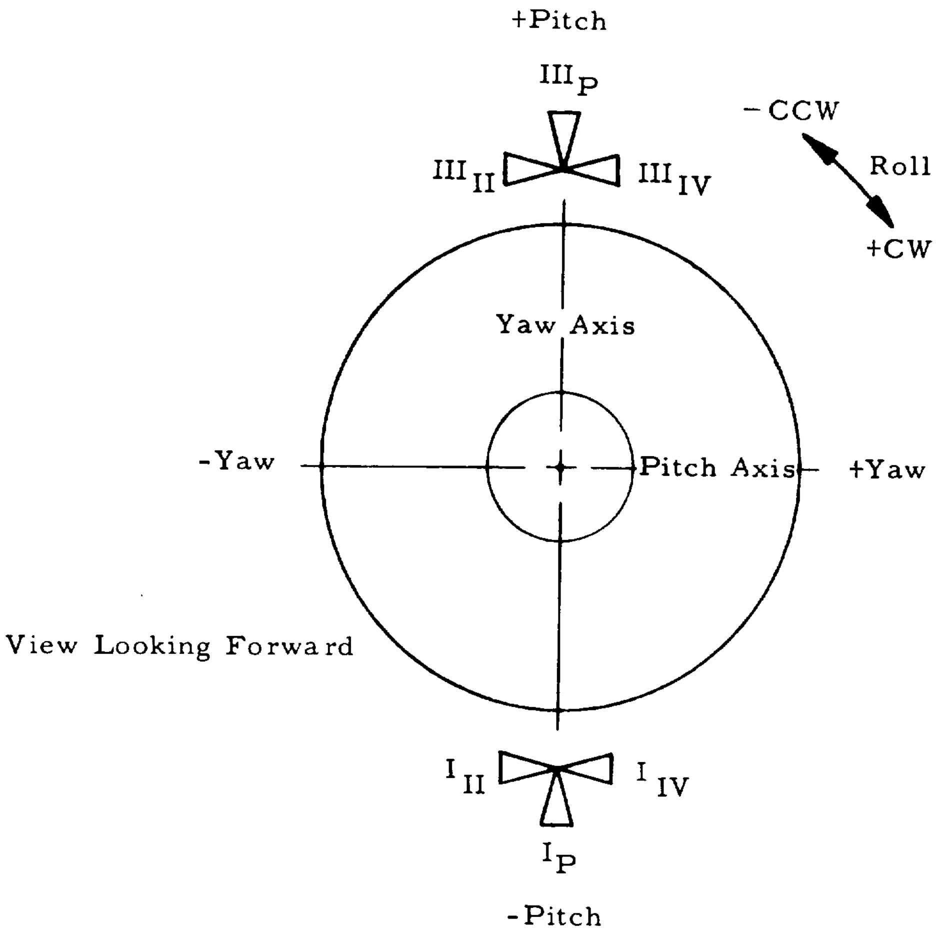
Figure 22-21. Auxiliary Propulsion Module

b. Ullage thrust for J-2 engine starting

The auxiliary propulsion system provides thrust for:

a. Roll control during stage mission
b. Ullage during S-II/S-IVB separation and engine start.
c. Hydrogen venting ullage
d. Attitude control during the coast periods. (Refer to Figure 22-19.)



3-174 Figure 22-22. Attitude Control Engine Locations

CHAPTER 4

SECTION XXIII

MECHANICAL SYSTEMS

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SECTION XXIII

MECHANICAL SYSTEMS

23-1. GENERAL.

The mechanical systems of the Saturn V launch vehicle include environmental control, engine gimballing, separation, ordnance, and platform gas-bearing supply system.

23-2. ENVIRONMENTAL CONTROL SYSTEM.

The Saturn V environmental control system controls the environment in certain compartments of the launch vehicle and Apollo payload. The system protects electrical and mechanical equipment from thermal extremes, controls humidity and provides an inert atmosphere for the vehicle compartments. Operation of the system is controlled by ground based equipment.

The environmental control system allows the use of "off the shelf" electrical components on board the vehicle which otherwise could not be used without elaborate provision for heat dissipation. The system is supplemented by a thermoconditioning unit for the cooling of instrumentation located in the instrument unit and the S-IVB forward compartments.

Environmental conditioning begins during the prelaunch phase upon the application of power to the launch vehicle and ends when the vehicle umbilicals are disconnected at lift off. The thermoconditioning unit continues to provide thermal protection to instrumentation mounted in the instrument unit and the S-IVB forward stage during the ascent, the earth orbital and the translunar phases of the mission. Thermoconditioning ends when the S-IVB/instrument unit is separated from the Apollo payload.

23-3. OPERATION.

The following vehicle and payload areas are conditioned by filtered and thermally controlled dry air or GN_2 supplied by ground equipment:

a. S-IC stage engine compartment

b. S-IC stage forward instrument containers

c. S-IC/SII interstage

d. S-II stage aft instrument containers.

e. S-II stage forward instrument containers

f. S-II/S-IVB interstage

g. Instrument unit including S-IVB stage forward compartment

The ground facilities also supply a thermally conditioned fluid to the thermoconditioning unit in the instrument unit throughout the prelaunch and launch phases of the mission.

At the start of the launch vehicle electrical equipment checkout during prelaunch, the environmental control system supplies cool air to all compartments containing electrical equipment. The cool air maintains electrical components in these compartments within design temperature limits. When loading of the hypergolic fuel for the auxiliary propulsion system (APS) of the S-IVB stage begins conditioned air is supplied to the S-II/S-IVB interstage. The temperature controlled air circulates through the APS modules maintaining the temperature critical fuel in a liquid state.

Prior to loading LOX in the S-IVB stage, warm air is delivered to the S-II/S-IVB interstage. Warm air is next delivered to the S-IC/S-II interstage and then to the S-IC engine compartment prior to loading LOX in the S-II stage and S-IC stage respectively. The warm air flow continues until 30 minutes before the start of LH_2 loading in the S-IVB stage.

The environmental control system medium is changed from air to GN_2 for all compartments and instrument containers a minimum of 30 minutes before the start of LH_2 loading in the S-IVB stage. This prevents possible fire or explosion by maintaining the O_2 content below the level which will support combustion and by preventing any significant accumulations of GH_2. The flow rates and temperature remain unchanged. (Figure 23-1.)

The Apollo payload is also conditioned by the environmental control system. The media, flow rate, temperature, and delivery schedules are determined by MSC.

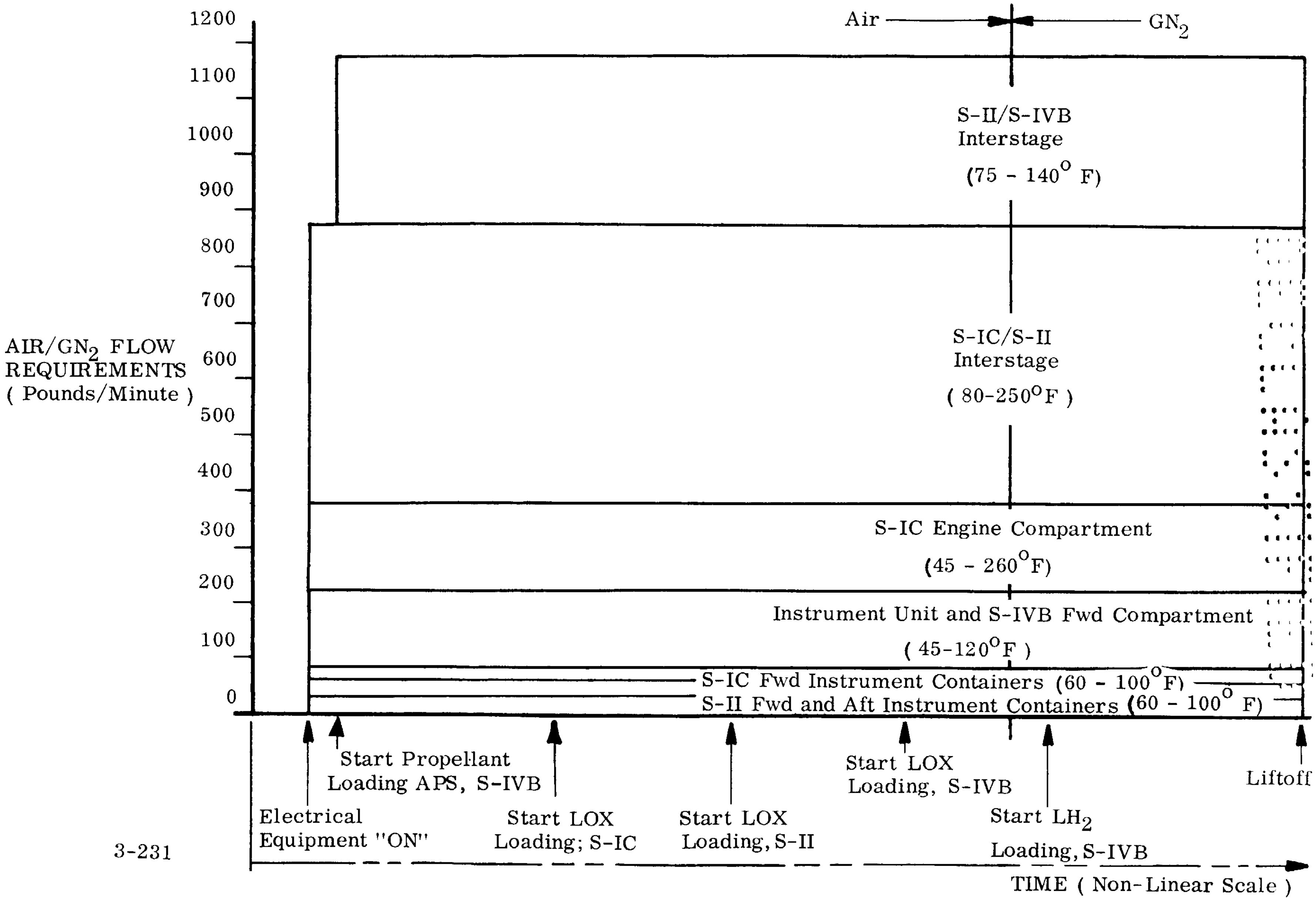

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Figure 23-1. Environmental Control System, Air/GN$_2$ Requirements

The vehicle thermoconditioning unit provides additional thermal conditioning for instrumentation mounted in the instrument unit and in the S-IVB stage forward compartment. Operation of the thermoconditioning unit begins at the start of the launch vehicle electrical checkout during prelaunch and continues until separation of the Apollo payload.

23-4. S-IC STAGE IMPLEMENTATION.

The environmental control system for the S-IC stage maintains the necessary temperature and humidity levels for the protection of instruments, electrical components and ordnance devices in the stage during the prelaunch and launch phase of the mission. The aft compartment, Figure 23-2, which comprises the area between the fuel container and firewall, receives conditioned air or GN_2 (45 to 260 degrees F and 0 to 43 grains per pound of dry air specific humidity), at a rate of 150 pounds per minute through two, 7-inch diameter umbilicals. The temperature within the compartment is maintained at approximately 80 degrees F $\pm$ 10 degrees F.

Instrument containers located in the forward compartment above the LOX container receive 38 pounds per minute of conditioned air or GN_2 (70 to 90 degrees F, 0 to 43 grains per pound of dry air, specific humidity) through one, 4-inch diameter umbilicals. Prelaunch and launch environmental control in the forward compartment outside of the instrument containers is accomplished by means of the S-II stage environmental control system. Environmental control during the ascent phase of the mission is provided by passive means. Thermal inertia and component insulation maintain temperatures within the design ranges.

23-5. S-II Stage IMPLEMENTATION.

The environmental control system for the S-II stage provides temperature and humidity control for the engine gimbal hydraulic systems, electrical components and ordnance devices located in the S-IC/S-II interstage and for instrument containers located in the aft and forward compartments. The system is operational during the prelaunch and launch phases of the mission. Conditioned air or GN_2 (80- to 250- degrees F) is supplied to the S-IC/S-II interstage at a flow rate of 500 pounds per minute through one, 8-inch by 17-inch umbilical, Figure 23-3.

Air or GN_2 at a temperature of 60 to 100 degrees F and a flow rate of 25 pounds per minute is supplied through one, 2-1/4 inch by 8-3/16 inch umbilical to the instrument

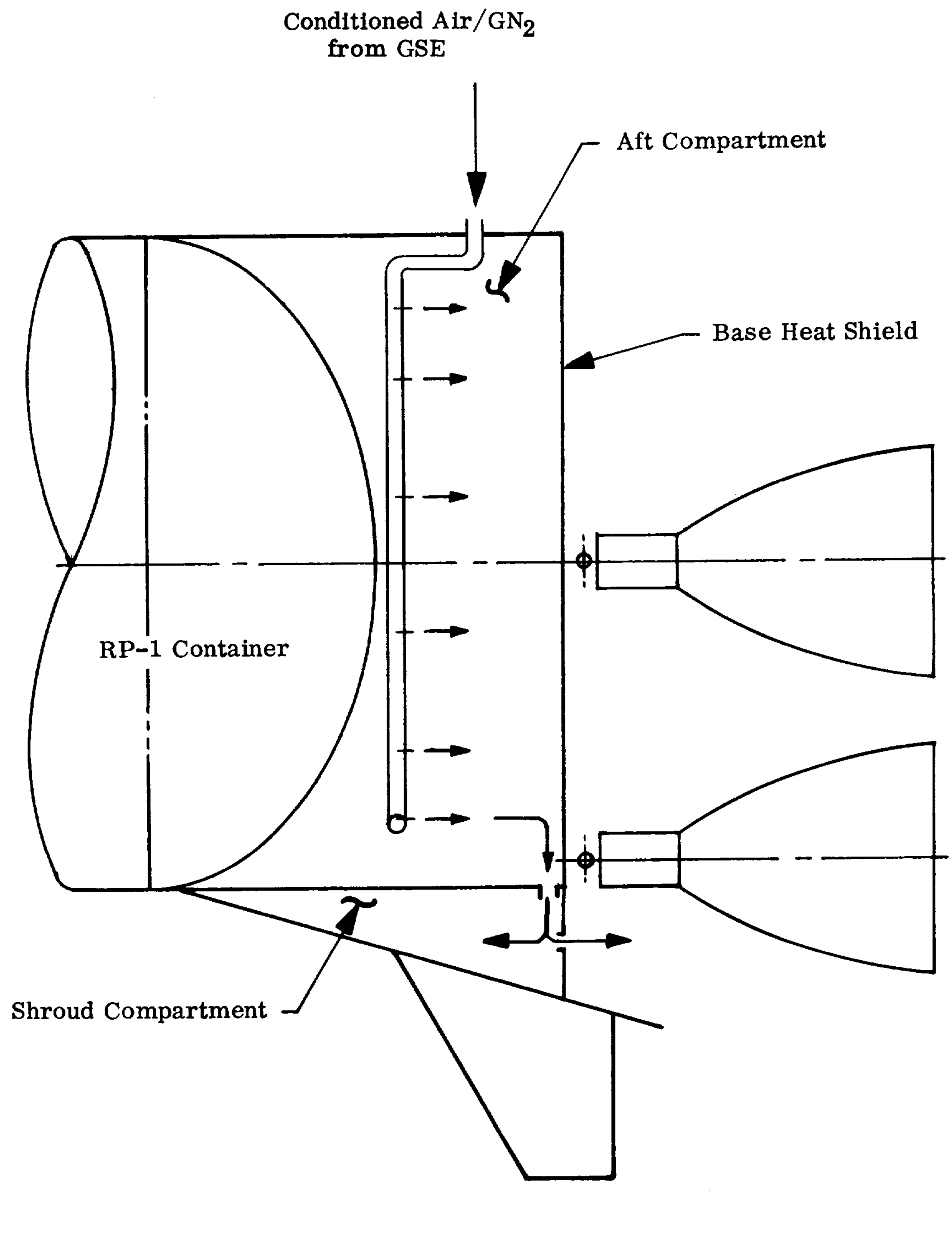

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Figure 23-2. Aft Compartment Environmental Control, S-IC

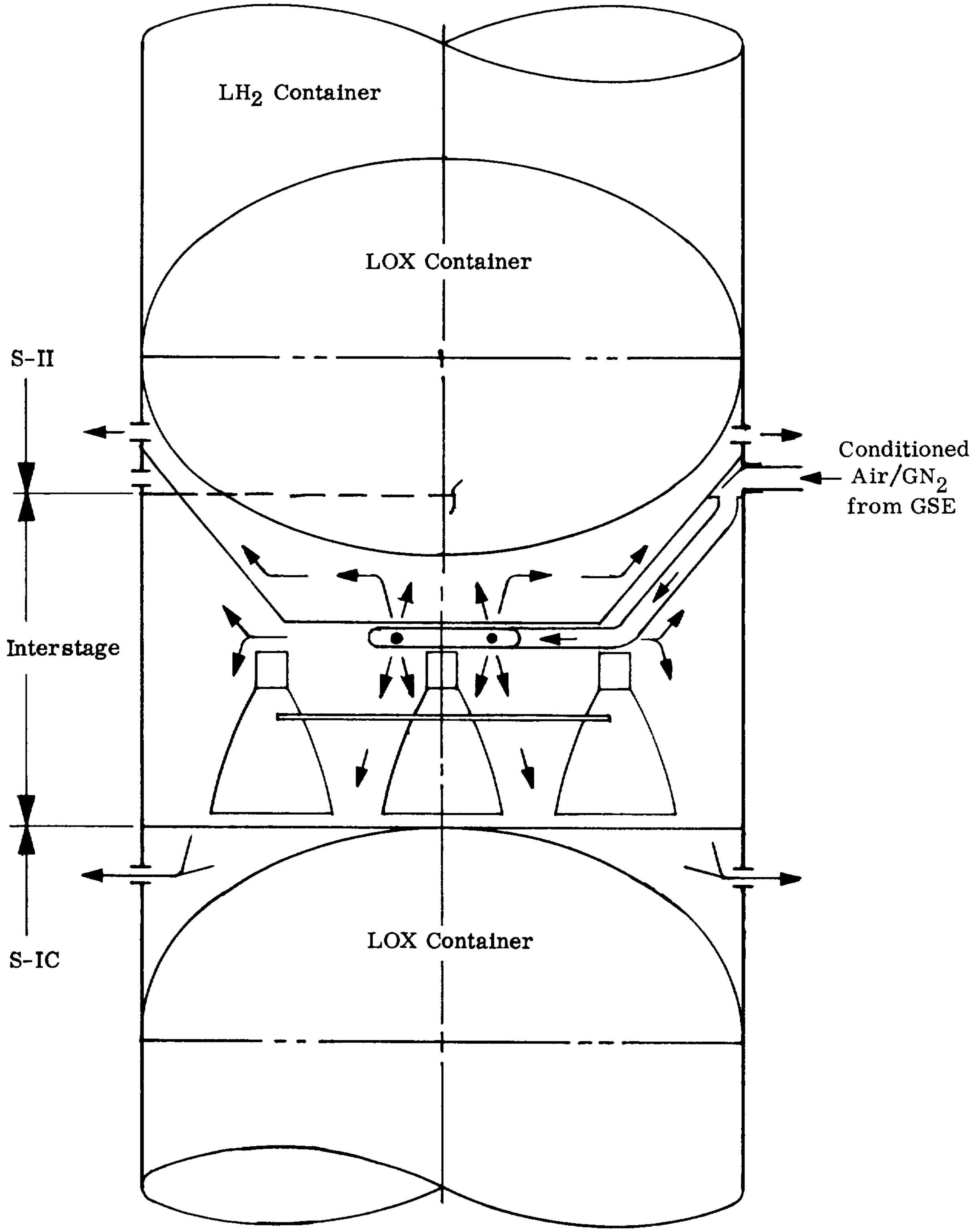

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Figure 23-3. Interstage Compartment Environmental Control, S-IC/S-II containers in the aft compartment and through one 4-inch-diameter umbilical to the instrument containers in the forward compartment.

The specific humidity in the system is maintained at 0 to 43 grains per pound of dry air. Pre-flight environmental control in the forward compartment outside of the instrument containers is accomplished by means of the S-IVB stage environmental control system. Environmental control during the ascent phase of the mission is provided by passive means.

23-6. S-IVB STAGE IMPLEMENTATION.

The environmental control system for the S-IVB stage provides temperature and humidity control for the engine gimbal-hydraulic system, electrical components and ordnance devices located in the aft compartment and for the APS modules during prelaunch and launch operations. The temperature of the aft compartment, which consists of the area beneath the LOX container and includes the area inside the S-II/S-IVB interstage, is controlled at 70$\pm$ 10 degrees F. Figure 23-4. Conditioned air or GN_2 (75 to 140 degrees F, 0 to 43 grains per pound of dry air specific humidity) is supplied at a flow rate of 300 pounds per minute through one 8-inch by 11-inch umbilical connection. The forward compartment of the S-IVB stage is conditioned by the instrument unit environmental control system. Environmental control of the S-IVB stage during ascent, earth orbital, and translunar trajectory phases of the mission is accomplished by passive means except for critical electrical equipment located in the forward compartment. Temperature sensitive electrical equipment in the forward compartment is mounted on cold plates which provide thermoconditioning from prelaunch until separation of the Apollo payload. The thermoconditioning system is described in paragraph 23-7.

23-7. INSTRUMENT UNIT IMPLEMENTATION.

The environmental control system for the instrument unit is accomplished by conditioned air or GN_2 (45 to 120 degrees F, 0 to 43 grains per pound of dry air, specific humidity) through one, 6-inch diameter umbilical at a flow rate of 150 pounds per minute. The temperature within the instrument unit is maintained at 40 to 70 degrees F. The instrument unit system provides conditioning for the S-IVB forward compartment.

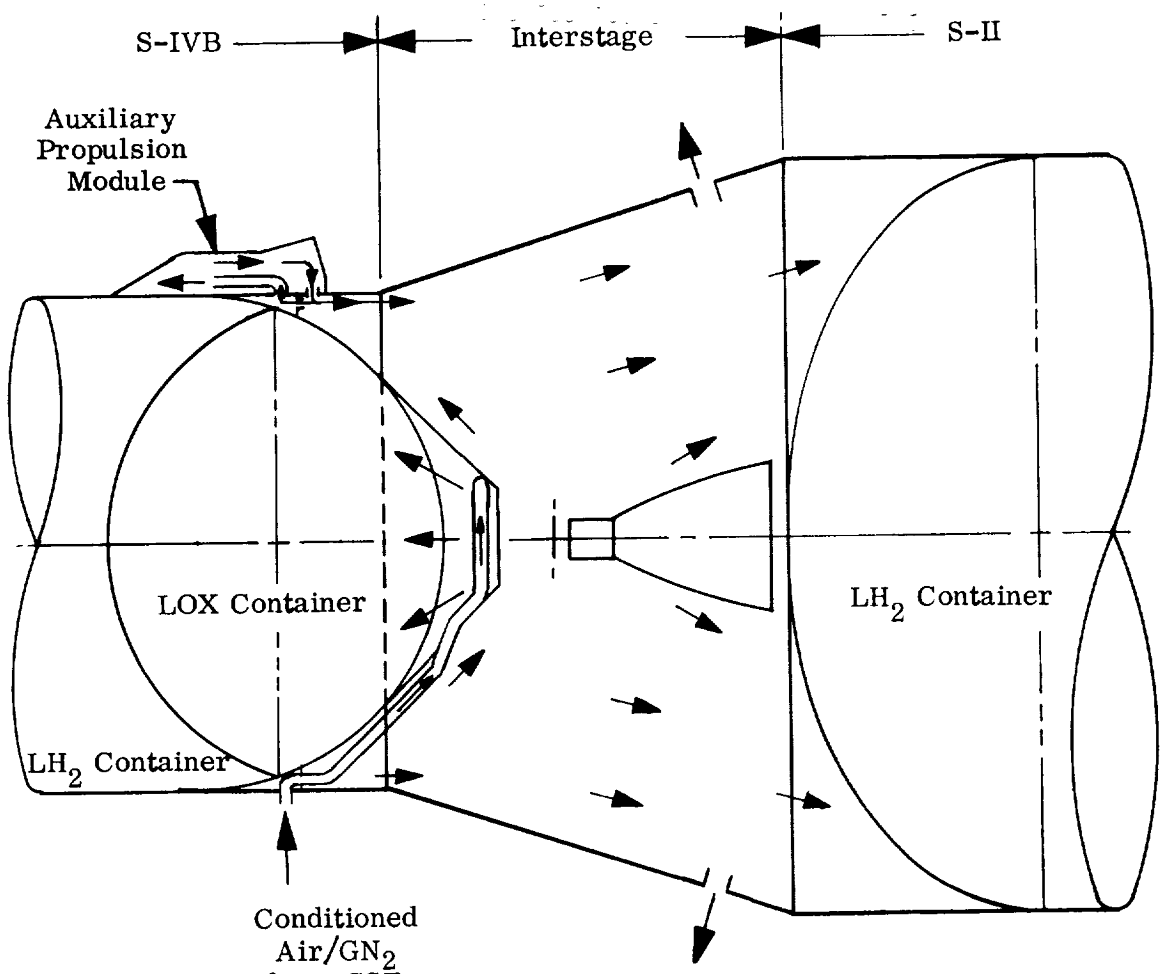

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Figure 23-4. Interstage Compartment Environmental Control, S-II/S-IVB

A thermoconditioning unit provides additional temperature control for instrumentation and the ST-124-M stabilized platform in the instrument unit, and for temperature sensitive equipment located in the forward compartment of the S-IVB stage. The unit is operational from prelaunch until Apollo payload separation. Thermoconditioning is accomplished by pumping a coolant solution (60-percent methanol/40 percent water, by weight) with corrosion inhibitors, through the thermoconditioning system to a heat sink that utilizes water as an expendable evaporant. The coolant circulates in a closed loop that includes the instrument unit cold plates, the ST-124-M inertial platform, and the thermoconditioning plates on which temperature sensitive components in the S-IVB stage, are mounted, Figure 23-5.

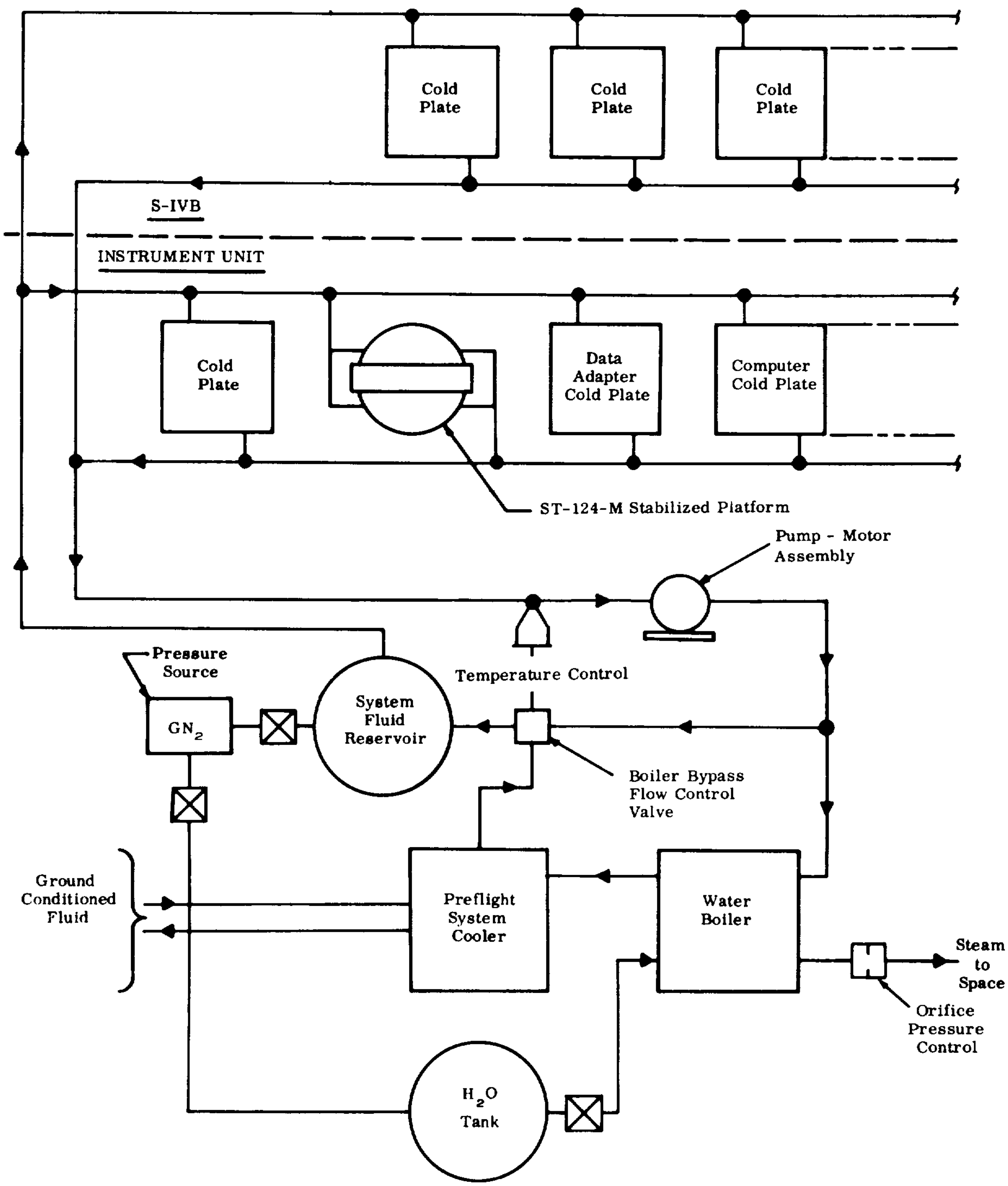

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Figure 23-5. Thermoconditioning System

The thermoconditioning system consists of the following components:

a. Thermoconditioning panels, or cold plates, on which the instrumentation is mounted. Heat transfer between the cold plate and the components takes place by conduction. The coolant circulates through tubes welded to the cold plates. Each panel or plate has a square surface area 30 inches by 30 inches and is capable of dissipating approximately 420 watts.

b. A methanol/water mixture coolant which acts as the heat transfer medium between the cold plates and the water boiler.

c. The water boiler which acts as the system heat sink. Hot coolant enters a heat exchanger where it is cooled by boiling water at reduced pressure. The water is stored in a reservoir with an expulsion diaphragm pressurized by nitrogen gas. After passing through a flow-control valve and the heat exchanger, the water vapor is vented to space.

d. A motor/pump assembly which circulates the coolant in the closed loop.

e. A boiler by-pass flow control valve which controls the coolant temperature by regulating the flow through the heat exchanger dependent upon the temperature of the coolant entering the pump.

During the prelaunch and launch phases of the mission ground conditioned fluid is circulated through the preflight system cooler which acts as the system heat sink for launch pad operation. The water boiler does not function as an active system heat sink until the vehicle reaches an altitude of approximately 115,000 feet.

23-8. ENGINE GIMBALLING SYSTEM.

The Saturn V engine gimballing system positions the gimballed engines of the active stage to provide the thrust vectors required for vehicle control. In performing this function, the gimballing system is controlled by commands initiated by the attitude control and stabilization function. (Refer to Paragraph 20-35.)

The engine gimballing system steers the vehicle along its trajectory by providing engine thrust vectors for pitch, yaw, and roll control (except for the S-IVB stage). The system is active during the ascent and the translunar trajectory phase of the mission (throughout S-IC stage, S-II stage, and S-IVB stage powered flight). As the vehicle ascends, in addition to the region of high aerodynamic pressure (35,000 to 50,000 feet), it may encounter other disturbances such as thrust misalignments and winds. The external forces produced on the vehicle by such disturbance are counteracted by gimballing the engines of the active stage providing thrust vectors which minimize vehicle structural loading and maintain the vehicle on trajectory.

After the S-IC stage has expended its useable propellants, the stage is separated from the vehicle and the gimballing system operation is switched to the gimballed engines of the S-II stage. When the S-II stage engines are cutoff at propellant depletion, the stage separates from the vehicle and the S-IVB stage engine performs the gimballing system functions. When the S-IVB stage and payload have obtained orbital velocity, and altitude, the S-IVB stage engines are cut off. The system is inactive during the earth orbital phase of the mission. The gimballing system is reactivated during the translunar trajectory phase during S-IV-B stage second burn. The system ceases to function after Apollo payload separation.

23-9. OPERATION.

The gimballed engines of the three Saturn V stages are positioned by means of similar servo actuator systems. Each of the four outboard F-1 engines of the S-IC stage are gimballed through a $\pm$ 5-degree, 9-minute square pattern for pitch, yaw and roll control, Figure 22-1. Similarly, the four outboard gimballed J-2 engines of the S-II stage provide pitch, yaw and roll control of the vehicle during the S-II stage burn. The S-II stage gimballed engines are capable of moving in a $\pm$ 7-degree square area, Figure 22-2. The single J-2 engine of the S-IVB stage is gimballed to provide pitch and yaw control of the vehicle. Roll control during S-IVB stage powered flight is accomplished by means of the roll control engines of the auxiliary propulsion system (Refer to Paragraph 22-58).

23-10. S-IC STAGE IMPLEMENTATION.

The gimballing system of the S-IC stage, illustrated in Figure 23-6, provides thrust vectoring for vehicle flight control of pitch, yaw and roll during first stage powered flight. Thrust vectoring is accomplished by positioning the four outboard gimballed engines by means of a servo actuator system. Two servo actuators, one in the vehicle pitch plane and one in the vehicle yaw plane, are required for each outboard engine. Pitch, yaw and roll control are resolved by the control computer into the proper combination of electrical commands for the actuators. Fuel, (RP-1) at approximately

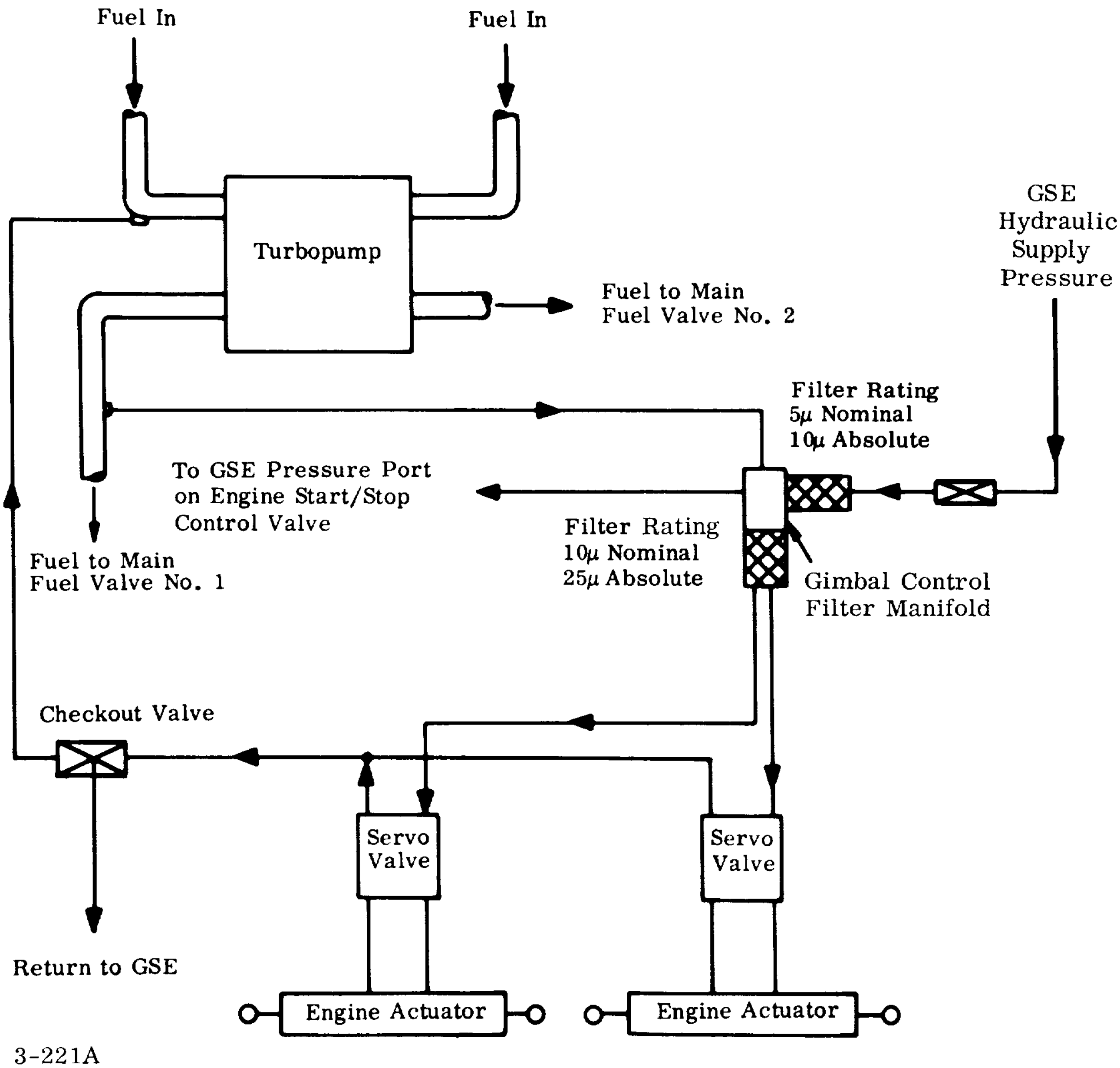

Figure 23-6. Gimballing System F-1 Engine

1800 psi is tapped off the engine turbopump discharge line. The fuel passes through the 25-micron filter and then into the high pressure port of the servo valve pilots. The valves direct the high-pressure fuel flow to the appropriate side of the engine actuators. The RP-1 from the discharge side of the actuator is returned to the turbopump fuel inlet. Maximum demand for RP-1 in this system is 235 gpm.

23-11. S-II STAGE IMPLEMENTAION.

A separate hydraulic system is used to provide thrust vector control on each of the four outboard gimballed J-2 engines. Each hydraulic system is a self-contained closed loop and includes a hydraulic pump, an auxiliary motor-pump, an accumulator-reservoir manifold assembly, and two servo actuators. The servo actuators move each gimbal engine in a ± 7-degree square pattern, one actuator operating in the pitch plane and the other in the yaw plane. The system is similar to that used on the Saturn I, S-I stage H-1 engines. (Refer to Paragraph 9-9 for additional information.) A typical hydraulic system schematic is shown in Figure 9-4.

23-12. S-IVB STAGE IMPLEMENTATION.

The S-IVB stage engine gimballing system positions the J-2 engine for vehicle control in the pitch and yaw planes. Two servo actuators receive hydraulic pressure from a hydraulic pump in a closed loop system. An auxiliary - electrically driven pump supplies pressure to the system prior to engine restart. The main engine driven hydraulic pump located on the engine-LOX auxiliary-drive pad is used during engine firing. Both pumps are of the variable delivery type to preclude undesirable heat generation during operation. The pumps, an accumulator, and a high-pressure relief valve, comprise the high-pressure system. The system accumulator supplies peak system demands and dampens high pulsations. The system is similar to that used on the Saturn I, S-I Stage (Refer to Paragraph 9-9).

23-13. SEPARATION SYSTEM.

The primary function of the Saturn V separation system is to provide positive separation of the S-IC stage from the S-II stage, and the S-II stage from the S-IVB stage during the ascent phase of the mission. (The following description does not include an explanation of the separation of the S-IVB/IU from the Apollo payload during the translunar trajectory phase of the mission).

To lift a given payload into orbit, it is desirable to use a launch vehicle of minumum weight. The design of a minimum-weight vehicle capable of lifting the payload required for the Apollo program necessitates the use of more than one propulsion stage when restricted to present space vehicle technology. During the flight of a multistage vehicle, as a stage is expended it is discarded and the next stage forward provides the thrust for continued payload boost.

23-14. OPERATION.

The Saturn V launch vehicle consists of three propulsion stages. The S-IC stage contains five F-1 engines, the S-II stage contains five J-2 engines, and the S-IVB stage has one J-2 engine. During the ascent phase of the mission, after the S-IC stage has expended its useable propellants it is separated from the launch vehicle and the S-II stage engines ignite to resume powered flight. The S-II stage in turn is discarded and the S-IVB stage ultimately achieves orbit along with the instrument unit and Apollo payload. Separation of the S-IVB stage/instrument unit from the Apollo payload occurs after S-IVB stage second burn during the translunar trajectory phase of the mission.

The S-IC stage is separated from the S-II stage using a dual plane separation scheme with a short coast mode which consists of severing the S-IC/S-II interstage in two planes. Separation occurs at two separate planes in order to detach first the depleted S-IC stage and then the S-IC/S-II interstage. The first separation plane is at MSFC station 1564, located aft of the S-II stage J-2 engine exit plane. Consequently, there is little danger of collision between the S-IC stage and the S-II stage J-2 engines, as the S-IC stage decelerates and falls away. Adequate clearance (6 feet) between the stages for S-II stage J-2 engines starting is achieved in a minimum amount of time allowing for the rapid recovery of control of the vehicle.

The second separation plane is at MSFC Station 1760 located at the forward face of the S-IC/S-II interstage.

The separation sequence is initiated approximately 146.6 seconds after liftoff with the shutdown of the center F-1 engine of the S-IC stage. Cutoff of the four outboard F-1 engines of the S-IC stage occurs about 4 seconds later when the S-IC propelland depletion signal is given. A controlled thrust termination of the four F-1 engines prevents attitude deviations which could occur from unsymmetrical booster burnout. A controlled cutoff is important because during separation there is a period of 3 to 5 seconds, between S-IC stage engine cutoff and S-II stage engine mainstage, during which the vehicle coasts in uncontrolled flight.

Following S-IC stage engine cutoff an acceleration switch triggers S-II stage ullage motor ignition when the vehicle acceleration decreases to 0.5 g's. The ullage motors provide the thrust required for propellant positioning for S-II stage engine starting and thrust buildup. Physical separation of the S-IC stage from the S-II stage begins by simultaneously severing the aft interstage and firing the S-IC stage retromotors. The retromotors decelerate the S-IC stage providing rapid and complete physical separation of the stages.

When clearance of 6 feet between the S-IC stage and the S-II stage engine exit plane is achieved, the S-II stage J-2 engines start sequence begins. The J-2 engines reach mainstage thrust 1.5 to 3.8 seconds after ignition. Second plane separation occurs after the J-2 engines are operating at full thrust. Physical separation of the S-IC/S-II interstage and the S-II stage occurs as the result of S-II stage acceleration and the axial load on the interstage due to J-2 engines' plume impingement.

Separation of the S-II stage from the S-IVB stage is initiated approximately 543.5 seconds after liftoff when propellant low level sensors initiate S-II stage engine cutoff. All five J-2 engines are cutoff at the same time. Two ullage engines which are part of the S-IVB auxiliary propulsion system (Refer to Paragraph 22-59) provide propellant positioning for S-IVB stage J-2 engine starting and thrust buildup.

The S-IVB aft skirt is severed at MSFC Station 2747 to achieve physical separation and retromotors are fired to decelerate the S-II stage. The S-II/S-IVB interstage remains with the S-II stage after separation.

Separation of the S-IVB stage/instrument unit from the Apollo payload occurs at the forward face of the instrument unit at MSFC Station 3259. There is no requirement for S-IVB stage/instrument unit deceleration during separation and consequently there are no retromotors on the S-IVB. The vehicle separation sequence is tabulated in Table 23-1.

23-15. S-IC STAGE IMPLEMENTATION.

The separation system components on the S-IC stage include eight retromotors located in pairs in the S-IC engine fairings. The motors are mounted at an angle of 7 degrees 30 minutes with the vehicle centerline. New forward thrust of seven retromotors is equal to or greater than the net thrust of the four outboard F-1 engines at the time of first plane separation. The total retromotor impulse is equal to or exceeds the total impulse of the four outboard F-1 engines during the time from 10 percent of engine mainstage thrust to zero thrust.

Table 23-1. S-IC/S-II and S-II/S-IVB Staging Sequence

Item	* Approximate Time	Remarks
S-IC stage center F-1 engine cutoff.	L. 0. + 146. 6 sec	
S-IC stage outboard F-1 engines cutoff.	—	Propellant depletion signal initiates engine cutoff.
S-II stage ullage motors ignition.	—	
LSC is fired to sever vehicle structure, and S-IC retromotors fire.	—	First plane separation occurs.
S-II stage, J-2 engine start sequence begins.	—	(Six feet clearance required between S-IC and S-II stages before engine start.)
J-2 engines operate at full thrust.	—	
LSC is fired to separate S-IC/S-II interstage from S-II stage.	—	Second plane separation occurs.
S-II stage J-2 engines cutoff.	L. 0. + 543. 5 sec	Propellant level sensors initiate J-2 engines cutoff.
S-IVB ullage engines fire.	—	(Ullage engines are part of the S-IVB auxiliary propulsion system.)
MDF fires to sever vehicle structure, and S-II retromotors fire.	—	Physical separation between S-II and S-IVB occurs. Interstage remains with S-II stage.
S-IVB stage J-2 engine ignition.	—	

*NOTE: Time values are based on preliminary information.

23-16. S-II STAGE IMPLEMENTATION.

Components of the S-II stage separation system include ullage motors and linear shaped charges.

Eight solid-propellant ullage motors are mounted 45 degrees apart on the S-IC/S-II interstage. The motor nozzels are canted 10 degrees from the S-II stage centerline to reduce the motor-out moment in the event of ullage motor malfunctions or thrust deviations. Each ullage motor produces a thrust of approximately 22, 800 pounds and burns for about four seconds.

Linear shaped charges (LSC) are used to physically sever the S-IC stage from the S-IC/S-II interstage during first plane separation, and the S-IC/S-II interstage from the S-II stage during second plane separation.

23-17. S-IVB STAGE IMPLEMENTATION.

The S-IVB stage separation system components include four retromotors and a mild detonating fuse (MDF).

Four solid-propellant retromotors radially mounted at 90 degrees intervals on the S-II/S-IVB interstage are used to decelerate the S-II stage during S-II stage, S-IVB stage separation.

An MDF is used to physically sever the S-II stage from the S-IVB stage during separation.

Retromotors are not required on the S-IVB stage for S-IVB/instrument unit separation from the Apollo payload. However, the Saturn V vehicle is designed with a structural capability for inclusion of two TX-280 solid-propellant retromotors on the S-IVB stage.

23-18. ORDNANCE SYSTEMS.

The mechanical operations performed during a Saturn V mission that require reliable, short-time high-energy, concentrated forces are performed by the ordnance system components. High reliability is achieved by providing redundant components throughout the system.

During S-IC/S-II and S-II/S-IVB staging, the vehicle structure is severed and ullage and retromotors are fired to provide auxiliary propulsion. Except for the S-IVB ullage requirements which are provided for by ullage engines of the auxiliary propulsion system, these functions are performed by ordnance system components. The physical separation of the S-IVB/instrument unit from the Apollo payload also is achieved by means of an ordnance device. Pyrotechnic - actuated cable cutters are used to release the horizon sensor protective dome. For range safety the ordnance system provides for the dispersal of vehicle propellants.

23-19. OPERATION.

Ordnance devices used on the Saturn V launch vehicle are operational during the ascent phase, and at the end of the translunar trajectory phase of the mission. Because of the potential hazard involved, the explosive initiators of ordnance devices are not installed, and the electrical circuits of the ordnance system are not completed until all personnel except the ordnance crew are clear of the launch pad.

23-20. Ascent Phase. Ordnance devices perform major functions during S-IC/S-II and S-II-S-IVB staging. Ullage motors provide vehicle acceleration for positioning of propellants in the S-II stage to prevent the admission of vapor into the propellant feed system in order to ensure successful ignition of the J-2 engines. Retromotors provide the thrust required to decelerate the S-IC stage and the S-II stage providing rapid and complete physical separation of these stages during S-IC/S-II and S-II/SIVB separation, respectively.

Physical separation of the stages is accomplished by means of a linear shaped charge (LSC) and a mild detonating fuse (MDF) which sever the launch vehicle structure at the separation planes.

A fibreglass dome-shaped cover is used to protect the temperature-sensitive horizon sensor from aerodynamic heating during first stage burn. The protective cover is jettisoned 4 to 10 seconds after S-IC/S-II stage separation. The jettison system employs pyrotechnic actuated cable cutters.

Throughout the ascent phase of the mission the range safety officer can terminate the flight any time the vehicle becomes a hazard by means of the propellant dispersion system. To attain high reliability, each stage (S-IC, S-II and S-IVB) has a separate and independent dispersion system. Upon receipt of coded signals by the vehicle from the range safety officer, the systems are actuated and the flight of the vehicle is terminated. The active stage engines are shut down and LSC's are ignited to open the propellant containers. For unmanned flights two signals are required from the range safety officer. The first signal arms the system and the second signal initiates the propellant dispersion sequence. On manned flights a time delay is built into the system between the receipt and the execution of the propellant dispersion command to allow for ejection of the CM by the LES before the LSC's are fired.

23-21. <u>Translunar Trajectory Phase.</u> Physical separation of the S-IVB/instrument unit from the Apollo payload occurs at the end of the translunar trajectory phase of the mission after S-IVB second burn. A LSC provides the cutting action to sever the vehicle structure.

23-22. S-IC STAGE IMPLEMENTATION.

Ordnance on the S-IC stage includes retromotors used during S-IC/S-II staging and propellant dispersion system ordnance.

23-23. <u>Retromotors</u>. Eight retromotors, located in pairs in each of the four S-IC engine fairings, provide deceleration of the S-IC stage during separation of the S-IC stage from the launch vehicle. The ignition charge is distributed to two pyrogen initiators in each motor by means of a confined detonating fuse (CDF) train which is connected to a detonator block. Two EBW detonators, installed in the detonator block, ignite the CDF train. Each detonator is fired by 2300$\pm$100 volts dc pulse from separate electronic bridge wire firing units upon receipt of a triggering signal from the control computer. Two firing units, and two EBW detonators are used to enhance the reliability of the system.

23-24. <u>Propellant Dispersion System Ordnance.</u> The Saturn V launch vehicle is equipped with a propellant dispersion system to provide for range safety during flight. The system ordnance for the S-IC stage consits of two separate and independent electronic bridge wire firing units which are connected to two EBW detonators. The detonators are tied to a CDF (confined detonating fuse) through a safety and arming (S&A) device. The firing units and the S&A device are similar to those used on the S-I stage of the Saturn I launch vehicle. Refer to Paragraph 9-25. Either detonator is capable of igniting the CDF. The CDF in turn initiates detonation of linear shaped charges (LSC) which are located to cut the LOX container on one side and the fuel container on the opposite side. The LSC installation consists of two parallel runs which cut a "window" opening in each container.

23-25. S-II STAGE IMPLEMENTATION. Ordnance on the S-II stage consists of ullage motors and LSC's used during S-IC/S-II staging and propellant dispersion system ordnance.

23-26 Ullage Motors. Eight ullage motors are used to provide acceleration to the S-II stage during S-IC/S-II staging for propellant positioning for J-2 engine starting. The firing circuit consists of two electronic bridge wire firing units, two EBW detonators, a detonator block and confined detonating fuses (CDF). Following receipt of the S-IC propellant depletion signal, the guidance computer initiates triggering of the firing units and 2300$\pm$100-volts dc are applied to the EBW detonators installed in the detonator block. A confined detonating fuse routed from the detonator block to each ullage motor propagates the firing charge at approximately 22,000 feet per second causing the eight ullage motors to fire simultaneously. Two pyrogen initiators mounted on the head of each ullage motor are fired by CDF fuse assemblies from separate manifolds to provide redundancy for motor ignition.

23-27. Linear Shaped Charges. Linear shaped charges (LSC) are used to physically sever the S-IC stage from the S-IC/S-II interstage during first plane separation, and the S-IC/S-II interstage from the S-II stage during second plane separation. Firing of each LSC is accomplished by an EBW system which includes two electronic bridge wire firing units, two EBW detonators and two LSC initiators. Signals from the guidance computer trigger the firing units causing discharge of a 2300$\pm$100-volt capacitor into the EBW detonators. The detonators initiate detonation of the LSC initiators which ignite the LSC at both ends.

23-28. Propellant Dispersion System Ordnance. The propellant dispersion system ordnance for the S-II stage is similar to that employed on the S-IC and the S-IVB stages of the Saturn V (Refer to Paragraph 23-24.) Two independent electronic bridge wire firing units are connected to two EBW detonators on one side of the safety and arming (S&A) device. In the armed position the S&A device transfers the shock wave from the EBW detonators to the confined detonating fuse (CDF) initiators. The CDF initiators ignite the CDF train which propogates the firing charge to 600 grains per foot LSC which ruptures the propellant containers.

23-29. S-IVB STAGE IMPLEMENTATION.

Ordnance on the S-IVB stage includes retromotors and a mild detonating fuse (MDF) used during S-II/S-IVB staging, and propellant dispersion system ordnance.

23-30. Retromotors. Four, TE-29-1B, solid-propellant retromotors, mounted on the S-II/S-IVB interstage, are used to decelerate the S-II stage during S-II/S-IVB staging. The ignition charge is distributed to two CDF initiators mounted on the head of each retromotor by means of two CDF fuse assemblies. Each of the eight CDF fuse assemblies is routed to a detonator block containing two EBW detonators which are used to ignite the fuse assemblies. A separate electronic bridge wire firing unit fires each EBW detonator upon receipt of a signal from the guidance computer. The ignition system is shown in Figure 23-7.

Retromotors are not required on the S-IVB stage of the Saturn V launch vehicle for the S-IVB and instrument unit separation from the Apollo Payload. However, the vehicle is designed with a structural capability for inclusion of two TX-280 solid-propellant retromotors on the S-IVB stage.

23-31. Mild Detonating Fuse (MDF). An MDF is used to physically sever the S-II stage from the S-IVB stage during S-II/S-IVB staging. Two redundant trains of 7 grains per foot MDF are installed in a groove in the aft skirt compression plate. A circumferential tension plate riveted to the aft skirt and bolted to the S-II/S-IVB interstage joins these structures at the separation plane. The thinnest section of the tension plate (7075-T6 aluminum) which is 0.040 inches thick, is located directly over the groove containing the MDF. When the MDF is fired, the tension plate is severed. The two trains of MDF are fastened to a detonator block at each end. Two EBW detonators, fired by separate electronic bridge wire firing units, are used to ignite each end of the MDF's. The MDF's are enclosed with a clear vinyl plastic cover approximately 0.020 inches thick. Installation details are shown in Figure 23-8.

23-32. Propellant Dispersion System Ordnance. The propellant dispersion system ordnance for the S-IVB stage is similar to that used on the S-IC and the S-II stages. (Refer to Paragraph 23-24). Propellant dispersion ordnance includes two

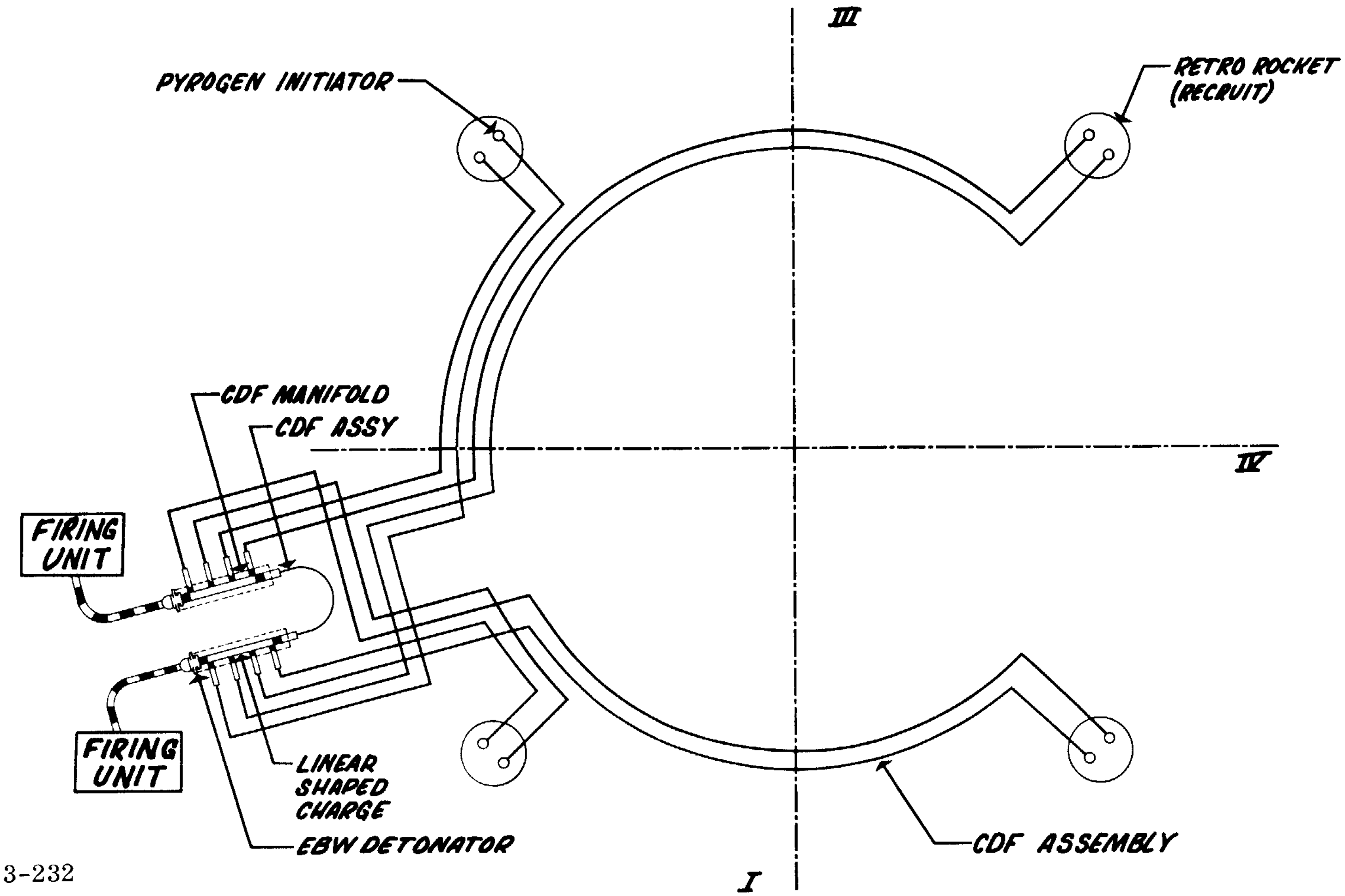

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Figure 23-7. Retromotor Ignition System

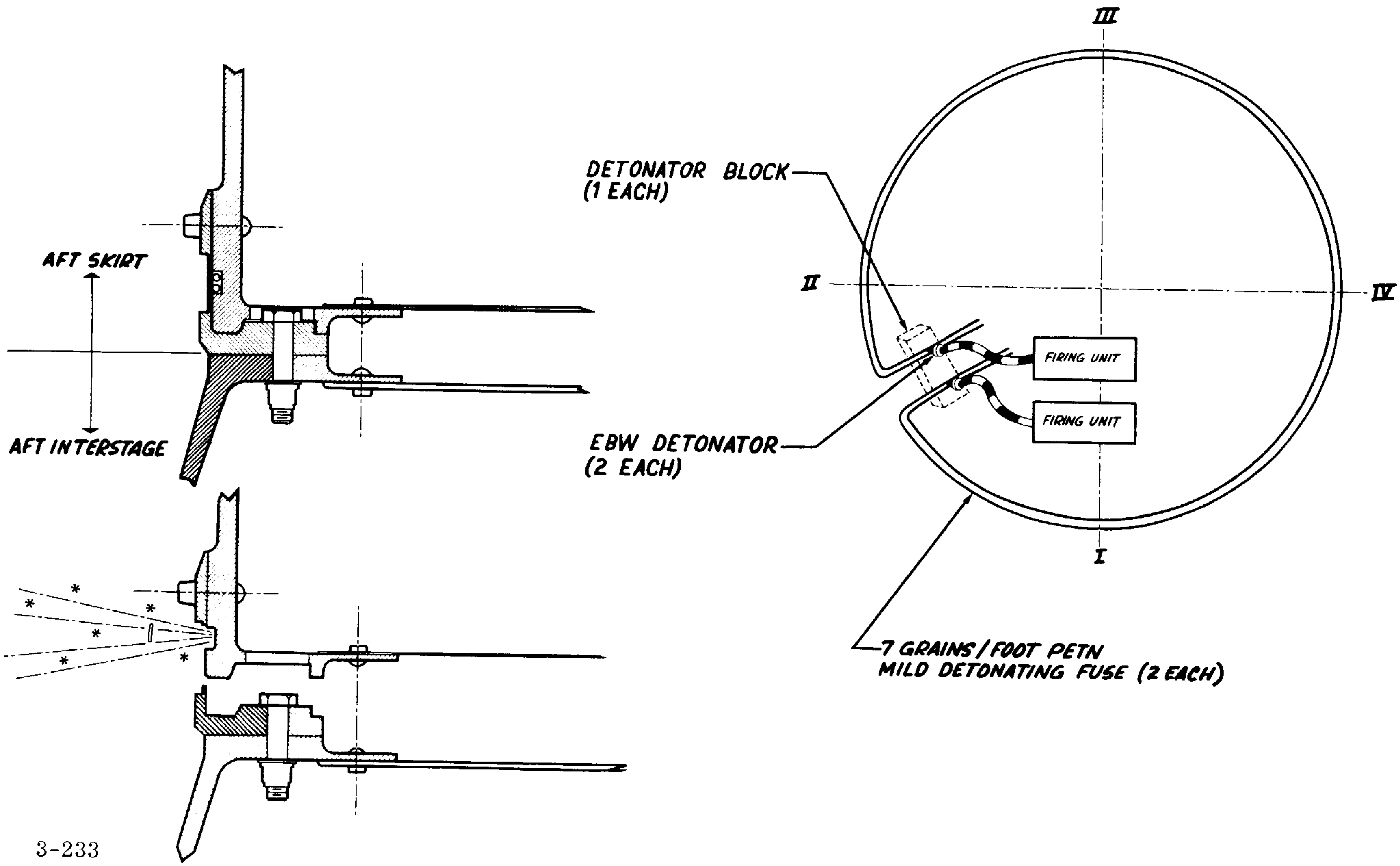

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Figure 23-8. MDF Installation, S-IVB Separation electronic bridge wire firing units, two detonators, a safety and arming device, confined detonating fuse (CDF) assemblies and linear shaped charges (LSC). LSC's located in the systems tunnel rupture the LH_2 container. An LSC ring is used to cut open the bottom of the LOX container. The LSC's are tied together by CDF.

23-33 INSTRUMENT UNIT IMPLEMENTATION.

A fiberglass dome-shaped cover is used to protect the temperature-sensitive horizon sensor from aerodynamic heating during ascent of the launch vehicle. The horizon sensor is located 18 inches aft of the forward interface of the instrument unit, Figure 21-14. Before the horizon sensor can become operative the protective dome must be jettisoned. The jettison system employs three pyrotechnic-actuated cable cutters for triple redundancy, three spring-loaded latches which clamp the dome to the vehicle skin, and a length of cable which encircles the base of the dome to hold the latches in the clamping position. When the pyrotechnic charges in the cable cutter are fired, the cable is cut in three places. The latches open under spring pressure and the dome is released. The system will operate satisfactorily if only one cable cutter actuates. Sea level atmospheric pressure at approximately 14.7 psia, sealed inside the dome, provides the thrust required to jettison the cover.

23-34. PLATFORM GAS-BEARING SUPPLY SYSTEM.

The platform gas-bearing supply system furnishes filtered GN_2 at a regulated pressure, temperature, and flow rate to the gas-bearings of the ST-124-M stabilized platform. The GN_2 is supplied to the stabilized platform from the start of checkout during prelaunch, until separation of the S-IVB/instrument unit from the Apollo payload at the completion of the translunar trajectory phase of the mission.

The system is similar to the platform gas-bearing supply system used on the Saturn I launch vehicle (Refer to Paragraph 9-33).

CHAPTER 4

SECTION XXIV

GROUND SUPPORT EQUIPMENT

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SECTION XXIV.
GROUND SUPPORT EQUIPMENT

24-1. GENERAL.

The Saturn V ground support equipment (GSE) includes all of the ground equipment required to support the fabrication, checkout, transportation, static testing, and launch operations related to the vehicle and its stages (S-IC, S-II and S-IVB) and instrument unit. The GSE in this section excludes GSE peculiar to launch operations which is described in Volume I. In supporting the above operations, the GSE is formed into functional ground system, subsystem, and unit configurations. The various configurations are employed as required at all locations involved in the research and development of the vehicle and its stages. Since the operation of each configuration may vary depending on the location where used, an operational description is not contained in this document. Instead, the major GSE is listed and primary functions described.

24-2. ELECTRICAL SUPPORT EQUIPMENT, SATURN V.

The Saturn V ESE is used during the checkout, static testing, and launching of the vehicle. The majority of this equipment is located at the Automatic Ground Checkout Station (AGCS). This ESE is classified as follows:

a. Monitoring and Control Equipment
b. System Integration Equipment
c. Networks, Distribution and Control Equipment
d. Ground Equipment Test Sets
e. Recording Group Equipment
f. Peripheral Equipment
g. Overall Test Equipment
h. Systems Integration Sets

With the exception of the monitoring and control equipment, and recording group equipment, MSFC is responsible for fabrication of all of the above. For these two classifications, MSFC has partial fabrication responsibility. A summary of the Saturn V ESE functions is given in Table 24-1.

Table 24-1. Electrical Support Equipment, Saturn V

Equipment	Function
Monitoring and Control Equipment	a. Provides monitoring and control of systems under test by means of panel meters, switches, light banks, and displays. b. Provides control and display equipment for the following systems: network, propulsion, navigation, measuring and RF, ordnance, emergency detection, mechanical, computer control and display, propellant loading, systems integration, test conductors console, and launch conductors console.
Systems Integration Equipment	Used for signal distribution to the stage GSE from the computer and from the computer to the monitoring and control consoles.
Networks, Distribution and Control Equipment	a. Provides proper distribution and sequencing of the control signals and power to the particular stage under test. b. Provides the capability of manual operation by means of switches for relay control and meters on the front panels.
Ground Equipment Test Set (GETS)	Provides signals for checking out the proper operation of the GSE prior to connecting it to the integrated vehicle or stage simulators.
Recording Group Equipment	Records all vehicle discrete outputs and inputs during the checkout sequence.
Peripheral Equipment	a. The peripheral equipment countdown clock provides the time base for countdown separation. The clock, synchronized with WWV, has a real-time readout capability and can supply real-time commands to the instrument unit guidance programmer through the RCA-110 computer. b. The signal conditioning equipment reduces the inputs from 28-volt dc to a standard 5-volt dc acceptable to the computer.

Table 24-1. Electrical Support Equipment, Saturn V (Cont'd)

Equipment	Function
Overall Test Equipment (OAT)	Simulates functions which actually cannot be performed by the systems under test because of the resulting hazardous conditions.
Systems Integration Sets (SIS)	Simulate interface signals between stages.

24-3. GROUND SUPPORT EQUIPMENT, S-IC.

The S-IC stage GSE equipments are classified as test, checkout and monitoring; transportation, protection and handling; maintenance; and propellant and gas servicing. Tables 24-2 through 24-5 list the equipments and functions of each classification.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC

Equipment	Function
Electrical Power Control Equipment	a. Provides a central source for all ground power control. b. Provides visual indications of voltages present in the stage.
DC Ground Power Station	Provides 28-volt dc power for the stage and GSE.
400-cps AC Ground Power Supply	a. Provides 115-volt, 400-cps, three-phase ac power for the stage and GSE.
Battery Test Set	Verifies that the stage batteries deliver the required outputs.
UDOP Test Set	Performs a complete evaluation of the stage UDOP tracking beacon.
AZUSA Test Bench	Performs a complete evaluation of the stage AZUSA tracking beacon.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Antenna Test Set	Used for checking voltage standing wave ratio (VSWR), attenuation, and load characteristics of the stage antennas and associated feed lines.
Antenna Couplers	Used as RF links between GSE and stage antennas. Limits spurious radiation in the test area.
Exploding Bridge Wire Set (EBW)	a. Provides stimuli used to checkout the EBW subsystem. b. Sensors monitor the subsystem and the test set evaluates the compatibility of the sensor response code with the stimuli output code. c. The compatibility or incompatibility results in the generation of a GO or NO-GO condition respectively.
Malfunction Detection Test Set	Checks the operation of the malfunction detection system by supplying the proper stimuli to simulate a malfunction and then checking for proper reaction time and signal output.
Thrust Vectoring Test Set	Checks the thrust vectoring control system by injecting the appropriate signals into the control system and monitoring the operation of the control system and associated engine displacement.
Stage Electronic Weighing System	Used to weigh the electronic complement of stage.
Range Safety Test Set	Monitors the Saturn V launch vehicle to ensure that the vehicle maintains the programmed liftoff and flight pattern within the limits specified for range safety.
Test Plate and Tool Kit	Used to perform leak and functional tests on the F-1 engine, and to seal the engine for pressure tests.
Electrical-Pneumatic Hydraulic Components Test Stand (GFE)	Used in performing tests on hydraulic, electrical, and pneumatic parts.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Hydraulic-Pneumatic Cable and Hose Cart	Used at each test location to store hook-up cables, hoses, and connectors required for the mechanical tests sets.
Ground Hydraulic Power Unit	Used in performing F-1 engine control system functional checks and engine gimballing checks.
Vibration Safety Cutoff	Used during engine static firing as a combustion stability monitor and control. Ensures that the engine is cut off before engine or stage damage is incurred due to extreme vibration levels.
Cryogenic Component Test Stand	Used for testing cryogenic components at actual operating temperatures.
Pneumatic and Hydraulic Hose Set	Used for flexible pneumatic and hydraulic hookups between the stage, GSE, and fixed plumbing for stage checkout.
Stage Work Platforms	Provide access to electrical and pneumatic test connections on the periphery of the stage.
Mechanical Adapter Kit	Supplied to preclude the possibility of mismatched pneumatic end connectors between the stage and ground termination points.
Engine Firing and Sequencing Test Set	Used during F-1 engine startup and shutdown sequences.
Engine and Propellant System Heatup Test Set	Used for checking the engine and propellant heater systems.
Command Destruct System Test Set	Used to verify the stage command destruct system. (The test set generates coded RF signals (stimuli) and monitors the command destruct system responses.)
RF Test Bench	Provides a central source of equipment and power used to calibrate, troubleshoot, and repair the RF equipment of the GSE.
Upper Stage Simulator	Provides the proper loads for circuits which normally terminate an upper stage.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
S-IC Simulator	Used to checkout the GSE.
Pneumatic Low-Pressure Supply Set	Used to pressurize stage low-pressure pneumatic systems prior to performance of stage and engine leak and pressure components tests.
Pneumatic High-Pressure Supply Set	Used to pressurize stage high-pressure pneumatic system prior to performance of stage and engine leak and pressure component tests.
Pressure Readout Pneumatic Set	a. Used to indicate actual supply pressure from the high and low-pressure supply sets. b. Facilitates the setting of pressure switches relief valves.
Fuel Tanking Simulator	Supplies evaluation signals to the fuel control panel.
Fuel Density Simulator	Supplies evaluation signals to the fuel density monitor panel.
LOX Tanking Simulator	Supplies signals to the LOX tanking control panel which allow the performance of the control panel to be evaluated.
Engine Simulator	Simulates the electrical network of the engine and verifies the operation of the GSE.
Pneumatic Flow Tester	Used to measure gas-flow rates in the stage pneumatic systems.
Leak Detector Set	Used to detect minute leaks in the stage pneumatic system, propellant and plumbing, and engine system.
Portable Gas Mixture Cart	Used for mixing the pressurizing nitrogen gas with the tracer gas used with the leak detecting equipment.
Electrical-Pneumatic Checkout Cart	Used to perform electro-pneumatic leak and functional tests on the F-1 engine.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
System Safety Monitor and Interface Equipment	Used to monitor all systems of the stage when the stage is under test.
Control and Monitor Console	a. Used to monitor and control the test and servicing complex.
	b. Can be used to override the computer and electrical launch equipment to stop test and remove power from the stage in the event of an emergency.
	c. Used to control the closed-loop television system.
Closed-loop Television System	Provides close range visual display of critical areas during the stage test and checkout.
PCM/FM Telemetry Ground Station	a. Verifies the proper operation of instrumentation system transducers.
	b. Used to test the airborne PCM/FM telemetry system.
FM/FM Telemetry Ground Station	a. Verifies the proper operation of instrumentation system transducers
	b. Used to test the airborne FM/FM telemetry system.
SS/FM Telemetry Ground	a. Verifies the proper operation of instrumentation system transducers.
	b. Used to test the airborne SS/FM telemetry system.
T/M Bench Test Station	Used to perform bench level tests on the components and assemblies of various telemetry systems.
Inverter Test Set	a. Controls the 400-cps primary ac power that is applied to the inverter.
	b. Performs tests on the inverter under various conditions of load, power factor, frequency, and other parameters.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Emergency Power Station	Provides 28-volt dc and 115-volt, 60 cps, three-phase ac power to critical circuits of stage and GSE if the DC and AC Power Stations lose primary power during test.
Distribution and Junction Boxes	Provide equipment flexibility, maintainability, and accessibility to stage and GSE test points.
Electronic Test Bench	Used for calibrating, trouble-shooting, and repairing GSE electronic equipment.
Tape Recorder	Used as a high-frequency recorder.
Direct Writing Recorder	Provides a semi-permanent visual record of a test. The chart is available for use upon completion of test.
Stage Wiring Checkout Set	Used to run continuity, short circuit, and ground isolation tests on the stage wiring system.
Instrumentation Test Station	a. Verifies the functions and calibration of the stage instrumentation system. b. Provides switching signals for instrumentation system signal conditioners.
Instrumentation Components Test Station	Used to test and calibrate the measurement transducers, signal conditioners, and instrumentation racks prior to installation into the instrumentation system.
Digital Data Acquisition (DDAS) Ground Station	Used for automatic checkout of some of the stage instrumentation.
Leak Detector Test Set	a. Used to detect the presence of dangerous vapors. b. Issues automatic warning when a contamination level is reached.
Stage DC Ground Power Supply	Provides dc ground power to the stage during test and checkout by means of two 28-volt dc, 250 amp units, one for each major stage bus.
GSE DC Power Supply	Provides dc power required by the test and checkout equipment by means of a single 28-volt dc, 500 amp unit.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
60 Cycle AC Ground Power Supply	Provides 120/208 volt, three phase, 60 cycle, 4 wire power to the ground computer.
Over-all Test (OAT) Battery	Provides ground power to the stage battery bus in lieu of the vehicle battery during test and checkout.
Computer Input/Output DC Power Supply	Provides minus 28-volt dc power required to operate relays in the computer discrete output distribution rack.
Audio Communication Equipment	Provides general and local area paging, and intercommunication equipment.
Stage Electronic Weighing System	Used to weigh a dry S-IC stage. The system consists of electronic weighing instruments and structural supports.
CRT Interface and Control Logic Equipment	The cathode ray tube (CRT) interface and control logic equipment contains the interface and control logic between the CRT display equipment and the ground computer.
Ground Computer System	Utilizes a general purpose digital computer for control testing, data recording, and data evaluation and display during stage testing.
Count-Clock System	Generates standard time, count time, programmable time interval signals, and square wave timing signals.
Input/Output Distribution Equipment	Insures computer interface compatibility with other test and checkout equipment.
Test Step Indicator Equipment	Accepts programmed computer output, decodes the information, transmits it to the appropriate test station, and displays the test step being accomplished at the individual test station.
Audio Communication Equipment DC Power Supply	Provides 28-volt dc, 100 amp power to the audio communication equipment.
Range Safety and Ordnance Test Set and Antenna	Used to test the stage range safety and ordnance systems in the installed state.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Electrical Networks Test Set	a. Commands the computer discrete outputs on and off, one at a time. b. Provides the necessary terminations and control signals from the upper stage, when the upper stage is not available. c. Provides the signals to program the stage switch selector and the computer interface to allow these signals to be controlled by the computer. d. Simulates electrical signals necessary to permit a complete simulated countdown and launch of the S-IC stage. e. Checks out the emergency detection system of the S-IC stage. f. Converts the count-clock system count-time signals from parallel form to serial form.
Signal Monitoring Equipment	Used to record selected stage pressures, events and analog signals.
Launch and Ignition Sequencing Equipment	Provides signals, in the proper sequence and time relationships, to control the launch functions required during count-down and firing and to control the ignition of the five engines of the S-IC stage.
Mechanical Test Control Equipment	Controls the Mechanical Test Station through control and display equipment, status displays and communications equipment.
Pneumatic Supply Unit	Controls the helium and either nitrogen or dry air pneumatic facilities for supplying the facility air supply, valve control pressure, LOX dome and gas generator LOX purge, fuel gas generator purge, pneumatic pressure test module, LOX container pre-pressurization, fuel container pre-pressurization, stand-by purge, and helium bottle fill.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Pneumatic Pressure Test Racks	Provide pneumatic pressures and flows to the S-IC stage during testing of the propulsion system.
Pneumatic Flowmeters Group	Measures pneumatic leakage rates and flow rates from various components and subsystems of the S-IC stage propulsion system.
Hydraulic Power Supply Unit	Supplies RJ-1 fluid to the S-IC stage through the hydraulic power distribution equipment to perform such functions as bleeding, engine start tests, and engine gimbal tests.
Portable Electrical Hydraulic Control Unit	Provides electrical control and monitoring of hydraulic power supplied to the stage.
Hydraulic Power Distribution Equipment	Routes and distributes hydraulic fluid from the hydraulic supply unit to the umbilical substitute connection in the stage test cells.
Non-Flight Checkout Instrumentation Equipment	Used during factory test and checkout to drive pressure displays in the mechanical test control equipment and to initiate emergency vent action upon overpressure detection.
Digital Data Acquisition System Ground Equipment	Performs demodulation, re-synchronization, digital data reconstruction, channel demultiplexing, serial to parallel conversion, buffering, digital-to-analog conversion, and provides for analog or digital displays for manual monitoring.
Integrated Telemetry Group Equipment	Receives, demultiplexes, and decodes telemetry signals from the stage during factory checkout.
RF Terminal Equipment	Serves as a distribution center for all telemetry RF and video signals arriving from the several test cells.
Remote Automatic Calibration Unit	Provides an automated method of calibration and test for the vehicle measurement subsystem.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Remote Automatic Calibration Portable Control Unit	Provides channel identification of all measurement channels, manual calibration of measurement signal conditioners, and manual calibration of the stage telemetry subcarrier oscillators.
DDAS Tape Recorder	Used for recording DDAS data during DDAS/computer testing.
Instrumentation Calibration Equipment	Provides the controls, power supply, and signal generator to calibrate the measurement and telemetry system.
Portable Instrumentation Calibration Equipment	Provides for ac amplifier calibration and temperature, pressure, and acoustic transducer stimulation and simulation.
Telemetry Digitizing Equipment	Receives the analog outputs of the PAM/FM/FM and the FM/FM discriminators from the integrated ground station for digitizing and programming into the stage DDAS format.
Electrical Distribution Equipment	Provides electrical interconnections between the test and checkout equipment and the stage to facilitate performance of system checkout versatility.
Electrical Interconnecting Cabling	Provides circuit continuity between the test and checkout equipment and the stage.
Umbilical Assembly Simulators	Provide fluid and electrical connections to the vehicle for testing and checkout.
Hose and Adapter Set	Provides flexible pneumatic hose lines for connecting pneumatic feed lines from the pneumatic supply unit to the pneumatic distributors on the forward and aft stage test platforms, and connecting test points on the S-IC stage to the pneumatic distributors on the forward and aft stage test platforms.
Pneumatic Distribution Equipment	Provides transition points between hard tubing runs and flexible hose lines.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Pneumatic System Leak Detector Set	Used for detecting pneumatic leaks in the stage propulsion system.
Antenna Checkout Set	Used to test the radiator VSWR, coaxial subsystem attenuation, and antenna system VSWR prior to system checkout of the completely assembled stage.
Stage Weighing Equipment	Used in determining the dry weight and longitudinal center of gravity of the S-IC stage.
Area Contamination Detection Equipment	Used to detect RP-1, RJ-1, and trichloroethylene vapors, and to monitor the content of oxygen.
Special Stage Electrical Maintenance Set	Provides special instruments and tools required to perform maintenance on installed stage electrical and electronic equipment.
Special Stage Mechanical Maintenance Set	Provides special instruments and tools required to perform maintenance on installed stage mechanical equipment.
Special Test Fixture Set	Used for sealing off or plugging sections of ducting and collecting leakage past valve seats during leak testing.
Horizontal Stage Internal Access Equipment	Provides access to the forward skirt, intertank, and thrust structure areas of the S-IC stage.
Manual Engine Actuator	Provides for moving or holding the outboard F-1 engine after the engine is stage mounted.
Engine Component Simulator Set	Provides substitutes for the spark exciter and monitor hardware and the turbine exhaust igniter hardware.
Ground Cooling Equipment	Provides a source of air for cooling electrical equipment installed in the forward skirt section of the S-IC stage.

Table 24-2. Test, Checkout and Monitoring Equipment, S-IC (Cont'd)

Equipment	Function
Ground Equipment Test Set (GETS)	Provides for electrical checkout and verification of the integrated test and checkout complex. The GETS takes the place of the stage for this purpose.
Special Test and Checkout Electrical Calibration and Maintenance Set	Used to perform calibration and maintenance of the electrical test and checkout equipment.
Special Test and Checkout Mechanical Calibration and Maintenance Set	Used to perform calibration and maintenance of the mechanical test and checkout equipment.
Data Processing Station	Used for processing large volumes of data at high speed. Utilizes a central processor control console, magnetic tape handler, magnetic tape controller, high speed printer, printer controller, card reader, card punch, and a paper tape reader and punch.
Forward Stage Test Platforms	Provide personnel and equipment access to various locations at the forward end of the stage while the stage is in a horizontal position.
Intertank Umbilical Platform	Provides personnel and equipment access to the umbilical plate and to the access door at the intertank area while the stage is in a horizontal position.
Aft Stage Test Platforms	Provide access to the aft end of the S-IC stage to support test and checkout operations.

Table 24-3. Transportation, Protection and Handling Equipment, S-IC

Equipment	Function
Main Stage Transporter Dolly	Used for moving the S-IC stage overland.
Main Stage Transporter Support	Supports the stage during all phases of horizontal assembly, movement, and testing in the factory and field.

Table 24-3. Transportation, Protection and Handling Equipment, S-IC (Cont'd)

Equipment	Function
Forward Handling Ring	Supports the forward end of the stage during transportation and handling.
Aft Support Cradle	Supports the aft end of the stage and provides tiedown fittings for stage tie-down to transporter support structure.
Top Lifting Sling	Used to lift the stage for handling and erection.
Bottom Lifting Sling	Used to lift the aft end of the stage for handling and erection.
Lifting Yoke	Used to attach the bottom lifting sling to hold down attachments.
Fin Sling	Used in lifting, handling, and installing or removing the stage fins.
Transportation Accessories Kit	a. Used to prepare the stage for transportation, protect small parts during transportation, and to tie down and block and shore the stage. b. Used to transfer the stage to supports during factory tests.
Shoring and Blocking Kit	Used for shoring and blocking the stage and main stage transporter support onto the barge.
Stage Attach Fittings	Used to prepare the stage for transportation, protect small parts during transportation and to prepare the stage for testing after transport.
Fin Cradle	Supports and protects the fins during transportation and storage.
Shroud Installation and Removal Equipment	Used to lift, handle, install, and remove the engine shrouds from the aft end of the S-IC stage.
Shroud Cradle	Provides rigid support and containment for the F-1 engine shroud during handling, storage, and shipping.

Table 24-3. Transportation, Protection and Handling Equipment, S-IC (Cont'd)

Equipment	Function
Component Containers and Covers Set	Provides packaging for shipment and storage of all components of the S-IC stage other than fins and shrouds, which are too sensitive to remain on the stage during launch or water transportation operations.
Desiccant Breather Set	Provides the equipment necessary for propellant container preservation.
Event Recorder	Monitors and records temperature, pressure, humidity, stresses,. and acceleration in critical areas of the stage during transportation.
S-IC Stage Weight Simulator	Simulates the S-IC stage weight, dimensions, center of gravity, and end mounting configuration for equipment and facility checkout usage.
Work Platforms and Bulkhead Protectors	Used in the maintenance of instrumentation, plumbing, and components in the upper LOX bulkhead area, the stage intertank area, and the thrust structure area, while the stage is in the vertical position.

Table 24-4. Maintenance Equipment, S-IC

Equipment	Function
Vacuum-Pressure Calibration Cart	Used to calibrate stage instruments which operate in a positive pneumatic pressure environment.
Pneumatic-Pressure Calibration Set	Used to calibrate stage instruments which operate in a positive pneumatic pressure environment.

Table 24-4. Maintenance Equipment, S-IC (Cont'd)

Equipment	Function
General Purpose Test Equipment	Used to support GSE calibration, trouble-shooting, and repair.
Hydraulic-Pressure Calibration Cart	Used to calibrate stage and support equipment hydraulic pressure switches and gages.
Temperature Calibration Cart	Used to calibrate stage and support equipment thermo-switches, temperature gages, and thermocouples.
Small Parts Handling and Special Tools Kit	Provides the equipment necessary to assemble, disassemble, maintain, inspect, and service the S-IC stage and GSE.

Table 24-5. Propellant and Gas Servicing Equipment, S-IC

Equipment	Function
RP-1 Filling and Draining Equipment	Used for RP-1 filling and draining through a single interface disconnect fitting.
RP-1 Fuel Level Adjustment System	Used to adjust fuel weight to within ± 0.25 percent of stage requirements.
LOX Replenishing System	Used to maintain a controlled supply of LOX at a tankage level within ± 0.25 percent of the stage weight tolerance.
LOX Filling and Draining System	Used for LOX filling and draining through two interface quick-disconnect fittings.
Preflight Ground Pressurization System	Provides preflight ground pressurization of fuel (RP-1) and helium containers.
Stage LOX Container Pressurization System	Provides preflight ground helium pressurization of the LOX container.
Preflight Purging System	a. Provides preflight purging of the F-1 engine pump shaft seals. b. Charges the stage inert gas supply for inflight use.

Table 24-5. Propellant and Gas Servicing Equipment, S-IC (Cont'd)

Equipment	Function
Instrument Unit Conditioner	Uses GN_2 to purge and condition the instrument unit.
Stage Compartment Conditioner	Uses GN_2 to purge and condition the forward end of the stage and between the propellant containers.
Stage Fuel and LOX Propellant Bubbling and Measurement System	Provides a helium gas supply for stage propellant bubbling and fuel and LOX measurement.
Stage Valves Actuation System	Provides a nitrogen gas supply for stage valve actuation.
Water Deluge System	Supply water to an interface disconnect fitting and to a pad spray system for fire or explosive hazard control.
F-1 Engine Servicing System	Provides nitrogen purging, trichlorethene flushing, deionized water charging, and RP-1 pressurized fluid for gimballing and engine valve actuation systems.

24-4. GROUND SUPPORT EQUIPMENT, S-II.

The S-II stage GSE is classified as test, checkout and monitoring; transportation, protection and handling; servicing; and auxiliary. Tables 24-6 through 24-9 list the GSE and functions of each classification.

Table 24-6. Test, Checkout, and Monitoring Equipment, S-II

Equipment	Function
Electric Cable Test Set	Used to test the cables which interconnect the stage component checkout GSE.
Test Conductor Console	Provides control and display of all necessary portions of the S-II GSE required for automatic checkout of stage systems.
Remote Power Distribution Rack	Provides remote control and transfer electrical power from facility power to stage.
Data Printout Set	Provides the permanent printed record essential for the tests performed by the S-II checkout GSE.
Command Destruct Receiver (CDR) Checkout Rack	Used to manually check out the command destruct receiver and associated antenna systems.
Computer Program Input Set	Provides the means for inserting new programmed material into the computer, thereby enabling the computer complex to perform the desired automatic checkout procedures.
Auxiliary Memory Rack	Used for data and program storage. Data received by the auxiliary memory from the computer will be stored until the computer requests it for data printout and for use at the data processing center. The program which the computer receives will be stored in the auxiliary memory. The test conductor will command the computer to search for a desired test program. The test program is then shifted into the computer memory and performed.

Table 24-6. Test, Checkout, and Monitoring Equipment, S-II (Cont'd)

Equipment	Function
Buffer Equipment Rack	Used to isolate and amplify signals between the computer and the GSE checkout station.
Local Digital Driver Link Rack	Provides the digital communication link between the computer complex and the remote digital drive link rack.
Remote Digital Driver Link Rack	Provides the digital communication link between the local digital drive link rack and the remote located checkout stations.
Computer Isolation Rack	Provides logic level conversion for all GSE buffer equipment input lines to the computer, and isolates computer frame ground from other GSE grounding.
Local Static Firing "A" Rack	Provides an interface for separation of the electrical checkout station at static firing sites. Provides for manual control of a limited number of functions required for static firing operations.
Hydraulic Power Console	Used to provide ground hydraulic power to check out rocket engine hydraulic system.
Local Static Firing "B" Rack	Provides vibration safety cutoff signals, displays of actuation of engine cutoff signals, and time code signals.
Pneumatic Checkout Console Set	Used to check out the S-II stage pressurization systems, and performs or assists in performing lead and functional checks on the engine and propellant systems.
Remote Static Firing "A" Rack	Provides an interface for separation of the electrical checkout station at static firing sites. Provides equipment necessary to detect and control hazardous conditions.

Table 24-6. Test, Checkout, and Monitoring Equipment, S-II (Cont'd)

Equipment	Function
Central Time Buffer Rack	Used to receive, translate, and amplify timing signals and apply these signals to computed and visual display stations.
Ground Equipment Test Set	Used to verify the functional readiness of the stage checkout GSE and aid in developing automatic checkout programs.
Engine System Flow Monitoring Unit	Provides a means for measuring and monitoring engine system bleed flow.
Automatic Checkout Computer	Used as part of the GSE for automatic program control cf S-II systems tests, preliminary data storage, and arithmetic operations.
Automatic Control Rack	Provides buffering and decoding functions for automatic control of the station, and routing of station response signals to the computer or to displays.
Manual Control and Display Rack	Provides the local control capability of the electrical checkout station, and displays the stage response signals during the checkout of the stage.
Signal Distribution Rack	Provides the electrical checkout station with the capability of selecting and distributing proper signals to the S-II stage or display panels as required.
Scanning Rack	Enables the automatic checkout computer to readily scan S-II stage hardware discrete signals.
Special Data Rack	Used for monitoring critical stage functions and commercial items to facilitate calibration and troubleshooting.
Station Control and Display Rack	Provides control of station power supplies, measuring instruments, limit detectors, echo checks; and displays for station status, test data, and stage responses.
Local Control and Display Rack	Provides facilities for the local control of engine stimuli and propellant fill, and associated displays.

Table 24-6. Test, Checkout, and Monitoring Equipment, S-II (Cont'd)

Equipment	Function
Stage Substitutes Rack	Provides necessary stimuli to verify proper functioning of the stage flight control, engine actuation, and separation systems.
Discrete Display Rack	Displays the discrete responses from the mechanical station, S-II stage systems, and the interlock relay rack.
Relay Interlock Rack	Provides relay interlock of commands to the S-II stage and responses from the S-II stage.
Automatic Control and Display Rack	Used to display the responses of the digital data acquisition station, accepts digitally encoded logic control signals from the general purpose computer, and decodes program commands and displays stage responses.
Manual Control and Display Rack	Provides the local operation of the digital data acquisition station and display distribution.
PCM Rack	Demodulates the 600 kc carrier signal from the DDAS system on board the S-II stage. Performs data regeneration, decommutation, and series to parallel conversion.
Automatic Control and Display Rack	Displays the responses from the telemetry station and the digital data acquisition station.
Digitizing System Rack	a. Converts PAM/FM/FM and FM/FM data to a PCM format for comparison in the computer complex with the digital data acquisition station. b. Permits computer selection of channels to be digitized for computer entry.
Computer Adapter Rack	Provides binary to binary-coded decimal (BCD) conversion and routes the BCD to displays located at other stations and the computer complex.

Table 24-6. Test, Checkout, and Monitoring Equipment, S-II (Cont'd)

Equipment	Function
Oscillograph Rack	Provides quick readout or permanent recording of the outputs from the single sideband rack, discriminator rack, decommutation rack, and PCM rack.
Decommutation Rack	Separates the time shared channels of the amplitude modulated commutated pulse train from the discriminator links.
PCM Rack	Decommutates the signal from the DDAS on board the S-II stage and allccates a particular signal to the desired display device.
Time Code Rack	Provides a time code signal for the various systems requiring a synchronous timing signal.
Automatic Checkout Program Set	Provides all of the digital computer instructions (program tapes) as required for particular stage systems. A separate program set will be provided for each functionally different stage checkout.
EBW Pulse Checker	Used during checkout to monitor the pulse output of EBW system firing units of the destruct system, separation system, and the ullage rocket motors.
Staging Area Cable Installation	Used to make the electrical interconnection between facility power, GSE, S-II stage, and computer complex.
Acceptance Stand No. 1 Cable Installation	Used on the acceptance test stand to connect the S-II stage to the controlling GSE through the facility wiring.
MTF Firing Control Center Cable Installation	Used to connect GSE in the firing control center to the stage through facility wiring.
Acceptance Stand No. 2 Cable Installation	Used on the acceptance test stand to connect the S-II stage to the controlling GSE through the facility wiring.

Table 24-7. Transportation, Protection, and Handling Equipment, S-II

Equipment	Function
S-II Stage Pallet	Supports the S-II stage in the horizontal position and provides a means of rotating the stage while on the transporter.
Transporter Forward Truck	Converts the S-II stage pallet into the roadable S-II stage transporter.
Transporter Aft Truck	Converts the S-II stage pallet into the roadable S-II stage transporter.
S-II Stage Interstage Pallet	a. Provides cradles for the horizontal support of stage. b. Used as the chassis for the stage transporter.
Interstate Transporter Forward Truck	Converts the forward end of the S-II stage pallet into an transporter configuration.
Interstate Transporter Aft	Converts the aft end of the S-II stage pallet into an transporter configuration.
Forward Stage Support Ring	Supports stage on transporter and provides attach points for forward hoisting frame.
Aft Stage Support Ring	Supports stage on transporter and provides attach points for aft hoisting frame.
Transport Illumination Set	Provides illumination for the S-II stage and surrounding areas during night highway transportation.
Interstage and Static Firing Skirt Sling	Used to hoist and maneuver static firing skirt and interstage.

Table 24-7. Transportation, Protection, and Handling Equipment, S-II (Cont'd)

Equipment	Function
Stage Front Cover; Stage Body Cover; Stage Aft Cover	Provides environmental protection for the stage against sand, dust, water, snow, etc., during handling, transportation, and storage.
Interstage Aft Cover	Provides environmental protection of the interstage during delivery and storage cycle.
Stage Fit-Up Fixture	Provides the facilities required for proof loading and testing the compatibility of the mating surfaces and external connections of the handling and transportation equipment used in conjunction with the S-II stage.
Engine Actuator Lock No. 1	Used to immobilize the engine actuator in its neutral position during ground handling and maintenance.
Engine Actuator Simulator	Strut used to replace engine actuator when the actuator is removed for maintenance.
Support Ring Segment Sling	Used to hoist and maneuver support ring segments during assembly or disassembly of the forward and aft stage support rings.
Ullage Rocket Sling	Used to hoist and install ullage rockets.
Stage Checkout Dolly Spacer	Supports the S-II stage on the stage checkout dolly during checkout of the stage.
Static Firing Skirt	Used in conjunction with the aft stage support ring to provide support for the aft end of the stage during transportation and handling operations. In addition, the skirt provides support for the stage during static firing.
Stage Erecting Sling	Attaches to the forward hoisting frame to provide a means for hoisting the S-II stage.
Forward Hoisting Frame	Distributes stage hoisting loads to the forward stage support ring.

Table 24-7. Transportation, Protection, and Handling Equipment, S-II (Cont'd)

Equipment	Function
Aft Hoisting Frame	Used for horizontal hoisting and when rotating from the horizontal to vertical or vertical to horizontal position.
Aft Interstage Dolly	Provides support and mobility for the aft interstage when in a vertical attitude.
Engine Component Manipulator	Used to handle engine and engine compartment components to facilitate installation and removal.
Engine Protective Frame	Used in conjunction with the aft stage cover to provide environmental and physical protection for the S-II stage engine compartment during handling and transportation.
Main Bus Battery Holder	Used in conjunction with the engine component manipulator to install and remove the main bus battery.
Engine Protective Frame Attitude Control Sling	Supports the engine protective frame during hoisting operations for installation on, or removal from the S-II stage.
Engine Protective Frame Segment Sling	Supports the engine protective frame segments during hoisting operations.
Interstage and Static Firing Skirt Segment Sling	Used to handle segments of the interstage and static-firing skirt.
Forward Hoisting Frame Holding Fixture	Supports the forward hoisting frame during assembly and disassembly.

Table 24-7. Transportation, Protection, and Handling Equipment, S-II (Cont'd)

Equipment	Function
Interstage Forward Support Ring	Provides a supporting structure to enable hoisting of the forward end of the S-II interstage.
Interstage Aft Support Ring	Provides a supporting structure for the aft end of the S-II interstage to support and maintain concentricity of the aft interstage during handling and checkout operations.
Engine Actuator Lock No. 2	Immobilizes the production engine actuator to prevent any relative motion between the actuator body and the rod during ground handling and maintenance.
Stage Storage Support Forward Stand	Provides support for the forward end of the S-II-F stage when storing in a horizontal position.
Stage Storage Support Aft Stand	Provides support for the aft end of the S-II-F stage when storing in a horizontal position.
Aft Hoisting Frame Access Ladder Sling	Supports the aft hoisting frame access ladder during hoisting operations.
Stage Guide Bracket Set	Provides positive control and guidance of the S-II stage during the final phases of lowering the stage on to the test stand.
Tag Lines Adapter Set	Provides attachment for tag lines which are used to guide the stage during erection.
Transporter Component Sling	Supports transporter components during hoisting operations for assembly and disassembly of the transporter.
Dock Loading Ramp; Dock Loading Ramp Sling	Used only at Port Hueneme for common bulkhead test speciment, fit-up fixture, and all systems test vehicle.
LH_2 Inlet Duct Handler; Inlet Duct Handler; Gas Spin Bottle Vertical Installer; LO_2 Feed System Handler; LH_2 Feed System Handler Exhaust System Handler; Sequence Control Package Handler; LO_2 Heat Exchanger Handler	Used in conjunction with the engine component manipulator to handle, remove, and install the various engine components as indicated in the title of the particular handler.

Table 24-8. Servicing Equipment, S-II

Equipment	Function
Pneumatic Servicing Console Set	Used to control the flow of pneumatic fluids to the stage for pressurization and purging purposes when the stage is being prepared for static firing.
Hydraulic Fluid Servicing Unit	Pre-filters hydraulic fluid prior to filling the hydraulic power console.
Hydraulic System Jumper Unit	Permits recirculation of hydraulic fluid through the hydraulic fluid servicing unit and fluid distribution system interconnecting the supply and return line.
Portable Vacuum Pump Unit	For re-evacuating vacuum-jacketed lines of fluid distribution systems and stage propellant feed lines.
Stage Area Fluid Distribution System	Used with pneumatic checkout console, and hydraulic power console to supply, distribute, and control the fluids required for checkout of the S-II stage.
Acceptance Stand No. 1 Fluid Distribution System	Used with swing arm pneumatic console, pneumatic checkout console, and hydraulic power console, to supply, distribute and control the fluids required for S-II stage checkout and static firing.
Acceptance Stand No. 2 Fluid Distribution System	Used with swing arm pneumatic console, pneumatic checkout console, hydraulic power console, and instrument unit air servicing unit to supply, distribute, and control the fluids required for checkout of the S-II stage.

Table 24-8. Servicing Equipment, S-II (Cont'd)

Equipment	Function
Electrical Container Air Servicing Unit	Consists of manual control blower and filter unit, used to supply continuous filtered air to stage disconnect.
Electrical Container GN_2 Thermal Control Unit	Controls, regulates, and heats GN_2 during and prior to propellant loading operation on static firing site.
Engine Compartment Environmental Control Unit	Supplies warm GN_2 to the S-II stage engine compartment purge system for temperature control and inerting purposes.
Hydraulic Accumulator Precharge Servicing Unit	Used to precharge the S-II stage accumulator-reservoir to the required pressure for hydraulic operations.

Table 24-9. Auxiliary Equipment, S-II

Equipment	Function
Umbilical Disconnect Carrier Plate Assembly Arm No. 3A	Provides mounting facilities for all GSE/stage disconnects.
Transport Instrumentation Unit	Used to monitor and record environmental conditions and acceleration loads on the S-II stage during the different modes of transportation.
Vertical Engine Compartment Platform	Provides access to engine compartment and serves as a drop screen to prevent damage to engines when the stage is in the vertical position.
Thrust Alignment Set	Used to verify the alignment of individual engine assemblies during engine installation.
LH_2 Container Servicing Mechanism	Used for "in tank" instrumentation, installation, and inspection.

Table 24-9. Auxiliary Equipment, S-II (Cont'd)

Equipment	Function
Forward Skirt Maintenance Walkway	Used during maintenance operations on the S-II and S-IVB stages.
LH_2 Container Servicing Clean Room	a. Provides uncontaminated areas for personnel, equipment, and LH_2 container during maintenance. b. Provides a clean atmosphere over the open LH_2 container and access into the container for servicing equipment and personnel.
Forward Stage Access Platform	Provides access to the forward hook, for attaching or detaching the hook to or from the stage erecting sling.
Static Firing Fragmentation Shield	Used to protect the equipment in the S-II stage engine compartment in the event of a J-2 engine explosion.
LH_2 Container Servicing Air Conditioner	Provides a clean atmosphere and continuous air purge during interior servicing of LH_2 container.
Engine Compartment Light Set	Provides floodlighting in the engine compartment to facilitate maintenance.
Auxiliary Engine Alignment Fixture	Used in the placement and alignment of the ullage engines relative to the S-II stage centerline.
Piston Position Indicator No. 1	Indicates the position of the piston in the preproduction hydraulic servo actuator or the angular position of a J-2 engine in terms of the actuator stroke.
Static Firing Heat Shield	Protects stage structure and equipment in the S-II stage engine section.
Piston Position Indicator No. 2	Indicates the position of the piston in the production hydraulic servo actuator or the angular position of a J-2 engine in terms of the actuator stroke.

Table 24-9. Auxiliary Equipment, S-II (Cont'd)

Equipment	Function
Hydraulic System Installation Fixture	Used to install or remove an intact hydraulic actuation system.
Hydraulic System Transportation Fixture	Used in the buildup and transportation of the hydraulic system.
Aft Hoisting Frame Access Ladder	Provides access to the centerlink of the aft hoisting frame during installation or removal when the S-II stage is in a horizontal position.
Pump Shaft Seal Visual Leakage Indicator	Used to monitor the main and auxiliary hydraulic pump main shaft leakage except during static firing and launching. (Consists of graduated plastic bottles.)
Umbilical Disconnect Carrier Plat Assembly Arm No. 4	Provides mounting facilities for all GSE stage disconnects.
Electrical Connector Cover Set	Used to seal umbilicals and open electrical sockets from dust and dirt.
Fluid Connector Cover Set	Used to seal all open fluid lines and propellant lines from dust and dirt.
Warning Streamer Set	Warning streamers marked "remove before flight," used during transportation and handling of stage.

24-5. GROUND SUPPORT EQUIPMENT, S-IVB.

The ground support equipment for the S-IVB stage of Saturn V is similar to that of the S-IVB stage of Saturn IB (Refer to Paragraph 17-4.)

CHAPTER 4

SECTION XXV

STAGE CONFIGURATIONS, SATURN V

LIST OF ILLUSTRATIONS

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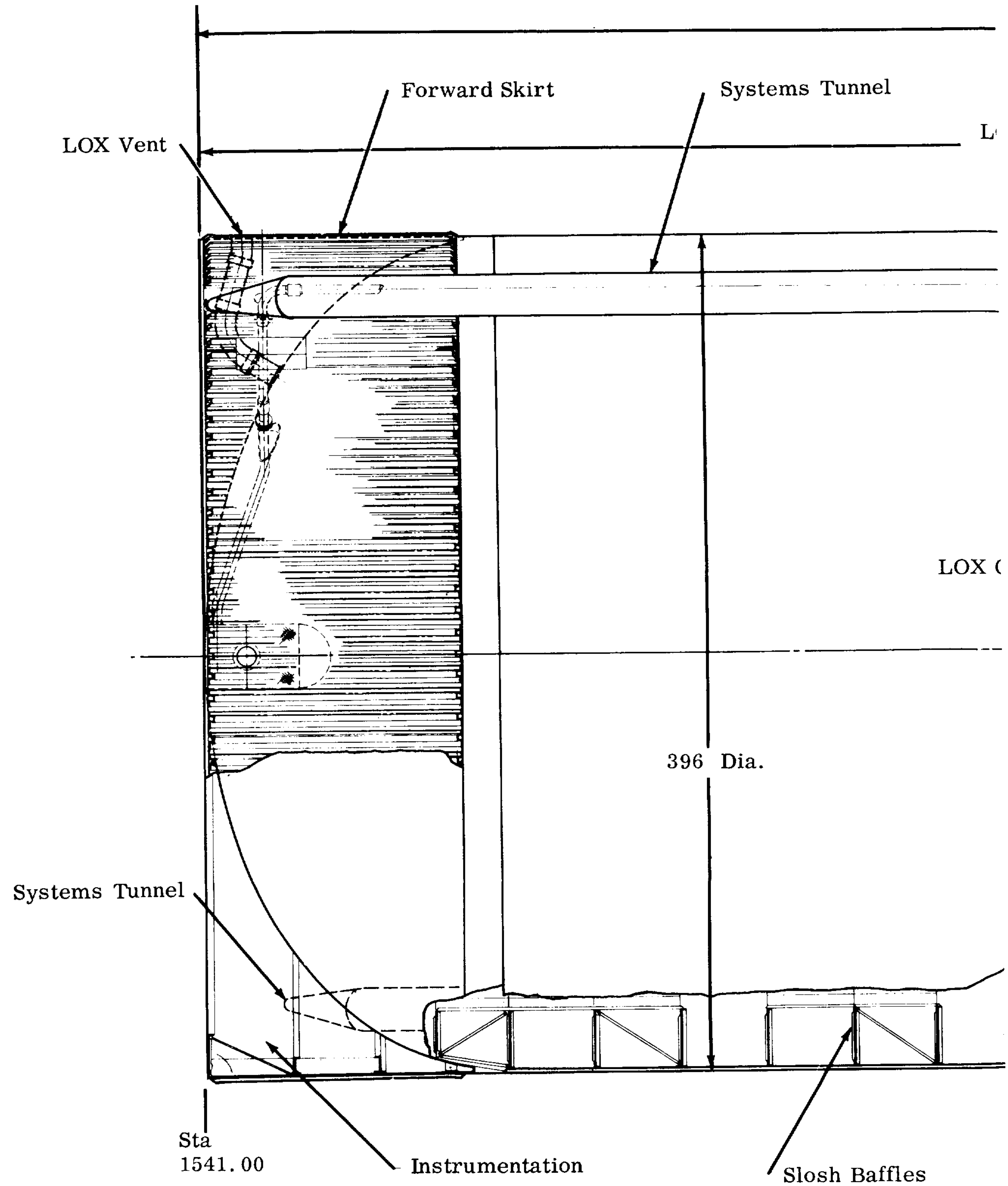

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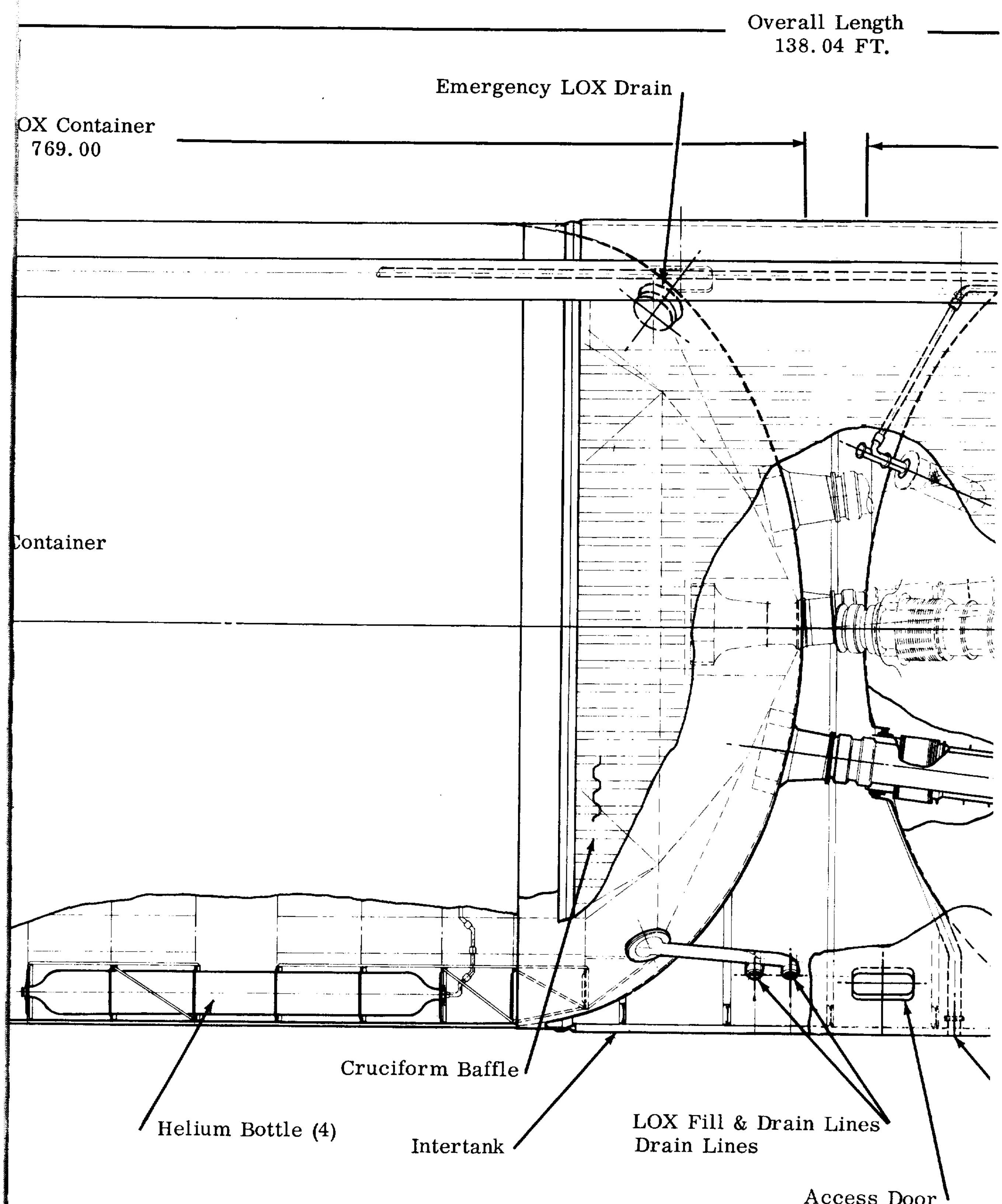

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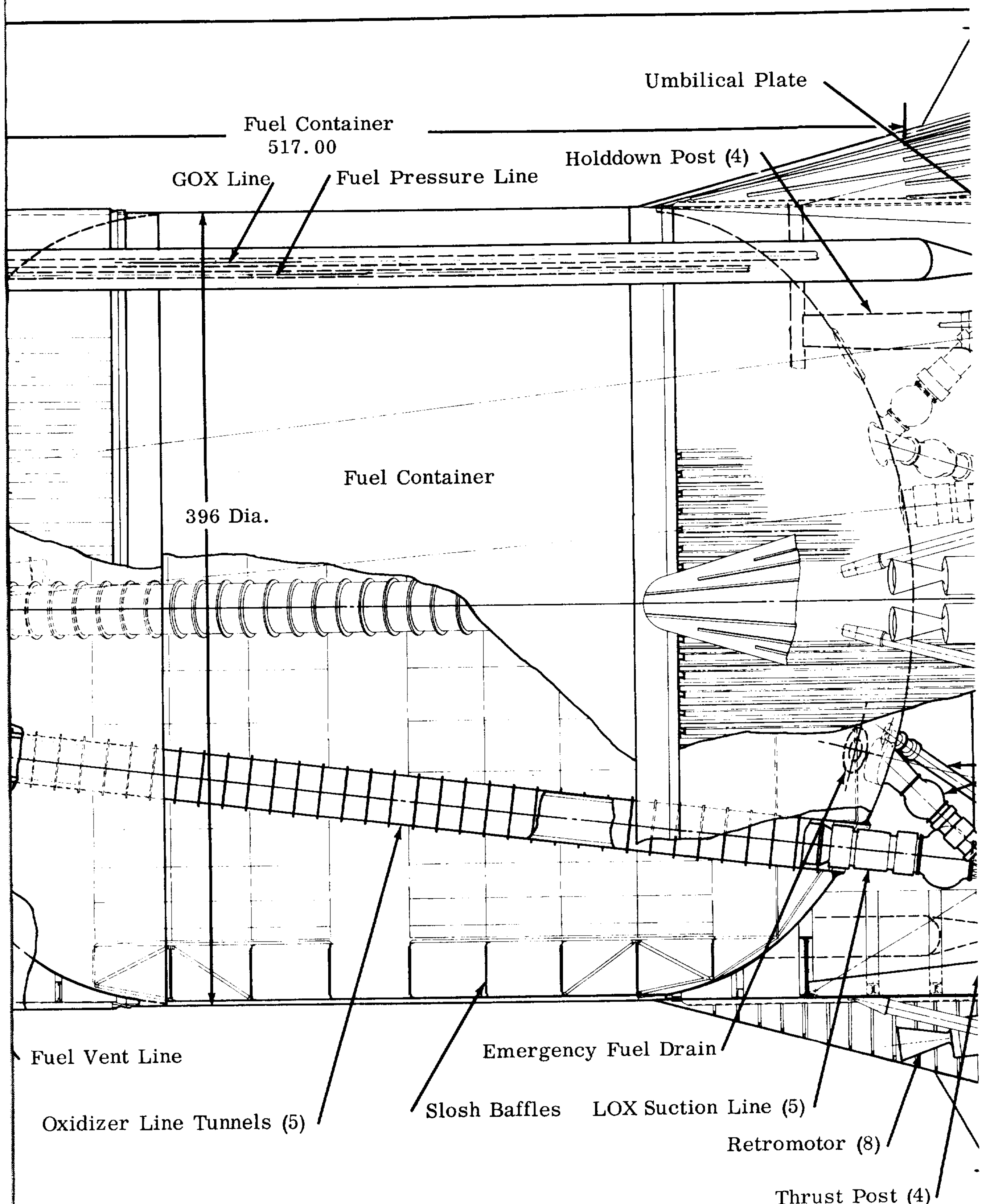

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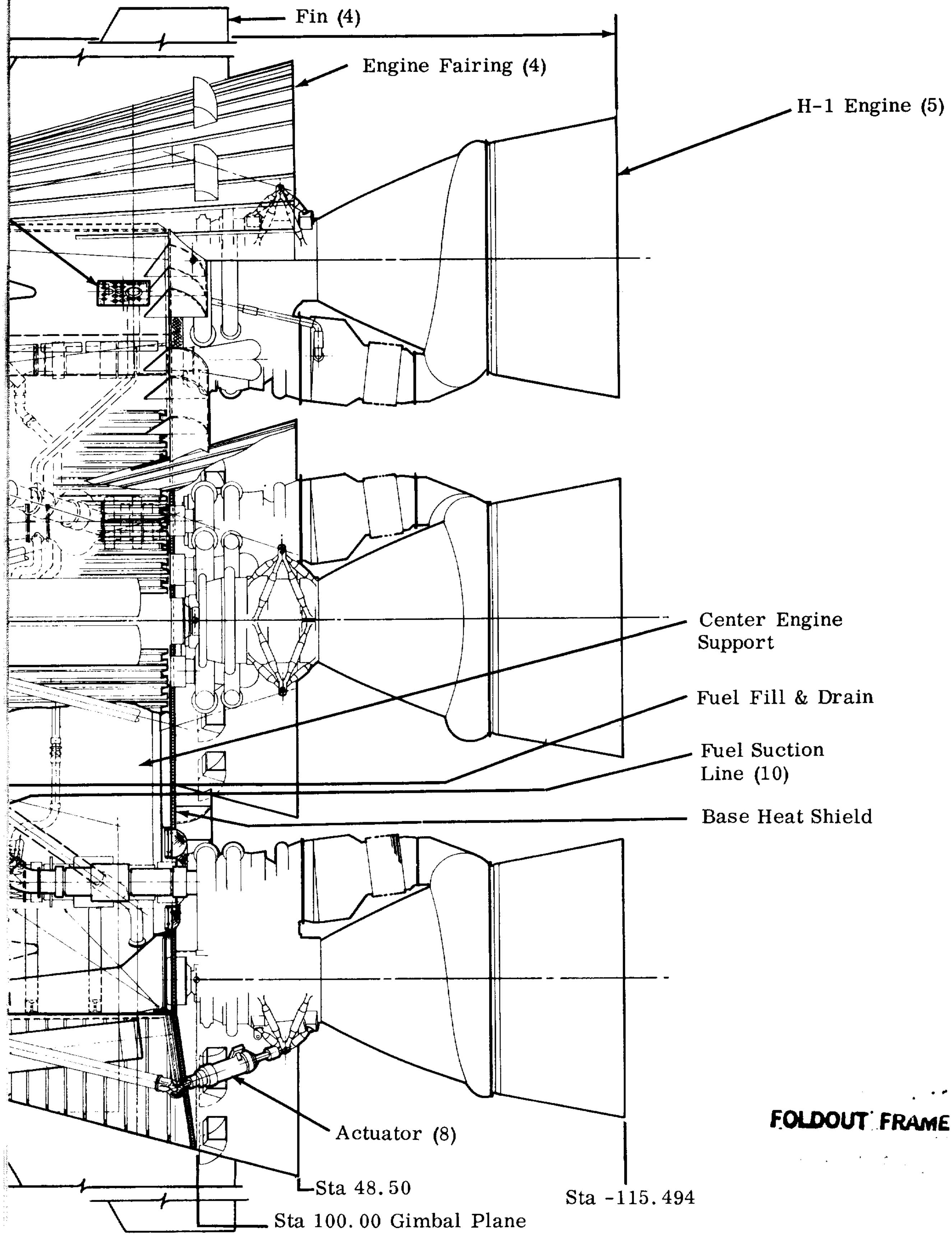

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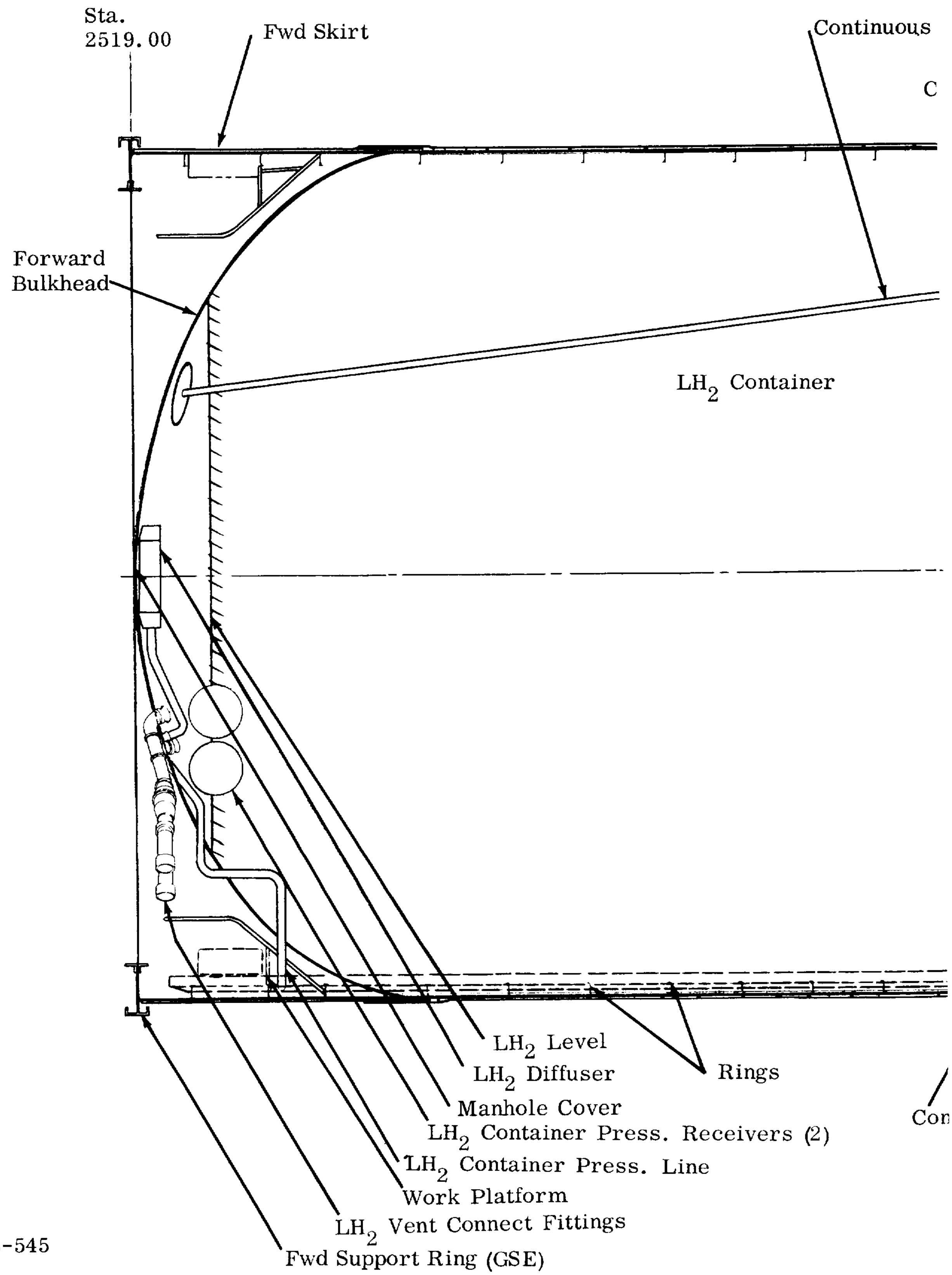
Figure 25-1. S-IC Inboard Profile



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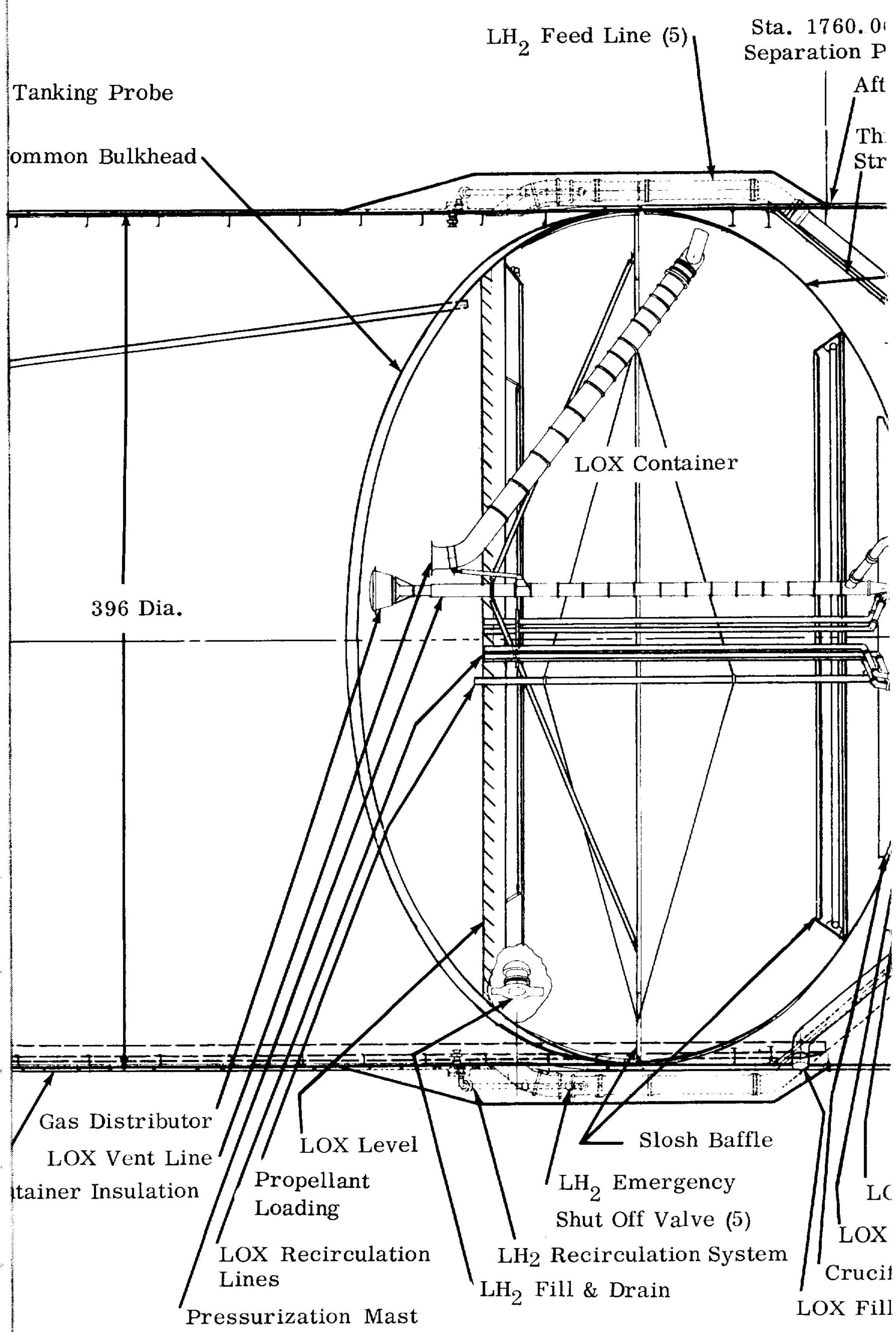

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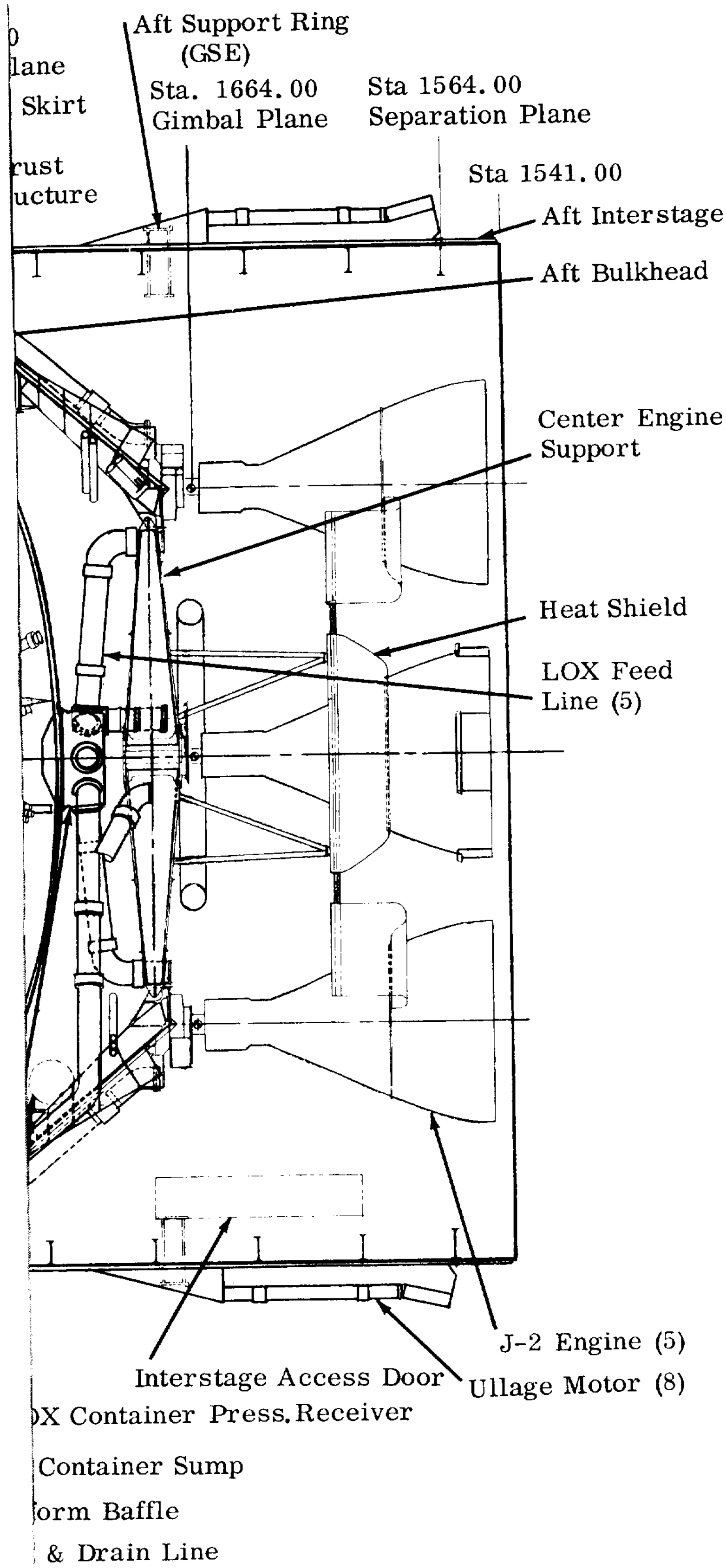

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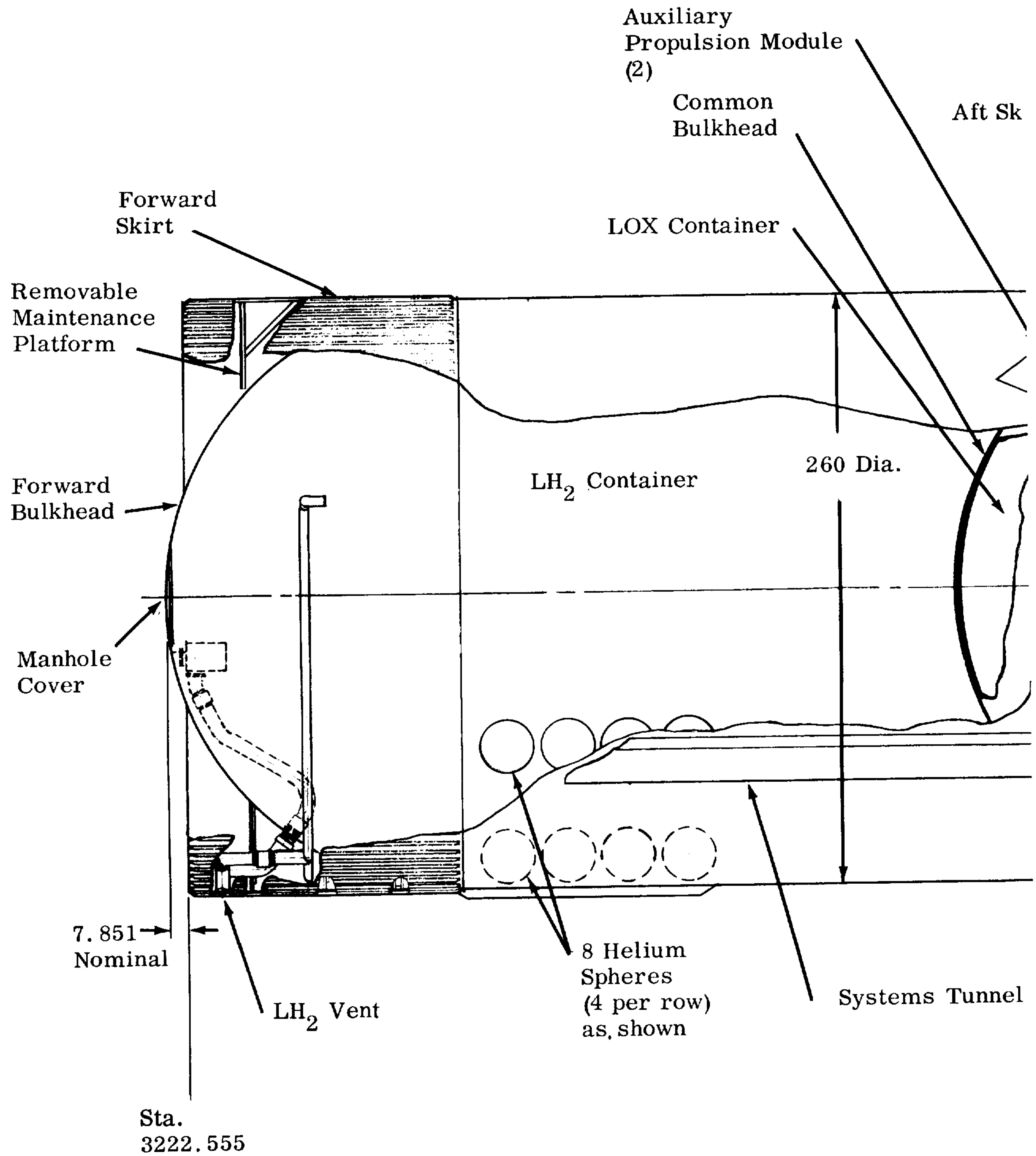
Figure 25-2. S-II Inboard Profile



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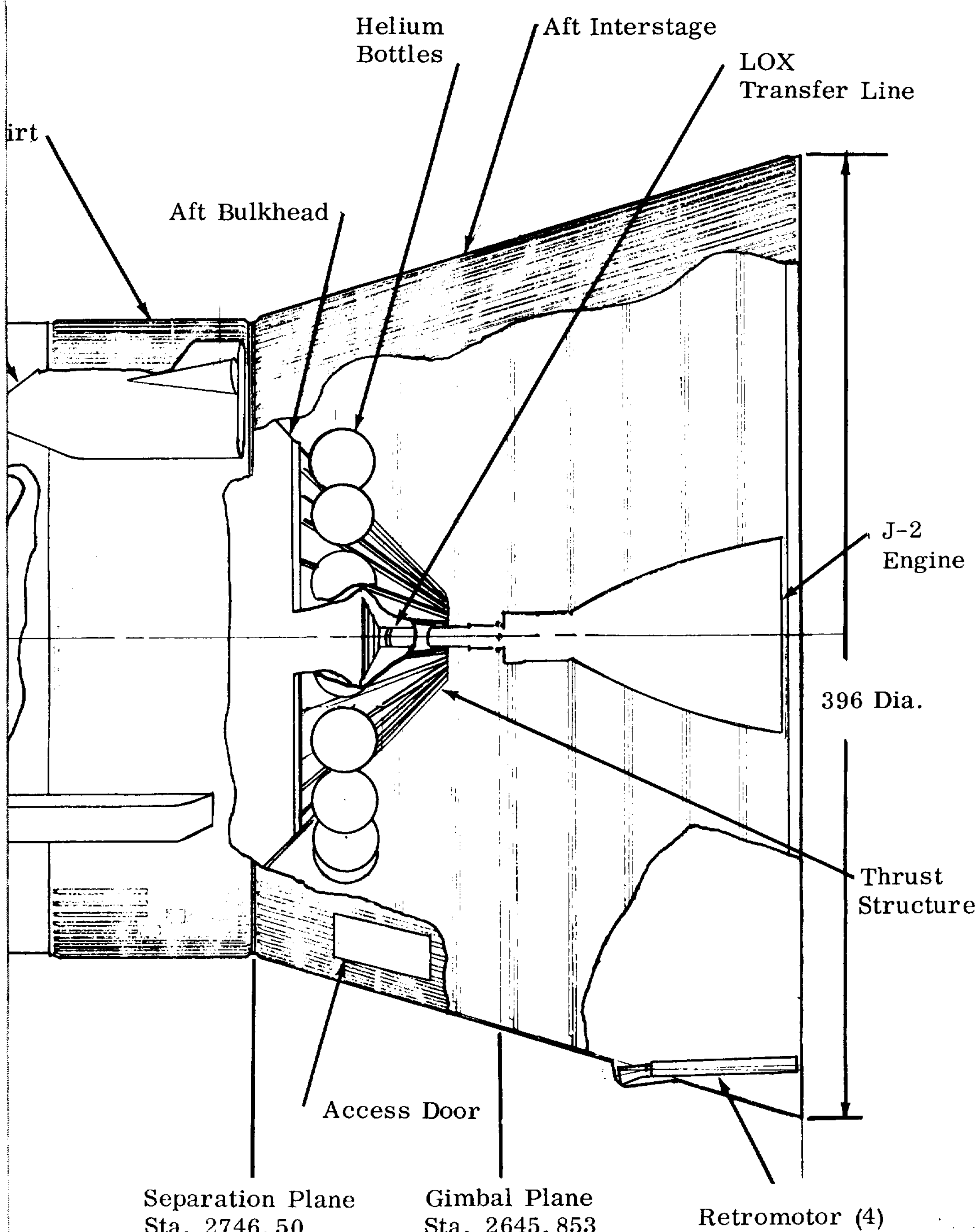

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Figure 25-3. S-IVB Inboard Profile, Saturn V

CHAPTER 5

SECTION XXVI

INTRODUCTION

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CHAPTER 5.
LAUNCH VEHICLE FACILITIES

This chapter contains a general description of the facilities involved with the design, development, and test of the Saturn I, Saturn IB, and Saturn V launch vehicles. Section XXVI contains an introduction to the facilities and Section XXVII relates the details of each facility. Section XXVIII contains a description of the logistics of the Saturn program.

SECTION XXVI.
INTRODUCTION

The launch vehicle facilities include both governmental and industrial facilities. The governmental facilities are located at the Marshall Space Flight Center at Huntsville, Alabama; the Michoud Operations, New Orleans, Louisiana; and the Mississippi Test Facility on the Pearl River in Hancock County, Mississippi. (The facilities of the Launch Operations Center at the Atlantic Missile Range, Cape Kennedy, Florida, are described in Volume III of the Apollo Systems Description.) The industrial facilities include the Douglas Aircraft Company plants at Santa Monica, Huntington Beach, and Sacramento, California; the North American Aviation plants at Tulsa, Oklahoma, and at Seal Beach, and Downey, California; the Rocketdyne Division, North American Aviation plants at Canoga Park and Santa Susana, California; and Pratt & Whitney Aircraft Company, West Palm Beach, Florida. The Boeing Company and the Chrysler Corporation perform manufacturing and testing operations at the Michoud Operations and the Mississippi Test Facility.

CHAPTER 5

SECTION XXVII

FACILITIES

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SECTION XXVII.
FACILITIES

27-1. MARSHALL SPACE FLIGHT CENTER.

The Marshall Space Flight Center is responsible for the design of the Saturn I, IB and V launch vehicles, and the fabrication of the first stage for the three vehicles. The first four Saturn I Block II, S-I stages, the S-IB dynamic test stage and facilities checkout stage, the S-IC dynamic test stage, systems checkout stage, static checkout stage, and the first flight stage are built and tested at MSFC. Instrument unit design, manufacture, and tests are also conducted at MSFC.

Test facilities at MSFC include a vertical structural test facility, component test facility, dynamic test facility, components and subassembly acceptance facility, load test annex, S-IC static test stand and blockhouse, F-1 and J-2 engine systems development test stands, S-I and S-IB static test stand and blockhouse, and other subassembly test facilities.

27-2. MICHOUD OPERATIONS.

The Michoud Operations are used by the Boeing Company for manufacturing and test operations of the S-IC facilities checkout stage and subsequent flight stages, and by the Chrysler Corporation for Chrysler manufactured S-I and S-IB stages.

Facilities at the Michoud Operations include a vertical assembly building, hydrostatic test and cleaning facility, Saturn dock, and other support complexes.

27-3. MISSISSIPPI TEST FACILITY.

The Mississippi Test Facility (MTF) provides acceptance test complexes for the S-IC and S-II stages.

The facilities at MTF include two stage-acceptance test stands, an instrumentation and control center, propellant ready storage and handling facility, high-pressure gas batteries, and test support buildings.

27-4. DOUGLAS AIRCRAFT COMPANY.

The Douglas Aircraft Company maintains facilities for research and development, qualification, production, and testing of the S-IV and S-IVB stages.

The facilities include research, development, assembly, manufacturing, and components test stands at Santa Monica, California; final assembly and checkout facilities at Huntington Beach, California; battleship testing, all-systems testing, attitude control motor tests, and static test facilities at Sacramento, California.

27-5. SPACE AND INFORMATION DIVISION, NORTH AMERICAN AVIATION.

The North American Aviation Space and Information Systems Division maintains facilities for research and development, qualification, production and testing of the S-II stage.

The facilities include research, development, cryogenic test, antenna test, processing, assembly, and electromechanical mock-up facilities at Downey, California; bulkhead fabrication, vertical assembly, hydro-static test, water conditioning, and final assembly at Seal Beach, California; battleship testing and all-systems testing at Santa Susana, California; and detail and subassembly operations at Tulsa, Oklahoma.

27-6. ROCKETDYNE DIVISION, NORTH AMERICAN AVIATION.

The Rocketdyne Division maintains manufacturing and test facilities at Canoga Park and Santa Susana, California for the development and production of the H-1 and F-1 rocket engines.

27-7. PRATT & WHITNEY AIRCRAFT COMPANY.

The Pratt & Whitney Aircraft Company maintains manufacturing and test facilities at West Palm Beach, Florida for the development and production of the RL10A-3 rocket engine.

CHAPTER 5

SECTION XXVIII

LOGISTICS

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SECTION XXVIII.
LOGISTICS

The logistics for the various stages of the Saturn launch vehicles is an important part of the program. The large size of the individual stages and the long distances between fabrication, test, and launch facilities demand thorough planning in order to keep delays due to handling and transportation to a minimum.

The first four S-I Block II stages will be manufactured and tested at MSFC. Upon completion of testing the stages will be transported by barge to the Cape Kennedy launch facility. Subsequent S-I stages will be manufactured at Michoud, Louisiana and at completion of testing, will be transported by barge to the Cape.

The S-IV stage will be manufactured and tested in California and shipped by freighter or special aircraft to Cape Kennedy.

The S-IB dynamic test stage and the facilities checkout stage will be manufactured and tested at MSFC. The facilities checkout stage will also be utilized at the Mississippi Test Facility and at the launch facility. Subsequent S-IB stages will be manufactured at Michoud, Louisiana, tested at Mississippi Test Facility, and shipped by barge to Cape Kennedy.

The S-IVB stage will be manufactured and tested in California and shipped by freighter or special aircraft to Cape Kennedy.

The S-IC dynamic test stage, systems checkout stage, static test stage, and the first flight stage will be manufactured and tested at MSFC. The S-IC facilities checkout stage and subsequent flight stages will be manufactured at Michoud, Louisiana, tested at Mississippi Test Facility, and shipped by barge to the Cape.

The S-II stage will be manufactured in California and transported by special ship to the Mississippi Test Facilitiy for test preparation and static firing. Upon completion of the test phase the S-II stage will be transported by ship to Cape Kennedy.

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