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VOYAGER DESIGN STUDY

VOLUME VI:

PROGRAM PLANS

(NASA

Oct. 15, 1963

(NASA NASw-696)

Prepared Under Contract NAS W-696

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for

no index

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

OFFICE OF SPACE SCIENCES

WASHINGTON, D.C.

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3536000

GENERAL ELECTRIC Co.

(2) MISSILE ~~AND SPACE~~ DIVISION

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VOLUME SUMMARY

The Voyager Design Study report is contained in six volumes, an appendix, and subcontractor reports. The volume numbers and their titles are as follows:

Volume No.	
I	Voyager Design Summary
II	Mission and System Analyses 1. Mission Analysis 2. Parametric System Performance 3. Voyager Systems 4. Reliability
III	System Design 1. Communications 2. Television 3. Radar 4. Guidance and Control 5. Propulsion 6. Power Supply Appendix (Classified)
IV	System Design 1. Entry/Lander 2. Orbiter
V	Sterilization
VI	Program Development Plans

Separate Reports from the following Companies are also included:

Aerojet-General Corp.
Barnes Engineering
Bell Aerosystems Co.
Conductron Corp.
Electro-Mechanical Research Inc.
General Electric Co.
    Light Military Equipment Dept.
General Precision Inc.
Hazeltine
North American Aviation Inc.
    Autonetics Division
    Rocketdyne Division
Radio Corporation of America
Rocket Research Corp.
Texas Instruments Corp.
Thiokol Chemical Corp.
    Elkton Division
    Reaction Motors Division

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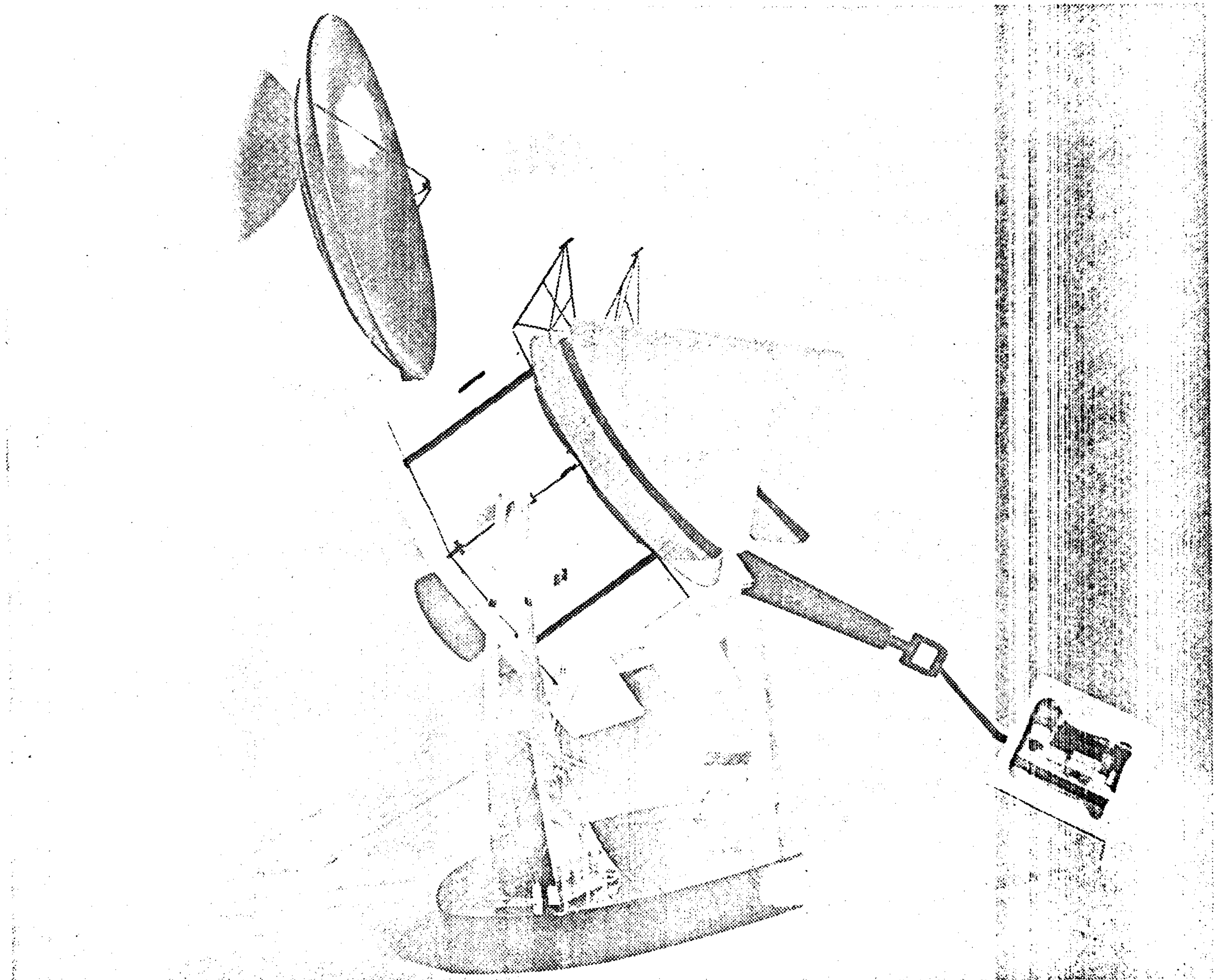


Voyager Spacecraft Model

VOLUME VI

CONTENTS

CONTENTS (Cont'd.)

ILLUSTRATIONS

TABLES

SECTION 1
INTRODUCTION

1.1 THE VOYAGER MISSION

The Voyager mission objective (Mars, 1969) is to provide scientific information about the planet Mars. This mission will acquire biological, geographical and geological data and increase our knowledge background for future manned missions.

1.2 THE VOYAGER SYSTEM

This mission will be performed by the Voyager Spacecraft System and its associated Ground Support System. A concept of the Spacecraft appears on the frontispiece of this volume.

The Spacecraft will consist of two major types of structural elements; the Orbiter Spacecraft, whose function it is to transport two Lander Spacecraft to a point 2 x 10^6 nautical miles from Mars where they will be ejected and brought to a soft landing on the planet. The Orbiter will then be injected into orbit about the planet, acquiring and relaying data to Earth and commands to the Lander for the duration of the mission. The scientific data will be acquired by a variety of sensors on both the Orbiter and Lander, processed and transmitted to Earth.

Major elements of the Orbiter and Lander Spacecraft Systems are shown on the diagrams presented in Figures 1.2-1 and 1.2-2.

These systems and the factors involved in their design, development and operation are technically described in the other volumes of this report.

1.3 THE VOYAGER PROGRAM

The following plan outlines the problems which will be encountered and tasks to be performed in the development, manufacture and test of the Voyager System for the Mars '69 mission.

The scope of development problems anticipated is quite broad due to the wide variety of components and techniques required to implement the system. The wide scope of these problems rather than serious depth of any selected few is the major factor determining program costs and schedule limitations.

Test plans are presented in some detail to convey an accurate picture of the nature and scope of the program.

The effects of Lander sterilization, high reliability requirements, long-term space soak, Martian environment and scientific data requirements, when combined create the distinctive or unusual development problems encountered in the Voyager Program. This plan points out the extent and nature of these problems and outlines approaches to their solution.

The following ground rules and definitions are applicable to the Mars '69 mission:

1.3.1 GROUND RULES

1. Launchings from Atlantic Missile Range (AMR)
2. Sterile Lander Spacecraft
3. Two flights per window
4. One back-up flight unit per window (includes Landers)

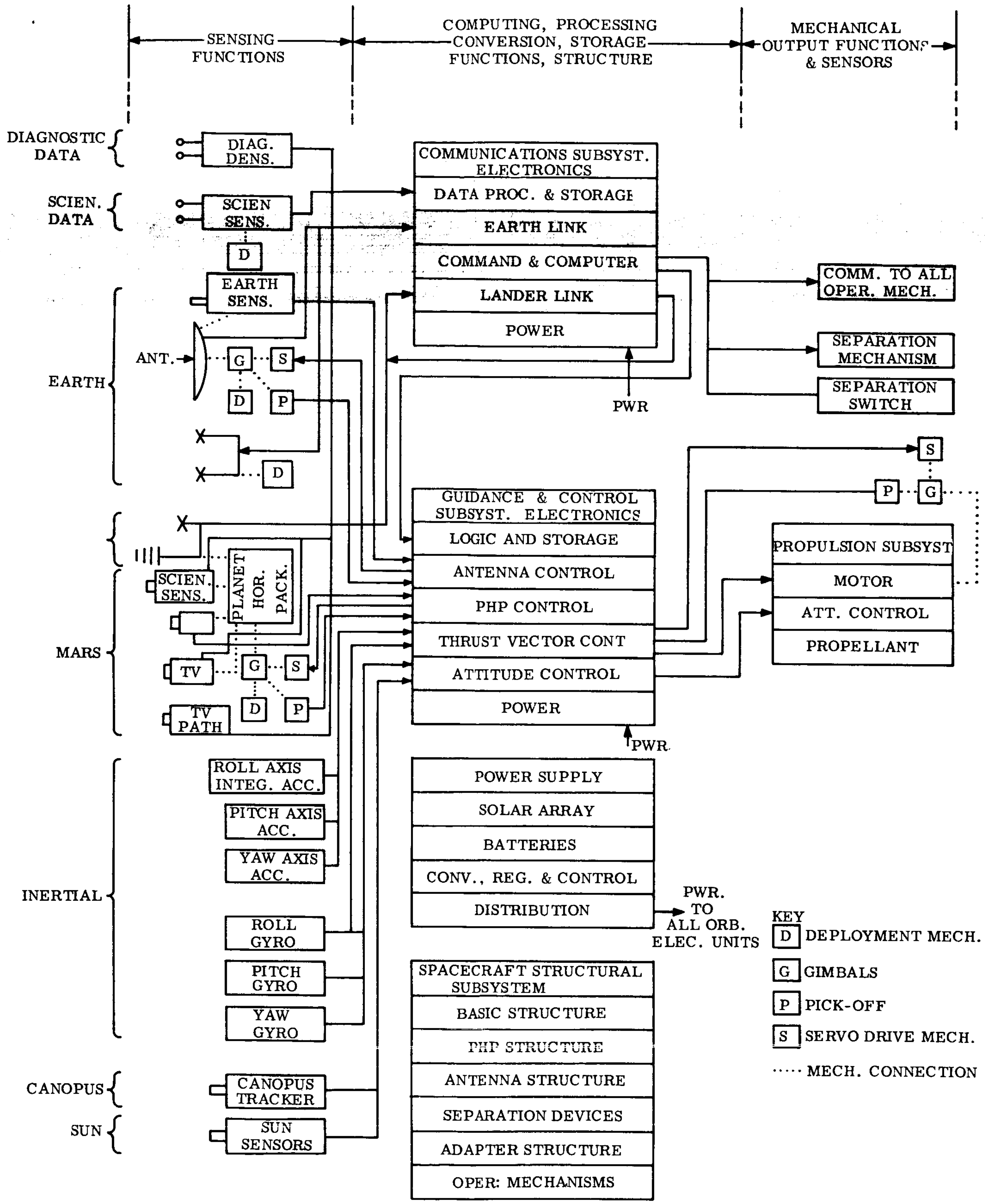


Figure 1.2-1. Orbiter Spacecraft System Diagram

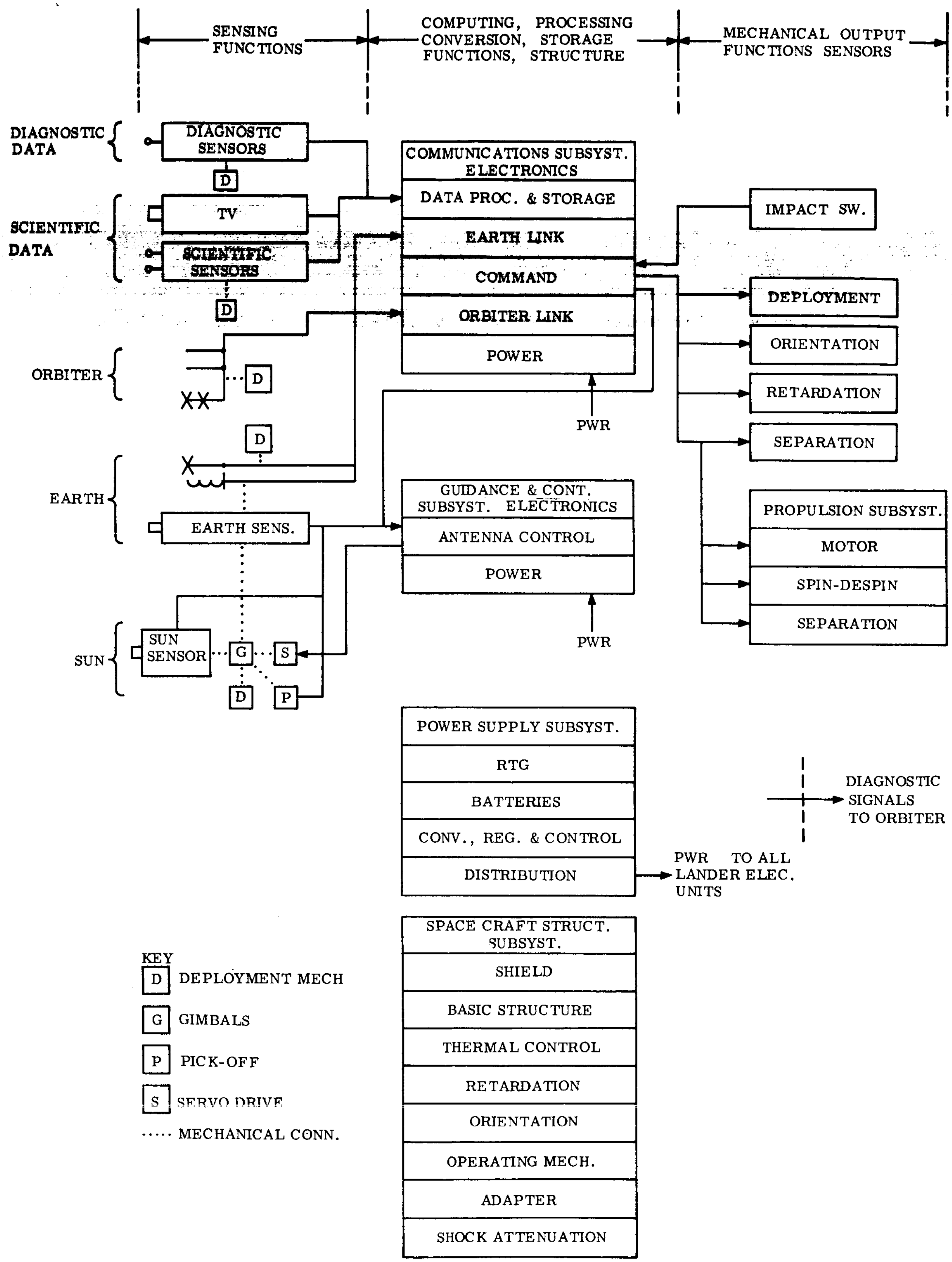


Figure 1.2-2. Lander Spacecraft System Diagram

5. 100% spares per window (includes one Orbiter and one complete, sterile Lander)
6. Three sets of GSE in field, three sets at factory
7. Spare GSE at critical component level
8. All component designs to be qualified above flight levels using two components per design. Quantities for qualification may exceed two for specific types of components (i.e., rocket engines, pyrotechniques, etc.).
9. One Orbiter System will be qualified above flight levels.
10. One complete Lander System will be qualified above flight levels.
11. One complete Voyager Spacecraft System will be qualified above flight levels.
12. Flight units will be in the field four months prior to launch.
13. Each delivered flight component will undergo 150 hours minimum thermal-vacuum acceptance test with last 100 hours failure free.
14. Each delivered flight system will undergo 1000 hours thermal-vacuum minimum acceptance test with last 700 hours at system level failure free.
15. Acceptance tests will be conducted at flight level environments where attainable.
16. Scientific payloads will be qualified at component level and as part of the system at system level.
17. Production of all equipments will be performed together.

A. Definitions

(1) Hardware

(a) Breadboard Hardware

Defines a set of parts and/or components assembled or interconnected to provide design development information on either components or subsystem. The hardware used herein is not usually packaged.

(b) Engineering Hardware

Defines that hardware developed from a breadboard into a packaged design defined by an engineering drawing. This packaged design (drawing and/or specification) must be suitable for use on the mockup and will define prototype hardware.

(c) Flight Hardware

Defines hardware built to final qualified design drawings and/or specifications. This hardware must be procured, fabricated, assembled and tested during the production of material , parts, methods, processes, tooling and test equipment.

(d) Prototype Hardware

Defines hardware built to engineering drawings and specifications by Manufacturing to provide further development testing which will result in the final design of components, subsystems and systems to be subjected to qualification testing.

(e) Qualification Hardware

Defines hardware built to final design drawings and/or specifications which have yet to be qualified. This hardware must be procured, fabrication, assembled and tested using production materials, parts, methods, processes, tooling and test equipment. This implies flight quality hardware.

(2) Systems

(a) Lander Spacecraft System

Defines the spacecraft(s) which separates from the Voyager Spacecraft for entry into the planet's atmosphere.

(b) Orbiter Spacecraft System

Defines the Voyager Spacecraft System less the Lander Spacecraft System(s).

(c) Sub-subsystem

Defines the next level of hardware grouping below the system level which consists of two or more components that provide a definable part of the subsystem performance and can be developed and tested as a group (i.e., the grouping of components that provide the command function are a sub-subsystem of the communications subsystem).

(e) Voyager System

Defines the Voyager Spacecraft System and its associated ground support equipment.

(f) Voyager Spacecraft System

Defines the Orbiter Spacecraft with Landers assembled and ready for launch.

(3) Testing

(a) Acceptance Testing

Defines that level of testing conducted on hardware which demonstrates the suitability of that particular component, subsystem and/or system for flight. This will be successfully conducted on all component and system hardware prior to acceptance-as-satisfactory for delivery for flight.

(b) Development Testing

Defines any and all testing conducted in the process of producing a final design (i.e., this can consist of breadboard, engineering and/or prototype hardware testing).

(c) Evaluation Testing

Defines the continuing development tests conducted to demonstrate performance capability, design margin and failure modes and effects. Testing will be conducted using prototype hardware.

(d) Qualification Testing

Defines tests conducted on qualification hardware at environmental levels above flight levels to demonstrate suitability of the design and the production processes.

(4) Miscellaneous

(a) Component

Defines the lowest level of hardware that constitutes a "black-box" which provides a definable part of the subsystem and/or sub-subsystem performance (i.e., command programmer is a component of the command sub-subsystem of the communications subsystem).

(b) Part

Defines the lowest level of hardware that can be defined by a drawing or specification (i.e., resistor, capacitor, bracket, etc.).

SECTION 2. SUMMARY

2.1 VOYAGER PROGRAM TASKS

The development of the Voyager Systems requires the solution of problems in the selection and application of materials and parts, design and manufacture of components and systems, and development of processes and unique facilities.

The more unique problems can be classified as being the result of one or more of the following:

1. sterilization effects and requirements
2. long-term space soak effects
3. high reliability requirements
4. entry and operation on Mars.

These problems, which are described in further detail in this volume, along with approaches to their solution, are summarized in Figure 2.1-1.

The types of tasks to be performed are indicated by program phases in Figure 2.1-2.

Voyager Program Schedules and Costs are summarized in Figures 2.1-3 and 2.1-4, respectively.

The Summary Schedule, Fig. 2.1-3 shows the Mars 1969 program schedule for two different starting dates. The recommended schedule, shown in wide lines, is based on a Preliminary Engineering Phase starting date of 4/1/64 with Design Engineering starting 10/1/64. This program length of 4-1/4 years from design to launch is consistent with schedule experience on other space programs.

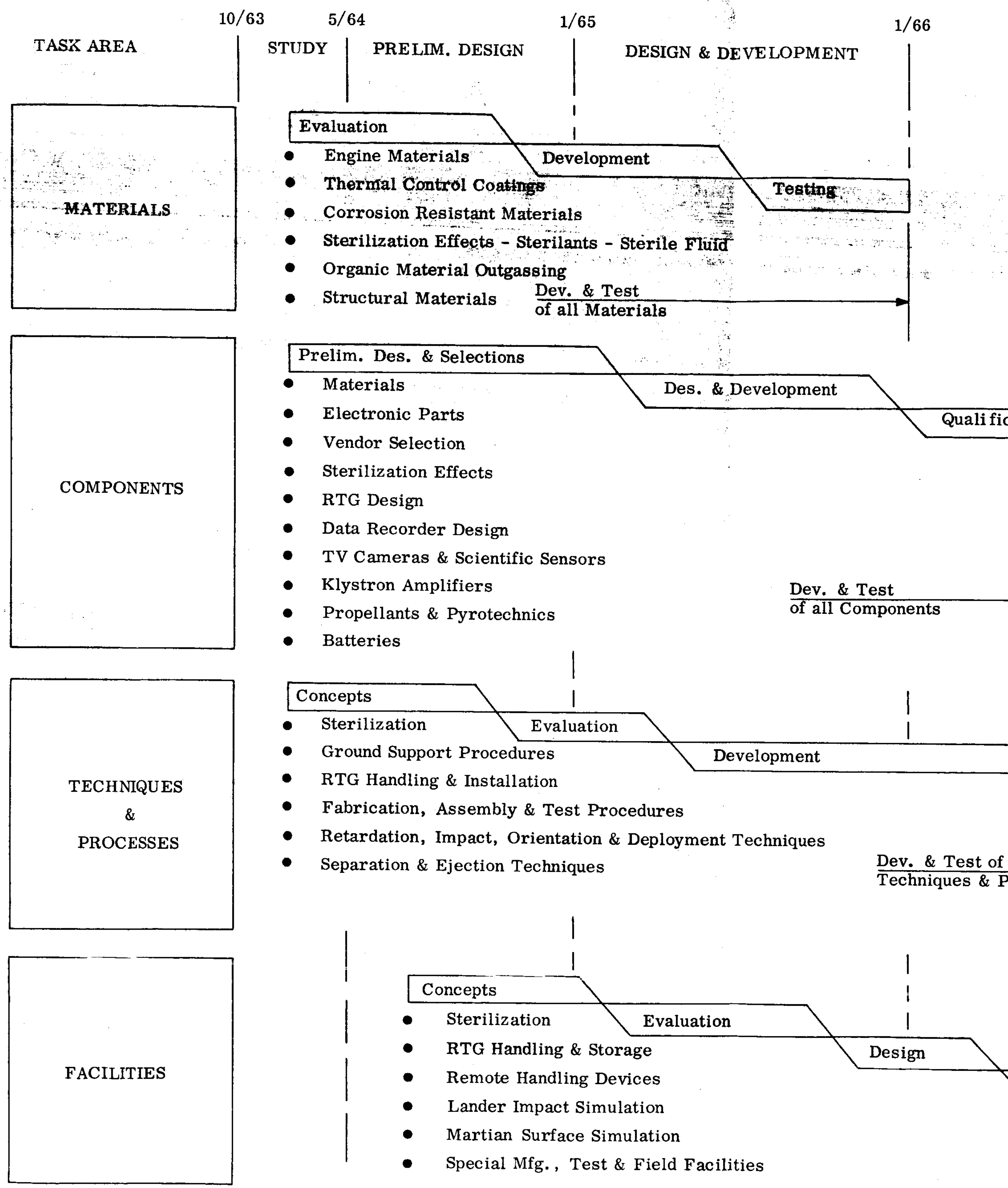
The second schedule, shown in narrow lines, represents a 9 month starting delay with Preliminary Engineering beginning on 1/1/65 and Design Engineering beginning on 7/1/65. This represents the shortest feasible schedule. Any additional starting delay would create a very high risk of failure to meet the Mars 1969 launch window.

DEVELOPMENT PROBLEMS SUMMARY

PROBLEM AREA	PROBLEMS	AFFECTED OR RELATED ITEMS & ACTIVITIES
LANDER STERILIZATION EFFECTS & REQUIREMENTS	EFFECTS ON MATERIALS	TV CAMERA TUBE SCREENS - PARACHUTE MATERIAL - SOLID PROPELLANT - ADHESIVES
	EFFECTS ON PARTS & COMPONENTS	DATA RECORDER - ELECTRONIC PARTS - VIDICON - IMAGE ORTHICON - PYROTECHNICS - SCIENTIFIC EXPERIMENTS
	EFFECTS ON LANDER SYSTEM DESIGN	LANDER STRUCTURE - CHECKOUT & SERVICING PROCESSES & ACCESS - STERILE BARRIER DESIGN - INTERFACES
	EFFECTS ON PROD. PROCESSES & FACILITIES	CLEAN AREAS & PROCESSES - TRAINING - HANDLING & PACKAGING
	EFFECTS ON FIELD PROCESSES & FACILITIES	FIELD ASSEMBLY & CHECKOUT PROCEDURES - FIELD STERILIZATION FACILITY - REMOTE HANDLING
LONG-TERM SPACE SOAK EFFECTS	PROPELLANT CORROSIVE EFFECTS	ORBITER ENGINE RESTART - PROPELLANT STORAGE TANK MATERIALS VALVES & FITTINGS
	BATTERY & PNEUMATIC SYST. LEAKAGE	SEALING METHODS & MATERIALS
	OPERATION OF MECHANISMS	BEARING & ACTUATOR DESIGN - DEVELOPMENT OF LUBRICANTS - BEARING MATERIALS
	SPACECRAFT THERMAL CONTROL	THERMAL CONTROL COATINGS - ACTIVE CONTROL SYSTEM DESIGN
	SIMULATED SPACE ENV. TESTING	THERMAL VACUUM TESTS - STIMULATION OF CONTROL SENSORS
HIGH RELIABILITY REQUIREMENTS	SYSTEM RELIABILITY	REDUNDANCY OF SYSTEMS AND COMPONENTS - ANALYSIS - APPORTIONMENT - TESTS - TRADEOFFS
	COMPONENT RELIABILITY & LIFE	TEST & SELECTION OF COMPONENTS, PARTS & MATERIALS - VENDOR SELECTION & CONTROL - STANDARDS
	MANUFACTURING & TESTING	CLEAN FACILITIES - TRAINING - QUALITY ASSURANCE PROCEDURES - QUALIFICATION TESTS
	HANDLING, SERVICING, CHECKOUT	EFFECTS OF FIELD OPERATIONS ON RELIABILITY - GROUND SUPPORT PROCEDURES & EQUIPMENT
ENTRY TO, LANDING & OPERATION ON MARS	ENTRY OF LANDER	ENTRY SENSING METHOD - PARACHUTE DEVELOPMENT & TESTS - HEAT SHIELD DEVELOPMENT & TESTS
	IMPACT SURVIVAL & ENVIRONMENT	SHOCK ABSORPTION STRUCTURE - ORIENTATION & THERMAL CONTROL DEVELOPMENT & TESTS
	EXPERIMENTS & DATA PROCESSING	DEVELOPMENT & TESTS OF DEPLOYMENT METHODS - DIRECT EARTH COMMUNICATION LINK DEVELOPMENT
OTHER PROBLEM AREAS	RADIOISOTOPE THERMOELECTRIC GEN.	DEVELOPMENT & TESTING - SIMULATION OF RADIOISOTOPE CAPSULE - HANDLING - INSTALLATION - AVAILABILITY
	DATA STORAGE & COMMUNICATION	DEVELOPMENT OF THERMOPLASTIC RECORDER - DEVELOPMENT OF KLYSTRON

Figure 2.1-1. Development Problems Summary

DEVELOPMENT TASK SUMMARY



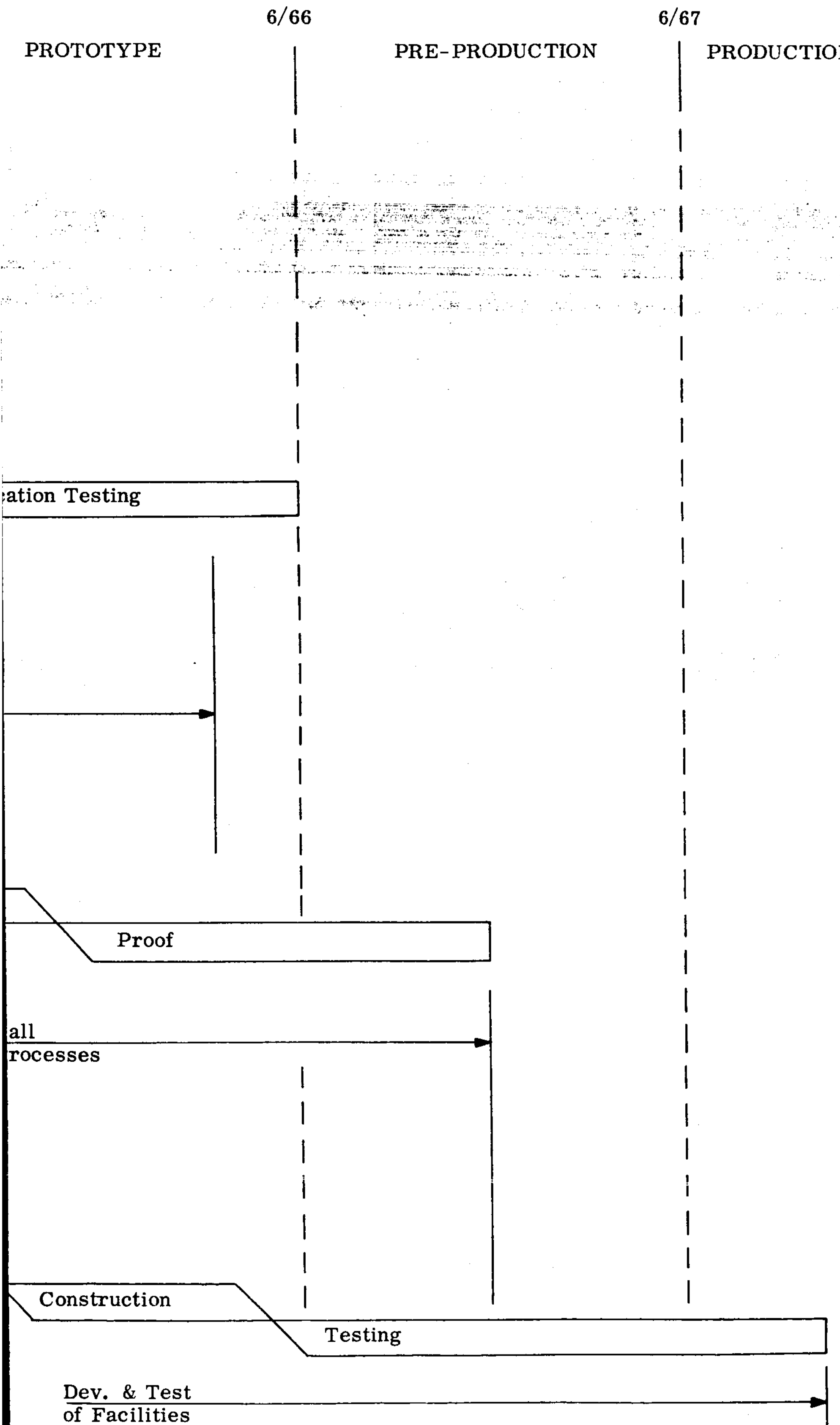


Figure 2.1-2. Development Tas
Summary

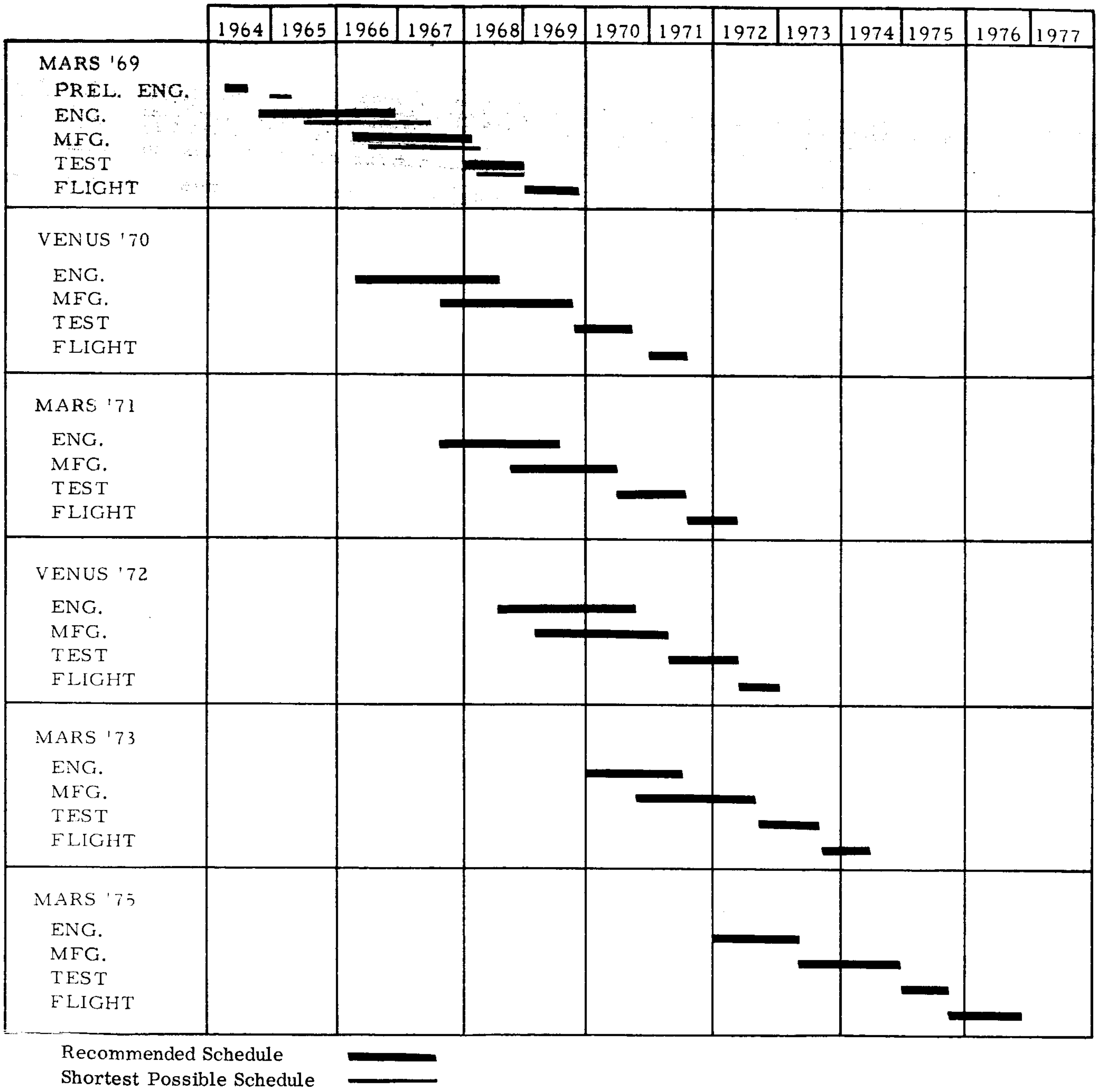


Figure 2.1-3. Voyager Summary Schedule

VOYAGER TOTAL PROGRAM COSTS*

MAJOR TASK	MARS					VENUS			PROGRAM
	69	71	73	75	Total	70	72	Total	Total
Development									
Lander	21.3	6.3	4.6	4.3	36.5	12.7	13.7	26.4	62.9
Orbiter	66.0	9.9	19.6	7.5	103.0	30.0	12.0	42.0	145.0
Total	87.3	16.2	24.2	11.8	139.5	42.7	25.7	68.4	207.9
Production									
Lander	14.2	12.5	12.5	12.5	51.7	4.5	12.7	17.2	68.9
Orbiter	25.0	21.3	18.5	15.4	80.2	21.3	21.3	42.6	122.8
Total	39.2	33.8	31.0	27.9	131.9	25.8	34.0	59.8	191.7
Systems	30.5	10.2	10.8	10.7	62.2	12.5	10.0	22.5	84.7
Support Equipment	23.4	1.5	1.5	0.7	27.1	2.2	0.8	3.0	30.1
Total	*180.4	61.7	67.5	51.1	360.7	83.2	70.5	153.7	514.4

*The MARS 1969 costs as shown above are for a hardware program start date of 10/1/64. A start date of 7/1/65 would result in total MARS 1969 costs of approximately $225 Million.

Figure 2.1-4. Voyager Total Program Costs

SECTION NO. 3
PROGRAM PLANS

3.1 INTRODUCTION TO PLANS

The plans which follow outline problems which will be encountered and tasks to be performed during all phases of the Voyager Program from the Study Program through development and manufacture to launch.

Particular attention is directed to critical or unusual development and testing problems. The comprehensive test plans provided indicate the broad scope of the test efforts required to properly test the Voyager Spacecraft and its many components.

3.2 VOYAGER STUDY PROGRAM - (PART 2) PLAN

The Voyager Study Program should be continued to acquire additional information needed to provide the proper basis for preliminary design activities. In particular, additional studies and investigations of critical components, interfaces and techniques will permit firm system requirements specifications to be established, with a high degree of confidence, so that required critical developments can be completed within the Voyager '69 schedule.

The outputs of Part 2 will consist of:

1. Firm System Functional Requirements Specifications - A complete listing of Voyager systems functions and sequences with sufficient detail to serve as a basis for detailed system engineering work leading to selection of parameters, definition of scientific experiments and firm determination of subsystem requirements. Results include definition of launch vehicle and launch complex interface functional requirements.

2. Preliminary Subsystem Specifications - A preliminary description of all subsystem functions and sequences in sufficient detail to serve as a basis for component requirements determination in the next phase.

3. Refined Program Plans - Refinement of estimates, schedules and plans based on information obtained during this phase.

Figure 3.2-1 outlines the major tasks for this phase.

3.3 PRELIMINARY DESIGN PHASE PLAN

The primary purpose of this phase will be to develop additional engineering information in order to provide a sound basis for flight hardware development which is to be performed during the following phases. The feasibility of various unique components and processes will be firmly established during this Preliminary Design Phase through additional studies, breadboard investigations, tests of processes and critical item evaluations.

The outputs of this phase will consist of:

1. Firm Subsystem Functional Requirements Specifications - A complete listing of all subsystem functions and sequences with sufficient detail to serve as a basis for detailed subsystem engineering, selection of parameters and determination of component requirements.

VOYAGER STUDY PROGRAM - (PART 2) PLAN

PHASE OUTPUT OBJECTIVES	MAJOR TASK AREAS	PROBLEMS
PRELIMINARY SUBSYSTEM AND FIRM SYSTEM FUNCTIONAL REQUIREMENTS SPECIFICATIONS	Data Storage	Reliability, life, sterilization, environment
	TV Camera Tubes	Reliability, life, sterilization, environment
	Klystron Amplifier	Reliability, life, sterilization, environment, weight
	RTG	Handling, installation, shielding, safety
	Propellant Corrosive Effects	Long-life requirements create corrosion problem
	Sterilization Effects	Definition of sterilization effects on materials and parts
	Sterilization Procedures	Definition of sterilization processes to be used
	Launch Vehicle Interfaces	Effects of launch vehicle on system design
	Launch Complex Interfaces	Effects of launch complex on GSE & Spacecraft requirements
	Sterile Propellant	Heat sterilization of solid propellant & pyrotechnics
	Prelim. Subsyst. Functional Req.	Determine feasibility of components & techniques
	System Functional Req.	Determine interfaces & subsystem limitations
	Scientific Experiments	Definition of scientific experiment interfaces
REFINED PROGRAM PLANS	Refined Cost Estimates	Definition of components, materials, subsystem, facilities
	Refined Schedules	System and subsystem definition
	Management Planning	Improved definition of requirements

APPROACH
Thermoplastic recording investigation
Image orthicon development, sterilizable vidicon development
Extensive development and test program
Early consultation with AEC, design studies
Corrosion-resistant material investigations
Studies & tests of effects on materials & components
Studies, experiments, assays & facility concepts
NASA liaison, selection of type of vehicle, study interface
NASA liaison, study AMR facilities & select for Voyager
Continued development, liaison with developing Gov't. agency
Continued studies & consideration of alternates
Continued studies
NASA liaison, TV camera investigations
Utilize information resulting from this phase
Utilize information resulting from this phase
Utilize information resulting from this phase

Figure 3.2-1. Voyager Study Program (Part 2) Plan

2. Preliminary System Design Requirements Specifications - A preliminary description of input, processing and output parameters of all systems, including definition of interface design requirements. This will include system accuracy and performance design requirements in sufficient detail to serve as a basis for subsystem and component design.

3. Firm Program Plans - Firm cost estimates, schedules and plans based on information obtained during this phase.

Figure 3.3-1 outlines major tasks for this phase.

3.4 SYSTEM DEVELOPMENT PLANS

The systems engineering tasks to be performed during the Voyager Program are summarized in Figure 3.4-1, which follows.

3.5 ORBITER SUBSYSTEMS DEVELOPMENT

The Orbiter Spacecraft System is comprised of the following subsystems:

1. Communications
2. Guidance and control
3. Power supply
4. Propulsion
5. Structure

The major problems to be encountered in the development of these subsystems and approaches to their solution follow in Figure 3.5-1 through Figure 3.5-5.

3.6 LANDER SUBSYSTEMS DEVELOPMENT

The Lander Spacecraft System contains the following subsystems:

1. Communications
2. Antenna control
3. Power supply
4. Propulsion
5. Structure

Major problems anticipated in the development of these subsystems and approaches to their solution follow in Figure 3.6-1 through Figure 3.6-5.

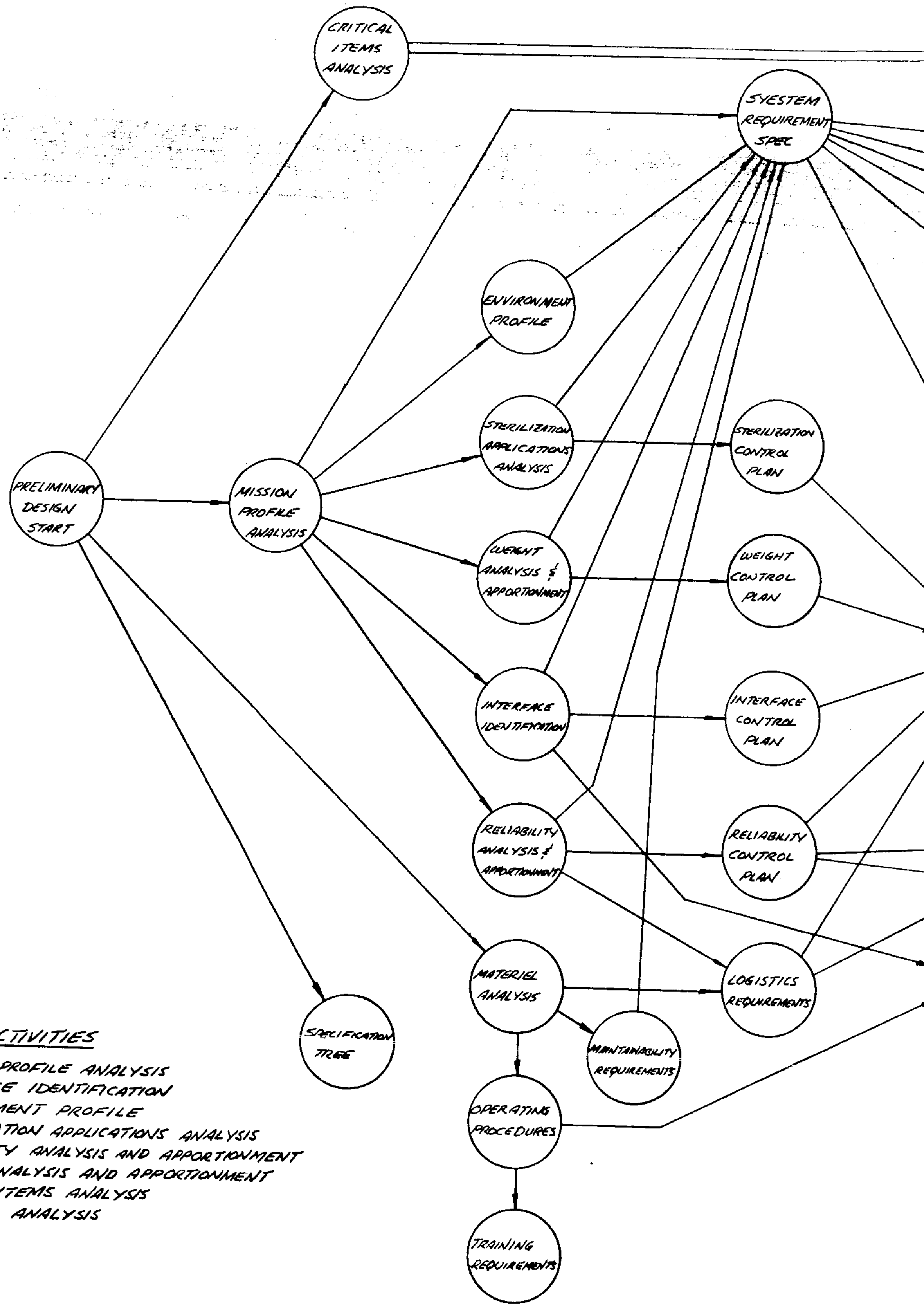
PRELIMINARY DESIGN PHASE PLAN

PHASE OUTPUT OBJECTIVES	MAJOR TASK AREAS	PROBL
FIRM SUBSYSTEM FUNCTIONAL AND PRELIMINARY SYSTEM DESIGN REQUIREMENTS SPECIFICATIONS	Develop Critcial Items	Sterilizable propellant, RTG, Klystron,
	Breadboard Unique Circuits	Data processing & storage, life, reliabi
	Tests of Unique Structures	Structural panels, RTG installation, han
	Tests of Unique Mechanisms	Operating mechanism designs, bearings
	Investigate Novel Processes	Sterilization, antenna fabrication, RTG
	Prelim. Comp. Requirements	Define critical components for purchase
	Subsystem Functional Req'ts	Define system requirements & compone
	System Design Requirements	Define interfaces & subsystem functions
FIRM PROGRAM PLANS	Firm Cost Estimates	Definition of Design Requirements
	Firm Schedules	System & subsystem definition
	Management Planning	Definition of requirements

EMS	APPROACH
TV, data recorder, thermal control	Develop & evaluate critical items
lity, programming	Evaluate experimental components in circuits
dling	Design and evaluate unique structures
, lubricants	Design and evaluate experimental models
installation	Same as above
or lubrication	Improve subsystem design definition
nt limitations	Continue studies & development of critical items
	Continue subsystem engineering & interface studies
	Base refinements on information resulting from this phase
	Base refinements on information resulting from this phase
	Base refinements on information resulting from this phase

Figure 3.3-1. Preliminary Design Phase Plan

VOYAGER PRELIMINARY DESIGN ACTIVITIES FLOW



MAJOR ACTIVITIES

MISSION PROFILE ANALYSIS
INTERFACE IDENTIFICATION
ENVIRONMENT PROFILE
STERILIZATION APPLICATIONS ANALYSIS
RELIABILITY ANALYSIS AND APPORTIONMENT
WEIGHT ANALYSIS AND APPORTIONMENT
CRITICAL ITEMS ANALYSIS
MATERIEL ANALYSIS

VOYAGER PRELIMINARY DESIGN ACTIVITIES FLOW

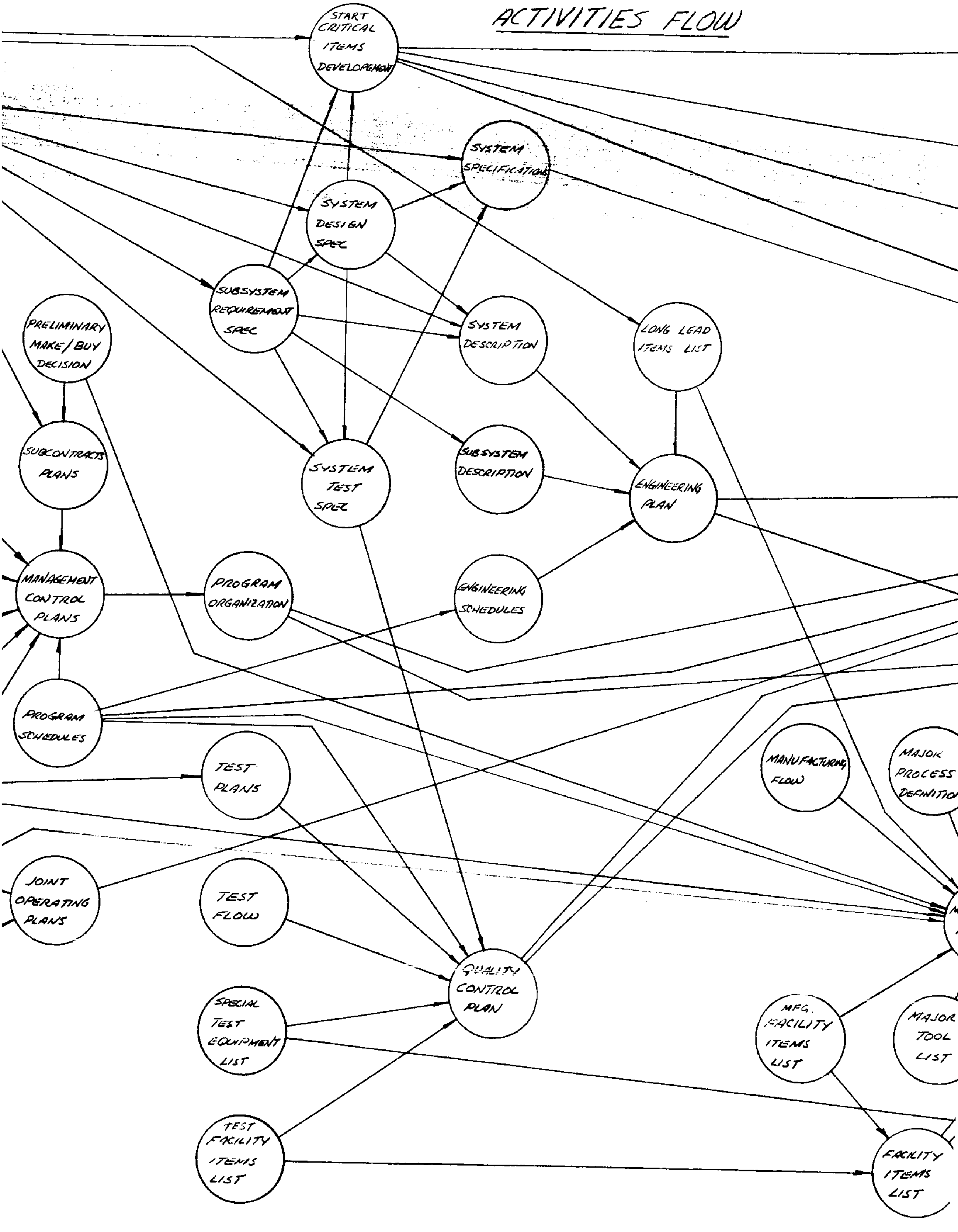


Figure 3.3-2. Voyager Preliminary Design Activities Flow

SYSTEM DEVELOPMENT PLAN

SYSTEM	SYSTEM DEVELOPMENT AREAS	
VOYAGER SPACECRAFT SYSTEM	Definition of Scientific Experiments	Determine effects of experiment design on spac
	Launch Vehicle Interfaces	Selection of vehicle type & definition of interfac
	Launch Complex Interfaces	Determine effects of launch site selection & def
	System Design Integration	Coordination of system requirements, weight c
	System Specifications	Preparation & revision of specifications to refl
ORBITER SPACECRAFT SYSTEM	Definition of System Requirements	Definition of Lander interfaces, launch vehicle
	Subsystem Design Integration	Establish subsystem requirements, weight appo
	Handling & Transportation	Coordination with design of handling, servicing
	Support Operations	Coordinate development of support operation pr
LANDER SPACECRAFT SYSTEM	Definition of System Requirements	Determine effects of experiment & data require
	Subsystem Design Integration	Establish subsystem requirements, weight app
	Handling & Transportation	Coordination with design of handling, servicing
	Sterilization	Coordination of sterilization effects, processes
	Support Operations	Coordinate development of support operation pr
GROUND SUPPORT SYSTEM	Definition of Support Requirements	Coordination with spacecraft design, define de
	Equipment Design Integration	Establish compatibility between various suppor
	Spacecraft Interfaces	Determine effects of detailed definition of inter
	Launch Complex Interfaces	Determine effects of definition of launch compl

TASKS
ecraft design; determine space, weight, power, mounting, deployment & environmental requirements, NASA liaison.
e effects on spacecraft weight design & schedule, NASA liaison.
inition on GSE & spacecraft design & schedule, NASA liaison, coordinate sterilization facility design with NASA & AMR.
ɒntrol, compatibility and integration of Lander & Orbiter systems.
ect actual system design throughout program.
shroud, scientific experiments, command & data link interfaces, GSE interface.
rtionment, coordinate designs, compatibility of subsystems.
& transportation equipment, shipping methods.
ocedures and orbiter spacecraft design.
ments on Lander design, orbiter, shroud & sterile barrier interfaces.
ɒrtionment, coordinate designs and compatibility of subsystems.
& transportation equipment, shipping methods.
& facility development with Lander design.
ocedures and Lander spacecraft design.
sign requirements, prepare specifications.
t equipment designs.
faces and weights on support equipment design & schedule.
ex on support equipment design & schedule.

Figure 3.4-1. System Development Plan

ORBITER COMMUNICATIONS SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
TV CAMERA	Image Orthicon	Environmental effects on tube automatic contr
DATA PROC. & STOR.	Data Recorder	Environment, reliability, life, weight
EARTH LINK	Klystron Power Amplifier	Development program for electrostatic focus
COMMAND	No Critical Elements	No critical problems
LANDER LINK	No Critical Elements	No critical problems
POWER CONV. & CONT.	No Critical Elements	No critical problems

ORBITER GUIDANCE AND CONTROL SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
TV PATH GUIDANCE	Image Orthicon	Dynamic range, data compression, interpreta reliability, life
LOGIC & STORAGE	No Critical Elements	Reliability, life
ANTENNA CONTROL	Earth Sensor	Reflected light at small earth-sun angles, reli
PHP CONTROL	No Critical Elements	Stray IR sources, servo sync. with TV camer
THRUST VECTOR CONT.	No Critical Elements	Disturbance torque from fuel sloshing, CG sh
ATTITUDE CONTROL	No Critical Elements	Reliability, life
POWER CONVERSION	No Critical Elements	Reliability, life

	APPROACH
l circuitry	Development and test program
	Thermoplastic recorder development (GE ATL)
	Development and life-test
	Normal development cycle expected
	Normal development cycle expected
	Normal development cycle expected

Figure 3.5-1. Orbiter Communications Subsystem

	APPROACH
ion, environment,	Experimental development program (GE-ATL) space flight tests (Piggy-Back)
	Normal development cycle expected
ıbility, life	Earth-sun simulation testing required
s, reliability, life	Mars-sun simulation testing required, normal development cycle expected
't, response time, reliability	Analog & physical simulation, fuel & oxidizer tank development, analysis & computer simulation, test firing
	Normal development cycle expected
	Normal development cycle expected

Figure 3.5-2. Orbiter Guidance and Control Subsystem

ORBITER POWER SUPPLY SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEM
SOLAR ARRAY	No Critical Elements	Vendor selection, filter selection, cover th materials, definition of radiation environm
BATTERIES	Batteries	Vendor selection, seal leakage, cycling cap
REGULATION & CONT.	No Critical Elements	Ripple, EMI, effects of radiation
DISTRIBUTION	No Critical Elements	Ripple, EMI

ORBITER PROPULSION SUBSYSTEM

SUB-SYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEM
MAIN PROPULSION	Thrust Chamber	High Temperatures, efficiency, weight
ATTITUDE CONTROL	Controls	Long-term leakage
	Subsystem Development	Verification of I_{sp} for minimum bit
PROPELLANT	Tanks, Valves, Fittings	Corrosion of propellant control compone tankage & expulsion configuration optimi

S	APPROACH
ickness, cell performance, ent	Free-space sun simulation testing, evaluation testing, environmental tests, performance testing
ability	Cell & battery environmental & performance testing
	Tests with breadboard loads, radiation effects testing, normal development cycle expected
	Tests with breadboard loads, normal development cycle expected

Figure 3.5-3. Orbiter Power Supply Subsystem

IS	APPROACH
	High-temperature material development required
	Development of fluid seals or other advanced concepts
	Testing using advanced thrust determination methods
ts, leakage in space environment, ation	Material development required, corrosion and leakage life tests, analytical study of propellant shift with selected control system design

Figure 3.5-4. Orbiter Propulsion Subsystem

ORBITER STRUCTURAL SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
BASIC STRUCTURE	Structural Panels	Honeycomb thin-skin combination, strength, f processes
	Thermal Control	Thermal control coatings, definition of enviro system development
PHP STRUCTURE	Thermal Control	Thermal control coatings, definition of enviro system development
ANTENNA STRUCT.	Antenna Dish	Dynamic load effects (stowed & deployed), the
SEP. DEVICES	Pyrotechnics & Mechanisms	Environmental effects, lander separation torq
ADAPTER STRUCT.	No Critical Elements	Vibration damping, interfaces
OPER. MECHANISMS	Bearings & Seals	Operation following long space soak (non-oper
	Actuators	Operation following long space soak (non-oper

	APPROACH
abrication & assembly	Process development, structural test models, definition of interfaces
nment, active control	Coating development, NASA liaison, analysis computer simulation, design & development tests
nment, active control	Coating development, NASA liaison, analysis computer simulation, design & development tests
rmal distortion, producibility	Analysis, thermal & vibration models
ies, reliability, sterilization	Error analysis, development and tests of materials and working models
	Normal development cycle expected, NASA liaison required
ating), reliability	Materials investigation & development, extensive testing
ating), reliability	Materials investigation & development, extensive testing

Figure 3.5-5. Orbiter Structural Subsystem

LANDER COMMUNICATIONS SUBSYSTEM

SUB-SYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
DATA PROC. & STOR.	Data Recorder	Environment, sterilization, development, prc
EARTH LINK	Klystron Amplifier	Development progress for electrostatic focus
COMMAND	No Critical Elements	No critical problems
ORBITER LINK	No Critical Elements	No critical problems
POWER CONV. & CONT.	No Critical Elements	No critical problems
TV CAMERAS	Vidicon Tube	Sterilization

LANDER ANTENNA CONTROL SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
ANTENNA SERVO	Earth Sensor	Acquiring and tracking earth from lander throu environment, reliability, sterilization

LANDER POWER SUPPLY SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
RTG	Radioisotope Capsule	Establish earth safety & planet contamination grc system design, handling, installation & test proc
BATTERIES	Batteries	Sterilization, leakage, environment, cell tests, v radiation
REG. & CONTROL	No critical elements	Sterilization, ripple, EMI, effects of radiation
DISTRIBUTION	No critical elements	Sterilization, ripple, EMI

	APPROACH
gress	Thermoplastic recorder development (GE-ATL) metal-backed magnetic tape development as alternate
	Development, life-testing and sterilization
	Normal development cycle expected
	Normal development cycle expected
	Normal development cycle expected
	Parts & materials sterilization compatibility development

Figure 3.6-1. Lander Communications Subsystem

	APPROACH
h Martian atmosphere,	Earth-sun simulation tests. Extensive development and testing required

Figure 3.6-2. Lander Antenna Control Subsystem

	APPROACH
und rules, RTG design, coolant edures, shielding requirements	AEC consultation, extensive design & development required
endor selection, effects of	Nickel-cadmium cells, sterilization & environmental tests
	Tests with breadboard loads, determine effects of sterilization and radiation, normal development cycle expected
	Tests with breadboard loads, determine effects of sterilization, normal development cycle expected

Figure 3.6-3. Lander Power Supply Subsystem

LANDER STRUCTURAL SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
ENGINE	Rocket	Sterilization of solid propellant, effects on perfo
SPIN-DESPIN	No Critical Elements	Reliability, sterilization of components, impinge
SEPARATION	No Critical Elements	Separation torques, impingement of particles on reliability

LANDER STRUCTURAL SUBSYSTEM

SUB-SUBSYSTEM OR COMPONENT	CRITICAL ELEMENT	PROBLEMS
SHIELD & BASIC STRUC.	Heat Shield	Re-entry simulation, ablation rate, thermal disto exposure to vacuum, Martian environment, radar
THERMAL CONTROL	Thermal Control	Thermal stability, integration with RTG, coating reliability, life, martian environment
RETARDATION	Parachute	Effects of space environment and sterilization on Martian atmosphere & high velocity
ORIENTATION	No critical elements	Martian surface characteristics & environment,
OPER. MECHANISMS	No critical elements	Deployment of scientific experiments & antenna, surface conditions
ADAPTER	No critical elements	Sterilization, reliability, interface design
SHOCK ATTENUATION	Crushable material	Determination of vehicle dynamics after impact, attenuation materials

	APPROACH
mance, reliability	Extensive development & testing program required, state-of-art advance required
nent of particles on orbiter	Normal development cycle expected
orbiter, sterilization,	Simulated separation tests, determination of separation torques

Figure 3.6-4. Lander Propulsion Subsystem

	APPROACH
rtion, sterilization, long transparent shield	Heat shield materials development, simulation of re-entry & impact, environmental & sterilization tests & further development of ESM material for radar transparency
;, sterilization effects,	Extensive development & testing, simulation of anticipated transit & Martian environments
material, design for	Development & testing of materials, studies of high velocity parachute design, experimental testing
:eliability, sterilization	Development & testing in anticipated Martian terrain
under anticipated Martian	Development & life testing of mechanisms, bearings, seals, lubricants
	Normal development cycle expected
characteristics of shock	Evaluate state-of-art, analysis of potential systems & development tests of most promising

Figure 3.6-5. Lander Structural Subsystem

3.7 INTEGRATED TEST PLAN

3.7.1 INTRODUCTION AND SUMMARY

A. Purpose of Test Plan

The test plan will outline the extensive program which must be undertaken:

1. To prove the feasibility of the proposed designs
2. To demonstrate that the completed units perform their functions successfully under ambient conditions and under space environmental operating conditions
3. To gather data to establish the achieved system reliability.

The test plan will provide design evaluation in depth at the material level, the basic part level, component level, subsystem level, and complete system level. At the completion of the test and evaluation phase of the program there will be reasonable assurance, backed up by many hours of test time and extensive test data, that the spacecraft system is capable of performing satisfactorily in flight.

B. Philosophy of Test Program

The basic philosophy of the proposed test program is to demonstrate by test that the proposed design approach is feasible and will perform satisfactorily, rather than to rely only upon a design analysis that would indicate the proposed approach should work. Therefore, testing and evaluation of all new designs is required to prove performance, and testing of all units is required to gather reliability data and demonstrate achieved reliability.

Testing at the system level of assembly will be conducted for extended periods of time under simulated space environmental conditions to prove performance and to provide reliability data on the long term operation of the spacecraft prior to flight.

For reliability demonstration, available component, subsystem and system test data will be gathered and simulated flight profiles compiled. At the completion of the test program, data will be available showing component and subsystem performance corresponding to an operating mission profile (except space storage) several times as long as operating time expected in flight. From the data on several mission profiles, an initial demonstration of achieved reliability will be made.

Test data will show the performance of the system, and its components and subsystems, rather than just give a functional "go-no-go" indication. The spacecraft will be designed so that accessible test points are included in the circuits. Detailed performance test data on the operation of the subsystem and the components can be gathered with a minimum of breaking into circuit or providing hastily contrived test connection. Availability of the test points will allow thorough and repeated testing in response to programmed stimuli and test sequences. It will be possible to determine performance degradation before it exceeds the specification limits. Thus an early prediction of tendency to vary or an early indication of an impending failure will be provided.

The test program will begin at the parts level, will include component and subsystem development tests, and will be carried through to complete systems tests under flight environments.

The objective of the parts test will be to eliminate from production use the parts which show parameter instability and the possibility of short life in flight unit application. Flight hardware will be built from parts selected for long life capability and screened and operated for an initial burn-in period to eliminate those parts with unstable parameters and the possibility of shortened life.

Extensive component and subsystem development testing will be performed to develop the components and to establish performance.

The in-house test program will be completed by the systems tests in which performance of the complete system will be demonstrated under ambient and thermal-vacuum conditions.

Results of the total test program including systems test data, subsystem test data, and component data will be used to establish the system performance and achieved reliability.

Testing of the spacecraft in the field will establish spacecraft launch vehicle interface, satisfactory survival through the sterilization processes on the Lander, and launch readiness.

C. Major Test Problems - Summary

(1) Sterilization

Achieving sterility and maintaining sterility through the test and launch sequence will be one of the major test program problems. (This is covered separately in Vol. V, Sterilization.)

(2) Field Test & Checkout with RTG Power Supply & Sterilization Requirement

Coupled with the sterilization requirement, the safe handling of fuel without compromising the achieved sterility of the Lander is a major handling problem. Equipment for accomplishing this handling and safe procedures must be worked out during the early design phases of the program.

(3) Thermal-Vacuum Test

Although many hours of thermal-vacuum tests have been completed in various facilities the completion of a long term thermal-vacuum test on a large spacecraft with high level true solar simulation will be a sizeable undertaking.

(4) Separation Tests

Separation tests of the Lander-Orbiter and simulation of zero gravity will be difficult to complete. Several possible approaches to this test are described in the test plan.

(5) Size of the Spacecraft

The large size of the spacecraft will make the accomplishment of acceleration test, humidity test, acoustic test, vibration test, and thermal-vacuum test expensive and difficult. Facilities for these tests have in general been designed for smaller size spacecraft.

D. Voyager Systems Development Test Summary (see Figure 3.7.1-1).

E. Voyager Development Test Summary - Materials, Components and Subsystems (see Figure 3.7.1-2).

F. Voyager Qualification Tests (see Figure 3.7.1-3).

G. Voyager Acceptance Tests (see Figure 3.7.1-4).

SYSTEMS DEVELOPMENT TEST SUMMARY

SYSTEM UNDER TEST	NATURE OF TESTS & DEVELOPMENT ACTIVITIES	PROBLEMS AND TA
VOYAGER SPACECRAFT SYSTEM STRUCTURAL PROTOTYPE	Handling Equip. & techniques	Personnel training, handling equipment suitabili
	Launch Vehicle Interface	Adequacy of access to mountings & service poin vehicle & shroud, fits, tolerances
	Pneumatic Tests	Optimizing routings, mounting of components fu leakage
	Static Load Tests	Adequacy of hard points for handling and accele flections, lateral vibration & handling loads
	Vibration Tests	Vibration levels for components & systems, str transmissibility, physical size of structure to b
	Acceleration Tests	Effects of acceleration on structure, physical s
	Separation Tests	Determine adequacy of separation device design
	Sterilization Shroud	Determine adequacy of sterilization barrier and
	Deployment Tests	Verify operation of deployment mechanisms
VOYAGER SPACECRAFT SYSTEM ELECTRICAL PROTOTYPE	Harnessing	Harnessing routings, compatibility of interconn
	Install subsystems	Compatibility of subsystems
	Energize & Check subsystems	Operation of subsystems
	Subsystem performance tests	Determine functional performance of each subs
	Lander sterilization & handling	Evaluate sterilization procedures, effectivenes training
	Evaluation of changes, retrofits	Evaluate design changes and retrofits
VOYAGER SPACECRAFT SYSTEM PROTOTYPE	Thermal mapping	Verification of calculated thermal environment evaluate thermal controls
	Complete system operation	Determine compatibility of systems and test ec
	Mission profile tests	Determine spacecraft performance during sim
	Removal & retest for degradation	Determine degrading effects of simulated spac

SKS	APPROACH
y	Evaluate prototype handling gear & train personnel through handling of prototype, prepare handling specifications and procedures.
e compatibility with launch	Mount assembled spacecraft to launch vehicle mock-up. Determine accessibility, fits, tolerances.
ıctional performance,	Route tubing & assemble pneumatics, perform functional & leakage tests.
ation loads, structural de-	Strain instrumentation and static loading by weights and jacks
ıctural characteristics & ; tested	Mount complete spacecraft in launch position on high capacity shaker facility. Map vibration levels at selected points.
ze of structure to be tested	Investigate use of large accelerator at Sandia or Edwards. Simulate launch acceleration.
	Develop methods for captive testing of spacecraft-adapter, adapter-booster and orbiter-lander separation devices. Perform tests.
shroud separation method	Set up spacecraft in safe area and fire separation mechanism, photographic monitoring.
	Set up spacecraft and exercise all deployment mechanisms, including high gain antenna, PHP and magnetometer boom.
ections	Install component mountings and harnesses, perform continuity checks.
	Continuity checks
	Energize one section or sub-subsystem at a time, check performance, verify operation and test procedures.
stem	Perform tests and record data on all subsystem functions. Simulate solar array & RTG power sources, RFI tests.
, handling and personnel	Remove lander, sterilize, package in sterile barrier, perform required tests, exercise all handling gear.
	Make changes in electrical prototype. Evaluate their effects on performance.
'or components, test &	Install in thermal-vacuum chamber with solar simulation. Perform extensive thermal mapping.
ipment	Operate complete spacecraft system and test equipment to be used in thermal vacuum tests.
ated flight	Reduce chamber pressure, simulate space heat sink and solar input, simulate mission profile with all systems operating in proper sequence. Monitor & record data.
light	Remove from chamber and retest.

Figure 3.7.1-1. Voyager Systems Development Test Summary

DEVELOPMENT TEST SUMMARY

TYPE OF TESTING	SUBJECT OF TEST	PROBLEMS
MATERIALS	Effects of Space Soak	Corrosion problems in long-term storage and ha propellant, bearings, lubricants, seals.
	Thermal Characteristics	Thrust chamber design, thermal control design, shield ablation.
	Sterilization Materials	Selection of materials capable of being sterilize
COMPONENTS	RTG	RTG performance and operational procedures.
	Planet, Sun and Star Sensors	Determine sensitivity, range, response, trackin capabilities
	Solar Cells	Determine cell performance, select filters, cov thickness.
	All Lander Components	Determine effects of heat sterilization on life, r ability and performance.
	Lander Heat Shield	Heat, shield adequacy determination.
	Sterile Barrier	Development of sterile barrier design, interface procedures.
SUBSYSTEMS AND SUB-SUBSYSTEMS	TV Path Guidance	Determine dynamic range, resolution and effecti of presentation.
	Thermal Controls	Develop thermal control designs.
	Spacecraft Structures	Determine structural adequacy (static and dynam
	Thrust-Vector Control	Determine effects of fuel sloshing, CG shifts, servo response time. Develop adequate design.
	Operating Mechanisms	Develop structures, bearings, seals, actuators, gimbals, lubricants.
	Spacecraft Power Supplies	Determine effects of actual loads, regulation effectiveness, EMI.

	APPROACH
illing of	Determine compatibility of titanium alloys with nitrogen tetroxide. Impact & life rests on submerged samples, development & test of bearings, seals and lubricants.
heat	Evaluation of ablative & radiation cooled refractory metals, coated tungsten, coated tantalum, & ceramic insert matls in firing tests. Irradiation testing of passive thermal control ctgs. Shield ablation tests.
.	Evaluation of sterile barrier materials and sterilants
	Tests of RTG will utilize simulated and actual radioisotope fuel capsule. AEC consultation.
g	Tests involving simulations of sun, Canopus earth, mars and star background. Determine effects of reflection and stray sources.
er	Free-space sun simulation, environmental and performance tests
eli-	Testing, design modifications and integration with system tests
	Simulation of entry with radiant energy lamps and a cooling plenum
s,	Tests of materials, attachments, interfaces, installation and removal, support operation procedures.
veness	Space flight testing (piggy back), NASA liaison
	Extensive testing required. Space environmental simulation, integrated testing with RTG, Martian environmental simulation.
ic)	Static load, shock and vibration tests.
	Test rocket firings, suitably instrumented, will be performed.
	Tests of mechanical and thermal models, space environmental life tests, non-operating, followed by operation.
	Testing with breadboard loads, measure performance, ripple and EMI.

Figure 3.7.1-2. Voyager Development Test Summary - Materials Components Subsystems

QUALIFICATION TESTS

CLASS OF ITEMS TESTED	TESTS PERFORMED	PROBLEMS AND TASK
COMPONENTS	Thermal Sterilization	Demonstrate operation after thermal sterilizati
	Gas Sterilization	Demonstrate effects of sterilant gas on compon heat sterilized.
	Shock, Vibration, Humidity, EMI	Demonstrate effects of environments in excess
	Acoustic Noise	Determine effects of noise levels obtained durir
	Thermal Vacuum	Determine operating characteristics in simulat Evaluate thermal balance design.
	Reliability Life	Evaluate drifts or changes, determine failure r reliability.
	Pyrotechnics	Determine suitability of squibs and other electr explosive devices.
LANDER SPACECRAFT SYSTEM	Transportation And Handling	Determine effects of handling and shipping.
	Sterilization Compatibility	Demonstrate performance following sterilizatic
	Environmental	Demonstrate performance following environme
VOYAGER SPACECRAFT SYSTEM	Temperature-Humidity	Demonstrate complete Voyager Spacecraft perf exposure.
	Powered Flight Vibration	Demonstrate complete Voyager Spacecraft perf exposure.
	Shock And Acceleration	Demonstrate complete Voyager Spacecraft perf exposure.
	Acoustic	Effects of sound levels encountered during pow size of vehicle to be tested
	Thermal vacuum	Determine effects of space environment.

	APPROACH
n.	Sterilize at 145°C, for 36 hrs, repeat 3 times, test performance to determine degradation.
nts which cannot be	Sterilize, purge with nitrogen and perform tests.
f flight levels.	Perform transportation and handling tests, shock, vibration, humidity and EMI tests.
g powered flight.	Suspend component in acoustic chamber; apply random noise at levels up to 160 DB. and determine effects on component performance.
d space environment.	Test in thermal-vacuum chamber; simulate solar radiation on externally mounted components.
odes, establish component	Perform mission profile tests under laboratory ambient conditions.
cally activated	Perform tests of resistance, firing, no-fire current, RF hazards, auto ignition, jolt tests, environmental, calorimeter firing.
	Apply shock, vibration and altitude environments to spacecraft container, with spacecraft packed for shipment. Demonstrate operation following exposure.
cycle.	Sterilize lander and perform complete performance demonstration.
tal exposure.	Perform temperature cycling, simulated flight shock and vibration tests.
rmance following	Perform temp-humidity cycling and performance demonstration tests.
rmance following	Perform test with components energized which would be operative during powered flight.
rmance following	Perform test with components energized which would be operative during powered flight.
red flight, physical	Suspend spacecraft in acoustic chamber and simulate powered-flight noise levels. Evaluate performance following test.
	Perform 1000-hr thermal-vacuum test. Measure performance during and after exposure.

Figure 3.7.1-3. Voyager Qualification Tests

ACCEPTANCE TESTS

ITEMS TESTED	TESTS PERFORMED	PROBLEMS AND
COMPONENTS	Inspection	Locate visible defects & damage; determine
	Ambient Performance	Verify operating characteristics & establis
	Environmental Tests	Demonstrate component performance under
	Reliability Life Tests	Eliminate early failures & unreliable comp changes & drifts, obtain reliability achieve
ORBITER SPACECRAFT SYSTEM	Pneumatic & Harness Checks	Determine integrity of piping & wiring.
	Subsystem Performance	Prove operability of all subsystems and re
	Alignment, Wt. & C.G.	Determine readiness of all sensors & mech tests; determine weight & CG.
LANDER SPACECRAFT SYSTEM	Alignment, Wt. & C.G.	Similar to orbiter above
	Subsystem Performance	Similar to orbiter above
	Vibration	Assure that lander can withstand flight vibr
VOYAGER SPACECRAFT SYSTEM	Mating Checks	Compatibility of launch vehicle, shroud and
	Support Equip. Compatibility	Compatibility with ground support equipmen
	Wt. & C.G. & Alignment	Determine weight, CG and thrust axis align
	System Performance	Determine response of all control loops and simulated mission performance
	EMI	Determine whether spurious component ope ference are caused by system operation; de
	Vibration Tests	Demonstrate that spacecraft will withstand
	Thermal Vacuum Functional Test	Measure performance during simulated pla
	Functional Performance	Determine changes produced by thermal va
	Alignment Recheck	Determine whether alignment has been dist

'ASKS	APPROACH
weight.	Use inspection procedures. Record weight in weight control log.
a performance base.	Set up & operate in accordance with specification. Set up test equipment & perform tests.
environmental conditions.	Perform vibration & thermal vacuum tests.
nents; evaluate component ıent data.	Perform mission profile tests in accordance with component specifications.
	Leak tests, continuity tests, visual examination, check of harness lengths & connections
liness for complete system test.	Perform complete subsystem adjustment and tests.
.nisms for complete system	Adjust & align all sensors, mechanisms and mechanical adjustments. Weigh & determine CG.
	Similar to orbiter above
	Similar to orbiter above
:tion levels.	Perform functional checks before during and after flight-level vibration
orbiter-lander interfaces	Mate orbiter & lander, mate to launch vehicle & shroud mockups
;	Exercise all GSE to be used in field (using a spacecraft simulator as necessary) to avoid tie-up of spacecraft
ıent with CG.	Weigh, determine CG using 3 scale method or balancing machine
overall system performance,	Operate complete system using simulated control inputs, insert commands from ground station.
'ation, communication inter- ermine EMI levels.	Operate complete spacecraft in a shielded area; exercise through all normal modes. Monitor noise levels.
light level vibration.	Vibration test along 3 axes; test & inspect for changes due to vibration
ıetary mission.	Install in thermal-vacuum chamber and perform simulated mission profile tests. Simulate high-gain antenna.
uum testing.	Perform complete performance tests at ambient conditions following thermal-vacuum test.
rbed by tests and handling.	Recheck alignments. Prepare for shipment

Figure 3.7.1-4. Voyager Acceptance Tests

3.7.2 HIGH RELIABILITY ELECTRONIC PARTS TESTING

A. General

The testing of electronic piece parts and microelectronic devices together with test data analysis and part handling procedures, will be a major step in assuring part quality and contributing to part reliability. The incoming parts will have been procured to a high reliability specification following careful type selection and will have been supplied by vendors selected for dependable delivery of high quality parts and subject to surveillance by the contractor's quality assurance engineers.

Acceptance of parts and microelectronic devices will be based upon investigation in three areas:

1. Visual and mechanical inspection and measurement of electrical parameters
2. Analysis of vendor-supplied test data by lot and part serial number to assure that:
   a. Each part has passed all vendor tests
   b. No inconsistencies exist in vendor test data
   c. Vendor's final parameter measurements agree with contractor's initial parameter measurements.
3. Screening tests intended to identify parts affected by manufacturing variables which could result in part failure or unacceptable degradation during the required operating life of the part. In these tests, environmental and electrical stresses are imposed on the parts to accelerate the dominant mechanisms, degrading substandard parts without affecting good parts. Acceptance is based upon percentage drift of key parameters. Screening tests are designed also to assure adequate mechanical strength, especially in the case of interconnections on microelectronic devices, both thin film and integrated circuit types.

B. Division of Testing Between Vendor and Contractor

In the following paragraphs are listed tests to be performed on typical parts and microelectronic devices. The division of testing between vendor and contractor is subject to negotiation for each type and vendor. For example, MIL-R-55182, General Specification for Established Reliability Film Fixed Resistors places responsibility for acceptance test (including "screening tests" as described in A. 3 above) on the vendor, reserving the Government's right to perform any of the tests. In procuring parts to this specification for Voyager, the contractor will perform at least the testing and analysis described in A. 1 and A. 2 above plus screening tests on a representative sample. These screening tests will include at least:

1. Temperature cycling - 5 cycles, +150 to -65°C; reject for mechanical damage or resistance change exceeding 0.25%
2. Overload - 2.25 to 5 times rated load for one hour; reject for physical damage or resistance change exceeding 0.25%.

In addition, the contractor may perform any of the specified tests either 100 percent or on a sampling basis if:

1. It is not possible for the vendor to perform all of the tests because of equipment, schedule, or other problems

2. Contractor's test results or analysis of vendor's test data leave some question as to the quality or reliability of the parts.

C. Typical Tests on Production Parts

(1) Semiconductor diode, General Electric MSD specification R4135, high reliability version of commercial types 1N647 and 1N649, 100 percent tests:

1. Temperature cycling - 10 cycles, +175 to -65°C, no power; reject for parameters out of purchase tolerances

2. Leak test - immerse part in dye and water at 500 psi followed by microscopic examination; reject for dye penetration

3. High current forward voltage scope check-reject for high V_F, instability, flutter or other defects

4. Storage life - 250 hours at 200°C, no power; reject for parameters out of purchase tolerances; reject entire lot if 10 percent of parts fail

5. Operation life - 250 hours at maximum rated power, 35 ± 15°C; rejection criteria same as for storage life

6. Back-bias bake - 168 hours at 100°C, reverse biased; reject for parameter drift exceeding specified limits or parameters out of purchase tolerances

7. Temperature coefficient - measure reverse current (I_R) at 0, +25, and +100°C; reject if I_R does not increase with increasing temperature

8. Other sampling tests - In addition to the 100 percent tests listed above are 2.5 percent AQL tests for visual/mechanical defects, physical dimensions, non-critical electrical parameters, variable frequency vibration, vibration fatigue, thermal shock, mechanical shock, constant acceleration, tension, surge current, and lead fatigue. At appropriate points in the testing, parts are measured for critical electrical parameters and rejected if out of purchase tolerances.

9. Radiation Testing - Because of the radiation environments to be encountered during the Voyager missions, radiation screening of semiconductors will be required for selection of individual parts showing adequate radiation resistance. Studies have shown that surface ionization effects can be expected from solar flare protons and that these effects vary widely from part to part in a production lot. Radiation evaluation tests will be performed to define screening conditions and criteria. It is expected that radiation screening will consist of a short time exposure to a Cobalt-60 source with parameter measurements made before and after exposure.

(2) Film resistor, General Electric MSD specification R4314, 1/10 watt, referencing MIL-R-55182, General Specification for Established Reliability Film Fixed Resistors:

(a) Acceptance Tests, 100 Percent

1. Temperature cycling - 5 cycles, +150°C to -65°C; reject for mechanical damage or resistance drift exceeding 0.25 percent

2. Overload - one hour at 5 times rated load; reject for physical damage or resistance drift exceeding 0.25 percent

3. Seal - hot oil bubble test, nominal sensitivity 10^{-5} atm cc/sec

4. DC resistance - within purchase tolerance

5. X-ray for mechanical defects (unless internal construction is visible through envelope).

(b) Acceptance Tests, Sampling

Visual and mechanical examination, resistance-temperature characteristic, solderability, dielectric withstanding voltage, insulation resistance, resistance to soldering heat, moisture resistance.

(c) Life Test: 250 Hours

Life test of 2,000 and 10,000 hours are required for qualification.

(d) Monthly and Quarterly Tests to Maintain Vendor Part Qualification Status

Low temperature operation and terminal strength followed by seal test; medium impact shock and high frequency vibration followed by seal test.

(3) Microelectronic Devices

Typical tests on both passive and active substrate devices will fall into the following classes:

1. Parameter measurements - more complex and critical than for discrete parts because each device tested is a complete circuit; a high level of operator training and test equipment sophistication is required

2. Screening tests to accelerate drift mechanism in resistor, capacitor and semiconductor elements similar to those used on corresponding discrete parts

3. Thermal cycling, centrifuging and impact shock to eliminate mechanical defects such as cracks, scatches and weak lead bonds and welds; microscopic examination where possible

4. Leak tests on hermetically sealed devices; bake tests to guard against coating breakdown on devices coated with urethanes or siloxanes

5. Radiation screening as for discrete semiconductors where applicable.

D. Handling of Parts and Data

To provide reliable testing of the quantities of parts required, minimizing handling damage and testing errors, special handling and test equipment will be used. Vendors will be required to deliver parts in styrofoam or other special containers giving individual protection to each part. In the contractor's parts testing facility following visual/mechanical inspection, high usage parts (resistors, capacitors, semiconductors) will be placed upon specially designed handling fixtures which will mate with automatic test equipment programmed for screening stresses or parameter measurements. Parts will remain on the handling fixtures until ready for transfer in protective (preferably sealed) containers to a bonded stockroom.

The use of automatic test equipment with punched card readout will:

1. Minimize part handling damage
2. Eliminate test errors due to handwritten data or incorrect reading of meters
3. Provide for prompt computer identification of rejects by comparing parameter measurements with fixed limits or calculating percentage drift
4. Facilitate computer studies of parameter distributions, drift characteristics, vendor performance, etc.
5. Provide test data records for retention.

E. Parts Test Equipment

In order to carry out the above described program of parts testing on a production basis to the detailed requirements detailed, it will be necessary to use automatic testing equipment, efficient parts handling equipment, and computers for handling the data. The following type equipment is required:

(1) Automatic resistor test station, automatic capacitor test station and automatic semiconductor test station to measure parameters of parts mounted on test boards, providing visual digital readout as well as punched card test data:

1. Resistor station to measure resistance with 0.05 percent accuracy
2. Capacitor station to measure capacitance at the commonly specified frequencies, also dissipation factor, leakage current, insulation resistance, and dielectric withstanding voltage
3. Semiconductor station to measure transistor leakage currents, breakdown voltage, current gain (DC and pulse) and saturation voltages, also diode forward voltage, reverse current, breakdown voltage and dynamic impedance.

(2) Screening test equipment, with sufficient flexibility to handle the required types and values, to apply temperature and/or electrical stresses to parts mounted on test fixtures:

1. Resistor power cycling - to apply voltage in programmed cycles at room temperature
2. Resistor overload - to apply specified overvoltage for specified time
3. Resistor temperature cycling - high/low temperature chambers, no power applied to resistors
4. Capacitor high temperature operation - to apply voltage in programmed cylces at elevated temperatures
5. Semiconductor back bias bake - to apply specified bias at elevated temperature
6. Semiconductor temperature coefficient - to measure leakage current on parts in temperature chamber at high and low temperatures

7. Semiconductor radiation screening - Cobalt-60 source with associated equipment

8. Microelectronic circuit screening - not yet fully defined but will require high/low temperature chambers wired for applying power and making measurements at specified temperatures, also centrifuge and shock test equipment.

(3) Handling fixtures are required for parts mounting during test cycles to minimize manual handling and facilitate rapid automatic testing of large quantities of parts. These fixtures must be able to withstand the temperature to which the parts are to be subjected without changing their electrical or mechanical characteristics.

(4) Electronic data processing equipment to accept punched card outputs from the automatic test stations and from vendor tests, process the data and provide rapid on-line accept/reject decisions in the form of punched cards and typewritten data sheets.

(5) Other Equipment

1. Seal test - dye penetrant pressurizing equipment for indicating gross leaks and helium mass spectrometer equipment for quantitative measurement of leak rate

2. X-ray equipment for inspection of parts for internal mechanical defects

3. Semiconductor parameter measurement equipment, not normally included in automatic test stations, to measure switching characteristics, hybrid parameters, frequency characteristics and capacitances

4. Microelectronic circuit parameter measuring equipment including XY recorder and sampling oscilloscope for measurement of switching characteristics

5. Parameter measurement equipment for relays, inductive devices and other low usage parts.

3.7.3 MATERIALS DEVELOPMENT AND TESTING

The long term flight of the Voyager system in a space environment will cause a number of material degradation problems. In the following sections the most critical problems for each subsystem are presented and discussed along with a planned approach to solve them. Because of the unique nature of the sterilization problem, planned development and testing programs for this area are presented separately.

A. Orbiter Materials Development

(1) Orbiter Propulsion Subsystem

(a) Thrust Chamber

1) Description of Problem

The high performance (specific impulse) of the liquid propellant rocket engine to be used for mid-course correction and placement of the vehicle in the planetary orbit will result in temperatures at which performance of the ablation thrust chamber currently being developed will be marginal. At wall temperatures exceeding $3000^{\circ}F$ the silica (SiO_2) used in fabrication of the chamber begins to melt and flow. Performance of silicon carbide throat inserts also becomes marginal. The rocket engine represents one of the most critical materials problems because of its weight, which is dependent upon performance and represents a large percentage of the vehicle total weight. Increased performance (lower weight) required increased operating temperatures.

2) Objectives

The materials development program will determine maximum performance available from the most advanced thrust chamber and throat insert materials available, and will provide test data to establish optimum use of available high performance materials.

3) Development Plan

Several of the most advanced thrust chamber materials, including both ablative and radiation cooled refractory metals, and ceramic insert materials will be evaluated in firing tests. Current ablative chambers and coated molybdenum alloy radiation cooled chambers will be used as standards. Coated tungsten and coated tantalum will be evaluated as advanced materials.

(b) Propellant Tank, Propellant Compatibility

1) Description of Problem

Titanium alloys (6A1 - 4V is being considered for use) are subject to a rapid oxidation reaction when exposed to oxidizers under certain conditions. Violent reactions have occurred when titanium vessels have been used to contain liquid or gaseous oxygen. Only a limited amount of evaluation has been done on the compatibility of titanium alloys with the nitrogen tetroxide (N_2O_4) oxidizer proposed for the Voyager engine. Additional testing will be required to demonstrate suitability of titanium alloys.

2) Objectives

The objectives of the test program will be to determine the hazards of storing N_2O_4 liquids and vapors in titanium tanks.

3) Development Plan

Impact tests will be conducted on titanium samples while submerged in N_2O_4. Impact tests, with and without penetration, will also be conducted on titanium containers filled with N_2O_4.

(c) Propellant Tank, Fabrication

1) Description of Problem

The 6A1-4V titanium alloy has been used for high pressure gas storage bottles for various space vehicles. After much difficulty, reliable fabrication procedures have been established for these tanks which are limited in diameter to approximately 15-20 inches. Sealing up to the large diameter with the thin wall needed for the Voyager will require process development with emphasis upon obtaining large diameter, thin-walled hemispheres with adequate weld joint reinforcements. Fusion welding and heat treatment processes to obtain required properties without excessive distortion must be developed.

2) Objectives

Objectives of the tank development program will be to establish processing and inspection requirements for fabrication of a large diameter, thin-walled, spherical propellant tank from 6A1-4V titanium alloy.

3) Development Plan

A systematic, well-planned program will be required to procure and machine forgings, develop welding schedules, define defect limits, and develop a heat treatment procedure. The program must be coordinated between design, manufacturing, and materials engineering. The materials portion of the program will consist of evaluating heat treat response of the forgings, determining notches and smooth mechanical properties of welded joints, after various welding/heat treatment combinations, and evaluating failures of scale model subscale and full scale hardware.

(d) Propellant Expulsion Devices

1) Description of Problem

In a rocket engine such as required for the Voyager, a positive expulsion device is required to cause the propellants to flow into the thrust chamber. Bladders from such materials as butyl rubber and teflon, have been successfully developed, but the state of the art is limited on large bladders. Butyl rubber is not compatible with N_2O_4. Expandable metallic bellows have undergone limited development.

2) Objectives

Objectives of this program will be the development and evaluation of large diameter teflon bladders and the development of techniques for the fabrication of large metallic bellows.

3) Development Plan

The current state of development will be reviewed to select the most promising techniques for fabrication of large diameter teflon bladders, e.g., laminated or nestled bladders. Flexure testing at room temperature and below, along with vibration testing will be used to evaluate the bladders.

A laboratory test, such as a flexure test, will be developed to correlate materials sample tests with behavior in the full scale tests. For the metallic bellows, the program will include selection of a material compatible with the propellants and developments of the forming, brazing, welding, and heat treatment processes required to fabricate the bellows.

(e) Component Corrosion

1) Description of Problem

The propulsion subsystem will contain component parts, such as solenoid injector valves, latch valves, calibrating orifices and expulsion devices, which must resist attack from the propellants for time periods up to one year. Past experience has shown that even small amounts of corrosion products can result in complete malfunction or erratic performance.

2) Objectives

The objectives of this program will be to insure that all materials used in construction of the various components of the propulsion subsystem are compatible with the propellants.

3) Development Plan

Each component will be analyzed to identify all constructional materials, both metallic and non-metallic. Long term (at least 6 months) corrosion tests will be conducted on these materials. In addition, actual hardware will be exposed to propellants for similar periods of time and then will be subjected to a functional test and disassembly for detailed examination.

(2) Pneumatics Subsystem

(a) Leakage

1) Description of Problem

As space vehicle attitude control requirements become more critical with longer life requirements, the leakage rate of pneumatic propulsion gases becomes critical. Current fittings, both AN and MS, have leakage rates which are marginal for current space application requirements.

2) Objectives

Objectives of this development program will be the selection of the optimum sealant for use in threaded joints, establishment of an application procedure, and evaluation of alternate methods such as brazing and welding.

3) Development Plan

Program to select sealants will evaluate various organic adhesives for sealing threaded joints in aluminum and stainless steel fittings. Precleaning techniques, advantages of primer coats, methods of application, and effects of sealants upon contamination level will be determined. Thermal cycling, vibration and particulate contamination tests will be required. For brazing and welding of joints, the program will concentrate upon developing processes which can be used to assemble the pneumatic subsystem within the vehicle.

(3) Power Subsystem

(a) Batteries

1) Description of Problem

Past experience with nickel-cadmium batteries, proposed for use in the Voyager, has shown them to have one rather serious limitation for usage in long life spacecraft. This is excessive leakage in the joint between the case and the positive terminal post. A ceramic insert is required in this joint for electrical insulation. The brazed joint between this insulator, the terminal, and the case has been subject to excessive leakage rates during operation of the battery.

2) Objectives

Objectives of a materials development program for the Ni-Cd battery is to develop a technique for positive sealing of the cell.

3) Development Plan

Various insulation materials, coatings, braze alloy, and brazing techniques will be studied to obtain a joint which will withstand thermal cycling without developing excessive leakage. Helium leak detectors will be used to measure seal leakage.

(4) Structure Subsystem

(a) High Gain Antenna

1) Description of Problem

The high gain antenna for the Voyager is to be a 10-foot parabola constructed from aluminum honeycomb. This antenna must be deployed from its launch position to its operating position during flight and will track the earth during planetary orbiting. This antenna will be one of the largest components deployed in space. Materials problems exist in the design and fabrication of locking and deployment mechanisms. These include corrosion and galling between mating parts sufficient to cause malfunction because of excessive friction. Programs are underway to select materials and coatings suitable for this type application, but it is anticipated that additional work will be required for the antenna locking and deployment mechanism. These problems are in addition to any antenna distortion problems caused by heating from the sun.

2) Objectives

This program will evaluate various combinations of materials, platings, and coatings to select those able to withstand launch vibrations and long term thermal-vacuum exposure without fretting or galling, or cold welding.

3) Development Plan

Hardware simulating that planned for the antenna locking and hardware mechanism will be made from various combinations of materials. Examples of materials considered include bronze, nitrided steels, hard chromium plating, electroless nickel plating, anodized aluminum, and teflon coatings. Test conditions will duplicate the stresses, including vibration, experienced during launch. Surface examination and coefficient of friction measurements will be used to select best combination.

(5) Thermal Control Subsystems

(a) Passive Control Coatings

1) Description of Problem

Organic coatings having the desired balance between emittance and solar absorptance are used extensively to maintain the critical internal temperature of space vehicles. As vehicle life requirements have increased, the degradation of these coatings resulting from exposure to space radiation has become a more serious problem. Limited studies in this area have shown significant increases occur in solar absorptance. For spacecraft with a mission life greater than 6 months, the changes result in undesirable temperature increases.

2) Objectives

This materials development program will determine the effects of space radiation, primarily ultraviolet radiation and protons, upon the thermal radiative properties of potential coatings. Special coatings will also be developed and evaluated. The final result of the program will be the selection of coatings with radiation resistance.

3) Development Plan

Commercial coatings and specially formulated coatings will be tested to determine thermal, radiative properties and processing cycles. Coatings showing applications potential will be exposed to ultraviolet and proton irradiation under vacuum for integrated doses representing the spacecraft mission environment. Thermal radiative properties will be determined after exposure to select the most suitable coatings.

(6) Component Outgassing

(a) Description of Problem

Most organic materials used in spacecraft tend to outgas in the vacuum of space. Considerable effort has been devoted to studying this phenomenon and its effects upon subsequent properties. One important aspect which has not received adequate attention is the effect on transmission of outgassing products which condense upon the spacecraft optics.

(b) Objectives

Materials which outgas and condense upon the lens materials used in spacecraft optics will be determined. The program will result in a listing of acceptable materials whose outgassing products can be tolerated.

(c) Development Plan

Combinations of various outgassing materials and lens materials will be exposed to thermal-vacuum cycles duplicating the expected operating temperatures. Lens materials will be those used in cameras, solar cells, cover glasses, sun sensors, and infrared radiometers. Outgassing materials will include thermal control coatings, adhesives, insulation and other organic constructional materials with line of sight exposure to the optics.

(7) Magnetic Properties

(a) Description of Problem

If the sensitivity of the magnetometer experiment to be utilized is high, it will be necessary to establish the required magnetic properties of materials used in the fabrication of the magnetometer, the boom, and the actuating mechanism. The primary requirement is that any material used must not generate or propagate magnetic fields to such an extent that the magnetometer readings become erroneous.

(b) Objectives

Identify the materials which can be used in the construction of the magnetometer, magnetometer actuating mechanism and support.

(c) Development Plan

The basic approach will be to evaluate existing data to establish a listing of acceptable materials which can be used on the spacecraft. In cases where data is not complete, it will be necessary to evaluate the magnetic performance of these materials in terms of composition, microstructure, and gross configuration.

B. Lander Materials Development

(1) Lander Capsule Shield Material

(a) Description of Problem

Although a number of shield materials have been developed for application to Earth reentry spacecraft, the material for Mars entry shield must be evaluated to assure that it is satisfactory for the mission and the selected entry trajectory.

(b) Objectives

The objective in this series of tests is to assure that the shield material and the design utilization of this material are satisfactory for the mission.

(c) Development Plan

Scale models of the Lander shield and structure will be inserted into the exhaust stream from a rocket in a static test stand firing and the ablation rate and the band temperatures will be measured. Pre-test and post-test measurements of the shield contours and thicknesses will be made.

(2) Materials Study - Shield Thermal Shock

(a) Test Objectives

To assure that the shield material and bond are able to withstand the thermal gradients of powered flight and entry.

(b) Test Description

Flat plate samples approximately 3 feet by 1 foot will be subjected to thermal shock cycles from -100°F to 1000°F in a restraining test fixture. Samples will be instrumented to obtain temperature and stress-strain data. Samples will be visually examined for interbond failures, crazing, etc.

(3) Materials Study - Aft Cover Ablation Tests

(a) Test Objectives

The objective of this test is to obtain ablation data to aid in the selection of an aft cover material.

(b) Test Description

Test samples of flat plates and oval discs will be tested in a low density, low heat rate environment. Measurements of the heat of ablation in various environments will be obtained. Tests will be made in the tandem Gerdian hypersonic air arc, or similar equipment.

(4) Materials Test - Aluminum Honeycomb

(a) Test Objectives

The objective of this series of tests is to obtain design data on the properties of commercial aluminum honeycomb.

(b) Test Description

A series of tests will be conducted on test samples of aluminum honeycomb to obtain data relative to brazing, crush-up, buckling and stress-strain for aluminum honeycomb. These data will be used to substantiate values used in the Lander structural design.

(5) Material Study - Fiber Glass Crush-up

(a) Test Objective

The objective of this series of tests is to obtain data on the crush-up properties of the fiber glass material proposed for use in the crush-up nose.

(b) Test Description

A series of tests will be conducted to obtain crush-up characteristics on fiber glass nose sections. The results of these tests will be used to substantiate the values used in the Lander design. The tests will be made by mounting fiber glass nose sections on the shock test machine and subjecting them to the shocks expected in landing on various types of terrain.

(6) Materials Study - Parachute Fabrics

(a) Test Objective

Determine the strength properties of decelerator material after high temperature soak (heat sterilization) and after long period of high vacuum soak (transit flight).

(b) Test Description

Material samples (fabric and webbing) identical to those selected for use in the supersonic decelerator and main chute designs will be subjected to the high temperatures and vacuum environment. Samples of the material exposed and not exposed to the heat and vacuum environments will be tested to obtain material strength data.

(7) Ethylene Oxide Compatibility Tests

(a) Test Objectives

There is a possibility that ethylene-oxide gas will be used to flush the Lander capsule during the final stages of the sterilization processes or during the pre-launch checks. The ethylene-oxide gas may have deleterious effects on the materials and components exposed to the gas. The objective of these tests is to determine the compatibility of the various materials which will be subjected to ethylene-oxide environment during sterilization operation.

(b) Test Description

Small samples of various materials will be subjected to an ethylene-oxide atmosphere and will then be tested for detrimental changes in properties. The testing must be accomplished in a test laboratory providing suitable handling equipment and safeguards for handling the toxic and explosive ethylene-oxide gas mixtures.

3.7.4 COMPONENT AND SUBSYSTEM DEVELOPMENT AND TESTING

Although the primary design approach will be to use developed components where they are available and will meet the requirements, there will be a number of cases where it will be necessary to develop new units. Some of these development problems are described in this section. Considerable testing will be performed to evaluate the feasibility of the selected designs and to determine the operating characteristics of the components and subsystems. These development tests are listed by subsystems:

1. Structural and Thermal Control
2. Guidance and Attitude Control
3. Electrical Power
4. Communications
5. Propulsion

A. Structure and Thermal Control Subsystem

(1) Spacecraft Structure Development

(a) Early Vibration and Mechanical Analog Modal Test of Voyager System Assembly

Objective: To determine the fundamental response frequencies, transmissibility and damping of the structure.

As early as practicable during the design phase of the program, a development test Orbiter structural subsystem will be manufactured and subjected to a mechanical analog model test. The test unit will be a rough structural model as nearly representative of the flight hardware as possible at this early stage of the program. Mass items (black boxes, propulsion tanks, etc.) need not be dynamically simulated; however, the dummy masses used should be such that the dynamic characteristics of the Orbiter structure are not altered.

Of the design changes which will occur as a result of other testing of components or of design refinements, only those involving no time penalty will be incorporated prior to conducting this early vibration test. The mechanical analog test, which is of a non-destructive nature, consists of a constant-force sinusoidal sweep from low to high frequencies; i.e., 5 cps to 2,000 cps, introduced along each of the three mutually perpendicular axes. The peak force will be only a fraction of the unit weight, thereby resulting in relatively low stress levels. The unit will be adequately instrumented with 3-axis accelerometers to obtain the fundamental modal shapes, modal frequencies, transmissibilities and dampings. These characteristics will be used to verify and improve the mathematical model of the Orbiter and adapter, thereby permitting refinements in the dynamic analysis and aiding in any succeeding design changes.

At the conclusion of the low-level testing, the structural subsystem will be subjected to higher level inputs directed along the roll axis (launch direction) only. These loads will include both sinusoidal and random inputs and will approach the expected flight levels of vibration, if it has been determined from the low-level data that these levels can be withstood by the structure. The higher level testing will give valuable information on the change of damping characteristics under higher loads and of the overall adequacy of the design concept.

(b) Static Test of Voyager Spacecraft Adapter Assembly

The same preprototype Orbiter adapter assembly used for the mechanical analog test described in 3.7.4A.(1)(a) will be subjected to a static test. Of the design changes which will ultimately occur as a result of static and vibration tests of components (see 3.7.4A.(2)), only those involving no time penalty will be incorporated prior to conducting this static test of the Orbiter adapter assembly. This static test will be only to limit load levels; and hence, will be essentially non-destructive. Probably only one loading condition will be simulated - that combination of axial and lateral accelerations considered most critical. Only a small amount of instrumentation will be used since the purpose of the test is to verify major load paths and to uncover any potential problems due to excessive deflections. An additional reason for minimizing the instrumentation and loading conditions is to make the results available as soon as possible so they may be incorporated in the prime design model used for final dynamic testing.

(2) Structural Component Testing

(a) Static Tests

Static tests of basic components of the Orbiter structure and adapter will be conducted to verify their stress and deflection analyses. Included will be:

1. Honeycomb shear panels and attachments

2. Pneumatic, fuel and oxidizer tanks

3. Tank trunnion fittings

4. Antenna support fittings

5. Lander tie-down fittings

6. Adapter attachment fittings

7. PHP attachment fittings

The components will be loaded in increments of the basic steady-state loads to failure. Adequate instrumentation will be provided to establish both deflections and strain distributions and levels. Test fixtures will be designed to properly simulate the conditions existing in the actual structure. The tests will be conducted at room temperature, but results will be corrected to the expected temperature in service. These static tests will be conducted as soon as possible in the design phase. This will permit any design changes made as a result of these static tests to be incorporated prior to conducting the prime design dynamic test, since it will be a requirement that this test structure is representative of flight hardware.

(b) Vibration Tests

Vibration testing of structural components which were not included in the early vibration structural subsystem, or have changed significantly since then shall be tested prior to the prime design dynamic test. These tests may include:

1. Component Mounting Panels

   Special attention will be given to evaluating the vibration levels that concentrated masses of components mounted on the panels will experience.

2. High Gain Antenna

3. Solar Cell Mounting Panels

Inputs will be determined from the mechanical analog test and sufficient instrumentation provided to insure obtaining enough design information so that these components can survive the prototype testing.

(3) High Gain Antenna Deployment Mechanism

Performance tests will be run to prove the performance of the high gain antenna deployment mechanism. Power input to the actuating mechanism must be measured. Operation of the actuation mechanism and the latching and release mechanism must be evaluated. Additionally, a materials compatibility and bearing problem study to determine long term cold welding, lubrication, and materials deterioration characteristics under conditions of high vacuum must be made.

Development work will be aimed at the detail bearing problem, actuator and release and latching mechanism problems; then as these components and materials are developed, the entire antenna deployment subsystem will be performance tested.

(4) Magnetometer and Boom and Mechanism

Operation of the magnetometer boom unfolding mechanism will be demonstrated. Bearing and material compatibility conditions will be evaluated. Long life evaluation of the mechanism which will periodically flip the magnetometer through 180° will be made.

A careful evaluation of the magnetic fields set up in the actuating mechanism will be made. Because of the low level magnetic field measurements to be made by the magnetometer, it will be necessary to make the field measurements in a special test facility set up with Helmholtz coils to cancel the Earth magnetic field.

(5) PHP Deployment Mechanism

The PHP deployment mechanism, requiring a double gimbal operating mechanism, will require a detailed evaluation. Bearing and lubrication effectiveness will be evaluated. Performance of co-axial cable connection between the PHP and the Orbiter will be measured, with special consideration being given to RF noise and signal attenuation characteristics. Performance of the actuation mechanism and reliability and wear characteristics must be measured.

(6) Lander Structural Static Load Test

(a) Test Objective

The objective of this test is to verify the structural adequacy of the Lander vehicle when simulated mission loads are applied.

(b) Description

A Lander structure will be placed on a load jig and instrumented to record strain. By means of hydraulic cylinders, loads will be applied until failure has occurred. From the test data, verification of the buckling instability analysis, and predicted effects of localized loads will result.

(7) Shield and Structure Thermal Distortion Test

(a) Test Objective

The objective of this test is to verify the adequacy of the shield and structure during the Mars entry condition.

(b) Description

By means of radiant energy lamps and a cooling plenum, the thermal gradient existing during the de-orbit and entry mission phase can be simulated. Strain gauges installed on a shield and structure assembly subjected to radiant energy heating and visual observation will be used to indicate satisfactory design.

(8) Impact and Crush-Up Tests

(a) Test Objective

The objective of this test is to obtain crush-up data and dynamic response characteristics of typical nose dome assemblies.

(b) Description

A total of 10 crush-up nose and side drops will be performed on structures having the correct weight and c.g. Two Lander structures will be dropped on rocky terrain at a nominal vertical 40 feet/second while two will be dropped in sandy terrain at 40 feet/second. If damage is negligible, an additional impact at 60 feet/second will be performed.

The structures will be instrumented with strain gauges and accelerometers to obtain dynamic response data.

(9) Lander Structure Vibration and Shock Test

(a) Test Objective

To evaluate the structural effects of dynamic vibration and shock environments.

(b) Description

A complete Lander, with structurally representative inoperable components and near-prime harnesses, will be mounted on an electrodynamic shaker and subjected to dynamic vibrations in the longitudinal and lateral directions. Following the vibration environment, the Lander shall be shocked for a total of 6 milliseconds in the longitudinal direction and 6 milliseconds in the lateral directions.

Low level vibration tests will be conducted to establish the resonance, bending modes, and transfer functions. Following the low level sine survey, vibration and shock testing will be conducted as indicated below.

(c) Test Equipment

1. The tests will be performed with a vibration test system having a sine and random test capability and the available force output equivalent to an MB-C-200 or C-210 system.

2. A rigid, light-weight vibration fixture will be required to support the Lander.

3. Accelerometers, amplifiers, power supplies and recorders are required to record the test data.

(10) Lander Structural Acceleration Tests

(a) Test Objective

To determine the structural adequacy of the Lander when subjected to the acceleration forces representing powered flight and re-entry conditions.

(b) Description

A structurally prime Lander, complete with inoperable components installed in an acceleration fixture and mounted on a centrifuge, will be operated such that acceleration forces representing critical powered flight and entry conditions can be simulated. Powered flight acceleration loads of 25g and entry loads of 125g are desired. Strain gauges and accelerometers will be used to monitor selected points in the vehicle.

(Note that this test will be repeated later to appropriate test levels as qualification test of prime production Lander.)

(c) Facilities Required

Hydraulic Centrifuge, Sandia or equivalent.

(11) Lander Supersonic Dynamic Damping Tests

(a) Test Objective

The objective of this test is to obtain experimental supersonic dynamic damping data in order to evaluate the design of the Lander.

(b) Description

A stainless steel scale model of the Lander capsule will be built and instrumented for wind tunnel testing. The model will be tested over a range of attack angles and at several possible Mars conditions. Testing can be acomplished at a facility such as the AEDC Tunnel B, Tullahoma, Tennessee.

(12) Lander Hypersonic Dynamic Damping Tests

(a) Test Objective

To obtain experimental dynamic damping data in the hypersonic regime.

(b) Description

Essentially, this test is an extension of the supersonic damping test into the hypersonic regime. Again, several attack angles and several possible atmospheres will be tested, using a scale model Lander capsule. Testing can be done at AEDC Tunnel B.

(13) Tip-Over Bar Actuator Support Fittings Load Test

(a) Test Objective

To determine that the Tip-Over Bar Actuator Support Fittings are structurally capable of withstanding loads imposed during the tip-over operation.

(b) Description

A Lander structure or representative element will be installed on a static load fixture and instrumented with strain gauges, stress coat and dial indicators. The fittings will be incrementally loaded to 150 percent load. All associated strains and deflections will be recorded and analyzed.

(14) Rotating Aft Bulkhead Bearing Load Test

(a) Test Objective

This series of tests will determine the friction losses in the Rotating Aft Bulkhead under various load conditions.

(b) Description

A Rotating Aft Bulkhead with its bearings will be mounted in a jig simulating the Lander structure. Various loading conditions will be imposed to simulate anticipated events after impact that will change load requirements. Torque required to rotate the bulkhead will be measured.

(15) Parachute Fittings Load Test

(a) Test Objective

To determine that the parachute fittings are structurally sound when subjected to Mars entry loads.

(b) Description

A Lander structure will be installed on a test fixture and instrumented with strain gauges and dial indicators. The parachute fittings will be loaded hydraulically and all associated strains will be recorded and analyzed.

(16) Lander Separation

Tests of the separation process are made more difficult by the need to separate the effects of the Earth gravity field from the test results. Several methods of performing the Orbiter-Lander separation tests are suggested.

1. The Voyager spacecraft with Landers can be suspended horizontally from flexible supports (as Bungee cord), with the Landers and the Orbiter suspended independently. The Landers must be supported through their center of gravity on a low-friction gimbal ring which allows them to spin up with a minimum restriction. The Orbiter must also be free to respond to the separation force.

   With the spacecraft supported as above, the separation spin-up mechanism is fired and the separation is monitored with high speed cameras. Measurements are made with accelerometers and other supporting instrumentation, which will measure the acceleration imparted to the Lander and the Orbiter. De-spin is monitored similarly.

2. Another approach to measuring separation, spin-up, and de-spin is to use a gimballed hanger to counterbalance the Lander vehicle on a test jig. The hanger will enable the Lander to "spin-up" while the net rocket separation force produces vertical rise of the counterbalance system. The interface between the Lander and test jig will be a mating ring employing pyrotechnic devices. Cameras will be used to record actuation of the mating ring disconnect sequence and the firing of the spin and de-spin rockets.

   This test will give a good picture of the separation and indicate any "tip-off" angle, but it will be difficult to demonstrate the momentum imparted to the Orbiter.

   The above tests must be performed in an explosion safe area.

3. Another feasible approach is to conduct the separation test from a drop tower during a free fall, thus minimizing the effect of the Earth gravity force on the test results.

   The Lander will be mounted to a structure representing the Orbiter interface and mass. The Lander and Orbiter model will be released from a drop tower in free fall. Upon command, the Lander will be ejected and spun-up by rockets. High speed cameras will record the sequence and measure the spin-rate. The Lander and Orbiter will be caught in nets. The tests will be performed on an early structural model of the Lander with a good operating separation and spin-up mechanism, rather than tie-up a complete functional unit for these tests.

   Tests will be conducted at a location such as the Sandia, New Mexico 180-foot drop tower, or an equivalent facility.

(17) Decelerator Ejection Tests

(a) Test Objectives

1. Verify that ejection charge weight is sufficient to attain required ejection velocity.

2. Evaluate structural integrity of mortar tube and support structure.

3. Obtain reaction loads during ejection for vehicle structure design.

(b) Description

At least ten ejection tests will be conducted. The test set-up will consist of the mortar tube with decelerator pack and lines mounted in a fixture instrumented to record velocity and reaction loads. Load cells will be used to measure reaction loads. High speed cameras will record the ejection process.

(c) Hardware Required

Ejection mortars	(4)
Dummy Decelerator Packs	(6)
Decelerator Packs	(4)
Decelerator Bridle	(10)

(18) Retardation Electrical System Breadboard Tests

(a) Test Objectives

1. Verify functional compatibility and electrical interface compatibility.

2. Demonstrate performance of the electrical sub-subsystem elements.

(b) Description

Recovery subsystem components (or electrical simulators) will be connected with the Lander electrical harness. Programmer input signals will be simulated and outputs monitored for correct sequence, timing and voltage levels.

(19) Retardation System Drop Test

(a) Test Objectives

1. Demonstrate system performance capability under simulated Mars entry conditions (i.e., high altitude, high velocity, low and high q).

2. Demonstrate structural integrity and system performance capability of components which were subjected to environments of sterilization and long vacuum soak.

(b) Description

Six successful high altitude system drop tests are to be conducted using helium filled balloons for obtaining ascent altitude. Drop test payload will be rocket boosted into Earth's upper atmosphere to obtain low density-high velocity deployment conditions.

The initial four tests are to be conducted on a system which has not been subjected to sterilization or vacuum soak. The final two tests will be conducted on retardation components which were exposed to sterilization and vacuum environments.

Tests can be conducted at Halloman AFB.

(c) Hardware Required

Drop Test Programmer	(3)
Boost Rockets (GFE)	
Adapter Section	(3)
Vehicle After Body	(6)

(20) Parachute Proof Test

(a) Test Objectives

1. Demonstrate load capability and system performance prior to conducting high altitude systems tests.

2. Verify peak opening loads for final chute sizing and specified reefing area.

(b) Description

The hyperflo and main chutes shall be tested individually at maximum q conditions. The two chutes as a system shall then be subjected to maximum q conditions to demonstrate successful deployment sequencing.

(c) Hardware Required

GFE Drop Bombs	(2)
Adapter Section	(2)
Hyperflo Chutes	(3)
Main Chutes	(3)
Programmer Circuit	(2)
Deceleration Ejection Mortar	(2)
Ejector Charges	(6)

(21) Retardation Wind Tunnel Tests

(a) Test Objective

To demonstrate the opening stability and performance characteristics of full scale Hyperflo Decelerator Chute.

(b) Description

Six supersonic low density deployment tests are to be conducted which will check canopy opening characteristics particularly under combined low q and high velocity conditions.

(c) Hardware Required

Hyperflo Chutes	(2)
Ejection Mortar	(2)
Test Rig (GFE)	

(22) Orientation and Deployment Test

(a) Test Objective

To demonstrate that the orientation rockets and stabilization arms satisfactorily perform under simulated Martian environment.

(b) Description

A Lander structure will be utilized to demonstrate the operation of tip over rocket, tip over bar, tip bar actuation, and orientation by means of rotating bulkhead. Three tests will be conducted wherein possible Lander attitude positions will be checked. The ability of the tip bar to operate, and success of the rocket tip over process and bulkhead rotation will all be appraised. Rocky, sandy, and combined rock/sand terrains will be utilized.

(c) Hardware Required

Lander structural subassembly with tip bar, arms and rotation rockets.

(23) Lander Environmental Control Subsystem

The Mars Lander environmental control is used to keep the payload temperatures between 60^{o}F and 100^{o}F. It must be capable of providing heat during transit (shade oriented) and the cold Mars night temperatures or provide cooling during the Mars days.

(24) Radiator and Heat Exchanger Performance

(a) Test Objectives

1. The objective of this test is to determine the performance of the various radiators and heat exchanger in dissipating the heat from the RTG and maintaining the temperature of the Lander payload.

2. Verify that the RTG radiator can maintain the Lander at acceptable operating temperatures during the Martian night periods.

(b) Description

The radiator and heat exchanger will be tested under simulated thermal loading and power conditions, including overloads. The tests will be conducted in a thermal vacuum chamber so that the environmental conditions of transit flight and of operating on the planet can be simulated. The tests will include the anticipated Martian night conditions. A Lander vehicle shall be placed within a test chamber and subjected to an absolute pressure of

1.5 psi and $-184^{\circ}F$ for a period of 24 hours. Any components that would be operating during the night will be simulated including the prime heat source - the RTG radiator. A prime RTG will not be used for this test, but will be thermally mocked-up and produce the correct amount of heat. Thermocouples will be used to monitor critical components.

(c) Facilities and Test Equipment

A vacuum chamber capable of reaching a hard vacuum of 10^{-6} torr, or better, and also capable of simulating the Martian atmosphere pressure and temperature is required.

B. **Guidance and Attitude Control Development**

(1) Guidance and Attitude Control Subsystem Development

Development of the guidance and control subsystem for Voyager does not represent an advance in the state-of-the-art, but rather the application of known technologies to the Voyager requirements. Development of the components represents the major portion of the guidance and control development engineering effort.

Special attention will be given to the propulsion - G & C interface and the interaction between control torques during acquisition, cruise and maneuvers, and disturbance torques, and particularly the effects of any cg shift on control of the thrust vector during rocket thrust. Development of a relatively high response thrust vector control to be actuated for several brief periods with long inactive times between, where the total burn time is three minutes or more, and where control of the thrust vector with near negligible reaction torques on the spacecraft is required, represents a considerable development effort.

Development of that portion of the attitude control sub-subsystem to perform acquisition, reacquisition, commanded turn maneuvers and limit cycle control may be accomplished using conventional analytical engineering techniques taking into account the effects of cross coupling between axes, the effects of disturbance torques from solar pressure and gravity gradient, the effects of noise on the sensors and servos, the mechanical and electrical null offset of sensors and servos and the reliability and stability of the components and servo loops.

Development of the optimum High Gain Antenna Control Subsystem to position the antenna boresight about two axes will take advantage of the near zero antenna unbalanced loads during the cruise mode which may not be completely duplicated to evaluate the design by testing at 1g. In addition, care must be exercised to take into account sources of reflected light off the spacecraft and solar impingement into the optics of the Long Range Earth Sensor.

Development of the optimum Planet Pointing Package Control Sub-subsystem are akin to those of the antenna control.

Emphasis on the development of the programmer will be on redundancy in design to provide the required reliability.

The major challenge offered by the Lander antenna control subsystem is the development of the servos to acquire the Earth with reliability, and withstand the hostile Martian environment. The Mars dust storms represent an unknown quantity so that protection must be provided for "worst estimate" conditions as far as gimbal drives are concerned, and a command back-up mode must be provided for acquisition if the sun is obscured.

(2) Engineering Approach to Solutions

In general, the guidance and control subsystem will be developed by following conventional engineering analysis and computer simulations to provide subsystem and component requirements and to evaluate designs. Computer simulations will be used to assist in the development and evaluation of the dynamics of the thrust vector control and attitude control loops. Attitude control sub-subsystem hardware will be evaluated on an air bearing simulator. Thrust vector control hardware will be evaluated on a rocket test stand for one and two axes. Discrepancies in results will be resolved by analysis and designs altered where required.

Evaluation of the antenna control loops and Plant Pointing Package control loops will be evaluated a single axis at a time with pseudo zero g along that axis and with a simulated load, to assist in the development of these servos. Effects of solar impingement and reflected light will be evaluated analytically and by test, using actual control hardware in the spacecraft configurations with a Sun, Earth and Mars simulators.

Early build up of a programmer and TV Path Guidance subassemblies for test to gain actual reliability data will assist in achieving a final design of the required reliability. These and other electronic components will be subjected to a burn in period to eliminate early random failure burn outs.

The following electronic components will be designed in breadboard form in accordance with the above requirements, tested, and block and/or circuit diagrams, and test results documented.

1. Logic and switching amplifier
2. Gyro electronics
3. Autopilot Servo amplifier
4. Accelerometer electronics
5. Power conversion subassembly
6. Spacecraft antenna servo amplifiers
7. Planet Pointing package servo amplifiers
8. Programmer (G and C functions)
9. TV path guidance subassembly electronics
10. Lander antenna servo amplifiers

Design layouts of the following mechanical components will be completed, and interfaces with the spacecraft assembly and booster resolved.

1. Gyro module
2. Autopilot servo valves and actuators
3. Integrating accelerometer module
4. Spacecraft antenna servo actuators and pickoffs
5. Plant pointing package servo actuators and pickoffs and cable unwind mechanism.
6. Lander antenna servo actuators and pickoffs

7. Outline of all electronic modules

8. TV path guidance subassembly

The following vendor items will be procured for the purpose of conducting critical tests to substantiate predicted performance.

1. Canopus tracker

2. Gyros

3. Sun Sensors

4. Long range earth sensor

5. Accelerometers

6. TV Path Guidance camera and optics

As testing of the above components proceeds, test results will be evaluated and necessary design changes will be made. The breadboarded components will be designed into their flight-packaged configurations and performance of these units under various test operating conditions and environments will be made. This series of evaluations will result in the prototype unit configuration, which will be qualification tested under flight-type environmental conditions.

Testing will be performed both at the component level and at the complete subsystem level to evaluate the performance interactions and complete subsystems response.

(3) Guidance and Attitude Control Tests

1. Performance tests will be conducted on the G&C attitude control subsystem, except the autopilot, on the bench and on an air bearing with simulated sun and canopus references and gyro references.

2. Performance tests will be conducted on the Hi Gain Antenna control subsystems, on the bench and using a simulated earth and simulated antenna load balanced to permit testing at 1g.

3. Performance tests will be conducted on the Planet Pointing Package Control subsystem on the bench and using a simulated planet and simulated planet pointing package load balanced to permit testing at 1g.

4. Single axis dynamic performance tests will be conducted on the thrust vector control loop of the attitude control subsystem including the gyro references the autopilot thrust vector control electronics, thrust vector control actuators, and rocket engine with the engine firing at nominal thrust level.

C. Electrical Power

(1) Development Plans

The development plans for this subsystem will be discussed, first as they apply to each component, and second in terms of the overall subsystem design and integration. This type of system is basically state-of-the-art, and no major development problems are anticipated. However, careful planning and engineering work will be required to insure that the resulting design represents the best that is possible for this type of system.

(a) Solar Array

Major steps in the development of the solar array include the following:

1. Verification of solar cell performance and vendor selection
2. Selection of filters
3. Selection of cover glass thickness
4. Selection of materials (cover glass, glass to cell bond, etc.)
5. Sub-module and module electrical and structural design

   (A sub-module is defined as a group of solar cells connected electrically in parallel and forming a single structural unit prior to assembly in a module. Ten cells per sub-module is a typical value. A module is defined as a number of sub-modules connected in electrical series and forming a single structural unit.)

6. Environmental testing to insure capability to withstand the expected environment
7. Performance testing to insure achievement of required performance

Verification of solar cell performance and vendor selection will be accomplished by obtaining samples of production lot cells from various vendors and running screening tests such as cell electrical contact termination, spectral response reflectivity, thermal cycling, and V-I curves at various temperatures. These data, when combined with such information as cost and vendor capability, will result in vendor selection.

Filter selection, which will interact with and be affected by cell vendor selection, will involve detailed analytical considerations of effects of various filters on cell thermal and electrical performance as well as performance and environmental testing of cover glass and filter considering such items as spectral transmission, effects of humidity, thermal and thermal cycling effects, and filter adherence.

Cover glass thickness will be determined based on radiation damage calculations and optimization studies of solar array area and weight requirements as functions of cover glass thickness. Because the 1969 flight date is during a period when solar flare activity is expected to be high, radiation damage will be an important consideration, and radiation tests will be conducted of complete cell-cover glass combinations to verify design adequacy.

As a result of past hardware programs, selection of materials is expected to be relatively straightforward with the chief determining factors being environmental considerations such as radiation damage and thermal cycling. Testing, principally of an environmental nature, will be required to verify materials adequacy in the specific design configuration selected.

Major considerations involved in sub-module and module electrical and structural design include: a) Voltage requirements and reliability as they affect selection of number of cells per sub-module, number of sub-modules per module, and interconnection of modules and use of diodes, b) Manufacturability, handling requirements, desire for verification testing of modules, and ease of replacement of defective modules as they affect module size and method of mounting modules on structure, and c) Radiation damage protection and vibration and stress criteria as they affect design of the module substrate structure.

Environmental testing to be performed to insure design adequacy includes, in addition to that previously mentioned, thermal cycling of sub-modules and thermal cycling, thermal vacuum, natural frequency determination, vibration, and acceleration testing of modules.

A major factor in carrying out performance testing is to insure that there is a proper correlation between spectrum and intensity of the light source used for testing and that of the sun in free space. From past programs, satisfactory procedures have been worked up to achieve such correlation. These show good agreement with other investigators and check well with results obtained at Table Mountain. Standard solar cells are used as references and both carbon-arc and tungsten light sources, after proper calibration, are used. The carbon-arc source provides the primary light standard, but the tungsten sources would be used for the bulk of the testing, particularly that involving large areas of cells.

(b) Nickel Cadmium Batteries

Major steps in the development of the batteries include the following:

1. Cell screening tests and vendor selection
2. Cell performance and environmental tests
3. Battery package design
4. Environmental testing to insure capability to withstand the expected environment
5. Performance testing to insure achievement of required performance

Vendor selection will be accomplished by ordering cells from several vendors and running screening tests including capacity, voltage-current characteristics, performance under cycling conditions, seal leakage, and preliminary vibration tests. These data, coupled with information on costs and vendor capability will result in vendor selection.

Following vendor selection, additional cells will be obtained and more extensive testing carried out to determine performance characteristics including cycling, both for individual cells and cells connected electrically into a battery. Testing will include thermal vacuum tests.

Battery packaging design and fabrication is expected to be performed by the battery vendor. Following receipt of complete, packaged batteries, additional environmental testing and performance testing will be carried out to verify the design. Environmental testing will include thermal, thermal-vacuum, vibration, acceleration, and shock.

(c) Battery Charge Regulator and Power Control Units

The basic development programs for these units are the same, so they will be discussed together. Major development steps include the following:

1. Circuit design
2. Construction and test of breadboards
3. Packaging design
4. Environmental and performance testing of packaged units to verify design adequacy

Circuit design will be based on specification requirements of these units. A major consideration will be reliability including carrying out reliability figure of merit analyses in order to establish the need and degree of improvement required through redundancy, derating, or major redesign in order to achieve the reliability required.

Several breadboards will be constructed to permit concurrent development in electronic, thermal/mechanical, and subsystem studies. Electronic tests will include investigation of such items as failure modes, early marginal testing, circuit value variations to establish tolerances, search for critical or sensitive areas, electromagnetic interference effects, and electrical performance including that under simulated duty cycles. Thermal/mechanical tests will evaluate temperature effects on performance, determine the proper heat transfer and balance, and to the extent possible in a non-packaged design, critical areas for vibration and shock. Subsystem breadboard tests will be used to optimize compatibility with the remainder of the subsystem as well as the loads.

When the breadboard designs are indicated as being satisfactory, packaging design will be carried out, and complete units will then undergo functional tests and environmental tests including vibration, acceleration, shock, thermal, and thermal-vacuum tests. Evaluation of electromagnetic interference effects will also be a key element in tests of the packaged units.

(d) Distribution Board

Major development steps include:

1. Determination of number and location of connections
2. Materials selection
3. Design of board, including connection methods
4. Environmental and performance testing to verify design adequacy

Of prime consideration in carrying out these steps will be electromagnetic interference effects which will dictate to a considerable extent the location of the various connections. Tests of the finished product, with connections, will include determination of electromagnetic interference. Design of connectors will be appreciably influenced by vibration considerations.

(e) Quick Disconnect

Major development steps include:

1. Determination of number and rating of connections required
2. Vendor selection
3. Environmental and performance testing to verify design adequacy

A thorough analysis of ground testing power and data requirements will be necessary in order to determine the number and rating of connections required. Following this, specifications will be prepared and vendor selection made on the basis of their ability to meet the specifications, costs, and their past performance. The final product will undergo complete environmental and performance testing to verify design adequacy.

(f) Harness and Connectors

Major development steps include:

1. Determination of interconnection diagram

2. Materials selection

3. Determination of wire size, wire shielding requirements, and type of connectors

4. Preliminary environmental and performance testing

5. Finalization of harness based on vehicle mock-up

6. Environmental and performance testing to verify design adequacy

A complete analysis of all vehicle electrical and diagnostic inter-connections will be required to determine the inter-connection diagram which identifies all leads, their function, and their end-point connections. Selection of materials will be based largely on environmental requirements utilizing experience on past programs. Wire size determination will involve consideration of allowable voltage drop and wire temperatures and will require determination of expected lead lengths. Electromagnetic interference considerations will be involved in determination of wire shielding requirements, relative physical function of various leads, and grounding techniques. The final harness physical shape and dimensions will be determined from a full scale vehicle mock-up with all components in their proper locations. Tests of the final configuration will include determination of electromagnetic interference effects.

(g) Lander RTG Power Supply

The RTG power supply will be purchased for the program as a developed unit constructed with existing state-of-the-art technique with the unit designed for the power requirements for the Voyager Program.

Tests will be performed by the power supply manufacturer to demonstrate that the unit will deliver rated power. The cooling capacity necessary to maintain rated operating temperature will be determined. Radiation levels from the power supply will be measured. All tests requiring the operation of the RTG with the isotope will be performed at the vendor's facility, where the required radiation protection and the required safeguards are available. The structural characteristics and the ability of the power supply to survive flight environments and with a sizeable margin of safety must be demonstrated.

For the in-house tests at the spacecraft contractor's facility the heat input to the RTG from the isotope will be simulated with an electrical heater. This will constitute a test of only the thermoelectric conversion efficiency and characteristics.

Techniques of handling the isotope and inserting it into the power supply will be evaluated so that the handling techniques are efficient by the time system testing must be conducted.

(2) Subsystem Design and Integration

Major development steps include the following:

1. Preliminary design of subsystem

2. Preparation of component and subsystem specifications

3. Breadboard testing of subsystem

4. Verification testing of subsystem

5. Overall cognizance and control of subsystem development including interfaces between subsystem components and with the remainder of the vehicle.

Based on power requirements, duty cycle, and consideration of interaction effects with the remainder of the vehicle, the preliminary design of the subsystem will result in selection of the type of components to be used, preparation of a functional description of each component and of the subsystem, preliminary determination of size and weight of components, determination of component and subsystem performance requirements, and preparation of a subsystem block diagram.

Component and subsystem specifications will then be prepared to form the basis for detailed design.

Subsystem breadboard testing will be carried out using breadboards of the battery charge regulator and power control units. If available, cells inter-connected to form batteries will be used; otherwise the battery characteristics will be simulated. Loads will be simulated, and a power source simulating the characteristics of the solar array will be used. These breadboard tests will investigate overall subsystem electrical performance and interaction between components. Included will be investigation of failure modes, performance margins, and effects of variation in circuit valves to establish tolerances, as well as determination of electromagnetic interference effects. Tests will include subsystem performance under simulated duty cycles.

Following delivery of packaged components, testing will be carried out to verify required subsystem performance using all components in their final form except the solar array, which will still be simulated. Simulated or actual loads will be used, depending upon availability, and anticipated duty cycles will be followed. Investigation of failure modes and electromagnetic interference effects will be repeated.

D. Communications Subsystem Development

(1) General Approach

Development of the electronic components required for the subsystem will follow a logical growth and improvement cycle. Where possible existing successful designs which can perform satisfactorily in the Voyager application, will be used. If existing basic designs can be used with modifications, the design will be modified, units built, and evaluated.

For new designs the design will be made, analyzed and a breadboard unit will be constructed, tested and evaluated. The breadboard unit will be tested in the subsystem. If the design is successful, the unit will be converted into prototype hardware having the configuration, size, weight and packaged design intended for flight use. Detailed evaluation and environmental tests will be made to prove the performance of the unit. Required design changes will be made and the revised design will be used to manufacture a production unit intended for qualification test.

The communications components for Orbiter and Lander will be functionally similar and will be developed concurrently.

(2) Major Development Problems

Major development problems in Communications Subsystem are:

(a) Development of Thermoplastic Recorder

The thermoplastic recorder, selected as the storage medium for all Orbiter data, must yet be developed and flight tested. As the unit is successfully developed it will be tested under the particular application conditions which apply to Voyager.

Since performance of the unit is imperative to success of the flight mission, it would be very desirable to flight test the recorder possibly as a passenger flight on an earlier program. If this is not possible, detailed performance checks will be made under thermal vacuum environmental conditions.

(b) Development of Electrostatically Focused Klystron

A thorough performance analysis under flight environment conditions must be made on the electrostatically focused klystron, selected as the S-Band power amplifier.

(c) Development of Electrostatic Image Orthicon

Since this unit has not yet been flown on a space flight, a thorough evaluation of its performance must be made, including performance under thermal vacuum conditions expected in flight.

(d) Development of Sterilizable Components for the Lander

One of the major development problems of the Voyager system will be to develop electronic components which will operate successfully and without drift after being subjected to the sterilization cycle at 135°C. An analysis must be made of the temperature capability of the components. Then a detailed evaluation of the ability of the components to withstand sterilization by heat must be made. Special emphasis will be given to checking deterioration of performance over a period of time, performance characteristic drift, and resultant reliability.

(e) Development of a data processing subsystem which will accept data at a high rate and communicate back to Earth via the Orbiter relay link.

(f) Other Orbiter-Lander communication subsystem development tests.

Table 3.7.4-1 shows other communication subsystem development tests which must be performed.

E. Propulsion Subsystem Development Tests

Development testing of the Propulsion subsystem will be performed at two levels: at the component level, and at the complete subsystem level. Working from the initial subsystem design, components will be procured and development tests will be conducted.

Component	Typical Functional Tests
Valves	Response time Valve leakage Power requirements (electrical) Fluid flow characteristics Burst pressure
Regulators	Pressure regulating characteristics Flow characteristics Burst pressure

TABLE 3.7.4-1. SUMMARY OF TYPICAL COMMUNICATION DEVELOPMENT TESTS

TEST	TEST OBJECTIVES	DESCRIPTION OF TEST
1. VHF Transmitter	Determine: (a) The effect of voltage variation on frequency and stability. (b) The heat dissipation characteristics. (c) The presence of spurious signal generation. (d) Frequency stability. (e) Power output. (f) Modulation sensitivity.	The Lander will utilize a VHF transmitter to relay data to the Orbiter. Output power will be approximately 25 watts at a frequency of 100 mc. Transmitter will be set up in a laboratory and operated under various voltage inputs from nominal rated value. The effects of voltage variations on frequency and stability and power output will be measured. Noise problems will be evaluated in a suitably shielded enclosure. Performance under flight environments will be checked. Heat dissipation under high vacuum conditions will be checked.
2. VHF Receiver	Determine receiver functional characteristics.	Receiver will be set up and operated under laboratory ambient conditions. Receiver input frequency characteristics, sensitivity, noise figure, oscillator stability will be determined. Performance under flight environment conditions will be measured.
3. S-Band Transponder	Verify transmitting, receiving functions of the S-band transponder.	Transmitter frequency, frequency stability, power output, modulation sensitivity, effects of input voltage on transponder characteristics will be determined. Presence of spurious signal generation will be investigated. Transponder receiver functions will include input frequency characteristics, sensitivity, oscillator stability, noise figure. Coherent mode of operation will be verified.
4. Klystron	Determine functional characteristics.	Output power, dissipation, efficiency, gain under various VSWR conditions and under various inputs will be checked. Effect of ripple, noise on high voltage input to klystron will be checked.

TABLE 3.7.4-1. SUMMARY OF TYPICAL COMMUNICATION DEVELOPMENT TESTS (Cont'd)

TEST	TEST OBJECTIVES	DESCRIPTION OF TEST
5. Command demodulator	1. Measure synchronization time with respect to signal-noise. 2. Measure error rate with respect to signal-noise. 3. Determine susceptibility to incorrect codes.	Output of a signal generator or a prepared program toegether with noise signals will be fed into the demodulator. Output will be compared to the input signal to determine ability of the demodulator to process command signals.
6. Programmer & Computational Unit	Ascertain that the programmer is compatible with input signals from the command demodulator and is able to perform required control and computational functions.	The programmer receives PCM format from the command demodulator and relays the required commands to the performing subsystem.
7. Entry Data Recorder	Verify that the data recorder can store and, upon command, relay the stored data at various rates below a fixed error rate. Demonstrate that the recorder will operate under entry-landing conditions.	The Lander communications system will contain a data recorder that can store the entry data during entry blackout. Briefly, scientific and diagnostic data is fed to a multicoder and thence to the data recorder for storage. After successful landing, the Orbiter will command the Lander to transmit the stored data and the data recorder will relay the data to the transmitter. Verification testing is required to assure that the recorder can store and process signals satisfactorily. Testing will include functional performance testing and environmental testing to entry-landing levels. Performance checks will include recorder, noise signal record-playback accuracy and loss of input data bits. A test program will be set up and processed through the recorder and playback data will be compared against the input.
8. Command Decoder	The objective of this test is to verify that the decoder can receive the synchronized/timed signals from the programmer and forward the proper	The Orbiter PHP will utilize a decoder that functions to feed the power conversion and control unit. A test is required to check compatibility with the power conversion and control unit and programmer. A

TABLE 3.7.4-1. SUMMARY OF TYPICAL COMMUNICATION DEVELOPMENT TESTS (Cont'd)

TEST	TEST OBJECTIVES	DESCRIPTION OF TEST
	PCM format to the power conversion and control unit.	command sequence will be fed to the decoder and its response to commands and ability to reject incorrect command sequences will be checked.
9. Data Processor Unit	Evaluate ability to handle digital and analog data with no error.	Process prepared program into data processor. Compare output data against input program for errors.
10. Diplexer	1. Determine that the diplexers are compatible with the antennas. 2. Measure minimum isolation. 3. Measure VSWR over the diplexer operating frequency.	Diplexer will be set up with appropriate signal generators and antenna impedances, and diplexer characteristics will be measured over operating frequency spectrum. Input switching, isolation, insertion loss and VSWR will be measured.
11. Antenna	Determine antenna characteristics	1. High gain antenna will be set up on antenna pedestal on antenna range and gain, beam width and antenna pattern will be determined. 2. Antenna gain and pattern will be measured for all antennas. 3. Effect of spacecraft configuration on antenna radiation pattern will be checked by measuring antenna pattern with antennas mounted on a simulated spacecraft, especially with the VHF antennas. Tests probably will be conducted on an isolated antenna range.
12. Power Converter and Control	1. Verify that the converter processes the input system power into the selected power requirements of individual components. 2. Determine the effect of a power input fluctuations on the output power of the converter.	The Lander utilizes a power converter to process battery power into proper component power inputs. It will be necessary to confirm that the converter operates properly and to determine the effects of failing battery power. Output power, voltage and regulation will be determined with normal battery voltage and with reduced voltages. Switching function of the power control unit must be verified.

TABLE 3.7.4-1. SUMMARY OF TYPICAL COMMUNICATION DEVELOPMENT TESTS (Cont'd)

TEST	TEST OBJECTIVES	DESCRIPTION OF TEST
13. High Voltage Power Supplies	Determine functional performance of power supplies including power output, voltage, frequency, and regulation AC component ripple.	Power supplies will be operated with simulated and dummy loads and performance will be checked with input voltage also below normal. Performance will be checked under flight environment conditions. Attention will be given to identifying heat transfer problems under high vacuum environment.
14. Communication Subsystem Functional Test	Determine the operating characteristics of the communications subsystem.	The communications subsystem will be built up as breadboards and functional characteristics of the subsystem will be determined. Component electrical interfaces will be established. As design changes are made, the subsystem will be updated from its breadboard status. Input parameters will be varied and operating characteristics measured. Transient and noise problems will be investigated.

Filters	Pressure drop Flow characteristics Efficiency Burst pressure
Relief valve	Operating pressure Burst pressure
Tank diaphragm	Burst strength Fatigue limits Compatibility with contaminants
Tanks	Burst pressure
Thrust chamber	Wear characteristics High temperature characteristics Uniformity of thrust alignment

Additionally the development program must include tests evaluating the component performance under the Voyager environmental conditions. These tests are followed up by the component qualification tests to demonstrate design adequacy.

Propulsion subsystem development will be going on simultaneously with the component development. The initial design will be "breadboarded" with the components which are available from similar designs and have similar characteristics as those called out for the Voyager Program. Propulsion subsystem characteristics will be determined from this initial breadboard. As the specified components can be developed and procured, they will be inserted into the breadboard and testing continued. Test results and evaluations will indicate changes required in the subsystem and components. Thus the design will be a reiterative process.

The following characteristics will be established:

1. Operating characteristics
2. Limit operation
3. Malfunction testing
4. Repeatability and reliability.

The malfunction testing will consist of building into the subsystem certain performance malfunctions or problems and determining system response. Special attention will be applied towards proving the function of the redundant components or loops in response to malfunctions. Operation of the subsystem with variable electrical power and control inputs will be evaluated.

System development testing must include an altitude firing test with control inputs into the subsystem control of propellant flow, thrust level, thrust chamber life and wear characteristics, and thrust vector control will be evaluated.

Evaluation of the subsystem under the Voyager environments with emphasis on vibration test must be made. This will be followed by the complete component and subsystem qualification program.

A qualification test of the Propulsion subsystem will be conducted in accordance with the Program specifications.

3.7.5 SPACECRAFT SYSTEMS DEVELOPMENT TEST PLAN

A. Philosophy

The primary purpose of the proposed system development tests is to determine subsystem compatibility, mechanical and electrical performance, provide operational data and design information, and develop, from test results, spacecraft vehicle specifications for the flight spacecraft production, and procedures to be used in testing the production spacecraft. It is recommended that three pre-prototype systems be utilized during the systems development phase of the Program. The need for three vehicles is based on the amount of testing required during the development program and the short period of time in which this testing must be accomplished. The availability of three development systems will allow several development sequences to proceed at the same time and permit the required development work to be completed on a shorter schedule.

The following philosophy will be followed during the systems development program:

1. The program will be designed to obtain a maximum amount of design information, hardware confirmation, and test development prior to Stage 4 preproduction design releases and prior to the qualification test program.

2. Development systems are exposed to expected ground, launch, and orbit environments simulated to the best extent possible, and consistent with the availability of time, funding, and facilities.

3. The intent will be to use supporting test equipment as close to final configuration as is feasible during the development program. This approach will serve to establish spacecraft and test equipment compatibility, to gain operating experience, and to reduce overall program costs.

4. Continual review and updating of the planned test programs will assure an effective balanced program growing on the basis of the test data and experience accumulated. Spacecraft test sequence will be reprogrammed as required to obtain the maximum amount of test data with the most effective utlization of the available spacecraft.

B. Test Plan

Three systems are utilized during the development phase of the program. They are identified as D-1, D-2, and D-3 spacecrafts. The test plan below is suggested to verify the operation of the system and to demonstrate the successful solution of the Voyager development problems.

(1) D-1 System Prototype Tests

(a) System Description

The first development system consists of a complete spacecraft structure with two Lander structures utilized primarily for dynamic, structure, and mechanical testing. This system is built to Stage 3 prototype releases and incorporates all of the design and development experience gained from the earlier structural development models.

The system includes operable, deployable mechanisms for the antenna, magnetometer and the PHP, and the adapter and Lander separation mechanisms. The actuating mechanisms (drive motors, actuators, etc.) for these units are required. However, the actual functional magnetometer, and antenna can be simulated with dummy components having similar configurations and masses, thus reducing the unit cost and avoiding early hardware schedule problems. The test spacecraft includes an early design functional pneumatic system for performing pneumatic tests. Other components affecting the mechanical and dynamic properties of the system are simulated by models having the same mass-volume characteristics with simulated mounting configurations. The use of mass simulated model components for the D-1 system may be necessary since many of the functional components will not be required for the planned tests and considerable cost savings can be realized. Also, the important dynamic tests can be initiated without waiting for the detail design of many of the components to be completed, and the components to be fabricated.

(b) Test Program

1) Lander-Orbiter Mechanical Compatibility

Although it is likely that the Lander and Orbiter will be developed as a parallel program and will share common interfacing tooling, to insure mechanical interface a compatibility test should be made with the Structural Model. This will provide a recheck of mating dimensions and fastening and separation devices. Additionally, the handling equipment required to lift, align, and mate the Lander to the Orbiter will be used and evaluated and handling experience will be gained. It is advantageous to gain this experience at this point rather than by working with a complete functional spacecraft.

2) Handling Procedures and GSE Compatibility

During the systems structural test, the handling equipment and techniques to be used during the program will be evaluated. Additional benefits gained from the activity would be the familiarization and training received by test personnel applicable during the later phases of the program, in-house and in the field. The experience gained in handling this large spacecraft during the development activities should eliminate many handling problems and preclude possible damage to production spacecraft.

GSE and in-house Handling equipment are utilized during this phase of activity. This includes dollies, slings, hoisting mechanisms, shipping containers, etc., for the Orbiter, the Lander, and the complete spacecraft. All of the handling methods to be used both in-house and at the field site are practiced using this equipment. Compatibility is checked and all mechanical fits and measurements verified. Based on the knowledge gained during this effort, detailed handling procedures and test equipment design changes will be specified for the production phases of the program.

3) Launch Vehicle Interface

Compatibility of the spacecraft with the mating Saturn VI launch vehicle interface and with the vehicle shroud must be verified during the development phase of the program. The D-1 development spacecraft will be mated to an accurate mockup of the adapter, and interfacing dimensions, electrical connections, and other interfaces will be checked. Special attention will be directed to mating bolt hole patterns and mechanical fits. Accessibility to electrical connectors and to the mounting bolts is verified. This mating check will be repeated for the adapter-launch vehicle interface.

Compatibility checks are made with a dummy or mockup Voyager shroud. Mechanical and dimensional checks are made, as well as electrical connections.

4) Pneumatic Tests

The D-1 development system provides the basic structure for performing the initial pneumatic system development and tests. The objective of this effort is to determine the best routing to be used for the pneumatic system, solve the interface, fit, and mounting problems, and to initiate functional performance and leak checks.

At the completion of the installation of the pneumatic system, the development spacecraft is moved into a pneumatic test cell and functional performance tests are made. The operation of the regulators, solenoids, nozzles, etc., is evaluated. The system is actuated from a control console to simulate the signal inputs to the various components. Proof pressure tests are performed to evaluate the design requirements of the system. Leak tests are made to evaluate the construction, connections, and components utilized in the pneumatic system. Halogen leak detectors and/or mass spectrometers are required for these tests.

5) Static Load Tests

Prior to dynamic testing, the spacecraft is subjected to static load tests which verify the structural characteristics of the system. These tests will be much simpler than the dynamic tests, and will not simulate the total flight loading condition, but they will provide the design confidence required for these tests. These tests will be an extension of the structural tests performed on only the bare structure and described in Paragraph 3.7.4A. The following tests should be made utilizing hydraulic jacks, weights, strain gages, etc., and measuring all resultant strains and deflections.

1. Check hard points for static load plus simulated acceleration load

2. Simulate acceleration and static loads for critical components

3. Simulate lateral loads expected from vibration and handling on items such as the Lander mounting points, rocket motor mounting points, structural panels, antenna, critical components, etc.

If subsequent investigation shows it is not possible to accomplish an acceleration test on a centrifuge, the above tests become more significant towards evaluating structural integrity.

6) Vibration Tests

The most critical structural development test is the system vibration test. With a light weight complex structure having a number of concentrated loads and experiencing the vibration stresses of launch, an intensive evaluation of the structure and method of anchoring the Landers, antenna, PHP and other loads is required.

The primary objectives of the vibration test of the system are as follows:

1. Evaluate the structural characteristics and dynamic properties of the spacecraft.

2. Determine vibration levels experienced by the components at selected spots on the spacecraft. This analysis applies particularly to the high gain antenna and the PHP which may experience vibration attenuation problems related to their mounting arrangements and positions.

3. Determine qualification and acceptance test vibration levels for components and systems. These specifications should be based on realistic data obtained during the development tests.

At the present time, it is difficult to realistically predict the vibration levels to be experienced by the spacecraft. This difficulty stems from the fact that the dynamic characteristics of the total launch vehicle and especially the Saturn VI Stage are as yet undefined.

It is expected that by the time the spacecraft development tests are being performed data will be available on the proposed launch vehicle. This will include a structural dynamic analysis and confirming test data obtained from measurement taken at selected monitoring points during static firing tests of the VI Stage. Also further flight data from the Saturn I and the I-B flights will identify the structural characteristics of the first two stages of the launch vehicle.

The spacecraft is monitored by recording the outputs of accelerometers and strain gages strategically placed on the structure. At completion of the low level sine tests the data is analyzed to determine if any severe attenuation or transmissibility problems exist which must be corrected before proceeding with the random or high level sweep tests. This test allows the vehicle to be vibrated nondestructively before experiencing flight vibration levels.

High level sweep tests should then be performed on each of the three mutually perpendicular axes through a spectrum of approximately 5 cps to 2000 cps with varying sine level inputs which correspond to or are higher than those levels expected in flight. The purpose of this test and following tests is to assess the capability of the spacecraft to withstand structural damage when exposed to the flight vibration environment.

At the completion of the high level sweep tests resonant dwell tests are performed at each of the critical resonant frequencies determined earlier. Data obtained on the launch vehicle will identify the expected dynamic input the spacecraft must be designed to withstand.

To accomplish the vibration test, the spacecraft is mounted (in its launch configuration) on a fixture simulating the spacecraft - adapter interface, and this fixture is mounted on a large vibration system (such as the MB-C210 with dual shakers, or equivalent). Because of the large size of the spacecraft, and especially if more than one shaker head is used, the supporting fixture design will be quite complex in order that a rigid, resonant free, yet light weight, fixture is provided. Additionally, the method of supporting or suspending the weight of the test unit without deflecting the shaker head excessively will require test design consideration. Tests are performed in three axes, using an oil film auxiliary table, hydro-pneumatic team tables, or equivalent support for tests in two of the axes.

Initial tests are performed in the three mutually perpendicular axes with low level sine sweeps. The frequency is cycled from approximately 5 cps to 2000 cps with increasing g loads being applied.

Random vibration tests in each of the three mutually perpendicular axes are performed. In all these tests accelerometer and strain gage data are recorded using magnetic tape recorders and/or high speed oscillographic recorders.

Any structural failures which occur during the vibration testing must be corrected and design changes incorporated. Data recorded at component positions during the vibration tests are analyzed and used in the formulation of component specifications for the component testing phases of the program.

7) Acceleration Tests

In order to verify that the spacecraft will withstand the flight loads, acceleration tests will be performed on the development system. The spacecraft will be mounted on a fixture in the normal launch configuration and it will be subjected to an acceleration level along the "launch axis" corresponding to at least powered flight levels for thirty minutes. Accelerometers and strain gages will provide readout of test results to the data recorders. Additionally the on-board spacecraft telemetry, by means of the diagnostic sensors, will monitor functional performance of the spacecraft during this simulated launch profile test. The spacecraft will be controlled by a simplified systems test set and data will be transmitted to a simplified ground station and recorders. At the completion of the acceleration test the spacecraft will be examined for any structural defects. Tests will be repeated with acceleration forces applied at the appropriate levels along the other major axes.

The test fixture will include an aerodynamic shroud, so that the spacecraft is not subjected to aerodynamic loading such as it will not see in flight.

Size and weight of the spacecraft will pose a major test limitation. The accelerator at Sandia with 10000-pounds weight capability and 450000g-lbs capacity and the one at Edwards Air Force Base should be capable of carrying the Spacecraft fixture weight and attain the required acceleration level; however, physical size limitations have not been checked and may not be compatible with the proposed design.

During the course of the testing program the need for the acceleration test will be evaluated. Depending on the results of the static loading tests and vibration tests, it may be possible to eliminate the test on the prototype and perform the acceleration test (on a complete functional spacecraft).

8) Separation Tests

Functional operation of the separation mechanism must be evaluated for the separation occurring between:

1. The flight vehicle shroud and the vehicle

2. The spacecraft adapter and the launch vehicle

3. The adapter and the spacecraft

4. Separation of the sterilization shroud

5. The Lander from the orbiting spacecraft

Separation tests will be conducted on the D-1 spacecraft to demonstrate the functional operation of the separation mechanisms. A separation command will be programmed to the separation explosive device or actuator and operation of the control circuits and actuating devices will be monitored. In the case of explosive devices, simulators will be substituted. Emphasis on these tests is to demonstrate functional performance of the circuits and components; the actual mechanism of separation at the separation process was tested and verified in the subsystem development tests described previously.

9) Deployment Tests

Tests must be made to verify the operation of the spacecraft deployable mechanisms. This includes the high gain antenna mechanism, the magnetometer boom and the PHP. The tests must be run in at least two positions of the spacecraft so that the loads due to the earth gravity field can be evaluated and factored out. Test evaluation must include monitoring the actuator outputs and operation of the latching and locking mechanism.

(2) D-2 System Prototype Tests

(a) System Description

The D-2 spacecraft is essentially a complete functional unit made up of an Orbiter and two Landers. It is constructed for the primary purpose of performing electronic and electromechanical component installation and subsystem and system performance testing. The components are assembled to prototype design releases and incorporate all development experience gained from the breadboard component design evaluations. All components and subsystems are functional. Simulators are used where it is not feasible to use the actual spacecraft unit. For example, functional output of the Lander RTG or the Orbiter solar array would be simulated.

The basic spacecraft structure need not be thermally coated or insulated. Primary emphasis will be on the installation, compatibility, and functional performance required for each of the electronic components and subsystems.

(b) Test Program

1) Component Installation and Subsystem Compatibility

Component layout and installation on the spacecraft structure will present the first basic problem on the D-2 prototype spacecraft.

The objective is to finalize the following problem areas:

1. Component location

2. Component mounting method and arrangement (utilizing dynamic structural analysis determined on the D-1 system)

3. Subsystem installation and harnessing arrangements

4. Harness routing installation and compatibility between subsystems.

All of the subsystems should be completely installed on the spacecraft and preliminary circuit checks made to verify the continuity and compatibility of all interconnections. When all of the installations have been accomplished, the spacecraft will provide a complete operational and functional system.

(c) Compatibility Test With Systems Checkout and Test Equipment

Once the subsystems have been completely installed, it will be necessary to verify that the subsystems and systems checkout and test equipment is compatible and operable with the spacecraft. To assure that faulty or incompatible test equipment does not cause test failure the compatibility check is completed before the spacecraft subsystems performance tests can be initiated.

(d) Subsystem Performance Evaluation

The objectives of these tests are to assure the functional performance of each subsystem, and to verify that the subsystems meet the design requirements and function satisfactorily. During these tests a continuous re-evaluation will be made of the design tradeoffs, and changes and redesign to improve performance will be taking place.

The following subsystem tests will be performed:

1) Structure Subsystem - Mechanical Properties Tests

The D-2 Development Vehicle should be complete enough that meaningful determination of weight, center of gravity and alignment can be made.

2) Electrical Power Subsystems

1. Power profile loading sequence
2. Subsystem power output versus operating requirements

3. Power regulation performance
4. Battery charge retention and charge rate
5. Power control unit circuit switching command and response
6. Power drop and loss measurements
7. Solar array shorting characteristics
8. Lander power supply output characteristics

Since it is not practical to utilize the solar array as a power source during these tests, it will be necessary to utilize a power supply simulator with power characteristics similar to the solar array.

Similarly, output of the Lander RTG unit must be simulated by using an electrical heater rather than the isotope to heat the power converter. Because of safety considerations, the isotope will not be used in any of the in-house spacecraft system tests.

3) Attitude Control and Guidance Subsystem Tests

A detailed analysis of the performance of the attitude control and guidance subsystems will be made on an air bearing supported 3-axis simulation test facility. This test will be performed at the subsystem level by mounting the attitude control components on a 3-axis motion simulator having a moment of inertia corresponding to that of the Voyager spacecraft. This test approach is taken rather than mounting the entire spacecraft on an air bearing because testing problems in mounting and testing the spacecraft make the spacecraft level of testing difficult, and compromise the test results. (See detailed analysis of feasibility of motion simulator test.)

With the subsystem mounted on the motion simulation facility the following tests are performed:

1. Attitude control subsystem power supply performance
2. Subsystem power requirements in various modes of operation
3. Star tracker and sun sensor polarity versus pneumatic component "polarity"
4. PHP and antenna drive polarity, torque, and position accuracy. Response of antenna drive loop, and sun sensors and star tracker tracking loops to input stimuli.
5. Response to all initial acquisition and reacquisition sequences
6. Correlation of attitude control telemetry monitoring signals to input stimuli
7. Stimulation and response of pitch rate gyro
8. Correctness of programmer functions
9. PHP Planet tracking capability evaluation.

The initial subsystem tests on the motion simulator establish the dynamic operation of the attitude control function in a manner which cannot be duplicated by static test. Upon the completion of the above tests on the motion simulator, the subsystem will be installed in the development spacecraft and an equivalent type of static open loop analysis will be made on the spacecraft. The subsystem will be energized and input stimuli will be fed into the subsystem and response will be monitored. A thorough evaluation and demonstration of performance must be made in the following areas:

1. Acquisition and reacquisition of the simulated sun and star targets and attitude control of the spacecraft about the axes fixed by these targets.

2. Response of the reaction control loop in maintaining attitude and fix on sun and star targets when the sensors are stimulated with input signals.

3. Response of the high gain antenna Earth tracking loop to input stimuli to the Earth sensor.

4. Response of the PHP attitude control loop in maintaining the PHP pointing towards a simulated planet target as various error inputs are fed into the planet sensor.

5. Response of the propulsion thrust vector control to input stimuli must be measured and evaluated.

In order to accomplish these tests, the motion simulator facility described in Paragraph 3.7.10D. will be required for the dynamic tests. The static tests will be powered, controlled, and monitored from the systems test set equipment array. It will contain power supplies, controls and instruments to set up, energize power and measure input to the entire spacecraft system. Additionally it will include the signals and simulator required to stimulate the spacecraft attitude control sensors and to measure the response of the attitude control subsystem. It will monitor the response of the sensors, the amplifiers, the control networks and the actuators and valves and the overall loop performance of the antenna control loop, the thrust vector control, and the mass expulsion attitude control loop.

4) Communication Subsystems

In addition to the communications function of transmitting telemetry data from the payload and from the diagnostic sensors, the communication subsystem has a command function and a tracking function. It receives and verifies and response to commands from the DSIF and it provides doppler tracking capability by using a coherent transponder.

The following tests will be performed:

1. Power control and conversion "subsystem" performance including response to main power bus over-voltage and under-voltage conditions.
2. S-band transponder and VHF transmitter power, frequency and stability.
3. Response of subsystem to programming sequences.
4. Correctness of command function - discrete and timed.
5. Command receiver acquisition time.
6. Performance of stored commands vs. Earth commands.
7. Correctness of timing functions.
8. Calibration and identification of telemetry data channels.
9. Correlation of processed telemetry data to known input stimuli.
10. Data storage and feedback verification, data transmission rate.
11. Operational checkout of communication links:

    Orbiter to Earth
    Lander to Orbiter
    Lander to Earth

Transmitter power, frequency, and deviation will be checked. Data transmitting rate for digital and analog data and storage capability will be checked.

Test of the communications (or Telemetry Tracking and Command Subsystem) is accomplished by energizing the spacecraft (including Landers) electrical power subsystem and the communications subsystems from the System Checkout and Test Set. The spacecraft communicates to and is monitored by a ground station. The ground station, including a computer has the capability of generating commands in digital word format, transmitting to the spacecraft and monitoring the response transmitted back through the telemetry transmitter. Likewise, response of the sensors to stimuli and known calibrated input levels will be monitored and calibration accomplished.

5) Propulsion Subsystem Tests

Testing of the propulsion subsystem as part of the systems development tests will be limited to loading the fuel tanks with simulated fuel, pressurizing the subsystem and monitoring the performance of the valves and regulator as input stimuli are provided from the attitude control subsystem. It is unlikely that a flight configuration nozzle will be installed in this spacecraft.

(e) Other Tests

1) Electromagnetic Interference Tests

The assembly of all of the subsystems on the D-2 pre-prototype spacecraft will provide a complete electrical system which can be evaluated for EMI or noise. The purpose of this testing is to evaluate the electrical noise characteristics of the spacecraft and to incorporate design modifications or shielding changes that are required to eliminate electrical noise or interference problems.

The completed spacecraft is installed in a shielded room. Each subsystem, and the complete system is subjected to a complete series of susceptibility and interference tests. All critical frequencies of the spacecraft and the launch system are evaluated and susceptible components earmarked for design modifications. The tests verify that the spacecraft system functional operation does not interfere with telemetry, communications, command, or payload functions and does not cause spurious operation or the malfunction of any electronic components.

Redesign will be accomplished, or filters and screening applied to correct any deficiencies, and additional tests made to evaluate the corrective action taken.

2) Magnetic Shielding

Depending upon the sensitivity of the magnetometer experiment, it may be necessary to check the spacecraft electrical system for stray magnetic fields which will interfere with the magnetometer measurements. Preferably this will be done at the component level. Each electrical component would be checked in a facility set up with Helmholtz coils energized to cancel the Earth's magnetic field as measured with the component de-energized. A similar type facility might be necessary to check the overall, or total system and assure that no spacecraft magnetic fields are being set up which interfere with the magnetometer measurements.

3) Sterilization Procedure Evaluation

Upon completion of the major portion of the subsystem functional performance tests, the D-2 system is used to verify the Lander sterilization and handling procedures.

The Lander capsule is disassembled from the spacecraft and placed in the sterilization preparation area. If any components are not heat sterilizable they will be removed from the Lander and sterilized by other means. The Lander is heat sterilized at 135°C for 24 hours. The components are reassembled under sterile conditions. A check is made for living organism count, if feasible,and the Lander is packaged in its bio-barrier.

Checkout of the sterilization handling procedures is accomplished by taking the Lander from the sterilization area, making required performance checks, and mating it to the Orbiter. Handling will correspond to that which the unit would see in the field. At the end of this cycle, the Lander sterility is assayed again. This test will serve to:

1. Establish the feasibility of the planned field handling equipment and techniques
2. Verify that proposed sterilization techniques are effective
3. Train the field crews prior to their handling a flight vehicle
4. Provide additional verification that the Lander will function satisfactorily after sterilization.

4) Retrofit and Failure Analysis

During the course of any development program, it is necessary to design, test, redesign, modify, retest. Because of the large number of redesigns and modifications that occur during a normal development program, it is necessary to have a spacecraft on which design changes and retrofits can be evaluated and failure analysis performed.

Since the D-2 spacecraft is essentially functionally complete, it presents an ideal system for performing this activity. During the course of the development program, design changes and retrofits will be made and evaluated on this spacecraft when it becomes available. After the modifications and retrofits have been performed, the subsystem which has been effected will be retested for compatibility and performance. This evaluation activity will enhance the development effort and will provide an active system for evaluating all changes made during the course of development.

Components which are removed from the production spacecraft due to malfunctions will be failure analyzed, and the information obtained will be incorporated into the component design. At times it may be desirable to attempt to duplicate the conditions occurring during a production unit failure using this development system.

5) Spares Burn-In

Additionally this system can be used to provide the component burn-in time for spares components so that spares installed after systems acceptance test will have had the benefit of burn-in time in systems application. Thus, spares would be not just out of stock, but would have been used for some period of time in systems operation. Thus, early component drift in characteristic or early failure occurrences should be reduced.

(3) D-3 System - Prototype Tests

(a) System Description

The D-3 development system or prototype spacecraft is a completely assembled system with all functional components, subsystems, thermal coatings, and insulation. The system is built to prototype design releases and incorporates all of the design and development experience gained in the earlier phases of the development program. The system is completely functional. The primary function of this spacecraft prototype is heat balance testing, improvement of subsystem and system performance, checkout of test equipment and testing procedures to be used in later testing phases and long term (1000 hours or more) thermal-vacuum testing.

(b) Test Program

1) Thermal Balance Test

As soon as a prototype structure (a stripped-down spacecraft) becomes available, it will be fitted for thermal balance tests. The objective of this early test will be to determine if there are any basic or major thermal control problems which must be corrected before design drawings are released for production. The tests provide the first verification of the calculated thermal environment for the components.

The spacecraft (including Landers) is mounted in a fixture and installed in the thermal-vacuum chamber having solar simulation capabilities of one solar constant. The spacecraft has a fixed orientation with respect to the simulated sun. The chamber pressure is to 10^{-6} torr or better, the cold walls stabilized at -300°F and the simulated sun turned on. The spacecraft is extensively instrumented with the thermocouples to provide detailed thermal mapping. Detailed evaluations are made of areas where components with large electrical dissipation loads are mounted.

After the initial tests are made and the evaluation of the thermal design is finalized, the D-3 spacecraft is completed. All components are installed and insulation and thermal coatings as required are applied.

The components and the spacecraft are instrumented with thermocouples and performance monitoring instrumentation within the limitations of available chamber penetrations. Test progress is monitored and controlled by the test equipment and instrumentation located outside the vacuum chamber. Additionally, commands and telemetry data are communicated by the spacecraft communications subsystem to the ground station.

The thermal-vacuum test provides an excellent opportunity to check out the sun, earth, planet and star sensors in environment simulating flight environment. To accomplish these tests, simulated sun, earth, planet and star sources are mounted in the chamber and the ability of the sensors and the control loops to respond to these sensors will be measured. The sensors will monitor the stimuli against a cold black simulated space environment.

At the completion of the spacecraft installation, the spacecraft system as well as all test equipment are checked for compatibility, and a functional checkout of the spacecraft with its vacuum test equipment is performed.

The chamber pressure is reduced to a pressure level of 10^{-6} torr or better, and the spacecraft is exposed to liquid nitrogen cooled chamber walls (-300°F) which simulate the heat sink of space and to solar input corresponding to near-Earth flight.

When conditions have stabilized in the chamber a complete operational test is made simulating the mission profile. The spacecraft is operated with all systems functioning in their proper sequence as well as all solar heat inputs being programmed to suit the orbital condition. Response of the subsystems to test inputs, stimuli, and commands is monitored. Functional performance of the spacecraft is checked and spacecraft operating temperatures and pneumatic leakage are monitored. The performance should be continuously monitored and all data analyzed for required modifications or corrections to the spacecraft prior to testing releases for production. Tests are as detailed as described for the D-2 spacecraft system functional performance tests.

At the completion of the test, the spacecraft is removed from the chamber and tested for any performance degradation. The evaluation outside the chamber has the advantage of the spacecraft and test equipment being accessible, and there are fewer limitations to the number of monitoring points due to vacuum chamber penetrations.

2) Reliability Test

When the D-3 system is available at the completion of the above tests, it will be used for a long term reliability or long life evaluation. This test will be a long term extension of the thermal-vacuum test with the system being exercised and evaluated periodically.

3.7.6 COMPONENT QUALIFICATION TEST PLAN

A. Definition of Qualification Test

The qualification tests are a series of tests performed on prime production hardware to certify that the design requirements have been achieved. The tests are normally run at test levels exceeding stress levels expected to be encountered in flight. This increased test level serves to establish the margin of safety in flight and to provide a margin to cover test unknowns to be encountered in handling and in flight. Also, it will provide some confidence to compensate for the small sample size.

Qualification tests are of such severity that they can be expected to shorten the life of the unit under test; therefore, components which have been qualification tested cannot be used on a flight vehicle, or for a spare unit.

B. Qualification Program Objectives

1. Verify that the component design meets the design specification performance requirements under flight environmental conditions or under more severe conditions.

2. Establish a performance margin of safety beyond the expected flight environments.

3. Collect performance data useable in measuring achieved component reliability and for predicting flight reliability.

4. Determine modes of failure of components and use this information for reliability improvement.

5. Determine product improvement design changes desirable for future production.

C. Qualification Program Ground Rules

1. In general, two components of each design type (except explosives, rocket engines, etc.) will be qualification tested. Qualification test of squibs and explosives will require a representative lot sample as described in the component specifications.

2. Component qualification will be completed before the first Voyager Program flight.

3. If any significant change is made in the design, or in the manufacturing processes, the component will be submitted for requalification.

4. All components submitted for qualification will be prime production units built by the Manufacturing Organization (not Engineering) to controlled drawings and specifications.

5. Components will be acceptance tested before qualification to assure that they are representative production units.

6. Test levels to which testing will be performed will be those specified in the Component Test Requirements Specification. The performance requirements will be as specified in the individual component specifications. (Test requirements for the Lander will be different from these for the Orbiter.) The component is performance tested prior to environmental exposure and following exposure to determine any degradation.

7. No repairs or adjustments on the components during the tests will be permitted. The need for adjustment or repair indicates that change in design, or in manufacturing and assembly techniques, or some other corrective action is required. A failure investigation will be conducted and appropriate corrective action taken to eliminate the recurrence of such failure. Component failure is defined as any deviation from performance specified in the component specification.

8. No attempt will be made to maintain sterilization of the component should any component be received from a vendor or from the manufacturing process in a sterile condition.

9. Test levels will be different for Lander and Orbiter, and will be as called out on component test specifications.

10. Functional performance tests are conducted on the component before exposure to the qualification test environment and after environment test to evaluate degradation caused by the environmental exposure.

11. Performance tests will be made using special test equipment as required. It will consist of power supplies, recorders, meters, timers, signal generators, programmers, oscilloscopes, and switches and controls as required for the particular component under test.

12. Acceptance Test - All components to be qualification tested will first be acceptance tested to insure that the units have been correctly assembled and are representative of typical production hardware.

13. Sterilization Test - The sterilization test will be performed on only those components which will be subjected to sterilization in the spacecraft application, presently limited to Lander components. When the sterilization test is performed, it will be the first test in the test sequence, with the other environmental test following. Thus, the effects of sterilization cycle can be evaluated in depth.

14. No fungus or salt spray test will be run.

D. Integrated Test Program Board

An Integrated Test Program Board will be established to analyze the data from the qualification testing program and determine whether the qualification test requirements have been fulfilled. The ITPB will monitor and direct the implementation of the qualification program. It will issue periodic reports giving the status of the qualification program. When failures occur during testing the ITPB will make recommendations on the qualification status of the components and the extent of required retesting.

E. Qualification Test Cycle

Summary of the tests is as follows:

1. Thermal Sterilization
2. Gas Sterilization

3. Handling Shock
   a. Drop test
   b. Bench Handling test
4. Humidity
5. Electromagnetic Interference
6. Static Acceleration
7. Vibration
8. Shock
9. Acoustic Noise
10. Thermal Vacuum
11. Sand and Dust

(a) Thermal Sterilization Test

1) Objectives

1. Demonstrate that those components which must be heat sterilized can operate after being subjected to the thermal sterilization cycle.

2. Determine the effectiveness of the sterilization cycle in eliminating living matter from the component.

2) Description of Test

Based on the system requirement for a sterilization cycle of 135°C for 24 hours the component sterilization test is performed at 145°C for 36 hours as described in the component test specification. Test time begins after temperature stabilization of the component has been achieved. Time required to reach stabilization of temperature must be recorded for use in estimating time required to accomplish the system sterilization prior to flight.

At the end of the test time the component is removed from the test chamber and allowed to stabilize to room temperature. This test is repeated three times to allow for the possibility that a flight component may require resterilization.

At the completion of the temperature cycles the performance of the component is tested to establish any degradation.

Since the object of this test is to determine capability of the component to withstand the test temperature, rather than to actually accomplish the sterilization, the performance test after the heating cycle may be performed under less than sterile conditions.

On a sampling basis it will be desirable to monitor the effectiveness of the thermal cycle in removing living matter and to evaluate the techniques of packaging and handling sterile components.

For a sample number of components going through the sterilization temperature cycle the component at the completion of the first thermal cycle will be handled and packaged under sterile conditions, then the number of surviving organisms will be measured. For the other components no attempt will be made to maintain sterility during test.

3) Facilities and Special Test Equipment Required

1. Temperature test chamber programmable to 145°C ±2°C for 36 hours. Chamber volume 3 ft x 3 ft x 3 ft to 4 ft x 4 ft x 4 ft as required so that test volume is at least twice test hardware volume.

2. Sterile test facility - required to perform test of effectiveness of heat cycle in producing sterility. (Equipment is described under Sterilization Plan).

(b) Gas Sterilization Test

1) Objectives

1. Verify that the ethylene oxide sterilization cycle produces no degradation of performance of the components.

2. Determine the effectiveness of the gas sterilization cycle in eliminating living matter from components.

2) Description of Test

The gas sterilization cycle will be used only on certain components which cannot be sterilized by heat and cannot be manufactured and delivered in a sterile state. The test cycle for the qualification test will be exposure to 12 percent ethylene oxide (88 percent freon 12, by weight) for a period of 32 hours. Temperature is maintained at 110 $\pm 10^{\circ}F$ with relative humidity of 35 to 90 percent.

At the end of the test cycle the sterilant will be purged with sterile dry nitrogen gas, then performance of the component will be measured.

On a sampling basis selected components will be provided sterile handling and the effectiveness of the gas sterilization in reducing living matter on the component will be evaluated.

3) Facilities and Test Equipment

1. Temperature test chamber with temperature programmable up to $150^{\circ}F$ and modified to handle ethylene oxide - freon 12 gas mixture.

2. Sterilization Area - The ethylene oxide testing requires an isolated, well-ventilated, protected test area providing personnel safeguards.

(4) Handling Shock Test

(a) Drop Test

1) Objectives

To verify that the component in its normal shipping container or packing can survive the shocks encountered in field handling.

2) Description of Test

The component in its minimum shipping container is subjected to drops onto a concrete floor landing on the container corners or edges. Six drops from heights specified in the component test specification constitute the test. Upon completion of the drop tests the component is inspected, energized, and operated and performance monitored.

3) Facilities and Test Equipment

Component test equipment to monitor component performance.

(b) Bench Handling

1) Objective

Verify that the component will survive the shocks of bench handling.

2) Description of Test

The component with at least one point or edge in contact with the wooden workbench and another edge or face at the angle specified in the component specification is allowed to drop through this angle to the table. Test is repeated for a total of 6 drops in 6 different positions.

At the completion of the test the component performance is evaluated.

3) Facilities and Test Equipment

Component performance test equipment.

(5) Transportation Vibration

(NOTE: The shipping container or packaging for the components will be designed to provide enough isolation and protection that in general the component will see very low levels of vibration during shipment, and the transportation vibration test can be omitted. Additionally any malfunction will be detected in subsequent tests.

(6) Humidity

(a) Objective

Verify that the component can operate after surviving extreme humidity conditions which can be encountered during unprotected phases of shipping and handling at the launch site, or other periods when the components are unprotected from the moisture.

(b) Description of Test

The component is tested in a temperature humidity chamber for the temperature-humidity cycle for test duration specified. Test cycle from ambient temperature to 160°F at 95% to 100% RH to ambient temperature is recommended. Test cycle to last 24 hours, 10 cycles required.

At the completion of the humidity-temperature cycles, the component is air dryed, then operated and performance tested.

(c) Facilities and Test Equipment Required

Temperature-humidity chamber programmable to cycle from ambient temperature up to 160°F with relative humidity of 95 to 100 percent. Chamber size for component test up to 3 x 3 x 3 feet or 4 x 4 x 4 feet.

(7) EMI Tests

(a) Objectives

All electrical and electronic components are tested to determine the amount of electrical interference they will radiate or conduct back into the connecting circuit.

(b) Description of Test

The component with required power supplies and loads is setup, and the radiated and conducted noise is measured over the specified frequency range. Testing will be to the requirements of MIL-I-26600.

(c) Facilities and Test Equipment

Screen room with attenuation of 100 db over a frequency range up to 10 gc and meeting requirements of MIL-STD-285 and MIL-I-26600.

Signal Generators
Antenna
Noise and field intensity meters and power meter.

(8) Static Acceleration

(a) Objectives

Verify that the design is capable of surviving the static acceleration to be experienced by the spacecraft during launch vehicle engine operation.

(b) Description of Test

Each component will be acceleration tested on a centrifuge along each of the three main axes and in each direction to the acceleration levels specified in the component test specification. Those components which normally function during the powered phase of flight will be performance tested and inspected for damage after the acceleration test.

(c) Facilities and Test Equipment

Accelerator - up to 400 pounds capacity. Acceleration 0 to about 40 g. Slip rings for power to the component and for monitoring instrumentation to be provided.

(9) Vibration

(a) Objectives

Determine the ability of the components to survive the vibration environment experienced during launch vehicle engine operation and during midcourse corrections.

(b) Description of Test

The components will be mounted to the vibration table or an oil film supported table using fixtures providing the necessary rigidity with no resonances. The vibration test will be performed with inputs along each of the three main axes in turn. Sine and random vibration tests are conducted in accordance with the specification.

Those components which normally operate during powered flight are energized and performance monitored during the vibration test. The other components are inspected and performance monitored at the completion of vibration test.

(c) Facilities and Test Equipment

1. Vibration Test Facility with sine and random test capability, MB-C125 shaker, or equivalent, with signal generators and filters and shaping network and automatic equalizer-analyzer for sine and random test

2. Auxiliary table, oil film supported, for performing tests in 3 axes

3. Overhead suspension system for suspending large and bulky components or assemblies, such as the high gain antenna, during test

4. Magnetic Tape controlled programmer used to program vibration test system in accordance with pre-recorded vibration programs

5. Tape recorders to monitor component response and performance

6. Accelerometers with power supplies and amplifiers to monitor vibration levels

7. Test fixtures to support component during test

8. Hydraulic Test Facility required to perform sinusoidal test at low frequency and with large displacements (if required by specification)

9. Component Performance Monitoring Instrumentation.

(10) Shock

(a) Objectives

Determine the response of the spacecraft components to shocks of separation and mid-course maneuvers during vehicle flight.

(b) Description

Mount the component on the platform of the shock test machine. A fixture with no low frequency resonances must be used. Shock test is performed in each direction along the three major axes. Shock magnitude is as described in the component test specification. Performance will be determined at the conclusion of the shock test.

As an alternate it may be more efficient to conduct the shock test on the vibration machine using a shaped pulse into the vibration system amplifier.

(c) Facilities and Test Equipment

1. Shock Test Machine with 100 pounds capacity at up to 50 g's with half sine wave and sawtooth shock pattern.

2. Oscilloscope for shock pattern measurement

(11) Acoustic Noise

(a) Objective

Determine response of the spacecraft components to the acoustic noise of powered flight.

(b) Description

The component is suspended from the test chamber by a soft suspension system having a low natural frequency. The component is tested at a sound pressure level of up to 160 db random noise over a broad spectrum. Test time will be 30 minutes. Only those components which operate during powered flight will be energized and operated during the acoustic test. The other components will be inspected for structural damage after test, then operated and performance tested.

(c) Facilities and Test Equipment

Acoustic test chamber with random sound generators capable of up to 160 db. Microphones and suitable instrumentation to measure and monitor sound input levels.

(12) Thermal Vacuum Test

(a) Objective

1. Determine the operating characteristics of the spacecraft components under thermal-vacuum environments corresponding to space flight application.

2. Determine whether thermal balance design is satisfactory.

(b) Description of Test

1) Components Mounted Internally in the Spacecraft

The component is mounted on a mounting plate, temperature controlled by a circulating fluid passing through a heat exchanger. Temperature is maintained at that temperature the component mounting in the spacecraft is designed to maintain.

Components located internally in the spacecraft are shielded from radiating directly to the vacuum chamber cryogenic walls by a shield simulating the spacecraft interior structure and maintained at its design temperature.

The component under test is operated in accordance with its individual component specification and the performance monitored. Test time under vacuum will be 1000 hours. Vacuum will be 10^{-6} torr or less. At the completion of test the component is removed from the vacuum chamber and performance tested under laboratory ambient conditions. Change in performance from original performance test is evaluated.

2) Components Mounted Externally to Spacecraft

The component is mounted on the temperature controlled mounting plate in the thermal vacuum chamber. Temperature is maintained at the design temperature of the component mounting surface on the spacecraft exterior. Simulated solar radiation programmed to the simulated mission requirements of flight from Earth to Mars or Venus is provided. The component mounting is exposed to the chamber cryogenic walls and to the simulated solar radiation corresponding to the component location on the spacecraft and considering mission profile and trajectory and spacecraft orientation with respect to the sun.

The component is operated and performance monitored in accordance with the individual component specification. The vacuum chamber is operated at 10^{-6} torr, or less, and test time will be 1000 hours at vacuum.

At the conclusion of the thermal vacuum test, the component performance is evaluated under laboratory ambient conditions and performance is compared with pre-vacuum test performance for change.

3) Facilities and Test Equipment

1. Component-size thermal vacuum chambers. Approximately 5 ft x 5 ft diameter vacuum chambers with 10^{-6} torr or better vacuum capability with liquid nitrogen cooled walls. Chamber equipped with temperature controlled component mounting surfaces controllable up to $200^{\circ}F$ and, possibly down to near liquid nitrogen

temperature. Small components, especially those not requiring solar simulation, can be tested in approximately 25 in. x 25 in. diameter chambers with cryogenic walls.

2. Solar Simulator required to illuminate the component under test. Output must be close to true solar spectral distribution as per Johnson's curve, collimated and having intensity of 1.4 KW/m^2 for near Earth flight, 2.67 KW/m^2 for Venus and 0.62 KW/m^2 for Mars.

(13) Sand, Dust Tests

Sand and dust tests will be conducted on Lander components to verify their ability to withstand the anticipated environments of Mars. Tests will be conducted in a commercial sand and dust test chamber.

F. Qualification Test of Special Components

Squibs and Other Electrically Actuated Explosive Devices

(1) Objectives

Determine that the design is satisfactory for application to the program.

(2) Description of Test

The test units for design qualification test will be selected in random fashion from a single lot of squibs.

The selected squibs will be further divided into test groups for submission to the various tests as required by the test specification. Squibs selected for testing are those which have gone through the regular production and quality control checks.

The following tests are conducted to qualify the explosive devices:

1. Resistance check of squib - each squib is checked

2. All Fire Test - the squibs of Group 1 will be fired in a Bruceton analysis to determine the minimum all-fire current. Initial firing current and incremental steps and sample size are defined in the component specification.

3. No fire tests - the squibs of Group 2 will be fired in a Bruceton type analysis to determine the maximum no-fire current (below which the squibs will not fire). Initial starting current and incremental steps for the Bruceton analysis are as listed in the specification.

4. Radio-frequency Hazard Test - the squibs of the third group from the lot will be tested to determine RF susceptibility to firing the squibs. Connected to an RF signal generator and impedance matching network, the level of RF power at various frequencies and the time required to fire will be determined by the Bruceton method of analysis.

5. Autoignition - the autoignition of each squib from the fourth group of the lot will be determined by placing each squib in a chamber at 160°F and raising the temperature gradually until the squib fires. (Those that do not fire above specified cut-off temperature will be set off electrically.)

6. Jolt Test - the squibs from the next group will be subjected the "jolt" test of MIL-STD-300 and then fired under ambient environment.

7. Environmental Tests - Groups of squibs as specified will be fired after exposure to conditions of temperature-humidity, and vibration. One group of squibs will be fired under vacuum.

8. Calorimeter Firing - Samples of the squibs are fired in a calorimeter and the heat of explosion measured.

(3) Facilities and Test Equipment

1. High-low temperature chamber - working volume of 2 x 2 x 2 feet and with temperature control from 200°F down to -80°F.

2. Squib Firing chamber

   a. With provision for maintaining high temperature to nearly 200°F low temperature to nearly -80°F and vacuum to 10^{-5} torr.

   b. With suitable firing timing circuits and pressure-time firing pattern measurement.

3. RF test equipment

   Signal generators
   Impedance matching networks
   Oscilloscope
   RF attenuators
   Power meter and related RF test equipment

4. Calorimeter for measuring heat of squib firing

5. 1000-volt megger tester, or equivalent

6. DC power supplies 24 volt, 0 to 5 amps.

3.7.7 SYSTEMS QUALIFICATION TEST PLAN

A. Definition of Qualification Test

The qualification program is a series of tests and evaluations made on a prime production spacecraft to demonstrate that the system design meets the spacecraft system design requirements. The spacecraft will be programmed through a series of tests, the test levels of which will be of a severity sufficient to assure a demonstration of performance exceeding environmental levels expected to be encountered in flight.

The test levels are regarded as high enough to reduce the potential operating life time of the hardware being tested, therefore test hardware will not be used in flight.

B. Objectives of Qualifications Test Program

1. Demonstrate that the spacecraft design has met the system design requirements and that the spacecraft is capable of performing under the expected flight conditions described in the system specifications.

2. Verify a margin of safety over expected flight environments and demonstrate that the system will function satisfactorily.

3. Establish reliability data showing achieved reliability under conditions comparable to or more severe than flight environments.

4. Determine modes of failure so that design improvements can be incorporated into the flight system prior to flight.

5. Develop skill and efficiency in operating and testing the first flight type system, improve test procedures, gain proficiency in operating the spacecraft and test complex so that operations in the acceptance tests and in the field will be efficient and free from operator and equipment errors.

C. Test Hardware Needed

The qualification test will be performed on a complete spacecraft system consisting of the Orbiter and the two Landers. The spacecraft will be of flight configuration, including the high gain antenna and normal flight configuration solar power equipment.

The system presented for test will be the first available prime production spacecraft, assembled by the Manufacturing Group and representative of a "typical" production flight unit. The test system will be subjected to an acceptance test prior to the qualification test to provide assurance that it is a representative and functional production spacecraft. The acceptance test will include as a minimum a system alignment, weight and center of gravity determination, pneumatic test, and a thorough functional performance test. These tests will serve as a reference against which to evaluate the effects of the environmental tests.

The shipping containers for the Landers, Orbiters and the antenna assembly, assuming these sub-assemblies are shipped separately, will be required to support the transportation and handling tests.

For the system qualification test flow plan see Figure 3.7.7-1.

D. Test Equipment and Facilities

Functional performance test equipment will be required to power the spacecraft, provide input stimuli and commands to the spacecraft, and measure and evaluate the response. This test equipment will consist of the automatic systems test and checkout set and the systems ground station with receiver discriminators, magnetic tape recorders, and some visual display and readout.

In addition, the special test equipment and facilities required to support particular tests are described with the test.

E. Qualification Test Ground Rules

(1) The Lander will be qualification tested as an individual system where required to establish its capability of performing under the particular environments it encounters in entry, descent, and landing.

(2) Where it is economical to combine the Lander and Orbiter tests into one systems test, this economy will be made. To conduct test of the spacecraft in the transit flight phase it is necessary to test a combined Orbiter-Lander.

(3) Qualification test of the Voyager Spacecraft System will be completed prior to the first launch.

(4) No repairs, replacement of components or significant adjustments to the system will be permitted during the course of testing without a study of the cause of failure and the effect on results of the tests completed so far. A failure investigation will be conducted and appropriate corrective action taken to eliminate the recurrence of such failure

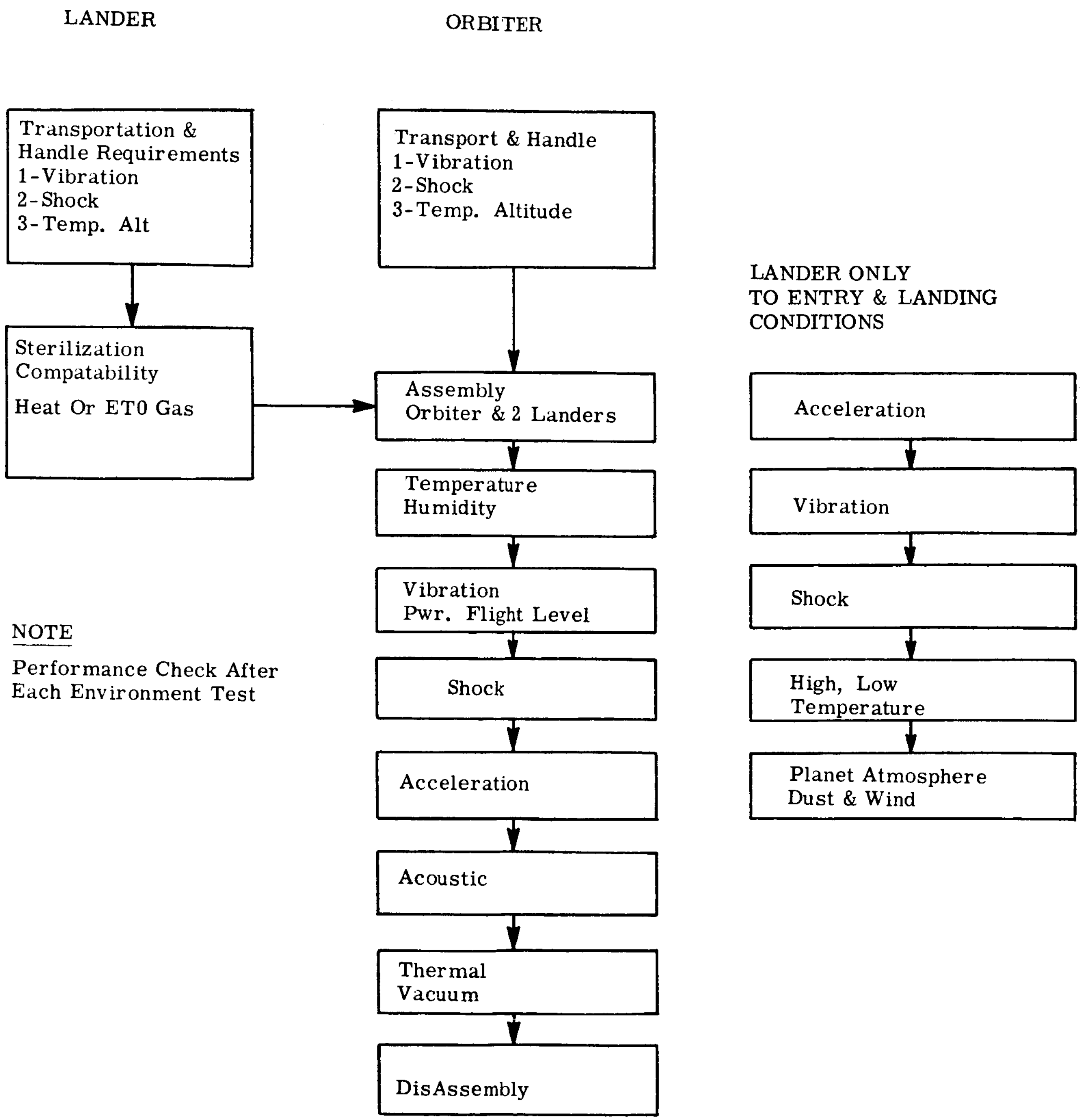


Figure 3.7.7-1. System Qualification Test Flow

or deviation from performance specification. Any action on failures during the qualification test will be taken by the Integrated Test Program Board after analysis of effects of failure on tests already completed.

F. The Integrated Test Program Board

The Integrated Test Program Board (ITPB) is established to monitor the qualification test program, analyze the test data, and determine whether the qualification test requirements have been fulfilled. The ITPB will review any failures and determine the effect on testing completed to date, and recommend the correct course of action to be taken towards further testing. The ITPB will monitor the implementation of the qualification program and issue periodic reports giving the status of the program.

G. System Qualification Test Cycle

The qualification test cycle shown in Figure 3.7.7-1 is proposed. Testing of the Lander and Orbiter separately in their transportation status will be performed to simulate the environment and the handling the units see during shipping.

The next series of tests is intended to subject the spacecraft system to the conditions it encounters during powered flight.

Finally, a series of tests on one Lander which has gone through the above cycle of tests, will determine the Lander performance through planet entry and landing.

A functional performance test of the unit under test is required after each environmental exposure. Performance after environmental exposure is evaluated against original performance and degradation is established.

In each case the performance test is made using the systems checkout and test set provided for systems (or Lander) test.

(1) Transportation and Handling Test - Lander and Orbiter

(a) Test Objectives

1. The transportation and handling tests are performed to establish that the flight vehicle is capable of surviving without failure the conditions it encounters in being shipped from the factory to the Florida launch site.

2. Since the transportation tests are run with the spacecraft assemblies in their respective shipping containers the test is also a qualification test of the shipping containers.

(b) Description

The Lander, Orbiter, Planet Horizontal Package (PHP) and the antenna assembly will be mounted in their respective shipping containers. The specified vibration and shock inputs will be applied to the base of the shipping containers through a rigid fixture. Using the same test set up the specified shock form can be programmed efficiently into the vibrator system amplifier and the shock test completed. At the completion of the handling and transportation tests the shipping containers and the spacecraft assemblies are visually examined for damage or failure.

Vibration tests are performed in accordance with specification MIL-STD-810, Method 514 with the vibration input to the shipping containers being:

1.3 g's	5 cps to 26 cps
0.036 inches double amplitude	26 cps to 50 cps.

Dwell time is 30 minutes at each resonant frequency.

Shock tests requirements:

Input to shipping containers: 30 g's 11 ms half sine wave shock input to each of 3 axes, 3 shocks each axis, each direction.

NOTE: It is assumed that the spacecraft will be shipped disassembled into the Lander, Orbiter, PHP, and the antenna assembly, rather than as one large assembly.

One of the major problems expected to be encountered in the transportation-vibration test will be to provide a rigid fixture suitable for supporting the hardware under test without introducing resonances into the test.

(c) Facilities

A large vibration test facility similar to the MB-C210 with force output per vibrator of 28,000 pounds is required to perform the vibration test. The amplifier and power supply is controlled by an automatic equalizer-analyzer control to compensate for table-fixture-test specimen loop resonances. Test time is thus reduced over the time required to set up manually the peak notch filters required to shape the vibration test frequency spectrum.

Shock tests can be performed using the same facility and controlling the amplifier with a programmed shock wave.

(2) Temperature-Altitude Transportation Test

(a) Test Objective

Demonstrate that the spacecraft assemblies will operate after exposure to the transportation conditions of an unpressurized high altitude aircraft.

(b) Description

The spacecraft assemblies - antenna, Orbiter, Lander and Planet Horizontal Package protected by their shipping containers are tested at a pressure corresponding to 50,000 feet altitude and at a temperature of -35^{o}F of restricted air transport. Test duration is 8 hours. At the end of the test, the assemblies are performance tested and inspected.

The temperature-altitude test is based on the assumption that the Orbiter assembly can be transported in an airplane. It is assumed that an aircraft such as the special purpose "Pregnant Guppy" B377PG by Aero Spacelines being built now for transportating the Saturn - S-IV stage will be capable of transporting the Orbiter assembly with some of the appendages such as the PHP, and the antenna support removed. If air transportation cannot be accomplished the above test will be deleted.

(c) Facilities and Equipment

An altitude temperature chamber capable of containing the individual spacecraft assemblies in their shipping containers is required. Temperatures range to -35^{O}F, altitude to 50,000 feet and chamber size up to 20 feet in diameter and 14 feet high.

(3) Sterilization Compatibility Test - Landers

(a) Test Objective

Verify that the spacecraft will perform within specification after being subjected to the sterilization cycle.

(b) Description

The sterilization test will be performed on the complete Lander minis any components which will not withstand sterilization by heat. The Lander will be placed in a temperature chamber and the temperature gradually raised until the Lander stabilizes at 145°C. This temperature will be held for 36 hours. Then the Lander is allowed to cool to room temperature. Test is repeated for an additional 2 test cycles to evaluate the effect of re-sterilization. At the completion of each of the temperature cycles, the performance of the Lander is evaluated to establish any performance degradation. Sterile handling procedures used will be those developed on the D-2 Development Spacecraft Tests described in Section 3.7.5B(2). The purpose of this test is to establish degradation due to the high temperature and to prove the feasibility of the handling procedures. At the completion of the test no subsequent efforts are made to maintain sterilization.

(c) Facilities and Special Test Equipment

1. Sterilization Facility

Sterilization facility will be a clean aseptic area where any required disassembly or rework can be performed and where the Lander can be heat sterilized at 145°C + 2°C for 36 hours. Facilities are available for sterilizing by gas or other means the components not heat sterilizable. Suitable work areas are available for performing the packaging of the Lander in a biological barrier after the sterilization has been completed.

The Sterilization Facility is described in the Sterilization Section of the report. See Volume V.

2. Lander Performance Test Equipment

The automatic system test and checkout set and other required Lander performance test equipment will be available to check out the Lander after each heat cycle.

(4) Temperature-Humidity Test - Voyager System

(a) Test Objective

Verify that the spacecraft will operate within specification after being subjected to a severe humidity environment similar to that likely to be encountered during operation and checkout on the launch pad.

(b) Description

The spacecraft will be exposed to 95% to 100% humidity cycles at a temperature of up to 160°F. The temperature will be cycled from 160°F to ambient temperature. Temperature cycling will induce the breathing in the system and be the more severe test condition. At the end of the cycling period, the spacecraft will be air-dryed and performance tested. Test performance data after the humidity test will be compared with the reference performance data to determine deterioration of performance.

Humidity test cycle is from ambient to 160°F 95% to 100% relative humidity (RH) in 2 hours. Maintain temperature for 6 hours. Over the next 16 hours, lower temperature to ambient again. Test cycle to be repeated 10 times, to make a 10 day test.

(c) Facilities

Humidity Chamber - Programmable 95 to 100% RH over temperature range of ambient to 125°F. Volume sufficient so spacecraft occupies no more than 50% of chamber volume. Approximately 20 feet diameter by 14 feet high.

System performance test equipment is necessary to measure performance of spacecraft.

System test fixture is required to support the spacecraft during test.

(5) Vibration Test - Powered Flight Environment

(a) Test Objective

To evaluate the performance of the Voyager Spacecraft when it is subjected to the sine and random vibration levels expected during the powered portion of flights.

(b) Description

To accomplish the vibration test the spacecraft is mounted to a rigid vibration fixture which will transmit vibration table acceleration levels without excessive magnification or attenuation throughout the test frequency range. The fixtures used will be those fabricated for the development vibration test and updated and modified as required by differences between the qualification spacecraft and the development unit.

Complex (sine and random) vibration tests will be performed along the 3 main vehicle axes to the levels and for the time duration called out in the Qualification Test Specification. Resonant frequencies will be maintained for the period of time specified.

The spacecraft will be in its powered flight status during the test. The system will be energized and data will be transmitted using the on board telemetry and communications equipment. Additionally, the spacecraft will be instrumented with accelerometers and strain gauges to record test levels and amplification within the spacecraft structure.

At the completion of the test, the entire spacecraft system will be exercised and the performance of the system will be evaluated and compared with pre-vibration test performance. A detailed visual examination will be performed to locate any structural failures.

(c) Facilities and Special Test Equipment

1. Large vibration test system of the capacity of the MB-C-210 with 2 vibrators or equivalent. System must have the signal generators, filtering and shaping networks and, preferably, the automatic equalizer-analyzer system required to shape the spectrum for a random test.

2. Auxiliary table, oil film supported, or equivalent support/suspension system to position and support the spacecraft to accomplish a 3 axis test.

3. Fixtures required to support the spacecraft for a 3 axis test.

4. Magnetic tape controlled programmer to program vibration tests in accordance with pre-recorded vibration profiles (if required by specification).

5. Tape recorders to monitor system response and performance.

6. Accelerometers with amplifiers and power supplies, and strain gauges and amplifiers to monitor vibration levels.

7. System performance test equipment.

   Systems checkout and test set including ground station to monitor performance of system during the vibration test.

(6) Shock-Powered Flight

Shock environment is expected to be covered by the vibration test and no separate test is required. If the shock requirement is found to be in excess of vibration requirements, the shock test will be made with the spacecraft on the vibrator. A shaped pulse of proper duration and amplitude will be programmed into the shaker system and the shock-vibration test completed in one set up.

(7) Acceleration-Powered Flight

(a) Test Objectives

1. Demonstrate that the structure can carry the stresses induced during powered flight.

2. Demonstrate that the system can operate while being subjected to high acceleration loads.

(b) Description

The spacecraft will be mounted on the accelerator and then spun up and performance of the components which operate during powered flight will be monitored. Tests will be performed in 3 axes, both directions as specified. At completion of test, the spacecraft will be inspected then performance tested. Tests will be set up and performed similar to the Development tests on D-1 spacecraft.

(c) Facilities

Large accelerator capable of driving 7,500 pound spacecraft and fixture to 10 g's in longitudinal direction. The Sandia centrifuge has the capability of accelerating 10,000 pounds to 450,000 g-pounds and should be suitable.

Fixture to support spacecraft for performing test along 3 axes in both directions. Performance test equipment to monitor system performance after acceleration.

(8) Acoustic Noise

(a) Test Objective

Determine whether the sound pressure levels encountered during engine operation will affect the performance of the spacecraft.

(b) Description

The spacecraft in launch configuration will be supported in the acoustic chamber by a soft suspension having a low natural frequency. The spacecraft will be subjected 160 db sound pressure level covering a broad spectrum of the audio range. Test time will be 30 minutes. Only those components which operate during powered flight will be energized and monitored. At the completion of the test, the system will be inspected for structural damage, then performance tested.

(c) Facilities

Acoustic chamber with random sound generators capable of up to 160 db. Chamber volume sufficient to suspend the entire spacecraft. Microphones and suitable instrumentation to measure sound level.

System performance test equipment is used to evaluate performance during and after test.

(9) Thermal-Vacuum Test

(a) Test Objectives

1. To provide assurance that the over-all design and especially the thermal design is capable of operating in a simulated space environment corresponding to Mars.

2. Measure the performance at the components and subsystems of the spacecraft under a relatively long term environmental test.

3. Collect data to establish the achieved reliability of operation in a simulated space environment.

(b) Description

The complete spacecraft will be mounted on a fixture in the vacuum chamber. Since the spacecraft maintains the same orientation to the sun during flight, the spacecraft will not be gimbal mounted under the solar simulator, but will remain fixed on the fixture.

The spacecraft will be mounted in its deployed configuration such as it maintains through the transit flight. The adapter will be off, and the deployment mechanism will be in the tracking position. After set-up, the spacecraft will be operated and performance checked on the telemetry diagnostic instrumentation. At the ground station, selected test circuits brought out through vacuum chamber penetrations will be monitored. After adequate performance of the spacecraft and the test equipment is assured, the chamber will be evaluated to 10^{-6} or better, and testing will begin. Solar illumination, ranging from 1.4 kw/m^2 of near Earth environment to 0.62 kw/m^2 of near Mars environment will be programmed to the spacecraft. The need for Earth albedo and Mars albedo simulation will be investigated. The spacecraft will be exposed to liquid nitrogen, helium cooled black walls (E = 0.9) of a temperature of about 100°K.

The thermal-vacuum test is an excellent opportunity to exercise and operate all the components and subsystems for a relatively long period of time in a space environment quite close to that expected in transit flight. The components and subsystems will be operated, stimulated and the responses and operating characteristics evaluated. Operation of the attitude control and guidance sensors and control loops in responding to programmable simulated sun, Earth, star and planet sources will be measured and evaluated. Pneumatic leakage will be checked. The complete thermal interactions of the spacecraft, the heat generating components and the sun and the cold black space environment will be evaluated.

At the conclusion of the test, the vacuum chamber is pumped up and a complete performance evaluation is made.

Vacuum test duration: 1000 hours.

(c) Facilities

Thermal-vacuum chamber - About 24 feet working diameter, minimum height at least 20 feet with vacuum capability of 10^{-6} torr or better. With 100°K walls with emissivity of 0.9.

Fixture to support the spacecraft - solar illuminator capable of illuminating an area approximately 20 feet in diameter to level of 1.4 kw/m programmable to 0.62 kw/m with uniformity and good collimation (about 5%).

System Performance Equipment - Systems Test set including ground station. Must also have capability of monitoring numerous temperature points, chamber pressure, solar illumination input and distribution.

(10) Lander Acoustic Noise

(a) Test Objective

Subject the Lander to an acoustic environment representative of entry on Mars and verify that no degradation has occurred relative to the performance of the required scientific experiments.

(b) Description

The Lander shall be placed in an acoustic chamber and subjected to an environment of 160 db for a period of ten minutes. At the end of this time, a performance check shall be conducted to verify that system performance is acceptable. The Lander shall be examined for structural misalignment or failure. A performance check shall be conducted following acoustic exposure.

(11) Lander Vibration/Shock Test

(a) Test Objective

To subject the Lander to a simulated Mars entry vibration and shock environment and determine if deterioration in vehicle structure, instrumentation, or functional performance occurs.

(b) Description

The Lander shall be placed on a vibration table of the capacity of the MB-C-200 or C-210 or equivalent rating and subjected to combined sine and random inputs. Vibration shall be conducted on 3 axes and accelerometers will be used to monitor vibration response on critical components. Following the vibration test, a performance check will be made to verify mission adequacy.

With the Lander in position on the vibration table, a series of programmed shocks will be imposed on the vehicle. Structural response and integrity will be evaluated. A performance check will be conducted following the shock test. Shocks will be conducted in two directions on three mutually perpendicular axes.

(12) Lander Acceleration Test

(a) Test Objective

The object of this test is to subject a Lander vehicle to simulated powered flight accelerations and entry to assure that no system degradation occurs.

(b) Description

A Lander vehicle shall be mounted on a fixture such that, when placed on the centrifuge, acceleration loads parallel and angled to the roll axis can be simulated. Maximum acceleration forces simulating 12g for powered flight and 125g for Mars entry are desired. During the test the Lander will be energized from the system test set and data will be transmitted from the Lander to the ground station. Data will be recorded on magnetic tape and analyzed. Additionally, output from accelerometers and their amplifiers, strain gauges and other instruments will be read off from the centrifuge slip rings and recorded and monitored.

Tests will be run at a large centrifuge such as Sandia, or Edwards Air Force Base.

(13) Lander Planet Atmosphere Test

(a) Test Objective

The object of this test is to verify that the Lander Spacecraft can operate in the anticipated Mars planet environment.

(b) Description

The test will be performed with the Lander Spacecraft operating in the reduced planet atmosphere pressure and high and low temperature, while being subjected to dust and simulated wind currents. Attention will be given to evaluating whether electrical circuits arc over or fail with reduced atmospheric pressure and whether mechanical devices can operate and deploy with sand and dust accumulation and under high and low temperatures.

(c) Facilities

A special test chamber corresponding somewhat to a conventional sand and dust chamber but with high and low temperature control and with reduced pressure capability will be required.

3.7.8 ACCEPTANCE TEST PLANS

A. Objectives of Acceptance Tests

Acceptance tests are formal documented tests performed as a quality assurance measure to verify the workmanship and practices in assembling the spacecraft and its components, and to demonstrate that component and systems performance meets the specified requirements. Acceptance tests verify the hardware performance under specified environments corresponding to flight environments.

Additionally, the components and system acceptance test will include an extended period of operation during which the units will go through their simulated mission profile. This testing is intended to take the components and the system through their high infant mortality failure rate regime of operation. Performance data from these extended operating tests will measure the achieved reliability of the system and help to predict reliability of performance in flight.

B. Ground Rules

For the Voyager Program acceptance tests will be performed at the component level and at the complete system level. (This does not imply that in-process type quality assurance tests will not be made at the part or module or assembly levels.)

Acceptance tests are performed on each spacecraft component intended for flight spacecraft application, for spares application, or for use in the qualification test program. Additionally, each component intended for application to GSE to be used for flight support or support of acceptance test or qualification tests is acceptance tested to AGE requirements.

Acceptance tests are performed on each spacecraft system intended for flight or for flight spare use or for application on qualification test programs.

Acceptance tests are performed on the Landers separately to include the entry, landing requirements.

C. Component Acceptance Test Cycle Summary

(1) Spacecraft

In general the spacecraft component acceptance test cycle consists of:

1. Visual inspection
2. Laboratory ambient performance test
3. Vibration test
4. Performance check and inspection to establish change in performance due to vibration test.
5. Thermal vacuum test
6. Performance check
7. Inspection and buy-off.

The above test sequence does not apply to explosive devices such as squibs, or to rocket engines.

Additional tests will be added to individual component test cycles as required.

(2) GSE

The GSE component acceptance tests include only a visual inspection and a laboratory performance test in accordance with the individual component specification. Environmental performance tests are not included.

(3) General Test Requirements

(a) Test Specifications

Component test specifications give the detailed performance requirements each component must meet.

(b) Adjustments and Repairs

No repairs or adjustments are made during the acceptance test cycle. The need for adjustment during the acceptance test cycle is regarded as a test failure. Failure analysis will be initiated and corrective action taken to prevent recurrence of the failure.

(c) Acceptance Test of Purchased Components

In many cases it is more efficient to perform the acceptance test of certain purchased items at the vendor facility. This is especially true where elaborate and expensive special testing facilities are required. For example, the acceptance test of RTG power supplies and the propulsion units will be performed at the vendor's facility. In these cases the acceptance test is performed by the vendor with the tests witnessed and approved by the quality control surveillance inspector. The same test requirements as for in-house manufactured items apply.

D. Spacecraft Component Test Cycle

(1) Inspection

(a) Test Objectives

A detailed examination and inspection will be made of all components presented for acceptance test. The objectives of these inspections are to:

1. Review the travel cards, and data sheets, and assure that the necessary receiving inspections and in-process checks and inspections have been accomplished, and the unit is ready for acceptance test

2. Examine the component and the drawings and specifications to assure that the unit has been built to the correct design change

3. Perform visual examination of the component to assure that no damage has occurred to the unit since it was given a detailed inspection and buy-off by the in-process inspector or the receiving inspector (for purchased items)

4. Weigh the component and enter the data into the weight control log to furnish data needed to maintain weight and center-of-gravity control within specifications.

(b) Facilities and Equipment

Inspection will be accomplished using conventional inspection and measuring equipment. Chiefly visual inspection will be involved, performed with the aid of stereo microscopes if required. Weighing will be performed on conventional platform balances.

(2) Performance Test - Laboratory Ambient

(a) Test Objectives

1. Verify that the component operates under controlled laboratory conditions

2. Establish a base of performance from which change of performance caused by environmental test can be evaluated

3. Verify compatibility of test equipment and procedures beyond that already accomplished in the development tests.

(b) Description

The component is set up and operated in the normal laboratory ambient environment. The expected normal input is provided to the component and the performance and response to the operating conditions called out in the component specification is monitored. The component is operated and the test monitored by the same test equipment required for the subsequent environmental test, as far as possible. Operation of electronic components under over-voltage and under-voltage conditions is included, as required by the component specification.

This same performance test is repeated after exposure to the vibration and thermal-vacuum tests.

(c) Facilities and Test Equipment

Component performance monitoring equipment such as power supplies, recorders, signal generators, ammeters, voltmeters, etc. are required. Standard type laboratory measuring equipment is used to measure performance. Where it will save operator time and reduce operator error, special test equipment will be assembled from these instruments, constructed to make testing as simple and nearly automatic and fool proof as possible. This same equipment is used to monitor subsequent environmental and post environmental tests.

(3) Vibration Test

(a) Test Objective

Demonstrate that the component can operate satisfactorily during or after vibration corresponding to that encountered during powered flight, separation and attitude control maneuvers.

(b) Description

The component is mounted on a resonant-free fixture by its normal mounting method and a vibration test in each of 3 axes with sine and random vibration input is performed. Test levels and duration are in accordance with the component test specification. Components which do not operate during powered flight need not be energized during test; at the completion of the vibration test they are examined for structural failure and their performance is checked using the laboratory performance test equipment described above. Components which operate during powered flight are energized and operated in flight mode during the vibration test and rechecked at the completion of the test.

(c) Facilities and Test Equipment

1. Vibration Test Facility with sine and random test capability, MB-C-125 shaker, or equivalent, with signal generators and filters and shaping network and automatic equalizer-analyzer for sine and random test

2. Auxiliary table, oil film supported, for performing tests in 3 axes for the larger components

3. Overhead suspension system - for suspending large and bulky components or assemblies - such as the high gain antenna - during test

4. Magnetic Tape controlled programmer used to program vibration test systems in accordance with pre-recorded vibration programs corresponding to launch vehicle vibration profile

5. Tape recorders to monitor component response and performance

6. Accelerometers with power supplies and amplifiers to monitor vibration levels

7. Test fixtures to support component during test

8. Hydraulic vibration test facility to perform sinusoidal test at low frequency and with large displacement if required by the specification

9. Performance monitoring instrumentation - special test equipment consisting of power supplies and instrumentation as required to energize the component, provide input stimuli, and monitor performance.

(4) Thermal-Vacuum Test

(a) Test Objectives

1. Demonstrate that the component will operate in accordance with the individual performance specification in a thermal-vacuum environment simulating the interplanetary thermal-vacuum environment as far as feasible

2. Demonstrate that the achieved thermal balance is satisfactory for space operation

3. Operate components for relatively long periods of time under simulated space application conditions to achieve component "burn-in" and reduce early high failure rate in system application

4. Obtain reliability performance data for component operating under simulated space application environment.

(b) Description

1) Components Mounted Internally in the Spacecraft

The component is mounted on a mounting plate and the temperature is controlled by a circulating fluid passing through a heat exchanger. Temperature is regulated to that which the component mounting in the spacecraft is designed to maintain.

Components located internally in the spacecraft are shielded from radiating directly to the vacuum chamber cryogenic walls by a shield of suitable emissivity and simulating the spacecraft interior structure in shadowing effect and maintained at its design temperature.

The component under test is operated in accordance with its individual component specification and the performance monitored. Test time under vacuum will be 150 hours in order to provide reliability data on component operation. Vacuum will be 10^{-6} torr or less.

At completion of test the component is removed from the vacuum chamber and performance tested under laboratory ambient conditions. Change in performance from original performance test is evaluated.

2) Components Mounted Externally to Spacecraft

The component is mounted on the temperature controlled mounting plate in the thermal-vacuum chamber. Temperature is maintained at the design temperature of the component mounting surface on the spacecraft exterior. Simulated solar radiation programmed to the simulated mission requirements of flight from Earth to Mars is provided. The component mounting is exposed to the chamber cryogenic walls and to the simulated solar radiation corresponding to the component location on the spacecraft and considering mission profile and trajectory and spacecraft orientation with respect to the sun.

The component is operated and performance monitored in accordance with the individual component specification.

Vacuum chamber is operated at 10^{-6} torr, or less, and test time will be 150 hours at vacuum.

At the conclusion of the thermal-vacuum test, the component performance is evaluated under laboratory ambient conditions and performance is compared with first test performance for change.

(c) Facilities and Test Equipment

1) "Component Size" Thermal-Vacuum Chambers

Approximately 5' x 5' dia. vacuum chambers with 10^{-6} or better vacuum capability with liquid nitrogen cooled walls capable of about 100°K. Chamber equipped with temperature controlled component mounting surfaces controllable from about 50°F to 200°F and down to near liquid nitrogen temperature, depending on the location and the temperature of the component mounting surface.

Small components, especially those not requiring solar simulation, can be tested in approximately 25 in. x 25 in. dia. chambers with cryogenic walls.

2) Solar Simulator

Solar Simulator is required to illuminate the component under test. Output must be close to true solar spectral distribution as per Johnson's curve, collimated, and having intensity of 1/4 KW/m^2 for near Earth flight, and 0.62 KW/m^2 for Mars.

3) Performance Monitoring Instrumentation

Special test equipment consisting of special power supplies and instrumentation to energize the component, provide input stimuli, and measure response.

(5) Inspection and Buy-Off

At the completion of the above test sequence a visual inspection of the tested component is made to determine that no damage has occurred to the component during the test cycle. The test data and the test requirements are examined and evaluated to determine that all requirements have been fulfilled. Test data is entered into the log book in accordance with the program requirements.

Review of the test program and test data with the customer's Quality Control representative as required by the program is held and the tested component is stamped off as accepted and submitted to bonded stock.

(6) Pyrotechnics Acceptance Tests

(a) Test Objective

Assure the quality of the explosive devices used for the flight units.

(b) Description

Acceptance test of the explosive devices will normally take place at the vendors facility under the surveillance of the vehicle contractor's Quality Control representative. Acceptance will be performed by lots. The lots will be further subdivided into test groups, which will be subjected to the tests specified in the test specifications.

NOTE: Design qualification and lot acceptance testing of the explosive devices will include virtually the same type of tests. Lot sizes will be different.

The following tests will be representative of the tests to be performed to accept a lot:

1) Electrical resistance measurement performed on 100% basis.

2) No-Fire Check

This establishes that no squibs will fire accidentally at the specified current for specified period of time. Test on 100% basis at specified fixed value current.

3) Dielectric Check

All squibs are subjected to the specified high voltage ac for specified number of seconds with voltage applied between the shorted leads and the test unit case. Value of resistance is measured.

4) All-Fire Test

One test group from the lot will be fired using a Bruceton type analysis to establish the minimum current value at which the all-fire test is begun as specified in the component specification.

5) No-Fire Test

The squibs from a second test group will be fired in a Bruceton type analysis to determine the current that squibs can withstand without firing. Initial current value for the Bruceton analysis will be specified in the component specification.

6) RF Firing

The squibs selected for the third test group of the lot will be fired by RF energy of specified frequency by connecting the squib to an RF signal generator with appropriate impedance matching network. The squibs are fired by RF power in a Bruceton analysis and the RF energy vs frequency curves are plotted.

7) Environmental Tests

Additional test groups from the acceptance lots are subjected to temperature-humidity, and vibration exposure as specified, and the electrical resistance of the squibs is measured.

8) Vacuum Firing

Samples from the lot that have been exposed to the environmental test are subjected to firing tests under vacuum.

9) Temperature Firing

Samples from the lot that have been exposed to the environmental tests above are subjected to firing tests under high and low temperatures as specified.

10) Calorimeter Firing

Samples of the squibs are fired in a calorimeter and the heat of explosion is measured.

(c) Facilities and Test Equipment

1) High-low temperature chamber working volume of 2 x 2 x 2 feet and with temperature control from 200°F down to -80°F.

2) Squib Firing Chamber

1. With provision for maintaining high temperature to nearly -80°F and vacuum to nearly 10^{-6} torr.

2. With suitable firing timing circuits and pressure time firing pattern measurement.

3) RF Test Equipment

1. Signal generators
2. Impedance matching networks
3. Oscilloscope
4. RF attenuators
5. Power meter and related RF test equipment

4) Calorimeter for measuring heat of squib firing.

5) 1000 V megger tester, or equivalent.

6) DC power supplies 24 volt, 0 to 5 amps.

(7) Propulsion Acceptance Tests

(a) Propulsion Component Acceptance

Acceptance test of the propulsion subsystem will be performed on the component and the subsystem level. Acceptance tests of the components will include the inspection and vibration tests outlined in the component acceptance test. For the mechanical components the thermal-vacuum test will in many cases not be meaningful and will be deleted.

(b) Propulsion Subsystem Acceptance

The complete subsystem must be performance evaluated as an acceptance test. This test is complicated by 2 factors:

1. The propellants are corrosive and must be thoroughly cleaned from the subsystem
2. The thrust chamber is ablative.

To accomplish the test it will be necessary to substitute a tank for the flight propellant tank and bladder assembly. This is necessary because of the danger of entrapping propellant in the tank and not being able to clean it. After a relatively long duration firing test the thrust chamber must be replaced and a calibration run of short duration performed on the new unit.

E. GSE Acceptance Tests

The Ground Support Equipment (GSE) unit will be inspected to verify correspondance to the applicable drawings and specifications. The units will be functionally tested to demonstrate performance as required by the GSE performance specifications. Tests will be conducted under laboratory ambient environmental conditions.

F. Spacecraft Systems Acceptance Tests

(1) Test Requirements

(a) Test Specifications

The system performance specification will detail the performance requirements to be met during the various tests.

(b) Adjustments and Repairs

No repairs or adjustments will be made during the acceptance test cycle. The need for any adjustment or repair during the test cycle will be regarded as a test failure. Failure analysis will be initiated and corrective action taken to prevent recurrence of the failure.

(c) Solar Array and High Gain Antenna Handling

Because of the potentially large size of the solar array and the high gain antenna and because of the lightweight construction, consideration will be given to the advantages of removing these assemblies during certain phases of the test cycle, and simulating their function. This will not be regarded as vehicle disassembly.

(d) The Lander capsule will be sterilized; the orbiting unit of the spacecraft will not be sterilized.

(e) Sterilization of the Lander will be performed on the complete Lander, rather than at the component level. Sterilization will be performed in the field, rather than in-house.

(f) Automatic performance test equipment will be favored because of reliability requirements and because of the extensive operating test requirements. See Figure 3.7.8-1.

(2) Spacecraft System Test Cycle

The spacecraft acceptance test will be subdivided into the Lander capsule acceptance test, and the complete spacecraft acceptance test. This is necessary because the environmental conditions that the Lander will encounter will be considerably more severe than those the orbiting vehicle will encounter. In addition to surviving launch and interplanetary journey to the planet, the Lander capsule must survive the heating, shock, vibration and acceleration of planet entry, descent through the planet atmosphere, and landing.

Therefore, the Lander will be acceptance tested, separately and delivered for assembly to the Orbiter. Testing of the complete system will then take place.

Similarly initial subsystem tests will be made on the Orbiter before the entire system is assembled, and testing will be completed on the complete Voyager Spacecraft System.

(3) Lander Acceptance Test Cycle

The Lander acceptance test cycle will follow the flow shown in Figure 3.7.8-2.

(a) Lander Systems Test

1) Adapter Subassembly

The complete adapter assembly will be assembled (including separation subsystem, less pyrotechnics) and will undergo sequencing check and firing checks using simulated pyrotechnics. Checks will also be performed on the thermal control subsystem.

2) Subsystem Calibration

With the subsystem on internal power the transmitters (relay link and back up link) will be adjusted for maximum power. Pre-emphasis adjustments on VCO's and deviation checks on real time data and playback data will be made.

Five (5) point VCO calibration check performed.

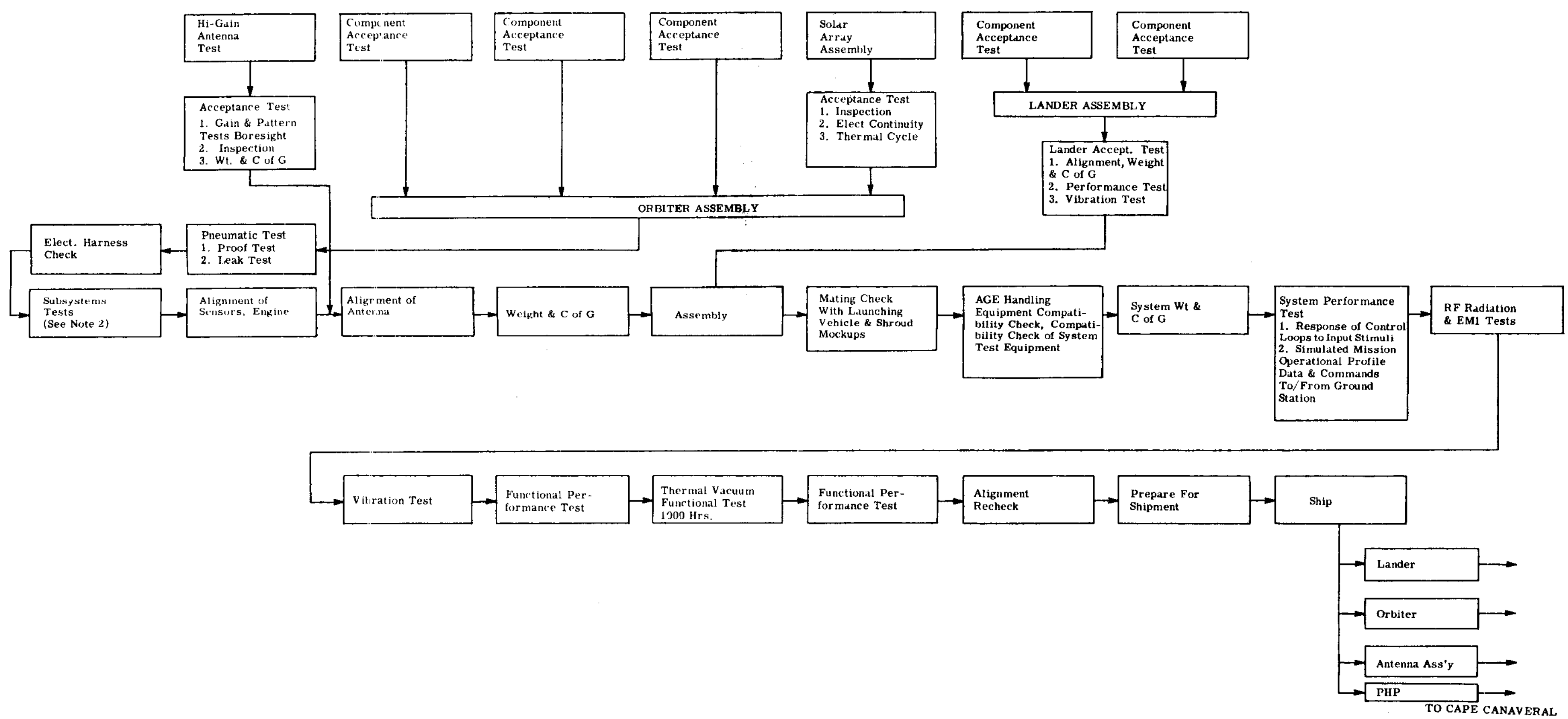


Figure 3.7.8-1. Acceptance Test Flow Diagram

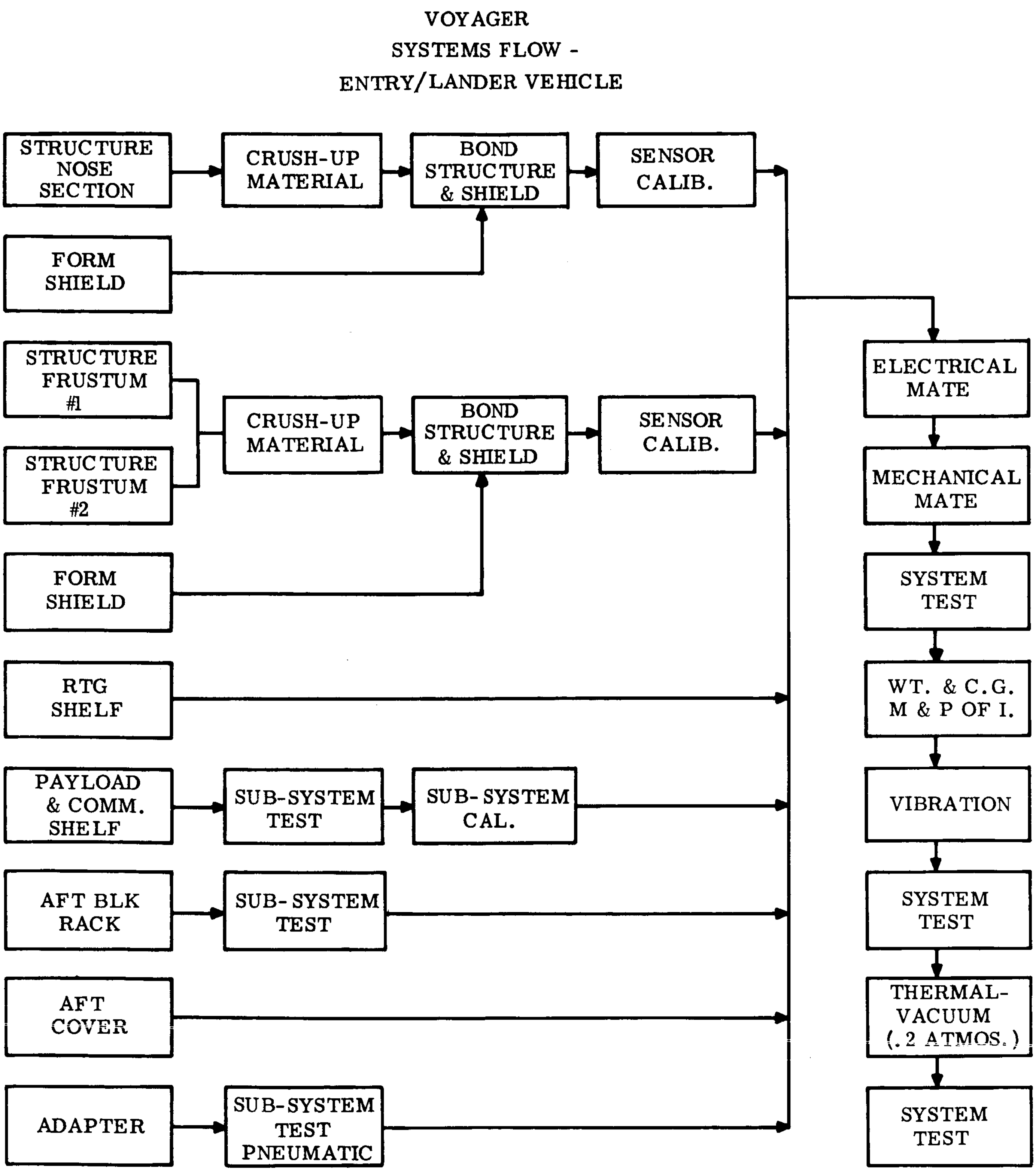


Figure 3.7.8-2. Voyager Systems Flow-Entry/Lander Vehicle

Real time and playback noise check.

Recorder playback and record speed check.

Simulate inputs to data processor decommutator.

Perform switching function from relay link to back up subsystem.

3) Electrically Mated Test

With the nose, frustrum no. 1 and 2, aft bulkhead, payload, and instrument shelf, and adapter electrically mated the complete system will be exercised to disclose any interference which may exist between any other subsystems. System electrical loading, command and telemetry shall be monitored to insure that system meets all requirements.

4) Mechanical Mate

The entire Entry/Lander Vehicle shall be mated to assure compatibility among the subassemblies.

5) System/Confidence Test

A system confidence test shall be performed after the Entry/Lander Vehicle is mechanically matched. This test shall consist of a simulated mission profile.

The complete Lander will be operated. Performance of each Lander subsystem and component will be monitored and evaluated as the Lander is exercised through a flight sequence profile.

Tests will include operation of the mechanical and deployment devices such as operation of the tip-over bar and anchor to simulate reorienting and stabilizing the Lander if it lands on the aft bulkhead.

Antenna deployment and orientation will be demonstrated and ability of the parabolic antenna to respond to simulated commands will be demonstrated. The electrical performance tests must include a demonstration that the RTG power unit is functional. The plan is to limit the handling of the radioactive isotope to the field. Operation of the RTG (at least the converter section) will be demonstrated by using an electrical heat source in place of the isotope. Using the electrical heat source, output of the RTG will be measured.

6) Weight and Balance, Moments and Products of Inertia Test

After the completion of the Systems Confidence Test the weight, cg, moments and products of inertia of the Lander will be measured.

7) Vibration

The complete Entry/Lander Vehicle will be vibrated at flight levels in all three (3) axes. A function check will be performed before, during, and after the Vibration test.

8) Altitude/Thermal-Vacuum Test

This environmental test is to be performed in order to simulate the Environmental profile through which Entry/Lander will have to perform.

The system shall operate at a vacuum condition, then the environment shall be changed to simulate the Mars atmosphere with a temperature cycling of -184 to 117°F. The Lander shall be operated in this environment for 500 hours trouble free.

9) Final Systems Confidence Test

After the completion of the Environmental test the Entry/Lander Vehicle shall be disassembled and inspected for degradation. A final system confidence test shall be performed which consists of a simulated mission profile, to assure the Entry/Lander Vehicle meets all system requirements and specifications.

10) Final Inspection and Shipping

After the completion of all tests, final inspection will be performed. A working logbook will be mainted on all major subassemblies. The logbook will contain a complete history of the vehicle from the component level to the shipped configuration.

Data from component test, subsystems and systems will be routed to data reduction for incorporation into a calibration book as quickly as possible after the completion of all testing.

(4) Spacecraft Acceptance Test Cycle (See Figure 3.7.8-1)

(a) Pneumatic Subsystem Test

1) Test Objectives

1. Assure that there are no leaks in the subsystem
2. Proof pressure test to establish safety.

2) Description

During the system assembly cycle, and while there is still good access to the pneumatic components, the system will be leak checked starting with gross leak checks and proceeding to tracing minute leaks in accordance with the system specification. Final testing may be performed in a leak tight housing using a helium mass spectrometer.

System will be proof pressure tested in pressure test cell to level in excess to that of normal operation. System will be pressurized to specified proof pressure level and held for specified period of time. Test will be conducted in a pressure safety cell with pressurization controlled from a control console outside the cell.

3) Facilities and Special Test Equipment

a) Systems pneumatic test laboratory with high pressure supply system for clean, dry nitrogen gas and with suitable pressure guages, regulators, and required safety features.

b) High pressure pneumatic test cell where high pressure pneumatic tests can be conducted with operator safety. Remote control is needed so test can be conducted with operator outside the test cell.

c) Leak detectors including halogen leak detectors and helium mass spectrometer.

d) Special test control console used to pressurize the pneumatic subsystem and to control and monitor the pneumatic tests. This will be built as special test equipment peculiar to the program requirements.

e) Leak Test Chamber

Special test equipment in which the spacecraft is housed while helium leak tests are performed.

f) "Clean Room"

The clean area is needed to perform any required disassembly or rework required. Must be large enough to handle complete vehicle system.

(b) Electrical Harness Test

1) Test Objectives

1. Assure that no damage has occurred to the electrical harness during its installation

2. Assure that the harness connections to the subsystems are compatible and that cable lengths and routing are satisfactory.

2) Description

The harness installation will be examined after it is installed in the spacecraft. Examination will be to assure no physical damage has occurred, that routing is correct, and that the various connectors and connections are compatible with the subsystem mating connections. Continuity checks will be performed. Hi-pot and megger tests will indicate any insulation damage.

3) Facilities and Special Test Equipment

Automated harness checker having the capability of automatically cycling through each circuit in the harness checking continuity, insulation resistance and ability to withstand voltage.

(c) Subsystem Tests

Providing the construction of the spacecraft will permit, prior to the final installation of the components and subsystems into the Orbiter, an electrical mating check will be performed. The components will be laid out on the honeycomb structural panels to which they normally mount. They will be cabled together and an operational checkout test will be performed. Purpose of the test is to find out any electrical incompatibility before the final installation is made.

Upon the completion of the above electrical mating tests the components are installed and the subsystem tests are completed. The tests are intended to prove that each subsystem is operable and ready for the complete system test.

1) Electrical Power Subsystem

a) Test Objectives

1. To establish that the subsystem will deliver power of the proper voltage, polarity, and frequency to the other subsystems

2. Demonstrate that the subsystem is capable of carrying power to the other subsystems in accordance with the calculated power profile curve

3. Determine subsystem regulation and loading characteristics

4. Demonstrate that the generator portion of the Lander RTG is operable by using an electric heat source to simulate the radioisotope heat source.

b) Description

The electrical harness is connected to the subsystem and the subsystem energized a "section" at a time so that the performance can be monitored starting with a simple circuit and adding additional circuits and components as checkout continues. Loading and regulation are monitored during this procedure. When the entire system electrical power check has been completed, a power profile loading test corresponding to mission requirements is conducted. Performance of the regulators and inverters and power supplies is determined.

The power source during the test is a simulated power supply to substitute for the photovoltaic supply (and the Lander RTG unit). Similarly, simulated loads may be required in cases where the actual flight load cannot be energized.

c) Facilities and Special Test Equipment

"Electrical Power Subsystem Tests" - An array of special test equipment providing a simulated system (and Lander) power supply, various recorders and meters for measuring the characteristics of the subsystem under varying loadings will be required. Additionally, simulated loads corresponding to the other subsystems are provided. This equipment will be incorporated with other subsystem test equipment into a Systems Test set. RTG source simulator - an electric heater to replace the isotope source and provide the required heat input to the RTG generator will be needed to check out the RTG power supply. (It is assumed that safety considerations will preclude operating the RTG unit prior to the final stages of checkout in the field.)

2) Structure and Temperature Control

NOTE: It is assumed that the structure welds and fabrication were inspected in detail during fabrication and prior to the installation of the components.

a) Test Objectives

1. Demonstrate that those components related to extending and folding mechanisms such as the antenna assembly and the Planet Horizontal Package function

2. Demonstrate that the temperature control mechanism is functional

3. Determine performance of the separation circuits and mechanism (without firing any squibs or explosive disconnects)

4. Determine the alignment of the basic structure.

b) Description

Input signals will be supplied to the control for the Planet Horizontal Package and the antenna mechanism, and the performance of the servo loops to input stimuli will be monitored. Probably during initial tests weights will simulate the paddles and the antenna, rather than use the real flight units.

The spacecraft will be mounted on a surface plate and the basic structural dimensions and alignment will be verified.

c) Facilities and Test Equipment

Special test equipment to monitor and control the operation of the structural subassemblies is required. This equipment is incorporated into the Systems Test set.

Handling fixtures for holding the antenna during weight and center of gravity measurements and during storage are required.

Optical alignment facility - to be described under paragraph 6) alignment.

3) Attitude Control and Guidance

a) Test Objectives

1. Demonstrate acquisition of simulated sun, star and earth by the acquisition loops

2. Measure performance of the tracking loops to input stimuli

3. Measure performance of the control loops for stabilization and antenna tracking of earth

4. Measure ability of the PHP to track stimuli corresponding to planet orbit

5. Demonstrate performance of autopilot systems to control input signals to propulsion/thrust control mechanism.

b) Description

Tests will be performed on the spacecraft system using the System Test set. Input stimuli will be programmed into the Attitude Control and Guidance Subsystem and the performance will be monitored. Operation of the jet nozzles and servo motors will be monitored. As required, simulators will be used to provide interfacing component or subsystem functions.

c) Facilities and Special Test Equipment

Systems Test set containing the required power supplies, signal generators, recorders, meters and necessary stimuli and monitoring devices will control the test. Programmable sun, star, Earth and Planet Mars simulators will be provided to test sensors and control loops. Test program will be automated as far as possible. System test fixture will be used to support the spacecraft during the ambient tests.

4) Propulsion

a) Test Objective

Demonstrate performance of the propellant flow control circuits of the propulsion system.

b) Description

Input stimuli will be programmed to the propulsion subsystem; and the response of the valves and controls will be measured. The fuel and oxidizer subsystem will not be charged even with simulated fuel in order to reduce the cleanliness maintenance problem. Only that portion of the system which can be energized without possibly contaminating the tanks and piping will be energized.

c) Facilities and Test Equipment

Systems Test Set

Pressurization control equipment to pressurize the fuel tanks.

5) Spacecraft Communications System

TABLE 3.7.8-1. TYPICAL ACCEPTANCE TESTS

	Name of Test	Objective	Description	Test Equipment Required
1.	Receiver, S-band (for each receiver)	To determine sensitivity of receiver and acquisition time.	Signals will be applied to the antenna input and output command demodulator will be monitored.	1) S-Band Transmitter 2) Command Simulator 3) Error Detection Unit 4) Attenuator 5) Clock
2.	Transmitter and Power Amp.	To determine frequency and power output of transmitter	Digital modulation will be simulated at the input to the transmitter and spectrum viewed and power measured.	1) Power Supply 2) High Power Load 3) Frequency Meter
3.	Command Subsystem	To determine operation of this subsystem	Command will modulate a simulated ground transmitter and output will be fed into antenna input. Commands will have various timetables and operation at correct time will be verified.	1) S-Band Transmitter 2) Command Simulator 3) Clock 4) Digital Error Unit 5) Monitor for PC&C unit
4.	Power Conversion and Control	To verify operation of PC&C unit.	Bus voltages will be varied over wide range and regulation can be measured.	1) Power supply 2) Voltmeter etc. 3) Loads
5.	Storage	Verify Wow and Flutter and Noise	Digital Data will be applied at input and played back at all possible speeds with voltage variations	1) Digital signal simulator 2) Error detection unit 3) Power Supply

TABLE 3.7.8-1. TYPICAL ACCEPTANCE TESTS (Cont'd)

	Name of Test	Objective	Description	Test Equipment Required
6.	Data Handling Subsystem	Verify Subsystem Operation	Analog and Digital data will be fed to digital handling processor. Output will be viewed in real time and after storage.	1) Data Simulator 2) Output Display 3) PC&C Simulator
7.	Sensor Calibration	Provide conversion factor for reduction of flight data	Apply Physical stimulus at each sensor. Check output of TLM.	1) Various stimuli 2) Ground receiver
8.	Simulated Flight	Demonstrate performance of complete subsystem.	Provide stimuli test patterns, changing bus voltages, communicate through antennas. Send commands; monitor outputs of PC&C. Operate Lander-Orbiter Link, Orbiter-Earth, and Lander-Earth links.	1) Ground antenna 2) Ground station

NOTE: The above test equipment will be part of the system checkout and test set and the systems ground station, both of which will be required as special test equipment.

6) Alignment

a) Test Objectives

1. Align sun sensor and star tracker to spacecraft reference directions

2. Align Orbiter engine center line to go through calculated center of gravity of the Orbiter

3. Align attitude control nozzles to vehicle references

4. Align earth center with respect to high gain antenna bore sight.

b) Description

The spacecraft will be mounted (using a suitable fixture) on a large surface plate. Optical tooling bars, theodolites and levels will be used to establish the vehicle reference planes and axes and to align the attitude control and guidance components and the Orbiter.

Alignment is performed before assembly of antenna for easy access to the spacecraft and to lessen possibility of damage to the antenna. Then the antenna is assembled and aligned.

c) Facilities and Special Test Equipment

Optical alignment laboratory with large surface plate, optical tooling bars, autocollimators, theodolites. Work area must be temperature and humidity controlled.

7) Weight and Center of Gravity

a) Test Objectives

1. Determine weight of the Orbiter.

2. Determine center of gravity of the Orbiter Vehicle.

b) Description

1. The Orbiter will be weighed and compared to the calculated weight and to the specification requirement.

2. Center of gravity of the spacecraft will be determined possibly by the use of a balance machine, or more likely by the three scale weighing method and calculation. Measurements along 2 planes will be required.

c) Facilities and Test Equipment

1. Platform scales having accuracy meeting the vehicle specification will be required. Three scales are needed.

2. Vehicle test fixtures will be required to support the vehicle during test. This will be furnished as special test equipment.

8) Launch Vehicle and Shroud Mating

a) Test Objective

Assure compatible interface between spacecraft and launch vehicle and between spacecraft and shroud.

b) Description

Use launch vehicle interface mockup and shroud and perform mating compatibility tests. Check dimensional compatibility and location and compatibility of mating connectors.

c) Facilities

1. Mockup of launch vehicle interface

2. Mockup of shroud.

9) GSE Handling Equipment and System Checkout Equipment Compatibility Test

a) Test Objectives

1. Assure that handling equipment to be used to support systems tests is compatible with spacecraft configuration

2. Assure that the systems checkout test set is operational and ready for spacecraft test

3. Assure the compatibility of the systems checkout test set with the spacecraft.

b) Description

Tests are performed early in the test cycle to assure that the handling equipment, especially the equipment to be used in the field, is compatible with the spacecraft.

The systems checkout test set is set up and checked out to assure that it is ready to support the tests. Checkout is made using a spacecraft simulator so that the spacecraft is not tied up unnecessarily. Compatibility checkout of the spacecraft and the test set is then performed. This test has the benefit of the tests performed previously using the development spacecraft.

c) Facilities and Test Equipment

System checkout and test set GSE handling equipment.

Spacecraft simulator - special test equipment built to simulate the GSE - spacecraft interfaces and the circuits in the spacecraft.

10) System Alignment, Weight, and Center of Gravity

a) Test Objectives

1. Check system alignment of complete spacecraft

2. Determine weight and center-of-gravity of the entire spacecraft and assure that main engine thrust is aligned with cg of spacecraft.

b) Description

1. The assembled spacecraft will be set up on a surface plate in the optical alignment area. Alignment of antenna, Orbiter rocket engine, Lander to Orbiter, and Orbiter sensors will be checked.

2. Following the alignment checks, the assembled spacecraft will be weighed and center-of-gravity checked using the three scale method.

   NOTE: For center-of-gravity in the second plane the measured value for the Landers and for the Orbiter separately will be algebraically combined to get the system cg. This may be necessary in preference to turning the spacecraft on its side with Earth gravity forces acting on Landers and Orbiter.

c) Facilities and Test Equipment

1. Optical alignment laboratory

2. Platform scale and fixtures

11) System Performance Tests

a) Test Objectives

1. Determine response of control loops to input stimuli and determine overall system performance

2. Perform simulated mission profile

3. Transmit data and commands to and from the system ground station

4. Establish base performance data against which performance after vibration and thermal-vacuum tests can be evaluated.

b) Description

The assembled spacecraft will be mounted on the system checkout fixture and programmed commands and simulated inputs will be fed into the control loops and output response will be monitored and evaluated. Commands will be put into the system from the ground station and response will be monitored. Telemeter data will be transmitted to the system ground station. The intent is to check performance of every component and every subsystem.

c) Facilities and Test Equipment

1. Power supply simulator to replace solar power supply and RTG's

2. System checkout and test set including programmed stimuli

3. System ground station

4. System test fixture

5. Ground cooling unit.

12) RF Radiation and EMI Tests

a) Test Objectives

1. Determine that system operation does not cause spurious operation or malfunction of the electronic components

2. Determine that electromagnetic radiation from system operation does not interfere with telemetry, communications or the experiments

3. Measure the level of RF radiation and electromagnetic interference generated during system operation through the mission profile.

b) Description

Set up and operate the complete spacecraft in a large shielded area. The spacecraft is operated through all modes of normal flight using simulated signals, if necessary, to complete the simulated mission profile. Noise level is monitored during the tests.

c) Facilities and Test Equipment Required

1. Large screen room with attenuation meeting the Interference Specification for the program

2. Signal generators and receivers to cover the frequency range of the interference specification

3. System checkout and test set to energize and operate spacecraft. System test fixture to support the spacecraft during test.

13) Vibration Test

a) Test Objective

Demonstrate that the spacecraft can withstand the vibration levels specified (and almost equal to expected flight levels of vibration) with no functional failure or structural failure or change of performance.

b) Description

The spacecraft is mounted to a rigid fixture which will transmit vibration table acceleration levels without excessive magnification or attenuation. Mounting is similar to that used to fasten the spacecraft to the launch vehicle adapter. The vibration program of sine and random vibration is programmed to correspond to the test specification. Those subsystems and components of the spacecraft which are normally energized and operated during powered flight are energized and operated and performance monitored during the vibration test.

Vibration test will be conducted along three axes to the requirements of the specification, using an oil film supported table, or equivalent, as required. Accelerometers at the spacecraft/fixture mounting monitor input levels.

Performance of the system during vibration test is evaluated by data transmitted by telemetry. At the completion of the vibration test the entire spacecraft system is performance tested and inspected to determine change of performance caused by the vibration test.

c) Facilities and Special Test Equipment

1. Shaker System with sine and random test capability over a spectrum of 5 to 2000 cps. Force rating sufficient to drive the vehicle (and fixture) to acceleration levels specified in the test specification. It is expected that at least dual MB-C-210, or equivalent is required for 7500 pound vehicle plus fixture

2. Control system equipped with signal generator, filters, shaping networks as required for the random program, and with automatic equalizer-analyzer to control the test

3. Tape recorder-programmer required to control vibration test for mission vibration simulators

4. Tape recorders and high speed oscillographs required to record input vibration levels, and measure response

5. Accelerometers, power supplies, and amplifier required to instrument the vibration levels

6. Systems checkout and test set and ground station required to energize, program, and checkout the spacecraft

7. Systems vibration fixture needed to support vehicle during the test

8. Auxiliary table, oil film supported, or equivalent mounting is required to support the spacecraft to accomplish test in three axes

9. Hydraulic vibrator is required if low frequency, large displacement vibration levels are in the specification.

14) Thermal-Vacuum Test

a) Test Objectives

1. Measure performance of the complete spacecraft under environmental conditions simulating as far as economically feasible those encountered during flight to the planet

2. Measure performance of the subsystems and components in the spacecraft under a relatively long time environmental test

3. Verify the operation of the thermal control

4. Collect data to establish the achieved reliability of operation in a simulated space environment.

b) Description

The complete spacecraft is installed on a test fixture in the thermal-vacuum chamber. The spacecraft is instrumented to monitor temperature at a number of critical locations. Test cables are provided from the spacecraft mating connectors to the chamber penetrations to the systems checkout and test set. Additionally, telemetry data is transmitted from the spacecraft (using a small substitute antenna as necessary) to an antenna on the chamber, through a suitable chamber penetration and to the ground station.

Solar simulation intended to match the Johnson curve is programmed to provide thermal input to the vehicle corresponding to the flight conditions. The spacecraft is mounted to be positioned with the proper orientation to the sun. Solar simulation is programmed to correspond to flight from the Earth atmosphere to Mars. Consideration will be given to the need for albedo simulation.

The vacuum chamber has nitrogen and helium cryogenic cooled black walls. Operating at 100°K. Vacuum requirement is 10^{-6} torr or better.

The spacecraft is set up in the chamber and tested out to insure validity of test set-up. Then the spacecraft is performance checked. The vacuum chamber is pumped down and the testing is begun. Performance is monitored at suitable intervals as the spacecraft is programmed to correspond to flight functions. Periodically test stimuli are introduced into the system and response monitored and performance evaluated. Test duration is planned to be 1000 hours failure free. At the completion of the vacuum test, chamber is pumped up and performance test is re-run and data evaluated.

c) Facilities and Test Equipment

1. Space simulator with capability of test to at least 10^{-6} torr. Liquid nitrogen, liquid helium cooled black walls with emissivity of 0.9 and temperature of $100^{o}K$. Solar illumination of at least 1.4 KW/m^2 for near Earth environment to 0.62 KW/m^2 for Mars with spectrum corresponding to Johnson curve and with 5% collimation and uniformity. Test diameter needed - about 20 feet

2. Vacuum-chamber vacuum and temperature instrumentation

3. Systems checkout and test set

4. System ground station

5. System test fixture.

15) Functional Performance Test

a) Test Objective

Detailed performance test to determine any change of performance as a result of the extended thermal-vaccum test.

b) Description - Same as initial systems performance tests.

c) Facilities and Test Equipment - Same as required for systems performance tests.

16) Alignment Recheck

a) Test Objective

Determine whether alignment has been distrubed as a result of testing and handling cycle.

b) Description - Same as the initial alignment check.

c) Facilities and Test Equipment - Same as optical alignment facility previously described.

17) Preparation for Shipment

a) Test Objectives

1. Review test data to determine completeness, adequacy, and evaluate performance

2. Perform any necessary inspections

3. Complete the log books

4. Secure customer buy-off and agreement to ship, in accordance with program ground rules.

b) Description

A final inspection is made to be certain no damage has occurred in testing and handling.

The travel records, inspection records, etc. are verified. Data is checked and the log book is completed. Customer buy-off is obtained.

3.7.9 FIELD TEST AND LOGISTICS PLAN

A. Field Test Philosophy

(1) Test - Launch Cycle

The test plan described assumes that the Apollo Saturn V type Integrated Transportation Launch complexes, Vertical Assembly Building and Transporter, and related equipment are not available to support the early Voyager launches. Also the use of Saturn 1B and the Stage VI would require some modification to the handling and checkout equipment. Tests described are not based on the use of this type Saturn V equipment, and the cycle includes tests and checks conventionally being made on present space launch programs.

(2) Test Requirements

The Reliability Program requires extensive operation of the system with no failures during the latter part of the test cycle. Thus, by the time the spacecraft is sent to the field it will have experienced considerable testing and its performance will have been evaluated in detail. Therefore, a minimum of field test failure and equipment rework is expected.

The intent of the Field Test Program is to make final preparations of the spacecraft for flight. Assurance must be given that the spacecraft is functioning adequately and that it is compatible with the launch vehicle. The launch support equipment must be checked out with the vehicle and preparations made for launch. The data handling and tracking and command network and Spacecraft Control Center must be checked out to assure it is ready to support a flight.

(3) Failures and Rework

In order to meet the Reliability Program requirements of operating the spacecraft system for an extended period of time with no failures, the spacecraft will be thoroughly checked out and tested before it is sent to the launch site. No extensive changes, rework, or modification may be made to the spacecraft after the completion of the systems tests. Field rework could result in unknown changes in performance and reliability.

Any unit substituted to replace a failed unit in the spacecraft must have gone through a test phase which would at least equal the testing that a component encounters with the in-house test program.

Component replacement in the field will be made only after thorough failure analysis. Intent is to perform any rework on units back at the factory rather than in the field.

(4) Spares

Consistent with the above concept of field rework, a "set" of flight spacecraft spare components will be available at the launch site and under the control of the Logistics group. The Logistics group will maintain the components in a controlled bonded stock area where the units are properly controlled and accounted. Spares will be released only after proper authorization following failure investigation and analysis.

Any spare component used in the system will have been subjected to a test program at least equivalent to the one which the original component was subjected to in the in-house systems acceptance test.

(5) Test Equipment

The test equipment to be used in the field will be similar to that used in-house. This requirement is made so that data collected in the field can be compared to test data gathered in-house with a reasonable degree of confidence. Additionally, there will be greater assurance of compatibility of test equipment to spacecraft. Also the test crews will have become experienced and expert in the use of the test equipment during the acceptance tests. For description of test equipment see Section 3.13 - Ground Support Plan.

(6) Flight Test Working Groups

Plans for the field test activities will be coordinated and worked out between the program contractors through the Flight Test Working Groups. The groups will be made up of representatives of the spacecraft contractor, the launch vehicle contractors, and the operating contractor for the DSIF and NASA. The Working Groups will establish the interface and working relationships between the program contractors. The vehicle handling and test procedures will be worked out by the Flight Test Working Groups.

B. Field Test Cycle - See Figure 3.7.9-1

(1) Receiving Inspection

The Voyager spacecraft will be shipped to the field in suitable shipping containers which will provide protection against transportation vibration and handling shock, and from extreme temperatures, humidity, and contamination. It is assumed that the Landers, the high gain antenna, and the Orbiter and the PHP will be packaged separately. Upon receipt in the field the spacecraft assemblies will be removed from the shipping containers and inspected for any sign of damage or any deficiency. Inspection of the test records and documentation will be made.

(2) Subsystem Checkout and Calibration

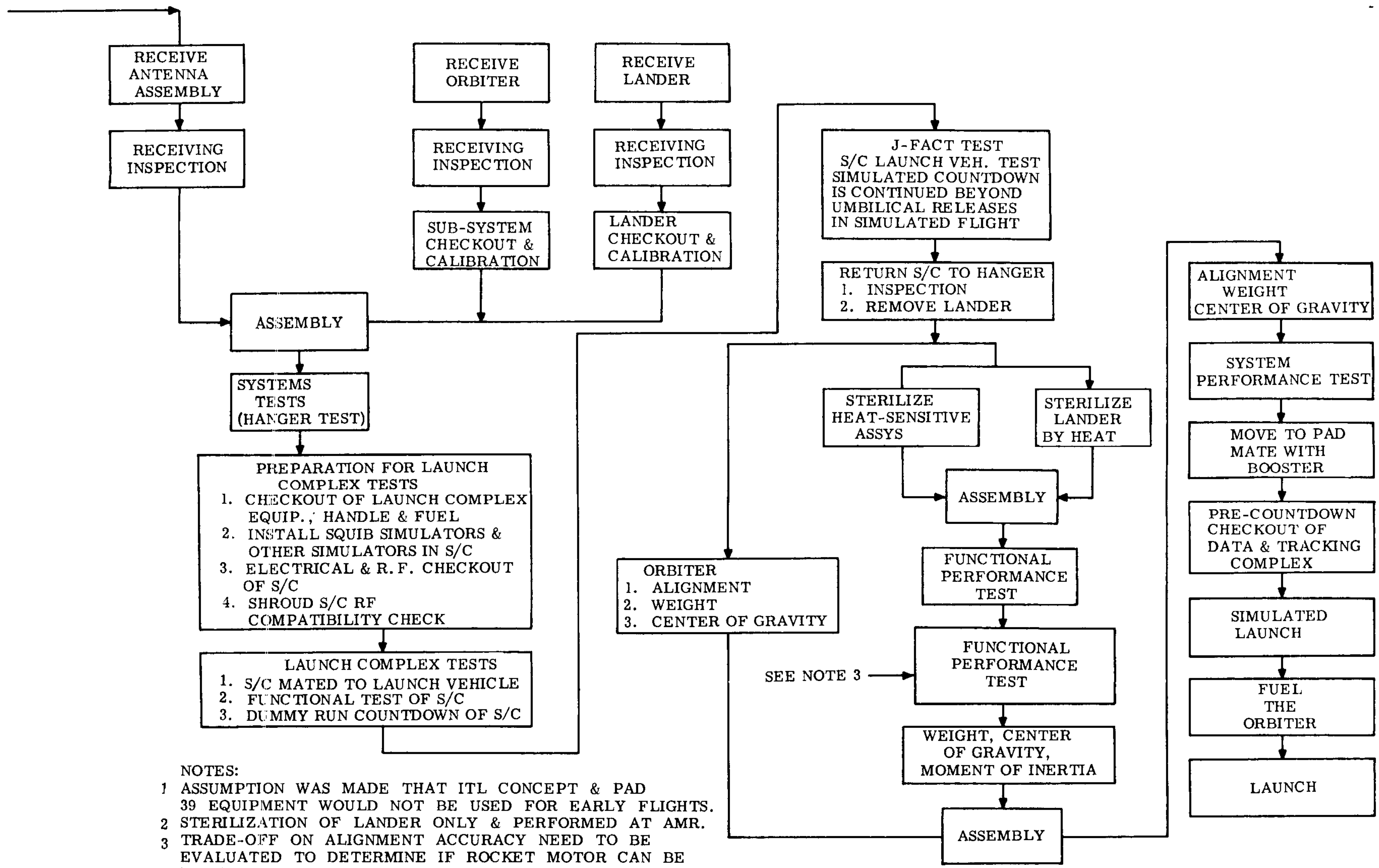
Initial checks will be performed on the pneumatics subsystem to insure that is is intact. The subsystem will be proof pressure tested. Although this is a repeat test it will be performed as an additional safeguard against a major failure. System leak checks are then performed to insure a tight system.

The subsystem checkout and calibration of the Orbiter and the Lander will be virtually a repetition of the tests run as part of the in-house acceptance test. Objective will be to insure that the subsystem is still functional and has not been degraded in performance during the transportation and handling cycle. Calibration of sensors and transducers will be performed to ascertain that the experiments and various diagnostic and measuring circuits are functional within the required degree of accuracy. Calibration consists of comparing at least three data points against the measurement calibration curve. If there if any out-of-tolerance deviation previously calibrated in the factor the complete calibration from that measurement must be replotted.

The Lander and antenna assembly are then assembled to the Orbiter, and systems tests begin in the hanger.

(3) Systems Performance Tests (Hangar)

The complete spacecraft is subjected to a performance test. Response of the subsystems to various programmed input stimuli is monitored and the spacecraft programmed through a simulated flight profile. This test is essentially a repetition of the test performed at the factory. Performance data from the test is evaluated against the data gathered in the factory tests and any degradation of performance is evaluated.



NOTES:
1 ASSUMPTION WAS MADE THAT ITL CONCEPT & PAD 39 EQUIPMENT WOULD NOT BE USED FOR EARLY FLIGHTS.
2 STERILIZATION OF LANDER ONLY & PERFORMED AT AMR.
3 TRADE-OFF ON ALIGNMENT ACCURACY NEED TO BE EVALUATED TO DETERMINE IF ROCKET MOTOR CAN BE INSTALLED AFTER C OF G & SYSTEM PERF. TEST
4 ASSUME RTG IS PERSONNEL RADIATION PROBLEM. RTG IS NOT REMOTE CONTROLLED. METHOD OF INSTALLING NOT YET ESTABLISHED

Figure 3.7.9-1. Field Test Flow Diagram

Spacecraft configuration for this test is the Orbiter with antenna assembly and adapter and the two Landers.

(4) Preparation for Launch Complex Tests

The launch complex tests are performed to insure that the spacecraft and the launch complex equipment are compatible and ready for flight. Testing is done as a separate test rather than delay the check till the Joint-Flight Acceptance Composite Test (J-FACT). Performing the separate test reduces the amount of time the launch vehicle and pad is required. Prior to the pad tests as many of the preparations as possible are made at the hangar because it is far easier to perform the work in the hangar than at the pad.

This "pre J-FACT" test also provides an opportunity to acquaint the field personnel with field test procedures and handling problems. (NOTE: It is assumed that the key test personnel also assisted in the factory acceptance tests.)

As preparation for the launch complex test the launch complex test equipment is checked out and certified as operational. A vehicle simulator would be very desirable as a tool in checking out the launch complex and insuring compatibility prior to connecting the spacecraft. The handling and fueling equipment can also be checked at this point using a spacecraft simulator. Squib simulators are installed and electrical checkout of the spacecraft is performed before the spacecraft leaves the hangar.

(5) Launch Complex Tests

The launch complex equipment is checked out with spacecraft simulator and launch vehicle simulator. The spacecraft is mated to the launch vehicle (at the pad). Functional test of the spacecraft are conducted. Blockhouse data links and communication links are checked.

The dummy run countdown of the spacecraft is conducted. This simulates the preflight countdown operations and the powered flight operations. The test objectives are:

1. Evaluate spacecraft operation in the launch complex environment
2. Functional compatibility of the spacecraft, the adapter and shroud is checked
3. Operation of launch complex equipment blockhouse equipment is verified
4. Countdown procedures are verified
5. Prepare the spacecraft for the J-FACT test compatibility test with the launch vehicle.

(6) J-FACT Test

The Joint-Flight Acceptance Composite Test is the basic test which establishes spacecraft/launch vehicle readiness for flight by demonstrating that the entire space vehicle and launch complex and tracking network operate correctly through countdown and simulated flight. The following tests are included:

1. Electrical compatibility of the overall space vehicle and operation through the flight profile
2. Vehicle power transfer from ground power to internal battery power
3. Complete functional and R-F radiation checkout in the launch configuration and with the gantry pulled away and the command and communication systems monitored by the local AMR and range tracking stations.

(7) Return Spacecraft to Hangar

The spacecraft is removed from the launch vehicle and returned to the Hangar. The Landers are removed and a detailed inspection made to determine status of the spacecraft and to determine whether any damage has occurred. The simulated RTG unit used to provide electrical power is removed. Preparations are made for the Lander sterilization.

If there are any components aboard the Lander which are not capable of withstanding the sterilization heat cycle they are removed from the Lander and sterilized separately by gas or other means as described in the Sterilization Plan.

Sterilization of the Lander is accomplished and the non heat sterilizable components (if any) are reinstalled under sterile conditions.

A functional performance test of the Lander is made to assure that no degradation of performance has resulted because of the handling and sterilization.

(8) Weight, Center of Gravity, Moment of Inertia

A quick recheck of weight, center of gravity, and moments of inertia of the Lander is made to assure that, if any disassembly and assembly was made, the Lander is still correctly aligned and balanced. The weight and center of gravity of the Orbiter are also checked, then the Landers and Orbiter are assembled and the checks are repeated for the spacecraft. The liquid fuel for the Orbiter is either simulated with a safe substitute liquid, or correction is calculated.

(9) System Performance Check

A system performance check will be made. The Lander rockets and all pyrotechnics are installed and the spacecraft is moved to the pad and mated with the launch vehicle.

(10) Precountdown Checkout

Countdown operation of the spacecraft and the launch vehicle is conducted. Checkout of the data and tracking net is made. During this operation all the world wide data links between the launch site and the DSIF tracking locations and the Spacecraft Control Center must be checked out and verified. The data handling links must be verified.

(11) Simulated Launch

The spacecraft and launch vehicle are programmed through a simulated launch program.

(12) Fuel the Orbiter, Install RTG Isotope

Because of the dangers in handling the liquid fuel it is suggested that fueling the Orbiter be delayed until this point. The tanks are fueled using remote control equipment.

The isotope fuel element is installed using remote handling equipment.

(13) Launch

3.7.10 ANALYSIS OF THE APPLICATION AND USE OF A THREE AXIS MOTION SIMULATOR FOR DEVELOPMENT AND ACCEPTANCE TESTING

A. Introduction

A study was conducted to evaluate the best test program for the control subsystem of the Voyager Spacecraft. The study evaluated the following tradeoff considerations:

1. Static versus Dynamic Tests
2. Subsystem versus Spacecraft Tests
3. Development versus Acceptance Tests
4. Ambient versus Environmental Tests

B. Results of Study

Based on the study which is detailed below, it is recommended that the control system on Voyager be tested on a subsystem basis; first on a set of development hardware and then on one set of the flight hardware, prior to installation in the spacecraft. These conclusions are reached on the basis of the test requirements for the spacecraft, the structural configuration of the spacecraft and the experience on other programs which the General Electric Company Spacecraft Department has worked on and is working on.

C. Trade-Off Analysis

(1) Test Plan

The analysis was based on the need for conducting control system tests which verify the capability of the subsystem to perform initial acquisition and reacquisitions, pointing accuracy tests, limit cycle operation and occulting tests.

(2) Basic Requirements

In order to adequately simulate the performance of a spacecraft control system, it is theoretically necessary to simulate the environment and orbital conditions to which the spacecraft is subjected. This task becomes extremely difficult, primarily because of the 1g gravity field under which these tests must be conducted. Because of this requirement, it is necessary that a continual coincidence of the center of gravity with the center of rotation be maintained. In order to meet this condition, a rigid structure is required, all component mass shifts must be avoided, compensation for gas depletion must be provided, and imbalance effects due to thermal gradients must be avoided.

(3) Dynamic Versus Static Tests

It is theoretically possible to test a control system either under static, that is, non-moving conditions or dynamic conditions which require vehicle motion. This essentially amounts to testing the control system in open-loop versus closed-loop fashion. Though static tests can be satisfactorily conducted when once the control system interactions are established, it must be proven at some point in the development program that under dynamic conditions, interactions between the various control loops will not take place. This, therefore, leads one to the conclusion that a dynamic test must be performed.

(4) Spacecraft Versus Subsystem Test

The need for dynamically testing the flight spacecraft creates an extremely difficult task of test simulation, primarily because the spacecraft is designed for optimum orbit performance and not for test simulation. In general, these two requirements conflict directly. The characteristics of the Voyager structure which, in particular, make spacecraft dynamic testing undesirable are as follows:

1. The vehicle consists of a thin web structure which, due to its size, will be quite flexible.

2. Uniball joints are used extensively in cross bracing. These joints contribute to mass shifts as proven by tests on the Nimbus spacecraft.

3. Extensive use of honeycomb panels with components mounted to them will lead to mass shifts.

4. The panels covered with solar cells with flex and deform under the influence of gravity.

5. The omnidirectional antennas on the spacecraft are flexible.

6. Mass shifts which arise due to antenna motion on its gimbal system and motion of the Planet Horizon Package on its gimbal system would be extremely difficult to compensate for. In addition, the torquers which drive these gimbal axes are not designed to move the full weight under the influence of the 1g gravity field

7. The center of gravity of the spacecraft is not accessible. In order to have access, the liquid rocket nozzle and mount would have to be removed and the fuel and oxidizer tanks would likewise have to be removed. In addition, as indicated in item 6, the antenna and PHP would have to be removed. Once these objects have been removed, a suitable means of attachment to the spacecraft would have to be provided so that all test equipment can be mounted thereon. As a result of all these factors, it is decided that tests should be run, on a simulated platform on a subsystem basis.

(5) Development Versus Acceptance Tests

A variety of different philosophies can and have been used in deciding which sets of hardware are to be tested on the motion simulator. It is certainly required that extensive tests be conducted on a subsystem basis for development purposes. Experience has shown, in particular, on the OAO Program in the General Electric Spacecraft Department, that because of changes in hardware, it is extremely important to conduct tests on a representative flight system. Therefore, both development and acceptance tests should be performed.

(6) Ambient Versus Environment Tests

Extensive experience in the General Electric Spacecraft Department has shown that adequate subsystem tests can be conducted at ambient conditions. The extreme difficulty of performing these tests under thermal-vacuum environment leads to an extensive design and development program of long time duration and high cost. It is best to evaluate the environmental performance of a subsystem under static conditions than to require the complexities of remote handling equipment which are needed for thermal-vacuum motion simulator tests. Because tests are conducted on a subsystem basis and are partly of development nature, it is strongly preferred that these tests be conducted in a room ambient environment.

D. Motion Simulator Facility

(1) General Arrangement

Figure 3.7.10-1 shows the general arrangement of the motion simulator test facility. It is planned to simulate the inertia of the spacecraft and mount the simulated inertia on a gas bearing approximately 15-inch diameter. All required sensors for the orbital control will be mounted on this platform. In addition, along the pitch axis of the platform an antenna gimbal and earth sensor will be mounted, and at the other end the PHP gimbal and planet sensor will be mounted. Placed in the room are the simulators representing Sun, Canopus, Mars and Earth. Instrumentation and measurement equipment is also located in the room as shown. The test cell will be sealed, to minimize drafts and draped with suitable material to prevent light reflection and to further prevent air currents in the room.

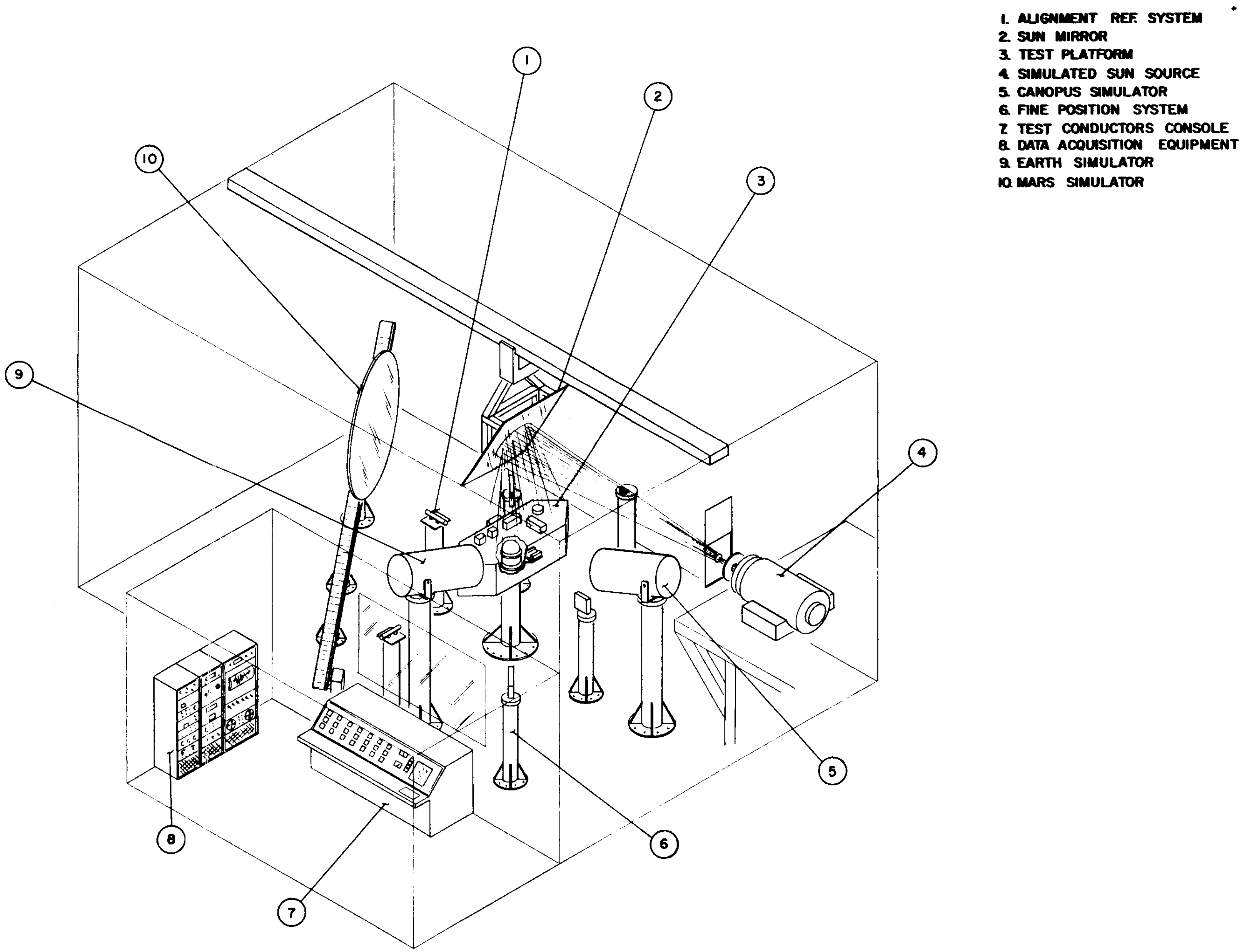


Figure 3.7.10-1. 3-Axis Motion Simulator

(2) Platform and Support

The bearing will be mounted on a straight non-magnetic stand. The structure and bearing will permit 360° of freedom motion about the vertical axis which is roll and ±60° of freedom about the two horizontal axes which are pitch and yaw. The platform will be designed so that its inertia can be readily varied to simulate the varying inertias of the spacecraft during its lifetime. Nozzles of different size and moment arms will also be used to achieve the simulation of varying inertias. A pair of balanced gas tanks will be mounted on the platform that will have sufficient capacity to conduct meaningful acquisition and holding tests. The sensors will be mounted so that they will have an unobstructed view of the simulators and yet possess the correct relationship with respect to the center of rotation. The gimbals which represent the antenna and PHP systems will be balanced about their axis of rotation so that no imbalance effects are introduced. Also mounted on the support stand are a set of actuators that will permit initial position and rate conditions to be applied to the control system. Means are also provided for measuring system imbalance.

(3) Simulators

Directly overhead along the roll axis will be a carbon arc which simulates the sun. Offset from this by a 40° angle will be an "Earth" simulator and 15° above the pitch axis will be the Canopus simulator. These two devices will be stellar simulators similar to the designs used on the OAO motion simulator. The three simulators mentioned so far will be fixed to room reference. Position of these simulators will be monitored by means of an optical alignment reference system. Simulation of Mars will be achieved by means of a fixed-size heated disk target moving along a track. This will permit dynamic vehicle inputs and achieve a variation in target size required by the eliptical Mars orbit. The heated platen will be surrounded by a cooled disk so that an adequate planet temperature interface is achieved.

(4) Balance System

In order to adequately simulate a two hour limit cycle, imbalance torques of less than 0.1 in. oz must be achieved for small angles. It is proposed that this be accomplished by means of the following:

1. A large gas bearing of high degree of sphericity; i.e., better than 50 micro inches.
2. Coarse manual balance weights and fine remote balance weights
3. Program balance compensation which will correct for imbalance as a function of tilt angle
4. Thermal and anisoelastic compensators to correct for structure deflections
5. Gas depletion compensation device
6. All harnessing will be conformal coated to prevent shifting and components will be mounted and packaged so that mass shifts are prevented. Batteries of the "starved" nickel-cadmium type will be used
7. Operating in conjunction with the equipment will be a command system which will be capable of driving the remote balance weights as a function of the measured residual imbalance.

(5) Readout and Instrumentation

In order to provide for data transmission from the subsystem, a PCM telemetry system will be provided. Digital to analogue converters will be provided and analogue recorders will provide readout of system performance. Instrumentation will be mounted on board the platform to permit position and rate readouts of platform motion about the pitch and yaw axes. The sensors used for this purpose will be connected to the telemetry system. Position about the roll axis will be obtained by means of an external electro-optical position tracker which will be servoed to follow platform motion. The technique of acquiring data could also be applied to the pitch and yaw axes by mounting the additional two trackers on the roll carriage. The equipment described so far will provide position data to 0.1°. In addition, for measurement of the fine pointing accuracy, a set of three fixed autocollimators will be provided that will monitor platform position. Position of the gimbal systems of antenna and PHP will be obtained by means of shaft position indicators attached to the gimbal drives. This information will be telemetered. All information and control of the entire test will be by means of a central console located in a console room, separate from the test facility. This console will provide for control and monitoring of the simulators, control of the command and balance system, and quick readout of position and rate information. The test conductor will have supervisory control of the entire test from this station.

3.8 RELIABILITY AND QUALITY ASSURANCE PLANS

The close and effective integration of Reliability and Quality Assurance activities in the Engineering, Manufacturing and Quality Control and Test areas of activity is vital to the ultimate success of the Voyager Program. Such an integrated approach is intended as an inherent characteristic of all elements of the overall Voyager Program Plan.

For the purpose of clarity and in keeping with the NASA Reliability and Quality Assurance documents (NPC-250-1 and NPC-202, 203 respectively) the Reliability Program Plan Summary and the Quality Control Plan have been separately documented. They have been jointly prepared and are presented in the following sections.

At each echelon of contracting and subcontracting these reliability and quality assurance documents are to be integrated and established as contractual requirements.

Figure 3.8-1 presents the Reliability Plan Summary information in chart form.

3.8.1 RELIABILITY PLAN SUMMARY

A. Introduction

(1) Scope

The Reliability Program Plan has been prepared in accordance with the guidelines of NASA Reliability Publication NPC-250-1, "Reliability Program Provisions for Space Systems Contractors", July 1963. It defines the management control systems and the major reliability engineering, test and evaluation elements necessary to assure a successful Voyager contractor reliability program.

(2) Background

A six-month study effort of the Voyager Program has been completed. Part of this effort has led to the determination of several problem areas, which may affect the probability of performing a successful Voyager mission. The nature of these problems and the significance of the Voyager Program impose a strict requirement for a concentrated effective reliability management system to assure an organized approach to their solution. These problems are summarized in terms of the various phases of the Voyager mission.

MISSION PHASES	PROBLEM AREAS
Design and Pre-launch	1. Sterilization effect on components. 2. Weight, volume, power, and thermal limitations. 3. Reliability demonstration before flight using small samples. 4. Testing required in simulated environment to obtain confidence in design.
Boost	1. Shock, vibration, acceleration, and thermal definition for new booster.
In-transit	1. High accuracy performance requirements. a. Guidance and navigation b. Attitude control c. Communications 2. Long-term space environmental conditions -temperature, vacuum, radiation, radio-isotopic power supply radiation. 3. Nine-month storage/cyclic operation and monitoring. Effects on:

RELIABILITY PLAN SUMMARY

PROBLEM AREA	TASK AREA	PROBLEMS
SYSTEMS RELIABILITY ANALYSIS & SYNTHESIS	Long Life Effects Data	Establishment of Mission reliability for 11,000 hour periods
	Systems confidence levels	Establishment of confidence in mission reliability de
	Environmental profile definition	Accurate & effective simulation of space, radiation, environments
	Operational cycling effects	Transient and differential peak stresses, "on-off" cy
SYSTEMS - RELIABILITY TRADE-OFFS	System definition	Establishment of: applicable details, optimal redunc success
	Failure mode & effects	Alternatives, interactions, transfer functions
STERILIZATION EFFECTS ON RELIABILITY	Heat Effects	Polymerization, chemical & physical changes
	Sterilant effects	Materials, processes, chemical & physical changes
	Tests of components	Effects of heat & sterilants on performance & reliabi
	System tests	Effects of sterilization on lander system
SUBSYSTEMS AND COMPONENT DESIGN & DEVELOPMENT	Subcontracting & purchasing	Contractual reliability requirements
	Design standards & approvals	Effective development and implementation
	Reliability problem recognition	Concentration on single prototype performance, cost ming
	Reliability monitoring	Management matrix, design reviews, management su
PARTS, MATERIALS, PROCESSES AND APPLICATION DATA	Component Development & Design	Early identification of needs, types, kinds, usage, a
	Process control	Adequate development, evaluation & control for all p contractors
	Parts and materials	Selection, specification, source selection
	Application Data	Stress, environment, derating, application effects
CONTROL	Eng'g, mfg, QC&T, field op'ns	Effective identification & implementation

	APPROACH
pperating & standby	Research existing, applicable data. Establish supplementary test programs.
nonstration	Math model development & evaluation of significance of test results at highest applicable echelon
lanetary & design	Establish limits, surveys, test requirements, and programs facilities and instrumentation requirements.
cle effects & risks	Controlled rates & limits, improved material combinations compatibility, new materials & processes
ancy, defined mission	Detailed estimations of preliminary designs, maximize attainable mission value per pound.
	Detailed designs, synthesis and design reviews
	Evaluate short & long time effects on characteristics for all approved parts, materials, processes.
	Investigate, evaluate all practicable alternatives -- establish reliability by procedures & controls.
lity	Performance & life testing of sterilized components, acceptance and qualification test requirements
	1000-hour performance-life tests of sterilized lander
	Specification S-31100, Plans at all applicable echelons per NPC-250-1, demonstration tests
	Approved high reliability parts, structural, circuit design required of all contractors by S-31100
delivery & program-	Early identification of quantitative demonstration testing requirements for life & reliability
pport	Detailed management plan & implementation.
plications, etc.	Early establishment of approved lists, integration, procedures & required usage by S-31100
ocesses by all	Applied materials & process research, development, evaluation, documentation & manufacturing & QC controls.
	Master specifications, approved lists, integrated usage, 100% screening, tests & controls
	Evaluation testing to minimize interpolation & extrapolation of performance & life effects data
	Management support, education, training, operator & process certification, conformance required

Figure 3.8-1. Reliability Plan Summary

MISSION PHASES	PROBLEM AREAS
	Standard electronics Thin film electronics Thermo-plastic recorders Klystrons TV cameras Mechanical, electro-mechanical, and pyrotechnic devices
Planetary Entry and Landing acceleration and thermal	1. Uncertainty of planetary environment. a. Temperature, atmospheric density, entry environment - effects on heat shield, retardation, thermal control and structure. b. Terrain conditions Impact absorption and orientation
Planetary Orbital Insertion and Operation	1. Up to three months operation. 2. Unknown environment surrounding planet. 3. Cyclic nature of operation. 4. High accuracy performance requirements. a. Attitude control stabilization b. Guidance and navigation c. Communications
Lander Surface Operation	1. Unknown planetary surface environment. a. High temperature - Venus b. Low temperature - Mars c. Terrain conditions 2. Time of Operation required. a. Several hours on Venus b. Several hours on Mars 3. Cyclic nature of operation

B. Reliability Program Management

(1) Reliability Program Plan

(a) Initial Issue

The essence of the Voyager Reliability Program is the Reliability Program Plan, which will be generated and submitted for approval within 60 days after contract award. As a part of the overall Voyager Program Plan, the tasks described in the Reliability Program Plan will be fully integrated with those performed by Engineering, Manufacturing, and Quality Control.

The Voyager Reliability Program Plan will describe in detail the specific tasks necessary to complete the reliability program, the methods and procedures for completing these tasks, the standards of performance for the tasks, and the control measures employed for assuring their completion.

(b) Revisions

The plan will be reviewed periodically and upon receipt of contract scope of work changes for resultant revisions which will be forwarded to NASA for approval within 30 days of the revision.

(2) Reliability Program Integration and Management

(a) Program Control - Reliability Management Matrix

The Reliability Plan will be implemented through several distinct and separate Reliability Management Systems. They are, however, related and make up the total Reliability Program. These systems have been integrated into a Reliability Management Matrix (RMM) which will provide management with the current status of all program reliability activities and allow for the continuous evaluation and control of reliability program elements (See Figures 3.8.1-1 and 3.8.1-2).

The essential elements of this management system are:

1. The principal events affecting reliability

2. The monitoring of hardware down to the component level

3. The indices of reliability measurement, i.e., Reliability Performance, Reliability Program Accomplishment, and Technical Evaluation.

Each principal reliability event is listed separately as a column heading in the RMM, while each equipment involved in the program, i.e., system, subsystem and component has a separate row heading. The events are measured against the hardware utilizing the three monitoring indices as mentioned above.

Through the continuously updated and revised RMM, the prime contractor and NASA program management will be constantly apprised of the reliability program status so that effective and timely action can be taken to redirect efforts on problem areas in order to insure success.

(b) Reliability Program Reviews

The reliability program has been organized and will be scheduled to permit a review of status at frequent intervals by the prime contractor and by NASA program management. The Reliability Management Matrix will form a basis for these reviews. Pertinent documentation and data generated during the program and review period will be presented for assessment of the reliability status.

Formal reviews will be conducted jointly by the contractor and NASA to assess the progress and effectiveness of the program. Schedule of the reviews will be incorporated in the Formal Reliability Program Plan.

(c) Progress and Control Reports

The contractor will submit brief weekly reports, periodic progress reports, and reliability program control reports in accordance with document NPC-250-1.

(d) Reliability Data Center

The contractor will maintain a complete centralized file of all reliability documentation for ready reference and to facilitate progress evaluation and monitoring, the data center will be under the control of the Reliability Manager.

(3) Reliability Education

Key personnel involved in the program will be apprised of the program objectives translated into terms of individual functional areas with stress on design simplicity and high component reliability. As the program progresses, information on problems affecting

DOCUMENT	NUMBER	ISSUED	REVISION	ISSUED	REMARKS
MISSION REQUIREMENTS SPEC.		40-64			
SYSTEM PERFORMANCE SPEC.		40-64			
SYSTEM RELIABILITY REQ. SPEC.		40-64			
AIRBORNE EQUIP. ENVIR. REQU. SPEC.		40-64			
GROUND EQUIP. ENVIR. REQU. SPEC.		40-64			
SYSTEM TEST REQUIREMENTS SPEC.		40-64			
DATA REQUIREMENTS SPEC.		40-64			
TERMINOLOGY GLOSSARY		40-64			
SUPPLIER REL. CONTROL STANDARD		40-64			
FINISH STANDARD		40-64			
SYSTEM FACILITIES REQU. SPEC.		40-64			
SYSTEM SELECTED PARTS LIST		40-64			
SYSTEM COMPONENTS TEST REQU. SPEC.		40-64			
INTERFACE SPEC. ORBITER LANDER		40-64			

DESIGN STUDY

PRELIMINARY DESIGN

DEVELOPMENT AND PROTOTYPE

SAMPLE

SYSTEM BREAKDOWN	MONITORING EVENT SPEC. NO.	DRWG NO	REL. EST.	APPORT DOC NO DATE	RELIABILITY DESIGN ESTIMATE DOC NO.	SCHED	ACTION ITEMS #OPEN / TOTAL #	PRED. INDEX	CONCEPTUAL DESIGN REVIEW DOC NO	SCHED.	ACTION ITEMS #OPEN / TOTAL #	RELIABILITY DESIGN ANALYSIS DOC NO	SCHED	ACTION ITEMS #OPEN / TOTAL #	DESIGN INDEX	SPECIFICATION REVIEW AND ITPB APPROVAL INITIAL	REVISION	DESIGN REV N DOC NO	SCHED
LANDER VEHICLE SYSTEM			.86[illegible]			47-64				41-64			39-66			52-66			20-65
										41-64									21-65
ELECTRICAL POWER AND DISTRIBUTION SUBSYSTEM			.933			46-64				41-64			34-66			11-66			1-65
RADIOISOTOPIC-THERMOELECTRIC GENERATOR			.993			40-64				41-64			49-65			8-66			1-65
BATTERY CHARGING REGULATOR			.999			40-64				41-64			52-65			9-66			1-65
NI-CAD BATTERY			.994			41-64				41-64			4-66			10-66			1-65
POWER CONTROLLER			.999			41-64				41-64			9-66			11-66			1-65
POWER DISTRIBUTION BOARD			.999			42-64				41-64			15-66			12-66			2-65
HARNESSES, CABLING AND CONNECTORS			.999			42-64				41-64			21-66			13-66			2-65
ORIENTATION SUBSYSTEM			.987			45-64				42-64			34-66			13-66			[illegible]-65
G-SWITCHES			.999			40-64				42-64			48-65			2-66			2-65
ARMING RELAYS			.999			41-64				42-64			50-65			3-66			2-65
DISARM RELAYS			.998			41-64				42-64			1-66			4-66			3-65
MERCURY SWITCHES			.999			42-64				42-64			3-66			5-66			3-65
MOTION DETECTORS			.999			43-64				42-64			6-66			6-66			3-65
TIMER			.999			44-64				42-64			8-66			7-66			3-65
MOTOR, GEAR TRAIN			.999			44-64				43-64			11-66			8-66			3-65
DEPLOYMENT MECHANISMS			.999			44-64				42-64			13-66			9-66			4-65
ELECTRO-MECHANICAL ACTUATORS			.999			47-64				43-64			16-66			10-66			6-65
TILT BAR			.999			45-64				43-64			18-66			11-66			4-65
HARPOONS			.999			45-64				43-64			20-66			12-66			4-65
SOLID ROCKETS			.998			45-64				43-64			21-66			13-66			4-65
RETARDATION SUBSYSTEM			.989			46-64				44-64			4-66			30-66			23-65
REMOTE ACTIVATED BATTERIES			.997			43-64				44-64			45-65			17-66			1-65
ARMING RELAYS			.999			44-64				44-64			51-65			18-66			3-65
G-SWITCHES			.999			44-64				44-64			1-66			19-66			2-65
TIMERS			.999			44-64				44-64			4-66			20-66			3-65
TIME DELAYS			.999			41-64				44-64			6-66			21-66			5-65
DROGUE MORTAR			.999			44-64				44-64			9-66			22-66			5-65
DECELERATION CHUTE			.999			42-64				44-64			11-66			23-66			6-65
AFT COVER EXPLOSIVE BOLTS			.999			47-64				45-64			14-66			24-66			6-65
INFLIGHT DISCONNECT			.999			41-64				45-64			16-66			25-66			7-65
MAIN PARACHUTE			.999			41-64				4[illegible]-64			18-66			26-66			7-65
SWIVEL, PARACHUTE			.999			47-64				45-64			19-66			27-66			7-65
REEF LINE CUTTERS			.999			40-64				45-64			20-66			28-66			7-65
CUTOFF FITTINGS			.999			40-64				45-64			21-66			29-66			5-65
PROPULSION AND SEPARATION SUBSYSTEM			.984			46-64				46-64			34-66			38-66			24-65
INFLIGHT DISCONNECT			.999			43-64				46-64			49-65			29-66			8-65
EXPLOSIVE BOLTS			.999			43-64				46-64			52-65			20-66			8-65
GAS TANK (SEPARATION)			.999			43-64				46-64			2-66			21-66			8-65
SQUIB VALVE (SEPARATION)			.999			44-64				46-64			5-66			32-66			9-65
NOZZLES (SEPARATION)			.996			42-64				46-64			7-66			33-66			10-65
GAS TANK (SPIN)			.999			42-64				46-64			10-66			34-66			9-65
SQUIB VALVE (SPIN)			.999			41-64				46-64			12-66			35-66			9-65
NOZZLES (SPIN)			.996			45-64				46-64			15-66			36-66			10-65
DELTA-V SOLID ROCKET			.999			45-64				46-64			17-66			37-66			5-65
ADAPTER EXPLOSIVE BOLTS			.999			45-64				46-64			21-66			38-66			6-65
THERMAL CONTROL SUBSYSTEM			.958			46-64				47-64			54-66			30-66			[illegible]
WATER TANK			.999			4[illegible]-64				47-64			50-65			14-66			10-65
SOLENOID VALVE			.995			48-64				47-64			[illegible]			15-66			10-65
WATER BOILER			.999			40-64				47-64			52-65			16-66			10-65
HEAT EXCHANGERS			.997			40-64				47-64			2-66			17-66			10-65
PUMPS			.999			40-64				47-64			4-66			18-66			11-65
ELECTRIC MOTORS			.999			41-64				47-64			5-66			19-66			11-65
SQUIB VALVES			.995			41-64				47-64			7-66			20-66			9-65
SQUIB VALVE AND GUILLOTINE			.995			41-64				47-64			8-66			21-66			9-65
CHECK VALVES			.995			44-64				48-64			10-66			22-66			12-65
SURFACE RADIATORS			.997			44-64				48-64			11-66			23-66			12-65
IN-TRANSIT RADIATOR			.997			44-64				46-64			14-66			24-66			12-65
ACCUMULATORS			.999			45-64				48-64			14-66			25-66			13-65
COMPONENT SURFACE PLATES			.997			45-64				48-64			16-66			26-66			13-65
MODULATION VALVES			.995			45-64				48-64			17-66			27-66			12-65
TEMPERATURE SENSOR			.999			45-64				49-64			19-66			28-66			13-65
TEMPERATURE CONTROLLER			.999			43-64				48-64			21-66			29-66			13-65
COMMUNICATIONS SUBSYSTEM			.968			40-64				49-64			[illegible]			21-66			[illegible]
VHF TRANSMITTER			.999			42-64				49-64			9-66			5-66			18-65
POWER AMPLIFIERS			.999			42-64				49-64			11-66			6-66			14-65
DIPLEXER (VHF)			.999			42-64				49-64			48-65			7-66			14-65
VHF RECEIVER			.999			43-64				49-64			13-66			8-66			16-65
WHIP ANTENNAS (VHF)			.999			44-64				49-64			49-65			9-66			17-65
OMNI ANTENNAS (VHF)			.999			44-64				49-64			50-65			10-66			17-65
KLYSTRON AND POWER SUPPLY			.997			45-64				49-64			15-66			11-66			16-65
DIPLEXER (SHF)			.999			41-64				49-64			51-65			12-66			14-65
TRANSPONDER			.995			41-64				50-64			17-66			13-66			14-65
COMMAND DEMODULATORS			.999			42-64				50-64			19-66			14-66			16-65
PROGRAMMER			.999			42-64				50-64			21-66			15-66			16-65
DATA MULTIPLEXER			.999			40-64				50-64			7-66			16-66			16-65
RECORDERS			.993			40-64				50-64			5-66			17-66			15-65
ANTENNA, HELICAL, (SHF)			.997			40-64				50-64			1-66			18-66			17-65
ANTENNA TRACKING SYSTEM			.997			45-64				50-64			3-66			19-66			17-65
OMNI ANTENNA, (SHF)			.997			41-64				50-64			52-65			20-66			17-65

RELIABILITY MANAGEMENT MATRIX

VOYAGER PROGRAM
(PAGE 1 OF 2)

PREPRODUCTION AND INTEGRATED TEST

PRODUCTION AND OPERATION

D 2	SUPPLIER RELIABILITY ACCEPTANCE		PARTS APPLICATION REVIEW			RELIABILITY DESIGN ANALYSIS				DESIGN REV NO 3			QUAL. APPROVAL		RELIA. DEMO.		FAILURE SUMMARY REPORT NO. DOCUMENT NO SCHED 39-65			EQUIP. REL. STATUS REPT. DOCUMENT NO SCHED 25-65		ENGINEERING CHANGE		
ACTION ITEMS OPEN / TOTAL #	DOC. NO.	SCHED	DOC. NO	SCHED	ACTION ITEMS # OPEN / TOTAL #	DOC. NO	SCHED	ACTION ITEMS # OPEN / TOTAL #	DESIGN INDEX	DOC. NO.	SCHED	ACTION ITEMS # OPEN / TOTAL #	DOC. NO	SCHED	DOC. NO	PERF INDEX 80%C	CUM FAILURES; SIGNIF # / TOTAL #	FAILURE ANALYSES DUE SIGNIF / # FAILURES	ACTION ITEMS OVERDUE / TOTAL #	PERF INDEX 80 %C	CONF AT APPORT VALUE	CHANGE NO.	EFFECTIVITY DATE	REVISED DESIGN INDEX
				—			31-67				2-65			13-67										
				—			26-67				3-65			34-66										
				38-66			40-66				22-65			21-66										
				43-66			45-66				22-65			25-66										
				6-67			6-67				22-65			26-66										
				2-67			4-67				23-65			27-66										
				48-66			50-66				23-65			29-66										
				11-67			13-67				23-65			32-66										
				—			26-67				4-65			26-66										
				47-66			49-66				24-65			15-66										
				11-67			13-67				27-65			16-66										
				42-66			44-66				27-65			17-66										
				5-67			7-67				25-65			18-66										
				7-67			9-67				26-65			19-66										
				40-66			42-66				24-65			20-66										
				9-67			11-67				24-65			21-66										
				44-66			46-66				22-65			22-66										
				3-67			5-67				29-65			23-66										
				50-66			52-66				28-65			24-66										
				1-67			3-67				26-65			25-66										
				38-66			40-66				28-65			26-66										
				—			26-67				6-65			43-66										
				38-66			40-66				27-65			30-66										
				40-66			42-66				27-65			31-66										
				42-66			44-66				25-65			32-66										
				2-67			4-67				25-65			33-66										
				44-66			46-66				29-65			34-66										
				6-67			8-67				29-65			35-66										
				46-66			48-66				30-65			36-66										
				4-67			6-67				30-65			37-66										
				48-66			50-66				31-65			38-66										
				8-67			10-67				31-65			39-66										
				50-66			52-66				31-65			40-66										
				10-67			12-67				31-65			41-66										
				52-66			2-67				32-65			42-66										
				—			26-67				6-65			[illegible]										
				38-66			40-66				32-65			42-66										
				41-66			43-66				32-65			43-66										
				[illegible]			45-66				33-65			44-66										
				46-66			48-66				34-65			45-66										
				48-66			50-66				35-65			46-66										
				51-66			1-67				33-65			47-66										
				1-67			3-67				34-65			48-66										
				4-67			6-67				35-65			49-66										
				6-67			8-67				29-65			50-66										
				10-67			12-67				30-65			51-66										
				—			26-67				7-65			43-66										
				38-66			40-66				36-65			27-66										
				39-66			41-66				37-65			28-66										
				40-66			42-66				36-65			29-66										
				42-66			44-66				36-65			30-66										
				43-66			45-66				37-65			31-66										
				45-66			47-66				37-65			32-66										
				46-66			48-66				38-65			33-66										
				48-66			50-66				38-65			34-66										
				49-66			51-66				39-65			35-66										
				51-66			1-67				39-65			36-66										
				1-67			3-67				39-65			37-66										
				3-67			5-67				40-65			38-66										
				5-67			7-67				39-65			39-66										
				7-67			9-67				39-65			40-66										
				9-67			11-67				40-65			41-66										
				11-67			13-67				41-65			42-66										
				—			26-67				6-65			34-66										
				38-66			40-66				41-65			15-66										
				40-66			42-66				41-65			19-66										
				41-66			43-66				42-65			20-66										
				43-66			45-66				43-65			21-66										
				44-66			46-66				44-65			22-66										
				46-66			48-66				44-65			23-66										
				47-66			49-66				45-65			24-66										
				49-66			51-66				42-65			25-66										
				50-66			52-66				43-65			26-66										
				52-66			2-67				45-65			27-66										
				2-67			4-67				45-65			28-66										
				4-67			6-67				46-65			29-66										
				6-67			8-67				46-65			30-66										
				8-67			10-67				47-65			31-66										
				10-67			12-67				47-65			32-66										
				11-67			13-67				47-65			33-66										

ESTIMATED DATE / ACTUAL DATE

Figure 3.8.1-1. Reliability Management Matrix - Lander System

DOCUMENT	NUMBER	ISSUED	REVISION	ISSUED	REMARKS
MISSION REQUIREMENTS SPEC.		40-64			
SYSTEM PERFORMANCE SPEC.		40-64			
SYSTEM RELIABILITY REQ. SPEC.		40-64			
AIRBORNE EQUIP. ENVIR. REQU. SPEC.		40-64			
GROUND EQUIP. ENVIR. REQU. SPEC.		40-64			
SYSTEM TEST REQUIREMENTS SPEC.		40-64			
DATA REQUIREMENTS SPEC.		40-64			
TERMINOLOGY GLOSSARY		40-64			
SUPPLIER RELIABILITY CONT. STAND.		40-64			
FINISH STANDARD		40-64			
SYSTEM FACILITIES REQU. SPEC.		[illegible]-64			
SYSTEM SELECTED PARTS LIST		40-64			
SYSTEM COMP. TEST REQU. SPEC.		[illegible]			
INTERFACE SPEC. ORBITER/LANDER		40-64			

DESIGN STUDY

PRELIMINARY DESIGN

DEVELOPMENT AND PROTOTYPE

SAMPL

SYSTEM BREAKDOWN	MONITORING EVENT		REL. EST.	APPORT.	RELIABILITY DESIGN ESTIMATE				CONCEPTUAL DESIGN REVIEW			RELIABILITY DESIGN ANALYSIS				SPECIFICATION REVIEW AND ITPS APPROVAL		DESIGN REV	
	SPEC. NO	DRWG. NO		DOC NO DATE	DOC. NO.	SCHED.	ACTION ITEMS #OPEN / TOTAL #	PRED. INDEX	DOC. NO.	SCHED.	ACTION ITEMS #OPEN / TOTAL #	DOC. NO.	SCHED.	ACTION ITEMS #OPEN / TOTAL #	DESIGN INDEX	INITIAL	REVISION	DOC. NO.	SCHED.
ORBITER SYSTEM			.733			47-64				41-64			39-66			52-66			20-66
POWER SUPPLY SUBSYSTEM			.900			46-64				41-64			34-66			11-66			21-65
SOLAR ARRAY			.999			45-64				41-64			30-66			12-66			5-65
REGULATORS			.942			43-64				41-64			[illegible]-66			11-66			8-65
BATTERIES			.962			42-64				41-64			20-66			10-66			10-65
ALL OTHER PARTS			.995			44-64				41-64			28-66			13-66			7-65
GUIDANCE & CONTROL SUBSYSTEM			.930			46-64				43-64			34-66			13-66			22-65
SUN SENSORS			.999			44-64				42-64			27-66			3-66			5-65
STAR TRACKER			.999			42-64				42-64			24-66			[illegible]-66			8-65
PROGRAMMER			.969			45-64				43-64			30-66			[illegible]-66			2-65
EARTH SENSOR			.995			43-64				42-64			23-66			[illegible]-66			7-65
IR SCANNER			.999			40-64				42-64			23-66			7-66			6-65
GYROS			.991			43-64				42-64			18-66			8-66			10-65
ACCELEROMETER			.999			42-64				42-64			22-66			4-66			4-65
THRUST VECTOR CONTROL			.998			43-64				42-64			13-66			9-66			7-65
ANTENNA ACTUATORS			.997			41-64				43-64			[illegible]			10-66			5-65
POWER AMPLIFIERS			.993			44-64				43-64			6-66			10-66			1-65
PHP SERVOS			.997			42-64				43-64			15-66			11-66			9-65
FEEDBACK & MODE CONTROL			.993			44-64				43-64			10-66			12-66			11-65
COMMUNICATIONS SUBSYSTEM			.884			46-64				45-64			34-66			32-66			23-65
TRANSPONDERS			.995			[illegible]				43-64			25-66			27-66			1-65
HI-GAIN ANTENNA			.952			40-64				44-64			25-66			18-66			2-65
DIPLEXERS			.999			[illegible]				44-64			23-66			20-66			6-65
OMNI ANTENNA			.999			30-64				44-64			18-66			19-66			7-65
POWER AMPLIFIERS			.998			[illegible]				44-64			26-64			21-66			[illegible]-65
H.V. POWER SUPPLY			.994			[illegible]				44-64			13-64			[illegible]			3-65
COMMAND DEMODULATOR			.981			43-64				44-64			30-64			[illegible]			8-65
VHF ANTENNA (YAGI)			.999			40-64				44-64			16-64			[illegible]			9-65
VHF ANTENNA (TURNSTILE)			.999			40-64				44-64			[illegible]-64			[illegible]			10-65
VHF TRANSMITTER			.989			45-64				44-64			8-64			[illegible]			5-65
VHF RECEIVERS			.998			43-64				44-64			21-64			[illegible]			11-65
DEMODULATOR (DATA)			.998			45-64				45-64			10-64			19-66			7-65
PROCESSOR			.995			45-64				48-64			30-64			27-66			12-65
COMMAND DECODER			.999			44-64				45-64			11-64			20-66			13-65
BUFFER UNIT			.999			42-64				45-64			15-64			17-66			14-65
RECORDING UNITS			.989			44-64				45-64			20-64			26-66			[illegible]-65
TV CAMERA			.997			45-64				45-64			26-64			30-66			15-65
PROGRAMMER			.997			45-64				45-64			7-64			30-66			3-65
POWER CONTROL & CONV.			.999			42-64				45-64			30-64			23-66			16-65
PREAMPLIFIER			.999			43-64				45-64			17-64			26-66			10-65
PROPULSION SUBSYSTEM			.994			36-64				46-64			34-64			[illegible]0-66			24-65

RELIABILITY MANAGEMENT MATRIX
VOYAGER PROGRAM
(PAGE 2 OF 2)

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PREPRODUCTION AND INTEGRATED TEST

O. 2	SUPPLIER RELIABILITY ACCEPTANCE		PARTS APPLICATION REVIEW			RELIABILITY DESIGN ANALYSIS				DESIGN REV. NO.3			QUAL. APPROVAL		RELIA. DEMO.		FAILURE SUMMARY REPORT NO. DOCUMENT NO. SCHED. 29-65			EQUIP. REL. STATUS REPT. DOCUMENT NO. SCHED. 25-65		ENGINEERING CHANGE		
ACTION ITEMS #OPEN / TOTAL #	DOC. NO.	SCHED.	DOC. NO.	SCHED.	ACTION ITEMS #OPEN / TOTAL #	DOC. NO.	SCHED.	ACTION ITEMS #OPEN / TOTAL #	DESIGN INDEX	DOC. NO.	SCHED.	ACTION ITEMS #OPEN / TOTAL #	DOC. NO.	SCHED.	DOC. NO.	PERF. INDEX 80%C	CUM. FAILURES: SIGNIF. # / TOTAL #	FAILURE ANALYSES DUE / SIGNIF. # FAILURES	ACTION ITEMS: OVERDUE / TOTAL #	PERF. INDEX 80 %C	CONF. AT APPORT. VALUE	CHANGE NO.	EFFECTIVITY DATE	REVISED DESIGN INDEX
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							28-67				3-65			34-66										
				43-66			8-67				20-65			20-66										
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ESTIMATED DATE / ACTUAL DATE

Figure 3.8.1-2. Reliability Management Matrix - Orbiter System

reliability will be obtained from such sources as design reviews, failure reports and analysis, and qualification test reports, and disseminated to all concerned personnel. The Voyager reliability material will be integrated into the existing contractor's reliability education programs so as to continuously acquaint personnel with the latest reliability principles, methods, and approaches.

(4) Subcontractor and Vendor Reliability

The major program activities will consist of the establishment of reliability requirements and controls imposed on the subcontractors and vendors, the selection of qualified suppliers, and surveillance and monitoring of the suppliers' reliability program. Reliability capability evaluation surveys of proposed vendors will be conducted, with the evaluation based on the vendor's reliability program, quality control system and examination of his facilities and past performance. Assistance will be given vendors in implementing reliability requirements imposed by the Voyager Program and in resolving reliability problems. Information obtained from vendor reliability data such as failure reports, failure analyses, test reports, etc., will be fed back to the vendor for corrective action. Progress reports on vendor reliability programs will be made, evaluating programs against requirements, identifying problems, and reporting on solutions.

(5) Government Furnished Property

Reliability data for Government Furnished Property (GFP) will be obtained with NASA assistance. In the event incomplete or no data is obtainable, the contractor will analyze the item insofar as practical and make a reliability estimate, which will be factored into the overall system for reference only. Reliability degradation issued by GFP will not be chargeable to the contractor's reliability performance.

C. Reliability Engineering

(1) Specifications

The contractor prepared Voyager specifications and revisions thereto shall be reviewed to insure that each contains the applicable requirements or instructions necessary to support the Voyager Reliability Program. The specifications to be reviewed shall be system requirements, environmental requirements, equipment design specifications (component and subsystem) and the supporting program specifications such as qualification and acceptance specifications. Applicable reliability specifications, such as subcontractor and vendor reliability requirements, will be prepared.

(2) Reliability Design Analysis

(a) Reliability Apportionment

The initial apportionment, to the functional level during preliminary design, will guide the subsystem and component engineers on the required redundancy and conceptual basis for design. A preliminary apportionment made at this level, during this design study, is presented in Figures 3.8.1-1 and 3.8.1-2.

Re-apportionments during the Development and Prototype phase and the Preproduction and Integrated Test phase, at the subsystem and component levels, will be included in component specifications. Apportionment factors to be employed will include the relative system complexity, severity of the operational environment (stress/duty), criticality and state-of-the-art factors as criteria of judging. The apportioned reliability indices are used for design, guidance, part evaluation and circuit design.

(b) Prediction Models

The initial step, that of defining mission success, must consider such things as the environment to be encountered during all phases of the mission, the intended performance capabilities and the mission time. The Reliability Block Diagram will be constructed by considering the functional relationships of parts, components, and subsystems which comprise the system and defining whether series, redundant parallel, majority logic, or other type situations exist in contributing towards the ultimate success of the mission.

The Reliability Mathematical Model will be derived by converting the functional block diagram into a Boolean expression which is then translated into a mathematical expression by means of the Laws of Probability. For those designs which do not lend themselves readily to an exact solution by the use of Boolean Algebra, matrix techniques are employed.

(c) Reliability Design Trade-off Analyses

Reliability trade-off analyses in accordance with the contractor's Reliability Trade-off Analysis Manual will be performed initially during the early stages of design and continually during the development phases. Trade-offs will include, as a minimum, the following parameters:

1. System Performance
2. Weight
3. Scientific Value
4. Failure Modes
5. Reliability
6. Cost, Design and Test
7. Schedule Commitments

(d) Failure Mode and Effects Analyses

Failure mode and effects analyses will be performed on each component and subsystem design to assess the relative likelihood of various failure modes occurring and their associated effect or contribution to system failure. Part Failure Mode Probability Tables are published in the Contractor's Reliability Analysis Manual (TRA-873-74) and other documents for certain types of parts while engineering judgment will be employed to estimate failure modes and their respective probabilities of occurrence where published information is not available. Failure Mode and Effects Analysis will be used as part of the complete Reliability Figure of Merit Analysis to identify potential problem areas early in the design phase.

(e) Parts Application and Usage Review

Part Application and Usage Data, provided by design engineers, will describe the piece parts utilized, the stresses encountered due to the outside environment and the circuit loading to be used to assess the failure rate in terms of failures per mission, cycle, or hour as applicable. Failure rates and failure distributions based upon GE-MSD High Reliability Parts programs will be used. Parts selection will be coordinated with the Division's parts specialists to assure optimum application of the parts utilized.

(f) Reliability Figure of Merit Analysis

The techniques and procedures to be employed for the design reliability evaluation of the Voyager system are presented in DOD Document PB 181 080, "Reliability Analysis Data for Systems and Component Design Engineers," (also published as the contractor's Document TRA-873-74). This computation of the Reliability Figure of Merit (RFM) will be performed by applying the parts failure rates obtained for electronic and mechanical parts and the results of structural, thermal and environmental analyses to the Reliability Mathematical Model which was previously derived. In addition, worst case drift and statistical tolerance analyses will be performed to include the failure probability by means of other than pure catastrophic failures.

(3) Design Review

Voyager design reviews will provide a detailed technical appraisal on the approach taken in the design to solve the particular problems involved, the immediate technical position at the time, and technical suggestions and recommendations for immediate and future steps to be taken, reliability considerations interface problems, etc.

Design reviews, with all operations concerned participating, will be conducted on the Voyager system, all its subsystems and components, at each design phase as indicated in the Reliability Monitoring Matrix. In addition, special reviews will be made of design manufacture cycle critical elements, such as design concepts depending on advances in the relevant state-of-the-art and unusual manufacturing techniques, sterilization and special test procedures or equipment.

Minutes of reviews will be issued to all operations concerned; and action items, recommendations, and suggestions will be sent to responsible persons for implementation. Reports on the implementation will be required of the responsible operations.

The design of all major subcontracted items will be reviewed in accordance with the contractor's Design Review Policies and Instructions.

(4) Design Changes

All design changes which may affect the inherent reliability of the Voyager will be evaluated by a reliability engineer assigned to the Voyager Design Change Board. A major requirement of this program is to assure that the reliability originally designed into the system is not degraded during the final design and manufacturing phases.

The proposed design change, which is reviewed by the Design Change Board for approval must satisfy all the requirements of performance, reliability, cost and schedule as required. As such, the reliability member of the Board will assure consideration of reliability.

(5) Failure Analysis and Reporting

Failure reports will be required from all test areas, including subcontractor, vendors, and field test sites, and will be accomplished at the component, subsystem, and system levels during Qualification Testing, Acceptance Operability Assurance, Special Evaluation Testing and Field Testing. Each failure will be reviewed and classified for significance (critical, major, minor, secondary, and design or non-design) prior to entering the report into the Mechanized Data System.

Formal Failure Analyses will be conducted on all critical and major failures by the Failure Analysis Board.

The Failure Analysis Board will review the history of the test failure, pinpoint the exact mode and cause of failure, determine the necessary corrective action, specify effectivity, assure that this corrective action is taken, and issue a Failure Analysis Report.

(6) Standards and Standardization

A discrete Standards Program based on the Program requirements will be established, particularly in the areas of parts, materials, processes, and technical procedures. Documentation will be issued containing the Voyager Selected Parts Lists, detailed purchase specifications for parts and materials, specifications covering processes and technical procedures, modular packaging and design practices, approved sources of supply, and vendor requirements for source testing and data.

(7) Component Parts

Since parts are a basic constraint upon system reliability, the Voyager Component Parts Program objective will be to assure that only parts of known capability and proven performance are used in design and manufacture, and are applied to assure maximum circuit function.

To achieve this objective a Voyager Selected Parts List will be issued. This list will initially include the contractor's high reliability parts, selected in a manner to minimize the number of specific part types. Only parts on the Voyager Selected Parts List will be used in the design.

Additions to the initial list can only be made by the Voyager Parts Selection Team which includes representatives from Design Engineering, Quality Control, Manufacturing, and Reliability.

The Voyager Component Parts Program will assure that parts are suitable with respect to mission environment and long-life requirements. Since considerable high reliability parts data are available, specific test programs will stress the comparative performance evaluation of newly developed component parts with that of existing parts of known capability.

D. Reliability Analysis and Evaluation

(1) Manufacturing Reliability Surveillance

Reliability personnel will establish requirements for the Voyager manufacturing reliability program, monitor the program and furnish consulting services. These requirements will include reviews of manufacturing plans, screening of selected parts, development of special handling procedures and process controls, and controlling the procedures for investigating and resolving manufacturing related problems affecting reliability of the end product. Cognizance of the problems will be maintained via failure reports, failure analyses, test data reports, etc.

(2) Quality Control Reliability Surveillance

Requirements for monitoring the program will include the study of critical inspection processes to determine the norm and variance of parameters significant to the quality and reliability of the product, the placement of control points in the process flow charts, the type and location of control points to measure the effectiveness of the manufacturing operation, and the review of initial manufacturing instructions and inspection procedures for simplicity of operation and the effect on product reliability.

(3) Integrated Test Program

A comprehensive test program will be integrated with the design, development and reliability evaluation program to assess the performance capability of the system, subsystems and components.

Test specifications will be prepared for each type of test within the integrated test program, incorporating statistical design of the test for the most effective data results for reliability evaluation. Such data will be sufficiently comprehensive to permit verification of reliability and life expectancy, interactions of components and identification of failure modes and effects as well as design and manufacturing weaknesses.

The results of the analyses of the test data will be used to effect corrective action and to provide the basis for reliability measurement/demonstration.

(4) Reliability Measurement/Demonstration

A reliability demonstration program similar to those established for previous programs will be utilized in the Voyager, using all applicable data from the integrated test programs.

The major elements of this program are:

1. The Reliability Demonstration Plan
2. The Mechanized Reliability Analysis System
3. The Equipment Reliability Status Report.

The Reliability Demonstration Plan will include:

1. The environmental profile (including countdown)
2. Schematics and descriptions of operation
3. Reliability block diagrams prepared from 2
4. Mathematical models prepared from 3
5. Data sources
6. Pass-fail criteria
7. A description of the Mechanized Reliability Analysis System
8. A description of the Equipment Reliability Status Report.

(5) Mechanized Reliability Data

(a) The Mechanized Reliability Analysis System which automates the processing and integrating of test data and the presentation of equipment reliability indices is described as follows:

1) All data from the various test activities will be collected in a controlled manner on uniform data sheets. The test data collected will include equipment identification, test description, details of the test environment and operating time, and associated failures, if any. This data will be recorded on IBM cards, processed on an IBM 7094 Electronic Data Processing Machine, and automatically assembled into the categories for the Voyager equipment.

2) This data is then converted into failure rate data and predicted reliability values for the Voyager components, subsystems, and system by using the appropriate statistical distributions and mathematical models.

(b) The Equipment Reliability Status Reports which represent the output of the Mechanized Analysis System will be issued monthly. The report will present:

1) A complete listing of equipment failure rates and reliability indices of Voyager equipment for each anticipated environmental condition and for operation across the overall duty/stress mission.

2) A discussion of problem areas and corrective action to be taken.

These reports provide management with a quantitative measure of reliability status while affording the design engineer an opportunity for any needed redesign early in the development program.

E. Reliability Documentation, Reports and Data Submittal

The formal program plan will include a complete listing of contractor reliability documentation and reports. It will identify those to be submitted to NASA as well as a schedule of issuance.

3.8.2 QUALITY ASSURANCE PLAN

A. Purpose of Quality Assurance Plan

This plan describes procedures, processes, and policies to be implemented to assure the development and manufacture of reliable, high quality hardware on schedule and at reasonable cost during the Voyager Development Program.

B. Philosophy of Voyager Quality Assurance Plan

The overriding reliability consideration in a complex spacecraft launch program such as Voyager is that the cost of a single launch is high, the number and duration of launch opportunities are limited, and a partial failure in a vehicle component can negate the value of a flight. Additionally, the vehicle will be subjected to critical performance requirements in launch, during the long space flight, and during the severe conditions of the Mars or Venus orbit, entry and landing. The program requirements for sterilization will introduce another new and severe performance requirement on the system. These considerations require high quality, highly reliable flight hardware. Because of the very limited flight windows available for launching on an optimum flight trajectory toward Venus or Mars, it is imperative that the system perform dependably and on time during the launch checkout and countdown.

The prime objective of the Quality Assurance Program is to provide the customer with a quality product on schedule at a reasonable cost which will perform the required functions with high reliability.

The Quality System consists of:

1. A network of quality assurance procedures and controls devised through practical development and manufacturing experience and study of program requirements and quality and reliability specifications such as the NASA series 200 specifications, the 250-1 proposed reliability requirements, and the MIL-Q-9858 requirements.

2. An information system for feedback of performance data, failures, and results of corrective action so that the effectiveness of the quality system can be evaluated.

3. An organization to carry out a sophisticated development program.

C. Elements of the Quality Assurance System

1. Quality Control Operating Procedures

2. Program Plan

3. New Design Control

    Designer - Q. C. Engr. Liaison

    Design Review

4. Incoming Material Control

    Vendor Selection

    Quality Assurance Provisions

    Vendor Surveillance

    Receiving Inspection

    Vendor Quality Audit

5. Product and Process Control

    Process Control

    Materials Control

    In-process Inspection

    Operator Training & Certification

    Quality Audit

    Master Defect Control System

    Non-Conforming Material Control

6. Component Test

    First Piece Evaluation

    Failure Analysis

7. Final Assembly

8. System Test

9. Test Equipment & Facilities

10. **Measuring & Test Equipment Control**

Standards Laboratory

Calibration Laboratories

Instrumentation and Gage Control

11. Field Test & Evaluation

12. Quality & Reliability Information Feedback

(1) Quality Control Operating Procedures

The procedures and policies to be followed are documented in the Quality Control Procedure Manual.

All procedures and policies contained in the manual are reviewed bv and co-ordinated with the affected operations and carry management approval, Before release, the procedures are reviewed by the local AFQCR or NASA representative. The manual is reviewed, revised and updated as required by changes in program requirements.

The Quality Control Procedure Manuals, and all other documentation affecting production quality are distributed to all personnel having a direct effect on production quality.

The manuals are serialized and maintained up-to-date through immediate distribution of revised or new instructions and procedures.

(2) Program Plan

At the initiation of the Program, a quality assurance and test plan is written providing a listing of the specific tasks to be performed to assure high quality hardware and a description of the approaches to be taken to accomplish these tasks. This plan, which is consistent with the overall program schedule and program ground rules, will serve as the guidelines for the quality assurance activities.

(3) New Design Control

(a) Designer-QC Engineer Liaison

Incorporation of quality requirements and provisions begins with the design. The Q. C. Engineer works with the designer to establish quality requirements which must be incorporated into the hardware. Required special test points or other testing provisions are incorporated during the design. The designer, in turn, gains the benefit of past quality and test data on similar hardware, problems in purchasing, building, and testing similar hardware. Initial monitoring for the use of preferred parts and preferred designs is introduced. Planning of the acceptance test and qualification test programs is initiated and specific test philosophies are formulated. As the design progresses and breadboards and prototype models are evaluated, test failures are examined and analyzed, and part failure rate data is collected and analyzed for required corrective action.

(b) Design Reviews

Periodic formal design reviews are held to analyze the design approach taken on specific hardware. The design review is performed by experts having skills in the manufacturing, test, and in the specific design skills involved. Outside consultants are invited as required.

The objective of the design review is to review and analyze the proposed design approach and provide qualified expert and independent evaluation. Thus the designer has the benefit of constructive criticism and critique from persons expert in all skills involved in the design and its implementation into flight hardware.

(4) Incoming Material Control

(a) Vendor Selection

When it is more economical and more effective the components for the spacecraft will be purchased from qualified suppliers of developed units. It is mandatory that material and components be purchased only from suppliers with a proven performance record. Prior to the submission of bids on hardware, surveys will establish the capability of vendors to supply high quality, high reliability components. Vendor manufacturing capability, his system for controlling hardware quality, methods for measuring achieved hardware quality, test and inspection capability and handling methods, and other factors influencing quality are evaluated. Ability to meet cleanliness and sterilization requirements of the Program will be evaluated.

Results of the vendor survey are recorded by a formal confidential facility survey report. This information showing manufacturing and quality assurance capability is used in making the hardware make-buy decisions and in deciding from which vendors to make the purchases. As a result of the vendor surveys, vendors and subcontractors are selected on the basis of performance and ability to deliver high quality material on schedule and at a reasonable cost.

(b) Vendor Quality Assurance Provisions

All material requests for purchased materials or components are processed through Quality Control for the incorporation of the quality assurance and reliability requirements prior to release of the document. These Quality Assurance Provisions (QAP) show the specific quality requirements, the approved parts and materials requirements, test requirements, inspections or any special handling requirements during the product manufacture and test. Sterilization and cleanliness requirements are identified and described. The vendor is required to furnish a list of materials used in the manufacture of his product. This materials list is reviewed to insure that the materials used are compatible with the program requirements. Materials not yet on the Approved Materials List must be qualified to the Program requirements by the vendor.

The Quality Assurance Provisions identify the acceptance test requirements. The document explains what operations, processes, and tests must be witnessed and approved by the quality control vendor surveillance inspector. This document becomes a part of the purchasing contract.

(c) Vendor Surveillance

The contractor will maintain the option of performing inspections, or witnessing and approving tests or critical processes performed at the vendors' plant. The final acceptance tests will be witnessed by the contractor's vendor surveillance inspector as spelled out in the QAP forming part of the purchasing document. Final buy-off is signified by the inspector's stamp.

(d) Receiving Inspection

Upon receipt in house, subcontractors' and vendors' materials are subjected to inspection to the extent necessary to assure conformance to specification. The extent of inspection of the material necessary may be dependent upon the vendor's documentary evidence of the quality controls maintained at his facility.

The Receiving Inspection Group has the capability of performing physical, chemical, and electronic test and analysis of the material received.

All materials that are being processed through Receiving Inspection are identified by punch cards as to type of inspection to be performed and degree of acceptability of material. Rejected material is clearly marked in order that it may not accidently enter the manufacturing process. The rejected material paper work is processed through the Material Review Board so that rapid corrective action can be recommended to the vendor and subcontractor.

(e) Vendor Quality Audit

The effectiveness of the subcontractor's quality program is measured by his performance in delivering high quality hardware. Quality performance data together with ability to deliver on schedule and at a reasonable price is considered in the establishment of vendor ratings. The vendor is evaluated and compared to other vendors of the same product. This information is the foundation for the vendor conferences where the vendor's quality performance record is discussed with the vendor's management team. Corrective action is instituted and any misunderstandings that exist with the vendor are resolved.

(5) Product & Process Control

(a) Process Control

The Quality System requires that all in-house work including fabrication, assembly, inspection, and test is described in succinct and clear work instructions for the operator and inspector. These instructions are in the form of detailed manufacturing, and quality control and test plans. Process Control engineers develop and implement inspection plans, and special process controls, and periodic evaluations of materials and processes are made. Quality standards such as wire dress samples, weld specimens, paint chips, etc., are prepared for use as visual aids to assure uniform high quality hardware. Support and assistance is given to shop supervision and other manufacturing and quality control functions in the interpretation of quality standards by troubleshooting quality problems and providing prompt disposition of substandard material and assuring follow-up on corrective measures.

Specifications will be issued for all the critical processes required for the manufacture of the Voyager Spacecraft. These will include metal joining, heat treatment, coating, plating, adhesive bonding, fabrication of plastic components such as Teflon bladders, etc. These specifications will establish the required properties, pre-processing controls, in-process controls, and evaluation of the finished product.

(b) Material Control

Requirements will be established to control the properties of all materials used. Detailed procedures will be prepared to control all critical processes. These will include surface preparations preparatory to coating, or bonding; special atmosphere requirements for thermal treatments; and curing cycles for coatings, adhesives, and other organic products.

Engineering properties, both physical and mechanical, which the resultant product must possess to ensure the desired reliabile performance will be detailed. These will also be a direct result of the materials application list and the close co-ordination maintained between the designer and the materials engineer. Included will be mechanical properties of welded and bonded joints, mechanical properties subsequent to heat treatment, stability after environmental preconditioning, transmittance, and other radioactive properties, leakage, and contamination levels. Ability to be sterilized by high temperature will be a requirement for materials used on the Lander Capsule.

The test procedures will include the frequency of sampling and the number and types of tests to be performed to assure that the property requirements have been met. These procedures will emphasize in-process control as well as the evaluation of the finished product. An optimum balance will be maintained between destructive testing of a percentage of the product and evaluation of representative test pieces. Standard test methods, including federal and ASTM methods, will be used where possible under the environmental conditions of the Voyager mission. Test methods will include the standard mechanical tests, special notch toughness tests, metallographic examination, spectrographic determination of radiative properties, and mass spectrographic and weight loss of outgassing products.

(c) In Process Inspection

Inspection stations are located in the manufacturing areas so that all material moving through the plant can be monitored and controlled. Special attention is given to critical operations such as welding of electronic parts in module fabrication, bonding of solar cells to supporting structure, welding of the structure, etc.

Inspectors are trained and adept in the specific types of inspection activity they will be performing. Training programs covering different manufacturing and inspection processes have been set up to qualify operators and inspectors. Periodic re-certification of operators is required.

Each inspection performed is certified by the inspector's stamp which indicates operation inspected, status of material and identification of the inspector performing the inspection.

(d) Operator and Inspector Training

Training programs will insure that manufacturing personnel and test operators and inspectors are thoroughly familar with the equipment they are using and the procedures they must follow. The training program will emphasize the high level of quality assurance and reliability requirements peculiar to the Program.

To conduct effectve in-process and final inspection of critical manufacturing processes, inspection personnel must be failiar with the basic principles of the processes, the potential problem areas, and the types of defects which can occur. In addition, they must have detailed knowledge of the test methods used to evaluate these processes. Training classes will be conducted to accomplish the above objectives with emphasis on the high reliability requirements associated with the Voyager Program.

Training programs will be organized and conducted by materials and processes engineers who have authorative knowledge of the processes involved. Among these critical processes are heat treatment, metal joining, adhesive bonding, surface cleaning, application of surface coatings, and plating. Inspection methods to be taught include radiography, penetrant inspection, standard mechanical tests, metallographic examination and leak detection using the mass spectrometer. Practical experience will be included wherever possible.

(e) Operator & Inspector Certification

Upon successfully passing a training course, the operators and inspectors will receive a certificate permitting them to perform on the particular process covered. The certification will remain valid as long as a satisfactory performance record is maintained. If new type processes or inspection techniques are introduced, certified operators and inspectors will receive the required training and their certificates will be updated.

(f) Quality Audit

To establish an independent impartial evaluation of quality assurance procedures, processes and techniques, a team of highly trained engineers and specialists will make systematic audits of manufacturing and testing facilities. Since this group is organizationally independent from the performing activities a true measure of the resulting product quality is more likely to be obtained. The Quality Audit function includes a review of contract requirements, evaluation of special processes, evaluation of the existence and adequacy of procedures, review of calibration activities, review of inspection and test practices, adherence to approved manufacturing processes and established procedures, evaluation of packaging, storage, and shipping practices and procedures.

All quality audits are performed on a random unannounced basis. Corrective action requests are sent on all deficiencies found, and follow-up audits are conducted to assure compliance. Results of all audits are summarized in a monthly report to management giving the quality performance, recommendations for corrective action, and a listing of quality problems remaining unresolved.

(g) Master Defect Control System

The master defect control system pinpoints specific fabrication defects, demands corrective action, and measures the effectiveness of the corrective action taken.

The inspection information is condensed and recorded on an inspection log. Defect codes, inspection events and work area codes, and inspector and operator identification are entered on the inspection log. The information is mechanically processed and a "Number Defective Table" compiled to determine out-of-control conditions.

(h) Non-Conforming Material Control

The control of non-conforming material is recognized as one of the most vital functions in the assurance of successful system performance. In recognition of these critical requirements, a Material Review Board is established, and functions as a primary source of data for quality assurance and reliability analysis.

When material is first found to depart from requirements, one of two courses of action is taken. If the departure is one of a non-functional nature, the discrepancy information is fed directly from the shop floor into the "Master Control System" and the discrepancy is corrected.

When functional failures occur, an Inspection Report is forwarded immediately to the Reliability function which chairs the Failure Analysis Board. If it is determined that the failure would have a critical effect on the vehicle or if it is of a recurring nature, a formal failure analysis is initiated and follow-up is provided until correction of the problem is assured.

All information relative to non-conforming material is forwarded to cost accounting for determination of manufacturing loss.

(6) Component Test

Tests will be performed at the component level in accordance with the test plan and specifications. Formal detailed test instructions to implement the test plan are prepared prior to the testing. They give a detailed description of the tests, include test connections, and provide data sheets for recording test data and the total test time accumulated. When components fail to meet specifications, troubleshooting is performed by qualified engineers.

(a) **First Piece Evaluation**

A through teardown analysis will be performed on the first part which satisfactorily passes all acceptance tests to determine degradation and wear mechanisms. Information from this analysis will be used to evaluate the effectiveness of the qualification tests, to predict the margin of safety, substantiate the previous choices of critical areas requiring close control during processing, to select additional areas for control, and to improve the design. In particular, quality of the fabrication processes, wear mechanisms, effects of temperature and vacuum exposure upon organic materials, deletrious effects of sterilization treatments (as applicable), corrosion from propellants and humidity tests, and damage as a result of vibration tests will be major items of interest.

(b) Failure Analysis

Complete teardown failure analyses will be conducted on all parts, components, or subsystems which fail qualificat ion or acceptance test. When conducted properly, a failure analysis yields invaluable information that can be used as a basis for selection of alternate materials, improvement of materials and processes control, and refinements in design, all which will increase reliability. The contractor will also review, and direct when necessary, failure analyses performed by vendors and subcontractors.

(7) Final Assembly Control and Evaluation

Bonded stock areas are maintained to accumulate acceptable hardware and to provide suitable protection accountability and control until hardware is assembled into the spacecraft.

Prior to the scheduled date of assembly, manufacturing operational planning, detailing the manufacturing process, is submitted to Quality Control Engineering for the integration of the in-process inspection points. Detailed quality control planning is initiated for each inspection point, delineating the steps necessary to assure quality conformance.

Design change accounting is maintained in the Final Assembly area utilizing electronic data processing equipment to document the configuration of all assembled equipment and to assure the incorporation of all mandatory changes. This data accumulated into a log containing all the applicable vendor and in-house data, and presenting a complete overall description of the equipment and test data up to the time of shipment.

(8) Subsystem and System Test

As major sub-assemblies and assemblies are completed, they are tested utilizing a combination of specially designed test equipment and instrumentation. Test instrumentation outling in detail the procedures to be followed and the data to be recorded are provided with the test equipment. Acceptance tests, including a variety of environmental and performance tests, are performed as outlined in the test plan and the spacecraft and end items are presented for formal customer buy-off. To assure product verification, all outgoing shipments are thoroughly inspected and minor defects are corrected before shipment.

(9) Test Equipment and Facilities

To perform component and system evaluations under the various environmental conditions of space flight and to measure the quality and reliability actually achieved in the manufactured products, certain major test facilities are required. This includes facilities such as:

1. Space environmental simulation facilities

2. Vibration test facilities

3. Parts test facilities

4. Module, microelectronic and component test equipment

5. Systems Test equipment

6. Optical alignment facilities

7. Pneumatic test equipment

8. Radio frequency interference test facilities

9. Materials and processes laboratory.

Additionally, capability of designing special test equipment to perform tests where existing standard equipment will not do the job is needed. The approach taken in the design of such equipment must be to make the test equipment reliable, and as simple and fool proof as possible. Where repetitive tests or a great number of tests are required, the test equipment must be designed for automatic operation when this has been determined to be the most economical approach and the approach that yields the most reliable test data.

Design capability is required in the areas of:

1. Electrical, electronic equipment

2. Optical test

3. Mechanical equipment

4. Pneumatic test

5. Material properties test

6. Solar, Stellar, Space Environment Simul.

(10) Measuring & Test Equipment Control

(a) Standards Laboratory

A modern well equipped, primary standards laboratory is required as part of the Quality Assurance Program. The reference standards have their calibration directly traceable to the National Bureau of Standards. The Standards Laboratory calibrates all transfer standards used by the Calibration Laboratory.

(b) Calibration Laboratory

The Calibration Laboratory calibrates, maintains, and controls all electronic, electro-mechanical and mechanical measuring equipment. All instruments and measuring devices are certified and calibrated in a temperature-humidity controlled environment, utilizing standards traceable to the National Bureau of Standards.

(c) Instrument and Gage Control

The Instrument and Gage Control operations procure, maintain, and control all measuring equipment; repair and service measuring equipment; operate instrument loan pools; and maintain inventory and location records of all test and measuring equipment. An automated electronic data control system maintains instrument periodic calibration records and schedules. This system of punched data cards records the location of the instrument or test equipment and its next calibration date; and assures the maintenance, calibration, and most efficient use of the inventory of available instrumentation.

(11) Field Test and Evaluation & Logistics Support

Field operations are maintained at the Missile Ranges to assure effective field test and flight support. Logistic support is provided to ship flight units and supporting GSE and spares to the field sites and to assure that the required equipment is available when it is needed. Consistent with program ground rules, modifications and retrofits are provided to maintain the latest equipment configuration.

All test and failure data initiated at a field site is documented on a Data Control Sheet. This data is transmitted to the data control office. Log books are maintained with each end item of field equipment and contain complete records of every significant event.

The FlightTest Engineering group handles all field test problems. This group performs the final checkout and launch activities, and it provides representation on the appropriate Flight Test Working Group.

(12) Quality and Reliability Information Feedback

A complete, effective quality and reliability system must of necessity include an efficient feedback mechanism to generate quality information leading to corrective action. All test data is continually monitored and analyzed, using modern statistical techniques. Vendor, manufacturing and field data is processed through electronic data processing machines to generate quality and cost trend charts and indices for use as an engineering guide for indicated product and process improvements.

Failures on components, modules, and parts are analyzed to establish cause or failure and required corrective action. A quality analysis activity combines and analyzes this data to spotlight correlations and indicators of possible continuing trouble areas. Functional failures of components and higher order assemblies are completely documented in periodic Failure Summary Reports.

If these failures are of a recurring nature or are deemed to have a critical effect on the vehicle, a formal analysis is made to give assurance that effective corrective action has been instituted. A Failure Analysis Board is estabilished for this purpose. Investigations are conducted until the cause of the failure has been found and eliminated.

Operating and non-operating time in each test environment is combined with the number of reliability failures to produce a confidence estimate of performance reliability. Reliability growth curves are plotted to depict the reliability improvement during the development cycle.

The thoroughly integrated Quality and Reliability Information Feedback System must be implemented to meet the growing stringent requirements of high reliability projects. The System completes the loop by enabling highly competent engineers to utilize inspection and test data to further improve the reliability of flight spacecraft.

3.9 STERILIZATION PLAN

The Lander Spacecraft will receive complete thermal sterilization at the launch site.

A primary design objective will be to employ in the Lander only those components and materials which will meet thermal sterilization qualification requirements. In the event that this objective cannot be met, heat sensitive components will be sterilized by other methods and inserted into the Lander by sterile processes, following Lander sterilization.

Strict cleanliness will be observed during manufacturing, testing and field operations to reduce the sterilization load factor.

A summary of sterilization problems and approaches is presented in Fig. 3.9-1.

A complete discussion of sterilization is contained in Volume V of this report.

STERILIZATION PLAN

PROBLEM AREA	TASK AREA	PROBLEMS
TRAINING & ORIENTATION	Vendors & subcontractors	Indoctrination & education of suppliers
	Personnel	Training of employees in sterilization methods
CLEAN MFG. & HANDLING PROCESSES	Vendor controls	Vendor facilities and methods
	Manufacturing controls	Maintenance of cleanliness
	Shipping controls	Contamination during packing, shipping, unpack
	AMR Assembly & test	Maintenance of cleanliness
STERILIZATION EFFECTS ON PERFORMANCE	Heat Effects	Propellants, pyrotechnics, electrical componen
	Sterilant effects	Materials, personnel, sterilant selection
	Tests of components	Effects of heat & sterilants on performance
	System tests	Effects of sterilization on lander system
PROCESSES & FACILITIES	Sterilization & assay methods	Determine effectiveness of processes
	Facility design & construction	Number of units to be processed, other progra
CLASS III COMPONENTS	Elimination of Class III	Difficulties in processing & installation
	Sterilization methods	Selection of sterilants & packaging
	Installation	Maintenance of sterility during installation
POST-STERILIZATION OPERATIONS	Check-out & servicing	Maintaining sterility of connections & acceses
	Handling	Maintaining sterility of handling points
	Radioisotope insertion	Maintaining sterility during radioisotope insert
	Flight protection	Sterile envelope design & removal

	APPROACH
	Establish sterilization training program.
	Sterilization training program.
	Establish requirements, surveys, specify processes.
	Establish & monitor processes & results.
ing	Establish & monitor processes & results.
	Establish clean facilities & processes at AMR.
ts	Heat effects testing and approval procedure
	Evaluate sterilants, processes, assembly processes.
	Performance & life testing of sterilized components
	Performance & life tests of sterilized lander
	Development & test assay methods, heat transfer studies
ı requirements	NASA liaison, consider other program requirements, plan & construction
	Development objective of 100% heat sterilization
	Development testing, process trials
	Development of optimum configuration, barriers, etc.
	Define requirements, evaluate designs.
	Define requirements, evaluate designs.
on	Define requirements, evaluate designs.
	Coordinate spacecraft & shroud design.

Figure 3.9-1. Sterilization Plan

3.10 MANUFACTURING PLAN

The Voyager Spacecraft will be manufactured with particular attention directed towards cleanliness and reliability. Subsequent sterilization of the Lander Spacecraft requires that it be fabricated, assembled and tested with strict cleanliness in order to reduce the sterilization load factor. In other respects no critical or unusual manufacturing problems are anticipated.

The following chart, Fig. 3.10-1 summarizes manufacturing plans, anticipated problems and approaches to their solutions.

PROBLEM AREA	NATURE OF PROBLEMS	PROBLEMS
CLEANLINESS & STERILIZATION CONTROL	Vendor control	Facilities, methods, indoctrination
	Fabrication, assembly & test	Clean & sterile process development & control
	Handling & storage	Contamination during handling or storage
	Personnel	Training, clothing, performance monitoring
MANUFACTURING RELIABILITY	Engineering Design	Effects on reliability
	Tools, fixtures & equipment	Maintenance, calibration, repairs, modificatio
	Process control	Effects of variations in processes
	Personnel	Workmanship, work habits
PROCESSES & TECHNIQUES	Honeycomb Thin-Skin Structure	Handling, bonding, inspection
	Parts Processing	Cleaning for clean environment assembly
	Assembly & Handling	Microminiaturized assembly, tubing welding, a
	Interface control	Launch vehicle adapter, lander-orbiter mating,
MANUFACTURING FACILITIES	Clean rooms	Area required, class 10K cleanliness
	Sterilization - Heat & Fluid	Sterilization of lander and its components
	Plastics & bonding	Thin-skin honeycomb, crush-up structure, hea
PURCH. & SUB-CONTR.	Purchase of Hardware	High-reliability & developmental hardware pur
FINAL ASSEMBLY	Material Handling	Handling light-weight structures
	Interconnections	Harness routing
SHIPPING	Pack & ship	Protective containers, routing

	APPROACH
	Vendor surveys and selection, specifications and work statement, training program
	Develop, evaluate & specify procedures & processes for clean environment assembly.
	Design, develop, evaluate & utilize material handling fixtures & containers, controlled environmental bonded stock areas with close humdity control.
	Develop training program, clothing & control methods.
	Producibility and manufacturing reliability engineering will participate in design, quality tests, change review.
s	Monitor effects of maintenance, modifications, calibration.
	Applications & process engineers will develop & monitor the necessary processes
	Reliability training, workmanship control
	Development of materials, processes, fixtures
	Develop processes, facilities, packaging.
ıtenna, solar array	Develop joining techniques & fixtures, handling fixtures.
shroud	NASA liaison, interface control master tooling and mock-ups
	Utilize laminar flow construction, plan & establish adequate clean facilities.
	Plan and establish adequate sterilization facilities to reduce field sterilization load factor.
shield	Plan and establish plastics & bonding facilities.
hase	Preparation of specifications and work statements, vendor liaison & control
	Develop material handling equipment, training of personnel.
	Utilization of mock-ups
	Design packaging, specify optimum transportation plan.

Figure 3.10-1. Manufacturing Plan

3.11 MANAGEMENT PLAN

3.11.1 VOYAGER PROGRAM MANAGEMENT REQUIREMENTS

The Voyager mission requirements of sterilization, high reliability and limited launch opportunity schedule impose unique program management tasks. Some of the tasks which demand particular management attention are discussed in the following paragraphs.

A. Sterilization Management

The requirements for sterile hardware can only be met by unusual management treatment of design specifications, manufacturing processes, test procedures and facilities.

A sterilization control group shall be established reporting directly to the Program Manager, with the responsibility of assuring that the Voyager spacecraft does not carry viable organisms to the planets.

A detailed Sterilization Program Plan shall be prepared which will describe the program to be carried out in the implementation of Voyager sterilization requirements from raw materials through lift-off. The plan shall include a complete listing of specific tasks, describe the methods, and detail the implementation and control procedures for assurance that sterilization requirements are fulfilled. In addition, the plan shall provide for indoctrination and training of design, manufacturing and test personnel to instill in them the continuous sterilization consciousness required to eliminate the human error and provide fulfillment of program scientific mission requirements. The Sterilization Program Plan shall be integrated with all participants throughout the program and shall be submitted to NASA for approval. Implementation of the approved plan shall be continuously monitored to assure adherence thereto.

The Sterilization Control Group, through the Program Manager, shall have approval authority over all program parameters affecting hardware sterilization, from material selection and application through facilities, processes, procedures, personnel training, training effectiveness, launch preparation and the resultant sterile condition of flight-ready hardware.

B. Reliability and Quality Assurance Management

The high reliability requirements for the Voyager Program, when coupled with heat sterilization, and the necessary use of toxic and corrosive propellants and sterilants present unique management problems.

A Reliability and Quality Assurance Management Group shall be organized reporting directly to the Voyager Program Manager. This leadership group shall have the responsibility of assuring that fulfillment of the high reliability requirements of the program is demonstrated.

A Reliability Program Plan shall be prepared which will describe the program to be carried out in the implementation of Voyager reliability requirements, from definitive procedures and design standards through demonstration. The plan shall include a complete listing of specific tasks, describe the methods and detail the implementation and control procedures for assurance that reliability requirements are fulfilled. The plan shall provide for reliability analyses and apportionment, design review and change control, reliability data collection, failure reporting and analysis, integrated test requirements, high-reliability parts evaluation and specification, and reliability measurement and demonstration. The plan shall be submitted to NASA for approval and implementation of the approved plan shall be continuously monitored to assure compliance.

The Reliability and Quality Assurance Group shall have approval authority over parts selection, changes to released design, qualification test requirements, hardware qualification and reliability demonstration.

C. Schedule Management

The limited launch opportunity dictates that no unrecoverable schedule slips can be allowed on the Voyager program. Schedules must be an important element contributing to management and technical decisions by both NASA and the contractor. The Program Manager must watch very closely the key events which lead to the solution of technical problems and design releases, with particular attention to items critical because of time or technology. Conservative scheduling must be practiced, with appropriate back-up solutions ready for timely introduction at pre-planned decision dates, if required. The highest degree of information accuracy must be used to establish attainable hardware schedules, allowing sufficient time for production testing and pre-launch operations.

A PERT/COST network shall be prepared for monitoring program schedule and cost performance. This will lead to identification of potential critical problems and enable implementation of timely preventive action by management.

D. Subcontractor Management

Forceful management of subcontracts will be necessary to overcome vendor or subcontractor potential "independence" in fulfilling schedule (and cost) objectives while meeting the scientific mission requirements of pre-sterilization cleanliness and high reliability.

Voyager program management personnel shall be placed in subcontractors' plants where necessary, to provide timely management action for problems and to assure compliance to all program requirements.

Special contract management tools, such as penalties for non-performance, and incentives keyed to successful progress from start to finish, shall be considered for use. Incentives or penalties must be of a magnitude which will assure that the subcontractor will apply in his plant whatever forces are necessary to meet his contract agreements.

E. System Integration

The Voyager Spacecraft System has many complex subsystems which are called upon to operate in a number of different modes throughout a single mission. System investigations and decisions must be promulgated with regard to interaction of subsystems (including the scientific payload, spacecraft, launch vehicle, handling, servicing, checkout and operating personnel, equipment, and procedures) during each mission segment. Decisions must be made considering variations in operating mode and resultant intra-system actions. Proposed changes to any part of the system (once defined) must be integrated throughout all affected areas to assure that every interaction is identified.

A System Engineering Group shall be organized to report directly to the Voyager Program Manager. This group will be responsible for technical liaison with NASA in establishing mission and system requirements and for the technical integration of all system parameters and subsystems. All interfaces shall be identified, defined and controlled by means of interface drawings, specifications, or other approved documentation. A Systems Integration Plan shall be prepared which will define the integration tasks, define the methods, and detail the implementation and control procedures to be used to assure that system integration is accomplished. This plan shall be prepared in two parts. Part I shall define the tasks, methods and procedures to be applied in the NASA/Contractor relationship and Part II shall define the contractor's internal integration relationships. Part I shall be submitted to NASA for approval. Implementation of the complete plan shall be continuously monitored to assure compliance thereto.

The Systems Engineering Group shall have approval authority over all system and subsystems specifications, interface documentation and changes to released drawings and specifications.

F. Launch Operations

The Voyager spacecraft system field operations will require special management attention because of the terminal sterilization, high reliability and strict schedule requirements. A complex facility will be required with the associated equipment and detailed operating and test procedures to be used by skilled technicians.

An Operational Engineering Group shall be organized reporting directly to the Program Manager. This group will be responsible for preparation of specifications for the field facility (including terminal sterilization parameters) and equipment, detailed methods and procedures for processing the Voyager Spacecraft from receipt at the field through final countdown, and specification of personnel skills and training requirements. This group shall also be responsible for liaison with the NASA Launch Operations agency.

The Operational Engineering Group shall have approval authority over support equipment specifications, changes to all released drawings and specifications, quantitative and qualitative personnel skill and training requirements, and (through the Program Manager) the flight-ready decisions.

3.12 FACILITIES PLAN

3.12.1 SUMMARY

This plan presents a discussion of facilities required to implement the Voyager Mars 1969 flight program. Existing facilities may fulfill many of the requirements, while some specialized requirements will exist for new facilities.

The facilities required to implement the Voyager program will be of the conventional type except for refinements necessary to satisfy the requirements of high reliability and sterilization. The need to attain high reliability in fabrication and assembly of hardware directs the use of large clean room areas. The need for reliability assessment and assurance directs the use of component, sub-assembly and system test facilities which simulate mission life environments.

The use of clean room areas is required also to achieve and maintain pre-sterilization cleanliness. In addition, facilities will be required for treating hardware with toxic and corrosive sterilants, as well as heat, for sterilization development.

See Figure 3.12.1-1 for a summary of the Facilities Plan.

3.12.2 TEST FACILITIES

A. Introduction

Development tests will require a broad range and depth of parametric test capabilities, qualification tests will require specific tests to specific depths, while acceptance tests will require specific tests to depths generally less than qualification tests. Facilities required for these three general categories of testing are discussed in the following paragraphs. Where utilization cycles permit, facilities may be used "across the board" or shared. Some specific areas, however, will require duplicate facilities because of parallel test schedules.

B. Engineering Development

Engineering development laboratories will require about 31,500 sq ft, two thirds of which must be high headroom (30 feet) space, the remainder being equally divided between normal and medium headroom area. In addition, outside facilities will be required for air drop tests and antenna development. Individual laboratories are discussed in the following paragraphs.

(1) System Development Laboratory

A system development laboratory will be required to perform system development testing and evaluation of the over-all electrical system, deployment devices and handling, servicing and checkout interfaces and procedures.

About 8,000 sq ft of high headroom space will be required to accommodate two spacecraft prototypes and one set of associated ground stations, handling, servicing and checkout equipment. An overhead crane will be required in addition to standard tools and test equipment.

(2) Structural Test Laboratory

A structural test laboratory is required in which to prove out the structural integrity of the vehicle, and supporting structures and components.

FACILITIES PLAN

PURPOSE OF FACILITY	TYPE OF FACILITY	PROBLEMS
DEVELOPMENT TESTING	Impact Testing	Crush-up design, lander payload survival
	Simulation of Martian Surface	Orientation tests, deployment of Antenna and exp
	Flight Test	TV path guidance, retardation device
	Propulsion Test	Lander and orbiter engine development
	Microbiology laboratory	Evaluation of sterilization processes
	Dynamic damping test	Determination of Lander dynamic characteristics
DEV., ACC. OR QUAL. TESTING	Simulation of Stimuli	Tests of sensors, components, subsystem and c
	Environmental Test	Thermal vacuum, acceleration, vibration, acous
	Attitude Control Test	Test of Orbiter attitude control
	Sterilization-Heat and Fluid	Sterilization compatibility of qualification items
	RTG Storage and Handling	Development and qualification tests of Lander po
MANUFACTURING	Plastics and Bonding	Honeycomb thin-skin bonding and handling.
	Clean Areas	Production of clean flight hardware.
FIELD OPERATIONS	Clean Areas	Spacecraft assembly and test without contaminat
	Sterilization	Lander and interface sterilization
	RTG Storage and Handling	Radiosotope storage and protection

	APPROACH
	Survey existing facilities and augment as required.
eriments	Terrain studies, simulation of wind, dust, surfaces
	NASA liaison-utilization of planned missile flights, balloon and aircraft flight tests
	Survey existing facilities and augment as required.
	Plan and establish suitable laboratory.
s	Survey existing facilities (AEDC).
ontrols	Simulations of earth, sun and mars, star background.
tic, humidity tests.	Survey existing facilities and augment as necessary.
	Development of air bearing test facility or equivalent
	Develop and establish suitable facilities.
wer supply	AEC liaison, plan and establish suitable facility
	Plan and establish suitable facility.
	Survey existing facilities and augment as required.
ion.	Survey AMR facilities, NASA liaison, construction.
	NASA liaison, surveys, plan and construct as required
	AEC and NASA liaison, plan and construct as required

Figure 3.12.1-1. Facilities Plan

The Laboratory should be equipped with static loading systems, strain recording instrumentation, deflection measuring instrumentation, and overhead crane. About 6400 square feet of high headroom floor space will be required for test preparation and performance.

In addition, 3600 square feet of high headroom floor space is required to perform vibration tests of structure assemblies and sub-assemblies. Two large vibrators (MB-C210 or equivalent) will be required. A comprehensive data acquisition system is required to record and measure acceleration responses.

(3) Environment Test Facilities

Environmental laboratory test facilities are required which include accelerators, shock testers, temperature chambers, thermal vacuum chambers, humidity chambers, vibrators, and associated instrumentation.

This facility (which may be centralized) would provide the environmental equipment to test components and subsystems.

Access to a large space simulator which includes cold walls, solar simulation and three-axis simulation is required (specified in system qualification section, paragraph 3.12.2-C).

(4) Pneumatics Laboratory

All assemblies of hydraulic and pneumatic hardware systems would be breadboarded and proof tested in this laboratory. A 6000 psi pressurizing gas supply system and high pressure test cell will be required.

Approximately 2400 square feet of medium high headroom is required for this laboratory.

(5) Ordnance Chambers

Explosive test chambers will be required for squibs, pyrotechnics, and other electro-explosive devices.

(6) Electric Power Laboratory

The power subsystem will be developed, breadboarded and evaluated in this laboratory. It should be equipped to assemble models of solar arrays, measure spectral response of solar cells, and make spectrophotometric measurements. Approximately 1000 square feet of normal height headroom is required.

(7) Guidance and Control Laboratory

This laboratory would be used to develop the guidance and control subsystem. It will need a capability to provide computer simulations representing the thrust vector control and attitude control loops of the system.

An area of 3200 square feet of normal headroom would be required for breadboarding and testing the attitude control system.

In addition a three axis simulation facility is required. Simulation of celestial reference bodies is required to provide attitude control system stimuli.

(8) Propulsion

Rocket engine performance tests will require access to static test stands, instrumented to record pressures, flows and thrust.

(9) Communications Laboratory

The communications system will be breadboarded for the command system for the Orbiter and Lander; the telemetry system for data collection, transmission and reception; and the television camera system for optics and operation characteristics.

An outdoor antenna range would be required for system evaluation, including ground station. Anticipated breadboarding area would require 2400 square feet of normal headroom. An RFI room enclosing 900 square feet of 20-foot high headroom will be required. These areas must be equippped with standard, electronic test gear.

(10) Sterilization Development Laboratory

The use of a sterilization facility is required to evaluate the capability of all materials used in the vehicle to be sterilized by heat or other means with no detrimental effects. This laboratory will require a high temperature chamber, and should provide for the safe storage, handling, application and evaluation of toxic and corrosive sterilants, such as ethylene oxide.

(11) Special Facilities

Evaluation of the Lander retardation equipment and erection mechanisms will require a test range for parachute drop tests over various terrains which simulate estimated surface conditions of the planet.

C. Qualification

(1) Spacecraft Components

A component environmental laboratory equipped with conventional equipment for shock, vibration, acceleration, humidity, noise, life, interference and possibly sand, and dust testing will be required. In addition, "sterilization effects" and thermal-vacuum life testing will be necessary.

Heat sterilizable components will require a temperature chamber programmed to achieve and maintain 145 $\pm$ 2^{o}C, about 4 ft by 4 ft by 4 ft in size. Components which require gas sterilization will require a temperature controlled gas chamber compatible to specified sterilants.

These facilities must be used in conjunction with a sterilized area equipped to assay the achieved sterility of the specimen.

Components will also require tests in a simulated space environment. These tests will require vacuum chambers with 10^{-6} torr capability and cryogenic walls. Component mounting surfaces must be temperature controllable from very low temperatures up to 200^{o}F. Components normally mounted external to the spacecraft will require exposure to solar inputs in accordance with Johnson's spectral distribution, collimated, with intensities of 0.62KW/m^2 to 2.67 KW/m^2.

(2) Spacecraft System

System qualification tests will require an area to accommodate a spacecraft system consisting of an Orbiter and two Landers, complete with antennas and solar array. Space must also be provided to accommodate ground handling, servicing, and checkout equipment.

The following chambers and equipment will be required:

Parameter	Range	Size or Load
Air Transportability Vibration	5 to 26 cps 1.3 g peak 26 to 50 cps .036 in. double amplitude	28,000 lb output each table
Temperature	to $-35^{o}F$	20 ft dia x 14 ft high
Altitude	59 - 50,000 feet	
Shock	30 g-11ms	
Sterilization Compatibility		
Heat	to $145^{o}C$	10 ft dia x 10 ft high
Gas	to 12% ethylene oxide	
Temperature/Humidity	to $125^{o}F$ 95 to 100% RH	20 ft dia x 14 ft high
Vibration	MB-C210 type or equiv.	2 tables, 28,000 lbs output each
Acceleration	to 10 g	7500 lb
Acoustic	59 160 db	30 ft x 30 ft x 30 ft
Thermal-vacuum	10-8 Torr. 100^{o}, .9 emissivity 0.62 to 1.4 kw/m^2 5% collimation	24 ft dia x 20 ft high illumination area of 20 ft dia

D. Quality Acceptance

A systems test area will be required for performing final factory testing on the Voyager System prior to shipment.

A pneumatics test area will be required including a 6000 psi pressurizing gas supply system, high pressure test cell with external control and monitoring equipment, and a leak test chamber. The pneumatics area must be located in or adjoining a clean room area to accommodate any disassembly or rework required as a result of testing.

The alignment of sensors, thrust chambers and antennas requires a temperature and humidity controlled area equipped with a large surface plate with optical tooling.

Weight and center of gravity measurement and adjustment will require an area large enough to accommodate the deployed Orbiter vehicle and associated holding fixtures. Three platform scales, and a Pelton or equivalent balancing machine will be required.

Mechanical interface checkout requires an area equipped with interface mockups of launch vehicle mating surface and shroud with mating connectors. This area may also be used for checking compatibility to handling fixtures and field checkout equipment.

RF radiation and EMI tests will require access to a large screen room equipped with signal generators and system test equipment.

Vibration acceptance tests require an area equipped with a shaker system with sine and random capability over a spectrum of 5 to 2000 cps. The mass of the vehicle and fixtures indicate that a dual MB-C210 or equivalent will be required. Sensors, recorders, and a programmed control system will also be required. A space simulator is required for thermal-vacuum testing. The capability must extend to 10^{-6} Torr with 0.9 emissivity cryogenic black walls, 0.62 to 1.4 KW/m^2 solar illumination with a Johnson spectral range and 45% collimation and uniformity. This chamber must be large enough for positioning and programming a specimen of 20 feet in diameter (Voyager antennas folded or simulated) and equipped with appropriate instrumentation for system performance tests, both hardwire and RF.

3.12.3 MANUFACTURING FACILITIES

A. Introduction

Voyager size, reliability, sterilization and production scheduling are the major factors affecting manufacturing facility requirements.

B. General Facilities

In addition to about 200,000 sq ft containing conventional shops, toolrooms, and stockrooms, about 100,000 sq ft of assembly and GSE space will be required.

The electronics, pneumatics, and solar cell assembly areas require about 28,000 sq ft of controlled environment space to meet manufacturing reliability objectives. Component test areas used in conjunction must also maintain controlled environments. Duplication of facility items will be required in some areas to accommodate loading imposed by the strict schedule, and controlled environment requirements. The plastics and bonding areas will need curing ovens about 10 ft by 10 ft by 7 ft high to accommodate high temperature curing of the Voyager adhesive and ablating-shield resin system.

C. Sterilization Facilities

Pre-sterilization cleaning and pilot sterilization development will require special facilities.

The controlled environment areas which are required to assure the hardware cleanliness necessary to achieve and maintain high manufacturing reliability standards will also control the pre-sterilization living organism load. In addition, about 28,000 sq ft of clean area will be required to surround the pilot sterilization chambers, and for the Lander assembly and test area.

The facilities required for pre-sterilization of flight hardware and for pilot sterilization development will consist of vapor de-greasers, ultrasonic cleaners, glove boxes, and heating ovens of component (4 ft x 4 ft x 4 ft) and assembly (10 ft x 10 ft x 7 ft) sizes installed in ultra-clean surroundings. Also a suitable gas chamber (2 ft x 2 ft x 4 ft) will be required for the application and evaluation of ethylene oxide or other sterilants.

A micro-biology laboratory will be required for assaying the effectiveness of sterilization media and methods. Pilot sterilization methods development and assay will serve to define the degree of cleanliness (allowable particles) required to attain absolute sterilization by subsequent use of heat or gas sterilants.

3.12.4 OPERATIONAL FACILITIES

A. Introduction

Facility requirements at AMR are based upon simultaneous processing of three Voyager vehicles as dictated by schedules to accommodate two launches within the approximate one-month launch window.

The facility described herein encompasses stationary building requirements only. It is assumed that unloading facilities from the shipping carrier such as docks, aircraft runways, etc., are already in existence and will be utilized. In addition, existing launch facilities will be used and handling and servicing equipment will be adapted to keep modification requirements to a minimum.

The facility requirements are based upon the following Ground Mission Profile.

Shipping components of the Voyager vehicle will comprise the following:

The Orbiter will be received at AMR in four (4) separate packages which will contain the orbiter structure, the antenna, the PHP and the booster adapter section.

Each Lander will be shipped separately in its own container.

Pyros, solid rockets, re-entry chutes, RTG fuel and servicing fluids and gases will be received individually.

Field confidence checks will not include weight and balance of any part of the vehicle.

Facilities for compressing gas to 6000 psi will be available at the site.

The Orbiter portion of the vehicle will not be sterilized.

The Lander portion of the vehicle will be sterilized within the field facility, and class III (sterile fabricated) components will have been hermetically sealed in covers which will require surface decontamination only.

Required local safety arrangements and resulting precautionary steps will be provided to permit installation of pyros and rockets in the Landers during the controlled assembly of the sterilized Lander.

Landers will be assembled to the Orbiter within the facility but the booster adapter, because of its size, will be handled as a separate unit.

B. Facility Requirements

(1) Spacecraft Assembly Building (SAB)

The SAB will accommodate receiving, checkout and assembly of the entire vehicle under one roof to assure optimum utilization of facilities and compactness of construction.

The SAB will provide inside clear dimensions of 230 ft x 250 ft and the Orbiter and Voyager assembly and checkout area will require a high bay to accommodate a 10-ton bridge type crane having a hook height of 30 feet.

The SAB will be partitioned into four (4) basic areas:

The Voyager and Orbiter assembly and checkout area - 30,000 sq ft
The Lander assembly, checkout and sterilization area - 14,000 sq ft
Administrative Office Space - 7,000 sq ft
Storage and Machinery Space - 7,200 sq ft

(2) Voyager and Orbiter Assembly and Checkout Area

The Voyager and Orbiter assembly and checkout area will be of hangar-type construction designed to support two (2) 10-ton, bridge-type cranes operating on the same tracks which will extend the full length of the area (220 ft). Cranes will provide a minimum height of 30 feet.

The apron end of the Voyager and Orbiter assembly and checkout area will require three (3) overhead-type doors having an overhead clearance of twenty (20) feet and a clear opening of 18 feet.

The floor shall be smooth concrete to permit wheeled mobility of vehicle handling equipment with or without the vehicle load.

Electrical checkout and vehicle and component test and alignment stands must be semi-permanently mounted to permit their stable use during the Voyager checkout phase, yet permit storage during the non-operational portions of the Voyager cycle (between launch opportunities).

(3) Lander Processing Area

The Lander processing area will be of finished interior construction. The area will include three (3) self-contained sterilization and reassembly chambers and an RTG Fuel Storage and Assembly Room as well as sufficient floor space to receive, inspect, checkout, prepare for sterilization, and store six Landers.

The Lander processing area will occupy space of 12,500 sq ft having nominal dimensions of 122 ft x 100 ft with a ceiling height and construction suitable to accommodate a 3 ton bridge-type crane having a minimum hook height of 15 feet. The crane will span 110 ft and will travel the full 100 ft depth of the area.

Electrical Ground Support Equipment (GSE) in the receiving, inspection and checkout area will be installed on a semi-permanent basis.

Two overhead doors having clear openings of 14 feet high by 14 feet wide, equally spaced in the end of the building, will be required. In addition, a 10 foot high by 14 foot wide door will be provided in the partition between the processing area and the equipment storage area.

The interior of the area will be constructed, and environmental control equipment provided to maintain the 12,500 sq ft area in a class 10K clean condition.

(a) Lander Sterilization, Assembly and Checkout Chambers

Three (3) sterilization, assembly and checkout chambers 15 ft x 25 ft x 18 ft high will be required to provide the necessary environment and facilities for processing the Lander through its terminal sterilization and confidence check.

Each chamber will be a self-contained unit capable of sterilization by a heat cycle of 135°C for 24 hours for Class I and II assemblies. An auxiliary compartment will also be included to permit sterilization by ethylene oxide of Class III components. This auxiliary compartment will be sealed off and insulated from the main heated compartment during the simultaneous sterilization cycle.

The main heating portion of the chamber will be insulated, lined with stainless steel and one 25 foot wall will be provided with bi-parting doors with a clear opening of 12 foot wide by 12 foot high. The other 25 foot wall will include three full size, glove box type suits connected and sealed to the inner chamber lining through accordian type, flexible tubes. These suits and tubes will be arranged to permit personnel to work manually in the chamber without degrading the sterility of the vehicle. The inside of the flexible tubes will open to the ambient of the processing area and will be supplied with air from blowers outside the chamber. During the sterilization cycle the tube openings in the wall will be closed off with insulated doors.

The chamber ambient will be controlled by ceiling to floor, recirculated, forced air. Heat will be furnished by zone controlled electric heaters in the ceiling air plenum, and post-sterilization cooling by a cooling evaporator in the recirculating system. The air supply will be filtered. The chamber will also contain a swinging, boom-type, monorail hoist of one ton capacity. Construction of the chamber will be blast proof to the extent necessary to safeguard surrounding areas and personnel in the event of a pyro or rocket ignition during the heat sterilization cycle.

Electrical GSE will be compartmented adjacent to the chamber, but the GSE operating units will be insulated from the heat. Readout instrumentation, connectors, and operating buttons will be available to the inside of the chamber but will be protected by flexible barriers of glass, metal or plastic.

Servicing fluids and gases will be stored outside the chamber and will be connected to the chamber through sterilizable wall fittings.

The Auxiliary Ethylene Oxide sterilization compartment will be constructed of polyethylene film on a stainless steel framework. The interior of the compartment, approximately 2 ft x 2 ft x 4 ft long will be arranged to facilitate complete surface contact between the gas sterilant and the components, and accessibility for installation of the components in the vehicle.

The facilities of the compartment will provide storage and distribution to the compartment of ethylene oxide at a low positive pressure. After the sterilization cycle of 18 hours, the compartment will be purged with dry sterile nitrogen followed by a purge with dry sterile air.

Special attention will be given to leak proofing the chamber to prevent the escape of ethylene oxide into surrounding areas.

(b) RTG Fuel Storage and Assembly Room

A room will be required for storage of the RTG fuel cartridges, and sterilized assembly of the components which make up the remote installation of the fuel cartridge while the vehicle is on the pad.

The room will occupy an area of 10 ft x 20 ft and will be 8 feet high. The room will be sectionalized into a cartridge storage area and an installation unit assembly room. The walls and sectionalizing partition will be constructed of a laminated structure comprising lining, insulation in the assembly area, 15 inch thick polyethylene shielding and exterior protection for the polyethylene.

The RTG fuel cartridge storage area will be arranged to support the fuel cartridges within their radioactive shields in the condition in which they are received from AEC.

The Assembly Area will be equipped with remote handling equipment designed to withdraw the cartridge from its original shielded container and insert and attach the cartridge to the pre-assembled remote loading mechanism.

The assembly area will also be provided with the environmental equipment and controls necessary to subject the remote loading mechanism to a heat sterilization cycle of 135°C for 24 hours.

(4) Environmental Machinery, GSE Storage and Laboratory Area

The environmental machinery, GSE storage and laboratory area will occupy 7200 sq ft and will be of sufficient roof height to accommodate a 3 ton, bridge-type crane with a minimum hook height of 15 feet.

The environmental machinery will consist of the units necessary to provide the required environments for the various sections of the Lander assembly, check-out and sterilization areas.

The GSE storage will consist of space for storing all handling, servicing and checkout equipment when not in use.

Laboratories will consist of partitioned areas and facilities to accommodate permanent calibration maintenance and test facilities for the operational GSE.

3.13 GROUND SUPPORT PLAN

3.13.1 INTRODUCTION

The Voyager Ground Support System will be used to perform handling, servicing and test operations on the Voyager, Orbiter and Lander Spacecraft. This plan indicates the scope of development problems anticipated and outlines plans for their solution.

3.13.2 SUPPORT EQUIPMENT ENGINEERING FUNCTIONS

A. The design and development of all handling, servicing and testing equipment required during all phases of work on the Voyager Spacecraft includes the following:

(1) Handling Equipment for:

1. Lifting
2. Orienting
3. Transporting and shipping
4. Mating
5. De-mating
6. Re-assembly of Spacecraft

(2) Servicing Equipment for:

1. Loading and unloading Orbiter propellants
2. Charging pressurized gas systems
3. Cooling electronics (during test)
4. Leak testing
5. Dynamic balancing
6. Determining weights and centers of gravity
7. Cooling radioisotope fuel capsules
8. Aligning of nozzles, sensors, cameras, etc.
9. Sterilizing and maintaining sterility of Landers
10. Loading Lander fluids
11. Arming squibs and rocket motors
12. Providing ethylene oxide flooding as required.

(3) Testing Equipment for:

1. Component, subsystem and system testing

2. Launch site testing and monitoring.

B. The formulation and execution of an engineering plan to provide a smooth and rapid flow of operations from factory to launch includes:

1. Packaging and shipping
2. Transportation at the launch site
3. Integration of range safety requirements
4. Integration of booster interfaces
5. Integration of launch site interfaces
6. Procurement and storage of fuels, sterilants, etc.
7. Servicing of Landers and Orbiter
8. Planning procedures for abort from the pad
9. Achieving and maintaining sterility of the Landers
10. Launch site monitoring
11. Final systems checkout
12. Launch site handling operation
13. On-pad testing
14. Personnel training.

3.13.3 SUPPORT EQUIPMENT ENGINEERING PROBLEMS

A. "Normal" problems of handling, servicing and checkout include:

1. When and where to conduct checkout testing
2. How much checkout testing to perform
3. Handling and servicing problems of a general nature.

B. Size of the Orbiter poses special handling problems.

1. Helicopter-barge sequences or special over-the-road techniques as may be required.

C. Loading of radioisotope fuel capsules in Landers.

1. Radiation hazard
2. Need for constant cooling
3. High weight and bulk of shielding
4. AEC licensing.

D. **Sterilization.**

1. **Special support equipment required for operations within sterile facility**
2. **Requirements and maintenance of facility.**

E. High reliability of support equipment required to meet limited launch window.

1. Extra development testing of support equipment
2. Qualification of some support equipment recommended
3. Redundancy in some designs required or stand-by equipment
4. Carefully conducted development plans including "walk-throughs," etc., required.

F. Subsequent to sterilization, Landers must remain sterile.

1. Sterile bag concept complicates handling and servicing.
2. Other concepts (including rigid containers for Landers) also have handling and servicing problems.

G. Reliability, safety and sterility requirements tend to be mutually exclusive.

1. When and where to arm squibs on sterilized Lander
2. Servicing operations on Orbiter with Lander RTG fuel capsules installed
3. When and how to install and arm rocket motors on Landers
4. Effects of ethylene oxide on personnel and equipment
5. Effects of heat sterilization on equipment
6. Difficulty in performing checkout tests subsequent to sterilization
7. When and how to load RTG fuel capsules.

H. Clean room requirements introduce equipment design problems during factory flow.

I. Cyclic usage of support equipment.

1. Storage problems.

J. Need for specialized launch facilities.

1. All requirements are not clear at the present time, but two launches plus flight-ready spare equipment within 30-day window dictates the need for extensive launch facilities.

3.13.4 SUPPORT EQUIPMENT ENGINEERING ASSUMPTIONS

A. Assume sterilization facility is at launch site (rather than at spacecraft fabrication area).

1. Cyclic usage-utilize for other programs

2. High original and operating costs discourage building two facilities.

3. To facilitate checkout testing and maximize chance of delivering sterile Lander, sterilization should occur as close as possible to launch.

B. Assume final systems checkout is at launch site.

1. Required by spacecraft complexity

2. Check required following sterilization.

C. Assume Orbiter moved by special means

1. Size of Orbiter requires helicopter-barge arrangement, special over-the-road techniques, or oversize aircraft for shipment from fabrication area to launch site.

D. Assume launch complex 37 type facilities.

1. If other facilities are provided flow plan will remain about the same; support equipment may change somewhat.

E. Assume sterile fluids introduced into Landers in sterilization facility.

F. Assume Orbiter to be fueled and pressurized on the pad.

1. Normal practice may require flyaway type disconnects

G. Assume RTG fuel capsules to be introduced into Landers at last possible point in pre- launch flow.

1. Radiation hazard (high shielding weight) precludes operation at other times.

2. Cooling requirements.

H. Assume Lander contains very few "Class III" components and that these few are designed for remote assembly.

1. If Lander contains high numbers of non-heat sterilizable items, cost and complexity of support equipment for sterilization will increase.

3.13.5 LAUNCH SITE FLOW PLAN

The flow plan (see Figure 3.7.9-1) follows the spacecraft and its components from arrival at the field to launch. Included are inspection, field testing, sterilization, transportation and servicing.

3.13.6 SUPPORT ENGINEERING EQUIPMENT LIST

Following is a list of equipment required for support of the Voyager Spacecraft. The number in parentheses following each item indicates an estimate of the number of delivered items required for support of the program. Development equipment is not included.

A. Spacecraft Mechanical GSE Items

1. Trailer, Servicing, Sterilizing Gas (3)
2. Test Stand, Spacecraft (4)
3. Personnel Work Stands (7)
4. Truck, Work and Assembly, Spacecraft (4)
5. Fixture, Handling, Spacecraft (4)
6. Prime Mover (GFE) (2)
7. Trailer, Transportation, Spacecraft (3)
8. Sling, Mating, Spacecraft (3)
9. Stand Work and Assembly Adapter (3)
10. Truck, Adapter (3)
11. Sling, Handling Adapter (3)
12. Container, Shipping, Adapter (4)

B. Spacecraft Electrical GSE Items

1. Computer (3)
2. Converter, Analog to Digital (3)
3. Test Set, RF, Spacecraft (5)
4. Simulators, Battery, Spacecraft (5)
5. Simulators, Squib (3)
6. Cable Set, Adapter, Spacecraft (5)
7. Cable Set, Ground Interconnection, SEE (5)
8. Recorder, Magnetic Tape (3)
9. Recorders, Analog (3)
10. Test Set, General Purpose (2)
11. Tapes, Computer, Programmed (3)
12. Power Supply and Switching Set (5)
13. Signal Generator (3)
14. Data Transmitter/Receiver (2)

C. Orbiter Mechanical GSE Items

1. Fixture, Handling, Orbiter (4)
2. Trailer, Servicing, Helium Gas (3)
3. Trailer, Servicing, Nitrogen Gas (3)
4. Detector Set, Leakage Rate Orbiter (3)
5. Container, Shipping, Orbiter (3)
6. Fixture, Handling, PHP (3)
7. Truck, Work and Assembly, PHP (3)
8. Test Stand, PHP (3)
9. Stimulator, Sun Sensor (3)
10. Stimulator, Canopus Sensor (3)
11. Stimulator, Planet Sensor (3)
12. Simulator, Inertia, PHP (3)
13. Simulator, Inertia, Antenna (3)
14. Detectors, Flow, Att. Control Nozzle (3)
15. Cooling System, Electronics, Orbiter (4)
16. Illuminator, Solar Array (3)

17. Container, Shipping, Antenna (3)
18. Fixture, Handling, Antenna (3)
19. Container, Shipping, Antenna (3)
20. Test Stand, Antenna (3)
21. Sling Set, Handling, Orbiter (3)
22. Alignment Set, Orbiter (3)
23. Transfer Unit, Fuel (2)
24. Transfer Unit, Oxidizer (3)
25. Trailer, Supply, Fuel (2)
26. Trailer, Supply, Oxidizer (2)
27. Detection System, Vapor (3)
28. Console, Control and Monitor, Propellants (2)
29. Suit, Protective, Propellant Handler's (16)
30. Training Set, Propellant Laoding (1)
31. Tester Set, Experiments, Orbiter (3)
32. Stimulator, Earth Sensor (3)
33. Truck, Work and Assy., Antenna (3)
34. Tool Kit, Special, Orbiter (3)
35. Fixture, Handling, Liquid Prop Engine (3)
36. Stand Support, Liquid Prop. Engine (3)
37. Sling, Handling, Liquid Prop. Engine (3)
38. Indicator, Gimbal Position, Liquid Prop. Engine (3)
39. Stand, Support, Magnetometer (3)
40. Stimulator, Magnetometer (3)
41. Stand Support, Honeycomb Panels (4)
42. Stand Support, Solar Arrays (4)
43. Stand Support, Solid Prop. Engine (4)
44. Covers, Protective, Component (4)
45. Sling Handling, PHP (3)
46. Sling Handling, Antenna (3)
47. Fixture, System Checkout (3)
48. Filter and Dryer, Auxiliary (3)

D. Orbiter Electrical GSE Items

1. Simulator, Earth Sensor (5)
2. Simulator, Sun Sensor (5)
3. Simulator, Canopus Sensor (5)
4. Simulator, Planet Sensor (5)
5. Receiver/Decommutator TLM Orbiter (5)
6. Command Generator, Orbiter (5)
7. Test Set, Experiments, Orbiter (2)
8. Cable Set, Adapter, Orbiter (2)
9. Simulator, Lander Telemetry (2)
10. Receiver, Lander Command (2)

E. Lander Mechanical GSE Items

1. Cradle, Lander (8)
2. Assembly and Checkout Stand, Lander (8)
3. Support Stand, Thermal Cover, Lander (8)
4. Support Stand, Adapter Section, Lander (8)
5. Support Stand, Nose Section, Lander (8)
6. Installation, Set, Remote, RTG Fuel Cart. (2)
7. Truck, Hand, Lander (4)
8. Cooling, System, Mobile, RTG Fuel Cart.(14)
9. Cooling System, Fixed, RTG Fuel Cart. (3)
10. Handling Set, Remote, RTG Fuel Cart. (2)

11. Tester Set, Experiments, Lander (4)
12. Handling Fixture, Adapter Section, Lander (4)
13. Sling Set, Handling, Lander (4)
14. Tool Kit, Special, Lander (3)
15. Container, Protective, Lander (8)
16. Sling, Mating, Lander (3)
17. Alignment Set, Lander (3)
18. Container, Shipping, Lander (6)
19. Handling Set, Remote, Lander (3)
20. Suit, Protective, Personnel (12)
21. Cooling System, Electronics, Lander (4)
22. Trailer, Servicing, Sterile Gas (3)
23. Trailer, Servicing, Sterile Coolant (3)
24. Handling Set, Rocket Motor, Lander (4)
25. Detector Set, Leakage Rate, Lander (4)
26. Detectors, Spin Nozzle Operation (4)
27. Training Set, RTG Fuel Cart. Loading (1)
28. Stand Support, Rocket Engine (4)
29. Sling, Handling, Rocket Engine (3)
30. Covers, Protective, Component (8)

F. Lander Electrical GSE Items

1. Receiver/Decommutator, TLM, Lander (2)
2. Command Generator, Lander (2)
3. Test Set, Experiments, Lander (2)
4. Simulators, RTG (6)
5. Cable Set, Adapter, Lander (2)
6. Entry Conditions Stimuli/ Simulator (2)

3.13.7 SUPPORT ENGINEERING SYSTEMS INTEGRATION AND DEVELOPMENT PLANS

The support of the Voyager Mars 1969 program entails the performance of many extremely complex functions. Examples of these are checkout testing, Lander sterilization, and the installation of radioisotope fuel capsules in the Landers. Not only are these operations complex, but they must be performed precisely and without delay if the launch window is to be met.

To attain the proficiency and reliability thus required, special techniques are called for in Support Engineering Systems integration and development, such as:

1. Early integration of field personnel into the design and system development cycle
2. Concurrent development of facilities with support equipment
3. Intensive training programs for field personnel
4. A comprehensive and carefully developed "walk-through" program in the field to develop proficiency and to validate equipment, facilities and procedures
5. Special development testing of certain groups of equipment (e.g., propellant loading equipment for the Orbiter, sterilization equipment for the Lander, and system checkout equipment).

The "walk-through" program is of special interest and importance. At this time, it might be considered to be a program utilizing three vehicles as they are made available and processing each through a complete field cycle from the unloading of shipping containers through a simulated launch. The first vehicle would be a mechanical model to check out all handling and some servicing equipment, including the operation of the sterilization facility. The second vehicle would include all of the features of the first, but would also carry electrical and electronic prototype gear and prototype systems such as the propellant system on the Orbiter, etc., so that systems tests and complex servicing could be conducted. The final vehicle would be an exact prototype of the flight system and the support operations would be carried out in precisely the fashion required for a flight vehicle. In the course of this operation, support equipment and facilities would be validated. If necessary, the operation could be repeated to attain the necessary confidence levels in men, equipment and procedures.

3.13.8 CHECKOUT PHILOSOPHY

Factory and field test equipment is utilized to verify that the system is built and operating to specifications, and that it survived the potentially-damaging environments experienced during acceptance testing, shipment to the field, sterilization, etc. To meet the latter requirement it is not sufficient to verify merely that the results of the test are within system tolerances, but rather the data for a particular system should be compared with previous data obtained for that system, and drifts or shifts of operating points detected. If these drifts or shifts are of sufficient magnitude, the system may be rejected even though it is still operating within limits.

The test philosophy of "end to end" testing is an approach designed to eliminate the need for test personnel to perform any disassembly to test a spacecraft. The concept consists of placing known stimuli before the front end of a subsystem and measuring the resulting system output; and, in theory at least, requires no breaking of flight connectors, etc., to perform the test. However, as a practical matter, some stimulation requirements may be extremely difficult or impossible to meet, and the vehicle should in these cases be designed to permit insertion of signals, etc., in a simple and reliable manner. The type of testing just described is the only type of test that should be run on a spacecraft from the time it is built until it is launched.

The Voyager checkout system must:

1. Accumulate and analyze large amounts of data during even one test sequence.
2. Be capable of conducting tests of long duration. The vacuum-thermal test, for example, is extended over 40 days during which time the spacecraft will be exercised periodically (at least twice a day).
3. Provide highly accurate and repeatable stimulation and measurement equipment for drift or shift detection.
4. Possess a high degree of flexibility to permit changes in test format and to provide adaptability to modifications of vehicle systems.

In view of the foregoing requirements, it is recommended that the system test equipment be a computer controlled test configuration.

The computer would:

1. Control the test sequences by setting up the peripheral equipment to operate in the desired manner.

2. Obtain measurements via the peripheral equipment.

3. Analyze, record, store, display, etc., the results obtained.

It can be a relatively small and slow-stored program computer with a good degree of priority interrupt capability. The outstanding requirement of the computer is that it generates mnemonic coding that is system-oriented and not computer-oriented.

The peripheral equipment provides the link between the computer and the spacecraft under test, and it should be capable of receiving either computer-generated or manual commands.

SECTION 4. PROGRAM SCHEDULES

4.1 MARS 1969

Two approaches to scheduling the Mars 1969 Voyager program were taken:

1. The development of a near optimum schedule, considering the complexity of the program, the requirement for high reliability and high probability of success on the first flight, a trade-off between schedule time and program costs, and a realistic starting date.

2. The development of a minimum time schedule which would permit a reasonable probability of success but with higher costs.

The detailed schedules shown in Figures 4.1-1, 4.1-2, and 4.1-3 are consistent with the first approach, and General Electric strongly recommends that they be followed. The minimum time schedule is shown by the narrow lines on the summary schedule, Figure 4.1-4.

The recommended schedules are based in part on the following ground rules:

1. Use 1965 state-of-the art in the design.
2. Deliver three flight systems.
3. Deliver one set of replaceable components as spares and one complete Lander as a spare.
4. Provide for sufficient development testing prior to start of manufacturing to allow high confidence in designs.
5. Schedule a complete and thorough test program consisting of:
    a. Development tests
    b. Qualification tests on two of each component
    c. Qualification tests on one complete system
    d. Thorough acceptance tests, including 150 hours thermal-vacuum tests on each component and 1000 hours thermal-vacuum tests on each system,
6. Deliver flight units to AMR four months prior to flight.

Given below are items of particular interest about the schedules:

1. The launch window for Mars 1969 is fixed and was used as a reference point from which to work back.

2. A four month preliminary design phase has been scheduled during which system and subsystem requirements will be firmly established. Since this phase will probably consist of parallel designs, two months have been allowed for NASA evaluation and selection of the design to be carried out.

3. Certain components have development cycles which could not be fitted into the overall schedule and these have been designated as critical items. Development of these items must be initiated during the preliminary design phase.

4. Development testing is done in parallel with the design engineering on the component, subsystem, and system levels. The system development tests will be performed on three prototype systems. The first unit will be a structural prototype to demonstrate the mechanical integrity of the system. The second will be the earliest electrically complete system unit that can be assembled, using prototype hardware to provide solutions to the many electrical problems bound to exist in the system (i.e., subsystem incompatibilities, E.M.I., test and support equipment bugs, etc.). The third unit will be a thermal prototype for thermal-vacuum demonstration of thermal design adequacy and system life. All subsystems and components are scheduled for similar

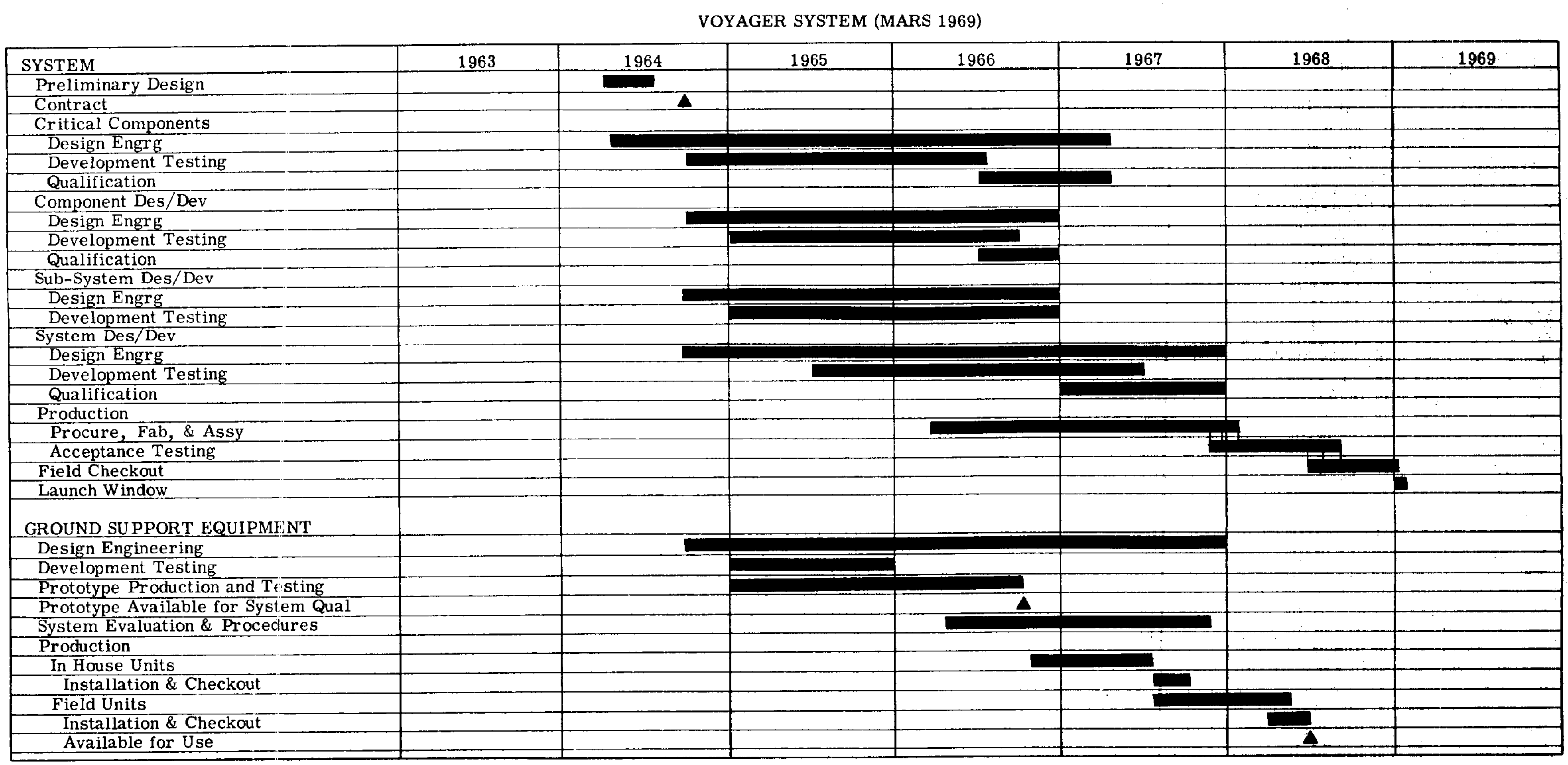
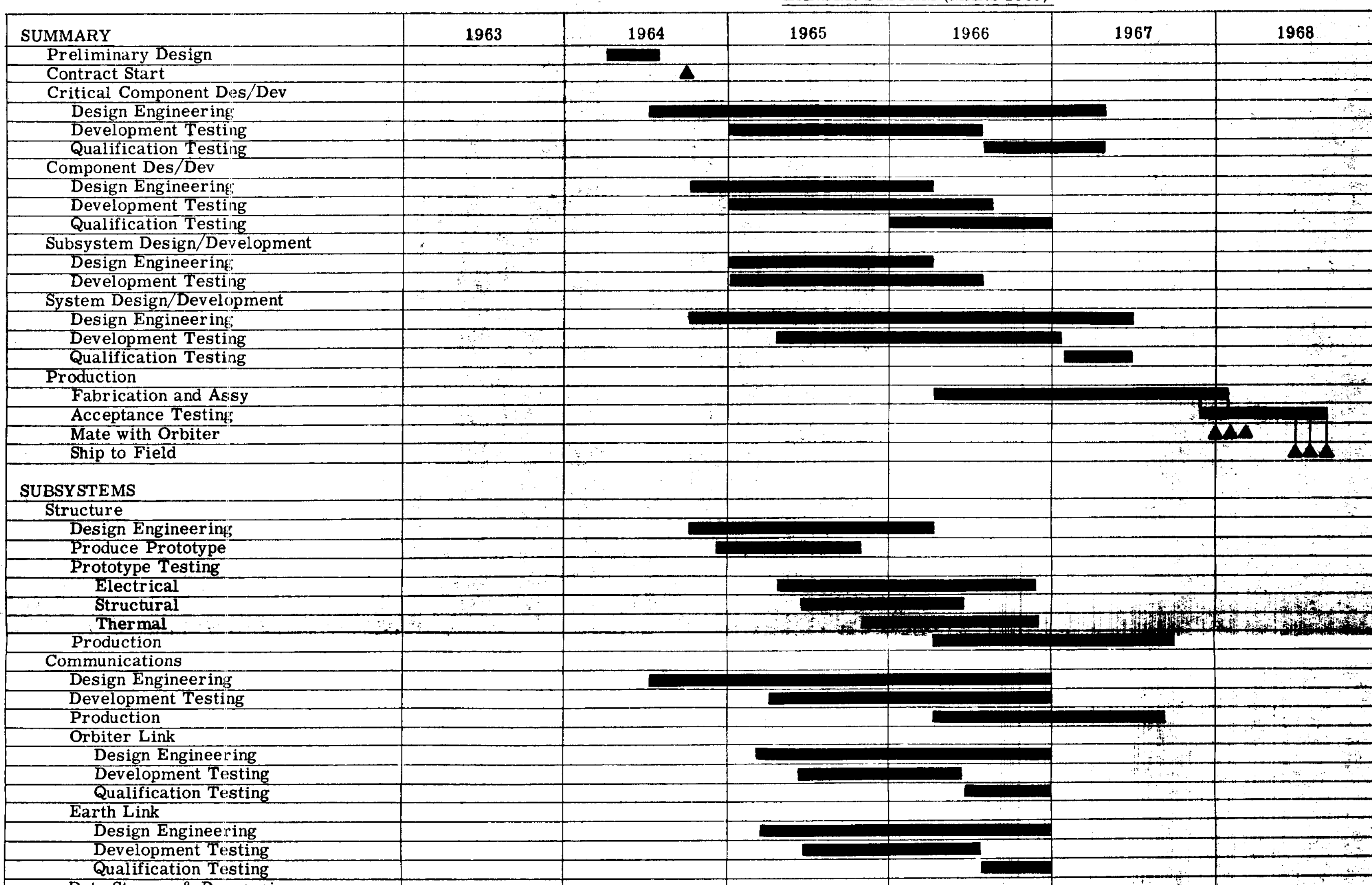


Figure 4.1-1. Voyager Program Schedule Chart

LANDER SYSTEM (MARS 1969)



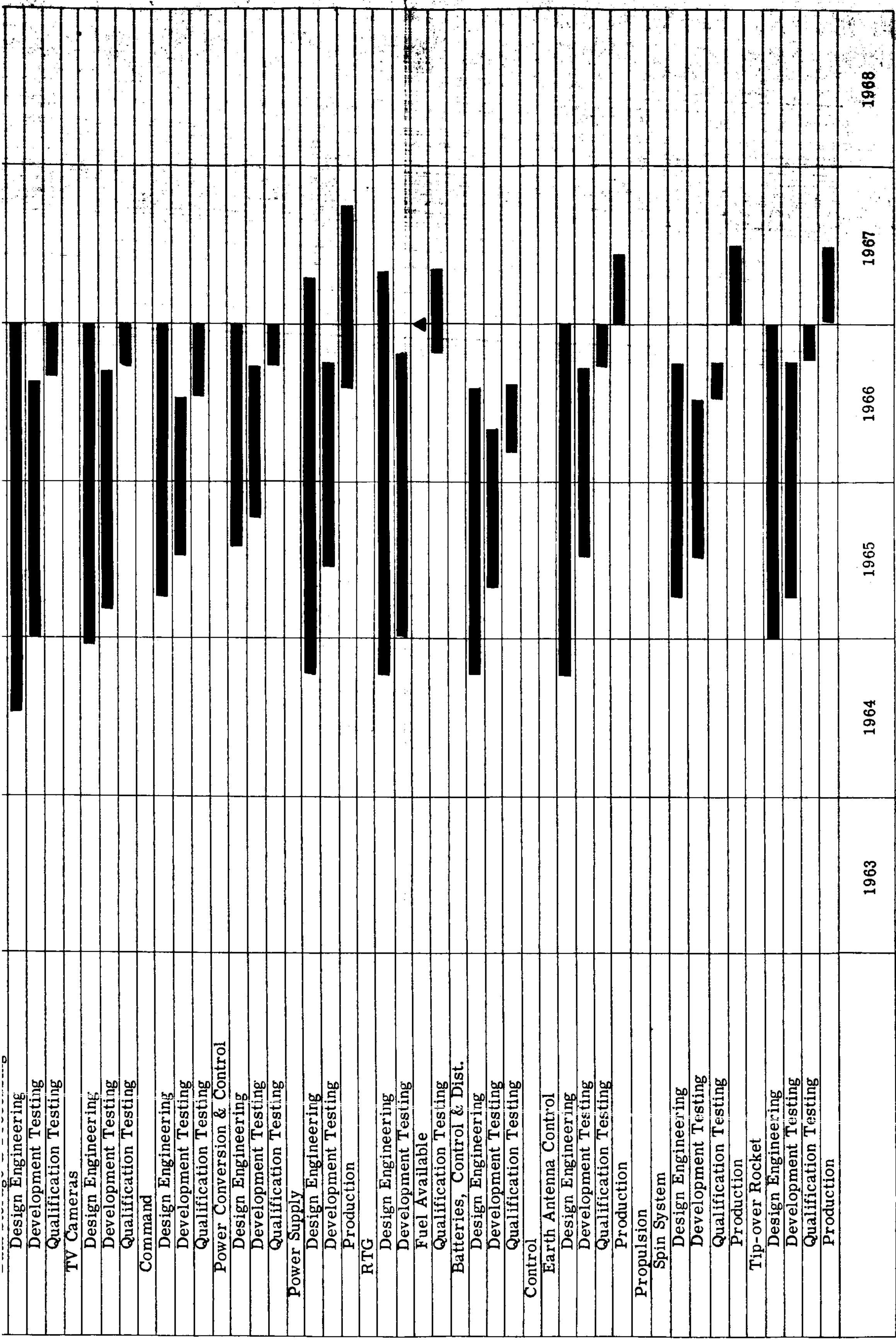
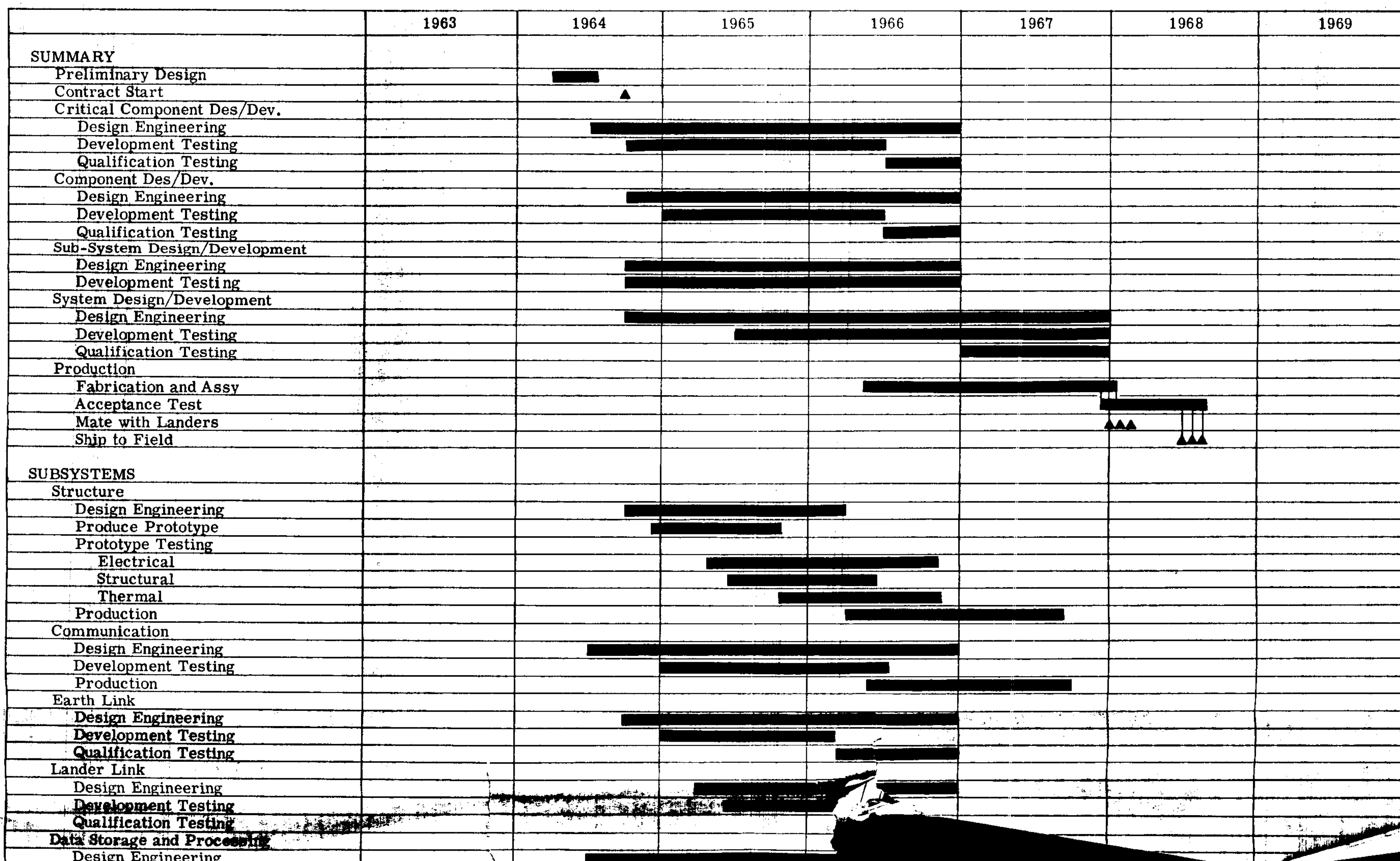


Figure 4.1-2. Lander Program Schedule Chart

ORBITER SYSTEM SCHEDULE (MARS 1969)



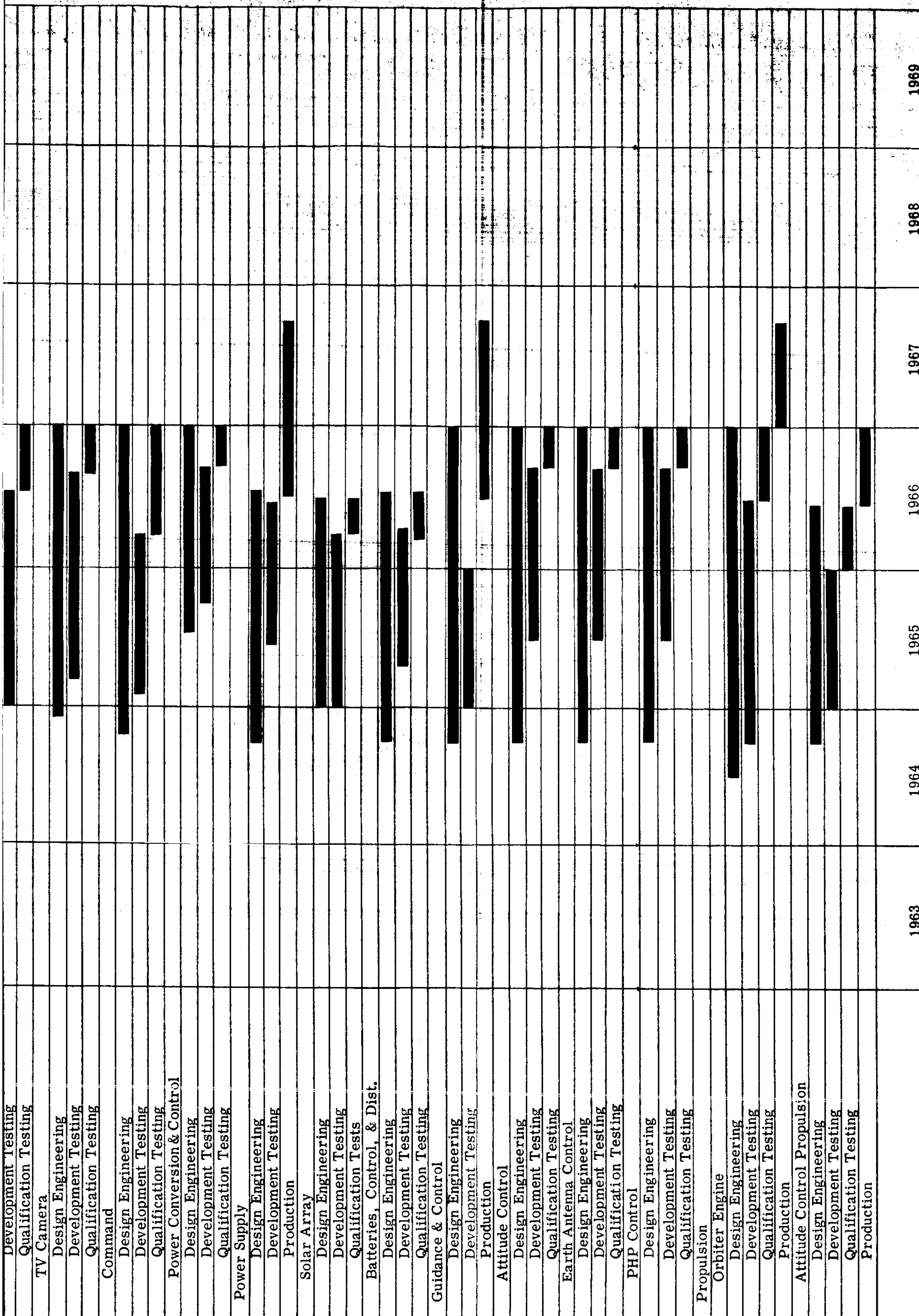


Figure 4.1-3. Orbiter Program Schedule Chart

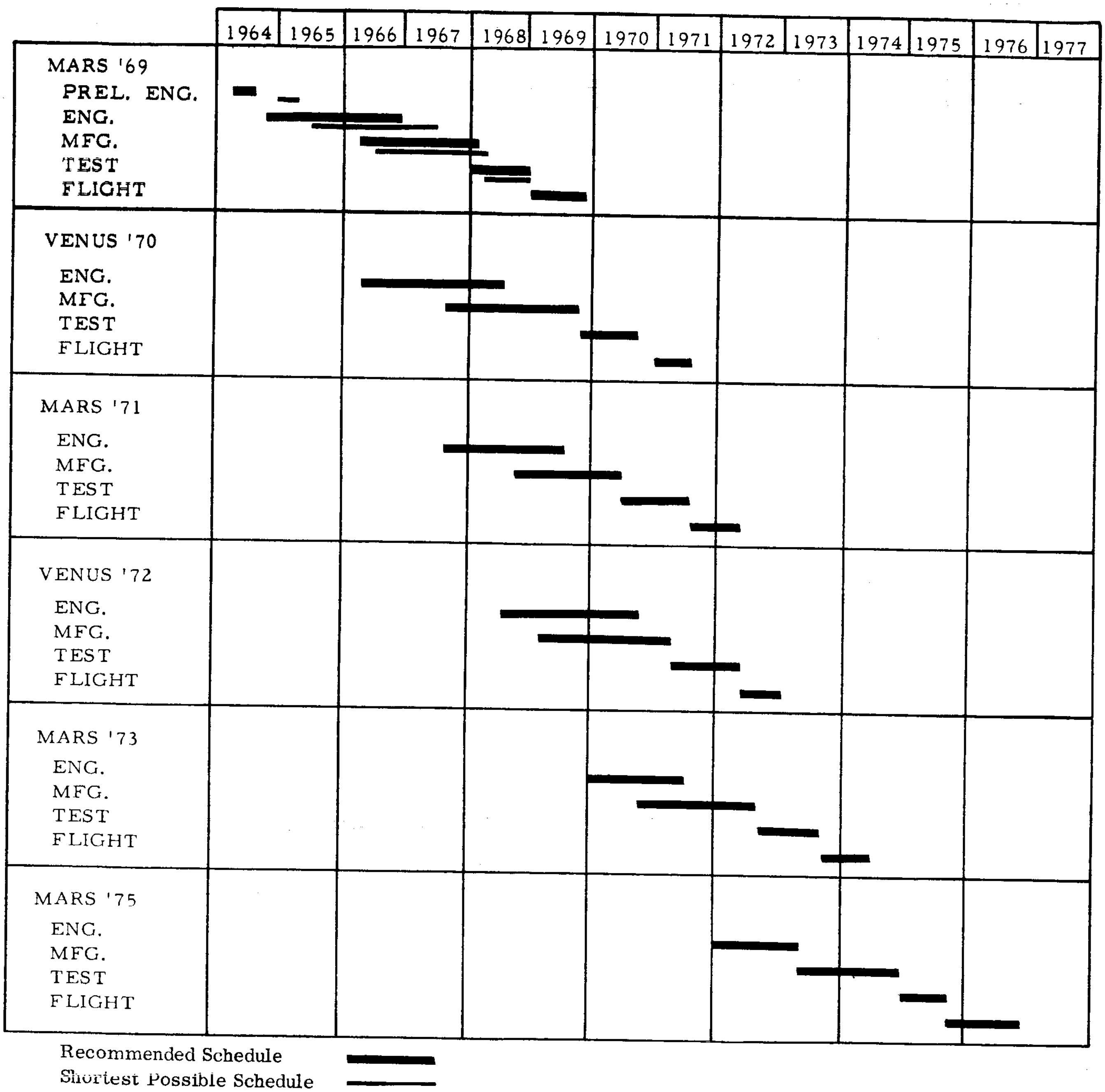


Figure 4.1-4. Voyager Program Summary Schedule

environmental and development testing so that each type of testing builds more and more confidence and limits failures in subsequent testing.

5. The qualification test program is scheduled in two parts, with the components being qualified during a six month program prior to initiation of a twelve month system qualification program. All qualification tests will be completed prior to start of system acceptance testing.

6. System acceptance tests are scheduled for a seven-month cycle. Considerable attention has been given to these tests, which are described in good detail in Section 3.7.8.F

7. The field check-out time is set at four months for each system, based on the experience of other large space programs. Systems will be delivered to the field at one month intervals.

8. It is anticipated that all Voyager components will use high reliability parts; this will add 16-20 weeks to the procurement times.

The schedule shown for Mars 1969 is realistic and attainable. However, if other considerations dictate that the first Voyager mission be delayed, the schedule shown can be moved, as is, to fit the later opportunity. With more time available, the schedule could be lengthened slightly to provide less overlap between development and production, but it does not appear that this would provide much gain.

The minimum time schedule is not recommended because of the larger overlap between the design and production phases with attendant increase in rework required, and the shortening of the test cycles with the attendant decrease in confidence in the equipment reliability.

4.2 VENUS 1967

Designs for a Venus 1967 Voyager were examined early in the Voyager study, and a schedule (Figure 4.2-1) developed for this mission, assuming a start date of January 1, 1964. This early start date appears to be completely unrealistic but, even so, the schedule is marginal and would require delivery of flight systems which were not sufficiently tested to provide a high confidence level.

Study of this schedule leads to a recommendation that a Voyager mission to Venus in 1967 not be attempted.

4.3 LATER MISSIONS

Detailed schedules for later missions were not developed, but the summary schedule shown in Figure 4.1-1 shows the engineering, manufacturing, and test cycles estimated for each of these missions.

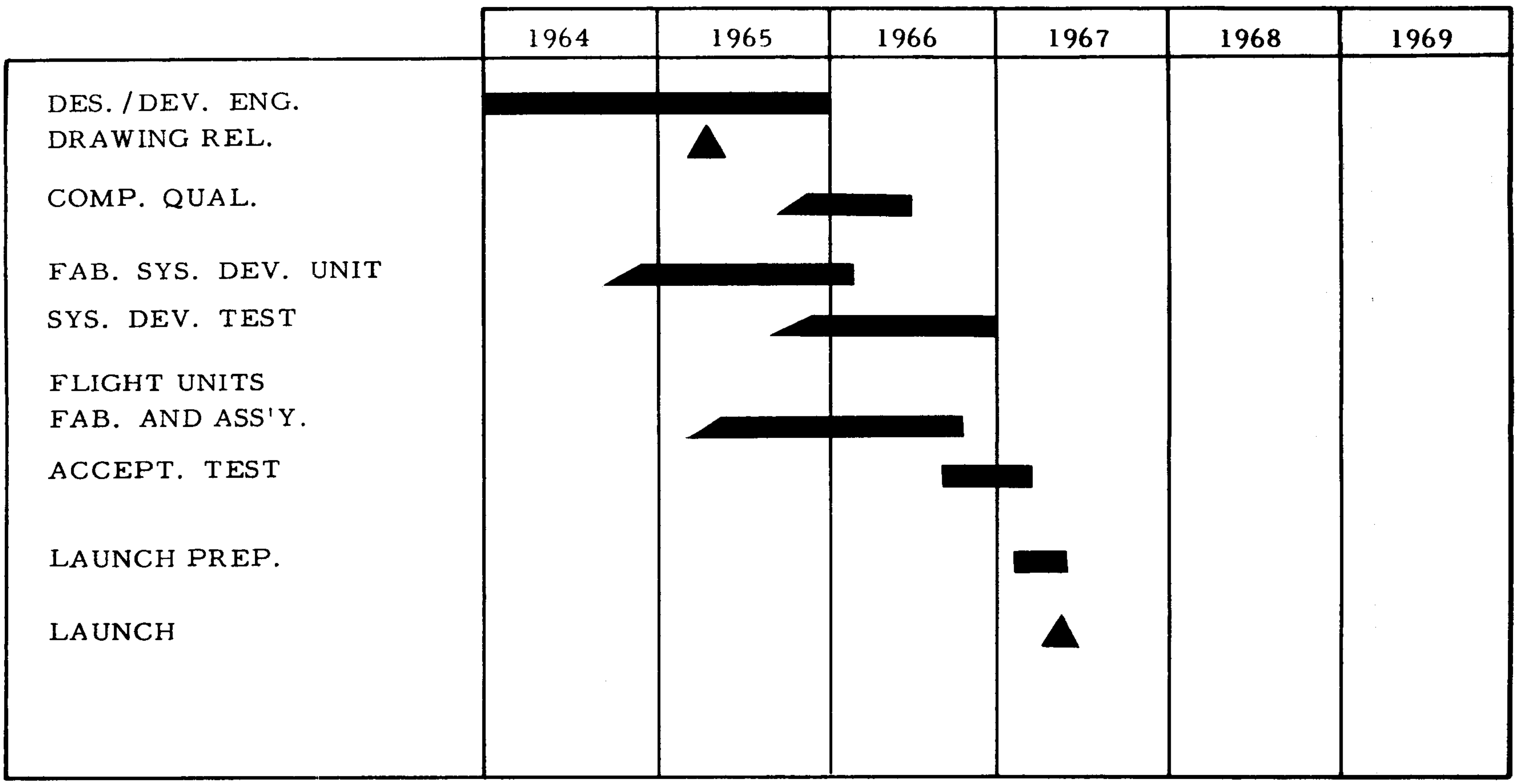


Figure 4.2-1. Venus 1967 Schedule

SECTION 5. PROGRAM COSTS

5.1 SUMMARY

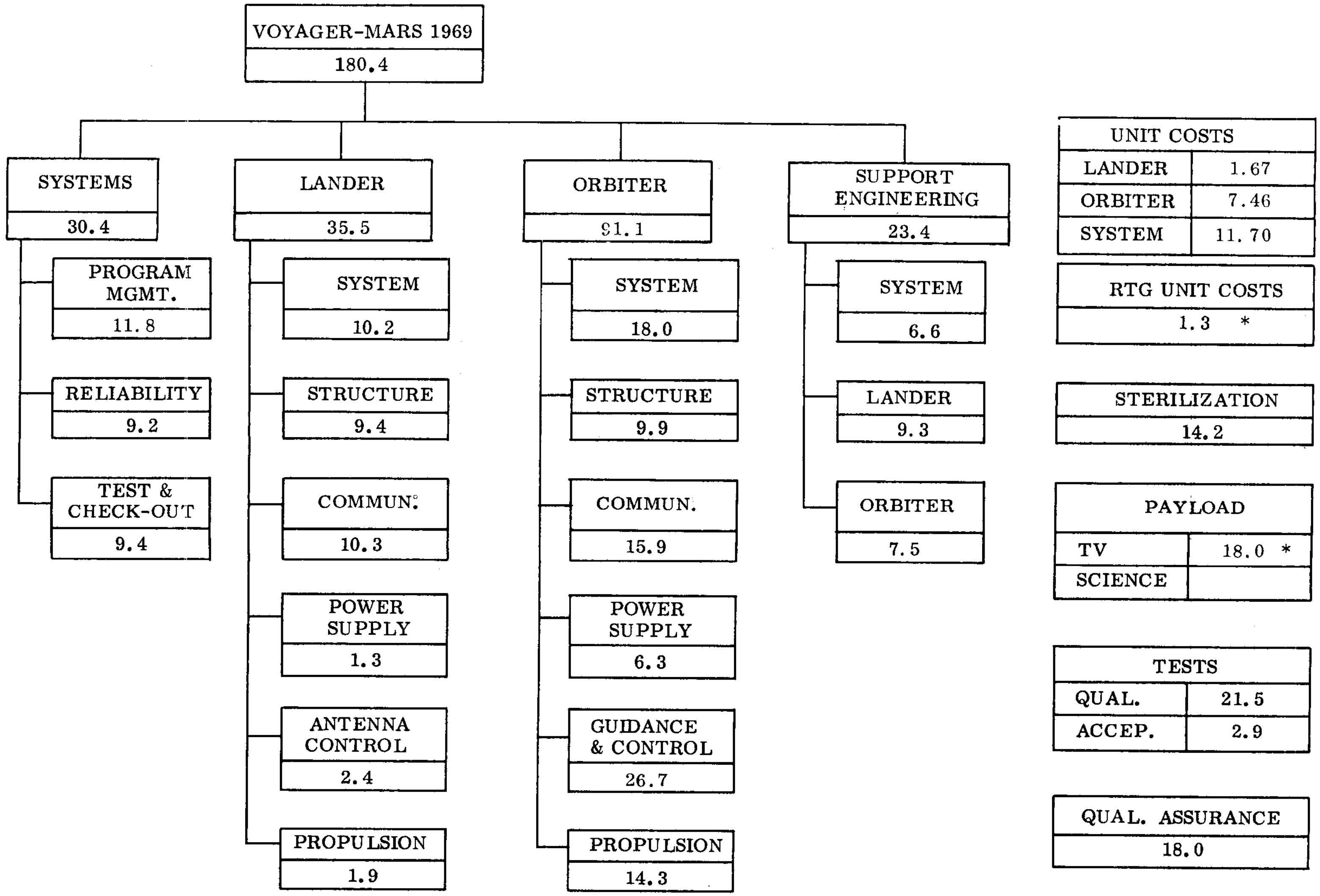
During the course of the Voyager study the system designated as Mars 1969 was specified in good detail, development and test plans written, schedules drawn, and costs estimated for the mission. Figure 5.1-1 shows the costs for this mission down to the subsystem level. Figures 5.1-2, 5.1-3, and 5.1-4 give a cost breakdown for each fiscal year quarters. Section 5.1-5 contains definitions of the cost elements described above.

The costs for Mars 1969 do not include costs for payload (TV or scientific), costs for the RTG power supplies included in the Landers, or any costs after the spacecraft is launched. Included in the costs are all development, manufacture, testing, and field checkout for three complete systems plus a complete set of spare components including one complete Lander as a spare.

Figure 5.1-1 lists several costs which have been broken out for special attention. Except where noted these costs are included in the mission costs also shown on the figure, and are shown separately only because they might be of particular interest.

Figure 5.1-5 shows a summary of the Voyager Total Program Costs. The costs for the later missions were obtained by assuming that all preceding missions were developed according to schedule and then by estimating the costs of the changes required for each new mission; section 5.3 discusses this further.

Figure 5.1-6 shows the expenditure rates by fiscal year for each mission assuming start of hardware design on October 1, 1964, for the Mars 1969 mission. Figure 5.1-7 shows the expenditure rate by fiscal year assuming a start date of July 1, 1965, for the Mars 1969 mission. This late start causes the cost estimate for Mars 1969 to increase to 225 million dollars.



NOTE: 1. COSTS IN MILLIONS OF DOLLARS

2. COSTS SHOWN ABOVE ARE FOR A HARDWARE PROGRAM WITH A START DATE OF 10-1-64. A START DATE OF 7-1-65 WOULD INCREASE THE COSTS TO APPROXIMATELY 225 MILLION.

* NOT INCLUDED IN COSTS

Figure 5.1-1. Voyager Program Costs - Mars 1969

OGRAM SUMMARY COSTS

F/Y 67 1	F/Y 67 2	F/Y 67 3	F/Y 67 4	F/Y 67 Total	F/Y 68 1	F/Y 68 2	F/Y 68 3	F/Y 68 4
738	721	755	734	2 948	752	720	702	608
665	661	605	407	2 338	360	358	349	320
653	583	390	448	2 074	812	666	956	1 343
2 056	1 965	1 750	1 589	7 360	1 924	1 744	1 999	2 271
3 598	2 144	1 516	1 405	8 663	1 026	726	469	383
6 265	2 647	609	154	9 675	140	28		
3 590	2 150	3 918	1 875	11 533	2 596	1 174	547	395
900	887	813	785	3 385	656	787	444	334
14 353	7 828	6 856	4 219	33 256	4 418	2 715	1 460	1 112
639	520	88	79	1 326	141	123		
1 367	872	185	130	2 554	130	117	29	16
2 458	1 943	602	542	5 545	252	198	17	5
521	492	606	551	2 170	631	550	149	140
4 985	3 827	1 481	1 302	11 595	1 154	988	195	161
1 635	1 483	1 312	915	5 345	712	578	422	
		1 194	2 390	3 584	1 194	607		
								296
1 635	1 483	2 506	3 305	8 929	1 906	1 185	422	296
23 029	15 103	12 593	10 415	61 140	9 402	6 632	4 076	3 840

Total	F/Y 69 1	2	3	Total	Program Grand Total
2 782	616	616	507	1 739	11 790
1 379	264	178	90	532	9 259
3 777	1 203	917	290	2 410	9 407
7 938	2 083	1 711	887	4 681	30 456
2 604	376			376	51 857
168					14 234
4 712	202	42		244	17 210
2 221	162			162	7 782
9 705	740	42		782	91 083
264					14 002
292					7 269
472					8 008
1 470					6 188
2 498					35 467
1 712					17 570
1 801					5 385
296	144	28		172	468
3 809	144	28		172	23 423
23 950	2 967	1 781	887	5 635	180 429

Figure 5.1-2. Voyager Mars 1969 Program Summary Costs

	F/Y 65 2	3	4	Total
LANDER SYSTEM				
Design & Development	174	261	307	742
Qualification Testing				
Manufacturing Hardware			10	10
Production Testing	9	9	9	27
	183	270	326	779
STRUCTURE				
Design & Development	355	813	877	2 045
Qualification Testing			9	9
Manufacturing Hardware			1	1
Production Testing			21	21
	355	813	908	2 076
COMMUNICATION				
Design & Development	491	665	747	1 903
Qualification Testing		16	35	51
Manufacturing Hardware				
Production Testing			16	16
	491	681	798	1 970
POWER SUPPLY				
Design & Development	26	148	171	345
Qualification Testing				
Manufacturing Hardware				
Production Testing				
	26	148	171	345
EARTH ANTENNA				
Design & Development	76	174	210	460
Qualification Testing				
Manufacturing Hardware				
Production Testing		9	15	24
	76	183	225	484
PROPULSION				
Design & Development	9	175	191	375
Qualification Testing			9	9
Manufacturing Hardware				
Production Testing				
	9	175	200	384
TOTAL LANDER	1 140	2 270	2 928	6 038

MARS 1969 LANDER DETAIL SUMMARY COSTS

F/Y 66					F/Y 67				
1	2	3	4	Total	1	2	3	4	Total
485	546	939	592	2 562	296	254	43	37	630
		781	1 003	1 784	1 062	624	185	130	2 001
18	27	27	25	97	4	10	30	48	92
639	75	100	86	900	41	26	82	94	243
1 142	648	1 847	1 706	5 343	1 403	914	340	309	2 966
853	833	158	138	1 982	114	96	21	21	252
46	69	219	197	531	96	82			178
341	405	147	151	1 044	362	356	327	312	1 357
168	178	159	159	664	183	160	201	188	732
1 408	1 485	683	645	4 221	755	694	549	521	2 519
238	213	170	130	751	64	64	12		140
254	535	324	216	1 329	160	145			305
14	23	221	337	595	1 642	994	141	138	2 915
146	144	212	224	726	216	220	228	220	884
652	915	927	907	3 401	2 082	1 423	381	358	4 244
149	158	96	82	485	57	30			87
	65	17	16	98	9				9
	8	18	26	52	59	70	52		181
		9		9	9	17	24		50
149	231	140	124	644	134	117	76		327
123	94	94	94	405	78	64			142
	75	100	30	205	26	21			47
5	12	12	21	50	307	427	14	14	762
11	30	24	43	108	21	28	30	14	103
139	211	230	188	768	442	540	44	28	1 054
50	50	147	110	357	30	12	12	21	75
16	35	184	172	407	14				14
	5	68	69	142	84	86	38	30	238
		23	30	53	41	41	41	35	158
66	90	422	381	959	169	139	91	86	485
3 556	3 580	4 249	3 951	15 336	4 985	3 827	1 481	1 302	11 595

F/Y 68					F/Y 69				Program Grand Total
1	2	3	4	Total	1	2	3	Total	
28	28			56					3 990
130	117	29	16	292					4 077
53	42	17	5	117					316
186	186	149	140	661					1 831
397	373	195	161	1 126					10 214
21	12			33					4 312
									718
94	89			183					2 585
194	146			340					1 757
309	247			556					9 372
71	71			142					2 936
									1 685
68	158			126					3 636
207	190			397					2 023
346	319			665					10 280
									917
									107
									233
									59
									1 316
									1 007
									252
16				16					828
16				16					251
32				32					2 338
21	12			33					840
									430
21	9			30					410
28	28			56					267
70	49			119					1 947
1 154	988	195	161	2 498					35 467

Figure 5.1-3. Mars 1969 Lander Detail Summary Costs

	F/Y 65			
	2	3	4	Total
ORBITER SYSTEM				
Design & Development	62	89	160	311
Qualification Testing				
Manufacturing Hardware			5	5
Production Testing				
	62	89	165	316
STRUCTURE				
Design & Development	316	539	649	1 504
Qualification Testing				
Manufacturing Hardware			12	12
Production Testing		21	37	58
	316	560	698	1 574
COMMUNICATION				
Design & Development	441	842	1 423	2 706
Qualification Testing				
Manufacturing Hardware				
Production Testing		14	26	40
	441	856	1 449	2 746
POWER SUPPLY				
Design & Development	28	149	410	587
Qualification Testing				
Manufacturing Hardware		5	12	17
Production Testing			9	9
	28	154	431	613
GUIDANCE & CONTROL				
Design & Development	457	1 614	2 419	4 490
Qualification Testing		26	184	210
Manufacturing Hardware			12	12
Production Testing			26	26
	457	1 640	2 641	4 738
PROPULSION				
Design & Development	125	2 458	170	2 753
Qualification Testing				
Manufacturing Hardware				
Production Testing		14	37	51
	125	2 472	207	2 804
TOTAL ORBITER	1 429	5 771	5 591	12 791

MARS 1969 ORBITER DETAIL SUMMARY COSTS

F/Y 66					F/Y 67				
1	2	3	4	Total	1	2	3	4	Total
367	1 819	4 293	2 745	9 224	1 571	460	66	66	2 163
	30	89	244	363	969	2 056	609	154	3 758
5	14	14	21	54	89	154	94	118	455
9	14	16	16	55	16	23	23	28	90
381	1 877	4 412	3 026	9 696	2 615	2 693	792	366	6 466
748	732	575	467	2 522	383	332	282	218	1 215
35	71	76	69	251	59	31			90
21	26	35	75	157	231	316	382	324	1 153
68	89	144	168	469	258	276	218	194	946
872	918	830	779	3 399	931	955	782	736	3 404
1 317	558	498	478	2 851	418	310	244	244	1 216
83	377	374	336	1 170	193	212			405
14	26	35	66	141	1 413	158	1 463	269	3 303
26	41	46	82	195	152	152	227	220	751
1 440	1 002	953	962	4 357	2 176	832	1 934	733	5 675
286	274	205	110	875	52	52	43	35	182
65	134	120	116	435	78	69			147
21	26	35	100	182	693	875	526	382	2 476
26	37	52	102	217	147	140	89	78	454
398	471	412	428	1 709	970	1 136	658	495	3 259
2 201	2 227	2 001	1 776	8 205	1 094	944	826	787	3 651
323	622	659	233	1 837	272	244			516
21	26	35	35	117	419	594	1525	761	3 299
63	106	189	248	606	251	227	206	215	899
2 608	2 981	2 884	2 292	10 765	2 036	2 009	2 557	1 763	8 365
2 475	206	1 409	96	4 186	80	46	55	55	236
	9	70	46	125	4 724	35			4 759
	5	5	14	24	745	53	28	21	847
64	66	82	76	288	76	60	50	50	245
2 539	286	1 566	232	4 623	5 625	203	133	126	6 087
8 238	7 535	11 057	7 719	34 549	14 353	7 828	6 856	4 219	33 256

F/Y 68					F/Y 69				Grand
1	2	3	4	Total	1	2	3	Total	Total
68	52			120					11 818
140	28			168					4 289
141	117	64	26	348	12			12	874
28	381	158	174	741	100			100	986
377	578	222	200	1 377	112			112	17 967
172	92	64	64	392	57			57	5 690
									341
204	149	116	68	537	64	14		78	1 937
183	118	80	52	433	23			23	1 929
559	359	260	184	1 362	144	14		158	9 897
188	132	126	106	552	108			108	7 433
									1 575
1 439	158	138	96	1 831	50	28		78	5 353
202	117	80	52	451				23	1 460
1 829	407	344	254	2 834	181	28		209	15 821
43	28	37	28	136	37			37	1 817
									582
212	100	64	50	426	46			46	3 147
50	30	28	5	113					793
305	158	129	83	675	83			83	6 339
500	367	201	144	1 212	146			146	17 704
									2 563
586	158	151	146	1 041	21			21	4 490
158	100	75	35	368	16			16	1 915
1 244	625	427	325	2 621	183			183	26 672
55	55	41	41	192	28			28	7 395
									4 884
14	492	14	9	529	9			9	1 409
35	41	23	16	115					699
104	588	78	66	836	37			37	14 387
4 418	2 715	1 460	1 112	9 705	740	42		782	91 083

Figure 5.1-4. Mars 1969 Orbiter Detail Summary Costs

VOYAGER MARS 1969 PR

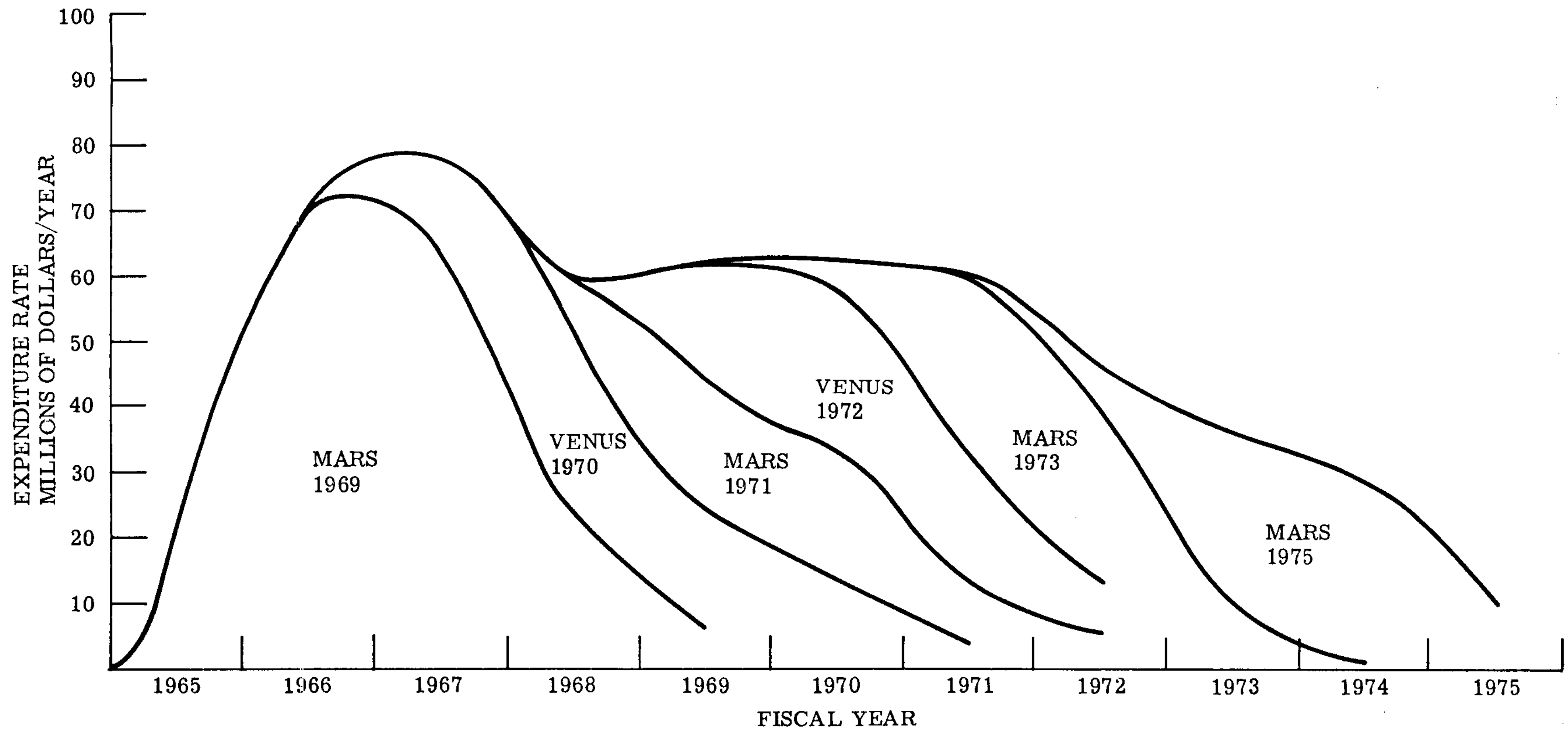
	F/Y 65 2	3	4	Total	F/Y 66 1	2	3	4	Total
VOYAGER SYSTEM									
Program Management	311	428	568	1 307	755	758	777	724	3 014
Reliability	194	858	956	2 008	868	801	673	660	3 002
System Test & C/O				—	21	96	531	498	1 146
	505	1 286	1 524	3 315	1 644	1 655	1 981	1 882	7 162
ORBITER SYSTEM									
Design & Development	1 429	5 691	5 231	12 351	7 354	5 816	8 981	5 672	27 863
Qualification Testing		26	184	210	506	1 243	1 388	1 044	4 181
Manufacturing Hardware		5	41	46	82	123	159	311	675
Production Testing		49	135	184	256	353	529	692	1 830
	1 429	5 771	5 591	12 791	8 238	7 535	11 057	7 719	34 549
LANDER SYSTEM									
Design & Development	1 131	2 236	2 503	5 870	1 898	1 894	1 604	1 146	6 542
Qualification Testing		16	53	69	316	779	1 625	1 634	4 354
Manufacturing Hardware			11	11	378	480	493	629	1 980
Production Testing	9	18	61	88	964	427	527	542	2 460
	1 140	2 270	2 628	6 038	3 556	3 580	4 249	3 951	15 336
GSE									
Design & Development	175	841	1 860	2 876	1 522	2 922	1 544	1 649	7 637
Manufacturing Hardware									
Operational I & C/O									
	175	841	1 860	2 876	1 522	2 922	1 544	1 649	7 637
TOTAL PROGRAM	3 249	10 168	11 603	25 020	14 960	15 692	18 831	15 201	64 684

VOYAGER TOTAL PROGRAM COSTS*

Major Task	MARS					VENUS			PROGRAM
	69	71	73	75	Total	70	72	Total	Total
Development									
Lander	21.3	6.3	4.6	4.3	36.5	12.7	13.7	26.4	62.9
Orbiter	66.0	9.9	19.6	7.5	103.0	30.0	12.0	42.0	145.0
Total	87.3	16.2	24.2	11.8	139.5	42.7	25.7	68.4	207.9
Production									
Lander	14.2	12.5	12.5	12.5	51.7	4.5	12.7	17.2	68.9
Orbiter	25.0	21.3	18.5	15.4	80.2	21.3	21.3	42.6	122.8
Total	39.2	33.8	31.0	27.9	131.9	25.8	34.0	59.8	191.7
Systems	30.5	10.2	10.8	10.7	62.2	12.5	10.0	22.5	84.7
Support Equipment	23.4	1.5	1.5	0.7	27.1	2.2	0.8	3.0	30.1
Total	180.4	61.7	67.5	51.1	360.7	83.2	70.5	153.7	514.4

*The MARS 1969 costs as shown above are for a hardware program start date of 10/1/64. A start date of 7/1/65 would result in total MARS 1969 costs of approximately $225 Million.

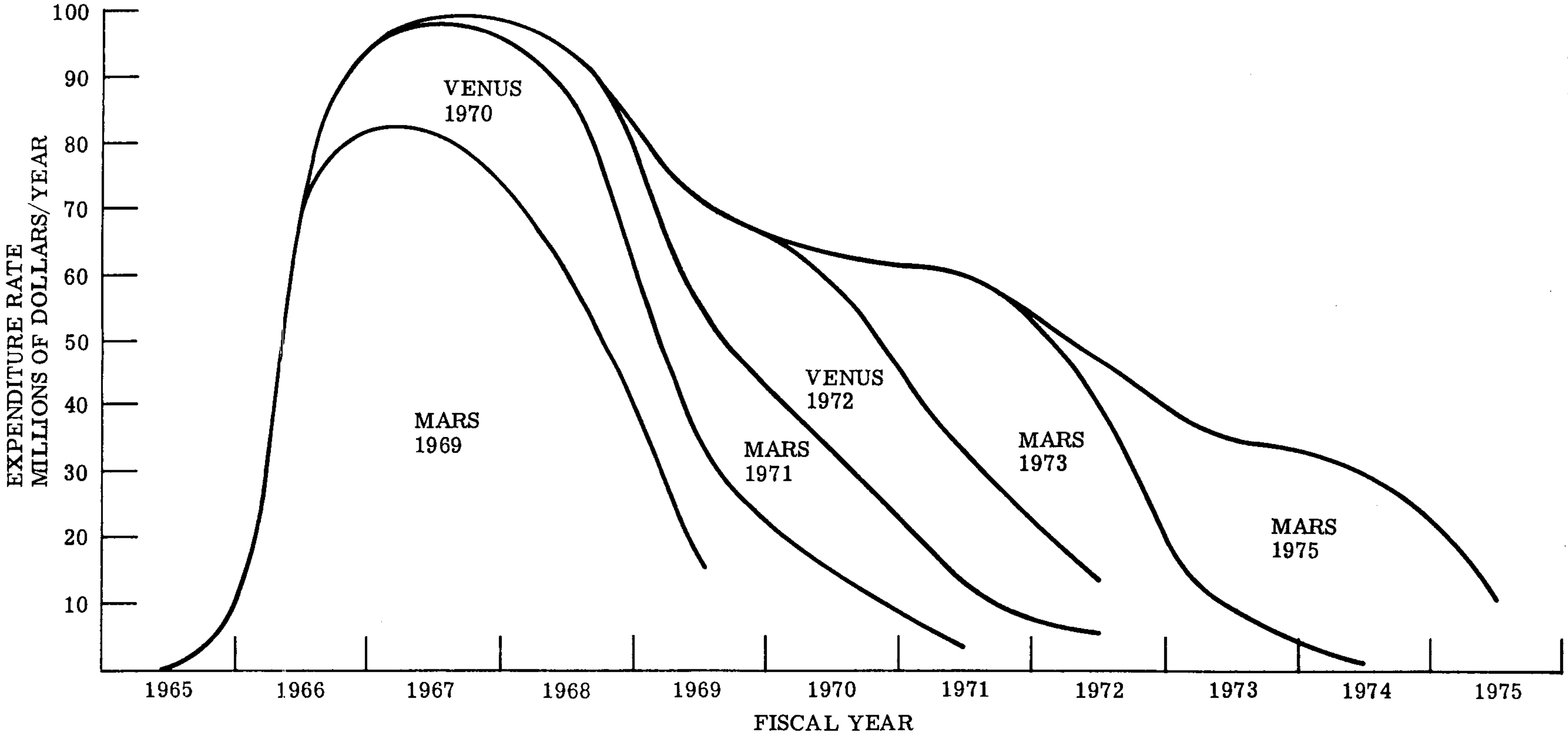
Figure 5.1-5. Voyager Total Program Costs



25	67	77	59	62	62	60	46	36	29	10

Total Costs by Fiscal Year in Millions of Dollars.

Figure 5.1-6. Voyager Spacecraft Cost Summary (Start 10/1/64)



	70	96	80	66	62	60	46	36	29	10

Total Costs by Fiscal Year in Millions of Dollars.

Figure 5.1-7. Voyager Spacecraft Cost Summary (Start 7/1/65)

5.2 MARS 1969 COST ELEMENTS

Figure 5.1-1 shows the costs for the Mars 1969 mission down to the subsystem level. Figures 5.2-1, 5.2-2, 5.2-3, and 5.2-4 give definitions of the elements contained in each of these costs.

Figure 5.1-1 also shows several cost items broken out for special attention. These cost items are defined and discussed briefly below.

5.2.1 UNIT COSTS

A. Lander

This cost includes all the procurement, fabrication, assembly, inspection, test, quality assurance, production engineering and program management costs required to produce an additional Lander System immediately following the production of the seven (7) flight units. It assumes the use of the same tooling, fixtures, test equipment, etc., as used for the original flight units. It also includes the acceptance test effort which is unique to the Lander System and the additional production engineering support and incremental program management costs.

B. Orbiter

This cost includes all the procurement, fabrication, assembly, inspection, test, quality assurance, production engineering and program management costs required to produce an additional Orbiter System immediately following the production of the three (3) flight units and one (1) set of spares. It assumes the use of the same tooling, fixtures, test equipment, etc., as used for the production of the original flight units. It also includes the acceptance test effort which is unique to the Orbiter System and the additional production engineering support and incremental program management costs.

C. System

This cost includes the above unit costs of two (2) Landers and one (1) Orbiter and the acceptance test effort unique to the complete system.

All of the above costs are for units F.O.B. Philadelphia. They do not include any effort beyond system acceptance test.

5.2.2 STERILIZATION

This cost includes a 20 per cent increase in Lander hardware development and production costs, an increase of approximately four (4) million dollars in ground support equipment costs, and an additional three (3) million dollars for a sterilization management and control effort and a parts test and evaluation program to determine the sterilization effects on reliability.

5.2.3 PAYLOAD TV

This cost covers the development and production of the TV subsystems (including optics) required for the three (3) Voyager systems, including spares. This would provide TV subsystems for the three (3) Orbiters with a set of spares and for the six (6) Landers with an additional set for the spare Lander.

5.2.4 PAYLOAD SCIENCE

Detailed estimates of payload science costs have not been made. Therefore, these have been excluded from all cost figures shown.

SUB-TASKS	ACTIVITIES	HARDWARE REQUIRED	COST DESCRIPTIONS
Program Management	Plans and Schedules	None	Includes all manpower required to perform these activities throughout the MARS 1969 Program.
	Program Control & Measurements	None	
	Reports & Documents	None	
	System Engineering & Integration	None	
	Contract Administration	None	
	Finance	None	
	Sterilization Management & Control	None	
Reliability	Parts Evaluation	Piece parts	
	Sterilization Effects	Piece parts	
	Parts Acceptance Test	All production piece parts	
	Analysis & Apportionment	None	
Systems Test & Checkout	System Development Testing	None costed - use Lander and Orbiter system development hardware	Includes all test manpower, special equipment & facility costs associated with Voyager system testing.
	System Qualification Testing	None costed - use Lander & Orbiter system qualification hardware	Includes all test manpower, special equipment & facility costs associated with Voyager system testing.
	System Acceptance Testing	Costed under Lander and Orbiter system	Includes all test manpower, special equipment & facility costs associated with Voyager system testing.
	Field Test & Checkout	None	Field equipment costs part of Support Equipment task.

Figure 5.2-1. Task Definitions for Mars 1969 Cost Elements - Voyager Systems

Batteries Regulation Control & Distribution	Qualification	(2) Sets of Qualification Hardware	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of man-power and equipment for qualification testing.
	Production-Mfg.	(3) Sets of Power Supply Subsystem flight hardware and (1) set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(3) Sets of Power Supply Subsystem flight hardware and (1) set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of orbiter system.
Guidance & Control Sub-System Consists of: Attitude Control Antenna Control PHP Control Thrust Vector Control Logic and Storage Power Conversion	Design & Development	Development Hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of man-power and equipment for qualification testing.
	Production-Mfg.	(3) Sets of Guidance & Control Subsystem flight hardware and (1) set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(3) Sets of Guidance & Control Subsystem flight hardware and (1) set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Propulsion-Sub-System Consists of: Main engine Orbit trim Attitude control	Design & Development	Development Hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware or equivalent quantities for rocket qualification	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the qualification testing costs of man-power and equipment.
	Production-Mfg.	(3) Sets of Propulsion Subsystem flight hardware and (1) set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(3) Sets of Propulsion Subsystem flight hardware and (1) set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.

Figure 5.2-2. Task Definitions for Mars 1969 Cost Elements - Orbiter System

TASK DEFINITIONS FOR MARS 1969 COST ELEMENTS-ORBITER SYSTEM

Sub-Tasks	Activities	Hardware Required	Cost Descriptions
System	Design & Development	(3) System prototypes-electrical, structural, thermal	Includes all orbiter system design and analysis and the fabrication, assembly and test costs of three orbiter system prototypes (less landers)
	Qualification	(1) System qualification unit	Includes all manufacturing and quality control costs to procure, fabricate, assemble & acceptance test this unit. Also includes test costs for qualifying orbiter less lander.
	Production-Final Assembly	(3) Flight Units	Includes all costs for final assembly of flight units including tooling.
	Production-Quality Control & Test (QC&T)	(3) Flight Units	Includes all costs for inspection & acceptance test of flight units during & after final assembly.
Structure-Sub-System Consists of: Spacecraft Structure PHP Structure Antenna Structure Antenna & PHP Drives & Deployment Separation Mechanisms	Design & Development	Development hardware	Includes all the design & analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of qualification hardware (not including structure) or equivalent quantities for pyrotechnics	Includes all mfg. & Q. C. costs associated with the hardware to be qualified and the costs of manpower and special equipment for qualification testing.
	Production-Mfg.	(3) Sets of structure flight hardware and one (1) set of spares (not including basic structures)	Includes all mfg. costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(3) Sets of structure flight hardware and one (1) set of spares (not including basic structures)	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Communication-Sub-System Consists of: Earth Link Lander Link Data Storage & Processing Command Power Conversion	Design & Development	Development Hardware	Includes all the design & analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of man-power and equipment for qualification [illegible]
	Production-Mfg.	(3) Sets of Communication Subsystem flight hardware and (1) set of spares	Includes [illegible] the ma[illegible] bly and test requi[illegible] hardware used. [illegible]
Consists of: Solar array	Production-QC&T	(3) Sets of Communicatio[illegible] Subsystem flight hardwa[illegible]	

Power Supply-Sub-System Consists Of: Batteries Regulation, control and distribution	Design & Development	Development Hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly, and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware or equivalent for batteries	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production - Mfg.	(6) Sets of Power Supply Subsystem Hardware and (1) set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production - QC&T	(6) Sets of Power Supply Subsystem Hardware and (1) set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Antenna Control-Sub-System	Design & Development	Development Hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg.	(6) Sets of Antenna Control Subsystem flight hardware and (1) set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(6) Sets of Antenna Control Subsystem flight hardware and (1) set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Propulsion-Sub-System Consists Of: Rocket Spin-de-Spin	Design & Development	Development Hardware	Includes all the design and analysis associated with this sub-system and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) or equivalent sets of qualification hardware	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of manpower and equipment for qualification testing.
	Production-Mfg.	(6) Sets of Propulsion Subsystem flight hardware and 1 (1) set of spares	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(6) Sets of Propulsion Subsystem flight hardware and 1 (1) set of spares	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.

Figure 5.2-3. Task Definitions for Mars 1969 Cost Elements - Lander System

TASK DEFINITIONS FOR MARS 1969 COST ELEMENTS-LANDER SYSTEM

Sub-Task	Activities	Hardware Required	Cost Description
System	Design & Development	Prototype hardware for lander system development test	Includes fabrication, assembly and test costs
	Qualification	(2) lander system qualification units	Includes all materials, fabrication assembly and acceptance test of both lander units. Also includes the qualification testing unique to the lander system on one (1) unit.
	Production-Final Assembly	(6) Flight Units and (1) spare Lander	Includes all costs for final assembly of flight units including tooling.
	Production-QC&T	(6) Flight Units and (1) spare Lander	Includes all costs for inspection and acceptance test of flight units during and after final assembly.
Structure-Sub-System Consists Of: Shield & Structure Environmental Control Retardation Orientation Deployment Adaptor	Design & Development	Development Hardware	Includes all the design and analysis associated with this sub-system and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware (not including structure	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of man-power and equipment for qualification testing.
	Production-Mfg.	(6) Sets of Structure Subsystem Hardware and (1) set of spares (not including structure)	Includes all manufacturing costs associated with this hardware up to but not including final assembly of system.
	Production-QC&T	(6) Sets of Structure Subsystem Hardware and (1) set of spares (not including structure)	Includes all inspection, test and quality assurance associated with this hardware up to but not including final assembly of system.
Communications-Sub-System Consists Of: Orbit Link Earth Link Data Storage & Processing Command Power Conversion	Design & Development	Development Hardware	Includes all the design and analysis associated with this subsystem and the materials, fabrication, assembly and test required for the development hardware used.
	Qualification	(2) Sets of Qualification Hardware	Includes all Mfg. & Q. C. costs associated with the hardware to be qualified and the costs of man-power and equipment for qualification testing.
	Production-Mfg.	(6) Sets of Communication Subsystem flight hardware	Includes all manufacturing costs associated with this [illegible]

SUB-TASK	ACTIVITIES	HARDWARE REQUIRED	COST DESCRIPTIONS
Spacecraft	Design, Development & Manufacture	Hardware unique to support of Voyager System	Includes all the costs associated with the design development and manufacture of this equipment.
Orbiter	Design, Development & Manufacture	Hardware unique to support of Orbiter System	Includes all the costs associated with the design, development and manufacture of this equipment.
Lander	Design, Development & Manufacture	Hardware unique to support of Lander System	Includes all the costs associated with the design, development and manufacture of this equipment.

Figure 5.2-4. Task Definitions for Mars 1969 Cost Elements - Support Equipment Systems

5.2.5 QUALIFICATION

These costs include the hardware, special test equipment and test effort required to conduct qualification tests on all the components of both the Lander and Orbiter. It also includes the hardware, test equipment and testing effort to conduct qualification tests of the Lander System, Orbiter System, and Voyager System.

5.2.6 ACCEPTANCE

This cost includes the acceptance tests conducted on the three (3) Voyager Systems prior to delivery. This is the testing unique to the Orbiter with the Landers mated. This cost covers the test effort only and not the cost of the Lander and Orbiter hardware as these are included in the Lander and Orbiter system costs.

5.2.7 QUALITY ASSURANCE

This cost includes the reliability program and all the other quality control functions involved in assuring the high quality required for Voyager in reliability and performance from piece parts to complete system.

5.3 LATER MISSION COST ELEMENTS

In arriving at the cost estimates for each of the missions after Mars 1969, consideration was given to the status of developments initiated for all earlier missions, the system design for the particular mission requiring changes to be made to the basic system, the probable changes to be imposed because of knowledge gained in earlier missions, and the schedules imposed. Each subsystem was then examined, changes noted, and estimates of the costs of development, production, systems and GSE were made. The costs obtained were tested by comparing them against other space program costs, and seem to be reasonably accurate.

The development costs shown in Figure 5.1-5 include the total labor and material costs for the design and development effort for each mission, including qualification or requalification, where required. This task also includes production engineering support for the flight units.

The production costs shown include the total manufacturing, quality control and acceptance test effort required to produce the flight hardware and spares for each mission.

The systems costs include the additional program management, reliability engineering and testing, system acceptance test, and field check-out costs associated with the particular mission.

The support equipment costs shown are for the design, production and check-out of additional support equipment or modification of existing support equipment, as required, for the particular mission.

Discussed below are some of the items taken into consideration in arriving at the cost estimate for each mission.

5.3.1 VENUS 1970

It was assumed that the results from a Venus 1967 Mariner "B" mission would be available prior to start of development for the Venus 1970 Voyager. The Lander was assumed to be a new development which would make use of the basic Mariner capsule design as well as the components developed for the Mars 1969 mission. A complete qualification program for the Lander system would be required. The Orbiter would require extensive modifications including new fuel tanks with subsequent engine and structural requalifications, the addition of a radar mapper, repackaging of the payload, modifications of the attitude control to accommodate the Venus parameters, a new adapter to carry a single small Lander, and a major redesign for thermal control. Partial system requalification would be required.

The production costs for the Lander are less than for Mars 1969 because of the small size of the Lander and because only four units are required. The Orbiter production costs were based on the Mars 1969 unit costs.

5.3.2 MARS 1971

The major differences in equipment required for this mission are caused by the growth of the Lander to 2,000 pounds. This requires the development of a new section to be added to the Mars 1969 Lander, repackaging of the Lander payload, a new rotating bulkhead, modified deployment mechanisms, a larger retardation subsystem, and a new interface section between the Landers and the Orbiter. The Orbiter will have some repackaging of the Planet Horizontal Package, a 15% larger solar array, and a 60% fuel load with consequent slosh problems. The structure must be tested with the heavy Landers.

The production costs were based on the Mars 1969 unit costs with some additional production tooling provided.

5.3.3 VENUS 1972

The scientific mission for Venus 1972 is to obtain more detailed information about the atmosphere and surface conditions. A new 2600 pound Lander will be required. This will be essentially a new development, although components developed for previous missions can be used. Complete qualification and development tests must be conducted. The Orbiter is modified by the removal of the radar mapper, repackaging of the PHP with new thermal controls, and a new interface section for the single large Lander. Structural tests must be conducted with the Lander.

The production costs for the Lander are higher because of new tooling required for the new design, as well as the increased size of the Lander. The Orbiter production costs are the same as for Venus 1970.

5.3.4 MARS 1973

The Mars 1973 Lander is very similar to that required for Mars 1971. The direct communication link with Earth will be the prime mode of communication, and modifications of the antenna will be required. Knowledge gained in earlier missions will undoubtedly require some equipment repackaging and modification. The Orbiter is changed considerably because of its modified scientific mission. The PHP is removed and new fuel tanks developed. This requires engine requalification and structural testing. Since the Orbiter will be placed in a low orbit, it must be sterilized.

The Lander production costs are the same as Mars 1971. The Orbiter production costs are decreased by the removal of equipment from the 1971 configuration but have been increased by 20% to allow for the cost of sterilization.

5.3.5 MARS 1975

The development costs will be incurred primarily because of changes required by knowledge gained in earlier missions.

The production costs of the Lander are unchanged from Mars 1973. The Orbiter production costs are reduced because of the elimination of the requirement for sterilization of the Orbiter.