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PHASE IA TASK B
FINAL TECHNICAL REPORT

VOYAGER SPACECRAFT

Volume 3
SPACECRAFT IMPLEMENTATION

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I. INTRODUCTION

This volume presents the result of the Phase IA, Task B, study for implementation of the spacecraft. It describes the effort required to design, develop, test, assemble, check out, launch, and support spacecraft flight operations for the 1971 mission. The plan applies to the spacecraft system only, although the OSE schedule interfaces are also identified. OSE implementation is discussed in Volume 4.

Section II presents the major milestone schedules for both Phase IB and II and discusses schedule philosophy. Section III discusses over-all test planning and the generation of the Voyager integrated test plan. System and subsystem development, assembly, checkout, systems test, launch, and mission support to implement the 1971 mission are discussed in Section IV. The section concentrates on identifying and scheduling the tasks that must be done to meet the launch date. Manufacturing tasks are briefly treated in terms of schedule requirements and equipment deliveries, but the general manufacturing plan is presented in Part 8 of Volume III of the Phase IB proposal.

At this time only enough effort has been devoted to the detailed planning of the sequence for the 1973 and subsequent missions to assure that the tasks are well within the available schedule time. Design efforts for the 1973 mission could begin early in 1969, and the fabrication and acceptance testing can be readily spaced to provide a stable level of manpower, equipment, and facilities.

Several appendixes are included to provide additional information on certain planning data (such as the assembly and test planning sheets), which would become cumbersome in the main text. The remaining appendixes are provided to suggest the scope of the plans that will be provided in Phase IB. The following plans are outlined in these appendixes:

- Assembly, test, and launch operations
- Alignment
- Magnetic properties control
- Contamination control

- Electromagnetic compatibility control
- Material acquisition
- Material handling and packaging

II. VOYAGER PROJECT SCHEDULES

1. PHASE IB SUMMARY SCHEDULE

The activities to be carried out during the implementation of Phase IB include a major system design effort occurring during the first three months of the program. This effort will be conducted by three teams of design personnel. One team will be located in residence at JPL's facilities to conduct spacecraft system engineering. This team will be composed of the personnel from Spacecraft Engineering Operations (see Voyager Organization Plan, submitted in response to RFP-VPMG 66-1a) and the various key subsystem design engineers. Concurrent with this effort, two teams will begin work at TRW. The first will undertake the subsystem detailed design and analysis required to provide data to the spacecraft engineering effort at JPL. The second will be up-dating and revising the various Voyager project management and implementation plans in accord with JPL requirements.

The TRW spacecraft system engineering effort at JPL will consist of analysis and design tasks leading to the establishment of spacecraft system, subsystem, and OSE design requirements, and preliminary system interface requirements, all based on the revised JPL guidelines and on the incorporation of Ranger and Mariner experience. This work will be reviewed at the first design audit occurring the sixth week. The TRW design team at JPL will forward the over-all design requirements to the various subprojects along with task statements of work. This team will provide the necessary guidance to each of the subsystem areas for implementing these tasks. Under the direction of JPL the team will coordinate spacecraft interface requirements with those of other systems in the Voyager program. A final spacecraft interface requirements document will then be prepared and submitted to JPL for approval and issuance subsequent to Phase IB completion.

The subsystem engineering effort at TRW will consist of an initial updating of subsystem design data and the initiation of design studies and analyses in accord with the directions of the system design team at JPL.

The subsystems groups will also define the requirements for critical breadboard testing and submit these requirements to the system design team and JPL for review and approval.

As the design effort continues, the spacecraft design team, in concert with the subsystem design groups, will evolve preliminary spacecraft system specifications, subsystem specifications, OSE specifications, and Part I, CEI specifications, during October of 1966. A second design audit will be held at this time to review and approve the specifications. These updated specifications then will form the basis for continuing design and analysis tasks, for identification of preliminary parts lists and critical long lead-time items, and for updating of the Phase II program plan and initiation of the detailed Phase II costing effort.

At this time, the spacecraft design team (including JPL personnel) will relocate at TRW facilities and the tasks of detailed analysis and design will continue. The results of these studies and analyses, and those obtained from the breadboard testing, will be combined in final design details leading to a third design audit and the preliminary design review (PDR) occurring during January of 1967. The input data to the design review will consist of:

- Spacecraft system specifications
- Subsystem specifications
- Part I, CEI specifications (partial)
- Design data package
- Integrated test plan
- System and subsystem OSE specifications
- Part I, CEI specifications (OSE)

At the completion of the PDR, the recommended changes will be incorporated in these documents, and the design requirements output of Phase IB will result in:

- Approved spacecraft system specification
- Approved subsystem specifications

- Final Part I, CEI specifications (partial)
- System Phase IB design data report
- Subsystem Phase IB design data reports
- Approved integrated test plan
- Approved system and subsystem OSE specifications
- Approved Part I, CEI (OSE) specifications

The project management and implementation activities occurring during Phase IB include the revision of the initial Phase IB and II management plans and the implementation of those plans applicable to Phase IB activities. The requirements for generating other plans not yet identified will be implemented as required. Major tasks, early in Phase IB, will be the updating of the Phase II program plan and updating the preliminary studies and analyses required in order to establish a firm set of premises for the Phase II costing effort. The finalization of the Phase II program plan and the initiation of the costing effort will begin during October of 1966. The output of this phase of the effort will consist of the following plans:

- Approved assembly and checkout plan
- Preliminary launch operations plan
- Preliminary mission support operations plan
- Updated Phase II organization plan
- Updated Phase II implementation plan
- Updated Phase II procurement plan
- Updated Phase II safety plan
- Updated Phase II project control plan
- Updated Phase II facilities plan
- Updated Phase II data management plan
- Updated Phase II quality program plan

- Updated Phase II reliability program plan
- Updated Phase II manufacturing plan
- Phase II cost and incentive proposal

A summary milestone schedule of these activities is shown in Figure 1. A more detailed milestone schedule for the Phase IB activities

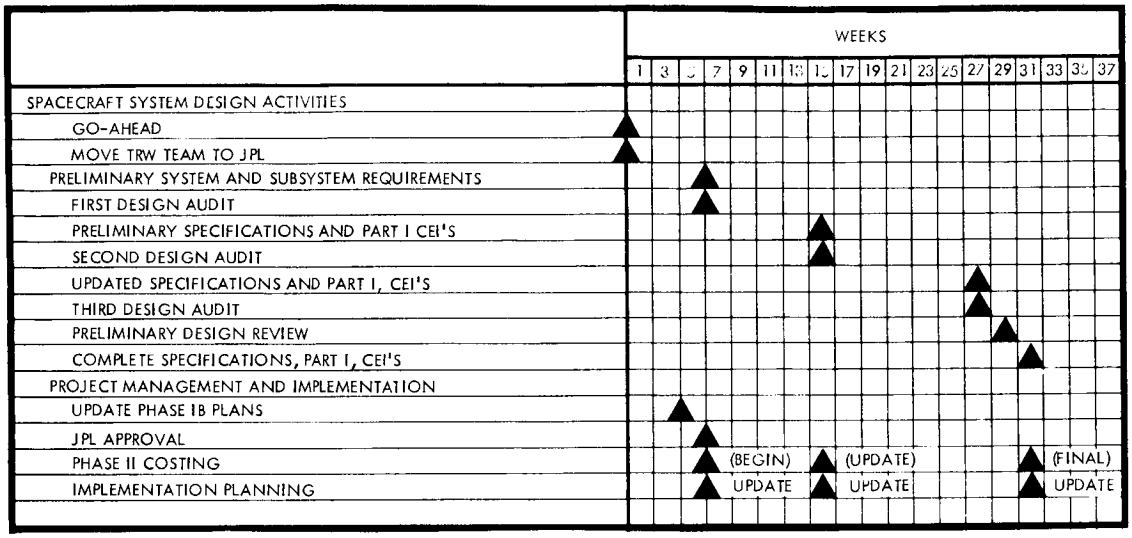


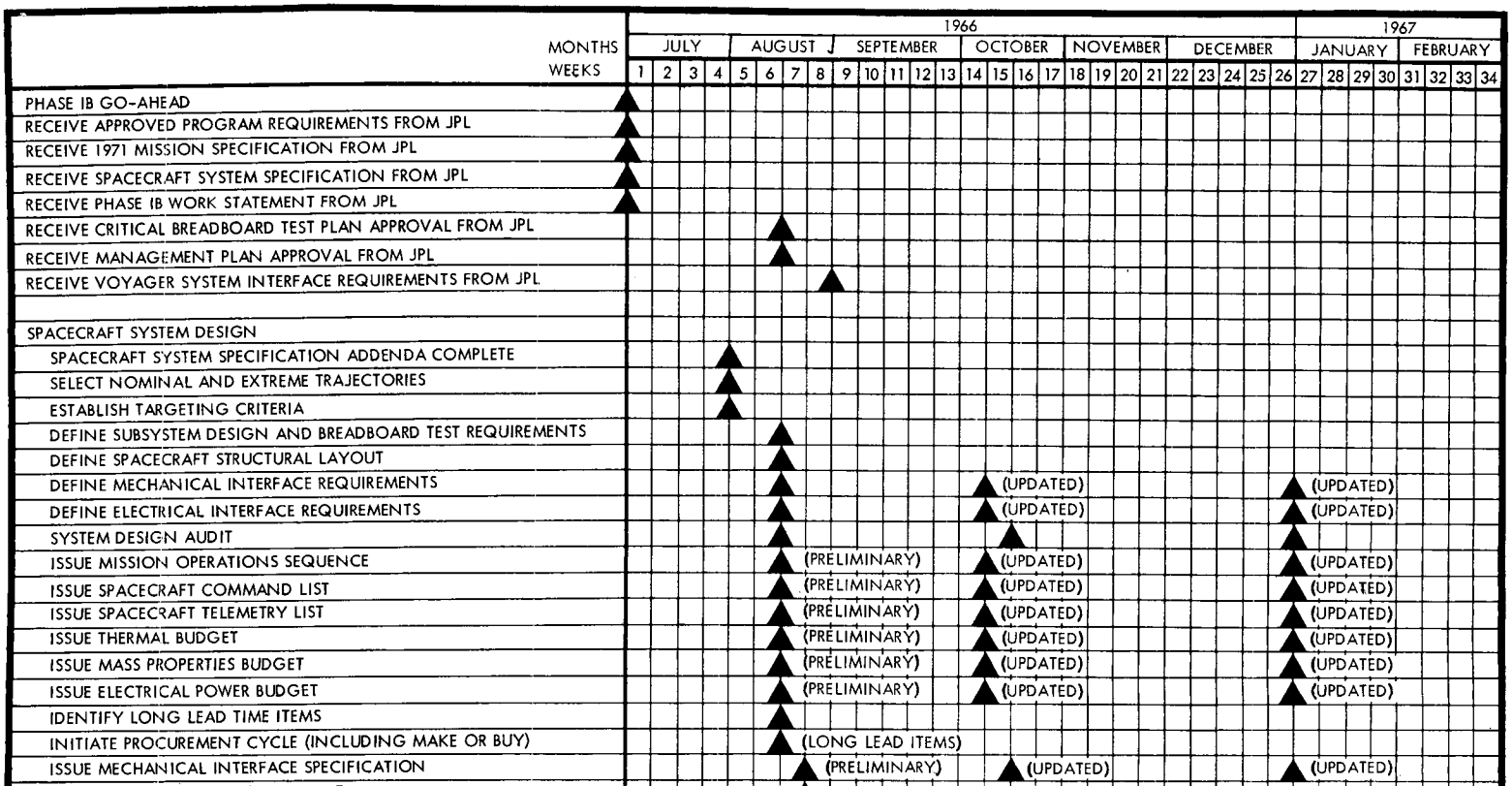
Figure 1. Phase IB Summary Schedule

is shown in Figure 2. This schedule describes the task-time relationship of the spacecraft design efforts and their relationship to the subsystem activities. In addition, the schedule shows the major activities planned during Phase IB for generating project management and implementation plans and the Phase II cost proposal.

2. PHASE II SUMMARY SCHEDULE

Figure 3 presents the major milestone schedules for the task-time relationships leading up to the 1971 launch. A preliminary list of subsystem deliveries is presented in Table 1. These relationships are based on a 5-day, 40-hour week.

Beginning early in the program, formal monthly evaluations of the details of the planned schedule will be made, as described in the Project Control Plan. If these evaluations identify problem areas, schedules


1966
1967
MONTHS
JULY
AUGUST
SEPTEMBER
OCTOBER
NOVEMBER
DECEMBER
JANUARY
FEBRUARY
WEEKS
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34
PHASE IB GO-AHEAD
RECEIVE APPROVED PROGRAM REQUIREMENTS FROM JPL
RECEIVE 1971 MISSION SPECIFICATION FROM JPL
RECEIVE SPACECRAFT SYSTEM SPECIFICATION FROM JPL
RECEIVE PHASE IB WORK STATEMENT FROM JPL
RECEIVE CRITICAL BREADBOARD TEST PLAN APPROVAL FROM JPL
RECEIVE MANAGEMENT PLAN APPROVAL FROM JPL
RECEIVE VOYAGER SYSTEM INTERFACE REQUIREMENTS FROM JPL
SPACECRAFT SYSTEM DESIGN
SPACECRAFT SYSTEM SPECIFICATION ADDENDA COMPLETE
SELECT NOMINAL AND EXTREME TRAJECTORIES
ESTABLISH TARGETING CRITERIA
DEFINE SUBSYSTEM DESIGN AND BREADBOARD TEST REQUIREMENTS
DEFINE SPACECRAFT STRUCTURAL LAYOUT
DEFINE MECHANICAL INTERFACE REQUIREMENTS
(UPDATED)
(UPDATED)
DEFINE ELECTRICAL INTERFACE REQUIREMENTS
(UPDATED)
(UPDATED)
SYSTEM DESIGN AUDIT
ISSUE MISSION OPERATIONS SEQUENCE
(PRELIMINARY)
(UPDATED)
(UPDATED)
ISSUE SPACECRAFT COMMAND LIST
(PRELIMINARY)
(UPDATED)
(UPDATED)
ISSUE SPACECRAFT TELEMETRY LIST
(PRELIMINARY)
(UPDATED)
(UPDATED)
ISSUE THERMAL BUDGET
(PRELIMINARY)
(UPDATED)
(UPDATED)
ISSUE MASS PROPERTIES BUDGET
(PRELIMINARY)
(UPDATED)
(UPDATED)
ISSUE ELECTRICAL POWER BUDGET
(PRELIMINARY)
(UPDATED)
(UPDATED)
IDENTIFY LONG LEAD TIME ITEMS
INITIATE PROCUREMENT CYCLE (INCLUDING MAKE OR BUY)
(LONG LEAD ITEMS)
ISSUE MECHANICAL INTERFACE SPECIFICATION
(PRELIMINARY)
(UPDATED)
(UPDATED)

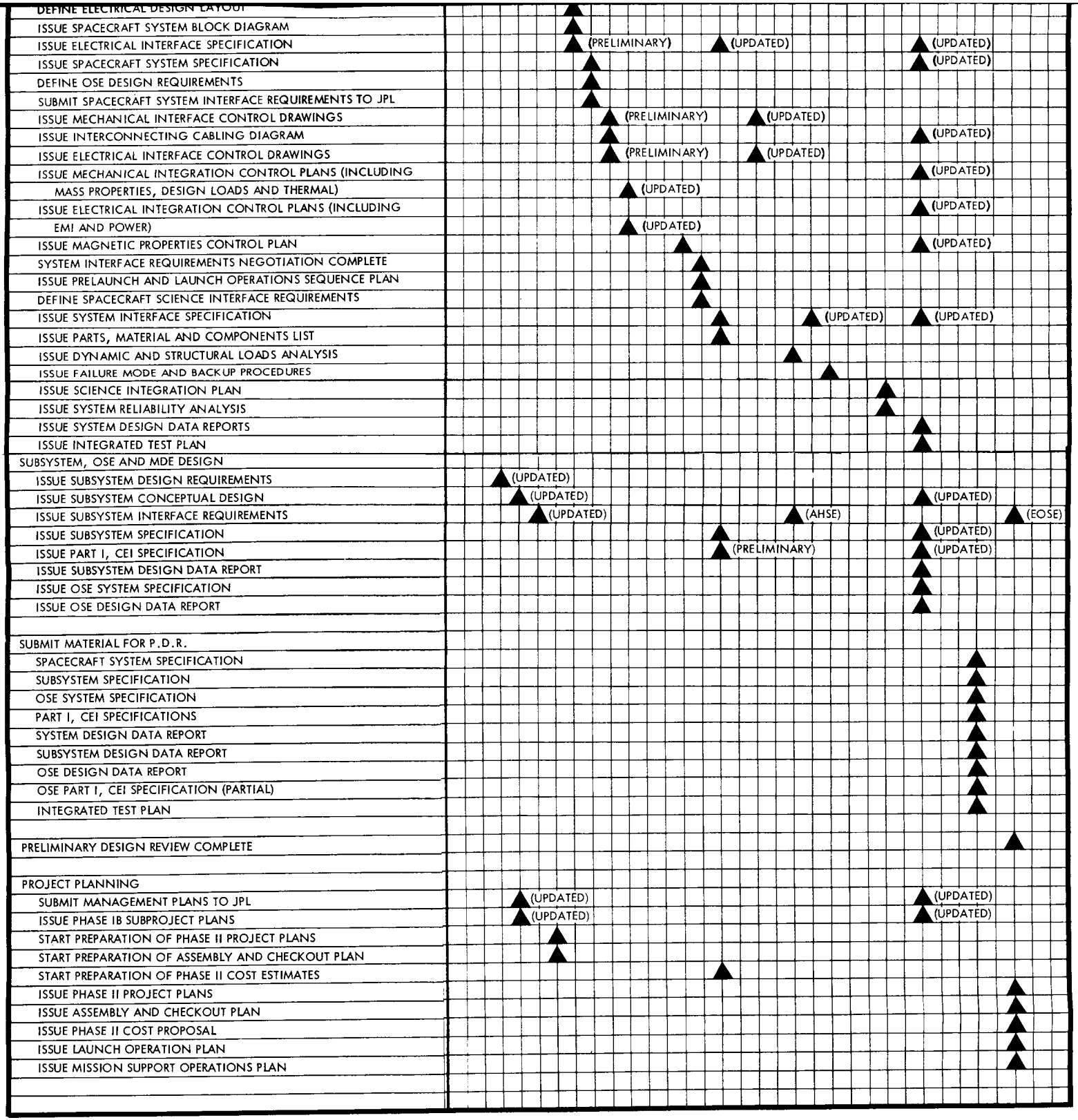
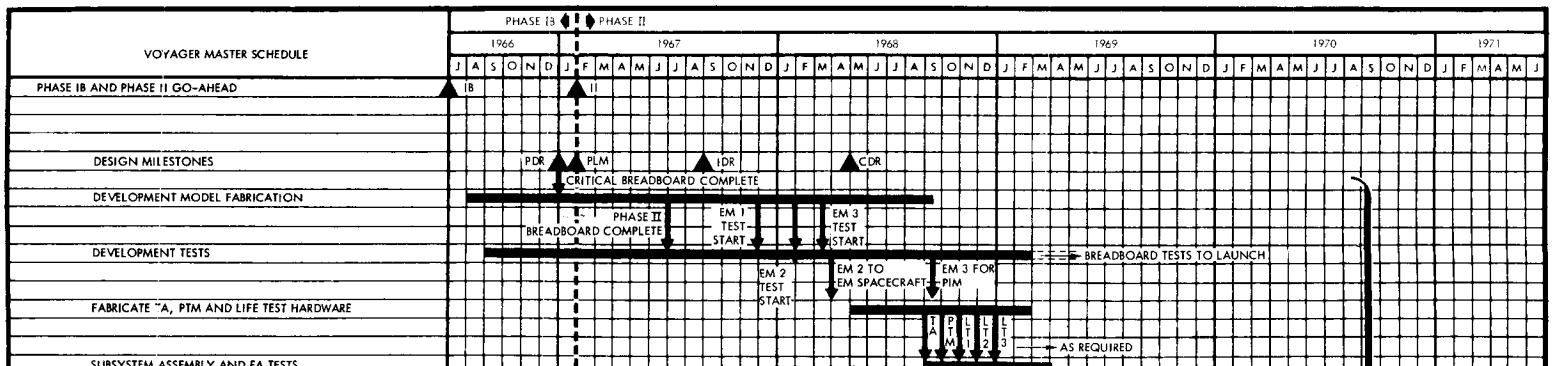


Figure 2. Phase IB Milestone Schedule


VOYAGER MASTER SCHEDULE
PHASE IB
PHASE II
1966
1967
1968
1969
1970
1971
J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J
PHASE IB AND PHASE II GO-AHEAD
IB
II
DESIGN MILESTONES
PDR
PLM
IDR
CDR
CRITICAL BREADBOARD COMPLETE
DEVELOPMENT MODEL FABRICATION
PHASE II BREADBOARD COMPLETE
EM 1 TEST START
EM 3 TEST START
DEVELOPMENT TESTS
BREADBOARD TESTS TO LAUNCH
EM 2 TEST START
EM 2 TO EM SPACECRAFT
EM 3 FOR PIM
FABRICATE "A, PTM AND LIFE TEST HARDWARE
T A
P T M
L T 1
L T 2
L T 3
AS REQUIRED
SUBSYSTEM ASSEMBLY AND EA TESTS

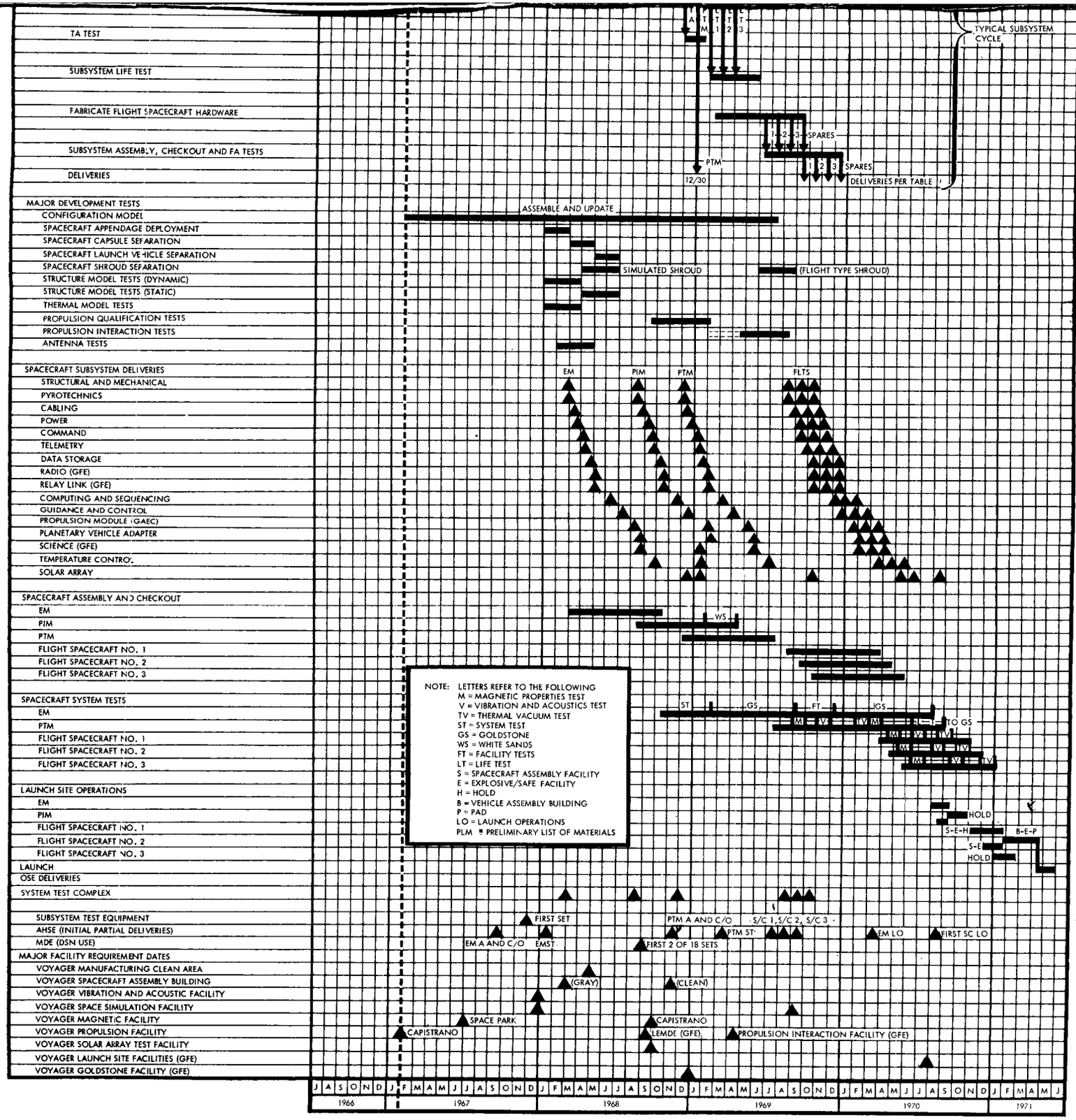


Figure 3. Phase II Milestone Schedule

Name and Identification Number		Separation Model	Config Model
0100	Structure and Mechanical	1/1/68	2/1/67
2200	Pyrotechnics	1/1/68	" "
0400	Cabling		" "
1400	Power (less solar array)		" "
0200	Command		" "
1200	Telemetry		" "
1900	Data Storage		" "
0600	Radio		" "
			" "
2100	Relay Link (GFE)		" "
0500	Computing and Sequencing		" "
0700	Guidance and Control		" "
0800	Propulsion Module	1/1/68[1,9]	" "
2000	Planetary Vehicle Adapter	1/1/68[10]	" "
2300	Science (GFE)		" "
1300	Temperature Control		" "
1400	Solar Array		" "

(1) On dock, TRW
(2) Simulated model, RCA
(3) Nonfiring model
(4) External mockup, TRW supplied
(5) As required for antenna test
(6) As required for thermal test
(7) As required for structural test

Table 1. Requirements for Subsystem Deliveries (Excluding S

ıration (12)	Structural Model (7)	Thermal Model (6)	Laboratory Engineering Model	Antenna Model	Engineeri Model
			(EM_1)		(EM_2)
	1/1/68	1/1/68		1/1/68[4]	3/1/68
					3/1/68
			1/13/68	1/1/68[5]	3/15/68
			1/20/68		3/25/68
			2/3/68		4/5/68
			2/9/69		4/11/68
			2/27/68		4/29/68
			3/1/68		RCA 5/3/68[1]
			3/14/68	1/1/68 TRW	5/17/68
			3/1/68[1]	1/1/68[1]	5/3/68[1]
			4/12/68		6/12/68
			5/2/68		7/5/68
	1/1/68[1]	12/1/68[1]		1/1/68[4]	8/7/68[1,3]
	1/1/68	1/1/68		1/1/68[4]	8/21/68
				1/1/68[4]	8/21/68[1]
	1/1/68	1/1/68		1/1/68[4]	10/1/68
	2/1/68[1,2]	1/1/68[1,2]	(RCA)	1/1/68[4]	12/15/68

(8) Convert propulsion qualification subsystem to PIM
(9) Lower interface hard mockup, mass, inertia, GAEC
(10) Preprototype structure, DAC
(11) Use RCA engineering model solar array
(12) Hard mockup, from respective contractors
(13) Use structural model solar array

pares)

ng	T/A	PIM (EM_3)	PTM	1	2	3
	12/30/67	8/16/68	12/2/68	8/17/69	9/15/69	10/16/69
	12/1/68	8/16/68	12/2/68	8/17/69	9/15/69	10/16/69
	8/30/68	8/29/68	12/13/68	9/2/69	10/2/69	11/2/69
	8/30/68	9/6/68	12/20/68	9/10/69	10/9/69	11/8/69
	12/1/68	9/19/68	1/3/69	9/23/69	10/22/69	11/22/69
	9/30/68	9/25/68	1/9/69	9/30/69	10/29/69	11/29/69
	8/30/68	10/11/68	1/27/69	10/15/69	11/13/69	12/13/69
	7/30/68	10/17/68[1]	2/1/69[1]	10/21/69[1]	11/20/69[1]	12/20/69[1]
	6/30/68	10/31/68	2/14/69	11/4/69	12/4/69	1/4/70
		10/17/68[1]	2/1/69[1]	10/21/69[1]	11/20/69[1]	12/20/69[1]
	11/15/68	11/25/68	3/12/69	12/1/69	12/29/69	1/29/70
	1/30/69	12/18/68	4/2/69	12/23/69	1/24/70	2/24/70
	9/15/68	2/3/69	5/3/69[1]	1/24/70[1]	2/24/70[1]	3/24/70[1]
	12/30/67	2/3/69	5/20/69	2/9/70	3/9/70	4/9/70
		1/15/69[1]	5/19/69[1]	2/6/70[1]	3/6/70[1]	4/6/70[1]
		1/15/69	7/1/69	3/20/70	4/17/70	5/17/70
,11		1/15/69[1,13]	10/13/69[1]	5/14/70[1]	6/24/70[1]	8/5/70[1]

will be revised when appropriate, or acceleration measures, including overtime, will be instituted when necessary.

When the flight spacecraft starts into the assembly and test operations, the formal schedule evaluation will be made weekly. When the flight spacecraft is shipped to ETR, the schedule evaluation will be made daily.

2.1 Schedule Generation

The schedule was generated by first defining the time before launch when it is necessary to initiate assembly and checkout of the first flight spacecraft. The time required was derived from a detailed, elapsed-time analysis of the tasks involved in the launch site operations, shipping, flight acceptance testing, and assembly and checkout operations. The next step defined the delivery date for each subsystem as a function of need date during the spacecraft assembly and checkout sequence. In turn, by accounting for the subsystem flight acceptance testing and manufacturing span, the start date for the manufacturing of each flight subsystem was defined. Thus a flight hardware drawing release date was established.

The start of PTM assembly and checkout operations was determined by scheduling completion of the major portion of the PTM type approval testing (i.e., magnetic, vibration, acoustics, and space simulation testing) one month prior to completion of assembly and checkout of the first flight spacecraft. This constraint then established the delivery dates for the PTM subsystem assemblies.

The drawing release dates for the fabrication of the subsystem type approval and PTM assemblies was set (for each subsystem) by the following constraints:

- The fabrication of the subsystem type approval models must precede the fabrication of the PTM assemblies by one month.

- Subsystem type approval testing must be complete prior to manufacturing flight hardware. This establishes the manufacturing drawing baseline dates (hence CDR). The fabrication and test cycle for the subsystem reliability demonstration units was inserted following the start of the PTM assemblies.

This process established the required CDR dates for each subsystem. The CDR dates for each subsystem then formed the basis on which each subproject engineer established Phase II implementation plans and schedules.

The subproject schedules supporting the major milestone schedule are presented in Section IV. Phase II subsystem milestones and their constraints are:

- Intermediate Design Review. Prepare updated subsystem specifications and complete Part I, CEI specifications for both flight equipment and OSE. Release engineering models (EM) No. 2 and 3 drawings to manufacturing.

- Critical Design Review. Prepare updated subsystem and Part I, CEI specifications, and preliminary Part II, CEI specifications for flight equipment and OSE. Complete partial type approval testing on either EM No. 1 or 2. Release drawings for type approval, PTM, and reliability demonstration models. Release OSE drawings for all units.

- Completion of Subsystem Type Approval Tests. Release updated drawing for flight spacecraft (Nos. 1, 2, 3, and spares).

- First Article Configuration Inspection (November 1970). Approve final Part II CEI specifications. Ship first flight spacecraft to Eastern Test Range.

2.2 Schedule Philosophy

The schedule provides for a high probability of meeting all program objectives with adequate time available for correcting any design deficiencies uncovered during the various testing phases. Beginning in Phase II, breadboard models of each subsystem will be fabricated and testing initiated to provide early data for design evaluation. The results of the breadboard tests will be used to correct or improve subsystem performance and will be reflected in updated subsystem and completed Part I, CEI specifications for the Intermediate Design Reviews (IDR). These results will also be used to prepare subsystem engineering model assembly drawings, which are then converted in engineering model subsystems. It is planned to build three engineering models of each subsystem. The first of these is used for design verification testing of

assemblies in the proposed flight configuration, and may be used to evaluate the subsystem performance under environments of temperature, vibration and thermal vacuum over specified ranges of power and signal anomalies. The second engineering model, built by the manufacturing organization to drawings approved at the IDR, may also be used for further testing or delivered directly to the engineering model spacecraft for early spacecraft compatibility testing. The results of subsystem engineering model testing (on either EM No. 1 or EM No. 2) are used to refine the design and update the specifications and final manufacturing drawings, in preparation for the critical design review (CDR). EM No. 3 subsystems are built for use in the propulsion interaction model (PIM) spacecraft.

Early in 1968, major development testing of separation models, structural model, thermal model, and full-scale antenna models are completed, and results are available early enough to include any design modifications in the PTM assemblies.

The assembly and checkout of engineering model spacecraft is initiated in February of 1968 and completed in mid-October of that year. The results accumulated from this operation provide the means of verifying spacecraft system, subsystem, OSE, facility, and procedure compatibility, thereby assuring high confidence in the timely completion of the ensuing operations on the PIM, PTM, and flight spacecraft.

The schedule also provides for the initiation of reliability demonstration testing (including life, wearout, and one-shot testing) approximately six months prior to the delivery of hardware to the first flight spacecraft.

The PTM spacecraft will have completed the most significant portion of its type approval test sequence (i.e., space simulation) one month prior to the completion of assembly operations on the first flight spacecraft. Experience indicates that, with the background of engineering model testing, the failures likely to be encountered during this phase of testing do not result in major design changes. The required changes can readily be accomplished on the flight spacecraft by schedule acceleration techniques.

The scheduled times for flight spacecraft assembly and test operations are identical to those for the EM, PIM, and PTM spacecraft. The application of a learning curve to these operations would result in reduced schedule times for the flight spacecraft operations. However, this factor was not applied at this time in order to provide a more conservative approach.

III. TEST PLANS AND PLANNING

1. INTRODUCTION

This section discusses the tests and related activities associated with the Voyager spacecraft project. It is the basis from which the preliminary integrated test plan will be derived for Part 2 of the Phase IB proposal. Integrated test planning will include a detailed evaluation of test adequacy and effectiveness. The plan presented here will be expanded to cover more detailed test requirements. Evaluations will be made of the completeness of the entire test program, adequacy of individual tests and cost effectiveness of alternate testing methods. The tests will be individually scheduled and the test equipment and facilities will be identified in terms of quantity and need date.

The main purpose of test planning at this time is to identify test hardware and prepare schedules for planning purposes. Preliminary integrated test planning has also been performed to assure efficient use of major test articles.

Since the scope of this plan covers all testing, from parts and materials to top-level system tests, it has many interfaces with other plans and documents. Table 2 presents these document interfaces. A major interface is with the subsystem development plans, Section IV.3. Each subsystem development plan includes a subsystem test program which describes or references all tests below the spacecraft system level on that subsystem. The subsystem test programs describe those aspects of testing peculiar to the subsystem and reference testing common to several subsystems. The standard sequence of tests for type approval and flight acceptance is described in this section. Other generalized test programs, including parts and materials, magnetics, electromagnetic compatibility, manufacturing, and reliability demonstration are described in other plans as noted on Table 2. System tests and combined systems tests are described in their entirety in Section IV.5 of this volume.

This section is divided into two parts. The first part is a summary of the test plan itself. Other documentation is referenced for detailed descriptions. The second part discusses the test plan implementation, including organization, integrated test planning, and data management.

Table 2 . Test Plan Document Interface

TYPE OF TEST	DOCUMENT
Tests Below the System Level	
Development	
Parts and materials	Preliminary Reliability Program Plan
Breadboard	Each Electronic Subsystem Development Plan
Engineering model	Each Electronic Subsystem Development Plan
Structural model	Structural and Mechanical Development Plan
Propulsion	Propulsion Subsystem Development Plan
Propulsion interaction	Spacecraft System Test
Thermal control model	Temperature Control Subsystem Development Plan
Antenna model	Radio Subsystem Development Plan
Electromagnetic compatibility	EMC Control Plan (Appendix E)
Magnetic evaluation	Magnetics Control Plan (Appendix C)
Other development	Each Subsystem Development Plan
OSE	OSE Implementation Plans (Vol 4)
Type Approval	
Electronic subsystem	Each Subsystem Development Plan
Structural model	Structural and Mechanical Development Plan
Separation	Structural and Mechanical Development Plan
Deployment	Structural and Mechanical Development Plan
Mechanical component	In applicable Subsystem Development Plans
Propulsion	Propulsion Subsystem Development Plan
Pyrotechnic lot qualification	Pyrotechnic Subsystem Development Plan

Table 2. Test Plan Document Interface (Continued)

TYPE OF TEST	DOCUMENT
Tests Below the System Level	
Type Approval	
Pressure vessel	Guidance and Control Subsystem Development Plan
Reliability demonstration	Preliminary Reliability Program Plan (also included in subsystem development plans.)
OSE	OSE Implementation Plans (Vol 4)
Electromagnetic compatibility	EMC Control Plan (Appendix E)
Magnetic evaluation	Magnetics Control Plan (Appendix C)
Flight Acceptance	
Manufacturing	Preliminary Manufacturing Plan
Electronic subsystem	Each Subsystem Development Plan
Structural and mechanical	Structural and Mechanical Development Plan
Propulsion	Propulsion Subsystem Development Plan
Pyrotechnics lot acceptance	Pyrotechnic Subsystem Development Plan
OSE	OSE Implementation Plan (Vol 4)
Electromagnetic compatibility	EMC Control Plan (Appendix E)
Magnetic evaluation	Magnetics Control Plan (Appendix C)
Systems Test and Combined System Tests	All system tests and combined system tests are described in Spacecraft System Test

2. TEST PLAN

The test nomenclature, such as development, type approval, flight acceptance, combined systems tests, system tests, and tests below the system level, has been taken from the guidelines and preliminary specifications provided by JPL. The test plan presented here is organized as follows:

- Tests below the system level
  - General test programs
  - Electronic subsystems
  - Other subsystems
- System tests
- Combined systems tests
- OSE tests

Subheadings in the tests below the system level are further subdivided into development, type approval, and flight acceptance as appropriate.

2.1 Tests Below the System Level

The tests in this section are divided into three groups: general test programs, electronic subsystems, and other subsystems. The general test programs are covered by other control plans (e.g. Magnetics Properties Control Plan) and apply to all subsystems; they are briefly summarized and referenced. The section on electronic subsystem tests describes the standard test program for electronic subsystems, including development, type approval, and flight acceptance tests. The section on other subsystems includes the nonelectronic subsystems such as mechanical and structural, propulsion, and temperature control. Tests tend to be unique to the subsystem, and are located in the corresponding subsystem development plan.

2.1.1 General Test Programs

The following test programs are general in that they cover more than one subsystem and are described in other control plans.

a. Parts and Materials Tests

The parts and materials tests will be in direct support of the approval and listing requirements set by the Parts, Materials, and Processes Selection Board. The functions of this board and its operating criteria for testing are established in the preliminary spacecraft system reliability plan. Specific categories of nonrecurring parts and materials tests include:

- Analysis of basic material properties
- Qualification tests
- Accelerated life tests
- Test-to-failure evaluations
- One-shot success testing

Specific categories of recurring parts and materials tests include:

- Parameter drift screening
- Periodic lot requalification
- Lot acceptance testing

The tests will also include the evaluation of magnetic properties and resistance to contamination-control conditioning. The magnitude of statistical-sample requirements for assuring reliability of parts and materials will be defined in the intermediate spacecraft system reliability plan scheduled for development in Voyager Phase IB.

b. Magnetic Tests

Magnetic properties testing is described in the Magnetics Properties Control Plan, Appendix C. Magnetic testing is planned at the parts or component level, the assembly level, and the spacecraft level. No magnetic testing is planned at the intermediate assembly level or at the subsystem level.

The tests that are to be performed as part of magnetic testing are:

- Measure the magnetic fields of components
- Measure magnetic fields both operating and nonoperating at the assembly level

- Map the field of selected assemblies in the operating and nonoperating modes
- Measure the coercive force at the assembly level
- Measure and map the magnetic fields nonoperating and operating at the spacecraft level
- Measure the coercive force at the spacecraft level

c. Electromagnetic Compatibility Tests

The electromagnetic compatibility (EMC) testing is described in Appendix E. A brief description of the types of EMC tests and where they are applied is listed below.

Breadboard Testing

- Determine breadboard noise and transient susceptibility threshold characteristics
- Investigate all breadboard signal characteristics in regards to affecting inverter and transformer-rectifier operation

Engineering Models

- Determine engineering model assembly noise and transient susceptibility threshold characteristics
- Determine engineering model subsystem noise and transient susceptibility threshold characteristics
- Investigate engineering model assembly signal characteristics in regards to affecting inverter and transformer-rectifier operation
- Investigate engineering model subsystem and spacecraft signal characteristics for their effect on inverter and transformer-rectifier operation
- Investigate engineering model electrical bonding, grounding, and shielding effectivity at the subsystem and spacecraft level
- Perform engineering model radiated interference measurements at the assembly, subsystem, and spacecraft levels
- Perform antenna conducted interference measurements at the assembly, subsystem, and spacecraft levels

All of the above tests will be repeated for the type approval units at the subsystem level after the engineering model testing phase has been completed.

The EMC testing for the PTM and flight models is identical to that of the engineering models and type approval models and will take place at the subsystem and spacecraft levels.

d. Manufacturing Tests

Manufacturing tests are conducted before and after conformal coating, encapsulation or enclosure for each electronic subassembly, assembly, and component. These tests, procedures to be used, and inspection requirements are referenced on the manufacturing shop order in accordance with the manufacturing flow sequences described in the manufacturing plan.

Manufacturing tests are performed at nominal input and load conditions to meet detailed output requirements. Procedures for these tests include: equipment requirements, test conditions, step-by-step instructions, and criteria for acceptability. Upon acceptance, following such tests, test data is included in and stored with the manufacturing data package as described in the quality assurance plan.

e. Reliability Demonstration Tests

Reliability demonstration tests for the spacecraft and its subsystem components are discussed in the Preliminary Spacecraft System Reliability Plan. The technical criteria for the planned test quantities and test intervals are given in Volume 1, Section IV. 4. The spacecraft level reliability test requirements are set at 1750 hours (at 40°C) in the space chamber. Individual subsystem component tests are classified as: life tests, wearout tests, and one-shot tests.

Component life tests are continued for either 6, 12, or 18 months depending upon their expected reliability levels. Component quantities and test environments are selected to obtain an equitable reliability assurance for all components. Wearout reliability tests are provided for selected components to assure the ability to perform beyond the maximum 1971 mission period. One-shot reliability tests are provided for selected components to provide an engineering confidence in the reliability of critical functional events.

2.1.2 Electronic Subsystems

A general test program for electronic subsystems is presented here. The specific test program for each electronic subsystem is presented in the subsystem test plan, and additions, deviations, and exceptions are noted there. The general test program is meant to provide a starting point for integrated test planning that provides optimum tests for each subsystem. This planning will continue during the preparation of the Phase IB proposal and during Phase IB.

The tests on the electronic subsystems are divided into three categories: development, type approval, and flight acceptance.

a. Development

A typical subsystem development test cycle begins with breadboard testing to develop the design details and, in addition, produces:

- Lists and specifications for material, parts, and processes
- Specifications for subcontract items
- High confidence system design data covering reliability, size, weight, volume, thermal dissipation, and power consumption
- Test procedures for engineering model tests

Thus the completion of breadboard testing provides detailed, high confidence data to the spacecraft system designers; provides detailed layout and schematics for the initiation of the design of the engineering models; and provides an early identification of parts and material requirements. This information is reviewed at the intermediate design review.

A typical breadboard test sequence is presented in Table 3, with a progression of tests from assembly level through subsystem level. Some of these tests are omitted if the test and schedule requirements are satisfied by engineering model tests.

The next major subsystems test phase consists of testing engineering models. The completion of this series of test produces:

- Released drawings and specifications
- Full design margin test results

Table 3. Breadboard Tests

NAME	DESCRIPTION
Assembly Level	
Temperature	Vary power supply ± 15% over TA temperature range. Then step-stress temperature at nominal power supply until performance is degraded.
Transient susceptibility	Subject assembly to transients, noise, and audio tones injected into the power supply lines and other critical signal lines. Vary noise and transients 15% in excess of TA levels.
Power frequency variation	Subject assembly to changes in the primary power frequency. Vary frequency in steps until performance is out of acceptable limits.
Additional parameters	Vary other critical parameters in discrete steps until normal circuit operation ceases.
Failure mode	Determine effects of failures of redundant parts.
Interface	Check interfaces between assemblies of a given subsystem; check critical interfaces between subsystems; check interface with EOSE.
Subsystem Level	(The following tests may be satisfied by engineering model tests.)
Margin	Vary voltage, temperature, noise injection, power frequency, power overshoot, and power transients as specified by Voyager test requirements specification.
Life	Continuously exercise subsystem breadboard; update as required and repeat margin tests at monthly intervals.

- Demonstration of size, weight, volume, thermal characteristics, power consumption, magnetic problem areas, intrasubsystem compatibility, and functional performance
- Test procedures for type approval (TA) testing
- Engineering models for the engineering model (EM) spacecraft

The successful completion of the EM test phase provides firm design data for the spacecraft subsystems design and supports the final release of drawings to enable manufacturing and subcontractors to proceed with a high confidence of producing reliable end items. This information is reviewed at the critical design review.

There will generally be three engineering models of each electronic assembly. The first is used for subsystem tests only. The second is used in the engineering model spacecraft, and the third is used in the propulsion interaction test model. The use of engineering models is proposed for the following reasons:

- Equipment almost identical to flight hardware can be produced with preliminary tooling early in the schedule
- Test procedures can be checked in an informal atmosphere
- Design changes can be incorporated before the critical design review
- OSE and computer programs can be debugged during the EM cycle
- Time and expense of EM tests will be compensated for by smoother flow of official TA and FA tests

The first engineering model may be made in engineering laboratories and does not require potting. The initial tests on this model are the same as the breadboard tests described above. Thus, when breadboards are not needed for design purposes, the breadboard tests may be replaced by engineering model tests when the schedule permits. Engineering model tests also include EMC and magnetics as described in the corresponding control plans. After assembly-level tests, the engineering model assemblies are integrated into a subsystem for subsystem-level tests. This EM subsystem may replace the breadboard subsystem for continued monitoring and tests.

The second engineering model of an assembly is used for the engineering model spacecraft. This model is made in the manufacturing area and is equivalent to flight hardware with respect to conformal coating and potting. The test program for this model is coordinated with the program for the first model so that a complete spectrum of environments is covered by the two models. For example, vibration to TA levels may be performed on the second model since the parts are potted.

The third engineering model is used in the propulsion interaction test model. It is given tests equivalent to flight acceptance to confirm that workmanship and quality standards have been satisfied prior to delivery to the PIM spacecraft.

b. Type Approval

Type approval tests are performed to verify that design requirements have been met in environments which are in excess of that expected during the mission. The tests are designed to be nondestructive. They include functional and environmental tests at the assembly and subsystem level. A typical minimum type approval test program is presented in Table 4. Additional environments, such as bench handling, transportation vibration, static acceleration, package drop, and explosive atmosphere will be considered and included if appropriate. Additional environmental exposures at subsystem level will also be considered and included if appropriate.

The standard type approval test concludes with a form of life test interspersed with environmental margin tests. This is not a pure form of life test, since failures are more likely to be caused by the repeated environmental extremes than by the extended operations. Additional life tests specifically designed to investigate expected life are discussed in the preliminary reliability program plan. These additional tests, along with the wearout tests and one-shot tests in the reliability demonstration are considered as an extension of the minimum type approval test program presented here.

Table 4. Type Approval Tests, Electronic Subsystems

NAME	DESCRIPTION
Assembly Level	(Subsystems must pass flight approval tests before starting type approval tests.)
Temperature	Perform all functions with ± 15% variations in external supply voltages over the type approval temperature range.
Vibration	Attach assembly to vibration exciter. Subject to complex and low frequency sine vibration in each of three mutually perpendicular axes.
Shock	Subject to shocks in three axes while nonoperative; perform functional tests before and after.
Humidity	Verify noncatastrophic performance with elevated humidity.
Sterilization compatibility	Expose to ethylene oxide at elevated temperature; determine effects visually and electrically.
Thermal shock	(For assemblies external to spacecraft only.) Place assembly in thermal vacuum chamber with lamps and cold walls; stabilize at high temperature, turn off lamps, leave off until specified low temperature is reached, and turn on lamps.
Subsystem Level	
Voltage and temperature	Perform all subsystem functions with variation, power overshoot, and power transients as specified.
Electromagnetic compatibility	Perform EMC tests as specified in the EMC control plan.

Table 4. Type Approval Test, Electronic Subsystems (Continued)

NAME	DESCRIPTION
Subsystem Level	
Thermal vacuum	Install subsystem on heat sink in thermal vacuum chamber. Verify subsystem operation and evacuate. Subject to short term low temperature exposure and long term high temperature exposure. Monitor performance and make functional tests as specified.
Life	After completion of environmental tests, cycle subsystem through its operating modes as a form of life test. Perform margin tests at two-month intervals; margin tests include variations in voltage, temperature, and power supply as specified.

c. Flight Acceptance

The purpose of flight acceptance testing is to confirm that workmanship and quality standards have been satisfied. Flight acceptance tests include tests at an assembly level and subsystem level. A typical flight acceptance test sequence on an electronic or electromagnetic subsystem is presented in Table 5.

This typical program is separately interpreted for each subsystem. For example, the C and S subsystem has only one assembly and so there is no difference between tests at the assembly level and subsystem level. Also the radio subsystem is given thermal vacuum tests at the assembly level and subsystem level. This is done both for technical reasons (cabling impedances are particularly critical in the radio subsystem) and for management reasons (this subsystem is made and tested by RCA and delivered to TRW).

Spares are given the same FA tests as the flight units. In addition, spares are functionally checked as a subsystem. All flight units and spares are delivered to integration stores after completion of FA tests.

Table 5. Flight Acceptance Tests, Electronic Subsystems

NAME	DESCRIPTION
Assembly Level	
Pre-environmental	Perform all assembly functions with variations in supply voltage, temperature, noise injection, power frequency variation, power overshoot, and power transients as specified.
Vibration	Attach assembly to the vibration exciter and subject to complex vibration in three mutually perpendicular axes.
Magnetic	Evaluate magnetics according to magnetic control plan.
Thermal vacuum	Attach assembly to heat sink in vacuum chamber while operating, subject to combined temperature vacuum test, controlling temperature by means of heat sink and its heat exchanger. If assembly is external to the spacecraft, use radiating mechanism for temperature control.
Post-environmental	Functional test to determine post-environmental performance.
Subsystem Level	
Voltage and Temperature	Perform all subsystem functions with variations in supply voltage, temperature, noise injection, power frequency variation, power overshoot, and power transients as specified for FA testing.

While in stores, they are removed for functional tests at two-month intervals to assure that no deleterious change has occurred.

2.1.3 Other Subsystems

At the same time that the electrical subsystems progress from breadboard through engineering and prototype model testing, each of the other subsystems is also being tested. The structural-mechanical (including planetary vehicle adapter), temperature control, and propulsion subsystems lend themselves to independent testing. Each test shares the general objectives of demonstrating functional performance of the design under the full range of operating conditions in combination with critical anticipated environments. The test articles used during this phase, as in the case of electrical testing, progressively approach the production configuration.

a. Structural and Mechanical Subsystem

This section describes the major structural-mechanical tests (including planetary vehicle adapter) necessary to support analysis and design. The subsystem test models include:

- The structural model (a full scale, flight configuration structure including propulsion module with tanks, flight vehicle adapter, and capsule interstage)
- The capsule-spacecraft separation model (full scale, mass-inertia models of spacecraft and capsule with a flight configuration capsule interstage)
- Spacecraft-launch vehicle separation model (full scale mass-inertia models of spacecraft and launch vehicle with a flight configuration planetary vehicle adapter)

The structural model, which will be used for both vibration and static testing, is considered the type approval test for demonstrating static structural integrity. The separation models, which are used to demonstrate the correct operation of the separation devices, are considered the type approval tests for the separation systems. The details of these tests and the element and component tests which precede the major tests (described above) are described in Section IV.3.2.

b. Temperature Control Subsystem

This section describes the major temperature control subsystem development tests. Development tests will be run on thermal insulation and engineering model louvers. The major thermal development tests are on the thermal model (a full scale structure, including the propulsion module, an inert rocket engine and nozzle, and thermal mockups of electronic components) and external equipment thermal models. Type approval tests include component tests on louvers, heaters, and thermostats, culminating with the complete subsystem test on the PTM. Flight acceptance tests include louver assemblies, heaters, and thermostats, whereas thermal insulation assemblies will be tested at the system level flight acceptance test. Details of the tests are included in Section IV.3.14.

c. Propulsion Subsystem

The major development tests for the propulsion subsystem can be categorized as follows:

- Component and preliminary configuration model tests
- Engineering model tests
- Propulsion integration tests

The tests start in Phase IB, proceed through components development, and culminate in the propulsion subsystem type approval test. Details of the tests are included in Section IV. 3.13.

2.2 System Tests

System tests include all operations from the first spacecraft integrated system test to mating of the flight capsule to the spacecraft at AFETR. All system level environmental testing is defined and the operations flow for each spacecraft is detailed. The system tests performed at AFETR prior to capsule mating are included in this definition. These tests are defined in Section IV.5, Spacecraft System Tests, and Section IV.6, Launch Site Operations. The planning and control operations necessary for performing these tests are discussed in Section IV.4.3, Planning and Control, and Section IV.8, Assembly and Test Implementation Planning.

2.3 Combined Systems Tests

Combined systems tests are defined as those tests which require an active interface with contractors not controlled by TRW. These tasks include tests performed with the flight capsule, DSN-spacecraft compatibility tests, mission support tasks, and joint launch vehicle-planetary vehicle tests. The planning tasks required for implementation of the combined systems test plan are also discussed. This information is found in Section IV.5, Spacecraft System Tests, Section IV.6, Launch Site Operations, Section IV.7, Mission Support Operations, and Section IV.8, Assembly and Test Implementation Planning.

2.4 Operational Support Equipment

2.4.1 Electrical Operational Support Equipment

While type approval and flight acceptance tests associated with spacecraft will not be run on EOSE, qualification or acceptance tests in the broad sense will be employed to verify that the EOSE has been properly built to specifications and performs the required functions. The implementation plan contained in the OSE volume (Section IV, Volume 4) discusses the activities to be performed in Phase IB to delineate specific tests on EOSE. Tests will include design verification, type approval tests, acceptance tests, and spacecraft-EOSE compatibility tests. All tests will be performed in accordance with TRW Systems quality assurance requirements. Comprehensive test data will be collected under computer control and off-line programs will permit retrieval of test data and formatting into standardized report forms.

2.4.2 Assembly, Handling, and Shipping Equipment

The specific tests planned for AHSE are noted in the subsystem functional description included in Volume 4. Generally, all AHSE is proof-load tested and functionally tested as applicable. Acceptance tests will be conducted on all AHSE prior to spacecraft use. Slings and functional AHSE will be periodically verified to assure safe operations. Detail test procedures will be identified during Phase IB and developed during Phase II.

3. TEST PLAN IMPLEMENTATION

This section describes the organization and some of the tasks required to implement the test plan. The first part discusses the Test Board and project test office, including tasks they will perform. The second part discusses the process of integrated test planning and the third part discusses some aspects of test data management.

3.1 Test Board and Project Test Office

The Test Board, as defined by the Voyager Spacecraft Organization Plan, has the responsibility for generating and monitoring the integrated test plan, reviewing detailed plans, and certifying successful completion of tests. The Test Board is a review board, with representatives from all operations of the Voyager Project Division, Quality Assurance, and JPL. The board is supported in its tasks by the project test office, which is a staff organization in Spacecraft Assembly and Test Operations. This office provides a continuing functional organization for planning, coordinating, monitoring, and reporting all tests. It collects data and makes recommendations for review and approval by the Test Board.

During Phase IB, a major task will be to update the integrated test plan. The associated schedules will be revised as required. All tests will be grouped into discrete packages for costing purposes. The Voyager spacecraft data center storage and retrieval methods will be utilized in implementing test planning. Standard formats for test data will be established. The integrated test plan will be coordinated with other plans. Test planning for preliminary design reviews will be supported. Special studies to optimize tests will be prepared as required.

During Phase II, maintenance of the integrated test plan will continue to be a major task. The Voyager spacecraft data center will be used to collect and store test data. Special documentation required for "key tests" will be collected and prepared. Tests will be monitored and special studies will be made of test failures and anomalies. Engineering change orders will be monitored for effects on test validity. Test planning for intermediate and critical design reviews will be supported. Test program status reports will be prepared monthly.

3.2 Integrated Test Planning

Integrated test planning coordinated by the Test Board and project test office will determine the scope of testing required as well as tradeoffs between tests, within the constraints of cost, time, and facility availability. The performing organizations do the planning required for implementation (who, how, when, where) while the integrated test planning is primarily concerned with why the test is being run and what will be done with the results. The integrated test planning provides the general test policies so that a consistent level of testing is provided for all subsystems. Guidance from JPL provides consistency of testing between the spacecraft and other Voyager systems. Test tradeoff studies in the Phase IA study have been made at a very broad level. One area is the multiple use of full scale models.

The major development test models that have multiple usage are:

- Structural test model
- Thermal test model
- Propulsion interaction test model
- Engineering model
- Propulsion qualification test model

The structural model is used to conduct both dynamic and static testing. At the completion of this cycle it is converted to a test bed for the boom and appendage deployment test and as a subsystem vibration test fixture, where required (i.e., cable subsystem). It is also used for match-mate tests, AHSE testing and nose fairing separation testing. At completion of the thermal model testing, the aft equipment module structure, and the thermal control hardware and appendages are used for the propulsion interaction test model. The propulsion qualification subsystem module is then used to complete the assembly of the propulsion interaction spacecraft. The engineering model and propulsion interaction model spacecraft are then used for early off-site facility and system interface testing.

Other areas of testing tradeoffs have been resolved only in preliminary form as required to define the implementation plan. Studies will be made of the following major areas during Phase IB proposal preparation and during Phase IB.

Choices of environments and environmental levels for type approval and flight acceptance tests will be studied. A task of system engineering will be to translate environmental predictions into meaningful specific test requirements. The effects of multiple environments will also be considered. Cost and time constraints limit the permutation of tests in all combinations of environments. The significant ones must, therefore, be selected. Typical combinations being considered are:

- Cold temperature and low supply power (during Mars eclipse)
- Vibration, shock, and electrical interferences (during launch and separation)
- Mechanical shock, cold temperature, and power supply transients (during retrofiring)

The extent to which environmental exposures should be repeated at an assembly and subsystem level will receive further study. A comprehensive test program requires complete functional tests at every level of assembly. However, environmental exposures, such as vibration, should be limited to avoid fatigue. In this implementation plan, the primary environmental tests are at the spacecraft and spares level, which is generally the assembly level. Other tests have been added, based on a preliminary evaluation of their worth.

Reliability demonstration tests are subject to tradeoff studies as a class. These are tests beyond those required for a minimum qualification or type approval test program. They include life tests, wearout tests, and one-shot tests. This class of testing is quite expensive and may be curtailed under budget pressures.

The propulsion interaction test has major cost and facilities implications. This test is included in the implementation plan for preliminary planning, but it is recognized that more study is required. The vibration interactions of interest to the guidance and control subsystems are primarily of low frequency--typically 1/4 cps. It is expected that the bending and vibration frequencies excited by propulsion will be an order of magnitude higher. Study during Phase IB may show that this test is of limited use for interactions between the propulsion and guidance and control.

3.3 Test Data Management

The test program depends on the data management system to provide access to detailed test information, provide clear indication of test interrelationships, provide identification of all tests, and provide capability for change of preliminary plans. The Test Board and project test office will work with the Configuration and Data Control Department to further establish the necessary systems.

In general, tests are identified by name and coded information which identifies such items as equipment number, task number, PERT number, type of test, and location of test. The Voyager spacecraft data center processes this information onto punched cards. Sorts and tab runs are then made to provide the desired grouping of tests and checks are made for missing and overlapping tests. Detailed test plans reports and data packages are also processed through the data management system.

IV. SPACECRAFT IMPLEMENTATION PLAN

1. INTRODUCTION

The preliminary implementation plan for the 1971 Voyager spacecraft shows design engineering as the major activity of Phase IB, and spacecraft system and subsystem development, through launch and mission operations, as the major activity during Phase II. Phase IB and Phase II activities are discussed and the relationships between these phases are indicated.

The organization of the material is as follows: spacecraft engineering, subsystem development and manufacturing, assembly and checkout, spacecraft system testing, and launch and mission support operations. As a rule, this section does not discuss organization or project control, the focus being on the identification and scheduling of the tasks that must be accomplished to meet the launch date. The one exception to this rule is the discussion of spacecraft engineering.

Spacecraft engineering (Section 2) is conducted by Spacecraft Engineering Operations of the Voyager Project Division. This section discusses mission and systems requirement analyses and the means by which those requirements are translated into systems, subsystems, and interface documentation. The organization, responsibilities, and working interfaces with both JPL and TRW subsystem development groups are indicated.

Subsystem development (Section 3) treats the engineering analysis, design and testing required to flight qualify and deliver flight hardware to the spacecraft assembly and checkout operations. Sections 4, 5, 6 and 7 discuss the sequence of operations beginning with spacecraft assembly and checkout and continuing through system test, launch, and mission support. Section 8 briefly describes the Phase IB and II planning activities which are required to update the following plans:

- Assembly and checkout
- System test plan
- Launch site operations
- Mission operations support

2. SPACECRAFT SYSTEM DEVELOPMENT

2.1 Introduction

This section discusses the spacecraft engineering tasks to be accomplished during Phase IB and Phase II by Spacecraft Engineering Operations in the Voyager Project Division. The organization and responsibilities of Spacecraft Engineering Operations are further discussed in Section 4.4 of the Voyager Organization Plan.

2.2 Spacecraft Engineering Milestones

The summary milestone chart, Figure 4, outlines the spacecraft engineering schedule during Phase IB. Certain spacecraft engineering activities, particularly those system analysis tasks to be performed in Phase II, must continue throughout Voyager development to provide support to the spacecraft system development and test program and to mission operations. Similarly, functions relating to system requirements and design integration are of a continuing nature and will provide sustained support in design reviews, and specification and documentation reviews. A schedule of the Phase II activities is shown in Figure 5.

2.3 Objectives of Spacecraft System Engineering

Spacecraft Engineering Operations will be responsible for design and technical development of the spacecraft system and the subsystems to assure that the system fulfills the mission objectives optimally. Withing the framework of the mission specification and spacecraft system specifications issued by JPL, spacecraft engineering will establish a system design approach and continually perform required tradeoff studies and monitor the functional relations among the spacecraft subsystems, the spacecraft and its operational support system, and the interfaces between the Voyager spacecraft system and other elements of the Voyager project.

To accomplish these objectives Spacecraft Engineering Operations will.

- Interpret the system and mission requirements, specifications, and constraints issued by JPL, and define the technical approach to be used by TRW and its subcontractors in accordance with mission objectives

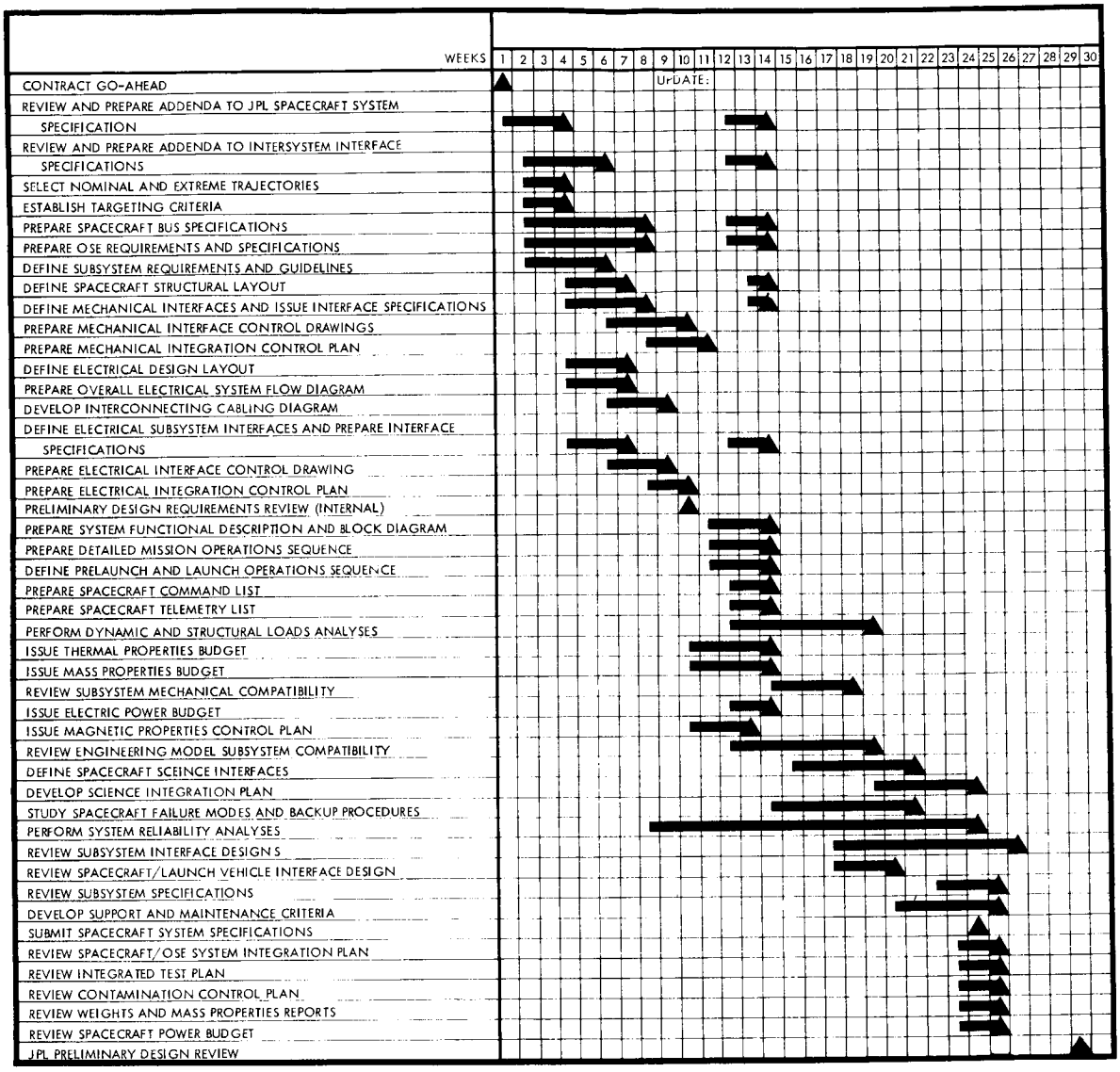


Figure 4. Spacecraft Engineering Implementation, Phase IB

- Generate the complete system level specification package including an Addendum to the JPL Voyager Spacecraft System Specifications, as necessary
- Generate intersystem interface specifications including an Addendum to the corresponding JPL interface specifications, as necessary
- Define design requirements for Voyager spacecraft subsystems; review and monitor subsystem specifications developed by all subprojects, with particular attention to subsystem interfaces

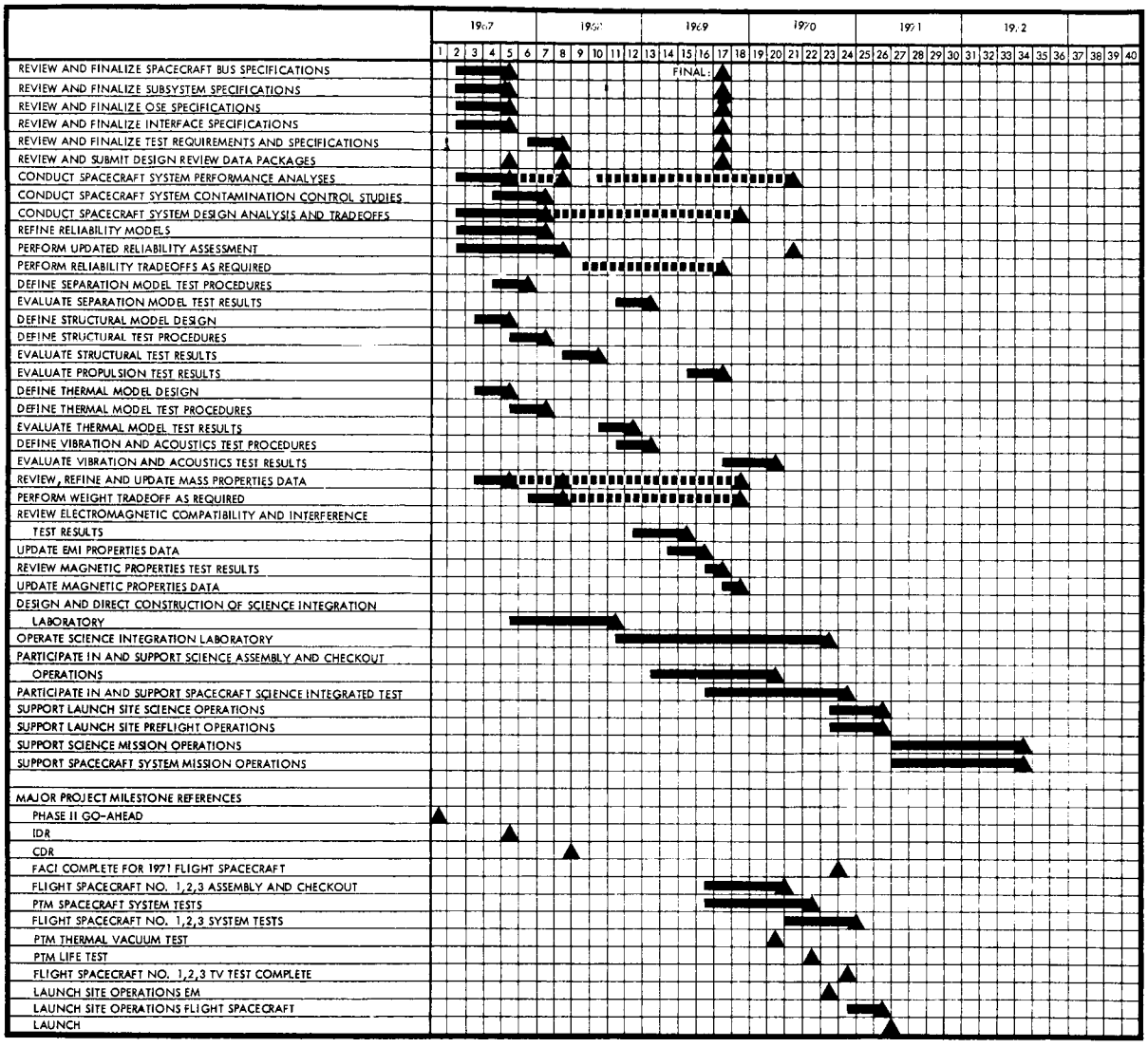


Figure 5. Spacecraft Engineering Implementation, Phase II

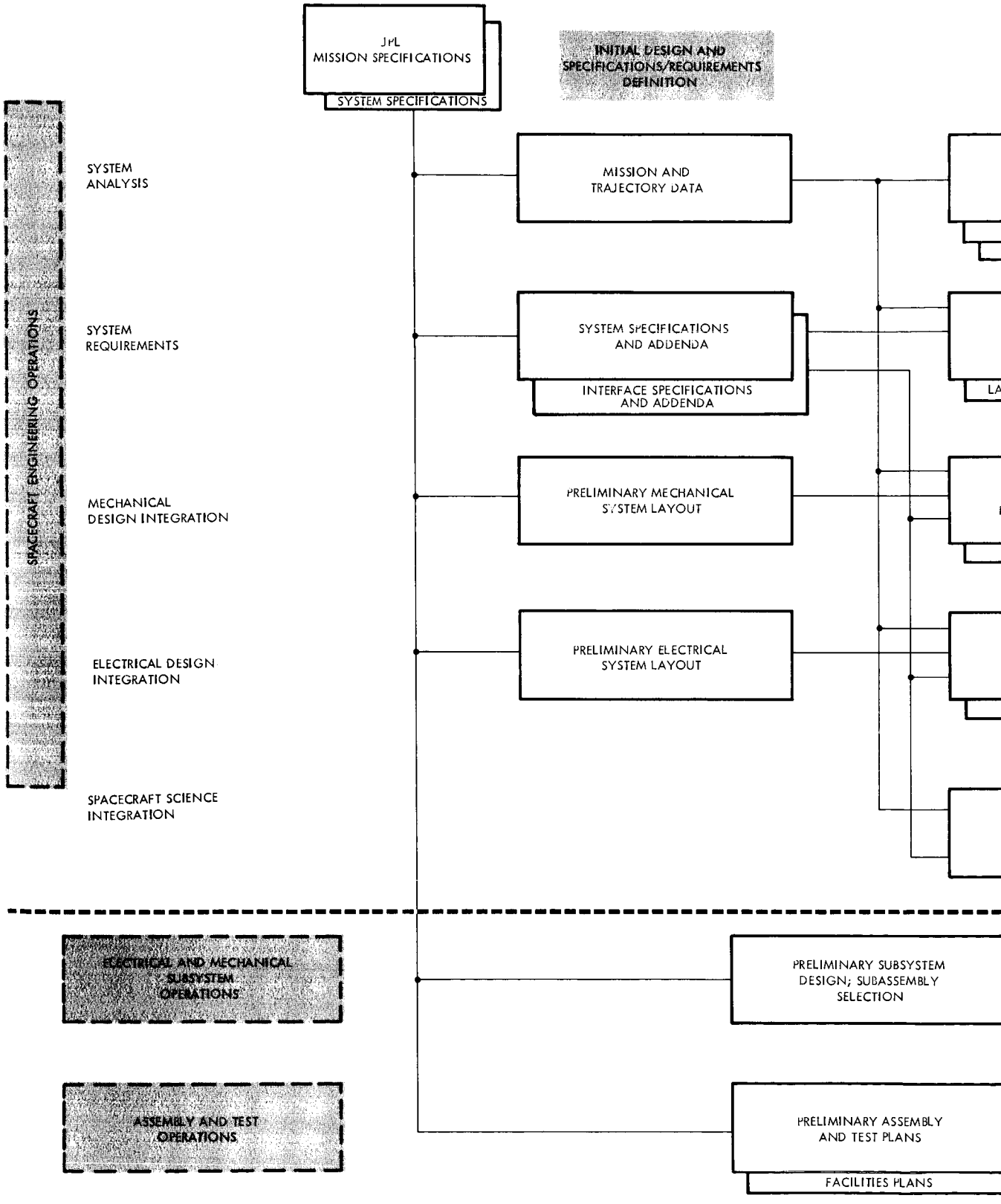
- Exercise responsibility for over-all spacecraft system design, including definition of the spacecraft configuration, preparation of functional flow diagrams, and definition of detailed operational sequences
- In cooperation with subsystem engineering, monitor the design evolution and implementation of an integrated spacecraft system and its operational support equipment
- Participate in the definition of an integrated spacecraft system test plan

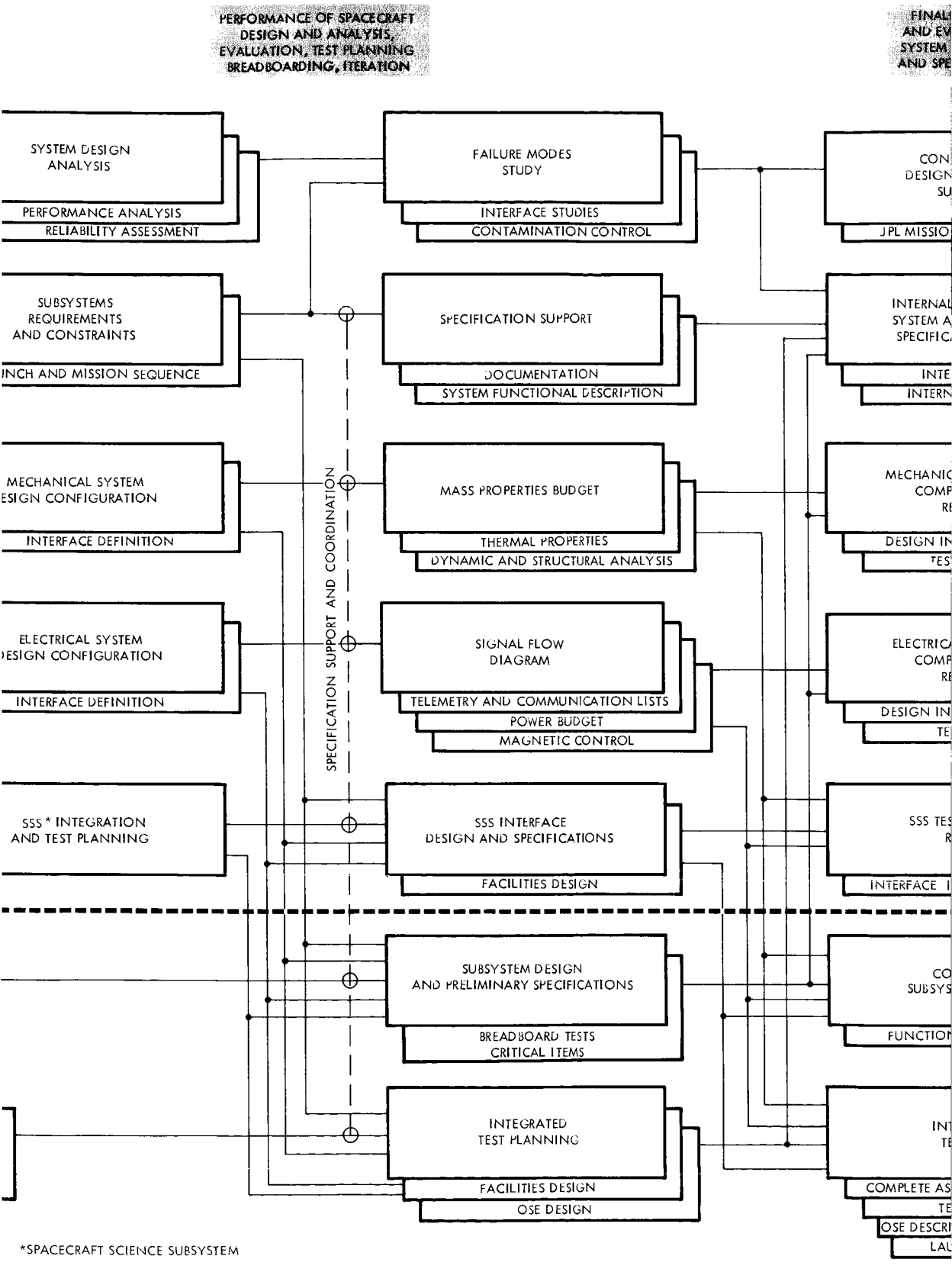
- Resolve design and development problem areas on the system level and perform tradeoffs, as necessary, to optimize system performance and reliability
- Formulate system and subsystem reliability models, coordinate and conduct reliability assessments, and assist the Reliability Department of Product Integrity Operations in spacecraft reliability implementation and monitoring
- Perform system and mission analyses in support of system and subsystem development, assembly and test, prelaunch and postlaunch operation
- Direct design, development, and documentation of all system electrical interfaces and implementation of the electrical interface control plan
- Direct design, development, and documentation of all system mechanical interfaces and implementation of the mechanical interface control plan
- Prepare the science payload interface design and monitor spacecraft science integration in cooperation with JPL
- Support JPL as directed, by performing mission engineering tasks, liaison, coordination with emphasis on intersystem interface areas, and evaluation of critical system performance data

2.4 Engineering Functions

The spacecraft engineering effort will be divided into five functional elements; the flow of work among these elements is shown in Figure 6. These elements correspond to the five departments of Spacecraft Engineering Operations. Four principal phases of spacecraft design are evident during Phase IB:

- Initial design and definition of specifications and requirements (Design audit No. 1)
- Spacecraft design and analysis, evaluation, test planning, and breadboarding of critical subassemblies. Iteration of these events is planned for this stage of spacecraft engineering (Design audit No. 2)
- Completion of system and subsystem design and evaluation, system description and specification packages subject to preliminary JPL review (Design audit No. 3)


JPL
MISSION SPECIFICATIONS
SYSTEM SPECIFICATIONS
INITIAL DESIGN AND SPECIFICATIONS/REQUIREMENTS DEFINITION
SPACECRAFT ENGINEERING OPERATIONS
SYSTEM ANALYSIS
SYSTEM REQUIREMENTS
MECHANICAL DESIGN INTEGRATION
ELECTRICAL DESIGN INTEGRATION
SPACECRAFT SCIENCE INTEGRATION
MISSION AND TRAJECTORY DATA
SYSTEM SPECIFICATIONS AND ADDENDA
INTERFACE SPECIFICATIONS AND ADDENDA
PRELIMINARY MECHANICAL SYSTEM LAYOUT
PRELIMINARY ELECTRICAL SYSTEM LAYOUT
ELECTRICAL AND MECHANICAL SUBSYSTEM OPERATIONS
ASSEMBLY AND TEST OPERATIONS
PRELIMINARY SUBSYSTEM DESIGN; SUBASSEMBLY SELECTION
PRELIMINARY ASSEMBLY AND TEST PLANS
FACILITIES PLANS


PERFORMANCE OF SPACECRAFT DESIGN AND ANALYSIS, EVALUATION, TEST PLANNING BREADBOARDING, ITERATION
SYSTEM DESIGN ANALYSIS
PERFORMANCE ANALYSIS
RELIABILITY ASSESSMENT
FAILURE MODES STUDY
INTERFACE STUDIES
CONTAMINATION CONTROL
SUBSYSTEMS REQUIREMENTS AND CONSTRAINTS
SPECIFICATION SUPPORT
DOCUMENTATION
SYSTEM FUNCTIONAL DESCRIPTION
MECHANICAL SYSTEM
INTERFACE DEFINITION
MASS PROPERTIES BUDGET
THERMAL PROPERTIES
DYNAMIC AND STRUCTURAL ANALYSIS
ELECTRICAL SYSTEM
INTERFACE DEFINITION
SIGNAL FLOW DIAGRAM
TELEMETRY AND COMMUNICATION LISTS
POWER BUDGET
MAGNETIC CONTROL
SPECIFICATION SUPPORT AND COORDINATION
SSS * INTEGRATION AND TEST PLANNING
SSS INTERFACE DESIGN AND SPECIFICATIONS
FACILITIES DESIGN
SUBSYSTEM DESIGN AND PRELIMINARY SPECIFICATIONS
BREADBOARD TESTS
CRITICAL ITEMS
INTEGRATED TEST PLANNING
FACILITIES DESIGN
OSE DESIGN
*SPACECRAFT SCIENCE SUBSYSTEM

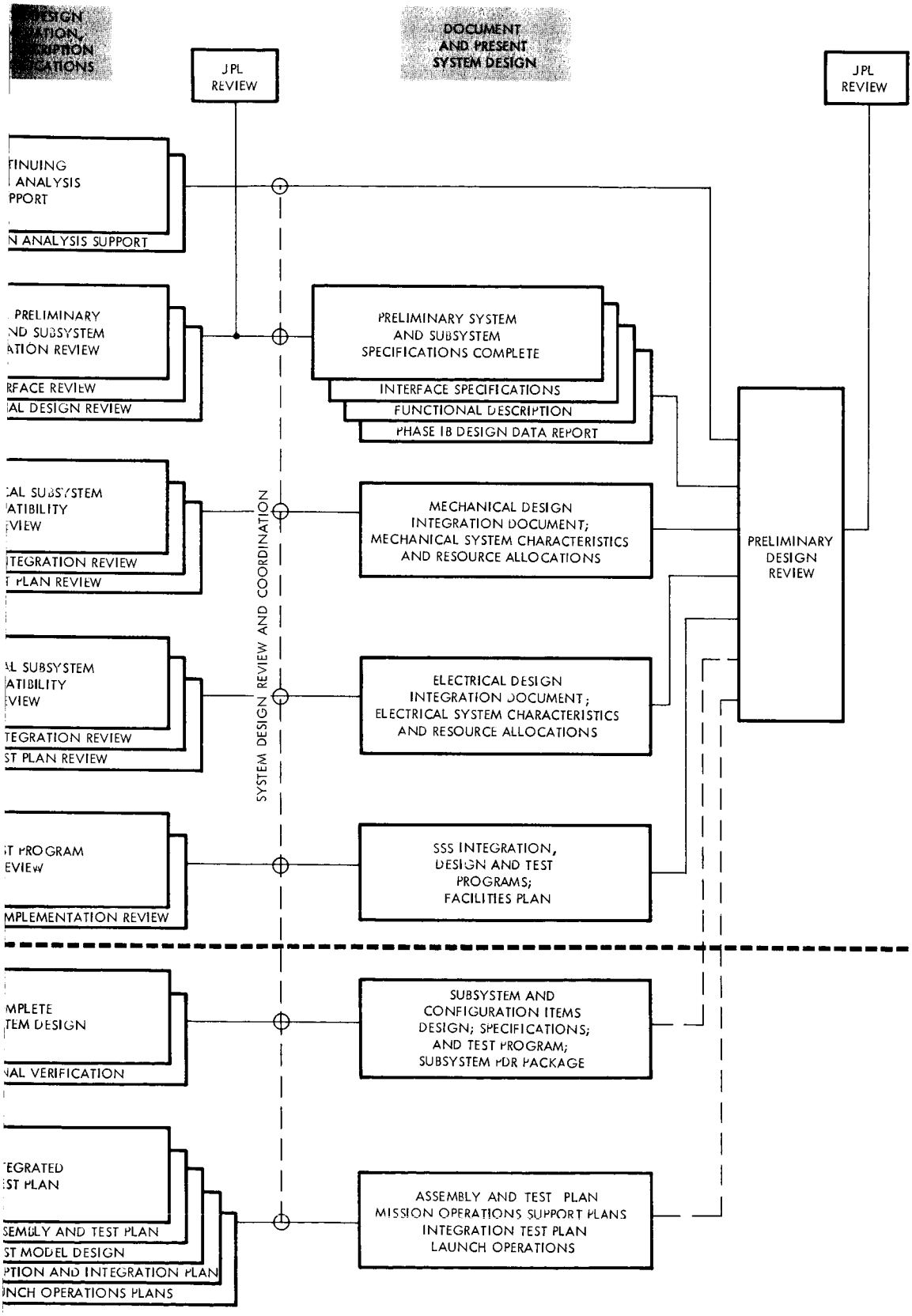


Figure 6. Summary Spacecraft Engineering Flow Diagram

- Documentation and submittal of system and subsystem design results and presentation at formal JPL Preliminary Design Review

Interactions of spacecraft engineering functions with other Voyager program activities within TRW, the NASA/JPL project organization, the major subcontractors, and the experimenters' organizations are shown in Figure 7 in terms of a matrix presentation.

2.5 System Analysis

2.5.1 Mission Analysis Studies

The System Analysis Department will conduct mission analyses, throughout Phases IB and II, to define mission and system characteristics in accordance with the Voyager mission objectives. The mission profile determined by JPL will be reviewed to derive basic criteria for the spacecraft system design and development. Subsequent performance studies conducted on a continuing basis will provide information on the ability of the spacecraft system to meet mission objectives. Tradeoff analyses will be required to resolve problem areas in system implementation, to determine preferred mission sequences, and to define alternate and preferred modes of operation. The following tasks will be included:

- Analysis of nominal and extreme transit trajectories and Mars orbits; definition of mission environmental parameters
- Establishment of targeting criteria and constraints
- Analysis of orbit determination accuracy
- Analysis of guidance accuracy requirements and capability
- Determination of maneuver sequences and maneuver accuracy
- Investigation of scientific coverage criteria and constraints
- Study of flight capsule trajectory and interface with flight spacecraft
- Analysis and tradoff studies involving alternate operation and maneuver sequences
- Determination of effect of variation in the mission profile from the nominal profile.

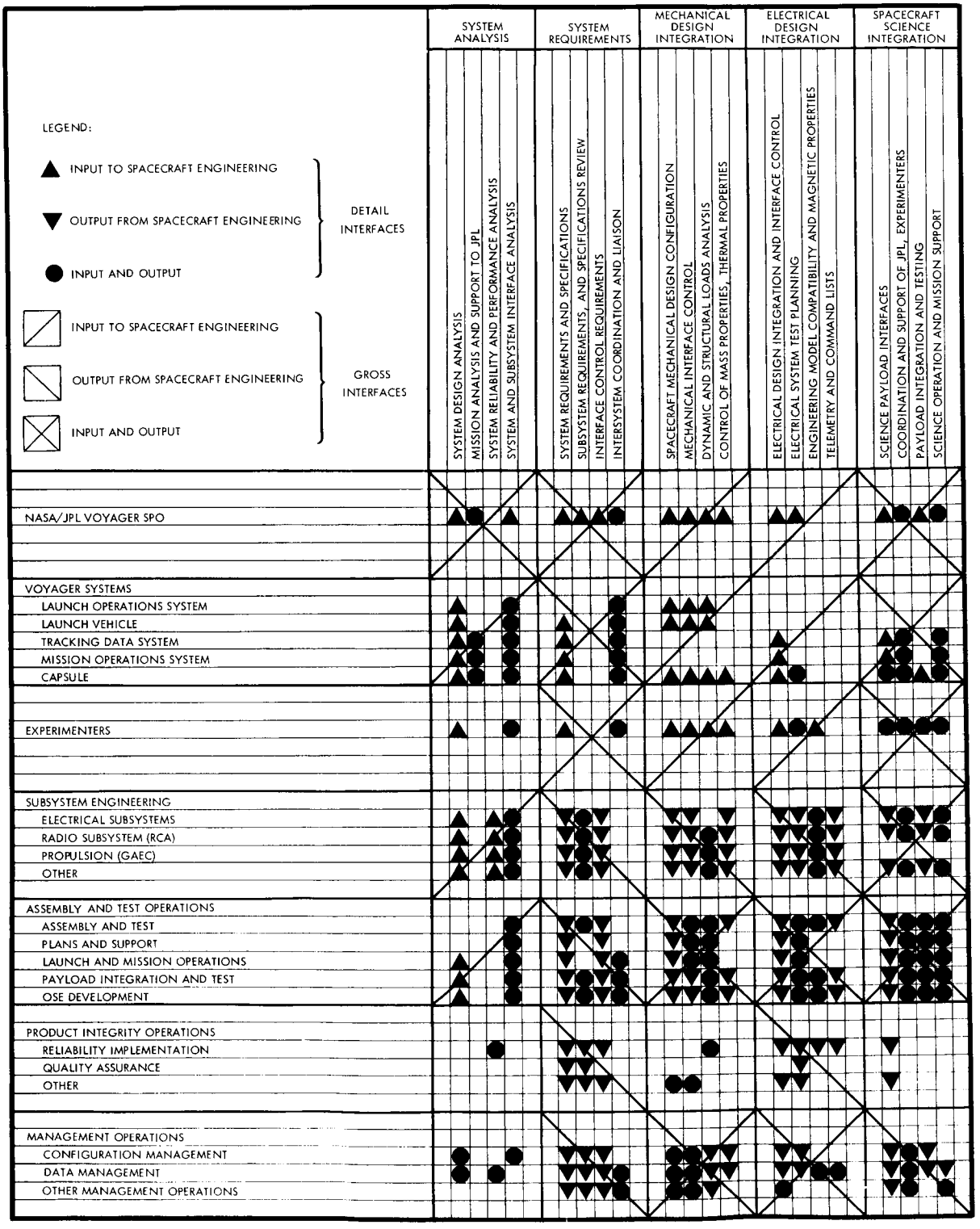


Figure 7. Spacecraft Engineering Work Breakdown and Functional Interfaces

2.5.2 Mission Engineering Support to JPL

The System Analysis Department will provide continuing support to JPL as directed, in the performance of mission engineering tasks. This effort is projected to extend through Phase II. This department will:

- Assist JPL in optimizing the mission profile for best use of launch window and for maximum scientific data return
- Study optimum deployment of science payload elements in mission sequence
- Study alternate modes of mission operation, science data coverage and communication
- Perform tradeoff studies as necessary to provide maximum quarantine assurance from system design and mission performance standpoint.

2.5.3 Intersystem Interface Analyses

The analysis of critical intersystem interfaces will require continued liaison with JPL and with various Voyager program elements, especially the deep space network. Interface analyses will include a continuous review of interactions and constraints imposed by spacecraft system implementation and integration of the spacecraft system with other program elements. These tasks will include the following:

- Determine DSIF loading throughout Voyager mission and during critical events
- Study capability of command and communication links for dealing with two spacecraft in transit and in Mars orbit
- Conduct communications link analyses to define the details of the DSIF spacecraft system interface
- Study operational and functional interfaces between planetary spacecraft, launch complex, and mission operations systems
- Study functional and hardware interfaces between planetary spacecraft and launch vehicle
- Study functional and hardware interfaces between flight spacecraft and flight capsule
- Study prelaunch interfaces of spacecraft and OSE with launch operations system.

2.5.4 System Design Analysis

In addition to the tasks identified below, systems analysis will be required in resolving unforeseen system implementation problems in support of subsystem engineering and assembly and test operations. Related tasks of system design analysis and review of the implementation of the system, the subsystems, and subsystem interfaces will be performed by the System Requirements Department (see Section 2.6). Tasks will include:

- Delineate the system design approach
- Participate in critical subsystem design decisions which affect over-all system performance and reliability
- Perform tradeoff studies and select preferred design solutions to minimize interface problems and to facilitate system integration
- Study allocation of priorities on spacecraft resources such as power, weight, volume, and thermal capacity
- Study adequacy of data link to monitor spacecraft performance; to detect and distinguish among failure modes; to provide information for ground control; to provide required science information. Perform detailed study of the data rate budget
- Study ability of spacecraft to meet Voyager mission requirements in terms of accuracy, operation sequence, payload deployment, capsule deployment, etc.
- Study ability of spacecraft to respond accurately to control from ground
- Study ability of spacecraft to maintain an environment suitable for successful operation of various subsystems
- Identify and analyze critical operating conditions and sequences and determine ability of spacecraft system to perform successfully during such conditions (e.g., maneuver command verification and solar eclipse survival).

2.5.5 System Reliability Studies

Throughout Phases IB and II the System Analysis Department will perform comprehensive reliability analyses, supporting Product Integrity Operations, in particular reliability studies pertaining to over-all system implementation and performance. These tasks will include the following:

- Establish reliability models and estimates (refinement of Phase IA data)
- Interpret mission reliability objectives, define system reliability criteria and update as required
- Perform reliability tradeoffs between subsystems during design evolution
- Participate in design reviews and take follow-up action as required
- Monitor failure reporting and corrective action on the system level.

2.5.6 Study of Failure Modes and Backup Procedures

A comprehensive analysis of failure modes at the system level will be conducted in support of the reliability analyses. These tasks will include:

- Study the ability of the flight spacecraft to accommodate failures while accomplishing the total mission or a partially successful mission
- Study and identify alternate modes of operation and backup modes applicable under various subsystem failures
- Investigate failure modes and failure detection procedures applicable in system prelaunch operations
- Investigate failure modes and failure detection procedures applicable during transit and Mars orbit
- Study backup modes of system operation and mission sequencing
- Determine communication requirements for failure detection and backup modes.
- Study alternate use of automatic and commanded mission sequences.

2.5.7 Contamination Control and Sterilization

The System Analysis Department and Product Integrity Operations will study the effect of contamination control and sterilization on system

design and system performance. This effort will complement requirements definition by the studies of sterilization procedures and will coordinate with Assembly and Test Operations. The following analyses will be performed.

- Establish contamination control requirements, policies and implementation plan
- Determine effect of contamination control and sterilization on system design
- Conduct tradeoff studies of system performance and reliability versus contamination control and sterilization objectives
- Monitor adherence to contamination control policies and planetary quarantine assurance.

2.6 System Requirements

The System Requirements Department will define and document system level requirements and specifications and will monitor adherence to these specifications throughout Phases IB and II. It will study the effect of such requirements on spacecraft design and operations. This activity is the focal point for the comprehensive review and application of JPL requirements and for the evaluation and feedback to JPL of the effect of such requirements on spacecraft design and operations. System design factors, such as spacecraft subsystem interfaces, test objectives, measurement and calibration guidelines, maintenance criteria, and spacecraft support system integration will be documented at the level of spacecraft specification and support system specification. The following tasks will be included:

- Interpret JPL mission objectives and system requirements in terms of spacecraft design objectives and spacecraft operation requirements
- Formulate and document system-level specifications for spacecraft and support system, and write addenda to JPL system specifications
- Provide coordination and liaison at the system level to achieve intersystem functional compatibility
- Provide continuous monitoring of system design and development for compliance with specifications and requirements

- Study the impact of specifications and specification changes on spacecraft design and operation
- Study and formulate specification change requirements
- Review and participate in writing engineering change proposals.

The specification tree for the 1971 Voyager spacecraft system illustrates the principal structure of spacecraft specifications on the system and subsystem levels. All addenda to JPL specifications, system, interface, environmental and requirement specifications are the subsystem and lower levels specifications generated by subsystem engineering in conformity with subsystem requirements and guidelines issued by this department. The department in turn will review the subsystem level specifications and assist in detailed formulation as required, through its specification support group. The objective is to adhere to a standardized specification format throughout the project.

With the support of the two Design Integration Departments the System Requirements Department will:

- Establish over-all system definition and layout
- Prepare system functional flow diagram
- Provide detailed functional description
- Maintain and update system configuration documents and flow diagrams
- Evaluate impact of subsystem engineering changes on over-all system configuration and functional flow.

The System Requirements Department will, on continuing basis, formalize and monitor interfaces between spacecraft subsystems. These tasks include:

- Define subsystem functional interface requirements
- Establish the interface control implementation plan

- Monitor interface design implementation
- Analyze constraints imposed on subsystem interfaces by critical mission sequences
- Maintain continuous interface monitoring effort and evaluate adherence to subsystem interface requirements and constraints.

The System Requirements Department will provide continuing assistance to JPL in assuring a fully-integrated spacecraft system design by performing the following tasks:

- Define intersystem interfaces and assist JPL in preparation of interface design specification and operation requirements
- Provide coordination of intersystem interface requirements and assist JPL in resolving interface conflicts
- Assist JPL in defining the environmental, mechanical, and electrical interfaces between the spacecraft system and the launch vehicle system
- Assist JPL in establishing a definition of the spacecraft system hardware and software interfaces with the mission operations system and the deep space network
- Determine operating constraints on DSN and on spacecraft.

The requirements for system maintenance and support equipment will be defined and formalized at the system level during Phase IB. Compliance with these requirements will be reviewed periodically during Phase II. Revisions of these requirements will be studied and coordinated with assembly and test operations. These tasks include:

- Define maintenance criteria and support system integration requirements
- Audit and review system maintenance and support system specifications
- Initiate action to resolve interface problems between spacecraft and support system.

The System Requirements Department will formally document and update design criteria, requirements, and specifications. These tasks include:

- Generate the over-all spacecraft spacecraft specification package on the system level
- Maintain complete, updated documents on all system-level requirements and specifications
- Maintain updated documents on system interface specifications affecting spacecraft and support system design and operation
- Review subsystem and subsystem-interface specifications and documentation and assure consistency and compatibility with system-level specifications and documents.

2.7 Spacecraft Science Integration

2.7.1 Interfaces and Design Integration

The Science Integration Department will be the focal point of spacecraft system design activity relating to the final integration of all science subsystems. This responsibility covers the definition and documentation of mechanical and electrical interfaces and generation of interface specifications, including the following tasks:

- Provide the principal point of contact for exchange of information relating to science payload integration between TRW and JPL
- Prepare specifications, procedures, drawings, and other data necessary to define and control the spacecraft science payload interface
- Assist JPL in the definition of the OSE/science support equipment interface
- Arrange for any design studies necessary to evaluate alternative interface design solutions or to solve problems which arise during compatibility testing
- Authorize implementation of the approved interface design
- Review spacecraft magnetic properties specification and test results and ensure adherence to magnetic control requirements imposed by science payload.

2.7.2 Assembly, Checkout and Test

Assembly of science payload system elements into the spacecraft system and test operations at various stages of spacecraft development will be conducted by Assembly and Test Operations but will remain

under the cognizance of the Science Integration Department. The department will exercise the following functions:

- Develop and specify test procedures in cooperation with JPL
- Assist JPL in calibration, trouble shooting, and repair of assemblies during the process of integration
- Maintain a science equipment log containing data on each science payload item
- Participate in reduction and analysis of test data and preparation of test reports, as directed
- Initiate and pursue follow-up action necessitated as a result of payload test data review.
- Initiate interface specification changes necessitated as a result of science payload integration and testing.

It is proposed to provide two identical spacecraft electrical simulators to be used for testing electrical compatibility of the spacecraft and science payload subassemblies. These simulators will be designed and developed as a part of the EOSE. One unit will be used for pre-assembly testing of science payload elements at JPL. ; The second unit will be located at TRW's assembly facility to permit convenient tests of science packages at times of removal from the complete spacecraft assembly, particularly for purposes of trouble shooting. The Science Integration Department will be responsible for defining the simulator design requirements and test procedures and for supervising its installation and use.

2.7.3 Operation Support

The Science Integration Department will provide support to JPL during all phases of science payload operation, as directed, and to the mission operations system during the operational life of the Voyager spacecraft. These responsibilities include:

- Train personnel for participation in preflight and flight operations involving the science payload and the interfacing spacecraft subsystems, such as DAE, command and sequencing and communications
- Assit JPL and MOS personnel in performing in-flight data analysis, trouble-shooting, etc. with emphasis on subsystem functions which interface with payload operations

- Assist JPL and MOS personnel in detecting emergency conditions and malfunctions, and in selecting backup modes for inflight science payload operations

2.7.4 Science Payload Integration Flow Chart

Figure 8 depicts the important events and operations during the development of the spacecraft system, the science payload, and relevant assembly and test functions. Dashed lines separate areas of TRW primary responsibility, for spacecraft development and integration, from areas of participation and support in science payload test and integration. These areas are delineated in accordance with provisions of the JPL Preliminary Mission Description and General System Specification (issued October 18, 1965) describing the spacecraft contractor's responsibility, and with subsequent directives received November 18, 1965.

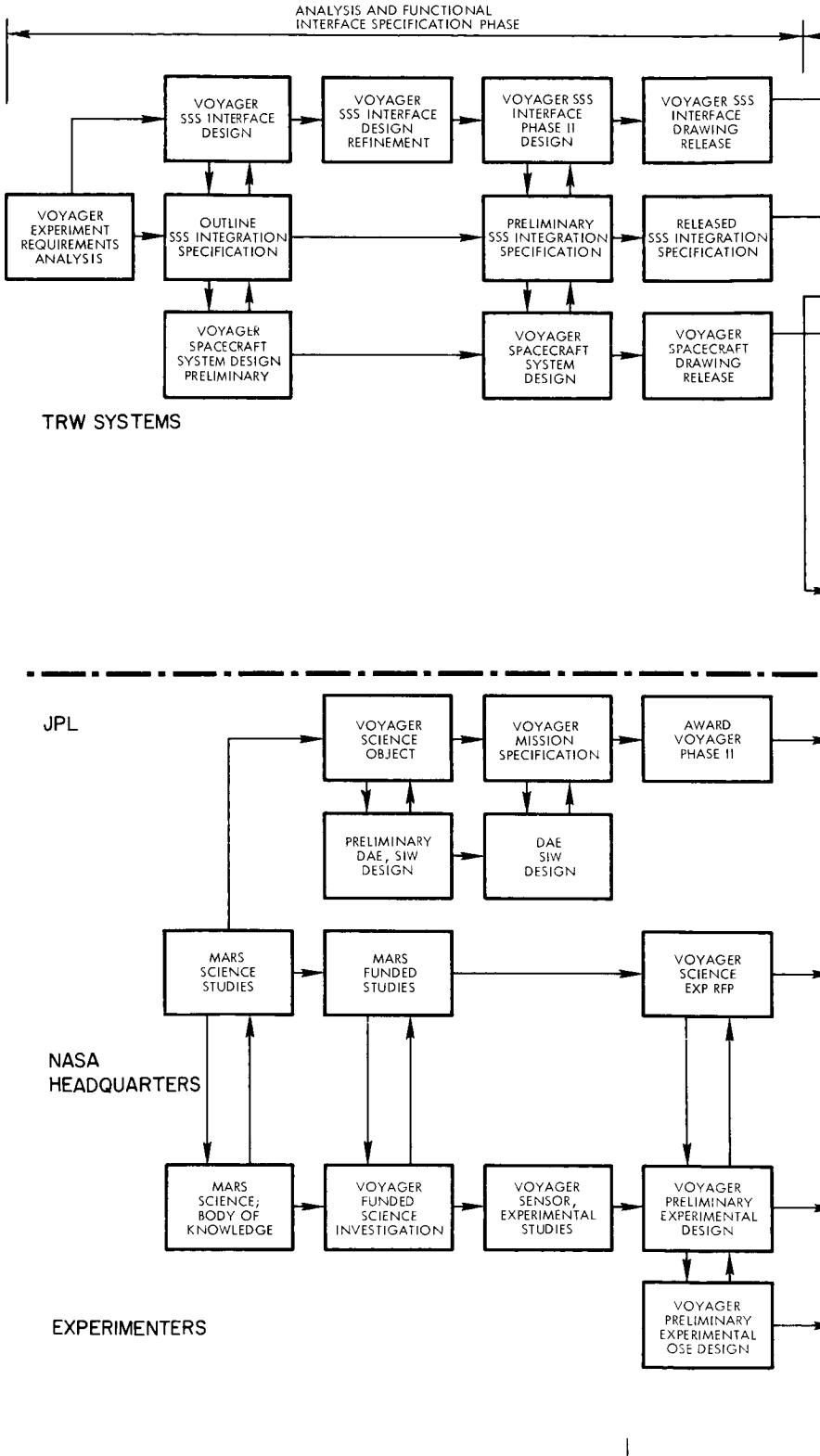
2.7.5 Science Payload Interface Requirements

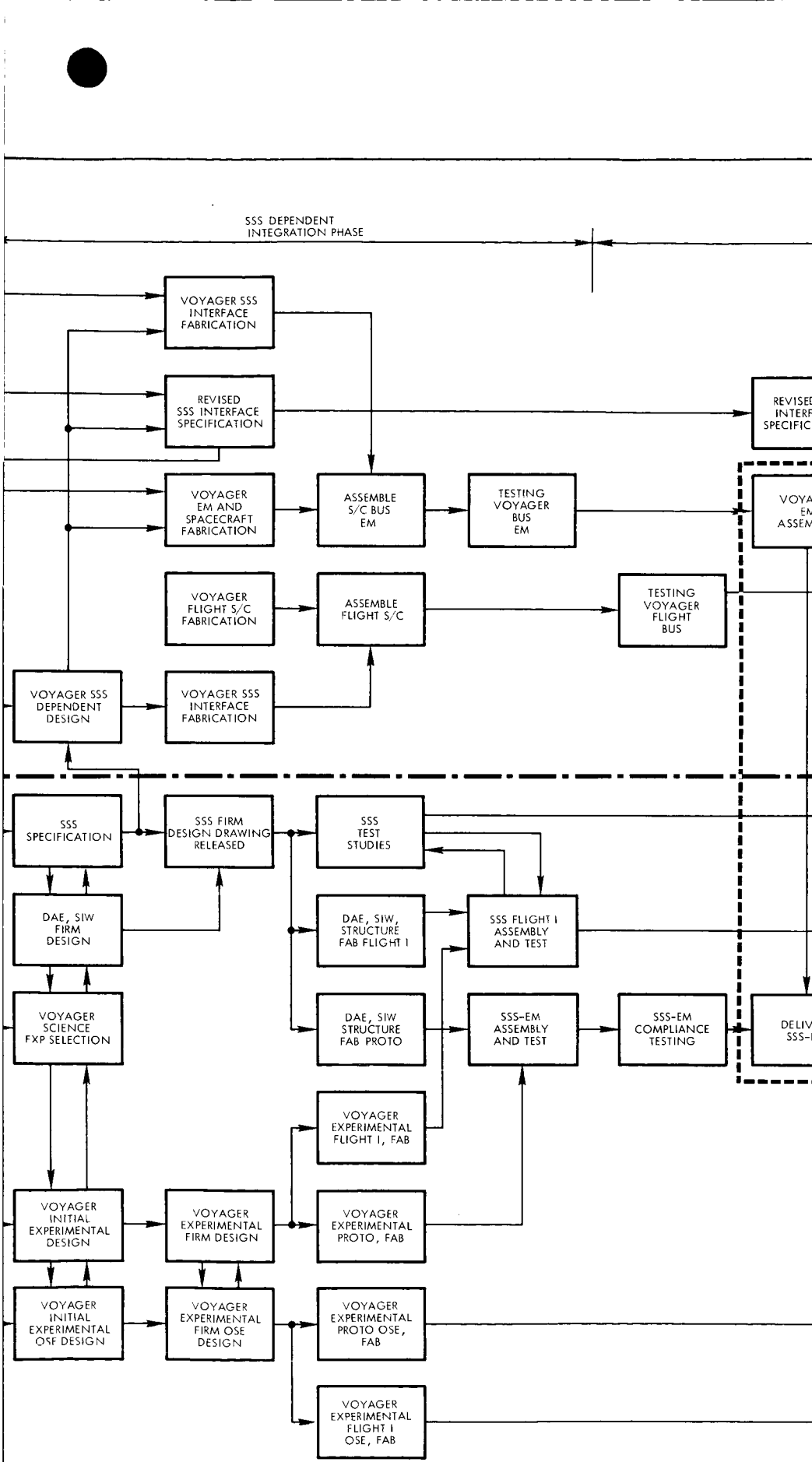
The matrix shown in Figure 7 provides a breakdown of possible interfaces existing between science payload elements (subdivided into body-mounted equipment, PSP equipment, and appendage-mounted equipment) interacting with functions and properties characterizing the spacecraft system design (see also Phase IA, Task A, Study Report, Vol. III, page 90).[1] The control and resolution of interface problems and the development of interface designs which conform to the above requirements will be the responsibility of the science payload integration department.

2.8 Mechanical Design Integration

The focal point for mechanical spacecraft design activities and for the coordination and integration of intersystem and subsystem mechanical interfaces will be the Mechanical Design Integration Department. This responsibility includes the generation and maintenance of mechanical interface specifications consistent with the system design requirements and restraints emanating from the System Requirements Department.

Specific responsibilities include analysis of system dynamic characteristics; development of external structural loads; mass properties control; analysis and specification of thermal control requirements and


ANALYSIS AND FUNCTIONAL
INTERFACE SPECIFICATION PHASE
VOYAGER SSS INTERFACE DESIGN
VOYAGER SSS INTERFACE DESIGN REFINEMENT
VOYAGER SSS INTERFACE PHASE II DESIGN
VOYAGER SSS INTERFACE DRAWING RELEASE
VOYAGER EXPERIMENT REQUIREMENTS ANALYSIS
OUTLINE SSS INTEGRATION SPECIFICATION
PRELIMINARY SSS INTEGRATION SPECIFICATION
RELEASED SSS INTEGRATION SPECIFICATION
VOYAGER SPACECRAFT SYSTEM DESIGN PRELIMINARY
VOYAGER SPACECRAFT SYSTEM DESIGN
VOYAGER SPACECRAFT DRAWING RELEASE
TRW SYSTEMS
JPL
VOYAGER SCIENCE OBJECT
VOYAGER MISSION SPECIFICATION
AWARD VOYAGER PHASE II
PRELIMINARY DAE, SIW DESIGN
DAE SIW DESIGN
MARS SCIENCE STUDIES
MARS FUNDED STUDIES
VOYAGER SCIENCE EXP RFP
NASA HEADQUARTERS
MARS SCIENCE; BODY OF KNOWLEDGE
VOYAGER FUNDED SCIENCE INVESTIGATION
VOYAGER SENSOR, EXPERIMENTAL STUDIES
VOYAGER PRELIMINARY EXPERIMENTAL DESIGN
VOYAGER PRELIMINARY EXPERIMENTAL OSE DESIGN
EXPERIMENTERS
PHASE IB
PHASE II


SSS DEPENDENT
INTEGRATION PHASE
VOYAGER SSS
INTERFACE
FABRICATION
REVISED
SSS INTERFACE
SPECIFICATION
VOYAGER
EM AND
SPACECRAFT
FABRICATION
ASSEMBLE
S/C BUS
EM
TESTING
VOYAGER
BUS
EM
VOYAGER
FLIGHT S/C
FABRICATION
ASSEMBLE
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COMPLIANCE
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FIRM DESIGN
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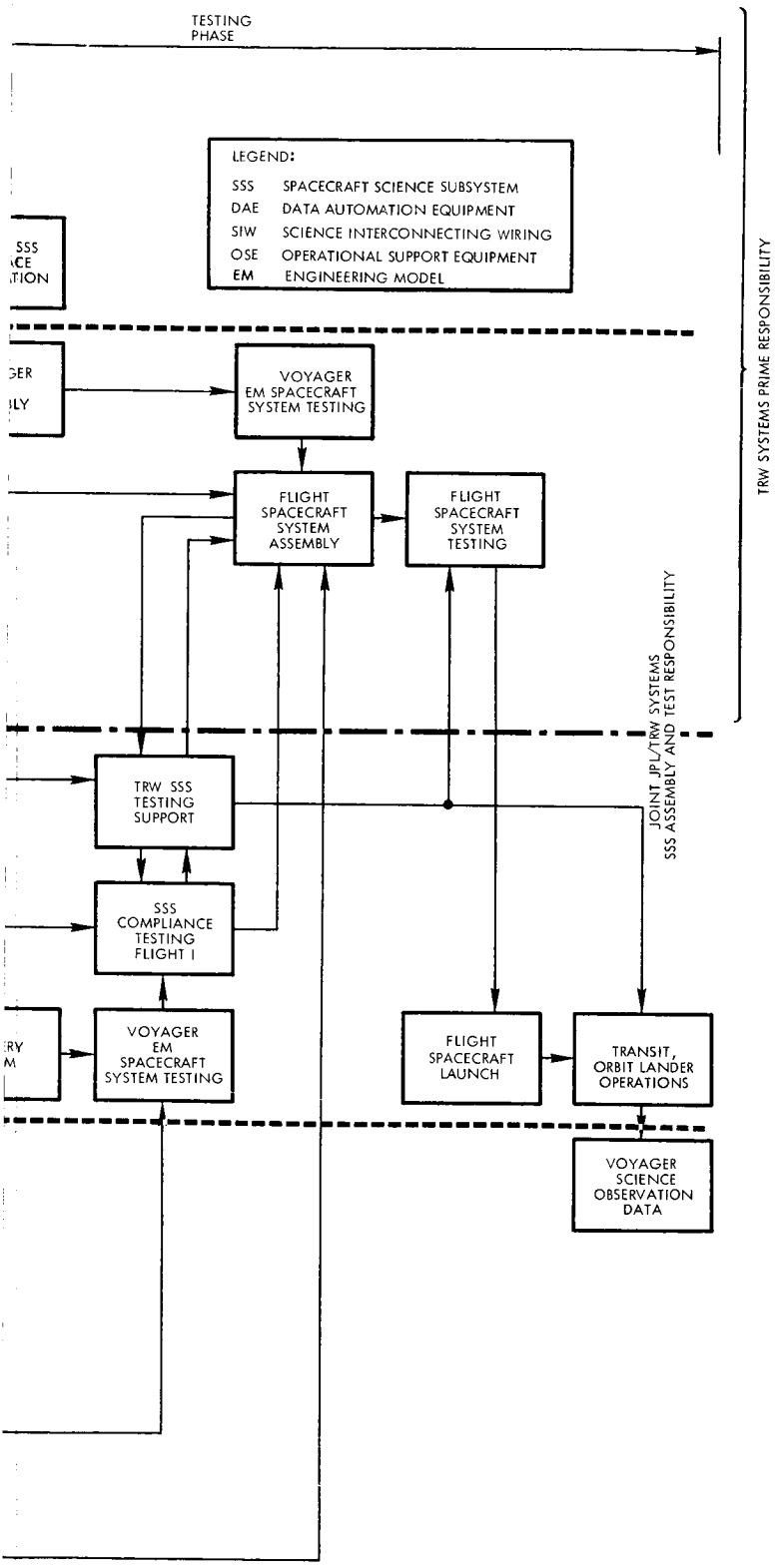


Figure 8. Spacecraft Science Design Integration

restraints; preparation, release, and maintenance of subsystem mechanical interface control drawings; assembly drawing preparation and related analysis of advanced or modified planetary vehicle configurations. In addition, the Mechanical Design Integration Department will have a key responsibility in the coordination and surveillance of subcontractor mechanical interfaces.

The mechanical design and integration responsibility exercised by the department will include the following tasks

- Direct the preparation of all configuration layouts and mechanical system design drawings
- Specify mechanical design constraints on subsystems
- Define and control mechanical and functional design compatibility of subsystems
- Review assembly compliance with compatibility and interchangeability requirements
- Prepare, release, and update spacecraft assembly drawings
- Define constraints on, and assure compatibility of appendage deployment
- Design and develop support structures for science payload subsystems
- Provide engineering support to mechanical assembly team
- Review and approve structural model test programs
- Participate in reliability assessment
- Participate in the selection of critical materials and processes
- Prepare mechanical interface specifications between planetary vehicles and launch vehicle structure
- Prepare interface specification between spacecraft and flight capsule
- Prepare, release, and update subsystem mechanical interface control drawings
- Define mechanical interface requirements and constraints for science payload assemblies

- Define design objectives for planetary spacecraft adapters
- Review design and implementation of separation joints and pyrotechnic subsystems
- Prepare conceptual design drawings for advanced missions and system modification, as needed
- Study effect of major subsystem changes on selected configuration
- Reappraise configuration and design relative to state-of-the-art advances
- Prepare interface specifications between aft equipment module structure and propulsion stage
- Coordinate and review with propulsion subcontractor the design and implementation of a compatible propulsion stage structural design
- Define weight allocations and other mass properties constraints upon propulsion stage and review their implementation
- Define the dynamic and structural interfaces and control implementation

Mass properties will be calculated, including weights, centers of gravity, moments and products of inertia, and mass distribution for the spacecraft and subassemblies. This task will include the following:

- Generate, maintain, and update a detailed weight budget for the spacecraft structure and all sub-assemblies
- Maintain mass properties records by using the TRW mass properties computer program; issue mass properties reports
- Hold weight review meetings as required, to exercise weight control over all subsystems and assemblies, to initiate weight reductions programs, and to perform weight tradeoffs, as required
- Participate in layout of assembly placement for optimization of weight, moments and products of inertia

- Determine actual weights of completed components and subassemblies and compare with the weight budget
- Obtain actual values of moments of inertia on the proof-test model and introduce into the mass properties computer program.

Similar studies will be conducted with respect to the thermal properties of the spacecraft. These will include:

- Analysis of the thermal environment during all flight phases and prior to launch
- Definition of thermal control requirements
- Definition of thermal interactions between subsystems
- Definition of thermal control interfaces with science payload and flight capsule
- Allocation of available power for heaters
- Study of effect of thermal emission from propulsion plumes
- Surveillance of thermal model test programs
- Review and approval of thermal model test programs.

Analysis of the dynamic behavior of the Voyager spacecraft during transportation, handling, launch, separation, midcourse corrections, Mars orbit injection, and capsule separation will be completed during Phase IB. Dynamic properties will be further analyzed and verified on the basis of test data during Phase II. The spacecraft responses to axial, lateral, and torsional dynamic loads during the boost phase will be analyzed including the responses of the spacecraft structure and subassemblies. The dynamic and structural loads analysis will include the following:

- Define maximum dynamic load environment expected during all phases of flight
- Analyze and define the dynamics of shroud separation, and separation of fore and aft planetary vehicles
- Analyze flight capsule and canister separation effects
- Analyze deployment of stowed appendages
- Study generation of internal loads

- Study the effect of support concepts and changes on system dynamic characteristics and response
- Define dynamic environment for each assembly
- Analyze g-loads on deployed appendages during orbit injection
- Define constraints imposed by separation sequence upon pyrotechnic subsystem
- Generate and maintain structural design criteria document for mechanical assemblies and subcontracted structure
- Study interaction between control system and appendages
- Provide surveillance of structural model test programs
- Investigate AHSE interfaces to ensure compliance with structural loading requirements

2.9 Electrical Design Integration

The Electrical Design Integration Department establishes requirements for all electrical subsystems and their integration into the spacecraft. The department has prime responsibility for electrical interfaces and the compatibility with the EOSE. The department establishes electrical power budgets and load criteria. In addition, it controls compatibility of power consuming subsystems with the power source, determines the electromagnetic environment of the spacecraft and electromagnetic subsystem compatibility. The department also controls magnetic properties of the over-all spacecraft design and its subsystems. These responsibilities entail:

- Establishment of electrical interface criteria and constraints among subsystems
- Design of the electrical cabling schematic and overall electrical system block diagrams internal to the spacecraft system and across intersystem interfaces. Makes use of the spacecraft configuration model as a design tool.
- Definition of system level test points in coordination with assembly and test operations

- Review of electrical integration test program for adequacy to establish adherence to interface specifications
- Definition and control of intersystem electrical interfaces: science payload, flight capsule, launch vehicle, launch complex, and planetary vehicle adapter
- Analysis of spacecraft electrical subsystem interactions
- Definition and maintenance of spacecraft power requirements
- Analysis of detailed operating sequences during the mission to determine detailed power profile
- Compilation, maintaining, and updating of power budgets
- Study of available power margins, contingencies, and backup power distribution to primary consumers

The department will prepare a specification defining the electromagnetic interference limits on electrical subsystems, will complete the EMC control plan, and will establish the over-all electromagnetic environment during prelaunch and flight operations. Compatibility of the electrical subsystems with this environment will be analyzed and monitored. This task will also include a study of signal levels and their compatibility with subsystem circuit design including circuit loading, shielding, and grounding. Signal coupling and interference will be analyzed and controlled in connection with integration tests. Test results will be reviewed and action for control of electromagnetic interference initiated as required. An outline of the EMC plan, to be detailed during Phase IB, is given in Appendix E.

The Electrical Design Integration Department will prepare detailed telemetry and command lists covering all phases of spacecraft system operation. The spacecraft instrumentation requirements will be defined to assure an adequate program of engineering measurements for telemetry to the ground and for detection and diagnosis of system malfunctions. Priority levels of engineering telemetry will be established to cover performance evaluation, diagnostic measurements and mission event verification.

Responsibility for the magnetic control program by this department will include the following tasks:

- Definition of magnetic design and control objectives
- Preparation of guidelines for magnetic properties of materials and parts, and approval authority as to acceptable parts and materials. This program imposes requirements on reliability, quality assurance, and procurement, and on the spacecraft design approach
- Preparation and maintenance of a magnetic control model
- Establishment of subsystem magnetic testing requirements and evaluation of test results for compliance with magnetic control policies.
- Establishment of spacecraft magnetic testing requirements and evaluation of test results to assure the meeting of magnetic control objectives
- Tradeoff studies as required to meet magnetic properties specifications
- Coordination with Science Integration Department to assure compliance with science payload magnetic constraints

A preliminary outline of the Magnetics Control Plan is presented in Appendix C.

2.10 Spacecraft Reliability Engineering

Spacecraft Engineering Operations will evaluate reliability models, estimates, and tradeoffs during Phases IB and II. Assumptions made for reliability of subsystems will be analyzed to establish relative subsystem criticality to the mission and to overall spacecraft system reliability. The function of spacecraft reliability engineering will include the responsibility for assessing reliability effects of subsystem interactions and interface design. Results of reliability analyses will establish design tradeoff criteria and constraints, in terms of weight, magnetic properties, contamination control, electromagnetic interference, circuit tolerance control, environmental control functions, maintainability, and testability.

Interactions of Reliability Engineering with the Reliability Department of Product Integrity Operations, with Electrical and Mechanical Subsystem Operations, and Assembly and Test Operations require close coordination among these groups.

The following specific tasks will be performed:

- Detailed definition of mission phases and events and the corresponding environment profiles
- System and subsystem reliability model formulation
- Study of system and subsystem reliability analysis techniques
- Reliability standards and policies for parts and materials
- Reliability standards and policies for parts derating and use
- Reliability tradeoff studies and status reporting
- Analysis of failure modes, effects, and criticality
- Reliability test planning
- Review of failure reports and test logs
- Corrective action planning and followup
- Coordination with launch site activities and launch operations to monitor system reliability status
- Coordination and liaison with JPL, as required
- Analysis of reliability implications and constraints of OSE design and OSE/spacecraft interfaces

3. SUBSYSTEM DEVELOPMENT

3.1 Introduction

This section discusses the spacecraft subsystem development efforts beginning in Phase IB and continuing to launch in Phase II. In general, the subsystems are discussed individually, the exceptions being the inclusion of the spacecraft-shroud adapter development in the structural mechanical subsystem discussion and the relay link antenna development in the radio subsystem discussion.

In the main, each subsystem discussion presents a summary discussion keyed to a development flow chart showing the sequence of major activities during Phase IB and II. A schedule is included in each discussion which relates the items in the flow charts to time. Key milestones are indicated.

During the progress of the development program certain key milestones were imposed on each of the subsystems (and OSE). The Phase IB milestones and their constraints were:

- Second Design Audit (October 1966)

  Preparation of preliminary subsystem specifications and Part I, CEI specifications for both flight hardware and OSE.

- PDR (January 15, 1967)

  Preparation of subsystem specification and final Part I, CEI specifications (partial) for both flight hardware and OSE and submittal of Phase IB design data reports.

Additional design audits are scheduled for the sixth and 26th week of Phase IB to provide interim project management review.

The Phase II milestones and their constraints were:

- Intermediate Design Review (as shown on subsystem schedules)

  Preparation of updated Subsystem Specifications and Part I, CEI specifications, complete for both flight equipment and OSE.

  Release of engineering model 2 and 3 drawing to manufacturing

- Critical Design Review (as shown on subsystem schedules)

  Preparation of updated subsystem and completed Part I, CEI specifications and preliminary Part 2, CEI specifications for flight equipment and OSE.

  Completion of partial type approval testing on either EM 1 or 2

  Release of drawings for type approval, PTM and reliability demonstration models.

  Release of OSE drawings for all units

- Completion of subsystem type approval tests (as shown on subsystem schedules)

  Release of updated drawing for flight spacecraft equipment (No's. 1, 2, 3, and spares)

- FACI (November 1970)

  Final Part II CEI specifications

  Ship first flight spacecraft to ETR

Another general ground rule affecting type approval and flight acceptance testing which bears emphasis here is that all assemblies (black

box) are flight acceptance tested per the requirements of Section III of this volume prior to being used for either subsystem type approval testing or for reliability demonstration testing. The subsystem type approval testing cycle then includes assembly testing through all environments (including magnetics and EMC) excepting thermal-vacuum. The thermal-vacuum testing is conducted on a configuration closely simulating the in-flight arrangement. Thus, prior to thermal-vacuum testing each subsystem is assembled and functionally checked as a subsystem, then exposed to the type approval thermal-vacuum test sequence and tested as required. At the completion of this cycle the subsystem is again subjected to a complete functional test over and above its specified operating limits of temperature, power and transient levels.

Flight acceptance testing includes environmental tests of vibration, thermal-vacuum, magnetics and EMC as an assembly. The subsystem is then integrated and tested as a subsystem over the specified operating limits of temperature, power and transient levels.

3.2 Structural and Mechanical Subsystem (Including Planetary Vehicle Adapter)

3.2.1 Description

The structural and mechanical subsystem consists of the flight capsule interstage structure, equipment mounting panels, aft equipment module, flight capsule emergency separation system, LEMDE propulsion module structure, and planetary vehicle adapter structure. The major components of the structural and mechanical subsystem (including adapter) and the respective contractors are illustrated on Figure 9.

The aft equipment module structure will have the following interfaces: high-gain antenna support, medium-gain antenna support, low-gain antenna support, planetary scan platform support, reaction control system pneumatics, science packages and antennas, solar array structure, the LEMDE propulsion module structure, and AHSE.

The LEMDE propulsion module structure will have interfaces with the following: LEMDE propulsion engine system, equipment mounting panels, capsule interstage structure, planetary vehicle adapter structure and the AHSE.

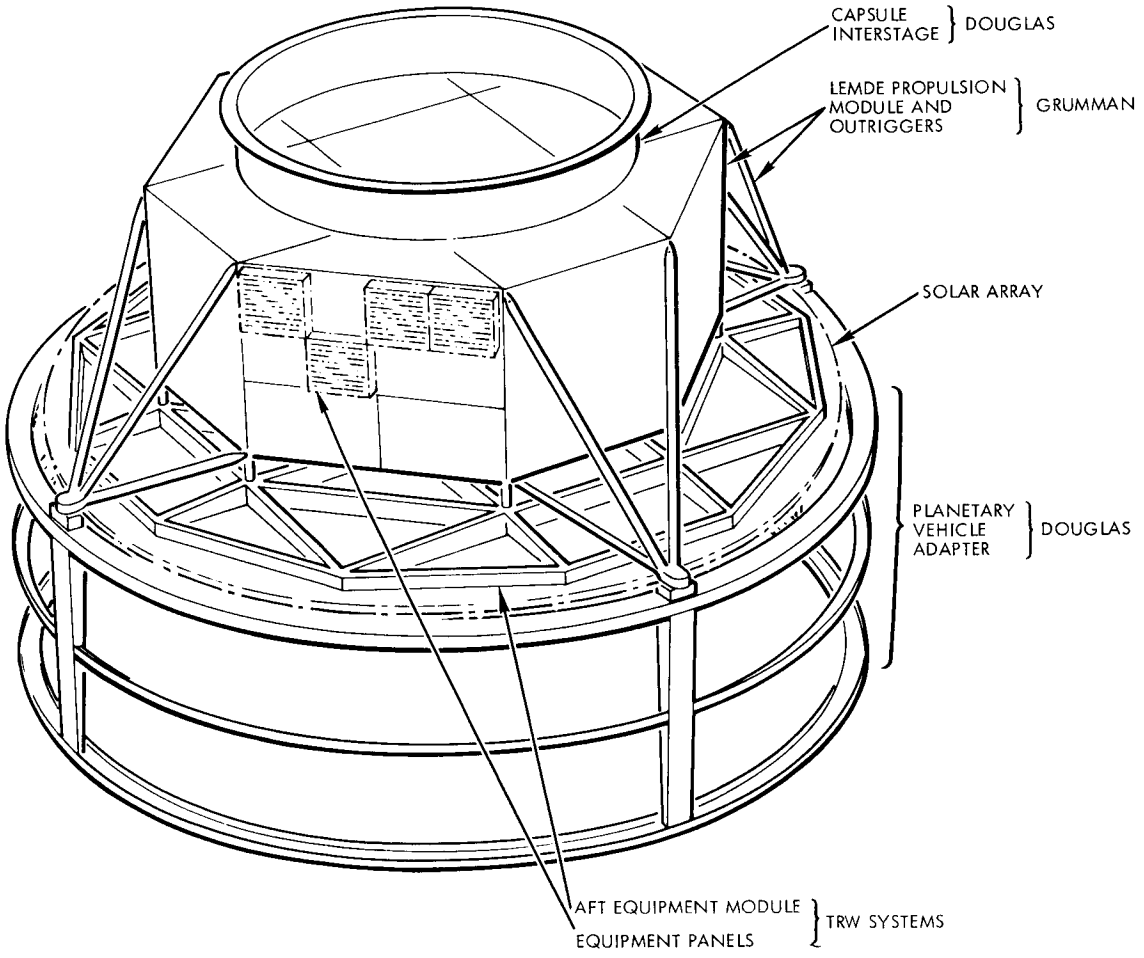


Figure 9. Structural and Mechanical Subsystem (Including Planetary Adapter)

The planetary vehicle adapter interfaces with the spacecraft at four hard points, known as separation joints. Each of the four hard points will have a tension bolt, a spring release, and a dual ordnance actuator. The bolt is to be held by a split nut which can be released by the action of either one of two pyrotechnic nuts. Shear will be taken by socket action at the interface.

The flight capsule interfaces with the capsule interstage structure and associated AHSE, and the planetary vehicle adapter structure interfaces with the flight nose fairing.

3.2.2 Summary

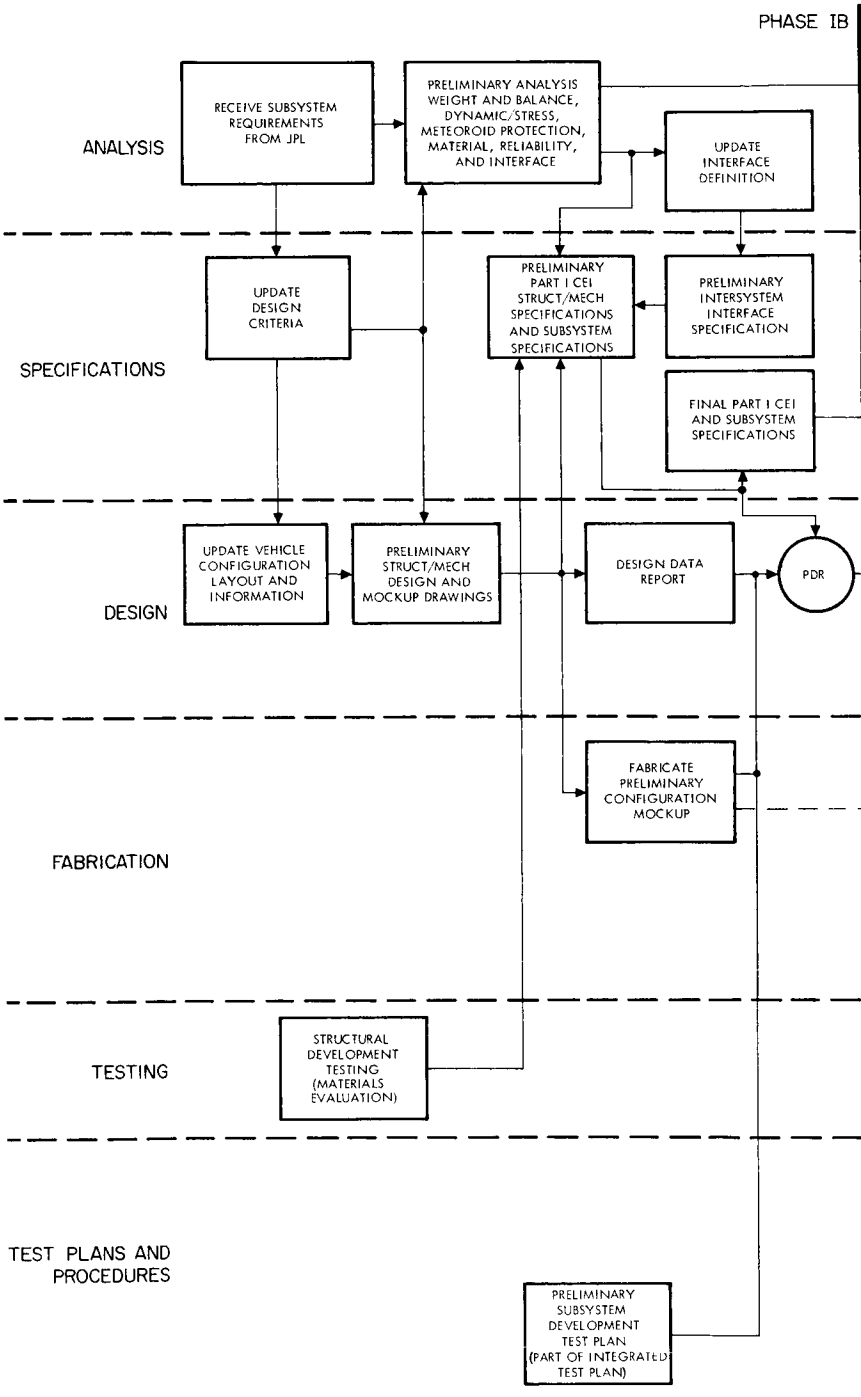
The scope and responsibility of the subsystem work includes: design; strength, dynamic, release separation, mass properties, and reliability analyses; preparation of specifications; testing, manufacturing, and liaison. The subsystem design represents no advance in the state of the art, therefore no new research programs are anticipated. A conservative design approach will be pursued with emphasis on minimum weight and cost, and simplicity. Structural and functional interfaces noted above will be developed.

The design and development tasks for the structural/mechanical subsystem (including planetary vehicle adapter) are shown in Figure 10 and on the schedule in Figure 11.

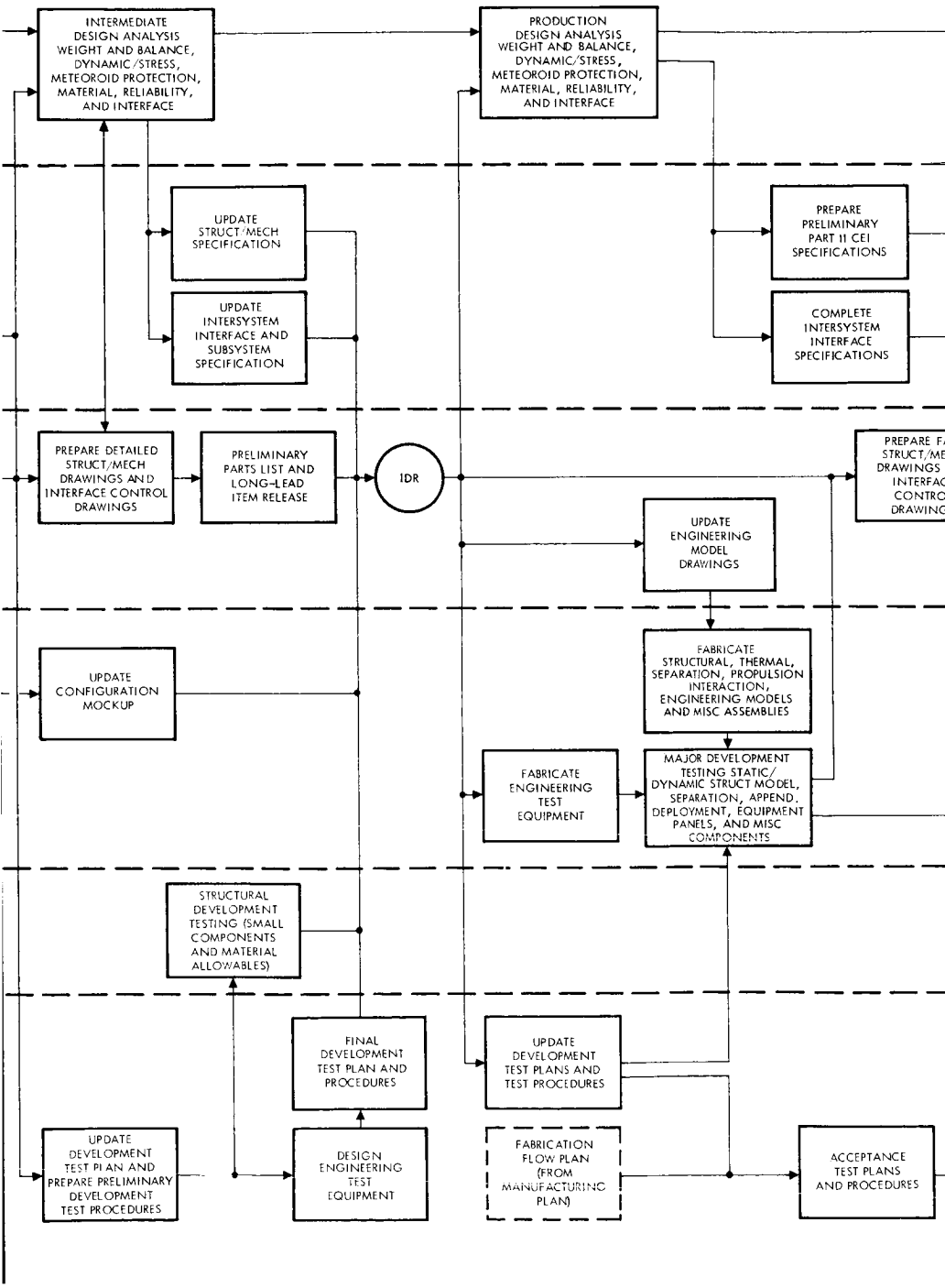
The tasks during Phase IB include the updating of the subsystems requirements, and preliminary design layouts and analyses leading up to the issuance, in October 1966, of preliminary subsystem and Part 1, CEI specifications. The design effort will then proceed to detailed layouts conforming to analysis of strength, rigidity, deflection, and weight. In addition, the structural interfaces will be defined. Prior to PDR, the subsystem specification, final Part I CEI specifications (partial), and the development test plans will be prepared. This material will be reviewed during the PDR and updated as required.

The effort prior to IDR will consist of detailed design and analyses, preparation of detailed drawings for the structural test model, interface control drawings and preparation of the preliminary list of materials. In addition, the effort will include development testing, preparation of final test plans and procedures, completion of Part I CEI specifications and updating interface specifications.

Following the IDR, the submitted items will be updated, and the engineering model structural drawings will be released for fabrication. The design effort will proceed into production (flight spacecraft) drawings. Simultaneously, production design analysis, preparation of preliminary Part II CEI specifications, and updating of interface control drawings and specifications will occur. Flight acceptance test plans and


PHASE IB
ANALYSIS
SPECIFICATIONS
DESIGN
FABRICATION
TESTING
TEST PLANS AND PROCEDURES
RECEIVE SUBSYSTEM REQUIREMENTS FROM JPL
PRELIMINARY ANALYSIS WEIGHT AND BALANCE, DYNAMIC/STRESS, METEOROID PROTECTION, MATERIAL, RELIABILITY, AND INTERFACE
UPDATE INTERFACE DEFINITION
UPDATE DESIGN CRITERIA
PRELIMINARY PART I CEI STRUCT/MECH SPECIFICATIONS AND SUBSYSTEM SPECIFICATIONS
PRELIMINARY INTERSYSTEM INTERFACE SPECIFICATION
FINAL PART I CEI AND SUBSYSTEM SPECIFICATIONS
UPDATE VEHICLE CONFIGURATION LAYOUT AND INFORMATION
PRELIMINARY STRUCT/MECH DESIGN AND MOCKUP DRAWINGS
DESIGN DATA REPORT
PDR
FABRICATE PRELIMINARY CONFIGURATION MOCKUP
STRUCTURAL DEVELOPMENT TESTING (MATERIALS EVALUATION)
PRELIMINARY SUBSYSTEM DEVELOPMENT TEST PLAN (PART OF INTEGRATED TEST PLAN)

PHASE II


INTERMEDIATE DESIGN ANALYSIS WEIGHT AND BALANCE, DYNAMIC/STRESS, METEOROID PROTECTION, MATERIAL, RELIABILITY, AND INTERFACE
PRODUCTION DESIGN ANALYSIS WEIGHT AND BALANCE, DYNAMIC/STRESS, METEOROID PROTECTION, MATERIAL, RELIABILITY, AND INTERFACE
UPDATE STRUCT/MECH SPECIFICATION
UPDATE INTERSYSTEM INTERFACE AND SUBSYSTEM SPECIFICATION
PREPARE PRELIMINARY PART II CEI SPECIFICATIONS
COMPLETE INTERSYSTEM INTERFACE SPECIFICATIONS
PREPARE DETAILED STRUCT/MECH DRAWINGS AND INTERFACE CONTROL DRAWINGS
PRELIMINARY PARTS LIST AND LONG-LEAD ITEM RELEASE
IDR
PREPARE F STRUCT/ME DRAWINGS INTERFAC CONTRO DRAWING
UPDATE ENGINEERING MODEL DRAWINGS
UPDATE CONFIGURATION MOCKUP
FABRICATE STRUCTURAL, THERMAL, SEPARATION, PROPULSION INTERACTION, ENGINEERING MODELS AND MISC ASSEMBLIES
FABRICATE ENGINEERING TEST EQUIPMENT
MAJOR DEVELOPMENT TESTING STATIC/ DYNAMIC STRUCT MODEL, SEPARATION, APPEND. DEPLOYMENT, EQUIPMENT PANELS, AND MISC COMPONENTS
STRUCTURAL DEVELOPMENT TESTING (SMALL COMPONENTS AND MATERIAL ALLOWABLES)
FINAL DEVELOPMENT TEST PLAN AND PROCEDURES
UPDATE DEVELOPMENT TEST PLANS AND TEST PROCEDURES
UPDATE DEVELOPMENT TEST PLAN AND PREPARE PRELIMINARY DEVELOPMENT TEST PROCEDURES
DESIGN ENGINEERING TEST EQUIPMENT
FABRICATION FLOW PLAN (FROM MANUFACTURING PLAN)
ACCEPTANCE TEST PLANS AND PROCEDURES

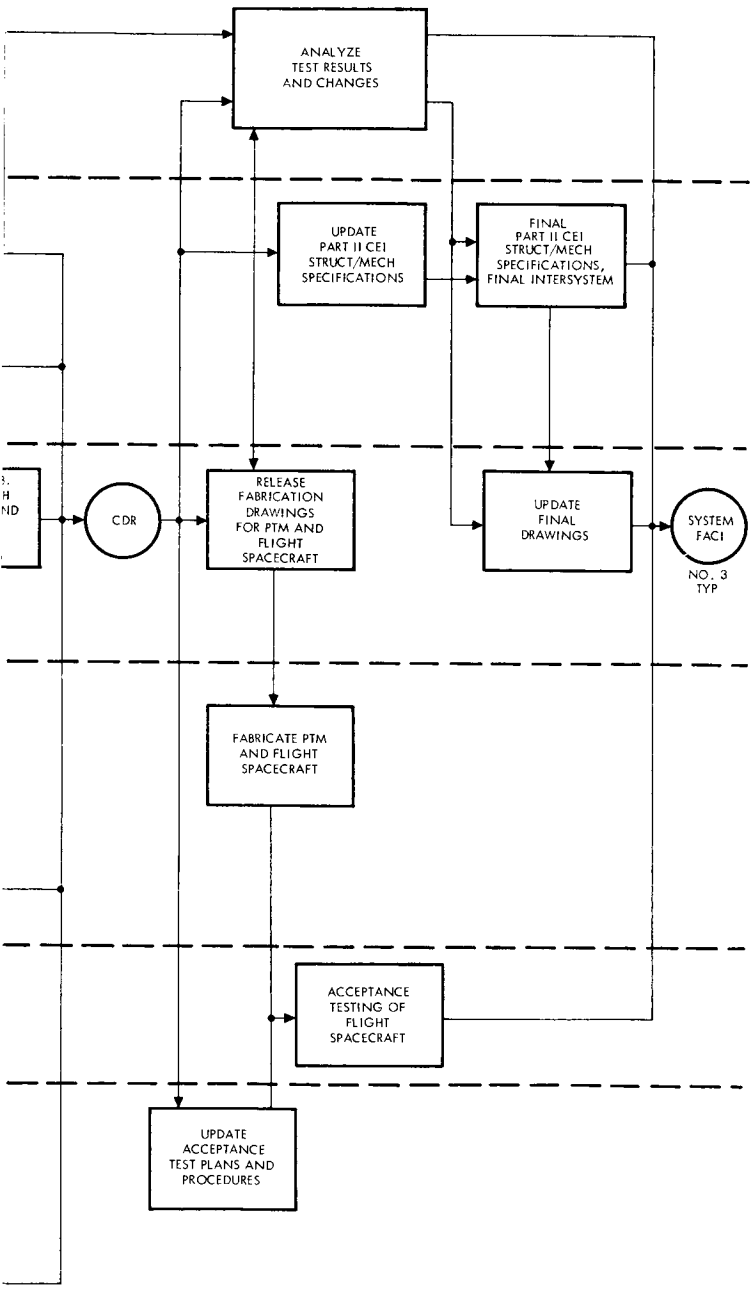


Figure 10. Design and Development Tasks for the Structural-Mechanical Subsystem

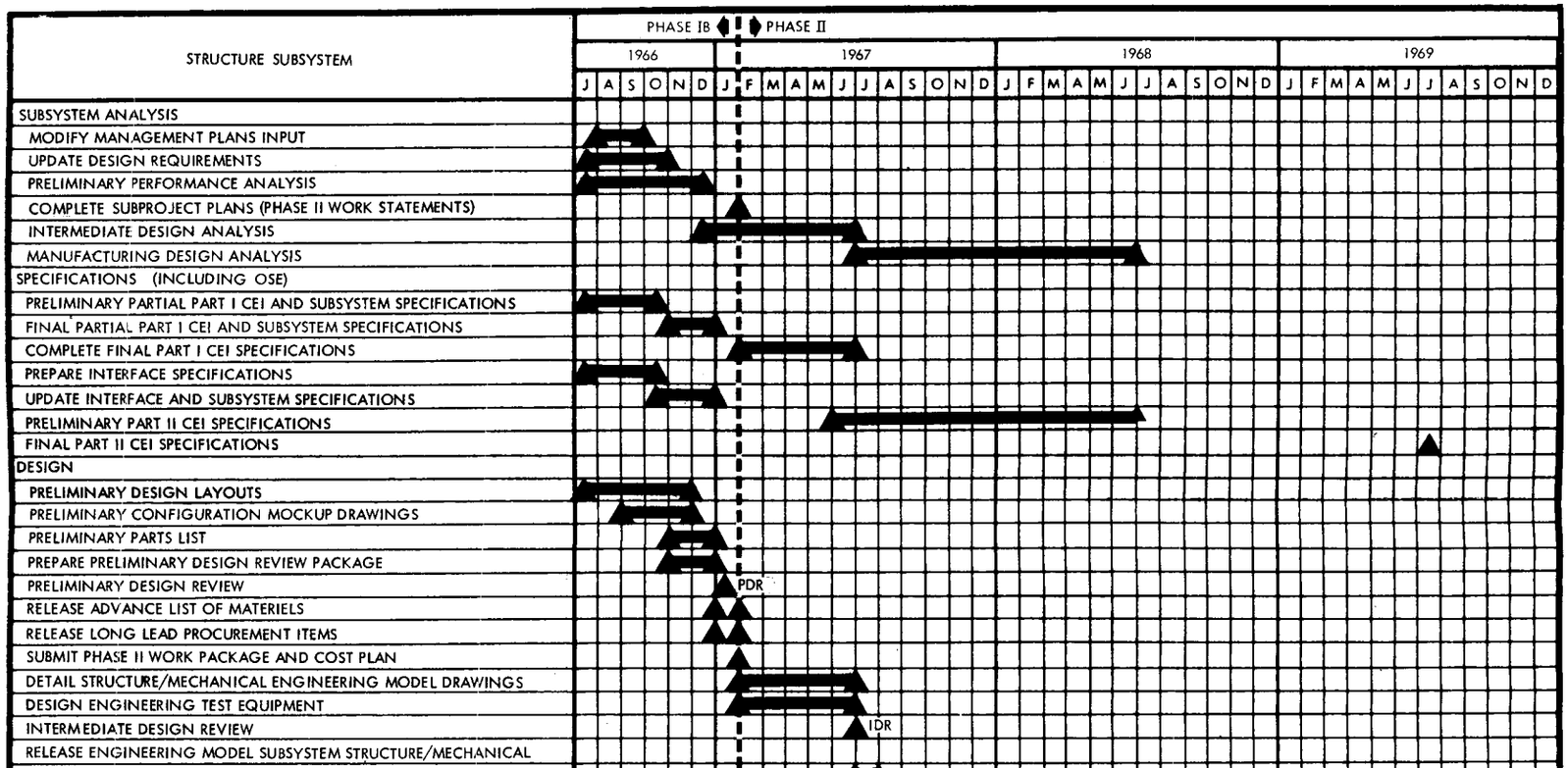

STRUCTURE SUBSYSTEM
PHASE IB
PHASE II
1966
1967
1968
1969
J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D
SUBSYSTEM ANALYSIS
MODIFY MANAGEMENT PLANS INPUT
UPDATE DESIGN REQUIREMENTS
PRELIMINARY PERFORMANCE ANALYSIS
COMPLETE SUBPROJECT PLANS (PHASE II WORK STATEMENTS)
INTERMEDIATE DESIGN ANALYSIS
MANUFACTURING DESIGN ANALYSIS
SPECIFICATIONS (INCLUDING OSE)
PRELIMINARY PARTIAL PART I CEI AND SUBSYSTEM SPECIFICATIONS
FINAL PARTIAL PART I CEI AND SUBSYSTEM SPECIFICATIONS
COMPLETE FINAL PART I CEI SPECIFICATIONS
PREPARE INTERFACE SPECIFICATIONS
UPDATE INTERFACE AND SUBSYSTEM SPECIFICATIONS
PRELIMINARY PART II CEI SPECIFICATIONS
FINAL PART II CEI SPECIFICATIONS
DESIGN
PRELIMINARY DESIGN LAYOUTS
PRELIMINARY CONFIGURATION MOCKUP DRAWINGS
PRELIMINARY PARTS LIST
PREPARE PRELIMINARY DESIGN REVIEW PACKAGE
PRELIMINARY DESIGN REVIEW
PDR
RELEASE ADVANCE LIST OF MATERIELS
RELEASE LONG LEAD PROCUREMENT ITEMS
SUBMIT PHASE II WORK PACKAGE AND COST PLAN
DETAIL STRUCTURE/MECHANICAL ENGINEERING MODEL DRAWINGS
DESIGN ENGINEERING TEST EQUIPMENT
INTERMEDIATE DESIGN REVIEW
IDR
RELEASE ENGINEERING MODEL SUBSYSTEM STRUCTURE/MECHANICAL

Figure 11. Schedule for Design and Development of Structure

2

procedures will be prepared, and major development tests will be conducted. The completion of the CDR is constrained by the completion of the foregoing--especially the major structural/mechanical subsystem (including adapter) tests. These tests include the structural model dynamic and static tests and the various separation tests (capsule emergency separation, spacecraft/launch vehicle separation and preliminary nose fairing separation tests).

After completing the CDR, submitted plans, drawings, procedures, and specifications will be updated and released. This includes the release of flight spacecraft structural drawings.

The effort during the next phase (CDR to FACI) consists of continuous subsystem support to fabrication, assembly, and testing of the proof test model (PTM) and flight spacecraft. Analysis, drawings, specifications and procedures will be continually updated as required in preparation for final submittal to the FACI.

3.2.3 Design and Development

This section details the tasks that will be performed in designing the structure. This includes analytical tasks, basic design layout, and detailing.

a. Analysis

The analytical tasks to be performed include an analysis of the structural design load criteria developed during the spacecraft systems analysis effort to define the specific critical design loading and rigidity requirements for each element of the structure; a strength analysis, a meteoroid protection analysis, a detailed dynamic response analysis, an acoustic response analysis, separation analysis, failure mode analysis, weights analysis, and reliability analysis.

Design Loads Analysis. The design loads which control the detailed design and layout of the spacecraft structure will be established from combined steady state and dynamic loads. The combined loads will be reduced to the internal loads used for analysis of critical structural elements. Design load factors, acceptable deflections, buckling loads, and fatigue limits will be defined. The mission profile will be analyzed

and critical loading conditions identified to establish loading intensities and dynamic environments. During spacecraft and mission development, design loads will be iterated as required.

Strength Analysis. Primary and secondary structural members will be analyzed to determine optimum strength-to-weight designs, discontinuity stresses, rigidity, deflections, and margins of safety. Conventional analytical techniques will be used to determine these characteristics for much of the structure. Where additional detailed evaluation is required, and specifically for the calculation of influence coefficients, the redundant force analysis method will be employed.

The structural components of the bus will be evaluated to determine structural integrity under quasi-static design load conditions and vibration test conditions.

Strength analysis will identify items requiring development tests. Such test data will supplement and be correlated with the analysis and design.

Meteoroid Protection Analysis. The meteoroid protection required to meet a realistic mission reliability goal will be determined. The effects of variations in environment which become available from other satellite data and various penetration equations will be used in the analysis. The results of this analysis provide constraints on the skin thickness and design of the structural panels.

Weight Analysis. Weights, centers of gravity, moments of inertia, and mass distributions will be generated commensurate with design details and actual hardware. These data will support the TRW mass properties computer program for the Voyager spacecraft.

Weight tradeoff studies will be conducted in such areas as joint design, insulation, and material selection. Weight optimization studies, as applied to sandwich structures, thermal isolators, and pressure vessel support, for example, will be initiated to minimize weight.

Dynamic Analysis. Investigation and analysis of the complex mass-spring system for the loads encountered during transportation,

launch, boost, transit, injection, and orbit will be continued. The powered flight environments will be examined in more detail and the dynamic response of the Voyager spacecraft will be carefully analyzed. Modal analysis techniques will be employed, using, in various stages of the program, a lumped parameter model, a Timoshenko beam model, and the JPL modal analysis program. Transient effects, such as shroud separation, spacecraft separation, appendage deployment, midcourse maneuvers, capsule canister ejection, capsule separation, and injection engine transients will be examined. Axial, lateral, and torsional response of the spacecraft and capsule will be predicted for the worst-case dynamic load conditions. Dynamic clearance with the shroud will be investigated. Finally, the early development test results will be used to verify or refine the preliminary analysis and the structural parameters. Certain structural components will be designed to satisfy stiffness criteria rather than load capability. Examples of this class of structure are the equipment panels, the outriggers supporting the bus at the flight capsule adapter, and the appendages in both boost and orbital configuration. Light exterior structure such as panels and louvers will be further analyzed for the acoustic environment.

Separation Analysis. Studies of the parallel separation systems will continue to assure that the final design will meet mission requirements. Tolerances in spring rate, spring force misalignments, and geometrical and mass property variations of the separation system elements (i.e., the booster, bus, and capsule), will be used in a digital separation program to determine resultant spacecraft attitude, tipoff rates, separation velocity, and accelerations. Tests will verify separation subsystem performance.

Deployment Analysis. The deployment of the various appendages will be analyzed in detail. The deployment of appendages such as the 12-foot low-gain antenna, the magnetometer boom, the PSP, the science antennas, and the high- and medium-gain antennas will be analyzed to assure that the velocities, accelerations, the transient shock and vibration effects associated with release and deployment are compatible with the system requirements, support structure, and appendage mounted equipment. An articulated, rigid body, analytical model will be used to analyze the various deployment sequences. Loads will be computed as a

function of deployment times. Early design development tests will be performed on flight type appendages to verify design characteristics. Later, deployment tests using flight approved hardware will verify design and performance margins.

b. Design

Structural configuration and complete interface definitions will be established in time to support design reviews.

Data on the capsule-interstage, interstage-LEMDE propulsion module, and the launch vehicle-planetary vehicle adapter interfaces will be developed and coordinated. Various conceptual interface designs, methods of attachment, and separation designs will be generated.

Preliminary layouts will be made for each candidate design. The candidate designs will be analyzed, tradeoff studies completed and selections made. The layouts of the configurations selected will be integrated with the layouts of the interfacing subsystems and specific interface requirements established.

The aft equipment module will be designed with insert patterns identical to those in the solar array modules. Consideration will be given to the relative expansion caused by temperature differences between the aft equipment module structure and the LEM structure.

Since the reaction control nozzles are mounted on the aft equipment module, the design will consider methods to minimize installation and alignment problems occurring prior to and during assembly and operations checkout.

The equipment mounting panel design will reflect the use of sandwich panel structure with standardized equipment mounting rails. The rails will provide additional panel stiffness, establish a common mounting for all standardized modules for ease of removal and checkout, and rigid supports for electrical harness. Thermal louvers will be mounted on the external surface of the panels.

The LEMDE propulsion module will be modified only as required by the Voyager system design requirements. The Voyager outriggers will be designed to attach to the same point on the LEMDE module

structure as the original outriggers. Two of the eight side panels are to be utilized as equipment mounting panels. Attachment fittings will be provided at the top and bottom surfaces for fastening the flight capsule interstage structure, the LEM engine, and the aft equipment module.

The flight capsule will be supported by a cylindrical capsule interstage structure mounted to the forward face of the LEMDE. The interstage will be designed to have sufficient strength, rigidity, and alignment characteristics to satisfy the flight capsule requirements. The attachment of the interstage structure to LEMDE will be accomplished through fittings equipped with thermal isolators.

The planetary vehicle adapter will be designed to be integral with the launch vehicle nose fairing, except for the four local interface fittings. The load will be introduced into the shroud uniformly. Stiffness characteristics of the adapter will be selected to maintain adequate rigidity.

3.2.4 Development Hardware and Test Program

Four major test models will be required to support analysis and design: structural model, capsule emergency separation model, spacecraft-launch vehicle separation model, and nose fairing separation model.

The structural model will be an accurate representation of the flight configuration consisting of the main propulsion module (including tanks, thrust support structure, plumbing), aft equipment module (including solar array modules and appendages), planetary vehicle adapter fittings, and capsule adapter. The structural model will be used for both vibration and static load testing.

The capsule emergency separation model will consist of a hard, mass-inertia mockup of the upper portion (interface) of the propulsion module, a mass-inertia model of the capsule, and a flight configuration capsule interstage with flight type pyrotechnic separation devices. The tests on this model are used to develop the design details of the separation system and, later, to verify the performance of the separation system in normal and simulated failure modes.

The spacecraft-launch vehicle separation model will consist of mass-inertia model of the Saturn IVB (with simulated fairing), a mass-inertia model of the planetary vehicle, and flight configuration planetary vehicle adapter fittings and pyrotechnic devices. Tests on this model are used to verify correct operation of the separation system and to determine separation velocities and tipoff rates. Failure modes are simulated to verify design margins.

The nose fairing separation model will consist of a GFE flight type nose fairing, a flight configuration planetary vehicle adapter, and the structural model spacecraft (with test capsule). Explosive separations of the fairings will be performed to determine the shock loads transmitted to the spacecraft, and to verify proper dynamic clearances between the planetary vehicle and the nose fairing. This test is scheduled for July 1969 and will constitute the final design verification test of the nose fairing/planetary vehicle separation. If the preliminary design of the nose fairing has progressed sufficiently to warrant building an engineering model at an earlier date, it is recommended that a separation test be performed in April 1968 to obtain early definition of problem areas.

The structural/mechanical subsystem (including adapter) development and type approval test program is summarized in matrix format in Table 6. The flight acceptance test program is summarized in Table 7.

Type approval and flight acceptance testing of pyrotechnic devices are included under the pyrotechnic subsystem, Section 3.3.

3.3 Pyrotechnic Subsystem

3.3.1 Description

The pyrotechnic subsystem consists of functional elements utilizing electro-explosive devices (EED) for actuation.

The subsystem encompasses the mechanical attach-release and control devices, EED, and associated power and control circuitry for all Voyager flight spacecraft associated pyrotechnically initiated operations as defined and described in Volume 2, Section 10.

Table 6. Structural-Mechanical Subsystem Development Type Approval Test Matrix

Name of Test	Item Being Tested	Purpose	Description	Test Equipment and Special Facilities
Vibration Survey (Dynamic Test)	Structural model	a) To determine within model limitations the fundamental resonances of the structure including frequencies, general vibration characteristics and transmissibility transfer functions for use in up-dating design and test criteria for assemblies and subassemblies b) To locate structural weaknesses and design faults as related to vibration characteristics c) To assure to a high level of confidence that the proof test model will successfully pass the type approval test.	The test article is the structural model including the main propulsion module (with tanks, plumbing, thrust supports, nonflight rated engine), aft equipment module, planetary vehicle adapter and simulated nose fairing structure, capsule adapter, dummy capsule, equipment panels, and solar array substrate. All load-carrying structures will be flight configuration, mass-inertia simulators will be used for all nonload carrying elements. The fullscale inertially similar structural model of the spacecraft with some simulated subassemblies will be vibrated in three orthogonal directions. The fundamental gross modes of the structure and the response of major components and subassemblies will be defined for low level sinusoidal input vibration applied at the spacecraft separation plane. The excitation level will be increased in increments up to qualification levels and response recorded. Necessary modifications will be incorporated during test series to prevent the occurrence of excessive amplification and/or potential failures. (Preliminary vibration surveys will be made to determine the effects of: 1) various appendage locations, 2) quantity of fuel in the propellant tanks, and 3) with and without dummy capsule.	Vibration system, fixtures, electronic counter, frequency meter, X-Y plotters, oscillographs, accelerometers, strain gages, dynamometers, panoramic sonic analyzer, amplifiers meters, tape recorders
Static Loading Test (Type Approval)	Structural model	To verify the structural integrity of the spacecraft, planetary vehicle adapter and capsule interstage structure to withstand design limit loads without permanent deformation and ultimate loads without failure for all critical design conditions. In addition, influence coefficients will be determined to corroborate and supplement analytical predictions of the structure.	The test article will be the structural model as previously described (vibration survey) except that the mass models (of nonstructural equipment) will be removed. The structural model and planetary vehicle adapter will be vertically cantilevered from a rigid simulated fairing structure. The capsule interstage will be installed and capsule loads will be introduced through a dummy capsule fixture. The spacecraft will be instrumented with strain gages at all critical locations and an independently supported deflection measuring system will be installed. Loads will be applied in increments up to limit and ultimate levels for all critical conditions and load orientations. Strain and deflection readings will be taken at each load increment.	Structural model, test fixture, hydraulic jacks, hydraulic power supply, Edison hydraulic pressure proportioning unit, load measuring devices, strain gages, dial gages, displacement transducers, recorders.
Spacecraft-Launch Vehicle Separation	Planetary vehicle adapter and separation system.	To demonstrate the correct operation of the spacecraft separation system. Determine separation velocity and tipoff rates.	The spacecraft will be simulated by a hard mass-inertial mockup of the spacecraft bus including the outrigger structure. The Saturn IV-B launch vehicle will be simulated by a mass inertial model with a simulated nose fairing. The mass models will be assembled using a flight configuration	Spacecraft and Saturn IV-B mass inertial models, one separation suspension test fixture, ordnance components, high-speed cameras, various transducers, power supplies, recorders.

Table 6. Structural-Mechanical Subsystem Development Type Approval Test Matrix (Continued)

Name of Test	Item Being Tested	Purpose	Description	Test Equipment and Special Facilities
			planetary vehicle adapters and pyrotechnics. The inertial models will be suspended as a double pendulum with a support from the center of gravity of each element with the roll axis in the horizontal plane. The two elements will be joined by the prototype separation system. The length of the pendulum supports will be selected to provide negligible horizontal forces for the separation distance over which measurements must be made. Measurements will be made of the separation velocities and tip-off rates.	
Nose Fairing Separations	Structural model, nose fairing and fairing separation systems.	o To verify proper operation of the adapter ring/nose fairing separation system. o To determine clearance between planetary vehicles and the nose fairing during separation. o To determine the shock loads transmitted to the spacecraft during nose fairing separations.	Place an assembly consisting of the structural model spacecraft, planetary vehicle adapter, lower fairing, upper fairing, and fairing explosive systems in a simulation chamber. Fire the explosive devices in the prescribed sequence to actuate fairing deployments. Measure and photograph the responses.	Space chamber, cameras, accelerometers, strain gages, recording system, test fixture.
Capsule Separation	Flight capsule interstage and emergency separation system flight configuration pyrotechnic devices.	To demonstrate the correct operation of the capsule interstage emergency separation system, determine separation velocities.	An inertial model consisting of the mass and c.g. capsule model and an inertial simulation of the spacecraft will be suspended as a double pendulum with a support from the center of gravity of each element with longitudinal axis in the horizontal plane. The two elements will be joined by the prototype separation system. The length of the pendulum supports will be selected to provide negligible horizontal forces for the separation distance over which measurements must be made. Measurements will be made of the separation velocities.	Capsule mass and c.g. model, inertially simulated spacecraft with flight configuration interface to interstage, one separation suspension test fixture, ordnance devices, high-speed cameras, power supplies, recorders, mass and c.g. model
Material Allowable	Structural elements (glass, laminates, honeycomb, plastics).	Laboratory investigations will be conducted to evaluate fabrication methods and to determine structural characteristics and physical properties.	Typical test parts will be fabricated in laboratory using proposed structural materials. These fabricated samples will be subjected to material and structural tests to evaluate their suitability for use in the spacecraft.	Jigs and fixtures, as required.
Insert Allowable	Structural elements with various insert configurations.	Measure the structural integrity of the design of the selected insert configuration	Apply a static load to the insert and determine its loading characteristics: Examine long term load characteristics of a loaded insert.	Static load machines fixture.
Components	Miscellaneous structural element detail components, i.e., springs, hold-back mechanism, hinge fittings, panels, joints.	To provide test information for the optimum design of detailed structural elements early in the design stages. The tests which will be performed in this category depend on detailed problems which arise during the design phase.	One or more of the proposed designs of the detailed structural element will be fabricated and subjected to the static or dynamic (including acoustic) loading which is expected during the subsequent testing or flight conditions. Measurements of the response (stress, deflection, acceleration, etc.) of the various proposed designs under the expected loadings will be made to provide a basis for the selection of the best design or to provide information for design changes.	Accelerometers, amplifiers, shake table, oscillators, leak detector, pressurization equipment, static testing machines, dial indicators, strain gages, load cells, testing frame
Appendage Deployment	Magnetometer boom and deployment mechanism Note: This test is typical of tests to be performed on the various deployable appendages.	To demonstrate flight worthiness of the magnetometer deployment characteristics. 1) release system operation, 2) latch system operation, 3) micro-switch operation, 4) deflection characteristics, 5) structural stresses, 6) angular accelerations, and 7) deployment time.	An EM magnetometer boom and deployment mechanism will be mounted on a stiffness simulated backup structure. Deployment and hold-down mechanisms shall be flight configuration hardware. A zero-g field will be simulated by a system of counterweights.	Test fixture, rotary position potentiometers, motion picture cameras (64 frames/sec), strain gages, accelerometers, linear motion transducers, amplifiers, connecting circuitry, recording equipment.

Table 7. Flight Acceptance Test Program Matrix

Name of Test	Item Being Tested	Purpose	Description	Test Equipment and Special Facilities
Separation Spring Calibration and Acceptance Test	Separation springs	To determine the spring characteristics and select flight sets	A deflection versus load curve will be established for each spring of the lot. Spring constant, and hysteresis loop will be established by loading springs on an accurate calibration fixture. Repeated loading characteristics, storage characteristics, life characteristics will be evaluated through repeated cycling and environmental exposures. A pedigreed batch will be set aside for flight.	Flight springs, calibration fixture, calibrated weights
Appendage Deployment	Magnetometer boom and deployment mechanism	To demonstrate flight worthiness of the magnetometer boom deployment characteristics. 1) Release system operation, 2) latch system operation, 3) microswitch operation, 4) deflection characteristics, 5) structural stresses, 6) angular accelerations, and 7) deployment time	The flight spacecraft magnetometer booms and deployment mechanism will be mounted on the engineering model spacecraft. Deployment and holddown mechanisms will be flight configuration hardware. A zero-g field will be simulated by a system of counterweights. A series of deployments will be run.	Test fixture, rotary position, potentiometers, motion picture cameras (64 frames/sec), strain gages, accelerometers, linear motion transducers, amplifiers, connecting circuitry, recording equipment

3.3.2 Summary

The flow chart, Figure 12, depicts the interrelation of functions necessary to the development of the pyrotechnic subsystem and its integration into the Voyager flight spacecraft. The schedule for these functions is shown in Figure 13.

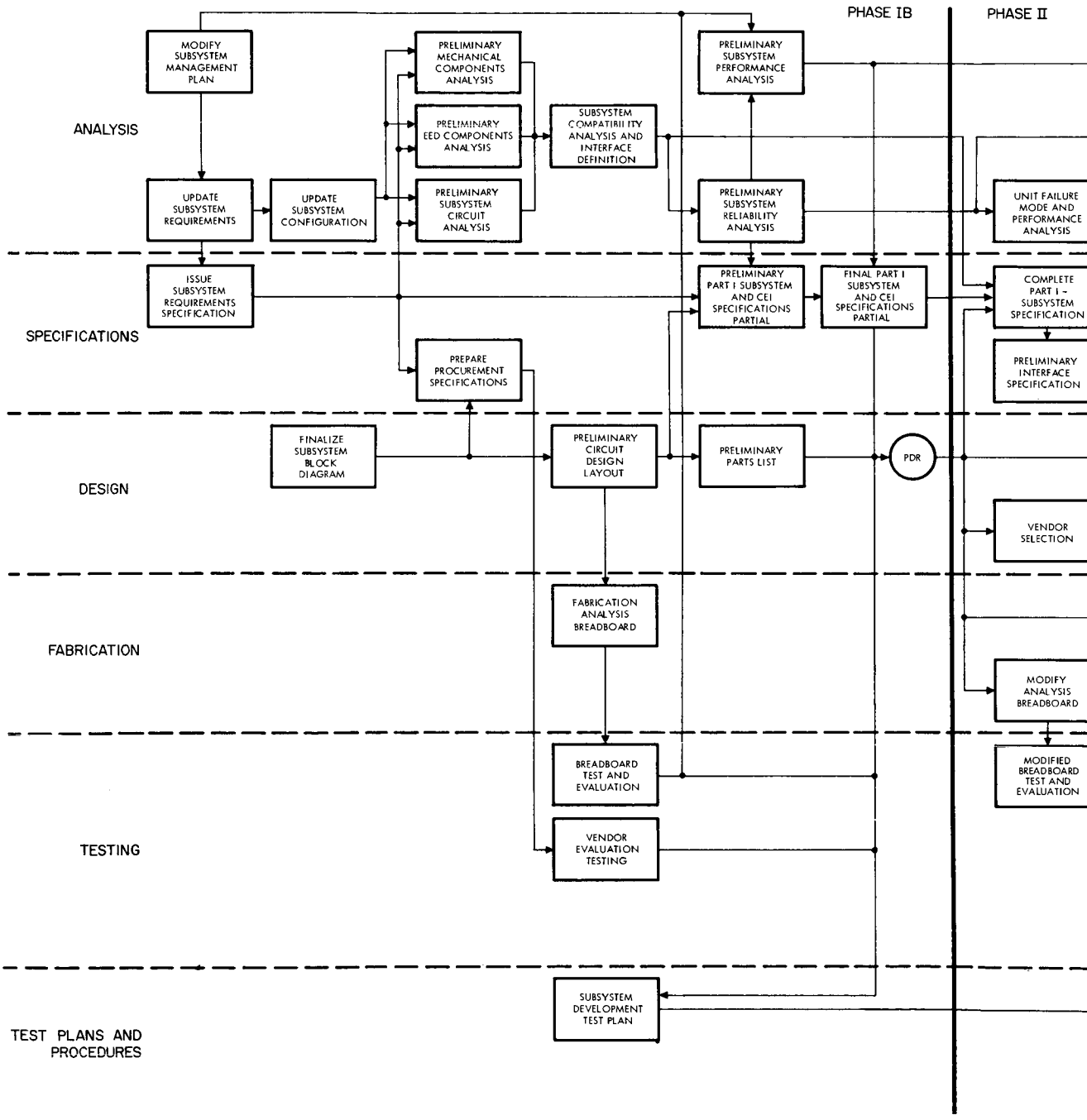
The Phase 1B effort will be initiated by an updating of the subsystem management plan and basic pyrotechnic subsystem requirements. This effort will be documented by the generation of a subsystem requirements specification. An analysis of the subsystem configuration based on the updated requirements will be made and a subsystem block diagram will be finalized.

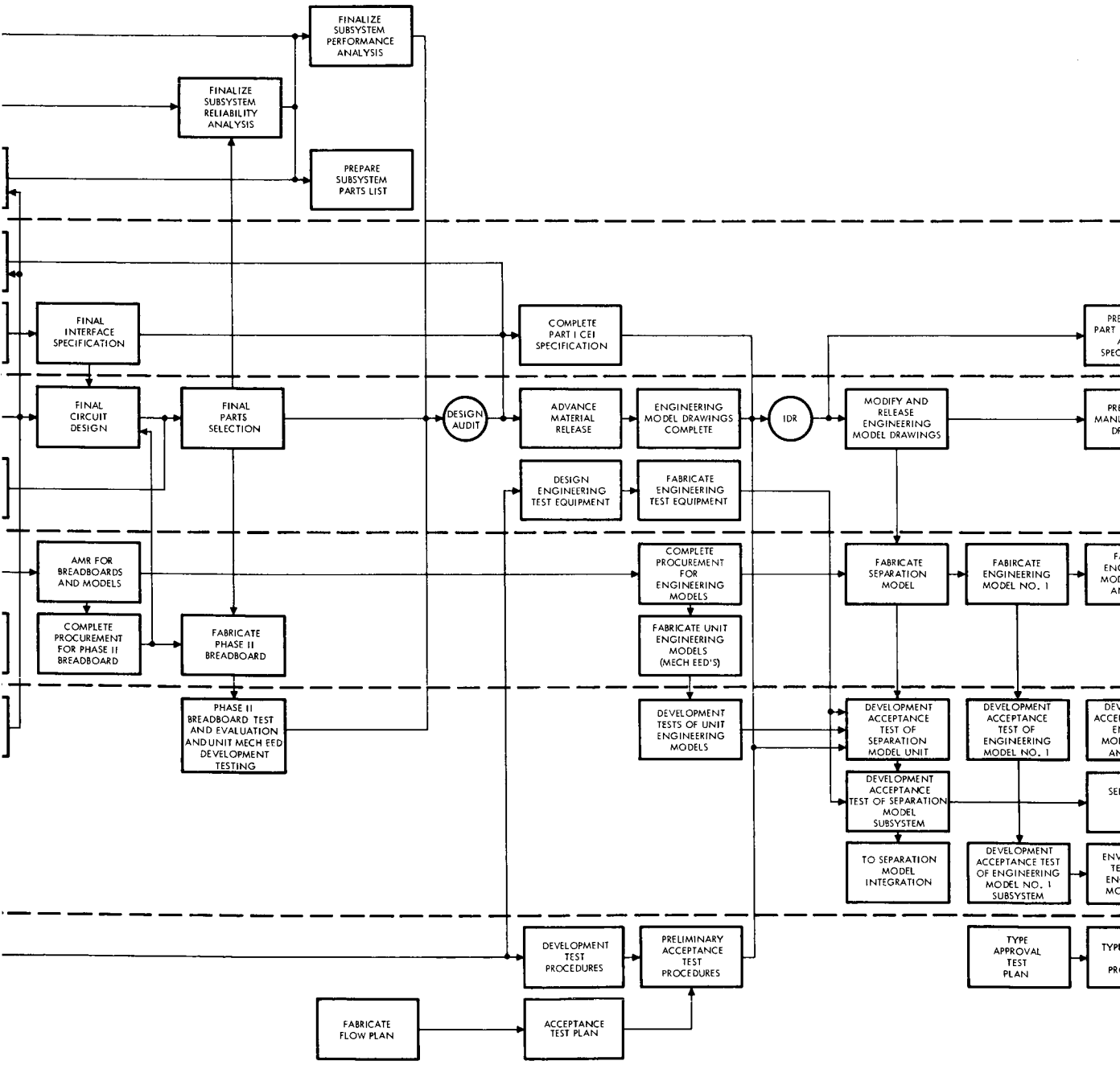
The identification of the subsystem requirements and configuration allows preliminary unit requirements to be defined. These items may then be utilized as a basis for preliminary design of the mechanical and electronic elements of the subsystem, followed by generation of a preliminary parts list. Additional data considered in the design are the results of the component analyses, compatibility analysis, and interface definition.

Breadboards of individual pyrotechnic power and control circuits will be fabricated during the Phase IB program from which data can be obtained to verify the validity of the design concepts. In addition, vendor component evaluation testing will be accomplished on hardware items and utilized in selecting vendors for these components toward the completion of the Phase IB effort.

With the design effort and testing in process, effort will also be directed toward a reliability analysis and performance analysis. Both of these analyses will extend beyond the Phase IB program to intermediate design review (IDR).

In the area of program documentation, the preliminary subsystem and Part I CEI specifications will be prepared by October, 1966. Prior to preliminary design review (PDR), the subsystem specification, a subsystem development test plan and a final Part I CEI (partial) specification will be available. A preliminary interface specification and design data report will also be completed during the Phase IB program.


PHASE IB
PHASE II
ANALYSIS
SPECIFICATIONS
DESIGN
FABRICATION
TESTING
TEST PLANS AND PROCEDURES
MODIFY SUBSYSTEM MANAGEMENT PLAN
UPDATE SUBSYSTEM REQUIREMENTS
UPDATE SUBSYSTEM CONFIGURATION
ISSUE SUBSYSTEM REQUIREMENTS SPECIFICATION
FINALIZE SUBSYSTEM BLOCK DIAGRAM
PRELIMINARY MECHANICAL COMPONENTS ANALYSIS
PRELIMINARY EED COMPONENTS ANALYSIS
PRELIMINARY SUBSYSTEM CIRCUIT ANALYSIS
PREPARE PROCUREMENT SPECIFICATIONS
SUBSYSTEM COMPATIBILITY ANALYSIS AND INTERFACE DEFINITION
PRELIMINARY CIRCUIT DESIGN LAYOUT
FABRICATION ANALYSIS BREADBOARD
BREADBOARD TEST AND EVALUATION
VENDOR EVALUATION TESTING
SUBSYSTEM DEVELOPMENT TEST PLAN
PRELIMINARY SUBSYSTEM PERFORMANCE ANALYSIS
PRELIMINARY SUBSYSTEM RELIABILITY ANALYSIS
PRELIMINARY PART I SUBSYSTEM AND CEI SPECIFICATIONS PARTIAL
PRELIMINARY PARTS LIST
FINAL PART I SUBSYSTEM AND CEI SPECIFICATIONS PARTIAL
PDR
UNIT FAILURE MODE AND PERFORMANCE ANALYSIS
COMPLETE PART I - SUBSYSTEM SPECIFICATION
PRELIMINARY INTERFACE SPECIFICATION
VENDOR SELECTION
MODIFY ANALYSIS BREADBOARD
MODIFIED BREADBOARD TEST AND EVALUATION


FINALIZE SUBSYSTEM PERFORMANCE ANALYSIS
FINALIZE SUBSYSTEM RELIABILITY ANALYSIS
PREPARE SUBSYSTEM PARTS LIST
FINAL INTERFACE SPECIFICATION
COMPLETE PART I CEI SPECIFICATION
FINAL CIRCUIT DESIGN
FINAL PARTS SELECTION
DESIGN AUDIT
ADVANCE MATERIAL RELEASE
ENGINEERING MODEL DRAWINGS COMPLETE
IDR
MODIFY AND RELEASE ENGINEERING MODEL DRAWINGS
DESIGN ENGINEERING TEST EQUIPMENT
FABRICATE ENGINEERING TEST EQUIPMENT
AMR FOR BREADBOARDS AND MODELS
COMPLETE PROCUREMENT FOR ENGINEERING MODELS
FABRICATE SEPARATION MODEL
FABIRCATE ENGINEERING MODEL NO. 1
COMPLETE PROCUREMENT FOR PHASE II BREADBOARD
FABRICATE PHASE II BREADBOARD
FABRICATE UNIT ENGINEERING MODELS (MECH EED'S)
PHASE II BREADBOARD TEST AND EVALUATION AND UNIT MECH EED DEVELOPMENT TESTING
DEVELOPMENT TESTS OF UNIT ENGINEERING MODELS
DEVELOPMENT ACCEPTANCE TEST OF SEPARATION MODEL UNIT
DEVELOPMENT ACCEPTANCE TEST OF ENGINEERING MODEL NO. 1
DEVELOPMENT ACCEPTANCE TEST OF SEPARATION MODEL SUBSYSTEM
TO SEPARATION MODEL INTEGRATION
DEVELOPMENT ACCEPTANCE TEST OF ENGINEERING MODEL NO. 1 SUBSYSTEM
DEVELOPMENT TEST PROCEDURES
PRELIMINARY ACCEPTANCE TEST PROCEDURES
TYPE APPROVAL TEST PLAN
FABRICATE FLOW PLAN
ACCEPTANCE TEST PLAN

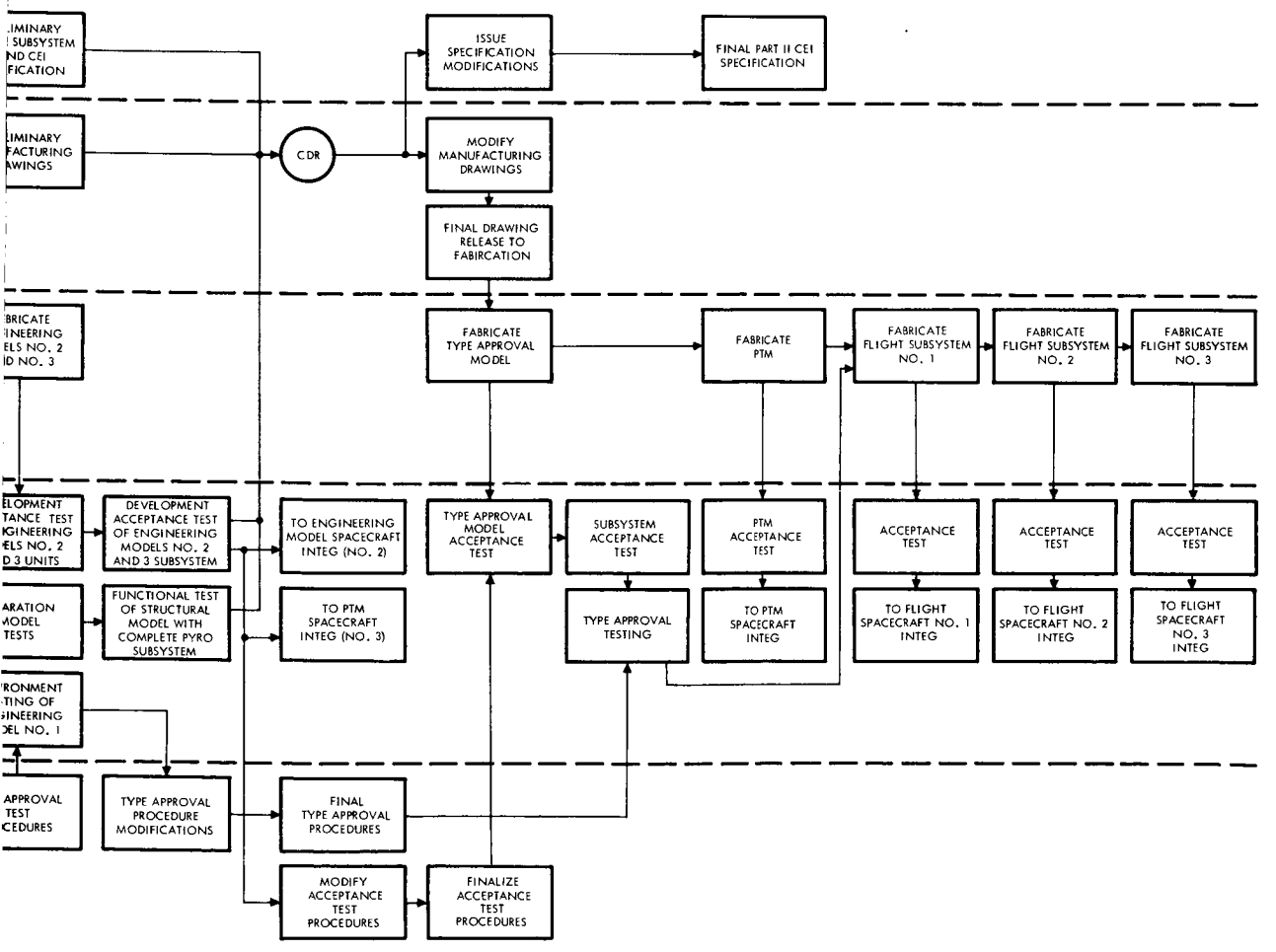
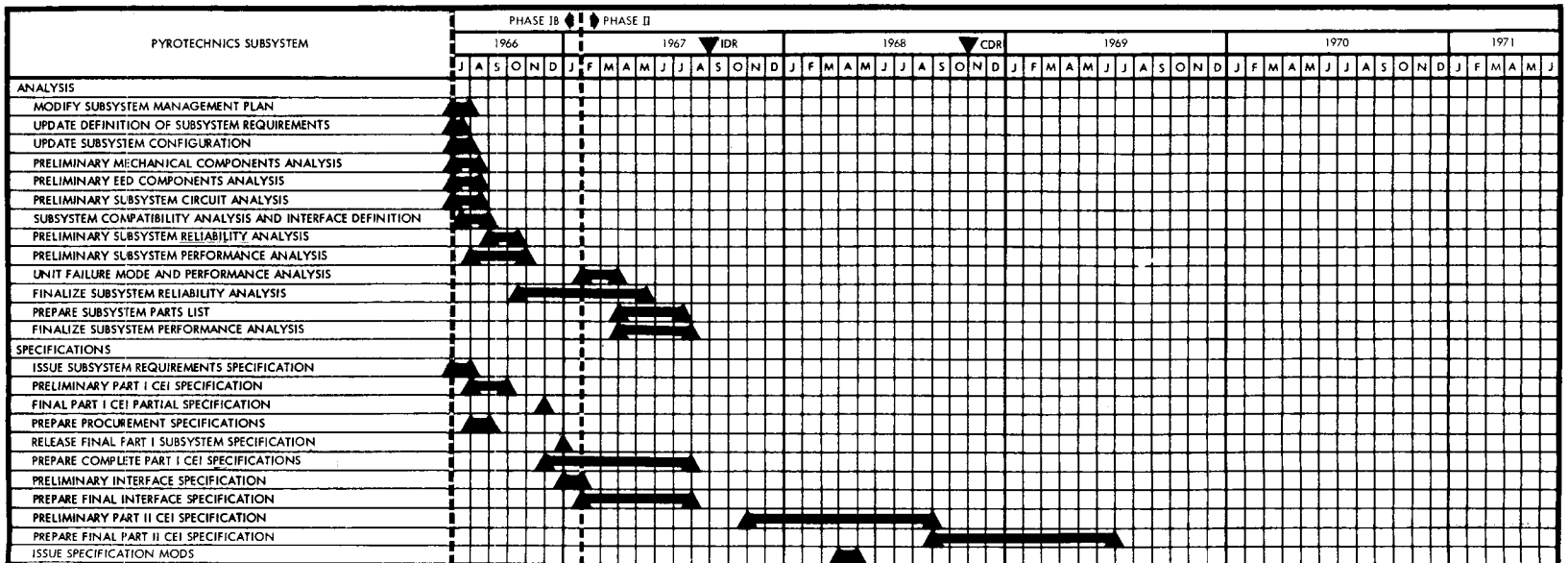


Figure 12. Design and Development of Pyrotechnics Subsystem

3


PYROTECHNICS SUBSYSTEM
PHASE IB
PHASE II
1966
1967
IDR
1968
CDR
1969
1970
1971
J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J
ANALYSIS
MODIFY SUBSYSTEM MANAGEMENT PLAN
UPDATE DEFINITION OF SUBSYSTEM REQUIREMENTS
UPDATE SUBSYSTEM CONFIGURATION
PRELIMINARY MECHANICAL COMPONENTS ANALYSIS
PRELIMINARY EED COMPONENTS ANALYSIS
PRELIMINARY SUBSYSTEM CIRCUIT ANALYSIS
SUBSYSTEM COMPATIBILITY ANALYSIS AND INTERFACE DEFINITION
PRELIMINARY SUBSYSTEM RELIABILITY ANALYSIS
PRELIMINARY SUBSYSTEM PERFORMANCE ANALYSIS
UNIT FAILURE MODE AND PERFORMANCE ANALYSIS
FINALIZE SUBSYSTEM RELIABILITY ANALYSIS
PREPARE SUBSYSTEM PARTS LIST
FINALIZE SUBSYSTEM PERFORMANCE ANALYSIS
SPECIFICATIONS
ISSUE SUBSYSTEM REQUIREMENTS SPECIFICATION
PRELIMINARY PART I CEI SPECIFICATION
FINAL PART I CEI PARTIAL SPECIFICATION
PREPARE PROCUREMENT SPECIFICATIONS
RELEASE FINAL PART I SUBSYSTEM SPECIFICATION
PREPARE COMPLETE PART I CEI SPECIFICATIONS
PRELIMINARY INTERFACE SPECIFICATION
PREPARE FINAL INTERFACE SPECIFICATION
PRELIMINARY PART II CEI SPECIFICATION
PREPARE FINAL PART II CEI SPECIFICATION
ISSUE SPECIFICATION MODS

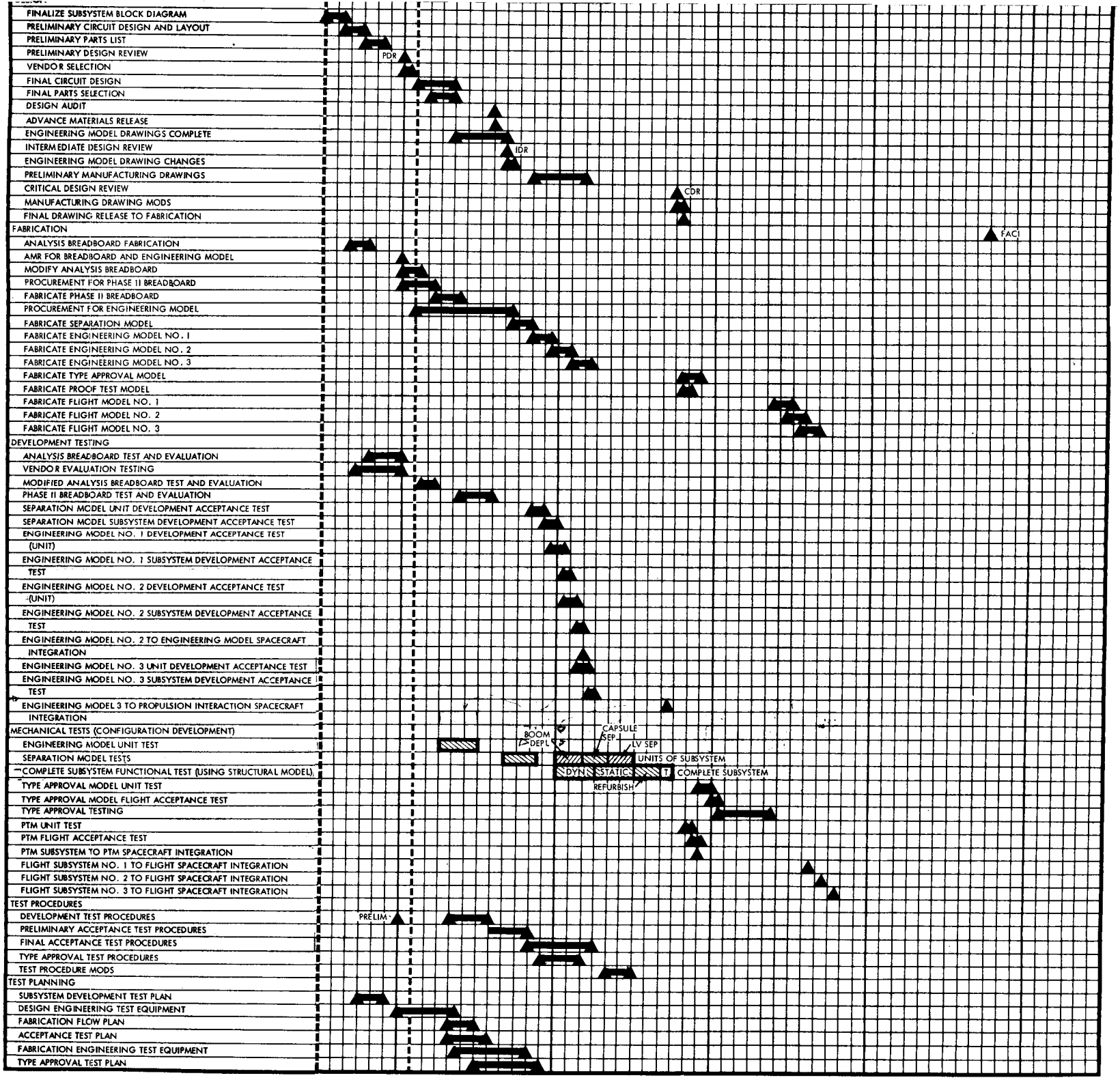


Figure 13. Schedule for Pyrotechnic Subsystem

These documents will be reviewed during the PDR in preparation for entering Phase II of the program.

The results of a unit failure mode analysis performed early in Phase II will be considered along with the Phase IB information in finalization of the subsystem design and parts selection. A Phase II breadboard will be fabricated from which data can be obtained for design verification. Engineering model drawings will be prepared prior to IDR which will be used for fabrication of engineering models (EM). Documentation which will have been finalized prior to IDR consists of the subsystem specification and Part I CEI specification.

The preliminary Part II CEI specification will be prepared prior to CDR and will be finalized prior to first article configuration inspection (FACI), following incorporation of any revisions dictated by test program results.

The development test program on breadboards, EM's, and other special-purpose models will be performed and the results analyzed. Appropriate modifications to the design and documentation will be made when indicated by equipment performance during the test program. Manufacturing drawings will be produced and following the critical design review, will be released to production for use in fabrication of the flight subsystems.

3.3.3 Design and Development

a. Subsystem Analysis

The basis for analysis of the subsystem is the requirements and constraints imposed from a number of sources, including range safety requirements, environmental requirements, interface requirements, etc. The culmination of identification of all subsystem requirements is the definition of the subsystem configuration which will best satisfy these requirements. With the establishment of an optimum configuration, it becomes possible to separate the mechanical and electronic portions of the subsystem.

An analysis of the electrical portions of the subsystem will be made, taking into account electromagnetic compatibility, magnetic compatibility, power consumption, thermal effects, packaging, and safety.

b. Pyrotechnic Control Assembly

The electrical design will utilize the analysis effort, subsystem requirements and block diagram as its starting point. The work which has been done to date has considered the division of design into circuit groups, which, when combined in an assembly, provide the total required capabilities. The circuit groups which have been identified are:

- Safe/arm
- Power conversion
- Energy storage
- Firing circuits

This subdivision will be re-evaluated as detailed design effort proceeds. The selected groups will then be packaged as modules which lend themselves to simplified fabrication. Combining a set of modules will result in the finished assembly.

c. Mechanical Devices

The mechanical devices identified to date can be categorized by type as:

- Release nuts
- Pin pullers
- Electrical disconnects

Testing of the various mechanisms will be performed as part of the testing program for the mechanical subsystem.

An additional device, not identified above, is the explosive valve. Although pyrotechnically actuated, this valve will be designed and tested as part of the propulsion subsystem.

d. Electro Explosive Devices

Since presently identified requirements have permitted the use of a proven device now being used as the Apollo standard initiator, an

extensive development program on EED is not planned, although possible improvements, such as an increased mating of connector shells before pins, will be studied.

3.3.4 Development Test Program

a. Pyrotechnic Control Assembly

The test program (summarized in Table 8) will be initiated on circuit breadboards for conceptual design verification and will proceed to

Table 8. Electrical Development Tests

Name of Test	Item Being Tested	Purpose	Description
Pyro Control Assembly Breadboard	Safe and arm circuits power conversion, energy storage, firing circuits	Perform an engineering evaluation of preliminary circuit design. Provide data on normal operation and susceptibility to voltage and frequency variations, load variations, temperature extremes, combined off-limit conditions with various failure modes induced.	The circuits will be operated under normal conditions using a simulated power source and load. The various parameters identified will be varied through a range in excess of anticipated type approval ranges and performance margin will be determined.
Pyro Control Assembly Engineering Model	EM No. 1, No. 2 and No. 3 assemblies	Same as the breadboard test except on the packaged model. In addition, provide data on electromagnetic compatibility, magnetic compatibility, vibration, and system integration.	In addition to testing described above, tests on conducted and radiated interference and measurements of magnetic stability and induced fields will be performed. Operation in an off-limit vibration environment will be determined. Functional operation as a part of a spacecraft total system operating into simulated EED's will be verified.
Flight Acceptance Test	EM No. 3 TA models, PTM's, flight subsystems	Confirm acceptable operation on a unit and subsystem level.	Functional operation will be verified under a variety of induced conditions, such as voltage and frequency variations, noise injection, temperature variations, etc. Magnetic testing will be accomplished as will vibration and thermal vacuum testing. Functional operation will be reverified. Performance as a subsystem will be determined.
Type Approval Test	TA models and EM No. 2	Confirm satisfaction of design requirements under environmental conditions in excess of those anticipated in flight.	Flight acceptance testing will be accomplished. This will be followed by type approval testing as defined in the JPL Voyager 1971 Performance and Design Specification, 17 September 1965.
Life Test	TA model	Confirm operability over extended period of time.	Subsystem will be operated continuously with functional operation verification every two months.

engineering model testing. The EM provides packaged circuits which can be subjected to a variety of environmental conditions. Since components are in a fixed relationship to each other at this point, meaningful EMC and magnetics data can be obtained. The TA model will provide the subsystem for qualification and testing. Simulated EED will be used for all testing up to the PTM; on the PTM live EED will be used for release operations.

b. Mechanical-Attach EED

In addition to component development tests and engineering unit model tests of the mechanical-attach EED's, the following major models will be required: configuration mockup, capsule emergency separation model, spacecraft-launch vehicle separation model, structural model (which will include a subsystem EM of the complete pyrotechnic subsystem), EM and the PTM. These models are described in Section 3.2. The capsule emergency separation, with flight type pyrotechnic devices, is used to demonstrate correct operation of the separation system. After the vibration and static tests on the structural model, it will be refurbished to include a complete pyrotechnic subsystem. The model will be vibrated and subjected to a series of functional tests involving repeated firings of the EED devices (with deployment of booms and appendages). This test is intended to assure that the PTM will successfully pass the TA test.

Tests on the EM will basically consist of electrical tests of subsystem interface compatibility. The tests of the PTM, which will include firing of EED and boom deployments (after vibration - acoustics and space simulation), constitutes the TA of the complete subsystem. This test is described in Section 4.

The pyrotechnic subsystem development, type approval, and flight acceptance test program is summarized in Table 9.

3.4 Cabling Subsystem

3.4.1 Description

The cabling subsystem consists of the wiring bundles, connectors, and junction boxes which provide for the signal and power flow between all

Table 9. Pyrotechnic Subsystem Test Matrix

Name of Test	Item Being Tested	Purpose	Description	Test Equipment and Special Facilities
Ordnance - Release Assembly Development Test	Development model release assemblies, i.e., pinpuller, cartridge, simulated interface structure.	To develop and establish the release mechanism-ordnance configuration.	Development release assemblies will be installed in test fixtures subjected to firing tests at an altitude environment. The effects of preconditioning environments (vibration, temperature, shock, etc.) and varying size of explosive charge will be determined.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber, shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Release Nut - Flight Spacecraft/Adapter Separation	EM release nut, threaded rod, debris catcher bolt, bolt ejector spring, split holding nut, pressure cartridge, simulated interface structure.	Evaluate functionability of the assembly and determine time to release simulated load and extent of outgassing.	The assembly will undergo vibration and shock environment then be placed in an ordnance thermal vacuum chamber, soaked at -300°F and fired.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Release Nut - Flight Spacecraft/Capsule Interstage Separation	EM release nut, threaded rod, debris catcher bolt, bolt ejector spring, split holding nut, pressure cartridge, simulated interface structure.	Evaluate functionability of the assembly and determine time to release simulated load and extent of outgassing.	The assembly will undergo vibration and shock environment then be placed in an ordnance thermal vacuum chamber, soaked at -300°F and fired.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Release Nut - Antenna Stowage Release	EM release nut, threaded rod, debris catcher bolt, bolt ejector spring, split holding nut, pressure cartridge, simulated interface structure.	Evaluate functionability of the assembly and determine time to release simulated load and extent of outgassing.	The assembly will undergo vibration and shock environment then be placed in an ordnance thermal vacuum chamber, soaked at -300°F and fired.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Dual Pin Puller - PSP Gimbal Uncaging.	EM modified pin puller pressure cartridge, simulated interface structure.	Evaluate functionability of the assembly and determine time to release simulated load and extent of outgassing.	The assembly will undergo vibration and shock environment then be placed in an ordnance thermal vacuum chamber, soaked at -300°F and fired.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Electrical Disconnect - EED Actuated Release.	EM connector with cable, EED actuated release mechanism, pressure cartridge, preloaded bungee cord, simulated interface structure.	Evaluation functionability of the assembly and determine time to release simulated load and extent of outgassing.	The assembly will undergo vibration and shock environment then be placed in an ordnance thermal vacuum chamber, soaked at -300°F and fired.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Complete Pyrotechnic Subsystem (Development Test).	Complete pyrotechnic subsystem mounted on structural model complete with appendages, capsule interstage and launch vehicle adapters and simulated nose fairing.	To demonstrate the correct operation of the pyrotechnic subsystem when subjected to a preconditioning environment.	The test article will be the structural model as previously described (vibration survey) except that the model will be refurbished to include complete engineering models of pyrotechnic subsystem (and the cabling subsystem). The full scale model will be vibrated in three orthogonal directions. The pyrotechnic release devices will be repeatedly fired and movable appendages deployed under a simulated zero "g" field. The spacecraft separation and capsule separation ordnance will be fired, (however, physical separations will not occur. Separation system tests are discussed below).	Vibration system, Fixtures, electronic counter, frequency meter, X-Y plotters, oscillographs, accelerometers, strain gages, dynometers, panoramic sonic analyzer, amplifiers, meters, tape recorders

Table 9. Pyrotechnic Subsystem Test Matrix (Continued)

Name of Test	Item Being Tested	Purpose	Description	Test Equipment and Special Facilities
Spacecraft - Launch Vehicle Separation Model (Type Approval)	Planetary vehicle adapter and separation system.	To demonstrate the correct operation of the spacecraft separation system. Determine separation velocity and tipoff rates.	These tests, which are described in section C-2 (structural-mech. subsystem), are type approval tests of the separation system and pyrotechnic devices.	Spacecraft and Saturn IVB mass inertial models, one separation suspension test fixture, ordnance components, high speed cameras, various transducers, power supplies, recorders.
Capsule Separation (Type Approval)	Flight capsule interstage and emergency separation system flight configuration pyrotechnic devices.	To demonstrate the correct operation of the capsule interstage emergency separation system, determine separation velocities.	These tests, which are described in section C-2 (structural-mech. subsystem), are type approval tests of the separation system and pyrotechnic devices.	Capsule mass and c.g. model, inertially simulated spacecraft with flight configuration interface to interstage, one separation suspension test fixture, ordnance devices, high speed cameras, power supplies, recorders, mass and c.g. model.
Ordnance -- Release Assembly-Extreme Environment Test (Type Approval Test)	Electroexplosive mechanical attach release devices, i.e., release nuts, pin puller, etc. Random selections from each lot buy of each type of device will be tested.	This test is to demonstrate that the margins of environment are satisfactory for the lot acceptance test.	The prototype release assemblies will be installed in a test fixture, subjected to one extreme environment (above qualification level), and test fired in orbit environment. 1) High Temperature 2) Low Temperature 3) Humidity 4) Vibration 5) Shock 6) Life Storage	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber, shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Ordnance -- Release Assembly-Proof Fire Test (Type Approval Test)	Electroexplosive mechanical attach release devices, i.e., release nuts, pin puller, etc. Random selections from each lot buy of each type of device will be tested.	This test is to demonstrate that the undercharge and overcharge ordnance loading tolerances will satisfactorily operate the devices.	The prototype release assembly will be loaded with 15 percent overcharge or 15 percent undercharge cartridges and fired in a vacuum environment. 1) Overcharge 2) Undercharge	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber, shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Ordnance -- Release Assembly (Flight Acceptance Test)	Random samples from flight lot of the above devices.	This test is to demonstrate the functional reliability of the flight lot ordnance devices.	The flight lot assemblies will be properly preloaded in a test fixture and subjected to qualification levels of the Voyager environment encountered during storage shipment, handling, preflight and flight. After exposure the devices will be functionally tested by firing the ordnance devices in an in-orbit environment. NOTE: This test is required for each separate buy of lots of ordnance and release devices.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber, shaker, shock test machine, oscilloscope and camera, vibration and shock test adapters, transducers.
Ordnance -- Satellite Separation System -- Release Assembly Cartridge Current Sensitivity Test (Safe Current Test) Flight Acceptance Test	Random samples from the flight lot buy of initiators (cartridges)	This test determines all fire and no fire current values by determining mean firing current and standard deviation around the mean.	Cartridges will be tested using the Bruceton up and down method to determine quantities of fire or no fire at incremental values about the mean firing current. 1) No fire 2) All fire Note: This test is required for each different cartridge type and will be repeated for additional lot buys of ordnance.	Test fixture, fragment shield, ordnance firing console, temp-humidity chamber, shaker, shock test machines, oscilloscope and camera, vibration and shock test adapters, transducers.

electrical elements of the spacecraft system. It is composed of the following major components:

- Main harness
- Intrapanel harnesses (including that portion of the test harness which is integral with the spacecraft)
- Radio frequency coaxial cabling
- Pyrotechnic harness
- J-boxes (four)

The major portion of the subsystem is located on equipment panels three and seven as shown on Figure 14. These panels are each divided

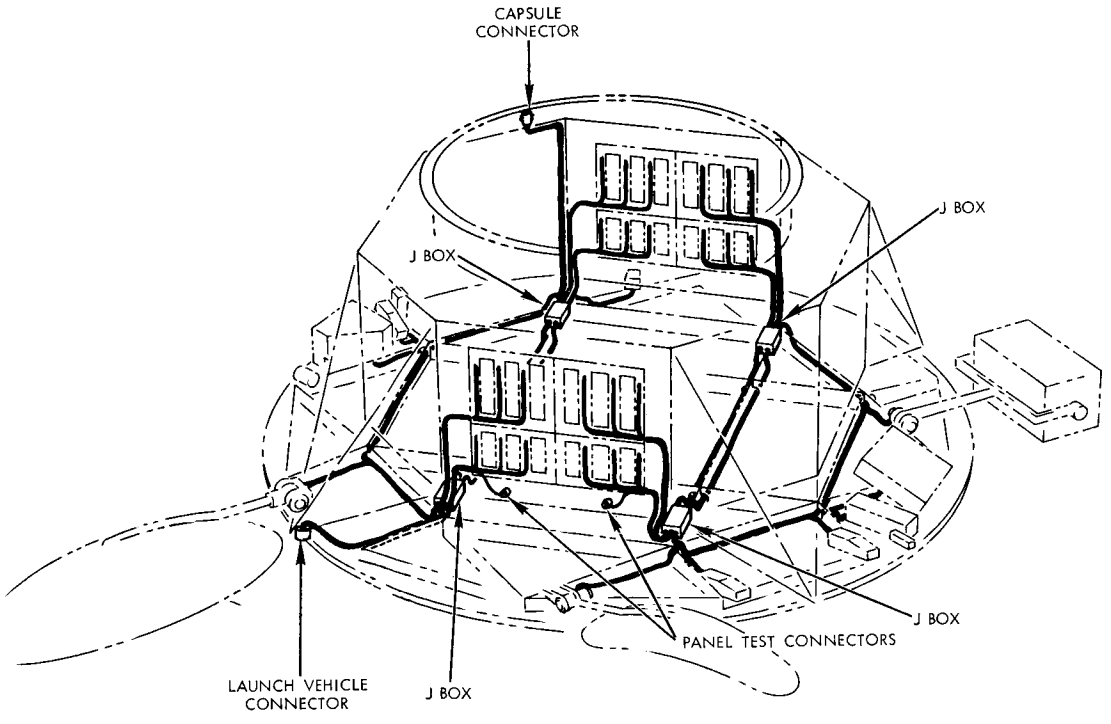


Figure 14. Voyager Spacecraft Cabling Layout

into four subpanels. Interconnecting harnesses are provided to connect the assemblies on one subpanel. Inputs, outputs, and telemetry test points are cabled to J-boxes. The main harness is routed in semicircles around the engine, following the structural beams. Subpanel test connectors and subsystem flyable test harnesses are provided for ground testing.

Microwave cabling uses semirigid coaxial cable; high-frequency and ultra-high frequency cabling is handled by flexible coaxial cable.

3.4.2 Summary

The tasks planned for the development of the cabling subsystem are shown in Figure 15. At the start of Phase IB, the updating of cabling subsystem analysis will commence. Preliminary Part I-CEI specifications and subsystem specifications will be generated and will form part of the preliminary design review (PDR) which is held prior to Phase II go-ahead. Also during Phase IB a preliminary parts list will be generated and long lead-time parts ordered. After the preliminary design review and Phase II go-ahead, parts will be procured for the engineering model (EM) cable subsystems. The cabling subsystem specifications will be updated as a result of the PDR. An intermediate design review will be held after which fabrication and test of the two EM subsystems will start.

After the testing phase has been completed, a critical design review (CDR) will be held, after which fabrication and test of the type approval (TA), proof test model (PTM), and flight cabling subsystems will commence. Also, all specifications will be updated as a result of the CDR.

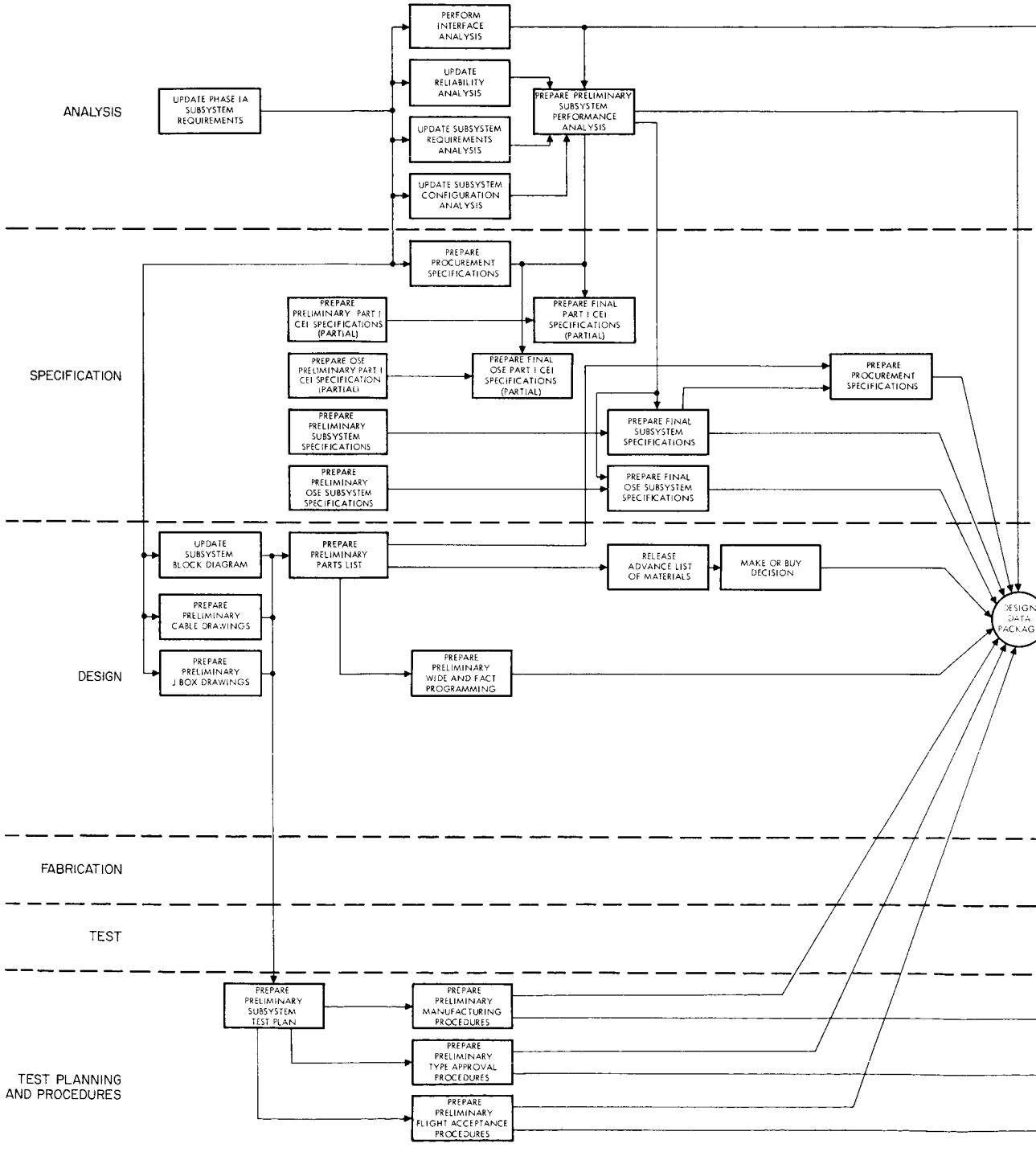
Figure 16 presents the schedule for the activities shown on the development flow chart.

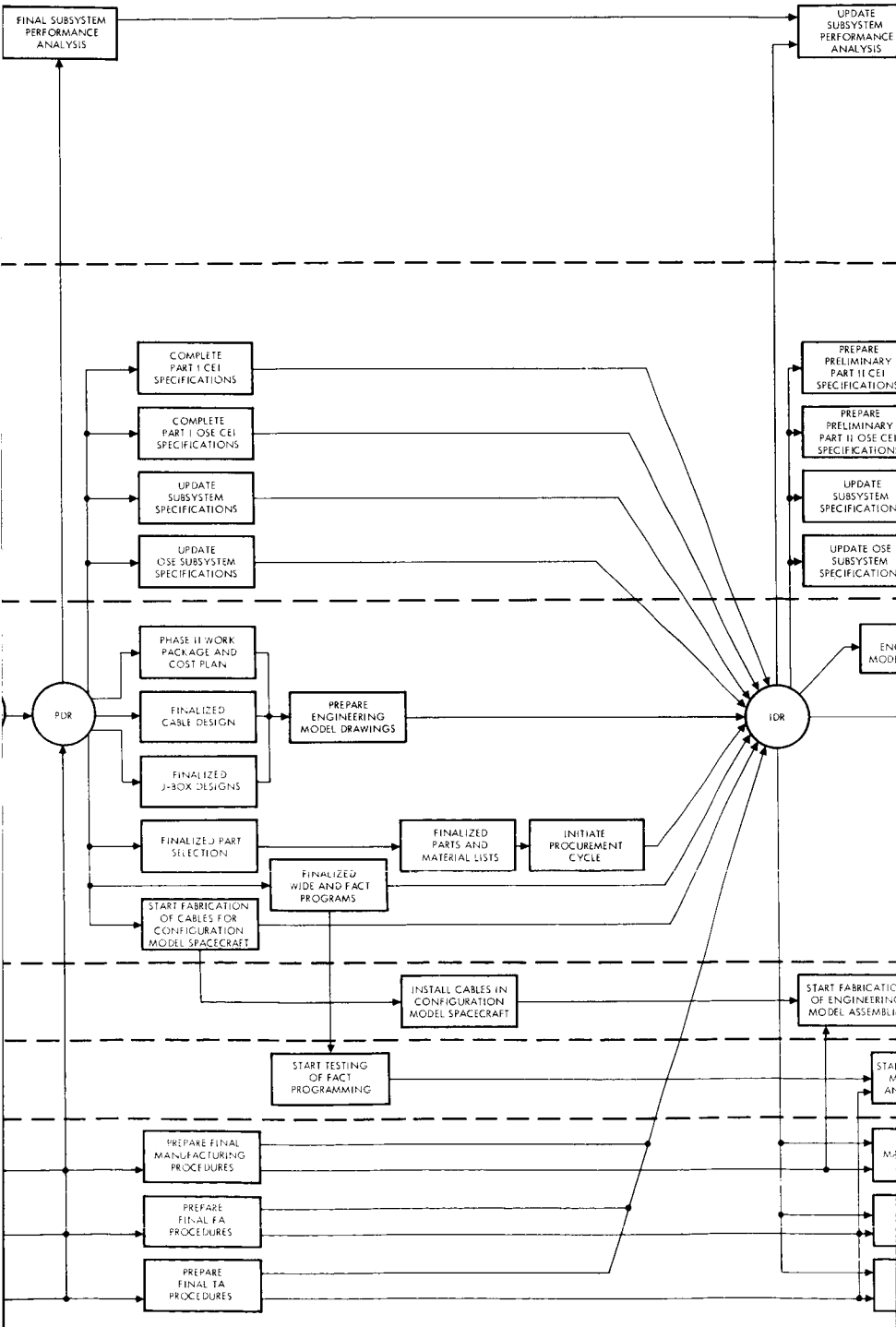
3.4.3 Design, Development, and Fabrication

a. Analysis and Design

After contract go-ahead, any new technical requirements will be analyzed and incorporated into the cabling subsystem design effort. Once the assembly and subsystem requirements are established, various types of analysis will be conducted. The major types of analysis that will be conducted upon the cabling subsystem are described in the following paragraphs.

The reliability of the subsystem will be analyzed in terms of its component parts. The results of this analysis will be used to improve the design and provide data for tradeoff studies. Vibration


ANALYSIS
SPECIFICATION
DESIGN
FABRICATION
TEST
TEST PLANNING AND PROCEDURES
UPDATE PHASE IA SUBSYSTEM REQUIREMENTS
PERFORM INTERFACE ANALYSIS
UPDATE RELIABILITY ANALYSIS
UPDATE SUBSYSTEM REQUIREMENTS ANALYSIS
UPDATE SUBSYSTEM CONFIGURATION ANALYSIS
PREPARE PRELIMINARY SUBSYSTEM PERFORMANCE ANALYSIS
PREPARE PROCUREMENT SPECIFICATIONS
PREPARE PRELIMINARY PART I CEI SPECIFICATIONS (PARTIAL)
PREPARE FINAL PART I CEI SPECIFICATIONS (PARTIAL)
PREPARE OSE PRELIMINARY PART I CEI SPECIFICATION (PARTIAL)
PREPARE FINAL OSE PART I CEI SPECIFICATIONS (PARTIAL)
PREPARE PROCUREMENT SPECIFICATIONS
PREPARE PRELIMINARY SUBSYSTEM SPECIFICATIONS
PREPARE FINAL SUBSYSTEM SPECIFICATIONS
PREPARE PRELIMINARY OSE SUBSYSTEM SPECIFICATIONS
PREPARE FINAL OSE SUBSYSTEM SPECIFICATIONS
UPDATE SUBSYSTEM BLOCK DIAGRAM
PREPARE PRELIMINARY PARTS LIST
RELEASE ADVANCE LIST OF MATERIALS
MAKE OR BUY DECISION
PREPARE PRELIMINARY CABLE DRAWINGS
PREPARE PRELIMINARY J BOX DRAWINGS
PREPARE PRELIMINARY WIDE AND FACT PROGRAMMING
DESIGN DATA PACKAG
PREPARE PRELIMINARY SUBSYSTEM TEST PLAN
PREPARE PRELIMINARY MANUFACTURING PROCEDURES
PREPARE PRELIMINARY TYPE APPROVAL PROCEDURES
PREPARE PRELIMINARY FLIGHT ACCEPTANCE PROCEDURES


FINAL SUBSYSTEM PERFORMANCE ANALYSIS
UPDATE SUBSYSTEM PERFORMANCE ANALYSIS
COMPLETE PART I CEI SPECIFICATIONS
COMPLETE PART I OSE CEI SPECIFICATIONS
UPDATE SUBSYSTEM SPECIFICATIONS
UPDATE OSE SUBSYSTEM SPECIFICATIONS
PREPARE PRELIMINARY PART II CEI SPECIFICATIONS
PREPARE PRELIMINARY PART II OSE CEI SPECIFICATIONS
UPDATE SUBSYSTEM SPECIFICATION
UPDATE OSE SUBSYSTEM SPECIFICATION
PHASE II WORK PACKAGE AND COST PLAN
FINALIZED CABLE DESIGN
FINALIZED J-BOX DESIGNS
PREPARE ENGINEERING MODEL DRAWINGS
PDR
IDR
FINALIZED PART SELECTION
FINALIZED PARTS AND MATERIAL LISTS
INITIATE PROCUREMENT CYCLE
FINALIZED WIDE AND FACT PROGRAMS
START FABRICATION OF CABLES FOR CONFIGURATION MODEL SPACECRAFT
INSTALL CABLES IN CONFIGURATION MODEL SPACECRAFT
START TESTING OF FACT PROGRAMMING
PREPARE FINAL MANUFACTURING PROCEDURES
PREPARE FINAL FA PROCEDURES
PREPARE FINAL TA PROCEDURES

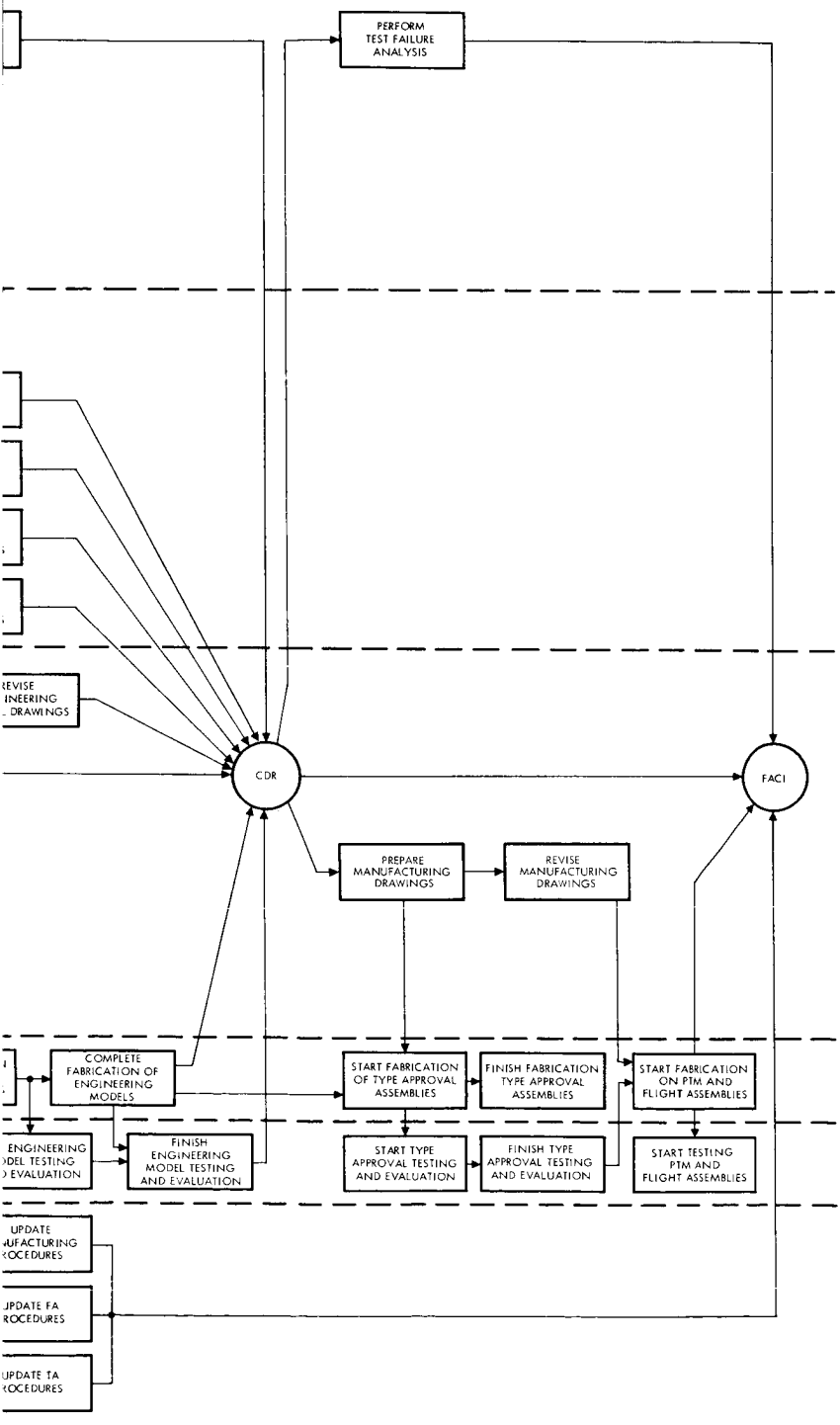


Figure 15. Design and Development of Cabling Subsystem

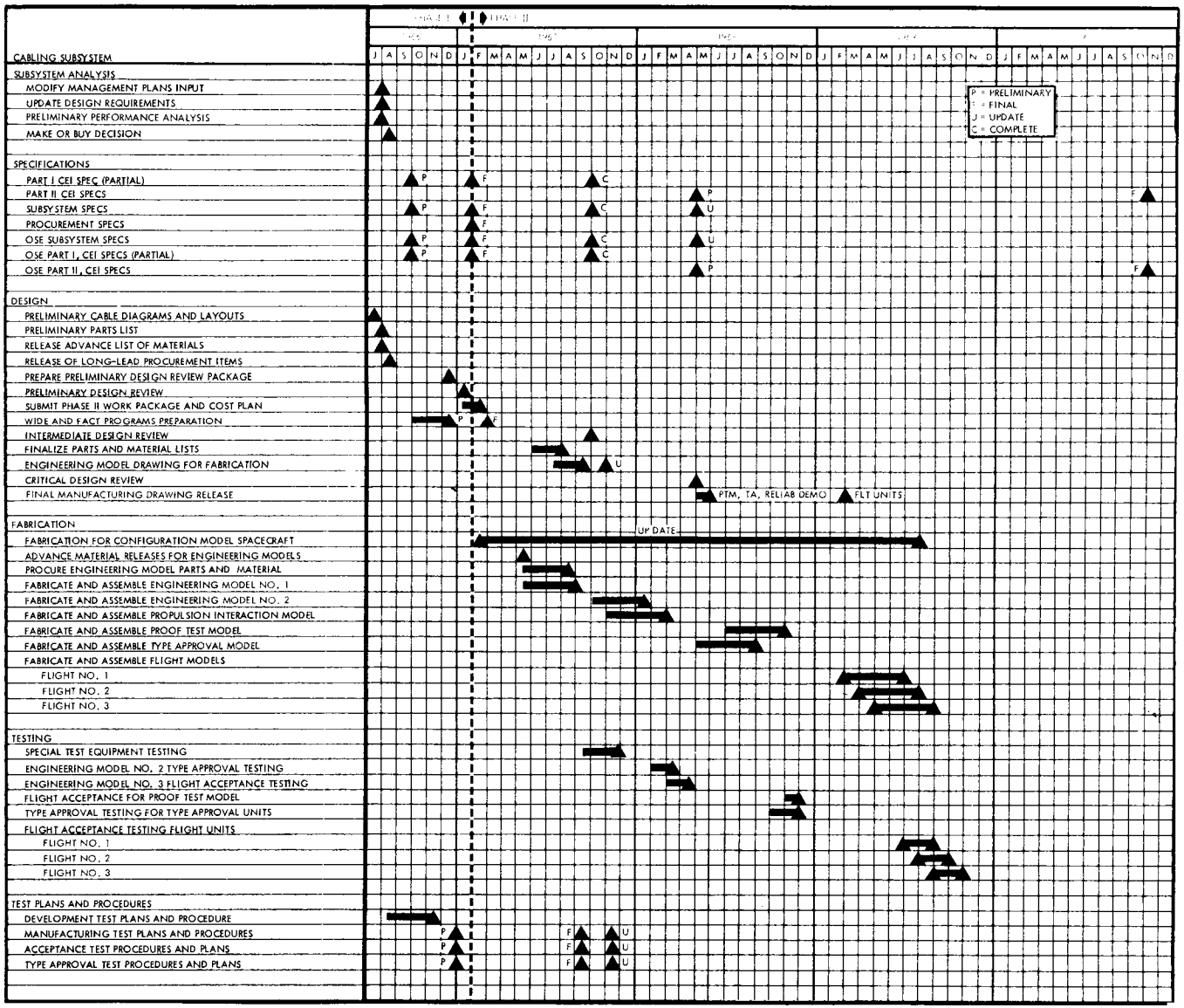


Figure 16. Cabling Subsystem Schedule

analysis will include consideration of cable routing and clamping in terms of the environment during boost and retro firing. The thermal constraint will be analyzed to provide temperature extremes to be used in subsystem design.

Power consumption will be analyzed to determine the proper wire gauges. Signal attenuation and VSWR will be analyzed for the microwave and high frequency cabling and connectors. Studies of shielding, grounding, and bonding techniques as well as selective cable grouping will be performed to minimize EMI and magnetic effects. Similar analyses will be conducted on the J-boxes. Weight estimates and verification will be made.

After all of the aforementioned analyses have been completed, a master cabling diagram will be drawn. The cabling list will be prepared using the WIDE (wiring integration design engineering) concept. In using this concept, every wire comprising the cabling subsystem will be listed on IBM cards using a suitable programming format as well as the wire type, length, input and output pin assignments, signal level, source impedance and load impedance. The IBM cards will be continuously updated and the computer will generate updated wire lists containing the above mentioned parameters. This cabling list will become the prime interface document for Voyager spacecraft subsystem.

The design of the cabling subsystem will also consider the following:

- Minimizing the number of necessary connectors
- Minimizing the number of necessary connections
- Accessibility
- Bend radius limits
- Cable routing and clamping

Once the cable design is firm, mockup cables and J-boxes will be fabricated and installed in the configuration model spacecraft. By utilizing the configuration model spacecraft, cable lengths, bend radii,

routing and clamping can be easily and properly determined at an early schedule time. EM's will be fabricated for each J-box using the cabling list as the major design tool. All of the above analyses will be employed in the design of the J-boxes. No breadboard testing is planned for the J-boxes.

b. Fabrication

A complete set of harnesses will be fabricated for the two EM spacecraft, using the information and knowledge gained from the configuration model spacecraft. Two complete sets of EM J-boxes will also be fabricated for the EM spacecrafts using information obtained from the mockups located in the configuration model spacecraft. After the EM cabling subsystems have successfully passed all tests, fabrication of the TA and PTM units will commence. After these units have passed TA tests, the cabling subsystems for the flight spacecrafts are fabricated. All cabling will be outgassed at the end of the fabrication cycle.

3.4.4 Tests

After fabrication, each EM cable and J-box will be hi-pot and continuity tested using the flexible automatic cable test (FACT) machine. All IBM cards that were developed for the WIDE program are adaptable for the FACT machine. After the hi-pot and continuity tests have been completed, a magnetics property check is made on each cable and J-box. The cables and J-boxes will next be installed in the structural model spacecraft for environmental testing. Once the cables have been connected to the J-boxes, a complete end to end continuity test is to be performed using the FACT machine. The environmental tests are to be performed using TA levels as a minimum on EM No. 1 and No. 2. The tests that are to be performed are vibration, shock, humidity, ethylene oxide compatibility, thermal shock, and thermal vacuum. EM No. 3 will undergo vibration and thermal vacuum testing at flight acceptance levels.

Hi-pot and continuity tests will be performed before and after these environmental tests. After successful completion of these tests, fabrication of TA units is started.

Testing of TA units will be identical to the tests that were run on the EM No. 1 and No. 2 cabling subsystems. For flight acceptance (FA)

testing, the PTM and flight cabling will be hi-potted and continuity checked. The PTM and flight J-boxes will be thermal vacuum and vibration tested to FA levels.

Electromagnetic compatibility tests will be performed on the cabling subsystem as part of the spacecraft system assembly and testing phase. The test matrix is shown in Table 10.

Table 10. Cabling Subsystem Test Matrix

	High-pot and continuity tests	Magnetic properties testing	Vibration (TA levels)'	Vibration (FA levels)	Thermal vacuum (TA levels)	Thermal vacuum (FA levels)	Shock testing	Humidity testing	Thermal shock	Electromagnetic compatibility tests(1)
EM No. 1 Assemblies	x				x					
EM No. 2 Assemblies	x	x	x		x		x	x	x	x
EM No. 3 Assemblies	x	x		x	x				x	
TA Assemblies	x	x	x		x		x	x	x	
PTM Assemblies	x	x		x	x					x
FA Assemblies	x	x		x(2)		x(2)				x

(1) Electromagnetics compatibility testing will not be accomplished at the assembly level, but rather, will be accomplished on a subsystem level as part of spacecraft assembly and test.

(2) Performed during spacecraft system test.

3.5 Power Subsystem

3.5.1 Description

The electrical power subsystem provides four basic forms of electrical power to the spacecraft bus cabling subsystem for utilization aboard the flight spacecraft and the flight capsule prior to separation.

Four power buses will supply: 1) direct array output, 2) unregulated battery output, 3) regulated 400 Hz-phase, and 4) regulated 4000 Hz-phase supply. Requirements for power characteristics other than these will be provided by power conditioning equipment within the individual user subsystems.

The Voyager power subsystem equipment consists of a fixed eight-panel solar array, three silver cadmium batteries, a power control unit (PCU), two shunt element assemblies, three battery regulators, two 4000 Hz inverters, two 400 Hz inverters, and a power distribution box. A precision oscillator and clock divide circuitry are located within the power subsystem to provide system synchronization and reduce the fundamental frequency down to the 400 Hz level.

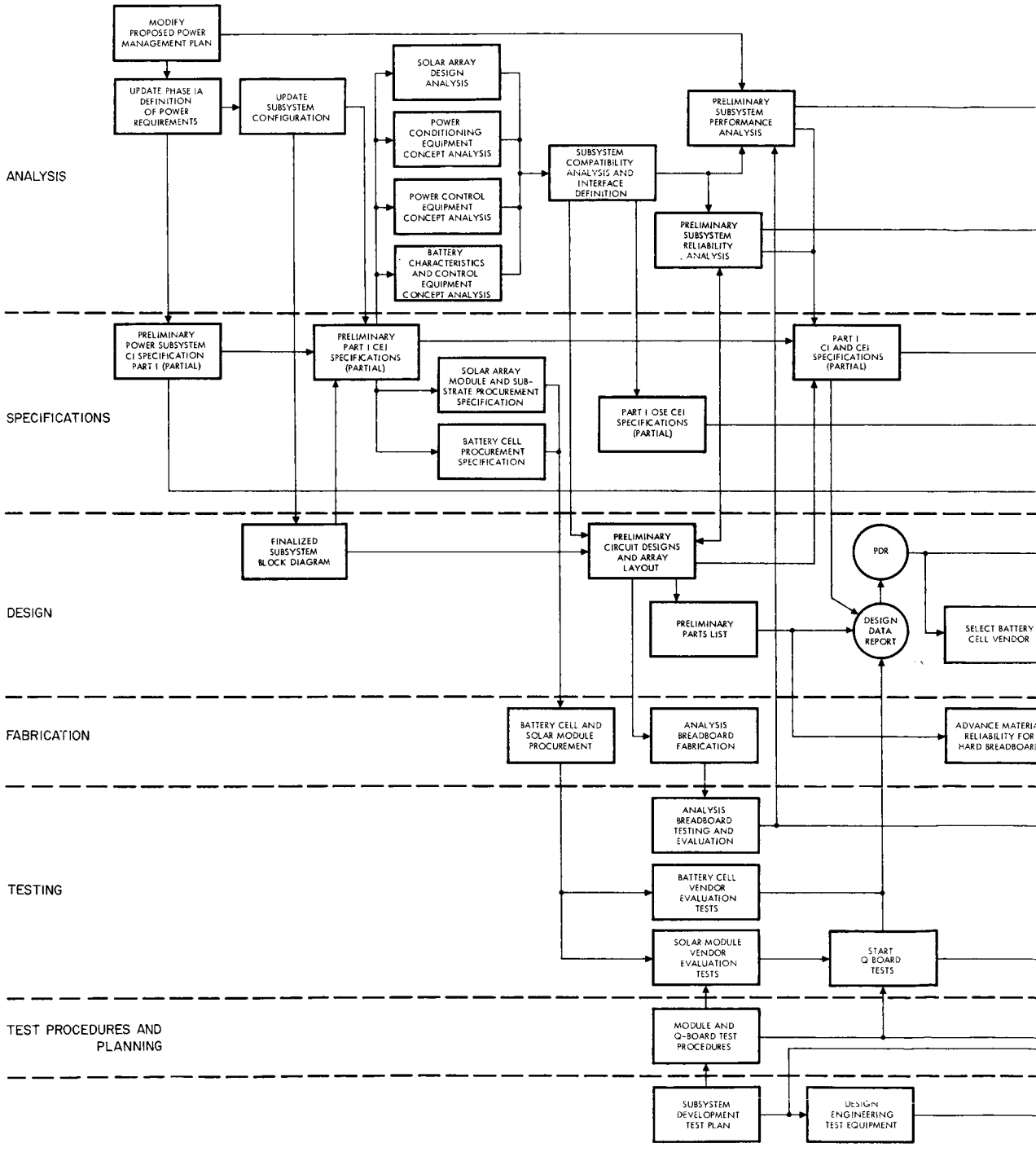
3.5.2 Summary

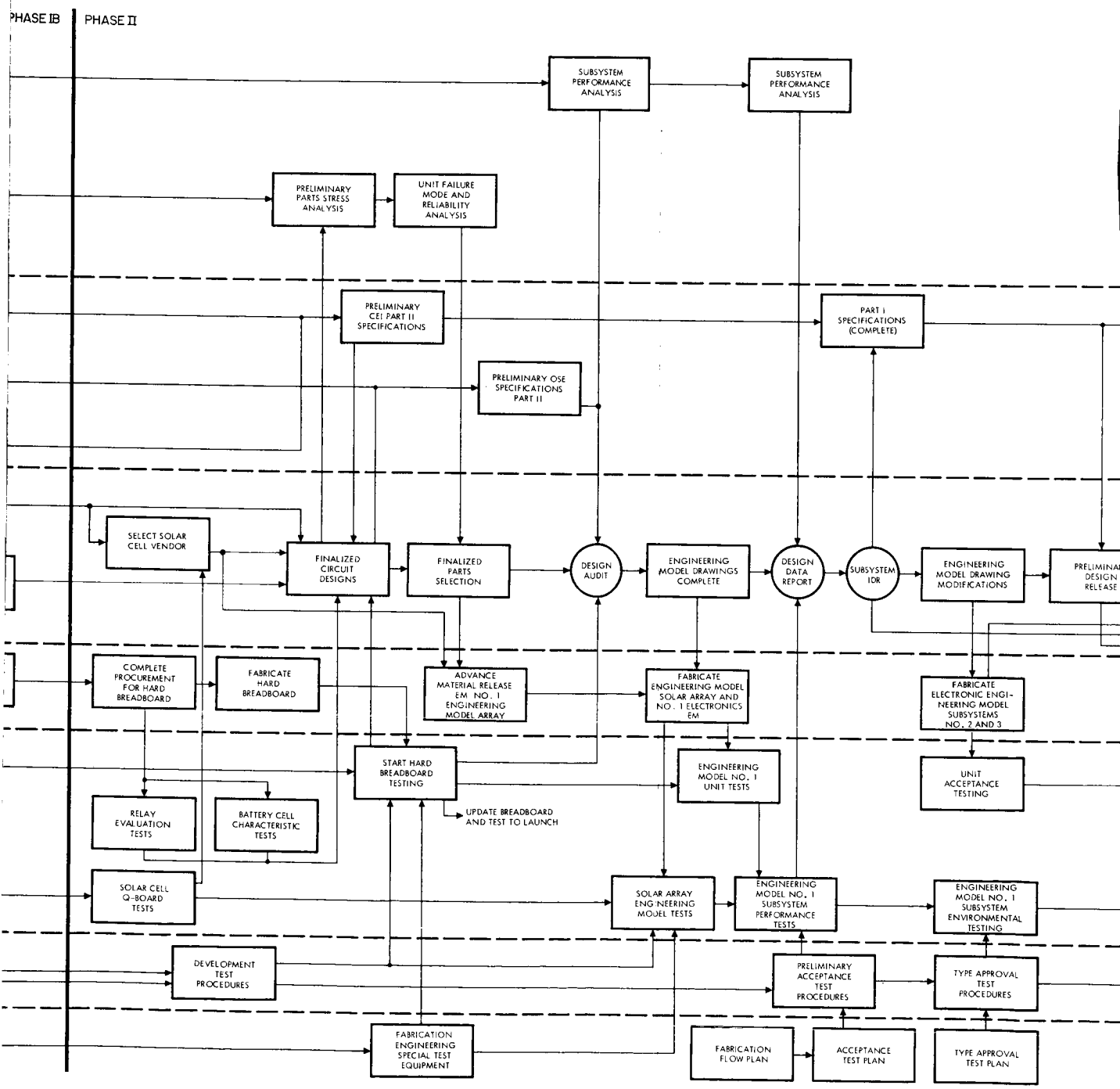
a. Development Flow Plan

The development effort will consist of an orderly progression of analysis, testing, and documentation in accordance with the development flow plan shown in Figure 17. The initial two months of Phase IB activity will accomplish modification of the proposed power management plan in accordance with suggestions of JPL, and updating of the Phase IA definition of power requirements. On this basis, subsystem requirements will be documented in the form of a power subsystem requirements specification and block diagram. Preliminary Part I CEI Specifications (partial) will be delivered to JPL 4 months after go-ahead.

Preliminary circuit design is supplemented by the fabrication and testing of critical circuit breadboards as required to investigate the characteristics and conceptual feasibility of critical circuit designs. Specific requirements for such Phase IB analysis breadboarding are discussed in paragraph 3.5.4 of development testing. Evaluation tests will be performed on sample quantities of solar cell modules and battery cells to provide data for vendor selection. Subsystem analysis of interfaces, reliability and performance will parallel the circuit design effort.

The Phase IB design effort is reviewed at the preliminary design review (PDR) for approval of the subsystem design concept and


MODIFY PROPOSED POWER MANAGEMENT PLAN
UPDATE PHASE IA DEFINITION OF POWER REQUIREMENTS
UPDATE SUBSYSTEM CONFIGURATION
SOLAR ARRAY DESIGN ANALYSIS
POWER CONDITIONING EQUIPMENT CONCEPT ANALYSIS
POWER CONTROL EQUIPMENT CONCEPT ANALYSIS
BATTERY CHARACTERISTICS AND CONTROL EQUIPMENT CONCEPT ANALYSIS
SUBSYSTEM COMPATIBILITY ANALYSIS AND INTERFACE DEFINITION
PRELIMINARY SUBSYSTEM PERFORMANCE ANALYSIS
PRELIMINARY SUBSYSTEM RELIABILITY ANALYSIS
ANALYSIS
PRELIMINARY POWER SUBSYSTEM CI SPECIFICATION PART I (PARTIAL)
PRELIMINARY PART I CEI SPECIFICATIONS (PARTIAL)
SOLAR ARRAY MODULE AND SUB-STRATE PROCUREMENT SPECIFICATION
BATTERY CELL PROCUREMENT SPECIFICATION
PART I OSE CEI SPECIFICATIONS (PARTIAL)
PART I CI AND CEI SPECIFICATIONS (PARTIAL)
SPECIFICATIONS
FINALIZED SUBSYSTEM BLOCK DIAGRAM
PRELIMINARY CIRCUIT DESIGNS AND ARRAY LAYOUT
PDR
PRELIMINARY PARTS LIST
DESIGN DATA REPORT
SELECT BATTERY CELL VENDOR
DESIGN
BATTERY CELL AND SOLAR MODULE PROCUREMENT
ANALYSIS BREADBOARD FABRICATION
FABRICATION
ANALYSIS BREADBOARD TESTING AND EVALUATION
BATTERY CELL VENDOR EVALUATION TESTS
SOLAR MODULE VENDOR EVALUATION TESTS
START Q BOARD TESTS
TESTING
MODULE AND Q-BOARD TEST PROCEDURES
TEST PROCEDURES AND PLANNING
SUBSYSTEM DEVELOPMENT TEST PLAN
DESIGN ENGINEERING TEST EQUIPMENT


PHASE IIB
PHASE II
SUBSYSTEM PERFORMANCE ANALYSIS
SUBSYSTEM PERFORMANCE ANALYSIS
PRELIMINARY PARTS STRESS ANALYSIS
UNIT FAILURE MODE AND RELIABILITY ANALYSIS
PRELIMINARY CEI PART II SPECIFICATIONS
PART I SPECIFICATIONS (COMPLETE)
PRELIMINARY OSE SPECIFICATIONS PART II
SELECT SOLAR CELL VENDOR
FINALIZED CIRCUIT DESIGNS
FINALIZED PARTS SELECTION
DESIGN AUDIT
ENGINEERING MODEL DRAWINGS COMPLETE
DESIGN DATA REPORT
SUBSYSTEM IDR
ENGINEERING MODEL DRAWING MODIFICATIONS
COMPLETE PROCUREMENT FOR HARD BREADBOARD
FABRICATE HARD BREADBOARD
ADVANCE MATERIAL RELEASE EM NO. 1 ENGINEERING MODEL ARRAY
FABRICATE ENGINEERING MODEL SOLAR ARRAY AND NO. 1 ELECTRONICS EM
FABRICATE ELECTRONIC ENGINEERING MODEL SUBSYSTEMS NO. 2 AND 3
START HARD BREADBOARD TESTING
ENGINEERING MODEL NO. 1 UNIT TESTS
UNIT ACCEPTANCE TESTING
UPDATE BREADBOARD AND TEST TO LAUNCH
RELAY EVALUATION TESTS
BATTERY CELL CHARACTERISTIC TESTS
SOLAR CELL Q-BOARD TESTS
SOLAR ARRAY ENGINEERING MODEL TESTS
ENGINEERING MODEL NO. 1 SUBSYSTEM PERFORMANCE TESTS
ENGINEERING MODEL NO. 1 SUBSYSTEM ENVIRONMENTAL TESTING
DEVELOPMENT TEST PROCEDURES
PRELIMINARY ACCEPTANCE TEST PROCEDURES
TYPE APPROVAL TEST PROCEDURES
FABRICATION ENGINEERING SPECIAL TEST EQUIPMENT
FABRICATION FLOW PLAN
ACCEPTANCE TEST PLAN
TYPE APPROVAL TEST PLAN

2 - A

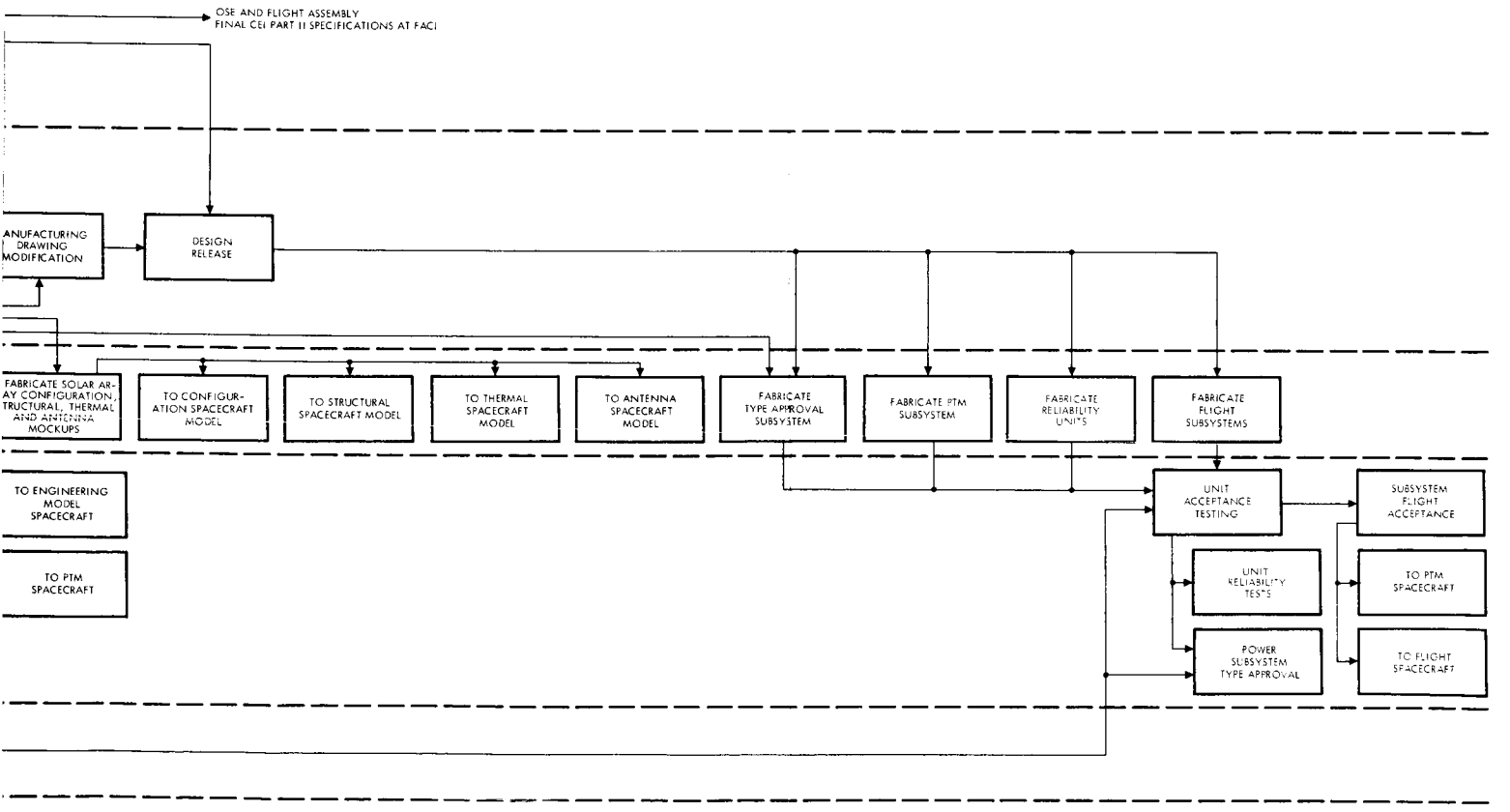


Figure 17. Design and Development of Electric Power Subsystem

3

final Part I CEI Specifications (partial). Following the PDR, Phase IB culminates with the following accomplishments:

- Release of subsystem specifications
- Release of Final Part I CEI Specifications (Partial)
- Selection of battery cell vendor
- Advance material release for Engineering Model No. 1 and Phase II breadboards
- Preparation of detailed power subsystem test plan
- Design of engineering test equipment
- OSE subsystem specification
- Preliminary Part I (Partial) Specifications for OSE
- OSE design data report
- Power subsystem design data report

Phase II development activity is keyed to two major design reviews designated as the intermediate design review (IDR) and the critical design review (CDR). In addition, approximately 2-1/2 months prior to the IDR, it is planned to conduct a less formal design audit. Limited breadboard test results, battery cell characteristics data, and solar module test data will be available at the design audit.

In preparation for the IDR, the design effort proceeds to completion of engineering model (EM) drawings and completion of Part I CEI specifications. Updating and continual testing of the breadboard proceeds until launch. EM No. 1 will be built in the Engineering Laboratory under direct supervision of the design engineers. The start of EM No. 1 fabrication will not be constrained by the IDR milestone, in order to permit early recognition and correction of packaging problems. However, fabrication of EM No. 2 and EM No. 3 will require IDR review and approval of detailed EM drawings, and specifications for parts, materials, and processes. The fabrication flow plan and preliminary acceptance test procedures will also be submitted for IDR review.

EM Nos. 2 and 3 will be fabricated by the Hardware Operations shops as a learning experience, following documented processes and utilizing flight type components whenever feasible. These subsystems will be acceptance tested utilizing acceptance test procedures and installed into the EM and PIM spacecraft. During EM fabrication, the solar array, structural, antenna, and thermal models will also be fabricated for use on test model spacecraft.

Comprehensive testing of EM No. 1 will precede the CDR, including test evaluation under environmental conditions which approximate subsystem type approval tests. Preliminary Part II specifications, manufacturing drawings, and acceptance test documentation will be reviewed at the CDR. Design release follows, after any required modifications.

Equipment fabricated after formal design release includes life test articles, the TA subsystem, subsystems for PTM and flight spacecraft, and flight spares. Release of flight subsystems and spares will be subject to successful completion of type approval tests.

Design, development, and fabrication of the solar array will be subcontracted to RCA; the battery assembly, battery charge regulator, battery boost regulator, and inverters are subject to a formal make-or-buy decision. The power subsystem hardware is listed in Table 11.

b. Schedules

Development will proceed in accordance with the schedule indicated in Figure 18. Early Phase II fabrication and testing of the breadboard, EM No. 1, battery cells, and solar cell Q-boards, necessitates an advance material release during Phase IB, for procurement of these items.

c. Test Program

Development testing of power subsystems and unit equipment is considered to include the following categories of tests:

- Battery cell evaluation tests
- Analysis breadboarding
- Parts and materials tests

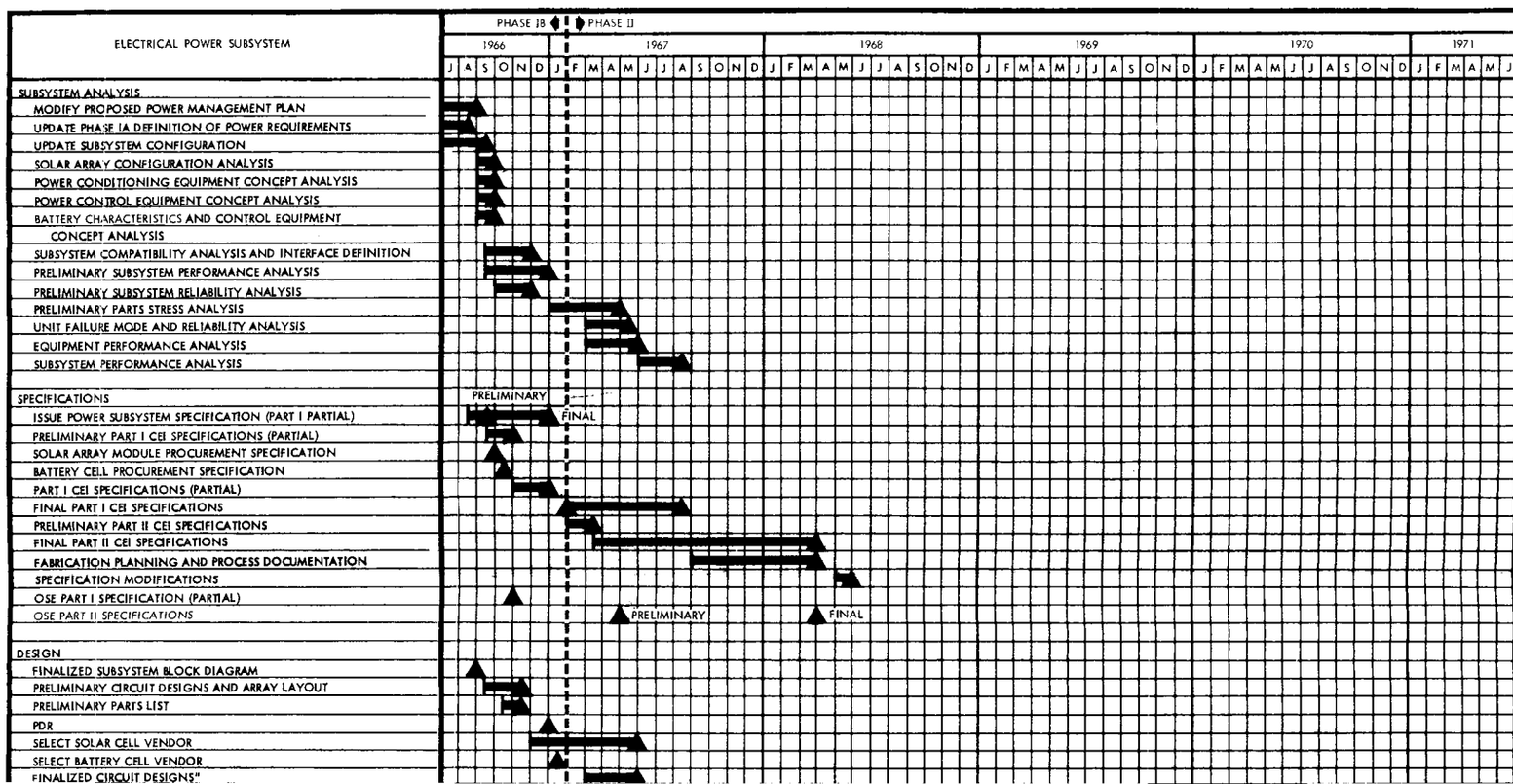
Table 11. Power Subsystem Equipment List

Equipment Name	Development: Subsystem: Critical Circuit Breadboards	Development: Subsystem: Breadboard Test Equipment	Development: Subsystem: Subsystem EM (Lab - EM No. 1)	Development: Subsystem: Vendor Evaluation Tests	Development: Subsystem: Magnetic Evaluation**	Development: Spacecraft: Subsystem EM (Product Line - EM No. 2 and 3)	Development: Spacecraft: Structural Model**	Development: Spacecraft: Thermal Model**	Development: Spacecraft: Antenna Model**	Development: Spacecraft: Configuration Model**	Subsystem Type Approval: TA Subsystem	Subsystem Type Approval: TA Subsystem	Spacecraft Type Approval: PTM Subsystem	Reliability Tests: Life Test Articles	Flight Hardware: Flight Subsystems	Flight Hardware: Flight Spares
Solar Array Panels	-	-	1	-	1	10	8	8	8	8	2		8	-	24	***
Power Control Unit		1	1	-		2	-	-	-	1	1		1	6	3	***
Shunt Element Assembly		1	2	-		4	-	-	-	2	2		2	7	6	***
Battery*	-	3	3	-		6	-	-	-	3	3		3	3	9	***
Battery Charge Regulator		3	3	-		6	-	-	-	3	3		3	4	9	***
4.0 Kc Inverter		1	2	-		4	-	-	-	2	2		2	5	6	***
400 Cps Inverter		1	2	-		4	-	-	-	2	2		2	5	6	***
Power Distribution Unit		1	1	-		2	-	-	-	1	1		1	4	3	***
Solar Array Q-Boards	-	6	-	-		-	-	-	-		-		-	-	-	-
Battery Cells*	-	25	-	75		-	-	-	-		-		-	-	-	-

*Quantity does not include attrition.
**Simulated array mockups.
***Spares to be determined during Phase IB.

- Equipment breadboard tests
- Solar cell Q-board tests
- Solar array EM tests
- Equipment EM tests
- Breadboard and EM subsystem tests.

The above tests are scheduled to yield appropriate data for the design reviews. Subsystem breadboard testing will continue until first launch; EM No. 1 subsystem environmental testing will be completed after the IDR in time for implementing necessary modifications prior to the CDR. Additional testing after design release will include subsystems integration testing utilizing solar array thermal, structural and antenna mockups; subsystem TA; component life testing; and FA testing for the PTM spacecraft and flight equipment. See Section 3.5.4.


ELECTRICAL POWER SUBSYSTEM
PHASE IB
PHASE II
1966
1967
1968
1969
1970
1971
J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J
SUBSYSTEM ANALYSIS
MODIFY PROPOSED POWER MANAGEMENT PLAN
UPDATE PHASE IA DEFINITION OF POWER REQUIREMENTS
UPDATE SUBSYSTEM CONFIGURATION
SOLAR ARRAY CONFIGURATION ANALYSIS
POWER CONDITIONING EQUIPMENT CONCEPT ANALYSIS
POWER CONTROL EQUIPMENT CONCEPT ANALYSIS
BATTERY CHARACTERISTICS AND CONTROL EQUIPMENT CONCEPT ANALYSIS
SUBSYSTEM COMPATIBILITY ANALYSIS AND INTERFACE DEFINITION
PRELIMINARY SUBSYSTEM PERFORMANCE ANALYSIS
PRELIMINARY SUBSYSTEM RELIABILITY ANALYSIS
PRELIMINARY PARTS STRESS ANALYSIS
UNIT FAILURE MODE AND RELIABILITY ANALYSIS
EQUIPMENT PERFORMANCE ANALYSIS
SUBSYSTEM PERFORMANCE ANALYSIS
SPECIFICATIONS
ISSUE POWER SUBSYSTEM SPECIFICATION (PART I PARTIAL)
PRELIMINARY
FINAL
PRELIMINARY PART I CEI SPECIFICATIONS (PARTIAL)
SOLAR ARRAY MODULE PROCUREMENT SPECIFICATION
BATTERY CELL PROCUREMENT SPECIFICATION
PART I CEI SPECIFICATIONS (PARTIAL)
FINAL PART I CEI SPECIFICATIONS
PRELIMINARY PART II CEI SPECIFICATIONS
FINAL PART II CEI SPECIFICATIONS
FABRICATION PLANNING AND PROCESS DOCUMENTATION
SPECIFICATION MODIFICATIONS
OSE PART I SPECIFICATION (PARTIAL)
OSE PART II SPECIFICATIONS
PRELIMINARY
FINAL
DESIGN
FINALIZED SUBSYSTEM BLOCK DIAGRAM
PRELIMINARY CIRCUIT DESIGNS AND ARRAY LAYOUT
PRELIMINARY PARTS LIST
PDR
SELECT SOLAR CELL VENDOR
SELECT BATTERY CELL VENDOR
FINALIZED CIRCUIT DESIGNS"

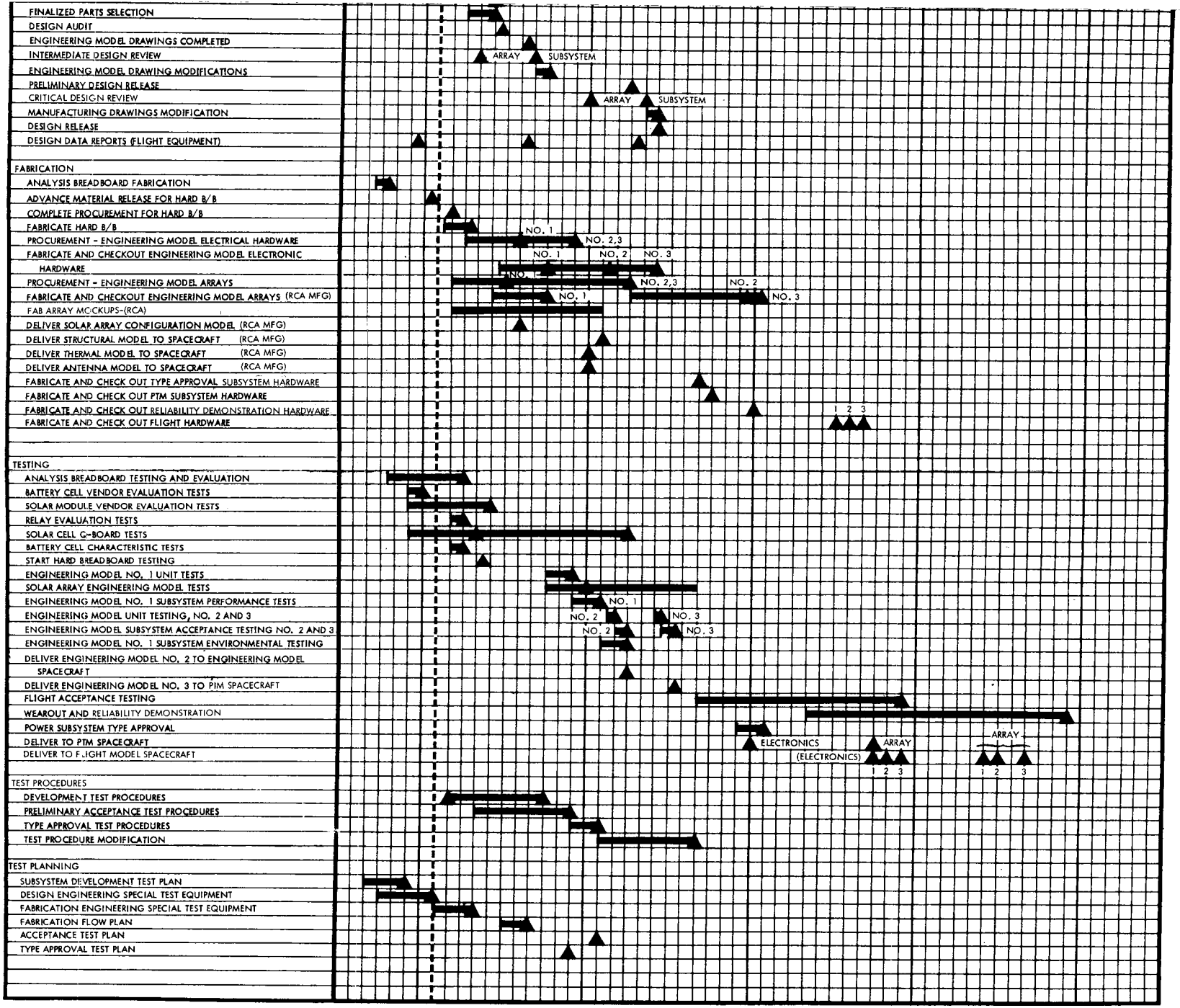


Figure 18. Schedule for Electric Power Subsystem

3.5.3 Analysis and Design

a. Power Subsystem

The subsystem development effort involves those tasks which treat the subsystem as a whole. These tasks are directed towards assuring compatibility of individual units within the subsystem as well as compatibility of the subsystem within the constraints and requirements of the spacecraft.

Surveillance of power margins as affected by revisions to spacecraft load requirements, array temperatures, definition of science loads, and effect of motor heating will be a continuous effort. An electric power management control plan will be implemented to provide systematic and timely acquisition of beneficial design data and load requirements

Subsystem design analyses will include steady-state and transient operations, potential failure modes, and comprehensive reliability assessment. Reliability gains possible from establishing emergency load priorities will be examined. Subsystem interfaces will be identified and requirements for test points and telemetry will be re-examined. DC voltage regulation limits will be a function of the battery boost regulator and the array shunt regulator characteristics. Analysis and test will assure that these limits are compatible with user equipment. Considerations of array area and temperatures will examine implications of potential improvements in solar cell design as well as over-all system tradeoffs of weight and size for missions beyond 1971.

The major subsystem analysis and design effort will be accomplished upon completion of environmental testing of EM No. 1. Power subsystem development design documentation includes the subsystem block diagram, subsystem specification, and contribution to Part I of CEI specifications.

b. Battery

The battery load requirements will be revised in the preparation of preliminary battery and cell specifications. These specifications,

together with supporting instructions and drawings, will be used to procure vendor evaluation cells for performance verification testing by TRW. Data from these cell tests will aid in the final selection of the battery cell vendor and in preparation of cell acceptance, battery acceptance, and environmental test procedures and provide preliminary characteristic data.

Battery packaging layouts will be initiated immediately after selection of the battery cell vendor, concurrent with thermal and structural design analysis. The scheme of charge control requires voltage measurement of individual cells and logic circuitry which sends a charge termination signal when any three single cell voltages exceed a selected reference voltage. A preliminary concept of these voltage sensors and circuitry will be established during Phase IB to permit accommodation by the packaging design. Battery magnetic moment effects will be minimized by proper arrangement of cell orientation and intercell wiring and connections.

The thermal design of the battery will involve the selection of insulating and bonding materials which satisfy the requirements of electrically isolating each cell from the base plate while providing heat conduction between them. The structural design of the battery will include analysis of internal pressure as well as shock and vibration factors.

The most probable failure modes in silver-cadmium batteries are:

1) Seal failures

- High internal pressure
- Mechanical damage
- Weld stresses
- Electrodeposition of braze alloys from seal weld

2) Short circuit failures

- Misalignment of one of the electrode placques

- Silver migration on ceramic insulation
- Insulation breakdown
- Impurities from fabrication processing
- Flaking plate material due to improper heat treating and/or excess material
- Battery connector shorts

3) Open circuit failures

- Connectors
- Seal leaks
- Broken plates due to dynamic environment.

These failure modes will be analyzed to determine the most reliable battery design.

TRW will maintain close surveillance over the battery cell vendor's test procedures, acceptance tests, failure reports and corrective actions. This activity will be supplemented as necessary by TRW participation in vendor analysis of critical design areas.

c. Power Control

The power control functions are packaged as three units designated as the power control unit (PCU), shunt element assembly (SEA), and the battery regulator.

Power Control Unit. The PCU provides voltage sensing and error signal amplification to control the SEA and battery regulators; logic and switching for battery charge control and control of redundant units; generation of a synchronization signal; and telemetry monitors of current and voltage. Upon definition of specific system requirements, detailed tradeoff studies will be made to establish detailed circuit and module concepts. The array shunt point and shunt element dissipation requirements are dependent on the load profile, the number of parallel panel sections, and the output characteristics of the array under various temperature conditions. Analysis will confirm compatibility of the selected operating point with requirements on the control and load dissipation of the shunt elements.

Thermal analysis of the PCU establishes maximum component temperature levels at critical modes of operation. Failure mode analysis includes effects of both open and short circuits under worst case voltage, current, and power characteristics of the components. Circuit and part redundancy of newly designed circuits will be selected for minimum power dissipation considering tradeoffs of gain requirements, reliability, and power conservation. Analysis and testing of the comparative merits and reliability of using relays or solid state switching for battery charge control will be performed to assure that maximum reliability is attained.

The high temperature sterilization may be a problem with some electronic components, notably tantalum capacitors. An evaluation program for selected critical parts will qualify their capability for survival without degradation at this temperature.

Battery Regulator. The battery regulator is packaged with the battery charge regulator and boost regulator. The proposed charge regulator has been designed and breadboard tested on individual silver-cadmium cells. The present circuit uses standard components assembled by welded wire techniques. Each individual cell voltage is measured and compared with a reference. When any cell reaches a voltage equal to or greater than the reference voltage, a signal is sent which terminates charge to the entire battery. The proposed method will require that three cells reach a maximum voltage level in order to terminate charge to the entire battery. The voltage limit at which this occurs varies as a function of temperature. The voltage-temperature function can be varied over a wide range by appropriate selection of network components. Four basis tasks remain before a complete cell level charge control model can be constructed.

1) Current Compensation. Since the limiting voltage of a silver-cadmium cell varies as a function of current, as well as temperature, the charge control system must be modified to include the voltage limiting as a function of current variation. Circuit development effort may include current compensation of voltage limit.

2) Charge Control - Parametric Data. Additional parametric data must be generated to supplement existing data for the complete expected range of operation of the Voyager

battery system. Tests will determine the variation of voltage limit as a function of temperature (at constant current) and current (at constant temperature).

3) Voting Logic Design. The charge control system can be designed so that a full charge signal is required from one or more cells in order to terminate charge to the entire battery. If a large number of cells must signal full charge before battery charge is terminated, the probability that failure of a single voltage sensing module will affect the battery charge operation decreases. Tests will determine the effect of cell mismatch and of multiple signal requirements upon the probability of severe overcharge of the weakest cell in the pack and confirm the selection of a three vote logic for the charge control.

4) Design of Integrated Circuitry. Because the voltage sensing modules are designed to operate directly on the voltage of a single cell (a maximum of 1.6 volts), micro-circuitry will be considered. This enables a decrease in the over-all weight of the charge control system, leads to decreased sensitivity of temperature variations, and improved battery packaging techniques.

Cell voltage sensing and voltage limiter logic for battery charge control will be packaged as an integral part of the battery. Current limiting elements will be packaged as part of the battery regulator.

A boost regulator design study will be made to compare circuit concepts for reliability and efficiency. Active element core losses and copper losses will be evaluated as a function of switching frequency. Although reliability is increased by redundant active elements within the regulator, this approach is wasteful of drive and forward drop losses. Furthermore, the use of three regulators will provide the required redundancy for reliability. Analysis will be made to compare this system with a system utilizing majority voting for the on-off control logic and a conventional boost regulator design. A bus voltage dead-band within the bus regulation tolerance must be utilized to prevent the array shunt regulator from hunting while the boost regulator is active. The use of a common voltage reference derived from the array shunt error amplifier in the PCU will eliminate nominal and long term drift considerations. Testing will be required to establish stability and adequacy of regulation control and dead-band separation.

Based upon these analyses and other subsystem requirements, specifications and functional schematic diagrams will be prepared. Performance requirements for each of approximately six functional modules will be established and module specifications issued. Detailed circuit design will proceed based upon these specifications. Thermal analysis will continue as a reiterative process as packaging layout of the modules and unit assembly progresses.

d. Power Conditioning

Power conditioning requirements will be provided by redundant sets of two inverter packages, with square wave output frequencies of 4.0 kc and 400 cps. Performance characteristics are within the capability of developed TRW design concepts. Detailed design and parts selection will particularly consider reliability and efficiency. Inherent in the generation and distribution of square wave AC is the requirement for detailed attention to RFI suppression and susceptibility. The inverters will accept a synchronization signal from the PCU to prevent beat frequencies.

Performance requirements for functional module assemblies will be established and module specifications issued. Thermal analysis of the preliminary package configuration for each inverter will determine maximum component temperature levels to enable assessment of the following performance parameters and design characteristics under operating conditions: regulation, distortion, turn-on and turn-off characteristics, electromagnetic interference, size and weight, efficiency, reliability, component stress levels, thermal considerations, and mechanical stress.

Thermal analysis will continue as a reiterative process as packaging layout of the modules and total inverter assembly progresses. Electromagnetic interference is a prime consideration in circuit design and layout of components. A modular packaging concept permits placement of parts according to circuit function with short connections between functions. Each inverter package layout will have a metallic enclosure constituting an uncluttered section which will be the connector area; input filters are imposed between the connector area and the inverter functional circuitry. The layout will minimize the effect of

electromagnetic fields external to the modules by proper orientation of components and modules. Conducted RFI will be cancelled by utilizing twisted pairs between modules. The modular concept also permits logical in-line checkouts and minimizes the number of connections between functions. Design analysis will be supported by breadboard and engineering model testing.

New magnetic component designs will be prepared as required, with the corresponding procedures for fabrication and test. It is planned to fabricate these magnetics at TRW.

e. Power Distribution Box

The power distribution box receives the output of inverters, batteries, and solar arrays, and distributes power to consuming subsystems in accordance with commands from the computing and sequencing subsystem. It serves as interface between the power subsystem and the main flight harness. The main electrical functions are connecting, switching, and circuit protection.

Solid state switching versus electromechanical relays will be examined to determine relative reliabilities, EMI effects, and possible interface problems between subsystems. Leakage current constraints may permit solid state switching in one case but not another. Protection devices and circuitry will be evaluated. Proposed telemetering points to monitor currents, voltages, and switching status will be updated during Phase IB.

f. Solar Array

Parametric design studies will be conducted of TRW-provided power requirements, eclipse criteria, and sun distance-angle profile to establish optimized array performance characteristics. Analyses will be performed involving weight and performance tradeoffs, magnetic field due to loop currents, reliability and failure modes, solar cell degradation, structural and thermal design, and ethylene oxide compatibility.

The design effort will stress selection of modules, materials, and fabrication techniques to accommodate the wide range of panel temperatures to be encountered. Analysis will be supplemented by Q-board testing of various proposed designs.

Detailed consideration of possible retrorocket plume contamination and type approval test temperature limits will be studied. Using margins specified by JPL for type approval test temperatures, the resultant test limits become +160 to -160°C. These temperatures are considered an advancement of the state-of-the-art. It is expected that the solar array thermal design will afford predictable and stable thermal performance permitting narrower temperature test margins. This will be the subject of a detailed study in Phase IB.

3.5.4 Test Program

a. Development Testing

Subsystem Development Tests. Subsystem development testing will be performed using both breadboard and engineering model equipment as shown in Table 12. Subsystem tests will utilize the equipment previously used in unit development testing. Breadboard bench testing of the subsystem will continue until first launch.

Table 12. Power Subsystem Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description
Subsystem Breadboard	Breadboards: inverters, PCU, shunt elements assembly, battery regulator, distribution unit	Continuous evaluation of system performance under simulated load conditions. Verify compatibility of PCU and loads. Confirm telemetry requirements	Measure efficiency, regulation; observe waveforms; study failure characteristics, performance switching functions and transients associated with load switching
Subsystem EM	EM's: inverters, PCU, shunt elements assembly, battery, battery regulator, distribution unit	Verify subsystem design in the laboratory. Study response of system to possible failures and develop corrective procedures. Assess electromagnetic compatibility at subsystem level	Operate the entire electrical subsystem less solar array with loads and temperature simulating launch, eclipse and cruise operations

A solar array simulator and dummy loads will be used for both breadboard and engineering model testing. EM's of the battery will be incorporated into the subsystem as part of the EM test phase.

Subsequent spacecraft integration tests will determine compatibility with system performance and electromagnetic interference requirements.

Battery Development Tests. A quantity of battery cells will be purchased during Phase IB for cell evaluation tests and acceptance tested in accordance with procedures of the cell procurement. After acceptance testing, all evaluation cells will be assessed in the following manner:

1) Mechanical evaluation tests

- Visual examination of workmanship
- Examination of supplier manufacturing and test records
- Weight control
- Dimension control
- Seals and weld analysis (X-rays and structural)

2) Performance evaluation tests

- Conditioning requirement analysis
- Storage capability (charge-discharge condition)
- Calibration cycle (capacity comparisons)
- Overcharge equilibrium measurements (pressure effect)
- Relationships of overcharge current, temperature, and voltage limits
- Internal impedance
- Short circuits and electrical leakage
- Thermal properties of cells will be measured using calorimetry and efficiency-energy balance for the determination of heat evolution during the various stages of operation of the cells. Cell heat capacity and thermal conductivity will be measured as required.
- Characterization data will be taken to provide information for determining the parameters of the charge control device. Existing TRW equipment and techniques will be used in these characerization tests.
- Control monitoring characteristics.

Complete breadboard subassemblies of the battery and charge control will be constructed and tested. Battery charge control failure

modes will be programmed and tested to determine the adverse effects upon battery operation. Failures of premature turn-off, failure to turn off, and other control failures will be simulated and their effects upon the over-all system assessed.

Tests will be performed to verify the reliability and performance of the control system design using an EM battery and battery charge control. Engineering model battery environmental tests will be performed to insure reliability of the mechanical, electrical, control, and thermal design characteristics. Mission profile life testing of batteries will be performed with a production model of the charge control to provide confidence in the life cycling capability of the system

Table 13 outlines the battery development test plan.

Table 13. Silver-Cadmium Battery Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description
Battery Cell Vendor Evaluation	Battery evaluation cells	Determination of acceptability of sample cells as obtained from various vendors for comparison	Measurement of capacity, weight, dimensions, internal resistance; detection of flaws, leaks, bulges, short circuits, excessive polarization, or improper or unusual electrical or pressure behavior
Battery Cell Characterization	Battery evaluation cells, with voltage and temperature sensors	To determine battery cell electrical properties: confirm current rates and voltage limit values as functions of cell temperature and usage cycle to enable confirmation of PCU design; generate parametric data of battery voltage characteristics as a function of cell temperature and charge current	Measurement of the relationship between terminal voltage, current, temperature and state of charge; evaluation of efficiency changes with relation to current, temperature and state of charge. Groups of cells are cycled at various current, voltage limits, temperatures, time cycles, with accurate automatic recording of currents and voltages
Battery Cell Pressure	Battery evaluation cells	Verification of battery cell internal pressure characteristics as a function of charge control	Cells are subjected to a program of charge cycles and internal cell pressures measured
Battery Thermal Property	EM battery	Determine relationship between temperature changes, current, voltage, and state of charge	Individual cells are cycled to a mission environment and temperatures measured
Battery Cell Life	Prototype cells	Determine cell characteristics after end-of-life incidents of charge cycle	Individual cells are maintained at a fixed thernal environment and subjected to charge/discharge cycles. Electrical characteristics periodically checked.
Battery Evaluation	EM battery	Verification of battery pack design prior to release for fabrication	Apply load profile and record relationship of voltage, current, temperature and state of charge; perform preliminary vibration, thermal vacuum and load cycle tests; test to failure

Power Control Development Tests. It will be highly advantageous to breadboard several regulator circuits to verify design concepts and obtain parametric performance data during Phase IB. These analysis breadboards would utilize stockroom parts and essentially encompass only critical circuitry rather than complete unit assemblies. The following general areas are identified for such Phase IB analysis breadboarding:

- The array shunt regulator circuitry and sequential shunt elements should be tested to determine practical limits on shunt element crossover voltages, over-all regulator bandwidth, response, regulation, and stability.

- Battery boost regulator circuitry should be tested to determine: operation of paralleled batteries with regard to load sharing, or capacity balancing; boost regulator - load bus regulator interfaces and operating characteristics in order to optimize regulation response and stability.

- Battery charge regulator breadboards should be evaluated with battery cells and cell voltage sensors for preliminary parametric data to enable detailed charge control circuit design. These tests should investigate the effect of cell mismatch and the requirement of multiple signals for voting logic design. The effect of current and battery temperature upon charge control constraints for the particular selected battery cell should be established.

- A partial system configurating of the shunt regulator, boost regulator and charge regulator circuitry should be assembled utilizing the previous breadboard equipment, to enable preliminary examination of the over-all regulation concept and to determine operating ranges and required control accuracies for each of the regulators.

Analysis breadboard testing will continue into Phase II as required. In addition, subsystem development testing will utilize a complete breadboard subsystem and complete engineering model. The breadboard will attempt to simulate critical aspects of component temperatures, interconnects, grounding and shielding to yield the most meaningful analysis of performance. High reliability parts will be used, or in case of schedule problems generic types of high reliability parts.

As part of the over-all subsystem, the breadboard will be continuously updated and cycled through a sequence of modes until first launch.

EM modules will be tested with emphasis on packaging effects and scrutiny of manufacturability. Each module and sets of interconnected modules will be functionally tested with simulated inputs and loads, over temperature extremes to confirm intitial thermal margins, performance analysis, and module interface problems. The engineering model modules will be assembled and tested as complete units and finally as a part of the EM subsystem. As individual units (i.e., the PCU, SEA, and battery regulator) limited environmental testing will be performed. A thermal profile of critical modules will be generated at ambient pressure and results compared with the thermal-vacuum analysis. The engineering model units are then subjected to a resonant frequency scan at low sinusoidal vibration to verify amplification factors assumed in structural analysis. The units are then made part of the EM subsystem for performance testing and comprehensive environmental testing to levels corresponding to TA.

Because normal anticipated variations in battery temperatures have a marked effect upon requirements for charge control operation, complete subsystem assemblies which duplicate the thermal mission load characteristics will be required. Battery and battery regulator failure modes will be programmed in tests to determine the adverse effects upon system operation. Simulated battery failures consist of conditions of cell short circuits, electrical leakage, mechanical leakage, and battery regulator failures.

The PCU utilizes switching functions in critical applications. An anlysis of the comparative merits and reliability of using relays or solid state battery circuit switching will be performed to assure that adequate reliability is attained. Selection of relays will be made only after extensive environmental testing and critical review of all relay characteristics and vendor quality control capabilities.

Selection and testing of parts for high reliability application will be conducted at the same time the tests are being performed to

evaluate equipment designs. EM test procedures are prepared concurrently with breadboard testing. These test procedures will be evaluated in subsequent engineering model testing and used as a guideline for production procedures and TA test procedures.

Test documentation prepared during the development phase will include: module test procedures; unit acceptance test procedures; and TA test procedures. Special test equipment for use in module and unit level development tests will be designed and fabricated by engineering. It will make maximum use of removable capital equipment available in the laboratory. The design of engineering development special test equipment will serve as the basis for corresponding production test equipment to be used for production acceptance checkout of modules, complete unit assemblies. Development tests proposed for power control equipment are listed in Tables 14 and 15.

Table 14. Development Tests for Shunt Element and Power Control Unit

Name of Test	Item Being Tested	Purpose and Objectives	Description
Breadboard Module	PCU breadboard module, breadboard shunt element assembly	Evaluate electrical performance of the modules and preliminary assessment for rating of critical components; study failure characteristics	Simulate range of inputs to individual modules and measure output characteristics. Repeat above measurements over rated temperature ranges
Breadboard	PCU breadboard consisting of interconnected circuit module breadboards, breadboard shunt element assembly	Evaluate compatibility of circuit modules and electrical performance of PCU with simulated inputs and loads performed prior to packaging design	Measure shunt drive signal voltage regulation, battery regulator drive voltage and power synchronization characteristics; measure allowable EMI susceptibility levels, shunt element dissipation and power consumption; observe waveforms and failure characteristics
Engineering Model Module	EM's of the PCU modules, breadboard shunt element assembly	Evaluate electrical performance of PCU modules; checkout module test procedures; test current and voltage telemetry monitors	Measure magnitude and volt-second integral of interference susceptibility at failure levels. Repeat above breadboard measurement over rated temperature ranges. Measure thermal profile; evaluate layout from production and test standpoint; calibrate telemetry outputs
Engineering Model	EM No. 1 of PCU consisting of mounted and interconnected PCU module, breadboard shunt element assembly	Evaluate electrical and environmental performance of finalized packaged design; checkout test procedures and finalize equipment specification; study failure characteristics under environmental extremes; confirm thermal design	Bench test performance per preliminary acceptance test procedure. Test vibration, shock, and thermal vacuum performance; evaluate package for production and test; measure radiated and conducted interference susceptibility; check stress of critical components

Table 15. Development Tests for Battery Regulator

Name of Test	Item Being Tested	Purpose and Objectives	Description
Charge Regulator Breadboard	Charge regulator breadboard	Evaluate electrical performance of design and primary assessment for rating of critical components; develop charge control system compatible with the battery temperature and voltage limiter sensors; investigate alternate methods of control logic in conjunction with battery cell overcharge characteristics; estimate power consumption	By simulation and the use of battery cells; inputs will be varied to simulate a range of battery voltage and temperature characteristics will be measured; charge limiting logic status recorded as a function of input variables
Boost Regulator Breadboard	Charge regulator breadboard	Evaluate electrical performance, reliability and efficiency of alternate circuit designs. Preliminary assessment of component ratings and losses as a function of switching frequency and loads; determine required regulation deadband between array shunt regulation and boost regulation	The boost regulator will be operated using a simulated power supply and bus voltage; simulate variation of battery and bus voltage and measure regulation characteristics, load current, circuit losses, and output ripple
Battery Regulator Engineering Model Module	Module package of EM No. 1, charge regulator, and boost regulator	Evaluate electrical performance of packaged modules; verify finalized module specifications and test procedures	Repeat above breadboard measurements in accordance with test procedures at ambient and overrated temperature ranges
Battery Regulator Engineering Model	EM No. 1 of battery regulator unit	Evaluate electrical performance of finalized package design, checkout test procedures and finalize equipment specification; detect failure under environmental extremes; confirm thermal design	

Inverter. The philosophy and scope of development testing for the inverters will be similar to that described for the power control equipment. Special load simulator test equipment will be designed and fabricated to simulate dynamic load characteristics and load pulse conditions. The circuit design of this equipment will be the basis for subsequent production bench testers. The development tests proposed for inverters are outlined in Table 16.

Power Distribution Unit Development Tests. The philosophy and scope of development testing for the PDU will be similar to that described for the power control equipment. Special load simulator test equipment will be designed and fabricated to enable performance evaluation and life cycling of the unit. The circuit design of this test equipment will be the basis for subsequent production bench testers. The tests proposed for inverters are given in Table 17.

Table 16. Inverter Development Tests

Name of Test	Item Being Tested	Purpose and Objectives	Description
Inverter Breadboard	Breadboards: 4.0 kc inverter, 410 cps inverter, 820 cps inverter	Perform an engineering evaluation of preliminary circuit design; gather initial data for efficiency reliability, thermal and performance analysis; evaluate compatibility of modules prior to packaging design effort	The inverters will be operated using a simulated source and simulated loads. Measurements of starting capabilities, load regulation, power consumption, semiconductor voltages, impedances, efficiency, voltage stress measurements, high temperature performance; observe waveforms and failure characteristics
Engineering Model Module	EM No. 1 modules	Evaluate electrical performance of packaged modules; finalize module specifications and test procedures	Measure performance parameters
Engineering Model Inverter	EM No. 1: 4.0 kc inverter, 410 cps inverter, 820 cps inverter	Performance and engineering evaluation to assure the ability of the unit to meet specified performance requirements; assess packaged design; checkout test procedures and finalize equipment specification; study failure characteristics under environmental extremes; confirm thermal design	The inverter shall be operated using a simulated power supply and simulated loads. Measurements of starting capabilities, load regulation stability, efficiency, high temperature performance, vibration effects, conducted and radiated interference, conducted susceptibility; establish thermal profiles, observe waveforms

Table 17. Power Distribution Unit Development Tests

Name of Test	Item Being Tested	Purpose and Objectives	Description
Breadboard	Assembly consisting of interconnected subassemblies	Preliminary evaluation of subassemblies and assembly for functional performance (switching, transients, overloads, limiting, acceptance of marginal commands, etc.)	Measure unit behavior under parameter variations (input, supply, load, noise, transients, temperature)
Engineering Model No. 1	EM No. 1	Evaluation of assembly for environmental, magnetic and electromagnetic performance. Determine heat rise at continuous loads leakage currents at temperature extremes, contact bounce and arcing	Measure performance under vibration, shock, etc. Measure magnetic and RFI characteristics

Solar Array Development Tests. Solar array development tests will be performed to probe design uncertainties, to aid in module vendor selection, to derive actual performance data, and to confirm the adequacy of the selected configuration, materials and processing. The development test program for the solar array will be conducted on the material, part, Q-board, panel, and array level with results in critical areas available for design reviews. Emphasis in this program will be

placed on the unique Voyager requirements of magnetic cleanliness, contamination control, and the wide range of temperature exposure. Table 18 outlines the scope of the solar array development test program.

Table 18. Solar Array Development Tests

Name of Test	Item Being Tested	Purpose and Objectives	Description
Part Evaluation	Solar cells, blocking diodes, telemetry temperature sensors, electrical connectors and wire from various vendors. Cover glass with ultraviolet reflective filters	Select a minimum of two vendors for each part for Q-board test program. Determine ultraviolet stability of optical coating	Evaluate vendor data and test results. Parts will not be tested until they are incorporated on Q-boards. Measure transmittance and examine coatings after exposure to temperatures and accelerated ultraviolet environment
Material Evaluation	Cell-glass adhesive, intercell connector material, insulation, module-insulation adhesive	Perform an initial material investigation to select candidate materials for Q-board testing	Conduct tests to determine temperature limitation of material, physical and mechanical properties, spectral response, uniformity of batches, lap shear strength of adhesives and compatibility with sterilization methods
Q-Board Evaluation (final part and material selection)	Q-boards that are small-scale mechanical and electrical representation of the solar array panels which have logical combinations of selected vendor parts and materials	To determine which parts and materials best meet the Voyager mission requirements	Perform environmental tests of thermal vacuum temperature cycling (thermal shock) increasing limit temperatures from FA to breadboard levels. Also perform vibration, humidity and ethylene oxide compatibility test to TA limits. Measure electrical output before and after each test and perform visual inspection
Engineering Model Test	EM solar array panel	To verify the design and fabrication technique	Perform FA tests and then TA tests
Antenna Mockup	Antenna mockup array	To provide normal conduction paths during range antenna pattern measurements without the use of actual cells	
Structural Mockup	Structural mockup array	To provide information for system level structural tests	System level structural tests
Thermal Mockup	Thermal mockup array	To check the thermal design of the spacecraft	Thermal-gradient or solar-simulation vacuum tests of the thermal model spacecraft provide an experimental check on thermal analysis
Magnetic Mockup	Magnetic mockup panels	To verify that flight panels wiring design results in acceptable current loop magnetic field	Measure magnetic field under simulated electrical load

Initial testing will be performed on modules composed of a small number of cells wired in a parallel electrical configuration. They will be tested to determine V-I characteristics, efficiency and contact peel. The test results will aid in the preliminary determination and

selection of a vendor for flight procurement, and also provide experimental data to support preliminary analytical design. After initial part and material tests are performed, samples of the most promising solar cell modules, diodes, intercell connecting materials, insulators, and adhesives will be fabricated together in logical combinations on substrates of reduced size to form Q-boards. These Q-boards will be exposed to those Voyager environments which experience dictates will prove most deleterious to survival and operation. Emphasis will be placed in the Q-board test program on thermal-vacuum cycling exposure, since this is considered the most severe environment the solar array must survive. By accelerating the cycling rate, fatigue data obtained is directly applicable to the mission. Testing will also include sterilization compatibility of the design by exposures to sterilization temperatures and ethylene-oxide for prolonged durations. Upon completion of the Q-board testing, statistical analysis of the results will be made to determine the effects and level of significance of each variable tested. With this information, a combination of parts, materials, processes, and substrate structure which best meet the anticipated Voyager conditions will be selected.

Using the final parts and materials selected in the Q-board program, an EM panel will be fabricated using flight fabrication procedures. This panel will be subjected to FA and TA levels and sequence of tests, to thoroughly verify the design and fabrication techniques and to check test procedures and equipment. A complete EM array consisting of eight panels will undergo FA at RCA and delivered to TRW for compatibility checkout with the EM spacecraft. Further evaluation of EM's will be offered by the two EM panels delivered to TRW and supplemented by six of the structural mockup panels to form an array for the PIM spacecraft test.

Mockups will be fabricated by RCA for the development test program. These mockups will conform to the existing flight design in outline and mounting dimensions and simulate those aspects of flight panels which will be under investigation. Five development mockups are planned: structural, antenna, configuration, magnetic, and thermal.

The structural mockup will be a dummy array conforming to existing flight design in outline, mounting dimensions, weight and c.g. location. Structural materials and fabrication will be as close to flight design as preliminary design and tooling permit. The structural mockup array will be used by TRW for vibration and static tests on the spacecraft structural model.

The antenna mockup will be a dummy array with the characteristic RF conductivity of the flight units, for TRW testing of spacecraft antennas.

The configuration mockup will be a dummy array. It will utilize flight design with regard to location, types of electrical connectors, outline and mounting dimensions. The configuration mockup will be utilized by TRW to check mounting fit clearances and connector access.

The magnetic mockup will be a dummy panel whose wiring and magnetic characteristics are representative of the flight model. This will be tested by RCA to evaluate the solar panel loop current field intensity under simulated spacecraft electrical loads and will remain at AED after testing is completed for any future magnetic investigation.

The thermal mockup will be a dummy array conforming to existing flight design in outline, mounting dimensions and thermal mass and surface finish. Appropriate resistors will be provided in the mockup to simulate flight power dissipations. This mockup will be furnished to TRW for design tests of the thermal model spacecraft.

b. Flight Acceptance Testing

FA testing includes performance checkout of individual units under limited environmental levels, and then after assembly into a flight configured subsystem, a final functional performance test to verify proper assembly and conformance with subsystem performance requirements. A solar array simulator will replace the actual array for subsystem level acceptance testing.

c. Subsystem Integration Testing

Initial electrical integration of the power subsystem with the total spacecraft is afforded by assembly and test of the EM spacecraft utilizing power subsystem, EM No. 2 with a complete solar array. These tests will provide a compatibility evaluation of electrical performance and interfaces at ambient conditions, as well as evaluation of subsystem checkout procedures. Further system level testing is afforded by assembly and test of the PIM spacecraft, utilizing power subsystem EM No. 3. The PIM array will consist of two live panels and six of the structural mockup panels also utilized on the structural spacecraft model.

d. Type Approval Testing

TA tests will verify the design under environments which exceed that expected during the mission. The tests include functional and environmental tests at the unit and subsystem level. The subsystem to be used for TA tests will have first successfully completed the FA testing sequence.

On a unit level the tests include pre-environmental functional checkout with voltage and temperature variations, complex sine vibration, shock, humidity, and ethylene oxide exposure. In addition, a solar array panel will be subjected to a profile of thermal vacuum cycling. The levels of vibration for individual components will be as determined by a vibration survey of the total spacecraft. Upon successful completion of unit environmental testing, these units will be included as part of a complete subsystem (exclusive of array), integrated on an equipment panel using flight cabling and attachments. TA continues on a subsystem level through performance testing under variation of source voltage and ambient temperatures, EMC, long-term thermal vacuum, and final ambient checkout.

e. Life Testing

Life tests will be conducted upon all power subsystem units to assess their inherent mean-time-to-failure characteristics (based upon a Poisson distribution) in relation to their design reliability objectives. These test units will be examined in terms of their cause of failure, periodicity and reprtition to afford a direct means of corrective redesign if such patterns occur. Life test units will be fabricated using fully controlled design released drawings, processes, and procedures.

3.5.5 Power Parameter Control Plan

Development of a satisfactory design for the electric power subsystem in accordance with system requirements, growth flexibility, and the overall development plan requires a closely coordinated effort of all design areas. Requirements for design documentation, data reporting, analyses and change control will be systematized to assure compatibility of system constraints with the electric power subsystem design.

In summary, it is proposed that a power management control plan will be implemented, capable of accomplishing the following objectives:

- Design control of power and energy utilization
- Design control of spacecraft power and energy source capability
- Identification of power critical areas and improvement potential
- Identification of power margins for mission planning
- Up-to-date utilization status
- Design control of the electric power subsystem
- Management visibility of design changes related to electrical energy requirements

A draft of this control plan will be included in the implementation plan for the Phase IB proposal. During the initial two months of Phase IB activity, it is planned to modify and implement this plan in accordance with JPL suggestions.

3.6 Command Subsystem Development

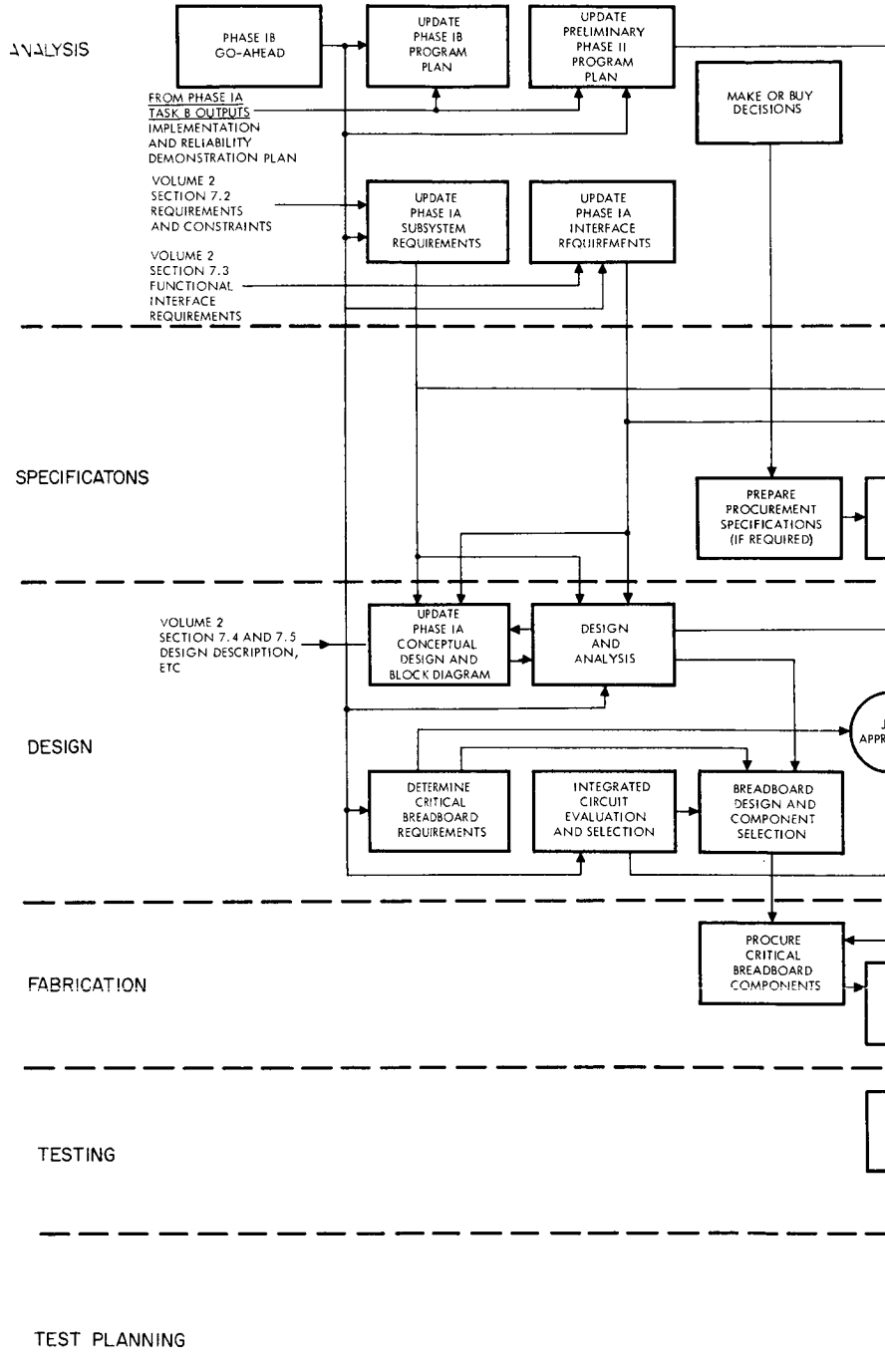
3.6.1 Description

The command subsystem consists of four units: two redundant command detector units whose outputs provide the inputs for two redundant decoder units.

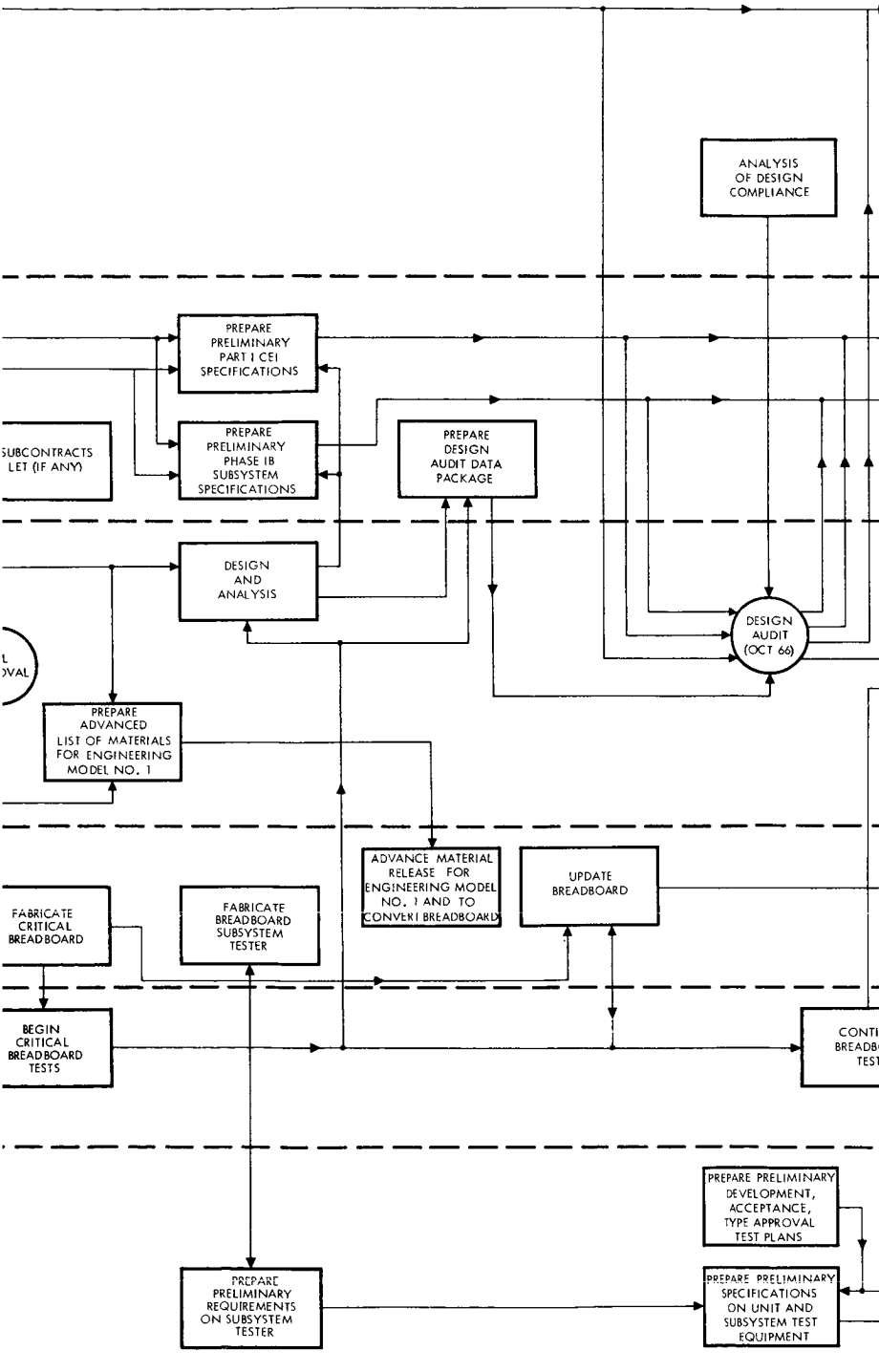
3.6.2 Summary

a. Development

Figure 19 is the development flow diagram for the command subsystem. The periods scheduled for the performance of each of the tasks shown in the flow diagram are presented in the schedule of Figure 20.


ANALYSIS
PHASE IB GO-AHEAD
UPDATE PHASE IB PROGRAM PLAN
UPDATE PRELIMINARY PHASE II PROGRAM PLAN
MAKE OR BUY DECISIONS
FROM PHASE IA TASK B OUTPUTS
IMPLEMENTATION AND RELIABILITY DEMONSTRATION PLAN
VOLUME 2 SECTION 7.2 REQUIREMENTS AND CONSTRAINTS
UPDATE PHASE IA SUBSYSTEM REQUIREMENTS
UPDATE PHASE IA INTERFACE REQUIREMENTS
VOLUME 2 SECTION 7.3 FUNCTIONAL INTERFACE REQUIREMENTS
SPECIFICATONS
PREPARE PROCUREMENT SPECIFICATIONS (IF REQUIRED)
VOLUME 2 SECTION 7.4 AND 7.5 DESIGN DESCRIPTION, ETC
UPDATE PHASE IA CONCEPTUAL DESIGN AND BLOCK DIAGRAM
DESIGN AND ANALYSIS
DESIGN
DETERMINE CRITICAL BREADBOARD REQUIREMENTS
INTEGRATED CIRCUIT EVALUATION AND SELECTION
BREADBOARD DESIGN AND COMPONENT SELECTION
PROCURE CRITICAL BREADBOARD COMPONENTS
FABRICATION
TESTING
TEST PLANNING

PHASE IB


ANALYSIS OF DESIGN COMPLIANCE
PREPARE PRELIMINARY PART I CEI SPECIFICATIONS
SUBCONTRACTS LET (IF ANY)
PREPARE PRELIMINARY PHASE IB SUBSYSTEM SPECIFICATIONS
PREPARE DESIGN AUDIT DATA PACKAGE
DESIGN AND ANALYSIS
DESIGN AUDIT (OCT 66)
PREPARE ADVANCED LIST OF MATERIALS FOR ENGINEERING MODEL NO. 1
ADVANCE MATERIAL RELEASE FOR ENGINEERING MODEL NO. 1 AND TO CONVERT BREADBOARD
UPDATE BREADBOARD
FABRICATE CRITICAL BREADBOARD
FABRICATE BREADBOARD SUBSYSTEM TESTER
BEGIN CRITICAL BREADBOARD TESTS
PREPARE PRELIMINARY DEVELOPMENT, ACCEPTANCE, TYPE APPROVAL TEST PLANS
PREPARE PRELIMINARY REQUIREMENTS ON SUBSYSTEM TESTER
PREPARE PRELIMINARY SPECIFICATIONS ON UNIT AND SUBSYSTEM TEST EQUIPMENT

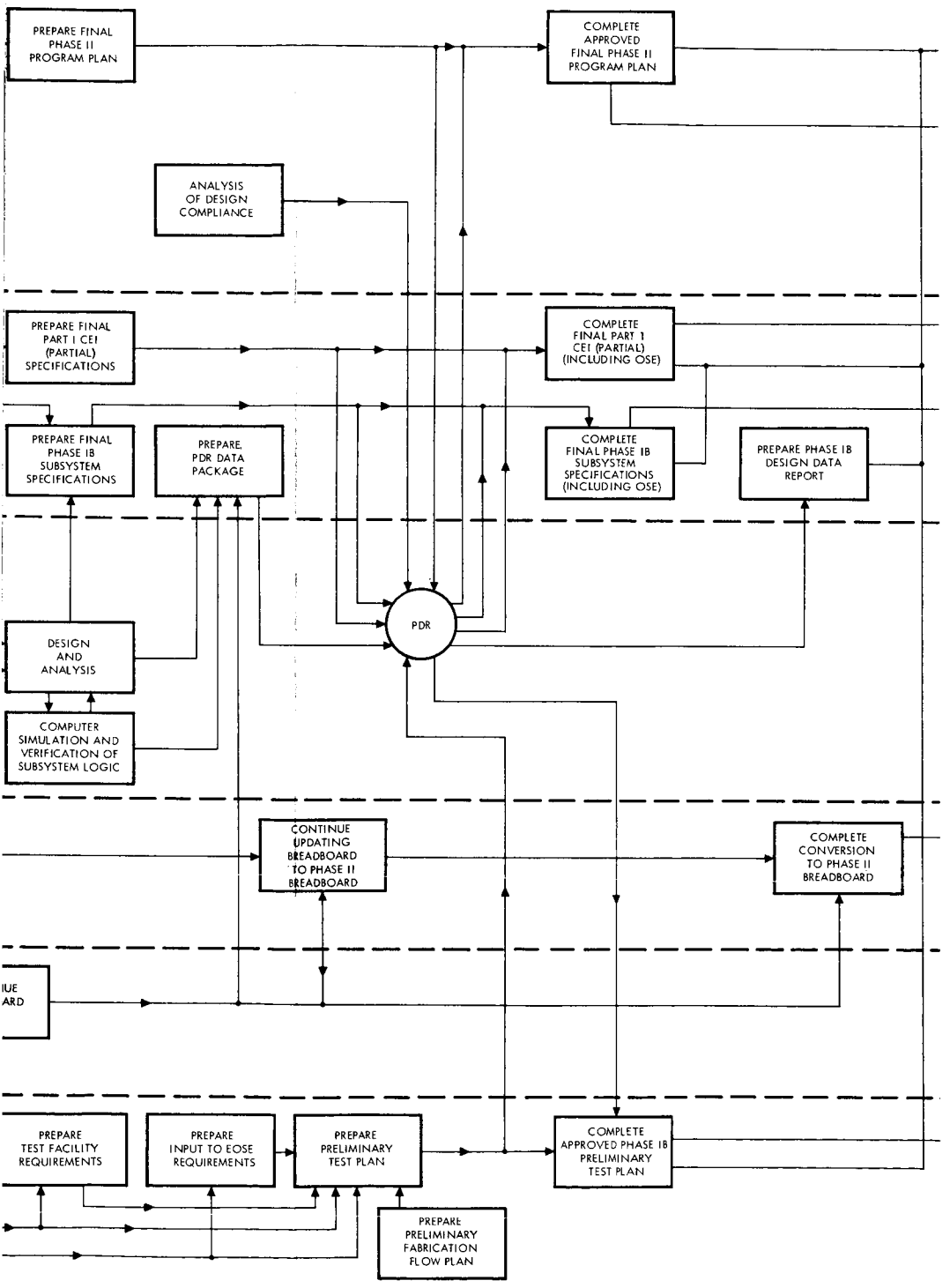
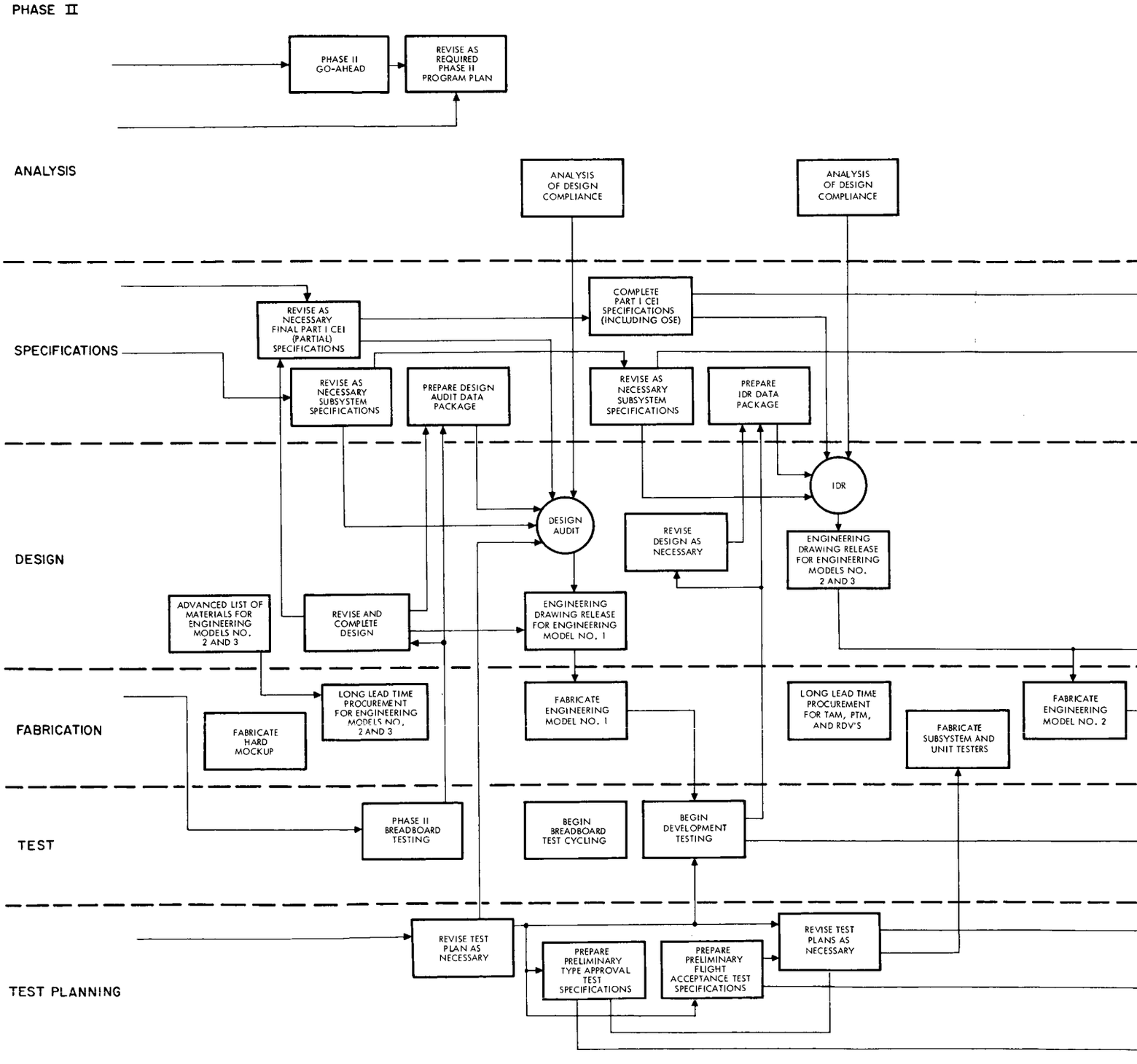
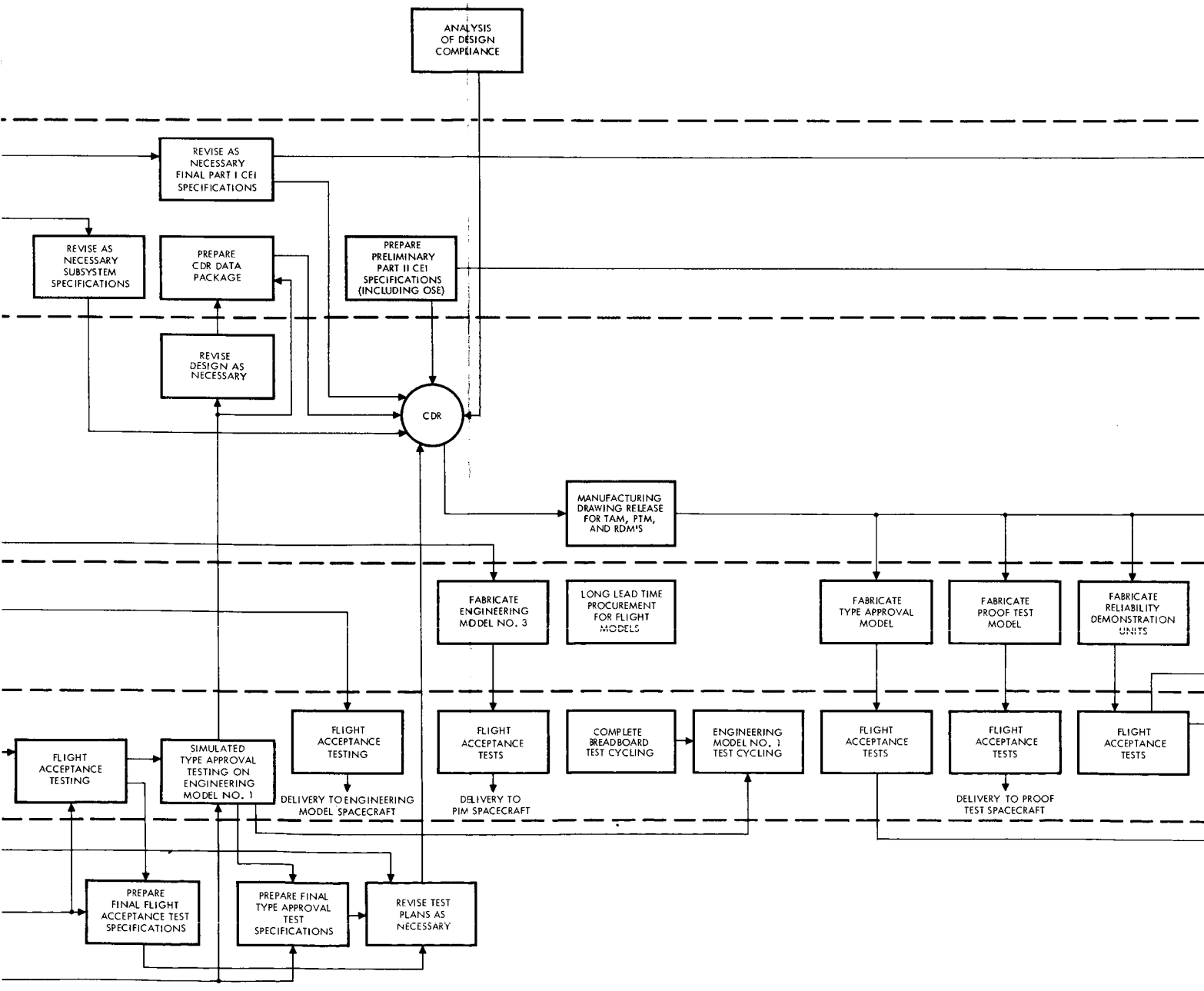


Figure 19. Design and Development of Command Subsystem

3


PHASE II
PHASE II GO-AHEAD
REVISE AS REQUIRED PHASE II PROGRAM PLAN
ANALYSIS
ANALYSIS OF DESIGN COMPLIANCE
ANALYSIS OF DESIGN COMPLIANCE
SPECIFICATIONS
REVISE AS NECESSARY FINAL PART I CEI (PARTIAL) SPECIFICATIONS
COMPLETE PART I CEI SPECIFICATIONS (INCLUDING OSE)
REVISE AS NECESSARY SUBSYSTEM SPECIFICATIONS
PREPARE DESIGN AUDIT DATA PACKAGE
REVISE AS NECESSARY SUBSYSTEM SPECIFICATIONS
PREPARE IDR DATA PACKAGE
IDR
DESIGN
DESIGN AUDIT
REVISE DESIGN AS NECESSARY
ENGINEERING DRAWING RELEASE FOR ENGINEERING MODELS NO. 2 AND 3
ADVANCED LIST OF MATERIALS FOR ENGINEERING MODELS NO. 2 AND 3
REVISE AND COMPLETE DESIGN
ENGINEERING DRAWING RELEASE FOR ENGINEERING MODEL NO. 1
FABRICATION
LONG LEAD TIME PROCUREMENT FOR ENGINEERING MODELS NO. 2 AND 3
FABRICATE HARD MOCKUP
FABRICATE ENGINEERING MODEL NO. 1
LONG LEAD TIME PROCUREMENT FOR TAM, PTM, AND RDV'S
FABRICATE SUBSYSTEM AND UNIT TESTERS
FABRICATE ENGINEERING MODEL NO. 2
TEST
PHASE II BREADBOARD TESTING
BEGIN BREADBOARD TEST CYCLING
BEGIN DEVELOPMENT TESTING
TEST PLANNING
REVISE TEST PLAN AS NECESSARY
PREPARE PRELIMINARY TYPE APPROVAL TEST SPECIFICATIONS
PREPARE PRELIMINARY FLIGHT ACCEPTANCE TEST SPECIFICATIONS
REVISE TEST PLANS AS NECESSARY

PHASE II


ANALYSIS OF DESIGN COMPLIANCE
REVISE AS NECESSARY FINAL PART I CEI SPECIFICATIONS
REVISE AS NECESSARY SUBSYSTEM SPECIFICATIONS
PREPARE CDR DATA PACKAGE
PREPARE PRELIMINARY PART II CEI SPECIFICATIONS (INCLUDING OSE)
REVISE DESIGN AS NECESSARY
CDR
MANUFACTURING DRAWING RELEASE FOR TAM, PTM, AND RDM'S
FABRICATE ENGINEERING MODEL NO. 3
LONG LEAD TIME PROCUREMENT FOR FLIGHT MODELS
FABRICATE TYPE APPROVAL MODEL
FABRICATE PROOF TEST MODEL
FABRICATE RELIABILITY DEMONSTRATION UNITS
FLIGHT ACCEPTANCE TESTING
SIMULATED TYPE APPROVAL TESTING ON ENGINEERING MODEL NO. 1
FLIGHT ACCEPTANCE TESTING
DELIVERY TO ENGINEERING MODEL SPACECRAFT
FLIGHT ACCEPTANCE TESTS
DELIVERY TO PIM SPACECRAFT
COMPLETE BREADBOARD TEST CYCLING
ENGINEERING MODEL NO. 1 TEST CYCLING
FLIGHT ACCEPTANCE TESTS
FLIGHT ACCEPTANCE TESTS
DELIVERY TO PROOF TEST SPACECRAFT
FLIGHT ACCEPTANCE TESTS
PREPARE FINAL FLIGHT ACCEPTANCE TEST SPECIFICATIONS
PREPARE FINAL TYPE APPROVAL TEST SPECIFICATIONS
REVISE TEST PLANS AS NECESSARY

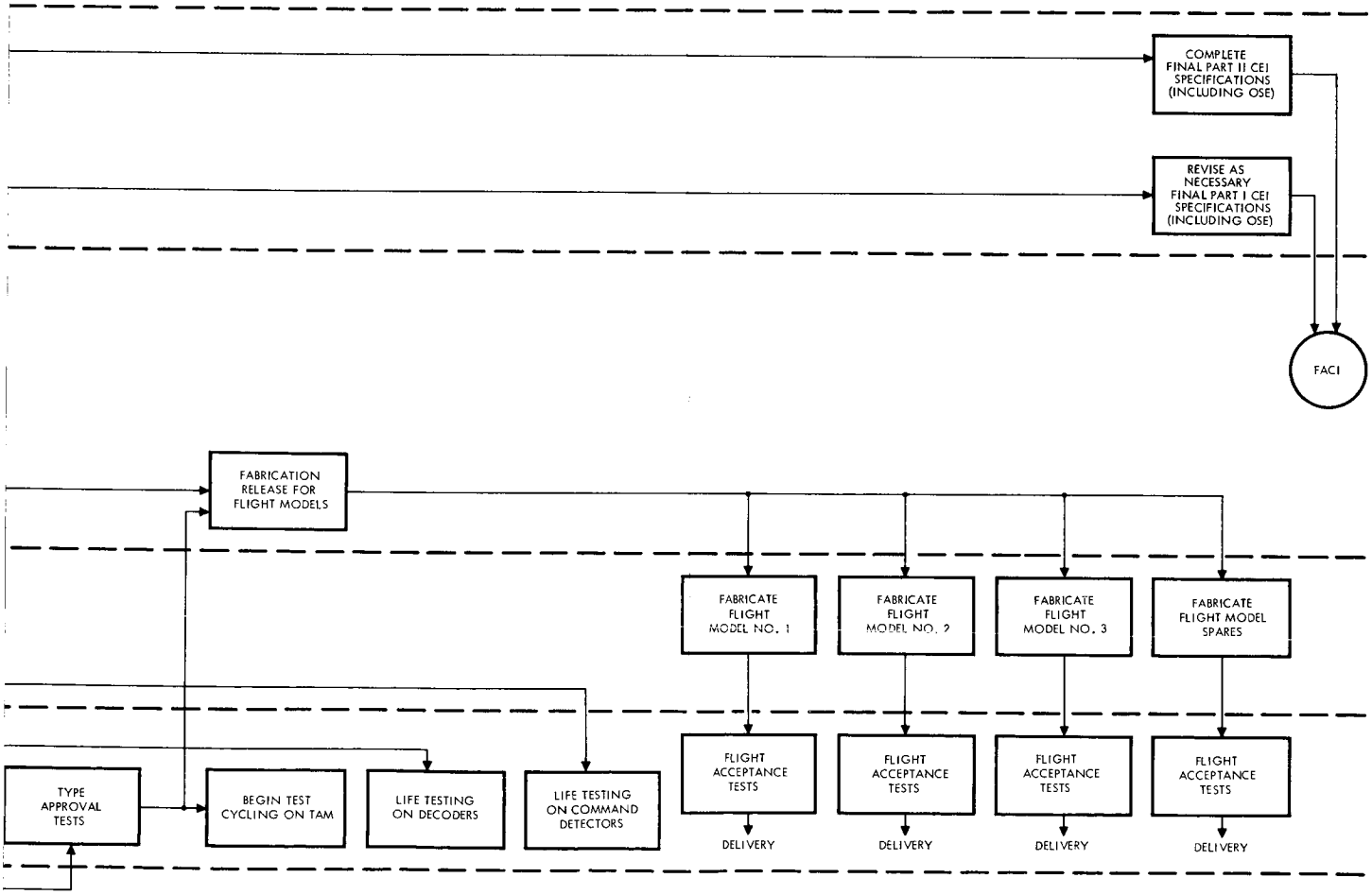


Figure 19. Design and Development of Command Subsystem (Continued)

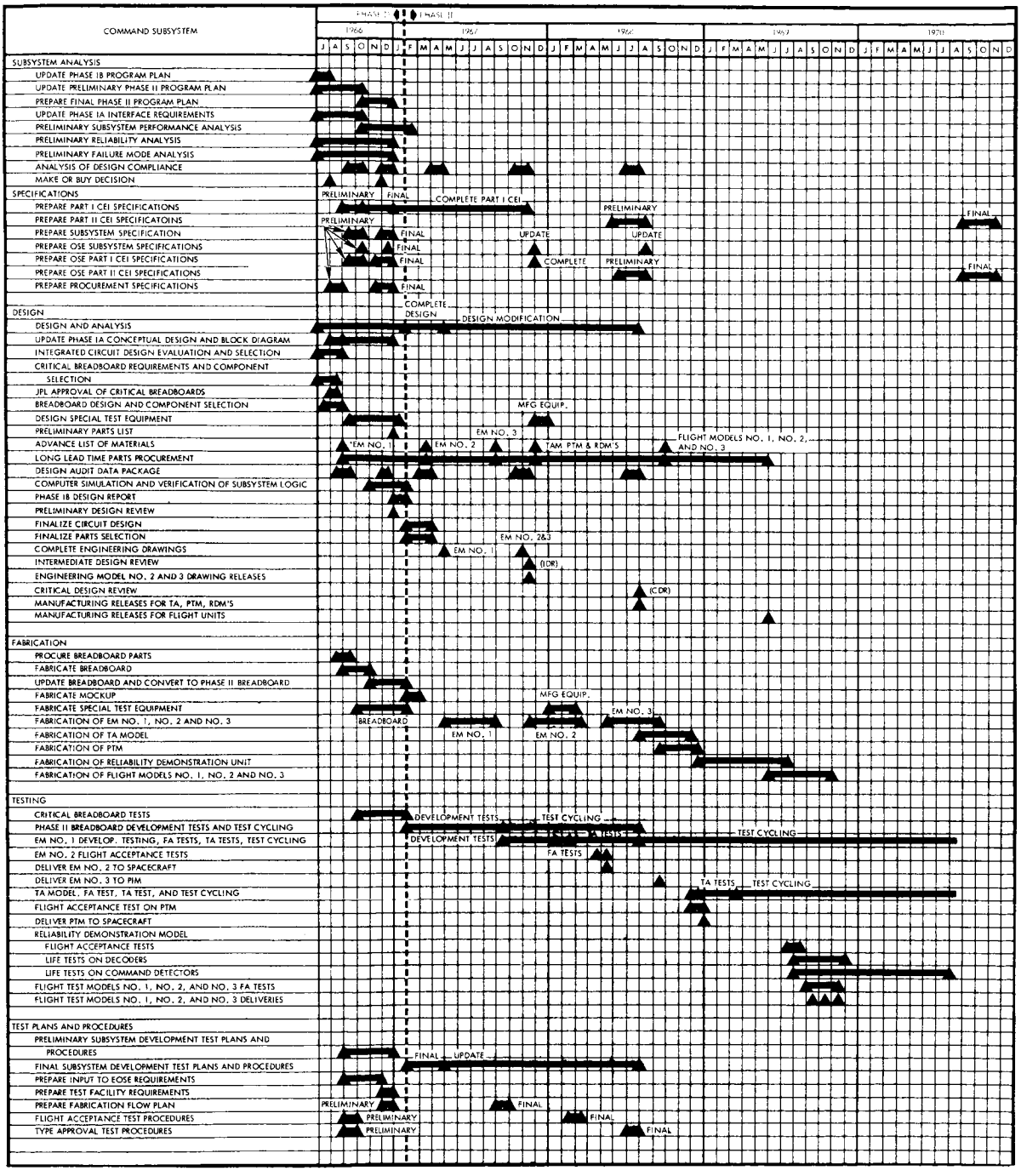


Figure 20. Schedule for Command Subsystem

During Phase IB the subsystem requirements are analyzed in detail, and the command list, logic design, circuit design, and the over-all packaging concept are developed in sufficient detail to permit the completion of the Phase II breadboard by the beginning of Phase II. Design audits in the 6th, 15th, and 26th weeks provide an opportunity to review the subsystem requirements and to determine whether the preliminary design concepts fulfill the requirements. A formal design review (PDR) in the 6th month integrates the results of design and analysis, examines the data of the critical breadboard tests, reviews the development test plans and over-all subproject plans, and provides direction for Phase II operations.

During Phase II detailed design is completed and engineering and flight models are fabricated and tested. In this phase there are essentially three design stages. The first ends at the design audit 3 months after the beginning. The second ends with the intermediate design review (IDR) and the third with the critical design review (CDR).

During the first stage the design is completed (except for changes that will be required as a result of development, simulated type approval, and type approval tests) and engineering drawings for EM No. 1 are released.

In the second stage results from development tests on EM No. 1 are utilized to revise the design as necessary before IDR and release of engineering drawings for EM No. 2 and 3.

In the third stage results from simulated type approval tests on EM No. 1 are utilized to revise the design as necessary before CDR and the release of manufacturing drawings for the type approval, proof test, and reliability demonstration models. After completion of type approval testing, fabrication release occurs for flight models and spares.

b. Schedule

The schedule for the command subsystem development program shown in Figure 20 is based upon a go-ahead date of 1 July 1966 and a production drawing release (for flight models) date of 1 June 1969. An

additional schedule restraint is the initiation of Phase II at 1 February 1967. All hardware, except for early critical breadboards, and the conversion of the critical breadboards into a Phase II breadboard, is fabricated in Phase II of the program.

3.6.3 Design, Development, and Fabrication

Most of the planned development program applies equally well to both the command detector and the decoder even though the former is primarily a communications type unit using analog equipment techniques while the latter uses digital equipment techniques. Thus, a general discussion applicable to both will be given below, followed by brief subsections indicating the main aspects of the development program peculiar to each.

a. Phase IB

Phase IB will be initiated by updating the subsystem program plan for Phase IB. This program plan will then serve to guide the efforts throughout Phase IB. Other initial efforts will update the Phase IA subsystem and interface requirements as the basis for preliminary Part I, CEI subsystem specifications.

Based upon the updated subsystem and interface requirements, the conceptual design and over-all block diagram (the general configuration) for the subsystem developed during Phase IA will be updated. This will furnish a major input to the design and analysis efforts.

An evaluation of various integrated circuits will be made at the start of Phase IB to permit an early selection of the general types of components and circuits to be used. This will permit more detailed design and analysis to begin earlier, and the timely procurement of long lead time high reliability components to be accomplished for EM No. 1 and the Phase II breadboard. After the conceptual design has been updated make-buy decisions will be made.

The design and analysis efforts will include logical design, circuit design, packaging concepts and design, failure mode analysis and design to avoid single failures causing catastrophic results, reliability

estimating, redundancy analysis and selection, performance analysis, estimation of the power, weight, thermal, magnetic and volume characteristics, and tradeoff analysis to permit optimum selection of design parameters.

Another initial effort will be the determination of the critical breadboards required. As indicated below, these are primarily for the command detector. In parallel with obtaining JPL approval of them, critical breadboard design and component selection will begin. After JPL approval, the procurement of breadboard parts and the construction of the breadboard will occur. It is planned to update the critical breadboard to convert it into a Phase II breadboard by the time Phase II begins.

A breadboard subsystem tester will also be constructed during Phase IB based upon preliminary requirements developed early in the program.

For the first design audit (as well as for PDR, IDR, CDR) a data package will be prepared containing results of tests and of design and analysis other than the inputs specifically shown on the flow diagram. These other inputs for the first design audit are:

- The Part I CEI specifications and subsystem specifications
- The preliminary Phase IB subsystem description
- An analysis of the design for compliance with the specifications.

During Phase IB there are essentially two design stages after the requirements design audit at the end of the first six weeks. The first ends at the design audit in October 1966 (15 weeks), the second at the PDR. The PDR is followed by a 2 week period in which any changes as a result of the review may be incorporated in the outputs of Phase IB.

The efforts conducted in the second design stage are similar to those in the first except that a computer simulation of the logic of the decoder is conducted for verification purposes, more test results are available, and considerable work is done on the preliminary test plan.

b. Phase II

During Phase II there are essentially three design stages. The first ends at the design audit 3 months after the beginning of Phase II, the second ends with the IDR and the third with the CDR.

During the first design stage the design is completed (except for changes that will be required as a result of development, simulated type approval, and type approval tests) and engineering drawings for EM No. 1 are released.

In the second stage results from development tests on EM No. 1 are utilized to revise the design as necessary before IDR and release of engineering drawings for EM No. 2 and 3.

In the third stage results from simulated type approval tests on EM No. 1 are utilized to revise the design as necessary before CDR and the release of manufacturing drawings for the type approval, proof test, and reliability models. If any troubles occur in the actual type approval testing a fourth design stage would occur covering necessary modifications before the release for flight models and spares.

c. Command Detector.

The following are the major features of the development program unique to the command detector. Acquisition time, efficiency both in communication power requirements and equipment complexity, and probability of acquiring will be analyzed. In addition, a study will be required on the command sync acquisition for the basic frequency offset technique and the pull-in characteristics in ambient noise. Analytical verification of the one-third probability of acquisition for the Mariner C will be attempted. Since the command sync acquisition time is so long, a better understanding of the mechanism is needed.

An extended analysis of the effects of filters on pseudo-noise synchronization will be performed. This study will more exactly develop the degradation in the pseudo-noise sync loop error function caused by various bandwidth restrictions. The carrier tracking loop introduces a high-pass characteristic ("droop") in the pulses. DSIF bandpass filters and spacecraft filters prior to the command detector cause rounding of

the square waves. Estimates of these effects are required to establish more realistic sync loop thresholds, in-lock detector threshold, and probability of false acquisition.

A study will be performed for the optimization of power division between data and sync in the command link. Present practice is to allocate power on the basis of the data requirements and a somewhat arbitrary sync threshold. A better approach is to minimize the total power required for both channels for a given data bit error rate performance. The best power split between data and sync will be determined such that the data performance is optimized regardless of any arbitrary sync threshold.

In support of the analysis of pseudo-noise synchronization acquisition, some experimental laboratory testing will be required to investigate: a) acquisition with respect to possible distortion by the phase-lock loop bandwidth under strong signal conditions, b) obtaining two or perhaps four pseudo-noise code sequences which are orthogonal to within small enough cross correlation, and c) obtaining bit error rates as low as 10^{-6}.

d. Decoder

The major features of the development program unique to the decoder are: a) obtaining long lead-time high-reliability components, b) making changes to the Phase II breadboard because of the printed circuit interconnections used, and c) the use of digital computer simulation of the logical design to verify it prior to fabrication.

3.6.4 Test Program

The test program will be initiated by breadboard tests on critical portions of the subsystem, principally covering experimental support to command detector analysis on the problems involved in: a) obtaining two or perhaps four pseudo-noise code sequences which are orthogonal to within small enough cross correlation; b) obtaining bit error rates as low as 10^{-6}.

The critical breadboards will be updated into Phase II breadboards to permit development tests on a full breadboard to begin at the initiation of Phase II. This will permit release of engineering drawings for EM

No. 1 three months after the beginning of Phase II. This early fabrication of EM No. 1 is required because of schedule constraints resulting from a) long lead-times involved in obtaining high-reliability components for the units throughout the development program; b) the importance of obtaining several months development testing on EM No. 1 before releasing engineering drawings for EM No. 2 and 3; c) the importance of completing simulated type approval testing on EM No. 1 before releasing manufacturing drawings for the type approval model which is the model used for type approval tests; and d) the requirements to complete type approval testing before releasing production drawings for the flight models.

The engineering model will be packaged identically to the flight models, thus permitting comprehensive developmental, flight acceptance, and simulated type approval tests to be conducted on it.

The command subsystem development, type approval, and flight acceptance test program is summarized in Table 19.

3.7 Telemetry Subsystem

3.7.1 Description

The telemetry subsystem consists of two redundant PCM encoders, each of which has, as major subelements, the following:

- Analog multiplexer and sequencer
- Analog-to-digital converter
- Pseudo-noise (PN) generator
- Digital multiplexer
- Modulator-mixer
- Capsule data buffer.

The multiplexer sequentially samples the analog data inputs, presenting them to the analog-to-digital converter for translation into a 7-bit digital word. Digital outputs of the analog-to-digital converter are applied to the digital multiplexer, as are all digital data, capsule data, the real-time science data, and the outputs of the data storage subsystem.

Table 19. Command Subsystem Development Tests

Name of Test	Item Being Tested	Purpose	Description	Special Test Equipment
Input Decoder Logic Test	Logical equations	Verify logical analysis and design	Logical equation test; bit-by-bit simulation of the operations of the input decoder	Logic equation simulator program and high speed computer
Output Decoder Logic Test	Logical equations	Verify logical analyses and design	Logical equations test; bit-by-bit simulation of the operations of the output decoder	Logic equation simulator program and high-speed computer
Pseudo-Noise Code Sequence Cross Correlation	Critical breadboard unit	To aid in determining whether two or four orthogonal codes can be obtained with sufficiently low cross correlation properties	To be determined	
Bit Error Rate for Command Detector	Critical breadboard unit	To aid in determining whether it is feasible to design for a bit error rate as low as 10^{-6}	Continual operation of the command detector with a subcarrier input with simulated noise	Unit tester
Breadboard Development Tests	Phase II breadboard	Perform an engineering evaluation of preliminary circuit and subsystem design. Provide data on normal operation, interface compatibility, susceptibility to transients and power frequency variations, load variations, temperature extremes, combined out of limit conditions with various failure modes induced.	The circuits will be operated under normal conditions using a simulated power source and load. The various parameters identified will be varied through a range in excess of anticipated type approval ranges and performance margin will be determined.	Subsystem and unit tester
Engineering Model Development Tests	EM No. 1	Same as the breadboard test except on the packaged model. In addition, provide data on electromagnetic compatibility, magnetic compatibility, vibration, and subsystem integration at T.A. levels.	In addition to testing described above, tests on conducted and radiated interference and measurements of magnetic stability and induced fields will be performed. Operation in an out of limit vibration environment will be determined.	Subsystem and unit tester
Flight Acceptance Test	EM's, type approval model, proof test model, reliability demonstration models, flight subsystems	Confirm acceptable operation on a unit and subsystem level.	Functional operation will be verified under a variety of induced conditions, such as voltage and power frequency variations, noise injection, temperature variations, etc. Magnetic testing will be accomplished as will vibration and thermal vacuum testing. Functional operation will be reverified. Performance as a subsystem will be determined.	Subsystem and unit tester
Type Approval Test	Type approval model	Confirm satisfaction of design requirements under environmental conditions in excess of those anticipated in flight.	Flight acceptance testing will be accomplished. This will be followed by subjection on a unit level to temperature variations above and below flight acceptance values, 3-axes complex vibration, shock, humidity, thermal shock, and sterilization compatibility. On a subsystem level, EMC and thermal vacuum testing will be performed as well as power supply variations.	Subsystem and unit tester
Life Cycling Tests	Type approval model, Phase II breadboard, EM No. 1	Confirm operability over extended period of time.	Subsystem will be operated continuously with functional operation verification every 2 months at extreme temperatures and power levels, and with noise injection, power frequency variation, power overshoot, and power transients	Subsystem and unit tester
Reliability Demonstration Tests	Command detector reliability demonstration units	To conduct reliability demonstration tests in accordance with the Voyager Reliability Program Plan	Four units will be operated under 40°C ambient temperature for 12 months	
Reliability Demonstration Tests	Decoder reliability demonstration units	To conduct reliability demonstration tests in accordance with the Voyager Reliability Program Plan	Six units will be operated under 30°C ambient temperature and vacuum conditions for 6 months.	

The digital multiplexer formats the telemetry data. The PN generator provides a 63-bit binary sequence for ground-station bit synchronization, which is combined with the serial data stream in the modulator-mixer. The resultant waveform is applied to the radio subsystem for transmission to earth.

3.7.2 Summary

a. Development Plan

Development of the telemetry subsystem is illustrated in the telemetry subsystem development flow chart, Figure 21.

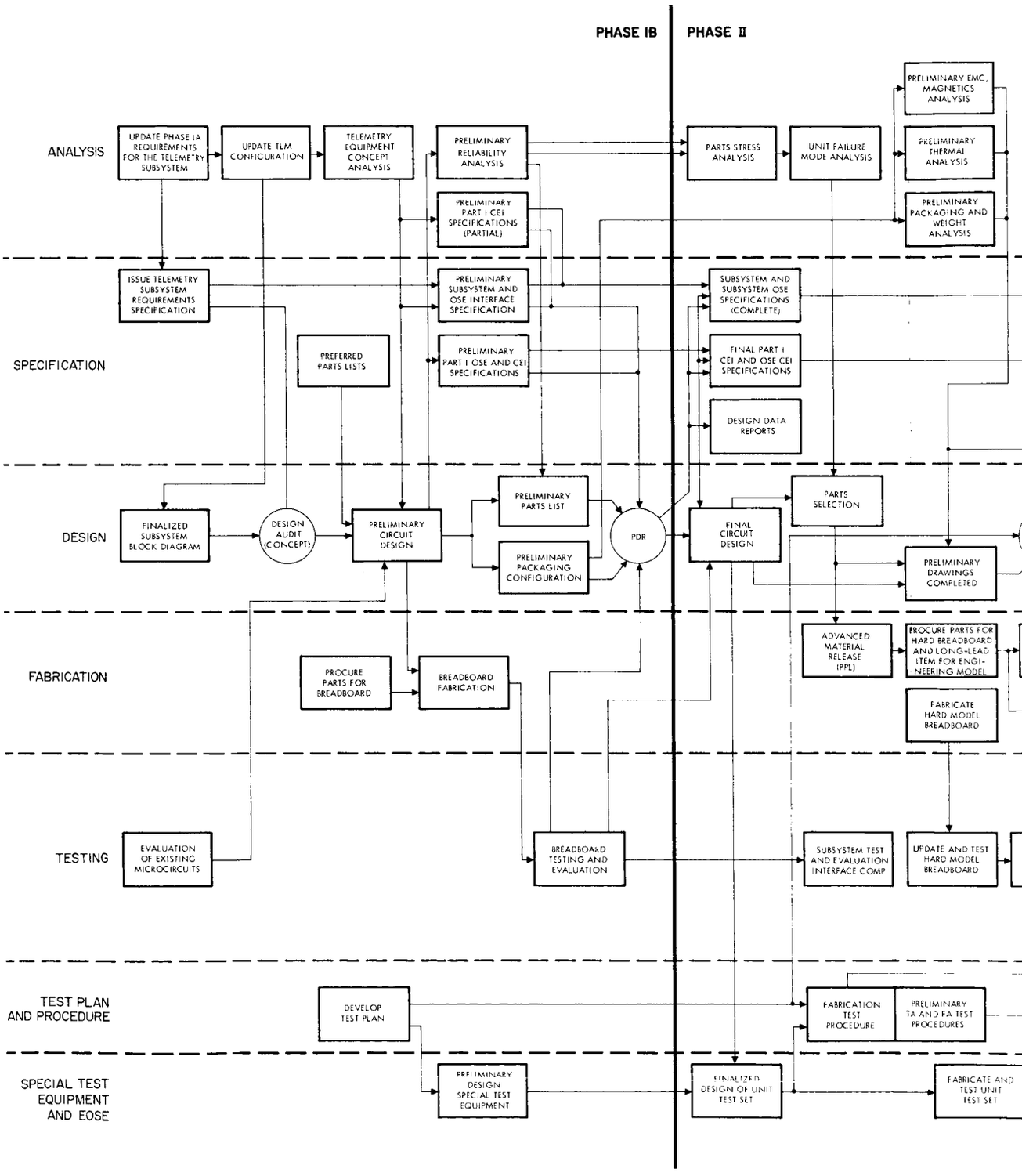
Phase IB activity will be directed toward the completion of a subsystem logic design. Interfaces with data storage, capsule and science will be defined. Modes, sampling rates, data rates, and formats will be established, based upon updated information.

Phase II development activity will be directed toward producing an engineering model subsystem. Breadboarding, detailed circuit design and testing are major activities in this period.

b. Schedule

The telemetry subsystem milestone schedule is shown in Figure 22. Phase IB will start with a review of the requirements generated in Phase IA, and modified by new mission requirements. Available information regarding capsule and science interfaces will be reviewed for their effect on the system logic diagram. Completion of the logic diagram leads to a design audit, verifying the conceptual design. Standard logic circuits will be investigated to determine applicability, and preliminary design and breadboards will be initiated for special circuit requirements. A preliminary subsystem specification and OSE specification will be completed, together with preliminary Part I CEI specifications for the subsystem units and OSE, in October 1966. Phase IB is concluded with a preliminary design review and preparation of design data reports and specifications.

Phase II efforts start by finalizing the subsystem logic design, releasing updated subsystem and OSE specification and final Part I


PHASE IB
PHASE II
ANALYSIS
SPECIFICATION
DESIGN
FABRICATION
TESTING
TEST PLAN AND PROCEDURE
SPECIAL TEST EQUIPMENT AND EOSE
UPDATE PHASE IA REQUIREMENTS FOR THE TELEMETRY SUBSYSTEM
UPDATE TLM CONFIGURATION
TELEMETRY EQUIPMENT CONCEPT ANALYSIS
PRELIMINARY RELIABILITY ANALYSIS
PRELIMINARY PART I CEI SPECIFICATIONS (PARTIAL)
PARTS STRESS ANALYSIS
UNIT FAILURE MODE ANALYSIS
PRELIMINARY EMC, MAGNETICS ANALYSIS
PRELIMINARY THERMAL ANALYSIS
PRELIMINARY PACKAGING AND WEIGHT ANALYSIS
ISSUE TELEMETRY SUBSYSTEM REQUIREMENTS SPECIFICATION
PRELIMINARY SUBSYSTEM AND OSE INTERFACE SPECIFICATION
SUBSYSTEM AND SUBSYSTEM OSE SPECIFICATIONS (COMPLETE)
PREFERRED PARTS LISTS
PRELIMINARY PART I OSE AND CEI SPECIFICATIONS
FINAL PART I CEI AND OSE CEI SPECIFICATIONS
DESIGN DATA REPORTS
FINALIZED SUBSYSTEM BLOCK DIAGRAM
DESIGN AUDIT (CONCEPT)
PRELIMINARY CIRCUIT DESIGN
PRELIMINARY PARTS LIST
PRELIMINARY PACKAGING CONFIGURATION
PDR
FINAL CIRCUIT DESIGN
PARTS SELECTION
PRELIMINARY DRAWINGS COMPLETED
ADVANCED MATERIAL RELEASE (PPL)
PROCURE PARTS FOR HARD BREADBOARD AND LONG-LEAD ITEM FOR ENGINEERING MODEL
FABRICATE HARD MODEL BREADBOARD
PROCURE PARTS FOR BREADBOARD
BREADBOARD FABRICATION
EVALUATION OF EXISTING MICROCIRCUITS
BREADBOARD TESTING AND EVALUATION
SUBSYSTEM TEST AND EVALUATION INTERFACE COMP
UPDATE AND TEST HARD MODEL BREADBOARD
DEVELOP TEST PLAN
FABRICATION TEST PROCEDURE
PRELIMINARY TA AND FA TEST PROCEDURES
PRELIMINARY DESIGN SPECIAL TEST EQUIPMENT
FINALIZED DESIGN OF UNIT TEST SET
FABRICATE AND TEST UNIT TEST SET

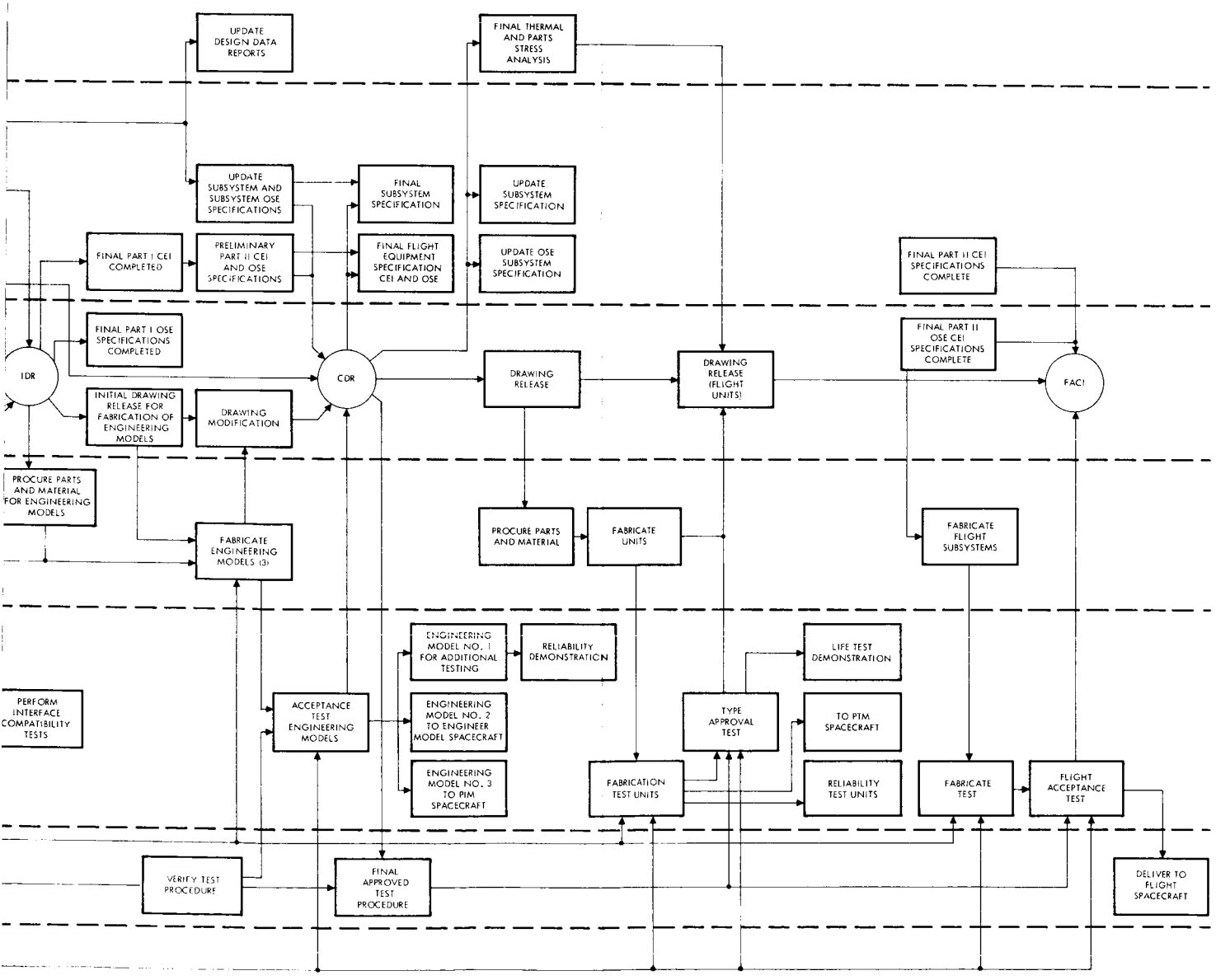


Figure 21. Design and Development of Telemetry Subsystem

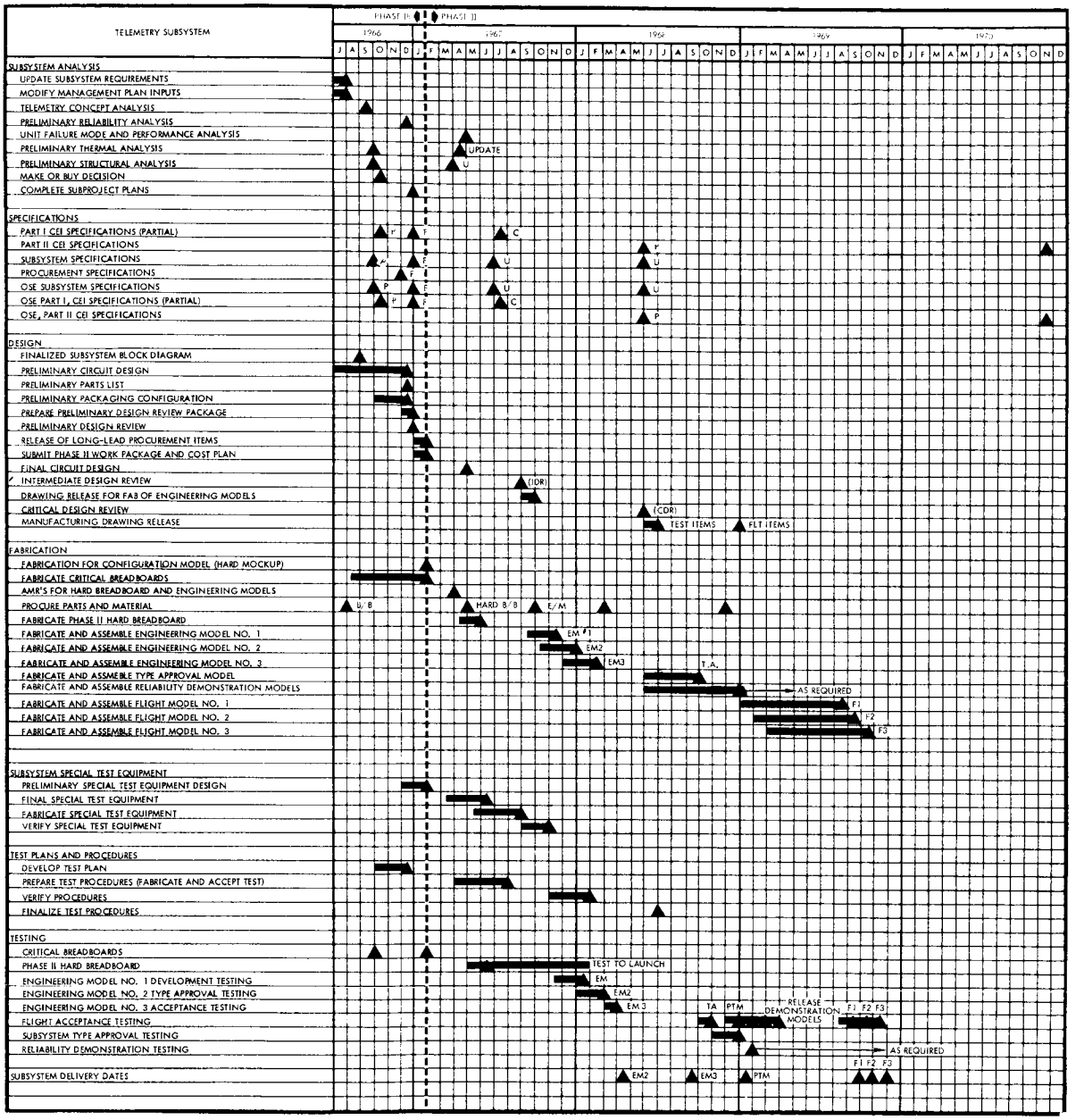


Figure 22. Schedule for Telemetry Subsystem

(partial) specifications for the subsystem units and OSE. Next, the subsystem breadboard is fabricated and tested, thermal and structural analyses are initiated, with the detailed packaging effort starting 2 months after Phase II go-ahead. Special test equipment requirements are finalized during the early period of Phase II such that their completion schedule will lead the fabrication cycle of the engineering models

(EM). Thus, at 6 months after Phase II start, sufficient information (regarding packaging details, test plan and breadboard test results) are available to conduct the intermediate design review (IDR).

Approval from the IDR will permit completion of all drawings and subsequent release of the drawings to start fabricating the EM's. Long lead items for the EM's are procured about a month prior to the time the IDR is to be conducted.

Three EM's are fabricated and tested. These units serve to verify the electrical and mechanical design of the PCM encoder. They are also used for verifying the special test equipment as well as the test procedures. Results obtained through the evaluation of the EM's are discussed in the critical design review (CDR) which is held around 15 months after Phase II go-ahead. Satisfactory findings from this review govern the decision to release the drawings for the fabrication of the type approval (TA), and proof test models (PTM).

3.7.3 Design and Development

a. Subsystem Analysis and Design

Telemetry modes and formats will be analyzed to determine the optimum implementation, based upon up-dated engineering, science, and capsule data requirements. Data compression requirements and implementation will be studied for feasibility. Information transmission methods such as biorthogonal coding will be examined. The effect and implementation of frequency multiplexing will be considered. Subsystem reliability and redundancy method will be reviewed.

In addition, the following physical analyses will be performed: thermal, weight, packaging, magnetic properties, power, and interface. The following compatibility analyses will be performed: sterilization, electromagnetic, and ethylene oxide.

b. PCM Encloder Analysis and Design

In the encoder, a final analog multiplexer circuit must be chosen. New devices such as the enhancement mode metal-oxide, silicon-field effect transistor will be tested in breadboard circuits.

Detailed design of the encoder logic will be completed, considering the most satisfactory method of incorporating reliability into the design.

3.7.4 Test Program

Development tests will be used to verify that the over-all subsystem requirements are met. The test matrix shown in Table 20 outlines some of the basic tests that will be performed during the development of the telemetry subsystem.

During the early stages of Phase IB a selected and limited amount of module circuit breadboards will be fabricated and tested. Standard digital circuits will be evaluated, and special circuits will be tested.

A breadboard of a single PCM encoder will be the next level of test. The breadboard will be tested to verify that all performance requirements, as set forth by the equipment specification, are met. For the PCM encoder some of the operational parameters that will be verified during the test are.

- Unit operates satisfactorily with the specified input signal characteristics
- Unit maintains the required analog encoding accuracy
- Unit operates properly at the various specified bit rates
- Unit formats properly at the various modes of operation
- Unit generates all the required interface signals
- Unit generates the required output PN coded data and is capable of modulating the data transmitter.

These parameters are tested while the assembly is subjected to a combination of the following conditions:

- TA temperature extremes
- ±15 percent power supply variation
- Transients, noise and audio tones 15 percent over the type approval levels
- ± 15 percent power frequency variation from TA levels.

Next, the PCM encoder breadboard is integrated into the subsystem breadboard, and the individual PCM encoder tests repeated. In

Table 20. Telemetry Subsystem Test Matrix

Name of Test	Item Tested	Purpose and Objective	Description
Circuit Design Evalution	Circuit breadboard	Determine electrical performance to verify that design conforms to component performance specifications	Measure operation with a variation of input conditions and parameters at ambient temperature
Design Verification	Subsystem breadboard	Demonstrate that the initial design chosen will meet preliminary design specifications	Conduct tests, measuring performance, during variation of parameters: temperature, power frequency and power transients, audio and noise.
Electrical Performance and life test	Subsystem breadboard	Determine electrical performance and to verify conformance to design specifications	Measure performance, test subsystem interfaces, operate unit continuously to launch, periodically check during a variation of input parameters.
Initial Electrical Performance (fabrication)	EM No. 1, 2 and 3	Determine electrical performance to verify that engineering model conforms to the performance specifications after assembly	Measure operation with specified range of input conditions at various temperatures and supply variations
Electrical Performance (thermal-vacuum)	EM No. 1, 2 and 3	Determine the electrical performance of the unit under thermal-vacuum	Measure operation with specified range of input conditions and monitor specifically any degradation due to thermal-vacuum environment
Electrical/ Mechanical Performance (vibration)	EM No. 2 and 3	Determine the electrical and mechanical integrity of the unit under vibration	Measure the operation with specified range of input conditions during and after vibration on 3 laxis. Check visually for any mechanical anomally.
Magnetic	EM No. 2 and 3	Determine magnetic field intensity to verify within the magnetic constraints	Measure magnetic field intensity in three planes while operating in each mode.
Ethylene Oxide compatibility	EM No. 2	Verify resistance to ethylene oxide environment	Expose EM to ethylene oxide gas at specified concentration and temperature for specified period
EMC	EM No. 1 and 2	Verify conformance to EMC performance specification	Measure level of susceptibility to and generation of conducted and radiated radio frequency interference
Other Environments	EM No. 2	Verify integrity of design for qualification environments	Subject EM to acceleration, shock and humidity environment and note performance at specified range of input
Type Approval tests	TA units	Demonstrate that flight subsystem will function properly at design qualification limits	Impose TA environmental limits on a unit, monitoring operation. Vary input parameters during test.
Life	Life test unit	Demonstrate that certain critical units meet the life requirements of the Voyager mission	Test one or more flight units, simulating as much as practicable actual operating conditions to destruction or to a point where the unit no longer meets minimum performance requirements
Compatibility	EM integrated into a subsystem	Demonstrate that the complete subsystem is compatible and it meets system and mission requirements	Measure over-all subsystem performance parameters while operating in various modes under specified input conditions.
Flight Acceptance tests	All flight units	Demonstrate that flight units will meet operating requirements during flight acceptance testing exposures	Subject assemblies to flight acceptance exposures and test system functional characteristics during exposure

addition, all interfaces between assemblies of the subsystem are thoroughly tested and checked and redundancy switching is tested. Interfaces with other subsystems will be tested with either subsystem breadboards or simulated subsystems. This breadboard, after all necessary tests are completed, will be updated, powered continuously, and tested periodically. This will continue until launch of the first spacecraft.

EM's of flight configuration are used in the next level of testing. There are three EM's. The first subsystem will be subjected to the same series of tests used to verify the design in the breadboard test phase. In addition, EMC and vibration testing will be performed.

The second EM subsystem will be tested to the TA test requirements, while the third EM will be tested to flight acceptance (FA) levels. Magnetic compatibility tests will be performed on EM No. 2 and 3.

The TA subsystem will first be tested to FA requirements, then to the TA test requirements. The TA subsystem, upon completion of these tests, will undergo a life test.

3.8 Data Storage Subsystem

3.8.1 Description

The data storage subsystem contains six tape recorders, each assigned a separate function. Each recorder of the subsystem is independent of the others, with separate power, data and control circuits. Table 21 indicates the functions and some of the important parameters of each recorder.

Five of the recorders store science data and receive data, clock and start-stop commands from the science subsystem. All of the recorders are played back through the telemetry subsystem, synchronously with the data rate clock, and receive start-stop commands (for playback) from the computing and sequencing subsystem. The maneuver recorder, under C and S control, is used to store telemetry data during maneuvers.

Table 21. Data Storage Tape Recorders

	Functions				
	Television	Spectrometer	IR Scanner	Fields and Part	Maneuver
Quantity per Subsystem	2	1	1	1	1
Capacity	10^8 bits	10^7 bits	10^7 bits	10^6 bits	1.5×10^6 bits
Input Rate	90 kbps	2 kbps	5 kbps	700 bps	234 bps
Output Rate	*	*	*	*	234 bps

*15, 7.5, 3.75, 1.875 kbps

3.8.2 Summary

a. Development Plan

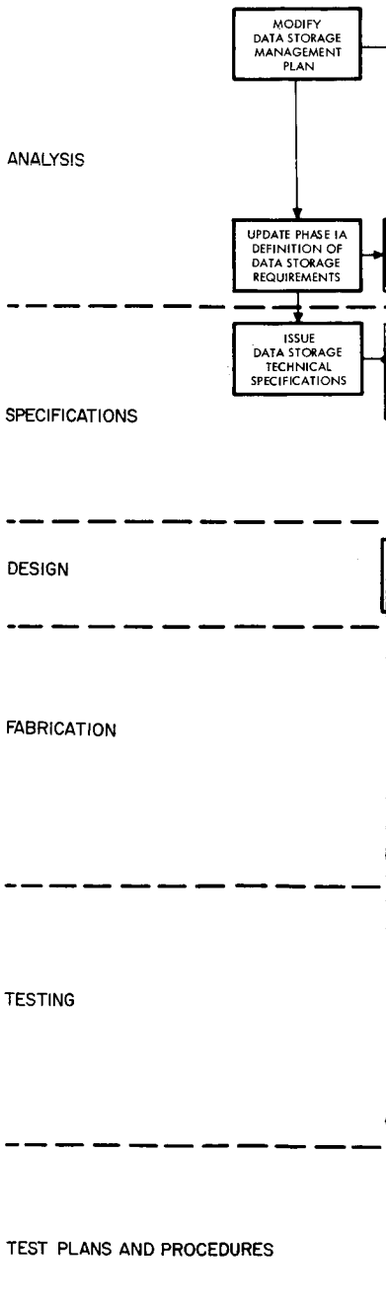
Development of the subsystem is illustrated in the flow chart of Figure 23.

Phase IB activity will be directed toward the completion of recorder specifications and the selection of a subcontractor for the fabrication of the subsystem recorders. Interfaces with telemetry and science subsystems will be defined, as will data rates and conceptual design. Some special circuit breadboarding will be performed.

Phase II development activity will be directed toward producing breadboards and an engineering model of each recorder. Detailed design, analysis and testing are major activities in this period.

b. Schedule

The subsystem milestone chart is shown in Figure 24. Phase IB will start with a review of the requirements generated in Phase IA, and modified by new mission requirements. Telemetry and science interfaces will be analyzed, and the recorder requirements finalized. A preliminary subsystem specification and OSE specification will be completed, together with preliminary Part I CEI specifications for the


MODIFY DATA STORAGE MANAGEMENT PLAN
ANALYSIS
UPDATE PHASE 1A DEFINITION OF DATA STORAGE REQUIREMENTS
ISSUE DATA STORAGE TECHNICAL SPECIFICATIONS
SPECIFICATIONS
DESIGN
FABRICATION
TESTING
TEST PLANS AND PROCEDURES

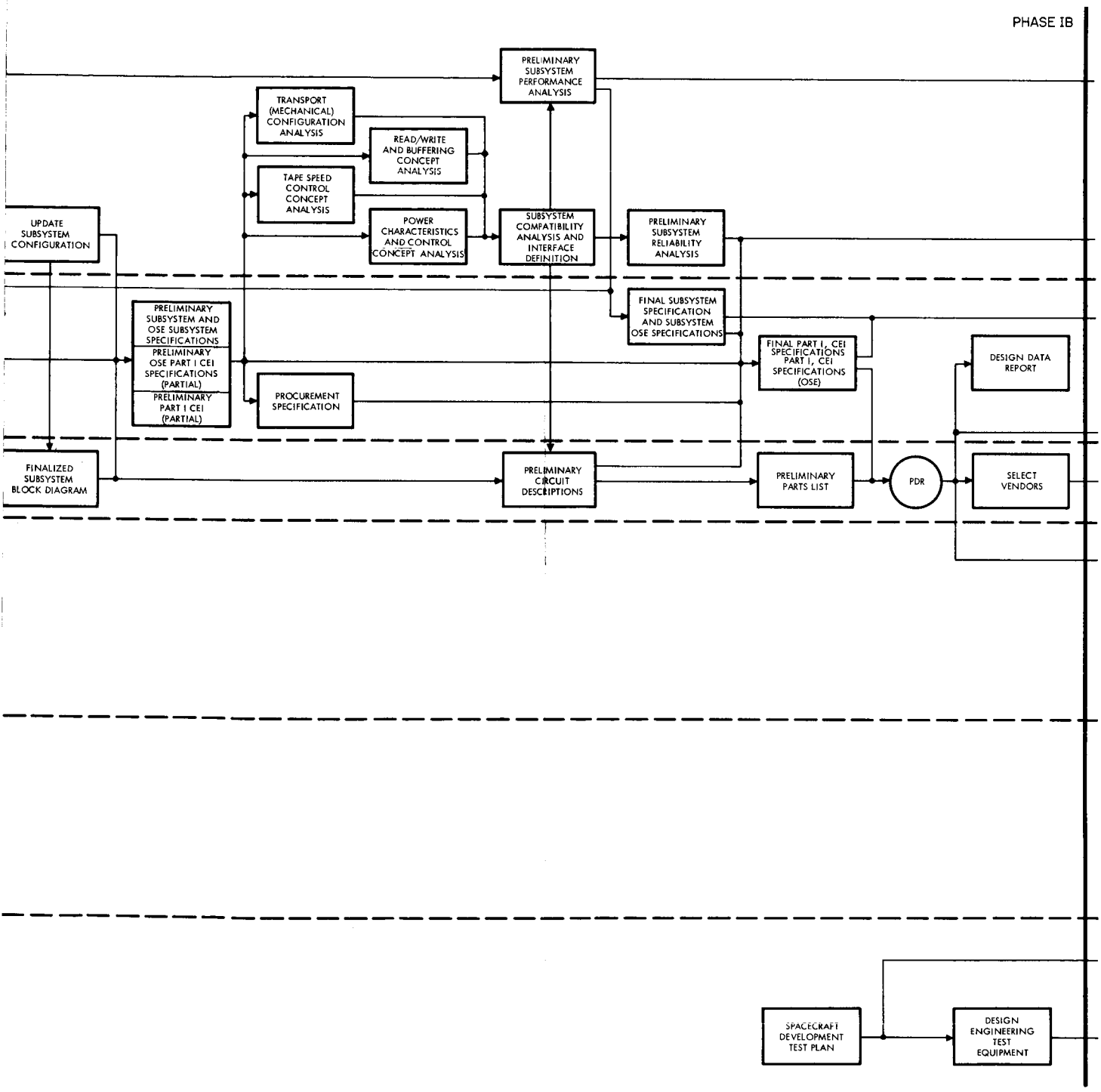
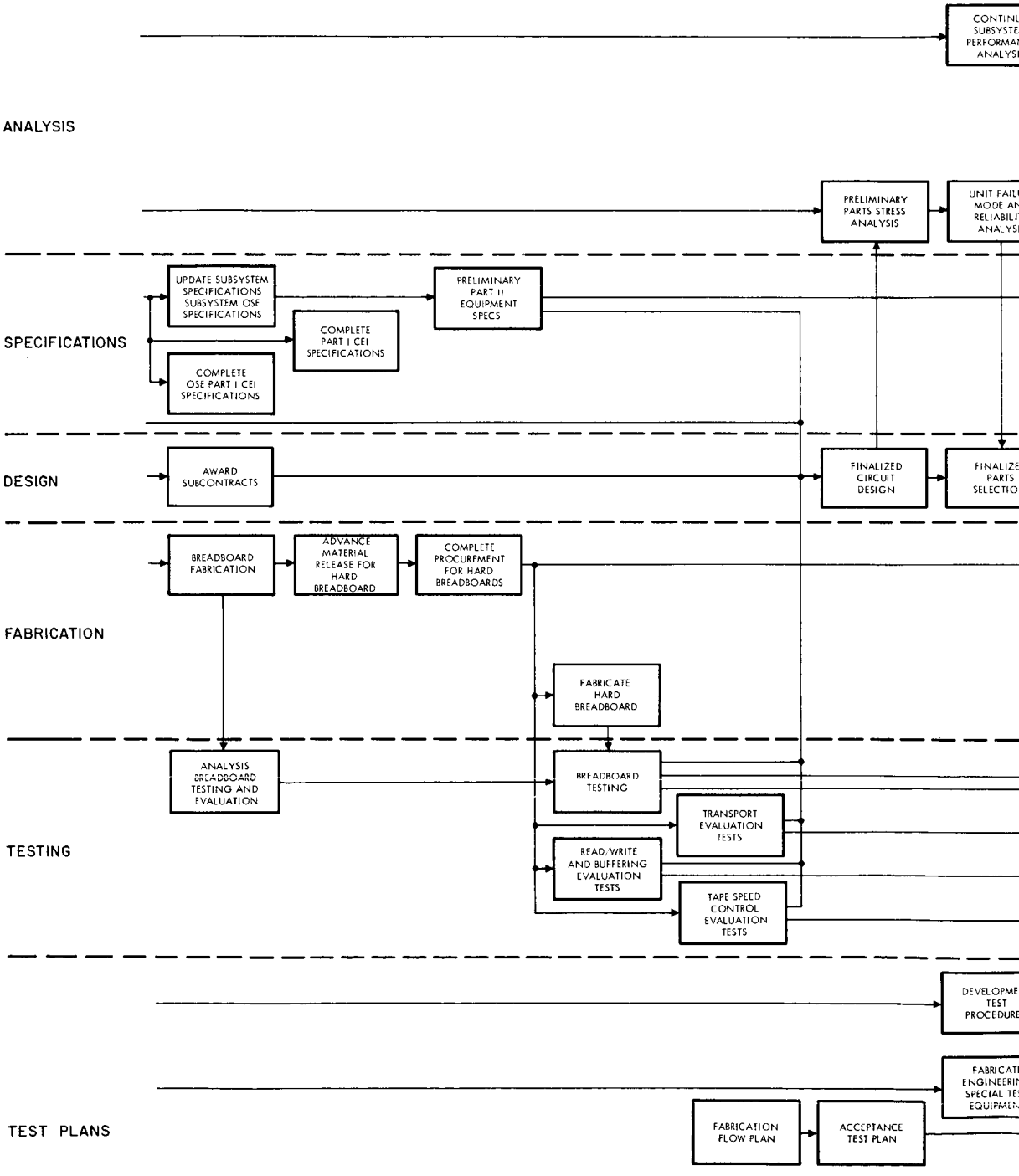
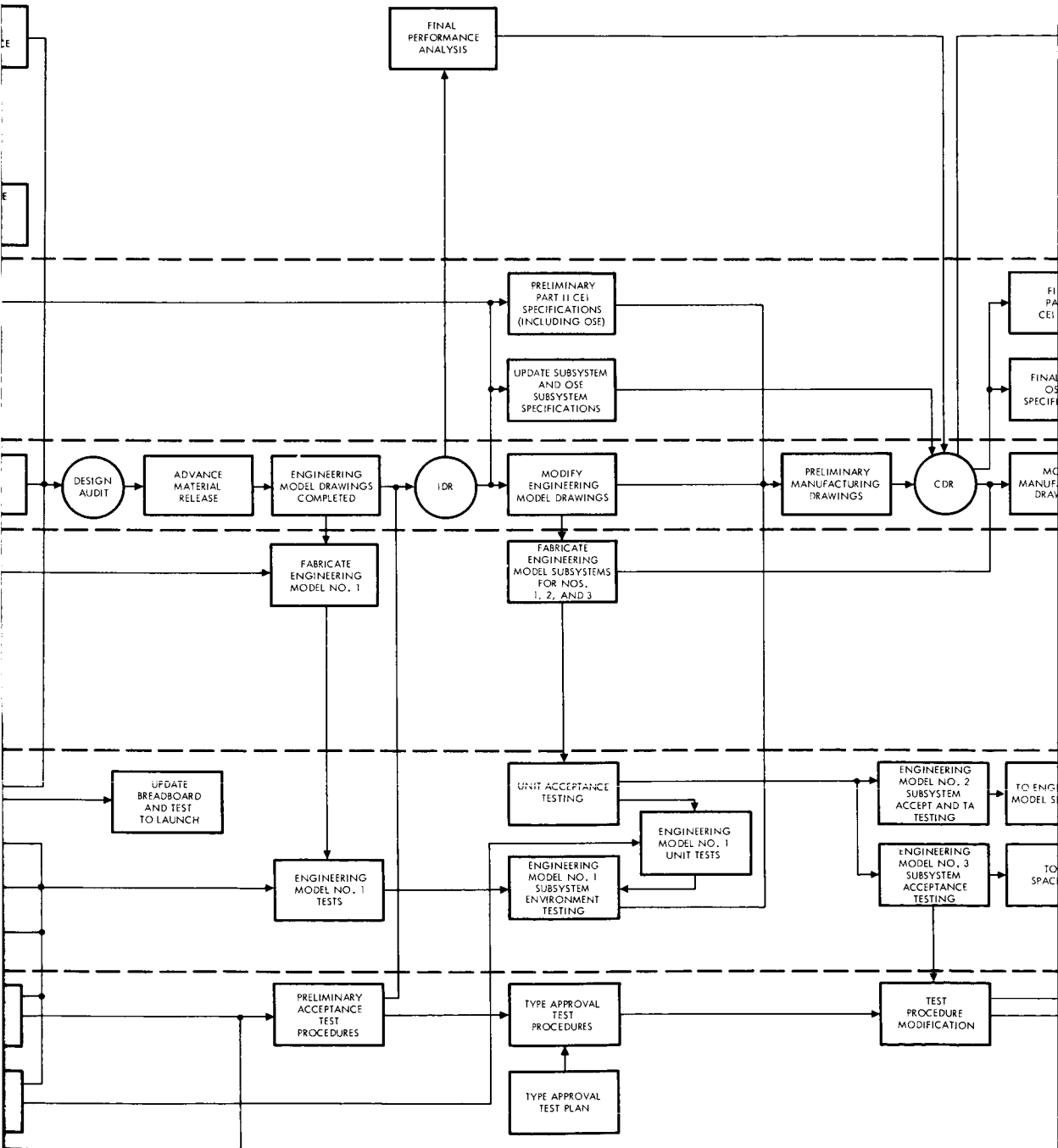


Figure 23. Design and Development of Data Storage Subsystem

PHASE II


CONTINU
SUBSYSTE
PERFORMA
ANALYSI
ANALYSIS
PRELIMINARY
PARTS STRESS
ANALYSIS
UNIT FAIL
MODE AN
RELIABILI
ANALYSI
SPECIFICATIONS
UPDATE SUBSYSTEM
SPECIFICATIONS
SUBSYSTEM OSE
SPECIFICATIONS
PRELIMINARY
PART II
EQUIPMENT
SPECS
COMPLETE
PART I CEI
SPECIFICATIONS
COMPLETE
OSE PART I CEI
SPECIFICATIONS
DESIGN
AWARD
SUBCONTRACTS
FINALIZED
CIRCUIT
DESIGN
FINALIZE
PARTS
SELECTIO
FABRICATION
BREADBOARD
FABRICATION
ADVANCE
MATERIAL
RELEASE FOR
HARD
BREADBOARD
COMPLETE
PROCUREMENT
FOR HARD
BREADBOARDS
FABRICATE
HARD
BREADBOARD
TESTING
ANALYSIS
BREADBOARD
TESTING AND
EVALUATION
BREADBOARD
TESTING
TRANSPORT
EVALUATION
TESTS
READ/WRITE
AND BUFFERING
EVALUATION
TESTS
TAPE SPEED
CONTROL
EVALUATION
TESTS
DEVELOPME
TEST
PROCEDURE
FABRICATI
ENGINEERIN
SPECIAL TE
EQUIPMEN
TEST PLANS
FABRICATION
FLOW PLAN
ACCEPTANCE
TEST PLAN


FINAL PERFORMANCE ANALYSIS
PRELIMINARY PART II CEI SPECIFICATIONS (INCLUDING OSE)
UPDATE SUBSYSTEM AND OSE SUBSYSTEM SPECIFICATIONS
DESIGN AUDIT
ADVANCE MATERIAL RELEASE
ENGINEERING MODEL DRAWINGS COMPLETED
IDR
MODIFY ENGINEERING MODEL DRAWINGS
PRELIMINARY MANUFACTURING DRAWINGS
CDR
FABRICATE ENGINEERING MODEL NO. 1
FABRICATE ENGINEERING MODEL SUBSYSTEMS FOR NOS. 1, 2, AND 3
UPDATE BREADBOARD AND TEST TO LAUNCH
UNIT ACCEPTANCE TESTING
ENGINEERING MODEL NO. 1 UNIT TESTS
ENGINEERING MODEL NO. 2 SUBSYSTEM ACCEPT AND TA TESTING
ENGINEERING MODEL NO. 1 TESTS
ENGINEERING MODEL NO. 1 SUBSYSTEM ENVIRONMENT TESTING
ENGINEERING MODEL NO. 3 SUBSYSTEM ACCEPTANCE TESTING
PRELIMINARY ACCEPTANCE TEST PROCEDURES
TYPE APPROVAL TEST PROCEDURES
TEST PROCEDURE MODIFICATION
TYPE APPROVAL TEST PLAN

2

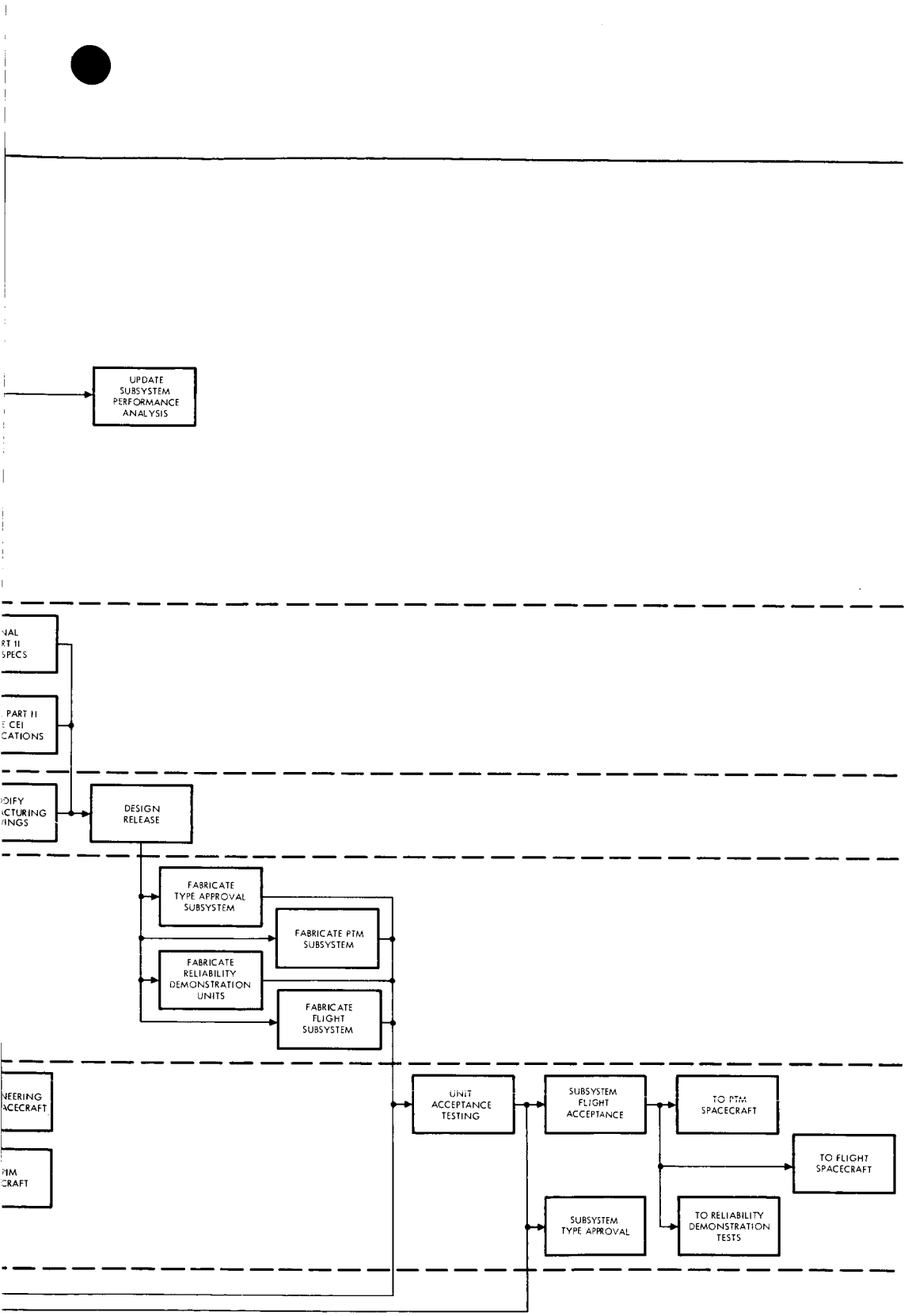


Figure 23. Design and Development of Data Storage Subsystem (Continued)

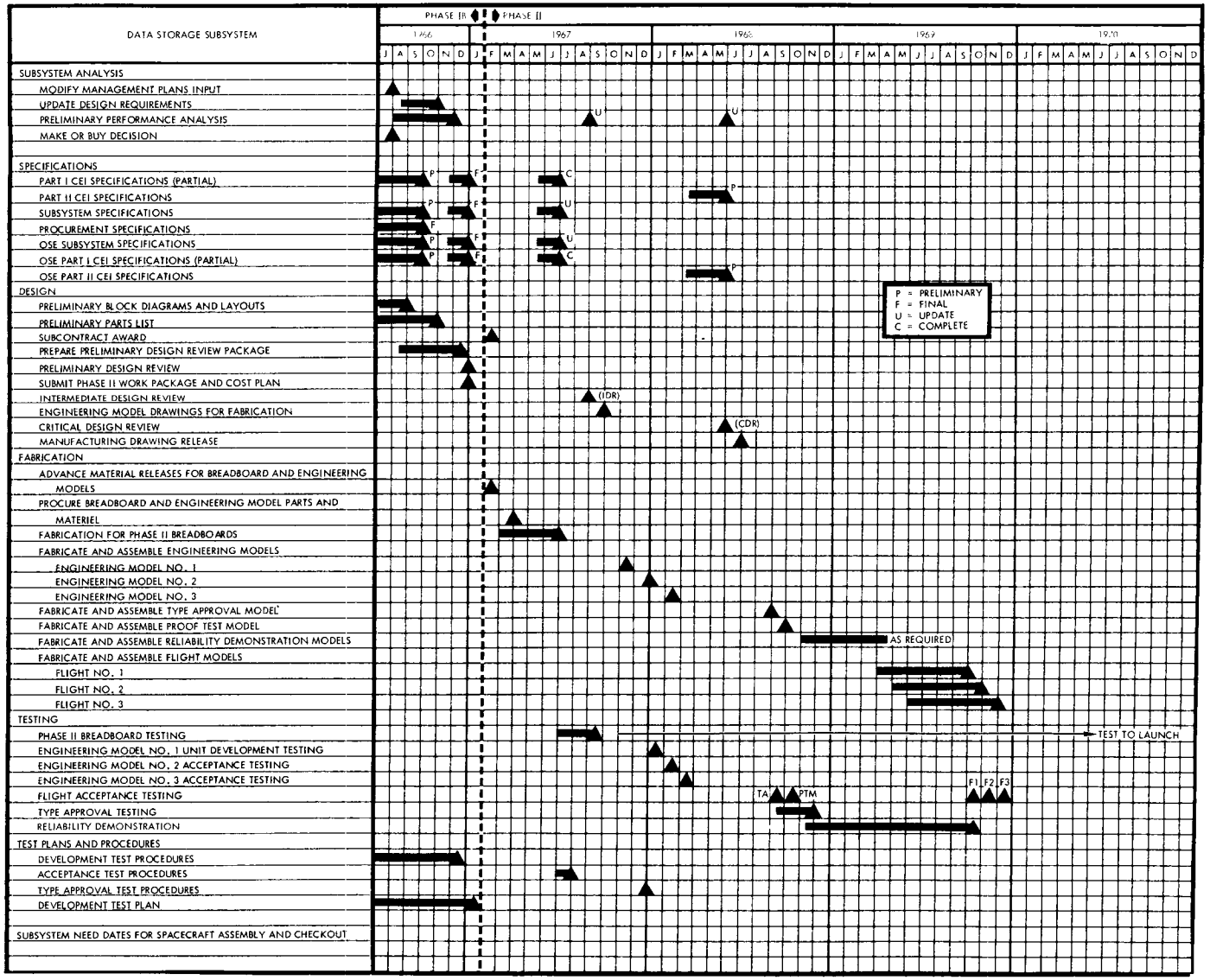


Figure 24. Schedule for Data Storage Subsystem

subsystem units and OSE. Subcontractor submittals will be reviewed. Phase IB is concluded with a preliminary design review and preparation of design data reports.

Phase II efforts start by finalizing the subsystem logic design, releasing a final subsystem and OSE specification and final Part I (partial) specifications for the subsystem units and OSE. A subcontract for recorder fabrication will be awarded. Next, the recorder breadboards are fabricated and tested. When breadboard test results are available, the IDR is held. Engineering model (EM) drawings are then released, and three EM's are fabricated. Completion of EM testing constrains the CDR, after which final drawings are released.

3.8.3 Design and Development

a. Subsystem Analysis and Design

Science data requirements will be studied to determine the parameters of each of the recorders. Mechanization of the recorder record-playback sequence will be studied to optimize the control signals. To determine the effect on each recorder, subsystem thermal, weight, packaging, power and magnetic analyses will be performed. Analyses will be performed to determine the requirements imposed by the need for ethylene oxide compatibility.

b. Recorder Analysis and Design

Recorder requirements will be analyzed to determine how many different units are required. To obtain maximum use from life test data, and to minimize design activity, it is necessary to have as few different components in the subsystem as possible. Once these requirements are established, an analysis will be performed to choose one or more types of tape transports.

The servo system required for synchronous playback of stored data will be analyzed, breadboarded, and tested. An effort will be made to utilize standard circuit modules, and an analysis will be performed. Bit dropout and recorder life requirements will be studied for their effect on recorder design. Redundant end of tape sensors will be designed.

Command and control circuits will be analyzed, with an emphasis on reliability and redundancy. Digital input and output circuits will be designed, considering the recorder interfaces with other subsystems. In each recorder, detailed analysis will be performed in the areas of weight, power, thermal interfaces and magnetic properties. Suitable telemetry sensors and signal conditioners will be designed. Effect of the momentum exchange with the spacecraft performance will be studied.

3.8.4 Test Program

During the early stages of Phase IB a selected and limited number of module circuit breadboards will be fabricated and tested. Standard digital circuits will be evaluated, and special circuits will be tested.

As listed in Table 22 a breadboard of each different type of recorder will be tested in Phase II to verify that all performance requirements, as set forth by the equipment specification, are met. For the recorders, some of the operational parameters that will be verified during the test are:

- Operation at different bit rates
- End of tape operation
- Control operation
- Bit dropout rate

These parameters are tested while the assembly is subjected to a combination of the following conditions:

- Type approval temperature extremes
- $\pm$ 15 percent power supply variation
- Transients, noise and audio tones 15 percent over the type approval levels
- $\pm$ 15 percent power frequency variation from type approval levels

Table 22. Data Storage Subsystem Tests

Test Title	Item Tested	Purpose and Objective	Description
Component Preconditioning	All electrical and mechanical where applicable. Motors, mylar belts, and storage tape.	To eliminate early mortality and bad lots.	Determine all samples meet component specifications and to eliminate components with unstable characteristics.
Circuit Design Evaluation	Breadboard Circuits	Verify circuit performance conforms to circuit specifications.	Measure circuit performance criteria while varying temperature and supply voltages to their tolerance extremes.
Unit Design Evaluation	Breadboard Mechanical and Electrical System	Verify unit feasibility and performance parameters conforms to system specifications	Measure unit performance criteria while varying temperature supply voltages, input signals, and output impedances to their tolerance extremes
Electrical and Mechanical Design Verification (Life Test)	Hard Model Breadboard	Demonstrate that all design aspects, including materials and processes meet minimum specification requirements	Worst-case voltage, temperature, noise, signals, transients, while monitoring system operating parameters. Operate continually until launch - update configuration as changes to system configuration occurs
System Compatibility	Hard Model Breadboard	Establish unit compatibility with interfacing subsystems - verifying unit operates to required specifications	Establish that all unit operating parameters meet specifications while integrated with other subsystems
In-line Production	All Production Hardware	Determine from the lowest level of assembly to the final system assembly that circuit, mechanical, and system operating requirements are met	Physical and electrical tests, imposing worst case limits where critical operation exists, to verify proper operating characteristics at various stages of assembly
Unit System Operating and Physical Design Verification	Engineering Model No. 1	Verify all design aspects meet unit specifications	Expose unit to tests verifying thermal, packaging, magnetic properties, electromagnetic controls, weight, physical dimensions, ethylene oxide compatibility, and interface (internal and external) designs are acceptable
Type Approval Tests	Engineering Model No. 2 Flight Unit No. 1	Establish the unit will survive type approval environmental exposures and meet operating requirements while exposed	Exposure unit to all type approval tests - vibration, acceleration, shock, acoustics,thermal vacuum, thermal shock, humidity, EMC, and magnetic properties.
Flight Acceptance Tests	All Flight Assemblies Engineering No. 3 Assemblies	Demonstrate flight units will meet operating requirements during exposure to acceptance tests	Subject the units to flight acceptance test exposures and test system functional characteristics during each exposure.
Reliability Demonstration Tests	Flight Units	Demonstrate that units will meet life and wearout requirements	Test units, simulating as much as practicable actual operating conditions, to demonstrate unit performance requirements will be met at least to the minimum operating life requirements and through a realistic wearout cycle

Interfaces with other subsystems will be tested with either subsystem breadboards or simulated subsystems. The recorder breadboards, after all necessary tests are completed, will be updated, powered continuously, and tested periodically. This will continue until launch of the first spacecraft.

EM's of flight configuration subsystems are used in the next level of testing. There are three engineering models. The first subsystem will be subjected to the same series of tests used to verify the design in the breadboard test phase. In addition, EMC and vibration testing will be performed.

The second EM subsystem will be tested to the type approval test requirements, while the third engineering model will be tested to flight acceptance levels. Magnetic compatibility tests will be performed on EM No. 1, 2 and 3.

The type approval subsystem will first be tested to flight acceptance requirements, then to the type approval test requirements. The type approval subsystem, upon completion of these tests, will undergo a life test. Reliability demonstration tests as defined in the Preliminary Reliability Plan will use units that have passed flight acceptance test.

3.9 Radio Subsystem

3.9.1 Introduction

The following equipment, to be supplied by RCA, comprises the S-band communication electronics units:

- Three S-band receivers for reception of command and ranging signals.
- Modulator-exciters to provide carrier phase-modulation and drive to the S-band power amplifier. Two modulator-exciters are included.
- 1-watt transmitter and power monitor for low power transmission and monitoring.
- Two S-band power amplifiers, each including a 50-watt TWT, power supply and power monitor.

- Receiver selector which accepts "in lock" signals from the three receivers plus commands from the computing and sequencing and command subsystems.

- Transmitter selector to choose the transmission mode, based upon inputs from on-board power monitors plus commands from the C and S and command subsystems.

In addition, RCA will supply the following microwave components:

- 4-port hybrid ring coupling the modulator-exciters and the S-band power amplifiers.

- Circular switches which connect the selected transmitter to the selected antenna.

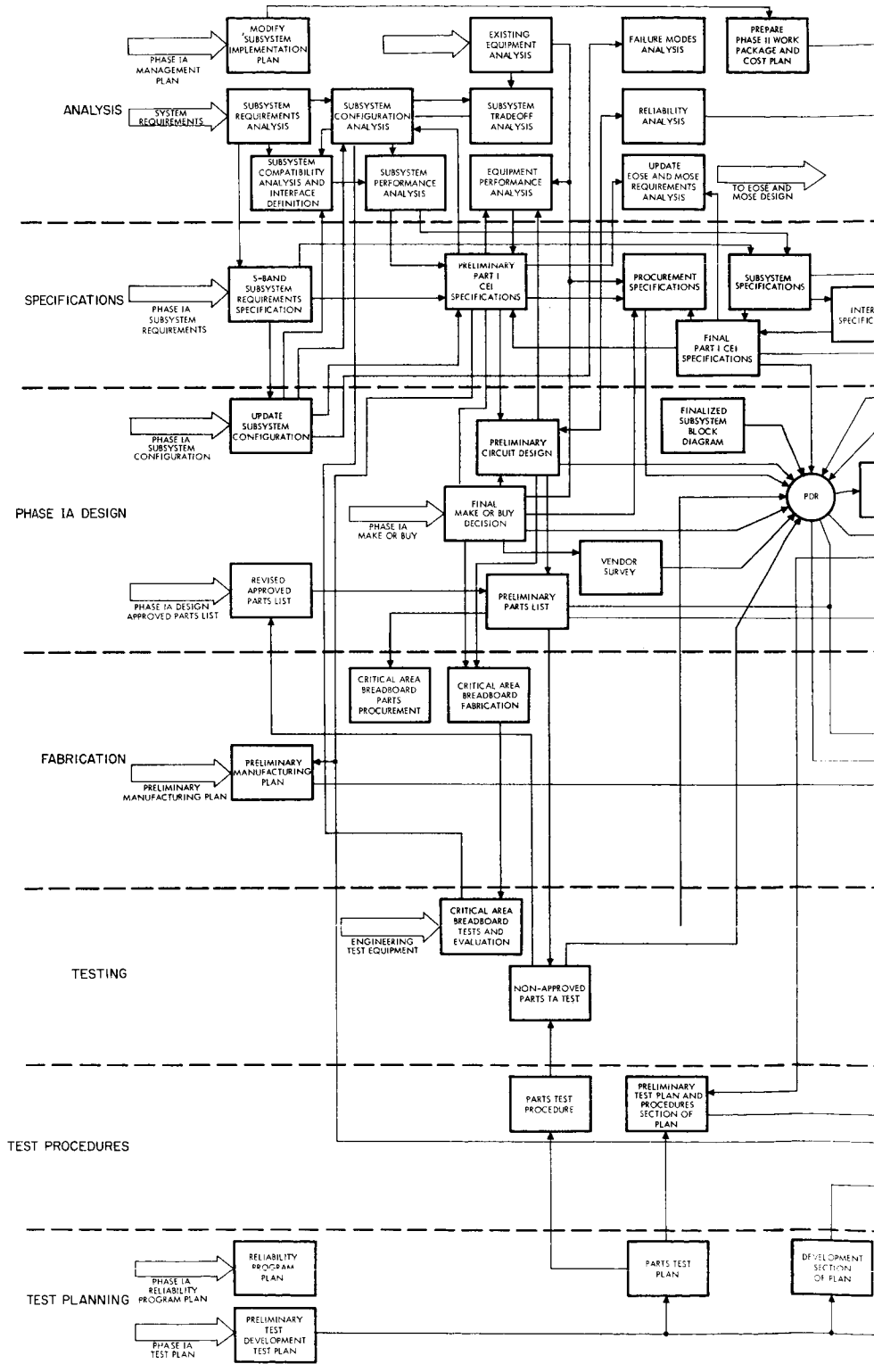
- Diplexers for each antenna.

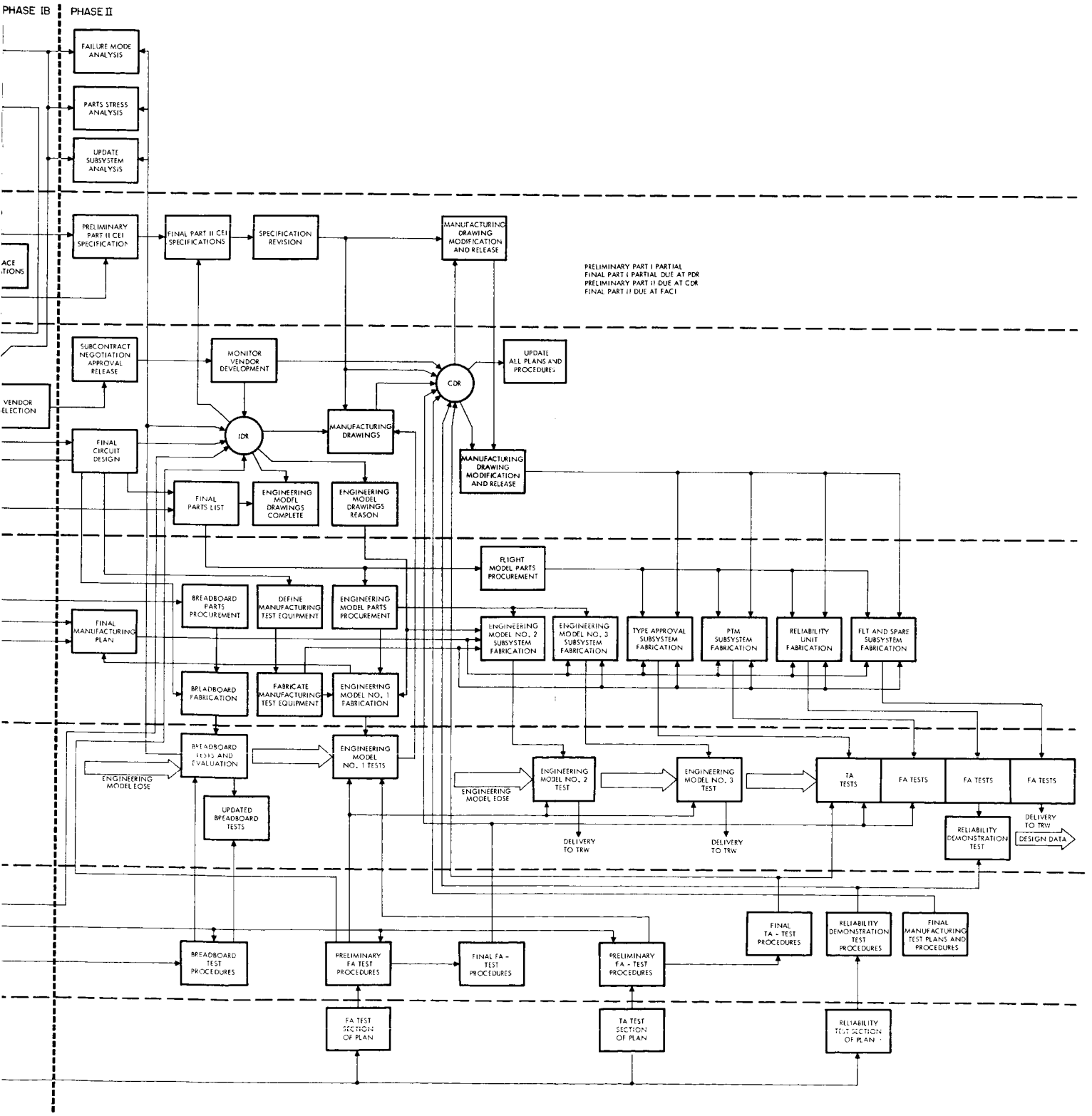
In the following discussion the above equipment is referred to collectively as the S-band subsystem.

The responsibilities of RCA, in connection with the S-band subsystem, include system engineering support, design, development, fabrication, and assembly of test and flight units, as well as development, type approval (TA) and flight acceptance (FA) testing on assembly and subsystem levels. RCA will deliver to TRW Systems assembled FA subsystems. RCA will perform life and reliability demonstration tests on the S-band subsystem and will provide support for field activities including system assembly and test at TRW, prelaunch, launch, and flight operations. EOSE for S-band subsystem checkout and for use in the STC will be supplied by RCA. In addition, the necessary bench checkout equipment for subassembly and assembly test will be designed and built. Mechanical operational support equipments for handling, transportation, assembly and test are also of RCA responsibility.

a. S-Band Subsystem Flow Plan

Starting with the results of the Phase IA study, and with additional technical direction which will be forthcoming from TRW, the development program for S-band communication electronics and microwave components will proceed through Phases IB and II, culminating in a demonstrated type-approved design. As indicated in Figure 25, S-band subsystem flow plan, the effort planned for Phase IB is devoted to


ANALYSIS
SPECIFICATIONS
PHASE IA DESIGN
FABRICATION
TESTING
TEST PROCEDURES
TEST PLANNING
PHASE IA MANAGEMENT PLAN
MODIFY SUBSYSTEM IMPLEMENTATION PLAN
SYSTEM REQUIREMENTS
SUBSYSTEM REQUIREMENTS ANALYSIS
SUBSYSTEM CONFIGURATION ANALYSIS
SUBSYSTEM TRADEOFF ANALYSIS
EXISTING EQUIPMENT ANALYSIS
FAILURE MODES ANALYSIS
PREPARE PHASE II WORK PACKAGE AND COST PLAN
RELIABILITY ANALYSIS
SUBSYSTEM COMPATIBILITY ANALYSIS AND INTERFACE DEFINITION
SUBSYSTEM PERFORMANCE ANALYSIS
EQUIPMENT PERFORMANCE ANALYSIS
UPDATE EOSE AND MOSE REQUIREMENTS ANALYSIS
TO EOSE AND MOSE DESIGN
PHASE IA SUBSYSTEM REQUIREMENTS
S-BAND SUBSYSTEM REQUIREMENTS SPECIFICATION
PRELIMINARY PART I CEI SPECIFICATIONS
PROCUREMENT SPECIFICATIONS
SUBSYSTEM SPECIFICATIONS
FINAL PART I CEI SPECIFICATIONS
PHASE IA SUBSYSTEM CONFIGURATION
UPDATE SUBSYSTEM CONFIGURATION
FINALIZED SUBSYSTEM BLOCK DIAGRAM
PRELIMINARY CIRCUIT DESIGN
PDR
PHASE IA MAKE OR BUY
FINAL MAKE OR BUY DECISION
VENDOR SURVEY
PHASE IA DESIGN APPROVED PARTS LIST
REVISED APPROVED PARTS LIST
PRELIMINARY PARTS LIST
CRITICAL AREA BREADBOARD PARTS PROCUREMENT
CRITICAL AREA BREADBOARD FABRICATION
PRELIMINARY MANUFACTURING PLAN
PRELIMINARY MANUFACTURING PLAN
ENGINEERING TEST EQUIPMENT
CRITICAL AREA BREADBOARD TESTS AND EVALUATION
NON-APPROVED PARTS TA TEST
PARTS TEST PROCEDURE
PRELIMINARY TEST PLAN AND PROCEDURES SECTION OF PLAN
PARTS TEST PLAN
DEVELOPMENT SECTION OF PLAN
PHASE IA RELIABILITY PROGRAM PLAN
RELIABILITY PROGRAM PLAN
PHASE IA TEST PLAN
PRELIMINARY TEST DEVELOPMENT TEST PLAN



2

Figure 25. Design and Development of S-Band Subsystem

obtaining performance and design requirements (partial Part 1 CEI specifications as defined in Voyager Configuration Management Manual) for the subsystem for each functional unit with supporting plans and analyses. Hardware activity, as now planned, will be limited to Phase II. An exception will be made if the conceptual design includes a low-noise preamplifier or other features which require breadboard testing in Phase IB to supplement design analyses.

Phase IB. The initial effort in Phase IB is an analysis of subsystem requirements which may be modified significantly from those analyzed during Phase IA. The uplink frequency, for example, may be changed to 1800 megahertz (MHz) This analysis and other development tasks will be performed in close cooperation with TRW in keeping with the technical interfaces involved. This activity will lead to the S-band subsystem requirements specification.

Long lead items will be identified in Phase IB and procured for use in Phase II breadboard circuits.

A re-examination of the tradeoffs resolved during Phase IA can then be made to satisfy any new subsystem requirements. In this connection, the interference and performance degradation of simultaneous ranging and telemetry must be re-evaluated and extended to include frequency multiplexed telemetry subcarriers. Existing and in-development S-band equipment will be evaluated for Voyager application as is, or with redesign. In particular, 50-watt traveling wave tube amplifier and electrastatic focused klystron programs now in the development stage, will be monitored and planning coordinated with the successful bidders. The configuration resulting from this effort will be the basis of the conceptual design. Critical development areas in the configuration will be identified for special planning to expedite in-house development or subcontractor effort. Provisional make or buy decisions made during the study will be reconsidered and final decisions made to permit vendor surveys and source selection processes and purchase specification preparation to start. Selection of a traveling wave tube amplifier development source will require careful screening, if this item is a contractor responsibility.

As the conceptual design evolves, test equipment requirements will be updated. Close liaison will be maintained between test planner, S-band equipment, EOSE and AHSE designers to ensure timely availability of test equipment and mechanical handling and shipping equipment.

Materials and parts will be evaluated for application in the detail design. Voyager requirements for magnetic cleanliness and sterilization emphasize the importance of this task. Selected parts and materials will be screened for conformance to program reliability, performance and magnetic standards. Parts and materials with insufficient reliability, magnetic or performance data will be subject to type approval testing during Phase IB.

With the conceptual design in progress, critical development items identified and parts and material information available, the effort will broaden out to several tasks performed in parallel. To investigate problems which cannot be resolved theoretically, such as the command subcarrier distortion due to the receiver subsystem, breadboards of critical elements may be constructed and tested. Electrical and mechanical interfaces will be monitored and defined to maintain compatibility as the subsystem design evolves. Reliability and failure mode analyses will be accomplished and other software prepared in Phase IB, such as management, test, safety, manufacturing, EMC, contamination control, and magnetic control plans, will be updated. Based on the preceding work, design specifications will be prepared for each functional unit completing the PDR prerequisites.

The preliminary design review, held near the end of Phase IB, will consider all aspects of the conceptual design including the results of all analyses, critical item breadboard tests, if any, and the Part I CEI (partial) specification.

The specifications will be issued with requests for bid on all buy items in accordance with the approved procurement plan. Bids will be available for subcontract negotiation at the beginning of Phase II.

Phase II. As soon as practicable after the start of Phase II, vendor selections will be made and subcontracts negotiated, approved and

let. Detailed test specifications and procedures will be completed for breadboard, EM and TA units.

Detailed design of in-house equipment will be accomplished using breadboards for initial design verification. Equivalent development effort will be performed by subcontractors. Using vendor-supplied breadboards of buy items, a breadboard subsystem will be integrated and tested.

The intermediate design review (IDR) will be held when circuit design and breadboard design verification tests are complete. EM fabrication must be undertaken before the intermediate design review, to accommodate schedule requirements. Part I, CEI specifications will be completed for this review.

Design changes arising from breadboard tests and the IDR will be incorporated and EM tests performed on assembly and subsystem levels. Complete product configuration definition will then be generated for the CDR constituting preliminary Part II, CEI specifications. When EM tests, product drawings and detail specifications are completed, the production design will undergo critical design review. Additional EM's will be fabricated, tested, and delivered for use in engineering and propulsion interaction spacecraft models.

CDR approval is required prior to initiation of flight item fabrication. The first production item of each type will be subject to TA testing at assembly and subsystem level. The next following production subsystem is intended for the PTM spacecraft. It will be fabricated and tested to FA standards. Additional production units will undergo reliability demonstration testing after FA. Successful completion of TA and reliability demonstration tests will signal the conclusion of the development effort.

On completion of the TA tests, subsystems for three flight spacecraft and spare equipment are fabricated and FA tested on subassembly and subsystem level prior to shipment to TRW.

b. Key Problem Areas

Development of the S-band subsystem does not present any apparent serious technical problems. The schedule appears to be ample to perform the necessary tasks with relatively minor risks.

Fulfilling the power amplifier requirement with a suitable TWTA or ESFK device will probably require a development effort since it is doubtful if existing devices are capable of meeting Voyager requirements. The subsystem development schedule incorporates such an effort although an existing ESFK design by Eimac or the projected TWTA development sponsored by JPL may have the desired features. It is also recognized that the power amplifier may be GFE or a directed procurement.

Communication with two spacecraft presents a possible interference problem between the two uplink frequencies. After analysis, this problem will be minimized by choice of channel frequencies, by properly selective receiver front ends and by careful spectrum examination to detect and remove spurious signals.

The choice of up-link frequency will influence receiver design. Major redesign of an existing equipment will be required to accommodate 1800 MHz while only minor modification would be required for 2100 MHz.

Magnetic field reduction will impose a requirement on parts and materials selection and packaging layout which can be met quite readily. No major problem is anticipated in meeting Voyager magnetic requirements using the methods described in the Phase IA, Task A Study Report.

Thermal, weight, power, reliability, and life requirements are nominal factors which will be satisfied by usual design and program implementation methods. Reliability considerations lead directly to the high degree of redundancy incorporated in the subsystem design. Test planning includes reliability demonstration tests as a further result of reliability and life requirements. The power amplifier development, as previously mentioned, is also affected.

The mission environment does not appear to present any difficult constraints on the S-band subsystem implementation with the single exception of operation through critical pressure during the ascent phase. The requirement for high power operation during launch is not apparent and it is therefore intended that the power amplifiers will not be operated during test or mission operations in the region of critical pressures to avoid possible breakdown problems.

c. Schedule

The schedule established for the S-band subsystem is based on the following key requirements:

- Definition of subsystem and assembly detailed design specification available for the PDR during Phase IB.
- Hardware effort confined to Phase II with the exception of critical development items.
- Completion of vendor and subcontractor source selection process, including approval by TRW and, if required, JPL, to accommodate subcontract releases during the first month of Phase II.
- Breadboard test results and detailed design available for the IDR.
- Breadboard tests to begin at least 3 months before engineering model fabrication.
- EM test results to be available for the CDR.
- TA tests completed before release for fabrication of flight 1, 2, 3 and spares.

The S-band communication schedule shown in Figure 26 is a conservative one with no critical schedule problems, although several considerations are relevant.

First, the 20-month period assumed for power amplifier development is defined as culminating in the delivery of the first flight accepted item for the PTM subsystem. Prior experience with Relay and Lunar Orbiter TWTA development indicates this to be more than ample but in keeping with conservative planning. Any remaining schedule uncertainty may be reduced further if the proposed JPL TWTA or ESFKA development contract is awarded prior to the start of Phase II.

The schedule shown includes several hardware activities which are initiated on the basis of analysis with incomplete or no design verification data in order to meet key requirements. Two of the three activities listed below involve relatively minor cost and schedule risks but result in continuity of effort and more uniform facility and manpower load. For the remaining instance, a limited amount of test data will be available to justify the step.

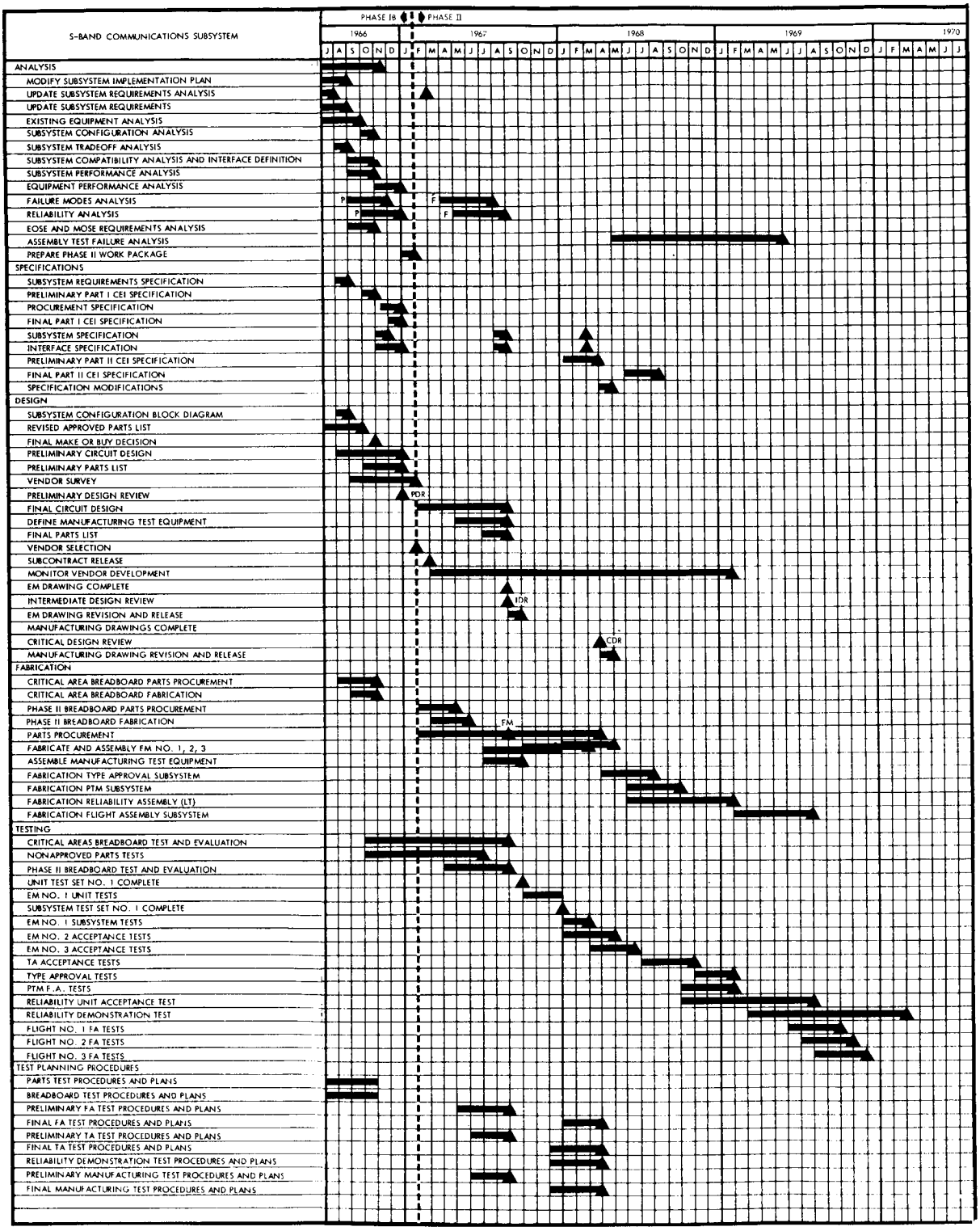


Figure 26. Schedule for S-Band Communications Subsystem

- EM parts are ordered before breadboard tests are initiated.
- Long-lead flight model parts are ordered before EM tests are completed.
- Fabrication of PTM and reliability demonstration test units is begun before FA test of the TA unit is started. It is assumed that EM tests will provide justification for this step.

3.9.2 Radio Subsystem Analysis

a. Subsystem Analyses

The Voyager mission imposes specific requirements on the radio subsystem beyond those encountered on earlier programs such as Ranger, Mariner, and Pioneer. In addition to the requirement for multiple planetary vehicle operation, improvements in system tracking performance, reduction of acquisition time and over-all improvements in telemetry and command threshold performance are desirable.

During Phase IB, the following analytical tasks will be performed:

- Determination of telemetry system degradation due to presence of turn-around ranging subcarrier.
- Determination of the effect of receiver phase-lock loop parameters on predetection bandwidth, command transfer function, acquisition characteristics and retransmitted signal parameters.
- Analysis of ring-around and false lock characteristics of the receiver/modulator/power amplifier loop, including a detailed spur analysis.
- Feasibility of reducing VCO and auxiliary oscillator phase instability, and effect on threshold and doppler performance.
- Optimization of phase-lock loop bandwidth, suppression factor and command subcarrier frequency considering the variation in vehicle velocity and acceleration throughout the mission.
- Distortion effects of phase-lock loop on command and range code sequences.
- Determination of the change in modulator-exciter characteristics under varying input and output impedance conditions.

- Stability characteristics of transistor and varactor multiplier chains.
- Determination of the effect of receiver and modulator-exciter parameters on the accuracy of relative and absolute ranging.
- Analyses will also be performed to guide design efforts in the areas of weight, structure, thermal, power consumption, reliability, magnetic properties, EMC, sterilization and intra- and inter-subsystem interfaces.
- Feasibility of using more than one coherency ratio to facilitate multiple planetary vehicle operation, and effect on receiver and phase-lock loop design requirements and DSIF operation.

b. Component Analyses

Four-Port Hybrid Ring and Low-Level Power Monitors. An analysis will be performed to determine the compatibility of the four-port hybrid ring with changing load conditions. This analysis is important in determining the degree of isolation provided by the four-port hybrid for various conditions of mismatch at the TWT amplifiers input. The analysis will determine the range of the TWT amplifier drive-power fluctuation, as well as modulator-exciter load-impedance variation.

Analyses will also be performed to determine the characteristics of the low-level power monitors under various impedances, drive levels, and environmental conditions in order to establish the interface with the transmitter selector.

To determine the capability of the proposed stripline design, the preliminary design of the four-port hybrid will be analyzed to determine the ability to meet insertion loss and isolation requirements, as well as establishing a package configuration compatible with Voyager mounting concepts.

Transponder (S-Band Receiver and Modulator-Exciter). This is the equipment most affected by the Voyager unique requirements. In order to detail the component performance and purchase specifications, some of the analytic work must first be concluded. Vendor quotations cannot, therefore, be solicited until early in the Phase II program. In

support of the analyses, it is desirable to use an available GFE transponder to construct a preliminary breadboard model of the radio subsystem.

Receiver and Transmitter Selectors. During Phase IB, the following tasks will be performed:

- Determine the probable failure modes of each unit. An investigation will also be made into methods for reducing these failure modes through the use of direct command override and component and circuit level redundancy.
- Investigate the possible interface characteristics which can be employed between the receiver and receiver selector with respect to optimum location of the quantizing circuits, and make a decision on the location which afford the maximum protection against incorrect operation.
- Investigate the possible interface characteristics which can be employed between the power monitor and transmitter selector with respect to voltage and impedance levels and with respect to signal comparison, and select the technique affording maximum protection against incorrect operation.
- Signal-to-noise and multiple carrier and subcarrier interference characteristics of the receiver, and effects of IF and video limiting.
- Reduction in acquisition time through use of noncarrier synchronous ranging, and effect on threshold performance.
- Investigation of range-code and reference signal interfaces between multiple receivers and modulator-exciters.
- Effect of change in up-link frequency to 1800 MHz with consequent increases in available bandwidth.
- Assessment of problems associated with magnetic cleanliness.
- Failure mode analysis of all elements of the subsystem and application of component and unit redundancy to improve subsystem reliability.
- Evaluation of reliability of candidate low noise S-band preamplifiers.

These and other related analyses will be iterative throughout Phase IB, based on the expectation that the required inputs data from JPL

will be continually updated as in-house and contractor studies proceed. Output data from the analyses will be presented to TRW as derived for comparison with JPL results and requirements.

An end item of the analyses will be a design study and evaluation of the applicability of existing or projected equipment, and a definition of the degree of required redesign.

Low-Power Transmitter. The low-power transmitter has requirements similar to the modulator-exciter, except for the higher output level. Also, the output impedance variations on the low-power transmitter can be greater than on the modulator-exciter.

During Phase IB, the following efforts will be performed in the area of design and development of the low-power transmitter:

- Additional analysis of the multiplier-chain stabilities will be performed. Analysis will include varactor-transistor multiplier-chain tradeoffs; an investigation of stress levels in a 1-watt, solid-state, S-band transmitter; compatibility of internal voltage levels with the partial-pressure launch environment; and spectral-purity requirements.
- A detailed vendor survey will be performed to support "Make-or-Buy" or redesign decisions.

Power Amplifier. It is assumed that the 50-watt TWT amplifier will be vendor supplied or GFE. Analyses will, however, be necessary to determine the required characteristics of high level power monitors, output bandpass filter and diplexer requirements, and the thermal and mechanical compatibility of the amplifier with the proposed radio subsystem packaging concept.

3.9.3 Design

a. General

During Phase IB, packaging, cabling and interface design data for the S-band radio subsystem will be generated, together with detailed functional specifications for all functional units of the subsystem.

During Phase II, major or complete mechanical redesign of existing receivers and modulator/exciters will be undertaken, and electrical redesign to the extent dictated by Phase IB analyses.

b. 50-Watt Power Amplifier

It is presently assumed that this will be a vendor-supplied or GFE item. During Phase IB, functional specifications, mechanical and electrical interface requirements will be agreed with TRW. Design investigations, including the use of split collector and modified helix construct ion in TWTA's, will be the subject of detailed investigation by potential vendors. The power amplifier will probably be a long-lead item.

c. Microwave Components

The microwave components of the S-band subsystem consist of the S-band switching network (four circulator switches), diplexers, and four port hybrid.

During Phase IB, the following effort will be performed in the area of design and development of the S-band switching network:

- Preparation of Part I, CEI specification (partial), which will include analyses to determine:
  a) Insertion loss requirements
  b) Isolation requirements
  c) Logic and failure-mode improvements
  d) Tradeoffs between continuous and latched-type switches
- Comprehensive vendor evaluations and a tentative selection, based upon the ability to satisfy the magnetic-cleanliness, performance, reliability, and schedule requirements of the Voyager spacecraft program.

The following efforts will be performed in the design and development of the diplexers:

- Preparation of the Part I, CEI specification (partial), which will include analyses to determine:
  a) Transmitter-to-receiver isolation requirements at the transmit frequency, receive frequency, and other frequencies that could cause receiver overloading or false lock
  b) Transmitter and receiver insertion-loss requirements

- Comprehensive vendor survey.

As a design task an investigation will be performed to determine the need for an instrumented testing program to determine the RF power handling characteristics of the switching matrix, diplexers, and RF interconnections.

For the four port hybrid the following tasks will be performed:

- Preparation of Part I, CEI specifications (partial) based on the following analysis:

  a) Isolation requirements

  b) Effect of power amplifier input VSWR variations (on and off) on isolation and power split.

d. Receiver and Transmitter Selectors

During Phase IB, specific design investigations will be necessary in order to meet the schedule requirements EM's in Phase II. These tasks will include:

- Preparation of Part I, CEI specifications (partial)
- Preliminary design of internal logic, based on maximum utilization of common circuit elements
- Definition and preliminary design of input and output buffer circuits and logic
- Design of packaging and preparation of engineering sketches for use in fabrication of EM's

Early in the Phase II part of the program, a breadboard model of each unit will be designed and constructed. This breadboard will consist of the circuits for the final design, but will be constructed using hand wired laboratory rack and panel mounting. Special test equipment required for breadboard testing will be designed and constructed, and performance and reliability tests will be performed. In particular, operation in the various failure modes will be verified, and corrective steps taken, where necessary, to minimize the effects of discrete failures on unit performance.

e. S-Band Receiver

For Phase IB, the following design and development tasks will be performed:

- Preparation of Part I, CEI specifications (partial) which will include analyses to determine:

  a) Predetection carrier tracking bandwidth

  b) Carrier tracking phase-lock loop requirements for earth parking orbit, cruise, trajectory correction maneuvers, deboost, and orbital operations.

  c) The effect of multiple uplink frequencies and IF/Video limiters on crosstalk.

  d) Receiver spurs-self lock.

- Comprehensive vendor survey.

f. Modulator-Exciter/Low Power Transmitter

The following Phase IB design and development tasks will be performed:

- Preparation of Part I, CEI specification (partial) which will include analyses to determine:

  a) Modulation linearity requirements for multiplexed signals

  b) Multiplier stability under various load conditions (VSWR)

  c) Transponder spurs (including range code input signals)—receiver lock.

- Comprehensive vendor survey.

3.9.4 Test Program

The S-band subsystem test program is divided into five major phases: parts, development, manufacturing, TA and FA testing. The program defines tests on parts, modules, components and subsystems. Breadboard, mockups, engineering models and flight-type equipment tests are specified. A matrix containing the test name, objective and description for each equipment type, is presented in Table 23.

Breadboards of both make-and-buy subassemblies will be tested early in the test program during the development phase. Assemblies, circuitry and packaging techniques as proposed for flight will be utilized

Table 23. S-Band Communications Electronics and Microwave Components Development Tests

Name of Test	Item to Be Tested: Component Level — Breadboard Models *	Engineering Models	Type Approval Model	Proof Test Model	Reliability Models	Flight Models	Spare Models	Subsystems — Breadboard Model	Engineering Model	Type Approval Model	Ty. Others	Purpose and Objectives	Description
Circuit design evaluation	X											Determine electrical performance breadboards to verify that design conforms to component performance specifications	Measure operation parameters with specified range of input conditions at ambient temperature
Design verification	Q											Demonstrate that the initial design chosen will meet preliminary design specifications	Conduct tests in the temperature chamber over the extreme qualification limits while measurements are made of the operating parameters with specified input conditions.
Electrical performance test (EPT)**	Q	Q	Q	X	X	X	X	Q	Q	Q	X	Determine electrical performance before and after environmental test sequence to verify conformance to the specification	Measure operating parameters with specified range on input conditions
Abbreviated performance test (APT)	X	X	X	X	X	X	X	X	X	X	X	Demonstrate that there is no degradation in electrical performance due to environmental exposure	Conduct tests before and after each exposure at ambient temperature and nominal line voltage.
Reliability testing	X											Provide test data that cannot be calculated or is required for verification	This series of breadboard tests includes margin failure mode, parts electrical stress measurements, and temperature hot spots of RF equipment.
Product examination		X	X	X	X	X	X	X	X	X	X	Demonstrate that equipment was built in accordance with drawings before testing starts	Inspect for compliance with the requirements for identification, markings, mounting, connectors, dimensions, center-of-gravity (e. g.)location.
Reliability demonstration life testing				X						X		Demonstrate that certain critical units meet the life requirements of the Voyager mission	Test one or more flight components to APT at ambient conditions continuously and to EPT at one-month intervals.
Margin tests	X	X							X			Provide a measure of design margin beyond those exposed by qualification limits	Test varying bus voltage, input-signal amplitudes, pulse duration, rise and fall times, frequencies, and repetition rates, where significant combinations of these may be tested.
Electromagnetic compatibility (EMC)		X	X					X	X	X		Verify conformance to EMC performance specification	Measure level of susceptibility to and generation of conducted and radiated RF interference
Magnetic testing	X	X	X	X	X	X	X	X	X	X		Determine magnetic field intensity to verify below acceptance limit	Measure magnetic field intensity in three axes while operating in each test mode while magnetized and demagnetized.
Sterilization compatibility test		X	X									Verify resistance to ethylene oxide environment and ability to meet performance specifications due to temperature sterilization	Expose to ethylene oxide gas at specified concentration and temperature for specified period. Expose to bulk temperature for specified period.
Environmental tests:												Verify integrity of design when equipment is tested in an environment equal to or greater than flight levels	Impose environmental limits on unit monitoring operating characteristics to insure proper operation or proper survival, whichever is applicable.
Vibration		Q	Q	X	X	X	X						
Thermal-vacuum	Q	Q	Q	X	X	X	X	Q	Q	Q	X		
Humidity		X	X										
Shock		X	X										
Thermal shock		X	X										
Handling, transport, and prelaunch environmental test				X								Determine ability of equipment to survive environment without electrical performance degradation	Includes bench handling, drop, transportation, vibration, explosive and corrosive atmospheres, humidity, contaminants, and magnetic field effects.
Subsystem compatibility									X	X		Demonstrate that the complete subsystem meets subsystem and mission requirements. Demonstrate that EOSE and subsystem are compatible	Measure over-all subsystem performance parameters while operating in various modes under specified input conditions using EOSE.

Q = Qualification level test more stringent than flight level.
* - Existing modified flight equipment from selected vendors.
** - Includes voltage, temperature, and noncatastrophic variations.

whenever possible. The tests performed will verify performance specification, evaluate failure modes and indicate necessary design modifications.

The breadboard assemblies will be subjected to performance test requirements in conformance with JPL General Specification for Performance and Design Requirements for the Voyager 1971 Spacecraft System.

The breadboard subassemblies will be interconnected to do preliminary checks of subsystem compatibility. Subsystem performance tests will be performed to confirm preliminary performance analysis. The breadboard equipment will then be subjected to extended testing for continued evaluation.

The EM's will also be tested both as assemblies and as a subsystem after integration. Three EM's will be fabricated. EM's are much closer to the flight models in all respects than the breadboards. As such, test results are more representative of flight equipment. In addition, certain testing which would prove inaccurate on the breadboard, such as detailed magnetic mapping, can now be performed. The first unit (EM No. 1) will be retained at RCA-AED. This model will consist of one assembly each of the required S-band complement. EM No. 1 will receive FA testing, followed by TA testing. This testing serves to verify the design to refine the FA and TA testing procedures and train personnel.

The remaining two EM's consist of an assembled S-band subsystem. They will receive FA testing at the component and subsystem level. They will then be delivered to TRW for integration into the EM spacecraft.

EM tests include, but are not limited to, vibration, thermal-vacuum, magnetic testing, and ethylene oxide compatibility tests. The unique requirements of the Voyager missions dictate specific attention to these areas.

The PTM model will consist of an assembled S-band subsystem. It will receive FA testing at the assembly and subsystem level. The integrated subsystem will be delivered to TRW for further testing on the PTM spacecraft.

The first subsystem to be fabricated to released drawings after the CDR is designated the TA unit. Assemblies and the subsystem will receive FA then TA testing. Each TA unit will be subjected to tests to demonstrate the adequacy of the design for its intended usage. The class of tests include type-approval tests and life and margin tests (reference: JPL General Specifications, parts 4.2.1.1.2.2 and 4.2.1.1.2.3) of performance at both ambient and simulated mission environments.

The reliability units will be composed of various quantities of each S-band subassembly consistent with reliability testing needs as defined in the Reliability Test Plan. Each subassembly will receive FA testing to verify conformity to performance specifications. They will then be subjected to a series of reliability demonstration tests designed to probe and confirm reliability analysis predictions.

Three flight models will be fabricated. Each will consist of a full complement of S-band subsystem assemblies. In addition, assemblies will be fabricated for flight spares.

All assemblies will receive FA testing. In addition, the three flight units will receive subsystem FA testing. The tests will demonstrate that each equipment has been fabricated to and performs in conformance with the design as represented by the drawing and specification.

Voyager unique tests include comprehensive magnetic and EMI testing and ethylene oxide compatibility tests. Performance tests are performed in accordance with TRW prescribed documents.

An EM and TA unit of each S-band subsystem component will be subjected to ethylene oxide sterilization. This will confirm compatibility early in the test program. Components will be subjected to TA levels of gas concentration, temperature, humidity, and exposure time. (Specific values will be contained in the contamination control plan.) Testing of the components before and after explosure will verify the absence of degradation due to the test exposure.

Magnetic testing of the S-band subsystem will be performed at the assembly and subsystem level of development testing. Testing will be performed on a nonmagnetic turntable utilizing a compensated flux-gate

magnetic probe. The magnetic field of each component will be measured with the component in a nonoperating and in an operating condition. The field will be measured in each of three mutually perpendicular axes. These measurements will be repeated after environmental testing to insure that the magnetic properties of the specimen have remained unchanged. In addition to the pre-environmental tests, the unpowered specimen will be measured after magnetization in a 100 gauss field and then after demagnetization.

Particular attention will be allotted to the magnetic testing of the 50 watt TWTA, if this device is used. Magnetic testing of this component includes compensation to reduce the magnetic field intensity at the position of the Voyager magnetometer probe relative to the TWTA.

Another component which will receive close scrutiny during magnetic testing will be the circulator. Particular attention will be paid to stray field measurements during the energized mode of operation.

EMC testing will be performed at both the assembly and subsystem levels. The EMC test will check equipment susceptibility to external sources, both by radiation and conduction. Self-generated RFI will also be investigated.

Particular emphasis will be given to this area during power amplifier testing. The application of high power RF transmitters, although proven on past programs, bears close scrutiny in the development phase. Testing will include measurement of spurious responses and intermodulation components. Similar measurements will be made at the subsystem level to determine the degree of interaction between components.

The types and quantities of developmental test hardware are listed in Table 24. Basically, the equipment is classified for development testing either on an assembly or subsystem basis. One unit of each S-band subassembly is allotted for EM assembly level development testing. Two S-band subsystems are designated for system level development testing. Each subsystem contains a complete complement of assemblies. One unit of a complete S-band subsystem is designated for TA testing.

Table 24. S-Band Subsystem Equipment Matrix for Development Testing

	Bread-board	EM1	EM2	EM3	Type Approval	Reliability Demonstration*
50 watt transmitter, power supply and monitor	2	1	2	2	2	5
4-port hybrid	2	1	1	1	1	6
Modulator exciter	2	1	2	2	2	4
1 watt transmitter and monitor	2	1	1	1	1	4
Receiver selector	2	1	1	1	1	6
S-band receivers	3	1	3	3	3	5
Diplexer	3	1	3	3	3	18
Circular switch	4	1	4	4	4	24
Transmitter selector	2	1	1	1	1	3

* Preliminary estimates

3.10 Antenna Development

During Phase IB, the antenna performance requirements and compatability with interfaces will be analyzed, utilizing the design approaches that were generated in Phase IA of the Voyager program as well as the modifications to these approaches dictated by the final mission specification. RF interfaces considered during the antenna development program will include rotary joints, shroud transmission windows, GFE-UHF antenna and RF transmission lines, and the spacecraft structure. Mechanical and thermal interfaces will include the gimbals and actuators, booms, structural mountings, and tie downs.

The three antennas in the radio subsystem include:

- Circular paraboloid, high-gain antenna with a conical horn feed

- Elliptical paraboloid, medium-gain antenna with a horn feed
- Cup turnstile, omnidirectional low-gain antenna

3.10.1 Summary

Development of the Voyager antennas is described in the antenna development flow chart, Figure 27. Phase IB will start with a review of the antenna requirements generated in Phase IA, as modified by new mission requirements. Each of the antennas will be analyzed for conceptual design, performance requirements, and compatibility within the radio subsystem and with other subsystems. The basic antenna structures as well as the allowable surface tolerance requirements will be established. Analysis of the various structures under the influence of thermal shock, mechanical loading, vibrations, and acceleration will be completed and their electrical performance calculated.

Phase II will initiate scale model antenna fabrication for breadboard evaluation of the low-gain and UHF antenna patterns, using a scale model spacecraft. Full-scale breadboards will also be used to obtain the low-gain antenna, UHF antenna, and feed horn parameters. Engineering models of the antennas will be fabricated from engineering model drawings. Complete testing of all antennas will be performed to allow completion of final design specifications. The RF data to be acquired will include antenna patterns, absolute gain, impedance, axial ratio, efficiencies, insertion losses and coupling measurements. Data acquired by environmental testing will include responses to shock, vibration, vacuum, humidity, magnetic, and ethylene oxide environments and will permit fabrication and test of type approval models. Once the type approval models have successfully passed the type approval testing sequence, fabrication of flight models can commence.

The schedule for the antenna development program (Figure 28) is based on a go-ahead date of 1 July 1966 and a final drawing release date of 31 July 1968. An additional schedule restraint is the initiation of Phase II on 1 February 1967. All hardware, including laboratory breadboard models, will be fabricated in Phase II.

Phase IB will be devoted to subsystem configuration definitions, antenna conceptual analysis, specifications, preliminary design, test


ANALYSIS
SPECIFICATION
DESIGN
FABRICATION
TESTING
TEST PLANS
AND PROCEDURES

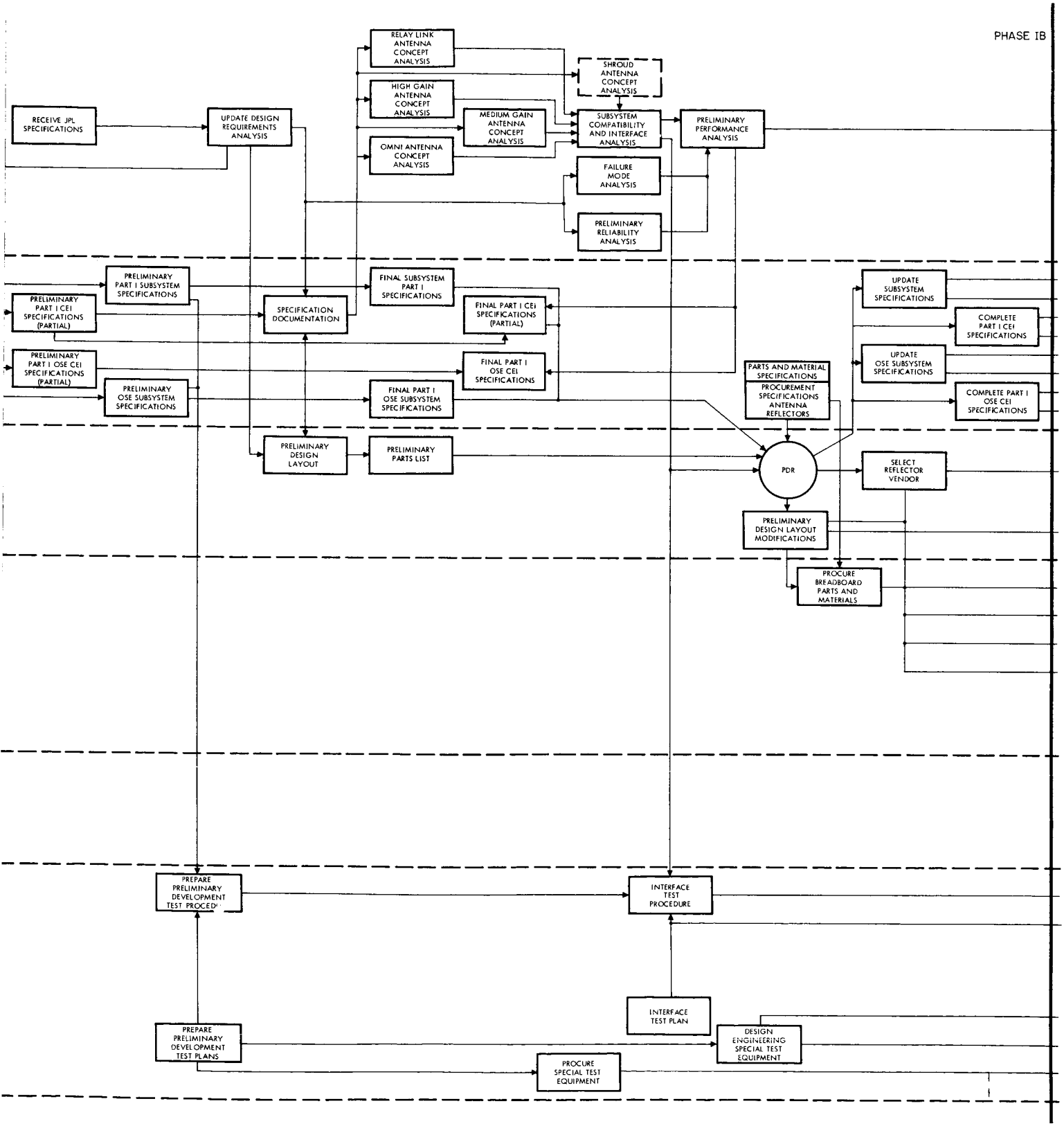
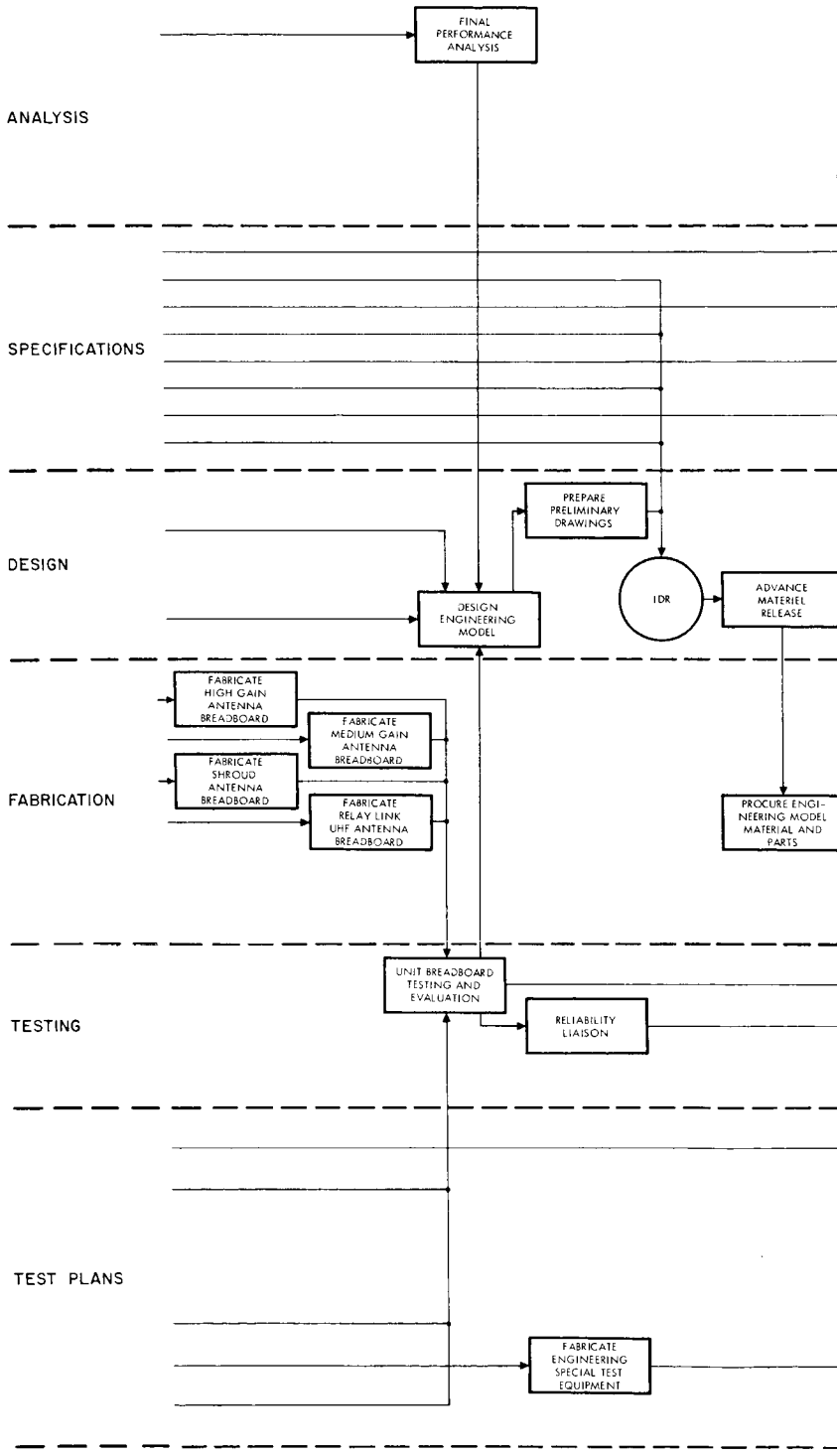


Figure 27. Design and Development of Antennas

PHASE II


FINAL PERFORMANCE ANALYSIS
ANALYSIS
SPECIFICATIONS
PREPARE PRELIMINARY DRAWINGS
DESIGN
IDR
ADVANCE MATERIEL RELEASE
DESIGN ENGINEERING MODEL
FABRICATE HIGH GAIN ANTENNA BREADBOARD
FABRICATE MEDIUM GAIN ANTENNA BREADBOARD
FABRICATE SHROUD ANTENNA BREADBOARD
FABRICATE RELAY LINK UHF ANTENNA BREADBOARD
FABRICATION
PROCURE ENGINEERING MODEL MATERIAL AND PARTS
UNIT BREADBOARD TESTING AND EVALUATION
RELIABILITY LIAISON
TESTING
TEST PLANS
FABRICATE ENGINEERING SPECIAL TEST EQUIPMENT

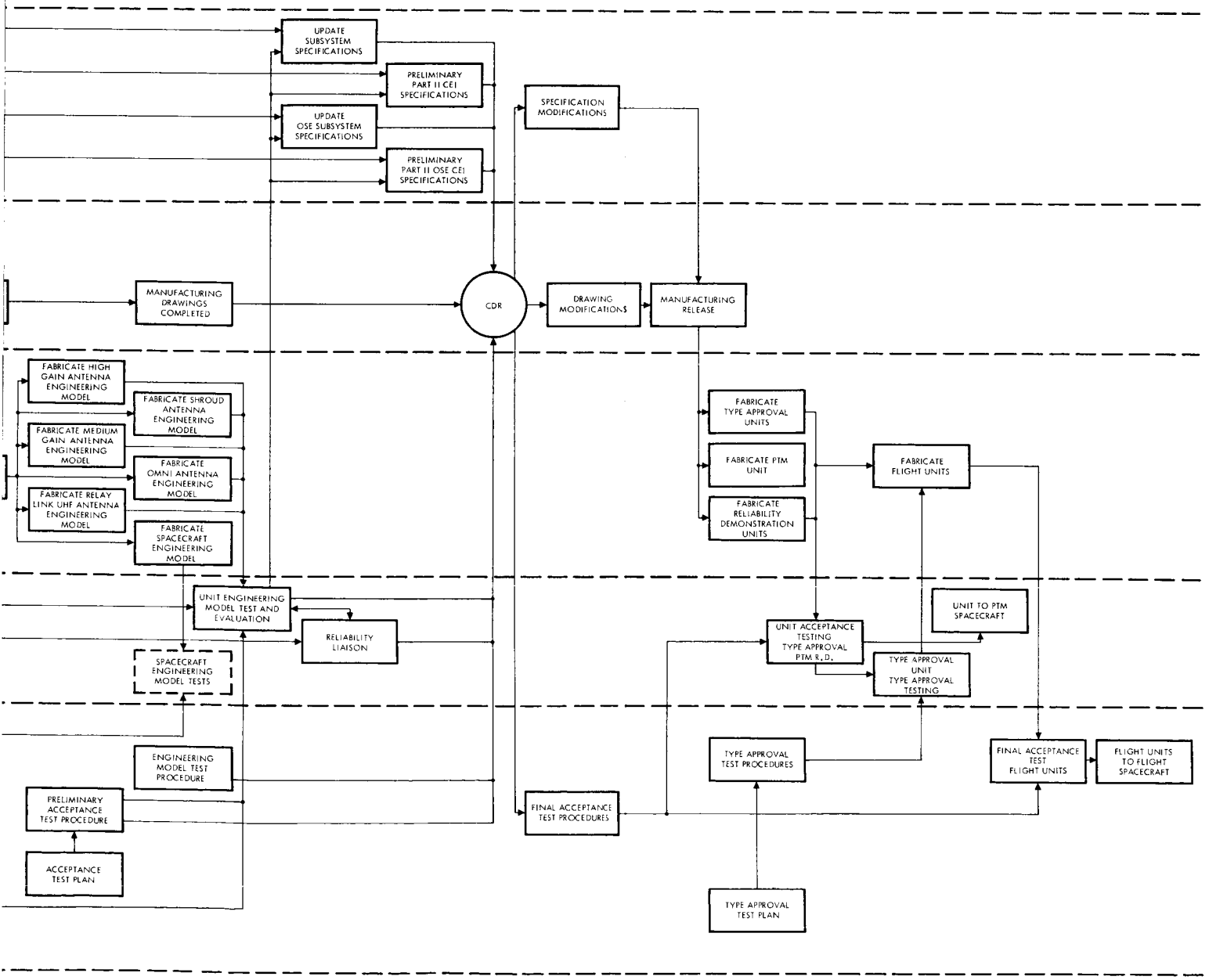


Figure 27. Design and Development of Antennas (Continued)

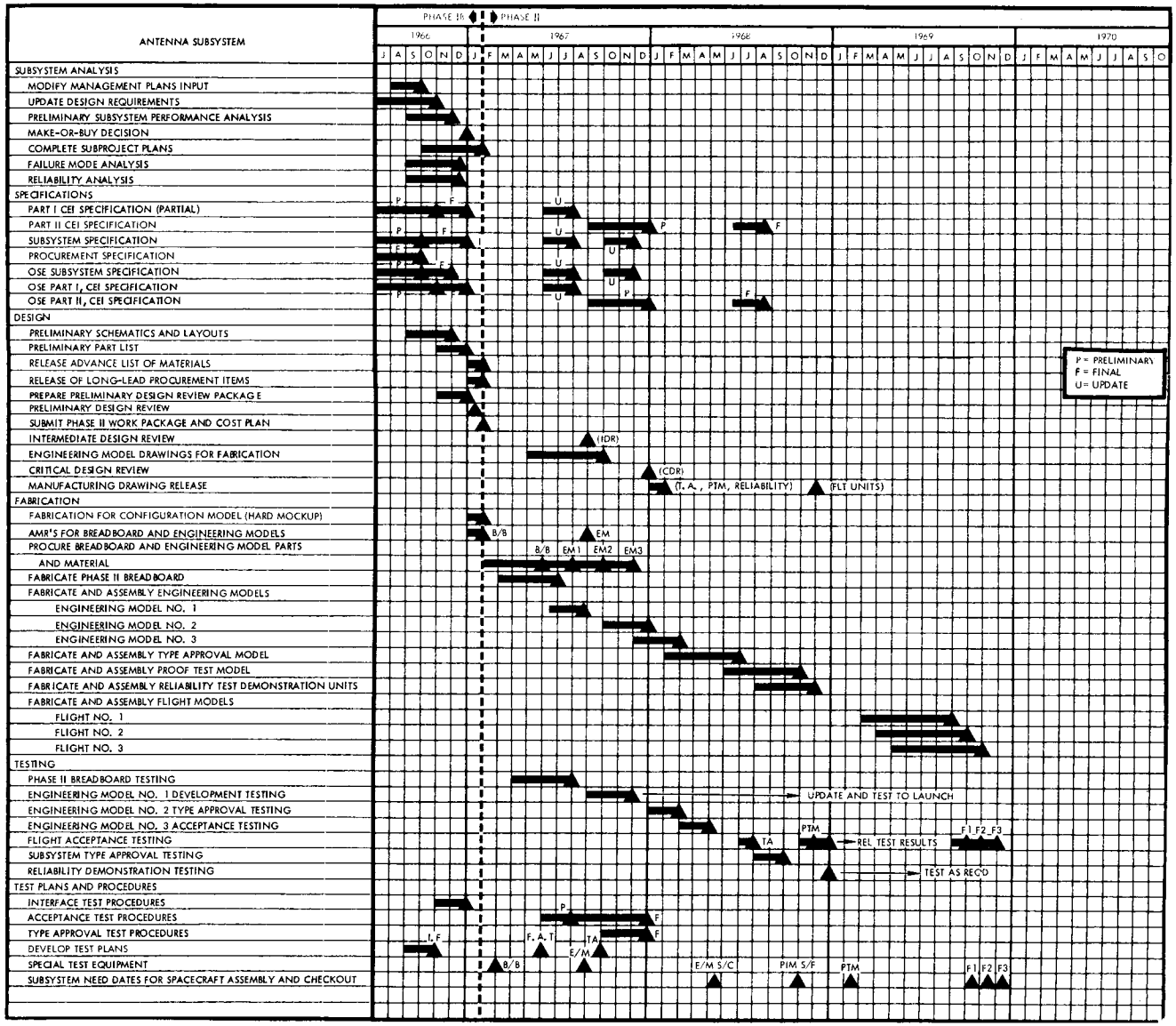


Figure 28. Schedule for Antennas

plans, test procedures, procurement of breadboard parts and materials, procurement of special test equipment, and interface and compatibility definitions and analyses. Phase IB will include the preliminary design review.

Phase II will include fabrication and test of breadboard engineering models, type approval models, proof test models, flight models, final test plans and test procedures, final specifications, critical design review, and final drawing release to manufacturing. Vendor and engineering liaison will continue until launch.

3.10.2 Design, Development, and Fabrication

a. Antenna Analysis and Design

Detailed testing that is peculiar to antennas is described in Table 24A. The over-all testing plan is described in Table 25.

The performance and interface compatibility of each antenna design will be evaluated as mission requirements are made firm. Cone and clock angles of selected ground stations with respect to spacecraft coordinates for nominal and worst-case trajectories from lift-off through six months after encounter will be studied. In addition, cone and clock angles of the capsule with respect to the spacecraft will be determined for nominal and worst-case trajectories from capsule separation to impact. The cone and clock angle data and range as a function of time of flight will establish the exact coverage required of the various antennas. All antenna failure and interference modes will be analyzed to determine their effects upon the over-all radio subsystem performance. A study will be performed to establish a method of checking the Voyager antennas in the on-stand prelaunch configuration, where all antennas are within the boost vehicle fairing and are inaccessable.

Firm performance requirements and the data obtained from a complete analysis of preliminary design concepts will be used to initiate detailed designs required for breadboard hardware.

High-Gain Antenna. Major components of the high-gain antenna include a parabolic reflector with support structure, a primary feed

Table 24A. Detailed Radio Subsystem Antenna Development Tests

NO.	TEST TITLE	TEST PURPOSE	DESCRIPTION
1	Scale Model Antenna Pattern Distribution	When interactions between antenna and spacecraft are significant, the antenna must be tested in the presence of the interacting systems. Interaction between antenna and spacecraft is expected from the omni-directional low gain and VHF antennas on Voyager.	Accurately scaled models are used for radiation pattern measurements when full-scale testing is not feasible because of the physical dimensions. Pattern distribution measurements are the three dimensional pattern measurements of relative power, per unit solid angle, radiated or received by the test antenna as a function of the polar coordinates centered at the antenna or spacecraft. The test model will be illuminated by a transmitting antenna of specified polarization and the received power will be recorded in specified conical planes about the antenna. A complete set of conical cuts will establish the three-dimensional coverage of the antenna. All tests will be performed on an antenna test range and in accordance with accepted TRW Systems antenna test procedures.
2	Scale Model Axial Ratio	Axial ratio measurements will establish ellipticities in the circularly polarized radiation fields as a function of the angular direction from the antenna axis.	For the same reasons noted in 1, scaled models may be used for these tests. Radiation patterns will be recorded at conical angles about the axis of the antenna as the antenna is illuminated by a linearly polarized transmitting antenna. The axial ratio will be determined from the ratio of the maximum to minimum field strengths representing orthogonal linearly polarized field components of the circularly polarized antenna. Tests will be performed as in 1.
3	Full Scale Pattern Distribution	The radiation characteristics of full scale antennas will be established as in 1.	Special test range preparation and calibration procedures will be employed in tests of Engineering Model and subsequent antennas. Range reflections will be checked out by testing antennas with known patterns. The illuminating antenna and model tower head will be aligned by mechanical and electrical boresighting procedures. Power and frequency will be monitored during the tests.
4	Full Scale Absolute Gain	Full scale absolute gain measurements establish the directive properties of the test antenna relative to a lossless isotropic radiator.	The recorded power level of the antenna under test will be compared with that of a known gain standard measured under identical test conditions. Radiation pattern measurement procedures of three will be used to measure the relative power radiated by the gain standard and the test antenna. Corrections will be made for relative mismatches and losses in the antennas and transmission lines.
5	Full Scale Impedance	Full scale impedance measurements will establish impedance mismatches between the antenna/spacecraft assembly and the transmission line.	Standard impedance measurements using a slotted line and VSWR meter will establish impedances as a function of frequency. Since measurements must be made in an environment that is free of obstructions which may reflect energy back to the antenna, special impedance towers or fixtures must be utilized. Impedance measurements must be performed on the full-scale antenna on a full-scale model of the spacecraft where interaction is evident.
6	Full Scale Axial Ratio	Full scale axial ratio measurements will establish ellipticities in the circularly polarized radiation fields as a function of the angular direction from the antenna axis.	Axial ratios will be determined as in 2.

Table 24A. Detailed Radio Subsystem Antenna Development Tests (Continued)

NO.	TEST TITLE	TEST PURPOSE	DESCRIPTION
7	Full Scale VSWR	Full scale VSWR measurements will establish impedance mismatches and insertion losses for the elements.	Voltage standing wave measurements are made by sampling the field along a transmission line connected to the test component. VSWR's are obtained from a VSWR indicator at each test frequency.
8	Full Scale Beamwidth	Full scale beamwidth measurements will assure compliance with radiation pattern coverage requirements.	Beamwidths will be established by pattern distribution and gain measurements in specified planes normal to the antenna aperture.
9	Insertion Loss	Insertion loss measurements will establish RF path losses through the components.	VSWR or RF power measurements will be used to determine insertion losses.
10	Isolation	Isolation measurements will establish RF leakage, or coupling between antenna elements as a function of frequency and/or location.	Isolation will be determined by measuring received power at one element as the other element is radiating. Measurements must be in an environment free of reflecting obstacles and RF sources.
11	Coupling	Coupling measurements will be made to establish the directivity of the coupler.	Measurement of power outputs relative to the input and measurement of insertion losses will establish the effective coupling or power division for the coupler.
12	Power Breakdown Measurements	Power breakdown measurements will assure the power handling capacity of the components under space environments.	High power will be applied to the test units in simulated space environments where breakdown or survival will be observed.

Table 25. Over-All Antenna Test Matrix

	Pattern Distribution	Gain	Impedance	Axial Ratio	Power Breakdown	Vibration	Shock	Thermal	Magnetic	Humidity	Flight Acceptance Test	Type Approval Test
High-Gain Antenna												
Breadboard assembly	X	X	X	X								
Engineering model assembly	X	X	X	X		X	X	X	X	X		
Type approval model assembly										X	X	X
Proof test model assembly											X	
Breadboard feed	X	X	X	X								
Engineering model feed	X	X	X	X	X	X	X	X				
Proof test model feed											X	
Medium-Gain Antenna												
Breadboard assembly	X	X	X	X								
Engineering model assembly	X	X	X	X		X	X	X	X	X		
Type approval model assembly										X	X	X
Proof test model assembly											X	
Breadboard feed	X	X	X	X								
Engineering model feed	X	X	X	X	X	X	X	X				
Proof test model feed											X	
Low-Gain Antenna												
Breadboard assembly	X	X	X	X								
Engineering model assembly	X	X	X	X	X	X		X	X	X		
Type approval model assembly										X	X	X
Proof test model assembly											X	
UHF Antenna												
Breadboard assembly	X	X	X	X								
Engineering model assembly	X	X	X	X		X		X	X	X		
Type approval model assembly										X	X	X
Proof test model assembly											X	

	VSWR	Insertion Loss	Power Breakdown	Vibration	Thermal	Flight Acceptance Test	Type Acceptance Test	
Rotary Joint								
Breadboard assembly	X	X						
Engineering model assembly	X	X	X	X	X			As unit and as part of gimbal assembly
Type approval model assembly						X	X	Part of gimbal assembly
Proof test model assembly						X		Part of gimbal assembly

antenna, and a coaxial feed cable. Studies will be performed to evaluate environmental, mechanical, and electrical requirements of each component to establish materials, manufacturing tolerances, manufacturing processes, and development test procedures. Design of the structure will be evaluated in detail through sample section fabrication and test techniques to establish optimum sizes and configurations in terms of minimum weight and deformation in the mission environments. The primary feed horn will be analyzed to assure proper illumination of the reflector and to assure its compatibility with mounting and feed line provisions. Radiation pattern degradation due to random and systematic reflector errors, aperture blockage by the feed and support structure, and misfocusing of the feed will be evaluated in detail to establish design goals and development test plans for the empirical evaluation of critical design parameters. All interface problems will be evaluated so that the complete antenna assembly performance can be evaluated with respect to the over-all mission requirements. The scattering of energy from the reflector surface will be evaluated with respect to thermal control of this area.

Medium-Gain Antenna. The major components of the medium-gain antenna include an elliptical paraboloid reflector with support structure, a primary feed antenna, and a coaxial feed cable. The analysis and design tasks required for the medium-gain antenna are identical to those required by the high-gain antenna. On the medium-gain antenna however, special emphasis will be placed on the evaluation of the asymmetrical aperture of the reflector and the shaping of the primary feed pattern for optimum illumination of the reflector.

Low-Gain Antenna. The low-gain antenna consists of a cup-turnstile element mounted on a deployable support boom. The major components will be the antenna element and the boom. The feasibility of utilizing the boom for the coaxial feed line, thereby reducing coaxial cable loses, will be evaluated. Mechanical, environmental, and electrical performance of each component will be analyzed to determine design parameters, assure survivability of the antenna during boost and during firing of the spacecraft engine, and verify the antenna radiation pattern coverage. Since the low-gain antenna will also be

used in the shroud, an analysis to evaluate the RF window design will also be undertaken. From the analysis of the low-gain antenna concepts, detailed component designs will be initiated and interface specifications for the shroud RF window will be generated. Thermal effects of engine plume heating will be analyzed.

b. Rotary Joint

The high- and medium-gain antenna gimbals require coaxial rotary joints to enable the transmission of RF energy between the transmitters and the antennas. The design is based upon the use of non-contacting, choke-coupled, coaxial rotary sections which are supported and aligned by the gimbal bearing system. The resulting joint is frictionless and sufficiently compact to fit within the gimbal axes. Mechanical, environmental, and functional performance analyses will be performed to establish optimum design parameters. The effects of multipacting within the rotary joint will be studied.

c. Capsule Radio Link Antenna

The capsule radio link antenna is a quad-spiral array whose elements are cavity-backed, two-arm Archimedian spirals. Each element is fed by a balun which may be integrated with the vacity of feed transmission line. A study will be performed to establish the feed network design including the hybrid ring power divider network. With the design of a feed network established pointing requirements for the antenna. The effects of thermal gradients upon antenna performance will be evaluated.

3.10.3 Test Program

a. Breadboard Tests

Initial antenna breadboard activity will consist of scale model testing. Electrically functional scaled models of the low-gain S-band antenna and the capsule link antenna will be utilized with a scaled Voyager flight spacecraft electrical model to determine antenna pattern distribution for suitable antenna mounting points on the spacecraft structure. The influence of the structure upon the patterns will be experimentally evaluated.

Later breadboard activity will include fabrication of operating antenna assemblies at full scale. The feed horn designs will be experimentally verified by measuring their radiation and impedance characteristics. The paraboloid illumination will be optimized and losses within the feed systems minimized. Parallel studies of material types and fabrication techniques will be undertaken to provide engineering model design data. Breadboard reflectors will be procured for measurements of secondary pattern characteristics and further optimization of the paraboloid illumination. The low-gain and UHF antennas will also be fabricated full scale to measure impedance and full-scale radiation characteristics. For final pattern distribution data a full-scale electrical breadboard of the Voyager spacecraft will be fabricated and utilized with the full-scale antenna breadboards. Rotary joint breadboards will be fabricated and evaluated for design usage and performance.

b. Engineering Model Tests

Engineering model (EM) activity will begin when design data has been obtained from the functional breadboards. Complete EM designs will be finished and accurate engineering models fabricated in accordance with the EM drawings. The completed EM's will be evaluated functionally and subjected to selected critical environmental exposures in accordance with preliminary proof test procedures. Variations in electrical performance will be determined between environmental exposures. Full inspection and reliability monitoring will prevail on all engineering models. Three sets of EM equipment will be fabricated for identical functional and environmental testing. The first set of EM will be retained by the antenna subproject; the second set will be integrated into the EM spacecraft, where complete subsystem testing will first be performed; and the third set will be integrated into the propulsion interaction model spacecraft. Magnetic compatibility testing will be performed on EM No. 2.

c. Type Approval Testing

Type approval tests are performed to verify that design requirements can be attained in environments which are in excess of

that expected during the mission. The tests are designed to be non-destructive. They include functional and environmental tests at the assembly and subsystem level. The tests are:

- Vibration
- Shock
- Magnetic
- Humidity
- Ethelyne oxide compatibility
- Thermal shock
- Electromagnetic compatability
- Thermal vacuum

d. Flight Acceptance Testing

Flight acceptance testing is to confirm that workmanship and quality standards have been satisfied. Flight acceptance tests include tests at an assembly level and at the subsystem level. Four tests are planned:

- Vibration
- Magnetic
- Thermal vacuum
- Electromagnetic compatibility

3.11 Computing and Sequencing Subsystem Development

The computing and sequencing (C and S) subsystem is a special-purpose sequencer containing a 256-word central core memory system, a system of clocks to time stored commands and data, a command input unit to load memory, sensor signal input logic to synchronize clock operations with external events, a function generator to issue pointing angles to the high-gain antenna, two accelerometer counters to provide engine cutoff signals, and special registers for telemetry sampling of C and S data and command events.

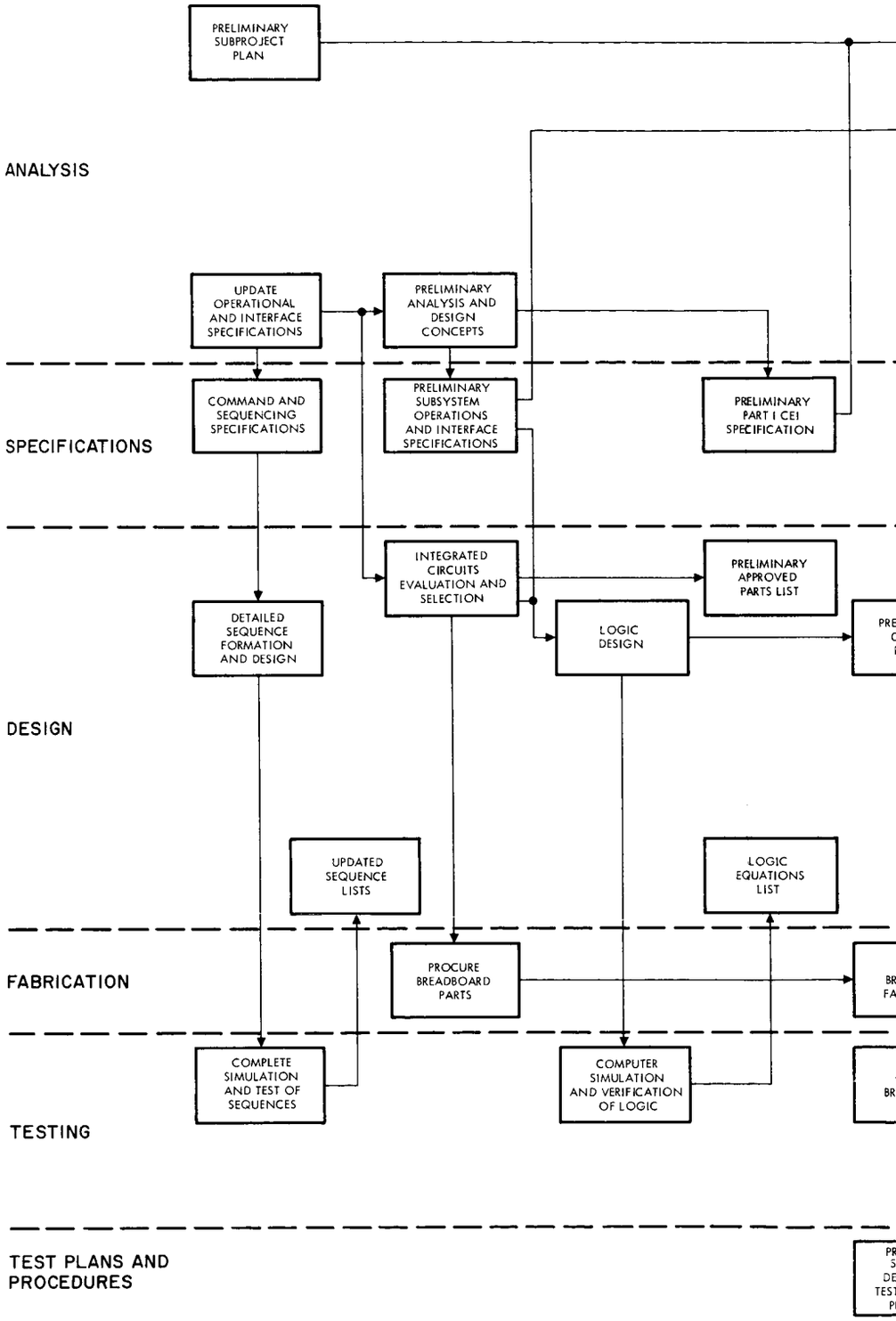
3.11.1 Summary

Development of the C and S is shown in Figure 29, which covers design and product development and manufacturing. During Phase IB the subsystem requirements are analyzed in detail, and the sequence formulation, logic design, circuit design, and the over-all packaging concept are developed to sufficient detail to permit critical breadboard testing. Design audits at the end of the sixth, 15th and 26th weeks provide opportunity to review the subsystem requirements and design concepts. A formal design review (PDR) in the 29th week integrates the results of design and analysis, examines the data of the critical breadboard tests, reviews the development test plans, and provides direction for Phase II operations.

During Phase II, detailed design is completed and engineering and flight models are fabricated and tested. A design audit is held to review the detail circuits and packaging designs and to examine the results of detail circuit breadboard tests before proceeding with the fabrication of engineering model (EM) No. 1. After EM No. 1 fabrication and testing an Intermediate Design Review (IDR) is conducted. This IDR examines the results of EM No. 1 testing, which includes acceptance and development testing, reviews the materials lists and type approval (TA) test plans for EM No. 1, and flight acceptance (FA) test plans for EM No. 2 and 3, and releases drawings for EM No. 2 and 3.

Following this review, EM's No. 2 and 3 are fabricated and FA tested. Both models are fabricated to full Voyager specifications in the manufacturing division and must pass flight acceptance tests. The results of EM No. 1 TA testing and EM No. 2 and 3 FA testing are reviewed at the critical design review (CDR) and approved manufacturing drawings for TA, PTM and reliability demonstration models are released. Fabrication release for flight spacecraft and space units occurs at the completion of subsystem TA testing.

The schedule in Figure 30 for the C and S subsystem development program is based on a go-ahead date of 1 July 1966 and a manufacturing drawing release date of 31 October 1968. An additional schedule restraint is the initiation of Phase II on 1 February 1967. All hardware,


PRELIMINARY SUBPROJECT PLAN
ANALYSIS
UPDATE OPERATIONAL AND INTERFACE SPECIFICATIONS
PRELIMINARY ANALYSIS AND DESIGN CONCEPTS
COMMAND AND SEQUENCING SPECIFICATIONS
PRELIMINARY SUBSYSTEM OPERATIONS AND INTERFACE SPECIFICATIONS
PRELIMINARY PART I CEI SPECIFICATION
SPECIFICATIONS
INTEGRATED CIRCUITS EVALUATION AND SELECTION
PRELIMINARY APPROVED PARTS LIST
DETAILED SEQUENCE FORMATION AND DESIGN
LOGIC DESIGN
DESIGN
UPDATED SEQUENCE LISTS
LOGIC EQUATIONS LIST
PROCURE BREADBOARD PARTS
FABRICATION
COMPLETE SIMULATION AND TEST OF SEQUENCES
COMPUTER SIMULATION AND VERIFICATION OF LOGIC
TESTING
TEST PLANS AND PROCEDURES

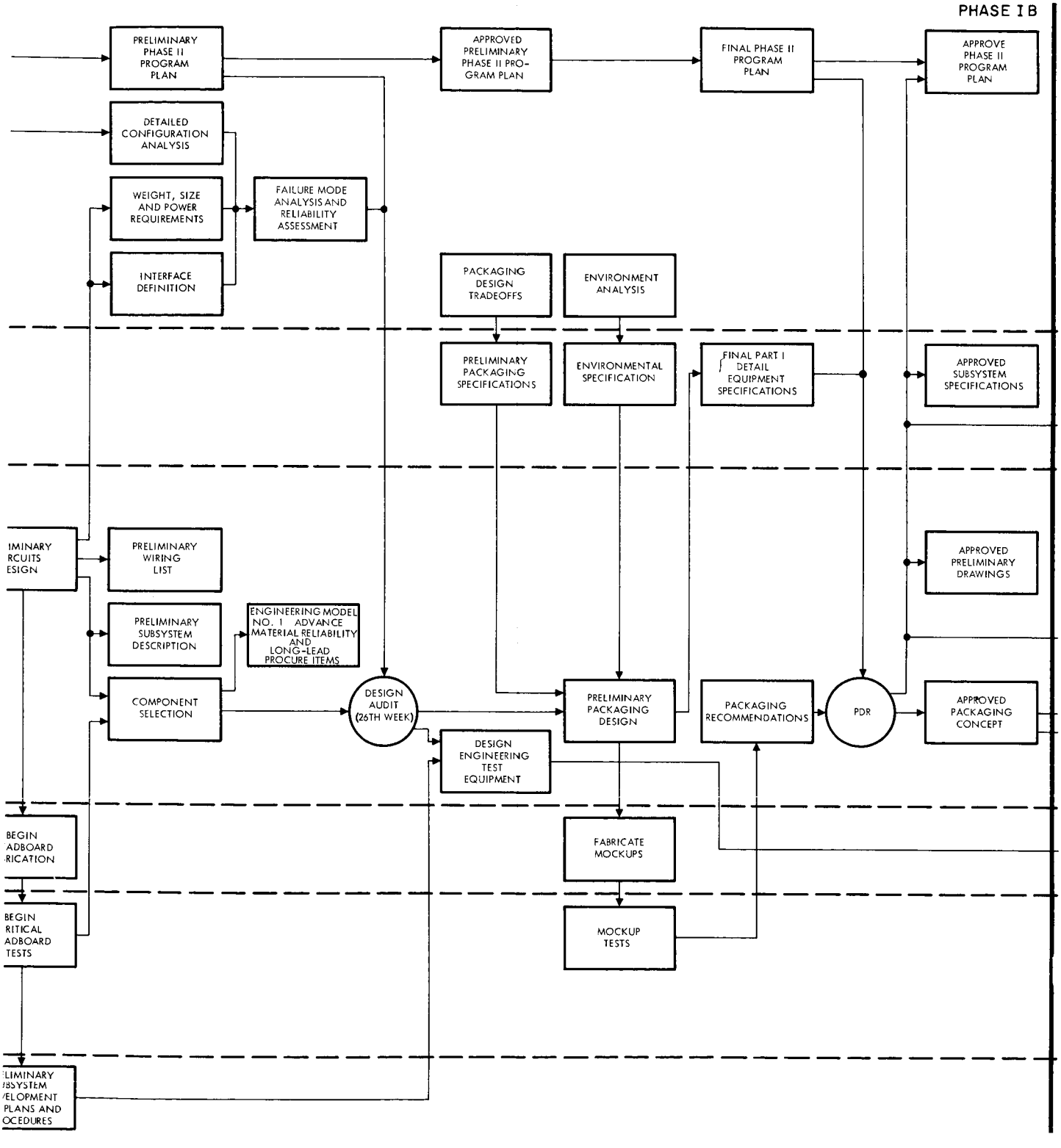
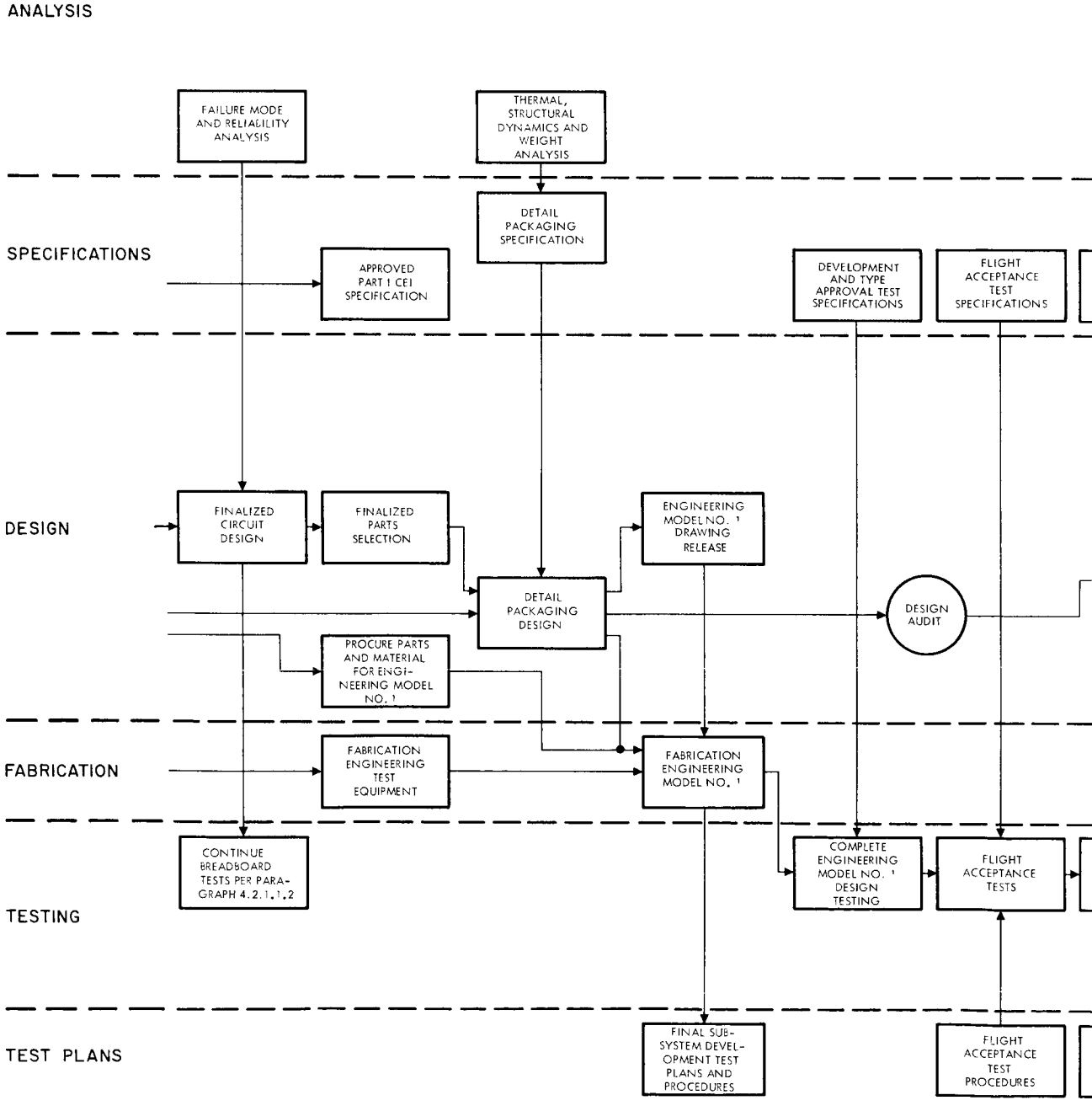
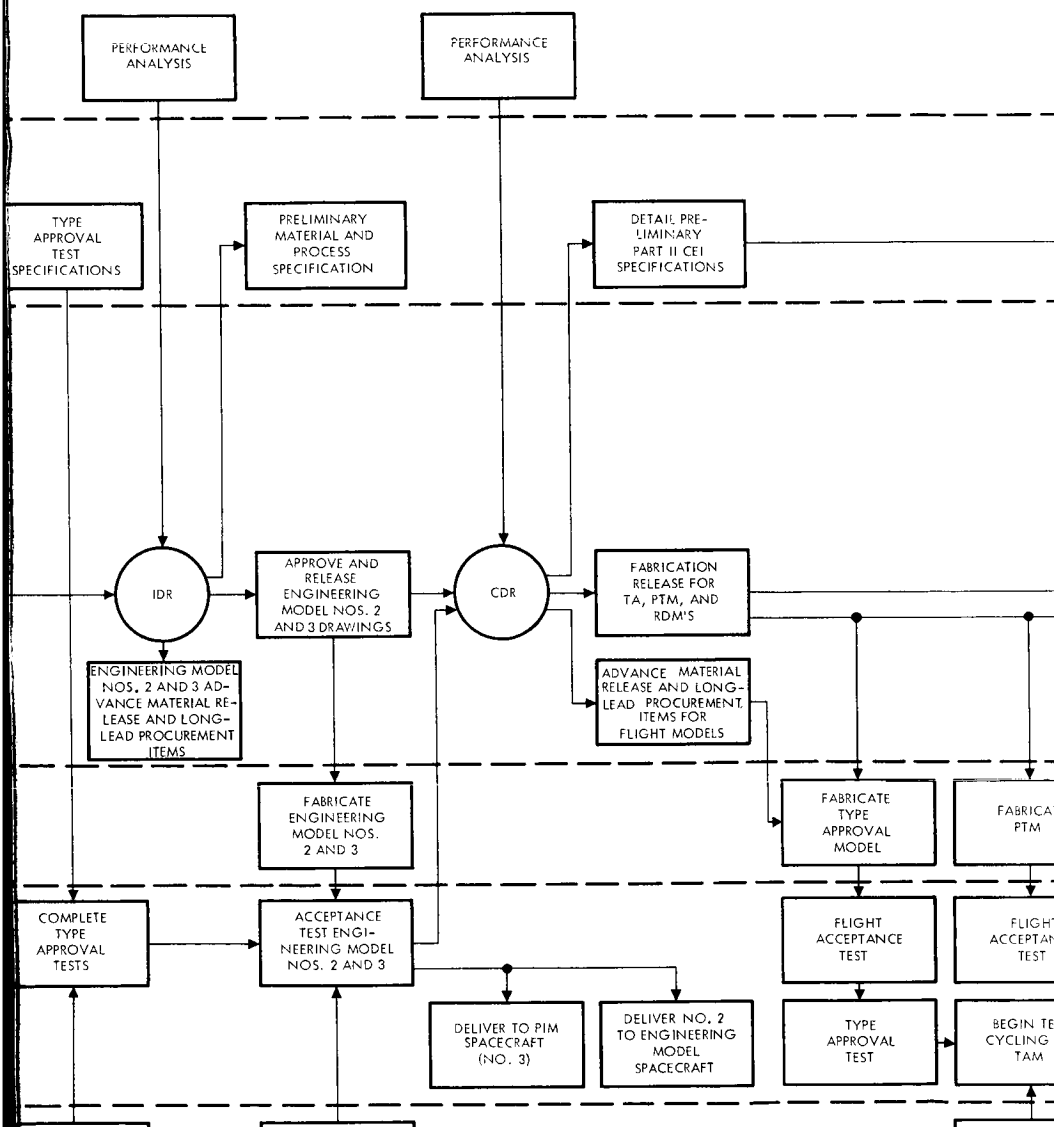


Figure 29. Design and Development of Computing and Sequencing Subsystem

PHASE II


ANALYSIS
FAILURE MODE AND RELIABILITY ANALYSIS
THERMAL, STRUCTURAL DYNAMICS AND WEIGHT ANALYSIS
SPECIFICATIONS
DETAIL PACKAGING SPECIFICATION
APPROVED PART 1 CEI SPECIFICATION
DEVELOPMENT AND TYPE APPROVAL TEST SPECIFICATIONS
FLIGHT ACCEPTANCE TEST SPECIFICATIONS
DESIGN
FINALIZED CIRCUIT DESIGN
FINALIZED PARTS SELECTION
ENGINEERING MODEL NO. 1 DRAWING RELEASE
DETAIL PACKAGING DESIGN
DESIGN AUDIT
PROCURE PARTS AND MATERIAL FOR ENGINEERING MODEL NO. 1
FABRICATION
FABRICATION ENGINEERING TEST EQUIPMENT
FABRICATION ENGINEERING MODEL NO. 1
TESTING
CONTINUE BREADBOARD TESTS PER PARAGRAPH 4.2.1.1.2
COMPLETE ENGINEERING MODEL NO. 1 DESIGN TESTING
FLIGHT ACCEPTANCE TESTS
TEST PLANS
FINAL SUBSYSTEM DEVELOPMENT TEST PLANS AND PROCEDURES
FLIGHT ACCEPTANCE TEST PROCEDURES


PERFORMANCE ANALYSIS
PERFORMANCE ANALYSIS
TYPE APPROVAL TEST SPECIFICATIONS
PRELIMINARY MATERIAL AND PROCESS SPECIFICATION
DETAIL PRE-LIMINARY PART II CEI SPECIFICATIONS
IDR
APPROVE AND RELEASE ENGINEERING MODEL NOS. 2 AND 3 DRAWINGS
CDR
FABRICATION RELEASE FOR TA, PTM, AND RDM'S
ENGINEERING MODEL NOS. 2 AND 3 ADVANCE MATERIAL RELEASE AND LONG-LEAD PROCUREMENT ITEMS
ADVANCE MATERIAL RELEASE AND LONG-LEAD PROCUREMENT ITEMS FOR FLIGHT MODELS
FABRICATE ENGINEERING MODEL NOS. 2 AND 3
FABRICATE TYPE APPROVAL MODEL
FABRICA PTM
COMPLETE TYPE APPROVAL TESTS
ACCEPTANCE TEST ENGI-NEERING MODEL NOS. 2 AND 3
FLIGHT ACCEPTANCE TEST
FLIGHT ACCEPTAN TEST
DELIVER TO PIM SPACECRAFT (NO. 3)
DELIVER NO. 2 TO ENGINEERING MODEL SPACECRAFT
TYPE APPROVAL TEST
BEGIN TE CYCLING TAM
TYPE APPROVAL TEST PROCEDURES
APPROVED ENGINEERING MODEL NOS. 2 AND 3 ACCEPTANCE PROCEDURES
TA TES PROCEDU ON TA

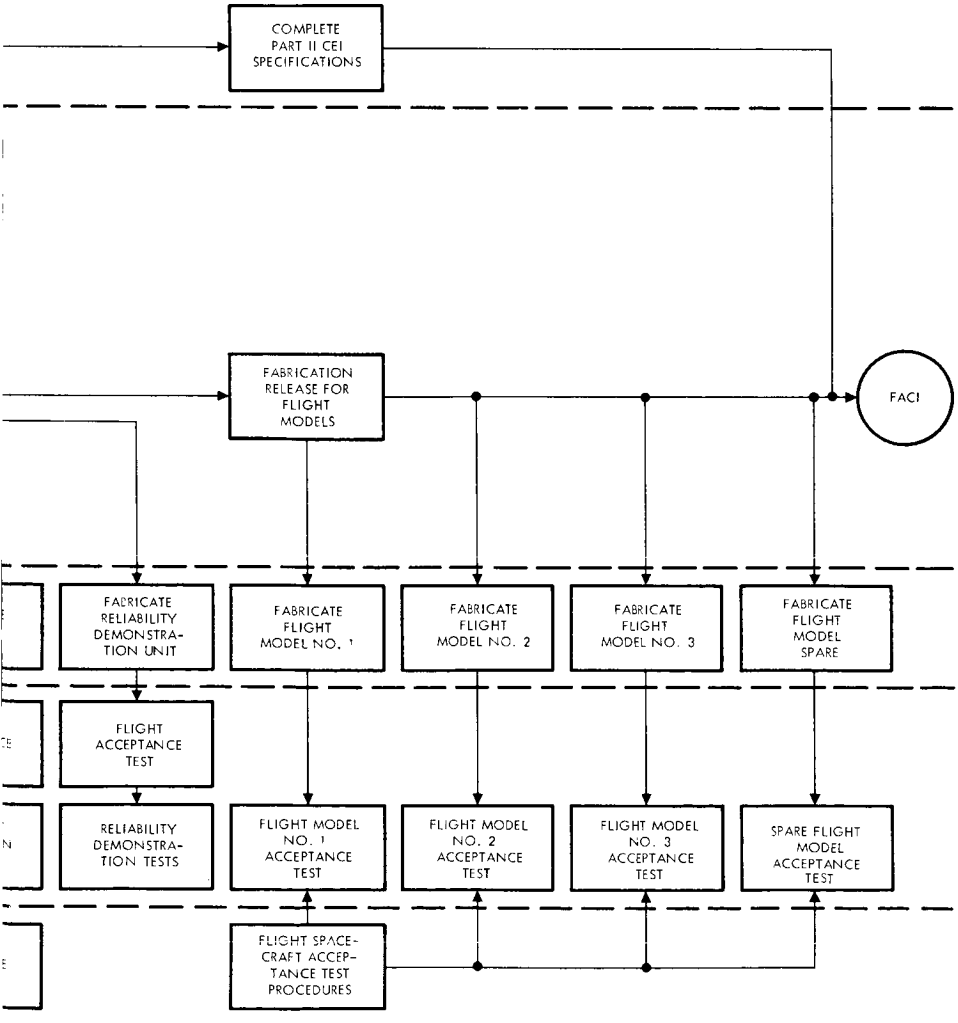


Figure 29. Design and Development of Computing and Sequencing Subsystem (Continued)

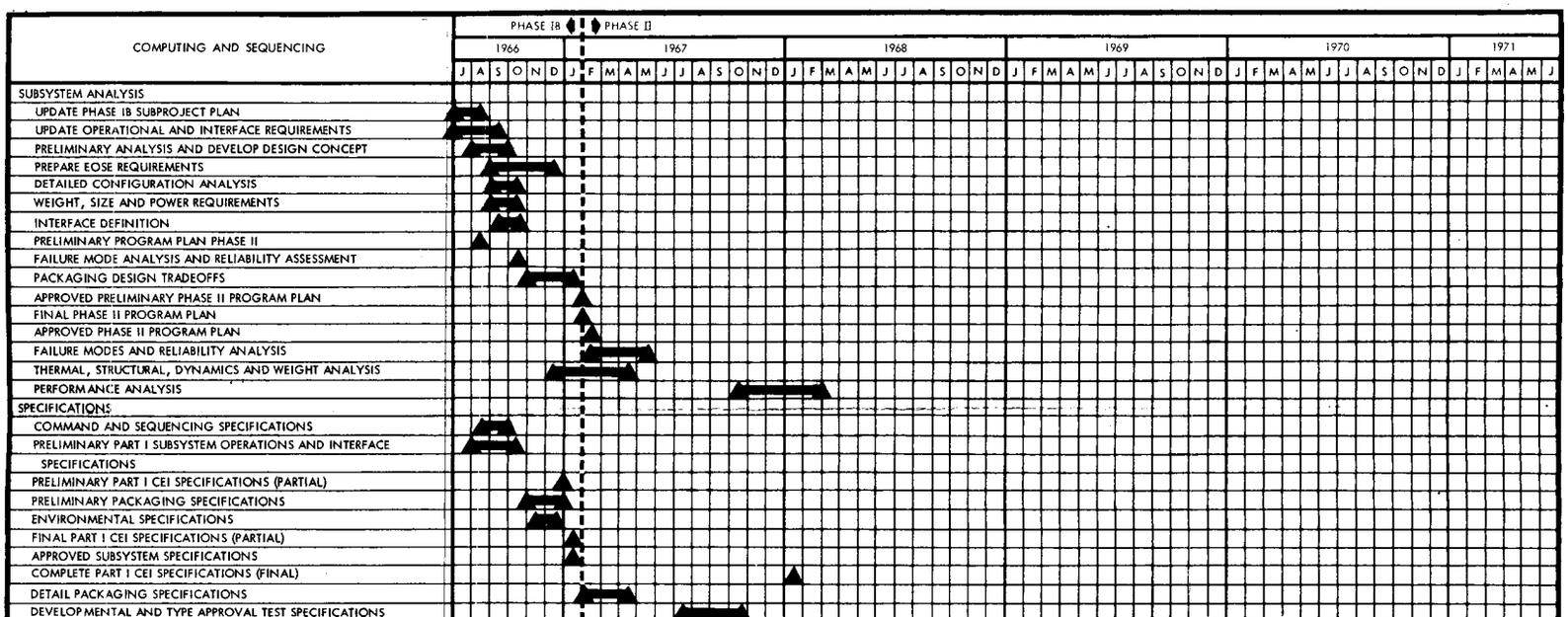

COMPUTING AND SEQUENCING
PHASE IB
PHASE II
1966
1967
1968
1969
1970
1971
J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J
SUBSYSTEM ANALYSIS
UPDATE PHASE IB SUBPROJECT PLAN
UPDATE OPERATIONAL AND INTERFACE REQUIREMENTS
PRELIMINARY ANALYSIS AND DEVELOP DESIGN CONCEPT
PREPARE EOSE REQUIREMENTS
DETAILED CONFIGURATION ANALYSIS
WEIGHT, SIZE AND POWER REQUIREMENTS
INTERFACE DEFINITION
PRELIMINARY PROGRAM PLAN PHASE II
FAILURE MODE ANALYSIS AND RELIABILITY ASSESSMENT
PACKAGING DESIGN TRADEOFFS
APPROVED PRELIMINARY PHASE II PROGRAM PLAN
FINAL PHASE II PROGRAM PLAN
APPROVED PHASE II PROGRAM PLAN
FAILURE MODES AND RELIABILITY ANALYSIS
THERMAL, STRUCTURAL, DYNAMICS AND WEIGHT ANALYSIS
PERFORMANCE ANALYSIS
SPECIFICATIONS
COMMAND AND SEQUENCING SPECIFICATIONS
PRELIMINARY PART I SUBSYSTEM OPERATIONS AND INTERFACE SPECIFICATIONS
PRELIMINARY PART I CEI SPECIFICATIONS (PARTIAL)
PRELIMINARY PACKAGING SPECIFICATIONS
ENVIRONMENTAL SPECIFICATIONS
FINAL PART I CEI SPECIFICATIONS (PARTIAL)
APPROVED SUBSYSTEM SPECIFICATIONS
COMPLETE PART I CEI SPECIFICATIONS (FINAL)
DETAIL PACKAGING SPECIFICATIONS
DEVELOPMENTAL AND TYPE APPROVAL TEST SPECIFICATIONS

Figure 30. Schedule for Computing and Sequencing Subsystem

except for early critical test breadboard, is fabricated in Phase II of the program. Vendor liaison and engineering liaision to manufacturing continues until launch time.

3.11.2 Design, Development, and Fabrication

a. Subsystem Analysis

The performance of the C and S and compatibility of the design with its interfaces must be evaluated as mission requirements are firmed. This analyses will be used to prepare firm requirements to initiate detailed designs required for the implementation of breadboard hardware.

A detailed study will be performed to develop a checkout procedure for the C and S in the prelaunch configuration. Provision must be made for a speeded-up test capability both for subsystem and system checkout, and operational procedure must be worked out in detail. Consideration must be given to the classification of commands into groups capable of being executed on the launch pad. These requirements must be established early enough to insure that they influence the C and S mechanization. Special logic and circuits have to be designed to permit the speedup of the sequencing operations, so decisions are required to define the method of speedup, whether only dead-time, i.e., time between commands, shall be reduced or whether some commands shall also be performed in reduced time, etc.

Detailed analyses are needed to provide ranges and limits on testing and to prepare developmental test plans. The organization of the test phases for flight acceptance and type approval must be examined in detail and test flow diagrams and schedules established. These plans must be completed in time to prepare data tapes for the unit test equipment and operational support equipment and must be ready before fabrication of the test unit is complete.

Prior to the fabrication of the first EM, detailed thermal, dynamics, and electrical noise analyses are required to design the C and S package. Fortunately, much of this analysis has already been done by TRW in its design and development of the LEM abort guidance

system and the results can be applied directly to the C and S package. The LEM abort guidance system has essentially the same environmental conditions as the Voyager C and S, especially since the same booster system is to be used, and most of the packaging problems have been analyzed and solved. It is anticipated that the LEM circuits can apply directly, so the analyses should also apply.

Analysis is also required to establish weight, size and power requirements of the C and S, and failure mode and reliability analysis is needed to assess the operational capability of the C and S. This data must be fed back to the design process to update the concepts.

A special problem is the analysis of the automatic antenna pointing function. The linear function approximation scheme will be examined in detail and error apportionment studies will be made. The cone and hinge angle functions will be formulated and compared to the requirements imposed by various trajectories. Computer simulations will be employed to provide evaluation data.

b. Design

The first task for the design of the C and S subsystem is to select and evaluate integrated circuits as determined by parameters evolved from the operational and interface requirements. The integrated circuits which are selected are then employed in the logic design. Other parameters associated with the logic design are determined by the operational and interface specifications and the control, command, and sequencing operations of the C and S subsystem. The results of the logic design, along with the selection of the integrated circuits, are used to establish the design requirements for the preliminary circuits. The results of the logic and preliminary circuits design contribute to the selection of the components to be used in the C and S subsystem and breadboard fabrication. The results of the critical breadboard tests are used in combination with the preliminary circuits design and logic design for components selection.

Upon completion of the second design audit, design of the engineering and test equipment for the C and S is begun. Preliminary packaging design using results of the packaging and environmental

analyses are completed and submitted with the preliminary design review (PDR). The PDR establishes the approved packaging concept. In addition, final circuit design for the C and S subsystem is frozen and the parts selection made. After the PDR, breadboard tests are continued and detailed packaging design is developed in accordance with the approved packaging concept and the final circuit design. The approved detail package design is the basis for fabrication of EM No. 1.

c. Hardware

A critical unit of the C and S subsystem operation is the memory system. The scan and time compare logic which controls cycling through memory for mode data and elapsed time comparison and for issuing memory data to the telemetry register will be breadboarded and are scheduled for the critical breadboard test phase.

Another unit of special importance in the C and S is the accelerometer counter. Special tests must be conducted as early as possible to evaluate its operation and for this reason critical breadboard tests are required.

d. Flight Data Tapes

An integral part of the design of the C and S is the formulation and preparation of detailed sequence lists to be stored in the C and S memory and controlled through the mode and time compare circuits. These sequences must be checked by computer simulation and flight data tapes must be prepared and delivered with the C and S.

Variations in command times, linear function approximations, and other mission or trajectory dependent factors have to be provided for alternate launch dates and trajectories. Careful analyses are required to establish the sensitivity of the C and S operations to parameter variations. Analytical techniques and computer simulations must be employed to estimate this sensitivity and to establish ways and means to update the C and S data at the launch station. The data update procedures must be compatible with the mission operations and ground support equipment.

3.11.3 Development Test Program

The development test program includes Phase IB as well as Phase II testing. As early as possible in the design of the C and S, breadboards must be fabricated to test critical items such as the memory system and the accelerometer counter logic. As the circuits and packaging details are established, breadboard testing continues and preparations are begun for fabricating and testing the EM's. Construction and testing of EM No. 1 proceeds through acceptance tests, development tests and type approval tests.

EM No. 2 and 3 receive FA testing prior to delivery to the EM and PIM spacecrafts, respectively. The breadboard tests continue to launch.

a. Phase IB Tests

Phase IB tests include sequence and logic simulation and checkout procedures and critical breadboard tests. Sequences have to be identified in detail and the logic equations have to be specified. Standard computer simulation techniques are employed to simulate the logic and to reduce and simplify the equations. One program iterates on the generation of the logic equations and removes unwanted logic and circuit redundancy. Second order optimization on power, cost, timing, and loading is performed to refine equations.

A loading check program examines the load distributions on the logic elements using initializing data with the equations. A simulation program checks the logical consistency of the detailed subsystem design using the bit-level detail logical equations.

Detailed breadboard testing evaluates the logical and electrical design by identifying worst-case parameters, studying transient responses, loading level and noise sensitivity, and establishing voltage and temperature limits.

Scopes, meters, oscillators, and pulse generators are used to isolate faults and electrical design weakness. Logic tests, voltage level tests, and response characteristics tests using a unit tester, evaluate the operational capability of the system under test.

Generally, the purpose of the breadboard tests is to ascertain whether parts meet the requirements of the circuit design specifications; to obtain derating, stability, and failure mode information; to determine whether screening procedures are advisable, and establish screening criteria; to determine changes in preliminary circuit specification; and to verify design margins and performance.

b. Phase II Tests

Phase II testing includes breadboard development and design margin testing; EM development, FA and TA testing; manufacturing tests; subsystem TA testing and subsystem FA testing.

Breadboard testing includes the further design and development testing of an updated breadboard model. This model is also used to determine the effects of temperature, power supply variations, and transient wave forms, and is exercised throughout the program until launch. These tests also include materials and process testing and parts selection testing.

EM No. 1 tests investigate the performance of the subsystem under various environmental conditions.

Manufacturing and in-process tests are employed to insure the correctness of the manufacturing process. These tests assure that parts and components are properly assembled and perform within specified limits. Components are selected for precision alignment or calibration, and performance checks are made on completed subassemblies. Tests are performed on components under extreme temperatures and voltages before they are assembled into higher units. Power supply and memory card subassemblies are tested by special purpose test sets which supply normal and abnormal voltages or master clock pulses, cycling time signals, mode select and static input signals as appropriate. Production model units of the C and S are functionally tested prior to the start of acceptance tests. Tests are run at nominal, high, and low voltages at room temperature.

Finally, environmental tests are used to demonstrate that the C and S under test is representative of that configuration and will satisfy

the operational and functional requirements of that model. The acceptance tests proceed sequentially with operational cycles to estimate the effect of the environments on C and S performance. The tests include vibration and thermal vacuum tests.

Critical environments tests are employed to provide confidence that the C and S will pass the design limit load tests and to verify safety margins for the Voyager mission. Selected worst-case environments and incrementally increasing dynamic tests are involved.

The test program discussed above is presented in more detail in Table 26.

3.12 Guidance and Control Subsystem

The major function of the guidance and control (G and C) subsystem is to establish and maintain three-axis stabilization of the spacecraft at all times after separation of the spacecraft from the launch vehicle. It also provides velocity data for trajectory and orbit adjustment modes, pointing controls for directional antennas, and signals based on Mars orbit events for the operation of science instruments.

Pitch and yaw error signals from the sun sensor and roll error signals from the Canopus sensor are used to control spacecraft attitude during cruise and orbit about Mars. The gyro reference assembly provides control signals to maintain inertial reference during the initial separation maneuver, midcourse corrections, orbit insertion and trim maneuvers, and during periods of sun or Canopus eclipse. The accelerometer supplies the signal to terminate engine firing during velocity corrections.

The control torques are produced by expulsion of nitrogen gas through nozzles located so as to produce couples about the spacecraft control axes. A two-level thrust system is utilized, the low-thrust mode during cruise and the high-thrust mode during all other flight phases. During propulsion maneuver modes, thrust vector control is provided by gimballing the engine in pitch and yaw with roll control supplied by the gas thrusters.

The antenna drive system, which is completely independent of other G and C subsystem functions, positions the high-gain antenna about its

Table 26. Computing and Sequencing Subsystem Development Tests

Name of Test	Item Being Tested	Purpose	Description	Special Equipment
Sequence Check	Command list order by occurance in subsets	To verify sequencing capability of C and S	Uses detailed sequences simulated on computer	Simulation routines
C and S Logic	Logic equations	To verify logic analyses and designs	Bit-by-bit simulation of operations of C and S	Logic simulation routines
Loading Check	Logic equations and initializing data	To check load distributions on logic elements	Uses binary electronics equipment check on equations	Computer program
Parts Selection	Integrated circuits, transistors, discretes	To obtain derating, stability and failure mode information	Evaluates devices for conformance to specs, vendor ratings, etc.	Wide range of electrical equipment and environmental test equipment
Circuit	Breadboard circuits	To verify design	Subject equipment to temperature, voltage extremes, noise	General Purpose (GP) test equipment, thermal chambers
Subassembly	Subassemblies of C and S	To verify design requirements, functional requirements, margins	Subjecting units to temperature and voltage extremes, noise	GP test equipment
Materials and Processes	Multilayer boards, solder processes, plastics, finishes, metals	To check conformance to requirements	Evaluate vendor products	GP test equipment
Dynamics Evlauation	Logic card frames	To verify structural adequacy	Vibrate items in three directions with increasing serverties	Vibration table and gages GP test equipment
Thermal Evaluation	Welded modules	To verify thermal assumptions	Measure stabilized temperature at various power dissipations	Vacuum chamber, temperature pickup, GP test equipment
Mockup	Wiring model of core stock	To verify wiring design	Wire core stock to surrounding boards to allow hinging of them	Ohmmeter
Circuit	Logic circuits breadboards	To verify design	Subject circuits to temperature, voltage and noise extremes	GP equipment, thermal chamber
C and S Operational	Preproduction model C and S	To verify operational and interface requirements	Provide input power and signal control, and output signal control	GP equipment, special test programs
In-process	Components, subassemblies and assemblies	To assure correctness of manufacturing process, check parts and units	Use C and S test set to exercise card functions	GP equipment, memory logic, power test sets, and unit test set
Flight acceptance	EM's, TA, and PTM model and all flight models	Confirm acceptable operation on a unit and subsystem level	Verify operation under a variety of induced conditions of voltage, frequency variations, noise injection, etc.	EOSE
Type approval	TA model and EM No. 1	Confirm satisfaction of design requirements under excess environment	FA tests first. Then temperature variations, vibration, shock, humidity, thermal shock, sterilization	EOSE and chamber
Life	TA Model	Confirm operability over extended period	Continuous operation at nominal conditions, periodic cycling over temperature power ranges	EOSE
Reliability demonstration	Reliability demonstration model	To conduct tests in accordance with Voyager reliability test plan	Four units: Operated under 40 degree ambient temperature for 12 months. Six units: Operated under 30 degree ambient temperature and vacuum conditions for 6 months	EOSE
Phase II breadboard	Subsystem breadboard model	To verify performance characteristics over: o Voltage and temperature extremes o Noise injection o Power frequency variation o Power transients	Update breadboard to simulate flight design; cycle through the input variations and environments; operate at nominal condition to launch	EOSE

hinge and shaft axes and the medium-gain antenna about its hinge axis in response to signals from the C and S or ground commands.

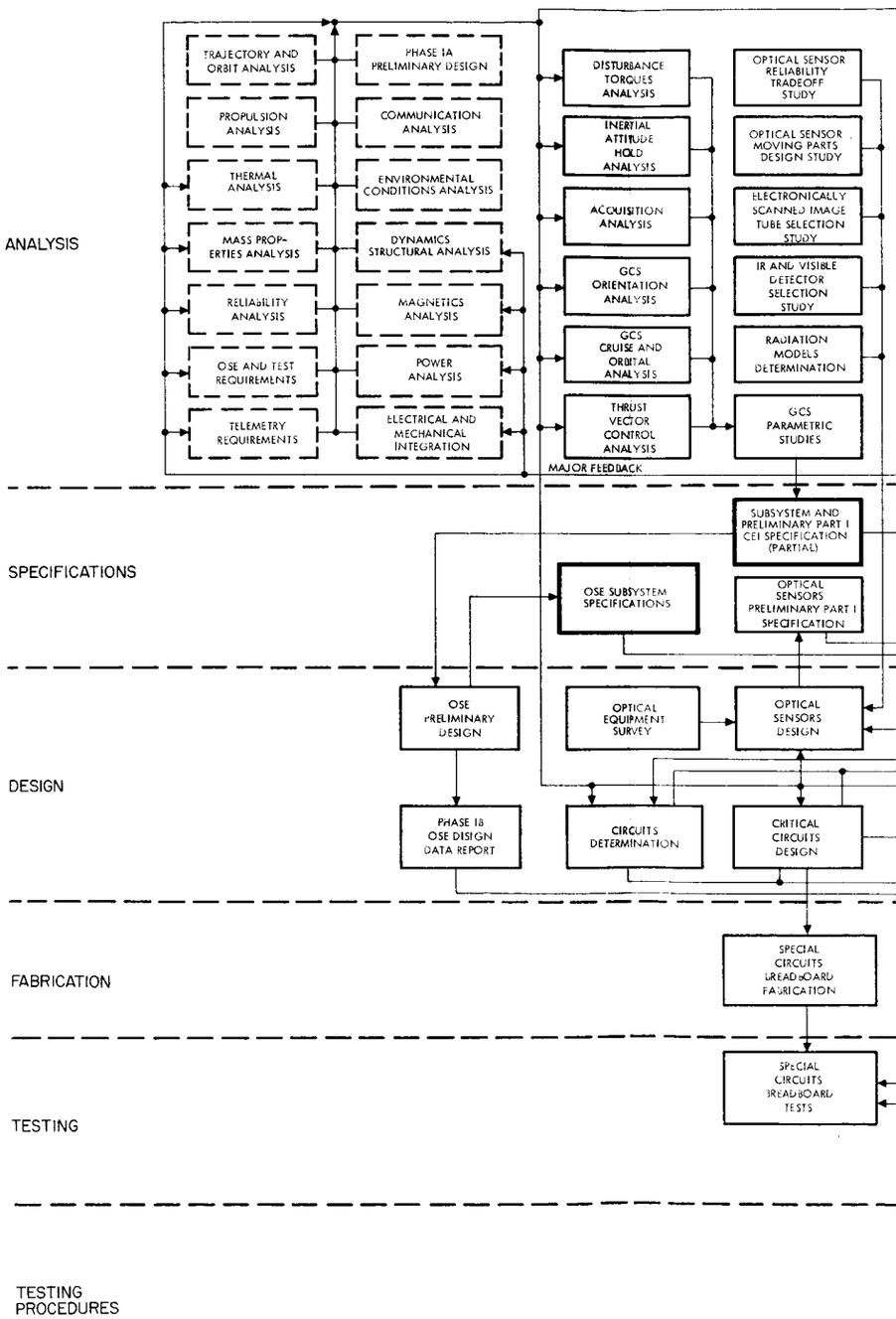
3.12.1 General

The engineering and support activities associated with the analysis, design, procurement, development, and testing of the G and C subsystem and its special test equipment are illustrated in Figure 31. The analyses required for design of the subsystem will be conducted primarily during Phase IB. The detailed design activities will largely occur during Phase II except that because of critical schedule requirements design and breadboard testing of the gyro reference assembly and critical electronics circuits will be conducted in Phase IB.

To complete development testing in time to meet critical drawing release dates, the preliminary procurement of gyros in Phase IB is indicated. Because of the long lead-time associated with high-reliability parts, preliminary procurement of critical electronic components will also be initiated in Phase IB. All other procurement will occur in Phase II.

Except for the design and breadboard testing previously mentioned all development and testing will be performed during Phase II, providing for a comprehensive program of testing including design verification, functional, environmental, life, type approval, and flight acceptance tests. Test articles will include breadboards, engineering models, proof test models, and flight models. The environmental testing will include thermal-vacuum, vibration, shock, humidity, thermal shock and magnetic properties tests.

The milestone chart in Figure 32 presents the events starting with the Phase IB go-ahead to delivery of developed subsystems to stores. Phase IB will start with a review of the requirements generated in Phase IA, modified by any new mission requirements. Guidance and control assembly, subsystem, and interface analyses will be initiated. A preliminary subsystem specification and OSE specification will be completed, together with preliminary Part I, CEI specifications for the subsystem and OSE assemblies. Subcontractor submittals will be reviewed and the Phase IB effort concluded with a preliminary design review.


ANALYSIS
SPECIFICATIONS
DESIGN
FABRICATION
TESTING
TESTING PROCEDURES
TRAJECTORY AND ORBIT ANALYSIS
PHASE 1A PRELIMINARY DESIGN
PROPULSION ANALYSIS
COMMUNICATION ANALYSIS
THERMAL ANALYSIS
ENVIRONMENTAL CONDITIONS ANALYSIS
MASS PROPERTIES ANALYSIS
DYNAMICS STRUCTURAL ANALYSIS
RELIABILITY ANALYSIS
MAGNETICS ANALYSIS
OSE AND TEST REQUIREMENTS
POWER ANALYSIS
TELEMETRY REQUIREMENTS
ELECTRICAL AND MECHANICAL INTEGRATION
DISTURBANCE TORQUES ANALYSIS
INERTIAL ATTITUDE HOLD ANALYSIS
ACQUISITION ANALYSIS
GCS ORIENTATION ANALYSIS
GCS CRUISE AND ORBITAL ANALYSIS
THRUST VECTOR CONTROL ANALYSIS
OPTICAL SENSOR RELIABILITY TRADEOFF STUDY
OPTICAL SENSOR MOVING PARTS DESIGN STUDY
ELECTRONICALLY SCANNED IMAGE TUBE SELECTION STUDY
IR AND VISIBLE DETECTOR SELECTION STUDY
RADIATION MODELS DETERMINATION
GCS PARAMETRIC STUDIES
MAJOR FEEDBACK
SUBSYSTEM AND PRELIMINARY PART I CEI SPECIFICATION (PARTIAL)
OSE SUBSYSTEM SPECIFICATIONS
OPTICAL SENSORS PRELIMINARY PART I SPECIFICATION
OSE PRELIMINARY DESIGN
OPTICAL EQUIPMENT SURVEY
OPTICAL SENSORS DESIGN
PHASE 1B OSE DESIGN DATA REPORT
CIRCUITS DETERMINATION
CRITICAL CIRCUITS DESIGN
SPECIAL CIRCUITS BREADBOARD FABRICATION
SPECIAL CIRCUITS BREADBOARD TESTS

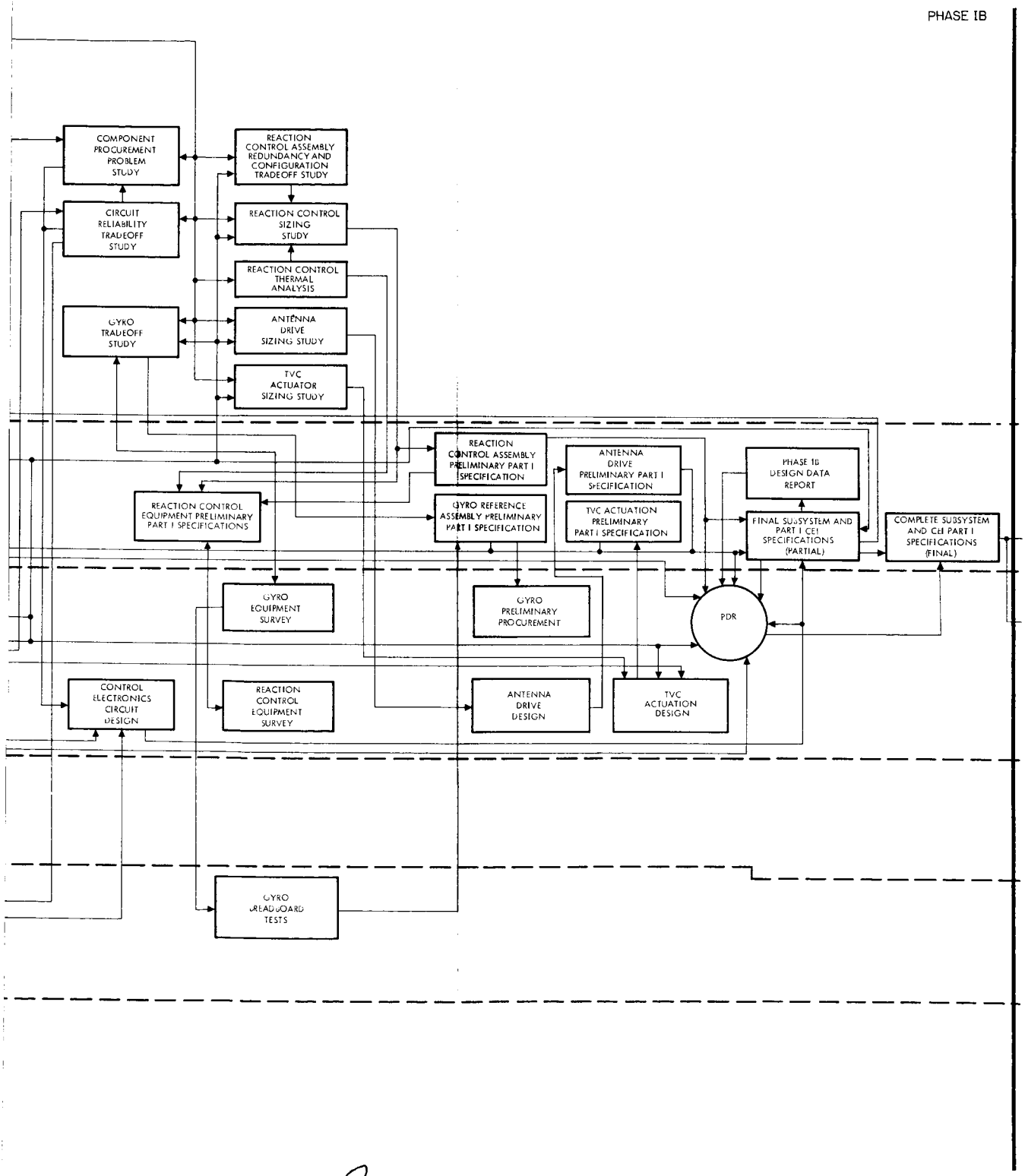
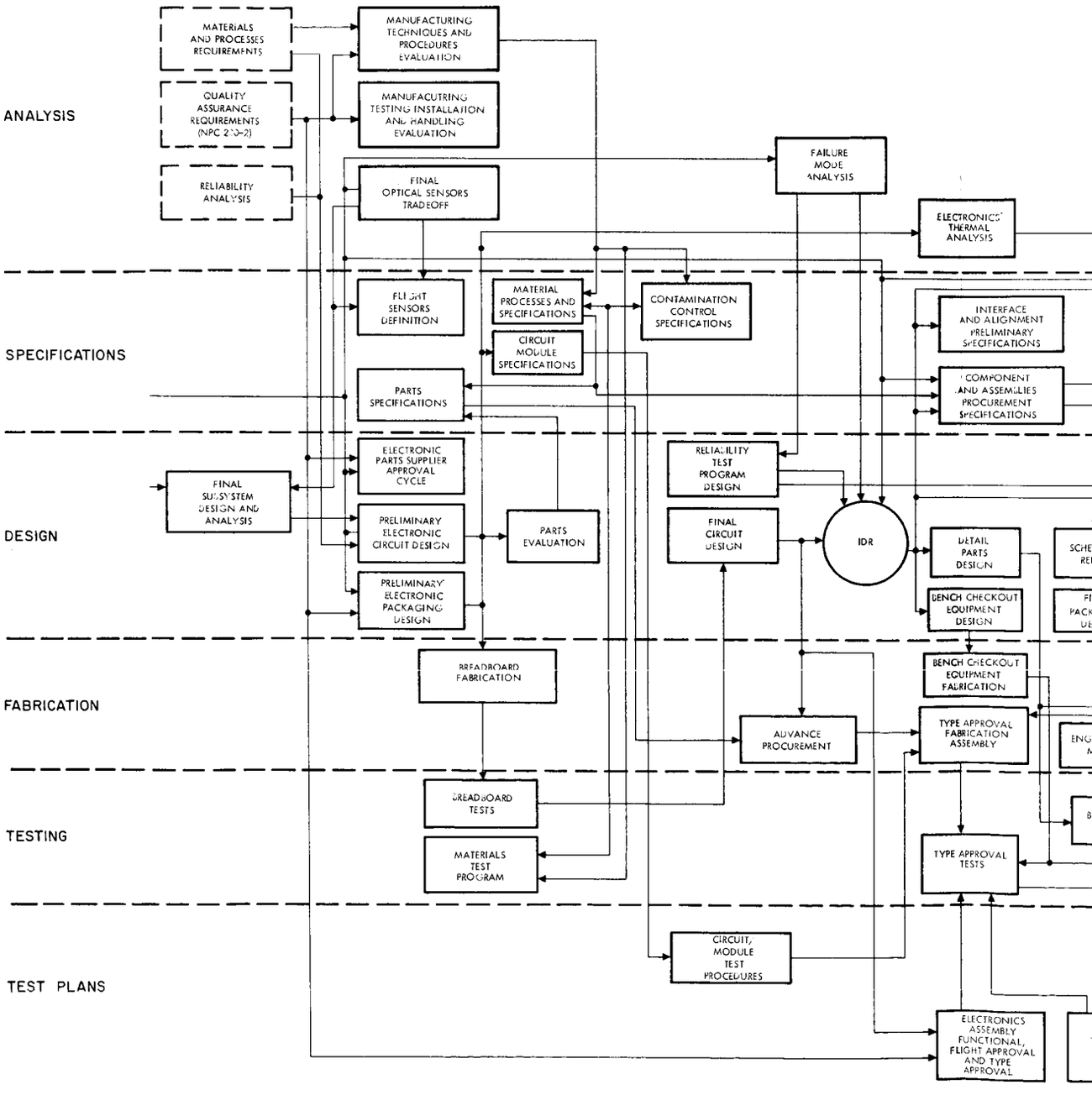


Figure 31. Design and Development of Guidance and Control Subsystem

PHASE II


ANALYSIS
SPECIFICATIONS
DESIGN
FABRICATION
TESTING
TEST PLANS
MATERIALS AND PROCESSES REQUIREMENTS
QUALITY ASSURANCE REQUIREMENTS (NPC 250-2)
RELIABILITY ANALYSIS
MANUFACTURING TECHNIQUES AND PROCEDURES EVALUATION
MANUFACTURING TESTING INSTALLATION AND HANDLING EVALUATION
FINAL OPTICAL SENSORS TRADEOFF
FAILURE MODE ANALYSIS
ELECTRONICS THERMAL ANALYSIS
FLIGHT SENSORS DEFINITION
MATERIAL PROCESSES AND SPECIFICATIONS
CONTAMINATION CONTROL SPECIFICATIONS
CIRCUIT MODULE SPECIFICATIONS
INTERFACE AND ALIGNMENT PRELIMINARY SPECIFICATIONS
PARTS SPECIFICATIONS
COMPONENT AND ASSEMBLIES PROCUREMENT SPECIFICATIONS
ELECTRONIC PARTS SUPPLIER APPROVAL CYCLE
RELIABILITY TEST PROGRAM DESIGN
FINAL SUBSYSTEM DESIGN AND ANALYSIS
PRELIMINARY ELECTRONIC CIRCUIT DESIGN
PARTS EVALUATION
FINAL CIRCUIT DESIGN
IDR
DETAIL PARTS DESIGN
BENCH CHECKOUT EQUIPMENT DESIGN
PRELIMINARY ELECTRONIC PACKAGING DESIGN
BREADBOARD FABRICATION
BENCH CHECKOUT EQUIPMENT FABRICATION
ADVANCE PROCUREMENT
TYPE APPROVAL FABRICATION ASSEMBLY
BREADBOARD TESTS
MATERIALS TEST PROGRAM
TYPE APPROVAL TESTS
CIRCUIT, MODULE TEST PROCEDURES
ELECTRONICS ASSEMBLY FUNCTIONAL, FLIGHT APPROVAL AND TYPE APPROVAL

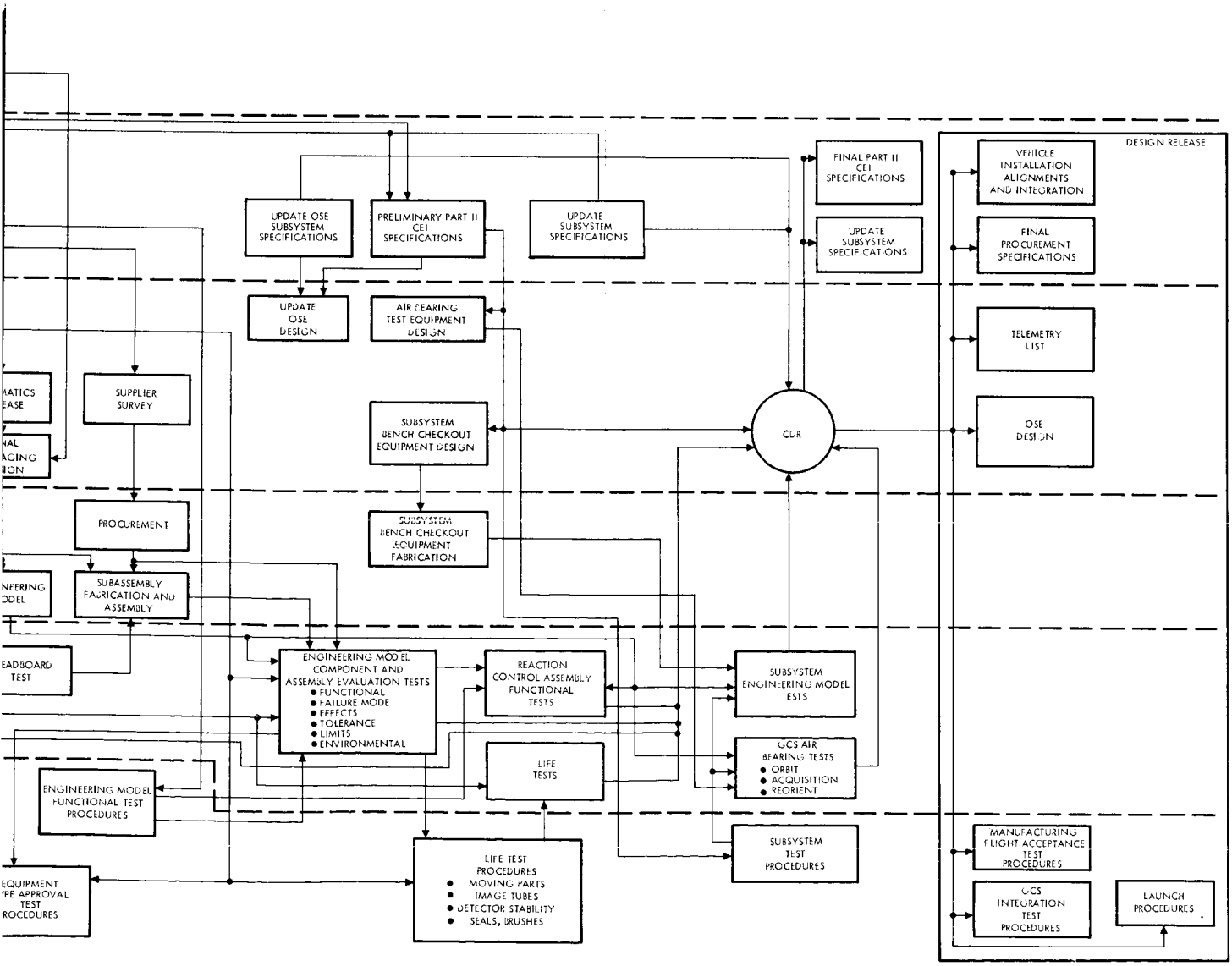


Figure 31. Design and Development of Guidance and Control Subsystem (Continued)

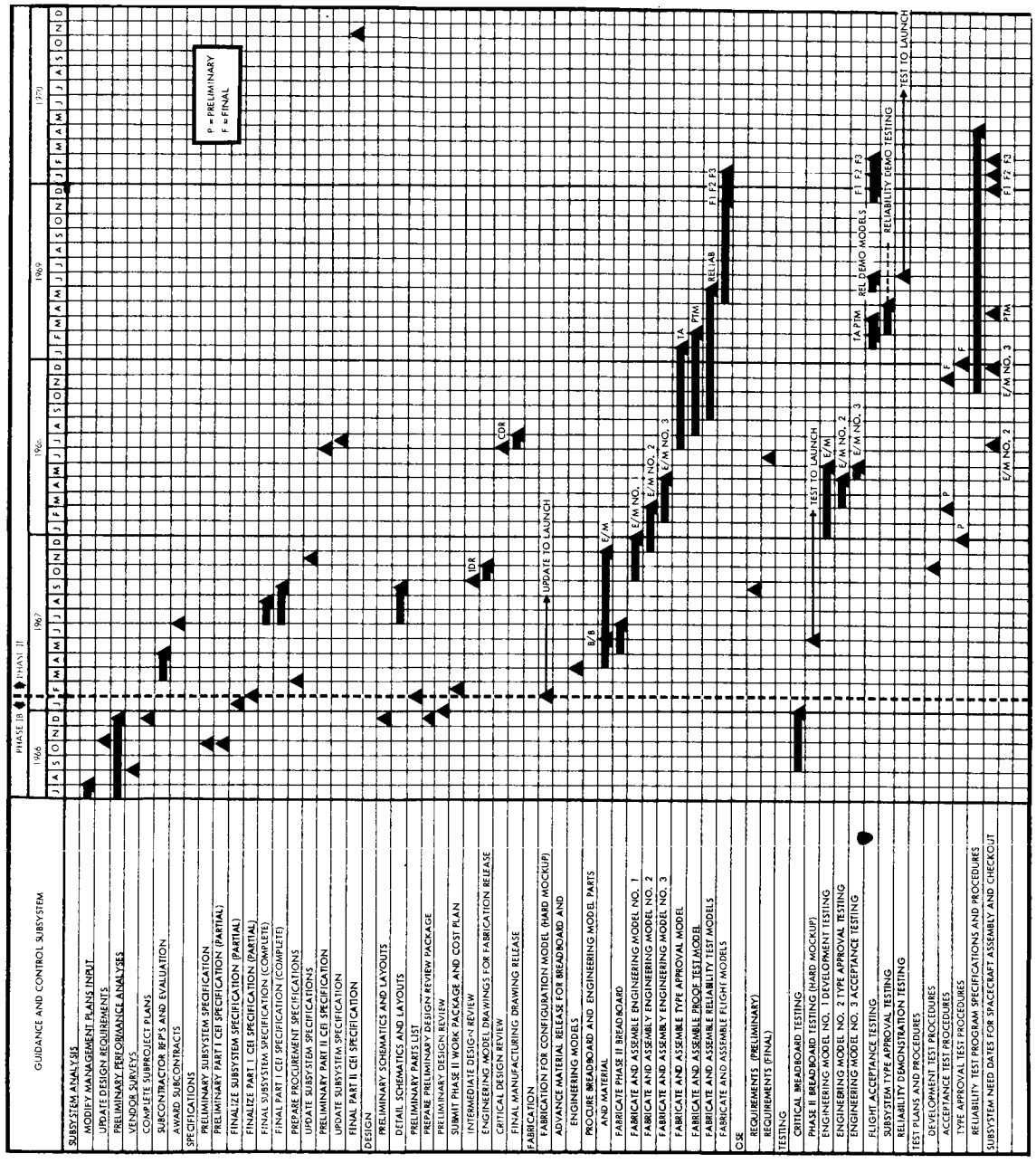


Figure 32. Schedule for Guidance and Control Subsystem

Phase II starts by completing subsystem specifications and guidance and control OSE subsystem specifications. Final Part I (partial) CEI specification for the subsystem and OSE assemblies will also be generated. Subsystem breadboard fabrication and testing effort will then start. At the completion of breadboard testing the intermediate design review will be held. All specifications will be completed as a result of the IDR, and all engineering model drawings will be released and the engineering model fabrication and testing effort started. Three complete engineering model subsystems will be fabricated and tested.

Results of evaluation of the engineering models are presented in the critical design review. Updated subsystem and Part I, CEI specifications and preliminary Part II, CEI specifications are also reviewed. Satisfactory findings from this design review govern the release of drawings for fabrication of the type approval (TA) and proof test model (PTM) assemblies. Final Part II, CEI specifications are prepared for the FACI in November 1970.

3.12.2 Design Studies

Analytical studies will define the detailed requirements for the G and C subsystem and the means for its implementation. The input data required will include:

- Sensor characteristics
- Accuracies (vehicle and antenna pointing)
- Moments of inertia
- Center of mass offset and uncertainties
- Control moment arms
- Acquisition time requirements
- Thrust level and thrust centerline uncertainty
- Accuracy requirements for midcourse corrections and orbital injection
- Disturbance inputs to spacecraft
- Tipoff rates
- Results of Phase IB preliminary design
- Structural deflection data
- Propellant tank geometry

a. Initial Hold

The G and C subsystem configuration and the required thrust level will be studied to assure that initial tipoff rates can be cancelled and the required planetary vehicle orientation maintained with respect to the separation attitude.

b. Acquisition

A detailed analysis will be completed of the acquisition scheme used to stabilize the spacecraft after separation from the boost vehicle and for subsequent acquisitions. The control authority required will be fully defined and the over-all acquisition process described in detail. Functional specifications for the control system will be prepared on the basis of this analysis. The output data will include definition of an acquisition sequence including time required for acquisition, control methods for acquisition, and functional specifications for the system.

c. Maneuvers

An analysis will be completed to determine the accuracy and response requirements for reorienting the spacecraft before midcourse corrections, capsule separation, and injection into orbit about Mars. Functional specifications for the control system to meet these requirements will be prepared.

d. Thrust Vector Control

A detailed analysis to determine the accuracy and response requirements of the thrust vector control mode will be made, considering structural bending and propellant slosh effects. The output will include thrust vector control accuracy requirements, response requirements, control torques, and functional specifications.

e. Attitude Control

A detailed analysis of the attitude control requirements for orbital and cruise phases of the mission will be completed. The output will include accuracy requirements, response requirements, control torques, and a functional specification for these modes.

f. Disturbance Torque Estimates

An analysis of the disturbances expected to act on the spacecraft in transit and in orbit about Mars will be made. Additional inputs for this study will include trajectory geometry and sequence of events, moments of inertia uncertainties, products of inertia and uncertainties, rotating equipment reaction torques, and solar pressure. The output of this analysis will be disturbance torque magnitude versus time and the classification of torques according to their cyclic or secular characteristics. Finally, an attitude control parametric study will be performed to optimize the G and C subsystem parameters, culminating in the formulation of a final G and C subsystem functional specification.

g. Alignment

Analyses will be performed to determine specific alignment requirements of orientation-sensitive assemblies. Error budgets will be established based on attitude control errors, assembly tolerances, and alignment capabilities. G and C subsystem assemblies requiring precise mechanical alignment include all optical sensors, inertial sensors, gas nozzles, and antenna drive shafts. Alignments prior to installation on the spacecraft include the alignment of gyros with respect to a reference plane on the gyro reference assembly, alignment of the accelerometer with respect to a reference plane on the accelerometer assembly, mounting and adjustment of the coarse sun sensors on the gas nozzle blocks, and alignment of the antenna drive axes.

h. Reliability

A subsystem reliability assessment based on failure rates of individual units will be completed, yielding a G and C subsystem reliability assessment and revision of unit reliability allocation.

i. General Assembly Level Design

The unit interface requirements and unit parameters will be determined, such as impedances, signal levels, gains, allowable errors, and time constants based on results of subsystem analysis and unit specifications. The result will be a subsystem detailed block diagram, updating of the functional system block diagram, and unit specifications.

j. Specific Assembly Level Design

The following studies will be among those conducted on the individual assemblies.

Control Electronics and Antenna Drive Electronics. A thorough analysis of the redundancy used will be conducted to insure compliance with the Voyager reliability requirements. A life test of the breadboard will also be conducted which will begin before construction of the engineering model and continue to the first launch, updated as required. Complete functional tests will be conducted periodically to insure proper operation. Any component or system failures of the breadboard will be thoroughly analyzed to determine the cause and the required modification, if any, to correct the problem. Critical low-level circuits will be analyzed to determine whether their relationship with other systems in the Voyager spacecraft will cause pickup problems. Any circuits which contain RFI problems will be analyzed to determine the filtering and shielding necessary to satisfy the spacecraft RFI and magnetic requirements. Switching control circuits will be used wherever possible to conserve power; however, for any circuits which dissipate large amounts of power, an analysis will be made of the thermal paths in the spacecraft, and in the individual subsystems, to insure proper operation of all components. High-reliability components will be used in the electronics. To obtain these parts and assure their use on the engineering model, orders will be placed within two weeks after the beginning of Phase II. Advanced material releases will be issued at the end of Phase IB to insure receipt in time for fabrication of the first production units.

Reaction Control Assembly. Besides the usual mechanical, thermodynamic, and fluid flow analyses associated with the reaction control assembly, additional activities will be pursued. To achieve the reliability goal a gas loading weight analysis will be performed which will consider first the amount of gas required due to disturbance torques, limit cycling, and maneuvers. Calculations based on probability of various failure modes will be conducted considering leakage and equipment failure modes such as a valve open, heater failure, out-of-tolerance regulation, and high-low thrust mode interchanging. Assembly mechanization will be optimized to reduce weight, power, and failure modes. To

demonstrate reliability, a life test program at the assembly and component level will be designed to give a reasonable confidence level, as well as comprehensive type approval and flight acceptance tests. These tests will be designed to uncover possible flaws through proper selection of stress levels, operation cycles and functions, and environments.

Pneumatics Cleanliness. Contamination control procedures will be reviewed to assure a high reliability baseline system. Vendor and in-house manufacturing, assembly, and test procedures and processes will be evaluated to assure a high level of cleanliness. An outline of the proposed contamination control plan is presented in Appendix D.

Inertial Sensor Selection. After establishing an error budget and preliminary specification, an equipment survey will be performed to determine the most suitable accelerometer and gyroscope(s). The lead time on these items will be approximately six months. Sensor magnetic shielding will be considered. To assure the required lifetime, the gyro will have a hydrodynamic spin bearing and the accelerometer will utilize a minimum number of moving parts (flexure or pivot supported proof mass), as compared to accelerometers with motors, ball bearings, pumps.

Gyro Reference Assembly. The following studies will be performed: establish an error budget for the gyro and electronic subassemblies; determine the transfer function of the assembly in the various modes of operation; analyze thermal, magnetic, alignment, mechanical, electrical, and reliability aspects.

Accelerometer Assembly. The following studies will be performed: establish an error budget for the accelerometer and electronic subassemblies; analyze the assembly performance throughout the range operation; analyze thermal, magnetic, alignment, mechanical, electrical, and reliability aspects.

Antenna Drive Mechanisms. It is a design requirement of the antenna drive mechanisms to hold the antennas such that no damage is incurred during spacecraft velocity corrections. These requirements impose loads on the drive gear and some of the drive components. The materials and lubricating techniques of the moving components exposed to

vacuum which were developed for OGO will be reviewed to consider the effects of heavier loading. Accelerated life tests will be required to verify materials and lubricating techniques. Several components of the drive mechanism are long lead-time items. The motor and gearhead will be a special design with features which will enhance its long life capabilities. The bellows, bearings, and wabble gear mesh will also be designed to special requirements for this application and thus require relatively long procurement cycles. The motor gearhead, bellows, large bore bearings, and output gear mesh will be purchased items. The remainder of the components will be fabricated at TRW. Fabrication of the individual components will require close tolerance machining of many surfaces. Individual components will be ultrasonically cleaned and bagged after machining. The drive will be assembled in an ultraclean facility to enhance the life and reliability of the drive mechanism.

Thrust Vector Control Actuators. Careful analysis and test of the engine gimbal system will be required to accurately define the TVC actuator loads. Information such as gimbal bearing friction, propellant line stiffness, and engine mass will be defined to compute the engine gimbal torques as a function of engine thrust, engine position, and spacecraft acceleration. An electromagnetic actuator of the same basic design as specified for the Voyager TVC actuators has been developed for the Apollo service module. The design modification required to meet the Voyager requirements are to shorten the length and reduce the output force required. The basic principle of operation of these actuators, whereby two counter-rotating clutches are employed to control the output shaft motion, has been incorporated on many actuators in the aircraft and space industry. A critical design feature of this actuator is the bellows to seal the output shaft in the presence of the travel requirements of the Voyager engine. However, the travel requirements of the Apollo service module actuator are nearly as large as the Voyager requirements. The employment of sufficient bellows convolutions will maintain the stresses in the bellows at a level compatible with the life requirements of Voyager. A load and life analysis of the bellows design will be performed to insure that it meets the design requirements. The complete actuator will be subcontracted. The actuator load analysis and design specification will

be completed at an early date to permit sufficient time for the procurement cycle.

Sensors. The following general analyses will be performed on all the G and C subsystem sensors. These represent the major areas of analysis associated with the sensor design and development.

- Select detector, optical system, scattered and reflected light elimination scheme. Develop search-track technique based on sensor requirements and detector data.
- Establish error budget, development signal processing, and logic based on sensor requirements and Voyager control sequences.
- Determine effects on sensor alignment of spacecraft mechanical and thermal deformation based on spacecraft layout, preliminary structural design, preliminary thermal design, and interface requirements.
- Establish final radiation models of the bodies to be sensed.
- Determine visible earth model for earth detector.
- Determine Canopus and star field background model for Canopus tracker.
- Determine visible models for the limb and terminator of Mars.
- Establish reliable target discrimination techniques for the Canopus tracker, sun sensor, and limb-terminator sensor during acquisition, cruise, and orbit modes.

Inputs for these special analyses will be the following:

- Available data on radiometric and photometric properties of Mars, Canopus, and the earth.
- Voyager trajectory and orbit parameters.
- Voyager sensor requirements.

k. Subsystem Interface

G and C subsystem requirements (power, structure, thermal, electrical integration, telemetry, command, computing and sequencing, spacecraft testing) will be coordinated based on:

- G and C subsystem specifications
- Unit specifications
- Unit drawings
- Specifications and drawings for other subsystems

The results will be alignment accuracy requirements and budgets, power consumption, thermal control requirements, wiring diagrams, telemetry lists, G and C subsystem requirements for spacecraft testing, and command lists.

1. Operational Support Equipment

Studies will be performed to determine specific OSE requirements. The most critical problem will be the design of sun, Canopus, and Mars simulators to stimulate the appropriate sensors during testing. Results of the analyses to establish models of the bodies to be sensed will be used as a baseline for simulator designs. Physical requirements and restrictions will also be determined. A survey will be conducted to determine what simulators are available.

Another problem is the design of a rate table to check out all three axes of the gyro reference assembly and the accelerometer. The rate table will also be used for alignment of the gyros with respect to the assembly reference plane. Size and weight of the assemblies to be tested, in addition to input rates and alignment requirements, must be determined before design can be initiated. The design will include a survey of existing equipment.

3.12.3 Development Test Program

The Voyager G and C subsystem development cycle provides for a comprehensive program of test to assure flight readiness of all units of the subsystem and of the subsystem itself. The test program is outlined in Table 27.

The following assemblies will be breadboarded: optical sensors (except Canopus sensor), control electronics, antenna drive electronics, gyro reference, accelerometer and selected portions of the reaction control assembly. The Canopus sensor and TVC actuators are conceived as design modifications of existing units and will probably be purchased

Table 27. Guidance and Control Tests

Name of Test	Item Tested	Purpose	Description
Circuit Design Evaluation	Circuit breadboards	Determine electrical per formance to verify that design meets performance requirements	Measure electrical characteristics when subjected to temperature and power supply variations over specified ranges.
Optical Sensors Design Feasibility	Electro-optical bread-boards	Determine electro-optical feasibility	Measure electro-optical character-istics when subjected to temperature and power supply variations over specified ranges.
Reaction Control Design Evaluation	Pneumatic breadboard	Verify that design meets performance requirements	Measure pneumatic parameters of lines and nozzles.
Component Evalua-tion	EM components	Verify that components meet design requirements	Measure performance parameters of purchased components such as gyros, accelerometers, electric motors, pneumatic valves and re-gulators, etc.
Design Verification	Breadboard electronics assemblies and EM of other assemblies	Determine design margins and demonstrate that designs will meet preli-minary design specificat-ions	Measure parameters when subjected to temperature and power supply variations over specified ranges. Step-stress until normal operation ceases. Demonstration of non-catastrophic performance in the presence of noise injection.
Interface Compati-bility	Breadboard/EM subsystem	Demonstrate that assembly interfaces are compatible and subsystem design meets requirements	Measure over-all subsystem per-formance parameters while operat-ing in various modes under speci-fied input conditions.
Subystem Develop-ment	Breadboard	Demonstrate that closed loop performance meets design requirements	Measure dynamic response charac-teristics of individual axis control channels.
3-axis Air Bearing Space Simulation	Breadboard/EM subsystem	Demonstrate closed loop functional performance about all control axes in all operating modes.	Operate scale model spacecraft in all control modes to show that sub-system will orient spacecraft in response to appropriate sensors.
Electrical Per-formance Life Test	Breadboard/EM subsystem	Determine electrical per-formance and verify con-formance to design speci-fications throughout life.	Continuously update to flight equiva-lent circuitry. At specified intervals measure parameters when subjected to specified input conditions. Operate continuously at nominal in-put conditions except as noted above.
Engineering Envi-ronmental Tests • Humidity • Vibration • Shock • Space Sim-ulation • EMC o Magnetic Compatibility • Ethylene Oxide Compatibility • Thermal Shock	EM No. 2	Verify units will survive specified vibration and shock limits, will operate in ethylene oxide environ-ment and will meet EMC and magnetic compati-bility design requirements.	Measure operating parameters during and/or after exposure to specified environmental conditions.
Type Approval Tests • Humidity • Vibration • Shock • Space Sim-ulation • EMC • Magnetic Compatibility • Ethylene Oxide Compatibility • Thermal Shock	TA units	Demonstrate flight units will function within specifi-cations at design qualifi-cation limits.	Impose qualification environmental limits on a unit monitoring operating parameters to insure proper opera-tion or survival, whichever is applicable.
Reliability Demonstrating	Life test units	Demonstrate that certain critical units meet the design life requirements per the requirements of the Preliminary Reliabi-lity Program Plan.	Operate flight units, simulating actual operating conditions as much as possible, to destruction or to a point where a unit no longer meets minimum performance requirements.
Flight Acceptance Tests • Vibration • Thermal Vacuum	Flight models EM No. 3 PTM	Demonstrate units will function within specifi-cations at design accept-ance limits	Impose acceptance environmental limits on a unit monitoring operating characteristics to insure proper operation or survival, whichever is applicable.

items. The antenna drive assemblies also represent design modifications of an existing unit. Therefore these units will not be breadboarded.

The breadboard tests are designed to verify functional performance and electro-optical feasibility and to determine design margins by step-stressing until units fail to meet design specifications. Tests will be performed on a subsystem consisting of breadboard and engineering model units to demonstrate closed-loop functional performance, interface compatibility, and that design requirements are met. In addition, a breadboard engineering model subsystem will be continuously updated to flight-equivalent circuitry, and at specified intervals performance parameters will be checked under special conditions. At all other times the subsystem will be continuously cycled in all operating modes at nominal input conditions.

Engineering model units will be subjected to a series of environmental tests including vibration, shock, thermal-vacuum, thermal shock, humidity, EMC, magnetic compatibility, and ethylene oxide compatibility to demonstrate that all design requirements are met.

A set of engineering model assemblies are to be supplied to the engineering model spacecraft, where further testing of the guidance and control subsystem will be performed. The guidance and control subsystem will be stimulated and the resulting reactions monitored.

In addition a set of engineering model assemblies are to be supplied to the propulsion interaction model spacecraft where the interfaces between the guidance and control subsystem and the propulsion subsystem will be thoroughly tested in a fully operating spacecraft during the engine firing.

A third set of engineering model assemblies will be mounted, as a subsystem, on a three-axes air bearing table and tested for an extended period of time. The tests will demonstrate closed loop functional performance about all control axis in all operating modes in response to appropriate sensor stimulus.

The subsystem type approval model will be exposed to type approval environmental limits to demonstrate design adequacy. Following completion of this test, the flight units are fabricated and FA tested prior to delivery to the flight spacecraft.

3.13 Propulsion Subsystem

3.13.1 Description

The Voyager propulsion subsystem will consist of the Apollo LEM descent stage with minor modifications needed to meet the Voyager mission requirements. The details of these modifications are described in Volume 2, Section IV. This plan describes the program for implementing these modifications through design, modification testing, and type approval demonstration tests.

The propulsion subsystem divides into two major areas; the engine and the feed system. The engine work will be accomplished at TRW while the feed system effort will take place at GAEC. The integration firings of the engine and feed system and the qualification of the propulsion subsystem is planned to be done at the NASA White Sands facility. This procedure would utilize experienced personnel developed during the LEMDE program, reducing program schedule time and cost.

3.13.2 Summary

a. Development Plan

The plan for Voyager is to make use of all work accomplished during the Apollo program that is applicable to the Voyager mission. The present LEMDE operates over a 10,500 to 1050 pound thrust range. Voyager requires operation at only two thrust levels, namely: 7750 and 1050 pounds. The LEMDE is already fully qualified for these levels; therefore, no additional thrust chamber development testing is planned to satisfy these Voyager requirements. In addition, the LEMDE mission duty cycle requires 417 seconds of firing at full thrust and 455 seconds at reduced thrust which compares favorably with the nominal Voyager requirement of 379 seconds at high thrust and 479 seconds at low thrust. Test data, presently available, at both high and low levels indicate no need for additional testing to verify these differences.

The Voyager mission, however, requires a much longer cruise time under deep space conditions than does the Apollo program; therefore, coincident with the initiation of Phase IB, ablative samples from LEMDE will be placed in a vacuum chamber and subjected to a 10-month exposure. Selected samples will be removed periodically to determine if any progressive rate of deterioration is taking place. In addition, valve components and pressurized tanks filled with N_2O_4 will be subjected to 18 months of this environment to evaluate the applicability of Apollo program solutions to the corrosion problems encountered in the Apollo program.

The modifications to the basic LEMDE propulsion subsystem to enable engine start and restart using positive expulsion start tanks will be developed as part of the ground test program. No zero g propellant settling tests are planned for the Voyager program because the tank configurations are almost identical and the Voyager settling thrust will provide four times the g level for settling experienced on the LEMDE vehicle. We can, therefore, use the same settling times as those used on LEMDE and be assured of a safety factor of at least four.

b. General Ground Rules

- Program planning will reflect compatibility with the Apollo program, i.e., noninterference of the Voyager propulsion module with the manned lunar landing program.
- The Voyager propulsion subsystem will make maximum utilization of LEMDE qualified subsystem equipment and qualification test results.
- Demonstration/qualification testing of "necessary changes" in LEMDE equipment will be accomplished to maximize likelihood of mission success.
- Major development testing will be accomplished without duplication of TRW or GAEC effort.
- Division of development and qualification testing will reflect TRW and Grumman's unique capabilities and economic utilization of resources.
- TRW will modify and qualify the LEM descent engine. TRW will also develop and qualify the engine control assembly (ECA). The engine testing will demonstrate

the suitability of the design with respect to the ablative nozzle extension, two level thrust performance, and engine operation under high rate thrust vectoring.

- Grumman will develop and qualify the descent engine oxidizer and fuel start tanks.
- Grumman will modify and qualify the ambient pressurization section and feed section.
- Grumman will conduct integrated propulsion module qualification testing (including the engine, ECA, propellant feed, and pressurization sections and tanks).

3.13.3 Design and Development

a. Phase IB

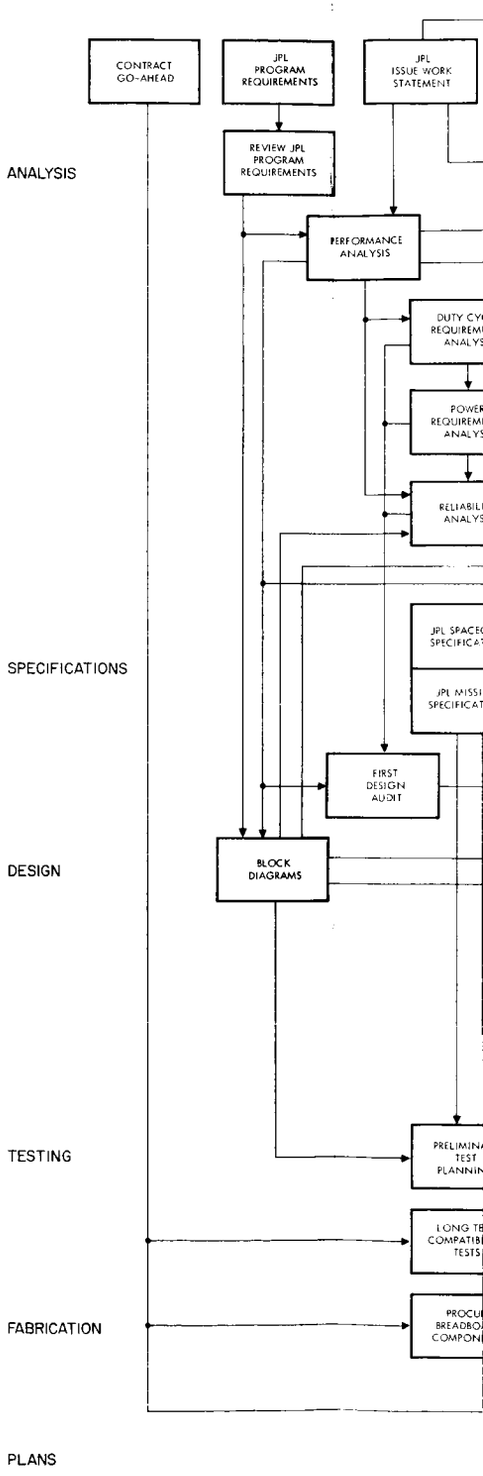
The program objectives to be completed during Phase IB will include propulsion subsystem analysis and design, preparation and issuance of specifications, and test planning for the Phase II effort.

The first tasks to be performed will be the updating of requirements, schematics, previously performed analyses, and block diagrams based on JPL inputs as shown on the flow chart of events per Figure 33. In mid August 1966 a design audit will be held to firm up these design requirements. Redefinition and updating of the design requirements will be used as a basis for complete performance analyses and for initiating the preliminary propulsion subsystem specification.

Based on the previous specifications, a revised subsystem schematic will be prepared and component, equipment, and interface requirements issued.

Preliminary equipment (i.e., Part I, CEI) and procurement specifications will also be issued in mid October 1966. Reliability analyses will be performed using the revised inputs.

Long lead-time items needed for the preliminary propulsion system configuration will be issued. The layout drawing for the preliminary propulsion configuration will also be initiated followed by assembly and detail drawings and an advanced list of materials for needed items.


CONTRACT GO-AHEAD
JPL PROGRAM REQUIREMENTS
JPL ISSUE WORK STATEMENT
REVIEW JPL PROGRAM REQUIREMENTS
ANALYSIS
PERFORMANCE ANALYSIS
DUTY CY
REQUIREM
ANALYS
POWER
REQUIREM
ANALYS
RELIABILI
ANALYS
JPL SPACEC
SPECIFICAT
JPL MISSI
SPECIFICAT
SPECIFICATIONS
FIRST DESIGN AUDIT
BLOCK DIAGRAMS
DESIGN
PRELIMINA
TEST
PLANNIN
TESTING
LONG TE
COMPATIBI
TESTS
PROCUI
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COMPONI
FABRICATION
PLANS

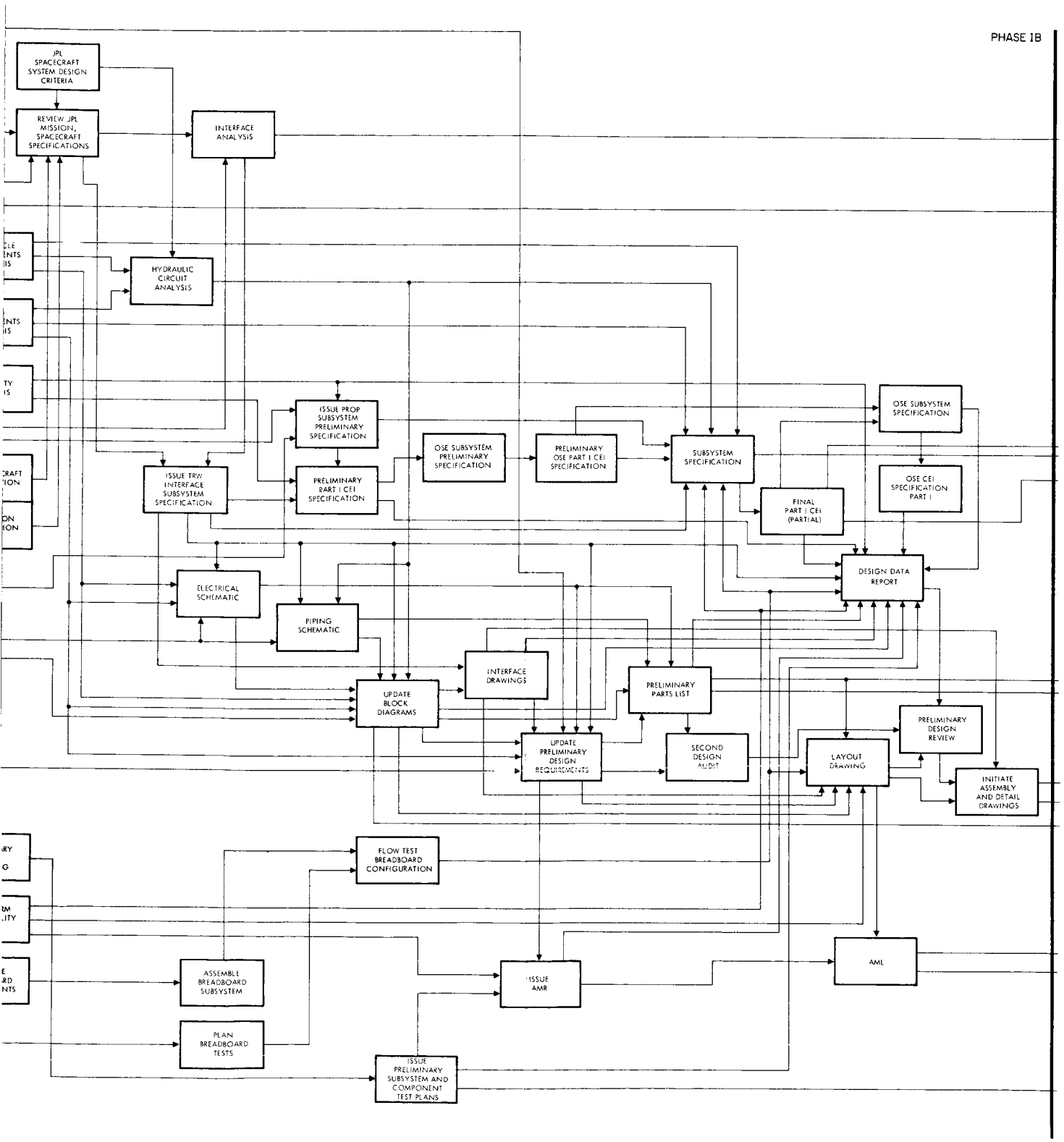
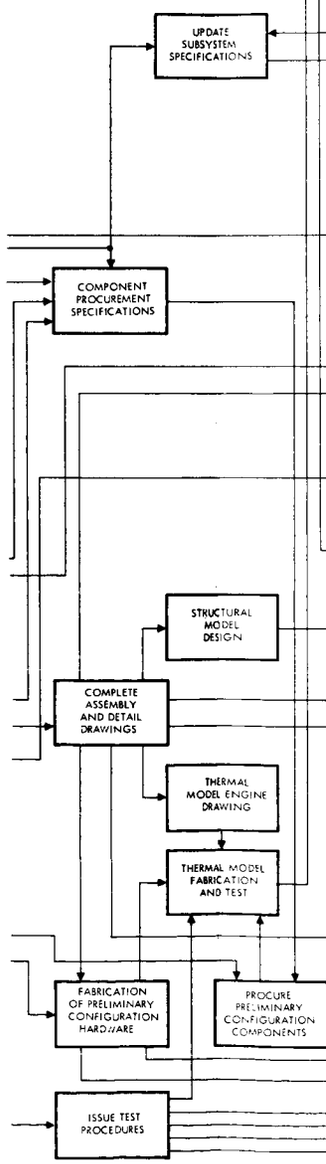


Figure 33. Design and Development of Propulsion Subsystem

PHASE II


UPDATE SUBSYSTEM SPECIFICATIONS
COMPONENT PROCUREMENT SPECIFICATIONS
STRUCTURAL MODEL DESIGN
COMPLETE ASSEMBLY AND DETAIL DRAWINGS
THERMAL MODEL ENGINE DRAWING
THERMAL MODEL FABRICATION AND TEST
FABRICATION OF PRELIMINARY CONFIGURATION HARDWARE
PROCURE PRELIMINARY CONFIGURATION COMPONENTS
ISSUE TEST PROCEDURES

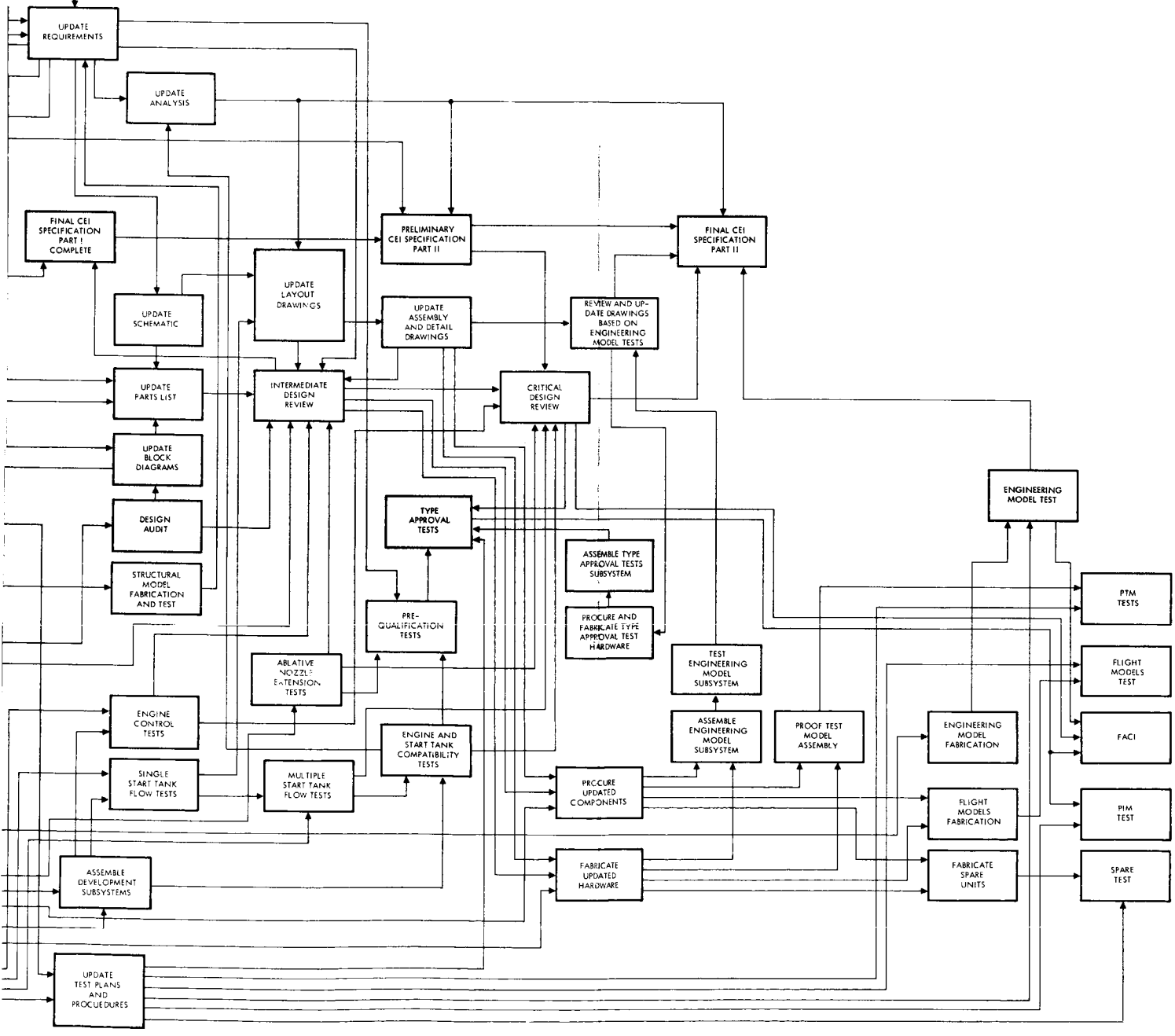


Figure 33. Design and Development of Propulsion Subsystem (Continued)

Twenty-six weeks after go-ahead a revision of the preliminary subsystem and Part I, CEI (partial) specifications will result in the first formal subsystem specification. Also the CEI specifications will be updated to the final partial Part I, CEI specification for transmission to JPL. A Phase IB design data report of all analysis and design data completed to date will also be compiled. The OSE specifications and OSE Part I, CEI specification will, in turn, be issued.

In addition to the above work, procurement of valves and control components will be initiated at Phase IB go-ahead in order to provide hardware for cold flow testing. The control components will be piped together to approximate the proposed design, and extensive bench testing will be performed to determine pressure drops and start tank-main tank transfer characteristics. The bench testing performed will provide a firm basis for the preliminary design configuration.

Following the PDR, any modifications required will be incorporated into the specifications and prospective component suppliers will be evaluated. During the design efforts, development test plans will be prepared and test equipment requirements will be established.

A major milestone chart depicting the activities to be completed during the Phase IB effort is shown as Figure 34.

b. Phase II

Propulsion subsystem activities during Phase II will be conducted as logical extensions of tasks initiated during Phase IB. Major efforts to be completed during Phase II are shown in Figure 33.

Performance analysis will be continued into the Phase II activity to update the parameters used in the evaluation of the preliminary design and hot firing tests initiated on the preliminary design hardware. Additional analytical effort will be expended in the areas of subsystem and component stress analysis, failure mode studies, and reliability prediction. All of the analytical effort and hot firing test results will be included in the engineering model design which will be reviewed during a July 1967 design audit and submitted as part of the intermediate design review (IDR) scheduled to occur during September 1967. Also reviewed

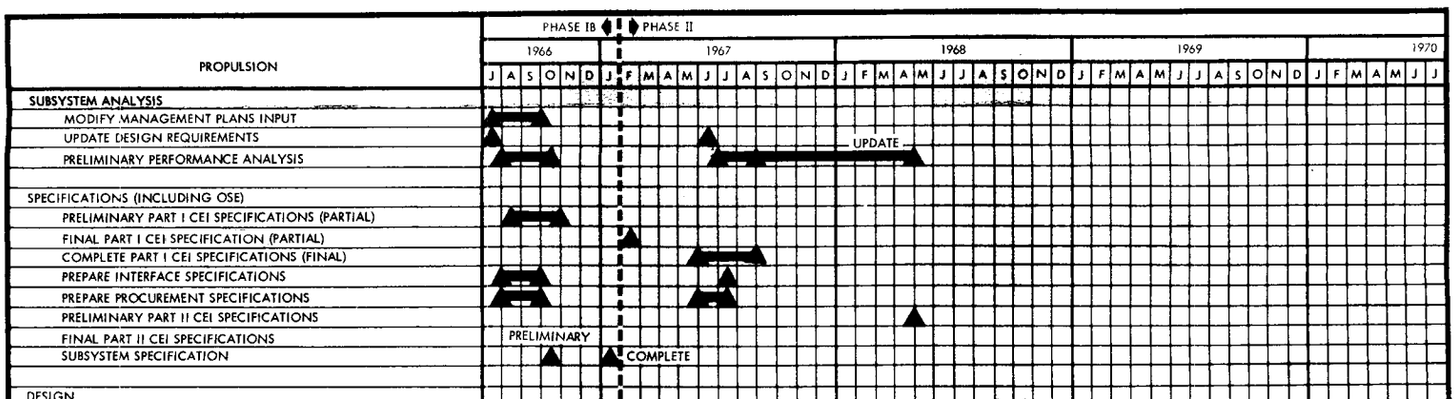

PROPULSION
PHASE IB
PHASE II
1966
1967
1968
1969
1970
J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J A S O N D J F M A M J J
SUBSYSTEM ANALYSIS
MODIFY MANAGEMENT PLANS INPUT
UPDATE DESIGN REQUIREMENTS
PRELIMINARY PERFORMANCE ANALYSIS
UPDATE
SPECIFICATIONS (INCLUDING OSE)
PRELIMINARY PART I CEI SPECIFICATIONS (PARTIAL)
FINAL PART I CEI SPECIFICATION (PARTIAL)
COMPLETE PART I CEI SPECIFICATIONS (FINAL)
PREPARE INTERFACE SPECIFICATIONS
PREPARE PROCUREMENT SPECIFICATIONS
PRELIMINARY PART II CEI SPECIFICATIONS
FINAL PART II CEI SPECIFICATIONS
SUBSYSTEM SPECIFICATION
PRELIMINARY
COMPLETE
DESIGN

Figure 34. Schedule for Propulsion Subsystem

at this time will be the updated subsystem specification, the completed Part I, CEI specifications, and OSE specifications. Continual updating of these specifications will occur, based on hot firing inputs of the engineering model configuration, prequalification, flight acceptance and type approval tests, and vehicle system tests as shown in the flow diagram Figure 33.

3.13.4 Test Program

The propulsion subsystem tests are divided into development, flight acceptance, and type approval tests. Table 28 presents a summary of all tests and gives hardware requirements for test and for flight spacecraft.

a. Development Tests

Each of the components used in the control system that are peculiar to the Voyager application will be subjected to a series of development tests to evaluate physical and functional characteristics. Components to be tested include the quad-redundant solenoid valve package, explosive valve package, and the normally-closed explosive valve. All of the components were selected on the basis of having been developed to a qualified flight status. Testing will be conducted to extend the qualified status to the complete range of conditions specified for the Voyager application.

Major emphasis in evaluation of the propellant feed system will be the development and qualification of the oxidizer and fuel starting system. Additional effort may be required to qualify the ambient pressurization blow system for midcourse firings if this is not accomplished on the LEMDE program.

Modifications to the basic LEMDE stage to include the restart capability with positive expulsion tanks (bellows) will be evaluated as part of a ground test program under 1 g operating conditions. The LEMDE flight test data will be used to extrapolate this testing to the zero g environment. Start system tests will be conducted in conjunction with engine firings and ablative nozzle extension development tests.

Tests of the propellant feed system will include flow at varying levels of tank pressure, temperature, and downstream pressure to

Table 28. Propulsion Subsystem Tests

Hardware	Test Type																									
	Hardware Requirements	Performance Tests	Pressure Drop and Flow Calibration	Functional	Leakage	Proof Pressure	Burst Pressure	Material Compatibility	Response Determination	Electrical Continuity	Power Utilization	Duty Cycle	Temperature Cycle	Thermal Vacuum	Functional Calibration	Life Cycle/ Endurance	Long Term Storage	Weight/ Dimensions	Magnetic Susceptibility	Radio Frequency Interference	Vibration	Shock	Sterilization Compatibility	Humidity	Acoustic Coupling
1. Component and Subsystem Modifications																										
Start System	6	x	x	x	x	x	x	x	x			x	x		x	x		x	x		x	x			
Flow Control System	5	x	x	x	x	x	x	x	x			x	x		x			x	x		x	x			
Propellant Feed System	2	x	x	x	x	x	x	x	x	x	x	x	x		x			x	x		x	x			
Ablative Nozzle Extension	10	x						x				x	x	x		x	x	x			x	x	x	x	
Water Cooled Combustion Chamber	3	x				x																			
1/2 inch Pyrotechnic Valves	100		x	x	x	x	x	x	x	x	x		x	x				x	x	x	x	x			x
1 inch Pyrotechnic Valves	100		x	x	x	x	x	x	x	x	x		x	x				x	x	x	x	x			x
1/2 inch Solenoid Valves	20		x	x	x	x	x	x	x	x	x	x	x	x	x	x		x	x	x	x	x			x
Pintle Actuator	20	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x		x	x	x	x	x			x
Propulsion Subsystem	4	x	x	x	x	x		x	x	x	x	x	x	x	x	x		x	x	x	x	x	x	x	x
2. Mission Hardware																										
Propulsion Subsystem	3	x	x	x	x	x			x	x	x			(1)	x			x	(1)	(1)	x				(1)
Spares	1	x	x	x	x	x			x	x	x			(1)	x			x	(1)	(1)	x				(1)
3. Type Approval Tests																										
Propulsion Subsystem	4	x	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Propellant Feed System	2	x	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
Flow Control System	3	x	x	x	x	x		x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x	x
4. Reliability Demonstration																										
Propulsion Subsystem	1	x	x	x	x	x			x	x	x	x	x	x	x	x	x	x	x	x	x	x		x	x
5. Vehicle System Development																										
PTM	1	x	x	x	x	x				x			x		x			x			x				x
PIM (use one TA test unit)	0				x					x					x										
EM(use one development model)	1	x	x	x	x	x			x	x	x				x			x	x		x	x			

Note: (1) accomplished during spacecraft FA test.

obtain operational flow characteristics. Sequencing of the flow control components will also be completed during this test activity to evaluate the influence of valve cycling on system characteristics and to obtain component reliability data.

No changes will be made to the control components of the LEMDE stage pressurization system. Development testing of this system to evaluate the effects of propellant contamination is not required because of the hermetic seal provided by the normally closed explosive valves. These valves retain the seal until just prior to deboost.

A single helium pressurization tank has been substituted for the two LEMDE tanks in order to obtain a proper balance of the Voyager spacecraft. The tank is of identical material as the LEMDE tanks and may be obtained from the same vendor. The tank has been qualified and, therefore, will be an off-the-shelf item.

The propellant feed tankage and relief valve assembly will remain unchanged except for the addition of the starting bellows assembly to the bottom cover plates, described in Volume 2, Section IV. Since the bellows assembly is attached only to the cover plates, it can be tested separately without affecting the main tanks. Tests of the starting bellows assembly will include cold flow tests at GAEC and hot firing engine integration tests at TRW.

The propellant feed manifolds will also remain identical to those used in LEM except for the removal of 27 inches of straight section in the vertical riser within the engine compartment. This change is required to accommodate the lowering of the engine described in Volume 1. Cold flow tests will be performed on the main tank and manifold systems to determine the flow characteristics of these assemblies.

Tests will be required to evaluate the ablative nozzle extension. This is planned as a minimal program, however, since the technology will have been developed during Apollo service module and ascent engine testing. This information will be used to design the required nozzle extension, thereby reducing cost and minimizing development time.

Engine feed system integration firings will be performed on the entire propulsion system. These tests, which demonstrate the compatibility of the engine and the feed system, could be done at the NASA White Sands facility. During this phase of the program, Voyager duty cycles will be simulated on the complete propulsion system under all the critical environments to insure passing the formal qualification on the first attempt.

b. Flight Acceptance (FA) Tests

The purpose of conducting FA tests of the propulsion subsystem is to confirm that workmanship and quality standards have been satisfied. Testing will be completed at the component and subsystem levels. Each rocket engine will be completely assembled in accordance with the drawings, then visually and dimensionally inspected before commencing the rocket engine tests. The rocket engine and components, as assembled for the inspection, will be subjected to the weight, magnetic, electromagnetic interference, static leakage, calibration, and additional tests specified in the acceptance test plan as shown in Table 29.

The injector will be mounted in a special combustion chamber designed to test the injector for chamber streaking. This special chamber will be run for sufficient time to demonstrate that no unacceptable streaking occurs during operation. Streaking requirements will be those determined during the LEMDE development program.

The rocket engine will then be operated for sufficient duration at minimum and maximum thrust levels to demonstrate compliance with specified performance ratings. The attitude of the engine and the sequence of its operation will be as specified in the FA test plan. The total combustion chamber operating time will not exceed 45 seconds for the acceptance tests. A stability test based on the information obtained from the development and qualification test programs will be conducted. The thrust vector location will be determined in accordance with specified engineering directives during the acceptance test program.

Acceptance of the rocket engine and its components will be predicated on maintenance of all parameters within the limits specified throughout all tests.

Table 29. Flight Acceptance Test Matrix

Hardware			Test Type				
			Vibration	Magnetic	Thermal Vacuum	Power Variations	Functional
Pressurization and Propellant Feed Assembly			x	x		x	x
ENGINE	ENGINE COMPONENTS	Quad Solenoid Valves	x		x	x	x
		Quad Check Valves	x				x
		1/2 Inch Explosive Valves*					
		Pressure Transducers	x		x	x	x
		Temperature Transducers					x
		Cavitating Venturi					x
		1/2 Inch Trim Orifice					x
		1 Inch Trim Orifice					x
		1 Inch Explosive Valves*					
		Filters	x		x		x
		Pintle Actuator	x		x	x	x
		Thrust Chamber Assembly	x	x			x
Propulsion Subsystem			x**	x**	x**	x	x

*Lot qualification acceptance testing

**Accomplished during spacecraft FA tests

Upon completion of the acceptance tests, the rocket engine will be subjected to a visual inspection. If any part is found to be defective an approved part will be supplied to replace it, and a suitable penalty test will be conducted. .

Whenever there is evidence of insufficient thrust or other malfunctioning, or evidence that the rocket engine is not meeting the specified requirements, the difficulty will be investigated and its cause

corrected before the test is continued. That portion of the test in which the difficulty was encountered will be repeated.

The combustion chamber will be rejected whenever the total running time accumulated during the tests specified exceeds the time allowed for acceptance test. Parts and other accessories from rejected rocket engines may be reused if such items meet the requirements for new parts.

In a similar manner the feed system will be assembled in accordance with the drawings, dimensionally inspected, and tested.

FA tests on the propulsion feed system include leak tests, mass properties determination, propellant loading and tanking tests (inert fluids), cold flow calibration tests, vibration, and space simulation testing. The vibration and space simulation tests will be performed as part of the spacecraft system test cycle.

Upon completion of the FA test program, the subsystem will be delivered.

c. Type Approval TA Tests

Type approval tests are conducted to demonstrate the ability of the hardware to satisfy the mission requirements and to demonstrate operational margins. Tests in this series are shown in Table 30. Functional and environmental tests will be conducted first at the component level and then at the subsystem level. Each item to be used for the TA test program will have first completed FA test. The usual procedure is to burst the hardware after TA test to demonstrate that adequate margins still exist and to disassemble the hardware to check for unusual wear or deterioration. This will be done on all units tested except one which will be delivered for the propulsion interaction test model. This model, when completing its test phase, will then undergo the hydraulic component burst and the physical inspection.

Table 30. Type Approval Test Matrix

Hardware			Test Type: Flight Acceptance Test	Temperature	Vibration	Acoustic Coupling	Shock	Humidity	Thermal Shock	Sterilization Compatibility	Radio Frequency Interference	Thermal Vacuum
		Pressurization and Propellant Feed Assembly	x	x	x	x	x	x	x	x		
ENGINE	ENGINE COMPONENTS	Quad Solenoid Valves	x	x	x	x	x	x	x	x		
		Quad Check Valves	x	x	x	x	x	x	x	x		
		1/2 Inch Explosive Valves	x	x	x	x	x	x	x	x		
		Pressure Transducers	x	x	x	x	x	x	x	x		
		Temperature Transducers	x	x	x	x	x	x	x	x		
		Cavitating Venturi	x							x		
		1/2 Inch Trim Orifice	x							x		
		1 Inch Trim Orifice	x							x		
		1 Inch Explosive Valves	x	x	x	x	x	x	x	x		
		Filters	x	x	x	x	x	x	x	x		
		Pintle Actuator	x	x	x	x	x	x	x	x		
		Thrust Chamber Assembly	x	x	x	x	x	x	x	x		
Propulsion Subsystem (1)			x		x	x	x			x	x	x

(1) Hot firing conducted after exposure to environments.

3.14 Temperature Control Subsystem

3.14.1 Description

The temperature control subsystem is responsible for maintaining satisfactory temperatures and/or thermal environments for the entire flight spacecraft for all phases of the mission.*

*Temperature control for government furnished external science has not been included in the development plan.

The flight spacecraft may be conveniently sectioned for the present purpose into the following groups:

- Main compartment - which includes the propulsion subsystem and the majority of the electronic equipment
- Solar array
- External equipment, such as the antennas, gimbals, attitude control lines and valves, and other components located external to the main compartment

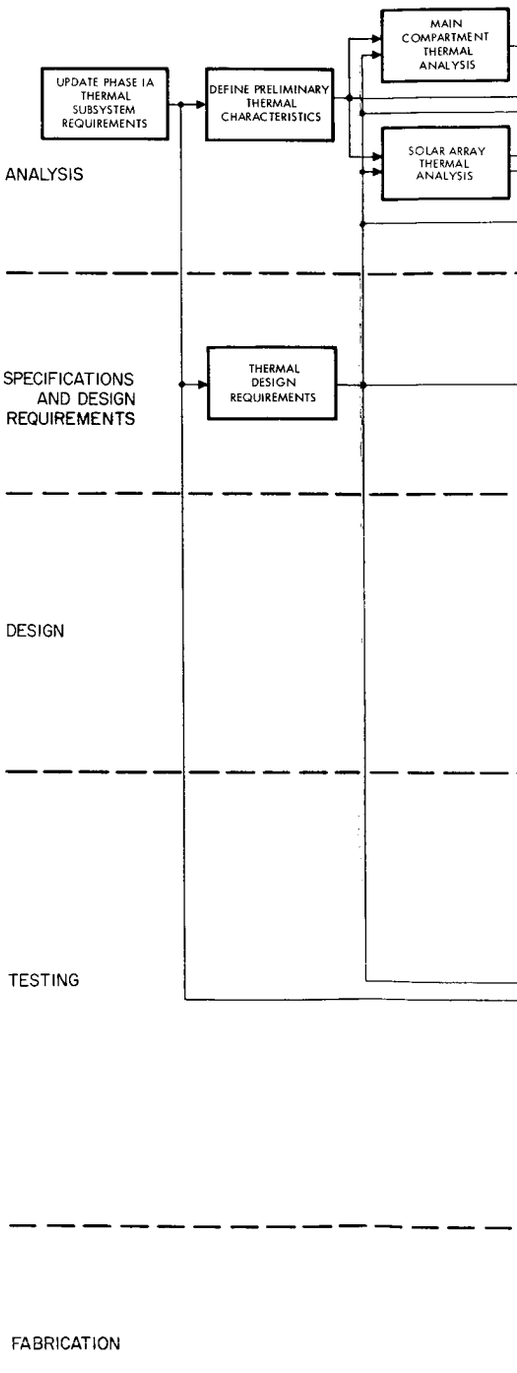
The main compartment is well insulated except for louver-covered radiators, which in a nominal spacecraft attitude do not view the sun. The solar array has its back surface insulated, which allows eclipse of longer duration, and reduces the radiant heat exchange between the array and the louver-covered radiators.

External equipment will utilize selected surface coatings, controlled exterior geometry, insulation, and where necessary, thermostatically controlled heaters to provide suitable temperature control.

3.14.2 Summary

The temperature control subsystem development flow chart is presented in Figure 35. The corresponding subsystem development schedule is shown in Figure 36. The majority of the development work is an iterative type of activity in which both thermal analysis and design are continually updated to reflect design changes and to incorporate development test results. The culmination of the temperature control development program is a series of thermal model tests, scheduled prior to the critical design review (CDR) and final drawing release. Type approval (TA) of the temperature control subsystem will also be accomplished through a series of tests, the principal one of the series being the thermal vacuum test of the proof test model (PTM), which incorporates flight-type functional equipment. That equipment which cannot be thermally qualified during the PTM TA test, because of facilities limitations, will be qualified separately. Equipment under this latter category are antennas, and other external packages.

The Phase IB design and analysis effort is characterized by the development of preliminary subsystem specifications, preliminary


UPDATE PHASE IA THERMAL SUBSYSTEM REQUIREMENTS
DEFINE PRELIMINARY THERMAL CHARACTERISTICS
MAIN COMPARTMENT THERMAL ANALYSIS
SOLAR ARRAY THERMAL ANALYSIS
ANALYSIS
SPECIFICATIONS AND DESIGN REQUIREMENTS
THERMAL DESIGN REQUIREMENTS
DESIGN
TESTING
FABRICATION

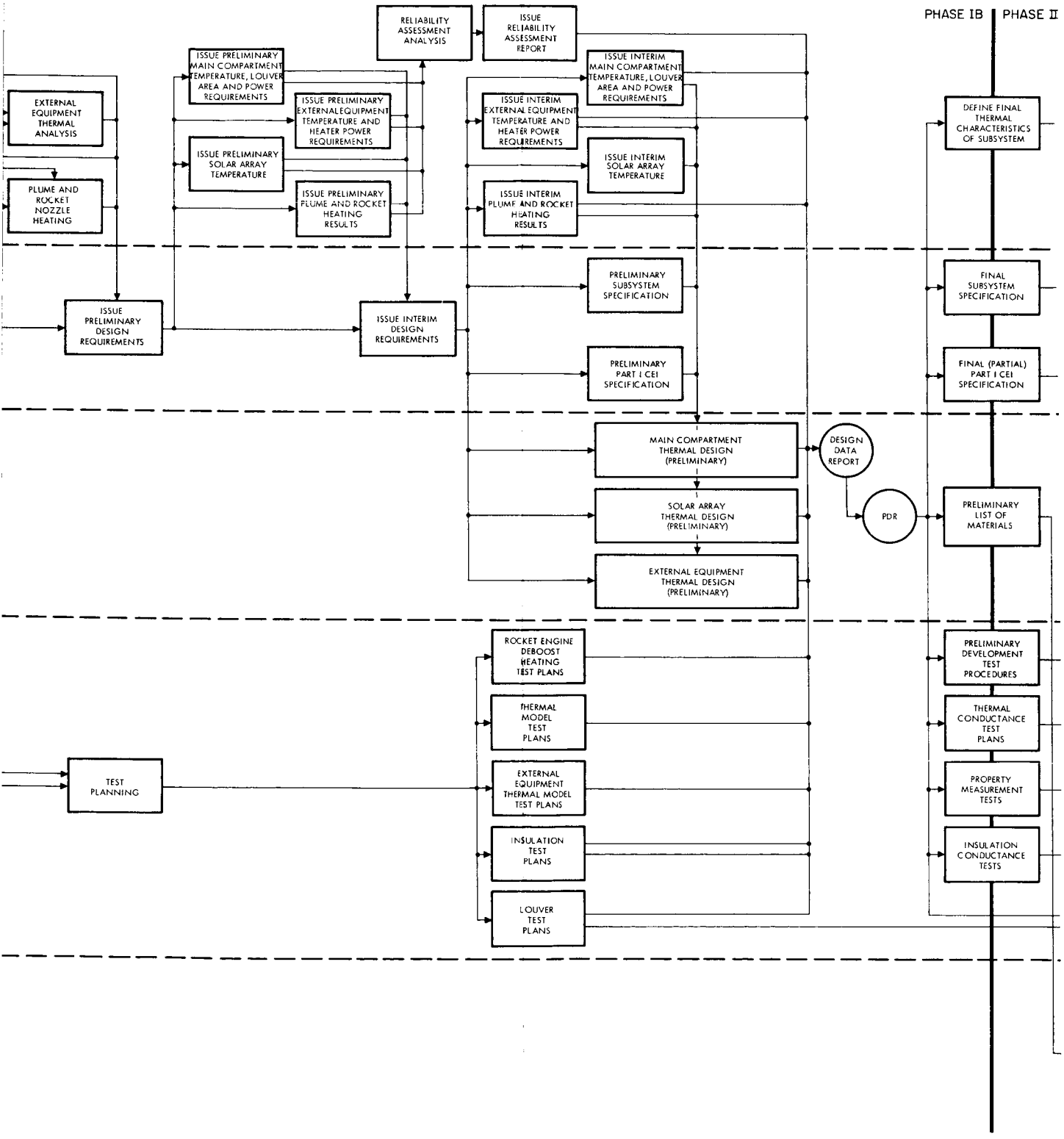
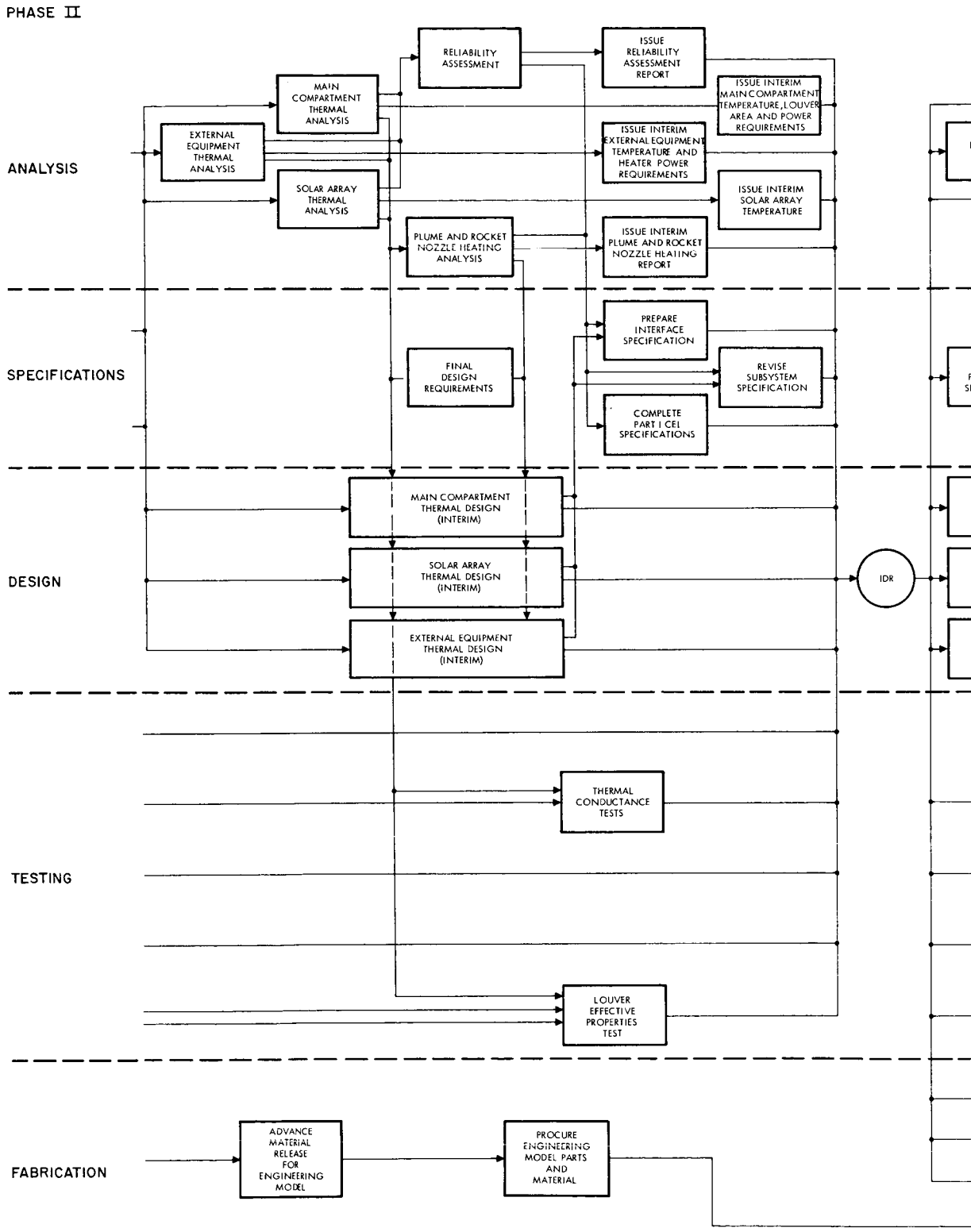
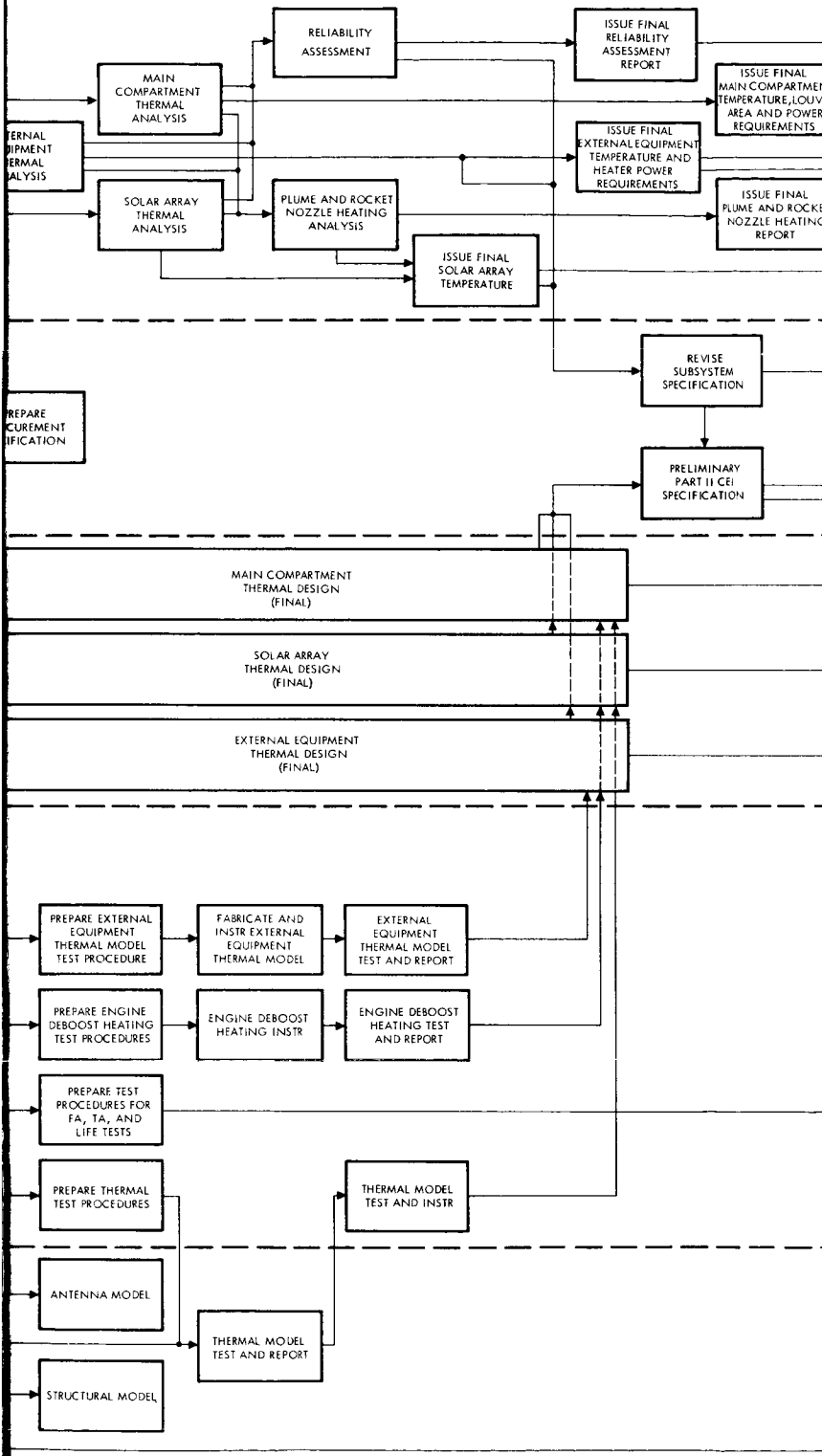


Figure 35. Design and Development of Temperature Control Subsystem


PHASE II
ANALYSIS
SPECIFICATIONS
DESIGN
TESTING
FABRICATION
RELIABILITY ASSESSMENT
ISSUE RELIABILITY ASSESSMENT REPORT
MAIN COMPARTMENT THERMAL ANALYSIS
ISSUE INTERIM MAIN COMPARTMENT TEMPERATURE, LOUVER AREA AND POWER REQUIREMENTS
EXTERNAL EQUIPMENT THERMAL ANALYSIS
ISSUE INTERIM EXTERNAL EQUIPMENT TEMPERATURE AND HEATER POWER REQUIREMENTS
SOLAR ARRAY THERMAL ANALYSIS
ISSUE INTERIM SOLAR ARRAY TEMPERATURE
PLUME AND ROCKET NOZZLE HEATING ANALYSIS
ISSUE INTERIM PLUME AND ROCKET NOZZLE HEATING REPORT
PREPARE INTERFACE SPECIFICATION
FINAL DESIGN REQUIREMENTS
REVISE SUBSYSTEM SPECIFICATION
COMPLETE PART I CEI SPECIFICATIONS
MAIN COMPARTMENT THERMAL DESIGN (INTERIM)
SOLAR ARRAY THERMAL DESIGN (INTERIM)
IDR
EXTERNAL EQUIPMENT THERMAL DESIGN (INTERIM)
THERMAL CONDUCTANCE TESTS
LOUVER EFFECTIVE PROPERTIES TEST
ADVANCE MATERIAL RELEASE FOR ENGINEERING MODEL
PROCURE ENGINEERING MODEL PARTS AND MATERIAL


RELIABILITY ASSESSMENT
ISSUE FINAL RELIABILITY ASSESSMENT REPORT
MAIN COMPARTMENT THERMAL ANALYSIS
ISSUE FINAL MAIN COMPARTMEN TEMPERATURE, LOUV AREA AND POWER REQUIREMENTS
TERNAL DIPMENT ERMAL ALYSIS
ISSUE FINAL EXTERNAL EQUIPMENT TEMPERATURE AND HEATER POWER REQUIREMENTS
SOLAR ARRAY THERMAL ANALYSIS
PLUME AND ROCKET NOZZLE HEATING ANALYSIS
ISSUE FINAL PLUME AND ROCKE NOZZLE HEATING REPORT
ISSUE FINAL SOLAR ARRAY TEMPERATURE
REVISE SUBSYSTEM SPECIFICATION
PREPARE CUREMENT IFICATION
PRELIMINARY PART II CEI SPECIFICATION
MAIN COMPARTMENT THERMAL DESIGN (FINAL)
SOLAR ARRAY THERMAL DESIGN (FINAL)
EXTERNAL EQUIPMENT THERMAL DESIGN (FINAL)
PREPARE EXTERNAL EQUIPMENT THERMAL MODEL TEST PROCEDURE
FABRICATE AND INSTR EXTERNAL EQUIPMENT THERMAL MODEL
EXTERNAL EQUIPMENT THERMAL MODEL TEST AND REPORT
PREPARE ENGINE DEBOOST HEATING TEST PROCEDURES
ENGINE DEBOOST HEATING INSTR
ENGINE DEBOOST HEATING TEST AND REPORT
PREPARE TEST PROCEDURES FOR FA, TA, AND LIFE TESTS
PREPARE THERMAL TEST PROCEDURES
THERMAL MODEL TEST AND INSTR
ANTENNA MODEL
THERMAL MODEL TEST AND REPORT
STRUCTURAL MODEL

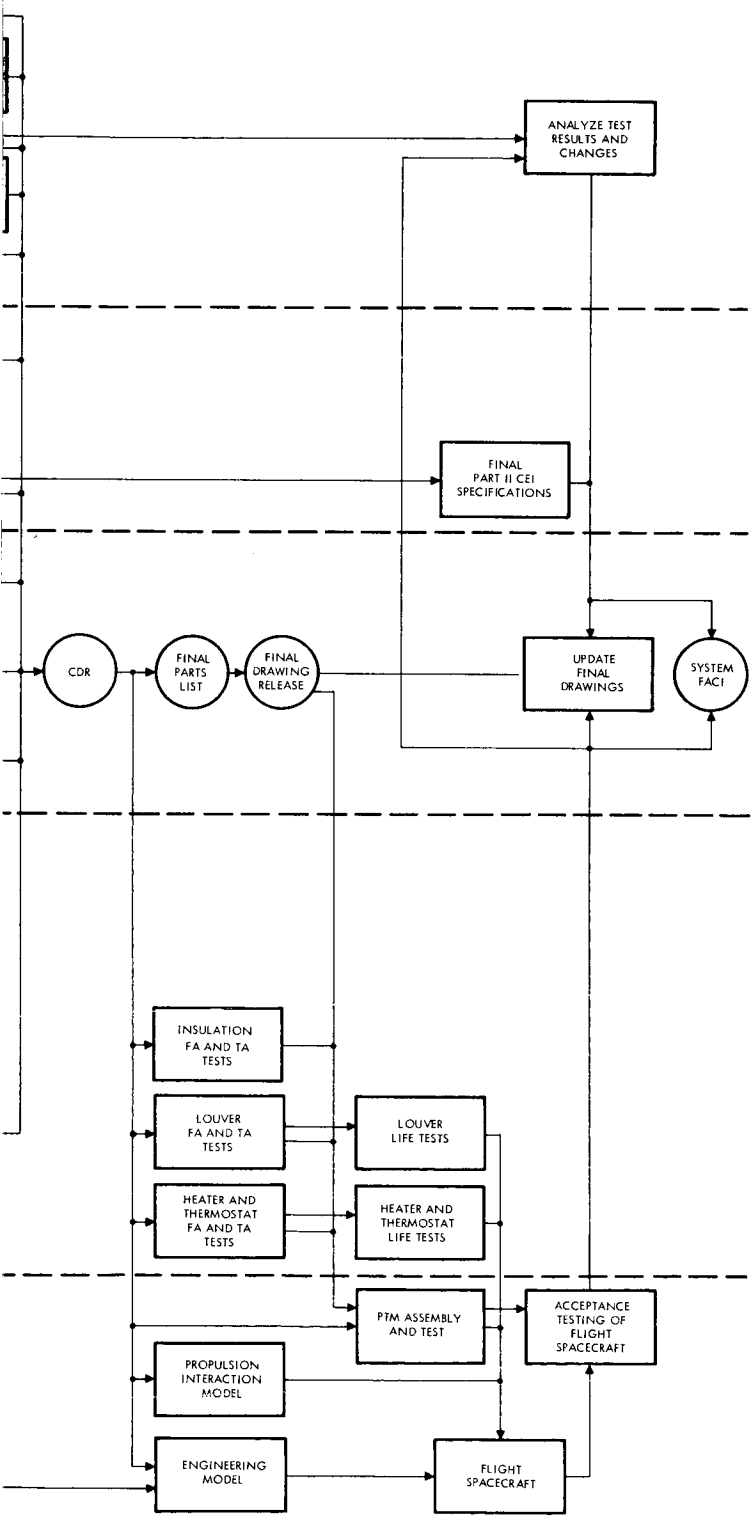


Figure 35. Design and Development of Temperature Control Subsystem (Continued)

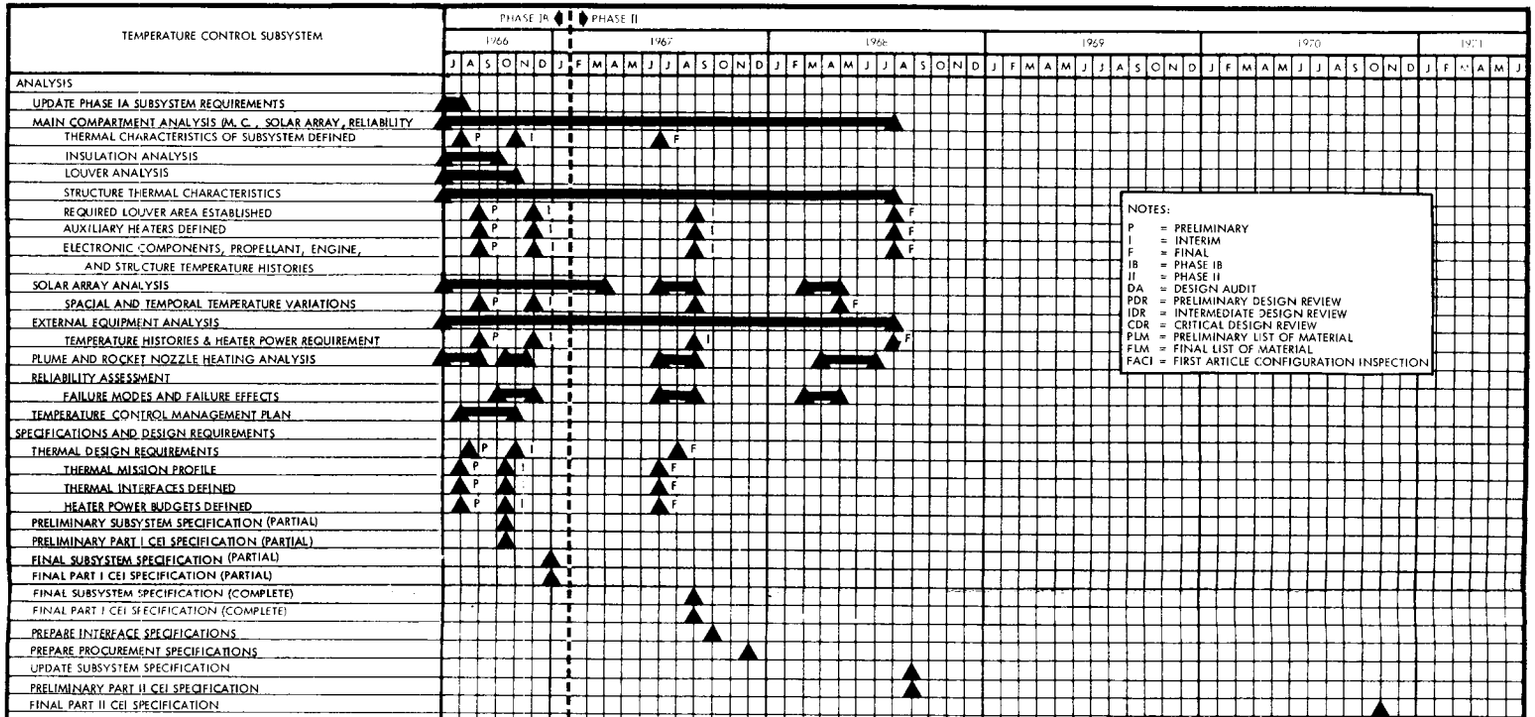

TEMPERATURE CONTROL SUBSYSTEM
PHASE IB
PHASE II
1966
1967
1968
1969
1970
1971
ANALYSIS
UPDATE PHASE IA SUBSYSTEM REQUIREMENTS
MAIN COMPARTMENT ANALYSIS (M. C. , SOLAR ARRAY, RELIABILITY
THERMAL CHARACTERISTICS OF SUBSYSTEM DEFINED
INSULATION ANALYSIS
LOUVER ANALYSIS
STRUCTURE THERMAL CHARACTERISTICS
REQUIRED LOUVER AREA ESTABLISHED
AUXILIARY HEATERS DEFINED
ELECTRONIC COMPONENTS, PROPELLANT, ENGINE,
AND STRUCTURE TEMPERATURE HISTORIES
SOLAR ARRAY ANALYSIS
SPACIAL AND TEMPORAL TEMPERATURE VARIATIONS
EXTERNAL EQUIPMENT ANALYSIS
TEMPERATURE HISTORIES & HEATER POWER REQUIREMENT
PLUME AND ROCKET NOZZLE HEATING ANALYSIS
RELIABILITY ASSESSMENT
FAILURE MODES AND FAILURE EFFECTS
TEMPERATURE CONTROL MANAGEMENT PLAN
SPECIFICATIONS AND DESIGN REQUIREMENTS
THERMAL DESIGN REQUIREMENTS
THERMAL MISSION PROFILE
THERMAL INTERFACES DEFINED
HEATER POWER BUDGETS DEFINED
PRELIMINARY SUBSYSTEM SPECIFICATION (PARTIAL)
PRELIMINARY PART I CEI SPECIFICATION (PARTIAL)
FINAL SUBSYSTEM SPECIFICATION (PARTIAL)
FINAL PART I CEI SPECIFICATION (PARTIAL)
FINAL SUBSYSTEM SPECIFICATION (COMPLETE)
FINAL PART I CEI SPECIFICATION (COMPLETE)
PREPARE INTERFACE SPECIFICATIONS
PREPARE PROCUREMENT SPECIFICATIONS
UPDATE SUBSYSTEM SPECIFICATION
PRELIMINARY PART II CEI SPECIFICATION
FINAL PART II CEI SPECIFICATION
NOTES:
P = PRELIMINARY
I = INTERIM
F = FINAL
IB = PHASE IB
II = PHASE II
DA = DESIGN AUDIT
PDR = PRELIMINARY DESIGN REVIEW
IDR = INTERMEDIATE DESIGN REVIEW
CDR = CRITICAL DESIGN REVIEW
PLM = PRELIMINARY LIST OF MATERIAL
FLM = FINAL LIST OF MATERIAL
FACI = FIRST ARTICLE CONFIGURATION INSPECTION

Figure 36. Schedule for Temperature Control Subsystem

Part I CEI specifications, and the temperature control design data report. The temperature control subsystem development test plans will also be generated. Prior to evolving individual test plans there will be a subsystem level test planning phase during which time test guide lines for the subsystem will be established. Final subsystem specifications and Part 1, CEI specifications (partial) will also be generated.

The temperature control design requirements document will provide subsystem design requirements and constraints, define thermal environments and interfaces, and allocate heater power budgets. Control over the thermal design of the flight spacecraft will be maintained through a series of design audits in addition to formal design reviews. The audits and design reviews will insure that design requirements are fulfilled. If audits and design reviews reveal major problem areas, the temperature control subprogram manager will establish a task force to resolve the problem. The goal of the task force will be to resolve the problem wholly within the domain of the subsystem without degrading subsystem performance. For example, a redistribution of subsystem power budgets or a renegotiation of thermal interfaces could be solutions for some types of thermal problems. If the problem cannot be resolved entirely within the domain of the temperature control subsystem, then the task force, via the subprogram manager, will present alternatives which could be used to resolve the problem. These recommendations may, for example, require an increase in the subsystem heater power budget, or an increase in weight allocation, or a relaxation of design requirements, or possibly a combination of these items. A product of the Phase IB effort will be a detailed temperature control management plan, giving the mechanics for implementing the plan. The mechanics of implementation may be adaptable from a similar plan generated for Apollo by TRW (Document No. 2141-6007-RU000). Briefly, this plan compares temperature requirements and predicted performance for all items in the Apollo CSM and LEM fluid loops establishing either problem areas or margins and trends of margins.

Commencing with Phase II, a preliminary list of materials will be issued, along with a release for long-lead items. There will be two

formal design reviews during Phase II. The first of these, the interim design review (IDR), will take place prior to engineering model drawing release. The CDR will occur prior to final drawing release.

3.14.3 Development and Fabrication

a. Analysis

The principal analytical efforts will be directly toward the development of detailed lumped node mathematical models of the flight spacecraft, using digital computer programs to solve the models. The analysis is an iterative process in which changes in configuration, thermal environments, component information and development test results are continually incorporated in the analysis. The confidence and experience gained in the correlation of analysis with test and flight data from the Vela, OGO and Pioneer spacecraft programs are directly applicable to Voyager.

The analyses are performed utilizing the TRW thermal analyser, shape factor programs, script F, and space heating programs. The TRW thermal analyzer lumped node finite difference digital computer program is very versatile and can accommodate not only radiative and conductive heat exchange but also convective heat transfer, as required for on-stand heating analysis. The space heating program evaluates direct solar irradiation, incident planetary albedo and planetary thermal radiation heat fluxes.

Reliability assessments of failure modes and failure effects upon the temperature control of the flight spacecraft will be performed. The assessment will consist of examining such effects as caused by failures of louvers, heaters and thermostats. The effects caused by material property degradation will also be examined.

Because it is necessary to calculate at least approximate operating temperatures for all internal and external components early in the spacecraft program, detailed thermal analysis will begin immediately in Phase IB, utilizing typical values of louver assembly performance, insulation effectiveness, thermal radiation properties, and interface conductances. As the program progresses and more detailed information becomes available from other analyses and the development test program

and better definition of the spacecraft components, these analyses will be refined.

Main Compartment. The mathematical model of the main compartment will embody the results of the development test program and of other analytical studies, such as those performed for the insulation assemblies, louvers, and for structural thermal characteristics.

The analytical effort will establish design information such as component placement, type of surface finishes, amount of louver area required, required insulation performance, and anticipated component temperature histories.

Solar Array. Solar array temperature variations, both spacial and temporal, will be predicted in order to determine the power generating history and mechanical integrity of the array. Array temperature prediction is one of the items usually most amendable to analysis. For the Voyager configuration, the support structure and the presence of the rocket nozzle may cause local problems which must be considered. The effects of rocket engine firing will also be determined.

External Equipment. Detailed thermal analysis will be performed on external equipment to determine the amount of insulation, surface coatings, geometry, and heater power requirements. The following is a list of the major items requiring analysis:

- Antennas
- Gimbals
- Attitude control lines and valves
- Sensors
- Pyrotechnics
- Cabling

Antenna pointing accuracies require that thermal distortions be minimized. Cabling must be examined to insure that their temperatures remain within bounds, especially flexible cabling such as those used for the antennas and planetary science package.

Plume and Rocket Nozzle Heating. Exhaust plume and rocket nozzle heating analysis will be performed. The analysis will determine the amount of ablative material required by the nozzle in order not to overheat the solar array, and characterize the thermal nature of the exhaust plume as it affects the spacecraft.

b. Design

An integral part of thermal analysis is thermal design, and for some activities there is not a clear distinction. The thermal design activity will have as its goal the avoidance of problems, with an aim of making the design amenable to analysis and test. In particular, adequate design margin will be provided to allow for the irreducible uncertainties such as 1) those associated with material properties measurements and the effects on surface properties caused by the prolonged exposure to mission environments, 2) component abnormal mode heat generation histories, and 3) state-of-the-art limitations in thermal-vacuum testing. In addition, the systematic identification and elimination of unreliable design features will be pursued.

The thermal design concept of the main compartment has been successfully used for the Mariner, OGO and Pioneer spacecraft. The chief problem peculiar to Voyager is its size relative to the total heat to be generated. This aspect of the flight spacecraft requires a high performance insulation system. The demands on performance are much greater than for previous space programs and, as such, greater care is required in the design, manufacture and installation of the insulation assemblies. Structural heat shorts, caused by solar array attachments, shroud attachment, struts, and the flight capsule adapter, also require greater care in design.

The louver design will be similar to those employed for the Mariner, OGO, and Pioneer programs, in that each louver blade is individually actuated by a bimetallic spring. A development effort will be conducted to improve the louver system for Voyager application.

The principal design effort of the solar array will be centered on reduction of adverse spacial temperature variations during steady

state and during transient periods. Insulation requirements will be optimized, considering survivability of array during long eclipses.

The external equipments design will concern itself with structural attachments, methods of insulating, surface finish, exterior surface geometry and heaters.

c. Fabrication

Development Models. The thermal models of the main compartment and external equipments will closely simulate the flight articles. The thermal model of the main compartment will consist of a flight propulsion module including propulsion tanks, louvers and insulation, a thermal mockup of the solar array, thermal mockups of electronic components, an inert rocket engine and nozzle, and a thermal equivalent of the flight capsule. The array mockup consists of flight type substrates which are modified on the external surface to simulate both the thermal capacity and the thermal radiation properties of the solar panels. The electronic component mockups are aluminum shells of the same dimensions and weight as a component being simulated with power dissipation provided by an internal resistor(s).

External equipment thermal models, like the main compartment thermal model, will be thermally equivalent to flight articles.

The rocket engine deboost heating test will be conducted concurrently with the propulsion subsystem development test program and as such, the temperature control subsystem will provide instrumentation and insure that the test model has adequate structure to ascertain heat soak back effects. The temperature control subsystem will also provide instrumentation for the propulsion interaction test.

Flight Spacecraft Hardware. The temperature control subsystem deliverable hardware consists of louver assemblies, insulation assemblies, heaters, and thermostats. The experience gained in manufacturing OGO, Pioneer, and Mariner C louver assemblies provides the advantage that high quality manufacturing techniques and processes already developed are directly applicable for Voyager. Louver parts will be fabricated, processed, and assembled utilizing proper tooling and assembly techniques

which will be defined in the Voyager Manufacturing Plan. Insulation fabrication and installation will rely on the results of the development program in conjunction with methods used on previous programs.

3.14.4 Test Program

Commitment of the thermal design to fabrication drawings will take place only after successful completion of the thermal model development tests. The temperature control subsystem test program is described in Table 31. A space simulation facility which includes solar simulation, will be developed to permit meaningful testing of the spacecraft. The facilities will allow study of solar illumination in abnormal as well as normal spacecraft attitudes.

a. Development Tests

The major development tests for the temperature control subsystem consist of the thermal model test and the external equipment thermal model tests. These tests will utilize vacuum chambers with liquid nitrogen cold walls and solar simulation. The tests will be conducted in environments essentially duplicating those which analysis indicates to be the most severe for each item under test. Structural heat leaks into or out of the test units will be simulated by driving boundary conditions for heat leaks to their analytically predicted (conservative) temperatures. During the tests the thermal design will be overstressed by varying heat dissipation of simulated components beyond their expected limits, both high and low, and by simulating eclipse of longer duration than expected. Boundary conditions will also be varied to find their effects upon the thermal design and the effects of having the simulated flight capsule attached to the main compartment will be determined. This type of testing allows a greater insight into those parameters most affecting the thermal design. Thus, if subsequent design changes are made, their influence on the thermal design can be adequately assessed.

During the insulation development test program, in addition to determining the effective conductance of insulation blankets, the effects of gravity upon insulation performance will be assessed. This latter aspect is of concern for blankets comprised of many layers, since gravity

Table 31. Temperature Control Subsystem Development Tests

Test Title	Purpose	Description	Test Unit and Equipment	Remarks
Development Tests				
Insulation Conductance Test	Determine insulation conductance for proposed insulation assemblies	Thermal conductance determined for insulation blankets under vacuum conditions with proper temperature ranges. The effects of gravity will be investigated.	Laboratory Samples Laboratory facilities, small cold wall vacuum chamber, cryostat, instrumentation, recorders	Several blanket configurations will be examined.
Louver Assembly Effective Properties Test	Determine effective surface properties of louver assemblies	Effective emissivity of louver assembly determined as well as effective solar absorptivity	Louver Assembly Laboratory facilities, recorders, power supplies, small cold wall vacuum chamber with solar source	Effective emissivity and absorptivity of the louver assembly will be determined as a function of temperature.
Thermal Radiation Properties Measurements	Determine thermal radiation properties of coatings and ultraviolet degradation where applicable	Sample thermal radiation properties measured. Exposed external materials to ultraviolet radiation; measure properties before and after exposure.	Laboratory Samples Heated cavity reflectometer, Gier-Dunkle integrating sphere, Beckman DS-2A modified integrating sphere, cooled radiometer total normal emittance stand, parabaloid reflectometer, ultraviolet degration facilities.	Materials measured will be those considered for use on the spacecraft, for which reliable data does not exist.
Thermal Conductance Test	Determine thermal conductance of components to mounting surface	Components mockup bolted to honeycomb panel. Mockups heated in vacuum, power and temperature gradients measured	Thermal Model Model Mockups, Honeycomb Panel Vacuum chamber, test units, instrumentation, recorders, power supplies	If analysis shows interface material needed for some components, then tests will include interface material.
Miscellaneous Thermal Tests as Required by Analysis	Determine adequacy of thermal design in which analysis indicates uncertainty	Models to be tested to those environments for which analysis indicates uncertainty in adequacy of thermal design	Thermal Model Mockups or Component EM's Vacuum chambers, solar simulators, infrared heaters, power supplies, instrumentation, recorders	Specific components requiring testing will be a resultant of detailed thermal analysis.
Thermal Model Test	Determine adequacy of main compartment thermal design and associated equipment	Space simulation test	Thermal Model of Main Compartment Space simulation chamber, solar simulators, instrumentation, recorders, power supplies	The effects of over stressing the thermal design will be determined for the main compartment and for those equipments which can be conveniently tested with it.
External Equipment Thermal Model Tests	Determine adequacy of the thermal design of the external equipment	Space simulation test	Thermal Model of External Equipments Space simulation chamber, solar simulators, instrumentation, recorders, power supplies	The effects of over stressing the thermal design will be determined.
Rocket Engine Deboost Heating Test	Evaluate heating rates from deboost engine firings and heat soak back effects	Static firing of deboost engine	Deboost Engine Firing Test Model Altitude simulation chamber, test unit, instrumentation	This test will be conducted concurrently with propulsion development tests.
Type Approval Test				
Heaters and Thermostats Type Approval Test	TA	Environmental test, vibration, temperature, thermal vacuum, shock, acceleration	Heater and Thermostats Environmental tests	
Louver Assembly Part 1 Type Approval Test	TA	Environmental test, vibration, temperature, thermal vacuum, shock, acceleration	Louver Assembly Environmental tests	

Table 31. Temperature Control Subsystem Development Tests (Continued)

Test Title	Purpose	Description	Test Unit and Equipment	Remarks
Louver Assembly Part 2 Type Approval Test	TA	Louver assembly mounted to guarded plate in vacuum chamber, effective emittance and open and closed temperatures determined	Part 1 Type Approval Louver Assembly Cold wall vacuum chamber variable temperature plate, instrumentation, recorders, power supplies, louver position indicator	This test will be conducted after Part 1 of the louver type approval test is completed.
Rocket Nozzle and Plume Heating Type Approval Test I	TA	Instrument PIM test model to determine heating effects caused by the deboost engine during retrofiring	Propulsion Interaction Test Model PIM test model, instrumentation, recorders	This test is one of a series used to qualify the thermal design.
Thermal Control Subsystem Type Approval Test II	TA	Space simulation test, solar simulation, eclipse duration	Proof Test Model Space simulation chamber, solar simulator, support fixture, infrared heaters, instrumentation, recorders	This is one of a series of tests comprising the TA test program for the thermal subsystem, external equipment which cannot be thermally qualified with the PIM will be qualified separately.
Flight Acceptance Tests				
Heaters and Thermostats Flight Acceptance Tests	FA	Heaters and thermostats applied to common plate in vacuum chamber, plate temperature cycled	Heaters and Thermostats Vacuum chamber, variable temperature plate, instrumentation recorders, power supplies	
Louver Assembly Flight Acceptance Test	FA	Louver assembly mounted to instrumented plate and checked for temperature settings	Louver Assembly Environmental chamber, variable temperature plate, instrumentation, power supplies, louver position indicator	
Thermal Control Subsystem Flight Acceptance Test	FA	Space simulator test, solar simulation	Flight Spacecraft Space simulation chamber, solar simulator, support fixture, infrared heaters, instrumentation	This test is one of a series of tests comprising the FA test program for the thermal subsystem, external equipment which cannot be thermally acceptance tested with the thermal vacuum test of the flight spacecraft will be tested separately.
Life Tests				
Heater and Thermostats Life Tests	Determine reliability of heaters and thermostats	Heaters and thermostats applied to common plate in vacuum chamber, plate temperature cycled	Heaters and Thermostats Vacuum chamber, variable temperature plate, instrumentation, recorders, power supplies	Reliability analysis will establish number of cycles. Information of life tests will be considered.
Louver Assembly Life Tests	Determine reliability of louver assembly	Louver assembly mounted to a heated plate in a vacuum chamber, temperature cycled	Louver Assembly Cold wall vacuum chamber, variable temperature plate, instrumentation, power supplies, louver position indicator	A reliability analysis will establish the number of cycles louver assembly is tested.

effects must be understood in order to properly interpret space simulation test data.

The louver development test program will be concerned with improving existing louver designs and with determining the thermal performances of the louver assembly, particularly when irradiated by the sun.

The engine will be statically fired at simulated altitude as part of the engine development test program and during the propulsion interaction test. Evaluation will be made of heat fluxes resulting from the exhaust plume and from the ablatively cooled rocket nozzle. The models will also be instrumented to evaluate heat conduction from the rocket engine.

b. Type Approval

Component level TA tests will be conducted for louver assemblies, heaters and thermostats. Insulation assemblies will be qualified during the structural vibration/acoustic test.

The TA test for the temperature control subsystem will consist of a series of tests. The principal test of this series will be the PTM test. Those items, such as the antennas, which may not be thermally qualified during the PTM test will be qualified separately.

c. Flight Acceptance

Louver assemblies, heaters, and thermostats will be FA tested at a component level, whereas insulation assemblies will be tested at the system level FA test. The temperature control subsystem will be tested in a series of FA tests, the principal test of this series being the thermal-vacuum test of the flight spacecraft.

4. SPACECRAFT ASSEMBLY AND CHECKOUT OPERATIONS

This section discusses the assembly and checkout operations for the various spacecraft models. These models include:

- Configuration model (CM)
- Engineering model (EM)
- Propulsion interaction model (PIM)
- Proof test model (PTM)
- First flight model
- Second flight model
- Third flight model

The section covers engineering tasks required to plan the operations, identifies the elements of planning and control needed to support the operation, discusses the science test facilities, defines the usage of an EM spacecraft, briefly discusses the assembly of the EOSE with the central data system to form a system test complex, and briefly summarizes the assembly and checkout sequence, including the basic assumptions used to develop this sequence. A detailed description of the assembly and checkout operations performed on each spacecraft is included in Appendix A of this volume.

4.1 Operations Engineering

A major engineering task associated with spacecraft assembly and checkout is that of insuring that spacecraft design and development is properly oriented toward assembly and test. Assembly and test engineers are assigned the task of maintaining current knowledge of the design details of both airborne and operational support equipment, and analyzing this data in terms of assembly and test operations. The results of these studies are used to feed back information to the design areas (in case of operational problems), and are the basis for detailed design of the assembly and test operations procedures, facilities, test equipment, and computer programs.

The operations design task includes the detailed analysis of the assembly and checkout requirements as determined by the spacecraft design. A configuration model spacecraft will be fabricated and continually updated to support this activity. The engineering personnel who participated in the initial design effort form the nucleus of an operations design group, which is augmented by other specialists from the spacecraft assembly and test operation. The analyses of the assembly and checkout requirements are used to form a detailed plan covering the identification and preparation of operating procedures, the detailed sequence of operations, the design of the EOSE and special test facilities, the design and implementation of the computer programs, and the assembly and test schedules. Continuous updating of the computer software and redesign of the physical elements is performed during the assembly and checkout sequence.

Upon completion of this task, the personnel from this group form the nucleus of the assembly and test crews, under the direction of the assembly and test manager.

4.2 Planning and Control Operations

A test operations planning and control group will be established as the focal point of all scheduling, planning, controls, and records. The scheduling effort will include the over-all spacecraft operations schedules, the required delivery dates for subsystem assembly into the spacecraft, and test facilities requirements. The controls effort includes hardware storage and maintenance of configuration status in accordance with the latest engineering changes. This group also provides support in expediting the delivery of spacecraft and OSE for spacecraft operations. In addition to controlling the normal records that flow with the spacecraft and OSE, this group has the responsibility to control procedures, computer software, and the magnetic tapes (via the data management system), containing information on subsystem trends collected during all phases of spacecraft testing.

4.3 Science Test Facilities

A science test facility will be supplied in the TRW Voyager assembly building. This facility will be equipped to support experimenter requirements. This facility will be under the direction and control of TRW for use by the experimenters and JPL.

4.4 Engineering Model Spacecraft

The primary purpose of an EM spacecraft is to allow an early definition of major spacecraft and OSE design problems prior to the assembly and checkout of flight configured hardware. The beginning of assembly and checkout of the EM spacecraft from EM subsystems is scheduled 9 months before the start of PTM spacecraft assembly. This provides adequate time for any redesign effort.

In addition to the design verification task, some test functions must be performed on a spacecraft rather than a spacecraft simulator. These functions include:

- Facilities checkout
- Procedure debugging
- Computer software debugging
- Operations crew training
- DSIF compatibility
- Launch vehicle compatibility.

The EM spacecraft allows these additional tasks to be performed and does not subject a flight configured spacecraft to possible damage that could result from design, facility or handling problems.

4.5 Subsystem EOSE and Central Data System

The assembly of the central data system with the subsystem test sets will be performed by assembly and test operations personnel. The assembly task is performed by this group to provide training for the personnel that will actually operate these system test complexes. The EOSE, assembled with the central data system, will form a system test complex. This assembly will follow the same basic flow as the assembly of a spacecraft. The final acceptance test of a system test complex will be demonstrated by performing an integrated system test with the EM spacecraft.

4.6 Spacecraft Assembly and Checkout Operations

This section presents a brief summary of the assembly and checkout sequence of all spacecraft models. One basic assumption made to develop this assembly sequence is that sufficient telemetry is available to perform fault isolation to the black box level. Test points required for system testing and subsystem interface checks will be brought, via a flyable test harness, to a subpanel test connector. Test points required for pad testing will be brought through the subsystem interface to an umbilical J-box. The combination of these test harnesses constitutes the direct access test circuitry. All other test points will be available at the individual boxes only and will appear on separate test connectors such that they can be used without requiring demating of the access test harness. Some system level tests, such as space simulation testing, may

require special test cables between the EOSE and the separate test connectors.

4.6.1 Subsystem Assembly

After each black box has completed environmental and flight acceptance tests, it will be delivered to the subsystem assembly area, where a system test complex has been assembled. Each electrical subsystem will be mechanically assembled to a spacecraft subpanel. The subpanel flight harness is then mechanically installed to the subpanel, electrically tested, and the connectors mated to the boxes. Upon completion of this electrical and mechanical assembly operation, each subsystem will be tested, as a subsystem, using the system test complex equipment. The following subsystem tests will be included:

- Perform all functions with ± 15% variations in external supply voltages over the flight acceptance temperature range.
- Perform all functions with nominal voltage and temperatures 15% in excess of flight acceptance limits expressed in degrees F.
- Exhibit noncatastrophic performance in the presence of noise injection, power frequency variation, power overshoot, and power transients, all at 15% in excess of those specified in the appropriate detailed specification, and demonstrate that components have not been degraded by the test.

These tests performed in the system test configuration will allow the necessary subsystem trend data to be compiled into a subsystem history log. Upon successful completion of the subsystem tests, the subsystems will be logged into a controlled stores area, along with the data logs, and will be withdrawn from stores when needed for spacecraft assembly.

4.6.2 Aft Equipment Module

The spacecraft aft equipment module will be mechanically installed on the handling fixture at the assembly and checkout area. The pyrotechnic and guidance and control pneumatics equipment will also be mechanically installed in the aft equipment module at this time.

4.6.3 Cabling Subsystem

The next subsystem to be installed and integrated will be the spacecraft harness and J-box subsystem. This operation will be split into two parts: the actual mechanical tiedown and cable routing operation, and the hi-pot and continuity check. The routing and cable tiedown will involve a certain amount of handling and flexing of the cables. This necessitates a continuity and hi-pot test to be conducted after the completion of this operation.

4.6.4 Power Subsystem

The power subsystem, with test batteries, is mounted to a subpanel. After complete acceptance tests, the power subsystem is delivered to the spacecraft. The power subpanel is mechanically mounted to the equipment mounting panel installation fixture and functionally tested for subsystem integrity. At the completion of the functional test, the spacecraft electrical interfaces to the power subsystem will be checked. The power subsystem distribution will also be checked at the appropriate end of the harness, including the distribution through the various J-boxes. The timing oscillator for the spacecraft system is checked and its performance validated at this time. Tests are conducted to investigate noise and transients, in addition to performing an end-to-end test of the power subsystem.

4.6.5 Command Subsystem

The command subsystem subpanel is mechanically mounted to the equipment mounting panel installation fixture after completion of the integration and test of the power subsystem. The subsystem is functionally tested and integrated with the power subsystem and noise and transient measurements made. This test will constitute an end-to-end test of the command subsystem and the power subsystem.

4.6.6 Telemetry Subsystem

The telemetry subsystem is installed next. At the completion of installation, functional test, and subsystem integration, the command and power subsystem telemetry outputs will be calibrated. The testing

concept utilized for the remainder of the spacecraft assembly and checkout operations will be to calibrate each subsystems' telemetry as that subsystem is integrated into the spacecraft.

4.6.7 Data Storage Subsystem

The data storage subsystem, after having been mechanically installed, functionally tested, and the spacecraft electrical interfaces checked, is tested to ascertain that all telemetry data words can be properly stored and read out for all data encoder formats, modes, and rates. Concurrently, the telemetry for the data storage subsystem is calibrated.

At this point, the harness assembly, power subsystem, command subsystem, telemetry subsystem, and data storage subsystem are integrated into the spacecraft. A compatibility test is performed to ascertain if interference exists between subsystems. If any interference exists, an investigation is made to determine the level of system degradation. This compatibility test investigates the system grounding with respect to the spacecraft skin and the negative side of the spacecraft batteries.

4.6.8 Radio Subsystem

After completion of the foregoing compatibility test, the radio subsystem subpanel is installed, tested, and integrated with the spacecraft. At this time, VSWR tests and insertion loss tests will be conducted on the RF cables, diplexers, and couplers. The integration of the radio subsystem provides the ability for downlink communications.

Upon completion of the radio subsystem integration test, a compatibility test between the radio subsystem and the rest of the integrated subsystems is performed. This test is to demonstrate that RF interference does not exist between the power, command, telemetry, data storage, and radio subsystems and that the subsystems are mutually compatible. This compatibility test also investigates whether or not RF multipaths and RF currents through the grounding system exist.

4.6.9 Computing and Sequencing Subsystem

The next subsystem subpanel to be installed will be the computing and sequencing subsystem. A functional test is performed to test the

integrity of this subsystem. Upon successful completion of the functional test, integration with the spacecraft is performed. During the integration test, functional operation of the power, telemetry, and data storage subsystems is demonstrated under the control of the computer and sequencer.

After integration and telemetry calibration of the computer and sequencer is completed, a compatibility test is performed to check for system interference with the computer and sequencer.

4.6.10 Guidance and Control Subsystem

Next, the guidance and control subsystem subpanel is installed and functionally tested. Sensors will be mounted and functionally tested, and compatibility with the radio subsystem will be investigated. At the completion of this test, the guidance and control telemetry words are calibrated. The same sequence used to perform previous compatibility tests is used for this compatibility test. This test checks grounding and interference between all subsystems installed to date.

4.6.11 Propulsion Subsystem

At this time, the propulsion subsystem module is mechanically installed to the aft equipment module, along with the planetary scan package adapter. Electrical interfaces with the telemetry subsystem and the guidance and control subsystem are checked. The full flight acceptance tests have been completed prior to delivery at TRW. A functional test is performed to demonstrate propulsion module operation with the command, telemetry, power, and guidance and control subsystems. Leak testing of the pressurization system is performed.

A compatibility test is performed to check RF interference, and grounding. This test is now essentially a spacecraft integrated system test, since all major spacecraft subsystems, excluding science, have been installed and tested. This assures maximum compatibility of the spacecraft subsystems before the integration of the planetary scan packages and the fixed science packages.

4.6.12 Planetary Scan Package

The planetary scan packages (PSP) are delivered as GFE; therefore, only the electrical interface, mechanical attachment and telemetry

calibration operations are performed. In addition, a functional test is conducted as specified by JPL. A compatibility test is performed to ensure that the spacecraft and all of the subsystems do not adversely affect proper PSP operation.

4.6.13 Fixed Science Packages

The fixed science packages (FSP) are mechanically attached, functionally checked, and the spacecraft electrical interfaces checked. The telemetry channels are calibrated concurrently with the interface test.

4.6.14 Science Payload Compatibility Test

At the completion of the FSP integration, a complete spacecraft compatibility test is performed to determine if interference exists between the spacecraft system and the PSP or FSP experiments. All telemetry calibrations are rechecked at this time to ensure that no significant changes have occurred from the time of initial calibration.

4.6.15 Temperature Control Subsystem

The installation of the temperature control subsystem is now completed with the exception of some of the thermal insulation. The sensors for this subsystem will have been installed during the harness assembly integration, and therefore, the telemetry will have been calibrated. The heaters, louvers, and thermostats are installed and functionally tested at this time.

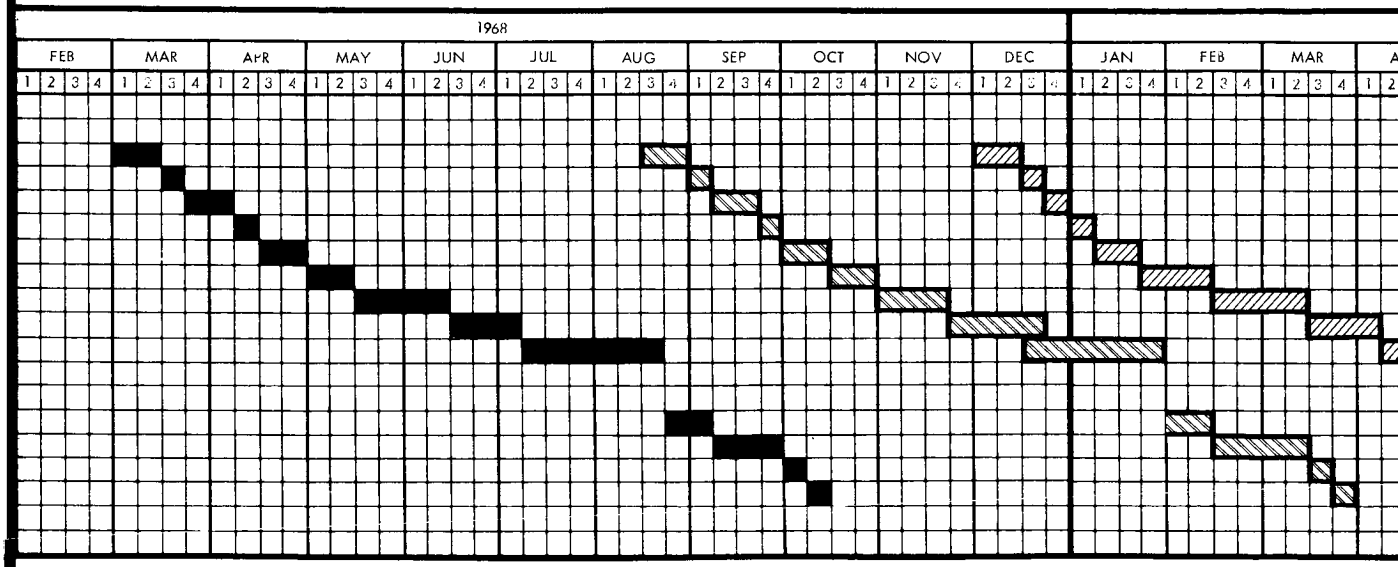
4.6.16 System Complex-Spacecraft Integrity Test

A final test constitutes a complete checkout of the ground station with all of the EOSE as a complete ground station-spacecraft system. Grounds, noise, interference and compatibility will be checked.

4.7 Assembly and Checkout Schedule

Figure 37 is the schedule for each spacecraft. The assembly and checkout tasks are listed in major test groupings. Facilities loading, hardware delivery schedules, and manpower requirements are derived from this schedule. Figure 38 shows the sequence of assembly and

SPACECRAFT ASSEMBLY AND CHECKOUT SCHEDULE	JAN		
	1	2	3
RECEIVE SPACECRAFT WAFER STRUCTURE, MOUNT TO VERTICAL STAND, INSTALL PYROTECHNIQUES S/S AND G AND C PLUMBING AND LEAK TEST (1-4a)*			
INSTALL SPACECRAFT HARNESS ASSEMBLY AND HI-POT AND CONTINUITY CHECK (5-5d)			
INSTALL, INTEGRATE AND TEST POWER SUBSYSTEM (6-8f) (LESS SOLAR ARRAYS)			
INSTALL, INTEGRATE AND CHECK COMMAND SUBSYSTEM (9-11e)			
INSTALL, INTEGRATE AND TEST TELEMETRY SUBSYSTEM (12-15)			
INSTALL, INTEGRATE AND TEST DATA STORAGE SUBSYSTEM MODULE (16-19)			
INSTALL, INTEGRATE AND TEST RADIO SUBSYSTEM AND RELAY LINK SUBSYSTEM (20-27)			
INSTALL, INTEGRATE AND TEST COMPUTER AND SEQUENCER MODULE (28-31)			
INSTALL, INTEGRATE AND TEST GUIDANCE AND CONTROL SUBSYSTEM MODULE (32-35)			
INSTALL, INTEGRATE AND TEST LEMDE MODULE AND P. S. P. VEHICLE ADAPTER (36-40)			
INSTALL, INTEGRATE AND TEST PLANETARY SCAN PACKAGE AND FIXED SCIENCE PACKAGE (41-47)			
INSTALL AND TEST TEMPERATURE CONTROL SUBSYSTEM (48-49)			
INSTALL, INTEGRATE AND TEST SPACECRAFT TEST CAPSULE (50-53)			



ENGINEERING MODEL SPACECRAFT

PROPULSION INTERACTION MODEL SPACECRAFT

PROOF TEST MODEL SPACECRAFT

1 FIRST FLIGHT SPACECRAFT

2 SECOND FLIGHT SPACECRAFT

3 THIRD FLIGHT SPACECRAFT

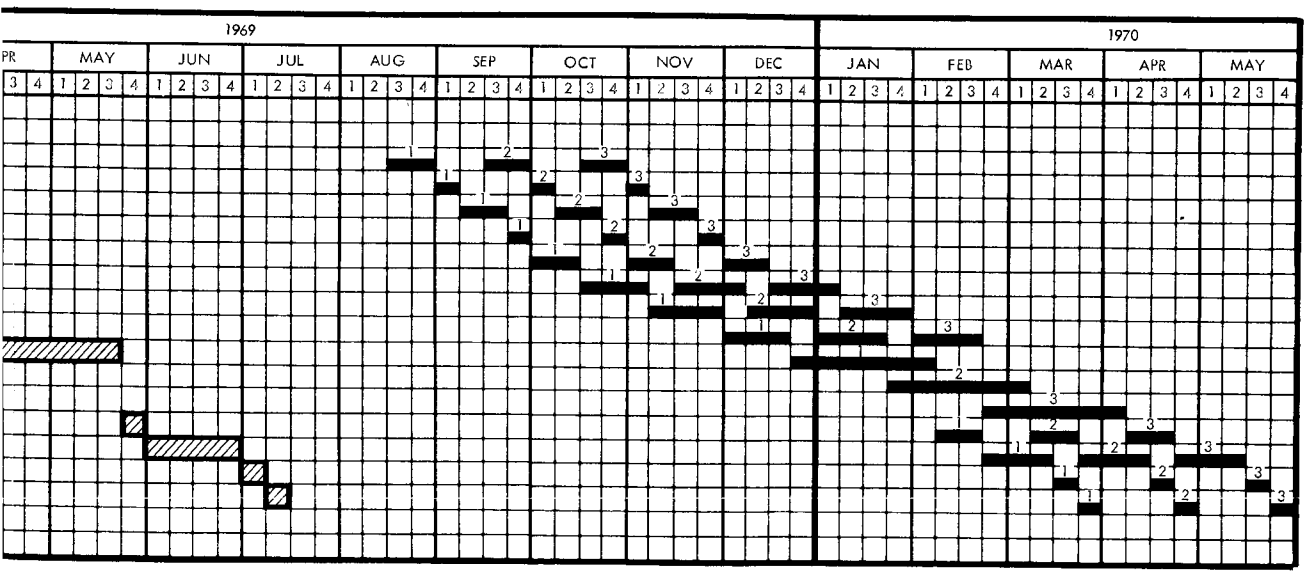
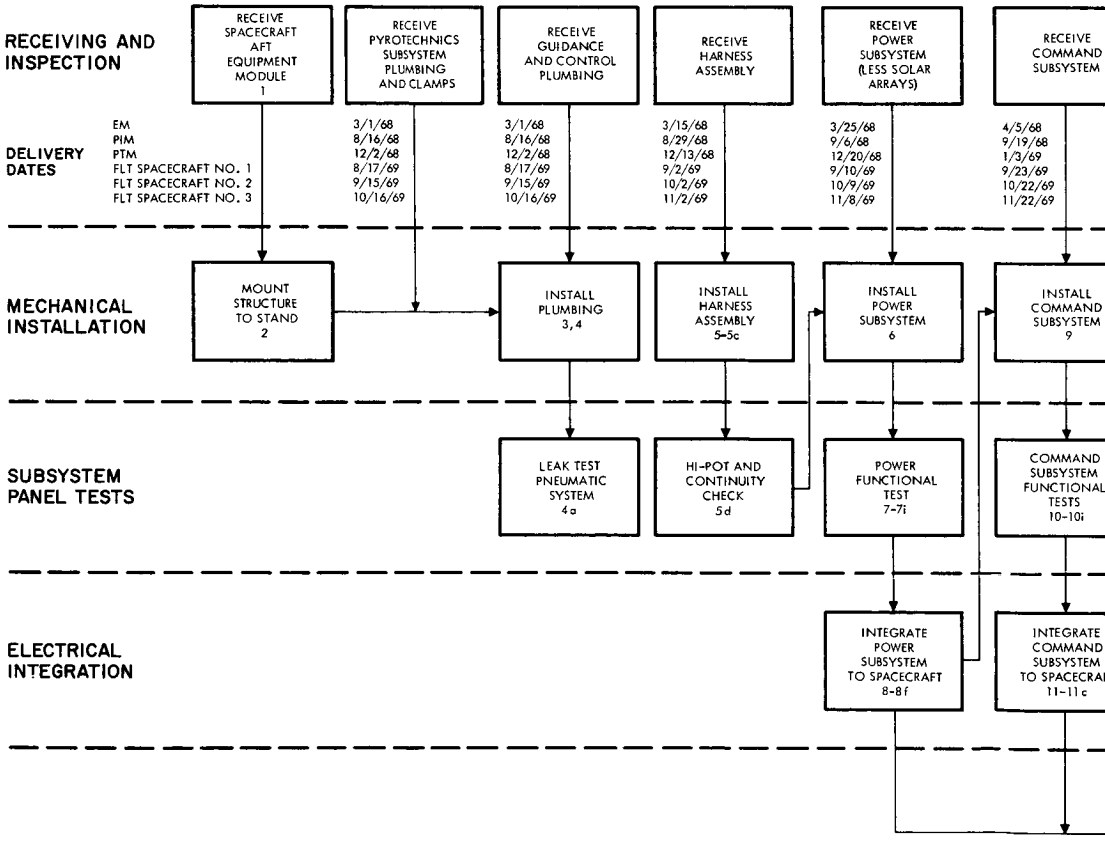
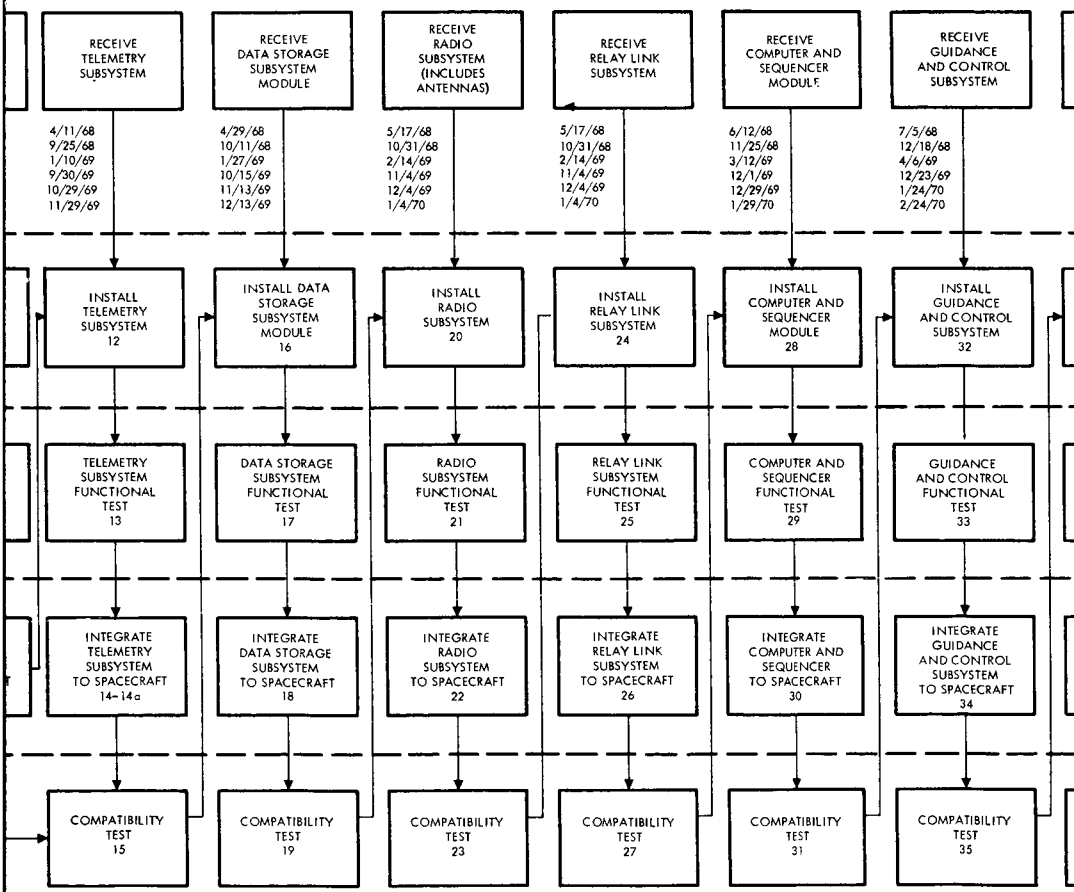


Figure 37. Spacecraft Assembly and Checkout Schedule


RECEIVING AND INSPECTION
RECEIVE SPACECRAFT AFT EQUIPMENT MODULE 1
RECEIVE PYROTECHNICS SUBSYSTEM PLUMBING AND CLAMPS
RECEIVE GUIDANCE AND CONTROL PLUMBING
RECEIVE HARNESS ASSEMBLY
RECEIVE POWER SUBSYSTEM (LESS SOLAR ARRAYS)
RECEIVE COMMAND SUBSYSTEM
DELIVERY DATES
EM
PIM
PTM
FLT SPACECRAFT NO. 1
FLT SPACECRAFT NO. 2
FLT SPACECRAFT NO. 3
3/1/68 8/16/68 12/2/68 8/17/69 9/15/69 10/16/69
3/1/68 8/16/68 12/2/68 8/17/69 9/15/69 10/16/69
3/15/68 8/29/68 12/13/68 9/2/69 10/2/69 11/2/69
3/25/68 9/6/68 12/20/68 9/10/69 10/9/69 11/8/69
4/5/68 9/19/68 1/3/69 9/23/69 10/22/69 11/22/69
MECHANICAL INSTALLATION
MOUNT STRUCTURE TO STAND 2
INSTALL PLUMBING 3,4
INSTALL HARNESS ASSEMBLY 5-5c
INSTALL POWER SUBSYSTEM 6
INSTALL COMMAND SUBSYSTEM 9
SUBSYSTEM PANEL TESTS
LEAK TEST PNEUMATIC SYSTEM 4a
HI-POT AND CONTINUITY CHECK 5d
POWER FUNCTIONAL TEST 7-7i
COMMAND SUBSYSTEM FUNCTIONAL TESTS 10-10i
ELECTRICAL INTEGRATION
INTEGRATE POWER SUBSYSTEM TO SPACECRAFT 8-8f
INTEGRATE COMMAND SUBSYSTEM TO SPACECRAI 11-11c
COMPATIBILITY INTERFERENCE TESTS


RECEIVE TELEMETRY SUBSYSTEM
RECEIVE DATA STORAGE SUBSYSTEM MODULE
RECEIVE RADIO SUBSYSTEM (INCLUDES ANTENNAS)
RECEIVE RELAY LINK SUBSYSTEM
RECEIVE COMPUTER AND SEQUENCER MODULE
RECEIVE GUIDANCE AND CONTROL SUBSYSTEM
4/11/68 9/25/68 1/10/69 9/30/69 10/29/69 11/29/69
4/29/68 10/11/68 1/27/69 10/15/69 11/13/69 12/13/69
5/17/68 10/31/68 2/14/69 11/4/69 12/4/69 1/4/70
5/17/68 10/31/68 2/14/69 11/4/69 12/4/69 1/4/70
6/12/68 11/25/68 3/12/69 12/1/69 12/29/69 1/29/70
7/5/68 12/18/68 4/6/69 12/23/69 1/24/70 2/24/70
INSTALL TELEMETRY SUBSYSTEM 12
INSTALL DATA STORAGE SUBSYSTEM MODULE 16
INSTALL RADIO SUBSYSTEM 20
INSTALL RELAY LINK SUBSYSTEM 24
INSTALL COMPUTER AND SEQUENCER MODULE 28
INSTALL GUIDANCE AND CONTROL SUBSYSTEM 32
TELEMETRY SUBSYSTEM FUNCTIONAL TEST 13
DATA STORAGE SUBSYSTEM FUNCTIONAL TEST 17
RADIO SUBSYSTEM FUNCTIONAL TEST 21
RELAY LINK SUBSYSTEM FUNCTIONAL TEST 25
COMPUTER AND SEQUENCER FUNCTIONAL TEST 29
GUIDANCE AND CONTROL FUNCTIONAL TEST 33
INTEGRATE TELEMETRY SUBSYSTEM TO SPACECRAFT 14-14a
INTEGRATE DATA STORAGE SUBSYSTEM TO SPACECRAFT 18
INTEGRATE RADIO SUBSYSTEM TO SPACECRAFT 22
INTEGRATE RELAY LINK SUBSYSTEM TO SPACECRAFT 26
INTEGRATE COMPUTER AND SEQUENCER TO SPACECRAFT 30
INTEGRATE GUIDANCE AND CONTROL SUBSYSTEM TO SPACECRAFT 34
COMPATIBILITY TEST 15
COMPATIBILITY TEST 19
COMPATIBILITY TEST 23
COMPATIBILITY TEST 27
COMPATIBILITY TEST 31
COMPATIBILITY TEST 35

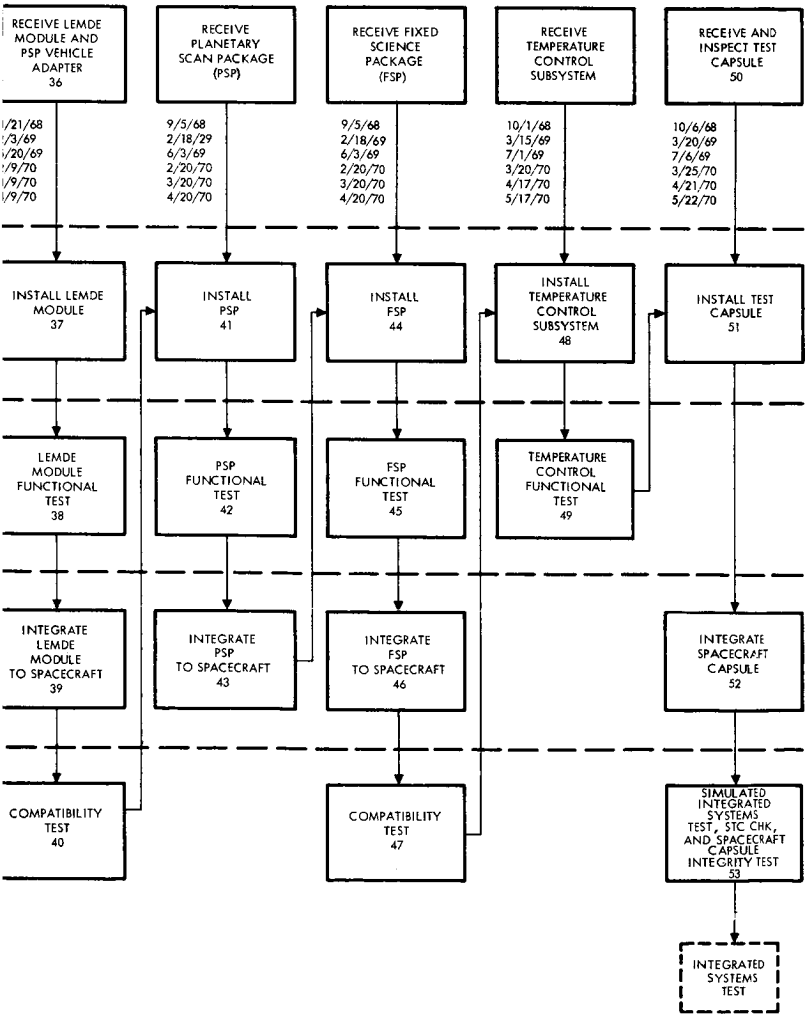


Figure 38. Spacecraft Assembly and Checkout Flow

operations tasks for each spacecraft. The starting date for each task is listed in Figure 37. The operations numbers, listed in each box, are defined in the detailed task descriptions located in Appendix A of this volume.

5. SPACECRAFT SYSTEM TESTS

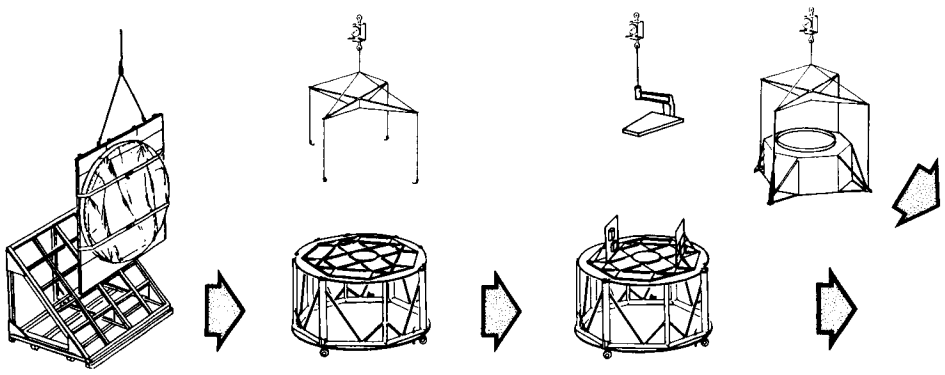
This section discusses all tests and operations performed on each spacecraft from the first integrated system test to shipment. A pictorial flow from assembly to shipment to AFETR is diagrammed in Figure 39. The test concepts underlying the test plan and sequence of operations are discussed. A brief description of the system test sequence is followed by flow charts and a schedule which defines the test program for each spacecraft. The schedule also shows test facility loading and will be used to develop manpower loading curves. Detailed functional descriptions of all tests and operations are included in Appendix A.

5.1 Test Philosophy

System testing of the engineering model (EM) spacecraft is performed primarily as a system compatibility and facility validation task. It will be used to verify OSE design, debug procedures and operations, and train personnel. The EM will be used at Goldstone to perfect the mission operation sequence and to verify initial deep space network (DSN)-spacecraft compatibility. The EM spacecraft and the propulsion interaction model (PIM) spacecraft will be used to validate launch site procedures, equipment and facilities.

The system testing of the proof test model (PTM) is aimed at system design verification and environmental type approval of flight type hardware. It will also serve to further debug procedures, operations, and OSE and to train personnel. Any design changes made as a result of the EM system tests will be specifically checked. The PTM will also be used to perform reliability life tests and to complete the DSN-spacecraft compatibility verification.

The acceptance testing of the flight spacecraft system is performed primarily as a workmanship verification. The major design problems will have been resolved by the EM, PIM, and PTM spacecraft.

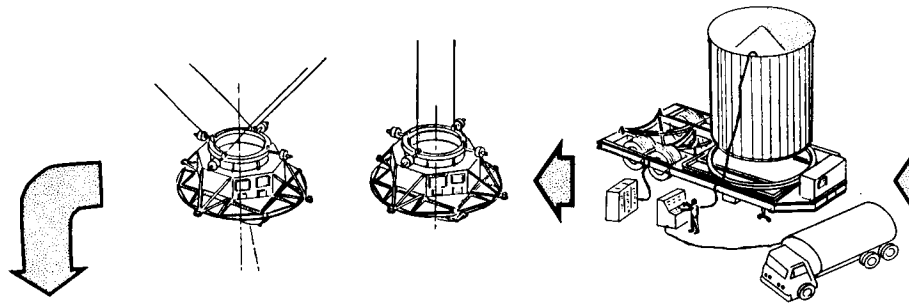


RECEIVE SPACECRAFT AFT EQUIPMENT MODULE

EQUIPMENT MODULE MOUNTED TO ASSEMBLY STAND

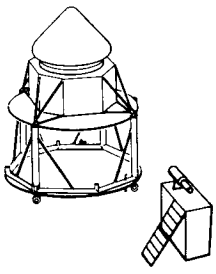
INSTALL POWER SYSTEM (INSTALL DUMMY ARRAY)

INSTALL LI
AFT EQUIP
EQUIPMEN

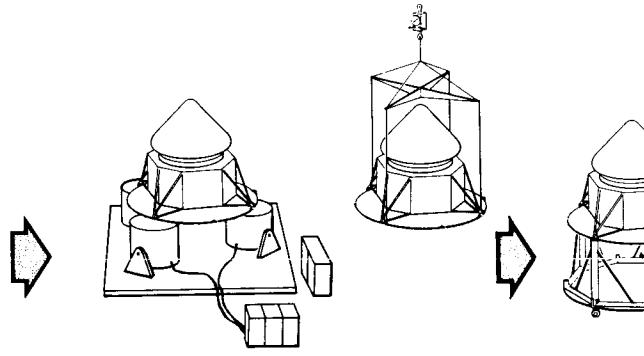


PERFORM WEIGHT AND C AND G

RECEIVE SPACECRAFT IN MASS PROPERTIES AREA AND LOAD INERT PROPELLANTS

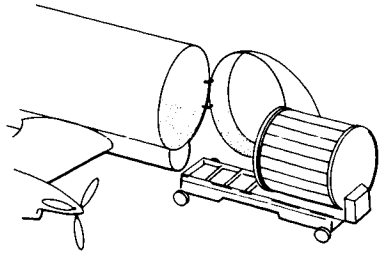


MOVE SPACECRAFT TO ALIGNMENT FACILITY AND PERFORM ALIGNMENTS

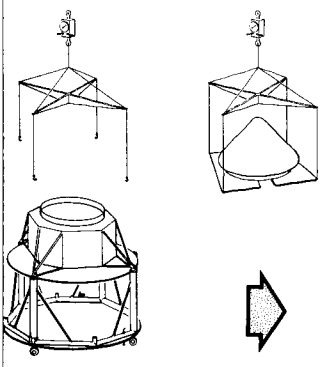


MOVE SPACECRAFT TO VIBRATION/ ACOUSTIC FACILITY AND PERFORM COMBINED VIBRATION/ACOUSTICS TEST

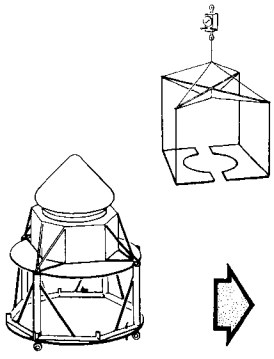
INSTALL SPACECRAF
IN VERTICAL STAND



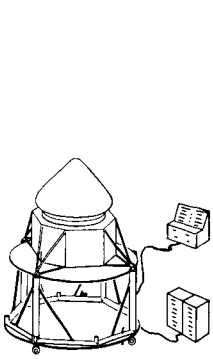
M
TI



EMDE MODULE TO SPACECRAFT
MENT MODULE AND INSTALL
T PANELS ON LEMDE MODULE

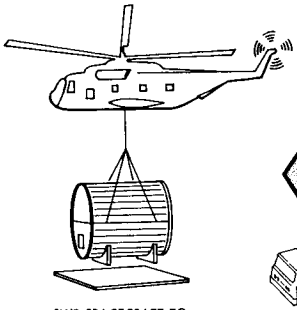


INSTALL TEST CAPSULE
TO THE SPACECRAFT

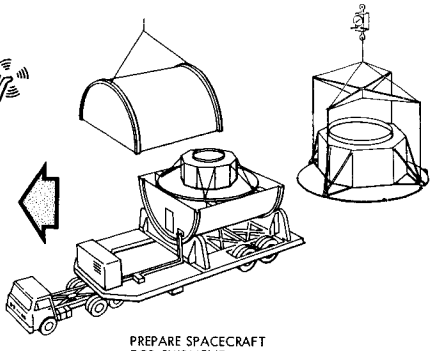


PERFORM INTEGRATED
SYSTEMS TEST

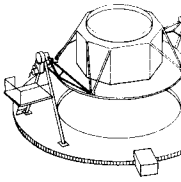
M
E



SHIP SPACECRAFT TO
REDONDO BEACH AREA



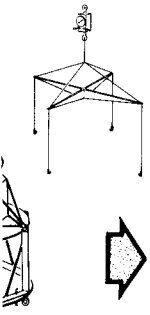
PREPARE SPACECRAFT
FOR SHIPMENT



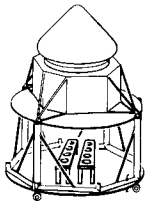
ASSEMBLE SPACECRAFT TO TEST S
(MAG FACILITY) AND PERFORM M
PROPERTIES TESTS

MAGNETIC STABILITY

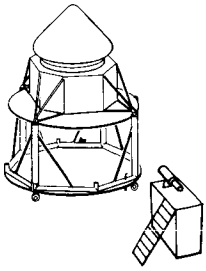
MAGNETIC MAPPING (US
COILS



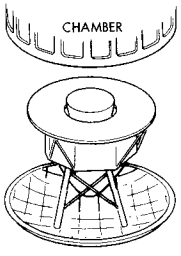
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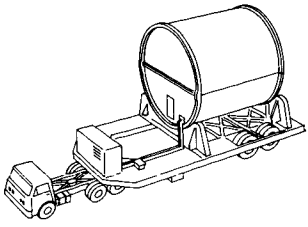
PERFORM FREE
MODE TEST



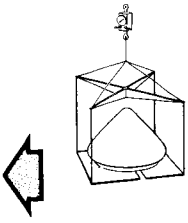
MOVE SPACECRAFT TO ALIGNMENT
FACILITY AND VERIFY ALIGNMENTS

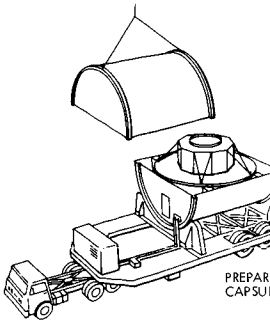


MOVE SPACECRAFT TO THERMAL V
FACILITY AND PERFORM SPACE SIM
TEST (INCLUDING OFF-AXIS AS RE



OVE SPACECRAFT AND
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PREPAR
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2

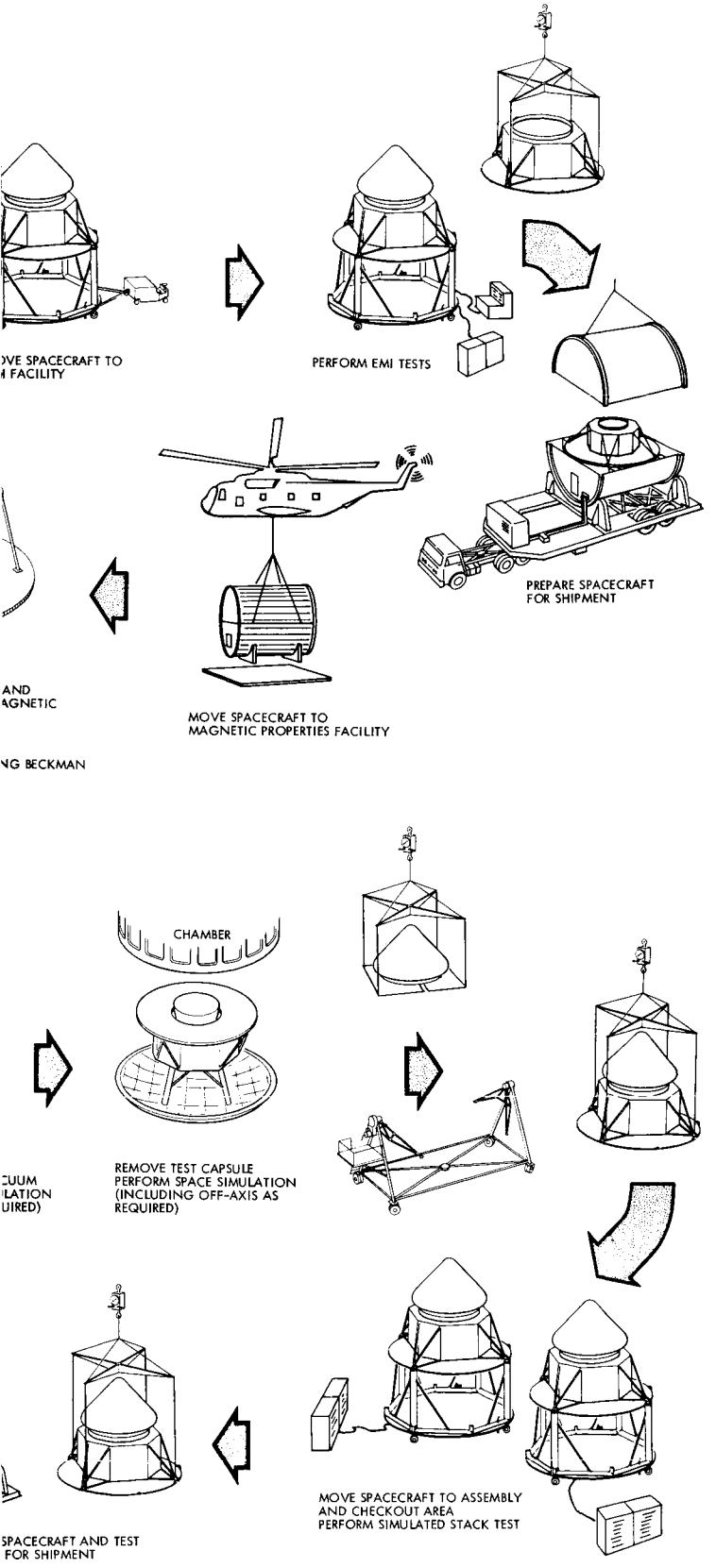


Figure 39. Operations for Assembly and Shipment of Spacecraft

During system testing, the electrical interfaces between the spacecraft and the EOSE will be minimized. Test cables constitute a non-flight configuration and can cause abnormal system operation as well as injecting unwanted noise. The goal will be to operate the spacecraft in a configuration as close as possible to a flight configuration. Sufficient spacecraft telemetry will be provided to isolate faults to the provisional spares level and to enable verification of command status. Certain commands are required for testing and will aid in keeping hardline use to a minimum. These commands will primarily be used to check redundant system operation.

Wherever possible, system test stimulation (external stimuli used to excite flight equipment, usually having only a mechanical interface with the spacecraft) will be used, rather than simulation, (signal injection) to perform an end-to-end system test. The same stimuli used during system tests will be used at the subsystem level. However, the subsystem test may incorporate additional stimulation or simulation.

It is assumed that a prototype capsule will be supplied at the completion of PTM spacecraft assembly to be used by all spacecraft as a test capsule. This test capsule will be required to survive type approval level environmental testing. A capsule simulator will be built to serve as an additional test capsule. The capsule simulator will not be exposed to environmental testing.

5.2 System Test Sequence Description

Because of the schedule constraints imposed by the simulated stack test, in which two spacecraft are tested together, the test sequence will not necessarily be identical for all spacecraft. The second constraint on the sequence is facility usage. All tests and operations are scheduled such that only one test area of each major type is required.

The sequence through the electromagnetic interference tests, magnetic properties, tests and previbration system tests for the EM is identical to the PTM and flight spacecraft sequence. The EM will not be exposed to environmental testing but it will be taken to each environmental test area for facility validation prior to testing of the PTM spacecraft.

The EM and the PTM are scheduled for shipment to Goldstone at various times begnining early in 1969 for deep space instrumentation facility compatibility checks. When a spacecraft is not available, magnetic test data tapes will be used for mission dependent equipment-deep space instrumentation facility compatibility checks. After the 1971 launch, the EM will be sent to Goldstone to replace the PTM as required.

The propulsion interaction model (PIM) spacecraft will be a complete spacecraft composed of engineering model subsystems. The spacecraft will be assembled and checked out at TRW to the point where the spacecraft is ready for propulsion subsystem integration. At that time the spacecraft and its STC will be shipped to the propulsion interaction test facility. The qualification model propulsion subsystem, which will have been at White Sands for subsystem qualification testing, will then be installed in the aft equipment module and integrated with the existing spacecraft. An EM or prototype science subsystem will then be installed and integrated along with the thermal control subsystem. An integrated system test and spacecraft alignments will then be performed. Appendix B of this volume describes the alignment task. A propulsion interaction test will be performed to verify proper operation of the guidance and control subsystem during operation of the propulsion subsystem and to verify compatibility of the spacecraft subsystems in that environment.

An integrated system test will be performed following the assembly and checkout of the PTM and flight spacecraft to verify functional integrity. All spacecraft subsystems and science instruments will be tested, as well as the spacecraft-capsule functions. This test is performed with a test capsule in place. The spacecraft will carry dummy solar arrays at this time.

Following the first integrated system test, the PTM will undergo an ethylene oxide exposure, followed by a system test to determine whether or not the system has been degraded. Following the system test, a series of special tests will be conducted. The first of these will be the power profile test to determine the power drain of the spacecraft in its various operating modes. This data is used for power allocation during flight. The next test, performed on the EM and PTM, is the failure mode and logic test, a detailed check of the on-board logic. A logic matrix will be

generated to verify that all combinations and permutations of the spacecraft logic are exercised. A system parameter variation test is then performed on all spacecraft followed by an electromagnetic interference test to determine system interference susceptibility and spacecraft system contribution to the command receiver noise level.

The spacecraft are taken to the Capistrano Magnetic Test Facility, where the spacecraft magnetic fields are mapped and magnetic stability is determined. Both dynamic and static fields will be measured. The magnetic fields of the flight capsule and the solar array will be mapped separately. The data will then be integrated to form a composite magnetic map. Each spacecraft will be returned and remapped following the vibration and thermal vacuum tests to determine whether or not the fields have changed due to testing and handling operations. Science interference and compatibility will be checked at this site since the electromagnetic noise level will be lower than at any other test facility. Upon return from the magnetic facility, the solar array will be installed, propellant and pneumatics tanks loaded with inert propellants and gas and mass property measurements made. Moments of inertia will be measured on the PTM only. Weight and center of gravity will be measured on all spacecraft. The spacecraft alignments will then be performed. All pneumatics and propellant systems will be leak checked and operational checks will be made of the appendage release system. (These last three tests will be repeated, following the vibration-acoustic environment test for flight spacecraft and following the shock test environment for the PTM, to determine any system degradation.)

Upon completion of these tests, the spacecraft will be prepared for the vibration-acoustic test. The spacecraft will be in flight configuration except for the thermal insulation, which may not be completely installed. Flight ordnance will be installed and armed for flight and a test capsule installed. All pneumatics systems will be pressurized to flight pressures. The propulsion subsystem tanks will be filled with inert propellants to simulate the most critical flight propellant loading condition. The spacecraft will be moved to the vibration-acoustic facility and an integrated system test performed to establish a pre-environmental baseline. A combined vibration-acoustic test will be performed with a hydraulic shaker

performing the low frequency vibration and a reverbent acoustic chamber generating the high frequency environment.

The PTM will undergo shock testing to verify that the firing of explosive devices and shroud jettison do not have an adverse effect on the spacecraft. A free mode test will then be performed with no electrical interfaces between the spacecraft and the EOSE. Mechanical interfaces with test stimulus will exist. The solar array will be illuminated and will power the spacecraft. This test has the dual purpose of serving as a post dynamic environment system test and as the solar array integration and compatibility test.

The spacecraft is then prepared for space simulation. The flight batteries will be installed and thermal insulation completed. Auxiliary heaters and special thermal vacuum test instrumentation will be installed. The test capsule will be installed and the spacecraft turned upside down for installation in the space chamber.

After installation of the spacecraft and test capsule in the chamber, an integrated system test will be performed to validate all EOSE and chamber cabling as well as to establish a system baseline. The thermal-vacuum exposure will be performed in two parts, the first part with the test capsule installed and the second part without the test capsule. Tests on the thermal model spacecraft and the PTM spacecraft may show that the effect of the capsule is negligible, in which case the first part of the test will be deleted for the flight models. Off-axis exposure to the solar simulator will be performed on the PTM spacecraft by rotating the spacecraft in its holding fixture. A system test will be performed before spacecraft removal from the chamber to determine any effect of the environmental exposure on system performance.

Prior to shipment, three system tests will be run, which require operating two spacecraft from one control point. The first is a simulated stack test, in which the spacecraft are placed as close to each other as possible, to determine whether or not interference problems exist in this configuration. The second test is a compatibility check with the launch complex equipment, which includes a practice countdown. The final test is a mission operations test. The mission dependent equipment and the

Space Flight Operational Facility, along with the STC, will perform a full system test for verification of readiness to ship the spacecraft to AFETR. The STC will be tied to the Space Flight Operational Facility via data links and communications nets.

5.3 System Test Schedule

Figure 40 is a schedule for each spacecraft. The tasks are listed in major test groupings. Facilities loading, hardware delivery dates, and manpower requirements are derived from this schedule. Figure 41 shows the sequence of operations for each spacecraft. The starting date for each task group is listed in Figure 40. The operations numbers listed in each box are defined in the detailed task descriptions of Appendix A.

6. LAUNCH OPERATIONS

This section discusses all tests and operations performed at the launch site. Included are operations performed to validate the STC, assembly, handling, and shipping equipment launch complex equipment and launch site facilities. Background information and launch philosophy are outlined, along with such topics as spares replacement and testing and schedule recovery from hardware failures. A sequence and description of operations for each spacecraft is outlined. The spacecraft involved are:

- Engineering model
- Propulsion interaction model
- First flight model
- Second flight model
- Third flight model

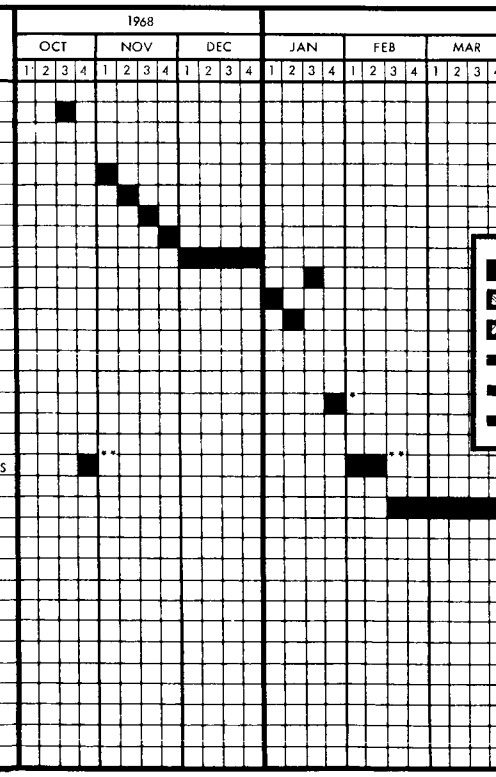
The schedule of operations and the flow charts define the launch site test program. The detailed description of tasks is included in Appendix A of this volume and is referenced to the schedule and flow chart by an operation number.

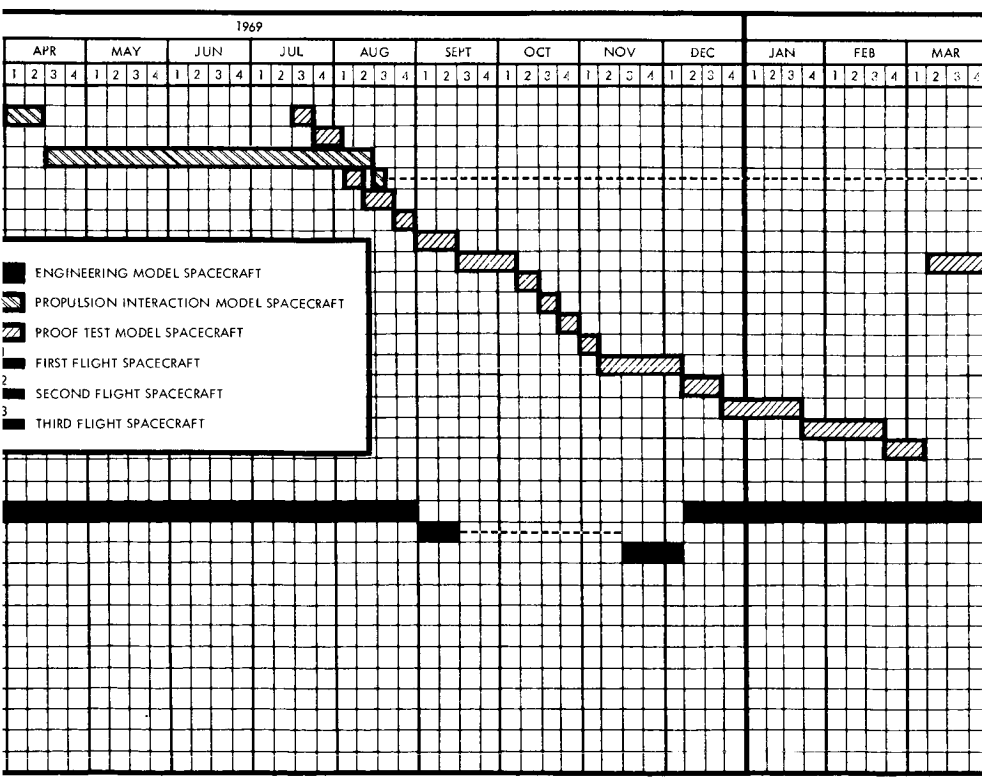
6.1 Planning Guidelines

Launch site operations will be supported by the same test conductors and crews used during factory assembly and test. The test crews will be

DETAIL FLOW OPERATIONS NUMBER	SPACECRAFT SYSTEMS TEST SCHEDULE
54	FIRST INTEGRATED SYSTEM TEST AND CRITIQUE
54a,b	ETHYLINE OXIDE COMPATIBILITY TEST AND 1ST
54c	PROPULSION INTERACTION TEST
55	POWER PROFILE TEST
57	FAILURE MODE LOGIC TEST
56	PARAMETRIC VARIATION TEST
58-62	EMI TESTS
63-69	MAGNETIC PROPERTIES TESTS
70-71	MASS PROPERTIES TEST
74,74a,75	ALIGNMENT TESTS
76,78-80	INSTALL SOLAR ARRAYS, LEAK, AND DEPLOYMENT TESTS
81,83	INSTALL FLIGHT ORDNANCES, 1ST AND CRITIQUE
84-87a	VIBRATION/ ACOUSTIC TESTS, INSTALL ON VERTICAL STAND
88-92	SHOCK TEST
93-98	FREE MODE, DEPLOYMENT, LEAK, AND ALIGNMENT TESTS
	1ST AND THERMAL VACUUM TESTS
105-107	INSTALL TEST CAPSULE, 1ST AND MOVE TO ASSEMBLY AREA
107a, 108, 109	SIMULATED STACK, PRACTICE COUNTDOWN AND MISSION OPERATION TESTS
109a, 110-113	DSIF COMPATIBILITY TESTS (GOLDSTONE)
76a	SPACECRAFT-VIBRATION FACILITY COMPATIBILITY CHECKS
99a	SPACECRAFT-THERMAL VACUUM FACILITY COMPATIBILITY CHECKS
119	RELAY LINK TEST
115, 116	SPACECRAFT-BOOSTER COMPATIBILITY TESTS
117, 118	LIFE TEST
114, 120	PREPARE AND SHIP SPACECRAFT TO CAPE

*FREE MODE TEST ONLY
**MISSION OPERATIONS TEST ONLY
***PRACTICE COUNTDOWN TEST ONLY




1969
APR
MAY
JUN
JUL
AUG
SEPT
OCT
NOV
DEC
JAN
FEB
MAR
1 2 3 4
ENGINEERING MODEL SPACECRAFT
PROPULSION INTERACTION MODEL SPACECRAFT
PROOF TEST MODEL SPACECRAFT
1 FIRST FLIGHT SPACECRAFT
2 SECOND FLIGHT SPACECRAFT
3 THIRD FLIGHT SPACECRAFT

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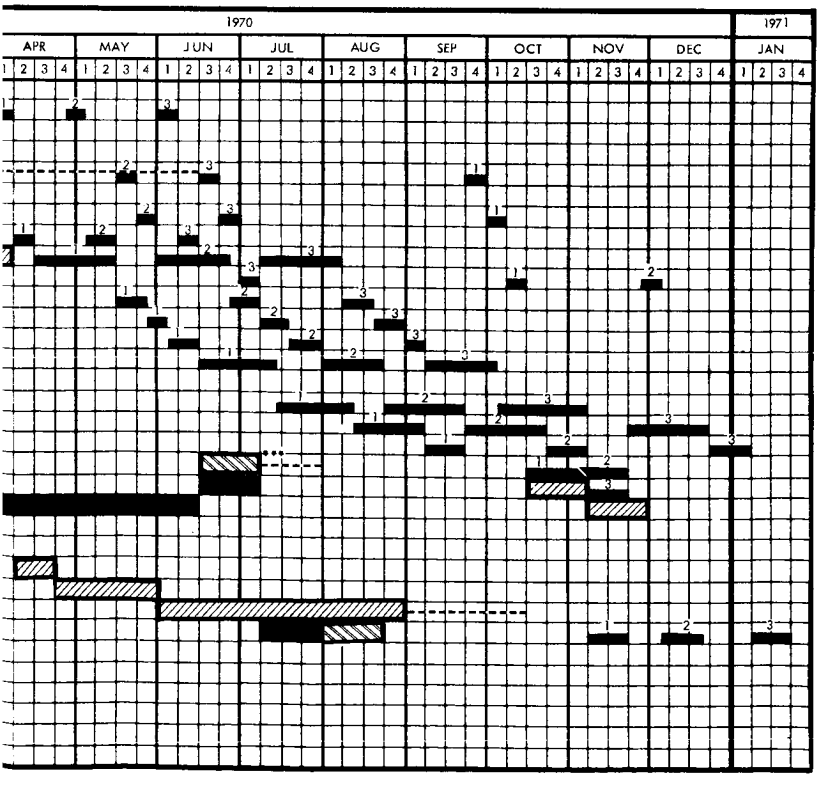
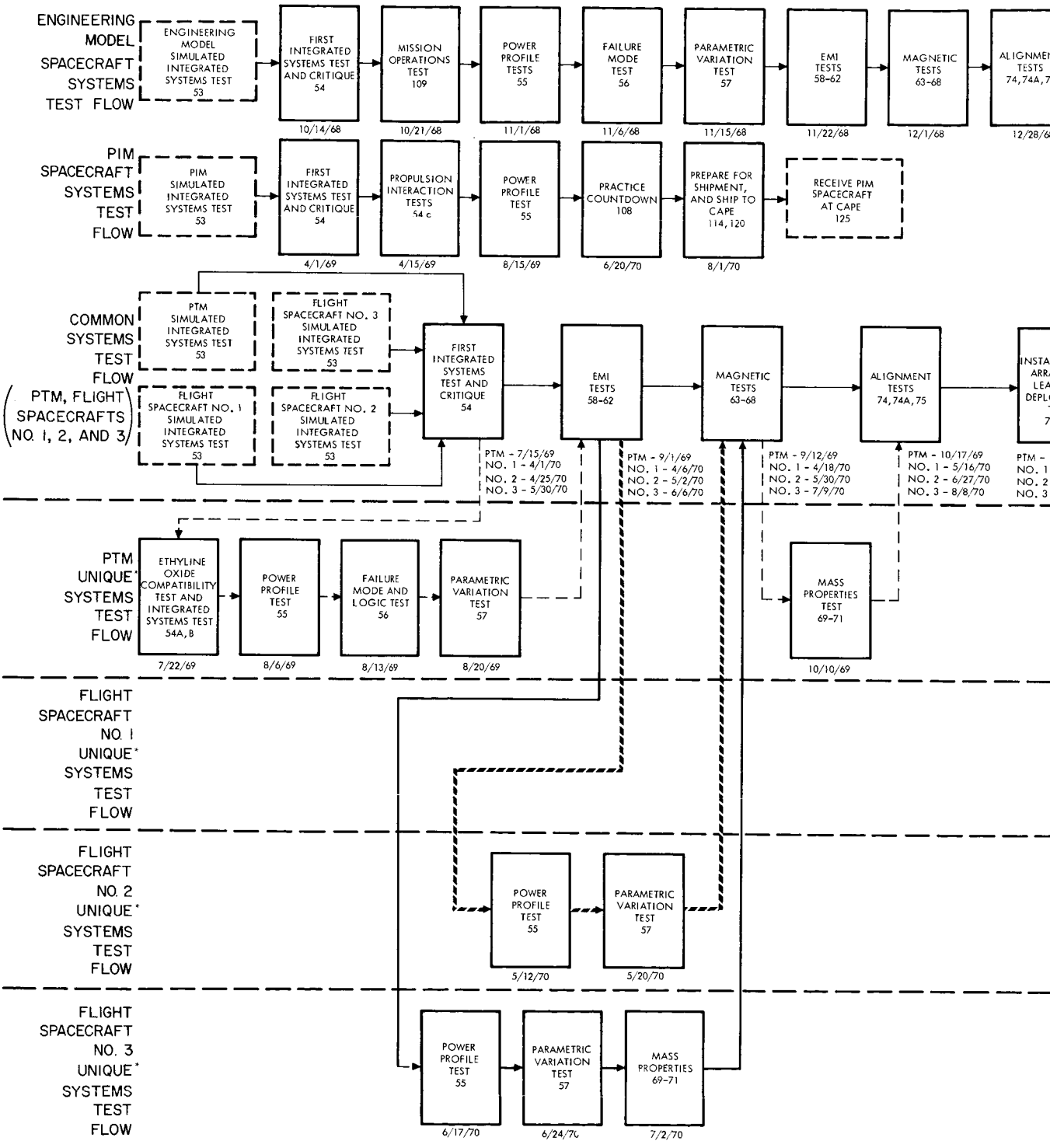
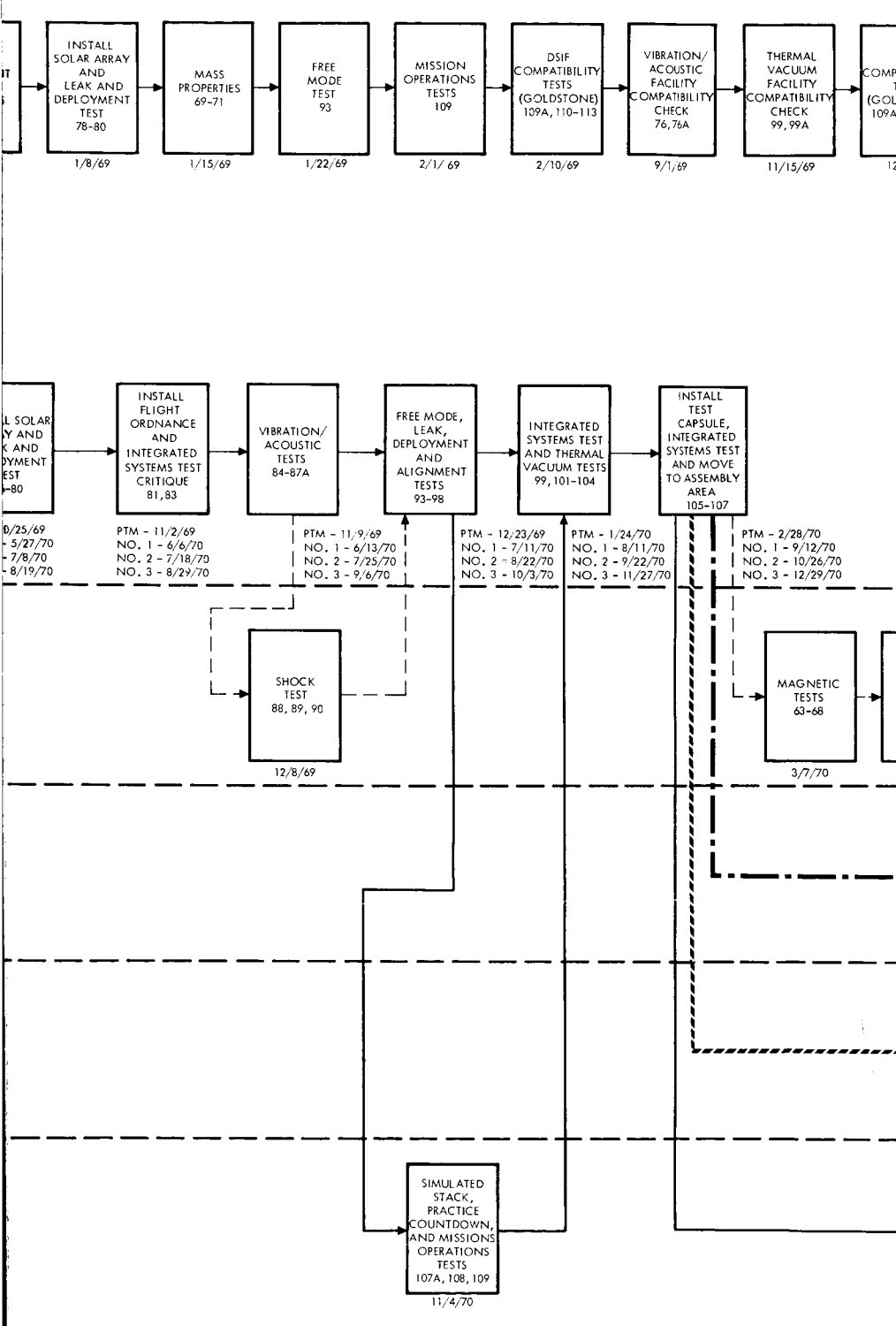


Figure 40. Spacecraft Systems Test Schedule


ENGINEERING MODEL SPACECRAFT SYSTEMS TEST FLOW
ENGINEERING MODEL SIMULATED INTEGRATED SYSTEMS TEST 53
FIRST INTEGRATED SYSTEMS TEST AND CRITIQUE 54
10/14/68
MISSION OPERATIONS TEST 109
10/21/68
POWER PROFILE TESTS 55
11/1/68
FAILURE MODE TEST 56
11/6/68
PARAMETRIC VARIATION TEST 57
11/15/68
EMI TESTS 58-62
11/22/68
MAGNETIC TESTS 63-68
12/1/68
PIM SPACECRAFT SYSTEMS TEST FLOW
PIM SIMULATED INTEGRATED SYSTEMS TEST 53
FIRST INTEGRATED SYSTEMS TEST AND CRITIQUE 54
4/1/69
PROPULSION INTERACTION TESTS 54.c
4/15/69
POWER PROFILE TEST 55
8/15/69
PRACTICE COUNTDOWN 108
6/20/70
PREPARE FOR SHIPMENT, AND SHIP TO CAPE 114, 120
8/1/70
RECEIVE PIM SPACECRAFT AT CAPE 125
COMMON SYSTEMS TEST FLOW
PTM, FLIGHT SPACECRAFTS NO. 1, 2, AND 3
PTM SIMULATED INTEGRATED SYSTEMS TEST 53
FLIGHT SPACECRAFT NO. 3 SIMULATED INTEGRATED SYSTEMS TEST 53
FLIGHT SPACECRAFT NO. 1 SIMULATED INTEGRATED SYSTEMS TEST 53
FLIGHT SPACECRAFT NO. 2 SIMULATED INTEGRATED SYSTEMS TEST 53
FIRST INTEGRATED SYSTEMS TEST AND CRITIQUE 54
EMI TESTS 58-62
MAGNETIC TESTS 63-68
ALIGNMENT TESTS 74,74A,75
PTM - 7/15/69
NO. 1 - 4/1/70
NO. 2 - 4/25/70
NO. 3 - 5/30/70
PTM - 9/1/69
NO. 1 - 4/6/70
NO. 2 - 5/2/70
NO. 3 - 6/6/70
PTM - 9/12/69
NO. 1 - 4/18/70
NO. 2 - 5/30/70
NO. 3 - 7/9/70
PTM - 10/17/69
NO. 1 - 5/16/70
NO. 2 - 6/27/70
NO. 3 - 8/8/70
PTM UNIQUE* SYSTEMS TEST FLOW
ETHYLINE OXIDE COMPATIBILITY TEST AND INTEGRATED SYSTEMS TEST 54A,B
7/22/69
POWER PROFILE TEST 55
8/6/69
FAILURE MODE AND LOGIC TEST 56
8/13/69
PARAMETRIC VARIATION TEST 57
8/20/69
MASS PROPERTIES TEST 69-71
10/10/69
FLIGHT SPACECRAFT NO. 1 UNIQUE* SYSTEMS TEST FLOW
FLIGHT SPACECRAFT NO. 2 UNIQUE* SYSTEMS TEST FLOW
POWER PROFILE TEST 55
5/12/70
PARAMETRIC VARIATION TEST 57
5/20/70
FLIGHT SPACECRAFT NO. 3 UNIQUE* SYSTEMS TEST FLOW
POWER PROFILE TEST 55
6/17/70
PARAMETRIC VARIATION TEST 57
6/24/70
MASS PROPERTIES 69-71
7/2/70


INSTALL SOLAR ARRAY AND LEAK AND DEPLOYMENT TEST 78-80
1/8/69
MASS PROPERTIES 69-71
1/15/69
FREE MODE TEST 93
1/22/69
MISSION OPERATIONS TESTS 109
2/1/69
DSIF COMPATIBILITY TESTS (GOLDSTONE) 109A, 110-113
2/10/69
VIBRATION/ ACOUSTIC FACILITY COMPATIBILITY CHECK 76,76A
9/1/69
THERMAL VACUUM FACILITY COMPATIBILITY CHECK 99, 99A
11/15/69
INSTALL FLIGHT ORDNANCE AND INTEGRATED SYSTEMS TEST CRITIQUE 81,83
PTM - 11/2/69
NO. 1 - 6/6/70
NO. 2 - 7/18/70
NO. 3 - 8/29/70
VIBRATION/ ACOUSTIC TESTS 84-87A
PTM - 11/9/69
NO. 1 - 6/13/70
NO. 2 - 7/25/70
NO. 3 - 9/6/70
FREE MODE, LEAK, DEPLOYMENT AND ALIGNMENT TESTS 93-98
PTM - 12/23/69
NO. 1 - 7/11/70
NO. 2 - 8/22/70
NO. 3 - 10/3/70
INTEGRATED SYSTEMS TEST AND THERMAL VACUUM TESTS 99, 101-104
PTM - 1/24/70
NO. 1 - 8/11/70
NO. 2 - 9/22/70
NO. 3 - 11/27/70
INSTALL TEST CAPSULE, INTEGRATED SYSTEMS TEST AND MOVE TO ASSEMBLY AREA 105-107
PTM - 2/28/70
NO. 1 - 9/12/70
NO. 2 - 10/26/70
NO. 3 - 12/29/70
SHOCK TEST 88, 89, 90
12/8/69
MAGNETIC TESTS 63-68
3/7/70
SIMULATED STACK, PRACTICE COUNTDOWN, AND MISSIONS OPERATIONS TESTS 107A, 108, 109
11/4/70

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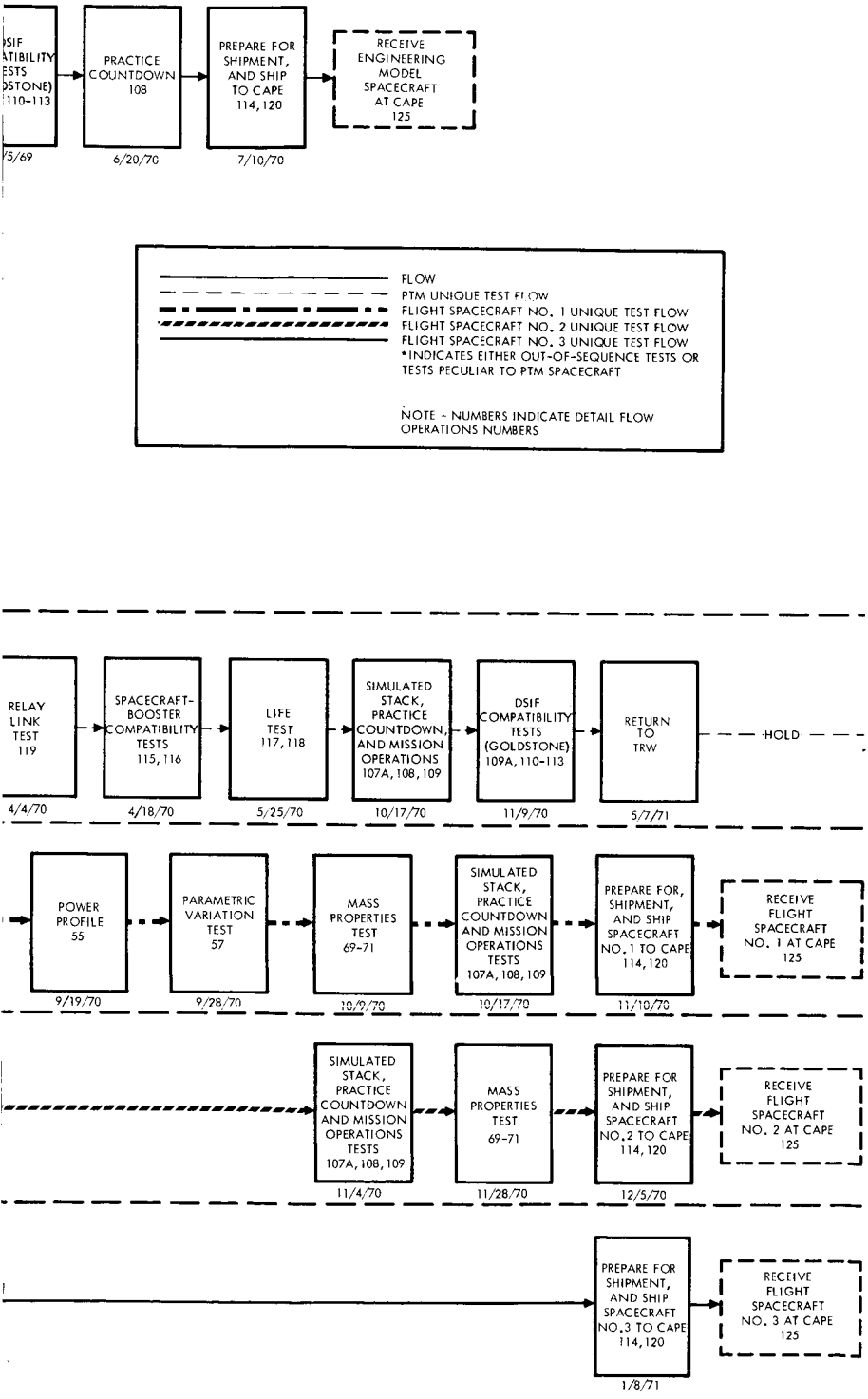


Figure 41. Spacecraft Systems Test Flow

integrated as part of the total launch operations crew along with other supporting contractor personnel. Operations planning and scheduling will be supported by TRW test crew personnel to ensure a smooth and integrated operation. TRW test personnel will be supported by resident TRW Kennedy Space Center (KSC) support personnel.

All spacecraft will arrive at AFETR assembled in a flight configuration. The launch flow assumes that the flight capsules will be tested, with propellants and ordnance loaded, in the vertical assembly building (VAB). If KSC safety does not permit this operation, VAB testing must be conducted using test capsules, in which case pad checkout time will increase since this now becomes the first place where the flight shroud, containing the flight planetary vehicles, is integrated and thus pad verification becomes a more detailed checkout operation.

The capability to load propellants in the explosive safe area (ESA) and KSC safety authorization to transport the planetary vehicles to the pad with propellants loaded and ordnance installed are required. These constraints are required in order to sterilize the planetary vehicles and seal the shroud prior to pad operations. No capability for propellant loading or unloading is provided for at the pad.

In the event of a minor failure on the flight spacecraft during launch operations, replacement parts are available down to the provisional spares level. A failed part in either the first or second flight spacecraft is replaced by duplicate hardware from the third flight spacecraft. The replacement parts brought to the launch site will be integrated and tested in the third flight spacecraft.

If a black box failure occurs on the launch pad, the individual planetary vehicle will be removed from the launch vehicle and moved back into the ESA. Here the shroud is unsealed, ordnance removed, and propellants unloaded. The failed unit is removed and the new unit from the third spacecraft is installed. A spacecraft functional test will be performed to verify the integrity of the unit. Ordnance and propellants are reloaded and leak tests conducted. The planetary vehicle is now sterilized and sealed before movement out to the pad. At the pad, the planetary vehicle will be reintegrated with the launch vehicle. Launch pad operations can then resume.

Turn around times for different types of failures at various schedule points will be studied and defined in Phase IB.

The spacecraft is designed to function independently of the flight capsule. If necessary a dummy capsule or ballast can be mated to the spacecraft and flown. It is also realized that some emergency could arise during the mission, prior to normal capsule separation, of such a nature that in order not to jeopardize the mission it would be required to jettison the flight capsule. In this case the spacecraft is capable of proceeding with the mission. If only one planetary vehicle or spacecraft is available for launch, ballast or engineering models may be substituted for the missing entities. Due to the short launch window, a partial mission success is more desirable than no mission at all.

6.2 Launch Site Sequence

Two STC's will be delivered to the launch site before the arrival of the EM spacecraft. Installation, checkout, and preliminary validation of the STC's will be completed prior to arrival of the engineering model spacecraft. An additional STC, required for the third flight spacecraft, will be received and validated before the first flight spacecraft arrives at the site. Testing with the EM spacecraft will complete the validation of the STC's. The three STC's are to be located in an annex of the ESA. All OSE necessary at the spacecraft will be located in the test area within the facility. The launch complex equipment will be located in the launch control center and launch umbilical tower.

The EM spacecraft and the PIM spacecraft are planned for arrival at KSC before the three flight spacecraft. These two spacecraft are used to debug assembly, handling and shipping equipment, validate OSE and to check out procedures, schedules and facilities at KSC. They provide the required launch operations training for key personnel from the flight spacecraft test crews. The shroud-launch vehicle interfaces will be checked out prior to arrival of a flight spacecraft.

The EM spacecraft arrives at the Cape Kennedy skid strip and is transported to the spacecraft assembly facility for inspection and checkout. During its stay in the spacecraft assembly facility, the EM spacecraft will

undergo receiving inspection and functional spacecraft tests. These checks will verify that the spacecraft has not been damaged during shipment. Load data taken by special shipping instrumentation will be analyzed for verification of the acceptability of the shipping operations. All OSE and facilities equipment in the spacecraft assembly facility will be validated with the EM spacecraft. An integrated system test will be run with the test capsule installed. At the completion of the integrated systems test, the EM is moved to the ESA.

In the ESA, alignment verification and weight and center of gravity checks are made with the engineering model to validate the required OSE. The spacecraft will then undergo separation and release tests followed by installation of ordnance simulators. The EM with its test capsule is then mated to a shroud section.

The PIM, upon arrival, will undergo the same checkout as the EM in the spacecraft assembly facility and ESA. When the PIM checkout is complete, the simulated planetary vehicles will be mated to their sections of the flight shroud. The flight shroud is composed of three sections, one for each planetary vehicle and a nose cap. Each planetary vehicle shroud section has a sterilization seal on the top and on the bottom.

A shroud cooling test will be performed to verify the ability of the launch vehicle cooling system to maintain a satisfactory operating environment for the spacecraft and to determine the required cooling system flow rates and temperature settings. This test will be performed with a GFE launch vehicle cooling system simulator. A modified integrated system test/compatibility test followed by a mock countdown will then be performed on each vehicle. These operations will validate the ESA and all OSE installed there. At the completion of this series of tests, the planetary vehicles, encapsulated in their respective shroud sections will be moved on a modified Saturn-IVB transporter to the VAB.

In the VAB, the shroud encapsulated planetary vehicles will be mated with the launch vehicle facility verification missile for mechanical

and electrical compatibility tests. The mechanical checks are the performance of the mating and shroud-launch vehicle alignment tasks. RF interference, mock flight readiness demonstration, mock countdown, integrated systems test and simulated flight tests will be performed. The Eastern Test Range and Deep Space Instrumentation Facility 71 will support these tests, as required, to verify their readiness to support the flight spacecraft. The shroud sections will then be brought back to the ESA for a practice of the final flight preparation operations.

All spacecraft pneumatics systems will be pressurized. Dummy ordnance will be installed and an inert propellant loading-unloading operation performed. Leak tests will be performed in conjunction with propellant loading operations. These operations will provide training for personnel and validation of equipment. An integrated system test will then be performed, followed by a sterilization (simulated) and encapsulation operation. The encapsulated vehicles are then moved to the launch pad.

At the launch pad, the shroud sections are mated with the launch vehicle facility verification missile. As a final validation the following series of practice tests will be run: RF interference, J-FACT, preflight readiness demonstration, mock countdown, and simulated flight. The Eastern Test Range and Deep Space Instrumentation Facility 71 will support these operations as required. At the completion of these tests, the simulated planetary vehicles are removed from the launch vehicle simulator and returned to the ESA. The shroud sections will be unsealed and spacecraft and test capsules removed. The spacecraft will remain in the ESA to assist as necessary in analyzing flight spacecraft problems.

The three flight spacecraft arrive at the launch site phased so that one spacecraft at a time is received and checked out in the spacecraft assembly facility. The testing sequence for the flight spacecraft is the same as that for the EM spacecraft in the spacecraft assembly facility.

Alignment and weight and center of gravity checks are made to each spacecraft in the ESA. The spacecraft will then undergo separation and appendage release tests. Ordnance simulators for use during the joint compatibility checks will be installed prior to the mating of the flight capsule to the spacecraft. Capsule compatibility checks, integrated

system test, and science-quiet tests are performed with the flight planetary vehicles. The science-quiet test will verify satisfactory operation of the science instruments with the spacecraft quiescent. The first two flight planetary vehicles are mated to their shroud sections at this point in their sequence. The launch preparation sequence for the third planetary vehicle ends at this point. It will be kept in the ESA and used as a backup. The third spacecraft will be kept in operating condition at all times. If a malfunctioning unit is found in either of the first two flight spacecraft, it will be replaced with a duplicate from the third spacecraft. The third flight spacecraft will then receive a spare part from stores and the part will be integrated into the system and completely tested. In case of a catastrophic failure of either of the prime flight spacecraft, spacecraft number three will become prime.

Each of the prime planetary vehicles, mated to its flight shroud section, is transported to the VAB for combined compatibility tests. The planetary vehicles are mated to the launch vehicle and aligned. The vertical assembly building checkout will include RF interference, flight readiness demonstration, integrated system test, mock countdown and simulated flight tests. At the conculsion of this test series, the planetary vehicles are removed from the launch vehicle one at a time and returned separately to the ESA on the modified Saturn S-IVB transporter.

All spacecraft pneumatics systems will be pressurized for flight. Flight ordnance and propellant loading operations are performed on each spacecraft followed by leak tests and an integrated system test. The sterilization and encapsulation operation is performed on each planetary vehicle. The vehicles are then transported to the launch pad.

On the launch pad, the planetary vehicles are mated to the launch vehicle and the flight shroud is aligned to the launch vehicle. RF interference, J-FACT, flight readiness demonstration, and mock countdown tests are performed prior to a precountdown operation. At the completion of the precountdown, launch control will initiate terminal countdown and liftoff.

6.3 Launch Site Operations Schedules

The operations schedule for each spacecraft is detailed in Figure 42. Tasks are listed by major test groupings. Figure 43 shows the sequence of operations for each spacecraft. The operations numbers, listed in each box, are defined in the detailed task description of Appendix A of this volume.

A gross pictorial representation of the launch site operations is shown in Figure 44. Operations sequence and equipment usage are shown for flight spacecraft.

7. MISSION SUPPORT OPERATIONS

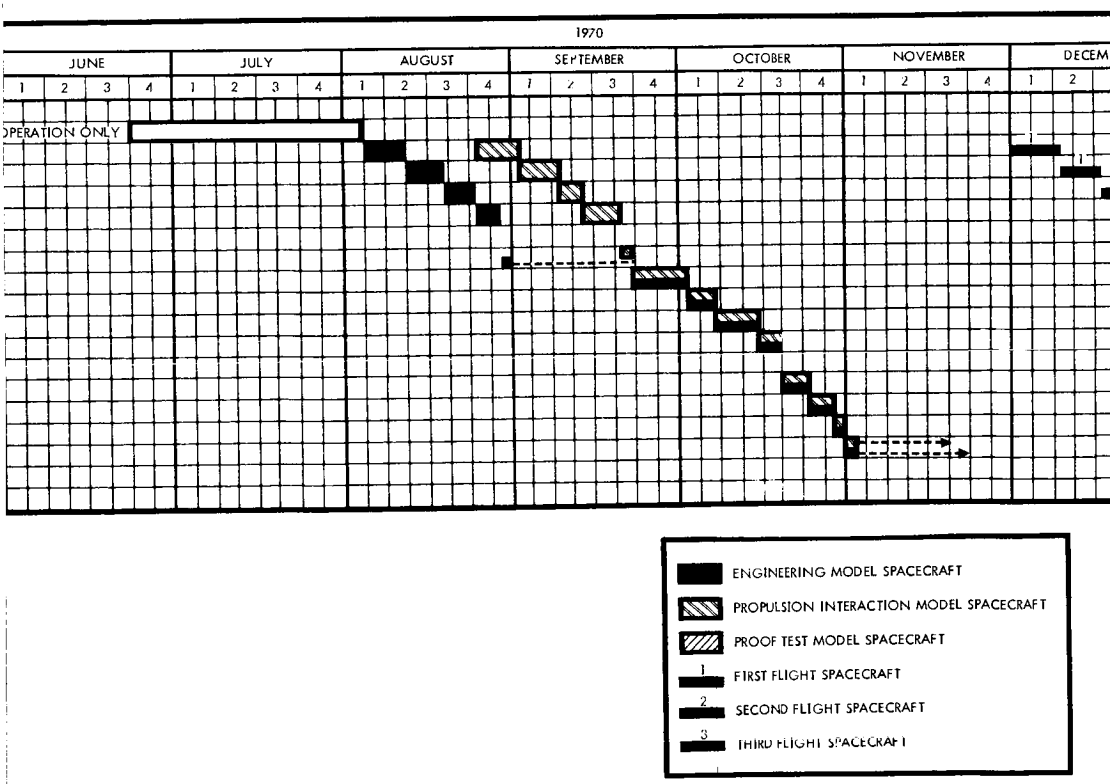
This section outlines the tasks to be performed by TRW to provide assistance to JPL mission operations. Tasks from preliminary planning to post mission evaluation are defined. It is the intention to concentrate on the interface checkout sequence of mission support operations that is required to prove DSN-spacecraft compatibility. Figure 45 shows the flow necessary to verify DSN spacecraft compatibility.

7.1 Operations

The development of software and mission dependent equipment will be completed prior to the assembly and checkout of the EM spacecraft. Software debugging and validation make use of a JPL-supplied DSIF-type computer, a set of mission dependent equipment and the EM spacecraft and STC. At the completion of theEM system test sequence, compatibility tests are conducted with the JPL computer. The JPL computer will control the spacecraft in various operational modes. In this configuration, an integrated system test is performed as acceptance of the DSIF-mission dependent equipment interface. All equipment and associated documentation used in the sell-off test will be shipped to Goldstone with the EM spacecraft for on-site compatibility testing.

At Goldstone the equipment is inspected, installed, and validated prior to checkout with the facility. The compatibility testing at Goldstone will use the DSIF antennas for command transmission to, and telemetry reception from, the EM spacecraft. Command transmitter power will be attenuated to simulate expected flight signal strength at the spacecraft.

	LAUNCH SITE OPERATIONS SCHEDULE
121-214	RECEIVE INSPECT AND VALIDATE SYSTEM TEST COMPLEX OSE
125-130	RECEIVE, TRANSPORT TO SAF AND INSPECT SPACECRAFT, INSTALL TEST CAPSULE AND FUNCTIONALLY TEST
131-133	BATTERY CAPACITY AND SOLAR ARRAY TESTS, 1ST AND CRITIQUE
134-136	REMOVE TEST CAPSULE AND SHIP TO ESA
137-139a	VERIFY ALIGNMENT WEIGHT AND C. G. TEST, SEPARATION AND RELEASE TEST, AND INSTALL ORDNANCE SIMULATORS
	INSTALL FLIGHT CAPSULE, COMPATABILITY TEST, 1ST AND CRITIQUE AND SCIENCE QUIET TEST
140a, 144, 145	CAPSULE ALIGNMENT AND IN-HANGAR BUTTON-UP
146-150	MATE BOTH SPACECRAFT TO SHROUD, SHROUD COOLING SYSTEM TEST, MODIFIED 1ST AND MOCK COUNTDOWN
151-154	MOVE TO VAB, MATE TO BOOSTER, ALIGNMENT AND 1ST
155-159	RFI AND FLIGHT READINESS TESTS, MOCK COUNTDOWN, AND SIMULATED FLIGHT TEST
160-161	MOVE TO ESA AND FLIGHT PRESSURIZE ALL SYSTEMS
162	INSTALL FLIGHT ORDNANCE
163-167	LOAD PROPELLANTS, LEAK TEST, 1ST AND STERILIZE
168-173	MOVE TO PAD, MATE TO BOOSTER, ALIGN, 1ST, RFI AND J-FACT TEST
174-178	FLIGHT READINESS, MOCK COUNTDOWN, PRE-COUNTDOWN, AND TERMINAL COUNTDOWN
179-180	REMOVE PLANETARY VEHICLE FROM BOOSTER AND RETURN TO ESA


1970
JUNE
JULY
AUGUST
SEPTEMBER
OCTOBER
NOVEMBER
DECEM
1 2 3 4
OPERATION ONLY
ENGINEERING MODEL SPACECRAFT
PROPULSION INTERACTION MODEL SPACECRAFT
PROOF TEST MODEL SPACECRAFT
1 FIRST FLIGHT SPACECRAFT
2 SECOND FLIGHT SPACECRAFT
3 THIRD FLIGHT SPACECRAFT

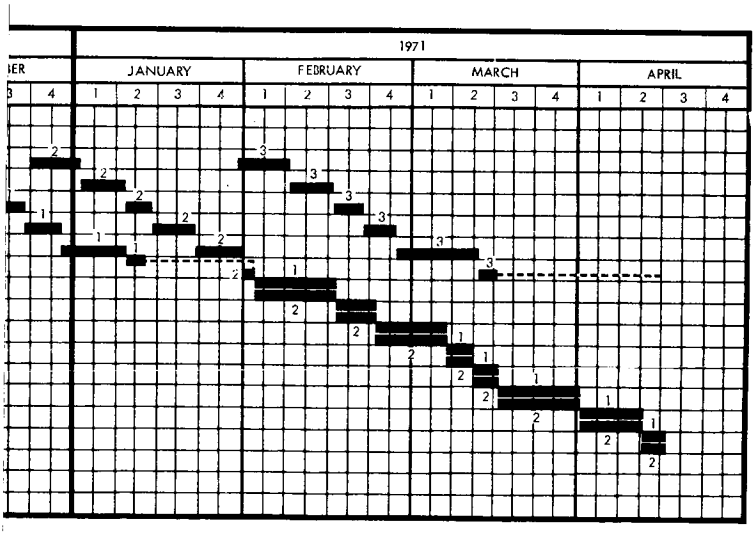
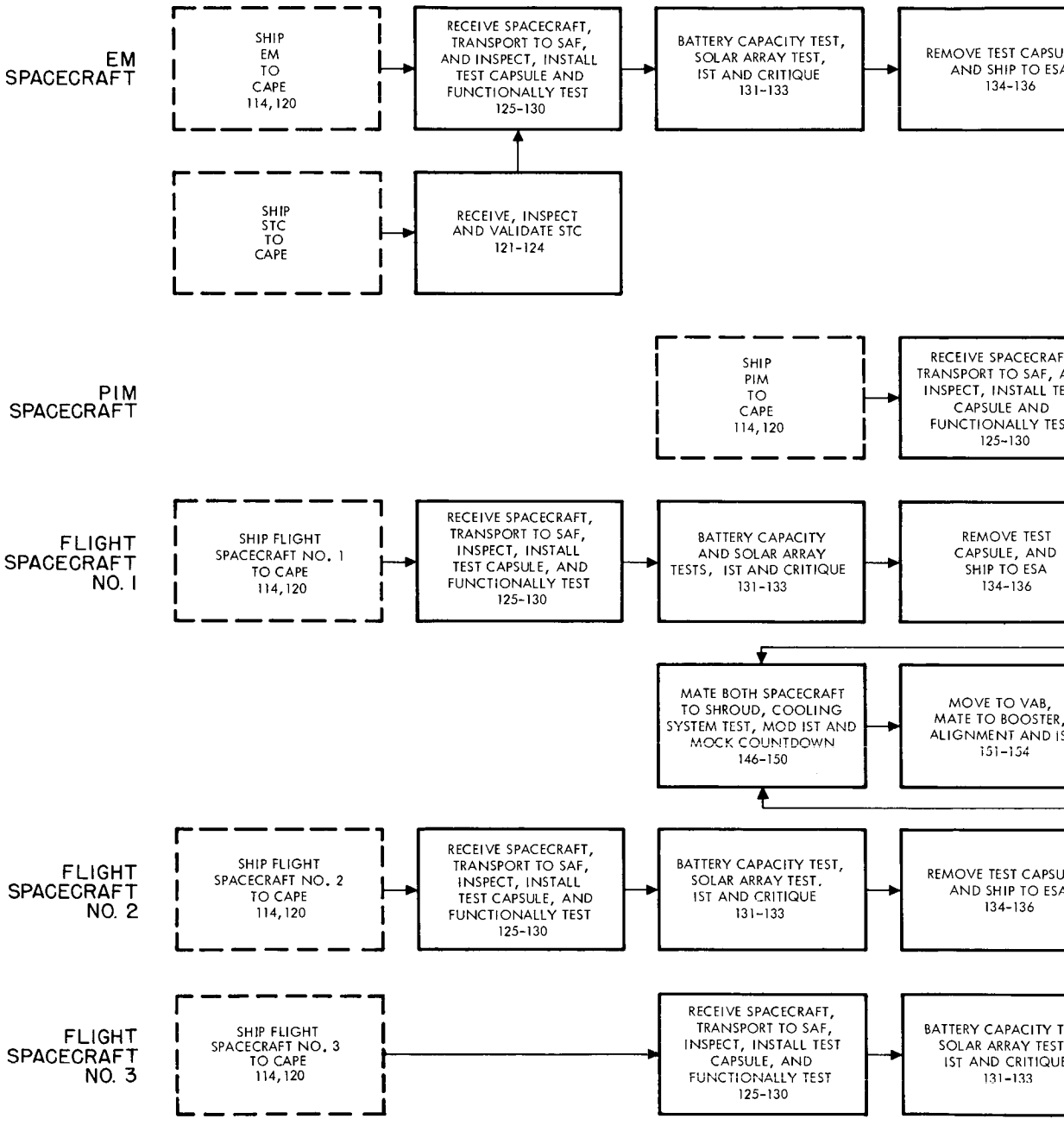
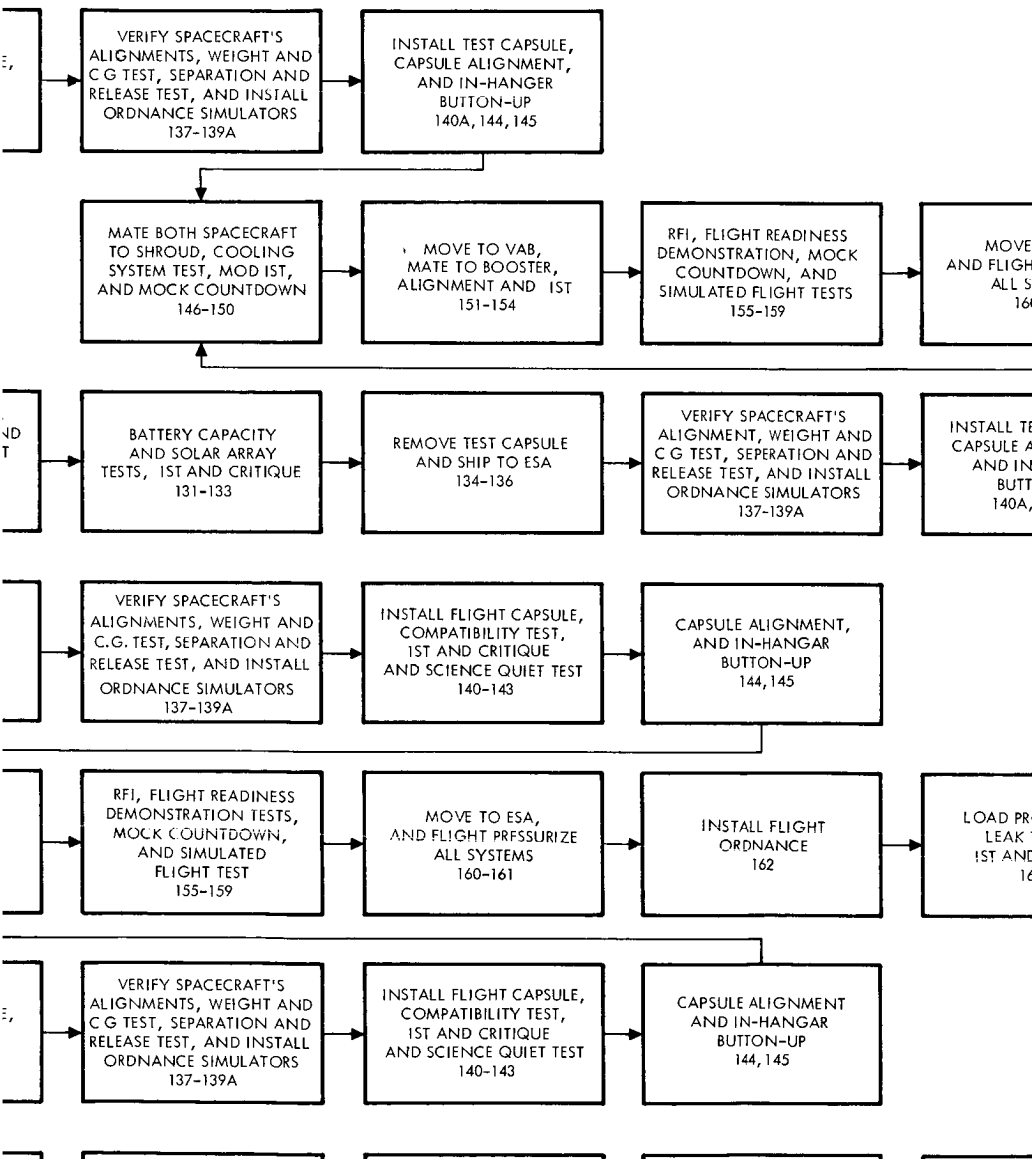


Figure 42. Launch Site Operations Schedule


EM SPACECRAFT
SHIP EM TO CAPE 114, 120
RECEIVE SPACECRAFT, TRANSPORT TO SAF, AND INSPECT, INSTALL TEST CAPSULE AND FUNCTIONALLY TEST 125-130
BATTERY CAPACITY TEST, SOLAR ARRAY TEST, IST AND CRITIQUE 131-133
REMOVE TEST CAPSU AND SHIP TO ESA 134-136
SHIP STC TO CAPE
RECEIVE, INSPECT AND VALIDATE STC 121-124
PIM SPACECRAFT
SHIP PIM TO CAPE 114, 120
RECEIVE SPACECRAF TRANSPORT TO SAF, A INSPECT, INSTALL TE CAPSULE AND FUNCTIONALLY TES 125-130
FLIGHT SPACECRAFT NO. 1
SHIP FLIGHT SPACECRAFT NO. 1 TO CAPE 114, 120
RECEIVE SPACECRAFT, TRANSPORT TO SAF, INSPECT, INSTALL TEST CAPSULE, AND FUNCTIONALLY TEST 125-130
BATTERY CAPACITY AND SOLAR ARRAY TESTS, IST AND CRITIQUE 131-133
REMOVE TEST CAPSULE, AND SHIP TO ESA 134-136
MATE BOTH SPACECRAFT TO SHROUD, COOLING SYSTEM TEST, MOD IST AND MOCK COUNTDOWN 146-150
MOVE TO VAB, MATE TO BOOSTER, ALIGNMENT AND IS 151-154
FLIGHT SPACECRAFT NO. 2
SHIP FLIGHT SPACECRAFT NO. 2 TO CAPE 114, 120
RECEIVE SPACECRAFT, TRANSPORT TO SAF, INSPECT, INSTALL TEST CAPSULE, AND FUNCTIONALLY TEST 125-130
BATTERY CAPACITY TEST, SOLAR ARRAY TEST, IST AND CRITIQUE 131-133
REMOVE TEST CAPSU AND SHIP TO ESA 134-136
FLIGHT SPACECRAFT NO. 3
SHIP FLIGHT SPACECRAFT NO. 3 TO CAPE 114, 120
RECEIVE SPACECRAFT, TRANSPORT TO SAF, INSPECT, INSTALL TEST CAPSULE, AND FUNCTIONALLY TEST 125-130
BATTERY CAPACITY T SOLAR ARRAY TEST IST AND CRITIQUE 131-133


VERIFY SPACECRAFT'S ALIGNMENTS, WEIGHT AND C G TEST, SEPARATION AND RELEASE TEST, AND INSTALL ORDNANCE SIMULATORS 137-139A
INSTALL TEST CAPSULE, CAPSULE ALIGNMENT, AND IN-HANGER BUTTON-UP 140A, 144, 145
MATE BOTH SPACECRAFT TO SHROUD, COOLING SYSTEM TEST, MOD IST, AND MOCK COUNTDOWN 146-150
MOVE TO VAB, MATE TO BOOSTER, ALIGNMENT AND IST 151-154
RFI, FLIGHT READINESS DEMONSTRATION, MOCK COUNTDOWN, AND SIMULATED FLIGHT TESTS 155-159
BATTERY CAPACITY AND SOLAR ARRAY TESTS, IST AND CRITIQUE 131-133
REMOVE TEST CAPSULE AND SHIP TO ESA 134-136
VERIFY SPACECRAFT'S ALIGNMENT, WEIGHT AND C G TEST, SEPERATION AND RELEASE TEST, AND INSTALL ORDNANCE SIMULATORS 137-139A
VERIFY SPACECRAFT'S ALIGNMENTS, WEIGHT AND C.G. TEST, SEPARATION AND RELEASE TEST, AND INSTALL ORDNANCE SIMULATORS 137-139A
INSTALL FLIGHT CAPSULE, COMPATIBILITY TEST, IST AND CRITIQUE AND SCIENCE QUIET TEST 140-143
CAPSULE ALIGNMENT, AND IN-HANGAR BUTTON-UP 144, 145
RFI, FLIGHT READINESS DEMONSTRATION TESTS, MOCK COUNTDOWN, AND SIMULATED FLIGHT TEST 155-159
MOVE TO ESA, AND FLIGHT PRESSURIZE ALL SYSTEMS 160-161
INSTALL FLIGHT ORDNANCE 162
VERIFY SPACECRAFT'S ALIGNMENTS, WEIGHT AND C G TEST, SEPARATION AND RELEASE TEST, AND INSTALL ORDNANCE SIMULATORS 137-139A
INSTALL FLIGHT CAPSULE, COMPATIBILITY TEST, IST AND CRITIQUE AND SCIENCE QUIET TEST 140-143
CAPSULE ALIGNMENT AND IN-HANGAR BUTTON-UP 144, 145
REMOVE TEST CAPSULE AND SHIP TO ESA 134-136
VERIFY SPACECRAFT'S ALIGNMENTS, WEIGHT AND C G TEST, SEPARATION AND RELEASE TEST AND INSTALL ORDNANCE SIMULATORS 137-139A
INSTALL FLIGHT CAPSULE, COMPATIBILITY TEST, IST AND CRITIQUE AND SCIENCE QUIET TEST 140-143

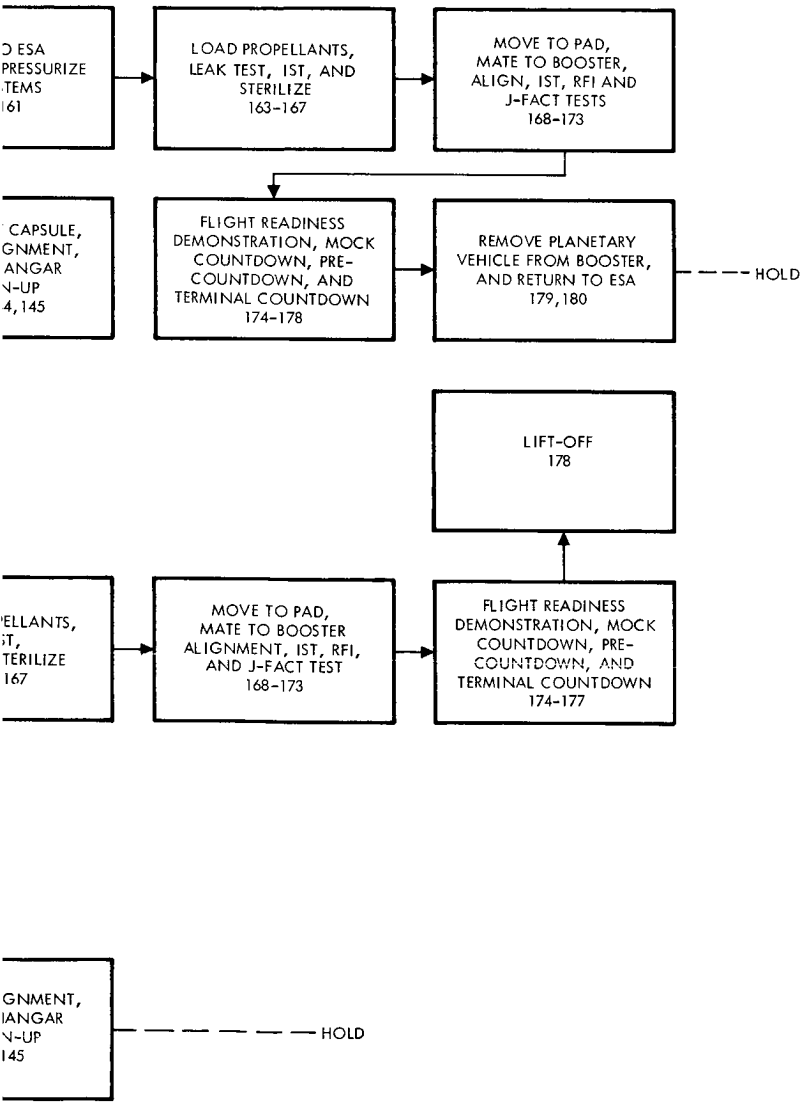
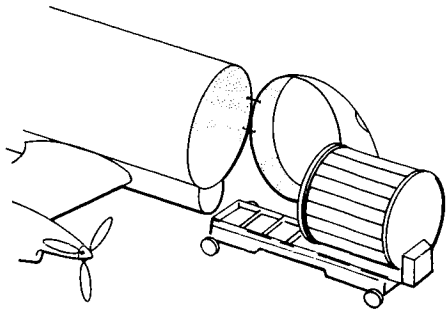
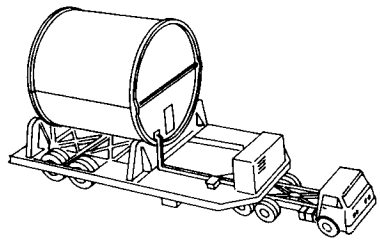


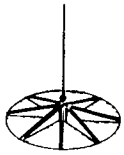
Figure 43. Launch Site Operations Flight Spacecraft Flow



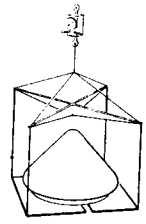
RECEIVE SPACECRAFT AND TEST CAPSULE AT KSC AIR STRIP

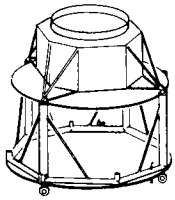


TRANSPORT SPACECRAFT TO SAF

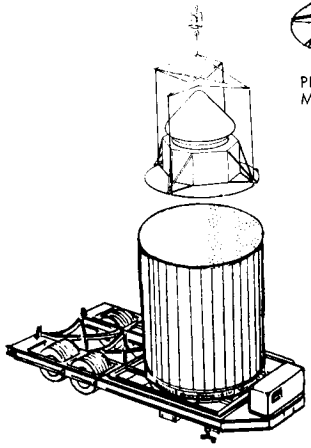


PREPARE PLANETARY VEHICLE FOR MOVEMENT OUTSIDE OF ESA

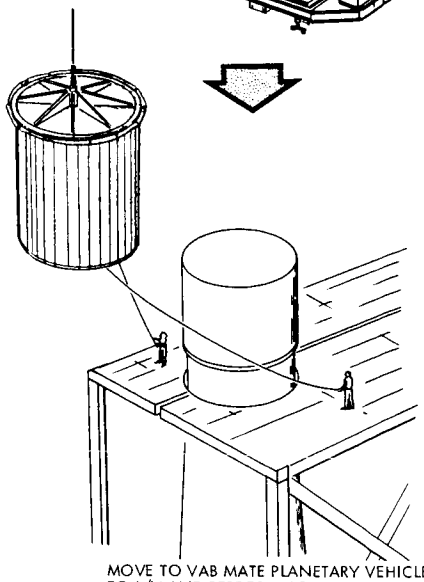




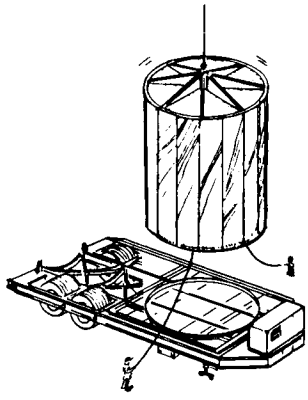
* INSTALL FLIGHT CAPSULE
* PERFORM FINAL IN-HANGER BUTTONUP OPS



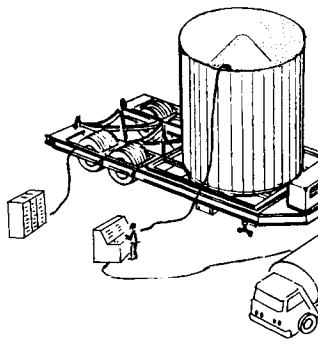
MATE SPACECRAFT TO FLIGHT SHROUD



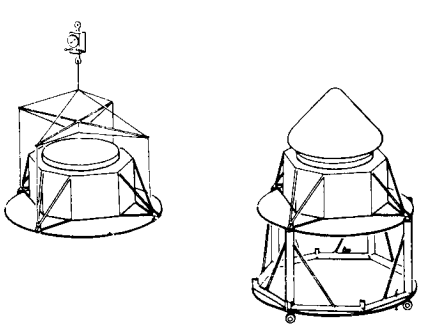
MOVE TO VAB MATE PLANETARY VEHICLES TO L/V AND PERFORM COMPATIBILITY TESTS



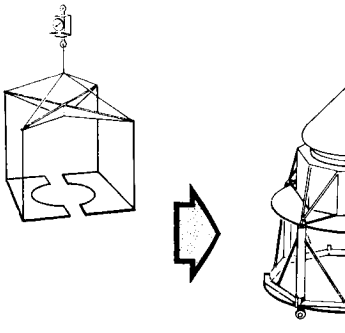
PREPARE PLANETARY VEHICLE FOR MOVEMENT TO ESA



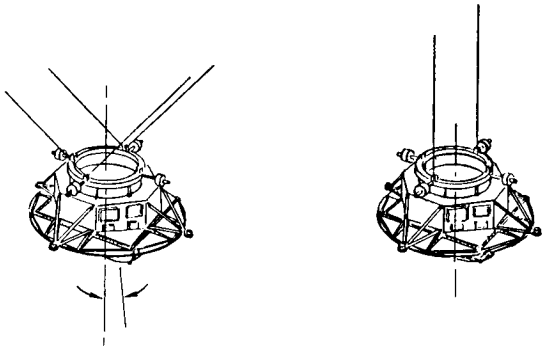
LOAD SPACECRAFT PROPELLA



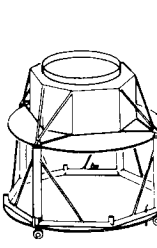
INSTALL TEST CAPSULE



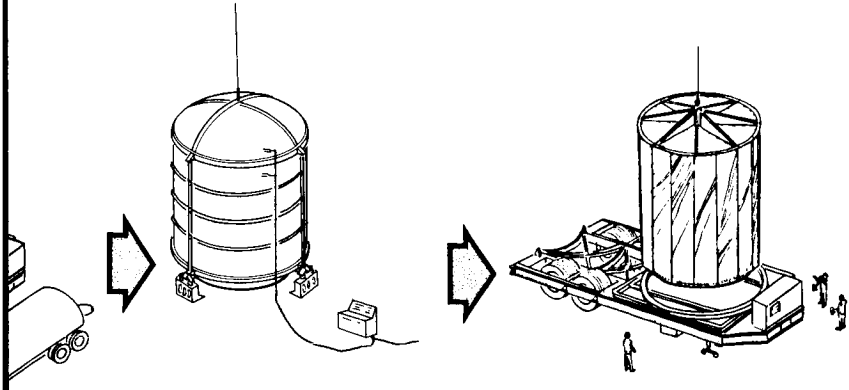
PERFORM F
SYSTEM TE



PERFORM WEIGHT AND
C AND G CHECKS



VERIFY SCS ALIGN



ITS

PERFORM PLANETARY
VEHICLE STERILIZATION

MOVE PLANETARY
VEHICLE TO PAD

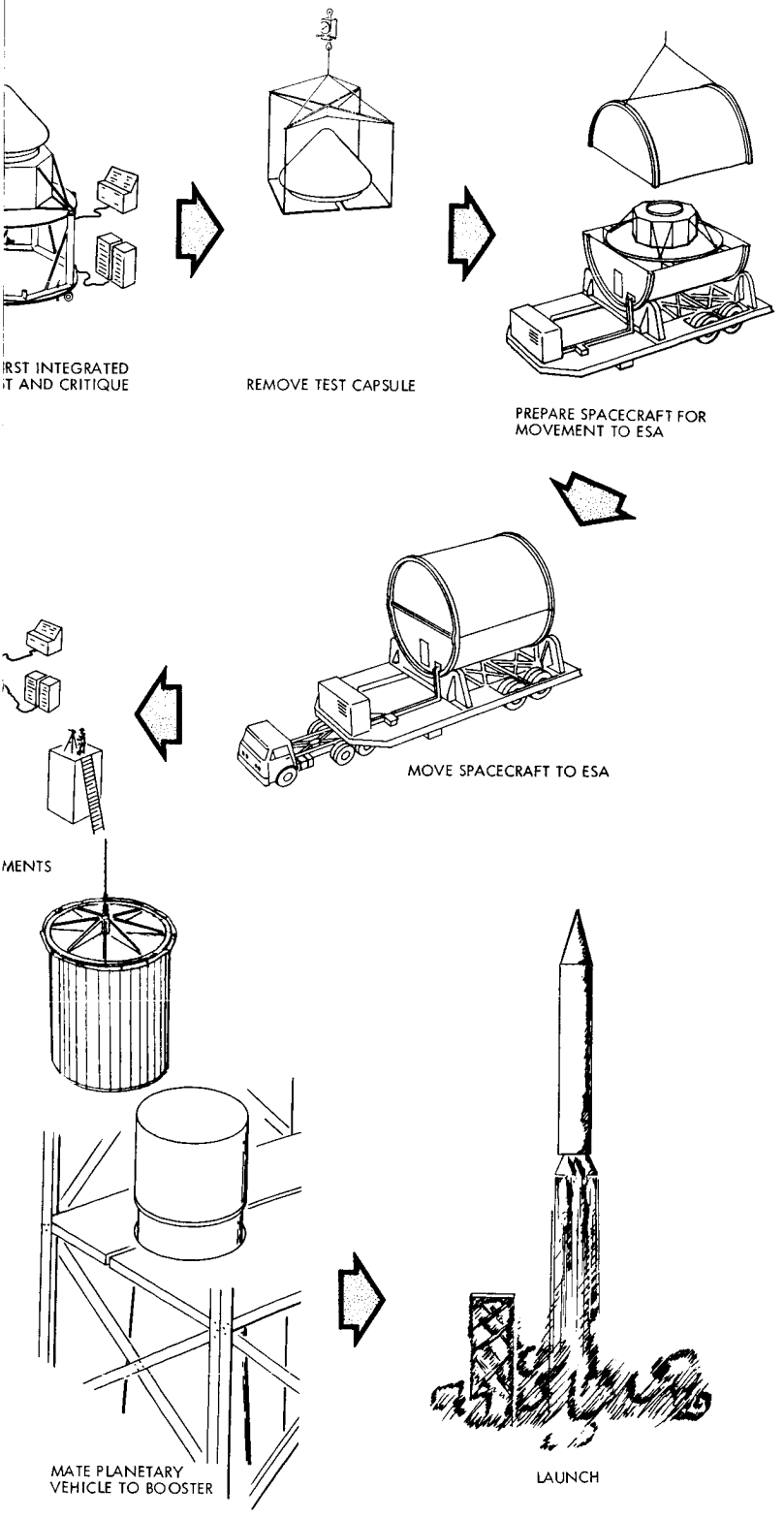


Figure 44. Launch Site Operations

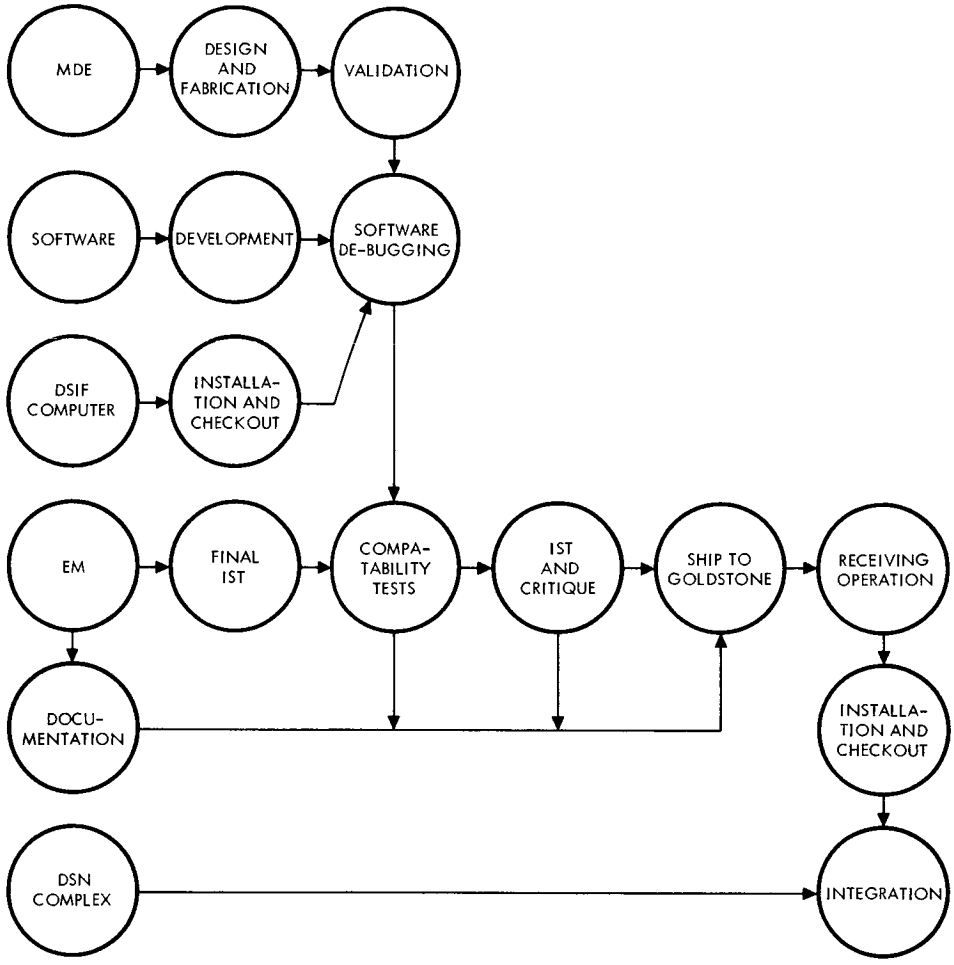


Figure 45. Mission Support Operations Compatibility Flow

The station telemetry receiver input will be attenuated and noise injected to simulate flight conditions. This series of tests using the capabilities of the DSN in conjunction with the EM spacecraft is scheduled to begin early in 1969. Simulated flight, midcourse maneuvers and Mars encounter simulation tests will be performed. It is assumed that the SFOF facility in Pasadena will be monitoring these operations with operational computer programs. The engineering model spacecraft will be returned to TRW for short periods of time for additional tasks. During those periods, compatibility testing at Goldstone will be conducted using magnetic tapes to simulate the spacecraft.

It is proposed that during the mission operations tests selected DSN personnel be brought to Goldstone for training. Classes would be

conducted in spacecraft system, subsystem, and mission operations. The training would include working with the EM during spacecraft tests as well as actually conducting interface checks between the DSIF and the mission dependent equipment.

A flyby will be made over individual DSIF's to verify RF transmission between the stations and the spacecraft. The aircraft used will contain equipment capable of sending a series of spacecraft signals to the ground. It will also be capable of receiving and identifying spacecraft commands. A status display available in both the ground station and the aircraft enables the respective operators to determine whether transmission was received correctly.

The EM spacecraft will leave Goldstone in mid-1970 for launch site validation. The PTM will arrive at Goldstone prior to launch for additional compatibility testing. The spacecraft will remain there through launch.

7.2 Additional Tasks

Listed below are some additional tasks in support of mission operations that will be performed by Voyager project personnel:

- Compile and revise as necessary all existing data pertinent to the understanding of the operational characteristics of the spacecraft, capsule, and all experiments. The documents will be subdivided into the various subsystems to facilitate use by subsystem personnel.

- Provide assistance in the definition and implementation of the communications network between the SFOF and the DSN stations, as well as secondary tracking stations in the STADAN network.

- Define the engineering and experiment computer program requirements to be used by JPL for both quick-look and long-term data processing, including a definition of the expceted and out-of-tolerance limits on spacecraft and experiment telemetry items.

- Define the real-time telemetry and communications requirements for the DSIF complex during the critical postlaunch, midcourse and orbit injection maneuvers.

- Define and coordinate data tape runs from the flight planetary vehicles during the final test phases. These tapes will include a simulation of anticipated inflight maneuvers as well as various

spacecraft and experiment modes. The tapes will be used by DSN stations for practice mission operations and to perform station validation.

- Generate detailed calibration data for all engineering telemetry items and those telemetry items considered crucial for the success of individual experiments. This data will be placed on calibration cards to be used with quick-look data displays.
- Generate a mission plan for each specific planetary vehicle which defines the operational requirements on the program.
- The Voyager project office will staff a spacecraft performance analysis team and supply personnel to the flight path analysis team for the 1971 mission.
- Install mission dependent equipment and validate associated software at the DSN stations.

8. ASSEMBLY AND TEST IMPLEMENTATION PLANNING

The Phase IB tasks will consist of updating and performing a detailed analysis of the plans prepared in Phase IA. Coordination with the design groups will ensure that the system design is compatible with the assembly, test, and launch operations requirements.

The assembly and test organization deliverable items at the completion of Phase IB will be:

- Assembly and checkout plan (including planning and control)
- Preliminary launch operation plan
- Preliminary mission support operations plan
- Facilities plan
- OSE specifications and implementation plan

By the sixth week of Phase IB the specific tasks and methods will be defined. The preliminary plans will be completed during the thirteenth week. The plans will then be reviewed, finalized and updated prior to the completion of Phase IB during the 26th week.

8.1 Assembly and Checkout Plan

The assembly and checkout plan will detail all operations and procedures to be performed from receipt of the integrated subsystems in the

spacecraft assembly area to shipment of the spacecraft to ETR. Detailed specification for all required procedures and computer programs will be included. These specifications will be formulated such that the actual procedure and computer program can be written using these specifications.

OSE specifications will be generated from these requirements. The integrated test plan will also be used to develop OSE requirements. Included will be the controls and procedures necessary to define: stores control, status reporting, hardware configuration, verification procedures and software configuration control, cost accounting and shipping, and the top operations schedules.

8.2 Preliminary Launch Operations Plan

The preliminary launch operations plan will specify all operations to be performed and procedures to be used from receipt of the spacecraft at ETR to launch, including spacecraft turnaround times and decision criteria as a function of specific failures. Launch site logistics and spares will be defined. Coordination with launch operations personnel at KSC will be required. The integrated test plan and the assembly and checkout plan will be used as prime reference documents. Procedures and computer program specifications will be included.

The organizational structure required to perform these tasks will be outlined and its interfaces with JPL, KSC and the remainder of TRW will be defined.

8.3 Preliminary Mission Support Operations Plan

During Phase IB, all of the mission support operations tasks will be defined and detailed. This will include such things as flow charting all mission support computer programs, detailing the functions to be performed at the various DSIF complexes and planning in depth the educational program for the orbital operations personnel. In addition to the tasks described above, the definition and planning of the software for experiment computer programs will be completed and detailed flow charts generated.

8.4 Facilities Plan

The Voyager facility needs will be reassessed, taking into account program changes directed by JPL. The requirements determined by the above reappraisal will result in reissuance of building and major test facility specifications to enable timely completion of new construction at TRW Space Park, and modification of off-site facilities required to meet program requirements (e. g. White Sands, Cape Kennedy, etc.).

Detailed schedules will be prepared indicating the manner in which facility needs will keep pace with program growth. A complete plan indicating all facility milestones will be issued at the termination of Phase IB.

8.5 OSE Specification and Implementation Plan

The initial Phase IB effort will concentrate on development of a complete list of all AHSE and EOSE required for system and subsystem development, manufacture and test. Subsequent to the development of the OSE lists, preliminary designs and specifications will be generated (by October 1966) for submittal to JPL. JPL final approved design specifications will be completed by the end of Phase IB. By the end of Phase IB, all OSE will be quantitized, scheduled, and allocated. Development of a Phase II work package and cost plan, which contains Phase II detail task descriptions and plans, will also be available as part of the Phase IB effort.

Management of Phase IB tasks will be exercised by monthly detailed design audits along with cost and PERT audits. The above audits will provide the required visibility in program problems and progress to assure completion of Phase IB with the highest degree of accomplishment of all tasks within the funds and time allocated.

The initial Phase II effort will include detail design of OSE, preparation of procedures and Part I, CEI specifications. The detailed design will be based on the Phase IB preliminary designs. Both test and operational procedures will be generated. The procedures and detailed designs will be reviewed at the critical design review, prior to finalization. Fabrication of OSE starts approximately seven months from start of Phase II

with deliveries to support all subsystem development tests and fabrication as well as spacecraft assembly and test activities. Detailed task and management plans will be generated by the conclusion of Phase IB.

APPENDIX A

ASSEMBLY, TEST, AND LAUNCH OPERATIONS

This appendix contains the detailed descriptions, in the form of tables, of the assembly and test operations. Three tables are included covering the following:

- Assembly and checkout operations
- Spacecraft test operations
- Launch operations

The off-site operations conducted with the proof test and propulsion interaction model spacecraft are contained in the spacecraft test operation tables.

In the left margin of the tables are the operation numbers. The flow charts, located in the text, are keyed on these operation numbers.

The tables contain a heading titled, "Equipment Required by OSE Code Numbers." The 3000 to 4000 series numbers refer to the electrical operational support equipment (EOSE), while the 5000 to 6000 series numbers refer to assembly, handling, and shipping equipment (AHSE).

In the right hand margin of the tables, under the heading, "Applicable Spacecraft," are listed the abbreviations for the engineering model (EM), propulsion interaction model (PIM), proof test model (PTM), first flight (Flt 1), second flight (Flt 2), and the third flight (Flt 3) model spacecraft. An "x" under these abbreviations means that particular operation applies to that spacecraft.

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
1	Receive Spacecraft Aft Equipment Module from Stores a) Unpack aft equipment module. b) Visually inspect for shipping damage. c) Move the aft equipment module to the assembly area.	Assembly and Checkout Facility	5001, 6005, 5004, 6001, 6002, 5008, 6003, 6004	X	X	X	X	X	X
2	Mechanically Mount the Spacecraft Aft Equipment Module on the Vertical Checkout and Assembly Stand	Assembly and Checkout Facility	5001, 6005, 5003, 5004, 5007, 6002, 5008, 5026, 6004, 6001	X	X	X	X	X	X
3	Install Pyrotechnics Subsystem Plumbing and Install Clamps in the Aft Equipment Module	Assembly and Checkout Facility	5003, 5004, 5007, 5008, 6202, 6203, 5036, 5037, 5038, 5039,	X	X	X	X	X	X
4	Install the Guidance and Control Plumbing and Install Clamps in the Aft Equipment Module a) Perform leak test on the pyrotechnic and control pneumatics systems in the aft equipment module.	Assembly and Checkout Facility	5003, 5004, 5007, 5008, 5036, 5037, 5038, 5701, 5702, 5703, 5704, 5705, 5003, 5004, 5007, 5008, 5036, 5037, 5038	X	X	X	X	X	X
5	Install the Spacecraft Harness Assembly in the Spacecraft Aft Equipment Module a) Route cables and clamp in place. b) Mechanically mount j-boxes. c) Electrically connect the harness assembly with the j-boxes. d) Perform the hi-pot and continuity check of the harness assembly. e) Mount the equipment mounting panel to the equipment mounting panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5901, 5038, 5036 Fact Machine	X	X	X	X	X	X
6	Install the Power Subsystem a) Draw a power subsystem subpanel from stores. b) Mechanically install the power subsystem subpanel on the equipment mounting panel installation fixture. c) Install dummy array.	Assembly and Checkout Facility	5003, 5007, 5008, 5801, 5803, 5804, 5806, 5808, 5809, 5802, 5036, 5038, 6004, 6001, 3100, 4100, 4200,	X	X	X	X	X	X
7	Perform the Power Subsystem Functional Test a) Connect the power subsystem EOSE to the input, output and test connectors on the power subsystem. b) Electrically connect all EOSE test sets to the STC interfaces. c) Perform the simulated solar array test. 1) Turn on the solar array simulator portion of the the EOSE and observe that voltage exists only at the appropriate points on each of the shunt regulators. 2) Set the solar array simulator to a 1.0 AU setting. 3) Vary the dummy loads from minimum to maximum noting that the proper error signal amplitudes exist at the output of the power control unit. 4) Repeat step 3) for solar array simulator settings of 1.2 and 1.67 AU.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 3100, 4100, 4200	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	5) With the solar array simulator adjusted to 1.0 AU, vary the dummy loads from minimum to maximum loading and observe that the proper regulators are operating in a non-conducting, linearly conducting or saturated state. 6) Repeat step e) for sun intensities of 1.2 and 1.67 AU. d) Battery control test 1) Using the ground power and the solar array simulator, charge each battery at each charge rate. 2) Measure the charge current applied to each battery for each charge rate. 3) Observe that the charging voltage for each battery is cut off when each battery is charged to capacity. 4) Reduce the solar array simulator power until it is unable to support the minimum load on the PCU. 5) Ascertain that the boost regulator portion of the battery regulator is able to maintain the system bus voltage at 50 VDC ±1 percent at minimum and maximum load demands. 6) Record the voltage when the booster regulator begins functioning. e) Perform power synch test 1) Measure the amplitude of each synch signal at each inverter connector. 2) Measure the frequency of each synch signal at each inverter connector. 3) Measure the rise and fall time of each synch signal at each inverter. 4) Measure the on/ off transients of each synch signal at each inverter. f) Perform 4.1 kc inverter tests 1) Check the output current and voltage at the 4.1 kc inverter dummy loads. 2) Observe the 4.1 kc inverter regulates within specification from a minimum and maximum load condition. 3) Determine that the 4.1 kc inverter regulates within specification from minimum to maximum input voltages. 4) Determine that the 4.1 kc inverter regulates within specification while operating under minimum input voltage and maximum load conditions. 5) Determine that the 4.1 kc inverter regulates within specification while operating under maximum input voltages and minimum load conditions. g) Perform the 410 cps inverter tests 1) Check the inverter output current and voltage at the dummy loads. 2) Determine that the inverter regulates within specification from minimum to maximum loads.								

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	3) Determine that the inverter regulates within specification from minimum to maximum input voltages. 4) Determine that the inverter regulates within specification while operating under minimum input voltages and maximum load conditions. 5) Determine that the inverter regulates within specification while operating under maximum input voltages and minimum load conditions. h) Determine that the master oscillator is within the following specifications: 1) Frequency 2) Amplitude 3) Wave shape 4) Impedence i) Perform failure mode test 1) The failure mode test will encompass the following investigations: • Loss of synch input to the PCU • Loss of synch input to each inverter • Battery failures • Battery regulator failures • Inverter failures								
8	Integrate the Power Subsystems in the Spacecraft a) Preset the power control unit to the most off condition. b) Remove the EOSE cables from the input/output connectors of the power subsystem c) Mate the flight harness to the input/output connectors of the power subsystem. d) Mate the EOSE subsystem solar simulator and panel control input cables to their respective flight harness terminations. e) Systematically commence a power turn-on sequence while monitoring open circuit voltages at appropriate flight harness connectors. f) Perform the following test: 1) Verify command capability of the PCU. 2) Verify ground battery charging through the flight harness. 3) Verify solar array battery charging through the spacecraft harness.	Assembly and Checkout Facility	5003, 5007, 5008, 5006, 5038, 5026, 6004, 6001 3100, 4100, 4200,	X	X	X	X	X	X
9	Install the Command Subsystem Subpanel a) Draw a command subsystem subpanel from stores. b) Mechanically install the command subsystem to the equipment mounting panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5301, 5806, 5038, 5036, 6004 3100, 4100, 4200, 3600	X	X	X	X	X	X
10	Perform the Command Subsystem Functional Test a) Connect the EOSE cables to the input/output connectors of the command subsystem. b) Connect the EOSE cables to the command decoder and command detector unit test connectors.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5036, 5038, 6004, 6001 3100, 4100, 4200, 3600	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	c) Apply power to the command subsystem and check the following: 1) Output voltages and currents at the dummy loads. 2) Thresholds 3) Frequencies d) Fail one detector and decoder e) Vary the input frequency and amplitude between minimum and maximum specifications and check the output voltages and current for deviation from the specifications at the dummy loads. f) Enable the failed detector and decoder. g) Disable the other detector and decoder. h) Vary the input frequency and amplitude between minimum and maximum specification and check the output voltages and current for deviation from the specifications at the dummy loads. i) Enable both detectors and decoders.								
11	Integrate the Command Subsystem in the Spacecraft a) Disconnect the input/output EOSE cables from the subsystem. b) Fuse the power lines to the command subsystem. c) Insure the power subsystem is in the most off condition. d) Connect the flight harness to the input/output connectors of the command subsystem. e) Command the PCU to apply power to the command subsystem. 1) Measure voltage and current at the power inputs to the command subsystem. 2) Measure noise, frequencies, and switching transients on all the input/output lines of the command subsystem. 3) Check command functions at all distribution points. • Voltage levels • Currents • Locations	Assembly and Checkout Facility	5003, 5007, 5006, 5036, 5038, 5806, 6004, 6001 3100, 4100, 4200, 3600	X	X	X	X	X	X
12	Mechanically Install the Telemetry Subsystem Subpanel a) Draw a telemetry subsystem module and container from stores. b) Install subsystem subpanel on aft equipment mounting panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5501, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500,	X	X	X	X	X	X
13	Perform the Telemetry Subsystem Functional Test a) Connect the EOSE cables to the input, output and test connectors on the telemetry subsystem. b) Power the subsystem and check: 1) ID words corresponding to all bit rates and formats. 2) Measure the subcarrier frequency and modulation index of the down link baseband signal. 3) Measure the frequency, pulse amplitude rise time, fall time, and the pulse width of all timing pulses.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001 3100, 4100, 4200, 3600, 3500	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	4) Repeat 3) for all bit rates. 5) Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all shift pulses. 6) Repeat 5) for all bit rates. 7) Measure the frequency, pulse amplitude, rise time, fall time, and pulse width for all inhibit pulses. 8) Repeat 7) for all bit rates. 9) Measure the frequency, pulse amplitude, rise time, fall time, and pulse width for all synch pulses. 10) Repeat 9) for all bit rates. Note: Noise, transient, and cross talk measurements will be conducted for items 2) through 10).								
14	Integrate the Telemetry System in the Spacecraft a) Disconnect the input/output EOSE cables from the telemetry subsystem. b) Insure that the power system is in the most off state. c) Fuse all power input lines to the telemetry subsystem. d) Connect the spacecraft flight harness to the input and output connectors of the telemetry subsystem. e) Command the power system on. f) Command the telemetry subsystem on. 1) Measure current and voltage at the input to the telemetry system. 2) Measure noise and switch transients on all the input and output lines of the telemetry subsystems. 3) Measure the distribution of all the telemetry subsystem outputs. 4) Repeat items 1) through 3) for all bit rates. g) Calibrate the telemetry signals from the following subsystem: 1) Power 2) Command 3) Telemetry	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001 3100, 4100, 4200, 3600, 3500	X	X	X	X	X	X
15	Perform a Compatibility Test Between the Power, Command, and Telemetry Subsystems a) Sequentially step the power subsystem through all modes of operation while monitoring the telemetry and test points from the power, command, and telemetry systems. b) Sequentially step the telemetry subsystem through all modes of operation while monitoring the telemetry and test points from the power, command, and telemetry systems. b) Sequentially step the command subsystem through all remaining commands while monitoring the telemetry and test points from the power, command, and telemetry systems.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
16	Install the Data Storage Subsystem Subpanel Module a) Draw data storage subsystem out of stores with container. b) Mechanically install the data storage subsystem to the spacecraft equipment panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5601, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700	X	X	X	X	X	X
17	Perform the Functional Test on the Data Storage Subsystem a) Electrically connect the EOSE cables to the input, output, and unit test connectors. b) Power the data storage subsystem and perform the following tests on each recorder: 1) Measure all command line voltages, current, noise, transients, and frequencies. 2) Load each data storage unit (DSU) with known data from the EOSE. 3) Recall the data from each DSU and measure noise, frequencies, amplitudes, and transients at the dummy loads. 4) Measure the rise time, fall time, amplitude, and pulse duration of the DSU data output signal at the DSU dummy memory readout. 5) Measure the rise time, fall time, amplitude, and pulse duration of the DSU and data pulse at the DSU. 6) Calibrate all telemetry signals from the data storage subsystem. Note: Noise, transient and cross talk measurements will be conducted for items 3) through 6). 7) Command off the power to the data storage subsystem and remove the fuses on the input power lines, and mate directly.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700	X	X	X	X	X	X
18	Integrate the Data Storage Subsystem with the Spacecraft Electrical System a) Remove the EOSE cables from the input and output connectors of the data storage subsystem. b) Place fuses in all of the power lines to the data storage subsystem. c) Insure the power system is in the most off condition. d) Connect the flight harness to the input and output connection of the data storage subsystem. e) Turn on the power subsystem and sequentially command the data storage subsystem on, while making the following measurements: 1) Voltage and current on all power input lines. 2) Noise, wave shapes, frequencies and transients on the power input lines. 3) Measure the rise time, fall time, amplitude, and pulse duration of the DSU input data signal at the DSU for each bit rate.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
19	Perform a Compatibility Test of the Following Subsystems 1) Power 2) Command 3) Telemetry 4) Data storage a) With the power subsystem on, command each of the four integrated subsystems on, in sequence, while monitoring all telemetry outputs and test points from each subsystem for the following interference: 1) Noise 2) Transients 3) Frequency (high and low) 4) Wave shape distortion 5) Data blanks 6) Voltage amplitude and phasing 7) Current amplitude and phasing b) Repeat a) for all bit rates and modes.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 3100, 4100, 4200, 3600, 3500, 3700	X	X	X	X	X	X
20	Mechanically Install the Radio Subsystem Subpanel to the Spacecraft Equipment Mounting Panel a) This equipment is delivered as an integrated subsystem from the vendor. b) Draw the radio subsystem module and container from stores. c) Mount the radio subsystem to the equipment mounting panel installation fixture. d) Install all antennas, diplexers, couplers, and filters mechanically to the spacecraft.	Assembly and Checkout Facility	5003, 5007, 5008, 5101, 5806, 5103, 5104, 5105, 5106, 5102, 5036, 5038, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400	X	X	X	X	X	X
21	Perform the Radio Subsystem Functional Test a) Connect the EOSE to the input, output, and test connectors. b) Power the radio subsystem and measure the following signals: 1) Voltage input amplitude 2) Current input amplitude 3) Noise on inputs 4) Frequency on inputs 5) Transients on inputs c) Measure the receiver thresholds. d) Measure the outputs of the receivers for the following: 1) Voltage amplitudes 2) Power 3) Quality of output signals 4) Frequency 5) Signal to noise ratio 6) Noise 7) Transients		5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	e) Measure the following signals on the 1 watt transmitter and power monitor: 1) Power output 2) Quality 3) Frequency 4) Signal level f) Measure the following signals on the modulator exciter: 1) Index of modulation 2) Voltage output amplitude 3) Current output amplitude 4) Modulator stability 5) Frequency 6) Phasing g) Measure the following signals on the 50-watt TWT: 1) Power input 2) RF power output 3) Frequency 4) Stability 5) Parasitic harmonics and sidebands h) Test the receiver selector switch for correct receiver selection and monitor the voltage and current input and output of the selector switch. i) Test the transmitter selector switch for the following: 1 1) Voltage required to operate 2) Current required to operate 3) Transmitter selection 4) Voltage output amplitude 5) Current output amplitude 6) Monitor for arcing or other malfunction indications during switching transmitters. j) Test the circulator switch for sequencing properly and measure the following: 1) Voltage input amplitude 2) Current input amplitude 3) Voltage output 4) Current output k) Check the diplexers for the following: 1) Frequency isolation 2) Attenuation to the antenna l) Four-part hybrid ring and power monitor checks: 1) Power 2) Frequencies 3) Noise 4) Impedance 5) Switching								

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
22	Integrate the Radio Subsystem with the Spacecraft Systems a) Disconnect the EOSE from the input and output connectors of the radio subsystem. b) Fuse the input power lines to the radio subsystem. c) Mate the input and output connector of the radio subsystem to the spacecraft flight harness. d) Measure the following on all antenna lines: 1) Insertion losses 2) VSWR 3) Isolation e) Turn the power subsystem on and command the radio subsystem on. f) Check all transmitters, sequentially, for the correct operation while monitoring the following: 1) Spectrum 2) Frequencies 3) Power input 4) Power output from antennas 5) Outputs from the power monitors 6) Harmonics 7) Correct antenna switching 8) Voltage to the radio subsystem 9) Currents drawn by the radio subsystem g) Check the operation of the uplink S-band receivers while monitoring the following: 1) Sensitivity and thresholds 2) Voltage outputs 3) Frequency 4) Loop stress h) Calibrate the radio subsystem telemetry output	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400	X	X	X	X	X	X
23	Perform a Compatibility Test of the Following Subsystems: 1) Power 2) Command 3) Telemetry 4) Data storage 5) Radio a) Sequentially turn on the subsystems while monitoring the telemetry outputs and subsystem test points from the five subsystems for the following: 1) Transients 2) Noise 3) Grounding problems 4) RF interference	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
24	Install the Relay Link Subsystem (GFE) to the Spacecraft Equipment Mounting Panel a) Draw the relay link subsystem module and container from stores. b) Mechanically install the relay link subsystem to the spacecraft equipment mounting panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400,	X	X	X	X	X	X
25	Perform the Relay Link Subsystem Functional Test a) Connect the EOSE to the input, output and test connectors of the relay link subsystem. b) Power the relay link subsystem and monitor the following: 1) Voltage input amplitude 2) Current input amplitude 3) Noise on input power 4) Noise on output signals 5) Transients 6) Voltage output amplitude 7) Current output amplitude 8) Recorder speed 9) Record function 10) Recorder playback function 11) Wave shape, levels, timing and index of tape playback.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400	X	X	X	X	X	X
26	Integrate the Relay Link Subsystem in the Spacecraft a) Remove the EOSE cables from the input and output connectors of the relay link subsystem. b) Fuse the power input lines to the relay link subsystem. c) Connect the spacecraft harness to the input and output connectors of the relay link subsystem. d) Measure the following parameters on the UHF antenna: 1) Insertion loss 2) VSWR 3) Isolation e) Check the following parameters on the relay link subsystem: 1) Receiver sensitivity 2) Receiver threshold 3) Receiver output level 4) Frequency 5) Tape recorder loading 6) Tape playback • Indexing • Timing • Pulse width • Amplitude of signal output • Signal to noise ratio		5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600, 3500, 3700, 3400	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	7) Transients 8) Noise 9) Data blanks f) Repeat e) for all bit rates and modes g) Remove the fuses from the input power lines and connect directly to the flight harness. h) Calibrate the relay link subsystem telemetry output signals.								
27	Perform a Compatibility Test of the Following Subsystems: 1) Power 2) Command 3) Telemetry 4) Data storage 5) Radio subsystem 6) Relay link subsystem a) Sequentially turn each subsystem on while monitoring the telemetry outputs and monitoring all test points from each subsystem for the following: 1) Noise 2) Parasitic oscillations 3) Transients 4) RF interference 5) Signal degradation 6) Degraded system performance	Assembly and Checkout Facility	3500, 3700, 3400, 5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3100, 4100, 4200, 3600	X	X	X	X	X	X
28	Install the Computing and Sequencing Subsystem Subpanel on the Spacecraft Equipment Mounting Installation Fixture a) Draw the computing and sequencing subsystem module and container from stores. b) Mechanically install the computing and sequencing subpanel to the spacecraft equipment mounting panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 5401, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200	X	X	X	X	X	X
29	Perform the Computer and Sequencer Functional Test a) Connect the EOSE cables to the input, output, and test connectors of the computing and sequencing subsystem. b) Measure the voltage and current input to C and S subsystem. c) Monitor all input and output signal levels for the following: 1) Noise 2) Frequency 3) Transient 4) Response 5) Pulse duration 6) Logic sequence 7) Timing signals 8) Command ability to the other subsystem 9) Speed up capability 10) Command discrete levels	Assembly and Checkout Facility	3500, 3700, 3400, 3100, 5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 4100, 4200, 3600, 3200	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
30	Integrate the Computing and Sequencing Subsystem into the Existing Systems on the Spacecraft a) Remove the EOSE cables from the input and output connectors of the computing and sequencing subsystem. b) Fuse all input power lines to the computing and sequencing subsystem. c) Command the computing and sequencing subsystem and measure the following signals: 1) Power drain by the subsystem. 2) Transmit each quantitative command from the command subsystem, and observe that each command was properly received and executed by monitoring the computer core dump. 3) As each command is sent in item 2) monitor that noise and transients are at acceptable levels. 4) Measure the amplitude and frequency of the down link PN subcarrier. 5) Measure the amplitude and frequency of all timing signals from the computing and sequencing subsystem. 6) Measure the frequency and stability of the master oscillator from the power subsystem to the computing and sequencing subsystem. d) Remove the fuses on the input power lines and mate the connector directly to the flight harness. e) Calibrate the telemetry signal outputs from the C and S subsystem.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200	X	X	X	X	X	X
31	Perform a Compatibility Test on the Following Subsystems: 1) Power 2) Command 3) Telemetry 4) Data storage 5) Radio 6) Relay link 7) Computing and sequencing • Sequentially turn each subsystem on while monitoring the telemetry outputs and all test points from each subsystem for the following: Noise Transients RF interference Signal degradation Degraded system performance Incompatibility between subsystems	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200	X	X	X	X	X	X
32	Install the Guidance and Control Subsystem Subpanel on the Equipment Mounting Panel of the Spacecraft a) Draw the guidance and control subsystem subpanel and container from stores. b) Mechanically mount the subsystem to the spacecraft equipment mounting panel installation fixture.	Assembly and Checkout Facility	5003, 5007, 5008, 5701, 5702, 5806, 5038, 5036, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
33	Perform the Guidance and Control Subsystem Functional Test a) Connect all EOSE cables to the input/output and test connectors of the guidance and control subsystem. Note: During this functional test, sensor simulators will be used to generate the error signals. b) Power the guidance and control system and measure the following signals: 1) Power into the system 2) Noise 3) Transients 4) Output amplitudes 5) Check logic for nonexisting error signal outputs. c) Simulate sun acquisition and monitor the following: 1) Observe the voltage appears where it should and not at any other location. 2) Simulate the pitch, roll and yaw gyro outputs and measure noise, correct gas valve actuations, threshold of the gas valves, power drawn by the gas valves. At the same time, calibrate telemetry for known error signal amplitudes from the gyro simulaters to compare against scale factor gains from the actual gyros later. 3) Simulate the sun sensor error signals and check gas valve actuations. Calibrate the telemetry output for the sun sensors error signals. Measure noise, transients and power to the valves. 4) Calibrate the sun intensity telemetry word for each sun sensor throughout the range of telemetry. The laboratory curves for each sun sensor (intensity versus voltage out) together with the digital word values will be inserted into the computer program. d) Simulate the earth sensor in the dark mode and measure noise and transients. 1) Simulate the earth sensor in the illuminated mode and measure noise and transients. 2) Measure voltage and current being drawn by the central system while the earth is in the illuminated mode. 3) Calibrate the telemetry words for earth dark, earth illuminated and determine the threshold of these modes. e) Simulate the Canopus sensor and measure the following: 1) Noise 2) Transients 3) Proper valve actuation for polarity changes. 4) At the center of the sensor field of view that gas actuations are non-existant.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	5) Telemetry signals are consistent with plus, minus, or center of the field of view of the sensor and that the valve telemetry indicates the proper valve firings. 6) Calibrate the telemetry output signals for the Canopus sensor error signals. 7) Monitor the logic sequence of the central control and logic assembly for the proper logic state. 8) Monitor for proper switch signals to the antennas during the Canopus logic checks.								
34	Integrate the Guidance and Control Subsystem with the Existing Spacecraft Systems a) Disconnect the EOSE cables from the input and output of the guidance and control subsystem. b) Fuse the input power lines. c) Connect the flight harness to the input and output of the subsystem. d) Install the inertial block housing the sensors, and perform the electrical interface as follows: 1) Perform gyro package checks. The gyro package checks are performed so that the gyro polarity and drift rates can be determined as part of the guidance and control testing phase. 2) Perform sun acquisition electrical tests. The sun acquisition electrical tests will be performed as follows: • Apply power to the spacecraft and command the gyros to on. • Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector of the gyro package. • Connect the gyro package to the spacecraft harness and measure the voltage and current drawn by the gyro spin motors (also measure turn on transient amplitude etc.). Note that noise and transients on these lines are within acceptable levels. • Check that voltage exists where it should and that no voltage exists on the remaining pins of each connector of the control signal electronics package. • Torque the spacecraft in the + yaw direction and measure the yaw gyro output signal amplitude. Note that the polarity is correct. • Torque the spacecraft in the - yaw direction and measure the yaw gyro output signal. Note that the polarity is correct. • Repeat previous two steps for the pitch and roll gyros. • With the spacecraft absolutely still, measure the noise amplitude on each gyro output line.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 5014, 5015, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	• By use of the torque current, increase the rate in each axis in each direction, separately, and note that the proper gas valve is actuated. • Determine the threshold rates in each axis which will just barely cause the gas valves to actuate. • Measure the voltage and current drawn from the secondary power supply subsystem by the control signal during gyro zero rate input conditions and during maximum rate inputs. Note that noise and transients are within acceptable levels. • Connect the sun sensors to the spacecraft harness. • Attach the sun sensor stimulus to each sun sensor. • Automatically simulate separation switches and check that the sun acquisition mode has started. • Transmit the backup command for starting the sun acquisition sequence. • Illuminate each sun sensor and check that voltage exists at each valve interface. • Stimulate each sun sensor and measure the voltage and current drawn from the secondary power supply subsystem by the control signal electronics package during each valve actuation. • Observe that when each sun sensor is stimulated the proper valve is opened. • Observe that when all five sun sensor elements are illuminated, no valves are actuated. 3) Perform Sun Acquisition Calibrations The sun acquisition calibration will be performed as follows: • The valve actuation signals will be calibrated by actuating each valve and noting the telemetry word values. • Gyro on/off calibrations will be performed by commanding them on and then off and the telemetry word value recorded. 4) Perform earth sensor electrical tests The earth sensor electrical tests will be performed as follows: • Turn on power to the spacecraft and command the earth sensor to on. • Observe that voltage exists where it should and that no voltage exists on the remaining pins of the earth sensor connector. • Connect the earth sensor to the spacecraft harness and measure the voltage and current drawn by the earth sensor from the secondary power subsystems. Note that noise and transients are within acceptable limits.								

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	• Darken the earth sensor aperture and measure the signal output amplitude. Note that noise and transients are within acceptable levels. • Attach the earth sensor stimulus to the earth sensor. • Illuminate the earth sensor and measure the sensor output signal amplitude. Note that noise and transients are within acceptable levels. • Measure the voltage and current drawn from the secondary power subsystem while the earth sensor is being illuminated Note that noise and transients are within acceptable limits. 5) Perform earth sensor calibrations The earth sensor calibrations will be performed as follows: • As the error signal level is varied the telemetry word values for this measurement will be recorded. These parameters as well as the laboratory bench test data (voltage versus intensity) will be inserted in the computer program. • Calibrate the earth sensor temperature sensor. 6) Perform Canopus acquisition tests The Canopus acquisition electrical tests will be performed as follows: • Turn on power to the spacecraft and command the Canopus sensor to on. • Observe that voltage exists where it should and that no voltage exists on the remaining pins of the Canopus sensor connector. • Connect the Canopus sensor to the spacecraft harness and measure the voltage and current drawn by the Canopus sensor. Note that noise and transients on these lines are within acceptable levels. • Attach Canopus sensor stimulus to the Canopus sensor. • Determine the threshold of the Canopus sensor field of view and note that the proper valves actuate. • Measure the voltage and current drawn by the Canopus sensor when each sensor half is illuminated. Note that noise and transients on these lines are within acceptable levels. • Illuminate the center of the Canopus sensor field of view and note that no valves are actuated. • Investigate the Canopus sensor signal output lines for out-of-tolerance transient and noise conditions when the center of the Canopus sensor is illuminated. • Command the spacecraft into the roll search mode and observe that the proper roll valves are actuated.								

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	• Investigate the Canopus sensor signal output lines for out-of-tolerance transient and noise conditions when the center of the Canopus sensor is illuminated. • Command the spacecraft into the roll search mode and observe that the proper roll valves are actuated. • Remove Canopus sensor illumination and observe that the guidance and control subsystem goes into the roll search mode. • Note that the airborne receivers switch to the omni antenna when the Canopus illumination is removed. 7) Perform canopus acquisition calibrations The Canopus acquisition calibrations will be performed as follows: • The Canopus sensor input signal level is varied, the telemetry word value for this measurement will be recorded. These parameters as well as the laboratory bench test data (voltage versus roll error in radians) will be inserted into the computer program. • The Canopus sensor intensity signal will be calibrated as in step above. • Calibrate the Canopus sensor temperature sensors.								
35	Perform a Compatibility Test of the Following Subsystem: 1) Power 2) Command 3) Telemetry 4) Data storage 5) Radio 6) Relay link 7) Computing and sequencing 8) Guidance and control • Sequentially turn power on each subsystem and step through all modes and bit rates while monitoring all telemetry channels and test points for the following: Noise Transients RF interference Noncompatibility between subsystems • Ship the PIM model spacecraft to White Sands, New Mexico. The completion of the integration and checkout tests of this spacecraft will be done at that location.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 5038, 5036, 6004, 6001, 5014, 5015, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300	X	X	X	X	X	X
36	Receive the LEMDE Module from Stores	Assembly and Checkout Facility	6530, 5001, 5004, 5008,	X	X	X	X	X	X
37	Install the Spacecraft Aft Equipment Module to the LEMDE Module a) Mechanically mount the spacecraft aft equipment structure to the LEMDE module.	Assembly and Checkout Facility	6530, 5001, 5003, 5005, 5008, 5806, 5036, 5038, 5026, 6532, 6533, 6534, 6549, 6550, 6551, 6004, 6001, 3500, 3700, 3400,	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	b) Mount LEMDE assembly to the vertical checkout and assembly stand. c) Install flight capsule interstage structure.		3100, 4100, 4200, 3600, 3200, 3300						
38	Perform LEMDE Module Functional Test a) Connect the EOSE to the LEMDE module input, output, and test connectors. b) Simulate error signals and gimbal the thrust chamber. c) Perform a cold flow propellant test. d) Simulate signals to test all actuators and valves. e) Verify all telemetry outputs from the LEMDE module.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 6004, 6001, 6007, 6006, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
39	Integrate the LEMDE Module to the Spacecraft a) Disconnect the EOSE from the input and output connectors of the LEMDE module. b) Fuse the power lines to the LEMDE module. c) Connect the flight harness to the LEMDE interfaces. d) Generate error signals from the C and S and guidance and control systems and measure the following signals: 1) Power 2) Noise 3) Transients 4) Calibrate all telemetry signals 5) Repeat items 1), 2), and 3) for all modes and bit rates.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
40	Perform a Compatibility Test of the Following Subsystems: 1) Power 2) Command 3) Telemetry 4) Data storage 5) Radio 6) Relay link 7) Computing and sequencing 8) Guidance and control 9) LEMDE module • Sequentially turn on the power to each subsystem and step through all modes and bit rates and perform all functions of each subsystem while monitoring telemetry and test points for the following: Noise Transients RF interference Noncompatibility between subsystems Note: This compatibility test will essentially be an integrated systems test of the spacecraft. This will assure maximum compatibility of the spacecraft subsystems before the integration of the planetary scan packages and the fixed science packages.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
41	Install the Planetary Scan Package (PSP) onto the Spacecraft a) Draw the planetary scan package from stores. b) Mechanically install the planetary scan packages onto the spacecraft.	Assembly and Checkout Facility	5003, 5004, 5007, 5008, 6301, 6302, 6303, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
42	Perform the PSP's Functional Test a) Connect the EOSE cables to the PSP's system. b) Perform the functional test to the specified requirements of the supplier.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
43	Integrate the PSP's to the Spacecraft Electrical System a) Remove the EOSE cables and fuse the interface to the spacecraft. b) Perform the functional test as supplied and monitor the following signals: 1) Noise 2) Power 3) Telemetry outputs 4) Functions c) Remove the fuses and connect directly to the flight harness.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
44	Install the Fixed Science Packages (FSP's) in the Spacecraft a) Draw the FSP's from stores. b) Mechanically install the FSP's in the spacecraft	Assembly and Checkout Facility	5003, 5004, 5007, 5008, 6401, 6402, 6403, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
45	Perform a Functional Test of the FSP's a) Connect the EOSE cables to the FSP's. b) Perform a specified functional test, supplied by GFE, to insure the complete operation of the FSP's.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
46	Integrate the FSP's to the Spacecraft Electrical System a) Fuse the spacecraft interfaces. b) Connect the flight harness to the FSP's. c) Perform the functional test and calibrate all telemetry channels. d) Remove the fuses and connect directly to the flight harness.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X
47	Perform a Compatibility Test of the PSP's and the FSP's with the Spacecraft Systems a) Sequentially turn on each subsystem while monitoring the telemetry and test points of the FSP's and PSP's to detect noise, transients or interference. b) Monitor all spacecraft telemetry and test points while sequentially stepping through all functions of the FSP's and PSP's to detect interference to the spacecraft system.	Assembly and Checkout Facility	5003, 5007, 5008, 5806, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3300, 3900	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
48	Install the Temperature Control Subsystem into the Spacecraft	Assembly and Checkout Facility	5003, 5007, 5008, 6101, 6105, 6102, 6106, 6103, 6107, 5806, 6108	X	X	X	X	X	X
49	Functionally Test the Temperature Control Subsystem a) Check all interfaces b) Perform thermistor verification c) Check all heaters 1) Measure power 2) Measure temperature d) Test louvers e) Check logic and thermostats for temperature heater turn on/off. f) Calibrate telemetry.	Assembly and Checkout Facility	5003, 5007, 5008, 6104, 5806	X	X	X	X	X	X
50	Receive the Test Capsule and Inspect in the Assembly Area	Assembly and Checkout Facility	5010, 5011, 5012, 5040	X	X	X	X	X	X
51	Install Test Capsule Mechanically to the Spacecraft	Assembly and Checkout Facility	5005, 5007, 5008, 5003, 5010, 5011, 5012, 5040, 6004, 6001	X	X	X	X	X	X
52	Perform Spacecraft-Capsule Integration a) Fuse all interfaces. b) Check power, noise, transients, commands, relay link communications and spacecraft capabilities. c) Remove all fuses and perform a specified functional test and calibrate all telemetry channels.	Assembly and Checkout Facility	5003, 5007, 5010, 6004, 6001, 3500, 3700, 3400, 3100, 4100, 4200, 3600, 3200, 3900	X	X	X	X	X	X
53	Perform a Simulated IST and Check System Test Complex and Spacecraft Capsule Integrity Test a) This test will follow the format of the first integrated systems test. b) Recheck all telemetry calibrations.	Assembly and Checkout Facility	5003, 5007, 5010, 6004, 6001, 3000, 4100, 4200	X	X	X	X	X	X
54	Perform First IST and Critique This test is designed to test the entire spacecraft to the fullest extent to determine satisfactory system operation. The test will be performed in as close to a flight configuration as possible with only those hard-lines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft. b) Remove all protective devices. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments.	Assembly and Checkout Facility	5003, 5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 6518, 5806, 5010, 6103, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 3000, 4200, 4100	X	X	X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT EM	PIM	PTM	FL1	FL2	FL3
	e) Perform an operational check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Perform a check of all ordnance circuits. h) Vary the up and down RF link levels. i) Vary all test stimuli over the expected flight ranges. j) Perform a simulated countdown including umbilical separation. k) Install all protective devices. All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become a part of the spacecraft test record. The critique will be a meeting of all cognizant personnel to discuss the results of the test. During this meeting, each subsystem engineer will sign off the IST data.								
54a	Perform an Ethyline Oxide Compatibility Test a) Remove all protective devices. b) Place biological test specimens at selected locations. c) Install a dummy shroud around the spacecraft. d) Evacuate the shroud. e) Purge the shroud with Ethyline Oxide for 24 hours. f) Evacuate the shroud. g) Purge the shroud with dry, sterile nitrogen for 8 hours. h) Remove the shroud. i) Remove and analyze the biological specimens. j) Visually inspect the entire spacecraft for any sign of corrosion or damage. k) Install protective devices.	Assembly and Checkout Facility	5003, 5007, 5806, 5010, 6103, 5046, 6004, 3000, 4200, 4100, 5008			X			
54b	Perform an IST and Critique This test will be performed to determine whether any damage or degradation of the spacecraft has occurred due to its exposure to the ethyline oxide purge. This test is designed to test the entire spacecraft to the fullest extent to determine satisfactory system operation. The test will be performed in as close to a flight configuration as possible with only those hardlines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft. b) Remove all protective devices. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments.	Assembly and Checkout Facility	5003, 5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 6518, 5806, 5010, 6103, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 3000, 4200, 4100			X			

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	e) Perform an operational check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Perform a check of all ordnance circuits. h) Vary the up and down RF link levels. i) Vary all test stimuli over the expected flight ranges. j) Perform a simulated countdown including umbilical separation. k) Install all protective devices. All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft test record. The critique will be a meeting of all cognizant personnel to discuss the results of the test. During this meeting, each subsystem engineer signs off the IST data.								
54c	Propulsion Interaction Test The spacecraft will be placed in a vacuum chamber at White Sands in which the modified LEMD engine can be fired and exhausted. All pertinent test points will be monitored, especially those from the guidance and control subsystem. Accelerometers will be placed at various locations to measure vibration levels and frequencies. Thermal Instrumentation will be installed to study effects of nozzle and plume heating. The spacecraft will be supported by a flexur system which will allow a maximum of freedom in all axes. The engine thrust vector will have been previously aligned through the center of gravity of the spacecraft. Propellants will be loaded and the chamber evacuated. The spacecraft will be placed in the midcourse maneuver mode and the engine fired in the low thrust mode. Effects of the engine firing on the system, especially the guidance and control gyros, will be monitored. The spacecraft will be placed in the deboost mode and the engine fired in the high thrust mode. Effects on the system will be monitored. Of prime concern in this test is verification that the guidance and control subsystem can control the spacecraft attitude during engine firing. Load levels at all appendage hinge points will be analyzed to verify that the structure will withstand engine firings.	White Sands Test Facility	6501 thru 6553 3000, 4200, 4100, 5010, 6103, 5806, 5008, 5007, 5058, 5059, 5011, 5012, 5040, 5009		X				
55	Perform Power Profile Test a) The flight sequence of events through sun acquisition will be followed and primary power drains monitored. b) Compare the primary power drains through sun acquisition with the trajectory information and ascertain that the battery capacity is adequate to support spacecraft operations until sun acquisition has been completed. c) Command the spacecraft to perform all of the cruise mode functions, monitoring all primary power drains.	Assembly and Checkout Facility	5003, 5007, 6103, 5806, 5010, 6004, 3000, 4100, 4200, 5008, 6540, 6541	X		X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	d) Compare the primary power drains with the trajectory information and ascertain that there is sufficient battery capacity to perform the midcourse maneuvers. e) Command the spacecraft to perform all of the Mars encounter functions, monitoring all primary power drains. f) Compare the primary power drains with the trajectory information and ascertain that sufficient battery capacity exists to perform the deboost and sun reacquisition modes for the Mars orbit operations. g) Command the spacecraft to perform all of the Mars orbiting functions, monitoring all primary power drains. h) Compare the primary power drains with the planned Mars orbit and ascertain that adequate battery capacity remains to carry the spacecraft through eclipses encountered during the Mars orbiting modes of operations.								
56	Perform Failure Mode Logic Test A complete check of all on-board logic will be performed. The test will be performed using a logic matrix such that all combinations and permutations are checked. This is primarily a test of the computer and sequencer subsystem.	Assembly and Checkout Facility	5003, 5007, 6103, 5806, 5010, 6004, 3000, 4100, 4200, 6540, 6541	X		X			
57	Perform Parameter Variation Test Certain selected critical system parameters will be varied from their nominal values to determine spacecraft behaviour under these conditions. Listed below are a number of parameters that will be varied: a) Solar array simulated power to match flight profile. b) Up and down link RF power levels to match flight profile. c) Canopus sensor cone angles. d) Midcourse and terminal maneuver turn angles. e) Midcourse engine burn duration. f) Antenna gimbal angles. g) Engine gimbal angles. h) Mechanical torques to rate gyros i) Spacecraft bus voltage.	Assembly and Checkout Facility	5003, 5007, 6103, 5806, 5010, 6004, 3000, 4100, 4200, 6540, 6541	X		X	X	X	X
58	Move Spacecraft to EMI Facility a) Remove spacecraft work stands and detach all test cables. b) Attach prime mover to vertical checkout and assembly stand and tow into EMI room.	Screen Room	5001, 5003, 5008, 6103, 5010, 5806, 5026, 6004, 6540, 6541	X		X	X	X	X
59	Perform Spacecraft Functional Test a) Detach prime mover. b) Position work stands.	Screen Room	5003, 5007, 5008, 6103, 5010, 5806, 6004, 3000, 4100, 4200, 5001, 6540, 6541	X		X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	c) Attach all required test cables and setup OSE. d) Perform a gross check of the spacecraft to verify proper spacecraft/STC operation.								
60	Perform EMI Tests a) Command the spacecraft and capsule subsystems through flight sequences and ascertain that there is no degradation or interference between subsystems. b) Irradiate the spacecraft with RF signals that correspond to the expected frequencies and levels from the launch vehicle system. c) Command the spacecraft and capsule subsystem through flight sequences and determine the frequencies and levels of RF radiation emitted from the spacecraft. d) Apply audio tones and tone bursts to the spacecraft primary bus system and observe each subsystem reaction, noting whether it is within specification. The data obtained during the assembly and checkout operation compatibility tests will be used as baseline information.	Screen Room	5003, 5007, 5008, 6103, 5010, 5806, 6004, 3000, 4100, 4200, 6540, 6541	X		X	X	X	X
61	Remove Test Capsule a) Position work stand and disconnect test cables. b) Remove work stand and connect prime mover to vertical checkout and assembly stand. c) Move vertical checkout and assembly stand out of EMI room. d) Reposition work stands. e) Attach capsule lift sling and hydraset to crane hook and attach sling to capsule. f) Position capsule transporter adjacent to vertical checkout and assembly stand. g) Remove capsule and place in transporter. h) Remove work stand and attach prime mover to vertical checkout and assembly stand. i) Tow vertical checkout and assembly stand into EMI room and detach prime mover. j) Position work stands around vertical checkout and assembly stand and attach test cables. k) Remove work stands.	High Bay Area with Crane	5001, 5003, 5006, 5007, 5008, 6103, 5010, 5011, 5012, 5806, 5040, 6004, 6540, 6541	X		X	X	X	X
62	Perform EMI Test a) Command the spacecraft subsystems through flight sequences and ascertain that there is no degradation or interference between subsystems. b) Irradiate the spacecraft with RF signals that correspond to the expected frequencies and levels from the launch vehicle system. c) Command the spacecraft subsystems through flight sequences and determine the frequencies and levels of RF radiation emitted from the spacecraft. d) Apply audio tones and tone bursts to the spacecraft primary bus system and observe each subsystem reaction, noting whether it is within specification.	Screen Room	5003, 6103, 5806, 5007, 3000, 4100, 4200, 5008, 6004, 6540, 6541	X		X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
63	Prepare Spacecraft for Shipment a) Position workstand and disconnect test cables. b) Remove workstand and connect prime mover to vertical checkout and assembly stand. c) Move vertical checkout and assembly stand to shipping area. d) Position spacecraft transporter and environmental cover adjacent to spacecraft. e) Remove spacecraft transporter environmental cover. f) Attach spacecraft lift sling to ring adapter. g) Lift spacecraft and mate to transporter, remove sling. h) Attach spacecraft transporter environmental cover to hook and mate to transporter. Remove sling. i) Attach prime mover to spacecraft transporter and tow to airlock. j) Start GN2 purge on transporter.	High Bay Area with Crane	5006, 5007, 5008, 6103, 5017, 5806, 5058, 5059, 5026, 5003, 5002, 5001, 5046, 6540, 6541	X		X	X	X	X
64	Move Spacecraft to Magnetic Properties Facilities a) Attach helicopter to transporter and remove from airlock. b) Proceed by flight/convoy to magnetics facility.		5058, Helicopter: 5056, 5055, 6540, 6541, 6103, 5806, 5059	X		X	X	X	X
65	Assemble Test Stand a) Position spacecraft transporter in magnetics facility airlock and clean all equipment. b) Attach spacecraft transporter cover sling and hydraset to ware hook and attach sling to cover. c) Remove cover, detach sling, attach spacecraft lifting sling to spacecraft and hook. d) Remove spacecraft from transporter and install on Magnetic Facility adapter/fixture. e) Install required EOSE. f) Move other equipment to storage area.	Magnetic Test Facility	5018, 5057, 5007, 5008, 5006, 6103, 5017, 5806, 5058, 5059, 3000, 4100, 4200, 5002, 6540, 6541	X		X	X	X	X
66	Perform Spacecraft Functional Test a) Position work stands. b) Attach all required test cables and set up OSE. c) Perform a gross check of the spacecraft to verify proper operation of the spacecraft/STC and the data link to Redondo Beach. c) The minimum amount of EOSE will be taken to this facility. Primary data processing will be at Redondo Beach.	Magnetic Test Facility	5018, 5057, 6103, 5806, 3000, 4100, 4200, 5007, 5008, 6540, 6541	X		X	X	X	X
67	Perform Magnetics Tests a) The magnetic field of the spacecraft will be mapped with no power applied to the spacecraft. b) The spacecraft will be permed and depermed and the change in the spacecraft magnetic field measured. Magnetic stability will be determined by this process.	Magnetic Test Facility	5018, 5057, 6103, 5806, 3000, 4100, 4200, 5007, 5008, 6540, 6541	X		X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	c) Each spacecraft subsystem will be commanded to various operating modes. While this is taking place, changes to the spacecraft magnetic fields are measured. d) The magnetometer will be calibrated with the spacecraft operating and also with all power off. e) For convenience and ease of handling, the solar array and the capsule will be mapped separately and and the fields folded to give a composite map. Past experience on OGO and Pioneer shows that this technique is technically justifiable.								
68	Perform Spacecraft/Science Interference Tests Interference between science packages and spacecraft subsystems and also intra and inter science packages will be checked. Calibration of some experiments will be performed at this time. Electromagnetic noise will be kept at a minimum at this facility.	Magnetic Test Facility	5018, 5057, 6103, 5806, 3000, 4100, 4200, 5007, 5008, 6540, 6541	X		X	X	X	X
69	Prepare Spacecraft for Shipment a) Position work stands and disconnect test cables. b) Attach spacecraft lift sling and hydraset to crane, attach sling to spacecraft. c) Hoist spacecraft and mate to transporter, remove sling. Install transporter environmental cover. d) Position spacecraft transporter in airlock. e) Start GN2 purge. f) Attach helicopter/transporter and proceed to assembly and checkout facility, Redondo Beach.	Magnetic Test Facility	5006, 5007, 5008, 6103, 5017, 5806, 5058, 5059, 5026, 5018, 5057, 5002, 5001, 5046, 6540, 6541	X		X	X	X	X
70	Move Spacecraft to Mass Properties Area a) Position spacecraft transporter in assembly and checkout facility airlock. b) Clean all equipment. c) Attach prime mover and tow transporter to mass properties area.	Assembly and Checkout Facility	5058, Helicopter: 5056, 5055, 6540, 6541, 6103, 5806, 5059	X		X	X	X	X
71	Perform Mass Properties Tests a) Remove transporter environmental cover. b) Attach mass properties sling and hydraset to crane and attach sling and load cells to spacecraft. c) Lift spacecraft. d) Move transporter to storage area. e) Weigh the spacecraft with load cells in three places. f) Using this weight data compute the center of gravity in two of the spacecraft axes. g) Tilt the spacecraft and reweigh. h) Use this new data to compute the center of gravity along the third spacecraft axis. i) Remove PTM from mass properties fixture and install on interim fixture. j) The moments of inertia will be measured on the PTM spacecraft only.	Assembly and Checkout Facility	5016, 5006, 5007, 5008, 5017, 5806, 5026, 6103, 6540, 6541	X		X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
72	Deleted								
73	Deleted								
74	Move Spacecraft to Alignment Facility a) Install spacecraft on vertical checkout and assembly stand. b) Install test capsule. 1) Position capsule in transporter adjacent to vertical assembly and checkout stand. 2) Attach capsule lift sling to hydraset and attach to capsule. 3) Lift capsule and mate to spacecraft. c) Attach prime mover and tow into alignment facility d) Detach prime mover.	Alignment Facility	5003, 5001, 5006, 5007, 5008, 5806, 5026, 5010, 5011, 5012, 5040, 5002, 6103, 6540, 6541	X	X	X	X	X	X
75	Perform Alignment Test and Calibrations The following will be mechanically aligned within specification and the electrical alignment compensated to reflect the true mechanical alignments: a) Spacecraft coordinate system b) All antennas c) All booms and latches d) All experiment sensors e) All guidance and control gas jets f) Retro propulsion engine nozzle g) All capsule alignments h) Sun sensors i) Canopus sensor j) Earth sensor The electrical alignments will be made using complete end-to-end testing. While the spacecraft is operated over the complete mission profile, all attitude sensors will reflect appropriate sensitivity during each phase of the mission profile. Alignment indicators will be installed on all alignment interfaces. The spacecraft will be moved to a given angle and its position optically checked. The gyros will then be torqued until the electrical null is reached. The angle through which the gyro moved will then be compared with the spacecraft position. The sun, earth and Canopus sensors will be checked by moving a calibrated stimulus through optically measured angles and checking system response.	Alignment Facility	5014, 5015, 5003, 5007, 5008, 6103, 5806, 5010, 3000, 4100, 4200, 6540, 6541	X	X	X	X	X	X
76	Move Spacecraft to Vibration Facility a) Position work stands, remove test cables. b) Remove dummy arrays.	Combined Vibration/ Acoustics Facility	5003, 5007, 5008, 5002, 5010, 5802-5809, 5006, 5026, 6103, 5019, 6540, 6541	X		X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	c) Install live solar arrays. d) Attach sling to crane and spacecraft. e) Hoist spacecraft, move to vibration facility, mate to vibration fixture.								
76a	Perform Spacecraft-Facility Compatibility Checks a) Verify mechanical fit of spacecraft and vibration fixture b) Verify handling procedures. c) Connect test cables and perform abbreviated functional test to verify facility compatibility.	Combined Vibration/ Acoustics Facility	5019, 5006, 5007, 5008, 5010, 5002,	X					
77	Deleted								
78	Deleted								
79	Perform Leak Tests a) Connect pneumatic and propellant fill and drain consoles to spacecraft. b) Pressurize spacecraft pneumatic and appendage release systems, fill propellant system with inert fluid and check for leaks. c) Disconnect and remove fill and drain consoles.	Combined Vibration/ Acoustics Facility	5705, 5007, 5008, 5010, 3900, 5019,			X	X	X	X
80	Perform Deployment Tests Each spacecraft appendage will be deployed in a simulated zero g field and observed to see that each appendage freely deploys, with no mechanical resistance or cable chafing due to electrical cables, mechanical failure, or misalignment.	Combined Vibration/ Acoustics Facility	5007, 5008, 5010, 5019,	X		X	X	X	X
81	Install Flight Ordnance a) At the safe-arm "J" box check that no voltage exists across any wire going to the ordnance devices to frame ground, using a range approved voltmeter. b) Arm the safe-arm "J" box and ascertain that battery voltage exists where it should and zero volts exist at the remaining pins of each connector. c) "Safe" the safe-arm "J" box and install all ordnance. d) At the safe-arm "J" box check that zero ohms exist across each ordnance bridgewire lead to frame ground using a range approved milliohmeter. e) At the safe-arm J-box determine that continuity exists through each ordnance bridgewire using an approved milliohmeter.	Combined Vibration/ Acoustics Facility	6202, 6203, 5007, 5008, 5010, 5019,			X	X	X	X
82	Deleted								
83	Perform IST and Critique This test is designed to test the entire spacecraft to the fullest extent to determine satisfactory system operation. The test will be performed in as close to a to a flight configuration as possible with only those hardlines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. This test will define the pre-dynamic environment baseline system operation. The test will include the following operations:	Combined Vibration/ Acoustics Facility	5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 6518, 5010, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6540, 6541, 3000, 4200, 4100, 5019			X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft. b) Remove all protective devices. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operational check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configuations and power levels the spacecraft will see in flight. g) Perform a check of all ordnance circuits. h) Vary the up and down RF link levels. i) Vary all test stimuli over the expected flight ranges. j) Perform a simulated countdown including umbilical separation. All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft test record. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer will sign off the IST data.								
84	Perform Combined Vibration/Acoustics Tests The purpose of this vibration/acoustic test is to demonstrate the capability of the planetary vehicle to withstand the mission vibration environments. It is expected that these environments will consist of sinusoidal and random inputs that could occur during the launch boost phase and the spacecraft retro propulsion phase of the mission sequence. All protective devices will be removed and ordnance will be armed. The vibration test will be performed as follows: a) Calibrate accelerometers b) Start vibrating spacecraft and search for mechanical resonances and amplifications. c) Perform sinusoidal vibration tests. d) Perform random vibration test. e) Repeat items b through d for each axis. The spacecraft will be electrically powered and all pneumatics lines and fuel vessels will be filled. The hydraulic shaker will perform vibrations to 600 cps. Exposure to frequencies above 600 cps will be accomplished acoustically.	Combined Vibration/ Acoustics Facility	5019, 5007, 5008, 5010, 3000, 4100, 4200			X	X	X	X
85	Remove Test Capsule a) Position work stands. b) Safe ordnance, install protective covers c) Attach capsule lift sling to hydraset and capsule d) Lift capsule and install in capsule transporter.	Combined Vibration/ Acoustics Facility	5006, 5007, 5008, 5010, 5011, 5012, 5040, 5019			X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT				
				EM	PIM	PTM	FL1	FL2	FL3
86	Perform Combined Vibration/Acoustics Test The purpose of this vibration/acoustic test is to demonstrate the capability of the spacecraft to withstand the mission vibration environments. It is expected that these environments will consist of sinusoidal and random inputs that could occur during the launch boost phase and the spacecraft retro propulsion phase of the mission sequence. All protective devices will be removed and ordnance will be armed. The vibration/acoustic test will be performed as follows: a) Calibrate accelerometers b) Start vibrating spacecraft and search for mechanical resonances and amplifications. c) Perform sinusoidal vibration test d) Perform random vibration test e) Repeat items b through d for each axis. The spacecraft will be electrically powered and all pneumatics lines and fuel vessels will be filled. The hydraulic shaker will perform vibrations to 600 cps. Exposure to frequencies above 600 cps will be accomplished acoustically.	Combined Vibration/ Acoustics Facility	5019, 5007, 5008, 3000, 4100, 4200,			X	X	X	X
87	Install Spacecraft in Vertical Stand a) Position work stands and remove test cables. b) Install all protective devices. c) Position vertical checkout and assembly stand in test area. d) Attach spacecraft lifting sling to hydraset and to spacecraft ring adapter. e) Hoist spacecraft and install on vertical checkout and assembly stand.	High bay area with crane	5003, 5006, 5007, 5008, 6103, 5002, 5805, 6540, 6541, 5019			X	X	X	X
87a	Install Test Capsule a) Position capsule in transporter adjacent to vertical checkout and assembly stand. b) Attach capsule lift sling to hydraset and attach to capsule. c) Lift capsule and mate to spacecraft.	High bay area with crane	5006, 5007, 5008, 5010, 5011, 5012, 5040, 6103, 5805, 5003, 6540, 6541, 6004, 6001			X	X	X	X
88	Move Spacecraft to Shock Test Area a) Connect required test cables. b) Install shroud and ordnance devices. c) Install shroud-restraining OSE. d) Install environmental cover. e) Attach prime mover and tow vertical checkout and assembly stand to shock test area.		5001, 5003, 5007, 5008, 5010, 5805, 5026, 6103, 6540, 6541			X			
89	Perform Spacecraft Functional Test This test will be used as a post vibration/preshock baseline to determine system degradation and damage. a) Position work stands.	Shock test area	5003, 5007, 5008, 5010, 3000, 4100, 4200, 6540, 6541			X			

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	b) Attach all required test cables and set up OSE. c) Perform a gross check of the spacecraft to verify proper operation of the spacecraft and the STC.								
90	Perform Shock Test a) Arm all shroud ordnance. b) Fire ordnance. c) Attach prime mover to vertical checkout and assembly stand and tow to assembly and checkout area. d) Remove environmental cover. e) Remove nose fairing. f) Position work stands.	Shock test area	5003, 5007, 5008, 5010, 6103, 5805, 6540, 6541			X			
91	Deleted								
92	Deleted								
93	Perform Free Mode Test The solar array protective covers will be removed and the array stimulated with solar simulation light banks. All test cabling will be disconnected from the spacecraft. Test stimuli which has only a mechanical interface with the spacecraft can remain installed and will be used during this test. A complete mission profile will be performed with only RF control and monitoring capability. Eclipse periods will be simulated in order to verify battery operation. This test will also serve as the post dynamic environment system test which will verify survivability of the spacecraft.	Assembly and Checkout Facility	5003, 5007, 5008, 6103, 5010, 6004, 5023, 3000, 4100, 4200, 6540, 6541, 5805	X		X	X	X	X
94	Perform Leak Tests All guidance and control, appendage release and LEMDE pneumatics and propellant systems will be leak tested. The purpose of the test is to verify that the leak rates are within specification, and have not increased due to the dynamic environmental exposure. At the completion of the leak checks, the propellant system will be drained and the pneumatics lines and tanks depressurized.	Assembly and Checkout Facility	5007, 5008, 6103, 5010, 5805, 6004, 5705, 5003, 3900, 6540, 6541			X	X	X	X
95	Perform Deployment Tests This test will be performed to verify that no damage has occurred due to the vibration exposure. Each appendage will be deployed in a simulated zero g field observing that each appendage freely deploys, with no mechanical resistance due to absence of heat, no chafing of electrical cables, no mechanical failures or misalignments.	Assembly and Checkout Facility	5007, 5008, 6103, 5010, 5805, 5020, 6004, 5003, 6540, 6541				X	X	X	X
96	Move Spacecraft to Alignment Facility a) Disconnect all test cables and move work stands back. b) Attach prime mover and tow vertical checkout and assembly stand into alignment facility.		5003, 5006, 5007, 5008, 6103, 5010, 5805, 6004, 5026, 5001, 6540, 6541, 5001				X	X	X	X
97	Perform Spacecraft Functional Test a) Detach prime mover. b) Position workstands.	Alignment Facility	5007, 5008, 6103, 5010, 5805, 6004, 6540, 6541, 3000, 4100, 4200, 5003, 5001				X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	c) Attach all required test cables and set up OSE. d) Perform a gross check of the spacecraft to verify proper spacecraft/STC operation.								
98	Perform Alignment Tests and Calibrations The following shall be mechanically aligned within specification and the electrical alignment compensated to reflect the true mechanical alignments. These tests are performed to verify that the alignments have not shifted due to the dynamic environmental exposure. a) Spacecraft coordinate system b) All antennas c) All booms and latches d) All experiment sensors e) All guidance and control gas jets f) Retro propulsion engine nozzle g) All capsule alignments h) Sun sensors i) Canopus sensor j) Earth sensor The electrical alignments shall be made using complete end-to-end testing. While the spacecraft is operated over the complete mission profile, all attitude sensors shall be checked for appropriate sensitivity during each phase of the mission profile. Alignment indicators shall be installed on all alignment interfaces. The spacecraft will be moved to a given angle and its position optically checked. The gyros will then be torqued until the electrical null is reached. The angle through which the gyro moved will then be compared with the spacecraft position. The sun, earth and Canopus sensors will be checked by moving a calibrated stimulus through optically measured angles and checking system response.	Alignment Facility	5003, 5014, 5015, 5007, 5008, 6103, 5010, 5805, 6004, 6541, 3000, 4100, 4200, 6540			X	X	X	X
99	Move Spacecraft to Thermal Vacuum Facility a) Attach spacecraft to planetary vehicle inverter. b) Turn spacecraft over such that the solar array is up. c) Hoist spacecraft and move with crane to thermal vacuum chamber. d) Install on thermal vacuum fixture and remove sling.		5009, 5007, 5008, 5805, 5021, 5022, 6103, 5010, 6004, 6540, 6541, 5026, 5013, 5002	X		X	X	X	X
99a	Perform Spacecraft-Facility Compatibility Checks a) Perform mechanical fit check of thermal vacuum fixture. b) Verify all handling procedures. c) Connect all test cables. d) Perform a spacecraft functional test with spacecraft inside the chamber and the chamber closed to verify spacecraft-chamber-STC compatibility.	Space simulation chamber	5007, 5008, 5010, 5021, 5022	X					
100	Deleted								

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
101	Perform IST and Critique This test is designed to test the entire spacecraft to the fullest extent to determine satisfactory system operation. The test will be performed in as close to a flight configuration as possible with only those hardlines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. This test will define the pre space simulation system baseline. The test will include the following operations: a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft. b) Remove all protective devices. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operational check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Perform a check of all ordnance circuits. h) Vary the up and down RF link levels. i) Vary all test stimuli over the expected flight ranges. j) Perform a simulated countdown including umbilical separation. All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft test record. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer will sign off the IST data.	Space simulation chamber	5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 6518, 5022, 5010, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6541, 3000, 4200, 4100, 5021, 6540			X	X	X	X
102	Perform Thermal Vacuum Test a) All test cabling, stimuli and OSE will have been completely checked out by the preceding system test. b) All thermal control and monitoring equipment will be installed and checked out prior to the system test. c) Some test cabling, in addition to that normally used in the Integrated System Test, will be connected to individual black boxes such that the ability to isolate faults to the provisional spares level is definitely established. d) A special mission profile will be developed which will exercise the system completely in each of its expected thermal environments. e) Complete, detailed checks will not be made. The emphasis will be on end-to-end testing and system reaction to gross stimulation. This is necessary due to the limited control over mechanical stimulus placed in the chamber.	Space simulation chamber	5010, 5021, 5022, 3000, 4100, 4200			X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	f) The spacecraft will be powered by stimulating the solar array with the solar simulator to the maximum extent possible. The solar simulator will be programmed for varying intensity over the mission profile. g) The thermal control equipment will be varied over the mission profile for simulated thermal environment. Special thermal instrumentation will be used to verify the design of the thermal control subsystem. h) The mission profile will include simulation of Mars orbital conditions to verify that the planetary vehicle can survive orbital thermal environments if the capsule is not released.								
103	Remove Test Capsule a) Open thermal vacuum chamber and traverse fixture and spacecraft outside chamber. b) Attach crane and install spacecraft on inverting fixture. c) Turn spacecraft right side up. d) Attach capsule lift sling and hydraset to crane hook and connect sling to capsule. e) Hoist capsule and install on transporter. f) Turn spacecraft upside down. g) Attach crane and install spacecraft on chamber fixture. h) Reinstall spacecraft in chamber.	Space simulation chamber	5007, 5008, 5021, 5010, 5011, 5012, 5040, 5006, 5013, 6004, 6001			X	X	X	X
104	Perform Thermal Vacuum Test This test will be performed to verify that the spacecraft can perform the required portions of the mission without the capsule. This will include simulation of boost and cruise phases as well as Mars orbit conditions. a) A special mission profile will be developed which will exercise the system completely in each of its expected thermal environments. b) Complete, detailed checks will not be made. The emphasis will be on end-to-end testing and system reaction to gross stimulation. This is necessary due to the limited control over mechanical stimulus placed in the chamber. c) The spacecraft will be powered by stimulating the solar array with the solar simulator to the maximum extent possible. The solar simulator will be programmed for varying intensity over the mission profile. d) The thermal control equipment will be varied over the mission profile for simulated thermal environment. Special thermal instrumentation will be used to verify the design of the thermal control subsystem.	Space simulation chamber	5021, 3000, 4100, 4200			X	X	X	X
105	Install Test Capsule a) Open thermal vacuum chamber and traverse fixture and spacecraft outside chamber. b) Attach crane and install spacecraft on inverting fixture.	Space simulation chamber	5006, 5007, 5008, 5021, 5010, 5011, 5012, 5040, 5013, 6004, 6001			X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	c) Turn spacecraft right side up. d) Position work stands around spacecraft. e) Attach capsule lift sling and hydraset to crane hook and connect sling to capsule. f) Hoist capsule and install on spacecraft; remove sling.								
106	Perform IST and Critique This test is performed as a complete check of the spacecraft to determine any damage or degradation caused by the thermal vacuum environmental exposure. The system test is designed to test the entire spacecraft to the fullest extent to determine satisfactory system operation. The test will be performed in as close to a flight configuration as possible with only those hardlines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft. b) Remove all protective devices. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operational check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Perform a check of all ordnance circuits. h) Vary the up and down RF link levels. i) Vary all test stimuli over the expected flight ranges. j) Perform a simulated countdown including umbilical separation. k) Install all protective devices.	Space simulation chamber	5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 5010, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000, 4200, 4100, 5021, 5022, 6540, 6541, 6004, 6001			X	X	X	X
107	Move Spacecraft to Assembly and Checkout Area a) Disconnect all test cables, remove work stands. b) Attach spacecraft sling and hydraset to crane hook and connect sling to spacecraft. c) Hoist spacecraft and move by crane to assembly and checkout area. d) Install in vertical checkout and assembly stand; remove sling.		5003, 5006, 5007, 5008, 6103, 5010, 5805, 5026, 5013, 5002, 6540, 6541				X	X	X	X
107a	Perform Simulated Stack Test This test will be a check of two spacecraft with test capsules, operating in close proximity. The effects of transmitting commands to both spacecraft simultaneously will be investigated. Mutual interference and compatibility will be checked. The capability to recieve telemetry and obtain ranging information simultaneously from two spacecraft will be verified.	Assembly and Checkout Facility	5003 (2), 5008 (2), 6103 (2), 5010, 5805 (2), 5007 (2), 5026 (2), 3000 4100, 4200, 5002, 6540, 6541				X	X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
108	Perform Practice Countdown This test will be run with two spacecraft and will simulate the actual countdown operation. Control and monitoring of the spacecraft will be through the umbilical and the RF link only. The spacecraft will be checked out serially. However, while one spacecraft is being tested, the other will be in launch configuration with the STC monitoring its status. This test will serve to debug procedures and operations as well as to further verify spacecraft integrity. A check of the shroud EOSE will be made at this time.	Assembly and Checkout Facility	5003 (2), 5008 (2), 6103 (2), 5010, 5805 (2), 5007 (2), 3000, 4100, 4200, 4000, 5026 (2), 6540, 6541	X	X	X	X	X	X
109	Perform Mission Operations Test This test will be the preshipment system test and as such will be a complete check of the system. a) A complete mission profile will be performed. b) All data will be transmitted to the SFOF via data link. c) Data analysis teams will participate at SFOF. d) Magnetic tapes will be made for checkout use at all tracking stations. Final buyoff will constitute approval to ship the spacecraft.	Assembly and Checkout Facility	5003, 5008, 6103, 5010, 5805, 5007, 3000, 4000, 4100, 4200, 6540, 6541	X		X	X	X	X
109a	Prepare Spacecraft and Test Capsule for Shipment a) Remove capsule and install in transporter. b) Position spacecraft transporter and cover adjacent to vertical checkout and assembly stand with prime mover. c) Attach spacecraft sling and hydraset to crane hook and connect sling to spacecraft. d) Hoist spacecraft and install on transporter, remove sling. e) Attach cover sling to hydraset and connect sling to cover. f) Hoist cover and install on transporter, remove sling. g) Move transporter into airlock with prime mover. Install spacecraft in shipping container.	Assembly and Checkout Facility	5006, 5007, 5008, 6103, 5017, 5058, 5059, 5026, 5010, 5011, 5012, 5040, 5805, 5002, 5001, 6540, 6541	X		X			
110	Move Spacecraft Test Capsule to Goldstone a) Connect helicopter transporter to spacecraft transporter and proceed to DSIF facility.		5058, 5059, 6103, 5805, 6540, 6541, 5017, Helicopter: 5056, 5055, 3000, 4100, 4200, 5010, 5011, 5012	X		X			
111	Assemble Spacecraft on Vertical Test Stand a) Position transporter next to vertical checkout and assembly stand. b) Attach cover sling and hydraset and connect sling to cover. c) Hoist cover and set aside; remove sling. d) Attach spacecraft sling to hydraset and connect sling to spacecraft.	Goldstone	5054-5056/5041-5048, 5041-5045, 5047, 5048/ 5056, 5055, 5003, 5002, 5001, 5007, 5008, 5010, 5011, 5012, 5040, 6540, 6541, 5026, 6103, 5805	X		X			

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	e) Hoist spacecraft and mate to fixture, remove sling. f) Position work stands around spacecraft. g) Install test capsule.								
112	Perform Spacecraft Functional Test a) Connect test cables to spacecraft. b) Set up and validate STC. c) Perform a gross check of the spacecraft to verify proper spacecraft-STC operation.	Goldstone	5003, 5008, 6103, 5010, 5805, 5007, 5006, 3000, 4100, 4200, 6540, 6541	X		X			
113	Perform DSIF Compatibility Tests These tests are described in detail in the mission support section of this volume. Briefly they include the following: a) Check MDE-DSIF compatibility. b) Check compatibility of the major elements of the SFOF. c) Provide crew training for DSIF personnel. d) Debug mission prodecures and operations.	Goldstone	5003, 5008, 6103, 5010 5805, 5006, 3000, 4100, 4200, 5007, 6540, 6541	X		X			
114	Prepare Spacecraft and Test Capsules for Shipment a) Remove test capsule and install in shipping container. b) Position work stands around spacecraft. c) Remove all test cables. d) Position spacecraft transporter and cover next to fixture. e) Attach spacecraft sling and hydraset to crane hook and connect sling to spacecraft. f) Hoist spacecraft and mate to transporter, remove sling. g) Attach spacecraft transporter cover sling to hydraset and connect sling to cover. h) Hoist cover and install on transporter; remove sling. i) Connect helicopter transporter to spacecraft transporter and proceed to airfield.	Assembly and Checkout Facility	5006, 5007, 5008, 6103, 5017. 5058, 5059, 5026, 5010, 5011, 5012, 5040, 5805, 5002, 5001, 6540, 6541	X	X		X	X	X
115	Move Spacecraft and Test Capsule to Booster Compatibility Area a) Install on vertical checkout assembly stand. b) Install capsule. c) The spacecraft will be moved into an area where it can be mated to the shroud and to the S-IVB instrumentation unit or simulator.	Assembly and Checkout Facility	Helicopter: 5056, 5055, 5058, 5059, 6103, 5805, 6540, 6541, 5017, 5003, 5002, 5001, 5007, 5008, 5010, 5011, 5012, 5040				X		
116	Perform Spacecraft-Booster Compatibility Test a) Mate planetary vehicle adapter to shroud. b) Mate Spacecraft to planetary vehicle adapter. c) Make all electrical connections.	Assembly and Checkout Facility	5035, 5009, 5028, 5029, 5030, 5031, 5010, 3000, 4000, 4200, 5002				X		

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	d) Install shroud seals. e) Mate shroud to instrumentation unit or simulator. f) Make all electrical connections. g) Connect umbilical to LCE and spacecraft. Use actual on-stand cable lengths. h) Perform gross system checks to verify OSE/ spacecraft operations. i) Verify all electrical and mechanical interfaces. j) Check antenna system VSWR.								
117	Move Spacecraft to Life Test Area a) Remove shroud seals. b) Install spacecraft lifting sling and lift spacecraft from shroud. c) Install spacecraft in vertical checkout and assembly stand. d) Remove planetary vehicle adapter from shroud. e) Attach prime mover and tow spacecraft to life test cell.	Assembly and Checkout Facility	5003, 5008, 5009, 6103, 5010, 5805, 5001, 5026, 5002, 5028, 5029, 5030, 5031, 5007, 6540, 6541			X			
118	Perform a 10-Week Life Test This test will be performed in an area where the spacecraft external ambient temperatures can be kept at approximately 60°C. The test will run continuously for 1750 hours. Life testing of the solar array will not be included in this test but will be performed separately. A mission sequence of events will be programmed which will provide complete end-to-end testing of the system. The sequence will be repeated at an event rate of 20 times expected mission usage. Testing of redundant components will be included to the maximum extent possible. "One shot" devices will not be tested. The sequence will be programmed such that night shift operations can be performed with a minimum test crew. Spacecraft failure will be defined as follows: a) A critical function fails and cannot be restored by ground command or redundant system operation. b) Premature wearout is observed in depletable sources of energy or material. (It may be required to perform this test in a vacuum chamber).	Assembly and Checkout Facility	5010, 5003, 5008, 5007, 3000, 4100, 4200			X			
119	Relay Link Test A detailed test of the capsule-spacecraft relay link will be made in which antenna pattern of both receiver and transmitter will be mapped. An analysis of the data will verify the capability of the relay link to provide capsule-spacecraft communications.	Antenna test range	5010, 5805, 5003, 5008, 5007, 6103, 3000, 4100, 4200, 6540, 6541			X			
120	Move Spacecraft and Test Capsule to Cape a) Remove helicopter/transporter from spacecraft transporter. b) Position cargo lift trailer end-to-end with spacecraft transporter and secure together. c) Transfer spacecraft on transporter sliding bed to the cargo lift trailer; disconnect transporter from trailer. d) Move trailer into position and transfer spacecraft to aircraft. e) Install test capsule transporter in aircraft.	Air Terminal	5049, 5050, 5051, 5052, 5053, 5058, 5059, 5805, 6103, 5007, 5008, 6201, 5010, 5011, 5012, 5040, 5033, 6540, 6541, 5017, 5055, 5056, 5048	X	X		X	X	X
121	Receive STC at KSC Air Strip a) Open aircraft and remove STC packing container tie-downs b) Remove containers from aircraft with fork-lift and place on truck.	KSC Air Strip	3000, 4000						

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
122	Transport STC to ESA a) Form convoy and proceed to ESA		3000, 4000						
123	Perform Receiving Inspection a) Place container in ESA airlock b) Open containers and perform receiving inspection c) Inspection will check for damage that might have occurred during shipping and handling operations	Explosive safe area	3000, 4000						
124	Perform STC Validation a) Set up equipment in ESA b) Interconnect all STC cabling. c) The STC will be functionally tested and proper operation verified.	Explosive safe area	3000, 4000						
125	Receive Spacecraft and Test Capsule at KSC Air Strip a) Unbutton aircraft and remove spacecraft transporter pallet tie-down b) Position cargo lift trailer in-line with aircraft and unload spacecraft transporter pallet c) Lower lift trailer scissor bed and align spacecraft transporter bed in-line with trailer d) Transfer spacecraft transporter pallet on to spacecraft transporter bed. e) Remove test capsule transporter and place on truck bed.		5049, 5050, 5051, 5052, 5053, 5058, 5059, 5805, 6103, 5007, 5008, 6201, 5010, 5011, 5012, 5040, 5033, 6540, 6541, 5017, 5048	X	X		X	X	X
126	Transport Spacecraft and Test Capsule to SAF a) Form convoy and proceed to SAF		5048, 5058, 5059, 5010, 5040, 5012, 5011, 6540, 6541, 6103, 5805	X	X		X	X	X
127	Perform Spacecraft Receiving Inspection a) Place spacecraft transporter in SAF airlock and remove prime mover b) Clean all exterior surfaces c) Attach spacecraft transporter cover lift sling and hydraset on crane hook and connect sling to cover d) Hoist cover and set aside; remove sling e) Attach spacecraft lift sling to crane hook and connect sling to spacecraft f) Hoist spacecraft and attach to vertical assembly and checkout stand, remove sling g) Position work stands around spacecraft h) Perform receiving inspection for any damage and/or contamination that might have occurred during shipping and handling operations i) Any damage and/or contamination will be corrected at this time	Spacecraft Assembly Facility	5048, 5006, 5058, 5059, 5026, 6540, 6541, 5002, 5003, 5002, 5001, 5007, 5008	X	X		X	X	X
128	Delete								

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
129	Install Test Capsule a) Position capsule in transporter adjacent to vertical assembly stand b) Attach capsule lift sling to hydraset and attach to capsule c) Lift capsule and mate to spacecraft	Spacecraft Assembly Facility	5001, 5003, 5006, 5007, 5008, 6103, 5010, 5011, 5012, 5805, 5040, 6540, 6541	X	X		X	X	X
130	Perform Spacecraft Functional Test a) Connect EOSE cables to spacecraft b) A functional checkout will be performed to verify proper spacecraft-STC operation.	Spacecraft Assembly Facility	5003, 5007, 5008, 6103, 5010, 5805, 3000, 6540, 6541	X	X		X	X	X
131	Perform Battery Capacity Tests a) The batteries will be fully discharged b) Charge slowly at a constant rate until batteries reach maximum charge c) Discharge slowly at a constant rate until fully discharged d) Charge slowly at a constant rate until batteries reach maximum charge	Spacecraft Assembly Facility	5003, 5007, 5008, 6103 5010, 5805, 3000, 6540, 6541	X	X		X	X	X
132	Perform Solar Array Tests a) Set up solar array checkout kit b) Remove solar array protective covers c) Perform inverse impedance test on each solar array panel d) Illuminate each array panel and measure the open circuit voltage and short circuit current e) Remove solar array checkout kit f) Install solar array protective covers	Spacecraft Assembly Facility	5003, 5005, 5007, 5008, 6103, 5010, 5805, 5807, 3000, 6540, 6541, 5802	X	X		X	X	X
133	Perform IST and Critique This test is designed to test the entire spacecraft to the fullest extent to determine satisfactory system operation. The test will be performed in as close to a flight configuration as possible with only those hard-lines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. This test will verify that no damage has occurred during shipping. The test will include the following operations: a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft. b) Remove all protective devices. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operation check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Perform check of all ordnance circuits	Spacecraft Assembly Facility	5003, 5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 5010, 6103, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 6001, 3000	X	X		X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	h) Vary the up and down RF link levels. i) Vary all test stimuli over the expected flight ranges. j) Perform a simulated countdown including unbilical separation. k) Install all protective devices. All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft record. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer will sign off the IST data.								
134	Remove Test Capsule a) Position work stands b) Attach capsule lift sling to hydraset and capsule c) Lift capsule and install in capsule transporter	Spacecraft Assembly Facility	5001, 5003, 5006, 5007, 5008, 6103, 5010, 5011, 5012, 5805, 5040, 6540, 6541	X	X		X	X	X
135	Prepare Spacecraft for Movement Outside SAF a) Remove all test cables b) Position spacecraft transporter and cover adjacent to vertical assembly and checkout stand with prime mover c) Attach spacecraft transporter cover lift sling and hydraset to crane hook and connect sling to cover d) Hoist cover and set aside; remove sling e) Attach spacecraft lift sling to Hydraset and connect sling to spacecraft. f) Hoist spacecraft and install in transporter; remove sling g) Attach spacecraft transporter cover lift sling to hydraset and attach to cover h) Hoist cover and install on transporter; remove sling i) Tow transporter into airlock	Spacecraft Assembly Facility	5001, 5003, 5006, 5007, 5008, 6103, 5805, 5026, 5048, 6540, 6541, 5058, 5059, 5002	X	X		X	X	X
136	Move Spacecraft to ESA a) Connect road tractor to spacecraft transporter and remove from airlock b) Form convoy and proceed to ESA c) Place spacecraft transporter in ESA airlock and remove prime mover d) Clean all exterior surfaces e) Attach prime mover to spacecraft transporter and position next to vertical assembly and checkout stand; disconnect prime mover f) Attach spacecraft transporter cover lift sling and hydraset to crane, connect sling to cover g) Hoist cover and set aside; remove sling h) Attach spacecraft lift sling to hydraset and connect sling to spacecraft i) Hoist spacecraft and install on vertical assembly and checkout stand		5048, 5058, 5059, 5008, 5805, 6103, 5003, 5007, 6540, 6541, 5006	X	X		X	X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	j) Position work stands and remove sling k) Attach spacecraft transporter cover sling to hydraset and connect to cover l) Hoist cover and install cover on transporter; remove sling								
136a	Move Test Capsule to ESA a) Engineering model test capsule will move to ESA in same convoy as spacecraft.		5010, 5011, 5012, 5040, 5048	X	X				
137	Verify SCS Alignments a) Set up optical alignment instrument b) Set up component alignment instrument c) Connect EOSE checkout console cables to spacecraft d) The following shall be checked mechanically for alignment within specification and the electrical alignment compensated to reflect the true mechanical alignments: 1) Spacecraft coordinate system 2) All antennas 3) All booms and latches 4) All experiment sensors 5) Guidance and control gas jets 6) Retropropulsion engine nozzle 7) Canopus sensor 8) Sun sensor 9) Earth sensor The electrical alignments shall be made using complete end-to-end testing. While the spacecraft is operated over the complete mission profile, all attitude sensors will be checked for appropriate sensitivity during each phase of the mission profile. Alignment indicators will be installed on all alignment interfaces. The gyros will be torqued to a given angle and the spacecraft then moved to null. The movement of the spacecraft to the null position will be optically checked. The sun and Canopus sensors will be checked by moving calibrated stimuli to known optical angles and checking system response.	Explosive safe area	5007, 5008, 6103, 5003, 5014, 5015, 5805, 6540, 6541, 3000	X	X		X	X	
138	Perform Spacecraft Weight and Center of Gravity Tests a) Disconnect all test equipment b) Attach weight and center of gravity sling and hydraset to crane hook and connect sling to spacecraft c) Remove all protective covers d) Hoist spacecraft e) Weigh the spacecraft with load cells in three places f) Using this weight data compute the center of gravity in two of the spacecraft axes g) Tilt the spacecraft and reweigh h) Use this new data to compute the center of gravity along the third spacecraft axis	Explosive safe area	5007, 5008, 6103, 5016, 5805, 5003, 5006, 3000, 5002, 6540, 6541	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	i) Traverse crane and install spacecraft on vertical assembly and checkout stand j) Position work stands around assembly and check-out stand and remove sling k) Replace all protective covers								
139a	Perform Separation and Release Test a) Connect test cables to spacecraft b) Connect pneumatic console to spacecraft c) Connect appendage deployment equipment to spacecraft d) Each spacecraft appendage will be released in a simulated zero g field using live ordnance, observing that each appendage freely releases with no mechanical restriction or cable chafing due to electrical cables, mechanical failure, or misalignment. e) Restore all appendages to flight configuration	Explosive safe area	5003, 5007, 5008, 6103, 6502, 5805, 5020, 5705, 3000, 6540, 6541	X	X			X	X
139	Install Ordnance Simulators a) Draw ordnance simulators from stores b) Test simulators with ordnance checkout kit c) Install ordnance simulators	Explosive safe area	5003, 5007, 5008, 6103, 5805, 6201, 6540, 6541	X	X			X	X
140	Install Flight Capsule a) Position capsule transporter next to vertical assembly and checkout stand with prime mover b) Attach capsule transporter cover sling and hydraset to crane hook, connect sling to cover c) Hoist cover, set aside d) Attach sling to capsule, hoist and mate to spacecraft, remove sling	Explosive safe area	5001, 5003, 5006, 5007, 6103, 5008, 5011, 5012, 5805, 5040, 5026, 6540, 6541					X	X X
140a	Install Test Capsules a) Position capsule transporter next to vertical assembly and checkout stand with prime mover b) Attach capsule transporter cover sling and hydraset to crane hook, connect sling to cover c) Hoist cover, set aside d) Attach sling to capsule, hoist and mate to spacecraft, remove sling Note: Test capsules will be used.	Explosive safe area	5010, 5001, 5003, 5006, 5007, 6103, 5008, 5011, 5012, 5805, 5040, 5026, 6540, 6541	X	X				
141	Perform Spacecraft-Capsule Compatibility Test a) Connect EOSE checkout console cables to spacecraft b) Check all signal line voltages and currents. Noise and transients levels must be within specified levels. c) Check that spacecraft radio subsystem does not interfere with the capsule and that the capsule does not interfere with the spacecraft d) Check the spacecraft-capsule operational communications e) Check all telemetry channel calibrations and outputs	Explosive safe area	5003, 5007, 5008, 6103, 5805, 3000, 6540, 6541,					X	X X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
142	Perform IST and Critique This test is designed to test the entire spacecraft to its fullest extent to determine satisfactory system operations. The test will be performed in as close to a flight configuration as possible with only those hardlines connected which are required to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect pneumatic, propulsion fill and drain, and STC OSE to spacecraft b) Remove all protective devices c) Perform an operational check of all subsystems d) Perform an operational check of all science instruments e) Perform an operation check of all spacecraft/capsule interfaces f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Perform check of all ordnance circuits h) Vary the up and down RF link levels i) Vary all test stimuli over the expected flight ranges j) Perform a simulated countdown including umbilical separation k) Install all protective devices All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft test record. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer will sign off the IST data.	Explosive safe area	5003, 5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 6521, 6103, 6522, 6523, 6524, 6527, 6525, 6526, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 6001, 3000	X	X		X	X	X
143	Perform Science Quiet Test a) This test will verify proper operation of science instruments. It will be accomplished with and without stimuluation to instruments. The test will be a modified system test. b) The modified system test will check all spacecraft operating modes with spacecraft equipment operating as appropriate. The interface between spacecraft and capsule experiments will be checked. All science instruments will be monitored for interference. c) The testing is conducted so as to minimize interference from facility activities, vehicles, personnel, etc.	Explosive safe area	5003, 5007, 5008, 6508, 6511, 6512, 6515, 6516, 6517, 6521, 6103, 6522, 6523, 6524, 6527, 6525, 6526, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 6001, 3000	X	X		X	X	X
144	Perform Spacecraft Capsule Vertical Alignment a) Set up optical alignment instrument b) Check alignment	Explosive safe area	5003, 5007, 5008, 6103, 5014, 5015, 5805, 6540, 6541, 5010	X	X		X	X	X
145	Perform Final In-Hangar Button-Up Operations a) Remove all test equipment from spacecraft b) Perform final design engineering inspection	Explosive safe area	5003, 5007, 5008	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
146	Mate Spacecraft to Flight Shroud a) Draw shroud sections and diaphrams and transporters from stores, move into ESA air-lock, remove prime mover. b) Clean all exterior surfaces c) Connect shroud and install lower diaphram. d) Attach spacecraft shroud to sterilization pressure dome. e) Attach spacecraft lift sling and hydraset to crane hook and connect sling to spacecraft. f) Hoist spacecraft and mate to shroud. g) Assemble upper shroud diaphram. h) Assemble work stands around unit.	Explosive safe area	5001, 5003, 5006, 5007, 5008, 5029, 5028, 5031, 5030, 5024, 5025, 5002	X	X		X	X	
147	Perform Shroud Cooling System Test a) Connect Saturn instrumentation unit cooling system shroud sections. b) Test will verify ability of the booster cooling system to maintain the spacecraft environment in the shroud within specified limits.	Explosive safe area	5007, 5008, 5025	X	X		X	X	
148	Perform Modified IST and Compatibility Tests This test is designed to test the entire spacecraft to the fullest extent possible with the spacecraft in a flight configuration and enclosed with the shroud. Umbilical cabling will be connected to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect STC OSE to planetary vehicle. b) Perform an operational check of all subsystems. c) Perform an operational check of all science instruments. d) Perform an operational check of all spacecraft/capsule interfaces. e) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. f) Vary the up and down RF link levels. g) Vary all test stimuli over the expected flight ranges. h) Perform a simulated countdown including unbilical separation. All data from this test will be recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft test record. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer will sign off the IST data.	Explosive safe area	5007, 5008, 6515, 6516, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 6001, 5025, 3000	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
149	Perform Mock Countdown This test will be run with two spacecraft and will simulate the actual countdown operation. Control and monitoring of the spacecraft will be through the umbilical and the RF link only. The spacecraft will be checked out serially. However, while one spacecraft is being tested the other will be in launch configuration with the STC monitoring its status.	Explosive safe area	5007, 5008, 6515, 6516, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 6540, 6541, 6004, 6001, 5025, 3000	X	X		X	X	
150	Prepare Flight Shroud/Planetary Vehicle for Movement Outside of ESA a) Position flight shroud/planetary vehicle transporter and cover in ESA airlock with prime mover; remove prime mover. b) Clean chamber and transporter. c) Attach flight shroud/planetary vehicle lift sling and hydraset to crane hook and connect sling to flight shroud. d) Install flight shroud/planetary vehicle on transporter, remove sling. e) Attach flight shroud/planetary vehicle transporter cover sling to hydraset and connect to flight shroud/planetary vehicle cover. f) Hoist flight shroud/planetary vehicle cover and attach to transporter. g) Connect road tractor to transporter.	Explosive safe area	5027, 5028, 5029, 5030, 5048, 5007, 5008, 5006	X	X		X	X	
151	Move to VAB a) Form convoy and proceed to Vertical Assembly Building.		5048, 5027, 5029, 5030,	X	X		X	X	
152	Mate Flight Shroud/Planetary Vehicles to Launch Vehicle a) Position transporter under crane. b) Attach flight shroud/planetary vehicle cover sling and hydraset to crane hook and connect sling to flight shroud/planetary vehicle cover. c) Hoist flight shroud/planetary vehicle cover and set aside, remove sling. d) Attach flight shroud/planetary vehicle lift sling to hydraset and connect sling to flight shroud/planetary vehicle. e) Hoist 1st flight shroud/planetary vehicle and mate to booster. f) Hoist 2nd flight shroud/planetary vehicle and mate to booster. g) Hoist nose fairing and mate to booster. Note: The EM and PIM will use the launch vehicle facility verification missile for all VAB operations.	Vertical Assembly Building	5009, 5008, 5027, 5029, 5030, 5048	X	X		X	X	
153	Perform Flight Shroud-Launch Vehicle Alignments Spacecraft-launch vehicle alignment will be performed to insure that the launched vehicle coordinate axis corresponds to the spacecraft axis within the required accuracy.	Vertical Assembly Building	5024, 5015, 5007, 5008, 5026, 5009	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT EM	PIM	PTM	FL1	FL2	FL3
154	Perform Planetary Vehicle System Test This test is designed to test the entire spacecraft to the fullest extent possible with the planetary vehicles on top of the launch vehicle. Umbilical cabling will be connected to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include: a) Connect STC OSE to planetary vehicle. b) Perform operational check of sensor stimuli. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operation check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Vary the up and down RF link levels. h) Vary all test stimuli over the expected flight ranges. All data from this test is recorded on magnetic tape. This data will be processed and stored; it will become part of the spacecraft test record. The spacecraft will be checked out serially. However, while one spacecraft is being tested the other will be in flight configuration with the STC monitoring its status. Two STC's will be used, one for each spacecraft. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer signs off the data.	Vertical Assembly Building	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000, 4000	X	X		X	X	
155	Perform RFI Tests A combined vehicle RF interference test is performed to ascertain that none of the launch vehicle transmitters or beacons interfere with or degrade the spacecraft transmitters or receivers. Likewise, the test is also performed to ascertain that the spacecraft transmitters do not interfere with or degrade the launch vehicle beacons, transmitters or receivers. The RFI compatibility test is to be performed as follows: a) Each launch vehicle beacon and transmitter is turned on one at a time. The spacecraft will ascertain that there is no interference with or degradation of the receiver or transmitter systems. b) Each spacecraft transmitter is turned on one at a time. The launch vehicle will ascertain that there is no degradation of or interference with the receiver or transmitter systems. c) All spacecraft and launch vehicle transmitters are turned on together and each vehicle will ascertain that there are no mutual degradations or interference with the various transmitting or receiving systems. d) All range functions will be exercises. The spacecraft will verify there is no interference.	Vertical Assembly Building	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000, 4000	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
156	Perform Flight Readiness Demonstration Test As far as the spacecraft is concerned, the FRD test is identical to the countdown.	Vertical Assembly Building	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000, 4000	X	X		X	X	
157	Perform Mock Countdown Both planetary vehicles will participate in the mock countdown activities. The countdown will approximate the actual countdown. Prior to the conclusion of the countdown each subsystem of each vehicle will have been checked.	Vertical Assembly Building	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000, 4000	X	X		X	X	
158	Perform Simulated Flight Test a) An abbreviated simulated flight will be run to verify spacecraft command and response capability. b) Spacecraft umbilical cable spinoff squib simulators will be fired. c) Spacecraft separation squib simulators will be fired.	Vertical Assembly Building	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000, 4000	X	X		X	X	
159	Prepare Flight Shroud/Planetary Vehicle for Movement Outside VAB a) Demate planetary vehicle from launch vehicle. b) Place planetary vehicle on transporter.	Vertical Assembly Building	5027, 5028, 5029, 5030, 5048, 5007, 5008, 5006	X	X		X	X	
160	Move to ESA Form convoy and proceed to ESA		5027, 5029, 5048, 5030	X	X		X	X	
161	Flight Pressurize All Systems a) Remove upper shroud diaphrams from shroud. b) Pressurize all pneumatic systems.	Explosive safe area	5027, 5029, 5048, 5030 5007 (2) 5008 (2), 5009, 3000 (2), 4000 (2), 5025 (2)	X	X		X	X	
162	Install Flight Ordnance 1) At the safe-arm "J" box check that no voltage exists across any wire going to the ordnance devices to frame ground using a range approved voltmeter. 2) Arm the safe-arm "J" box and ascertain that battery voltage exists where it should and zero volts exists at the remaining pins of each connector. 3) "Safe" the safe-arm "J" box and install all ordnance. 4) At the safe-arm "J" box check that zero ohms exists across each ordnance bridgewire lead to frame ground using a range approved milliohmmeter. 5) At the safe-arm "J" box determine that continuity exists through each ordnance bridgewire using an approved milli-ohmmeter.	Explosive safe area	5025 (2), 5007, 5008, 6201, 6202, 6203, 6502, 3000 (2), 4000 (2)				X	X	
163	Load Spacecraft Propellants a) Connect propellant fill and drain console to spacecraft.	Explosive safe area	5007, 5008, 5025 (2), 6508, 6511, 6512, 6515, 6517, 6518, 6520, 6521				X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	b) Load propellants. Note: EM and PIM will be loaded with inert propellants.								
164	Perform Leak Tests The guidance and control, pyrotechnic and the liquid engine system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak rates are within specification and that no damage has been experienced due to handling of the spacecraft.	Explosive safe area	5705, 5025(2), 5008, 5007, 3000(2), 4000(2)	X	X		X	X	
165	Perform Integrated System Test and Critique This test is designed to test the entire spacecraft to the fullest extent possible with the spacecraft in a flight configuration and enclosed within the shroud. Umbilical cabling will be connected to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect STC OSE to planetary vehicle. b) Perform operational check of sensor stimuli. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operational check of all spacecraft/capsule interfaces. f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Vary the up and down RF link levels. h) Vary all test stimuli over the expected flight ranges. i) Perform a simulated countdown including umbilical separation. All data from this test is recorded on magnetic tape. This data will be processed and stored, it will become part of the spacecraft test record. The spacecraft will be checked out serially. However, while one spacecraft is being tested the other will be in launch configuration with the STC monitoring its status. Two STC's will be used, one for each spacecraft. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer signs off the IST data.	Explosive safe area	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	
166	Perform Planetary Vehicle Sterilization a) Disconnect all test cables. b) Connect sterilization unit to spacecraft-shroud modules. c) The planetary vehicle will undergo surface sterilization which will be performed by soaking the planetary vehicle in an environment of ethyline oxide using the shroud as a sterilization container. At this point, the shroud will be sealed.	Explosive safe area	5007, 5008, 5034, 5025(2)		X	X		X	X

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
167	Prepare Planetary Vehicle for Movement Outside of ESA a) Disconnect sterilization unit from shroud. b) Evacuate the shroud. c) Purge the shroud with dry, sterile nitrogen for 8 hours. d) Position flight shroud/planetary vehicle transporter and cover in ESA airlock with prime mover, remove prime mover. e) Attach flight shroud/planetary vehicle lift sling and hydraset to crane hook and connect sling to flight shroud. f) Install flight shroud/planetary vehicle on transporter, remove sling. g) Attach flight shroud/planetary vehicle transporter cover sling to hydraset and connect to flight shroud/planetary vehicle cover. h) Hoist flight shroud/planetary vehicle cover and attach to transporter. i) Connect load tractor to transporter.	Explosive safe area	5009, 5007, 5008, 5027-5030, 5048	X	X		X	X	
168	Move Planetary Vehicle to Pad Form convoy and proceed to launch complex.		5027, 5029, 5030, 5048						
169	Mate Planetary Vehicle to Launch Vehicle a) Position transporter on pad under gantry crane. b) Attach transporter cover sling and hydraset to gantry crane hook and connect sling to cover. c) Hoist cover clear and set aside, remove sling. d) Attach shroud section lift sling to hydraset and connect sling to shroud section. e) Hoist 1st shroud section and mate to booster. f) Hoist 2nd shroud section and mate to booster. g) Hoist nose fairing section and mate to booster. Note: The EM and PIM will use the launch vehicle facility verification missile for all PAD operations.	PAD	5009, 5008, 5027, 5029, 5030, 5048	X	X		X	X	
170	Perform Flight Shroud-Launch Vehicle Alignment a) Set up optical alignment instrument. b) Align shroud axes to launch vehicle axes.	PAD	5024, 5015, 5007, 5008, 5026, 5009	X	X		X	X	
171	Perform Planetary Vehicle System Test and Critique This test is designed to test the entire spacecraft to the fullest extent possible with the spacecraft on top of the booster. Umbilical cabling will be connected to aid telemetry in fault isolation to the provisional spares level or to allow testing of redundant elements. The test will include the following operations: a) Connect STC OSE to planetary vehicle. b) Perform an operational check of sensor stimuli. c) Perform an operational check of all subsystems. d) Perform an operational check of all science instruments. e) Perform an operational check of all spacecraft/capsule interfaces.	PAD	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	f) Perform a mission sequence of events correlated with expected configurations and power levels the spacecraft will see in flight. g) Vary the up and down RF link levels. h) Vary all test stimuli over expected flight ranges. i) Perform a simulated countdown including umbilical separation. All data from this test is recorded on magnetic tape. This data will be processed and stored, it will become part of the spacecraft test record. The spacecrafts will be checked out serially. However, while one spacecraft is being tested the other will be in launch configuration with the STC monitoring its status. Two STC's will be used, one for each spacecraft. The critique will be a meeting of all cognizant personnel to discuss the results of this test. During this meeting, each subsystem engineer signs off the data.								
172	Perform RFI Tests A combined vehicle RF interference test is performed to ascertain that none of the launch vehicle transmitters or beacons interfere with or degrade the spacecraft transmitters or receivers. Likewise, the test is also performed to ascertain that the spacecraft transmitters do not interfere with or degrade the launch vehicle beacons, transmitters or receivers. The RFI compatibility test is to be performed as follows: a) Each launch vehicle beacon and transmitter is turned on one at a time. The spacecraft will ascertain that there is no interference with, or degradation of the receiver or transmitter systems. b) Each range beacon and transmitter is turned on one at a time and both the launch vehicle and the spacecraft will ascertain that there is no degradation of or interference with the receiver or transmitter systems. c) Each spacecraft transmitter is turned on one at a time and the launch vehicle will ascertain that there is no degradation of or interference with the receiver or transmitter systems. d) All spacecraft, booster, and launch vehicle transmitters are turned on together and each vehicle will ascertain that there are no mutual degradations or of interference with the various transmitting of receiving systems.	PAD	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	
173	Perform J-Fact Test The purpose of the J-FACT test is to checkout the post injection portions of the mission profile. a) The installation of the nose fairing separation squib simulators are included in the J-FACT test preparations. b) The following spacecraft related post launch functions will be monitored and checked. 1) Nose fairing separation	PAD	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PIM	PTM	FL1	FL2	FL3
	2) Launch vehicle umbilical separation 3) Spacecraft separation from the launch vehicle c) As the spacecraft itself does not control any of the above functions, the J-FACT test, as far as the spacecraft is concerned, will serve as a practice countdown.								
174	Perform Flight Readiness Demonstration Test As far as the spacecraft is concerned, the FRD test is identical to the countdown.	PAD	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	
175	Perform Mock Countdown The mock countdown will approximate the terminal countdown. The entire terminal countdown procedure will be conducted with the exception of lift-off.	PAD	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	
176	Perform Pre-Countdown	PAD					X	X	
177	Perform Terminal Countdown a) During terminal countdown, the launch vehicle is fueled and the gantry removed. b) This is a launch vehicle task.	PAD	5008, 5009, 6508, 6511, 6512, 6515, 6517, 6520, 6521, 6522, 6523, 6524, 6525, 6526, 6527, 6528, 6529, 6537, 6538, 3000(2), 4000(2)	X	X		X	X	
178	Lift Off						X	X	
179	Remove Planetary Vehicles from Launch Vehicle a) Position shroud transporter beneath Gantry Crane Hook on launch pad b) Connect transporter cover sling and hydraset to crane hook and connect sling to cover c) Hoist cover and set aside, remove sling d) Attach shroud lift sling to hydraset and connect sling to shroud section. e) De-mate nose-fairing and place in transporter f) De-mate 1st shroud section and place in transporter. g) De-mate 2nd shroud section and place in transporter. h) Attach transporter cover lift sling to hydraset and connect sling to cover i) Hoist cover, install on transporter, remove sling j) Connect prime mover to transporter and proceed to ESA.	PAD	5008, 5009, 5027, 5029, 5030, 5048	X	X				
180	Move Planetary Vehicles to ESA Form convoy and proceed to ESA		5027, 5029, 5048, 5030	X	X				

APPENDIX B

VOYAGER MECHANICAL ALIGNMENT PLAN

This alignment plan presents a method for the alignment of the Voyager spacecraft and forms the basis for detailed equipment and facility requirements and procedures.

The alignment technique is based upon using existing, commerially-available equipment since such equipment is capable of accuracies an order of magnitude greater than any Voyager requirement.

Most of the critical alignment requirements on the Voyager spacecraft are angular. The basic reference system which best meets such requirements is a spherical coordinate system determined by the fine sun sensor and the Canopus sensor, as shown in Figure B-1. Most items

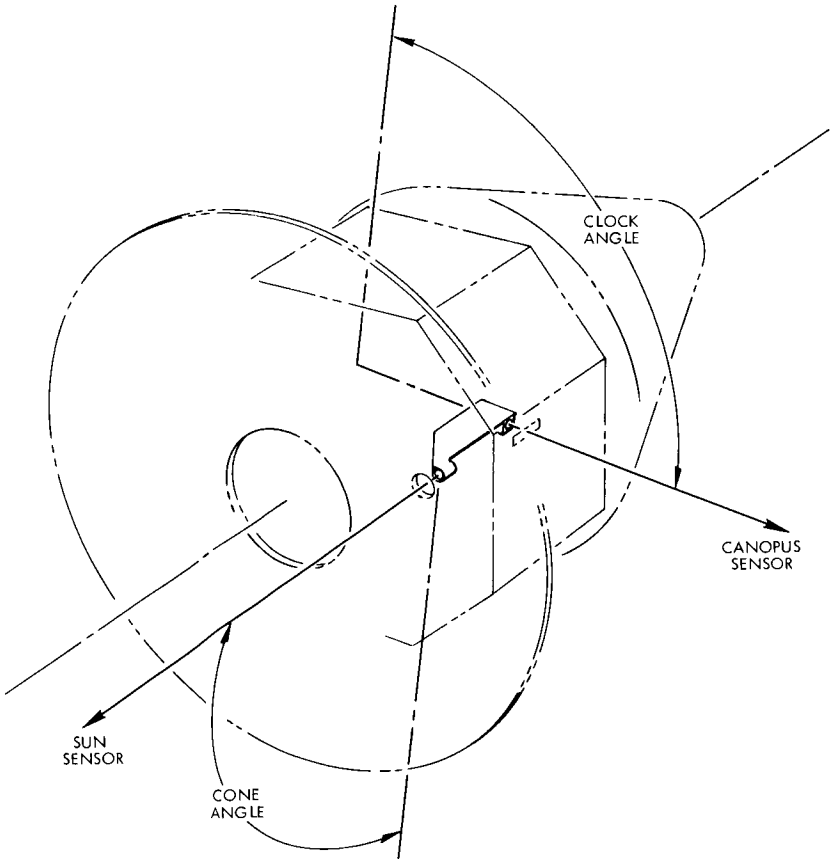


Figure B-1. Basic Reference Coordinate System

requiring alignment have one axis which must be aligned to the fine sun sensor axis, with the cone angle set to zero degrees plus or minus a tolerance; and another (orthogonal) axis set to some clock angle, plus or minus a tolerance, relative to the Canopus sensor axis. Therefore, the fine sun sensor and the Canopus sensor are used as basic references for all final alignments. Except for some special cases (e.g., thrust axis offsets and mass properties determinations), radial and axial locations are not measured.

It is necessary for practical reasons to establish a reference on the spacecraft structure which is basic to all alignments. This reference is a plane described by three of the four hard points at the spacecraft separation plane. This plane is aligned to local gravity by a system of 10 arc-sec sight levels as a preliminary step to the installation of the sensors which define the basic reference system.

1. SUN SENSOR REFERENCE

The sun sensor axis, by definition, defines the zero reference for cone angle. A collimating mirror (AHSE No. 5014) is installed on the sun sensor assembly and aligned on the bench perpendicular to the sun sensor line of sight. This mirror then provides a reference surface for final alignments. The alignment of the sun sensor on the spacecraft is accomplished by leveling the structure to reference marks on the hard mounting points. The mirror on the sun sensor is then viewed with an auto-collimating plumb alignment assembly (AHSE No. 5015) and the sun sensor assembly is adjusted until it is plumb within 10 arc-sec (0.003 degree). After the sun sensor has been aligned to local gravity, all alignments to the sun sensor axis can be made with standard leveling instruments, such as machinist's levels, sight levels, and theodolites. Accuracies of 10 arc-sec are common with such instruments. The plumb alignment assembly is left in place during alignment operations and is periodically checked to assure that the sun sensor is still vertical. This check consists of turning on the collimator lamp and seeing that the returned reticle image is still superimposed on the instrument reticle image.

2. CANOPUS SENSOR REFERENCE

The Canopus sensor axis defines the zero reference for clock angle. A collimating mirror (AHSE No. 5014) is installed on the sensor assembly and aligned on the bench to the sensor line of sight. After installation of the sensor assembly on the spacecraft, the mirror is viewed with an auto-collimating theodolite (AHSE No. 5015). The horizontal axis of the mirror defines the zero reference clock angle, the azimuth zero setting of the theodolite, and the zero setting of the vertical assembly and alignment stand (AHSE No. 5003). The clock angles of other items to be aligned are measured by rotating the alignment stand and the theodolite as necessary and reading the angles indicated. Standard theodolites, such as are used in second-order surveys, will provide angular measurement accuracies of better than 10 arc-sec.

3. AXIAL AND RADIAL LINEAR MEASUREMENTS

For the determination of mass properties, thrust-axis offsets, et al, axial and radial linear measurements are necessary. With the basic vehicle reference established (i. e., the sun sensor line of sight normal to local horizontal), these measurements are made with standard industrial optical equipment such as jig transits or theodolites and optical tooling scales in combination with special fixtures and alignment targets (AHSE No. 5014).

4. FLIGHT CAPSULE

The flight capsule must be aligned to the geometric axis of the flight spacecraft in order to hold the overall c. g. offset within the allowable envelope. This alignment is accomplished by the traditional method of sighting on capsule reference targets with two jig transits arranged to sweep orthogonal planes which intersect along the vehicle longitudinal axis. These transits are set up using the sun sensor-Canopus sensor system as a basic reference.

5. END-TO-END CHECKS

All components of the spacecraft are automatically checked end-to-end to the basic reference when the techniques described above are used.

APPENDIX C

MAGNETIC PROPERTIES CONTROL PLAN

1. INTRODUCTION

This section discusses the tasks that will be accomplished to ensure that the spacecraft magnetic characteristics satisfy the mission requirements. These tasks form the basis for the Magnetic Control Plan which will be submitted as part of the Implementation Plan for Part 2 of the Phase IB proposal.

The four main elements involved in maintaining control of the magnetic properties of the Voyager spacecraft are:

a. Analysis of the configuration and magnetic requirements resulting in derived allocations.

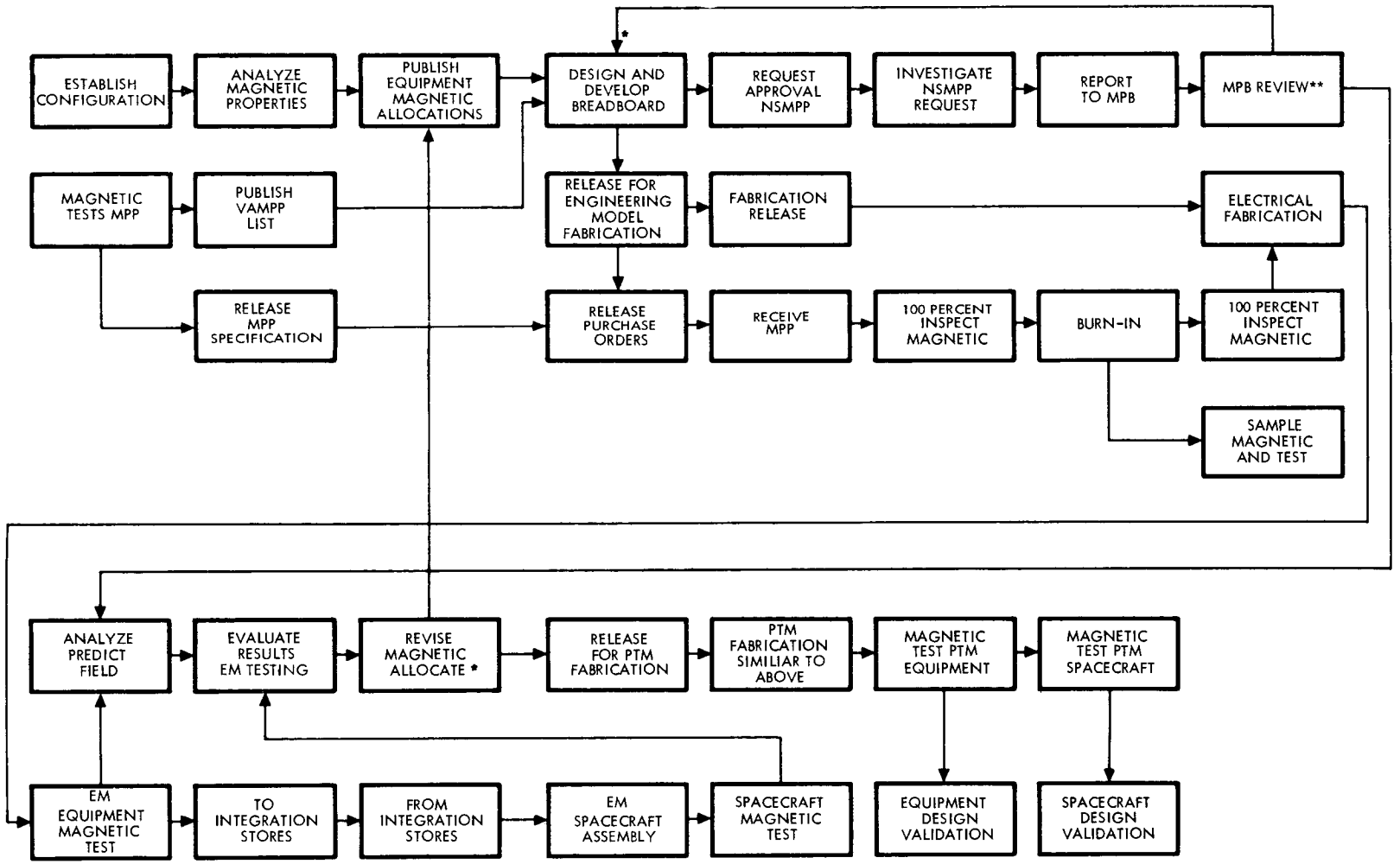
b. The evaluation of materials, parts, and processes for magnetic properties as well as other important properties to establish the list of Voyager approved materials, parts and processes.

c. The development of the equipment for the spacecraft within the established constraints and the evaluation of the resulting fields, using the engineering model hardware.

d. The validation of the adequacy of the design for magnetic levels by performing the following tests:

- Measuring magnetic fields in both operating and non-operating modes at the assembly level
- Field mapping of selected assemblies in the operating and nonoperating modes at the assembly level
- Measuring the residual magnetism and net coercive force at the assembly level
- Measuring and mapping the magnetic fields in nonoperating and operating modes at the spacecraft level
- Measuring the residual magnetism and net coercive force at the spacecraft level

Figure C-1 shows the information flow and critical steps in the process.



PTM PROOF TEST MODEL
MPP MATERIALS, PARTS AND PROCESSES
MPB MAGNETIC PROPERTIES BOARD
VAMPP VOYAGER APPROVED MATERIALS, PARTS AND PROCESSES
NSMPP NONSTANDARD MATERIALS, PARTS AND PROCESSES

*CORRECTIVE ACTION TESTS
**COMPLETED PRIOR TO SECOND DESIGN REVIEW

Figure C-1. Magnetic Control Flow Diagram

Spacecraft Engineering Operations is responsible for evaluating the costs, reliabilities, and schedule risks to establish favorable trade-offs between an adequate deployment device for the magnetometer and controlling the magnetic properties of the spacecraft. Long booms involve excessive weight, present difficult alignment problems, and, to some extent, affect reliability. On the other hand, a preliminary analysis has indicated that controlling the magnetic properties of the spacecraft equipment sufficiently well to permit mounting the magnetometer directly on the spacecraft would be very expensive, and would involve high development and schedule risks. It presently appears that a boom length between 10 and 20 feet will be adequate to permit the use of a reasonably inexpensive magnetic control program.

Once this decision has been made, Spacecraft Engineering Operations is responsible for establishing the allocations for spacecraft equipment with respect to magnetic properties. A magnetic property board will report to the manager of Spacecraft Engineering Operations and will be responsible for establishing the allocations, for evaluating requests for deviation from allocations, and for the approval of magnetic properties which are called out in equipment specifications.

The Electrical and Mechanical Operations groups are responsible for developing the spacecraft equipment within the constraints established by magnetic allocations. They will be assisted by the product engineering laboratory in carrying out investigations of materials, parts and processes, including magnetic requirement evaluations. They are also responsible for establishing the design requirements for materials, parts and processes and evaluating development risk, schedule risk and cost risk to determine the need for testing new materials, parts and processes.

Product Integrity Operations is responsible for establishing the scope of any sample tests involved, for evaluating process control procedures and evaluating test procedures. Product Integrity Operations is also responsible to see that proper tests are conducted and proper data records maintained for magnetic property traceability.

The Electrical and Mechanical Operations groups are responsible for fabricating spacecraft equipment in accordance with approved specifications and drawings. The engineering model equipment will be fabricated to evaluate the efficacy of the design, including the control of magnetic properties. Measurements will be made to confirm the predicted field levels for items of equipment and for the spacecraft. Magnetic testing will be carried out on the material, parts and processes involved in the basic manufacturing process, on the equipment items (black boxes) and on the spacecraft. Specifications and drawings will be revised to incorporate design features reflecting the experience obtained in working with the engineering model spacecraft.

The PTM spacecraft will be fabricated in accordance with the revised documents. Magnetic testing will be carried out at the same three levels of the fabrication process to validate the adequacy of the control plan to produce desired results.

All these tests will be carried out on 100 per cent of the parts for engineering models and PTM assemblies. It may subsequently prove practical to employ sampling tests.

2. ESTABLISHING MAGNETIC ALLOCATIONS

2.1 Guidelines

Previous experience with OGO, Pioneer, and other spacecraft indicates that the specified magnetic field of 1γ at the magnetometer is difficult to achieve without deploying the magnetometer, but relatively straightforward to achieve with a boom 10 to 20 feet long. It appears practical to establish a maximum magnetic moment of approximately 500γ referred to 1 foot for each item of equipment, and to require that all structural items be nonmagnetic as far as possible. Such magnetic field levels are consistent with results achieved on OGO. Application of these levels on the basis of permitting the magnetic moment to be random when below this level and requiring the vector moment to be made part of the specification when the moment is above the allocated level will permit the prediction of total spacecraft field with the required accuracy.

2.2 Current Effects

The use of a boom approximately 15 feet long places the magnetometer between 15 and 35 feet from current loops. It appears that over this range of distance, there will be little or no effect from such loops. Preliminary analysis indicates that the current loop restrictions imposed by the low frequency EMI specification will be more severe than the specified DC field at the magnetometer.

2.3 Magnetic Materials

It may prove necessary to use some magnetic materials in the spacecraft. Their use will be avoided if at all possible, but certain functions may require their use. The presence of magnetic materials must be controlled with respect to two properties. First the specified minimum fields, both magnetized and demagnetized must be met. Second, the net coercive force must be controlled sufficiently well to permit using the anhysteretic assumptions involved in matrix transformation from the induction field to the field in the magnetometer. This latter effect is not critical for Voyager in the measurement of low magnitude fields. However, it is very important to the adequate testing of the equipment and the spacecraft in the earth's field, requiring extreme care with respect to the magnetic history of the items under tests.

3. DEVELOPMENT CYCLE

3.1 Voyager Approved Material, Parts and Processes

At the beginning of the development cycle it is expected that information from JPL test experience and from TRW experience on previous programs involving magnetic constraints will permit establishing a reasonably comprehensive list of magnetically clean materials, parts and processes.

3.2 Nonstandard Materials, Parts and Processes

As the development proceeds, designers will undoubtedly request permission from the magnetic properties board to use alternative parts. Each request will be processed to solve the problem with acceptable solutions being in the following order of priority:

- Find a standard alternative
- Standardize the nonstandard materials, parts and processes with the specified constraints (in cooperation with the vendor)
- Accept the nonstandard materials, parts and processes with the deviations.

3.3 Magnetic Property Board

The magnetic property board will control the approval of materials, parts and processes. They will take the initiative at the first design review to see to it that all materials, parts and processes are properly identified and action is taken to establish control. They will also be available to act on requests of the development engineer at any time.

3.4 Breadboard Tests

Magnetic tests of the breadboard equipment will only be indicative of the order of magnitude of the magnetic moment. It is not anticipated that full magnetic control will be effective at the time of fabrication of the breadboard. Consequently, quantitative predictions of overall spacecraft field levels cannot be made from breadboard test data.

4. SPECIFICATIONS

4.1 Materials, Parts and Processes Specifications

All specifications and drawings for materials, parts and processes will include a magnetic constraints specification (material or material plus processing callouts on fabricated parts is adequate for a magnetic constraint). It will not be acceptable to depend on the vendor to continue to produce acceptable parts based only on test of available catalog parts.

4.2 Equipment Specification

Each equipment specification will contain a specified magnetic level. It is expected that the equipment will fall within the allocated magnetic field. The specification will use the allocated field as a maximum value. If it is expected that the item of equipment will have a field exceeding the allocation, the specification will include the magnetic vector dipole moment.

4.3 Test Specifications/Procedures

Magnetic tests will be included for each item of equipment. To the greatest extent possible, uniform test specifications and procedures will be followed for materials, for parts and processes, for equipment, and for the spacecraft. For any item having significant coercive effects, special tests and procedures will be prepared.

5. DESIGN EVALUATION

5.1 Materials, Parts and Processes Inspection

All parts, samples, or materials and all processes will be 100 percent inspected for magnetic constraints conformance. Magnetic tests will be included as one of the parameters measured before and after burn-in. For parts which are serialized for screening, magnetic traceability will be maintained to part serial numbers. For any parts which are not serialized for screening, magnetic traceability will be maintained to designated batches.

5.2 Intermediate Assembly Level Testing

It is not anticipated that magnetic testing of intermediate assembly level will be required. However, if the assembly process is such that it would be extremely expensive, or involve high schedule risk, to remove an item of equipment from the spacecraft, it may be necessary to test the magnetic moment at some intermediate level.

5.3 Assembly Level Testing

On the engineering models, field mapping of equipment may be carried out on selected items of equipment to verify analytical predictions of magnetic properties. Such tests may be carried out even though the equipment meets the magnetic allocations. As mentioned before, equipment not meeting magnetic allocations always requires field mapping for magnetic test. Subsequent to engineering model test, it is anticipated that field mapping will not be used in equipment meeting its magnetic allocations. Field measurements will be performed in operating and nonoperating modes on all assemblies. The coercive force of each assembly will be measured. Coercive force measurements at the

assembly level will not be performed after the engineering model assembly testing phase.

5.4 Subsystem Level Testing

Subsystem level magnetic testing is not planned at this time because it is felt that the engineering information obtained could not justify the cost of the operation.

5.5 Spacecraft Level Testing

All spacecraft models except the propulsion integration model spacecraft will be tested. The near magnetic field will be mapped in detail to permit accurate predictions of the field at the magnetometer. Three or four magnetometers, mounted at various distances, will be used. The spacecraft will be rotated to map the fields. The testing will be done at a magnetically quiet location. In addition to this the spacecraft coercive force will also be measured for each spacecraft.

The stability of the field with changes in operating modes will be evaluated using a sensitive magnetometer in a coil system to read small changes in level.

APPENDIX D

VOYAGER CONTAMINATION CONTROL PLAN

1. INTRODUCTION

This plan describes the contamination control program to be applied to the Voyager spacecraft project. The program is based on an analysis of the effect on mission success of both biological and nonbiological contaminants. The program consists basically of reducing the number of extraneous particles by mechanical means and destroying microorganisms by chemical or heat sterilization.

This program is to be conducted in close liaison with JPL to assure coordination of contamination control plans, restraints, and techniques of implementation optimal to mission requirements. The details of implementation of the contamination control requirements for hardware elements of the Voyager spacecraft will be developed during Phase IB.

Voyager spacecraft reliability is a major consideration in the assessment of cleanliness controls and techniques to be applied.

The biological contamination control plan satisfies the mission requirement that the probability of contaminating Mars be 10^{-4}, by precluding the chance of contaminating the Voyager landing capsule through emissions of viable particles from the flight spacecraft. The plan establishes control of the biological load buildup during manufacture of the spacecraft, and specifies necessary sterilization procedures for spacecraft subsystems and the completed spacecraft assembly.

The nonbiological contamination control plan has as its objective the manufacture of a clean spacecraft with the elimination of all contaminants which could detrimentally affect the performance of the vehicle. The program is oriented more to contamination prevention than after-the-fact curative measures. The basic methods of the plan are environmental control during fabrication and assembly by the use of clean rooms, institution of effective cleaning procedures at various stages of manufacture, and contamination controlled storage and transportation.

2. BIOLOGICAL CONTAMINATION CONTROL

The trajectory of the Voyager flight spacecraft will be chosen such that the probability of the spacecraft entering the Martian atmosphere satisfies the Mars quarantine constraint. This is not necessarily true of efflux from the spacecraft. Gases and particles emitted by the Voyager flight spacecraft have the possibility of impinging on the lander during flight (after the lander sterilization canister has been removed), or upon entering the Martian atmosphere. These ejecta include midcourse propulsion exhaust, cold gas for attitude control, surface coating particles, outgassing and sublimation products, and particles chipped from such areas as insulation film layers and honeycomb structures by micrometeoroid or other damage. An analysis of the probabilities of contaminating Mars through these sources is presented in Phase IA, Task B Final Report, Volume 1.

To preclude the possibility of violation of the Mars quarantine constraint by these sources, the following decontamination processes will be employed.

- Use of self-sterilizing surface coatings
- Filtration and sterilization of gases
- Internal sterilization of subsystems which would produce spacecraft efflux
- Thermal sterilization of sandwich or laminated surfaces subject to micrometeoroid impact.
- Surface sterilization of the spacecraft with ethylene oxide gas.

Controls that will be established to ensure effective decontamination include the following:

- Minimizing surface contamination through clean room handling, periodic cleaning, and storage and transportation in closed containers.
- Training of personnel in executing the contamination control plan.
- Monitoring of facilities and personnel to insure conformance to requirements.

- Use of proven sterilization processes, equipment, and sterility certification methods.

Fabrication and assembly will be conducted in rooms meeting the requirements of Class 100,000 or better (Federal Standard No. 209) and additional limits establishing permissible concentration of viable particles in the ventilation inlet air, in the room air, and in surface fallout. Microbioassays will follow the procedures delineated in NASA "Requirements for Bioclean Facilities". Number of people within the rooms will be limited and all persons will be required to wear clean lintless coats, head and shoe covering and lintless gloves. Air washes will be required of personnel before they don clean overgarments.

Voyager subsystem designs will be critically examined during Phase IB to uncover other items which may contribute contaminated efflux. Design modifications or decontamination procedures to eliminate such problems will be defined and incorporated in the contamination control plan.

Surface sterilization of the completed flight spacecraft and the lander capsule will be conducted in a biological shroud after completion of systems tests and mating of the capsule and spacecraft. Handling controls to prevent rupture of the capsule seal during mating operations will be established.

Materials and components used in construction of the spacecraft will be tested during Phase IB for compatibility with the sterilant gas to ensure no degradation of spacecraft life as a result of the sterilization cycle. Materials and components will be excepted from the testing when sufficient data are already available through NASA and its contractors.

The fungus resistance of spacecraft materials will be established as well. Materials will be tested for fungus resistance in accordance with MIL-E-5272, except those materials for which data is already available. Results will be incorporated in the Voyager approved materials list.

A program to indoctrinate and train personnel in the objectives of the contamination control program, and the required skills and procedures for its successful execution, will be implemented during Phase IB and II

of the Voyager program. Management, engineering, and manufacturing personnel will participate.

The following documents will be prepared during Phase IB.

- Contamination Monitoring Procedures
- Bacteriological Filter Specifications
- Cleaning Equipment and Fluids Specifications
- Decontamination Processes Specifications
- Ethylene Oxide Sterilization Process Specification
- Mobile Sterilization Unit Specification
- Contamination Control Indoctrination and Training Course
- Specifications for Self-sterilizing Formulations
- Detailed Contamination Control Procedures for Each Vital Subsystem

3. NONBIOLOGICAL CONTAMINATION CONTROL

The previous section dealt with control of living organisms which could contaminate the environs to which the Voyager spacecraft will be introduced. This section deals with the control of inert particles which could have deleterious effects on the performance of the spacecraft rather than on the target environs. Since particles can contain living microorganisms, control of the number of dirt particles which may find their way into the spacecraft also contributes to control of biological contaminants.

Discussed in this section are means for maintaining a clean atmosphere during the assembly, fabrication and test of the spacecraft and its subsystems by such methods as air filtration and personnel control. Most important is the reduction of particle introduction by training, education, and dress of the personnel involved, since they are the major source of dirt particles in a clean room activity.

Cleanliness requirements will be specified by the Voyager contamination control plan and will be in conformance with JPL requirements.

Specifications are prepared to control the cleaning or clean room practices for the required components and assemblies. Detailed procedures and materials to clean, package, and assemble designated components are established and specified.

Cleaning equipment such as solvent and cleaning solution pump units, flushing consoles, ultrasonic units, and drying equipment are constructed of low particle-producing materials with filtration provided between the equipment and the component being cleaned. The design of cleaning equipment and their location are chosen to minimize their contribution to room contamination. A cleaning solvent recovery system is utilized wherever practical to conserve solvents.

Cleaning and testing fluids are prefiltered to the cleanliness level defined by the contamination control requirements. Particle counts are taken on the filtered fluids as a control measure. Nonvolatile residue tests are performed when necessary. Cleaning and testing gases also are prefiltered to meet contamination control requirements. Vendor shipments of gas are checked for dew point and nonvolatile hydrocarbon content. All expendable materials such as identification inks, cleaning cloths, and writing materials are selected by the contamination control activity if they are to be used in environmentally controlled areas. Contamination control provisions are made in process specifications for all flight spacecraft components. Special attention is provided those operations in which there is production of chips, burrs, filings.

In interconnecting electronic parts and components by one or more of the currently available joining methods, such as resistance welding, soldering, and wire wrapping, serious reliability problems may be encountered if the joining surfaces are corroded. Corrosion of joints may occur if the parts are stored over long periods, unprotected, in an uncontrolled environment. Corrosion may also occur if parts are excessively handled by human hands.

Experience has shown that after precleaning, electronic hardware for use in spacecraft can be safely assembled and stored in an environment meeting the requirements of Federal Standard 209, Class 100,000 clean

room or equivalent. The temperature, humidity, and dust count requirements of the 100,000 clean room are adequate to provide the kind of protection from contamination that ensures reliable hardware performance. Before removal from these areas to contamination controlled storage or subsequent assembly areas, the assemblies are packaged in protective containers, such as dust-free bags or other specially designed protective dustfree boxes.

Precision mechanical and electromechanical devices requiring tolerances as close as a few microns are cleaned and assembled in suitable controlled environments to prevent the inclusions of micron size particles that can cause galling and seizure of critical surfaces. Sealing of these clean assemblies by dust-free containers is important in order that they may be stored and handled under less stringent environmental conditions, with high assurance that they will remain clean.

Hardware having stringent requirements for non-magnetic properties is fabricated, processed and assembled in specifically designated areas, where magnetic materials are not processed.

The final assembly of the flight spacecraft subsystems is conducted in a high reliability assembly and checkout area. Physical contamination is minimized through personnel and environmental control.

Installation of the flight capsule on the flight spacecraft will be conducted in the explosion-proof facility at Cape Kennedy, under clean room conditions meeting the requirements of Federal Standard 209, Class 100,000 facility. The flight spacecraft with the flight capsule will be enclosed in the nose fairing under similar conditions.

Complete documentation is obtained through design drawings and process specifications. Materials are controlled by government or industry and TRW specifications. No deviations are allowed from the specifications without written approval from engineering.

4. ADDITIONAL CONTROLS

All personnel who clean or assemble critical components are trained and certified in this operation. Only those who have completed

the training course and successfully passed the written tests are authorized for clean room work. The training program integrates the biological and nonbiological contamination control requirements.

Contamination control requirements compatible with the contamination control plan are imposed on subcontractor furnished assemblies. Subcontractors are required to submit their contamination control plan for review and approval.

Subcontractor facilities are reviewed for conformance to contamination control procedures and certified by Quality Assurance.

The minimum identification on cleaned parts consists of the certification stamp of the employee who cleaned the part; the part number and serial number; the data of cleaning and the specification to which the part was cleaned.

APPENDIX E

ELECTROMAGNETIC COMPATIBILITY CONTROL PLAN

The success of the Voyager mission requires the development of an electromagnetically compatible system. This task involves the analysis and definition of the system EMC requirements, implementation of the EMC control requirements, design verification, and integrated system testing. EMC criteria documents to be issued will consist of EMC requirements and an EMC control plan which details the techniques for implementing the design requirements on both the system and subsystem level. Design support will be provided in the application of these criteria to all levels of design through direct consultation and support of the design reviews. Design verification testing will consist of breadboard tests, engineering model evaluation, type approval testing and proof test model verification. Portions of the integrated system test will be devoted to determining the EMC of the system, with emphasis on potential problem areas identified through previous EMC analysis.

The flow diagram, Figure E-1, shows the events necessary for the successful development of a compatible system. Table E-1 is a tabulation of the tasks associated with each of the major events shown on the flow diagram. A typical test requirements list on a subsystem level is given in Table E-2.

The system test requirements will be generated to demonstrate system compatibility. The types of tests to be performed will include measurements of spacecraft generated noise, evaluation of system susceptibility thresholds, and a demonstration that an adequate compatibility margin exists in the design.

Figure E-2 is a tentative schedule of major EMC control events, dependent upon the deliverable hardware schedules. The key items in the schedule are the electromagnetic compatibility requirements document, and the subsystem design implementation criteria (control plan) document. The requirements documents will be released three months after Phase IB; the implementation criteria document will be released 2-1/2 months later. Any additional schedule dates will be hardware-dependent.

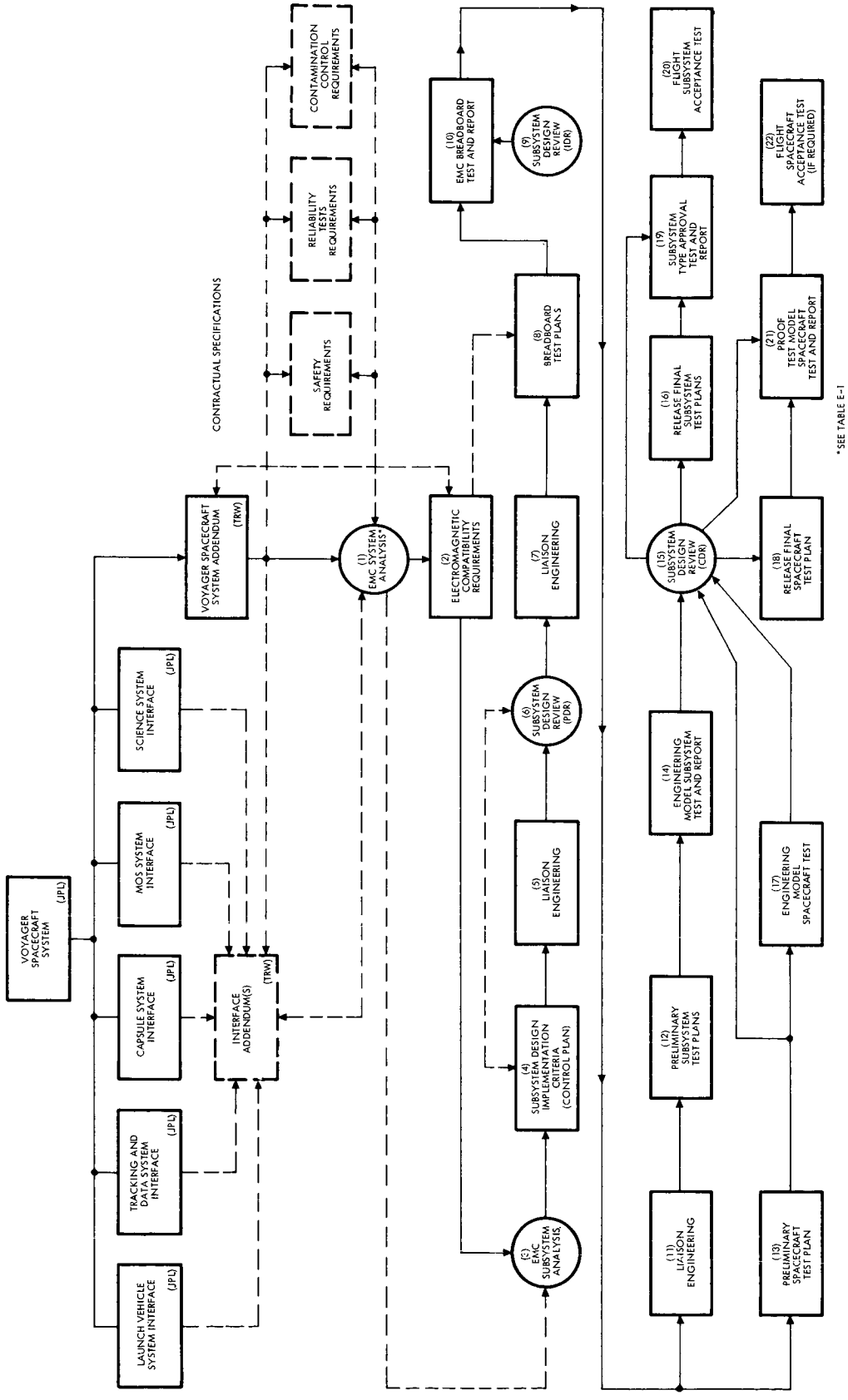


Figure E-1. EMC Control Implementation Program Flow

Table E-1. Electromagnetic Control Implementation Plan Outline

Event	Objective	Description	System Article(s)
1. System Analysis	Determine critical system compatibility areas and required design margins.	The various subsystems and their interfaces will be reviewed to determine possible generated interference and critical susceptibility levels. Compatibility margins will be determined.	Voyager spacecraft, spacecraft support system, and mission dependendent equipment system.
2. EMC Requirements	Generate the EMC requirements specification for the Voyager program.	Specify the system design requirements for electrical bonding, grounding, shielding, cabling, circuit isolation, power quality, and generated interference susceptibility limits.	Spacecraft, including all subsystems and experiments, EOSE, and mission dependent equipment.
3. EMC Subsystem Analysis	Determine subsystem implementation criteria.	Each subsystem will be analyzed in terms of the system requirements (2). Required implementation techniques will be evaluated to determine most efficient and effective approach to complying with the system requirements.	All spacecraft and EOSE equipment.
4. Subsystem Design Implementation Criteria (control plan)	Generate subsystem implementation criteria document (control plan).	All the preferred approaches for each subsystem will be defined. Details of electrical bonding, grounding, and shielding will be specified for each equipment. The required test and applicable limits will be specified, along with recommended approaches to reducing generated interference and raise susceptibility thresholds.	All spacecraft and EOSE equipment.
5. Liaison Engineering	Coordinate EMC design requirements and implementation criteria with subsystem designers.	Subsystem requirements and implementation techniques will be clarified when required. In cases of design conflicts, alternate approaches will be devised. Adequacy of alternate approaches will be verified by analysis and laboratory tests where necessary.	All spacecraft and EOSE equipment.
6. Subsystem Design Review (PDR)	Review preliminary subsystem designs.	Subsystem design recommendations will be made when required. Problems encountered in complying with EMC requirements will be discussed.	All spacecraft and EOSE equipment.
7. Liaison Engineering	Coordinate EMC design requirements and implementation criteria with subsystem designers.	The activities of (5) will be continued. Any changes in implementation techniques as a result of (6) will be coordinated.	All spacecraft and EOSE equipment.
8. Breadboard Test Plans	Specify the EMC engineering evaluation test for selected subsystems.	Subsystem such as power, thermal, pyrotechnic, etc., will have breadboard test to investigate operation characteristics as they affect power quality and susceptibility thresholds. The types of tests are listed in Table G-2.	Spacecraft and EOSE power subsystems, spacecraft pyrotechnic firing circuits, thermal control electronics, etc.
9. Subsystem Design Review (IDR)	Review subsystem design for adequacy of EMC control.	The subsystem designs will be reviewed for adequacy of EMC control. Problem areas will be discussed and recommendations for implementation techniques will be made.	All spacecraft and EOSE equipment.

Table E-1. Electromagnetic Control Implementation Plan Outline (Continued)

Event	Objective	Description	System Article(s)
10. EMC Breadboard Test	Investigate conducted interference and susceptibility characteristics on selected subsystems.	Test will be performed as specified in (8). Results will be issued to responsible design engineers along with recommendations for corrective action where required.	Spacecraft and EOSE power subsystems. Spacecraft pyrotechnic firing circuits, thermal control electronics, etc.
11. Liaison Engineering	Coordinate breadboard test results. Close out any action items resulting from design review.	The detailed breadboard test results and recommendations will be coordinated with responsible design engineers. Any further investigations or studies will be initiated at this time.	Spacecraft and EOSE power subsystems. Spacecraft pyrotechnic firing circuits, thermal control electronics, etc.; plus any equipment interfaces which result from event (9).
12. Preliminary Subsystem Test Plans	Specify the EMC test and methods for verifying adequacy of subsystem design.	Detail test plans will be generated in accordance with the system EMC requirements of event (2).	All spacecraft equipment and selected EOSE equipment.
13. Preliminary Spacecraft EMC Test Plan	Specify the system level EMC evaluation test.	As a parallel effort with events (11) and (12), System Level Test Plan will be prepared to investigate suspect problem areas. A complete set of tests in accordance with the requirements of event (2) will be defined.	Engineering model spacecraft and EOSE equipment.
14. Engineering Model Subsystem Test and Report	Verify adequacy of subsystem EMC design.	Test will be performed in accordance with event (12). Any out of tolerance conditions will be explored to determine feasibility of corrective action.	All spacecraft equipment and selected EOSE equipment.
15. Subsystem Design Review (CDR)	Final review of subsystem design for adequacy of EMC control.	The results and recommendations of the engineering model test will be presented. Preliminary drafts of the subsystem type approval unit test plans will be presented.	All spacecraft equipment.
16. Release Final Subsystem Test Plans	Update subsystem EMC test requirement for type approval units.	Update the test and methods necessary to verify design compliance with the requirements of event (2).	All spacecraft equipment.
17. Engineering Model Spacecraft Test	Investigate system level compatibility problem areas.	Test will be performed on the engineering model spacecraft as specified in event (13) for compatibility problem areas such as conducted, radiated susceptibility thresholds.	Engineering model spacecraft and EOSE equipment.
18. Release Final Spacecraft System Test Plan	Update EMC test and methods to verify system compliance with the requirements of event (2).	The system requirements of event (2), modified by the results of the engineering model test, will be translated into a released EMC system test plan.	Proof test model spacecraft and EOSE equipment.
19. Subsystem Type Approval Tests and Reports	Demonstrate adequacy of subsystem design by compliance to system requirements.	Test will be performed in accordance with event (16).	All spacecraft equipment.

Table E-1. Electromagnetic Control Implementation Plan Outline (Continued)

Event	Objective	Description	System Article(s)
20. Flight Subsystem Acceptance Test	Demonstrate that flight articles are similar in design to type approval units.	Test will consist primarily of an electrical bonding test on the equipment mechanical structure. Power quality test will be made on selected equipment.	All spacecraft equipment.
21. Proof Test Model Spacecraft Test	Demonstrate compatible system design complying with requirements of event (2).	Test will be performed in accordance with event (18).	Proof test model spacecraft and EOSE equipment.
22. Flight Spacecraft Acceptance Test	Demonstrate flight spacecraft are compatible.	Selected portions of event (18) test plan will be performed to demonstrate any marginal conditions which may have been experienced in the test of event (21).	Flight spacecraft and EOSE equipment.

Table E-2. Subsystem Test Requirements

Test Component	Conducted Interference (Power Quality)	Radiated Interference	Antenna Conducted	Conducted Susceptibility	Radiated Susceptibility	Intermodulation Susceptibility
Thermal control 4C2 heaters and thermostats	Scope measurements of power bus switching transients to verify suppression technique. Prepare report.	None	N/A	None	N/A	N/A
S-Band Radio (RCA-supplied) A) S-Band receiver (Engineering model) and/or prototype	Broadband CW, 15kc to 25mc. N/A for TRW.	CW, 150kc to 25mc, and 35mc to 10gc Broadband, 150kc to 25 mc and 35mc to 400mc. N/A for TWR.	150 kc to 10 gc. Signal level limit to be determined. N/A for TRW.	AF conducted on power lines. Transient conducted, signal levels to be determined. N/A for TRW.	150kc to 10gc, 30 percent modulated with 400cps and 1kc at power level of 1 watt to antenna. N/A for TRW.	Frequency to be determined. Not applicable for TRW.
B) 1 watt transmitter	CW, 15kc to 25mc.	CW, 150kc to 25mc; 35mc to 10gc.	150kc to 10gc. Signal level limits to be determined.	AF conducted. Transient conducted. Signal level to be determined.	15kc to 10gc, 30 percent modulated with 400cps and 1kc. Power of 1 watt to antenna	N/A
C) Modulator-exciter and TWT + power supplies diplexer	CW, 15kc to 25mc. Broadband, 15kc to 25mc. N/A for TRW.	CW, 150kc to 25mc, 35 mc to 10gc. Broadband, 150kc to 25mc, 35mc to 400mc. N/A for TRW.	150kc to 10gc Signal level limit to be determined. N/A for TRW.	AF conducted. Transient conducted. Signal level to be determined. N/A for TRW.	150kc to 10gc, 30 percent modulated with 400cps and 1kc at power level of 1 watt to antenna. N/A for TRW.	N/A
D) UHF receiver and demodulator (GFE)	CW, 15kc to 25mc. Broadband, 15kc to 25mc. N/A for TRW	CW and broadband, 150 kc to 25mg. CW, 35mc to 1gc. Broadband, 35mc to 400mc. N/A for TRW.	150 kc to 10 gc. Signal level limit to be determined. N/A for TRW.	AF conducted. Transient conducted. Signal levels to be determined. N/A for TRW.	150kc to 10gc, 30 percent modulated with 400 cps and 1kc at power level of 1 watt to antenna. N/A for TRW.	Frequency to be determined. Not applicable for TRW.
UHF preamplifier (GFE)	CW and Broadband. 15kc to 25mc.	CW and broadband, 15kc to 25mc. CW, 35mc to 1gc. Broadband, 35mc to 400mc.	150 kc to 10 gc. Signal level limit to be determined. N/A for TRW.	AF conducted. Transient conducted. Signal level to be determined.	150kc to 10gc, 30 percent modulated with 400 cps and 1kc. Power level of 1 watt to antenna.	Frequency to be determined
4.0 KC inverter	CW and Broadband, 15 kc to 25mc, or scope photos made in accordance with power quality test methods of EMC spec. Conducted measurement to be made on breadboard also.	CW and broadband, 150 kc to 25mc. Broadband, 35 mc to 400mc.	N/A	AF conducted. Transient conducted. Signal level to be determined.	15kc to 1gc, 30 percent modulated with 400cps and 1kc. Power level of 1 watt to antenna.	N/A
(PCU and battery regulator)	Broadband, 15kc to 25 mc or scope photos made in accordance with power quality test methods of EMC specification.	Broadband, 150kc to 25mc, and 35mc to 400 mc.	N/A	AF conducted Transient conducted. Signal level to be determined.	15kc to 1gc, 30 percent modulated with 400cps and 1kc. Power level of 1 watt to antenna.	N/A

Table E-1. Subsystem Test Requirements (Continued)

Test Component	Conducted Interference (Power Quality)	Radiated Interference	Antenna Conducted	Conducted Susceptibility	Radiated Susceptibility	Intermodulation Susceptibility
400 CPS inverter	CW and broadband, 15kc to 25mc or scope photos made in accordance with power quality test methods of EMC specification.	CW and broadband, 150kc to 25mc; CW, 35mc to 1gc; broadband, 35 to 400 mc.	N/A	AF conducted. Transient conducted. Signal level to be determined.	15kc to 1gc, 30 percent modulated with 400 cps and 1kc. Power level of 1 watt to antenna.	N/A
Computing and sequencing and command subsystem	CW and broadband, 15kc to 25 mc and/or scope photos of power bus for power quality.	CW and broadband, 150kc to 25mc; CW, 35MC to 1GC; broadband, 35mc to 400mc.	N/A	AF conducted. Transient conducted. Signal level to be determined.	15kc to 10gc, 30 percent modulated with 400 cps and 1kc. Power level of 1 watt to antenna.	N/A
Data handling subsystem	CW and broadband, 15kc to 25mc and/or scope photos of power bus for power quality.	CW and broadband, 150kc to 25mc; CW, 35mc to 1gc; broadband, 35 to 400mc.	N/A	AF conducted. Transient conducted. Signal level to be determined.	15kc to 1gc, 30 percent modulated with 400 cps and 1kc. Power level of 1 watt to antenna.	N/A
Guidance and control subsystem	CW and broadband, 15kc to 25mc and/or scope photos of power bus for power quality.	CW and broadband, 150kc to 25mc; broadband, 35 to 400mc.	N/A	AF conducted. Transient conducted. Signal level to be determined.	15kc to 1gc, 30 percent modulated with 400 cps and 1kc. Power level of 1 watt to antenna.	N/A
High gain antenna assembly	Broadband, 15kc to 25mc and/or scope photos for power quality.	Broadband, 150kc to 25 mc; 35 to 400mc	N/A	AF conducted. Transient conducted. Signal level to be determined.	N/A	N/A
PSP gimbal drive and electronics	Broadband, 15kc to 25 mc and/or scope photos for power quality.	Broadband, 150kc to 25 mc; 35 to 400mc.	N/A	AF conducted. Transient conducted. Signal level to be determined.	N/A	N/A
Scientific instrumentation	Broadband and CW, 15 kc to 25mc and/or scope photos for power quality. N/A for TRW.	Broadband and CW, 150 kc to 25mc; 35mc to 10 gc. N/A for TRW.	150kc to 10gc. Signal level to be determined. N/A for TRW.	AF conducted. Transient conducted. Signal level to be determined. N/A for TRW.	15kc to 10gc, 30 percent modulated with 400 cps and 1kc. Power level of 1 watt to antenna. N/A for TRW.	Frequency to be determined. Not applicable to TRW.

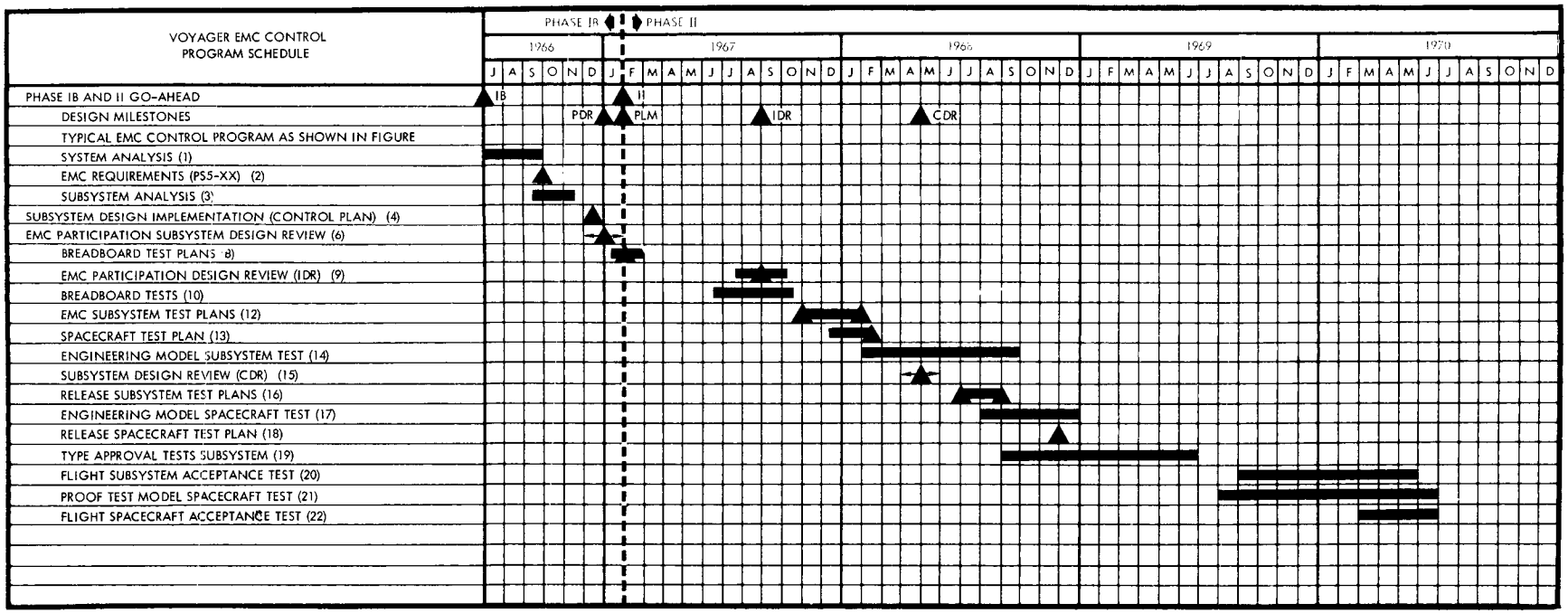


Figure E-2. Voyager EMC Control Program Schedule

The implementation criteria document, item 4 of Figure E-1 and Table E-1, will be the system and subsystem design working document. This document will contain detailed design criteria in the areas of electrical bonding, grounding, shielding, cabling, power quality, electromagnetic interference generation, and susceptibility margins. Some preliminary details concerning bonding, grounding, shielding, and cabling are contained in Volumes 1 and 2 under the electrical integration and cabling subsystem sections.

APPENDIX F

MATERIAL ACQUISITION

During Phase IB considerable effort will be expended in developing sources, determining requirements, and establishing delivery schedules for procurement of parts, materials, and subcontract items. Parts to be procured for Phase IB breadboard testing and evaluation will be commercial or military specification off-the-shelf items. Delivery of such items requires 8 to 16 weeks after issuance of the purchase requisition. The material acquisition flow is shown in Figure F-1. A more detailed discussion of material acquisition will be presented in the procurement plan submitted in Part 1 of the Phase IB proposal.

Phase II material acquisition becomes considerably more complex due to the Voyager specifications that must be imposed both on parts and on suppliers. Capacitors procured to the military specification level take 8 weeks, whereas a similar capacitor procured to a TRW high reliability specification may take as long as 32 weeks. Procurement of flight approved gyroscopes requires approximately 40 weeks to deliver (first units) after award of contract. However, for planning and scheduling purposes, additional time for in-house administrative processing, JPL/NASA approval, and material handling must be added to the manufacturer's production time. Special effort, when required, can be applied to accelerate this effort.

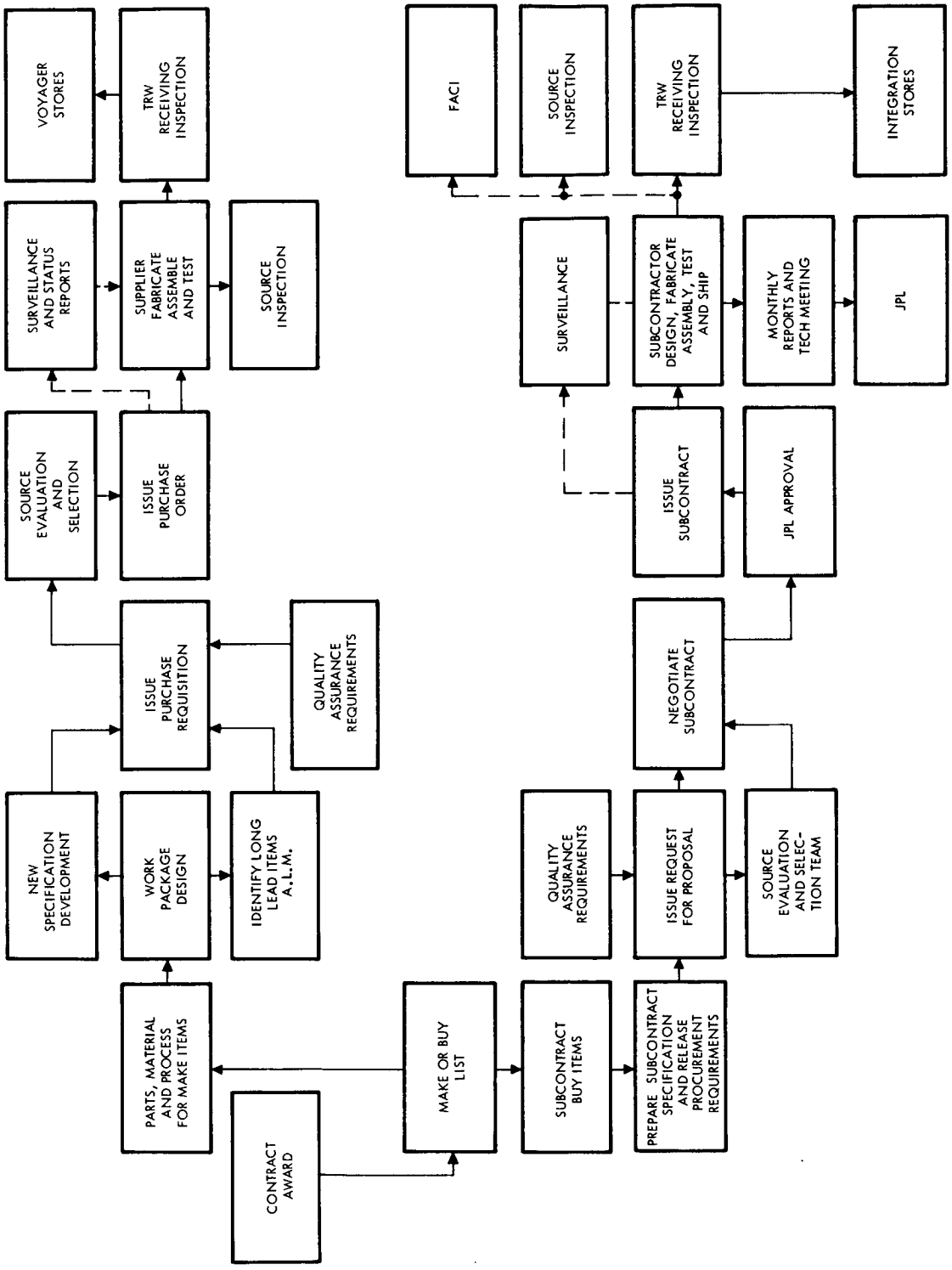


Figure F-1. Voyager Material Acquisition Flow

APPENDIX G

MATERIAL HANDLING AND PACKAGING PLAN OUTLINE

This plan outline describes the material handling and packaging program to be applied to the Voyager spacecraft. The program will consider every requirement for the protection of parts and assemblies against damage and contamination. Voyager reliability and personnel safety are the prime considerations in the assessment of handling and packaging techniques. These techniques take into account the following:

- Storage requirements
- Protection against adverse environments (humidity, temperatures, etc.)
- Shock and vibration (shipping)
- Inter- and intra-plant transporting
- Clean room requirements (handling during assembly under contamination-free environments)
- Control of hazardous or dangerous materials, radioactive material and the reduction of magnetic fields to tolerable levels

In particular, the material handling and packaging program will:

1) Provide for the design of appropriate handling and shipping equipment, and procedures for the use of such equipment.

2) Establish a training program that will provide positive motivation for good workmanship, as well as specific training to aid supervisory personnel and technical personnel in the promotion of handling and packaging skills.

The program will schedule development of necessary procedures and specifications in the early stages of the project, to be implemented during Phase II by the material handling and packaging organization.