NASA CR71508

VOYAGER SPACECRAFT SYSTEM

FINAL TECHNICAL REPORT

VOLUME D

DESIGN FOR 1969 TEST SPACECRAFT

prepared for

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

UNDER
CONTRACT NO. 951111
JULY 1965

THE BOEING COMPANY • AERO-SPACE DIVISION • SEATTLE, WASHINGTON

THE BOEING COMPANY

SEATTLE, WASHINGTON 98124

LYSLE A. WOOD
VICE PRESIDENT-GENERAL MANAGER
AERO-SPACE DIVISION

July 29, 1965

Jet Propulsion Laboratory
California Institute of Technology
4800 Oak Grove Drive
Pasadena, California

Gentlemen:

This technical report culminates nearly three years of Mariner/Voyager studies at Boeing. During this time, we have gained an appreciation of the magnitude of the task, and feel confident that the experience, resources and dedication of The Boeing Voyager Team can adequately meet the challenge.

The Voyager management task is accentuated by three prime requirements: An inflexible schedule of launch opportunities; the need for an information-retrieval system capable of reliable high-traffic transmission over interplanetary distances; and a spacecraft design flexible enough to accommodate a number of different mission requirements. We believe the technical approach presented here satisfies these design requirements, and that management techniques developed by Boeing for space programs will assure delivery of operable systems at each critical launch date.

Mr. E. G. Czarnecki has been assigned program management responsibility. His group will be ably assisted by Electro-Optical Systems in the area of spacecraft power, Philco Western Development Laboratories will be responsible for telecommunications, and the Autonetics Division, North American Aviation will provide the auto-pilot and attitude reference system. This team has already demonstrated an excellent working relationship during the execution of the Phase IA contract, and will have my full confidence and support during subsequent phases.

This program will report directly to George H. Stoner, Vice President and Assistant Division Manager for Launch and Space Systems. Mr. Stoner has the authority to assign the resources necessary to meet the objectives as specified by JPL.

The Voyager Spacecraft System represents to us more than a business opportunity or a new product objective. We view it as a chance to extend scientific knowledge of the universe while simultaneously contributing to national prestige and we naturally look forward to the opportunity of sharing in this adventure.

Lysle A. Wood

Lysle A. Wood

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INTRODUCTION

In fulfillment of the Jet Propulsion Laboratory (JPL) Contract 951111, the Aero-Space Division of the Boeing Company submits the Voyager Spacecraft Final Technical Report. The complete report, responsive to the documentation requirements specified in the Statement of Work, consists of the five following documents:

VOLUME	TITLE	BOEING DOCUMENT NUMBER
A	Preferred Design Flight Spacecraft and Hardware Subsystems	D2-82709-1
	Part I	
	Section 1.0 Voyager 1971 Mission Objectives and Design Criteria	
	Section 2.0 Design Characteristics and Restraints	
	Section 3.0 System Level Functional Descriptions of Flight Spacecraft	
	Part II	
	Section 4.0 Functional Description for Spacecraft Hardware Subsystems	
	Part III	
	Section 5.0 Schedule and Implementation Plan	
	Section 6.0 System Reliability Summary	
	Section 7.0 Integrated Test Plan Development	
B	Alternate Designs Considered-Flight Spacecraft and Hardware Subsystems	D2-82709-2
C	Design for Operational Support Equipment	D2-82709-3
D	Design for 1969 Test Spacecraft	D2-82709-4
E	Design for Operational Support Equipment for 1969 Test Flight Spacecraft	D2-82709-5

For convenience the highlights of the above documentation have been summarized to give an overview of the scope and depth of the technical effort and management implementation plans produced during Phase IA. This summary is contained in Volume O, Program Highlights and Management Philosophy, D2-82709-O. A number of supporting documents are provided to furnish detailed information developed through the course of the contract and to provide substantiating reference material which would not otherwise be readily available to JPL personnel. Additionally, a full scale mock-up of the preferred design spacecraft has been assembled. This mock-up, shown in Figure -1, has been delivered to JPL. The mock-up has been provided with the view that it would be of value to JPL in subsequent Voyager Spacecraft System planning. Mr. William M. Allen, President of The Boeing Company, Mr. Lysle A. Wood, Vice-President and AeroSpace Division General Manager, Mr. George H. Stoner, Vice-President and Assistant Division Manager responsible for Launch and Space Systems activities, and Mr. Edwin G. Czarnecki, Voyager Program Manager, are shown with the mockup.

During the three month period covered by Contract 951111, Boeing has:

1) Performed system analysis and trade studies necessary to achieve an optimum or preferred design of the Flight Spacecraft.
2) Determined the requirements and constraints which are imposed upon the Flight Spacecraft by the 1971 mission and by the other systems and elements of the project, including the science payload.
3) Developed functional descriptions for the Flight Spacecraft and for each of its hardware subsystems, excluding the science payload.

Figure 1: Preferred Design Mockup

Left to Right:

William M. Allen
Edwin G. Czarnecki
Lysle A. Wood
George H. Stoner

4) Determined the requirements for the Flight Spacecraft associated Operational Support Equipment (OSE) necessary to accomplish the Voyager 1971 mission.

5) Developed a preliminary design of the OSE.

6) Developed functional descriptions for the OSE.

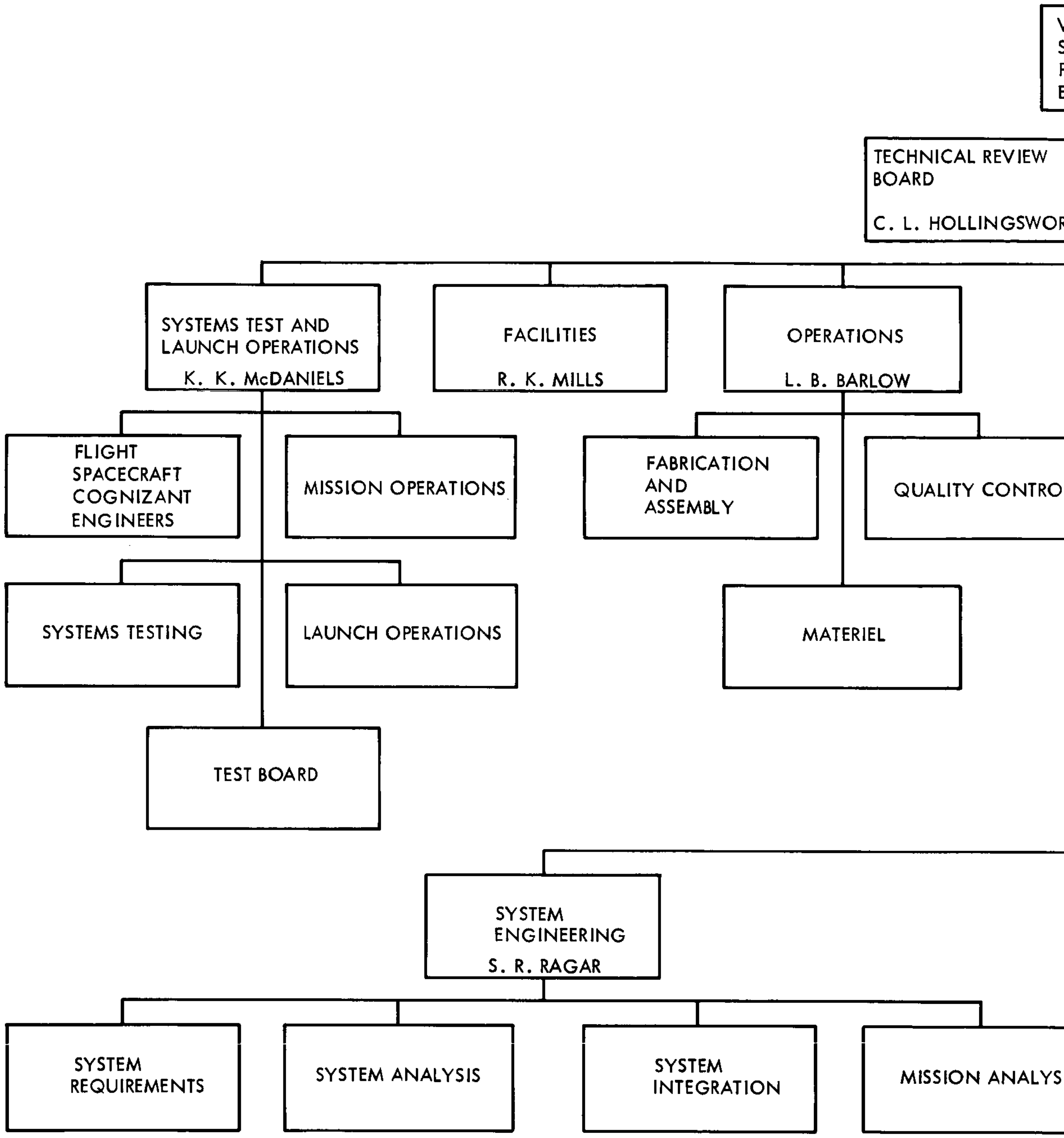
7) Determined the objectives of a 1969 test flight and the design of the 1969 Test Flight Spacecraft using the Atlas/Centaur Launch Vehicle. An alternate test flight program is presented which utilizes the Saturn IB/Centaur Launch Vehicle.

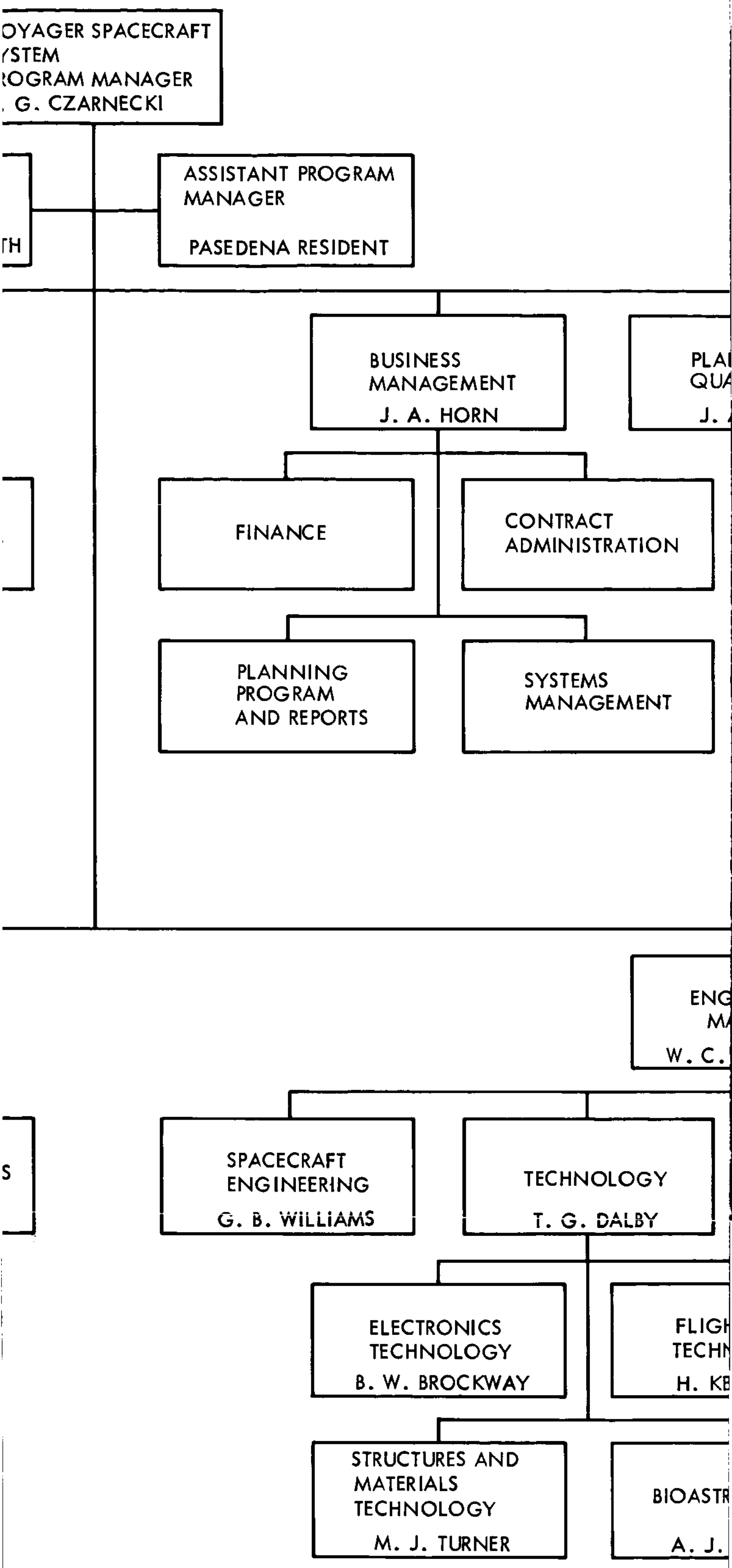
8) Developed functional descriptions for the Flight Spacecraft Bus, and its hardware subsystems, and OSE for the 1969 test spacecraft.

9) Updated and supplemented the Voyager Implementation Plan originally contained in the response to JPL Request for Proposal 3601.

The Voyager program management Team, shown in Figure -2 is under the direction of Mr. Edwin G. Czarnecki. Mr. Czarnecki is the single executive responsible to JPL and Boeing management for the accomplishment of the Voyager Spacecraft Phase IA, and will direct subsequent phases of the program. He reports directly to Mr. George H. Stoner who has the authority to commit those corporate resources necessary to fulfill JPL's Voyager Spacecraft System objectives.

Although Boeing has a technical management capability in all aspects of the Voyager Program, it is planned to extend this capability in depth through association with companies recognized as specialists in certain fields. Use of team members to strengthen Boeing's capability


TECHNICAL REVIEW BOARD
C. L. HOLLINGSWOR
SYSTEMS TEST AND LAUNCH OPERATIONS
K. K. McDANIELS
FACILITIES
R. K. MILLS
OPERATIONS
L. B. BARLOW
FLIGHT SPACECRAFT COGNIZANT ENGINEERS
MISSION OPERATIONS
FABRICATION AND ASSEMBLY
QUALITY CONTRO
SYSTEMS TESTING
LAUNCH OPERATIONS
MATERIEL
TEST BOARD
SYSTEM ENGINEERING
S. R. RAGAR
SYSTEM REQUIREMENTS
SYSTEM ANALYSIS
SYSTEM INTEGRATION
MISSION ANALYS


OYAGER SPACECRAFT
YSTEM
ROGRAM MANAGER
. G. CZARNECKI
ASSISTANT PROGRAM
MANAGER
PASEDENA RESIDENT
TH
BUSINESS
MANAGEMENT
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PROGRAM
AND REPORTS
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W. C.
S
SPACECRAFT
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TECHNOLOGY
T. G. DALBY
ELECTRONICS
TECHNOLOGY
B. W. BROCKWAY
FLIGH
TECHN
H. KE
STRUCTURES AND
MATERIALS
TECHNOLOGY
M. J. TURNER
BIOASTR
A. J.

Space

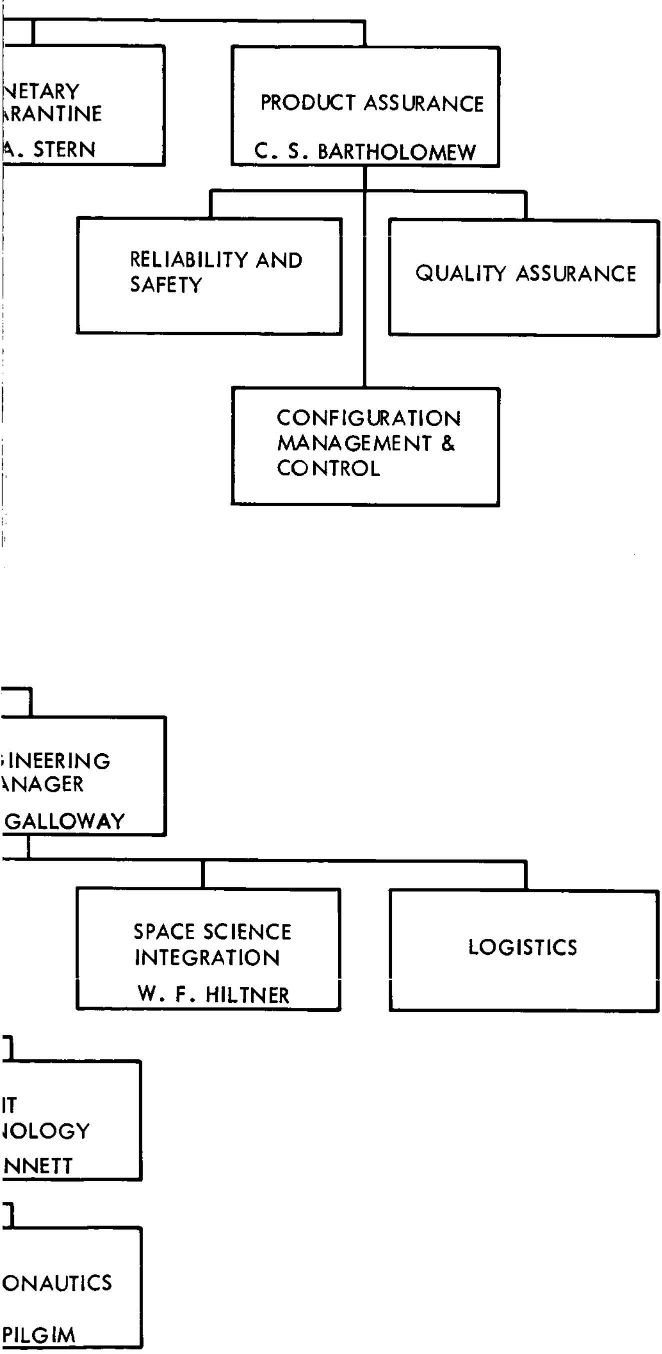
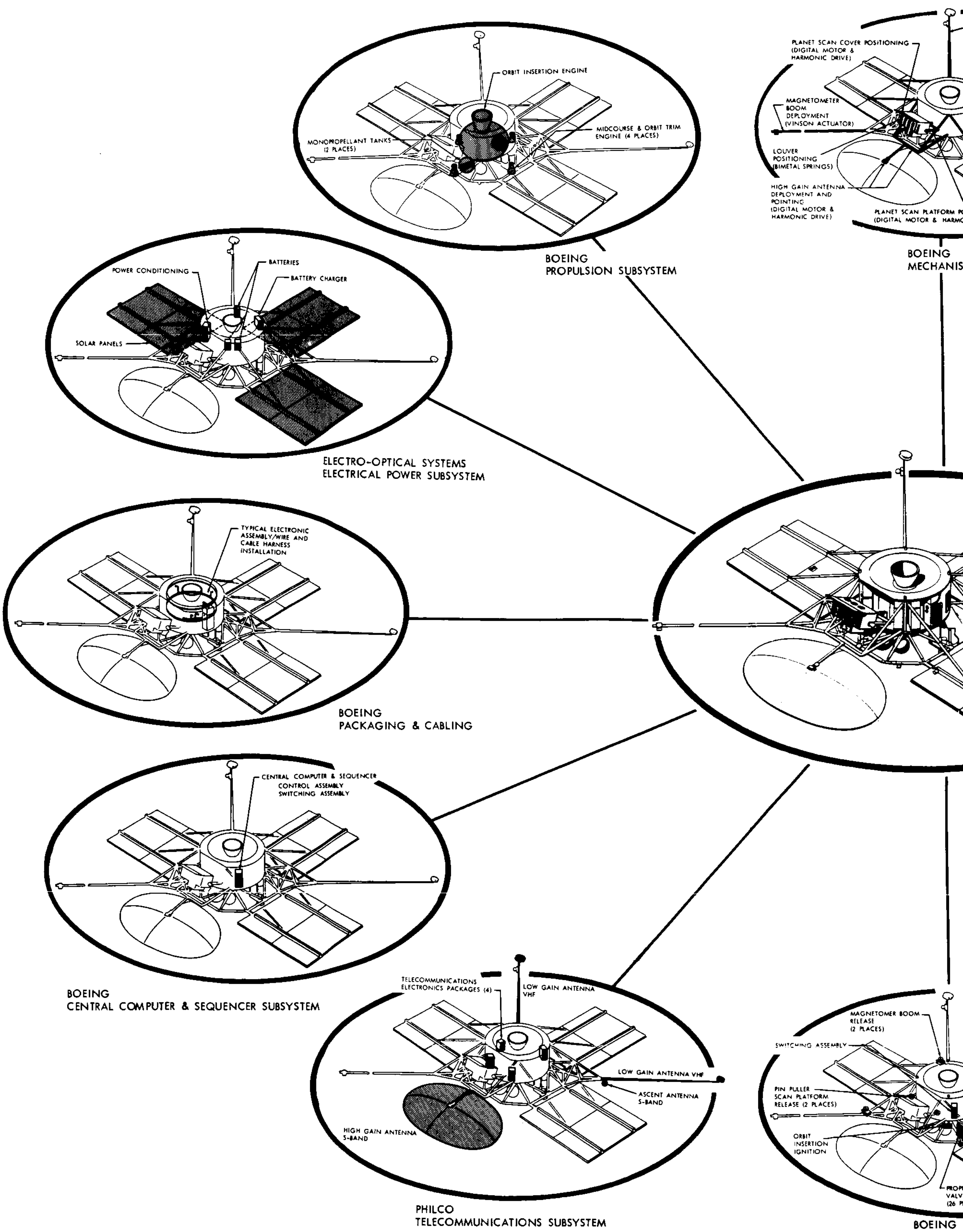


Figure 2 Boeing Voyager
craft Systems Management Structure

was considered early during pre-proposal activities. The basic concept was to add team members who would complement Boeing experience and capability, and significantly improve the amount and quality of technical and management activities. Based upon competitive considerations including experience and past performance and giving strongest emphasis to technical qualifications and management willingness to support the Voyager effort, Autonetics, Philco Western Development Laboratories, and Electro-Optics Systems were chosen as team members. This team arrangement, subject to JPL approval, is shown in Figure -3. The flight spacecraft design and integration task to be accomplished by this team is illustrated in Figure -4. Discussions leading to the formation of this team were initiated late in 1964, formal work statement agreements have been arrived at, and there has been a continuous and complete free exchange of information and documentation; permitting the Boeing team to satisfy JPL's requirements in depth and with confidence.

BOEING VOYAGER TEAM VOYAGER SPACECRAFT AND SPACE SCIENCES PAYLOAD INTEGRATION CONTRACTOR The Boeing Company Seattle, Washington Mr. E. G. Czarnecki - Program Manager		
SUBCONTRACTOR	SUBCONTRACTOR	SUBCONTRACTOR
Autonetics, North American Aviation Anaheim, California	Philco, Western Development Lab. Palo Alto, California	Electro-Optical Systems, Inc. Pasadena, California
o Autopilot and Attitude Refer- Subsystem	o Telecommunication Subsystem	o Electrical Power Subsystem
Mr. R. R. Mueller Program Manager	Mr. G. C. Moore Program Manager	Mr. C. I. Cummings Program Manager

FIGURE -3


ORBIT INSERTION ENGINE
MIDCOURSE & ORBIT TRIM ENGINE (4 PLACES)
MONOPROPELLANT TANKS (2 PLACES)
BOEING
PROPULSION SUBSYSTEM
PLANET SCAN COVER POSITIONING (DIGITAL MOTOR & HARMONIC DRIVE)
MAGNETOMETER BOOM DEPLOYMENT (VINSON ACTUATOR)
LOUVER POSITIONING (BIMETAL SPRINGS)
HIGH GAIN ANTENNA DEPLOYMENT AND POINTING (DIGITAL MOTOR & HARMONIC DRIVE)
BOEING
POWER CONDITIONING
BATTERIES
BATTERY CHARGER
SOLAR PANELS
ELECTRO-OPTICAL SYSTEMS
ELECTRICAL POWER SUBSYSTEM
TYPICAL ELECTRONIC ASSEMBLY/WIRE AND CABLE HARNESS INSTALLATION
BOEING
PACKAGING & CABLING
CENTRAL COMPUTER & SEQUENCER
CONTROL ASSEMBLY
SWITCHING ASSEMBLY
BOEING
CENTRAL COMPUTER & SEQUENCER SUBSYSTEM
TELECOMMUNICATIONS ELECTRONICS PACKAGES (4)
LOW GAIN ANTENNA VHF
LOW GAIN ANTENNA VHF
ASCENT ANTENNA S-BAND
HIGH GAIN ANTENNA S-BAND
PHILCO
TELECOMMUNICATIONS SUBSYSTEM
MAGNETOMER BOOM RELEASE (2 PLACES)
SWITCHING ASSEMBLY
PIN PULLER SCAN PLATFORM RELEASE (2 PLACES)
ORBIT INSERTION IGNITION
BOEING

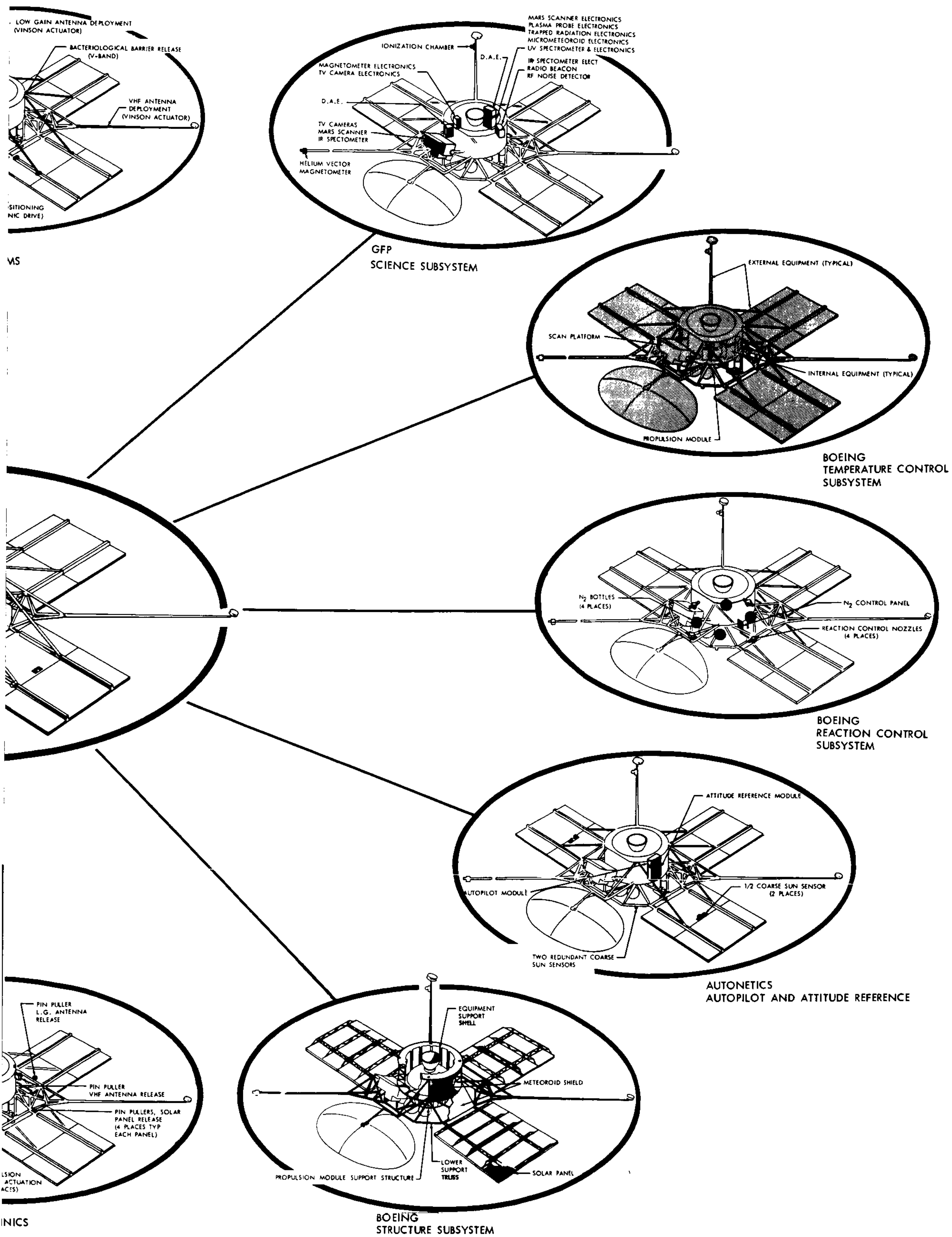


Figure 4: Voyager Flight Spacecraft Subsystem Integration

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The objective of the Voyager test program is to provide maximum probability of success of the 1971 mission. The 1969 Test Flight, as a part of the complete test program, can make a significant contribution toward meeting this objective. As in the ground test program described in Section 7, Volume A, the value of the 1969 Test Flight's contribution is determined by the extent to which the 1969 Test Flight elements, procedures, and flight profile correspond to those for the 1971 mission. The ideal 1969 Test Spacecraft, then, should be a duplicate of the 1971 spacecraft, utilize the 1971 OSE, and follow the 1971 flight profile. Since this reasoning indicates the desirability of using a Saturn IB/Centaur launch vehicle, this option was investigated and compared with the Atlas/Centaur-launched 1969 Test Flight required in the contract work statement.

This volume, D2-82709-4, Design for the 1969 Test Spacecraft, is in three parts:

1) Part I defines the major objectives of a 1969 Test Flight, compares several means by which it might be accomplished, and presents recommendations.
2) Part II describes a 1969 spacecraft which is boosted by an Atlas/Centaur. This spacecraft does not carry a Mars orbit insertion engine nor a Flight Capsule but otherwise is identical to the 1971 design except as noted below.
3) Part III describes a 1969 spacecraft which is boosted by a Saturn IB/Centaur. This spacecraft design is identical to the 1971 design.

Parts II and III primarily describe the differences between the 1969 Test Spacecraft and the 1971 spacecraft described in Volume A. Where the description is identical to that in Volume A, it is incorporated by reference rather than repetition.

The three parts are summarized below.

Part I--1969 Test Flight

The Boeing Company recommends that a 1969 Test Flight be included in the Voyager program and believes it will make a significant contribution to the success of the 1971 mission. The 1969 Test Flight, with either of the recommended test spacecraft, can be phased with the 1971 mission schedule so that test data is provided in time for corrective action to be taken in the 1971 spacecraft.

The 1969 Test Flight is a phase of a total test program, progressing from component testing to flight testing, with each phase contributing to the success of the 1971 mission. Its particular virtue is that it is an opportunity to bring all system elements together for the first time in an actual prelaunch and flight, and with mission operations environment. In addition to being a test of the spacecraft subsystems, singly and together, the 1969 Test Flight will test all interfaces such as operation procedures, personnel proficiency and the operational support equipment subsystems and their compatibility with each other and the spacecraft. Defining and solving potential problems in these test areas in 1969 will enhance the probability of mission success in 1971.

Although phased differently, the same ground testing is accomplished for the 1971 Voyager program with or without the 1969 Test Flight. The existence of a 1969 Test Flight will provide the discipline to assure timely testing early in the program. In addition the 1969 Test Flight will increase confidence that possible adverse interactions or multistress environmental effects encountered in actual mission flight either do not exist or can be corrected.

The recommended test spacecraft and flight profiles are described below in order of preference.

1) A test spacecraft that is a duplicate of the 1971 spacecraft, which is launched by a Saturn IB/Centaur, and which is placed in Mars orbit after separating a simulated Flight Capsule into a trajectory away from Mars.
2) A test spacecraft which is a minimum modification from the 1971 spacecraft, which is launched by an Atlas/Centaur, and which is placed on a Mars flyby trajectory.

These recommendations result from an evaluation of the relative contributions to the 1971 mission made by several alternatives using either launch vehicle.

Although the flight profiles recommended for each of the alternative test spacecraft are the most desirable in each case, significant data can still be obtained if the corresponding target launch dates are not met. If the 1969 Saturn/Centaur test were unable to meet the launch dates required for a Mars orbit, it would still be possible to launch a Mars flyby trajectory. Failing that, a heliocentric orbit which simulates a Mars flyby could be attempted. If the 1969 Atlas/Centaur test were unable to launch in time for a Mars flyby, it could be placed on a simulated Mars flyby trajectory.

Part II--1969 Test Spacecraft--Atlas/Centaur

The 1969 Atlas/Centaur Test Spacecraft is the 1971 spacecraft modified to meet the launch vehicle envelope and weight capability. The 1971 spacecraft subsystems are used except for the orbit insertion engine, the science payload, the VHF antenna and receiver, and the magnetometer. The size and arrangement of some subsystem elements have been changed but the 1971 subsystem schematic diagrams are still applicable. The major items in this latter category are:

1) The high-gain telecommunications antenna is folded differently and is an 8-foot circular dish in 1969, instead of the 8- by 12-foot 1971 antenna.
2) Three of the 1971 solar-cell panel sections are used, instead of six, and are folded differently.
3) Tankage for midcourse propulsion and attitude control is reduced in size. The number of tanks is the same.
4) Two, instead of three, 1971 battery sections are used.

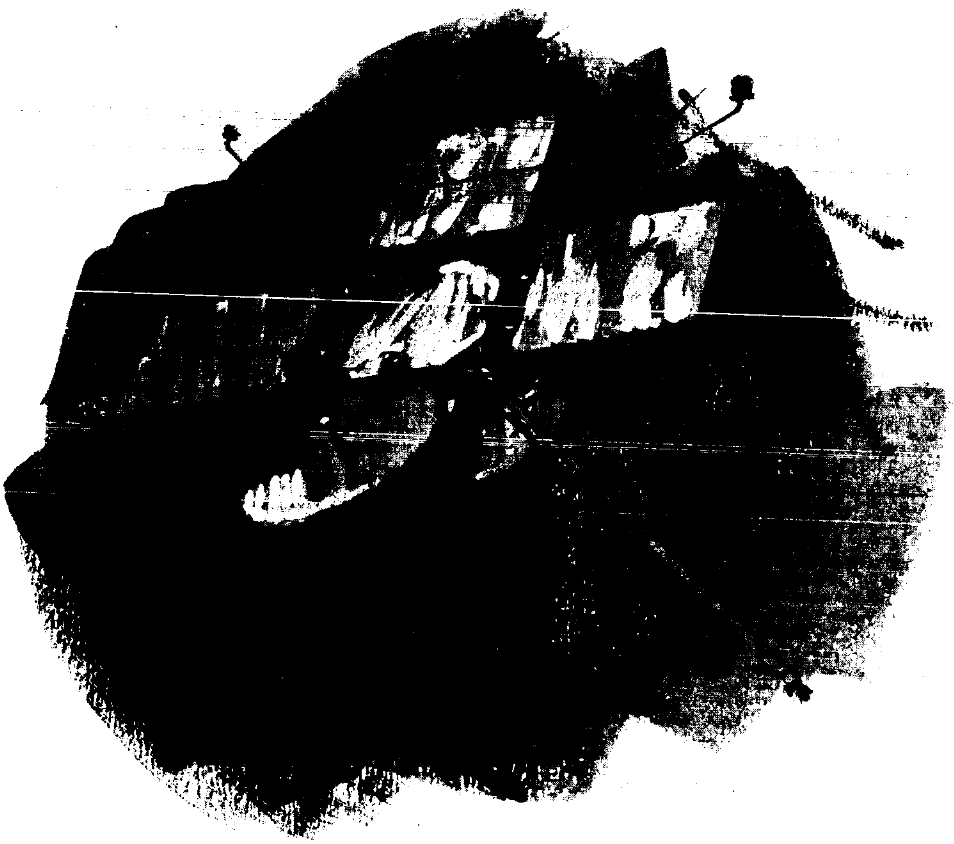
A 54-inch cylindrical extension is added to the basic Atlas/Centaur nose fairing described in the Voyager Mission Guidelines to accommodate the test spacecraft. The spacecraft weight placed on trans-Mars trajectory is well below the Atlas/Centaur capability for a 1969 Mars flyby.

Part III--1969 Test Spacecraft--Saturn IB/Centaur

The 1969 Saturn IB/Centaur Test Spacecraft is a duplicate of the 1971 spacecraft, except for increased propellant loading in the identical 1971 orbit insertion motor case as required to achieve desirable Mars orbit in 1969. A simulated science payload and a simulated Flight Capsule are used. It is recommended that the Flight Capsule contractor provide the simulated Flight Capsule.

Coordination with S-IB launch vehicle supplier indicates launch vehicle lead times would permit delivery to support 1969 launch if orders are received by early 1967.

The Saturn IB/Centaur has the capability to place the 1969 Planetary Vehicle on a trajectory towards Mars which will permit the test spacecraft, after separation of the simulated Flight Capsule in a direction away from Mars, to insert itself into a Mars orbit.



PART I - 1969 TEST FLIGHT

I-1.0 INTRODUCTION

The purpose of Part I of this three-part volume is to delineate the proper role of a 1969 Test Flight in the overall Voyager program. The discussion is divided into: development of a 1969 test philosophy; a definition and evaluation of candidate 1969 Test Flights, a discussion of launch vehicle availability and 1969 Test Flight decision options; and a summary of a total test plan incorporating the 1969 Test Flight as an integral part. Part I discusses these points in order.

In support of the general objective of increasing the probability of success of the 1971 mission, more specific objectives of the 1969 Test Flight are:

1) Serve as an integral link between the ground system integration tests and the 1971 mission;
2) Serve as a correlation checkpoint with ground life tests;
3) Integrate and qualify subsystems into spacecraft, and spacecraft into Planetary Vehicle, in a space environment;
4) Integrate the spacecraft with the OSE and MOS, and prove satisfactory interaction;
5) Verify design criteria used in establishing design specifications;
6) Demonstrate operations procedures and personnel proficiency under realistic mission conditions;
7) Uncover unforeseen problems in time to take corrective action for 1971 Voyager launch.

To meet its objectives, the spacecraft and ground hardware elements should be flight-qualified, and should have completed as much of the formal 1971 ground testing as practical prior to 1969 flight. The information determined from the test flight will influence both the design and the ground test program for 1971.

I-2.0 TEST PHILOSOPHY

The objective of a test program is to verify the design to specified requirements and to uncover and correct deficiencies in a complete system prior to the time when that system is to perform the mission for which it is designed. In the Voyager program, this objective is to be achieved prior to a fixed launch period, and with a high reliability.

Some of the general considerations of testing are as follows. Tests must be designed to uncover unknowns. Criteria must be established for determining failure to pass tests. Preflight testing should consider wear-out modes of the equipment. A properly-related test philosophy affects all parts of a scientific program such as Voyager, from the setting of goals through development and design, to fabrication and assembly, as well as the parts and system tests. Tests such as development tests, type approval tests, system integration tests, and flight acceptance tests should be directly related to, and be an outgrowth of, the test philosophy. Seen in this light, defining the test philosophy is a significant portion of the management and system-engineering function.

It is generally accepted that capability and reliability are designed and built into hardware. Testing will determine, with continuing analysis of results, the degree of success, definition of failure modes, and will form the basis for needed changes. Testing must uncover any significant deficiency or critical condition.

A complete test program involves four levels of testing which are generally time-phased--but overlapping--to permit flow of information from a following phase to influence preceding phases, as well as furnish a basis for the succeeding phase. These phases are:

1) Development testing of components, processes, breadboard models, and prototype models to determine that the basic design is sound, that materials and components have been correctly selected, and that fabrication procedures are satisfactory;
2) Verification testing of final hardware to achieve maximum assurance that the subsystems and the spacecraft as a whole are capable of surviving launch and performing the intended mission function. Test conditions are more severe than those expected during the mission;
3) Flight acceptance testing to ensure that no defects have been introduced in the manufacturing process. Test conditions are, in general, consistent with those expected during the mission;
4) Flight testing, including all system elements, to define the ability of the system to operate in the actual environment and to demonstrate the compatibility of the major system elements.

Previous Boeing experience with the Minuteman missile test program included all four testing levels and involved numerous interfaces and wide geographical dispersion of system elements, agencies, and contractors.

Particularly analogous to the potential role of the 1969 Test Flight in the Voyager program is the Minuteman Vandenberg test program. This program brought many system elements together in an operational configuration for the first time and indicated the equipment, personnel, and procedural changes required to ensure that the delivered system would operate as intended. Shake-down testing of this type, which the 1969 Test Flight would provide, is believed highly desirable for the Voyager program. This not only will include the operation of the spacecraft in flight but will include with equal importance the ground operations including procedures and personnel, before and during the spacecraft flight. There must be a first Voyager flight. If it is done in 1969, the benefits will accrue to the 1971 mission. If the 1969 Test Flight is not made, the first actual mission will be in 1971.

A potential disadvantage of including a 1969 Test Flight in the Voyager program is that earlier design decisions and design freeze dates are required. Sufficient testing to provide reasonable probability of success for 1969 is included in current schedule planning.

In conclusion, the recommendations that the 1969 Test Flight be included in the Voyager program, and that the 1971 system elements be used in the test flight, are based on previous experience with complex systems and vehicles. Some of these, such as Minuteman, included large-volume follow-on programs; others, such as the X-20, Lunar Orbiter, and HiBEX, are very similar to Voyager in the extensive testing required, the small number of deliverable articles, and the high reliability requirements.

I-3.0 TEST FLIGHT EVALUATION

A comparative evaluation of candidate 1969 Test Flights has been made to form the basis for the selection of the recommended launch vehicle and flight profile. The results are shown in Figure I-1.

For each of the two launch vehicles, Atlas/Centaur and Saturn IB/Centaur, three candidate flight profiles that correspond to launch vehicle/test spacecraft capabilities were considered. These are indicated in the left column. For each spacecraft function in the 1971 flight sequence, indicated across the top of the figure, the applicability of the 1969 Test Flight was subjectively evaluated considering the capability of the particular spacecraft, the degree of its commonality with the 1971 spacecraft, and the degree to which 1971 OSE and procedures were simulated. This evaluation of applicability, ranging from zero to one, is shown above the diagonal line in each block.

Although the applicability values assigned are in most cases subjective, the relative values do provide a basis for assessment. As an extreme example, consider Item 6.0, "Spacecraft-Capsule Separation." The Atlas/Centaur-launched test spacecraft carries no Flight Capsule, while the Saturn IB/Centaur-launched test spacecraft supports a simulated Flight Capsule which will be separated in the same manner in 1969 as in 1971. The applicability factor is therefore zero for the former and 1.0 for the latter.

Each function in the flight sequence was assigned a probability of successful accomplishment shown by the number at the top of each column.

SPACECRAFT	FLIGHT PROFILE	1.0 PRELAUNCH AT ETR
		1.00
1969 TEST SPACECRAFT ATLAS/CENTAUR LAUNCH	Mars Flyby	0.6 / 0.6
	Heliocentric (Simulated Mars Flyby)	0.6 / 0.6
	Earth Orbit	0.6 / 0.6
1969 TEST SPACECRAFT SATURN IB/CENTAUR LAUNCH	Mars Orbit	1.0 / 1.0
	Mars Flyby	1.0 / 1.0
	Heliocentric (Simulated Mars Flyby)	1.0 / 1.0

SPACECRAFT FLIGHT SEQUENCE — 1971 MISSION				
.0 LAUNCH & NJECTION	3.0 ACQUISITION	4.0 INTERPLANETARY CRUISE	5.0 INTER. TRAJ. CORRECTION	6.0 S CAPS
0.90	0.90	0.85	0.85	
.8 / 0.72	0.8 / 0.72	0.9 / 0.76	0.8 / 0.68	0
.8 / 0.72	0.8 / 0.72	0.9 / 0.76	0.8 / 0.68	0
.6 / 0.54	0.6 / 0.54	0.7 / 0.60	0.6 / 0.68	0
1.0 / 0.9	1.0 / 0.9	1.0 / 0.85	1.0 / 0.85	1.0
1.0 / 0.9	1.0 / 0.9	1.0 / 0.85	1.0 / 0.85	1.0
1.0 / 0.9	1.0 / 0.9	1.0 / 0.85	1.0 / 0.85	1.0

PACECRAFT ULE SEPAR.	8.0 SPACECRAFT CRUISE	13.0 SPACECRAFT ORBIT INSERTION	14.0 ORBITAL OPERATIONS	Σ
0.80	0.73	0.65	0.45	
0	0.9 / 0.66	0 / 0	0.1 / 0.04	4.19
0	0.9 / 0.66	0 / 0	0 / 0	4.13
0	0.7 / 0.51	0 / 0	0 / 0	3.47
0.8	1.0 / 0.73	1.0 / 0.65	1.0 / 0.45	7.13
0.8	1.0 / 0.73	0.8 / 0.52	0.1 / 0.04	6.59
0.8	1.0 / 0.73	0.6 / 0.39	0 / 0	6.42

Figure I-1: Applicability to 1971 Mission of 1969 Test Flight

These were taken directly from the "Voyager 1971 Mission Specification," JPL Project Document No. 45, except for intermediate events which were taken from straight-line interpolations between the end-points defined in the specification.

The value of the 1969 Test Flight to the 1971 mission for each function is then the product of the applicability number and the probability of accomplishing the function. This product is shown below the diagonal line in each block.

The final step in the assessment of each launch vehicle (and corresponding test spacecraft) / flight profile combination is the summation of the products appearing in each box. This summation is in the right column, with the highest value indicating the most desirable combination.

Some functions are more important to demonstrate than others. Prelaunch activities may be duplicated without a spacecraft flight. However, a spacecraft has not yet been placed in orbit around another planet, and the value of demonstrating this is very high. A weighing factor for each function could be employed to account for the difference in value. If this were done, the differences shown in the summation would be accentuated.

It is concluded that the order of preference for the 1969 Test Flight is:

1) Saturn Booster

    a) Mars orbit--Saturn IB/Centaur launch;

b) Mars flyby--Saturn IB/Centaur launch;

c) Simulated Mars-flyby--Saturn IB/Centaur launch.

2) Atlas Booster

a) Mars flyby-Atlas/Centaur launch;

b) Simulated Mars-flyby--Atlas/Centaur launch;

c) Earth orbit--Atlas/Centaur launch.

I-4.0 LAUNCH VEHICLE AVAILABILITY

The re-order lead-times for Atlas/Centaur and Saturn IB/Centaur have been estimated in coordination with the several suppliers. The estimated lead-times from order placement to availability on the launch pad for payload integration are 19 months for the Atlas/Centaur and 21 months for the Saturn IB/Centaur. An additional two months is allowed for payload integration and preflight testing.

The general approach has been to form a schedule which has maximum flexibility to adapt to new knowledge, and to engineering/management decisions. This has resulted in a booster-decision date which can be as late as 1 February 1967.

The necessary schedule adjustments of spacecraft subsystems to meet a 1969 Test Flight have been made using either an Atlas/Centaur or Saturn IB/Centaur. If the Saturn IB/Centaur is to be considered for 1969 test, preimplementation authorization will be required by February, 1966 for procurement of the orbit-insertion motor. Schedule estimates have been received from the supplier indicating a first-demonstration of the orbit-insertion motor in February 1967, the same month as the required launch-vehicle decision.

The spacecraft designs, with the exception of the orbit-insertion engine, are similar except for the relatively minor changes mentioned elsewhere; so work done prior to the booster-selection decision will be largely recoverable regardless of which choice is made.

The schedule in Figure I-2 shows the delivery dates of Saturn IB/Centaur launch vehicles currently on order, the pertinent dates for the Atlas/ Centaur and Saturn IB/Centaur configurations, a schedule breakdown for the orbit-insertion motor (since it appears to be the pacing item) and the assembly and test schedule for the first available 1969 spacecraft (the spare).

I-5.0 TEST PLAN

Considered as a part of the total series of tests which are required to achieve Voyager mission success in 1971, time limitations will limit the degree of ground testing for the 1969 Test Spacecraft as compared to the 1971 spacecraft. It is, however, necessary to complete sufficient testing to assure an adequate probability of success as well as a degree of design maturity that will minimize the probability of significant hardware changes between the 1969 and 1971 spacecraft. The latter requirement is reflected in the timing of significant events shown in the detailed program schedules of Section 5.0, Volume A.

Ground test assurance will be obtained through comprehensive testing on the 1969 proof test and compatibility test models in addition to flight acceptance testing (FAT) of the Flight Spacecraft. The proof test model will be the first 1969 flight configuration system to be

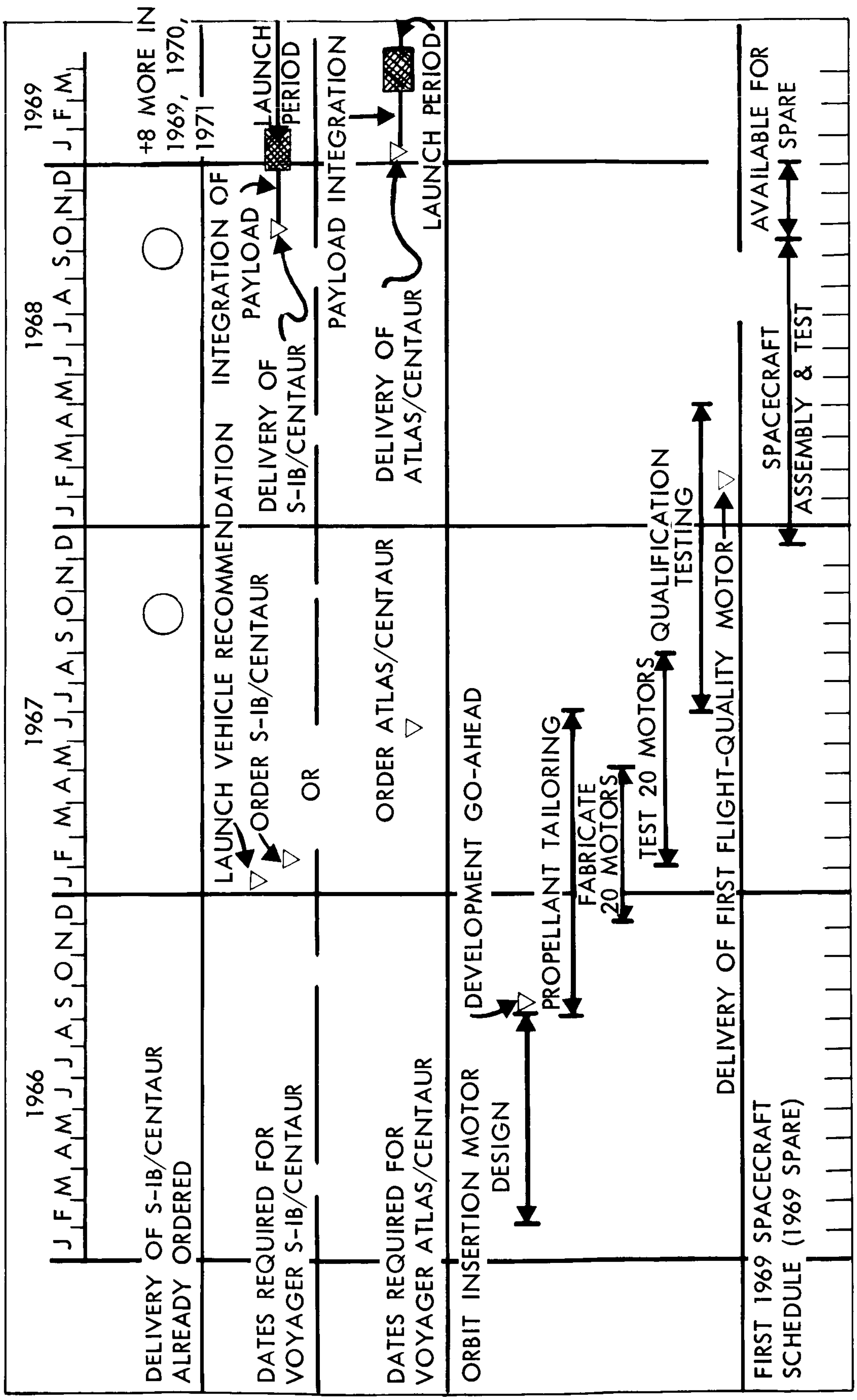
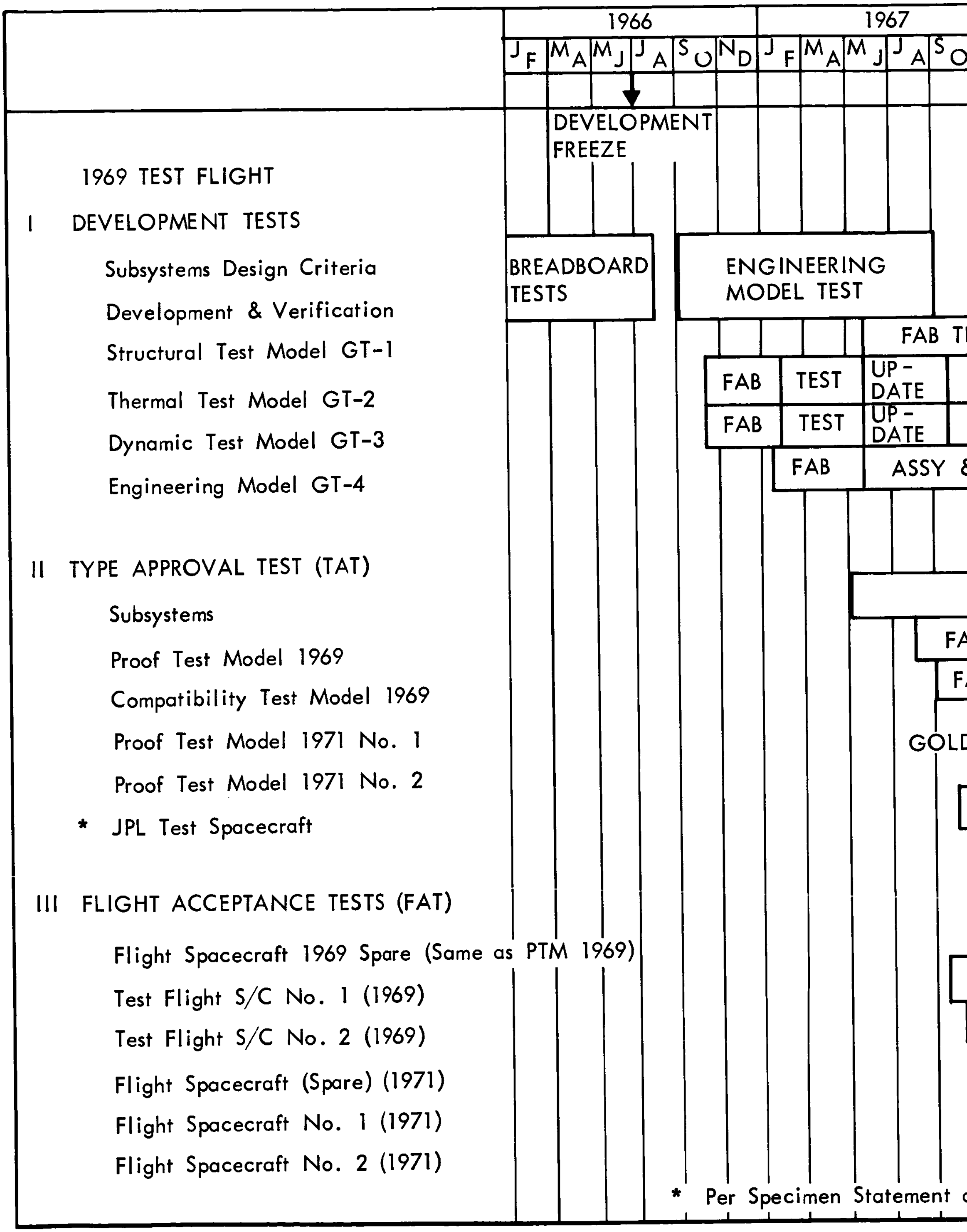


Figure I-2: Launch-Vehicle Availability

tested and will undergo a combined FAT/TAT program of sufficient scope to demonstrate adequacy of the system design for its test flight. Levels of testing on the PTM will be limited, however, such that its capability to serve as a spare Flight Spacecraft will not be compromised. The compatibility test model will undergo ambient system level FAT including EMI, launch countdown simulations and other special tests pertinent to both the Goldstone and ETR compatibility tests. Because of schedule restrictions space simulation, vibration and other tests not critical to the compatibility tests will not be conducted on the compatibility test model. The Flight Spacecraft will undergo complete FAT, both ambient and environmental, prior to delivery to ETR. The following table summarizes the amount of 1971 type spacecraft ground testing that will be accomplished prior to the 1969 launch.

	Test Type		
Test Level	TAT (%)	FAT (%)	COMPATIBILITY (%)
Subsystems	100	100	100
Complete 1969 Test Spacecraft			
(1) Atlas/Centaur Launch	(Modified Test)	100	100
(2) Saturn IB/Centaur Launch	(Plus limited level TAT)	100	100

Summary test plans for the Voyager program are shown in schedule form in Figure I-3, with a Saturn IB/Centaur-launched 1969 Test Spacecraft, and in Figure I-4, with an Atlas/Centaur-launched test spacecraft. It will be noted that data from the test flight is available during the assembly and test period of the 1971 proof test models and during the fabrication phase of the 1971 Flight Spacecraft. The 1969 Test Flight also provides about 9 months of life testing in advance of the ground life tests.


1966
1967
J F M A M J J A S O N D
J F M A M J J A S O
DEVELOPMENT FREEZE
1969 TEST FLIGHT
I DEVELOPMENT TESTS
Subsystems Design Criteria
Development & Verification
Structural Test Model GT-1
Thermal Test Model GT-2
Dynamic Test Model GT-3
Engineering Model GT-4
II TYPE APPROVAL TEST (TAT)
Subsystems
Proof Test Model 1969
Compatibility Test Model 1969
Proof Test Model 1971 No. 1
Proof Test Model 1971 No. 2
* JPL Test Spacecraft
III FLIGHT ACCEPTANCE TESTS (FAT)
Flight Spacecraft 1969 Spare (Same as PTM 1969)
Test Flight S/C No. 1 (1969)
Test Flight S/C No. 2 (1969)
Flight Spacecraft (Spare) (1971)
Flight Spacecraft No. 1 (1971)
Flight Spacecraft No. 2 (1971)
BREADBOARD TESTS
ENGINEERING MODEL TEST
FAB
TEST
UP - DATE
FAB
TEST
UP - DATE
FAB
ASSY
* Per Specimen Statement

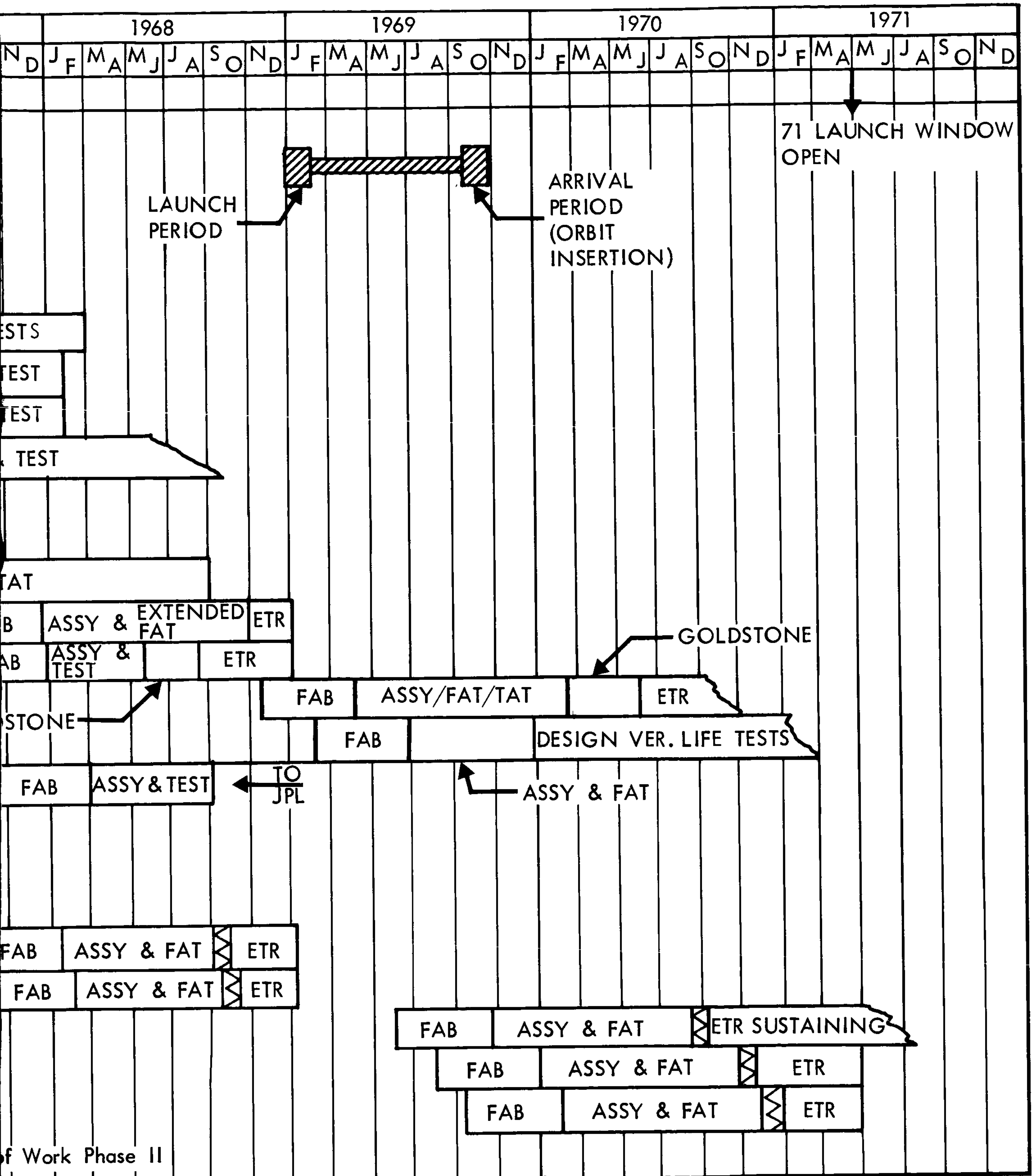


Figure 1-3: Integrated Test Program Schedule
1969 Saturn IB/Centaur Test Flight
1971 Saturn IB/Centaur Mission

2

	1966	1967
	J F M A M J J A S O N D	J F M A M J J A S O N D
	DEVELOPMENT FREEZE	
1969 TEST FLIGHT		
I DEVELOPMENT TESTS		
Subsystems Design Criteria	BREAD BOARD TESTS; ENGINEERING MODEL TESTS	ENGINEERING MODEL TESTS
Development & Verification	BREAD BOARD TESTS; ENGINEERING MODEL TESTS	ENGINEERING MODEL TESTS
Structural Test Model GT-1		FAB TEST
Thermal Test Model GT-2	FAB	TEST UPDATE TES
Dynamic Test Model GT-3	FAB	TEST UPDATE TES
Engineering Model GT-4		FAB ASSY
II TYPE APPROVAL TESTS (TAT)		
Subsystems		TA
Proof Test Model 1969		F
Compatibility Test Model 1969		F
Proof Test Model 1971 No. 1		
Proof Test Model 1971 No. 2		
* JPL Test Spacecraft		FA
III FLIGHT ACCEPTANCE TESTS (FAT)		
Flight Spacecraft 1969 Spare (Same as Compat. Test Model 1969)		
Test Flight S/C No. 1 (1969)		
Test Flight S/C No. 2 (1969)		
Flight Spacecraft (Spare) (1971)		
Flight Spacecraft No. 1 (1971)		
Flight Spacecraft No. 2 (1971)		

* Per Specimen Statement-of-Work F

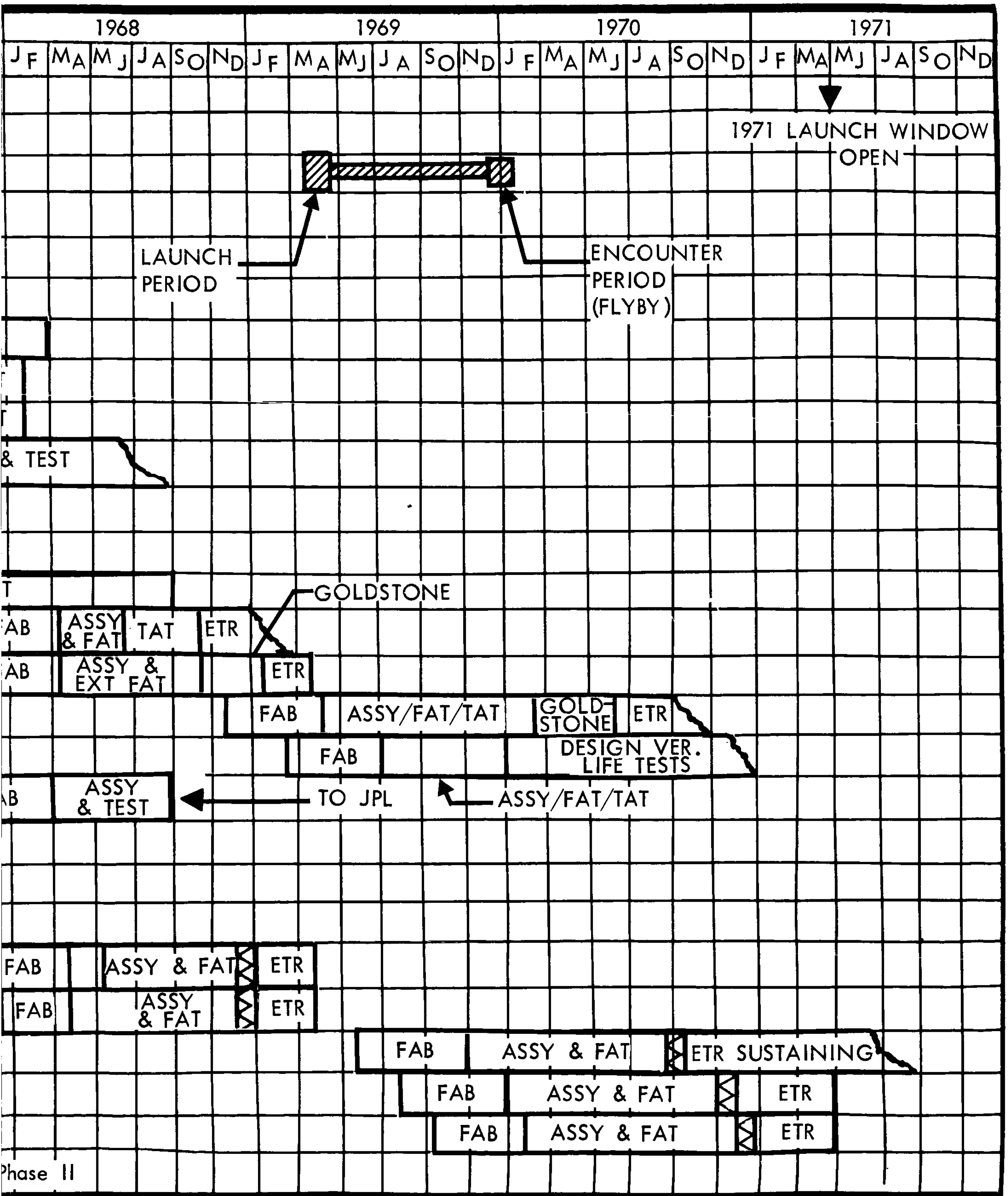


Figure 1-4: Integrated Test Program Schedule
1969 Atlas/Centaur Test Flight
1971 Saturn IB/Centaur Mission

2

The Atlas/Centaur-launched test flight has the advantage of a later launch period, which permits more extensive ground testing prior to flight. The Saturn IB/Centaur-launched test flight, in addition to providing more significant data, provides that data at an earlier time in the program, thus allowing more time for corrective action.

I-6.0 CONCLUSIONS AND RECOMMENDATIONS

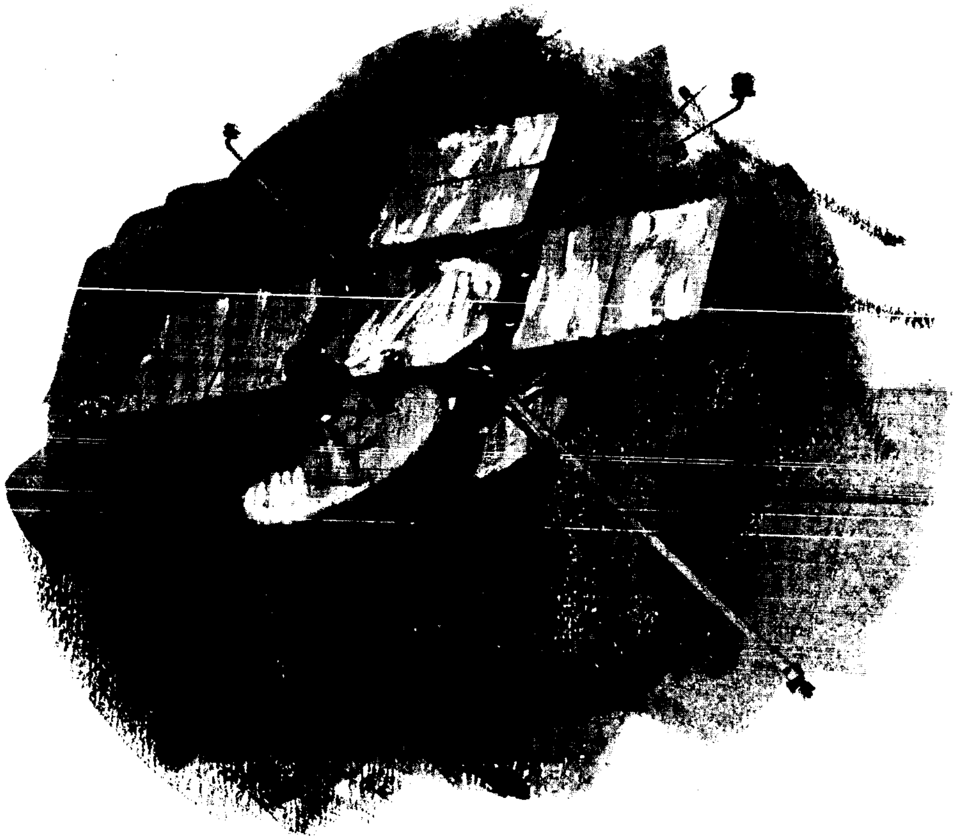
Consideration has been given to several 1969 Test Flight options and spacecraft configurations and to two launch vehicles. The recommended 1969 Test Flight program has the advantages of providing direct support to and enhancement of the 1971 Voyager program and of providing flexibility of implementation through multiple launch date and launch vehicle choices.

The objective of the 1969 Test Flight is to enhance the probability that the 1971 mission will be successful. An evaluation of the potential ways of achieving this objective, and of the value of any 1969 Test Flight, results in the following recommendations:

1) A 1969 Test Flight can be accomplished within a properly phased 1971 Voyager Program, is a desirable part of the program, and is recommended.
2) The 1969 launch vehicle should be a Saturn IB/Centaur. The 1969 Test Flight should be as nearly like the 1971 mission as possible and use a 1971 spacecraft with a simulated Flight Capsule and a simulated science payload. The simulated Flight Capsule should be separated as planned for 1971, but away from the planet. After the separation, the spacecraft should be inserted into a Mars orbit.

3) If the Saturn IB/Centaur cannot be made available, a 1969 Mars flyby using an Atlas/Centaur launch vehicle and a 1971 spacecraft bus with minimum modification is recommended. The principal modifications are deletion of the orbit insertion engine, three of the six solar cell panels, and two of the three battery sections; reductions in propellant tank volume and high-gain antenna area; and revised solar-cell panel and antenna folding geometry.

4) If the Mars orbit launch period is missed, the Saturn IB/Centaur-launched test spacecraft should be placed on a Mars flyby trajectory. If this opportunity is also missed, the test spacecraft should be placed on an heliocentric orbit simulating a Mars flyby. In the case of the Atlas/Centaur-launched test spacecraft, a simulated Mars flyby test flight should be made if the Mars flyby launch period is missed.



PART II - 1969 TEST SPA[illegible]
ATLAS/CENTAUR

PART II--1969 TEST SPACECRAFT-ATLAS/CENTAUR

This part of the document establishes the requirements and descriptions for a test spacecraft and its associated subsystems in performance of a 1969 Test Flight using an Atlas/Centaur launch vehicle. The statements herein are presented in a form that establishes the differences between the 1969 test program and the 1971 Voyager mission and configuration. The latter are presented in Volume A,D2-82709-1 of this series of documents.

II-1.0 1969 TEST SPACECRAFT TEST PROGRAM OBJECTIVES AND DESIGN CRITERIA -- ATLAS/CENTAUR

II-1.1 1969 TEST SPACECRAFT INTRODUCTION-- ATLAS/CENTAUR

Two types of test flights using Atlas/Centaur launch vehicles are considered for the 1969 Test Flight; a primary Mars flyby and a heliocentric flight as an alternate. In keeping with test flight objectives, the 1969 Test Spacecraft will be identical with the 1971 Flight Spacecraft configuration except for the differences caused by flight plan, weight, and volume limits. The principal differences are omission of the orbit insertion engine, science payload (including magnetometer and boom), scan platform instruments, VHF antenna, and Flight Capsule. Reduced size and modified folding arrangement of solar cell panels and antennas is also necessary.

II-1.2 1969 TEST FLIGHT OBJECTIVES --ATLAS/CENTAUR

The objective of the 1969 Test Flight is stated in Part I of this document.

II-1.3 1969 TEST SPACECRAFT TEST PROGRAM RESTRAINTS--ATLAS/CENTAUR

1) Two Atlas/Centaur launch vehicles will be provided for the 1969 opportunity, with the capability of launching two spacecraft in a 30-day period.

2) Launch and prelaunch operations will be conducted at AFETR Complex 36A and 36B, using facilities installed for the Centaur R&D program.

3) Spacecraft prelaunch assembly and checkout will be conducted at the Spacecraft Checkout Facility (Building AO).

4) An explosive safe facility will be used for propellant and gas loading, final spacecraft alignment, installation of other hazardous components, and spacecraft encapsulation within the nose fairing.

5) The planetary quarantine requirements as stated in Volume A will be respected.

6) Payload variation with the Atlas/Centaur versus C_3 for a due east launch and a 70-degree to 108-degree launch azimuth is shown in Figure II-1.

7) The Atlas/Centaur is as described in Appendix A of JPL Project Document No. 46, "Voyager 1971 Mission Guidelines," with one variation. The nose fairing as described in the referenced document will be modified by the addition of a 54-inch-long, 10-foot-diameter cylindrical section. Figure II-2 presents the estimated launch probability of success for the Atlas/Centaur versus calendar time for shroud lengths of 6 feet (surveyor shroud) and 10.5 feet (Voyager shroud). The shroud length is defined as the circular cylinder length measured from launch vehicle station 219 forward to the start

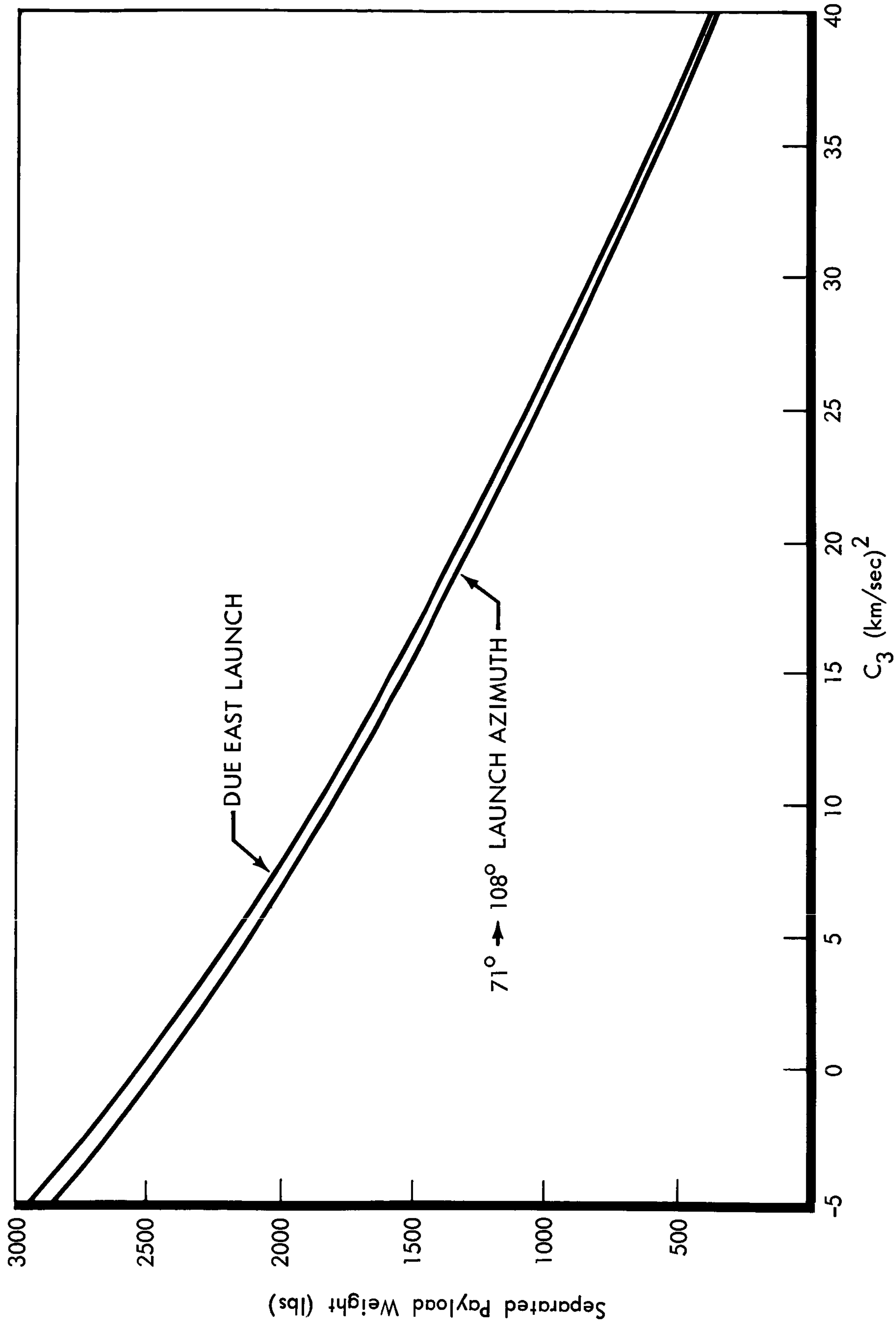


Figure II-1: Separated Payload Capability versus Energy — Atlas/Centaur

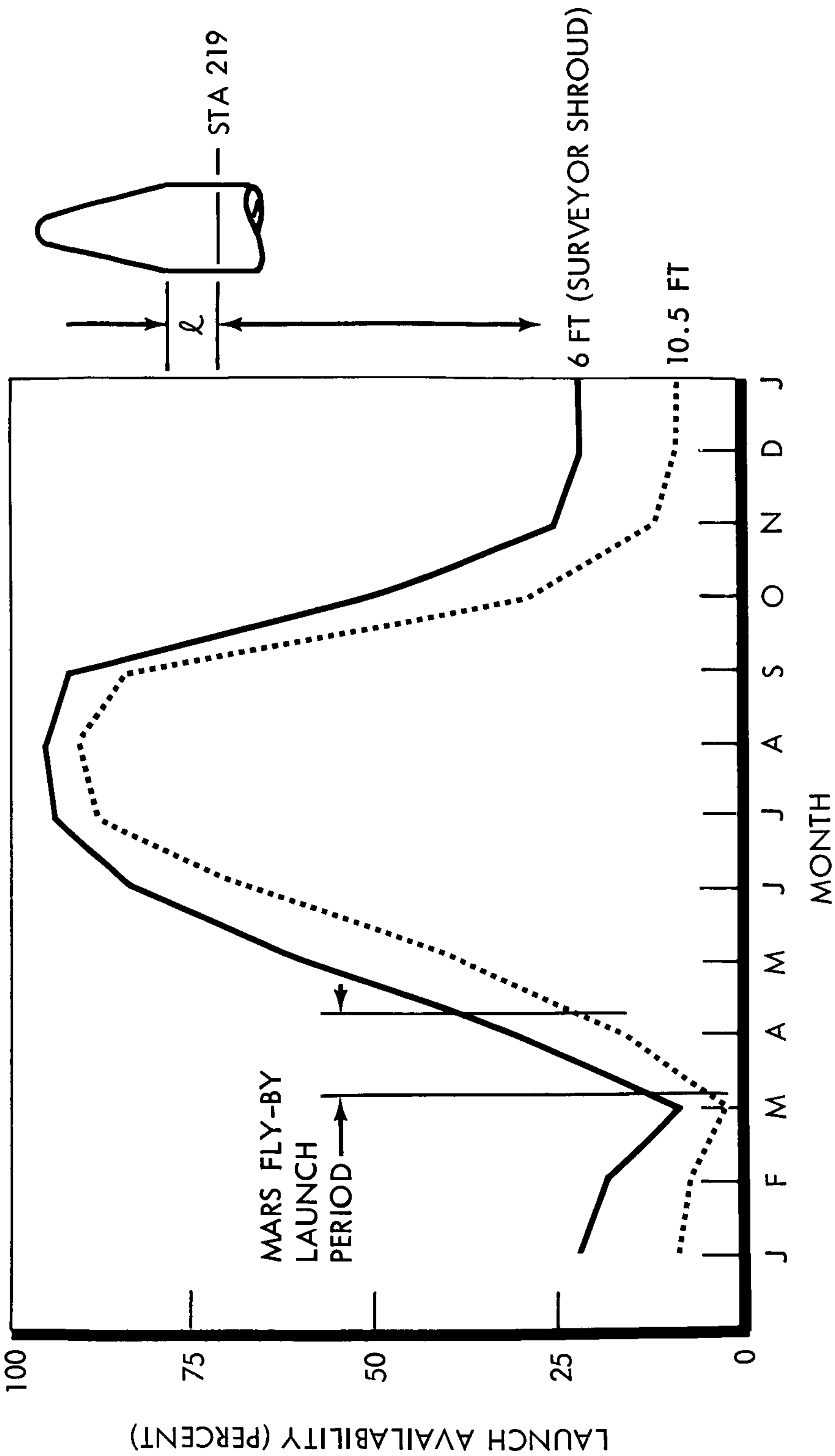


Figure II-2: Estimated Launch Availability Due to Launch Vehicle Structural Capability for Atlas/Centaur of the nose fairing. This is also shown in Figure II-2. The analysis method used in developing the figure was a static rigid body trim condition. Loads were increased by 12 percent to account for aeroelastic effects. The analysis determined the bending moment distribution and associated allowable $q\alpha$ within the structural allowables and engine deflection limitations of the Atlas booster. Gust bending moment obtained from NASA TM-X-1094 and adjusted for shroud length was subtracted out leaving a bending moment available for wind shear. The allowable wind speed was then computed using the bending moment available for winds, trajectory data, and an estimate of the α response induced by wind shear. The data of NASA TN-D-1274 shown in Figure II-3 was used to determine the monthly wind probability associated with the allowable wind speed computed for each shroud length. Figure II-2 shows that launch availability for the 10.5-foot shroud can be as low as 2 percent during March, the windiest month, and as high as 91 percent during August. For the 1969 Mars Flyby launch opportunities, the launch availability will vary from a low of approximately 7 percent (March 10, 1969) to a high of approximately 22 percent (April 9, 1969). Additional analysis and expanding of the above data is desirable prior to launch.

II-1.4 SCHEDULE CRITERIA -- ATLAS/CENTAUR

The 1969 Mars opportunity places absolute constraints upon the schedule for the Mars flyby test. However, there is a schedule extension available with the alternate test program. Regardless of the 1969 test all design and fabrication of flight systems, subsystems, and components must be compatible with obtaining significant data for 1971 Voyager application.

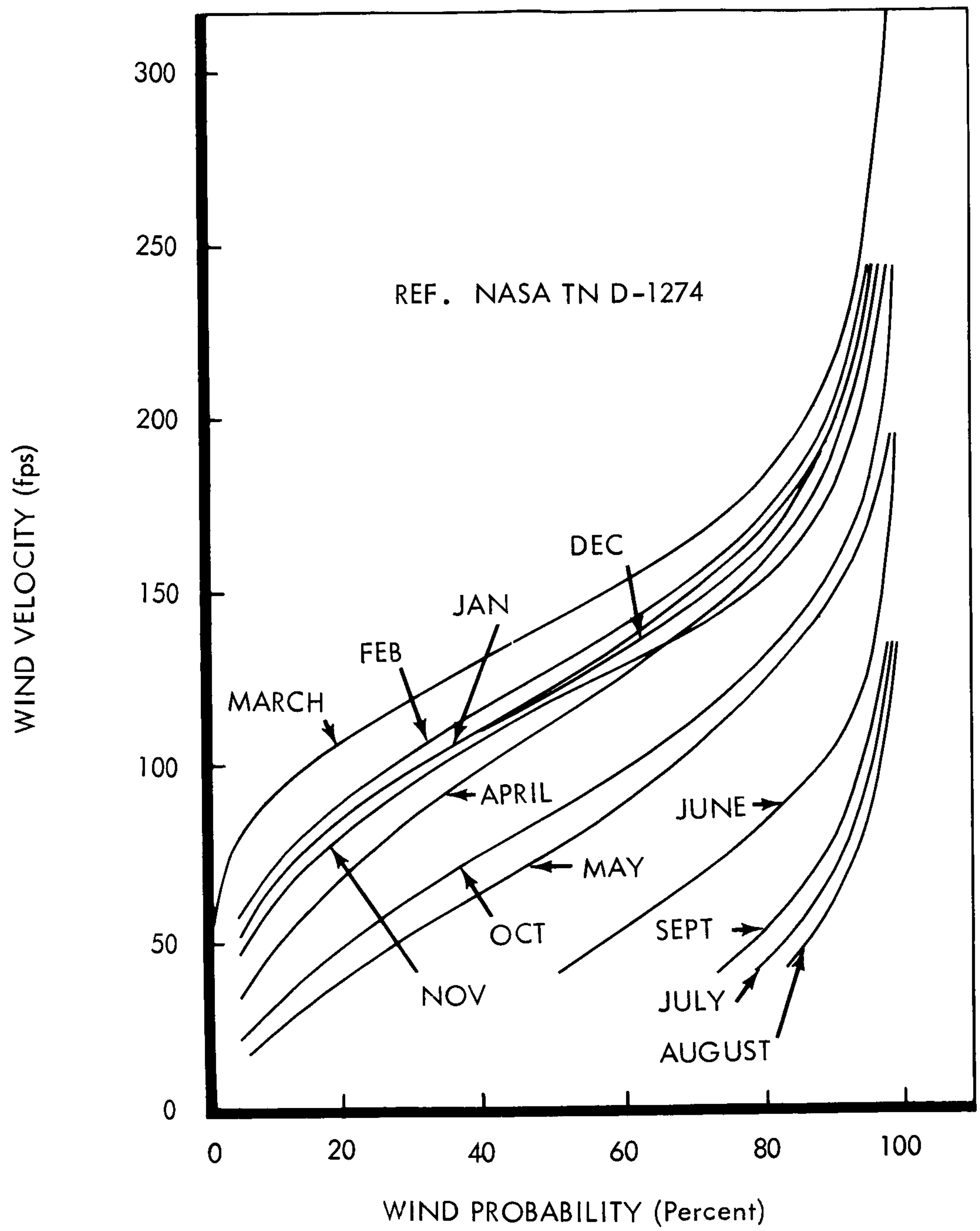


Figure II-3: Wind Probability vs Wind Speed

II-1.5 1969 TEST SPACECRAFT WEIGHT - ATLAS/CENTAUR

The total spacecraft weight (including spacecraft to launch vehicle adapters) for the three 1969 test programs must be no more than 1700 pounds. The separated spacecraft weight must be no more than 1500 pounds.

II-1.6 1969 TEST SPACECRAFT COMPETING CHARACTERISTICS -- ATLAS/CENTAUR

1971 spacecraft priorities will be used where there are conflicting technical requirements.

II-2.0 1969 TEST SPACECRAFT DESIGN CHARACTERISTICS AND RESTRAINTS -- ATLAS/CENTAUR

II-2.1 1969 TEST SPACECRAFT DESIGN CHARACTERISTICS -- ATLAS/CENTAUR

II-2.1.1 1969 Test Program Profiles

II-2.1.1.1 Mars Flyby

The primary 1969 Test Flight is a Mars flyby using a Type II trajectory, a C_3 of 9.0 $(km/sec)^2$, and launch dates from March 10 through April 9, 1969. Flyby distance from Mars will be of a magnitude which will provide data for the 1971 Voyager mission and which will respect the planetary quarantine requirements.

II-2.1.1.2 Heliocentric Flight

The alternate test flight considers a heliocentric flight of the same duration and distance as established for the 1971 Voyager mission. The heliocentric flight will place the test spacecraft on a trajectory which will intercept the Mars orbit at the point where Mars would be in 1971. The flight will use a Type I trajectory, a C_3 of 15.5 $(km/sec)^2$, and launch dates from May 7 through June 22, 1969. Open launch dates are available for a heliocentric flight which places the test spacecraft at Mars distance out of the orbit plane.

II-2.1.2 Subsystems

With the exclusion of the science subsystem the characteristics and restraints for the 1969 Test Spacecraft subsystems are the same as those established in Section 2.1.3 of Volume A, D2-82709-1, for the 1971 preferred design.

Detailed subsystem functional descriptions are contained in Section II-4.0.

II-2.2 1969 TEST SPACECRAFT DESIGN RESTRAINTS -- ATLAS/CENTAUR

The restraints imposed on the subsystems by the system and other subsystems (including environmental restraints) for the 1969 test program are the same as those established in Volume A for the 1971 mission. The restraints imposed by the Atlas/Centaur launch vehicle are stated in Appendix A of JPL Project Document No. 46, "Voyager 1971 Mission Guidelines."

II-2.3 1969 TEST SPACECRAFT GUIDANCE AND NAVIGATION MANEUVER ACCURACY AND PROPULSION REQUIREMENTS -- ATLAS/CENTAUR

II-2.3.1 Description

The maneuver accuracy and propulsion requirements for the 1969 Test Flight are the same as described in Volume A for the 1971 Preferred Design. The approach to aim-point biasing during midcourse is identical to that established in Section 3.1 of Volume B, D2-82709-2.

II-2.3.2 1969 Test Flight Benefits

The 1969 Test Flight will provide a test of capability to accurately guide the spacecraft to a successful encounter using three midcourse maneuvers.

II-2.4 1969 TEST SPACECRAFT AIMING POINT SELECTION -- ATLAS/CENTAUR

II-2.4.1 Description

The aiming point for a Mars flyby trajectory in 1969 is selected by a procedure similar to that described in Section 2.4 of Volume A, D2-82709-1. The principal difference is that occultation of the Sun and Canopus, undesirable for the 1971 mission, may be specifically desired for the 1969 Test Flight to verify predicted spacecraft performance under occultation conditions. Minor changes in the aim points for occultation result from the differences in arrival geometry, and from the concern with occultation after periapsis, for a flyby as well as before. Also, the aiming point selection is not limited by any considerations of the desired orbital parameters, such as inclination and periapsis altitude.

II-2.4.2 1969 Test Flight Benefits

As an experiment, the aiming point can be selected such that occultation of the Sun or Canopus occurs. This would help to determine the spacecraft's behavior under occultation conditions similar to those encountered during the Mars-orbit phase of the 1971 Voyager mission.

The final selection of an aiming point for a particular trajectory depends upon analysis of the length of time the spacecraft should be occulted. Ideally, occultation on the flyby should last slightly longer than the maximum occultation in orbit for the 1971 mission. For most of the trajectories under consideration, the range of aiming points giving occultation of both the Sun and Canopus is quite limited, so the duration of occultation cannot be varied for both. One must aim either to occult both bodies, and accept whatever duration of occultation results, or to occult only one of the bodies for a specified length of time. If two spacecraft are launched, one can be aimed for occultation of the Sun, and the other, for Canopus occultation.

For one representative trajectory, an aiming point with $|\bar{B}| = 10{,}000$ km and a θ of 120 degrees gives solar occultation only. Canopus occultation occurs for the same $|\bar{B}|$ with θ's between 195 and 300 degrees. For this particular transit trajectory, Earth occultation also occurs for those aiming points which give occultation of both the Sun and Canopus.

II-2.5 1969 TEST SPACECRAFT PARTS, MATERIALS AND PROCESSES--ATLAS/CENTAUR

II-2.5.1 Description

Parts, materials, and processes for the 1969 Test Spacecraft will be the same as described in Section 2.5 of Volume A, D2-82709-1, for the 1971 Preferred Design and will be subjected to the same controls.

II-2.5.2 1969 Test Flight Benefits

The 1969 Test Flight will provide verification testing of critical material systems and parts through functional performance in the true trans-Mars environment. Changes in performance resulting from the environmental exposure of the 1969 Test Flight can be interpreted to estimate performance in the 1971 mission;

1) Thermal control coating performance will be determined by spacecraft temperature data.

2) The change in performance in solar cells will be indicated by power generation data.

3) Bearing lubricant and gear performance of scan platform and antennas will be evaluated by functional operation of mechanisms.

II-3.0 1969 TEST SPACECRAFT SYSTEM LEVEL FUNCTIONAL DESCRIPTION OF FLIGHT SPACECRAFT - ATLAS/CENTAUR

II-3.1 1969 TEST SPACECRAFT STANDARD TRAJECTORIES - ATLAS/CENTAUR

Three trajectories have been investigated for the 1969 Test Flight: (1) Mars flyby, (2) heliocentric orbit, and (3) Earth orbit. The fly-by tests pass as close to Mars as the planetary quarantine constraints allow. In any year, a heliocentric flight is possible which has all the characteristics of the interplanetary transfer for 1971.

In the heliocentric flight, an encounter of Mars is not attempted. Rather, the launch and subsequent operations are made to hold as closely as possible to those for the 1971 mission. In this way, the spacecraft follows essentially the same path in space, but it arrives at Mars' orbit at a time when Mars is elsewhere in its orbit. In this way, all trajectory aspects of the transit are simulated. Considering a 1500-pound spacecraft and a 200-pound adapter, the Atlas/Centaur can provide a maximum C_3 of 12 km^2/sec^2. The 1971 mission is based upon a maximum C_3 of 18 km^2/sec^2 so some relaxation must be made to the launch period length. For a C_3 of 12 $(km/sec)^2$ and an approach speed, with respect to the selected aim point, of 3.5 km/sec, the spacecraft can be launched from April 30 to June 10. Similarly, all other types of potential 1971 missions can be simulated.

Alternatively, if less complete trajectory simulation is acceptable, trajectories with C_3 = 12 km^2/sec^2 or less can be launched almost any day of the year. Such a flight would not necessarily pass through Mars orbit, but would attain representative distances from Earth in representative time.

The Earth orbits are elliptical with perigees of 200 nautical miles and apogees of 21,700 nautical miles and 30,600 nautical miles and with periods of 12 hours and 18 hours, respectively. The various trajectory characteristics are outlined for each considered test flight in Table II-1 in conjunction with allocated weight for the test spacecraft.

Figure II-4 is a basic trajectory design chart for 1969 Type II Mars test flight which shows C_3, launch date, and arrival date in black. The declination of the geocentric asymptote (DLA) is shown in red. Only the Type II trajectories are shown since the Type I trajectories have high geocentric declinations which do not satisfy the launch azimuth requirements.

II-3.2 1969 TEST SPACECRAFT HELIOCENTRIC ORBIT DETERMINATION CAPABILITY --ATLAS/CENTAUR

II-3.2.1 Description

The heliocentric orbit determination capability for the 1969 Test Flight will not be significantly different from the capability discussed in Volume A, Section 3.2, and Volume B, Section 3.1.

II-3.2.2 1969 Test Flight Benefits

II-3.2.2.1 Mars Flyby

The 1969 Test Flight will provide information that will allow refinement of the physical constants that affect heliocentric orbit determination

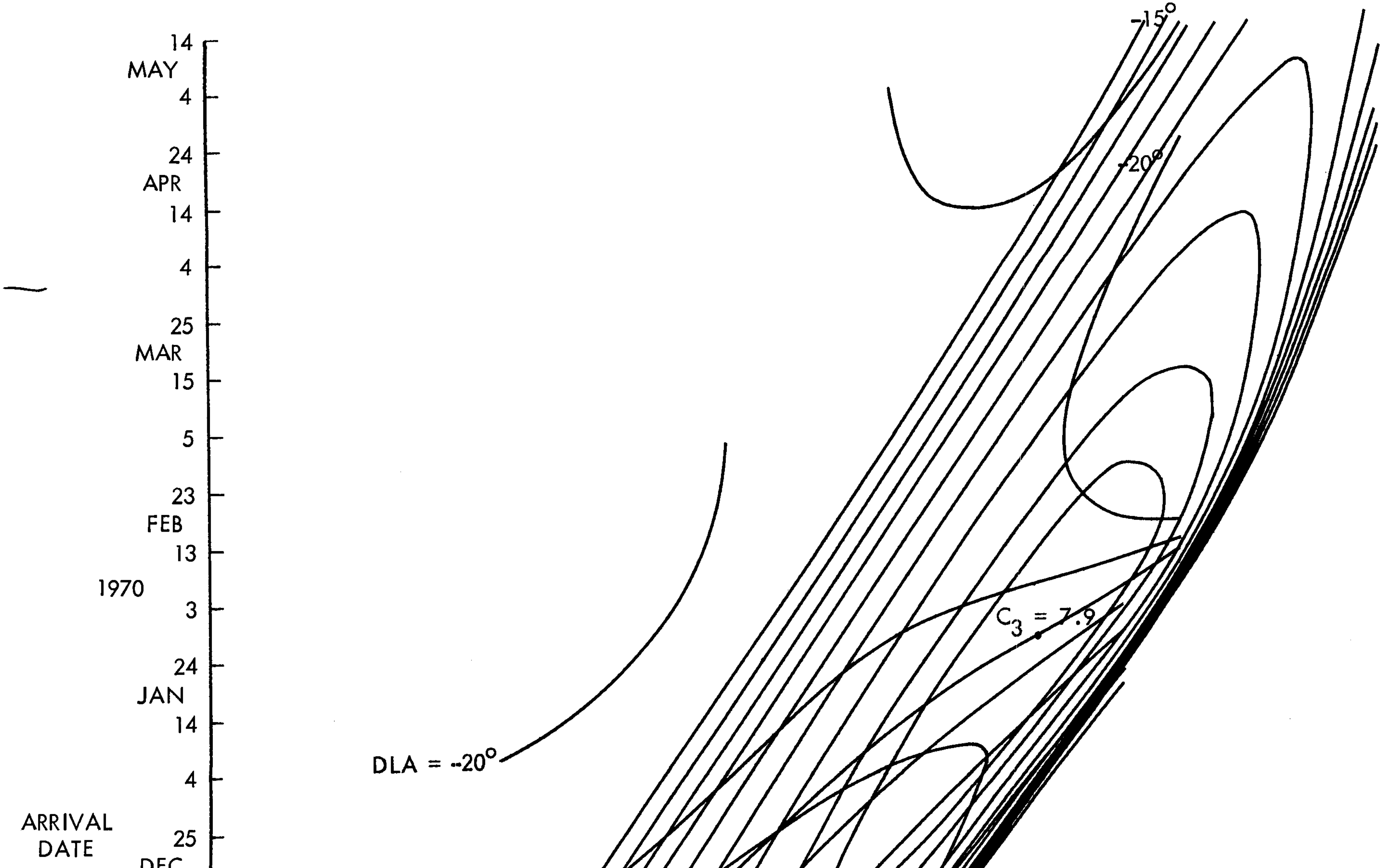
MISSION	BOOSTER	LAUNCH PERIOD (DAYS)	TRAJ TYPE	LAUNCH DATES
Flyby — Type II High DLA	Atlas/Centaur	30	II	Mar 10, 1969/Apr 9, 1969
Heliocentric 50-Day Launch Period	Atlas/Centaur	50	I	May 5, 1969/June 26, 196
Heliocentric 42-Day Launch Period	Atlas/Centaur	42	I	Apr 30, 1969/June 10, 196
12-Hour Period Orbit	Atlas/Centaur	Not Restricted	—	Spring or Summer, 1969
18-Hour Period Orbit	Atlas/Centaur	Not Restricted	—	Spring or Summer, 1969

1) At location in space where Mars will be in 1971 opportunity

Table II-1: Atlas/Centaur Trajectory Characteristics

ARRIVAL DATES	REQUIRED C_3 $(km/sec)^2$	LAUNCH VEHICLE SEPARATED WEIGHT CAPABILITY (lbs)	TRANSIT TIMES (DAYS)	DLA RANGE	COMMUNICATION DISTANCE (km)
Dec 18, 1969/Jan 22, 1970	9.0	1850	283/288	$-5^{\circ}/-28^{\circ}$	262×10^6
Dec 15, 1969 1	8/18	1390	172/222	$-22^{\circ}/-28^{\circ}$	158×10^6
Oct. 31, 1969/ Nov. 8, 1969 1	9/12	1700	176/220	$-33^{\circ}/-18^{\circ}$	110×10^6
	$\Delta V=8115$	3900	--	--	--
	$\Delta V=8680$	3500	--	--	--

ALLOCATED WEIGHT			
ADAPTERS		SPACECRAFT SEPARATED WEIGHT	TOTAL WEIGHT
ABOVE FIELD JOINT	BELOW FIELD JOINT		
50 lbs	150 lbs	1500 lbs	1700 lbs



ARRIVAL DATE
1970
14
MAY
4
24
APR
14
4
25
MAR
15
5
23
FEB
13
3
24
JAN
14
4
25
DEC
DLA = -20°
-15°
-20°
C_3 = 7.9

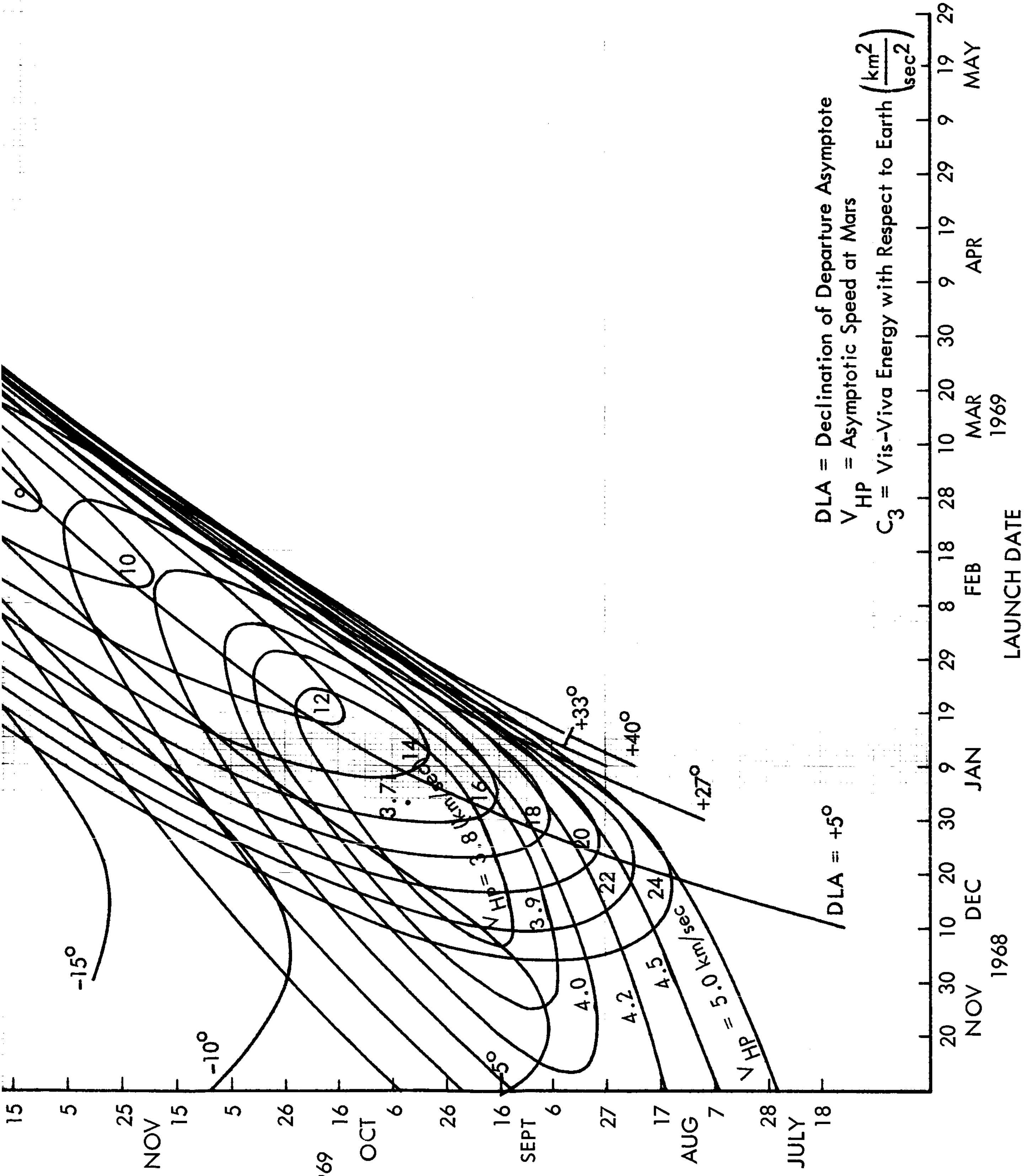


Figure II-4: Earth-Mars Transfer Characteristics —1969 Type II

2

capability, including the astronomical unit, the Mars ephemeris, and the Mars gravitational parameters. This will provide significant improvement in the 1971 mission orbit determination capability.

II-3.2.2.2 Heliocentric Flight

The benefits accuring from a heliocentric flight are the same as for the Mars flyby with the significant exceptions that no data on Mars ephemeris or Mars gravitational parameters will be obtained.

II-3.3 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, SPACECRAFT COMPONENTS DESIGN PARAMETERS - ATLAS/CENTAUR

II-3.3.1 Description

The Spacecraft Components Design Parameters Sheets (SCDPS) serve as a control for weight, power, volume, and thermal operating ranges of all the 1969 Test Spacecraft components. Table II-2 identifies the changes, deletions and additions (by subsystem and component nomenclature, and item number) from the list described in Section 3.3 of Volume A, D2-82709-1, 1971 Preferred Design. A description of the mass properties for the 1969 Test Spacecraft is found in Section II.4.4.6.

II-3.4 1969 TEST SPACECRAFT EQUIPMENT ELEMENT IDENTIFICATION --ATLAS/CENTAUR

II-3.4.1 Description

The methods and procedures for hardware and software identification for the 1969 Test Spacecraft are the same as described in Section 3.4 of Volume A, D2-82709-1, 1971 Preferred Design.

Table II-2: 1969 ATLAS/CENTAUR TEST SPACECRAFT COMPONENTS DESIGN PARAMETERS

DELETIONS

Subsystem	Assembly/Component	Item No.	No. Required 1971 Spacecraft
Telecommunications	Relay Antenna VHF	1.2.1.1.3.5	1
Telecommunications	Relay Radio Subsystem	1.2.1.1.4	1
Spacecraft Structures	VHF Antenna Supports	1.2.1.7.1.6.3	1
Mechanisms	VHF Antenna Mech.	1.2.1.8.3	1
Mechanisms	Science Boom	1.2.1.8.5	1
Mechanisms	Bacteriological Barrier	1.2.1.8.6	1
Temperature Control	Heat Shield Solid Motor	1.2.1.9.1.1	1
Temperature Control	Insulation Blanket	1.2.1.9.1.2	1
Temperature Control	Thermal Shroud Aft Side Area	1.2.1.9.1.4	1
Temperature Control	Louvers--Prop. Module	1.2.1.9.1.7	-
Orbit Injection Propulsion	--	1.2.2.2	1
Science Instrumentation	--	1.2.3.1	1
Science Data Automation	--	1.2.3.2	1
Electrical Power	Solar Panel Strut	1.2.1.6.1.4	6

ADDITIONS

Subsystem	Assembly/Component	Item No.	No. Required 1971 Spacecraft
Telecommunications	Multiplexer/Encoder (for engineering data)	1.2.1.1.2.2.13	1 (5 lbs)

CHANGES

Subsystem	Assembly/Component	Item No.	Changes 1971 to 1969	Parameters
Telecommunications	Tape Recorders	1.2.1.1.2.1.1	2 1	Quantity
Telecommunications	High-Gain Antenna	1.2.1.1.3.2	8'x12'ellipse - 8' circular	Size/Shape
Telecommunications	Spacecraft Antenna Subsystems	1.2.1.1.3	54.3 lb. 42.5 lb.	Weight

Table II-2: 1969 ATLAS/CENTAUR TEST SPACECRAFT COMPONENTS DESIGN PARAMETERS

CHANGES (Continued)

Subsystem	Assembly/Component	Item No.	Changes 1971 to 1969		Parameters
Reaction Control	N_2 Tanks	1.2.1.4.1	128.0 lb	54.0 lb	Size/Weight
Reaction Control	N_2 Propellant	1.2.1.4.11	60.0 lb	25.0 lb	Weight
Power Electrical	Solar Panel Array	1.2.1.6.1	284.0 lb	154.0 lb	Size/Weight
Power Electrical	Substrate Assembly	1.2.1.6.1.3	12	9	Quantity
Power Electrical	DC Motor	1.2.1.6.1.6	6	3	Quantity
Power Electrical	Panel Boost Latch	1.2.1.6.1.7	12	6	Quantity
Power Electrical	Squib Double Bridge	1.2.1.6.1.8	12	6	Quantity
Power Electrical	Panel Deploy Latch	1.2.1.6.1.9	6	3	Quantity
Power Electrical	Photovoltaic All Assembly	1.2.1.6.1.10	48,708	24,354	Quantity
Power Electrical	Battery Installation	1.2.1.6.2	3	2	Quantity
Power Electrical	Battery Charger and Failure Sensor	1.2.1.6.3.1	3	2	Quantity
Spacecraft Structure	Equipment Support Structure	1.2.1.7.1	373.2 lb	305.0 lb	Weight
Spacecraft Structure	Prop./Reaction Control Support Structure	1.2.1.7.2	124.6 lb	40.0 lb	Weight
Mechanisms	High-Gain Antenna Mech. Instl.	1.2.1.8.1	31.5 lb	25.0 lb	Weight
Temperature Control	Thermal Shroud Aft Base Area	1.2.1.9.1.5	4.4 lb	1.4 lb	Weight
Instl. Cables & Tubing	Spacecraft Propulsion Subsy, Cabling/Harness Inst.	1.2.1.11.3	10.0 lb	8.0 lb	Weight
Midcourse Correction Propulsion	Propellant System--Inerts	1.2.2.1.2	75.0 lb	39.0 lb	Weight
Midcourse Correction Prop.	Exp. Hydrazine-Usable	1.2.2.1.3	395.0 lb	38.0 lb	Weight
Midcourse Correction Prop.	Pressurization Systems	1.2.2.1.4	16.0 lb	15.0 lb	Weight

II-3.4.2 1969 Test Flight Benefits

Use of the methods and procedures for the 1969 Test will provide early training and experience which can be readily applied to the 1971 Voyager program.

II-3.5 1969 TEST SPACECRAFT FLIGHT EQUIPMENT LAUNCH VEHICLE INTERFACE REQUIREMENTS--ATLAS/CENTAUR

II-3.5.1 Description

The 1969 Test Spacecraft flight equipment launch vehicle interface requirements are different to the extent of differences between the Atlas/Centaur and Saturn IB/Centaur. These include but are not limited to:

1) The 1969 test uses a modified Surveyor shroud;
2) The spacecraft envelope dimensions are different (see Para. II-3.10);
3) There is no Flight Capsule in the 1969 test flight.

II-3.5.2 1969 Test Flight Benefits

The 1969 Test Flight provides a limited opportunity to prove interface compatibility between the Centaur and the Flight Spacecraft. These benefits are limited to those interfaces identical to the Saturn IB/Centaur launch vehicle.

II-3.6 1969 TEST SPACECRAFT FLIGHT EQUIPMENT TELEMETRY CRITERIA - ATLAS/CENTAUR

II-3.6.1 Description

The primary objective of the telemetry equipment is to measure, with high resolution and accuracy, the spacecraft subsystem performance parameters and encode the information on subcarriers with low error probability for subsequent rf transmission.

The 1969 Test Flight telemetry equipment are identical to the 1971 Voyager spacecraft equipment as described in Volume A, D2-82709-1 except for incorporation of the capability for multiplexing additional engineering measurements. The test spacecraft provides the necessary capabilities and functions for the telemetry equipment design verification. On the basis that capsule engineering and science data will not be available for the 1969 Test Flight, equivalent simulation will be provided to exercise the telemetry modes, techniques, and equipment within the weight limitations of the Atlas/Centaur launch vehicle. In lieu of actual science data, the telemetry channels are to be used for additional spacecraft engineering measurements in conjunction with spacecraft equipment design verification.

The feasible test flights are essentially Mars flyby and Heliocentric trajectory.

For a Mars flyby test flight, Mode 6, planetary science high bit rate, will be exercised prior to encounter. The rf-link calculations indicate operation at encounter will be in the rf-link blackout region. Other modes will be exercised throughout the test flight for transmission of engineering data.

The Voyager 1969 Test Spacecraft will be equipped with telemetry and communications equipment capable of transmitting spacecraft data to the

DSIF. The transmitted data consists of spacecraft engineering, and simulated high-rate science data. If high-bit-rate performance data is required, it can replace the simulated data on command. The capsule engineering data channel is used for input of additional spacecraft engineering data. The capsule-to-spacecraft-telemetry on-board interface is exactly simulated. During critical spacecraft maneuvers, spacecraft telemetry data can be redundantly stored on-board for subsequent retransmission.

The 1969 Test Flight telemetry equipment, data formatting, modulation techniques, data storage modes, and on-board core storage and tape recorders are identical to the 1971 spacecraft telemetry equipments except that only one tape recorder will be used to save weight. Since no capsule is carried the radio relay subsystem will not be on-board.

The measurements required for the 1971 flight will be made on the 1969 Test Flight using the regular engineering multiplexer-encoder. Additional slow-sampling-rate data will be telemetered using another multiplexer-encoder unit whose output feeds the capsule data channel. It is expected that this second unit will be identical to the engineering multiplexer-encoder in design.

Since a 48-kilocycle bit rate capability exists, it can be used for low-accuracy transient-type information such as midcourse propulsion motor pressure or attitude control parameters that could not be telemetered at the slow bit rates. Encoding at 6 bits results in a single channel 8-kilocycle sample rate capability which can be multiplexed or time shared.

Simulation of the planetary or cruise science data for checkout of Modes 5 and 6 will be accomplished using known sequences of data bits. For example, an 11-stage shift register with feedback will generate a fixed recycling PN sequence of $2^{11}-1$ (i.e. 2047) bits in length which can be easily synchronized with the block coder or tape recorder bit rates. Since the exact pattern is known, the data will be automatically reduced to determine the system error rate. Switching from Mode 5 to 6 is done by speeding up the input clock to the shift register and will not change the ground data processing equipment. In this way, planetary or cruise science data channels can be evaluated.

II-3.6.2 1969 Test Flight Benefits

Since the 1969 Test Flight telemetry equipment will be identical to the 1971 Voyager spacecraft equipment except for incorporation of the capability of multiplexing additional engineering measurements, the 1969 Test Flight will greatly intensify and accelerate experience with and evaluation of the 1971 Voyager spacecraft telemetry equipments and techniques.

II-3.7 1969 TEST SPACECRAFT FLIGHT EQUIPMENT TELEMETRY CHANNEL LIST-ATLAS/Centaur

II-3.7.1 Description

The total flight data measurements requirements for the 1969 Test Spacecraft are listed in Table II-3. The list is a modification to the one provided for the 1971 preferred design to reflect the deletions and additions necessitated by the 1969 spacecraft configuration and test flight. The deletions and additions are as follows:

Table II-3: FLIGHT DATA MEASUREMENT LIST

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An/Dig.	Rate Samples/Second
	ANTENNA					
1	High-Gain Antenna Hinge Angle	Angle	$0-360^{\circ}$	$1/2^{\circ}$	10 bit	1/60
2	High-Gain Antenna Swivel Angle	Angle	$0-360^{\circ}$	$1/2^{\circ}$	10 bit	1/60
3	High-Gain Antenna RF Input	RF Pwr	0-50 w	2%	0-50 mw	1/600
4	Low-Gain Antenna RF Input	RF Pwr	0-50 w	2%	0-50 mw	1/600
5	High-Gain Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200
6	Low-Gain Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200
7	High-Gain Antenna Hinge Motor Temp	Temp	$\pm 300^{\circ}F$	2%	± 2.5 v	1/1200
8	High-Gain Antenna Swivel Motor Temp	Temp	$\pm 300^{\circ}F$	2%	± 2.5 v	1/1200
9	High-Gain Antenna Hinge Torque	Inch-Pounds	0-100	2%	0-5 v	1/60
10	High-Gain Antenna Swivel Torque	Inch-Pounds	0-100	2%	0-5 v	1/60
11-18	High-Gain Antenna Dish Temperature	Temp	$\pm 150^{\circ}F$	2%	± 2.5 v	1/60
	RADIO					
19	Receiver AGC (Coarse)	Signal Strength	-90 to -162 dbm	2%	0-5 v	1/60
20	Receiver AGC (Fine)	Signal Strength	-125 to -150 dbm	2%	0-5mv	1/600
21	Receiver Static Phase Error	Phase Angle	$\pm 20^{\circ}$	2%	± 2.5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An/Dig.	Rate Samples/Second
	RADIO (Continued)					
22	Receiver L.O. Drive	Osc Pwr	0-1 mw	2%	0-50 mw	1/600
23	Exciter Output Power	Osc Pwr	0-0.2 w	2%	0-5 v	1/600
24	TWT Helix Voltage	Volts	0-2 kv	2%	0.5 v	1/1200
25	TWT Helix Current	ma	0-20 ma	2%	0-5 v	1/1200
26	TWT Coil Current	ma	0-100 ma	2%	0-5 v	1/1200
27	Exciter Voltage 1	Volts	0 to -25 volts	2%	±2.5 v	1/1200
28	Exciter Voltage 2	Volts	0 to -15 volts	2%	±2.5 v	1/1200
29	Crystal OSC Temperature (oven)	Temp	25 to 150°F	2%	0-5 v	1/1200
30	Command Det Mon	Status	yes/no	discrete	1 bit	1/60
31	TWT A Temp	Temp	±150°F	5%	±2.5 v	1/1200
32	TWT B Temp	Temp	±150°F	5%	±2.5 v	1/1200
33	Receiver A Temp	Temp	±150°F	5%	±2.5 v	1/1200
34	Receiver B Temp	Temp	±150°F	5%	±2.5 v	1/1200

TABLE II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An/Dig.	Rate Samples/Second
	TELEMETRY AND DATA STORAGE SYSTEM					
35	Recorder "A" Pressure	x port press	0-4 psia	2%	0-5 v	1/1200
36	Mode Status	Modes	0-4 psia	discrete	2 bits	1/60
37	Event Counter A	Event	0-100	discrete	7 bits	1/60
38	Event Counter B	Event	0-100	discrete	7 bits	1/60
39	Recorder A Temp	Temp	±150°F	5%	±2.5 v	1/1200
40-46	Electronics Package Temp	Temp	±150°F	2%	±2.5 v	1/1200
	ATTITUDE REFERENCE/ AUTOPILOT					
47	Pri. Sun Sen Fine Pitch Error			5%	0-5 v	1/60
48	Pri. Sun Sen Fine Yaw Error			5%	0-5 v	1/60
49	Pri. Sun Sen Coarse Pitch Error			5%	0.5 v	1/60
50	Pri. Sun Sen Coarse Yaw Error			5%	0.5 v	1/60
51	Sec. Sun Sen Fine Pitch Error			5%	0.5 v	1/60
52	Sec. Sun Sen Fine Yaw Error			5%	0.5 v	1/60
53	Sec. Sun Sen Coarse Pitch Error			5%	0.5 v	1/60
54	Sec. Sun Sen Coarse Yaw Error			5%	0.5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An/Dig.	Rate Samples/Second
	ATTITUDE REFERENCE/ AUTOPILOT (Continued)					
55	Sun Sensor Pitch Error			5%	0.5 v	1/60
56	Sun Sensor Yaw Error			5%	0.5 v	1/60
57	Sun Sensor Yaw Acquisition			5%	discrete	1/60
58	Sun Sensor Pitch Acquisition			5%	discrete	1/60
59	Pri. Sun Sen Pitch Acquisition			5%	discrete	1/60
60	Pri. Sun Sen Yaw Acquisition			5%	discrete	1/60
61	Sec. Sun Sen Pitch Acquisition			5%	discrete	1/60
62	Sec. Sun Sen Yaw Acquisition			5%	discrete	1/60
63	Canopus Pitch Gimbal Angle			5%	0-5 v	1/60
64	Canopus Acquisition			5%	discrete	1/60
65	Canopus Roll Error			5%	±2.5 v	1/60
66	Canopus Star Magnitude			5%	±2.5 v	1/60
67	Pri. Canopus Sen Acquisition			5%	discrete	1/60
68	Sec. Canopus Sen Acquisition			5%	discrete	1/60
69	Pri. Canopus Sen Roll Error			5%	±2.5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An/Dig.	Rate Samples/Second
	ATTITUDE REFERENCE/ AUTOPILOT (Continued)					
70	Sec. Canopus Sen Roll Error			5%	±2.5 v	1/60
71	Pri. Canopus Star Magnitude			5%	±2.5 v	1/60
72	Sec. Canopus Star Magnitude			5%	±2.5 v	1/60
73	Attitude Ref Case Temp			1%	0-5 v	1/60
74	IRV Pwr Supply 1			5%	0-5 v	1/60
75	IRV Pwr Supply 2			5%	0-5 v	1/60
76	IRV Pwr Supply 3			5%	0-5 v	1/60
77	Roll Gyro Sel Verify			5%	discrete	1/60
78	Pitch Gyro Sel Verify			5%	discrete	1/60
79	Yaw Gyro Sel Verify			5%	discrete	1/60
80	Roll Gyro Rate			5%	0-5 v	1/60
81	Pitch Gyro Rate			5%	0-5 v	1/60
82	Yaw Gyro Rate			5%	0-5 v	1/60
83	Roll Gyro "A" Pos			5%	0-5 v	1/60
84	Roll Gyro "B" Pos			5%	0-5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An/Dig.	Rate Samples/Second
	ATTITUDE REFERENCE/ AUTOPILOT (Continued)					
85	Pitch Gyro "A" Pos			5%	0-5 v	1/60
86	Pitch Gyro "B" Pos			5%	0-5 v	1/60
87	Yaw Gyro "A" Pos			5%	0-5 v	1/60
88	Yaw Gyro "B" Pos			5%	0-5 v	1/60
89	EMA Accel. + Output			5%	0-5 v	1/60
90	EMA Accel. - Output			5%	0-5 v	1/60
91	Bell Accel. + Output			5%	7 bit	1/60
92	Bell Accel. - Output			5%	7 bit	1/60
93	EMA DC Amp			5%	0-5 v	1/60
94	Bell DC Amp			5%	0-5 v	1/60
95	Trisafe Amp 1 Output			5%	0-5 v	1/60
96	Trisafe Amp 2 Output			5%	0-5 v	1/60
97-104	Trisafe Amp 2 Output			5%	0-5 v	1/60
105-110	Trisafe Pwr Supply 1			5%	0-5 v	1/60
111-126	Jet Vane 1 Feedback			5%	0-5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An.Dig.	Rate Samples/Second
	<u>CENTRAL COMPUTER & SEQUENCER</u>					
127-133	Programmer Data	Mode Cond	Instruct Words	absolute	22 bits	1/60
134	Programmer Parity	Ind	yes/no	discrete	1 bit	1/60
135	Vehicle Time	Time	233 hrs	1/10 sec	22 bits	1/60
136	Command Verify	Verify	Command Word	absolute	26 bits	1/60
137	A/C Mode Select	Mode Cond	yes/no	discrete	3 bits	1/60
138	Power Mode Select	Cond	yes/no	discrete	1 bit	1/600
139	Unit Temperature	Temp	$\pm 150^{\circ}$	1%	± 2.5 v	1/1200
140	Science Data Mode Select	Mode Cond	yes/no	discrete	3 bits	1/60
141	Command Message Identification	Mode Cond	yes/no	discrete	6 bits	1/60
142	Processor 1, 2	Mode Cond	yes/no	discrete	1 bit	1/60
143	Command Decoder 1, 2	Mode Cond	yes/no	discrete	1 bit	1/60
	<u>REACTION CONTROL</u>					
144	A/C Tank 1 Pressure	psia	0-375	2%	0-5 v	1/60
145	A/C Tank 2 Pressure	psia	0-375	2%	0-5 v	1/60
146	Engine Pitch Act + (OIP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An.Dig.	Rate Samples/Second
	<u>REACTION CONTROL</u> (Continued)					
147	Engine Pitch Act + (MCP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600
148	Engine Yaw Act + (OIP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600
149	Engine Yaw Act + (MCP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600
150	Jet Driver Pitch +	Cond	yes/no	discrete	1 bit	1/1200
151	Jet Driver Pitch -	Cond	yes/no	discrete	1 bit	1/1200
152	Jet Driver Yaw +	Cond	yes/no	discrete	1 bit	1/1200
153	Jet Driver Yaw -	Cond	yes/no	discrete	1 bit	1/1200
154	Jet Driver Roll +	Cond	yes/no	discrete	1 bit	1/1200
155	Jet Driver Roll -	Cond	yes/no	discrete	1 bit	1/1200
156	A/C Gas Temp 1	Temp (Farenheit)	$\pm 150^{\circ}$	1%	0-5 v	1/1200
157	A/C Gas Temp 2	Temp (Farenheit)	$\pm 150^{\circ}$	1%	0-5 v	1/1200
	<u>ELECTRICAL POWER</u>					
158	Solar Panel 1 Current	Amp	0-10 A	2%	0-5 v	1/60
159	Solar Panel 2 Current	Amp	0-10 A	2%	0-5 v	1/60
160	Solar Panel 3 Current	Amp	0-10 A	2%	0-5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An.Dig.	Rate Samples/Second
	<u>ELECTRICAL POWER</u> (Continued)					
161	Battery B1 Voltage	Volts	0-50 v	2%	0-5 v	1/60
162	Battery B2 Voltage	Volts	0-50 v	2%	0-5 v	1/60
163	Battery B1 Discharge	Amp	0-15 A	2%	0-5 v	1/60
164	Battery B2 Discharge	Amp	0-15 A	2%	0-5 v	1/60
165	Battery B1 Charge	Amp	0-5 A	2%	0-5 v	1/60
166	Battery B2 Charge	Amp	0-5 A	2%	0-5 v	1/60
167	Unregulated DC Bus Current	Amp	0-30 A	2%	0-5 v	1/60
168	Unregulated DC Bus Voltage	Volt	0-50 v	2%	0-5 v	1/60
169	Prime Telecom Regulator	Amp	0-15 A	2%	0-5 v	1/600
170	Standby Telecom Regulator	Amp	0-15 A	2%	0-5 v	1/600
171	Prime Spacecraft Regulator	Amp	0-15 A	2%	0-5 v	1/600
172	Standby Spacecraft Regulator	Amp	0-15 A	2%	0-5 v	1/600
173	Regulated DC Bus (Spacecraft)	Volt	0-50 v	2%	0-5 v	1/600
174	Regulated DC Bus (Telecom)	Volt	0-50 v	2%	0-5 v	1/600
175	400 cps 1 Ø Bus Voltage	Volt	0-30 v	2%	0-5 v	1/600

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An, Dig.	Rate Samples/Second
	<u>ELECTRICAL POWER</u> (Continued)					
176	400 cps 1 Ø Inverter Current	Amp	0-3 A	2%	0-5 v	1/600
177	400 cps 3 Ø Bus Voltage	Volt	0-30 v	2%	0-5 v	1/600
178	400 cps 3 Ø Inverter Current	Amp	0-3 A	2%	0-5 v	1/600
179	2400 cps Inverter Prime	Amp	0-5 A	2%	0-5 v	1/600
180	2400 cps Inverter Standby	Amp	0-5 A	2%	0-5 v	1/600
181	Power Sync Frequency	Cycles/ Second	350 to 450 cps	2%	0-5 v	1/600
182	Power Sync Frequency	Kc/Sec	2.2 to 2.6 kc	2%	0-5 v	1/600
183	Solar Panel 1 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
184	Solar Panel 2 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
185	Solar Panel 3 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
186	Regulator Output Ripple	MV	0-500 mv	2%	0-5 v	1/1200
187	Battery B1 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
188	Battery B2 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
189	Telecom Regulator 1 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
190	Telecom Regulator 2 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	ELECTRICAL POWER (Continued)	^{o}F				
191	Spacecraft Regulator 1 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
192	Spacecraft Regulator 2 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
193	PS&L Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
194	2.4 kc Inverter 1 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
195	2.4 kc Inverter 2 Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
196	400 cps 1 Ø Inverter Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
197	400 cps 3 Ø Inverter Temp	^{o}F	$\pm 150^{o}$	5%	± 2.5 v	1/1200
198 - 207	Relay RL-1 thru RL-10 Status	Cond	yes/no	discrete	10 bits Total	1/1200
	THERMAL CONTROL					
208	Louver 1 Position	Angle	$0-90^{o}$	2%	0.5 v	1/1200
209	Louver 2 Position	Angle	$0-90^{o}$	2%	0.5 v	1/1200
210	Louver 3 Position	Angle	$0-90^{o}$	2%	0.5 v	1/1200
211	Louver 4 Position	Angle	$0-90^{o}$	2%	0.5 v	1/1200
212	Louver 5 Position	Angle	$0-90^{o}$	2%	0.5 v	1/1200
213	Louver 6 Position	Angle	$0-90^{o}$	2%	0.5 v	1/1200

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident No.	Measurement	Unit	Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	THERMAL CONTROL (Continued)					
214	Coldplate 1 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
215	Coldplate 2 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
216	Coldplate 3 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
217	Coldplate 4 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
218	Coldplate 5 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
219	Coldplate 6 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
220	Coldplate 7 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
221	Coldplate 8 Temp	Temp	$\pm150^{\circ}$	5%	±2.5 v	1/1200
222	M/C Motor Shield Temp	Temp	0-1K°F	5%	0.5 v	1/1200
223	M/C Motor Shield Temp	Temp	0-1K°F	5%	0.5 v	1/1200
224 - 253	Spacecraft Temperatures		(to be determined)			
	PROPULSION					
254	Gas Supply Pressure	psia	0-4000	5%	0-5 v	1/60
255	Fuel Tank Pressure	psia	0-400	5%	0-5 v	1/60
256	Gas Supply Temperature I	°R	300-600	5%	0-5 v	1/60

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	PROPULSION (Continued)					
257	Gas Supply Temperature II	^oR	300-600	5%	0-5 v	1/60
258	Fuel Tank Temperature I	^oR	400-600	5%	0-5 v	1/600
259	Fuel Tank Temperature II	^oR	400-600	5%	0-5 v	1/600
260	Motor Temperature	^oR	400-1500	5%	0-5 v	1/60
261	Engine Valve Driver-Fuel	Cond	yes/no	discrete	1 bit	1/60
262 - 302	M/C Motor Chamber Pressure	psia	0-15 k	2%	0-5 v	1/60
	MECHANISMS					
303	Solar Panel 1 Deploy	Cond	yes/no	discrete	1 bit	1/600
304	Solar Panel 2 Deploy	Cond	yes/no	discrete	1 bit	1/600
305	Solar Panel 3 Deploy	Cond	yes/no	discrete	1 bit	1/600
306						
307	Scan Platform Position	Angular Position	0-90^o	2%	0.5 v	1/600
308	Scan Platform Deploy	Cond	yes/no	discrete	1 bit	1/1200
309	Optics Cover Position	Cond	yes/no	discrete	1 bit	1/1200
310	Gimbal Motor 1 Temp	Temp	$\pm 150^o F$	2%	± 2.5 v	1/1200

Table II-3: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	<u>MECHANISMS</u> (Continued)					
311	Gimball Motor 2 Temp	Temp	$\pm 150^{\circ}F$	2%	± 2.5 v	1/1200
312 - 317	Scan Platform Axis I Drive Torque	Inch-Pound	0-100	2%	0.5 v	1/60
318 - 323	Scan Platform Axis II Drive Torque	Inch-Pound	0-100	2%	0.5 v	1/60

1) Antenna--Added high-gain antenna motor torque measurements on both hinge and swivel motors, and temperature measurements on the dish;
2) Telemetry and Data Storage--Deleted measurements on Recorder B and added temperature measurements for electronic packages;
3) Relay Radio--Deleted all measurements;
4) Attitude Reference--Added 12 error and magnitude measurements, and several discrete function measurements;
5) Autopilot--Added eight error and rate measurements;
6) Electrical Power--Deleted the measurements on Battery B3;
7) Thermal Control--Added approximately 30 temperature measurements and six louver measurements;
8) Propulsion--Added four midcourse propulsion motor pressure measurements;
9) Mechanisms--Added torque measurements on the drive motors of scan platform;

II-3.8 1969 TEST SPACECRAFT GUIDANCE AND NAVIGATION MANEUVER ERROR ALLOCATIONS AND ANALYSIS--ATLAS/CENTAUR

II-3.8.1 Description

The detailed allocation of maneuver error to component error sources will nto vary for the 1969 test spacecraft from that described in Section 3.8 of Volume A, D2-82709-1, 1971 Preferred Design.

II-3.9 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, FLIGHT SEQUENCE--ATLAS/CENTAUR

II-3.9.1 Description

The 1969 Test Spacecraft functions are basically similar to the functions described in Section 3.9 of Volume A. The test flight profile is divided into nine gross functional areas shown in Figure II-5. A detailed expansion of the 13 test spacecraft subsystems are shown in Figure II-6.

In summary, the differences from the 1971 mission are:

1) The countdown and launch sequence for the 1969 Atlas/Centaur Test Flight is different from the 1971 Saturn IB/Centaur mission;
2) Additional engineering instrumentation data acquisition is substituted for cruise science data acquisition;
3) The capsule-flight spacecraft separation sequence is deleted.

II-3.9.2 1969 Test Flight Benefits

The 1969 Atlas/Centaur Test Flight will provide an opportunity to prove the flight sequence functions well in advance of the 1971 mission, therefore guarding against any unforeseen problems that might prejedice 1971 mission performance.

II-3.10 1969 TEST SPACECRAFT FLIGHT EQUIPMENT SPACECRAFT LAYOUT AND CONFIGURATION--ATLAS/CENTAUR

The 1969 Test Spacecraft is the same as that described in Section 3.10 of Volume A, D2-82709-1 for the 1971 Preferred Design except as described below. The configuration drawing, isometric sketches of its structure, and its propulsion module, a sketch of a solar panel, and an inboard profile drawing are shown. A configuration trade study drawing and a short trade discussion are included. Mass properties and nose fairing modifications are also discussed. Figure II-7 is a pictorial view of the test spacecraft.

1969 TEST FLIGHT ATLAS/CENTAUR	PRELAUNCH AT ETR	LAUNCH AND INJECTION	ACQUISITION	INTERPLA CRUISE
LAUNCH VEHICLE SYSTEM	System checkout, mating, and combined systems testing	Temperature control for Spacecraft; Telemetry from Test Spacecraft; Shroud separation; Orientation in parking orbit through Spacecraft separation; Assurance that Launch Vehicle meets non-impacting requirements		
SPACECRAFT SYSTEM (See Figure II-6 for detailed expansion)	System checkout of test Spacecraft; Fuel and install pyrotechnic at Explosive Safe Facility; Mate to adapter and shroud; transport to pad; mate Launch Vehicle and testing; Power to subsystems	Temperature control after shroud off; Telemetry to Launch Vehicle; Separate from Launch Vehicle; Make engineering measurements; Environmental control; Power to subsystems	Acquire reference objects; Provide power to subsystems; Environmental control; Provide RF signal to Earth; Make engineering measurements	Attitude Power to tems; Pro signal to Make int etary eng measurem Environm control
CAPSULE SYSTEM	Not Applicable			
MISSION OPERATIONS SYSTEM	Operational Readiness test; Spacecraft Deep Space Network compatibility test	Monitor telemetry; Receive and evaluate Eastern Test Range prediction	Monitor telemetry; Send backup commands; Send Deep Space Network Station prediction	Monitor t and contr Spacecra necessary mine traj characte
LAUNCH OPERATIONS SYSTEM	Scheduling and coordination of Eastern Test Range activities	Control launch; Monitor telemetry; Provide tracking to injection		
DEEP SPACE NETWORK	Checkout and training; Support Mission Operations System	Provide tracking as required; Support Mission Operations System	Provide two-wo	

NETARY	INTERPLANETARY TRAJECTORY CORRECTION	SIMULATE CAPSULE SEPARATION AND PREPARE FOR MARS ENCOUNTER	MARS FLYBY ACQUIRE AND TRANSMIT DATA	COAST BEYOND MARS;ACQUIRE AND TRANSMIT DATA MAKE TRAJECTORY MANEUVER	ANALYZE DATA AND REPORT
ontrol; ubsys- ide RF Earth; rplan- ineering ents; ntal	Provide power to subsystems; Orient thrust vector; Provide ΔV; Temperature control; Make engineering measurements; Provide RF signal to Earth; Reacquisition	Provide power to subsystems; Environmental control; Maintain attitude references; Provide RF signal to Earth; Conduct encounter checkout; Provide commands to simulate capsule separation	Provide power to subsystems; Environmental control; Maintain attitude references; Provide RF signal to Earth; Make engineering measurements	Provide power to subsystems; Environmental control; Maintain attitude references; Provide RF signal to Earth; Make engineering measurement; orient thrust vector; provide ΔV; Earth reacquisition.	
elemetry ol Test t as ; Deter- ectory istics	Monitor telemetry; Determine and send midcourse command; Control Spacecraft as necessary; Determine trajectory characteristics	Monitor telemetry; determine trajectory characteristics	Monitor telemetry; Determine orbit characteristics	Monitor telemetry on signal strength decay; determine and send trajectory correction command; control spacecraft as necessary; determine trajectory characteristics	Analyze all data and prepare report on mission after termination.
y communication lock; Provide tracking support (Mission Operations Systems)					

Figure II-5: Mission Profile — 1969 Atlas/Centaur FlyBy

2

1.0
PRELAUNCH
AT ETR

GROUND COMPLEX (MOS, LOS, & DSN)	A. OPERATIONAL READINESS TEST, S/C - DSN B. COMPATIBILITY TEST C. SCHEDULE AND COORDINATE ETR ACTIVITIES D. C/O AND SUPPORT MOS (DSN ONLY) E. LOAD CC&S WITH FLIGHT PROGRAM
SPACECRAFT TELECOMMUNICATIONS	SUBSYSTEM C/O AND STATUS MONITORING.
CENTRAL COMPUTER AND SEQUENCER (CC&S)	A SUBSYSTEM C/O AND STATUS MONITORING B COMMAND OTHER SUBSYSTEMS FOR C/O AND STATUS MONITORING C READY ALL SUBSYSTEMS FOR LAUNCH D LOAD CC&S WITH FLIGHT PROGRAM
ATTITUDE REFERENCE SUBSYSTEM AUTOPILOT SUBSYSTEM REACTION CONTROL SUBSYSTEM (RCS)	SUBSYSTEM C/O AND STATUS MONITORING
MIDCOURSE CORRECTION PROPULSION SYSTEM	SUBSYSTEM C/O AND STATUS MONITORING
ELECTRICAL POWER SUBSYSTEM	SUBSYSTEM C/O AND STATUS MONITORING
S/C STRUCTURE SUBSYSTEM S/C MECHANISMS SUBSYSTEM INSTALLATIONS CABLES & TUBING	SUBSYSTEM C/O AND STATUS MONITORING
TEMPERATURE CONTROL SUBSYSTEM	A SUBSYSTEM C/O AND STATUS MONITORING B TEST S/C COOLING BY GROUND AIR SUPPLY
PYROTECHNIC SUBSYSTEM	SUBSYSTEM C/O AND STATUS MONITORING
ENGINEERING INSTRUMENTATION	SUBSYSTEM C/O AND STATUS MONITORING

2.0 LAUNCH & INJECTION (INCLUDES COUNTDOWN)	3.0 ACQUISITION	4.0 IN CR
MOS/LOS A. COMMAND LIFTOFF B. TRACK VEHICLE DURING BOOST C. SUPPLY FLIGHT COMMANDS (AS REQUIRED) D. RECEIVE AND ANALYZE DATA FROM S/C AND BOOSTER DSN A. STANDBY ON ALERT B. COMMUNICATE WITH ETR CHANGE FROM MOS/LOS TO DSN CONTROL UPON INJECTION INTO TRANS-MARS ORBIT C. PROVIDE SFCF/DSIF WITH ANTENNA SEARCH DATA DSN A. RECEIVE ANTENNA SEARCH DATA B. SEARCH FOR AND ACQUIRE S/C C. ESTABLISH AND VERIFY CONTROL OF S/C D. TRACK S/C (ONE-WAY) E. RECEIVE AND ANALYZE ENGINEERING DATA	DSN A. MONITOR BOOSTER SEPARATION B. MONITOR SOLAR PANELS DEPLOYMENT C. MONITOR ANTENNA DEPLOYMENT D. PROVIDE COMMANDS TO BACK UP CC&S AS REQUIRED E. TRACK S/C F. RECORD DATA G. MONITOR AND VERIFY ACQUISITION OF SUN H. MONITOR AND VERIFY ACQUISITION OF CANOPUS I. COMMAND REACQUISITION OF CANOPUS AS REQUIRED J. MONITOR S/C TRAJECTORY K. UPDATE CC&S TRAJECTORY PARAMETERS FOR PITCH L. COMPUTE CC&S PITCH & ROLL POLARITY	DSN A. TRACK SPACECRAFT B. RECEIVE AND DISPL C. MONITOR S/C STATU D. PROCESS DATA ON E COMMANDS (STORED
A. TRANSMIT ENGINEERING DATA VIA CENTAUR TELEMETRY B. TRANSMIT ENGINEERING DATA VIA LOW POWER LAUNCH EXCITER C. RECEIVE POWER FROM E/P D. RECORD ENGINEERING DATA	A. TRANSMIT ENGINEERING DATA VIA LOW POWER LAUNCH EXCITER B. RECEIVE POWER FROM E/P C. DETECT & SEND TO CC&S COMMAND SIGNALS FROM EARTH D. TRANSMIT CELESTIAL REFERENCE ACQUISITION TO DSN E. TRANSMIT VERIFICATION OF DEPLOYMENT OF SOLAR PANELS, ANT, ETC TO EARTH F. TWO-WAY TRACKING	A. TRANSMIT ENGINEERING D B. TRANSMIT CONE ANGLE SE C. EXERCISE AND CALIBRAT D. SWITCH FROM LOW-GAIN E. PROVIDE RANGING SIGNA F. SWITCH TRANSMISSION F AMPLIFIER AND DATA MO
PROVIDE BACKUP COMMANDS AS REQUIRED	A. COMMAND SIGNALS TO PYROTECHNICS AND MECHANISMS TO DEPLOY: SOLAR PANELS, HI-GAIN ANTENNA, VHF ANTENNA, LO-GAIN ANTENNA B. SWITCH ON GYROS & SELECT MODES C. RECEIVE SUN PRESENCE SIGNAL D. SWITCH ON BATTERY CHARGER E. TURN "ON" CANOPUS TRACKER F. SWITCH ROLL CONTROL TO CANOPUS SENSOR G. RECEIVE CANOPUS PRESENCE OUTPUT SIGNAL H. TRANSMIT VERIFICATION TO TELECOMMUNICATIONS SUBSYSTEM (CANOPUS & SUN) I. PERFORM COMMAND FUNCTIONS FOR CANOPUS OVERRIDE AS REQUIRED.	A. COMMAND TELECOM. TO DA B. BATTERY CHARGER ON AS C. COMMAND CRUISE ENGINE D. COMMAND DATA RECORDER E. COMMAND TWTA ON F. SWITCH CANOPUS ANGLE A G. INITIATE ENGINEERING DA H. COMMAND TELECOM. CHAN I. UPDATE HI-GAIN ANTENNA J. COMMAND TELECOM - CHA K. COMMAND RECALIBRATION
A. ATTITUDE REFERENCE-GYROS OFF DURING LAUNCH B. AUTOPILOT — OFF C. RCS — OFF	A. RECEIVE CC&S COMMAND, SWITCH ON GYROS B. DAMP ROTATION C. GYRO CONTROL, YAW AND PITCH SPACECRAFT TO ACQUIRE SUN D. RELAY SUN ACQUISITION SIGNAL TO CC&S E. TURN ON CANOPUS SENSOR (AUTOPILOT CONTROL) F. ROLL TO ACQUIRE CANOPUS G. RELAY ACQUISITION SIGNAL (CANOPUS) TO CC&S H. PERFORM CANOPUS OVERRIDE ROLL MANEUVER AS REQUIRED	A. UPDATE CANOPUS CONE ANG B. MAINTAIN S/C ATTITUDE TO
A. PROVIDE BATTERY POWER TO: • ENGINEERING INSTRUMENTATION • AUTOPILOT SUBSYSTEM • TELEMETRY SUBSYSTEM • CC&S • ATTITUDE REFERENCE B. PROVIDE ENGINEERING DATA FOR TELECOMMUNICATIONS SUBSYSTEM	A. ACTIVATE SOLAR POWER SYSTEM AFTER SUN ACQUISITION (AUTOMATIC) B. TRANSMIT "VOLTAGE SATSIFACTORY" SIGNAL TO CC&S C. PROVIDE POWER TO: • T/M • CC&S • TEMP CONTROL • AUTOPILOT • PYROTECHNICS • MECHANISM	A. PROVIDE SOLAR ELECTRIC • T/M • CC&S • ATTITUDE RE B. CHARGE BATTERIES
A. PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT	A. PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT B. DRIVE SOLAR PANELS TO LIMIT STOPS C. PROVIDE OUT AND LOCK SIGNALS TO T/M. D. DRIVE HI-GAIN ANTENNA TO OPERATING POSITION AND LOCK E. DRIVE VHF AND OMNI-ANTENNAS TO OPERATING POSITION AND LOCK	A. SUPPORT S/C ASSEMBLIES B. SUPPORT S/C COMPONENTS C. MAINTAIN ADEQUATE ALIGN D. PROVIDE ACCEPTABLE STA
A. PROVIDE HEAT SINK COOLING CAPABILITY TO SHROUD JETTISON B. TEMPERATURE CONTROL AFTER SHROUD JETTISON	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CO
	RECEIVE SQUIB FIRING SIGNALS FOR DEPLOYMENT OF: • SOLAR PANEL • LO-GAIN ANTENNA • HI-GAIN ANTENNA • VHF ANTENNA	
ENGINEERING INSTRUMENTATION "ON"	ENGINEERING INSTRUMENTATION "ON"	A. RECEIVE CRUISE ENGINEE B. ACTIVATE CRUISE ENGINE C. ACQUIRE DATA D. TRANSMIT DATA TO TELE E. RECEIVE POWER FROM E/F F. RECALIBRATE ENGINEERI

...TERPLANETARY ...RUISE → 5.0 INTERPLANETARY TRAJECTORY CORRECTION → 6.0 SIMULATE CAPSULE SEPARATION & PREPARE FOR MARS ENCOUNTER

[1] THESE SIGNALS SIMULATED TO CHECK COMMAND AND SEQUENCING CAPABILITY FOR 1971 MISSION

	5.0 INTERPLANETARY TRAJECTORY CORRECTION	6.0 SIMULATE CAPSULE SEPARATION & PREPARE FOR MARS ENCOUNTER	
...AY ENGINEERING DATA ...S & SIMULATE MONITORING OF CAPSULE ...ARTH TO OBTAIN GUIDANCE ... COMMANDS & START TIMES)	DSN A. PROVIDE ROLL TURN MAGNITUDE & DIRECTION B. PROVIDE YAW TURN MAGNITUDE & DIRECTION C. PROVIDE MIDCOURSE MOTOR BURN DURATION AND ΔV D. PROVIDE INITIATE MANEUVER SEQUENCE COMMAND E. RECEIVE VERIFICATION OF CORRECTION MANEUVER • DEGREES OF YAW AND ROLL • THRUST DURATION • REACQUISITION OF CELESTIAL REFERENCES F. TRACK SPACECRAFT	DSN A. INITIATE ENCOUNTER PREPARATION SEQUENCE B. RECEIVE SPACECRAFT ENGINEERING DATA C. INITIATE PRE-SEPARATION SEQUENCE [1] D. COMMAND SEPARATION [1] E. SIMULATE VERIFICATION OF CAPSULE PROGRAM [1] F. SIMULATE VERIFICATION OF SEPARATION [1] G. TRACK SPACECRAFT	D A B C D E F
...ATA VIA DATA MODE NO. 2 (OPTIONAL MODE NO. 3) ...TTINGS (CANOPUS TO EARTH) ... HI-GAIN ANTENNA T + 40 DAYS ...TO HIGH-GAIN ANTENNAS T + 80 DAYS ...L TO A MAX RANGE OF 8.0×10^8 km (NOMINAL) ...ROM LAUNCH EXCITER TO TWT POWER ...DE NO. 2	A. RECEIVE ROLL TURN COMMAND AND RELAY TO CC&S B. RECEIVE YAW TURN COMMAND AND RELAY TO CC&S C. RECEIVE MOTOR BURN DURATION AND ΔV, RELAY TO CC&S D. RECEIVE INITIATE MANEUVER COMMAND AND RELAY TO CC&S E. SWITCH TO DATA MODE NO. 1 (FOR LATE MIDCOURSE MANEUVERS) F. RELAY VERIFICATION OF CORRECTION MANEUVER TO DSN G. STORE AND/OR TRANSMIT ENGRG DATA TO DSN (MODE 3)	A. RECEIVE ENCOUNTER SEQUENCE COMMAND AND RELAY TO CC&S B. SWITCH TO DATA MODE NO. 2 C. SIMULATE TRANSMISSION OF CAPSULE ENGINEERING DATA TO DSN [1] D. RECEIVE PRE-SEPARATION SEQUENCE COMMAND AND RELAY TO CC&S [1] E. RECEIVE SEPARATION COMMAND FROM DSN & ROUTE TO CC&S [1] F. STORE AND/OR TRANSMIT ENGINEERING DATA TO DSN (MODE 3)	
...TA MODE NO. 2 ... REQUIRED ...ERING ON (WARMUP) ...S ON ...S REQUIRED ...TA ACQUISITION ...GE FROM OMNI TO HI-GAIN ANTENNA ... POSITION (AS REQUIRED) ...NGE DATA TRANSMISSION RATES ... OF ENGINEERING ELEMENTS AS REQUIRED	A. COMMAND TELECOM. TO DATA MODE NO. 1 B. RECEIVE TRAJECTORY CORRECTION PARAMETERS C. SWITCH GYROS TO RATE CONTROL D. COMMAND ANGULAR MANEUVER E. RECEIVE GYRO SIGNAL & SUM FOR ANGULAR MANEUVER F. SWITCH ATTITUDE REFERENCE TO AUTOPILOT SUBSYSTEM G. REMOVE PROPULSION INHIBIT H. PROVIDE COMMAND TO INITIATE ATTITUDE MANEUVER FOR ΔV APPLICATION I. RECEIVE ACCELEROMETER SIGNAL & INTEGRATE FOR ΔV MANEUVER J. DISARM PROPULSION SYSTEM K. COMMAND REVERSE ANGULAR MANEUVER L. COMMAND REACQUIRE CELESTIAL REFERENCES M. COMMAND AUTOPILOT SELECTION CONTROL N. BACKUP COMMANDS AS REQUIRED	A. LOAD CAPSULE WITH 5 OR MORE COMMANDS [1] B. COMMAND CAPSULE SEQUENCING [1] C. COMMAND TELECOM — SWITCH TO DATA MODE NO. 1 D. COMMAND DATA TAPE RECORDER "OFF" E. COMMAND GYROS ACTIVATE (CAPSULE) [1] F. COMMAND GYROS ESTABLISH POLARITY [1] G. COMMAND GYROS TO ATTITUDE CONTROL MODE [1] H. COMMAND CAPSULE ACTIVATE ENTRY SCIENCE [1] I. COMMAND CAPSULE ACTIVATE TELECOMM & VERIFY RADIO LINK (VHF) [1] J. SELECT T/M MODE FOR TRANSMISSION OF CAPSULE DATA [1] K. COMMAND [1] • BIOBARRIER SEPARATION • ELECTRICAL CONNECTION SEPARATION • S/C ORIENT TO SEPARATION ATTITUDE • VERIFY SEPARATION L. COMMAND REORIENT S/C & REACQUIRE CELESTIAL REFERENCES M. BACKUP COMMANDS AS REQUIRED N. RECEIVE S/C ORIENTATION PARAMETERS	
...LE UPON COMMAND ... CELESTIAL REFERENCES DURING CRUISE	A. SWITCH TO GYRO CONTROL B. ROLL S/C & VERIFY C. YAW S/C & VERIFY D. AUTOPILOT • PROVIDE COMMAND TO TVC DURING MOTOR BURN • TURN ON ACCELEROMETERS • PROVIDE ACCELERATION DATA TO CC&S • TURN OFF ACCELEROMETER E. YAW BACK TO ORIGINAL ATTITUDE & VERIFY F. REACQUIRE SUN (FINE SUN SENSOR) G. ROLL TO ORIGINAL ATTITUDE H. LOCK ON CANOPUS & VERIFY I. SWITCH AUTOPILOT FROM GYRO CONTROL TO CELESTIAL REFERENCE CONTROL	A. SWITH TO GYRO CONTROL B. PERFORM ROLL TURN & VERIFY C. PERFORM YAW TURN & VERIFY D. MAINTAIN ATTITUDE DURING SEPARATION E. PERFORM YAW TURN (INVERSE TO E) & VERIFY F. PERFORM ROLL TURN (INVERSE TO D) & VERIFY G. REACQUIRE CELESTIAL REFERENCES H. RETURN TO CELESTIAL REFERENCE ATTITUDE CONTROL MODE	
	A. ARM PRESSURIZATION SYSTEM (FIRE SQUIB VALVE) B. ARM PROPELLANT FEED SUBSYSTEM (FIRE SQUIB VALVE) C. FIRE PROGRAMMED ENGINES (OPERATE SOLENOID VALVE) D. SHUT DOWN ENGINE E. ISOLATE PROPELLANT FEED SUBSYSTEM F. ISOLATE PRESSURIZATION SUBSYSTEM		
...AL POWER • AUTOPILOT • TEMP CONTROL ...F	A. PROVIDE ELECTRIC BATTERY POWER TO: • T/M • REACTION CONTROL • CC&S • PROPULSION • AUTOPILOT • TEMP CONTROL • ATTITUDE REFERENCE • PYROTECHNIC	A. PROVIDE ELECTRICAL BATTERY POWER TO: • T/M • TEMP CONTROL • REACTION CONTROL • CC&S • PYROTECHNICS • AUTOPILOT • ATTITUDE REFERENCE	
... ...NMENT BETWEEN COMPONENTS ...TIC AND DYNAMIC LOAD ENVIRONMENTS	A. SUPPORT S/C ASSEMBLIES B. SUPPORT S/C COMPONENTS C. MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D. PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS	A. SUPPORT S/C ASSEMBLIES B. SUPPORT S/C COMPONENTS C. MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D. PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS	
...NTROL	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL	
	A. REMOVE PROPULSION SYSTEM INHIBIT B. PROVIDE IGNITION C. INHIBIT PROPULSION SYSTEM	A. RECEIVE SIGNAL TO REMOVE CAPSULE SEPARATION INHIBIT [1] B. RECEIVE SIGNAL TO FIRE CAPSULE UMBILICAL SEPARATION [1] C. RECEIVE SIGNAL TO FIRE BIO-BARRIER REMOVAL [1] D. RECEIVE SIGNAL TO FIRE CAPSULE SEPARATION [1] E. RECEIVE SIGNAL FOR SEPARATION INHIBIT [1]	
...RING DATA "ON" COMMAND ...ERING INSTRUMENTATION ...COMMUNICATIONS SUBSYSTEM ...NG DATA AS REQUIRED	ENGINEERING INSTRUMENTATION "ON"	ENGINEERING INSTRUMENTATION "ON"	

7.0 FLYBY MARS, ACQUIRE & TRANSMIT DATA → 8.0 COAST BEYOND MARS, ACQUIRE & TRANSMIT DATA, MAKE TRAJECTORY CORRECTION MANEUVER → 9.0 ANALYZE DATA & REPORT

7.0	8.0
SN . RECEIVE, DECODE, AND ANALYZE ENGINEERING DATA . TRACK SPACECRAFT . INITIATE ENCOUNTER ENGRG INSTRUMENTATION . VERIFY FLY BY PROGRAM SEQUENCING IN OPERATION . VERIFY POSITION OF SCAN PLATFORM . PROVIDE COMMANDS TO BACK UP CC&S AS REQUIRED	DSN A. RECEIVE, DECODE AND ANALYZE ENGINEERING DATA B. TRACK SPACECRAFT C. RECEIVE VERIFICATION OF TRAJECTORY CORRECTION MANEUVER D. RECEIVE VERIFICATION OF ELECTRIC POWER SWITCH E. RECEIVE PROGRAMMED TELEMETRY SIGNAL TO MONITOR SIGNAL STRENGTH DECAY
A. RELAY CELESTIAL REFERENCE ACQUISITION TO DSN B. TRANSMIT ENGINEERING DATA VIA DATA MODE 3, 4, 5. C. RECORD ENGINEERING DATA AS COMMANDED D. UPDATE HI-GAIN ANTENNA POSITION AS COMMANDED E. TRANSMIT PROGRAMED DATA THROUGH MARS ATMOSPHERE TO DSN TO MONITOR ATTENUATION	A. RELAY CELESTIAL REFERENCE ACQUISITION TO DSN B. TRANSMIT ENGINEERING DATA VIA DATA MODE 3, 4, 5 C. RECORD ENGINEERING DATA AS COMMANDED D. UPDATE HI-GAIN ANTENNA POSITION AS COMMANDED E. RECEIVE AND RELAY COMMAND FOR MISSION TERMINATION F. TRANSMIT PROGRAMMED DATA ON COMMAND TO ALLOW DSN MONITOR OF SIGNAL STRENGTH DECAY G. RECEIVE AND STORE DATA FOR TRAJECTORY CORRECTION MANEUVER
A. COMMAND UPDATING OF HI-GAIN ANTENNA POSITION B. PROVIDE SIGNAL TO PYROTECHNICS & MECHANISMS TO DEPLOY SCAN PLATFORM C. SWITCH DATA MODES AS NECESSARY D. SELECT RECORDED ENGINEERING DATA READOUT TO TELECOM E. SWITCH ATTITUDE CONTROL TO OCCULTATION MODE AS REQUIRED F. COMMAND SHIFT FROM SOLAR PANEL TO BATTERY POWER G. COMMAND SIMULATION OF TV CAMERA RELAY H. BACKUP COMMANDS AS REQUIRED	A. COMMAND UPDATING OF HI-GAIN ANTENNA POSITION B. SWITCH DATA MODES AS NECESSARY C. SELECT PROGRAMMED DATA FOR TRANSMITTAL TO MONITOR SIGNAL DECAY D. SWITCH ATTITUDE CONTROL TO OCCULTATION MODE AS REQUIRED E. COMMAND TRAJECTORY CORRECTION MANEUVER F. COMMAND SWITCH FROM BATTERY TO SOLAR PANEL G. BACK UP COMMANDS AS REQUIRED
A. MAINTAIN SOLAR LOCK AND REACQUIRE FOLLOWING OCCULTATION B. MAINTAIN CANOPUS LOCK AND REACQUIRE FOLLOWING OCCULTATION C. UPDATE CANOPUS ANGLE ON COMMAND	A. POSITION S/C FOR TRAJECTORY CORRECTION MANEUVER AS REQUIRED B. UPDATE CANOPUS ANGLE ON COMMAND C. REACQUIRE REFERENCES FOLLOWING TRAJECTORY CORRECTION MANEUVER
	A. ARM PRESSURIZATION & PROPELLANT FEED SUBSYSTEMS B. PROVIDE THRUST (FIRE PROGRAMMED ENGINES) FOR TRAJECTORY CORRECTION MANEUVER IF REQUIRED C. TERMINATE THRUST ON COMMAND D. ISOLATE PROPELLANT FEED SUBSYSTEM E. ISOLATE PRESSURIZATION SUBSYSTEM
A. PROVIDE ELECTRICAL SOLAR POWER TO: • T/M • AUTOPILOT • REACTION CONTROL • CC&S • ATTITUDE REFERENCE • TEMP CONTROL B. CHARGE BATTERIES	A. PROVIDE ELECTRIC SOLAR POWER TO ALL SYSTEMS FROM SOLAR PANELS • T/M • ATTITUDE REFERENCE • TEMP CONTROL • AUTOPILOT • CC&S • MIDCOURSE PROPULSION • PYROTECHNICS B. CHARGE BATTERIES
A. PROVIDE LOCKING OF SCAN PLATFORM IN EXTENDED POSITION B. PROVIDE TV CAMERA RELAY ACTUATION SIMULATION C. PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT D. SUPPORT S/C ASSEMBLIES & COMPONENTS E. MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS F. PROVIDE ACCEPTABLE STATIC & DYNAMIC LOAD ENVIRONMENTS	PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT
PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL
RECEIVE SQUIB FIRING SIGNAL FOR UNLATCHING OF SCAN PLATFORM	A. REMOVE PROPULSION SYSTEM INHIBIT B. PROVIDE IGNITION C. INHIBIT PROPULSION SYSTEM
A. RECEIVE ENCOUNTER ENGRG DATA "ON" COMMAND B. ACTIVATE ENCOUNTER ENGRG INSTRUMENTATION C. ACQUIRE DATA D. TRANSMIT DATA TO TELECOMMUNICATION SUBSYSTEM E. RECEIVE POWER FROM E/P F. RECALIBRATE ENGRG INSTRUMENTATION AS REQUIRED	ENGINEERING INSTRUMENTATION "ON"

Figure II-6 : 1969 Atlas / Centaur Flyby

4

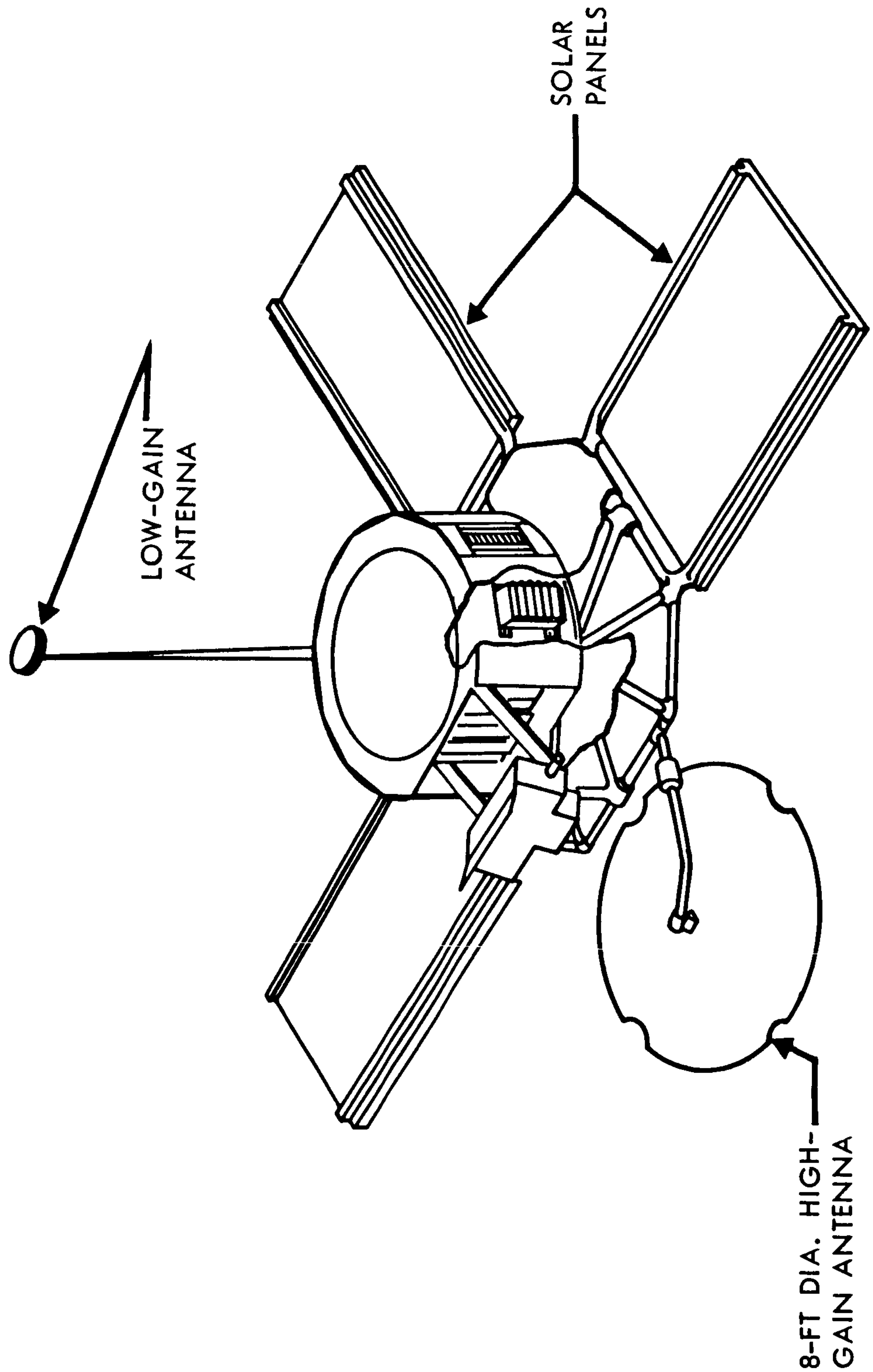
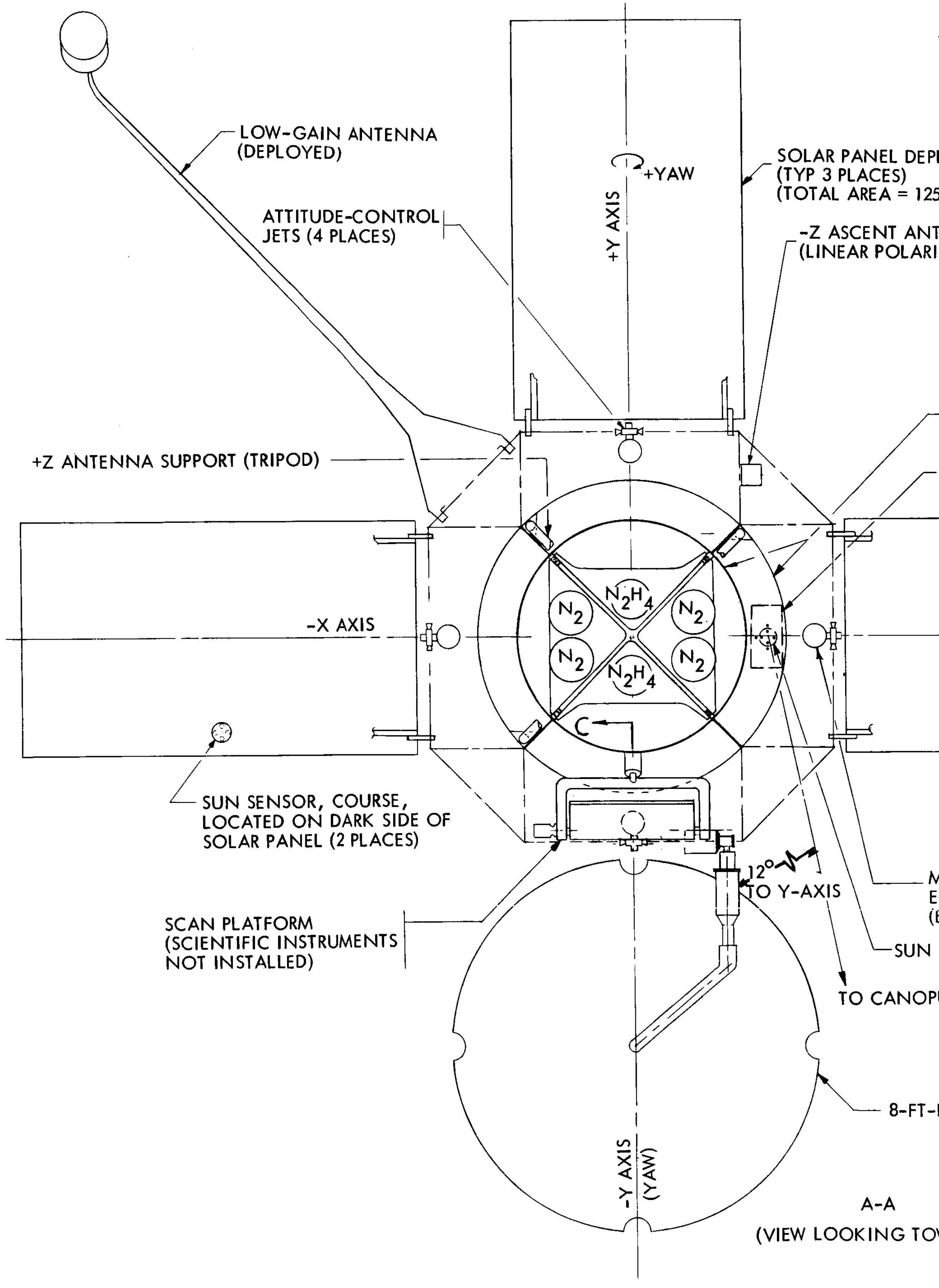


Figure II-7: 1969 Test Spacecraft — Atlas/Centaur

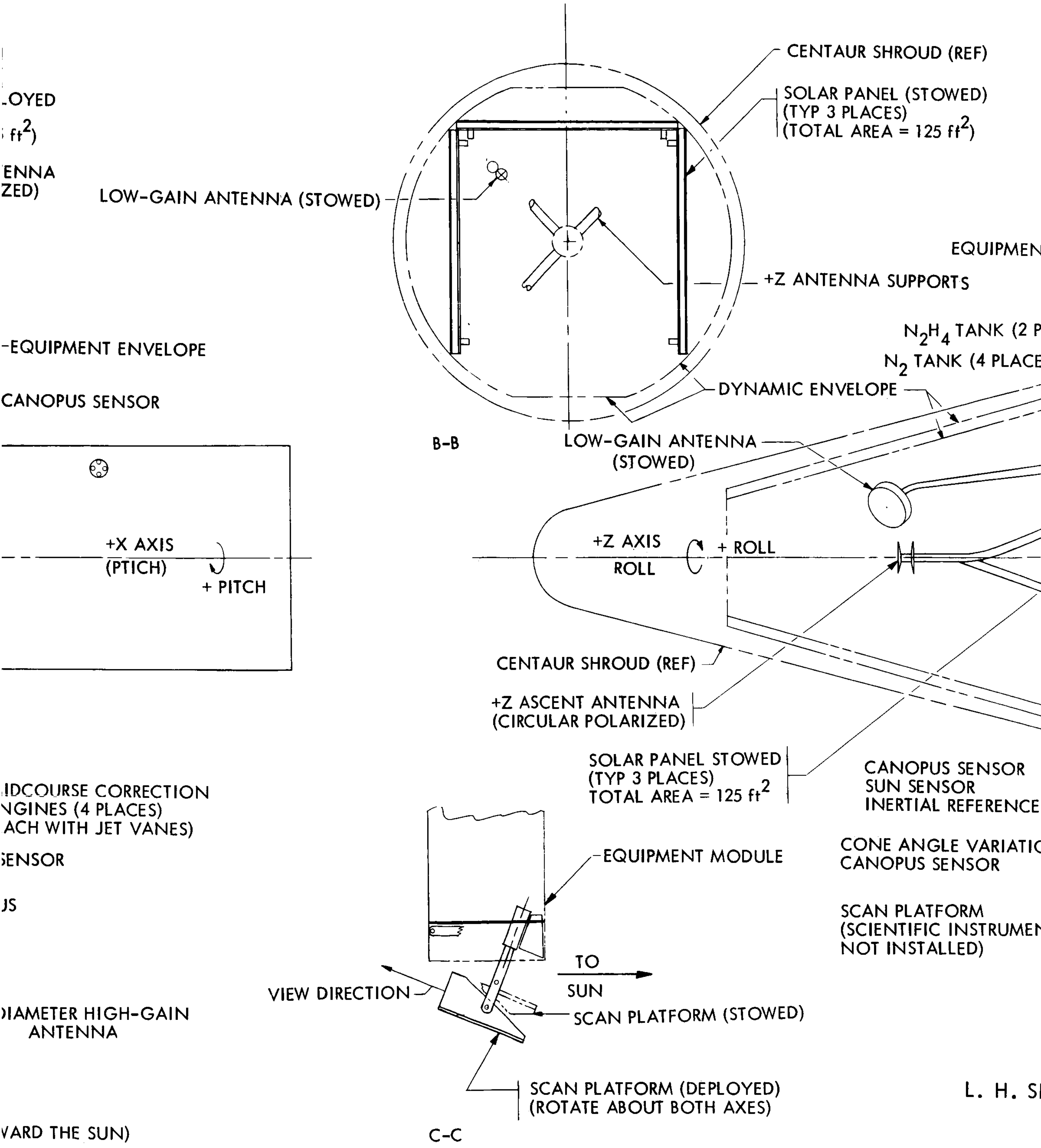
II-3.10.1 Configuration

Figure II-8 shows the spacecraft configuration. Figure II-9 is the inboard profile drawing. The spacecraft hardware is identical to that of the 1971 spacecraft with the following exceptions:

1) The orbit insertion engine and its heat shield are deleted;
2) The VHF antenna and relay radio are deleted;
3) The hydrazine and nitrogen tanks are reduced in size and are made into an assembly that occupies the same location and mounting as the deleted orbit insertion motor. This assembly is identified as a part of the propulsion module. The 1971 tank supports are omitted. Figure II-10 is a sketch of the 1969 propulsion module. Schematically, the plumbing and wiring diagrams are identical with the 1971 configuration;
4) The solar shield is moved forward to clear the ascent antenna (discussed later). The number of solar panels is reduced from six to three; however, identical panels are used. Figure II-11 shows how the 1971 panels are attached to separate spars having hinges adapted to the 1969 Test Spacecraft. The orientation of the panels during boost is similar for both spacecraft in that the plane of the panel is generally parallel to the booster thrust direction; however, as shown the 1969 panels are rotated 90 degrees to stay within the limits of the nose fairing.
5) The scan platform scientific instruments (TV cameras, scanner, and IR spectrometer) are removed; however, the platform, its actuators and electronics are retained. The platform envelope is reduced in size to fit within the envelope of the Centaur shroud; however, the louver area is not changed. The louver area is maintained so that an



LOW-GAIN ANTENNA
(DEPLOYED)
+YAW
+Y AXIS
SOLAR PANEL DEPL
(TYP 3 PLACES)
(TOTAL AREA = 125
ATTITUDE-CONTROL
JETS (4 PLACES)
-Z ASCENT ANT
(LINEAR POLARI
+Z ANTENNA SUPPORT (TRIPOD)
N_2H_4
N_2
N_2
N_2
N_2
N_2H_4
-X AXIS
C
SUN SENSOR, COURSE,
LOCATED ON DARK SIDE OF
SOLAR PANEL (2 PLACES)
12°
TO Y-AXIS
SCAN PLATFORM
(SCIENTIFIC INSTRUMENTS
NOT INSTALLED)
SUN
TO CANOPU
8-FT-
-Y AXIS
(YAW)
A-A
(VIEW LOOKING TO



CENTAUR SHROUD (REF)
SOLAR PANEL (STOWED)
(TYP 3 PLACES)
(TOTAL AREA = 125 ft^2)
-OYED
ENNA
ZED)
LOW-GAIN ANTENNA (STOWED)
EQUIPMEN
+Z ANTENNA SUPPORTS
N_2H_4 TANK (2 P
N_2 TANK (4 PLACE
-EQUIPMENT ENVELOPE
DYNAMIC ENVELOPE
CANOPUS SENSOR
B-B
LOW-GAIN ANTENNA
(STOWED)
+X AXIS
(PTICH)
+ PITCH
+Z AXIS
ROLL
+ ROLL
CENTAUR SHROUD (REF)
+Z ASCENT ANTENNA
(CIRCULAR POLARIZED)
SOLAR PANEL STOWED
(TYP 3 PLACES)
TOTAL AREA = 125 ft^2
CANOPUS SENSOR
SUN SENSOR
INERTIAL REFERENCE
IDCOURSE CORRECTION
NGINES (4 PLACES)
ACH WITH JET VANES)
SENSOR
JS
EQUIPMENT MODULE
CONE ANGLE VARIATIC
CANOPUS SENSOR
SCAN PLATFORM
(SCIENTIFIC INSTRUMEN
NOT INSTALLED)
TO
SUN
VIEW DIRECTION
SCAN PLATFORM (STOWED)
IAMETER HIGH-GAIN
ANTENNA
SCAN PLATFORM (DEPLOYED)
(ROTATE ABOUT BOTH AXES)
L. H. SI
VARD THE SUN)
C-C

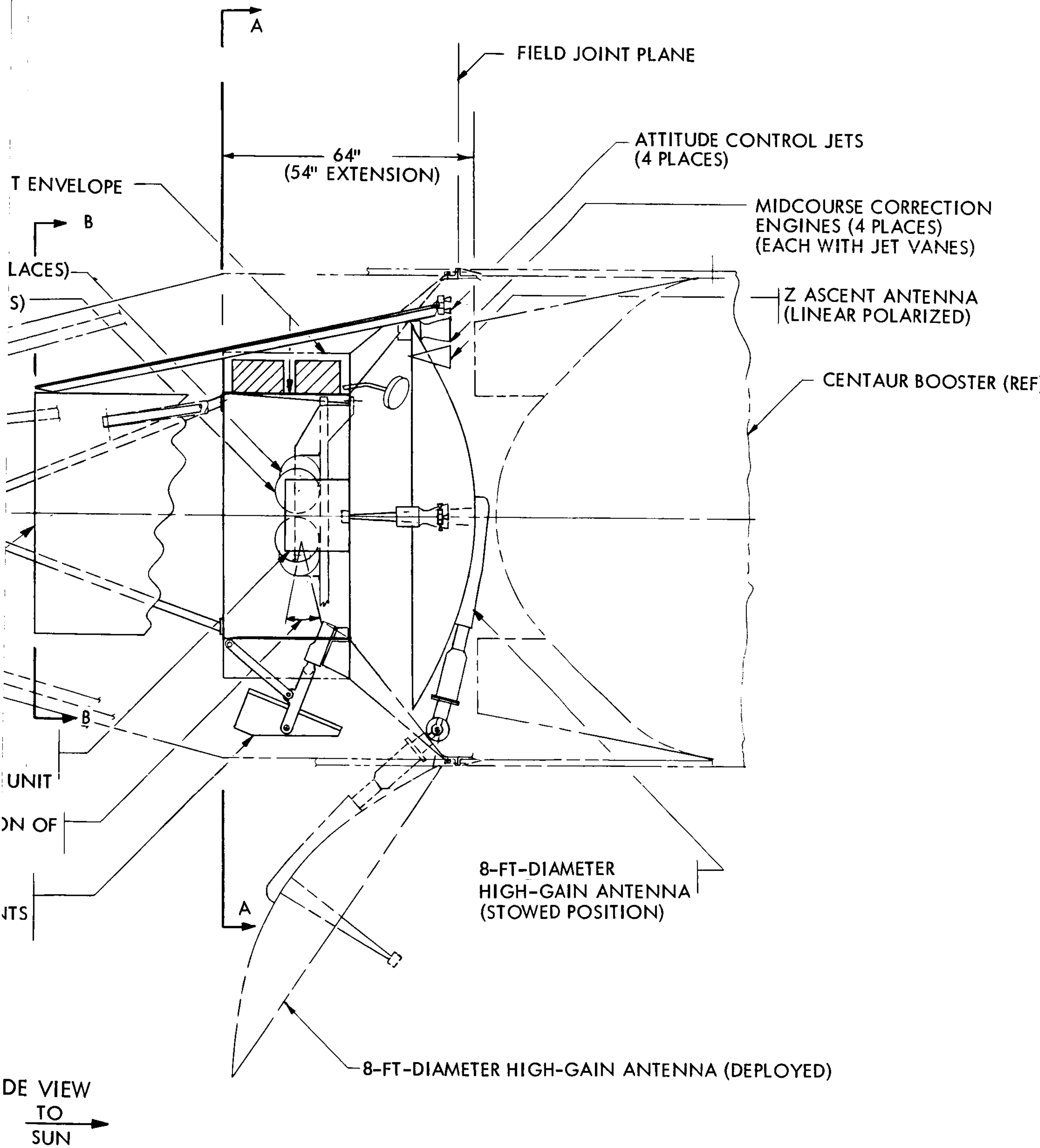
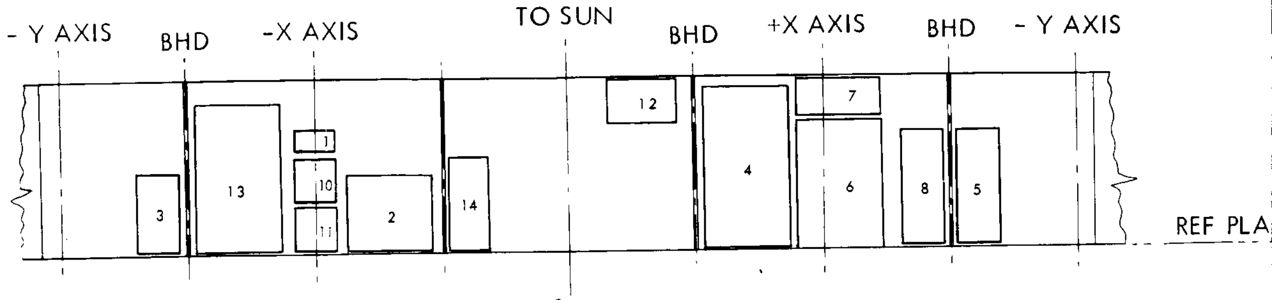
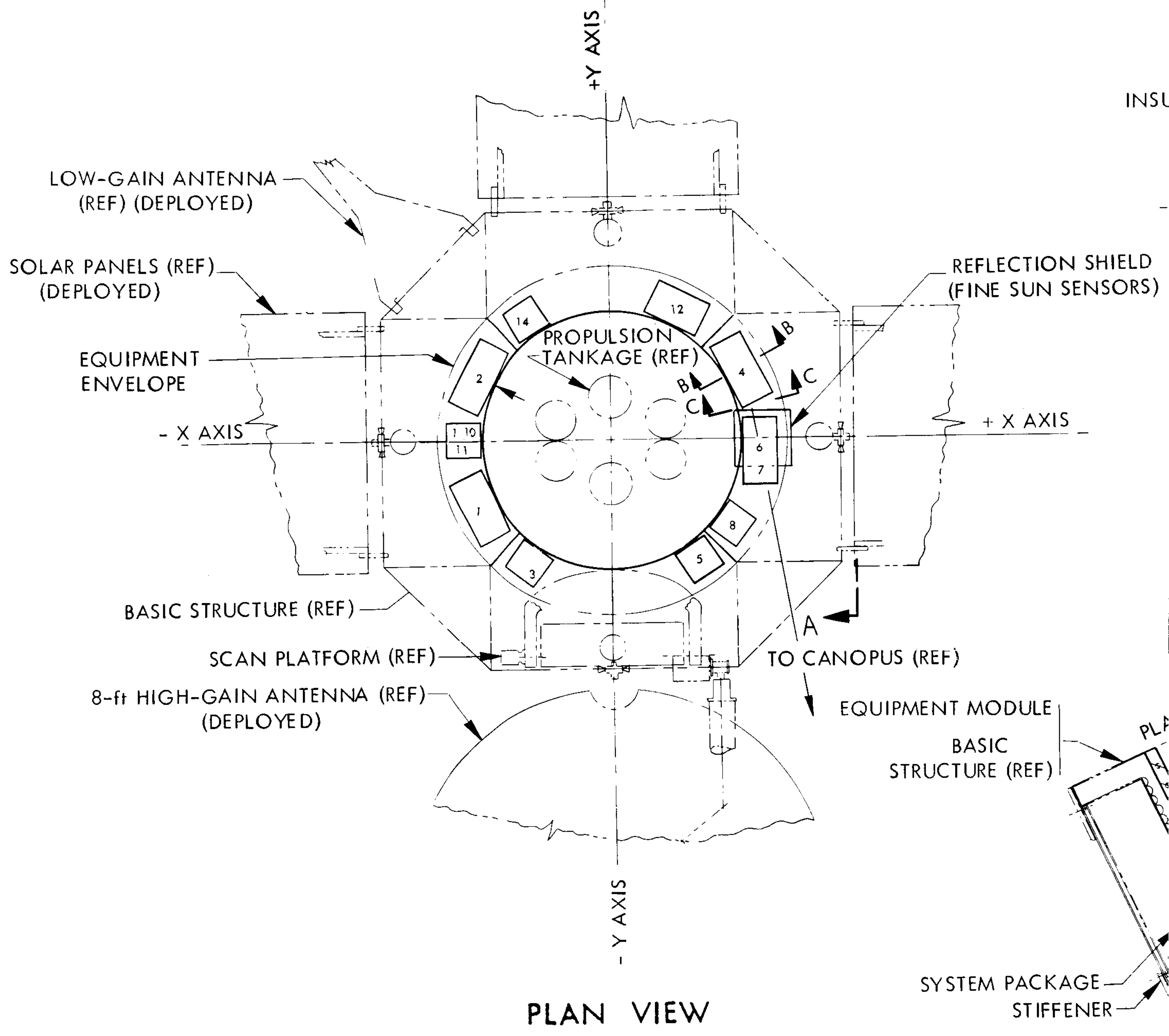


Figure II-8: Configuration — 1969 Test Spacecraft — Atlas/Centaur

3

DEVELOPED VIEW OF ELECTRONIC PACKAGING





PLAN VIEW

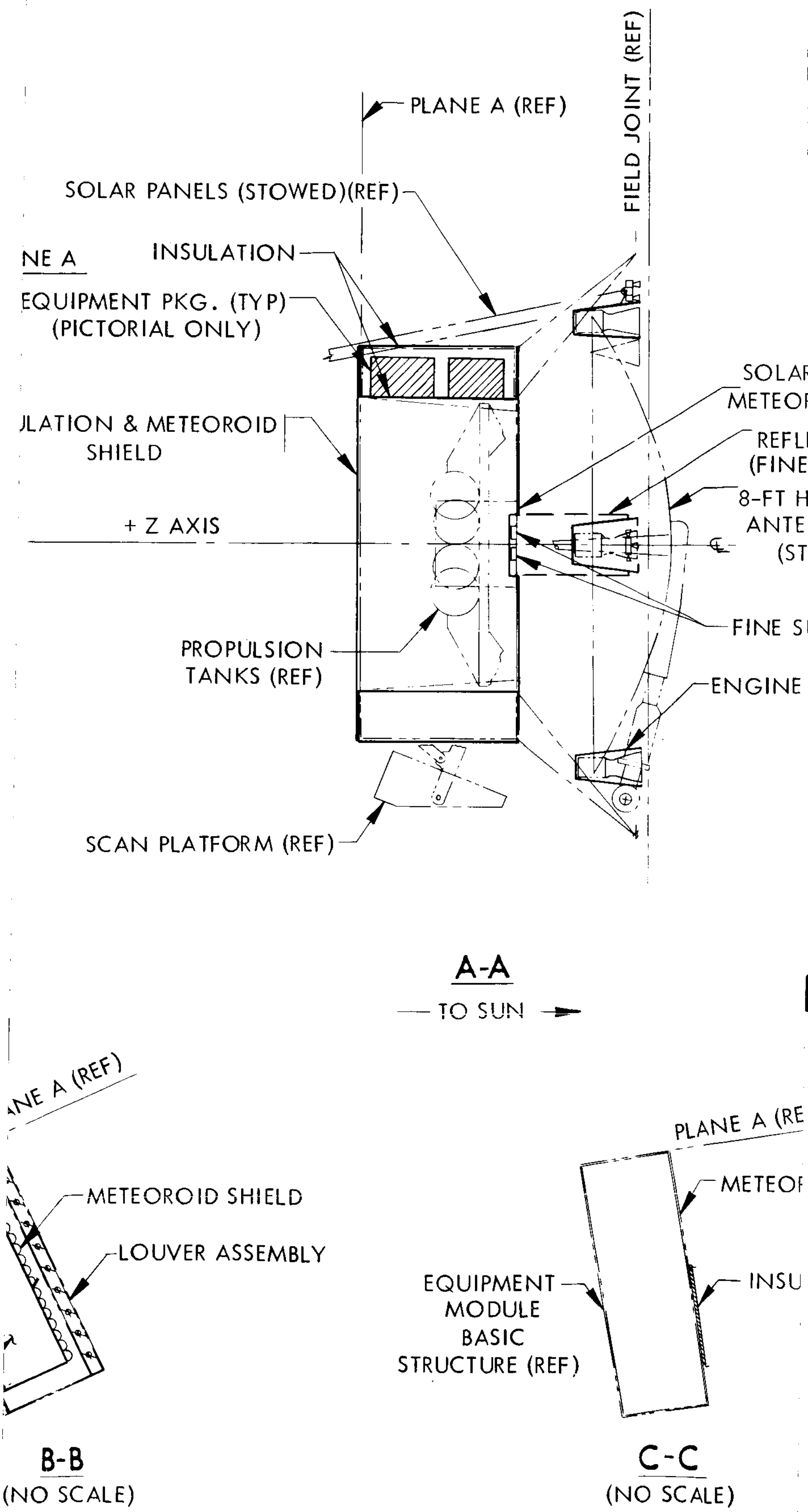


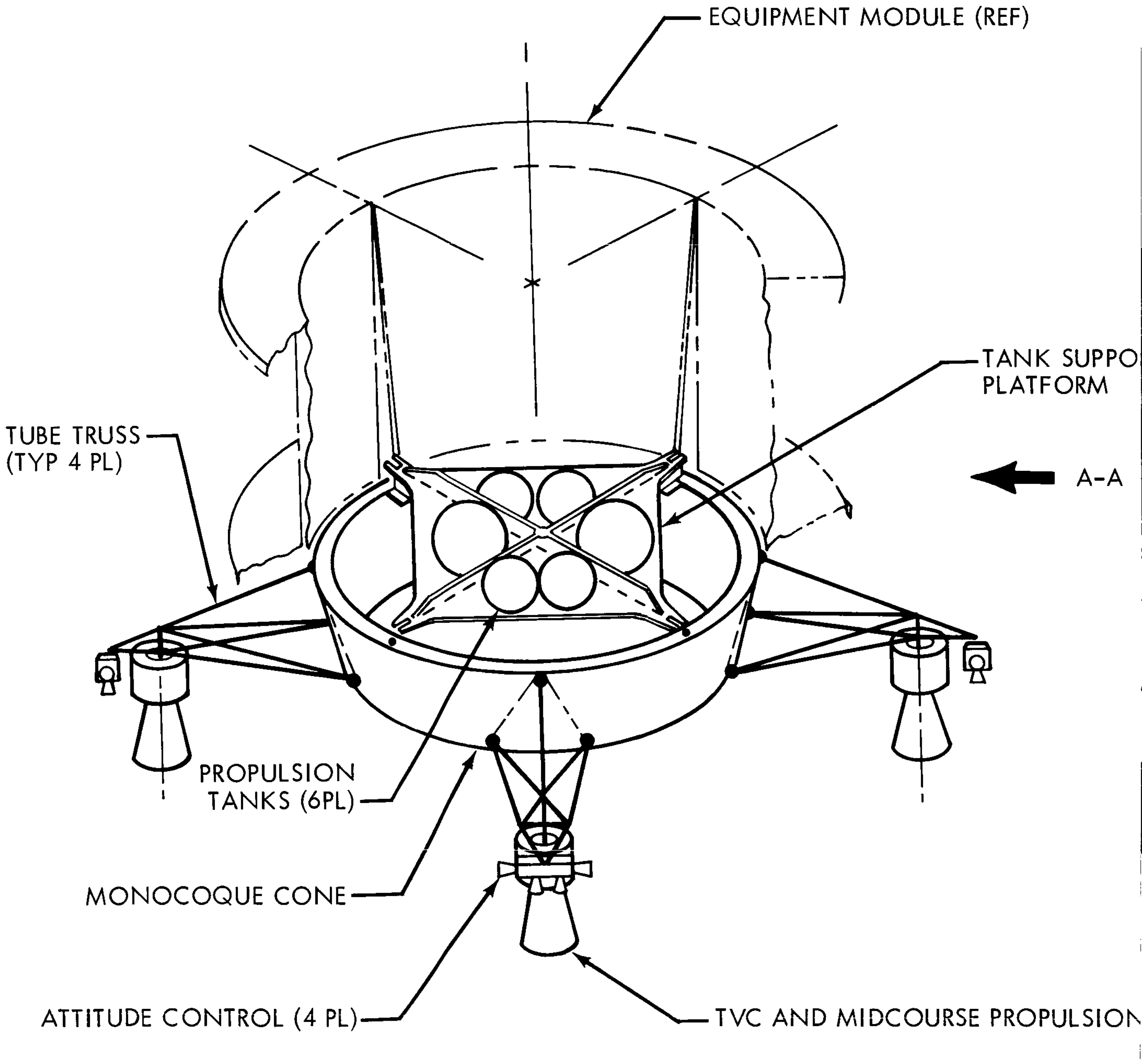
Figure II-9: Inboard Pr

2

: SHIELD &
ROID SHIELD
ECTION SHIELD
SUN SENSORS)
IGH-GAIN
NNA (REF)
OWED)
JN SENSORS
SHIELD (TYP)
OID SHIELD
.ATION

BOX	ITEM	SUBSYSTEMS
1		RELAY-RADIO SUBSYSTEM
	1	DETECTOR & BIT SYNCHRONIZER
	2	RECEIVER
2	1	TELEMETRY & DATA STORAGE
	2	PLANETARY SCIENCE TAPE RECORDER
	3	TAPE RECORDER 1
3	1	TELEMETRY PROCESSOR
	2	ENG. CORE MEMORY
	3	CAPSULE CORE MEMORY
	4	MULTIPLEXER-ENCODER & MODULATOR
	5	POWER SUPPLY
4		SPACECRAFT RADIO
	1	RF POWER AMPLIFIER
	2	POWER SUPPLY
	3	CIRCULATOR SWITCH
	4	PREAMPLIFIER
	5	DI-PLEXER
	6	PRESELECTOR
	7	INSULATOR
	8	TWT AMPLIFIERS 1 & 2
	9	INSULATOR (3)
	10	BANDPASS FILTER (3)
	11	BAND STOP
	12	HYBRID
	13	CIRCULATOR SWITCH
	14	STRUCTURE
5	1	RECEIVERS & EXCITERS (2)
	2	RECEIVERS (2)
	3	COMMAND DETECTORS (2)
	4	RANGING UNIT (2)
	5	EXCITER & LAUNCH TRANSMITTER
	6	LAUNCH TRANSMITTER
	7	REDUNDANCY CONTROL & LOGIC
6		ATTITUDE REFERENCE
	1	IRU
	2	CANOPUS
	3	SUN SENSOR
7		AUTOPILOT
	1	SYNCHRONIZER
8		CENTRAL COMPUTER & SEQUENCER
	1	CONTROL ASSEMBLY
	2	SWITCHING ASSEMBLY
9		DELETED
10		BATTERY
11		BATTERY
12	1	BATTERY-CHARGER BOOSTER ASSEMBLY
	2	P.S. & L
	3	
	4	SENSOR
13	1	D.C./D.C. REGULATOR
	2	FAILURE SENSOR
14	1	D.C./A.C. 2400 INVERTER
	2	FAILURE SENSOR D.C./A.C.
	3	D.C./A.C. 400 ~ 1θ

file — 1969 Test Spacecraft — Atlas/Centaur


EQUIPMENT MODULE (REF)
TANK SUPPO
PLATFORM
A-A
TUBE TRUSS
(TYP 4 PL)
PROPULSION
TANKS (6PL)
MONOCOQUE CONE
ATTITUDE CONTROL (4 PL)
TVC AND MIDCOURSE PROPULSION

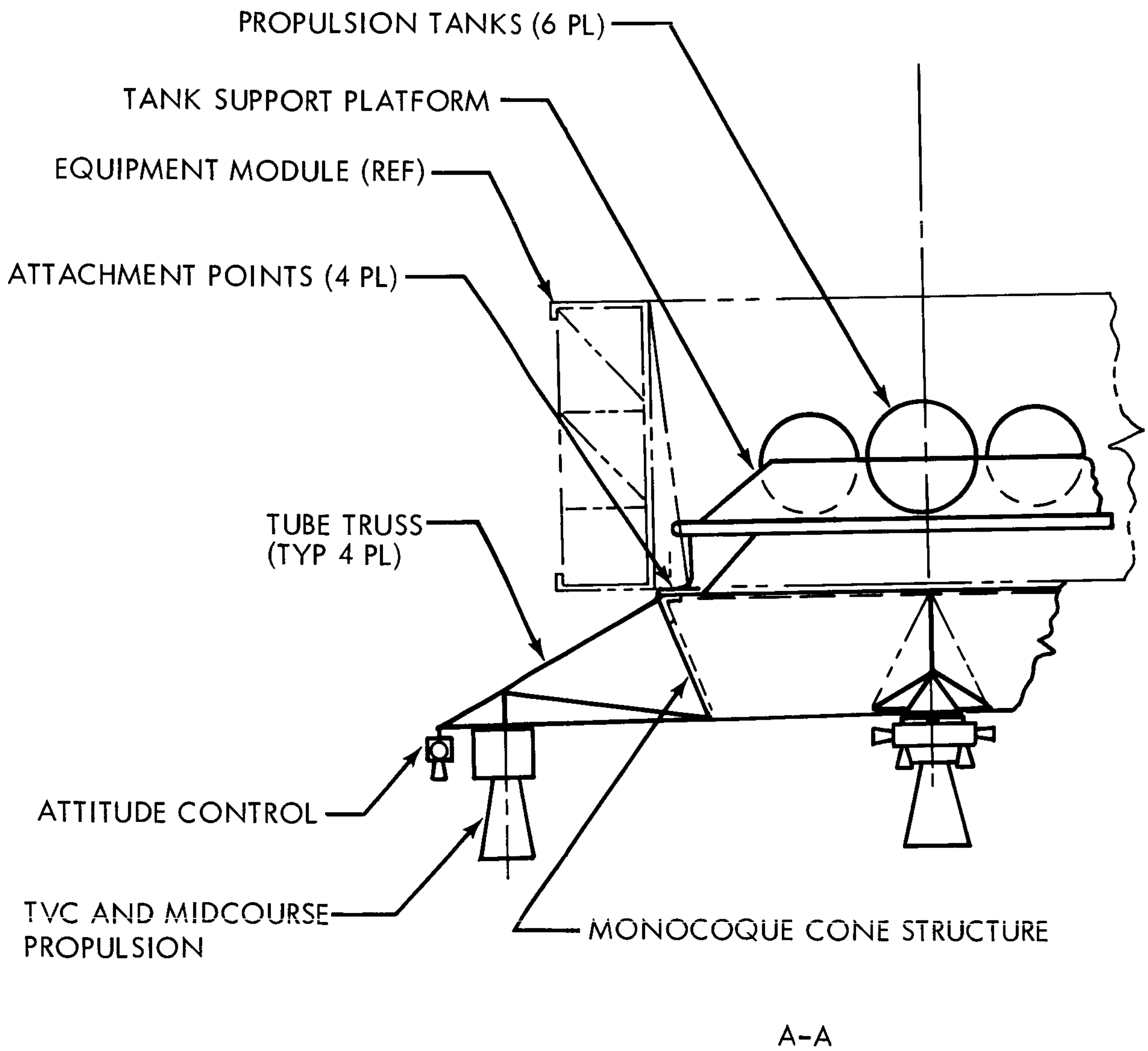
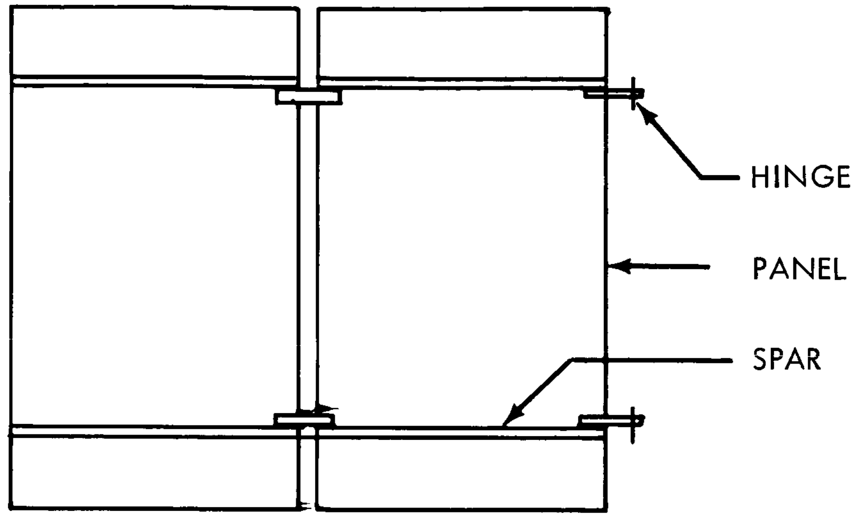
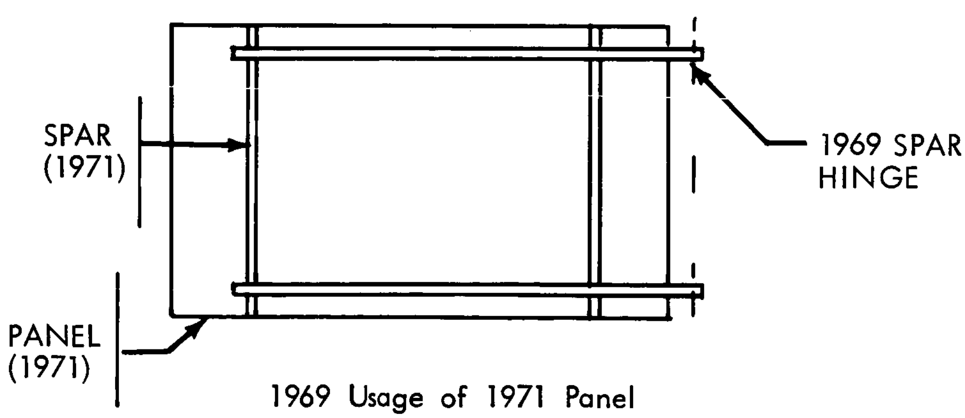


Figure II-10: Propulsion Module — 1969 Test Spacecraft — Atlas/Centaur

V



1971 SOLAR PANEL



1969 Usage of 1971 Panel

Figure II-11: Solar Panel Attachment — 1969 Test Spacecraft — Atlas/Centaur accurate environmental control test of the platform package can be made; there is ample storage battery capacity to supply the simulated heat load for a reasonable time period. Since the solar panels are adequate for battery recharge, more than one test can be made.

6) The science payload, data automation equipment, magnetometer boom, one battery, and one data recorder are deleted.

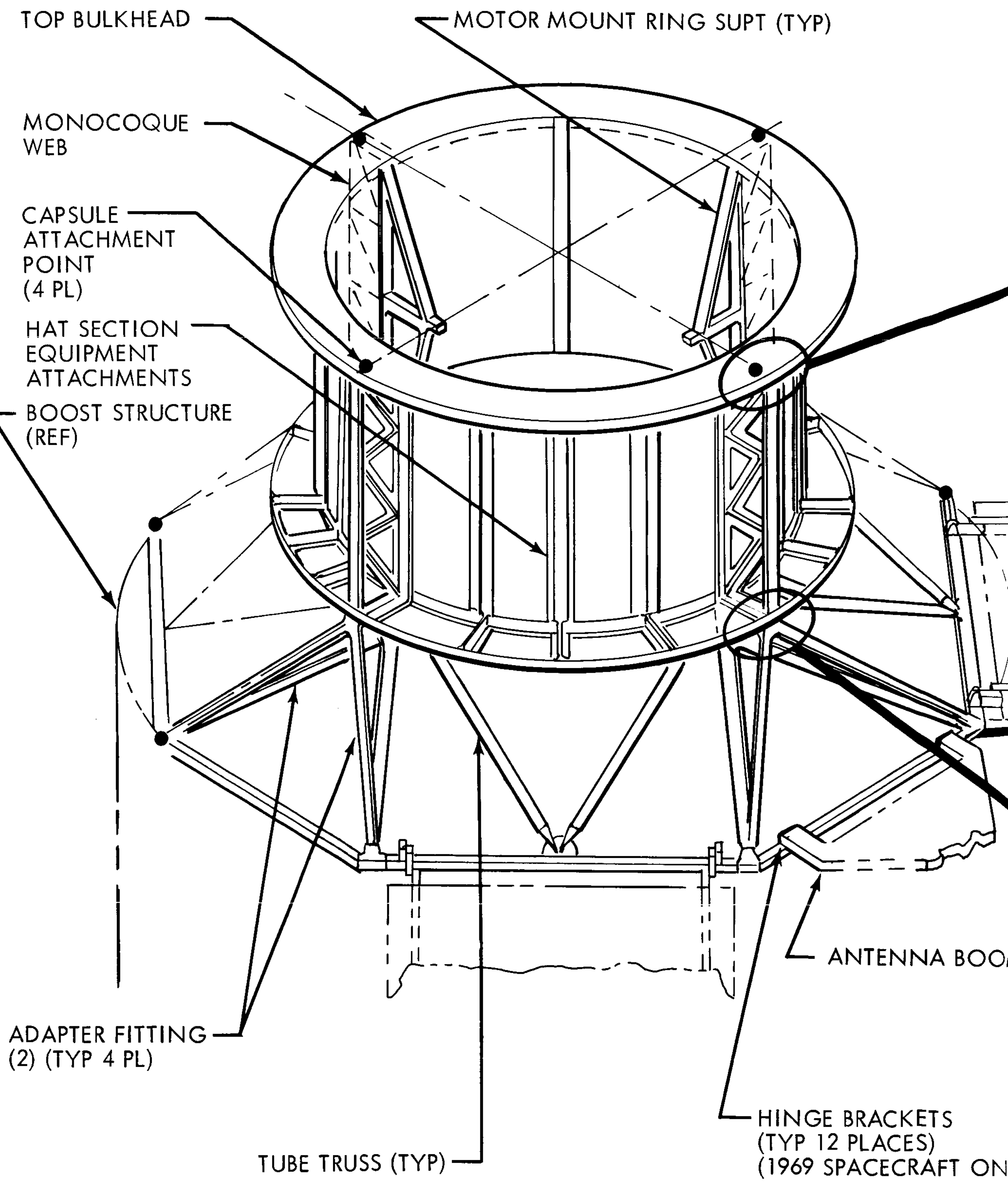
7) Equipment boxes identical with the 1971 boxes are located as shown in Figure II-9. This rearrangement is necessary to maintain the center-of-gravity location.

8) The high-gain antenna size is reduced to an 8-foot diameter, its stowed position is changed to fit within the nose fairing envelope. The antenna deployment mechanisms are identical but the supporting linkages are changed.

9) The low-gain antenna boom is shortened to maintain the same relative position to the reduced size solar panel outer corners. This is done in order to more nearly simulate the 1971 configuration's rf reflective pattern. The deployment mechanisms and number of hinge lines are identical.

10) Figure II-12 is an isometric drawing of the structural design for the 1969 Test Spacecraft. The structure is identical to that of the 1971 spacecraft except as follows:

    a) The structure lying outside a 10-foot diameter is omitted; these omissions are the 3-boom supports and the 3-solar panel hinge points. Note that hard points of the 1971 spacecraft structure which are within a 10-foot diameter support the three deployable antennas and the three solar panels of the 1969 Test Spacecraft. Hinge brackets are added to the members connecting these hard points.



TOP BULKHEAD
MOTOR MOUNT RING SUPT (TYP)
MONOCOQUE WEB
CAPSULE ATTACHMENT POINT (4 PL)
HAT SECTION EQUIPMENT ATTACHMENTS
BOOST STRUCTURE (REF)
ANTENNA BOO
ADAPTER FITTING (2) (TYP 4 PL)
HINGE BRACKETS (TYP 12 PLACES) (1969 SPACECRAFT ON
TUBE TRUSS (TYP)

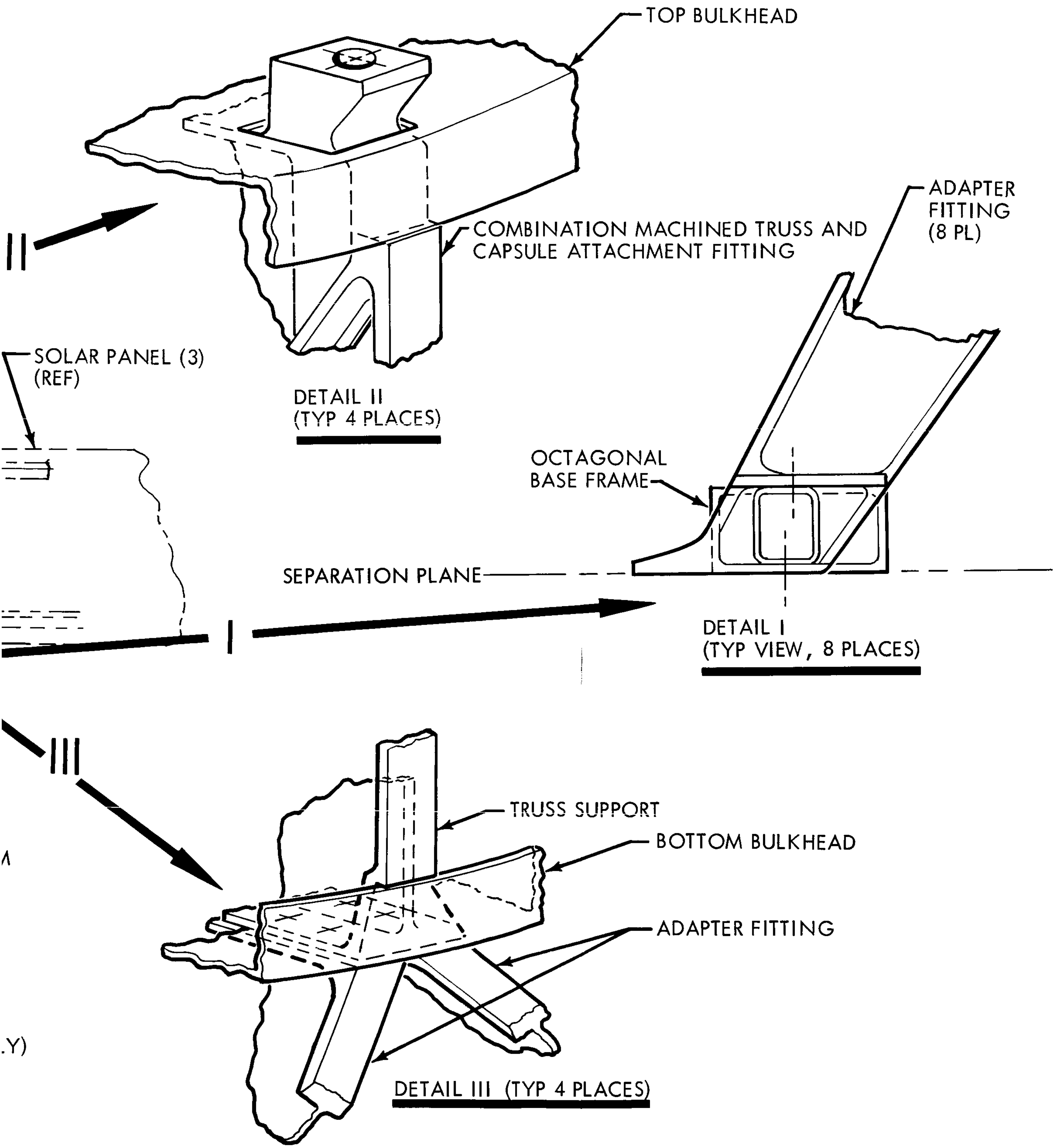


Figure II-12: Structure Module 1969 Test Spacecraft — Atlas/Centaur

2

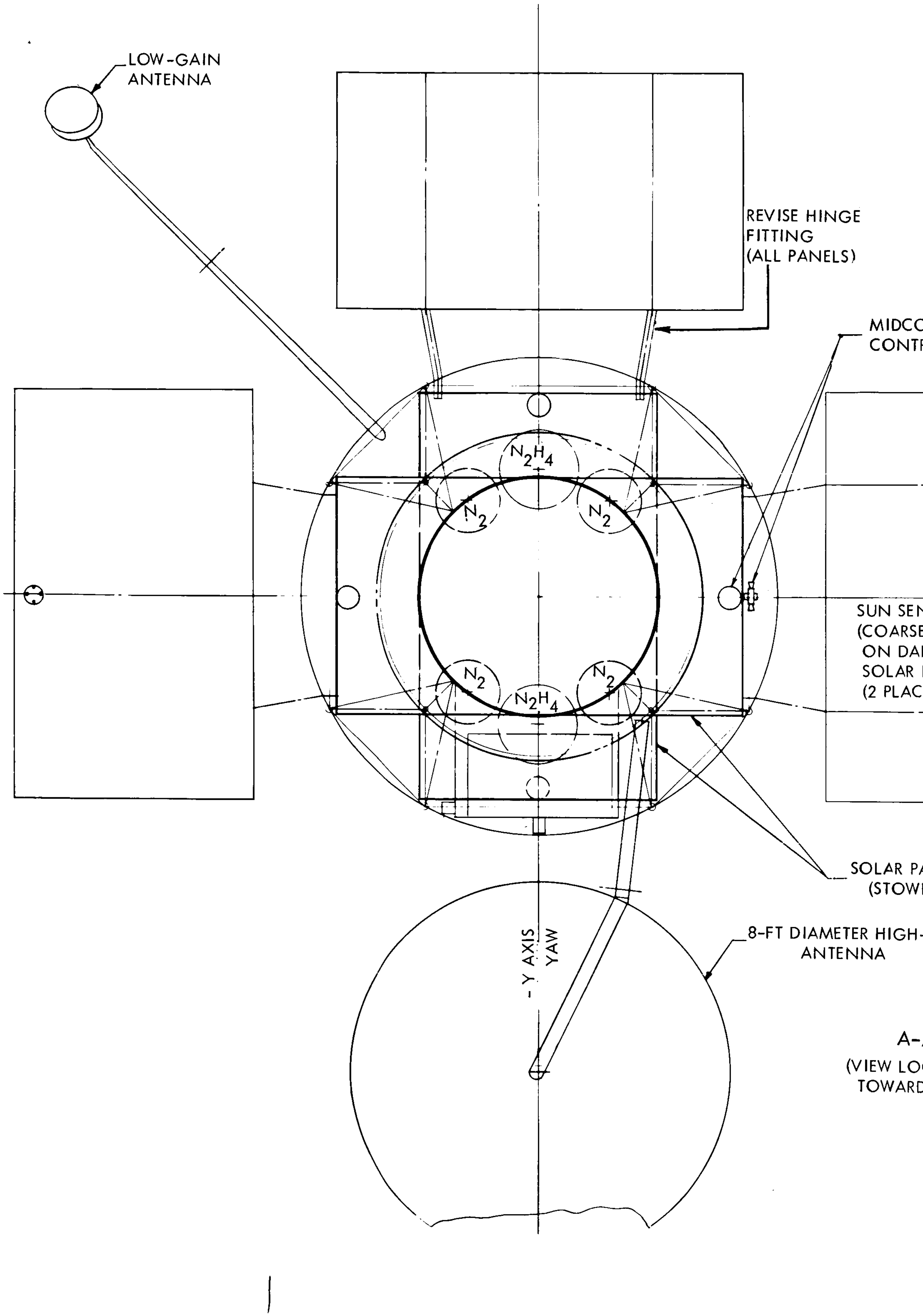
11) The +Z ascent antenna occupies the 1971 Flight Capsule location. The antenna support consists of a tripod assembly using three of the four Flight Capsule mountings to the spacecraft. One leg of the tripod is a waveguide; it fastens to the mount adjacent to the antenna electronics.

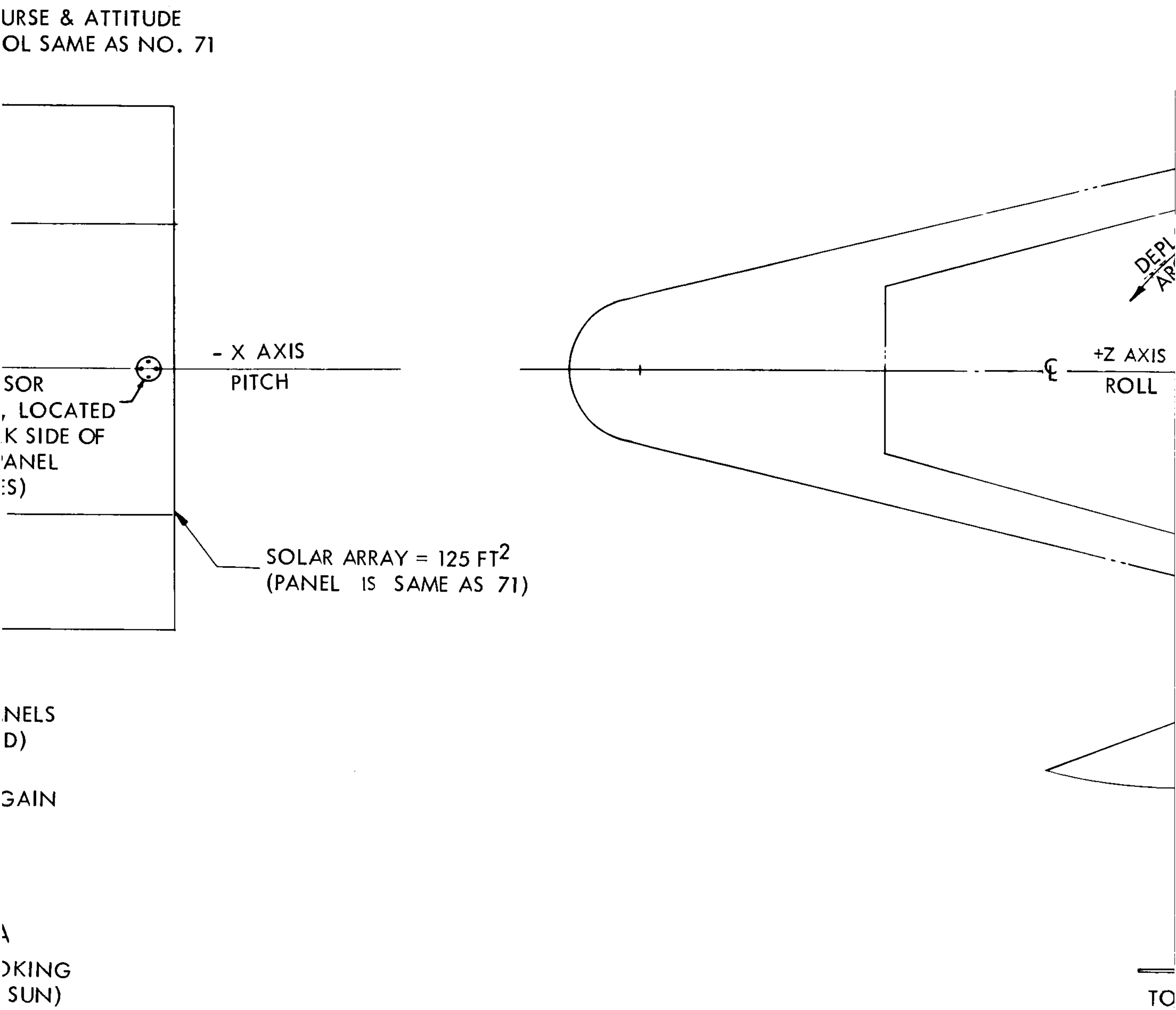
II-3.10.2 Center of Gravity Consideration (See Mass Properties, Section II-4.4.7)

With the differences discussed above, the center-of-gravity location is maintained within the 1-inch envelope allowed for the payload of the Atlas/Centaur launch vehicle.

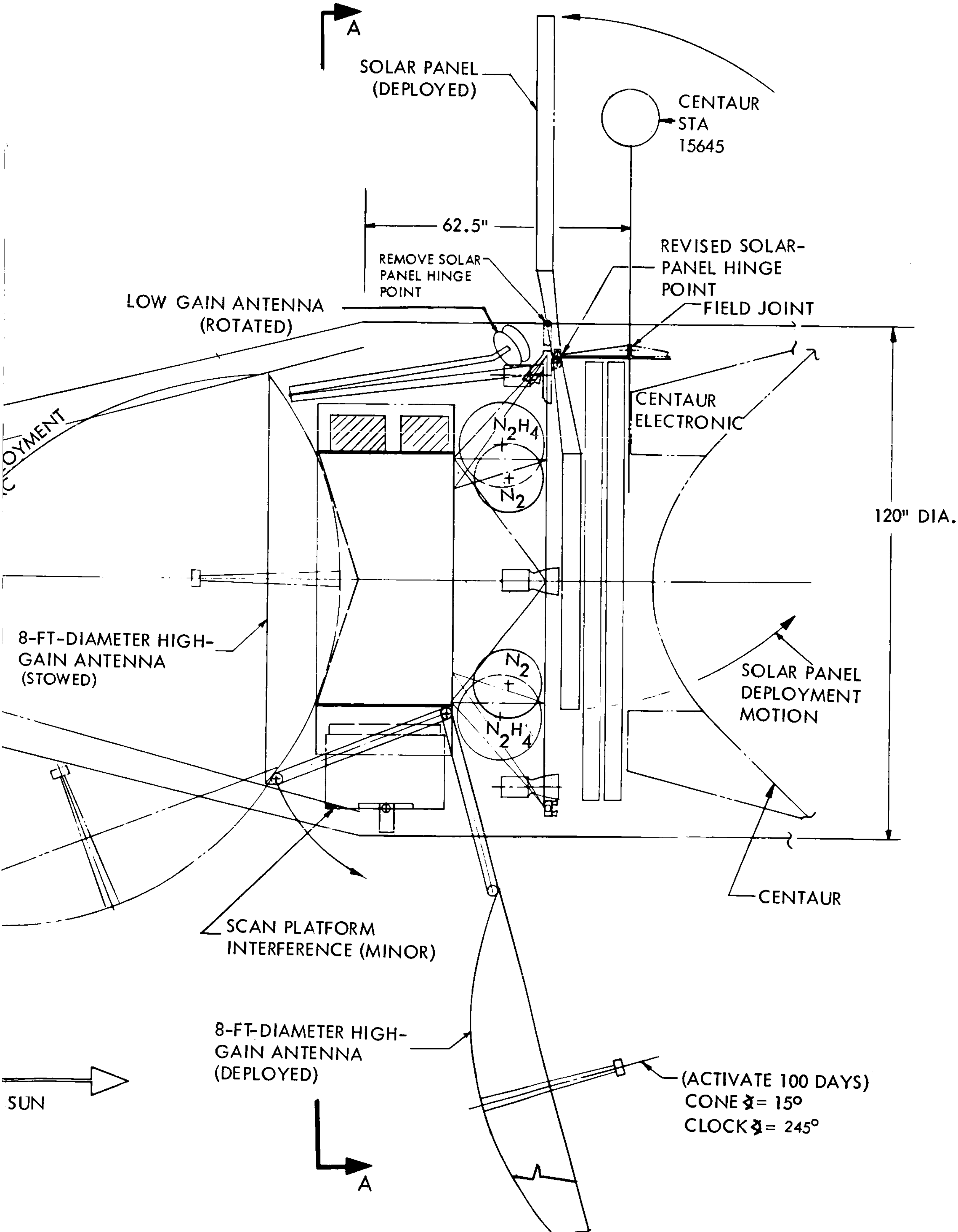
II-3.10.3 Configuration Trade Study

Another configuration considered for the 1969 Test Flight is shown in Figure II-13. In this configuration, the 1971 solar panels (unchanged except in number) are stowed aft across the body diameter and the 8-foot diameter antenna is located on the forward side of the spacecraft. This configuration has the advantage in that the antenna deployment linkages and mechanisms identical to the preferred 1971 spacecraft design are used. However, the major disadvantage is the problem occurring at spacecraft separation in attempting to pull the solar panels from within the cylindrical section between the field joint and the spacecraft separation plane. Redesign of the separation and support structure would be required to eliminate this condition. Also the stowed orientation of the solar panels is not an accurate simulation of the 1971 spacecraft.


LOW-GAIN ANTENNA
REVISE HINGE FITTING (ALL PANELS)
MIDCO
CONTR
N_2H_4
N_2
N_2
N_2
N_2
N_2H_4
SUN SEN
(COARSE
ON DA
SOLAR
(2 PLAC
SOLAR PA
(STOW
8-FT DIAMETER HIGH-
ANTENNA
-Y AXIS
YAW
A-
(VIEW LO
TOWARD


URSE & ATTITUDE
OL SAME AS NO. 71
- X AXIS
PITCH
SOR
, LOCATED
K SIDE OF
'ANEL
:S)
SOLAR ARRAY = 125 FT2
(PANEL IS SAME AS 71)
+Z AXIS
ROLL
NELS
D)
GAIN
KING
SUN)
TO

Figure



1-13: Alternate Configuration — 1969 Test Spacecraft — Atlas/Centaur

3

II-3.10.4 Nose Fairing Modification

The 1969 Test Spacecraft requires a 54-inch cylindrical extension of the standard nose fairing described in the Voyager 1971 Mission Guidelines.

II-3.11 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, PLANETARY QUARANTINE--ATLAS/CENTAUR

II-3.11.1 Description

The decontamination requirements considered for the 1969 Test Spacecraft are the same as described in Section 3.11 of Volume A, D2-82709-1, 1971 Preferred Design.

II-3.11.2 1969 Test Flight Benefits

The requirements are established to provide verification that the equipment and structure are not degraded by the decontamination treatment. In addition, techniques can be developed and tested in the areas of handling and maintenance of the decontaminated spacecraft. These tests and developments will result in increased confidence that the 1971 mission will be able to meet the planetary quarantine constraint without jeopardizing the probability of mission success.

II-3.12 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, CLEANLINESS--ATLAS/CENTAUR

II-3.12.1 Description

Cleanliness requirements for the 1969 Test Flight are identical to those described in Section 3.12 of Volume A, D2-82709-1, 1971 Preferred Design.

II-3.13 1969 TEST SPACECRAFT MAGNETICS--ATLAS/CENTAUR

II-3.13.1 Description

Magnetic design constraints for the 1969 Test Flight are identical to those defined in Section 3.13 of Volume A, D2-82709-1, 1971 Preferred Design.

II-3.14 1969 TEST SPACECRAFT RADIATION EFFECTS--ATLAS/CENTAUR

II-3.14.1 Description

A general treatment of the effects of radiation on spacecraft components and subsystems is given in Section 3.14 of Volume A, D2-82709-1.

II-4.0 1969 TEST SPACECRAFT FUNCTIONAL DESCRIPTION FOR HARDWARE SUBSYSTEMS--ATLAS/CENTAUR

II-4.1 1969 TEST SPACECRAFT TELECOMMUNICATIONS--ATLAS/CENTAUR

II-4.1.1 Description

The telecommunications subsystem will be identical to that described in Section 4.1 of Volume A, D2-82709-1 for the 1971 Preferred Design, with the following exceptions:

1) The 8- by 12-foot elliptical high-gain antenna will be replaced by an 8-foot circular paraboloid. This is the maximum size antenna that can be incorporated in the nose fairing. This change results in a 2-decibel reduction in link gains. All margins quoted in Volume A, Section 4.1 for operations using the high-gain antenna must be adjusted accordingly. The resultant performance of affected modes is plotted in Figures II-14 and II-15 for the nominal 1969 trajectory.
2) The relay radio subsystem and VHF antenna will not be incorporated in the 1969 Test Spacecraft because there are no capsule communications

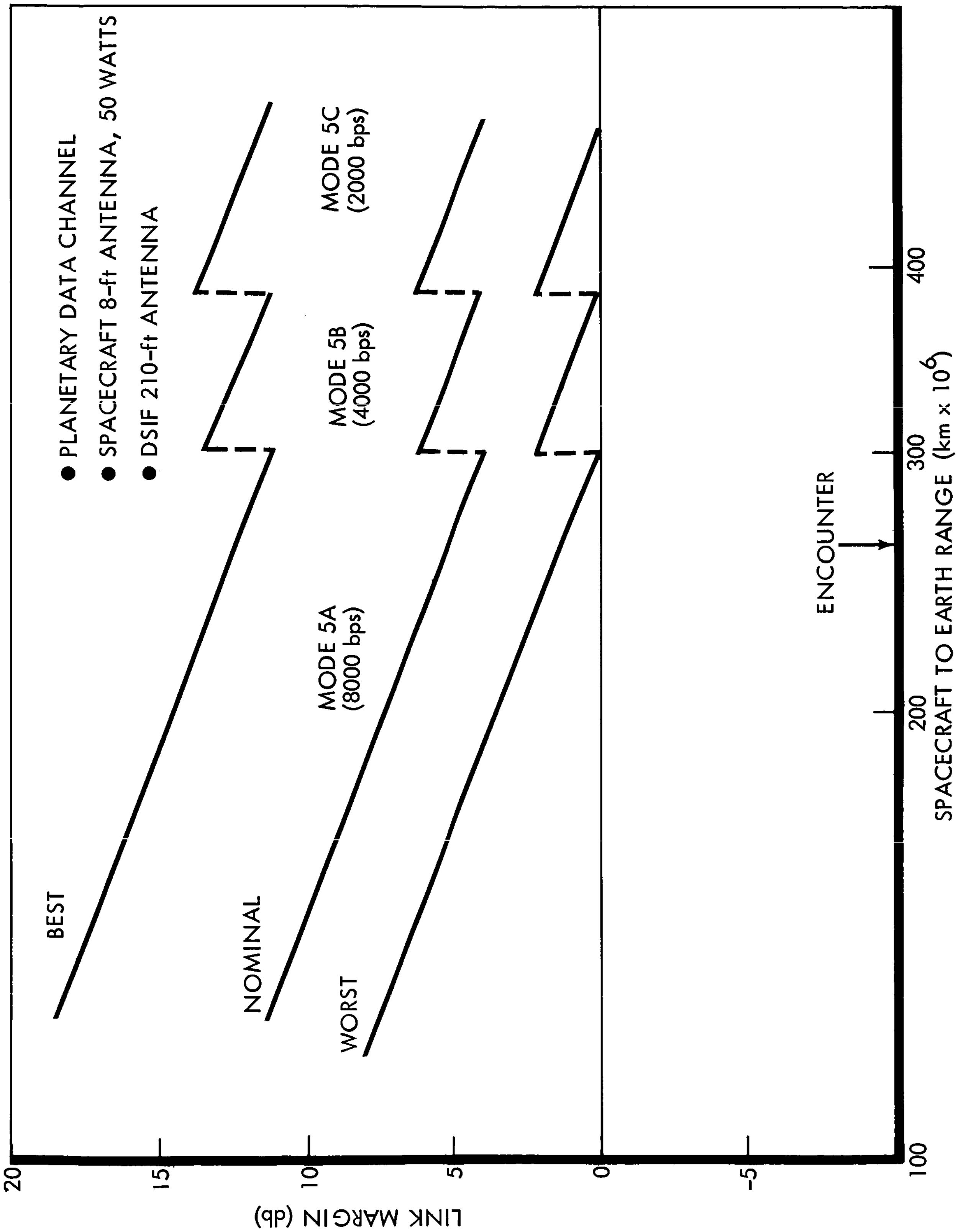


Figure II-14: Voyager Telemetry Link — 1969 Test Flight

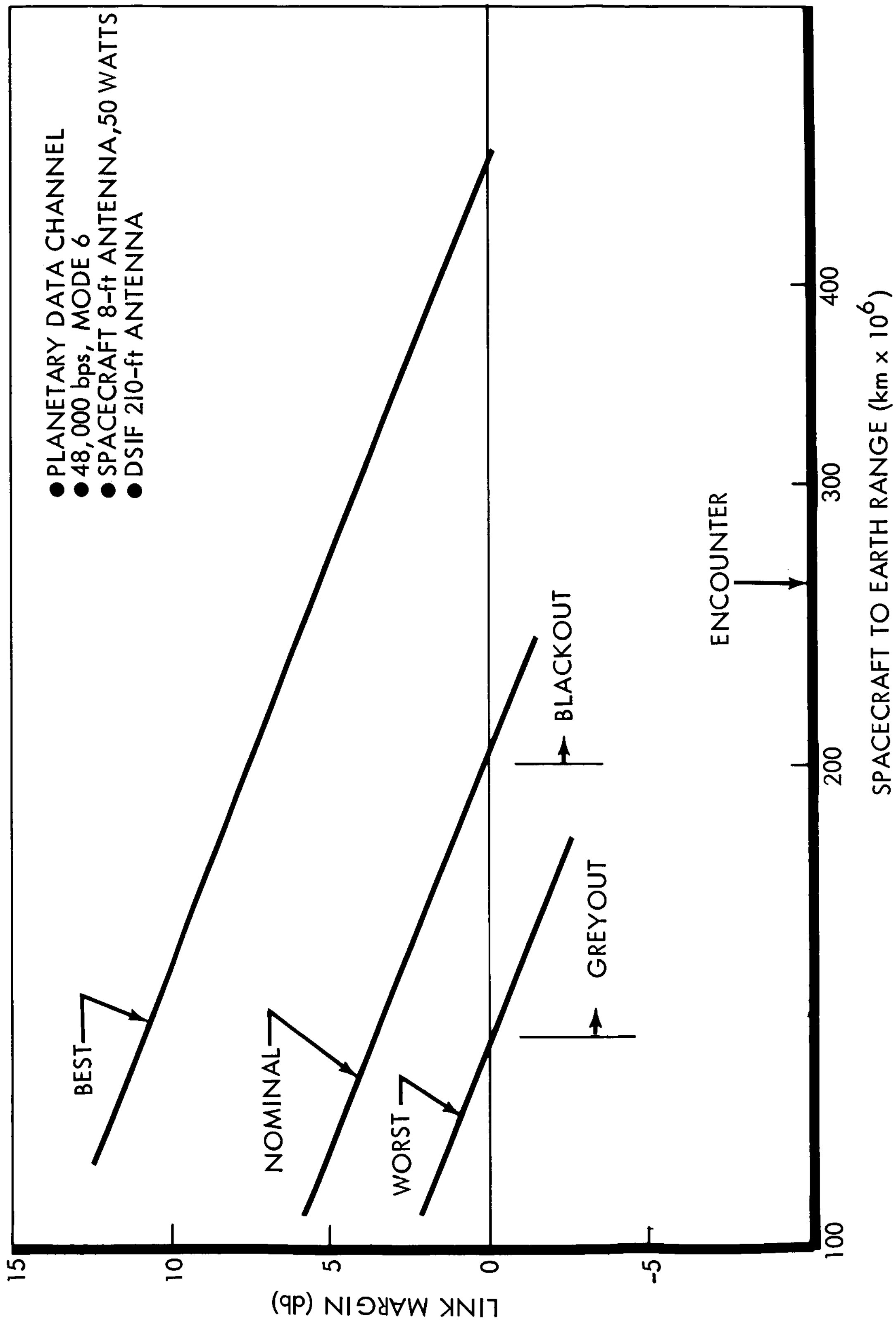


Figure II-15: Voyager Telemetry Link — 1969 Test Flight during this test flight. The capsule input channel to the telemetry and data storage subsystem will be used as an additional channel for engineering data measurement.

3) A special multiplexer will be incorporated to accommodate the expanded number of test points on spacecraft subsystems from which data will be telemetered during the test. This multiplexer will interface with the telemetry and data storage subsystems via the relay radio subsystem input channel and will provide an exact simulation of the normal interface at this point.

4) Only one planetary science tape recorder will be carried to save weight. Provision will be made to verify the proper operation of recorder sequencing functions. The simulation of real time and delayed planetary science data modes is described in Section II 3.6. It should be noted from Figure II-15 that checkout of the real-time planetary science mode must be accomplished prior to encounter because of the long trajectory for this test flight.

5) The cruise science input channels will be used for engineering data signals or simulated signals will be provided to verify proper operations.

II-4.1.2 1969 Test Flight Benefits

The telecommunications subsystem proposed for the 1971 Voyager mission includes several advances in equipment configuration and in data processing and handling. The major advantages of a test flight prior to the 1971 mission are as follows:

1) All elements of the subsystem will receive extended life testing under the full set of multistress environmental conditions they will

experience in 1971. This will be a much more comprehensive environmental test of the integrated system than can be achieved on the ground.

2) The operational software and MDE to be used at the deep space stations and the space flight operations facility can be completely exercised over a simulated 1971 mission profile (excluding the Mars orbit phase). This will be of significant value for personnel training and for discovery of any discrepancies or deficiencies that remain after ground simulation and checkout.

3) Verification of the predicted performance of high data rate modes at encounter ranges can be achieved by generating a bit stream simulating the high rate output of the data automation system. In particular, this will demonstrate the performance of the high-gain antenna pointing system coupled with vehicle dynamics.

II-4.2 1969 TEST SPACECRAFT ELECTRICAL POWER--ATLAS/CENTAUR

II-4.2.1 Description

The 1969 electrical power subsystem will be identical with the 1971 subsystem as described in Section 4.2 of Volume A, D2-82709-1, except for changes to accommodate the differences in environment and launch vehicle, decreased electrical load, and additional engineering test data.

Accordingly, the 1969 electrical power subsystem deviates from the 1971 subsystem only in the following features:

1) Deletion of half of the solar array area. The 1969 solar array will consist of three panels each of which has one structural section instead of two. The means of panel stowage and deployment is modified and is discussed in Section II-3.10.
2) Deletion of one of the three silver-cadmium (AgO-Cd) batteries. With two batteries, total capacity is sufficient to enable completion of the mission with one battery failed.
3) Addition of sensors or monitoring points to provide more extensive engineering test data.

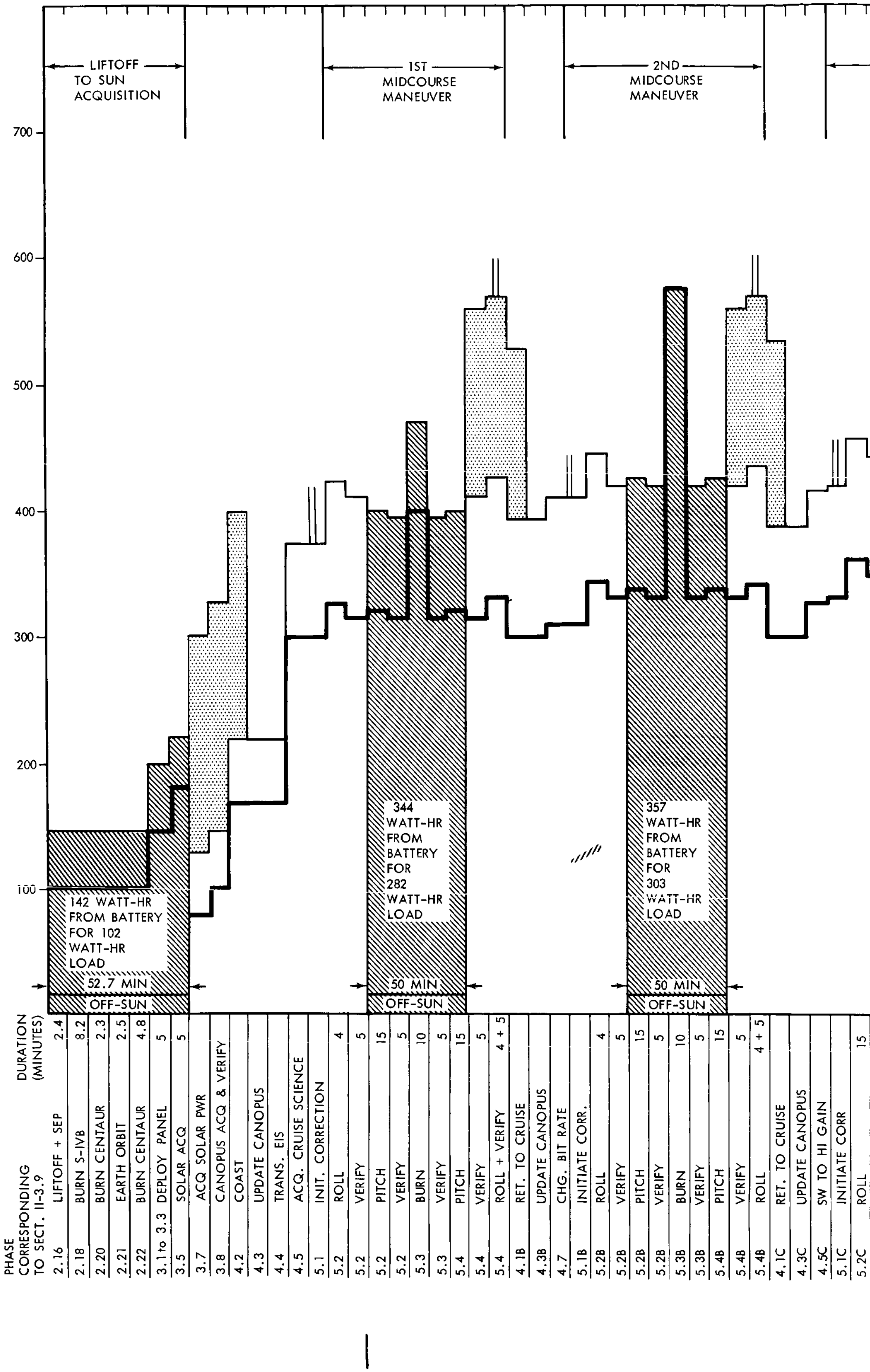
II-4.2.1.1 Requirements and Constraints

The operational requirements and constraints are generally the same for 1969 Test Flight as for the 1971 mission. The total load power required (a.c. plus d.c.) is represented by the heavy line (lower) profile shown in Figure II-16. Time periods in which the solar array is off-Sun are shown shaded and the watt-hours for that period are given in the shaded blocks.

II-4.2.1.2 Subsystem Functional Description

This section describes the difference in design and operation between the 1969 power subsystem and the basic subsystem described in Volume A.

Component Description--Tables II-4, II-5, and II-6 list the differences between the 1969 and 1971 electrical power subsystems.



LIFTOFF TO SUN ACQUISITION
1ST MIDCOURSE MANEUVER
2ND MIDCOURSE MANEUVER
700
600
500
400
300
200
100
WATTS
142 WATT-HR FROM BATTERY FOR 102 WATT-HR LOAD
52.7 MIN
OFF-SUN
344 WATT-HR FROM BATTERY FOR 282 WATT-HR LOAD
50 MIN
357 WATT-HR FROM BATTERY FOR 303 WATT-HR LOAD
PHASE CORRESPONDING TO SECT. II-3.9
DURATION (MINUTES)
2.16 LIFTOFF + SEP 2.4
2.18 BURN S-IVB 8.2
2.20 BURN CENTAUR 2.3
2.21 EARTH ORBIT 2.5
2.22 BURN CENTAUR 4.8
3.1 to 3.3 DEPLOY PANEL 5
3.5 SOLAR ACQ 5
3.7 ACQ SOLAR PWR
3.8 CANOPUS ACQ & VERIFY
4.2 COAST
4.3 UPDATE CANOPUS
4.4 TRANS. EIS
4.5 ACQ. CRUISE SCIENCE
5.1 INIT. CORRECTION
5.2 ROLL 4
5.2 VERIFY 5
5.2 PITCH 15
5.2 VERIFY 5
5.3 BURN 10
5.3 VERIFY 5
5.4 PITCH 15
5.4 VERIFY 5
5.4 ROLL + VERIFY 4 + 5
4.1B RET. TO CRUISE
4.3B UPDATE CANOPUS
4.7 CHG. BIT RATE
5.1B INITIATE CORR.
5.2B ROLL 4
5.2B VERIFY 5
5.2B PITCH 15
5.2B VERIFY 5
5.3B BURN 10
5.3B VERIFY 5
5.4B PITCH 15
5.4B VERIFY 5
5.4B ROLL 4 + 5
4.1C RET. TO CRUISE
4.3C UPDATE CANOPUS
4.5C SW TO HI GAIN
5.1C INITIATE CORR
5.2C ROLL 15

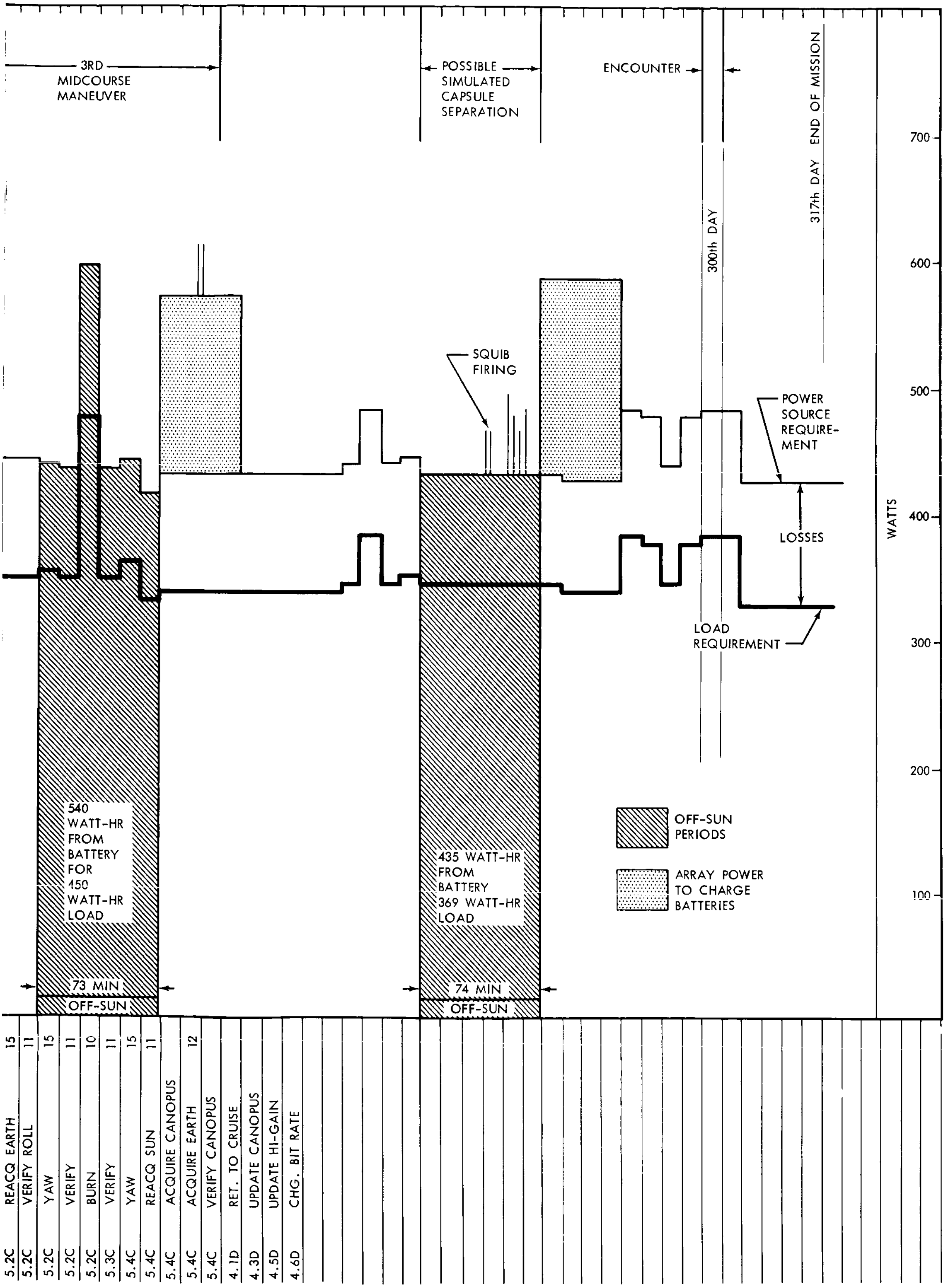


Figure II-16: Load Profile Plus Source Profile — Atlas/Centaur

2

Table II-4: QUANTITY DIFFERENCES

	1969	1971
Batteries	2	3
Battery Chargers	2	3
Solar Panel Structural Sections (4 electrical sections/Structural Section)	3	6

Table II-5: SIGNIFICANT ELECTRICAL DIFFERENCES

	1969	1971
Solar Array		
Nominal operating voltage output near Earth	61V	60V
Nominal operating voltage output near Mars	63V	65V
Maximum power near Earth ($55^{\circ}C$)	955W	1910W
Maximum power at end of mission	511W ($3^{\circ}C$)	685W ($-18^{\circ}C$)
Battery		
Watt hour capacity	1640	2460

Table II-6: SIGNIFICANT PHYSICAL DIFFERENCES

	1969	1971
Solar Array		
Area--Planform	120.5 ft^2	241 ft^2
Area--Substrate surface	118 ft^2	236 ft^2
Area--Solar cell mounting	104.9 ft^2	209.8 ft^2
Cell active area	99.5 ft^2	199 ft^2
Major structural section per panel	1	2
Structural subsections per panel (no difference)	2	4
Total Cells per Array	24,354	48,708
Battery	2	3
Power Conditioning Equipment		
Battery charger/failure sense unit	2	3
Subsystem Total Weight		
Solar array	164 lbs(est)	284 lbs
Batteries	82 lbs	123 lbs
Power conditioning electronics	48 lbs	50 lbs
Total Weight	294 lbs	457 lbs

II-4.2.1.3 Solar Array

Electrical Description--The solar array provides primary electrical power to the power subsystem by conversion of the solar energy. The array is composed of 24,354 solar cells mounted on three panels, and is capable of providing 511 watts at end of mission (8 months) with worst-case degradation (K_t = 0.63). The power requirement for the solar panel is approximately 383 watts. Thus, adequate capacity will be available from the panel. At 5 percent off the maximum power point (not including battery charging) when oriented within ±5 degrees of the Sun, the total mission degradation (K_t) is estimated to be 0.63. See Section 4.2.9 of D2-82709-1 for the detailed calculation of K_t. The solar flare degradation assumed for 1969 results in 20.5-percent power degradation as compared with 28 percent for the 1971 case, since the latter includes degradation for the 6 months in orbit.

Design Optimization--The use of three standard structural sections will provide 12 electrical sections. The power safety margin including the 10-percent safety factor is 179 watts which permits the complete loss of three electrical sections without decreasing power from the power subsystem to an unacceptable level.

II-4.2.1.4 Battery

Description--The mission requirements of the battery for the 1969 Test Flight are:

1) 38 volts (minimum);
2) 373-watt load;
3) 9.8-ampere current;
4) 222 watt-hours of energy;
5) 5.85 ampere-hours capacity;
6) 0.6-hour discharge duration;
7) Maximum of three maneuver cycles with long periods of recharge.

The preferred approach is to use two of the AgO-Cd batteries planned for 1971 mission even though one battery provides more than adequate energy capacity. This approach assures reliability of operation and more nearly duplicates the 1971 subsystem operational features.

II-4.2.2 1969 Test Flight Benefits

Added confidence in the successful operation of the electrical power subsystem during the 1971 mission will be gained through a 1969 Test Flight in the following ways:

1) Better judgment on solar proton radiation damage effects;
2) Verification of subsystem temperature;
3) Consideration will be given to further tests of battery cycling, panel stresses, and temperatures and other orbiting effects by deliberate misorientation of the spacecraft to simulate Mars occultation. This simulation would occur after flyby, and would be limited by the capability of other subsystems, telemetry blackout by the Sun, and other factors.

II-4.3 1969 TEST SPACECRAFT PROPULSION--ATLAS/CENTAUR

II-4.3.1 Description

This subsystem is identical to that described in Section 4.3 of D2-82709-1 except no solid-propellant-motor orbit insertion (including thrust vector control) capability is provided. For the midcourse correction propulsion system, the fuel and tankage are reduced to provide for 45 meters per second instead of 75 meters per second, and the orbit trim maneuver is deleted. The midcourse propulsion system will retain its 1971 configuration (except for fuel tank capacity) in that pressurant and propellant isolation valves and plumbing are provided for four midcourse corrections. The propulsion module weight is 200 pounds.

II-4.3.2 1969 Test Flight Benefits

A 1969 Test Flight will provide the means to demonstrate the operation of the midcourse correction propulsion system under the environmental conditions of a 1971 mission. In particular, it will enable verification of the spontaneous catalyst ignition system used on the monopropellant engines, under space operating conditions. In addition, repeated engine start and shutdown sequences are obtainable through the backup isolation valve circuits, within the limits of expendable propellant and pressurant.

II-4.4 1969 TEST SPACECRAFT ENGINEERING MECHANICS--ATLAS/CENTAUR

Summary--This section discusses the differences between the engineering mechanics described in D2-82709-1, Section 4.4 of this study report and the engineering mechanics used in the 1969 Test Spacecraft for the following subsystems and disciplines:

1) Temperature Control;
2) Packaging and Cabling;
3) Structure;
4) Mechanisms;
5) Pyrotechnics;
6) Mass Properties.

II-4.4.1 Thermal Control Subsystem

II-4.4.1.1 Description

The temperature control techniques and major elements are the same as for the 1971 spacecraft and as defined in D2-82709-1, Section 4.2.1. The design of thermal separation between basic modules makes possible design revisions in one module with only minor thermal effects on the others. Important thermal interfaces revised in the 1969 Test Spacecraft are modified to simulate the 1971 spacecraft characteristics. These include the following:

1) Flight Capsule Thermal Interface--Shields are used to simulate radiation blockage of the louvers by the capsule. Heat losses from the spacecraft to the capsule are simulated by a combination of insulation and heat shorts; the heat shorts are variable to permit adjustment during ground tests.

2) Exterior Boom Thermal Interfaces--Heat exchange is matched by using the operational spacecraft joint, with radiation tabs to simulate the booms.

3) Body Mounted Exterior Equipment--Heat losses from the spacecraft are matched with radiation tabs simulating the equipment which has been deleted from the spacecraft.

4) Propulsion Module Thermal Interface--Elimination of the orbit insertion engine and changes to tank geometry and arrangement introduce significant thermal differences, which cannot be completely compensated. For example, orbit engine head heat source cannot be simulated. However, design of the heat shield, conduction members, and insulation are tailored so the 1969 spacecraft thermal characteristics match the 1971 spacecraft.

5) Solar Thermal Shield--The stowage of the high-gain antenna requires the relocation of the solar shield. However, the thermal environment for the propulsion module and equipment module is thermally duplicated to match those of the 1971 spacecraft.

II-4.4.1.2 1969 Test Flight Benefits

A comparison of the thermal control similarities between the 1969 Test Flight and the 1971 mission is shown and summarized in Table II-7. The summary shows that both the Mars flyby and the heliocentric trajectories are beneficial, and that thermal control similarity is preserved nearly to the same extent that the configurations are the same. The test flight will adequately simulate all important parameters, except the thermal performance of the operational spacecraft propulsion module, the effects of the orbit insertion engine plume heating, and the variation in Mars orbit heat load due to Sun occultation.

The test flight provides confidence in the thermal control system through demonstration in the realistic total space environment, and through confirming the 1971 spacecraft design parameters by correlating theoretical calculations with actual values from the test flight. This logic makes test flight measurements particularly attractive in determining the effect of:

1) Proton and alpha particle radiation on the emissive properties of surface coatings; and
2) The solar intensity at Mars (thermal design of the high-gain antenna is based upon a low solar intensity).

II-4.4.2 1969 Spacecraft Cabling and Packaging Subsystem

II-4.4.2.1 Description

The cabling and packaging methods for the 1969 Test Flight are identical to that described in D2-82709-1, Section 4.4.2 for the 1971 spacecraft

Table II-7: THERMAL CONTROL SIMILARITY OF 1969 TEST FLIGHT TO 1971 MISSION

	Mars Flyby or Heliocentric Trajectory
Internal Bus Equipment	
Louver Control	Good
Equipment Packaging for Thermal Performance	Good
Thermal Coupling and Heat Leak	Fair
External Bus Equipment	
Surface Coatings	Good
Solar Panels	Good
High-Gain Antenna	Good
Omni Antenna	Good
Boost Heating	Fair
Plume Heating	None
Propulsion Module	
Solar Shield	Poor
Louver Control	Good
Thermal Coupling and Heat Leak	Poor
Thermal Interface	
Science Platform	None
Capsule	None
Electrical	Fair
Instrumentation	Fair

except for the deletion of those packages defined in Section II-3.10. The deletion of certain packages requires the following revisions to be made to the 1969 Test Spacecraft:

1) All packages not removed will be shifted a minimum amount to provide the spacecraft with the proper center of gravity. However, the functional and thermal relationships of the packages will not be changed.
2) Package interconnecting cable harness will be revised to provide for rearrangement of the equipment packages and the deleted packages.

II-4.4.3 1969 Spacecraft Structure Subsystem

II-4.4.3.1 Description

The spacecraft structural subsystem is basically the same as that described in D2-82709-1, Section 4.4.3 of this study report. The use of the Atlas/Centaur nose fairing requires that some modification be made to the basic configuration. These are identified in Section II-3.10. The structural modification required to accept the configuration revisions are as follows:

1) Primary Structure--The support truss assemblies which support the low-gain antenna, VHF antenna, and magnetometer booms are removed. The lower support truss frame members require additional bending capability to accommodate the appendage loads which are introduced eccentric to the node point of the ring frame members.
2) Propulsion/Reaction Control Structure--A separate revised structural system to support the midcourse propulsion and reaction control

subsystem is attached to the existing propulsion/reaction control support points.

3) Solar Panels--Two additional spars per panel are required oriented 90 degrees to the 1971 solar panel spars.

4) High-Gain Antennas--Relocation of the high-gain antenna attachment requires a revision to the lower support truss frame to accommodate the new antenna hinge and support fittings. The antenna boom and deployment structure is revised to be consistent with the new antenna.

II-4.4.3.2 1969 Test Flight Benefits

The benefit of a test flight to the structural subsystem is minor. However, confidence is increased in predicting structural response in maintaining the critical spacecraft component alignment and providing an adequate dynamics environment to the presence of heat and loads inputs during the 1971 mission.

II-4.4.4 1969 Mechanisms Subsystem

II-4.4.4.1 Description

The spacecraft mechanisms subsystem is identical to that described in D2-82709-1, Section 4.4.4, in that the same bearings, materials, gears and procedures are used in all instances. However, the following revisions in configuration are required to fit within the Atlas/Centaur nose fairing envelope:

1) Low-Gain Antenna Deployment Mechanism--Due to the difference in stowed position and deployment geometry, the deployment hinge mount

is changed. The Vinson actuator is used to deploy the antenna and the locking mechanisms are identical although oriented in a new plane.

2) High-Gain Antenna Deployment and Pointing Mechanism--The deployment linkage is changed due to the different antenna stowed position. However, the deployment principle is similar so that bearings, actuators, and locks may be evaluated. The antenna pointing mechanism is unchanged.

3) Scan Platform--The scan support yoke is shorter to provide proper clearance with the extended nose fairing dynamic envelope. This does not effect the operation of the deployment and drive mechanism which is identical to the 1971 installation. The boost support linkage is shorter but the method of releasing is identical.

4) Magnetometer Deployment Mechanism--This entire component is deleted.

5) Bacteriological Barrier Release Mechanism--This entire component is deleted since there is no capsule on the flight.

II-4.4.4.2 1969 Test Flight Benefits

The benefits from a test flight are:

1) The performance evaluation of the mechanisms will disclose areas of probable malfunction and the degree of rectification required.

2) Critical and long-lead-time component failure would be determined, therefore providing adequate modification allowances for follow-on spacecraft.

3) The test would demonstrate that the hinge mechanisms have not distorted thermally to prevent deployment and that the proper selection of bearing materials and fits have been made.

4) The confidence level of the predicted performance of the mechanisms subsystem for the 1971 spacecraft will be increased by the 1969 test, since all mechanisms will be operated in a true space environment which is difficult to simulate on the ground. Of particular concern in ground testing is the simulation of the combined effects of zero gravity, vacuum, radiation, time, meteoroid and operating stresses.

5) In addition to demonstrating by the normal release-and-lock position indicators that each mechanism performs satisfactorily, the test flight spacecraft mechanisms will be instrumented to obtain torque measurements for both drive units on the scan platform and for the high-gain antenna drive units.

Determination of these values under true flight conditions would either confirm or refute the design concept and permit timely rectification that would greatly enhance the probability of success of the 1971 mission.

II-4.4.5 Pyrotechnics Subsystem

II-4.4.5.1 Description

The pyrotechnic subsystem is identical to that described in D2-82709-1, Section 4.4.5 except that load simulating devices will be used where pyrotechnic devices have been omitted by the deletion of the magnetometer, VHF antenna, and orbit-insertion engine.

II-4.4.5.2 1969 Test Flight Benefits

The 1969 Test Flight will verify the functional operation of the pyrotechnic subsystem in an actual space environment.

II-4.4.6 Mass Properties

II-4.4.6.1 Summary Weight Statement

Consistent with design changes discussed in the previous sections, a weight statement for the 1969 Voyager Test Spacecraft (Atlas/Centaur launch vehicle) is presented below. For comparative purposes, the weight statement for the 1971 Voyager spacecraft is also presented.

	1969 Test Spacecraft	1971 Spacecraft
SPACECRAFT BUS		
Spacecraft Telecommunications	177	207
Attitude Reference Subsystem	51	51
Autopilot Subsystem	11	11
Reaction Control Subsystem	103	212
Central Computer & Sequencer Subsystem	58	58
Electrical Power Subsystem	283	457
Spacecraft Structure Subsystem	305	373
Spacecraft Mechanisms Subsystem	39	59
Temperature Control Subsystem	36	36
Pyrotechnic Subsystem	(included in CC&S)	
Instl. Cables/Harness Assemblies	100	100
Weight Contingency--Bus	137	186
TOTAL SPACECRAFT BUS	1300	1750
SPACECRAFT PROPULSION INSTALLATION		
Midcourse Correction Prop. Subsystem	(114)	(508)
Rocket Engine System	22	22
Propellant System Inerts	39	75
Pressurization Plumbing	15	16
Hydrazine--Useable	38	395
Orbit Insertion Prop. System	(--)	(2686)
Rocket Engine, Solid--Inerts	--	272
Thrust Vector Control Unit	--	108
Solid Propellant--Useable	--	2306
Spacecraft Structure--Prop. Inst.	(40)	(125)
Temperature Control--Prop. Inst.	(23)	(71)
Inst. Cables/Harness--Prop. Inst.	(8)	(10)
Weight Contingency--Prop. Inst.	(15)	(100)
Contingency--	15	62
Contingency--Fiberglas to Ti Eng Case	-	38
TOTAL PROPULSION INSTALLATION	200	3500

	1969 Test Spacecraft	1971 Spacecraft
SPACECRAFT SCIENCE PAYLOAD		
Science Payload Instrumentation	--	195
Science Payload-Data Automation	--	55
TOTAL SCIENCE PAYLOAD	--	250
Flight Capsule System	--	2300
SEPARATED PLANETARY VEHICLE	1500	7800
Spacecraft Adapter and Support Above Field Joint	50	250
Spacecraft Support Below Field Joint	150	250
TOTAL	1700	8300

II-4.4.6.2 Sequential Mass Properties

Sequential weight, center of gravity, and mass moments of inertia are presented below based on maximum usage of expendables. Because the mid-course propellant and reaction control gas are located very close to both the spacecraft centerline and the spacecraft center of gravity, their usage exerts a negligible effect on both center of gravity and mass moments of inertia.

Condition	Weight (lb)	CG STA (in)	I_z (slug - ft^2)	I_x (slug - ft^2)	I_y (slug - ft^2)
Gross	1500	29.0	1210	740	530
Gross Less Expendables	1438	29.0	1210	740	530

II-4.5 1969 TEST SPACECRAFT SCIENCE--ATLAS/CENTAUR

No provisions have been allocated for a science payload on the 1969 Test Flight; however, structural support, electrical power, and telemetry is available should science experiments be required.

II-4.6 1969 TEST SPACECRAFT ATTITUDE REFERENCES AND AUTOPILOT SUBSYSTEM--ATLAS/CENTAUR

II-4.6.1 Description

The attitude references and autopilot subsystem proposed for the 1969 Test Flight is identical to the preferred design described in Section 4.6 of Volume A, D2-82709-1 except for the following changes.

II-4.6.1.1 Equipment Modifications and Telemetry

The 1971 equipment will be modified to convert the output signal of the test connector for the OSE to an inflight capability.

The normal mission sequence will exercise all modes of operation of the attitude reference and autopilot subsystems except orbit-insertion TVC. External dummy loads will be provided for the secondary injection valves to exercise the valve drivers and associated switching functions in a simulated maneuver. Changes in spacecraft inertia will necessitate changes in autopilot gain in order to preserve stability.

II-4.6.2 1969 Test Flight Benefits

Partial confirmation of the operation of the subsystem can be obtained from a simulation test program on Earth. However, complete confirmation can be obtained only through test under actual operating conditions. For example, an important function of the gyros is to maintain vehicle attitude during maneuvers and occultation in order to point the high-gain antenna. It is anticipated that the long-term drift bias stability of the gyro is less than 0.05 degree per hour, which is adequate; however, there is no data on long-term operation at zero acceleration to confirm this. Consequently, a test flight similar to the 1971 Voyager flight would provide data required to confirm the bias stability of the gyro under zero-g conditions as well as prove the validity of subsystem operation within the combined effects of induced and natural environments.

II-4.7 1969 TEST SPACECRAFT REACTION CONTROL SUBSYSTEM--ATLAS/CENTAUR

II-4.7.1 Description

The reaction control subsystem for the Atlas/Centaur 1969 Test Flight is the same subsystem as described in Paragraph 4.7 of D2-82709-1 except that approximately half of the propellant is required as for the 1971 Preferred Design. The reduction is fuel is accomplished by a reduction in the size of the tanks. Rearrangement of the tanks within the spacecraft is necessary.

II-4.7.2 1969 Test Flight Benefits

The test flight will prove the capability of the specific reaction control hardware to operate for extended periods of time in space. Space environment and time are the only criteria for the test.

Temperatures and pressures of the propellant tanks will be determined during prolonged space exposure.

II-4.8 1969 TEST SPACECRAFT CENTRAL COMPUTER AND SEQUENCER--ATLAS/CENTAUR

II-4.8.1 Description

The central computer and sequencer subsystem is the same as described in Section 4.8 of Volume A, D2-82709-1.

II-4.8.2 1969 Test Flight Benefits

The 1969 Test Flight will prove the reliability and versatility of the CC&S to meet the 1971 mission requirements, constraints, and objectives. Specifically the benefits derived are as follows:

1) Reliability--Command Message Handling--The command link and CC&S command message handling will be verified proving ground-to-spacecraft bit error rate, command verification, command acceptance, and command execution. By transmission of a sufficient number of real-time commands under all Earth weather conditions the ground-to-spacecraft transmission bit error rate may be established. Resultant single and multiple bit error rates will determine the necessity of additional error checking or correcting codes within the command message. Command verification is of major importance, providing ground assurance that transmitted commands arrived successfully at the spacecraft. However, command verification at great distances from Earth requires time, and under emergency conditions may be too long. Establishing the command verification technique for command acceptance within the CC&S will be of major benefit to ensure 1971 mission success. As the CC&S executes the stored program sequence, it will be desirable to establish the validity of execution of the CC&S over prolonged periods of time under actual flight environmental

conditions. Thus flight program execution reliability will be demonstrated as a result of the 1969 Test Flight. If failures occur during flight in various command paths either within the CC&S or in other subsystems, the test flight will determine the value of the work-around methods and procedures established within the CC&S.

2) Versatility--The 1969 Test Flight will demonstrate the ability of the CC&S to accept changes to the stored preplanned sequence of events at any time as dictated by mission circumstances. The tests will include demonstrations of switchover between the two redundant CC&S control assembly logic processing units. Override capability of real-time ground initiated commands will be demonstrated.

PART III--1969 TEST SPACECRAFT--SATURN IB/CENTAUR

This part of the document establishes the requirements and descriptions for a Voyager Test Spacecraft and its associated subsystems in performance of a 1969 Test Flight using a Saturn IB/Centaur launch vehicle. Only the exceptions from the descriptions stated in Volume A, D2-82709-1, "1971 Preferred Design," are cited.

III-1.0 1969 TEST SPACECRAFT TEST OBJECTIVES AND DESIGN CRITERIA

III-1.1 1969 TEST SPACECRAFT INTRODUCTION--SATURN IB/CENTAUR

One principal type of test flight using Saturn IB/Centaur boosters is considered for the 1969 Test Flight. The test flight is similar to the 1971 Voyager mission in that the test spacecraft will orbit Mars. Unlike the 1971 mission, however, the 1969 Test Flight will employ a simulated Flight Capsule that, at the appropriate point on the trajectory, will be deflected away from Mars. Also, the science payload is omitted from the 1969 Test Spacecraft.

III-1.2 1969 TEST FLIGHT OBJECTIVES--SATURN IB/CENTAUR

The objective of the 1969 Test Flight is to enhance the probability of successful accomplishment of the 1971 Voyager objectives. Toward this goal, the test flight and test spacecraft will be identical to the 1971 Voyager.

III-1.3 1969 TEST SPACECRAFT TEST FLIGHT RESTRAINTS--SATURN IB/CENTAUR

The test flight restraints are the same as those established in Section 1.3 of Volume A, D2-82709-1, except that in 1969, a nominal launch period of 30 days will be provided for a Mars orbiting test flight. Later launch opportunities are available for Mars flyby or heliocentric trajectories if the prime launch period is missed.

III-1.4 1969 TEST SPACECRAFT DESIGN CRITERIA--SATURN IB/CENTAUR

Design of 1969 Test Spacecraft components will be identical with 1971 Flight Spacecraft component design to achieve meaningful test flight data. In all other respects, design criteria are as established in Volume A for the 1971 Voyager design.

III-1.4.1 Schedule Criteria

Since the 1971 Mars opportunity is a rigid constraint upon the Voyager program, it follows that all facets of the test flight must be accomplished on a schedule that is compatible with obtaining significant data for incorporation into the 1971 Voyager program schedule.

III-1.5 1969 TEST SPACECRAFT WEIGHT--SATURN IB/CENTAUR

The launch weight of the test spacecraft together with the simulated Flight Capsule and adapters must be no more than 8300 pounds.

III-1.6 1969 TEST SPACECRAFT, COMPETING CHARACTERISTICS--SATURN IB/CENTAUR

1971 spacecraft priorities will be used where there are conflicting technical requirements.

III-2.0 1969 TEST SPACECRAFT DESIGN CHARACTERISTICS AND RESTRAINTS

III-2.1 1969 TEST SPACECRAFT DESIGN CHARACTERISTICS--SATURN IB/CENTAUR

This section presents a general description of the 1969 Voyager Test Spacecraft and the considered test flight profile using a Saturn IB/Centaur launch vehicle.

III-2.1.1 General System Description

The elements of the system are the launch vehicle system, the test spacecraft system and the simulated Flight Capsule system. As with the 1971 Voyager program, the capsule and the spacecraft are collectively called the Planetary Vehicle.

III-2.1.1.1 Launch Vehicle System

The launch vehicle system consists of the three stages of the launch vehicle, the nose fairing over the Planetary Vehicle, and associated supporting equipment. The three-stage launch vehicle includes the S-IB, S-IVB, and Centaur stage as described in Volume A, D2-82709-1, "1971 Preferred Design."

III-2.1.1.2 Planetary Vehicle

Test Spacecraft--The test spacecraft is the same as the Flight Spacecraft described in Volume A, D2-82709-1, except that the 1969 Test Spacecraft will not have a science subsystem.

Simulated Flight Capsule--A simulated Flight Capsule is included in the Saturn-IB/Centaur-launched 1969 Test Flight to measure the maximum number of parameters that could affect the spacecraft.

III-2.1.2 Test Flight Profile

With some variation, the test flight profile for the 1969 Test Flight will be the same as that described in Volume A, D2-82709-1, for the 1971 Voyager mission. The basic differences between the 1969 and 1971 Mars orbit are that the 1969 test flight will incorporate:

1) Type II transfer trajectories;
2) A C_3 of 17.0 $(km/sec)^2$;
3) Launch dates from December 30, 1968, to January 29, 1969;
4) An 18-hour orbit period with maximum propellant load allowed by the 1971 orbit insertion motor case;
5) The simulated Flight Capsule will be deflected away from Mars.

III-2.1.3 Subsystems

With the exclusion of the science subsystem, the characteristics and restraints for the 1969 Test Spacecraft subsystems are the same as those described in Section 2.1.3 of Volume A, D2-82709-1, for the 1971 preferred design.

III-2.2 1969 TEST SPACECRAFT DESIGN RESTRAINTS--SATURN IB/CENTAUR

The restraints imposed on the subsystems by the system and other subsystems (including the environmental restraints) for the 1969 Test Flight are the same as those established in Volume A for the 1971 mission.

III-2.3 1969 TEST SPACECRAFT GUIDANCE AND NAVIGATION MANEUVER ACCURACY AND PROPULSION REQUIREMENTS--SATURN IB/CENTAUR

III-2.3.1 Description

The maneuver accuracy and propulsion requirements for the 1969 Test Flight are the same as described in Volume A for the 1971 preferred design. The approach to aim-point biasing during midcourse is identical to that established in Section 3.1 of Volume B, D2-82709-2.

III-2.3.2 1969 Test Flight Benefits

The 1969 Test Flight will provide a test of capability to accurately guide the spacecraft to a successful encounter using three midcourse maneuvers.

The test flight will also provide a test of capability to trim the orbit about Mars.

III-2.4 1969 TEST SPACECRAFT AIMING POINT SELECTION--SATURN IB/CENTAUR

III-2.4.1 Description

The aiming-point selection for the 1969 Mars orbital test flight is identical to that described in Section 2.4, Volume A, D2-82709-1. Minor changes in the occultation charts result from differences in the transit trajectories.

III-2.5 1969 TEST SPACECRAFT PARTS, MATERIALS AND PROCESSES--SATURN IB/CENTAUR

III-2.5.1 Description

Parts, materials, and processes for the 1969 Test Spacecraft will be the same as described in Section 2.5 of Volume A, D2-82709-1, 1971 preferred design, and will be subjected to the same controls.

III-2.5.2 1969 Test Flight Benefits

The 1969 Test Flight will provide verification testing of critical material systems and parts through functional performance in the Mars mission environment. Changes in performance resulting from the environmental exposure of the 1969 Test Flight can be interpreted to estimate performance in the 1971 mission.

1) Thermal control coating performance will be determined by spacecraft temperature data.
2) The change in performance in solar cells will be indicated by power generation data.
3) Bearing lubricant and gear performance will be evaluated by functional operation of mechanisms.

III-3.0 1969 TEST SPACECRAFT SYSTEM LEVEL FUNCTION DESCRIPTION OF FLIGHT SPACECRAFT

III-3.1 1969 TEST SPACECRAFT STANDARD TRAJECTORIES--SATURN IB/CENTAUR

III-3.1.1 Description

The 1971 Voyager mission will be performed with an 18-hour orbit period about Mars and a required orbit insertion ΔV of 5700 fps. For the Mars orbiting 1969 Test Flight, using the Saturn IB/Centaur launch vehicle, orbit periods of 18, 30, and 60 hours have been examined. The important trajectory characteristics are summarized in Table III-1 along with a table of the allocated weights for the 1969 Planetary Test Vehicle compared with the 1971 Planetary Vehicle allocated weight.

1) 18-Hour Orbit Period--This mission, which is most similar to the 1971 Voyager mission, was investigated on the basis of two different launch periods: 60 days and 30 days. The 18-hour orbit test flight and 60-day launch period requires an insertion ΔV of 7900 fps, which is beyond the capability of the propulsion system. By reducing the launch period to 30 days, the ΔV required would be reduced to 6500 fps, which is within propulsion system capability by adding propellant to the identical orbit insertion motor case. The simulated Flight Capsule will be decreased by a corresponding amount. Alternatively, an 18-hour orbit with a 60-day launch period can be obtained with a larger injected weight (about 8850 pounds) on the Mars transit trajectory or with a lesser weight (1620 pounds) inserted into the Mars orbit.

2) 30-Hour Orbit Period--To achieve the same ΔV of 5700 fps, which will be used in the 1971 mission, the 1969 orbit period is increased to 30 hours. The launch period, however, is 1 day.

3) 60-Hour Orbit Period--The maximum safe orbit (due to orbit perturbations) is the 60-hour orbit. It can be performed with a 30-day launch period and a required orbit insertion ΔV of 5900 fps. This test flight can also be performed with a ΔV of 5700 fps if the spacecraft weight inserted into Mars orbit is 1940 pounds. The largest launch period for this test flight is 22 days.

MISSION	BOOSTER	LAUNCH PERIOD (DAYS)	TRAJ. TYPE	LAUNCH DATES	ARRIVAL DATES
Mars Orbit with 18-hr Period	Saturn IB/Centaur	60	II	Dec. 22, 1968/ Feb. 20, 1969	Aug. 24, 19 Nov. 25, 19
Mars Orbit, 18-hr Period		30	II	Dec. 30, 1968/ Jan. 29, 1969	Sept. 16, 19 Oct. 26, 19
Mars Orbit, 30-hr Period		1	II	Jan. 5, 1969	Sept. 29, 1
Mars Orbit, 60-hr Period		30	II	Dec. 30, 1968/ Feb. 1, 1969	Sept. 18, 19 Oct. 28, 19
Mars Orbit, 60-hr Period		22	II	Jan 2, 1969/ Jan. 24, 1969	Sept. 21, 19 Oct. 21, 19
Mars Flyby		122	II	Dec. 27, 1968/ April 28, 1969	Sept. 8, 196 Feb. 5, 197

Saturn IB/Centaur

Saturn IB/Centaur

Saturn IB/Centaur

Saturn IB/Centaur

Saturn IB/Centaur

Capsule Separated Weight
Capsule Adapter & Sterilization Canister
Total Capsule Weight
Spacecraft Bus, including Science
Spacecraft Propulsion
Total Spacecraft Weight
Separated Planetary Vehicle Weight
Spacecraft Adapter and Support above Field Joint
Planetary Vehicle Weight
Spacecraft Support Below Field Joint
Total

Table III-1: Saturn IB/Centaur — Trajectory Characteristics

	REQUIRED C_3 (km^2/sec^2)	LAUNCH VEHICLE WEIGHT CAPABILITY (lb)	TRANSIT TIMES (DAYS)	DLA RANGE	COMMUNICATION DISTANCE (km)	INSERTION V REQUIREMENTS (ft/sec)
69/ 69	17	9200	245/280	$5^\circ/12^\circ$	198×10^6	7900
69/ 69	17	9200	260/270	$5^\circ/12^\circ$	170×10^6	6550
69	17	9200	270	5°	145×10^6	5700
69/ 69	17	9200	260/270	$5^\circ/12^\circ$	165×10^6	5900
69/ 69	16	9200	260/275	$5^\circ/12^\circ$	$128 \times 10^6/272 \times 10^6$	5700
9/)	18	9050	255/280	$5^\circ/33^\circ$	170×10^6	—

WEIGHT ALLOCATIONS (lb)

1969 TEST		1971 VOYAGER	
1418		1950	
350		350	
	1768		2300
2000		2000	
4032		3500	
	6032		5500
	7800		7800
250		250	
	8050		8050
250		250	
	8300		8300

2

All of the trajectories considered meet the aiming point constraints described in Section 2.4 of Volume A, D2-82709-1. A design chart (Figure III-1) shows the detailed trajectory information. Lines of constant DLA have been superimposed in red on this curve. The DSN orbit determination constraints of DLA = ± 5 degrees have been shaded. Lines of constant approach velocity are indicated by the blue lines. A minimum V_{HP} of 3.7 km/sec is achievable in 1969.

III-3.1.2 1969 Test Flight Benefits

In analyzing the described trajectories, orbital periods, launch periods, and insertion ΔV's it is recommended that the trajectory that provides an 18-hour orbit period with a ΔV of 6550 fps and a 30-day launch period be adopted for the 1969 Test Flight. This particular mission is the closest in similarity to the 1971 Voyager mission; consequently, the test results would greatly enhance the confidence in a 1971 mission success.

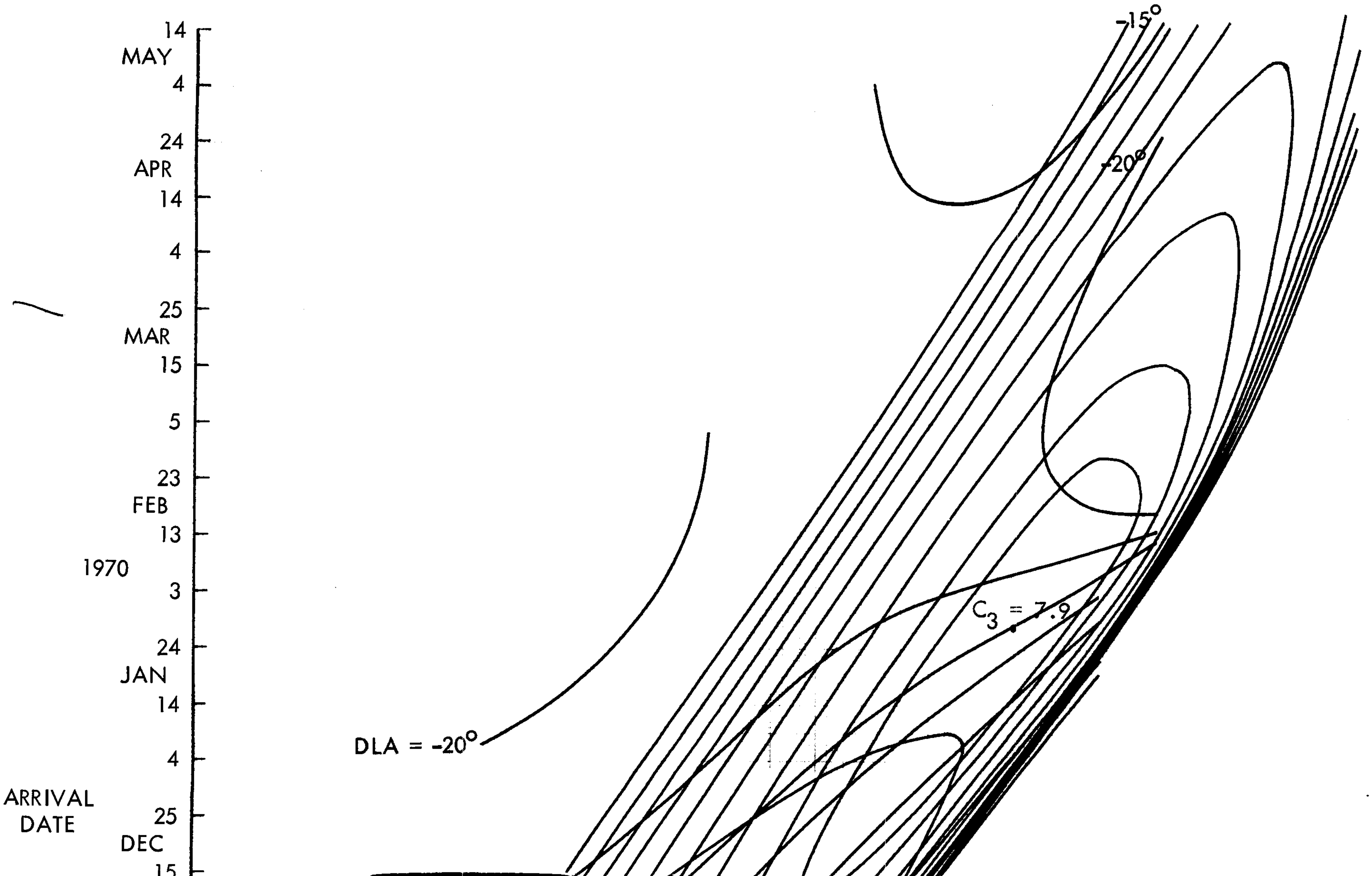
III-3.2 1969 TEST SPACECRAFT ORBIT DETERMINATION CAPABILITY--SATURN IB/CENTAUR

III-3.2.1 Description

The orbit determination capability for the 1969 Test Flight will not vary significantly from the capability discussed in Volume A, Section 3.2, and Volume B, Section 3.1.

III-3.2.2 1969 Test Flight Benefits

The 1969 Test Flight will provide information that will allow refinement of the physical constants that affect heliocentric orbit determination capability, including the astronomical unit, the Mars ephemeris, and the Mars gravitational parameters. This will provide significant improvement


ARRIVAL DATE
1970
14
MAY
4
24
APR
14
4
25
MAR
15
5
23
FEB
13
3
24
JAN
14
4
25
DEC
15
DLA = -20°
-15°
-20°
C3 = 7.9

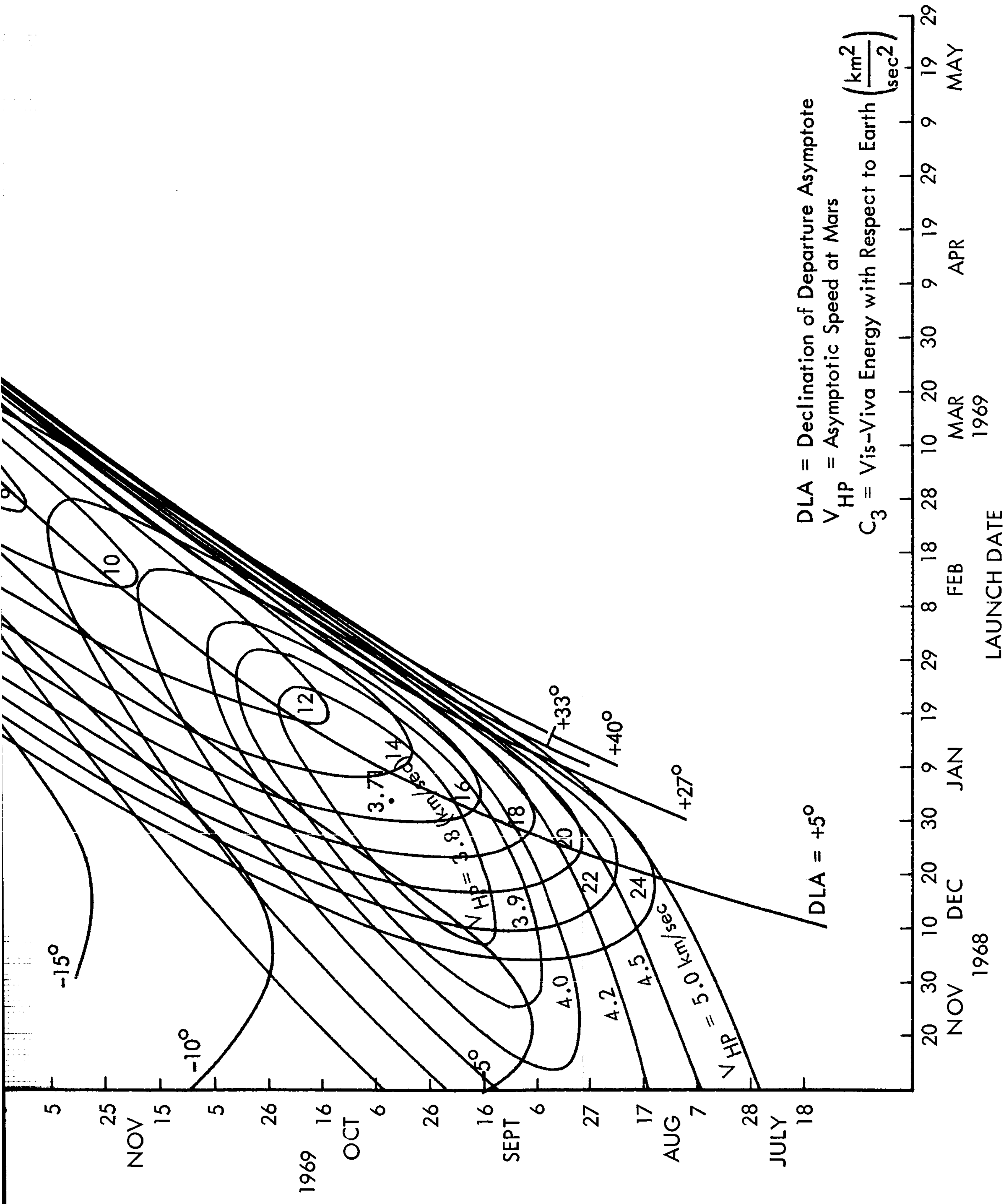


Figure III-1: Earth-Mars Transfer Characteristics —1969 Type II

2

in the 1971 mission heliocentric orbit determination capability. The Mars orbit phase mission will provide a test of orbit determination for an orbiter around a distant body. This problem is discussed in Volume B, Section 3.1.

III-3.3 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, SPACECRAFT COMPONENTS DESIGN PARAMETERS--SATURN IB/CENTAUR

III-3.3.1 Description

In Volume A, D2-82709-1, Section 3.3, specifications controlling weight, volume, power, and thermal operating ranges are tabulated for each subsystem on sheets headed "Spacecraft Components Design Parameters." These sheets (SCDPS) are also applicable to the Saturn IB/Centaur-launched 1969 Test Spacecraft except for deletion of science instrumentation and science data automation equipment and the addition of one multiplexer-encoder unit for additional engineering performance data.

III-3.4 1969 TEST SPACECRAFT EQUIPMENT ELEMENT IDENTIFICATION--SATURN IB/CENTAUR

III-3.4.1 Description

The methods and procedures for hardware and software identification for the 1969 Test Spacecraft are the same as described in Section 3.4 of Volume A, D2-82709-1, "1971 Preferred Design."

III-3.4.2 1969 Test Flight Benefits

Use of the methods and procedures for the 1969 Test Flight will provide early training and experience that can be readily applied to the 1971 Voyager program.

III-3.5 1969 TEST SPACECRAFT FLIGHT EQUIPMENT LAUNCH VEHICLE INTERFACE REQUIREMENTS--SATURN IB/CENTAUR

III-3.5.1 Description

The 1969 Test Spacecraft flight equipment to launch vehicle interface requirements are the same as those described in Subsection 3.8, Volume A, D2-82709-1.

III-3.5.2 1969 Test Flight Benefits

The 1969 Test Flight provides an opportunity to prove interface compatibility well in advance of the 1971 mission, therefore guarding against any unforeseen launch vehicle interface problems that might prejudice 1971 schedules.

III-3.6 1969 TEST SPACECRAFT FLIGHT EQUIPMENT TELEMETRY CRITERIA--SATURN IB/CENTAUR

III-3.6.1 Description

The primary objective of the telemetry equipment is to measure, with high resolution and accuracy, the spacecraft subsystem performance parameters and encode the information on subcarriers with low-error probability for subsequent RF transmission.

The 1969 Test Flight telemetry equipment is identical to the 1971 Voyager Spacecraft equipment as described in Volume A, D2-82709-1 except for incorporation of the capability for multiplexing additional engineering measurements. The test spacecraft will provide the necessary capabilities and functions for the telemetry equipment design verification. On the basis that spacecraft science data will not be available for the 1969 Test Flight, equivalent simulation will be provided to exercise the telemetry modes, techniques, and equipment. In lieu of actual science data, the telemetry channels are to be used for additional spacecraft engineering measurements in conjunction with spacecraft equipment design verification.

For the Mars orbiting and flyby test flights (Mode 6), the planetary science high-bit rate will be exercised before encounter. The rf link calculations indicate operation at encounter may be in the rf link blackout region. Other modes will be exercised throughout the test flight for transmission of engineering data.

The Voyager 1969 Test Spacecraft will be equipped with telemetry and communication equipment capable of transmitting spacecraft data to the DSIF. The transmitted data consists of spacecraft engineering, and simulated high-rate science data. If high-bit-rate performance data is required, it can replace the planetary science simulated data on command.

In addition, the cruise science channel is used to input additional spacecraft engineering data, which will approximately simulate cruise science data. The capsule to spacecraft telemetry on board and separation interfaces are exactly simulated. During critical spacecraft maneuvers, spacecraft telemetry data can be redundantly stored on board for subsequent retransmission.

The 1969 Test Flight telemetry equipment data formatting, modulation techniques, data storage modes, and onboard core storage and tape recorders are identical to 1971 spacecraft telemetry equipment. The use of a simulated Flight Capsule with VHF capability will check out the relay radio subsystem and bus interfaces before, during, and after separation. If required, simulated capsule performance data can be acquired through the capsule telemetry channel as shown in Figure III-2. The measurements required for the 1971 flight will be made on the 1969 Test Flight using the regular engineering multiplexer-encoder. Additional engineering performance data will be telemetered using another multiplexer-encoder unit whose output feeds the cruise science data channel. It is expected that this second unit will be identical to the engineering multiplexer-encoder in design.

Because a 48-kc-bit-rate capability exists, it can be used for low-accuracy transient information such as rocket motor pressure or attitude control parameters that could not be telemetered at the slow bit rates. Encoding at 6 bits results in a single-channel 8-kc sample rate capability that can be multiplexed or time-shared.

Simulation of the planetary science data for checkout of Modes 5 and 6 will be accomplished using known sequences of data bits. For example, an 11-stage shift register with feedback will generate a fixed-recycling PN sequence of $2^{11}-1$ (i.e., 2047) bits in length that can be easily synchronized with the block coder or tape recorder bit rates. Because the exact pattern is known, the data will be automatically reduced to determine the system error rate. Switching from Mode 5 to 6 is done by speeding up the input

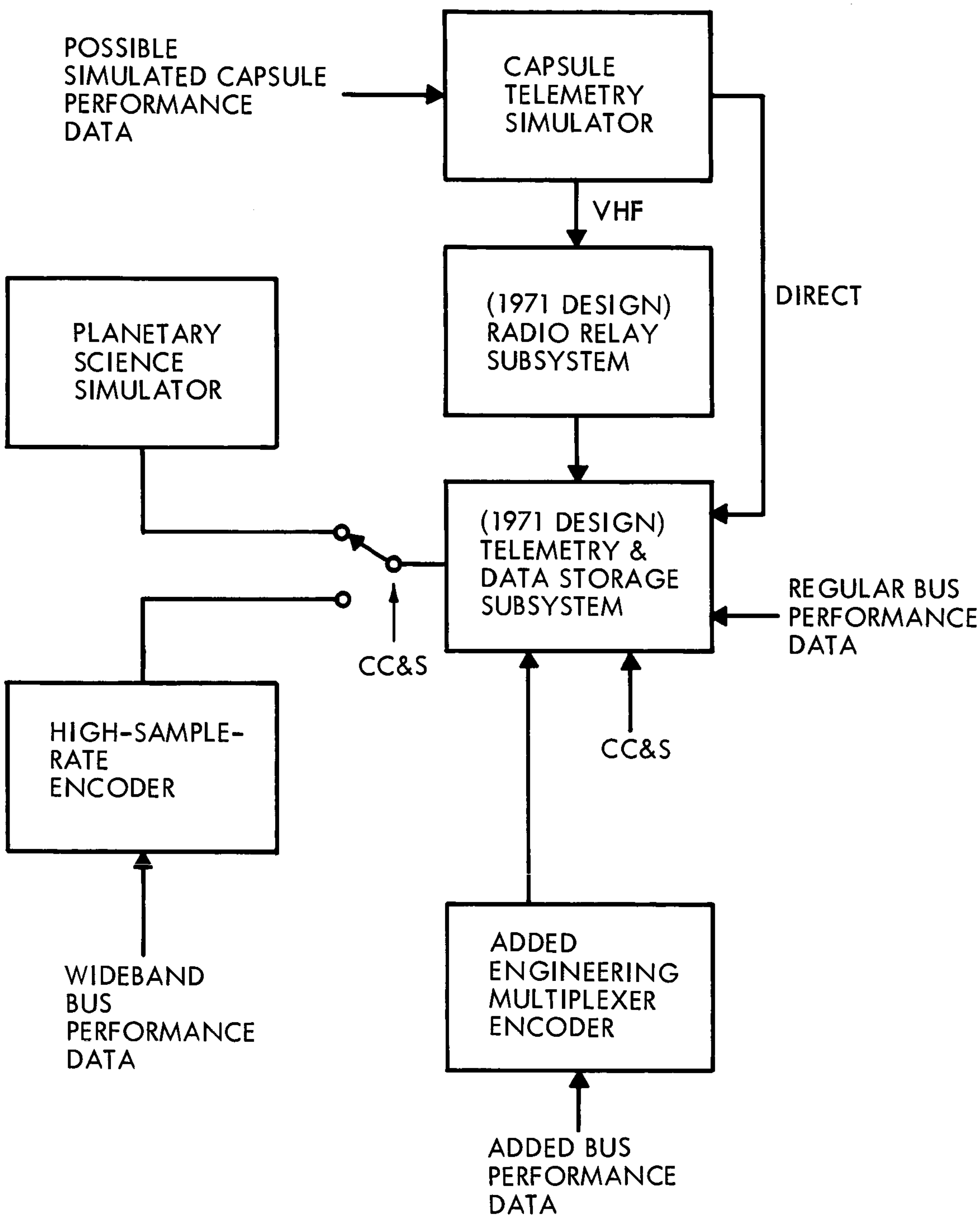


Figure III-2: 1969 Test Flight Telemetry Subsystem Function clock to the shift register and will not change the ground data processing equipment. In this way, the planetary science data channel can be evaluated. Exact configuration of the simulator cannot be established without more knowledge of the data formats.

III-3.6.2 1969 Test Flight Benefits

Since the 1969 Test Flight telemetry equipment will be identical to the 1971 Voyager spacecraft equipment except for the incorporation of the capability of multiplexing additional engineering measurements; the 1969 Test Flight will greatly intensify and accelerate experience with and evaluation of the 1971 Voyager Spacecraft telemetry equipment and techniques.

III-3.7 1969 TEST SPACECRAFT FLIGHT EQUIPMENT TELEMETRY CHANNEL LIST--SATURN IB/CENTAUR

III-3.7.1 Description

The flight data measurement requirement in Table III-2 comprise the required telemetry measurements for the Saturn IB/Centaur-launched 1969 Test Spacecraft.

Table III-2: FLIGHT DATA MEASUREMENT LIST

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	ANTENNA					
1	High-Gain Antenna Hinge Angle	Angle	0-360°	1/2°	10 bit	1/60
2	High-Gain Antenna Swivel Angle	Angle	0-360°	1/2°	10 bit	1/60
3	High-Gain Antenna rf Input	rf Pwr	0-50 w	2%	0-50 mw	1/600
4	Low-Gain Angenna rf Input	rf Pwr	0-50 w	2%	0-50 mw	1/600
5	High-Gain Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200
6	Low-Gain Antenna Deploy	Position	yes/no	discrete	1 bit	1/1200
7	High-Gain Antenna Hinge Motor Temp	Temp	±300°F	2%	±2.5 v	1/1200
8	High-Gain Antenna Swivel Motor Temp	Temp	±300°F	2%	±2.5 v	1/1200
9	High-Gain Antenna Hinge Torque	Inch-Pounds	0-100	2%	0-5 v	1/60
10	High-Gain Antenna Swivel Torque	Inch-Pounds	0-100	2%	0-5 v	1/60
11-18	High-Gain Antenna Dish Temperature	Temp	±150°F	2%	±2.5 v	1/60
	RADIO					
19	Receiver AGC (Coarse)	Signal Strength	-90 to -162 dbm	2%	0-5 v	1/60
20	Receiver AGC (Fine)	Signal Strength	-125 to -150 dbm	2%	0-5 mv	1/600
21	Receiver Static Phase Error	Phase Angle	±20°	2%	±2.5 v	1/60

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	RADIO (Continued)					
22	Receiver LO Drive	Osc Pwr	0-1 mw	2%	0-50 mw	1/600
23	Exciter Output Power	Osc Pwr	0-0.2 w	2%	0-5 v	1/600
24	TWT Helix Voltage	Volts	0-2 kv	2%	0-5 v	1/1200
25	TWT Helix Current	ma	0-20 ma	2%	0-5 v	1/1200
26	TWT Coil Current	ma	0-100 ma	2%	0-5 v	1/1200
27	Exciter Voltage 1	Volts	0 to -25 volts	2%	± 2.5 v	1/1200
28	Exciter Voltage 2	Volts	0 to -15 volts	2%	± 2.5 v	1/1200
29	Crystal OSC Temperature (oven)	Temp	25 to $150^{o}F$	2%	0-5 v	1/1200
30	Command Det Mon	Status	yes/no	discrete	1 bit	1/60
31	TWT A Temp	Temp	$\pm 150^{o}F$	5%	± 2.5 v	1/1200
32	TWT B Temp	Temp	$\pm 150^{o}F$	5%	± 2.5 v	1/1200
33	Receiver A Temp	Temp	$\pm 150^{o}F$	5%	± 2.5 v	1/1200
34	Receiver B Temp	Temp	$\pm 150^{o}F$	5%	± 2.5 v	1/1200
	RELAY RADIO					
35	Receiver AGC	Signal Strength	-90 to -150 dbm	2	0-5 v	1/600
36	Receiver Static Phase Error	Phase Angle	$\pm 20^{o}$	2	± 2.5 v	1/600
37	Receiver LO Drive	Power	0-1 mw	2	0-5 v	1/600
38	Receiver Temperature	Temp	$\pm 150^{o}F$	2	± 2.5 v	1/1200

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	<u>TELEMETRY AND DATA STORAGE SYSTEM</u>					
39	Recorder "A" Pressure	x port press	0-4 psia	2%	0-5 v	1/1200
40	Mode Status	Modes	0-4 psia	discrete	2 bits	1/60
41	Event Counter A	Event	0-100	discrete	7 bits	1/60
42	Event Counter B	Event	0-100	discrete	7 bits	1/60
43	Recorder A Temp	Temp	$\pm 150^{\circ}F$	5%	± 2.5 v	1/1200
44-50	Electronics Package Temp	Temp	$\pm 150^{\circ}F$	2%	± 2.5 v	1/1200
	<u>ATTITUDE REFERENCE AND AUTOPILOT</u>					
51	Pri Sun Sen Fine Pitch Error			5%	0-5 v	1/60
52	Pri Sun Sen Fine Yaw Error			5%	0-5 v	1/60
53	Pri Sun Sen Coarse Pitch Error			5%	0.5 v	1/60
54	Pri Sun Sen Coarse Yaw Error			5%	0.5 v	1/60
55	Sec Sun Sen Fine Pitch Error			5%	0.5 v	1/60
56	Sec Sun Sen Fine Yaw Error			5%	0.5 v	1/60
57	Sec Sun Sen Coarse Pitch Error			5%	0.5 v	1/60
58	Sec Sun Sen Coarse Yaw Error			5%	0.5 v	1/60

TABLE III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rate Samples/Second
	<u>ATTITUDE REFERENCE AND AUTOPILOT (Continued)</u>					
59	Sun Sensor Pitch Error			5%	0.5 v	1/60
60	Sun Sensor Yaw Error			5%	0.5 v	1/60
61	Sun Sensor Yaw Acquisition			5%	discrete	1/60
62	Sun Sensor Pitch Acquisition			5%	discrete	1/60
63	Pri Sun Sen Pitch Acquisition			5%	discrete	1/60
64	Pri Sun Sen Yaw Acquisition			5%	discrete	1/60
65	Sec Sun Sen Pitch Acquisition			5%	discrete	1/60
66	Sec Sun Sen Yaw Acquisition			5%	discrete	1/60
67	Canopus Pitch Gimbal Angle			5%	0-5 v	1/60
68	Canopus Acquisition			5%	discrete	1/60
69	Canopus Roll Error			5%	$\pm$2.5 v	1/60
70	Canopus Star Magnitude			5%	$\pm$2.5 v	1/60
71	Pri Canopus Sen Acquisition			5%	discrete	1/60
72	Sec Canopus Sen Acquisition			5%	discrete	1/60
73	Pri Canopus Sen Roll Error			5%	$\pm$2.5 v	1/60

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	<u>ATTITUDE REFERENCE AND AUTOPILOT</u> (Continued)					
74	Sec Canopus Sen Roll Error			5%	±2.5 v	1/60
75	Pri Canopus Star Magnitude			5%	±2.5 v	1/60
76	Sec Canopus Star Magnitude			5%	±2.5 v	1/60
77	Attitude Ref Case Temp			1%	0-5 v	1/60
78	IRV Pwr Supply 1			5%	0-5 v	1/60
79	IRV Pwr Supply 2			5%	0-5 v	1/60
80	IRV Pwr Supply 3			5%	0-5 v	1/60
81	Roll Gyro Sel Verify			5%	discrete	1/60
82	Pitch Gyro Sel Verify			5%	discrete	1/60
83	Yaw Gyro Sel Verify			5%	discrete	1/60
84	Roll Gyro Rate			5%	0-5 v	1/60
85	Pitch Gyro Rate			5%	0-5 v	1/60
86	Yaw Gyro Rate			5%	0-5 v	1/60
87	Roll Gyro "A" Pos			5%	0-5 v	1/60
88	Roll Gyro "B" Pos			5%	0-5 v	1/60

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	ATTITUDE REFERENCE AND AUTOPILOT (Continued)					
89	Pitch Gyro "A" Pos			5%	0-5 v	1/60
90	Pitch Gyro "B" Pos			5%	0-5 v	1/60
91	Yaw Gyro "A" Pos			5%	0-5 v	1/60
92	Yaw Gyro "B" Pos			5%	0-5 v	1/60
93	EMA Accel + Output			5%	0-5 v	1/60
94	EMA Accel - Output			5%	0-5 v	1/60
95	Bell Accel + Output			5%	7 bit	1/60
96	Bell Accel - Output			5%	7 bit	1/60
97	EMA d.c. Amp			5%	0-5 v	1/60
98	Bell d.c. Amp			5%	0-5 v	1/60
99	Trisafe Amp 1 Output			5%	0-5 v	1/60
100	Trisafe Amp 2 Output			5%	0-5 v	1/60
101-108	Trisafe Amp 2 Output			5%	0-5 v	1/60
109-114	Trisafe Pwr Supply 1			5%	0-5 v	1/60
115-130	Jet Vane 1 Feedback			5%	0-5 v	1/60

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	CENTRAL COMPUTER AND SEQUENCER					
131-137	Programmer Data	Mode Cond	Instruct Words	absolute	22 bits	1/60
138	Programmer Parity	Ind	yes/no	discrete	1 bit	1/60
139	Vehicle Time	Time	233 hrs	1/10 sec	22 bits	1/60
140	Command Verify	Verify	Command Word	absolute	26 bits	1/60
141	A/C Mode Select	Mode Cond	yes/no	discrete	3 bits	1/60
142	Power Mode Select	Cond	yes/no	discrete	1 bit	1/600
143	Unit Temperature	Temp	±150°	1%	±2.5 v	1/1200
144	Science Data Mode Select	Mode Cond	yes/no	discrete	3 bits	1/60
145	Command Message Identification	Mode Cond	yes/no	discrete	6 bits	1/60
146	Processor 1, 2	Mode Cond	yes/no	discrete	1 bit	1/60
147	Command Decoder 1, 2	Mode Cond	yes/no	discrete	1 bit	1/60
	REACTION CONTROL					
148	A/C Tank 1 Pressure	psia	0-375	2%	0-5 v	1/60
149	A/C Tank 2 Pressure	psia	0-375	2%	0-5 v	1/60
150	Engine Pitch Act + (OIP)	Angle	±5°	2%	±2.5 v	1/600

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	REACTION CONTROL (Continued)					
151	Engine Pitch Act + (MCP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600
152	Engine Yaw Act + (OIP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600
153	Engine Yaw Act + (MCP)	Angle	$\pm 5^{\circ}$	2%	± 2.5 v	1/600
154	Jet Driver Pitch +	Cond	yes/no	discrete	1 bit	1/1200
155	Jet Driver Pitch -	Cond	yes/no	discrete	1 bit	1/1200
156	Jet Driver Yaw +	Cond	yes/no	discrete	1 bit	1/1200
157	Jet Driver Yaw -	Cond	yes/no	discrete	1 bit	1/1200
158	Jet Driver Roll +	Cond	yes/no	discrete	1 bit	1/1200
159	Jet Driver Roll -	Cond	yes/no	discrete	1 bit	1/1200
160	A/C Gas Temp 1	Temp (Farenheit)	$\pm 150^{\circ}$	1%	0-5 v	1/1200
161	A/C Gas Temp 2	Temp (Farenheit)	$\pm 150^{\circ}$	1%	0-5 v	1/1200
	ELECTRICAL POWER					
162	Solar Panel 1 Current	Amp	0-10 A	2%	0-5 v	1/60
163	Solar Panel 2 Current	Amp	0-10 A	2%	0-5 v	1/60
164	Solar Panel 3 Current	Amp	0-10 A	2%	0-5 v	1/60

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	ELECTRICAL POWER (Continued)					
165	Battery B1 Voltage	Volts	0-50 v	2%	0-5 v	1/60
166	Battery B2 Voltage	Volts	0-50 v	2%	0-5 v	1/60
167	Battery B3 Voltage	Volts	0-50 v	2%	0-5 v	1/60
168	Battery B1 Discharge	Amp	0-15 A	2%	0-5 v	1/60
169	Battery B2 Discharge	Amp	0-15 A	2%	0-5 v	1/60
170	Battery B3 Discharge	Amp	0-15 A	2%	0-5 v	1/60
171	Battery B1 Charge	Amp	0-5 A	2%	0-5 v	1/60
172	Battery B2 Charge	Amp	0-5A	2%	0-5 v	1/60
173	Battery B3 Charge	Amp	0-5 A	2%	0-5 v	1/60
174	Unregulated d.c. Bus Current	Amp	0-30 A	2%	0-5 v	1/60
175	Unregulated d.c. Bus Voltage	Volt	0-50 v	2%	0-5 v	1/60
176	Prime Telecom Regulator	Amp	0-15 A	2%	0-5 v	1/600
177	Standby Telecom Regulator	Amp	0-15 A	2%	0-5 v	1/600
178	Prime Spacecraft Regulator	Amp	0-15 A	2%	0-5 v	1/600
179	Standby Spacecraft Regulator	Amp	0-15 A	2%	0-5 v	1/600

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	<u>ELECTRICAL POWER</u> (Continued)					
180	Regulated d.c. Bus (Spacecraft)	Volt	0-50 v	2%	0-5 v	1/600
181	Regulated d.c. Bus (Telecom)	Volt	0-50 v	2%	0-5 v	1/600
182	400 cps 1 ϕ Bus Voltage	Volt	0-30 v	2%	0-5 v	1/600
183	400 cps 1 ϕ Inverter Current	Amp	0-3 A	2%	0-5 v	1/600
184	400 cps 3 ϕ Bus Voltage	Volt	0-30 v	2%	0-5 v	1/600
185	400 cps 3 ϕ Inverter Current	Amp	0-3 A	2%	0-5 v	1/600
186	2400 cps Inverter Prime	Amp	0-5 A	2%	0-5 v	1/600
187	2400 cps Inverter Standby	Amp	0-5 A	2%	0-5 v	1/600
188	Power Sync Frequency	Cycles/ Second	350 to 450 cps	2%	0-5 v	1/600
189	Power Sync Frequency	Kilocycles/ Second	2.2 to 2.6 kc	2%	0-5 v	1/600
190	Solar Panel 1 Temp	^{o}F	$\pm150^{o}$	5%	±2.5 v	1/1200
191	Solar Panel 2 Temp	^{o}F	$\pm150^{o}$	5%	±2.5 v	1/1200
192	Solar Panel 3 Temp	^{o}F	$\pm150^{o}$	5%	±2.5 v	1/1200
193	Regulator Output Ripple	Milli- volts	0-500 mv.	2%	0-5 v	1/1200
194	Battery B1 Temp	^{o}F	$\pm150^{o}$	5%	±2.5 v	1/1200

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	ELECTRICAL POWER (Continued)					
195	Battery B2 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
196	Battery B3 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
197	Telecom Regulator 1 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
198	Telecom Regulator	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
199	Spacecraft Regulator 1 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
200	Spacecraft Regulator 2 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
201	PS&L Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
202	2.4 kc Inverter 1 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
203	2.4 kc Inverter 2 Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
204	400 cps 1 ϕ Inverter Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
205	400 cps 3 ϕ Inverter Temp	^oF	$\pm 150^o$	5%	± 2.5 v	1/1200
206-215	Relay RL-1 thru RL-10 Status	Cond	yes/no	discrete	10 bits Total	1/1200
	THERMAL CONTROL					
216	Louver 1 Position	Angle	$0-90^o$	2%	0-5 v	1/1200
217	Louver 2 Position	Angle	$0-90^o$	2%	0-5 v	1/1200

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	THERMAL CONTROL (Continued)					
218	Louver 3 Position	Angle	$0-90^{\circ}$	2%	0-5 v	1/1200
219	Louver 4 Position	Angle	$0-90^{\circ}$	2%	0-5 v	1/1200
220	Louver 5 Position	Angle	$0-90^{\circ}$	2%	0-5 v	1/1200
221	Louver 6 Position	Angle	$0-90^{\circ}$	2%	0-5 v	1/1200
222	Coldplate 1 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
223	Coldplate 2 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
224	Coldplate 3 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
225	Coldplate 4 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
226	Coldplate 5 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
227	Coldplate 6 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
228	Coldplate 7 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
229	Coldplate 8 Temp	Temp	$\pm 150^{\circ}$	5%	± 2.5 v	1/1200
230	M/C Motor Shield Temp	Temp	$0-1K^{\circ}F$	5%	0-5 v	1/1200
231	O/I Motor Shield Temp	Temp	$0-1K^{\circ}F$	5%	0-5 v	1/1200
232-261	Spacecraft Temperatures			(to be determined)		

Table III-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	<u>PROPULSION</u>					
262	Gas Supply Pressure	psia	0-4000	5%	0-5 v	1/60
263	Fuel Tank Pressure	psia	0-400	5%	0-5 v	1/60
264	Gas Supply Temperature I	^{o}R	300-600	5%	0-5 v	1/60
265	Gas Supply Temperature II	^{o}R	300-600	5%	0-5 v	1/60
266	Fuel Tank Temperature I	^{o}R	400-600	5%	0-5 v	1/600
267	Fuel Tank Temperature II	^{o}R	400-600	5%	0-5 v	1/600
268	Motor Temperature	^{o}R	400-1500	5%	0-5 v	1/60
269	Engine Valve Driver-Fuel	Cond	yes/no	discrete	1 bit	1/60
270-310	M/C Motor Chamber Pressure	psia	C-15 k	2%	0-5 v	1/60
	<u>MECHANISMS</u>					
311	Solar Panel 1 Deploy	Cond	yes/no	discrete	1 bit	1/600
312	Solar Panel 2 Deploy	Cond	yes/no	discrete	1 bit	1/600
313	Solar Panel 3 Deploy	Cond	yes/no	discrete	1 bit	1/600
314	Magnetometer Boom Deploy	Cond	yes/no	discrete	1 bit	1/600

Table II-2: FLIGHT DATA MEASUREMENT LIST (Continued)

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
	MECHANISMS					
315	Scan Platform Position	Angular Position	0-90°	2%	0-5 v	1/600
316	Scan Platform Deploy	Cond	yes/no	discrete	1 bit	1/1200
317	Optics Cover Position	Cond	yes/no	discrete	1 bit	1/1200
318	Gimbal Motor 1 Temp	Temp	±150°F	2%	±2.5 v	1/1200
319	Gimbal Motor 2 Temp	Temp	±150°F	2%	±2.5 v	1/1200
320-326	Scan Platform Axis I Drive Torque	Inch-Pound	0-100	2%	0-5 v	1/60
327-331	Scan Platform Axis II Drive Torque	Inch-Pound	0-100	2%	0-5 v	1/60

III-3.8 1969 TEST SPACECRAFT GUIDANCE AND NAVIGATION MANEUVER ERROR-ALLOCATIONS AND ANALYSIS--SATURN IB/CENTAUR

III-3.8.1 Description

The detailed allocation of maneuver error to component error sources will not vary for the 1969 Test Spacecraft from that described in Section 3.8 of Volume A, D2-82709-1, "1971 Preferred Design."

III-3.9 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, FLIGHT SEQUENCE--SATURN IB/CENTAUR

III-3.9.1 Description

The 1969 Test Spacecraft functions are basically the same as those described in Section 3.9 of Volume A, D2-82709-1. The test flight profile is divided into 11 gross functional areas shown in Figure III-3. A detailed expansion of the 13 test spacecraft subsystems are shown in Figure III-4.

In summary, the 1969 Saturn IB/Centaur Test Flight configuration is identical to the 1971 Mars orbital mission except that:

1) There is no science payload;

2) The Flight Capsule is replaced by a simulated Flight Capsule.

III-3.9.2 1969 Test Flight Benefits

The 1969 Saturn IB/Centaur Test Flight provides an opportunity to prove flight component functions well in advance of the 1971 mission, thereby guarding against any unforseen problems that might prejudice 1971 mission performance.

1969 TEST FLIGHT SIB/CENTAUR	PRELAUNCH AT ETR	LAUNCH AND INJECTION	ACQUISITION	I C
LAUNCH-VEHICLE SYSTEM	System checkout; mating; and combined systems testing	Temperature control for spacecraft; Telemetry from test spacecraft; Shroud separation; Orientation in parking orbit through spacecraft separation; assurance that Launch Vehicle meets non-impact requirements		
SPACECRAFT SYSTEM (See Figure III-4 for a detailed expansion)	System checkout; Spacecraft simulated capsule mating and combination testing; fuel and install pyrotechnics at Explosive Safe Facility; Mate to adapter and shroud; transport to pad; Launch Vehicle mate and testing; Power to subsystems and simulated capsule	Temperature control after shroud off; Telemetry to launch vehicle; Separate from Launch Vehicle; Make engineering measurements; Environmental control; Power to subsystems	Acquire reference objects; Provide power to subsystems; Environmental control; Provide RF signal to Earth; Make engineering measurements	A p P t e s c
CAPSULE SYSTEM	Final checkout of simulated capsule circuits and final sterilization at Explosive Safe Facility	Telemetry to Spacecraft; Temperat Make engineering measurements		
MISSION OPERATIONS SYSTEM	Checkout and training; Support Mission Operations System	Monitor telemetry; Receive and evaluate Eastern Test Range prediction	Monitor telemetry; Send backup commands; Send Deep Space Network Station prediction	
LAUNCH OPERATIONS SYSTEM	Scheduling and coordination of Eastern Test Range activities	Control launch; Monitor telemetry; Provide tracking to injection		
DEEP SPACE NETWORK	Operational readiness test; Spacecraft Deep Space Network compatibility test	Provide tracking as required; Support Mission Operations System		

NTERPLANETARY RUISE	INTERPLANETARY TRAJECTORY CORRECTION	SPACECRAFT SIMULATED CAPSULE SEPARATION AND DEFLECTION	SIMULATED CAPSULE CRUISE (AWAY FROM MARS)
ttitude control; ower to subsystems; rovide RF signal o Earth; Make ngineering mea- urements; Envir- nmental control	Provide power to subsystems; Orient thrust vector; Provide ΔV; Temperature control; Make engineering measurements; Provide RF signal to Earth; Reacquisition	Provide preseparation commands to simulated capsule; Provide telemetry relay from simulated capsule; Orient separation vector; Separate sterilization canister and maintain contamination requirements; Maintain temperature control; Make engineering measurements; Provide RF signal to Earth	Provide commar to and from simulated capsu
ure control after shroud off;		Provide separation-sequence simulation signals; Separate from spacecraft	Provide simula signals to spacecraft
Monitor telemetry and control planetary vehicle, as necessary; Determine orbit characteristics	Monitor telemetry; Determine and send midcourse command; Control spacecraft as necessary; Determine orbit characteristics	Monitor telemetry; Determine and send separation and deflection commands; Determine orbit characteristics	Determine orbit characteristics of simulated capsule until missi termination
Provide two-way communication lock; Provide tracking support (Miss			

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2

	PREPARE FOR MARS ENCOUNTER	TEST SPACECRAFT ORBIT INSERTION	TEST SPACECRAFT ORBIT OPERATIONS	ANALYZE DATA AND REPORT
ıds le	Provide power to subsystesm; Environmental control; Maintain attitude reference; Provide RF signal to Earth; Prepare for orbital insertion sequence	Orient retrothrust vector; Confirm attitude to Earth; Provide ΔV; Provide power & temperature control; Make engineering measurements; Provide RF signal; Reacquisition	Orient planetary instrumentation; Make engineering measurements; Store and read out data; Attitude control; Provide power, temperature control, RF signal, & environmental control	
ed				
ɔn	Monitor telemetry; control spacecraft as necessary; Determine orbit characteristics			Analyze all data and prepare report on mission after termination
	ion Operations System)			

gure III-3: 1969 Saturn S-IB/Centaur Orbiter — Mission Profile

3

1.0 FRELAUNCH AT ETR → 2.0 LAUNCH & I (INCLUDES (

GROUND COMPLEX (MOS, LOS & DSN)	A) OPERATIONAL READINESS TEST, S/C-DSN B) COMPATIBILITY TEST C) SCHEDULE AND COORDINATE ETR ACTIVITIES D) C/O & SUPPORT MOS (DSN ONLY) E) LOAD CC&S WITH FLIGHT PROGRAM	MOS/LOS A) COMMAND LIFTOFF B) TRACK VEHICLE DURING BOOST C) SUPPLY FLIGHT COMMANDS (AS REQUIRED) D) RECEIVE & ANALYZE DATA FROM S/C & BOOSTER DSN A) STANDBY ON ALERT B) COMMUNICATE WITH ETR CH UP A. B. C. D. E.
SPACECRAFT TELECOMMUNICATIONS	SUBSYSTEM C/O & STATUS MONITORING	A) TRANSMIT ENGINEERING DATA VIA B) TRANSMIT ENGINEERING DATA VIA C) RECEIVE POWER FROM E/P
CENTRAL COMPUTER AND SEQUENCER (CC&S)	A) SUBSYSTEM C/O & STATUS MONITORING B) COMMAND OTHER SUBSYSTEMS FOR C/O & STATUS MONITORING C) READY ALL SUBSYSTEMS FOR LAUNCH D) LOAD CC&S WITH FLIGHT PROGRAM	PROVIDE BACKUP COMMANDS AS RFQ
ATTITUDE REFERENCE SUBSYSTEM AUTOPILOT SUBSYSTEM REACTION CONTROL SUBSYSTEM (RCS)	SUBSYSTEM C/O & STATUS MONITORING	A) ATTITUDE REFERENCE GYROS OFF B) AUTOPILOT — OFF C) RCS — OFF
MIDCOURSE CORRECTION PROPULSION SYSTEM ORBIT INJECTION PROPULSION SYSTEM	SUBSYSTEM C/O & STATUS MONITORING	
ELECTRICAL POWER SUBSYSTEM	SUBSYSTEM C/O & STATUS MONITORING	A) PROVIDE ENGINEERING DATA FOR TE B) PROVIDE BATTERY POWER TO. • TELECOMMUNICATIONS • AUTOPILOT SUBSYSTEM • CC&S • •
S/C STRUCTURE SUBSYSTEM S/C MECHANISMS SUBSYSTEM INSTALLATION CABLES & TUBING	SUBSYSTEM C/O & STATUS MONITORING	A) PROVIDE PHYSICAL SUPPORT FOR AL B) PROVIDE ATTACHMENT FOR SIMULATE C) SUPPORT FLIGHT CAPSULE
TEMPERATURE CONTROL SUBSYSTEM	A) SUBSYSTEM C/O & STATUS MONITORING B) TEST SPACECRAFT COOLING SUPPLIED BY CENTAUR	A) PROVIDE HEAT SINK COOLING CAPAB B) TEMPERATURE CONTROL AFTER SHROU
PYROTECHNIC SUBSYSTEM	SUBSYSTEM C/O & STATUS MONITORING	
ENGINEERING INSTRUMENTATION	SUBSYSTEM C/O & STATUS MONITORING	ENGINEERING INSTRUMENTATION O
DATA AUTOMATION EQUIPMENT (GFE)		

REPEAT UP TO FOU

(IN)JECTION (C)OUNTDOWN) → 3.0 ACQUISITION → 4.0 INTERPLANETARY CRUISE →

	3.0 ACQUISITION	4.0 INTERPLANETARY CRUISE	
(…)NGE FROM MOS/LOS TO DSN CONTROL (O)N INJECTION INTO TRANSMARS ORBIT PROVIDE SFCF/DSIF WITH ANTENNA SEARCH DATA SN RECEIVE ANTENNA SEARCH DATA SEARCH FOR & ACQUIRE S/C ESTABLISH & VERIFY CONTROL OF S/C TRACK S/C (ONE WAY) RECEIVE & ANALYZE ENGR. DATA	DSN A) MONITOR BOOSTER SEPARATION B) MONITOR SOLAR PANEL DEPLOYMENT C) MONITOR ANTENNA DEPLOYMENT D) PROVIDE COMMANDS TO BACKUP CC&S AS REQUIRED E) TRACK S/C F) REC DATA G) MONITOR & VERIFY ACQUISITION OF SUN H) MONITOR & VERIFY ACQUISITION OF CANOPUS I) COMMAND REACQUISITION OF CANOPUS (AS REQUIRED) J) MONITOR S/C TRAJECTORY K) UPDATE CC&S TRAJECTORY PARAMETERS FOR PITCH, YAW & ROLL L) COMPUTE CC&S PITCH, YAW & ROLL POLARITY	DSN A) TRACK S/C B) RECEIVE ENGINEERING DATA C) MONITOR S/C & SIMULATED CAPSULE STATUS D) PROCESS DATA ON EARTH TO OBTAIN GUIDANCE COMMANDS (STORED COMMANDS & START TIMES)	A) B) C) D) E) F)
(CE)NTAUR TELEMETRY (LO)W-POWER LAUNCH EXCITER	A) TRANSMIT ENGINEERING DATA VIA LOW-POWER LAUNCH EXCITER B) RECEIVE POWER FROM E/P C) DETECT AND SEND TO CC&S COMMAND SIGNALS FROM EARTH D) TRANSMIT CELESTIAL REFERENCE ACQUISITION TO DSN E) TRANSMIT VERIFICATION OF DEPLOYMENT OF SOLAR PANELS, ANTENNAS, ETC TO EARTH F) TWO-WAY TRACKING	A) TRANSMIT ENGINEERING DATA VIA DATA MODE NO. 2 (OPTIONAL MODE NO. 3) B) TRANSMIT SIMULATED CAPSULE ENGINEERING DATA C) TRANSMIT ANGLE SETTINGS (CANOPUS TO EARTH) D) EXERCISE & CALIBRATE HIGH-GAIN ANTENNA T + 40 DAYS E) SWITCH FROM LOW-GAIN TO HIGH-GAIN ANTENNA T + 80 DAYS F) PROVIDE RANGING SIGNAL TO A MAXIMUM RANGE OF 8.0×10^6 km (NOMINAL) G) SWITCH TRANSMISSION FROM LAUNCH EXCITER TO TWT POWER AMPLIFIER AND DATA MODE NO. 2	A) B) C) D) E) F) G) H) I)
IRED	A) COMMAND SIGNALS TO PYROTECHNICS & MECH. TO DEPLOY: • SOLAR PANELS • HI-GAIN ANTENNA • VHF ANTENNA • LO-GAIN ANTENNA B) SWITCH ON GYROS AND SELECT MODES C) ENABLE SUN SENSOR ROLL AND PITCH CONTROL D) RECEIVE SUN PRESENCE SIGNAL E) TURN ON CANOPUS TRACKER F) SWITCH ROLL CONTROL TO CANOPUS SENSOR G) RECEIVE CANOPUS PRESENCE OUTPUT SIGNAL H) TRANSMIT VERIFICATION TO TELECOMMUNICATION SUBSYSTEM (CANOPUS & SUN) I) PERFORM COMMAND FUNCTIONS FOR CANOPUS OVERRIDE AS REQUIRED J) BACKUP COMMANDS AS REQUIRED	A) COMMAND TELECOMMUNICATION TO DATA MODE NO. 2 B) COMMAND CRUISE ENGINEERING INSTRUMENTS ON (WARMUP) C) COMMAND DATA RECORDERS ON D) COMMAND TWTA ON E) SWITCH CANOPUS ANGLE AS REQUIRED F) INITIATE CRUISE ENGINEERING DATA ACQUISITION G) COMMAND TELECOMMUNICATION CHANGE FROM OMNI TO HI-GAIN ANTENNA H) UPDATE HI-GAIN ANTENNA POSITION (AS REQUIRED) I) COMMAND TELECOMMUNICATIONS - CHANGE DATA TRANSMISSION RATES J) COMMAND RECALIBRATION OF ENGINEERING INSTRUMENTS AS REQUIRED	A) B) C) D) F) F) G) H) I) J) K) L) M) N)
(DU)RING LAUNCH	A) RECEIVE CC&S COMMAND, SWITCH ON GYROS, RCS & AUTOPILOT B) DAMP ROTATION C) GYRO CONTROL YAW & PITCH SPACECRAFT TO ACQUIRE SUN D) RELAY SUN ACQUISITION SIGNAL TO CC&S E) TURN ON CANOPUS SENSOR (AUTOPILOT CONTROL), ROLL 360° TO PROVIDE STAR MAP F) ROLL TO ACQUIRE CANOPUS G) RELAY ACQUISITION SIGNAL (CANOPUS TO CC&S) H) PERFORM CANOPUS OVERRIDE ROLL MANEUVER AS REQUIRED	A) UPDATE CANOPUS-CONE-ANGLE ON COMMAND B) SWITCH AUTOPILOT TO CRUISE MODE C) MAINTAIN S/C ATTITUDE TO CELESTIAL REFERENCE DURING CRUISE	A) B) C) D) E) F) G) H) I)
			MI(D…) A) B) C) D) E) F)
(TEL)ECOMMUNICATIONS SUBSYSTEM (AT)TITUDE REFERENCE SUBSYSTEM (EN)GINEERING INSTRUMENTATION	A) PROVIDE POWER TO PYROTECHNIC SUBSYSTEMS FOR SQUIB FIRINGS B) PROVIDE POWER TO MECHANISMS C) ACTIVATE SOLAR POWER SYSTEM AFTER SUN ACQUISITION (AUTOMATIC) D) TRANSMIT "VOLTAGE SATISFACTORY" SIGNAL TO CC&S E) PROVIDE POWER TO TELEMETRY, CC&S, THERMAL CONTROL, AUTOPILOT & ENGRG INSTRUMENTATION	A) PROVIDE ELECTRICAL SOLAR POWER TO: • TELEMETRY • TEMPERATURE CONTROL • AUTOPILOT • SIMULATED CAPSULE • CC&S • ATTITUDE REFERENCE • ENGR INSTR B) CHARGE BATTERIES	A)
EQUIPMENT CAPSULE	A) PROVIDE PHYSICAL SUPPORT FOR ALL EQUIPMENT B) DRIVE SOLAR PANELS TO LIMIT STOPS C) PROVIDE OUT & LOCK SIGNALS TO TELEMETRY D) DRIVE HIGH-GAIN ANTENNA TO OPERATING POSITION & LOCK E) DRIVE VHF & OMNI ANTENNAS TO OPERATING POSITION & LOCK F) DRIVE MAGNETOMETER BOOM TO OPERATING POSITION & LOCK G) SUPPORT SIMULATED CAPSULE	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D) PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS E) SUPPORT SIMULATED CAPSULE	A) B) C) D) E)
(…)LITY UP TO SHROUD JETTISON JETTISON	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL	
	RECEIVE SQUIB FIRING SIGNALS FOR DEPLOYMENT OF • SOLAR PANEL • LOW-GAIN ANTENNA • HI-GAIN ANTENNA • VHF ANTENNA • MAGNETOMETER BOOM		A) B) C)
	ENGINEERING INSTR ON	A) RECEIVE "CRUISE ENGINEERING INSTRUMENTS ON" COMMAND B) ACTIVATE CRUISE ENGINEERING INSTRUMENTATION C) TRANSFER DATA TO DAE D) RECEIVE POWER FROM E/P E) RECALIBRATE ENGINEERING INSTRUMENTATION AS REQUIRED	A) B)
		A) RECEIVE DAE "ON" COMMANDS B) PROCESS AND TRANSFER DATA TO TM	

2

TIMES

5.0 INTERPLANETARY TRAJECTORY CORRECTION

6.0 SPACECRAFT-SIMULATED CAPSULE SEPARATION & DEFLECTION

7.0 SIMULATED CAPSULE CRUISE AWAY FROM MARS

8.0 PREPARE FOR MARS ENCOUNTER

9.0 TEST S ORBIT

5.0	6.0	7.0 / 8.0	9.0
PROVIDE ROLL TURN MAGNITUDE & DIRECTION PROVIDE YAW TURN MAGNITUDE & DIRECTION PROVIDE MIDCOURSE MOTOR BURN DURATION AND ΔV PROVIDE INITIATE MANEUVER SEQUENCE COMMAND RECEIVE VERIFICATION OF CORRECTION MANEUVER DEGREES OF YAW & ROLL THRUST DURATION REACQUISITION OF CELESTIAL REFERENCES TRACK SPACECRAFT	A) INITIATE PRESEPARATION SEQUENCE B) RECEIVE SIMULATED CAPSULE-STATUS DATA C) COMMAND SEPARATION D) VERIFY SIMULATED CAPSULE PROGRAM E) VERIFY SEPARATION F) TRACK SIMULATED CAPSULE & SPACECRAFT	A) TRANSMIT ORBIT INJECTION PARAMETERS • ROLL MAGNITUDE & DIRECTION • YAW MAGNITUDE & DIRECTION • MOTOR-BURN START TIME B) TRACK FLIGHT SPACECRAFT C) RECEIVE, REDUCE, & DISPLAY ENGINEERING DATA D) PROVIDE COMMANDS TO BACK UP CC&S AS REQUIRED	A) RECEIVE, DECODE, & DISPL B) RELAY UPDATED ORBIT INJE C) PROVIDE COMMAND TO INI D) PROVIDE BACKUP COMMAN E) TRACK S/C F) VERIFY ORBIT INSERTION
RECEIVE ROLL AND YAW TURN COMMAND & RELAY TO CC&S RECEIVE ANTENNA POSITIONING DATA, RELAY TO CC&S RECEIVE MOTOR BURN DURATION AND ΔV, RELAY TO CC&S RECEIVE INITIATE MANEUVER COMMAND & RELAY TO CC&S SWITCH TO DATA MODE NO. 1 (FOR LATE MIDCOURSE MANEUVERS) RELAY VERIFICATION OF CORRECTION MANEUVER TO DSN TRANSMIT ENGR DATA STORE AND/OR TRANSMIT ENGINEERING DATA TO DSN (MODE NO. 3) TRANSMIT ENGINEERING (SPACECRAFT & CAPSULE) PLUS CRUISE ENGINEERING TO DSN (MODE NO. 2)	A) RECEIVE PRE SEPARATION SEQUENCE COMMAND & RELAY TO CC&S B) SWITCH TO DATA MODE NO. 2 C) TRANSMIT SIMULATED CAPSULE ENGINEERING DATA TO DSN D) RECEIVE SEPARATION COMMAND FROM DSN & ROUTE TO CC&S E) STORE AND/OR TRANSMIT ENGINEERING DATA TO DSN (MODE NO. 3) TRANSMIT ENGINEERING (SPACECRAFT & SIMULATED CAPSULE) PLUS CRUISE ENGINEERING TO DSN (MODE NO. 2)	A) RECEIVE ROLL & YAW TURN MAGNITUDE & DIRECTION COMMANDS & RELAY TO CC&S B) RECEIVE ANTENNA POSITIONING DATA & RELAY TO CC&S C) RECEIVE MOTOR-BURN START TIME. RELAY TO CC&S D) RECEIVE SIMULATED CAPSULE DATA, CONDITION & RELAY TO GROUND E) TRANSMIT FLIGHT SPACECRAFT ENGINEERING, SIMULATED CAPSULE ENGRG, & ENGRG INSTRUMENT DATA TO GROUND VIA MODE NO. 2	A) UPDATE ROLL & YAW TURN B) UPDATE MOTORBURN START C) UPDATE INITIATE-MANEUVE D) SWITCH TO DATA MODE NO. E) REACQUIRE EARTH AFTER R F) RELAY VERIFICATION OF M
COMMAND TELECOMMUNICATION TO DATA MODE NO. 1 RECEIVE TRAJECTORY CORRECTION PARAMETERS SWITCH GYROS TO RATE CONTROL COMMAND ANGULAR MANEUVER RECEIVE GYRO SIGNALS AND SUM FOR ANGULAR MANEUVERS SWITCH ATTITUDE REFERENCE TO AUTOPILOT SUBSYSTEM ARM PROPULSION SYSTEM PROVIDE COMMAND TO INITIATE ATTITUDE MANEUVER & ΔV MANEUVER RECEIVE ACCELEROMETER SIGNALS AND INTEGRATE FOR ΔV MANEUVER DISARM PROPULSION SYSTEM COMMAND REVERSE-ANGULAR MANEUVER COMMAND REACQUIRE CELESTIAL REFERENCES COMMAND AUTOPILOT SELECTION CONTROL BACKUP COMMANDS AS REQUIRED	A) LOAD SIMULATED CAPSULE WITH 5 OR MORE COMMANDS B) COMMAND SIMULATED CAPSULE SEQUENCING C) COMMAND TELECOMM SWITCH TO DATA MODE NO. 1 D) COMMAND GYROS ACTIVATE (SIMULATED CAPSULE) E) COMMAND GYROS ESTABLISH POLARITY (SIMULATED CAPSULE) F) COMMAND GYROS TO ATTITUDE CONTROL MODE (SIMULATED CAPSULE) G) COMMAND DATA TAPE RECORDER "OFF" H) COMMAND SIMULATED CAPSULE ACTIVATE ENTRY SCIENCE I) COMMAND SIMULATED CAPSULE ACTIVATE TELECOMM & VERIFY RADIO LINK (VHF) J) SELECT T/M MODE FOR TRANSMISSION OF SIMULATED CAPSULE DATA K) COMMAND • BIOBARRIER SEPARATION • ELECTRICAL CONNECTION SEPARATION • S/C ORIENT TO SEPARATION ATTITUDE • VERIFY SEPARATION L) COMMAND REORIENT S/C & REACQUIRE CELESTIAL REFERENCES M) BACKUP COMMANDS AS REQUIRED N) RECEIVE S/C ORIENTATION PARAMETERS	A) COMMAND SWITCH TO DATA MODE NO. 2 B) COMMAND CANOPUS ANGLE SETTING C) COMMAND ANTENNA STEP (AS REQUIRED) D) RECEIVE ORBIT INJECTION PARAMETERS E) PROVIDE SIGNAL TO PYROTECHNICS & MECHANISMS TO DEPLOY SCAN PLATFORM (CRUISE ENGINEERING INSTRUMENTS T/M OFF) AND SWITCH TO DATA MODE NO. 1 F) SWITCH ON DATA RECORDER — COMMAND TELECOMMUNICATIONS RECEIVE SIMULATED CAPSULE DATA G) RECORD ENCOUNTER ENGINEERING DATA H) BACKUP COMMANDS AS REQUIRED	A) UPDATE STORED ROLL, PIT B) UPDATE STORED VELOCITY C) INITIATE MANEUVER SEQUE D) COMMAND ORIENTATION OF E) COMMAND THRUST FOR ORI F) COMMAND RETURN TO CRU G) BACKUP COMMANDS AS RE
SWITCH TO GYRO CONTROL ROLL S/C AND VERIFY YAW S/C AND VERIFY AUTOPILOT • PROVIDE COMMAND TO TVC DURING MOTOR BURN • TURN ON ACCELEROMETERS • PROVIDE ACCELERATION DATA TO CC&S • TURN OFF ACCELEROMETERS YAW BACK TO ORIGINAL ATTITUDE & VERIFY REACQUIRE SUN (FINE SUN SENSOR) & VERIFY ROLL TO ORIGINAL ATTITUDE LOCK ON CANOPUS & VERIFY SWITCH AUTOPILOT FROM GYRO CONTROL TO CELESTIAL REFERENCE CONTROL	A) SWITCH TO GYRO CONTROL B) PERFORM ROLL TURN & VERIFY C) PERFORM YAW TURN & VERIFY D) MAINTAIN ATTITUDE DURING SEPARATION E) PERFORM YAW TURN (INVERSE TO C) & VERIFY F) PERFORM ROLL TURN (INVERSE TO B) & VERIFY G) REACQUIRE CELESTIAL REFERENCES H) RETURN TO CELESTIAL REFERENCE ATTITUDE CONTROL MODE	A) UPDATE CANOPUS ANGLE ON COMMAND B) SWITCH AUTOPILOT TO CRUISE MODE C) MAINTAIN SPACECRAFT ATTITUDE TO CELESTIAL REFERENCES DURING CRUISE	A) SWITCH TO GYRO CONTRO B) PROVIDE ROLL TO PROPER C) YAW TO PROPER ATTITUD D) AUTOPILOT-PROVIDE COM E) TURN ON ACCELEROMETER F) PROVIDE ACCELEROMETER G) TURN OFF ACCELEROMETER H) YAW BACK AFTER MOTOR I) SUN ACQUISITION & VER J) ROLL BACK TO PROGRAM K) CANOPUS ACQUISITION L) SWITCH AUTOPILOT FROM
COURSE PROPULSION SYSTEM ARM PRESSURIZATION SYSTEM (FIRE SQUIB VALVE) ARM PROPELLANT FEED SUBSYSTEM (FIRE SQUIB VALVE) FIRE PROGRAMMED ENGINES (OPERATE SOLENOID VALVE) SHUT DOWN ENGINES ISOLATE PROPELLANT FEED SUBSYSTEM ISOLATE PRESSURIZATION SUBSYSTEM			ORBIT INSERTION ENGINE A) ARM TVC B) ARM IGNITER C) FIRE MOTOR, PROVIDE T D) TERMINATE THRUST (MOT
CONDITIONED SOLAR OR BATTERY POWER TO: • TELEMETRY • REACTION CONTROL • PROPULSION • CC&S • PYROTECHNICS • TEMPERATURE CONTROL • AUTOPILOT • ATTITUDE REFERENCE • ENGRG INSTR	A) PROVIDE CONDITIONED SOLAR OR BATTERY POWER TO: • T/M • ATTITUDE REF • SIMULATED CAPSULE • CC&S • REACTION CONTROL • AUTOPILOT • TEMP. CONTROL • PYROTECHNICS • ENGINEERING INSTRUMENTATION	A) PROVIDE CONDITIONED ELECTRICAL SOLAR POWER TO: • T/M • ATTITUDE REF • ENGR INSTR • CC&S • REACTION CONTROL • AUTOPILOT • TEMPERATURE CONTROL B) CHARGE BATTERIES	PROVIDE CONDITIONED • T/M • CC&S • ATTITUDE REFERENCE • AUTOPILOT
SUPPORT S/C ASSEMBLIES SUPPORT S/C COMPONENTS MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS SUPPORT SIMULATED CAPSULE	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D) PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS E) SUPPORT SIMULATED CAPSULE	A. SUPPORT S/C ASSEMBLIES B. SUPPORT S/C COMPONENTS C. MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D. PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS E. DRIVE SCAN PLATFORM TO DEPLOYED POSITION & LOCK	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENT C) MAINTAIN ADEQUATE ALIG D) PROVIDE ACCEPTABLE STA
PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE CONTROL	PROVIDE TEMPERATURE C
REMOVE PROPULSION SYSTEM INHIBIT PROVIDE IGNITION INHIBIT PROPULSION SYSTEM	A) RECEIVE SIGNAL TO REMOVE SIMULATED CAPSULE SEPARATION INHIBIT B) RECEIVE SIGNAL TO FIRE SIMULATED CAPSULE UMBILICAL SEPARATION C) RECEIVE SIGNAL TO FIRE BIO-BARRIER REMOVAL D) RECEIVE SIGNAL TO FIRE SIMULATED CAPSULE SEPARATION E) RECEIVE SIGNAL FOR SEPARATION INHIBIT	RECEIVE SQUIB FIRING SIGNAL FOR UNLATCHING OF SCAN PLATFORM	A) RECEIVE & EXECUTE CO INHIBIT FOR PROPULS B) RECEIVE & EXECUTE SI ENGINE START C) RECEIVE & EXECUTE C D) RECEIVE & EXECUTE SI INHIBIT
RECEIVE CC&S COMMAND "CRUISE ENGINEERING INSTRUMENTS OFF" TURN OFF ENGINEERING INSTRUMENTS	RECEIVE CC&S COMMAND CRUISE ENGINEERING INSTRUMENTS "OFF" TURN OFF CRUISE ENGINEERING INSTRUMENTS	A) RECEIVE "CRUISE ENGINEERING INSTRUMENTS ON" COMMAND B) ACQUIRE ENCOUNTER ENGINEERING DATA C) TRANSMIT CONDITIONED DATA TO DAE D) RECEIVE POWER FROM E/P E) RECALIBRATE ENGRG INSTRUMENTS AS REQUIRED	ENGINEERING INSTR ON
		A) RECEIVE DAE "ON" COMMAND B) PROCESS & TRANSFER DATA TO TM	

3

Figu

PACECRAFT
INSERTION

10.0
TEST SPACECRAFT
ORBITAL OPERATIONS

11.0
ANALYZE DATA & REPORT

Y ENGINEERING DATA TION PARAMETERS IATE INSERTION MANEUVER S TO CC&S AS REQUIRED	A) RECEIVE AND ANALYZE ENGINEERING DATA B) TRACK S/C C) PROVIDE DATA FOR ORBITAL TRIM (IF REQUIRED) D) RECEIVE VERIFICATION OF ORBITAL TRIM MANEUVER E) RECEIVE & DISPLAY SIMULATED CAPSULE DATA F) TERMINATE MISSION
MAGNITUDE & DIRECTION COMMANDS & RELAY TO CC&S TIME COMMAND & RELAY TO CC&S R & RELAY TO CC&S 2 OR 5A AS REQUIRED LL & YAW TURNS & MOTOR BURN NEUVERS	A) TRANSMIT ENGINEERING DATA VIA DATA MODE 5 & 6 B) RECORD ENGINEERING DATA AS COMMANDED C) RECEIVE AND RELAY DATA FOR ORBIT TRIM IF REQUIRED D) RECEIVE, STORE, & RELAY SIMULATED CAPSULE DATA ON COMMAND (MODES 5 & 6)
H & YAW MAGNITUDE & SIGN MAGNITUDE ICE S/C TO INSERTION ATTITUDE IT INSERTION SE ATTITUDE QUIRED	A) COMMAND ORBITAL TRIM MANEUVERS (IF REQUIRED) B) SWITCH DATA MODES AS NECESSARY C) RECEIVE ANY ORBITAL OPERATION CHANGES AND SEQUENCES D) COMMAND POSITIONING OF SCAN PLATFORM E) SWITCH ON ORBITAL ENGINEERING INSTRUMENTS F) SELECT RECORDED ENGINEERING DATA READOUT TO TELECOMMUNICATIONS G) BACKUP COMMANDS AS REQUIRED H) STORE DATA FOR ORBIT TRIM (IF REQUIRED) I) TERMINATE MISSION ON COMMAND
L ATTITUDE & VERIFY & VERIFY MAND TO TVC DURING MOTOR BURN DATA TO CC&S BURN FY ED ATTITUDE VERIFY GYRO CONTROL TO CELESTIAL REFERENCE CONTROL	A) MAINTAIN S/C ATTITUDE TO CELESTIAL REFERENCES B) SWITCH AUTOPILOT TO GYRO CONTROL DURING OCCULTATION C) REACQUIRE CELESTIAL REFERENCES FOLLOWING OCCULTATION D) REORIENT S/C FOR ORBITAL-TRIM MANEUVER USING GYRO CONTROL E) PROVIDE TVC DURING ENGINE FIRING F) REACQUIRE CELESTIAL REFERENCES FOLLOWING MANEUVERS G) UPDATE CANOPUS CONE ANGLES ON COMMAND
RUST & TVC R BURNS TO DEPLETION)	MID-COURSE A) ARM PRESSURIZATION & PROPELLANT FEED SUBSYSTEMS B) PROVIDE THRUST (FIRE PROGRAMMED ENGINES) FOR ORBIT TRIM IF REQUIRED C) TERMINATE THRUST ON COMMAND D) ISOLATE PROPELLANT FEED SUBSYSTEM E) ISOLATE PRESSURIZATION SUBSYSTEM
SOLAR OR BATTERY POWER TO: • TEMPERATURE CONTROL • PULSE TO PYROTECHNICS SUBSYSTEM • PULSE TO PROPULSION • ENGR INSTR	A) PROVIDE CONDITIONED SOLAR OR BATTERY POWER TO: • T/M • AUTOPILOT • CC&S • TEMPERATURE CONTROL • ATTITUDE REFERENCE SUBSYSTEM • STRUCTURES & MECHANISMS • ENGINEERING INSTR B) CHARGE BATTERIES
S NMENT BETWEEN COMPONENTS TIC AND DYNAMIC LOAD ENVIRONMENTS	A) SUPPORT S/C ASSEMBLIES B) SUPPORT S/C COMPONENTS C) MAINTAIN ADEQUATE ALIGNMENT BETWEEN COMPONENTS D) PROVIDE ACCEPTABLE STATIC AND DYNAMIC LOAD ENVIRONMENTS E) POSITION SCAN PLATFORM
NTROL	PROVIDE TEMPERATURE CONTROL
MMAND SIGNAL TO REMOVE ION ENGINE NAL FOR PROPULSION MMAND SIGNAL FOR PROPULSION ENGINE STOP NAL FOR PROPULSION	
DURING ORBIT INSERTION	A) TURN ON "ORBITAL ENGINEERING INSTRUMENTS" B) ACQUIRE "ORBITAL ENGINEERING INSTRUMENTS" DATA C) TRANSFER CONDITIONED DATA TO DAE D) RECALIBRATE INSTRUMENTS AS REQUIRED
	A) RECEIVE DAE "ON" COMMAND B) PROCESS AND TRANSFER DATA TO TELECOMMUNICATIONS

4

re III-4: 1969 Saturn IB/Centaur Orbiter; Subsystem Correlation Chart

III-3.10 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, SPACECRAFT LAYOUT AND CONFIGURATION--SATURN IB/CENTAUR

III-3.10.1 Description

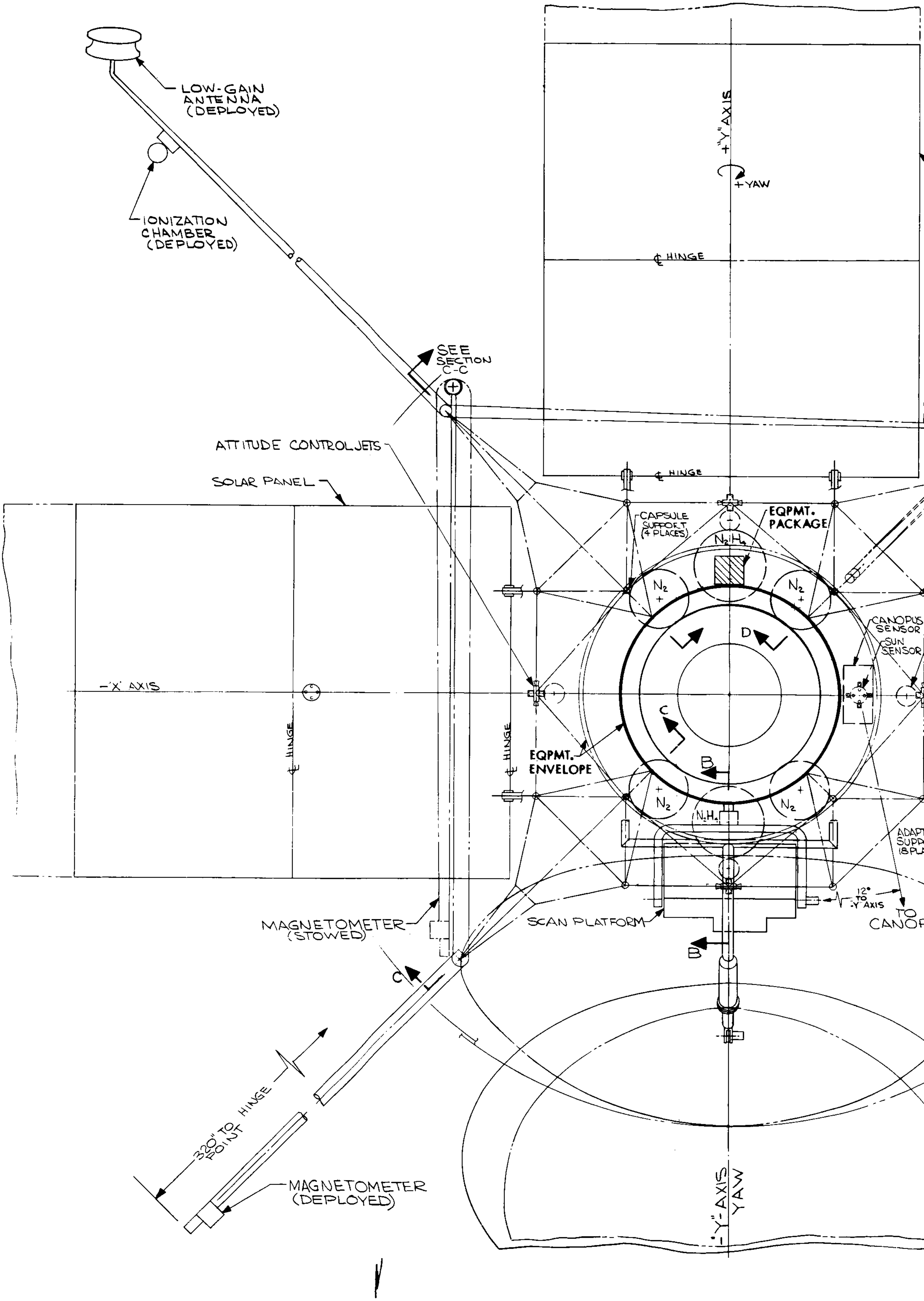
This section defines the configuration and layout of the 1969 Saturn-IB/Centaur-launched Mars orbital flight test spacecraft. The configuration is identical to the 1971 Voyager Flight Spacecraft described in Section 3.1 of D2-82709-1; the drawings, hardware definition, and analysis contained therein are applicable to this vehicle except as noted.

The Saturn-IB/Centaur-launched test spacecraft has the inherent capability of test-flying all 1971 hardware components in an environment representative of the 1971 mission and at the same time demonstrating the 1971 operations. Should it be desired to include scientific or developmental experiments along with the 1969 Test Flight, this configuration also has the potential to accomodate the following, alone or in combination:

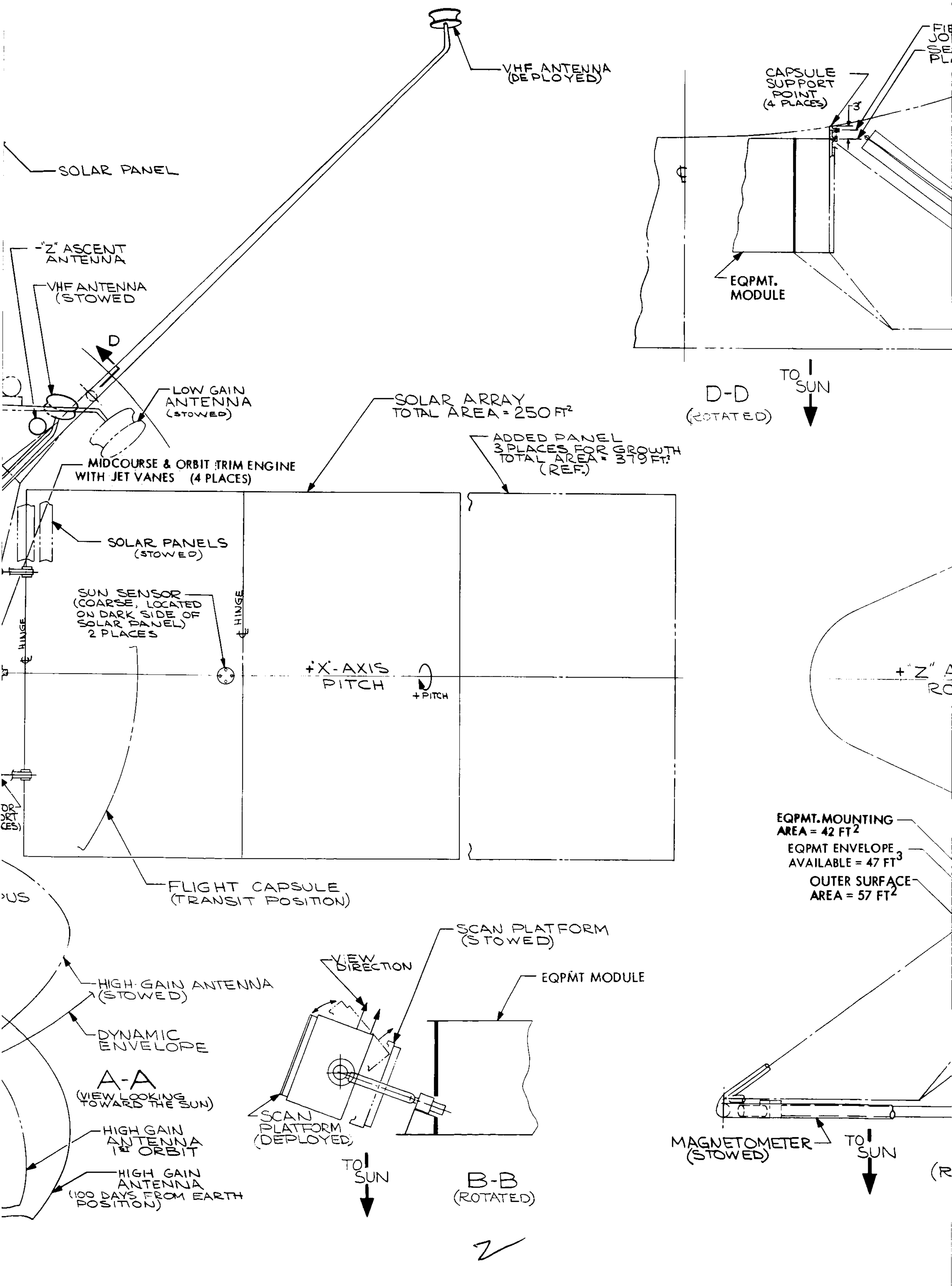
1) Cruise and orbital experiments;
2) Mars atmosphere experiments;
3) Lander development tests.

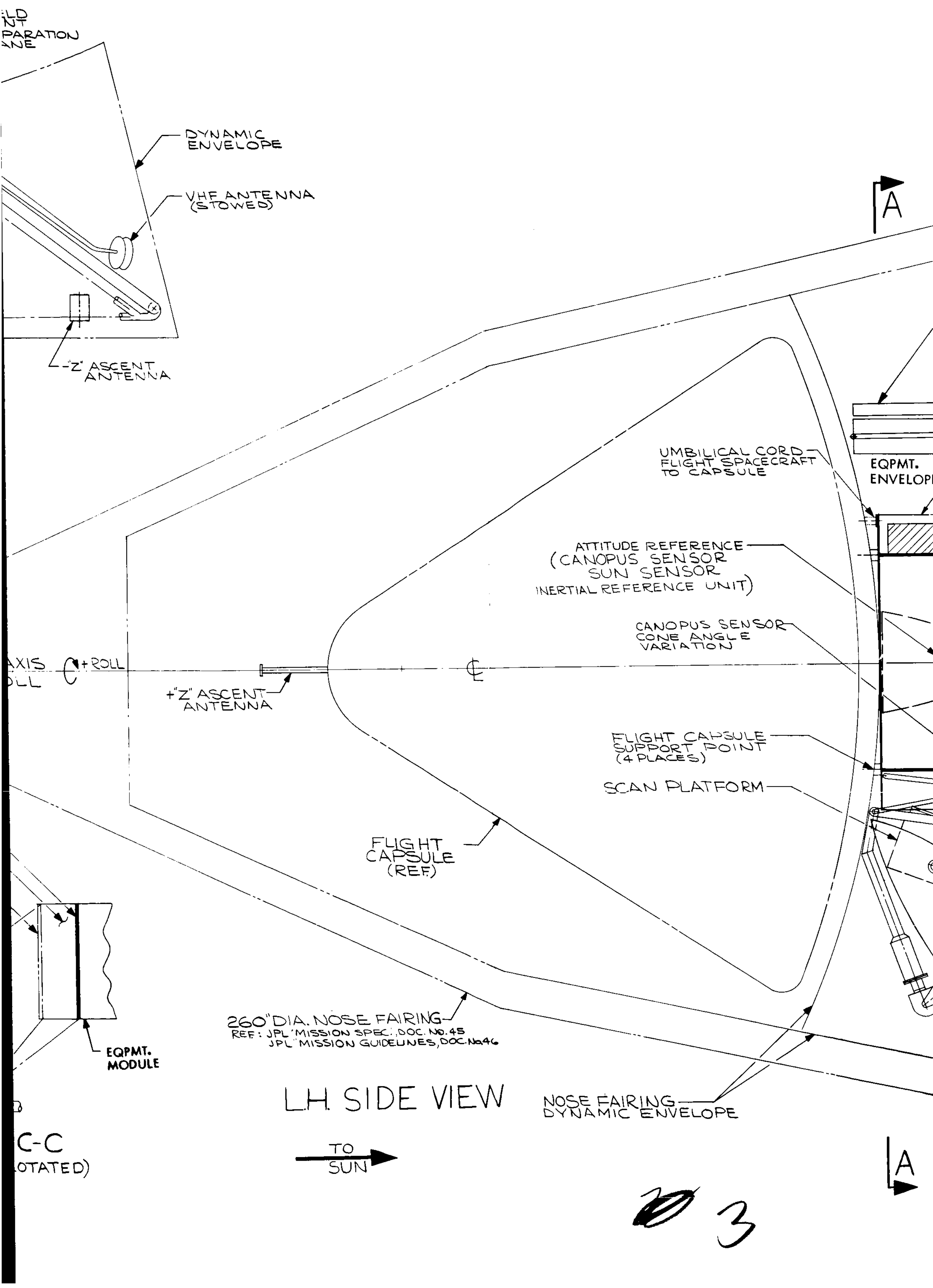
III-3.10.1.1 Applicable Documentation and Drawings

Documentation and drawings for the 1969 Test Flight Spacecraft are the same as the 1971 Voyager spacecraft. The 1971 spacecraft general-arrangement drawing, 25-50034A, Sheet 1, is repeated in this section for convenience as Figure III-5.



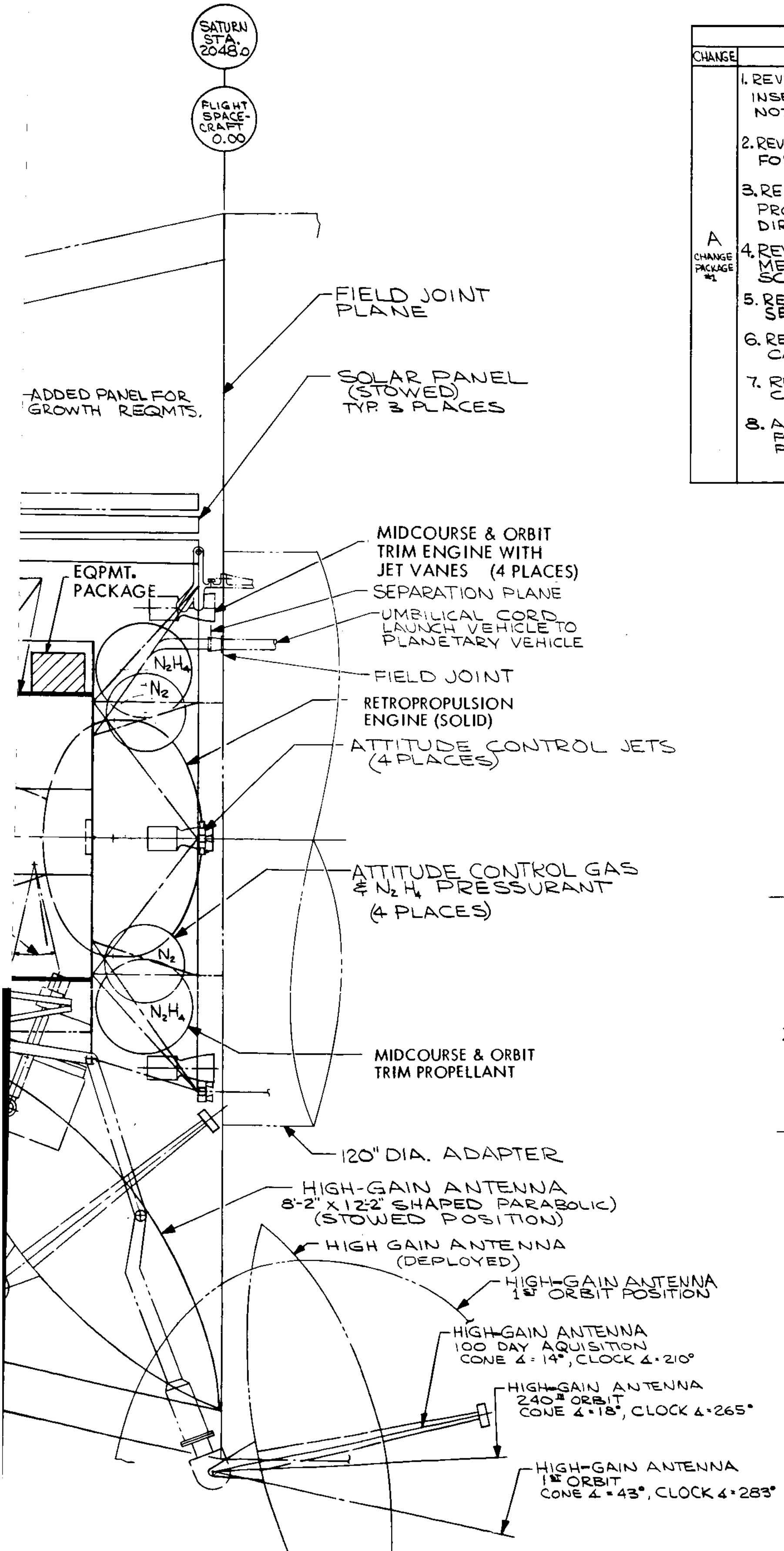
LOW-GAIN ANTENNA (DEPLOYED)
IONIZATION CHAMBER (DEPLOYED)
SEE SECTION C-C
ATTITUDE CONTROL JETS
SOLAR PANEL
-"X" AXIS
℄ HINGE
+"Y" AXIS
+YAW
CAPSULE SUPPORT (4 PLACES)
EQPMT. PACKAGE
N_2H_4
N_2
CANOPUS SENSOR
SUN SENSOR
D
C
B
EQPMT. ENVELOPE
12° TO -Y AXIS
TO CANOP
SCAN PLATFORM
MAGNETOMETER (STOWED)
320" TO HINGE POINT
MAGNETOMETER (DEPLOYED)
-"Y" AXIS YAW


VHF ANTENNA (DEPLOYED)
SOLAR PANEL
"Z" ASCENT ANTENNA
VHF ANTENNA (STOWED)
D
LOW GAIN ANTENNA (STOWED)
SOLAR ARRAY TOTAL AREA = 250 FT²
ADDED PANEL 3 PLACES FOR GROWTH TOTAL AREA = 379 FT² (REF.)
MIDCOURSE & ORBIT TRIM ENGINE WITH JET VANES (4 PLACES)
SOLAR PANELS (STOWED)
SUN SENSOR (COARSE, LOCATED ON DARK SIDE OF SOLAR PANEL) 2 PLACES
HINGE
+X-AXIS PITCH
+PITCH
FLIGHT CAPSULE (TRANSIT POSITION)
CAPSULE SUPPORT POINT (4 PLACES)
3"
EQPMT. MODULE
TO SUN
D-D (ROTATED)
EQPMT. MOUNTING AREA = 42 FT²
EQPMT ENVELOPE AVAILABLE = 47 FT³
OUTER SURFACE AREA = 57 FT²
SCAN PLATFORM (STOWED)
VIEW DIRECTION
EQPMT MODULE
HIGH-GAIN ANTENNA (STOWED)
DYNAMIC ENVELOPE
A-A (VIEW LOOKING TOWARD THE SUN)
HIGH GAIN ANTENNA 1st ORBIT
HIGH GAIN ANTENNA (100 DAYS FROM EARTH POSITION)
SCAN PLATFORM (DEPLOYED)
TO SUN
B-B (ROTATED)
MAGNETOMETER (STOWED)
TO SUN

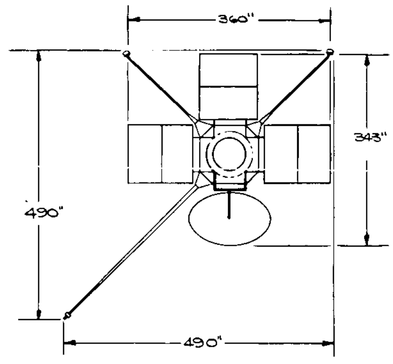


LH. SIDE VIEW

3



REVISIONS				
CHANGE	DESCRIPTION	DWN.	CHECK	APPROVAL
A CHANGE PACKAGE #1	1. REVISED SOLID PROPELLANT ORBIT INSERTION ENGINE CASE AND NOZZLE GEOMETRY. 2. REVISED ACTUATION MECHANISM FOR THE HIGH-GAIN ANTENNA 3. REVISED HINGING & STOWAGE PROVISIONS OF THE OMNI-DIRECTIONAL ANTENNA BOOM. 4. REVISED DEPLOYMENT & ACTUATION MECHANISM FOR THE PLANET SCAN PLATFORM. 5. REVISED CAPSULE BARRIER SEPARATION DEVICE. 6. REVISED ATTITUDE REFERENCE CALLOUT. 7. REVISED SPACECRAFT AXES CALLOUT. 8. ADDED TWO OMNI ANTENNAS FOR LAUNCH AND EARLY PHASES OF CRUISE.	[illegible] 7/15	[illegible] 7/16	[illegible] 7/18



SPACECRAFT DEPLOYED
(VIEW LOOKING TOWARD SUN)
SCALE 1/100 SIZE

"THIS DRAWING CONTAINS BOEING COMPANY INFORMATION AND SHOULD BE DISSEMINATED WITHIN YOUR COMPANY ON A NEED-TO-KNOW BASIS AND MAY NOT BE TRANSMITTED OUTSIDE YOUR COMPANY WITHOUT APPROVAL OF THE BOEING COMPANY"

Figure III-5: Mars Exploration Flight Spacecraft

4

III-3.10.1.2 Functional Description

The only significant differences between the 1971 Flight Spacecraft and the 1969 Test Spacecraft are:

1) Additional test data instrumentation is carried on the 1969 Test Spacecraft which will utilize the excess capacity of the telecommunications system made available by the elimination (or reduction) in capsule and/or science payload data loads.

2) Consideration is given to providing maximum propellant loading on the 1969 Test Flight (as described in Section III-4.3 of this document) to permit greater flexibility of Mars orbit selection. The same motor case is used in both the 1969 and 1971 configurations.

III-3.10.1.3 Interface Definition

A simulated Flight Capsule reduced in weight to compensate for the additional propellant required is included in the 1969 test configuration to verify the interfaces between the capsule and the nose fairing, Spacecraft Bus, spacecraft propulsion, and spacecraft adapters during all phases of the test flight. In all probability, the simulated Flight Capsule will be supplied by the capsule contractor.

III-3.10.1.4 Physical Characteristics and Constraints

By increasing the propellant weight to the maximum capacity of the motor case, an 18-hour orbit can be attained. The higher ΔV required in 1969 does not result in a 1969 launch weight problem because the additional propellant weight can be compensated for by reducing the weight allocated to the capsule. The reduction in capsule weight will not significantly affect the correlation of the 1969 test results to the 1971 Voyager program.

III-3.10.2 1969 Test Flight Benefits

Successful completion of one or more of the Saturn-IB/Centaur-launched 1969 Test Flights will have a major effect on the probability of 1971 mission success in that:

1) An integrated, full-duration field test of the complete 1971 Flight Spacecraft and its OSE will have been conducted in the 1971 mission environment.

2) Additional spacecraft test data instrumentation will enable closer evaluation of subsystem performance to the component level far lower than available during the 1971 flight. Corrective action can be taken before the 1971 launch to enhance the component reliability.

III-3.11 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, PLANETARY QUARANTINE--SATURN IB/CENTAUR

III-3.11.1 Description

Decontamination requirements for the 1969 Test Spacecraft are the same as those established for the 1971 Voyager spacecraft as stated in Section 3.11 of Volume A, D2-82709-1.

III-3.12 1969 TEST SPACECRAFT FLIGHT EQUIPMENT, CLEANLINESS--SATURN IB/CENTAUR

III-3.12.1 Description

Cleanliness requirements for the 1969 Test Flight are identical to those described in Section 3.12 of Volume A, D2-82709-1, "1971 Preferred Design."

III-3.13 1969 TEST SPACECRAFT MAGNETICS--SATURN IB/CENTAUR

III-3.13.1 Description

Magnetic design constraints for the 1969 Test Flight are identical to those defined in Section 3.13 of Volume A, D2-82709-1, "1971 Preferred Design."

III-3.14 1969 TEST SPACECRAFT RADIATION EFFECTS--SATURN IB/CENTAUR

III-3.14.1 Description

A general treatment of the effects of radiation on spacecraft components and subsystems is given in Section 3.14 of Volume A, D2-82709-1.

III-3.14.2 1969 Test Flight Benefits

The ability of the spacecraft to operate successfully in the Mars orbital environment will provide additional assurance that radiation effects will not degrade the 1971 mission.

III-4.0 1969 TEST SPACECRAFT FOR HARDWARE SUBSYSTEM

III-4.1 1969 TEST SPACECRAFT TELECOMMUNICATIONS--SATURN IB/CENTAUR

III-4.1.1 Description

The telecommunications subsystem will be identical to that described in Section 4.1 of Volume A, D2-82709-1, for the 1971 preferred design with the following exceptions:

1) An extra multiplexer will be incorporated to accomodate the expanded number of test points on the spacecraft subsystem from which data will be telemetered during the test. The multiplexer

will interface with the telemetry and data storage subsystem via the cruise science input channel.

2) Provision will be made to simulate data inputs for the real-time and delayed planetary science modes.

III-4.1.2 1969 Test Flight Benefits

The telecommunication subsystem proposed for the 1971 Voyager mission includes several advancements in equipment configuration and in data processing and handling. The major advantages of a 1969 Test Flight before the 1971 mission are:

1) All elements of the subsystem will receive extended life testing under the full set of multistress environmental conditions they will experience in 1971. This will be a much more comprehensive environmental test of the integrated system than can be achieved on the ground.

2) The operational software and MDE to be used at the Deep Space Network stations and the Space Flight Operations Facility can be completely exercised over a full mission profile. This will be of significant value for personnel training and for discovery of any discrepancies or deficiencies that remain after ground simulation and checkout.

3) Verification of the predicted performance of high-data-rate modes at encounter ranges can be achieved by generating a bit stream simulating the high-rate output of the data automation system. In particular this will demonstrate the performance of the high-gain antenna-pointing system coupled with vehicle dynamics.

4) A simulated Flight Capsule, equipped with VHF-relay-link communication equipment will be carried on a Saturn-IB/Centaur-launched test flight. Separation of this capsule on a nonimpacting trajectory will permit checkout of VHF-link performance exclusive of Martian atmosphere or surface proximity effects.

III-4.2. 1969 TEST SPACECRAFT ELECTRICAL POWER--SATURN IB/CENTAUR

III-4.2.1 Description

The electrical power subsystem for the 1969 Test Spacecraft will be identical to that for the 1971 mission as described in Section 4.2 of Volume A, D2-82709-1 except for the addition of sensors or monitoring points to provide for more extensive engineering data.

III-4.2.2 1969 Test Flight Benefits

Added confidence in the successful operation of the electrical power subsystem during the 1971 mission will be gained through a 1969 Test Flight in the following ways:

1) Better judgment on solar proton radiation damage effects will take into account the statistical probability that the 1971 environment is similar to the 1969 environment.
2) Subsystem temperatures will be verified.
3) Battery cycling, panel stresses, temperatures, and other effects will be verified through actual exposure to space conditions during cyclic occultations.

III-4.3 1969 TEST SPACECRAFT PROPULSION--SATURN IB/CENTAUR

III-4.3.1 Description

This subsystem is identical to that described in Section 4.3 of D2-82709-1 with one exception. The solid-propellant orbit insertion motor case will be used for maximum propellant load, providing a ΔV of 6550 fps, in lieu of 5700 fps and an 18-hour orbit period about Mars. A separate qualification testing program will be required for the orbit insertion motor with the increased propellant loading.

III-4.3.2 1969 Test Flight Benefits

The 1969 Test Flight will demonstrate the suitability of the midcourse correction and orbit propulsion system design concept, the selection of components, and the operating modes under the environmental conditions of a 1971 mission.

In particular, the following data would be obtained:

1) Determine monopropellant engine operating performance in space environment using the spontaneous catalyst ignition.
2) Determine operating reliability of solenoid isolation valves in the pressurization and propellant-feed portions of the system for repeated engine start and shutdown sequences.
3) Evaluate orbit-insertion solid-propellant motor start, burn, and thrust-vectoring characteristics following long-term storage in deep space environment. It is noted that the 1969 Test Flight of the motor with maximum propellant load imposes a more stringent test of the motor case than the 1971 mission.

III-4.4 1969 TEST SPACECRAFT ENGINEERING MECHANICS--SATURN IB/CENTAUR

Summary--This section is identical to the engineering mechanics described in Section 4.4 of Volume D2-82709-1 except that, with the use of a simulated capsule, revised science payload and revised orbit insertion engine propellant loading the 1969 Test Spacecraft mass properties will be revised as follows.

The mass properties of the Voyager 1969 Test Spacecraft (Saturn IB/Centaur launch vehicle) are equal to the values presented for the 1971 Voyager spacecraft, subject to the following exceptions. The 1971 spacecraft has a solid-propellant load of 2306 pounds, which is 532 pounds less than the motor design capacity of 2838 pounds. The 1969 Test Spacecraft will use this maximum solid-propellant capacity, in combination with an off-loaded simulated Flight Capsule so that the separated planetary test vehicle weighs 7800 pounds. The mass properties will change to the anticipated nominal values given below:

Condition	Weight (lb)	CG STA. (in.)	I_z (slug-ft^2)	I_x (slug-ft^2)	I_y (slug-ft^2)
Initial Deployed Condition	7800	42.3	4890	5060	5370
Start Capsule Separation	7703	42.7	4860	5030	5340
Start Orbit Insertion	5933	24.6	3002	1640	1950

III-4.5 1969 TEST SPACECRAFT SCIENCE--SATURN IB/CENTAUR

No provisions have been made for a science payload on the 1969 Test Flight; however, structural support, electrical power, and telemetry is available should science experiments be required.

III-4.6 1969 TEST SPACECRAFT ATTITUDE REFERENCES AND AUTOPILOT SUBSYSTEM--SATURN IB/CENTAUR

III-4.6.1 Description

The attitude reference and autopilot subsystem for the 1969 Test Flight with the Saturn IB/Centaur launch vehicle is identical to the equipment described in Volume A, Section 4.5, for the 1971 mission.

III-4.6.2 1969 Test Flight Benefits

Partial confirmation of the operation of the subsystem can be obtained from a simulation test program on Earth. However, complete confirmation can be obtained only through test under actual operating conditions. For example, an important function of the gyros is to maintain vehicle attitude during maneuvers and occultation in order to point the high-gain antenna. It is anticipated that the long-term drift bias stability of the gyro is less than 0.05 degree per hour, which is adequate; however, there is no data on long-term operation at zero acceleration to confirm this. Consequently a test flight similar to the 1971 Voyager would provide data required to confirm the bias stability of the gyro under zero-g conditions as well as prove the validity of subsystem operation within the combined effects of induced and natural environments.

III-4.7 1969 TEST SPACECRAFT REACTION CONTROL SUBSYSTEM--SATURN IB/CENTAUR

III-4.7.1 Description

The reaction control subsystem for the Saturn IB/Centaur 1969 Test Flight is the same subsystem as described in Section 4.7 of D2-82709-1.

III-4.7.2 Test Flight Benefits

The test flight will prove the capability of the specific reaction control hardware to operate for extended periods of time in space. Space environment and time are the only criteria for the test.

Temperatures and pressures of the propellant tanks will be determined during prolonged space exposure.

III-4.8 1969 TEST SPACECRAFT CENTRAL COMPUTER AND SEQUENCER--SATURN IB/CENTAUR

III-4.8.1 Description

The central computer and sequencer subsystem for the 1969 Test Flight is exactly the same as that described in D2-82709-1, Section 4.8, for the 1971 preferred design. The 1969 Test Flight will carry a simulated Flight Capsule that will provide dynamic simulated electrical interfaces representative of those of the 1971 Flight Capsule. The 1969 capsule will simulate the 1971 capsule in all mission phases before Mars encounter and will be ejected in a manner similar to that of the 1971 capsule. The CC&S will run a complete 1971 program of commands and switching to the simulated capsule.

III-4.8.2 1969 Test Flight Benefits

The specific benefits of a 1969 Saturn IB/Centaur Test Flight to the CC&S subsystem are as follows:

1) Reliability--The bit error rate of the DSIF to spacecraft telecommunication subsystem command link will be exactly defined, and will verify error checking or correcting codes within the command message. In addition, the performance of the CC&S in accepting, verifying, and executing these real-time and stored commands will be demonstrated over the long mission duration and in the actual environment of the 1971 mission.

   Hardware failures and degradations occurring in the CC&S sub-system during the 1969 flight will provide information on the adequacy of the ground test program, the reliability of the selected parts and tolerances, and the adequacy of the reliability design including redundancy, work-around techniques, critical-path analysis, stored program techniques, and data transfer techniques.

2) Versatility--The 1969 Test Flight will demonstrate the ability of the CC&S to accept changes to the stored preplanned sequence of events at any time as dictated by mission circumstances. The tests will include demonstration of switchover between the two redundant CC&S control assembly logic processing units. Override capability of real-time ground-initiated commands will be demonstrated.

ERRATA

The Boeing Company Document No. D2-82709-4, Voyager Spacecraft System Final Technical Report

VOLUME D

"Design for 1969 Test Spacecraft"

Page No.	Paragraph, Table, or Figure No.	
		Part I
I-15	Figure I-3	INTEGRATED TEST PROGRAM SCHEDULE 1969 SATURN IB/CENTAUR TEST FLIGHT Replace figure with attached corresponding revised schedule.
I-17	Figure I-4	INTEGRATED TEST PROGRAM SCHEDULE 1969 ATLAS/CENTAUR TEST FLIGHT Replace figure with attached corresponding revised schedule.
		Part II
II-6	Para. II-2.1.1.2	Heliocentric Flight Revise third sentence which reads: "The flight will use a Type I trajectory, a C_3 of 15.5 $(km/sec)^2$, and launch dates from May 7 through June 22, 1969." to read as follows: "The flight will use a Type I trajectory, a maximum C_3 of 15.5 $(km/sec)^2$, and launch dates from May 7 through June 22, 1969 with a separated spacecraft weight of 1500 pounds."

ERRATA (Continued)

VOLUME D

Page No.	Paragraph, Table, or Figure No.	
II-8	Para. II-2.1.2	Subsystems Revise the paragraph, "With the exclusion of the science subsystem the characteristics and restraints for the 1969 Test Spacecraft subsystems are the same as those established in Section 2.1.3 of Volume A, D2-82709-1, for the 1971 preferred design." to read as follows: "With the exclusion of the science subsystem, the Spacecraft/Capsule RF link, and the orbit insertion engine, the characteristics and restraints are the same as those established in Section 2.1.3 of Volume A, D2-82709-1, for the 1971 preferred design."
II-9	Para. II-2.3.1	Description (Maneuver Accuracy) Revise the paragraph which reads: "The maneuver accuracy and propulsion requirements for the 1961 Test Flight are the same as described in Volume A for the 1971 preferred design." to read: "The maneuver accuracy and propulsion requirements for the 1969 Test Flight are the same as those described in Volume A for the 1971 preferred design except for the deletion of the orbit trim ΔV requirements."
II-11	Para. II-2.5.1	Description (Parts, Materials and Processes) Delete this entire subparagraph including the title.
II-11	Para. II-2.5.2	1969 Test Flight Benefits Change the paragraph number to II-2.5.1.

ERRATA (Continued)

VOLUME D

<table>
<tr><th>Page No.</th><th>Paragraph, Table, or Figure No.</th><th></th></tr>
<tr><td>II-15</td><td>Table II-1</td><td>ATLAS/CENTAUR TRAJECTORY CHARACTERISTICS

1) Add the following footnote:

"ΔV = impulse total velocity from 100-nautical-mile circular parking orbit."

2) The last column on the right entitled:

"Communication Distance"

should read:

"Maximum Communication Distance at Encounter."</td></tr>
<tr><td>II-32 and II-33</td><td>Table II-3</td><td>FLIGHT DATA MEASUREMENT LIST

Delete identification numbers 146, 147, 148, 149 (Engine Pitch and Yaw Actuators). Add a duplicate set of identification numbers 150 through 157.</td></tr>
<tr><td>II-51</td><td>Figure II-9</td><td>INBOARD PROFILE--1969 TEST SPACECRAFT--ATLAS/CENTAUR

1) Revise callout at right of figure to read as follows:

<table><tr><th>BOX</th><th>ITEM</th><th>SUBSYSTEMS</th></tr><tr><td>1</td><td></td><td>DELETED</td></tr><tr><td>2</td><td></td><td></td></tr></table>
2) In the plan view, lower left of figure, delete the number "1" from the box numbered 1, 10, 11. The adjoining large box presently numbered "1" should be numbered "13."

3) Delete the box and number "1" from the developed view in the upper left of the figure.</td></tr>
</table>

ERRATA (Continued)

VOLUME D

Page No.	Paragraph, Table, or Figure No.	
		Part III
III-4	Para. III-2.1.2	Test Flight Profile The subindenture number 2 which reads: "A C_3 of 17.0 $(km/sec)^2$" should read: "A maximum C_3 of 17.0 $(km/sec)^2$"
III-5	Para. III-2.4.1	Description (Aiming Point) The first line which reads: "The aiming-point selection for the 1969 Mars...." should be revised to read: "The aiming-point selection process for the 1969 Mars.... ."
III-6	Para. III-2.5.1	Description (Parts, Materials and Processes) Delete this entire subparagraph including the title.
III-6	Para. III-2.5.2	1969 Test Flight Benefits Renumber the paragraph to III-2.5.1.
III-6	Para. III-3.1.1	Description (Standard Trajectories) The first line which reads: "The 1971 Voyager mission will be performed with an 18-hour.... ." should be revised to read: "The 1971 Voyager mission can, for example, be performed with an 18-hour.... ."

ERRATA (Continued)

VOLUME D

Page No.	Paragraph, Table, or Figure No.	
III-9	Table III-1	SATURN IB/CENTAUR--Trajectory Characteristics

1) The middle column entitled

"REQUIRED C_3 (km^2/sec^2)"

should be titled to read:

"MAXIMUM REQUIRED C_3 (km^2/sec^2)"

2) The title in the next to the last column on the right which reads:

"COMMUNICATION DISTANCE (km)"

should be revised to read:

"MAXIMUM COMMUNICATION DISTANCE, AT ENCOUNTER (km)"

3) In the title of the last column on the right, a Δ should be added before the "V" to make the title read:

"INSERTION ΔV REQUIREMENTS (ft/sec)"

4) Move the five Saturn IB/Centaur callouts in the second column up to coincide with the missions. The first two columns should read as follows:

MISSION	BOOSTER
Mars Orbit with 18-hr Period	Saturn IB/Centaur
Mars Orbit, 18-hr Period	Saturn IB/Centaur
Mars Orbit, 30-hr Period	Saturn IB/Centaur
Mars Orbit, 60-hr Period	Saturn IB/Centaur
Mars Orbit, 60-hr Period	Saturn IB/Centaur
Mars Flyby	Saturn IB/Centaur

ERRATA (Continued)

VOLUME D

Page No.	Paragraph, Table, or Figure No.	
III-15	Para. III-3.3.1	Description (Design Parameters) The second sentence which reads: "These sheets (SCDPS) are also applicable to the Saturn IB/Centaur-launched 1969 Test Spacecraft except for deletion of science instrumentation and science data automation equipment and the addition of one multiplexer-encoder unit for additional engineering performance data." should be revised to read as follows: "These sheets (SCDPS) are also applicable to the Saturn IB/Centaur-launched 1969 Test Spacecraft except for the deletion of science instrumentation and the addition of one multiplexer-encoder unit for additional engineering performance data."
III-27 and III-28	Table III-2	FLIGHT DATA MEASUREMENT LIST Replace identification numbers 150, 151, 152 and 153 with the attached items. Add a duplicate set of identification numbers 154 through 161.
III-50	Para. III-4.3.1	Description (Propulsion) The second sentence which reads: "The solid-propellant orbit insertion motor case will be used for maximum propellant loading, providing a ΔV of 6550 fps, ..." should be revised to read: "The solid-propellant orbit insertion motor case will be used for maximum propellant loading, providing a ΔV of 6640 fps, ..."

Table III-2: FLIGHT DATA MEASUREMENT LIST

in part:

Ident. No.	Measurement	Parameter Unit	Parameter Range	Required Accuracy	Signal An./Dig.	Rates Samples/Second
150	Injection valve pitch +			2%	0-5V	1/60
151	Injection valve pitch -			2%	0-5V	1/60
152	Injection valve yaw +			2%	0-5V	1/60
153	Injection valve yaw -			2%	0-5V	1/60

	1966	1967
	DEVELOPMENT FREEZE	
I. DEVELOPMENT TESTS		
SUBSYSTEMS DESIGN CRITERIA DEVELOPMENT & VERIFICATION	BREAD-BOARD TESTS	ENGINEERING MODEL TESTS
STRUCTURAL TEST MODEL GT-1		FAB
THERMAL TEST MODEL GT-2	1969 FAB	TEST 1971
DYNAMIC TEST MODEL GT-3	FAB	TEST
ENGINEERING MODEL GT-4		FAB
II. TYPE APPROVAL TESTS (TAT)		
SUBSYSTEMS		
PROOF TEST MODEL 1969		
COMPATIBILITY TEST MODEL 1969		
PROOF TEST MODEL 1971 NO. 1		
PROOF TEST MODEL 1971 NO. 2		
*JPL TEST SPACECRAFT		
III. FLIGHT ACCEPTANCE TESTS (FAT)		
FLIGHT SPACECRAFT 1969 SPARE (SAME AS COMPAT TEST MODEL 1969)		
TEST FLIGHT S/C NO. 1 (1969)		
TEST FLIGHT S/C NO. 2 (1969)		
FLIGHT SPACECRAFT (SPARE) (1971)		
FLIGHT SPACECRAFT NO. 1 (1971)		
FLIGHT SPACECRAFT NO. 2 (1971)		

* Per Specimen Statement of Work

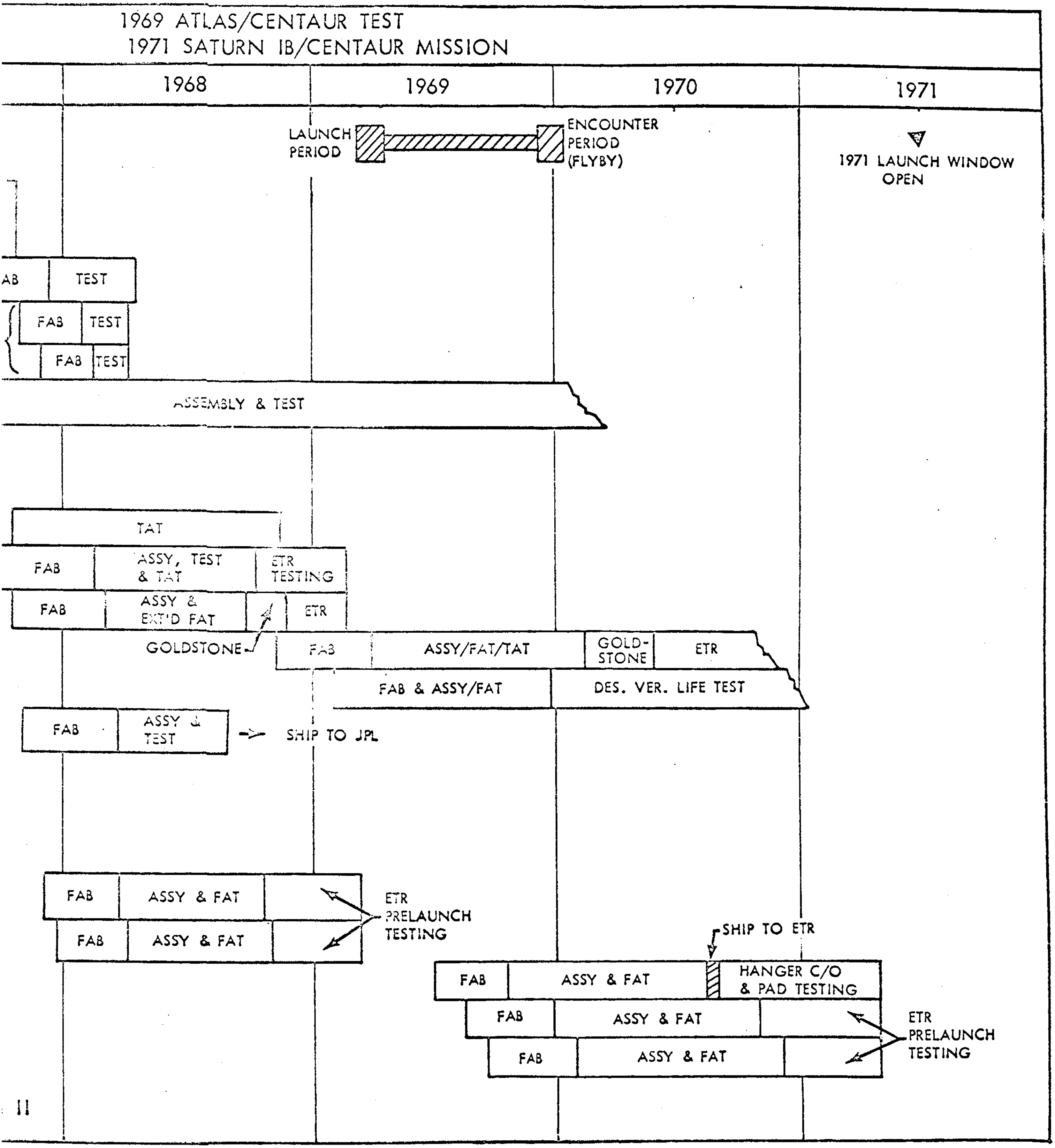


Figure 1-4: Integrated Test Program Schedule

1969 SATUR
1971 SATUR

	1966	1967
	DEVELOPMENT FREEZE ▽	
I. DEVELOPMENT TESTS		
SUBSYSTEMS DESIGN CRITERIA DEVELOPMENT & VERIFICATION	BREADBOARD TESTS	ENGINEERING MODEL TESTS
STRUCTURAL TEST MODEL GT-1		FAB
THERMAL TEST MODEL GT-2	1969	FAB TEST 1971 F
DYNAMIC TEST MODEL GT-3		FAB TEST
ENGINEERING MODEL GT-4		FAB
II. TYPE APPROVAL TESTS (TAT)		
SUBSYSTEMS		
PROOF TEST MODEL 1969		FA
COMPATIBILITY TEST MODEL 1969		F
PROOF TEST MODEL 1971 NO. 1		
PROOF TEST MODEL 1971 NO. 2		
* JPL TEST SPACECRAFT		
III. FLIGHT ACCEPTANCE TESTS (FAT)		
FLIGHT SPACECRAFT 1969 SPARE (SAME AS PTM 1969)		
TEST FLIGHT S/C NO. 1 (1969)		
TEST FLIGHT S/C NO. 2 (1969)		
FLIGHT SPACECRAFT (SPARE) (1971)		
FLIGHT SPACECRAFT NO. 1 (1971)		
FLIGHT SPACECRAFT NO. 2 (1971)		

*Per Specimen Statement of Work Phase II

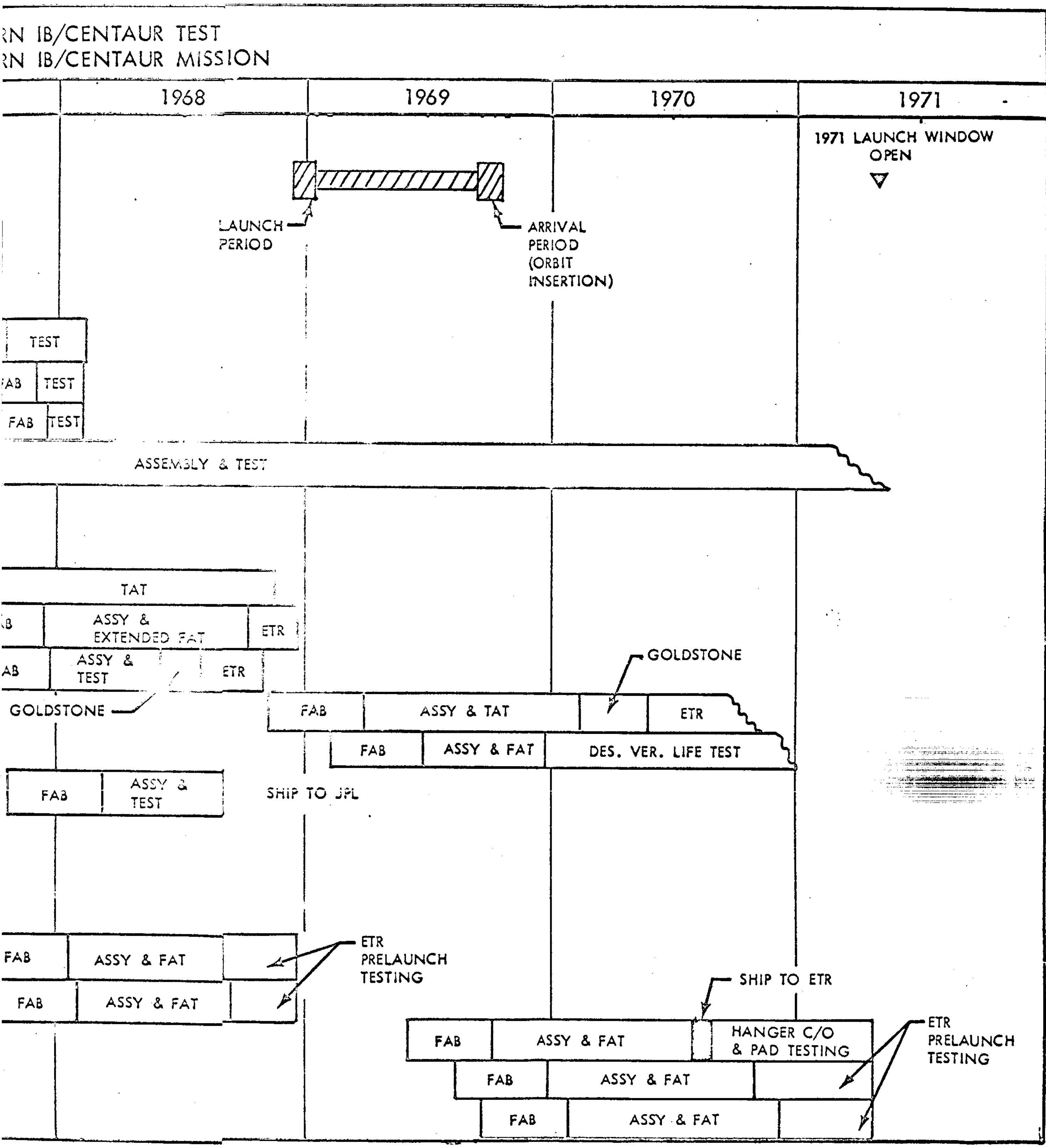


Figure 1-3: Integrated Test Program Schedule