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PHASE 1A STUDY REPORT

VOYAGER SPACECRAFT

VOLUME 3
VOYAGER PROGRAM PLAN

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TRW SYSTEMS GROUP

Redondo Beach, California

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ILLUSTRATIONS (Continued)

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I. INTRODUCTION

This volume presents the results of the Phase IA study with respect to implementation of the spacecraft. It includes a description of the effort required to design, develop, test, assemble, check out, launch, and support spacecraft flight operations for both the 1969 flight test and the 1971 mission. The plan essentially applies to the spacecraft only, although the OSE schedule interfaces of the system and its subsystems are also identified. OSE implementation is discussed in full in Volume 6, and the special characteristics of OSE implementation for the 1969 flight are discussed in Volume 7.

Section II presents the major milestone schedules for both Phase IB and II and discusses schedule philosophy, critical areas, and schedule tradeoffs. Section III is devoted to a discussion of the effects of the 1969 flight test program on the 1971 mission in terms of advantages and disadvantages. Section IV is a discussion of over-all test planning and control and the generation of the Voyager integrated test plan.

Section V contains a somewhat detailed description of engineering, test, spacecraft assembly, checkout, systems test, launch, and mission support operations during Phases IB and II to implement both the 1969 flight test and 1971 missions. The manufacturing tasks are briefly treated in terms of schedule requirements and a preliminary estimate of the equipment to be delivered and the time needed for their manufacture. Only a minor effort has been given at this time to the detailed planning of the sequence for the 1973 and subsequent missions. The tasks necessary to implement the missions after 1971, in the light of the design, development, and test status at the time of the 1971 launch, are well within the available schedule time. Design efforts for a 1973 mission could begin as early as 1968, and the fabrication and acceptance testing could be readily spaced to provide a stable level of effort in terms of manpower, equipment, and facilities loading.

As a general rule, Section V does not discuss organization or project control, the focus being on the identification and scheduling of the tasks that must be done to meet the launch dates. The implementation tasks discussed in Section V begin with the systems engineering effort, which converts mission analysis into system design requirements. The spacecraft development group in turn converts these requirements to subsystem design requirements. These design requirements, through the development process, are translated into manufactured and tested equipment which is then assembled into spacecraft models for test and launch.

Several appendices are included to provide additional information on certain planning tasks. Some of these appendices provide detailed planning data (such as the assembly and test planning sheets), which would become cumbersome in the main text. The remaining appendices are provided in outline form to suggest the scope of the plans that need to be provided in Phase IB. These appendices include:

- Assembly and checkout
- Reliability program planning
- Magnetic control plan
- Contamination control
- Equipment list

The policy used in generating the schedules and task descriptions in Sections II and V has been that the 1969 flight test effort is an integral portion of the development cycle of the spacecraft for the 1971 mission. To this end, the ground rule for the design of the 1969 spacecraft is to retain a one-to-one identity with the elements of the 1971 spacecraft, within the constraints imposed by the difference in launch vehicle capability and the absence of scientific objectives. Thus the 1969 spacecraft design is identical to the 1971 design in the elements of the electrical subsystems but differs in over-all size, solar array configuration, structure, propulsion (e.g., there is no solid engine), science payload, and certain deployable elements. The panels used for mounting equipment on the 1969 spacecraft are identical to those used on the 1971 spacecraft; four being used on the 1969 design, compared to six on the 1971. The subsystem equipment is mounted on three of these panels and are identical for both the 1969 and 1971 designs. The fourth panel is used for mounting experiment equipment in the 1971 version; for the 1969 version this panel could support additional equipment if desired.

The solar array is the same in terms of the module design and number of modules per string but differs in terms of the layout of the module on a deployable panel and the reduced number of parallel strings. The six-foot antenna and drives are identical for both the 1969 and 1971 designs. The three-foot medium-gain antenna is replaced on the 1969 design by an additional low-gain antenna. The low-gain antennas are identical for both spacecraft. The 1969 stabilization and control system uses the same valve arrangement and electronics as the 1971 but has different tankage and a smaller nozzles. The midcourse engine for the 1969 spacecraft, including the valving and plumbing, is also identical with the exception of the deletion of one propellant tank.

An additional common element exists in the design concept of retaining a modular approach to both the 1969 and 1971 spacecraft. Thus, the various critical electrical subsystems and certain elements of the structure (e.g., equipment mounting panels, drive gears, bearings, thermal louvers, insulation materials, and pressure bottles) are of a modular design for both spacecraft systems. With this degree of similarity between the two configurations, a significant portion of the design effort is simply an extension of the 1969 design and test effort.

Those designs unique to the 1971 spacecraft will be instituted in parallel with the 1969 design. The 1969 ground test program provides early design verification data to the 1971 subsystem design (see Section IV 4).

II. SUMMARY OF MAJOR MILESTONE SCHEDULES

1. INTRODUCTION

This section presents the major milestone schedules for the Phase IB effort, 1969 flight, and 1971 missions, and a combination schedule of the 1969 test flight and 1971 and 1973 missions. It also discusses the basis for these schedules and identifies the critical areas and possible tradeoffs.

For reasons stated elsewhere in this volume, the 1971 mission program is considered to be an extension of the 1969 flight test development effort with the exception of those design efforts not common to both designs. It is expected that in each area of specialization, when common designs exist, the same group will perform both efforts. In those design areas where the 1971 mission differs from the 1969 flight test, there is sufficient time in the schedule to allow a major effort to be expended on the 1969 test with a smaller parallel group developing the 1971 designs and, as the 1969 design effort decreases those personnel will be diverted to the 1971 mission.

2. PHASE IB SCHEDULE

The preliminary design definition activities for the 1971 Voyager spacecraft, and corresponsing 1969 test flight, cover an eight-month span, starting in early January 1966. The products of this activity are complete functional specifications, system and subsystem, and OSE and Phase II implementation planning documents. The over-all 1971 mission spacecraft is not by itself schedule constrained; however, the objectives of a test flight of parallel design in 1969 will require earlier design definition for 1971 than might normally be attempted. The objective of obtaining maximum design verification from the test flight requires common design efforts wherever possible. The Phase IB activities have been approached with this objective in mind.

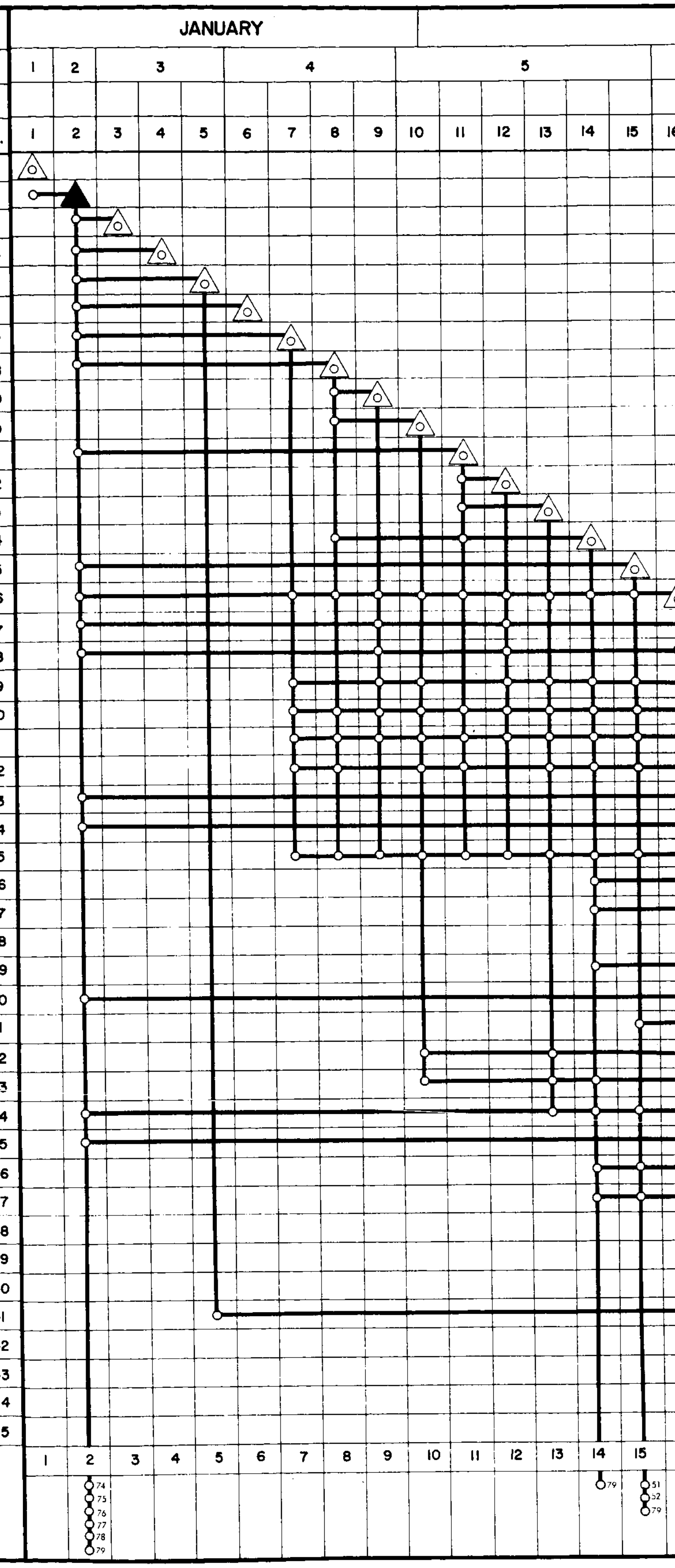
Figure 2-1 presents the Phase IB schedule in the TRW format of a simplified, uniform milestone matrix integration technique (SUMMIT) to highlight the prerequisite data required in accomplishing the selected milestones displayed. The activities involved in subsystem development are shown in detail in Section V4. Phase IB activities will be planned using PERT before the phase begins. Figure 2-1 provides a summary of the same activities and events. Upon receipt of the revised Voyager guidelines and specifications, system engineering will proceed, supported by subsystem engineering parametric data, in defining the system requirements imposed upon the spacecraft. This data results from the mission engineering, involving trajectory analysis, mission sequence of events, loads criteria, and guidelines of the mission experiment requirements. Specific system requirements data listed as milestone events will lead toward the preliminary design requirements review scheduled for the 7th week, following which the data is available for subsystem design analysis, leading toward the completion of subsystem and system design concepts early in the 12th week. Design Review No. 1 (12th week) establishes the subsystem baselines from which further definition results, using breadboard testing in conjunction with the design development processes. The purpose of Design Review No. 1 is to make certain that the system and subsystem requirements are clearly defined and that the conceptual design approach satisfies all requirements. The material to be reviewed includes: technical contract requirements, block diagrams, schematics, layouts, equipment specifications, technical work statements for system and subsystem design, development and test, and advanced procurement requirements.

JPL participation in all design reviews is expected. In addition a formal JPL review is scheduled for the 14th week to assess the system and subsystem concepts and to solidify the spacecraft interfaces. The JPL design review will result in release of updated subsystem requirements, interface specifications, and purchase orders for long lead time, high reliability parts.

The activities following this key design review lead to subsystem preliminary designs for both the 1971 and 1969 test flight, culminating in Design Review No. 2, scheduled for the 26th week. Intermediate milestones involving reliability assessment and materials and process specifications submitted are also shown. Design Review No. 2 verifies the adequacy of implementations of design concepts. The material to be reviewed includes:

- Detailed layouts and schematics
- Lists of material, parts and processes, and related specifications
- Results of development tests
- Reliability data
- Specifications for subcontract items
- Weight, volume, power requirements

TARGET	MONTH	
	WEEK	
	EXPECTED/ACTUAL COMPLETION	
MILESTONE EVENT		NO.
CONTRACT GO AHEAD		1
REVISED JPL MISSION SPECIFICATION AND TASKS ESTABLISHED		2
PHASE IB WORK PACKAGE DEVELOPMENT PLANS ISSUED		3
MODIFIED PROJECT MANAGEMENT, DOCUMENTATION AND CONFIGURATION CONTROL PLANS SUBMITTED		4
PREFERRED PARTS LIST ISSUED, PROCUREMENT PLAN SUBMITTED		5
UPDATED SAFETY AND FACILITIES PLANS SUBMITTED		6
UPDATED RELIABILITY AND WEIGHTS BUDGET ISSUED 1969, 1971		7
EXTREME AND NOMINAL 1969 REFERENCE TRAJECTORIES SELECTED		8
PRELIMINARY 1969 SEQUENCE OF EVENTS SELECTED		9
PRELIMINARY 1969 LOADS CRITERIA ISSUED		10
EXTREME AND NOMINAL 1971 REFERENCE TRAJECTORIES SELECTED		11
PRELIMINARY 1971 SEQUENCE OF EVENTS SELECTED		12
PRELIMINARY 1971 LOADS CRITERIA ISSUED		13
THERMAL ENVIRONMENT - DEFINED 1969 AND 1971		14
MAGNETIC CONTROL PLAN ISSUED		15
COMMUNICATION AND DATA HANDLING PRELIMINARY DESIGN REQUIREMENTS AND DSIF INTERFACE DEFINITION ISSUED (1969 AND 1971)		16
COMMAND LIST ISSUED 1969 AND 1971		17
TELEMETRY REQUIREMENTS ISSUED 1969 AND 1971		18
CONTROL SEQUENCING AND COMMAND PRELIMINARY DESIGN REQUIREMENTS ISSUED 1969 AND 1971		19
MIDCOURSE PROPULSION PRELIMINARY DESIGN REQUIREMENTS ISSUED 1969 - 1971		20
SPACECRAFT STABILIZATION AND CONTROL - (REACTION CONTROL) - PRELIMINARY DESIGN REQUIREMENTS ISSUED 1969 AND 1971		21
GUIDANCE SENSOR PRELIMINARY DESIGN REQUIREMENTS ISSUED		22
UPDATED QUALITY PLAN SUBMITTED (45 DAYS)		23
UPDATED RELIABILITY PLAN SUBMITTED (45 DAYS)		24
PRELIMINARY DESIGN REQUIREMENTS REVIEW		25
POWER BUDGET ISSUED 1969, 1971		26
TELEMETRY LISTS ISSUED 1969 AND 1971		27
SPACECRAFT INSTRUMENTATION REQUIREMENTS ISSUED (1969 AND 1971)		28
MASS PROPERTIES BUDGET REVISED AND ISSUED		29
SYSTEM MAINTAINABILITY REQUIREMENTS ISSUED		30
OPERATIONAL CONSTRAINTS ON SYSTEM DESIGN REVIEWED		31
SPACECRAFT DYNAMICS ANALYSIS COMPLETED		32
STRUCTURAL LOADS ANALYSIS COMPLETED		33
PRELIMINARY EXPERIMENT INTERFACE SPECIFICATION ISSUED		34
PRELIMINARY EMC PLAN ISSUED		35
SYSTEM AND SUBSYSTEM DESIGN CONCEPTS COMPLETE		36
PRELIMINARY SPACECRAFT CONFIGURATION LAYOUT COMPLETED		37
SPACECRAFT MOCKUPS STARTED		38
DESIGN REVIEW NO. 1 (1969 AND 1971)		39
APPROVAL OF BREADBOARD TEST PLANS COMPLETED		40
START BREADBOARD TESTING		41
SPACECRAFT - LVS INTERFACE SPECIFICATION COMPLETED		42
SPACECRAFT - CAPSULE INTERFACE SPECIFICATION COMPLETED		43
SPACECRAFT - DSN AND MOS INTERFACE SPECIFICATION COMPLETED		44
PRELIMINARY RELIABILITY ASSESSMENT COMPLETED		45



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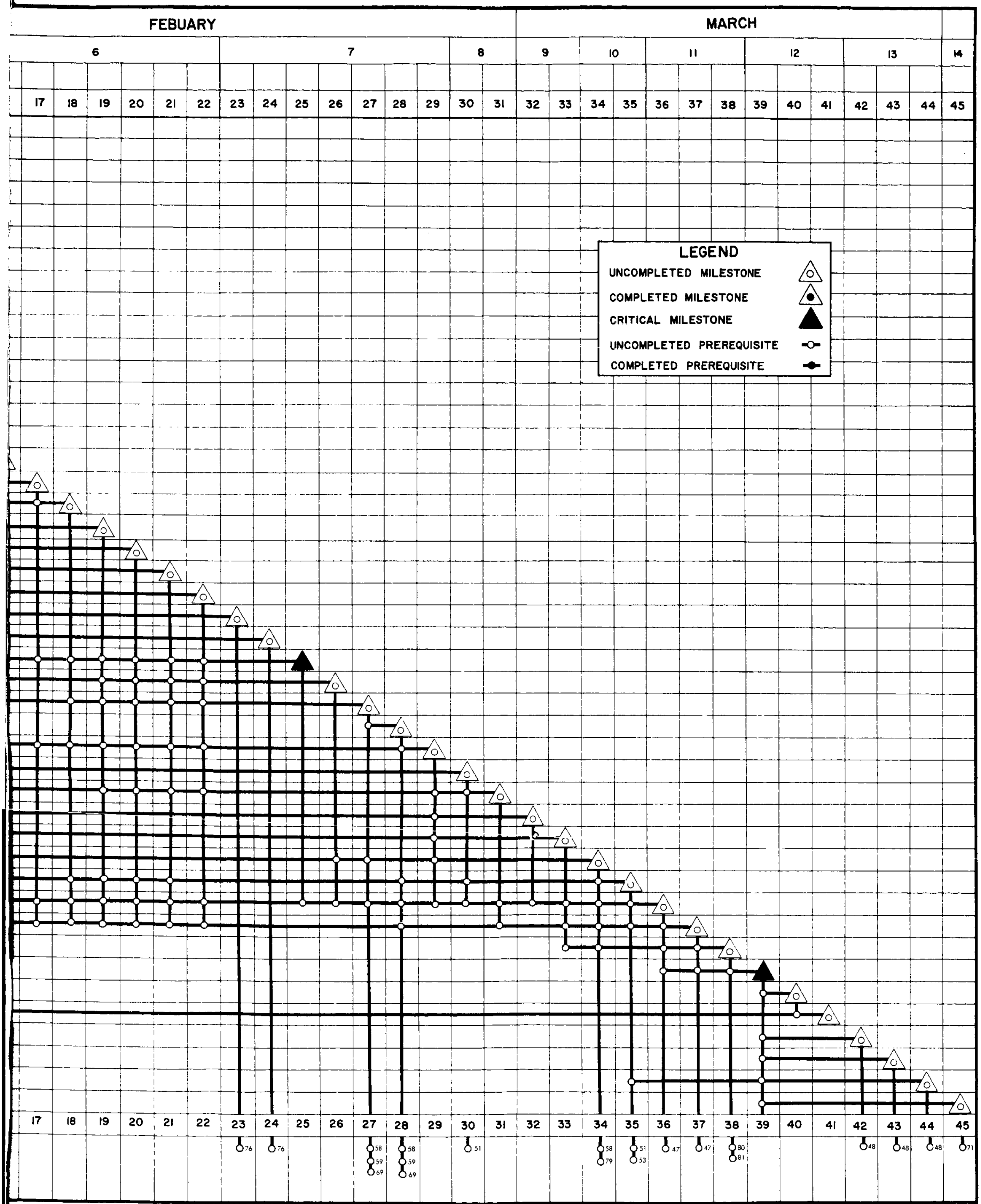
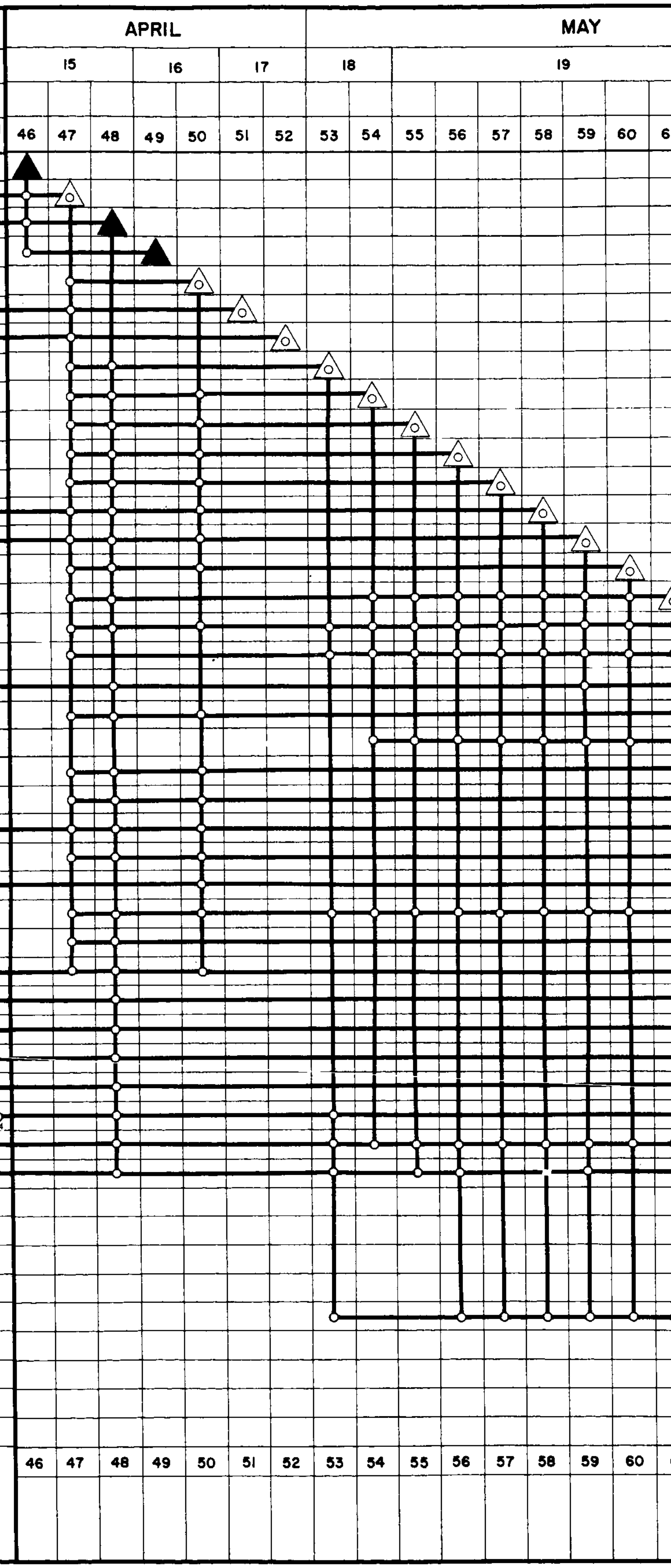


Figure 2-1. Phase IB Milestone Schedule

2

TARGET	MONTH	
	WEEK	
	EXPECTED/ACTUAL COMPLETION	

MILESTONE EVENT	NO.	COLUMN A
JPL DESIGN REVIEW	46	
UPDATED SUBSYSTEM DESIGN REQUIREMENTS RELEASED	47	36 37
INTERFACE SPECIFICATIONS RELEASED	48	42 43 44
LONG LEAD TIME HIGH RELIABILITY PARTS PURCHASE ORDERS RELEASED	49	
MATERIAL TESTS COMPLETED	50	
ELECTRICAL SPACECRAFT DESIGN INTEGRATION TESTING REQUIREMENTS ISSUED	51	15 30
MECHANICAL SPACECRAFT DESIGN INTEGRATION TESTING REQUIREMENTS ISSUED	52	15
PRELIMINARY CABLE AND PLUMBING ROUTING DESIGN COMPLETED	53	
PRELIMINARY STRUCTURAL SUBSYSTEM DESIGN COMPLETED 1969	54	
PRELIMINARY MIDCOURSE ENGINE DESIGN COMPLETED 1969 AND 1971	55	
PRELIMINARY THERMAL CONTROL SUBSYSTEM DESIGN COMPLETED 1969 AND 1971	56	
PRELIMINARY STABILIZATION AND CONTROL SUBSYSTEM DESIGN COMPLETED 1969	57	
PRELIMINARY CENTRAL SEQUENCING AND COMMAND SUBSYSTEM COMPLETED 1969 AND 1971	58	27 28 34
PRELIMINARY COMMUNICATIONS AND DATA HANDLING SUBSYSTEM COMPLETED 1969 AND 1971	59	27 28
PRELIMINARY POWER SUBSYSTEM DESIGN COMPLETED 1969	60	
1969 MASS PROPERTIES, POWER BUDGET AND TELEMETRY BUDGET ISSUED	61	
PRELIMINARY POP AND BODY MOUNTED EXPERIMENT PACKAGE DESIGN COMPLETED	62	
OSE END ITEM LIST AND PRELIMINARY DESIGN COMPLETED - 1969	63	
MDE-DSN INTERFACE SPECIFICATION RELEASED	64	44
PRELIMINARY STRUCTURE SUBSYSTEM DESIGN COMPLETED 1971	65	
1969 DESIGN REVIEW AND RELIABILITY ASSESSMENT	66	
PRELIMINARY RETROPROPULSION MOTOR DESIGN COMPLETED 1971	67	
PRELIMINARY STABILIZATION AND CONTROL DESIGN COMPLETED 1971	68	
PRELIMINARY CAPSULE COMMUNICATIONS AND DATA HANDLING SUBSYSTEM DESIGN COMPLETED 1971	69	27 28
PRELIMINARY POWER SUBSYSTEM DESIGN - 1971	70	
MATERIAL AND PROCESS SPECIFICATIONS SUBMITTED	71	45
DESIGN REVIEW NO. 2 (1969 AND 1971)	72	
OSE DESIGNS COMPLETED AND SUBMITTED	73	
MANUFACTURING PLAN SUBMITTED	74	2
ASSEMBLY AND CHECKOUT PLAN SUBMITTED	75	2
INTEGRATED TEST PLAN SUBMITTED	76	2 23 24
CONTAMINATION CONTROL PLAN SUBMITTED	77	2
LAUNCH OPERATIONS PLAN SUBMITTED	78	2
EXPERIMENT DESIGN INTEGRATION PLAN SUBMITTED	79	2 14 15 34
UPDATED CONFIGURATION MODEL 1969, COMPLETED	80	38
UPDATED CONFIGURATION MODEL - 1971 COMPLETED	81	38
1971 RELIABILITY ASSESSMENT COMPLETE	82	
JPL DESIGN REVIEW MEETING	83	
PHASE II WORK PACKAGE AND COST PLAN SUBMITTED	84	
SPACECRAFT FUNCTIONAL SPECIFICATION SUBMITTED - 1969 AND 1971	85	
SUBSYSTEM FUNCTIONAL SPECIFICATION SUBMITTED - 1969 AND 1971	86	
OSE FUNCTIONAL SPECIFICATION SUBMITTED - 1969, 1971	87	
ALL CIRCUIT DIAGRAMS RELEASED - 1969	88	
STRUCTURAL DRAWINGS FOR TEST MODELS RELEASED - 1969	89	
	90	



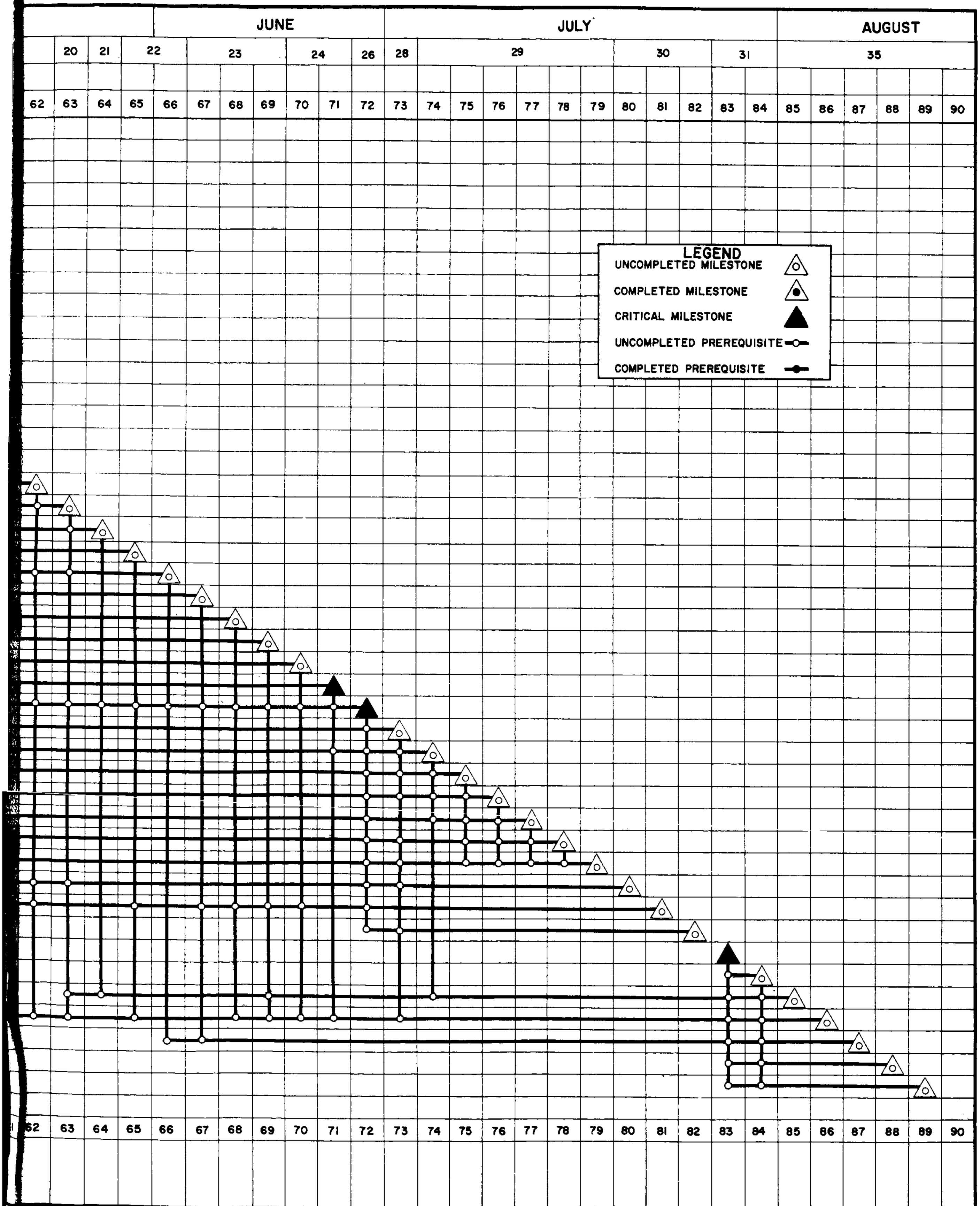


Figure 2-1. Phase IB Milestone Schedule (Continued)

2

As further subsystem design definition proceeds, the Phase II implementation plan is completed, spacecraft configuration models for 1971 and 1969 are completed, and system reliability assessments finished leading toward the second key JPL design review (28th week). The models are presented at this time and the previously submitted Phase II planning documents reviewed.

The system and subsystem functional specification, OSE specification, Phase II work package, and work plan will be submitted as revised by the design review. Structural model drawings for the 1969 test flight are prepared during Phase IB and released at the end of this phase.

A vital factor in the schedule is the early definition of both 1971 and 1969 spacecraft, with early preparation for the 1969 test flight using as many common subsystem designs as possible. Thus the configuration models are proposed for Phase IB construction to be used as design tools for spacecraft configuration development, followed by configuration control during Phase II.

3. PHASE II SCHEDULES

Figure 2-2 presents the summary schedule of the task-time relationships proposed for the 1969 flight test and the 1971 and 1973 missions. Figure 2-3 presents the summary schedule of the task-time relationships, for the 1969 flight test. The summary schedule of the task-time relationships for the 1971 mission is shown in Figure 2-4. Figure 2-5 presents the detailed test facilities schedule for the spacecraft flight approval portion of the 1971 mission. Figure 2-6 presents a detailed schedule of the PTM type approval testing for 1971.

The schedules were generated for each launch 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 operation, shipping, spacecraft flight acceptance testing, and assembly and checkout operations for both the 1969 and

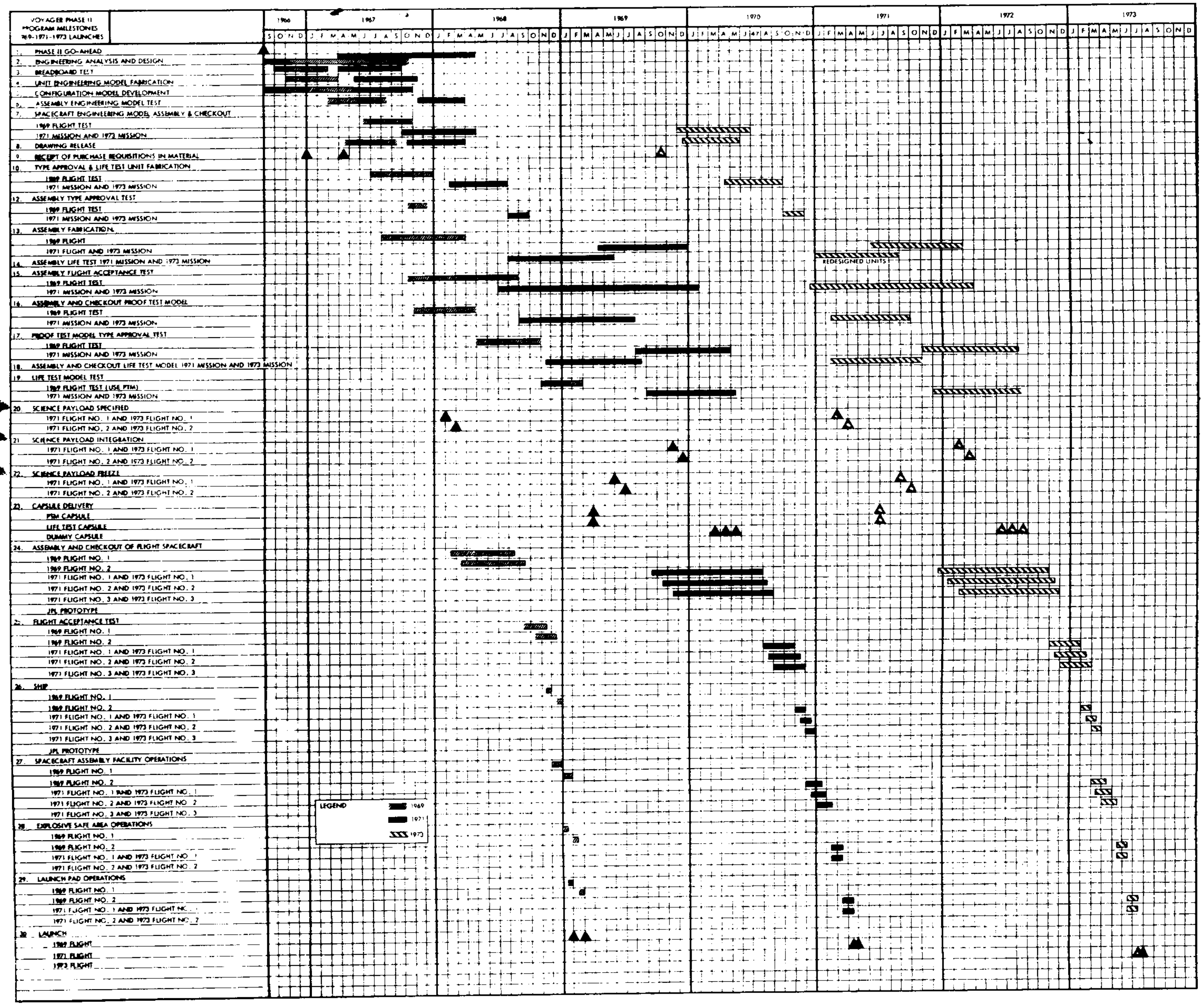


Figure 2-2. Voyager Phase II Program Milestones, 1969-1971-1973 Launches

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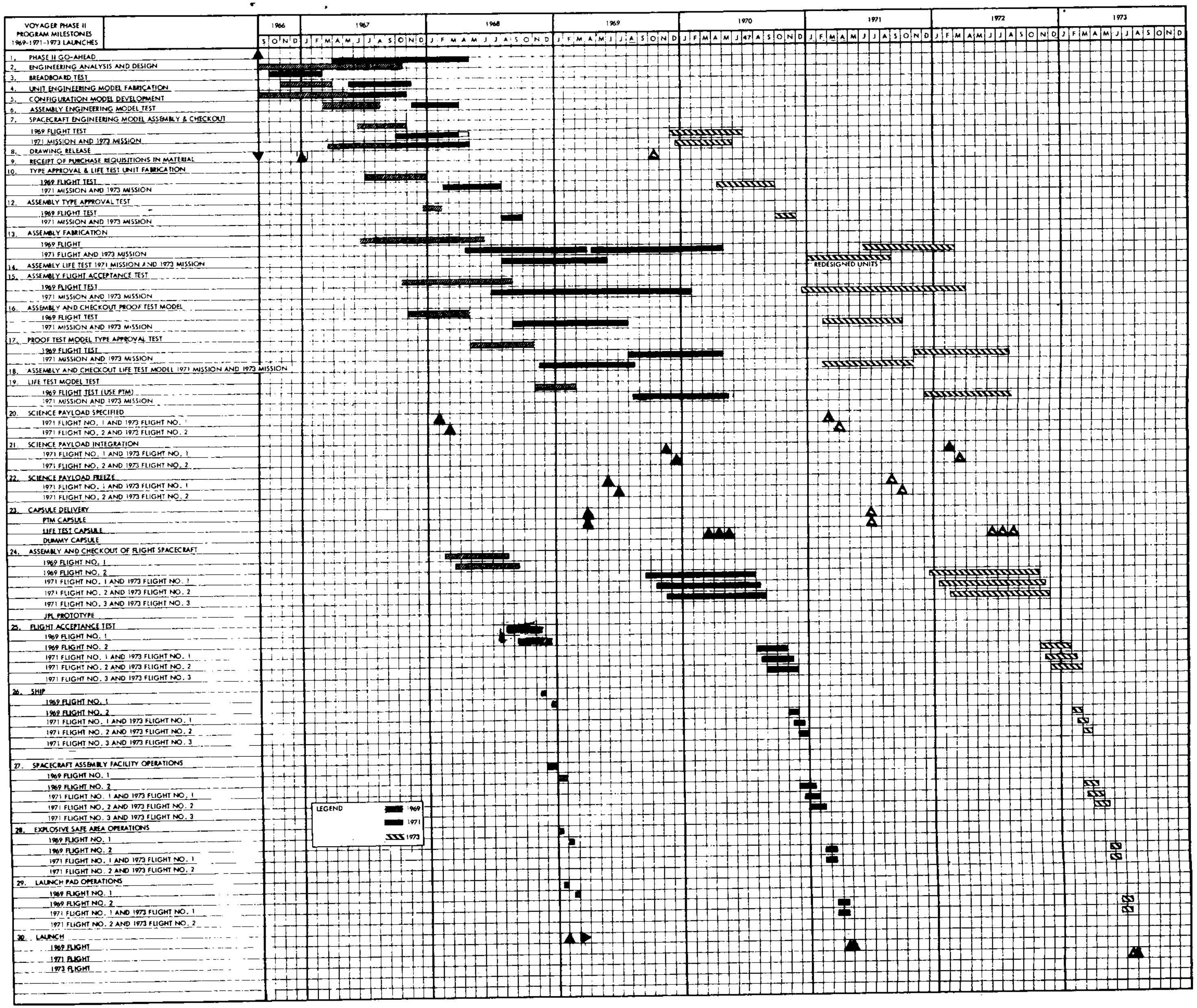
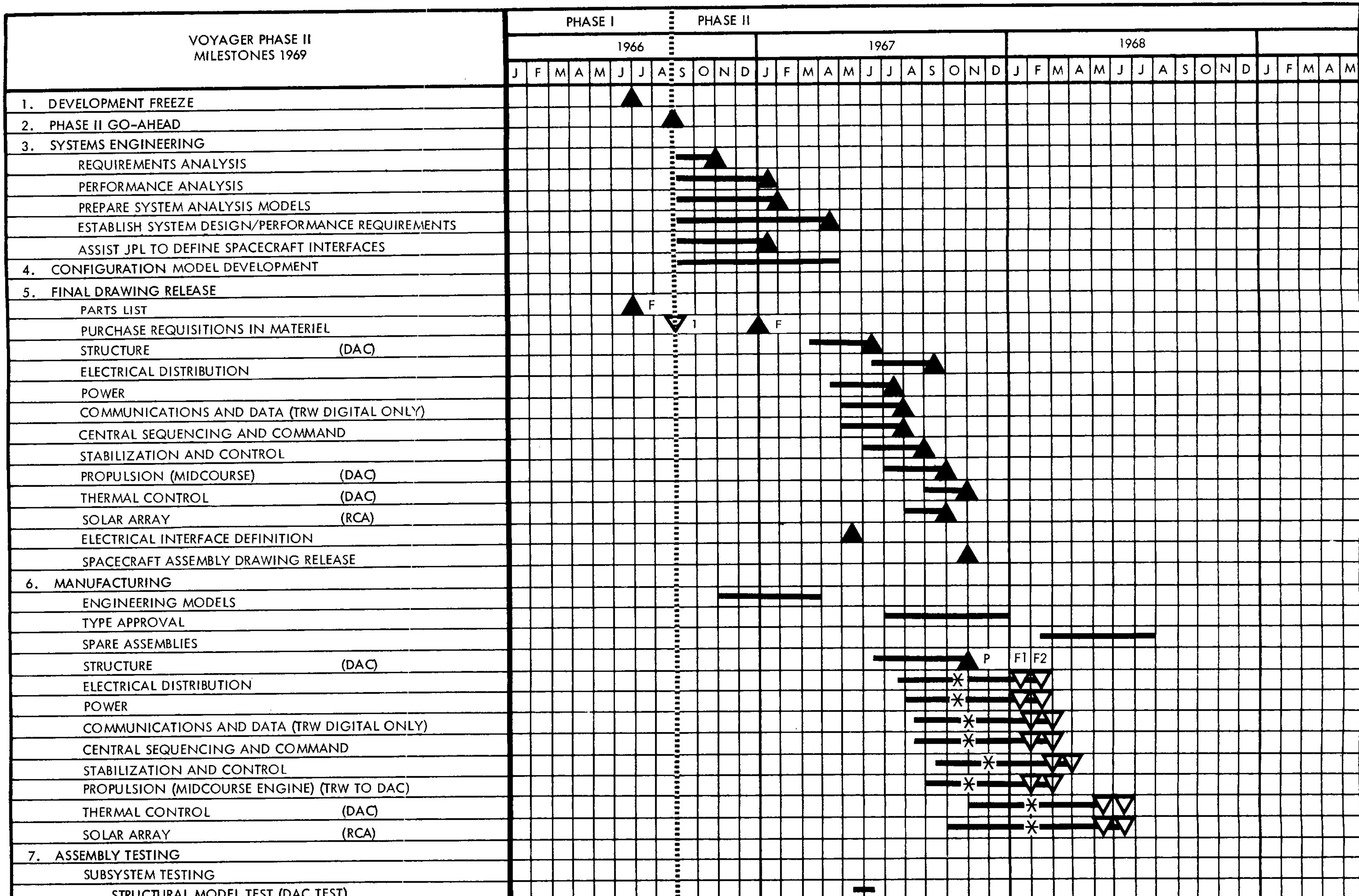
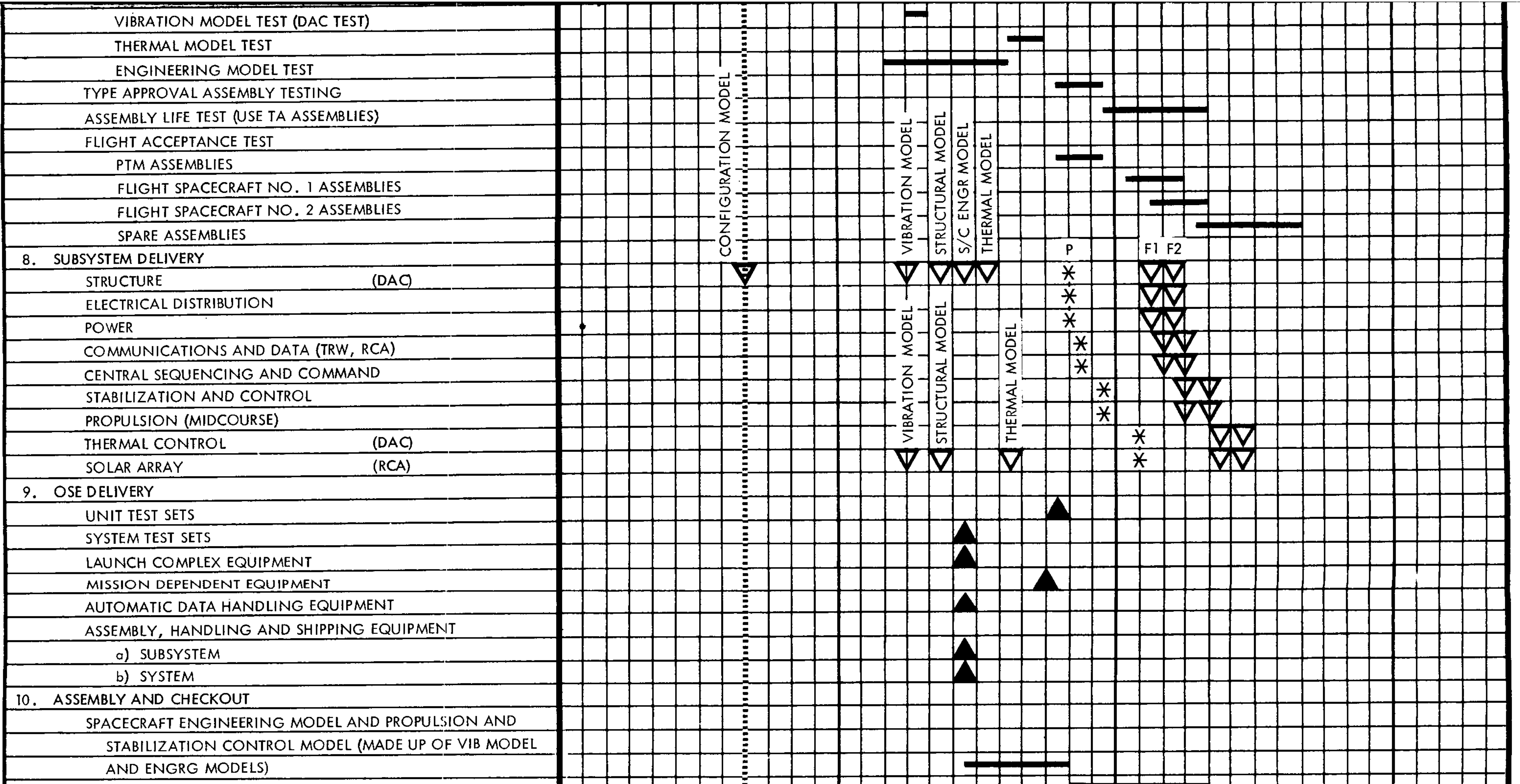


Figure 2-2. Voyager Phase II Program Milestones, 1969-1971-1973 Launches


VOYAGER PHASE II
MILESTONES 1969
PHASE I
PHASE II
1966
1967
1968
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
1. DEVELOPMENT FREEZE
2. PHASE II GO-AHEAD
3. SYSTEMS ENGINEERING
REQUIREMENTS ANALYSIS
PERFORMANCE ANALYSIS
PREPARE SYSTEM ANALYSIS MODELS
ESTABLISH SYSTEM DESIGN/PERFORMANCE REQUIREMENTS
ASSIST JPL TO DEFINE SPACECRAFT INTERFACES
4. CONFIGURATION MODEL DEVELOPMENT
5. FINAL DRAWING RELEASE
PARTS LIST
F
PURCHASE REQUISITIONS IN MATERIEL
1
F
STRUCTURE (DAC)
ELECTRICAL DISTRIBUTION
POWER
COMMUNICATIONS AND DATA (TRW DIGITAL ONLY)
CENTRAL SEQUENCING AND COMMAND
STABILIZATION AND CONTROL
PROPULSION (MIDCOURSE) (DAC)
THERMAL CONTROL (DAC)
SOLAR ARRAY (RCA)
ELECTRICAL INTERFACE DEFINITION
SPACECRAFT ASSEMBLY DRAWING RELEASE
6. MANUFACTURING
ENGINEERING MODELS
TYPE APPROVAL
SPARE ASSEMBLIES
STRUCTURE (DAC)
P
F1
F2
ELECTRICAL DISTRIBUTION
POWER
COMMUNICATIONS AND DATA (TRW DIGITAL ONLY)
CENTRAL SEQUENCING AND COMMAND
STABILIZATION AND CONTROL
PROPULSION (MIDCOURSE ENGINE) (TRW TO DAC)
THERMAL CONTROL (DAC)
SOLAR ARRAY (RCA)
7. ASSEMBLY TESTING
SUBSYSTEM TESTING
STRUCTURAL MODEL TEST (DAC TEST)


VIBRATION MODEL TEST (DAC TEST)
THERMAL MODEL TEST
ENGINEERING MODEL TEST
TYPE APPROVAL ASSEMBLY TESTING
ASSEMBLY LIFE TEST (USE TA ASSEMBLIES)
FLIGHT ACCEPTANCE TEST
PTM ASSEMBLIES
FLIGHT SPACECRAFT NO. 1 ASSEMBLIES
FLIGHT SPACECRAFT NO. 2 ASSEMBLIES
SPARE ASSEMBLIES
8. SUBSYSTEM DELIVERY
STRUCTURE (DAC)
ELECTRICAL DISTRIBUTION
POWER
COMMUNICATIONS AND DATA (TRW, RCA)
CENTRAL SEQUENCING AND COMMAND
STABILIZATION AND CONTROL
PROPULSION (MIDCOURSE)
THERMAL CONTROL (DAC)
SOLAR ARRAY (RCA)
9. OSE DELIVERY
UNIT TEST SETS
SYSTEM TEST SETS
LAUNCH COMPLEX EQUIPMENT
MISSION DEPENDENT EQUIPMENT
AUTOMATIC DATA HANDLING EQUIPMENT
ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT
a) SUBSYSTEM
b) SYSTEM
10. ASSEMBLY AND CHECKOUT
SPACECRAFT ENGINEERING MODEL AND PROPULSION AND
STABILIZATION CONTROL MODEL (MADE UP OF VIB MODEL
AND ENGRG MODELS)
CONFIGURATION MODEL
VIBRATION MODEL
STRUCTURAL MODEL
S/C ENGR MODEL
THERMAL MODEL
P
F1 F2
VIBRATION MODEL
STRUCTURAL MODEL
THERMAL MODEL

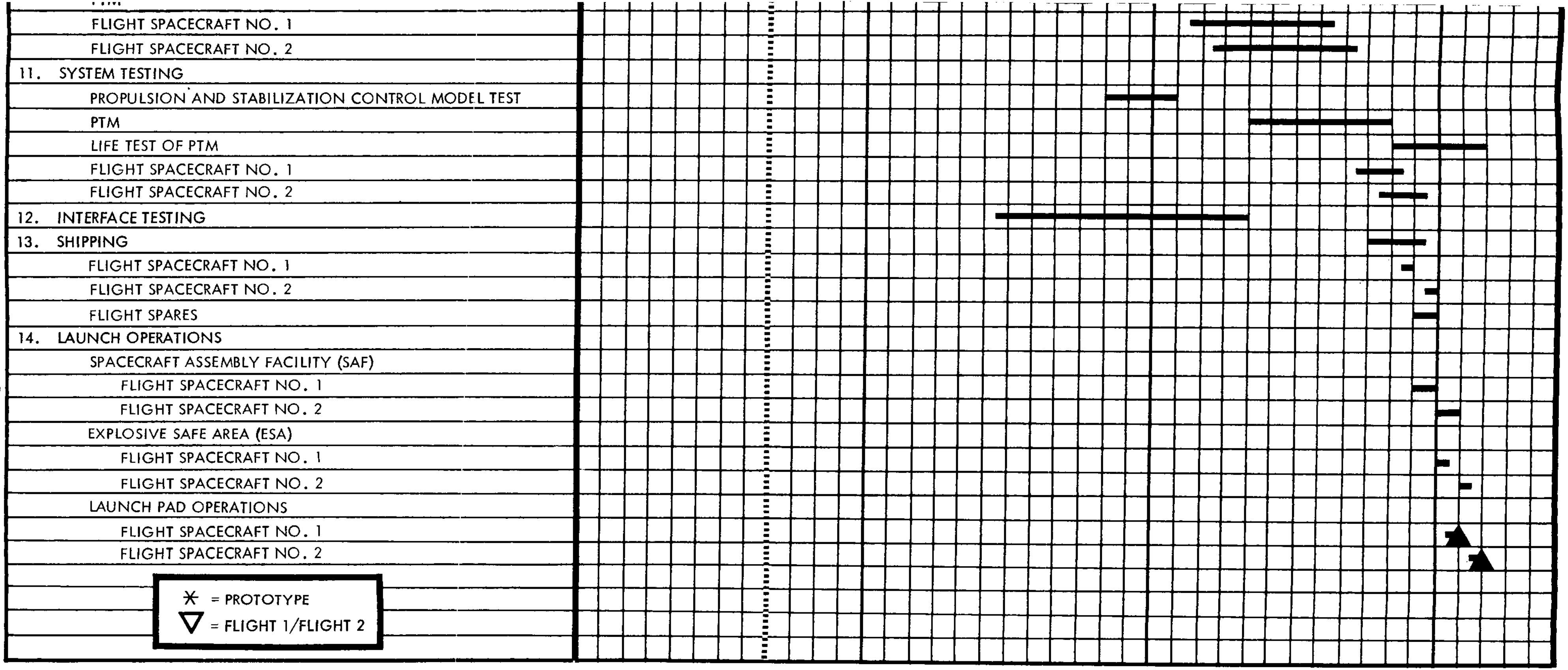
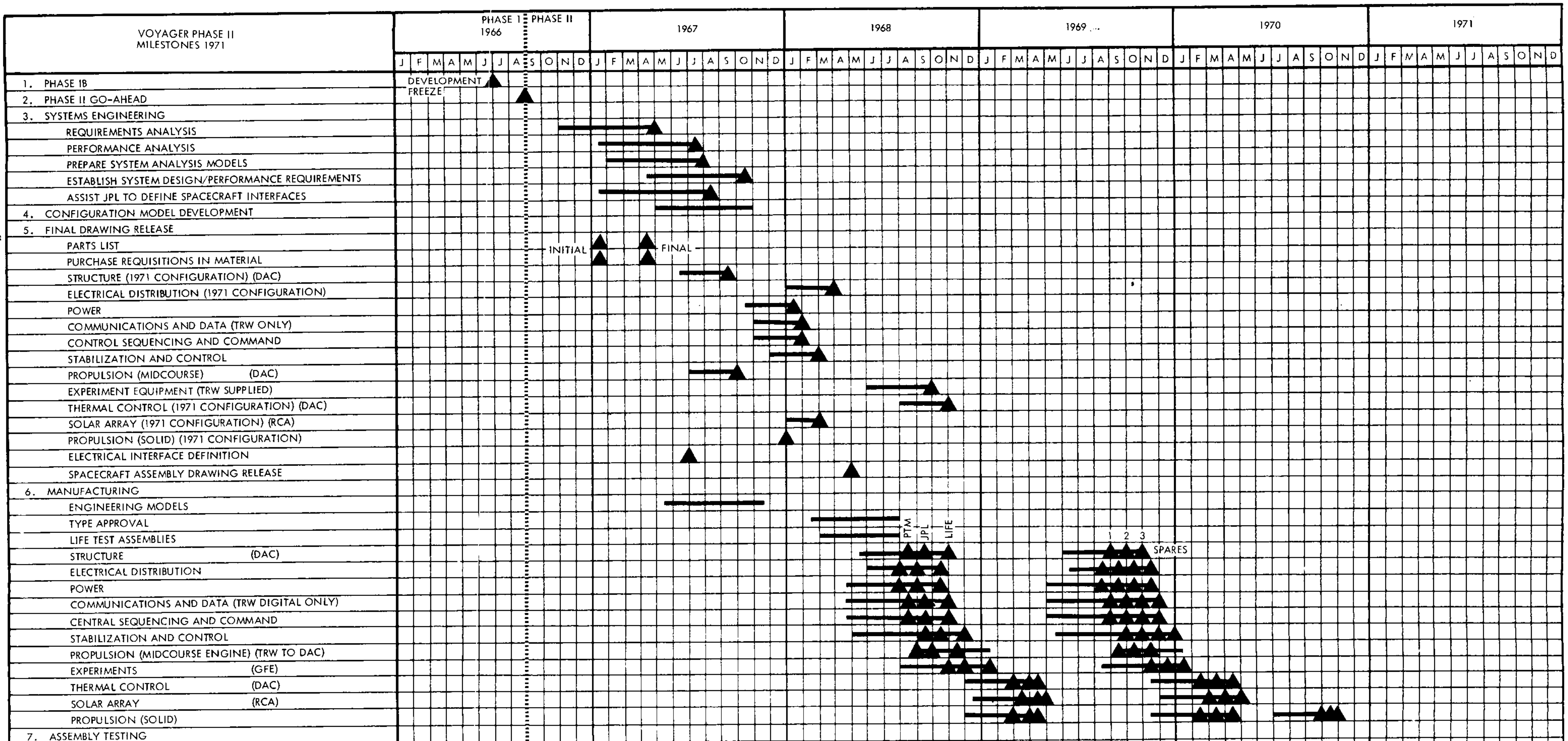
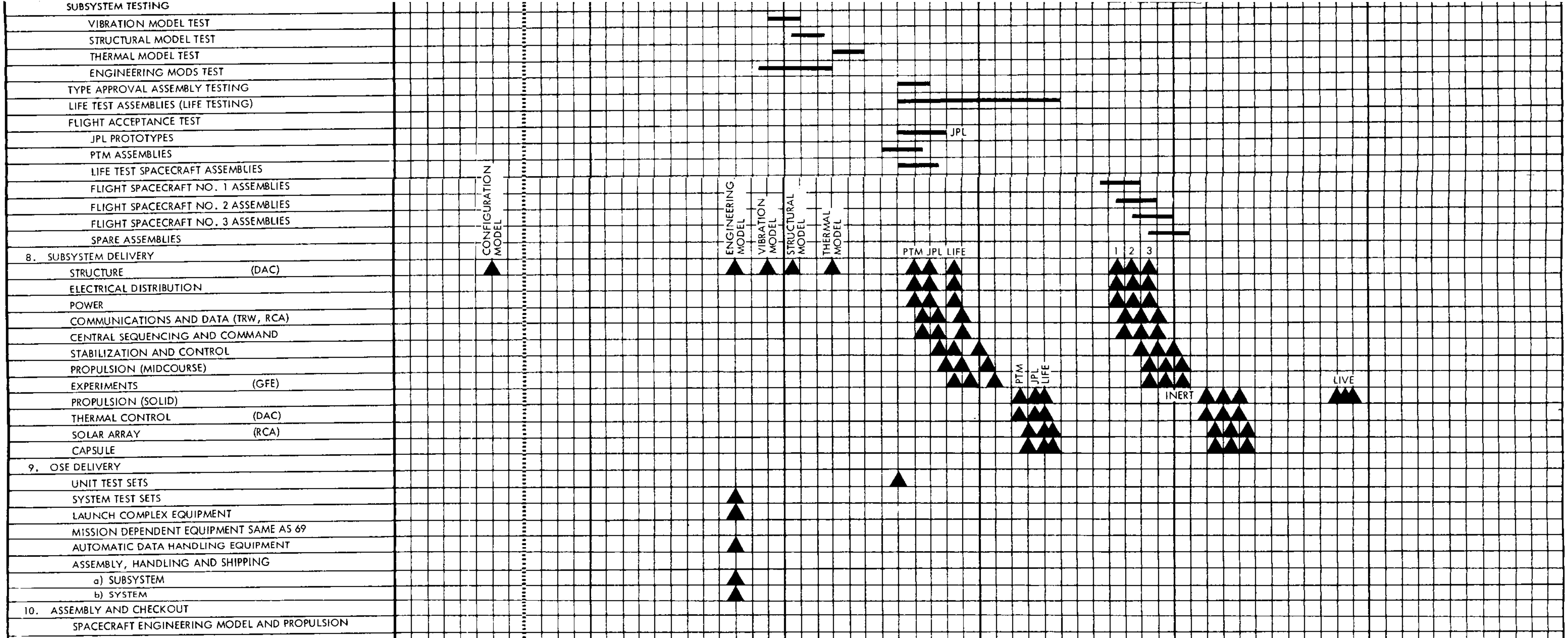


Figure 2-3. Voyager Phase II Milestones, 1969


VOYAGER PHASE II
MILESTONES 1971
PHASE 1
PHASE II
1966
1967
1968
1969
1970
1971
J F M A M J J A S O N D
1. PHASE IB
2. PHASE II GO-AHEAD
3. SYSTEMS ENGINEERING
REQUIREMENTS ANALYSIS
PERFORMANCE ANALYSIS
PREPARE SYSTEM ANALYSIS MODELS
ESTABLISH SYSTEM DESIGN/PERFORMANCE REQUIREMENTS
ASSIST JPL TO DEFINE SPACECRAFT INTERFACES
4. CONFIGURATION MODEL DEVELOPMENT
5. FINAL DRAWING RELEASE
PARTS LIST
PURCHASE REQUISITIONS IN MATERIAL
STRUCTURE (1971 CONFIGURATION) (DAC)
ELECTRICAL DISTRIBUTION (1971 CONFIGURATION)
POWER
COMMUNICATIONS AND DATA (TRW ONLY)
CONTROL SEQUENCING AND COMMAND
STABILIZATION AND CONTROL
PROPULSION (MIDCOURSE) (DAC)
EXPERIMENT EQUIPMENT (TRW SUPPLIED)
THERMAL CONTROL (1971 CONFIGURATION) (DAC)
SOLAR ARRAY (1971 CONFIGURATION) (RCA)
PROPULSION (SOLID) (1971 CONFIGURATION)
ELECTRICAL INTERFACE DEFINITION
SPACECRAFT ASSEMBLY DRAWING RELEASE
6. MANUFACTURING
ENGINEERING MODELS
TYPE APPROVAL
LIFE TEST ASSEMBLIES
STRUCTURE (DAC)
ELECTRICAL DISTRIBUTION
POWER
COMMUNICATIONS AND DATA (TRW DIGITAL ONLY)
CENTRAL SEQUENCING AND COMMAND
STABILIZATION AND CONTROL
PROPULSION (MIDCOURSE ENGINE) (TRW TO DAC)
EXPERIMENTS (GFE)
THERMAL CONTROL (DAC)
SOLAR ARRAY (RCA)
PROPULSION (SOLID)
7. ASSEMBLY TESTING
DEVELOPMENT FREEZE
INITIAL
FINAL
PTM
JPL
LIFE
1
2
3
SPARES


SUBSYSTEM TESTING
VIBRATION MODEL TEST
STRUCTURAL MODEL TEST
THERMAL MODEL TEST
ENGINEERING MODS TEST
TYPE APPROVAL ASSEMBLY TESTING
LIFE TEST ASSEMBLIES (LIFE TESTING)
FLIGHT ACCEPTANCE TEST
JPL PROTOTYPES
PTM ASSEMBLIES
LIFE TEST SPACECRAFT ASSEMBLIES
FLIGHT SPACECRAFT NO. 1 ASSEMBLIES
FLIGHT SPACECRAFT NO. 2 ASSEMBLIES
FLIGHT SPACECRAFT NO. 3 ASSEMBLIES
SPARE ASSEMBLIES
8. SUBSYSTEM DELIVERY
STRUCTURE (DAC)
ELECTRICAL DISTRIBUTION
POWER
COMMUNICATIONS AND DATA (TRW, RCA)
CENTRAL SEQUENCING AND COMMAND
STABILIZATION AND CONTROL
PROPULSION (MIDCOURSE)
EXPERIMENTS (GFE)
PROPULSION (SOLID)
THERMAL CONTROL (DAC)
SOLAR ARRAY (RCA)
CAPSULE
9. OSE DELIVERY
UNIT TEST SETS
SYSTEM TEST SETS
LAUNCH COMPLEX EQUIPMENT
MISSION DEPENDENT EQUIPMENT SAME AS 69
AUTOMATIC DATA HANDLING EQUIPMENT
ASSEMBLY, HANDLING AND SHIPPING
a) SUBSYSTEM
b) SYSTEM
10. ASSEMBLY AND CHECKOUT
SPACECRAFT ENGINEERING MODEL AND PROPULSION
JPL
CONFIGURATION MODEL
ENGINEERING MODEL
VIBRATION MODEL
STRUCTURAL MODEL
THERMAL MODEL
PTM JPL LIFE
1 2 3
PTM
JPL
LIFE
INERT
LIVE

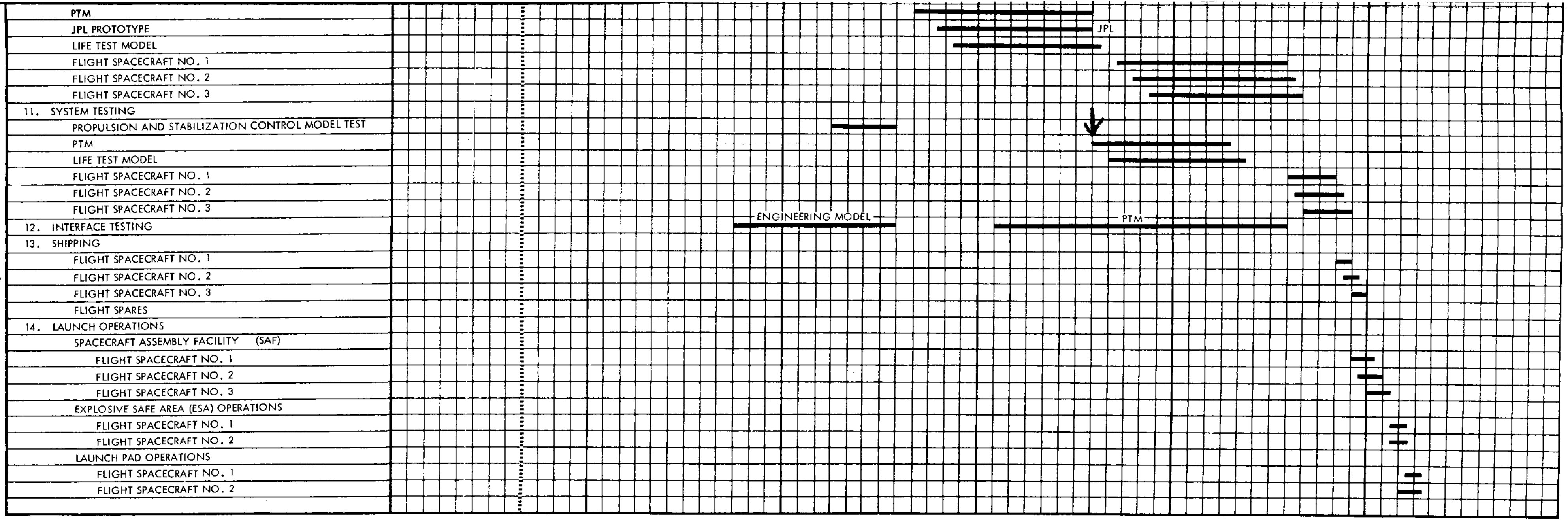


Figure 2-4. Voyager Phase II Milestones, 1971

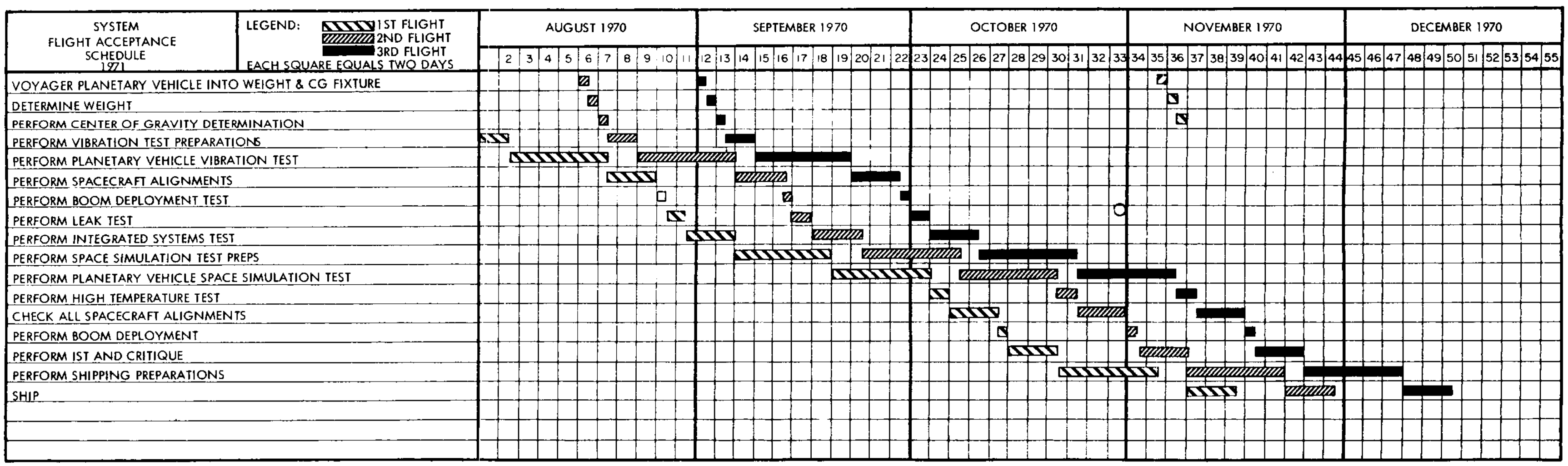


Figure 2-5. System Flight Acceptance Schedule, 1971

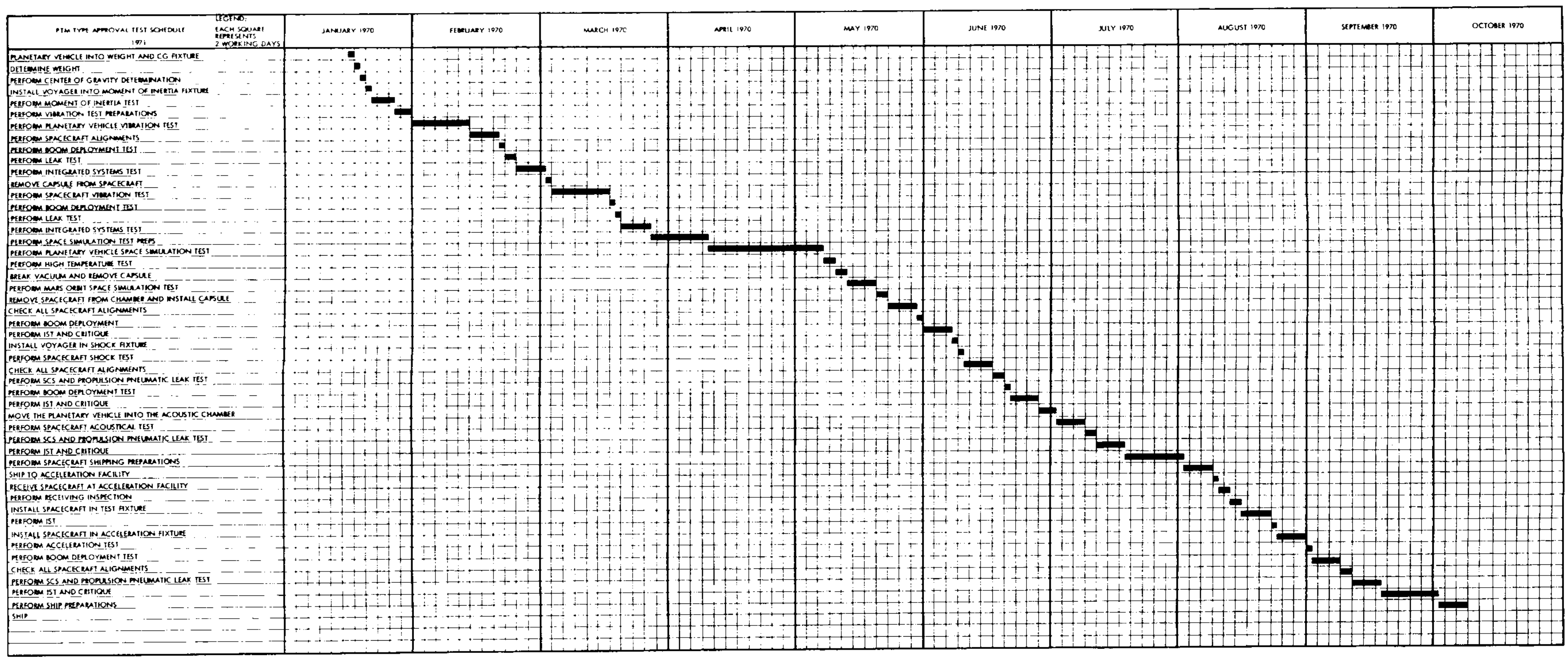


Figure 2-6. PTM Type Approval Test Schedule, 1971

1971 efforts. The next step was to define the delivery date for each subsystem as a function of the need date during the flight 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 the flight subsystems was defined. Next the time needed before the final drawing release was defined to enable manufacturing to plan and begin fabrication. However, the drawing release date required for the fabrication of flight spacecraft assemblies does not account for the requirements imposed by the need for fabricating (to flight drawing and procedures) the various type approval, proof tested model and life test assemblies.

It is at this point in the construction of the schedule that the policy diverges between the 1969 and 1971 effort because of the differences in the amount of schedule time left from the start of Phase II to the beginning of the manufacturing cycle and the requirements for fabricating and testing the various type approval, proof tested model, and life test models. Here the 1969 schedule requires a degree of concurrency to allow for the insertion of the type approval and life-test test programs, whereas the 1971 schedule allows considerable margins for a conservative approach in that subsystem type approval and subsystem life testing can be completed before start of fabrication of flight type subsystems.

To complete the 1969 test flight schedule, the type approval test of the subsystem units is scheduled to be completed simultaneously with the completion of the fabrication of the units for the proof test model spacecraft. This concurrency is expected to result in only minor modifications on the proof tested model units. The subsystem drawing release date was obtained from the times needed for fabrication and type approval test.

The time available from the start of Phase II to the start of manufacturing of the 1971 flight units allows a degree of freedom not contained in the 1969 schedule. There are two basic choices of how best to use the available time. One choice would be to delay the 1971 drawing release date sufficiently to allow any 1969 ground test results to be included in the 1971 design. This approach then would require that a series of test models would immediately precede the start of fabrication of the flight unit. The other choice is to continue the design effort from the end of the 1969 design effort and release the final 1971 drawings as soon thereafter as possible.

This second approach has been selected, for several reasons. The advantages lie in the ability to start fabrication of the 1971 type approval, life test, and proof test model units at an early date, thus, allowing these units to accumulate a significant test history prior to fabrication of the 1971 flight units. This approach can still allow for the inclusion of design changes resulting from the 1969 test program while at the same time providing test, procedural, and fabrication data useful to the correction of unforeseen discrepancies in the 1971 flight units.

Table 2-1 summarizes some of the additional considerations involved in scheduling the 1969 flight test and the 1971 mission efforts. Table 2-2 presents the objectives for both the 1969 flight test and the 1971 mission.

The schedules are based on a 5-day week, single shift operation thus providing for accelerated effort if it is required.

Beginning early in the program, formal periodic evaluation of the details of the planned schedule versus the actual schedule will search for slippages. If this evaluation suggests that the planned schedule is slipping the following will be accomplished:

a) A re-evaluation of the rest of the planned schedule will be made utilizing program experiences of that time and an updated schedule will be issued.

b) If the updated schedule still shows a slip from the actual, means for accelerating the work including overtime, will be used to bring the schedule up to date for the next review.

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

Table 2-1. Schedule Philosophy

- Maximum use of all data learned on the 1969 flight test
- Minimum interference of the 1969 flight test on the 1971 mission schedule
- No scientific objectives on the 1969 flight test
- Conservative estimates of all tasks associated with the 1969 flight test and the 1971 mission
- The ability to take full advantage of Phase IB for Phase II tasks if required
- 1969 flight test launch on 1 February 1969
- First 1971 mission launch on 15 April 1971
- Ship two flight spacecraft to ETR in 1969
- Use PTM spacecraft for life testing in 1969
- No special magnetic requirements in 1969 – however, magnetic testing will be started at the subsystem and system level (in-plant testing) to ascertain possible trouble areas for the 1971 design
- Minimum test program on PTM in 1969
- Two launches in 1969, 1 month apart
- Maximum use of the spacecraft engineering model in 1969 and 1971 for interface testing.

Table 2-2. Schedule Objectives and Achievement

Schedule Objectives	Schedule Achievement	
	1969	1971
• Final purchase requisitions 8 months before start of manufacture of type approval units.	6 months before start (1)	10 months before start
• Final drawing release for each subsystem 4 months before completion of first manufactured item.	2-1/2 months before completion (2)	4 months before completion
• Type approval testing of assemblies complete before completion of flight acceptance testing of proof test models assemblies.	Complete 1 month prior to completion of proof test model assemblies	Complete 2 weeks (3) after completion of proof test model assemblies
• Spacecraft engineering model assembly and checkout completed before starting proof test model assembly and checkout tasks.	Complete 2 weeks prior to start of proof test model	Complete 4 months prior to start of proof test model
• Proof test model assembly and checkout tasks completed before completing first flight spacecraft assembly and checkout.	Complete 2 weeks prior to completion of first flight spacecraft	Complete 1-1/2 months prior to start of first flight spacecraft
• Proof test model testing completed before completion of first flight acceptance test.	Complete 2 weeks prior to completion of first flight spacecraft	Complete 3-1/2 months prior to start of first flight spacecraft

1. Refer to Section II. 4.2.1a
2. Refer to Section II. 4.2.1
3. This ground rule not significant in 1971 due to available time to make corrections before the flight units are tested.

4. CRITICAL AREAS AND TRADEOFFS

4.1 Introduction

This section discusses the critical schedule areas and associated tradeoffs in the Voyager implementation plan. Critical areas are defined as those where a failure to successfully complete a given event on time has a high probability of either delaying the launch date or of launching without sufficient test confirmation (i.e., lower probability of mission success). Critical areas are discussed in terms of two periods of the Voyager implementation plan. The first period is concerned with the development cycle which begins with Phase IB and ends at the completion of subsystem drawing release and type approval testing. The second period overlaps the first, beginning with the drawing release date and ends at launch. Tradeoffs in terms of preventive action, the use of longer work weeks, and double shifts are identified in Section 3 above. The following sections discuss the critical areas for the 1969 test flight and 1971 mission respectively.

4.2 1969 Test Flight

4.2.1 Development Cycle

Evaluations of all subsystems have been made for schedule criticality. All subsystems were found to have schedule-critical items associated with the procurement of magnetically acceptable and high-reliability parts. Otherwise, the design and development of these subsystems contains no critical items except as discussed here.

a. Parts

The use of high reliability parts is considered a prerequisite to the 1969 program if it is to be a meaningful test for the 1971 mission. The procurement cycle for such parts (see Section V.5) can require as long as 14 months. The need for these parts, to be included in the type approval and flight units occurs 10 months after Phase II starts. Procurement of these parts thus is a critical area for the 1969 effort.

Three options in approaching this problem appear feasible:

- Purchasing high reliability parts from existing production which has established production standards commensurate with the Minuteman type of reliability requirements (e.g., typical of the Motorola, Fairchild, Texas Instrument, Minneapolis Honeywell productions). This has a disadvantage in that the parts specifications may not satisfy either JPL or TRW requirements or that the parts may be unacceptable from a magnetics standpoint.

- Procuring parts for the 1969 effort which are from a lot formation to be qualified to TRW/JPL standards but which are withdrawn for use after parameter drift screening but before lot qualification. This runs the risk that if a part fails the qualification phase, the unit will either have to be rebuilt or accepted "as is" with the pursuant risk.

- Provide an approved parts list early in Phase IB from which the designs must be selected. Identify, during Phase IB breadboard testing, those parts requiring a special effort to qualify as additions to the approved list, and negotiate an early purchase release during Phase IB for long lead time parts.

Of these alternates, TRW recommends the use of the last in that it provides high confidence in meeting schedules and attaining a more reliable design. This approach has been included in the Phase IB schedule in that the approval of the parts list and procurement plan is scheduled during the first month, the purchase orders for long lead time, high reliability parts and components occur in the fourth month.

b. Structure

The need date for the first flight type structure for 1969 occurs 14 months after Phase II start. To attain a high confidence in the structural design at this point it is necessary to have completed the vibration survey and structural tests sufficiently in advance of this need date to include any required changes. This indicates a need for these test structures within seven months of Phase II start, a critical schedule area. The tradeoffs here include either completing detailed structural layouts during Phase IB at the penalty of higher Phase IB costs or accepting a slip in the structural testing phase, which runs the risk of slipping the PTM and flight spacecrafts. TRW and Douglas recommend that the former alternative be selected to reduce the risk of slipping the 1969 launch. The early vibration survey test also provides and acceptable schedule margin for the design and development of the solar array panels. Additionally, a configuration model of the spacecraft will be completed during Phase IB to aid in the placement of subsystems, plumbing routing, mechanical interferences and cable routing.

c. Midcourse Propulsion

The need date for the midcourse monopropellant engine is set by the PTM spacecraft at 18 months after Phase II start. In order to achieve a high confidence in the engine system design at this time it is necessary to complete assembly testing at the propulsion subsystem level and system test in the propulsion and stabilization control model. The latter requirement dictates midcourse propulsion system delivery at 14 months after Phase II start.

The need date defines a critical schedule area unless the design and development testing is initiated during Phase IB. TRW proposes to proceed with the detailed design of the prototype system beginning in June of 1966, with design verification testing in July and August 1966. This tradeoff insures a higher cost in Phase IB but provides a corresponding higher degree of confidence in achieving the 1969 schedule.

d. Spacecraft Stabilization and Control

The need date for delivery of components and parts to begin fabrication of the type approval hardware for the stabilization and control subsystem is 10 months after Phase II start. Thus breadboard testing must be completed and engineering model design started during Phase IB to enable the release of engineering model drawings early in Phase II to avoid a critical schedule area. The other critical area involves the long lead time associated with the procurement of the gyro assembly. The gyro package procurement includes early specification of the gyro design and subcontractor selection. These factors may lead to additional cost for the gyro procurement but ensures the availability of test data from the subsystem and system engineering model in support of the final drawing release cycle.

e. Communications and Data Handling

The fabrication of the subsystem type approval units are scheduled for the 11th month of Phase II, with a drawing release cycle extending from the 8th to the 11th month. This requirement induces several critical schedule areas in the development of both the communications and data handling subsystems. To avoid any delay in the 1969 schedule the following tradeoffs are proposed:

- Tape recorder development with three speed features will require development effort during Phase IB to meet the 1969 schedule. This will include the fabrication of an engineering model with breadboard circuitry.
- Antenna gimbal drives will be designed and prototype models built and tested.

This approach again represents a tradeoff of higher Phase IB costs for an increased confidence in meeting the 1969 launch schedule.

f. Power

A critical factor in the attainment of the 1969 schedule is the power subsystem development, which includes the design of the solar array for the low temperature condition.

- Q-boards of solar panel segments need to be fabricated and tested over the extremes of temperature, and in particular below $-120^{\circ}C$. Materials tests will be made to evaluate cell performance and mechanical problems associated with the glass solar cell cover.

4.2.2 Fabrication, Assembly, and Test

The fabrication, assembly and test cycle begins with the fabrication of the type approval units and the assembly of the spacecraft engineering model. The type approval fabrication cycle for each subsystem is keyed to the drawing release cycle and the availability of the parts required. As discussed in Section 2.2.1, adequate precautions have been taken to ensure a high confidence in meeting the drawing release dates and the parts availability. No critical problem areas are seen in the fabrication of the type approval and subsequent PTM and flight units for the electrical units nor for the fabrication of the structure, solar array, and thermal control equipment. A critical area does exist, however, because of the concurrency of type approval testing and the fabrication of the PTM and flight units. A failure at this point will require either a redesign, a parts change, or a process change and could (depending on the nature of the required change) slip the delivery of the PTM and flight units. Although the likelihood of a design failure is low in the light of the development and test cycle on the engineering model, a parts change or a process change could induce a one- to two-week delay. If a failure is found, the technique used to circumvent delay will include:

a) A task force working on an accelerated schedule to determine the cause of the failure, recommend corrective action, and expedite the rework through the fabrication and test cycle.

b) In the event that the rework schedule is such that the above action is not completed in time to meet the need date for either the PTM or flight spacecraft, substitute hardware (e.g., engineering models) will be used to continue the assembly and checkout sequence. The spacecraft assembly and checkout schedule can be adjusted to accommodate the reworked unit at a later point in the schedule. Here again the use of an extended work week, overtime, and double shifts can be used to include the assembly and checkout of the reworked unit in the PTM or flight spacecrafts on a non-nominal interference basis.

The failure of the flight unit during flight approval testing presents a similar problem and requires a similar approach to recover. Here, however, the types of failures encountered are likely to be more of the workmanship and random part failure than of design deficiency. These types of failures do not present the likelihood of long rework and retest cycle in that replacements can be made and rework done on spare assemblies already in process.

The next possible critical area occurs during the assembly and checkout operations of the PTM and flight spacecraft, arising from the constraint that each operation on the flight spacecraft be preceded by the completion of that operation on the PTM. The kinds of problems encountered at this point in the schedule usually include mechanical interferences, intersubsystem electrical incompatability, OSE incompatibility, and procedural and computer programming difficulties. The likelihood of these types of problems occurring is low because of the previous experience gained in the assembly and checkout of the spacecraft engineering model and the updated spacecraft configuration model.

Mechanical interferences can be checked against the configuration model as part of the subsystem flight approval cycles. Electrical, procedural, and programming difficulties can be obvaited by temporarily bypassing that step in the sequence while the changes required are checked out on the engineering model, and by assigning a special task force on an extended work schedule to work out the solution. The deficiency can then be corrected later in the schedule on a noninterference basis. The choice of which of the two approaches to use depends on the nature and severity of the problem. Sufficient schedule margin over the flight spacecraft assembly and checkout sequence can be readily maintained.

A critical area may exist in the event a major failure occurs during PTM type approval testing. The first two months of testing include vibration and space simulation testing. This phase of testing uncovers most existing design deficiencies. It is programmed for completion one month prior to the beginning of flight spacecraft acceptance testing, which allows incorporation, on an accelerated basis, of reasonable changes.

Additional slack time is still available beyond the completion of flight acceptance testing in two ways. The first exists because of the conservative launch site schedule of 2 months, which can possibly be shortened to 1 month by an accelerated effort and by the real possibility of shipping the spacecraft in a completely assembled configuration, and performing a systems test without breaking configuration, and then proceeding with on-stand operations. (This is being done on the OGO-C launch from WTR, reducing the launch site time to at least half of that otherwise required.) The second possibility lies in utilizing the remaining portion of the launch window.

4.3 1971 Mission

The 1971 mission schedule (Figure 2-4) shows that there are no critical schedule areas in the development cycle. The drawing release cycle occurs during late 1967 and early 1968, thus providing a development time of approximately 24 months from Phase IB start or 16 months from Phase II start. This time is considered more than adequate considering that much of the 1971 designs are identical to those for 1969 and that the fabrication and test of the 1971-peculiar engineering models are scheduled for completion prior to the start of the 1971 drawing release cycle. In addition, the subsystem fabrication and type approval cycle is such as to allow 7 months for design adjustments if needed before beginning fabrication of the flight hardware. The start of flight fabrication is so placed as to allow for the inclusion of the 1969 test results up to and including the early portions of the test flight as well as the results of the 1971 subsystem life testing.

In the case of a failure in the 1969 test flight, there is still sufficient time to include changes in the 1971 spacecraft as late as 14 months after 1969 launch. A failure occurring during PTM type approval testing is most likely to occur during vibration or space simulation testing. This portion of the PTM tests is completed by the end of December 1969, allowing approximately 6 months to include design refinements. The 1971 life test model is scheduled to enter life test in August of 1969 and could proceed as long as 8 months before a detected failure would pose a 1971 launch schedule problem.

III. EFFECTS OF THE 1969 TEST FLIGHT ON THE 1971 MISSION

The 1969 test flight program will contribute significantly towards improving the probability of a successful 1971 mission. The 1969/1971 subsystem and system designs are essentially identical as indicated in Section I above. Differences exist in structural loading and thermal and electromagnetic interactions, which are attributable to the particular configuration arrangements, power availability and weight. The boost phase environment differs in that a different launch vehicle is planned, but the ensuing phase provides an accurate simulation of the coast environment.

The major factor which contributes to improving the success of the 1971 mission is the completion of the 1969 ground and flight test program (see Sections IV 3 and V 6). The 1969 ground test program begins to provide significant data on the performance of the subsystems during the engineering model phase. The problems arising from packaging provide meaningful data for gaining confidence in the final design. The engineering model test phase provides performance data over a wide range of design conditions such as temperature, vibration, magnetic characteristics, and power levels. Additional confidence is attained in terms of subsystem size, weight, and power consumption. Subsystem testing, using engineering models, also provides for a verification of internal subsystem and OSE compatibility. The extension of engineering model testing to the spacecraft levels provides for testing of intersubsystem compatibility, over-all spacecraft performance characteristics, magnetic characteristics, final verification of configuration arrangement, electromagnetic interface, and OSE and facilities checkout.

The completion of the 1969 subsystem type approval testing provides for high confidence in the proper functioning under severe environment conditions and verifies the procedures and processes used in the manufacturing phase. Failures uncovered during this test phase are useful in correcting design deficiences in the 1971 hardware. The extension of type approval testing to the proof test model spacecraft will again prove the performance characteristics of the major portions of the electrical subsystem which are applicable to the 1971 design. The process of assembly and checkout of the 1969 proof test model provides an opportunity to validate a large portion of the 1971 operational support equipment, assembly and checkout procedures, computer programs, and test facilities.

An important test benefit is provided by the 1969 ground test program in terms of providing reliability data on parts, subsystems and systems. Life testing of the 1969 proof test model spacecraft (see Section IV 3.7.2) will add to the confidence in the ability of the subsystem designs to survive the expected life requirements.

The completion of the 1969 launch and prelaunch operations with the two flight spacecraft and the engineering model spacecraft will provide a means of rehearsing and validating much of the 1971 operational support equipment, launch control equipment, procedures, checkout operations, on-stand operations, and terminal count procedure.

The data received from the 1969 test flight through powered flight and guidance acquisition will further ensure confidence in the subsystem designs. As the flight progresses, more meaningful data on the performance and survival of the subsystems will add confidence in the success of the 1971 mission. Failures occurring during early flight will provide design data for use in the 1971 design.

The conduct of the 1969 test flight effort also provides additional confidence in the success of the 1971 mission in the following areas:

a) Crew Training. The assembly, checkout, test, and launch crews will receive real experience in the conduct of their respective operations. The conduct of the engineering model and proof test model interface also add to crew training at the Deep Space Network, Spacecraft Flight Operations Facility and mission operations support centers.

b) Procedure and Computer Program Checkout. A large portion of 1969 test procedures and computer programs will be directly applicable to the 1971 mission. The 1969 test effort provides an opportunity to validate these documents.

c) OSE Checkout. Here again, a great deal of the OSE used in the 1969 effort is identical to that used for the 1971 mission, and the conduct of the 1969 effort provides an early opportunity to revise and validate this equipment and to improve the design in terms of failure detection.

d) Test Facility Checkout. It is planned to use the same test facilities for the 1969 test flight spacecraft as for the 1971 mission spacecraft. The use of the 1969 equipment in these facilities will provide a high confidence in their design and operations.

e) Manufacturing Checkout. The identical designs of much of the equipment fabricated for both the 1969 and 1971 programs permit a checkout of the manufacturing processes, assembly, lines, test equipment, and software controls. This will contribute to the confidence in fabricating high quality 1971 equipment and on-schedule performance. The qualification of the various vendors and subcontractors will be verified.

f) Schedule Confidence. The performance of the 1969 program provides high confidence through learning in performing to the 1971 schedule. Thus, the ability to "launch on time" is greatly enhanced by the 1969 effort.

g) Customer Interface. The working relationship between JPL and TRW will be completely worked out in every phase of the program prior to the 1971 mission.

h) Subcontract Interface. The working relationship between TRW and its subcontractors will be completely worked out in every phase of the program prior to the 1971 mission.

i) Tests. It will not be necessary to repeat breadboard, engineering model, type approval, and life testing on the assembly level for assemblies that are not redesigned or changed from the 1969 flight to the 1971 flight.

j) Drawings. It will not be necessary to release new drawings for assemblies that are not redesigned or changed from the 1969 flight to the 1971 flight.

k) Spares. Spare 1969 assemblies that are not redesigned or changed can be used for the 1971 spacecraft.

IV. TEST PLANNING

1. INTRODUCTION

The test activity and the test management program required for the Voyager project will begin with the selection of parts and continue through subsystem development, system assembly and checkout, systems testing, and launch. In this context, the test plan described in the following paragraphs has been constructed so that it is applicable to both the 1969 flight test spacecraft and to the spacecraft systems required for the 1971 Voyager mission. For additional clarification, the plan contains a separate discussion which describes the effect that the test data required from the 1969 flight test will have on the 1971 mission.

The over-all responsibility for implementing the test plan is the function of a special organization, the Test Office, reporting to the project manager. Supporting the project manager and the Test Office will be a Test Board (Figure 4-1), which has as its function the establishment

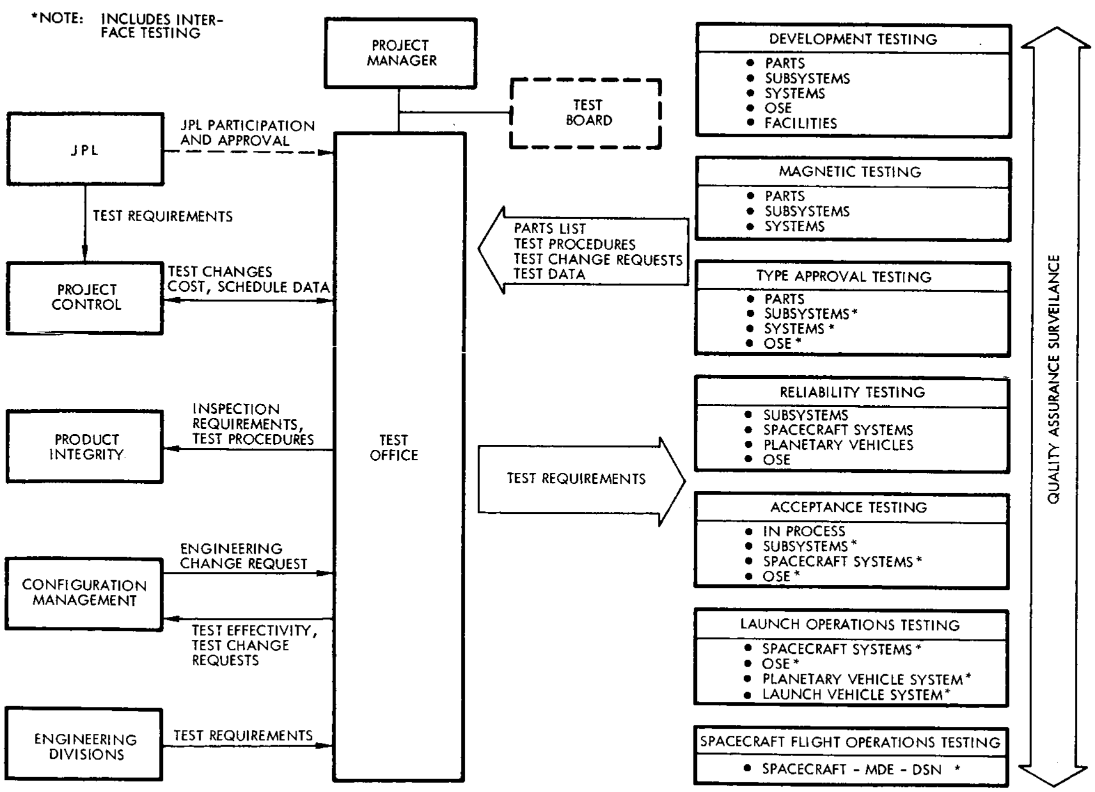


Figure 4-1. Interaction of Test Office with the Major Program Elements

of policy for activities directly relating to the test program. This board will review the functions and technical output of the Test Office and will coordinate with the Jet Propulsion Laboratories in appraising and updating the integrated test plan and project management activities.

2. THE TEST OFFICE

The importance of reliability and the magnitude of the test program required to develop the Voyager spacecraft has necessitated the establishment of a central test planning control, review, and reporting function. TRW proposes to satisfy these requirements by establishing a Test Office. The personnel assigned to this office will include full-time senior engineers who are experienced in each of the test disciplines. These engineers will report to a chief test engineer. Other support required by this office will be obtained from the responsible organizations.

Reporting directly to the Voyager project manager, the Test Office will be responsible for the following tasks:

a) Plan and implement a parts and materials program in accordance with Paragraph 3.9 of NPC-250-1.

b) Establish development, type approval, magnetics, reliability, interface, and design verification test requirements.

c) Prepare and maintain the integrated test plan; define the role of each test in the evaluation of system performance and reliability.

d) Establish the test plan schedule, evaluate and approve detailed test procedures in light of the test requirements, monitor test performance, and evaluate test results.

e) Define the use of test results in assessing the validity of reliability models and in correcting design deficiencies.

f) Participate in formal design reviews and approve detailed test plans as follows:

   First Design Review. Review and analyze the subsystem development test program; review and approve detailed test procedures for breadboard testing; review proposed parts list and test program; identify parts requirements and prepare for JPL approval per Paragraph 3.9 of NPC 250-1.

Second Design Review. Reivew, analyze, and critique breadboard test results; review and approve subsequent testing, including engineering model test plan and detailed test procedures, as well as preliminary plan for qualification testing.

Third Design Review. Review, analyze, and critique results of engineering model test phase, review and approve subsystem type approval test plan and procedures for manufacturer in-process tests and flight acceptance test; review and approve preliminary test plan for spacecraft assembly, checkout, test approval, flight approval, life testing; establish schedule for submission of detailed test schedule including dates for submission of detailed test procedures for approval.

g) Prepare monthly test program status reports and update the integrated test plan; coordinate test change requirements requirements with the Test Board, with JPL, and with program management.

h) Monitor each engineering change order for inclusion in test plans and procedures; advise the Configuration Management Board of the impact of the engineering change order on test validity.

i) Maintain a current log of all tests, test results, and failure reports; prepare a comprehensive analysis of test results for each functional element of the spacecraft system and subsystem to estabish the level of confidence in the adequacy of the system design to satisfy the Voyager mission requirements.

3. INTEGRATED TEST PLAN

3.1 Scope

The integrated test plan will provide for the sequential testing of spacecraft parts, subassemblies, assemblies, subsystems, and the completed spacecraft. The exceptions to this sequence involve parts selections which occur during the development and magnetic testing and the parts procurement activity that continues throughout the life of the project. Because of the possible schedule effect, these tests must be identified early in the program.

After the selection and testing of parts, and when parts magnetic testing has been completed, the development test cycle will be initiated. This test cycle constitutes the initial phase of subsystem testing; it will include breadboard testing, testing of the engineering model, type approval model tests, and life tests.

The next phase of subsystem testing includes the in-process manufacturing tests and subsystem flight acceptance testing.

The system test cycle begins with tests of the spacecraft engineering model and continues through the proof test model (PTM), the life tests, and the flight spacecraft acceptance testing. A similar test cycle (see Volume 6) will occur during the development of the OSE system.

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, and 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.

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
- Demonstration of size, weight, and 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 systems design and supports the final release of drawings to enable manufacturing and subcontractors to proceed with a high confidence of producing reliable end items.

The next phase of subsystem testing encompasses the type approval and life testing of items produced in accordance with final released drawings and specifications. Successful completion of this test phase obtains a high level of confidence for the subsystem design. Since failures occurring at any point during type approval and life testing may create a design or schedule slippage problem, any such failures will be reported to the Test Office as part of the normal TRW failure reporting system. Follow-up action is instituted if the cause of failure will affect the design, schedule, or reliability.

The next phase of subsystem testing occurs when the flight hardware is manufactured. This test sequence includes in-process testing and environmental acceptance testing. These tests make it possible to evaluate workmanship and reliability during the fabrication of subsystem units.

At the system level, the first tests occur during the assembly and checkout of the engineering model spacecraft. This in-process testing sequence demonstrates:

- Intersubsystem compatibility
- OSE-spacecraft compatibility
- Final procedures and computer programs for PTM assembly and test
- Spacecraft environmental test facilities
- Spacecraft magnetic properties

- Spacecraft-DSN compatibility
- Spacecraft electromagnetic compatibility
- Crew training

Successful completion of the in-process test phase establishes high confidence in the functional operation of the spacecraft system and its associated OSE.

The subsequent spacecraft testing activity includes type approval, flight acceptance, and life testing. The successful completion of the type approval test demonstrates high confidence in the design and fabrication of the spacecraft system; flight acceptance testing identifies correctness of workmanship and qualification of the spacecraft for flight; and life testing provides data relating to the expected life of the spacecraft system.

3.2 Voyager Project Test Matrix

The test matrix shown in Table 4-1 describes the sequence of test events and the elements of equipment involved. The column at the left of the matrix lists the elements to be tested in a program sequence from start to completion. Each facet of the testing program is discussed in a time-sequenced order and is keyed to the paragraph numbers shown in the corresponding columns across the top of the page. The test program consists of the six major phases listed below:

- Parts Selection (Section 3.3). Testing associated with the selection of parts
- Magnetic Testing (Section 3.4). All testing required to meet magnetic requirements

Table 4-1. Voyager Project Test Matrix

Equipment Tested	3.3 Parts Selection	3.4 Magnetic	3.5 Development: Design Margin	3.5 Development: Environmental	3.5 Development: Internal Subsystem	3.5 Development: Inter-Subsystem	3.6 Manufacturing: Part Reliability	3.6 Manufacturing: In Process Test	3.6 Manufacturing: Flight Acceptance Test	3.7 Type Approval Testing: Proof Test	3.7 Type Approval Testing: Life Test	3.7 Type Approval Testing: Design Margin & Failure Modes	3.7 Type Approval Testing: Interface Test	3.8 Subsystem Assembly and Test: Assembly & Check-out	3.8 Subsystem Assembly and Test: Flight Acceptance Test	3.8 Subsystem Assembly and Test: Launch Operation Test
Parts	x	x					x									
Breadboard	x	x	x	x	x											
Engineering Model	x	x	x	x	x	x										
Subassembly								x								
Assemblies		x						x	x	x	x	x		x		
Partial Subsystem													See Table 4-2	x		
Subsystems		x	x	x					x	x	x	x		x		
Systems:																
Engineering Model		x			x											
Proof Test Model		x								x	x (1969 only)	x		x		
Flight Spacecraft		x												x	x	x
Life Test Spacecraft		x									x			x	x	
OSE (details Volume 6)					x									x		

- Development Testing (Section 3.5). All testing on breadboard and engineering models associated with design margin, environmental, and intrasubsystem testing to evaluate the feasibility of a particular design concept

- Manufacturing Testing (Section 3.6). All testing associated with parts reliability, in-process testing, and assembly and subsystem flight acceptance

- Type Approval and Interface Testing (Section 3.7). All proof testing, life testing, and design margin testing on flight type hardware; all possible interfaces, intrasubsystem, intersubsystem, and spacecraft external interface testing

- Assembly Testing (Section 3.8). All integration testing flight acceptance testing, and launch operations testing associated with the total spacecraft.

The interface type approval test program shown in column 3.7 of the matrix (Table 4-1) is described in Table 4-2. During succeeding discussions (Paragraphs 3.3 through 3.8) the terms "Parts," "Subassemblies," "Assembly," "Subsystem," and "System" are defined as follows:

- Parts. The next level of complexity below subassembly which can be tested and has parameters that can be evaluated.

- Subassembly. The next level of complexity below an assembly, or a significant portion of an assembly, which when integrated with other subassemblies or parts, forms an assembly.

- Assembly. The next level of complexity below a subsystem, which when integrated with other selected subassemblies, forms a subsystem.

- Subsystem. A major, substantially independent functional grouping of equipment, which when assembled and combined with all other subsystems, forms a system.

- System. One of the principal functioning entities comprising the Voyager space system. A system is the major subdivision of a space system; e.g., launch vehicle system, spacecraft system.

Table 4-2. Interface Type Approval Testing

Test	Purpose	Equipment Used	When
1. Subsystem-panel compatibility in the electrical sense, not necessarily mechanical	Verify that the individual subsystem black boxes can operate together as a system.	Prototype subsystem-panel in EM spacecraft	Prior to PTM assembly
2. Subsystem compatibility tests, electrical and mechanical	Ascertain that all of the subsystem interfaces perform properly, including noise and transients tests, signal compatibility, and RF compatibility.	Flight approval subsystems in PTM spacecraft	Prior to flight spacecraft assembly
3. Subsystem-OSE compatibility, panel-OSE compatibility	Verify that the OSE and spacecraft are compatible. The compatibility tests include systems test set EOSE, the panel test EOSE, and the mechanical OSE.	EM and PTM systems from (1 and 2) above and panel assembly from (1 and 2) above	Prior to delivery to systems test area
4. Intersubsystem (system), STC compatibility	Verify that the individual subsystems are not being interferred with by another subsystem and that a given subsystem interferes with no other subsystem.	EM spacecraft and PTM	Completed on EM and verified on PTM
5. Spacecraft-LCE Compatibility	Verify that the hangar assembly area, block house, and gantry facilities are ready to support the launch of two flight spacecraft.		
a. Redondo Beach		ECM and PTM spacecraft at Redondo Beach	Completed prior to start of flight spacecraft assembly and checkout
b. AFETR		EM and/or PTM spacecraft at ETR	Completed prior to flight spacecraft flight approval
6. Spacecraft-capsule compatibility	Ascertain that the spacecraft does not interfere with or degrade the capsule operation, that the capsule does not interfere with or degrade the spacecraft operation.	PTM spacecraft and PTM (type) capsule	Prior to assembly and checkout of flight spacecraft
7. Spacecraft-DSN-MDE compatibility	Verify that the TRW-supplied equipment is compatible with the DSIF and SFOF facilities.	Spacecraft simulator and MDE and PTM-MDE	Completed prior to start of PTM test. During PTM test.
Spacecraft communications-MDE-DSIF compatibility	Verify that the spacecraft telemetry data is compatible with the DSIF and SFOF equipment, and that the DSIF equipment is compatible with the spacecraft airborne receivers	Spacecraft simulator with proto subsystems MDE at DSIF	
8. Spacecraft MOS	Establish a RF or hardline link between TRW and JPL to verify that the DSIF and SFOF equipment and computer programs work properly. It is planned that the EM spacecraft will be transported to the Goldstone DSIF station for compatibility testing.	a) MDE installed at DSIF b) Software completed c) Spacecraft simulator d) and/or EM RF link to Goldstone e) EM spacecraft at Goldstone	prior to completion of PTM test
9. Spacecraft, launch vehicle system a. Interface adapter, Centaur	Verify that the spacecraft can be mated properly to the Centaur launch vehicle and that adequate clearance exists between the spacecraft and nose fairing. In addition, all electrical umbilical functions will be checked through the Centaur to the spacecraft, and the RF nose fairing coupler losses will be determined.	PTM, Centaur adapter and nose fairing at Sycamore Canyon, if schedules prohibit using structural model	After completion of PTM test (schedule dependent)
b. Launch comples		First 1. Spacecraft simulator 2. LCE at ETR 3. LV simulator and comples	As early as possible, using L V simulator and LCE at launch complex
		Second 1. EM/PTM spacecraft 2. LCE at ETR 3. LV vehicle complex 4. PTM capsule	Immediately after erection of LV test vehicle
10. AHSE spacecraft compatibility	Verify that the various handling fixtures are compatible with both the spacecraft and capsule	1. Structural model 2. AHSE	Prior to use on PTM
11. Test facilities-spacecraft compatibility	Ensure that each test facility is compatible with the spacecraft.	1. EM for electrical checks and structural model for mechanical check	At least 2 months prior to use by PTM
o Chamber o Shaker o Acoustic o Magnetic		2. Verify with PTM	At least 1 month prior to use by flight spacecraft
12. AFETR-LV-spacecraft systems compatibility	Ensure that each spacecraft facility is compatible with the spacecraft.	PTM	" "
13. Spacecraft, spacecraft science payload compatibility	Verify that the various scientific experiments do not interfere with the spacecraft operation and that the spacecraft operations do not interfere with any experiment operation	PTM spacecraft	Prior to PTM test

3.3 Parts Selection

The primary aim of the parts selection task is:

- Selection of part types which have previously been qualified to space application environments similar to the Voyager program
- Establishment of suitable controls to assure that part applications are well within the capabilities of individual part types
- Identify as critical items those parts which are new or life-limited and to establish controls and design procedures to control the application of these items
- Selections and/or development and enforcement of part specifications which will contain complete part descriptions, performance requirements, associated test procedures, qualification, inspection, and preconditioning requirements
- Selection of new parts

3.4 Magnetic Testing

3.4.1 Parts

Parts and materials to be used on the Voyager spacecraft will be tested for magnetic cleanliness at incoming inspection in four phases:

- Preliminary tests and studies to determine what parts are inherently nonmagnetic; these are known as Class I parts
- Determination of the magnetic field criteria for all other single parts using a preliminary parts list. All parts which are expected to exhibit some small amount of residual magnetism are known as Class II parts
- Tests of all parts considered for the, spacecraft to generate a magnetically clean approved parts list
- Incoming inspection test, 100 per cent at the part or module level.

The criterion for the nonmagnetic Class I parts is less than 1 gamma at 3 inches after exposure to a magnetizing field of 100 gauss. The criterion for Class II parts (expected to have some permanent magnetic field) is that the maximum magnetic field measured at 3 inches from the center of the part should not exceed 5 γ after exposure to a 100-gauss field when the parts leads have been trimmed to 1/8 inch. These parts criteria are used in the generation of a magnetically approved parts list. However, there are some "problem parts" whose field cannot be brought down to below 5 γ at 3 inches after magnetization. These problem parts are sometimes approved for limited use provided there is no nonmagnetic replacement for the part and redesign is impractical, the field of the part is not extremely high, and only a small number are used on each spacecraft. These problem parts will be kept to a minimum in the spacecraft.

Tests will be performed to qualify parts and materials as magnetically clean according to the magnetically clean parts criteria. This information will be incorporated into the JPL-approved Voyager approved parts list. This list designates those parts which must be used wherever possible in the design of the spacecraft. When a subprogram manager feels that a part not on the approved parts list must be used, it is required that a parts deviation form be filled out and the part sent in for magnetic test. If the part does not satisfy the magnetically clean parts criteria it will be considered a special problem part. An entensive search for the nonmagnetic equivalent part is then initiated.

3.4.2 Magnetic Testing of Subassemblies and Assemblies

During the development phase, breadboard circuits, especially those containing high current levels, will be tested. The purpose of breadboard testing is to determine whether circuit currents will create a magnetic field problem. When the assembly layout and packaging is designed, all possible means are taken to minimize the field. In general, all engineering models suspected of having troublesome magnetic characteristics are tested, both operating and nonoperating, to verify that the dynamic magnetic field of the assembly caused by current flow has been minimized.

Particular attention will be given to solar cells to check for magnetic effects of manufacturing procedures. Complete magnetic field measurements will also be obtained for the mounting arrangement of the traveling wave tubes. During verification tests at Table Mountain, the solar array for the PTM spacecraft will be exposed to natural sunlight, and effects of current loops in the array will be measured. For this reason, it is not necessary to determine the permanent magnetic field of the solar array during spacecraft tests.

Each type approval and flight assembly model will be placed in a magnetic test fixture and its magnetic properties determined using the coilless method of testing. This method consists of measuring the magnetic field with flux-gate probes compensated to remove effects of the earth's field. In the first type approval magnetic test (pre-environment) the magnetic field of the assembly is measured as received; no attempt is made to magnetize or demagnetize the assembly. In the second type approval magnetic test (post-environment), and for flight units, the magnetic field of the assembly is measured in three conditions: as received, after magnetization in a 100-gauss field, and after demagnetization. Measurements will be made with the assembly both operating and non-operating.

3.4.3 Magnetic Testing of Spacecraft

Magnetic tests of the spacecraft are the same as for subassemblies and assemblies except that the assembled spacecraft will be tested. The level of the magnetizing field is 25 gauss.

3.5 Development Testing

Breadboard testing provides the designer with a means for assessing performance with minimal effort and delay, but the usual open breadboard format, while facilitating circuit layout and revisions, has insufficient resemblance to the flight configuration to yield generally applicable data.

Engineering models, close to flight configuration, extend the valid area of development testing beyond the limitations of the breadboard. Although they are available later than the breadboard units, engineering models permit design testing at a relatively early stage in the program and make it possible to verify compatibility with the operational support equipment, to train test personnel, and to check test procedures which will be employed for flight units.

Using the breadboard and engineering models, four types of testing are categorized as development tests:

- Design margin testing determines the validity of design margins. In each case significant stress parameters are applied in increasing steps starting at flight levels and going up to design maximums.

- Environmental tests are performed to the extent possible to obtain early information on environmental effects on designs. The breadboard testing may be limited to high and low temperature testing due to the limited validity of other environmental exposures. Engineering models, however, are exposed to all possible environments.

- Internal subsystem testing is started at the breadboard level and continues at the engineering model level to obtain early elimination of intrasubsystem problems.

- Intersubsystem testing is started with the engineering models and continues in the spacecraft engineering model assembly to obtain early elimination of intersubsystem problems.

3.6 Manufacturing Testing

The three types of testing categorized as manufacturing tests consist of:

- Part reliability testing

- Manufacturing in-process testing

- Manufacturing flight acceptance testing.

Parts electrical and environmental testing will be performed on all part types used on the Voyager spacecraft for the purpose of predicting reliability with a high confidence factor. A typical part reliability testing sequence is shown in Figure 4-2. Parts testing as distinguished from parts screening does not necessarily increase reliability of the parts but increases the confidence factor associated with the reliability prediction of a specific lot.

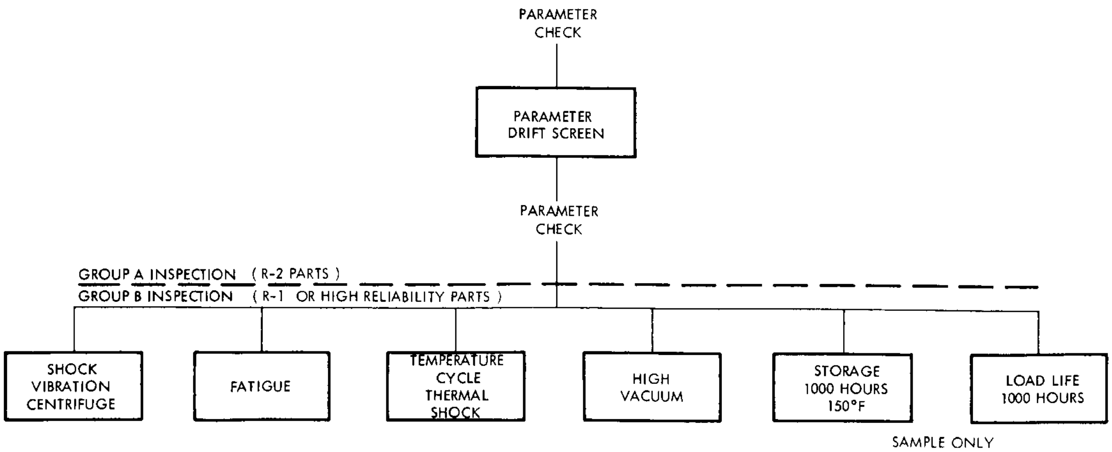


Figure 4-2. Typical High Reliability Parts Testing Sequence

Checkout tests will be conducted on electrical subassemblies and assemblies during their fabrication to assure their electrical integrity prior to type approval or acceptance testing. Thus, potential delays resulting from deficiencies are greatly reduced.

Acceptance tests for assemblies and subsystems consist of subjecting assemblies and subsystems to the kinds of environmental exposure levels anticipated during launch and orbit. The test levels and exposure will be defined in TRW specifications. A typical assembly acceptance test sequence is shown in Figure 4-3. The purpose of these tests is to assure the performance requirements have been met, that the equipment is free from defective workmanship, and that it will survive the flight environments. The environmental exposures during acceptance test differ from qualification test in that only two stresses are considered, vibration and thermal-vacuum. These stresses are described below:

- Vibration. Flight assemblies will be subjected to vibration tests. Only sinusoidal vibrations will be applied.

- Thermal-Vacuum. Thermal-vacuum tests will be performed with the assembly mounted in a manner thermally simulating the attachment of the assembly to the spacecraft structure. Tests will be conducted at maximum and minimum predicted assembly temperatures. The assembly

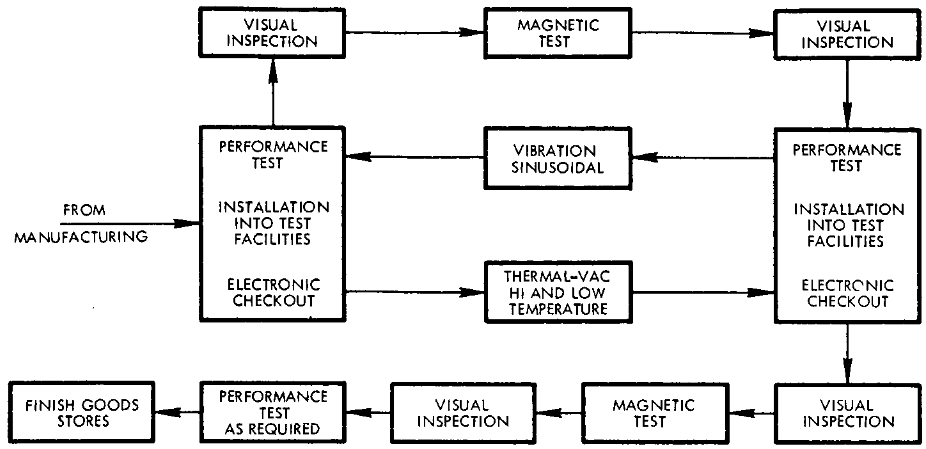


Figure 4-3. Typical Voyager Flow Chart, Assembly Flight Approval

will be sufficiently instrumented to insure measurement of realistic assembly maximum and minimum temperatures. During evacuation the assembly will be operated in the condition typical of the launch phase, and corona effects will be monitored throughout evacuation. Tests will be conducted under stabilized temperature and pressure conditions with the assembly operating. For cyclically-operated assemblies ("on-off" orbital operation), cold start capability will be demonstrated during the exposure. Performance of the assembly will be verified during and after the exposure.

3.7 Type Approval Testing

3.7.1 Proof Testing

Type approval tests are performed on type approval assemblies and the proof test model spacecraft for purposes of qualifying the design. (Figure 4-4.) The tests will be conducted in accordance with TRW-prepared and JPL-approved environmental specifications. Assemblies and spacecraft to be tested will be subjected to the following environmental exposures:

- Vibration
- Shock
- Humidity
- Linear Acceleration
- Magnetic Properties
- Temperature
- Acoustics
- Space Simulation (thermal-vacuum)

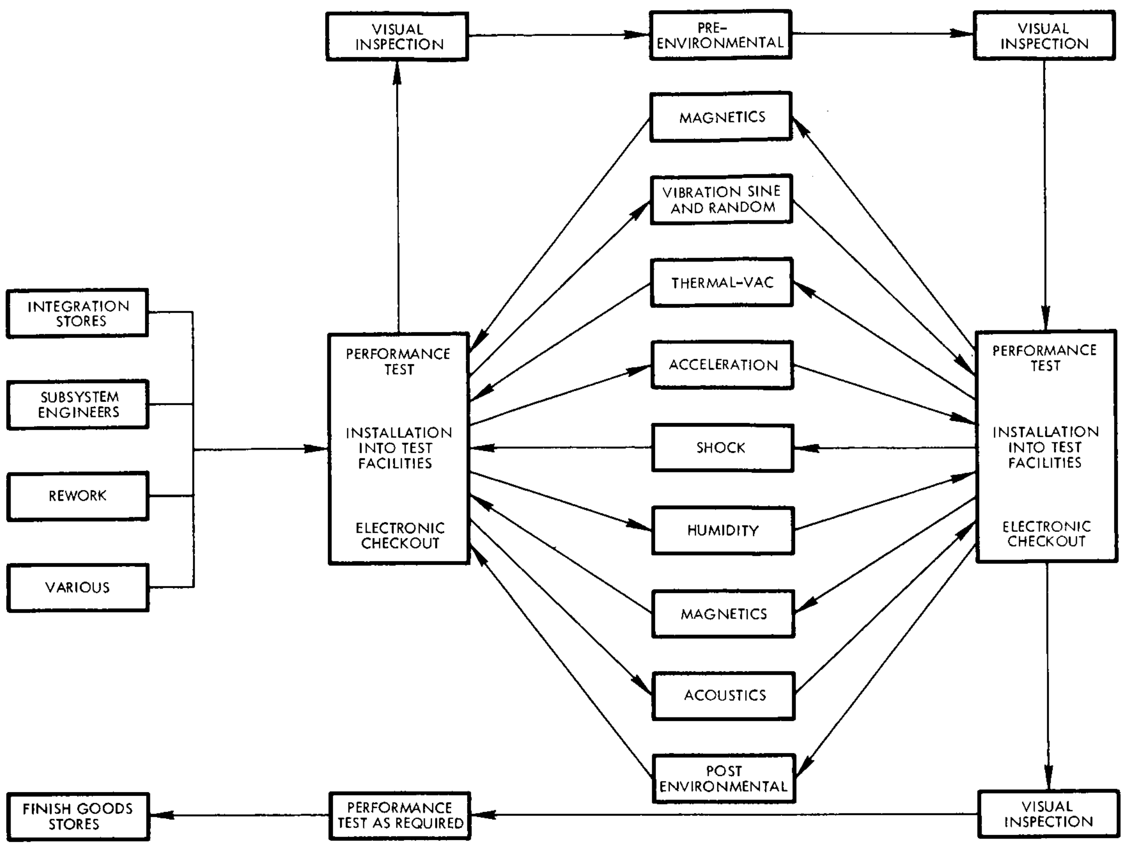


Figure 4-4. Type Approval Testing

These tests are a series, more stringent from an environmental viewpoint than are conditions anticipated for transportation, handling, storage, launch, and flight. The type approval articles are considered to be flight type hardware. Spacecraft type approval testing is described in subsection 6 of Section V.

Particular emphasis will be placed upon searching for design deficiencies and keeping accurate test records, failure and rejection reports, and engineering data. Production engineering and fabrication personnel will be kept completely informed of design deficiencies as they are revealed and their recommendations will be solicited so that the effects on the fabrication and acceptance test cycle can be minimized.

3.7.2 Life Testing

As a supplement to margin testing, life testing is important for its contribution to the demonstration of reliability. Life testing accomplishes this in two ways. First, the repetitive performance of certain equipment functions verifies the absence of systematic failures caused by fatigue or wearout (if the number of cycles is large enough). Second, the accumulation of operation time at mission levels contributes to the validation of functional performance over the specified test time.

It is not intended that every assembly be subjected to life test since such a procedure, although technically useful, is excessively costly and time consuming. The complete Voyager test program includes life testing at several levels, all contributing to the demonstration of assembly reliability. The only assemblies that will be considered for component level life testing will be those for which existing reliability and life data, from prior flight experience or from parts level tests, are incomplete. Life testing at the assembly level has obvious advantages over tests conducted on a complete system because early testing of assemblies makes it possible to proceed more rapidly with subsystem development. This advantage is also valid (to a lesser extent), when compared to subsystem level testing. The absence of interactions is the prime disadvantage encountered in component-level life tests; for this reason, each subsystem will be analyzed in terms of its in-line effect on reliability and the requirements for subsystem life testing will be based on the criteria thus obtained.

The project schedule does not provide for real-time mission life testing of components before the first flight. The importance of life tests is such, however, that the required testing should begin as soon as possible and continue after the launch. The results of such tests will be useful in several ways. First, if a systematic wearout or life-limiting mode is evidenced in time, a redesign may be instituted. If a failure mode is revealed during the life test conducted after the launch of a flight spacecraft, it may provide important data that flight operations personnel can use in the preparation of programs designed to avoid or counteract that failure mode. Conversely, if a failure occurs in the flight vehicle, the life test spacecraft will be a useful model to test the effect of corrective commands. Finally, the results of the real-time life test will be available for later flights.

a. Subsystem Testing

One approach to life-testing electrical subsystem under thermal-vacuum environment is based on the assumption that reliability can be demonstrated by testing sufficient numbers of subsystems for a prescribed amount of time (and duty cycle) so that the product of the number of subsystems and the test duration time (and operating cycles) is equal to, or greater than the product of the predicted mean time between failure (MTBF) and an assigned factor, K. This factor is assigned as a confidence indication and will vary between 2 and 8 depending on the functional characteristics of the subsystem and its in-line effect on the over-all system reliability. Thus, if a given subsystem has a MTBF of 3000 hours, a K factor of 2, using a test duration of 4000 hours, the minimum number of subsystems to be tested would be 2. This approach will be used for subsystem life testing, however, consideration of cost and schedule, will probably require deviations.

For mechanical environments, the test approach will be planned on the basis of the time a component or subsystem is exposed to the test environments rather than on MTBF. This method is proposed because (in the mechanical aspect) the MTBF would be very much in excess of the exposure time. Applying this reasoning, viz., the product of the number of subsystem elements and the test time must exceed the product of exposure time and the K factor, it follows that one sample more than satisfies this criteria, however, TRW proposes to subject no less than two subsystems to mechanical life tests. The sample size, in this case, would be expanded by the inclusion of the type approval and flight acceptance vibration tests.

To demonstrate reliability, selected quantities of single-occurrence functional elements (such as pyrotechnics) will be obtained from a common lot. Such elements will be identified and a statistical test rationale will be derived during the Phase IB and II development cycles.

Mechanical subsystems, such as structural, thermal louvers, and deployable booms, will need to be subjected to design margin testing, under adverse conditions, to establish possible failure modes; functional acceptance tests will be required to ensure performance.

b. System Life Testing

The economic and time restraints of testing limit the number of spacecraft life tests models to one or two. Here again, confidence is bolstered by the test of the proof test model, the engineering model, and the flight spacecraft. Thus, TRW proposes that only one life test spacecraft be tested for the 1971 mission and that the proof test model be used as a life test model for the 1969 test flight. In this test configuration, the test environment would be limited to thermal-vacuum conditions. The following test approaches might be employed:

- Install life test spacecraft in the thermal-vacuum chamber at vacuum; operate spacecraft at a nominal temperature as established by solar simulation tests; cycle through mission sequence periodically such that a sufficient number of cycles are obtained to satisfy a reliability assessment; establish test duration on the basis of the MTBF of various subsystem elements as weighed by their in-line effect on mission reliability. Once this point is reached, the test environment could be made more severe (e.g., higher temperature) and the test repeated.

- Test configuration same as above, test duration set by real-time.

- Spacecraft the same as above, but set an arbitrary test cycle of 40 days at nominal temperature, 40 days at elevated temperatures, and 40 days at depressed temperatures. The choice and rationale of the system level life testing will require further study and definition during Phase IB.

3.7.3 Design Margin Testing

Design margin testing makes use of the T/A units and PTM spacecraft that have been proof tested. For each element, significant stresses will be applied in increasing steps beginning at type levels and continuing to the design maximums.

3.7.4 Interface Testing

All interface tests will occur at the earliest opportunity; such tests will be initiated at the lowest practical assembly level and continued through the highest assembly levels.

3.8 Assembly Testing

The subsystems will be subjected to checkout tests as they are assembled to form the spacecraft. This procedure will insure that the functional integrity of subsystems and the system is maintained prior to spacecraft type approval or acceptance testing. An example of assembly testing is shown in subsection 6 of Section V.

Spacecraft flight approval tests are designed to ensure that the flight and life test spacecraft have been properly fabricated and assembled, that performance meets specifications, and that the integrated spacecraft is ready for launch. Acceptance testing combines electrical and mechanical functional tests performed during or after the subject items have been exposed to space simulation and vibration environments at stress levels commensurate with the projected launch and orbital environments. The proposed spacecraft acceptance test cycle is contained in Appendix A.

Spacecraft space simulation testing will be performed under vacuum conditions with realistic solar simulation. This will require a vacuum chamber with a high quality collimated solar beam approximately 23 feet in diameter. The detailed design requirements for this facility will be provided in the proposal for Phase IB.

Subsystems and systems of the spacecraft will be subjected to checkouts tests during the launch operations to assure the integrity of the subsystems and systems prior to launch. An example of launch operations testing is contained in subsection 6 of Section V.

4. EVENT TEST MATRIX

To evaluate the test program in terms of the Voyager mission, a matrix of mission events versus testing levels will be maintained. Each cell of the event test matrix will contain both the environmental parameters (such as vibration and temperature and the elements common to each event (i.e., verification of command received, function initiated). This event test matrix will serve two major functions: first, when a new test is planned, the test parameters and a list of the elements to be tested will be incorporated in the matrix and the matrix will then show to what extent the planned test duplicates other tests; second, periodic examination of the event test matrix will indicate where insufficient testing efforts are likely to occur. Since hardware items are not shown on the event test matrix (in contrast to the over-all test matrix shown in Table 4-1), the event test matrix will present a mission oriented picture of the testing program. The use of this matrix as a test planning tool will make it possible to maintain a more uniform test density. An example of the event test matrix illustrating the details contained in a single cell is shown in Figure 4-5.

5. EFFECTS OF TESTING 1969 FLIGHT TEST SPACECRAFT ON THE 1971 MISSION

The 1969 subsystem and system designs are essentially identical to those of the 1971 design with the exception of those factors attributable to the differences in spacecraft arrangement and weight such as distribution of structural loading and thermal and electromagnetic interactions. Thus, the 1969 test program can provide early performance, design verification, reliability, and environmental test data of direct use in the design and test effort for the 1971 flight spacecraft.

The schedule for significant tests of the 1969 spacecraft is shown in Figure 4-6 as solid bars; the cross-hatched areas are the similar efforts for the 1971 program. Table 4-3 summarizes the benefits that

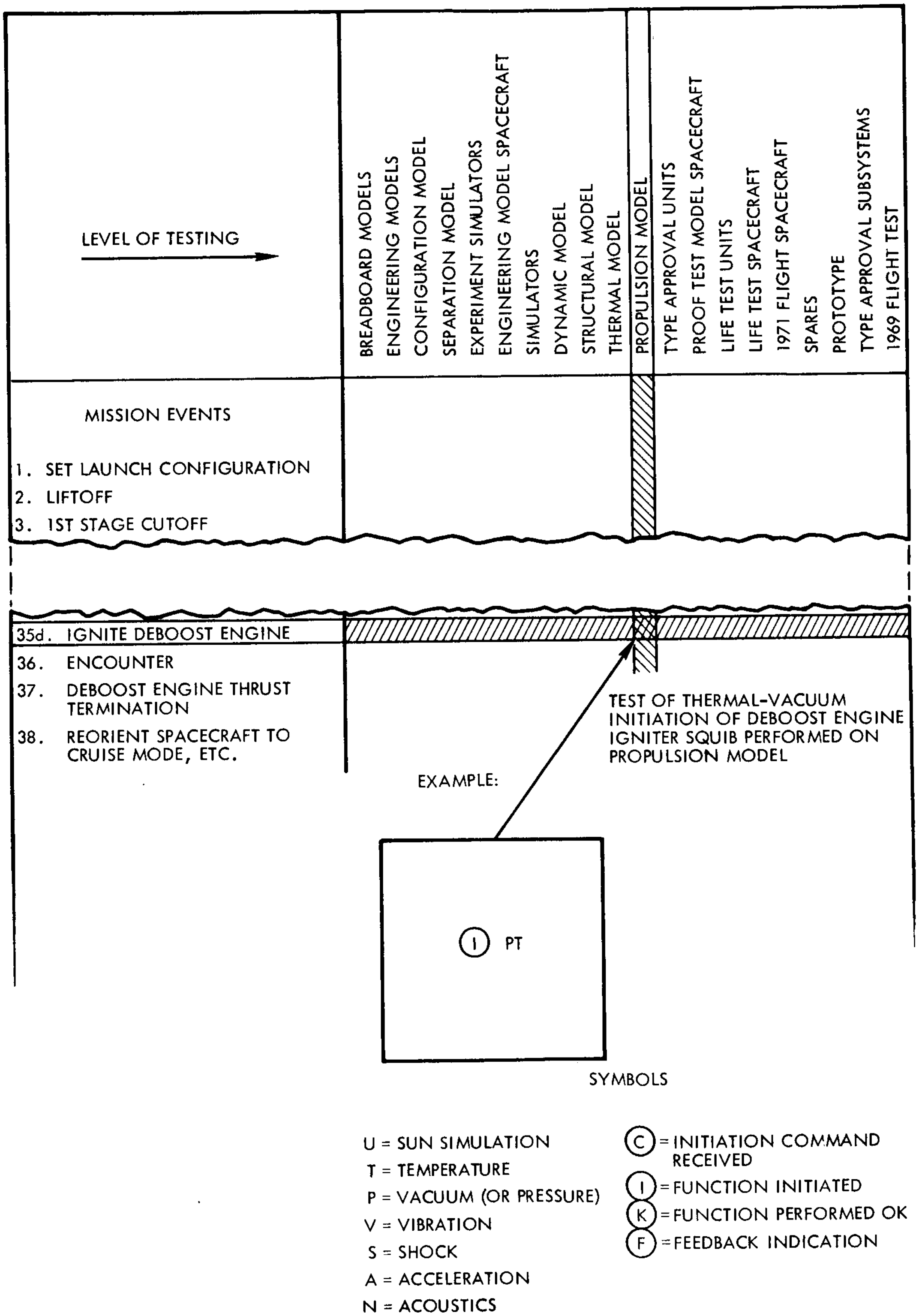


Figure 4-5. Event Test Matrix

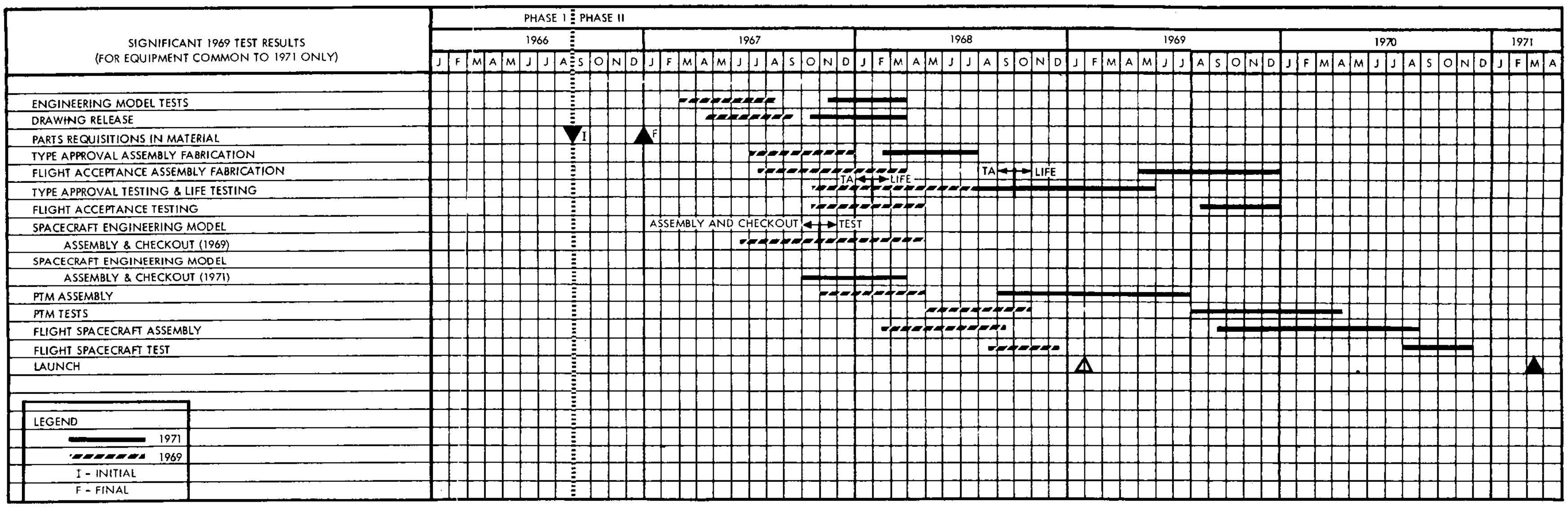


Figure 4-6. Significant 1969 Test Results Schedule

Table 4-3. Effects of 1969 Test Program on 1971 Mission Design and Test

1969 Test Phase	Benefits	As Scheduled	Minimum Concurrency* — Allow for a slip in TA unit test to start of 1971 flight assembly fabrication*	Maximum Concurrency** — Allow for a further slip of proof test model testing
		1971 Schedule Margin (months) — Time from Completion of 1969 Event to Final Release of 1971 Drawings		
Parts tests	High reliability parts Inclusion of unique parts after adequate tests	13	20	24
Subsystem engineering model tests	Released drawings and specifications Demonstration of Size Weight Thermal properties Power Performance Internal compatibility Magnetic properties OSE compatibility	7	14	18
Subsystem type approval tests	Complete subsystem design verification Confidence in design capability in environmental extremes Verification of manufacturing process Verification of magnetic properties	2	10	14
Subsystem life tests	High confidence in life capability	-4	7	11
Spacecraft engineering model tests	Demonstrate compatibility with OSE Software Launch vehicle Facilities Subsystem interactions EMC	-1	8	12
Proof test model tests	Crew training Higher confidence in EM test results Launch survival	-7	4	8
Spacecraft flight acceptance tests	Higher confidence	-9	2	6
Launch		-10	1	5

* 1971 type approval model completed at start of fabrication of flight units.

** First four months of 1971 proof test model testing completed 3 months before flight spacecraft assembly and checkout.

will be derived from the 1969 test program and describes the opportunity to apply the results of such benefits to the 1971 effort in terms of the schedule margins available relative to the final design release for the 1971 type approval hardware. The margin times are given in calendar months for three conditions: the first assumes that the 1971 schedule will not be slipped and that the drawing release date will remain firm at 1 October 1968; the second condition (minimum concurrency) provides for the completion of type approval testing just before fabrication of the 1971 flight units is initiated; the third (maximum concurrency) provides for delaying the 1971 spacecraft proof tests for four months beyond the nominal schedule. If this third schedule approach becomes necessary, vibration and space simulation tests on the proof test model would be completed three months before the assembly and checkout of the 1971 flight spacecraft is concluded.

In combination, Table 4-3, and the schedule of significant 1969 test results shown in Figure 4-6 illustrate that, in the time frame available in the 1971 mission schedule, it is readily feasible to complete rework or redesign to compensate for a problem which is discovered while the 1969 breadboard models are being tested. If, for example, the tasks involved in repairing a subsystem occupy three months of the available 13-month 1971 schedule period, 10 months will still remain as a safety margin. On the other hand, if a failure is not discovered until prior to launch at the end of the 1969 program, no schedule margin will be available after the appropriate corrections have been made. Thus, if the requirement for a schedule margin is imposed (in the 1971 time frame), the schedule will have to be shifted to the "maximum concurrency" previously defined.

Parts testing will include the parts qualification and selection program. This program will verify that the parts selected and the capability of the participating vendors will satisfactorily provide the kind of high-reliability parts required for the 1971 mission subsystems..

As shown in Table 4-3, design problems discovered by the end of the 1969 subsystem engineering model test phase can be readily accommodated without disturbing the projected schedule for 1971. Since any failure that might occur during these model tests can be compensated for within the projected time frame, the opportunity is available to gain additional insight to the possible failure modes. Thus, at this juncture, the 1971 design effort will have been reinforced by an additional confidence level concerning the size, weight, power, thermal characteristics, reliability, and performance of the subsystem elements.

At the successful completion of the 1969 subsystem type approval test phase, an adequate subsystem design will have been established. It is at this point in the 1969 test program that a test failure requiring a major redesign effort would slip the original 1971 schedule (see Table 4-3). However, the 1971 schedule margin time will still be adequate (i. e., 10 to 14 months) to provide a high confidence of successfully attaining the 1971 mission if either of the proposed concurrency schedules is adopted.

The subsystem life test phase for the 1971 mission will be completed during the 1969 subsystem life test program. Successful completion of the 1969 subsystem life test will provide for extremely high confidence in the subsystem design. Here, again, a major failure will cause a slip in the original 1971 schedule, but there will still exist a schedule margin of from four to six months after the redesign and retest effort.

The next test phase (see Table 4-3) consists of the assembly and checkout of the 1969 engineering model spacecraft. These tests will constitute the first system interaction verification; upon its successful completion, all of the problems associated with the design should have been resolved. Also at this time, interfaces such as those between the spacecraft and the OSE, the spacecraft and the test facilities, and the spacecraft and the software, will have been verified. In the event that similar failures occur during this time frame, there will still be a sufficient schedule margin for minimum concurrency and a more than adequate schedule margin for maximum concurrency. At the end of this phase of testing, crew training will also have been completed.

The successful completion of the proof test model test phase will provide sufficient confidence to proceed with the 1969 launch and will support a comparable level of confidence in the success of the 1971 launch. In the event of a failure during this phase, there is still adequate time to incorporate changes in the 1971 flight spacecraft and, on a high effort basis, changes in the 1969 flight test vehicle.

The final phase of the 1969 ground test program culminates at launch. The survival of the spacecraft through the powered flight, injection, guidance, acquisition, and first midcourse maneuver will support a high level of confidence in the probability that the 1971 mission will be successful. In the event of a failure at launch there still remains (conservatively) a two-to-four-month schedule margin in which to execute a redesign and retest program for the 1971 mission.

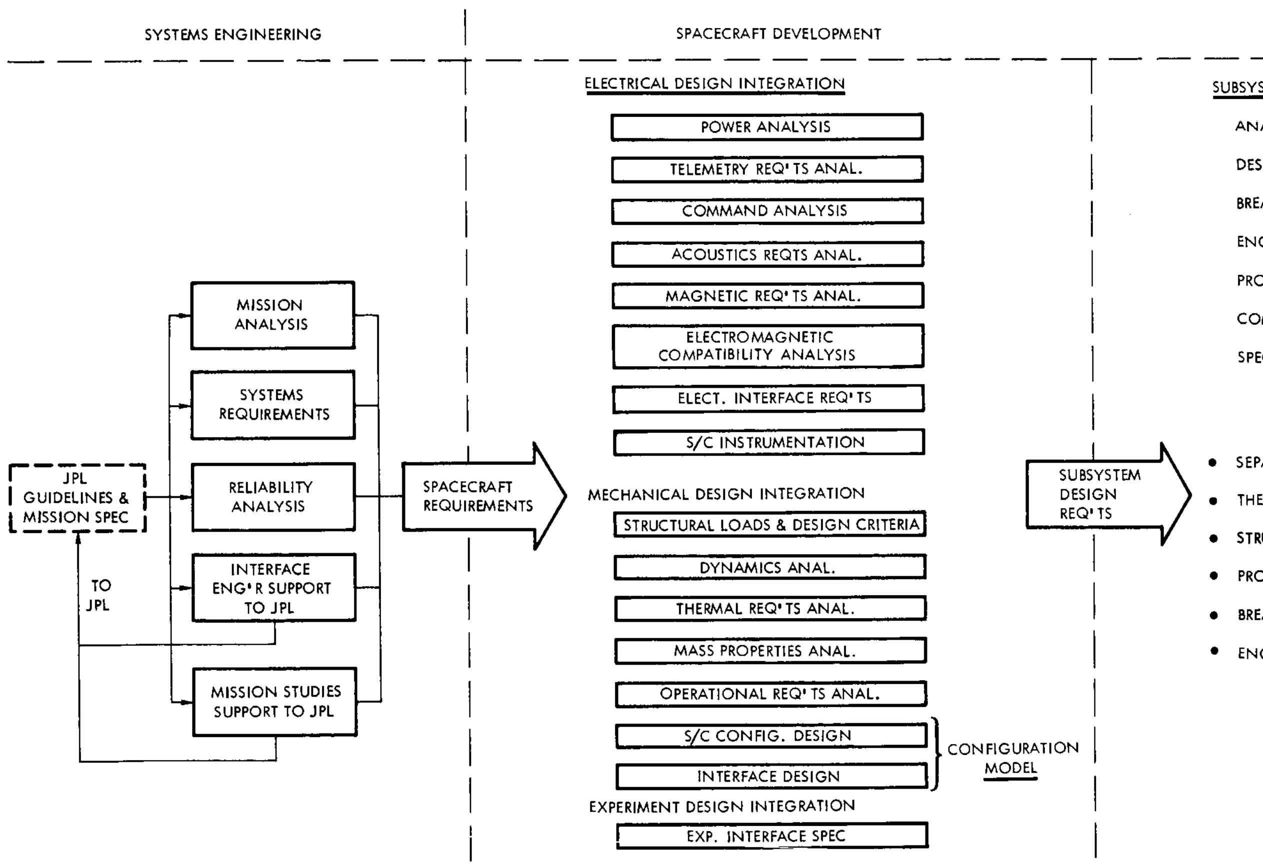
Since subsequent failures in the 1969 flight test vehicle provide decreasing time to include corrections, whether the 1969 flight results can be useful for the 1971 mission depends on the nature of the failure and the magnitude of the redesign effort, unless, of course, the failure is of such magnitude as to suggest that the 1971 launch should be postponed. Since the true maximum degree of concurrency is not shown in Table 4-3 (i. e., completing a fix, installing it in the proof test model and flight spacecrafts, and testing the proof test model in concurrence with the flight spacecraft acceptance testing), it is still possible to include changes in the flight spacecraft beyond the limits of the scheduled periods.

V. IMPLEMENTATION PLAN

1. INTRODUCTION

This section presents the preliminary implementation plan for the Voyager spacecraft. The plan includes design engineering as the major activity of Phase IB and Voyager development through mission operations as Phase II. The discussion generally treats both phases as one continuing effort, however, although the schedules and related discussion identify those efforts associated with each phase. Moreover, the test flight planned for 1969 launch is considered a part of the over-all development of the spacecraft and is therefore included in this volume. The over-all implementation of the Voyager spacecraft is diagrammed in Figure 5-1.

The discussion is organized by system engineering, spacecraft development, spacecraft assembly and checkout, spacecraft testing, launch operations and mission support operations. A final section discusses the further planning tasks needed during Phase IB to prepare for Phase II. Systems engineering (Subsection 2) discusses the mission and requirements analysis, systems documentation, and engineering reliability means of which the mission is converted into system design requirements on the spacecraft and interface requirements on the planetary vehicle. Spacecraft development (Subsection 3) in turn converts these system requirements into subsystem interface and design requirements. Subsection 4, subsystem development, a part of spacecraft development, treats the engineering analysis, design, and testing required to flight qualify the equipment. Subsystem type approval and life testing culminates the Voyager spacecraft development discussion. Subsection 6 presents the spacecraft assembly and checkout operations for the 1969 test flight and 1971 Voyager mission flight spacecraft, followed by the spacecraft testing. Prelaunch and launch operations for both the 1969 and 1971 flights are also discussed in Subsystem 6, followed by the mission support operations planning.



SYSTEMS ENGINEERING
SPACECRAFT DEVELOPMENT
JPL GUIDELINES & MISSION SPEC
TO JPL
MISSION ANALYSIS
SYSTEMS REQUIREMENTS
RELIABILITY ANALYSIS
INTERFACE ENG'R SUPPORT TO JPL
MISSION STUDIES SUPPORT TO JPL
SPACECRAFT REQUIREMENTS
ELECTRICAL DESIGN INTEGRATION
POWER ANALYSIS
TELEMETRY REQ'TS ANAL.
COMMAND ANALYSIS
ACOUSTICS REQTS ANAL.
MAGNETIC REQ'TS ANAL.
ELECTROMAGNETIC COMPATIBILITY ANALYSIS
ELECT. INTERFACE REQ'TS
S/C INSTRUMENTATION
MECHANICAL DESIGN INTEGRATION
STRUCTURAL LOADS & DESIGN CRITERIA
DYNAMICS ANAL.
THERMAL REQ'TS ANAL.
MASS PROPERTIES ANAL.
OPERATIONAL REQ'TS ANAL.
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INTERFACE DESIGN
CONFIGURATION MODEL
EXPERIMENT DESIGN INTEGRATION
EXP. INTERFACE SPEC
SUBSYSTEM DESIGN REQ'TS

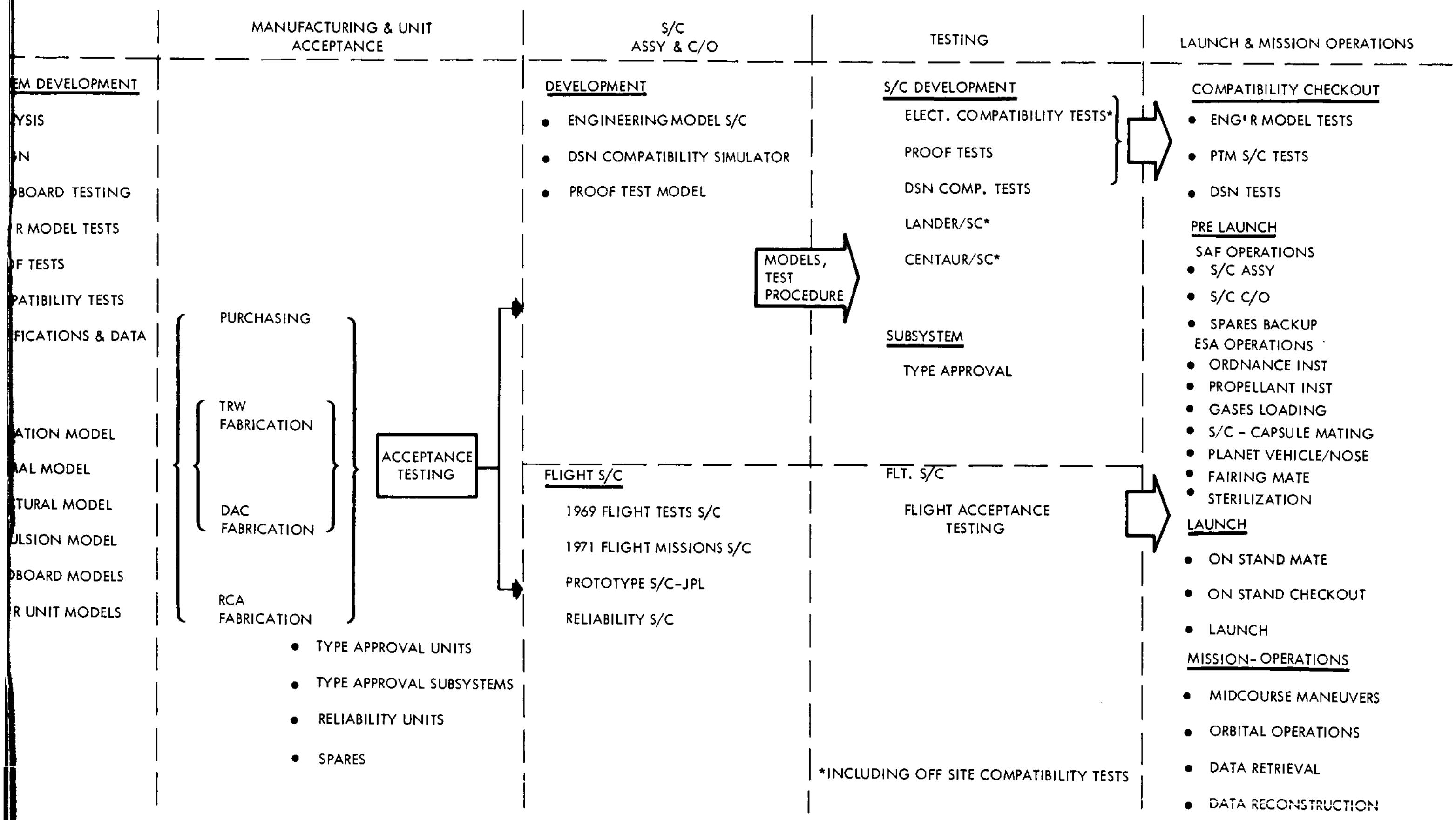


Figure 5-1. Voyager Program Implementation

2. SYSTEM ENGINEERING

Under the direction of JPL the primary task of system engineering for both the Voyager 1971 mission and the 1969 test flight is to ensure that the Voyager spacecraft system meets the requirements of the Voyager mission specification and that the reliability of the flight spacecraft is maximized within established constraints. To accomplish this objective, system engineering will formulate the approaches to be used in preliminary design and in later phases by the system and subsystem elements of the program to assure the evolution of a fully integrated system on all levels of engineering development.

System engineering effort will be devoted to detailed quantitative evaluation of the over-all system implementation and the results of the subsystem engineering phases. Among the responsibilities of system engineering will be the task of monitoring program activities in relation to meeting final program goals on the systems level.

Specific tasks to be performed by the system engineering team will include the following:

- Provide mission engineering support to JPL in the refinement of the Voyager 1971 mission definition and conduct mission studies to assist in definition of mission design
- Assist JPL in establishing a definition of the environmental, mechanical, and electrical interface 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 (DSN). Prepare and maintain communications link analyses that define the details of the Deep Space Instrumentation Facility (DSIF) spacecraft systems interface.
- Develop requirements on the functional interface between the spacecraft science subsystem and the remainder of the spacecraft system
- Ensure that the spacecraft system will satisfy the contamination constraint

- Develop reliability assessments and allocations and review the design from an over-all reliability point of view.

These individual tasks are facilitated by structuring the system engineering activities along the lines of system analysis, system requirements, and reliability, in such a way that they remain closely interrelated.

System analysis will be conducted to investigate, select, and optimize elements of the mission profile and to study in depth the general problems associated with spacecraft design and subsystem interfaces. Interaction problems and trade-offs among subsystem engineering activities will be interpreted and resolved using system analysis concepts and procedures.

The system requirements activity establishes a comprehensive hierarchy of requirements, criteria, and specifications from system through subsystem levels based upon compatibility with the Voyager mission specifications. These tasks include careful interpretation of priorities, resolution of conflicting subsystem design objectives, and continuous attention to changing system and subsystem performance capabilities throughout the pre-design, design and program development phases.

The reliability analysis activity formulates reliability models and policies, monitors adherence by program elements to established reliability goals, and ensures that all implementation activities remain in keeping with the highest system reliability consistent with the established constraints.

2.1 System Analysis

TRW will conduct mission studies, as requested by JPL, to assist in the definition of an optimum mission profile. Such studies will include the following subjects:

- Trajectories
- Guidance accuracy
- Communication performance
- Orbit determination accuracy
- Maneuvers
- Failure modes
- Targeting criteria
- Effects of constraints imposed by other systems on the design and operation of the flight spacecraft

The tasks performed under system analysis will encompass fundamental studies pertaining to the above technical disciplines to ensure penetration in depth of potentially critical design interfaces and to arrive at the technically most promising design approach.

A second class of problems is those that arise during the process of design evolution and need prompt attention by system analysis to assure a solution consistent with the over-all requirements of the system and its subsystems. During the course of the Voyager spacecraft development, TRW will analyze or review the spacecraft system design to investigate such factors as:

a) The adequacy of the data link to monitor planetary vehicle performance, to distinguish among failure modes, to provide information for ground control, and to provide the required science information.

b) The ability of the flight spacecraft to accommodate failures while accomplishing the total mission or partially successful mission, to provide the attitude accuracy required by the mission, to respond accurately to control from the ground, to meet the requirements established by the spacecraft science payload and the flight capsule, and to maintain an environment suitable to the successful operation of its own hardware subsystem, the spacecraft science payload, and the flight capsule.

c) The ability of the spacecraft bus, including propulsion, to meet the requirements of the Voyager mission specification.

2.2 System Requirements

The transformation of over-all system objectives and requirements into a set of hardware and associated software is controlled by a hierarchy of comprehensive statements covering both qualitative characteristics and quantitative design parameters for the system at all levels. Thus the requirements data become the medium for establishing well-defined design areas. Conversely, this data serves to represent the system design in such a way that it defines system performance and allows evaluation of the design for its adequacy in meeting the over-all goals.

The system requirements work area can be thought of in terms of the following tasks:

- Organizing and structuring the total requirements documentation package
- Generating the TRW spacecraft requirements documentation at the system level in the light of JPL requirements on spacecraft design and operation
- Supporting JPL in the definition of intersystem interfaces such as between the spacecraft and launch vehicle and capsule, and the transformation of such interfaces into spacecraft requirements data
- Coordinating and auditing within the TRW Voyager project the interpretation of and compliance with system requirements as embodied in spacecraft design

2.2.1 Requirements Documentation

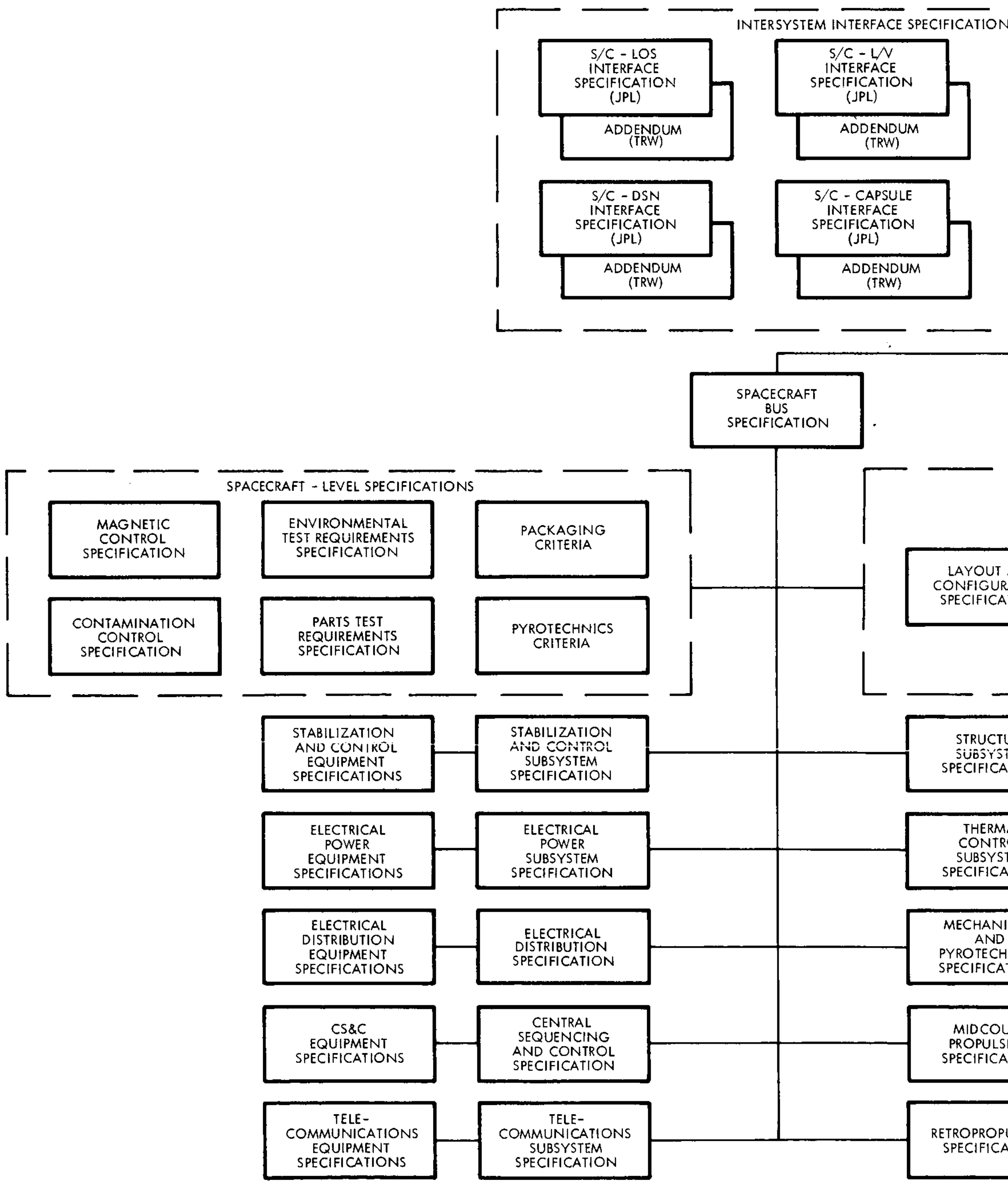
The system requirements documentation is formalized in a specification package. The organization of this package along with a definition of the scope and content of the individual documents is developed in the form of a specification plan. A hierarchy related documents is contemplated starting at the system or mission-oriented level and extending down to the configured item or individual components level. This general organization is shown in the specification tree of Figure 5.2.

2.2.2 System-Level Requirements Definition

The system requirements function 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. Additional material regarding such system requirements as developed by TRW is coordinated and documented in the corresponding system-level in-house requirements documents, including mission-oriented data such as the prelaunch and flight sequence, telecommunications guidelines, reliability requirements, mission operations requirements, maneuver and accuracy requirements, and trajectory considerations. System design factors such as spacecraft subsystem boundaries and interface requirements, test objectives, measurement guidelines, maintenance criteria, and spacecraft-support system integration are also documented at the level of the spacecraft specification and the support system specification.

2.2.3 Interface Engineering Support to JPL

The system requirements function provides in-house project direction in support of JPL for the definition of intersystem interfaces between the spacecraft and the launch vehicle, the LOS, the capsule, the DSN, the MOS and the science subsystem. Various technical specialist areas such as mechanical design, structures, electrical distribution, thermal control, and telecommunications will be brought into play as required to carry out special studies involving interface design. Functional interface considerations such as loads and environment definition and operations will also be covered. The result of such activities will be to arrive at a suitable definition of all interface items, taking spacecraft and other system considerations into account. These interface definitions will be transformed into suitable requirements data.



INTERSYSTEM INTERFACE SPECIFICATION
S/C - LOS INTERFACE SPECIFICATION (JPL)
ADDENDUM (TRW)
S/C - L/V INTERFACE SPECIFICATION (JPL)
ADDENDUM (TRW)
S/C - DSN INTERFACE SPECIFICATION (JPL)
ADDENDUM (TRW)
S/C - CAPSULE INTERFACE SPECIFICATION (JPL)
ADDENDUM (TRW)
SPACECRAFT BUS SPECIFICATION
SPACECRAFT - LEVEL SPECIFICATIONS
MAGNETIC CONTROL SPECIFICATION
ENVIRONMENTAL TEST REQUIREMENTS SPECIFICATION
PACKAGING CRITERIA
CONTAMINATION CONTROL SPECIFICATION
PARTS TEST REQUIREMENTS SPECIFICATION
PYROTECHNICS CRITERIA
STABILIZATION AND CONTROL EQUIPMENT SPECIFICATIONS
STABILIZATION AND CONTROL SUBSYSTEM SPECIFICATION
ELECTRICAL POWER EQUIPMENT SPECIFICATIONS
ELECTRICAL POWER SUBSYSTEM SPECIFICATION
ELECTRICAL DISTRIBUTION EQUIPMENT SPECIFICATIONS
ELECTRICAL DISTRIBUTION SPECIFICATION
CS&C EQUIPMENT SPECIFICATIONS
CENTRAL SEQUENCING AND CONTROL SPECIFICATION
TELE-COMMUNICATIONS EQUIPMENT SPECIFICATIONS
TELE-COMMUNICATIONS SUBSYSTEM SPECIFICATION

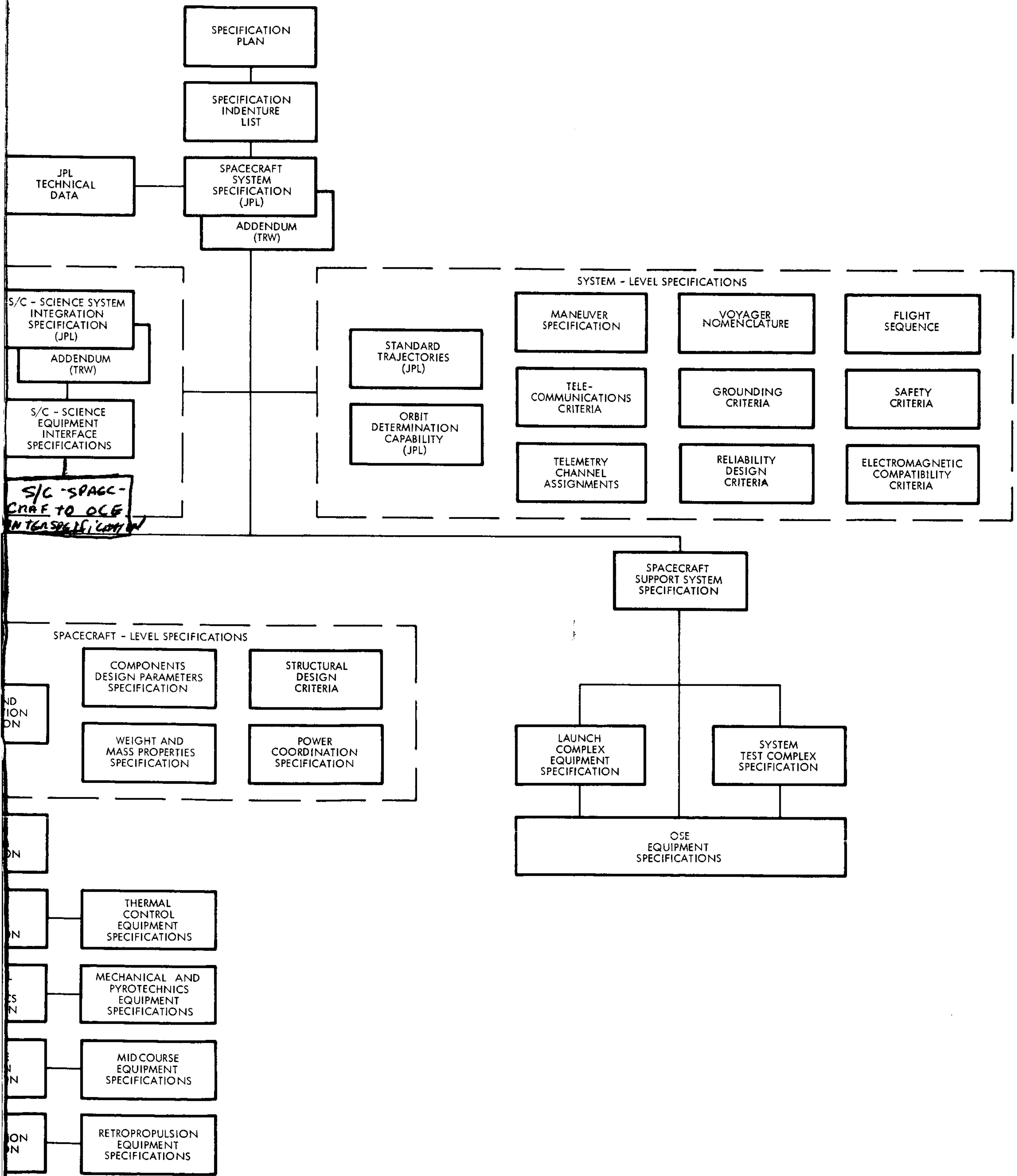


Figure 5-2. 1971 Voyager Spacecraft System Specification Tree

2.2.4 Requirements Audit and Analysis

The system requirements function serves to establish and maintain a continuous audit of the analysis and design activities as these interact with and are embodied in the system requirements data package. Operations and test plans are reviewed for consistency with the program goals and system requirements. Specification documents below the system level are reviewed in a similar fashion, with subsystem interface implementation given particular attention at this time. Most of this review activity is informal, but is also formalized at the scheduled design reviews, when design data packages are prepared and presented by the responsible engineer for each design area.

2.3 Reliability

For the Phase IB proposal and subsequent phases of the Voyager program, TRW will present its reliability program plan, a summary of which is contained in Appendix B. Because reliability is a valuable engineering tool in arriving at design decisions, it becomes an activity upon which systems engineering relies heavily. Systems engineering during Phase IB will include the continuing evaluation of reliability models, estimates, and tradeoffs. Design commitments made for reliability of the subsystems will be analyzed in accordance with their relative criticality to the mission as established by JPL. Results of reliability analyses will also constitute significant design criteria and constraints as applied to weight, magnetic properties, contamination control, electromagnetic interference, circuit tolerance control, maintainability and environment control functions, and element testability.

3. SPACECRAFT SYSTEM DEVELOPMENT

The analysis, design, and development tasks for the spacecraft as a system are organized into electrical design integration, mechanical design integration, experiment integration, and spacecraft development planning. In general, electrical design integration controls spacecraft system design and electrical interfaces and budgets power, telemetry, command and other electrical consumption. Mechanical design integration governs spacecraft configuration and interface designs, and budgets and controls the spacecraft mechanical properties. The experiment task is one of establishing requirements on the spacecraft and experimenters in conjunction with JPL and later implementing these requirements to provide a comprehensive Voyager mission. Finally, the spacecraft test planning implements the spacecraft development test requirements which form a part of the integrated test plan as approved by the Test Board.

3.1 Electrical Design Integration

The electrical design integration of the Voyager spacecraft consists of a number of systems analysis tasks, the establishment of electrical interface criteria and constraints, the determination of system level test points, and the specific determiration and coordination of the spacecraft electrical interfaces with the science payload, the lander capsule and the launch vehicle and launch complex equipment as specified in the mission requirements. The product of certain of these analyses is a set of requirements for subsidiary hardware for the electrical distribution subsystem.

The Voyager approved standard parts and material application lists, a key requirement for all electrical design, results from the electrical design integration effort. These lists evolve early in Phase IB and are updated as new requirements are generated. The OGO and Pioneer list will be the basis for the initial lists, tailored as necessary to meet Voyager requirements.

During the electrical subsystem design definition, worst case circuit analysis will be conducted. The results of individual subsystem analysis will be integrated to formulate the spacecraft total worst case analysis thus identifying critical parts and establishing the basis for part level reliability applications.

3.1.1 System Requirements Analysis

During Phase IB a detailed analysis of the Voyager functions which influence the electrical systems design will be conducted to define the requirements upon the spacecraft subsystems to ensure compatible interaction among subsystems and to determine potential problem areas. The analyses are continuing tasks and will proceed throughout the design and test phases of the Voyager program.

The analysis of total spacecraft power requirements will be updated from the preliminary information available and maintained throughout the program. Detailed operating configurations, in conjunction with total mission sequencing and operations, will be established and electrical load profiles generated for design and operations planning.

As a result of the systems test planning development, requirements for systems level test points will be defined to be implemented in the spacecraft integrated design. The design integration and the system test design will be studied concurrently to optimize both the quality of systems testing and the implementation of the test points. The test point implementation will define the hardline test connectors which will determine the EOSE interface with the spacecraft. In conjunction with the test planning, an analysis of the launch site testing will produce requirements for test points and control lines. The implementation of these will define electrical interfaces with the launch complex.

The preliminary telemetry measurement assignments will be analyzed to determine the adequacy of measurements of system parameters and the relative importance of each measurement. These measurements will be coordinated among the subsystem design groups, the test planning and launch operations groups, and the missions operations planning group. The assignment of measurement points will be analyzed from the standpoint of spacecraft state-of-health determination, the capability for diagnostic and failure analysis, the determination of the proper operation of redundant operating equipment, and the instrumentation of the detailed flight sequence of events. Specific attention will be devoted to establish the engineering instrumentation for the 1969 test spacecraft to permit maximum evaluation of the Voyager capability prior to the 1971 mission.

The on-board sequencing and control functions and the ground command functions will be re-evaluated with particular attention to backup capability of functions critical to the success of the mission and to the selection of redundant on-board subsystem equipment. The detailed operations of each of the subsystems will be analyzed to determine areas where the reliability of the subsystem can be measurably improved by the injection of backup signals, either by on-board generation or ground command. Additional analysis of the requirements during the flight for the verification of data prior to the initiation of critical event sequences and the need for enabling signals from the ground for these critical sequences will be conducted. Methods of optimizing the control of these sequences or events from the point of view of reliability will be re-examined. Optimum reliable control of spacecraft occurs when the event is initiated by a previously verified on-board generated control signal simultaneously backed by a ground command. Operating situations which for any reason prevent this simultaneous control of the critical operations will be analyzed in detail to maximize the over-all mission reliability.

3.1.2 Interface Analysis

Phase IA studies have generated preliminary system functional diagrams and descriptions. Phase IB studies will provide functional specifications to permit detailed electrical interface designs to be implemented during Phase II.

The intersubsystem interfaces within the spacecraft will be analyzed in detail to ensure the proper functioning of the integrated spacecraft systems. The electrical connections of one subsystem to another will be examined in every case to determine that the signal levels, circuit loading, and shielding and grounding implementation are compatible. Multiple connections of subsystems or elements within subsystems to single signal source will be examined to assure that no detrimental coupling from one user to the others exist through the impedance of the signal source.

Electrical outputs from each subsystem to the signal conditioning and data handling equipment will be examined to assure that each measurement signal is compatible with the capabilities of the signal conditioning equipment, that end-to-end calibration and measurement accuracy are maintained, and that there are no common impedances among the measurements which will allow errors to be introduced into one measurement by another.

The interface between the spacecraft systems and the science payload and science DAE will require detailed definition and analysis during the next phases of the Voyager program. Certain assumptions have been made during Phase IB which have attributed functional capabilities to and requirements for the science DAE and individual experiments. In conjunction with JPL, the functional and circuit interfaces between the spacecraft and the scientific equipment will be defined in detail and a total electrical interface established.

The interface between the lander capsule and the spacecraft remains to be established in detail. As in the science payload the total electrical interface between the lander capsule and the spacecraft will be detailed in conjunction with JPL.

The interface between the launch vehicle and the launch complex equipment will require detailed definition in conjunction with JPL.

3.1.3 Electromagnetic Compatibility

To achieve systems electromagnetic compatibility, it is necessary to develop, on an over-all systems basis, interference limits and methods of control of desired and undesired electromagnetic energy and the protection of sensitive circuitry. The criteria and controls will thus be establised and implemented on all spacecraft systems and those interfacing with the spacecraft in mutual effort with JPL.

Once the electromagnetic interference limits have been established and the environment defined (including the ground environment at the launch site and the flight environment), an electromagnetic compatibility control plan will be developed. In addition to systems and subsystems design and analysis studies, the control plan will establish a consistent and practical ground philosophy and the methods of implementing the criteria for bonding, shielding, circuit isolation, and interconnect cabling. The plan will contain requirements for the management mechanics to ensure that effective electromagnetic control engineering will be reflected in equipment and subsystem designs in accordance with the established criteria and methods.

3.1.4 Magnetic Control

The magnetic control program proposed by TRW consists of the following approach:

- Careful magnetic design integration and control
- Materials and parts guidelines
- Vendor control
- Subsystem magnetic testing
- Spacecraft magnetic testing

a. Magnetic Design Integration

An operational directive for the control of magnetic properties (see Appendix C) will be prepared to specify the magnetic requirements, identify the general approach to magnetic control, assign organization responsibilities for the magnetic control activities, and plan the orderly sequence of these activities. Early in the program, the acceptable levels of magnetic fields will be defined for the subsystems and units of the spacecraft. The magnetic field at the magnetometer sensor will be calculated on the basis of magnetic field measurements of materials and equipment and the positioning of the units on the spacecraft. A magnetic analysis will define the magnetic requirements for equipment and parts and identify the areas of significant design change to obtain acceptable magnetic system characteristics. From the magnetic analysis a list of magnetically acceptable parts will be generated as requirements to the standard approved parts program and a magnetic properties test specification and procedure generated for components, assemblies, and the spacecraft.

b. Material and Parts Guidelines

The materials and parts used for the Voyager spacecraft will be approved for proper magnetic properties before they are incorporated in the approved parts and material list. The program of magnetics control imposes requirements on reliability and quality assurance for procurement purposes and the spacecraft design approach. The magnetic control plan will contain these guidelines and will be submitted to JPL during Phase IB.

c. Subsystem and System Testing

Certain breadboards will be tested to evaluate ways of reducing the magnetic fields. During assembly and subsequent proof testing the subsystem assemblies and the spacecraft will be evaluated for magnetic fields. The magnetic testing is discussed in the subsystem development discussion and in the spacecraft test section (subsection 6). The design and development integration associated with spacecraft magnetic requirements will be controlled through test specifications and procedures as outlined in the Magnetic Control Plan.

b. Vendor Control

To control the magnetic properties of vendor procured items, provisions in each contract will stipulate a maximum allowable magnetic field. The resulting assemblies will undergo magnetic testing by TRW as monitored by Quality Assurance.

3.1.5 Voyager System Instrumentation

A detailed analysis in Phase IB of the engineering instrumentation requirements will define the optimum system instrumentation based upon system operational and final design parameters. The instrumentation requirements for each subsystem will be coordinated among subsystem design groups. This analysis will incorporate the results of reliability analyses and such other factors as redundancies, diagnostic and failure analysis, telemetry capability, and ground data handling requirements. Specific attention will be directed to establishing the engineering instrumentation for the 1969 test flight to permit maximum evaluation of the Voyager capability prior to the 1971 mission.

Tradeoffs will be required throughout the system between degree of refinement or diagnostic capability and added weight and complexity, particularly with respect to event measurements. For example, the receipt of a ground command might require adding a transducer with its additional weight and circuitry. The point in the chain of events at which a particular measurement is taken becomes a matter of compromise. The tradeoffs will be evaluated based on the purpose of the individual measurement, the degree of complexity or weight involved, possible alternatives, and the implications on over-all system operation.

3.2 Mechanical Design Integration

3.2.1 Spacecraft Requirements Analysis

Data required for analysis of the mechanical characteristics of the spacecraft are obtained from the system engineering mission analysis, the JPL mission specifications, and the launch vehicle system data. The spacecraft analyses include:

- Dynamics analysis
- Structural loads and design criteria
- Thermal requirements
- Mass properties analysis
- Operational influences.

a. Dynamic Analysis

Analyses of the dynamic behavior of the Voyager spacecraft during transportation, launch and boost, separation, midcourse velocity corrections, and Mars orbit injection will be refined during Phase IB, generally on the basis of existing digital computer programs. All significant tolerances in system characteristics will be examined to ensure satisfactory operational performance of the spacecraft. The analytical work will be supplemented by test data during Phase II.

The dynamic environment defined by anticipated ground handling and transportation procedures will be applied to the spacecraft design to ensure that the spacecraft will not be adversely affected by these environments. The effects of launch and boost environments on the spacecraft will be determined. The spacecraft will be analyzed for axial, lateral, and torsional responses under the vibration levels associated with the launch booster. Dynamic clearance between the spacecraft and shroud will be determined. Axial response will be computer using lumped spring-mass model simulation. The nonuniform lateral and torsional mass and stiffness distributions of the spacecraft and supports along with a lumped spring-mass injection motor simulation will be input to an available digital computer program. The program solves the Timoshenko beam equations subject to the appropriate boundary conditions and furnishes generalized model characteristics. Spacecraft responses will then be computed by modal techniques. Dynamic responses of components and spacecraft structure will also be assessed by a vibration survey development test. A spacecraft structural model will be gradually subjected to vibration which simulates the maximum environment expected during flight. Structural responses will be recorded and used along with the analytical results to ensure adequacy of equipment isolation and over-all spacecraft design.

Dynamic load factors will be evaluated for each of the following additional mission events:

- Centaur shroud separation
- Deployment of the scientific packages and the high-gain and low-gain antennas
- Midcourse velocity correction of the spacecraft
- Ejection of outer portions of lander shroud
- Separation of lander from spacecraft
- Ignition and burning of the Mars injection motor.

The results of these calculations will be incorporated in the design of the spacecraft to insure that all operational tolerances are maintained throughout the mission.

A parametric study of the booster-spacecraft separation will be performed. Performance characteristics of the retrorockets and the calculated thrust misalignment will be combined with various geometrical and mass misalignments of the booster and spacecraft. These data will be used to determine the resulting spacecraft altitude, tipoff rates, and separation velocities.

The spring separation of the spacecraft from the remaining portion of the lander capsule will be analyzed by means of an existing digital computer program and the results compared with separation tests. A sufficient number of tests will be made to obtain a statistical representation of the tipoff resulting from all tolerances of alignment and disturbing torques which can arise from the release system and separation springs.

b. Structural Loads and Design Criteria

The structural loads requirements and design criteria established during Phase IA will be updated during Phase IB, based upon the Voyager requirements established by JPL. Specific loads and criteria will be established for the 1969 test flight and 1971 missions for the booster-Voyager combinations. The structural design criteria document will specify all structural design requirements for the spacecraft. The results of the dynamic analyses will be combined with the static loads analyses to arrive at combined static-dynamic load criteria.

c. Thermal Requirements Analysis

As discussed in more detail in subsection 4.2, the thermal environment to which the spacecraft will be subjected throughout the mission will be updated as the first thermal task. This environment includes on-stand heating, radiant heating from the fairing, aerodynamic heating after fairing jettison, non-nominal attitude with respect to the sun varying solar intensity throughout the mission, radiative heating from the deboost motor plume, eclipse, and the Martian orbital environment. The magnitude of the on-stand heating will be determined from the duty cycle schedule of the spacecraft equipment during on-stand checkout. The radiant heating from the fairing will be determined from the parametric curves of internal fairing temperatures as a function of time and fairing insulation utilized. The aerodynamic heating after fairing jettison will be determined by straightforward aerodynamic heating computation utilizing the 3σ low launch trajectory. The heating rates during the time when the spacecraft is in a non-nominal attitude with respect to the sun will be determined from the sun-look angle versus time information available from the launch trajectory analysis. The varying solar intensity throughout the mission is a straightforward calculation. The calculation of the magnitude of the radiative heat input to the spacecraft from the molten alumina particles in the deboost motor will be performed utilizing the TRW wake analysis program, a program tested and proved during the Vela and Minuteman programs.

The lengths of the eclipses which may be experienced in the region of the earth or during the Martian orbit will be determined from the trajectory analysis. The planetary heating environment experienced by the spacecraft when it is near the earth or Mars will be computed utilizing trajectory information and the TRW planetary heating program.

These environments will be incorporated in the detailed thermal analyses conducted for all portions of the spacecraft. The resulting response of all elements of the spacecraft to these inputs, coupled with the internal power dissipation modes of the mission, will determine the thermal design of the spacecraft and any launch restraints, if required.

d. Mass Properties Analysis

Mass properties will be calculated, including weights, centers of gravity, moments and products of inertia, and mass distribution for the Voyager and subassemblies. This task will be implemented in accordance with MIL-M-38310 as follows:

- Mass properties records will be maintained by using the TRW mass properties computer program to compile and compute weights, centers of gravity, moments of inertia, products of inertia, and mass distribution
- Weight review meetings will be held, as required, during the design to review the weights of all components and the weight tradeoff studies and to initiate action to effect weight reductions
- During production, actual weights of completed components and subassemblies will be entered in the computer program
- Experimental values of moments of inertia will be obtained on the proof test model to verify the computer program.

The relationship of mass properties studies with other tasks is depicted in Figure 5-3 . The flow of data through this circular path is continuous throughout design and development.

e. Operational Influences

All functions from spacecraft integration through launch will be analyzed to establish design constraints on the spacecraft and its subsystems, and to identify the detailed requirements for OSE. The assumptions used during Phase IA to establish preliminary OSE designs will be corrected as a reult of the functional analysis and incorporated into the requirements for launch support equipment.

The same functional analysis will establish the requirements for mission dependent equipment and the facility requirements at the ETR in support of the spacecraft during prelaunch and launch activities.

3.2.2 Configuration Design

The Phase IA layout will be updated based upon additional mission definition and spacecraft requirements received from JPL, with special

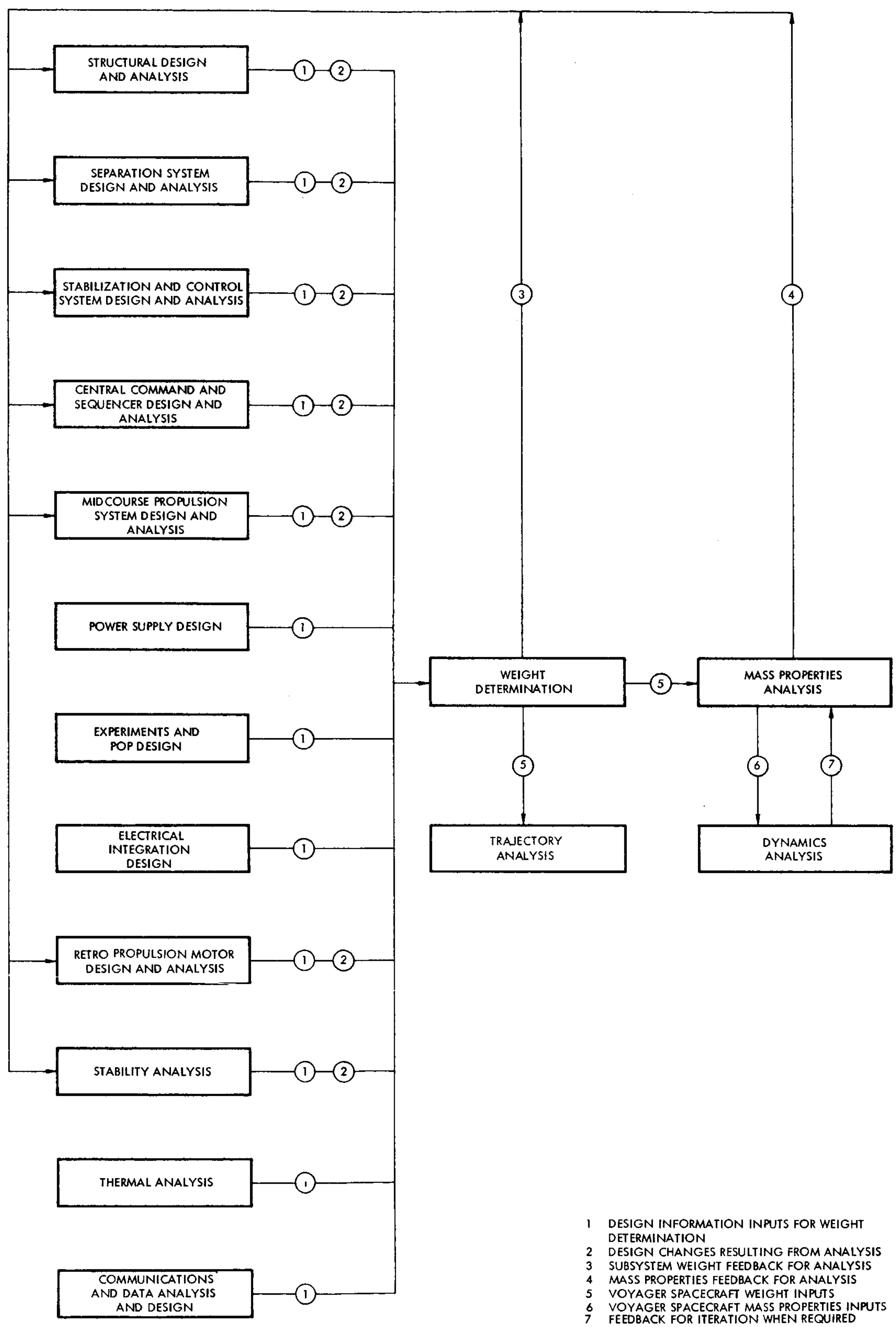


Figure 5-3 . Mass Properties Analysis, Task Interrelationships attention to location of the equipment to provide optimum mass properties distribution, thermal environment, and access for assembly and test. The structural load paths will be optimized and design requirements established for the structural subsystem. Final selection of ordnance systems will be made after the mission sequence of events is made final. The interfaces with the Centaur and the flight capsule will also be established.

A metal model for physical design intergration will be constructed using soft tooling. The configuration model will be used for many purposes, the most important of which are:

- Physical layout checks
- Physical configuration control
- Plumbing routing development
- Electrical harness routing verification
- Mechanical functional demonstrations
- Fastener definitions.

The mechanical design will be studied to be certain that it readily permits maintenance during all phases of spacecraft ground life. Of particular importance is the remove-and-replace capability of units without destroying the validity of previously tested portions of the spacecraft.

The design constraints resulting from the thermal analysis will be incorporated in the studies of mechanical design integration together with special attention to location and orientation of the antenna systems to provide continuous earth viewing with a minimum of antenna gimbal motion. The location and orientation of the planet-oriented package will also be studied in depth to permit Mars orientation with a minimum of maneuvers and with accurate knowledge of the view direction of the POP experiments and cameras. The body-mounted experiment and guidance sensor viewing requirements will be further defined to optimize the locations of experiments on the spacecraft and to permit off-line experiment alignment.

3.2.3 Spacecraft Interface Definition

Mechanical design integration will require definition and control of the following interfaces:

- Spacecraft to Centaur adapter
- Spacecraft to OSE
- Spacecraft subsystems
- Spacecraft to flight capsule.

The interface document prepared during Phase IA, defining the interface between the spacecraft and the Centaur adapter and fairing, will be revised. The requirements resulting from the interface between the spacecraft and Centaur will be used as input to the structural loads analysis and will result in design criteria for the spacecraft and its subsystems for the 1969 test flight and the subsequent Mars missions. These criteria will also establish some of the requirements for development and design verification tests.

Interface requirements of the spacecraft will be coordinated with JPL and the Centaur contractor. Dynamic data resulting from the Voyager spacecraft design will be forwarded to JPL and the Centaur contractor in sufficient time to permit the Centaur contractor to conduct the system (booster plus payload) dynamic analysis.

The mechanical interface between the spacecraft and the OSE will be defined in the mechanical OSE interface specification. These interfaces include attach points on the spacecraft for lifting and handling, service interfaces for fluid and electrical connections, OSE dynamic and shock requirements for spacecraft handling and transportation, and thermal requirements for protection equipment.

The subsystem interface specification prepared during Phase IA will be definitively prepared during Phase IB and released early in Phase II.

The interface between flight spacecraft and the flight capsule will be designed during Phase IB as a support task to JPL. Interface control specifications will be prepared describing this, including mechanical, electrical, environmental, and safety aspects.

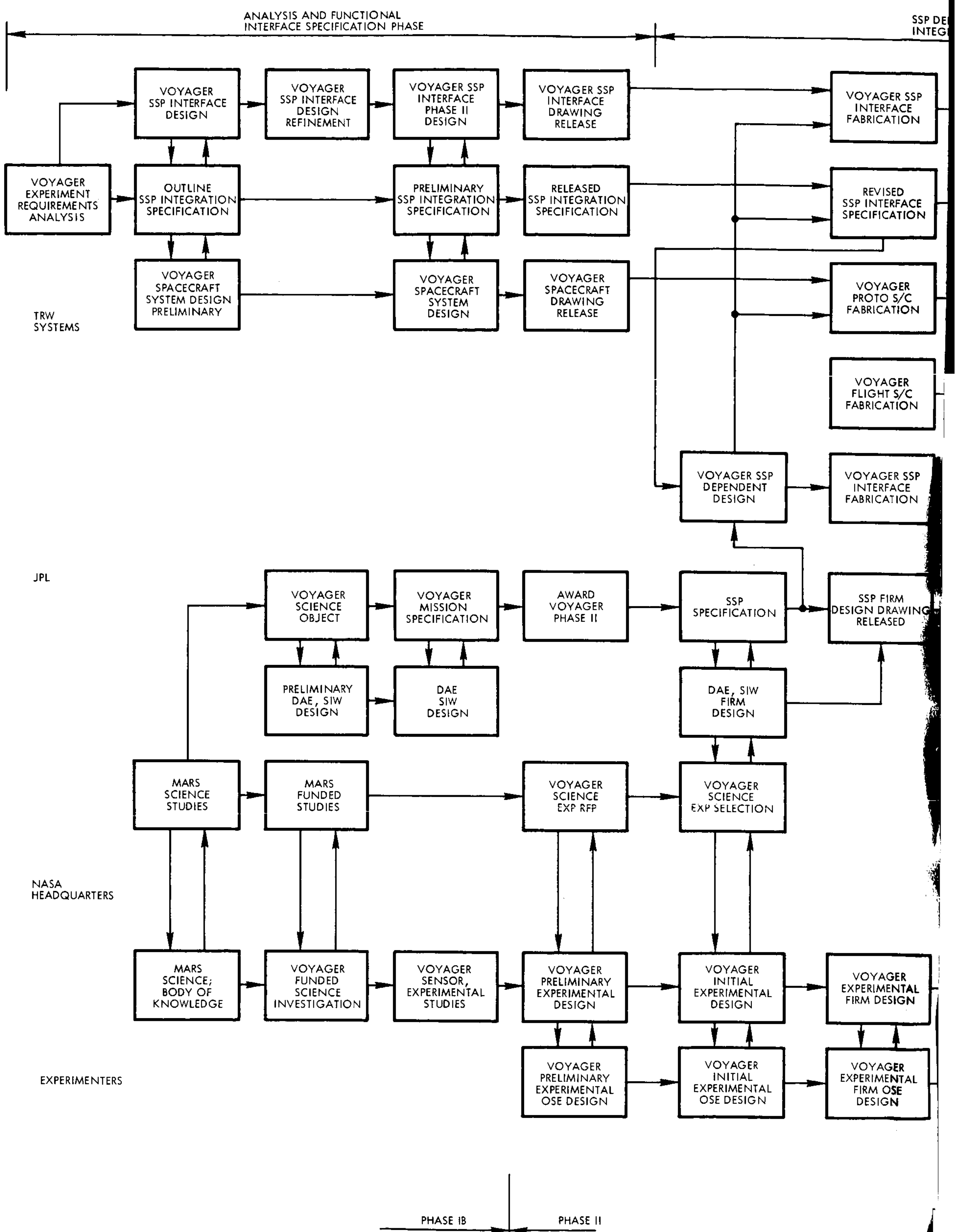
3.3 Experiment Integration

Design integration with respect to the science subsystems covers all TRW activities relating to the subsystems, from initial liaison to postlaunch support to JPL. The three major TRW tasks will be detailed definition of the spacecraft science payload interface with the spacecraft bus, integration of the SSP into the spacecraft bus, and testing of the SSP. Much of what is described in this section is based on TRW experience with the OGO program. A much more detailed statement of the proposed techniques and procedures than is given here has been prepared and is available upon request. During Phase IB an experiment design integration plan will be presented to JPL defining the experiment design integration role.

Figure 5-4 is an over-all flow chart indicating the three major phases of the integration task and the functional relationships of the elements in each phase. Figure 5-5 shows the proposed schedule.

3.3.1 SSP Integration Management

An SSP integration manager will be assigned who will have overall responsibility for SSP integration, for liaison, and for coordination. He will have current and detailed knowledge of the spacecraft interface with SSP, and will understand the purpose and operation of the experiments. He will have final responsibility within TRW for the experiment interface designs, the integration procedures, and the experiment testing program. He will work closely with JPL and with the experimenters to coordinate all aspects of the SSP and to resolve any differences or discrepancies. In the absence of JPL or experimenter personnel, he will represent them to other elements of the Voyager program, in both managerial and technical matters. He will appoint a staff of responsible engineers, one for each three or four experiments. The responsible engineer will handle all integration tasks for that experiment including



ANALYSIS AND FUNCTIONAL INTERFACE SPECIFICATION PHASE
SSP DE
INTEG
VOYAGER SSP INTERFACE DESIGN
VOYAGER SSP INTERFACE DESIGN REFINEMENT
VOYAGER SSP INTERFACE PHASE II DESIGN
VOYAGER SSP INTERFACE DRAWING RELEASE
VOYAGER SSP INTERFACE FABRICATION
VOYAGER EXPERIMENT REQUIREMENTS ANALYSIS
OUTLINE SSP INTEGRATION SPECIFICATION
PRELIMINARY SSP INTEGRATION SPECIFICATION
RELEASED SSP INTEGRATION SPECIFICATION
REVISED SSP INTERFACE SPECIFICATION
VOYAGER SPACECRAFT SYSTEM DESIGN PRELIMINARY
VOYAGER SPACECRAFT SYSTEM DESIGN
VOYAGER SPACECRAFT DRAWING RELEASE
VOYAGER PROTO S/C FABRICATION
TRW SYSTEMS
VOYAGER FLIGHT S/C FABRICATION
VOYAGER SSP DEPENDENT DESIGN
VOYAGER SSP INTERFACE FABRICATION
JPL
VOYAGER SCIENCE OBJECT
VOYAGER MISSION SPECIFICATION
AWARD VOYAGER PHASE II
SSP SPECIFICATION
SSP FIRM DESIGN DRAWING RELEASED
PRELIMINARY DAE, SIW DESIGN
DAE SIW DESIGN
DAE, SIW FIRM DESIGN
MARS SCIENCE STUDIES
MARS FUNDED STUDIES
VOYAGER SCIENCE EXP RFP
VOYAGER SCIENCE EXP SELECTION
NASA HEADQUARTERS
MARS SCIENCE; BODY OF KNOWLEDGE
VOYAGER FUNDED SCIENCE INVESTIGATION
VOYAGER SENSOR, EXPERIMENTAL STUDIES
VOYAGER PRELIMINARY EXPERIMENTAL DESIGN
VOYAGER INITIAL EXPERIMENTAL DESIGN
VOYAGER EXPERIMENTAL FIRM DESIGN
EXPERIMENTERS
VOYAGER PRELIMINARY EXPERIMENTAL OSE DESIGN
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PHASE IB
PHASE II

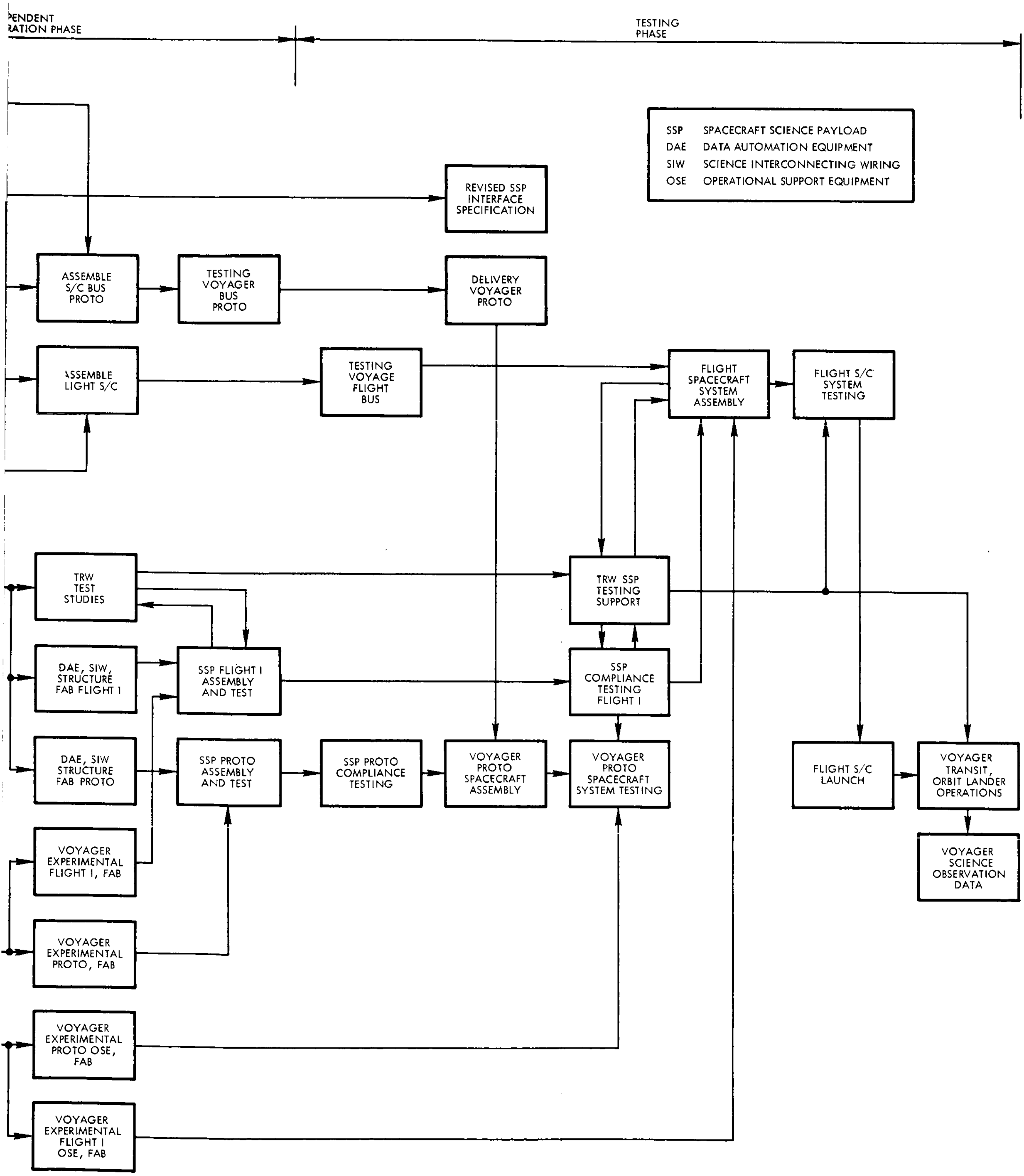
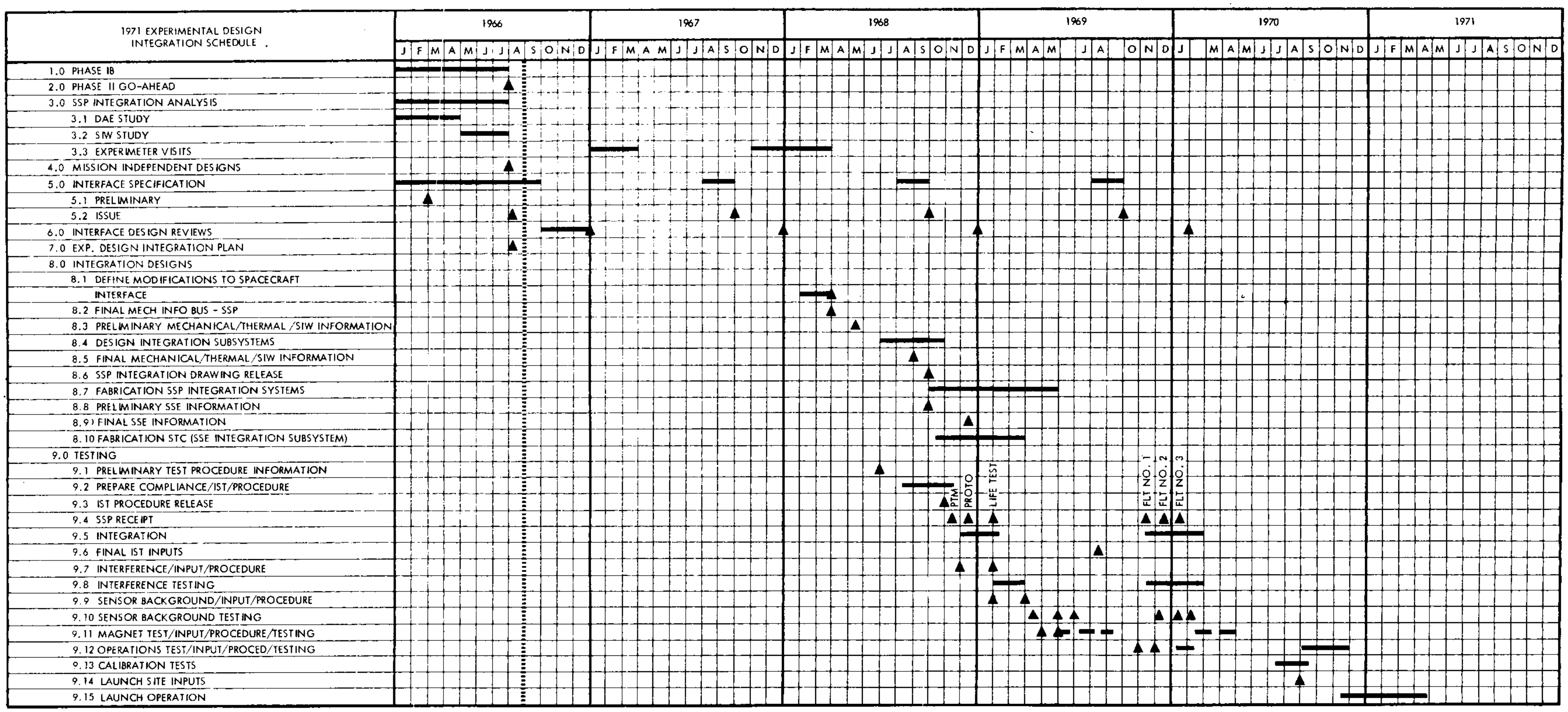


Figure 5-4. Spacecraft Science Design Integration

2



GLOSSARY

SSP - SPACECRAFT SCIENCE PAYLOAD
SIW - SCIENCE INTEGRATION WIRING
DAE - DATA AUTOMATION EQUIPMENT
SSE - SPACE SUPPORT EQUIPMENT

Figure 5-5. 1971 Experimental Design Integration Schedule evaluation, surveillance, and overall integration hardware provisions. This technique was developed for the OGO experiment integration and has proved highly satisfactory.

3.3.2 Interface Definition Phase

During Phase IB the initial task will be to collect as much information as possible regarding the SSP and its experiments and coordinate this data with concurrently available design data on the spacecraft bus in order to define the SSP-spacecraft bus interface. The first step will be analysis of requirements to determine what characteristics these impose on the interface. At this point it will be possible to define precisely only those features of the interface which are mission independent, but when these are defined a specification can be prepared. A preliminary SSP integration specification will be issued in late 1966 to define the mission independent interface.

As soon as the SSP specification is released by JPL, the SSP integration effort will increase the level of operation. On the basis of the SSP specification, new requirements analyses will be made to define the mission-dependent characteristics of the SSP/spacecraft bus interface, and will define specific hardware requirements for each experiment, such as mounting provisions on the bus, harnessing, shielding, thermal control, and similar physical requirements. The electrical interface will also be defined in detail. Although it is planned to keep the electrical interface as simple as possible by incorporating the data automation equipment (DAE) into the SSP, there will be some minimum interface requirements remaining.

The final and essential output of the interface definition effort will be a released SSP integration specification, which will specify completely the interface requirements with respect to mechanical, electrical, thermal, telemetry, magnetic, and orientation characteristics. The electrical interface will include timing and synchronization signals, logical control, and commands as well as specifications for noise and impedances in both directions. Any requirements concerning particulate radiation or electromagnetic radiation will be included in this procedure.

In addition, a supplement to the interface specification will be issued to provide the experimenters with necessary background information such as the interface circuits, the grounding system, the timing signal characteristics, the transfer characteristics of the spacecraft data handling subsystems, and appropriate supporting data. For the same purpose, the experimenters will be provided with a brochure describing the spacecraft bus and its functions.

The interface definition effort will be heaviest at the beginning, but will continue over much of Phase II. Interface design review meetings will be held regularly with JPL and the experimenters in order to assure complete mutual understanding of requirements on both sides of the interface.

Figure 5-6 identifies various interface characteristics and experiments which may be expected to pose special problems.

3.3.3 Integration Phase

The integration phase will begin with the first deliveries of experiment hardware to TRW. By this time an SSP integration laboratory will have been established and equipped; it will be staffed by the SSP integration manager, and his responsible engineers plus the necessary technical and clerical support personnel.

Once an experiment has been delivered to the SSP Integration Laboratory, it is subjected basically to two tests. The first is a compliance test which determines whether the experiment complies with the SSP integration specification and is therefore compatible with the spacecraft. The second is a functional test which determines whether the experiment operates in accordance with its own specification. Both tests will be repeated several times, and the functional test will be repeated at every step of the testing schedule up to the launch pad under a great variety of conditions.

Certain experiments will be delivered to TRW individually (if, for example, they are appendage-mounted) while others will be assembled together with other components of the SSP at JPL and delivered as a complete assembly. The TRW experiment responsible engineer will design a compliance test procedure which will be completed and approved before receipt of the package. The functional test procedures will be prepared by the respective experimenters for use with their SSE.

		BODY MOUNTED EQUIPMENT							PLANET ORIENTED SENSORS						APPENDAGE(S) MOUNTED EQUIPMENT				
		ENERGETIC PARTICLE DETECTORS	PLASMA DETECTORS	MASS SPECTROMETERS	ABSORPTION SPECTROSCOPY	LOW FREQUENCY RADIO RECEIVERS	MICROMETEOROID DETECTORS	DATA AUTOMATION EQUIPMENT	CAMERAS	SPECTROMETERS	RADIOMETERS	PHOTOMETERS	METEORPHOTOMETER	POLARIMETER	MAGNETOMETERS	MICROMETEOROID DETECTORS	ION TRAPS MASS SPECTROMETERS	SOLAR MONITORS	PLASMA PROBES
ELECTRICAL INTERFACE	NOISE SUPPRESSION					●		●	●						●		●		
	CABLING DESIGN	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●
	LOGICAL CONTROL	●			●			●	●	●	●	●			●			●	
	TIMING SYNCHRONIZATION	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●	●
	BULK STORAGE	●						●	●	●	●						●	●	
	POWER AND DUTY CYCLE							●	●	●	●	●		●					
MECHANICAL INTERFACE	APERTURES		●	●	●					●	●						●		●
	ALIGNMENT				●				●	●	●			●	●			●	
	SHOCK AND VIBRATION		●	●			●		●	●						●			
	LOOK ANGLES	●	●		●				●	●		●	●				●		●
	DEPLOYMENT					●									●	●	●	●	●
	POINTING				●				●	●	●	●	●	●	●			●	
THERMAL INTERFACE	SURFACE PROPERTIES	●	●		●				●	●	●		●				●	●	●
	TEMPERATURE			●	●			●			●				●	●	●	●	●
	TEMPERATURE GRADIENTS			●	●				●	●			●						
	APERTURE SIZING AND COVERS		●		●				●	●	●	●	●	●					
	DUTY CYCLES			●	●			●	●	●	●	●	●		●	●			●
	SPECIAL COOLING			●	●					●	●								
BACKGROUND INTERFACE	LIGHT LEAKAGE				●				●	●	●	●	●	●					
	FIELDS OR INTERFERENCE	●		●											●	●	●	●	●
	MASS DISTRIBUTION	●															●		●
	OUTGASSING									●									
	ELECTROSTATIC SURFACES		●	●													●		●

Figure 5-6. Possible Voyager SSP Experiments and Special Interface Requirements

Upon delivery of the experiment package to TRW, compliance and functional tests are first performed as bench tests under ambient conditions. Ordinarily these tests are performed by the TRW engineer with the assistance of the experimenter. The package is then integrated into an experiment subsystem and the tests are repeated.

They are repeated during and after environmental testing of the assembly (or of the package, if it is tested individually). The package is then certified by a JPL representative as approved for installation on the spacecraft bus.

This procedure makes use of the spacecraft simulator (to determine compliance) and the experiment SSE (to determine function). The experiment package is operated in all of its modes, with all possible input variations and combinations consistent with normal operation. Data is fed to the laboratory computer and printed out on a high-speed printer.

After the package has been certified for integration, it is integrated into the spacecraft bus in accordance with a procedure prepared by the responsible engineer and coordinated with JPL and the experimenter. The first step is mechanical installation on the spacecraft. Then the experiment is electrically connected to the spacecraft through a fuse box which permits manual completion and interruption of each line, providing a test point for each line, and fuse protection for both spacecraft and experiment. Once the gross electrical characteristics of the interface are found satisfactory, the fuse box is replaced by an interface test box, which provides a test point for each line. Amplitude and noise measurements are made on each line, and if they are within acceptable limits, the test box is removed and the experiment connected directly to the spacecraft.

At this point a functional test is run, complete with acquisition and printout of data. This requires appropriate stimulation of each experiment with external sources or internal calibration devices. The data handling system and experiment are operated in all their modes. Test results are retained for comparison with later runs of the same experiment. If the results are satisfactory, the particular experiment may be considered to be integrated.

3.3.4 SSP Integrated System Testing

Although we have previously identified the compliance functional tests as the major types of test, there are a number of other tests of the experiments which are essential and are grouped in the category of special tests. These are performed after integration of the SSP into the spacecraft. They may be scheduled before or during the various repetitions of the integrated systems test (IST).

The IST itself is intended to simulate observatory operation during the entire mission, and therefore includes operation of the experiments and recording of the resulting data. The following special tests are performed.

a. Flight Spacecraft Interference Test

The flight spacecraft interference test locates any interexperiment interference in the presence of the operating spacecraft subsystems. It consists of data runs with the complete flight spacecraft, with each experiment operated in its various modes (particularly in the most sensitive mode). This test occurs early in the cycle to allow for investigation and remedy of any interference problems.

b. Sensor Background Test

The sensor background test records realistic interference measurements on certain RF sensitive experiments. Since it is desirable to minimize background noise, the test is performed in a relatively open area at a time when all facility power is off (except that required for the test). Background measurements are made with all spacecraft systems off, and repeated as units are turned on one at a time to a full-on condition.

c. Magnetic Properties Testing

The required high sensitivity of the magnetometer experiments imposes very severe requirements on the spacecraft with respect to keeping magnetic fields to a minimum. A fairly elaborate test program measures the magnetic characteristics of the spacecraft and calibrates and checks the magnetometer experiments.

The first tests map the permanent and induced magnetic fields of the complete spacecraft, determine the possible variations in this field under worst-case magnetic conditions, and reduce the permanent field, if necessary, through compensation. The appendages and solar arrays may be tested separately from the spacecraft for these tests.

The second tests are made on the operating spacecraft and are intended to determine the interference seen by the magnetometer and VLF experiments. A special test is made for the benefit of those experimenters having charged-particle detectors; a mu-meson background radiation test, performed with the particle detectors operating continuously over a 12-hour period to monitor cosmic ray background. From this the experimenters can derive a sensitivity figure for the experiments concerned.

d. Tape Simulation of Transit/Orbital/Landed Operations

A magnetic tape will be prepared to simulate the operation of a flight spacecraft throughout the mission, with respect to the spacecraft and SSP operation and generation of data from the observation of physical events. This tape will serve as a basic tool for checkout of the MOS, DSIF, and SFOF networks, as well as for quick-look and production SSP data reduction programs. Each experimenter will be provided with a copy of the test representing his experiment so that he can simulate his own data analysis.

e. Calibration Tests

Provision will be made for each experimenter to calibrate his own experiment after the entire cycle of system level observations have been completed but before launch operations begin. This requires about 50 to 75 in-line hours of spacecraft test time. For this reason, calibration tests should be strictly scheduled and carefully monitored to make sure that no testing is left to this time wher. it might have been performed at an earlier stage.

3.4 Spacecraft Development Test Planning

Under the direction and control of the Test Board and in accordance with the integrated test plan, the detailed implementation of the system development test plan will be performed.

3.4.1 Test Planning

As the design of the spacecraft and subsystems become firm early in Phase IB, the preliminary test plan will be modified to reflect the specific needs for a program of tests for development and design verification at all levels of equipment complexity. The spacecraft development tests will be incorporated into the integrated test plan which will be submitted during Phase IB.

A preliminary development plan has evolved from Phase IA, based upon the selected Voyager design. In addition to breadboard, unit, and subassembly tests of the electronics equipment, TRW Systems will use the engineering models, after unit development tests, for a system electrical compatibility test, where the equipment is integrated into a complete spacecraft engineering model. This series of tests also permits a complete compatibility evaluation of the spacecraft and its corresponding electrical operational support equipment.

The test program for each subsystem is presented in Section 4, with corresponding matrix of subsystem development and design verification tests planned during Phase IB and II.

The development of procedures is similar to the task of hardware development, i.e., early preparation of development procedures will be modified as equipment changes occur and requirements for testing changes.

During Phase IB, the critical test procedures will be identified and prepared; these procedures will be revised and updated during Phase II. Early development procedures will be prepared to form the basis for a formal procedure for design verification testing, acceptance, and qualification testing.

Test reports are prepared for each test or series of tests performed. In addition, development test reports are prepared for all significant development tests. All other test data is recorded in the engineering record books.

The engineering activities of electrical design integration require early system evaluation of the electrical portions of the spacecraft and use the spacecraft engineering model as the primary development tool followed by design verification on the spacecraft proof test model.

The mechanical spacecraft design integration activity employs configuration models during Phase IB as its initial development tools in establishing system requirements. Thereafter, the mechanical development proceeds primarily on each subsystem followed by spacecraft design verification on the spacecraft proof test model. Thus, the structural design verification, using the structural model, becomes a primary task of the structural subsystem (Section 4.1), and the thermal spacecraft development tests are included in the thermal subsystem (Section 4.2). The propulsion integration testing has been included as part of the propulsion subsystem (Section 4.3). The system mass c.g. properties will be incorporated into a subsystem separation model. The separation development testing is discussed in Section 4.1, structural subsystem. A spacecraft propulsion and stabilization control model is used for propulsion interaction tests.

3.4.2 Test Models

The spacecraft test models planned for Voyager consist of the following (in addition to engineering breadboards, unit models, structural, thermal, separation, and propulsion models as discussed within subsystem development, Section 4):

- Configuration model
- Spacecraft engineering model
- Spacecraft simulators
- Spacecraft propulsion and stabilization control model
- Proof test model
- Reliability life test

Figure 5-7 presents the general time phase relationship of these models and summarizes their test application.

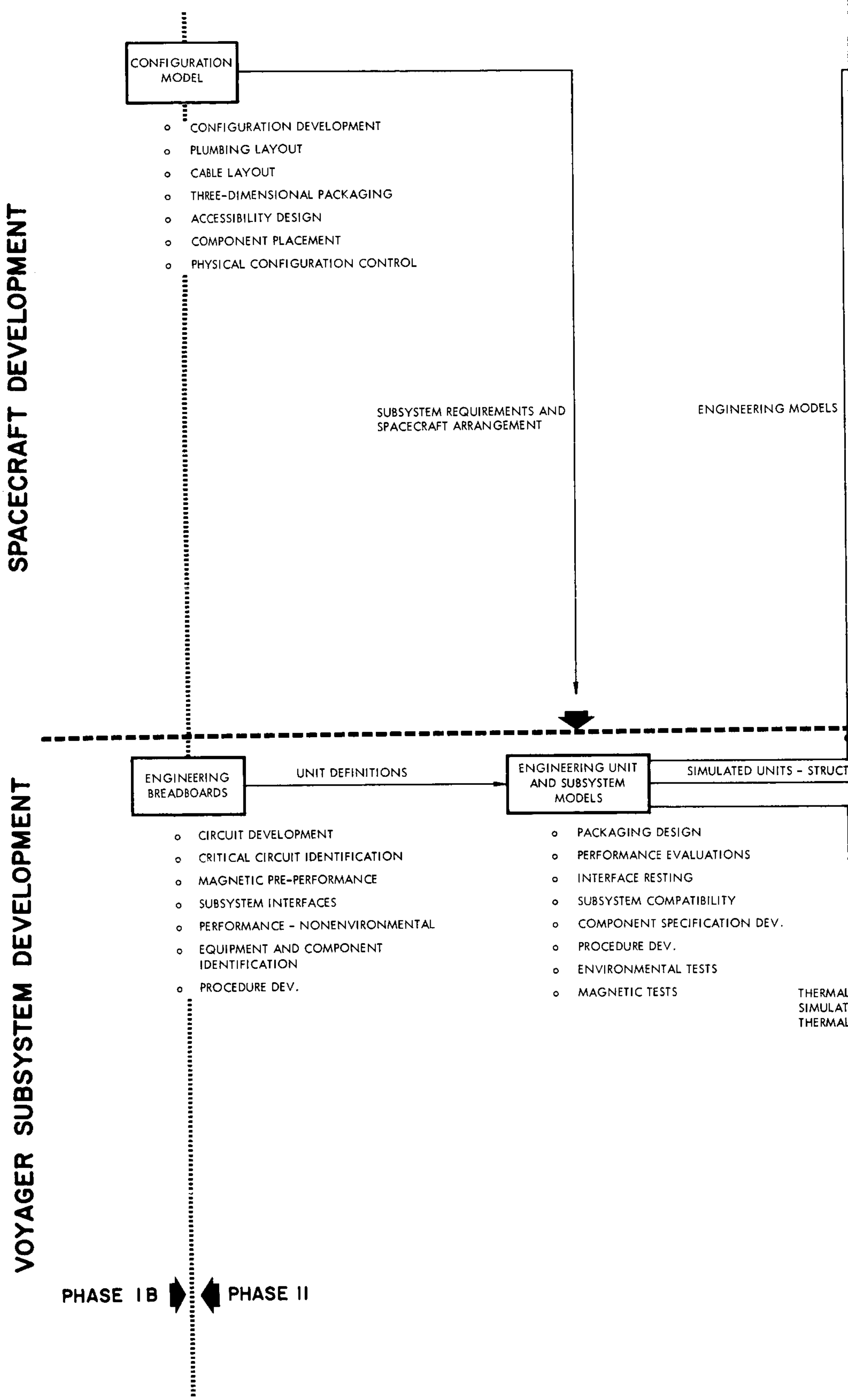
a. Configuration Model

The configuration model is employed as a system design development tool for three-dimensional layout checks and component placement evaluation, including access checks. Harness and plumbing routing configurations are developed using this model. It also becomes a continuous physical configuration control model.

The model is constructed early in Phase II from temporary tooling. The structure is of metal construction using available materials and gages, but the outline dimensions of individual pieces are retained. The structure is updated as design changes occur.

The electronic units are constructed to simulate only their housing. Aluminum sheet is employed to duplicate outline dimensions. Physical connections, i.e., fasteners and electrical connectors, are actual hardware although not required to be flight qualified.

The nitrogen and propellant tanks are initially mock-ups, but are replaced later with actual hardware. The valves, lines, and fittings are actual nonflight-rated hardware to accommodate plumbing routing design. The retropropulsion motor subsystem utilizes an inert mock-up. Antennas, booms, and the planet-oriented package will be deployable mock-ups. The solar panels are similar in configuration to flight panels except solar cells will be mock-ups.


SPACECRAFT DEVELOPMENT
CONFIGURATION MODEL
o CONFIGURATION DEVELOPMENT
o PLUMBING LAYOUT
o CABLE LAYOUT
o THREE-DIMENSIONAL PACKAGING
o ACCESSIBILITY DESIGN
o COMPONENT PLACEMENT
o PHYSICAL CONFIGURATION CONTROL
SUBSYSTEM REQUIREMENTS AND SPACECRAFT ARRANGEMENT
ENGINEERING MODELS
VOYAGER SUBSYSTEM DEVELOPMENT
ENGINEERING BREADBOARDS
UNIT DEFINITIONS
ENGINEERING UNIT AND SUBSYSTEM MODELS
o CIRCUIT DEVELOPMENT
o CRITICAL CIRCUIT IDENTIFICATION
o MAGNETIC PRE-PERFORMANCE
o SUBSYSTEM INTERFACES
o PERFORMANCE - NONENVIRONMENTAL
o EQUIPMENT AND COMPONENT IDENTIFICATION
o PROCEDURE DEV.
o PACKAGING DESIGN
o PERFORMANCE EVALUATIONS
o INTERFACE RESTING
o SUBSYSTEM COMPATIBILITY
o COMPONENT SPECIFICATION DEV.
o PROCEDURE DEV.
o ENVIRONMENTAL TESTS
o MAGNETIC TESTS
PHASE IB
PHASE II

1

SPACECRAFT ENGINEERING MODEL

DEVELOPMENT

- o ELECTRICAL SYSTEM COMPATIBILITY
- o SUBSYSTEM ELECTRICAL INTERFACE COMPATIBILITY
- o EMC
- o COMPUTER PROGRAM C/O
- o OSE AND ENVIRONMENTAL TEST EQUIP. C/O
- o THERMAL VAC. TESTS FACILITY C/O
- o VIBRATION TESTS FACILITY C/O
- o ACOUSTICAL TESTS FACILITY C/O
- o MAGNETIC TESTS FACILITY C/O
- o PROCEDURES DEVELOPMENT
- o PERSONNEL TRAINING

COMPATIBILITY TESTS

- o OSE
- o LCE - AFETR
- o S/C - CENTAUR ELECT.
- o S/C - SHROUD RF COUPLING TESTS
- o INITIAL EXPERIMENT
- o S/C - LAUNCH FACILITY ELECT. C/O
- o S/C - DSN EQUIPMENT C/O

SPACECRAFT SIMULATORS

- o DSIF COMPATIBILITY
- o SFOF COMPATIBILITY
- o MISSION DEPENDENT EQUIP C/O

PROPULSION AND SOLARIZATION AND CONTROL MODEL

- o SIMULATED MIDCOURSE INTERACTION TEST
- o SIMULATED RETRO PROPULSION INSTEACTION TEST
- o AUTOPILOT PERFORMANCE

DYNAMIC MODEL
ENGINEERING UNIT MODELS

TEST DATA

JRALLY

STRUCTURAL SUBSYSTEM MODEL

- o VIBRATION SURVEY
- o STATIC LOAD TESTS
- o TYPE APPROVAL - STRUCTURAL SUBSYSTEM
- o MECHANICAL OSE INTERFACE COMPATIBILITY TESTS

PROOF TEST AND TYPE APPROVAL UNITS

- o UNIT QUALIFICATION
- o TYPE APPROVAL DEMONSTRATION

HARDWARE
D UNITS -
.Y

THERMAL SUBSYSTEM MODEL

- o THERMAL VACUUM TESTS
- o MISSION SIMULATION
- o LOUVER CYCLING
- o HEAT FLUX AND HEAT SHORT TESTS

PROPULSION MODEL

- o VIBRATION TEST
- o MISSION SIMULATION
- o MIDCOURSE PROPULSION PERFORMANCE
- o THERMAL PERFORMANCE

SEPARATION MODEL

- o LOWER INTERFACE SEPARATION TESTS
- o UPPER INTERFACE SEPARATION TESTS

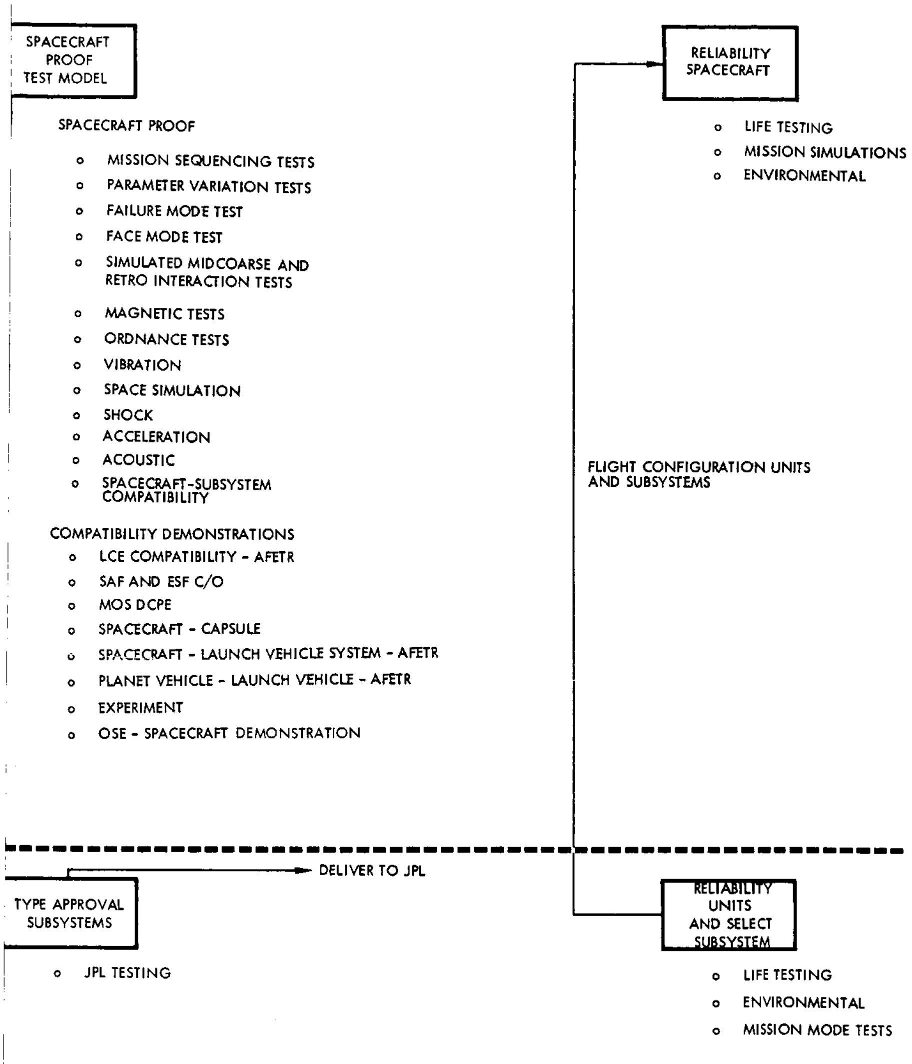


Figure 5-7. Voyager Development Models

3

b. Spacecraft Engineering Model

After each subsystem has received unit level development tests on its electrical portions, the engineering models are delivered to the spacecraft assembly area for assembly into a system electrical model. The model consists of a prototype structure containing the electromagnetic and conductive characteristics of the spacecraft. The engineering units are mounted to this structure. All electrically-operated devices are included on the model except squibs and ordnance, which are simulated. A regulated power supply is used in place of the solar array.

A complete operating engineering model of the communications and data subsystem, central stabilization and control, and command and sequencing subsystems will be used for assembly, and system testing. The spacecraft engineering model is under minimum configuration control surveillance, and has a configuration status and operating log which is verified by quality assurance personnel.

The spacecraft engineering model is primarily used for electrical design integration. It also provides for procedures, development, and personnel training. The development tests planned for this model include:

- Electrical system compatibility
- Subsystem electrical interface compatibility
- Electromagnetic compatibility
- Operational computer program checkout
- DSN equipment checkout
- Environmental facilities and special test equipment checkout

The electrical compatibility tests using the spacecraft engineering model include:

- Spacecraft - OSE
- Spacecraft - Centaur
- Spacecraft - Shroud RF Coupling
- Spacecraft - Experiments
- Spacecraft - Launch Facilities
- Spacecraft - Launch Complex Equipment

The byproducts of the electrical tests using the engineering model spacecraft are the verification of the approach to spacecraft assembly and test and the development of test procedures. However, this testing will be conducted in accordance with informal test procedures. Equipment will be operated primarily to provide information by which spacecraft performance characteristics can be evaluated. Varying the test sequence or approach will be permitted to evaluate problem areas which may arise. The electrical system testing will be designed to determine margins of safety of various functional and performance characteristics.

The electromagnetic compatibility tests will search for potential compatibility problems, rather than simply demonstrate conformance with a particular performance specification. Any EMC problems or border-line conditions found will be investigated and remedial measures developed. Any corrective measures will be confirmed by subsequent tests. The tests will be performed in two parts, first in a simulated flight configuration to ascertain system compatibility, and, second, in conjunction with the OSE. During each of these tests, critical circuits will be monitored using laboratory instrumentation in addition to real time monitoring and recording of system outputs. Monitoring points will be selected to preclude injection of spurious signals or alteration of circuit parameters. During the tests, each unit and subsystem will be exercised in accordance with typical operational sequencing, while critical circuits and the system outputs are monitored to detect any undesired response, malfunction, or degradation of system outputs. The procedure developed during the engineering EMC tests will be used as a basis for a detailed step-by-step compatibility test procedure for formal acceptance testing.

c. Spacecraft Simulators

The spacecraft simulator employs actual subassemblies and additional equipment to demonstrate the compatibility between flight spacecraft and the DSIF. It consists of a test transponder, magnetic tape containing demodulated data, error rate tester, and a data format generator. The model is used for operational tests of the mission-dependent equipment supplied to the DSIF and SFOF.

d. Spacecraft Propulsion and Stabilization Control Model

A systems control model will be constructed using the structural subsystem dynamic model as the basic frame and installing all subsystem engineering models after their respective development tests. Dummy solar panels with the correct mass and center of gravity are employed in lieu of actual solar cells. The inertial guidance sensors, gyro reference assembly, and the other electronic equipment in the spacecraft control loop are employed. A live retropropulsion motor and monopropellant propulsion subsystem complete the model, including liquid thrust vector control and jet vane actuation.

The tests planned for this model will assess the capability of the autopilot system during the operation of retropropulsion and midcourse engines. One of the test objectives is to verify that the dynamic properties of the spacecraft structure will not degrade the control performance. This test is specified as a required design verification in the Voyager mission specification. The test can be conducted in several ways depending on JPL's direction. To permit a realistic test, the spacecraft will require a suspension system allowing three degress of angular freedom and a soft translational support system all enclosed in a thermal-vacuum facility. The detailed objectives of such tests and the corresponding facility design requirements will be a subject for further study during Phase IB.

e. Proof Test Model

The proof test spacecraft is a fully, flightready prototype which is released to manufacturing drawings, assembled and tested in accordance with approved procedures. It is subjected to design verification tests to environmental levels in excess of those predicted for actual flight. The following tests will be accomplished on the PTM: (see subsection 6 for details).

- Mission Sequencing. Sequence spacecraft through all possible flight operating modes in a compressed time scale with detailed monitoring and analysis of spacecraft behavior.

- Parameter Variation Test. Selected functions will be varied from their nominal values to determine spacecraft behavior under these conditions.

- Failure Mode Test. Investigation of the effects of selected failures that cannot be easily analyzed otherwise. Redundant circuits that cannot be tested during the normal mission sequence test will be tested.

- Free Mode Test. Disconnect spacecraft from all tests equipment, power from on-board batteries and solar panels, and test using radio command link through a limited mission sequence.

- Simulated Midcourse and Retro-Interaction Test. Verify stabilization and control system capable of maintaining and controlling the spacecraft attitude during midcourse propulsion and that the dynamic properties of the spacecraft structure do not degrade autopilot performance.

- Magnetic Testing. Magnetometer mappings to determine the perm and current fields of the spacecraft.

- Ordnance Test. Demonstrate ability to fire actual ordnance within protective cannisters to protect the spacecraft and demonstrate that mechanical devices actually operate.

- Vibration Testing. Demonstrate ability of the spacecraft to operate satisfactorily during and after exposure to vibration levels greater than those expected during the boost phase of flight.

- Space Simulation Test. Expose spacecraft to thermal-vacuum profile simulating the mission environment.

- Shock. Expose spacecraft to the shock loads encountered during shroud and spacecraft separation.

- Acceleration. Expose spacecraft to the acceleration profile simulating the boost and retropropulsion environment.

- Acoustic. Demonstrate ability of the spacecraft to operate satisfactorily during and after exposure to acoustic levels greater than those expected during the boost phase of the flight.

The PTM is used for compatibility tests at the contractors facility, at off-site locations and at the launch site.

- Spacecraft - Mechanical OSE Compatibility. The ability of the MOSE to provide the correct support will be demonstrated. The structural subsystem model is initially used for these compatibility tests with the first article of each item of mechanical support equipment. Later the OSE is checked with the proof test model prior to use with the flight spacecraft.

- Spacecraft - Launch Vehicle System Compatibility. The spacecraft will be mated and tested with the Centaur stage, Centaur adapter, and nose fairing to establish compatibility between the spacecraft and the booster. The PTM model will be used for this test initially at the booster contractors facility and later at AFETR.

- Planet Vehicle - Launch Complex Equipment. The PTM will be used with a flight capsule and mated to the launch vehicle at AFETR on the launch pad for early evaluation of the Voyager OFSC compatibility to the launch vehicle system.

- Spacecraft - Subsystem Mechanical Compatibility. The spacecraft performance and mechanical compatibility will be demonstrated using the PTM and will include deployment of all booms and appendages.

- Spacecraft - Experiment. The PTM will contain the flight experiments. A major test effort is devoted to this experiment interface compatibility evaluation including thermal, electrical, and mechanical operational tests.

- Spacecraft - Capsule Compatibility. The PTM is used to test the capsule interface initially at the contractor's facility with a fully simulated GFE supplied capsule and later at AFETR using a flight capsule. Complete electrical and mechanical tests will be conducted under simulated space environments.

e. Reliability Life Test Spacecraft Model

The reliability spacecraft model is the same configuration as the qualified spacecraft and is subjected to repetitive test cycles in a simulated hard vacuum mission environment, during which time tests are performed in a manner paralleling actual mission use. The results of these tests will be used to evaluate compliance with reliability requirements and will aid in the establishment of reliability confidence levels.

4. SUBSYSTEM DEVELOPMENT

This section discusses the spacecraft subsystem development as a portion of the over-all Voyager development program for both Phase IB and II tasks for the 1969 test flight and 1971 Voyager mission. Many of the subsystems are developed early for the 1971 mission and are flight tested during the 1969 launch. The major exception is the structural subsystem, although design commonality of electronic equipment mounting panels is retained. The retropropulsion subsystem and mission experiments are not required for the 1969 flight, and thus have a more leisurely development schedule.

The subsystems discussed are grouped as follows:

- Structural subsystem, including all pyrotechniques and separation
- Thermal control including louvers and insulation
- Midcourse propulsion subsystem and retropropulsion motor and controls
- Stabilization and control subsystem, including optical sensors, gyros, attitude control, and equipment for angular orientation and maneuvers
- Central sequencing and command subsystem

- Communication and data handling subsystem
- Power, including solar array, batteries, and power conditioning and control
- Planet-oriented package and body-mounted science payload, including mechanical integration equipment for mounting all experiments
- Electrical distribution subsystem of spacecraft, including cabling and junction boxes

Certain of the analysis and design activities are common to all subsystems, such as reliability analysis, maintainability, design review activities, specifications, planning, and reporting. Estimates of the reliability of the designs will appear as a part of the mathematical model constructed for each subsystem. Reliability goals and objectives will be realigned as a result of the reliability prediction. This continuing analysis reflects the level of design for which the estimates are made and will be updated as the design is refined and the test data is made available.

Failure mode, effect, and criticality analysis (FMEA) will consider every component of each subsystem, show for each component its mode of failure and the effects of each failure mode on the subsystem, spacecraft, and mission; permit the determination of critical items; and rank the components in order of criticality. This procedure has been effectively applied during Phase IA. During Phase IB, work on each subsystem will be updated. An FMEA will be done for each design change following configuration selection and design release. As alternate design methods evolve, tradeoffs are made considering the interactions of weight, availability, state of the art, cost, and reliability. Quantitative and qualitative analyses are conducted for effective results. To prevent launch delays, or even more serious mishaps, it is vital that maintainability principles be thoroughly exercised in each subsystem design.

Documented formal design reviews will be conducted in accordance with the program milestone schedule as a comprehensive evaluation of all pertinent aspects of the design, that is, relaibility, performance, value engineering, weight, manufacturing and tooling, human factors maintainability, test operations, safety, and quality assurance. Two such reviews will be conducted during the Voyager Phase IB effort, one at the end of 12 weeks following issuance of all spacecraft requirements and specifications and the second during the sixth month.

Phase II test planning is conducted by each subsystem engineer as a part of the integrated test plan covering the development test activities. Periodic progress reports will include design and development activities at the time intervals required by JPL. The results of progressive test activities will be included for each subsystem.

4.1 Structural Subsystem

4.1.1 Summary

The structural subsystem consists of the basic structure, equipment panels, engine mount structures, tank support structure, deployment devices, solar panels, support brackets, and the separation equipment. The Douglas Aircraft Company has been selected as a major subcontractor to provide this subsystem supporting both phases of the Voyager program. The scope of subsystem work includes design; strength, dynamic, and separation analyses; mass properties and reliability analyses; testing, and development liaison. Figure 5-8 is a development chart for the structural subsystem.

The proposed structural design will not involve any new development programs. The development anticipated is easily resolved through normal processes. Representative of design challenges are the evaluation of tank support to accomplish minimum restraint and structural integrity to satisfy the dynamic loading, and structural designs to maintain alignment between critically interrelated spacecraft components.

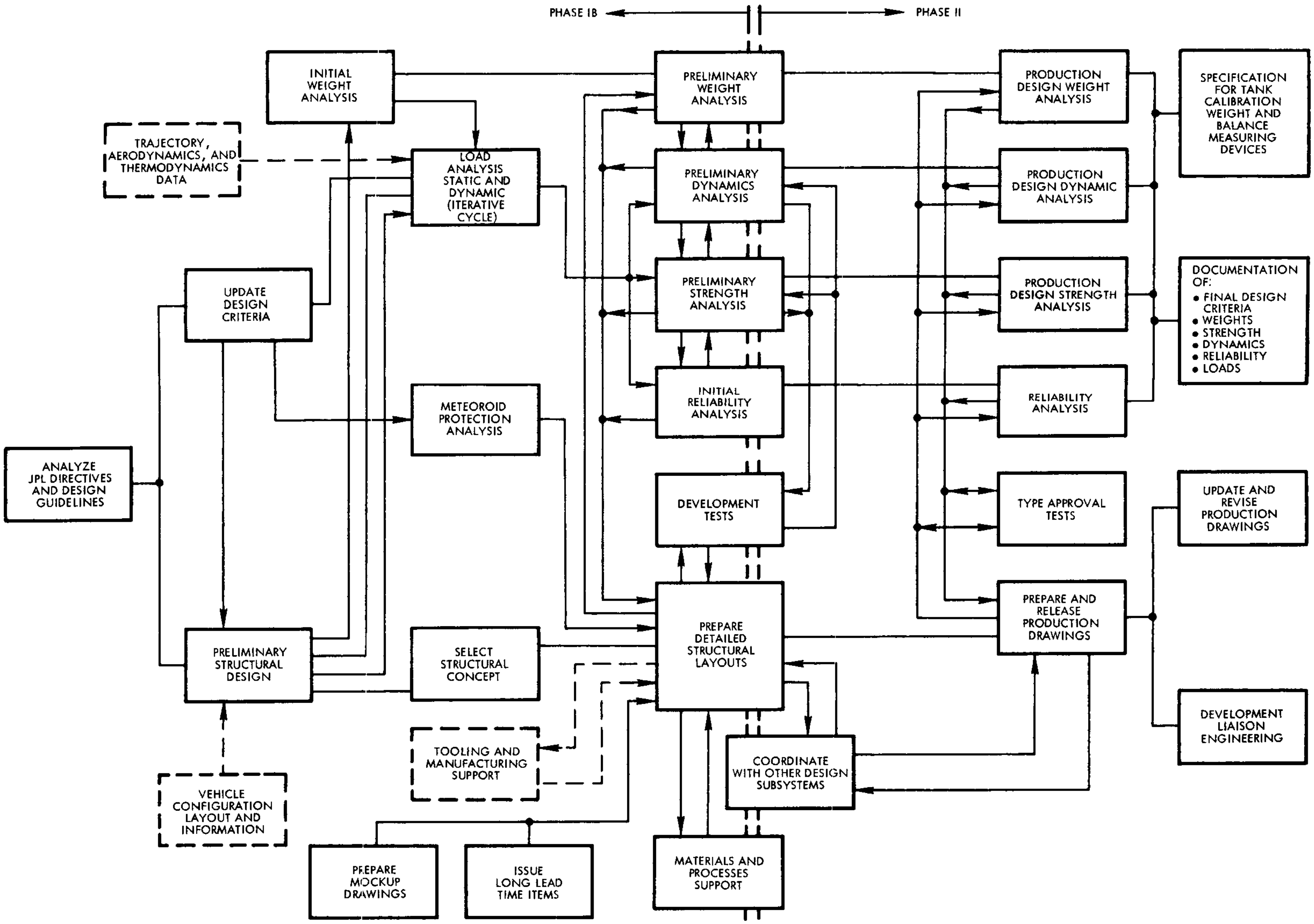


Figure 5-8. Structures Development Flow Chart

The separation system includes separation at the upper and lower field joints of the spacecraft. The components consist of mechanical attach-release devices actuated by electroexplosive devices (EED), which are actuated by a signal and power supplied by the Centaur stage for the lower separation and by the spacecraft for the capsule container shield separation at the upper field joint. It is anticipated that standard mechanical items can be used for spacecraft separation. The electro-explosive devices are Apollo standard initiators, except for ensuring that the shield circuit is completed before contact is made with the bridge pins, which may require changes to the standard qualified item. However, an external contact mounted on the electrical connector may be developed. Analysis of the detailed solution will be conducted during Phase IB.

Anticipated problems are those of tolerances and fit between the two matching surfaces, indicating a requirement for two interface plates to be produced for use by the Centaur, the capsule contractor, and the spacecraft builder. The separation subsystem will be verified by an operating test mockup of the spacecraft.

During Phase IB, both 1969 and 1971 structural designs will be pursued, the conceptual designs will be established in 3 months followed by preliminary design layouts. The 1969 test and 1971 flight spacecraft configuration models will be fabricated followed by fundamental specifications. A minimum of testing will be required in the area of design information tests used for selecting materials and critical processes. A design freeze is required for the 1969 flight at the end of Phase IB to accommodate the schedule and to permit early attention to the structural model design and tests early in Phase II.

4.1.2 Subsystem Analysis Tasks

The tasks performed during the development of the structural subsystem include an analysis of the structural design load criteria developed during the spacecraft systems analysis effort to define the specific critical design loading and deflection conditions 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.

a. Design Loads Analysis

The design loads which control the detailed design and layout of the spacecraft structure consist of the combined static and dynamic loads which are used for strength and deflection analyses. Dynamic loads are used to determine the response of critical structural elements and provide inputs for defining acceptable deflection, buckling loads, and fatigue limits. The basic static and dynamic analyses required to identify the spacecraft structural loads and corresponding structural criteria for all phases of the mission are performed as part of the spacecraft design effort (see Section V 3.2). The mission profile will be analyzed and critical loading conditions identified. The results of these studies are used to define loading intensities and dynamic environments for use in the detailed design and layout of specific elements of the structural subsystems. Since many or all of the parameters used in this analysis vary during spacecraft and mission development, design loads will be iterated as required. The resulting outputs are used for final strength and dynamic analyses of structural members.

b. Strength Analysis

The continuing strength analysis conducted during both Phase IB and II require input data including static and dynamic loads, structural design layouts and details, temperatures, deflection limitations, and weight constraints. Primary and secondary structural members will then be analyzed to determine optimum strength-to-weight designs, discontinuity stresses, rigidity, deflections, and margins of safety.

Standard analysis techniques will be used to determine these characteristics for much of the structure. Where more detailed evaluation is required, and specifically for the calculation of influence coefficients, the redundant force analysis method will be employed.

Strength analysis is used to identify items requiring development tests which supplement analytical techniques. Such test data will be correlated with analysis and design. The strength analysis also provide data for vehicle design and for such analyses as mass properties, dynamics and reliability.

c. Meteoroid Protection Analysis

The meteoroid protection required to meet a realistic mission reliability 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.

d. Dynamic Analysis

The spacecraft environment may be summarized as three sources of dynamic loading: 1) ignition shock; engine mechanical vibration; aerodynamic and maneuver loads, and shroud jettison shock transmitted through the launch vehicle structure; 2) liftoff and maximum dynamic pressure external noise; and 3) shock and vibration generated by the spacecraft (separation and retrothrust). Standard analysis, Atlas, Thor, and Saturn flight data, and structural data from JPL may be used to predict inputs to the spacecraft for the first two sources of dynamic loading. The third source, plus shroud jettison, will require special study.

During Phase IB, an analytical model of the spacecraft structure for the 1969 preliminary design will be prepared similar to the model used in the Phase IA study for the 1971 preliminary design. When spacecraft structural design is sufficiently detailed, spring constants and weights will be calculated and inserted in the models, and modal response will be calculated via a computer program. Frequencies, mode shapes, and (after insertion of input loads) deflections and accelerations will be obtained for the 1969 and 1971 preliminary designs. As JPL has the responsibility for over-all dynamic criteria for the flight, the spacecraft loads generated by TRW will be coordinated with JPL so that combined spacecraft and launch vehicle modal response may obtained and the Voyager structural design criteria completed. The resulting dynamic loading criteria is also applicable to design restraints for the MOSE design.

Critical substructure is then analyzed for dynamic response. Deflections will be kept within safe limits, dynamic buckling prevented, and fatigue stress loads generated for use in strength analysis. Flight subsystem coordination is required for critical items involving weight, stiffness, and dynamic loads. Structural development tests for these items will be planned. Specifications for the procedures and load levels will be prepared for these dynamic tests during Phase IB.

Acoustic noise levels at the spacecraft will be relatively low, 140 to 142 db over-all. It has been Douglas/MSSD experience that at these levels only thin or large sheet panels are susceptible to acoustically induced damage; therefore, only thermal louver panel acoustic tests are contemplated. Test reports will be prepared, both to document the tests and to aid spacecraft design. Douglas tests will be coordinated with TRW subsystem development tests.

To confirm the predicted dynamic environments and spacecraft responses, as well as provide a record for failure analysis purposes, a flight dynamics measurement plan for the 1969 mission will be prepared. Some of the measurements undoubtedly will not be repeated in 1971, while others may be unique for the 1971 mission. The plan will also include the types of high response telemetry which is required of the spacecraft on the Centaur stage. FM/FM telemetry is not currently planned for the spacecraft and it may be more convenient to transmit data through the Centaur stage up to stage separation.

A Phase II task will be to determine predicted flight dynamic environmental loads for the 1973 mission and to refine, as needed, those for the 1969 and 1971 missions. The analytical models for the 1969 and 1971 structure will be updated for the production designs as soon as they are sufficiently detailed. Special dynamic analyses as needed will be performed including response of the spacecraft to shroud jettison and spacecraft separation. Updated dynamic characteristics and loads will be coordinated with JPL for final design and possible changes to the structural design criteria.

Type approval dynamic tests will be planned and requirements developed. Shock test will be performed to simulate the most critical conditions during Phase II.

During Phase II the 1969 launch flight data will be analyzed and compared with predicted values for the 1971 mission.

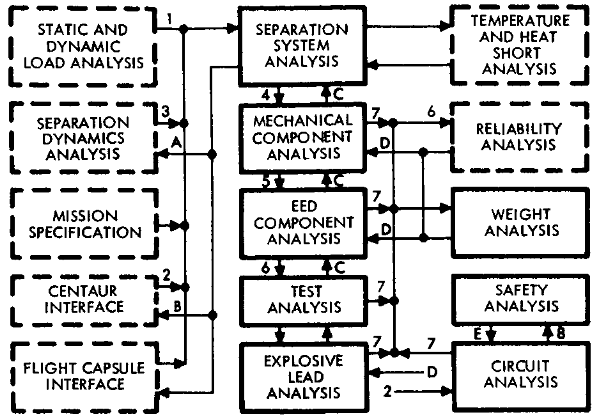
e. Separation Analysis

Separation analysis will incorporate the separation rate and interface requirements to establish component equipment requirements and criteria. Figure 5-9 shows the interrelationships of the separation analysis.

f. Weights Analysis

Weights, centers of gravity, moments of inertia, mass distributions, material breakdowns for costing purposes, and time histories will be generated commensurate with final design details. 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 attachment, and material selection. Parametric weight data will be generated to provide a basis for system sizing and mission definition. Weight optimization studies, such as determining if lightening holes can be cut out of certain frames, or if



INPUTS

1. WORST CASE LOADS ON INTERFACE
2. DIMENSION AND CIRCUIT AT INTERFACE
3. RATE OF SEPARATION
4. TYPE OF MECHANICAL RELEASE MECHANISM
5. EED SYSTEM CRITERIA
6. TEST REQUIREMENTS
7. RELIABILITY AND WEIGHT INFORMATION
8. CIRCUIT ANALYSIS
9. HEAT PATH INFORMATION

OUTPUTS

A TIME FOR STAGE RELEASE
B DIMENSIONS AND CIRCUIT AT INTERFACE
C COMPONENT CRITERIA
D RELIABILITY AND WEIGHT ANALYSIS
E SAFETY ANALYSIS
F HEAT SHORT AND TEMPERATURE INFORMATION

Figure 5-9. Separation Analysis Task Interrelationships

support structure can be scalloped, will be initiated to insure an optimum weight system.

As parts are manufactured during Phase II, the mass properties of the components, assemblies, and the complete vehicle will be physically measured to verify the computer program. The mass characteristics of the vehicle will be determined prior to launch.

4.1.3 Design

a. Structural Design

The approach to structural design is one of evolving structural configurations sufficiently early for the 1971 mission to permit direct application wherever possible on the 1969 test flight. Phase IB is devoted to configuration design of both flight configurations with this common design in mind. The subject configurations of the Phase IA study have shown that the equipment panels for the spacecraft can be identical. Six panels are employed for the 1971 spacecraft (two blank panels and four equipment panels); four panels of which are used for the 1969 flight.

Of the four equipment panels, three are devoted to subsystem equipment common to both flights, the fourth panel is experiment-peculiar electronic equipment and thus is uncommon in that little experiment equipment is required for 1969. Phase IB will result in the two structural configurations and will employ separate configuration models to derive the structural designs. The same development team will conduct these preliminary design activities to assure common design approach between spacecraft. Complete layouts of the structural subsystem will be completed during Phase IB in sufficient detail to permit structural model fabrication to proceed early in Phase II for the 1969 test subsystem. The equipment panels and structural portions of the propulsion system are therefore developed early for the 1969 flight and directly applicable to the 1971 configuration.

The structural members supporting equipment panels, solar panels, antennas, and the propulsion subsystem employ the same design techniques for both spacecraft. Because of variations in launch vehicle payload capability dynamic environment, and vehicle geometry, the general design configuration is sufficiently different to require separate development. Little advantage will be obtained for the 1971 spacecraft as a result of the earlier 1969 configuration for these structural members other than verification of analysis techniques and ground test program evaluation. Some training and procedures development can be realized.

The common electronic equipment panels employ a well-established design approach using honeycomb panel structure with standardized equipment mounting rails. The rails provide additional panel stiffness and establish a common mounting for all standardized modules for ease of removal and accessibility for test. The quarter-scale mockup constructed during Phase IA will be used to establish a full-scale configuration model during Phase IB, permitting further definition of equipment arrangement, plumbing, routing, and cabling.

The interface design between the spacecraft and Centaur adapter and nose fairing for the 1969 spacecraft and additionally between the spacecraft and capsule for the 1971 mission will be defined and coordinated with JPL and other responsible contractors. As the four panel spacecraft configuration (1969) results in a different interface, a spacecraft adapter is required between the Centaur and spacecraft bus. The 1969 spacecraft adapter will employ sheet stringer construction design to adequately redistribute the loads (six points on Centaur to four points on the spacecraft).

The Phase II design activity consists of detailed parts design and preparation and release of manufacturing drawings. The production phase of planning, tooling, and manufacturing will be contained in the manufacturing plan submitted in Phase IB. Configuration control is initiated after the subsystem baseline design review.

b. Separation Design

Two separation functions are required for 1971 spacecraft while only one is required for the 1969 test flight. The separation analysis will result in alternate concepts. Layouts of these candidate concepts will lead toward concept selection. The configuration selected will be considered with the interfacing contractor and interface design. The same separation technique will be used for both interfaces and employed on the 1969 test flight. After selecting the general methods of separation a survey of standard mechanical components will be made, and one type will be selected.

The mission specification clearly calls out the Apollo standard initiator, except for the ground constraint. Methods for meeting this requirement must be devised. Once approach is to rivet a leaf spring into a groove in the receptable end of the initiator. A matching short, gold-plated, alignment boss or pin could be built on the connector. These two parts would mate and connect before the bridge pins connect. This approach and others must be defined, analyzed, and one selected.

The selected design will result in detail development test part drawings of the components during Phase II. Separation model drawings of the separation subsystem will be made. These drawings must be of sufficient detail that a test model of the separation subsystem and immediate spacecraft/launch vehicle interface can be built. The model will retain the proper mass center of gravity properties so that meaningful separation components fit and separation-proof tests can be performed. A complete functional design specification for the subsystem and each of the components will result at the end of Phase IB.

4.1.4 Development Test

Test of the structural subsystem utilizes two models each for the 1969 and 1971 spacecraft. The models are identical and used for concurrent static and dynamic tests during development and subsystem type approval tests.

A separation model is employed to verify the separation design. The flight separation hardware is employed in each of the two separation interfaces for the 1971 designs, while the 1969 separation hardware is employed with its model.

Individual structural components will be statically tested prior to subsystem structural tests. Table 5-1 shows the development tests prepared for Phase II. Table 5-2 shows the type approval tests.

a. Static Structural Model

The static structural model consists of prototype flight hardware with simulated mass and center of gravity component equipment loading the panels in place of flight electronics.

Table 5-1. Phase II Development Test Matrix

Test Item	Parameters Measured	Test Categories (Explosure Level)				
		Temperature (^{o}F)	Vacuum (torr)	Dynamic Load	Static Load	Other
Main Body						
Structural Subsystem						
Candidate bus panels	Strain, deflections, temperature, thermal conductivity, and specific heat	-100 to + 300	10^{-6}	not applicable	limit load and ultimate load	meteorite bombardment 28,000 ft/sec
Equipment support attachment	Dynamic response, force, strain, and deflections	-100 to + 300	not applicable	sinusoidal vibration	limit load and ultimate load	not applicable
Candidate solar panel designs	Dynamic response, strain, deflections, and force	not applicable	not applicable	accoustic vibration	limit load and ultimate load	not applicable
Structural joints and splices	Strain, force, and deflections	not applicable	not applicable	not applicable	limit load and ultimate load	not applicable
Thermal block joints	Strain, force, and deflections	not applicable	not applicable	not applicable	limit load and ultimate load	not applicable
Frangible nut separation Subsystem	Temperature, dynamic response, outgassing, current and voltage wave frames and response, and trigger voltage	-100 to + 300	10^{-6}	random vibration		electromagnetic
Omni antenna deploy mechanism	Temperature, dynamic response, and heat transfer	-100 to + 300	not applicable	sinusoidal vibration	not applicable	functional
Magnetometer boom and deploy mechanism	Temperature, dynamic response, heat transfer, strain, and deflections	-100 to + 300	10^{-8}	sinusoidal vibration	not applicable	functional
Solar panel deployment mechanism (1969 only)	Temperature, dynamic response, and heat transfer	-100 to + 300	not applicable	sinusoidal vibration	not applicable	functional

Table 5-2. Type Approval Tests

	Pre Exposure Magnetic Properties	Humidity Test	Salt Spray	Random Vibrations	Sine Vibration	Shock	Thermal Cycling	Vacuum	Temperature	Acceleration	Functional	Static Load	Electrical Bonding	Spring Rates
Structural model											x	x	x	x
Dynamic model				x	x						x		x	x
Separation model											x		x	
Spacecraft Centuar separation joint				x	x				x			x	x	
Engine thrust structure (retropropulsion)				x	x				x			x	x	x
Propellant tank mounting brackets				x	x	x						x		x
Omni antenna deployment mechanism				x	x				x		x			
Antenna brackets and deploy mechanism				x	x	x			x			x		x
Science package mounting brackets and deploy mechanism				x	x	x			x			x		x
Magnetometer boom and deploy mechanism				x	x	x			x		x	x		x
Detonator, electric	x	x	x	x	x			x	x	x	x			
Harness, deploy mechanism subsystem		x	x	x	x			x	x	x	x			
Harness, separation subsystem		x	x					x	x		x			
Flangible nut separation subsystem	x	x	x	x	x		x	x	x	x	x			
Hinge bearings (louvers)				x	x			x	x			x		
Solar panel deploy mechanism (1969 spacecraft only)				x	x				x		x			
Hinge bearings (solar panel) (1969 spacecraft only)				x	x			x			x	x		

The structural test model will be mounted to a test fixture and subjected to static flight load tests to verify the structural characteristics of the system. These tests will be an extension of the structural tests performed on the individual components. The tests will be made to accomplish the following:

- Check hardpoints for static load plus simulated acceleration load
- Simulate acceleration and static loads for critical components
- Simulate lateral loads expected from vibration and handling on items such as lander mounting points, rocket motor mounting points, structural panels, antenna, and critical components.

The structural model will be used during these tests to accomplish type approval of the subsystem.

b. Dynamic Model

The dynamic model is the same configuration as the static model used above. The static and dynamic models will also be used for testing, mechanical OSE compatibility testing, and continued verification testing.

Dynamic tests constitute one of the critical structural tests for the following reasons:

- Axial load factors are relatively small
- Structure is extremely lightweight, redundant, and complex
- There are a number of concentrated loads

For these reasons, dynamically induced stresses experienced during boost phase and stage separations will design and/or contribute significantly to design loads for most structural elements.

Structural dynamic type approval tests will be performed to provide data for extensive structural analyses to establish the spacecraft structural integrity for flight dynamic loads, to eliminate design weaknesses, to revise and/or confirm the spacecraft dynamic model, and to evaluate methods of support for the landers, solar panels, and o other appendages.

The primary objectives of the vibration test of the model are:

- Evaluate and define the structural dynamic properties of the spacecraft
- Determine vibration levels experienced by spacecraft components, high gain antenna, solar panels, and magnetometer boom as a result of their mounting arrangement and positioning.
- Determine qualification and acceptance test vibration levels of components and systems. These specifications should be based on realistic data obtained during the development tests

c. Separation Test Model

Functional operation of the separation subsystem must be valuated for the following:

- The separation of the spacecraft adapter and the launch vehicle adapter
- The separation of the sterilization cover from the spacecraft

Separation tests will be conducted on the separation test model to demonstrate the functional operation of the separation mechanism. A separation command will be programmed to the explosive devices and operation of the control circuits will be monitored. Type approval of the separation system is demonstrated during this series of tests. Component proof tests will be conducted on frangible nuts, electronic detonators, and harness systems, as shown in Table 5-2

4.1.5 Schedules

The schedules for Phase IB and II structural/mechanical subsystem development are shown in Figures 5-10 and 5-11, respectively.

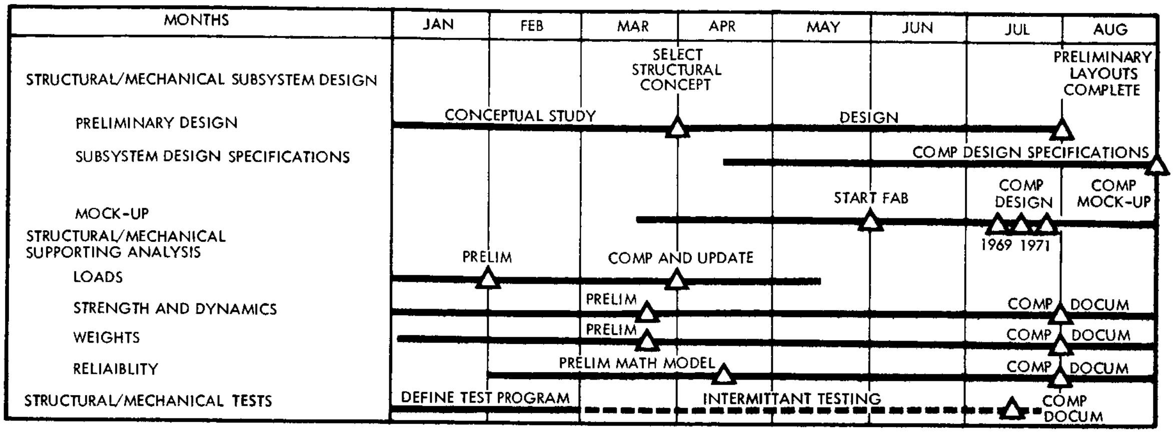


Figure 5-10. Phase IB Structural Subsystem Schedule

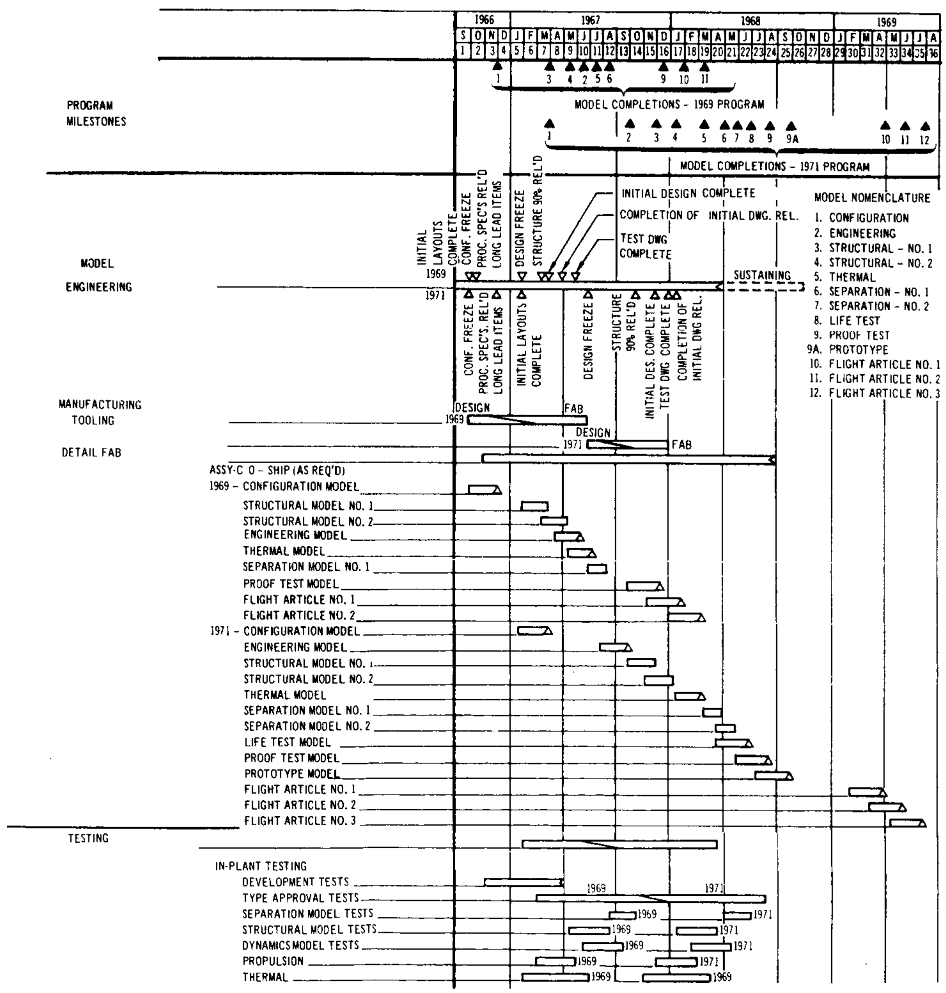


Figure 5-11. Phase II Structural Subsystem Schedule

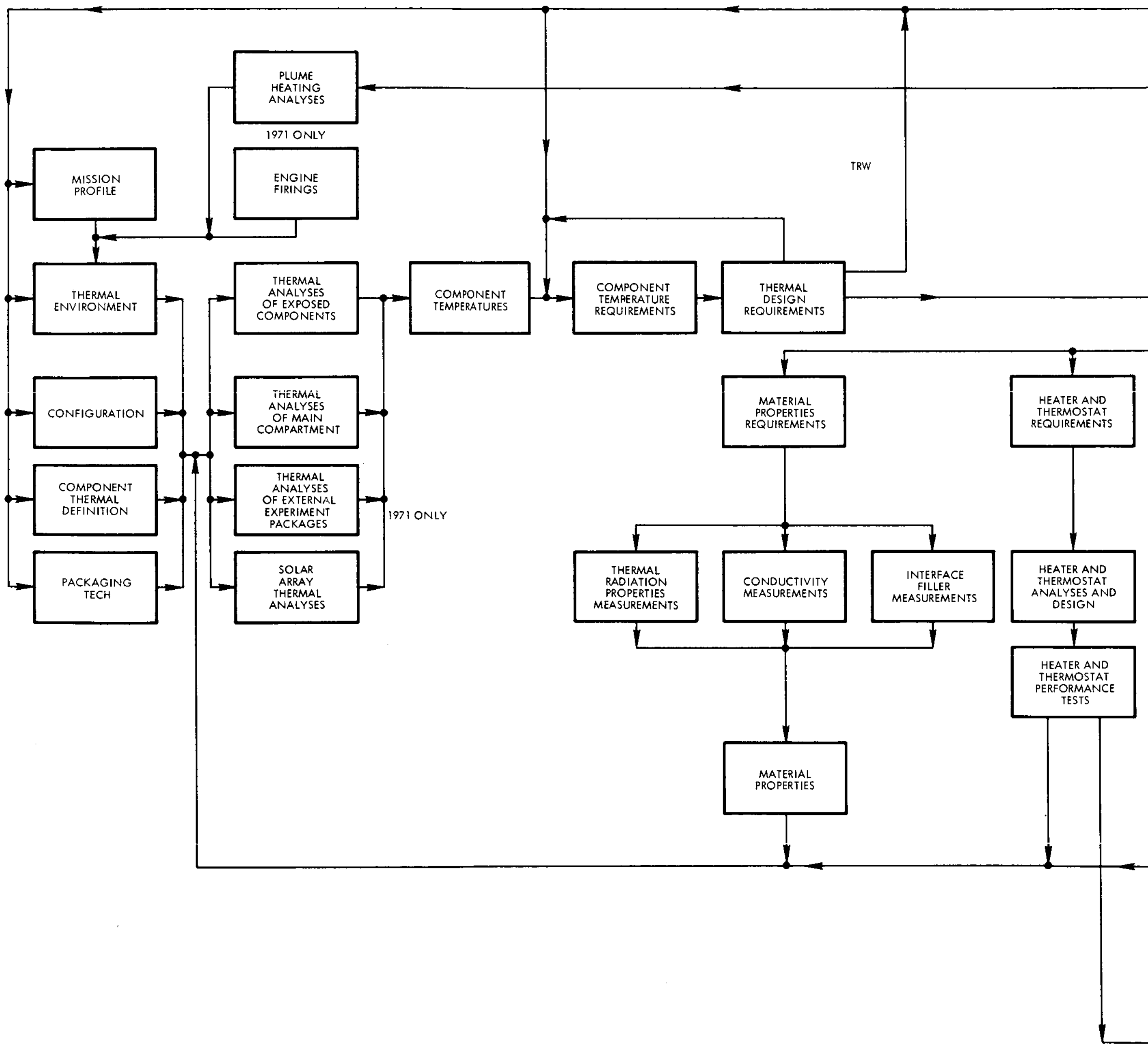
4.2 Thermal Control Subsystem

The thermal control development plan presented in the following sections is based upon the 1971 Voyager mission. Due to the differences in configuration and thermal environment between the 1969 and 1971 missions, they require separate development efforts. However, much of the information obtained during the 1969 mission development in terms of thermal control subassembly design and performance will be applicable to the 1971 mission. The manner in which the development of the thermal control system for the 1969 mission varies from that of the 1971 mission is discussed in Section 4.2.8.

The development of the thermal control system for Voyager is similar to that of the OGO, Vela, and Pioneer programs. Essentially it consists of iterative detailed thermal analyses of on-board equipment supported and verified by thermal testing. The analysis is performed utilizing the TRW thermal analyzer, shape factor, and other computer programs. It iterates upon changes in configuration, thermal environment, component information, and information obtained from thermal testing. The thermal testing is performed for a dual purpose. The initial thermal testing provides information on the elements of the thermal control system (i.e., louvers, insulation) which is used as input information to the thermal analyses. The final thermal testing is performed on engineering thermal models characteristic of the flight hardware to verify the performance of the thermal control system.

The activities planned for the design and development of the Voyager thermal control system are diagrammed in Figure 5-12. The following key areas are indicated:

a) The physical configuration, on the basis of which the thermal analyses will begin, is obtained from structural drawings and weight lists. Changes in configuration must be assessed for their influence on the thermal control system by updating the thermal analyses involved.


PLUME HEATING ANALYSES
1971 ONLY
ENGINE FIRINGS
TRW
MISSION PROFILE
THERMAL ENVIRONMENT
CONFIGURATION
COMPONENT THERMAL DEFINITION
PACKAGING TECH
THERMAL ANALYSES OF EXPOSED COMPONENTS
THERMAL ANALYSES OF MAIN COMPARTMENT
THERMAL ANALYSES OF EXTERNAL EXPERIMENT PACKAGES
SOLAR ARRAY THERMAL ANALYSES
1971 ONLY
COMPONENT TEMPERATURES
COMPONENT TEMPERATURE REQUIREMENTS
THERMAL DESIGN REQUIREMENTS
MATERIAL PROPERTIES REQUIREMENTS
HEATER AND THERMOSTAT REQUIREMENTS
THERMAL RADIATION PROPERTIES MEASUREMENTS
CONDUCTIVITY MEASUREMENTS
INTERFACE FILLER MEASUREMENTS
HEATER AND THERMOSTAT ANALYSES AND DESIGN
HEATER AND THERMOSTAT PERFORMANCE TESTS
MATERIAL PROPERTIES

(1)

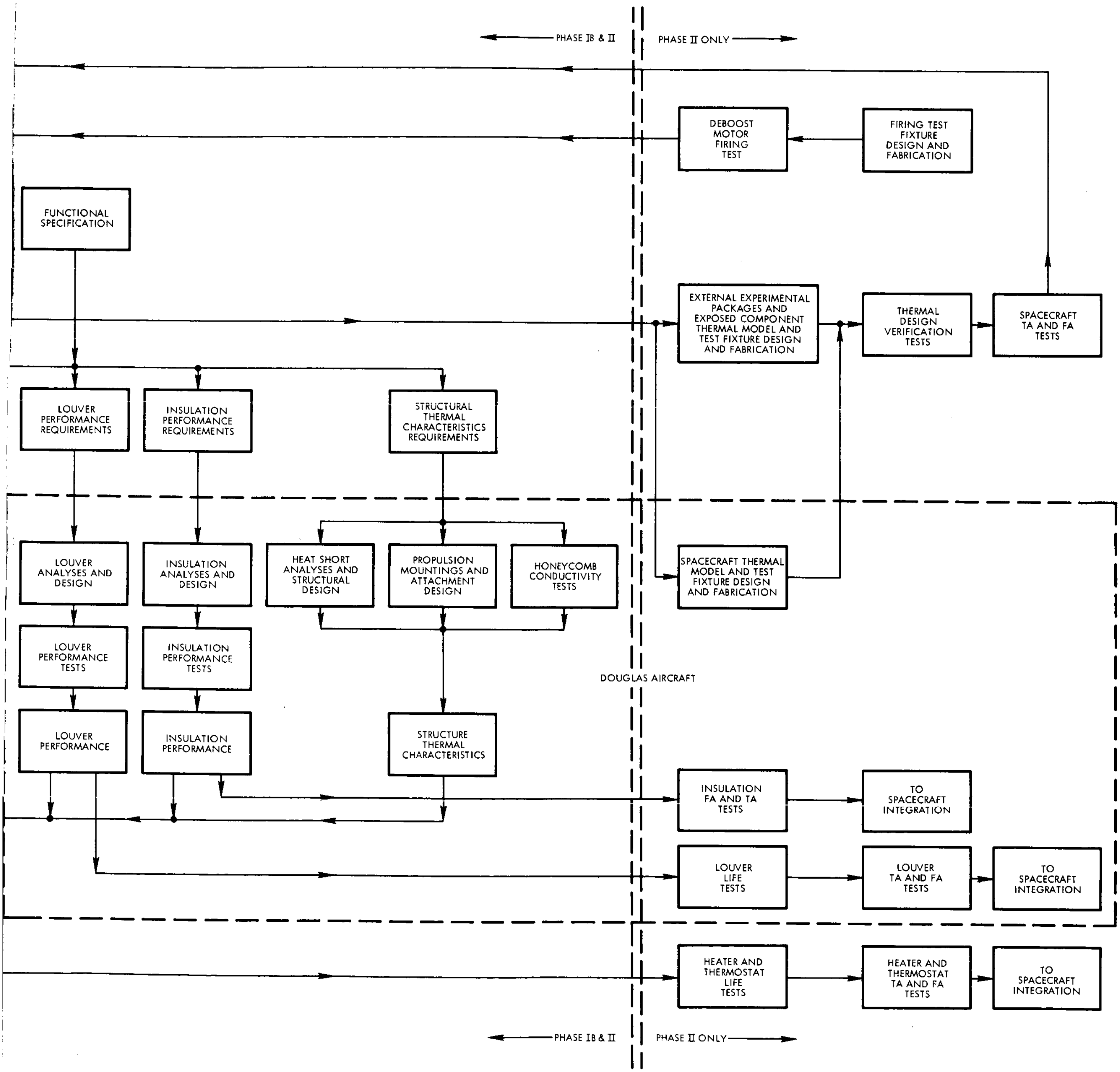


Figure 5-12. Thermal Control Subsystem Development

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b) The total thermal environment to which the spacecraft is subjected throughout the mission is determined. This environment includes on-stand heating, radiant heating from the fairing, aerodynamic heating after fairing jettison, non-nominal attitude with respect to the sun prior to orientation, varying solar intensity throughout the mission, radiative heating from the deboost motor plume, eclipse, and Martian emitted and reflected solar heating.

c) Thermal definition of the electronic components is required from all other subsystems and experimenters. This definition includes allowable temperature limits, power dissipation, duty cycle, mounting base area, and requirements for insulation blanket penetration.

d) The detailed thermal analyses will determine the coating and finish callout, component placement, and amount of active thermal control required to meet the required temperature limits.

e) The louver system, insulation, and structural characteristics analysis, design, and test provides both subsystem hardware and performance information about the hardware. This information is utilized to update the detailed thermal analyses.

f) The deboost motor firing test in Phase II will provide the information necessary to determine the heat shield and insulation required for this thermal environment (1971 mission only).

g) The thermal control design verification tests of Phase II will provide verification of the analysis and design, as well as the data necessary to trim the thermal control system.

The schedule for the design and development of the thermal control subsystem is shown in Figures 5-13. 5-14, and 5-15.

4. 2. 1 Spacecraft Thermal Analysis

a) Thermal Environment

In order to design a system which will provide adequate thermal control throughout the mission, it is necessary first to assess

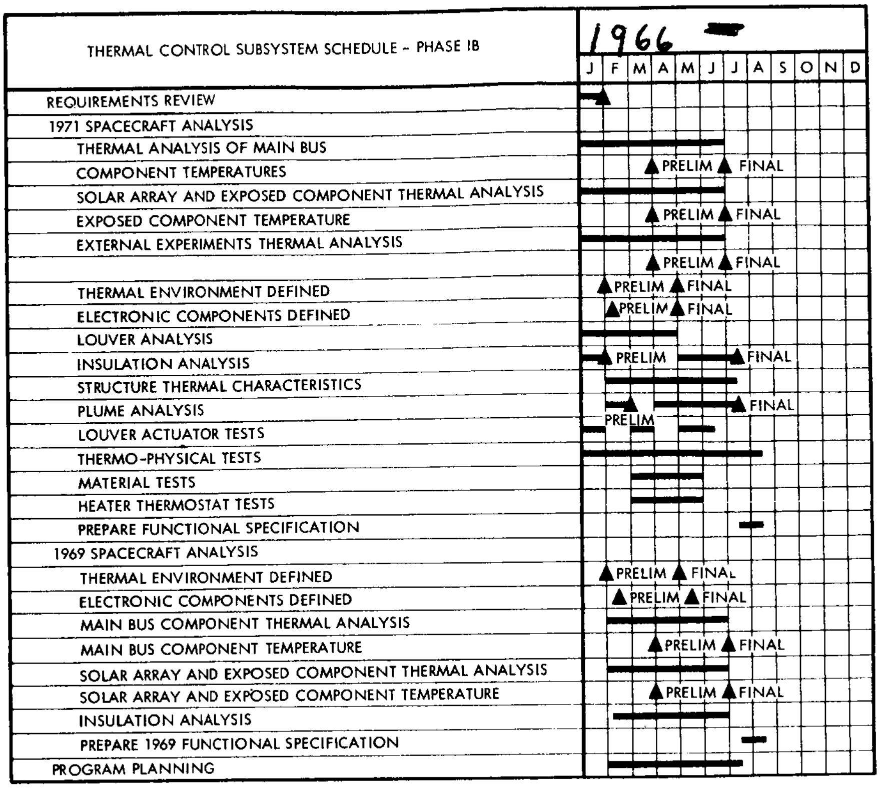

Figure 5-13. Thermal Control Subsystem Schedule, Phase IB

the thermal environment to which the spacecraft will be subjected throughout the mission. This environment includes on-stand heating, radiant heating from the fairing, aerodynamic heating after fairing jettison, non-nominal attitude with respect to the sun prior to orientation, varying solar intensity throughout the mission, radiative heating from the deboost motor plume, eclipse, and the Martian orbital environment. The magnitude of the on-stand heating will be determined from the duty cycle schedule of

the spacecraft equipment during on-stand checkout. The radiant heating from the fairing will be determined from the parametric curves of internal fairing temperatures as a function of time and fairing insulation utilized. The aerodynamic heating after fairing jettison will be determined by computation utilizing the 3σ low launch trajectory. The heating rates during the time when the spacecraft is in a non-nominal attitude with respect to

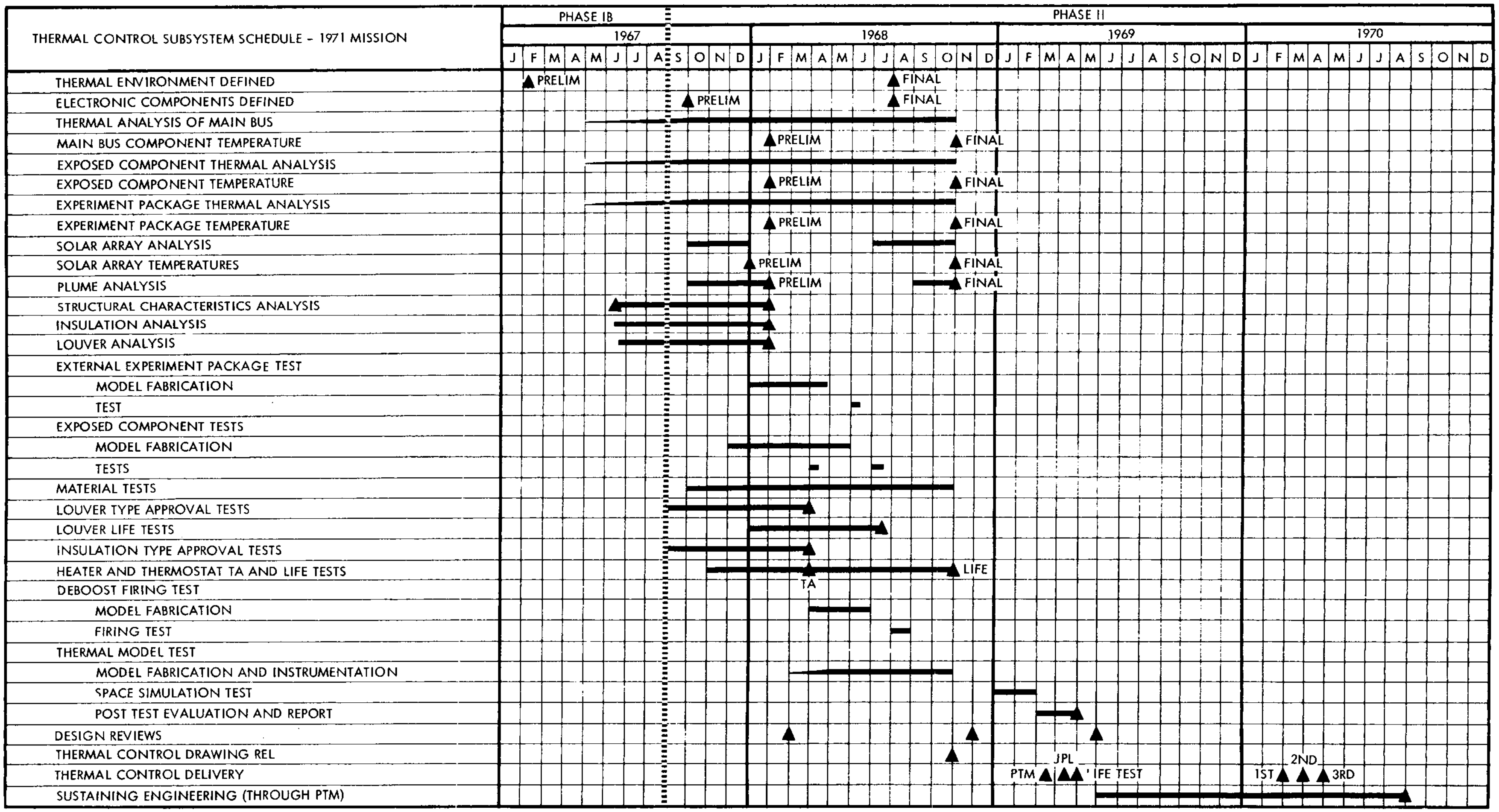

Figure 5-14. Thermal Control Subsystem Schedule, 1971 Mission

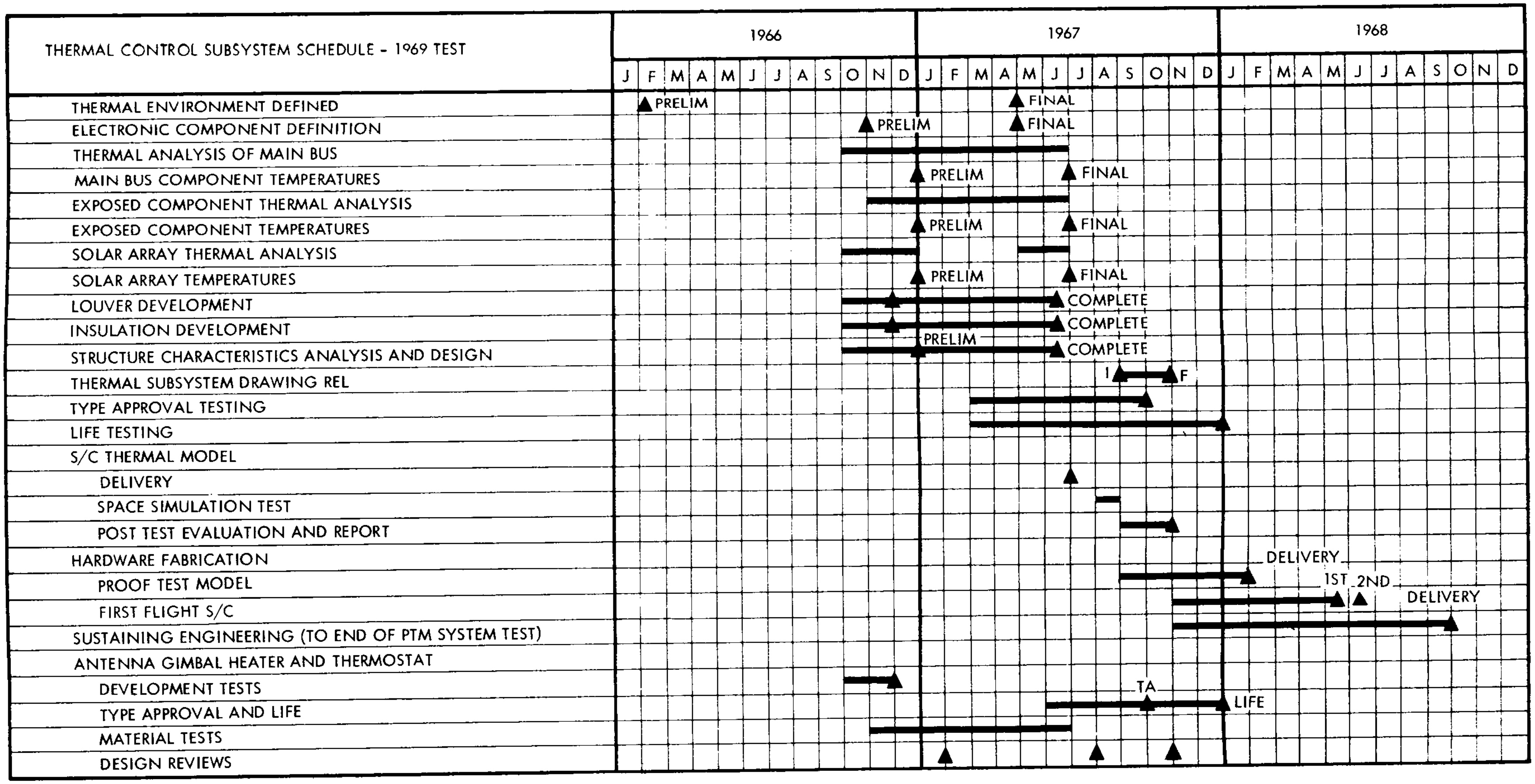

Figure 5-15. Thermal Control Subsystem Schedule, 1969 Test

the sun will be determined from the sun-look angle versus time information available from the launch trajectory analysis. The varying solar intensity throughout the mission is a straightforward calculation. The calculation of the magnitude of the radiative heat input to the spacecraft from the molten alumina particles in the deboost motor will be performed by Douglas. The lengths of the eclipses which may be experienced in the region of the earth or during the Martian orbit will be determined from the trajectory analysis. The planetary heating environment experienced by the spacecraft when it is in the proximity of earth or Mars will be computed utilizing trajectory information and the TRW planetary heating program.

These environments will be input to the detailed thermal analyses conducted for all portions of the spacecraft. These inputs and the resulting response of all elements of the spacecraft to these inputs, coupled with the internal power dissipation modes of the mission, will determine the thermal design of the spacecraft and any launch restraints if required.

b) Detailed Thermal Analyses

Detailed thermal analyses of all elements of the spacecraft will be conducted for all phases of the mission environment. This analysis will rely heavily upon the TRW thermal analyzer program, developed specifically to solve thermal problems involving any combination of the convection, conduction, or radiation modes of heat transfer. The program utilizes the electrical analogy for a lumped parameter network and offers no limit to the network size other than computer capacity.

Detailed thermal computer models will be constructed for all major thermally-controlled compartments such as the main bus and the external experiment packages. The level of detail will be such that the mounting base temperatures will be computed for each component in the compartment considering radiative and conductive heat transfer for the thermal environments in space. The effect of convective heat transfer will be considered for the thermal analysis of the on-stand operation. These analyses will determine the requirements for the amount of active thermal control louver area, the insulation effectiveness, thermal coatings and finishes, and placement of high and low power dissipation components.

Separate models will be constructed for those portions of the spacecraft which are not compartmentized but are exposed to the space environment such as the sun sensors, horizon scanners, antennas, and gimbals. Solar array temperatures will he supplied to the power subsystem to allow the choice of an optimum solar cell-cover glass-filter combination. Temperature differences on the array will be determined for all the environments of the mission and supplied to the power subsystem to allow an assessment of the problems of voltage mismatch. The analysis for the sun sensors, horizon scanners, antennas, and gimbals will determine thermal coatings and finishes, heater power, and thermostat, and insulation requirements such that the equipment is maintained within acceptable temperature limits without degradation of its operating efficiency.

Because it is necessary to calculate at least approximate operating temperatures for all the internal and external components early in the spacecraft program, the detailed thermal analysis will begin immediately in Phase IB, utilizing typical values of louver system performance, insulation effectiveness, thermal radiation properties, and interface conductances. As the program progresses and more detailed information becomes available from other analyses and test programs (i.e., the design and evaluation of the louver system) and better definition of the spacecraft components, these analyses will be refined. A final updating of the detailed thermal models will occur after the thermal design verification tests of the spacecraft.

4.2.2 Analysis of Thermal Assemblies

The thermal design analysis effort to be performed is divided into four parts: thermal analysis of the louvers, insulation, heat shorts, and the effects of the propulsion system on the spacecraft.

a. Louvers

To assist in the selection of a construction material and technique for the louvers, the effects of various materials and fabrication techniques will be analyzed in terms of over-all louver assembly performance. The design of the actuation mechanism will be analyzed to determine degradation in louver performance due to heat leak through the actuation mechanism.

b. Spacecraft Insulation

As the performance requirements and a description of the environment become more clearly defined, the total insulation requirement in terms of insulation thickness and number of reflective sheets will be upgraded. A parametric analysis will be conducted for a range of environmental and insulation characteristics.

c. Heat Shorts

Heat short analyses will be conducted to determine the effective thermal conductance for the structural members in question. Calculating heat loss through each member to optimize thermal design provides the principal analytical tool for these analyses. Included will be examination of heat shorts such as the solar array attach ring, solar array struts, lander attach area, antenna boom, mapping package boom, and attitude control lines.

d. Propulsion System

Radiant and convective heat fluxes from the nozzle and the plume during engine firing are examined, defining insulation requirements for those surfaces exposed to such heating. The effective conductance of attachment members between the structure and the engine will be calculated to estimate engine soakback heating and to select attachment members designs. At the completion of firing, after the engine components have cooled, the heat leak from the spacecraft out through the propulsion system will be predicted.

4.2.3 Functional Specifications and Program Plans

The software output of Phase IB will be the thermal control subsystem functional specification for the 1969 test spacecraft and the 1971 flight spacecraft. Thermal control subsystem inputs will be provided for the manufacturing, assembly and checkout, integrated test, and launch operations plans.

4.2.4 Development Design Fabrication

The thermal models to be used in the design verification tests of the main spacecraft bus and the exposed experiment packages will very closely simulate the flight articles. These models will consist of flight-type structures including insulation and louvers, thermal mockups of the solar array, thermal mockups of the electronic components, and both an inert and expended deboost motor. The array mockups consist 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 cells. The electronic component mockups are aluminum shells of the same dimension and weight as the component being simulated with power dissipation provided by an internal resistor. The inert deboost motor will be used for tests of conditions before firing, and the expended case for those after firing.

The model used to determine the effect of the deboost motor firing test will consist of sufficient spacecraft structure to support the deboost motor and heatshield (if analysis indicates the necessity of a heat shield) during static firing tests.

Fabrication of other development hardware will be conducted for the investigation of local thermal problems for those instances where analysis indicates the possibility of a thermal problem.

4.2.5 Development Design

After conceptual design, detailed layouts will be started, and a list of specifications written covering parts, materials, processes, tooling, and subcontract items. Test programs will be initiated to check vendors' parts and subsystem configurations. These tests will include, but not be limited to, vibration, g-loading, heat, cold, hard vacuum, and operational tests.

The final step of the development program will consist of product design devoted to packaging, thermal model fabrication and test, a final design review, and release of drawing to production fabrication.

Test procedures will be written for the final type approval test program. A final assessment of specification compliance and spacecraft interface compatability will be confirmed. Documentation will be produced to cover ground handling, grid area assembly, and checkout procedure.

4.2.6 Development Testing

Table 5-3 presents the thermal control system development and type approval test matrix.

a. Design Verification Tests

The main spacecraft bus and the external experiment packages will have their thermal design verified by space simulation testing utilizing a vacuum chamber with liquid nitrogen-filled cold walls to simulate the heat sink of space and solar simulation. These tests will be conducted for the environments which the analysis indicates to be the most severe for each package. Structural heat leaks into or out of the packages will be simulated by driving the boundary condition for the heat leak to its analytically predicted temperature. Infrared heat inputs to the packages such as would occur from the solar array will be simulated by a mocked-up solar array, and as would occur due to planetary infrared emission will be simulated by infrared heater elements. It is anticipated that additional design verification or engineering thermal model tests will be conducted in a similar manner on additional portions of the spacecraft, which further analysis indicates may pose thermal problems. Likely candidates for this category of testing are the gimbals for the antenna

Table 5-3. Thermal Control Test Matrix

Test Title	Purpose	Test Unit	Description	Test Equipment	Remarks
Exposed experiment packages design verification test	Verify thermal analysis and design	Thermal models of the exposed experiment packages	Space simulation test. Test conditions simulate all thermal environments to which the packages are sensitive	Space simulation chamber, carbon arc solar simulators, thermocouples, recorders, power supplies, radiometers	To separate tests.
Main spacecraft bus design verification test	Verify thermal analysis and design	Thermal model of the main spacecraft bus	Space simulation test. Test conditions simulate all thermal environments to which the main spacecraft bus is sensitive	Space simulation chamber, solar simulators, thermocouples, recorders, power supplies, radiometers	
Deboost motor firing tests	Evaluate heating rates from deboost motor firing and insulation qualities of heat shield and deboost motor structural assembly	Deboost motor firing test model	Static firing of deboost motor at simulated altitude. Radiant heat fluxes and temperatures monitored	Altitude simulation chamber, test unit, thermocouples, recorders, radiometers, high-speed cameras, thermistors	Choice of heat shield material will be made from heat shield insulation tests
Materials thermal radiation properties measurements	Determine thermal radiation properties of coatings and ultraviolet degradation, where applicable	Laboratory samples	Sample thermal radiation properties measured. Expose external materials to ultraviolet radiation. Measure properties before and after exposure	Heated cavity reflectometer, Gier-Dunkle integrating sphere, Beekman DK-2A modified integrating sphere, cooled radiometer total normal emittance stand, paraboloid reflectometer, UV degradation facility	Materials measured will be those considered for use on spacecraft. Advantage will be taken of data obtained from previous contracts
Interface filler conductance tests	Determine thermal conductance of interface filler material under those components where analysis indicated need	Thermal model mockups, honeycomb panel sections	Component mockup bolted to honeycomb with interface filler. Mockups heated in vacuum and temperature gradients measured	Vacuum chamber, test units, thermocouples, recorders, power supplies	It will be possible to run more than one unit at a time, thus reducing vacuum chamber time
Miscellaneous thermal tests as required by analysis	Determine the adequacy of thermal designs in which analysis indicates uncertainty	Thermal model mockups or component engineering models	Models to be tested to those environments for which analysis indicates uncertainty in adequacy of thermal design	Vacuum chambers, solar simulators, infrared heaters, power supplies, thermocouples, recorders, radiometers	Specific components to be tested will be determined by detailed thermal analysis. Likely candidates are the gimbals, sun sensors, and horizon scanners.
Heater and thermostat life tests	Determine reliability of heaters and thermostats	Heaters and thermostats	Heaters and thermostats applied to common plate in vacuum chamber, plate temperature cycled	Vacuum chamber, variable temperature plate, thermocouples, recorders, voltmeters, power supplies	
Heater and thermostat type approval tests	Type approvel	Type approval heaters and thermostats	Environmental test, vibration, temperature, thermal vacuum, shock, acceleration	Environmental test	
Thermal control subsystem type approval test	Type approval	Proof test model	Space simulation test, solar simulator intensity 20% above and 20% below realistic levels	Space simulation chamber, solar simulator, support fixture, capsule simulator	

and external experiment package, the horizon scanners, and the sun sensors. The Phase IB analyses will indicate particular problem areas for updated planning of Phase II tests.

b. Deboost Motor Firing Test (Phase II)

The deboost motor will be statically fired at simulated altitude as part of the motor qualification, allowing evaluation of the heat flux resulting from the molten alumina particles in the plume. If analysis has indicated that a heat shield is required, this test will also serve as a design verification test of the heat shield. In addition, the model will be instrumented to evaluate the magnitude of the heat soak-back by conduction from the hot rocket motor casing after firing through the structural attachment. The model will be instrumented with thermocouples and narrow angle radiometers to monitor heat fluxes and temperature distribution.

c. Material Properties Tests

Tests will be conducted on laboratory samples to determine thermal properties for those coatings for which data acquired on previous programs is not adequate. Coatings that will be exposed to solar irradiation in orbit will be exposed to ultraviolet radiation in the laboratory to determine the extent of degradation of the thermal radiation properties. The TRW ultraviolet degradation facility consists of a series of small vacuum chambers with temperature-controlled sample holders ranged around a xenon lamp to expose the samples for various periods and levels.

d. Louver Blade Tests

A series of louver blades will be subjected to a series of structural tests to determine their torsional and bending strength, as well as their ability to resist handling during fabrication and assembly.

e. Louver Actuator Mechanisms

Various candidate louver actuator mechanisms will be tested to determine their output force as a function of the temperature change and ability to withstand the vacuum environment of space. The mechanisms will be thermally cycled in a space chamber and the output rotation measured as a function of temperature.

f. Surface Finishes Tests

A series of tests will be conducted on typical surface finishes for the louvers, substantiating thermal analysis and surface degradation effects leading toward surface finish, and the material selection in the louver blades.

g. Outgassing Tests

All components of the louver system will be tested to determine their relative outgassing characteristics. Components which outgas excessively will be redesigned to eliminate or minimize the use of outgassing materials.

h. Vibration Tests

A typical louver panel will be fabricated for vibration testing. The specimen will be subjected to launch vehicle vibration environments.

i. Life-Cycle Tests

A louver assembly complete with actuation mechanism, simulated cold plate, and heat source will be fabricated and installed in the vacuum chamber for thermal performance life tests.

These tests will be monitored and the results evaluated in terms of the comparison between predicted and actual behavior of each thermal component. The design of each component in the thermal control subsystem will be modified according to the results of the test in an effort to obtain optimum performance. Descriptions of test programs, procedures and results will also be presented in final report form.

j. Test Matrix

The development and type approval tests matrices are listed in Tables 5-4 and 5-5.

Table 5-4. Thermal Control Subsystem Development Test Matrix

Test Item	Parameters Measured	Temperature Exposure Level (°F)	Vacuum Exposure Level (torr)	Test Categories Dynamic Load Exposure Level	Solar Simulation Exposure Level (w/sq cm)
PHASE IB					
Types of coatings	Solar absorptivity, thermal emissivity, outgassing, temperature, time	-100 to +300	10^{-8}	Not applicable	140
Types of insulation	Temperature, outgassing, specific heat, thermal conductivity, time	-100 to +300	10^{-8}	Not applicable	140
Candidate louver panels	Dynamic response, temperature, outgassing, specific heat, thermal conductivity, time	-100 to +300	10^{-8}	Sinusoidal vibration and shock	Not applicable
Actuator (thermal louvers)	Temperature, force, dynamic response, time	-100 to +300	10^{-8}	Random vibration shock acceleration	140
PHASE II					
Thermal louver assembly	Temperature, solar absorptivity, thermal emissivity, dynamic response	-100 to +300	10^{-8}	Random and sinusoidal vibration	140

Table 5-5. Thermal Control Subsystem Type Approval Test Matrix

	Pre Exposure Magnetic Properties	Random Vibrations	Sine Vibration	Thermal Shock	Thermal Cycling	Vacuum	Test Temperature	Acceleration	Functional	Electrical Bonding	Acoustics	Solar Absorptivity	Thermal Emissivity	Thermal Conductivity
Thermal Actuators	x	x	x		x	x	x	x	x					
Thermal Louvers		x	x		x	x	x	x	x					
Thermal Louver Subsystem(Life Test)		x	x		x	x	x	x	x	x				
Propellant Thermal Protection System						x	x							x
Impingement on optical coatings							x					x	x	

4.2.8 <u>Differences Between 1969 Mission and 1971 Mission Development</u>

Much of the development of the thermal control system for the 1969 mission will be applicable to the 1971 mission in the areas of the development of the thermal control subassemblies, equipment mounting panel and solar array substrate conductance measurements, thermal radiation property measurements, and portions of the interface filler conductance tests. However, due to the difference in configuration of the spacecraft main bus for the two missions it will be necessary to conduct two separate detailed thermal analyses for the bus. The detailed thermal analyses and space simulation tests conducted for the gimbals, antennas, sun sensors, solar array, and horizon scanners for the 1969 mission will be at least partially applicable to the 1971 mission, dependent upon the amount of change in configuration and thermal environment.

4.3 Propulsion Subsystem

The Voyager propulsion subsystem consists of a monopropellant midcourse engine and a retropropulsion solid propellant motor.

4.3.1 Midcourse Propulsion Subsystem

The design approach in Phase IA for the midcourse propulsion subsystem (MPS) was to devise the simplest system, in terms of the number and types of components and the interactions between the components and other spacecraft subsystems, consistent with the Voyager performance, duty cycle, and reliability requirements. The development program thus requires no state-of-the-art improvement in any of the components. The majority of the effort is involved in characterizing the system performance over all operating conditions and qualifying the components and the system to Voyager specifications. The development program shown in Figure 5-16 is compatible with delivery of a flight qualified system for a 1969 mission.

Design and development of the MPS is divided into two categories: 1) component development, prequalification and performance determination, and 2) system characterization and qualification. Since much of the engine system is essentially identical to flight qualified hardware, feasibility type testing in heavyweight hardware of these components is not required, and all testing can be conducted with flightweight hardware. Although considerable development history exists for the engine, the tank and expulsion device will require a new design and concomitant development effort.

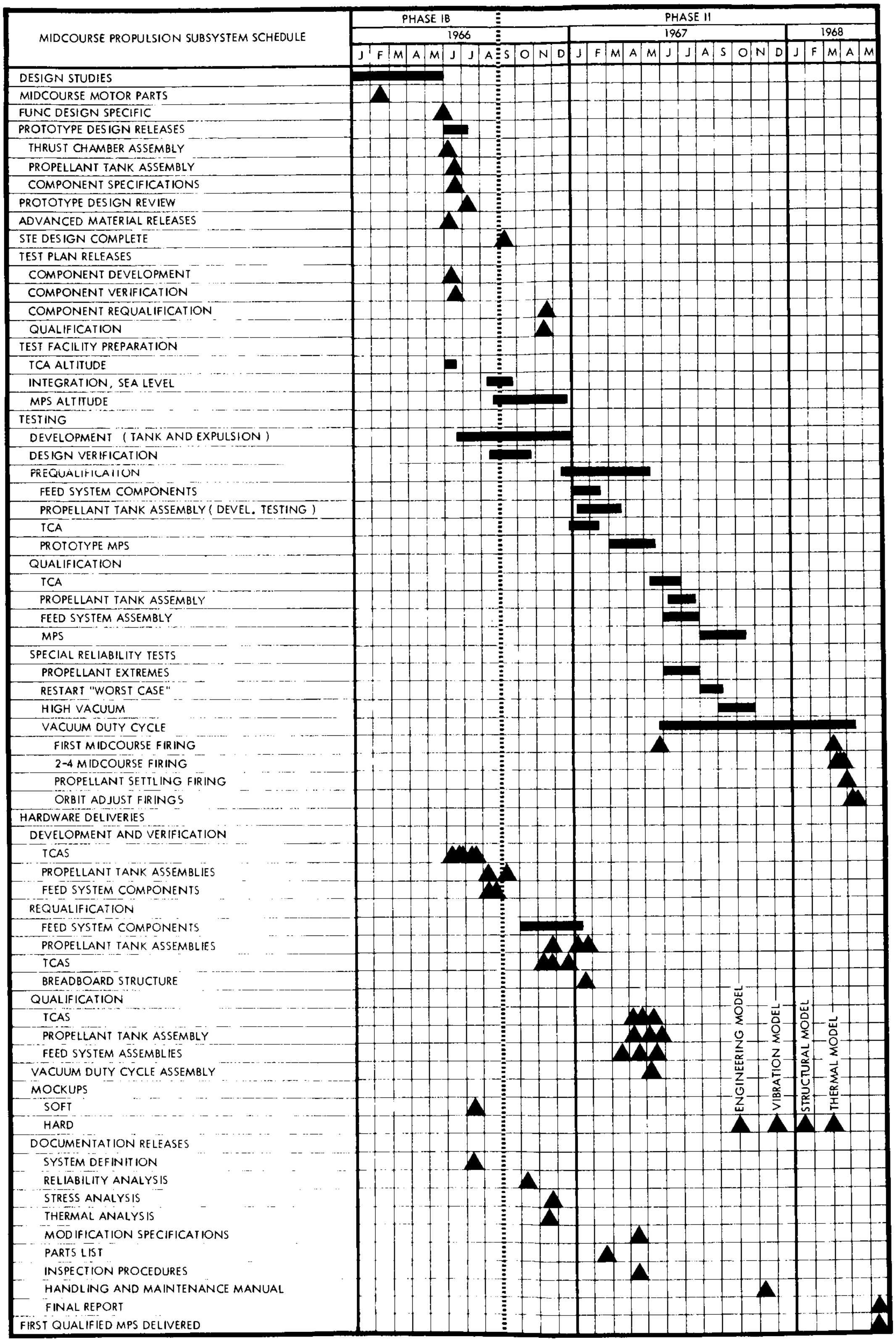

Figure 5-16. Midcourse Propulsion Subsystem Schedule

a. Analysis and Design Studies

To assist in confirming that the design of the MPS will meet all requirements, particularly in the areas of packaging, temperatures, vibration or propellant slosh mode interactions with the spacecraft vehicle, design studies of these and other problem areas will be conducted. The effects of the particular duty cycle requirements on engine integrity, heat transfer into the other Voyager vehicle structure and systems, and dynamic field interference with possible spacecraft design experiments will be examined in detail. On establishment of a prototype propulsion subsystem design configuration, a detailed analysis of the hydraulic characteristics of the MPS will be made.

During the course of the initial design verification testing, preliminary analytical studies and tests will be conducted to determine the mass properties of the MPS. Of particular importance will be data gathered on the center of gravity shift with various percentages of the full propellant load with the liquid restrained by the positive expulsion bladder. Other mass properties will be determined such as weight, center of gravity, moments of inertia, and mass distribution.

b. Design Specification

The detailed design and layout of the flight prototype MPS assembly will be completed within the first few weeks of the Phase II program, including any changes resulting from the preliminary testing during the verification phase of the test program. During this period, specifications will be prepared and a hard mockup constructed to ensure interface compatibility between the MPS and the Voyager vehicle

c. Component Verification and Qualification Testing

A series of prequalification component verification tests will be carried out on each of the components to verify their acceptability in the MPS prior to the initiation of systems tests. These tests will be conducted against specifications generated to meet the needs of the Voyager vehicle system. In this phase of the program all components will be subjected to the test shown in the text matrix, Table 5-6. It is anticipated that the valves selected for this application will have already passed similar qualification tests in other space vehicles qualification programs. Testing of the two unqualified hardware items, the thrust chamber assembly and the propellant tank assembly, will be necessary.

Thrust Chamber Assembly. The thrust chamber design, i.e., thrust level, injector concept, jet vane design, and chamber materials, is similar to the JPL Ranger motor. A similar motor built and tested at TRW Systems has demonstrated the ability to operate in the blow-down mode and has shown satisfactory performance of the Shell 405

Table 5-6. Prequalification Test Matrix

Item	Proof Pressure	Vibration	Acceleration	Shock	Temperature	Humidity	Altitude (6 days at less than 20 nm Hg	Pressure Cycling (400 cycles)	Leak Checks	Combined Environment	Burst Test
Pressurant Fill Valve	x	x	x	x	x	x	x				x
Explosive Valves and Solenoid Valve	x	x	x	x	x	x	x		x	x	x
Propellant Fill Valve	x	x	x	x	x	x	x				x
Rocket Engine Assembly	x	x	x	x		x	x		x		
Propellant Tank	x	x		x	x	x	x	x	x		x

catalyst. Hence, the feasibility is established and the development effort will be utilized to optimize the catalyst bed design, characterize the transient performance with the flight valve configuration, and conduct environmental and performance evaluation of the prototype configuration.

A series of tests will be conducted to obtain a catalyst bed design which gives stable combustion, reliable ignition, and maximum performance. Because of the relatively high ammonia dissociation associated with the spontaneous catalyst, it is important to arrive at a bed depth which produces minimum ammonia dissociation. A change in ammonia dissociation from 40 to 50 per cent represents a decrease of three seconds specific impulse, equivalent to approximately eight pounds of propellant in the Voyager MPS. However, no compromise in ignition reliability or combustion stability will be made to achieve higher specific impulse.

The possible degradation of the spontaneous catalyst under prolonged vacuum exposure will be investigated in laboratory scale during the development program. In theory, the loss of activity under vacuum conditions should not be significant; this has not been verified by experiment. Therefore, four catalyst samples will be tested for activity in the laboratory. One sample will serve as a control, and the other three will be tested after 30, 60, and 90 days of vacuum exposure. The use of the spontaneous catalyst is not considered to be a high risk approach, and these tests are proposed as a relatively low cost precautionary measure. However, should problems such as loss of activity or physical strength be observed, design alternatives could be instituted.

During this phase, a catalyst bed will be assembled and subjected to vibration to determine its compatibility with the flight environment. This test will be the chronological subjection of the catalyst in a prototype thrust chamber to boost phase vibration, a hot firing of the thrust chamber through the midcourse cycle, vibration per the retrothrust specification, and firing through the orbit injection duty cycle.

The purpose of this test is to identify bed strength problems, if any, early in the development program to preclude delay in the subsequent qualification program.

The environmental and performance evaluation is conducted as follows:

> Transient Performance. Despite the fact that the explosive actuated flow control valves will have predictable and reproducible action times, start and shut-down transients will vary somewhat over the range of operating conditions. Consequently, a series of tests will be required to characterize the transient performance. These tests will be conducted in a test rig, which simulates the hydraulic characteristics of the flight feed system, or in an actual flight unit.
>
> Environmental Testing. Following the catalyst bed optimization, which will define the steady state performance, and the transient characterization tests, the thrust chamber assembly will be subjected to a series of tests including acceleration, shock, vibration, vacuum storage, and humidity. Typically, a thrust chamber, complete with a flight vavle package and simulated jet vane actuators would be mounted on a shake table in a support equivalent to the flight mount. The assembly will then be subjected to vibration at specified values in three orthogonal directions to obtain resonant frequencies and transmissibility factors. The engine will then be hot fired to a duty cycle in excess of the anticipated flight requirements. The assembly, with the explosive valves replaced, will also be subjected to additional altitude and humidity tests and firings at extremes of temperature during the component verification test phase.

Propellant Tank Assembly. During the Phase IB design studies, analyses will be conducted on the flowdown characteristics of gas pressurization and propellant feed system. The tank and expulsion system designs will consider environmental influences, such as propellant sloshing, axial acceleration, vibration, leakage, expulsion efficiency, long-time storage. The developmental test program will permit systematic evaluation of the dynamic and static characteristics under flight conditions and serve to verify the designs selected.

The propellant feed system developmental tests are divided into three categories: 1) tank development, 2) expulsion system development, and 3) combined tank and expulsion tests. The types of tests to be included are: structural physical properties, proof pressure, vibration, acceleration, shock, pressure cycling, leakage, and burst tests. Additional testing on the positive displacement bladders could include expulsion efficiency, long-term storage and helium premeability testing.

d. System Verification and Prequalification Tests

At the completion of component testing, a complete breadboard MPS will be assembled and tested at simulated altitude. It is planned to fuel the system with hydrazine and pressurant and allow it to stand for approximately seven days. During this period the system will be monitored for propellant leakage or pressure decay. The system will then be fired at a simulated altitude in a duty cycle simulating the mission, except for the extended coast.

Other system tests at extremes of temperature environment will also be required to characterize the system completely.

e. Qualification and Acceptance Tests

Qualification testing takes place during the period from the 50th to the 72nd week of the program. Acceptance tests will occur at approximately equal intervals through the end of the program. The final specifications for these tests must be established before the details of this test program can be developed. Therefore, the procedures described in the following paragraphs are tentative and are used to indicate the probable time required for the entire program.

Qualification tests will be performed on the system to provide information on possible malfunction effects and safety limits. The system assembly will be mounted to a structure designed to duplicate the mounting points of the Voyager vehicle. No qualification testing at the component level is anticipated because of the selection of previously qualified components and an extensive prequalification test program.

Each delivered system will be subjected to a series of component tests which will demonstrate that the system will perform within specification limits. Typical acceptance procedures are as follows:

- Propellant tanks

  Proof pressure
  Leakage

- Solenoid valves

  Proof pressure
  Leakage
  Previbration functional test
  Combined sine and random vibration test
  Post vibration functional and leakage test

- Explosive actuated valves

  Proof test

- Thrust chamber

  All delivered thrust chambers will be required to be functionally tested through a series of two hot firings and a vibration schedule. Engine calibration and flow measurements tests will be made during this acceptance test series. Valves will be simulated with calibrated orifices and flow control will be by a solenoid valve.

  Each injector valve assembly will be calibrated for operating flow and pressure drop. After this, the assembly will be cleaned and attached to the flight model combustion chamber and will then be leak tested and fired for 10 seconds to obtain performance data. After firing, the thrust chamber assembly will be vibration and leak tested, and then fired again for 10 seconds to confirm performance. Both performance tests will be conducted at simulated altitude conditions. These performance tests will demonstrate conformance to engine thrust level and specific impulse specifications.

Data recorded during each engine acceptance firing series will include thrust, chamber pressure, fuel flow rate, thrust chamber outer wall temperature, fuel temperature, and nozzle exit plane ambient pressure.

4.3.2 Retropropulsion Motor

The schedule for the proposed retropropulsion motor development plan is shown in Figure 5-17. As this figure shows, the program is composed of a design study phase, a design and development phase, a qualification phase, and a flight phase. System tests with the motor integrated into the vehicle are discussed under the vehicle system development plan.

a. Design Studies

In Phase IB the tasks will consist of evaluation of the effects of updated retropropulsion motor performance, interface, and envelope requirements and a detailed preliminary design study of the motor. The first category will include tradeoff and optimization studies of performance parameters such as thrust, chamber pressure, and expansion ratio. In the latter category, practical designs will be evolved for the grain, case, nozzle, igniter, insulation, and thrust vector control. On the basis of this work, a detail design specification will be prepared and submitted to prospective retropropulsion motor subcontractors. This work will also form the basis for evaluation of the vendors' proposals, and preparation of the required subsystem functional specification. Selection of the retropropulsion motor subcontractor and approval of this selection will complete the Phase IB accomplishments.

b. Design and Development

The subcontractor will complete the detailed design with a drawing release six months after Phase II initiation. The development testing scheduled during this period will demonstrate the feasibility

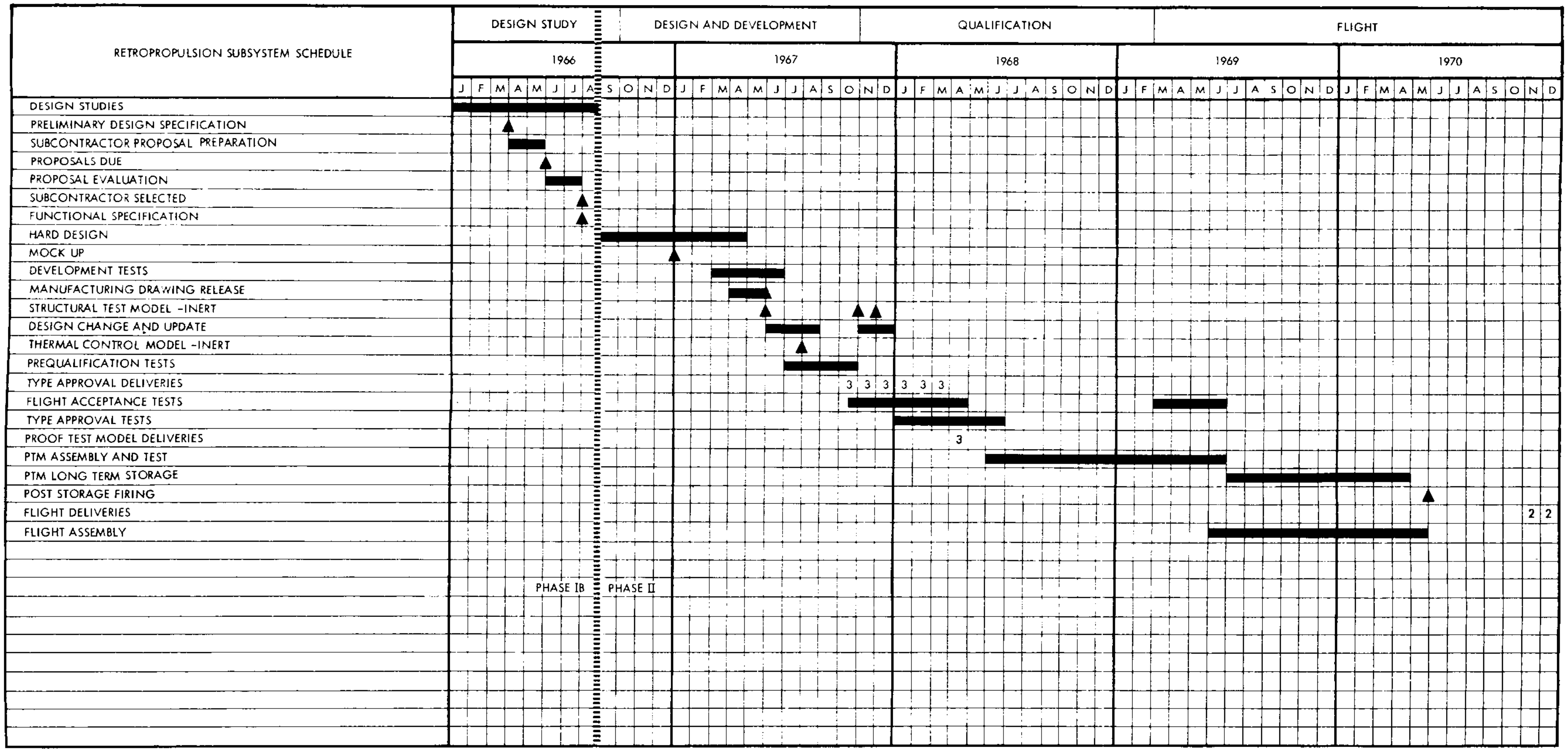

Figure 5-17. Retropropulsion Subsystem Schedule

of component design approaches. This testing includes structural tests of the pressure vessel, nozzle, and attachment skirt; static firings in heavyweight cases for evaluation of grain and nozzle design; cold-flow tests for injectant location optimization; and component evaluation tests on the safe and arm device, igniter, and TVC system components such as injectors, injectant bottles and the pressurization source. This testing is summarized in Table 5-7. Prior to qualification, a series of tests will be run on the complete flight-weight motor and TVC system to determine design and performance characteristics. Tests will be made under conditions which are more severe than qualification to determine performance margins and establish design confidence. This test series is summarized in Table 5-8.

During the design and development phase, deliveries of various inert models are required to support various test vehicles. An approximate time scale for these deliveries is indicated on Figure 5-17.

c. Qualification

Qualification consists mainly of the type approval test to qualify the retropropulsion motor for flight. Prior to initiation of the type approval program each motor will be subjected to flight acceptance testing. Motors will be delivered during this phase for use in the proof test model. A breakdown of the tests proposed for the type approval test program is given in Table 5-9.

d. Flight Models

The manufacture and flight acceptance testing of the flight motors will also include the proof test model life test following long-term storage.

Table 5-7. Development Test Program

Table Development Test Program

No. of Tests	Components*	Purpose	Conditions	Data
3	Case and nozzle	Verify structural analysis	Hydrostatic pressure to motor	Pressure strain
3	Case	Determine case yield	Hydrostatic pressure to burst	Pressure strain
3	Nozzle (excluding TVC)	Determine nozzle integrity and erosion rate	Static fire nozzle on test motor under design mass flow and gas temperature conditions	Measure nozzle integrity and thrust erosion, compute lateral shift in centroid of throat
2	Inert loaded motor with attachment ring	Evaluate attachment ring design and failure criteria	Load to flight conditions, then to failure	Deflection strain
25	Initiator	Evaluate functioning time and output; establish reliability trends	Temperature condition; static test at ambient pressure	Firing current, prefire and post-fire resistance, pressure history
10	Pyrogen igniter with safe and arm	Evaluate performance; establish reliability trends	Temperature, vacuum condition; static test at ambient pressure	Firing current, pre-fire and post-fire resistance, pressure history
50	TVC injector and flow controller	Evaluate injector pattern and flow control perfòrmance	Ambient temperature and pressure	Pressures, spray pattern, flow rates
4	TVC injectors and retro nozzle	Optimize injector location	Ambient (cold flow)	Pressure profile, flow rate
10	TVC pressurization source	Evaluate gas flow rate and temperature	Ambient temperature and pressure	Temperatures, flow rates, pressures
2	TVC injectant tank	Evaluate compatibility expulsion efficiency	Ambient expulsion tests	Pressures, flow rates
2	TVC injectant tank	Evaluate bottle strength	Hydrostatic pressure to burst	Pressure strain
3	TVC subsystem	Evaluate system performance	Ambient pumping system test; simulated firing	Pressures, flow rates, temperatures
3	Nozzle closure	Evaluate blow-out characteristics	Hydrostatic pressure simulating ignition start-up	Closure integrity, blow-out pressure
3	Heavywall motor	Evaluate motor ballistics and ignition characteristics	Temperature condition; static test at ambient pressure	Pressure, thrust, ignition timing

*Development tests on components and the motor will be conducted concurrently with failure mode analyses which will indicate the exact type and extent of testing to be done. Therefore, the test plan shown here is meant to illustrate the type of tests anticipated and is not limited to precisely the tests shown should additional tests be deemed necessary.

Table 5-8. Prequalification Test Program

No. of Tests	Components	Purpose	Conditions	Data
4	Flightweight motor (including TVC)	Evaluate motor design and performance characteristics	Temperature condition; static test in ambient and altitude pressure environments	Pressures, thrust, temperatures, photography
6	Flightweight motor (including TVC)	Establish design confidence prior to undertaking qualification phase	Condition to environmental extremes 20% greater than nominal flight extremes; static test altitude back pressure	Pressure, thrust, temperatures, photography
10	Spent flightweight case/nozzle assembly from preceding tests	Determine failure criteria; establish reliability trends	Hydrostatic pressure to failure	Pressure strain

Table 5-9. Type Approval Test Program

Test Description		Motor Number 1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18
Temperature Conditioning:	Ambient	x	x	x	x	x					x		x		x				
	Low						x	x				x		x		x	x		x
	High								x	x								x	
Pressure	Ambient												x	x					
	Altitude	x	x	x	x	x	x	x	x	x	x	x			x	x	x	x	
Vibration											x	x							
Vibration/acceleration													x	x					
Shock/acceleration															x	x			
Centrifuge fire													x	x					
Drop																			x

4.4 Stabilization and Control Subsystem

4.4.1 Summary

This implementation plan presents the engineering activities concerning the analysis, design, procurement, development, and testing of the stabilization and control subsystem and its equipment, assemblies, parts, and special test equipment. The development task flow is shown in Figure 5-18.

The majority of the equipment proposed for the subsystem presents no development problems.

A system problem associated with the midcourse velocity correction and deboost phases of the Mars trajectory is the thrust vector offset angle resulting from the proximity of the engine gimbal point to the center of gravity and the center of gravity offset envelope. Based on the selected configuration geometry and the presently specified lateral center of gravity offset envelope, maximum trim thrust vector deflections of 1.7 and 2.9 degrees occur for the midcourse velocity correction and deboost phases, respectively. Should these offset angles result in unacceptable velocity errors, the thrust vector offset can be effectively reduced by increasing the control moment arm length, reducing the acceptable center of gravity offset envelope, or compensating for the offset through the SCS. Of the three alternatives, reducing the acceptable center of gravity offset envelope appears most desirable.

In order to complete development testing in time to meet the 12-month Phase II drawing release date for the 1969 test flight, the need to start procurement of gyros during Phase IB is indicated.

4.4.2 Analysis and Design

Various analyses are required for design of the subsystem optical sensors, gyro reference assembly, reaction thrust control, jet vane actuator, and electronics. These analyses will be conducted during Phase IB, continuing where necessary into Phase II.

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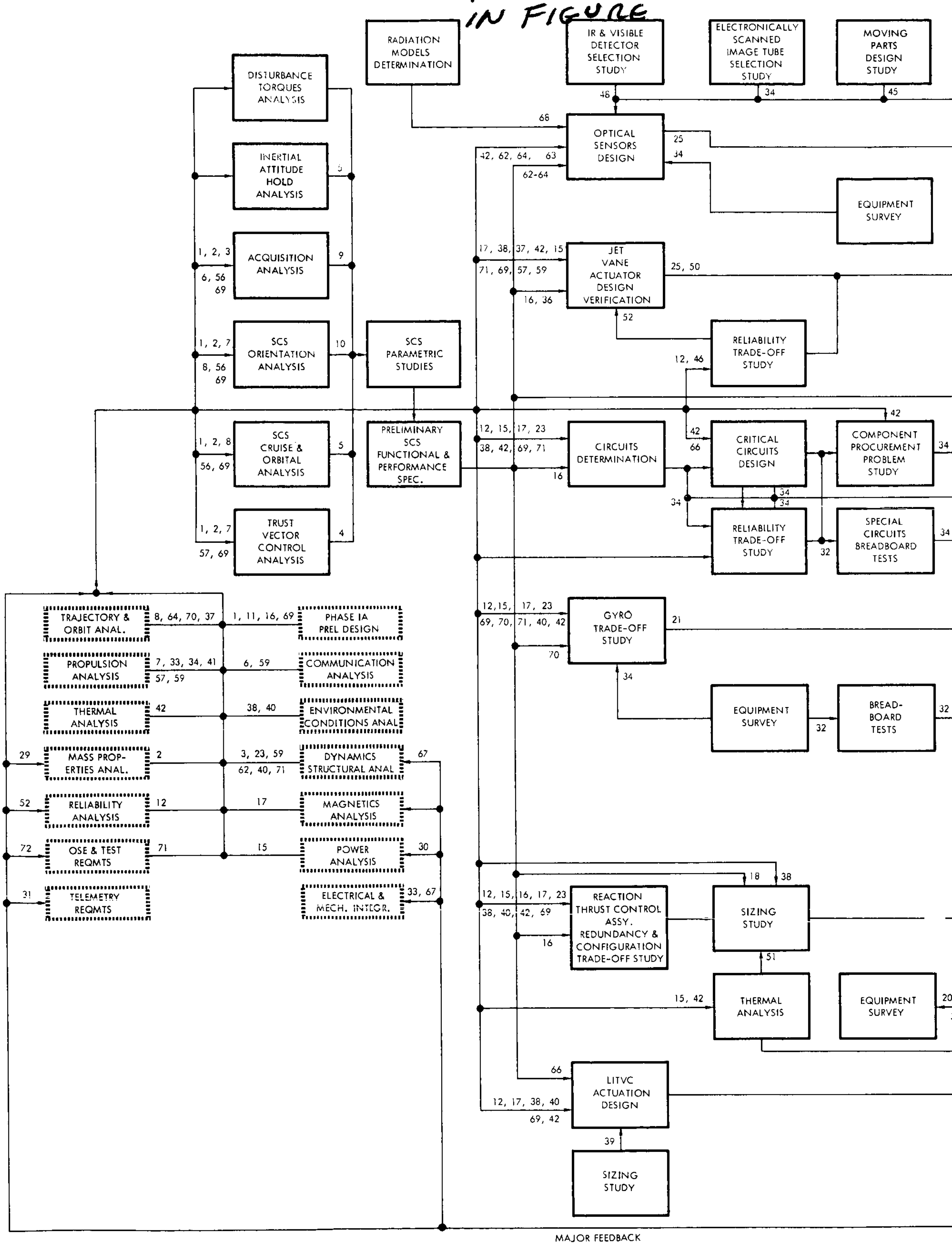

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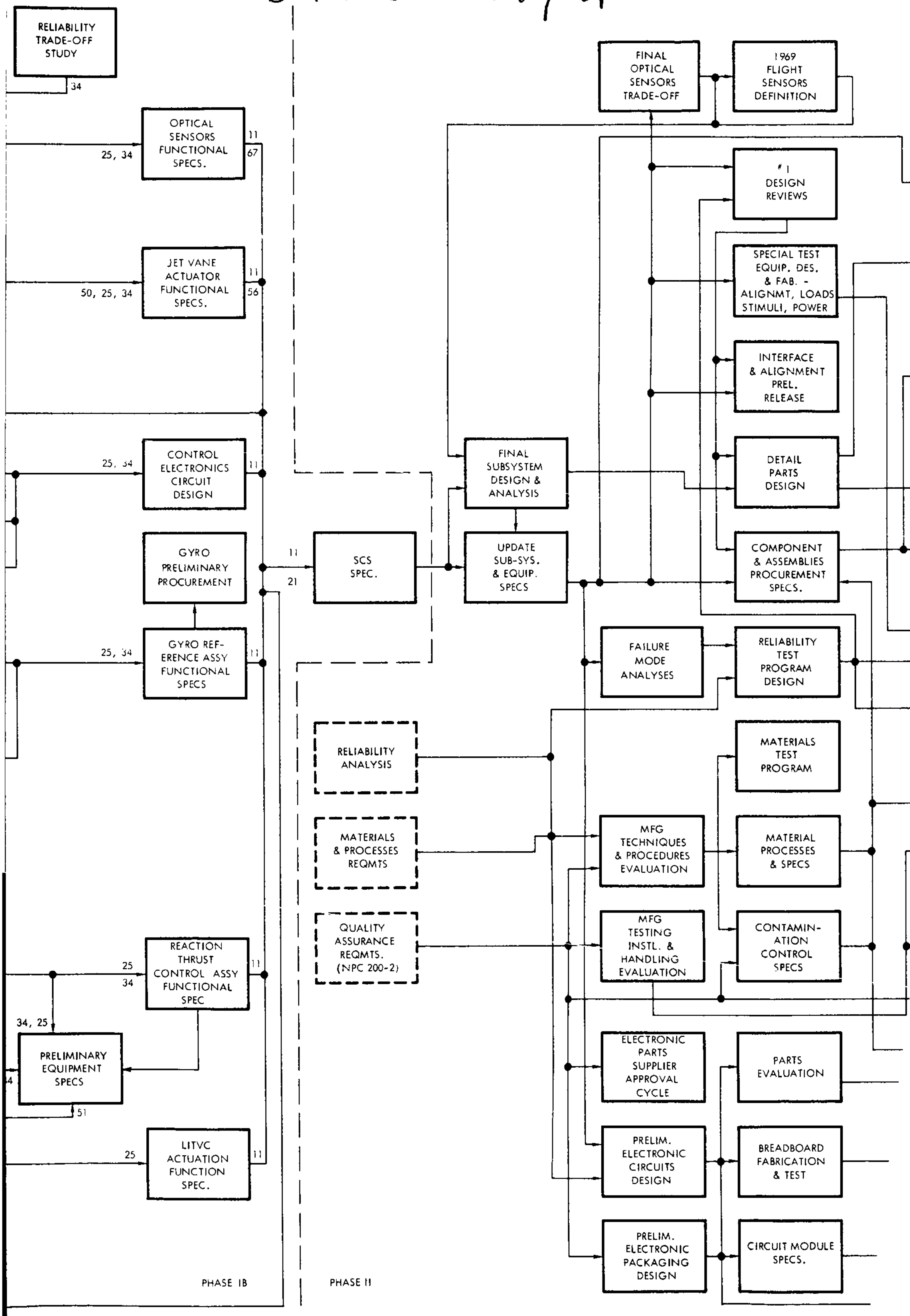

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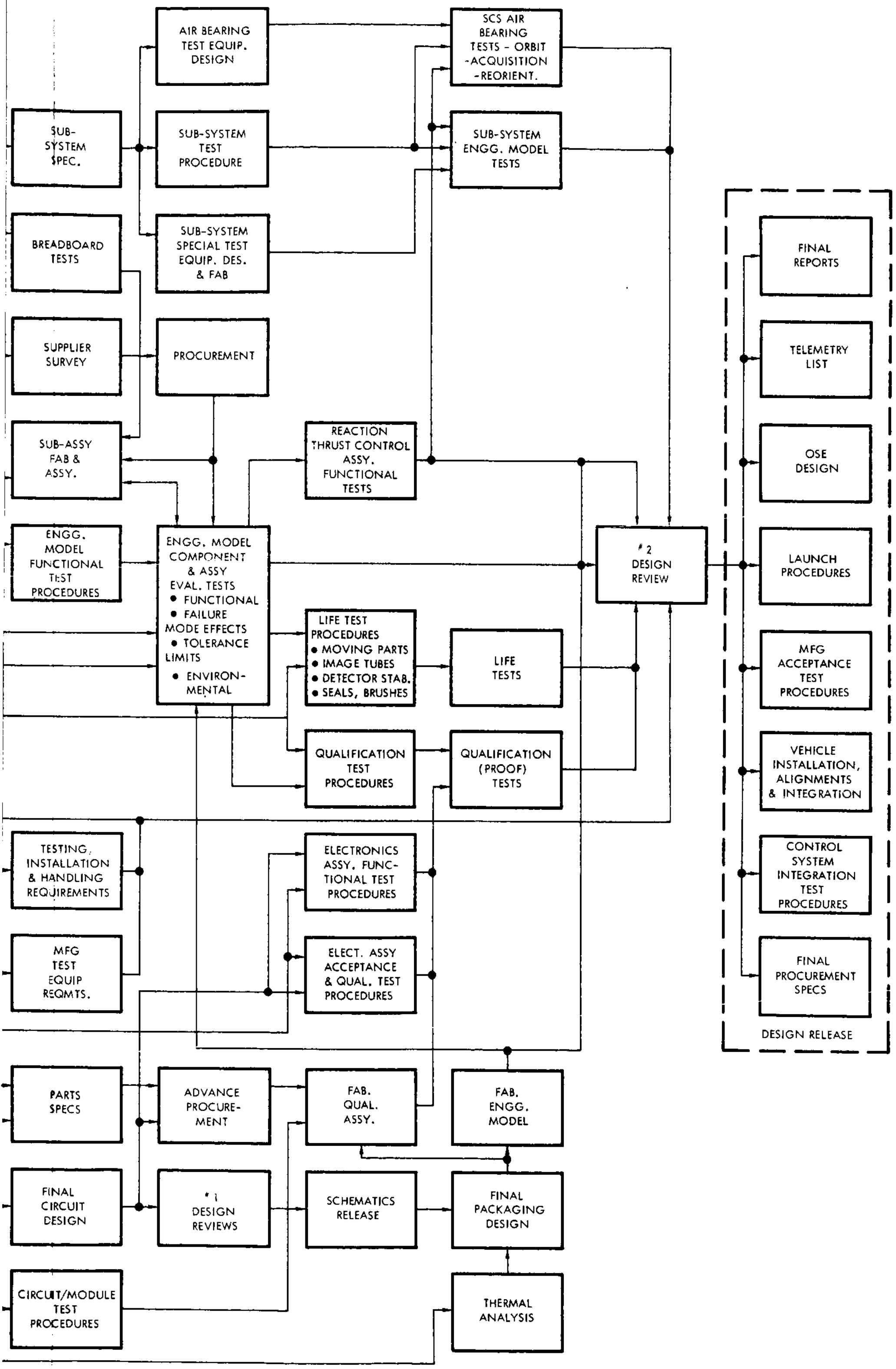


igure 5-18. Stabilization and Control Subsystem Development

3

The detailed design activities will be primarily conducted during Phase II. However, two activities will be conducted in Phase IB, the gyro reference assembly and the control electronics assembly. Due to the schedule-critical deadlines of the gyro and high reliability electronics parts, breadboard tests will be conducted on these two assemblies during Phase IB.

a. Subsystem Analysis

The following subsystem analyses will be performed requiring input data such as view angles, input characteristics, sensitivities, accuracies, moments of inertia, center of gravity offset and uncertainties, control moment arms, tipoff rates, acquisition time requirements, thrust level and thrust centerline uncertainty, accuracy requirements for midcourse corrections and orbital injection, disturbance inputs to spacecraft, and results of Phase IA subsystems preliminary design.

Acquisition. A detailed analysis will be completed of the acquisition scheme used to initially stabilize the spacecraft after separation from the boost vehicle and for subsequent acquisitions. This analysis will define an acquisition scheme including sequencing, time required for acquisition, control methods for acquisition, and functional specifications for the control system.

Alignment. Upon completion of the spacecraft layout, preliminary structural, thermal, and interface design requirements, a complete system alignment analysis will be conducted to establish the effective sensor alignment due to spacecraft mechanical and thermal deformation. The requirements for installation alignment will also be established.

Attitude Orientation Requirements. An analysis will be performed to determine the accuracy and response requirements for orienting the spacecraft prior to performing course corrections, capsule separation, and injection into orbit about Mars. Functional specifications for the control system to meet these requirements for positioning midcourse and deboost motor will result.

Thrust Vector Control. Detailed analysis of the SCS requirements for the orbital and cruise phases of the mission, including the requirements for precise attitude orientation of the experiment package during orbit, will be performed to provide functional specifications for the control system. TVC accuracy, response, and control requirements will result.

Disturbance Torque Estimates. An analysis of the disturbances expected to act on the spacecraft in transit and in orbit about Mars will be performed resulting in disturbance torque magnitude versus time and cyclic/secular torques classification.

Parametric Studies. The above analyses will permit stabilization and control parametric studies to be performed and will culminate in the formation of the final SCS functional specification best fitted to the over-all mission objectives.

b. Optical Sensors Analysis and Design

A number of equipment analyses is required to select requirements for optical sensors.

Target Radiation Analysis. The available data on earth, Mars star fields, and Canopus and star fields about Canopus will be studied together with Mariner C data. Using Voyager trajectory data control sequences and the optical sensor requirements, models will be established for determining the various bodies to be sensed. The analysis will establish target discrimination logic requirements. Preliminary analyses in these areas are presented in Appendix B of Volume 5.

Electro-Optical Analysis. From the sensor requirements and detector data, the choice of detector will be made. The optical requirements will be established by analysis and a configuration will be selected. The optical designs will consider the problems of scattered and reflected light. The search and track requirements for the star sensor will be established and the necessary functional techniques will be developed.

Error Analysis. The error budget will be established, based on the sensor requirements and the Voyager control sequences. The error analysis will include parametric studies of the signal processing and logic circuits.

Sensor Design. The design activities for optical sensors include:

- Detailed electro-optical design and detector-to-optics design integration. The design of the optical sensors employs proven approaches using design techniques and sensors with demonstrated flight experience. Particular emphasis will be placed on the specific design requirements imposed by the Voyager mission.
- Mechanical design including structure and mechanisms
- Thermal design
- Electronics detail design to implement signal processing and logic circuits. The application of redundancy techniques will be investigated further. Critical circuit factors will be identified and evaluated.
- Design of sensor stimuli and other special purpose fixtures and test equipment. Special techniques requirements for the sensor stimuli will be identified and the design implications established.

c. Gyro Reference Assembly Analysis

Analyses of electronic circuits and gyro parameters to determine a transfer function of the gyro reference assembly in various modes will include:

- Rate mode
- Position mode
- Precision turn mode

Using the spacecraft turning rate data, an analysis of the gyro parameters will evolve a voltage proportional to the spacecraft turning rate for each of these modes.

Design of the gyro control loop, current supply, and heater control will be conducted during Phase IB and continued during Phase II. The selection of the gyro for the reference package will be complete in Phase IB because of procurement lead time. The thermal design of the interface between the gyro reference package and the spacecraft will include calculations made to determine the desired characteristics of the mounting surface to achieve the desired thermal impedance.

d. Reaction Control Analysis and Design

The analyses associated with the reaction thrust control involve gas weight and thrust dynamic determinations.

Gas Weight. An analysis will be conducted to determine the amount of gas to be carried based on probability of various failure mode effects. Data on leakage, valve open, heater failures, disturbance torques, and various probabilities will be employed to establish these gas requirements.

Thrust Dynamics Analysis. An analysis will be conducted to determine the thrust rise and decay and impulse variation versus time on during operation of the reaction control system. Sizing data on lines, valves, and nozzles will be utilized in conjunction with valve characteristics and environmental conditions for this analysis.

Design specifications for components such as solenoid valves, pressure regulators, and transducers will be created for procurement of components. Detailed mechanical design of lines, pressure vessels, and nozzles completes the reaction thrust control design activities. The design approach to reaction control has been utilized on many spacecraft. In addition, the Voyager design will employ high and low thrust level roll reaction control features.

Special attention to magnetic cleanliness, magnetic field cancellation, and system magnetic control will be considered in the application of magnetic valving in conjunction with the program magnetic control requirements. Techniques developed in OGO and Pioneer will be used to control the magnetic fields.

e. Electronics

A parametric tolerance analysis will be conducted employing worst case conditions of all circuits to verify that all components are used within their specification limits. A preliminary circuit tradeoff analysis will be conducted using the reliability apportionment, parts and functional specifications, and early circuit designs. This analysis coupled with Phase IB breadboard tests will provide information for part specifications, circuit revisions, and reliability data. The use of Voyager approved parts will be employed.

The critical circuit factors such as low signal level, noise problems, and filters will be identified and evaluated during Phase IB breadboard tests. The types of electronic circuits and preliminary design will be fully evaluated (Phase IB) in order to identify the high reliability parts required and to initiate early procurement of the long lead items for the 1969 test flight.

f. Jet Vane Actuator

Two primary analyses will be conducted on the jet vane actuator. First, the stress analysis will be conducted to determine the stress on the actuator due to thrust loads on the vane. This analysis employs the jet vane sizing information and the thrust load parameters to establish actuator design requirements. Then a magnetic properties analysis is performed to estimate the magnetic fields produced by the actuator motor and to determine the resulting effects on the experiments. The analysis is required to establish the magnetic design requirements on the actuator.

The actuator has been used on Mariner and other programs and becomes an adaptation for Voyager peculiar requirements. Thus, the design activities consist of determining specific actuator requirements, generating specifications, submitting purchase requisitions, vendor surveys, design reviews, and vendor liaison. The actuator will in all probability be a subcontracted item.

g. Spacecraft Subsystem Design Analysis

The results of the previous analyses, the subsystem and unit specifications, and the other subsystem data will be utilized to integrate the spacecraft subsystem design.

A unit characteristics analysis will be conducted to determine, allocate, and coordinate the unit interface requirements and unit parameters such as impedances, signal levels, gains, allowable errors, and time constants. The analysis will result in a detailed subsystem block diagram and updating of unit and functional specifications.

The interfaces with other Voyager subsystems will be evaluated to coordinate stabilization and control subsystem requirements, including power, structure alignment, thermal, electrical integration, telemetry, and spacecraft testing. The results of this analysis will establish or modify accuracy requirements and budgets, power consumption, thermal control requirements, wiring diagrams, telemetry lists, and spacecraft subsystem requirements of spacecraft testing.

Finally a complete assessment of the subsystem reliability will be made.

4.4.4 Test Program

Two breadboard tests are planned for Phase IB, the gyro reference assembly and the control electronics assembly. The development lead time for gyro's require early breadboarding and procurement to accommodate the drawing release dates associated with the 1969 test flight. The control electronics assembly (CEA) requires early development attention because of the long lead time associated with high reliability parts. The CEA breadboard tests are planned during Phase IB to define the components required and release purchase orders for these long lead items. Procurement associated with engineering models also will be initiated for both the gyro reference assembly and the control electronics assembly. All other breadboards and engineering model tests are scheduled for the early months of Phase II and can be accomplished in the lead time available.

The test program required to develop and qualify the stabilization control subsystem and its complement of units is shown in the test matrix, Table 5-10.

4.4.5 Subsystem Schedule

Figure 5-19 presents the development schedule for the stabilization and control subsystem. Both Phase IB and II are shown for the 1969 test flight. The 1971 equipment will generally be the same as employed during the test flight except for sizing and equipment reliability redundancy applications. Early development and resulting tests associated with the 1969 launch will provide high assurance of success during the 1971 and subsequent mission opportunities.

Table 5-10. Stabilization and Control Subsystem Test Matrix

Equipment	Test Title	Purpose	Test Article	Test Equipment
	Type approval tests	Verify that flight type unit will operate within specifications after exposure to type approval level shake and vibration and will operate within specifications at type approval level thermal-vacuum conditions	Flight unit	Vibration test equipment, thermal vacuum, DC voltmeter, DC power supply, position control transmitter, position repeater, dekavider, mechanical test fixture
	Acceptance tests	Verify flight unit will operate within specification after exposure to acceptance level shake and vibration and will operate within specifications at acceptance level thermal-vacuum conditions	Flight unit	Vibration test equipment, thermal vacuum, DC voltmeter, DC power supply, position control transmitter, position repeater, dekavider, mechanical test fixture
	Magnetic properties test	Determine magnetic field characteristics	Flight unit	Power supply, magnetic test facility
Electronics	Breadboard test	Discover problems resulting from temperature and electrical testing; determine the electrical characteristics	Breadboard control electronics	Test console, temperature control chamber, capital electronic equipment
	Engineering model-tests	Determine grounding and signal cross coupling problems Verify expected performance for electrical and temperature testing Determine the necessary production tests to be performed	Engineering model - control electronics	Console, temperature control, capital electronic equipment
	Type approval test	Discover any structural, electrical, temperature, and magnetic field problems	Prototype-control electronics	Environmental Laboratory, thermal vacuum, shake, shock, and magnetic field equipment
	Acceptance test	Verify unit fabrication is correct and that unit electrically and mechanically withstands all expected environments and electrical conditions	Control electronics, prototype spacecraft model, and flight and spares	Capital electrical equipment, environmental equipment
Subsystem development	Breadboard subsystem test	Determine the compatibility of units and make preliminary measurements of functional parameters so changes can be implemented if required	Breadboards of electronic units and engineer models of other units or simulators	Electrical power supplies, digital voltmeters, voltmeters AC and DC, oscilloscopes, recorders, counters, stimuli for sensors, holding fixtures, turntable
	Engineering model test	Determine the compatibility and functioning of the units as a system and as units in the system	Engineering models of all SCS units	Spacecraft power supply or simulator, cables, voltmeters, AC, DC, digital, oscilloscopes, recorders, counter, stimuli for sensors, holding fixtures, turntables, interconnection and switching rack, test facility-low sensor interference provisions, alignment equipments-levels-autocollimators
	Three axis - air bearing space simulation test	Verify the functioning of the subsystem by performing closed loop tests of all maneuvers; check logic, sequencing, commands, and functional parameters	Engineering or type approval models of SCS units	Air bearing simulator, air bearing simulator test facility, stimuli for sensors, spacecraft structure simulator, telemetry set, gas supply, battery chargers, battery set, interconnecting cables, command transmitter and receiver, pneumatic system, recorders, alignment and balancing equipment-autocollimators, levels, motion picture cameras

Table 5-10. Stabilization and Control Subsystem Test Matrix (Continued)

Equipment	Test Title	Purpose	Test Article	Test Equipment
Optical sensors	Breadboard tests	Determine electro-optical feasibility	Breadboard sensor	
	Engineering model functional tests	Determine over-all functional feasibility	EM sensors	Sensor stimuli test console
	Type approval tests	Qualify sensor design for flight	Type approval sensor	
	Environmental tests	Evaluate performance of engineering models under various environmental stresses	EM sensors	Thermal vacuum chamber, vibration tables, shock tables centrifuge
	Acceptance tests	Establish functional performance Uncover workmanship errors	Flight sensor	
	Life tests	Evaluate reliability	Sensors	
	Magnetic properties test	Assure the meeting of specified magnetic properties	Flight units	Test console, magnetic test facility
Gyro reference assembly	Gyro acceptance test	Determine if gyro meets minimum requirements	Gyro	Gyro test set
	Determination of gyro parameters	Determine for engineering purposes the gyro drift, torquer scale factor, temperature sensitive coefficients and vibration sensitivity	Gyro	Gyro test set
	Gyro life and stability test	Reliability information	Gyro	Gyro test set
	Determination of current generator parameters	Determine for engineering purposes the current output and the temperature sensitive coefficients	Current generator	Ammeter and temperature controlled oven
	Design verification of gyro reference assembly	Determine rate and position scale factors about the three reference axes Determine temperature sensitive coefficients	Gyro reference assembly	Gyro reference assembly test set
	Gyro reference assembly life and stability test	Reliability information	Gyro reference assembly	Gyro reference assembly test set
	Type approval test	Determine rate and position scale factors about the three reference axes; determine temperature sensitive coefficients	Gyro reference assembly	Gyro reference assembly test set
	Acceptance test	Determine rate and position scale factors about the three reference axes; determine temperature sensitive coefficients	Gyro reference assembly	Gyro reference assembly test set
Reaction Control Assembly	Breadboard test	Determine line drops, nozzle parameters, system dynamics	Breadboard plumbing	Pneumatic supply and control console (PSCC)
	Nozzle and heater tests	Determine thrust, flow, and specific impulse	Nozzle and heater assembly	PSCC, electrical power supply, current and power meters, vacuum chamber
	Component developmental functional tests	Evaluate functional performance	Engineering models	PSCC, temperature and vacuum chamber, oscilloscopes, meters
	Component developmental environmental tests	Evaluate performance as a function of environmental stress	Engineering models	PSCC, temperature and vacuum chamber, oscilloscopes, meters, vibration acceleration, and shock test equipment
	Assembly functional tests	Assure performance of the assembly as a unit	Engineering models	PSCC, temperature and vacuum chamber, oscilloscopes, meters, vibration acceleration
	Proof and burst pressure tests	Assure structural integrity and safety factors	Prototype components	PSCC, safety chamber
	Type approval tests	Formally assure mission compatibility by overstress testing	Flight models	PSCC, temperature and vacuum chamber, oscilloscopes, meters, vibration acceleration, and shock test equipment
	Life tests	Assure reliable operation during expected life	Flight models	PSCC, temperature and vacuum chamber, oscilloscopes, meters
	Acceptance test	Assure quality and performance of flight units	Flight models	PSCC, temperature and vacuum chamber, oscilloscopes, meters and vibration test equipment
Jet vane actuator	Engineering environmental tests	Verify engineering unit will survive specified vibration and shock levels and operate in space environment (thermal vacuum)	Prototype	Vibration test equipment, thermal vacuum, DC voltmeter, DC power supply, position control transmitter, position repeater, dekavider, mechanical test fixture
	Functional test	Verify engineering unit meets all design requirements other than environmental	Prototype	DC voltmeter, DC power supply, position control transmitter, position repeater, dekavider, D. C. Megger, torque gauge, leak detector, mechanical test fixture

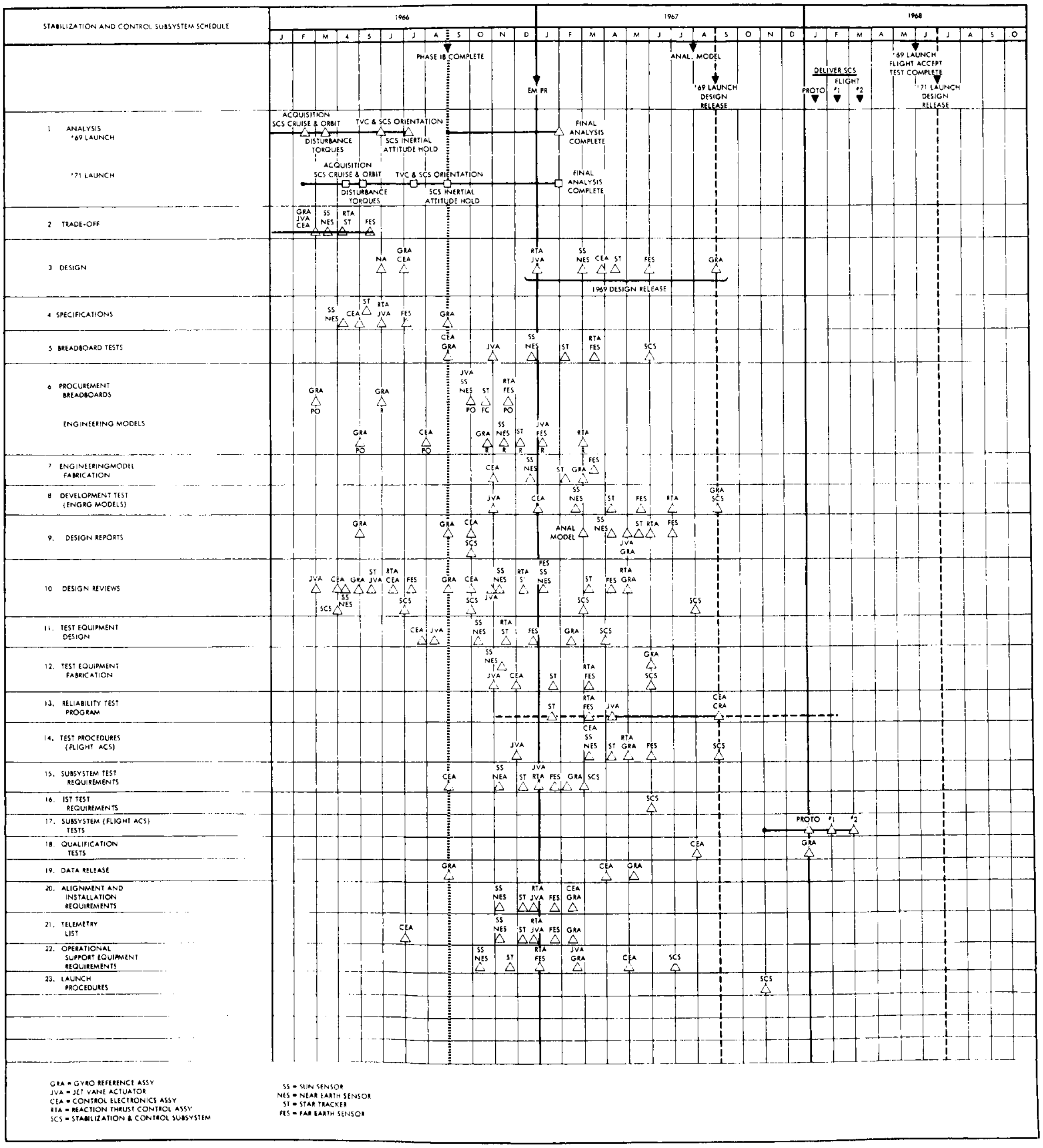


Figure 5-19. Stabilization and Control Subsystem Schedule

4.5 Central Sequencing and Command Subsystem

The development plan for the central sequencing and command subsystem for the 1971 Voyager mission is presented in this section. This effort is similar to the effort required for the 1969 mission since essentially the same equipment configuration is expected to be used. Most of the information obtained during the 1969 development in terms of the central sequencing and command subassembly (CS and C) design and performance will therefore be directly applicable to the 1971 mission. The differences stem primarily from the detailed specification of functional requirements, since the later mission includes capsule separation, Mars retropropulsion, and orbit maneuvers, whereas the earlier mission only involves simulated versions of these maneuvers.

The development of the CS and C subsystem for Voyager is similar to that of the Mariner C CC and S and command decoder unit, the Pioneer, OGO, and Comsat command distribution units and the Apollo LEM abort guidance computer. It consists of iterated detailed requirements determination, and logic, circuit, packaging, and reliability analyses of the subassemblies and of the integrated system, supported by thermal, vibration, and shock tests. The analysis is performed using analytical techniques and computer simulations. It iterates upon changes in requirements, environmental conditions, system configurations, component information, and information obtained from the various tests performed on the units and integrated system. The initial tests provide new inputs to the design analysis and packaging techniques. The final tests are performed on the engineering models characteristic of the flight hardware to verify the performance of the sequencing and command system.

The activities planned for the design and development of the CS and C are presented on Figure 5-20. The schedule for Phases 1B and II is shown in Figure 5-21. A summary description of the plan follows.

4.5.1 General Approach

Except for the special applications that are involved for the Voyager mission and the new circuits that have to be designed to meet them, all

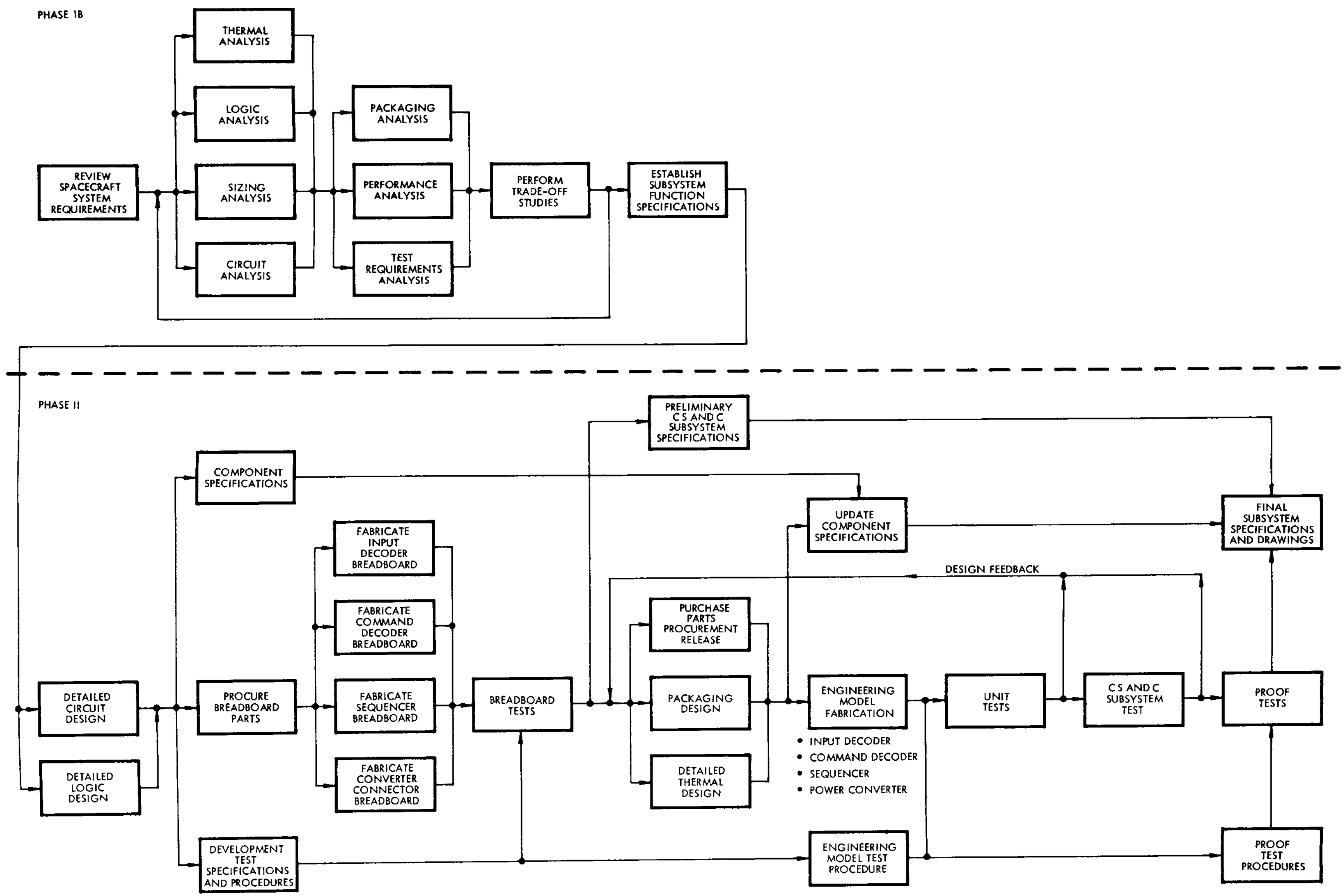


Figure 5-20. Central Sequencing and Command Subsystem Development Flow

CENTRAL SEQUENCING & COMMAND SUBSYSTEM	PHASE 1							
	1966							
	J	F	M	A	M	J	J	A
CONTRACT GO-AHEAD								
CS&C WORK PACKAGE APPROVED								
CS&C REQUIREMENTS ISSUED								
ANALYSIS								
CIRCUIT ANALYSIS				P			F	
LOGIC ANALYSIS				P			F	
THERMAL ANALYSIS								F
ENVIRONMENT ANALYSIS				P				F
RELIABILITY ANALYSIS							F	
COMMAND LIST ISSUED			P				F	
CS&C TELEMETRY REQUIREMENTS ISSUED			P			F		
CS&C POWER REQUIREMENTS ISSUED			P			F		
DESIGN								P
SEQUENCER								P
POWER CONVERTER								P
INPUT DECODER								
COMMAND DECODER								
LONG LEAD HIGH RELIABILITY PARTS ORDERED								
PARTS LIST ISSUED							P	
DESIGN [redacted] REVIEWS		NO. 1		NO. 2				
DEVELOPMENT FABRICATION								
BREADBOARD PARTS ORDERED								
BREADBOARD FABRICATION								
ENG'R MODEL FABRICATION								
PROOF TEST MODELS								
TYPE APPROVAL UNITS & SUBSYSTEM								
TESTING								
SEQUENCER BREADBOARD								
INPUT DECODER BREADBOARD								
COMMAND DECODER BREADBOARD								
POWER CONVERTER BREADBOARD								
ENGINEERING MODEL TESTS								
PROOF TESTS - PERF., THERMAL VAC., SHOCK, VIBRATION								
LIFE TESTING								
DOCUMENTATION								
TEST PLAN								
COMPONENT SPECIFICATIONS							P	
UNIT SPECIFICATION							P	
SUBSYSTEM FUNCTIONAL SPEC.							P	
DELIVERIES - SC&S								
TYPE APPROVAL - JPL								
PTM S/C								
PROTOTYPE S/C								
ENG'R MODEL S/C								
FLT S/C NO. 1								
FLT S/C NO. 2								
FLT S/C NO. 3								
LIFE TEST S/C								

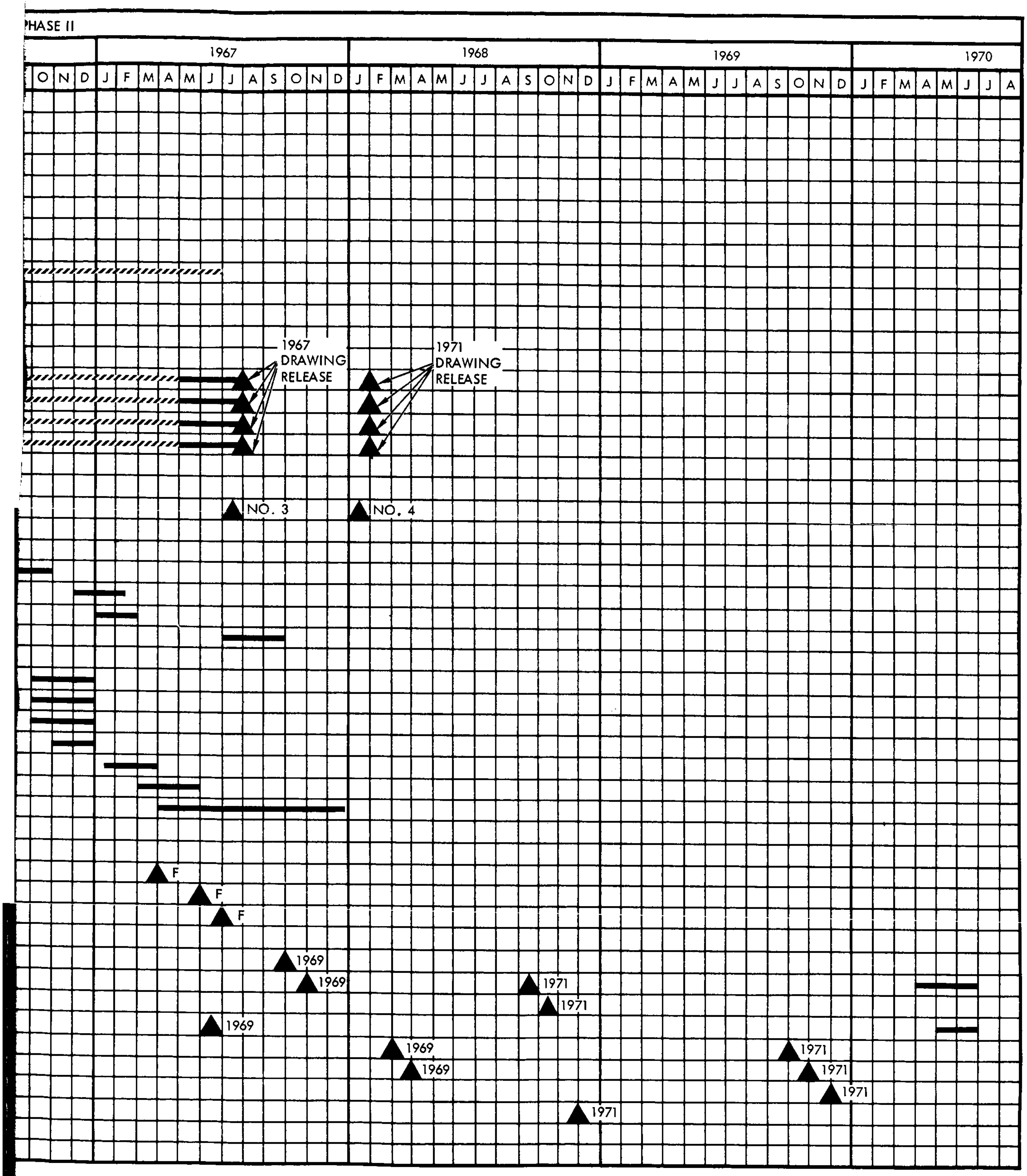


Figure 5-21. Central Sequencing and Command Subsystem Schedule

of the elements in the CS and C are standard items well within the state of the art. The CS and C subsystem has been deliberately constrained to a well developed standard state-of-the-art design. Its electrical, magnetic, structural, thermal, and reliability characteristics have been based on devices currently under funded development or production by TRW Systems. On other programs, the detailed electrical specifications have been negotiated, sources selected, and devices already received, tested, and used. Similar or identical devices will be used on the CS and S so that high reliability as well as minimum cost and schedule difficulties will be assured. Although no new problems are anticipated, the fact that a new configuration is being implemented means that detailed analyses must be made. For example, size, weight and power requirements have to be determined. The registers, counters, decoding matrix, and memory have to be sized. The decoding, control, and enable logic have to be formulated and the circuits designed and sized for power, weight, and reliability. The structural integrity in the anticipated physical environment must be established. Consideration must also be given to reduce the susceptibility of the CS and C to electrical, magnetic, and radiation environments. The design criteria must assure adequate circuit margins for long life and stability. This is particularly true of the crystal oscillator and the divide circuitry which provides the spacecraft frequencies and timing signals. Precedence for adequate margins has already been set in the Mariner C CC and S and other inhouse designs and will be continued in this program.

4.5.2 Analysis and Design

a. Requirements Analysis

Supporting analysis will be provided to establish the functional requirements of the CS and C subsystem. The effects of the requirements on the design will be fed back to the systems analysis and to the design of the other subsystems. Tradeoffs will be conducted to establish optimum interface conditions and to define the CS and C design constraints. Detailed design implications will be fed back to iterate on the functional requirements.

b. Logic Analysis

A system of logic equations will be developed for the input and command decoders and for the sequencer to define the CS and C functions (see Volume 5). These equations must be analyzed for their compliance with the requirements and for internal consistency. In support of the analysis use will be made of logic simulation techniques programmed on the IBM 7094. Since the logic must be adapted to the special requirements of the mission, the effort will, for the most part, involve new formulation.

c. Circuits Analyses

Analysis will be performed on the new circuits designed to mechanize the logic equations and to form the power converter. The results of such an analysis will yield confidence values of reliability, worst case effects, parameter variations, drift stability, component redundancy, crosstalk potential, and dynamic and static response.

Analysis will be made of input and command decoder tolerances to a combination of white noise and spurious signals coupled with extreme drifts of the component to determine the effect on false command completion.

Integrated circuits will be purchased and qualification tested to meet the mission reliability requirements. The oscillator will be selected to meet the long-term stability requirements for the system. Special circuitry will be designed as required and tested to meet the conditions discussed above.

d. Packaging Analysis

A packaging analysis will be conducted to determine structural integrity based on size and weight constraints, and the thermal, RFI, and radiation environment.

Analysis of the CS and C packaging will be made to establish that it meets the environmental requirements, and that outline dimensions, weights, centers of gravity and moments of inertia are compatible with the flight model spacecraft dynamics and thermal control.

e. <u>Testing Requirements Analysis</u>

Analysis will be conducted to determine what test levels are required to enable the subsystem to survive the environments of transportation, launch, cruise, and the Mars orbital mission.

The four major units of the CS and C subassembly (input decoder, command decoder, sequencer, power supply) will be fabricated and tested as independent entities. Each unit will be tested and qualified, wherever possible, to the appropriate environmental specifications. Vibration and accelerated life tests on a sample basis may be incorporated at this level in order to test for any unknown failure modes. Finally, upon integration of these units into a CS and C subassembly the total unit will be vibrated and tested to an appropriate thermal vacuum environment.

The proposed development tests are summarized in Table 5-11.

Table 5-11. Design and Development Test Summary for Central Sequencing and Command Subsystem

Table 5-11. Design and Development Test Summary for Central Sequencing and Command Subsystem

Test Title	Purpose	Test Unit	Description	Test Equipment	Remarks
Input decoder logic test	Verify logical analyses and design	Logical equations	Logical equation test; bit-by-bit simulation of the operations of the input decoder	Logic equation simulator program and high-speed computer	Removes internal inconsistencies
Command decoder logic test	Verify logical analyses and design	Logical equations	Logical equations test; bit-by-bit simulation of the operations of the command decoder	Logic equation simulator program and high-speed computer	Removes internal inconsistencies
Sequencer logic test	Verify logical analyses and design	Logical equations	Logical equations test; bit-by-bit simulation of the operations of the sequencer	Logic equation simulator program and high-speed computer	Removes internal inconsistency
CS and C logic test	Verify integrated system logic	Logical equations	Logical equations test; simulates integrated operations of the CS and C	Logic equation simulator and computer	Checks overall consistency
Oscillator drift test	Verify long-term stability of oscillator	Oscillator	Tracks the frequency of the oscillator to determine the variation from nominal	Drift test oscillator	
CS and C Input/output test	Breadboard model evaluation	Engineering breadboard	Provides input power, simulates input interface, generates input data (direct and quantitative commands), furnishes loads for output lines, and tests output signals	Subsystem test set	A self-contained, rack-mounted unit with power supply, tape reader, frequency source, test control unit and cabling.
CS and C input/output test	Engineering model evaluation of packaging design at environmental extremes	Engineering model CS and C	Subject CS and C to environmental conditions, provide power, simulate input interface, generate input data, furnish loads for output lines, test output signals	Subsystem test set	
CS and C life tests	Determine reliability of system	CS and C	System applied to common plate in vacuum chamber	Vacuum chamber, variable temperature plate, thermocouples, recorders, voltages, power supplies	
CS and C type approval tests	Type approval	Type approval CS and C	Environmental test, vibration, temperature, thermal vacuum, shock, acceleration	Environmental test	
CS and C type approval	Type approval	Proof test model	Space simulation test, solar simulator intensity 20% above and 20% below realistic levels	Space simulation chamber, solar simulator, support fixture, capsule simulator	

4.6 Communications and Data Handling Subsystems

4.6.1 Summary

The major components which form the communications and data handling subsystems are as follows:

a) Elliptical paraboloid, high-gain antenna with a conical horn feed

b) Circular paraboloid, medium-gain antenna with a conical horn feed

c) S-band cup turnstile, low-gain antenna

d) Diplexers, hybrid coupler, and RF circulator switches

e) S-band receiver

f) Signal processor

g) Exciter-modulator

h) S-band power amplifier and associated power supply

i) VHF receiver and demodulator

j) VHF turnstile antenna

k) Digital telemetry unit

l) Magnetic core memory

m) Signal conditioner

n) Tape recorders

RCA as a major subcontractor has design responsibilities for items e through i and TRW has design responsibility for the remainder, as well as over-all subsystems design responsibility.

The approach to development of the subsystems for 1971 is one of early development and flight test on the 1969 test flight to the maximum extent possible. All electronic equipment mounted on the modularized equipment panels of the spacecraft will be identical even to reliability redundancy except for the equipment used for experiment data on the

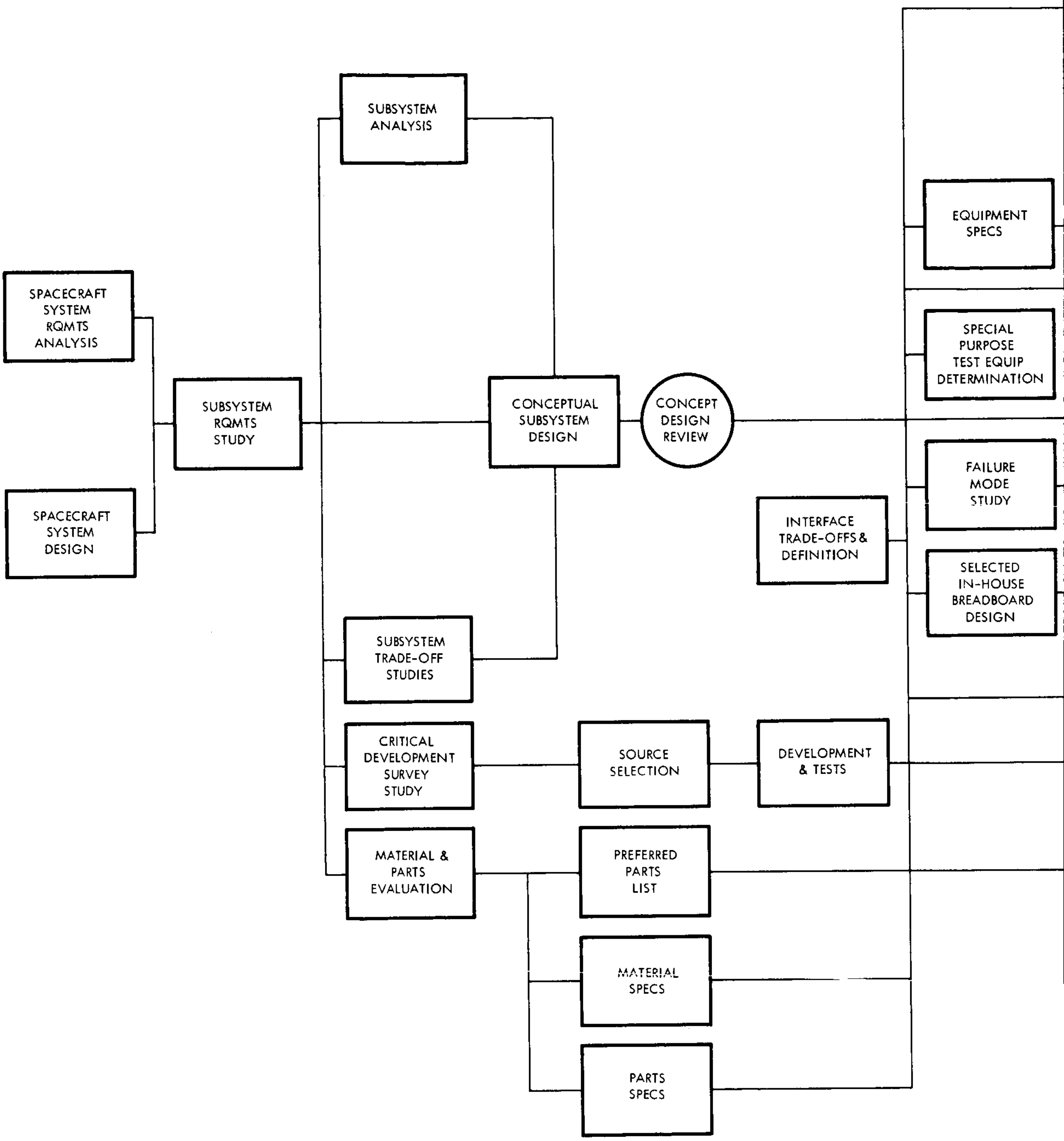
1971 mission. Three panels of electronic equipment including communication and data handling are identical in the 1969 and 1971 configurations. The elliptical paraboloid antenna is identical for both flights; two S-band cup turnstile low-gain antennas are used for 1969 while only one is employed for 1971; a circular paraboloid medium-gain antenna is used for 1971 but is not used for 1969; and the VHF turnstile antenna (capsule link) is not used on 1969 since no capsule is carried.

Development of the communications and data handling subsystem proceeds through Phases IB and II in the manner depicted in Figure 5-22.

The communications subsystem will be designed to minimize long-lead development and to utilize, wherever possible, off-the-shelf components and state-of-the-art techniques. In the power amplifier area it is planned to use the Apollo 20-watt traveling wave tube which has been flight qualified and will have been flown on the Apollo earth-orbit mission before the Voyager launch. Considerable attention will be devoted to studying the reliability of the tube for this particular application. Extensive testing will be initiated during Phase IB and continued into Phase II with the tube being subjected to the failure modes and power supply variations possible during the mission. In addition, it will be tested to the required environmental limits so that a complete reliability assessment of the TWT can be determined. Apollo test results will be received and the data incorporated wherever possible.

During Phase IB a survey will be made on the possibility of using a low-noise preamplifier using tunnel-diodes or hot-carrier diodes to improve the performance of the S-band command link. Although tunnel-diode amplifiers are already operational, insufficient life-test data is available for adequate reliability definition. It should prove relatively simple to add the TDA to the system should satisfactory results be achieved during the Phase IB test evaluation study (see Volume 5, Section 1.5).

Development of the tape recorder for bulk storage will receive close attention to maximise the use of off-the-shelf equipment. Areas which will required some development effort are as follows:


SPACECRAFT SYSTEM RQMTS ANALYSIS
SPACECRAFT SYSTEM DESIGN
SUBSYSTEM RQMTS STUDY
SUBSYSTEM ANALYSIS
CONCEPTUAL SUBSYSTEM DESIGN
CONCEPT DESIGN REVIEW
SUBSYSTEM TRADE-OFF STUDIES
CRITICAL DEVELOPMENT SURVEY STUDY
MATERIAL & PARTS EVALUATION
SOURCE SELECTION
DEVELOPMENT & TESTS
PREFERRED PARTS LIST
MATERIAL SPECS
PARTS SPECS
INTERFACE TRADE-OFFS & DEFINITION
EQUIPMENT SPECS
SPECIAL PURPOSE TEST EQUIP DETERMINATION
FAILURE MODE STUDY
SELECTED IN-HOUSE BREADBOARD DESIGN

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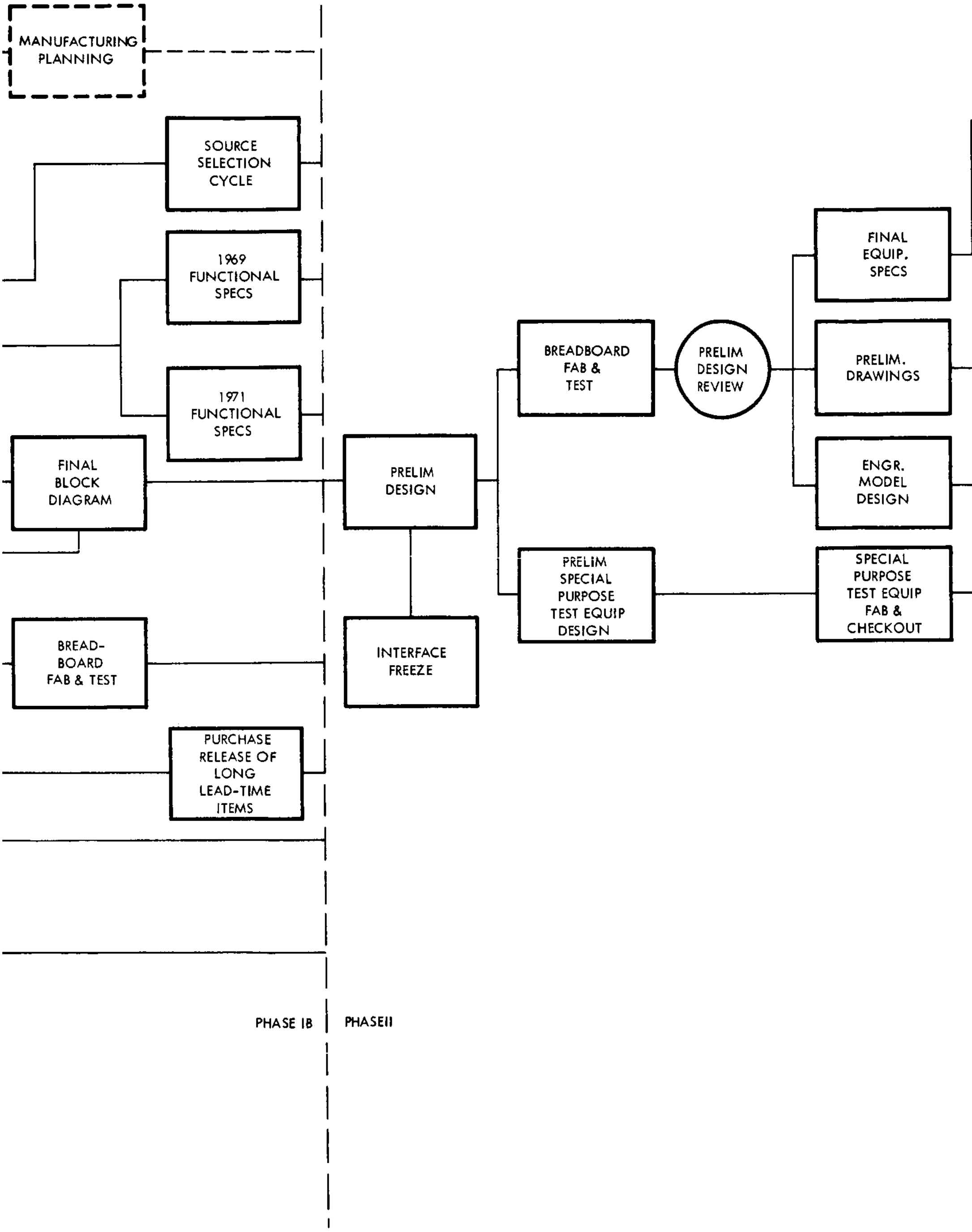
MANUFACTURING PLANNING
SOURCE SELECTION CYCLE
1969 FUNCTIONAL SPECS
1971 FUNCTIONAL SPECS
FINAL BLOCK DIAGRAM
BREAD-BOARD FAB & TEST
PURCHASE RELEASE OF LONG LEAD-TIME ITEMS
PRELIM DESIGN
INTERFACE FREEZE
BREADBOARD FAB & TEST
PRELIM SPECIAL PURPOSE TEST EQUIP DESIGN
PRELIM DESIGN REVIEW
FINAL EQUIP. SPECS
PRELIM. DRAWINGS
ENGR. MODEL DESIGN
SPECIAL PURPOSE TEST EQUIP FAB & CHECKOUT
PHASE IB
PHASEII

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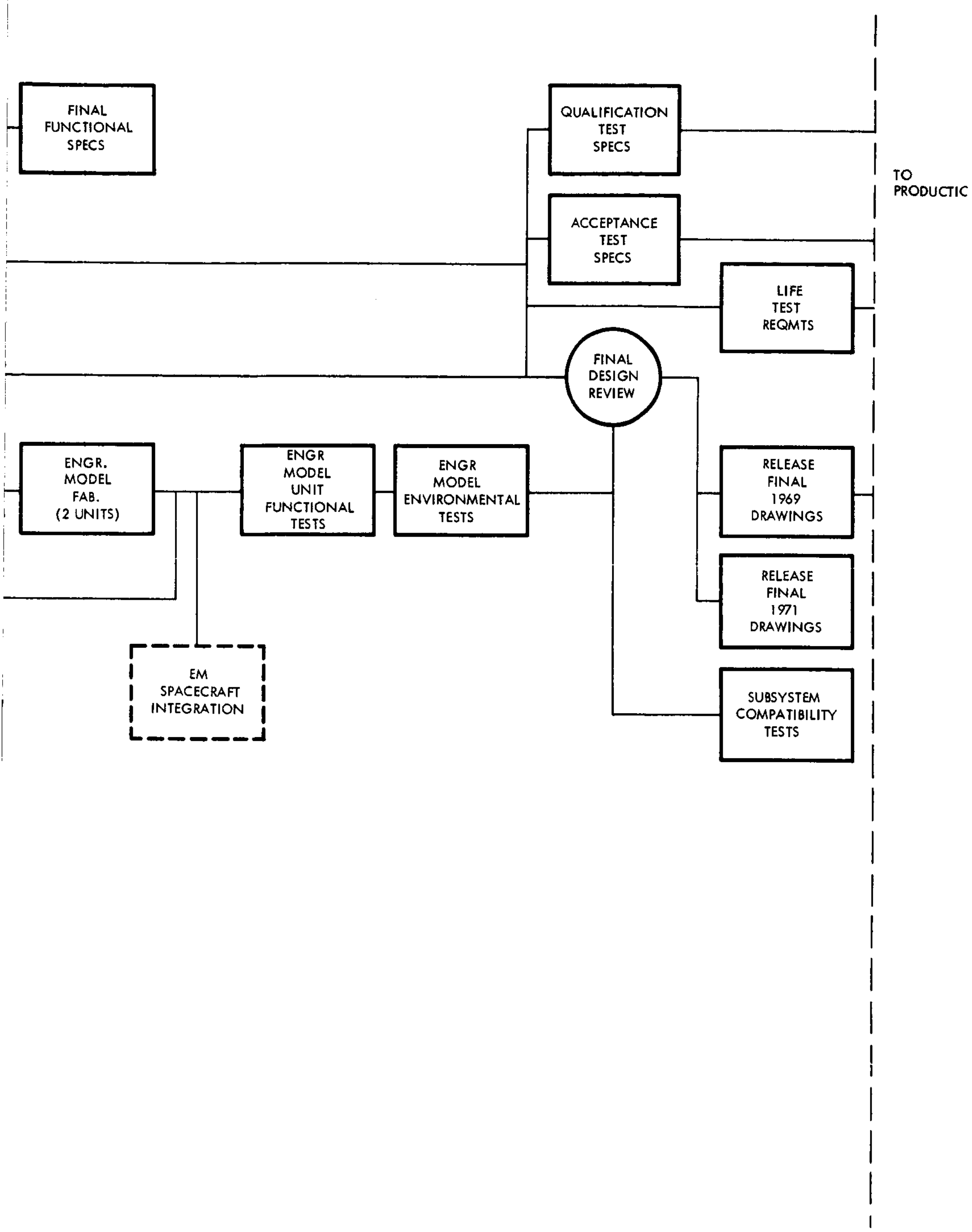


Figure 5-22. Communication and Data Handling Development

(3)

a) The servo system required to control tape speed during the playback will be investigated in detail since it is required to synchronize the tape recorded data with a signal clock. Control servo loops, and the available motors will be investigated.

b) Various techniques will be investigated for buffering the tape recorded data to synchronize it with the system clock.

c) Integrated circuits will be studied to insure maximum use in the system. Particular, a strong effort will be placed on the use of sense, DC, or differential amplifiers for recovery of data from the read heads.

Development effort will also be required in the microwave area, consisting of fabrication of several prototype horn radiators, simulation of the feed-support transmission line for each design, and measurement of characteristics of each, both in free space and in conjunction with a paraboloid reflector. Various techniques will be investigated for their suitability is suppressing undesirable radiation modes. Most of these are standard practice and will be employed in conjunction with the measurements indicated above.

The radiation pattern of the low-gain antenna system will be investigated. The requirement to provide wide coverage while providing at least 2-db gain is incompatible with a single aperture. The use of two apertures mechanically integrated but electrically separated offers the best choice of achieving the desired results without the use of switches. Electrical decoupling of one aperture from the other by 5 to 10 db will satisfy the early flight requirements. Later flight requirements will be satisfied by the primary antenna. The amount of decoupling and the angular displacement of the two apertures will be investigated. Since the pattern will be affected by the spacecraft, development tests will be accomplished with the antennas attached to a mock-up of the spacecraft; making use of a scale model of the spacecraft at the appropriately scaled frequency. Radiation patterns of various mechanical configuration will be measured, each with varying degrees of coupling between the two apertures. The configuration yielding the widest coverage with the least interference between the apertures and by the spacecraft will be incorporated into the spacecraft antenna system.

4.6.2 Analysis

During Phases IB and II, analytical studies supported by equipment analyses will be performed before communication subsystem configuration is frozen. These studies, some of which are extensions of those conducted in Phase IA, will be establish the basis for determination of subsystem performance, reliability, modulation and synchronization techniques, operational modes, and configuration requirements.

An investigation of the applicability of planetary and adaptive range codes with respect to efficiency, acquisition time, resolution, and accuracy will be conducted. The more efficient adaptive codes would permit ranging with lower power gain, as well as shorter acquisition time.

The intermodulation effects in a two-channel system occasioned by filtering will be studied. The composite data-pulse-sync signal will suffer intermodulation distortion in passing through RF and IF filters; these effects on data and sync will be assessed. Both the telemetry and command channels will require investigation.

An extension of the analysis performed in Phase IA on PN synchronization acquisition for telemetry will be required. A comparison of the offset frequency technique versus automatic acquisition by code-stepping will be investigated. Acquisition time, efficiency both in communication power requirements and equipment complexity, and probability of acquiring are the significant comparison parameters. 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 1/3 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. For example, if failure to lock on an initial sweep can be recognized, the sweep can be accelerated until the vicinity of the next lock point is reached. This will improve the effect of the high probability of failure-to-lock on a single trail.

An extended analysis of the effects of practical filters on PN synchronization will be performed. This study will more exactly develop the degradation in the PN sync loop error function caused by various bandwidth restrictions. The carrier tracking loop introduces a high-pass characteristic ("droop") in the pulses. DSIF telemetry 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.

Additional study beyond that made in Phase IA 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. Consideration will be given to the best choice of data subcarrier frequency, considering the lower limit set by carrier loop tracking and the upper limit set by subcarrier phase jitter. Once the data subcarrier frequency is known, 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 addition, an extension of the carrier-data, including sync power division optimization discussed in Appendix D, Volume 5, will be needed. The analyses discussed in Appendix D must be extended to a two-channel system where data and sync are separately affected by a noisy carrier reference.

Current power budgets for the three different links involved during the Phase IB and II programs will be maintained. These budgets will be updated periodically as more data on the subsystem becomes available.

A transponder spurious response analysis will be performed. The response of the frequency tracking loop will be investigated, taking into account the possible interference modes of the spacecraft receiver which may arise from self-generation of undesired responses and cross-coupling between receivers and between transmitter and receiver. In addition, the phase distortion will be studied to apportion the distortion budget between the various subsystem equipments.

Over-all reliability analyses will be made in the following areas:

a) A reliability assessment will be conducted to obtain estimates for each subsystem within the communication subsystem for the purpose of determining the reliability of the individual links. The reliabilities are computed using parts list information and best failure rate information in connection with actual subsystem configuration, including all redundancy, along with the established stress levels.

b) Parts evaluation will be conducted to establish a preferred parts list. The evaluation will give consideration in terms of environment, magnetics, shelf-life, parameter drift, and operating life.

c) A failure mode and effect study will be performed to to estabilish redundancy requirements and total communication subsystem failure modes. The study will be based upon the calculated reliability and mission requirements. Different redundant configurations within existing constraints will be investigated.

d) Circuit analysis will be performed on each subassembly within the communication subsystem to prove worst-case to end-of-mission operating requirements are satisfied. Design data and breadboard test results will be used as primary input for these analyses.

A packaging and layout analysis including thermal and RF shielding studies will be performed to determine the best construction and fabrication to insure structural integrity, ease of reproducability, assembly, and test.

An analysis will be conducted to determine the effects of various types of errors in the construction of the high and medium-gain antennas. This analysis is for the purpose of evaluating the effect of random and periodic errors on the gain and sidelobe level of the secondary pattern. In addition, a study will be made to determine cone and clock angles of the spacecraft with respect to earth, and spacecraft with respect to Mars, for all possible trajectories from lift-off through Mars orbiting. The variation of these angles as a function of time will determine the exact coverage required on the various antenna subsystems to satisfy mission requirements.

4.6.3 Design

The design effort will be divided between Phases IB and II; the subsystem and individual equipment design specifications, including preliminary interfaces, will be determined during IB, and detailed equipment design will be completed during Phase II. Key areas will be studied with the initiation of some breadboard designs during Phase IB, e.g., TWT, tape recorder, and selected antenna elements.

Some preliminary design studies will be necessary in Phase IB to establish approved preferred parts lists and to determine where new parts and material specifications will be required. The preparation of these specifications will be initiated during the second half of Phase IB.

a. S-Band Receiver, Exciter-Modulator, and Low-Power Amplifier

In consideration of reliability, risk, and schedule requirements, a transponding system will be selected from an existing design or as an adaptation of an existing design, e.g., those for LEM, Apollo CSM, Mariner C, Lunar Orbiter, or Pioneer. Factors involved in the selection will be performance, packaging constraints, modifications needed to meet magnetic cleanliness requirements, ethylene oxide compatibility, and acceptability of existing parts against those established for Voyager. One of the above sources, the Pioneer transponder alone was designed to meet magnetic cleanliness requirements, whereas only the Mariner C transponder has had space flight experience.

It is planned that a thorough program of investigation be conducted on two or more transponder designs during Phase IB, to encompass the following:

1) Analysis of design changes and compromises required to accommodate Voyager performance specifications, Voyager approved parts list, ethylene oxide sterilization, and magnetic cleanliness

2) Evaluation of the qualification requirements for critical nonstandard parts

3) Evaluation of the manufacturers' processes and controls, down to the part level

4) Development and evaluation of engineering breadboards and models of modified design areas

Implementation will require a competitive bid program early in Phase IB with an award to the several manufacturers having the best promise of hardware success. One of the designs evaluated will then be selected in Phase II for detailed design, fabrication, test, and flight hardware delivery. No schedule problems are anticipated during Phase II. The Lunar Orbiter transponder development, an extension of the Mariner C design, required 11 months to prototype qualification.

b. Power Amplifier

Twenty-watt TWTA's have been qualified for the Apollo program. However, a study will be required in Phase IB to assess the reliability of these tubes in view of Voyager mission requirements. Consequently a reliability test program will be initiated early in Phase IB to assure that there are no problems associated with the various flight spacecraft failure modes and environments.

c. Command Detector

No problem areas are foreseen in the design of the command detector except the magnetic cleanliness and parts qualification exercise which applies to all elements of the subsystems. Phase II offers no schedule problem, prototype qualification occurring within 14 months.

d. VHF Receiver

The implementation requirements for the VHF receiver will depend to some extent on the type of link established for capsule-spacecraft communications. However, spacecraft AM and FM receivers are in the indistrial inventory and, other than magnetic cleanliness and parts analysis during Phase IB, little development is required.

e. Data Handling

The design effort in the data handling subsystem will commence with the evaluation of existing microcircuit modules with respect to

Voyager requirements to establish whether modifications are needed. Some new or special circuit modules may have to be developed to meet the design requirements. In addition, a standardization study will be conducted to minimize the number of different types of modules and the operation of modules will be analytically and experimentally verified over the temperature range.

Based on the required encoding accuracy the number of bits and techniques for A-D conversion will be re-evaluated. The present scheme is based on 7-bit conversion accuracy and gated comparators, but a 6-bit system with simple diode gating might be sufficient. During Phase IB, a preliminary detailed subsystem block diagram will be prepared within the constraints of weight, power, flexibility and reliability. Special consideration will be given to re-examining the formats and modes established in Phase IA in view of new information on the experiments and engineering measurements. During the early part of Phase II, the detailed design will be completed and breadboard testing will be conducted.

f. Data Storage

Early in the development of the recorder the interface must be defined in detail, including the input and output data signals as well as the control functions, clock, and synchronizing signals. The means for commanding the recorder into its various modes of operation, will be studied together with techniques for controlling the tape recorder.

The requirement to synchronize stored data with the main clock requires attention in the design of the drive system. The speed changes required will need special attention. Studies will be made to decide whether belt transmissions, clutches, or orther techniques should be used to meet the read-and-write drive requirements. The selection of the drive motor will be coordinated with the choice of the servo scheme and will involve a survey of the motor manufacturers to seek the most reliable motor.

Integrated circuits, where proven, will be used for the read-and-write amplifiers, and logic and control circuits. At the present time, end-of-tape sensors in satellite recorders are not considered reliable enough for the Voyager program; it is possible that redundant techniques are the only solution.

g. Antenna Subsystems

In Phase IB initial study of Voyager antennas will be centered upon the theoretical aspects of large aperture antennas, with emphasis the constraints imposed by the electrical performance of the feeds, transmission lines, and actuator mechanisms under the influence of the environments. The basic structure as well as the surface tolerance requirements will be established. Analyses of the various structures under the influences of thermal, vibration, acceleration, and shock loading will be completed and their electrical performance will be determined analytically.

Breadboard activity will include investigations of the antenna patterns of the low-gain and VHF antennas on a scale model of the spacecraft, as well as full-scale models of the low-gain, VHF, and feed horns for the paraboloids. Pattern, gain, and impedance data as well as axial ratio measurements will be obtained from the full-scale model. Some full-scale breadboarding will be required of the paraboloid and drive mechanism.

Engineering models of the antenna subsystems will be fabricated and tested from the engineering model drawings. Complete testing of all portions of the subsystems will be performed to allow final design specifications to be written. The data to be acquired will include antenna patterns, absolute gain, impedance, axial ratio, efficiencies and insertion losses, coupling measurements, and testing under environments which are felt to be critical loadings for the components. The engineering models will be assembled into the subsystems and tested as complete assemblies as well, to provide functional data.

The gimballing of the high and low antenna assemblies is planned to be accomplished by adopting the OGO solar panel drive to the Voyager application. Since the Voyager requirements are similar to those of OGO, and flight experience and life testing have been accomplished with good results, no major problems are anticipated. The electronic circuitry will also be based upon OGO experience.

4.6.4 Magnetics

The communication subsystem will be divided into two sections for magnetic considerations, those units that are the same or similar to units flown on other spacecraft and not considered problem, and those that are a problem.

Falling into the first category are such assemblies as the receivers, command detectors, modulator exciters, DC converters to power the RF amplifiers, demodulators, VHF preamplifiers, signal conditioner, and core storage unit. These assemblies are not a problem in the sense that acceptably small magnetic fields can be obtained (4 to 8γ at 1 foot) if careful parts screening and material control is instituted concurrent with initial breadboard design. Modification and parts substitution in completed units may result in need for extensive redesign.

The remainder of the various assemblies in this subsystem will be approached as potential magnetic problems. These are discussed below.

a. Power Amplifiers

Although the TWT is listed in the problem area, the success in compensating similar assemblies on such programs as Pioneer, along with the careful positioning and rotation of the unit on the spacecraft in relation to the magnetometer sensor, can result in fields of 0.1γ at the sensor. If a klystron is used on later missions to obtain higher RF power levels, the lack of magnetic focusing for such a unit reduces this to a normal assembly involving only kovar to glass sealing.

b. Circulators

Similar circulators flown on Mariner have exhibited a field of 7.5γ at 12 inches, the majority of which is stray field from the energizing current required to hold the switch in a preferred position. It is not known if the magnetic field can be reduced by better shielding or by magnetic compensation. Further studies will be made on this assembly.

c. Antenna Assemblies

Although the antenna dishes are expected to be nonmagnetic, the means of orienting these dishes involves torque motors and a magnetic pickoff. Reduction in the magnetic field of these assemblies is expected, by careful control of the motor windings to minimize the leakage fields, matching of the permanent magnets, and using preferred shielding and compensation techniques.

d. Digital Telemetry Units

Integrated circuits will be used extensively in the digital telemetry units. Studies of the magnetic properties of various types of circuits from four different suppliers indicate a magnetic field of 21γ at 3 inches after magnetization. Since the majority of this is due to the case and leads, it is probable that an optimum type of packaging using a nonmagnetic material can reduce the field to that caused by the leads. This could be minimized by trimming back lead lengths to something less than 1/8 inch.

e. Tape Recorders

Tests on the OGO recorder show them to be quite magnetic. Within the tape transporter, three magnetic latching relays, a DC erase head, and a negator spring (used for tape tension between the feed and take-up reel) were the main contributors. With solid state switching to replace the relays or by shielding and compensating these relays together with changing to an AC type erase head and using a nonmagnetic material for the negator spring will probably reduce this unit to the magnetic limits.

4.6.5 Test

Development tests will be used to corroborate the analyses investigate and to verify that over-all system requirements are met. Two complementary sets of development units will be used, breadboard and engineering models.

a. Breadboard Test

In the microwave area, breadboard activity will progress in the form of scale model testing during Phase IB. This testing will consist of evaluating the low-gain antenna patterns using a scale model of the spacecraft. In addition, full-scale experimental testing of the feed horns for the parabolic antenna will be conducted during Phase IB to determine the efficiency of illumination of the aperture and the leakage energy through the aperture surface.

Early in Phase IB, and continuing into Phase II, extensive engineering reliability testing of the traveling wave tube will be carried out. The tests to be performed will establish DC power supply interface requirements, operational failure modes, and their effects on reliability.

In support of the analysis of PN synchronization acquisition, some experimental laboratory testing will be required to investigate acquisition with respect to possible distortion by the phase-lock loop bandwidth under strong signals condition.

A selected and limited amount of module circuit breadboard testing in the data handling area will be conducted during the latter half of Phase IB. Investigation of certain microelectronic components will be evaluated during these tests so that an early design on new modules can be expedited at the start of Phase II.

During Phase IB almost all units for the 1969 test flight will require some breadboard testing if final drawings are to be released for 6 to 9 months after the start of Phase II. The breadboard circuit tests outlined in the test matrix, Table 5-12, will consist of low and high qualification temperature levels, to ascertain conformance to their appropriate equipment specifications.

Table 5-12. Communications and Data Handling Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description
Circuit design evaluation	Breadboard	Determine electrical performance to verify that design conforms to component performance specifications.	Measure operation parameters with specified range of input conditions at ambient temperature.
Design Verification	Breadboard	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.
Special antenna evaluation	Scale model antennas	Determine with scale models the antenna coverage and directivity levels to ascertain that mission requirements are met.	Perform tests in conjunction with a scale model spacecraft on which model antennas are mounted.
Initial electrical performance	Engineering model	Determine electrical performance to verify that engineering model conforms to component performance specifications.	Measure operating parameters with specified range of input conditions at ambient temperature.
Magnetic	Engineering model	Determine magnetic field intensity to verify below acceptance limit.	Measure magnetic field intensity in three planes while operating in each mode test while magnified and demagnified.
Ethylene oxide compatibility	Engineering model	Verify resistance to ethylene oxide environment.	Expose engineering model to ethylene oxide gas at specified concentration and temperature for specified period.
EMC	Engineering model	Verify conformance to EMC performance specification.	Measure level of susceptibility to and generation of conducted and radiated radio frequency interference.
Electrical performance	Engineering model	Determine electrical performance before, during, and after environmental exposure to verify conformance to specification	Measure operating parameters with specified range on input conditions.
Environmental	Engineering model	Verify integrity of design for qualification environments	Subject engineering model to thermal-vacuum vibration, sustained acceleration, shock, and humidity environments.
Special antenna design verification	Full-scale antenna engineering model	Determine efficiency of illumination of aperture, side-lobes levels, directivity through integration of patterns, and leakage energy through the aperture surface.	Measure performance parameters on a full-scale directional or semi-directional antenna without the influence of spacecraft presence.
Qualification	First production flight unit	Demonstrate flight units will function properly at design qualification limits.	Impose qualification environmental limits on a unit monitoring operating characteristics to insure proper operation or proper survival, whichever is applicable.
Life	Flight units	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	Unit engineering models 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.

Next the breadboard units will be interconnected to check critical subsystem compatibility, mutual interference, DC voltage and signal variations, and presence of spurs. The breadboards will be used throughout the design effort to conduct special tests required by design modifications or performance changes.

b. Engineering Model Test

The engineering model testing will encompass all performance and environmental tests required to verify the adequacy of the design.

The drives for gimballing the parabolic antennas will be fabricated and tested in the engineering model configuration. There are no plans for breadboard testing these drives; over 10,000 hours of life testing has been accumulated on the drive to date.

Since two engineering models of each unit will be fabricated, plans are to test one as a unit and the other as a part of the engineering model spacecraft system. The engineering model tests afford early evaluation of flight configuration interface design and over-all integrated performance. The first engineering model fabricated will be designated for the unit tests. It will be inspected for mounting, connectors, dimensions, weight and center of gravity locations. The units are then tested in accordance with the test matrix. At the conculsion of the unit level testing, the individual units will be integrated to form partial, or complete subsystems and subjected to compatibility testing.

For the subsystem test setup breakout cables between boxes will expedite testing. Units are tested in flight spacecraft layout configuration so that proper lengths of coaxial cable can be utilized for determining line losses. Successful performance of the subsystem tests will confirm satisfactory subsystem operation and reduce the possibility of incompatibility problems with other subsystems.

Life testing on the final choice of low-noise preamplifiers to be evaluated will be started during the second half of Phase IB can be made at the outset of Phase II.

4.6.6 Schedule

The communications and data handling milestone schedule is depicted in Figure 5-23. To meet the 1969 need dates the manufacturing drawings are required 6 to 9 months after Phase II go ahead. Prior milestones have evolved from this constraint. As soon as the subsystem requirements are reasonably defined, a conceptual design approach will be developed.

The traveling wave tube procurement will be initiated in the second month of Phase IB so that reliability testing on the TWT can begin no later than the fifth month and completed by the end of Phase IB.

Critical items of development planning will be completed shortly after the beginning of Phase IB. The subcontractor for the development of the tape recorder will be chosen within the first few weeks of Phase IB. This item is extremely important from the aspect of having a breadboard model completely fabricated and tested by the end of Phase IB. The key to this critical area is to determine by the start of Phase II the preliminary design for the servo system to control the speed during playback and the technique for buffering the tape recorder data for synchronization with the system clock.

In an effort to improve the relay link performance, development will be pursued during Phase IB so that frequency uncertainties over a long period can be decreased through crystal development or oscillator circuit stability advancements.

In addition, it will be necessary to commence scale model spacecraft antenna fabrication, low-gain antenna subsystem tests, and full scale feed horn model testing during the five months of Phase IB in order to meet the 1969 flight need date on a timely basis.

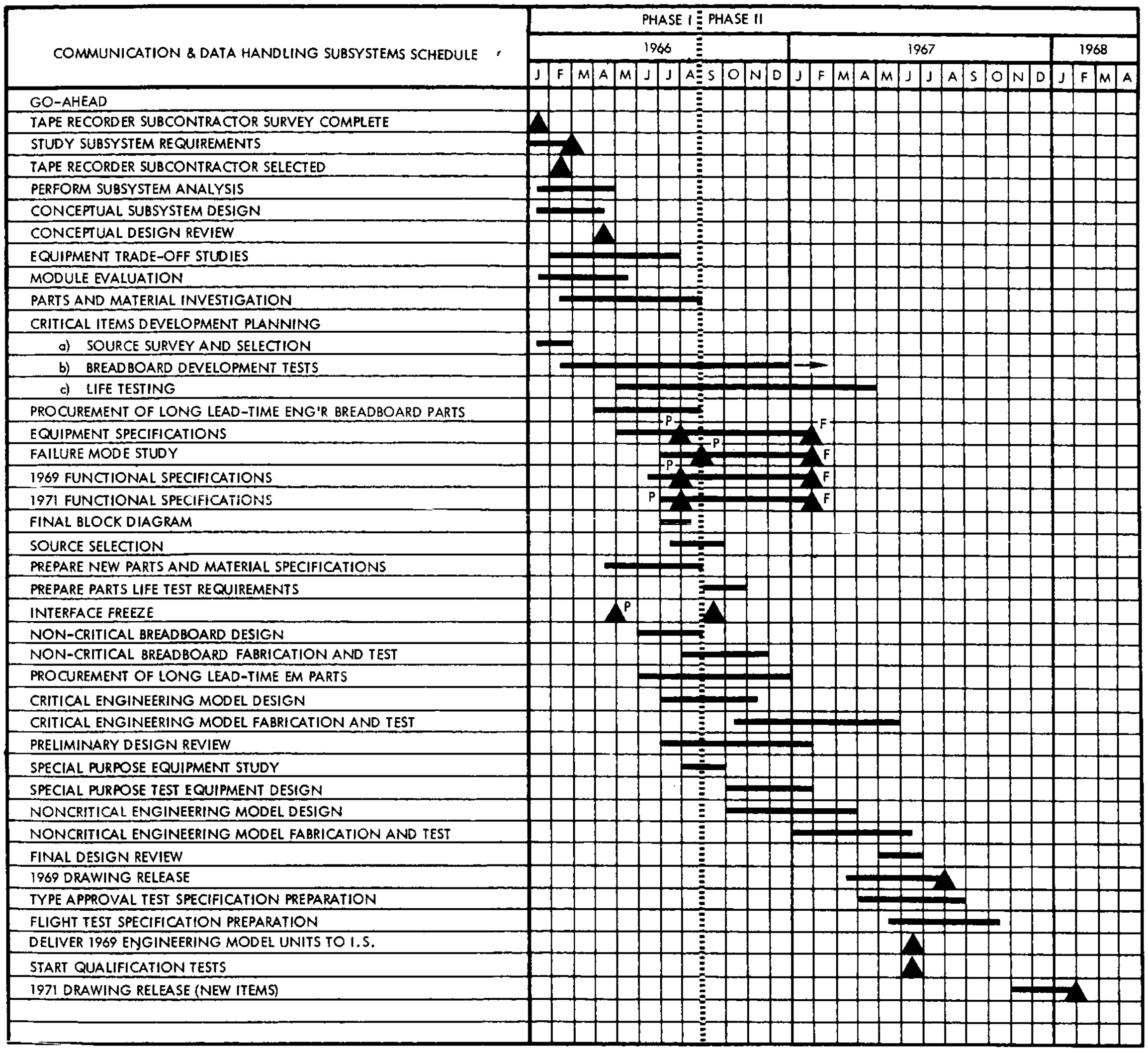


Figure 5-23. Communication and Data Handling Subsystems Schedule

4.7 Power Subsystem

4.7.1 Summary

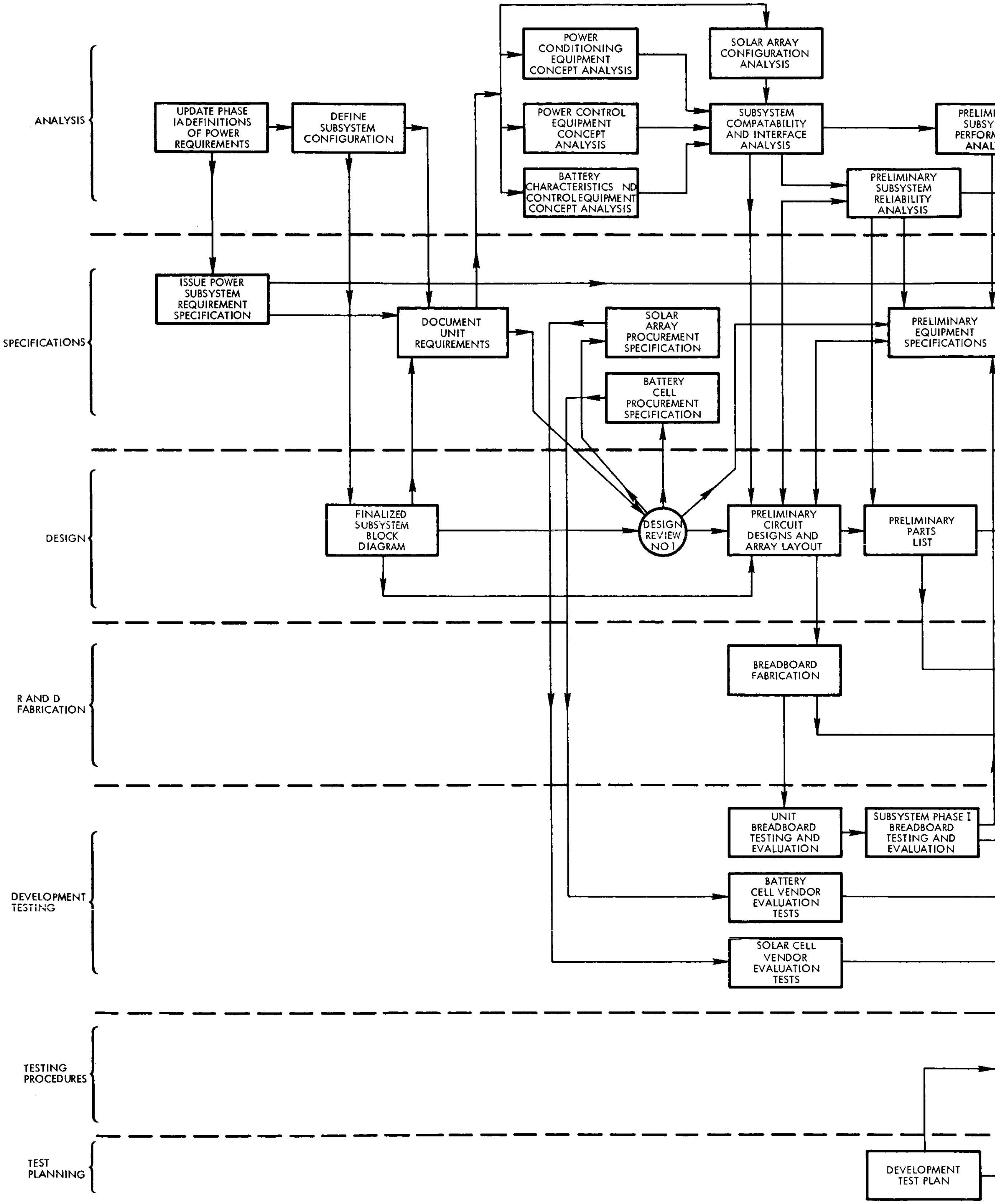
In the development of the spacecraft power subsystem TRW proposes to be the source for the battery pack, power control unit, shunt elements assembly, battery regulator, and the inverters; it is proposed to subcontract the design, development, and fabrication of the solar array to RCA. The development of the subsystem consists of the activities shown in Figure 5-24 on the schedule in Figure 5-25.

a. Development Problems and Approach

In the preliminary design of the power subsystem, it has been assumed that the sun will be eclipsed after the first month in orbit at Mars, and that these eclipses may be as long as 2.3 hours. The expected low temperature which will be reached by the solar array during the longest eclipses (approximately $-160^{\circ}C$) is a problem which requires careful attention during Phases IB and II. TRW has been faced with similar problems in the OGO program, where array temperatures of $-160^{\circ}C$ were expected under certain orbital conditions. OGO solar panels have been qualified to $-140^{\circ}C$. However, only three thermal cycles were required in the OGO qualification specification, whereas Voyager will experience a much larger number of eclipses during its six months' life in orbit at Mars. Similarly, RCA has qualified solar panels for the Lunar Orbiter program down to $-120^{\circ}C$, for up to 600 thermal cycles.

The low temperature problem will be approached in Phase IB through an engineering sample testing program designed to evaluate the temperature cycling behavior of sample cell modules, bondings, and substrates. Several options will be available in the event that the desired low temperature qualification is not achieved by the Phase IB freeze date:

- Suffer the power, weight, and size needed to keep the array warmer during the longest eclipses.


ANALYSIS
SPECIFICATIONS
DESIGN
R AND D FABRICATION
DEVELOPMENT TESTING
TESTING PROCEDURES
TEST PLANNING
UPDATE PHASE IA DEFINITIONS OF POWER REQUIREMENTS
DEFINE SUBSYSTEM CONFIGURATION
POWER CONDITIONING EQUIPMENT CONCEPT ANALYSIS
POWER CONTROL EQUIPMENT CONCEPT ANALYSIS
BATTERY CHARACTERISTICS ND CONTROL EQUIPMENT CONCEPT ANALYSIS
SOLAR ARRAY CONFIGURATION ANALYSIS
SUBSYSTEM COMPATABILITY AND INTERFACE ANALYSIS
PRELIMINARY SUBSYSTEM RELIABILITY ANALYSIS
ISSUE POWER SUBSYSTEM REQUIREMENT SPECIFICATION
DOCUMENT UNIT REQUIREMENTS
SOLAR ARRAY PROCUREMENT SPECIFICATION
BATTERY CELL PROCUREMENT SPECIFICATION
PRELIMINARY EQUIPMENT SPECIFICATIONS
FINALIZED SUBSYSTEM BLOCK DIAGRAM
DESIGN REVIEW NO 1
PRELIMINARY CIRCUIT DESIGNS AND ARRAY LAYOUT
PRELIMINARY PARTS LIST
BREADBOARD FABRICATION
UNIT BREADBOARD TESTING AND EVALUATION
SUBSYSTEM PHASE I BREADBOARD TESTING AND EVALUATION
BATTERY CELL VENDOR EVALUATION TESTS
SOLAR CELL VENDOR EVALUATION TESTS
DEVELOPMENT TEST PLAN

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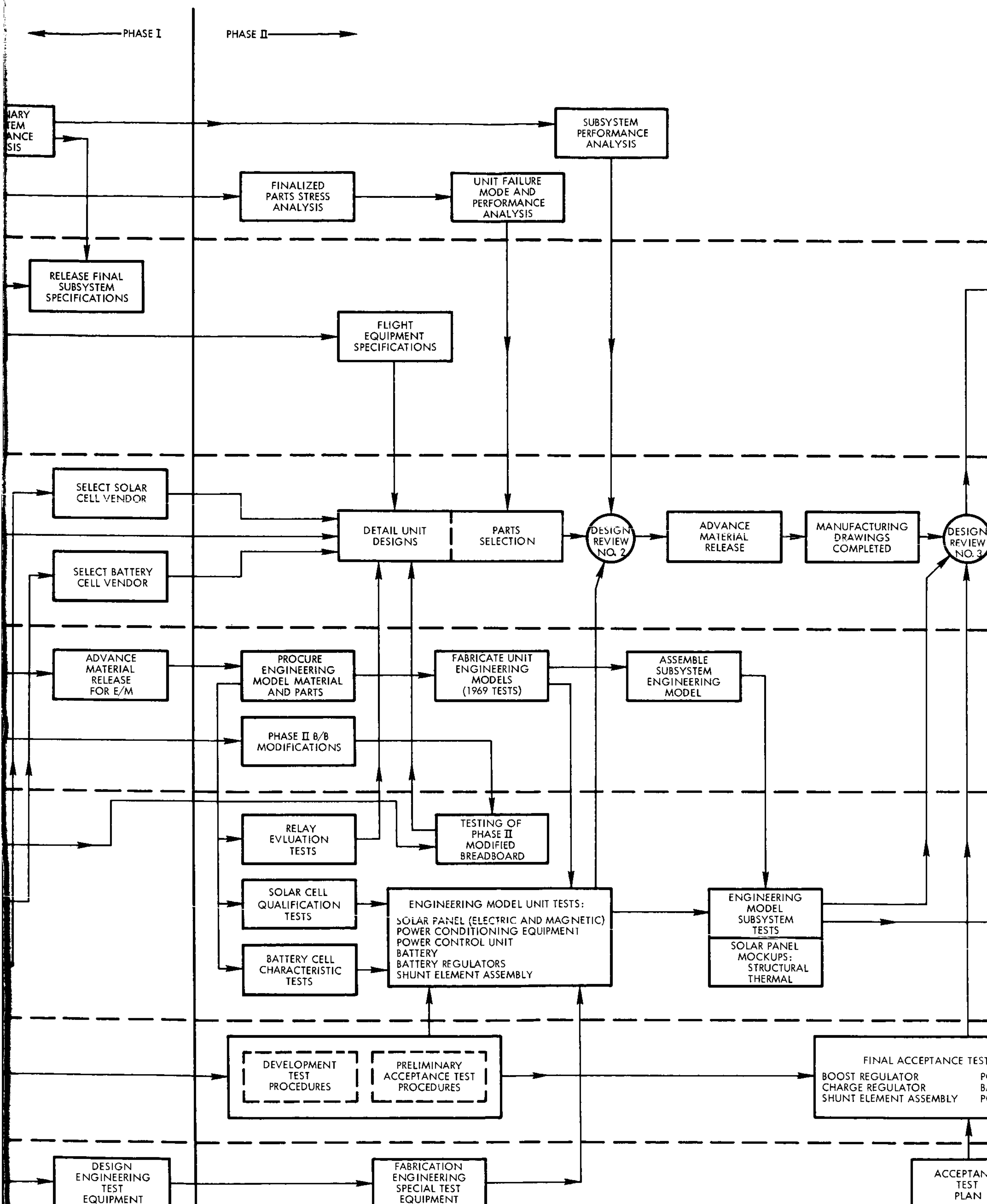
PHASE I
PHASE II
...NARY ...TEM ...ANCE ...SIS
SUBSYSTEM PERFORMANCE ANALYSIS
FINALIZED PARTS STRESS ANALYSIS
UNIT FAILURE MODE AND PERFORMANCE ANALYSIS
RELEASE FINAL SUBSYSTEM SPECIFICATIONS
FLIGHT EQUIPMENT SPECIFICATIONS
SELECT SOLAR CELL VENDOR
DETAIL UNIT DESIGNS
PARTS SELECTION
DESIGN REVIEW NO. 2
ADVANCE MATERIAL RELEASE
MANUFACTURING DRAWINGS COMPLETED
DESIGN REVIEW NO. 3
SELECT BATTERY CELL VENDOR
ADVANCE MATERIAL RELEASE FOR E/M
PROCURE ENGINEERING MODEL MATERIAL AND PARTS
FABRICATE UNIT ENGINEERING MODELS (1969 TESTS)
ASSEMBLE SUBSYSTEM ENGINEERING MODEL
PHASE II B/B MODIFICATIONS
RELAY EVLUATION TESTS
TESTING OF PHASE II MODIFIED BREADBOARD
SOLAR CELL QUALIFICATION TESTS
ENGINEERING MODEL UNIT TESTS:
SOLAR PANEL (ELECTRIC AND MAGNETIC)
POWER CONDITIONING EQUIPMENT
POWER CONTROL UNIT
BATTERY
BATTERY REGULATORS
SHUNT ELEMENT ASSEMBLY
ENGINEERING MODEL SUBSYSTEM TESTS
SOLAR PANEL MOCKUPS: STRUCTURAL THERMAL
BATTERY CELL CHARACTERISTIC TESTS
DEVELOPMENT TEST PROCEDURES
PRELIMINARY ACCEPTANCE TEST PROCEDURES
FINAL ACCEPTANCE TEST...
BOOST REGULATOR
CHARGE REGULATOR
SHUNT ELEMENT ASSEMBLY
DESIGN ENGINEERING TEST EQUIPMENT
FABRICATION ENGINEERING SPECIAL TEST EQUIPMENT
ACCEPTAN... TEST PLAN

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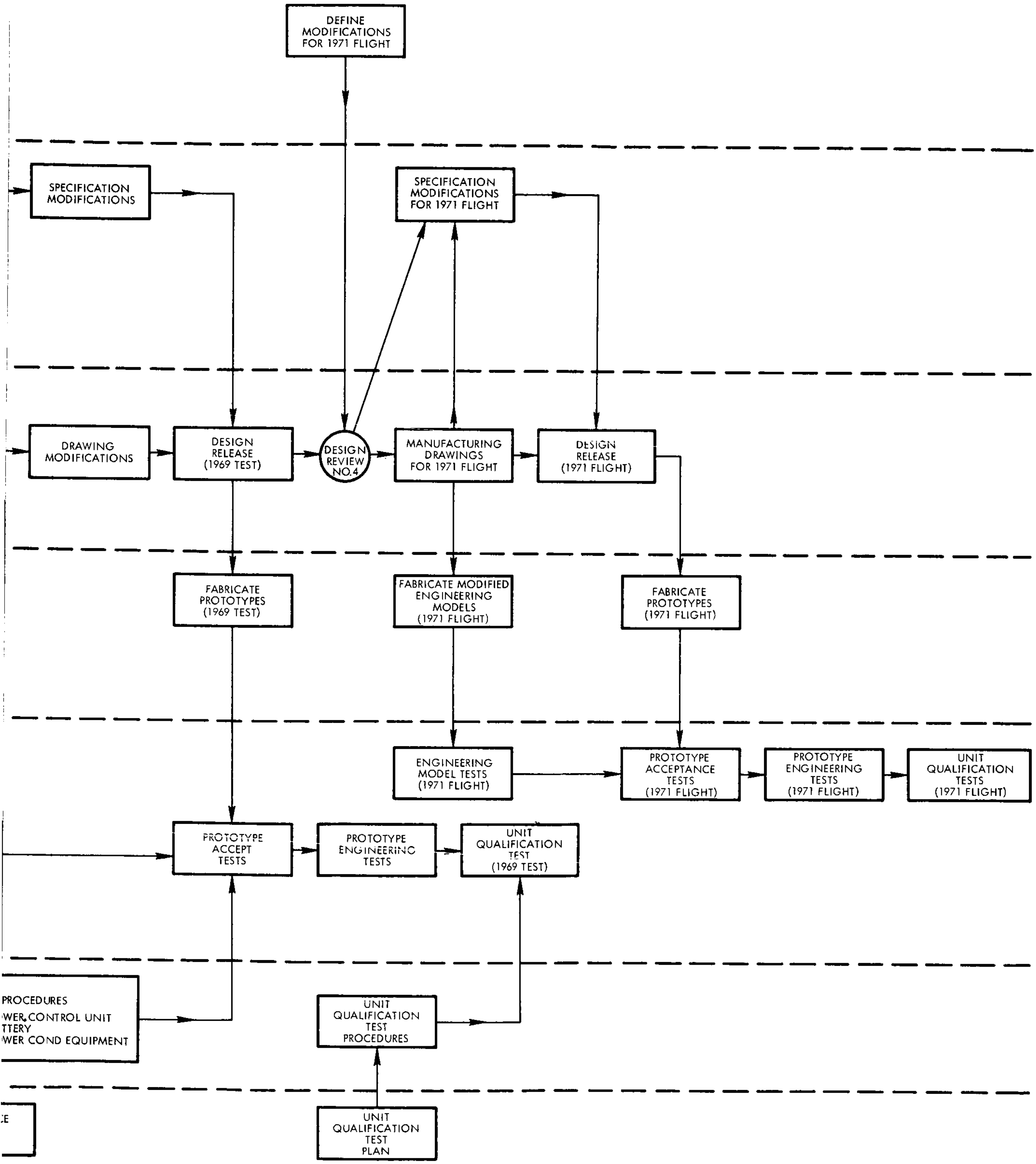

Figure 5-24. Power Subsystem Development

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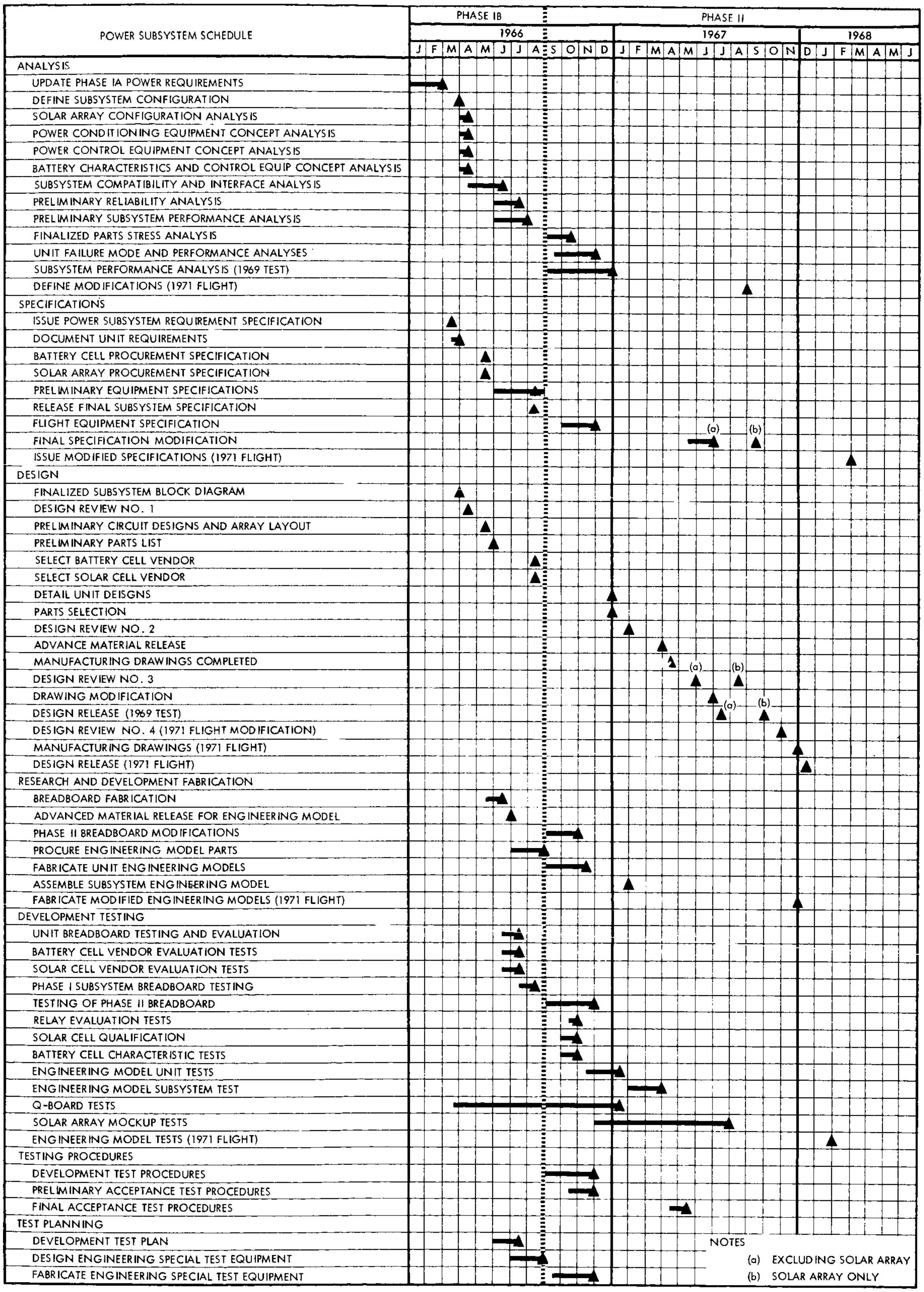


Figure 5-25. Power Subsystem Development Schedule

- Adopt a more efficient design in a tradeoff of increased performance and mission capability during the early months against the probabilities of achieving an orbit which occults the sun during the later months

- Apply a compromise approach to the 1969 mission and continue design efforts with the expectation of finding an acceptable low temperature design before a 1971 configuration freeze

The latter could result in a 1971 design which has not been flight evaluated in 1969, although it is likely that the major design features will be common.

4.7.2 Analysis and Design

The Phase IA analysis of the total subsystem will be refined to include any revised system requirements. Revised requirements for each unit of the subsystem will be issued, and a subsystem specification will be released including updated power requirements and data and interface criteria. The subsystem electrical interfaces will be defined at the schematic level, including unit testability after spacecraft installation, interconnection with the electrical integration subsystem, and other spacecraft equipment. The performance analysis will include steady-state and transient operational analysis, failure mode analysis, and reliability assessments. This analysis is updated throughout the development phase. Finally the formal final subsystem block diagram, specification and performance analysis reports are released.

a. Solar Array

Preliminary solar cell, cover glass, and module specifications will be generated and sent to potential suppliers of solar modules. Briefing sessions will be held with vendors regarding fabrication techniques, design, and costs. A source will be selected and justified.

Characteristic I-V curve data will be measured for typical cells and modules supplied by the vendor. Based upon updated power requirements, a preliminary solar array configuration will be established, analytically degrading the typical I-V curves to yield design array output for beginning of life, cruise, and orbit around Mars. A parametric study of the orbit, time after insertion, and panel design characteristics will be made. Estimation of anticipated radiation effects on the solar array will be made using available data, which should include results of the Mariner 4 mission.

A detailed preliminary reliability analysis will be performed on the specific array design taking into account the failure modes and their effects on array performance from launch to end-of-life.

From the array configuration analysis and structural interface inputs, a preliminary array layout will be made to achieve the required number of modules in series, and the required number of parallel module strings. The layout will also include preliminary detailed wiring between module strings, diode packaging and assembly details, thermistor details for temperature telemetry, and voltage and current telemetry component boards. Intermodule wiring will consider magnetic moment effects. The layout and a preliminary parts list will be generated by Design Review No. 2 at the end of the fifth month of Phase II.

Analysis will be supported by testing of Q-boards and panel mockups.

b. <u>Battery</u>

The battery load requirements will be revised to provide up-to-date inputs to the design analysis, which results in the preparation of final interface definitions, and battery and cell specifications. These specifications, together with supporting instructions and drawings, will constitute a preliminary design. Battery magnetic moment effects will be minimized in the design by proper arrangement of cell orientation and intercell wiring and connections.

Evaluation cells will be procured for performance verification tests. Data from these cell tests will aid in the preparation of cell acceptance, battery acceptance, and qualification test procedures.

Battery packaging layouts will then be initiated, concurrently with thermal and structural analysis. Production drawings will not be released until completion of engineering model environmental tests.

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) <u>Seal Failures</u>

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

2) <u>Short Circuit Failures</u>

- Misalignment of one of the electrode placques
- Silver migration on ceramic insulator
- Insulator breakdown
- Impurities from fabrication processing
- Flaking plate material due to improper heat treating and/or excess material
- Battery connector shorts

3) <u>Open Circuit Failures</u>

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

These failure modes will receive an engineering analysis 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

Power control includes three units designated as the power control unit, shunt elements assembly (SEA), and the battery regulator.

Power Control Unit. The power control unit provides voltage sensing and error signal amplification to control the SEA; and battery regulators, sensing, logic and relays for control of redundant power system units; synchronization signals; and telemetry monitors of current and voltage. The proposed mode of array voltage control is within the capability of existing TRW developed circuit concepts. The supplementary functions related to control of redundant units, conditioning of telemetry signals, and generation of synchronization signals also present no new development problems. Upon definition of specific system requirements, tradeoff studies will be made to enable detailed circuit and module designs. The array shunt point and shunt element dissipation requirements are dependent on the loads profile, the number of parallel connected array sections, and the output characteristics of the array under various operating conditions.

Thermal analysis of the PCU establishes maximum component temperature levels at critical modes of operation. Failure modes analysis includes effects of both open and short circuit conditions under worst case voltage, current, and power characteristics of the components. An analysis 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.

Charge Regulator. A battery charge control technique has been designed, breadboarded, and 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 the cell reaches a voltage equal to or greater than the reference voltage, a signal is sent which terminates charge to the entire battery. In this way, the first cell in the battery to reach a maximum voltage level will terminate charge to the entire battery. The voltage limit at which this occurs is varied as a function of temperature. The voltage-temperature function can be varied over a wide range by appropriate selection of network components. Four basic tasks remain before a complete cell level charge control model can be constructed:

1) Current Compensation. 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. Several circuit concepts have been designed at TRW, although not yet implemented. Circuit development effort will 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). These data will be obtained using a battery characterization test program.

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 one of these cells will overcharge increases; 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 the appropriate design of voting logic for the charge control device.

4) Design of Integrated Circuitry. Because the voltage sensing modules may be designed to operate directly on the voltage of a single cell (a maximum of 1.6 volts), micro-

circuits can be used. This enables a considerable decrease in the over-all weight of the charge control system and leads to increased reliability, decreased sensitivity to temperature variations, and improved batter packaging techniques.

The voltage sensing portion of the battery charge control will be mounted on individual battery cells and packaged as an integral part of the battery. The current limiting element will be packaged as a part of the battery regulator. Engineering model battery and battery charge control tests will be performed to verify the reliability and performance of the control system design. Complete breadboard subassemblies of the battery and charge control will be constructed and tested. Charge control failure modes will be programmed and tested to determine the adverse effects upon system operation. Simulated failures of premature turn-off signal, failure to turn off, and other failures will be simulated and their effects upon the over-all system assessed.

Boost Regulator. A design study comparing circuit approaches for optimum efficiency includes active element, core, and copper loss 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. Further analysis will be made to compare with a system utilizing majority voting for the on-off control logic and a conventional boost regulator design.

Based upon these analyses and other subsystem requirements, specifications and functional schematic diagrams will be prepared. Performance requirements for each functional module 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 Analysis and Design

Power conditioning requirements will be provided by three inverter packages, with output frequencies of 4.1 kc, 820, and 410 cps.

The performance requirements are within the capability of conventional design concepts.

After final definition of load and voltage requirements, detailed design will be implemented to maximize reliability and efficiency. Inherent in the generation and distribution of square wave AC is the requirement for detailed attention to RFI supression and susceptibility.

Thermal analysis of the preliminary package configuration for each inverter will establish maximum component temperature levels and will include an assessment of the following performance parameters and design characteristics:

- Regulation
- Distortion
- Turn-on and turn-off characteristics
- Electromagnetic interference
- Size and weight
- Efficiency
- Reliability
- Component stress levels
- Thermal considerations
- Mechanical stress

These analyses will support formal design reviews, and together with other subsystem requirements will result in specifications for each of the inverters. Performance requirements for functional module assemblies will be established and module specifications issued. Detailed circuit design will proceed based on these specifications.

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 interconnections 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 attempt to cancel generated fields by proper orientation of components and modules. Conducted RFI and magnetic fields will be cancelled by utilizing twisted pairs between modules. Design analysis will be supported by breadboard and engineering model testing.

Design reviews and documentation during the development phase of the inverters will be similar to the corresponding procedures for fabrication and test.

4.7.3 Subsystem Development Tests

Subsystem development testing will be performed using both breadboard and engineering model equipment as shown in Table 5-13. Both series of tests will utilize the equipment previously used in unit development testing. Breadboard bench testing of the subsystem will be essentially concluded at the time of the second design review. Performance testing of the engineering model subsystem will have been completed prior to Design Review No. 3. As individual units, the breadboards and engineering models will have completed engineering testing prior to subsystem testing.

A solar array simulator and dummy loads will be used for both breadboard and engineering model testing. Engineering models of the battery will be incorporated into the subsystem as part of the engineering model test phase. Subsequent spacecraft integration tests will determine the total system compatibility for performance and electromagnetic interference.

a. Solar Array Development Tests

Development tests are performed to probe design uncertainties and to confirm the adequacy of analytically-derived design solutions. For the tests three complementary sets of development components are planned: Q-boards, mockups, and engineering models.

Table 5-13. Power Subsystem Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description	Test Equipment and Special Facilities
Subsystem breadboard	Breadboards: inverters, power control unit, shunt elements assembly, battery regulator	Preliminary 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	Oscilloscopes, electric meters, recorders, power supply, dummy loads, simulated solar array, test battery
Subsystem engineering model	Engineering models: inverters, power control unit, shunt elements assembly, battery, battery regulator	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	Recorders, load banks, Oscilloscopes, Oil bath, small temperature chamber, solar array simulator

Q-boards are solar panel sections manufactured to specifications employing flight processes and materials. Q-board testing, while providing the designer with means for assaying performance with minimal effort and delay, is limited by the smaller size of panels as to the types of tests which yield meaningful data. Thermal, structural, and magnetic mockups are duplicates of the flight component with respect to the particular design area. Accurate mockups will be made available for timely system level tests in each design area. Design changes suggested by the systems tests will be fed back to the design effort as early as possible.

Q-Board Tests. The prime virtue of Q-board tests is their ability to provide performance data during the preliminary design phase. The test sequence (Figure 5-26) is designed to furnish the most useful information first. The Q-board tests shown in the test matrix (Table 5-14) are intended to yield preliminary data on the materials and processes employed in solar panel fabrication. Possible degradations due to temperature extremes and temperature cycling of particular interest to the Voyager mission, are tested by subjecting Q-boards to thermal vacuum cycling. By accelerating the cycling rate, the fatigue data obtained is applicable to the mission. A Q-board will be magnetically tested for data on materials and processes. Current loops will be checked although differences from flight configuration will limit the usefulness of this data. Solar panel materials and processing will be checked for compatibility with ethylene oxide gas by exposing Q-boards to an excess concentration of the gas over a prolonged period. Static and vibration tests will validate or modify analytically-derived mechanical characteristics. The final test planned for solar array Q-boards is an exposure to humidity to determine the effects on materials and manufacturing techniques. After each environmental test, insulation resistance and I-V output characteristics under artificial illumination will be checked for degradation.

Table 5-14. Solar Array Development Test Matrix

				Test Equipment and Special Facilities
Cell evaluation	Solar cells from various vendors	Determine electrical and mechanical properties after various environmental conditions. Vendor evaluation	Measure output characteristics after temperature cycling, high temperature exposure, and high humidity environmental exposure. Determine contact strength after high temperature exposure. Mechanical and radiation tests	Solar simulator, digital voltmeters, recorders, load box, standard cells, temperature chamber, vacuum chamber microscope, weight
Coverglass mechanical evaluation	Cover slides with UV reflective filters	Determine stability of optical coatings	Measure transmittance and examine coatings after exposure to temperature cycling, humidity at elevated temperatures	Perkin-Elmer spectrophotometer, microscope, humidity chamber
Panel evaluation test - Q-board evaluation	Q-boards built in accordance with production processes	Verify integrity of design and processes under ambient and environmental conditions	Perform environmental tests thermal-vacuum test, sustained acceleration test, shock test, vibration test, in accordance with specified requirements. Measure electrical output before and after each test, and perform visual inspection. Ethylene oxide compatability test.	Tungsten light table and associated measurement equipment, temperature chamber, shake table, shock table, shock equipment, vibration table, stereomicroscope
Life evaluation	Flight type cells with filter and covers	Determine UV stability of cover, adhesives and cells	Expose cells to an accelerated UV environment	
Type approval	Type approval model	Final design verification of operational readiness	In accordance with contractually approved test procedure	

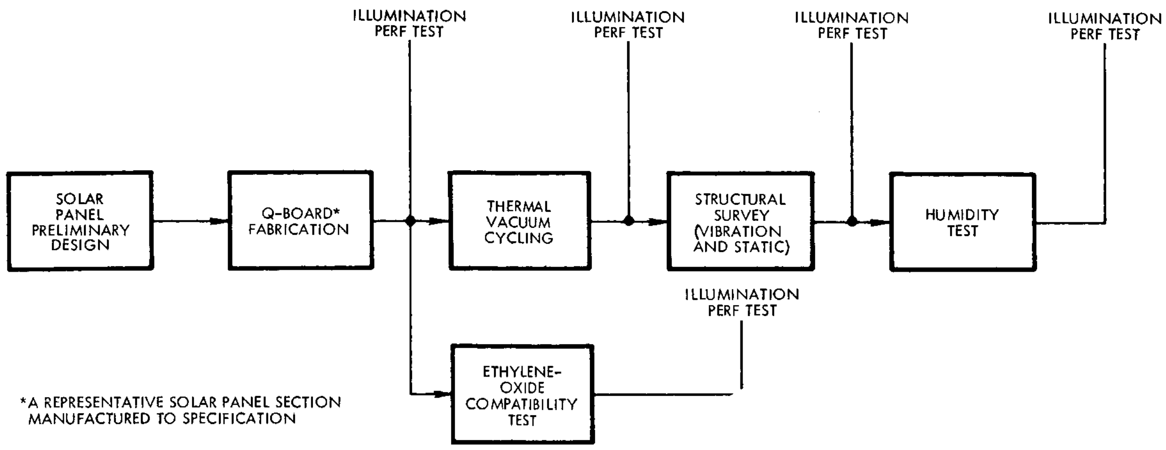


Figure 5-26. Q-Board Development Tests

Mockup Tests. Mockups will be fabricated for thermal, structural, and magnetic testing. A thermal model will be integrated and tested by TRW. Since test results are required during the development phase before flight configuration components become available, RCA will supply for this use a thermal mockup of the solar array. The thermal mockups will conform to the existing flight design in outline and mounting dimensions, and in thermal surface finish.

Two structural test dummies of the solar array will be furnished to TRW for vibration and static tests on spacecraft structural models conforming to the flight design in outline, mounting dimensions, weight and cg location, and material and fabrication

To determine the solar array magnetic field intensity, a dummy array will be provided using conducting strips in place of solar cells. The circuit paths will accurately duplicate the flight design so that the magnetic field intensity due to current loops will correlate with the flight array. Magnetic test results will be available for Design Review No. 2. Structural and thermal test results will be available for Design Review No. 3. Table 5-15 presents the characteristics and environments of tests by components of the solar array.

Table 5-15. Solar Panel Development Test and Evaluation Matrix

	Solar Cells	Cover Slides	Solar Cell Modules	Q-boards	Complete Panels
Characteristics:					
Examination of product and weight	X	X	X	X	X
V-I characteristic	X		X	X	X
Dimensional check	X	X	X	X	X
Magnetic field			X	X	X
Contact peel	X		X		
Transmittance and cut-off		X			
Environments:					
Temperature cycling	X	X		X	X
Thermal-vacuum				X	X
Vibration				X	X
Tungsten light	X		X	X	X
Natural sunlight				X	X
Humidity			X	X	X
Shock				X	X
Static load				X	X
Ethylene oxide compatibility			X	X	

b. Battery Development Tests

A quantity of battery cells will be purchased for cell evaluation tests and acceptance tested in accordance with procedures described in the cell specification. After acceptance testing, all evaluation cells will be further assessed in the following manner:

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)

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 characterization tests.
- Control monitoring characteristics

Engineering model battery tests will be performed to insure reliability of the mechanical, electrical, control, and thermal design characteristics. Because normal anticipated variations in battery temperatures have a marked effect upon requirements for charge control operation, complete breadboard subsystem assemblies which duplicate the thermal mission load characteristics will be required.

Battery failure modes will be programmed in tests to determine the adverse effects upon system operation. The simulated failures consist of conditions of cell short circuits, electrical leakage, mechanical leakage, and battery regulator failures.

Mission profile life testing will be performed with a simulated or production model of the charge control to provide confidence in the life cycling capability of the system.

Tables 5-16 and 5-17 outline the battery development test plan.

c. Power Control and Regulator Development Tests

Development testing (Table 5-18) will utilize one breadboard model and one engineering model of the power control unit, shunt elements assembly, and battery regulator unit. The breadboard model differs from the engineering model (and subsequent flight configuration) in that packaging, interconnections,and part reliability is not a consideration of the layout of fabrication

The PCU breadboard consists of an interconnected set of breadboard modules. Since wiring is well spread out, radiated and conducted interference tests are not performed. No thermal gradient problems are simulated and the dissipative parts have a conservative heat sink. Each circuit breadboard module will be functionally tested with simulated inputs and loads. These module tests will be conducted over temperature extremes to establish thermal margins.

The circuit module breadboards are then assembled into a breadboard PCU which will be similarly tested, in conjunction with the shunt elements, with simulated inputs and loads for an evaluation of performance. The results of the breadboard PCU tests are then compared with the test results at the module level to determine if module interface problems exist.

Table 5-16. Silver Cadmium Battery Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description	Test Equipment and Special Facilities
Battery cell 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	Automatic cycle tester, oil bath, heat exchange plates, bristol recorder, digital voltmeter, leak detector, millivolt meter, regulated power supplies, audio-oscillator and AM amplifier
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 power control unit 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	TRW characterization tester, digital data recording console, Bristol recorders, digital voltmeter, temperature oil bath
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	safety chamber, strain gages, recorders, power supply
Battery thermal property	Engineering model battery	Determine relationship between temperature changes, current, voltage, and state of charge	Individual cells are cycled to a mission environment and temperatures measured	Beckman thermometers, Bristol recorders, mounting hardware, regulated power supply
Battery cell life	Prototype cells	Determine cell characteristics after end-of-life incidents of charge cycle	Individual cells are maintained at a fixed thermal environment and subjected to charge/discharge cycles. Electrical characteristics periodically checked	Automatic cycle tester, oil bath, characterization tester
Battery evaluation	Engineering model 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	Oscilloscope, power supply, temperature-vacuum chamber, shake table, signal generator, impedance bridge, programmable load bank, recorders
Type approval	Type approval model	Final design verification of operational readiness	In accordance with approved test procedure	

Table 5-17. Battery Regulator Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description	Test Equipment and Special Facilities
Charge regulator breadboard	Charge regulator breadboard	Evaluate electrical perform- of alternate circuit design and primary assessment for rating of critical components; develop charge control system compatible with the battery temp. 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 re-corded as a function of input variables.	Oscilloscope, simulated loads, power supply, meters, recorders, test battery
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.	Oscilloscope, simulated loads, power supply, meters, recorders, test battery
Battery regulator engineering model module	Module package of engineering model, charge regulator, and boost regulator	Evaluate electrical performance of packaged modules; verify finalized module specs and test procedures	Repeat above breadboard measurements in accordance with test procedures at ambient and over-rated temperature ranges.	Oscilloscope, simulated loads, power supply, meters, recorders, test battery, temperature chamber
Battery regulator engineering model	Engineering model of battery regulator unit	Evaluate electrical performance of finalized package design, checkout test procedures and finalize equipment spec; detect failure under environmental extremes; confirm thermal design		Oscilloscope, simulated loads, power supply, meters, recorder, test battery, temperature chamber, thermal-vacuum chamber, vibration table
Type approval	Type approval model	Final design unification of operational readiness	In accordance with approved test procedure	

Table 5-18. Shunt Element and Power Control Unit Development Text Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description	Test Equipment and Special Facilities
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	Oscilloscopes, meters, recorders, power supply, solar array simulator, load simulator, temperature chamber, oscillator
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 sync. characteristics; measure allowable EMI susceptibility levels, shunt element dissipation and power consumption; observe waveforms and failure characteristics	Oscilloscopes, meters, recorders, power supply, solar array simulator, temperature chamber, oscillator
Engineering model module	Engineering models of the PCU modules, breadboard shunt element assembly	Evaluate electrical performance of PCU modules; check-out 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	Oscilloscopes, meters, recorders, power supply, solar array simulator, load simulator, temperature chamber, oscillator, wave analyzer, special test equipment
Engineering model	Engineering model of PCU consisting of mounted and interconnected PCU module, breadboard shunt, element assembly	Evaluate electrical and environmental performance of finalized packaged design; check-out 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	Oscilloscopes, meters, recorders, battery power source, solar array simulator, load simulator temperature chamber, oscillator, shake table, shock tower, thermal vacuum chamber, wave analyzer, special test equipment
Type approval	Type approval model	Final design verification of operational readiness	In accordance with approved test procedure	

Battery regulator tests will be made to assess compatibility of the array shunt regulator system with the regulation performance of the battery boost regulator. On a unit basis, these tests will be made using an array simulator, batteries, and active load simulators. The design of the simulator must be compared with that of the array performance and such parameters as capacitance, frequency response, as well as I-V characteristics.

Load simulators must duplicate such load parameters as turn-on surge, load impedance, and induced current. System stability tests will be performed utilizing circuitry which accurately simulates the entire characteristic curve including dynamic characteristics.

Comprehensive testing of the charge regulator will be made during development of the battery cell, charge monitor testing, as well as for subsequent verification of compatibility with the charge control design. Development testing of the battery regulator unit will proceed through the usual schedules of breadboard and engineering model prior to subsequent total system evaluation.

d. Power Conditioning Development Tests

The philosophy and scope of development testing for the inverters will be similar to that described above for the power control unit.

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 testers. The test proposed for inverters is outlined in Table 5-19.

4 7.4 Type Approval Tests

Type approval of the electric power subsystem will be on a unit basis, subject to further subsystem assessment as part of the spacecraft integration and spacecraft type approval testing. One

Table 5-19. Inverter Development Test Matrix

Name of Test	Item Being Tested	Purpose and Objectives	Description	Test Equipment and Special Facilities
Inverter breadboard	Breadboards: 4.1 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	Recorders, oscilloscope, simulated loads, impedance bridge, power supply, oscillator, wave analyzer, temperature chamber
Engineering model module	Engineering Model modules	Evaluate electrical performance of packaged modules: finalize module specs and test procedures	Measure performance parameters	Recorders, oscilloscope, simulated loads, impedance bridge, simulated power supply, oscillator, wave analyzer, temperature chamber
Engineering model inverter	Engineering models: 4.1 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: check-out 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	Recorders, oscilloscope, simulated loads, impedance bridge, simulated power supply, oscillator, wave analyzer, temperature chamber, shake table
Type approval test	Type approval model	Final design verification of operational readiness	In accordance with approved test procedure	

test article of each equipment will be subjected to environmental conditions applicable to the operational phases of storage, handling, stanby, launch, deployment, and flight of electric power equipment. The level of environment will be more severe than expected operational conditions in order to provide greater assurance of detecting design deficiencies. The test conditions are not intended to exceed design margins or to excite unrealistic modes of failure: should this occur, appropriate waivers will apply

Test articles will be identical to flight articles except for the solar panels, which will be configured to simulate magnetic, thermal, and dynamic characteristics by the partial use of dummy cells, partial panels, and mockup of conductor paths

In general, before the environmental tests the unit will be subjected to comprehensive functional tests under standard ambient conditions and a record made of all data necessary to determine compliance with the applicable equipment specification. These data will provide the basis for checking satisfactory performance of the equipment during or after environmental tests.

Degradation or change in performance of any assembly which exceeds limits established by its specification and applicable test procedure during any test period will be considered as a failure. Testing will be discontinued until the malfunction (including design defects) is corrected. If the corrective action consists of simple repair, such as replacement with identical parts, only that test procedure under which failure occurred will be repeated in its entirety without equipment failure before proceeding to the next test. If corrective action, such as redesign, is required, the test procedure under which failure occurred will be repeated as well as all other tests affected by the redesign.

An allowance for mechanical damage to solar cells will be reflected in the solar array type test procedure. Such deviation will consist of allowable cell cover cracks per module and percent area delamination, if analysis indicates that the array performance is still within specification.

The type approval test sequence will be governed by the following:

- Examination of product will be performed prior to each functional testing.
- Functional tests will be performed prior to, during (where appropriate), and following environmental testing. The functional testing to be performed prior to the next environmental test.
- Vibration and shock will precede thermal-vacuum testing.
- Magnetic properties determination will be performed prior to and following vibration testing.
- Humidity tests will be conducted last.
- Other environmental testing may be performed in any sequence.

Type approval test procedures will be prepared for each individual unit, which specify in detail the operating and nonoperating environments, simulation of environment, level of environment, special test apparatus, test measurements and sequence of testing and test procedures.

4.8 Planet-Oriented Package Subsystem

4.8.1 Summary

The planet-oriented package (POP) will provide the means of precision pointing for science instruments requiring articulation with respect to the spacecraft while it is in a Mars orbit. The POP subsystem consists of a payload structure mounted to the spacecraft by a double gimbal drive mechanization, the associated drive electronics, and the Mars horizon scanner. The experiment interface design, interface specification and design integration is discussed in Section V, paragraph 3.3 as a spacecraft development consideration.

The subsystem development consisting of analysis, testing, and documentation, is displayed in the development flow shown in Figure 5-27. The development schedule for POP is shown in Figure 5-28. The POP subsystem development effort involves those tasks which treat the subsystem in a manner to assure compatibility of individual units within the POP, as well as the POP and the spacecraft. Subsystem development will be a continuous function to establish design, specify, and test the POP for compatible interfaces with other spacecraft equipment. Analysis of the overall problem will be supported by breadboards and engineering models.

4.8.2 Analysis and Design

The analyses during the design phase will be as follows:

a. Preliminary Design Analysis

The preliminary design analysis will determine the general packaging arrangement and size of parts, materials, processes, and other information which would permit the initiation of the design layout. As a part of this analysis consideration will be given to the mechanical, electrical, and thermal interfaces with the POP, and the dynamic and static loads the POP will undergo. The type of drive motor, the gear ratio of the drive mechanism, the gimbal rates, and travel will also be studied; the POP mass properties will be calculated. Optical analysis for the Mars horizon scanner will be performed to establish the optical radiation levels which dictate the design requirements for the scanner.

A primary design objective for the POP and the body-mounted experiment package design will be to provide flexibility to accommodate a number of experiments and experiment changes both during development and between launch opportunities. This is accommodated by standardized mounting interfaces, and provision of extra electrical leads through the gimbal drives, accomplished at only a slight weight penalty.

b. Stress Analysis

Stress analysis will be conducted to insure that the sizes, weights, and materials provided in the initial design layout are optimized to withstand dynamic and static loads for the design life of the POP.

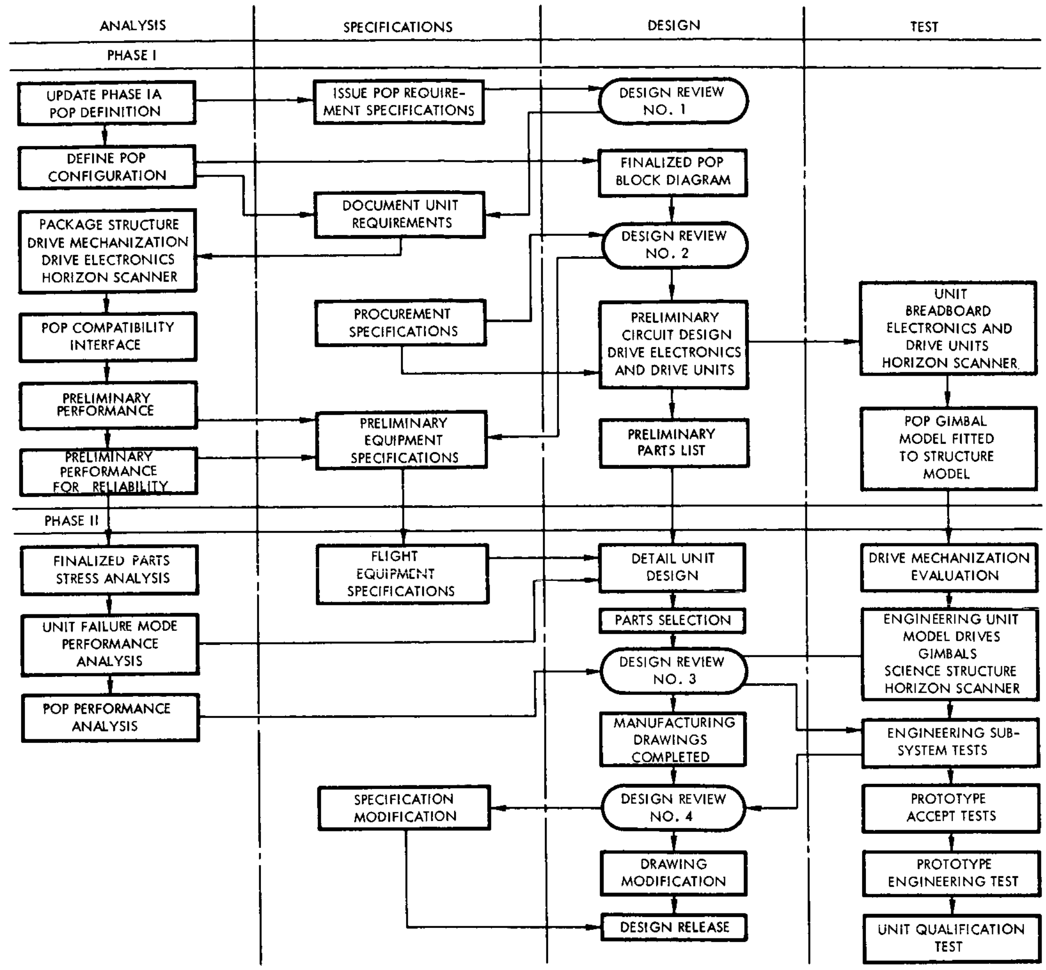


Figure 5-27. Planet-Oriented Package Subsystem Development Flow

c. Thermal Analysis

The thermal analysis will determine the thermal limitations of the parts, materials, and processes and adjust the over-all design as required to insure reliable performance within the thermal requirements of the science payload. Duty cycles and the power dissipation of the drive motors and science payload will be evaluated in terms of their thermal effects.

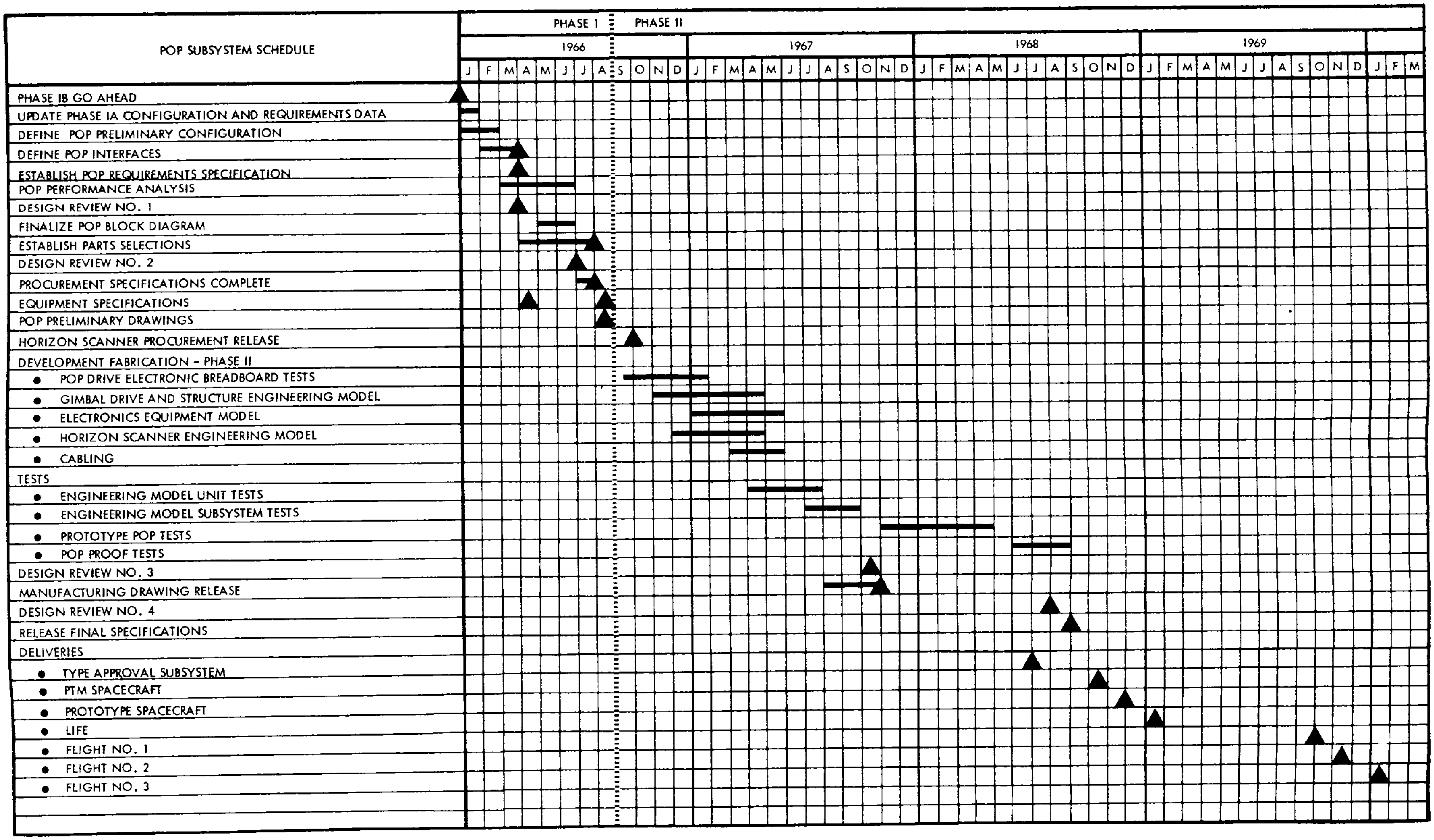


Figure 5-28. Planet-Oriented Package Subsystem Development Schedule

Based on these analyses, the design of the POP will be completed and preliminary manufacturing drawings prepared. Engineering models will be fabricated from these drawings and a series of engineering tests run on these models, including assessment of performance after vibration, acceleration, impact shocks, and thermal vacuum soak. Any necessary design changes will be incorporated into the design prior to release of manufacturing drawings.

4.8.3 Development Test

Early in Phase II, breadboards of the electronics will be constructed and subjected to performance tests in order to develop the drive electronic circuitry, and establish requirements on other spacecraft subsystems. Development tests are conducted during Phase IB as the POP will not be on the 1969 flight test, permitting adequate development time for the 1971 mission.

Engineering tests planned for Phase II include vibration, acceleration, shock, thermal vacuum, and humidity, on the following unit models:

- Horizon scanner
- Gimbal drive and structure
- Electronics and cabling

These engineering models will also be integrated for POP subsystem testing. A second POP engineering model will be fabricated, tested, and delivered to the spacecraft engineering model for spacecraft electrical compatibility tests.

Two prototype models will evolve from the engineering model tests. One will be used for proof test, and the second model delivered to JPL for type approval testing.

The Mars horizon scanner will be subcontracted after complete requirements are established. Requests for proposals, vendor surveys, and release of the horizon scanner subcontract will be completed within 9 months after Phase IB go-ahead. Early procurement of the scanner will permit complete reliability testing to be accomplished in support of the 1971 mission.

4.9 Electrical Distribution Subsystem

4.9.1 Summary

The electrical distribution subsystem consists of electrical interconnecting cabling, junction boxes, test and umbilical connectors, power switching, and ordnance initiation circuitry. The development of this hardware is discussed in this section and is shown in Figure 5-29.

The spacecraft design integration tasks are discussed under spacecraft development, subsection 3.1. Figures 5-30 and 5-31 show the 1969 and 1971 development schedules, respectively, covering both Phase IB and II development of the electrical distribution subsystem.

No major problems are anticipated in the design of the electrical distribution hardware. Design and development of the cabling and junction boxes will proceed according to techniques which have been proven on current programs. Circuit design and development is necessary in the area of power switching without utilizing electromechanical relays and the capacitor discharge initiation of pyrotechnic devices. However, it is anticipated that these circuits will use available components and will involve no new problems.

4.9.2 Analysis and Design

Analyses necessary for the design of the electrical distribution hardware consist of the extraction of systems design requirements and the analysis of methods of implementing these requirements. The major inputs criteria are general packaging, electromagnetic interference control, magnetic field control, and systems test points requirements including the resulting electrical operational support equipment hardline interfaces.

Having established the requirements, the subsequent analyses will produce design guides for the cable and junction box designers. These guides will define the types of wiring to be used, where twisting and shielding will be used, criteria for the selection of wire sizes, a plan for grounding, bonding, and shielding, and guides for the allowable signal circuits which can be grouped together in a cable bundle.

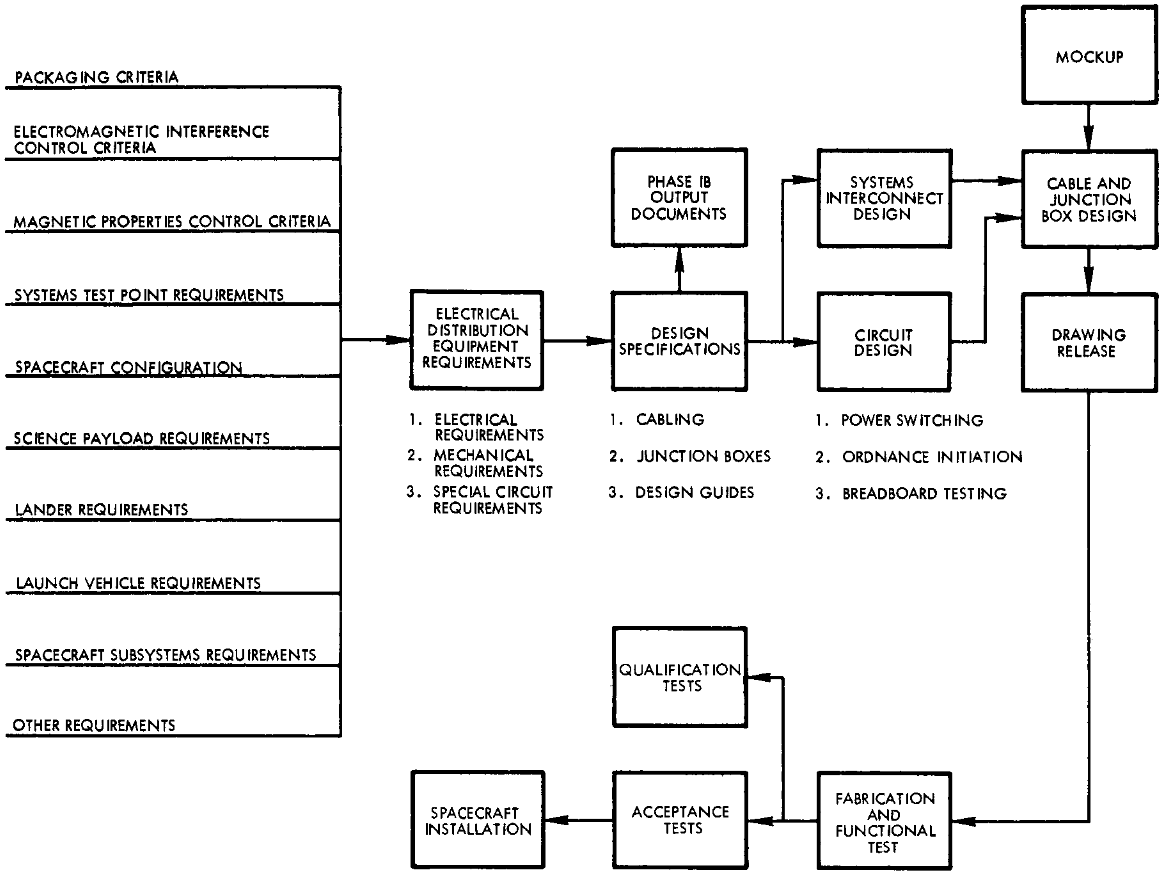


Figure 5-29. Design and Development Flow Electrical Distribution Subsystem

A detailed analysis of all electrical interface characteristics will be made to optimize the electrical interconnections. Participation in all electrical subsystems design reviews will be maintained to further this effort.

In conjunction with structures, packaging, thermal, and other design personnel, the detailed cable routing and panel interfaces will be defined and maintained using a spacecraft configuration model as a design tool.

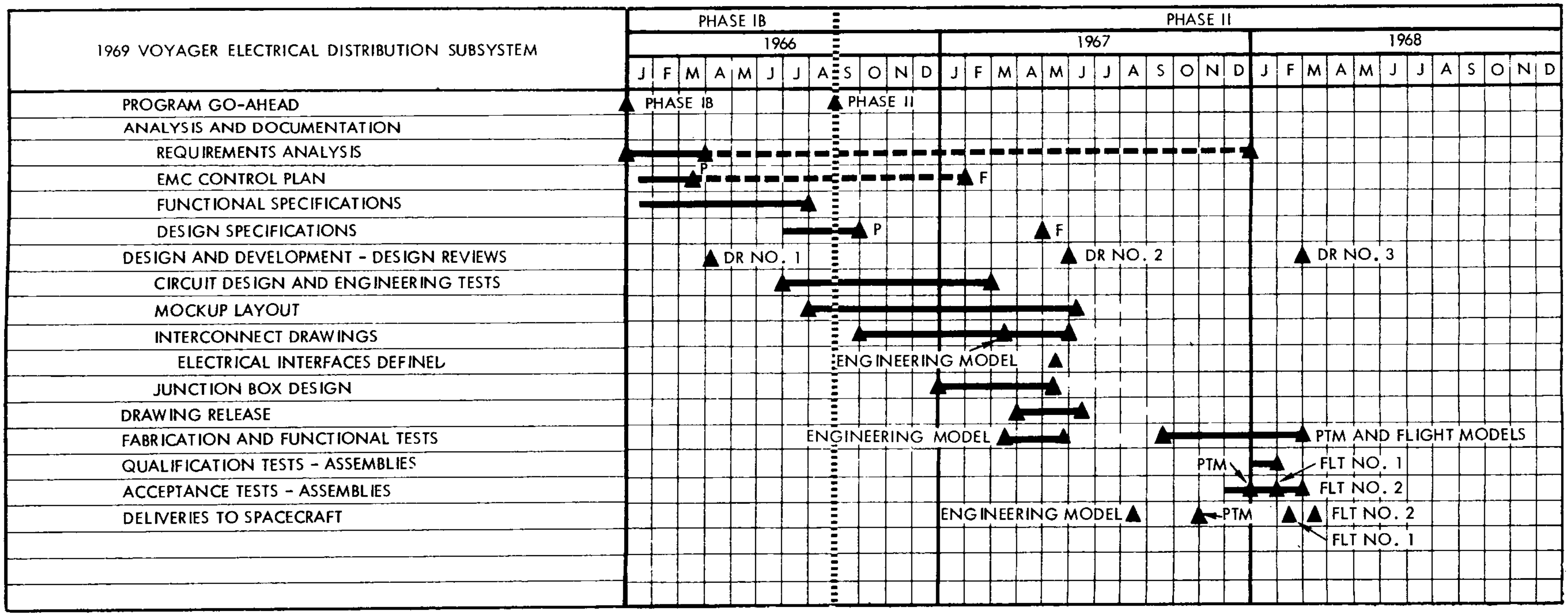


Figure 5-30. 1969 Electrical Distribution Subsystem Schedule

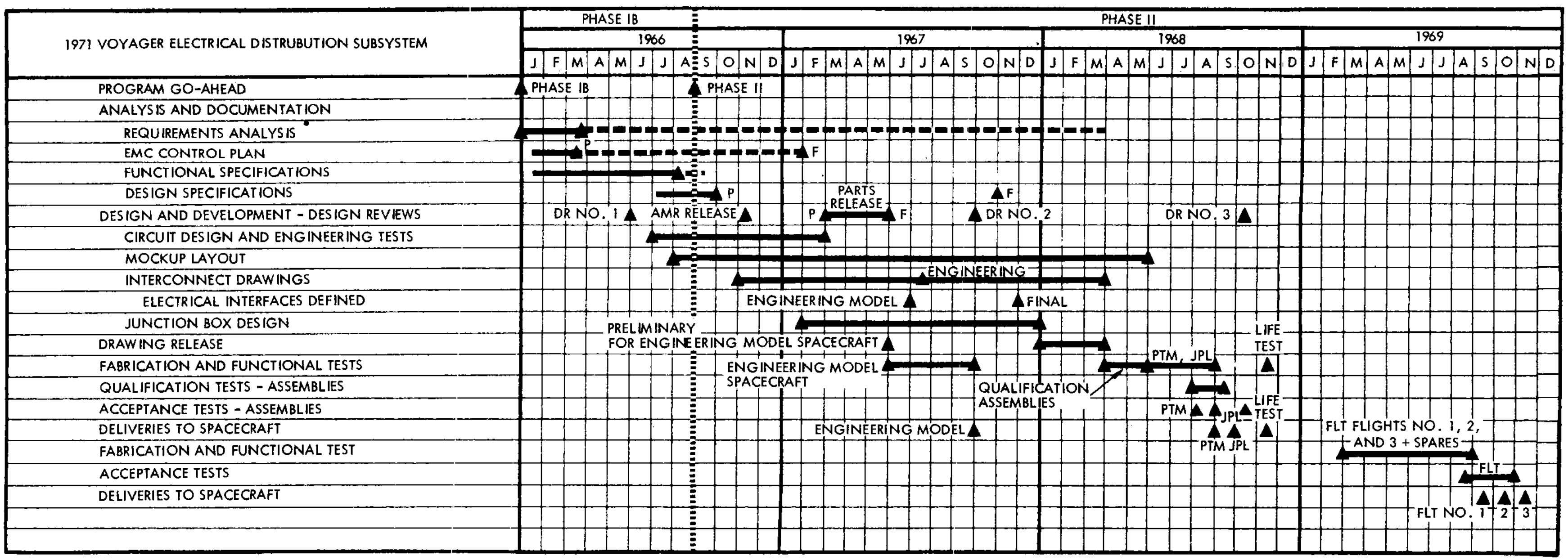


Figure 5-31. Electrical Distribution Subsystem Schedule, 1971

4.9.3 Test Program

a. Development Testing

A minimum amount of development testing is required for this subsystem. The circuit designs for the power switching circuitry and the ordnance initiation circuitry will require some breadboard and engineering model environmental testing. The remainder of the hardware has a considerable systems level and flight test history and, in addition, its configuration is such that meaningful tests of the hardware are minimal.

b. Proof Testing

It is anticipated that cabling and junction boxes which do not contain active circuitry will not require a complete subsystem proof test program. It is considered that proof test level vibration and thermal-vacuum testing should prove adequate to qualify junction boxes containing only passive circuits and components. A meaningful vibration and thermal-vacuum test of interconnect cabling can be made only on a spacecraft model because the mechanical characteristics are determined by the spacecraft installation.

Where junction boxes contain active circuitry a full qualification test sequence will be conducted on flight configuration samples.

c. Test Procedures

Formal test procedures will be generated for each item of separable hardware in its configuration prior to spacecraft installation. This will include each of the interconnect cables and each of the junction boxes. Fabrication test procedures and qualification and acceptance test procedures, containing specific test requirements for the individual item of hardware will define and document the tests including fabrication testing through spacecraft installation.

4.9.4 1969 Flight Test Spacecraft

Unlike the majority of the spacecraft electrical subsystems, the 1969 electrical distribution subsystem will differ considerably in form and function from the 1971 Voyager spacecraft. Although the functional requirements upon the subsystem are the same as those for 1971, the detailed configuration is entirely dependent upon the exact equipment complement, the structural and configuration layout, and the interfaces with the defined science experiments and the launch vehicle.

The majority of the analyses to be conducted for the 1971 Voyager will be applicable to the 1969 flight test spacecraft, but the detailed interconnecting cabling and junction box configurations will be tailored to the specific requirements of the 1969 mission.

The same design tools will be utilized, the spacecraft configuration mockup used for cable routing purposes and for layout of the spacecraft black box assemblies. The same electromagnetic compatibility criteria and control methods should apply.

The design layout and interconnect cabling will proceed essentially in parallel with interface definitions and final configuration required earlier for the 1969 mission than for the 1971 mission.

The same criteria for testing will apply for the 1969 assemblies as for the 1971 assemblies. The junction boxes containing passive circuitry will be exposed to qualification levels of vibration and thermal-vacuum testing; those containing active circuitry will receive a full qualification test explosure sequence. Interconnect cabling will receive only insulation resistance and continuity testing prior to spacecraft installation.

5. MANUFACTURING AND MATERIAL ACQUISITION

This section provides a brief description of the manufacturing and material acquisition tasks pertinent to the Voyager project. The detailed plans will be submitted in response to the Phase IB request for proposal.

5.1 Manufacturing

The manufacturing tasks for the Voyager project include design liaison, identification of the equipment and quantities to be delivered, detailed manufacturing planning and scheduling, design and fabrication of production tooling and test equipment, fabrication, and flight approval testing.

A preliminary equipment list has been prepared (given in Appendix E) and the schedule requirements to fabricate the equipment to meet the delivery dates have been analyzed and defined. Schedules for the manufacturing activities at TRW, RCA, and Douglas are presented in Figure 5-32 and 5-33. The preliminary plans for fabrication and assembly of the structural, thermal, and propellant feed assemblies for the 1969 and 1971 spacecraft are sketched in Figures 5-34 and 5-35, respectively.

As items are fabricated for the Voyager spacecraft they will undergo flight approval tests, as diagrammed in Figure 5-36.

5.2 Material Acquisition

The tasks associated with the procurement of long lead time, high reliability electrical parts and certain other specific equipment (e.g., gyro reference assemblies, three speed tape recorders) require that a definitive material acquisition plan be formulated early during Phase IB. These tasks are briefly outlined in PERT format in Figure 5-37, with typical setback times shown. Typical procurement time for parts requiring a full qualification program is shown as approximately 49 weeks; for parts requiring parameter drift screening, 42 weeks; and for subcontracted items, 56 weeks. A detailed material acquisition plan will be prepared in response to the Phase IB request for proposal. Key milestones required for the updating and implementation of this plan are shown in the Phase IB schedule in Section II

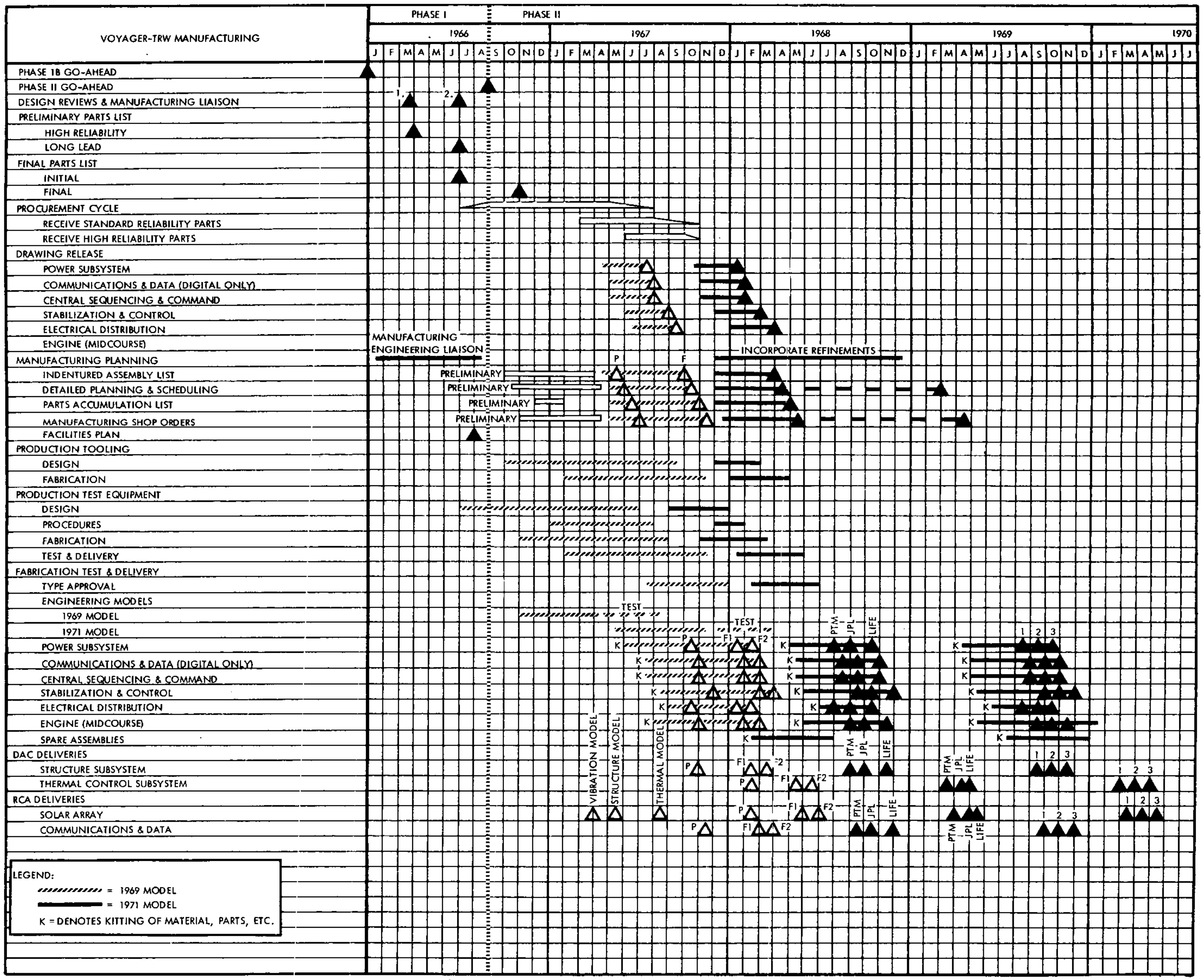


Figure 5-32. Voyager Manufacturing Schedule

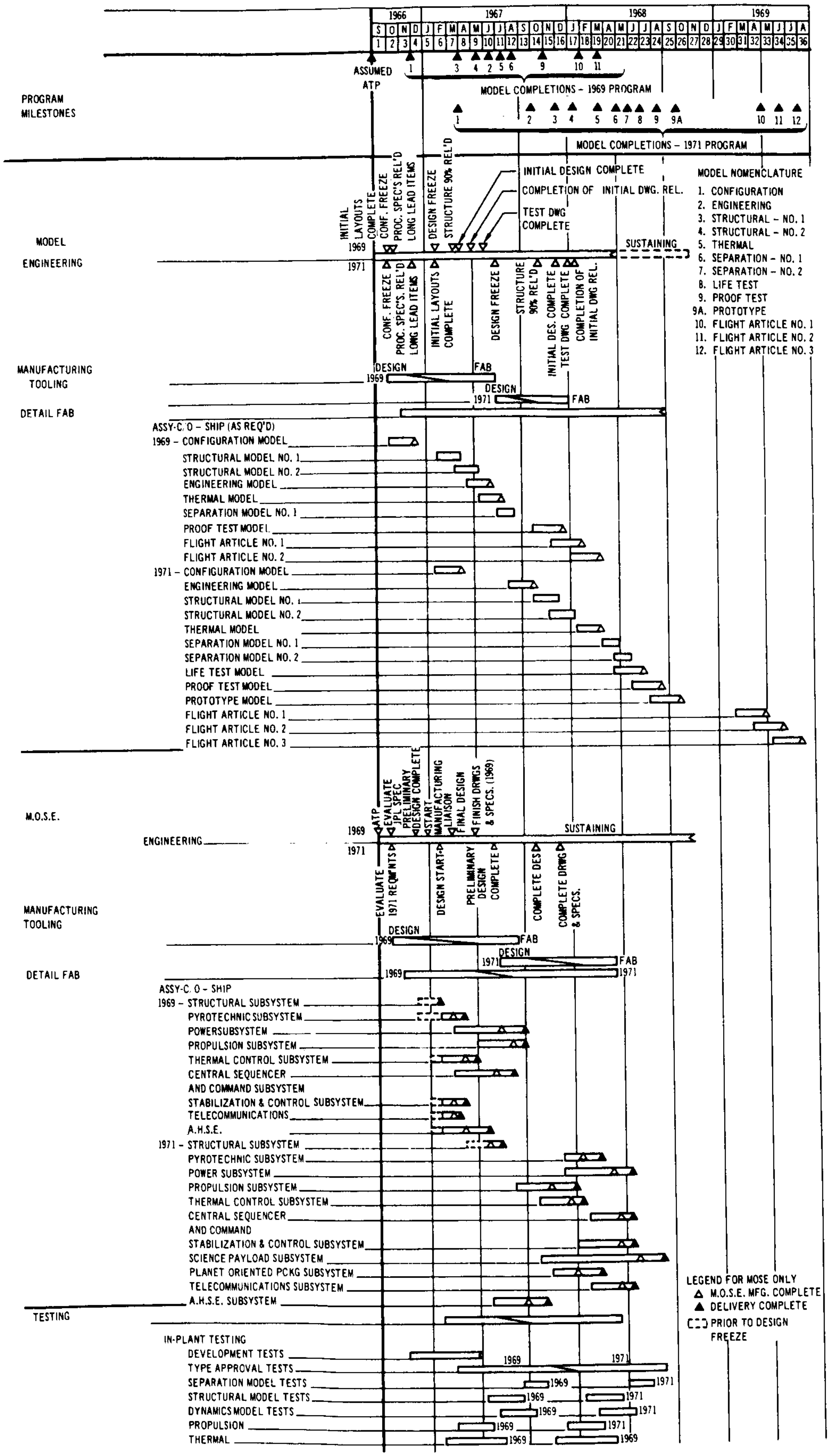


Figure 5-33. Preliminary Master Summary Schedule Phase II

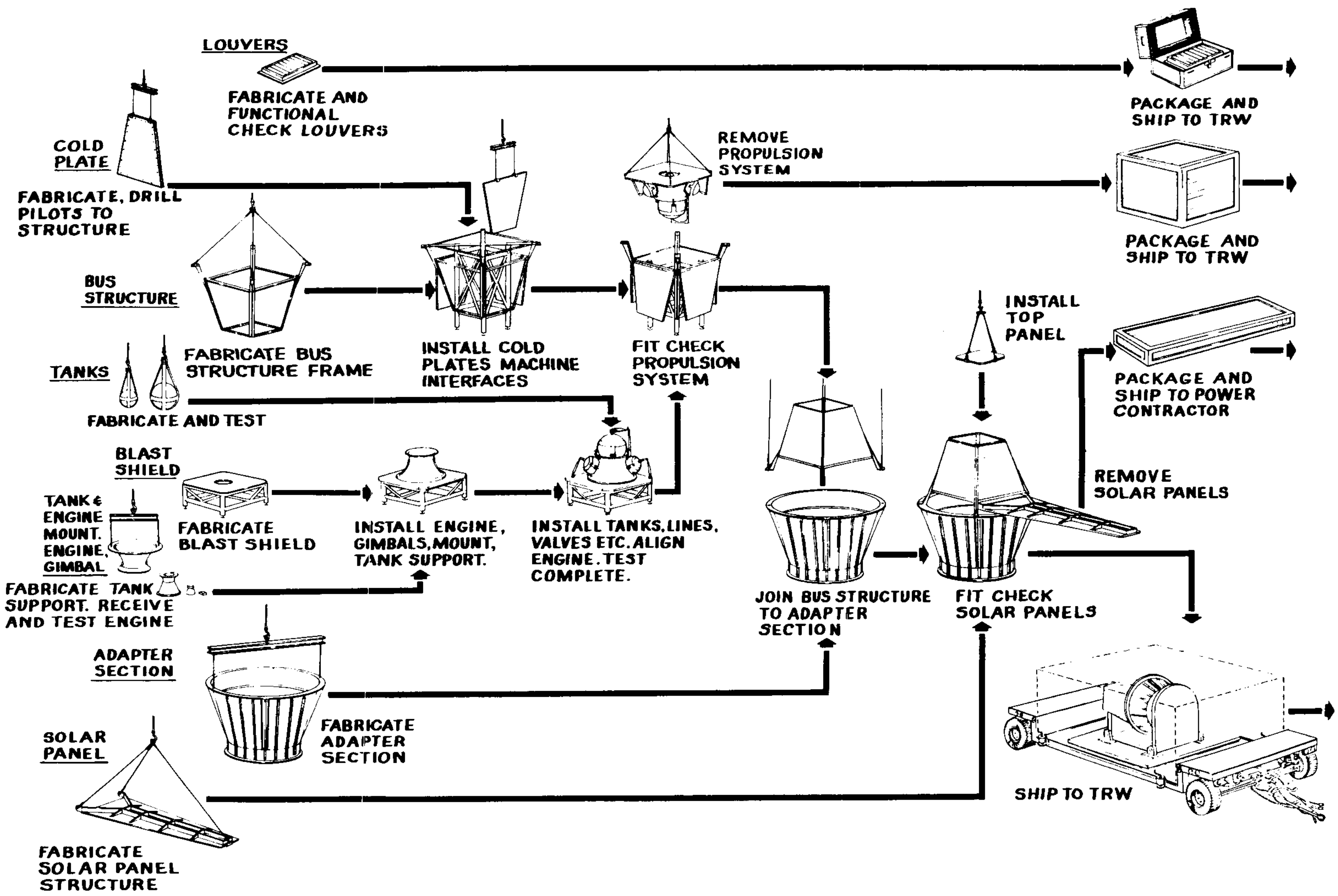
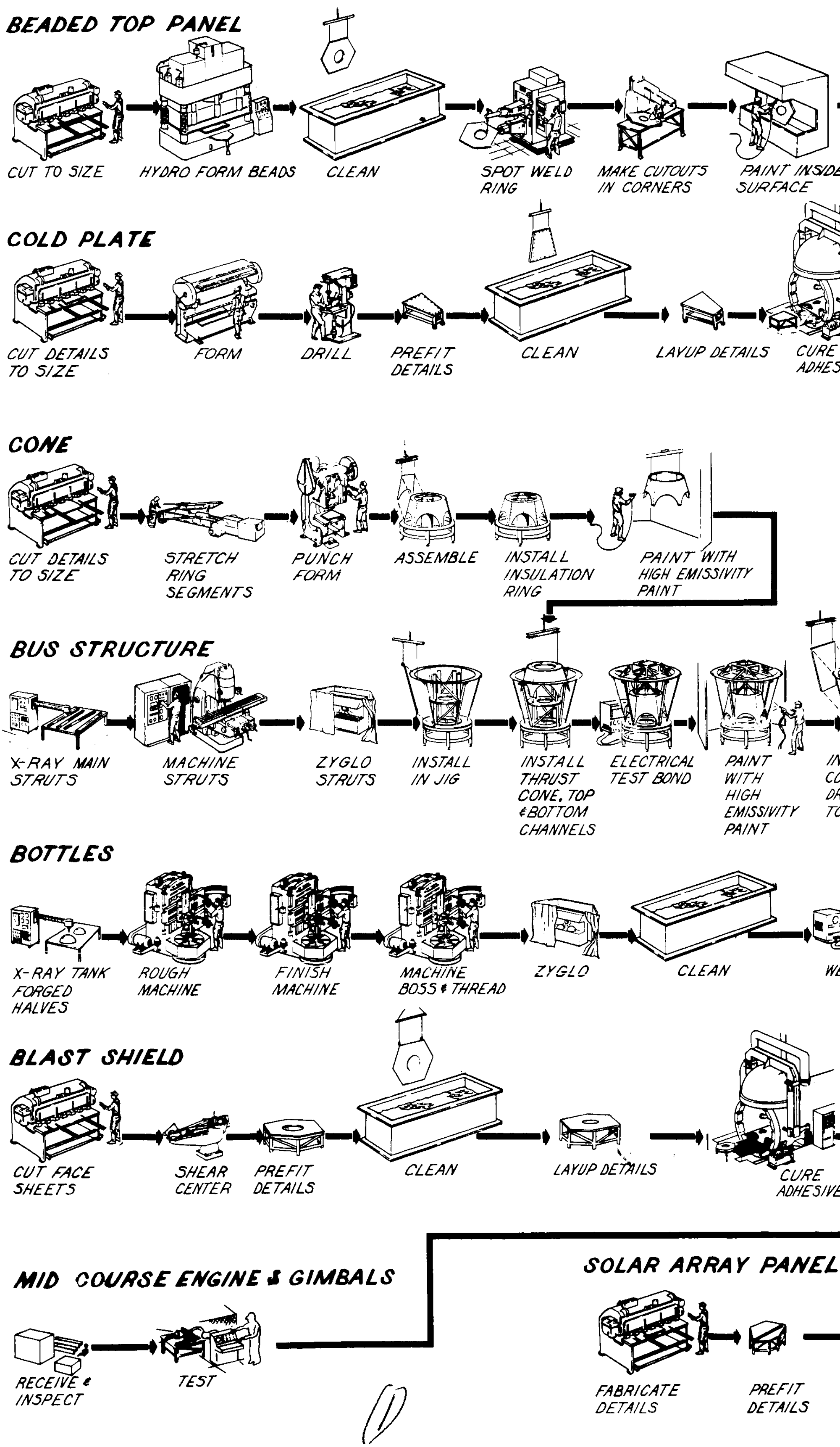
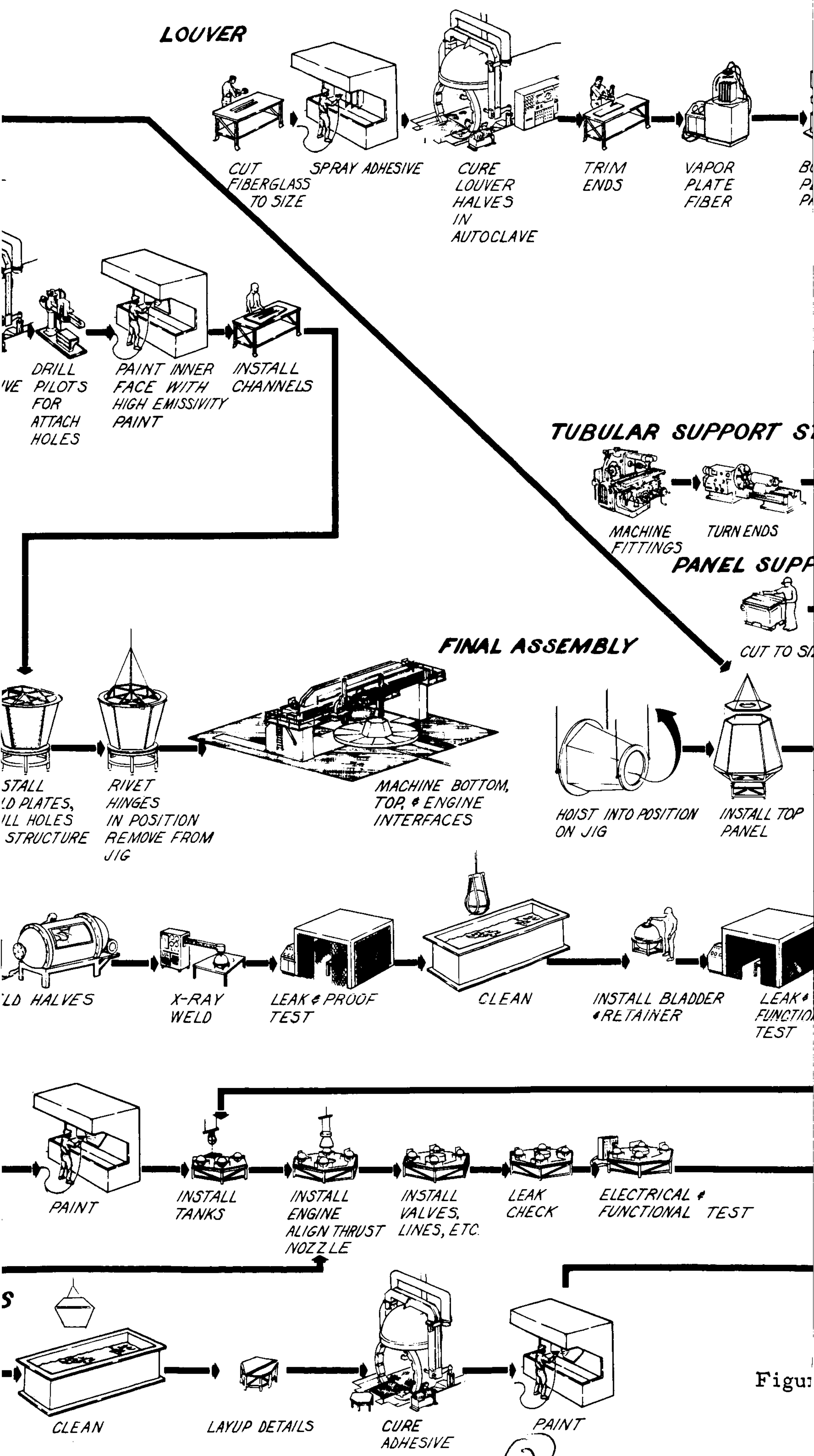


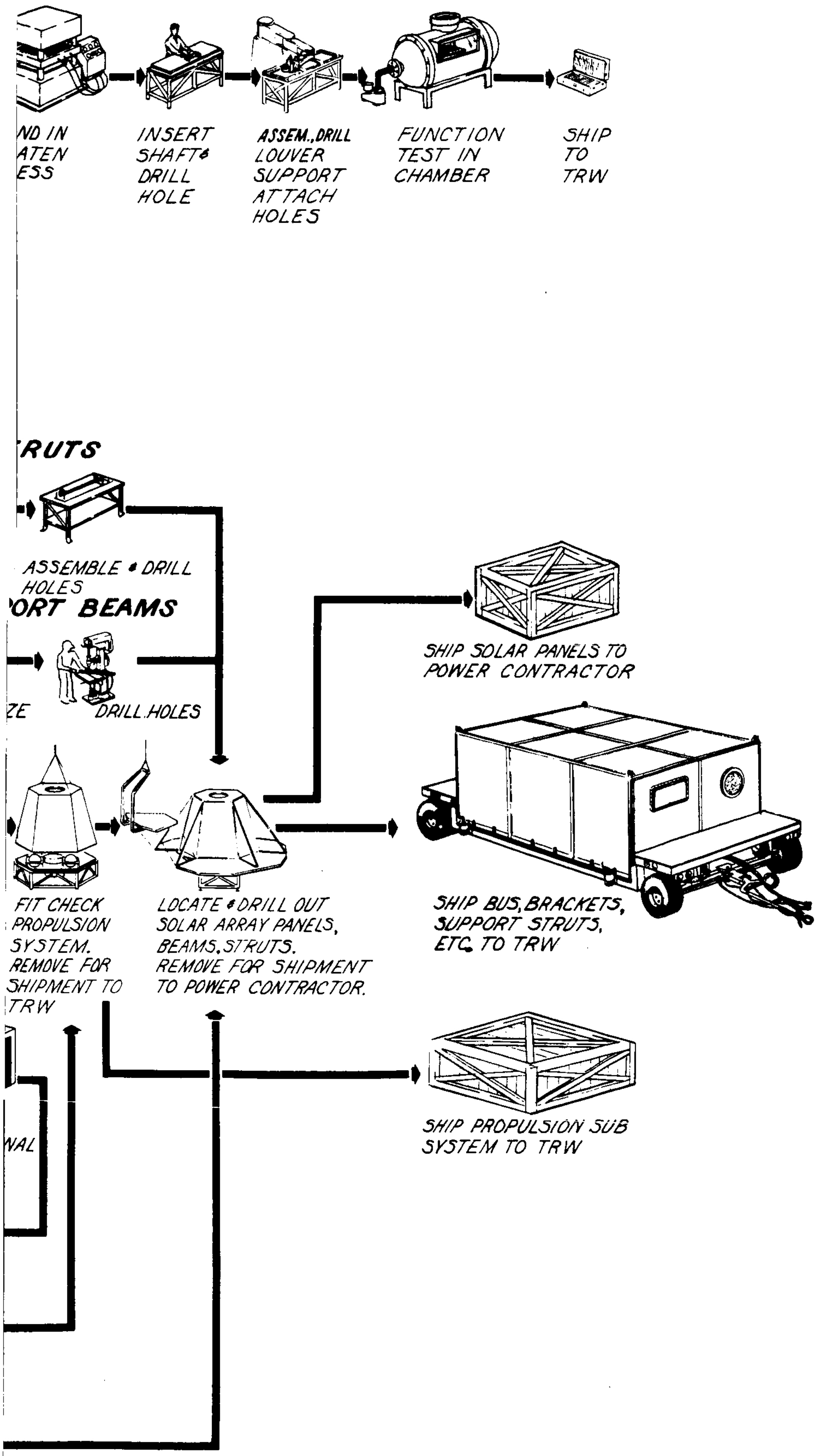
Figure 5-34. Fabrication and Assembly of the 1969 Voyager Planetary Vehicle


BEADED TOP PANEL
CUT TO SIZE
HYDRO FORM BEADS
CLEAN
SPOT WELD RING
MAKE CUTOUTS IN CORNERS
SURFACE
COLD PLATE
CUT DETAILS TO SIZE
FORM
DRILL
PREFIT DETAILS
CLEAN
LAYUP DETAILS
CURE
CONE
CUT DETAILS TO SIZE
STRETCH RING SEGMENTS
PUNCH FORM
ASSEMBLE
INSTALL INSULATION RING
PAINT WITH HIGH EMISSIVITY PAINT
BUS STRUCTURE
X-RAY MAIN STRUTS
MACHINE STRUTS
ZYGLO STRUTS
INSTALL IN JIG
INSTALL THRUST CONE, TOP & BOTTOM CHANNELS
ELECTRICAL TEST BOND
PAINT WITH HIGH EMISSIVITY PAINT
BOTTLES
X-RAY TANK FORGED HALVES
ROUGH MACHINE
FINISH MACHINE
MACHINE BOSS & THREAD
ZYGLO
CLEAN
BLAST SHIELD
CUT FACE SHEETS
SHEAR CENTER
PREFIT DETAILS
CLEAN
LAYUP DETAILS
CURE
MID COURSE ENGINE & GIMBALS
RECEIVE & INSPECT
TEST
SOLAR ARRAY PANEL
FABRICATE DETAILS
PREFIT DETAILS
10


LOUVER
CUT FIBERGLASS TO SIZE
SPRAY ADHESIVE
CURE LOUVER HALVES IN AUTOCLAVE
TRIM ENDS
VAPOR PLATE FIBER
DRILL PILOTS FOR ATTACH HOLES
PAINT INNER FACE WITH HIGH EMISSIVITY PAINT
INSTALL CHANNELS
TUBULAR SUPPORT S
MACHINE FITTINGS
TURN ENDS
PANEL SUPP
FINAL ASSEMBLY
STALL
LD PLATES,
LL HOLES
STRUCTURE
RIVET HINGES IN POSITION REMOVE FROM JIG
MACHINE BOTTOM, TOP, & ENGINE INTERFACES
HOIST INTO POSITION ON JIG
INSTALL TOP PANEL
LD HALVES
X-RAY WELD
LEAK & PROOF TEST
CLEAN
INSTALL BLADDER & RETAINER
TEST
PAINT
INSTALL TANKS
INSTALL ENGINE ALIGN THRUST NOZZLE
INSTALL VALVES, LINES, ETC.
LEAK CHECK
ELECTRICAL & FUNCTIONAL TEST
S
CLEAN
LAYUP DETAILS
CURE ADHESIVE
PAINT

Figu

(2)



re 5-35. Fabrication and Assembly of the
1971 Voyager Planetary Vehicle

3

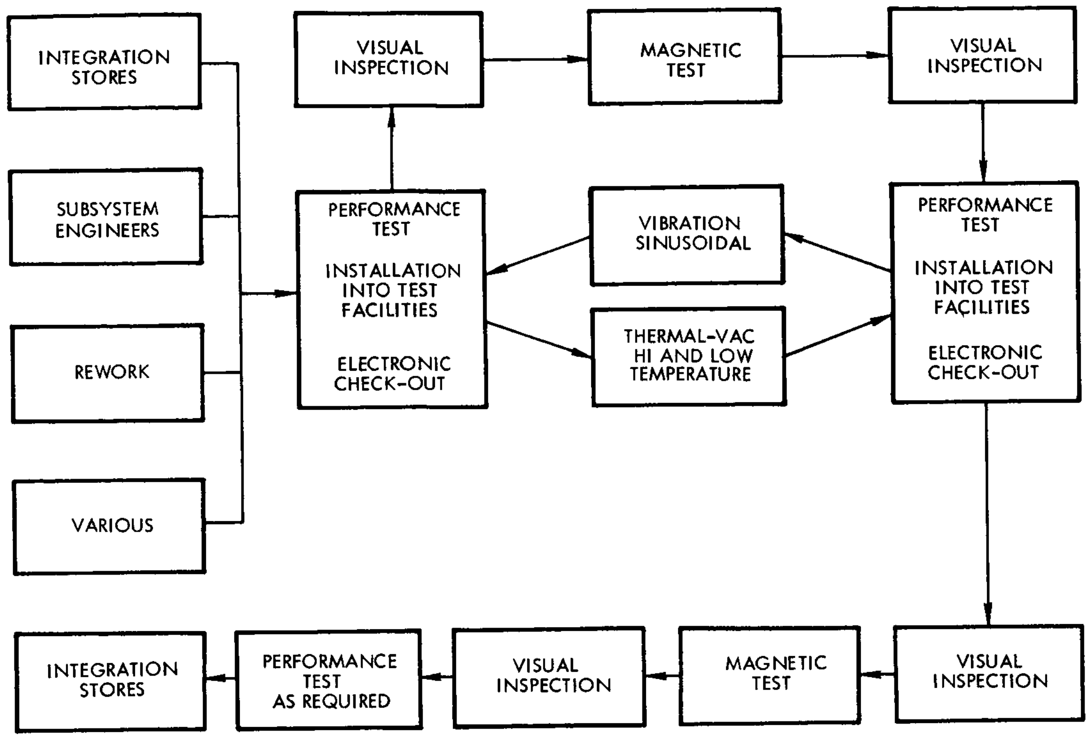


Figure 5-36. Assembly Flight Approval Test Flow

6. SPACECRAFT ASSEMBLY, CHECKOUT, TEST, LAUNCH AND MISSION SUPPORT OPERATIONS

6.1 Introduction

This section discusses the assembly and checkout, test, launch, and mission support operations for the various spacecraft models associated with the 1969 test flight and the 1971 mission. The 1971 spacecraft models include:

- Spacecraft Engineering Model (S/C EM)
- Spacecraft Propulsion and Stabilization and Control Model
- Proof Test Model (PTM)
- Life Test Model (LTM)

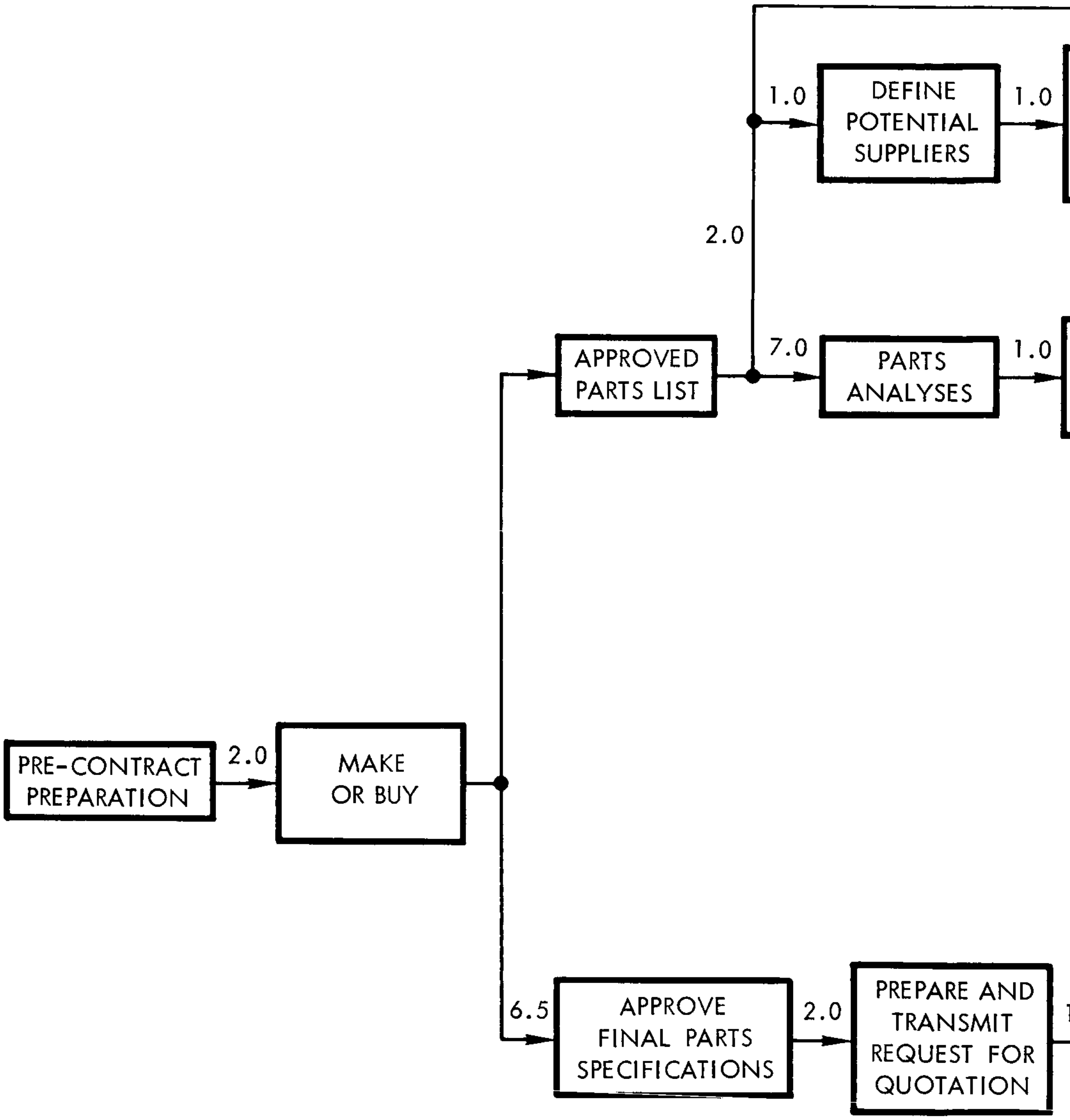
- Simulators
- First Flight Spacecraft (FS-1)
- Second Flight Spacecraft (FS-2)
- Third Flight Spacecraft (FS-3)

The presentation first discusses the engineering tasks required to plan the operations, identifies the elements of planning and control to support the operation, and finally presents a narrative description of the assembly, checkout, test, launch, and mission support operations. The description is provided in the form of operations flow charts and text, with a more detailed step-by-step description supplementing the text in the form of tabular descriptions keyed to the flow charts by operation numbers. The detailed tabular descriptions are given in Appendix A along with a duplicate copy of the operations flow charts.

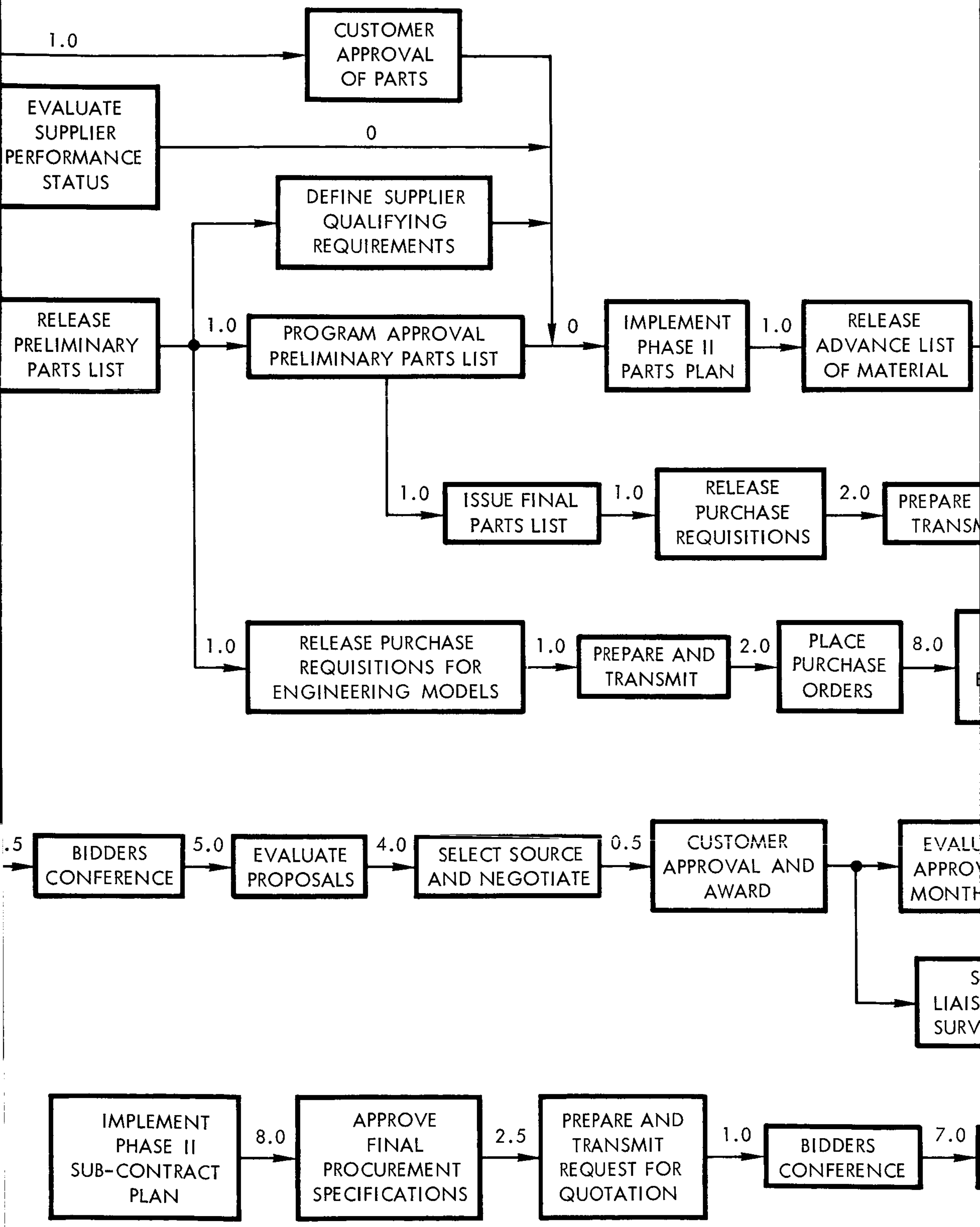
Since the assembly, checkout, test, launch, and mission support operations for the 1969 test flight are essentially identical to those for the 1971 mission spacecraft, they are not repeated here. However, flow charts and tables describing these operations as pertinent to the 1969 spacecraft are also included in Appendix A. The spacecraft planned for the 1969 test flight are as follows:

- Spacecraft Engineering Model (S/C EM)
- Spacecraft Propulsion and Stabilization and Control
- Proof Test Model (PTM) (also used for the life tests)
- Simulators
- First Flight Spacecraft (FS-1)
- Second Flight Spacecraft (FS-2)

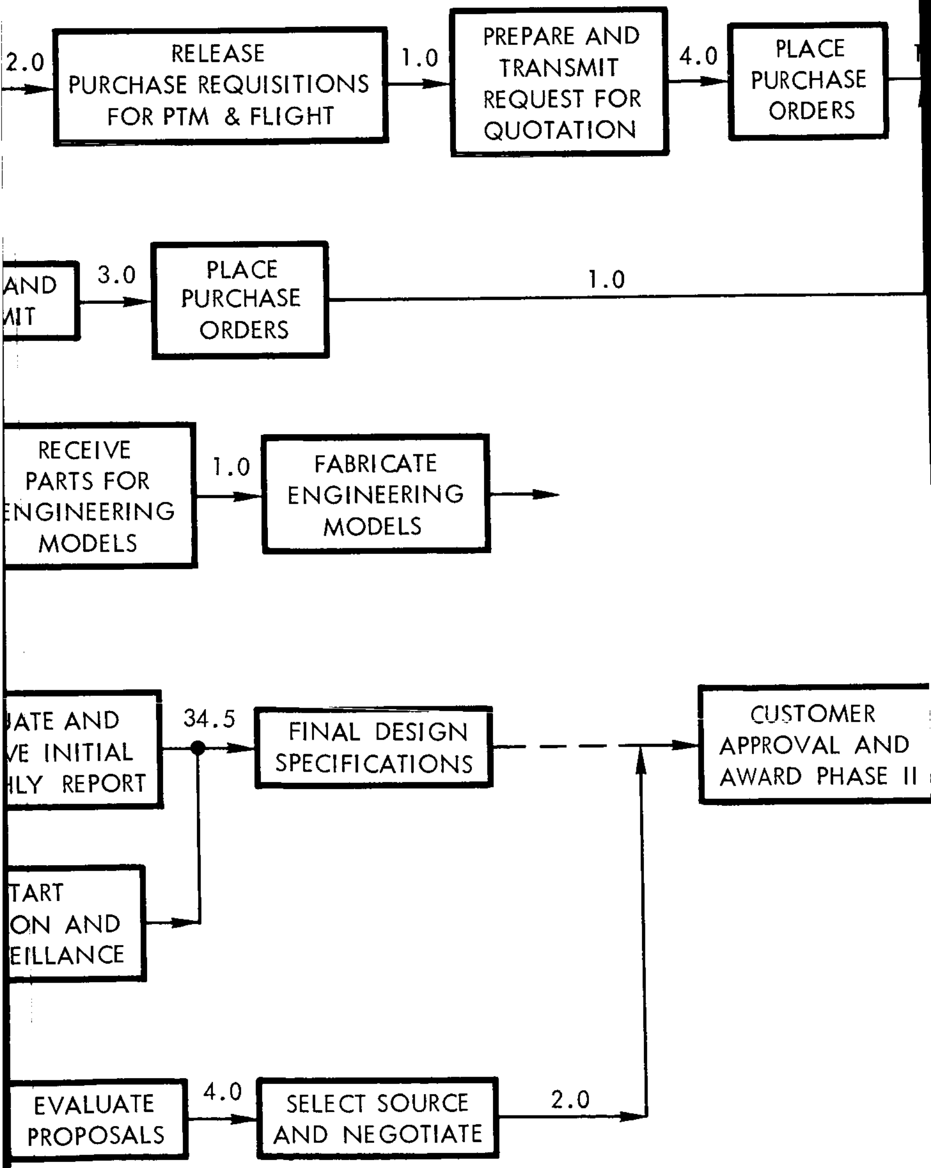
The launch operation plan for the 1969 test flight will parallel that of the 1971 mission in that although only two flight spacecraft are programmed for launch, the 1969 spacecraft engineering model will act as the third spacecraft for rotating spares.


PRE-CONTRACT PREPARATION
2.0
MAKE OR BUY
APPROVED PARTS LIST
6.5
APPROVE FINAL PARTS SPECIFICATIONS
2.0
PREPARE AND TRANSMIT REQUEST FOR QUOTATION
2.0
1.0
DEFINE POTENTIAL SUPPLIERS
1.0
7.0
PARTS ANALYSES
1.0

(1)

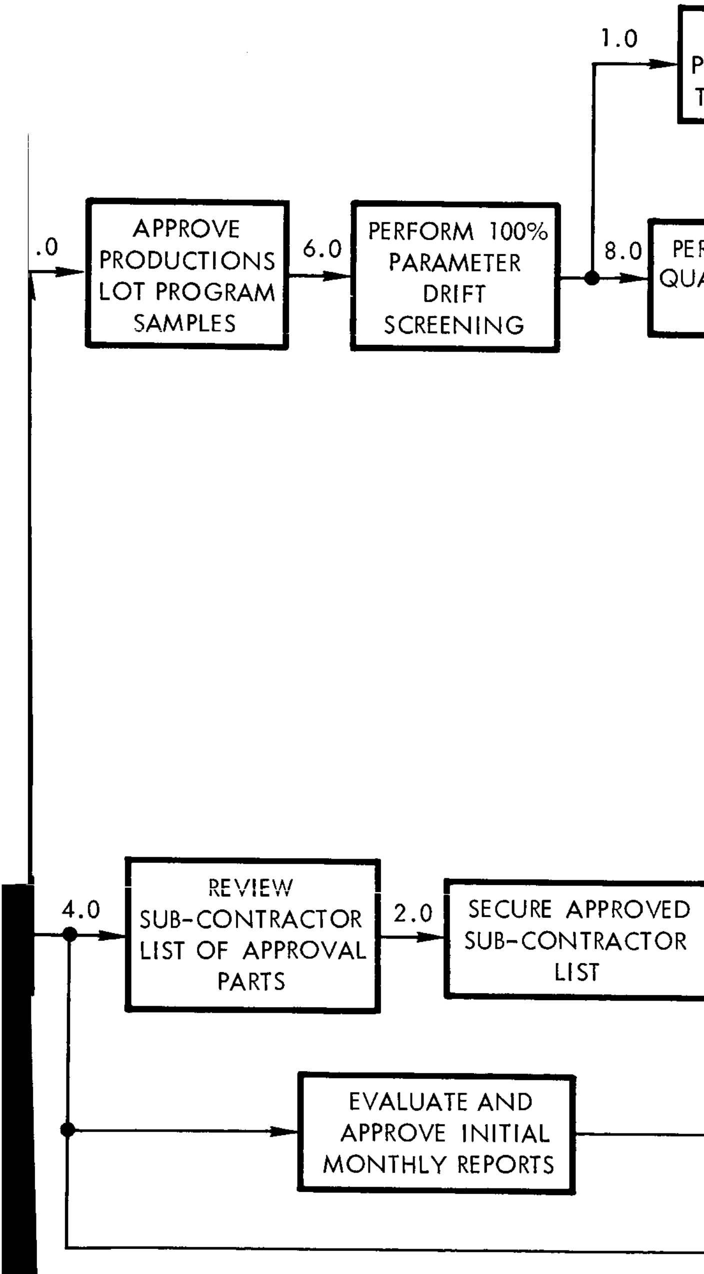

1.0
CUSTOMER APPROVAL OF PARTS
EVALUATE SUPPLIER PERFORMANCE STATUS
0
DEFINE SUPPLIER QUALIFYING REQUIREMENTS
RELEASE PRELIMINARY PARTS LIST
1.0
PROGRAM APPROVAL PRELIMINARY PARTS LIST
0
IMPLEMENT PHASE II PARTS PLAN
1.0
RELEASE ADVANCE LIST OF MATERIAL
1.0
ISSUE FINAL PARTS LIST
1.0
RELEASE PURCHASE REQUISITIONS
2.0
PREPARE
1.0
RELEASE PURCHASE REQUISITIONS FOR ENGINEERING MODELS
1.0
PREPARE AND TRANSMIT
2.0
PLACE PURCHASE ORDERS
8.0
.5
BIDDERS CONFERENCE
5.0
EVALUATE PROPOSALS
4.0
SELECT SOURCE AND NEGOTIATE
0.5
CUSTOMER APPROVAL AND AWARD
IMPLEMENT PHASE II SUB-CONTRACT PLAN
8.0
APPROVE FINAL PROCUREMENT SPECIFICATIONS
2.5
PREPARE AND TRANSMIT REQUEST FOR QUOTATION
1.0
BIDDERS CONFERENCE
7.0

2



2.0
RELEASE PURCHASE REQUISITIONS FOR PTM & FLIGHT
1.0
PREPARE AND TRANSMIT REQUEST FOR QUOTATION
4.0
PLACE PURCHASE ORDERS
AND
MIT
3.0
PLACE PURCHASE ORDERS
1.0
RECEIVE PARTS FOR ENGINEERING MODELS
1.0
FABRICATE ENGINEERING MODELS
ATE AND
VE INITIAL
HLY REPORT
34.5
FINAL DESIGN SPECIFICATIONS
CUSTOMER APPROVAL AND AWARD PHASE II
TART
ON AND
EILLANCE
EVALUATE PROPOSALS
4.0
SELECT SOURCE AND NEGOTIATE
2.0

3


1.0
.0
APPROVE PRODUCTIONS LOT PROGRAM SAMPLES
6.0
PERFORM 100% PARAMETER DRIFT SCREENING
8.0
4.0
REVIEW SUB-CONTRACTOR LIST OF APPROVAL PARTS
2.0
SECURE APPROVED SUB-CONTRACTOR LIST
EVALUATE AND APPROVE INITIAL MONTHLY REPORTS

4

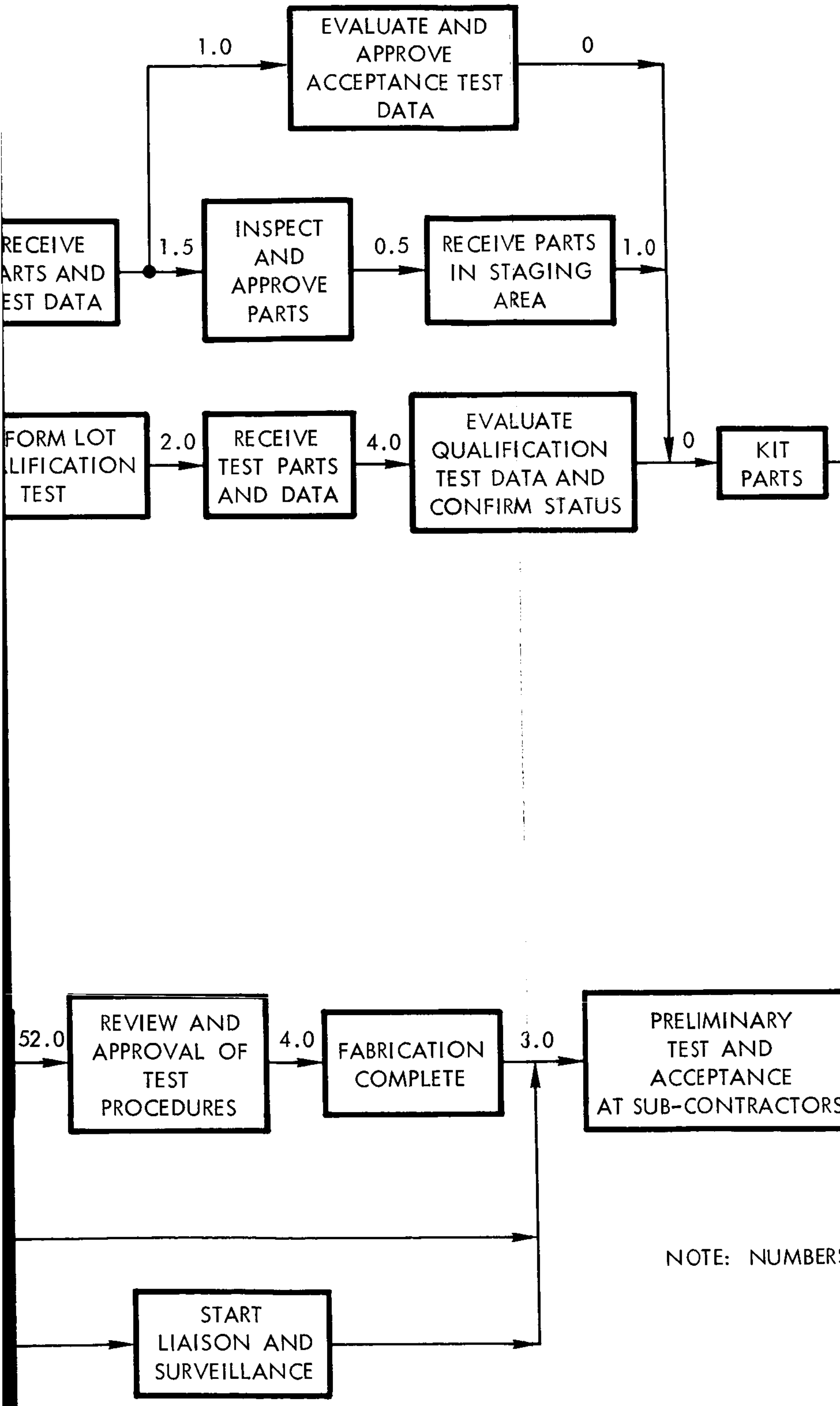
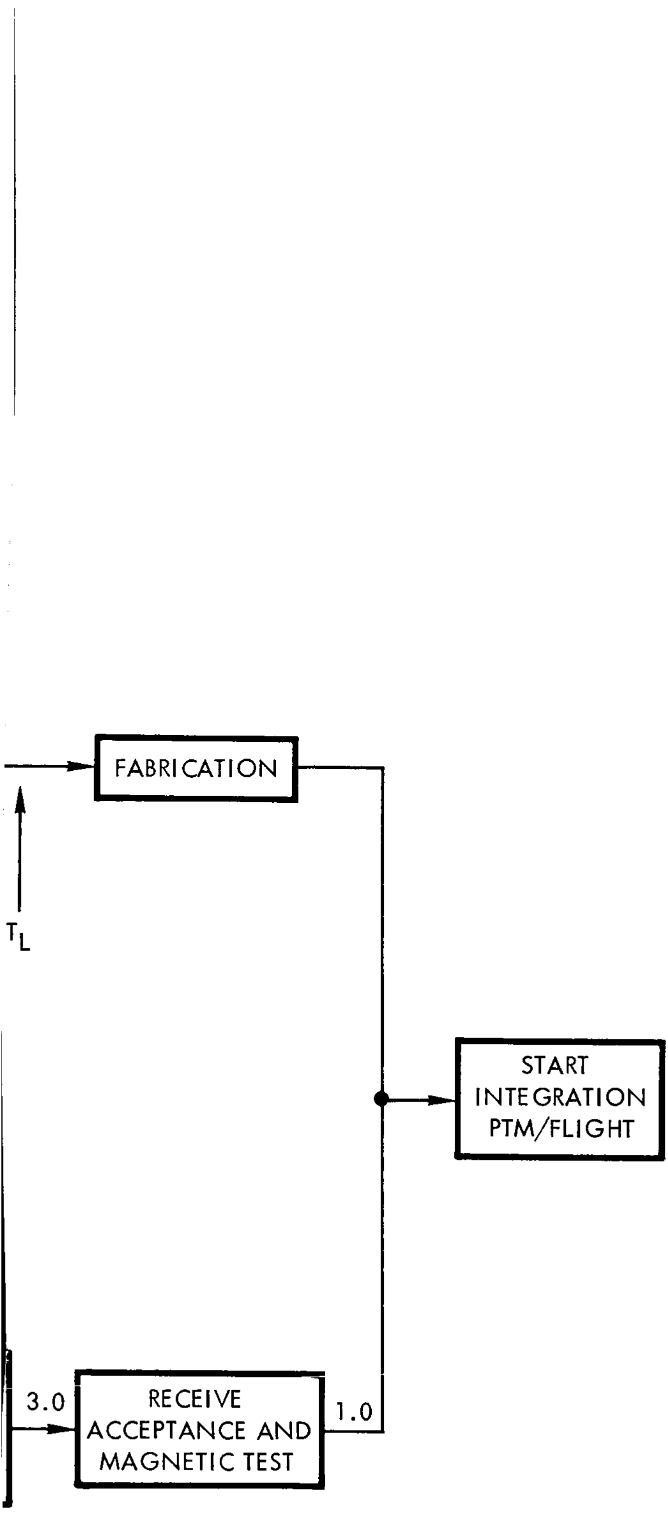

EVALUATE AND APPROVE ACCEPTANCE TEST DATA
1.0
0
RECEIVE
ARTS AND
EST DATA
1.5
INSPECT AND APPROVE PARTS
0.5
RECEIVE PARTS IN STAGING AREA
1.0
FORM LOT
LIFICATION
TEST
2.0
RECEIVE TEST PARTS AND DATA
4.0
EVALUATE QUALIFICATION TEST DATA AND CONFIRM STATUS
0
KIT PARTS
52.0
REVIEW AND APPROVAL OF TEST PROCEDURES
4.0
FABRICATION COMPLETE
3.0
PRELIMINARY TEST AND ACCEPTANCE AT SUB-CONTRACTORS
NOTE: NUMBERS
START LIAISON AND SURVEILLANCE

Figure 5-37.

5



REFER TO SCHEDULE WEEKS

Materiel Procurement Set-Back Schedule

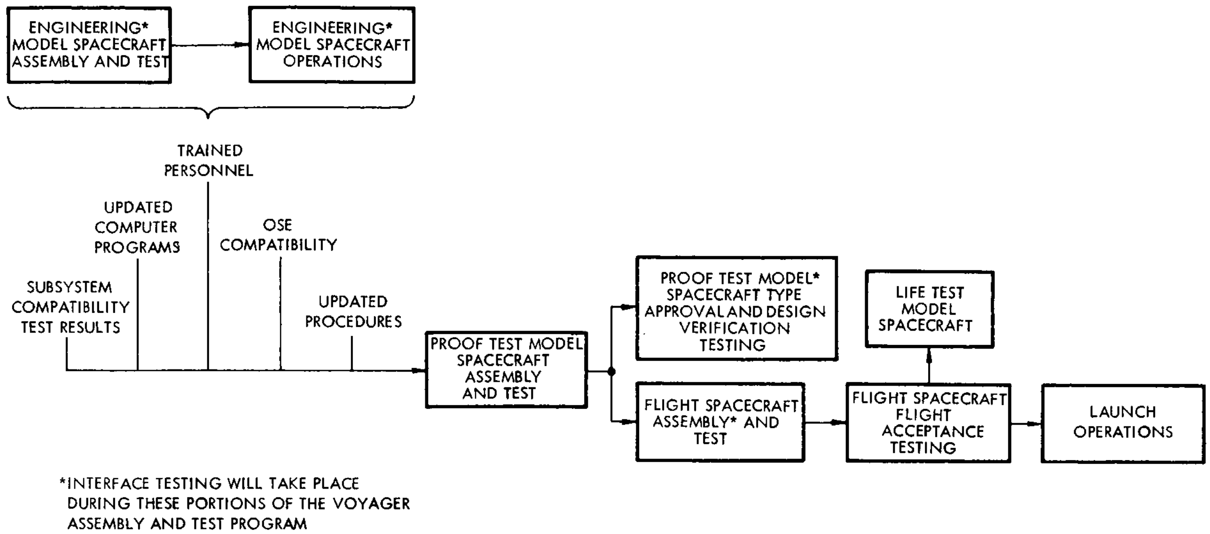


Figure 5-38. Voyager Spacecraft Top Assembly Flow

6.2.3 Data Management

The large amount of spacecraft performance data generated in the course of assembly and test operations requires the formation of an engineering group to control and identify the data. The spacecraft test data will be identified, time tagged, reduced (as required), quick-look data issued for analysis, and final data packages prepared. This group will also operate and maintain the data centers.

6.2.4 Operations Planning and Control

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 equipment for assembly into the spacecraft, and test facilities schedules. The controls effort includes the storage and maintenance of configuration status of all spacecraft hardware in accordance with the latest configurations. This group also provides support in expediting the delivery of equipment for use in spacecraft operations.

Figure 5-38 presents a top assembly, checkout, and test flow diagram which identifies the arrangement and sequence among the various spacecraft models. Figure 5-39 shows a brief pictorial flow of the major elements of the assembly and test flow.

6.2 Operations Engineering

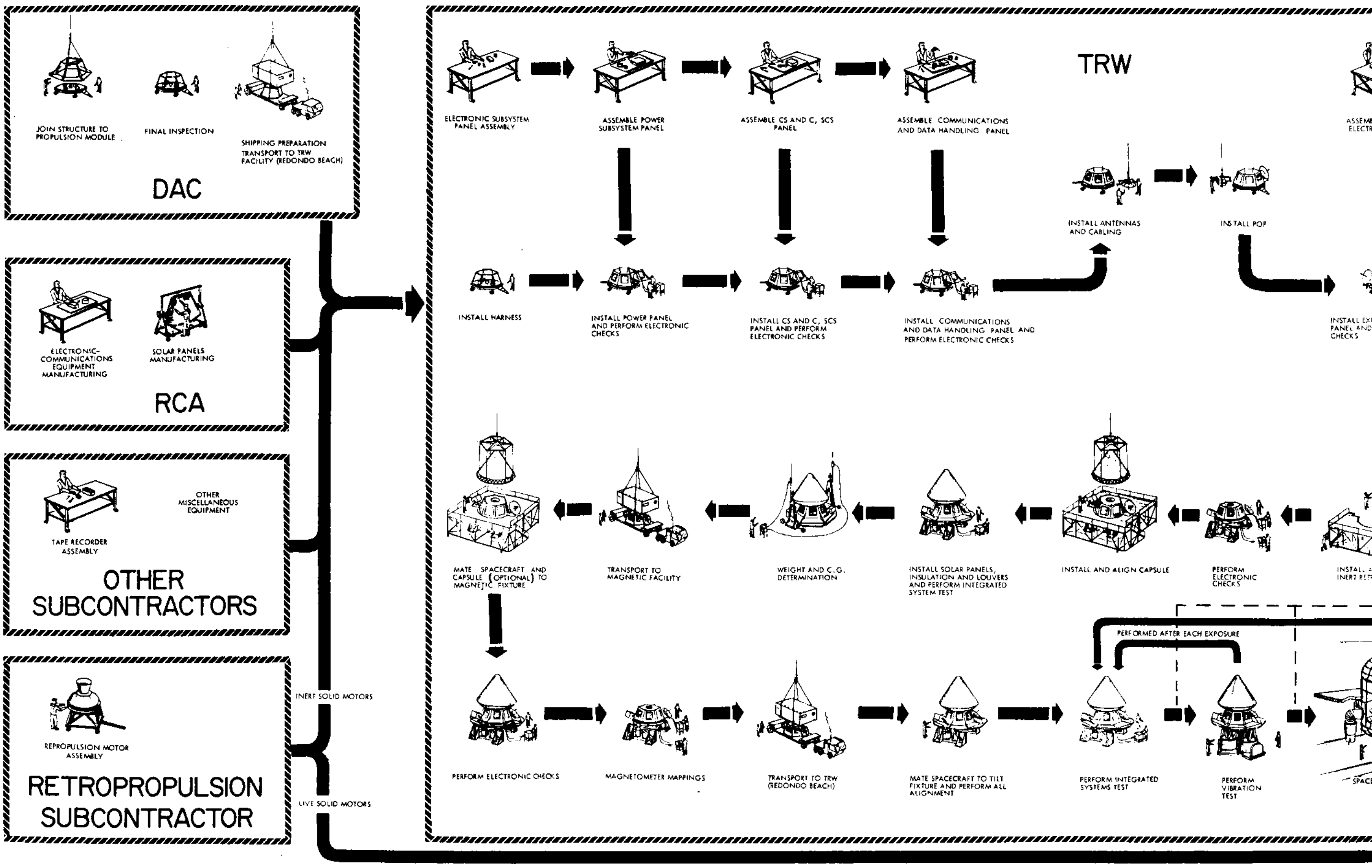
6.2.1 Design Integration

A major engineering task associated with spacecraft assembly and test is that of active interaction with the spacecraft design and development. To this end 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 ease of assembly and test. The results of these studies are used to feed back information to the design areas (in the event of operational problems) and as the basis for detailed design of the assembly and test operations procedures, facilities, test equipment, and computer programs.

6.2.2 Operations Design

The operations design task includes the detailed analysis of the assembly, checkout, and test requirements as determined by the spacecraft system design. The engineering personnel who participated in the initial design effort form the nucleus of this group, and the group is augmented by other specialists from the spacecraft assembly and test laboratory. The analyses of the assembly, checkout, and test requirements are used to design a detailed plan covering the identification and preparation of operating procedures, the detailed sequence of operations, the design of the test setup and special test facilities, the design and implementation of computer programs, and the assembly, checkout, and test schedules. Continuous updating and redesign of these elements is performed during the assembly and test phase.

Personnel of this group then form the nucleus of the assembly and test crews, under the direction of the spacecraft test manager.


JOIN STRUCTURE TO PROPULSION MODULE
FINAL INSPECTION
SHIPPING PREPARATION
TRANSPORT TO TRW FACILITY (REDONDO BEACH)
DAC
ELECTRONIC-COMMUNICATIONS EQUIPMENT MANUFACTURING
SOLAR PANELS MANUFACTURING
RCA
TAPE RECORDER ASSEMBLY
OTHER MISCELLANEOUS EQUIPMENT
OTHER SUBCONTRACTORS
REPROPULSION MOTOR ASSEMBLY
RETROPROPULSION SUBCONTRACTOR
INERT SOLID MOTORS
LIVE SOLID MOTORS
TRW
ELECTRONIC SUBSYSTEM PANEL ASSEMBLY
ASSEMBLE POWER SUBSYSTEM PANEL
ASSEMBLE CS AND C, SCS PANEL
ASSEMBLE COMMUNICATIONS AND DATA HANDLING PANEL
INSTALL HARNESS
INSTALL POWER PANEL AND PERFORM ELECTRONIC CHECKS
INSTALL CS AND C, SCS PANEL AND PERFORM ELECTRONIC CHECKS
INSTALL COMMUNICATIONS AND DATA HANDLING PANEL AND PERFORM ELECTRONIC CHECKS
INSTALL ANTENNAS AND CABLING
INSTALL POP
MATE SPACECRAFT AND CAPSULE (OPTIONAL) TO MAGNETIC FIXTURE
TRANSPORT TO MAGNETIC FACILITY
WEIGHT AND C.G. DETERMINATION
INSTALL SOLAR PANELS, INSULATION AND LOUVERS AND PERFORM INTEGRATED SYSTEM TEST
INSTALL AND ALIGN CAPSULE
PERFORM ELECTRONIC CHECKS
PERFORMED AFTER EACH EXPOSURE
PERFORM ELECTRONIC CHECKS
MAGNETOMETER MAPPINGS
TRANSPORT TO TRW (REDONDO BEACH)
MATE SPACECRAFT TO TILT FIXTURE AND PERFORM ALL ALIGNMENT
PERFORM INTEGRATED SYSTEMS TEST
PERFORM VIBRATION TEST

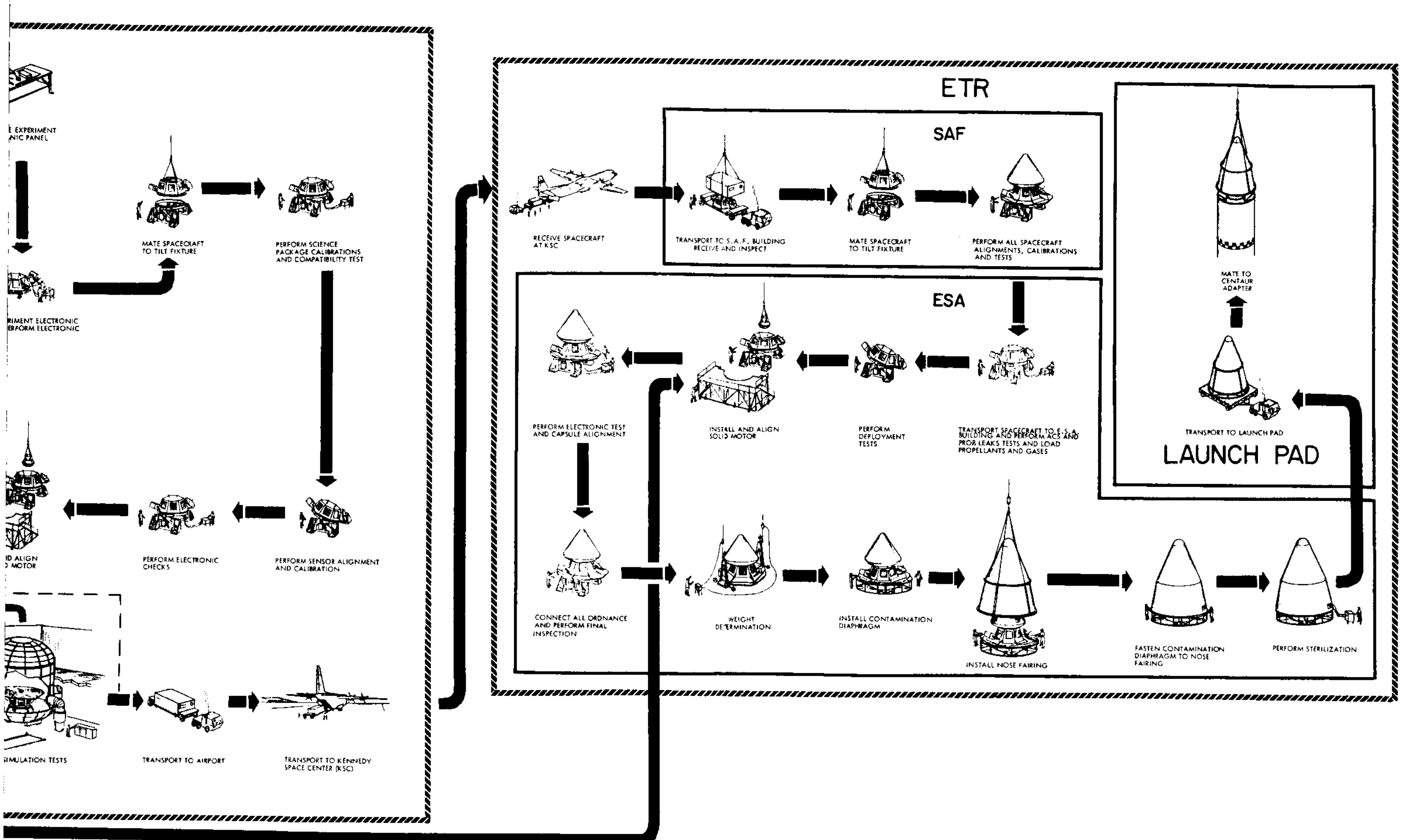


Figure 5-39. Voyager Planetary Vehicle Assembly and Checkout Operations

2

6.3 1971 Spacecraft Engineering Model Assembly, Checkout, and Test

6.3.1 Introduction

This section provides a description of the assembly and test operations planned for the engineering model spacecraft. The configuration of the engineering model is described and the interface testing tasks are identified. Finally, a more detailed description of the assembly and test operation is presented. Figure 5-40 shows a flow diagram which identifies the sequence of tasks.

a. Configuration

The subsystem configuration of the engineering model spacecraft is as follows:

a) Power, less solar arrays

b) Communications and data handling

c) Stabilization and control

d) Central sequencer and command

e) Pyrotechnics

f) Midcourse engine

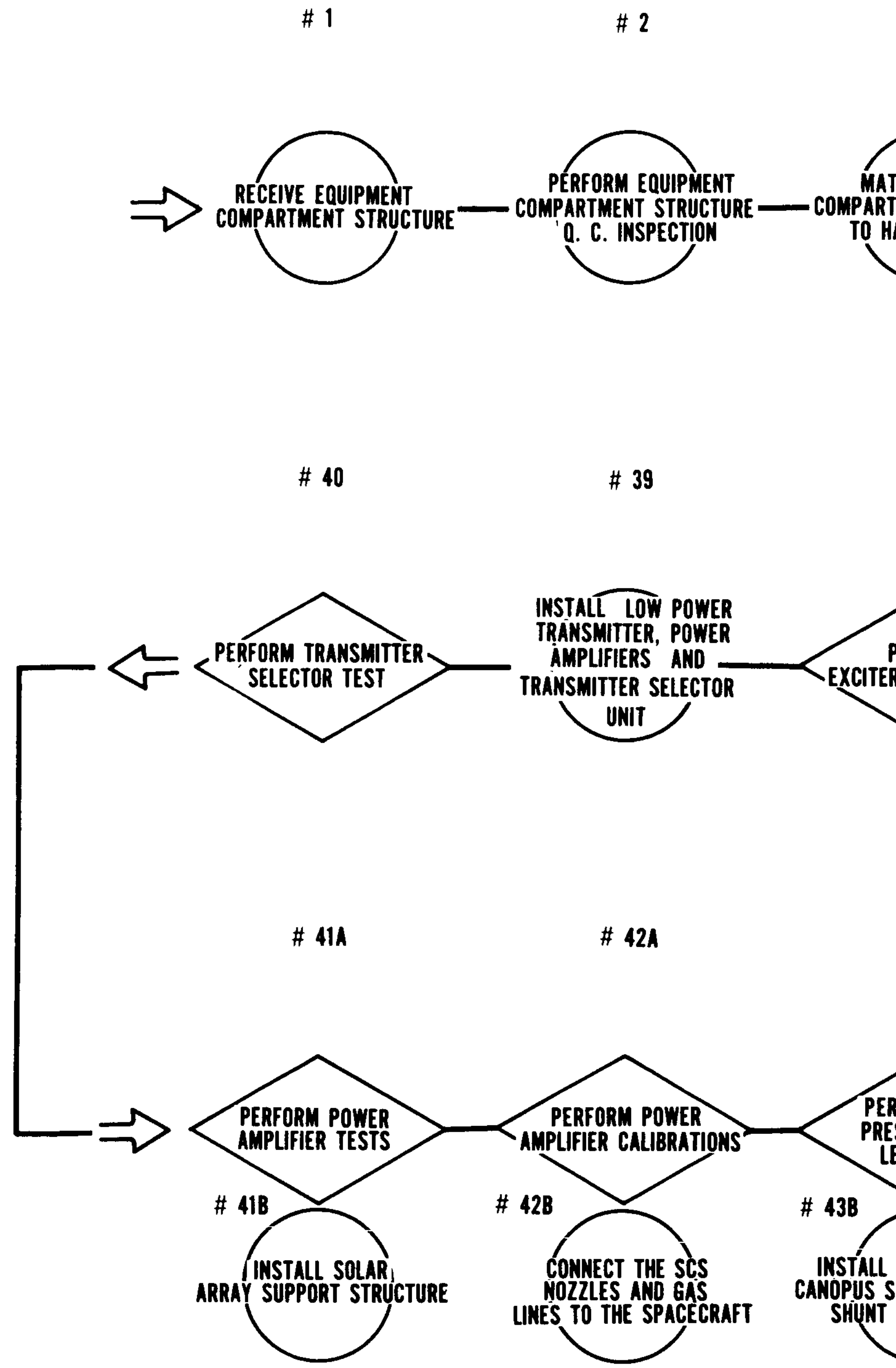
g) Inert solid motor, including an operational thrust vector control system

h) Planet-oriented package, less experiments

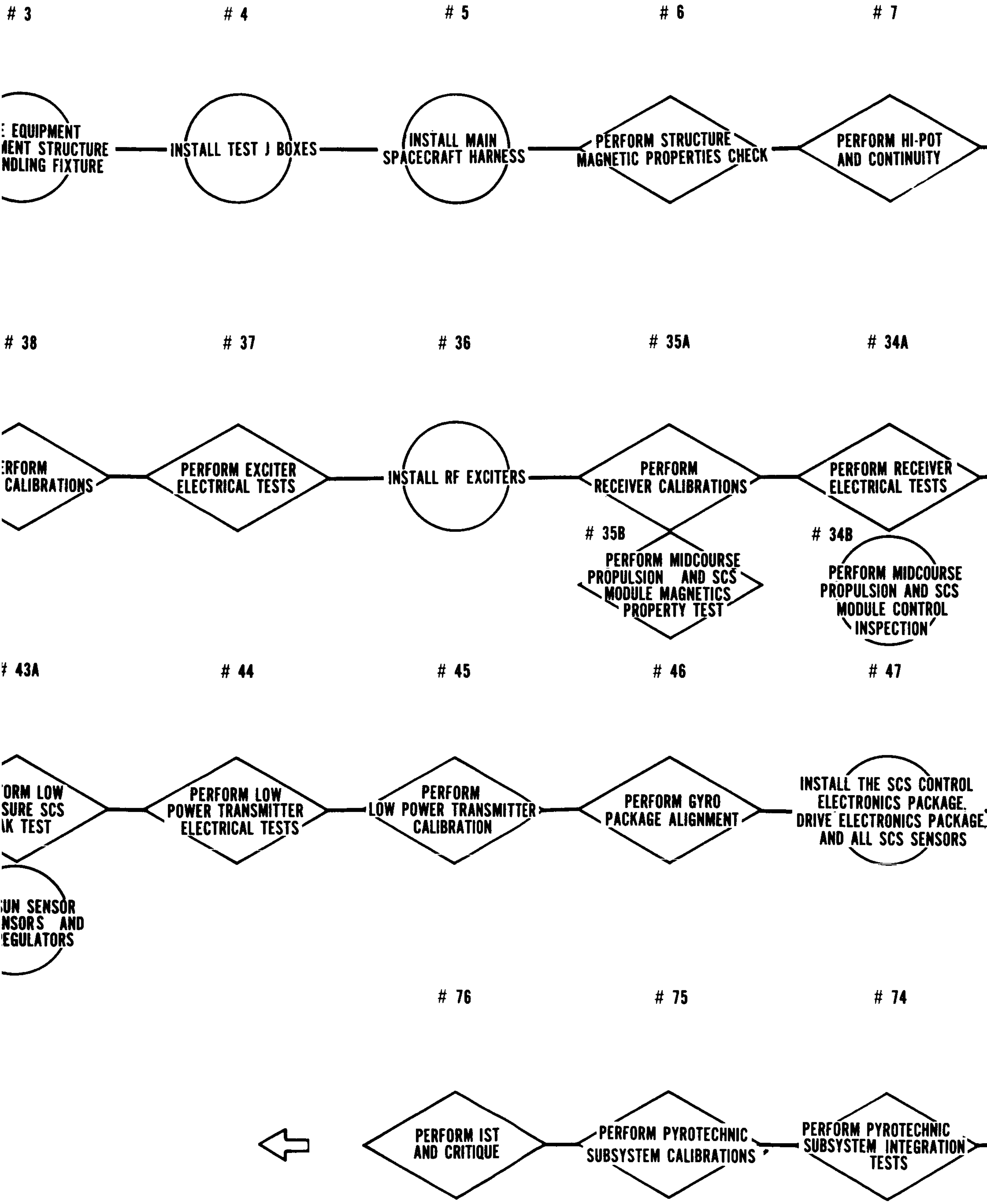
i) Experiments (it is not planned to install experiments in the engineering model permanently, but some experiments will be installed for the purpose of an early compatibility test)

j) The capsule subsystem will not be installed: a dummy capsule will be installed for match mate and nose fairing clearance checks

k) The thermal control subsystem will not be installed

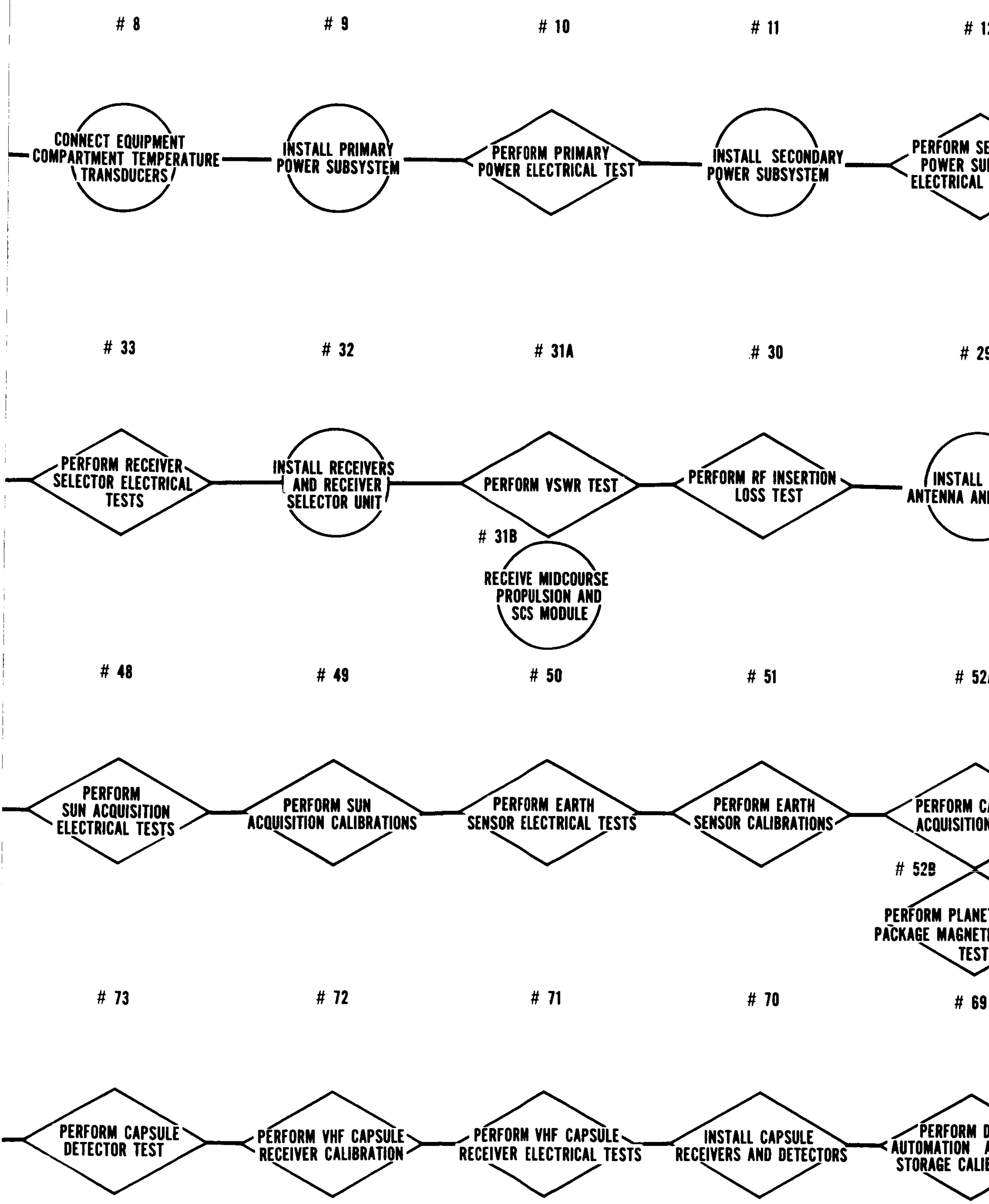

1
RECEIVE EQUIPMENT COMPARTMENT STRUCTURE
2
PERFORM EQUIPMENT COMPARTMENT STRUCTURE Q. C. INSPECTION
40
PERFORM TRANSMITTER SELECTOR TEST
39
INSTALL LOW POWER TRANSMITTER, POWER AMPLIFIERS AND TRANSMITTER SELECTOR UNIT
41A
PERFORM POWER AMPLIFIER TESTS
42A
PERFORM POWER AMPLIFIER CALIBRATIONS
41B
INSTALL SOLAR ARRAY SUPPORT STRUCTURE
42B
CONNECT THE SCS NOZZLES AND GAS LINES TO THE SPACECRAFT
43B

0

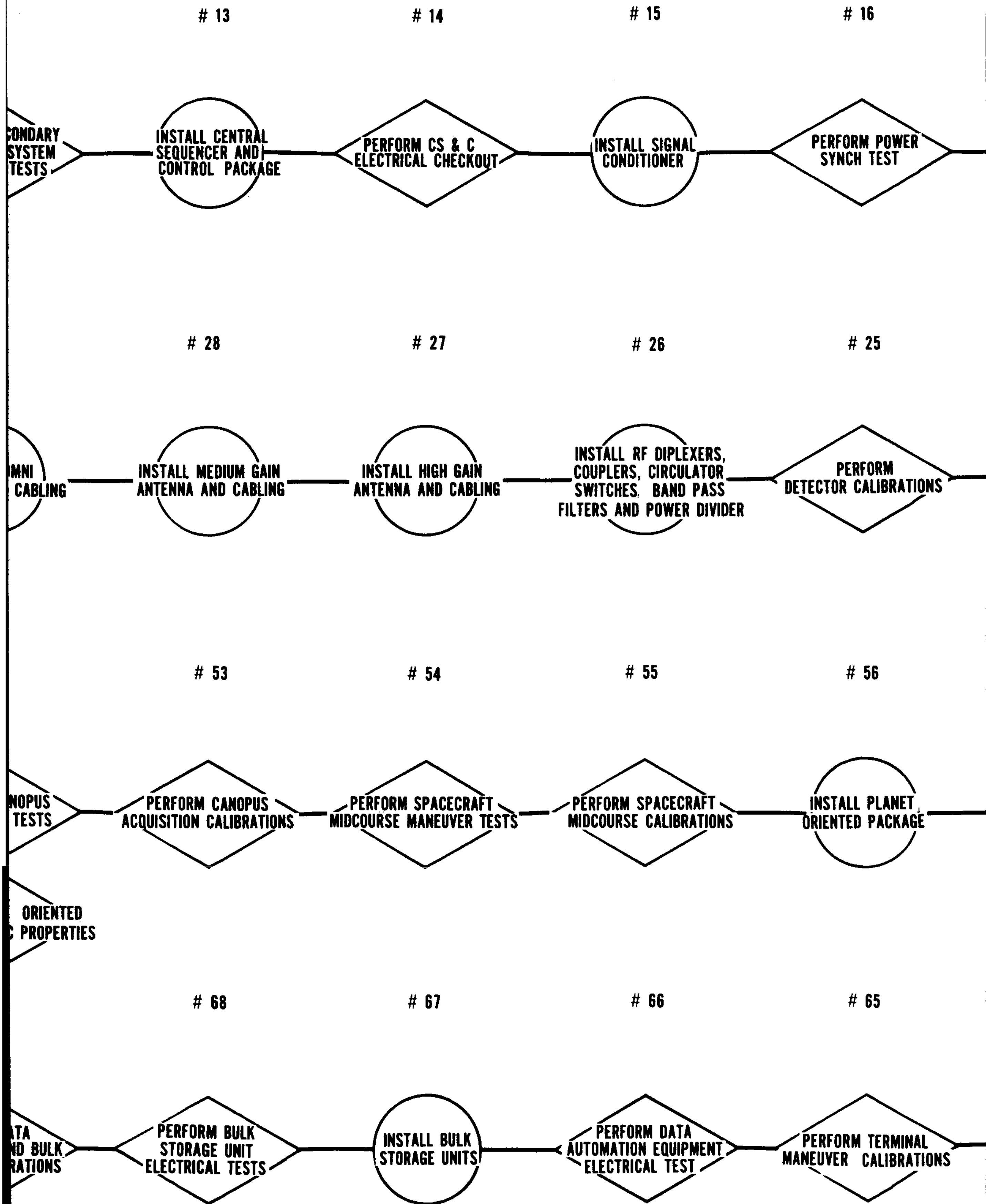


3
: EQUIPMENT
MENT STRUCTURE
NDLING FIXTURE
4
INSTALL TEST J BOXES
5
INSTALL MAIN SPACECRAFT HARNESS
6
PERFORM STRUCTURE MAGNETIC PROPERTIES CHECK
7
PERFORM HI-POT AND CONTINUITY
38
:RFORM
CALIBRATIONS
37
PERFORM EXCITER ELECTRICAL TESTS
36
INSTALL RF EXCITERS
35A
PERFORM RECEIVER CALIBRATIONS
35B
PERFORM MIDCOURSE PROPULSION AND SCS MODULE MAGNETICS PROPERTY TEST
34A
PERFORM RECEIVER ELECTRICAL TESTS
34B
PERFORM MIDCOURSE PROPULSION AND SCS MODULE CONTROL INSPECTION
43A
'ORM LOW
SURE SCS
AK TEST
UN SENSOR
NSORS AND
EGULATORS
44
PERFORM LOW POWER TRANSMITTER ELECTRICAL TESTS
45
PERFORM LOW POWER TRANSMITTER CALIBRATION
46
PERFORM GYRO PACKAGE ALIGNMENT
47
INSTALL THE SCS CONTROL ELECTRONICS PACKAGE, DRIVE ELECTRONICS PACKAGE, AND ALL SCS SENSORS
76
PERFORM IST AND CRITIQUE
75
PERFORM PYROTECHNIC SUBSYSTEM CALIBRATIONS
74
PERFORM PYROTECHNIC SUBSYSTEM INTEGRATION TESTS

2



8
CONNECT EQUIPMENT COMPARTMENT TEMPERATURE TRANSDUCERS
9
INSTALL PRIMARY POWER SUBSYSTEM
10
PERFORM PRIMARY POWER ELECTRICAL TEST
11
INSTALL SECONDARY POWER SUBSYSTEM
1
PERFORM SE POWER SU ELECTRICAL
33
PERFORM RECEIVER SELECTOR ELECTRICAL TESTS
32
INSTALL RECEIVERS AND RECEIVER SELECTOR UNIT
31A
PERFORM VSWR TEST
31B
RECEIVE MIDCOURSE PROPULSION AND SCS MODULE
30
PERFORM RF INSERTION LOSS TEST
2
INSTALL ANTENNA AN
48
PERFORM SUN ACQUISITION ELECTRICAL TESTS
49
PERFORM SUN ACQUISITION CALIBRATIONS
50
PERFORM EARTH SENSOR ELECTRICAL TESTS
51
PERFORM EARTH SENSOR CALIBRATIONS
52
PERFORM C ACQUISITION
52B
PERFORM PLANE PACKAGE MAGNET TEST
73
PERFORM CAPSULE DETECTOR TEST
72
PERFORM VHF CAPSULE RECEIVER CALIBRATION
71
PERFORM VHF CAPSULE RECEIVER ELECTRICAL TESTS
70
INSTALL CAPSULE RECEIVERS AND DETECTORS
69
PERFORM D AUTOMATION A STORAGE CALI



13
14
15
16
CONDARY
SYSTEM
TESTS
INSTALL CENTRAL SEQUENCER AND CONTROL PACKAGE
PERFORM CS & C ELECTRICAL CHECKOUT
INSTALL SIGNAL CONDITIONER
PERFORM POWER SYNCH TEST
28
27
26
25
MNI
CABLING
INSTALL MEDIUM GAIN ANTENNA AND CABLING
INSTALL HIGH GAIN ANTENNA AND CABLING
INSTALL RF DIPLEXERS, COUPLERS, CIRCULATOR SWITCHES, BAND PASS FILTERS AND POWER DIVIDER
PERFORM DETECTOR CALIBRATIONS
53
54
55
56
NOPUS
TESTS
PERFORM CANOPUS ACQUISITION CALIBRATIONS
PERFORM SPACECRAFT MIDCOURSE MANEUVER TESTS
PERFORM SPACECRAFT MIDCOURSE CALIBRATIONS
INSTALL PLANET ORIENTED PACKAGE
ORIENTED
PROPERTIES
68
67
66
65
TA
D BULK
RATIONS
PERFORM BULK STORAGE UNIT ELECTRICAL TESTS
INSTALL BULK STORAGE UNITS
PERFORM DATA AUTOMATION EQUIPMENT ELECTRICAL TEST
PERFORM TERMINAL MANEUVER CALIBRATIONS

4

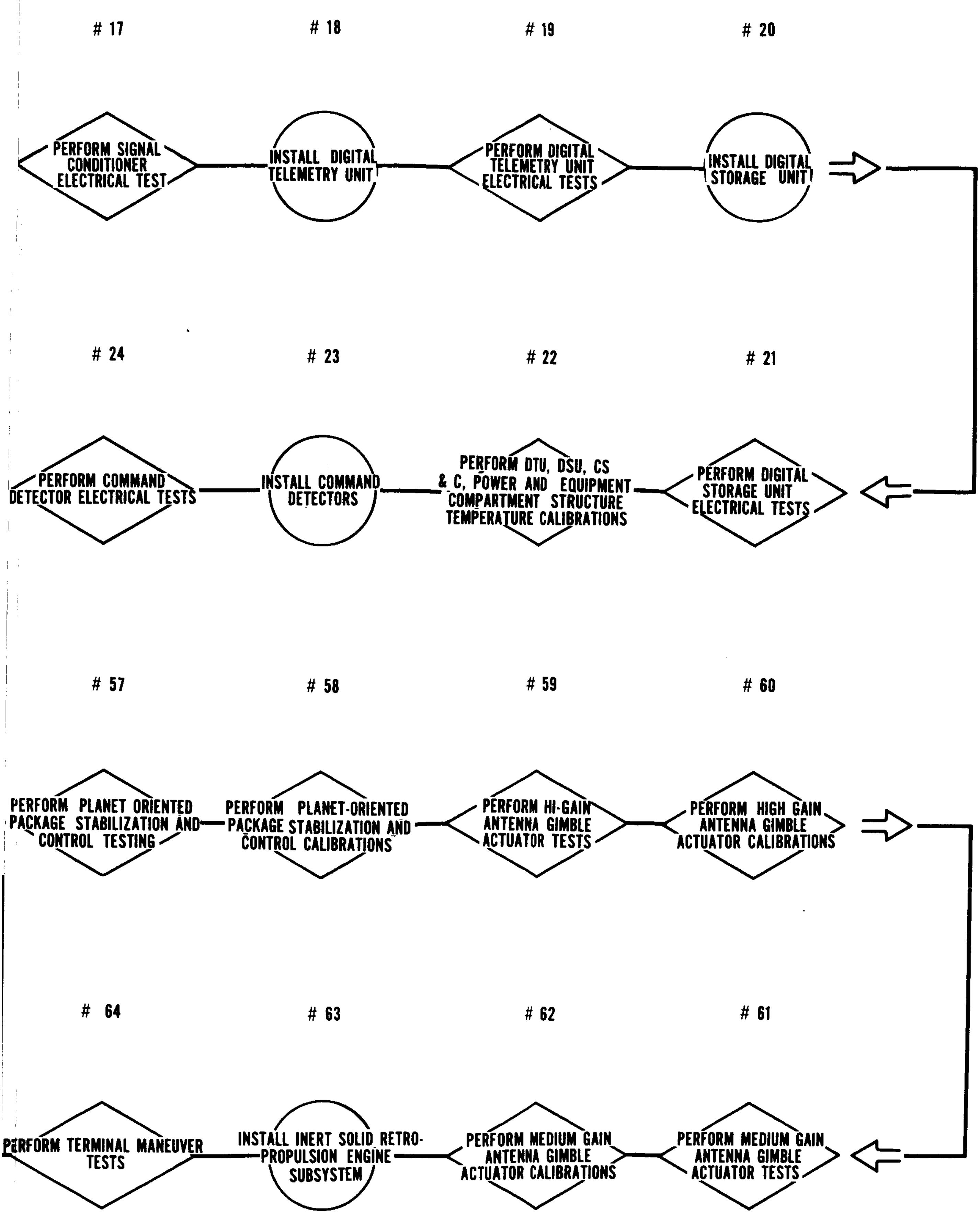


Figure 5-40. 1971 Engineering Model Spacecraft Assembly and Test

5

b. Spacecraft EM Tasks

The primary functions of the spacecraft engineering model (Figure 5-41) are to establish system and subsystem compatibilities, verify and validate OSE compatibilities with the spacecraft, to provide the PTM and flight spacecrafts with trained personnel, and to provide operational spacecraft procedures and computer programs for in-house testing, launch operations, and DSIF and SFOF operations.

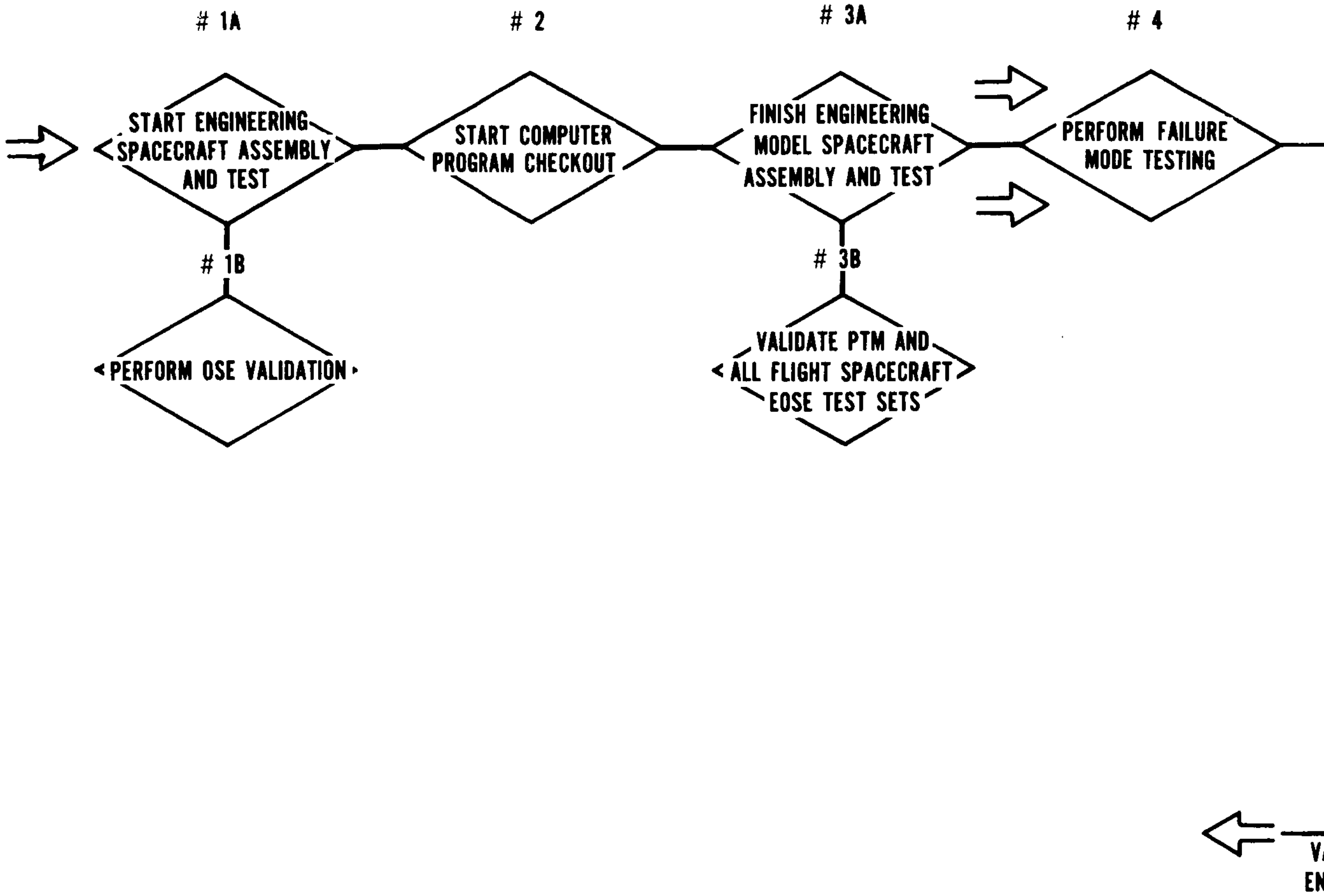
The tasks planned for the spacecraft engineering model are as follows:

- Establishing system and subsystem compatibilities
- Early checkout of the spacecraft electrical and mechanical OSE
- Personnel familiarization and training
- Debugging and bench checkout of all computer programs
- Debugging and checkout of all specialized OSE and cabling such as: thermal vacuum (space simulation), vibration, acceleration, acoustical, magnetic properties, launch site (primarily with PTM)
- Debugging and checkout of TRW-supplied DSN and mission dependent equipment
- Match mate with Centaur stage and nose fairing
- Nose fairing RF coupler loss determination

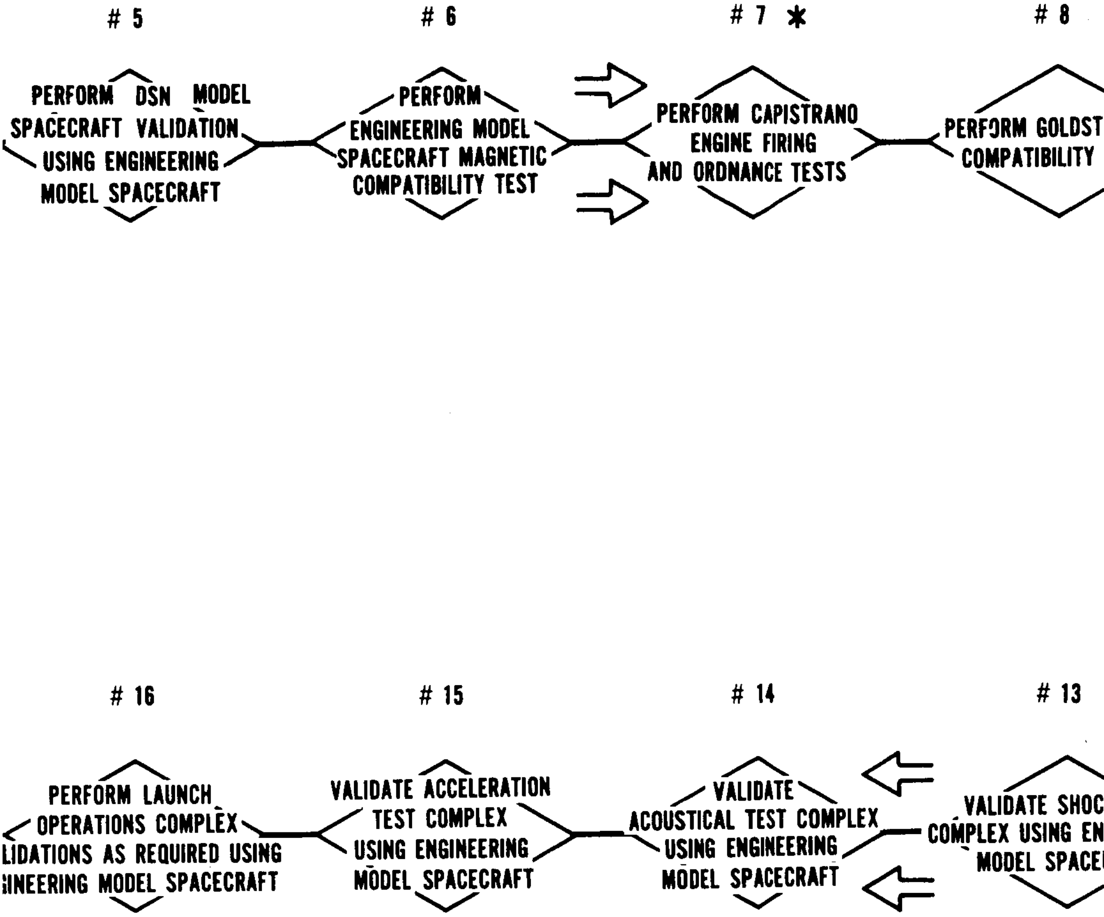
6.3.2 Spacecraft Engineering Model Assembly and Checkout Procedure

a. General

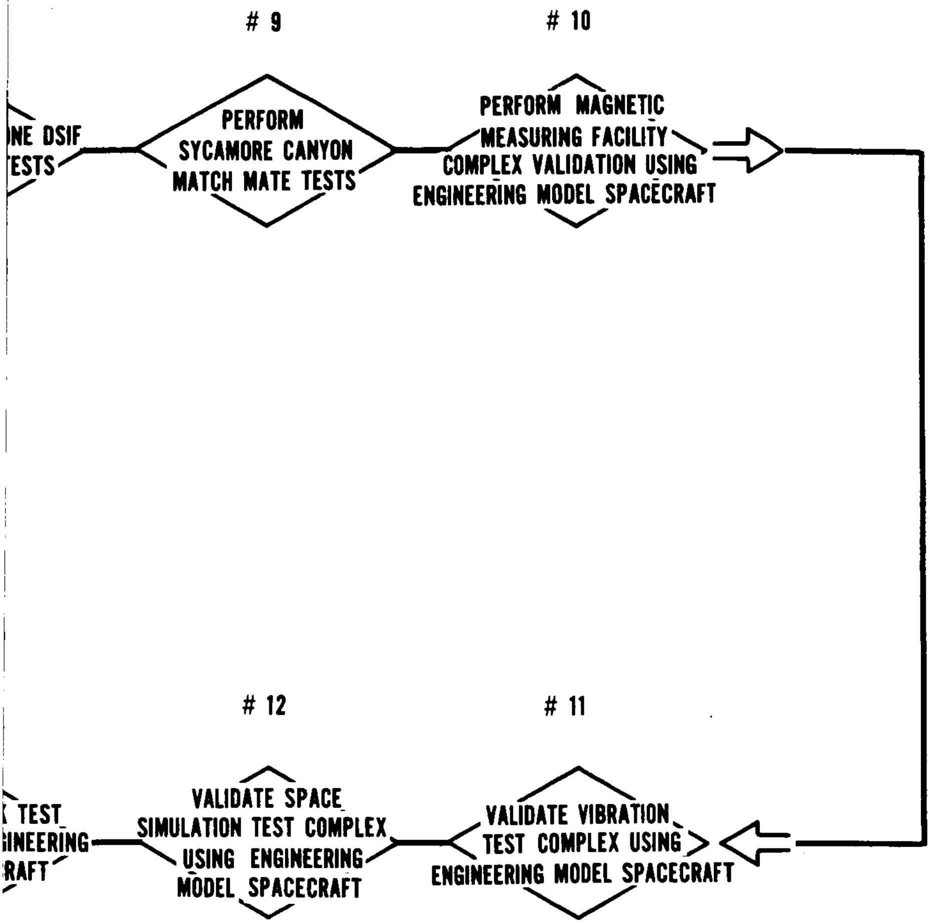
One basic policy adopted during spacecraft assembly and checkout operations is that the subsystem assembly and checkout operations are conducted off the spacecraft assembly line on their respective equipment mounting panels. The advantage of this approach is that of conserving



1A
START ENGINEERING SPACECRAFT ASSEMBLY AND TEST
1B
PERFORM OSE VALIDATION
2
START COMPUTER PROGRAM CHECKOUT
3A
FINISH ENGINEERING MODEL SPACECRAFT ASSEMBLY AND TEST
3B
VALIDATE PTM AND ALL FLIGHT SPACECRAFT EOSE TEST SETS
4
PERFORM FAILURE MODE TESTING



* NOTE PERFORMED WITH



THE PROPULSION AND STABILIZATION CONTROL ENGINEERING MODEL SPACECRAFT

Figure 5-41 Engineering Model Spacecraft Operations

3

schedule time with the subsystem assembly and checkout operations conducted in parallel with spacecraft operations. The other basic policy requires that the assembly and test sequence be logically ordered so as to minimize the need for repeating portions of tests previously completed or for breaking an already validated connector. This results in a sequence which begins with the installation of the spacecraft harness for accessibility reasons, the next addition being the power subsystem to provide the proper power for subsequent subsystems, etc. The sequence chosen based on this logic is shown in Figure 5-40.

The spacecraft equipment compartment structure, after having been received from Douglas, will be inspected for damage from shipping and handling operations. The equipment compartment structure will be mechanically mated to the handling fixture.

b. Power Subsystem

The first subsystem to be installed and electrically integrated will be the power subsystem for the reasons stated above. The electrical checkout will be split into two parts: the primary and the secondary power subsystem. After the EOSE electrical interfaces with the spacecraft have been checked, the primary power subsystem checkout will be initiated. Basically, the primary power subsystem tests consist of verifying that the solar array power can be controlled such that it can supply the proper charge to the battery and at the same time sustain the spacecraft load demands. The secondary power subsystem tests consist of verifying that the secondary power supply outputs are within specification as the spacecraft primary power bus is varied within specifications. The power subsystem will have incorporated a sufficient number of test points at each individual black box such that system noise and transients can be effectively monitored.

c. Central Sequencer and Control

After the power subsystem has been tested, the central sequencer and control subsystem will be installed in order to provide for power switching and subsequent signal switching. Thus this provides a means for end-to-end checking as the spacecraft assembly progresses, in contrast to intermediate checking between subsystems. The CS&C testing consists of ascertaining that commands can be properly received from the command detector and acted upon and verifying that the internally timed commands are sent and acted upon properly. The CS&C power input lines and appropriate signal lines will be made available via test points on the individual black boxes so that system noise and transients may be effectively monitored.

d. Signal Conditioner

The signal conditioner is installed next to accommodate the processing of telemetry analog data.

e. Digital Telemetry Unit

The digital telemetry unit is installed next. The testing philosophy utilized for the remainder of the spacecraft assembly and test phase is that as each black box in integrated into the spacecraft, its telemetry calibration will be accomplished concurrently.

The Digital Telemetry Unit Electrical tests consist of ascertaining that the input data to the DTU is proper and that the output data is in the proper format for all DTU modes and bit rates with the correct word value. All DTU timing signals will be checked for the correct amplitude, rise and fall time, frequency, and pulse width.

f. Digital Storage Unit

The digital storage unit is installed and tested to ascertain that telemetry data words can be properly stored and read out for all DTU formats, modes, and bit rates. At this point it is possible for data to be transmitted or stored via hardline for any DTU format, mode, or bit rate and the bench check of all computer programs can commence. The computer program bench checks are to be done in parallel with the normal spacecraft assembly and test operations. Computer programs will be made identical, whenever possible, to those required for the DSIF and SFOF operations to simplify the writing of computer programs used during mission operations.

A sufficient number of DTU and DSU test points will be made available on the individual black boxes such that all telemetered parameters can be properly calibrated and system noise and transients can be monitored. There will be a sufficient number of telemetry transducers in each black box so that the operational status of each subsystem can be monitored with a minimum of hardlines. Historically, there is never a sufficient number of spacecraft transducers. The advantage to having sufficient test points and telemetry transducers is that it is not necessary to disconnect spacecraft cables for telemetry calibrations and noise and transient investigations, thus preserving configuration validation (and also saving wear on connectors).

g. Command Detectors

The command detectors are installed to establish an operational RF up-link system. The command detectors will be checked to ascertain that the detectors, after receiving ground commands, can properly act upon and execute them. While each ground command is being acted upon by the command detectors, the CS&C interface will be monitored noting that the CS&C reacts properly to each ground command.

h. Communication Equipment

All spacecraft antennas and cables will be installed at this time so that end-to-end RF VSWR and insertion loss tests can be performed.

After the VSWR and insertion loss tests have been completed, the receivers will be installed and electrically integrated. Each command will be transmitted from the ground transmitter via the RF link, noting proper reaction of the CS&C. Commands will also be transmitted through each antenna as part of the receiver electrical tests. Once it has been ascertained that commands can be transmitted to, and properly received by the spacecraft, the receiver threshold sensitivity will be determined.

A sufficient number of test points will be made available so all telemetered parameters can be properly calibrated and system noise and transients can be properly monitored.

At this point in the spacecraft testing, the RF up-link has been completely electrically checked and calibrated and all commands will be sent via RF link from the ground transmitter to the airborne receivers. Command hardlines will be used only for trouble shooting. Proper spacecraft reception of commands will be verified by monitoring the spacecraft reaction to each command and the command blip strip via telemetry.

With the spacecraft RF up-link established, transmitter selector, exciters, low power transmitter, and the power amplifiers will be installed and electrically integrated. The capability of the transmitter to SELECTOR TO select each power amplifier and the low gain transmitter will be checked by monitoring each CS&C output to the transmitter selector noting that the selector output is proper for each command. Each transmitter will be selected and the RF power output and frequency monitored. In addition, each transmitter RF output will be monitored for spurious harmonics. The transmitter will be modulated at each bit rate by the DTU output. While each transmitter is being modulated, the modulation index will be checked at each bit rate. The transmitters will then be connected to the spacecraft antenna system. The RF down-link having been completely integrated, the RF signal will be demodulated at the telemetry EOSE and processed. Henceforth, all telemetry will be processed via air link from the spacecraft transmitters to the ground receiver. Telemetry hardlines will only be used for trouble shooting or when the ground receiver is being interfered with. Hardline data will always be recorded during spacecraft tests.

A sufficient number of test points will be made available such that all telemetered parameters can be calibrated and so that system noise and transients can be properly monitored.

i. Pneumatics

The midcourse propulsion and stabilization and control pneumatics module will then be attached to the equipment compartment structure and the pressure transducers calibrated via telemetry.

j. Stabilization and Control

The stabilization and control subsystem is now in a position to be completely installed and electrically integrated. After all of the SCS electronic boxes and sensors have been installed, the sun acquisition portion of the stabilization and control subsystem will be electrically integrated. This portion of the SCS subsystem is divided into three basic parts:

- Gyro electrical integration
- Fine and coarse sun sensor integration
- Telemetry calibrations

The gyro package and SCS electronics after being electrically integrated will be mechanically torqued using the spacecraft tilt fixture. While the gyro package is being torqued, the gas jet actuations will be monitored for proper polarity, and the gyro rate at which the gas ceases to actuate will be determined.

After each sun sensor has been electrically integrated, it will be illuminated using the sun sensor EOSE, and the gas jet actuation will be monitored for proper polarity.

The calibration of the sun acquisition mode requires calibrations of the following parameters: gyro on-off signal, gyro generator outputs, valve actuations, sun sensor intensity, and all SCS electronics package temperatures. Each package used for sun acquisition testing and calibrations will have sufficient test points so that calibrations can be performed and noise and transient measurements properly made.

After the earth sensor and its electronics have been electrically integrated, the earth sensor will be illuminated using the earth sensor EOSE. While the earth sensor is being illuminated, its signal amplitude will be monitored as a go-no-go function. The earth sensor calibrations will be accomplished primarily by signal injection.

The third portion of the SCS integration and test is Canopus acquisition. After the Canopus sensor and electronics have been electrically integrated, the sensors will be illuminated using the Canopus sensor EOSE. The resulting gas jet actuations will be monitored for proper polarity. The Canopus sensor calibrations will be accomplished by signal injection. Sufficient test points will be provided to allow for Canopus acquisition calibrations and transient and noise monitoring.

The present policy for calibrating the stabilization and control sensors such as gyros, sun sensors, earth sensors, and Canopus sensors is as follows. Each sensor will be supplied to the assembly and test facility with a set of laboratory bench calibration curves. As previously mentioned, the calibration of these sensors is accomplished by signal injection, i.e., the sensor will be replaced by a suitable signal generator. The signal generator voltage amplitude will be varied and the corresponding telemetry word monitored. The telemetry word values and the generator voltage along with the laboratory bench calibrations will be inserted into the computer programs. The disadvantage to this approach is that the sensors have to be removed from the spacecraft for calibration checks; the advantage is that large quantities of complicated EOSE are not necessary as part of the systems test set EOSE since the final calibrations are done in the laboratory. Furthermore, the necessary spacecraft system test EOSE simulation for each sensor simply becomes an on-off stimulus whose amplitude or intensity does not become important. However, an investigation will be undertaken during Phase IB to ascertain whether the SCS sensors can be adequately stimulated while mechanically mated to the spacecraft.

The spacecraft midcourse maneuver equipment is the next portion of the stabilization and control subsystem to be electrically integrated into the spacecraft. The midcourse maneuver testing is in three parts: spacecraft orientation changes, jet vane orientation, and midcourse motor burn duration. The roll and pitch turn magnitude and polarity will be transmitted to the spacecraft via RF link. After the turn commands have been transmitted, the gyro torquing current amplitude and time duration will be monitored for each polarity. While the gyros are being torqued, the gas jet actuations will be monitored for proper polarity; this information will be transmitted to the spacecraft and the resulting jet vane angle monitored. The midcourse motor burn duration information will be transmitted to the spacecraft and the midcourse motor stop and start signals time interval monitored. Sufficient test points will be made available so that all midcourse maneuver calibrations can be properly accomplished and noise and transients successfully monitored.

k. POP

At this stage of the SCS testing, the planet-oriented package will be attached to the spacecraft and electrically integrated. The POP package consists of the following units: planet-oriented package boom, planet-oriented package gimbal actuators, and the Mars horizon scanners. The POP experiments will not be installed in the engineering model spacecraft.

After the planet-oriented package subsystem has been installed and electrically integrated, the Mars horizon scanners will be stimulated using the horizon scanner EOSE and the reaction of the gimbal actuator measured. The horizon scanner is stimulated again so that the gimbal actuators slew in the opposite direction. This is repeated for the remaining gimbal actuator.

l. Antenna Gimballing

The high-gain and medium-gain antenna articulation tests are performed after the POP package articulation test. After the high- and medium-gain antennas have been electrically integrated, each gimbal actuator will be commanded to slew; the direction and slewing rate will be checked for each actuator. Each gimbal actuator will be commanded to slew in the opposite direction and the slew rate checked.

A sufficient number of test points will exist such that the POP package, high-gain and medium-gain antenna gimbal actuators can be properly calibrated and the noise and transients properly monitored.

m. <u>Solid Engine</u>

An inert solid motor is installed at this time since the thrust vector control must be made available to support the terminal maneuver portion of the SCS testing phase. The terminal maneuver portion of the SCS testing phase will be accomplished as follows. After the thrust vector control portion of the solid retropropulsion subsystem has been electrically integrated, the spacecraft will be rotated about the pitch axis by means of the tilt fixture. While the spacecraft is being rotated, the thrust vector control gas injectors will be monitored to ascertain that gas is flowing out of the proper injector. The spacecraft will be rotated in the opposite direction and the injectors monitored. The above will be repeated for spacecraft rotation about the yaw axis.

n. EXPERIMENT DATA HANDLING

The data automation and bulk storage subsystems will then be installed and electrically integrated; the rise and fall time, pulse width and pulse amplitude will be measured, using black box test points, and all timing signals, shift signals, sync signals and inhibit signals will be monitored. Once it has been ascertained that the data automation signals are within specification for all bit rates and modes, the capsule and experiment simulator will be connected to the spacecraft. The capsule and experiment simulator insures that both the data automation system and the computer programs are functioning properly.

After the data automation system testing has been completed, next the bulk storage units will be installed and electrically integrated; then the rise and fall time, pulsewidth and amplitude of the bulk storage input and output data signals will be monitored for all bit rates and modes. When it has been ascertained that the signals are within specification data from the capsule and experiment simulator will be read into the bulk storage unit. The capability to read data into the ground computer simultaneously while data is being read into the bulk storage unit will exist within the spacecraft for all modes and formats. The reason for this is that the data stored in the ground computer can then be compared bit by bit by a special computer subroutine with the data stored in the bulk storage unit.

A sufficient number of test points will exist such that the data automation and bulk storage subsystems can be adequately calibrated, and the noise and transients properly monitored.

o. VHF Communications

The capsule VHF receiver and detector will be installed and electrically integrated, after which the receiver sensitivity will be determined using the capsule simulator. The receiver signal will be modulated by the capsule simulator and the telemetered data fed into the ground computer via the S-band link. Concurrently, the computer data will be monitored for proper format and word values.

p. Pyrotechnics

The pyrotechnic subsystem testing will be accomplished as follows. It will be ascertained that the spacecraft is in the "safe" condition. Then each squib connector pin will be monitored for continuity to frame ground. Next, each squib will be commanded to the "fire" condition, and the firing voltage monitored. The pyrotechnic EOSE will be connected to each squib bridge wire interface. Each squib will again be commanded to the fire condition noting that the EOSE indicates an "all-fire" condition. This will be done when the battery is at its lower voltage limit. A sufficient number of test connectors will exist so that ordnance calibrations can be properly conducted, test points will exist so the "safe" or "armed" condition of each pyrotechnic device can be determined.

q. Integrated System Test

The last task to be performed as part of the engineering model assembly and test is the integrated system test. This task is designed to test the spacecraft to the fullest extent possible without breaking any spacecraft or EOSE connectors. The mission sequence of events will be closely followed and the spacecraft configuration will match the flight configurations.

The solar array simulated power will be varied to match the sun intensity levels that will be encountered during the various phases of the mission profile. The up and down link RF power levels will be varied to match the levels that would exist due to stabilization and control maneuvers and changes in distance between the spacecraft and the earth. Parameters such as Canopus sensor cone angles, midcourse and terminal maneuver turn angles, midcourse correction jet vane angles, and midcourse correction engine burn time will be varied during certain portions of the integrated system test to detect failures that might remain undetected if the same quantitive values for the above parameters were used for each phase of the integrated system test.

As a part of the integrated system test a practice countdown will be performed, including a free mode test. When the practice countdown progresses to the point of liftoff, the umbilical cable and all other test cables will be disconnected except the solar array simulated power connector. The spacecraft will be exercised in this manner up to and through the midcourse maneuver portion of the mission profile, using battery and solar array simulated power. This constitutes the free mode test and is used to verify proper spacecraft operations in the absence of OSE and umbilical cables.

During all integrated system tests telemetered data will be recorded on magnetic tape. All spacecraft data will be monitored and checked for proper values by a data team comprised of a subsystem representative from all subsystem areas, with JPL invited to participate.

TRW proposes a combination of RF and wire telemetry links between TRW and the SFOF operations in Pasadena and quick-look SFOF operation at the Goldstone DSIF. The participation of JPL personnel during integrated systems test and the data evaluation will provide training for later mission operations.

During the integrated system test a minimum of test cables and EOSE will be utilized since EOSE cables constitute a nonflight spacecraft configuration; this requires a sufficient amount of telemetry transducers so that the spacecraft subsystem can be adequately monitored without EOSE.

6.4 Engineering Model Operations

The spacecraft engineering model having completed assembly and checkout, will enter into the spacecraft engineering model operations phase (Figure 5-41), starting with failure mode testing. The failure mode test will investigate effects of selected failure modes and redundant circuit failures. When necessary, the engineering model black boxes will be opened and modified to effect the failures.

Next, a preliminary electromagnetic compatibility test will ascertain that there are no radiated or induced interfering signals with experiments, spacecraft subsystems, and launch vehicle. The spacecraft will be irradiated with the calculated design levels of RF signals.

The spacecraft engineering model will be shipped to the Goldstone DSIF facility to verify that the DSIF Goldstone and SFOF Pasadena software is compatible with spacecraft operations and that the spacecraft can be commanded from the DSIF Goldstone station.

The spacecraft engineering model will be transported to the Sycamore Canyon facility for launch vehicle electrical tests to test the mechanical interfaces between the Centaur and the spacecraft, including a nose fairing clearance test. All spacecraft umbilical functions will be checked using the launch pad EOSE; the RF nose fairing antenna coupling will be determined.

The next task is to use the EM to validate the magnetic properties test site. The validation would include specialized EOSE and MOSE and system test set, and specialized cabling. The vibration, space simulation, shock acoustical, and acceleration test facility complexes will also be validated using the engineering model spacecraft. As a final task the spacecraft engineering model will be shipped to the AFETR to support the launch facility area checkout as required.

TRW is investigating the desirability of transporting the spacecraft engineering model to a solar array testing facility, such as Table Mountain, to perform solar array spacecraft compatibility tests. The tests would involve powering the spacecraft from prototype arrays and monitoring battery charge control for various spacecraft load conditions.

Another test being investigated is the use of the spacecraft engineering model as a propulsion test vehicle to check stabilization and control subsystem performance during engine firing. The test requires altitude simulation to obtain meaningful data. A detailed study will be made during Phase IB to investigate techniques and facilities capable of supporting the test.

6.5 Deep Space Network Model Testing

The Deep Space Network model is a group of specialized test equipment consisting of the following items:

- Test transponder package
- Magnetic tapes
- Capsule telemetry simulator
- Capsule VHF transmitter

The test transponder simulates the spacecraft RF subsystem. The normal input and output RF connections to the DSIF station are made via the station test diplexer. The test transponder will be capable of being modulated by the magnetic tape recordings of biphase-modulated telemetry data and the capsule telemetry simulator. The capsule VHF transmitter will also be modulated by the capsule telemetry simulator.

The DSN model is a secondary means of testing the DSIF spacecraft interfaces, the primary method being the tests with the spacecraft engineering model at the Goldstone DSIF facility.

The mission dependent test equipment consists of the following items:

- PN generators
- Command encoders

- Test equipment, including an oscilloscope, frequency counter, RF power meter, RF signal generator, power supplies, spectrum analyzer, digital voltmeter, and vacuum tube voltmeter
- Spacecraft status displays
- RF patch panel
- Bit error rate checker
- Computer buffer

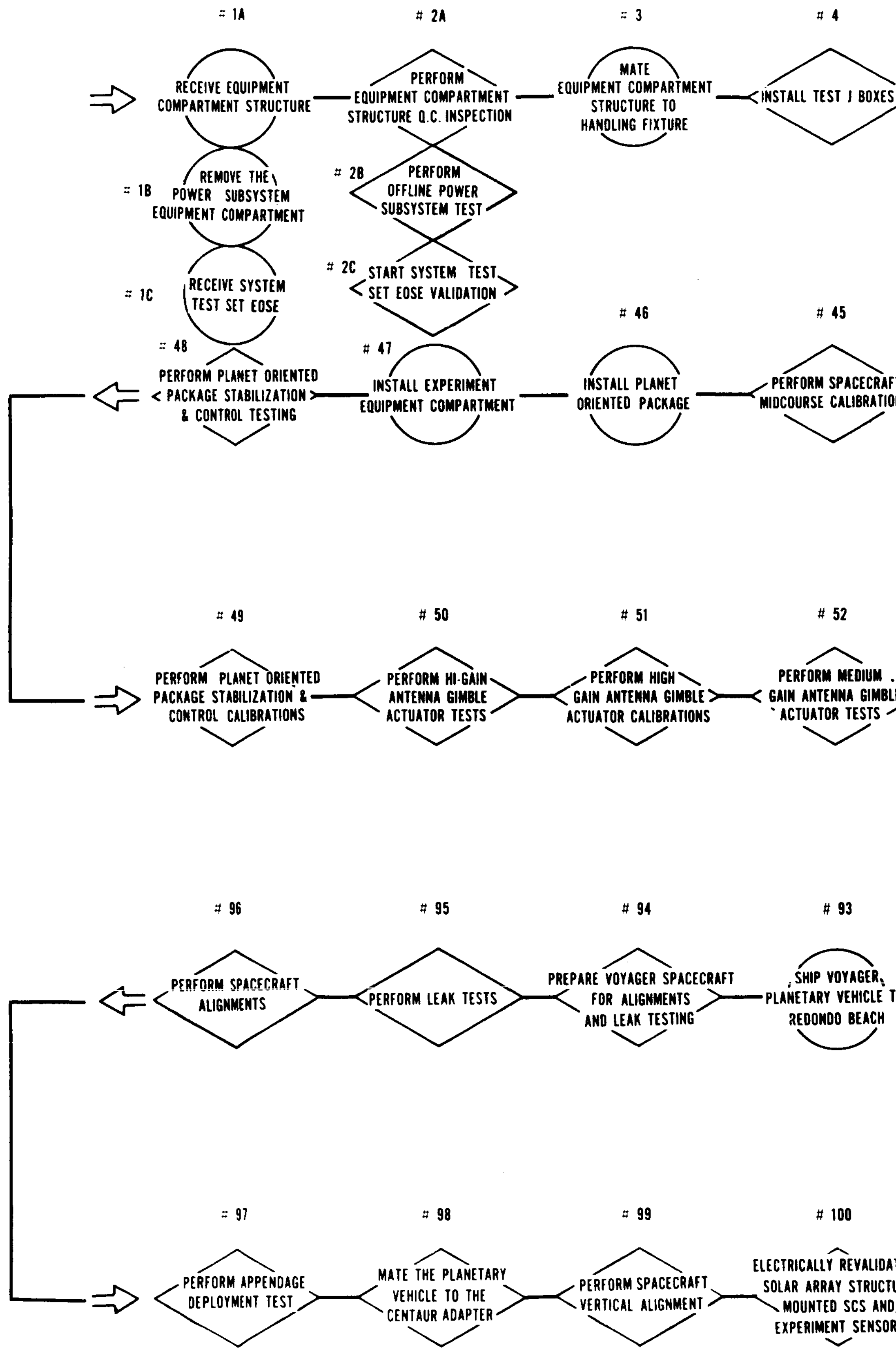
All of this equipment will be tested with the spacecraft engineering model at TRW and again when the spacecraft engineering model is delivered to the Goldstone DSIF station.

6.6 Proof Test Model

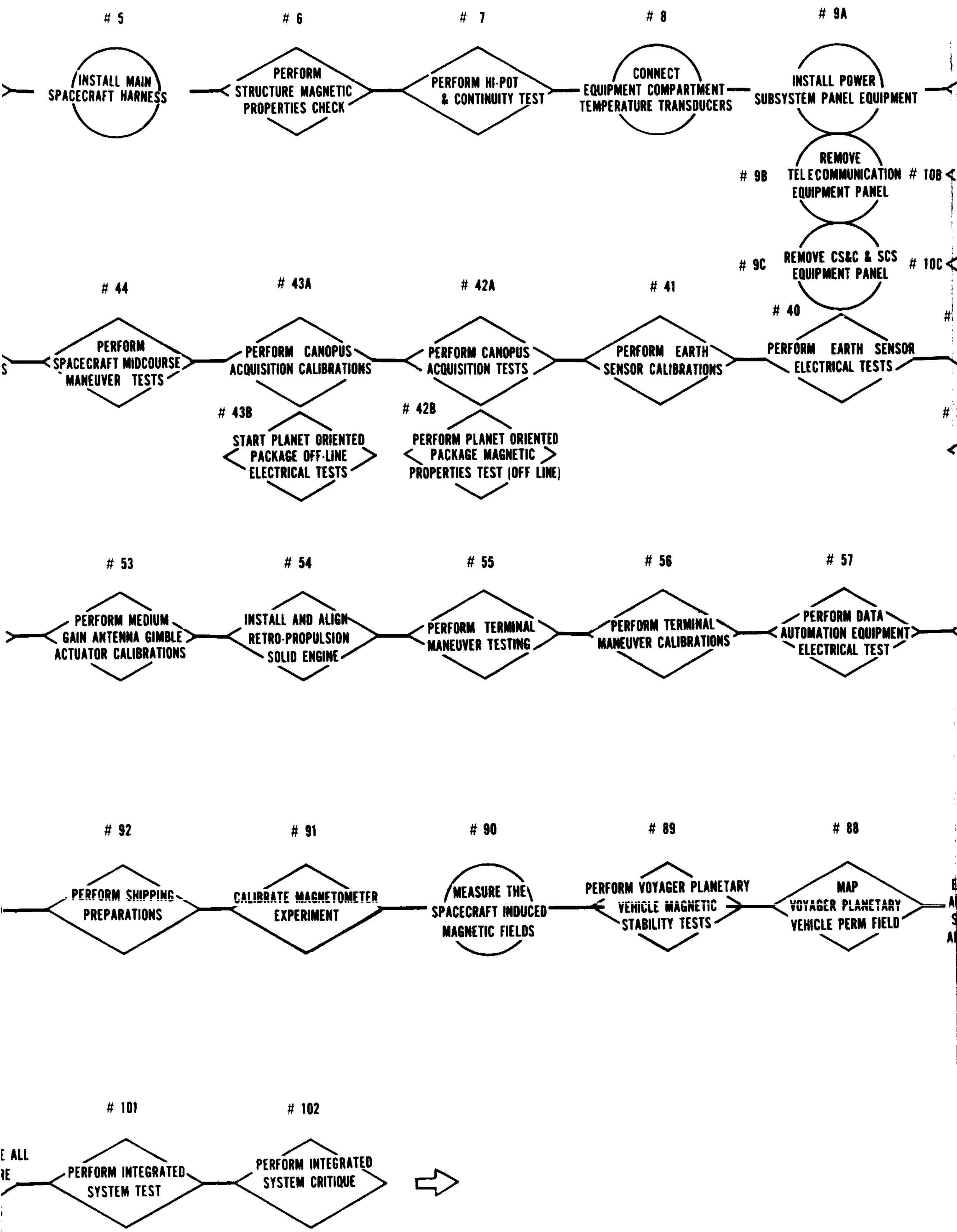
The proof test model spacecraft is a complete spacecraft whose various subsystems have been subjected to flight acceptance testing. Each subsystem will be identical to those of the three flight spacecraft and the life test model. The mechanical and electrical OSE will be identical to that of the three flight units and the life test model; the OSE will have been validated using the spacecraft engineering model. The computer programs used with the proof test model will be identical to those of the flight type spacecrafts and will be validated using the spacecraft engineering model.

The major differences between the proof test model (PTM) and the engineering model are that the science and test capsule PTM subsystems will be installed and electrically tested as part of the PTM assembly integrated into the proof test model spacecraft. Figure 5-42 is a flow diagram of the assembly and checkout sequence for the PTM. Further detail is given in Appendix A.

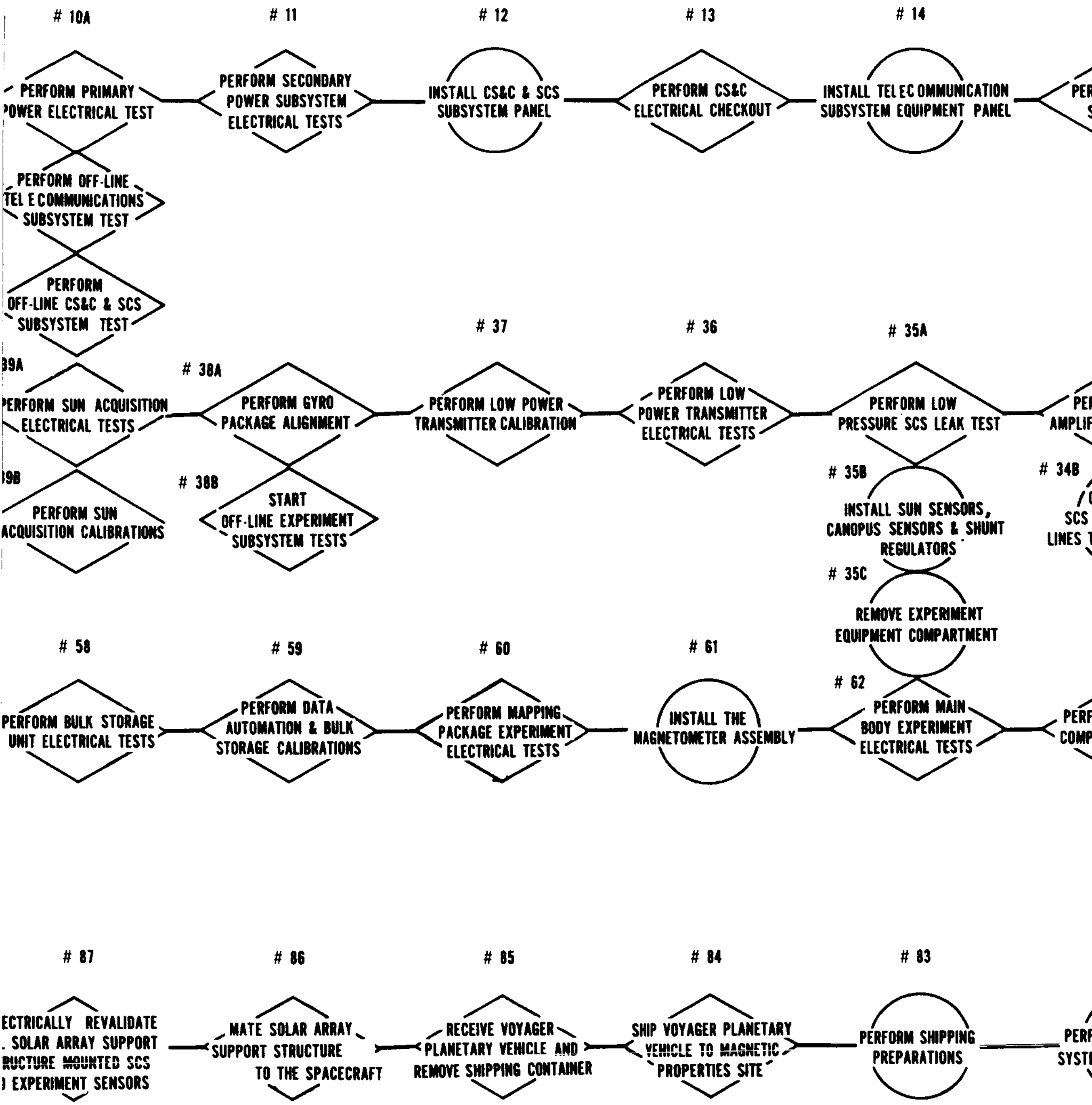
Each subsystem will be assembled and checked out as an off-line function, i.e., the respective equipment panel or panels will be removed from the spacecraft structure and delivered to the subsystem assembly area. Here the various elements of the subsystem will be mechanically



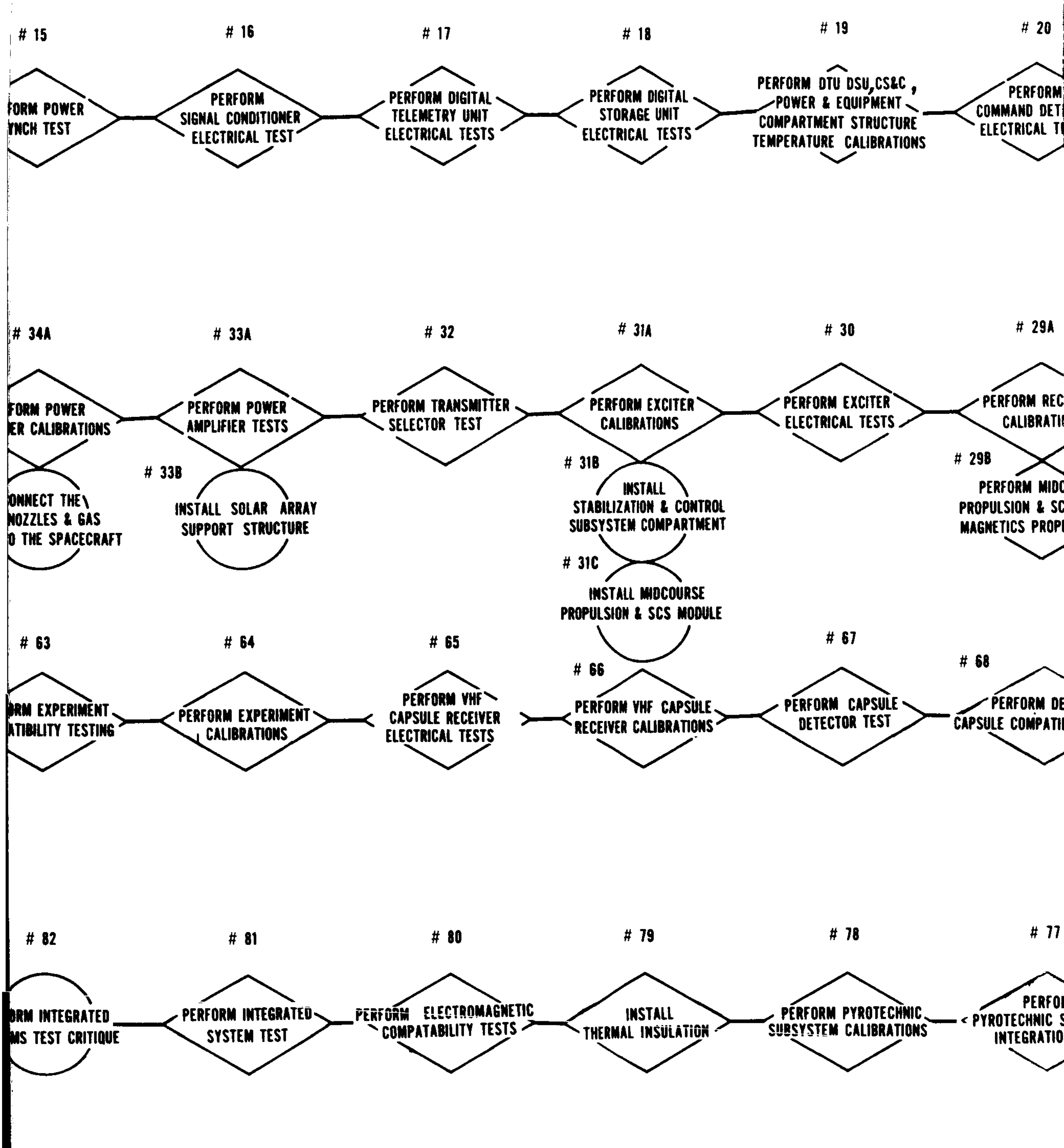
1A
RECEIVE EQUIPMENT COMPARTMENT STRUCTURE
2A
PERFORM EQUIPMENT COMPARTMENT STRUCTURE Q.C. INSPECTION
3
MATE EQUIPMENT COMPARTMENT STRUCTURE TO HANDLING FIXTURE
4
INSTALL TEST J BOXES
1B
REMOVE THE POWER SUBSYSTEM EQUIPMENT COMPARTMENT
2B
PERFORM OFFLINE POWER SUBSYSTEM TEST
1C
RECEIVE SYSTEM TEST SET EOSE
2C
START SYSTEM TEST SET EOSE VALIDATION
48
PERFORM PLANET ORIENTED PACKAGE STABILIZATION & CONTROL TESTING
47
INSTALL EXPERIMENT EQUIPMENT COMPARTMENT
46
INSTALL PLANET ORIENTED PACKAGE
45
PERFORM SPACECRAF
MIDCOURSE CALIBRATIO
49
PERFORM PLANET ORIENTED PACKAGE STABILIZATION & CONTROL CALIBRATIONS
50
PERFORM HI-GAIN ANTENNA GIMBLE ACTUATOR TESTS
51
PERFORM HIGH GAIN ANTENNA GIMBLE ACTUATOR CALIBRATIONS
52
PERFORM MEDIUM
GAIN ANTENNA GIMBL
ACTUATOR TESTS
96
PERFORM SPACECRAFT ALIGNMENTS
95
PERFORM LEAK TESTS
94
PREPARE VOYAGER SPACECRAFT FOR ALIGNMENTS AND LEAK TESTING
93
SHIP VOYAGER
PLANETARY VEHICLE T
REDONDO BEACH
97
PERFORM APPENDAGE DEPLOYMENT TEST
98
MATE THE PLANETARY VEHICLE TO THE CENTAUR ADAPTER
99
PERFORM SPACECRAFT VERTICAL ALIGNMENT
100
ELECTRICALLY REVALIDA
SOLAR ARRAY STRUCTU
MOUNTED SCS AND
EXPERIMENT SENSOR



5
INSTALL MAIN SPACECRAFT HARNESS
6
PERFORM STRUCTURE MAGNETIC PROPERTIES CHECK
7
PERFORM HI-POT & CONTINUITY TEST
8
CONNECT EQUIPMENT COMPARTMENT TEMPERATURE TRANSDUCERS
9A
INSTALL POWER SUBSYSTEM PANEL EQUIPMENT
9B
REMOVE TELECOMMUNICATION EQUIPMENT PANEL
10B
9C
REMOVE CS&C & SCS EQUIPMENT PANEL
10C
44
PERFORM SPACECRAFT MIDCOURSE MANEUVER TESTS
43A
PERFORM CANOPUS ACQUISITION CALIBRATIONS
42A
PERFORM CANOPUS ACQUISITION TESTS
41
PERFORM EARTH SENSOR CALIBRATIONS
40
PERFORM EARTH SENSOR ELECTRICAL TESTS
43B
START PLANET ORIENTED PACKAGE OFF-LINE ELECTRICAL TESTS
42B
PERFORM PLANET ORIENTED PACKAGE MAGNETIC PROPERTIES TEST (OFF LINE)
53
PERFORM MEDIUM GAIN ANTENNA GIMBLE ACTUATOR CALIBRATIONS
54
INSTALL AND ALIGN RETRO-PROPULSION SOLID ENGINE
55
PERFORM TERMINAL MANEUVER TESTING
56
PERFORM TERMINAL MANEUVER CALIBRATIONS
57
PERFORM DATA AUTOMATION EQUIPMENT ELECTRICAL TEST
92
PERFORM SHIPPING PREPARATIONS
91
CALIBRATE MAGNETOMETER EXPERIMENT
90
MEASURE THE SPACECRAFT INDUCED MAGNETIC FIELDS
89
PERFORM VOYAGER PLANETARY VEHICLE MAGNETIC STABILITY TESTS
88
MAP VOYAGER PLANETARY VEHICLE PERM FIELD
101
PERFORM INTEGRATED SYSTEM TEST
102
PERFORM INTEGRATED SYSTEM CRITIQUE



10A
PERFORM PRIMARY POWER ELECTRICAL TEST
11
PERFORM SECONDARY POWER SUBSYSTEM ELECTRICAL TESTS
12
INSTALL CS&C & SCS SUBSYSTEM PANEL
13
PERFORM CS&C ELECTRICAL CHECKOUT
14
INSTALL TELECOMMUNICATION SUBSYSTEM EQUIPMENT PANEL
PERFORM OFF-LINE TELECOMMUNICATIONS SUBSYSTEM TEST
PERFORM OFF-LINE CS&C & SCS SUBSYSTEM TEST
PERFORM SUN ACQUISITION ELECTRICAL TESTS
PERFORM SUN ACQUISITION CALIBRATIONS
38A
PERFORM GYRO PACKAGE ALIGNMENT
38B
START OFF-LINE EXPERIMENT SUBSYSTEM TESTS
37
PERFORM LOW POWER TRANSMITTER CALIBRATION
36
PERFORM LOW POWER TRANSMITTER ELECTRICAL TESTS
35A
PERFORM LOW PRESSURE SCS LEAK TEST
35B
INSTALL SUN SENSORS, CANOPUS SENSORS & SHUNT REGULATORS
35C
REMOVE EXPERIMENT EQUIPMENT COMPARTMENT
34B
58
PERFORM BULK STORAGE UNIT ELECTRICAL TESTS
59
PERFORM DATA AUTOMATION & BULK STORAGE CALIBRATIONS
60
PERFORM MAPPING PACKAGE EXPERIMENT ELECTRICAL TESTS
61
INSTALL THE MAGNETOMETER ASSEMBLY
62
PERFORM MAIN BODY EXPERIMENT ELECTRICAL TESTS
87
86
MATE SOLAR ARRAY SUPPORT STRUCTURE TO THE SPACECRAFT
85
RECEIVE VOYAGER PLANETARY VEHICLE AND REMOVE SHIPPING CONTAINER
84
SHIP VOYAGER PLANETARY VEHICLE TO MAGNETIC PROPERTIES SITE
83
PERFORM SHIPPING PREPARATIONS



15
FORM POWER
YNCH TEST
16
PERFORM SIGNAL CONDITIONER ELECTRICAL TEST
17
PERFORM DIGITAL TELEMETRY UNIT ELECTRICAL TESTS
18
PERFORM DIGITAL STORAGE UNIT ELECTRICAL TESTS
19
PERFORM DTU DSU, CS&C, POWER & EQUIPMENT COMPARTMENT STRUCTURE TEMPERATURE CALIBRATIONS
20
PERFORM
COMMAND DET
ELECTRICAL T
34A
FORM POWER
ER CALIBRATIONS
ONNECT THE
NOZZLES & GAS
O THE SPACECRAFT
33A
PERFORM POWER AMPLIFIER TESTS
33B
INSTALL SOLAR ARRAY SUPPORT STRUCTURE
32
PERFORM TRANSMITTER SELECTOR TEST
31A
PERFORM EXCITER CALIBRATIONS
31B
INSTALL STABILIZATION & CONTROL SUBSYSTEM COMPARTMENT
31C
INSTALL MIDCOURSE PROPULSION & SCS MODULE
30
PERFORM EXCITER ELECTRICAL TESTS
29A
PERFORM REC
CALIBRATI
29B
PERFORM MIDC
PROPULSION & SC
MAGNETICS PROP
63
RM EXPERIMENT
ATIBILITY TESTING
64
PERFORM EXPERIMENT CALIBRATIONS
65
PERFORM VHF CAPSULE RECEIVER ELECTRICAL TESTS
66
PERFORM VHF CAPSULE RECEIVER CALIBRATIONS
67
PERFORM CAPSULE DETECTOR TEST
68
PERFORM DE
CAPSULE COMPATI
82
RM INTEGRATED
MS TEST CRITIQUE
81
PERFORM INTEGRATED SYSTEM TEST
80
PERFORM ELECTROMAGNETIC COMPATABILITY TESTS
79
INSTALL THERMAL INSULATION
78
PERFORM PYROTECHNIC SUBSYSTEM CALIBRATIONS
77
PERFO
PYROTECHNIC S
INTEGRATIO

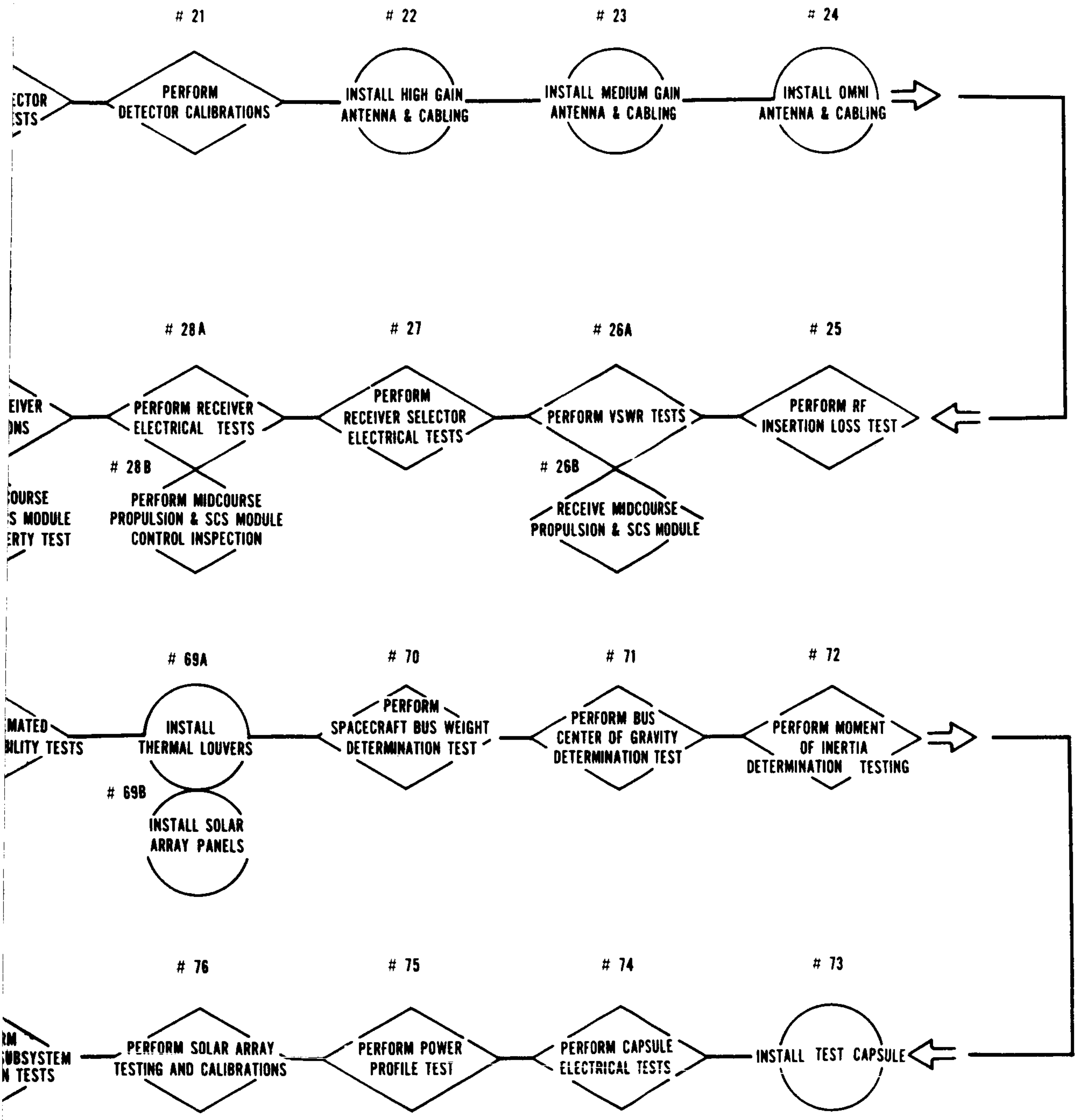


Figure 5-42. 1971 Proof Test Model Spacecraft Assembly and Checkout

installed. At this point, the various break-in and fuse boxes will be installed and the subsystem OSE and antenna interface equipment connected. When the subsystem testing is completed, it will be installed in the spacecraft in the same sequence as for the engineering model and further checks made.

The next step (beyond that for the EM) in the PTM sequence is the installation of each planet-oriented experiment sensor. The following is a list of POP component systems that will be installed off-line:

- Planet-oriented package boom
- Planet-oriented package gimbal actuators
- Mars horizon scanners
- Television experiment sensors
- Ultraviolet spectrometer sensors
- Scan radiometer experiment
- Meteoroid flash experiment sensors
- Infrared spectrometer sensors
- POP intercabling
- POP thermal insulation

The gimbal actuators, cabling, experiments, and sensors will be electrically tested off-line using the experiment equipment compartment and associated EOSE. The POP experiment off-line testing will use the experiment panel EOSE and the experiment spacecraft simulators.

The experiment subsystem panel will then be mechanically installed into the spacecraft. The planet-oriented package articulation and the high- and medium-gain antenna articulation will be tested in the same manner as with the engineering model. The PTM terminal maneuver testing and calibration will also be done in the same manner as with the engineering model.

The bulk storage and data automation electrical testing and calibration will be performed in the same manner as the spacecraft engineering model electrical checkout.

Each experiment package and sensor will be integrated into the spacecraft and the turn-on transient of each experiment measured. The experiment interface with the spacecraft data automation system will be tested by monitoring the rise time, fall time, pulsewidth and pulse amplitude of all data and timing signals under specified operating conditions. Each POP experiment will be stimulated using the experiment EOSE and the EOSE and telemetry response checked. The main body experiments, including magnetometer checkout, is to be performed as follows: each experiment electronics package and sensor will be electrically integrated into the spacecraft. As a part of the experiment integration, noise, and transient conditions will be monitored on the secondary power and signal line for each experiment. Each experiment will be stimulated using the experiment EOSE to test that each experiment is operating properly.

A major testing portion of the science subsystem tests is the experiment compatibility testing. The purpose of the experiment compatibility tests is to:

a) Uncover any interference between experiments

b) Demonstrate that each subsystem does not interfere with any experiment data

c) Demonstrate that each experiment does not degrade the spacecraft operation, in particular that the radio propagation experiments do not degrade the RF subsystem.

Finally, each experiment will be calibrated using both external and built-in radioactive sources. Wherever possible built-in radioactive sources or voltages will be used for experiment calibration. The magnetometer calibration will take place at the magnetic properties facility.

The capsule receiver and demodulator electrical tests and calibrations will be performed in the same manner as the spacecraft engineering model tests. The separated capsule tests are primarily RF tests and to ascertain that the capsule RF subsystem and the spacecraft and experiment subsystems do not cause interference.

The thermal louvers and the solar array panels are installed before the weight determination test. The spacecraft will be weighed at three different points on the spacecraft structure, the total of the three weights determining the spacecraft weight. The three weights will also permit calculating the spacecraft center of gravity in two axes. The c.g. of the third spacecraft axis is determined by tilting the spacecraft to an accurately known vertical angle, and the weighing repeated.

The moment of inertia tests are performed on the proof test model spacecraft only. They are performed by swinging the spacecraft as a pendulum in an appropriate fixture. The moment of inertia about each spacecraft axes will be determined.

Next, the PTM capsule subsystem will be installed in the spacecraft. As soon as the capsule has been electrically integrated the capsule RF subsystem will be tested. The capsule RF tests, like the separated capsule test, determines that the capsule subsystem does not interfere with the spacecraft or experiment subsystem operations and that, in turn, the spacecraft or experiment subsystem does not interfere with normal capsule operation.

The power profile test next will determine the spacecraft subsystem power demands on the power subsystem during each part of the mission profile. The power profile test will be performed as follows:

a) The flight sequence of events until sun acquisition will be followed and primary drains monitored.

b) The primary power drains until sun acquisition will be compared with the trajectory information to determine that the battery capacity is adequate to support spacecraft operations until sun acquisition.

c) The spacecraft will be commanded to perform all of the cruise functions, while all primary power drains are monitored.

d) The primary power drains will be compared with the trajectory information to ascertain that sufficient battery capacity remains to perform the midcourse maneuvers.

e) The spacecraft will be commanded to perform all of the cruise mode and Mars encounter functions while all primary power drains are monitored.

f) The primary power drains will be compared with the trajectory information to ascertain that sufficient battery capacity exists to perform the deboost and sun reacquisition modes for the Mars orbit operations.

g) The spacecraft will be commanded to perform all of the Mars orbiting functions, while all primary power drains are monitored.

h) The primary power drains will be compared with the trajectory information to ascertain that sufficient battery capacity remains to carry the spacecraft through the sun eclipses encountered during the Mars orbits.

The spacecraft ordnance tests will be performed in the same manner as the spacecraft engineering model.

Next the proof test model solar array testing will take place. Each solar array section will be illuminated using the solar array EOSE, and the short-circuit current and open-circuit voltage measured. An inverse impedance measurement will be performed on each solar array string as part of the solar array testing phase.

The last part of the spacecraft build-up is the installation of all thermal insulation, before electromagnetic compatibility tests since the insulation may also serve as RF insulation. The electromagnetic compatibility test checks that no spacecraft subsystem interferes with another subsystem and that no spacecraft subsystem will interfere with the launch vehicle for every spacecraft electrical configuration. The spacecraft operations will be performed as follows:

a) Command the spacecraft subsystems through every combination of the flight sequences and ascertain that there is no degradation or interference between subsystems.

- Canopus sensor
- Gas jet
- High-gain antenna
- High-gain antenna latch
- Medium-gain antenna
- Medium- gain antenna latch
- Mapping package
- Low-gain antenna
- Magnetometer experiment
- Magnetometer boom latch
- Spacecraft vertical alignments

The final test conducted as part of the FTM assembly and test is appendage deployment. Each spacecraft appendage will be deployed in simulated zero g, observing that each appendage freely deploys with no mechanical resistance or cable chaffing.

6.7 Type Approval Testing

The PTM type approval testing sequence is shown in Figure 5-43 and the PTM test schedule in Figure 2-6.

The proof test model spacecraft weight, center of gravity, and moment of inertia determinations will take place in the same manner as was done during the FTM assembly and test, the only difference being that during the assembly and test phase the capsule was not installed.

Before the PTM spacecraft vibration test, test accelerometers for measuring vibration forces will be installed in the spacecraft. The spacecraft is mated to the vibration fixture and a random vibrational search is made for mechanical force amplifications; next, low frequency sinusoidal vibration forces will be applied to the spacecraft; and last, an omnidirectional input of random vibration will be applied. The three vibration tests are to be done in each spacecraft axis.

The capsule will be removed from the spacecraft so that the forces that would be experienced by the spacecraft during the retropropulsion engine fire phase can be adequately simulated. After the capsule has been

b) Irradiate the over-all spacecraft with RF signals that correspond to the expected frequencies and levels from the Saturn IB and Centaur launch vehicle system.

c) Command the spacecraft subsystems through all the Voyager flight sequences and determine the frequencies and levels of all radiation that are emitted from the spacecraft.

d) Apply audio tones and tone bursts to the spacecraft primary bus system and observe each subsystem reaction.

The integrated system test is to be performed in the same manner as on the spacecraft engineering model.

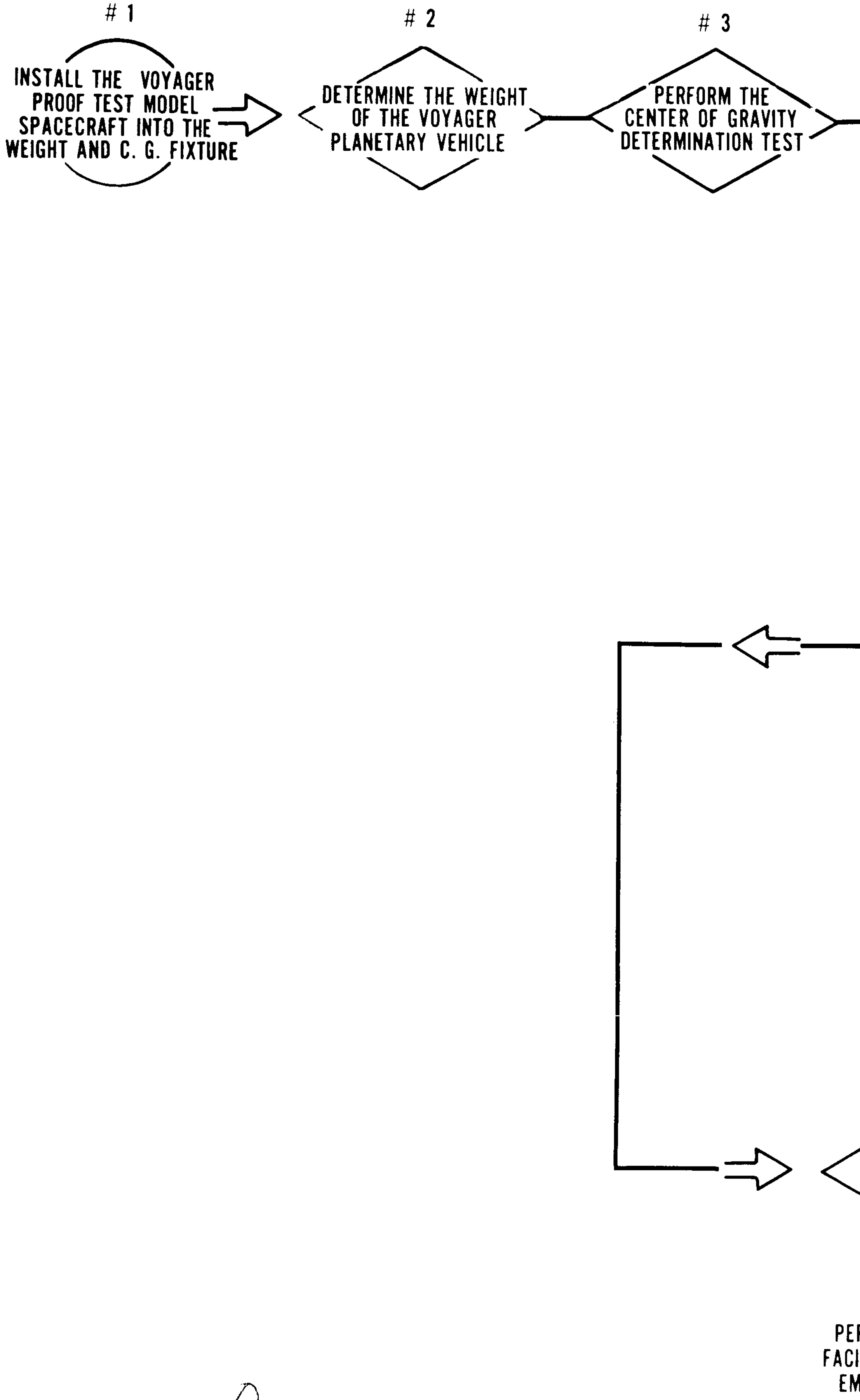
At the magnetics facility the spacecraft magnetic property test will measure the spacecraft perm field and the spacecraft induced magnetic fields to determine the stability of the spacecraft perm field and to calibrate the magnetometer.

The integrated systems test is once again performed as the last spacecraft electrical test before the type approval testing. The integrated systems test will be performed in the same manner as that of the engineering model spacecraft.

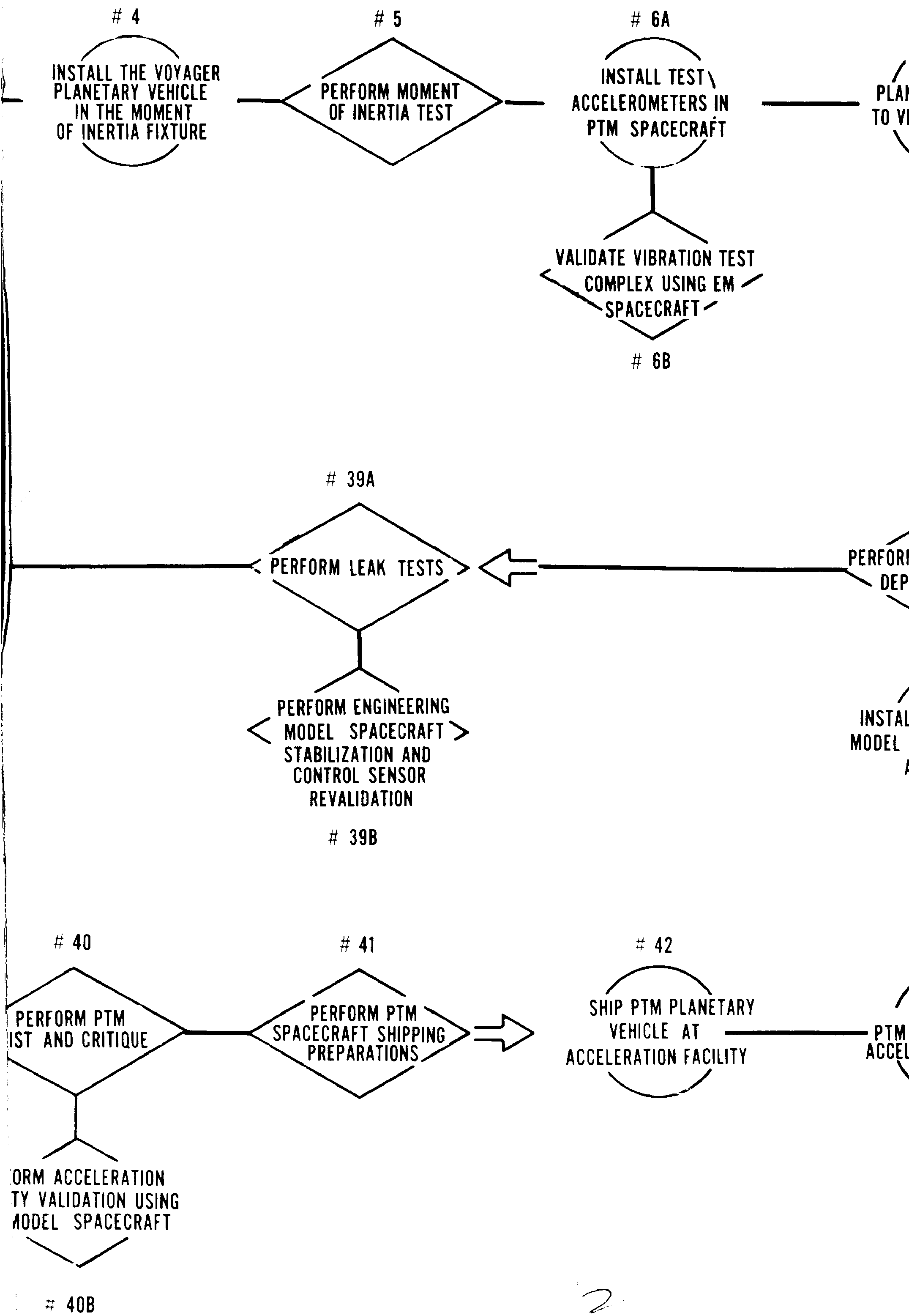
The spacecraft will then be leak tested to insure that no leaks exist in the spacecraft vessels, plumbing, valves, or regulators. The SCS pneumatic subsystem, the midcourse correction engine subsystem, and the solid engine thrust vector control subsystem will be leak tested.

All spacecraft units that require alignment will be optically aligned to flight specifications in preparation for the spacecraft type approval testing. These units are as follows:

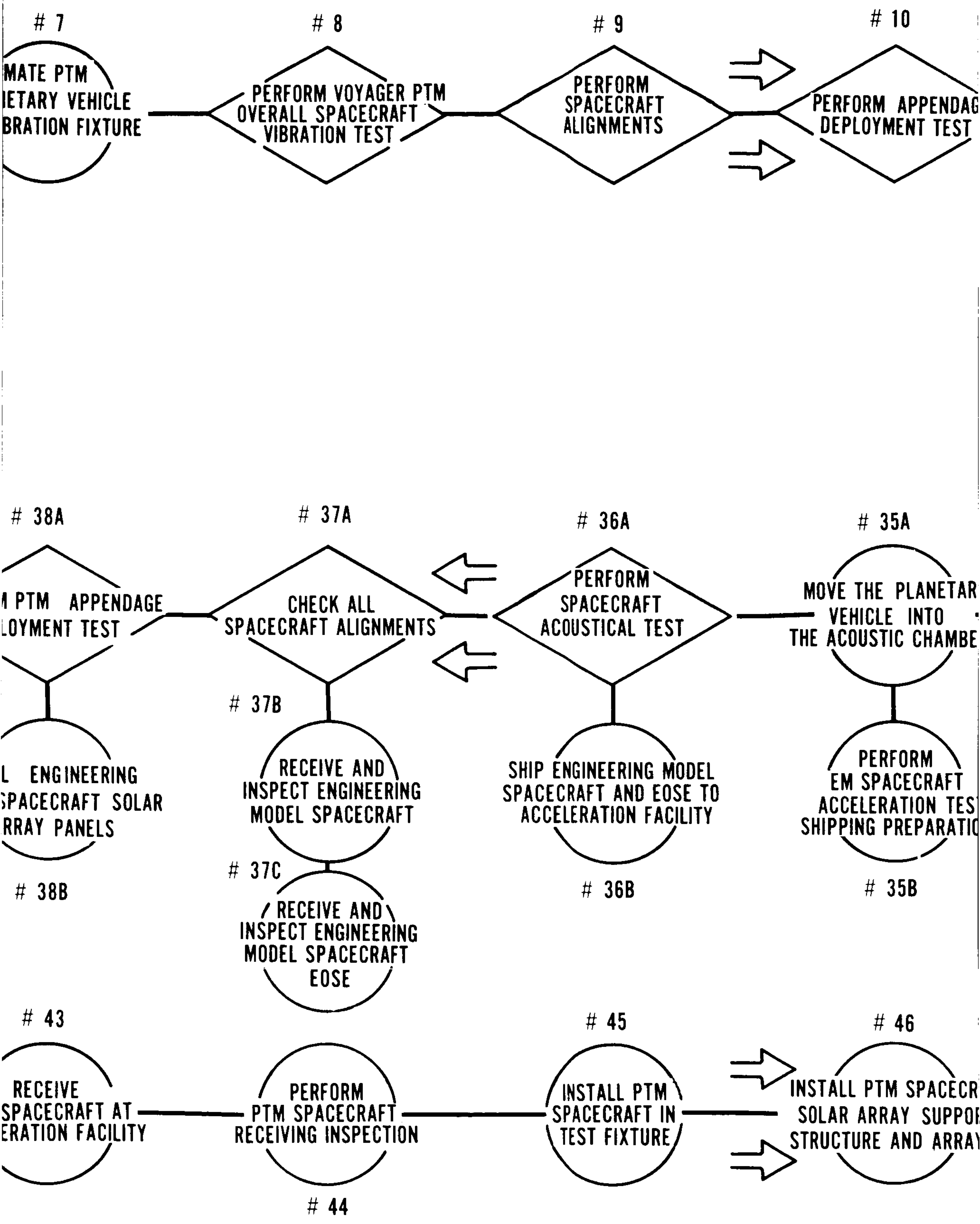
- Solid retropropulsion motor
- Monopropellant motor
- Capsule
- Gyro
- Sun sensor



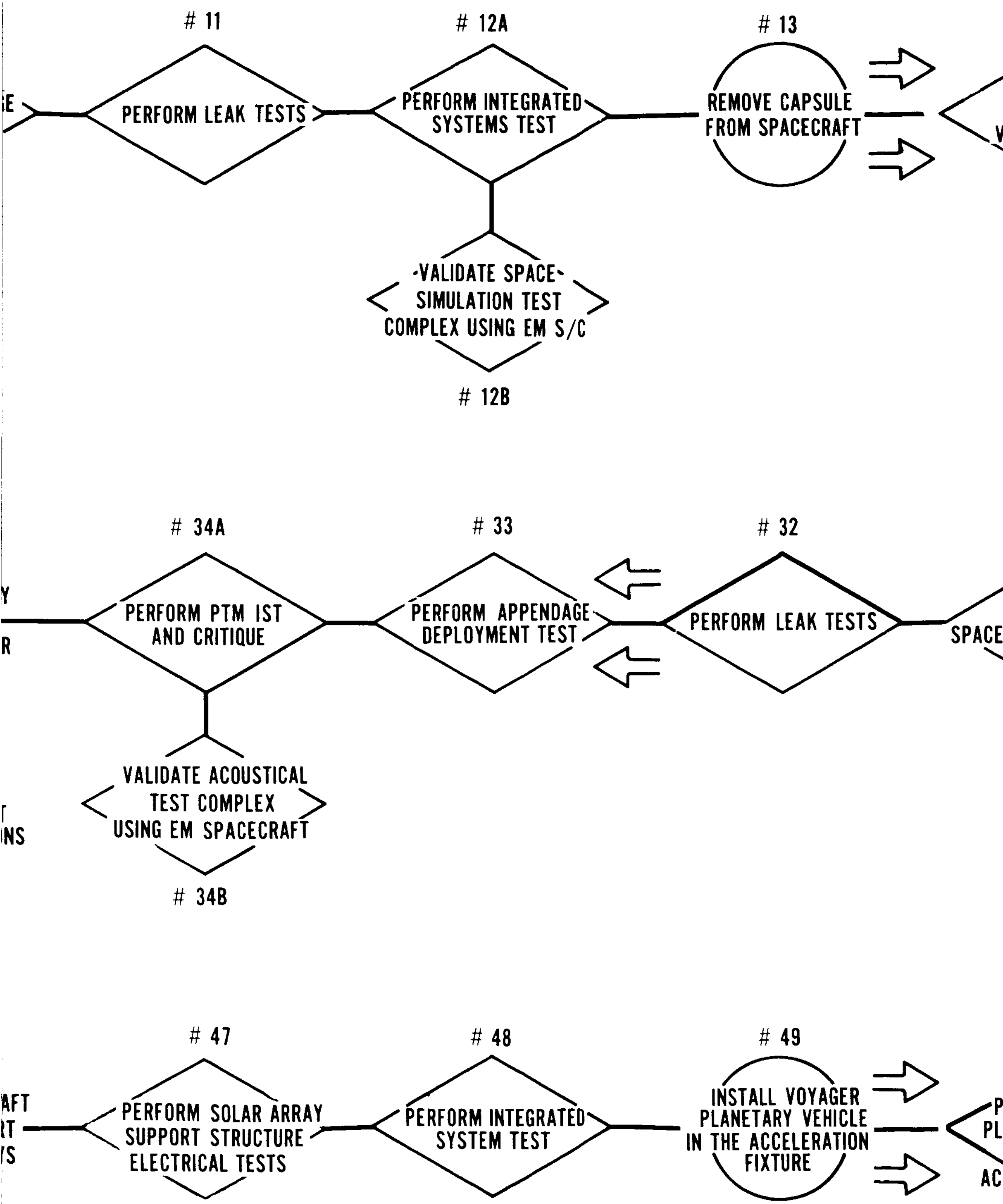
1
INSTALL THE VOYAGER PROOF TEST MODEL SPACECRAFT INTO THE WEIGHT AND C. G. FIXTURE
2
DETERMINE THE WEIGHT OF THE VOYAGER PLANETARY VEHICLE
3
PERFORM THE CENTER OF GRAVITY DETERMINATION TEST



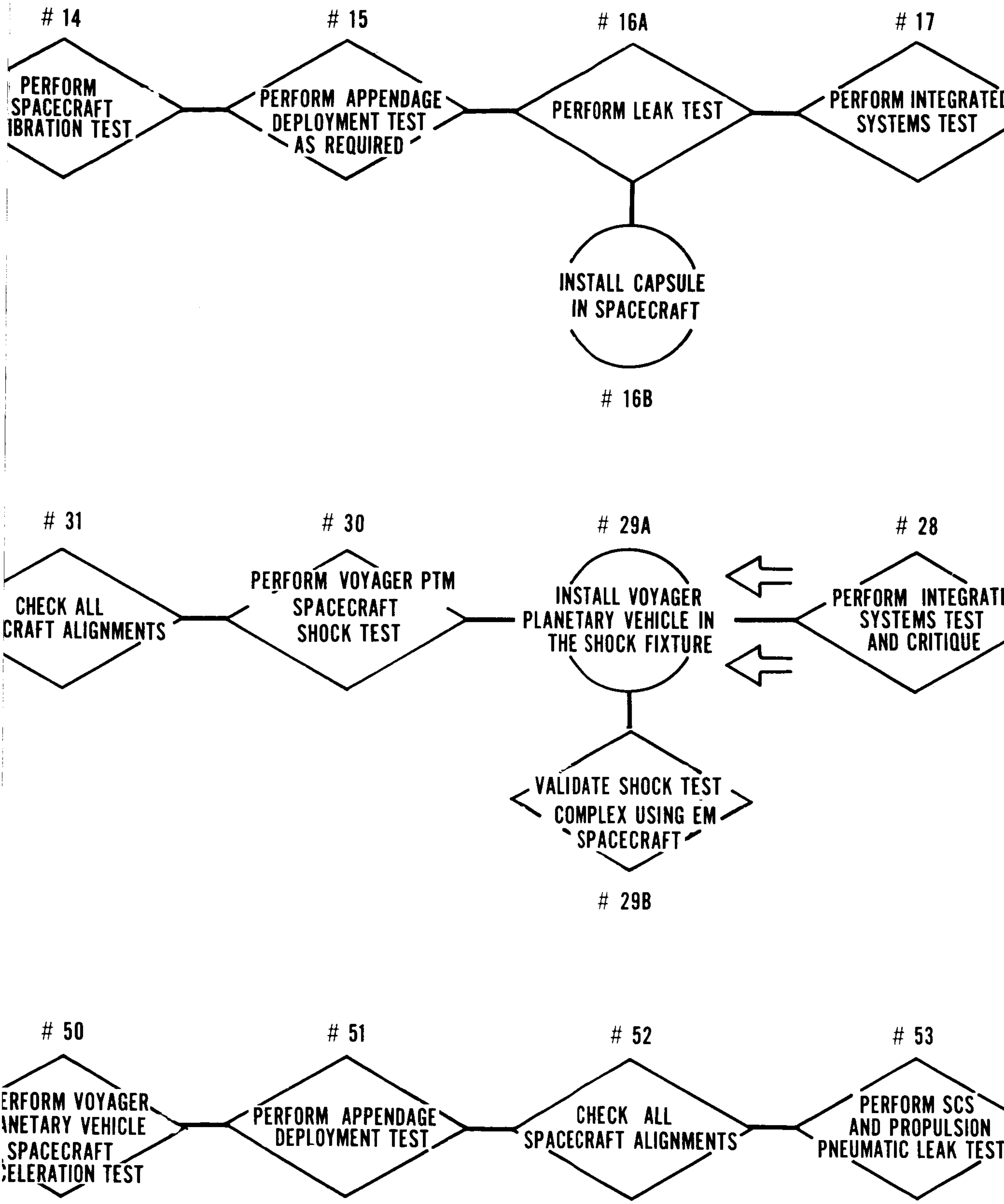
4
INSTALL THE VOYAGER PLANETARY VEHICLE IN THE MOMENT OF INERTIA FIXTURE
5
PERFORM MOMENT OF INERTIA TEST
6A
INSTALL TEST ACCELEROMETERS IN PTM SPACECRAFT
PLAN
TO VI
VALIDATE VIBRATION TEST COMPLEX USING EM SPACECRAFT
6B
39A
PERFORM LEAK TESTS
PERFOR
DEP
PERFORM ENGINEERING MODEL SPACECRAFT STABILIZATION AND CONTROL SENSOR REVALIDATION
39B
INSTAL
MODEL
40
PERFORM PTM
IST AND CRITIQUE
41
PERFORM PTM SPACECRAFT SHIPPING PREPARATIONS
42
SHIP PTM PLANETARY VEHICLE AT ACCELERATION FACILITY
PTM
ACCEL
ORM ACCELERATION
TY VALIDATION USING
MODEL SPACECRAFT
40B



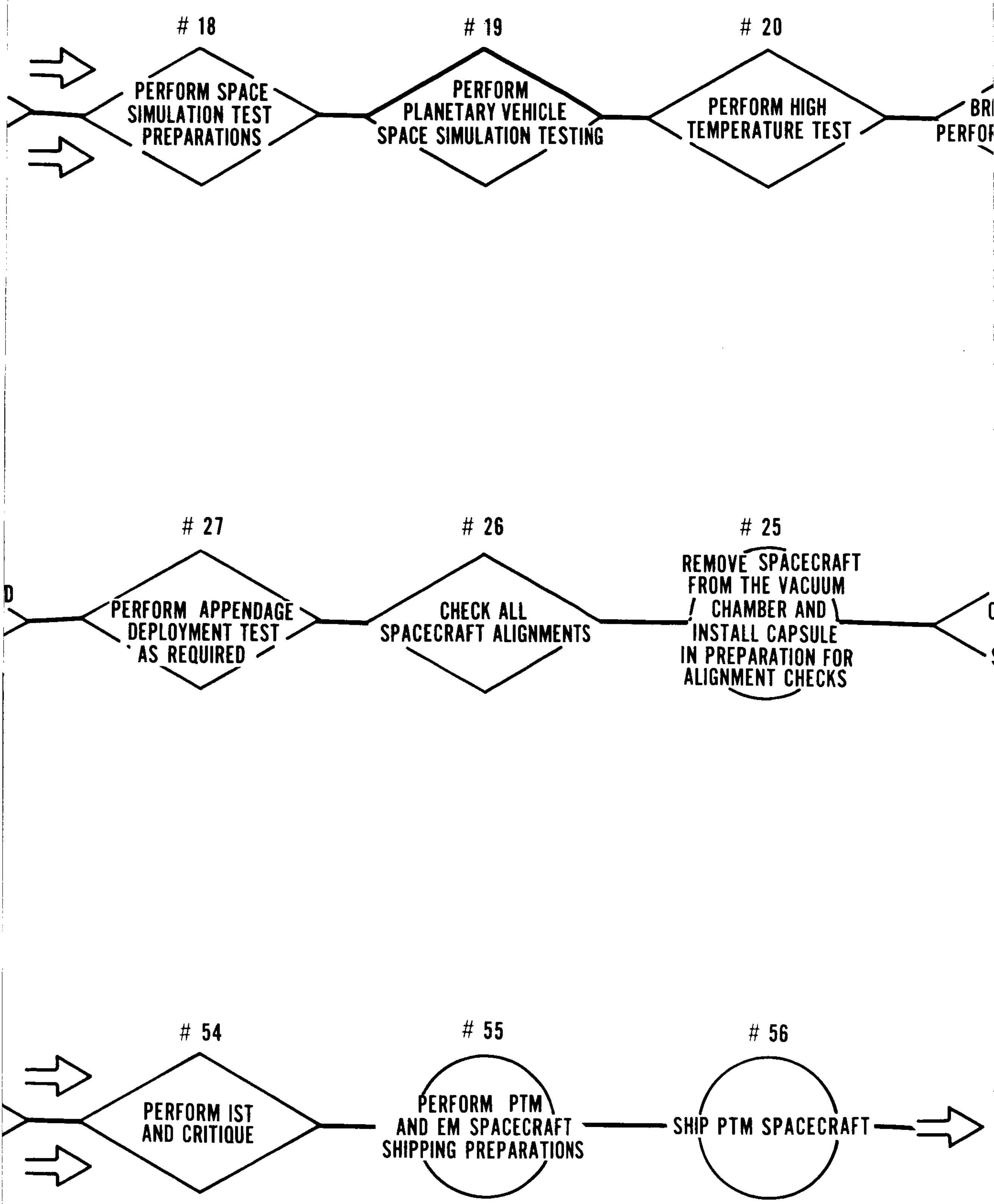
7
MATE PTM
IETARY VEHICLE
BRATION FIXTURE
8
PERFORM VOYAGER PTM
OVERALL SPACECRAFT
VIBRATION TEST
9
PERFORM
SPACECRAFT
ALIGNMENTS
10
PERFORM APPENDAG
DEPLOYMENT TEST
38A
M PTM APPENDAGE
LOYMENT TEST
37A
CHECK ALL
SPACECRAFT ALIGNMENTS
36A
PERFORM
SPACECRAFT
ACOUSTICAL TEST
35A
MOVE THE PLANETAR
VEHICLE INTO
THE ACOUSTIC CHAMBE
L ENGINEERING
SPACECRAFT SOLAR
RRAY PANELS
38B
37B
RECEIVE AND
INSPECT ENGINEERING
MODEL SPACECRAFT
37C
RECEIVE AND
INSPECT ENGINEERING
MODEL SPACECRAFT
EOSE
SHIP ENGINEERING MODEL
SPACECRAFT AND EOSE TO
ACCELERATION FACILITY
36B
PERFORM
EM SPACECRAFT
ACCELERATION TES
SHIPPING PREPARATI
35B
43
RECEIVE
SPACECRAFT AT
ERATION FACILITY
PERFORM
PTM SPACECRAFT
RECEIVING INSPECTION
44
45
INSTALL PTM
SPACECRAFT IN
TEST FIXTURE
46
INSTALL PTM SPACECR
SOLAR ARRAY SUPPO
STRUCTURE AND ARRA



11
PERFORM LEAK TESTS
12A
PERFORM INTEGRATED SYSTEMS TEST
13
REMOVE CAPSULE FROM SPACECRAFT
VALIDATE SPACE SIMULATION TEST COMPLEX USING EM S/C
12B
34A
PERFORM PTM IST AND CRITIQUE
33
PERFORM APPENDAGE DEPLOYMENT TEST
32
PERFORM LEAK TESTS
SPACE
VALIDATE ACOUSTICAL TEST COMPLEX USING EM SPACECRAFT
34B
47
PERFORM SOLAR ARRAY SUPPORT STRUCTURE ELECTRICAL TESTS
48
PERFORM INTEGRATED SYSTEM TEST
49
INSTALL VOYAGER PLANETARY VEHICLE IN THE ACCELERATION FIXTURE


14
PERFORM SPACECRAFT IBRATION TEST
15
PERFORM APPENDAGE DEPLOYMENT TEST AS REQUIRED
16A
PERFORM LEAK TEST
17
PERFORM INTEGRATED SYSTEMS TEST
INSTALL CAPSULE IN SPACECRAFT
16B
31
CHECK ALL CRAFT ALIGNMENTS
30
PERFORM VOYAGER PTM SPACECRAFT SHOCK TEST
29A
INSTALL VOYAGER PLANETARY VEHICLE IN THE SHOCK FIXTURE
28
PERFORM INTEGRATI SYSTEMS TEST AND CRITIQUE
VALIDATE SHOCK TEST COMPLEX USING EM SPACECRAFT
29B
50
ERFORM VOYAGER ANETARY VEHICLE SPACECRAFT CELERATION TEST
51
PERFORM APPENDAGE DEPLOYMENT TEST
52
CHECK ALL SPACECRAFT ALIGNMENTS
53
PERFORM SCS AND PROPULSION PNEUMATIC LEAK TEST

5



18
PERFORM SPACE SIMULATION TEST PREPARATIONS
19
PERFORM PLANETARY VEHICLE SPACE SIMULATION TESTING
20
PERFORM HIGH TEMPERATURE TEST
27
PERFORM APPENDAGE DEPLOYMENT TEST AS REQUIRED
26
CHECK ALL SPACECRAFT ALIGNMENTS
25
REMOVE SPACECRAFT FROM THE VACUUM CHAMBER AND INSTALL CAPSULE IN PREPARATION FOR ALIGNMENT CHECKS
54
PERFORM IST AND CRITIQUE
55
PERFORM PTM AND EM SPACECRAFT SHIPPING PREPARATIONS
56
SHIP PTM SPACECRAFT

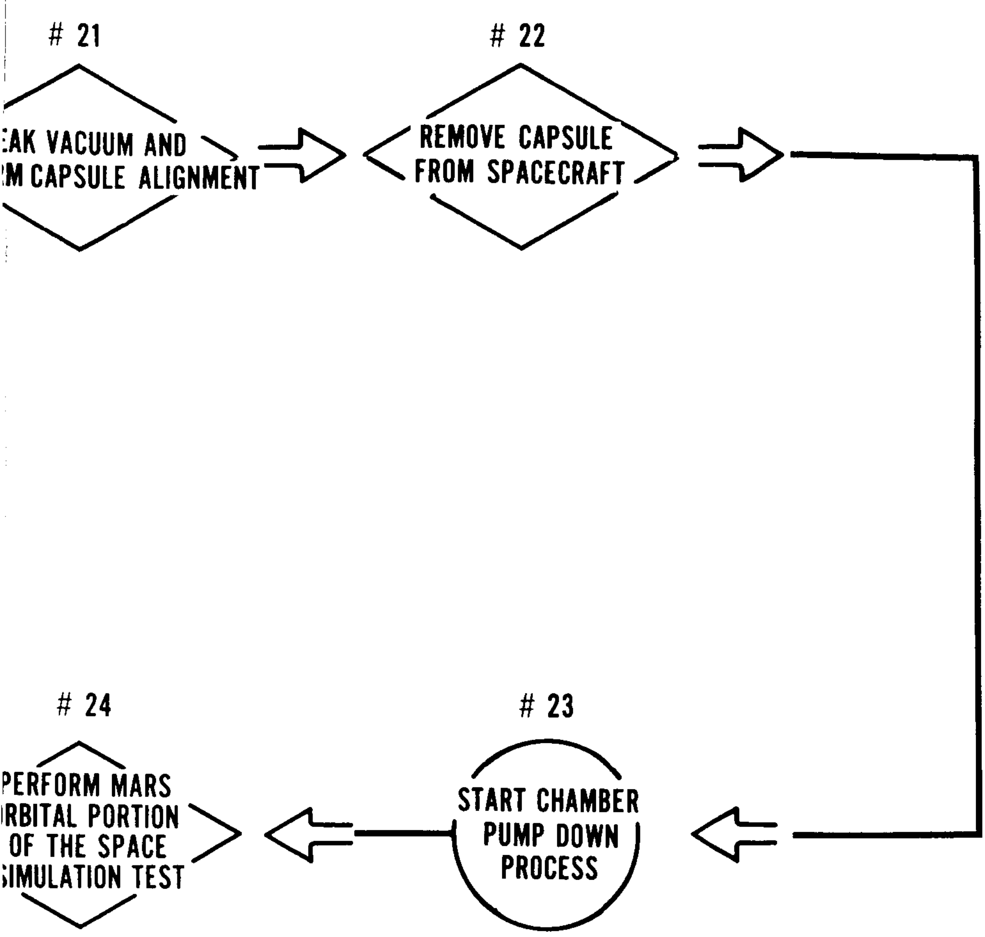


Figure 5-43. 1971 Mission Proof Test Model Spacecraft Type Approval Testing

(6)

removed, the vibration test is repeated. Each spacecraft subsystem will be electrically powered and sequenced corresponding to the portion of the mission profile undergoing vibrational testing. Between the two vibration tests (with and without capsule), all spacecraft alignments will be checked for shifts due to the applied vibrational forces.

The appendage deployment test will be performed in a simulated zero g field using live ordnance to ascertain that each appendage freely deploys. Next a leak test will be performed to ascertain that the SCS pneumatic, midcourse correction engine and the solid retropropulsion engine thrust vector control feed systems have survived the vibration test. An integrated systems test will be performed to ascertain that there has been no electrical degradation due to vibration testing. At the conclusion of the retropropulsion phase of the vibration test the spacecraft alignment, appendage deployment, leak, and integrated system tests will be repeated to ascertain that the spacecraft has mechanically and electrically survived the retropropulsion phase of vibration testing.

The spacecraft engineering model will be used to validate the space simulation test complex as part of the space simulation testing. The space simulation preparation for the PTM will consist of the following tasks:

a) Install heaters in the spacecraft as required

b) Install thermalcouples in the spacecraft

c) Attach capsule to spacecraft

d) Install the spacecraft in the simulation fixture

e) Functional test as a final verification of the space simulation electrical complex and MOSE

When the proper chamber pressure has been reached, the vacuum chamber cold walls will be activated and the spacecraft allowed to temperature soak. When the spacecraft has reached the temperature anticipated during the spacecraft separation portion of the mission sequence, the spacecraft sun acquisition mode simulator will be initiated.

The spacecraft will be electrically powered and sequenced during the space simulation test following the mission profile. The sun simulator will be varied throughout the test to correspond to the intensities encountered during the various phases of the mission profile. The solar array outputs will be monitored to determine that the array output electrically meets the required specifications.

The space simulation chamber temperature will then be allowed to increase to the specified upper temperature limit. Each spacecraft subsystem will then be exercised and monitored for proper operation. After the capsule has been removed from the spacecraft, the spacecraft will undergo a high- and low-temperature test and simulated Mars orbit testing including eclipse simulation. After the completion of space simulation tests, appendage alignment, appendage deployment, and leak tests, an integrated system test will be performed in the same manner as after vibration testing.

The PTM capsule will be reinstalled in the PTM spacecraft, and the shock test initiated. The shock tests simulate shocks encountered by the spacecraft during the liftoff shroud jettison or retropropulsion engine firing. The spacecraft will be electrically powered and actuated corresponding to the applicable portions of the mission profile. After the shock test has been completed, all spacecraft alignment will be checked for shifts. All spacecraft appendages will then be deployed in a simulated zero g field, using live ordnance. After leak tests, a spacecraft integrated systems test will be performed to verify that the spacecraft suffered no adverse electrical or mechanical effects as a result of the shock test.

Next the acoustical test will simulate forces encountered by the spacecraft and capsule during the liftoff phases. The spacecraft will be electrically powered and actuated corresponding to the applicable portions of the mission profile. After the acoustical test, all spacecraft alignments will be checked for shifts and all spacecraft appendages deployed in a simulated zero g field, using live ordnance, followed by leak tests and integrated systems test.

The acceleration test will simulate forces encountered by the spacecraft and capsule during the liftoff and retropropulsion maneuver phases. The spacecraft will be electrically powered and actuated corresponding to the applicable portions of the mission profile. Following acceleration, all spacecraft alignments will be checked for shifts, appendages deployed in a simulated zero g field, using live ordnance, leak tests carried out, and an integrated systems test completed.

After the completion of the final integrated system test, the spacecraft and associated OSE will be placed in shipping containers and shipped to the AFETR to support the launch complex facility validations.

6.8 Flight and Life Test Spacecraft Assembly and Checkout

The flight spacecraft assembly and checkout will be performed precisely as for the proof test model spacecraft with the exception of moment of inertia determination.

6.9 Flight and Life Test Spacecraft Acceptance Testing

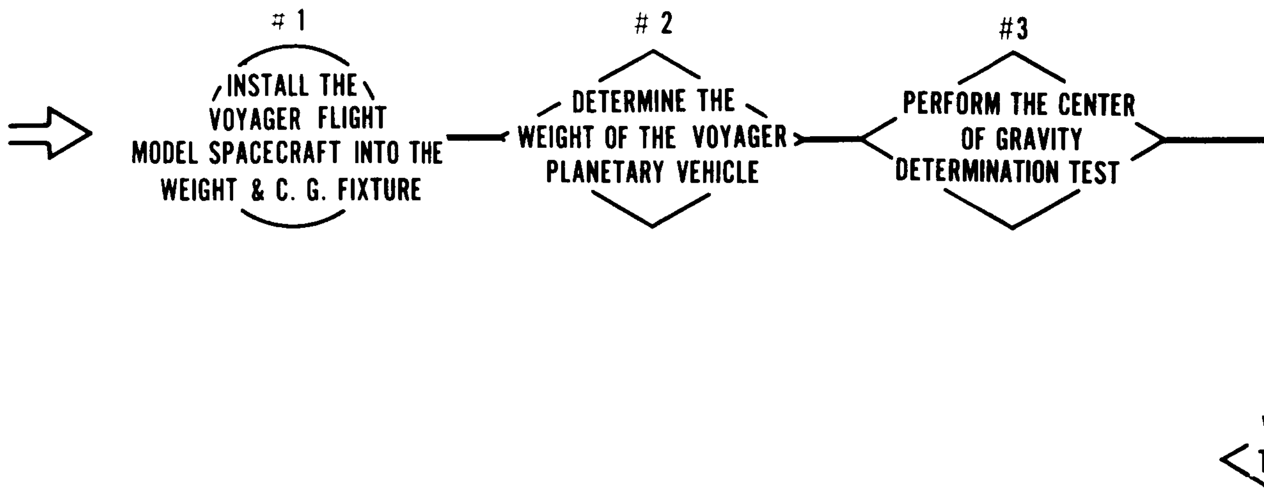
As shown in Figure 5-44, the flight spacecraft will undergo vibration and space simulation testing only. The vibration and space simulation testing will be performed in the same manner as for the proof test model but with levels commensurate with flight environment. Shock, acoustical, and acceleration tests will not be performed on the flight spacecrafts.

It is not planned that humidity testing be performed at the spacecraft level. A description of spacecraft life testing is discussed in Section IV 3. 7. 2.

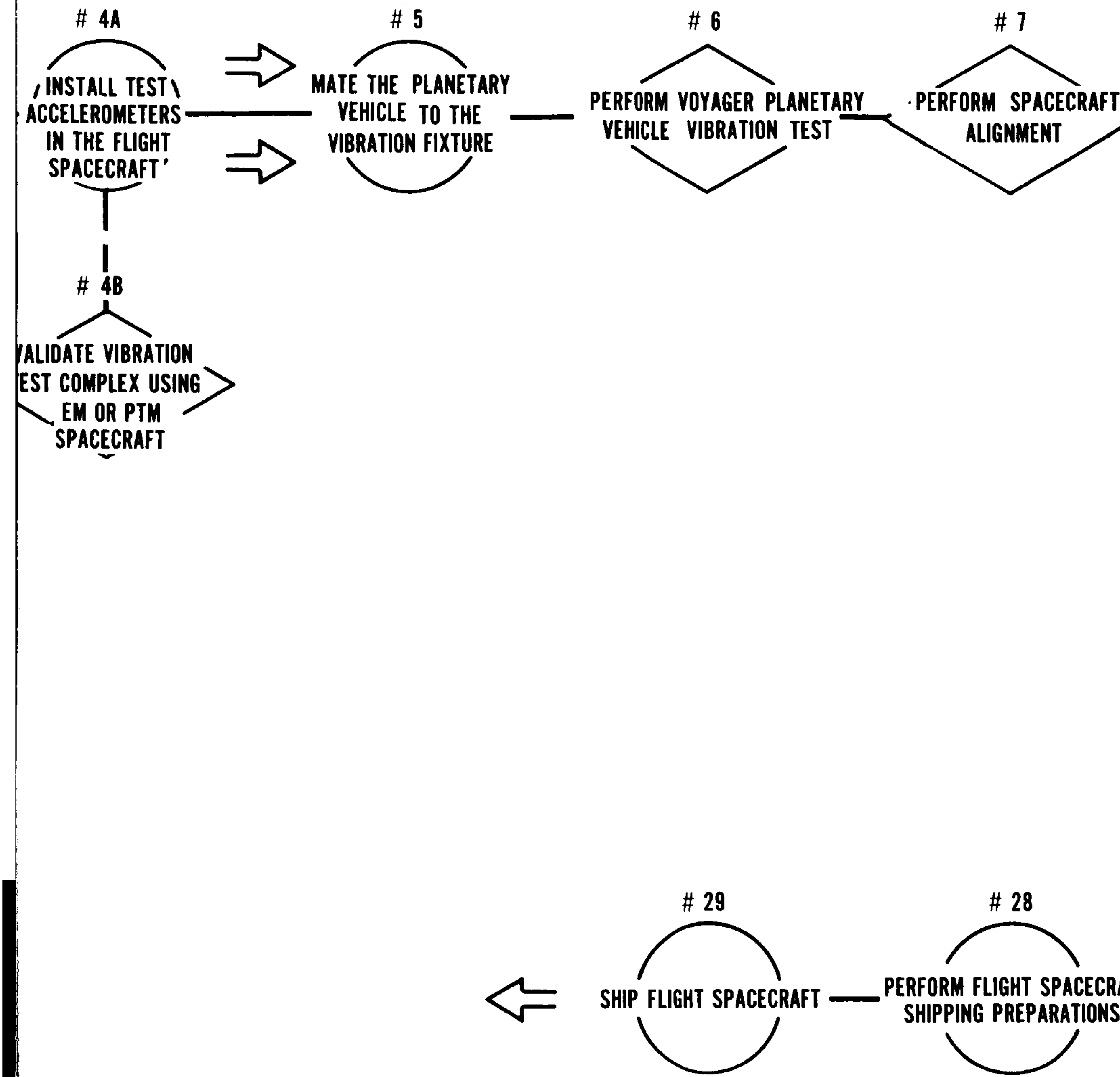
6.10 Spacecraft Launch Operations

The launch site operations sequence is shown in Figure 5-45.

The proof test model spacecraft and OSE will be received and inspected for shipping and handling damage. The spacecraft solar array support structure will be mated to the spacecraft and arrays installed.

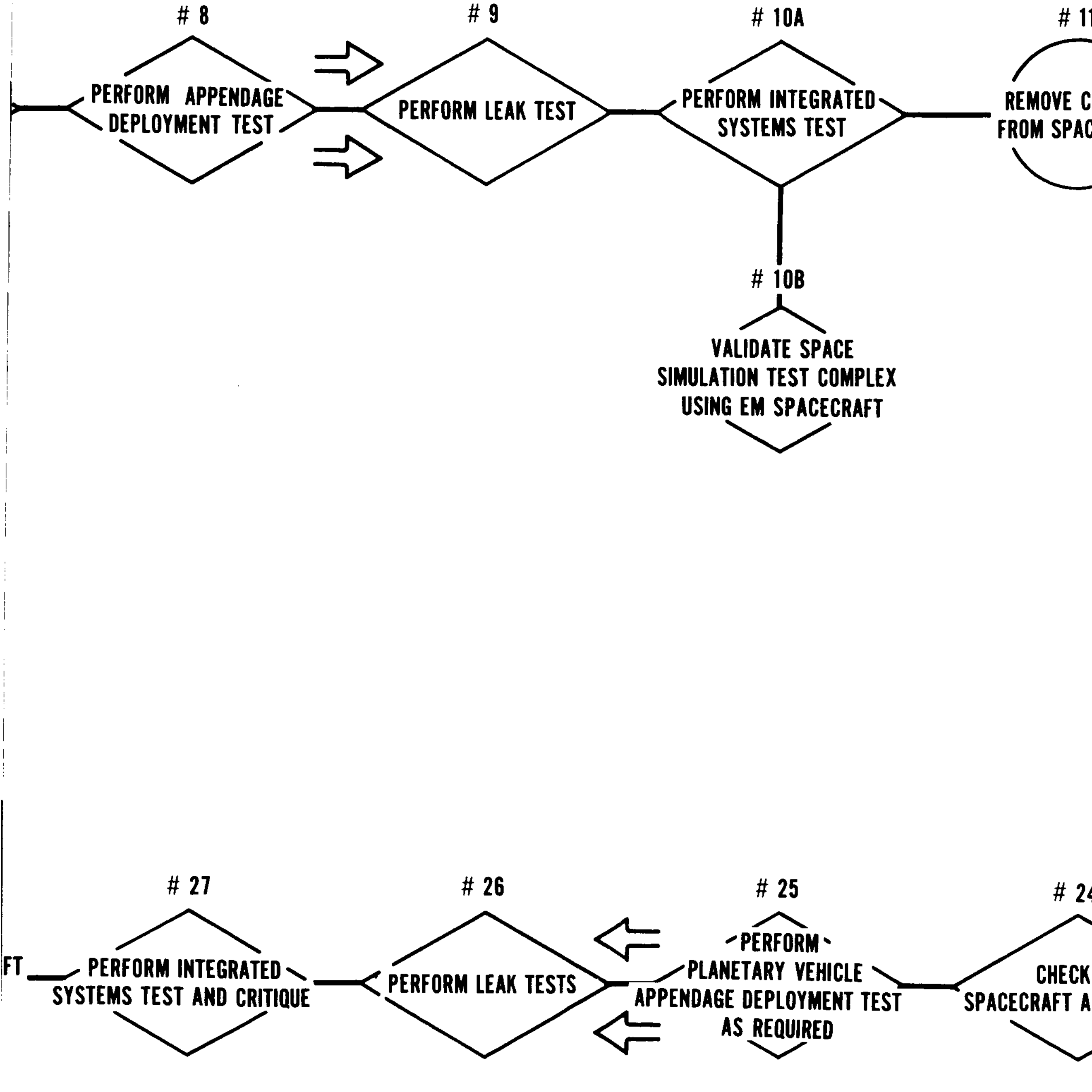


1
INSTALL THE VOYAGER FLIGHT MODEL SPACECRAFT INTO THE WEIGHT & C. G. FIXTURE
2
DETERMINE THE WEIGHT OF THE VOYAGER PLANETARY VEHICLE
#3
PERFORM THE CENTER OF GRAVITY DETERMINATION TEST

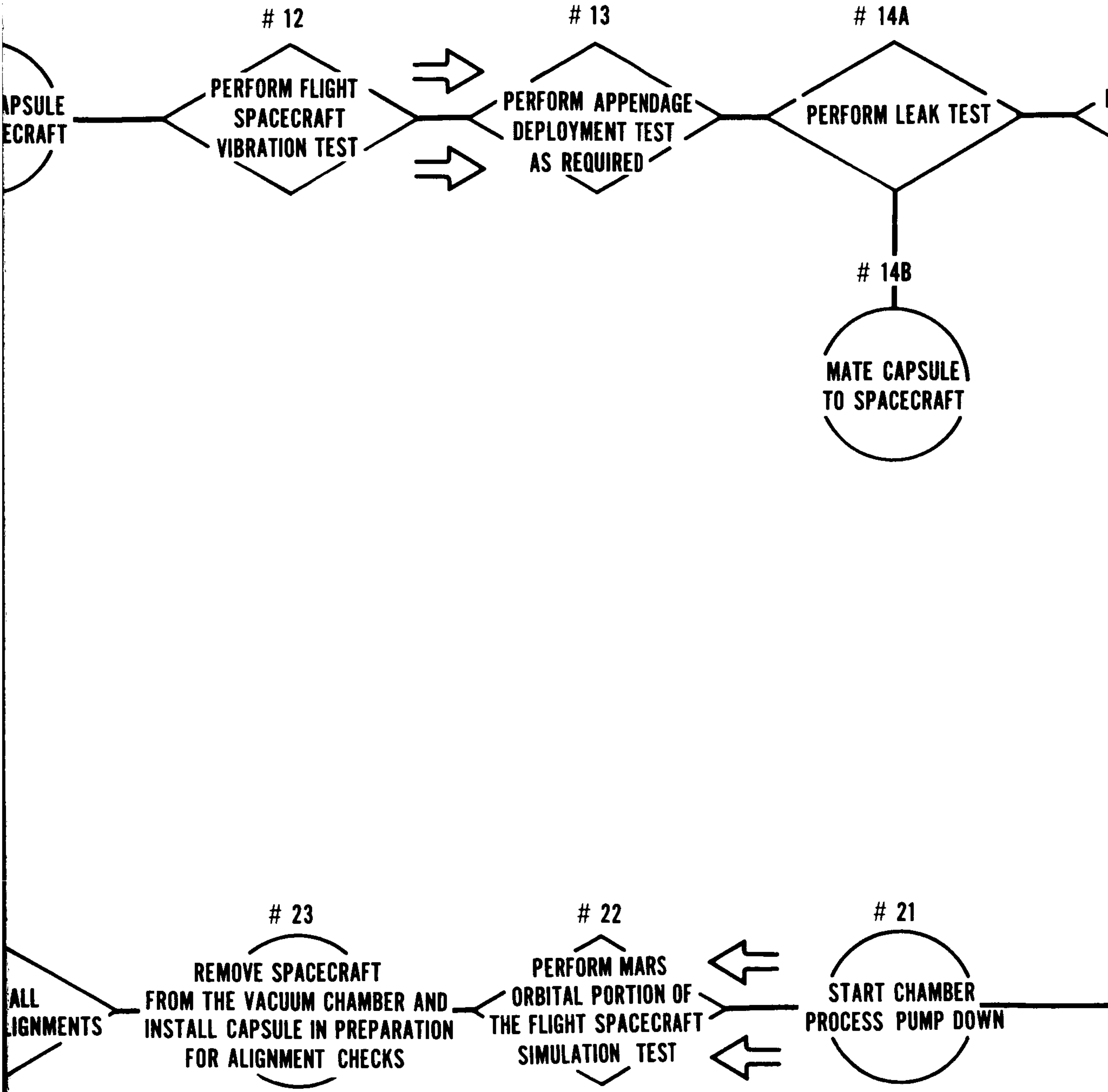


4A
INSTALL TEST ACCELEROMETERS IN THE FLIGHT SPACECRAFT
5
MATE THE PLANETARY VEHICLE TO THE VIBRATION FIXTURE
6
PERFORM VOYAGER PLANETARY VEHICLE VIBRATION TEST
7
PERFORM SPACECRAFT ALIGNMENT
4B
ALIDATE VIBRATION EST COMPLEX USING EM OR PTM SPACECRAFT
29
SHIP FLIGHT SPACECRAFT
28
PERFORM FLIGHT SPACECR SHIPPING PREPARATIONS

2



8
PERFORM APPENDAGE DEPLOYMENT TEST
9
PERFORM LEAK TEST
10A
PERFORM INTEGRATED SYSTEMS TEST
10B
VALIDATE SPACE SIMULATION TEST COMPLEX USING EM SPACECRAFT
11
REMOVE C
FROM SPAC
27
FT
PERFORM INTEGRATED SYSTEMS TEST AND CRITIQUE
26
PERFORM LEAK TESTS
25
PERFORM PLANETARY VEHICLE APPENDAGE DEPLOYMENT TEST AS REQUIRED
24
CHECK
SPACECRAFT A



12
PERFORM FLIGHT SPACECRAFT VIBRATION TEST
13
PERFORM APPENDAGE DEPLOYMENT TEST AS REQUIRED
14A
PERFORM LEAK TEST
14B
MATE CAPSULE TO SPACECRAFT
21
START CHAMBER PROCESS PUMP DOWN
22
PERFORM MARS ORBITAL PORTION OF THE FLIGHT SPACECRAFT SIMULATION TEST
23
REMOVE SPACECRAFT FROM THE VACUUM CHAMBER AND INSTALL CAPSULE IN PREPARATION FOR ALIGNMENT CHECKS

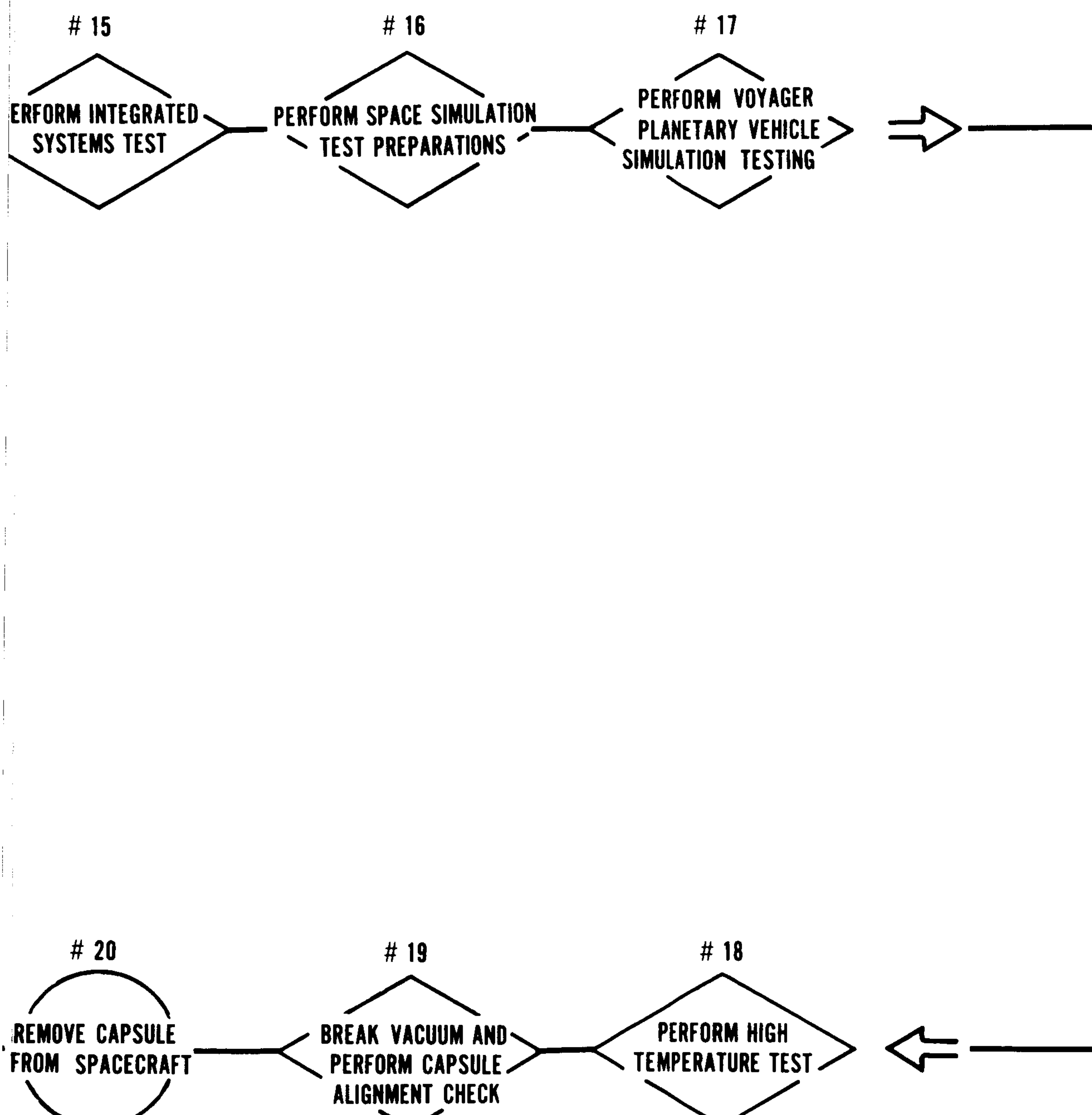
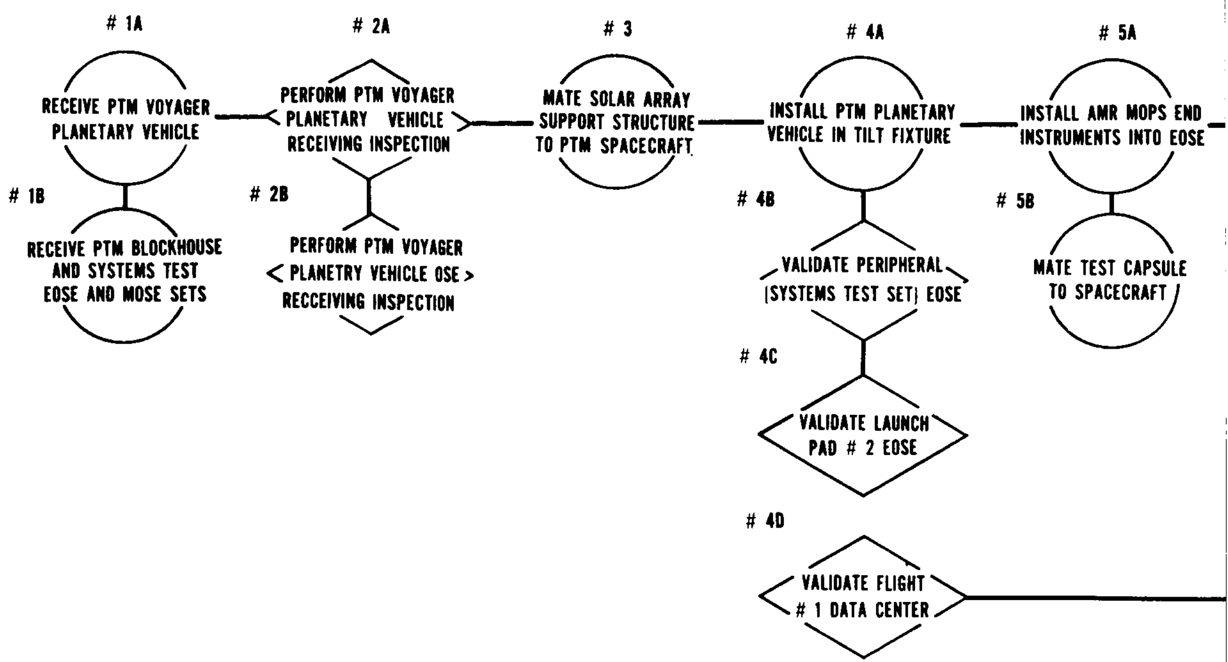


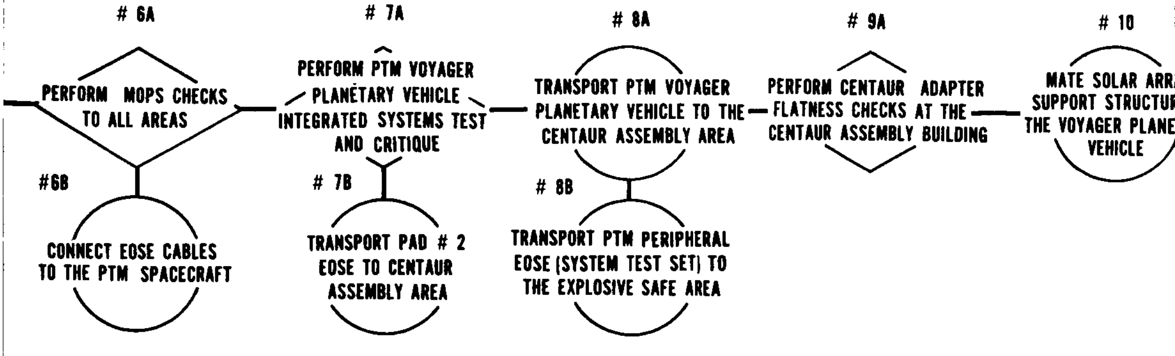
Figure 5-44. 1971 Voyager Flight Model Spacecraft Flight Approval Testing

5

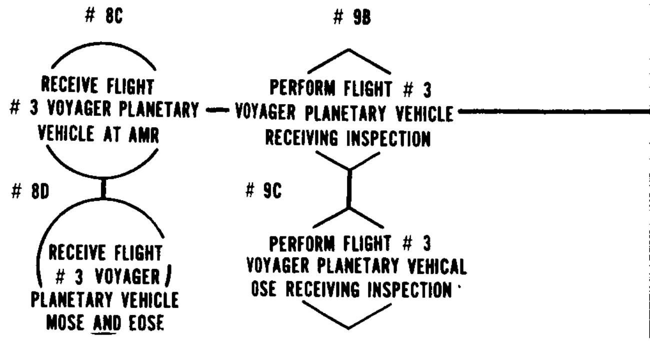


1A
RECEIVE PTM VOYAGER PLANETARY VEHICLE
1B
RECEIVE PTM BLOCKHOUSE AND SYSTEMS TEST EOSE AND MOSE SETS
2A
PERFORM PTM VOYAGER PLANETARY VEHICLE RECEIVING INSPECTION
2B
PERFORM PTM VOYAGER PLANETRY VEHICLE OSE RECCEIVING INSPECTION
3
MATE SOLAR ARRAY SUPPORT STRUCTURE TO PTM SPACECRAFT
4A
INSTALL PTM PLANETARY VEHICLE IN TILT FIXTURE
4B
VALIDATE PERIPHERAL (SYSTEMS TEST SET) EOSE
4C
VALIDATE LAUNCH PAD # 2 EOSE
4D
VALIDATE FLIGHT # 1 DATA CENTER
5A
INSTALL AMR MOPS END INSTRUMENTS INTO EOSE
5B
MATE TEST CAPSULE TO SPACECRAFT

1

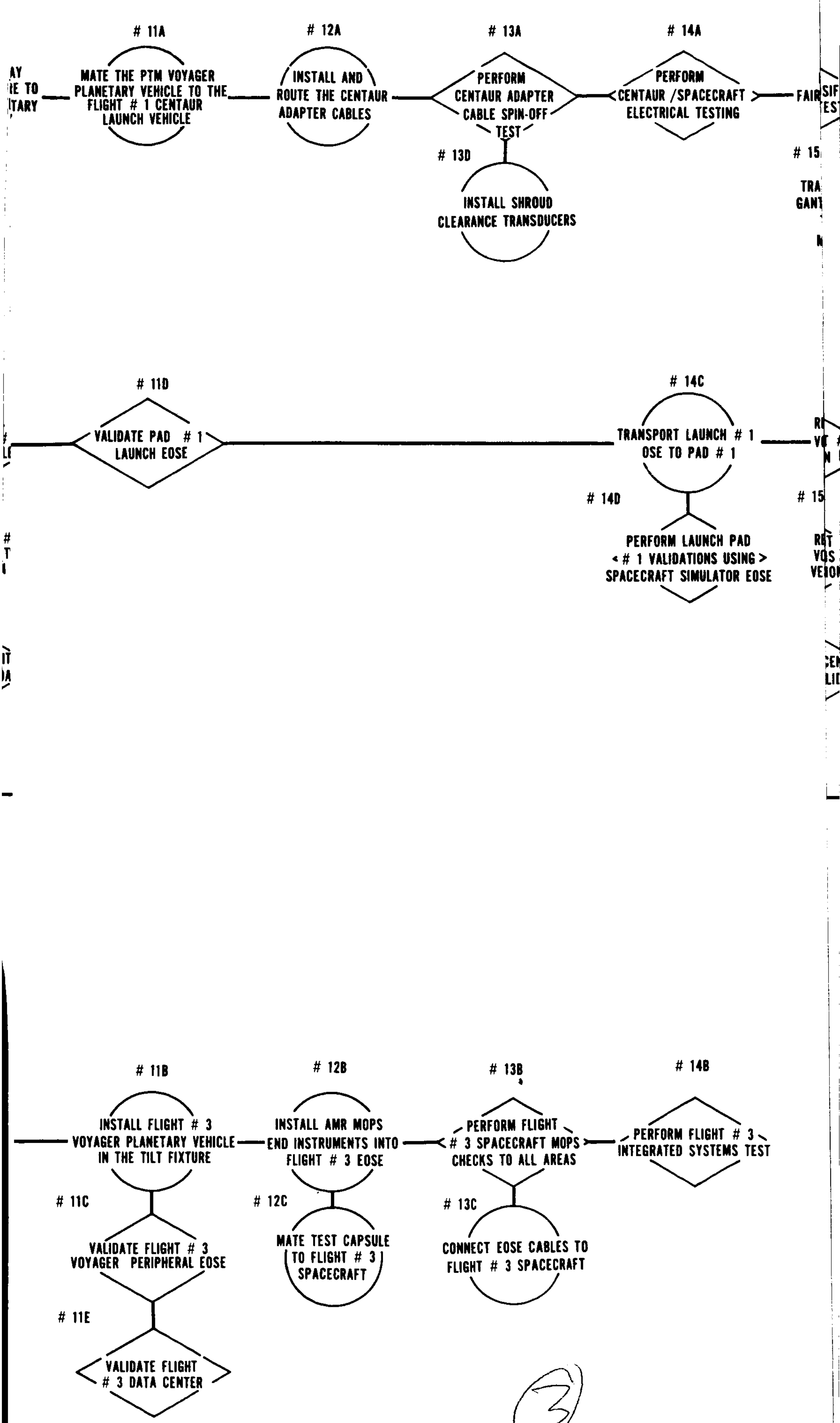


6A
PERFORM MOPS CHECKS TO ALL AREAS
#6B
CONNECT EOSE CABLES TO THE PTM SPACECRAFT
7A
PERFORM PTM VOYAGER PLANETARY VEHICLE INTEGRATED SYSTEMS TEST AND CRITIQUE
7B
TRANSPORT PAD # 2 EOSE TO CENTAUR ASSEMBLY AREA
8A
TRANSPORT PTM VOYAGER PLANETARY VEHICLE TO THE CENTAUR ASSEMBLY AREA
8B
TRANSPORT PTM PERIPHERAL EOSE (SYSTEM TEST SET) TO THE EXPLOSIVE SAFE AREA
9A
PERFORM CENTAUR ADAPTER FLATNESS CHECKS AT THE CENTAUR ASSEMBLY BUILDING
10
MATE SOLAR ARR
SUPPORT STRUCTU
THE VOYAGER PLANE
VEHICLE



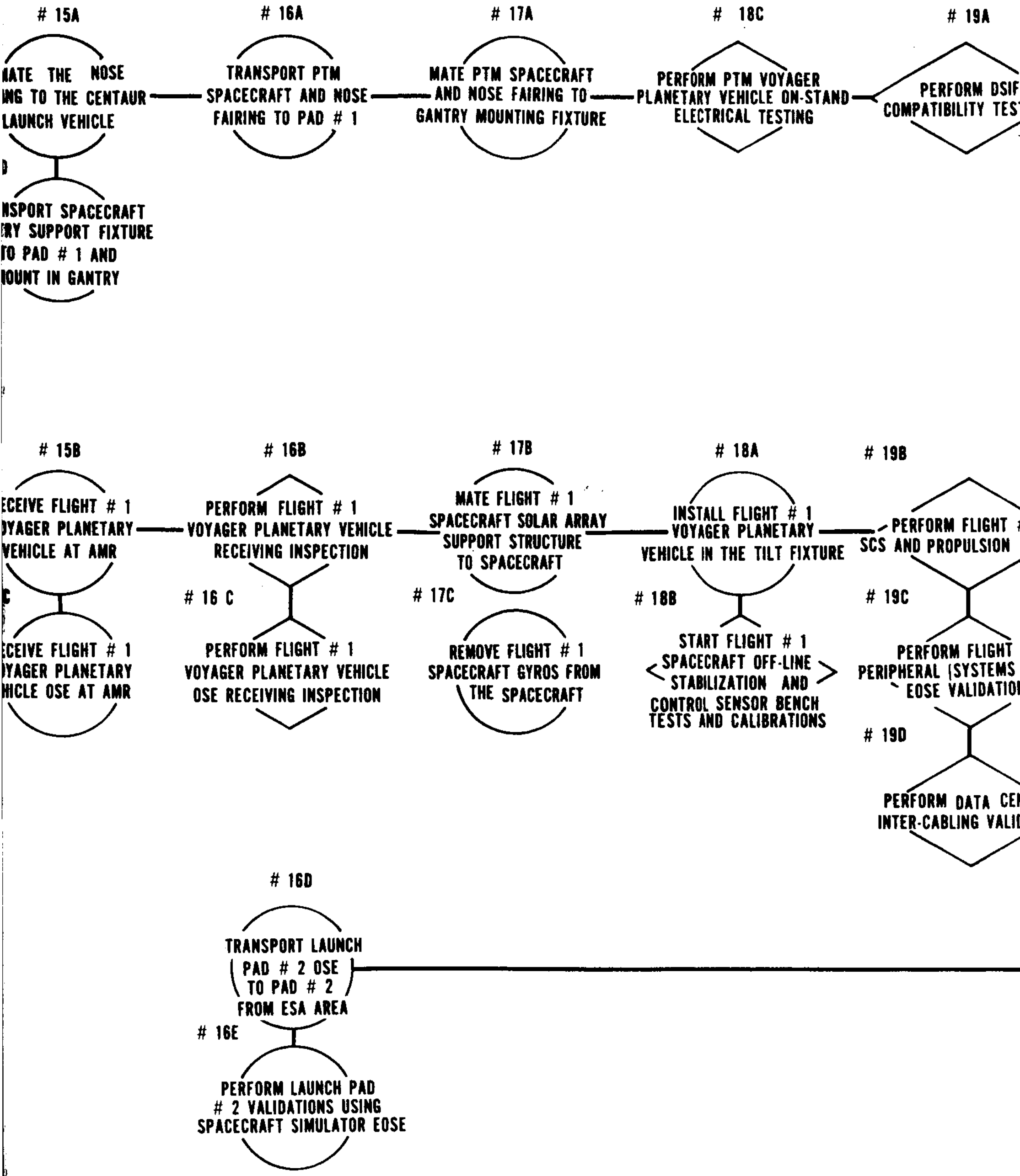
8C
RECEIVE FLIGHT # 3 VOYAGER PLANETARY VEHICLE AT AMR
8D
RECEIVE FLIGHT # 3 VOYAGER PLANETARY VEHICLE MOSE AND EOSE
9B
PERFORM FLIGHT # 3 VOYAGER PLANETARY VEHICLE RECEIVING INSPECTION
9C
PERFORM FLIGHT # 3 VOYAGER PLANETARY VEHICAL OSE RECEIVING INSPECTION

2



11A
MATE THE PTM VOYAGER PLANETARY VEHICLE TO THE FLIGHT # 1 CENTAUR LAUNCH VEHICLE
12A
INSTALL AND ROUTE THE CENTAUR ADAPTER CABLES
13A
PERFORM CENTAUR ADAPTER CABLE SPIN-OFF TEST
13D
INSTALL SHROUD CLEARANCE TRANSDUCERS
14A
PERFORM CENTAUR /SPACECRAFT ELECTRICAL TESTING
11D
VALIDATE PAD # 1 LAUNCH EOSE
14C
TRANSPORT LAUNCH # 1 OSE TO PAD # 1
14D
PERFORM LAUNCH PAD # 1 VALIDATIONS USING SPACECRAFT SIMULATOR EOSE
11B
INSTALL FLIGHT # 3 VOYAGER PLANETARY VEHICLE IN THE TILT FIXTURE
11C
VALIDATE FLIGHT # 3 VOYAGER PERIPHERAL EOSE
11E
VALIDATE FLIGHT # 3 DATA CENTER
12B
INSTALL AMR MOPS END INSTRUMENTS INTO FLIGHT # 3 EOSE
12C
MATE TEST CAPSULE TO FLIGHT # 3 SPACECRAFT
13B
PERFORM FLIGHT # 3 SPACECRAFT MOPS CHECKS TO ALL AREAS
13C
CONNECT EOSE CABLES TO FLIGHT # 3 SPACECRAFT
14B
PERFORM FLIGHT # 3 INTEGRATED SYSTEMS TEST

3



15A
IATE THE NOSE
ING TO THE CENTAUR
LAUNCH VEHICLE
NSPORT SPACECRAFT
RY SUPPORT FIXTURE
TO PAD # 1 AND
IOUNT IN GANTRY
16A
TRANSPORT PTM
SPACECRAFT AND NOSE
FAIRING TO PAD # 1
17A
MATE PTM SPACECRAFT
AND NOSE FAIRING TO
GANTRY MOUNTING FIXTURE
18C
PERFORM PTM VOYAGER
PLANETARY VEHICLE ON-STAND
ELECTRICAL TESTING
19A
PERFORM DSIF
COMPATIBILITY TES
15B
ECEIVE FLIGHT # 1
YAGER PLANETARY
VEHICLE AT AMR
CEIVE FLIGHT # 1
YAGER PLANETARY
HICLE OSE AT AMR
16B
PERFORM FLIGHT # 1
VOYAGER PLANETARY VEHICLE
RECEIVING INSPECTION
16 C
PERFORM FLIGHT # 1
VOYAGER PLANETARY VEHICLE
OSE RECEIVING INSPECTION
17B
MATE FLIGHT # 1
SPACECRAFT SOLAR ARRAY
SUPPORT STRUCTURE
TO SPACECRAFT
17C
REMOVE FLIGHT # 1
SPACECRAFT GYROS FROM
THE SPACECRAFT
18A
INSTALL FLIGHT # 1
VOYAGER PLANETARY
VEHICLE IN THE TILT FIXTURE
18B
START FLIGHT # 1
SPACECRAFT OFF-LINE
STABILIZATION AND
CONTROL SENSOR BENCH
TESTS AND CALIBRATIONS
19B
PERFORM FLIGHT
SCS AND PROPULSION
19C
PERFORM FLIGHT
PERIPHERAL (SYSTEMS
EOSE VALIDATIO
19D
PERFORM DATA CE
INTER-CABLING VALI
16D
TRANSPORT LAUNCH
PAD # 2 OSE
TO PAD # 2
FROM ESA AREA
16E
PERFORM LAUNCH PAD
2 VALIDATIONS USING
SPACECRAFT SIMULATOR EOSE

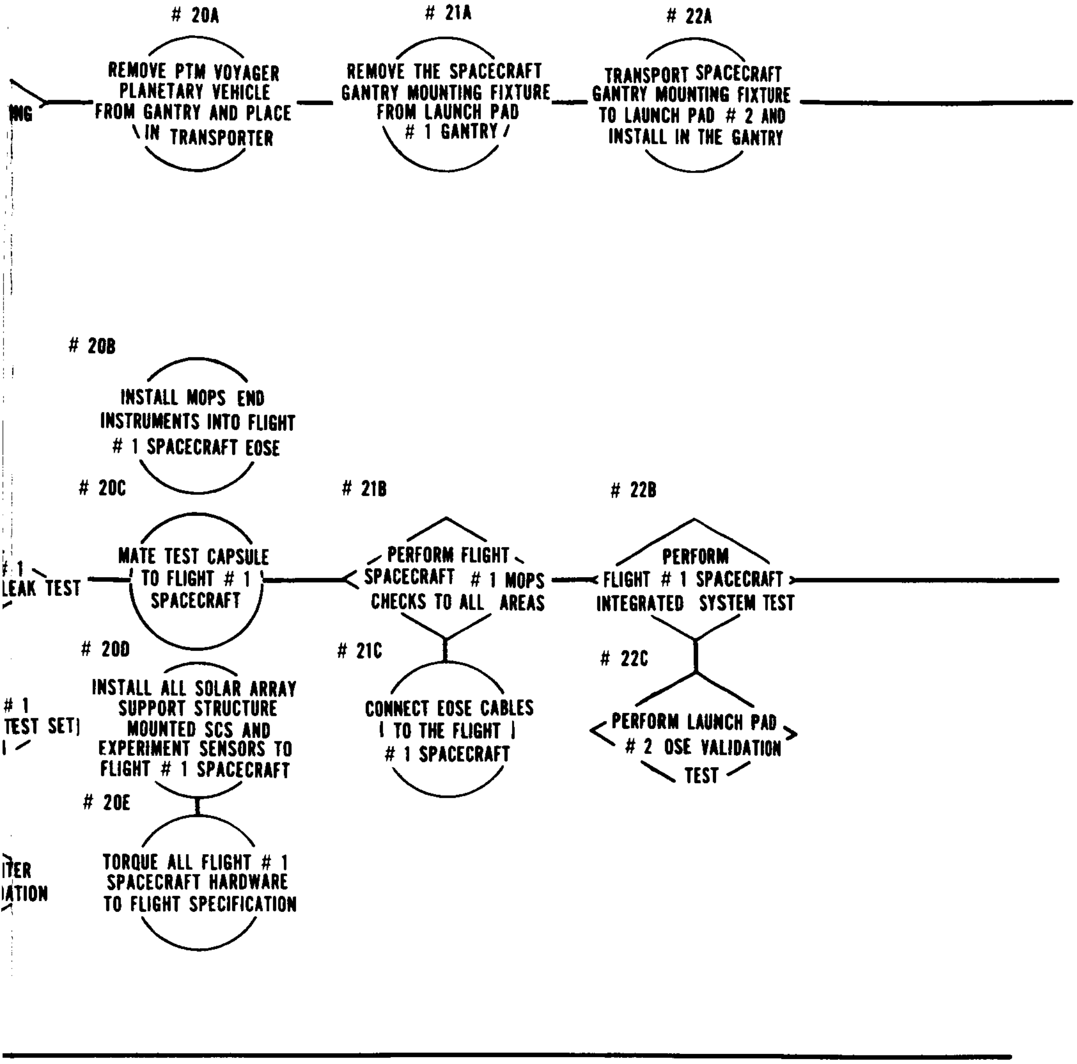
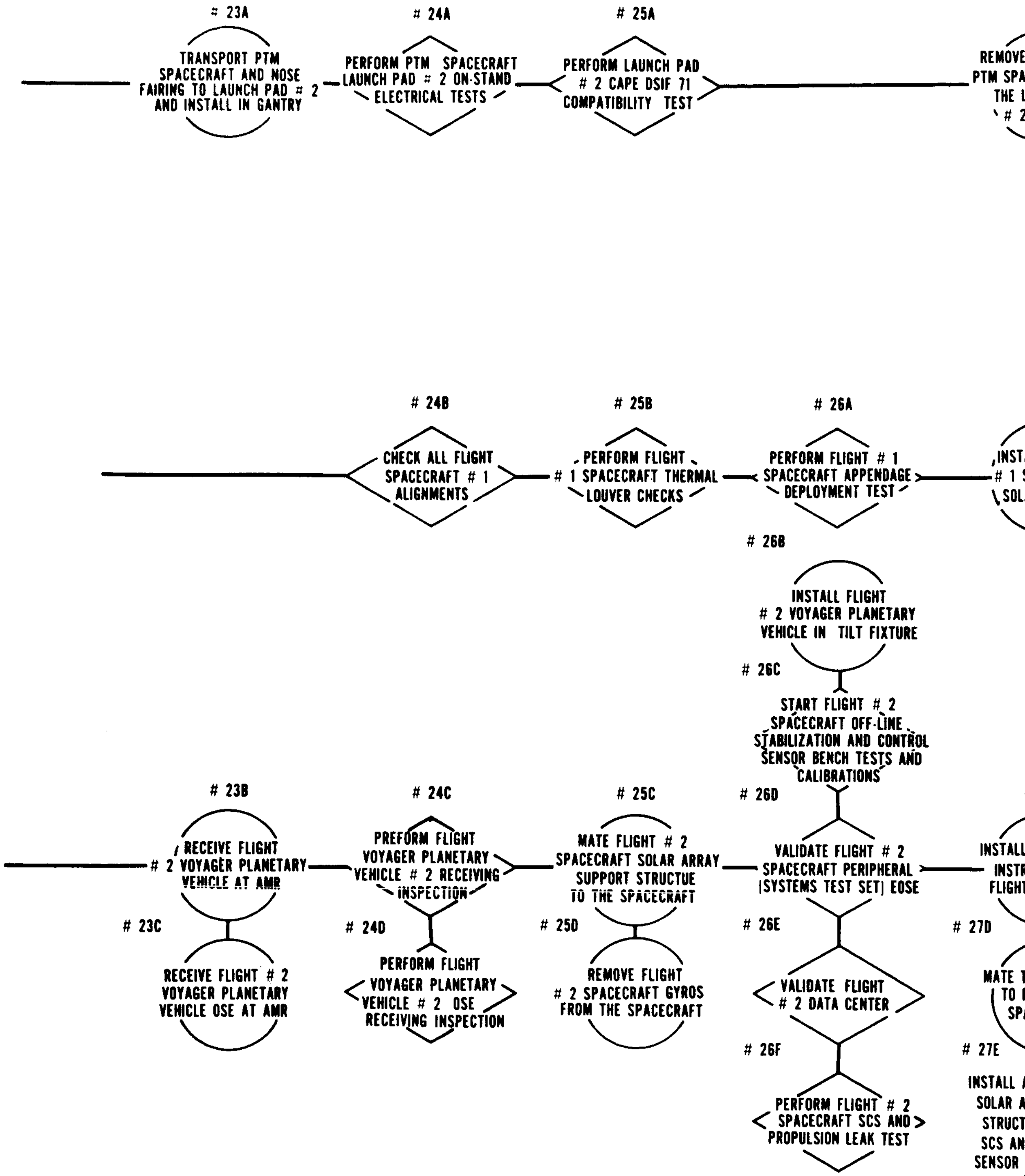


Figure 5-45. 1971 Voyager Launch Operations

5

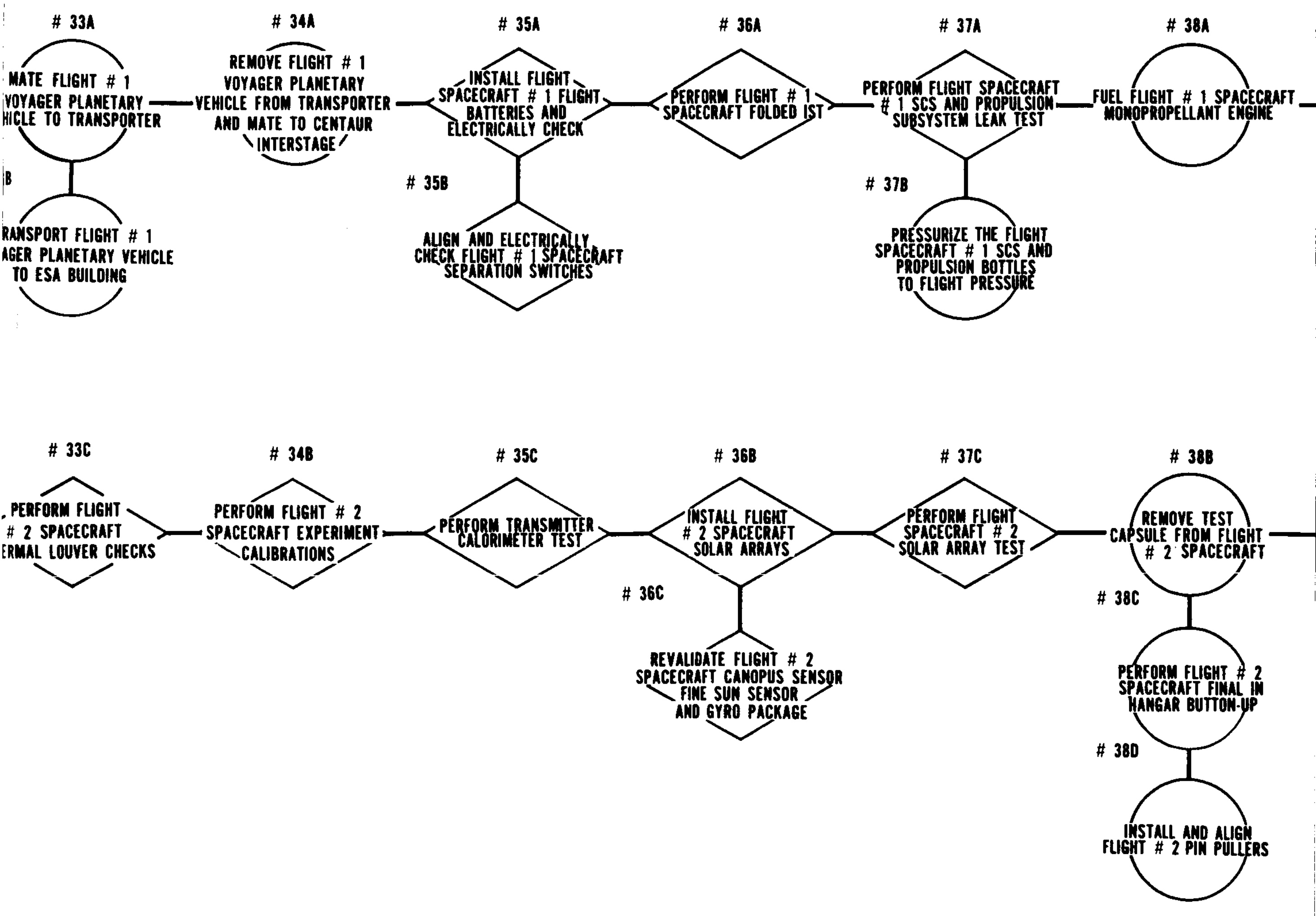


23A
TRANSPORT PTM SPACECRAFT AND NOSE FAIRING TO LAUNCH PAD # 2 AND INSTALL IN GANTRY
24A
PERFORM PTM SPACECRAFT LAUNCH PAD # 2 ON-STAND ELECTRICAL TESTS
25A
PERFORM LAUNCH PAD # 2 CAPE DSIF 71 COMPATIBILITY TEST
REMOVE
PTM SPA
THE L
2
24B
CHECK ALL FLIGHT SPACECRAFT # 1 ALIGNMENTS
25B
PERFORM FLIGHT # 1 SPACECRAFT THERMAL LOUVER CHECKS
26A
PERFORM FLIGHT # 1 SPACECRAFT APPENDAGE DEPLOYMENT TEST
INST
1 S
SOL
26B
INSTALL FLIGHT # 2 VOYAGER PLANETARY VEHICLE IN TILT FIXTURE
26C
START FLIGHT # 2 SPACECRAFT OFF-LINE STABILIZATION AND CONTROL SENSOR BENCH TESTS AND CALIBRATIONS
23B
RECEIVE FLIGHT # 2 VOYAGER PLANETARY VEHICLE AT AMR
24C
PREFORM FLIGHT VOYAGER PLANETARY VEHICLE # 2 RECEIVING INSPECTION
25C
MATE FLIGHT # 2 SPACECRAFT SOLAR ARRAY SUPPORT STRUCTUE TO THE SPACECRAFT
26D
VALIDATE FLIGHT # 2 SPACECRAFT PERIPHERAL (SYSTEMS TEST SET) EOSE
INSTALL
INSTR
FLIGHT
23C
RECEIVE FLIGHT # 2 VOYAGER PLANETARY VEHICLE OSE AT AMR
24D
PERFORM FLIGHT VOYAGER PLANETARY VEHICLE # 2 OSE RECEIVING INSPECTION
25D
REMOVE FLIGHT # 2 SPACECRAFT GYROS FROM THE SPACECRAFT
26E
VALIDATE FLIGHT # 2 DATA CENTER
27D
MATE T
TO
SP
26F
PERFORM FLIGHT # 2 SPACECRAFT SCS AND PROPULSION LEAK TEST
27E
INSTALL
SOLAR
STRUCT
SCS AN
SENSOR

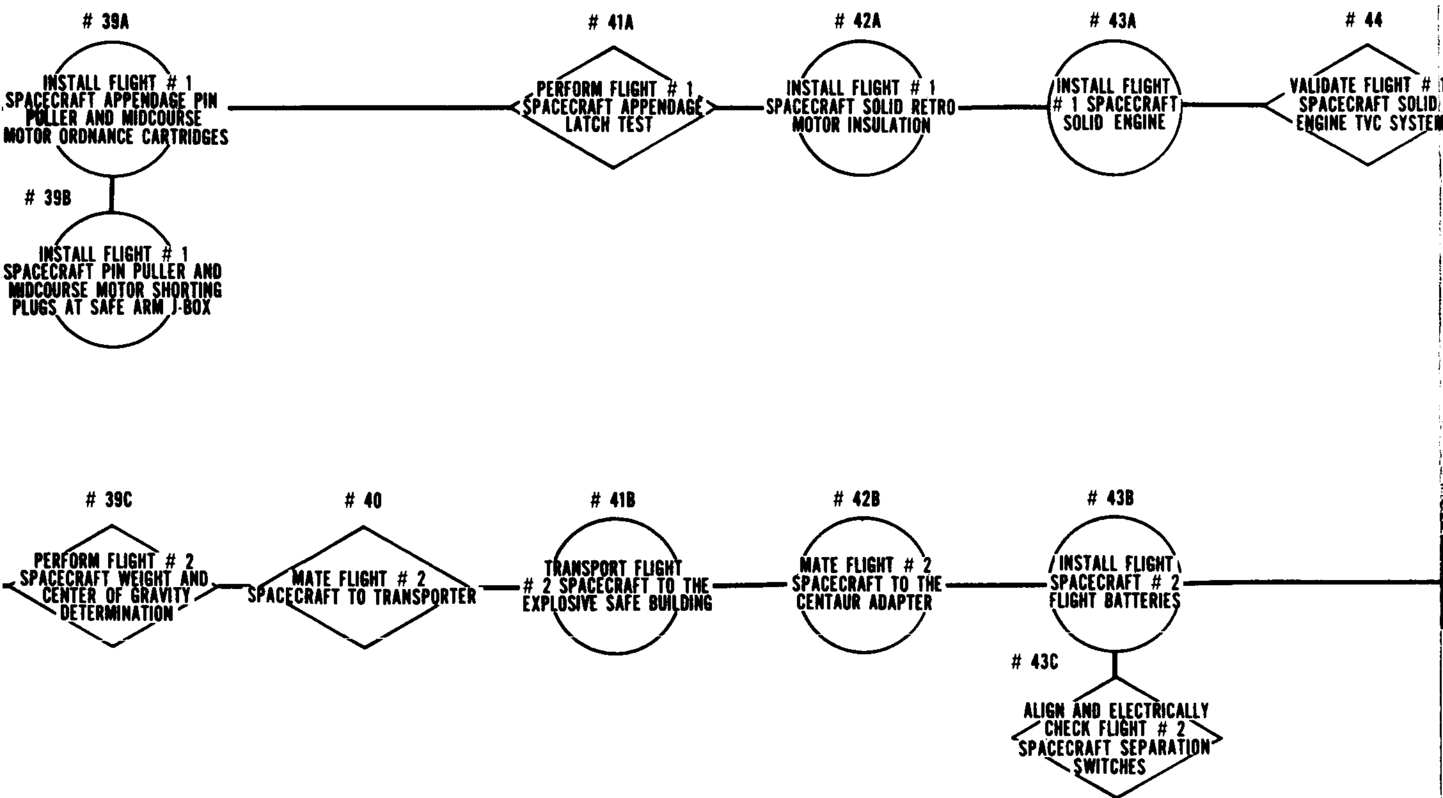
D

27A
28A
REMOVE THE SPACECRAFT GANTRY SUPPORT FIXTURE FROM LAUNCH PAD # 2 GANTRY
28D
TRANSPORT PTM SPACECRAFT TO ESA AREA
29A
TRANSPORT THE SPACECRAFT GANTRY SUPPORT FIXTURE TO THE ESA AREA
30B
VALIDATE ESA TEST COMPLEX USING PTM SPACECRAFT
27B
28B
PERFORM FLIGHT SPACECRAFT # 1 EXPERIMENT CALIBRATIONS
29B
PERFORM FLIGHT # 1 SPACECRAFT TRANSMITTER CALORIMETER TEST
30A
PERFORM FLIGHT # 1 SPACECRAFT SOLAR ARRAY TEST
31A
REMOVE TEST CAPSULE FROM FLIGHT # 1 SPACECRAFT
31B
PERFORM FINAL FLIGHT # 1 IN HANGAR BUTTON-UP
31C
INSTALL AND ALIGN FLIGHT # 1 APPENDAGE PIN PULLERS
32A
PERFORM FLIGHT # 1 SPACECRAFT WEIGHT AND CENTER OF GRAVITY DETERMINATION
27C
28C
CONNECT EOSE CABLES TO FLIGHT # 2 SPACECRAFT
28E
TORQUE ALL FLIGHT # 2 SPACECRAFT HARDWARE TO FLIGHT SPECIFICATION
29C
PERFORM FLIGHT SPACECRAFT # 2 INTEGRATED SYSTEMS TEST
31D
CHECK ALL FLIGHT # 2 SPACECRAFT ALIGNMENTS
32B
PERFORM FLIGHT # 2 SPACECRAFT DEPLOYMENT TEST

2



33A
MATE FLIGHT # 1 VOYAGER PLANETARY HICLE TO TRANSPORTER
B
RANSPORT FLIGHT # 1 AGER PLANETARY VEHICLE TO ESA BUILDING
34A
REMOVE FLIGHT # 1 VOYAGER PLANETARY VEHICLE FROM TRANSPORTER AND MATE TO CENTAUR INTERSTAGE
35A
INSTALL FLIGHT SPACECRAFT # 1 FLIGHT BATTERIES AND ELECTRICALLY CHECK
35B
ALIGN AND ELECTRICALLY CHECK FLIGHT # 1 SPACECRAFT SEPARATION SWITCHES
36A
PERFORM FLIGHT # 1 SPACECRAFT FOLDED IST
37A
PERFORM FLIGHT SPACECRAFT # 1 SCS AND PROPULSION SUBSYSTEM LEAK TEST
37B
PRESSURIZE THE FLIGHT SPACECRAFT # 1 SCS AND PROPULSION BOTTLES TO FLIGHT PRESSURE
38A
FUEL FLIGHT # 1 SPACECRAFT MONOPROPELLANT ENGINE
33C
PERFORM FLIGHT # 2 SPACECRAFT ERMAL LOUVER CHECKS
34B
PERFORM FLIGHT # 2 SPACECRAFT EXPERIMENT CALIBRATIONS
35C
PERFORM TRANSMITTER CALORIMETER TEST
36B
INSTALL FLIGHT # 2 SPACECRAFT SOLAR ARRAYS
36C
REVALIDATE FLIGHT # 2 SPACECRAFT CANOPUS SENSOR FINE SUN SENSOR AND GYRO PACKAGE
37C
PERFORM FLIGHT SPACECRAFT # 2 SOLAR ARRAY TEST
38B
REMOVE TEST CAPSULE FROM FLIGHT # 2 SPACECRAFT
38C
PERFORM FLIGHT # 2 SPACECRAFT FINAL IN HANGAR BUTTON-UP
38D
INSTALL AND ALIGN FLIGHT # 2 PIN PULLERS



39A
INSTALL FLIGHT # 1 SPACECRAFT APPENDAGE PIN PULLER AND MIDCOURSE MOTOR ORDNANCE CARTRIDGES
39B
INSTALL FLIGHT # 1 SPACECRAFT PIN PULLER AND MIDCOURSE MOTOR SHORTING PLUGS AT SAFE ARM J-BOX
41A
PERFORM FLIGHT # 1 SPACECRAFT APPENDAGE LATCH TEST
42A
INSTALL FLIGHT # 1 SPACECRAFT SOLID RETRO MOTOR INSULATION
43A
INSTALL FLIGHT # 1 SPACECRAFT SOLID ENGINE
44
VALIDATE FLIGHT # 1 SPACECRAFT SOLID ENGINE TVC SYSTEM
39C
PERFORM FLIGHT # 2 SPACECRAFT WEIGHT AND CENTER OF GRAVITY DETERMINATION
40
MATE FLIGHT # 2 SPACECRAFT TO TRANSPORTER
41B
TRANSPORT FLIGHT # 2 SPACECRAFT TO THE EXPLOSIVE SAFE BUILDING
42B
MATE FLIGHT # 2 SPACECRAFT TO THE CENTAUR ADAPTER
43B
INSTALL FLIGHT SPACECRAFT # 2 FLIGHT BATTERIES
43C
ALIGN AND ELECTRICALLY CHECK FLIGHT # 2 SPACECRAFT SEPARATION SWITCHES

4

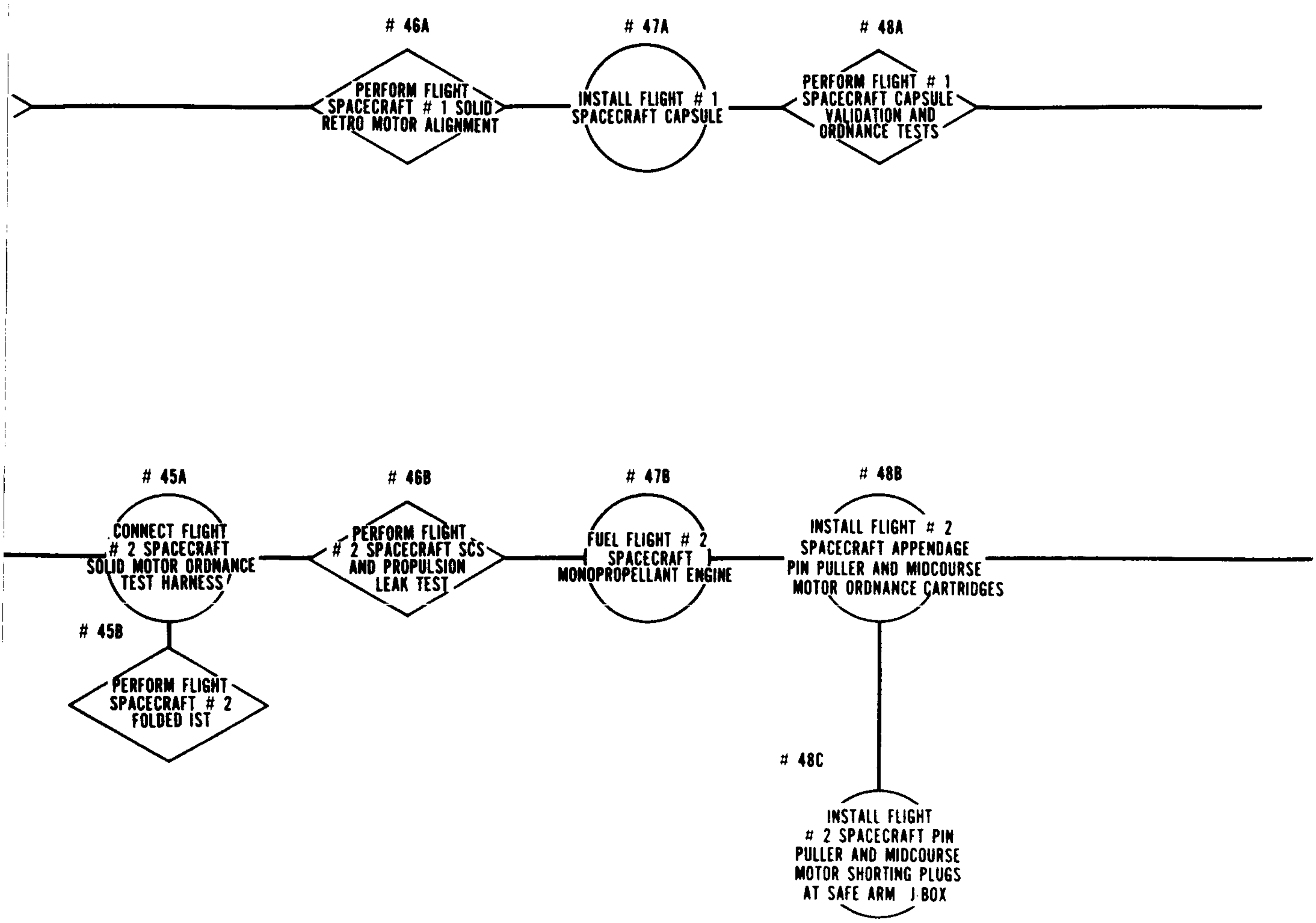
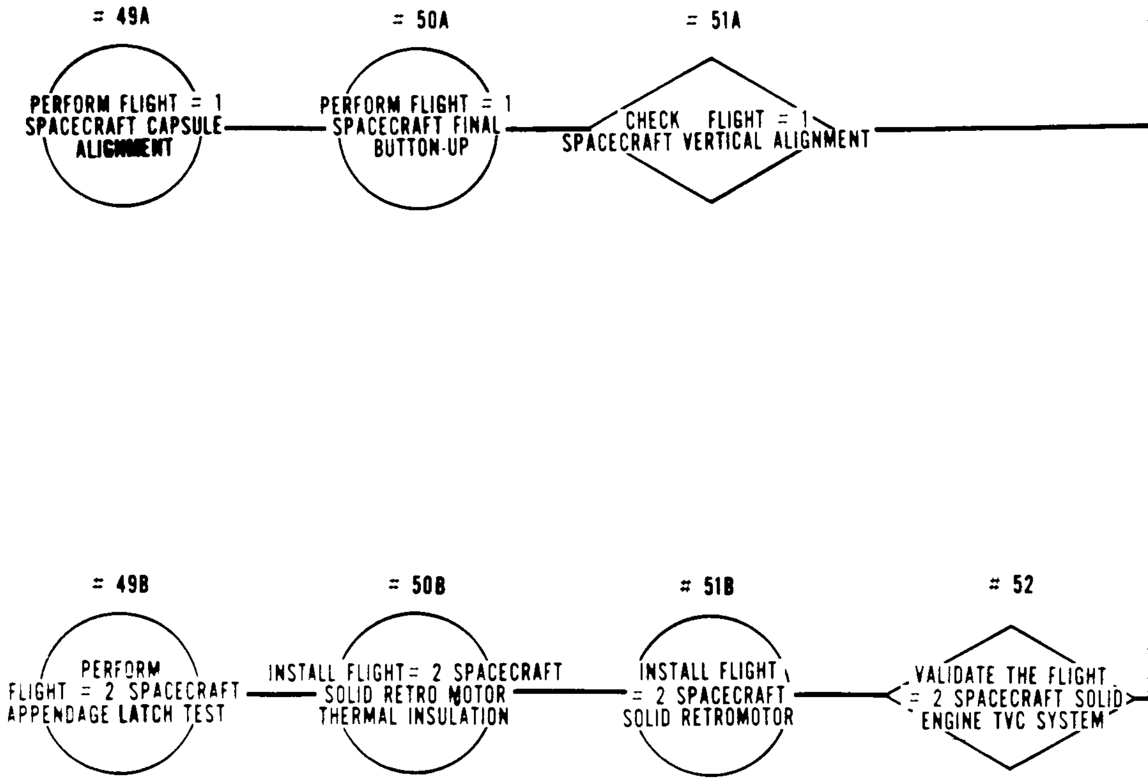
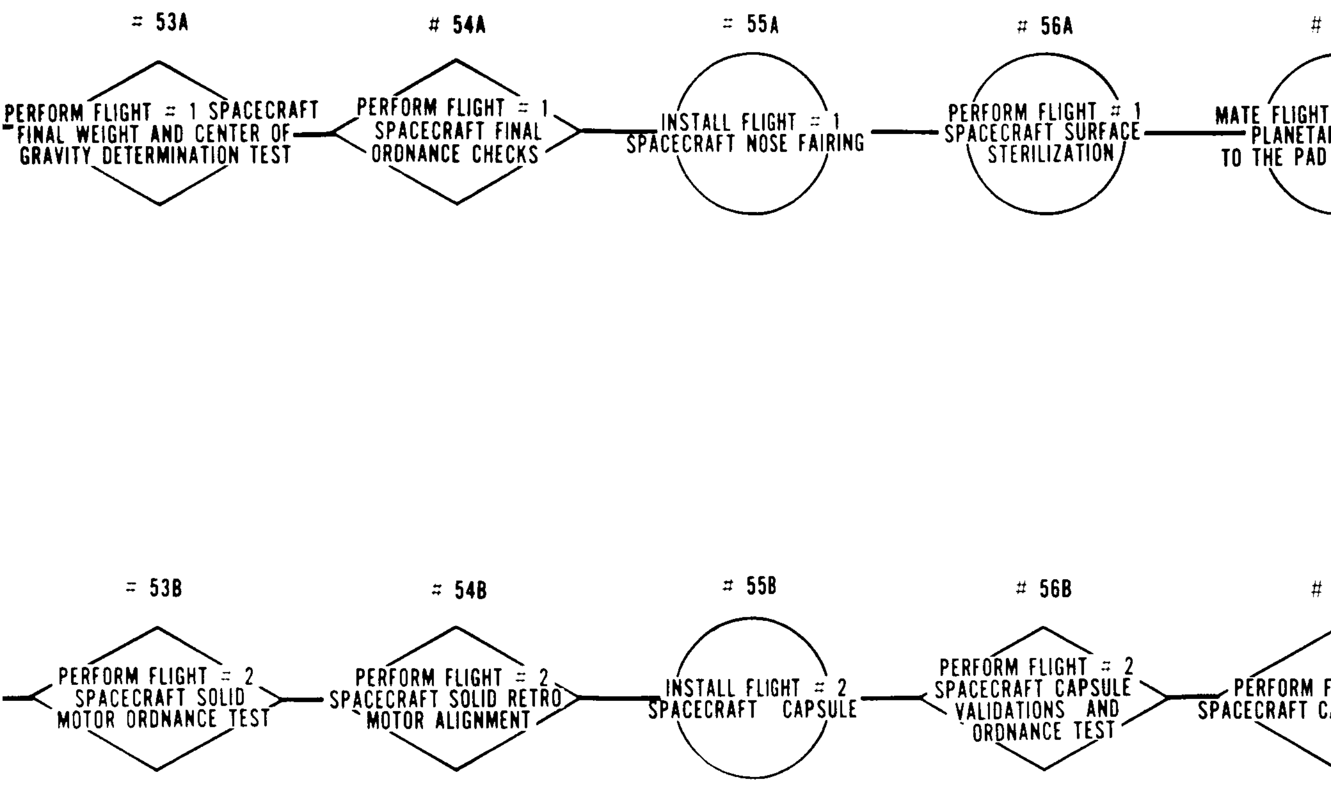


Figure 5-45. 1971 Voyager Launch Operations (Continued)

5

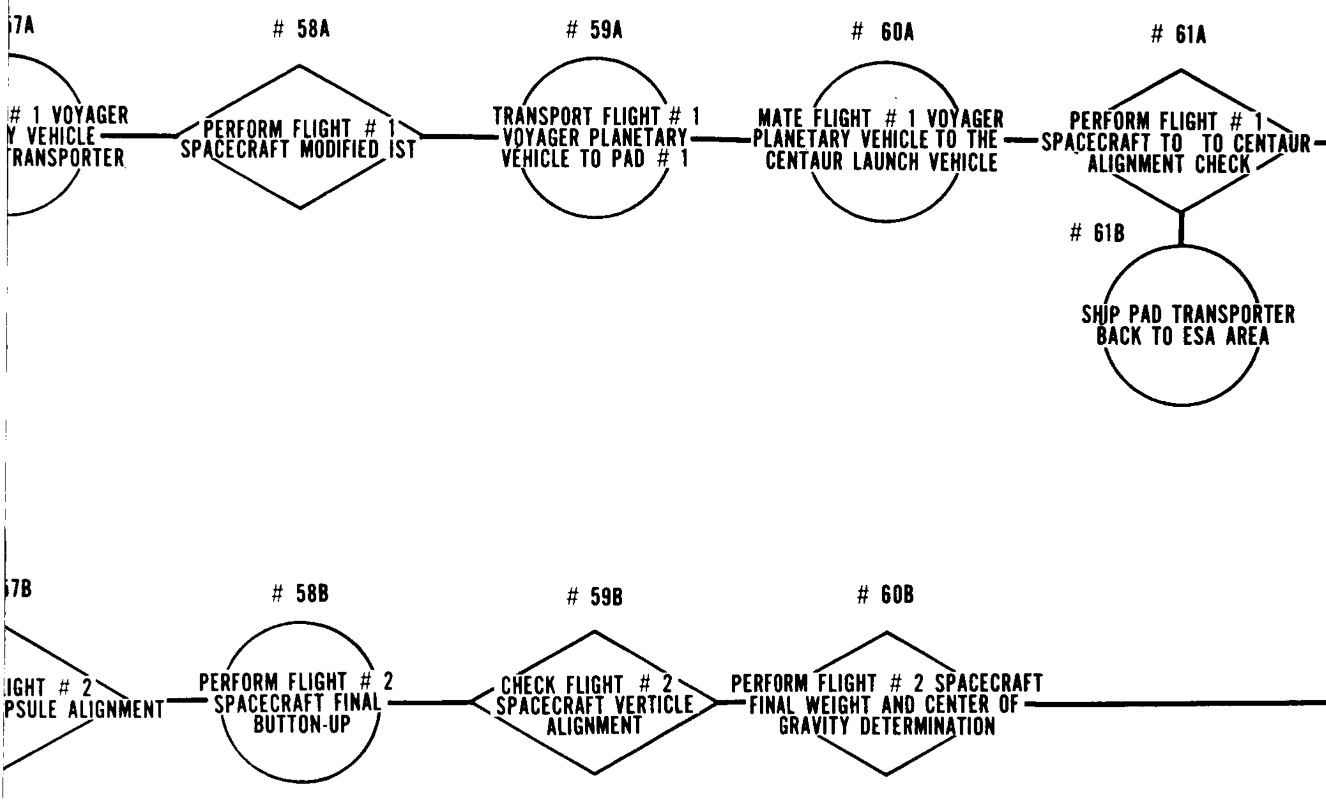


≠ 49A
PERFORM FLIGHT ≠ 1 SPACECRAFT CAPSULE ALIGNMENT
≠ 50A
PERFORM FLIGHT ≠ 1 SPACECRAFT FINAL BUTTON-UP
≠ 51A
CHECK FLIGHT ≠ 1 SPACECRAFT VERTICAL ALIGNMENT
≠ 49B
PERFORM FLIGHT ≠ 2 SPACECRAFT APPENDAGE LATCH TEST
≠ 50B
INSTALL FLIGHT ≠ 2 SPACECRAFT SOLID RETRO MOTOR THERMAL INSULATION
≠ 51B
INSTALL FLIGHT ≠ 2 SPACECRAFT SOLID RETROMOTOR
≠ 52
VALIDATE THE FLIGHT ≠ 2 SPACECRAFT SOLID ENGINE TVC SYSTEM



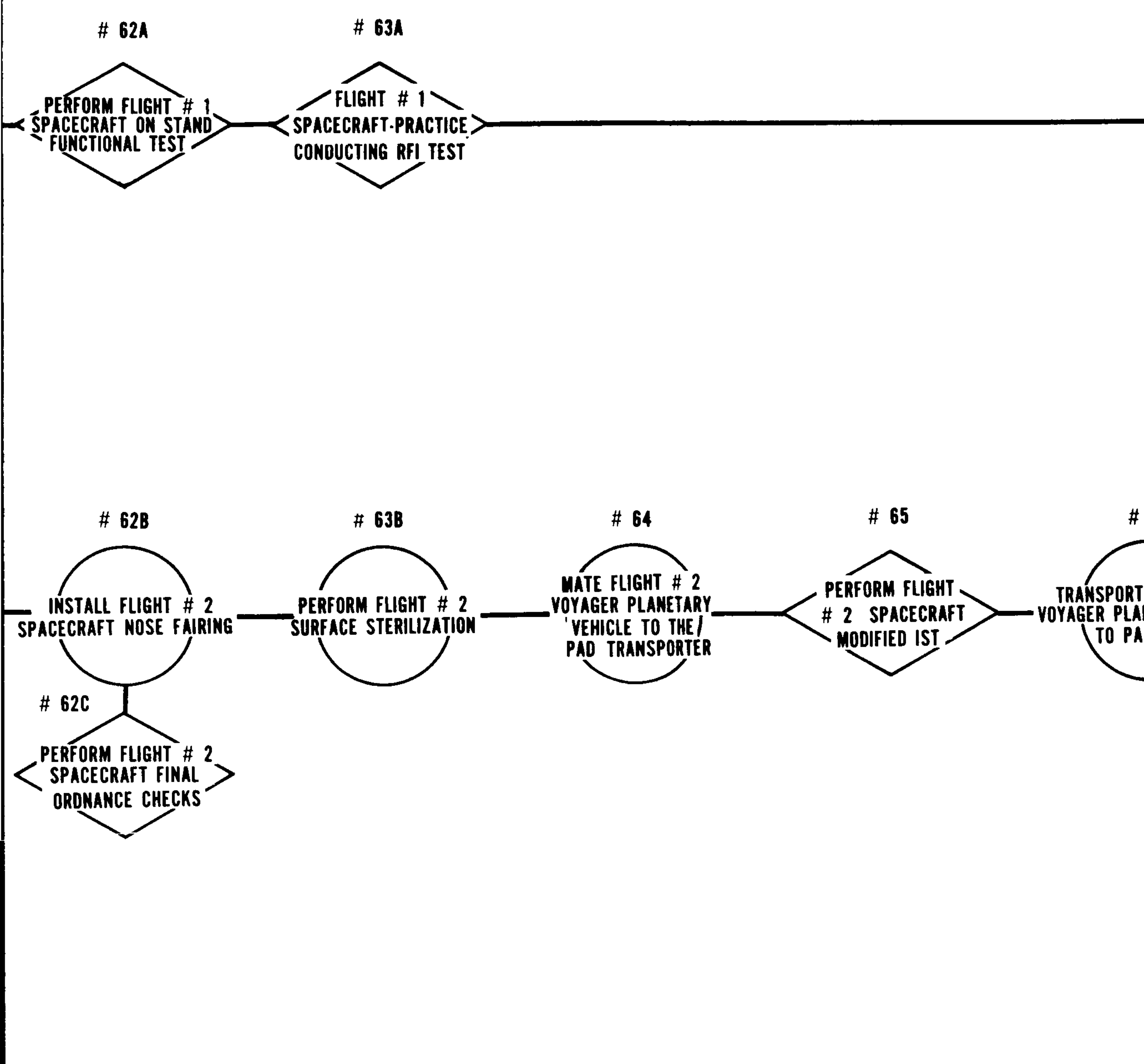
53A
PERFORM FLIGHT # 1 SPACECRAFT FINAL WEIGHT AND CENTER OF GRAVITY DETERMINATION TEST
54A
PERFORM FLIGHT # 1 SPACECRAFT FINAL ORDNANCE CHECKS
55A
INSTALL FLIGHT # 1 SPACECRAFT NOSE FAIRING
56A
PERFORM FLIGHT # 1 SPACECRAFT SURFACE STERILIZATION
#
MATE FLIGHT
PLANETA
TO THE PAD
53B
PERFORM FLIGHT # 2 SPACECRAFT SOLID MOTOR ORDNANCE TEST
54B
PERFORM FLIGHT # 2 SPACECRAFT SOLID RETRO MOTOR ALIGNMENT
55B
INSTALL FLIGHT # 2 SPACECRAFT CAPSULE
56B
PERFORM FLIGHT # 2 SPACECRAFT CAPSULE VALIDATIONS AND ORDNANCE TEST
#
PERFORM F
SPACECRAFT C

2



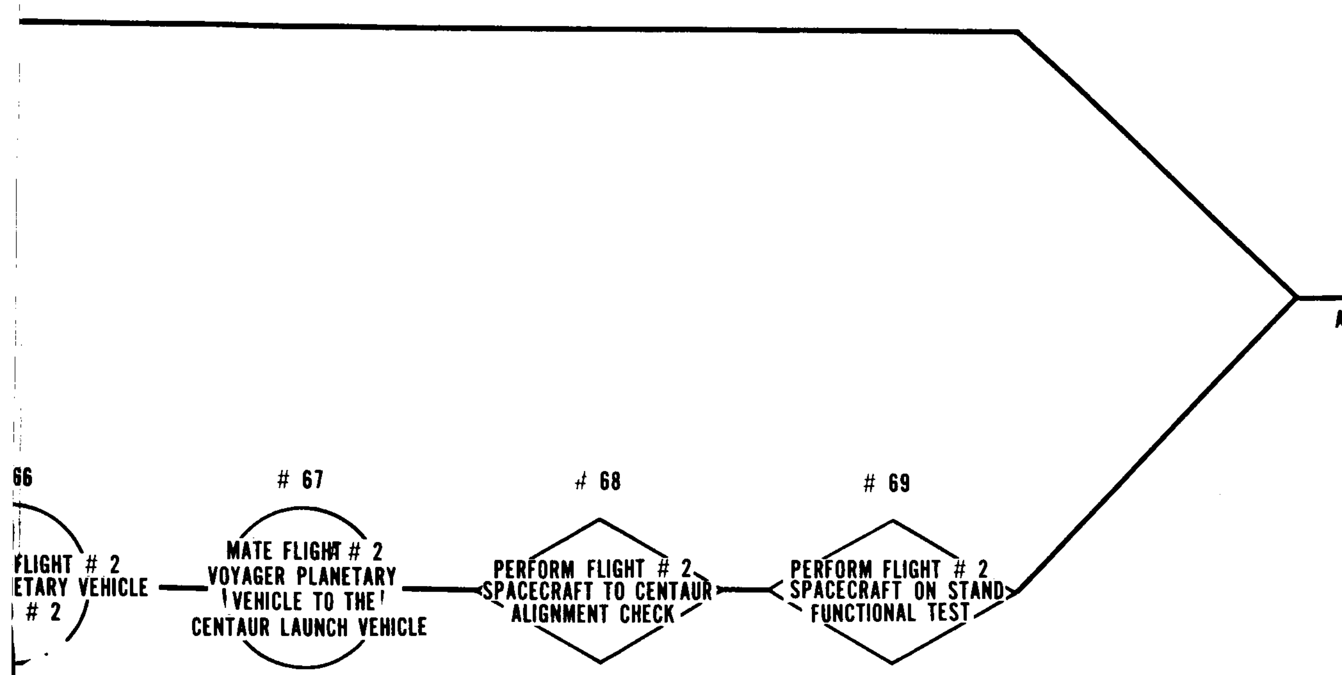
7A
1 VOYAGER
VEHICLE
TRANSPORTER
58A
PERFORM FLIGHT # 1 SPACECRAFT MODIFIED IST
59A
TRANSPORT FLIGHT # 1 VOYAGER PLANETARY VEHICLE TO PAD # 1
60A
MATE FLIGHT # 1 VOYAGER PLANETARY VEHICLE TO THE CENTAUR LAUNCH VEHICLE
61A
PERFORM FLIGHT # 1 SPACECRAFT TO TO CENTAUR ALIGNMENT CHECK
61B
SHIP PAD TRANSPORTER BACK TO ESA AREA
7B
IGHT # 2
PSULE ALIGNMENT
58B
PERFORM FLIGHT # 2 SPACECRAFT FINAL BUTTON-UP
59B
CHECK FLIGHT # 2 SPACECRAFT VERTICLE ALIGNMENT
60B
PERFORM FLIGHT # 2 SPACECRAFT FINAL WEIGHT AND CENTER OF GRAVITY DETERMINATION

3



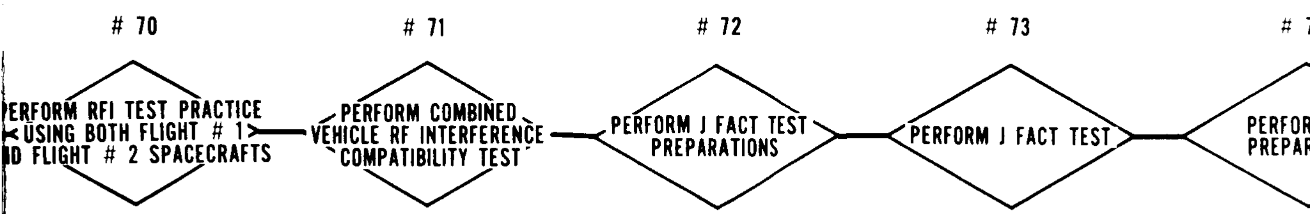
62A
PERFORM FLIGHT # 1 SPACECRAFT ON STAND FUNCTIONAL TEST
63A
FLIGHT # 1 SPACECRAFT-PRACTICE CONDUCTING RFI TEST
62B
INSTALL FLIGHT # 2 SPACECRAFT NOSE FAIRING
62C
PERFORM FLIGHT # 2 SPACECRAFT FINAL ORDNANCE CHECKS
63B
PERFORM FLIGHT # 2 SURFACE STERILIZATION
64
MATE FLIGHT # 2 VOYAGER PLANETARY VEHICLE TO THE PAD TRANSPORTER
65
PERFORM FLIGHT # 2 SPACECRAFT MODIFIED IST
TRANSPORT

4



66
FLIGHT # 2
ETARY VEHICLE
2
67
MATE FLIGHT # 2
VOYAGER PLANETARY
VEHICLE TO THE
CENTAUR LAUNCH VEHICLE
68
PERFORM FLIGHT # 2
SPACECRAFT TO CENTAUR
ALIGNMENT CHECK
69
PERFORM FLIGHT # 2
SPACECRAFT ON STAND
FUNCTIONAL TEST

3



70
ERFORM RFI TEST PRACTICE
USING BOTH FLIGHT # 1
D FLIGHT # 2 SPACECRAFTS
71
PERFORM COMBINED
VEHICLE RF INTERFERENCE
COMPATIBILITY TEST
72
PERFORM J FACT TEST
PREPARATIONS
73
PERFORM J FACT TEST
PERFOR
PREPAR

6

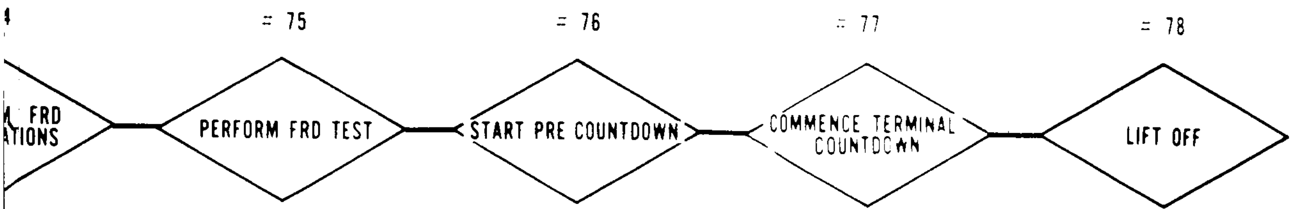


Figure 5-45. 1971 Voyager Launch Operations (Continued)

All structure-mounted stabilization and control subsystem and experiment sensors will be installed to the solar array support structure. Concurrently, all OSE will be validated and the spacecraft mounted on the tilt fixture. After OSE validation, the SCS and experiment sensors will be electrically validated in the spacecraft, and the proof test model capsule mated to the proof test model spacecraft and validated. As an off-line task, the MOPS end instruments will be installed into all applicable EOSE and checked with all areas. The proof test model spacecraft integrated systems test will be performed, proving that the spacecraft is working properly and can proceed with its assigned tasks.

The proof test model spacecraft and Launch Pad No. 1 EOSE will be transported to the Centaur assembly area to support the Centaur-spacecraft interface testing. Concurrently, the peripheral EOSE will be transported to the explosive safe area. The proof test model spacecraft mechanical interface tests at the Centaur assembly area are:

- Centaur-spacecraft interstage fit and alignment tests
- Install and route interstage cables
- Nose fairing clearance tests
- Spin-off connector clearance test

The proof test model spacecraft electrical interface tests at the Centaur assembly area are:

- Validate all umbilical electrical functions
- Validate all Centaur-initiated spacecraft ordnance functions
- Determine nose fairing RF coupler losses

The spacecraft gantry support fixture will be transported to Pad No. 1. The proof test model spacecraft and nose fairing, having concluded the interface tests at the Centaur assembly area, will be transported to Pad No. 1 and mated to the spacecraft gantry support fixture. The spacecraft will be electrically powered from the blockhouse EOSE which had previously been validated using the spacecraft simulator. The proof test model spacecraft on-stand electrical tests are:

- Validate all electrical umbilical functions using blockhouse EOSE
- Determine RF nose fairing coupler losses
- Determine RF nose fairing air loss between the DSIF station and the spacecraft and between the spacecraft assembly area and the spacecraft.

In addition, the on-stand air conditioning, purging, and sterilization equipment compatibility tests will be performed.

The proof test model spacecraft and gantry support fixture will be transported to Pad No. 2 and the launch pad tests repeated.

At the conclusion of the Pad No. 2 testing, the proof test model spacecraft and spacecraft gantry support fixture will be transported back to the spacecraft explosive safe area, for validating the STC and associated capsule equipment.

Concurrently with the AFETR testing on the proof test model, the flight 1, 2, and 3 spacecraft and OSE are received and inspected at the spacecraft assembly and test facility. The flight SCS and experiment sensors will be bench tested and calibrated while the solar array support structure is being mated to the flight spacecraft. After the solar array support structure is mated to the flight spacecraft, which in turn has been mated to the tilt fixture, the SCS and experiment sensor will be mounted to the supported structure and electrically validated. While the SCS and experiment sensors are being calibrated in the laboratory, all of flight EOSE will be validated. The MOPS ETR end instruments will be installed and checked in each applicable EOSE.

The flight spacecraft integrated systems test will be performed, demonstrating that each spacecraft is performing properly. At the conclusion of the flight, No. 3 spacecraft will become a means of acquiring electrical running time of all spare black boxes. No spare black box will be placed upon the first or second flight spacecraft unless it has been taken from the third flight spacecraft.

The first and second flight spacecraft SCS pneumatic system and midcourse correction engine leak tests will be performed to prove that the SCS and midcourse correction engine tanks are in a condition to be filled to flight levels. Next, all spacecraft alignments will be checked to insure that there have been no alignment shifts during shipping and handling. After the alignment checks, the spacecraft thermal louvers will be tested using the spray technique. Each louver will be sprayed with a highly evaporative fluid to cool and actuate the louvers. After the louver tests have been completed, the experiment calibrations will commence. No experiment will be removed during the calibration; and all calibrations must be performed with the experiments installed in the spacecraft.

After the experiment calibrations the RF transmitter calorimeter test will be performed, measuring each spacecraft transmitter to the nearest 0.1 db. The following in-hangar test will be performed on the solar array:

- Perform inverse impedance test on each solar array panel
- Illuminate each array panel and measure the open circuit-voltage and short-circuit current

All flight items will be torqued to specification and thermal control surfaces, SCS and experiment sensors, solar array cells and safety wiring, RF connectors, and other applicable spacecraft hardware cleaned. All appendage flight pin pullers and the flight retropropulsion engine thermal insulation will be installed.

The spacecraft will be transported to the explosive safe area and mechanically mated to the Centaur adapter. The separation switches will be adjusted and electrically tested and the flight batteries installed and electrically tested, unless it has been determined that the flight batteries will be installed in the spacecraft assembly area. A folded integrated systems test will be performed, that is, no spacecraft appendages will be articulated. The flight spacecraft midcourse correction and SCS penumatic system will be pressurized to flight levels. The midcourse correction engine will be fueled and its ignitor ordnance cartridges installed with shorting plugs connected. Next, the pin puller ordnance cartridges will be installed in each pin puller and the shorting plugs installed.

Each spacecraft appendage will be manufally deployed, observing that the appendage freely deploys with no chaffing or restriction.

The solid retropropulsion engine and the flight capsule will be installed and aligned to the spacecraft. After the flight capsule has been aligned to the spacecraft, a capsule interface test will be conducted. All electrical and mechanical interfaces added since the hangar testing will be checked. All sensors and the solar arrays will be cleaned. Spacecraft vertical alignment will be checked. A final weight and center of gravity determination will be made.

The spacecraft ordnance tests will be performed as follows. Ascertain that the spacecraft is in a safe condition by observing that no voltage exists across each ordnance device and that no resistance exists across each ordnance device connector pin to frame ground. Next each ordnance device will be commanded to the armed condition and the proper voltage monitored at the input to each ordnance switch. The spacecraft will again be commanded to the ordnance safe condition, rechecked and connections completed. The spacecraft nose fairing will be installed and the spacecraft and its associated subsystems will be gas sterilized using the nose fairing as a sterilization container. The spacecraft will then undergo a modified integrated systems test which will grossly check each subsystem. After the modified integrated systems test has been completed, the spacecraft will be transported to Pad No. 1 and mated and aligned to the launch vehicle.

The on-stand functional test will in clude the following interfaces:

- All spacecraft umbilical functions between the spacecraft and the PAD No. 1 blockhouse

- Wideband video pair system between the spacecraft and the data centers

- RF link between the spacecraft and the data center

- RF link between the spacecraft and the DSIF station

Once the spacecraft interfaces have been tested, the radio frequency interference test will be performed. It is expected that only the spacecraft will participate in this test.

All No. 1 flight spacecraft on-stand activities will cease until the No. 2 flight spacecraft is mated to the launch vehicle at Pad No. 2, following testing activities identical to those for flight No. 1. From this point on, both the flight No. 1 and No. 2 spacecraft will participate concurrently in the remaining on-stand testing activities.

A combined vehicle RF interference test is performed to ascertain that none of the Centaur or Saturn transmitters or beacons interfere with the spacecraft transmitters or receivers and vice versa. The RFI compatibility test will be performed as follows:

- Each Saturn beacon and transmitter is turned on one at a time and both the Centaur and the spacecraft will ascertain that there is no interference with or degradation of the receiver or transmitter systems.

- Each Centaur beacon and transmitter is turned on one at a time and both the Saturn and the spacecraft will ascertain that there is no degradation of or interference with the receiver or transmitter systems.

- Each spacecraft transmitter is turned on one at a time and both the Saturn and Centaur vehicles will ascertain that there is no degradation of or interference with the receiver or transmitter systems.

- All spacecraft, Centaur, and Saturn transmitters are turned on together and each vehicle will ascertain that there are no mutual degradations of or interference with the various transmitting or receiving systems.

The J FACT test preparations are divided into the following tasks:

- Installation of the nose fairing separation squib simulators

- Installation of the spacecraft umbilical cable spin-off connector squid simulators

- Installation of the spacecraft separation squib simulators

The remainder of the day will be spent practicing the J FACT test procedure. It is expected that only the spacecraft will participate in this particular activity. After the J FACT test preparations have been completed, the J FACT test itself will check out the post-injection portion of the mission profile. The following spacecraft-related postlaunch functions will be monitored and checked.

- Nose fairing separation

- Spacecraft umbilical cable separation

- Spacecraft separation from the Centaur

Since 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.

Next, the FRD test preparations will take place followed by the FRD, duplicating the countdown with respect to the spacecraft.

The last launch task will be the actual launch vehicle countdown. The countdown is divided into two activities: the precountdown and the terminal countdown. Both spacecraft will participate in the precountdown activities. Prior to the conclusion of these activities each subsystem of each spacecraft will have been checked for proper operation. At the conclusion of the precountdown activities a decision will be made as to whether flight No. 1 or No. 2 spacecraft will be launched.

6.11 Mission Operations Support

Mission operation support begins during the spacecraft engineering model assembly and test, when the orbital operations computer programs will be tested. Listed below are some additional tasks that will be performed by the operations personnel during the course of spacecraft testing:

a) 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 the various subsystem personnel.

b) Define and coordinate the implementation of the communications network between the central control at JPL and the DSIF stations, as well as secondary tracking stations in the STADAN network and downrange postlaunch tracking and data acquisition stations.

c) Define the engineering and experiment computer programs to be used at JPL for both quick-look and long-term data processing, including a definition of the expected and out-of-tolerance limits on major spacecraft and experiment telemetry items.

d) Define the real-time telemetry and communications requirements for the DSIF complex during the critical postlaunch and in-flight maneuvers.

e) Define and coordinate a data tape run from the flight equipment during the final test phases. This tape will include a simulation of all anticipated in-flight maneuvers as well as all conceivable spacecraft and experiment modes.

f) Generate detailed calibration data for all engineering items and those items considered crucial for the success of individual experiments. This data will be integrated on calibration cards to be used with quick-look data displays.

g) Generate a mission plan for each specific spacecraft which defines the operational requirements of the program.

7. PHASE IB IMPLEMENTATION PLANNING

The implementation planning for Phase IB consists of updating plans submitted during the Phase IB proposal and preparing additional Phase II planning documents.

The management plans to be updated and submitted during the eight-month Phase IB preliminary design phase include:

- Project Control Plan
- Safety Plan
- Facilities Plan
- Quality Assurance Plan
- Reliability Program Plan
- Configuration Control Plan
- Documentation Plan
- Procurement Plan

In addition, plans for the detailed implementation of the Phase II, development phases will be prepared and submitted including:

- Manufacturing Plan
- Integrated Test Plan
- Assembly and Checkout Plan
- Launch Operations Plan
- Magnetic Control Plan
- Contamination Control Plan
- Electromagnetic Control Plan
- Experiment Design Integration Plan

APPENDIX A

ASSEMBLY, TEST, AND LAUNCH OPERATIONS

This appendix contains the relatively detailed descriptions, in the form of tables and flow charts, of the assembly and test operations for both the 1969 and 1971 missions. Nine tables and flow charts are included, covering the following:

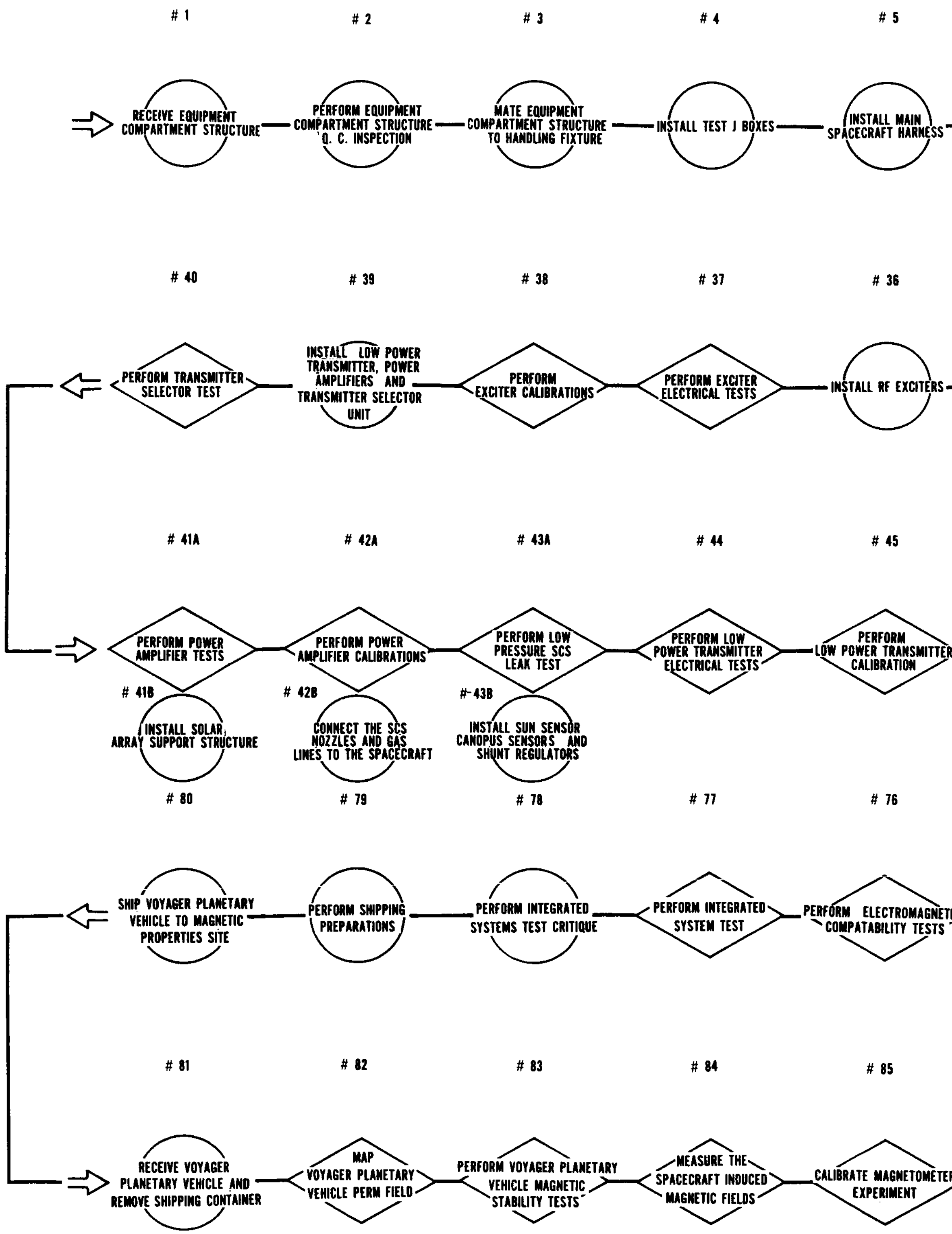
1969

1. Proof Test Model Assembly and Checkout
2. Proof Test Model Type Approval Testing
3. Flight Spacecraft Flight Approval Testing
4. Launch Operations

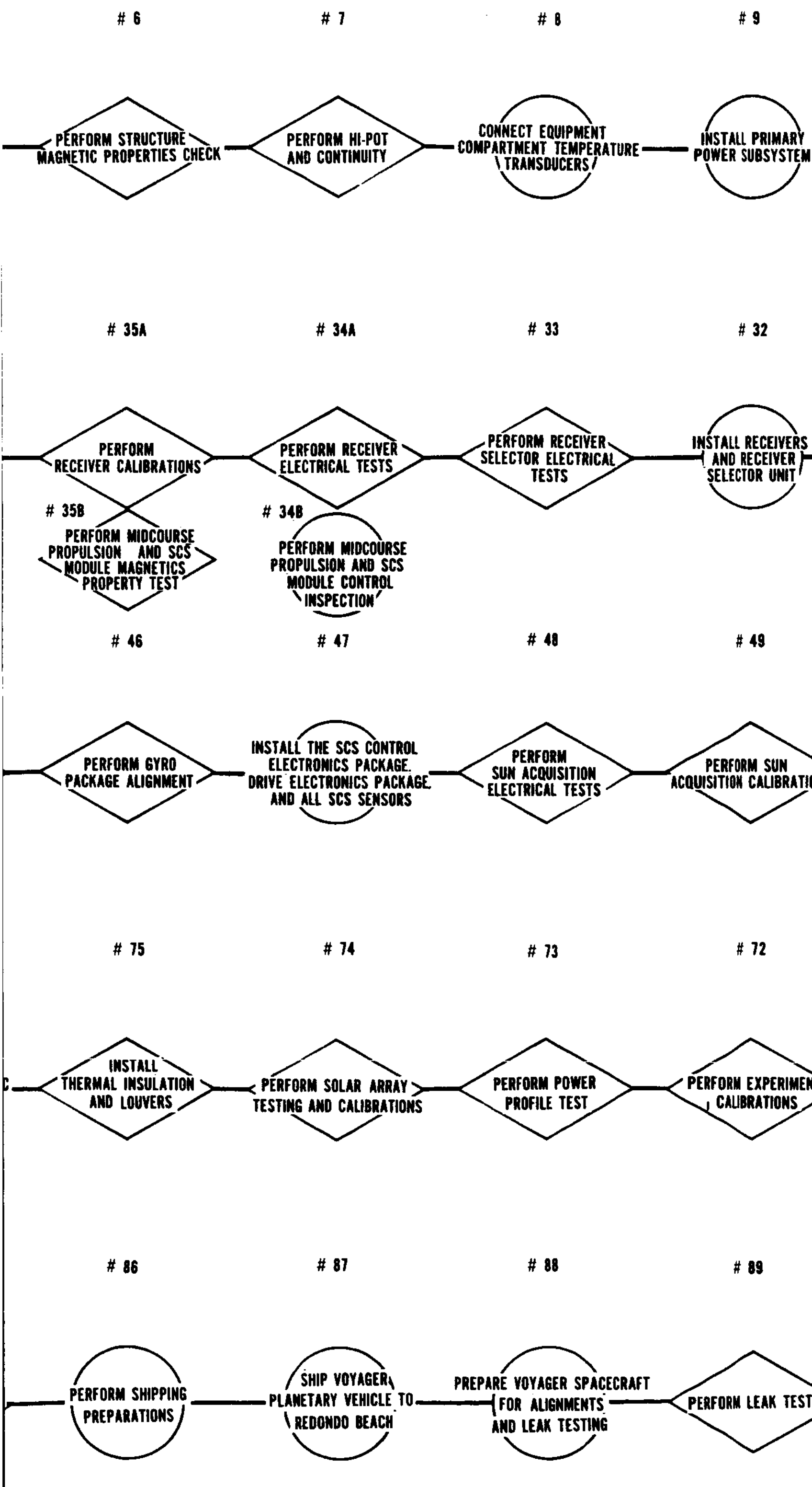
1971

1. Engineering Model Assembly and Checkout
2. Proof Test Model Assembly and Checkout
3. Proof Test Model Type Approval Testing
4. Flight Spacecraft Flight Approval Testing
5. Launch Operations

No table is supplied for the 1969 engineering model assembly and test since these activities are identical to the 1971 engineering model assembly and test. Similarly, the flight model assembly and test activities for both 1971 and 1969 missions are not recorded since they are identical to those of the proof test model assembly and test, with the exception of the moment of inertia test which will not be included during flight model assembly and test.



1
RECEIVE EQUIPMENT COMPARTMENT STRUCTURE
2
PERFORM EQUIPMENT COMPARTMENT STRUCTURE Q. C. INSPECTION
3
MATE EQUIPMENT COMPARTMENT STRUCTURE TO HANDLING FIXTURE
4
INSTALL TEST J BOXES
5
INSTALL MAIN SPACECRAFT HARNESS
40
PERFORM TRANSMITTER SELECTOR TEST
39
INSTALL LOW POWER TRANSMITTER, POWER AMPLIFIERS AND TRANSMITTER SELECTOR UNIT
38
PERFORM EXCITER CALIBRATIONS
37
PERFORM EXCITER ELECTRICAL TESTS
36
INSTALL RF EXCITERS
41A
PERFORM POWER AMPLIFIER TESTS
42A
PERFORM POWER AMPLIFIER CALIBRATIONS
43A
PERFORM LOW PRESSURE SCS LEAK TEST
44
PERFORM LOW POWER TRANSMITTER ELECTRICAL TESTS
45
PERFORM LOW POWER TRANSMITTER CALIBRATION
41B
INSTALL SOLAR ARRAY SUPPORT STRUCTURE
42B
CONNECT THE SCS NOZZLES AND GAS LINES TO THE SPACECRAFT
43B
INSTALL SUN SENSOR CANOPUS SENSORS AND SHUNT REGULATORS
80
SHIP VOYAGER PLANETARY VEHICLE TO MAGNETIC PROPERTIES SITE
79
PERFORM SHIPPING PREPARATIONS
78
PERFORM INTEGRATED SYSTEMS TEST CRITIQUE
77
PERFORM INTEGRATED SYSTEM TEST
76
PERFORM ELECTROMAGNETI
COMPATABILITY TESTS
81
RECEIVE VOYAGER PLANETARY VEHICLE AND REMOVE SHIPPING CONTAINER
82
MAP VOYAGER PLANETARY VEHICLE PERM FIELD
83
PERFORM VOYAGER PLANETARY VEHICLE MAGNETIC STABILITY TESTS
84
MEASURE THE SPACECRAFT INDUCED MAGNETIC FIELDS
85
CALIBRATE MAGNETOMETER EXPERIMENT



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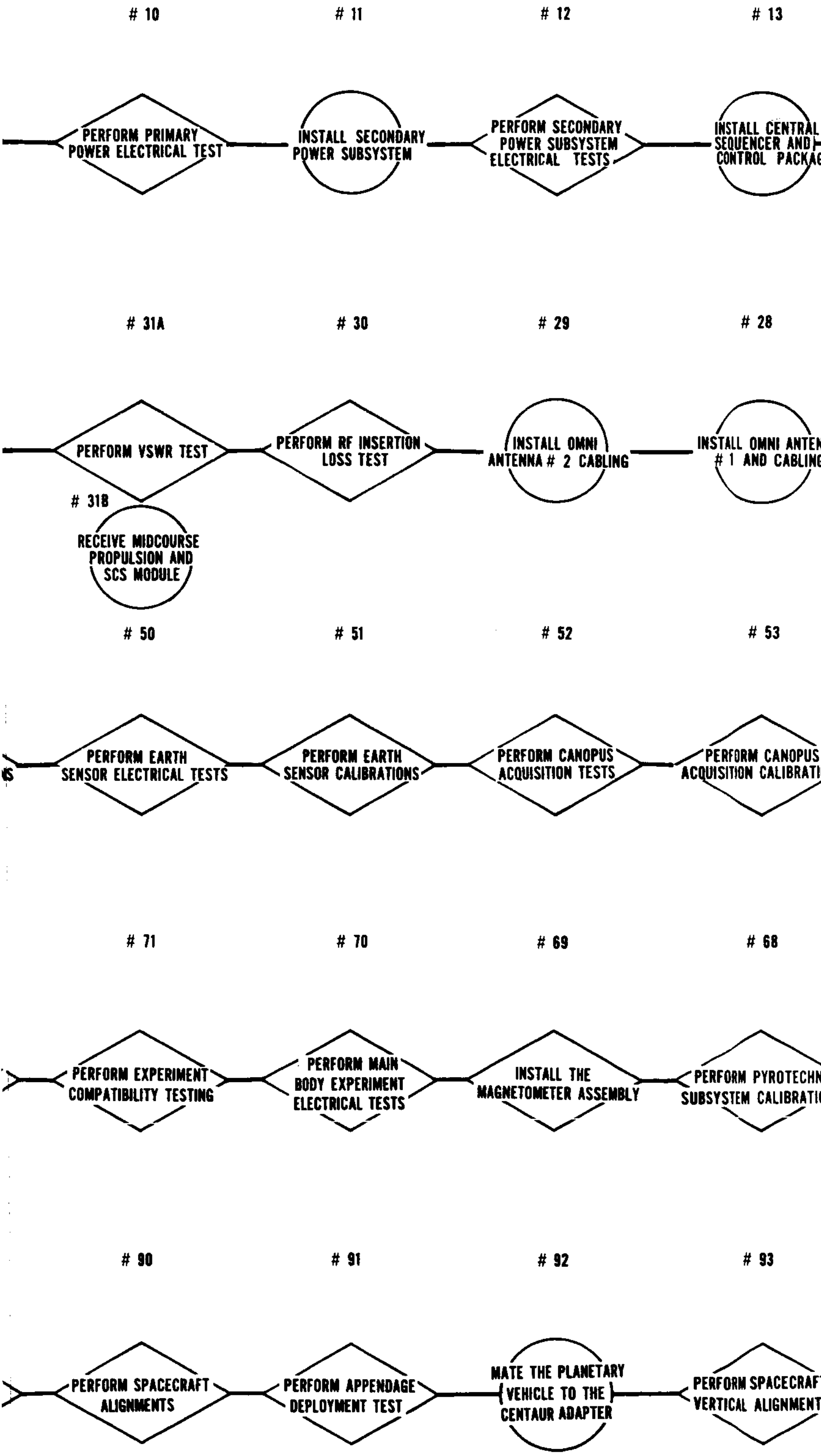

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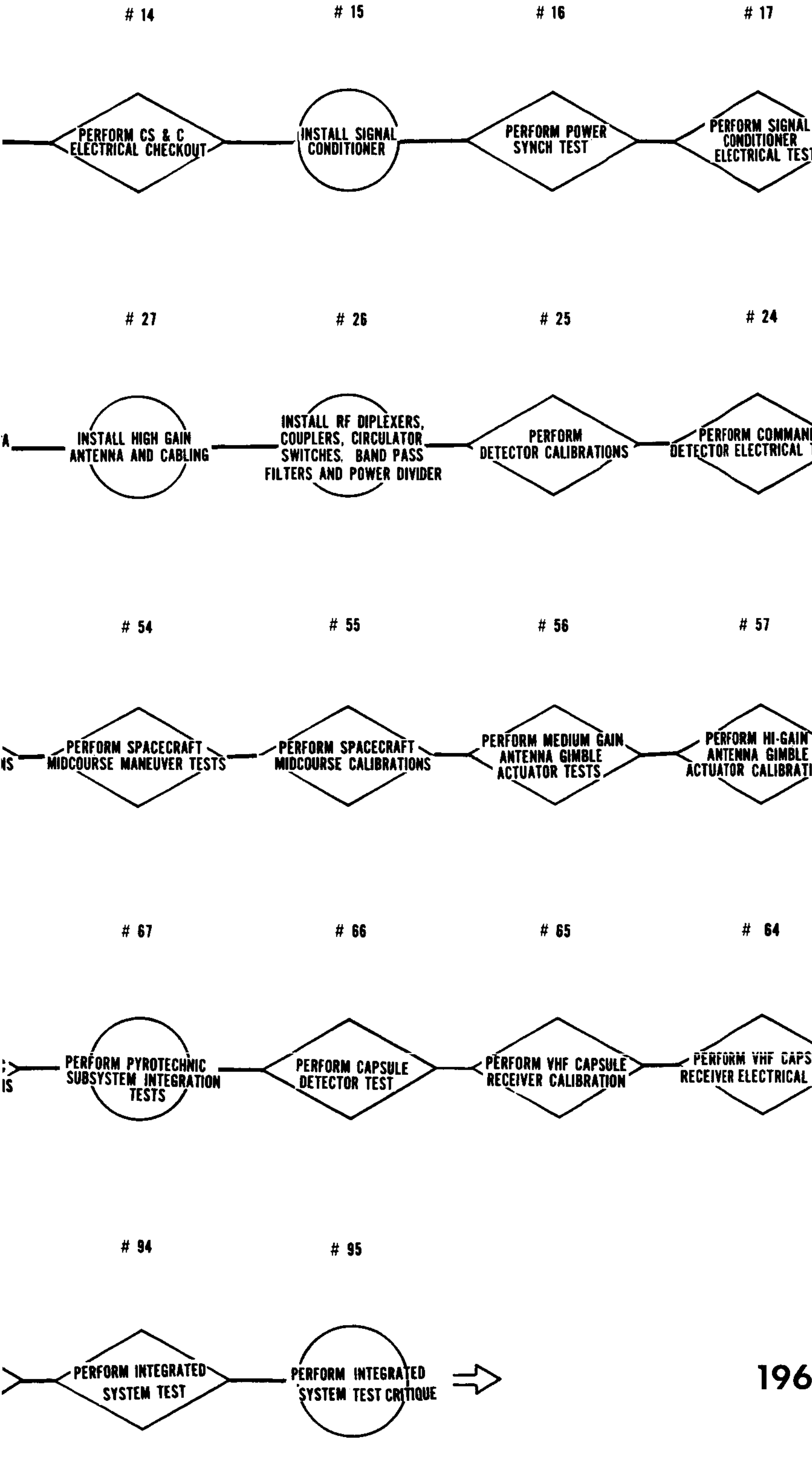

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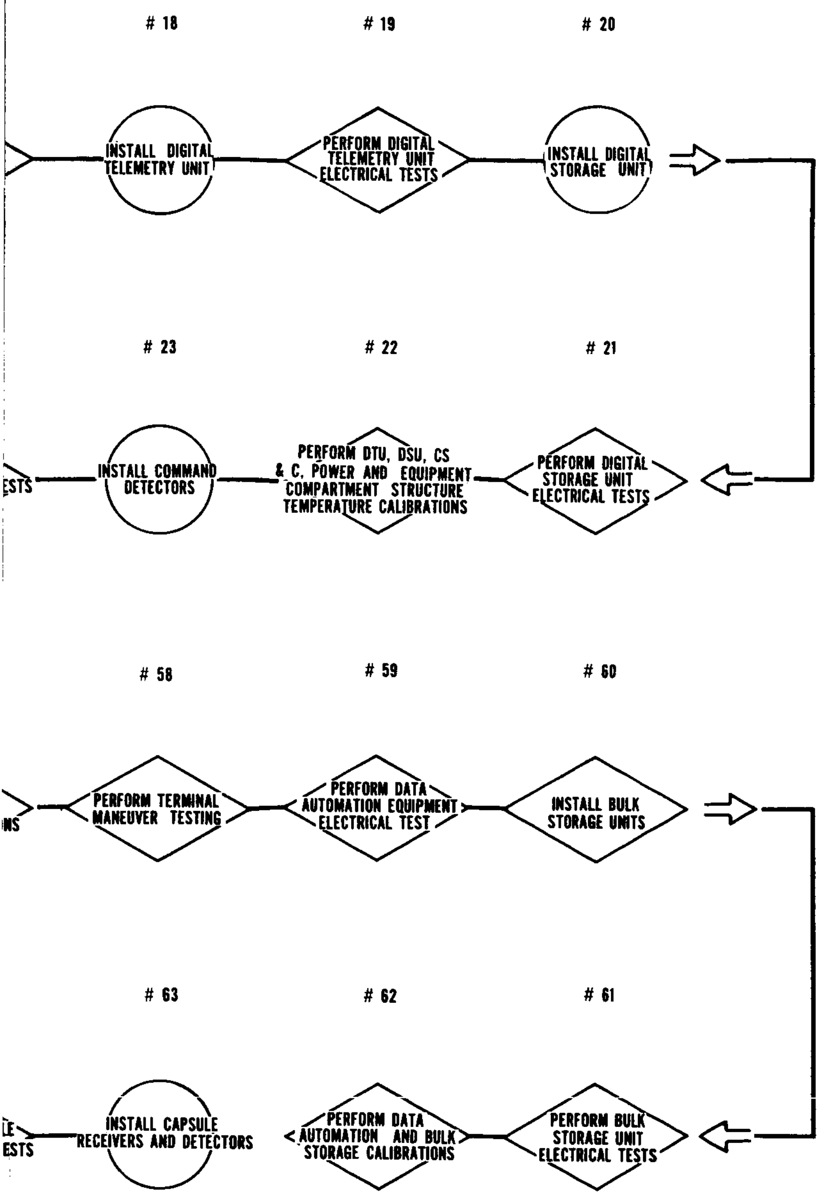

9 PROOF TEST MODEL S/C ASSEMBLY & TEST

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Functional Flow Drawing Title and No. **Proof Test Model Spacecraft Assembly and Checkout** Revision ______ Date ______ Approval ______ Page No. 1

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1A	Receive Equipment Compartment Structure The spacecraft equipment compartment structure will be received from Douglas Aircraft Co. in the following configuration: a. Solar array support structure not installed b. Main spacecraft harness not installed c. Thermal insulation not installed d. Thermal louvers not installed e. Propulsion system not installed f. Equipment compartment structure temperature transducers installed g. High-gain antenna and support structure not installed h. OMNI antenna and boom not installed i. Magnetometer and boom not installed j. TRW quality control buy-off will be performed at Douglas Aircraft Co.	Tools to uncrate structure	Equipment list	None
1B	Receive Systems Test Set EOSE	None	Equipment list	None
2A	Perform Equipment Compartment Structure Quality Control Inspection Quality control inspection is mainly for shipping damage as the equipment compartment structure will have been already bought off at Douglas Aircraft Co.	None	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
2B	**Start System Test Set EOSE Validation** The system test set EOSE will be validated for two reasons. a. To ensure that the EOSE has survived the shipping and handling operations b. To familiarize test crews with the EOSE	System test set validation sets	Procedures	None
3	**Mate Equipment Compartment Structure to Handling Fixture** a. Mate MOSE adapter to spacecraft structure b. Mate MOSE adapter and spacecraft to handling fixture	Handling sling, adapter handling fixture, Protective covers, hand tools	Procedures	None
4	**Install Test J Boxes** Install all electrical test J boxes to support the hi-pot and continuity test	Hand tools, torque wrench	Procedure	None
5	**Install Main Spacecraft Harness** Install main spacecraft electrical harness and connect to J boxes	Hand tools, torque wrench, handling sling	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
6	**Perform Structure Magnetic Properties Check** The equipment compartment magnetic properties check will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the magnetic field of the equipment compartment structure mounted in handling fixture c. Analyze all variations between readings and repeat if necessary	Magnetic measuring equipment, handling fixture, protective covers, handling slings	Procedure	Area in building free of large magnetic fields
7	**Perform Hi-Pot and Continuity** This is to be accomplished using a Huges FACT machine or equivalent. Whereever possible the test will be run end to end through all J boxes	Huges FACT machine or equivalent, cable adapters, FACT machine programs	Procedure	None
8	**Connect Equipment Compartment Temperature Transducers** Solder all temperature transducers to main spacecraft harness	Soldering iron, solder, insulation	None	None
9	**Install Primary Power Subsystem**	Hand tools, torque wrench	Procedure	None

Functional Flow Drawing Title and No. Proof Test **Model Spacecraft Assembly and Checkout** Revision ______ Date ______ Approval ______ Page No. 4

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
10	**Perform Primary Power Electrical Test** The primary power subsystem consists of the following items: batteries, power control unit, shunt regulators, and battery boost regulator. The subsystem electrical tests will be performed as follows: a. Integrate power OSE b. Perform bus open circuit checks using the external power mode c. Perform bus open circuit checks using the spacecraft batteries d. Perform bus open circuit checks using solar array simulated power e. Load electrical bus using dummy loads and electrically test the power control unit and shunt regulators using the spacecraft batteries and the solar array simulator. Commands will be simulated by using an external power supply that will be part of one of the load boxes f. Remove loads from bus and connect boost regulator g. Power boost regulator from external power and measure output current h. Load boost regulator output and measure the input and output voltage and current. Also note that noise on the output lines is within acceptable limits. Note: All loads are to be applied at the users side of the harness.	Voltmeters, ammeters, oscilloscope, power supply, power EOSE, series fuse boxes, in-line test connectors	Procedure	None
11	**Install Secondary Power Subsystem**	Hand tools, torque wrench	Procedure	None

Functional Flow
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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
12	**Perform Secondary Power Subsystem Electrical Tests** **The secondary power subsystem consists of the following items:** a. 4.1 kc — 1ϕ inverter b. 820 cps — 2ϕ inverter c. 410 cps — 1ϕ inverter **The secondary power subsystem test will be performed as follows:** **a.** Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the 4.1 kc primary power input **b.** Connect 4.1-kc inverter to the spacecraft main harness **c.** Check 4.1-kc inverter open circuit voltage by powering the bus on external power **d.** Load 4.1-kc inverter using dummy loads and check output current and voltage **e.** Repeat steps a through c for the 820 and 410 cps inverters Note: All load boxes are to be applied at the users side of the harness	**Ammeters, voltmeters, oscilloscope, power EOSE, series fuse boxes, in-line test connectors**	**Procedure**	**None**
13	**Install Central Sequencer and Control Package**	**Hand tools**	**None**	**None**

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
14	**Perform CS and C Electrical Checkout** **The central sequencer and control unit electrical check-out will be performed as follows:** **a.** Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the CS and C power input connector **b.** Connect the CS and C to the spacecraft harness and measure the voltage and current drawn by the CS and C. Also note that noise and transients are at acceptable levels **c.** Connect command detector format generator to the CS and C at the detector side of the spacecraft harness **d.** Check all of the power control unit commands as follows: 1) Open all command lines from the CS and C at the PCU side of the spacecraft harness 2) Transmit all PCU commands via the command format generator 3) Observe the open circuit command signal voltage at the PCU 4) Close the command lines to the PCU and retransmit the PCU commands via the command format generators 5) Monitor the command voltage and current at the PCU 6) Observe command signal lines and note that noise and transients are at acceptable levels 7) Observe that the PCU reacts properly to the CS and C commands	Command format generators, voltmeters, oscilloscope, ammeter, power EOSE, series fuse boxes, in-line test connectors, command matrix monitor	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	e. Check the open circuit voltage of the remaining discrets command lines from the CS and C at the side of the spacecraft harness. Note: The noise and transient levels on each of the remaining command signal lines will be checked during the electrical integration of the remaining subsystems f. Transmit each quantitative command from the format generator and observe that each command was properly received by observing the command matrix monitor g. Measure the amplitude and frequency of the down link PN subcarrier. h. Measure the amplitude and frequency of all timing signals from the CS and C			
15	<u>Install Signal Conditioner</u>	Hand tools, torque wrench	Procedure	None
16	<u>Perform Power Synch Test</u> The power synch tests will be performed in the following manner: a. Apply external power to the spacecraft and observe the open circuit frequency, rise time, fall time pulse width, and amplitude of each synch pulse from the CS and C to the boost regulator and each inverter b. Connect the synch pulse to the boost regulator and observe the frequency, rise time, fall time, pulse width, and amplitude of each pulse	Oscilloscope, in-line test connector	Procedure	None

Functional Flow Drawing Title and No. Proof Test **Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 8

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Observe the boost regulator 50 vdc output noise d. Note that noise and transients are within acceptable limits e. Repeat the above steps for each inverter			
17	**Perform Signal Conditioner Electrical Test** a. Turn on external power to spacecraft and check that voltage exists where it should and no voltage exists on the remaining pins at the signal conditioner power input connector b. Connect signal conditioner to secondary power subsystem c. Measure voltage and current drawn by signal conditioner from the secondary power subsystem	Voltmeter, ammeter, series fuse boxes	Procedure	None
18	**Install Digital Telemetry Unit**	Hand tools, torque wrench	Procedure	None
19	**Perform Digital Telemetry Unit Electrical Tests** The DTU electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins at the DTU power input connector b. Connect the DTU to the 4.1-kc inverter and measure the voltage and current drawn by the DTU. Also note that noise and transients are at acceptable levels	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator, analog word simulator	Procedure	None

Functional Flow
Drawing Title and No. Proof Test **Model Spacecraft** **Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 9

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. **Measure command line signal voltage and current drawn for each commanded bit rate, format and mode of operation. Also note that noise and transients are acceptable levels**			
	d. **Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all timing pulses at the users side of the harness. This is to be done for each bit rate**			
	e. **Measure the frequency, pulse amplitide, rise time, fall time, and the pulse width of all shift pulses at the users side of the harness. This is to be done for each bit rate**			
	f. **Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all synch pulses at the users side of the harness**			
	g. **Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all inhibit pulses at the users side of the harness. This is to be done for each bit rate**			
	h. **Check ID words corresponding to all bit rates and all formats using the telemetry data display EOSE**			
	i. **Loop check all analog words by applying a DC voltage at the senders side of the harness and reading out the decimal word at the telemetry data display EOSE**			
	j. **Loop check all digital words by applying a digital signal at the senders side of the harness and reading out the decimal word at the telemetry display EOSE**			
	Note: Noise, transient and cross talk measurements will be conducted for items c through g			
	k. **Measure the subcarrier frequency and modulation index of the down link baseband signal**			

Functional Flow Drawing Title and No. Proof Test **Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 10

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
20	<u>Install Digital Storage Unit</u>	Hand tools, torque wrenches	Procedure	None
21	<u>Perform Digital Storage Unit Electrical Tests</u> The digital storage unit electrical testing will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the DSU power connector b. Connect the DSU to the spacecraft harness and measure the voltage and current drawn by the DSU. Also note that noise and transients are at acceptable levels c. Measure all command line voltages and currents for each DSU command. Also note that noise and transients are at acceptable levels d. Measure the rise time, fall time, amplitude, and pulse duration of the DSU input data signal at the DSU for each bit rate e. Measure the rise time, fall time, amplitude, and pulse duration of the DSU data output signal at the DTU during memory readout f. Measure the rise time, fall time, amplitude, and pulse duration of the DSU index pulse at the DTU Note: Noise, transient and cross talk measurements will be conducted for items d through f	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator, analog word format generator		

Functional Flow
Drawing Title and No. Proof Test **Model Spacecraft Assembly and Checkout** Revision ______ Date ______ Approval ______ Page No. 11

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
22	**<u>Perform DTU, DSU, CS and C, Power and Equipment Compartment Structure Temperature Calibrations</u>** **These calibrations will be handled as follows:** **a.** DTU temperature calibrations will be accomplished by replacing the transducer with precision resistors and noting the word value at the telemetry data display EOSE for each resistor value. The word values together with the factory transducer curves complete the calibration. Next, these parameters will be incorporated into the computer programs. The DTU analog to digital converter reference words are to be simply noted and recorded **b.** DSU temperature calibrations will be accomplished as in Task 22. a. 1. **c.** CS and C temperature calibrations will be accomplished as in Task 22. a. 1 **d.** Primary power calibrations will be accomplished by varying the load current and line voltage and monitoring the voltage and current with meters. The telemetry word values for each voltage and current will be recorded. These parameters will be inserted into the computer programs. Secondary power calibrations will be accomplished in the same manner as the primary power calibrations **e.** Equipment compartment structure temperature calibrations will be accomplished as in Task 22. a. 1 **f.** At the telemetry data display EOSE, verify that each command sent during items a through c above indicates the proper telemetry word value	Voltmeter, ammeter, decade resistance box, data center computer programs, telemetry data display EOSE, power supply, power EOSE, series fuse boxes, in-line test connectors	Procedure	None

Functional Flow Drawing Title and No. Proof Test **Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 12

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
23	<u>Install Command Detectors</u>	Hand tools, torque wrench		
24	<u>Perform Command Detector Electrical Tests</u> The command detector electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the command detector connectors b. Connect the detectors to the spacecraft harness and measure the secondary power supply voltage and current drawn by the detectors. Also note that noise and transients are at acceptable levels c. Measure the detector, output peak to peak amplitude at the CS and C input in the presence of a simulated receiver signal (command encoder EOSE) d. Measure the bit synch rise time, fall time, pulse width, and amplitude e. Check that each command processor can be addressed only one separate address. f. Check each detector synch lock operation with the command encoder g. Transmit each discrete command via the command encoder and observe that each command was received by observing the command matrix monitor h. Repeat the above for the redundant detector. Note that quantitative commands from each detector will be monitored during stabilization and control subsystem checkout	Power EOSE, voltmeter, ammeter, series fuse box, in-line test connector, command matrix, monitor command encoder	Procedure	None

Functional Flow Proof Test **Model Spacecraft**
Drawing Title and No. **Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. **13**

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
25	**Perform Detector Calibrations** **The detector temperature calibrations will be accomplished as in task 22.a.1.**	**Power EOSE, command encoder, resistor decade box, operational data center command, matrix monitor, in-line test connector**	**Procedure**	**None**
26	**Install RF Diplexers, Couplers, Circulator Switches Band Pass Filters and Power Dividers**	**Hand tools, torque wrenches**	**Procedure**	**None**
27	**Install High-Gain Antenna and Cabling** **This task is broken up into several subtasks as follows:** **a. Install high-gain antenna** **b. Connect, route, and clamp cabling** **c. Articulate antenna and check for cable chaffing and clearance** **d. Latch antenna in place**	**Hand tools, torque wrench, antenna drive EOSE**	**Procedure**	**None**
28	Install OMNI Antenna No. 1 and Cabling **This task is broken up into several subtasks as follows:** **a. Install medium-gain antenna** **b. Connect, route, and clamp cabling** **c. Articulate antenna and check for cable chaffing and clearance** **d. Latch antenna in place**	**Hand tools, torque wrench, antenna drive EOSE**		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
29	Install OMNI Antenna No. 2 and Cabling **This task is broken up into several subtasks as follows:** a. Install OMNI antenna to OMNI antenna boom b. **Install antenna and boom to spacecraft** c. **Connect, route, and clamp cabling** d. **Deploy and latch boom observing cable clearance and that no chaffing takes place** e. **Latch antenna boom in place**	**Hand tools, torque wrench**	**Procedure**	**None**
30	**Perform RF Insertion Loss Test** **The RF insertion loss determination will take place as follows:** a. **Connect the diplexers, couplers, bandpass filters, power monitors, and circulator switches to the RF cable harness system** b. **Measure the insertion loss between the receivers and the high-gain antenna** c. **Measure the insertion loss between the receivers** and OMNI antenna No. 1 d. **Measure the insertion loss between the receivers** and OMNI antenna No. 2 e. **Measure the insertion loss between the power amplifiers and the high-gain antenna** f. **Measure the insertion loss between the power** amplifiers and OMNI antenna No. 1 g. **Measure the insertion loss between the power** amplifiers and OMNI antenna No. 2 h. **Measure the insertion loss between the exciters and the high-gain antenna** i. **Measure the insertion loss between the exciters** and OMNI antenna No. 1 j. **Measure the insertion loss between the exciters** and OMNI antenna No. 2 k. **Measure the insertion loss between the exciters and the power amplifiers**	**RF converter, adapters, RF generator, RF power meter**	**Procedure**	**None**

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
31A	<u>Perform VSWR Tests</u> The VSWR tests will be performed as follows: a. After the insertion loss test has been completed, connect the high-gain and omni antennas to the RF cable harness b. Measure the VSWR between the receivers and the high-gain antenna c. Measure the VSWR between the receivers and OMNI antenna No. 1 d. Measure the VSWR between the receivers and OMNI antenna No. 2 e. Measure the VSWR between the power amplifiers and the high-gain antenna f. Measure the VSWR between the power amplifiers and OMNI antenna No. 1 g. Measure the VSWR between the power amplifier and OMNI antenna No. 2 h. Measure the VSWR between the exciters and the high-gain antenna i. Measure the VSWR between the exciters and OMNI antenna No. 1 j. Measure the VSWR between the exciters and OMNI antenna No. 2	RF connecter adapters, RF generator, RF couplers, VSWR meter, notch filters	Procedure	None
31B	<u>Receive Midcourse Propulsion and SCS Module</u> The midcourse propulsion and SCS module will be received from Douglas consisting of the following: a. Monopropellant engine and control valves b. Monopropellant engine feed system c. Monopropellant engine pressurization system d. Stabilization and control subsystem gas system e. Jet vane assemply installed in engine Note: Final TRW Quality Control buy-off will be performed at Douglas			

Functional Flow Proof Test **Model Spacecraft**
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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
32	**Install Receivers and Receiver Selector Unit**	Hand tools, torque wrench		
33	**Perform Receiver Selector Electrical Tests** The receiver electrical tests will be performed as follows: a. Apply external power to the spacecraft and check that voltage exists where it should and that no voltage exists at the remaining pins of the receiver selector connectors. b. Connect the receiver selector to the spacecraft harness and measure the voltage and current drawn by the selector. Also note that noise and transients are at acceptable levels c. Connect the receiver signal simulator to the receiver selector d. Simulate each receiver present signal and observe that the proper receiver is selected e. Simulate all combinations of the three receiver present signals and observe that the proper receiver is selected f. Simulate the loss of sun-Canopus and observe that receiver No. 1 is selected	Power EOSE, voltmeter, ammeter, oscilloscope, receiver, selector, simulator	Procedure	None
34A	**Perform Receiver Electrical Tests** The receiver electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector b. Connect each receiver to the spacecraft harness and measure the voltage and current drawn by each receiver. Note that noise and transients are within acceptable levels	RF EOSE, command encoder, command matrix monitor, voltmeter, ammeter, power EOSE, series fuse boxes, in-line test connectors	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. **Measure the modulation index of test transmitter output while it is being modulated with the command encoder and determine that it is within specification. This is to be done with and without the ranging signal**			
	d. **Connect the receiver to a strong hardline signal from the RF EOSE (-110 dbm) and acquire**			
	e. **Modulate the test transmitter (RF EOSE) with the command encoder and note that commands can be received and properly acted upon by the CS and C using each receiver through each antenna. This is to be accomplished by observing the command matrix monitor and by monitoring the appropriate telemetry word. Verify that the airborne receiver will acquire while the ground transmitter is being ramped at the maximum specified rate for given signal strengths**			
	f. **Determine the signal strength at which the receiver thresholds or drops out of lock**			
	g. **Verify that the receiver will stay acquired for the maximum specified ramp rate for given signal strengths**			
	h. **Repeat above for the redundant receiver**			
34B	**Perform Midcourse Propulsion and SCS Module Control Inspection** **Quality control inspection is mainly for shipping damage as the module has previously been bought off at Douglas Aircraft Co. by TRW personnel**	**None**	**Procedure**	**None**

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
35A	**Perform Receiver Calibrations** The receiver calibrations will be performed as follows: a. Receiver temperature calibrations will be accomplished as in Task 22. a. 1 b. The airborne receivers will be dropped in and out of lock by removing the test transmitter signal and noting that the telemetry indication is proper c. A precisely known signal level is fed into a precision step attenuator. A known signal strength can now be calculated for each attenuator setting. Each power level will be correlated with telemetry output d. After the receivers have been required by the test transmitter, the test transmitter frequency is varied and the loop stress telemetry output noted. All of the above parameters will be inserted into the computer programs	RF EOSE, command encoder, power EOSE, RF attenuators, calorimeter, data center, in-line test connector	Procedure	None
35B	**Perform Midcourse Propulsion and SCS Module Magnetics Property Test** The midcourse propulsion and SCS module magnetic properties check will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the magnetic field of the bus structure mounted in handling fixture c. Analyze all variations between readings and repeat if necessary	Magnetic measuring equipment, handling fixture, protective covers, handling slings	Procedure	Area in building free of large magnetic fields

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Assembly and Checkout   Revision ____ Date ____ Approval ____

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
36	<u>Install RF Exciters</u>	Hand tools, torque wrench	Procedure	None
37	<u>Perform Exciter Electrical Tests</u> The exciter electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector b. Connect the exciter to the spacecraft harness and measure the voltage and current drawn by the driver. Note that noise and transients are within acceptable limits c. Measure the rise time, fall time, and amplitude of the exciter modulation for each bit rate d. Remove modulation and measure the exciter RF power and frequency at the exciter output e. Measure the exciter modulation index with and without the ranging signal present f. Investigate driver output for spurious harmonics using a spectrum analyzer g. Connect the exciter output of the RF harness and ascertain that data can be received by the ground receiver (RF EOSE) through each antenna via air link h. Command the exciter to the coherent mode of operation and observe that driver output is 240/221 times the frequency of the ground transmitter i. Repeat above for the redundant exciter	RF EOSE, command encoder, power EOSE voltmeter, ammeter, series fuse boxes, in-line test connector, spectrum analyzer	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
38A	**Perform Exciter Calibrations** The exciter calibrations will be performed as follows: a. Exciter temperature calibration to be performed as in Task No. 22.a.1 b. Coherent/noncoherent mode to be performed by commanding the driver to the coherent and noncoherent modes of operation and noting that the proper telemetry word value exists	Power EOSE, RF EOSE, decade resistance box, series fuse boxes	Procedure	None
39	**Install Low Power Transmitter, Power Amplifiers, and Transmitter Selector Unit**	Hand tools, torque wrench		
40	**Perform Transmitter Selector Test** The transmitter selector electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins on each selector connector b. Connect the transmitter selector to the spacecraft harness and measure the voltage and current drawn from the secondary power supply subsystem. Note that noise and transients are within acceptable levels c. Simulate the appropriate transmitter modes via ground commands and CS and C back-up commands and ascertain that the proper transmitter was selected by monitoring the selector outputs.			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
41A	<u>Perform Power Amplifier Tests</u> The power amplifier tests will be performed as follows: a. Turn on external power to the spacecraft and command the power amplifier on b. Connect dummy loads to the power amplifier output c. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector d. Connect the power amplifier power to the spacecraft harness and measure the voltages and currents drawn by power amplifiers. Note that noise and transients are within acceptable levels e. Measure the power amplifier RF output power f. Measure the power amplifier modulation index with and without the ranging signal g. Measure the power amplifier output for spurious harmonics using a spectrum analyzer h. Connect the power amplifier to the RF cable harness i. Observe that telemetry can be received by the ground receiver (RF EOSE) from each antenna via air link j. Repeat for the redundant power amplifier	Power meter, NF-112 analyzer, power EOSE, RF EOSE, series fuse box, in-line test connectors	Procedure	None
41B	<u>Install Solar Array Support Structure</u>	Hand tools, torque wrenches		
42A	<u>Perform Power Amplifier Calibrations</u> The power amplifiers calibrations will be performed as follows: a. Temperature calibration will be performed as in Task 22.a.1 b. Step attenuators will be placed in the RF lines and the attenuator power measured. The measured power for each attenuator step is correlated with the telemetry output words. These parameters will be	Step attenuator, decade resistor box, Power EOSE RF EOSE, power meter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	inserted into the computer programs			
42B	Connect the SCS Nozzles and Gas Lines to the Spacecraft The SCS nozzles and gas lines will be connected to the spacecraft SCS pneumatics system	Hand tools	Procedure	None
43A	Perform Low Pressure SCS Leak Test The purpose of the low pressure leak test is to ascertain that the SCS pneumatic system leak rate is grossly within specification	Leak test console		
43B	Install Sun Sensor, Canopus Sensors and Shunt Regulators	Hand tools	Procedure	None
44	Perform Low Power Transmitter Electrical Tests a. Turn on external power to the spacecraft and command the low power transmitter on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins c. Connect the low power transmitter to the spacecraft harness and measure the voltage and current drawn from the secondary power subsystem. Note that noise and transients are within acceptable limits d. Measure the low power transmitter output and frequency e. Measure the low power transmitter output for spurious harmonics using a spectrum analyzer f. Measure the low power transmitter output modulation index g. Connect the low power transmitter to the RF cable harness h. Observe that telemetry can be received by the ground receiver through each antenna via air link	Voltmeter, ammeter, RF power meter, NF-112 analyzer, oscilloscope series fuse box, spectrum analyzer, oscilloscope, RF frequency counter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
45	<u>Perform Low Power Transmitter Calibration</u> The low power transmitter calibration will be performed as follows: a. Temperature calibration will be performed as in Task 22. a. 1 b. Step attenuators will be placed in the RF lines and the attenuator power measured. The measured power for each attenuator step is correlated with the telemetry output words. These parameters will be inserted into the computer programs	Step attenuator, decade resistor box, power EOSE RF EOSE, power meter	Procedure	None
46	<u>Perform Gyro Package Alignment</u> The gyro package alignment is performed so that the gyro scale factors can be determined as part of the SCS testing phase	Gyro alignment set	Procedure	None
47	<u>Install the SCS Control Electronics Package, Drive Electronics Package and All SCS Sensors</u> The above items will be installed in the spacecraft in preparation for the SCS testing phase	Hand tools, torque wrenches		
48	<u>Perform Sun Acquisition Electrical Tests</u> The sun acquisition electrical tests will be performed as follows: a. Apply external power to the spacecraft and command the gyros to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector of the gyro package c. 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). Note that noise and transients on these lines are within acceptable levels	SCS EOSE, power EOSE, voltmeter, ammeter oscilloscope jet vane, angle MOSE, in-line test connector, series fuse box	Procedure	Tilt fixture should experience zero floor vibrations

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. 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 e. Connect the control signal electronics package to the spacecraft harness and measure the voltage and currents drawn by the control signal electronics package. Note that noise and transients on these lines are within acceptable levels f. Torque the tilt fixture in the +yaw direction at a known rate and measure the yaw gyro output signal amplitude. Note that the polarity is correct g. Torque the tilt fixture in the -yaw direction at a known rate and measure the yaw gyro output signal. Note that the polarity is correct h. Repeat Step f for the pitch and roll gyros i. With the spacecraft absolutely still, measure the noise amplitude on each gyro output line j. Increase the rate in each axis in each direction and note that the proper gas valve is actuated k. Determine the threshold rates in each axis which will just barely cause the gas valves to actuate l. 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 m. Connect the sun sensors to the spacecraft harness n. Attach the sun sensor stimulus to each sun sensor o. Connect a voltmeter in place of each gas valve solenoid p. Manually actuate separation switches and check that the sun acquisition mode has started q. Transmit the back-up command for starting the sun acquisition sequence r. Illuminate each sun sensor and check that voltage exists at each valve interface			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	s. Connect each valve to the spacecraft harness t. 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 u. Observe that when each sun sensor is stimulated the proper valve is opened v. Observe that when all five sun sensor elements are illuminated, no valves are actuated			
49	Perform Sun Acquisition Calibrations The sun acquisition calibration will be performed as follows: a. The sun intensity signals will be simulated by replacing the sun sensor with a signal generator. As the voltage is varied, the telemetry word value is recorded for each sun sensor. The laboratory curves for each sun sensor (intensity versus voltage out) together with the digital word values will be inserted into the computer program b. The valve actuation signals will be calibrated by actuating each valve and noting the telemetry word values c. Control signal electronics package temperature calibration will be performed as per Task 22.a.1 d. Sun sensor temperatures calibration will be performed as per Task 22.2.1 e. The gyro temperature will be calibrated as per Task 22.a.1 f. Gyro on/off calibrations will be performed by commanding them on and then off and the telemetry word value recorded	Resistance decade box, Power EOSE, SCS EOSE, series fuse boxes, signal generator, in-line test connector	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	g. Gyro pick-off outputs will be replaced with a signal generator. As the signal generator amplitude is varied, the telemetry word value is monitored. These parameters together with the laboratory bench test data (rate versus output voltage) will be inserted into the computer programs			
50	<u>Perform Earth Sensor Electrical Tests</u> The earth sensor electrical tests will be performed as follows: a. Turn on external power to the spacecraft and command the earth sensor to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the earth sensor connector c. 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 d. Darken the earth sensor appeture and measure the signal output amplitude. Note that noise and transients are within acceptable levels e. Attach the earth sensor stimulus to the earth sensor f. Illuminate the earth sensor and measure the sensor output signal amplitude. Note that noise and transients are within acceptable levels g. 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	SCS EOSE, power EOSE, earth sensor stimulus, voltmeter, ammeter, oscilloscope, series fuse box		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
51	<u>Perform Earth Sensor Calibrations</u> The earth sensor calibrations will be performed as follows: a. The earth sensor will be replaced by a suitable signal generator. As the signal generator 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 b. The earth sensor temperature calibration will be performed as in Task 22.a.1	Signal generator, in-line connector, voltmeter, power EOSE, data center		
52	<u>Perform Canopus Acquisition Tests</u> The Canopus acquisition electrical tests will be performed as follows: a. Turn off external power to the spacecraft and command the Canopus sensor to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the Canopus sensor connector c. 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 d. Attach Canopus sensor stimulus to the Canopus sensor e. Illuminate each half of the Canopus sensor field of view and note that the proper valves actuate when each half is illuminated f. 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	Voltmeter, ammeter, oscilloscope power EOSE, SCS EOSE	Procedure	None

Operation No	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	g. Illuminate the center of the Canopus sensor field of view and note that no valves are actuated h. Investigate the Canopus sensor signal output lines for out-of-tolerance transient and noise conditions when the center of the Canopus sensor is illuminated i. Command the spacecraft into the roll search mode and observe that the proper roll valves are actuated j. Remove Canopus sensor illumination and observe that the SCS subsystem goes into the roll search mode k. Note that the airborne receivers switch to the OMNI antenna when the Canopus illumination is removed			
53	<u>Perform Canopus Acquisition Calibrations</u> The Canopus acquisition calibrations will be performed as follows: a. The Canopus sensor will be replaced by a suitable signal generator. As the generator 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	Signal generator, resistor decode box, power EOSE, data center	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. The Canopus sensor intensity signal will be performed as in Step a above c. The Canopus sensor temperature calibrations will be performed as in Task 22.a.1			
54	<u>Perform Spacecraft Midcourse Maneuver Tests</u> The spacecraft maneuver testing will be accomplished as follows: a. Enter the roll turn and polarity information into the command detector b. Execute the roll turn command, measure and time the gyro output and input signals and note that noise and transients are within acceptance levels c. Note that the proper gas valves are actuated while the gyro is being torqued d. Repeat Steps b and c for the opposite polarity turn e. Repeat Steps a through d for the pitch turn f. Load velocity increment information into the command detector to activate jet vane control noting that the proper voltage amplitudes exist at each jet vane actuator connector g. Repeat Step f for the opposite polarity h. Connect the midcourse motor jet vanes to the spacecraft harness re-insert the velocity increment and measure the voltage and current drawn by each jet vane actuator. Note that noise and transients on these lines are within acceptable limits i. Measure the jet vane angle with respect to the sun-line j. Repeat Step i for the opposite polarity velocity increment	ECS EOSE, power EOSE, voltmeter, ammeter, oscilloscope, jet vane, angle MOSE, in-line test connector, series fuse box	Procedure	Tilt fixture should experience zero floor vibrations

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	k. Enter midcourse motor burn duration information into the command detector and measure the mid-course motor ignitor firing voltage and turnoff voltage and the time duration between the turn on signal and turn off signal			
55	<u>Perform Spacecraft Midcourse Calibrations</u> The spacecraft maneuver calibrations will be performed as follows: a. Jet vane actuator temperature calibrations are to be performed as per Task 22. a. 1 b. Jet vane angle calibartions are to be performed by turning the jet vanes to known angles and recording the telemetry word values. These parameters are then inserted into the computer program	Power EOSE, SCS EOSE angle gauges decade re-sistor box in-line test connector	Procedure	None
56	<u>Perform High-Gain Antenna Gimble Actuator Tests</u> The gimble actuator tests will be performed as follows: a. Turn on external power to the spacecraft and command the antenna to slew b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the gimble actuator connectors c. Measure the drive signal amplitude d. Repeat Steps a, b, and c for the remaining gimble axis e. Connect the gimble actuators to the harness and command the gimble to slew	Voltmeter, ammeter, power EOSE, command EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	f. Measure the voltage amplitude and current drawn by the drive electronics from the secondary power subsystem while the gimble is being slewed noting that noise and transient conditions are within specification g. Repeat Step f for each gimble in each direction h. Observe that the gimble slews at the proper rate for each direction			
57	<u>Perform High Gain Antenna Gimble Actuator Calibrations</u> The actuator calibrations will be performed as follows: a. The actuator temperature calibrations will be performed as per Task 22.a.1 b. Gimble angle calibrations will be performed by slewing each gimble to a known angle and observing and recording the telemetry word values. These parameters are then inserted into the computer program	Resistor decade box, gimble angle indicator power EOSE, command EOSE, data center	Procedure	None
58	<u>Perform Terminal Maneuver Testing</u> The terminal maneuvers are accomplished in the same manner as the midcourse maneuvers	Voltmeter, ammeter, oscilloscope, SCS EOSE, power EOSE, data center	Procedure	Darkened room

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
59	**Perform Data Automation Equipment Electrical Test** The data automation electrical test will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins at the DAE power input connector b. Connect the DAE to the 4.1 kc inverter and measure the voltage and current drawn by the DAE and note that noise and transients are at acceptable levels c. Measure command line voltage and current drawn for each bit rate, format, and mode of operation, note that noise and transients are at acceptable levels d. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all timing pulses at the experimenters side of the harness. This is to be done for each bit rate. e. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all shift pulses at the experimenters side of the harness. This is to be done for each bit rate f. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all synch pulses at the experimenters side of the harness. This is to be done for each bit rate g. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all inhibit pulses at the experimenters side of the harness. This is to be done for each bit rate	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope series fuse boxes, in-line test connectors, digital word data format generator, analog word simulator		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	h. Check ID words corresponding to all bit rates and all formats using the telemetry data display EOSE. i. Loop check all analog words by applying a DC voltage at the experimenters side of the harness and reading out the decimal word at the telemetry data display EOSE. j. Loop check all digital words by applying a digital signal at the senders side of the harness and reading out the decimal word at the telemetry data display EOSE. Note: Noise, transient and cross talk measurements will be conducted for items c. through g.			
60	<u>Install Bulk Storage Units</u>	Hand tools, torque wrenches	Procedure	None
61	<u>Perform Bulk Storage Unit Electrical Tests</u> The bulk storage unit electrical testing will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the bulk storage power connector. b. Connect the bulk storage to the spacecraft harness and measure the voltage and current drawn by the bulk storage. Also note that noise and transients are at acceptable levels.	Fully operated data center, operational computer programs, telemetry data display, EOSE ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Measure all command line voltages and currents for each bulk storage command. Also note that noise and transients are at acceptable levels.	connectors, digital word data format generator		
	d. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage input data signal at the bulk storage for each bit rate.			
	e. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage data output signal at the DAE during memory readout.			
	f. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage index pulse at the DAE.			
	Note: Noise, transient and cross talk measurements will be conducted for items d. through f.			
62	Perform Data Automation and Bulk Storage Calibrations These calibrations are temperature calibrations and will be performed as follows: a. DAE temperature calibration is to be performed as per Task 22.a.1. b. Bulk storage temperature calibration is to be performed as per Task 22.a.1.	Power EOSE, data center, resistor decade box, command EOSE, in-line test connector	Procedure	None
63	Install Capsule Receivers and Detectors	Hand tools, torque wrenches	Procedure	None
64	Perform VHF Capsule Receiver Electrical Tests	RF EOSE, command matrix		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	The receiver electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector. b. Connect each receiver to the spacecraft harness and measure the voltage and current drawn by each receiver noting that noise and transient conditions are within specification. c. Connect the receiver to a strong signal from the capsule EOSE (-110 dbm) and acquire. d. Determine the signal at which the receiver thresholds or drops out of lock. e. Modulate the capsule simulator and measure the receiver output signal amplitude. f. Repeat the above steps for the redundant receiver.	monitor, voltmeter, ammeter, power EOSE, series fuse boxes, in-line test connectors, capsule, simulator		
65	<u>Perform VHF Capsule Receiver Calibration</u> a. Receiver temperature calibrations will be accomplished as in Task 22.a.1. b. The airborne receivers will be dropped in and out of lock by removing the capsule simulator signal and noting that the telemetry indication is proper. c. A precisely known signal level is fed into a precision step attenuator. A known signal strength can now be calculated for each attenuator setting. Each power level is correlated with telemetry output. These parameters are entered in the computer program.	RF EOSE, power EOSE, RF attenuators, calorimeter, data center, in-line test connector	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
66	<u>Perform Capsule Detector Test</u> The capsule detector will be tested as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the detector power connector. b. Connect the detector to the spacecraft harness and measure the voltage and current drawn by the detector. Also note that noise and transients are at acceptable levels. c. Acquire the capsule simulator and measure the amplitude, rise time, and fall time of the detector output signal.			
67	<u>Perform Pyrotechnic Subsystem Integration Tests</u> The pyrotechnic subsystem integration encompasses the following areas: a. Experiment ordnance b. Experiment boom ordnance c. High-gain antenna boom ordnance d. Medium-gain antenna boom ordnance e. Low-gain antenna boom ordnance f. Planet-oriented package boom ordnance g. Midcourse correction motor ordnance h. Solid retropropulsion engine ordnance i. Capsule separation ordnance The pyrotechnic subsystem ordnance tests will be performed as follows:	Ordnance EOSE, system test set EOSE	Procedure	None

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Operation No	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. Ascertain that the pyrotechnic subsystem is in a safe condition by monitoring across each squib bridge wire connector a dead short. b. Command each squib to the fire condition and monitor the "firing" voltage at each squib bridge wire connector. c. Connect the pyrotechnic EOSE to each squib connector and command each squib to the "fire" condition. d. Ascertain that an "all-fire" indication exists for each squib actuation. e. Command each squib to the "fire" condition using under-voltage conditions and ascertain that a "no-fire" condition exists for each squib actuation.			
68	<u>Perform Pyrotechnic Subsystem Calibrations</u> The pyrotechnic subsystem will be calibrated by commanding each squib actuation and monitoring each telemetry word for correct value. NOTE: Steps 59 through 67 are performed to gain engineering data for the 1971 mission			
69	<u>Install the Magnetometer Assembly</u> The magnetometer assembly consists of magnetometer sensors and magnetometer sensor boom.	Hand tools, torque wrenches	Procedure	None
70	<u>Perform Main Body Experiment Electrical Tests</u> The main body experiment electronics and sensors consist of the following items: a. Meteoroid impact experiment b. Magnetometer electronics	Voltmeter ammeter, oscilloscope, power EOSE, command EOSE, experiment EOSE, series fuse boxes	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	The main body electrical testing will be performed as follows: a. Turn on external power to the spacecraft and command each experiment to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each experiment electronics connector c. Connect each experiment electronics package to the spacecraft harness and measure the voltage and current drawn by each electronic package from the secondary power supply d. At each main body sensor package observe that voltage exists where it should and that no voltage exists on the remaining pins of each sensor connector e. Connect each sensor to the spacecraft harness f. Measure the voltage and current drawn by each main body experiment from the secondary power supply subsystem g. Measure the noise content on all main body experiment power and signal lines observing that the noise content is within specified levels h. Measure the rise time, fall time, pulse duration, and amplitude of the turn-on transient of each main body experiment i. Stimulate each experiment and determine that each experiment is working properly by using both EOSE and telemetry information			

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
71	<u>Perform Experiment Compatibility Testing</u> The experiment capability tests will be performed as follows: a. Ascertain that each experiment test source does or does not interfere with another experiment b. Exercise each spacecraft subsystem and ascertain that each subsystem does not interfere or degrade any experiment data c. Exercise each experiment and ascertain that each experiment does not degrade the spacecraft operation. In particular ascertain that the radio propagation experiments do not degrade the RF subsystem	Complete set of system test EOSE, spectrum analyzer	Procedure	None
72	<u>Perform Experiment Calibrations</u> The magnetometer calibration will take place at the magentometer site. The Meteoroid Impact experiment will be calibrated by signal injection using a suitable signal generator	Complete set of systems test EOSE	Procedure	None
73	<u>Perform Power Profile Test</u> The power profile tests will be performed as follows: a. The flight sequence of events up until sun acquisition will be followed and primary power drains monitored b. Compare the primary power drains up until sun acquisition with the trajectory information and ascertain that the battery capacity is adequate to support spacecraft operations until sun acquisition has been completed	Recorders, current probe, complete set of systems EOSE, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Command the spacecraft to perform all of the cruise mode functions monitoring all primary power drains d. Compare the primary power drains with the trajectory information and ascertain that sufficient battery capacity remains to perform the midcourse maneuvers e. Command the spacecraft to perform all of the cruise mode and Mars encounter functions monitoring all primary power drains			
74	<u>Perform Solar Array Testing and Calibrations</u> The solar array testing and calibrations will be accomplished as follows: a. Illuminate each solar array string and measure the short circuit current and open circuit voltage b. Perform inverse impedence tests on each solar array string	Resistor decade box, solar array test set, voltmeter, ammeter	Procedure	None
75	<u>Install Thermal Insulation and Louvers</u> The thermal insulation and louvers will be installed in preparation for the electromagnetic compatibility testing, magnetic property testing, and thermal vacuum testing	Hand tools, torque wrenches		
76	<u>Perform Electromagnetic Compatibility Tests</u> The electromagnetic compatibility tests will be performed as follows: a. Command the spacecraft subsystems through every combination and permutation of the flight sequences and ascertain that there is no degradation of interference between subsystems b. Irradiate the spacecraft with RF signals that correspond to the expected frequencies and levels from the Saturn and Centaur over-all launch vehicle system	Complete set of systems test EOSE, electromagnetic test set	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Command the spacecraft subsystems through every combination and permutation of the Voyager flight sequences and determine the frequencies and levels of all radiation that is emitted from the spacecraft d. Apply audio tones and tone bursts to the spacecraft primary bus system and observe each subsystem reaction noting that it is within specification			
77	<u>Perform Integrated System Test</u> The integrated systems test rigidly follows the flight sequence of events. Each Voyager space subsystem is tested to the maximum level and proper operation is verified by using the systems test EOSE and the data center to carefully reduce all telemetry data	Complete set of systems test EOSE	Procedure	None
78	<u>Perform Integrated Systems Test Critique</u> The integrated system test critique is a meeting of all cognizant personnel to discuss the results of the integrated systems test. It is during this meeting that each subsystem engineer signs off the IST data	None	Records to be signed off	None
79	<u>Perform Shipping Preps</u> The spacecraft booms and other appendages are folded and latched and the spacecraft is placed in the shipping container. Next desicate is placed inside of the container and the container sealed. The shipping container and spacecraft are purged with dry nitrogen. Note that it will be necessary to remove the array panels and support structure for shipment	Slings, shipping containers, purging equipment	Procedure	Crane with hook height of ____

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
80	Ship Voyager Planetary Vehicle to Magnetic Properties Site The spacecraft and shipping container will be shipped to the test site. During shipment the shipping container is purged with dry nitrogen	Helicopter sling	Procedure	
81	Receive Voyager Planetary Vehicle and Remove Shipping Container After removing the shipping containers, the spacecraft is next placed on magnetic properties test fixture and torqued down	Magnetic properties	Procedure	Crane with hook height of ______
82	Map Voyager Planetary Vehicle Perm Field The magnetic field of the spacecraft is measured with no power applied	Magnetic properties measuring equipment	Procedure	Low magnetic ambient field
83	Perform Voyager Planetary Vehicle Magnetic Stability Tests The spacecraft will be permed and depermed and the change in the spacecraft magnetic field measured	Magnetic properties measuring equipment, magnetizing coils	Procedure	None
84	Measure the Spacecraft Induced Magnetic Fields Each spacecraft subsystem will be commanded to perform every combination and permutation of the possible operating modes. While this is taking place, the spacecraft magnetic fields are measured	Magnetic properties measuring equipment, complete set of system test EOSE, long EOSE cables, coil system to buck out earths magnetic field	Procedure	Low magnetic ambient field

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
85	<u>Calibrate Magnetometer Experiment</u> The magnetometer boom is deployed and the magnetometer extended into the coil system. Precision currents are fed through the coil system to generate known magnetic field strengths, as well as to buck out the effects of the earth's field. EOSE measurements and telemetered data are compared with the known fields generated by the coil system. These parameters are entered into the computer program.	Coil system to buck out earth's magnetic field, complete set of systems test EOSE, long EOSE cables, slings, handling fixture	Procedure	Crane with hook height of ____
86	<u>Perform Shipping Preparations</u> The Voyager planetary vehicle booms and other appendages are folded and latched. The spacecraft is placed in the shipping container. Next desicate is placed in the shipping container.and the container sealed. The shipping container and spacecraft are purged with dry nitrogen. Note that it will be necessary to remove the array panels and support structure for shipment.	Slings, shipping containers, purging equipment		
87	<u>Ship Voyager Planetary Vehicle to Redondo Beach</u> After magnetic testing the spacecraft is to be placed into the shipping container and sealed with desicate. The nitrogen purging equipment is next attached and purging started. The spacecraft and shipping container are shipped via helicopter back to Redondo Beach.	Slings, shipping container, desicate, purging equipment, helicopter	Procedure	Crane with hook height of ____
88	<u>Prepare Voyager Spacecraft for Alignments and Leak Testing</u> After the spacecraft has been removed from the shipping contained it will be placed upon the tilt fixture and the solar array panels and support structure installed in preparation for spacecraft alignments and leak testing.	Slings, tilt fixture, torque wrench	Procedure	Crane with hook height of ____

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
89	<u>Perform Leak Tests</u> After the vibration test has been completed, the SCS pneumatic system, the monopropellant engine system and the solid engine TVD system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced dur to shipping and handling operations. During this leak test all tank pressure and temperature calibrations will take place.	System test set EOSE, SCS leak test console, propulsion leak test console	Procedure	None
90	<u>Perform Spacecraft Alignments</u> After the leak test has been completed, all spacecraft alignments will be checked. Listed below are all of the alignments that will be checked: a. Solid retropropulsion motor b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Omni antenna alignments l. Omni antenna boom latch alignments m. Magnetometer experiment alignments n. Magnetometer boom latch alignments o. Planetary vehicle vertical alignments p. Medium-gain antenna alignments q. Medium-gain antenna latch alignments	Alignments sets, autocollimators	Procedure	None

Functional Flow Drawing Title and No. **Proof Test Model Spacecraft Type Approval Testing** Revision ______ Date ______ Approval ______ Page No. 44

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
91	**Perform Appendage Deployment Test** **After the alignment test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure or misalignment.**	**Systems test set EOSE, deployment fixtures**	**None**	**None**
92	**Mate the Planetary Vehicle to the Centaur Adapter**	**Slings, torque wrenches, tag lines**	**Procedure**	**Crane with hook height of ____**
93	**Perform Spacecraft Vertical Alignment** **The spacecraft vertical alignment will be checked optically and scribe marks used as reference points, once the alignment has been completed.**	**Spacecraft vertical alignment set**	**Procedure**	**None**
94	Perform Integrated System Test The integrated system test is performed at this time to establish base line conditions prior to undergoing type approval testing.	**System test set EOSE**	**Procedure**	**None**
95	Perform Integrated System Test and Critique The integrated system test critique is a meeting of all cognizant personnel to discuss the results of the integrated system test. It is during this meeting that each subsystem engineer signs off the IST data.	None	Records to be signed off	None

1

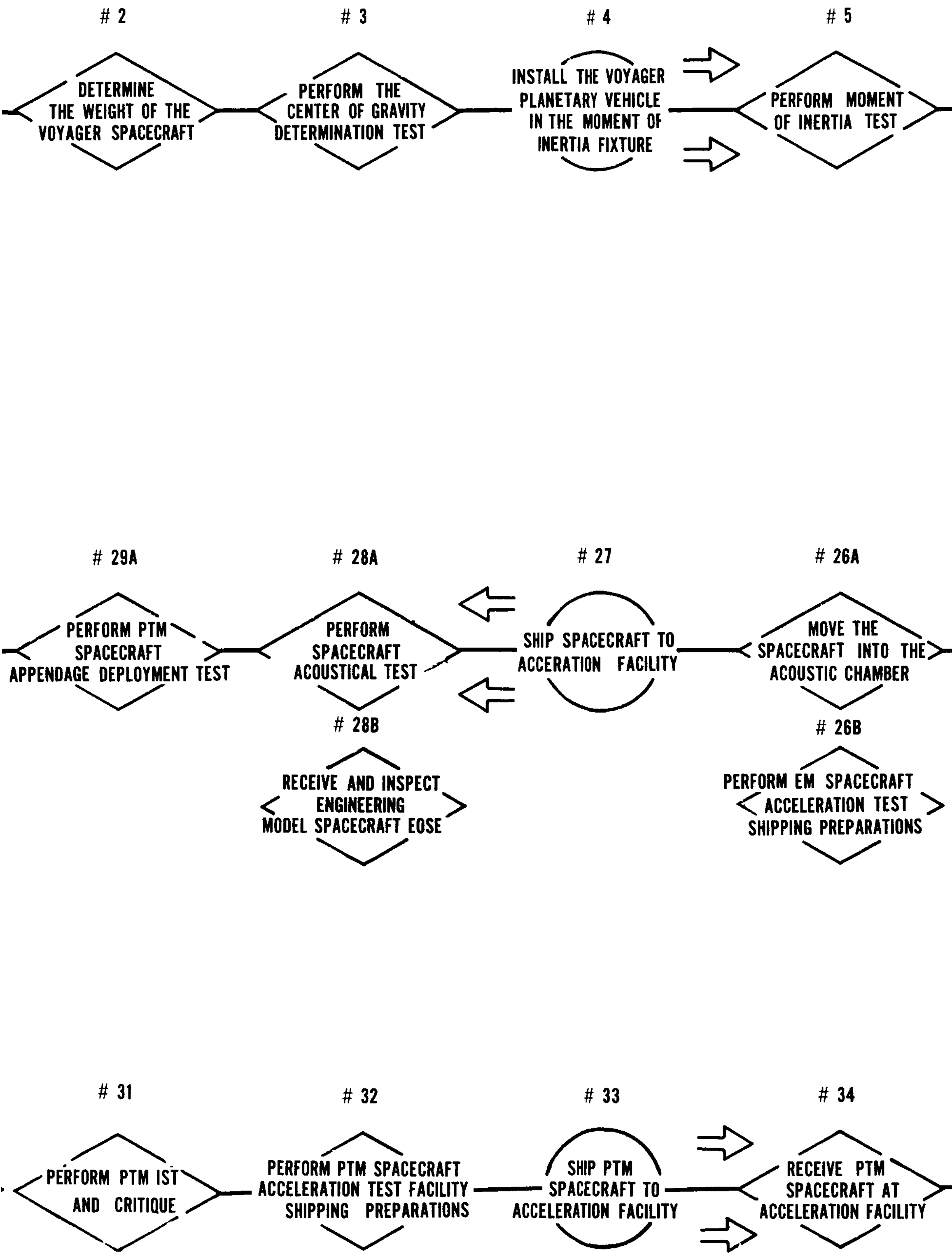
INSTALL THE PROOF TEST MODEL PLANETARY VEHICLE INTO THE WEIGHT AND C G FIXTURE

30A

PERFORM PTM SCS AND PROPULSION PNEUMATIC LEAK TEST

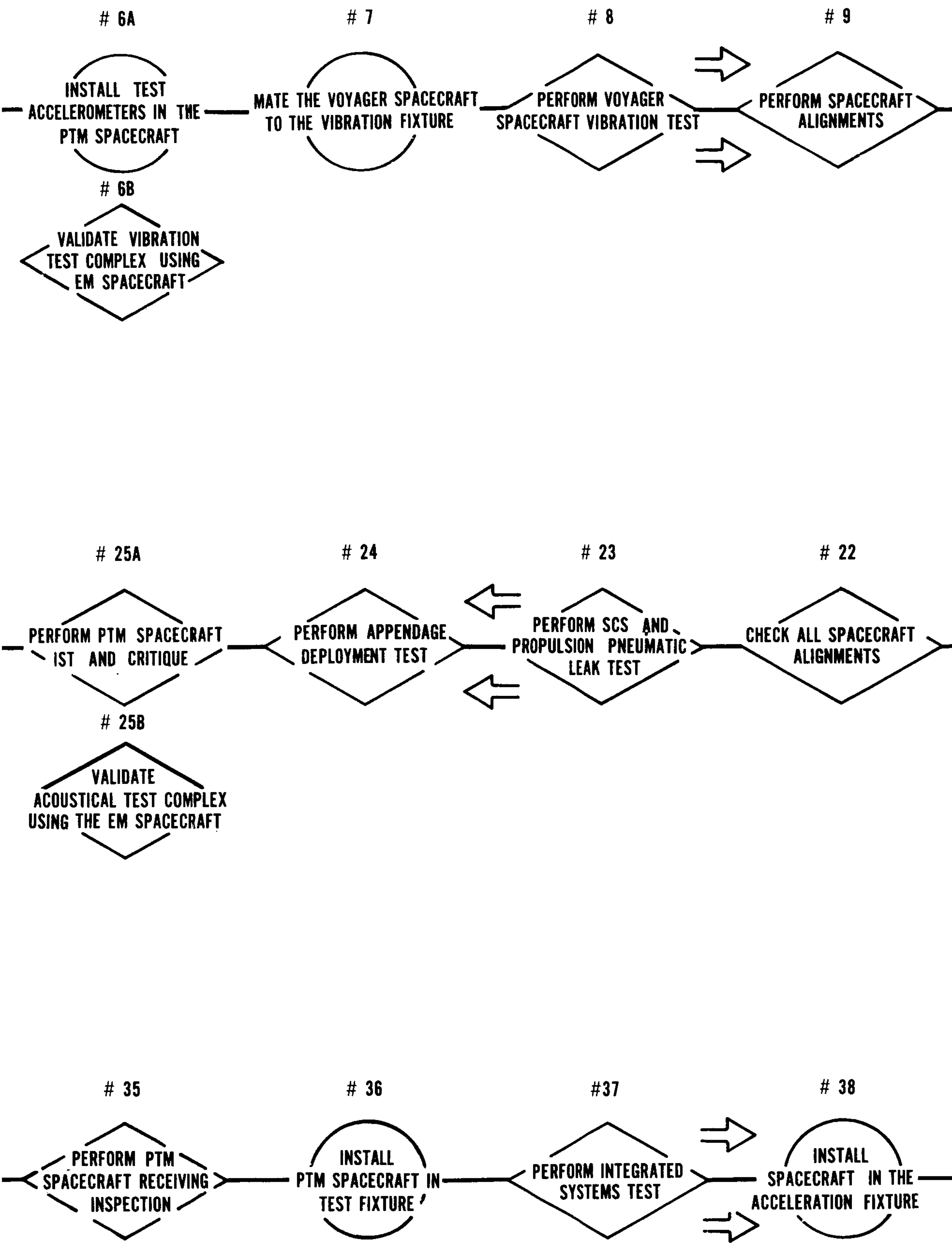
30B

VALIDATE ACCELERATION TEST COMPLEX USING ENGINEERING MODEL SPACECRAFT

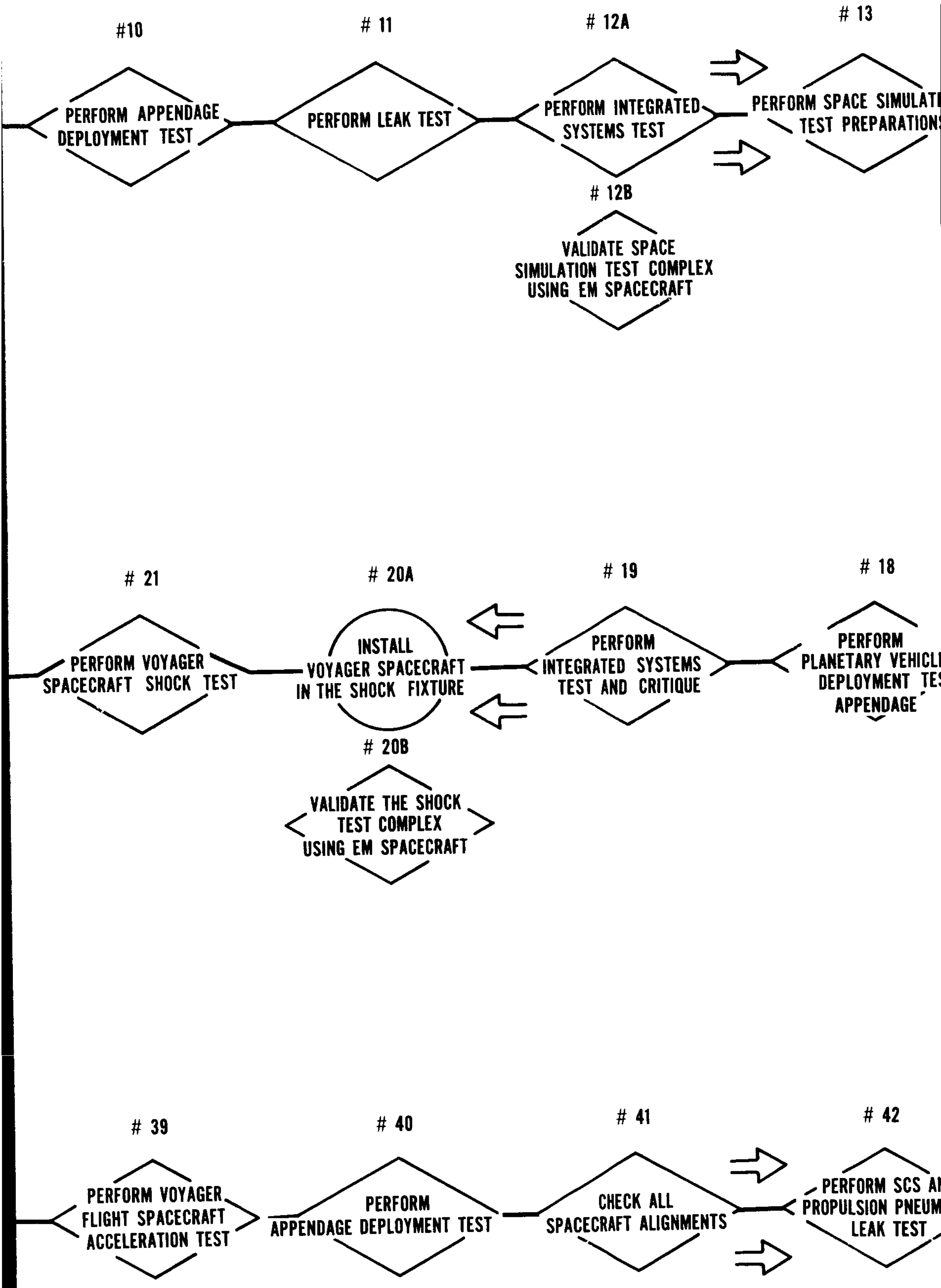


2
DETERMINE THE WEIGHT OF THE VOYAGER SPACECRAFT
3
PERFORM THE CENTER OF GRAVITY DETERMINATION TEST
4
INSTALL THE VOYAGER PLANETARY VEHICLE IN THE MOMENT OF INERTIA FIXTURE
5
PERFORM MOMENT OF INERTIA TEST
29A
PERFORM PTM SPACECRAFT APPENDAGE DEPLOYMENT TEST
28A
PERFORM SPACECRAFT ACOUSTICAL TEST
28B
RECEIVE AND INSPECT ENGINEERING MODEL SPACECRAFT EOSE
27
SHIP SPACECRAFT TO ACCERATION FACILITY
26A
MOVE THE SPACECRAFT INTO THE ACOUSTIC CHAMBER
26B
PERFORM EM SPACECRAFT ACCELERATION TEST SHIPPING PREPARATIONS
31
PERFORM PTM IST AND CRITIQUE
32
PERFORM PTM SPACECRAFT ACCELERATION TEST FACILITY SHIPPING PREPARATIONS
33
SHIP PTM SPACECRAFT TO ACCELERATION FACILITY
34
RECEIVE PTM SPACECRAFT AT ACCELERATION FACILITY

2


6A
INSTALL TEST ACCELEROMETERS IN THE PTM SPACECRAFT
6B
VALIDATE VIBRATION TEST COMPLEX USING EM SPACECRAFT
7
MATE THE VOYAGER SPACECRAFT TO THE VIBRATION FIXTURE
8
PERFORM VOYAGER SPACECRAFT VIBRATION TEST
9
PERFORM SPACECRAFT ALIGNMENTS
25A
PERFORM PTM SPACECRAFT IST AND CRITIQUE
25B
VALIDATE ACOUSTICAL TEST COMPLEX USING THE EM SPACECRAFT
24
PERFORM APPENDAGE DEPLOYMENT TEST
23
PERFORM SCS AND PROPULSION PNEUMATIC LEAK TEST
22
CHECK ALL SPACECRAFT ALIGNMENTS
35
PERFORM PTM SPACECRAFT RECEIVING INSPECTION
36
INSTALL PTM SPACECRAFT IN TEST FIXTURE
#37
PERFORM INTEGRATED SYSTEMS TEST
38
INSTALL SPACECRAFT IN THE ACCELERATION FIXTURE

3



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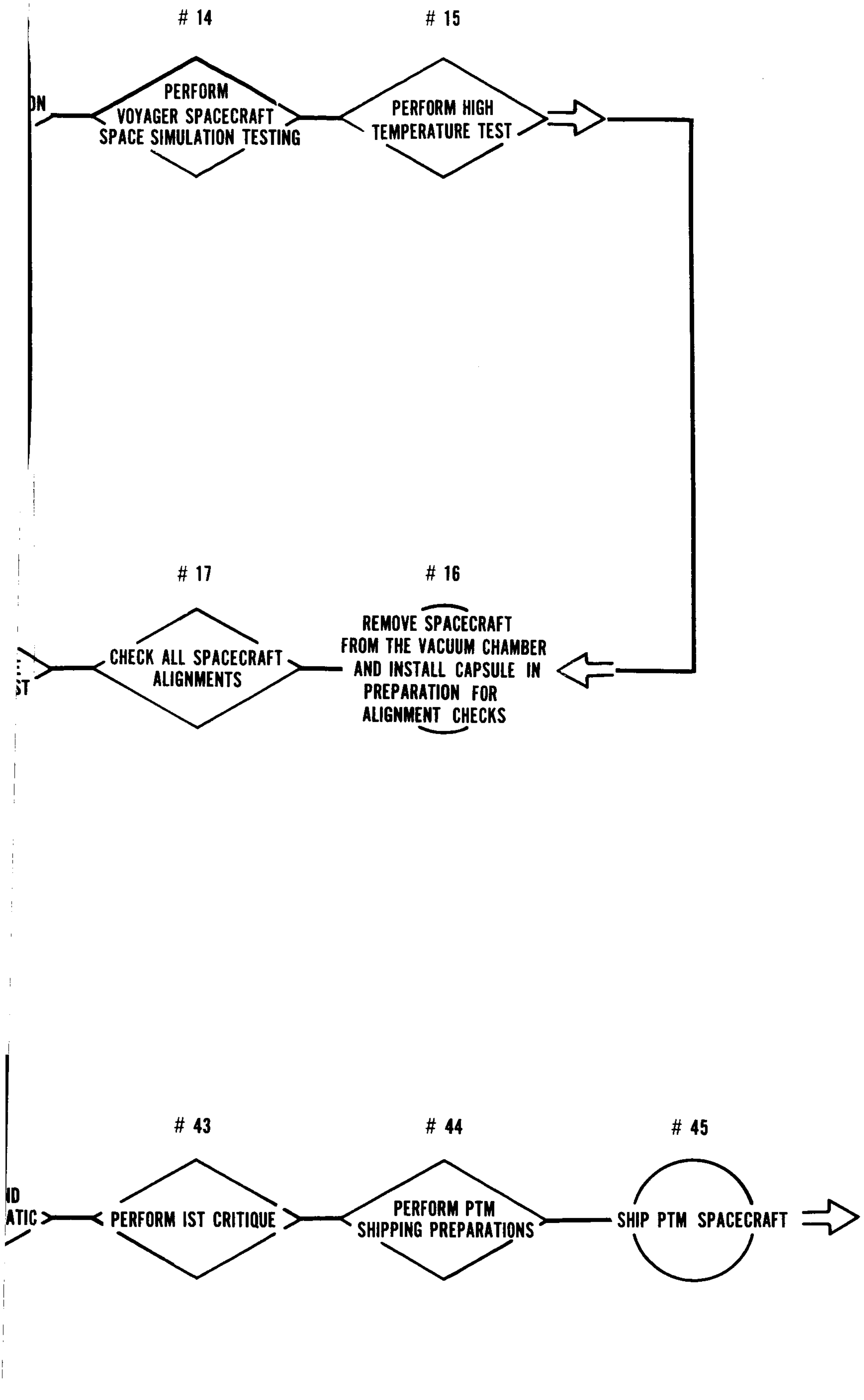
1969 PROOF TEST M



5

ODEL S/C TYPE APPROVAL TESTING

Functional Flow Proof Test Model Spacecraft
Drawing Title and No. Type Approval Testing Revision ______ Date ______ Approval ______ **Page No.** 1 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1	Install the Proof Test Model Planetary Vehicle into the Weight and Center of Gravity Fixture	Hand tools, torque wrenches, c. g. fixture	None	Some means of hoisting the spacecraft into the c. g. fixture
2	Determine the Weight of the Voyager Spacecraft The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axis. Note that the weight of the spacecraft less capsule was determined during assembly and test.	Load cells and associated electronics, c. g. fixture	Procedure	None
3	Perform the Center of Gravity Determination Test The center of gravity for two of the spacecraft axes was determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axis. Note that the center of gravity determination of the spacecraft less capsule was determined during assembly and test.	C. g. fixture	Procedure	None
4	Install the Voyager Planetary Vehicle in the Moment of Inertia Fixture	Inertia fixture, slings	None	Some means of hoisting the spacecraft into the inertia fixture
5	Perform Moment of Inertia Test The moments of inertia about the roll axis and the maximum and minimum moments about the transverse axis will be determined and compared with design requirements. Note that the moment of inertia determination of the spacecraft less capsule was determined during assembly and test.	Timer	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
6A	Install Test Accelerometers in the PTM Spacecraft Test accelerometers will be used to monitor the forces acting on the spacecraft during the vibration test.	Test accelerometers, accelerometer electronics	Procedure	None
6B	Validate Vibration Test Complex Using EM Spacecraft The engineering model spacecraft will be utilized to verify the vibration test cabling and EOSE.	Vibration test EOSE, vibration test cables	Procedure	None
7	Mate the Voyager Spacecraft to the Vibration Fixture	Vibration fixture, slings	None	Crane with hook height of ____
8	Perform Voyager Spacecraft Vibration Test The purpose of the vibration test is to demonstrate the capability of the flight spacecraft to withstand the mission vibration environments as specified in the Voyager mission environmental specification. It is expected that these environments will consist of low frequency sinusoid and random inputs that could occur during the launch boost phase. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels will be filled to flight specifications.	Complete set of EOSE vibration tables, vibration transducers and recorders, pressurization console, fueling consoles	Procedure	Vibration fixtures, vibration tables
9	Perform Spacecraft Alignments After the vibration test has been completed, all spacecraft alignments will be checked. Listed below are all of the alignments that will be checked: a. Monopropellant motor alignment b. Gyro alignments c. Sun sensor alignments d. Canopus sensor alignments e. Gas jet alignments f. High-gain antenna alignments	Alignment sets, autocollimators	Procedure	None

Functional Flow Proof Test Model Spacecraft
Drawing Title and No. Type Approval Testing **Revision** ______ **Date** ______ **Approval** ______ **Page No.** 3 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	g. High-gain antenna latch alignments h. Omni antenna alignments i. Omni antenna boom latch alignments j. Magnetometer experiment alignments k. Magnetometer boom latch alignments l. Planetary vehicle vertical alignments			
10	<u>Perform Appendage Deployment Test</u> After the vibration test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test set, EOSE, deployment fixtures	None	None
11	<u>Perform Leak Test</u> After the vibration test has been completed, the SCS pneumatic system and the monopropellant engine system will be leak tested. The purpose of this test is to ascertain that the pneumatic lead and flow rates are within specification and that no damage was experienced dur to vibration.	SCS leak test console, propulsion lead test console	Procedure	None
12A	<u>Perform Integrated Systems Test</u> The integrated systems test will be performed at the conclusion of the vibration test. The purpose of the integrated systems test is to ascertain that there has been no degradation in the Voyager spacecraft subsystems due to vibration testing.	Complete set of systems EOSE and cabling	Procedure	Electrical outlets
12B	<u>Validate Space Simulation Test Complex Using EM Spacecraft</u> Concurrently, while the integrated systems test is being conducted, the engineering model spacecraft will be utilized to verify the space-simulation test cabling, EOSE and mechanical fixtures.	Complete set of systems EOSE and cables, ESM model spacecraft	Procedure	Electrical outlets

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 4 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
13	Perform Space Simulation Test Preparations The space simulation test preparations consist of the following tasks: a. Install heaters in the planetary vehicle b. Install thermocouples in the planetary vehicle c. Installation of the planetary vehicle into the simulation fixture d. Functional test as a final verification of the space simulation electrical complex and mechanical MOSE.	Sun source, Canopus source, heaters, thermocouple standard solar cells, gas actuator monitoring EOSE	Procedure	Vacuum chamber, electrical outlets for EOSE
14	Perform Voyager Spacecraft Space Simulation Testing The spacecraft simulation testing will be performed as follows: a. When the proper pressure has been reached, the vacuum chamber cold walls will be turned on and the spacecraft allowed to temperature soak b. When the spacecraft has reached the temperature that would be expected during the spacecraft separation portion of the mission sequence, the spacecraft sun acquisition mode will be initiated c. After the SCS sun acquisition testing has been completed, the solar array testing sequence will commence. The solar array testing phase will consist of the following: 1. The sun simulator output intensity and dispersion will be determined by using standard solar cells 2. The Voyager spacecraft solar array output will be monitored to determine that the solar array output performance meets specification 3. The primary power charge control subsystem will be exercised and the performance will be monitored for proper operation. For each charge rate the following relationships must hold: solar array current = shunt regulator current + bus current + battery current.	Same as above	Procedure	Vacuum chamber

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 5 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. Following the solar array testing phase of the space simulation test, the Canopus acquisition tests will start. The ability of the Canopus sensor and associated electronics to perform to specification will be monitored. e. After Canopus has been acquired, the cruise science will be turned on and the ability to perform to specifications will be monitored. f. The next event to be checked out in the flight sequence of events will be the midcourse maneuvering sequence. The spacecraft turn maneuver will be performed in each axis in each d direction. The midcourse correction engine jet vane angles will be commanded and checked in each direction. The motor burn time will correspond to the maximum burn time that can be commanded to the spacecraft. The ability to perform to specification of the midcourse sequencing will be monitored. g. It should be mentioned that both the SCS and propulsion leak testing will take place throughout the space simulation test. h. Post midcourse maneuver cruise mode testing is as follows: 1) Sun acquisition established 2) Canopus acquisition established 3) spacecraft powered from the sun simulation source 4) All cruise science on 5) The RF up and down link (coherent) operation established All subsystem performance data will be monitored to ascertain that the Voyager spacecraft performs within specified limits.			
	<u>Perform High Temperature Test</u>	None	Procedure	None
	The cold walls will be turned off and the spacecraft temperature allowed to rise to upper specification limit. When the spacecraft has reached its upper limits, each subsystem will be exercised and monitored for proper operation.			

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing **Revision** ______ **Date** ______ **Approval** ______ **Page No.** 6 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
16	<u>Remove Spacecraft from the Vacuum Chamber and Install Capsule in Preparation for Alignment Checks</u>	Slings, capsule handling fixture, spacecraft handling fixture	Procedure	Crane with hook height of ______
17	<u>Check all Spacecraft Alignments</u> All spacecraft alignments will be checked for shifts due to thermal effects. Listed below are the spacecraft alignments that will be checked: a. Monopropellant motor alignment b. Gyro alignment c. Sun sensor alignments d. Canopus sensor alignments e. Gas jet alignments f. High-gain antenna alignments g. High-gain antenna latch alignments h. Omni antenna alignments i. Omni antenna latch alignments j. Magnetometer experiment alignment k. Magnetometer experiment latch alignment l. Planetary vehicle vertical alignments	Complete compliment of alignment sets, auto-collimators, bench marks	Procedure	Bench marks
18	<u>Perform Appendages Deployment Test</u> After the space simulation test has been completed, each spacecraft appendage will be deployed. The appendage will be deployed in a simulated zero g field using live ordnance, observing that each freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test EOSE, deployment fixtures	Procedure	None

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing **Revision** ______ **Date** ______ **Approval** ______ **Page No.** 7 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
19	Perform Integrated Systems Test and Critique The IST is performed at this time for two reasons: a. To verify that the Voyager spacecraft and all of its subsystems operate properly at atmospheric pressure. Often failures due to vacuum become evident only when the chamber vacuum is released. b. To perform any subsystem test that could not adequately be performed to mechanical and electrical constraints that are incurred when operating a spacecraft in a space simulator.	Complete set of systems test EOSE	Procedure	Electrical outlets
20A	Install Voyager Spacecraft in the Shock Fixture	Slings, spacecraft handling fixture, shock fixture	Procedure	Electrical outlets
20B	Validate the Shock Test Complex Using EM Spacecraft Concurrently the electrical compatibility model spacecraft will be used to validate the systems test set and MOSE and EOSE that will be used for shock testing.	ECM, spacecraft, complete set of shock EOSE, shock transducers and electronics	Procedure	Electrical outlets
21	Perform Voyager Spacecraft Shock Test The purpose of the shock test is to demonstrate the capability of the flight spacecraft to withstand the mission shock environments as specified in the Voyager mission environmental specification. Note that the spacecraft will be electrically powered and that all pneumatic and fuel vessels will be filled to flight specifications.	Shock test fixture, shock test transducers and electronics, shock test EOSE	Procedure	Shock test fixture, electrical outlets

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
22	Check all Spacecraft Alignments All spacecraft alignments will be checked for shifts due to the abovementioned shock environments. Listed below are the spacecraft alignments that are to be checked: a. Monopropellant motor alignment b. Gyro alignments c. Sun sensor alignments d. Gas jet alignments e. Canopus sensor alignments f. High-gain antenna alignments g. High-gain antenna latch alignments h. Omni antenna alignments i. Omni antenna latch alignments j. Magnetometer experiment alignment k. Magnetometer experiment latch alignment l. Planetary vehicle vertical alignment	Complete compliment of alignmen sets, auto-collimators	Procedure	None
23	Perform SCS and Propulsion Pneumatic Leak Test The stabilization and control and the monopropellant propulsion engine subsystems will be tested for leaks that may have been incurred during the shock test.	Propulsion leak test console, SCS leak test console	Procedure	None
24	Perform Appendage Deployment Test After the vibration test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure, or misalignment.	None	None	None
25A	Perform PTM Spacecraft IST and Critique The integrated systems test will be performed to verify that the Voyager spacecraft and all of its subsystems have successfully survived the shock test.	Complete set of systems EOSE	Procedure	Electrical outlets

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 9 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
25B	Validate the Acoustical Test Complex Using the EM Spacecraft			
26A	Move the Spacecraft into the Acoustic Chamber	Handling fixture, slings, transporter	Procedure	Overhead crane with hook height of ______
26B	Perform EM Spacecraft Acceleration Test Shipping Preparations The spacecraft shipping preparations will include both the engineering model spacecraft and its system test set EOSE. The engineering model is to be used to check out the acceleration complex. The spacecraft and EOSE will be placed in shipping containers and purged with dry nitrogen.	Slings, handling fixtures, shipping containers, purging equipment	Procedure	None
27	Ship Spacecraft to Acceleration Facility	Same as above	Procedure	None
28A	Perform Spacecraft Acoustical Test The purpose of the shock test is to demonstrate the capability of the flight spacecraft to withstand the acoustical environments as specified in the Voyager mission environmental specification. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels filled to flight specification.	Acoustical test transducers and electronics, acoustical test EOSE and test cables, noise generators	Procedure	Acoustical chamber, electrical outlets
28B	Receive and Inspect Engineering Model Spacecraft EOSE	Handling fixtures, slings	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
29	Perform PTM Spacecraft Appendages Deployment Test After the acoustical test has been completed in each axis, each spacecraft appendage will be deployed. The appendages will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test, EOSE deployment fixtures	Procedure	None
	a. Mate EM spacecraft to acceleration fixture b. Validate systems test set EOSE	Sligns, handling fixture	Procedure	Crane with hook height of ______, acceleration machine
30A	Perform PTM SCS and Propulsion Pneumatic Leak Test The stabilization and control and monopropellant propulsion engine subsystems will be tested for leaks that may have been encountered during acoustical testing.	SCS leak test console, midcourse motor leak test console	Procedure	None
30B	Perform Acceleration Facility Validation The engineering model spacecraft and EOSE will be used to validate the acceleration facility cabling and specialized EOSE and MOSE.	System test set EOSE, voltmeters, ammeters	Procedure	None
31	Perform PTM IST and Critique The integrated system test will be performed to verify that the spacecraft and all of its subsystems have successfully survived the acoustical test.	Complete set of systems test EOSE	Procedure	Electrical outlets
32	Perform PTM Spacecraft Acceleration Test Shipping Preparations The spacecraft shipping preparations will include both the spacecraft and its system test set EOSE. The spacecraft and EOSE will be placed in shipping containers and purged with dry nitrogen.	Slings, handling fixtures, shipping containers, purging equipment	Procedure	None

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 11 1969

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
33	Ship PTM Spacecraft at Acceleration Facility	Slings, handling fixtures, shipping containers, purging equipment	Procedure	None
34	Receive PTM Spacecraft at Acceleration Facility	None	None	None
35	Perform PTM Spacecraft Receiving Inspection The receiving inspection will be an inspection for shipping and handling damage.	None	Procedure	None
36	Install PTM Spacecraft in Test Fixture	Slings, handling fixture	Procedure	Crane with hook height of ______
37	Perform IST The integrated system test will be performed to verify that the spacecraft has incurred no damage due to shipping and handling.	Systems test set	Procedure	None
38	Install Spacecraft in the Acceleration Fixture	Slings, spacecraft handling fixture, acceleration fixture	Procedure	Acceleration machine, electrical outlets
39	Perform Voyager Flight Spacecraft Acceleration Test The purpose of the acceleration test is to demonstrate the capability of the Voyager flight spacecraft to withstand the mission acceleration environments as specified in the Voyager Mission Environmental Specification. Note that the spacecraft will be electrically powered and that all pneumatic and fuel vessels will be filled to flight specifications.	Complete set of acceleration EOSE, acceleration test transducers and electronics	Procedure	Acceleration machine, electrical outlets

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ____ Date ____ Approval ____ Page No. 12

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
40	<u>Perform Appendage Deployment Test</u> After the vibration test has been completed in each axis, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalingment.	Systems test EOSE, deployment fixtures	Procedure	None
41	<u>Check All Spacecraft Alignments</u> All spacecraft alignments will be checked for shifts due to the abovementioned acceleration environments. Listed below are the spacecraft alignments to be checked: a. Monopropellant motor alignments b. Gyro alignments c. Sun sensor alignments d. Canopus sensor alignments e. Gas jet alignments f. High-gain antenna alignments g. High-gain antenna latch alignments h. Omni antenna alignments i. Omni antenna latch alignments j. Magnetometer experiment alignment k. Magnetometer experiment latch alignment l. Planetary vehicle vertical alignments	Complete compliment of alignment sets, autocollimators	Procedure	None
42	<u>Perform SCS and Propulsion Pneumatic Leak Test</u> The stabilization and control subsystem and the monopropellant propulsion engine subsystem will be tested for leaks that may have been incurred during acceleration testing.	SCS leak test console, midcourse motor leak test console	Procedure	None
43	<u>Perform IST Critique</u> The integrated systems tests will be performed to verify that the spacecraft and all of its subsystems have successfully survived the acceleration test.	Complete set of systems test EOSE	Procedure	

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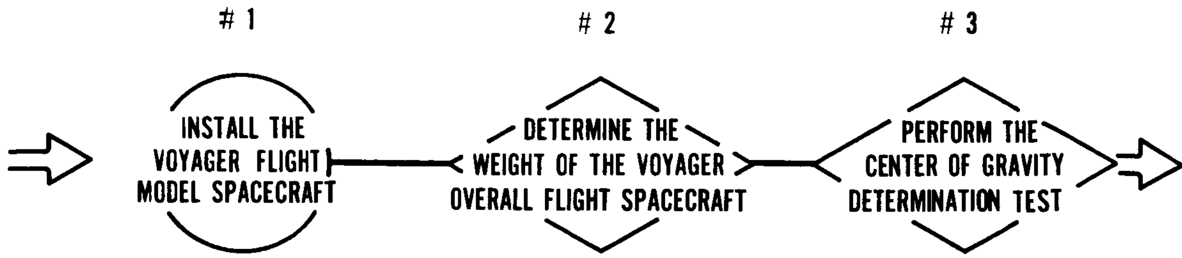
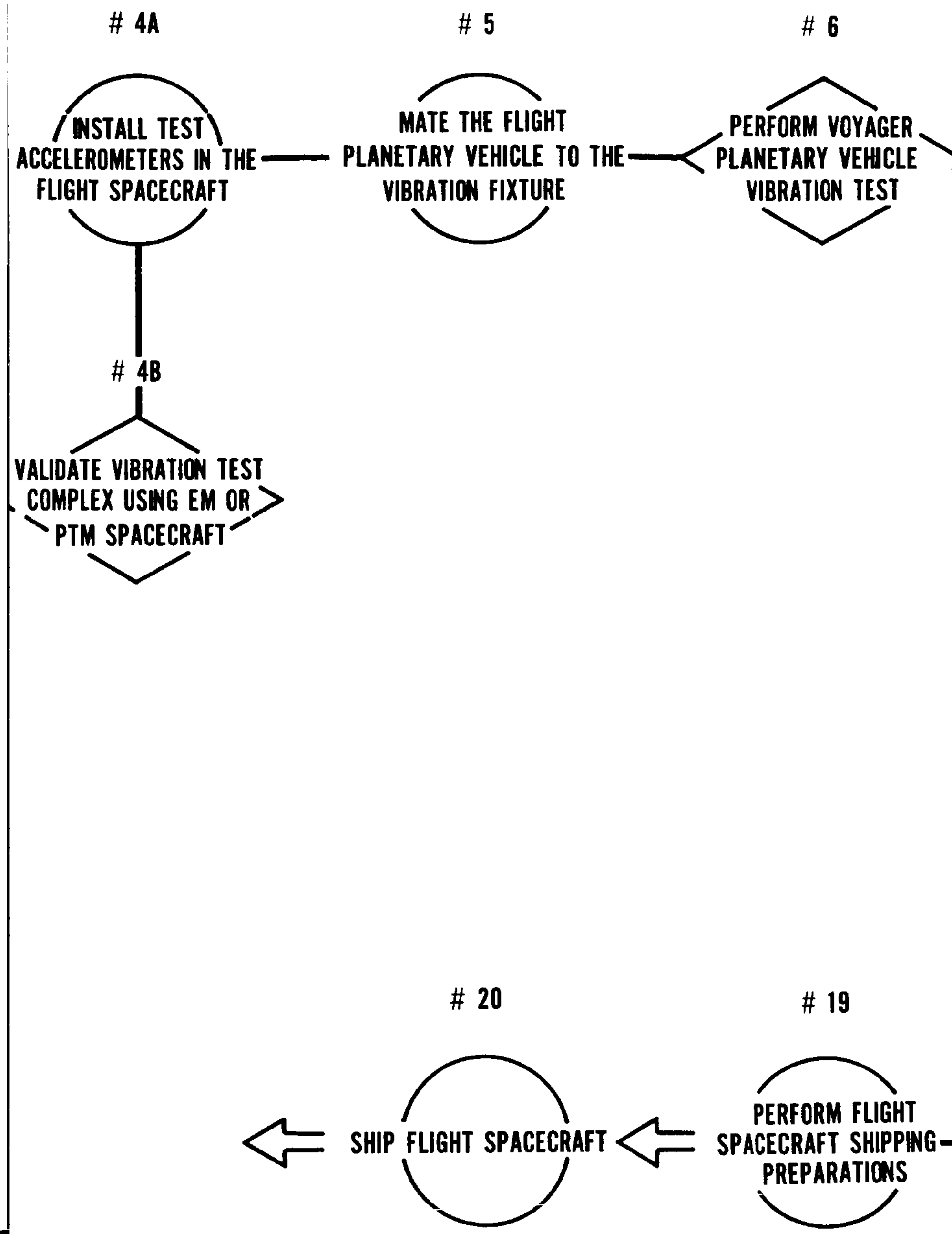
Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 13

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
44	Perform PTM Shipping Preparations The spacecraft shipping preparations will involve both the spacecraft and its system test set EOSE. The spacecraft and EOSE will be placed in shipping containers and purged with dry nitrogen.	Slings, handling fixtures, purging equipment	Procedure	Crane with hook height of ______
45	Ship PTM Spacecraft	Slings, handling fixtures, purging equipment	Procedure	Crane with hook height of ______

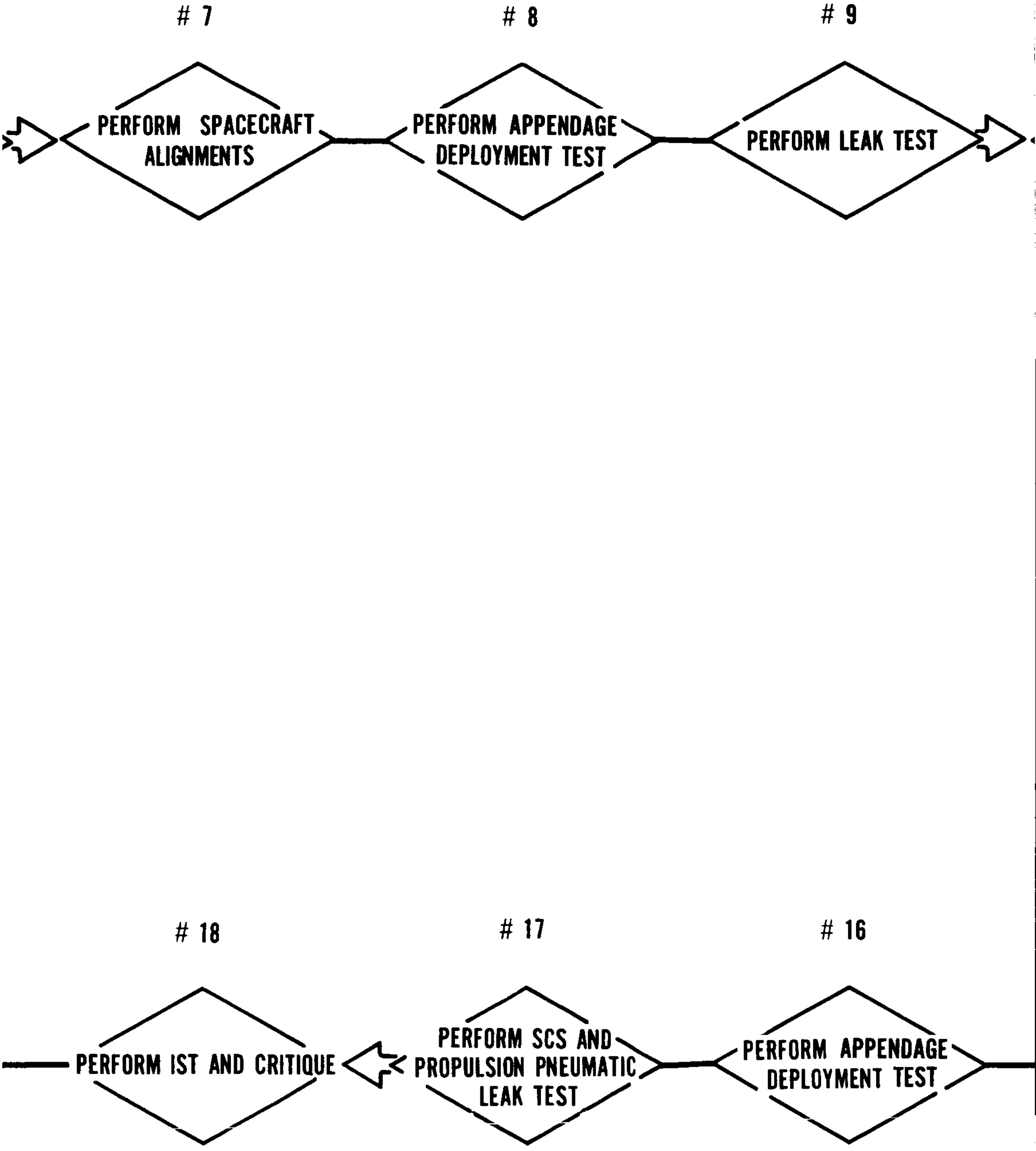


1
INSTALL THE VOYAGER FLIGHT MODEL SPACECRAFT
2
DETERMINE THE WEIGHT OF THE VOYAGER OVERALL FLIGHT SPACECRAFT
3
PERFORM THE CENTER OF GRAVITY DETERMINATION TEST



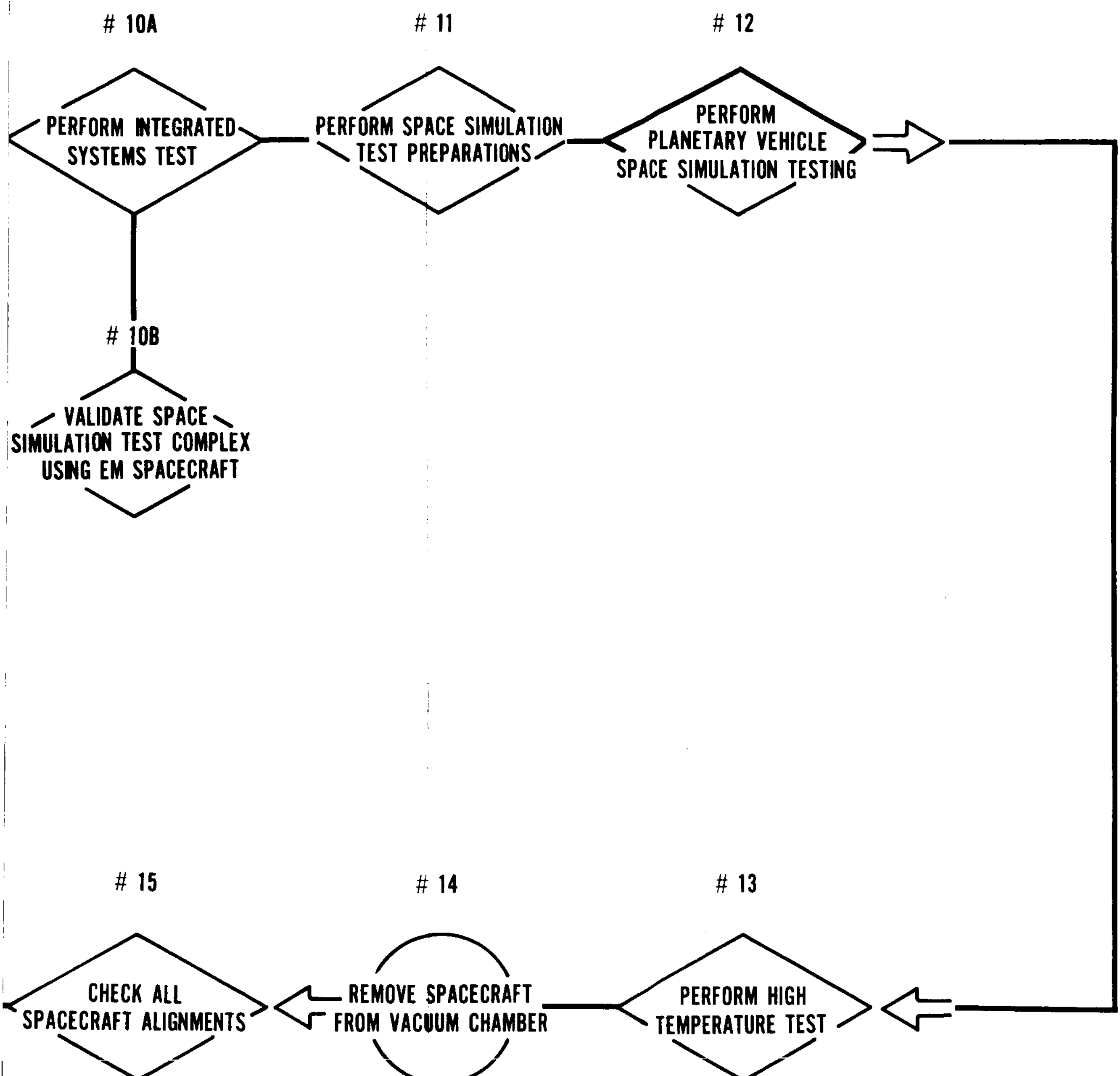
4A
INSTALL TEST ACCELEROMETERS IN THE FLIGHT SPACECRAFT
5
MATE THE FLIGHT PLANETARY VEHICLE TO THE VIBRATION FIXTURE
6
PERFORM VOYAGER PLANETARY VEHICLE VIBRATION TEST
4B
VALIDATE VIBRATION TEST COMPLEX USING EM OR PTM SPACECRAFT
20
SHIP FLIGHT SPACECRAFT
19
PERFORM FLIGHT SPACECRAFT SHIPPING PREPARATIONS

2



1969 VOYAGER FLIGHT M

3



ODEL S/C FLIGHT APPROVAL TESTING

4

Functional Flow Drawing Title and No. Flight Model Spacecraft Flight Approval Testing Revision ______ Date ______ Approval ______ Page No. 1

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1	Install the Voyager Flight Model Spacecraft	Hand tools, torque wrenches, c.g. fixture	None	Some means of hoisting the spacecraft into the c. g. fixture
2	Determine the Weight of the Voyager Planetary Vehicle The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. Note that the weight of the spacecraft less capsule was determined during assembly and test.	Load cells and associated electronics, c.g. fixture	Procedure	None
3	Perform the Center of Gravity Determination Test The center of gravity for two of the spacecraft axes was determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axes. Note that the center of gravity determination of the spacecraft less capsule was determined during assembly and test.	C.g. fixture	Procedure	None
4A	Install Test Accelerometers in the Flight Spacecraft Test accelerometers will be used to monitor the forces acting on the spacecraft during the vibration test.	Test accelerometers, accelerometer electronics	Procedure	None
4B	Validate Vibration Test Complex Using EM or PTM Spacecraft The engineering model or PTM spacecraft will be utilized to verify the vibration test cabling and EOSE.	Vibration test EOSE, vibration test cables	Procedure	None

Functional Flow
Drawing Title and No. Flight Model Spacecraft Flight Approval Testing   Revision ______ Date ______ Approval ______   Page No. 2

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
5	<u>Mate the Planetary Vehicle to the Vibration Fixture</u>	Vibration fixture, slings	None	Crane with hook height of ______
6	<u>Perform Voyager Planetary Vehicle Vibration Test</u> The purpose of the vibration test is to demonstrate the capability of the planetary vehicle to withstand the mission vibration environments as specified in the Voyager mission environmental specification. It is expected that these environments will consist of low frequency sinusoid and random inputs that could occur during the launch boost phase and the spacecraft retropropulsion phase of the mission sequence. The vibration test will be performed as follows: a. Calibrate accelerometers b. Start vibrating spacecraft and search for mechanical resonances and amplifications c. Perform frequency vibration test d. Perform random vibration test e. Repeat items b through d for each axis Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels will be filled to flight specifications.	Complete set of EOSE vibration tables, vibration transducers and recorders, pressurization console, fueling consoles	Procedure	Vibration fixtures, vibration tables
7	<u>Perform Spacecraft Alignments</u> After the vibration test has been completed, all spacecraft alignments will be checked as follows: a. Monopropellant motor alignment b. Gyro alignments c. Sun sensor alignments d. Canopus sensor alignments e. Gas jet alignments f. High-gain antenna alignments	Alignment sets, autocollimators	Procedure	None

Functional Flow
Drawing Title and No. Flight Model Spacecraft Flight Approval Testing   Revision ______ Date ______ Approval ______ Page No. 3

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	g. High-gain antenna latch alignments h. Omni antenna alignments i. Omni antenna boom latch alignments j. Magnetometer experiment alignments k. Magnetometer boom latch alignments l. Planetary vehicle vertical alignments			
8	Perform Appendage Deployment Test After the vibration test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure or misalignment.	Systems test set EOSE, deployment fixtures	None	None
9	Perform Leak Test After the vibration test has been completed, the SCS pneumatic system and the monopropellant engine system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
10A	Perform Integrated Systems Test The integrated systems test will be performed at the conclusion of the vibration test. The purpose of the integrated systems test is to ascertain that there has been no degradation in the Voyager over-all spacecraft subsystems due to vibration testing.	Complete set of systems EOSE and cabling	Procedure	Electrical outlets
10B	Validate Space Simulation Test Complex Using EM Spacecraft Concurrently, while the integrated systems test is being conducted, the engineering model spacecraft will be utilized to verify the space-simulation test cabling, EOSE and mechanical fixtures.	Complete set of systems EOSE and cables, EM model spacecraft	Procedure	Electrical outlets

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
11	<u>Perform Space Simulation Test Preparations</u> The space simulation preparations consist of the following tasks: a. Install heaters in the planetary vehicle b. Install thermocouples in the planetary vehicle c. Installation of the planetary vehicle into the simulation fixture d. Functional test as a final verification of the space simulation electrical complex and mechanical MOSE.	Sun source, Canopus source, heaters, thermocouple standard solar cells, gas actuator monitoring EOSE	Procedure	Vacuum chamber, electrical outlets for EOSE
12	<u>Perform Planetary Vehicle Space Simulation Testing</u> The spacecraft simulation testing will be performed as follows: a. When the proper pressure has been reached, the vacuum chamber cold walls will be turned on and the spacecraft allowed to temperature soak b. When the spacecraft has reached the temperature that would be expected during the spacecraft separation portion of the mission sequence, the spacecraft sun acquisition mode will be initiated. c. After the SCS sun acquisition testing has been completed, the solar array testing sequence will commence. The solar array testing phase will consist of the following: 1) The sun simulator output intensity and dispersion will be determined by using standard solar cells 2) The planetary vehicle solar array output will be monitored to determine that the solar array output performance meets specification 3) The primary power charge control subsystem will be exercised and the performance will be monitored for proper operation. For each charge rate the following relationship must hold: solar array current = shunt regulator current + bus voltage + battery current.	Sun source, Canopus source, heaters, thermocouple standard solar cells, gas actuator monitoring EOSE	Procedure	None

Functional Flow Drawing Title and No. Flight Model Spacecraft Flight Approval Testing Revision ____ Date ____ Approval ____ Page No. 5

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. Following the solar array testing phase of the space simulation test, the Canopus acquisition tests will start. The ability of the Canopus sensor and associated electronics to perform to specification will be monitored. e. After Canopus has been acquired, the cruise science will be turned on and the ability to perform to specifications will be monitored. f. The next event to be checked out in the flight sequence of events will be the midcourse maneuvering sequence. The spacecraft turn maneuvers will be performed in each axis in each direction. The midcourse correction engine jet vane angles will be commanded and checked in each direction. The motor burn time will correspond to the maximum burn time that can be commanded to the spacecraft. The ability to perform to specifications of the midcourse sequencing will be monitored. g. It should be mentioned that the SCS and the midcourse correction engine leak testing will take place throughout the space simulation test. h. Post midcourse maneuver cruise mode testing. The cruise mode testing mode is as follows: 1) Sun acquisition established 2) Canopus acquisition established 3) Spacecraft powered from the sun simulation source 4) All cruise science on 5) The RF up and down link (coherent) operation established All subsystem performance data will be monitored to ascertain that the Voyager planetary vehicle performs within specified limits.			
13	<u>Perform High Temperature Test</u> The cold walls will be turned off and the spacecraft temperature allowed to rise to upper specification limit. When the spacecraft has reached its upper limits, each subsystem will be exercised and monitored for proper operation.	None	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
14	Remove Spacecraft From Vacuum Chamber	Slings, spacecraft handling fixture	Procedure	Crane with hook height of ______
15	Check All Spacecraft Alignments All spacecraft alignments will be checked for shifts due to thermal effects. Listed below are the spacecraft alignments that will be checked: a. Monopropellant motor alignment b. Gyro alignments c. Sun sensor alignments d. Canopus sensor alignments e. Gas jet alignments f. High-gain antenna alignments g. High-gain antenna latch alignments h. Omni antenna alignments i. Omni antenna boom latch alignments j. Magnetometer experiment alignments k. Magnetometer boom latch alignments l. Planetary vehicle vertical alignments	Complete compliment of alignment sets, autocollimators	Procedure	Bench marks
16	Perform Appendage Deplotment Test After the vibration test has been completed in each axis, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test EOSE, deployment fixtures	Procedure	None
17	Perform SCS and Propulsion Pneumatic Leak Test The stabilization and control subsystem and the monopropellant propulsion engine subsystem will be tested for leaks that may have been encountered during vibration testing.	SCS leak test console, midcourse motor leak test console	Procedure	None

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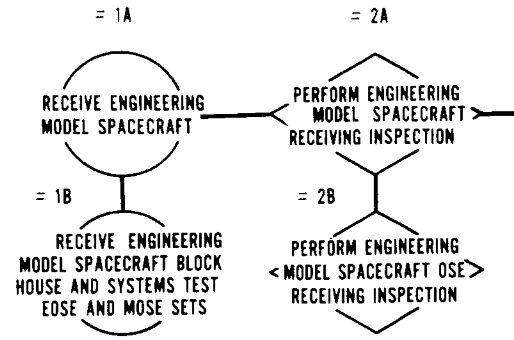
Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
18	<u>Perform IST and Critique</u> The integrated system test will be performed to verify that the over-all spacecraft and all of its subsystems have successfully survived the vibration test.	Complete set of systems test EOSE	Procedure	Electrical outlets for EOSE
19	<u>Perform Flight Spacecraft Shipping Preparations</u> The spacecraft shipping preparations will include loading both the spacecraft and the systems test set EOSE into the shipping containers. After the spacecraft and the EOSE have been packed, each shipping container will be purged with dry nitrogen.	Slings, handling fixtures, shipping containers, purging equipment	Procedure	Crane with hook height of ______
20	<u>Ship Flight Spacecraft</u>	Slings, handling fixtures, purging equipment	Procedure	Crane with hook height of ______



NOTE: EM MODEL DAT
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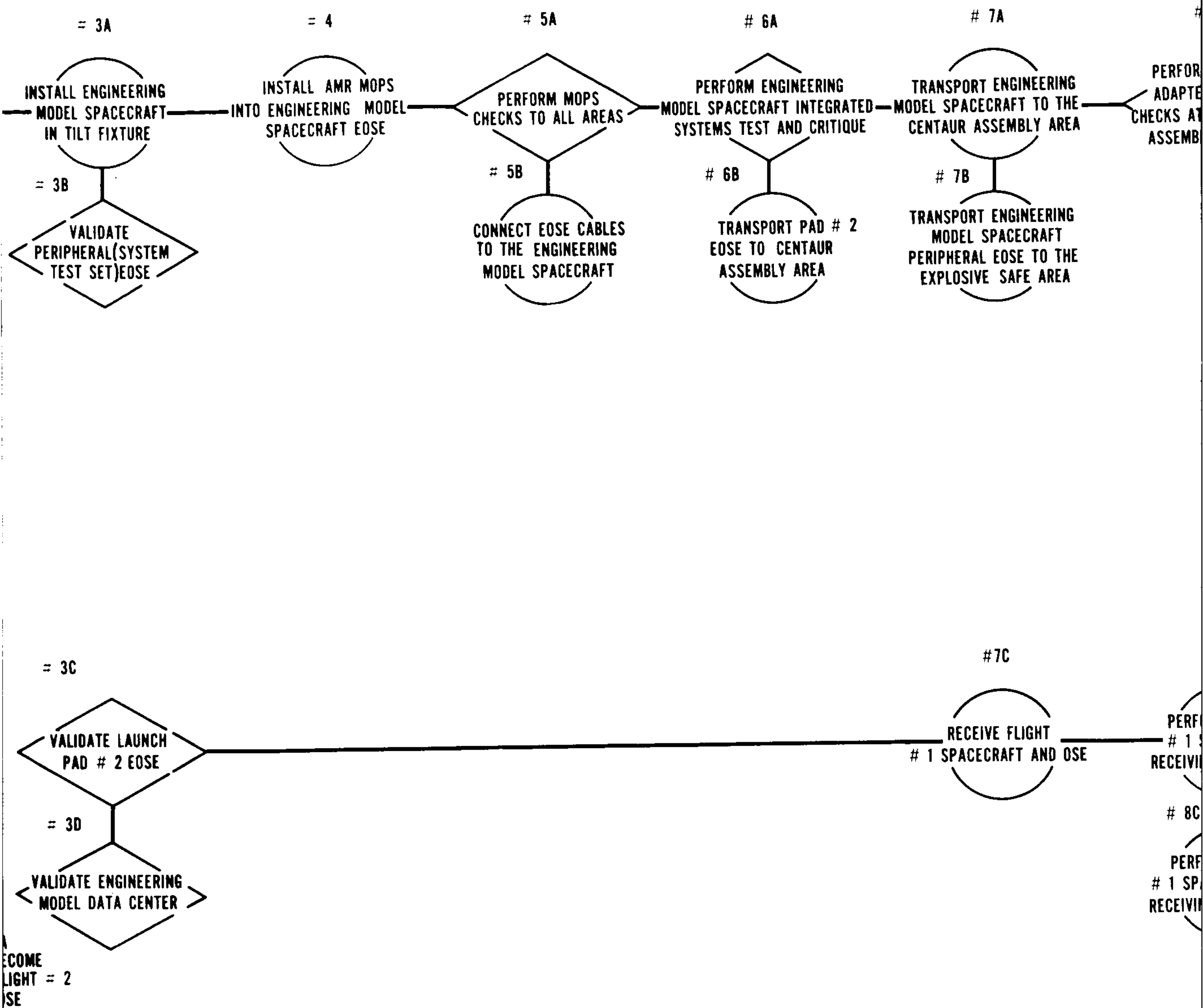
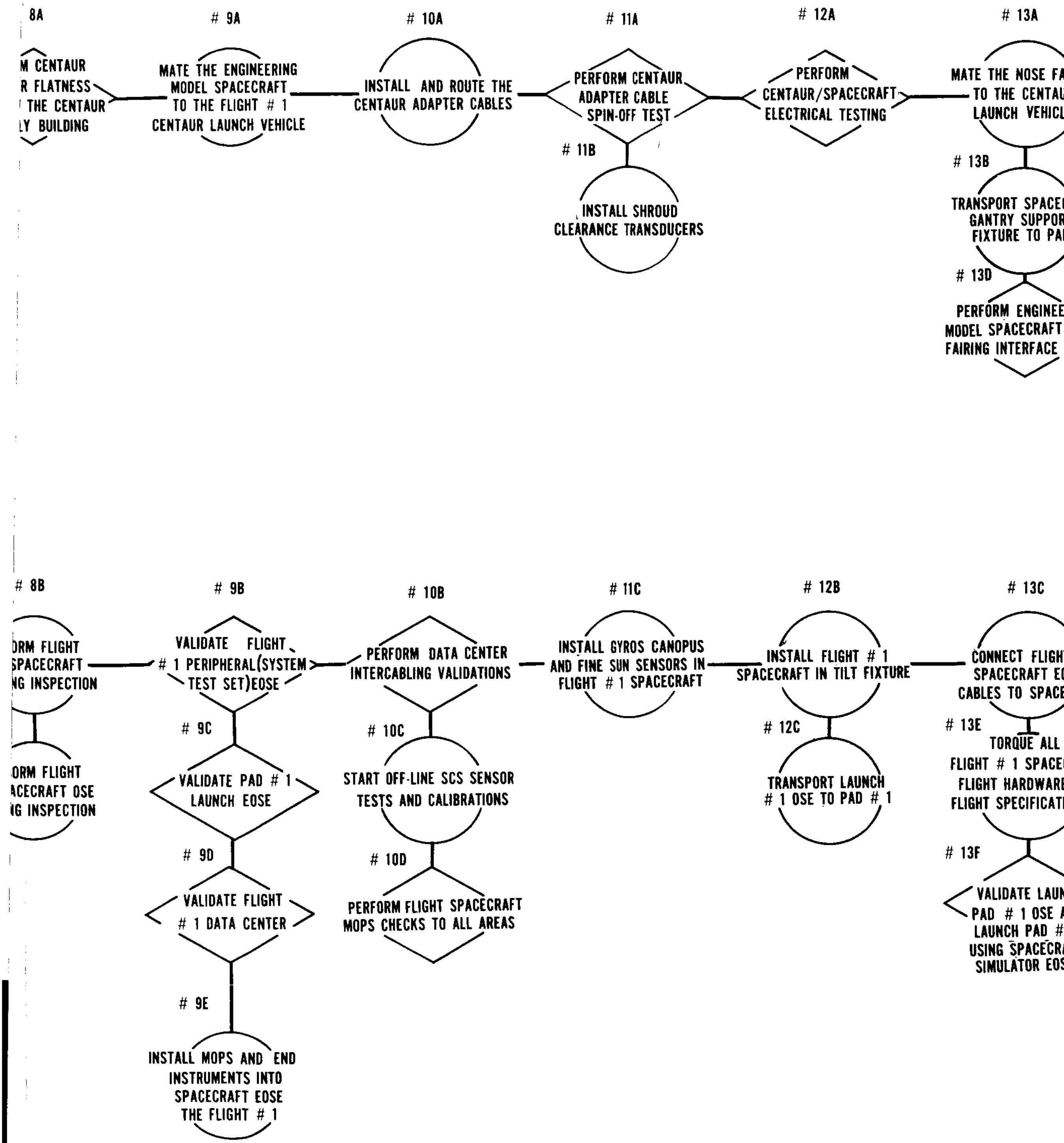
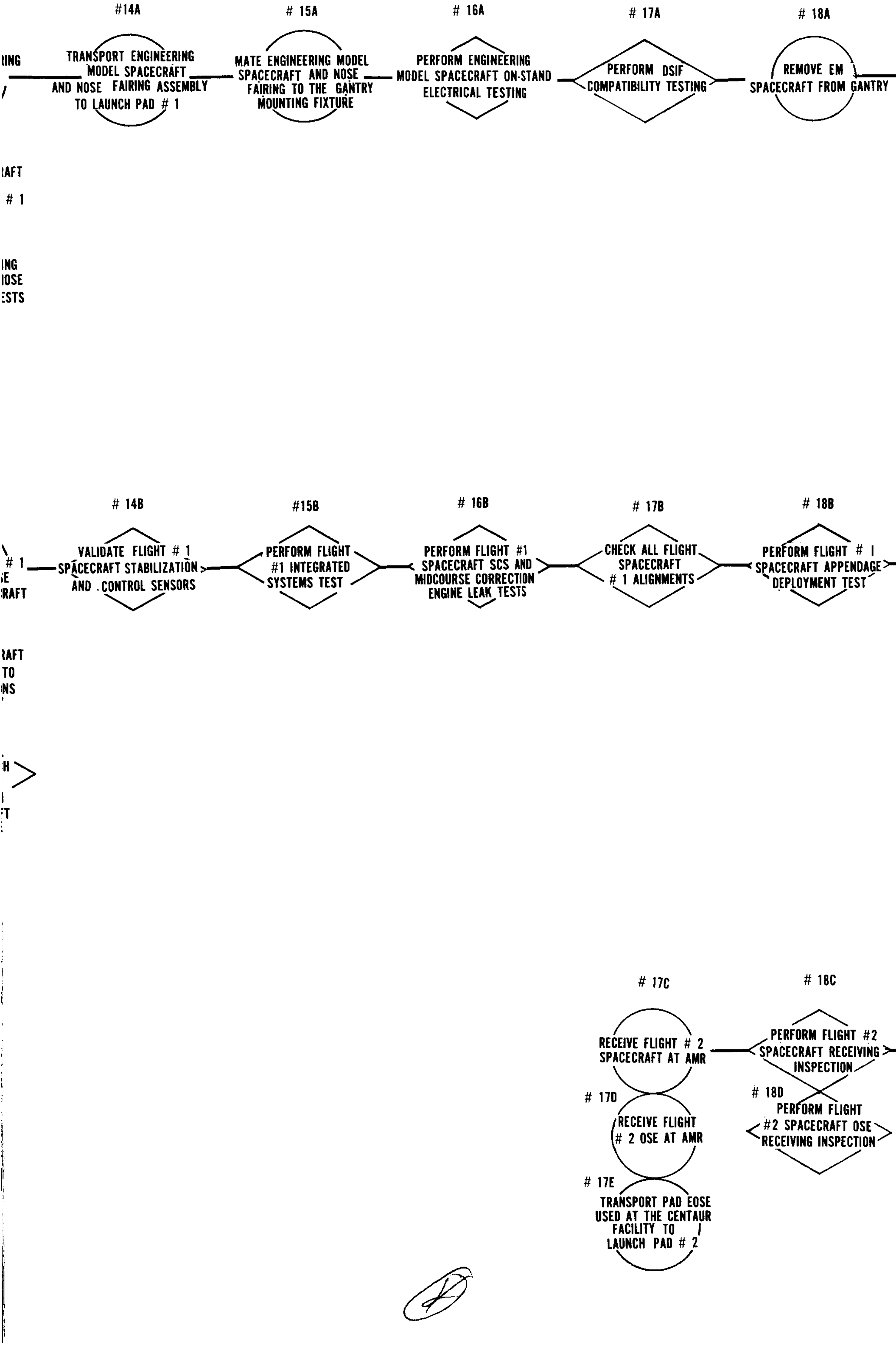
SPACECRAFT E

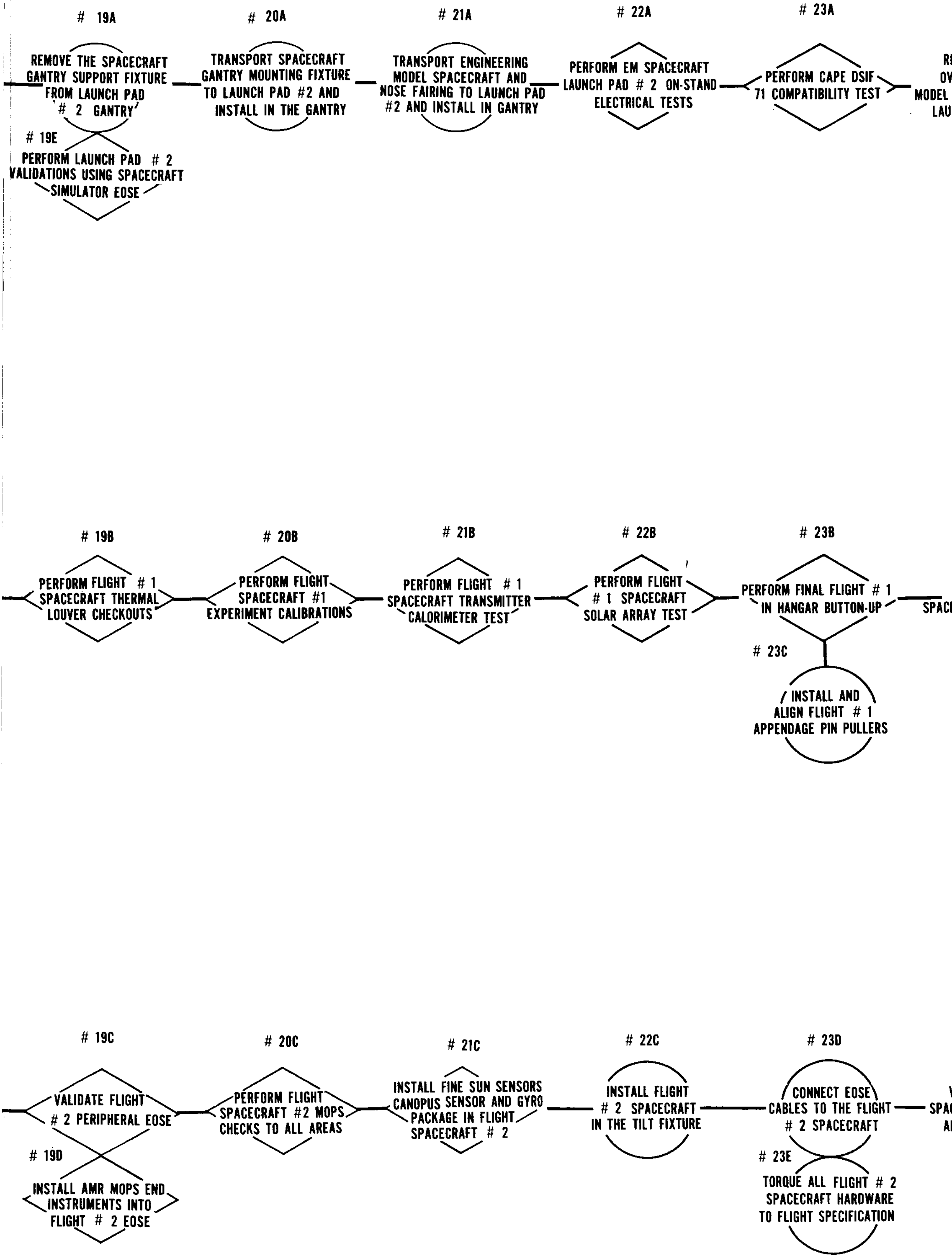


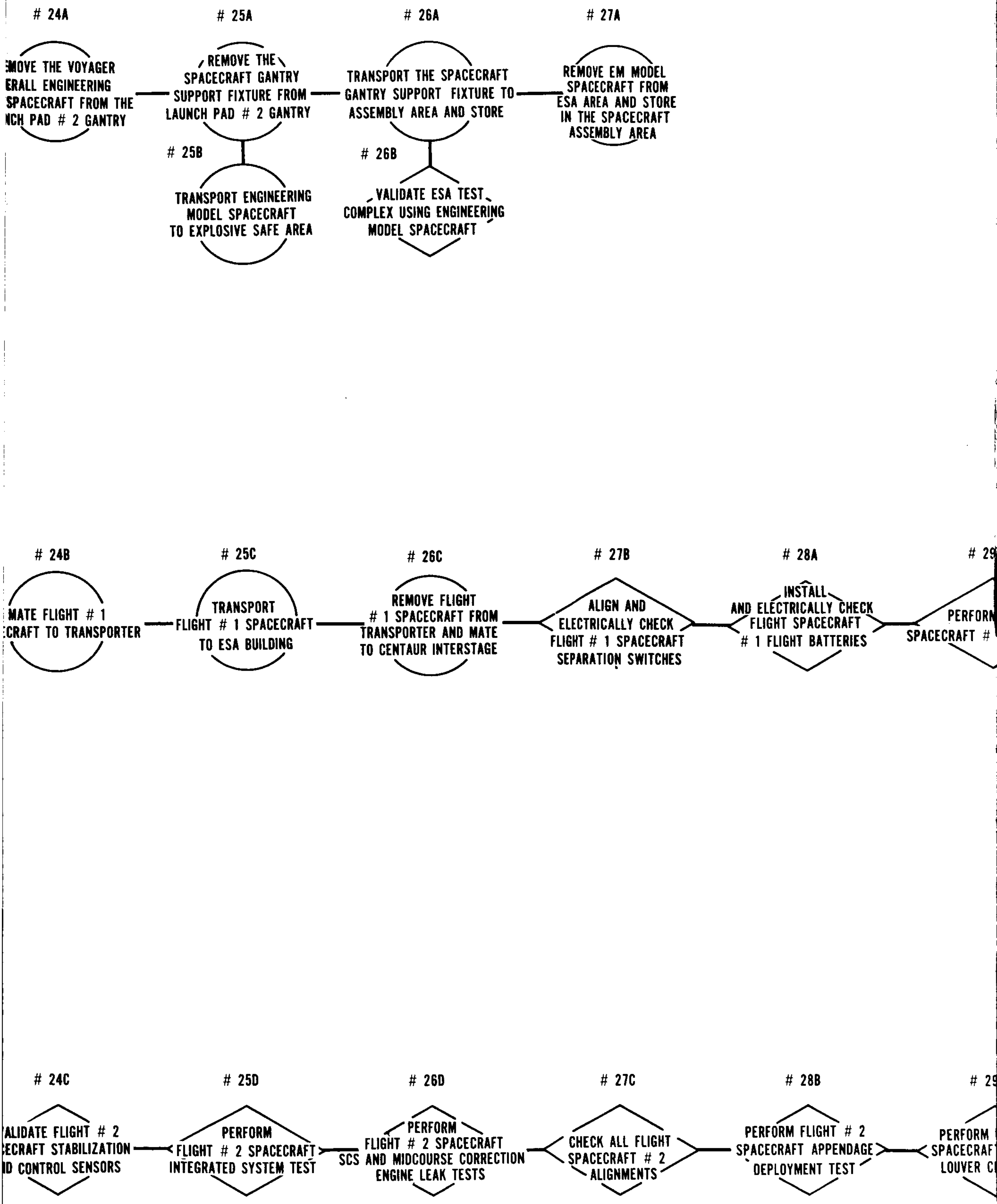
= 3A
INSTALL ENGINEERING MODEL SPACECRAFT IN TILT FIXTURE
= 3B
VALIDATE PERIPHERAL(SYSTEM TEST SET)EOSE
= 4
INSTALL AMR MOPS INTO ENGINEERING MODEL SPACECRAFT EOSE
5A
PERFORM MOPS CHECKS TO ALL AREAS
5B
CONNECT EOSE CABLES TO THE ENGINEERING MODEL SPACECRAFT
6A
PERFORM ENGINEERING MODEL SPACECRAFT INTEGRATED SYSTEMS TEST AND CRITIQUE
6B
TRANSPORT PAD # 2 EOSE TO CENTAUR ASSEMBLY AREA
7A
TRANSPORT ENGINEERING MODEL SPACECRAFT TO THE CENTAUR ASSEMBLY AREA
7B
TRANSPORT ENGINEERING MODEL SPACECRAFT PERIPHERAL EOSE TO THE EXPLOSIVE SAFE AREA
= 3C
VALIDATE LAUNCH PAD # 2 EOSE
= 3D
VALIDATE ENGINEERING MODEL DATA CENTER
#7C
RECEIVE FLIGHT # 1 SPACECRAFT AND OSE
8C


8A
M CENTAUR R FLATNESS THE CENTAUR LY BUILDING
9A
MATE THE ENGINEERING MODEL SPACECRAFT TO THE FLIGHT # 1 CENTAUR LAUNCH VEHICLE
10A
INSTALL AND ROUTE THE CENTAUR ADAPTER CABLES
11A
PERFORM CENTAUR ADAPTER CABLE SPIN-OFF TEST
11B
INSTALL SHROUD CLEARANCE TRANSDUCERS
12A
PERFORM CENTAUR/SPACECRAFT ELECTRICAL TESTING
13A
MATE THE NOSE FA TO THE CENTAU LAUNCH VEHICL
13B
TRANSPORT SPACE GANTRY SUPPOR FIXTURE TO PA
13D
PERFORM ENGINEE MODEL SPACECRAFT FAIRING INTERFACE
8B
ORM FLIGHT SPACECRAFT NG INSPECTION
ORM FLIGHT ACECRAFT OSE IG INSPECTION
9B
VALIDATE FLIGHT # 1 PERIPHERAL (SYSTEM TEST SET) EOSE
9C
VALIDATE PAD # 1 LAUNCH EOSE
9D
VALIDATE FLIGHT # 1 DATA CENTER
9E
INSTALL MOPS AND END INSTRUMENTS INTO SPACECRAFT EOSE THE FLIGHT # 1
10B
PERFORM DATA CENTER INTERCABLING VALIDATIONS
10C
START OFF-LINE SCS SENSOR TESTS AND CALIBRATIONS
10D
PERFORM FLIGHT SPACECRAFT MOPS CHECKS TO ALL AREAS
11C
INSTALL GYROS CANOPUS AND FINE SUN SENSORS IN FLIGHT # 1 SPACECRAFT
12B
INSTALL FLIGHT # 1 SPACECRAFT IN TILT FIXTURE
12C
TRANSPORT LAUNCH # 1 OSE TO PAD # 1
13C
CONNECT FLIGH SPACECRAFT E CABLES TO SPACE
13E
TORQUE ALL FLIGHT # 1 SPACE FLIGHT HARDWARE FLIGHT SPECIFICAT
13F
VALIDATE LAUN PAD # 1 OSE LAUNCH PAD # USING SPACECR SIMULATOR EO



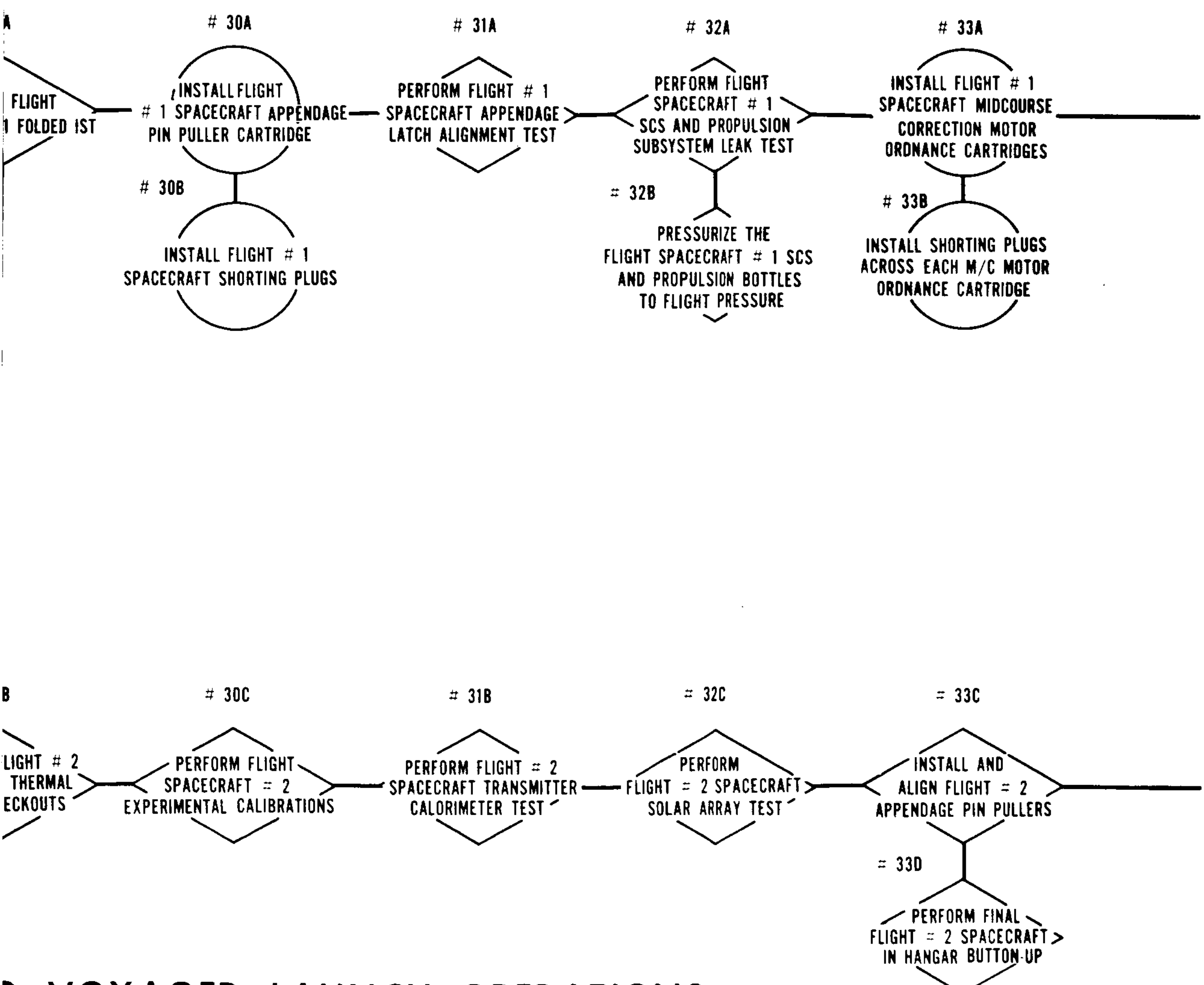
#14A
TRANSPORT ENGINEERING MODEL SPACECRAFT AND NOSE FAIRING ASSEMBLY TO LAUNCH PAD # 1
15A
MATE ENGINEERING MODEL SPACECRAFT AND NOSE FAIRING TO THE GANTRY MOUNTING FIXTURE
16A
PERFORM ENGINEERING MODEL SPACECRAFT ON-STAND ELECTRICAL TESTING
17A
PERFORM DSIF COMPATIBILITY TESTING
18A
REMOVE EM SPACECRAFT FROM GANTRY
14B
VALIDATE FLIGHT # 1 SPACECRAFT STABILIZATION AND CONTROL SENSORS
#15B
PERFORM FLIGHT #1 INTEGRATED SYSTEMS TEST
16B
PERFORM FLIGHT #1 SPACECRAFT SCS AND MIDCOURSE CORRECTION ENGINE LEAK TESTS
17B
CHECK ALL FLIGHT SPACECRAFT # 1 ALIGNMENTS
18B
PERFORM FLIGHT # 1 SPACECRAFT APPENDAGE DEPLOYMENT TEST
17C
RECEIVE FLIGHT # 2 SPACECRAFT AT AMR
18C
PERFORM FLIGHT #2 SPACECRAFT RECEIVING INSPECTION
17D
RECEIVE FLIGHT # 2 OSE AT AMR
18D
PERFORM FLIGHT #2 SPACECRAFT OSE RECEIVING INSPECTION
17E
TRANSPORT PAD EOSE USED AT THE CENTAUR FACILITY TO LAUNCH PAD # 2


19A
REMOVE THE SPACECRAFT GANTRY SUPPORT FIXTURE FROM LAUNCH PAD # 2 GANTRY
19E
PERFORM LAUNCH PAD # 2 VALIDATIONS USING SPACECRAFT SIMULATOR EOSE
20A
TRANSPORT SPACECRAFT GANTRY MOUNTING FIXTURE TO LAUNCH PAD #2 AND INSTALL IN THE GANTRY
21A
TRANSPORT ENGINEERING MODEL SPACECRAFT AND NOSE FAIRING TO LAUNCH PAD #2 AND INSTALL IN GANTRY
22A
PERFORM EM SPACECRAFT LAUNCH PAD # 2 ON-STAND ELECTRICAL TESTS
23A
PERFORM CAPE DSIF 71 COMPATIBILITY TEST
19B
PERFORM FLIGHT # 1 SPACECRAFT THERMAL LOUVER CHECKOUTS
20B
PERFORM FLIGHT SPACECRAFT #1 EXPERIMENT CALIBRATIONS
21B
PERFORM FLIGHT # 1 SPACECRAFT TRANSMITTER CALORIMETER TEST
22B
PERFORM FLIGHT # 1 SPACECRAFT SOLAR ARRAY TEST
23B
PERFORM FINAL FLIGHT # 1 IN HANGAR BUTTON-UP
23C
INSTALL AND ALIGN FLIGHT # 1 APPENDAGE PIN PULLERS
19C
VALIDATE FLIGHT # 2 PERIPHERAL EOSE
19D
INSTALL AMR MOPS END INSTRUMENTS INTO FLIGHT # 2 EOSE
20C
PERFORM FLIGHT SPACECRAFT #2 MOPS CHECKS TO ALL AREAS
21C
INSTALL FINE SUN SENSORS CANOPUS SENSOR AND GYRO PACKAGE IN FLIGHT SPACECRAFT # 2
22C
INSTALL FLIGHT # 2 SPACECRAFT IN THE TILT FIXTURE
23D
CONNECT EOSE CABLES TO THE FLIGHT # 2 SPACECRAFT
23E
TORQUE ALL FLIGHT # 2 SPACECRAFT HARDWARE TO FLIGHT SPECIFICATION



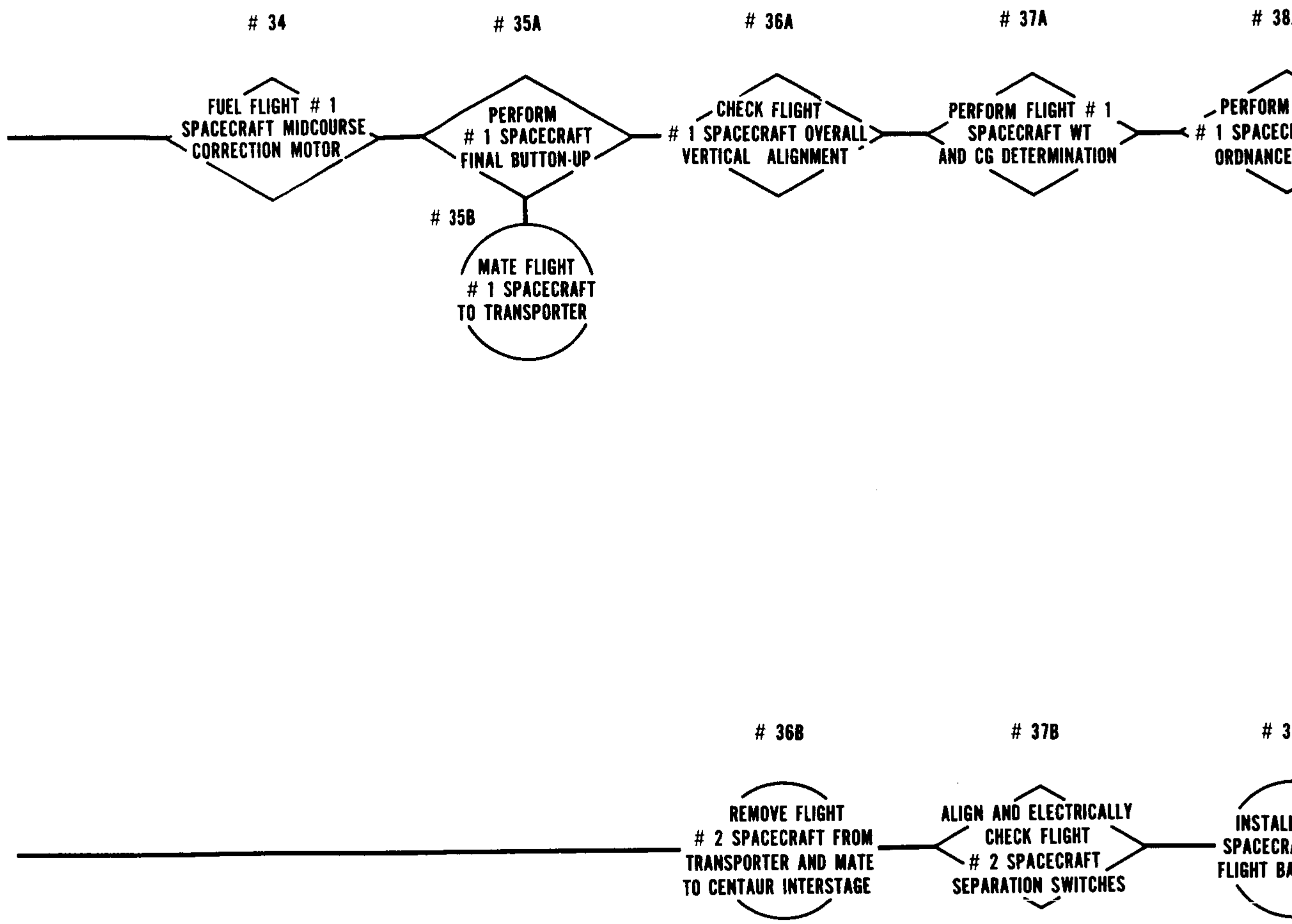
24A
EMOVE THE VOYAGER ERALL ENGINEERING SPACECRAFT FROM THE NCH PAD # 2 GANTRY
25A
REMOVE THE SPACECRAFT GANTRY SUPPORT FIXTURE FROM LAUNCH PAD # 2 GANTRY
25B
TRANSPORT ENGINEERING MODEL SPACECRAFT TO EXPLOSIVE SAFE AREA
26A
TRANSPORT THE SPACECRAFT GANTRY SUPPORT FIXTURE TO ASSEMBLY AREA AND STORE
26B
VALIDATE ESA TEST COMPLEX USING ENGINEERING MODEL SPACECRAFT
27A
REMOVE EM MODEL SPACECRAFT FROM ESA AREA AND STORE IN THE SPACECRAFT ASSEMBLY AREA
24B
MATE FLIGHT # 1 CRAFT TO TRANSPORTER
25C
TRANSPORT FLIGHT # 1 SPACECRAFT TO ESA BUILDING
26C
REMOVE FLIGHT # 1 SPACECRAFT FROM TRANSPORTER AND MATE TO CENTAUR INTERSTAGE
27B
ALIGN AND ELECTRICALLY CHECK FLIGHT # 1 SPACECRAFT SEPARATION SWITCHES
28A
INSTALL AND ELECTRICALLY CHECK FLIGHT SPACECRAFT # 1 FLIGHT BATTERIES
29
PERFORM SPACECRAFT #
24C
ALIDATE FLIGHT # 2 ECRAFT STABILIZATION D CONTROL SENSORS
25D
PERFORM FLIGHT # 2 SPACECRAFT INTEGRATED SYSTEM TEST
26D
PERFORM FLIGHT # 2 SPACECRAFT SCS AND MIDCOURSE CORRECTION ENGINE LEAK TESTS
27C
CHECK ALL FLIGHT SPACECRAFT # 2 ALIGNMENTS
28B
PERFORM FLIGHT # 2 SPACECRAFT APPENDAGE DEPLOYMENT TEST
2
PERFORM SPACECRAF LOUVER C

6



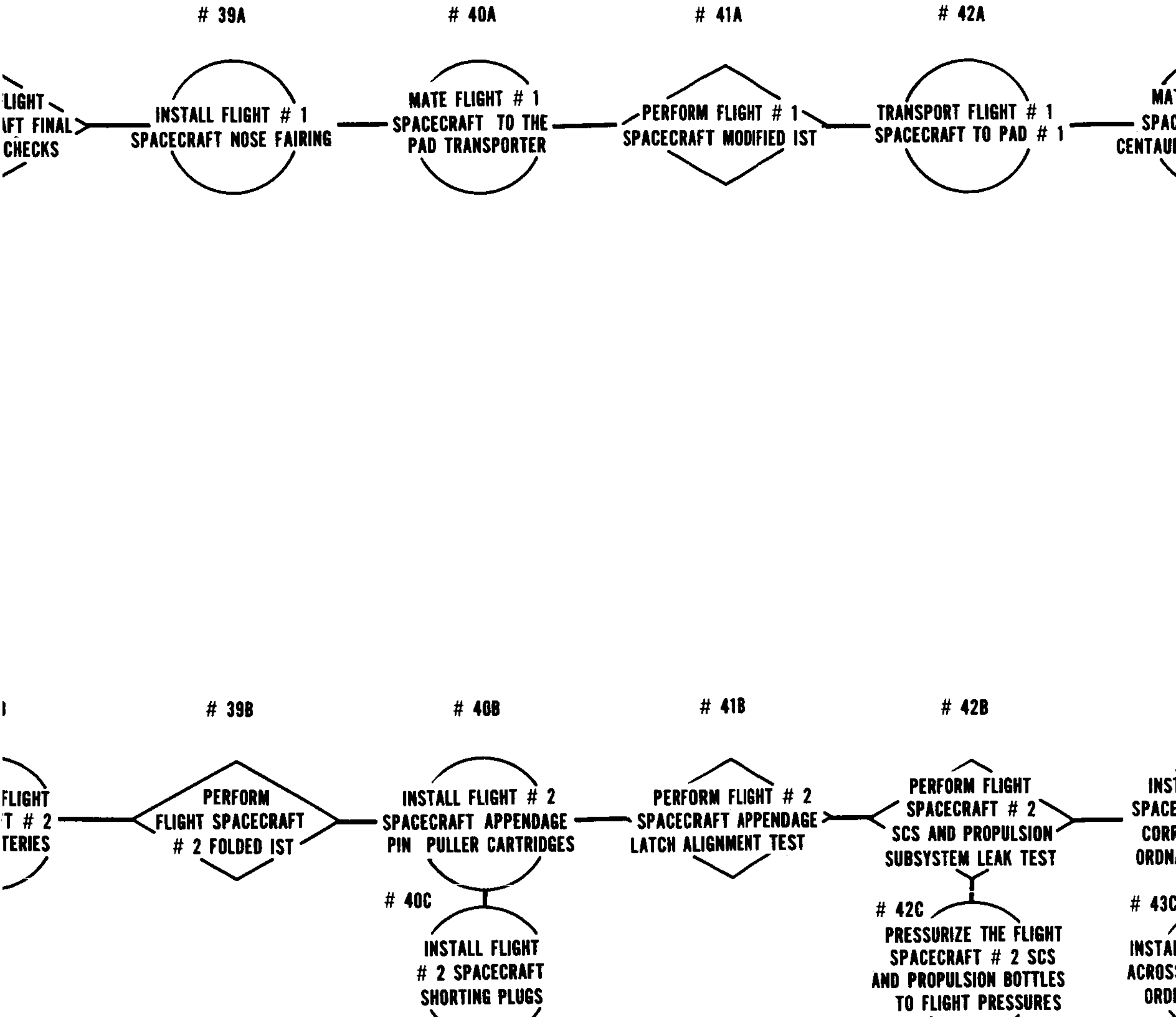
9 VOYAGER LAUNCH OPERATIONS

7



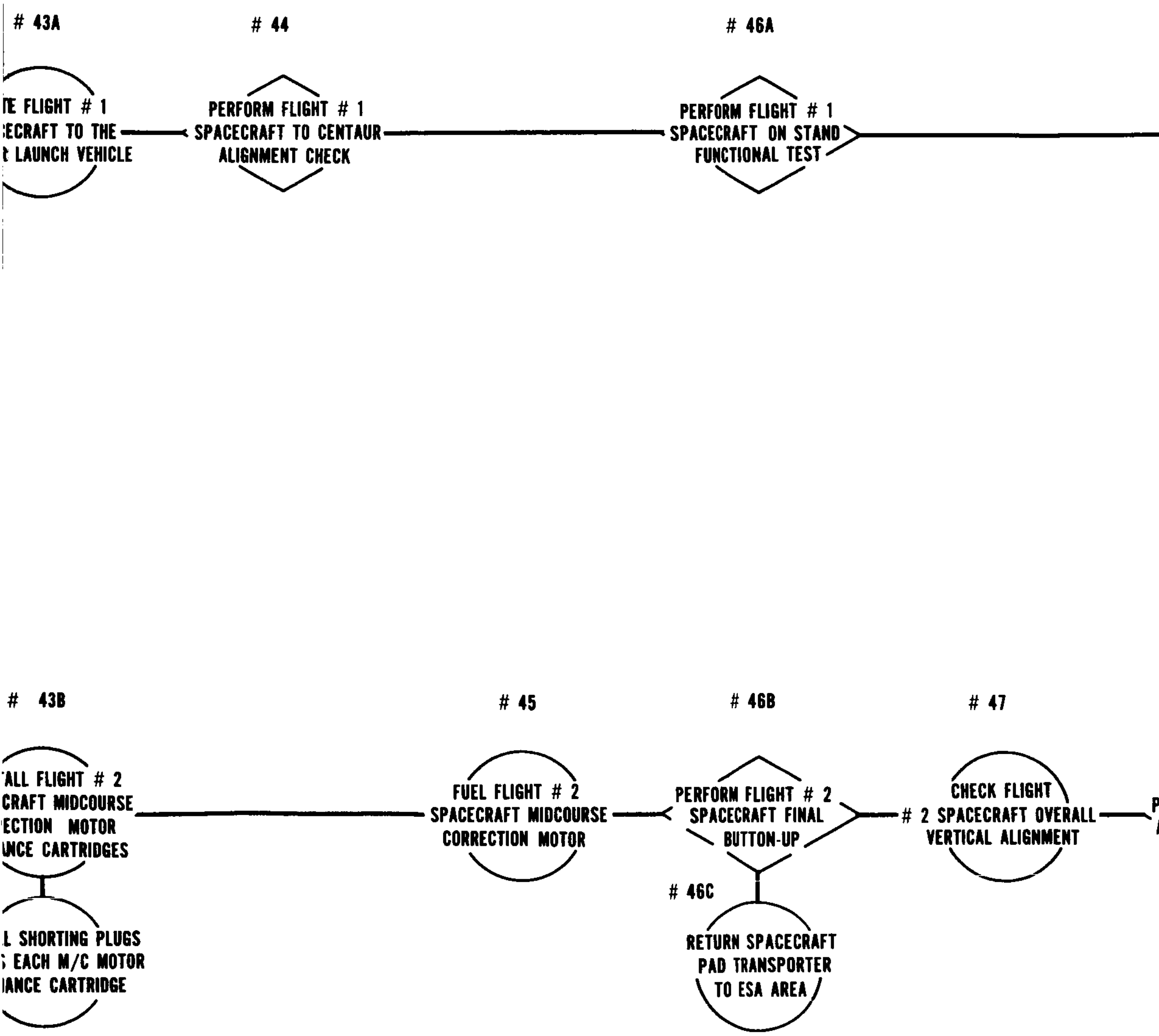
34
FUEL FLIGHT # 1 SPACECRAFT MIDCOURSE CORRECTION MOTOR
35A
PERFORM # 1 SPACECRAFT FINAL BUTTON-UP
35B
MATE FLIGHT # 1 SPACECRAFT TO TRANSPORTER
36A
CHECK FLIGHT # 1 SPACECRAFT OVERALL VERTICAL ALIGNMENT
37A
PERFORM FLIGHT # 1 SPACECRAFT WT AND CG DETERMINATION
PERFORM
ORDNANCE
36B
REMOVE FLIGHT # 2 SPACECRAFT FROM TRANSPORTER AND MATE TO CENTAUR INTERSTAGE
37B
ALIGN AND ELECTRICALLY CHECK FLIGHT # 2 SPACECRAFT SEPARATION SWITCHES

1



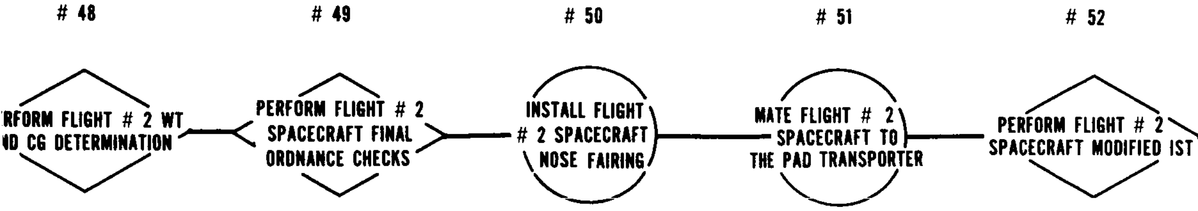
39A
INSTALL FLIGHT # 1 SPACECRAFT NOSE FAIRING
40A
MATE FLIGHT # 1 SPACECRAFT TO THE PAD TRANSPORTER
41A
PERFORM FLIGHT # 1 SPACECRAFT MODIFIED IST
42A
TRANSPORT FLIGHT # 1 SPACECRAFT TO PAD # 1
39B
PERFORM FLIGHT SPACECRAFT # 2 FOLDED IST
40B
INSTALL FLIGHT # 2 SPACECRAFT APPENDAGE PIN PULLER CARTRIDGES
40C
INSTALL FLIGHT # 2 SPACECRAFT SHORTING PLUGS
41B
PERFORM FLIGHT # 2 SPACECRAFT APPENDAGE LATCH ALIGNMENT TEST
42B
PERFORM FLIGHT SPACECRAFT # 2 SCS AND PROPULSION SUBSYSTEM LEAK TEST
42C
PRESSURIZE THE FLIGHT SPACECRAFT # 2 SCS AND PROPULSION BOTTLES TO FLIGHT PRESSURES

2



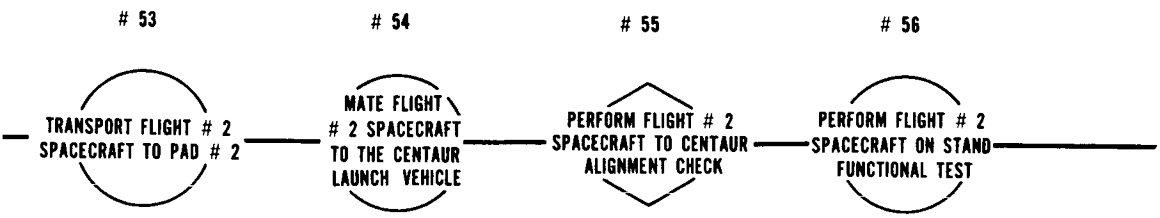
43A
TE FLIGHT # 1
ECRAFT TO THE
LAUNCH VEHICLE
44
PERFORM FLIGHT # 1
SPACECRAFT TO CENTAUR
ALIGNMENT CHECK
46A
PERFORM FLIGHT # 1
SPACECRAFT ON STAND
FUNCTIONAL TEST
43B
ALL FLIGHT # 2
CRAFT MIDCOURSE
ECTION MOTOR
NCE CARTRIDGES
L SHORTING PLUGS
EACH M/C MOTOR
NCE CARTRIDGE
45
FUEL FLIGHT # 2
SPACECRAFT MIDCOURSE
CORRECTION MOTOR
46B
PERFORM FLIGHT # 2
SPACECRAFT FINAL
BUTTON-UP
46C
RETURN SPACECRAFT
PAD TRANSPORTER
TO ESA AREA
47
CHECK FLIGHT
2 SPACECRAFT OVERALL
VERTICAL ALIGNMENT

3

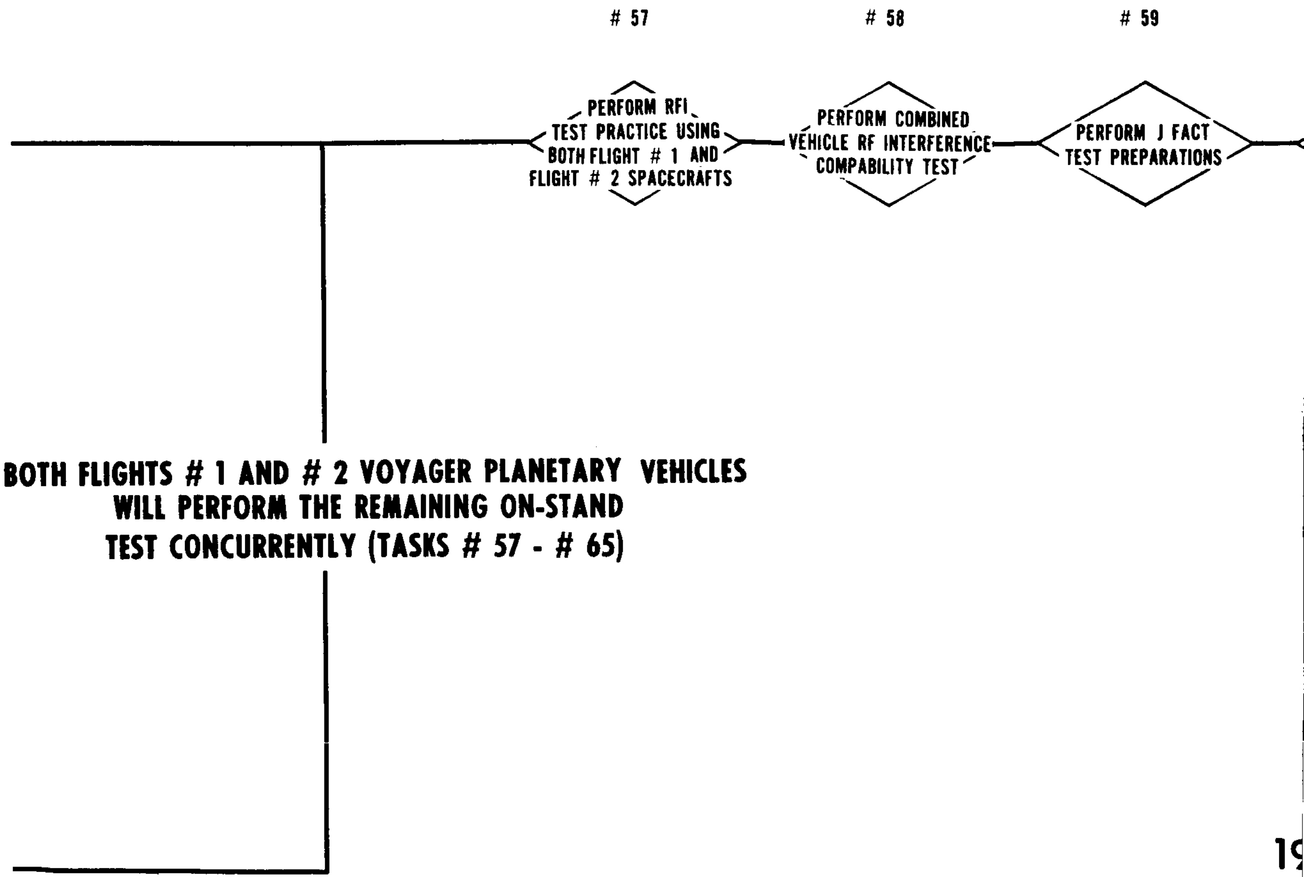


48
RFORM FLIGHT # 2 WT
ND CG DETERMINATION
49
PERFORM FLIGHT # 2
SPACECRAFT FINAL
ORDNANCE CHECKS
50
INSTALL FLIGHT
2 SPACECRAFT
NOSE FAIRING
51
MATE FLIGHT # 2
SPACECRAFT TO
THE PAD TRANSPORTER
52
PERFORM FLIGHT # 2
SPACECRAFT MODIFIED IST

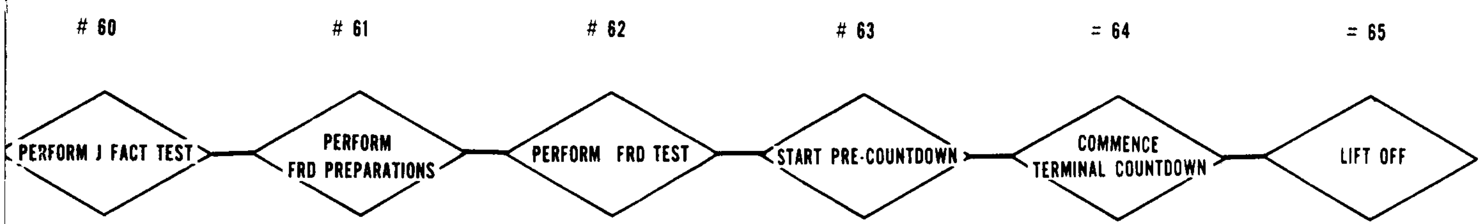
3



1969 VOYAGER LAUNCH OPERATIONS



57
58
59
PERFORM RFI TEST PRACTICE USING BOTH FLIGHT # 1 AND FLIGHT # 2 SPACECRAFTS
PERFORM COMBINED VEHICLE RF INTERFERENCE COMPABILITY TEST
PERFORM J FACT TEST PREPARATIONS
BOTH FLIGHTS # 1 AND # 2 VOYAGER PLANETARY VEHICLES WILL PERFORM THE REMAINING ON-STAND TEST CONCURRENTLY (TASKS # 57 - # 65)



969 VOYAGER LAUNCH OPERATIONS

2

Functional Flow
Drawing Title and No. Launch Operations   Revision ____ Date ____ Approval ____ Page No. 1

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1A	Receive Engineering Model Spacecraft	Slings, handling fixtures, transporter	Procedure	None
1B	Receive Engineering Model Spacecraft Blockhouse and Systems Test EOSE and MOSE Sets Both the spacecraft and the OSE will be delivered to the skid strip by air. From the skid strip the spacecraft and EOSE will be transported to the hangar.	Slings, handling fixtures, transporter	Procedure	None
2A	Perform Engineering Model Spacecraft Receiving Inspection	None	None	None
2B	Perform Engineering Model Spacecraft OSE Receiving Inspection Receiving inspection will be held for damage that might have been incurred during shipping and handling operations.	None	None	None
3A	Install Engineering Model Spacecraft in Tilt Fixture The spacecraft is to be installed in the tilt fixture in preparation for the integrated systems test.	Hand tools		
3B	Validate Peripheral (System Test Set) EOSE	Peripheral EOSE validation test set	None	None
3C	Validate Launch Pad No. 2 EOSE	Spacecraft simulator	Procedure	None
3D	Validate Engineering Model Data Center The peripheral EOSE and the data center will be validated in preparation for the integrated systems test.	Computer validation tapes, data center validation test set		

Functional Flow Drawing Title and No. Launch Operations Revision ______ Date ______ Approval ______ Page No. 2

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
4	Install AMR MOPS End Instruments into Engineering Spacecraft EOSE	MOPS end instruments	None	None
5A	Perform MOPS Checks to all Areas The AMR intercommunication net will be checked by contacting each Voyager station using the MOPS end instrument selector switch. Each end instrument in each area will be checked in this manner.	None	List of MOP channel assignments	None
5B	Connect EOSE Cables to the Engineering Model Spacecraft The EOSE cables will be connected to the spacecraft in preparation for the integrated systems test.	None	Procedure	None
6A	Perform Engineering Model Spacecraft Integrated Systems Test and Critique The engineering model spacecraft integrated systems test will be performed to verify that the spacecraft and all of its subsystems have successfully survived the shipping and handling operations.	Complete set of systems test EOSE	Procedure	None
6B	Transport Launch Pad No. 2 EOSE to Centaur Assembly Area The pad EOSE will be shipped to the Centaur assembly building and utilized to checkout the Centaur/Voyager spacecraft electrical interfaces.	Slings, EOSE handling fixtures, transporters	Procedure	None
7A	Transport Engineering Model Spacecraft to the Centaur Assembly Area	Slings, EOSE handling fixtures, transporters	Procedure	Adequate door width to get spacecraft through

Functional Flow
Drawing Title and No Launch Operations Revision ____ Date ____ Approval ____ Page No. 3

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
7B	<u>Transport Engineering Model Spacecraft Peripheral EOSE to the Explosive Safe Area</u>	Slings, EOSE handling fixtures, transporters	Procedure	None
7C	<u>Receive Flight No. 1 Spacecraft and OSE</u> The Flight No. 1 spacecraft, MOSE and EOSE will be delivered to the skid strip at AMR. Next the spacecraft and associated OSE will be delivered to the spacecraft assembly hangar.	Slings, OSE handling fixtures, transporters	Procedure	None
8A	<u>Perform Centaur Adapter Flatness Checks at the Centaur Assembly Building</u>	Centaur adapter alignment set	Procedure	None
8B	<u>Perform Flight No. 1 Spacecraft Receiving Inspection</u> The Flight No. 1 and 2 Centaur adapter flatness checks will be performed to ascertain that the Centaur mating surfaces and the spacecraft mating surfaces are absolutely flat and level.	None	None	None
8C	<u>Perform Flight No. 1 Spacecraft OSE Receiving Inspection</u> Concurrently, the Flight No. 1 spacecraft and OSE receiving inspection tasks will take place. The receiving inspections are performed mainly to ascertain that no damage to the spacecraft or OSE was incurred due to shipping and handling operations	None	None	None
9A	<u>Mate the Engineering Model Spacecraft to the Flight No. 1 Centaur Launch Vehicle</u>	Hand tools, torque wrenches, slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of

Functional Flow
Drawing Title and No Launch Operations Revision ______ Date ______ Approval ______ Page No. 4

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
9B	Validate Flight No. 1 Peripheral (System Test Set) EOSE	Peripheral EOSE validation set	Procedure	None
9C	Validate Pad No. 1 Launch EOSE The PTM spacecraft will be mated to the first flight Centaur vehicle in preparation for Centaur spacecraft electrical tests. Next, the Flight No. spacecraft will be mated to the tilt fixture in preparation for the Flight No. 1 integrated systems test.	Spacecraft simulator	Procedure	None
9D	Validate Flight No. 1 Data Center Concurrently, the pad EOSE for Pad No. 2, the Flight No. 1 peripheral EOSE, and the Flight No. 1 data center will be validated.	Computer validation tapes, data center validation test set	Procedure	None
9E	Install MOPS End Instruments into the Flight No. 1 Spacecraft EOSE The AMR MOPS end instruments will be installed into the Flight No. 1 EOSE and connected to the AMR intercommunications system.	None	None	None
10A	Install and Route the Centaur Adapter Cables The Centaur adapter cables will be installed at this time to support the Centaur umbilical tests.	Hand tools	Procedure	None
10B	Perform Data Center Intercabling Validations The data center intercabling validations between the engineering model and Flight No. 1 data centers will be performed in preparation for the IST.	Intercabling validation set	Procedure	Data center intercabling

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Drawing Title and No. Launch Operations Revision ______ Date ______ Approval ______ Page No. 5

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
10C	<u>Start Off-line SCS Sensor Tests and Calibrations</u> The gyros, fine sun sensors, and Canopus sensors will be bench tested in the SCS laboratory and given a final calibration.	SCS Bench Test Equipment	Procedure	SCS Laboratory
10D	<u>Perform Flight Spacecraft No. 1 MOPS Checks to All Areas</u> The AMR intercommunications net will be checked by contacting each Voyager station using the MOPS end instruments selector switch. Each end instument in each area will be checked in this manner.	None	None	None
11A	<u>Perform Centaur Adapter Cable Spin-Off Test</u> The Centaur adapter cable spin-off test will not be done live. It is merely a mechanical test to ascertain that the adapter cables at the time of separation will fall freely away from the spacecraft.	None	None	None
11B	<u>Install Shroud Clearance Transducers</u> The shroud clearance transducers will be installed on the engineering model spacecraft and connected to the electronics in preparation for the shroud interface testing.	Hand tools	Procedure	None
11C	<u>Install Gyros, Canopus and Fine Sun Sensors in Flight No. 1 Spacecraft</u> The sensors will be installed in the spacecraft in preparation for the IST test.	Hand tools, torque wrenches	Procedure	None
12A	<u>Perform Centaur/Spacecraft Electrical Testing</u> The Centaur/spacecraft electrical test will be performed as follows: a. Continuity test the Centaur adapter cabling.	Launch pad EOSE, voltmeter oscilloscope	Procedure	Electrical outlets

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Connect the adapter cabling to the EOSE and the spacecraft. c. Apply external power to the spacecraft and determine that the adapter cabling line drops are within specification. d. The spacecraft spin-off separation signals from the Centaur will be checked for no-fire conditions and all-fire conditions. e. All other umbilical signal functions will be tested for proper operation.			
12B	Install Flight No. 1 Spacecraft in Tilt Fixture	Hand tools, torque wrenches	Procedure	None
12C	Transport Launch No. 1 OSE to Pad No. 1 The Pad No. 1 OSE will be transported to Pad No. 1 in support of the launch complex testing phase.	Slings, OSE handling fixtures, transporters	Procedure	None
13A	Mate the Nose Fairing to the Centaur Launch Vehicle	Hand tools, torque wrenches, slings, shroud handling fixture	Procedure	Overhead crane with hook height of
13B	Transport Spacecraft Gantry Support Fixture to Pad No. 1.	Slings, handling fixture, transporter	Procedure	Overhead crane with hook height of
13C	Connect Flight No. 1 Spacecraft EOSE Cables to Spacecraft The EOSE cables will be connected to the spacecraft in preparation for the IST test.	None	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
13D	<u>Perform Engineering Model Spacecraft Nose Fairing Interface Tests</u> The nose fairing will be lowered over the PTM spacecraft and mated to the Centaur launch vehicle. The nose fairing interface test is comprised of two parts. a. Shroud clearance determination b. RF shroud coupler losses	Slings, shroud, handling fixture	Procedure	None
13E	<u>Torque All Flight No. 1 Spacecraft Flight Hardware to Flight Specifications</u> All flight hardware will be torqued to flight specifications as part of the button up procedure.			
13F	<u>Perform Launch Pad No. 1 Validations Using Spacecraft Simulator EOSE</u> The pad validations are comprised of the following tests: a. Determine primary power line drops between the spacecraft and the blockhouse. b. RF up and down link power loss determinations c. Electrically check all of spacecraft umbilical functions between the spacecraft and blockhouse. d. Check the wideband video pair system between the spacecraft assembly area and the spacecraft.	Spacecraft simulator, launch pad EOSE	Procedure	All Voyager Pad Modifications Completed
14A	<u>Transport Engineering Model Spacecraft and Nose Fairing to Launch Pad No. 1</u>	Slings, handling fixture, transporter	Procedure	Overhead crane with hook height of
14B	<u>Validate Flight No. 1 Spacecraft Stabilization and Control Sensors</u> The gyros, Canopus sensors, and fine sun sensors will be electrically revalidated in the spacecraft as part of the IST preparations.	System test set	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
15A	Mate Engineering Model Spacecraft and Nose Fairing to the Gantry Mounting Fixture	Slings, handling fixtures	Procedure	Overhead crane with hook height of
15B	Perform Flight No. 1 Integrated Systems Test The Flight No. 1 spacecraft integrated systems test will be performed to verify that the flight spacecraft and all of its subsystems have survived the shipping and handling operations.	Complete set of systems test EOSE	Procedure	None
16A	Perform Engineering Model Spacecraft On-Stand Electrical Testing The engineering model spacecraft on-stand testing phase is comprised of the following tests: a. Determine primary power line drops between the spacecraft and the blockhouse b. RF up and down link power loss determination c. Electrically check all of spacecraft umbilical functions between the spacecraft and blockhouse d. Check the wideband video pair system between the spacecraft assembly area and the spacecraft	Blockhouse EOSE, data center	Procedure	Wideband video pair, system MOPS to all
16B	Perform Flight No. 1 Spacecraft SCS and Midcourse Correction Engine Leak Tests Both the stabilization and control subsystem and midcourse correction engine subsystem will be leak tested for leaks that could have been incurred during shipping and handling operations.	Leak test consoles	Procedure	Minimal personnel present
17A	Perform DSIF Compatibility Testing While the engineering model spacecraft is on-stand, the Cape DSIF 71 station compatibility test will be performed. The following measurements will be taken by the DSIF station:	Blockhouse EOSE	Procedure	RF clearance

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. Relative RF power measurements b. Frequency measurements c. Modulation index measurements d. Airborne command receiver best lock frequency determination e. Airborne command receiver zero loop stress frequency determination			
17B	<u>Check all Flight Spacecraft No. 1 Alignments</u> All Flight No. 1 spacecraft alignments will be checked for shifts due to transportation and handling operations.	Complete complement of alignment sets	Procedure	None
17C	<u>Receive Flight No. 2 Spacecraft at AMR</u>	Transporters	Procedure	None
17D	<u>Receive Flight No. 2 OSE at AMR</u> Both the Flight No. 2 spacecraft and OSE will be delivered to the AMR skid strip by air. From the skid strip, the Flight No. 2 spacecraft and OSE will be transported to the spacecraft assembly area.	Transporters	Procedure	None
17E	<u>Transport Pad EOSE Used at the Centaur Facility to Launch Pad No. 2</u> The Pad No. 2 OSE will be transported from the Centaur Assembly facility to Pad No. 2 in support of the launch complex phases of testing.	Slings, OSE handling fixtures, transporters	Procedure	None
18A	<u>Remove EM Spacecraft from Gantry</u>	Slings	Procedure	Crane service
18B	<u>Perform Flight No. 1 Spacecraft Appendage Deployment Test</u> Each spacecraft appendage will be manually deployed, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment as a result of the shipping and handling operations.	None	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
18C	Perform Flight No. 2 Spacecraft Receiving Inspection	None	None	None
18D	Perform Flight No. 2 Spacecraft OSE Receiving Inspection The spacecraft and OSE receiving inspections are performed mainly to ascertain that no damage to the spacecraft or OSE was incurred due to shipping and handling operations.	None	None	None
19A	Remove the Spacecraft Gantry Support Fixture From Launch Pad No. 1 Gantry The spacecraft gantry support fixture will be removed from launch Pad No. 1 gantry and placed in the transporter in preparation for moving to Pad No. 2	Slings	Procedure	Overhead crane with hook height of
19B	Perform Flight No. 1 Spacecraft Thermal Louver Checkout The Flight No. 1 spacecraft thermal louvers will be tested by stimulating them with a highly evaporative liquid and observing that proper operation exists.	Evaporative liquid	Procedure	None
19C	Validate Flight No. 2 Peripheral EOSE The data center and peripheral EOSE will be validated at this time to support the IST.	Peripheral EOSE validation set	Procedure	None
19D	Install AMR MOPS end instruments into Flight No. 2 EOSE The AMR MOPS end instruments will be installed in the EOSE and connected to the AMR MOPS intercommunications system.	MOPS end instruments	None	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
19E	Perform Launch Pad No. 2 Validations Using Spacecraft Simulator EOSE The pad validations are comprised of the following tests: a. Determine primary power line drops between the spacecraft and the blockhouse. b. RF up and down link power loss determinations. c. Electrically check all of spacecraft umbilical functions between the spacecraft and blockhouse. d. Check the wideband video pair system between the spacecraft assembly area and the spacecraft.	Capsule simulator, launch pad EOSE	Procedure	All Voyager pad modifications completed
20A	Transport Spacecraft Gantry Mounting Fixture to Launch Pad No. 2 and Install in the Gantry The spacecraft gantry mounting fixture will be transported to Pad No. 2 and installed in the gantry in preparation for the Pad No. 2 on-stand tests.	Transporter	Procedure	None
20B	Perform Flight Spacecraft No. 1 Experiment Calibrations The Flight No. 1 spacecraft experiment calibrations will be performed to insure that optimum experiment performance will be achieved during flight.	Complete set of systems test and experiment EOSE	Procedure	None
20C	Perform Flight Spacecraft No. 2 MOPS Checks to All Areas The AMR intercommunications net will be checked by contacting each Voyager station using the MOPS end instruments selector switch. Each end instrument in each area will be checked in this manner.	None	Channel assignment list	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
21A	<u>Transport Engineering Model Spacecraft and Nose Fairing to Launch Pad No. 2 and Install in Gantry</u> The engineering model spacecraft will be delivered to Pad No. 2 and mated in the spacecraft gantry mounting fixture in preparation for Pad No. 2 on-stand testing.	Transporter, slings handling fixture	Procedure	Crane with hook height of
21B	<u>Perform Flight No. 1 Spacecraft Transmitter Calorimeter Test</u> The first flight spacecraft transmitter calorimeter test will be performed to accurately measure the driver and power amplifier RF power delivered to the antenna system.	RF calorimeter	Procedure	None
21C	<u>Install Fine Sun Sensors, Canopus Sensor, and Gyro Package in Flight Spacecraft No. 2</u> The fine sun sensor, Canopus sensor, and gyro package installation will be performed in preparation for the IST.	Hand tools	Procedure	None
22A	<u>Perform EM Spacecraft Launch Pad No. 2 On-Stand Electrical Tests</u> The on-stand tests will be performed as follows: a. Determine primary power line drops between the engineering model spacecraft and blockhouse No. 2. b. RF power up and down link power loss determination. c. Electrically check all of the spacecraft umbilical functions between the spacecraft and the blockhouse. d. Check the wideband video pair system between gantry No. 2 and the spacecraft assembly area.	Blockhouse EOSE, data center	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
22B	Perform Flight No. 1 Spacecraft Solar Array Test The Flight No. 1 spacecraft solar array testing will be performed as follows: a. Perform inverse impedence test on each solar array panel. b. Illuminate each array panel and measure the open circuit voltage and short circuit current.	Solar array integration test EOSE	Procedure	None
22C	Install Flight No. 2 Spacecraft in the Tilt Fixture	Hand tools, torque wrenches, slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of
23A	Perform Cape DSIF 71 Compatibility Test The DSIF compatibility test will encompass the following tests: a. Relative power measurements between Pad No. 2 and the DSIF station. b. Engineering model spacecraft down link frequency measurement. c. Engineering model down link modulation index measurement. d. Airborne command receiver best lock frequency determination. e. Airborne receiver loop stress frequency determination.	Blockhouse EOSE, DSIF station	Procedure	Range clearance
23B	Perform Final Flight No.1 in Spacecraft Hangar Button-up The final in-hangar button-up is only a partial button-up in support of the actual bench. The partial button-up will include such things as, insulation, installation,	Hand tools, torque wrenches	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	cleaning of solar arrays, cleaning of Canopus sensor, and cleaning of sun sensors, and torquing all electronic equipment panels to specification.			
23C	<u>Install and Align Flight No. 1 Spacecraft Appendage Pin Pullers</u> The Flight No. 1 spacecraft pin pullers will be installed and aligned, insuring that proper appendage deployment will occur during flight. The pin puller alignments will be performed in two steps: a. Align pin pullers b. Check pin puller alignment by manually deploying each appendage and noting that mechanical hang-up does not occur.	Hand tools, torque wrenches	Procedure	None
23D	<u>Connect EOSE Cables to the Flight No. 2 Spacecraft</u> The EOSE Cables will be connected to the Flight No. 2 spacecraft in preparation for the Flight No. 2 integrated system test.	None	Procedure	None
23E	<u>Torque all Flight No. 2 Spacecraft Hardware to Flight Specification</u>	Torque wrenches	None	None
24A	<u>Remove the Voyager Engineering Model Spacecraft from the Launch Pad No. 2 Gantry</u> The engineering model spacecraft will be removed from the gantry and placed in the transporter in preparation for moving to the explosive safe area.	Slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of
24B	<u>Mate Flight No. 1 Spacecraft to Transporter</u> The Flight No. 1 spacecraft is to be removed from the tilt fixture and mated to the transporter in preparation for spacecraft testing at the explosive safe area.	Hand tools, transporter, purging equipment	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
24C	Validate Flight No. 2 Spacecraft Stabilization and Control Sensors The gyros, Canopus sensors, and fine sun sensors will be electrically revalidated in the spacecraft as part of the IST preparations.	System test set	Procedure	None
25A	Remove the Spacecraft Gantry Support Fixture from Launch Pad No. 2 Gantry The spacecraft gantry support fixture will be removed from Pad No. 2 gantry and placed in the transporter in preparation for moving to the spacecraft assembly area.	Slings	Procedure	Overhead crane with hook height of
25B	Transport Engineering Model Spacecraft to Explosive Safe Area The EM spacecraft will be transported from Pad No. 2 to the explosive safe area to validate the test complex.	Transporter	Procedure	None
25C	Transport Flight No. 1 Spacecraft to ESA Building The Flight No. 1 spacecraft is to be transported to the explosive safe area to support the tests that are to be performed in that area.	Transporter, purging equipment, tractor	Procedure	None
25D	Perform Flight No. 2 Spacecraft Integrated System Test The Flight No. 2 spacecraft integrated system test will be performed to verify that the spacecraft and all of its subsystems have successfully survived the shipping and handling operations.	Complete complement of systems test OSE	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
26A	Transport the Spacecraft Gantry Support Fixture to Assembly Area and Store The spacecraft gantry support fixture will be transported to the spacecraft assembly building and stored.	Transporter	None	None
26B	Validate ESA Test Complex Using Engineering Model Spacecraft The explosion safe area facility complex validations are performed in two parts. a. Validation of all OSE using the EM spacecraft b. Validation of the data lines going to the data centers in the spacecraft assembly area	System test set, EM spacecraft, spacecraft handling fixture		
26C	Remove Flight No. 1 Spacecraft from Transporter and Mate to Centaur Interstage The flight No. 1 spacecraft will be removed from the transporter and mated to the Centaur interstage in preparation for separation switch alignments and electrical checks.	Hand tools, torque wrench	Procedure	Overhead crane with hook height of ____
26D	Perform Flight No. 2 Spacecraft SCS and Midcourse Correction Engine Leak Tests Both the stabilization and control subsystem and midcourse correction engine subsystem will be leak tested for leaks that could have been incurred during shipping and handling operations.	Leak test	Procedure	Minimal personnel present
27	Remove Engineering Model Spacecraft from ESA Area and Store in the Spacecraft Assembly Area			

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
27B	Align and Electrically Check Flight No. 1 Spacecraft Separation Switches The Flight No. 1 spacecraft separation switches will be aligned and electrically checked to insure that the acquisition phases of the spacecraft mission profile will be properly accomplished.	Hand tools, torque wrench, separation switch alignment set		
27C	Check All Flight Spacecraft No. 2 Alignments All Flight No. 2 spacecraft alignments will be checked for shifts due to transportation and handling operations.	Complete complement of alignment sets	Procedure	None
28A	Install and Electrically Check Flight Spacecraft No. 1 Flight Batteries Concurrently, the Flight No. 1 spacecraft battery will be installed and electrically tested to insure that the proper cell voltages exist under load and to insure that the battery charges and discharges properly.	Torque wrenches, hand tools	Procedure	None
28B	Perform Flight No. 2 Spacecraft Appendage Deployment Test Each spacecraft appendage will be manually deployed observing that each freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment as a result of the shipping and handling operations.	None	Procedure	None
29A	Perform Flight Spacecraft No. 1 Folded IST The flight No. 2 spacecraft folded integrated system test will be performed to insure that the spacecraft is ready to proceed to the on-stand testing phase.	Complete set of systems test and experiment EOSE	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
29B	<u>Perform Flight No. 2 Spacecraft Thermal Louver Check-out</u> The Flight No. 2 spacecraft thermal louvers will be tested by stimulating them with a highly evaporative liquid and observing that proper operation exists.	Evaporative liquid	Procedure	None
30A	Install Flight No. 1 Spacecraft Appendage Pin Puller Cartridges Install Flight No. 1 spacecraft ordnance cartridges in each pin puller and torque to flight specification.	Torque wrench	Procedure	None
30B	<u>Install Flight No. 1 Spacecraft Shorting Plugs</u> After the pin puller cartridges have been installed, shorting plugs will be installed across each cartridge.	Shorting plugs	None	None
30C	<u>Perform Flight Spacecraft No. 2 Experiment Calibrations</u> The Flight No. 2 spacecraft experiment calibrations will be performed to insure that optimum experiment performance will be achieved during flight.	Complete set of systems test and experiment EOSE	Procedure	None
31A	<u>Perform Flight No. 1 Spacecraft Appendage Latch Test</u> The Flight No. 1 spacecraft pin puller alignments are to be checked by manually deploying each appendage and noting that each appendage properly latches and unlatches.	None	Procedure	None
31B	<u>Perform Flight No. 2 Spacecraft Transmitter Calorimeter Test</u> The spacecraft transmitter calorimeter test will be performed to accurately measure the driver and power amplifier RF power delivered to the antenna system	RF calorimeter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
32A	<u>Perform Flight Spacecraft No. 1 SCS and Propulsion Subsystem Leak Test</u> The Flight No. 1 spacecraft stabilization and control subsystem and the propulsion subsystem will be leak tested.	SCS leak test console, propulsion leak test console	Procedure	None
32B	<u>Pressurize the Flight Spacecraft No. 1 SCS and Propulsion Bottles to Flight Pressure</u> After the leak tests each subsystem will be pressurized to flight levels as part of launch preparations.	SCS leak test console, propulsion leak test console	Procedure	None
32C	<u>Perform Flight No. 2 Spacecraft Solar Array Test</u> The Flight No. 2 spacecraft solar array testing will be performed as follows: a. Perform inverse impedence test on each solar array panel b. Illuminate each array panel and measure the open circuit voltage and short circuit current.	Solar array integration test EOSE	Procedure	None
33A	<u>Install Flight No. 1 Spacecraft Midcourse Correction Motor Ordnance Cartridges</u>	Hand tools, torque wrenches	Procedure	None
33B	<u>Install Shorting Plugs Across Each Midcourse Correction Cartridge</u>	None	None	None
33C	<u>Install and Align Flight No. 2 Appendage Pin Pullers</u> As part of the button-up procedure, the flight pin puller will be installed and aligned. The pin puller alignments will take place as follows: a. Align pin pullers b. Check pin puller alignment by manually deploying each appendage and noting that each appendage latches and unlatches properly	Hand tools, torque wrenches, puller alignment set	Procedure	

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
33D	Perform Final Flight No. 2 Spacecraft in Hangar Button-Up The final in-hangar button-up is only a partial button-up in support of the actual launch. The partial button-up will include such things as, insulation, installation, cleaning of solar arrays, cleaning of Canopus sensor, and cleaning of sun sensors, and torquing all electronic equipment panels to specification.	Hand tools, torque wrenches	Procedure	None
34	Fuel Flight No. 1 Spacecraft Midcourse Correction Motor to Flight Levels	Motor fuelding EOSE set	Procedure	None
35A	Perform Flight No. 1 Spacecraft Final Button-Up The Flight No. 1 spacecraft final button-up will be performed to insure that all electrical and mechanical interfaces added since the hangar testing operations have been properly mated. In addition, all sensors and solar arrays will be cleaned with suitable solvents.	Torque wrenches, cleaning solvents, solvent applicators	Procedure	None
35B	Mate Flight No. 2 Spacecraft to Transporter The Flight No. 1 spacecraft is to be removed from the tilt fixture and mated to the transporter in preparation for spacecraft testing at the explosive safe area.	Hand tools, transporter, purging equipment	Procedure	None
36A	Check Flight No. 1 Spacecraft Over-all Vertical Alignment The Flight No. 1 spacecraft over-all vertical alignment will be performed to insure that the spacecraft will separate properly from the launch vehicle.	Spacecraft vertical alignment set	Procedure	
36B	Remove Flight No. 2 Spacecraft from Transporter and Mate to Centaur Interstage The Flight No. 2 spacecraft will be removed from the transporter and mated to the Centaur interstage in preparation for separation switch alignments and electrical checks.	Hand tools, torque wrench	Procedure	Overhead crane with hook height of ______

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
37A	<u>Perform Flight No. 1 Spacecraft Weight and Center of Gravity Determination</u> The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. The spacecraft will be used to determine the center of gravity of the third spacecraft axis.	Hand tools, torque wrenches, center of gravity fixture, load cells and associated electronics	None	Some means of hoisting the spacecraft into the Center of Gravity fixture
37B	<u>Align and Electrically Check Flight No. 2 Spacecraft Separation Switches</u> The Flight No. 2 spacecraft separation switches will be aligned and electrically checked to insure that the acquisition phases of the spacecraft mission profile will be properly accomplished.	Hand tools, torque wrench, separation switch alignment set	Procedure	None
38A	<u>Perform Flight No. 1 Spacecraft Final Ordnance Checks</u> The final ordnance checks will be performed as follows: a. At the safe-arm J-box check that no voltage exists across the wires going to each ordnance device b. At the safe-arm J-box check that zero ohms exist across each ordnance wire to ground by using a range approved milli-ohmmeter c. At the safe-arm J-box determine that continuity exists through each ordnance bridge wire by using a range approved milli-ohmmeter. d. Arm the safe-arm J-box and check that battery voltage exists where it should and no voltage exists on the remaining pins of each connector. e. "Safe" the safe-arm J-box and check that zero ohms exists across each ordnance device to frame ground. f. Connect each ordnance device to the safe-arm J-box.	Complete complement of ordnance test equipment	Procedure	None

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Operation No	Task Description	Equipment Required	Documentation Required	Special Facilities Required
38B	Install Flight Spacecraft No. 2 Flight Batteries Concurrently, the Flight No. 2 spacecraft battery will be installed and electrically tested to insure that the proper cell voltages exist under load and to insure that the battery charges and discharges properly.	Torque wrenches, hand tools	Procedure	None
39A	Install Flight No. 1 Spacecraft Nose Fairing The Flight No. 1 spacecraft nose fairing will be placed over the Flight No. 1 spacecraft in preparation for the on-stand testing phase.	Slings, nose fairing handling fixture	Procedure	Overhead crane with hook height of ____
39B	Perform Flight Spacecraft No. 2 Folded IST The flight No. 2 spacecraft folded integrated system test will be performed to insure that the spacecraft is ready to proceed to the launch stand testing phase.	Complete set of systems test and experiment EOSE	Procedure	N ne
40A	Mate Flight No. 1 Spacecraft to the Pad Transporter The Flight No. 1 spacecraft will be mated to the pad transporter in preparation for shipment to Pad No. 1.	Slug, spacecraft nose fairing handling fixture	Procedure	Overhead crane with hook height of ____
40B	Install Flight No. 2 Spacecraft Appendage Pin Puller Cartridges The Flight No. 2 spacecraft ordnance cartridges will be installed in each pin puller and torques to flight specification.	Torque wrench	Procedure	None
40C	Install Flight No. 2 Spacecraft Shorting Plugs After the pin puller cartridges have been installed, shorting plugs will be installed across each cartridge.	Shorting plugs	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
41A	Perform Flight No. 1 Spacecraft Modified IST The Flight No. 1 spacecraft modified integrated system test is designed to verify that there has been no degradation of spacecraft performance during the ESA build-up and testing phase and is ready to proceed with the on-stand testing activities.	Complete set of systems test EOSE	Procedure	None
41B	Perform Flight No. 2 Spacecraft Appendage Latch Test The Flight No. 2 spacecraft pin puller alignments are to be checked by manually deploying each appendage and noting that each appendage properly latches and unlatches.	None	Procedure	None
42A	Transport Flight No. 1 Spacecraft to Pad No. 1 The Flight No. 1 spacecraft will be transported to Pad No. 1 to support the spacecraft final on-stand activities.	Pad transporter, tractor, purging equipment, slings, spacecraft handling fixture		
42B	Perform Flight Spacecraft No. 2 SCS and Propulsion Subsystem Leak Test The Flight No. 2 spacecraft stabilization and control subsystem and the propulsion subsystem will be leak tested.	SCS leak test console, propulsion leak test console	Procedure	None
42C	Pressurize the Flight Spacecraft No. 2 SCS and Propul- Bottles to Flight Pressure After the leak tests, each subsystem will be pressurized to flight levels as part of launch preparations.	SCS leak test console, propulsion leak test console	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
43A	Mate Flight No. 1 Spacecraft to the Centaur Launch Vehicle The Flight No. 1 spacecraft will be hoisted to the top of the gantry and mated to the Centaur launch vehicle stage.	Slings, spacecraft handling fixture, hand tools, torque wrenches	Procedure	Overhead crane with hook height of ____
43B	Install Flight No. 2 Spacecraft Midcourse Correction Motor Ordnance Cartridges	Hand tools, torque wrenches	Procedure	None
43C	Install Shorting Plugs Across Each Midcourse Correction Motor Ordnance Cartridge	None	None	None
44	Perform Flight No. 1 Spacecraft to Centaur Alignment Check The purpose of this check is to ascertain that the spacecraft coordinate axes system is aligned properly to the Centaur coordinate axes system.	Spacecraft Centaur alignment set, torque wrenches	Procedure	None
45	Fuel Flight No. 2 Spacecraft Midcourse Correction Motor to Flight Levels	Motor fueling EOSE set	Procedure	None
46A	Perform Flight No. 1 Spacecraft On-Stand Functional Test The Flight No. 1 spacecraft on-stand functional test is designed to checkout the following interfaces: a. All spacecraft umbilical functions between the spacecraft and the Pad No. 1 blockhouse b. Wideband video pair system between the spacecraft and the data centers c. RF link between the spacecraft and the data center d. RF link between the spacecraft and the DSIF station. Note: The Flight No. 1 spacecraft on-stand testing will hold until arrival of Flight No. 2 spacecraft at Pad No. 2.	Hangar data		Spacecraft cooling MOPS, primary EOSE power

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
46B	<u>Perform Flight No. 2 Spacecraft Final Button-Up</u> The Flight No. 2 spacecraft final button-up will be performed to insure that all electrical and mechanical interfaces added since the hangar testing operations have been properly mated. In addition, all sensors and solar arrays will be cleaned with suitable solvents.	Torque wrenches, cleaning solvents, solvent applicators	Proque	None
46C	<u>Return Spacecraft Pad Transporter to ESA Area</u>	Tractor	None	None
47	<u>Check Flight No. 2 Spacecraft Over-all Vertical Alignment</u> The Flight No. 1 Spacecraft over-all vertical alignment will be performed to insure that the spacecraft will separate properly from the launch vehicle.	Spacecraft vertical alignment set	Procedure	None
48	<u>Perform Flight No. 2 Weight and Center of Gravity Determination</u> The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axis. Note that the center of gravity determination of the spacecraft less capsule was determined during assembly and test.	Hand tools, torque wrenches, CG fixture, load cells, and associated electronics	None	Some means of hoisting the spacecraft into the CG fixture
49	<u>Perform Flight No. 2 Spacecraft Final Ordnance Checks</u> The final ordnance checks will be performed as follows: a. At the safe-arm J-box check that no voltage exists across the wires going to each ordnance device. b. At the safe-arm J-box check that zero ohms exist across each ordnance wire to ground by using a range approved milli-ohmmeter. c. At the safe-arm J-box determine that continuity exists through each ordnance bridge wire by using a range approved milli-ohmmeter.	Complete complement of ordnance test equipment	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. Arm the safe-arm J-box and check that battery voltage exists where it should and that no voltage exists on the remaining pins of each connector. e. "Safe" the safe-arm J-box and check that zero ohms exists across each ordnance device to frame ground. f. Connect each ordnance device to the safe-arm J-box.			
50	Install Flight No. 2 Spacecraft Nose Fairing The Flight No. 2 spacecraft nose fairing will be placed over the Flight No. 2 spacecraft in preparation for the on-stand testing phase.	Slings, nose fairing handling fixture	Procedure	Overhead crane with hook height of ____
51	Mate Flight No. 2 Spacecraft to the Pad Transporter The Flight No. 2 spacecraft will be mated to the pad transported in preparation for shipment to Pad No. 2.	Slug, spacecraft nose fairing handling fixture	Procedure	Overhead crane with hook height of ____
52	Perform Flight No. 2 Spacecraft Modified IST The Flight No. 2 spacecraft modified integrated system test is designed to verify that there has been no degradation of spacecraft performance during the ESA build-up and testing phase and is ready to proceed.	Complete set of systems test EOSE	Procedure	None
53	Transport Flight No. 2 Spacecraft to Pad No. 2 The Flight No. 2 spacecraft will be transported to Pad No. 1 to support the spacecraft final on-stand launch activities.	Pad transporter, tractor, purging equipment, slings, spacecraft handling fixture		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
54	<u>Mate Flight No. 2 Spacecraft to the Centaur Launch Vehicle</u> The Flight No. 2 spacecraft will be hoisted to the top of the gantry and mated to the Centaur launch vehicle.	Slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of _____
55	<u>Perform Flight No. 2 Spacecraft to Centaur Alignment</u> The Flight No. 2 spacecraft to Centaur alignment check is performed to ascertain that the spacecraft coordinate system is aligned to the Centaur coordinate system within the required accuracy.	Spacecraft Centaur alignment set, torque wrenches	Procedure	None
56	<u>Perform Flight No. 2 Spacecraft On-Stand Functional Test</u> The Flight No. 2 spacecraft on-stand functional test is designed to checkout the following interfaces: a. All spacecraft umbilical functions between the spacecraft and the Pad No. 2 blockhouse. b. Wideband video pair system between the spacecraft and the data centers c. RF link between the spacecraft and the data center d. RF link between the spacecraft and the DSIF station.	Hangar data center, complete set of pad EOSE purging equipment	Procedure	Spacecraft cooling MOPS, primary EOSE power
57	<u>Perform RFI Test Practice Using Both Flight No. 1 and Flight No. 2 Spacecrafts</u> The RFI test practice is repeated again because this is the first time both Flight No. 1 and 2 are operating at the same time, affording experience in operating and coordinating two spacecrafts and two data centers at once.	Hangar data centers, data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft cooling, primary EOSE power, MOPS
58	<u>Perform Combined Vehicle RF Interference Compatibility Test</u> The combined vehicle RF interference test is performed to ascertain that none of the Centaur or Saturn transmitters or beacons interfere with or degrade the spacecraft transmitters or receivers. Likewise, the test is performed	Hangar data centers data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	to ascertain that the spacecraft transmitters do not interfere with or degrade the Centaur or Saturn vehicle beacons, transmitters, or receivers. The RFI compatibility test is to be performed as follows: a. Each Saturn beacon and transmitter is turned on one at a time and both the Centaur and the spacecraft will ascertain that there is no interference with, or degradation of the receiver or transmitter systems. b. Each Centaur beacon and transmitter is turned on one at a time and both the saturn vehicle and the spacecraft will ascertain that there is no degradation or interference with the receiver or transmitter systems. c. Each spacecraft transmitter is turned on one at a time and both the Saturn and Centaur vehicles will ascertain that there is no degradation of or interference with the receiver or transmitter systems. d. All spacecraft Centaur and Saturn transmitters are turned on together and each vehicle will ascertain that there are no mutual degradations of or interference with the various transmitting of receiving systems.			
59	Perform J FACT Test Preparations The J FACT test preparations are broken up into the following subtasks: a. The installation of the nose fairing separation squib simulators. b. The installation of the spacecraft umbilical cable spin-off connector squib simulators c. The installation of the spacecraft separation squib simulators The remainder of the day is to be spent in practicing the J FACT test procedure. It is expected that only the spacecraft will participate in this particular activity.	Hangar data centers, data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing

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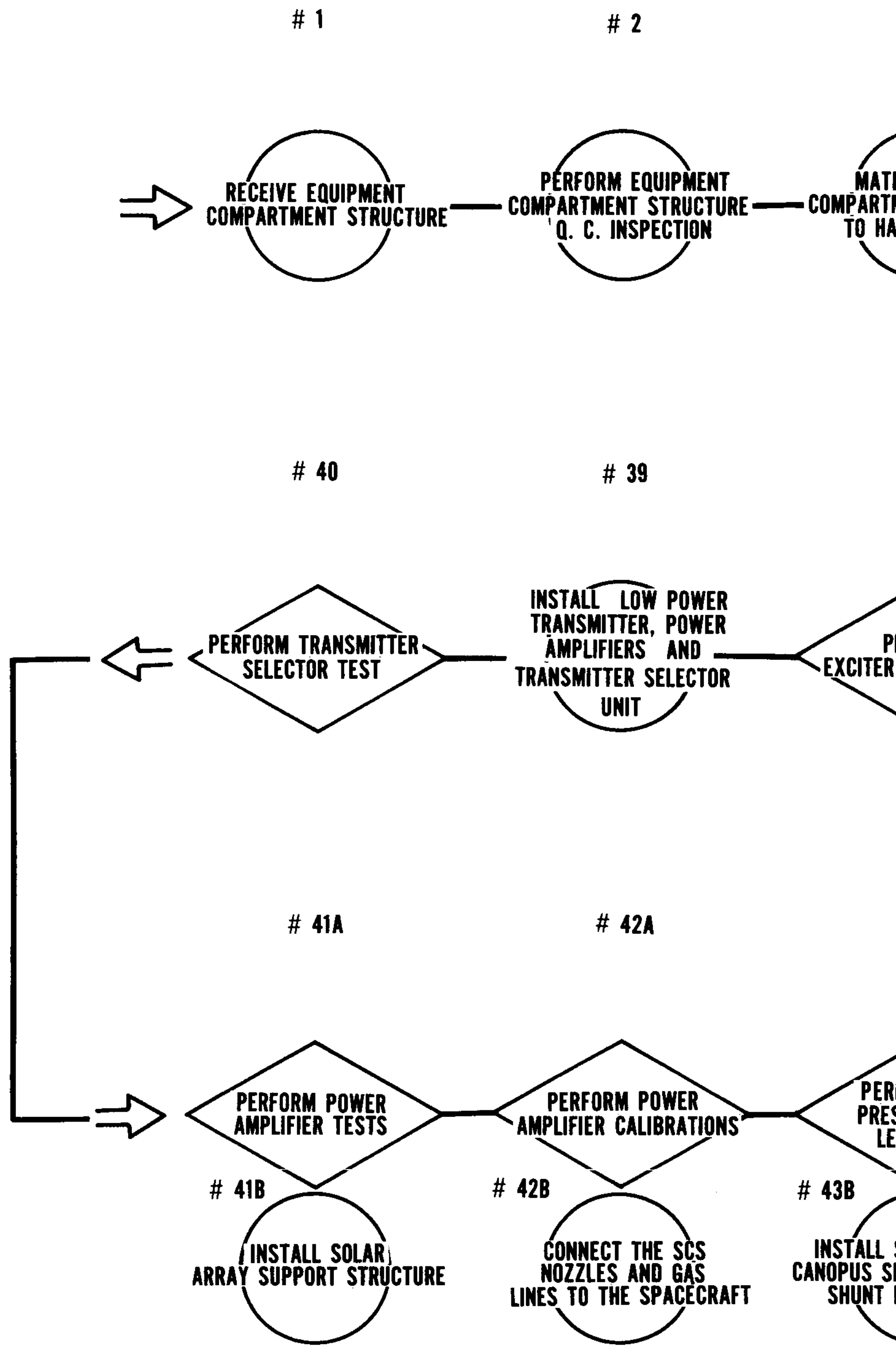
Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
64	Commence Terminal Countdown During terminal countdown, the launch vehicles will be fueled with oxidizer and the gantry removed.	Hangar data centers, data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft colling, MOPS
65	Lift Off	Hangar data center, data center, data center interpatching	Procedure	MOPS, range firing

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
60	<u>Perform J FACT Test</u> The purpose of the J FACT test is to check out the post-injection portions of the mission profile. The following spacecraft-related postlaunch functions will be monitored and checked. a. Nose fairing separation b. Spacecraft umbilical cable separation c. Spacecraft separation from the Centaur vehicle 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.	Hangar data centers, data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power
61	<u>Perform FRD Preparations</u> As far as the spacecraft is concerned, the flight readiness demonstration preparations will consist of practicing the FRD procedure. It should be mentioned that the FRD test is identical to the countdown in regards to spacecraft activities	Hangar data denters, data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power
62	<u>Perform FRD Test</u>	Hangar data centers, data center, interpatching pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing
63	<u>Start Pre-countdown</u> Both spacecrafts will participate in the pre-countdown activities. Prior to the conclusions of the pre-countdown activities each spacecraft subsystem will have been checked. At the conclusion of the pre-countdown activities a decision will be made as to whether Flight No. 1 or No. 2 spacecraft will be launched.	Hangar data centers, data center, interpatching	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing



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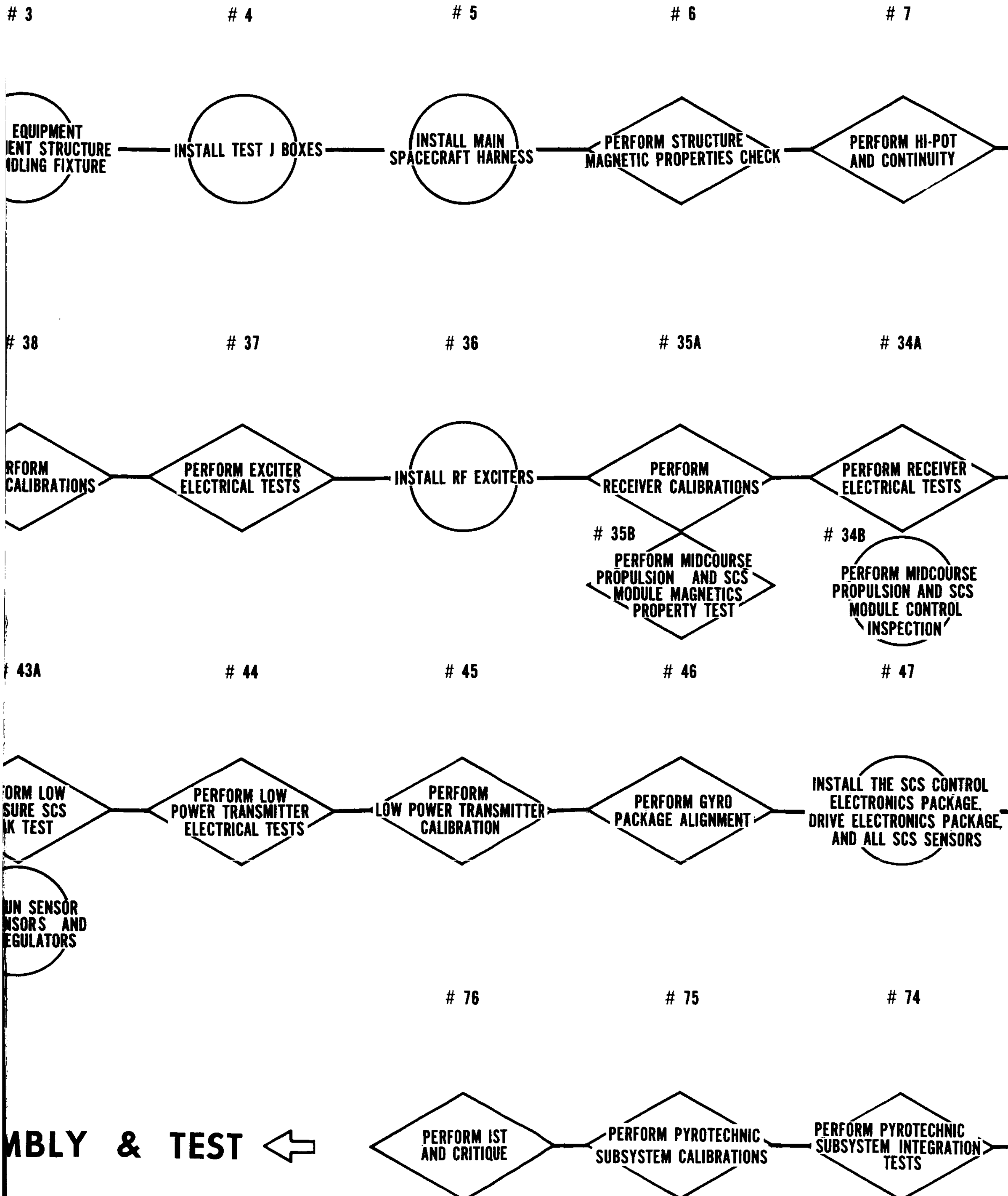

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EQUIPMENT
IENT STRUCTURE
IDLING FIXTURE
INSTALL TEST J BOXES
INSTALL MAIN SPACECRAFT HARNESS
PERFORM STRUCTURE MAGNETIC PROPERTIES CHECK
PERFORM HI-POT AND CONTINUITY
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35A
34A
RFORM
CALIBRATIONS
PERFORM EXCITER ELECTRICAL TESTS
INSTALL RF EXCITERS
PERFORM RECEIVER CALIBRATIONS
PERFORM RECEIVER ELECTRICAL TESTS
35B
PERFORM MIDCOURSE PROPULSION AND SCS MODULE MAGNETICS PROPERTY TEST
34B
PERFORM MIDCOURSE PROPULSION AND SCS MODULE CONTROL INSPECTION
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ORM LOW
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K TEST
PERFORM LOW POWER TRANSMITTER ELECTRICAL TESTS
PERFORM LOW POWER TRANSMITTER CALIBRATION
PERFORM GYRO PACKAGE ALIGNMENT
INSTALL THE SCS CONTROL ELECTRONICS PACKAGE, DRIVE ELECTRONICS PACKAGE, AND ALL SCS SENSORS
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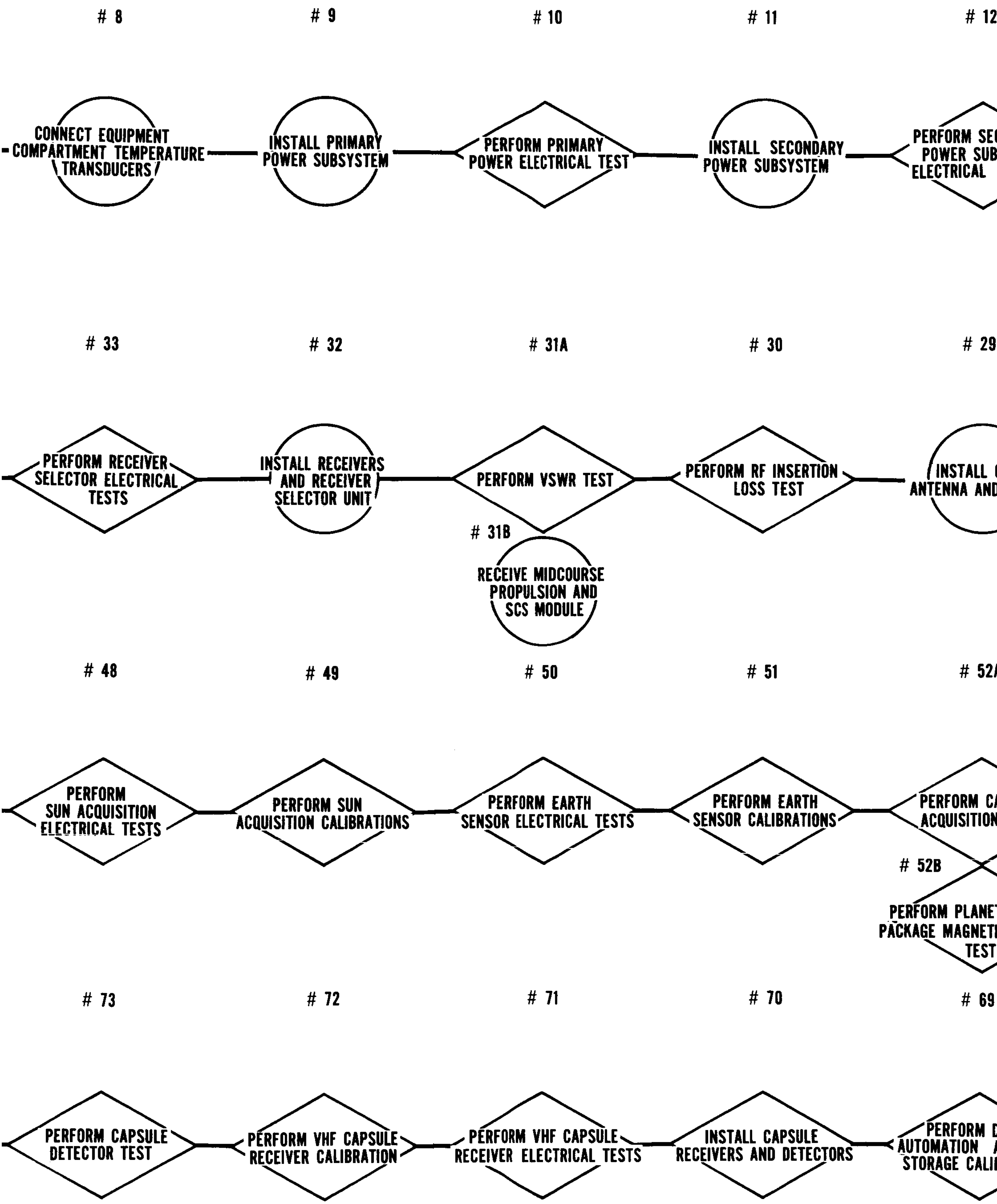
MBLY & TEST
PERFORM IST AND CRITIQUE
PERFORM PYROTECHNIC SUBSYSTEM CALIBRATIONS
PERFORM PYROTECHNIC SUBSYSTEM INTEGRATION TESTS



8
CONNECT EQUIPMENT COMPARTMENT TEMPERATURE TRANSDUCERS
9
INSTALL PRIMARY POWER SUBSYSTEM
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PERFORM PRIMARY POWER ELECTRICAL TEST
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INSTALL SECONDARY POWER SUBSYSTEM
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PERFORM
POWER
ELECTRICAL
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PERFORM RECEIVER SELECTOR ELECTRICAL TESTS
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INSTALL RECEIVERS AND RECEIVER SELECTOR UNIT
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RECEIVE MIDCOURSE PROPULSION AND SCS MODULE
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INSTALL
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PERFORM SUN ACQUISITION ELECTRICAL TESTS
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PERFORM SUN ACQUISITION CALIBRATIONS
50
PERFORM EARTH SENSOR ELECTRICAL TESTS
51
PERFORM EARTH SENSOR CALIBRATIONS
52
PERFORM
52B
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TEST
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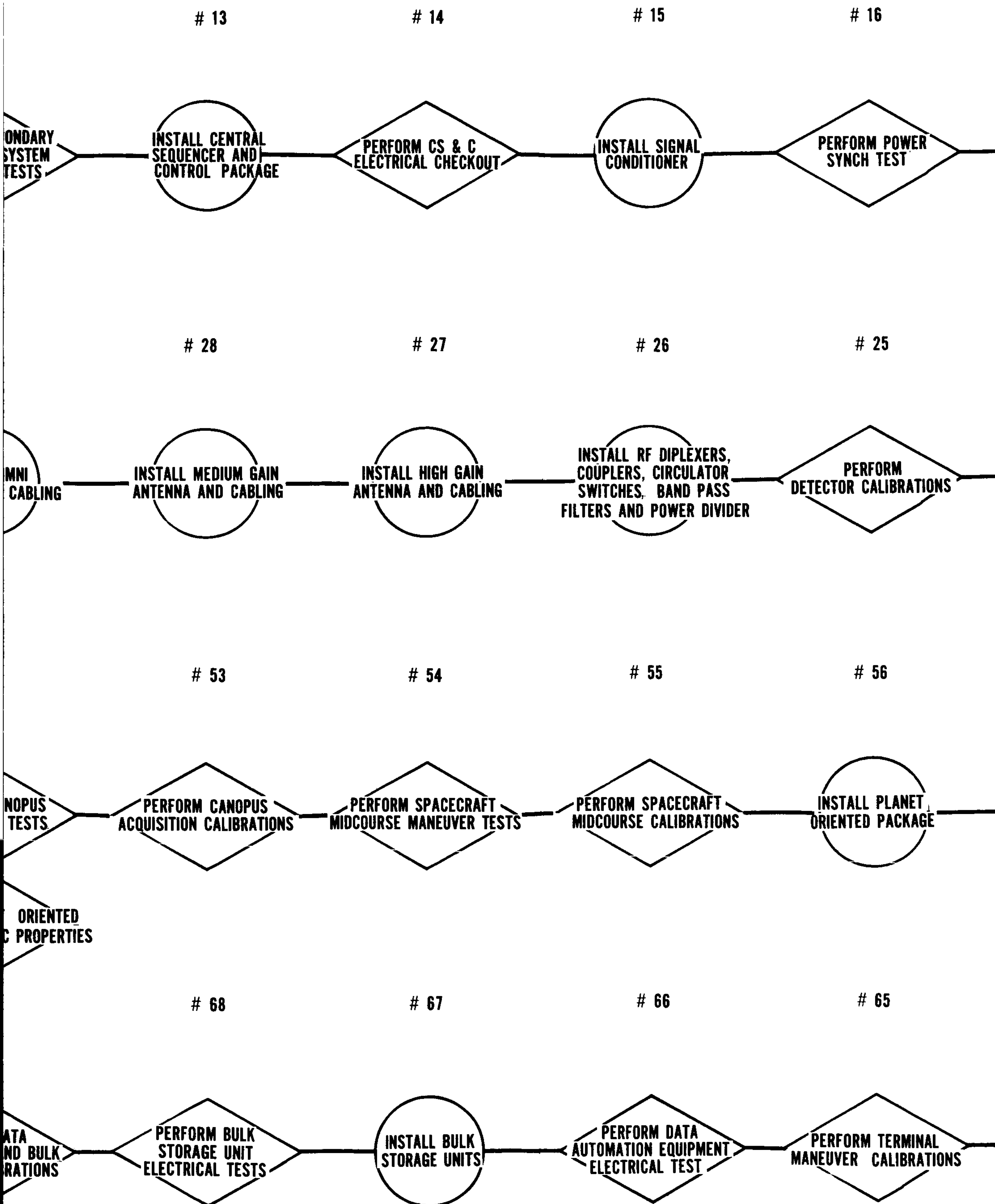
PERFORM CAPSULE DETECTOR TEST
72
PERFORM VHF CAPSULE RECEIVER CALIBRATION
71
PERFORM VHF CAPSULE RECEIVER ELECTRICAL TESTS
70
INSTALL CAPSULE RECEIVERS AND DETECTORS
69
PERFORM
AUTOMATION
STORAGE

3



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ONDARY SYSTEM TESTS
INSTALL CENTRAL SEQUENCER AND CONTROL PACKAGE
PERFORM CS & C ELECTRICAL CHECKOUT
INSTALL SIGNAL CONDITIONER
PERFORM POWER SYNCH TEST
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MNI CABLING
INSTALL MEDIUM GAIN ANTENNA AND CABLING
INSTALL HIGH GAIN ANTENNA AND CABLING
INSTALL RF DIPLEXERS, COUPLERS, CIRCULATOR SWITCHES, BAND PASS FILTERS AND POWER DIVIDER
PERFORM DETECTOR CALIBRATIONS
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NOPUS TESTS
PERFORM CANOPUS ACQUISITION CALIBRATIONS
PERFORM SPACECRAFT MIDCOURSE MANEUVER TESTS
PERFORM SPACECRAFT MIDCOURSE CALIBRATIONS
INSTALL PLANET ORIENTED PACKAGE
ORIENTED
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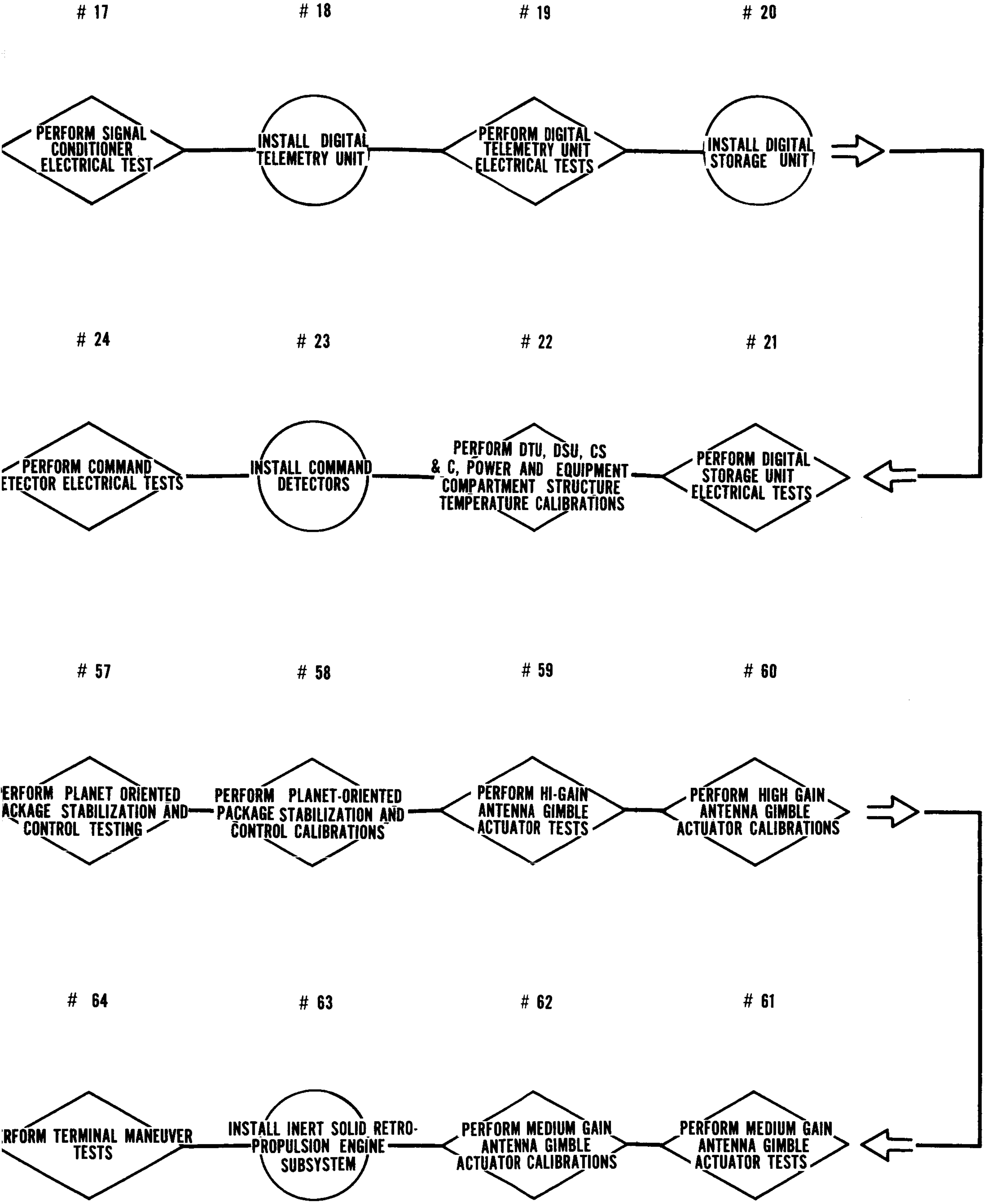
PERFORM BULK STORAGE UNIT ELECTRICAL TESTS
INSTALL BULK STORAGE UNITS
PERFORM DATA AUTOMATION EQUIPMENT ELECTRICAL TEST
PERFORM TERMINAL MANEUVER CALIBRATIONS



17
PERFORM SIGNAL CONDITIONER ELECTRICAL TEST
18
INSTALL DIGITAL TELEMETRY UNIT
19
PERFORM DIGITAL TELEMETRY UNIT ELECTRICAL TESTS
20
INSTALL DIGITAL STORAGE UNIT
21
PERFORM DIGITAL STORAGE UNIT ELECTRICAL TESTS
22
PERFORM DTU, DSU, CS & C, POWER AND EQUIPMENT COMPARTMENT STRUCTURE TEMPERATURE CALIBRATIONS
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INSTALL COMMAND DETECTORS
24
PERFORM COMMAND ETECTOR ELECTRICAL TESTS
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ERFORM PLANET ORIENTED ACKAGE STABILIZATION AND CONTROL TESTING
58
PERFORM PLANET-ORIENTED PACKAGE STABILIZATION AND CONTROL CALIBRATIONS
59
PERFORM HI-GAIN ANTENNA GIMBLE ACTUATOR TESTS
60
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61
PERFORM MEDIUM GAIN ANTENNA GIMBLE ACTUATOR TESTS
62
PERFORM MEDIUM GAIN ANTENNA GIMBLE ACTUATOR CALIBRATIONS
63
INSTALL INERT SOLID RETRO-PROPULSION ENGINE SUBSYSTEM
64
RFORM TERMINAL MANEUVER TESTS

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1A	<u>Receive Equipment Compartment Structure</u> The spacecraft equipment compartment structure will be received from Douglas Aircraft Co. in the following configuration: a. Solar array support structure not installed b. Main spacecraft harness not installed c. Thermal insulation not installed d. Thermal louvers not installed e. Propulsion system not installed f. Equipment compartment structure temperature transducers installed g. Planet-oriented package and support fixture not installed h. High-gain antenna and support structure not installed i. Medium-gain antenna and boom not installed j. Omni antenna and boom not installed k. Magnetometer and boom not installed l. Solid inert motor not installed m. TRW quality control buy-off will be performed at Douglas Aircraft Co.	Tools to un-crate structure	Equipment list	None
1B	<u>Receive Systems Test Set EOSE</u>	None	Equipment list	None
2A	<u>Perform Equipment Compartment Structure Quality Control Inspection</u> Quality control inspection is mainly for shipping damage as the equipment compartment structure will have been already bought off at Douglas Aircraft Co.	None	Procedure	None

Functional Flow
Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 2

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
2B	<u>Start System Test Set EOSE Validation</u> The system test set EOSE will be validated for two reasons. a. To ensure that the EOSE has survived the shipping and handling operations b. To familiarize test crews with the EOSE	System test set validation sets	Procedures	None
3	<u>Mate Equipment Compartment Structure to Handling Fixture</u> a. Mate MOSE adapter to spacecraft structure b. Mate MOSE adapter and spacecraft to handling fixture	Handling sling, adapter handling fixture, Protective covers, hand tools	Procedures	None
4	<u>Install Test J Boxes</u> Install all electrical test J boxes to support the hi-pot and continuity test	Hand tools, torque wrench	Procedure	None
5	<u>Install Main Spacecraft Harness</u> Install main spacecraft electrical harness and connect to J boxes	Hand tools, torque wrench, handling sling	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
6	**Perform Structure Magnetic Properties Check** The equipment compartment magnetic properties check will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the magnetic field of the equipment compartment structure mounted in handling fixture c. Analyze all variations between readings and repeat if necessary	Magnetic measuring equipment, handling fixture, protective covers, handling slings	Procedure	Area in building free of large magnetic fields
7	**Perform Hi-Pot and Continuity** This is to be accomplished using a Huges FACT machine or equivalent. Whereever possible the test will be run end to end through all J boxes	Huges FACT machine or equivalent, cable adapters, FACT machine programs	Procedure	None
8	**Connect Equipment Compartment Temperature Transducers** Solder all temperature transducers to main spacecraft harness	Soldering iron, solder, insulation	None	None
9	**Install Primary Power Subsystem**	Hand tools, torque wrench	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
10	<u>Perform Primary Power Electrical Test</u> The primary power subsystem consists of the following items: batteries, power control unit, shunt regulators, and battery boost regulator. The subsystem electrical tests will be performed as follows: a. Integrate power OSE b. Perform bus open circuit checks using the external power mode c. Perform bus open circuit checks using the spacecraft batteries d. Perform bus open circuit checks using solar array simulated power e. Load electrical bus using dummy loads and electrically test the power control unit and shunt regulators using the spacecraft batteries and the solar array simulator. Commands will be simulated by using an external power supply that will be part of one of the load boxes f. Remove loads from bus and connect boost regulator g. Power boost regulator from external power and measure output current h. Load boost regulator output and measure the input and output voltage and current. Also note that noise on the output lines is within acceptable limits. Note: All loads are to be applied at the users side of the harness.	Voltmeters, ammeters, oscilloscope, power supply, power EOSE, series fuse boxes, in-line test connectors	Procedure	None
11	<u>Install Secondary Power Subsystem</u>	Hand tools, torque wrench	Procedure	None

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 5

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
12	**Perform Secondary Power Subsystem Electrical Tests** **The secondary power subsystem consists of the following items:** a. 4.1 kc — 1ϕ inverter b. 820 cps — 2ϕ inverter c. 410 cps — 1ϕ inverter **The secondary power subsystem test will be performed as follows:** **a.** **Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the 4.1 kc primary power input** **b.** **Connect 4.1-kc inverter to the spacecraft main harness** **c.** **Check 4.1-kc inverter open circuit voltage by powering the bus on external power** **d.** **Load 4.1-kc inverter using dummy loads and check output current and voltage** **e.** **Repeat steps a through c for the** 820 and 410 cps inverters **Note: All load boxes are to be applied at the users side of the harness**	**Ammeters, voltmeters, oscilloscope, power EOSE, series fuse boxes, in-line test connectors**	**Procedure**	**None**
13	**Install Central Sequencer and Control Package**	**Hand tools**	**None**	**None**

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 6

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
14	Perform CS and C Electrical Checkout The central sequencer and control unit electrical checkout will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the CS and C power input connector b. Connect the CS and C to the spacecraft harness and measure the voltage and current drawn by the CS and C. Also note that noise and transients are at acceptable levels c. Connect command detector format generator to the CS and C at the detector side of the spacecraft harness d. Check all of the power control unit commands as follows: 1) Open all command lines from the CS and C at the PCU side of the spacecraft harness 2) Transmit all PCU commands via the command format generator 3) Observe the open circuit command signal voltage at the PCU 4) Close the command lines to the PCU and retransmit the PCU commands via the command format generators 5) Monitor the command voltage and current at the PCU 6) Observe command signal lines and note that noise and transients are at acceptable levels 7) Observe that the PCU reacts properly to the CS and C commands	Command format generators, voltmeters, oscilloscope, ammeter, power EOSE, series fuse boxes, in-line test connectors, command matrix monitor	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	e. Check the open circuit voltage of the remaining discrets command lines from the CS and C at the side of the spacecraft harness. Note: The noise and transient levels on each of the remaining command signal lines will be checked during the electrical integration of the remaining subsystems f. Transmit each quantitative command from the format generator and observe that each command was properly received by observing the command matrix monitor g. Measure the amplitude and frequency of the down link PN subcarrier h. Measure the amplitude and frequency of all timing signals from the CS and C			
15	<u>Install Signal Conditioner</u>	Hand tools, torque wrench	Procedure	None
16	<u>Perform Power Synch Test</u> The power synch tests will be performed in the following manner: a. Apply external power to the spacecraft and observe the open circuit frequency, rise time, fall time pulse width, and amplitude of each synch pulse from the CS and C to the boost regulator and each inverter b. Connect the synch pulse to the boost regulator and observe the frequency, rise time, fall time, pulse width, and amplitude of each pulse	Oscilloscope, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Observe the boost regulator 50 vdc output noise d. Note that noise and transients are within acceptable limits e. Repeat the above steps for each inverter			
17	<u>Perform Signal Conditioner Electrical Test</u> a. Turn on external power to spacecraft and check that voltage exists where it should and no voltage exists on the remaining pins at the signal conditioner power input connector b. Connect signal conditioner to secondary power subsystem c. Measure voltage and current drawn by signal conditioner from the secondary power subsystem	Voltmeter, ammeter, series fuse boxes	Procedure	None
18	<u>Install Digital Telemetry Unit</u>	Hand tools, torque wrench	Procedure	None
19	<u>Perform Digital Telemetry Unit Electrical Tests</u> The DTU electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins at the DTU power input connector b. Connect the DTU to the 4.1-kc inverter and measure the voltage and current drawn by the DTU. Also note that noise and transients are at acceptable levels	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator, analog word simulator	Procedure	None

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 9

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Measure command line signal voltage and current drawn for each commanded bit rate, format and mode of operation. Also note that noise and transients are acceptable levels			
	d. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all timing pulses at the users side of the harness. This is to be done for each bit rate			
	e. Measure the frequency, pulse amplitide, rise time, fall time, and the pulse width of all shift pulses at the users side of the harness. This is to be done for each bit rate			
	f. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all synch pulses at the users side of the harness			
	g. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all inhibit pulses at the users side of the harness. This is to be done for each bit rate			
	h. Check ID words corresponding to all bit rates and all formats using the telemetry data display EOSE			
	i. Loop check all analog words by applying a DC voltage at the senders side of the harness and reading out the decimal word at the telemetry data display EOSE			
	j. Loop check all digital words by applying a digital signal at the senders side of the harness and reading out the decimal word at the telemetry display EOSE Note: Noise, transient and cross talk measurements will be conducted for items c through g			
	k. Measure the subcarrier frequency and modulation index of the down link baseband signal			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
20	<u>Install Digital Storage Unit</u>	Hand tools, torque wrenches	Procedure	None
21	<u>Perform Digital Storage Unit Electrical Tests</u> The digital storage unit electrical testing will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the DSU power connector b. Connect the DSU to the spacecraft harness and measure the voltage and current drawn by the DSU. Also note that noise and transients are at acceptable levels c. Measure all command line voltages and currents for each DSU command. Also note that noise and transients are at acceptable levels d. Measure the rise time, fall time, amplitude, and pulse duration of the DSU input data signal at the DSU for each bit rate e. Measure the rise time, fall time, amplitude, and pulse duration of the DSU data output signal at the DTU during memory readout f. Measure the rise time, fall time, amplitude, and pulse duration of the DSU index pulse at the DTU Note: Noise, transient and cross talk measurements will be conducted for items d through f	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator, analog word format generator		

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 11

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
22	<u>Perform DTU, DSU, CS and C, Power and Equipment Compartment Structure Temperature Calibrations</u> These calibrations will be handled as follows: a. DTU temperature calibrations will be accomplished by replacing the transducer with precision resistors and noting the word value at the telemetry data display EOSE for each resistor value. The word values together with the factory transducer curves complete the calibration. Next, these parameters will be incorporated into the computer programs. The DTU analog to digital converter reference words are to be simply noted and recorded b. DSU temperature calibrations will be accomplished as in Task 22. a. 1. c. CS and C temperature calibrations will be accomplished as in Task 22. a. 1 d. Primary power calibrations will be accomplished by varying the load current and line voltage and monitoring the voltage and current with meters. The telemetry word values for each voltage and current will be recorded. These parameters will be inserted into the computer programs. Secondary power calibrations will be accomplished in the same manner as the primary power calibrations e. Equipment compartment structure temperature calibrations will be accomplished as in Task 22. a. 1 f. At the telemetry data display EOSE, verify that each command sent during items a through c above indicates the proper telemetry word value	Voltmeter, ammeter, decade resistance box, data center computer programs, telemetry data display EOSE, power supply, power EOSE, series fuse boxes, in-line test connectors	Procedure	None

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ____ Date ____ Approval ____ Page No. 12

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
23	<u>Install Command Detectors</u>	Hand tools, torque wrench		
24	<u>Perform Command Detector Electrical Tests</u> The command detector electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the command detector connectors b. Connect the detectors to the spacecraft harness and measure the secondary power supply voltage and current drawn by the detectors. Also note that noise and transients are at acceptable levels c. Measure the detector output peak to peak amplitude at the CS and C input in the presence of a simulated receiver signal (command encoder EOSE) d. Measure the bit synch rise time, fall time, pulse width, and amplitude e. Check that each command processor can be addressed only one separate address f. Check each detector synch lock operation with the command encoder g. Transmit each discrete command via the command encoder and observe that each command was received by observing the command matrix monitor h. Repeat the above for the redundant detector. Note that quantitative commands from each detector will be monitored during stabilization and control subsystem checkout	Power EOSE, voltmeter, ammeter, series fuse box, in-line test connector, command matrix, monitor command encoder	Procedure	None

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ______ Date ______ Approval ______ Page No. 13

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
25	<u>Perform Detector Calibrations</u> The detector temperature calibrations will be accomplished as in task 22.a.1.	Power EOSE, command encoder, resistor decade box, operational data center command, matrix monitor, in-line test connector	Procedure	None
26	<u>Install RF Diplexers, Couplers, Circulator Switches Band Pass Filters and Power Dividers</u>	Hand tools, torque wrenches	Procedure	None
27	<u>Install High-Gain Antenna and Cabling</u> This task is broken up into several subtasks as follows: a. Install high-gain antenna b. Connect, route, and clamp cabling c. Articulate antenna and check for cable chaffing and clearance d. Latch antenna in place	Hand tools, torque wrench, antenna drive EOSE	Procedure	None
28	<u>Install Medium Gain Antenna and Cabling</u> This task is broken up into several subtasks as follows: a. Install medium-gain antenna b. Connect, route, and clamp cabling c. Articulate antenna and check for cable chaffing and clearance d. Latch antenna in place	Hand tools, torque wrench, antenna drive EOSE		

Functional Flow Drawing Title and No. **Engineering Model Spacecraft Assembly and Checkout** Revision ______ Date ______ Approval ______ Page No. 14

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
29	Install Omni Antenna and Cabling This task is broken up into several subtasks as follows: a. Install omni antenna to omni antenna boom b. Install antenna and boom to spacecraft c. Connect, route, and clamp cabling d. Deploy and latch boom observing cable clearance and that no chaffing takes place e. Latch antenna boom in place	Hand tools, torque wrench	Procedure	None
30	Perform RF Insertion Loss Test The RF insertion loss determination will take place as follows: a. Connect the diplexers, couplers, bandpass filters, power monitors, and circulator switches to the RF cable harness system b. Measure the insertion loss between the receivers and the high-gain antenna c. Measure the insertion loss between the receivers and the low-gain antenna d. Measure the insertion loss between the receivers and the medium-gain antenna e. Measure the insertion loss between the power amplifiers and the high-gain antenna f. Measure the insertion loss between the power amplifiers and the low-gain antenna g. Measure the insertion loss between the power amplifiers and the medium-gain antenna h. Measure the insertion loss between the exciters and the high-gain antenna i. Measure the insertion loss between the exciters and the low-gain antenna j. Measure the insertion loss between the exciters and the medium-gain antenna k. Measure the insertion loss between the exciters and the power amplifiers	RF converter, adapters, RF generator, RF power meter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
31A	<u>Perform VSWR Tests</u> The VSWR tests will be performed as follows: a. After the insertion loss test has been completed, connect the high-gain and omni antennas to the RF cable harness b. Measure the VSWR between the receivers and the high-gain antenna c. Measure the VSWR between the receivers and the medium-gain antenna d. Measure the VSWR between the receivers and the low-gain antenna e. Measure the VSWR between the power amplifiers and the high-gain antenna f. Measure the VSWR between the power amplifiers and the medium-gain antenna g. Measure the VSWR between the power amplifier and the low-gain antenna h. Measure the VSWR between the exciters and the high-gain antenna i. Measure the VSWR between the exciters and the medium-gain antenna j. Measure the VSWR between the exciters and the low-gain antenna	RF connecter adapters, RF generator, RF couplers, VSWR meter, notch filters	Procedure	None
31B	<u>Receive Midcourse Propulsion and SCS Module</u> The midcourse propulsion and SCS module will be received from Douglas consisting of the following: a. Monopropellant engine and control valves b. Monopropellant engine feed system c. Monopropellant engine pressurization system d. Stabilization and control subsystem gas system e. Jet vane assemply installed in engine Note: Final TRW Quality Control buy-off will be performed at Douglas			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
32	Install Receivers and Receiver Selector Unit	Hand tools, torque wrench		
33	Perform Receiver Selector Electrical Tests The receiver electrical tests will be performed as follows: a. Apply external power to the spacecraft and check that voltage exists where it should and that no voltage exists at the remaining pins of the receiver selector connectors. b. Connect the receiver selector to the spacecraft harness and measure the voltage and current drawn by the selector. Also note that noise and transients are at acceptable levels c. Connect the receiver signal simulator to the receiver selector d. Simulate each receiver present signal and observe that the proper receiver is selected e. Simulate all combinations of the three receiver present signals and observe that the proper receiver is selected f. Simulate the loss of sun-Canopus and observe that receiver No. 1 is selected	Power EOSE, voltmeter, ammeter, oscilloscope, receiver, selector, simulator	Procedure	None
34A	Perform Receiver Electrical Tests The receiver electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector b. Connect each receiver to the spacecraft harness and measure the voltage and current drawn by each receiver. Note that noise and transients are within acceptable levels	RF EOSE, command encoder, command matrix monitor, voltmeter, ammeter, power EOSE, series fuse boxes, in-line test connectors	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Measure the modulation index of test transmitter output while it is being modulated with the command encoder and determine that it is within specification. This is to be done with and without the ranging signal d. Connect the receiver to a strong hardline signal from the RF EOSE (-110 dbm) and acquire e. Modulate the test transmitter (RF EOSE) with the command encoder and note that commands can be received and properly acted upon by the CS and C using each receiver through each antenna. This is to be accomplished by observing the command matrix monitor and by monitoring the appropriate telemetry word. Verify that the airborne receiver will acquire while the ground transmitter is being ramped at the maximum specified rate for given signal strengths f. Determine the signal strength at which the receiver thresholds or drops out of lock g. Verify that the receiver will stay acquired for the maximum specified ramp rate for given signal strengths h. Repeat above for the redundant receiver			
34B	<u>Perform Midcourse Propulsion and SCS Module Control Inspection</u> Quality control inspection is mainly for shipping damage as the module has previously been bought off at Douglas Aircraft Co. by TRW personnel	None	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
35A	<u>Perform Receiver Calibrations</u> The receiver calibrations will be performed as follows: a. Receiver temperature calibrations will be accomplished as in Task 22.a.1 b. The airborne receivers will be dropped in and out of lock by removing the test transmitter signal and noting that the telemetry indication is proper c. A precisely known signal level is fed into a precision step attenuator. A known signal strength can now be calculated for each attenuator setting. Each power level will be correlated with telemetry output d. After the receivers have been required by the test transmitter, the test transmitter frequency is varied and the loop stress telemetry output noted. All of the above parameters will be inserted into the computer programs	RF EOSE, command encoder, power EOSE, RF attenuators, calorimeter, data center, in-line test connector	Procedure	None
35B	<u>Perform Midcourse Propulsion and SCS Module Magnetics Property Test</u> The midcourse propulsion and SCS module magnetic properties check will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the magnetic field of the bus structure mounted in handling fixture c. Analyze all variations between readings and repeat if necessary	Magnetic measuring equipment, handling fixture, protective covers, handling slings	Procedure	Area in building free of large magnetic fields

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
36	<u>Install RF Exciters</u>	Hand tools, torque wrench	Procedure	None
37	<u>Perform Exciter Electrical Tests</u> The exciter electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector b. Connect the exciter to the spacecraft harness and measure the voltage and current drawn by the driver. Note that noise and transients are within acceptable limits c. Measure the rise time, fall time, and amplitude of the exciter modulation for each bit rate d. Remove modulation and measure the exciter RF power and frequency at the exciter output e. Measure the exciter modulation index with and without the ranging signal present f. Investigate driver output for spurious harmonics using a spectrum analyzer g. Connect the exciter output of the RF harness and ascertain that data can be received by the ground receiver (RF EOSE) through each antenna via air link h. Command the exciter to the coherent mode of operation and observe that driver output is 240/221 times the frequency of the ground transmitter i. Repeat above for the redundant exciter	RF EOSE, command encoder, power EOSE voltmeter, ammeter, series fuse boxes, in-line test connector, spectrum analyzer	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
38A	<u>Perform Exciter Calibrations</u> The exciter calibrations will be performed as follows: a. Exciter temperature calibration to be performed as in Task No. 22.a.1 b. Coherent/noncoherent mode to be performed by commanding the driver to the coherent and noncoherent modes of operation and noting that the proper telemetry word value exists	Power EOSE, RF EOSE, decade resistance box, series fuse boxes	Procedure	None
39	<u>Install Low Power Transmitter, Power Amplifiers, and Transmitter Selector Unit</u>	Hand tools, torque wrench		
40	<u>Perform Transmitter Selector Test</u> The transmitter selector electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins on each selector connector b. Connect the transmitter selector to the spacecraft harness and measure the voltage and current drawn from the secondary power supply subsystem. Note that noise and transients are within acceptable levels c. Simulate the appropriate transmitter modes via ground commands and CS and C back-up commands and ascertain that the proper transmitter was selected by monitoring the selector outputs.			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
41A	Perform Power Amplifier Tests The power amplifier tests will be performed as follows: a. Turn on external power to the spacecraft and command the power amplifier on b. Connect dummy loads to the power amplifier output c. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector d. Connect the power amplifier power to the spacecraft harness and measure the voltages and currents drawn by power amplifiers. Note that noise and transients are within acceptable levels e. Measure the power amplifier RF output power f. Measure the power amplifier modulation index with and without the ranging signal g. Measure the power amplifier output for spurious harmonics using a spectrum analyzer h. Connect the power amplifier to the RF cable harness i. Observe that telemetry can be received by the ground receiver (RF EOSE) from each antenna via air link j. Repeat for the redundant power amplifier	Power meter, NF-112 analyzer, power EOSE, RF EOSE, series fuse box, in-line test connectors	Procedure	None
41B	Install Solar Array Support Structure	Hand tools, torque wrenches		
42A	Perform Power Amplifier Calibrations The power amplifiers calibrations will be performed as follows: a. Temperature calibration will be performed as in Task 22.a.1 b. Step attenuators will be placed in the RF lines and the attenuator power measured. The measured power for each attenuator step is correlated with the telemetry output words. These parameters will be	Step attenuator, decade resistor box, Power EOSE RF EOSE, power meter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	inserted into the computer programs			
42B	Connect the SCS Nozzles and Gas Lines to the Spacecraft The SCS nozzles and gas lines will be connected to the spacecraft SCS pneumatics system	Hand tools	Procedure	None
43A	Perform Low Pressure SCS Leak Test The purpose of the low pressure leak test is to ascertain that the SCS pneumatic system leak rate is grossly within specification	Leak test console		
43B	Install Sun Sensor, Canopus Sensors and Shunt Regulators	Hand tools	Procedure	None
44	Perform Low Power Transmitter Electrical Tests a. Turn on external power to the spacecraft and command the low power transmitter on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins c. Connect the low power transmitter to the spacecraft harness and measure the voltage and current drawn from the secondary power subsystem. Note that noise and transients are within acceptable limits d. Measure the low power transmitter output and frequency e. Measure the low power transmitter output for spurious harmonics using a spectrum analyzer f. Measure the low power transmitter output modulation index g. Connect the low power transmitter to the RF cable harness h. Observe that telemetry can be received by the ground receiver through each antenna via air link	Voltmeter, ammeter, RF power meter, NF-112 analyzer, oscilloscope series fuse box, spectrum analyzer, oscilloscope, RF frequency counter	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
45	<u>Perform Low Power Transmitter Calibration</u> The low power transmitter calibration will be performed as follows: a. Temperature calibration will be performed as in Task 22.a.1 b. Step attenuators will be placed in the RF lines and the attenuator power measured. The measured power for each attenuator step is correlated with the telemetry output words. These parameters will be inserted into the computer programs	Step attenuator, decade resistor box, power EOSE, RF EOSE, power meter	Procedure	None
46	<u>Perform Gyro Package Alignment</u> The gyro package alignment is performed so that the gyro scale factors can be determined as part of the SCS testing phase	Gyro alignment set	Procedure	None
47	<u>Install the SCS Control Electronics Package, Drive Electronics Package and All SCS Sensors</u> The above items will be installed in the spacecraft in preparation for the SCS testing phase	Hand tools, torque wrenches		
48	<u>Perform Sun Acquisition Electrical Tests</u> The sun acquisition electrical tests will be performed as follows: a. Apply external power to the spacecraft and command the gyros to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector of the gyro package c. 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). Note that noise and transients on these lines are within acceptable levels	SCS EOSE, power EOSE, voltmeter, ammeter oscilloscope jet vane, angle MOSE, in-line test connector, series fuse box	Procedure	Tilt fixture should experience zero floor vibrations

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. 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			
	e. Connect the control signal electronics package to the spacecraft harness and measure the voltage and currents drawn by the control signal electronics package. Note that noise and transients on these lines are within acceptable levels			
	f. Torque the tilt fixture in the +yaw direction at a known rate and measure the yaw gyro output signal amplitude. Note that the polarity is correct			
	g. Torque the tilt fixture in the -yaw direction at a known rate and measure the yaw gyro output signal. Note that the polarity is correct			
	h. Repeat Step f for the pitch and roll gyros			
	i. With the spacecraft absolutely still, measure the noise amplitude on each gyro output line			
	j. Increase the rate in each axis in each direction and note that the proper gas valve is actuated			
	k. Determine the threshold rates in each axis which will just barely cause the gas valves to actuate			
	l. 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			
	m. Connect the sun sensors to the spacecraft harness			
	n. Attach the sun sensor stimulus to each sun sensor			
	o. Connect a voltmeter in place of each gas valve solenoid			
	p. Manually actuate separation switches and check that the sun acquisition mode has started			
	q. Transmit the back-up command for starting the sun acquisition sequence			
	r. Illuminate each sun sensor and check that voltage exists at each valve interface			

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	s. Connect each valve to the spacecraft harness t. 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 u. Observe that when each sun sensor is stimulated the proper valve is opened v. Observe that when all five sun sensor elements are illuminated, no valves are actuated			
49	Perform Sun Acquisition Calibrations The sun acquisition calibration will be performed as follows: a. The sun intensity signals will be simulated by replacing the sun sensor with a signal generator. As the voltage is varied, the telemetry word value is recorded for each sun sensor. The laboratory curves for each sun sensor (intensity versus voltage out) together with the digital word values will be inserted into the computer program b. The valve actuation signals will be calibrated by actuating each valve and noting the telemetry word values c. Control signal electronics package temperature calibration will be performed as per Task 22.a.1 d. Sun sensor temperatures calibration will be performed as per Task 22.2.1 e. The gyro temperature will be calibrated as per Task 22.a.1 f. Gyro on/off calibrations will be performed by commanding them on and then off and the telemetry word value recorded	Resistance decade box, Power EOSE, SCS EOSE, series fuse boxes, signal generator, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	g. Gyro pick-off outputs will be replaced with a signal generator. As the signal generator amplitude is varied, the telemetry word value is monitored. These parameters together with the laboratory bench test data (rate versus output voltage) will be inserted into the computer programs			
50	Perform Earth Sensor Electrical Tests The earth sensor electrical tests will be performed as follows: a. Turn on external power to the spacecraft and command the earth sensor to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the earth sensor connector c. 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 d. Darken the earth sensor appeture and measure the signal output amplitude. Note that noise and transients are within acceptable levels e. Attach the earth sensor stimulus to the earth sensor f. Illuminate the earth sensor and measure the sensor output signal amplitude. Note that noise and transients are within acceptable levels g. 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	SCS EOSE, power EOSE, earth sensor stimulus, voltmeter, ammeter, oscilloscope, series fuse box		

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
51	<u>Perform Earth Sensor Calibrations</u> The earth sensor calibrations will be performed as follows: a. The earth sensor will be replaced by a suitable signal generator. As the signal generator 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 b. The earth sensor temperature calibration will be performed as in Task 22.a.1	Signal generator, in-line connector, voltmeter, power EOSE, data center		
52A	<u>Perform Canopus Acquisition Tests</u> The Canopus acquisition electrical tests will be performed as follows: a. Turn off external power to the spacecraft and command the Canopus sensor to on b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the Canopus sensor connector c. 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 d. Attach Canopus sensor stimulus to the Canopus sensor e. Illuminate each half of the Canopus sensor field of view and note that the proper valves actuate when each half is illuminated f. 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	Voltmeter, ammeter, oscilloscope power EOSE, SCS EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	g. Illuminate the center of the Canopus sensor field of view and note that no valves are actuated h. Investigate the Canopus sensor signal output lines for out-of-tolerance transient and noise conditions when the center of the Canopus sensor is illuminated i. Command the spacecraft into the roll search mode and observe that the proper roll valves are actuated j. Remove Canopus sensor illumination and observe that the SCS subsystem goes into the roll search mode k. Note that the airborne receivers switch to the omni antenna when the Canopus illumination is removed			
52B	<u>Perform Planet-Oriented Package Magnetic Properties Test (Off Line)</u> The magnetics test will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the handling fixture magnetic field stability c. Measure the magnetic field of the planet-oriented package while mounted in the handling fixture	Protective covers, handling slings, magnetic field measuring equipment	Procedure	Area in building free of large magnetic fields (less than 50 gamma ambient field)
53	<u>Perform Canopus Acquisition Calibrations</u> The Canopus acquisition calibrations will be performed as follows: a. The Canopus sensor will be replaced by a suitable signal generator. As the generator 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	Signal generator, resistor decode box, power EOSE, data center	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. The Canopus sensor intensity signal will be performed as in Step a above c. The Canopus sensor temperature calibrations will be performed as in Task 22.a.1			
54	<u>Perform Spacecraft Midcourse Maneuver Tests</u> The spacecraft maneuver testing will be accomplished as follows: a. Enter the roll turn and polarity information into the command detector b. Execute the roll turn command, measure and time the gyro output and input signals and note that noise and transients are within acceptance levels c. Note that the proper gas valves are actuated while the gyro is being torqued d. Repeat Steps b and c for the opposite polarity turn e. Repeat Steps a through d for the pitch turn f. Load velocity increment information into the command detector to activate jet vane control noting that the proper voltage amplitudes exist at each jet vane actuator connector g. Repeat Step f for the opposite polarity h. Connect the midcourse motor jet vanes to the spacecraft harness re-insert the velocity increment and measure the voltage and current drawn by each jet vane actuator. Note that noise and transients on these lines are within acceptable limits i. Measure the jet vane angle with respect to the sun-line j. Repeat Step i for the opposite polarity velocity increment	ECS EOSE, power EOSE, voltmeter, ammeter, oscilloscope, jet vane, angle MOSE, in-line test connector, series fuse box	Procedure	Tilt fixture should experience zero floor vibrations

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	k. Enter midcourse motor burn duration information into the command detector and measure the midcourse motor ignitor firing voltage and turnoff voltage and the time duration between the turn on signal and turn off signal			
55	<u>Perform Spacecraft Midcourse Calibrations</u> The spacecraft maneuver calibrations will be performed as follows: a. Jet vane actuator temperature calibrations are to be performed as per Task 22. a. 1 b. Jet vane angle calibrations are to be performed by turning the jet vanes to known angles and recording the telemetry word values. These parameters are then inserted into the computer program	Power EOSE, SCS EOSE angle gauges decade resistor box in-line test connector	Procedure	None
56	<u>Install Planet-Oriented Package</u> The planet-oriented package consists of the following items: a. Planet-oriented package boom b. Planet-oriented package gimble actuators c. Mars horizon scanners	Hand tools, torque wrenches	Procedure	None
57	<u>Perform Planet Oriented-Package Stabilization and Control Testing</u> The planet-oriented and stabilization and control testing will take place as follows: a. Connect the planet-oriented package boom connector to the main spacecraft harness	Voltmeter, ammeter, oscilloscope, power EOSE, SCS EOSE, horizon scanner stimulus	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Apply external power to spacecraft and command the planet-oriented package control system to on			
	c. Measure voltage where it should be and no voltage on the remaining pins of each connector of the drive electronics package			
	d. Connect the drive electronics package to the spacecraft harness and measure the voltage and current drawn by the drive electronics package			
	e. Measure voltage where it should be and no voltage on the remaining pins of each connector of the Mars horizon scanner package			
	f. Connect the Mars horizon scanner package to the boom harness			
	g. With the horizon scanner package sensors completely darkened, observe that the horizon scanner output and drive electronics package output signal noise and transient levels are within acceptable levels			
	h. Attach the horizon scanner stimulus EOSE to the horizon scanner package			
	i. Stimulate each horizon scanner and note that the proper gimble slews in the proper direction at the proper rate			
	j. Repeat Step i for the opposite polarity for each scanner			
	k. Slew each gimble by means of the horizon scanner stimulus and measure the drive voltage amplitude at the gimble actuators. Measure the voltage and current drawn by the drive electronics package from the secondary power subsystem noting that noise and transient conditions are within specification			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
58	<u>Perform Planet-Oriented Package Stabilization and Control Calibrations</u> The planet-oriented package stabilization and control calibrations will take place as follows: a. The planet-oriented package temperature will be calibrated as per Task 22.a.1 b. Each gimble will be turned to known angles during which the telemetry word values are monitored and recorded. The gimble angle information and the telemetry word values are inserted into the computer programs c. The horizon scanner sensor output calibrations will be performed by replacing the horizon scanner with a signal generator and opening each gimble control loop. As the signal generator output amplitude is varied the telemetry word values are monitored and recorded. These parameters along with the horizon scanner laboratory bench information (scanner error in degrees versus output voltage) are inserted into the computer program	Resistor decade box, gimble, angle indicator, signal generator, power EOSE data center	Procedure	None
59	<u>Perform High-Gain Antenna Gimble Actuator Tests</u> The gimble actuator tests will be performed as follows: a. Turn on external power to the spacecraft and command the antenna to slew b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the gimble actuator connectors c. Measure the drive signal amplitude d. Repeat Steps a, b, and c for the remaining gimble axis	Voltmeter, ammeter, power EOSE, command EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	e. Connect the gimble actuators to the harness and command the gimble to slew f. Measure the voltage amplitude and current drawn by the drive electronics from the secondary power subsystem while the gimble is being slewed noting that noise and transient conditions are within specification g. Repeat Step f for each gimble in each direction h. Observe that the gimble slews at the proper rate for each direction			
60	<u>Perform High Gain Antenna Gimble Actuator Calibrations</u> The actuator calibrations will be performed as follows: a. The actuator temperature calibrations will be performed as per Task 22. a. 1 b. Gimble angle calibrations will be performed by slewing each gimble to a known angle and observing and recording the telemetry word values. These parameters are then inserted into the computer program	Resistor decade box, gimble angle indicator, power EOSE, command EOSE, data center	Procedure	None
61	<u>Perform Medium-Gain Antenna Gimble Actuator Tests</u> The gimble actuator tests will be performed as follows: a. Turn on external power to the spacecraft and command the antenna to slew b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the gimble actuator connectors. c. Measure the drive signal amplitude	Voltmeter, ammeter, power EOSE, command EOSE	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. Connect the gimble actuator to the harness and command the gimble to slew e. Measure the voltage amplitude and current drawn by the drive electronics from the secondary power subsystem while the gimble is being slewed noting that noise and transient conditions are within specification f. Repeat Step e for each gimble in each direction g. Observe that each gimble slews at the proper rate in each direction			
62	Perform Medium-Gain Antenna Gimble Actuator Calibrations The actuator calibrations will be performed as follows: a. The actuator temperature calibrations will be performed as per Task 22.a.1 b. Gimble angle calibrations will be performed by slewing the gimble to a known angle and observing and recording the telemetry word values. These parameters are then inserted into the computer program	Resistor, decade box, gimble angle indicator, power EOSE, command EOSE, data center	Procedure	None
63	Install Inert Solid Retropropulsion Engine Subsystem	Hand tools, torque wrenches	Procedure	None
64	Perform Terminal Maneuver Testing The terminal maneuver testing will be performed as follows: a. Perform terminal turn maneuvers via the CC and S. These maneuvers are accomplished in the same manner as in the midcourse turn maneuvers	Voltmeter, ammeter, oscilloscope, SCS EOSE, power EOSE, data center, solid motor pressurizing test set	Procedure	Darkened room

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Perform spacecraft deboost tests 1) Connect cold gas supply to the solid motor TVC test connector 2) Perform the spacecraft terminal maneuver and command the thrust vector control system to on 3) Torque the tilt fixture in the pitch axis and monitor the pitch injector signal amplitude at the solid retromotor 4) Repeat Step 3) for the opposite direction 5) Repeat Step 3) for the yaw axis in both directions 6) Connect the solid motor thrust vector control system to the spacecraft harness 7) Torque the tilt table in the pitch and yaw axis in both polarities 8) While the spacecraft is being torqued, observe that gas is flowing through the proper injector 9) While the spacecraft is being torqued, measure the voltage and current drawn from the secondary power supply subsystem and note that noise and transients are within acceptable limits 10) Connect the ordance EOSE to the solid motor ignitor system 11) Transmit the solid motor ignition command and verify performance by observing the EOSE (all fire indication) and the telemetry indications			
65	<u>Perform Terminal Maneuver Calibrations</u> The terminal maneuver calibrations will be performed as follows: a. Solid motor thrust vector control temperature calibration will be performed as per Task 22.a.1 b. Each thrust vector control injector actuator will be energized and the telemetry word monitored correct value	Ordance EOSE		

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
66	<u>Perform Data Automation Equipment Electrical Test</u> The data automation electrical test will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins at the DAE power input connector b. Connect the DAE to the 4.1 kc inverter and measure the voltage and current drawn by the DAE and note that noise and transients are at acceptable levels c. Measure command line voltage and current drawn for each bit rate, format, and mode of operation, note that noise and transients are at acceptable levels d. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all timing pulses at the experimenters side of the harness. This is to be done for each bit rate. e. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all shift pulses at the experimenters side of the harness. This is to be done for each bit rate f. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all synch pulses at the experimenters side of the harness. This is to be done for each bit rate g. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all inhibit pulses at the experimenters side of the harness. This is to be done for each bit rate	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator, analog word simulator		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	h. Check ID words corresponding to all bit rates and all formats using the telemetry data display EOSE. i. Loop check all analog words by applying a DC voltage at the experimenters side of the harness and reading out the decimal word at the telemetry data display EOSE. j. Loop check all digital words by applying a digital signal at the senders side of the harness and reading out the decimal word at the telemetry data display EOSE. Note: Noise, transient and cross talk measurements will be conducted for items c. through g.			
67	<u>Install Bulk Storage Units</u>	Hand tools, torque wrenches	Procedure	None
68	<u>Perform Bulk Storage Unit Electrical Tests</u> The bulk storage unit electrical testing will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the bulk storage power connector. b. Connect the bulk storage to the spacecraft harness and measure the voltage and current drawn by the bulk storage. Also note that noise and transients are at acceptable levels.	Fully operated data center, operational computer programs, telemetry data display, EOSE ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Measure all command line voltages and currents for each bulk storage command. Also note that noise and transients are at acceptable levels.	connectors, digital word data format generator		
	d. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage input data signal at the bulk storage for each bit rate.			
	e. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage data output signal at the DAE during memory readout.			
	f. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage index pulse at the DAE.			
	Note: Noise, transient and cross talk measurements will be conducted for items d. through f.			
69	<u>Perform Data Automation and Bulk Storage Calibrations</u>	Power EOSE, data center, resistor decade box, command EOSE, in-line test connector	Procedure	None
	These calibrations are temperature calibrations and will be performed as follows:			
	a. DAE temperature calibration is to be performed as per Task 22.a.1.			
	b. Bulk storage temperature calibration is to be performed as per Task 22.a.1.			
70	<u>Install Capsule Receivers and Detectors</u>	Hand tools, torque wrenches	Procedure	None
71	<u>Perform VHF Capsule Receiver Electrical Tests</u>	RF EOSE, command matrix		

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	The receiver electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector. b. Connect each receiver to the spacecraft harness and measure the voltage and current drawn by each receiver noting that noise and transient conditions are within specification. c. Connect the receiver to a strong signal from the capsule EOSE (-110 dbm) and acquire. d. Determine the signal at which the receiver thresholds or drops out of lock. e. Modulate the capsule simulator and measure the receiver output signal amplitude. f. Repeat the above steps for the redundant receiver.	monitor, voltmeter, ammeter, power EOSE, series fuse boxes, in-line test connectors, capsule, simulator		
72	<u>Perform VHF Capsule Receiver Calibration</u> a. Receiver temperature calibrations will be accomplished as in Task 22.a.1. b. The airborne receivers will be dropped in and out of lock by removing the capsule simulator signal and noting that the telemetry indication is proper. c. A precisely known signal level is fed into a precision step attenuator. A known signal strength can now be calculated for each attenuator setting. Each power level is correlated with telemetry output. These parameters are entered in the computer program.	RF EOSE, power EOSE, RF attenuators, calorimeter, data center, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
73	<u>Perform Capsule Detector Test</u> The capsule detector will be tested as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the detector power connector. b. Connect the detector to the spacecraft harness and measure the voltage and current drawn by the detector. Also note that noise and transients are at acceptable levels. c. Acquire the capsule simulator and measure the amplitude, rise time, and fall time of the detector output signal.			
74	<u>Perform Pyrotechnic Subsystem Integration Tests</u> The pyrotechnic subsystem integration encompasses the following areas: a. Experiment ordnance b. Experiment boom ordnance c. High-gain antenna boom ordnance d. Medium-gain antenna boom ordnance e. Low-gain antenna boom ordnance f. Planet-oriented package boom ordnance g. Midcourse correction motor ordnance h. Solid retropropulsion engine ordnance i. Capsule separation ordnance The pyrotechnic subsystem ordnance tests will be performed as follows:	Ordnance EOSE, system test set EOSE	Procedure	None

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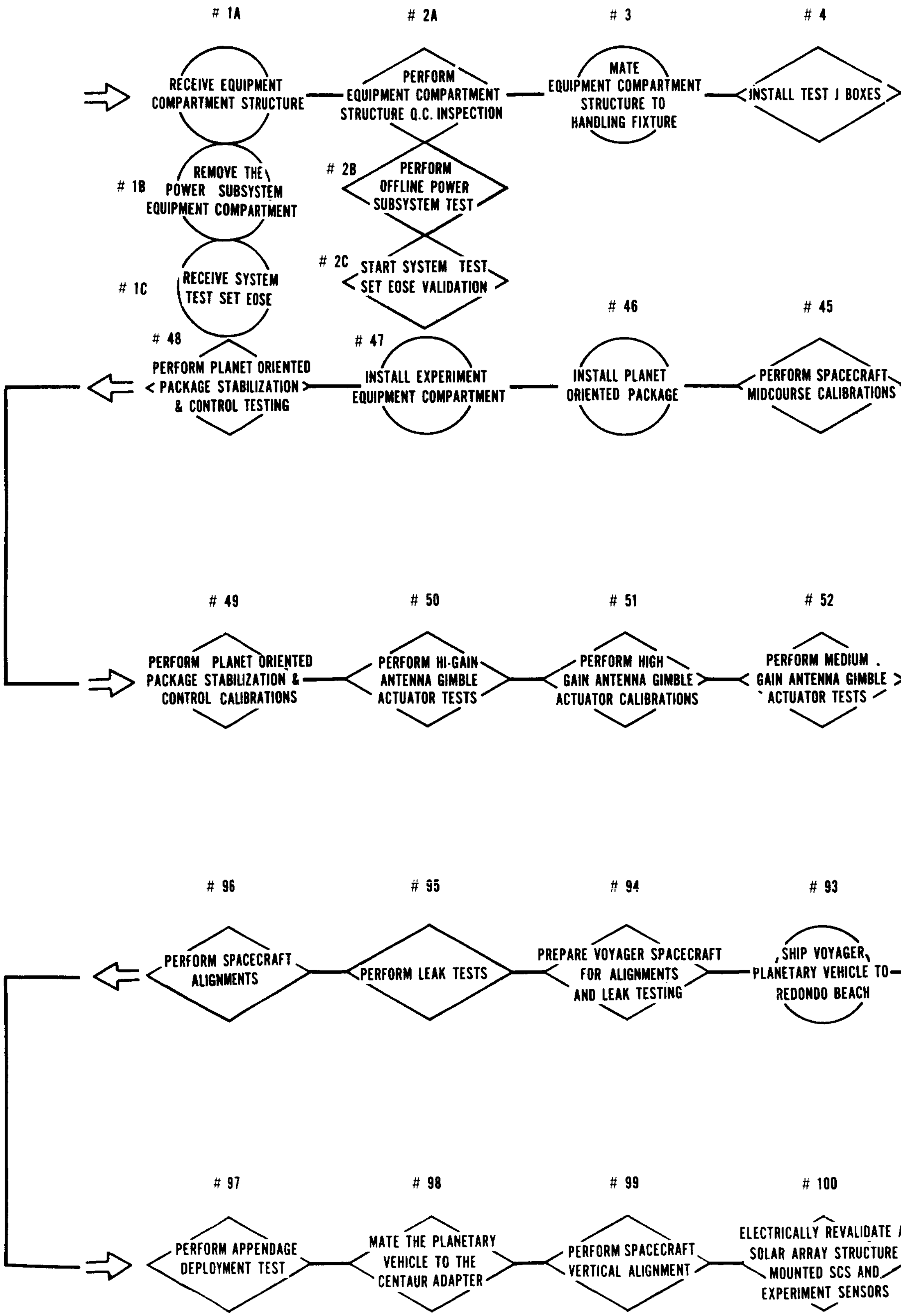
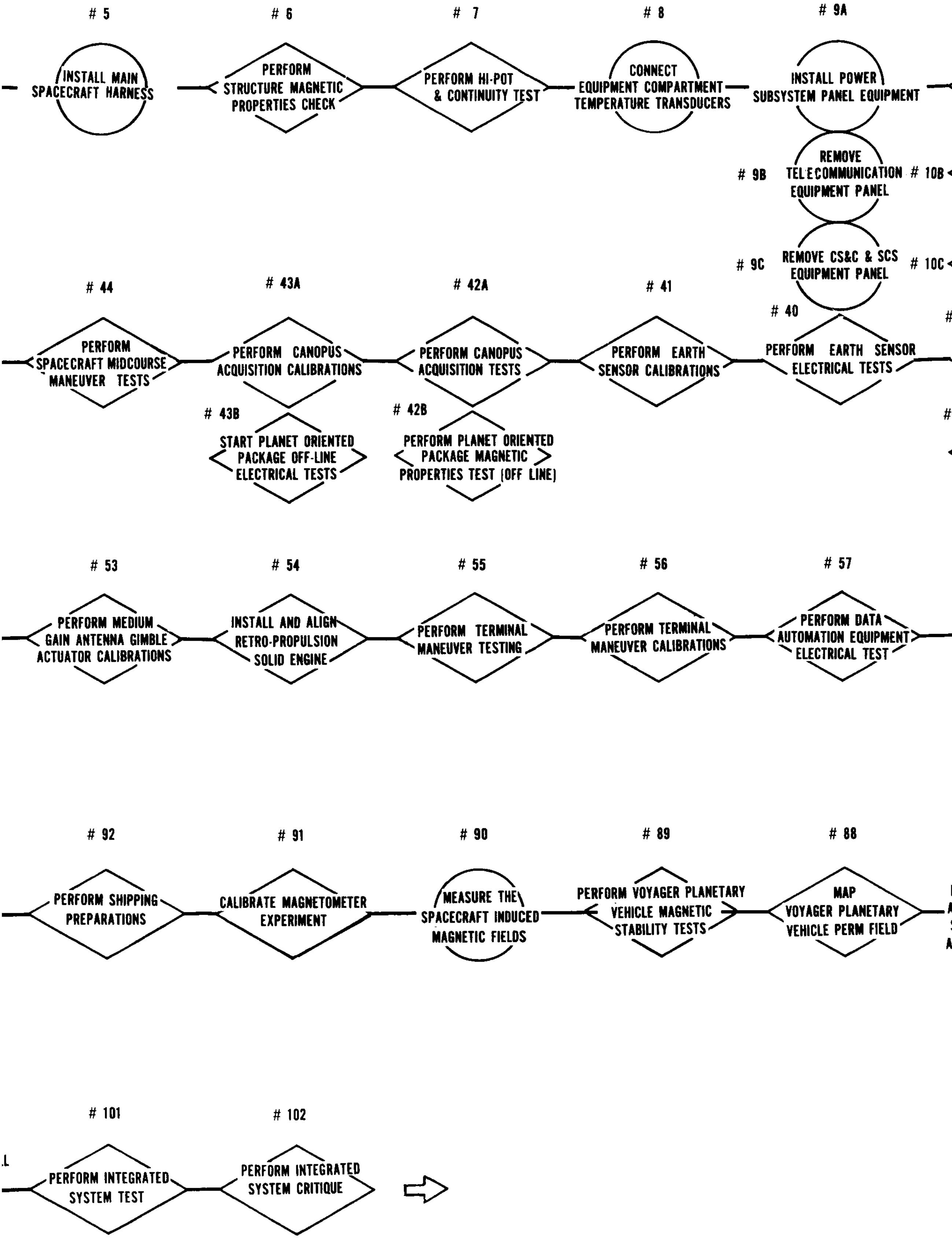
Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. Ascertain that the pyrotechnic subsystem is in a safe condition by monitoring across each squib bridge wire connector a dead short. b. Command each squib to the fire condition and monitor the "firing" voltage at each squib bridge wire connector. c. Connect the pyrotechnic EOSE to each squib connector and command each squib to the "fire" condition. d. Ascertain that an "all-fire" indication exists for each squib actuation. e. Command each squib to the "fire" condition using under-voltage conditions and ascertain that a "no fire" condition exists for each squib actuation.			
75	<u>Perform Pyrotechnic Subsystem Calibrations</u> The pyrotechnic subsystem will be calibrated by commanding each squib actuation and monitoring each telemetry word for correct value.			
76	<u>Perform IST and Critique</u>			

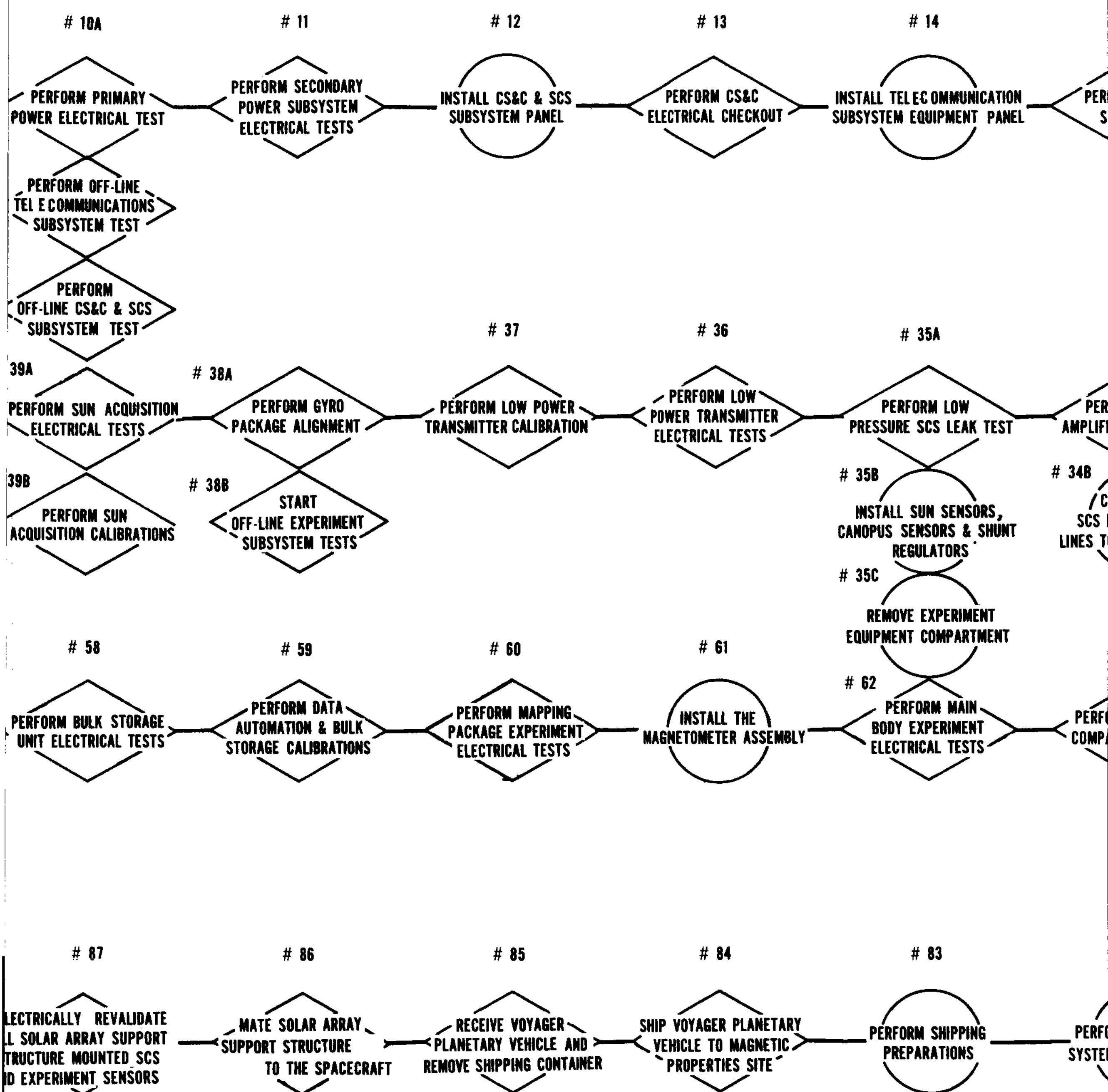


1A
RECEIVE EQUIPMENT COMPARTMENT STRUCTURE
2A
PERFORM EQUIPMENT COMPARTMENT STRUCTURE Q.C. INSPECTION
3
MATE EQUIPMENT COMPARTMENT STRUCTURE TO HANDLING FIXTURE
4
INSTALL TEST J BOXES
1B
REMOVE THE POWER SUBSYSTEM EQUIPMENT COMPARTMENT
2B
PERFORM OFFLINE POWER SUBSYSTEM TEST
1C
RECEIVE SYSTEM TEST SET EOSE
2C
START SYSTEM TEST SET EOSE VALIDATION
48
PERFORM PLANET ORIENTED PACKAGE STABILIZATION & CONTROL TESTING
47
INSTALL EXPERIMENT EQUIPMENT COMPARTMENT
46
INSTALL PLANET ORIENTED PACKAGE
45
PERFORM SPACECRAFT MIDCOURSE CALIBRATIONS
49
PERFORM PLANET ORIENTED PACKAGE STABILIZATION & CONTROL CALIBRATIONS
50
PERFORM HI-GAIN ANTENNA GIMBLE ACTUATOR TESTS
51
PERFORM HIGH GAIN ANTENNA GIMBLE ACTUATOR CALIBRATIONS
52
PERFORM MEDIUM GAIN ANTENNA GIMBLE ACTUATOR TESTS
96
PERFORM SPACECRAFT ALIGNMENTS
95
PERFORM LEAK TESTS
94
PREPARE VOYAGER SPACECRAFT FOR ALIGNMENTS AND LEAK TESTING
93
SHIP VOYAGER PLANETARY VEHICLE TO REDONDO BEACH
97
PERFORM APPENDAGE DEPLOYMENT TEST
98
MATE THE PLANETARY VEHICLE TO THE CENTAUR ADAPTER
99
PERFORM SPACECRAFT VERTICAL ALIGNMENT
100
ELECTRICALLY REVALIDATE SOLAR ARRAY STRUCTURE MOUNTED SCS AND EXPERIMENT SENSORS

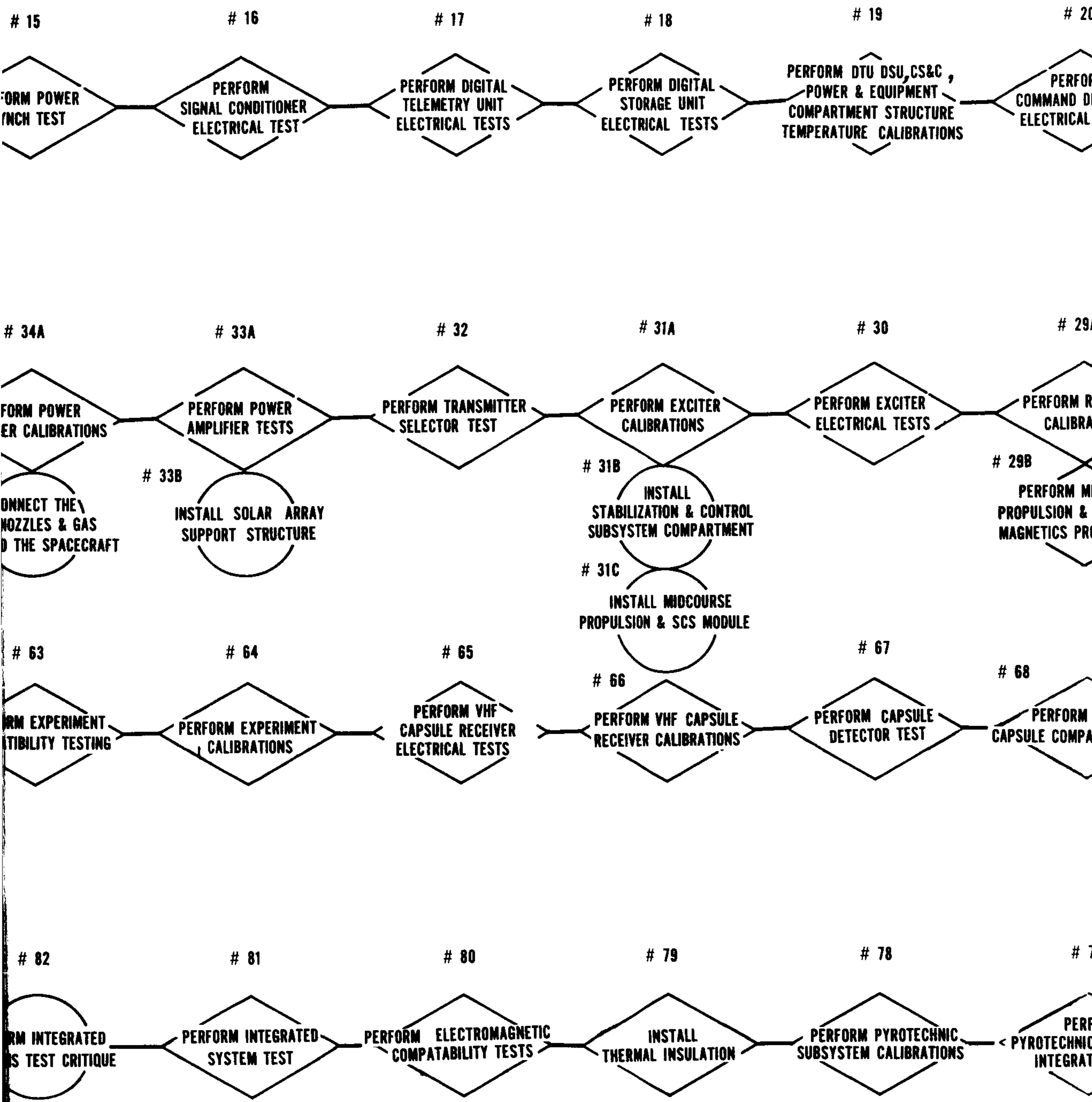


5
INSTALL MAIN SPACECRAFT HARNESS
6
PERFORM STRUCTURE MAGNETIC PROPERTIES CHECK
7
PERFORM HI-POT & CONTINUITY TEST
8
CONNECT EQUIPMENT COMPARTMENT TEMPERATURE TRANSDUCERS
9A
INSTALL POWER SUBSYSTEM PANEL EQUIPMENT
9B
REMOVE TELECOMMUNICATION EQUIPMENT PANEL
10B
9C
REMOVE CS&C & SCS EQUIPMENT PANEL
10C
44
PERFORM SPACECRAFT MIDCOURSE MANEUVER TESTS
43A
PERFORM CANOPUS ACQUISITION CALIBRATIONS
42A
PERFORM CANOPUS ACQUISITION TESTS
41
PERFORM EARTH SENSOR CALIBRATIONS
40
PERFORM EARTH SENSOR ELECTRICAL TESTS
43B
START PLANET ORIENTED PACKAGE OFF-LINE ELECTRICAL TESTS
42B
PERFORM PLANET ORIENTED PACKAGE MAGNETIC PROPERTIES TEST (OFF LINE)
53
PERFORM MEDIUM GAIN ANTENNA GIMBLE ACTUATOR CALIBRATIONS
54
INSTALL AND ALIGN RETRO-PROPULSION SOLID ENGINE
55
PERFORM TERMINAL MANEUVER TESTING
56
PERFORM TERMINAL MANEUVER CALIBRATIONS
57
PERFORM DATA AUTOMATION EQUIPMENT ELECTRICAL TEST
92
PERFORM SHIPPING PREPARATIONS
91
CALIBRATE MAGNETOMETER EXPERIMENT
90
MEASURE THE SPACECRAFT INDUCED MAGNETIC FIELDS
89
PERFORM VOYAGER PLANETARY VEHICLE MAGNETIC STABILITY TESTS
88
MAP VOYAGER PLANETARY VEHICLE PERM FIELD
101
PERFORM INTEGRATED SYSTEM TEST
102
PERFORM INTEGRATED SYSTEM CRITIQUE

2



10A
PERFORM PRIMARY POWER ELECTRICAL TEST
PERFORM OFF-LINE TEL E COMMUNICATIONS SUBSYSTEM TEST
PERFORM OFF-LINE CS&C & SCS SUBSYSTEM TEST
11
PERFORM SECONDARY POWER SUBSYSTEM ELECTRICAL TESTS
12
INSTALL CS&C & SCS SUBSYSTEM PANEL
13
PERFORM CS&C ELECTRICAL CHECKOUT
14
INSTALL TEL EC OMMUNICATION SUBSYSTEM EQUIPMENT PANEL
PER
S
39A
PERFORM SUN ACQUISITION ELECTRICAL TESTS
39B
PERFORM SUN ACQUISITION CALIBRATIONS
38A
PERFORM GYRO PACKAGE ALIGNMENT
38B
START OFF-LINE EXPERIMENT SUBSYSTEM TESTS
37
PERFORM LOW POWER TRANSMITTER CALIBRATION
36
PERFORM LOW POWER TRANSMITTER ELECTRICAL TESTS
35A
PERFORM LOW PRESSURE SCS LEAK TEST
35B
INSTALL SUN SENSORS, CANOPUS SENSORS & SHUNT REGULATORS
35C
REMOVE EXPERIMENT EQUIPMENT COMPARTMENT
PER
AMPLIFI
34B
C
SCS
LINES T
58
PERFORM BULK STORAGE UNIT ELECTRICAL TESTS
59
PERFORM DATA AUTOMATION & BULK STORAGE CALIBRATIONS
60
PERFORM MAPPING PACKAGE EXPERIMENT ELECTRICAL TESTS
61
INSTALL THE MAGNETOMETER ASSEMBLY
62
PERFORM MAIN BODY EXPERIMENT ELECTRICAL TESTS
PERF
COMP
87
LECTRICALLY REVALIDATE
L SOLAR ARRAY SUPPORT
TRUCTURE MOUNTED SCS
D EXPERIMENT SENSORS
86
MATE SOLAR ARRAY SUPPORT STRUCTURE TO THE SPACECRAFT
85
RECEIVE VOYAGER PLANETARY VEHICLE AND REMOVE SHIPPING CONTAINER
84
SHIP VOYAGER PLANETARY VEHICLE TO MAGNETIC PROPERTIES SITE
83
PERFORM SHIPPING PREPARATIONS
PERF
SYSTE



15
FORM POWER
NCH TEST
16
PERFORM SIGNAL CONDITIONER ELECTRICAL TEST
17
PERFORM DIGITAL TELEMETRY UNIT ELECTRICAL TESTS
18
PERFORM DIGITAL STORAGE UNIT ELECTRICAL TESTS
19
PERFORM DTU DSU, CS&C, POWER & EQUIPMENT COMPARTMENT STRUCTURE TEMPERATURE CALIBRATIONS
20
PERFORM
COMMAND DE
ELECTRICAL T
34A
FORM POWER
ER CALIBRATIONS
ONNECT THE
NOZZLES & GAS
D THE SPACECRAFT
33A
PERFORM POWER AMPLIFIER TESTS
33B
INSTALL SOLAR ARRAY SUPPORT STRUCTURE
32
PERFORM TRANSMITTER SELECTOR TEST
31A
PERFORM EXCITER CALIBRATIONS
31B
INSTALL STABILIZATION & CONTROL SUBSYSTEM COMPARTMENT
31C
INSTALL MIDCOURSE PROPULSION & SCS MODULE
30
PERFORM EXCITER ELECTRICAL TESTS
29A
PERFORM RE
CALIBRAT
29B
PERFORM MID
PROPULSION & S
MAGNETICS PRO
63
RM EXPERIMENT
TIBILITY TESTING
64
PERFORM EXPERIMENT CALIBRATIONS
65
PERFORM VHF CAPSULE RECEIVER ELECTRICAL TESTS
66
PERFORM VHF CAPSULE RECEIVER CALIBRATIONS
67
PERFORM CAPSULE DETECTOR TEST
68
PERFORM D
CAPSULE COMPAT
82
RM INTEGRATED
S TEST CRITIQUE
81
PERFORM INTEGRATED SYSTEM TEST
80
PERFORM ELECTROMAGNETIC COMPATABILITY TESTS
79
INSTALL THERMAL INSULATION
78
PERFORM PYROTECHNIC SUBSYSTEM CALIBRATIONS
77
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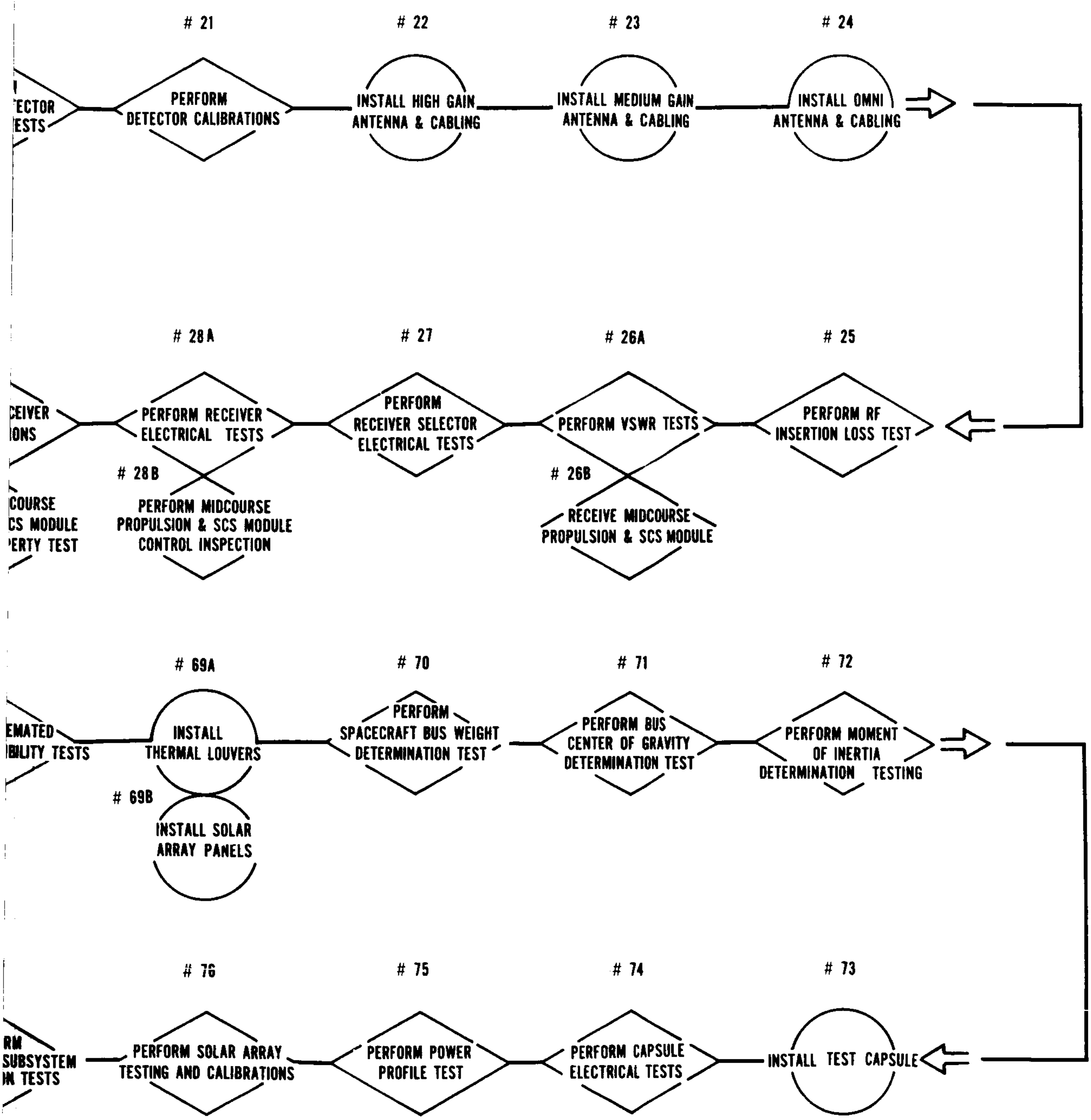
4



1971 PROOF TEST MODEL S/C ASSEMBLY AND TEST

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1A	Receive Equipment Compartment Structure The spacecraft equipment compartment structure will be received from Douglas Aircraft Co. in the following configuration: a. Solar array support structure not installed b. Main spacecraft harness not installed c. Thermal insulation not installed d. Thermal louvers not installed e. Propulsion system not installed f. Equipment compartment structure temperature transducers installed g. Planet-oriented package and support fixture not installed h. High-gain antenna and support structure not installed i. Medium-gain antenna and boom not installed j. Omni antenna and boom not installed k. Magnetometer and boom not installed l. Solid inert motor not installed m. TRW quality control buy-off will be performed at Douglas Aircraft Co.	Tools to uncrate structure	None	None
1B	Remove the Power Subsystem Equipment Compartment The power subsystem equipment compartment will be removed from the spacecraft and individual power modules mechanically installed off-line in the compartment.	Hand tools	None	None
1C	Receive System Test Set EOSE	None	Equipment list	None
2A	Perform Equipment Compartment Structure Quality Control Inspection Quality control inspection is mainly for shipping damage as the equipment compartment structure will have been already bought off at Douglas Aircraft Co.	None	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
2B	Perform Off-Line Power Subsystem Test The power subsystem will be completely checked as a subsystem off-line.	Power subsystem panel EOSE	None	None
2C	Start System Test Set EOSE Validation The system test set EOSE validation will take place for two reasons: a. To ensure that the EOSE has survived the shipping and handling operations b. To familiarize the test crews with the EOSE			
3	Mate Equipment Compartment Structure to Handling Fixture This task is a two step task: mate MOSE adapter to spacecraft structure and handling fixture.	Handling sling, adapter handling fixture, protective covers, hand tools	Procedure	None
4	Install Test J Boxes Install all electrical test J boxes to support the hi-pot and continuity test.	Hand tools, torque wrench	Procedure	None
5	Install Main Spacecraft Harness Install main spacecraft electrical harness and connect to J boxes	Hand tools, torque wrench, handling sling	Procedure	None
6	Perform Structure Magnetic Properties Check The equipment compartment magnetic properties check will be conducted as follows:	Magnetic measuring equipment, handling fixture,		

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. Measure the magnetic field of the handling fixture b. Measure the magnetic field of the equipment compartment structure mounted in handling fixture c. Analyze all variations between readings and repeat if necessary	Protective covers, handling slings	Procedure	Area in building free of large magnetic fields
7	<u>Perform Hi-Pot and Continuity Test</u> This is to be accomplished using a Huges FACT machine or equivalent. Wherever possible the test will be run end to end through all J boxes.	Huges FACT machine or equivalent, cable adapters, FACT machine programs	Procedure	None
8	<u>Connect Equipment Compartment Temperature Transducers</u> Solder all temperature transducers to main spacecraft harness.	Soldering iron, solder, insulation	None	None
9A	<u>Install Power Subsystem Panel Equipment</u> The power subsystem black boxes will be installed on the PTM power panel in preparation for off line testing.	Hand tools, torque wrench	Procedure	None
9B	<u>Remove Telecommunication Equipment Panel</u> The telecommunications panel equipment will be removed from the spacecraft and the individual modules mechanically installed in the equipment compartment in preparation for off line testing.	Hand tools	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
9C	Remove CS and C and SCS Equipment Panel The CS and C and SCS equipment panels will be removed from the spacecraft and the individual power modules mechanically installed off-line in the equipment compartment in preparation for off-line testing.	Hand tools	None	None
10A	Perform Primary Power Electrical Test The primary power subsystem consists of the following items: a. Batteries b. Power control unit c. Shunt regulators d. Battery boost regulator The subsystem electrical tests will be performed as follows: a. Integrate power EOSE b. Perform bus open circuit checks using the EOSE external power mode c. Perform bus open circuit checks using the spacecraft batteries d. Perform bus open circuit checks using solar array simulated power e. Load electrical bus using dummy loads and electrically test the power control unit and shunt regulators using the spacecraft batteries and the solar array simulator. Commands will be simulated by using an external power supply that will be part of one of the load boxes f. Remove loads from bus and connect boost regulator g. Power boost regulator from external power and measure output current	Voltmeters, ammeters, oscilloscope, power supply, power EOSE, series fuse boxes, in-line test connectors	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	h. Load boost regulator output and measure the input and output voltage and current. Also note that noise on the output lines is within acceptable limits. Note: All loads are to be applied at the users side of the harness.			
10B	<u>Perform Off-Line Telecommunications Subsystem Test</u> The telecommunications panel electronics will be completely checked as a subsystem, off-line using the panel test EOSE.	Telecommunications subsystem panel EOSE		
10C	<u>Perform Off-Line CSØC and SCS Subsystem Test</u> The central sequencer and control subsystem and the stabilization and control subsystem panel electronics will be completely checked out as individual subsystems using the panel test EOSE.	CSØC and SCS subsystem penel EOSE,		
11	<u>Perform Secondary Power Subsystem Electrical Tests</u> The secondary power subsystem consists of the following items: a. 4.1 kc, 1 Ø inverter b. 820 cps, 2 Ø inverter c. 410 cps 1 Ø inverter The secondary power subsystem test will be performed as follows: a. Connect 4.1-kc inverter to the spacecraft main harness b. Check 4.1-kc inverter open circuit voltage by powering the bus on external power c. Load 4.1-kc inverter using dummy loads and check output current and voltage d. Repeat steps a, b and c for the 820-cps and 410 cps inverters. Note: All load boxes are to be applied at the users side of the harness.	Ammeters, voltmeters, oscilloscope power EOSE, series fuse boxes, in-line test connectors	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
12	Install CSØC and SCS Subsystem Panel	Hand tools	None	None
13	Perform CSØC Electrical Checkout The central sequencer and control unit electrical checkout will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the CSØC power input connector b. Connect the CSØC to the spacecraft harness and measure the voltage and current drawn by the CSØC. Also note that noise and transients are at acceptable levels c. Connect command detector format generator to the CSØC at the detector side of the spacecraft harness d. Check all of the power control unit commands as follows: 1) Open all command lines from the CSØC at the PCU side of the spacecraft harness 2) Transmit all PCU commands via the command format generator 3) Observe the open circuit command signal voltage at the PCU 4) Close the command lines to the PCU and re-transmit the PCU commands via the command format generators 5) Monitor the command voltage and current at the PCU 6) Observe command signal lines and note that noise and transients are at acceptable levels 7) Observe that the PCU reacts properly to the CSØC commands	Command format generators, voltmeters, oscilloscope, ammeter, power EOSE, series fuse boxes, in-line test connectors, command matrix monitor	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	e. Check the open circuit voltage of the remaining discrets command lines from the CS$\emptyset$C at the side of the spacecraft harness. Note: The noise and transient levels on each of the remaining command signal lines will be checked during the electrical integration of the remaining subsystems f. Transmit each quantitative command from the format generator and observe that each command was properly received by observing the command matrix monitor g. Measure the amplitude and frequency of the down link PN subcarrier h. Measure the frequency and amplitude of all CS$\emptyset$C timing signals			
14	<u>Install Telecommunication Subsystem Equipment Panel</u>	Hand tools, torque wrench	Procedure	None
15	<u>Perform Power Synch Test</u> The power synch tests will be performed in the following manner: a. Apply external power to the spacecraft and observe the open circuit frequency, rise time, fall time, pulse width, and amplitude of each synch pulse from the CS$\emptyset$C to the boost regulator and each inverter b. Connect the synch pulse to the boost regulator and observe the frequency, rise time, fall time, pulse width, and amplitude of each pulse. c. Observe the boost regulator 50 vdc output noise d. Note that noise and transients are within acceptable limits e. Repeat the above Steps for each inverter	Oscilloscope, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
16	Perform Signal Conditioner Electrical Test The signal conditioner electrical test will be performed as follows: a. Turn on external power to spacecraft and check that voltage exists where it should and none exists on the remaining pins at the signal conditioner power input connector. b. Connect signal conditioner to secondary power subsystem. c. Measure voltage and current drawn by signal conditioner from the secondary power subsystem.	Voltmeter, ammeter, series fuse boxes	Procedure	None
17	Perform Digital Telemetry Unit Electrical Tests The digital telemetry unit electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins at the DTU power input connector. b. Connect the DTU to the 4.1 kc inverter and measure the voltage and current drawn by the DTU. Also note that noise and transients are at acceptable levels. c. Measure command line signal voltage and current drawn for each commanded bit rate, format, and mode of operation. Also note that noise and transients are acceptable levels. d. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all timing pulses at the users side of the harness. This is to be done for each bit rate. e. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all shift pulses at the users side of the harness. This is to be done for each bit rate.	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator, analog word simulator	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	f. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all synch pulses at the users side of the harness. g. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all inhibit pulses at the users side of the harness. This is to be done for each bit rate. h. Check ID words corresponding to all bit rates and all formats using the telemetry data display EOSE. i. Loop check all analog words by applying a DC voltage at the senders side of the harness and reading out the decimal word at the telemetry data display EOSE. j. Loop check all digital words by applying a digital signal at the senders side of the harness and reading out the decimal word at the telemetry display EOSE. Note: Noise, transient and cross talk measurements will be conducted for items c through g. k. Measure the subcarrier frequency and modulation index of the downlink baseband signal.			
18	Perform Digital Storage Unit Electrical Tests The digital storage unit electrical testing will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the DSU power connector. b. Connect the DSU to the spacecraft harness and measure the voltage and current drawn by the DSU. Also note that noise and transients are at acceptable levels. c. Measure all command line voltages and currents for each DSU command. Also note that noise and transients are at acceptable levels. d. Measure duration of the DSU input data signal at the DSU for each bit rate.	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, in-line test connectors, digital word data format generator		

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	e. Measure the rise time, fall time, amplitude, and pulse duration of the DSU data output signal at the DTU during memory reqdout. f. Measure the rise time, fall time, amplitude, and pulse duration of the DSU index pulse at the DTU. Note: Noise, transient, and cross talk measurements will be conducted for items d through f.	Analog word format generator		
19	<u>Perform DTU, DSU, CSØC, Power and Equipment Compartment Structure Temperature Calibrations</u> These calibrations will be handled as follows: a. DTU temperature calibrations will be accomplished by replacing the transducer with precision resistors and noting the word value at the telemetry data display EOSE for each resistor value. The word values together with the factory transducer curves complete the cablication. Then these parameters will be incorporated into the computer programs. The DTU analog to digital converter reference words are to be simply noted and recorded. b. DSU temperature calibrations will be accomplished as in Task 19.a. c. CSØC temperature calibrations will be accomplished as in Task 19.a. d. Primary power calibrations will be accomplished by varying the load current and line voltage and monitoring the voltage and current with meters. The telemetry word values for each voltage and current will be recorded. These parameters will be inserted into the computer programs. Secondary power calibrations will be accomplished in the same manner as the primary power calibrations. e. Equipment compartment structure temperature calibrations will be accomplished in Task 19.a. f. At the telemetry data display EOSE, verify that each command sent during items a through e above indicates the proper telemetry word value.	Voltmeter, ammeter, decade resistance box, data center computer programs, telemetry data display EOSE, power supply, power EOSE, series fuse boxes, in-line test connectors	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
20	<u>Perform Command Detector Electrical Tests</u> The command detector electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the command detector connectors. b. Connect the detectors to the spacecraft harness and measure the secondary power supply voltage and current drawn by the detectors. Also note that noise and transients are at acceptable levels. c. Measure the detector output peak to peak amplitude at the CSØC input in the presence of a simulated receiver signal (command encoder EOSE). d. Measure the bit synch rise time, fall time, pulse width, and amplitude. e. Check that each command processor can be addressed by one and only one separate address. f. Check each detector synch lock operation with the command encoder. g. Transmit each discrete command via the command encoder and observe that each command was received by observing the command matrix monitor. h. Repeat the above for the redundant detector. Note that quantitative commands from each detector will be monitored during stabilization and control subsystem checkout.	Power EOSE, voltmeter, ammeter, series fuse box, in-line test connector, command matrix, monitor command encoder	Procedure	None
21	<u>Perform Detector Calibrations</u> The detector calibrations will be accomplished as follows: a. DTU temperature calibrations will be accomplished by replacing the transducer with precision resistors and noting the word value at the telemetry data display	Power EOSE, command encoder, resistor decade box, operational data center command,	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	EOSE for each resistor value. The word values together with the factory transducer curves complete the calibration. Then these parameters will be incorporated into the computer programs. The DTU analog to digital converter reference words are to be simply noted and recorded.	matrix monitor, in-line test connector		
22	<u>Install High Gain Antenna and Cabling</u> This task is broken up into several subtasks as follows: a. Install high-gain antenna b. Connect, route, and clamp cabling c. Articulate antenna and check for cable chaffing and clearance d. Latch antenna in place	Hand tools, torque wrench, antenna drive EOSE	Procedure	None
23	<u>Install Medium Gain Antenna and Cabling</u> This task is broken up into several subtasks as follows: a. Install medium-gain antenna b. Connect, route, and clamp cabling c. Articulate antenna and check for cable chaffing and clearance d. Latch antenna in place	Hand tools, torque wrench, antenna drive EOSE		
24	<u>Install Omni Antenna and Cabling</u> This task is broken up into several subtasks as follows: a. Install omni antenna to omni antenna boom b. Install antenna and boom to spacecraft c. Connect, route, and clamp cabling d. Deploy and then latch boom observing cable clearance and that no chaffing takes place e. Latch antenna boom in place	Hand tools, torque wrench	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
25	<u>Perform RF Insertion Loss Test</u> The RF insertion loss determination will take place as follows: a. Connect the diplexers, couplers, bandpass filters, power monitors, and circulator switches to the RF cable harness system b. Measure the insertion loss between the receivers and the high-gain antenna c. Measure the insertion loss between the receivers and the low-gain antenna d. Measure the insertion loss between the receivers and the medium-gain antenna e. Measure the insertion loss between the power amplifiers and the high-gain antenna f. Measure the insertion loss between the power amplifiers and the low-gain antenna g. Measure the insertion loss between the power amplifiers and the medium-gain antenna h. Measure the insertion loss between the exciters and the high-gain antenna i. Measure the insertion loss between the exciters and the low-gain antenna j. Measure the insertion loss between the exciters and the medium-gain antenna k. Measure the insertion loss between the exciters and the power amplifiers	RF converter adapters, RF generator, RF power meter	Procedure	None
26A	<u>Perform VSWR Tests</u> The VSWR tests will be performed as follows: a. After the insertion loss test has been completed, connect the high-gain and omni antennas to the RF cable harness	RF connecter adapters, RF generator, RF couplers, VSWR meter, notch filters	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Measure the VSWR between the receivers and the high-gain antenna c. Measure the VSWR between the receivers and the medium-gain antenna d. Measure the VSWR between the receivers and the low-gain antenna e. Measure the VSWR between the power amplifiers and the high-gain antenna f. Measure the VSWR between the power amplifiers and the medium-gain antenna g. Measure the VSWR between the power amplifier and the low-gain antenna h. Measure the VSWR between the exciters and the high-gain antenna i. Measure the VSWR between the exciters and the medium-gain antenna j. Measure the VSWR between the exciters and the low-gain antenna			
26B	Receive Midcourse Propulsion and SCS Module The midcourse propulsion and SCS module will be received from Douglas consisting of the following: a. Monopropellant engine and control valves b. Monopropellant engine feed system c. Monopropellant engine pressurization system d. Stabilization and control subsystem gas system e. Jet vane assembly installed in engine Note: Final STL Quality Control buy-off will be performed at Douglas.			

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
27	<u>Perform Receiver Selector Electrical Tests</u> The receiver electrical tests will be performed as follows: a. Apply external power to the spacecraft and check that voltage exists where it should and that no voltage exists at the remaining pins of the receiver selector connectors. b. Connect the receiver selector to the spacecraft harness and measure the voltage and current drawn by the selector. Also note that noise and transients are acceptable levels. c. Connect the receiver signal simulator to the receiver selector. d. Simulate each receiver present signal and observe that the proper receiver is selected. e. Simulate all combinations of the three receiver present signals and observe that the proper receiver is selected. f. Simulate the loss of sun-Canopus and observe that receiver No. 1 is selected.	Power EOSE, voltmeter, ammeter, oscilloscope, receiver, selector, simulator	Procedure	None
28A	<u>Perform Receiver Electrical Tests</u> The receiver electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector. b. Connect each receiver to the spacecraft harness and measure the voltage and current drawn by each receiver. Note that noise and transients are within acceptable levels. c. Measure the modulation index of test transmitter output while it is being modulated with the command	RF EOSE, command encoder, command matrix, monitor, voltmeter, ammeter, power EOSE series fuse boxes	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	encoder and determine that it is within specification. This is to be done with and without the ranging signal. d. Connect the receiver to a strong hardline signal from the RF EOSE (-110 dbm) and acquire. e. Modulate the test transmitter (RF EOSE) with the command encoder and note that commands can be received and properly acted upon by the CS/C using each receiver through each antenna. This is to be accomplished by observing the command matrix monitor and by monitoring the appropriate telemetry words. Verify that the airborne receiver will acquire while the ground transmitter is being ramped at the maximum specified rate for given signal strengths. f. Determine the signal strength at which the receiver thresholds or drops out of lock. g. Verify that the receiver will stay acquired for the maximum specified ramp rate for given signal strengths. h. Repeat above for the redundant receiver			
28B	Perform Midcourse Propulsion and SCS Module Control Inspection Quality control inspection is mainly for shipping damage as the module has previously been bought off at Douglas by TRW personnel.	None	Procedure	None
29A	Perform Receiver Calibrations The receiver calibrations will be performed as follows: a. Receiver temperature calibrations will be accomplished as in Task 19.a.	RF EOSE, command encoder, power EOSE, RF attenuators, calorimeter, data center, in-line test connector		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. The airborne receivers will be dropped in and out of lock by removing the test transmitter signal and noting that the telemetry indication is proper. c. A precisely known signal level is fed into a precision step attenuator. A known signal strength can now be calculated for each attenuator setting. Each power level will be correlated with telemetry output. d. After the receivers have been acquired by the test transmitter, the test transmitter frequency is varied and the loop stress telemetry output noted. All of the above parameters will be inserted into the computer program.			
29B	<u>Perform Midcourse Propulsion and SCS Module Magnetics Property Test</u> The midcourse propulsion and SCS module magnetic properties check will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the magnetic field of the bus structure mounted in handling fixture c. Analyze all variations between readings and repeat if necessary	Magnetic measuring equipment, handling fixture, protective covers, handling slings	Procedure	Area in Building free of large magnetic fields
30	<u>Perform Exciter Electrical Tests</u> The exciter electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector. b. Connect the exiciter to the spacecraft harness and measure the voltage and current drawn by the driver. Note that noise and transients are within acceptable limits.	RF EOSE, command encoder, power EOSE, voltmeter, ammeter, series fuse boxes, in-line test connector, spectrum analyzer		

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Measure the rise time, fall time, and amplitude of the exciter modulation for each bit rate. d. Remove modulation and measure the exciter RF power and frequency at the exciter output. e. Measure the exciter modulation index with and without the ranging signal present. f. Investigate driver output for spurious harmonics using a spectrum analyzer. g. Connect the exciter output of the RF harness and ascertain that data can be received by the ground receiver (RF EOSE) through each antenna via air link. h. Command the exciter to the coherent mode of operation and observe that driver output is 240/221 times the frequency of the ground transmitter. i. Repeat above for the redundant exciter			
31A	<u>Perform Exciter Calibrations</u> The exciter calibrations will be performed as follows: a. Exciter temperature to be performed as in Task No. 19.a. b. Coherent/noncoherent mode to be performed by commanding the driver to the coherent and noncoherent modes of operation and noting that proper telemetry word value exists.	Power EOSE, RF EOSE, decade resistance box, series fuse boxes	Procedure	None
31B	<u>Install Stabilization and Control Subsystem Compartment</u>	Hand tools, torque wrenches	Procedure	None
31C	<u>Install Midcourse Propulsion and SCS Module</u>	Hand tools, torque wrenches		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
32	<u>Perform Transmitter Selector Test</u> The transmitter selector electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins on each selector connector. b. Connect the transmitter selector to the spacecraft harness and measure the voltage and current drawn from the secondary power supply subsystem. Note that noise and transients are within acceptable levels. c. Simulate the appropriate transmitter modes via ground commands and CS∅C backup commands and ascertain that the proper transmitter was selected by monitoring the selector outputs.	Power EOSE, voltmeter, ammeter, osciloscope	Procedure	None
33A	<u>Perform Power Amplifier Tests</u> The power amplifier tests will be performed as follows: a. Turn on external power to the spacecraft and command the power amplifier on. b. Connect dummy loads to the power amplifier output. c. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector. d. Connect the power amplifier power to the spacecraft harness and measure the voltages and currents drawn by power amplifiers. Note that noise and transients are within acceptable levels. e. Measure the power amplifier RF output power f. Measure the power amplifier modulation index with and without the ranging signal. g. Measure the power amplifier output for spurious harmonics using a spectrum analyzer.	Power meter, NF-112 analyzer, power EOSE, RF EOSE, series fuse box, in-line test connectors	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	h. Connect the power amplifier to the RF cable harness i. Observe that telemetry can be received by the ground receiver (RF EOSE) from each antenna via air link. j. Repeat for the redundant power amplifier			
33B	Install Solar Array Support Structure	Hand tools, torque wrenches		
34A	Perform Power Amplifier Calibrations The power amplifier calibrations will be performed as follows: a. Temperature calibration will be performed as in Task 19.a. b. To accomplish the power monitor calibrations, step attenuators will be placed in the RF lines and the antenna power measured. The measured power for each attenuator step is correlated with the telemetry output words. These parameters will be inserted into the computer programs.	Step attenuator, decade resistor box, power EOSE, RF EOSE power meter	Procedure	None
34B	Connect the SCS Nozzles and Gas Lines to the Spacecraft The SCS nozzles and gas lines will be connected to the spacecraft SCS pneumatics system.	Hand tools	Procedure	None
35A	Perform Low Pressure SCS Leak Test The purpose of the low pressure leak test is to ascertain that the SCS pneumatic system leak rate is grossly within specification.	Leak test console		
35B	Install Sun Sensors, Canopus Sensors and Shunt Regulators	Hand tools	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
35C	Remove Experiment Equipment Compartment The experiment equipment compartment will be removed from the spacecraft and individual black boxes installed in preparation for off line experiment testing.	Hand tools	None	None
36	Perform Low Power Transmitter Electrical Tests The lower power transmitter electrical tests will be performed as follows: a. Turn on external power to the spacecraft and command the low power transmitter on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins. c. Connect the lower power transmitter to the spacecraft harness and measure the voltage and current drawn from the secondary power subsystem. Note that noise and transients are within acceptable limits. d. Measure the low power transmitter power output and frequency. e. Measure the low power transmitter output for spurious harmonics using a spectrum analyzer. f. Measure the low power transmitter output modulation index. g. Connect the low power transmitter to the RF cable harness. h. Observe that telemetry can be received by the ground receiver through each antenna via air link.	Voltmeter, ammeter, RF power meter, NF-112 analyzer oscilloscope, series fuse box, spectrum analyzer, RF frequency counter	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
37	Perform Low Power Transmitter Calibration The low power transmitter calibration will be performed as follows: a. Temperature calibration will be performed as in Task 19.a. b. To accomplish the power monitor calibrations, step attenuators will be placed in the RF lines and the antenna power measured. The measured power for each attenuator step is correlated with the telemetry output words. The parameters will be inserted into the computer programs.	Step attenuator, decade resistor box, power EOSE, RF EOSE, power meter	Procedure	None
38A	Perform Gyro Package Alignment The gyro package alignments are performed so that the gyro scale factors can be determined as part of the SCS testing phase.	Gyro alignment set	Procedure	None
38B	Start Off-line Experiment Subsystem Tests	Experiment panel EOSE	Procedure	None
39A	Perform Sun Acquisition Electrical Tests The sun acquisition electrical tests will be performed as follows: a. Apply external power to the spacecraft and command the gyros on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each connector of the gyro package. c. Connect the gyro package to the spacecraft harness and measure the voltage and current drawn by the gyro	SCS EOSE, power EOSE, voltmeter, ammeter, oscilloscope, jet vane angle MOSE, in-line test connector, series fuse box	Procedure	Tilt fixture should experience zero floor vibrations

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	spin motors. (Also measure turn on transient amplitude). Note that noise and transients on these lines are within acceptable levels.			
	d. 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.			
	e. Connect the control signal electronics package to the spacecraft harness and measure the voltage and currents drawn by the package. Note that noise and transients on these lines are within acceptable levels.			
	f. Torque the tilt fixture in the +yaw direction at a known rate and measure the yaw gyro output signal amplitude. Note that the polarity is correct.			
	g. Torque the tilt fixture in the -yaw direction at a known rate and measure the yaw gyro output signal. Note that the polarity is correct.			
	h. Repeat Step f for the pitch and roll gyros.			
	i. With the spacecraft absolutely still, measure the noise amplitude on each gyro output line.			
	j. Increase the rate in each axis in each direction and note that the proper gas valve is actuated.			
	k. Determine the threshold rates in each axis which will just barely cause the gas valves to actuate.			
	l. Measure the voltage and current drawn from the secondary power supply subsystems by the control signal during zero gyro rate input conditions and maximum rate inputs. Note that noise and transients			

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	are within acceptable limits. m. Connect the sun sensors to the spacecraft harness. n. Attach the sun sensor stimulus to each sun sensor. o. Connect a voltmeter in place of each gas valve solenoid. p. Manually actuate separation switches and check that the sun acquisition mode has started. q. Transmit the back-up command for starting the sun acquisition sequence. r. Illuminate each sun sensor and check that voltage exists at each valve interface. s. Connect each valve to the spacecraft harness. t. Stimulate each sun sensor and measure power supply subsystem by the control signal electronics package during each valve actuation. u. Observe that when each sun sensor is stimulated the proper valve is opened. v. Observe that when all of the five sun sensor elements are illuminated, no valves are actuated.			
39B	<u>Perform Sun Acquisition Calibrations</u> The sun acquisition calibration will be performed as follows: a. The sun intensity signals will be simulated by replacing the sun sensor with a signal generator. As the voltage is varied the telemetry word value	Resistance decade box, power EOSE SCS EOSE, series fuse boxes, signal generator,	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	is recorded. This is to be done for each sun sensor. The laboratory curves for each sun sensor (intensity versus voltage out) together with the digital word values inserted into the computer program. b. The valve actuation signals will be calibrated by merely actuating each valve and noting the telemetry word values. c. Control signal electronics package temperature calibration will be performed as per task 19.a. d. Sun sensor temperature calibration will be performed as per task 19.a. e. The gyro temperature will be calibrated as per 19.a. f. Gyro on/off calibrations will be performed as follows: The gyros will be simply commanded on and then off and the telemetry word value simply recorded. g. Gyro pick-off outputs will be replaced with a signal generator. As the signal generator amplitude is varied the telemetry word value is monitored. These parameters together with the laboratory bench test data (rate versus output voltage) will be inserted into the computer programs.	in-line test connector		
40	Perform Earth Sensor Electrical Tests The earth sensor electrical tests will be performed as follows: a. Turn on external power to the spacecraft and command the earth sensor to on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the earth sensor connector. c. 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.	SCS EOSE, power EOSE, earth sensor stimulus, voltmeter, ammeter, oscilloscope, series fuse box		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. Darken the earth sensor appeture and measure the signal output amplitude. Note that noise and transients are within acceptable levels. e. Attach the earth sensor stimulus to the earth sensor f. Illuminate the earth sensor and measure the output signal amplitude. Note that noise and transients are within acceptable limits. g. 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			
41	Perform Earth Sensor Calibrations The earth sensor calibrations will be performed as follows: a. The earth sensor will be replaced by a suitable signal generator. As the signal generator 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 into the computer program. b. The earth sensor temperature calibration will be performed as in step 19.a.	Signal generator, in-line connector, voltmeter, power EOSE data center		
42A	Perform Canopus Acquisition Tests The Canopus acquisition electrical tests will be performed as follows: a. Turn off external power to the spacecraft and command the Canopus sensor on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the Canopus sensor connector.	Voltmeter, ammeter, oscilloscope, power EOSE, SCS EOSE	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. 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. d. Attach Canopus sensor stimulus to the Canopus sensor. e. Illuminate each half of the Canopus sensor field of view and note that the proper valves actuate when each half is illuminated. f. 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. g. Illuminate the center of the Canopus sensor field of view and note that no valves are actuated. h. Investigate the Canopus sensor signal output lines for out-of-tolerance transient and noise conditions when the center of the Canopus sensor is illuminated. i. Command the spacecraft into the roll search mode and observe that the proper roll valves are actuated. j. Remove Canopus sensor illumination and observe that the SCS subsystem goes into the roll search mode. k. Also note that the airbourne receivers switch to the omni antenna when the Canopus illumination is removed.			
42B	<u>Perform Planet Oriented Package Magnetic Properties Test (Off Line)</u> The magnetics test will be conducted as follows: a. Measure the magnetic field of the handling fixture b. Measure the handling fixture magnetic field stability	Protective covers, handling slings, magnetic field measuring equipment	Procedure	Area in building free of large magnetic fields (less than 50 gamma ambient field)

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Measure the magnetic field of the planet-oriented package while mounted in the handling fixture.			
43A	Perform Canopus Acquisition Calibrations The Canopus acquisition calibrations will be performed as follows: a. The Canopus sensor will be replaced by suitable signal generator. As the generator 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. b. The Canopus sensor intensity signal will be performed as in Task a above. c. The Canopus sensor temperature calibrations will be performed as in Task 19.a.	Signal generator, resistor decode box, power EOSE, data center	Procedure	None
43B	Start Planet-Oriented Package Off-Line Electrical Tests The planet-oriented package system will be completely checked out off-line including both the experiments and the SCS articulation system.			
44	Perform Spacecraft Midcourse Maneuver Tests The spacecraft maneuver testing will be accomplished as follows: a. Enter the roll turn and polarity information into the command detector b. Execute the roll turn command and measure and time the gyro output and input signals. Also,	SCS EOSE, power EOSE, voltmeter, ammeter, oscilloscope, jet vane angle MOSE, in-line test connector,	Procedure	Tilt fixture should experience zero floor vibrations

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	note that noise and transients are within acceptance levels. c. Note that the proper gas valves are actuated while the gyro is being torqued. d. Repeat Steps b and c for the opposite polarity turn. e. Repeat Steps a through d for the pitch turn. f. Load velocity increment information into the detector to activate jet vane control noting that the proper voltage amplitudes exist at each jet vane actuator connector. g. Repeat Step f for the opposite polarity. h. Connect the midcourse motor jet vanes to the spacecraft harness re-insert the velocity increment and measure the voltage and current drawn by each jet vane actuator. Note that noise and transients on these lines are within acceptable limits. i. Measure the jet vane angle with respect to the sun-line. j. Repeat Step i for the opposite polarity velocity increment. k. Enter midcourse motor burn duration information into the command detectors and measure the midcourse motor ignitor firing voltage and turn off voltage and the time duration between the turn on signal and the turn off signal.	series fuse box		
45	<u>Perform Spacecraft Midcourse Calibrations</u> The spacecraft maneuver calibrations will be performed as follows: a. The jet vane actuator temperature calibrations are to be performed as per Task 19.a. b. Jet vane angle calibrations are to be performed by turning the jet vanes to known angles and recording the telemetry word values. These parameters are then inserted into the computer program.	Power EOSE, SCS EOSE, angle gauges, decade resistor box, in-line test connector	Procedures	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
46	<u>Install Planet-Oriented Package</u> The planet-oriented package consists of the following items: a. Planet-oriented package boom. b. Planet-oriented package gimble actuators c. Mars horizon scanners d. Television experiment sensors e. Ultraviolet spectrometer sensors f. Scan radiometer experiment g. Infrared spectrometer sensors h. Meteoroid flash experiment sensors i. Planet-oriented package intercabling j. Planet-oriented package insulation Note: Assume that all of the above items have been tested and assembled into one package, i.e. planet oriented package prior to this time.	Hand tools, torque wrenches	Procedure	None
47	<u>Install Experiment Equipment Compartment</u>	Hand tools	Procedure	None
48	<u>Perform Planet-Oriented Package Stabilization and Control Testing</u> The planet oriented and stabilization and control testing will take place as follows: a. Connect the planet-oriented package boom connector to the main spacecraft harness. b. Apply external power to spacecraft and command the planet oriented package control system to on. c. Check that voltage exists where it should and that no voltage exists on the remaining pins of each	Voltmeter, ammeter, oscilloscope power EOSE SCS EOSE, horizon scanner stimulus		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	connector of the drive electronics package. d. Connect the drive electronics package to the spacecraft harness and measure the voltage and current drawn by the drive electronics package. e. Check that voltage exists where it should be and that no voltage exists on the remaining pins of each connector of the Mars horizon scanner package. f. Connect the Mars horizon scanner package to the boom harness. g. With the horizon scanner package sensros completely darkened, observe that the horizon scanner and drive electronics package output signal noise and transient levels are within acceptable levels. h. Attach the horizon scanner stimulus EOSE to the horizon scanner package. i. Stimulate each horizon scanner and note that the proper gimble slews in the proper direction at the proper rate. j. Repeat Step i for the opposite polarity for each scanner. k. Slew each gimble by means of the horizon scanner stimulus and measure the drive voltage amplitude at the gimble actuators. Also measure the voltage and current drawn by the drive electronics package from the secondary power subsystem noting that noise and transient conditions are within specification.			
49	<u>Perform Planet Oriented Package Stabilization and Control Calibrations</u> The planet-oriented package stabilization and control calibrations will take place as follows: a. Planet-oriented package temperature will be calibrated as per Task 19.a. b. Each gimble will be turned to known angles during which the telemetry word values are monitored and recorded. The gimble angle information and the telemetry word values are inserted into the computer programs.	Resistor decade box, gimble angle indicator, signal generator, power EOSE data center	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. The horizon scanner sensor output calibrations will be performed b replacing the horizon scanner with a signal generator and opening up each gimble control loop. As the signal generator output amplitude is varied the telemetry word values are monitored and recorded. These parameters along with the horizon scanner laboratory bench information (scanner error in degrees versus output voltage) are inserted into the computer program.			
50	<u>Perform High-Gain Antenna Gimble Actuator Tests</u> The gimble actuator tests will be performed as follows: a. Turn on external power to the spacecraft and command the antenna to slew. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the gimble actuator connectors. c. Measure the drive signal amplitude. d. Repeat steps a, b, and c for the remaining gimble axis. e. Connect the gimble actuators to the harness and command the gimble to slew. f. Measure the voltage amplitude and current drawn by the drive electronics from the secondary power subsystem while the gimble is being slewed noting that noise and transient conditions are within tolerance. g. Repeat Step f for each gimble in each direction. h. Observe that each gimble slews at the proper rate in each direction.	Voltmeter, ammeter, power EOSE, command EOSE	Procedure	None
51	<u>Perform High-Gain Antenna Gimble Actuator Calibrations</u> The actuator calibrations will be performed as follows:	Resistor decade box, gimble angle indicator,	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. The actuator temperature calibrations will be performed as per Task 19.a. b. Gimble angle calibrations will be performed by slewing each gimble to a known angle and observing and recording the telemetry word values. These parameters are then inserted into the computer program.	power EOSE, command EOSE, data center		
52	<u>Perform Medium-Gain Antenna Gimble Actuator Tests</u> The gimble actuator tests will be performed as follows: a. Turn on external power to the spacecraft and command the antenna to slew. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the gimble actuator connectors. c. Measure the drive signal amplitude. d. Connect the gimble actuator to the harness and command the gimble to slew. e. Measure the voltage amplitude and current drawn by the drive electronics from the secondary power subsystem while the gimble is being slewed noting that noise and transient conditions are within specification. f. Repeat Step e for each gimble in each direction. g. Observe that each gimble slews at the proper rate in each direction.	Voltmeter, ammeter, power EOSE command EOSE	Procedure	None
53	<u>Perform Medium-Gain Antenna Gimble Actuator Calibrations</u> The actuator calibrations will be performed as follows: a. The actuator temperature calibrations will be performed as per Task 19.a.	Resistor decade box, gimble angle indicator, power EOSE command EOSE, data center	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Gimble angle calibrations will be performed by slewing the gimble to a known angle and observing and recording the telemetry word values. These parameters are then inserted into the computer program.			
54	<u>Install and Align Retropropulsion Solid Engine</u>	Hand tools, torque wrenches solid motor alignment set	Procedure	None
55	<u>Perform Terminal Maneuver Testing</u> The terminal maneuver testing will be performed as follows: a. Perform terminal turn maneuvers via the CC&S. These maneuvers are accomplished in the same manner as in the midcourse turn maneuvers. b. Perform capsule separation via the command decoder. This test will be performed as follows: 1. The safe-arm capsule separation system is checked out. When the ordance lines are safed, no voltage exists across the firing lines and zero ohms continuity will exist between each line to the spacecraft frame ground. When the separation lines are armed, approximately 28 volt will exist across the lines. 2. The ordance EOSE is connected to the capsule separation lines. 3. The capsule separation command is transmitted to the spacecraft via the ground transmitter. 4. Proper indications should be observed on the ordnance EOSE (all fire) and also via telemetry.	Voltmeter, ammeter, oscilloscope ordnance EOSE, SCS EOSE, power EOSE data center Solid motor pressurizing test set, voltmeter, ammeter, oscilloscope	Procedure	Darkened room

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. Perform spacecraft deboost tests. 1. Connect cold gas supply to the solid motor TVC test connector 2. Perform the spacecraft terminal maneuver and command the thrust vector control system to on. 3. Torque the tilt fixture in the pitch axis and monitor the pitch injector signal amplitude at the solid retromotor. 4. Repeat Step 3 for the opposite direction. 5. Repeat Step 3 for the yaw axis in both directions. 6. Connect the solid motor thrust vector control systemto the spacecraft harness. 7. Torque the tilt table in the pitch and yaw axis in both polarities. 8. While the spacecraft is being torqued, observe that gas is flowing through the proper injector. 9. While the spacecraft is being torqued, measure the voltage and current drawn from the secondary power supply subsystem. Also note that noise and transients are within acceptable limits.			
56	<u>Perform Terminal Maneuver Calibrations</u> The terminal maneuver calibrations will be performed as follows: a. Solid motor thrust vector control temperature calibration is to be performed as per Step 19.a.	Ordnance EOSE		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Thrust vector control injector actuation will be energized and the telemetry word monitored correct value.			
57	<u>Perform Data Automation Equipment Electrical Test</u> The data automation electrical test will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins at the DAE power input connector. b. Connect the DAE to the 4.1 kc inverter and measure the voltage and current drawn by the DAE. Also note that noise and transients are at acceptable levels. c. Measure command line voltage and current drawn for each bit rate, format and mode of operation. Also note that noise and transients are at acceptable levels. d. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all timing pulses at the experimenters side of the harness. This is to be done for each bit rate. e. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all shift pulses at the experimenters side of the harness. This is to be done for each bit rate. f. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all synch pulses at the experimenters side of the harness. This is to be done for each bit rate. g. Measure the frequency, pulse amplitude, rise time, fall time, and the pulse width of all inhibit pulses at the experimenters side of the harness. This is to be done for each bit rate. h. Check ID words corresponding to all bit rates and all formats using the telemetry data display EOSE.	Fully operational data center, operational computer, programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, In-line test connectors, digital word data format generator, analog word simulator		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	i. Loop check all analog words by applying a DC voltage at the experimenters side of the harness and reading out the decimal word at the telemetry data display EOSE. j. Loop check all digital words by applying a digital signal at the senders side of the harness and reading out the decimal word at the telemetry data display EOSE. Note Noise, transient and cross talk measurements will be conducted for items c through g.			
58	<u>Perform Bulk Storage Unit Electrical Tests</u> The bulk storage unit electrical testing will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the bulk storage power connector. b. Connect the bulk storage to the spacecraft harness and measure the votage and current drawn by the bulk storage. Also note that noise and transients are at acceptable levels. c. Measure all command line voltages and currents for each bulk storage command. Also note that noise and transients are at acceptable levels. d. Measure the use time, fall time, amplitude and pulse duration of the bulk storage input data signal at the bulk storage for each bit rate. e. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage data output signal at the DAE during memory readout. f. Measure the rise time, fall time, amplitude, and pulse duration of the bulk storage index pulse at the DAE. Note: Noise, transient and cross talk measurements, will be conducted for items d through f.	Fully operational data center, operational computer programs, telemetry data display EOSE, ammeter, voltmeter, oscilloscope, series fuse boxes, In-line test connectors, digital word data format generator	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
59	<u>Perform Data Automation and Bulk Storage Calibrations</u> These temperature calibrations will be performed as follows: a. DAE temperature calibration is to be performed as per Task 19.a. b. Bulk storage temperature calibration is to be performed as per Step 19.a.	Power EOSE data center, resistor decade box, command EOSE, in-line test connector	Procedure	None
60	<u>Perform Mapping Package Experiment Electrical Tests</u> The mapping package experiment tests will be performed as follows: a. Turn on external power to the spacecraft and command each experiment to on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each experiment electronic package connector. c. Connect each experiment electronic package to the spacecraft harness and measure the voltage and current drawn by the electronic packages from the secondary power subsystem. d. At each mapping package experiment sensor, observe that voltage exists where it should and that no voltage exists on the remaining pins of each sensor connector. e. Connect each sensor to the spacecraft harness. f. Measure the voltage and current drawn from the secondary power supply subsystem by each experiment. g. Measure the noise content on all mapping package experiment power and signal lines observing that the noise content is within specified levels. h. Measure the rise time, fall time, pulse duration, and amplitude of the turn-on transient of each experiment.	Voltmeter, ammeter, power EOSE command EOSE, experiment EOSE, series fuse boxes, in-line test connectors	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	i. Stimulate each experiment and determine that each experiment is working properly by using both EOSE and telemetry information.			
61	<u>Install the Magnetometer Assembly</u> The magnetometer assembly consists of magnetometer sensors and magnetometer sensor boom.	Hand tools, torque wrenches	Procedure	None
62	<u>Perform Main Body Experiment Electrical Tests</u> The main body experiment electronics and sensors consist of the following items: a. Meteoroid impact experiment. b. Plasma experiment. c. Cosmic ray experiment. d. Trapped radiation experiment. e. Ionosphere experiment. f. Magnetometer electronics The main body electrical testing will be performed as follows: a. Turn on external power to the spacecraft and command each experiment to on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of each experiment electronics connector. c. Connect each experiment electronics package to the spacecraft harness and measure the voltage and current drawn by each electronic package from the secondary power supply. d. At each main body sensor package observe that voltage exists where it should and that no voltage exists on the remaining pins of each sensor connector. e. Connect each sensor to the spacecraft harness.	Voltmeter ammeter, oscilloscope power EOSE, command EOSE, experiment EOSE, series fuse boxes	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	f. Measure the voltage and current drawn by each main body experiment from the secondary power supply subsystem. g. Measure the noise content on all main body experiment power and signal lines observing that the noise content is within specified levels. h. Measure the rise time, fall time, pulse duration, and amplitude of the turn-on transient of each main body experiment. i. Stimulate each experiment and determine that each experiment is working properly by using both EOSE and telemetry information.			
63	<u>Perform Experiment Compatibility Testing</u> The experiment capability tests will be performed as follows: a. Ascertain that each experiment test source does or does not interfere with another experiment. b. Exercise each spacecraft subsystem and ascertain that each subsystem does not interfere or degrade any experiment data. c. Exercise each experiment and ascertain that each experiment does not degrade the spacecraft operation. In particular ascertain that the radio propagation experiments do not degrade the RF subsystem.	Complete set of system test EOSE, spectrum analyzer	Procedure	None
64	<u>Perform Experiment Calibrations</u> The magnetometer calibration will take place at the magnetometer site.	Complete set of systems test EOSE, radiation sources	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
65	<u>Perform VHF Capsule Receiver Electrical Tests</u> The receiver electrical tests will be performed as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of each connector. b. Connect each receiver to the spacecraft harness and measure the voltage and current drawn by each receiver noting that noise and transient levels are within specification. c. Connect the receiver to a strong signal from the capsule EOSE (-110 dbm) and acquire. d. Determine the signal strength at which the receiver thresholds or drops out of lock. e. Modulate the capsule simulator and measure the receiver output signal amplitude. f. Repeat the above steps for the redundant receiver.	RF EOSE, command matrix monitor, voltmeter, ammeter, power EOSE, series fuse boxes, in-line test connectors, capsule simulator	Procedure	None
66	<u>Perform VHF Capsule Receiver Calibrations</u> The receiver calibrations will be performed as follows: a. Receiver calibrations will be accomplished as in Task 19.a. b. The airborne receivers will be dropped in and out of lock by removing the capsule simulator signal and noting that the telemetry indication is proper. c. A precisely known signal level is fed into a precision step attenuator. A known signal strength can now be calculated for each attenuator setting. Each power level is correlated with telemetry output. These parameters are then entered into the computer program.	RF EOSE, power EOSE, RF attenuators, calorimeter, data center, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
67	<u>Perform Capsule Detector Test</u> The capsule detector will be tested as follows: a. Turn on external power to the spacecraft and check that voltage exists where it should and that no voltage exists on the remaining pins of the detector power connector. b. Connect the detector to the spacecraft harness and measure the voltage and current drawn by the detector. Also note that noise and transients are at acceptable levels. c. Acquire the capsule simulator and measure the amplitude, rise time, and fall time of the detector output signal.			
68	<u>Perform Demated Capsule Compatibility Tests</u> The demated capsule compatibility tests are mainly RF tests and consist of the following: a. Activate the capsule RF system and ascertain that the capsule RF system does not interfere with or degrade the spacecraft up or down link communications system. b. Rotate the capsule through 360 degrees and ascertain that the capsule RF system does not interfere with or degrade with the spacecraft up or down link communication system. c. Exercise the spacecraft through all of the possible up and down link communication configurations and ascertain that the spacecraft does not interfere with the capsule communications subsystem. d. Rotate the capsule through 360 degrees and ascertain that the spacecraft and all of the possible up and down link RF configurations do not interfere with the capsule communications subsystem.	Complete set of system EOSE, spectrum analyzer	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
69A	Install Thermal Louvers Thermal louvers are to be installed on each spacecraft side panel and torqued to flight specifications.	Hand tools, torque wrenches		
69B	Install Solar Array Panels The solar array panels are installed at this time to support the weight determination task.			
70	Perform Spacecraft Bus Weight Determination Test	Slings, hydra set, weighing fixture	Procedure	Overhead crane with hook height of ____
71	Perform Bus Center of Gravity Determination Test The spacecraft center of gravity will be determined by analytical evaluation of the data obtained from the weighing operations.	Slings, hydra set, c.g. fixture	Procedure	Overhead crane with hook height of ____
72	Perform Moment of Inertia Determination Testing While nonoperative the spacecraft moments of inertia about the sun line and the maximum and minimum moments about transverse axis will be determined and compared with design requirements.	Slings, moment of inertia fixture, timer	Procedure	Overhead crane with hook height of ____
73	Install Test Capsule The test capsule will be installed and lightly torqued down.	Slings, hydra set, torque wrenches	Procedure	Overhead crane with hook height of ____

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
74	**Perform Capsule Electrical Tests** The capsule electrical tests will be performed as follows: a. Apply external power to the spacecraft and command spacecraft capsule power to on. b. Observe that voltage exists where it should and that no voltage exists on the remaining pins of the capsule interface connectors. c. Connect the capsule to the spacecraft electrical harness and measure the voltage and current drawn by the capsule from the spacecraft power system during all capsule electrical modes noting that noise and transient levels are within specification. d. Observe that the capsule RF subsystem does not interfere with or degrade the spacecraft up and down link spacecraft RF systems. e. Rotate the capsule through 360 degrees and observe that the capsule RF subsystem does not interfere with or degrade the spacecraft up and down link RF systems. f. Observe that the spacecraft up and down link RF subsystems do not interfere with or degrade the capsule electrical and RF operations. g. Rotate the capsule through 360 degrees and observe that the spacecraft up and down link RF subsystems do not interfere with the capsule electrical and RF operations.	Ammeter, voltmeter, series fuse box, complete set of systems test EOSE		
75	**Perform Power Profile Test** The power profile tests will be performed as follows: a. The flight sequence of events up until sun acquisition will be followed and primary power drains monitored.	Recorders, current probe, complete set of systems test EOSE, in-line test connector	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	b. Compare the primary power drains up until 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. d. Compare the primary power drains with the trajectory information and ascertain that sufficient battery capacity remains to perform the midcourse maneuvers. e. Command the spacecraft to perform all of the cruise mode and Mars encounter functions monitoring all primary power drains. f. Compare the primary power drains, the trajectory information and ascertain that sufficient battery capacity exists to perform the deboost and sun re-acquisition 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 trajectory battery capacity remains to carry the spacecraft through the sun eclipses that are encountered during the Mars orbiting modes of operations.			
76	<u>Perform Solar Array Testing and Calibrations</u> The solar array testing and calibrations will be accomplished as follows: a. Illuminate each solar array string and measure the short circuit current and open circuit voltage. b. Perform inverse impedence tests on each solar array string.	Resistor decade box, solar array test set, voltmeter, ammeter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. The solar array temperature calibrations will be performed as per Task 19.a.			
77	Perform Pyrotechnic Subsystem Integration Tests The pyrotechnic subsystem integration encompasses the following areas: a. Experiment ordnance b. Experiment boom ordnance c. High-gain antenna boom ordnance d. Medium-gain antenna boom ordnance e. Low-gain antenna boom ordnance f. Planet-oriented package boom ordnance g. Midcourse correction motor ordnance h. Solid retropropulsion engine ordnance i. Capsule separation ordnance The pyrotechnic subsystem ordnance tests will be performed as follows: a. Ascertain that the pyrotechnic subsystem is in a safe condition by monitoring across each squib bridge wire connector a dead short. b. Command each squib to the fire condition and monitor the squib "firing" voltage at each squib bridge wire connector. c. Connect the pyrotechnic EOSE to each squib connector and command each squib to the "fire" condition. d. Ascertain that an "all fire" indication exists for each squib actuation. e. Command each squib to the "fire" condition using under voltage conditions and ascertain that a "no fire" condition exists for each squib actuation.	Ordnance EOSE, system test set EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
78	<u>Perform Pyrotechnic Subsystem Calibrations</u> The pyrotechnic subsystem will be calibrated by firing each simulated ordnance device and recording the corresponding telemetry word value.	Systems test set EOSE	Procedure	None
79	<u>Install Thermal Insulation</u> The thermal insulation will be installed in preparation for the electromagnetic compatibility testing, magnetic property testing, and thermal vacuum testing.			
80	<u>Perform Electromagnetic Compatibility Tests</u> The electromagnetic compatibility tests will be performed as follows: a. Command the spacecraft subsystems through every combination and permutation of the flight sequences and ascertain that there is no degradation of interference between subsystems. b. Irradiate the spacecraft with RF signals that correspond to the expected frequencies and levels from the Saturn and Centaur over-all launch vehicle system. c. Command the spacecraft subsystems through every combination and permutation of the Voyager flight sequences and determine the frequencies and levels of all radiation that is emitted from the spacecraft. d. Apply audio tones and tone bursts to the spacecraft primary bus system and observe each subsystem reaction noting that it is within specification.	Complete set of systems test EOSE, electromagnetic test set	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
81	<u>Perform Integrated System Test</u> The integrated systems test rigidly follows the flight sequence of events. Each Voyager space subsystem is tested to the maximum level and proper operation is verified by using the systems test EOSE and the data center to carefully reduce all telemetry data.	Complete set of systems test EOSE	Procedure	None
82	<u>Perform Integrated Systems Test Critique</u> The integrated system test critique is a meeting of all cognizant personnel to discuss the results of the integrated systems test. It is during this meeting that each sub-system engineer signs off the IST data.	None	Records to be signed off	None
83	<u>Perform Shipping Preps</u> The spacecraft booms and other appendages are folded and latched and the spacecraft is placed in the shipping container. Next desicate is placed inside of the container and the container sealed. The shipping container and spacecraft are purged with dry nitrogen. Note that it will be necessary to remove the array panels and support structure for shipment.	Slings, shipping containers, purging equipment	Procedure	Crane with hook height of ____
84	<u>Ship Voyager Planetary Vehicle to Magnetic Properties Site</u> The spacecraft and shipping container will be shipped to the test site. During shipment the shipping container is purged with dry nitrogen.	Helicopter sling	Procedure	

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
85	Receive Voyager Planetary Vehicle and Remove Shipping Container The spacecraft is next placed on magnetic properties test fixture and torqued down.	Magnetic properties fixture	Procedure	Crane with hook height of ____
86	Mate the Solar Array Support Structure and Array to the Spacecraft	Hand tools, torque wrenches	Procedure	None
87	Electrically Revalidate All Solar Array Support Structure Mounted SCS and Experiment Sensors			
88	Map Voyager Planetary Vehicle Perm Field The magnetic field of the spacecraft is measured with no power applied.	Magnetic properties measuring equipment	Procedure	Low magnetic ambient field
89	Perform Voyager Planetary Vehicle Magnetic Stability Tests The spacecraft will be permed and depermed and the change in the spacecraft magnetic field measured.	Magnetic properties measuring equipment, magnetizing coils	Procedure	None
90	Measure the Spacecraft Induced Magnetic Fields Each spacecraft subsystem will be commanded to perform every combination and permutation of the possible operating modes. While this is taking place, the spacecraft magnetic fields are measured.	Magnetic properties measuring equipment, complete set of system test EOSE, long EOSE cables, coil system to buck out earths magnetic field	Procedure	Low magnetic ambient field

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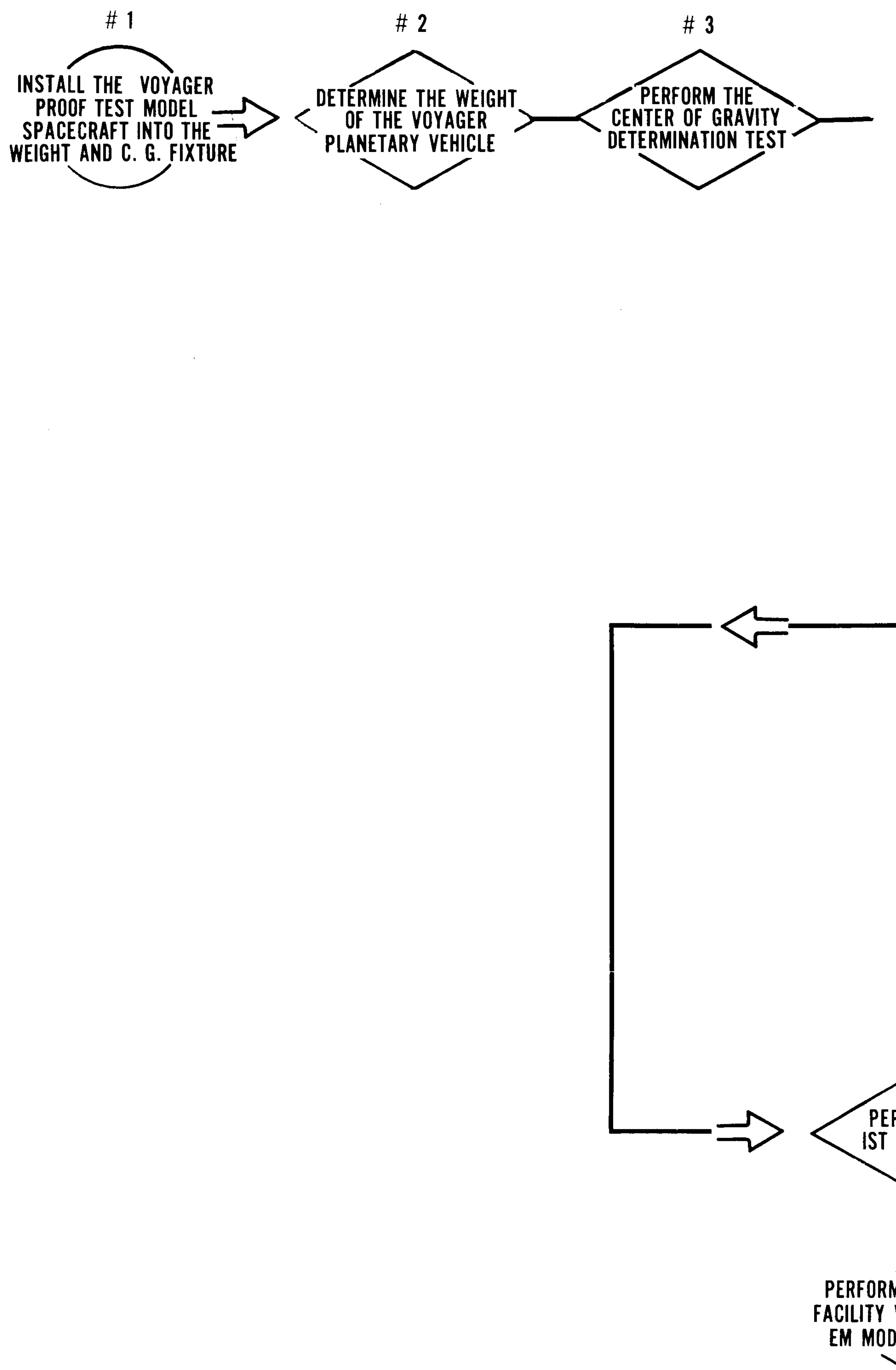
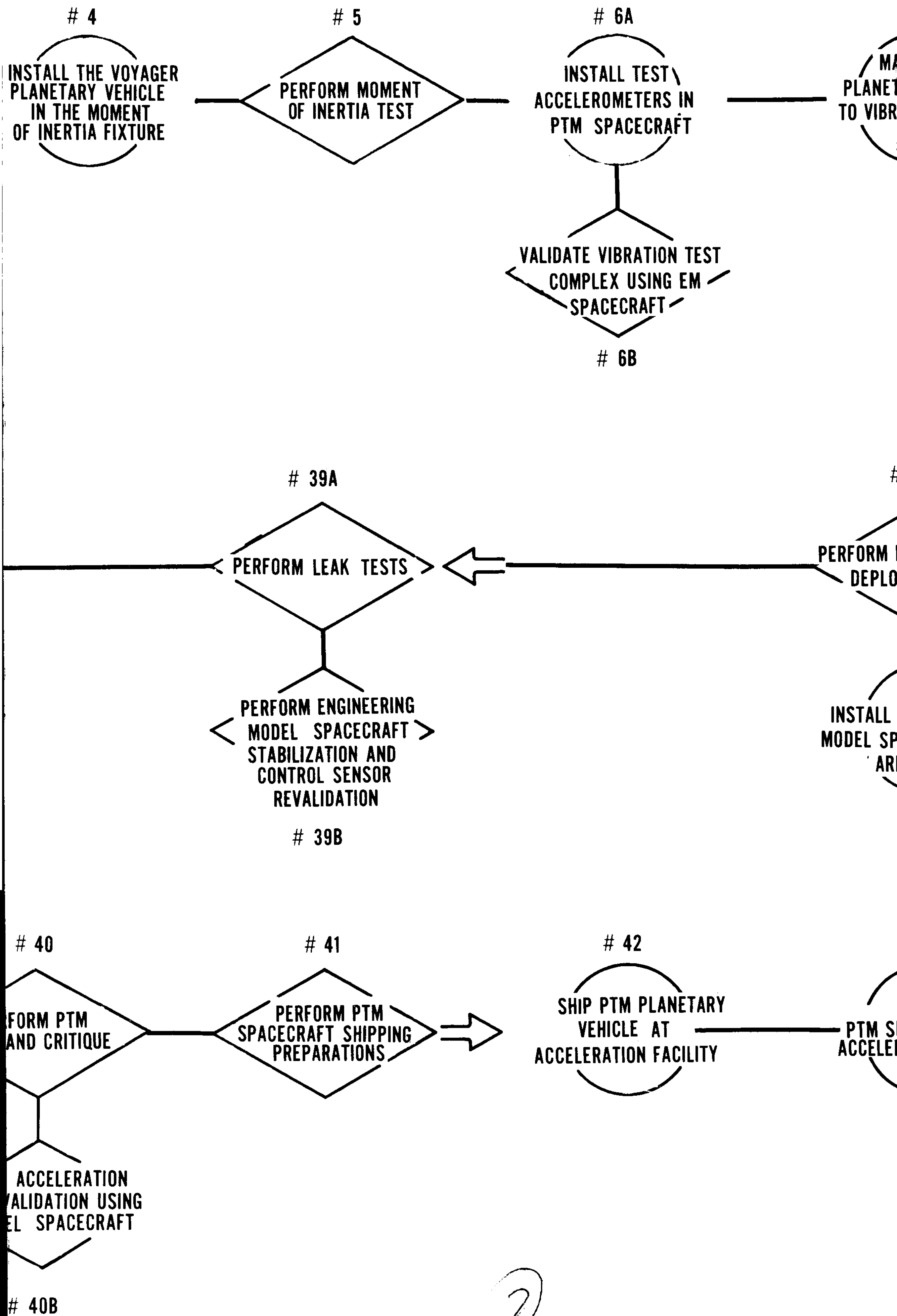
Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
91	<u>Calibrate Magnetometer Experiment</u> The magnetometer boom is deployed and the magnetometer extended into the coil system. Precision currents are fed through the coil system to generate known magnetic field strengths, as well as to buck out the effects of the earth's field. EOSE measurements and telemetered data are compared with the known fields generated by the coil system. These parameters are entered into the computer program.	Coil system to buck out earth's magnetic field, complete set of systems test EOSE, long EOSE cables, slings, handling fixture	Procedure	Crane with hook height of ____
92	<u>Perform Shipping Preparations</u> The Voyager planetary vehicle booms and other appendages are folded and latched. The spacecraft is placed in the shipping container. Next desicate is placed in the shipping container and the container sealed. The shipping container and spacecraft are purged with dry nitrogen. Note that it will be necessary to remove the array panels and support structure for shipment.	Slings, shipping containers, purging equipment		
93	<u>Ship Voyager Planetary Vehicle to Redondo Beach</u> After magnetic testing the spacecraft is to be placed into the shipping container and sealed with desicate. The nitrogen purging equipment is next attached and purging started. The spacecraft and shipping container are shipped via helicopter back to Redondo Beach.	Slings, shipping container, desicate, purging equipment, helicopter	Procedure	Crane with hook height of ____
94	<u>Prepare Voyager Spacecraft for Alignments and Leak Testing</u> After the spacecraft has been removed from the shipping contained it will be placed upon the tilt fixture and the solar array panels and support structure installed in preparation for spacecraft alignments and leak testing.	Slings, tilt fixture, torque wrench	Procedure	Crane with hook height of ____

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
95	<u>Perform Leak Tests</u> After the vibration test has been completed, the SCS pneumatic system, the monopropellant engine system and the solid engine TVD system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced dur to shipping and handling operations. During this leak test all tank pressure and temperature calibrations will take place.	System test set EOSE, SCS leak test console, propulsion leak test console	Procedure	None
96	<u>Perform Spacecraft Alignments</u> After the leak test has been completed, all spacecraft alignments will be checked. Listed below are all of the alignments that will be checked: a. Solid retropropulsion motor b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Omni antenna alignments l. Omni antenna boom latch alignments m. Magnetometer experiment alignments n. Magnetometer boom latch alignments o. Planetary vehicle vertical alignments p. Medium-gain antenna alignments q. Medium-gain antenna latch alignments	Alignments sets, auto-collimators	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
97	Perform Appendage Deployment Test After the alignment test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure or misalignment.	Systems test set EOSE, deployment fixtures	None	None
98	Mate the Planetary Vehicle to the Centaur Adapter	Slings, torque wrenches, tag lines	Procedure	Crane with hook height of ____
99	Perform Spacecraft Vertical Alignment The spacecraft vertical alignment will be checked optically and scribe marks used as reference points, once the alignment has been completed.	Spacecraft vertical alignment set	Procedure	None
100	Electrically Revalidate All Solar Array Structure Mounted SCS and Experiment Sensors	System test set EOSE	Procedure	None
101	Perform Integrated System Test The integrated system test is performed at this time to establish base line conditions prior to undergoing type approval testing.	System test set EOSE	Procedure	None
102	Perform Integrated System Test Critique The integrated system test critique is a meeting of all cognizant personnel to discuss the results of the integrated system test. It is during this meeting that each subsystem engineer signs off the IST data.	None	Records to be signed off	None

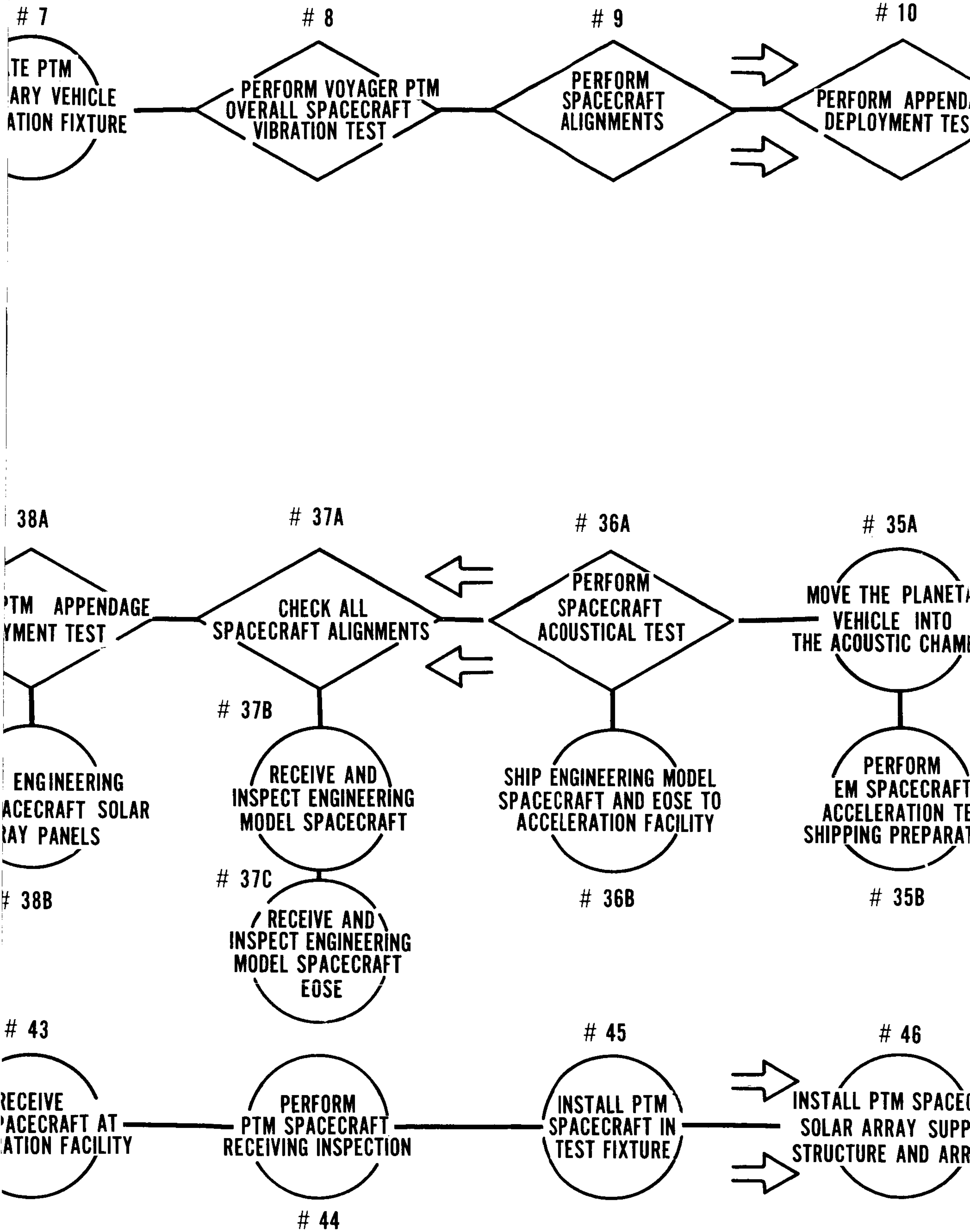


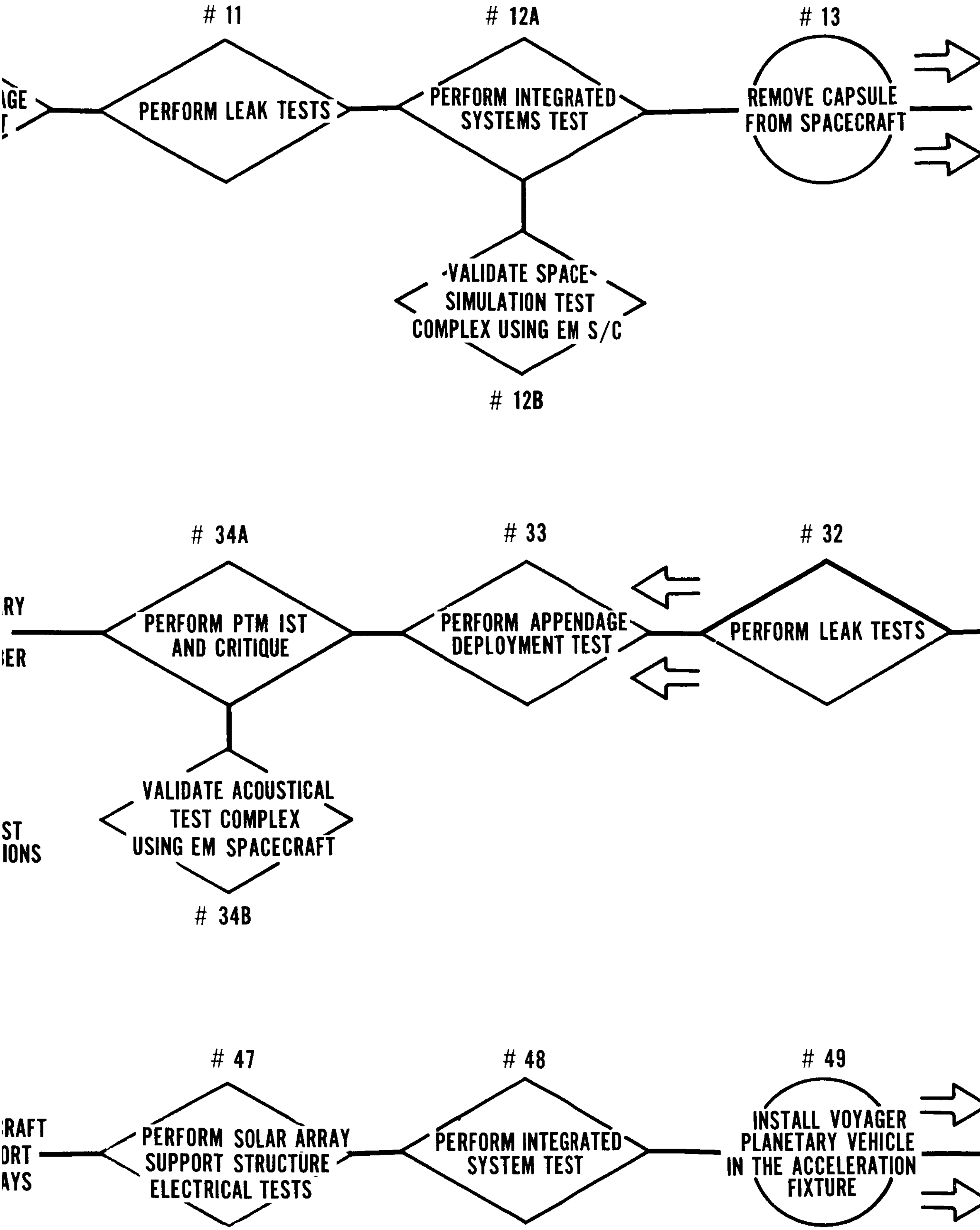
1
INSTALL THE VOYAGER PROOF TEST MODEL SPACECRAFT INTO THE WEIGHT AND C. G. FIXTURE
2
DETERMINE THE WEIGHT OF THE VOYAGER PLANETARY VEHICLE
3
PERFORM THE CENTER OF GRAVITY DETERMINATION TEST
PERFORM
FACILITY



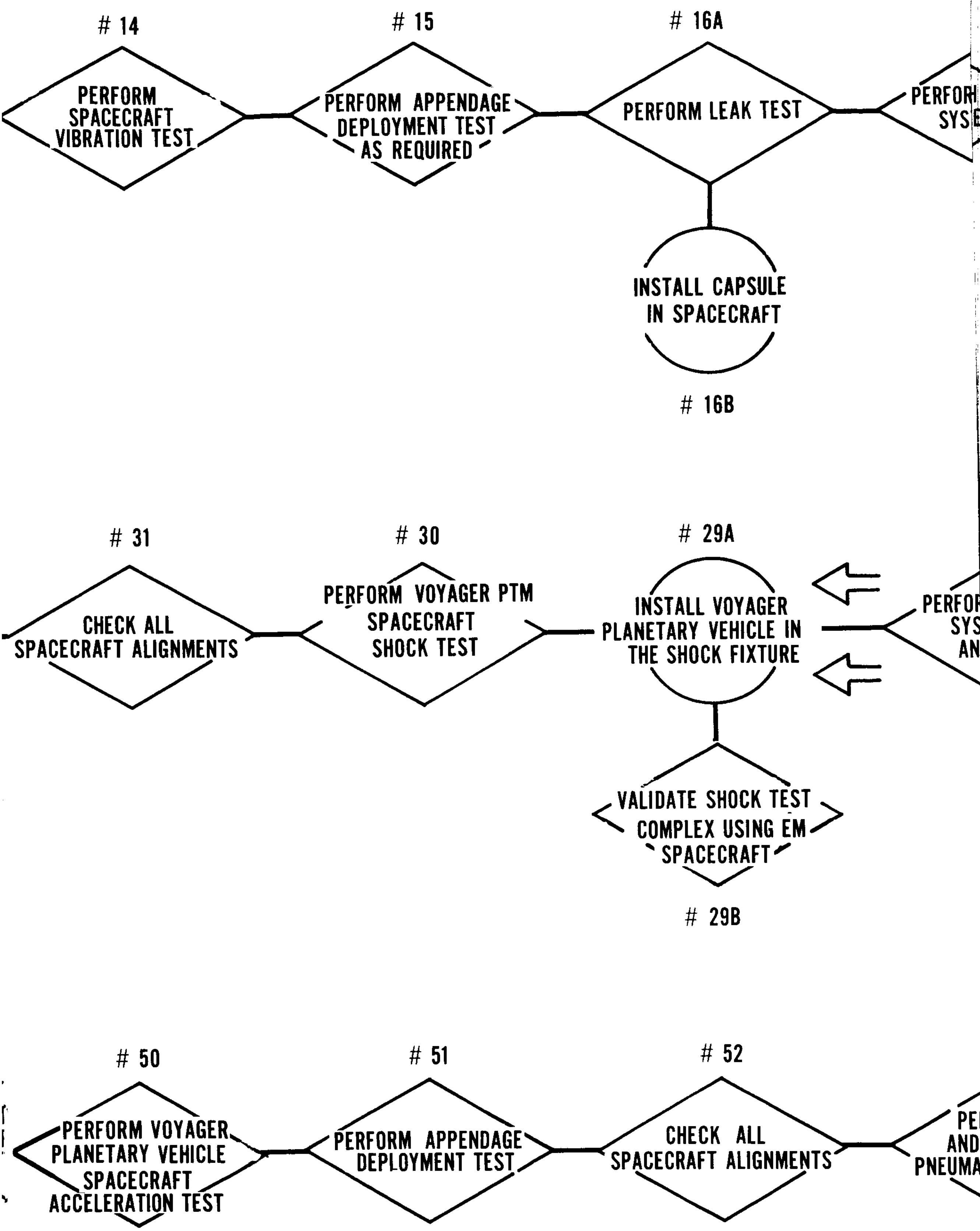
4
INSTALL THE VOYAGER PLANETARY VEHICLE IN THE MOMENT OF INERTIA FIXTURE
5
PERFORM MOMENT OF INERTIA TEST
6A
INSTALL TEST ACCELEROMETERS IN PTM SPACECRAFT
VALIDATE VIBRATION TEST COMPLEX USING EM SPACECRAFT
6B
39A
PERFORM LEAK TESTS
PERFORM ENGINEERING MODEL SPACECRAFT STABILIZATION AND CONTROL SENSOR REVALIDATION
39B
40
41
PERFORM PTM SPACECRAFT SHIPPING PREPARATIONS
42
SHIP PTM PLANETARY VEHICLE AT ACCELERATION FACILITY
40B

2


7
TE PTM
ARY VEHICLE
ATION FIXTURE
8
PERFORM VOYAGER PTM
OVERALL SPACECRAFT
VIBRATION TEST
9
PERFORM
SPACECRAFT
ALIGNMENTS
10
PERFORM APPEND
DEPLOYMENT TES
38A
PTM APPENDAGE
YMENT TEST
37A
CHECK ALL
SPACECRAFT ALIGNMENTS
36A
PERFORM
SPACECRAFT
ACOUSTICAL TEST
35A
MOVE THE PLANET
VEHICLE INTO
THE ACOUSTIC CHAM
ENGINEERING
ACECRAFT SOLAR
RAY PANELS
38B
37B
RECEIVE AND
INSPECT ENGINEERING
MODEL SPACECRAFT
37C
RECEIVE AND
INSPECT ENGINEERING
MODEL SPACECRAFT
EOSE
SHIP ENGINEERING MODEL
SPACECRAFT AND EOSE TO
ACCELERATION FACILITY
36B
PERFORM
EM SPACECRAFT
ACCELERATION TE
SHIPPING PREPARAT
35B
43
RECEIVE
PACECRAFT AT
ATION FACILITY
PERFORM
PTM SPACECRAFT
RECEIVING INSPECTION
44
45
INSTALL PTM
SPACECRAFT IN
TEST FIXTURE
46
INSTALL PTM SPACE
SOLAR ARRAY SUPP
STRUCTURE AND ARR

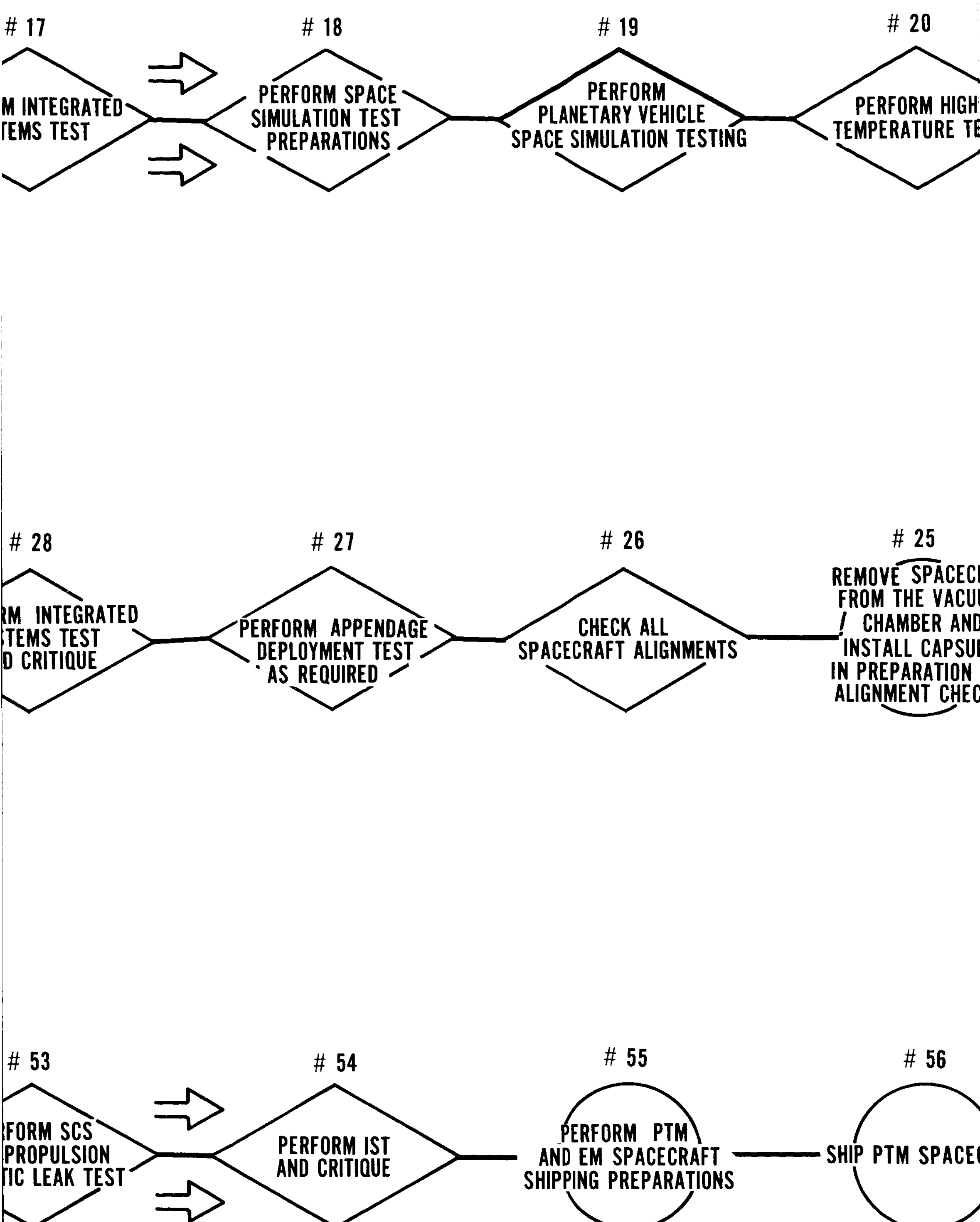


11
PERFORM LEAK TESTS
12A
PERFORM INTEGRATED SYSTEMS TEST
13
REMOVE CAPSULE FROM SPACECRAFT
VALIDATE SPACE SIMULATION TEST COMPLEX USING EM S/C
12B
34A
PERFORM PTM IST AND CRITIQUE
33
PERFORM APPENDAGE DEPLOYMENT TEST
32
PERFORM LEAK TESTS
VALIDATE ACOUSTICAL TEST COMPLEX USING EM SPACECRAFT
34B
47
PERFORM SOLAR ARRAY SUPPORT STRUCTURE ELECTRICAL TESTS
48
PERFORM INTEGRATED SYSTEM TEST
49
INSTALL VOYAGER PLANETARY VEHICLE IN THE ACCELERATION FIXTURE


14
PERFORM SPACECRAFT VIBRATION TEST
15
PERFORM APPENDAGE DEPLOYMENT TEST AS REQUIRED
16A
PERFORM LEAK TEST
INSTALL CAPSULE IN SPACECRAFT
16B
31
CHECK ALL SPACECRAFT ALIGNMENTS
30
PERFORM VOYAGER PTM SPACECRAFT SHOCK TEST
29A
INSTALL VOYAGER PLANETARY VEHICLE IN THE SHOCK FIXTURE
VALIDATE SHOCK TEST COMPLEX USING EM SPACECRAFT
29B
50
PERFORM VOYAGER PLANETARY VEHICLE SPACECRAFT ACCELERATION TEST
51
PERFORM APPENDAGE DEPLOYMENT TEST
52
CHECK ALL SPACECRAFT ALIGNMENTS

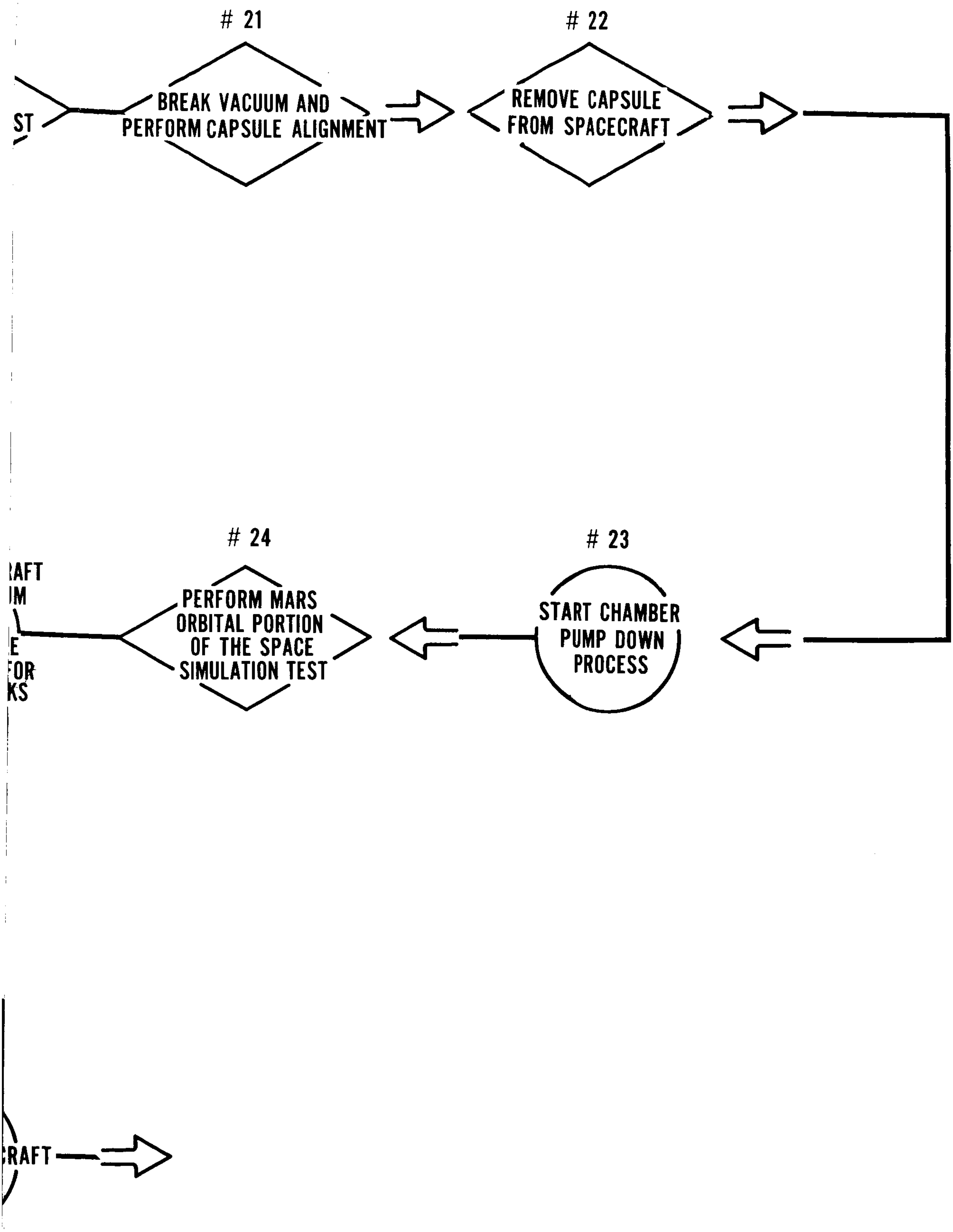
VOYAGER 1

5



71 MISSION PROOF TEST MODEL S/C TYP

6



E APPROVAL TESTING

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Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ____ Date ____ Approval ____ Page No. 1

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1	<u>Install the Voyager Proof Test Model Planetary Vehicle into the Weight and c.g. Fixture</u>	Hand tools, torque wrenches, c.g. fixture	None	Some means of hoisting the spacecraft into the c.g. fixture
2	<u>Determine the Weight of the Voyager Planetary Vehicle</u>			
	The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. Note that the weight of the spacecraft less capsule was determined during assembly and test.	Load cells and associated electronics, c.g. fixture	Procedure	None
3	<u>Perform the Center of Gravity Determination Test</u>	C.g. fixture	Procedure	None
	The center of gravity for two of the spacecraft axes was determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axis. Note that the center of gravity determination of the spacecraft less capsule was determined during assembly and test.			
4	<u>Install the Voyager Planetary Vehicle in the Moment of Inertia Fixture</u>	Inertia fixture, slings	None	Some means of hoisting the spacecraft into the inertia fixture
5	<u>Perform Moment of Inertia Test</u>			
	The moments of inertia about the roll axis and the maximum and minimum moments about the transverse axis will be determined and compared with design requirements. Note that the moment of inertia determination of the spacecraft less capsule was determined during assembly and test.	Timer	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
6A	Install Test Accelerometers in the PTM Spacecraft Test accelerometers will be used to monitor the forces acting on the spacecraft during the vibration test.	Test accelerometers, accelerometer electronics	Procedure	None
6B	Validate Vibration Test Complex Using EM Spacecraft The engineering model spacecraft will be utilized to verify the vibration test cabling and EOSE.	Vibration test EOSE, vibration test cables	Procedure	None
7	Mate the Voyager Planetary Vehicle to the Vibration Fixture	Vibration fixture, slings	None	Crane with hook height of ______
8	Perform Voyager Spacecraft Vibration Test The purpose of the vibration test is to demonstrate the capability of the planetary vehicle to withstand the mission vibration environments as specified in the Voyager mission environmental specification. It is expected that these environments will consist of low frequency sinusoid and random inputs that could occur during the launch boost phase and the spacecraft retropropulsion phase of the mission sequence. The vibration test will be performed as follows: a. Calibrate accelerometers b. Start vibrating spacecraft and search for mechanical resonances and amplifications c. Perform frequency vibration test d. Perform random vibration test e. repeat items b through d for each axis. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels will be filled to flight specification.	Complete set of EOSE vibration tables, vibration transducers and recorders, pressurization console, fueling consoles		
9	Perform Spacecraft Alignments After the vibration test has been completed, all spacecraft alignments will be checked. Listed below are all of the alignments that will be checked:	Alignment sets, autocollimators	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. Solid retropropulsion motor b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Omni antenna alignments l. Omni antenna boom latch alignments m. Magnetometer experiment alignments n. Magnetometer boom latch alignments o. Planetary vehicle vertical alignments p. Medium-gain antenna alignments q. Medium-gain antenna latch alignments			
10	Perform Appendage Deployment Test After the vibration test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test set EOSE, deployment fixtures	None	None
11	Perform Leak Tests After the vibration test hasbeen completed, the SCS pneumatic system, the monopropellant engine system and the solid engine TVC system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
12A	Perform Integrated Systems Test The integrated systems test will be performed at the conclusion of the vibration test. The purpose of the integrated systems test is to ascertain that there has been no	Complete set of systems EOSE and cabling	Procedure	Electrical outlets

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	degradation in the Voyager Planetary Vehicle subsystems due to vibration testing.			
12B	Validate Space Simulation Test Complex Using EM Spacecraft Concurrently, while the integrated systems test is being conducted, the engineering model spacecraft will be utilized to verify the space-simulation test cablin g, EOSE, and mechanical fixtures.	Complete set of systems EOSE and cables, ESM model spacecraft	Procedure	Electrical outlets
13	Remove Capsule from Spacecraft	Hand tools	Procedure	Vibration fixtures, vibration table
14	Perform Spacecraft Vibration Test The purpose of the spacecraft vibration test is to demonstrate the capability of the spacecraft to withstand the vibrations that would be expected during the retropropulsion maneuver. The vibration test will be performed as follows: a. Calibrate accelerometers b. Start vibrating spacecraft and search for mechanical resonances and amplifications c. Perform frequency vibration test d. Perform random vibration test e. Repeat items b through d for each axis. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels will be filled to flight specification.	Systems test, EOSE vibration table, pressurization consoles, fueling consoles		
15	Perform Appendage Deployment Test as Required After the vibration test has been completed, each applicable spacecraft appendage will be deployed. The appendage will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test set EOSE, deployment fixtures		

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ____ Date ____ Approval ____ Page No. 5

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
16A	<u>Perform Leak Test</u> After the vibration test has been completed, the SCS pneumatic system, the monopropellant engine system and solid engine TVC subsystem will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
16B	<u>Install Capsule in Spacecraft</u>			
17	<u>Perform Integrated Systems Test</u> The integrated systems test will be performed at the conclusion of the vibration testing phase of type approval testing. The purpose of the integrated systems test is to ascertain that there has been no degration in the Voyager planetary vehicle subsystems due to vibration testing.	Complete set of systems EOSE and cabling	Procedure	Electrical outlets
18	<u>Perform Space Simulation Test Preparations</u> The space simulation preparations consist of the following tasks: install heaters in the planetary vehicle, install thermal couples in the planetary vehicle, install planetary vehicle into the simulation fixture; perform functional tests as a final verification of the space simulation electrical complex and mechanical MOSE.	Sun source, Canopus source, heaters, thermocouple standard solar cells, gas actuator monitoring EOSE	Procedure	Vacuum chamber; electrical outlets for EOSE
19	<u>Perform Voyager Planetary Vehicle Space Simulation Testing</u> The spacecraft simulation testing will be performed as follows: a. When the proper pressure has been reached, the vacuum chamber cold walls will be turned on and the spacecraft allowed to temperature soak b. When the spacecraft has reached the temperature that would be expected during the spacecraft separation portion of the mission sequence, the spacecraft sun acquisition mode will be initiated.	Sun source, Canopus source, heaters, thermocouple, standard solar cells, gas actuator monitoring EOSE	Procedure	Vacuum chamber

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. After the SCS sun acquisition testing has been completed, the solar array testing sequence will commence. The solar array testing phase will consist of the following: 1) The sun simulator output intensity and dispersion will be determined by using standard solar cells 2) The planetary vehicle solar array output will be monitored to determine that the solar array output performance meets specification. 3) The primary power charge control subsystem will be exercised and the performance will be monitored for proper operation. For each charge rate the following relationship must hold: solar array current = shunt regulator current + bus current + battery current			
	d. Following the solar array testing phase of the space simulation test, the Canopus acquisition tests will start. The ability of the Canopus sensor and associated electronics to perform to specification will be monitored.			
	e. After Canopus has been acquired, the cruise science will be turned on and the ability to perform to specifications will be monitored.			
	f. The next event to be checked out in the flight sequence of events will be the midcourse maneuvering sequence. The spacecraft turn maneuvers will be performed in each axis in each direction. The midcourse correction engine jet vane angles will be commanded and checked in each direction. The motor burn time will correspond to the maximum burn time that can be commanded to the spacecraft. The ability to perform to specification of the midcourse sequencing will be monitored.			
	g. It should be mentioned that the SCS, the midcourse correction engine, and the solid engine TVC system leak testing will take place throughout the space simulation test.			

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	h. Post midcourse maneuver cruise mode testing is as follows: 1) Sun acquisition established 2) Canopus acquisition established 3) Spacecraft powered from the sun simulation sourse 4) All cruise science on 5) The RF up and down link (coherent)operation established. All subsystem performance data will be monitored to ascertain that the Voyager planetary vehicle performs within specified limits. i. During the encounter mode of testing the SCS approach guidance will be checked out as well as the capsule separation circuitry. Both subsystems will be checked for proper operation.			
20	Perform High Temperature Test The cold walls will be turned off and the spacecraft temperature allowed to rise to upper specification limit. When the spacecraft has reached its upper limits, each subsystem will be exercised and monitored for proper operation.	None	Procedure	None
21	Break Vacuum and Perform Capsule Alignment Check The chamber vacuum will be released and the capsule alignment checked. The capsule alignment will consist of merely observing that scribe lines on both the capsule and spacecraft have not shifted due to thermal effects.	Slings, capsule handling fixture	Procedure	Crane with hook height of ______
22	Remove Capsule from Spacecraft	Hand tools, capsule handling fixture	Procedure	Overhead crane with hook length of ______
23	Start Chamber Pumpdown Process			

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 8

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
24	<u>Perform Mars Orbital Portion of the Space Simulation Test</u> The Mars orbital testing will be performed as follows: a. When the chamber has reached the proper pressure, the cold walls will be turned on. When the spacecraft has reached the lower temperature limit, spacecraft power will be turned on and each subsystem checked for proper operation. This test ensures that the spacecraft will survive the Mars eclipse. b. The Mars sun intensity level will be established c. The retropropulsion subsystem will be tested for proper operation. This test will include the testing of the solid engine thrust vector control system and the engine ignition system. d. The Mars orbital portion of the SCS subsystem will be checked for proper operation. e. The planet-oriented package and associated experiment packages will be checked for proper operation. f. All other subsystems will be checked for proper operation. g. The cold walls will be turned off and the spacecraft temperature allowed to reach its upper limit. h. When the spacecraft has reached its upper temperature limit, each spacecraft subsystem will be checked for proper operation.	Complete set of EOSE and cables	Procedure	Space simulation chamber
25	<u>Remove Spacecraft from the Vacuum Chamber and Install Capsule in Preparation for Alignment Checks</u>	Slings, capsule handling fixture, spacecraft handling fixture	Procedure	Crane with hook height of ______
26	<u>Check all Spacecraft Alignments</u> All spacecraft alignments will be checked for shifts due to thermal effects. Listed below are the spacecraft alignments that will be checked: a. Solid retropropulsion motor alignment b. Monopropellant motor alignment c. Capsule alignments	Complete complement of alignment sets, autocollimators, bench marks	Procedure	Bench marks

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	d. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Medium-gain antenna alignments l. Medium-gain antenna latch alignments m. Omni antenna alignments n. Omni antenna latch alignments o. Magnetometer experiment alignment p. Magnetometer experiment latch alignment q. Planetary vehicle vertical alignments			
27	Perform Boom Deployment Test As Required After the space simulation test has been completed, each spacecraft applicable appendage will be deployed. The appendate will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment. Currently an investigation is being undertaken to ascertain the feasibility of performing appendage deployment tests in the space simulation.	Systems test EOSE, deployment fixtures	Procedure	None
28	Perform Integrated Systems Test and Critique The IST is performed at this time for two reasons: To verify that the planetary vehicle and all of its subsystems operate properly at atmospheric pressure. Often failures due to vacuum become evident only when the chamber vacuum is released; and to perform any subsystem test that could not adequately be performed to mechanical and electrical constraints that are incurred when operating a spacecraft in a space simulator.	Complete set of systems test OESE	Procedure	Electrical outlets

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
29A	<u>Install Voyager Planetary Vehicle in the Shock Fixture</u>	Slings, spacecraft handling fixture, shock fixture	Procedure	Electrical outlets
29B	<u>Validate the Shock Test Complex Using EM Spacecraft</u> Concurrently the electrical compatibility model spacecraft will be used to validate the systems test set and MOSE and EOSE that will be used for shock testing.	ECM spacecraft, complete set of shock EOSE, shock transducers and electronics	Procedure	Electrical outlets
30	<u>Perform Voyager Spacecraft Shock Test</u> The purpose of the shock test is to demonstrate the capability of the planetary vehicle to withstand the mission shock environments as specified in the Voyager mission environmental specification. Note that the spacecraft will be electrically powered and that all pneumatic and fuel vessels will be filled to flight specifications.	Shock test fixture, shock test transducers and electronics, shock test EOSE	Procedure	Shock test fixture electrical outlets
31	<u>Check All Spacecraft Alignments</u> All spacecraft alignments will be checked for shifts due to the above mentioned shock environments. The spacecraft alignments to be checked are: a. Solid retropropulsion motor alignment b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Gas jet alignments g. Canopus sensor alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Medium-gain antenna alignments l. Medium-gain antenna latch alignments m. Omni antenna alignments	Complete complement of alignment sets, autocollimators	Procedure	None

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ____ Date ____ Approval ____ Page No. 11

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	n. Omni antenna latch alignments o. Magnetometer experiment alignment p. Magnetometer experiment latch alignment q. Planetary vehicle vertical alignment			
32	Perform Leak Tests After the vibration tests have been completed, the SCS pneumatic system, the monopropellant engine system, and the solid engine TVC system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
33.	Perform Appendage Deployment Test After the vibration test has been completed, each applicable spacecraft appendage will be deployed. The appendage will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure,or misalignment.	Systems test set EOSE, Deployment fixtures	None	None
34A	Perform PTM IST and Critique The integrated systems test will be performed to verify that the planetary vehicle and all of its subsystems have successfully survived the shock test.	Complete set of systems test OSE	Procedure	Electrical outlets
34B	Validate the Acoustical Test Complex Using EM Spacecraft Concurrently the electrical compatibility model spacecraft will be used to validate the systems test set and MOSE and EOSE that will be used for acoustical testing.			
35A	Move the Planetary Vehicle into the Acoustic Chamber	Handling fixture, slings, transporter	Procedure	Overhead crane with hook height of ____

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing — Revision ______ Date ______ Approval ______ Page No. 12

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
35B	Perform EM Spacecraft Acceleration Test Shipping The spacecraft shipping preparations will include both the engineering model spacecraft and its system test set EOSE. The engineering model will be used to check out the acceleration complex. The solar arrays, support structure, and equipment mounted on the array structure will be removed from the spacecraft for shipment.	Slings, handling fixtures, shipping containers, purging equipment	Procedure	None
36A	Perform Spacecraft Acoustical Test The purpose of the acoustical test is to demonstrate the capability of the planetary vehicle to withstand the acoustical environments as specified in the Voyager mission environmental specification. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels filled to flight specification.	Acoustical test transducers and electronics, acoustical test EOSE and test cables, noise generators	Procedure	Acoustical chamber, electrical outlets
36B	Ship Engineering Model Spacecraft and EOSE to Acceleration Facility	Slings, handling fixtures, shipping containers, purging equipment	Procedure	None
37A	Check All Spacecraft Alignments All spacecraft alignments will be checked for shifts due to the above mentioned acoustical environments. Listed below are the spacecraft alignments that are to be checked: a. Solid retropropulsion motor alignment b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments	Complete complement of alignment sets, autocollimator	Procedure	None

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ____ Date ____ Approval ____ Page No. 13

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	j. Medium-gain antenna alignments k. Medium-gain antenna latch alignments l. Mapping package alignments m. Omni antenna alignments n. Omni antenna latch alignments o. Magnetometer experiment latch alignment p. Magnetometer experiment alignment q. Planetary vehicle vertical alignments			
37B	Receive and Inspect Engineering Model Spacecraft	Handling fixtures, slings	Procedure	Crane with hook height of ______
37C	Receive and Inspect Engineering Model Spacecraft EOSE	Handling fixtures, slings	Procedure	None
38A	Perform PTM Appendage Deployment Test After the acoustical test has been completed in each axis, each spacecraft appendage will be deployed. The appendage will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test EOSE, deployment fixtures	Procedure	None
	a. Mate EM spacecraft to acceleration fixture b. Validate systems test set EOSE	Slings, handling fixture	Procedure	Crane with hook height of ______
38B	Install Engineering Model Solar Array Panels	Hand tools, torque wrenches	Procedure	None
39A	Perform Leak Tests After the vibration tests have been completed, the SCS pneumatic system, the monopropellant engine system and the solid engine TVC system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
39B	<u>Perform Engineering Model Stabilization and Control Sensor Revalidation</u> The stabilization and control sensor revalidation tests will be performed as follows: a. Attach solar array support structure to the spacecraft b. Electrically revalidate each solar array support structure mounted stabilization and control sensor.	Voltmeter, ammeter, systems test set EOSE, hand tools, torque wrenches	Procedure	None
40A	<u>Perform PTM IST and Critique</u> The integrated system test will be performed to verify that the planetary vehicle and all of its subsystems have successfully survived the acoustical test.	Complete set of systems test EOSE	Procedure	Electrical outlets
40B	<u>Perform Acceleration Facility Validation</u> The engineering model spacecraft and EOSE will be used to validate the acceleration facility cabling and specialized EOSE and MOSE.	System test set EOSE, voltmeter, ammeter		
41	<u>Perform PTM Spacecraft Shipping Preparations</u> The spacecraft shipping preparations will include shipping both the planetary vehicle and the systems test set EOSE. The solar arrays and the solar array support structure will be removed from the spacecraft for shipment.	Slings handling fixtures, shipping containers, purging equipment	Procedure	None
42	<u>Ship PTM Planetary Vehicle to Acceleration Facility</u> TRW believes that the data obtained from this type of test is not significant in view of the more scarce loading obtained during vibration testing; however it is included for reference in the event JPL deems it necessary.	Slings, handling fixtures, shipping containers, purging equipment	Procedure	None
43	<u>Receive PTM Spacecraft at Acceleration Facility</u>	None	Procedure	None

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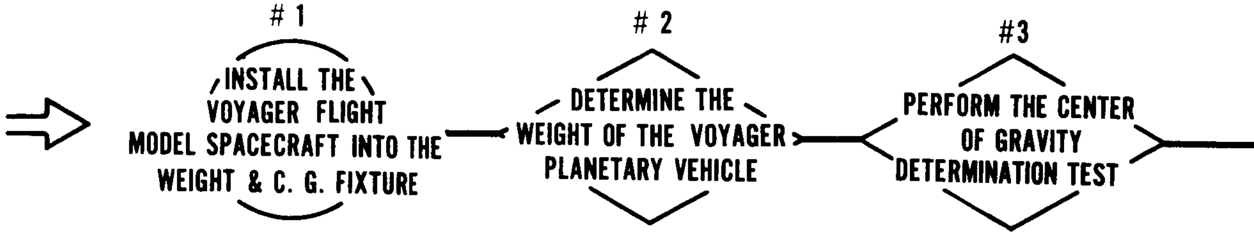
Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 15

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
44	Perform PTM Spacecraft Receiving Inspection The receiving inspection will be for shipping and handling damage.	None	Procedure	None
45	Install PTM Spacecraft in Test Fixture	Slings, handling fixture,	Procedure	Crane with hook height of ______
46	Install PTM Spacecraft Solar Array Support Structure and Arrays	Hand tools, torque wrenches	Procedure	None
47	Perform Solar Array Support Structure Electrical Tests The solar array support structure testing and calibrations will be accomplished as follows: illuminate each solar array string and measure the short circuit current and open circuit voltage; perform inverse impedence tests on each solar array string; electrically revalidate each array-mounted stabilization and control subsystem and experiment instrument.	Solar array test set, voltmeter, ammeter, systems test set EOSE	Procedure	None
48	Perform Integrated System Test The integrated system test will be performed to verify that the spacecraft has incurred no damage due to shipping and handling.	Systems test set	Procedure	None
49	Install Voyager Planetary Vehicle in the Acceleration Fixture	Slings, spacecraft handling fixture, acceleration fixture	Procedure	Acceleration machine, electrical outlets

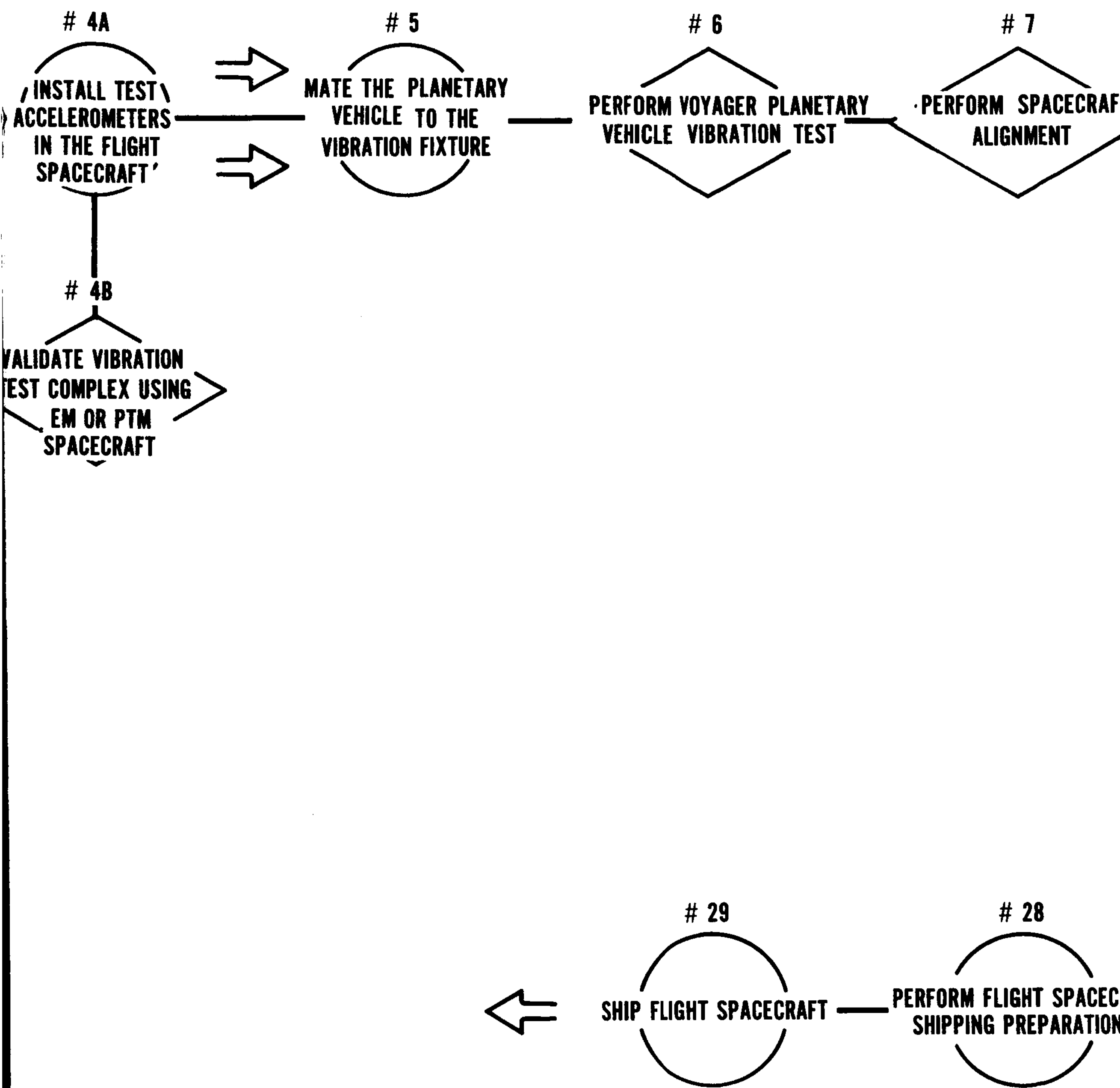
Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
50	<u>Perform Voyager Planetary Vehicle Acceleration Test</u> The purpose of the acceleration test is to demonstrate the capability of the Voyager planetary vehicle to withstand the mission acceleration environments as specified in the Voyager mission environmental specification. Note that the spacecraft will be electrically powered and that all pneumatic and fuel vessels will be filled to flight specifications.	Complete set of acceleration OESE, acceleration test transducers and electronics	Procedure	Acceleration machine, electrical outlets
51	<u>Perform Appendage Deployment Test</u> After the acceleration test has been completed in each axis, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test EOSE, deployment fixtures	Procedure	None
52	<u>Check All Spacecraft Alignments</u> All spacecraft alignments will be checked for shifts due to the above mentioned acceleration environments. Listed below are the spacecraft alignments that are to be checked. a, Solid retropropulsion motor alignment b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Medium-gain antenna alignments l. Medium-gain antenna latch alignments m. Omni antenna alignments n. Omni antenna latch alignments	Complete complement of alignment sets, autocollimators	Procedure	None

Functional Flow Drawing Title and No. Proof Test Model Spacecraft Type Approval Testing Revision ______ Date ______ Approval ______ Page No. 18

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	o. Magnetometer experiment alignment p. Magnetometer experiment latch alignment q. Planetary vehicle vertical alignments			
53	<u>Perform SCS and Propulsion Pneumatic Leak Test</u> The stabilization and control subsystem and the monopropellant propulsion engine subsystem will be tested for leaks that may have been incurred during acceleration testing	SCS leak test console, midcourse motor leak test console	Procedure	None
54	<u>Perform IST and Critique</u> The IST will be performed to verify that the planetary vehicle and all of its subsystems have successfully survived the acceleration test.	Complete set of systems test EOSE	Procedure	
55	<u>Perform PTM and EM Spacecraft Shipping Preparations</u> The spacecraft shipping preparations will include both the spacecraft and their system test set EOSE. The solar arrays and equipment mounted on the arrays as well as the support structures will be removed from the spacecraft for shipment.	Slings, handling fixtures, purging equipment	Procedure	Crane with hook height of ______
56	<u>Ship PTM Spacecraft</u>	Slings, handling fixtures, purging equipment	Procedure	Crane with hook height of ______

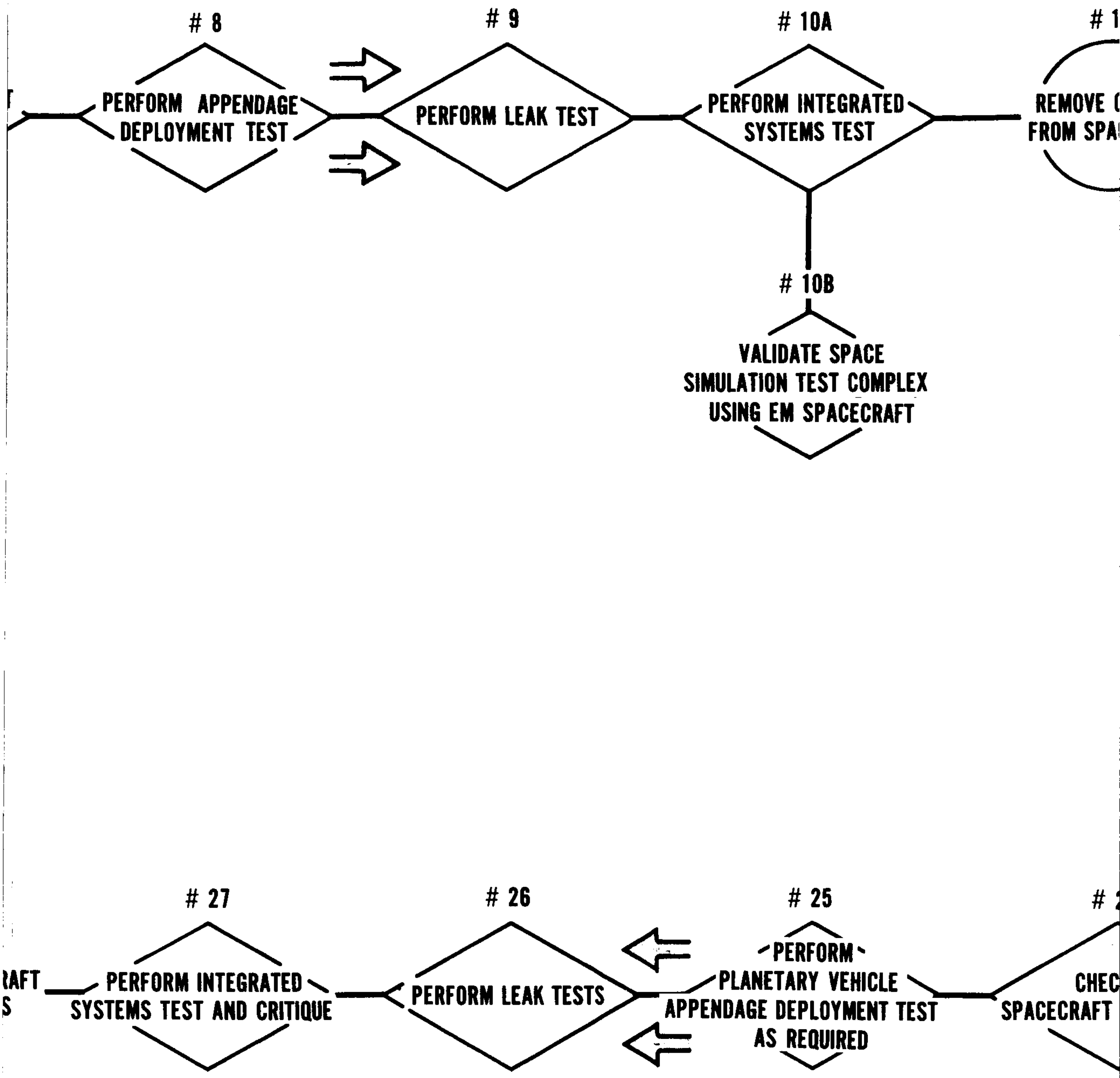


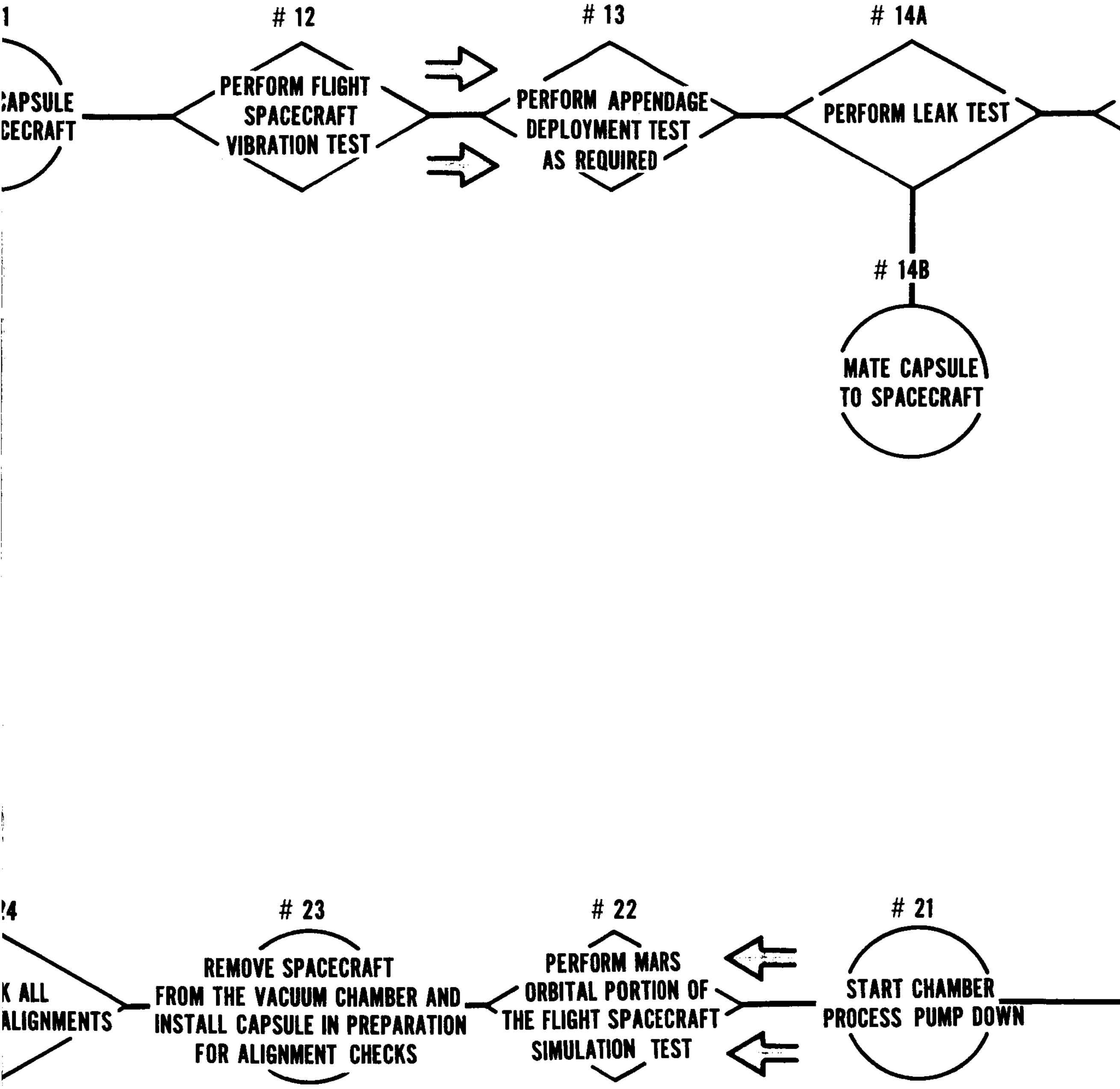
1
INSTALL THE VOYAGER FLIGHT MODEL SPACECRAFT INTO THE WEIGHT & C. G. FIXTURE
2
DETERMINE THE WEIGHT OF THE VOYAGER PLANETARY VEHICLE
#3
PERFORM THE CENTER OF GRAVITY DETERMINATION TEST



4A
INSTALL TEST ACCELEROMETERS IN THE FLIGHT SPACECRAFT
5
MATE THE PLANETARY VEHICLE TO THE VIBRATION FIXTURE
6
PERFORM VOYAGER PLANETARY VEHICLE VIBRATION TEST
7
PERFORM SPACECRAF
ALIGNMENT
4B
VALIDATE VIBRATION
EST COMPLEX USING
EM OR PTM
SPACECRAFT
29
SHIP FLIGHT SPACECRAFT
28
PERFORM FLIGHT SPACEC
SHIPPING PREPARATION

2


8
PERFORM APPENDAGE DEPLOYMENT TEST
9
PERFORM LEAK TEST
10A
PERFORM INTEGRATED SYSTEMS TEST
1
REMOVE
FROM SPA
10B
VALIDATE SPACE SIMULATION TEST COMPLEX USING EM SPACECRAFT
27
PERFORM INTEGRATED SYSTEMS TEST AND CRITIQUE
26
PERFORM LEAK TESTS
25
PERFORM PLANETARY VEHICLE APPENDAGE DEPLOYMENT TEST AS REQUIRED
CHEC
SPACECRAFT



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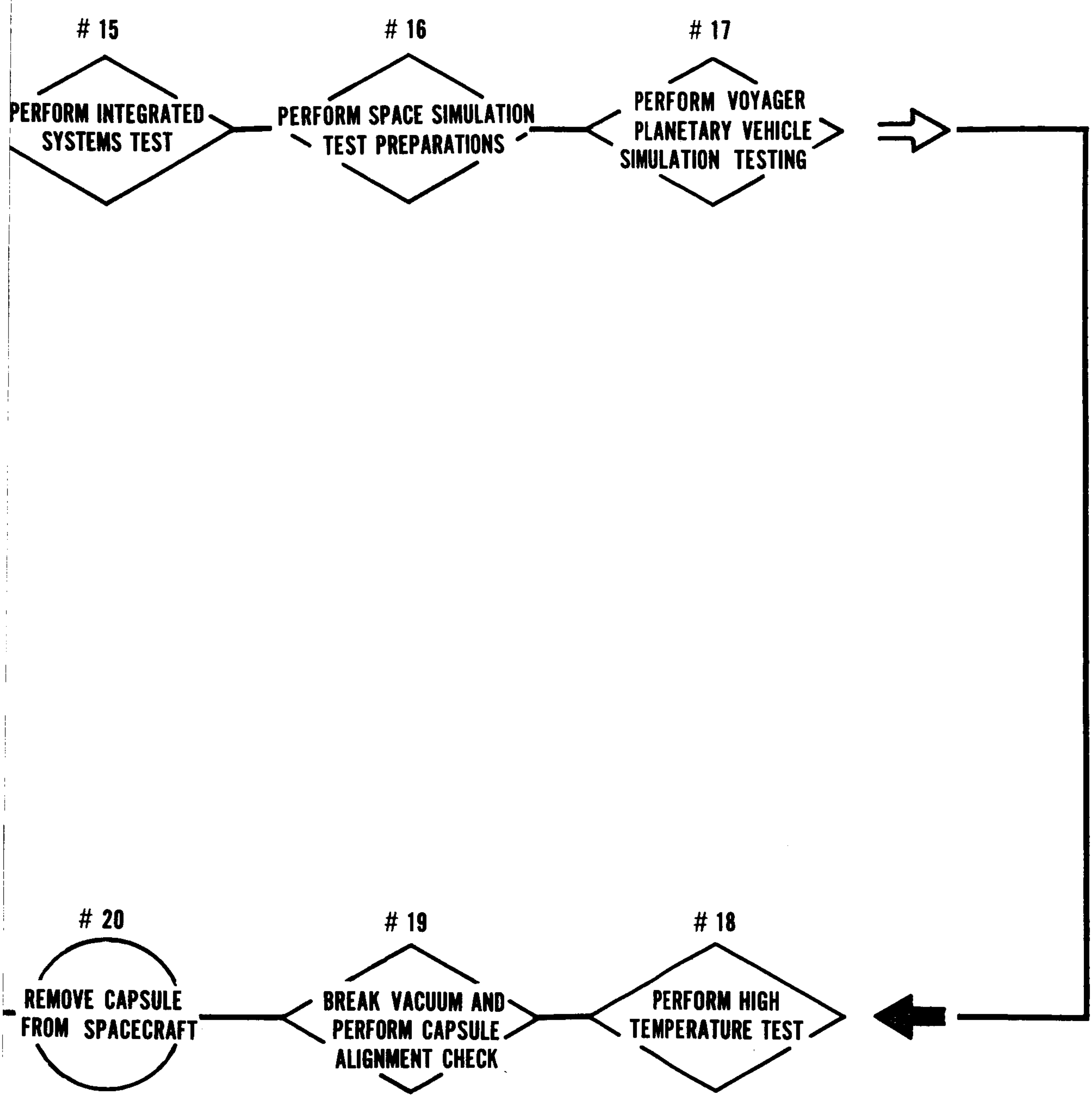

AFT FLIGHT APPROVAL TESTING

5

Functional Flow Flight Model Spacecraft
Flight Approval Testing
Drawing Title and No. ________ **Revision** ______ **Date** ______ **Approval** ________ **Page No.** 1 1971

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1	<u>Install the Voyager Flight Model Planetary Vehicle into the Weight and C.G. Fixture</u>	Hand tools, torque wrenches, c.g. fixture	None	Some means of hoisting the spacecraft into the c.g. fixture.
2	<u>Determine the Weight of the Voyager Planetary Vehicle</u> The spacecraft will be weighted using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. Note that the weight of the spacecraft less capsule was determined during assembly and test.	Load cells and Associated Electronics, c.g. fixture	Procedure	None
3	<u>Perform the Center of Gravity Determination Test</u> The centers of gravity fcr two of the spacecraft axes were determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axes. Note that the center of gravity determination of the spacecraft less capsule was determined during assembly and test.	C.g. fixture	Procedure	None
4A	<u>Install Test Accelerometers in the Flight Spacecraft</u> Test accelerometers will be used to monitor the forces acting on the spacecraft during the vibration test.	Test accelerometers, accelerometer electronics.	Procedure	None
4B	<u>Validate Vibration Test Complex Using EM or PTM Spacecraft</u> The Engineering model or PTM spacecraft will be utilized to verify the vibration test cabling and EOSE.	Vibration test EOSE, vibration test cables	Procedure	None

Functional Flow Flight Model Spacecraft Flight Approval Testing
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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
5	Mate the Planetary Vehicle to the Vibration Fixture	Vibration fixture, slings	None	Crane with hook weight of ______
6	Perform Voyager Planetary Vehicle Vibration Test The purpose of the vibration test is to demonstrate the capability of the planetary vehicle to withstand the mission vibration environments as specified in the Voyager mission environmental specification. It is expected that these environments will consist of low frequency sinusoid and random inputs that could occur during the launch boost phase and the spacecraft retropropulsion phase of the mission sequence. The vibration test will be performed as follows: a. Calibrate accelerometers b. Start vibrating spacecraft and search for mechanical resonances and amplifications c. Perform frequency vibration test d. Perform random vibration test e. Repeat items b through d for each axis. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels will be filled to flight specifications.	Complete set of EOSE vibration tables, vibration transducers and recorders, pressurization console, fueling consoles	Procedure	Vibration fixtures, vibration tables
7	Perform Spacecraft Alignments After the vibration test has been completed, all spacecraft alignments will be checked. Listed below are all of the alignments that will be checked: a. Solid retropropulsion motor b. Monopropellant motor alignment c. Capsule alignments d. Gyro alignments e. Sun sensor alignments f. Canopus sensor slignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments	Alignment sets, autocollimators	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	k. Omni antenna alignments l. Omni antenna boom latch alignments m. Magnetometer experiment alignments n. Magnetometer boom latch alignments o. Planetary vehicle vertical alignments p. Medium-gain antenna alignments q. Medium-gain antenna latch alignments			
8	Perform Appendage Deployment Test After the vibration test has been completed, each spacecraft appendage will be deployed. Each appendage will be deployed in a simulated zero g field using live ordnance observing that each appendage freely deploys, with no mechanical resistance or cable chaffing due to electrical cables, mechanical failure or misalignment.	Systems test set EOSE, deployment fixtures	None	None
9	Perform Leak Test After the vibration test has been completed, the SCS pneumatic system, the monopropellant engine system and the solid engine TVC system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
10A	Perform Integrated Systems Test The integrated systems test will be performed at the conclusion of the vibration test. The purpose of the integrated systems test is to ascertain that there has been no degradation in the Voyager planetary vehicle subsystems due to vibration testing.	Complete set of systems EOSE and cabling	Procedure	Electrical outlets

Functional Flow Flight Model Spacecraft Flight Approval Testing
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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
10B	Validate Space Simulation Test Complex Using EM Spacecraft Concurrently, while the integrated systems test is being conducted, the engineering model spacecraft will be utilized to verify the space-simulation test cabling, EOSE, and mechanical fixtures.	Complete set of systems EOSE and cables, EM model spacecraft	Procedure	Electrical outlets for EOSE
11	Remove Capsule from Spacecraft	Hand tools,	None	None
12	Perform Flight Spacecraft Vibration Test The purpose of the spacecraft vibration test is to demonstrate the capability of the spacecraft to withstand the vibrations expected during the retropropulsion maneuver. The vibration test will be performed as follows: a. Calibrate accelerometers b. Start vibrating spacecraft and search for mechanical resonances and amplifications c. Perform frequency vibration test d. Perform random vibration test e. Repeat items b through d for each axis. Note that the spacecraft will be electrically powered and all pneumatic and fuel vessels will be filled to flight specification.	Systems test EOSE, vibration table, pressurization consoles, fueling consoles		
13	Perform Appendage Deployment Test as Required After the vibration test has been completed, each spacecraft appendage will be deployed. The appendage will be deployed in a simulated zero g field using live ordnance, observing that each 900 K cable appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment.	Systems test set EOSE, deployment fixtures		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
14A	Perform Leak Test After the vibration test has been completed, the SCS pneumatic system, subsystem monopropellant engine system and the solid engine TVC will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
14B	Mate Capsule to Spacecraft	Hand tools torque wrench		
15	Perform Integrated Systems Test The integrated systems test will be performed at the conclusion of the vibration testing phase of flight approval testing. The purpose of the integrated systems test is to ascertain that there has been no degradation in the Voyager planetary vehicle subsystems due to vibration testing.	Complete set of systems EOSE and cabling	Procedure	Electrical outlets for EOSE
16	Perform Space Simulation Test Preparations The space simulation preparations consist of the following tasks: a. Install heaters in the planetary vehicle b. Install thermal couples in the planetary vehicle c. The installation of the planetary vehicle into the simulation fixture d. Functional test as a final verification of the space simulation electrical complex and mechanical MOSE.	Sun source, Canopus source, heaters, thermocouple, standard solar cells, gas actuator monitoring EOSE	Procedure	Vacuum chamber, electrical outlets for EOSE

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
17	<u>Perform Over-all Voyager Planetary Vehicle Simulation Testing</u> The spacecraft simulation testing will be performed as follows: a. When the proper pressure has been reached, the vacuum chamber cold walls will be turned on and the spacecraft allowed to temperature soak. b. When the spacecraft has reached the temperature that would be expected during the spacecraft separation portion of the mission sequence, the spacecraft sun acquisition mode will be initiated. c. After the SCS sun acquisition testing has been completed, the solar array testing sequence will commence. The solar array testing phase will consist of the following: 1) The sun simulator output intensity and dispersion will be determined by using standard solar cells. 2) The spacecraft solar array output will be monitored to determine that the solar array output performance meets specification. 3) The primary power charge control subsystem will be exercised and the performance will be monitored for proper operation. For each charge rate the following relationship must hold: solar array current = shunt regulator current + bus current + battery current. d. Following the solar array testing phase of the space simulation test, the Canopus acquisition tests will start. The ability of the Canopus sensor and associated electronics to perform the specification will be monitored. e. After Canopus has been acquired, the cruise science will be turned on and the ability to perform to specifications will be monitored.	Sun source, Canopus source, heaters, thermocouple, standard solar cells, gas actuator monitoring EOSE		

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	f. The next event to be checked out in the flight sequence of events will be the midcourse maneuvering sequence. The spacecraft turn maneuvers will be performed in each axis in each direction. The midcourse correction engine jet vanes will be commanded and checked in each direction. The motor burn time will correspond to the maximum burn time that can be commanded to the spacecraft. The ability to perform to specification of the midcourse sequencing will be monitored. g. It should be mentioned that the SCS, the midcourse correction engine and the solid motor TVC system leak testing will take place throughout the space simulation test. h. Post midcourse maneuver cruise mode testing. The cruise mode testing mode is as follows: 1) Sun acquisition established 2) Canopus acquisition established 3) Spacecraft powered from the sun simulation source 4) All cruise science on 5) The RF up and down link (coherent) operation simulated with hard line All subsystem performance data will be monitored to ascertain that the Voyager planetary vehicle performs within specified limits. i. During the encounter mode of testing the SCS approach guidance will be checked out as well as the capsule separation circuitry. Both subsystems will be checked for proper operation.			
18	<u>Perform High Temperature Test.</u> The cold walls will be turned off and the spacecraft temperature allowed to rise to upper specification limit. When the spacecraft has reached its upper limits, each subsystem will be exercised and monitored for proper operation.	None	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
19	Break Vacuum and Perform Capsule Alignment Check The chamber vacuum will be released and the capsule alignment checked. The capsule alignment will consist of merely observing that scribe lines on both the capsule and spacecraft have not shifted due to thermal effects.	Slings, capsule handling fixture	Procedure	Crane with hook height of ________
20	Remove Capsule From Spacecraft	Hand tools, capsule handling fixture	Procedure	Overhead crane with hook length of ________
21	START Chamber Pumpdown Process			
22	Perform Mars Orbital Portion of the Flight Spacecraft Simulation Test The Mars orbital testing will be performed as follows: a. When the chamber has reached the proper pressure, the cold walls will be turned on. When the spacecraft has reached the lower temperature limit, spacecraft power will be turned on and each subsystem checked for proper operation. The purpose of this test is to simulate the Mars Eclipses. b. The Mars sun intensity level will be established. c. Next, the retropropulsion subsystem tested for proper operation. This test would include the testing of the solid engine thrust vector control system and the engine ignition system. d. The Mars orbital portion of the SCS subsystem will be checked for proper operation. e. The planet oriented package and associated experiment packages will be checked for proper operation. f. All other subsystems will be checked for proper operation. g. The cold walls will be turned off and the spacecraft temperature allowed to reach its upper limit.	Complete set of EOSE and cables	Procedure	Space simulation chamber

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	h. When the spacecraft has reached its upper temperature limit, each spacecraft subsystem will be checked for proper operation.			
23	Remove Spacecraft from the Vacuum Chamber and Install Capsule in Preparation for Alignment Checks	Slings, capsule handling fixture, spacecraft handling fixture	Procedure	Crane with hook height of ______
24	Check all Spacecraft Alignments All spacecraft alignments will be checked for shifts due to thermal effects. Listed below are the spacecraft alignments that will be checked: a. Solid retropropulsion motor alignment b. Monopropellant motor alignment c. Capsule alignments c. Gyro alignments e. Sun sensor alignments f. Canopus sensor alignments g. Gas jet alignments h. High-gain antenna alignments i. High-gain antenna latch alignments j. Mapping package alignments k. Medium-gain antenna alignments l. Medium-gain antenna latch alignments m. Omni antenna alignments n. Omni antenna latch alignments o. Magnetometer experiment alignment p. Magnetometer experiment latch alignment q. Planetary vehicle vertical alignments	Complete complement of alignment sets autocollimators bench marks	Procedure	Bench Marks

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
25	<u>Perform Planetary Vehicle Appendage Deployment Test As Required</u> After the space simulation test has been completed, each spacecraft appendage will be deployed. The appendage will be deployed in a simulated zero g field using live ordance, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure, or misalignment. An investigation is underway to ascertain the possibility of doing deployment tests in the space simulation chamber.	Systems test EOSE, deployment fixtures	Procedure	None
26	<u>Perform Leak Test</u> After the vibration test has been completed, the SCS pneumatic system and the monopropellant engine system will be leak tested. The purpose of this test is to ascertain that the pneumatic leak and flow rates are within specification and that no damage was experienced due to vibration.	SCS leak test console, propulsion leak test console	Procedure	None
27	<u>Perform Integrated Systems Test and Critique</u> The IST is performed at this time for two reasons: a. To verify that the Voyager planetary vehicle and all of its subsystems operate properly at atmospheric pressure. Often failures due to vacuum become evident only when the chamber vacuum is released. b. To perform any subsystem test that could not adequately be performed to mechanical and electrical constraints that are incurred when operating a spacecraft in a space simulator.	Complete set of systems test EOSE	Procedure	Electrical outlets

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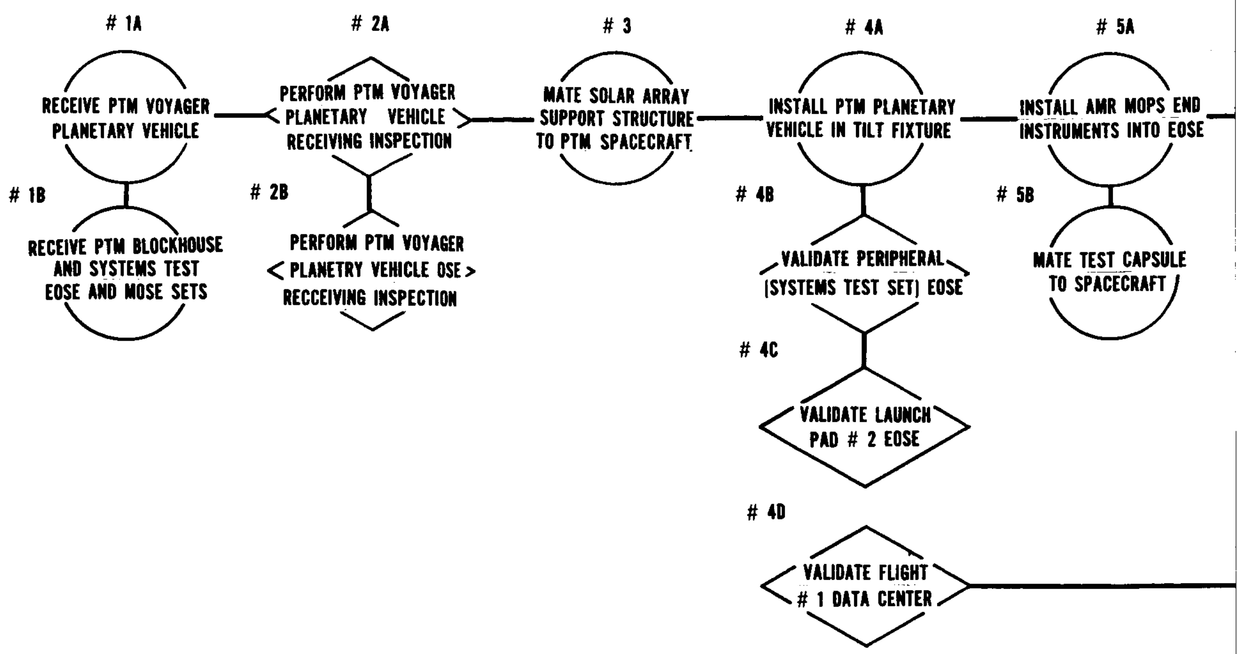
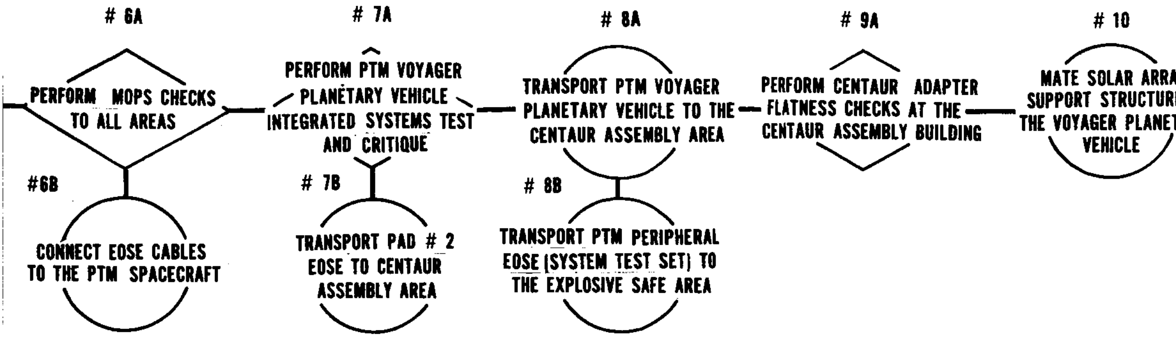
Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
28	<u>Perform Flight Spacecraft Shipping Preparations</u> The spacecraft shipping preparations will include both the spacecraft and its system test set EOSE. The solar arrays support structure and equipment mounted on the array structure will be removed from the spacecraft for shipment. The spacecraft, arrays and array support structure will be placed in shipping containers and purged with dry nitrogen.	Slings, handling fixtures, shipping containers, purging equipment	Procedure	Crane with hook height of ______
29	<u>Ship Flight Spacecraft</u>	Slings, handling fixtures, purging equipment	Procedure	Crane with hook height of ______

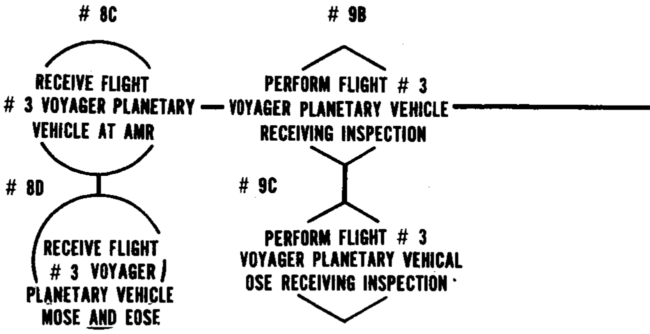


1A
RECEIVE PTM VOYAGER PLANETARY VEHICLE
1B
RECEIVE PTM BLOCKHOUSE AND SYSTEMS TEST EOSE AND MOSE SETS
2A
PERFORM PTM VOYAGER PLANETARY VEHICLE RECEIVING INSPECTION
2B
PERFORM PTM VOYAGER PLANETRY VEHICLE OSE RECCEIVING INSPECTION
3
MATE SOLAR ARRAY SUPPORT STRUCTURE TO PTM SPACECRAFT
4A
INSTALL PTM PLANETARY VEHICLE IN TILT FIXTURE
4B
VALIDATE PERIPHERAL (SYSTEMS TEST SET) EOSE
4C
VALIDATE LAUNCH PAD # 2 EOSE
4D
VALIDATE FLIGHT # 1 DATA CENTER
5A
INSTALL AMR MOPS END INSTRUMENTS INTO EOSE
5B
MATE TEST CAPSULE TO SPACECRAFT

0

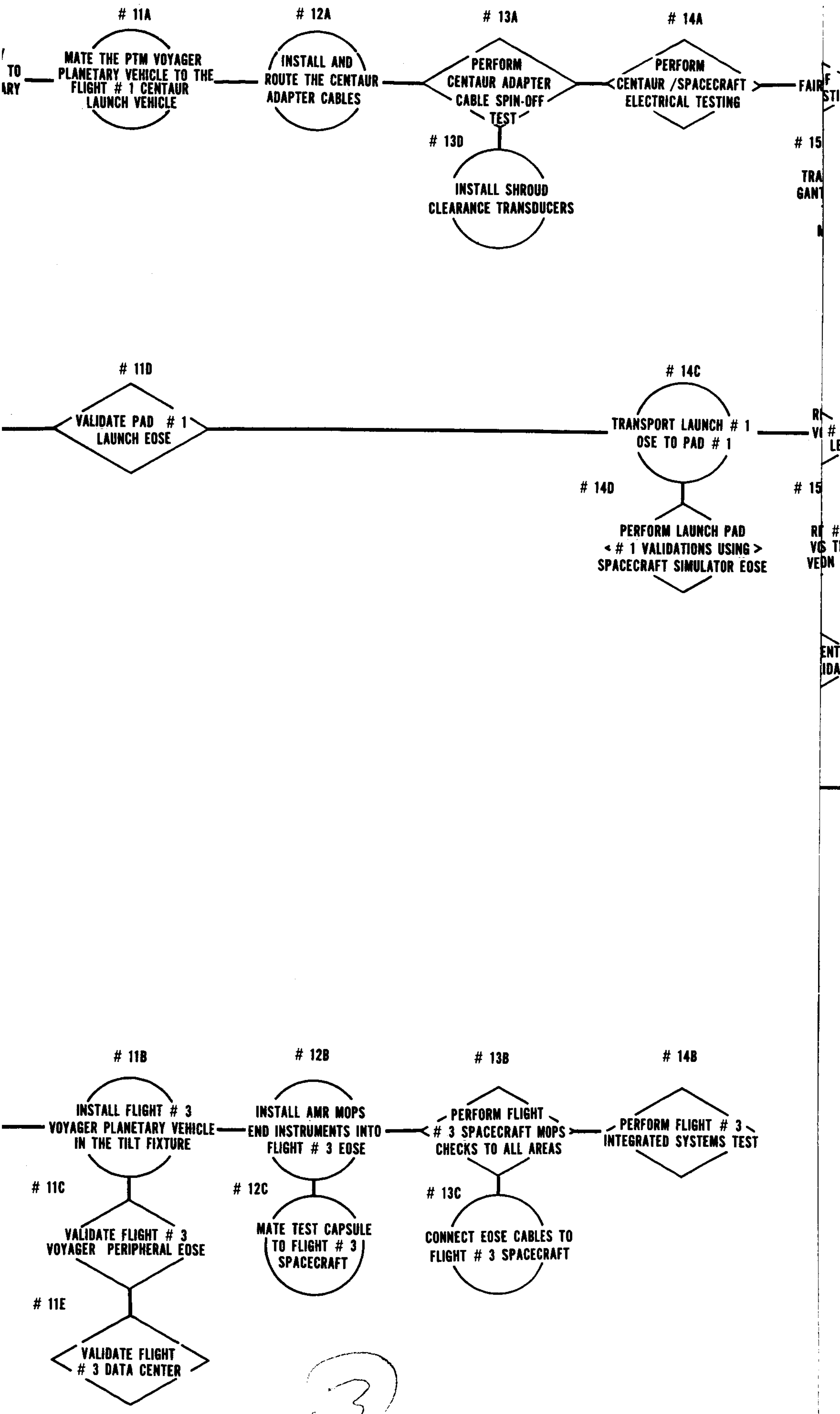


6A
PERFORM MOPS CHECKS TO ALL AREAS
#6B
CONNECT EOSE CABLES TO THE PTM SPACECRAFT
7A
PERFORM PTM VOYAGER PLANETARY VEHICLE INTEGRATED SYSTEMS TEST AND CRITIQUE
7B
TRANSPORT PAD # 2 EOSE TO CENTAUR ASSEMBLY AREA
8A
TRANSPORT PTM VOYAGER PLANETARY VEHICLE TO THE CENTAUR ASSEMBLY AREA
8B
TRANSPORT PTM PERIPHERAL EOSE (SYSTEM TEST SET) TO THE EXPLOSIVE SAFE AREA
9A
PERFORM CENTAUR ADAPTER FLATNESS CHECKS AT THE CENTAUR ASSEMBLY BUILDING
10
MATE SOLAR ARRA
SUPPORT STRUCTUR
THE VOYAGER PLANET
VEHICLE



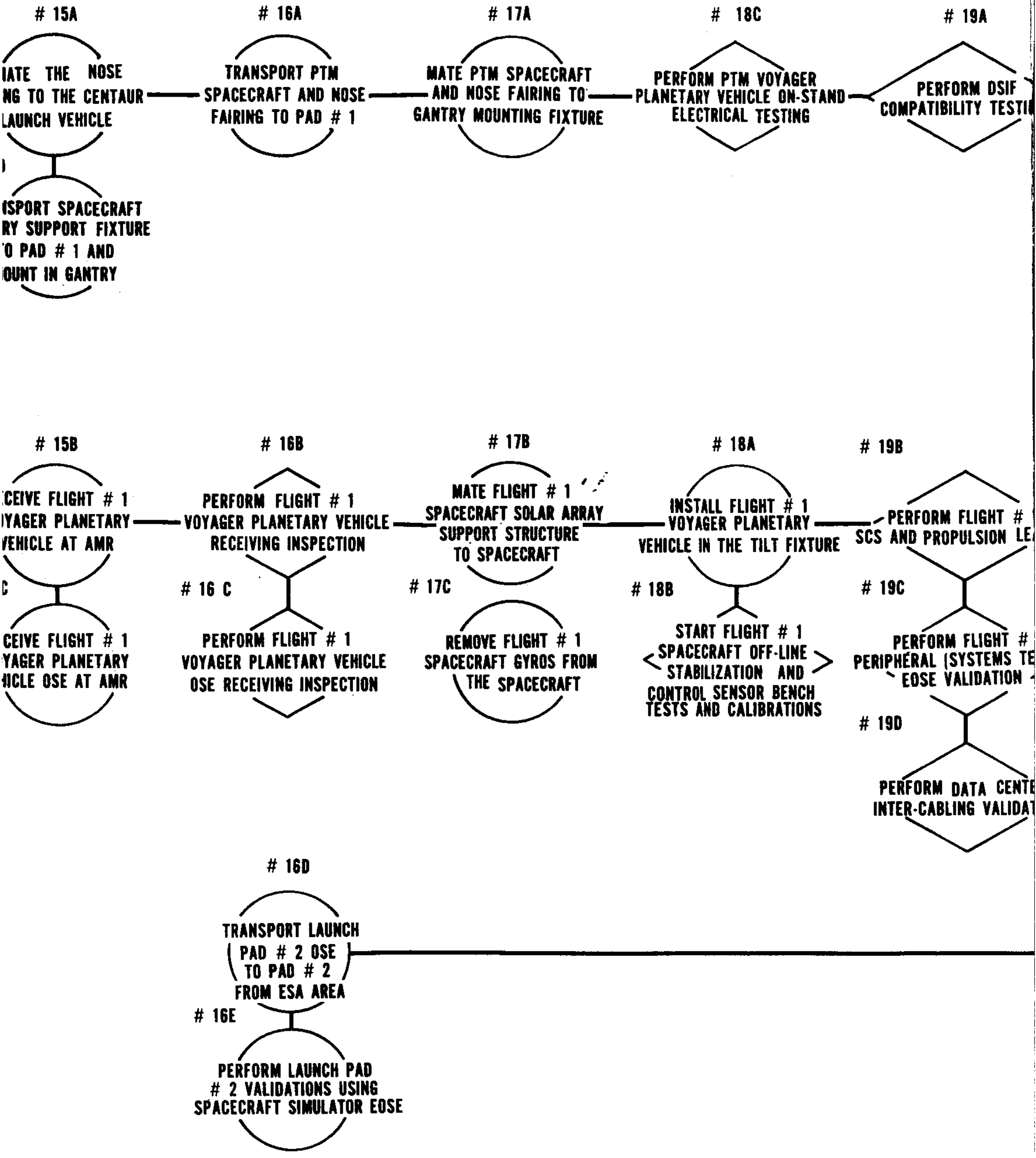
8C
RECEIVE FLIGHT # 3 VOYAGER PLANETARY VEHICLE AT AMR
8D
RECEIVE FLIGHT # 3 VOYAGER PLANETARY VEHICLE MOSE AND EOSE
9B
PERFORM FLIGHT # 3 VOYAGER PLANETARY VEHICLE RECEIVING INSPECTION
9C
PERFORM FLIGHT # 3 VOYAGER PLANETARY VEHICAL OSE RECEIVING INSPECTION

2



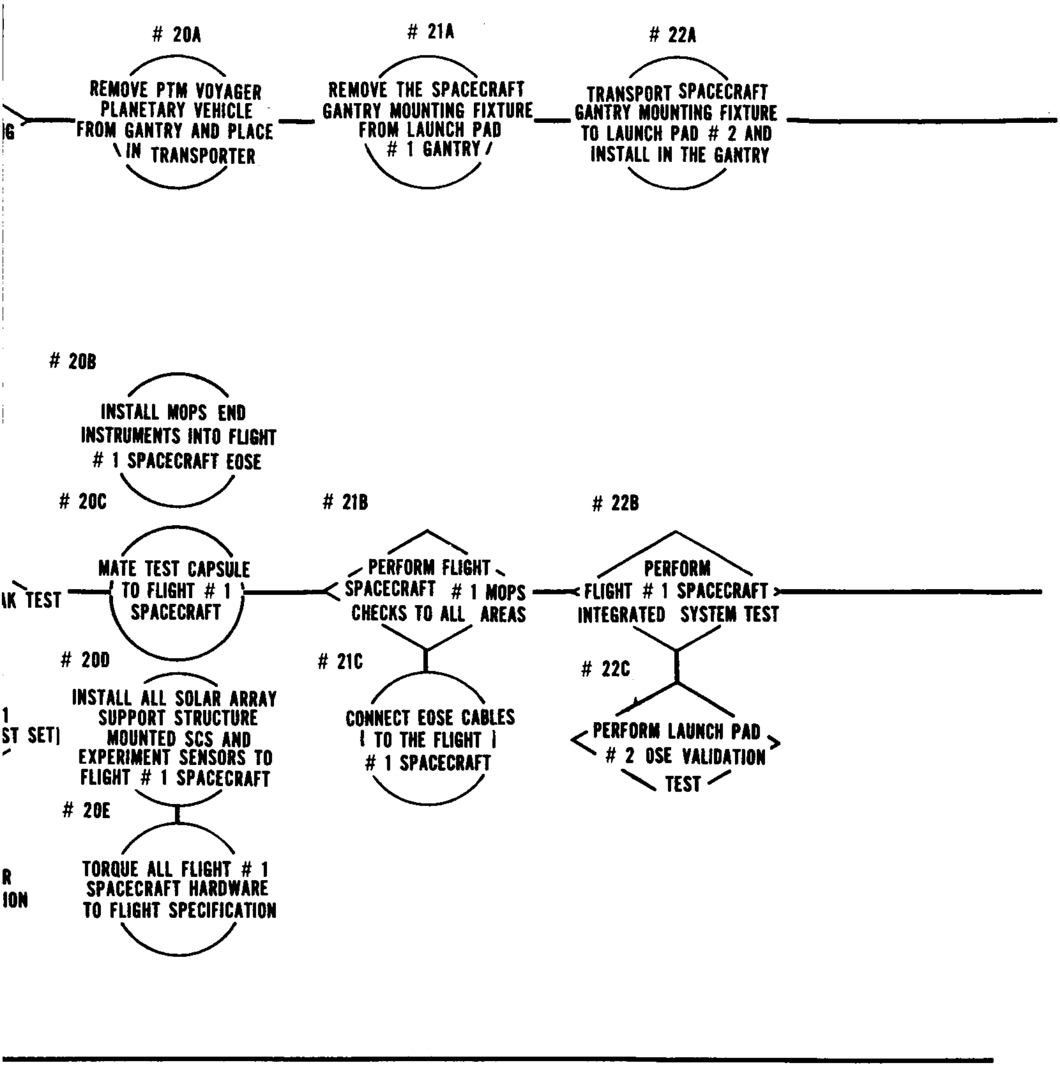
TO
ARY
11A
MATE THE PTM VOYAGER PLANETARY VEHICLE TO THE FLIGHT # 1 CENTAUR LAUNCH VEHICLE
12A
INSTALL AND ROUTE THE CENTAUR ADAPTER CABLES
13A
PERFORM CENTAUR ADAPTER CABLE SPIN-OFF TEST
13D
INSTALL SHROUD CLEARANCE TRANSDUCERS
14A
PERFORM CENTAUR /SPACECRAFT ELECTRICAL TESTING
11D
VALIDATE PAD # 1 LAUNCH EOSE
14C
TRANSPORT LAUNCH # 1 OSE TO PAD # 1
14D
PERFORM LAUNCH PAD # 1 VALIDATIONS USING SPACECRAFT SIMULATOR EOSE
11B
INSTALL FLIGHT # 3 VOYAGER PLANETARY VEHICLE IN THE TILT FIXTURE
11C
VALIDATE FLIGHT # 3 VOYAGER PERIPHERAL EOSE
11E
VALIDATE FLIGHT # 3 DATA CENTER
12B
INSTALL AMR MOPS END INSTRUMENTS INTO FLIGHT # 3 EOSE
12C
MATE TEST CAPSULE TO FLIGHT # 3 SPACECRAFT
13B
PERFORM FLIGHT # 3 SPACECRAFT MOPS CHECKS TO ALL AREAS
13C
CONNECT EOSE CABLES TO FLIGHT # 3 SPACECRAFT
14B
PERFORM FLIGHT # 3 INTEGRATED SYSTEMS TEST

3



15A
IATE THE NOSE
NG TO THE CENTAUR
LAUNCH VEHICLE
ISPORT SPACECRAFT
RY SUPPORT FIXTURE
'O PAD # 1 AND
OUNT IN GANTRY
16A
TRANSPORT PTM
SPACECRAFT AND NOSE
FAIRING TO PAD # 1
17A
MATE PTM SPACECRAFT
AND NOSE FAIRING TO
GANTRY MOUNTING FIXTURE
18C
PERFORM PTM VOYAGER
PLANETARY VEHICLE ON-STAND
ELECTRICAL TESTING
19A
PERFORM DSIF
COMPATIBILITY TESTI
15B
CEIVE FLIGHT # 1
YAGER PLANETARY
VEHICLE AT AMR
CEIVE FLIGHT # 1
YAGER PLANETARY
HICLE OSE AT AMR
16B
PERFORM FLIGHT # 1
VOYAGER PLANETARY VEHICLE
RECEIVING INSPECTION
16 C
PERFORM FLIGHT # 1
VOYAGER PLANETARY VEHICLE
OSE RECEIVING INSPECTION
17B
MATE FLIGHT # 1
SPACECRAFT SOLAR ARRAY
SUPPORT STRUCTURE
TO SPACECRAFT
17C
REMOVE FLIGHT # 1
SPACECRAFT GYROS FROM
THE SPACECRAFT
18A
INSTALL FLIGHT # 1
VOYAGER PLANETARY
VEHICLE IN THE TILT FIXTURE
18B
START FLIGHT # 1
SPACECRAFT OFF-LINE
STABILIZATION AND
CONTROL SENSOR BENCH
TESTS AND CALIBRATIONS
19B
PERFORM FLIGHT #
SCS AND PROPULSION LE
19C
PERFORM FLIGHT #
PERIPHERAL SYSTEMS TE
EOSE VALIDATION
19D
PERFORM DATA CENTE
INTER-CABLING VALIDAT
16D
TRANSPORT LAUNCH
PAD # 2 OSE
TO PAD # 2
FROM ESA AREA
16E
PERFORM LAUNCH PAD
2 VALIDATIONS USING
SPACECRAFT SIMULATOR EOSE

4



1971 VOYAGER LAUNCH OPERATIONS

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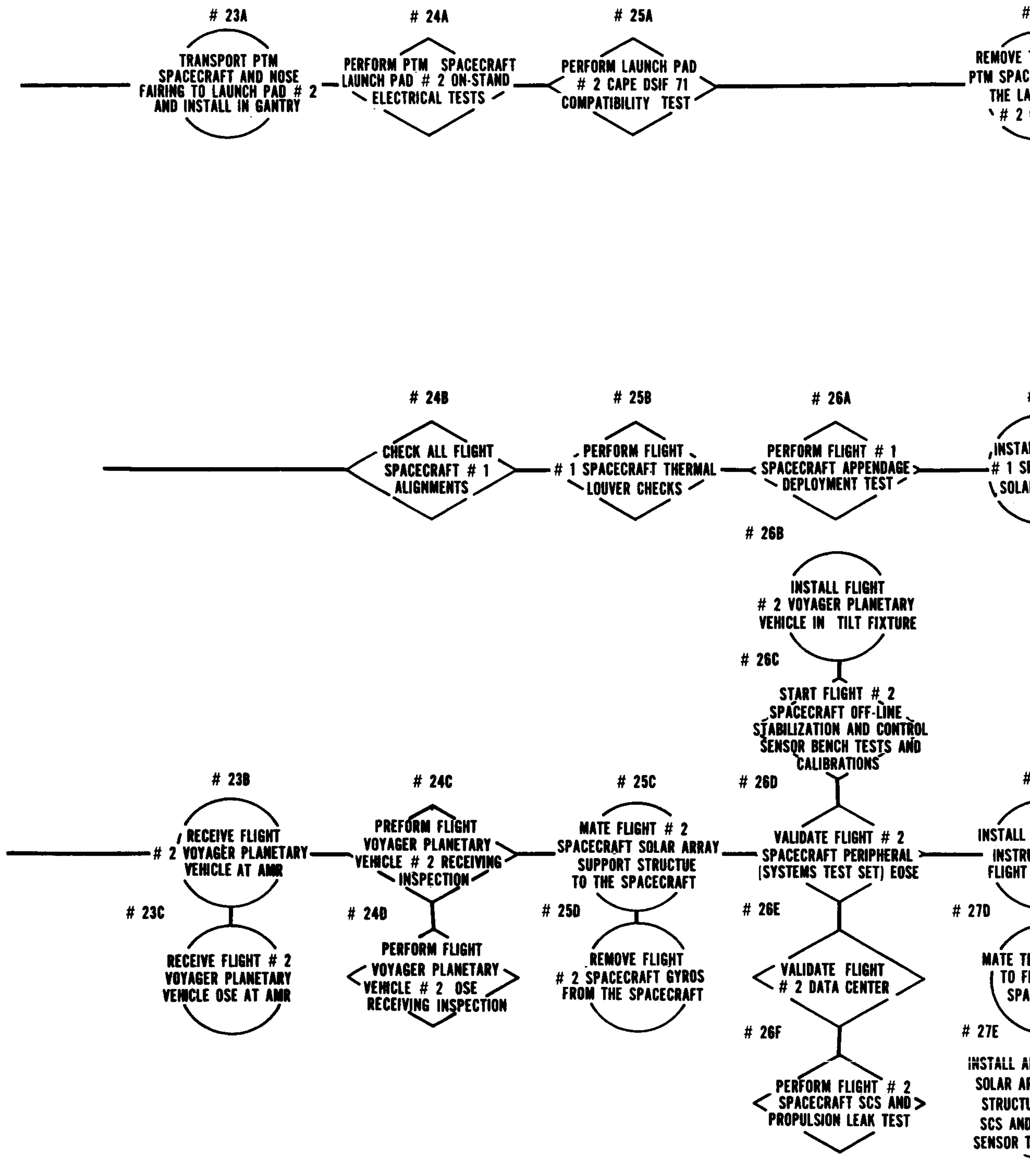
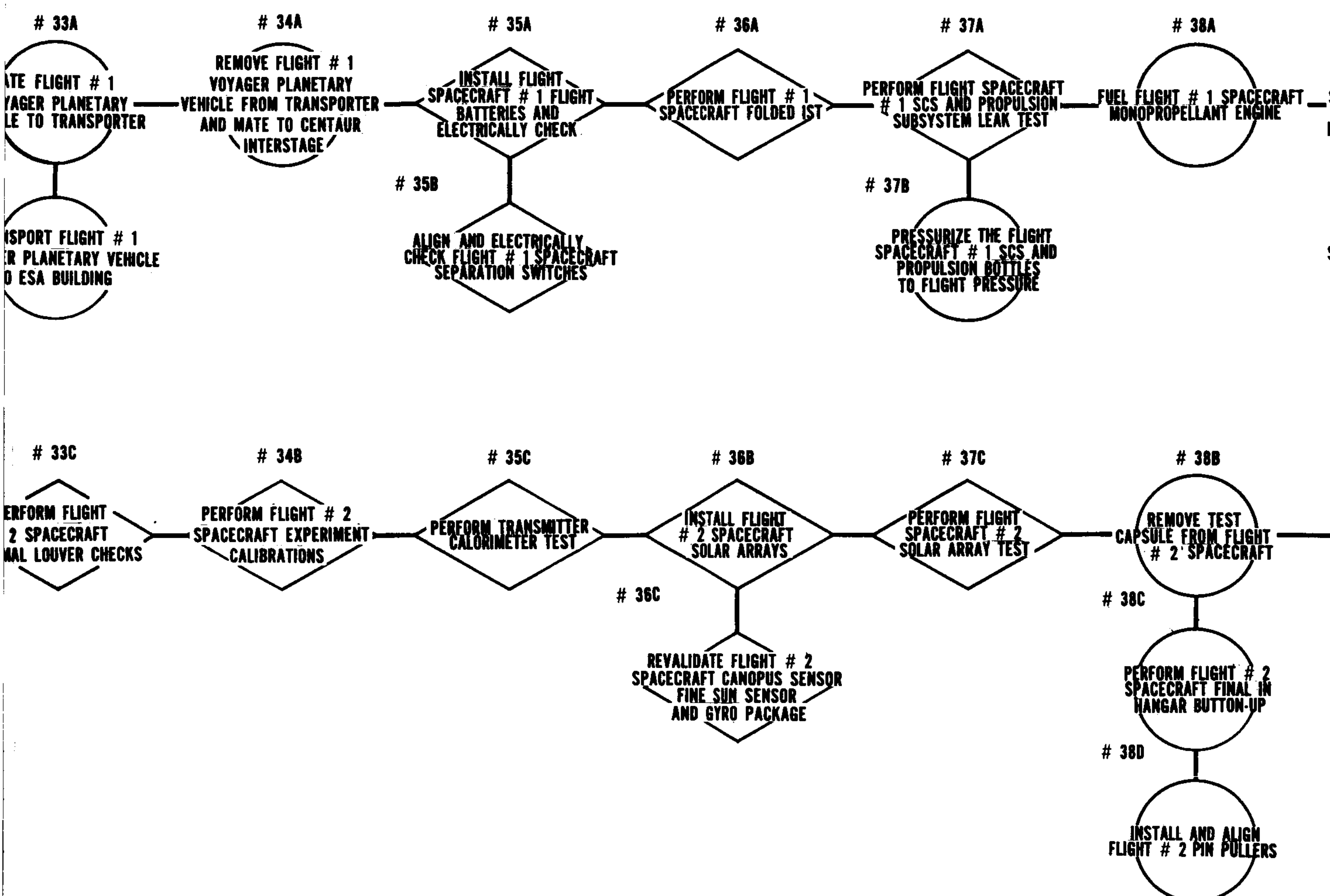


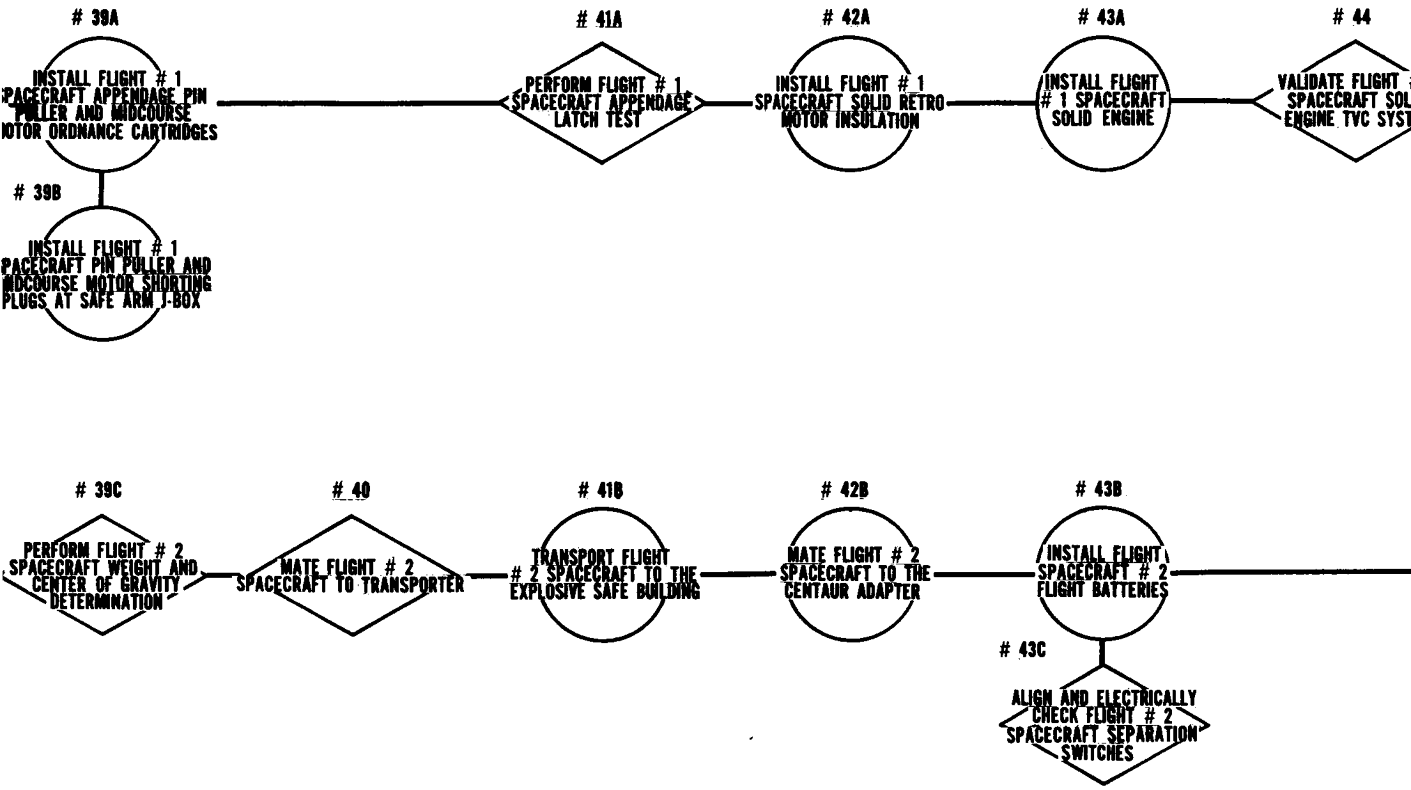
23A
TRANSPORT PTM SPACECRAFT AND NOSE FAIRING TO LAUNCH PAD # 2 AND INSTALL IN GANTRY
24A
PERFORM PTM SPACECRAFT LAUNCH PAD # 2 ON-STAND ELECTRICAL TESTS
25A
PERFORM LAUNCH PAD # 2 CAPE DSIF 71 COMPATIBILITY TEST
24B
CHECK ALL FLIGHT SPACECRAFT # 1 ALIGNMENTS
25B
PERFORM FLIGHT # 1 SPACECRAFT THERMAL LOUVER CHECKS
26A
PERFORM FLIGHT # 1 SPACECRAFT APPENDAGE DEPLOYMENT TEST
26B
INSTALL FLIGHT # 2 VOYAGER PLANETARY VEHICLE IN TILT FIXTURE
26C
START FLIGHT # 2 SPACECRAFT OFF-LINE STABILIZATION AND CONTROL SENSOR BENCH TESTS AND CALIBRATIONS
23B
RECEIVE FLIGHT # 2 VOYAGER PLANETARY VEHICLE AT AMR
23C
RECEIVE FLIGHT # 2 VOYAGER PLANETARY VEHICLE OSE AT AMR
24C
PREFORM FLIGHT VOYAGER PLANETARY VEHICLE # 2 RECEIVING INSPECTION
24D
PERFORM FLIGHT VOYAGER PLANETARY VEHICLE # 2 OSE RECEIVING INSPECTION
25C
MATE FLIGHT # 2 SPACECRAFT SOLAR ARRAY SUPPORT STRUCTUE TO THE SPACECRAFT
25D
REMOVE FLIGHT # 2 SPACECRAFT GYROS FROM THE SPACECRAFT
26D
VALIDATE FLIGHT # 2 SPACECRAFT PERIPHERAL (SYSTEMS TEST SET) EOSE
26E
VALIDATE FLIGHT # 2 DATA CENTER
26F
PERFORM FLIGHT # 2 SPACECRAFT SCS AND PROPULSION LEAK TEST
27D
27E

28A
REMOVE THE SPACECRAFT GANTRY SUPPORT FIXTURE FROM LAUNCH PAD # 2 GANTRY
29A
TRANSPORT THE SPACECRAFT GANTRY SUPPORT FIXTURE TO THE ESA AREA
30B
VALIDATE ESA TEST COMPLEX USING PTM SPACECRAFT
28D
TRANSPORT PTM SPACECRAFT TO ESA AREA
27B
28B
PERFORM FLIGHT SPACECRAFT # 1 EXPERIMENT CALIBRATIONS
29B
PERFORM FLIGHT # 1 SPACECRAFT TRANSMITTER CALORIMETER TEST
30A
PERFORM FLIGHT # 1 SPACECRAFT SOLAR ARRAY TEST
31A
REMOVE TEST CAPSULE FROM FLIGHT # 1 SPACECRAFT
32A
PERFORM FLIGHT # 1 SPACECRAFT WEIGHT AND CENTER OF GRAVITY DETERMINATION
31B
PERFORM FINAL FLIGHT # 1 IN HANGAR BUTTON-UP
33B
31C
INSTALL AND ALIGN FLIGHT # 1 APPENDAGE PIN PULLERS
27C
28C
CONNECT EOSE CABLES TO FLIGHT # 2 SPACECRAFT
29C
PERFORM FLIGHT SPACECRAFT # 2 INTEGRATED SYSTEMS TEST
31D
CHECK ALL FLIGHT # 2 SPACECRAFT ALIGNMENTS
32B
PERFORM FLIGHT # 2 SPACECRAFT DEPLOYMENT TEST
28E
TORQUE ALL FLIGHT # 2 SPACECRAFT HARDWARE TO FLIGHT SPECIFICATION

2

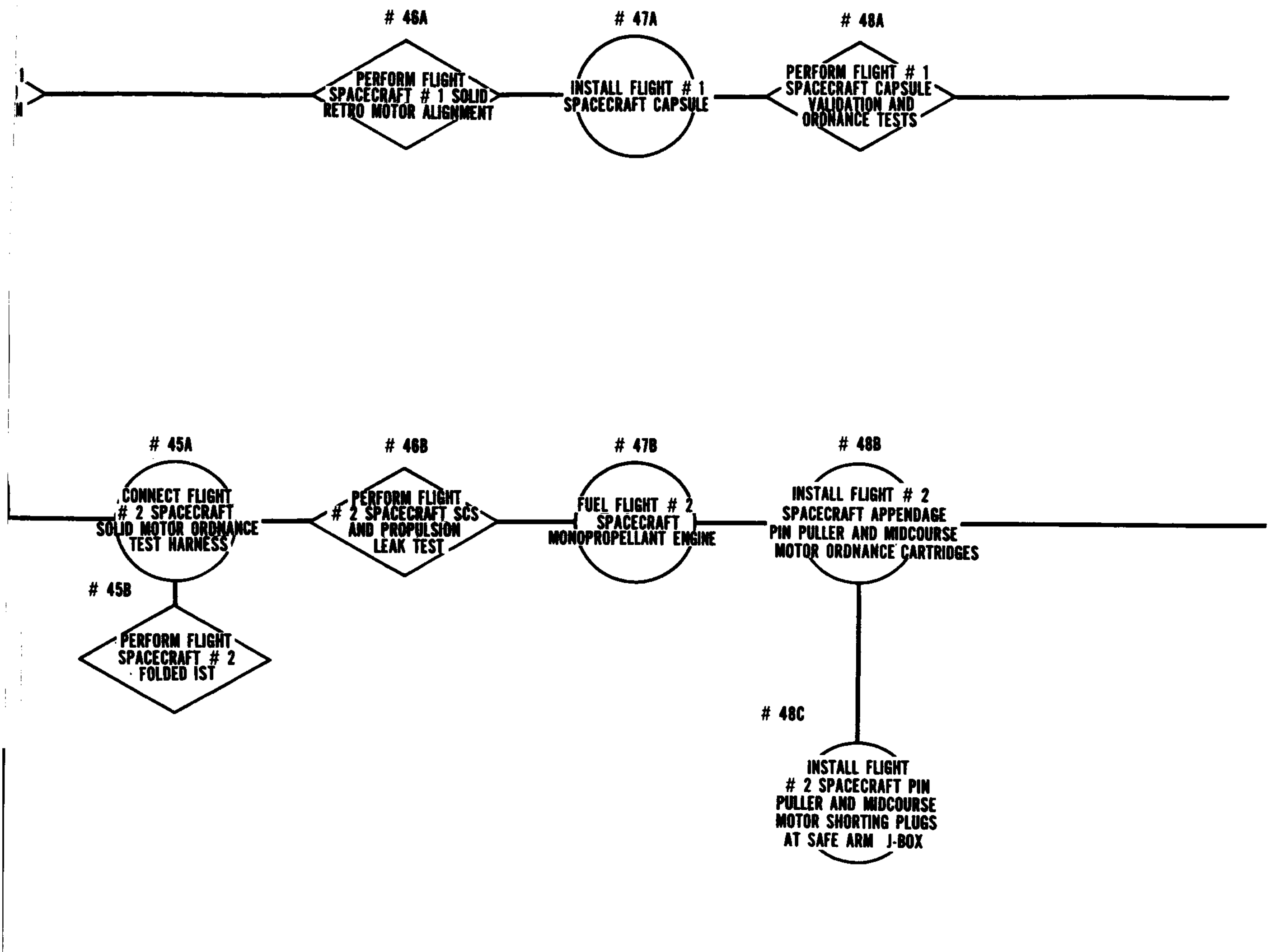


33A
ATE FLIGHT # 1
YAGER PLANETARY
LE TO TRANSPORTER
ISPORT FLIGHT # 1
R PLANETARY VEHICLE
D ESA BUILDING
34A
REMOVE FLIGHT # 1
VOYAGER PLANETARY
VEHICLE FROM TRANSPORTER
AND MATE TO CENTAUR
INTERSTAGE
35A
INSTALL FLIGHT
SPACECRAFT # 1 FLIGHT
BATTERIES AND
ELECTRICALLY CHECK
35B
ALIGN AND ELECTRICALLY
CHECK FLIGHT # 1 SPACECRAFT
SEPARATION SWITCHES
36A
PERFORM FLIGHT # 1
SPACECRAFT FOLDED IST
37A
PERFORM FLIGHT SPACECRAFT
1 SCS AND PROPULSION
SUBSYSTEM LEAK TEST
37B
PRESSURIZE THE FLIGHT
SPACECRAFT # 1 SCS AND
PROPULSION BOTTLES
TO FLIGHT PRESSURE
38A
FUEL FLIGHT # 1 SPACECRAFT
MONOPROPELLANT ENGINE
33C
ERFORM FLIGHT
2 SPACECRAFT
MAL LOUVER CHECKS
34B
PERFORM FLIGHT # 2
SPACECRAFT EXPERIMENT
CALIBRATIONS
35C
PERFORM TRANSMITTER
CALORIMETER TEST
36B
INSTALL FLIGHT
2 SPACECRAFT
SOLAR ARRAYS
36C
REVALIDATE FLIGHT # 2
SPACECRAFT CANOPUS SENSOR
FINE SUN SENSOR
AND GYRO PACKAGE
37C
PERFORM FLIGHT
SPACECRAFT # 2
SOLAR ARRAY TEST
38B
REMOVE TEST
CAPSULE FROM FLIGHT
2 SPACECRAFT
38C
PERFORM FLIGHT # 2
SPACECRAFT FINAL IN
HANGAR BUTTON-UP
38D
INSTALL AND ALIGN
FLIGHT # 2 PIN PULLERS



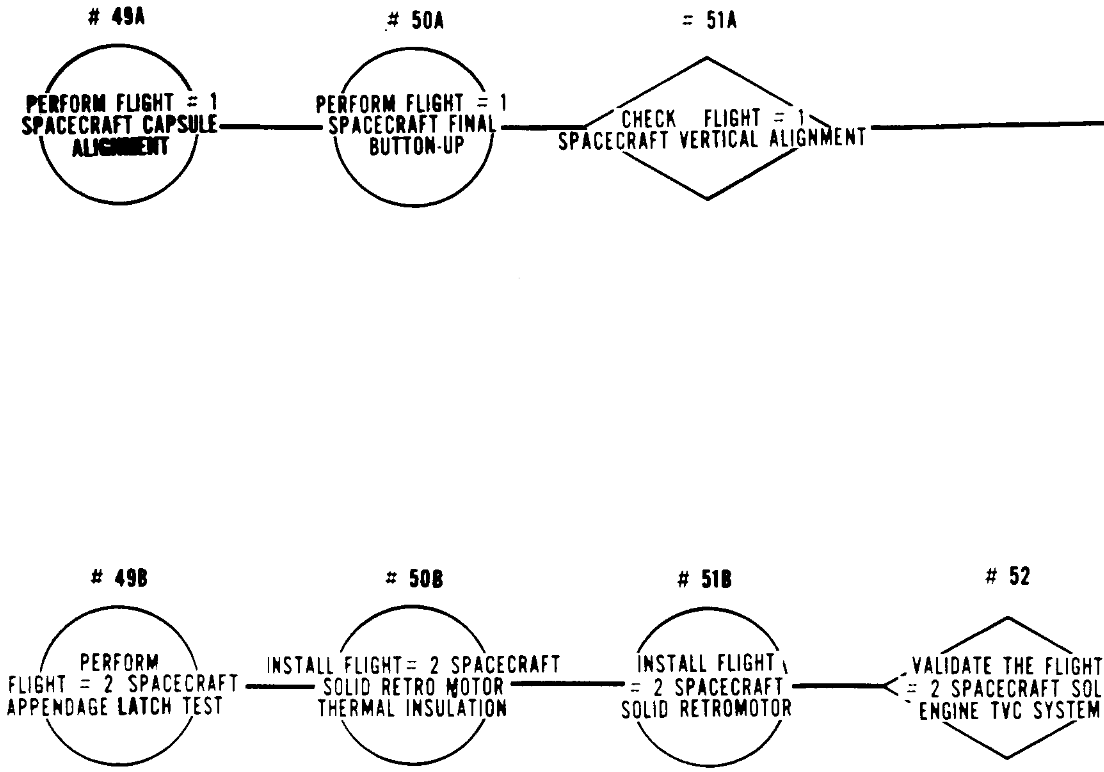
39A
INSTALL FLIGHT # 1 SPACECRAFT APPENDAGE PIN PULLER AND MIDCOURSE MOTOR ORDNANCE CARTRIDGES
39B
INSTALL FLIGHT # 1 SPACECRAFT PIN PULLER AND MIDCOURSE MOTOR SHORTING PLUGS AT SAFE ARM J-BOX
41A
PERFORM FLIGHT # 1 SPACECRAFT APPENDAGE LATCH TEST
42A
INSTALL FLIGHT # 1 SPACECRAFT SOLID RETRO MOTOR INSULATION
43A
INSTALL FLIGHT # 1 SPACECRAFT SOLID ENGINE
44
VALIDATE FLIGHT
SPACECRAFT SOL
ENGINE TVC SYST
39C
PERFORM FLIGHT # 2 SPACECRAFT WEIGHT AND CENTER OF GRAVITY DETERMINATION
40
MATE FLIGHT # 2 SPACECRAFT TO TRANSPORTER
41B
TRANSPORT FLIGHT # 2 SPACECRAFT TO THE EXPLOSIVE SAFE BUILDING
42B
MATE FLIGHT # 2 SPACECRAFT TO THE CENTAUR ADAPTER
43B
INSTALL FLIGHT SPACECRAFT # 2 FLIGHT BATTERIES
43C
ALIGN AND ELECTRICALLY CHECK FLIGHT # 2 SPACECRAFT SEPARATION SWITCHES

4

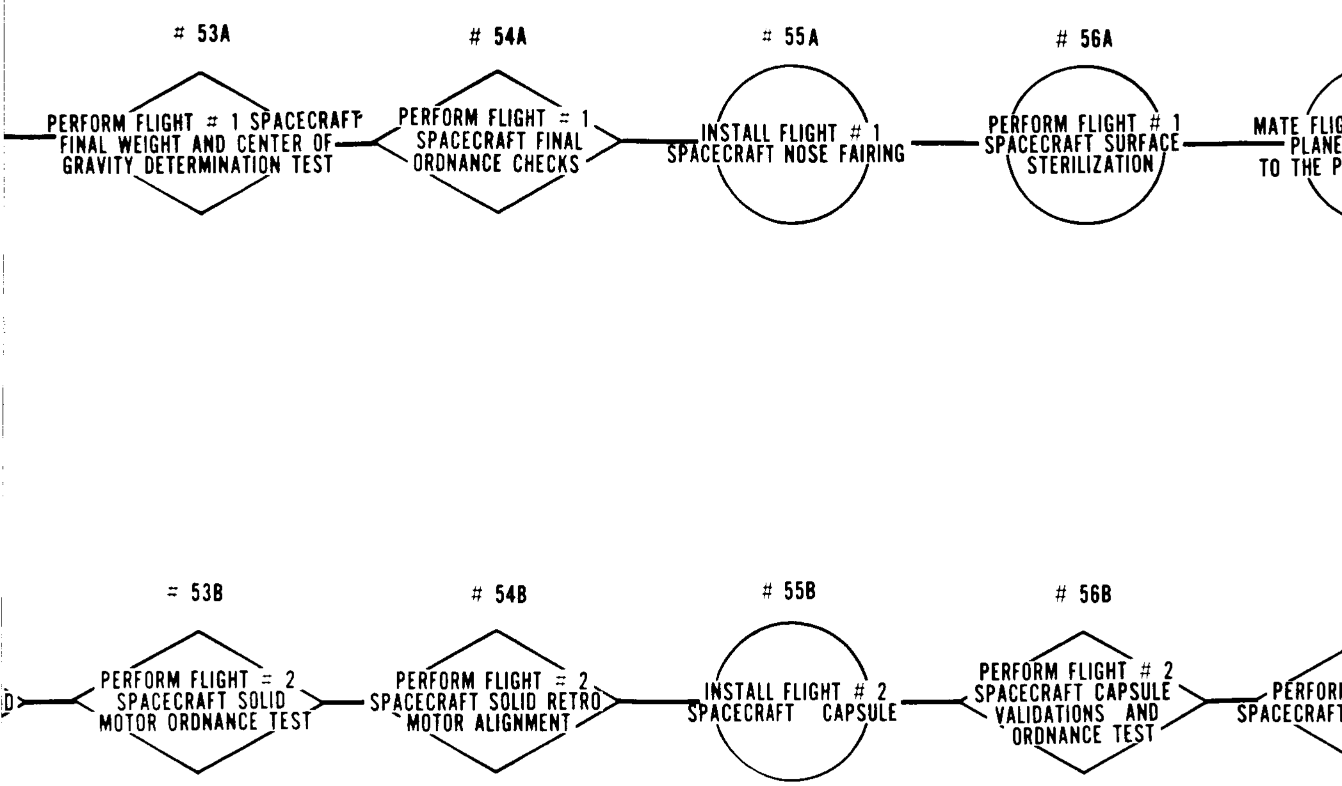


1971 VOYAGER LAUNCH OPERATIONS

5

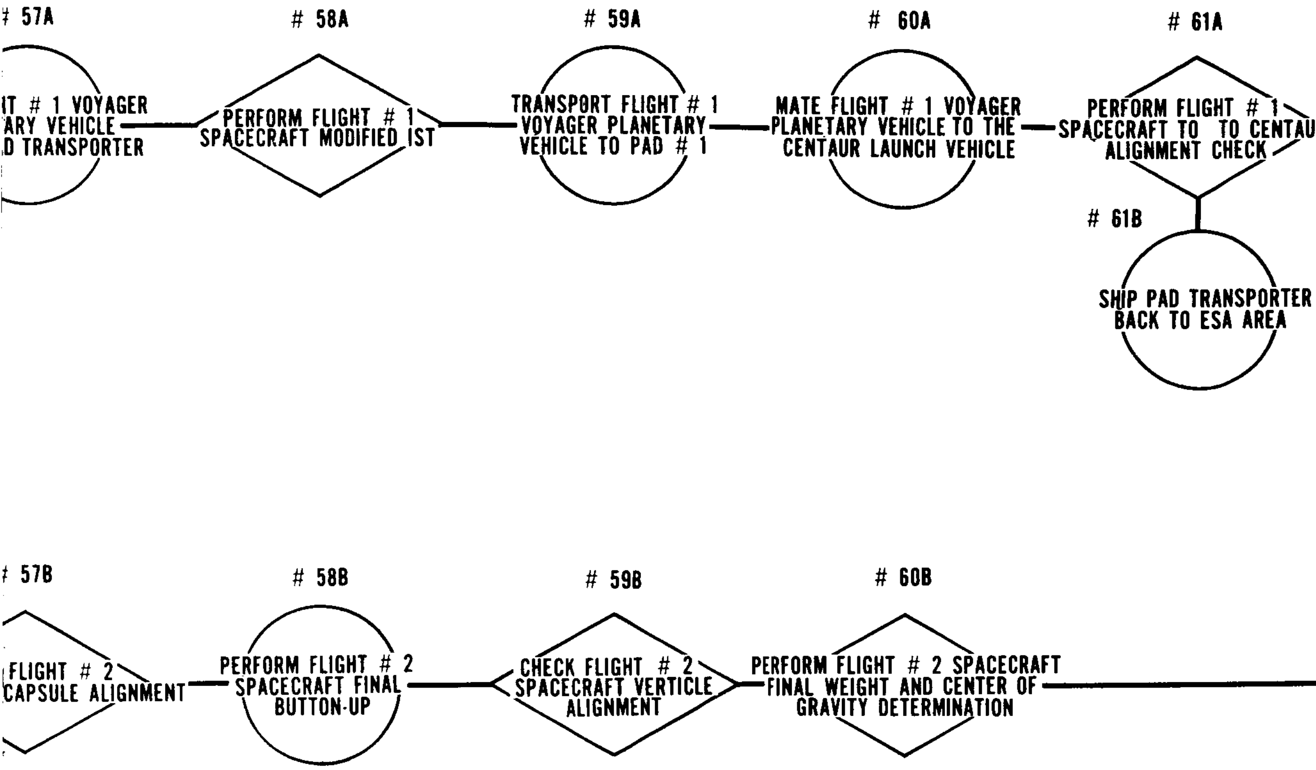


49A
PERFORM FLIGHT = 1 SPACECRAFT CAPSULE ALIGNMENT
50A
PERFORM FLIGHT = 1 SPACECRAFT FINAL BUTTON-UP
= 51A
CHECK FLIGHT = 1 SPACECRAFT VERTICAL ALIGNMENT
49B
PERFORM FLIGHT = 2 SPACECRAFT APPENDAGE LATCH TEST
= 50B
INSTALL FLIGHT = 2 SPACECRAFT SOLID RETRO MOTOR THERMAL INSULATION
51B
INSTALL FLIGHT = 2 SPACECRAFT SOLID RETROMOTOR
52
VALIDATE THE FLIGHT = 2 SPACECRAFT SOL
ENGINE TVC SYSTEM



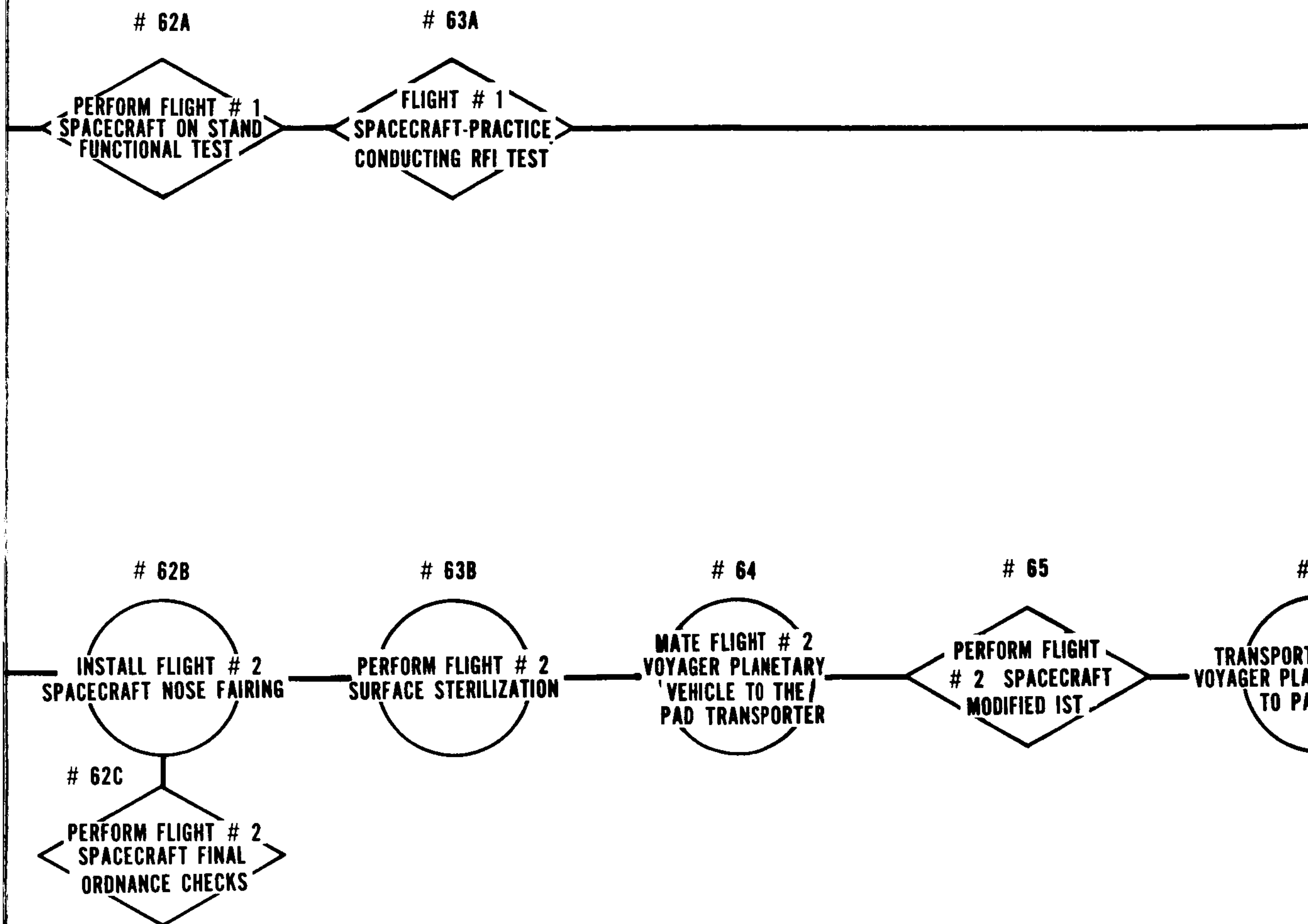
53A
PERFORM FLIGHT # 1 SPACECRAFT FINAL WEIGHT AND CENTER OF GRAVITY DETERMINATION TEST
54A
PERFORM FLIGHT # 1 SPACECRAFT FINAL ORDNANCE CHECKS
55A
INSTALL FLIGHT # 1 SPACECRAFT NOSE FAIRING
56A
PERFORM FLIGHT # 1 SPACECRAFT SURFACE STERILIZATION
53B
PERFORM FLIGHT # 2 SPACECRAFT SOLID MOTOR ORDNANCE TEST
54B
PERFORM FLIGHT # 2 SPACECRAFT SOLID RETRO MOTOR ALIGNMENT
55B
INSTALL FLIGHT # 2 SPACECRAFT CAPSULE
56B
PERFORM FLIGHT # 2 SPACECRAFT CAPSULE VALIDATIONS AND ORDNANCE TEST

2



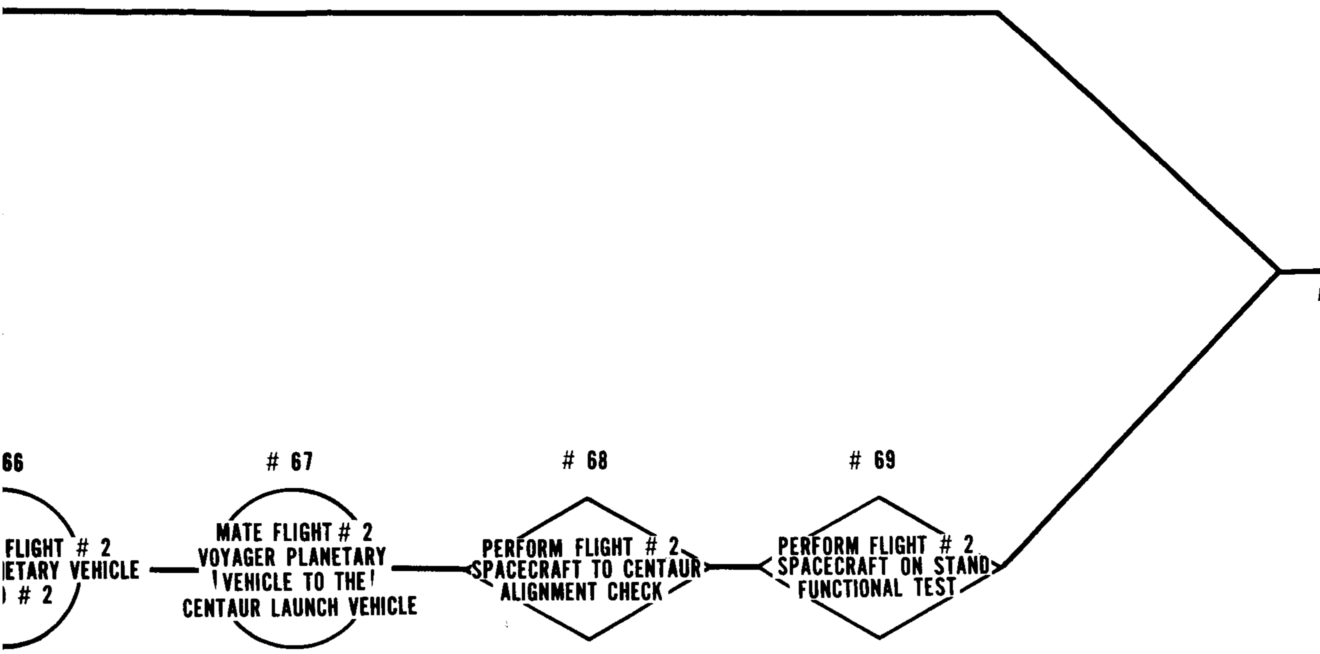
57A
IT # 1 VOYAGER
ARY VEHICLE
D TRANSPORTER
58A
PERFORM FLIGHT # 1
SPACECRAFT MODIFIED IST
59A
TRANSPORT FLIGHT # 1
VOYAGER PLANETARY
VEHICLE TO PAD # 1
60A
MATE FLIGHT # 1 VOYAGER
PLANETARY VEHICLE TO THE
CENTAUR LAUNCH VEHICLE
61A
PERFORM FLIGHT # 1
SPACECRAFT TO TO CENTAUR
ALIGNMENT CHECK
61B
SHIP PAD TRANSPORTER
BACK TO ESA AREA
57B
FLIGHT # 2
CAPSULE ALIGNMENT
58B
PERFORM FLIGHT # 2
SPACECRAFT FINAL
BUTTON-UP
59B
CHECK FLIGHT # 2
SPACECRAFT VERTICLE
ALIGNMENT
60B
PERFORM FLIGHT # 2 SPACECRAFT
FINAL WEIGHT AND CENTER OF
GRAVITY DETERMINATION

3

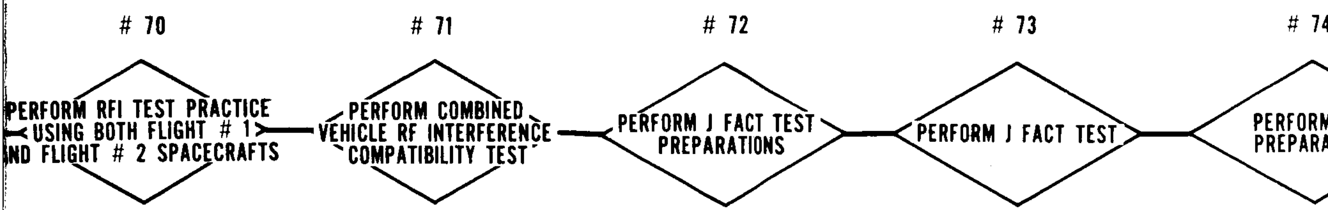


62A
PERFORM FLIGHT # 1 SPACECRAFT ON STAND FUNCTIONAL TEST
63A
FLIGHT # 1 SPACECRAFT-PRACTICE CONDUCTING RFI TEST
62B
INSTALL FLIGHT # 2 SPACECRAFT NOSE FAIRING
62C
PERFORM FLIGHT # 2 SPACECRAFT FINAL ORDNANCE CHECKS
63B
PERFORM FLIGHT # 2 SURFACE STERILIZATION
64
MATE FLIGHT # 2 VOYAGER PLANETARY VEHICLE TO THE PAD TRANSPORTER
65
PERFORM FLIGHT # 2 SPACECRAFT MODIFIED IST
#
TRANSPORT

4


66
FLIGHT # 2
IETARY VEHICLE
) # 2
67
MATE FLIGHT # 2
VOYAGER PLANETARY
VEHICLE TO THE
CENTAUR LAUNCH VEHICLE
68
PERFORM FLIGHT # 2
SPACECRAFT TO CENTAUR
ALIGNMENT CHECK
69
PERFORM FLIGHT # 2
SPACECRAFT ON STAND
FUNCTIONAL TEST

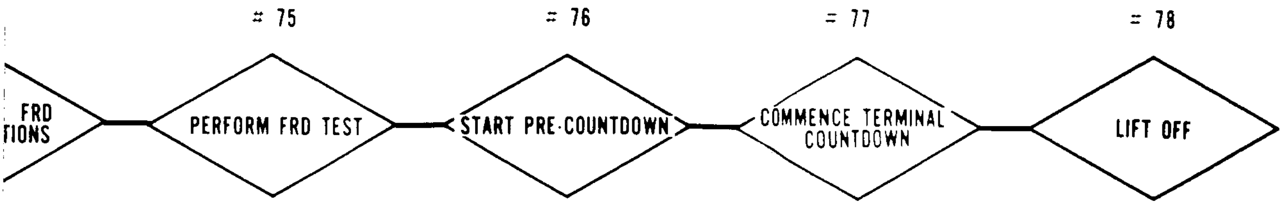
5



70
PERFORM RFI TEST PRACTICE USING BOTH FLIGHT # 1 AND FLIGHT # 2 SPACECRAFTS
71
PERFORM COMBINED VEHICLE RF INTERFERENCE COMPATIBILITY TEST
72
PERFORM J FACT TEST PREPARATIONS
73
PERFORM J FACT TEST
74
PERFORM
PREPARA

1971 VOYAG

6



ER LAUNCH OPERATIONS

Functional Flow
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Operation No	Task Description	Equipment Required	Documentation Required	Special Facilities Required
1A	Receive PTM Voyager Planetary Vehicle	Slings, handling fixtures, transporter	Procedure	None
1B	Receive PTM Blockhouse and Systems Test EOSE and MOSE Both the spacecraft and the OSE will be delivered to the skid strip by air. From the skid strip the spacecraft and EOSE will be transported to the hangar.	Slings, handling fixtures, transporter	Procedure	None
2A	Perform PTM Voyager Planetary Vehicle Receiving Inspection	None	None	None
2B	Perform PTM Voyager Planetary Vehicle OSE Receiving Inspection Receiving inspection will be an inspection for damage that might have been incurred during shipping and handling operations.	None	None	None
3	Mate Solar Array Support Structure to PTM Spacecraft The solar array support structure having been removed from the spacecraft as a part of the shipping preparations will be installed at this time and torqued to flight specification.	Hand tools, torque wrenches	Procedure	None
4A	Install PTM Planetary Vehicle in Tilt Fixture	Hand tools	Procedure	Overhead crane
4B	Validate Peripheral EOSE (Systems Test Set)	Peripheral EOSE validation test set	None	None
4C	Validate Launch Pad No. 2 EOSE	Spacecraft simulator	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
4D	Validate Flight No. 1 Data Center The spacecraft is to be installed in the tilt fixture in preparation for the integrated systems test. Concurrently, the peripheral EOSE and the data center will be validated in preparation for the integrated systems test.	Computer validation tapes, data center validation test set	Procedure	None
5A	Install AMR MOPS End Instruments into EOSE	MOPS end instruments	None	None
5B	Mate Test Capsule to Spacecraft The AMR end instruments will be installed in the EOSE and connected to the AMR MOPS intercommunications system. Concurrently, with this task, the test capsule will be installed to the spacecraft in preparation for the PTM integrated systems test. Last the solar arrays will be mated to the spacecraft and torqued to flight specification.	Hand tools, torque wrenches	Procedure	Overhead crane with hook height of ______
6A	Perform MOPS Checks to All Areas	None	List of MOP channel assignments	None
6B	Connect EOSE Cables to the PTM Spacecraft The AMR intercommunication net will be checked by contacting each Voyager station by using the MOPS end instrument selector switch. Each end instrument in each area will be checked out in this manner. Concurrently, the EOSE cables will be connected to the spacecraft in preparation for the integrated systems test.	None	Procedure	None
7A	Perform PTM Voyager Planetary Vehicle Integrated Systems Test and Critique	Complete set of systems test EOSE	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
7B	Transport Pad No. 2 EOSE to Centaur Assembly Area The Voyager PTM spacecraft integrated systems test will be performed to verify that the PTM spacecraft and all ot its subsystems have successfully survived the shipping and handling operations. Concurrently, the Pad EOSE will be shipped to the Centaur assembly building. The Pad EOSE will be utilized to checkout the Centaur/Voyager spacecraft electrical interfaces.	Slings, EOSE handling fixtures, transporters	Procedure	None
8A	Transport PTM Voyager Planetary Vehicle to the Centaur Assembly Area	Slings, EOSE handling fixtures, transporters	Procedure	Adequate door width to get spacecraft through
8B	Transport PTM Peripheral EOSE (System Test Set) to the Explosive Safe Area	Slings, EOSE handling fixtures, transporters	Procedure	None
8C	Receive Flight No. 3 Voyager Planetary Vehicle at AMR	Slings, spacecraft handling fixture, transporters	Procedure	None
8D	Receive Flight No. 3 Voyager Planetary Vehicle MOSE and EOSE The PTM spacecraft will be shipped to the Centaur assembly area in preparation for the Centaur/Voyager spacecraft electrical interface test. The PTM peripheral EOSE will be shipped to the explosive safe area in preparation for the folded integrated system test. Concurrently, with the above operations, both the Flight No. 3 spacecraft and MOSE and EOSE will be delivered to the skid strip at AMR. Next the spacecraft and associated OSE will be delivered to the spacecraft assembly hangar.	Slings, OSE handling fixtures, transporters	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
9A	Perform Centaur Adapter Flatness Checks at the Centaur Assembly Building	Centaur adapter alignment set	Procedure	None
9B	Perform Flight No. 3 Voyager Planetary Vehicle Receiving Inspection	None	None	None
9C	Perform Flight No. 3 Voyager Planetary Vehicle OSE Receiving Inspection	None	None	None
	The Flight No. 1 and 2 Centaur adapter flatness checks will be performed to ascertain that the Centaur and spacecraft mating surfaces are absolutely flat and level. Concurrently, the Flight No. 1 spacecraft and OSE receiving inspections are performed mainly to ascertain that no damage to the spacecraft or OSE was incurred due to shipping and handling operations.			
10	Mate Solar Array Support Structure to Spacecraft	Hand tools, torque wrenches	Procedure	None
	The solar array support structure having been removed from the spacecraft as part of shipping preparations will be installed.			
11A	Mate the PTM Voyager Planetary Vehicle to the Flight No. 1 Centaur Launch Vehicle	Hand tools, torque wrenches, spacecraft handling fixture	Procedure	Overhead crane with hook height of ______.

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
11B	Install Flight No. 3 Voyager Planetary Vehicle in the Tilt Fixture	Hand tools, torque wrenches, slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of ____.
11C	Validate Flight No. 3 Voyager Planetary Vehicle Peripheral EOSE	Peripheral EOSE validation set	Procedure	None
11D	Validate Pad No. 1 Launch EOSE	Spacecraft simulator	Procedure	None
11E	Validate Flight No. 3 Data Center The PTM Spacecraft will be mated to the first flight Centaur vehicle in preparation for Centaur spacecraft electrical tests. Next, the Flight No. 3 spacecraft will be mated to the tilt fixture in preparation for the Flight No. 3 integrated systems test. Concurrently, the EOSE for Pad No. 2, the Flight No. 3 peripheral EOSE and the Flight No. 3 data center will be validated.	Computer validation tapes, data center validation test set	Procedure	None
12A	Install and Route the Centaur Adapter Cables	Hand tools	Procedure	None
12B	Install AMR MOPS End Instruments into Flight No. 3 EOSE	MOPS end instruments	None	None
12C	Mate Test Capsule to Flight No. 3 Spacecraft The AMR MOPS end instruments will be installed in the EOSE and connected to the AMR MOPS intercommunications system. Concurrently, the test capsule will be mated to the Flight No. 3 spacecraft. Last, the Centaur adapter cables will be installed, routed and clamped in preparation for the Centaur/spacecraft electrical interface tests.	Slings, capsule handling fixture	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
13A	Perform Centaur Adapter Cable Spin-off Test	None	None	None
13B	Perform Flight No. 3 Spacecraft MOPS Checks to all Areas	None	List of MOPS channel assignment	None
13C	Connect EOSE Cables to Flight No. 3 Spacecraft The Centaur adapter cable spin-off test will not be done live; it is merely a mechanical test to ascertain that the adapter cables at the time of separation will fall freely away from the spacecraft.	None	None	None
13D	Install Shroud Clearance Transducers The AMR intercommunications net will be checked by contacting each Voyager station by using the MOPS end instrument selector switch. Each end instrument in each area will be checked out in this manner. Concurrently, the EOSE cables will be connected to the Flight No. 3 Spacecraft. Last, the shroud clearance transducers will be installed on the PTM spacecraft and connected to the electronics in preparation for the shroud interface testing.			
14A	Perform Centaur/Spacecraft Electrical Testing	Launch pad EOSE, voltmeter, oscilloscope	Procedure	Electrical outlets
14B	Perform Flight No. 3 Integrated Systems Test	Complete set of systems test EOSE	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
14C	<u>Transport Launch Pad No. 1 OSE to Pad No. 1</u> The Centaur/spacecraft electrical test will be performed as follows: a. Continuity test the Centaur adapter cabling. b. Connect the adapter cabling to the EOSE and the spacecraft. c. Apply external power to the spacecraft and determine that the adapter cabling line drops are within specification. d. The spacecraft spin-off separation signals from the Centaur will be checked for no-fire conditions and all-fire conditions. e. All other umbilical signal functions will be tested for proper operation through the Centaur. Concurrently, the Flight No. 3 Spacecraft integrated systems test will be performed to verify that the flight spacecraft and all of its subsystems have survived the shipping and handling operations. Last, the Pad No. 1 OSE will be transported to Pad No. 1 in support of the launch complex phases of testing and validated.	Slings, OSE handling fixtures, transporters		
14D	<u>Perform Launch Pad No. 1 Validations Using Spacecraft Simulator EOSE</u> The pad validations are comprised of the following tests: a. Determine primary power line drops between the spacecraft and the blockhouse. b. RF up and down link power loss determinations c. Electrically check all of spacecraft umbilical functions between the spacecraft and blockhouse. d. Check the wideband video pair system between the spacecraft assembly area and the spacecraft.	Capsule simulator, launch pad EOSE	Procedure	All Voyager pad modification completed.

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
15A	Mate the Nose Fairing to the Centaur Launch Vehicle	Hand tools, torque wrenches, slings, shroud handling fixture		Overhead crane with hook height of ________.
15B	Receive Flight No. 1 Voyager Planetary Vehicle at AMR	None	None	None
15C	Receive Flight No. 1 Voyager Planetary Vehicle OSE at AMR	None	None	None
15D	Transport Spacecraft Gantry Support Fixture to Pad No. 1 and Mount in Gantry The nose fairing will be lowered over the PTM spacecraft and mated to the Centaur launch vehicle. The nose fairing interface test is comprised of two parts: shroud clearance determination and RF shroud coupler losses. Concurrently, the Flight No. 1 spacecraft will be received at the AMR skid strip and transported to the spacecraft assembly area. Last, the spacecraft gantry support fixture will be transported to the Pad No. 1 gantry and installed in preparation for the PTM spacecraft on-stand testing.	Slings, handling fixture, transporter	Procedure	Overhead crane with hook height of ______.
16A	Transport PTM Spacecraft and Nose Fairing to Pad No. 1	Slings, handling fixture, transporter	Procedure	Overhead crane with hook height of _______.
16B	Perform Flight No. 1 Voyager Planetary Vehicle Receiving Inspection	None	None	None
16C	Perform Flight No. 1 Voyager Planetary Vehicle OSE Receiving Inspection	None	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
16D	Transport Launch Pad No. 2 OSE to Pad No. 2 From ESA Area The PTM spacecraft and nose fairing will be transported to Pad No. 1. The Pad EOSE will also be transported, installed and validated at Pad No. 2. Concurrently, the Flight No. 1 spacecraft and OSE receiving inspections will take place. The receiving inspections are visual inspections for damage that may have been incurred during shipping and handling operations.			
16E	Perform Launch Pad No. 2 Validations Using Spacecraft Simulator EOSE The pad validations are comprised of the following tests: a. Determine primary power line drops between the spacecraft and the blockhouse. b. RF up and down link power loss determinations c. Electrically check all of spacecraft umbilical functions between the spacecraft and blockhouse. d. Check the wideband video pair system between the spacecraft assembly area and the spacecraft.	Capsule simulator, launch pad EOSE	Procedure	All Voyager pad modification completed
17A	Mate PTM Spacecraft and Nose Fairing to the Gantry Mounting Fixture	Slings, handling fixtures	Procedure	Overhead crane with hook height of ______.
17B	Mate Flight No. 1 Spacecraft Solar Array Support Structure to the Spacecraft	Hand tools, torque wrenches	Procedure	None
17C	Remove Flight No. 1 Spacecraft Gyros from the Spacecraft The PTM spacecraft and nose fairing will be mated to the gantry mounting fixture to support the on-stand testing phase.	Hand tools, torque wrenches	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	Concurrently, the solar array support structure, having been removed from the flight spacecraft, will be installed and torqued to flight specification. Last, the flight gyro package will be removed from the spacecraft to support the final off-line stabilization and control sensor bench tests and calibrations.			
18A	Install Flight No. 1 Voyager Planetary Vehicle in the Tilt Fixture	Slings, handling fixture, torque wrench	None	Overhead crane with hook height of ______.
18B	Start Flight No. 1 Spacecraft Off-Line Stabilization and Control Sensor Bench Tests and Calibrations	SCS bench part, EOSE	Procedure	SCS laboratory
18C	Perform PTM Voyager Planetary Vehicle On-stand Electrical Testing The Flight No. 1 spacecraft will be installed in the tilt fixture to support the remaining tests. Concurrently, the Canopus sensor, the gyro package, and the fine sun sensors will be bench checked and final calibrations will be performed. Note that it is assumed that these items were removed as part of the shipping preparations. The PTM spacecraft on-stand testing phase is comprised of the following tests: a. Determine primary power line drops between the spacecraft and the blockhouse. b. RF up and down link power loss determination. c. Electrically check all of spacecraft umbilical functions between the spacecraft and blockhouse. d. Check the wideband video pair system between the spacecraft assembly area and the spacecraft.	Blockhouse EOSE, data center	Procedure	Wideband video pair system MOPS

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
19A	Perform DSIF Compatibility Testing	Blockhouse EOSE	Procedure	RF clearance
19B	Perform Flight No. 1 SCS and Propulsion Leak Test	SCS and propulsion leak test consoles	Procedure	None
19C	Perform Flight No. 1 Peripheral (Systems Test Set) EOSE Validation	Peripheral EOSE validation sets	Procedure	None
19D	Perform Data Center Inter-cabling Validation While the PTM spacecraft is on-stand, the DSIF 71 station compatibility test will be performed. The following measurements will be taken by the DSIF station: a. Relative RF power measurements b. Frequency measurements c. Modulation index measurements d. Airborne command receiver best lock frequency determination e. Airborne command receiver zero loop stress frequency determination Concurrently, the Flight No. 1 spacecraft SCS and propulsion leak tests will be performed. The leak tests are performed to ascertain that leaks do not exist that may have been caused by the shipping and handling operations. Last, the Flight No. 1 spacecraft OSE and data center intercabling validations will be performed in preparation of the integrated systems test.	Data center intercable validation set	Procedure	Data center intercabling system

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
20A	Remove PTM Voyager Planetary Vehicle from Gantry and Place in Transporter	Slings, handling fixtures	Procedure	Crane service
20B	Install MOPS End Instruments into Flight No. 1 Spacecraft EOSE	None	None	None
20C	Mate Test Capsule to Flight No. 1 Spacecraft	Slings, capsule handling fixture	Procedure	Crane with hook height of ______.
20D	Install All Solar Array Support Structure Mounted SCS and Experiment Sensors to Flight No. 1 Spacecraft	Hand tools, torque wrench	Procedure	None
20E	Torque All Flight No. 1 Spacecraft Hardware to Flight Specification	Torque wrenches	None	None
	The PTM will be removed from the gantry in preparation for launch complex No. 2 testing. Concurrently, the AMR MOPS end instruments will be installed into the flight No. 1 EOSE and connected to the AMR intercommunications system. Last, the capsule will be installed in the Flight No. 1 spacecraft in preparation for the Flight No. 1 integrated system test.			
	The SCS and experiment sensors were originally removed from the spacecraft as part of shipping preparations and will be installed at this time.			
	All spacecraft flight hardware will be torqued to flight specification initiating the button up procedure.			
21A	Remove the Spacecraft Gantry Mounting Fixture from Launch Pad No. 1 Gantry	Slings, handling fixtures	None	Crane with hook height of ______.

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
21B	Perform Flight Spacecraft No. 1 MOPS Checks to all Areas	None	None	None
21C	Connect EOSE Cables to the Flight No. 1 Spacecraft The gantry mounting fixture will be removed from Pad No. 1 gantry in preparation for Pad No. 2 on-stand testing. Concurrently, the AMR intercommunications net will be checked by contacting each Voyager station by using the MOPS end instruments selector switch. Each end instrument in each area will be checked out in this manner. Last, the EOSE cables will be connected to the spacecraft in preparation for the Flight No. 1 integrated system test.	None	Procedure	None
22A	Transport Spacecraft Gantry Mounting Fixture to Launch Pad No. 2 and Install in the Gantry	Transporter	Procedure	None
22B	Perform Flight No. 1 Spacecraft Integrated System Test	Complete compliment of systems test OSE	Procedure	None
22C	Perform Launch Pad No. 2 OSE Validation Tests The spacecraft gantry mounting fixture will be transported to Pad No. 2 and installed in the gantry in preparation for the launch pad on-stand tests. Concurrently, the Flight No. 1 spacecraft integrated system test will be performed to verify that the spacecraft and all of its subsystems have successfully survived the shipping and handling operations.	Spacecraft simulator	Procedure	None
23A	Transport PTM Spacecraft and Nose Fairing to Launch Pad No. 2 and Install in Gantry	Transporter, slings, handling fixture	Procedure	Crane with hook height of ______.

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
23B	Receive Flight No. 2 Voyager Planetary Vehicle at AMR	Transporters	Procedure	None
23C	Receive Flight No. 2 Voyager Planetary Vehicle OSE at AMR The PTM spacecraft will be delivered to Pad No. 2 and mated in the spacecraft gantry fixture in preparation for on-stand testing. Both the Flight No. 2 spacecraft and OSE will be delivered to the AMR skid strip by air. From the skid strip, the spacecraft and OSE will be transported to the spacecraft assembly area.	Transporters	Procedure	None
24A	Perform PTM Spacecraft Launch Pad No. 2 On-stand Electrical Tests	Blockhouse EOSE, data center	Procedure	Range clearance
24B	Check All Flight Spacecraft No. 1 Alignments	Complete compliment of alignment sets	Procedure	None
24C	Perform Flight Voyager Planetary Vehicle No. 2 Receiving Inspection	None	None	None
24D	Perform Flight Voyager Planetary Vehicle No. 2 OSE Receiving Inspection The PTM spacecraft Pad No. 2 on-stand electrical tests will be performed as follows: a. Determine primary power line drops between the PTM spacecraft and blockhouse No. 2. b. RF power up and down link power loss determination. c. Electrically check all of the spacecraft umbilical functions between the spacecraft and the blockhouse. d. Check the wideband video pair system between gantry No. 2 and the spacecraft assembly area.	None	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	All flight spacecraft No. 1 alignments will be checked for shifts due to transportation and handling operations. The flight spacecraft No. 1 fine sun sensors, Canopus sensor, and gyro package will be given a final bench check and calibration in preparation for launch. No. 2 flight spacecraft and OSE receiving inspection will be inspected for damage that might have been incurred during shipping and handling operations.			
25A	Perform Launch Pad No. 2 Cape DSIF 71 Compatibility Test	Blockhouse EOSE, DSIF station	Procedure	Range clearance
25B	Perform Flight No. 1 Spacecraft Thermal Louver Checks	Evaporative liquid	Procedure	None
25C	Mate Flight No. 2 Spacecraft Solar Array Support Structure to the Spacecraft	Hand tools, torque wrenches	Procedure	None
25D	Remove Flight No. 2 Spacecraft Gyros from the Spacecraft The DSIF compatibility test will encompass the following tests: a. Relative power measurements between Pad No. 2 and the DSIF station. b. Engineering model spacecraft down-link frequency measurement. c. Engineering model down-link modulation index measurement. d. Airborne command receiver best lock frequency determination. e. Airborne receiver zero loop stress frequency determination.	Hand tools, torque wrenches, blockhouse EOSE, DSIF station	None	None

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Operation No	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	The Flight No. 1 spacecraft thermal louvers will be tested by stimulating them with a highly evaporative liquid and observing that proper operation exists. Concurrently, the solar array support structure, having been removed from the spacecraft as part of shipping preparations, will be installed. Last, the gyro package will be removed from the spacecraft to support the off-line laboratory final bench checks and calibrations.			
26A	Perform Flight No. 1 Spacecraft Appendage Deployment Test	None	Procedure	None
26B	Install Flight No. 2 Voyager Planetary Vehicle in Tilt Fixture	Hand tools, torque wrenches, slings, handling fixture	Procedure	None
26C	Start Flight No. 2 Spacecraft Off-line Stabilization and Control Sensor Bench Tests and Calibrations	SCS bench part, EOSE	Procedure	SCS laboratory
26D	Validate Flight No. 2 Spacecraft Peripheral (Systems Test Set) EOSE	Peripheral EOSE validation set	Procedure	None
26E	Validate Flight No. 2 Data Center The DSIF compatibility test will encompass the following tests: a. Relative power measurements between Pad No. 2 and the DSIF station. b. PTM spacecraft down-link frequency measurement. c. PTM down-link modulation index measurement. d. Airborne command receiver best lock frequency determination.	Computer validation tapes, data center validation set	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
26E	e. Airborne receiver zero loop stress frequency determination.			
26F	<u>Perform Flight No. 2 Spacecraft SCS and Propulsion Leak Test</u> The Flight No. 2 spacecraft will be installed in the tilt fixture to support the remaining tests. Concurrently, the Canopus sensor, the gyro package and the fine sun sensors will be bench checked and final calibrations will be performed. Note that it is assumed that these items were removed as part of the shipping preparations. Next, the Flight No. 2 EOSE and data center will be validated to support the integrated system test. The stabilization and control subsystem and midcourse correction engine subsystem will be leak tested to ascertain that no leaks have occurred that might have been caused by shipping and handling operations. Last, each appendage will be manually deployed, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure or misalignment as a result of the shipping and handling operations.	SCS and propulsion leak test consoles	Procedure	None
27A	<u>Remove the Voyager PTM Spacecraft from the Launch Pad No. 2 Gantry</u>	Slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of ______.
27B	<u>Install Flight No. 1 Spacecraft Solar Arrays</u>	Hand tools, torque wrench	Procedure	None
27C	<u>Install MOPS End Instruments in Flight No. 2 EOSE</u>	MOPS end instruments	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
27D	Mate Test Capsule to Flight No. 2 Spacecraft	Sling, capsule handling fixture, torque wrench	Procedure	None
27E	Install All Flight No. 2 Solar Array Support Structure Mounted SCS and Experiment Sensors to Spacecraft The PTM spacecraft will be removed from the gantry and placed in the transporter in preparation for moving to the spacecraft assembly area. Next, the flight solar arrays will be attached to the spacecraft in support of the solar tests. The AMR MOPS end instruments will be installed in the Flight No. 2 EOSE. Concurrently, the capsule will be installed to the Flight No. 2 spacecraft in preparation for the integrated system test. Last, the SCS and experiment sensors originally removed from the spacecraft as part of shipping preparations will be installed.	Hand tools, torque wrench	Procedure	None
28A	Remove the Spacecraft Gantry Support Fixture from Launch Pad No. 2 Gantry	Slings	Procedure	Overhead crane with hook height of ______.
28B	Perform Flight Spacecraft No. 1 Experiment Calibrations	Complete set of systems test and experiment EOSE	Procedure	None
28C	Connect EOSE Cables to the Flight No. 2 Spacecraft	None	None	None
28D	Transport PTM Spacecraft to ESA Area	Transporter	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
28E	Torque All Flight No. 2 Spacecraft Hardware To Flight Specification The spacecraft gantry support fixture will be removed from Pad No. 2 gantry and placed in the transporter in preparation for moving to the spacecraft ESA area. The Flight No. 2 spacecraft experiment calibrations will be performed to insure that optimum experiment performance will be achieved during flight. During the experiment calibrations the EOSE cables will be connected to the Flight No. 2 spacecraft in preparation for the integrated system tests. Next, the PTM spacecraft will be transported from Pad No. 2 to the spacecraft assembly area for storage. Last, all spacecraft Flight No. 2 spacecraft hardware will be torqued to flight specification initiating the button up procedure.	Hand tools, torque wrenches	None	None
29A	Transport the Spacecraft Gantry Support Fixture to the Spacecraft Assembly Area	Transporter	None	None
29B	Perform Flight No. 1 Spacecraft Transmitter Calorimeter Test	RF Calorimeter	Procedure	None
29C	Perform Flight Spacecraft No. 2 Integrated Systems Test The spacecraft gantry support fixture will be transported to the spacecraft assembly building and stored. The first flight spacecraft transmitter calorimeter test will be performed to accurately measure the driver and power amplifier RF power delivered to the antenna system. Last, the Flight No. 2 integrated systems test will be performed to verify that the spacecraft survived the shipping and handling operations.	Complete set of systems test and experiment EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
30A	Perform Flight No. 1 Spacecraft Solar Array Test The Flight No. 1 spacecraft solar array testing will be performed as follows: a. Perform inverse impedence test on each solar array panel. b. Illuminate each array panel and measure the open circuit voltage and short circuit current.	Solar array integration test EOSE	Procedure	None
30B	Validate ESA Test Complex Using Proof Test Model Spacecraft The ESA test complex will be validated using the proof test model spacecraft.			
31A	Remove Test Capsule from Flight No. 1 Spacecraft	Hand tools	None	None
31B	Perform Final Flight No. 1 in Hangar Button-up	Hand tools, torque wrenches	Procedure	None
31C	Install and Align Flight No. 1 Appendage Pin Pullers	Hand tools, torque wrenches	Procedure	None
31D	Check all Flight No. 2 Spacecraft Alignments The capsule will be removed from the Flight No. 1 spacecraft in preparation for ESA solid retromotor installation. Last, the final in-hangar button-up will be performed. The in-hangar button-up is only a partial button-up in support of the actual bench. The partial button-up will include such things as, insulation, installation, cleaning of solar arrays, cleaning of Canopus sensor, cleaning of sun sensors, and torquing all electronic equipment panels to specification. The Flight No. 1 spacecraft pin pullers	Complete compliment of alignment sets, SCS bench test equipment	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	will be installed and aligned, insuring that proper appendage deployment will occur during flight. The pin puller alignments will be performed in two steps: align pin pullers and check pin puller alignment by manually deploying each appendage and noting that mechanical hang-up does not occur. Last, the Flight No. 2 spacecraft alignment checks will take place to insure that misalignments due to shipping and handling operations have not been incurred.			
32A	<u>Perform Flight No. 1 Spacecraft Weight and Center of Gravity Determination</u>	Hand tools, torque wrenches, center of gravity fixture, load cells and associated electronics, center of gravity fixture	Procedure	
32B	<u>Perform Flight No. 2 Spacecraft Appendage Deployment Test</u> The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. The center of gravity for two of the spacecraft axes was determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axis.	None	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	Last, each spacecraft appendage will be deployed. Each appendage will be manually deployed, observing that each appendage freely deploys with no mechanical restriction or cable chaffing due to electrical cables, mechanical failure or misalignment as a result of the shipping and handling operations.			
33A	Mate Flight No. 1 Voyager Planetary Vehicle to Transporter	Hand tools, transporter purging equipment	Procedure	None
33B	Transport Flight No. 1 Voyager Planetary Vehicle to ESA Building	Transporter, purging equipment, tractor	Procedure	Police escort
33C	Perform Flight No. 2 Spacecraft Thermal Louver Checks The Flight No. 1 spacecraft is to be transported to the explosive safe area to support the tests that are to be performed in that area. Concurrently, the thermal louver checks are performed by spraying them with a highly evaporative liquid and observing that proper operation takes place.	Evaporative liquid	Procedure	None
34A	Remove Flight No. 1 Voyager Planetary Vehicle From Transporter and Mate to Centaur Interstage	Hand tools, torque wrench	Procedure	Overhead crane with hook height of ______
34B	Perform Flight No. 2 Spacecraft Experiment Calibrations The Flight No. 1 spacecraft will be removed from the transporter and mated to the Centaur interstage in preparation for separation switch, alignment, and electrical checks. Concurrently, the Flight No. 2 spacecraft calibrations will be performed to insure that optimum experiment performance will be achieved during flight.	Complete set of systems test and experiment EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
35A	Install Flight Spacecraft No. 1 Flight Batteries and Electrically Check	Hand tools, torque wrench	Procedure	None
35B	Align and Electrically Check Flight No. 1 Spacecraft Separation Switches	Hand tools, torque wrench, separation switch, alignment set	Procedure	None
35C	Perform Flight No. 2 Spacecraft Transmitter Calorimeter Test The Flight No. 2 spacecraft batteries will be installed and electrically tested to insure that the proper cell voltages exist under load and to insure that the battery charges and discharges properly. Concurrently the Flight No. 1 spacecraft separation switches will be aligned and electrically checked to insure that the acquisition phases of the spacecraft mission profile are properly accomplished. Last, the Flight No. 2 spacecraft transmitter calorimeter test will be performed to accurately measure the driver and power amplifier RF power delivered to the antenna system.	Power EOSE, command EOSE, calorimeter	Procedure	None
36A	Perform Flight No. 1 Spacecraft Folded IST	Complete set of systems test and experiment EOSE	Procedure	None
36B	Install Flight No. 2 Spacecraft Solar Arrays	Hand tools, torque wrenches	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
36C	Revalidate Flight No. 2 Spacecraft Canopus Sensor, Fine Sun Sensor and Gyro Package The Flight No. 1 spacecraft folded integrated system test will be performed to insure that the Flight No. 1 spacecraft is ready to launch. No appendages will be articulated during this test. Last, the Flight No. 2 solar arrays will be installed in preparation of the solar array electrical tests.			
37A	Perform Flight Spacecraft No. 1 SCS and Propulsion Subsystem Leak Test	SCS leak test console, propulsion leak test console	Procedure	None
37B	Pressurize the Flight Spacecraft No. 1 SCS and Propulsion Bottles to Flight Pressures The Flight No. 1 spacecraft stabilization and control subsystem and the propulsion subsystem will be leak tested. After the leak tests each subsystem will be pressurized to flight levels as part of launch preparations.	SCS leak test console, propulsion leak test console	Procedure	None
37C	Perform Flight Spacecraft No. 2 Solar Array Test	Solar array integration and test EOSE	Procedure	None
38A	Fuel Flight No. 1 Spacecraft Monopropellant Engine	Monopropellant engine fueling set	Procedure	None
38B	Remove Test Capsule From Flight No. 2 Spacecraft	Hand tools, torque wrench, sling capsule handling fixture	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
38C	Perform Flight No. 2 Spacecraft Final in Hangar Button-up	Hand tools, torque wrenches	Procedure	None
38D	Install and Align Flight No. 2 Pin Pullers The Flight No. 1 monopropellant engine will be fueled in preparation for final on-stand activities. Concurrently, the Flight No. 2 spacecraft capsule will be removed in preparation for the installation of the solid retromotor at the ESA facility. After the motor has been removed, the final in-hangar button-up will proceed. As a part of the button-up procedure, the flight pin pullers will be installed and aligned. The pin puller alignments will take place as follows: a. Align pin pullers. b. Check pin puller alignment by manually deploying each appendage and noting that each appendage latches and unlatches properly.	Hand tools, torque wrenches, puller alignment set	Procedure	None
39A	Install Flight No. 1 Spacecraft Appendage Pin Puller and Midcourse Motor Ordnance Cartridges	Torque wrench	Procedure	None
39B	Install Flight No. 1 Spacecraft Pin Puller and Midcourse Motor Shorting Plugs at the Safe-Arm "J" Box The ordnance cartridges will be installed in each pin puller and midcourse motor actuator and torqued to flight specification. After the pin puller cartridges have been installed, shorting plugs will be connected at the safe-arm "J" box across each bridgewire.	Shorting plugs	None	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
39C	Perform Flight No. 2 Spacecraft Weight and Center of Gravity Determination The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. The center of gravity for two of the spacecraft axes was determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axis.	Hand tools, torque wrenches, c.g. fixture, load cells and associated electronics,	None	None
40	Mate Flight No. 2 Spacecraft to Transporter The Flight No. 2 spacecraft will be mated to the transporter in preparation for shipment to the explosive safe area.	Slings, spacecraft handling fixture, transporter	Procedure	Overhead crane with hoo height of ________.
41A	Perform Flight No. 1 Spacecraft Appendage Latch Test	None	Procedure	None
41B	Transport Flight No. 2 Spacecraft to the Explosive Safe Building The Flight No. 1 spacecraft pin puller alignments are to be checked by manually deploying each appendage and noting that each appendage properly latches and unlatches. Concurrently, the Flight No. 2 spacecraft will be transported to the explosive safe area for the installation of the solid retromotor and ordnance devices.	Transporter, tractor, purging equipment	Procedure	None
42A	Install Flight No. 1 Spacecraft Solid Retro Motor Insulation	None	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
42B	Mate Flight No. 2 Spacecraft to the Centaur Adapter The Flight No. 1 spacecraft solid retromotor thermal insulation will be installed in preparation for the installation of the solid motor. Concurrently, the Flight No. 2 spacecraft will be removed from the transporter and installed on the Centaur interstage in preparation for separation switch alignments and electrical checks.	Slings, spacecraft handling fixture, torque wrenches		Overhead crane with hook height of ______.
43A	Install Flight No. 1 Spacecraft Solid Engine	Sling, solid motor handling fixture, torque wrenches	Procedure	Overhead crane with hook height of ______.
43B	Install Flight Spacecraft No. 2 Flight Batteries	Torque wrenches, hand tools	Procedure	None
43C	Align and Electrically Check Flight No. 2 Spacecraft Separation Switches The Flight No. 1 spacecraft solid retromotor will be installed into the spacecraft as part of the final spacecraft buildup. Concurrently, the Flight No. 2 spacecraft battery will be installed and electrically tested to insure that the proper cell voltages exist under load and to insure that the battery charges and discharges properly.	Power EOSE, separation switch, alignment set	Procedure	None
44	Validate Flight No. 1 Spacecraft Solid Engine TVC System The flight thrust vector control subsystem interface with the stabilization and control subsystem will be thoroughly tested and calibrated.	System test set EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
45A	Connect Flight No. 2 Spacecraft Solid Motor Ordnance Test Harness	Solid motor ordnance test set	Procedure	None
45B	Perform Flight Spacecraft No. 2 Folded IST The Flight No. 1 spacecraft solid motor ordnance testing will be performed as follows: a. Connect shorting plug to ordnance safe-arm J-box. b. At the solid motor ordnance connector, observe that short circuits exist across each ordnance device by measuring resistance with a range approved milli-ohmmeter. c. Connect the solid motor ordnance cable to the spacecraft harness. The Flight No. 2 spacecraft folded integrated system test will be performed to insure that the spacecraft is ready to proceed to the launch testing phase.	Complete set of systems test and experiment EOSE	Procedure	None
46A	Perform Flight Spacecraft No. 1 Solid Retromotor Alignment	Solid motor alignment set, hand tools, torque wrenches	Procedure	None
46B	Perform Flight No. 2 Spacecraft SCS and Propulsion Leak Test Pressurize Flight No. 2 spacecraft SCS and propulsion bottles to flight pressure. The Flight No. 1 spacecraft solid retromotor alignment is to be performed to insure that solid motor coordinate axis correspond to the spacecraft axis to within the required accuracy. Concurrently, the stabilization and control subsystem and the propulsion subsystem will be leak tested. After the leak tests have been completed, each subsystem will be pressurized to full flight levels as part of the launch preparations.	SCS leak test console, propulsion leak test console	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
47A	Install Flight No. 1 Spacecraft Capsule (Assume capsule weight has been previously determined)	Sling, capsule handling fixture, torque wrenches	Procedure	Overhead crane with hook height of ______.
47B	Fuel Flight No. 2 Spacecraft Monopropellant Engine The Flight No. 1 spacecraft capsule will be installed in the spacecraft as part of the final build-up for launch testing. Concurrently, the Flight No. 2 midcourse engine will be fueled in preparation for on stand testing.	Mono-propellant fueling set	Procedure	None
48A	Perform Flight No. 1 Spacecraft Capsule Validation and Ordnance Tests	Capsule ordnance test set, system test set EOSE	Procedure	None
48B	Install Flight No. 2 Spacecraft Appendage Pin Puller and Midcourse Motor Ordnance Cartridges	Torque wrench	Procedure	None
48C	Install Flight No. 2 Spacecraft Pin-Puller and Midcourse Motor Shorting Plugs at Safe Arm J Box The flight No. 1 spacecraft capsule validation and ordnance testing will be performed as follows: a. Check all signal line voltages and currents noting that noise and transient levels are within specified levels b. Check that the spacecraft RF subsystems do not interfere with the flight capsule and that the flight capsule does not interfer with the spacecraft. The ordnance tests will be performed as follows:	Shorting plugs	None	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	1) Connect capsule shorting plugs to the ordnance safe-arm J box. 2) At the capsule ordnance connector, observe that short circuits exist across each ordnance device by measuring resistance with a range approved milli-ohmmeter. 3) Connect the capsule ordnance connector to the spacecraft harness. The Flight No. 2 spacecraft ordnance cartridges will be installed in each actuator and torqued to flight specification. After the pin puller and midcourse motor cartridges have been installed, shorting plugs will be installed across each cartridge at the safe arm J box.			
572 49A	Perform Flight No. 1 Spacecraft Capsule Alignment	Capsule alignment set, torque wrenches	Procedure	None
49B	Perform Flight No. 2 Spacecraft Appendage Latch Test The Flight No. 1 spacecraft capsule alignment will be performed to insure that the capsule coordinate axis corresponds to the spacecraft axis to within the required accuracy. Concurrently, the Flight No. 2 spacecraft pin puller alignments are to be checked by manually deploying each appendage and noting that each appendage properly latches and unlatches.	None	Procedure	None
50A	Perform Flight No. 1 Spacecraft Final Button-up	Torque wrenches	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
50B	Install Flight No. 2 Spacecraft Solid Retro Motor Thermal Insulation The Flight No. 1 spacecraft final button-up will be performed to insure that all electrical and mechanical interfaces added since the hangar testing operations have been properly mailed. All sensors and the solar arrays will be cleaned with suitable solvents. The Flight No. 2 spacecraft solid motor thermal insulation will be installed in preparation for the installation of the solid retromotor.	None	Procedure	None
51A	Check Flight No. 1 Spacecraft Vertical Alignment	Spacecraft vertical alignment set	Procedure	None
51B	Install Flight No. 2 Spacecraft Solid Retromotor The Flight No. 1 spacecraft vertical alignment will be performed to insure that the spacecraft will separate properly from the launch vehicle. Concurrently, the flight spacecraft No. 2 solid retromotor will be installed as part of the final spacecraft build up.	Sling, solid motor, handling fixture	Procedure	Overhead crane with hook height of ______.
52	Validate the Flight No. 2 Spacecraft Solid Engine TVC System The flight thrust vector control subsystem interfaces with the stabilization and control subsystem and will be thoroughly tested and calibrated.	System test set EOSE	Procedure	None
53A	Perform Flight No. 1 Spacecraft Final Weight and Center of Gravity Determination Test	Slings, spacecraft handling fixtures, weight and center of gravity fixture	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
53B	Perform Flight No. 2 Spacecraft Solid Motor Ordnance Test The spacecraft will be weighed using load cells in three places. The weight data will be used to compute the center of gravity in two of the spacecraft axes. The center of gravity for two of the spacecraft axes was determined from the spacecraft weighing exercise. The spacecraft will be tilted and the resulting three weights will be used to determine the center of gravity of the third spacecraft axis. Concurrently, the Flight No. 2 spacecraft solid motor ordnance testing will be performed as follows: a. Connect shorting plug to ordnance safe arm "J" box. b. At the solid motor ordnance connection, observe that short circuits exist across each ordnance device by measuring resistance with a range approved milli-ohmmeter. c. Connect the solid motor ordnance cable to the spacecraft harness.	Solid motor ordnance test set	Procedure	None
54A	Perform Flight No. 1 Spacecraft Final Ordnance Checks	Complete complement of ordnance test equipment	Procedure	None
54B	Perform Flight No. 2 Spacecraft Solid Retromotor Alignment The final ordnance checks will be performed as follows: a. At the safe-arm "J" box check that no voltage exists across the wires going to each ordnance device. b. At the safe arm "J" box check that zero ohms exist across each ordnance wire to ground by using a range approved milli-ohmmeter.	Solid motor, alignment set, hand t tools, torque wrenches	Procedure	None

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	c. At the safe arm "J" box determine that continuity exists through each ordnance bridge wire by using a range approved milli-ohmmeter. d. Arm the safe arm "J" box and ascertain that battery voltage exists where it should and that zero volts exists at the remaining pins of each connector. e. "Safe" the safe arm "J" box and correct the ordnance jumper. Concurrently, the Flight No. 2 spacecraft solid motor coordinate axis correspond to the spacecraft axis within the required accuracy.			
55A	Install Flight No. 1 Spacecraft Nose Fairing	Slings, nose fairing, handling fixture	Procedure	Overhead crane with hook height of ____.
55B	Install Flight No. 2 Spacecraft Capsule The Flight No. 1 spacecraft nose fairing will be placed over the Flight No. 1 spacecraft in preparation for the on-stand testing phase. Concurrently, the Flight No. 2 spacecraft capsule will be installed in the spacecraft as part of the final buildup. Note that Flight No. 2 capsule weight has previously been determined.	Sling, capsule handling fixture	Procedure	Overhead crane with hook height of ____.
56A	Perform Flight No. 1 Spacecraft Surface Sterilization	Sterilization set	Procedure	None
56B	Perform Flight No. 2 Spacecraft Capsule Validations and Ordnance Test The Flight No. 1 spacecraft will undergo surface sterilization which will be performed by soaking the spacecraft in an environment of a sterilizing gas using the nose fairing as a sterilization container. The capsule validations will be performed as follows:	Capsule ordnance set, system test set EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	a. Check all signal line voltages and currents noting that noise and transients levels are within specified levels. b. Check that spacecraft RF subsystem does not interfere with the capsule and that the capsule does not interfere with the spacecraft. Perform the capsule ordnance tests as follows: a. Connect capsule shorting plugs to the ordnance safe arm "J" box. b. At the capsule ordnance connectors, observe that short circuits exist across each ordnance device by measuring resistance with a range approved milli-ohmmeter.			
57A	<u>Mate Flight No. 1 Voyager Planetary Vehicle to the Pad Transporter</u>	Sling, spacecraft, nose fairing, handling fixture	Procedure	Overhead crane with hook height of ______.
57B	<u>Perform Flight No. 2 Spacecraft Capsule Alignment</u> The Flight No. 1 spacecraft will be mated to the pad transporter in preparation for shipment to Pad No. 1. Concurrently, the Flight No. 2 spacecraft capsule alignment is to be performed to insure that the capsule coordinate axis corresponds to the spacecraft axis to within the required accuracy.	Capsule alignment set	Procedure	None
58A	<u>Perform Flight No. 1 Spacecraft Modified IST</u>	Complete set of systems test EOSE	Procedure	None

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
58B	Perform Flight No. 2 Spacecraft Final Button-up The Flight No. 1 spacecraft modified integrated system test will be performed to insure that the spacecraft is ready for transportation to Pad No. 1. Concurrently, the Flight No. 2 spacecraft final button-up will be performed to insure that all electrical and mechanical interfaces added since the hangar testing operations have been properly mated. In addition all sensors and solar arrays will be cleaned with suitable solvents.	Torque wrenches, cleaning solvents, solvent applicators	Procedure	None
59A	Transport Flight No. 1 Voyager Planetary Vehicle to Pad No. 1	Pad transporter tractor, purging equipment, slings, spacecraft handling fixture		
59B	Check Flight No. 2 Spacecraft Vertical Alignment The Flight No. 1 spacecraft will be transported to Pad No. 1 to support the spacecraft final on stand launch activities. Concurrently, the Flight No. 2 spacecraft alignment will be performed to insure that the spacecraft will separate properly from the launch vehicle.	Spacecraft vertical alignment set	Procedure	None
60A	Mate Flight No. 1 Voyager Planetary Vehicle to the Centaur Launch Vehicle	Slings, spacecraft handling fixture, hand tools, torque wrenches	Procedure	Overhead crane with hook height of ______.

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
60B	Perform Flight No. 2 Spacecraft Final Weight and Center of Gravity Determination. The Flight No. 1 spacecraft will be hoisted to the top of the gantry and mated to the Centaur launch vehicle stage. Last, the spacecraft will be weighed in three places using load cells. The center of gravity of two of the spacecraft axes was determined from the weight determination test. The third axis will be determined by tipping the spacecraft. The resulting three weights determine the center of gravity of the third spacecraft axis.	Weight and center of gravity, fixture, load cells and electronics, slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of ______.
61A	Perform Flight No. 1 Spacecraft to Centaur Alignment Check	Spacecraft/ centaur alignment set, torque wrenches	Procedure	None
61B	Ship Pad Transporter Back to ESA Area	Tractor	None	None
62A	Perform Flight No. 1 Spacecraft on Stand Functional Test	Hangar data center, complete set of pad EOSE		Spacecraft cooling, MOPS, primary EOSE power
62B	Install Flight No. 2 Spacecraft Nose Fairing The Flight No. 1 spacecraft on-stand functional test is designed to checkout the following interfaces: a. All spacecraft umbilical functions between the spacecraft and the Pad No. 1 blockhouse. b. Wideband video pair system between the spacecraft and the data centers. c. RF link between the spacecraft and the data center. d. RF link between the spacecraft and the DSIF station.	Slings, nose fairing, handling fixture	Procedure	Overhead crane with hook height of ______.

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
	Concurrently, the Flight No. 2 spacecraft nose fairing will be placed over the Flight No. 2 spacecraft in preparation for the on-stand testing phase.			
62C	Perform Flight No. 2 Spacecraft Final Ordnance Checks The Flight No. 1 spacecraft to Centaur alignment check is performed to ascertain that the spacecraft is aligned to the Centaur coordinate system to within the required accuracy. While the Flight No. 1 spacecraft to Centaur alignment is taking place, the pad transporter is to be returned to the explosive safe area. Concurrently, the Flight No. 2 spacecraft final ordnance checks will be performed as follows: 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. At the safe-arm "J" box check that zero ohms exists across each ordnance bridge wire lead to frame ground using a range approved milli-ohmmeter. c. At the safe-arm "J" box determine that continuity exists through each ordnance bridgewire using an approved milli-ohmmeter. d. 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. e. "Safe" the safe-arm "J" box and connect the ordnance jumper connector.	Complete complement of ordnance test equipment	Procedure	None
63A	Flight No. 1 Spacecraft-Practice Conducting RFI Test	Hangar, data center, complete set of pad EOSE	Procedure	Pad cooling MOPS, pri mary EOSE power

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
63B	Perform Flight No. 2 Surface Sterilization The Flight No. 2 spacecraft will prepare for and practice the RFI test. The purpose of practicing the RFI test is to degug the procedure and launch crew familiarization. It is expected that only the spacecraft will participate. Concurrently, the Flight No. 2 spacecraft will undergo surface sterilization in an environment of sterilizing gas using the nose fairing as a sterilization container. NOTE: The Flight No. 1 spacecraft testing will temporarily cease after the RFI practice test until the Flight No. 2 spacecraft catches up.	Surface sterilization set	Procedure	None
64	Mate Flight No. 2 Voyager Planetary Vehicle to the Pad Transporter The Flight No. 2 spacecraft will be mated to the pad transporter in preparation for shipment to Pad No. 2.	Slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of ______.
65	Perform Flight No. 2 Spacecraft Modified IST. The Flight No. 2 spacecraft modified integrated system test is designed to verify that there has been no degradation of spacecraft performance during the ESA build-up and testing phase.	Complete set of systems test EOSE	Procedure	None
66	Transport Flight No. 2 Voyager Planetary Vehicle to Pad No. 2 The Flight No. 2 spacecraft will be transported to Pad No. 2 to support the spacecraft final on-stand launch activities.	Pad transporter, tractor, purging equipment	Procedure	Police escort

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
67	Mate Flight No. 2 Voyager Planetary Vehicle to the Centaur Launch Vehicle The Flight No. 2 spacecraft will be hoisted to the top of the gantry and mated to the Centaur launch vehicle.	Slings, spacecraft handling fixture	Procedure	Overhead crane with hook height of ____.
68	Perform Flight No. 2 Spacecraft to Centaur Alignment Check The Flight No. 2 spacecraft to Centaur alignment check is performed to ascertain that the spacecraft coordinate system is aligned to the Centaur coordinate system within the required accuracy.	Spacecraft/ Centaur alignment set, torque wrenches	Procedure	None
69	Perform Flight No. 2 Spacecraft On-stand Functional Test The Flight No. 2 spacecraft on-stand functional test is a test designed to checkout the following interfaces: a. All spacecraft umbilical functions between the spacecraft and the Pad No. 2 blockhouse. b. Wideband video pair system between the spacecraft and the data centers. c. RF link between the spacecraft and the data center. d. RF link between the spacecraft and the DSIF station.	Hangar, data center, complete set of pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power
70	Perform RFI Test Practice Using Both Flight No. 1 and Flight No. 2 Spacecrafts The RFI test practice is repeated again because this is the first time that both flight No. 1 and 2 are operating at the same time, affording experience in operating and coordinating two spacecrafts and two data centers at once.	Hangar, data centers, data center interpatching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
71	<u>Perform Combined Vehicle RF Interference Compatibility Test</u> The combined vehicle RF interference test is performed to ascertain that none of the Centaur or Saturn 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 centaur or saturn vehicle beacons, transmitters or receivers. The RFI compatibility test is to be performed as follows: a. Each Saturn beacon and transmitter is turned on one at a time and both the centaur and the spacecraft will ascertain that there is no interference with, or degradation of the receiver or transmitter systems. b. Each Centaur beacon and transmitter is turned on one at a time and both the saturn 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 both the Saturn and Centaur vehicles will ascertain that there is no degradation of or interference with the receiver or transmitter systems. d. All spacecraft, Centaur, and Saturn transmitters are turned on together and each vehicle will ascertain that there are no mutual degradations of or interference with the various transmitting of receiving systems.	Hangar, data centers, data center interpatching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing

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Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
72	Perform J FACT Test Preparations The J FACT test preparations are broken up into the following subtasks: a. The installation of the nose fairing separation squib simulators. b. The installation of the spacecraft umbilical cable spin-off connector squib simulators. c. The installation of the spacecraft separation squib simulators. The remainder of the day is to be spent in practicing the J FACT test procedure. It is expected that only the spacecraft will participate in this particular activity.	Hangar, data centers, data center inter-patching, pad EOSE, purging equipment	Procedure	Spacecraft cooling MOPS, primary EOSE power, range firing
73	Perform J FACT Test The purpose of the J FACT test is to checkout the post injection portions of the mission profile. The following spacecraft related postlaunch functions will be monitored and checked. a. Nose fairing separation. b. Spacecraft umbilical cable separation. c. Spacecraft separation from the Centaur vehicle. 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.	Hangar, data centers, data center inter-patching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power
74	Perform FRD Preparations As far as the spacecraft is concerned, the flight readiness demonstration preparations will consist of practicing the FRD procedure. It should be mentioned that the FRD test is identical to the countdown in regards to spacecraft activities.	Hangar data centers, data center interpatching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power

Functional Flow
Drawing Title and No. Launch Operations Revision ______ Date ______ Approval ______ Page No. 42

Operation No.	Task Description	Equipment Required	Documentation Required	Special Facilities Required
75	Perform FRD Test	Hangar, data centers, data center interpatching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing
76	Start Pre-countdown Both spacecrafts will participate in the pre-countdown activities. Prior to the conclusion of the pre-countdown activities each subsystem of each spacecraft will have been checked. At the conclusion of the pre-countdown activities a decision will be made as to whether Flight No. 1 or No. 2 spacecraft will be launched.	Hangar, data centers, data center, interpatching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing
77	Commence Terminal Countdown During terminal countdown, the launch vehicles will be fueled with oxidizer and the gantry removed.	Hangar, data centers, data center interpatching, pad EOSE, purging equipment	Procedure	Spacecraft cooling, MOPS, primary EOSE power, range firing
78	Lift Off	Hangar, data center, data center inter-patching	Procedure	MOPS, range firing

APPENDIX B
RELIABILITY PROGRAM PLANNING

For Phase IB of the Voyager program, TRW will draft a reliability program plan in accordance with the NASA Reliability Publication NPC 250-1. Certain features of the plan may be noted in advance.

First, the plan will be of major scope and will call for a reliability effort that operates throughout the life of the program.

Second, the plan will be organized in accordance with NPC-205-1. It will contain a detailed account of tasks, milestones, and level of effort needed to fulfill the mission reliability requirements established by JPL Project Document No. 45 (V-MA-004-001-14-03, Preliminary Voyager 1971 Mission Specification, May 1, 1965). The plan will also follow the guidelines in the TRW Reliability Manual and draw upon applicable DAC and RCA reliability procedures. Fortunately, all three companies already pursue basically similar methods.

Third, the plan will identify three areas of special importance in reliability program planning, as follows:

a) Subcontractors. Paragraph 2.6 of NPC-250-1, relating to subcontractor and supplier control, will be applied.

b) Testing. The test board will schedule specific tests of all levels of material as required to meet reliability verification requirements.

c) Quality Assurance. Reliability tasks (per NPC-250-1) will be smoothly coordinated with quality assurance tasks (per NPC-200-2 and -3) and such coordination provided for in the reliability program plan and the reliability assurance plan.

Fourth, the plan will identify and describe 16 reliability task elements, framed in accordance with NPC-205-1, that are necessary to meet reliability program requirements.

We foresee the development and coordination of the plan moving through three steps: 1) a preliminary plan to be submitted as part of the Phase IB proposal, 2) an intermediate plan growing out of program level-of-effort negotiations, and 3) a final plan for NASA/JPL formal review and approval.

The remainder of this appendix is devoted to the 16 reliability tasks.

Task 1 - Reliability Program Management

Reliability program management will focus strongly on systems engineering during Phase IB and on product-design in Phase II. To coordinate prime and subcontractor efforts, a joint reliability concil will be formed at the start of Phase IB. Project task planning will give equal attention to spaceborne and critical ground equipment.

Task 2 - NASA/JPL Liaison

TRW recognizes the broad system and mission responsibilities borne by NASA/JPL and the necessity for effective liaison on all critical reliability matters. As presently foreseen, specific formal liaison actions will include: NASA/JPL approval of the reliability program plan; reviews per NPC-250-1, paragraph 2.3; and independent assessments, per NPC-250-1, paragraph 1.4.2. NASA/JPL will also take part in joint reliability council meetings, design reviews (per NPC-250-1, paragraph 3.6.1), failure reporting-corrective action cycles, and review of test data. Finally, the status of all reliability action items will be reported currently and in a format designed to facilitate accurate monitoring and assessment by NASA/JPL.

Task 3 - Reliability Program Plans

Current guidelines for reliability program planning apply to the preliminary plan to be prepared in response to the RFP for Voyager Phase IB. The preliminary plan will be composed of identified tasks, along with project schedules and milestones. Specific reliability organizations with responsibile personnel will be shown for TRW and its major subcontractors. Detailed descriptions will be provided for the parts and materials plan and the design review plan for Voyager as required by Appendix B of NPC-250-1. Further detailed considerations for other task areas will be included in the intermediate and final plans and in cooperation with NASA/JPL reliability activities during the Voyager Phase IB study interval.

Task 4 - Reliability Models and Estimates

During the Phase IA study, numerous system and subsystem reliability models were employed in arriving at Voyager designs. The modeling techniques (which are described in Chapter 8 of TRW Systems' Reliability Manual) will be reviewed in the light of Voyager program needs when the intermediate or final reliability program plans are drafted.

Task 5 - Reliability Tradeoff

Reliability objectives will differ for each mission because each flight varies with respect to launch opportunity, mission purpose, scientific payload, weight reserve, etc. Within any mission plan, design commitments made for reliability must be traded off for various subsystems in accordance with their relative criticality to the mission. These constraining issues include those given in the Preliminary Voyager 1971 Mission Specification for the primary mission objectives (page 5) and the competing characteristics factors (page 21) pertinent to spacecraft and capsule mode priorities. There are various tradeoff areas where reliability is a significant constraint. These will be enumerated in the reliability program plans and will include the weight versus reliability tradeoff exercise discussed in Volume 4, Section III. 4, of the report, and used in this study to arrive at the preliminary Voyager spacecraft design.

Task 6 - Reliability Input to Specifications

The plan will contain a schedule of detailed events whereby numerical reliability requirements will be invoked for Voyager subsystems and elements. These requirements will be based upon analyses of the kind discussed here. In the conduct of the Phase IA study, reliability requirements for Voyager subsystems have been established and included in Volume 2 for the 1971 spacecraft and Volume 7 for the 1969 spacecraft. These requirements have been established as design goals commensurate with the achievements of the maximum level of Voyager mission success in accordance with the established (numerical) primary mission objectives for reliability. Continued inputs to specifications for reliability requirements will be planned and integrated with the other scheduled design and manufacturing events for systems, subsystems, equipment, and parts level materiel.

Task 7 - Parts and Materials

Fulfillment of valid reliability predictions and achievement of reliable end products depends upon appraisal and control of Voyager material at the level of parts and materials. TRW will present parts and materials evaluation and control practices suitable to Voyager as part of the preliminary reliability program plan. The specifics of these practices are to be consistent with the stated requirements of the JPL Preliminary Voyager 1971 Mission Specification, Section 6, paragraph 2, as interpreted in response to the primary Voyager mission objectives.

Task 8 - Design Constraint Planning

In addition to parts and material considerations, design constraints will apply to weight, magnetic properties, contamination control, electromagnetic interference, circuit tolerance control, maintainability features, environment control functions, and element testability. In each case, reliability analysis and judgment factors will constitute significant flight spacecraft design criteria and constraints in accordance with paragraph 6 of the JPL Preliminary Voyager 1971 Mission Specification. The underlying objectives of the reliability program plan will be to relate all reliability-oriented design constraints into a coherent plan. At the outset, broad design constraints will take the form of structural-design safety factors, electronic part derating policies, thermal excursion maxima, etc. In each of these instances, preliminary constraints were set for the designs evolved during Phase IA and are inherent in the reliability apportionment ground rules documented in Volume 2, Section 3, for the 1971 Voyager spacecraft.

Task 9 - Mission Reliability Analysis

In arriving at meaningful design goals for Voyager subsystems and equipment, we have made physical interpretations of the probabilistic mission objectives given in the JPL Preliminary Voyager 1971 Mission Specification. These interpretations must be updated and all variances in design achievement (relative to the specified goals established) interpreted in terms of the over-all Voyager mission success probability. Such mission reliability analyses will proceed in accordance with the reliability model updating provisions of NPC-250-1, paragraph 3.3.

Task 10 - Design Review

Effective reviews of Voyager system, subsystem, and equipment designs are important to the reliability effort. The preliminary reliability program plan will detail an approach (per NPC-250-1, Appendix B) and will provide representative review meeting agenda and technical review criteria. The design review meetings for all subcontract design phases will be chaired by the Voyager subcontractor project manager. All design reviews will be attended by the key technical design engineers for the contractor or subcontractor plus responsible reliability experts. The latter will validate and follow up all action items pertinent to the tasks outlined in this document. It is expected that NASA/JPL representatives will participate in all final-stage design reviews in accordance with NPC-250-1, paragraph 3.6.

Task 11 - Reliability Test Program Planning

Voyager spacecraft development and verification tests will include those designed specifically to yield statistical verification of reliability requirements as well as tests designed to assure functional capabilities and "worst case" qualifications as provided by NASA Document NPC-200-2, paragraph 4.3. Similar requirements and tests will be imposed on critical ground operating equipment. As noted in NPC-250-1, paragraph 4.1, the contractor (and subcontractor) reliability organizations will not normally have primary responsibility for testing; however, they will be responsible for ensuring that the integrated test plan (Section IV) provides for economical and timely reliability evaluation at the system, subsystem, and component levels. The reliability program also embraces economical reliability test planning at the parts and materials level. This responsibility will be outlined in the preliminary plan.

Task 12 - Reliability Test Data Reduction

A key responsibility will be to accumulate and interpret the reliability data from all Voyager tests. While special attention will be paid to tests specifically intended for reliability verification, pertinent data will be collected from all other Voyager tests, from NASA and other sources, to provide a composite engineering evaluation of Voyager materiel reliability. In reducing and interpreting test data, consideration will be given to the combined statistical and engineering confidences associated with the various compromises made for sample sizes, environmental simulations, mission time and actuation replicas, system configuration variations, and test and measurement facilities. A preliminary evaluation of such practical compromises as they pertain to Voyager appears in Section IV of this volume.

Task 13 - Failure-Corrective Action

TRW as well as DAC and RCA have all had direct and practical experience with failure reporting and corrective action systems corresponding to NPC-250-1, paragraph 3.7. A detailed description of failure reporting and corrective action procedures, organizational responsibilities, and report formats will be included in the intermediate reliability program plan. This system will embrace both reliability and quality assurance and will provide for smooth data and action controls across the contractor, subcontractor, and intra-company organizational boundaries. The system will incorporate strict reporting, analysis, and corrective feedback for fabrication, handling, test, checkout, and operational phases. Malfunction analysis procedures will include thorough documentation of malfunction events and use of the most experienced personnel to render decisions of malfunction categorization, corrective measure action, and case disposition.

Task 14 - Reliability Progress Reporting

Progress reporting requirements are prescribed in NPC-250-1, Section 5. These include brief weekly summaries, periodic progress reports (coincident with Voyager project progress reports), and reliability program control reports as separate fiscal and management portions of the Voyager project report required by the contract. Status reporting will cover all the action items in the approved reliability program plan.

Task 15 - Subcontractor Reliability Controls

Subcontractor project managers will direct interface operations (meetings, schedules, follow-up, funding) between TRW and its subcontractors. However, the reliability program manager will establish technical requirements for subcontractor reliability and verify successful completion. DAC and RCA will establish for TRW approval (and incorporation in the over-all reliability program plan), separate task definitions and schedules for the reliability areas under their cognizance. The specific details of a Voyager plan for subcontractor reliability control will be included in the intermediate reliability program plan as prescribed in NPC-250-1, Appendix C, and related directly to fiscal and over-all project schedule factors.

Task 16 - Reliability Training

Reliability training activities in accordance with NPC-250-1, paragraph 2.5, will be coordinated with the Voyager quality training plan in accordance with NPC-200-2, paragraph 13. Details will be spelled out in the intermediate reliability program plan.

APPENDIX C

MAGNETIC CONTROL PLAN OUTLINE

1. INTRODUCTION

The Voyager project requires the establishment of a magnetics control plan, which will be directed by a magnetic control group within the Voyager organization. The outline of this plan, which will be described in detail in the Phase IB proposal, is presented below, described in terms of the tasks to be accomplished during both Phase IB and Phase II.

2. GENERAL

A general description of the tasks to be accomplished in providing effective magnetic control includes:

a) Participation in the design of the various subsystem assemblies and the over-all spacecraft to control the type and positioning of the components to minimize permanent fields and wiring techniques to reduce stray fields.

b) Control of assembly and processing operations to prevent magnetic contamination of clean materials. Past experience has shown that assemblies like fiberglass antennas have become contaminated and magnetic.

c) Magnetic receiving tests on all materials and components to be used in the spacecraft.

d) Complete magnetic testing of all assemblies, both operating and static, and in the magnetized and demagnetized condition.

e) Participation in the spacecraft layout of the subsystems and orientation of assemblies within a subsystem to minimize the magnetic field of the spacecraft seen by the magnetometer sensor.

f) Testing of the spacecraft in the nonoperating condition to determine the permanent field, and in all operating and failure modes to determine the stray field.

3. PHASE IB TASK OUTLINE

The following tasks are considered necessary for the implementation of the Phase IB magnetics control program. The same approach has been successfully used on the OGO and Pioneer programs and more recently on the USAF 2029 program.

General

a) Provide a personnel and funding plan for the tasks associated with the Voyager magnetic control program.

b) Plan and coordinate magnetic tests in conjunction with other TRW departments; this includes breadboard tests and special component tests. Extensive special component tests are often necessary in developing techniques for minimizing magnetic fields in specific problem areas.

c) Coordinate assembly magnetic test requirements within TRW. The magnetic test requirements of each assembly fabricated must be designed to yield the maximum useful information about the magnetic properties of the assembly while in no way jeopardizing the function of the assembly.

d) Coordinate assembly testing within TRW. The contents of each magnetic test procedure for assemblies fabricated by TRW must be determined, including determining distances at which measurements are made and the operating modes to be exercised during the test.

e) Coordinate assembly magnetic properties and test requirements with subcontractors. Requirements for the magnetic properties and testing of subcontracted assemblies must be determined.

f) Coordinate assembly testing with subcontractors. Magnetic test procedures prepared by the subcontractors will be reviewed.

g) Planning and coordination of spacecraft testing. Spacecraft magnetic testing requirements will be defined, and tests to meet these requirements devised. Special test equipment necessary for spacecraft testing will be designed.

h) Preparation of contractually required reports, i. e., progress reports, material reports, and other contractually required documentation including the preparation of a preliminary spacecraft magnetic test plan.

Parts Testing

a) Preliminary tests and studies leading to establishment of general criteria for parts testing. These studies and tests define the magnetic level above which parts are considered unacceptable for the Voyager mission and below which parts are considered acceptable for the Voyager mission. This level is influenced by many factors, such as the total parts count of the spacecraft, the length of the magnetometer boom, and the relative locations of the assemblies.

b) Assist in parts and materials tests and selection.

c) Attend parts deviation meetings as a parts deviation board member. Parts deviation board meetings are set up to incorporate new parts on the approved parts list. Parts are investigated to determine their magnetic characteristics and previous reliability history. Acceptable parts are then incorporated into the approved parts list. If the part is unacceptable a search is initiated to find a suitable substitute.

d) Study and recommend solutions for troublesome parts which are magnetic and functionally replaceable with nonmagnetic substitutes.

e) Generate specific criteria for incoming inspection of all parts and materials. Parts and materials to be used on the Voyager spacecraft will be magnetically screened at incoming inspection. The parts list is divided into two classes: Class I parts which are nonmagnetic and Class II parts which are magnetic. All Class I parts are tested to a general magnetic test procedure containing the criteria for failure. Each Class II part type is handled individually. The criteria for failure for each Class II part type is established. This criteria together with the Class II parts incoming inspection magnetic test procedure are used to screen Class II parts at incoming inspection.

Design and Development

a) Generate magnetic control guidelines specific to the Voyager program.

b) Participate in breadboard tests and analyze results. Breadboard tests will be conducted and the results analyzed to diagnose potential problem areas.

c) Participate in system and assembly design reviews. Each system and assembly will be carefully analyzed to determine whether magnetic fields are minimized and, based on the analyses, recommendations will be made.

d) Assist subcontractors in the areas of magnetic control. Magnetic control guidelines will be supplied to the subcontractors and TRW will assist the subcontractors in establishing magnetic control programs.

e) Assist subcontractors in setting up facilities and magnetic testing techniques. TRW experience in the field of magnetic measurements will be made available to the subcontractors to assist them in establishing their magnetic test facilities and magnetic testing techniques.

f) Study magnetic problem areas and recommend solutions. System and assembly magnetic field problems will be studied and recommendations made.

g) Determine magnetic criteria for each assembly. Magnetic field criteria will be established based on the maximum allowable field at the magnetometer sensor, the position of the assembly relative to the sensor, and the number and the nature of the parts in the assembly.

h) Perform solar panel and solar array tests. On past programs it has been shown that solar arrays can be manufactured to be completely nonmagnetic when nonoperating. In the operating mode, stray fields have been very accurately predicted and eventually reduced to extremely low levels (0.1 gamma at the sensor) by making use of a mock-up of the array. Copper strips were used to simulate the sheets of current produced in the solar cells. Wiring routes were traced exactly to duplicate the interconnection wiring. The various panels were then energized by passing currents through them. Not only is this system representative of the actual array but it lends itself to simulating any failure modes that might occur. It is proposed that this simulation should be verified and that an array of mock-up panels be used to determine the stray magnetic field due to the solar array at the position of the magnetometer sensor (see Figure C-1, mock-up of 2029 solar array).

Procedures and Specifications

a) Prepare parts and materials incoming inspection procedures. The magnetic test procedure for Class I and Class II incoming inspection will be prepared.

b) Prepare assembly magnetic test procedures. Magnetic test procedures for the magnetic testing of each assembly fabricated by TRW will be prepared.

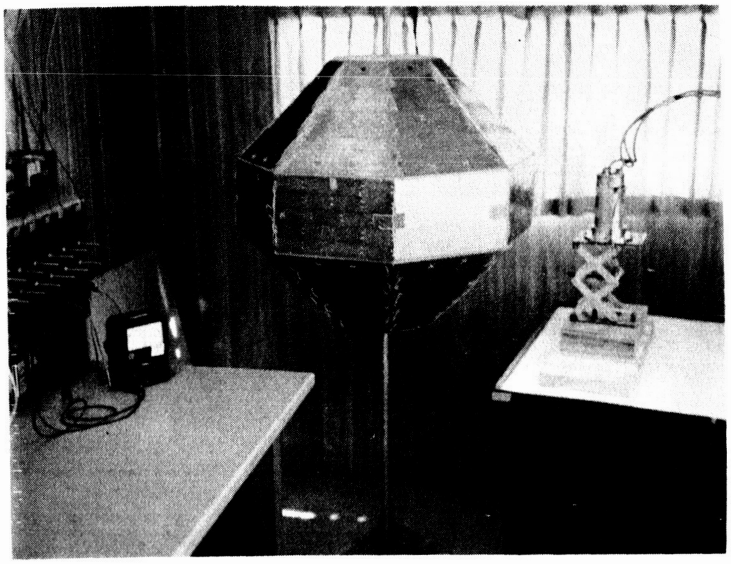

Figure C-1. Mock-up of Solar Array, 2029 Program

Spacecraft Testing

a) Design special fixtures and test equipment for spacecraft perming, deperming, and mapping. Using the coilless method in determining the magnetic field of the spacecraft necessitates a handling fixture to rotate the spacecraft about two axes. This method of mapping the spacecraft, while not as accurate, is far less costly than using a coil system. If greater accuracy is required in determining the spacecraft magnetic field, a coil system far larger than the present Malibu facility will have to be constructed. If the coils are accurately controlled with regard to drift, the spacecraft need not be rotated to ascertain the off-set. If, on the other hand, the coils do drift, a fixture like that used on the coilless method will be required (see Figures C-2 and C-3). Fixtures will also be required to hold the spacecraft while the boom-mounted experiment sensor is positioned in a coil facility, such as that at Malibu, for interference and calibration tests. The same fixtures can be utilized for positioning the spacecraft within the perming and deperming coils. On the instrument side, commercial test equipment will be used wherever possible in the test setup necessary for this operation.

b) Design for special test equipment and holding fixtures for solar array testing. Unless swayed by other considerations, it is not planned to test the solar array as part of the complete spacecraft. From past experience, nothing is achieved by having this in the measuring facility along with the spacecraft since it is not contributing power and therefore exhibiting no stray field. Statically, it should be nonmagnetic and certified by individual panel measurements. A fixture is required to hold the array while being illuminated and the stray field measurements carried out. Design of the load banks and switching units along with the measuring equipment is also required for this test.

c) Plan and calibrate site equipment. The planning of site test equipment will be strongly influenced by past experience obtained on the OGO and Pioneer programs, and similar test equipment necessary for the mapping of the spacecraft will be used. Calibration of site equipment and earth's gradients is made against a proton magnetometer.

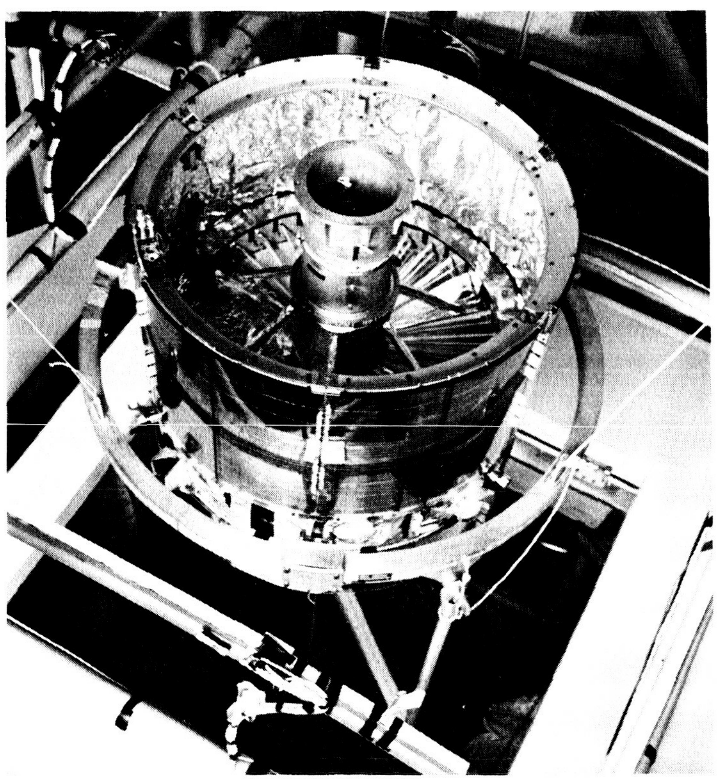

Figure C-2. Pioneer Handling Fixture in Coils

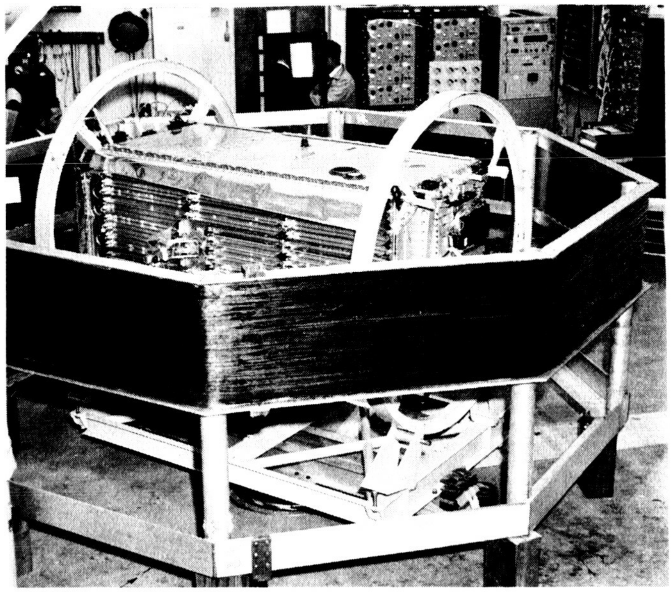

Figure C-3. Handling Fixture for OGO Perming-Deperming Tests

4. PHASE II TASK OUTLINE

Documentation

Preparation of a final spacecraft magnetic test plan.

Parts Testing

Assist in the resolution of problems arising at the magnetic incoming inspection of Voyager parts and materials.

Design and Development

Perform engineering model tests on assemblies fabricated by TRW. Since engineering models are constructed to prove out a design and are as nearly identical as possible to the flight design, engineering model tests are extremely valuable in determining the stray magnetic fields due to current loops. If the stray field of the assembly proves troublesome at this point, modifications can be made to minimize the stray fields in time to be incorporated into subsequent units.

Procedures and Specifications

a) Prepare preliminary solar array magnetic test procedures
b) Prepare final solar array magnetic test procedures
c) Prepare preliminary spacecraft magnetic test procedures
d) Prepare final spacecraft magnetic test procedures

Assembly Testing

a) Analysis of assembly test data for assemblies fabricated and tested by TRW. Complete copies of the assembly test data are provided to the Magnetic Control Group. This data is analyzed and evaluated to determine acceptability relative to the Voyager mission and compliance with magnetic control procedures. The results of the analysis are forwarded to the Voyager Project.

b) Analysis and monitoring of assembly tests performed by subcontractors. Assembly testing performed by subcontractors will be monitored. The results of the assembly testing will be analyzed relative to the Voyager mission requirements and in compliance with magnetic control procedures. The analysis will be forwarded to the Voyager Project and to the subcontractor.

c) Evaluation of the assembly magnetic test data relative to the spacecraft magnetic properties. The results of the assembly magnetic tests are compiled to present an up-to-date estimate of the spacecraft magnetic field at the position of the magnetometer sensor.

Spacecraft Testing

a) Construct special test equipment and fixtures for solar array testing.

b) Perform solar array testing.

c) Perform solar array magnetic test data analysis.

d) Construct special test equipment and fixtures for spacecraft testing.

e) Calibrate site and equipment for spacecraft tests.

f) Perform dry runs to verify the compatibility of site and test procedures. These tests will also provide invaluable experience for the Voyager test crews and help to minimize unnecessary and avoidable delays in the spacecraft tests.

g) Perform spacecraft tests.

h) Perform spacecraft magnetic test data analysis.

i) Participate with the experimenter in any required calibration tests of the spacecraft/magnetometer sensor combination.

APPENDIX D
CONTAMINATION CONTROL

1. INTRODUCTION

It is the purpose of this planning document to state the guidelines to be followed for an organized approach to the evolution of an effective contamination control plan. This control plan will be separated into two areas: nonbiological and biological contamination control.

2. NONBIOLOGICAL CONTAMINATION CONTROL

2.1 Introduction

Contamination control procedures will be essential during construction of the flight spacecraft to:

- Achieve the highest degree of functional reliability
- Preclude failure of sensitive instrumentation due to contamination
- Minimize the degree of microbiological contamination during fabrication
- Eliminate the presence of magnetized chips, filings, and other products.

Following is a discussion on cleanliness requirements, methods of obtaining cleanliness, controls used to maintain cleanliness, and contamination inspection procedures.

2.2 Requirements

2.2.1 Cleanliness Requirements

Cleanliness requirements will be specified by Quality Assurance, and are to be in conformance with JPL requirements. All components and assemblies requiring any level of cleaning or clean room practices will be so stated on the engineering drawings. TRW will specify the methods and materials to clean, package, and assemble designated components.

2.2.2 Support Facilities

TRW will specify the cleanliness requirements for facilities. Clean rooms will meet the requirements of Federal Standard 209 or its equivalent. Laminar flow benches and portable work stations will be used inside the clean rooms when more stringent controls are required.

2.2.3 Cleaning Equipment

Cleaning equipment such as solvent and cleaning solution pump units, flushing consoles, ultrasonic units, and drying equipment will be constructed of low particle-producing materials with filtration provided between the equipment and the component being cleaned.

2.2.4 Cleaning and Testing Materials

Cleaning and testing fluids will be prefiltered to the cleanliness level defined by Process Engineering to meet design engineering requirements. Particle counts will be taken on the filtered fluids as a control measure. Nonvolatile residue tests will be performed when necessary. Cleaning and testing gases will be prefiltered to meet design engineering requirements. Vendor shipments of gas will be checked for dew point and nonvolatile hydrocarbon content. All expendable materials such as identification inks, cleaning cloths, writing materials, and tote boxes will be selected by Process Engineering if they are to be used in environmentally-controlled areas.

2.3 Methods of Contamination Control

2.3.1 Critical Components

Contamination control provisions will be made in process specifications for all flight spacecraft components. Special attention will be provided those operations in which there is production of chips, burrs, filings, and other products in which magnetic fields may be established by the fabrication processes. Components will be precleaned to remove corrosion, scale, and flux, prior to final cleaning. The level of cleanliness will be specified by Design Engineering and approved by Quality Assurance.

2.3.2 Final In-plant Assembly

The final assembly of the flight spacecraft subsystems will be in a high reliability assembly and checkout area. Physical contamination will be minimized through personnel and environmental control.

2.3.3 Packaging and Shipping

Packaging of cleaned parts and assemblies will be in tamper-proof containers meeting or exceeding the cleanliness conditions under which each unit was fabricated. Whenever necessary, temperature, humidity, and pressure will be controlled in shipping containers.

2.3.4 The Planetary Vehicle

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 or its equivalent. The precise level of control will be determined by Quality Assurance. The flight spacecraft including the flight capsule will be enclosed in the nose fairing under similar conditions.

2.4 Documentation

Complete documentation will be obtained through design drawings. Materials will be controlled by government or industrial specifications. No deviations will be allowed from the specifications without written approval from Design Engineering.

2.5 Controls

2.5.1 Personnel Training and Certification

The Industrial Training Department will train and certify all personnel who will clean or assemble critical components. Only those who have completed the training course and successfully passed the written tests will be authorized for clean room work. The training program will include the following:

- A general introduction concerning the significance of contamination as it relates to the Voyager program
- Familization with the approved materials to be used in cleaning and packaging

- Specific techniques of cleaning, clean assembly, and packaging
- Discipline of dress when working in clean rooms
- A written examination

2.5.2 Quality Assurance

Quality Assurance will maintain surveillance over all contamination control requirements and processes. Subcontractors and vendors will be certified and a list of approved sources will be maintained. All sources will be recertified at regular intervals.

2.5.3 Verification of Cleanliness

All parts will be subjected to a visual examination immediately after cleaning. The effectiveness of the process will be maintained and controlled by conducting sample tests as follows: The parts will be washed with a known volume of solvent and a particle count will be performed on the effluent. If required, the nonvolatile residue content of the effluent will be determined.

2.5.4 Identification of Item Cleanliness

The minimum identification on cleaned parts will consist of the certification stamp of the employee who cleaned the part; the part number and serial number; the date of cleaning; and the specification to which the part was cleaned.

3. BIOLOGICAL CONTAMINATION CONTROL

3.1 Introduction

In order to meet the JPL requirement of a one part in 10^4 chance of biologically contaminating Mars in any one launch attempt, extensive measures will be taken to insure sterilization of the capsule and spacecraft effluents.

Voyager flight capsules will be sterilized and delivered to the Cape Kennedy explosion proof facility under conditions established to maintain their sterility. However, the exterior of the capsule biological barrier (cannister) and of the flight spacecraft will be contaminated, and hence it will be necessary to sterilize the external surfaces of the flight spacecraft and the flight capsule cannister after installation within the nose fairing to assure the biological contamination requirements.

Another means by which the flight spacecraft may contaminate the flight capsule is from gases ejected by the attitude control and midcourse correction systems. Some small fraction of these gases will surely be on trajectories intercepting Mars and another fraction will distribute itself around the flight spacecraft. Prior to separation from the spacecraft the cannister will be removed from the capsule, resulting in the capsule being ejected through a potentially contaminating cloud; to reduce this the cold gases and the cold gas systems will be sterilized.

3.2 Requirements

3.2.1 Attitude Control and Midcourse Correction Systems

Hardware associated with the attitude control and midcourse correction systems may be sterilized either by dry heat or with a gas purge with 12 per cent ethylene oxide and 88 per cent freon (12-88). The dry heat sterilization would require special handling of the system during installation to avoid microbiological contamination. The simpler procedure would be to purge the tanks, valves, and lines before the filling operations with 12-88 but after the systems have been assembled within the flight spacecraft.

The hydrazine [redacted] fuel under consideration for the monopropellant is self sterilizing. Therefore the fuel and its containers will be sterile, however, the jets through which the fuels will be emitted will not be sterile nor will the brief contact with the fuel during firing be sufficient to sterilize them. It will be necessary to surface sterilize them with 12-88.

The cold gas system will also be purged with 12-88 prior to filling. The cold gases will be filled through sterile high pressure microbiological filters. The filters will be selected from those currently under investigation by NASA contractors.

Final assembly of the flight spacecraft with the flight capsule will be conducted in the explosion safe facility. If the sterilization is to be conducted in the same explosion safe facility, the spacecraft and capsule assembly will be enclosed in the nose fairing. The biological shroud will be assembled at the base of the nose fairing and the entire unit purged with 12-88. Any time the barrier is penetrated the unit will be resterilized. The planetary vehicle will then be mated to Centaur without disrupting the integrity of the sterility barrier. It may be necessary to purge the planetary vehicle after it is mated with Centaur; at this point, with the Centaur shroud in place, it will be possible to also surface sterilize Centaur. Figure D-1 presents the functional flow diagram of this procedure.

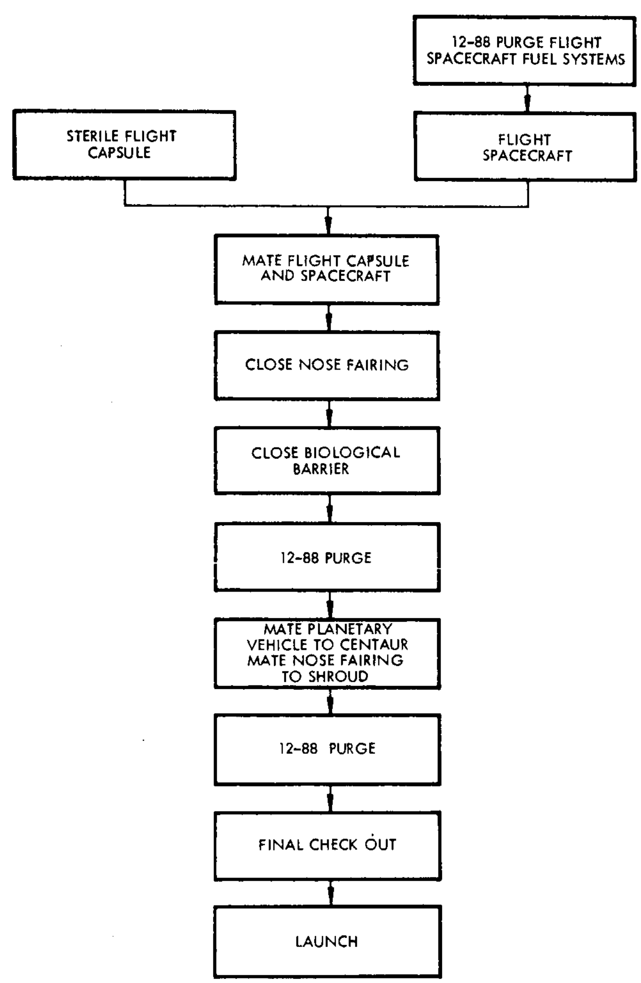

Figure D-1. Functional Flow Diagram of Voyager Flight Spacecraft Surface Sterilization Process

If the sterilization is to be conducted on the stand, the spacecraft and capsule assembly will be mated with the Centaur; the biological shroud will be assembled and the nose fairings installed. Sterilization of the planetary vehicle will then be conducted, and, if desired, surface sterilization of the Centaur can be accomplished.

3.2.2 Facilities

Fabrication of units and structures whose surfaces will be externally exposed in the flight spacecraft will be accomplished under clean room conditions. The degree of cleanliness required will be determined by Quality Assurance. All clean room procedures will be reviewed from the standpoint of minimizing the microbiological contamination during fabrication. This will be performed to ensure that sterilization be accomplished during the time period designated for the sterilization process.

3.2.3 Sterilization Requirements

In order to achieve ethylene oxide sterilization it is essential to recognize the complexities of the process. Success is dependent upon integration of ethylene oxide concentration with time, temperature, and humidity. Other factors such as the nature of materials, gas penetration into difficult areas, and resistance of the microorganisms are equally important. Therefore, final values for the various parameters will depend upon the ability of the planetary vehicle and its enclosure to tolerate the stress. The following conditions are considered to be optimum for achieving a 5 to 6 hour sterilization: temperature: $55^{\circ}C$, humidity: 50 per cent RH, and positive gas pressure as required:

A typical standard gas sterilizing cycle is as follows:

a) Preconditioning phase in which an initial vacuum is drawn on a preheated system and the unit is humidified.

b) The 12-88 is introduced via a heat exchanger until the required pressure is reached at which time the gas flow is discontinued.

c) 4 to 6 hour exposure period

d) Evacuation of the unit and a terminal vacuum is drawn

e) The unit is returned to atmospheric pressure by introducing filtered air to prevent recontamination

It is recognized that the optimum conditions of vacuum and pressure will not be tolerated by the nose fairing and microbiological shroud. Tradeoffs will have to be made depending on engineering constraints.

The parameters of time, temperature, and humidity will be experimentally determined when all constraints are defined.

3.3 Methods

Design engineering will provide the sterilization unit. Port attachments will be needed in the design of the nose fairing for attachment of the sterilization purge unit. Additional ports in the nose fairing will be needed through which sterilization controls may be inserted and withdrawn.

Sterilization requirements will be experimentally determined in the Douglas Microbiology Laboratory.

3.4 Personnel

All microbiological assay work will be performed by trained microbiologists. The sterilization program will be supervised by microbiologists versed in the problems of contamination control, ethylene oxide sterilization, and hardware constraints.

3.5 Sterilization Controls

The sterilization controls will be selected from: commercially available strips, NASA recommendations, and preparation assembled in the Douglas Microbiology Laboratory. Controls will be inserted through ports in the nose fairing and exposed to the sterilization cycle. All controls will be removed following sterilization and assayed for the achievement of sterility.

3.6 Final Assembly

If the planetary vehicle is sterilized in the explosion safe facility, it will be transported to the launch site and mated to Centaur without violating the integrity of the sterility barrier. A terminal ethylene oxide purge will be conducted prior to final circuit check and launch.

APPENDIX E
EQUIPMENT LIST

This appendix contains the preliminary equipment lists for the 1969 flight test (Table E-1) and the 1971 spacecraft (Table E-2), together with the equipment lists for the mechanical and electrical operational support equipment to support both spacecraft (Table H-3).

Table E-1. 1969 Flight Test Equipment List

1969 VOYAGER EQUIPMENT LIST

Structural Subsystem Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks *- These units also used for life test.
Main Body														
Bus frame	1	1		1		1		2	1		1	2	2	*Mounting panels also serve as meteoroid protection panels
Mounting panels*	4	4		4		4		8	4		4	8	8	
Meteoroid protection panels forward	1	1		1		1		2	1		1	2	2	
Meteoroid protection panels aft	1	1		1		1		2	1		1	2	2	
Propulsion attach brackets	1	1		1		1		2	1		1	2	2	
Centaur to space capsule support cone	1	1		1		1		2	1		1	2	2	
Centaur - spacecraft separation joint	1	1		1	1	1		2	1		1	2	2	
Bolts and separation nut	4	4		4	4	4		8	4		4	8	8	
Antenna														
Antenna mount support structure	1		1	1		1		2	1		1	2	2	

1969 VOYAGER EQUIPMENT LIST

Structural Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks
														*- These units also used for life test.
Deployed Appendages														
LF antenna	1		1	1		1		2	1		1	2	2	

1969 VOYAGER EQUIPMENT LIST

Thermal Control Subsystem		Development								T.A. and Reliabil.		Flight		
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2)	Thermal Model	Type Approval Units *	Proof Test Model *	Flight Spacecraft 1969 (2)	Spares	Remarks
Flight Spacecraft														*- These units also used for life test.
Louvers	63	See 1971 List	See 1971 List	63				120	63	See 1971 List	63	126	126	
Forward aluminized mylar insulation blanket	1			1				2	1		1	2	2	
Aft insulation blanket	1			1				2	1		1	2	2	
Side panel aluminized blanket	6			6				12	6		6	12	12	
Midcourse motor insulation blanket	1			1				2	1		1	2	2	
Heaters	3			3				6	3		3	6	6	
Thermostats	3			3				6	3		3	6	6	

1969 VOYAGER EQUIPMENT LIST

Propulsion Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks
Engine (monopropellant)	1	See 1971 List	See 1971 List	1		1	1	2	1	See 1971 List	1	2	2	*- These units also used for life test.
Propellant Tanks (N_2N_4)	1			1		1	1	2	1		1	2	2	
Monopropellant engine valve module	1			1		1	1	2	1		1	2	2	
Propellant fill valve	1			1		1	1	2	1		1	2	2	
Helium fill valves	1			1		1	1	2	1		1	2	2	
Pressure transducers	4			4		4	4	8	4		4	8	8	
Temperature transducers	2			2		2	2	4	2		2	4	4	
Propellant feed system plumbing	1			1		1	1	2	1		1	2	2	
Pressurization system plumbing	1			1		1	1	2	1		1	2	2	
Monopropellant engine thrust structure	1			1		1	1	2	1		1	2	2	
Propellant tank support	1			1		1	1	2	1		1	2	2	

1969 VOYAGER EQUIPMENT LIST

Stabilization and Control Subsystem — Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks
Sensors (Optical)		See 1971 List	See 1971 List				See 1971 List			See 1971 List				*- These units also used for life test.
Canopus sensor	1			1		1		2	1		1	2	2	
Course sun sensor	4			4		4		8	4		4	8	8	
Fine sun sensor	1			1		1		2	1		1	2	2	
Earth sensor	1			1		1		2	1		1	2	2	
Sensors (Inertial)														
Control gyros & elect. package	1			1		1		2	1		1	2	2	
Actuators (Electromechanical)														
Thrust vector control actuator (midcourse)	4			4		4		8	4		4	8	8	
Reaction Thrust Control														
Regulator	2			2		2		4	2		2	4	4	
Solenoid valve	12			12		12		24	12		12	24	24	
Transducers	4			4		4		8	4		4	8	8	
Pressure vessel	2			2		2		4	2		2	4	4	

1969 VOYAGER EQUIPMENT LIST

Stabilization and Control Subsystem

Equipment Item	Quantity per spacecraft	Development								T. A. and Reliabil.		Flight		Remarks
		Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units *	Proof Test Model *	Flight Spacecraft 1969 (2)	Spares	*- These units also used for life test.
Fill valve	2	See 1971 List	See 1971 List	2		2	See 1971 List	4	2	See 1971 List	2	4	4	
Nozzle	12			12		12		24	12		12	24	24	
Plumbing set	2			2		2		4	2		2	4	4	
Electronics														
Control signal elect. package	1			1		1		2	1		1	2	2	
Control drive elect. package	1			1		1		2	1		1	2	2	

3

1969 VOYAGER EQUIPMENT LIST

Central Sequencing and Command Subsystem — Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks
Sequencer	2	See 1971 List	See 1971 List	2		2	2	4	2	See 1971 List	2	4	4	*- These units also used for life test.
Decoder, command	2			2		2	2	4	2		2	4	4	
Decoder, input	2			2		2	2	4	2		2	4	4	
Power	2			2		2	2	4	2		2	4	4	

1969 VOYAGER EQUIPMENT LIST

Communications and Data Subsystem		Development								T. A. and Reliabil.		Flight		
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units *	Proof Test Model *	Flight Spacecraft 1969 (2)	Spares	Remarks
Signal conditioner	1			1		1		2	1		1	2	2	*- These units also used for life test.
S-band receiver	3			3		3		1	3		3	6	6	
Command detector	1			1		1		2	1		1	2	2	
Preamplifier	1			1		1		2	1		1	2	2	
VHF receiver	1			1		1		2	1		1	2	2	
PCM encoder	2			2		2		4	2		2	4	4	
Transmitter selector	1			1		1		2	1		1	2	2	
Buffer storage unit	1			1		1		2	1		1	2	2	
Receiver selector	1			1		1		2	1		1	2	2	
Bulk storage unit	1			1		1		2	1		1	2	2	
Circulator switch antenna gimbals	4	See 1971 List	See 1971 List	4		4	See 1971 List	8	4	See 1971 List	4	8	8	
Power supply 20w	2			2		2		4	2		2	4	4	
Power amplifier 20w	2			2		2		4	2		2	4	4	
Low power transmitter	1			1		1		2	1		1	2	2	

1969 VOYAGER EQUIPMENT LIST

Communications and Data Subsystem		Development								T.A. and Reliabil.		Flight		
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units *	Proof Test Model *	Flight Spacecraft 1969 (2)	Spares	Remarks
4 port power divider	1			1		2	1		1			2	2	*- These units also used for life test.
High gain antenna	1			1		1		2	1		1	3	2	
High gain antenna feed	1			1		1		2	1		1	3	2	
Aux. low gain antenna	1	2	2	1		1		2	1		1	3	2	
Diplexer	3			3		3		6	3		3	9	6	
Low gain antenna	1			1		1		2	1		1	3	2	
Circulator switch	2			2		2		4	2		2	6	4	
Rotary joint	2			2		2		4	2		2	6	4	
High gain antenna mount structure	1	See 1971 List	See 1971 List	1		1	See 1971 List	2	1	See 1971 List	1	3	2	
High gain antenna actuator gimbal	1			1		1		2	1		1	3	2	

1969 VOYAGER EQUIPMENT LIST

Power Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model Mockup	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks
Solar panel sections			4											*- These units also used for life test.
Solar array	1			1				2	1	1	1	2	2	
Shunt elements	1			1		1		2	1		1	2	2	
Power control unit	1			1		1		2	1		1	2	2	
Battery packs	2			2		2		4	2		2	4	4	
Battery cells		See 1971 List	See 1971 List				See 1971 List			See 1971 List				
Charge current regulator	2			2		2		4	2		2	4	4	
Boost voltage regulator	2			2		2		4	2		2	4	4	
300 w 4.1 kc inverter	2			2		2		4	2		2	4	4	
20w 820 cps inverter	2			2		2		4	2		2	4	4	
50w 410 cps inverter	2			2		2		4	2		2	4	4	
Solar panel brackets	1			1		1		2	1	1	2	2	2	
Solar panel structure folding	3			1		3		2	1	3	2	2	2	

Power Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model Mockup	T.A. and Reliabil.: Type Approval Units *	T.A. and Reliabil.: Proof Test Model *	Flight: Flight Spacecraft 1969 (2)	Flight: Spares	Remarks
Solar panel structure fixed	1	See 1971 List	See 1971 List	1		1		2	1	1	2	2	2	*- These units also used for life test.
Solar panel actuation system	3			4		3		8	4	3	8	8	8	
Solar panel release system	1			1		1		2	1	1	2	2	2	

1969 VOYAGER EQUIPMENT LIST

Electrical Distribution Subsystem

Equipment Item	Quantity per spacecraft	Development								T. A. and Reliabil.		Flight		Remarks
		Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model	Type Approval Units *	Proof Test Model *	Flight Spacecraft 1969 (2)	Spares	*- These units also used for life test.
Interconnecting cables	20	See 1971 List	1	20		20		40	20	See 1971 List	20	60		Thermal model representative wire bundles
Junction boxes	2		1	2		2		4	2		2	6		

Table E-2. 1971 Flight Test Equipment List

1971 VOYAGER EQUIPMENT LIST

Structural Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model	T.A. and Reliability: Type Approval Units	T.A. and Reliability: Proof Test Model	T.A. and Reliability: Reliability and Life Test Units	T.A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Main Body																		
Bus frame	1	1		1		1		2	2		1	3	1	3	1	1	2	
Mounting panels	6	6		6		6		12	6		6	3	6	18	6	6	12	
Meteoroid protection panels forward	1	1		1		1		2	1		1	3	1	3	1	1	22	
Meteoroid protection panels aft	1	1		1		1		2	1		1	3	1	3	1	1	12	
Propulsion solid support cone	1	1		1		1		2	1		1	3	1	3	1	1	2	
Spacecraft Capsule bolt and separation nut	3	3		3		3		6			3	3	3	9	3	3	6	
Centaur spacecraft separation bolt and nut	3	3		3		3		6			3	3	3	9	3	3	6	
Flight Capsule																		
Bus-capsule interface fitting	6	1		6		6		12	6		6		6	18	6		6	

1971 VOYAGER EQUIPMENT LIST

Structural Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model	T. A. and Reliability: Type Approval Units	T. A. and Reliability: Proof Test Model	T. A. and Reliability: Reliability and Life Test Units	T. A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
<u>Antenna</u>																		
Antenna mount structure	1	1		1		1	1	2	1		1		1	3	1		1	
<u>Deployable appendages</u>																		Typical booms $\approx$ 10 feet long
Magnetometer boom	1		1	1		1		2	1		1		1	3	1	1	2	
LF antenna boom	1		1	1		1		2	1		1		1	3	1	1	2	

1971 VOYAGER EQUIPMENT LIST

Thermal Control Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model	T.A. and Reliability: Type Approval Units	T.A. and Reliability: Proof Test Model	T.A. and Reliability: Reliability and Life Test Units	T.A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Flight Spacecraft																		
Louvers	63		50	63				126	63	63	63	189	63	189	63	63	126	
Forward aluminized mylar insulation blanket	1		1	1				2	1	1	1	3	1	3	1	1	2	
Aft insulation blanket aluminized mylar and refrosil batt	1		1	1				2	1	1	1	3	1	3	1	1	2	
Side panel aluminized mylar blanket	6		6	6				12	6	6	6	3	6	18	6	6	12	
Midcourse motor insulation blanket (aluminum foil-fiberglass paper)	1		1	1				2	1	1	1	3	1	3	1	1	2	
Heaters	3		1	3				6	3	3	3	3	3	9	3	3	6	

1971 VOYAGER EQUIPMENT LIST

Thermal Control Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2)	Development: Thermal Model	T.A. and Reliability: Type Approval Units	T.A. and Reliability: Proof Test Model	T.A. and Reliability: Reliability and Life Test Units	T.A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Thermostats	3		1	3				6	3	3	3	3	3	9	3	3	6	
Solid motor insulation blanket (aluminum foil-fiberglass paper)	1		1	1				2	1	1	1	3	1	3	1	1	2	
<u>Planet Oriented Package</u>																		
Aluminized Mylar blanket	1		1	1				2	1		1	3	1	3	1	1	2	
Heater	1		1	1				2	1		1		1	3	1	1	2	
Thermostats	1		1	1				2	1		1		1	3	1	1	2	
<u>External Experiment Package</u>																		
Aluminized mylar blanket	1		1	1				2	1		1	3	1	3	1	1	2	
Heater	1		1	1				2	1		1	3	1	3	1	1	2	
Thermostat	1		1	1				2	1		1		1	3	1	1	2	

1971 VOYAGER EQUIPMENT LIST

Propulsion Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliability: Type Approval Units	T.A. and Reliability: Proof Test Model	T.A. and Reliability: Reliability and Life Test Units	T.A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Monopropellant engine thrust structure	1	1		1		1	1	2	1	1	1	3	1	3	1	1	2	
Propellant tank supports	2	2		2		2	2	4	2	2	2	3	2	6	2	2	4	
Monopropellant engine valve module	1	1		1		1	1	2	1	1	1	3	1	3	1	1	2	
Monopropellant engine	1	1		1		1	1	2	1	1	1	3	1	3	1	1	2	
Solid propellant motor	1	1		1		1	1	2	1	1	1	3	1	3	1	1	2	
Propellant tanks (W_2H_4)	2	2		2		2	2	4	2	2	2	3	2	6	2	2	4	
Helium fill valves	2	2		2		2	2	4	2	2	2	3	2	6	2	2	4	
Propellant fill valve	2	1		1		1	1	2	1	2	1	3	1	3	1	1	2	
Pressure transducers	4	4		4		4	4	8	4	4	4	3	4	12	4	4	8	

1971 VOYAGER EQUIPMENT LIST

Propulsion Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliability: Type Approval Units	T.A. and Reliability: Proof Test Model	T.A. and Reliability: Reliability and Life Test Units	T.A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Temperature transducers	2	2		2		2	2	4	2	2	2	3	2	6	2	2	4	
Propellant feed system plumbing set	1	1		1		1	1	2	1	1	1	3	1	3	1		2	
Pressurization system plumbing set	1	1		1		1	1	2	1	1	1	3	1	3	1		2	

1971 VOYAGER EQUIPMENT LIST

Stabilization and Control Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	T.A. and Reliability: Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Sensors (Optical)																		
Canopus sensor	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Course sun sensor	4	4	8	4		4	4	8	4	4	4	5	4	12	4	4	4	
Fine sun sensor	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Terminal guidance sensor	1	1	2	1		1	1	2	1	1	1	5	2	6	2	2	2	
Earth sensor	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Sensors (Inertial)																		
Control gyros and electrical package	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Actuators (electro-mechanical)																		
Thrust vector control actuator (midcourse)	4	4	8	4		4	4	8	4	4	4	5	4	12	4	4	8	
Thrust vector control actuator (orbit injection)	4	4	8	4		4		8	4	4	4	3	4	12	4	4	8	

1971 VOYAGER EQUIPMENT LIST

Stabilization and Control Subsystem		Development								T. A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Reaction Thrust Control																		
Regulator	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	
Solenoid valve	12	12	24	12		12	12	24	12	12	12	3	12	36	12	12	24	
Transducers	4	4	8	4		4	4	8	4	4	4	3	4	12	4	4	8	
Pressure vessel	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	
Fill valve	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	
Nozzle	12	12	24	12		12	12	24	12	12	12	3	12	36	12	12	24	
Plumbing set	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	
Electronics																		
Control signal electrical package	1	1	2	1		1	1	2	1	1	1	3	1	3	1	1	2	
Control drive electrical package	1	1	1	1		1	1	2	1	1	1	3	1	3	1	1	2	

1971 VOYAGER EQUIPMENT LIST

Central Sequencing and Command		Development								T. A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Input decoder	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Command decoder	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Sequencer	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Power	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	

1971 VOYAGER EQUIPMENT LIST

Power Subsystem		Development								T.A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Batteries	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Inverter (300 w, 4.1 kc)	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Inverter (20 w, 820 cy)	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Inverter (50 w, 410 cy)	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Regulator	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Power control unit	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Shunt element assembly	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Solar panel brackets	6	1		6		6		12	6	6	6	3	6	18	6	6	12	
Solar panel structure	6	1		6		6		12	6	6	6	3	6	18	6	6	12	
Solar panels	6		12	6		6		12	6	6	6	3	6	18	6	6	12	

1971 VOYAGER EQUIPMENT LIST

Science Subsystem		Development								T. A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Body Mounted																		
Meteoroid impact (4 sensors)	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Magnetometer	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Plasma (2 sensors)	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Cosmic ray (4 sensors)	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Trapped radiation	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Ionosphere experiment	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Meteoroid flash	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
IR spectrometer	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Data auto equip.	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
TV experiment	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
UV spectrometer	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Scan radiometer	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	

1971 VOYAGER EQUIPMENT LIST

Science Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T. A. and Reliability: Type Approval Units	T. A. and Reliability: Proof Test Model	T. A. and Reliability: Reliability and Life Test Units	T. A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Planet oriented package mount	1	1		1		1		2	1		1	3	1	3	1		2	
POP Mounted																		
Experiment equipment	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Interconnecting wiring	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Mars sensor	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Scan platform actuator	1	1	2	1		1	1	2	1	1	1	3	1	3	1	1	2	
Antenna servos	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	
Gimbals	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	
Housing	1	1	2	1		1	1	2	1	1	1	3	1	3	1	1	2	
Servos	2	2	4	2		2	2	4	2	2	2	3	2	6	2	2	4	

1971 VOYAGER EQUIPMENT LIST

Electrical Distribution Subsystem		Development								T.A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Interconnecting cables	20		1	20		1	1	*	20	20	1	1	1	3	1	20	1	*Thermal model, representative wire bundle
Junction boxes	2		1	2		1	1	1	2	2	1	1	1	3	1	2	1	

1971 VOYAGER EQUIPMENT LIST

Communications and Data Subsystem

Equipment Item	Quantity per spacecraft	Development: Engineering Breadboards	Development: Engineering Models	Development: Configuration Model Mockup	Development: Separation Model	Development: Spacecraft Engineering Model	Development: Simulators	Development: Structural Model (2) Mockup	Development: Thermal Model Mockup	T.A. and Reliability: Type Approval Units	T.A. and Reliability: Proof Test Model	T.A. and Reliability: Reliability and Life Test Units	T.A. and Reliability: Life Test Spacecraft (1)	Flight: Flight Spacecraft 1971 (3)	Flight: Prototype for JPL (1)	Flight: Type Approval Subsystems	Flight: Spares	Remarks
Power supply, 20 w	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Power amplifier, 20 w	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Preamplifier	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
VHF receiver	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Command detector	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Low power transmitter, 1 w	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
PCM encoder	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Signal conditioner	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Buffer storage unit	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Bulk storage unit	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Capsule demodulator	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	

1971 VOYAGER EQUIPMENT LIST

Communications and Data Subsystem		Development								T. A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model Mockup	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2) Mockup	Thermal Model Mockup	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Modulator exciter (150 mw)	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
Command detector	2	2	4	2		2	2	4	2	2	2	5	2	6	2	2	4	
S-band receiver	3	3	6	3		3	3	6	3	3	3	5	3	9	3	3	6	
Receiver selector	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Transmitter selector	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
Circulator switch	4	4	8	4		4	4	8	4	4	4	5	4	12	4	4	8	
Diplexer	3	3	6	3		3	3	6	3	3	3	5	3	9	3	3	6	
4 port power divider	1	1	2	1		1	1	2	1	1	1	5	1	3	1	1	2	
High gain antenna reflector (6 ft)	1	1	2	1		1		1	1	1	1	3	1	3	1	1	2	
High gain antenna feed	1	4	2	1		1		1	1	1	1	3	1	3	1	1	2	
Medium gain antenna ref.	1	3	2	1		1		1	1	1	1	3	1	3	1	1	2	

1971 VOYAGER EQUIPMENT LIST

		Development								T.A. and Reliability				Flight				
Equipment Item	Quantity per spacecraft	Engineering Breadboards	Engineering Models	Configuration Model	Separation Model	Spacecraft Engineering Model	Simulators	Structural Model (2)	Thermal Model	Type Approval Units	Proof Test Model	Reliability and Life Test Units	Life Test Spacecraft (1)	Flight Spacecraft 1971 (3)	Prototype for JPL (1)	Type Approval Subsystems	Spares	Remarks
Medium gain antenna feed	1	3	2	1		1		1	1	1	1	3	1	3	1	1	2	
Low gain antenna	1	3	2	1		1		1	1	1	1	3	1	3	1	1	2	
VHF antenna	1	2	2	2		1		1	1	1	1	3	1	3	1	1	2	
Rotary joint	3	6	6	3		3		3	3	1	3	3	3	9	3	3	6	
High gain antenna mount structure	1	1	1	1		1		1	1	1	1	3	1	3	1	1	2	
Medium gain antenna mount structure	1	1	1	1		1		1	1	1	1	3	1	3	1	1	2	
High gain antenna actuator gimbal	1	2	1	1		1		1	1	1	1	3	1	3	1	1	2	
Medium gain antenna actuator gimbal	1	2	1	1		1		1	1	1	1	3	1	3	1	1	2	

Table E-3. Operational Support Equipment

System Test Complex Unit Test Sets	Quantity Required 1971
Command Data Handling Subsystem	
• S-band communications unit test set	7*
• UHF communications unit test set	7
• Command decoder unit test set	7
• Data handling unit test set	7
Stabilization and Control Subsystem	
• Rate gyro assembly unit test set	6
• Sun sensor and near earth detector unit test set	6
• Star sensor unit test set	6
• Stabilization and control electronics assembly unit test set	6
Central Sequencing and Command Subsystem	
• Central sequencing and command unit test set	6
Power Subsystem	
• Solar panel unit test set	6
• Power inverter unit test set	6
• Battery control unit test set	6
• Power control electronics assembly unit test set	6
• Battery unit test set	6

*Quantity of 7 for 1971 consists of 1 new Unit Test Set in addition to the requirements of the 6 Unit Test Sets developed for 1969

	Quantity Required	
	1971	GFE
Electrical Distribution Subsystem		
• Electrical distribution unit test set	3	
Planet Oriented Package Subsystem		
• Planet oriented package unit test set	6	
Propulsion Subsystem	-	
System Test Sets	9	9a
Communications Data Handling System		
Launch Complex Equipment		
• STS	-	-
• ADAS	-	-
• Monitor console	2	-
• RF console	2	-
Mission Dependent Equipment	4	2b

a = SDS-930 computer

b = SDS-910 computer

System Test Complex Unit Test Sets	Quantity Required 1969
Command Data Handling Subsystem	
• S-band communications unit test set	6
• UHF communications unit test set	6
• Command decoder unit test set	6
• Data handling unit test set	6
Stabilization and Control Subsystem	
• Rate gyro assembly unit test set	5
• Sun sensor and near earth detector unit test set	5
• Star sensor unit test set	5
• Stabilization and control electronics assembly unit test set	5
Central Sequencing and Command Subsystem	
• Central sequencing and command unit test set	5
Power Subsystem	
• Solar panel unit test set	5
• Power inverter unit test set	5
• Battery control unit test set	5
• Power control electronics assembly unit test set	5
• Battery unit test set	5

	Quantity Required	
Electrical Distribution Subsystem	1969	GFE
• Electrical distribution unit test set	2	
Planet Oriented Package Subsystem		
• Planet oriented package unit test set	5	
Propulsion Subsystem		
System Test Sets	4	
Communications Data Handling System	4	9a
Launch Complex Equipment		
• STS	-	-
• ADAS	-	-
• Monitor console	2	-
• RF console	2	-
Mission Dependent Equipment	4	2b

a = SDS-930 computer

b - SDS-910 computer

Nomenclature	Use Location: Transportation	Subcontractor Plant	TRW Plant	Remote Test Sites	JPL	AFETR	Quantity Required 1971
Assembly, Handling and Shipping Equipment (Flight Spacecraft and Over-all Flight Spacecraft) (OSE/VS-3-140)							
Transporter, Flight Spacecraft	x						4
Assembly, Handling and Tilt Fixture			x		x	x	7
Transport Recorder	x						4
Fixture, Weight, Center of Gravity and Moment of Inertia			x			x	2
Shipping Container Group Standard Modules	x	x	x	x	x	x	50
Work Platforms, Mobile		x	x			x	7
Adapter Kit, Centaur/Shroud Transporter						x	2
Sling Assembly, Planetary Vehicle and Nose Fairing						x	2
Purge Unit, Freon/Ethylene Oxide						x	2
Planetary Vehicle/Nose Fairing Mating and Assembly Fixture Fixture						x	2
Sling, Flight Capsule			x	x	x	x	2
Hoist Beam and Sling, Flight Spacecraft		x	x	x	x	x	4
Tag Lines						x	2
Platform Launch Stand Access						x	2
Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle	x	x	x	x	x	x	4
Environmental Cover, Flight Spacecraft	x				x		4
Hoist Sling, Environmental Cover		x	x	x	x	x	4
Platform, Auxiliary Access			x	x	x	x	6
Science Payload Subsystem (OSE/VS-4-210)			x				
Alignment Fixture, Science Payload			x		x	x	4
Shipping Container, Experiment Booms	x						5
Communications and Data Handling Subsystems (OSE/VS-4-310)							
Dolly, 6' Parabolic Antenna			x		x	x	4
Hoist Beam 6' Parabolic Antenna			x		x	x	4
Shipping Container, 3' Parabolic Antenna	x						5
Shipping Container, 6' Dish Antenna	x						1
Shipping Container, Low gain Antenna	x						5
Shipping Container, Flight Capsule Receiving Antenna	x						5
Stabilization and Control Subsystem (OSE/VS-4-410)							
Alignment Fixture, Stabilization and Control Nozzles			x		x	x	4
Protective Covers, Stabilization and Control Nozzles	x	x	x	x	x	x	28
Power Subsystem (OSE/VS-4-460)							
Assembly and Handling Frame, Solar Panel Segment	x	x	x	x	x	x	30
Protective Cover, Solar Panel Segment	x	x	x		x	x	30
Shipping Container, Solar Panel Segment	x						15
Handling Dolly, Solar Panel Segment		x	x		x	x	18
Sling Assembly, Solar Panel Segment		x	x		x	x	6
Shipping Container, Battery	x						10
Shipping Container, Power Amplifier	x						2

	Use Location						
Nomenclature	Transportation	Subcontractor Plant	TRW Plant	Remote Test Sites	JPL	AFETR	Quantity Required 1971
Thermal Control Subsystem (OSE/VS-4-510)							
Assembly and Handling Fixture, Spacecraft Louvers		x	x		x	x	20
Shipping Container, Spacecraft Louvers	x						5
Handling and Shipping Container, Insulation	x	x	x				4
Structural Subsystem Equipment (OSE/VS-4-520)							
Dolly, Structural Sections		x	x		x		4
Shipping Containers, Miscellaneous Spacecraft Structure	x						4
Sling, Propulsion/Pneumatic Structural Section		x	x	x		x	4
Interface Match Tool, Spacecraft/Flight Capsule		x	x				2
Interface Match Tool, Spacecraft/Centaur Adapter		x	x				2
Pyrotechnic Subsystem (OSE/VS-4-530)							
Shipping Container, Explosive Train	x						4
Handling Case, Arming Kit				x		x	2
Planet Oriented Package Subsystem (OSE/VS-4-580)							
Assembly Fixture and Dolly, POP			x		x	x	4
Shipping Container, POP	x						2
Hoist Beam, POP			x		x	x	3
Propulsion Subsystem (OSE/VS-4-610)							
Sling, Retropropulsion Motor		x	x		x	x	4
Dolly, Retropropulsion Motor		x	x		x	x	4
Alignment Fixture, Retropropulsion Motor			x			x	4
Alignment Fixture, Midcourse Engine			x	x		x	4
Shipping Container, Midcourse Engine	x						2
Pneumatic Test Set		x	x	x		x	3
Pneumatic Fill Cart		x		x		x	3
Propellant Transfer and Handling Cart		x		x		x	3
Alignment Fixture, Midcourse Engine/Steering Vanes			x	x		x	4
Universal Handling Fixture, Hydrazine/Helium Tank		x	x			x	4
Sling, Hydrazine/Helium Tank		x	x			x	3

Nomenclature	Use Location: Transportation	Subcontractor Plant	TRW Plant	Remote Test Sites	AFETR	Quantity Required 1969
Assembly, Handling and Shipping Equipment (Flight Spacecraft and Over-all Flight Spacecraft)(OSE/VS-3-140)						
Transporter, Flight Spacecraft	x					3*
Assembly, Handling and Tilt Fixture			x		x	5*
Transport Recorder	x					3*
Fixture, Weight, Center of Gravity and Moment of Inertia			x		x	2*
Shipping Container Group, Standard Modules	x	x	x	x	x	50*
Work Platforms, Mobile		x	x		x	5*
Hoist Beam and Slings, Flight Spacecraft		x	x	x	x	4
Tag Lines					x	2*
Platform, Launch Stand Access					x	2
Universal Mounting Ring, Flight Spacecraft and Planetary Vehicle	x	x	x	x	x	4
Environmental Cover, Flight Spacecraft	x					3*
Hoist Sling, Environmental Cover		x	x	x	x	3*
Platform, Auxiliary Access			x	x	x	6*
Transporter Adapter Cradle, 1969 Test Spacecraft						3
Communications and Data Handling Subsystems (OSE/VS-4-310)						
Dolly, 6' Parabolic Antenna			x		x	3*
Hoist Beam, 6' Parabolic Antenna			x		x	3
Shipping Container, 6' Dish Antenna	x					1*
Shipping Container, Low Gain Antenna	x					4*
Stabilization and Control Subsystem (OSE/VS-4-410)						
Alignment Fixture, Stabilization and Control Nozzles			x		x	3*
Protective Covers, Stabilization and Control Nozzles	x	x	x	x	x	24
Power Subsystem (OSE/VS-4-460)						
Assembly and Handling Frame, Solar Panel Segment	x	x	x	x	x	12
Protective Covers, Solar Panel Segment	x	x	x		x	12
Shipping Container, Solar Panel Segment	x					6
Handling Dolly, Solar Panel Segment		x	x		x	8
Sling Assembly, Solar Panel Segment		x	x		x	5
Shipping Container, Battery	x					10*
Shipping Container, Power Amplifier	x					2*
Thermal Control Subsystem (OSE/VS-4-510)						
Assembly and Handling Fixture, Spacecraft Louvers		x	x		x	16*
Shipping Container, Spacecraft Louvers	x					4*
Handling and Shipping Container, Insulation	x	x	x			3*

*1969 uses 1971 equipment as is or with removable MOD kits

Nomenclature	Use Location: Transportation	Subcontractor Plant	TRW Plant	Remote Test Sites	AFETR	Quantity Required 1969
Structural Subsystem Equipment (OSE/VS-4-520)						
Dolly, Structural Sections		x	x			3
Shipping Containers, Miscellaneous Spacecraft Structure	x					3
Sling, Propulsion/Pneumatic Structural Section		x	x	x	x	4
Interface Match Tool, Spacecraft/Centaur Adapter		x	x			2
Pyrotechnic Subsystem (OSE/VS-4-530)						
Handling Case, Arming Kit				x	x	2
Propulsion Subsystem (OSE/VS-4-610)						
Alignment Fixture, Midcourse Engine			x	x	x	4*
Shipping Container, Midcourse Engine	x					2*
Pneumatic Test Set		x	x	x	x	2*
Pneumatic Fill Cart		x		x	x	2*
Propellant Transfer and Handling Cart		x		x	x	2*
Alignment Fixture, Midcourse Engine/Steering Vanes			x	x	x	4*
Universal Handling Fixture, Hydrazine/Helium Tank		x	x		x	4*
Sling, Hydrazine/Helium Tank		x	x		x	3*

*1969 uses 1971 equipment as is or with removable MOD kits

SIGNIFICANT ERRATA. TRW Systems, Phase 1A Study Report, Voyager Spacecraft

August 11, 1965

Volume 1. Summary

Substitute new p. 79 attached.

Volume 2. 1971 Voyager Spacecraft

p. 18. Item h) "necessary landed operations" should read "necessary lander operations."

p. 143. Section 3.4.1.a. second line should read "threshold of 0.25 gamma"

p. 282. Lines 3 and 4. Delete "or incorrect spacecraft address"

p. 284. Figure 5. Change "128 Word DRO Core Memory" to "256 Word DRO Core Memory"

p. 327. Denominator of second term on right hand side of equation should read

$$\left(\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1\right)\left(N - 1\right)$$

p. 351. Figure 1, Section F-F. "separation nut" should read "bolt catcher"

Volume 3. Voyager Program Plan

Substitute new p. 12 attached.

p. 13. Figure 2-3. PTM Assemblies in item 7 move 1.5 months to right

p. 16. Figure 2-6. First milestone date should be September 1, 1969, instead of mid-January 1970, and all subsequent dates should be correspondingly adjusted 4.5 months earlier.

p. 20. Table 2-2. Third item in 1969 column should read "coincident with completion of proof test model assemblies. Fifth item in this column change "2 weeks" to "3.5 months." Fourth item in 1971 column, change "4 months" to "5 months."

p. 67. Figure 5-2. Under Intersystem Interface Specification add a block entitled "Spacecraft to OSE Interface Specification"

p. 120. Last line of paragraph c should read "shown in Table 5-2."

p. 126. Figure 5-13. Year should be 1966 instead of 1965.

p. 153. Figure 5-18. Ignore all numbers associated with lines in figure.

p. 167. Figure 5-21. In line 20 change "design revisions" to "design reviews"

p. 254. Second paragraph, third line, "The capability of the transmitter to select" should read "The capability of the transmitter selector to select."

p. 258. Section heading n should read Experiment Data Handling

p. 604. Section 3.2.1 beginning of second paragraph should read "The hydrazine fuel . . . "

Volume 4. Alternate Designs: Systems Considerations

p. 103. Figure 3-19. Caption should read "Radial Center of Mass. . . "

p. 151. Last paragraph, second line, "For the baseline, the reliability. . ." should read "The reliability . . . "

p. 158. 8th line, replace "0.06 pound/watt" by "0.6 pound/watt"

p. 215. Figure 3-50. Dot in ellipse at right should be 0.

p. 230. Section 5.3.2, second paragraph, 7th line, should read "Figure 3-52."

p. 239. Second line, "with a variable V" should read "with a variable ΔV"

p. 247. First line, "3250 km/sec" should read "3.250 km/sec"

p. 261. Figure 3-64. Interchange coordinates, clock angle and cone angle

p. 293. Figure 3-81. An arrow should connect "Low-gain spacecraft antenna" and the dashed line at 73 $\times$ 10^6 km

Volume 4. Alternate Designs: Systems Considerations Appendix

p. 6. Figure A-2. The shaded portion under the lower curve should extend to the right only as far as 325 lb.

p. 9. Table A-1, part (1). In last column heading change "W_3" to "W_1". In part (4) last column heading change "W_3" to "W_4"

p. 22. Second line below tabulation, replace "575 × 35" by "570 × 35"

p. 29. Tabulation at bottom of page, change "18" to "30" and "400" to "240"

p. 207. Numerator of equation for λ best at bottom of page should read "0.0201," and numerator of equation for λ worst should read "9.21"

p. 209. Table 5B, fifth line. Delete "$\times 10^{-}$." Also p. 213, Table 7A, seventh line, and p. 232, Table 3B, fifth line.

p. 217. Top portion of Table 9B should be labeled "primary mode" instead of "other modes"

p. 326. In equations following words "clearly" and "thus" insert "$>$" before second summation.

Volume 5. Alternate Designs: Subsystem Considerations

p. 3-15 Fifth line, "... is extended, spacecraft" should read "... is extended, two spacecraft"

p. 3-38 Last line, change "$= \frac{32}{4500} = M$" to "$\left(\frac{32}{4500}\right)\left(M\right)$"

p. 3-51 Two equations at bottom of page should read

$$D = 4\pi A/\lambda^2$$

$$A = \frac{D\lambda^2}{4\pi} = \frac{1000\,\lambda^2}{4\pi}$$

p. 3-67 Third line, last parenthesis "$\left(\frac{\pi}{2} + \phi\right) -$"

p. 3-82 6th line should read "50 degrees" instead of "50-140 degrees," and seventh line should read "140 degrees" instead of "50-140 degrees"

p. 3-111 Last line, change "50 Mc" to "1 Mc"

p. 3-137 Item g) for "... followed by 5 frames of real time" substitute "... followed by 11 frames of low rate science data and 5 frames of real time" pp. 3-150 and 3-151 are interchanged.

p. 3-156 Last line, should read "gates, a 7 bit"

p. 5-21 Second paragraph, third line, for "others since they are" substitute "others which are"

p. 5-33 Bjork equations should identify 0.18 as an exponent, and the exponent for (ρ_p/ρ_t) in the Hermann and Jones equation should be 2/3 in both cases.

p. 5-33 Figure 5-12 should be replaced with Figure C-7 of Appendix C.

p. 5-40 Three lines above Table 5-10 substitute "permanent set" for "experiment"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix I

p. B-11 Bottom of page, for "$r^{2/3}$" substitute "$(V/C)^{2/3}$ r"

p. C-4 The title of Figure C-2 should read "Figure C-2. Meteoroid Influx Rate Circular Orbit Mars", and the title of Figure C-3 should read "Figure C-3. Meteoroid Influx Rate Cruise"

p. C-5 At bottom of page, add the following: "*Within 50,000 km of Mars"

p. C-6 Line 13 should read: "... of low density (ρ_p < 2.4 gm/cm^3..."

p. C-6 Figure C-4. The ordinate "2" should read "100"

pp. C-17 C-21 The figures C-6 and C-7 on pages C-17 and C-21 should be reversed.

p. C-28 The title of Figure C-8 should read "Meteoroid Shield Test Specimen"

p. C-29 The title of Figure C-9 should read "Cutaway of Meteoroid Shield Test Specimen

p. C-34 In Section 1.8 the first sentence should be replaced by the following two sentences: "Preceding sections of this appendix contain derivations of the probability of penetrations of the spacecraft outer skin by meteoroids. It is clear that to design an outer skin of sufficient thickness to reduce the probability of no penetrations to a low level, such as 0.05 to 0.01, would be prohibitive in terms of the weight required."

p. C-35 In the first equation, the expression "(t in m^2)" in two places should read "(t in cm)" and "A" in two places should read "(A in m^2)"

p. C-38 In Table C-2, all values in inches should be in centimeters. A zero should be inserted immediately following the decimal point, for example: (0.020-inch) = 0.05080, (0.020-inch) = 0.06096, (0.020-inch) = 0.04064, etc.

p. C-40 In Section 1.8.7 Computation of R_i's, the sixth line should read "... than 10^6 are neglected"

p. C-45 In listing under "Values of t Used for Extreme Environment Analysis," under Inch, the first number should read 0.020 instead of 0.202

p. C-52 In 1.10 NOMENCLATURE, "K_2" should be defined as "$K^{-2/3}(4 \pm 2)$" and "B" should be

$$\frac{1000\ \rho_t V^2}{9.806\, H_t}$$

pp. C-150 and C-151 should be reversed.

p. C-208 Along the ordinate in the graph, "Stress $\times 10^{-3}$" should read "Stress $\times 10^{-2}$"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix II

p. F-23 Lines 7 and 10 change all subscript τ to T

p. F-24 Line 14, change "ME_i" to "mE_i"

p. F-29 Figure F-9 title should be "Reflection Phase Angle ϕ (deg)" and Figure F-10 title should be "Reflection Magnitude R"

p. F-30 Last line, change "0.27" to "0.175"

p. F-31 Lines 14 and 15, change "14,700 ft/sec to 460 ft/sec" to 14,700 ft/sec minus 460 ft/sec" and "14,700 ft/sec to 10,000 ft/sec" to "14,700 ft/sec minus 10,000 ft/sec"

p. F-32 Last line in item 4), change "27 per cent" to "17.5 per cent"

p. F-35 Table F-4, under Assumed Parameter for item 2 insert "$\pm 2 \times 10^{-5}$", for item 3 insert "$\pm 3 \times 10^{-5}$", and for item 4 insert "$\pm 2 \times 10^{-5}$"

p. F-53 Item d. Noise Figure, change "4 db" to "3.5 db"; Gain, change "20 db" to "10 db", last line change "10 db" to "4 db"

p. F-58 Figure F-21. Change 102 kc to 112 kc.

p. F-59 Line 22, change to "M_1 = 21.5 deg or 0.375 radians (rms, peak)"

p. F-60 Line 2, change to

$$"M_2 = \sqrt{(1.1)^2 - (0.375)^2}"$$

p. F-60 Line 3, change to "M_2 = 1.03 radians (rms) or 1.46 radians (peak)"

p. G-6 Paragraph 1.4, second line, change "from $E_M = 10' E_o$ to $10^4 E_o$ . . ." to read "from $E_M = 10^{-1} E_o$ to $10^4 E_o$ . . ."

Volume 6. <u>Operational Support Equipment</u>

p. 25 Figure 6. Caption should be "Typical Grounding Scheme"

p. 39 Section 1.3.3, change opening of first sentence to read "Launch pad equipment consists of the ground power and RF consoles and the test flight program power and control equipment . . ."

p. G-31 Figure 1. Lines enclosing Data Format Generator should be solid.

p. G-102 Last line substitute "4500" for "45"

p. G-113 In Section 4.4.2, change "25 per cent" to "250 per cent"

p. G-184 Section 4.5, substitute "6.5 feet" for "six feet"

p. G-311 Fifth line, change "30 per cent" to "20 per cent"

p. G-398 Section 4.2 should begin with "The hoist beam is . . ."

p. G-419 Second line "4 optical alignment targets" instead of 8. Same correction top of p. G-421.

p. G-423 Section 4.9.2, substitute "20 per cent" for "50 per cent"

Volume 7. <u>1969 Flight Test Spacecraft and OSE</u>

p. 90 First line should read "Launch pad equipment consists of the ground power and RF consoles and . . . "

p. 107 Last line, change Volume 5 to Volume 6.