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FINAL REPORT
VOYAGER SPACECRAFT
PHASE B, TASK D

VOLUME II (BOOK 5 OF 5)

SYSTEM DESCRIPTION

PREPARED FOR

GEORGE C. MARSHALL SPACE FLIGHT CENTER

UNDER MSFC CONTRACT NO. NAS8-22603

GENERAL ELECTRIC
MISSILE AND SPACE DIVISION
Valley Forge Space Technology Center
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VOLUME SUMMARY

The Voyager Phase B, Task D Final Report is contained in four volumes. The volume numbers and titles are as follows:

Volume I	Summary
Volume II	System Description
Book 1	Guidelines and Study Approach, System Functional Description
Book 2	Telecommunication
Book 3	Guidance and Control Computer and Sequencer Power Subsystem Electrical System
Book 4	Engineering Mechanics Propulsion Planet Scan Platform
Book 5	Design Standards Operational Support Equipment Mission Dependent Equipment
Volume III	Implementation Plan
Volume IV	Engineering Tasks
Book 1	Effect of Capsule RTG's on Spacecraft
Book 2	Applicability of Apollo Checkout Equipment
Book 3	Central Computer
Book 4	Mars Atmosphere Definition
Book 5	Photo-Imaging

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VOYAGER TASK D
VOLUME II
SYSTEM DESCRIPTION

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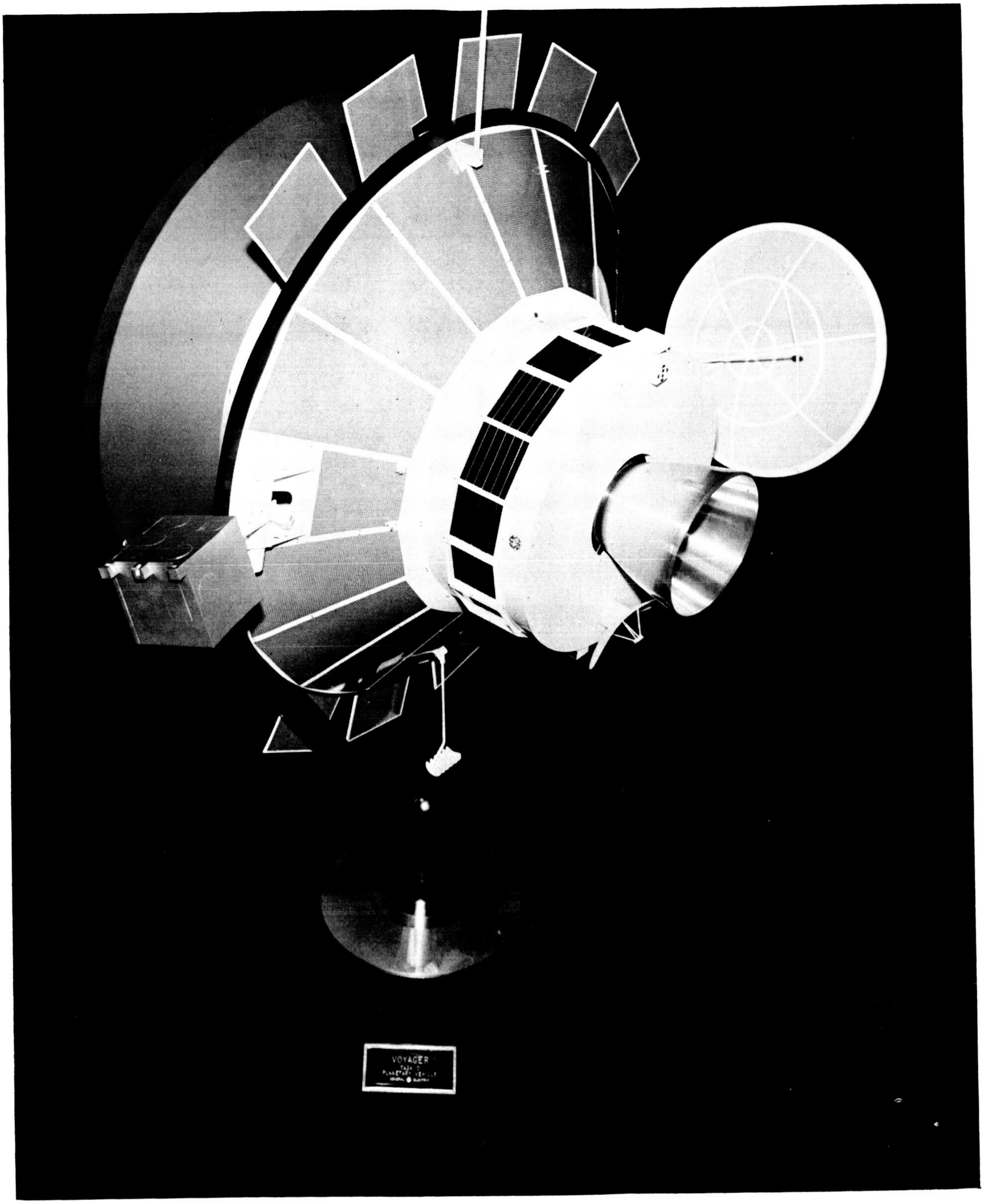

Voyager Planetary Vehicle (Task D)

VOYAGER TASK D
Volume II
PREFACE

This volume describes the design of the Voyager Spacecraft System, the Operational Support Equipment requirements, and the Mission Dependent Equipment requirements resulting from the system update study.

The mission concept for Voyager has not changed substantially since the previous Phase B, Task B study in late 1965. The Saturn V Launch Vehicle is used to inject two identical planetary vehicles on a Mars trajectory. Each planetary vehicle consists of a flight spacecraft and a flight capsule and, after separation from the Saturn V, the two vehicles provide complete mission redundancy. The flight spacecraft serves as a bus to deliver the flight capsule into Mars orbit from which it subsequently descends and soft lands to carry out surface experiments. The flight spacecraft then carries out an orbiting science mission for periods ranging from six months for early missions to two years for subsequent missions.

The flight spacecraft developed in this system update is shown in the illustration on the page opposite. This design is described in detail in this volume which is organized in the following major sections:

Section	Subject	Identification No.
I	Guidelines and Study Approach	VOY-D-100
II	System Functional Description and Analysis	VOY-D-200
III	Subsystem Functional Description and Analysis	VOY-D-300
IV	Design Standards	VOY-D-400
V	Operational Support Equipment	VOY-D-500
VI	Mission Dependent Equipment	VOY-D-600

Section I describes the study approach and discusses major constraints and guidelines that were imposed, with emphasis on requirements or guidelines which have changed since the last Voyager System design study.

Section II is a system level description of the resulting spacecraft design and its interfaces with other systems. Major system analyses and trade studies, such as trajectory and orbit selection, are covered.

Section III describes the baseline design of each subsystem, with discussion of alternates that were considered in arriving at the selected design.

Section IV covers some limited areas of design standards to be applied to the Voyager spacecraft.

Section V is an analysis of Operational Support Equipment (OSE) requirements and an evaluation of a number of OSE concepts with selection of a preferred approach.

Section VI analyzes the space flight operation together with the current and planned capabilities of the deep space network to define probable requirements for mission dependent hardware and software to support the mission.

SECTION IV
VOY-D-400
DESIGN STANDARDS

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VOY-D-400
DESIGN STANDARDS

1. INTRODUCTION

During the system update, some effort was applied in the area of design standards for Voyager. This is a critical activity in that adequate design standards must be available very early in the design cycle if they are to be effective, and if the reliability and economic benefits associated with their use are to be realized.

Specifically, the standardized approach to packaging of the electronic equipment was reviewed and updated; MSFC preferred parts and materials lists were reviewed for adequacy for Voyager; and development of standard design and test techniques for microelectronic devices was pursued.

2. SPACECRAFT ELECTRONIC EQUIPMENT PACKAGING

2.1. APPLICABLE DOCUMENTS

2.1.1. Government

MIL-STD-275A	Printed Wiring for Electronic Equipment
MIL-C-26482	Connectors, Electric, Circular Miniature Quick Disconnect
MIL-W-16878	Wire, Electrical, Insulated, High Temperature
MIL-S-7742	Screw Threads, Standard Aeronautical

2.1.2. General Electric

S-30000	MSD Design Requirements for Electronic Modules
S-30023	Cross Wire Resistance Welding Process
S-30100	MSD Design Requirements for the Soldering of Electrical Connections

2.2. PHYSICAL DESCRIPTION

The principal restraints governing the packaging design are (1) a lightweight, flexible and compact configuration for the spacecraft electronic equipment; (2) mounting and interconnection for electronic parts so that they will perform reliably and efficiently during launch and long-time exposure to space environment, and (3) use of standardization to the greatest possible extent, to allow parallel development of electronics and vehicle structure.

2.2.1. Description

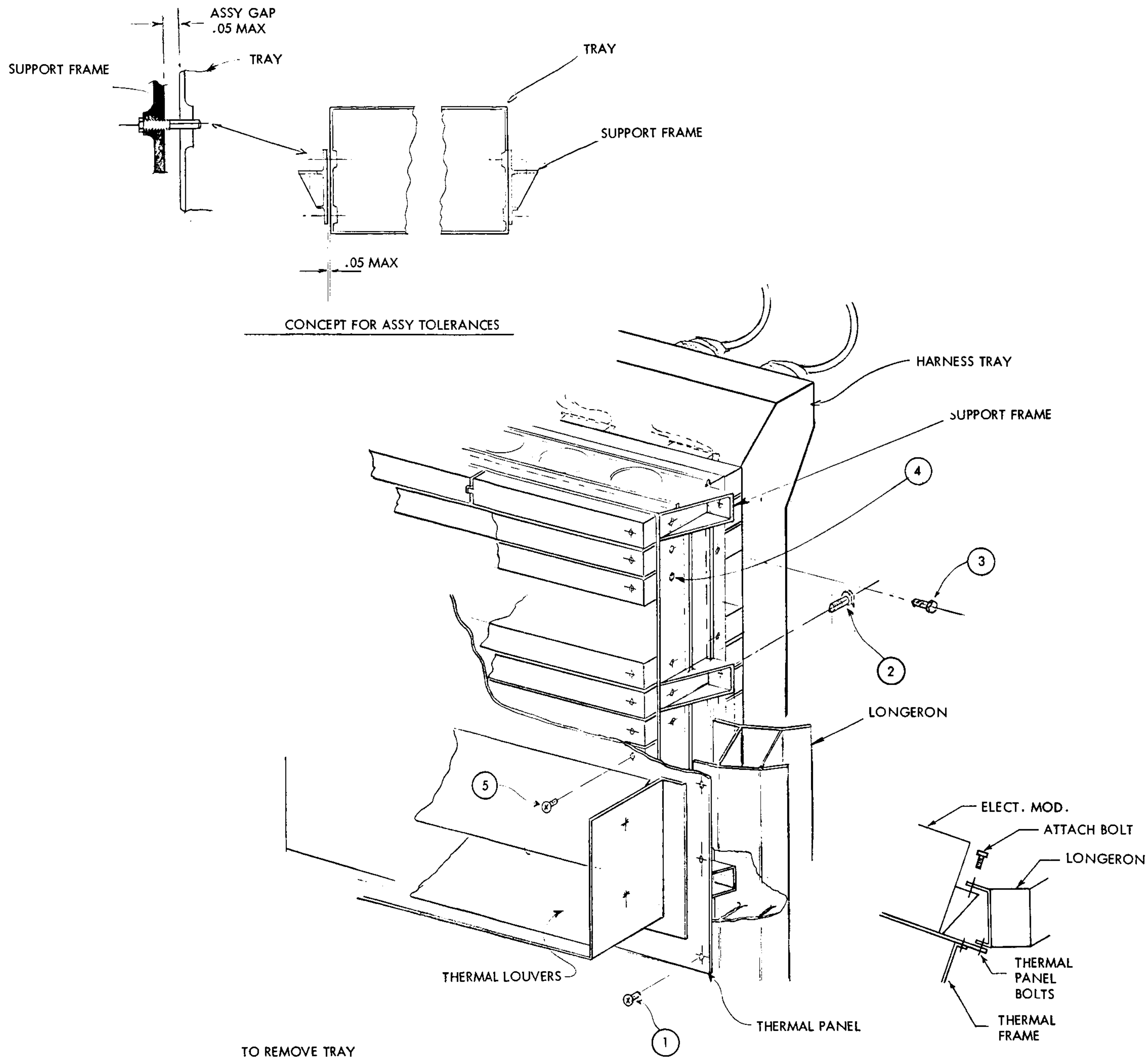
The spacecraft electronic equipment packages are modular assemblies of standard size and shape for mounting in the 16 bays of the Spacecraft Bus. (See Figure 1). When assembled, these self-contained packages give rigid support to the electronic components and interconnections against dynamic and static loads, and provide conductive heat paths from the dissipating parts to the thermal radiating surface.

A standard approach is specified with the provision that nonstandard solutions to accommodate special problems may be used with proper approvals. The standard design uses three levels of interconnections, as defined below:

- Level I - A functional grouping of parts interconnected and encapsulated to form a module.
- Level II - A subassembly of parts and modules in a machined housing of standard profile.
- Level III - An assembly of Level II Subassemblies and plates mounted to, and made integral with, the vehicle structure.

2.2.1.1. Assembly

The assembly package consists of up to 13 Level II subassemblies sandwiched between two plates as shown in Figure 2. The inner plate, in the form of a shallow tray, provides a mounting base for the subassemblies as well as a supporting structure for the assembly



TO REMOVE TRAY

1 REMOVE BOLT 1S FROM THERMAL PANEL
2 REMOVE BOLT 5s FROM THERMAL PANEL
3 REMOVE BOLT 3 & 4 FROM FRAME
4 REMOVE BOLT FROM CONNECTOR ON HARNESS TRAY
5 WITHDRAW TRAY

TO REMOVE WHOLE BOX

1 REMOVE BOLT 1's FROM AROUND THERMAL PANEL
2 BOLT ON HANDLING FIXTURE
3 REMOVE BOLT 2's FROM MAIN FRAME. REMOVE FROM INSIDE
4 WITHDRAW BOX

Figure 1. Proposed Electronic Module Assembly

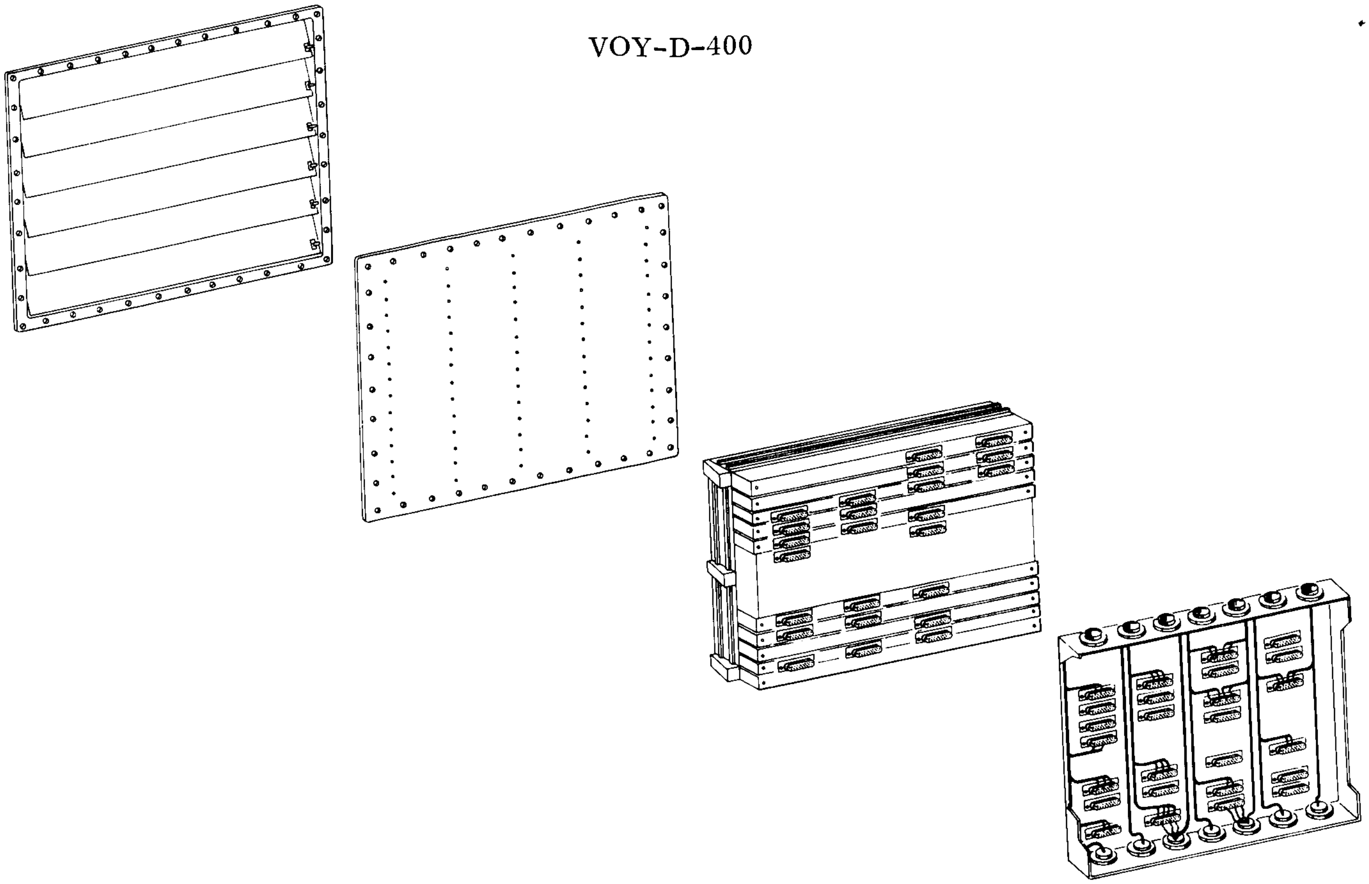


Figure 2. Electronic Assembly (Exploded View)

harness and connectors. This harness tray is a magnesium structure, 18.0 by 19.5 by 4.0 inches, which provides locations for 60 float mounted subassembly connectors and 16 system connectors. The lower eight locations have been designated for both test points and system harness connections, and the upper eight reserved for mating with the spacecraft ring harness. In total, provision is made for 976 connections into and out of each standard assembly and 3000 pins can be made available for subassembly interwiring.

The outer plate, stabilized by the subassembly chassis acts as a radiating surface for heat rejection to space.

2.2.1.2. Subassembly

The electronic subchassis is of standard profile. Two dimensional standards are specified; 8.625 by 6.0 by 1.25 inches and 18.0 by 6.0 by 1.25 inches. The 8.625 inch subassemblies are used in pairs, locked and bolted together to satisfy the structural requirements. An offset web of 0.062-inch section, rib stiffened, is included, pre-drilled with a 0.100 staggered grid hold pattern as shown in Figure 3. The subchassis is a machined housing of HM21A-T8 magnesium, suitably finished for chemical compatibility and thermal control. The housing contains integral fittings at each end for mounting into the vehicle longerons. Non-magnetic attachments are used to mount the subchassis to the harness tray and thermal panel. They are located in a standard pattern on each side of the housing. These attachments are spaced to provide the required stabilization to the thermal panel.

Each 18-inch subchassis has four standard mounting locations for 50 pin, Cannon Golden D connectors. An 8.6-inch subchassis has 2 connector locations. Electrical continuity from the subassembly circuitry to the assembly harness is accomplished by a connector module. (See Figure 4). The connector modules contain guide pins for alignment with the mating connectors which are float mounted in the assembly harness tray. This technique provides for straight, in-line engagement and disengagement of connectors, reducing the possibility of bending connector pins.

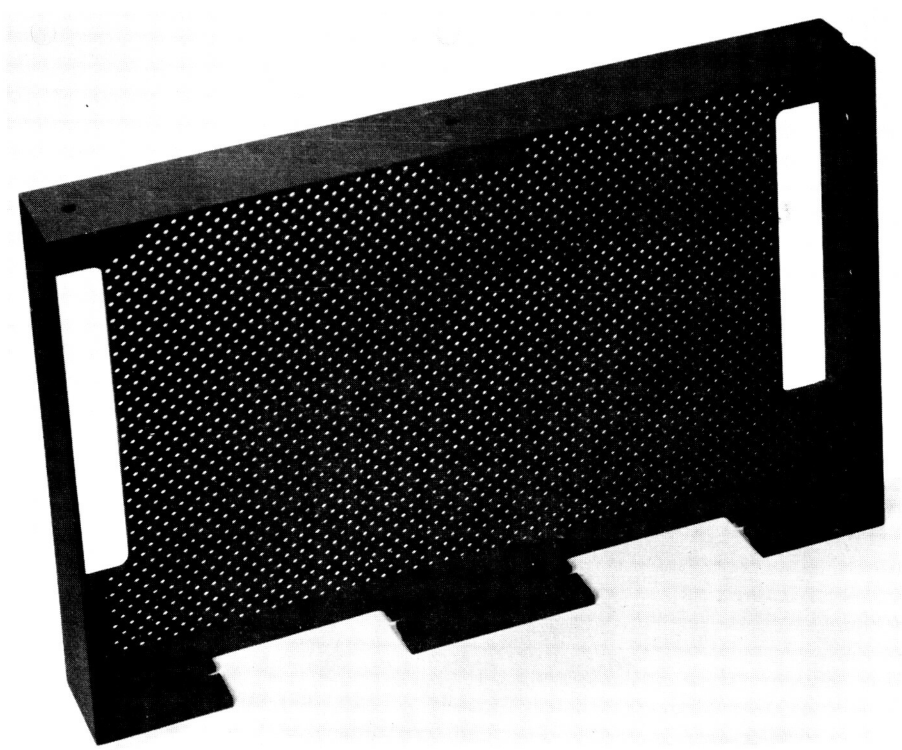


Figure 3. Subchassis Design

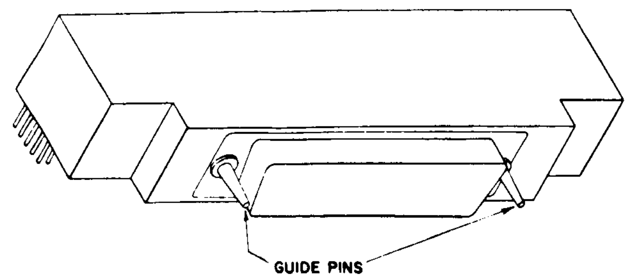


Figure 4. Connector Module

Interconnection of modules in the subassemblies is accomplished by a double sided printed wiring board. The connection between the two circuit patterns is effected by a plated-through hole and a "Z" bar connection.

Indexing of the subassemblies is accomplished by varying the orientation of the "D" shape of the Cannon connector 180 degrees. Since there are four connector locations on each subchassis, there are 16 possible keyed positions; more than enough to provide unique locations for the 13 subassemblies.

2.2.1.3. Modules

Discrete part circuits are packaged into encapsulated cordwood modules consistent with circuit performance requirements. The module dimensions are standardized to assure maximum utilization of the given subchassis area without sacrificing flexibility. (See Figure 5.) In conformity with the principle of minimizing techniques and materials, the design and

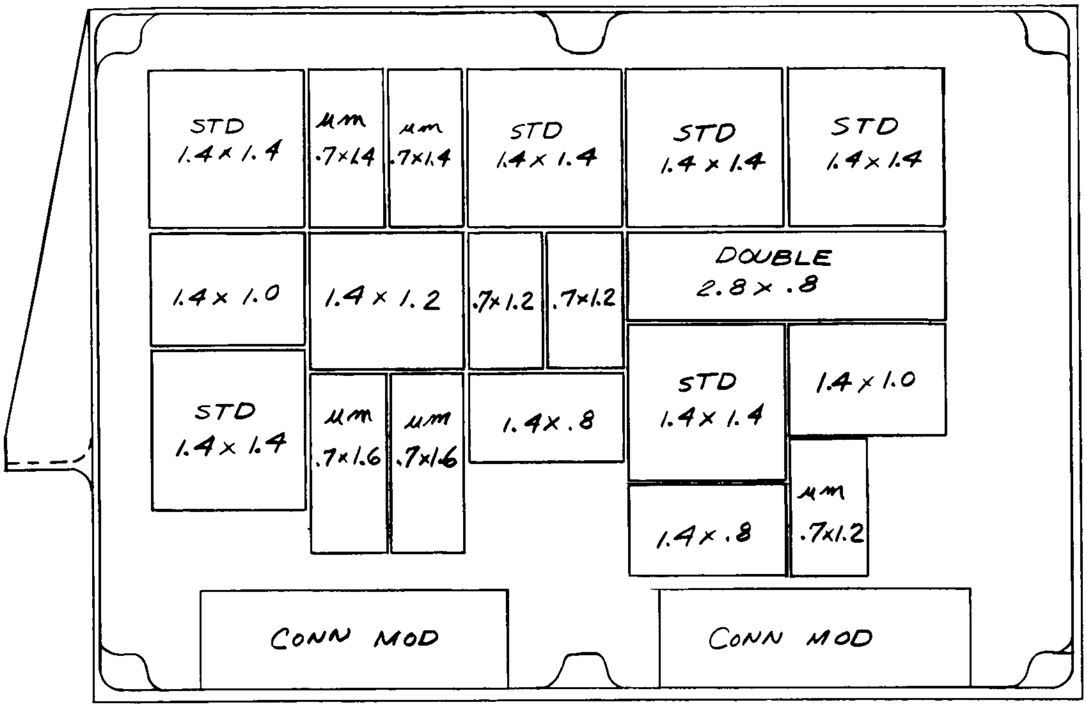


Figure 5. Typical Submodule

fabrication of the cordwood modules will be in accordance with GE specification S-30002, or approved alternative. The encapsulated design provides protection against shock and vibration environments and adequate thermal conductivity to insure component operating temperature within acceptable limits. Buffer coatings are specified to guarantee compatibility of the encapsulating compound with fragile parts.

The packaging of microelectronics is limited to the use of hermetically sealed flat packs mounted by suitable means on double sided printed wiring boards; or 3-dimensional construction of flat packages in special welded modules, as shown in Figure 6.

2.3. PERFORMANCE PARAMETERS

2.3.1. Dynamic Response

The electronic subassemblies, housed in a rib stiffened machined chassis and assembled as shown in Figure 7 have been vibration tested, and exhibited a composite response of 400 cps or higher, with an amplification factor of 11.

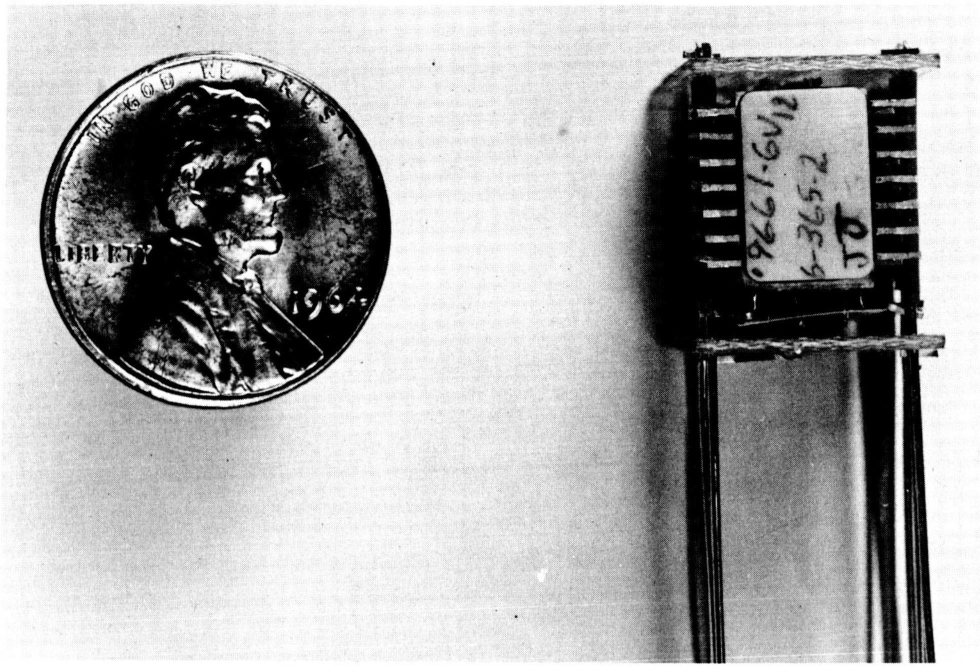


Figure 6. Flat Pack Module

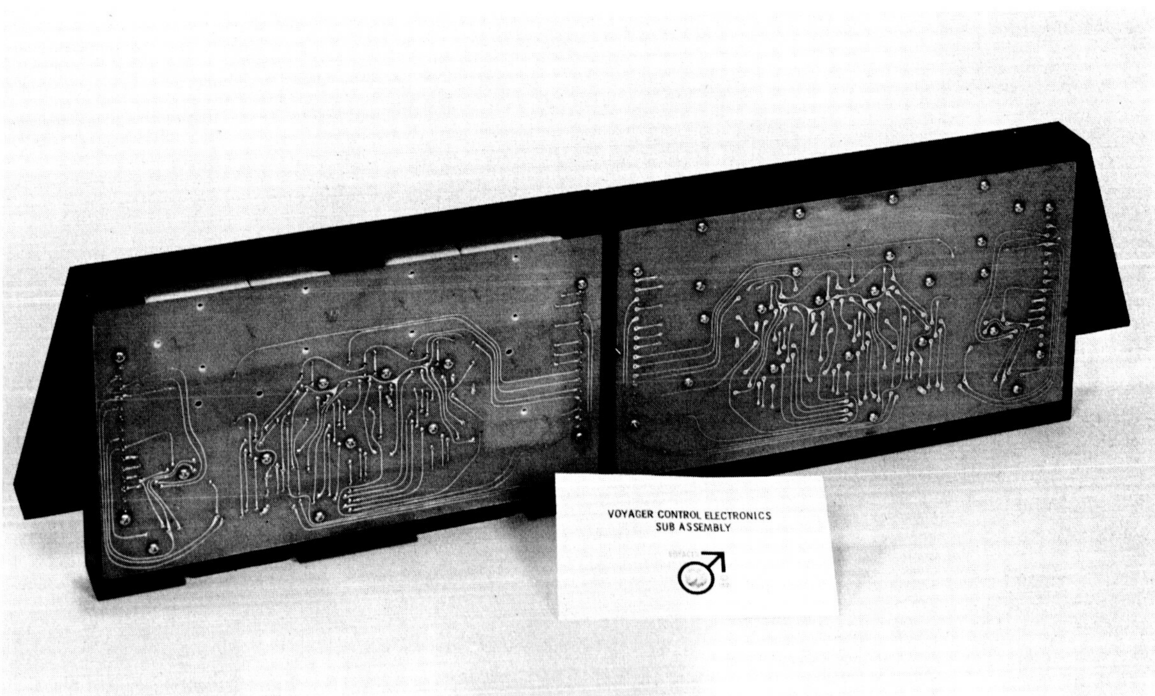


Figure 7. Electronic Subassembly

2.3.2. Thermal Performance

The longest conductive thermal path from a dissipating part to the radiating panel is six inches. Series thermal joints are limited to three: part-to-module encapsulating material; module-to-subassembly web; and subassembly to radiating panel.

In the event of temperature control shutter failure, thermal contact maintained between the subassembly and the harness assembly tray provides a secondary heat rejection path by radiation exchange within the spacecraft.

2.3.3. Magnetic Cleanliness

Considerations of magnetic cleanliness are extended down to the module level through specification of non-magnetic part lead materials, interconnections and hardware. Deviations from this requirement will be allowed only if it can be shown that the substitution of materials is necessary to assure proper performance and/or reliability of the unit.

2.3.4. Environmental Protection

Encapsulated construction and conformal coating is used to assure protection of the parts against handling, dirt, humidity and corrosive atmospheres.

2.3.5. Maintainability

The modular packaging design facilitates the repair and revision of electronic assemblies. It is possible to replace any subassembly without mechanically or electrically disturbing any other subassembly.

2.4. FUNCTIONAL DESCRIPTION

A Level III assembly occupies approximately two cubic feet and may contain up to 112 pounds of electronic equipment. A view of a typical assembly is shown in Figure 2. It is comprised of three sections, as shown: a) the thermal control panel, b) a group of electronic subassemblies, and c) the harness subassembly.

The electronic package configuration is similar to that used in Task B study, except that the subassemblies are attached to an intermediate tie plate, which is then attached to the vehicle longerons. A thermal plate, bolted to the subassemblies, covers the entire area of the bay. An overhang dimension of 5 inches in height and 4 inches in width is added to the electronic assembly dimensions, giving a fin area of 3.5 square feet for the thermal plate. The depth of the electronic subassembly is 6 inches, with an additional 4 inches allowed for connectors and wiring.

Total volume of electronic equipment (subassemblies) is estimated to be 13.36 cubic feet. The equipment is allocated among the various bays by a functional breakdown, so as to avoid having more than one subsystem occupying a single bay. The nine functional subsystems are listed in Table 1.

Table 1. Functional Subsystems

Description		Wt. (lb.)	Volume (cu. ft.)	Connectors* Required
I	Power	321.85	5.75	30
II	Science	185.80	5.28	Later
III	Data Storage	78.60	2.98	8
IV	Telemetry	37.40	1.30	16
V	Command	39.10	1.30	12
VI	Radio	87.20	3.02	8
VII	Relay Radio	56.00	1.76	12
VIII	C&S	51.10	1.46	13
IX	G&C	61.60	1.76	13
Total		918.65	24.51	

*Connectors based on the use of MIL-type, having capacity of up to 61 pins.

Twenty four percent spare volume is provided in the equipment module to allow for growth and flexibility. Sixteen system connectors are provided on each electronic assembly. Additional items which were considered in the study as tradeoffs, include:

a. Area required to dissipate heat from TWT.

b. Weight of an electronic assembly.

c. Easy access to connectors for mating and demating, and to mounting hardware for installation and removal of assemblies and subassemblies.

d. Equipment allocation should accommodate Spacecraft mass balance, thermal balance and proximity requirements of certain subsystems, e.g., Radio S/S next to High Gain Antenna, Science Electronics near Planet Scan Platform, C&S and G&C near sensors and science, Data Encoder and Storage near Science DAE and Radio. A typical equipment arrangement for a sixteen sided vehicle is shown in Table 2.

2.4.1. Level III Assembly

The electronic mounting assembly, consisting of the subassemblies mounted to the harness tray (See Figure 2) is attached to the vehicle structural longerons by means of bolts passed through fittings on the assembly sideplates. After the mounting assembly is in place, the thermal control panel is bolted to the subassemblies and the spacecraft structure, completing the load and thermal paths to the spacecraft frame. The outer panel need not be attached until after the system checkout is complete. This facilitates removal and replacement of assemblies as required during testing.

At the conclusion of the Task B Study, it was recognized that a problem existed with regard to the accessibility of individual subassemblies after the thermal panel was bolted in place, e.g., at system thermal vacuum test. Packaging investigations made during the system update provided a solution to this problem.

The selection of a 120 in. dia. Electronic Equipment Module provides sufficient clearance for an operator to stand inside the spacecraft structure even with the Engine Module in place.

Table 2. Equipment Allocations

Bay	Description	Wt. (lb.)	Subassy. Weight	Subassy. Volume (cu. in.)	Packaging Factor (lb./cu. in.)	No. Connectors
1	Power	98.45	96.65	2020	0.048	10
2	Power	111.95	102.75	2020	0.051	10
3	Science E.	65.60	61.00	1750	0.035	Later
4	Spare	4.2	-	-	-	-
5	Science E.	65.60	61.00	1750	0.035	Later
6	Science DAE	54.60	50.00	1750	0.029	10
7	Data Storage	37.30	31.30	1280	0.024	4
8	Data Storage	41.30	35.30	1280	0.028	4
9	Telemetry	37.40	27.20	945	0.029	16
10	Command	39.10	29.90	945	0.032	12
11	Radio	32.60	26.60	1080	0.025	4
12	Radio	54.60	46.70	1550	0.030	4
13	Radio Rel.	56.00	50.00	1750	0.029	4
14	C&S	51.10	42.00	1220	0.034	13
15	Power	111.45	102.00	2020	0.051	10
16	G&C	61.60	43.60	1750	0.025	13
			806.00 lb.	23,110 cu. in.		

Total available Subassembly

Packaging volume - 30,400 cu. in.

Spare volume - 7,290 cu. in.
or 24 %

This allows the inboard structural bolts to be removed with the assembly in final configuration. Consequently, it is not necessary to remove the thermal tie down bolts and thermal plate to remove or replace a level III assembly.

The harness tray connector mounting bracket was modified to allow the connectors to face inboard, thus providing visual and hand access for reliable mating and demating of system electrical connectors. (See Figure 1.)

2.4.2. Level II Subassembly

In order to tie the electronic assemblies to the equipment module structure, it is necessary that the level II chassis span the 18 inch width between longerons.

In applications where a subchassis width of less that the 18 inches is desired, two 8.625 inch subassemblies are bolted and locked together so that the structural integrity of the assembly is maintained.

To accommodate bulky, non-standard components such as transformers, filter chokes and capacitors, gyros, tape recorders and radio equipment, the standard profile subchassis is allowed to vary in integral multiples of its unit thickness maintaining standard connector and mounting insert locations. High thermal dissipating parts or modules are located near the edge of the subassembly that is adjacent to the temperature control plate. Thermal loading of subassemblies is controlled to insure that part operating temperatures are within the limits for which they have been qualified.

Subassemblies are functional and testable units of the subsystem. The sandwich design of the subassemblies provides adequate stiffness to insure that components, modules and printed wiring boards are not damaged from deflections caused by shock and vibration.

All connections between the outgoing module terminals and the printed wiring board are done only on the exposed side of the board to allow visual verification of the connections. The module terminal clearance hole through the web is provided with annular insulation in the form of a molded insert.

Subassemblies containing repetitive circuits requiring more than two layers of interconnections are packaged in a sandwich configuration (See Figure 8). Two double sided printed wiring boards are used to obtain four layers of wiring. Encapsulated cordwood modules are assembled between a top wiring board and the offset web. Alternate rows of modules are reversed to mate into their respective wiring boards. Titanium fasteners are used to seat all modules firmly against the web for maximum heat transfer and to secure the printed wiring boards. Interconnection between the boards is accomplished by interconnecting modules or sections of flat cable, soldered to terminals and electrically and mechanically secured to the boards. Subassemblies containing large variations in part sizes, or requiring part value changes for tuning and adjustment, are packaged in a flat layout. Those circuits which are not part of the tuning and adjustment are packaged in cordwood modules. The tuning and/or select-at-test parts are mounted to the printed wiring boards, through a cutout in the web, with their leads wired to terminals mounted on the boards. In the event that any of these parts have high thermal dissipations, they will be mounted on the web with their leads wired to insulated terminals passing through the web and into the printed wiring board.

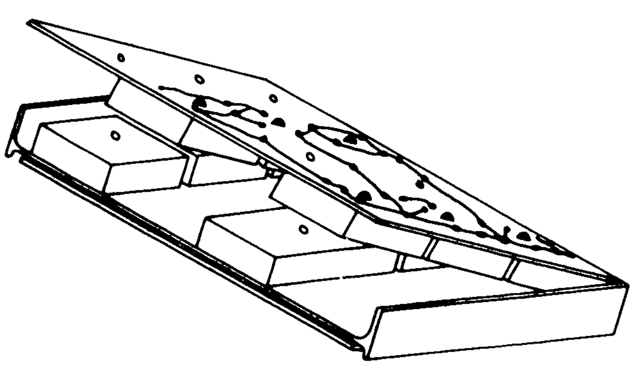


Figure 8. Electronic Tray (Exploded View)

3. PARTS AND MATERIALS

3.1. INTRODUCTION

Two efforts were carried out in the parts and materials area as part of the system update. The first was a preliminary investigation of the effect of ethylene oxide (ETO) on spacecraft materials. Early in the study an ETO surface treatment of the Planetary Vehicle encapsulated in the shroud was discussed as a means of insuring no violation of planetary quarantine. This requirement was later removed. The second effort involved a preliminary review of MSFC-PPD-600 Preferred Parts List, Volume 1, and its associated Design Guide Lines to assess its adequacy for the Voyager project.

3.2. ETO DECONTAMINATION EFFECT ON MATERIALS

Various references for the effects of ETO on materials were reviewed. JPL Document 50503-ETS, January 1966, was used as the ETO decontamination test procedure in the study. The baseline spacecraft configuration was studied for identification of possible condensation points during the decontamination cycle. Due to the complexity of the spacecraft and the problem of maintaining a uniform temperature as prescribed in the test procedure, approximately two hundred prime retention points were identified in the configuration regardless of the position (horizontal or vertical) of the spacecraft during the decontamination process. In addition a study of the envisioned facility for the decontamination process was performed. It was concluded that maintaining the controls of the ETO/Freon mixture, and the water vapor ratio at the specified temperature 123°F were critical. Specific details are contained in VOY-P-TM-23 "ETO Decontamination Effects Upon Materials and Systems," dated August 11, 1967. This interim report concluded the work on ETO since the requirement was removed.

While detailed material effects were not determined, it is certain that an ETO decontamination requirement would entail a need for numerous development tests and would have a degrading effect on the system reliability.

3.3. REVIEW OF MSFC PREFERRED PARTS LIST AND DESIGN GUIDE LINES

To provide a preliminary estimate of electronic part usage on the Voyager program, an analysis was performed of two Voyager subsystems, the Guidance and Control Subsystem and the Computer and Sequencer Subsystem. It is estimated that the two subsystems are representative of the rest of the Voyager Spacecraft, and that they contain most of the individual part types which will be required. Based on the part types identified, MSFC-PPD-600 was reviewed to determine whether these part types were adequately covered.

It was found that most of the part types identified are already within the scope of MSFC-PPD-600. A few items were recommended as additions such as field effect transistors, linear integrated circuits, and RFI filters. All parts and materials on the list were reviewed for adequacy for a long life space program. Recommendations were made for the addition of some items which have proven capability.

The design Guide Lines were reviewed in depth and again several recommendations made for expansion. Examples of additions recommended include:

a. Application Data

Radiation effects
End-of-life limits
Worst case design data and procedures
Semiconductor "do's" and "don'ts"
Expansion of all part category discussion
High voltage considerations
Grounding considerations
Reliability tradeoffs

b. Processes

Tightening torque for bolted joints
Staking of bolted joints
Metal forming
Welding & brazing

The detailed recommendations resulting from this evaluation are contained in Milestone Report, VOY-P-TM-26, "Review of MSFC PPD-600 Vol 1, Preferred Parts List and Design Guidelines for Application to the Voyager Project", issued September 11, 1967.

4. CIRCUIT ANALYSIS OF SIGNETIC DEVICES

4.1. INTRODUCTION

The heuristic approach to electronic equipment for future spacecraft is a microelectronic design. Present day acceptance tests give some unknown degree of confidence in the margin of operation of such equipment when acceptance tested by varying temperature and input voltage. In fact, these tests are not designed to prove that the equipment has some given amount of margin and that they will operate in a space environment for the intended spacecraft mission of one year, two years, etc. The purpose of this study is to determine the feasibility of designing acceptance tests capable of predicting the performance of electronic equipment over extended time. Throughout this investigation, the Signetics Series 400 digital microcircuits were subjected to analyses, since they are being used on a General Electric long life space program and are tentatively planned for Voyager.

Data obtained through circuit analyses, manufacturer's specifications, manufacturer's "Reliability Testing Program Status" memoranda, and in-house life tests are summarized in this report.

4.2. GENERAL APPROACH

Through an investigation of data obtained through:

a. Computer Analyses

b. Manufacturers Specification Sheets

c. Manufacturers Memoranda "Reliability Testing Program Status"

The dependence of the input/output electrical characteristics of the microcircuits of the Signetics Series 400 versus changes in selected variables were isolated. Selected variables include:

a. Temperature

b. Supply Voltage

c. Electrical Characteristics of elements which make up the integrated circuits. Elements include resistors, diodes and transistors.

4.3. COMPUTER ANALYSES

The analyses of the Signetics 400 Series microcircuits, SE416J, SE424J, SE455J, and SE480J, were performed using ECAP (Electronic Circuit Analysis Program). Equivalent circuits representing each of the gates of the 400 series were utilized. The equivalent circuits were constructed using data obtained through:

a. Circuit Chip Examination by Microscope

b. Black Box Measurements

c. Microcircuit Probing

d. Manufacturers Specifications

More than sixty computer runs were performed to produce nine acceptable analyses. Each successive run made on a given microcircuit produced refinements culminating in the accuracy of the final analyses. Specific details pertaining to the acquisition of data used in equivalent circuits needed for the analyses as well as one sample computer analysis output and explanation are contained in Milestone Report, VOY-P-TN-24 (Circuit Analysis of Signetic Devices - 9/11/67).

4.4. RESULTS

Data obtained through this investigation are presented in Tables 3 and 4.

Table 3 tabulates effects of changes in temperature and supply voltage on the operating characteristics of microcircuits of the 400 series. Column 3 lists the percentage change in the output voltage which results from a one percent change in supply voltage. Column 4 presents the percentage change in the output voltage which results in a one degree increase in operating temperature.

Table 4 identifies the most critical circuit elements and electrical parameters within the microcircuits and explains the effects of their degradation on circuit performance.

Table 3. Temperature Change Effect on Voltage Supply

		Voltage out versus:		
Gate	State	Temperature	Supply Voltage	Comments
SE416J	0	$+.015/^{\circ}C$	-0.60	@ Constant Current @ 25°C
SE416J	1	$+0.038/^{\circ}C$	+1.4	@ Constant Current
SE424J	0	$+.1/^{\circ}C$	-0.63	@ Constant Current @ 25°C
SE424J	1	$+.13/^{\circ}C$	+1.6	@ Constant Current
SE455J	0	$+.085/^{\circ}C$	-0.1	@ Constant Current @ 25°C
SE455J	1	$+.042/^{\circ}C$	+1.4	@ Constant Current
SE480J	0	$+.12/^{\circ}C$	-0.61	@Constant Current @ 25°C

<u>LEGEND</u>

0 - Logical "0"

1 - Logical "1"

Table 4. Critical Circuit Elements & Parameters

Critical Circuit Element	Effect on Circuit Operation
Diode forward voltages and transistor collector - base and emitter - base forward voltages	Forward voltages decrease and temperature is increased. This results in an overall loss in DC noise margin at elevated temperature.
Transistor current amplification factors	Transistors whose collector emitter terminals from the microcircuit output terminals must have current amplification factors large enough to insure saturation in the "0" state. Betas are typically four times greater than that magnitude which would produce deleterious effects.

4.5. SUMMARY AND CONCLUSIONS

Data obtained through computer analyses was used in determining the margin of operation designed into the integrated circuits. Combining this with data supplied by the manufacturer allowed comprehensive evaluation of circuit operation. Data describing circuit degradation versus time can now be used in designing acceptance tests to evaluate the operating lifetime of equipment constructed from Signetics Series 400 devices.

As a result of this investigation, it is feasible to develop acceptance tests capable of simulating the operation of electronic equipment for end of life predictions.

SECTION V
OPERATIONAL SUPPORT EQUIPMENT

VOY-D-500
OPERATIONAL SUPPORT EQUIPMENT REQUIREMENTS

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SECTION V
OPERATIONAL SUPPORT EQUIPMENT

VOY-D-500
OPERATIONAL SUPPORT EQUIPMENT REQUIREMENTS

1. INTRODUCTION

This section, dealing with OSE requirements for the updated Voyager spacecraft system, reports the results of a checkout system conceptual trade study. This trade study was conducted in accordance with guidelines and objectives received from the Marshall Space Flight Center. (These are summarized in Section 2).

Four concepts were considered for Voyager Launch support and system testing. These had been utilized on the Apollo Spacecraft, Saturn Launch Vehicle, Mariner, and Nimbus Programs.

Due to the constraints imposed by the launch facilities, and due also to the unique nature of the launch-ready Voyager planetary vehicle, separate checkout configurations are recommended for the system test situation and the pre-launch situation.

For support of the launch, system test equipment would back up the hardwire manual consoles (in the Launch Control Center) in a manner analogous to the Apollo checkout system. (Implementation of this approach could be accomplished using Apollo/Saturn checkout system facilities, such as the hardwires between LC #39 and the LCC, the data/control links, and timing and synchronizing equipment.)

For system testing, the checkout concepts used on the Apollo and Mariner programs were found to be suitable for Voyager, with a preference for the Mariner approach.

Additional factors, however, which should be considered in the spacecraft OSE system selection include:

- maximizing use of available checkout hardware at all locations versus impact on schedules, economics, and performance
- assessment of possibility of program conflicts for use of existing KSC facilities and checkout equipment.
- interface with Capsule OSE and Science (Spacecraft and Capsule) OSE.

Volume IV, Book 2, "Applicability of Apollo Checkout Equipment", discusses the extent of the applicability of the Apollo/Saturn systems and identifies specific equipment which could be considered for incorporation into the Voyager OSE.

A generic description of checkout equipment for the component (sub system) level of testing; and updated assembly, handling and shipping equipment (AHSE) requirements (to reflect the 1973 ground mission and planning) are also included in this section.

2. OSE OBJECTIVES, CRITERIA, AND CONSTRAINTS

2.1. OBJECTIVES

The principal objective of the Voyager electrical OSE is to provide the means for accomplishing the test program required in building a spacecraft for the long mission life. The gross elements of this test program are module assembly and subsystem checkout tests; final assembly and systems checkout tests at the factory; system checkout test at the launch area; and prelaunch and launch operations. Accordingly, the electrical OSE is provided in three configurations:

a. The Bench Test Equipment (BTE) is the equipment required to support module assembly and subsystem testing.

b. The System Test Equipment (STE) is the equipment required to support subsystem integration during final assembly, system test at the factory and system test at Kennedy Space Center (KSC).

c. The Launch Complex Equipment (LCE) is that required to support prelaunch and launch operations with the spacecraft mounted on the launch vehicle.

A corollary to the requirement to support test is that the OSE must be able to stimulate the spacecraft and measure its outputs in such a fashion as to provide data indicative of trends in the performance of the spacecraft. In this manner it will be possible to assess its performance as a function of time. The OSE should have the capability to stimulate the spacecraft and collect the data for trend analysis. However, the processing of this data should be performed after the test on a general purpose computer system.

In order to accomplish these objectives the OSE must have the capability to exercise the spacecraft and its subsystems through all its standard and secondary modes of operation. It must be capable of adapting to run standard modes of operation and have the flexibility to support troubleshooting to the degree necessary to locate faults down to the level required by the type of test being performed. The OSE should be designed so that it is easy to differentiate between faults in the spacecraft and faults in the OSE. The principal objectives of the assembly, handling, and shipping equipment is to provide the capability for lifting, holding, positioning, transporting and aligning the spacecraft and its parts during the ground mission.

2.2. OSE DESIGN CRITERIA

The basic OSE design criteria is the use of proven techniques with state of the art equipment and methods. In particular, heavy emphasis shall be placed on the use of techniques already proven on earlier space programs such as Apollo/Saturn, Mariner, Nimbus, etc. This course capitalizes on past experience and provides equipment that is the natural extension of previous equipment.

Within this framework, the OSE should have the following capability:

a. Stimulate the spacecraft so that it appears to exist in its normal operating environment. This includes generation, processing, transmission and distribution of commands to the spacecraft and its interface equipment.

b. Collect, process, transmit distribute and display data for real time and non-real time analysis.

c. Automatic processing of the data by a general purpose digital computer.

d. Execution of automatic test sequences with manual (operator) override; semiautomatic test sequencing and manual control of the entire test sequence.

e. Provide a complete record of the test in the form of magnetic tape recording. This shall provide the basis for post test data processing as well as on the spot recall of test data for the analysis of anomalies which occur during the execution of a test.

f. Provide an effective man/machine interface in the form of consoles for operator command and control and display devices for data presentation.

g. Provide diagnostic programs and test routines to facilitate fault isolation to the required level.

h. Provide OSE self test and diagnosis routines to assess the condition of the equipment and to assist in the isolation of malfunctions and their repair.

i. Provide redundant paths for functions critical to the spacecraft launch operation and functions related to hazardous operations. This includes hardware bypass of digital data links where necessary. These shall be provided so that the OSE does not cause or contribute to failure to launch and so that hazardous operations can be accomplished without danger to the launch operations personnel.

j. Operate independently of the launch vehicle support equipment.

k. Provide the capability to support the testing of any Voyager spacecraft with any set of OSE.

l. Provide the capability to test the spacecraft without having the OSE affect the spacecraft's performance. In addition, a failure in the OSE should not contribute to or cause a failure in the spacecraft.

m. Provide the capability to adapt to changes in the spacecraft as its design evolves.

2.3. OSE DESIGN CONSTRAINTS

The basic OSE philosophy is to provide general purpose equipment with general purpose capability and a high degree of flexibility. This approach must be constrained to reflect the environment in which the OSE operates and the specific tasks it must accomplish. In general, the equipment must be reliable, maintainable and produceable at a reasonable cost. It also must be able to perform its job within the confine of LC 39 at KSC as well as at the various systems test facilities such as the thermal vacuum test facility and the SDF at MSFC. As a result, the following design constraints apply:

a. Design and construction shall be in accordance with the appropriate customer and contractor standards.

b. State-of-the-art designs and off-the-shelf components shall be preferred to designs which require extensive development work.

c. Interchangeability shall be provided by using standardized component, circuit cards, modules, power supplies and test equipment.

d. Standard signals available at LC-39 shall be used with the OSE. For example, the system shall use the Saturn timing equipment when at LC-39 and a comparable system at all other locations.

e. A single point ground system shall be used in accordance with that used for Saturn V.

f. Provision shall be made for transfer to emergency power sources when necessary.

g. The OSE shall be capable of interfacing with the deep space network, the launch vehicle electrical support equipment, the facility instrumentation system and the capsules OSE.

3. SYSTEM TEST AND LAUNCH COMPLEX EQUIPMENT

3.1. GENERAL

Volume III of this report, the "Implementation Plan," describes the ground flow plan for the spacecraft. The sequence, location and nature of assembly and test operations, starting with factory level and going through launch, are given in these flow plans. The STE is shown to be used both in the factory, to support buildup and test of the spacecraft, and at the launch site for pre-launch system testing. For the purpose of this study, a standard set of System Test Equipment for both factory and launch site has been assumed. Use of the STE to support intermediate test activities, on some spacecraft models, is also probable. Similarly, launch complex equipment will be used in the early factory part of the ground flow to establish compatibility with spacecraft.

The STE is used in three major portions of the factory assembly and test operations; (1) the integration of subsystems and components into the spacecraft, (2) the determination of performance margins and interactions, and (3) the simulation of the operational mission. These are the key activities in factory level spacecraft testing (environment simulation is a controlled variable during some system tests). This level of testing is also required at the vicinity of the launch site, as part of post-shipment system verification. Hence, the system test equipment is needed here, also.

The STE must have the capability to verify proper interfaces between, and interdependent operations of, the spacecraft's subsystems. It should be in substantially the same configuration for testing flight spacecraft, as for testing earlier, non-flight test model spacecraft.

Different units will probably be required for testing the different spacecraft models. The test schedule information presented in Volume III is based upon a high degree of overlapping test phasing, and approximately six to seven STE's will be required to support this schedule. This estimate will vary with the over-all program management tradeoff between economic and schedule risk.

In conducting a quantitative measurement of performance of the spacecraft, the STE is the principal engineering tool of the test/design personnel. The STE must have characteristics which permit and facilitate engineering personnel to establish exactly what performance characteristics have been obtained. This is a broader job than measurements intended to establish whether a performance parameter is between some predetermined upper and lower limit. It involves giving operating personnel sufficient direct and unobscured capability to establish what the actual characteristics and interactions are (including those not anticipated). For example, the ability of the system to establish or maintain synch, or any other proper operating mode, under controlled repeatable degrees of degradation of other parameters such as off-frequency, low power, noise, etc., is required. The suitability of the STE insofar as facilitating engineering analysis of the tests, development acceptance, verification, etc., is influenced by the basic concepts to be used in developing the STE, as well as the design features of the hardware. The STE is also required to facilitate the recognition or identification of performance trends which develop. The concept and design which is developed for Voyager's STE should reflect this requirement.

The STE is also the principal engineering tool available to Engineering in conducting simulated mission-type testing. This involves using the STE as a substitute for the deep space net to control and analyze the performance (including backup modes of operation) of the spacecraft in a simulated mission sequence. The STE concept, design, and implementation should maximize the capability of the engineering team personnel using it, to conduct such simulations. This use mode of the STE is the initial way in which engineering capability is built up to "fly" the spacecraft, become aware of its operating characteristics, and, most significantly, build the analytical and empirical basis for being able to analyze and correct (or work around) some anomalous behavior or deviation from prior behavior.

In the buildup process, wherein flight or flight-type components become integrated into a spacecraft configuration, the STE is required to take over (from bench test equipment used for component test) the role of providing ability to verify, measure, control, etc., the spacecraft interfaces that become integrated. One of the important capabilities that the STE must implement is to assure that spacecraft component flight qualification status is not prejudiced. The engineering team using the equipment must have maximum provision in the STE design, for display and control characteristics which match their need for rapid analysis and avoidance of improper control of spacecraft parameters.

3.2. CONCEPTS

There have been several types of STE concepts which have been developed and implemented on various aerospace programs. At best, these concepts proved to be optimally suited for the unique requirements of the particular program; at worst, the final design and implementation of the concept was made to work so as to be successfully used on the particular space program.

It had been recognized that the Voyager mission (and the probable Voyager spacecraft characteristics) represented a unique situation in regard to requirements imposed upon OSE. These mission-oriented aspects include the following OSE requirements which, in combination (if not necessarily independently), impose a unique OSE requirement:

a. Long-time interval of operation (interplanetary journey), occurring prior to key mission events, and occurring after encounter. Then, the desirability of long-life operation of its own science payload in orbit around MARS, after the SLS mission has been completed.

b. Complexity represented by the functions required to be designed into the spacecraft. This includes its "automatic" operation capability, the command repertoire required, control and processing of information, and scope of operational and functional interfaces.

c. The schedule constraint imposed upon testing operations by the existence of a fixed and absolute launch window, as well as by obvious economic constraints which are imposed by dependence upon the Saturn V launch vehicle and its related facilities.

These kinds of considerations make it appropriate that, in generating preliminary requirements for OSE to support Voyager 1973, principal concern be given to determination of which basic concept would be most suitable for Voyager.

A major guideline used during the Task D study, was to work at the conceptual level, rather than the design level, as the general requirements of spacecraft tests were expected to be a firmer data base than a design which was in the process of being updated and subject to trade-off decisions.

Another major guideline was to determine the preferred OSE concept by considering the OSE configuration required to support the launch pad operations as the determining one, and then to work back to system tests of the spacecraft, component tests, etc., in order that an emergent concept would have maximum interface commonality between spacecraft and OSE.

The final major guideline was the consideration of concepts in use by current/recent space programs, so that Voyager might profit from them, and the consideration that concepts were to be formed in a manner that would permit accommodating the Saturn V, Complex No. 39 imposed constraints.

Concepts for OSE for the 1973 Voyager Spacecraft are listed and defined. Some of these concepts have been implemented for more than one program and the actual OSE designs of these separate implementations have, in some cases, substantial differences. The attempt has been made, therefore, to treat concepts and implementations separately in this report.

The concepts defined below are applicable to system integration and performance testing, and pre-launch of a complete spacecraft system, but not necessarily applicable to component, assembly, or subsystem testing, even though some of these concepts were originally formulated and implemented with this objective. If it is considered desirable to view the concept with this greater scope of applicability, the concepts fall into two categories, dependent upon whether or not separate "bench test equipment" is needed for component level testing.

3.2.1. Concept "A"

Concept "A" (shown in Figure 1) is primarily a remote control concept which enables a test complex, located in a system test area or facility, to conduct tests on a remote spacecraft

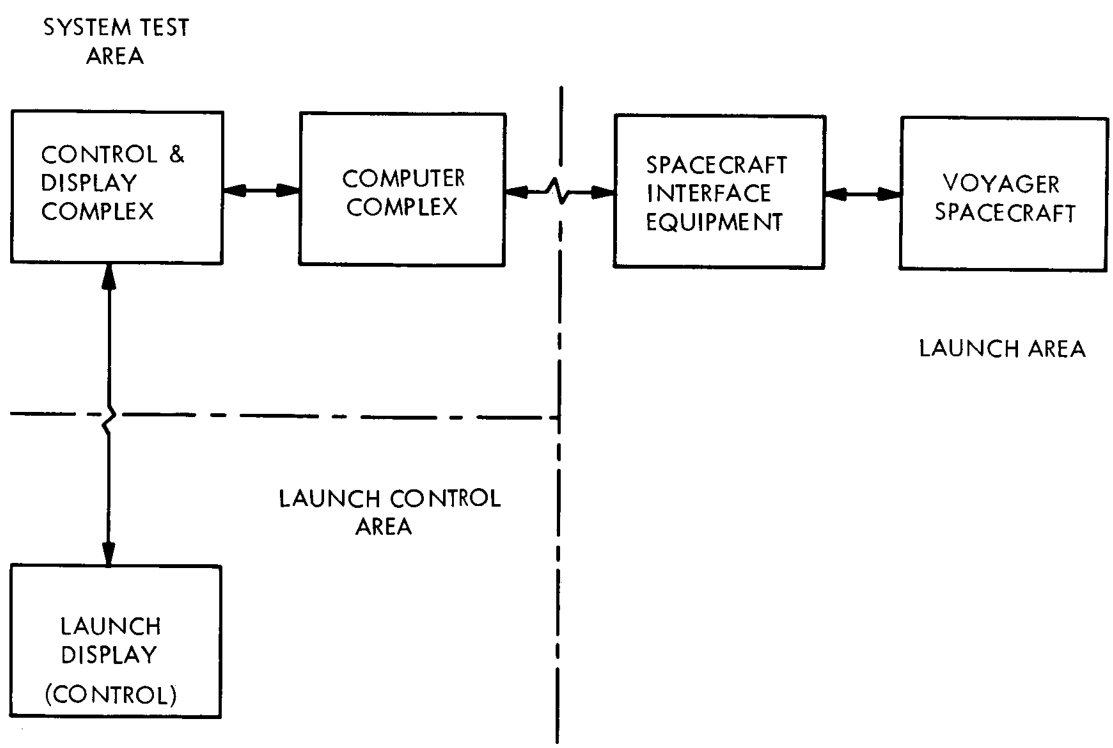


Figure 1. Concept "A"

as it is processed through test facilities to the launch area. It also provides status information to a remote launch control area. In this concept, there is data exchange between an up-link and down-link on a one-for-one basis (zero order data compression). For every operation that is initiated in the system test area, and implemented by the command computer in the form of a digital message to the remote Spacecraft Interface Equipment (SIE), there is one function altered in this interface equipment. An example would be a switch closure in the SIE. Similarly, for each sampling of a monitoring point in the spacecraft, one word is returned to the display.

The degree to which operation of this type of checkout system is automated is not relevant to the concept. It could be programmed to be completely manual (each operator action causes the up-link function of the computer to, in turn, complete one discrete function in the SIE, and each corresponding datum sent back to the operator for inspection, analysis and decision). Alternately, it can be programmed so as to proceed through sequences of test routines, using the computer capability to analyze and decide if responses were proper. The operator displays are then used only for monitoring.

A feature of this concept is program control. This is common to any concept which utilizes general purpose computation equipment and digital data flow only between the system test equipment and a remote spacecraft. Required configuration changes are implemented through programming. The design of such a system should exploit this feature to achieve flexibility and economy.

3.2.2. Concept "B"

This concept is shown in Figure 2.

It is somewhat similar to concept "A" in several respects, especially in the fact that the control and display complex and computer complex can conduct test operations equally effectively upon a spacecraft remotely located at either a test site or at a launch site. The major

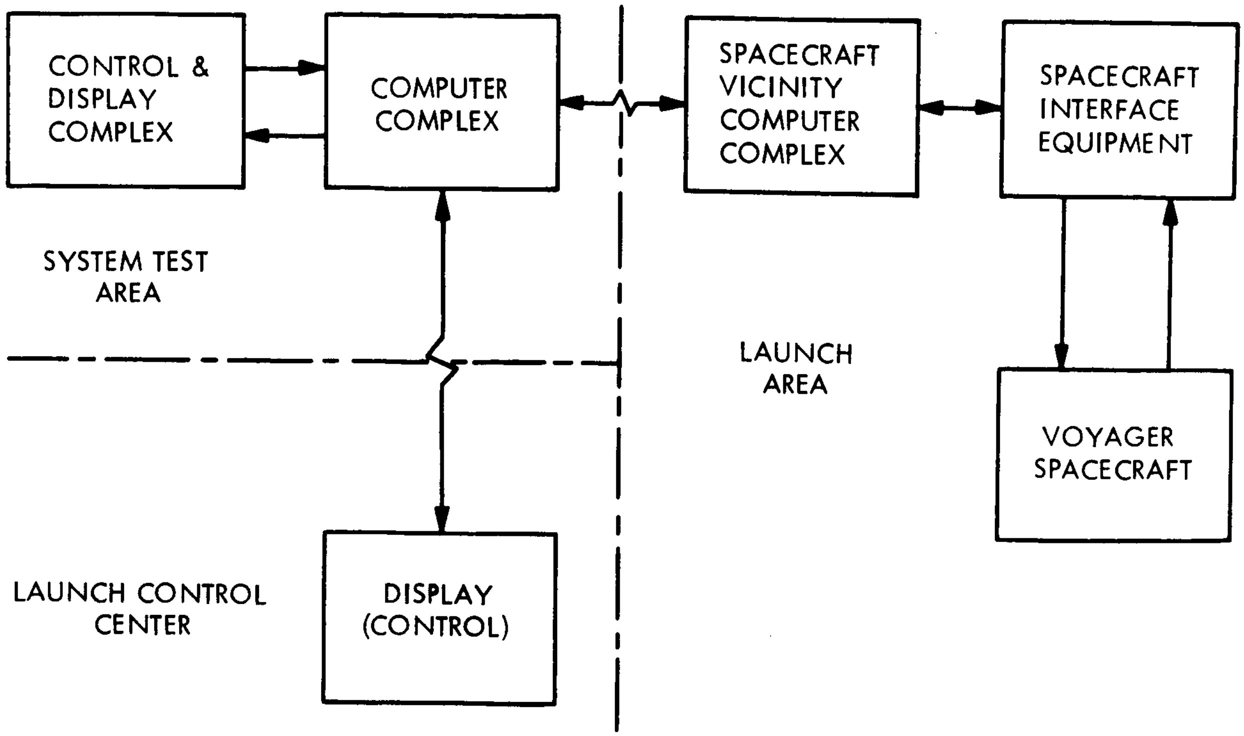


Figure 2. Concept "B"

difference in concept lies in having a second computer complex in physical proximity to the spacecraft and SIE. This second computer complex accompanies the spacecraft and the other interface equipment through the ground flow up to launch.

This "vicinity" computer complex allows both up-link and down-link data compression to be achieved, and is accordingly a somewhat more sophisticated concept than "A". In the "up" direction, a command request, initiated by the test operator's controls (control and display complex), causes the system test area's computer complex to generate a command word. This word is transmitted to the remote computer complex and can be used to control pre-programmed test sequences and logic which, in turn, controls the operation of the SIE. Down-link data compression can be achieved by optimum allocation of computer loads. For example, the remote computer complex can be made to reject unchanged data samples and send to the displays only indications of changes from expected or established values. The data compression concept, to be valid for a given application, should meet two criteria;

one is that the total test load should justify two computer complexes, and the other is that the allocation of programs to the two can be done such that data compression can be implemented while maintaining a balance between the two loads.

It should be noted that this concept, and also concept "A", can be augmented with parallel hardwires from the control and display complex to the SIE. These could be used for back-up or for critical "safety" circuits (such as EED control), or both. The salient point is that the use or absence of parallel hardwires is a feature of a particular design and is not central to the concept.

3.2.3. Concept "C"

Concept "C", shown in Figure 3, can be called a "blockhouse" concept. This concept has been the one almost universally implemented in spacecraft, satellite and similar programs

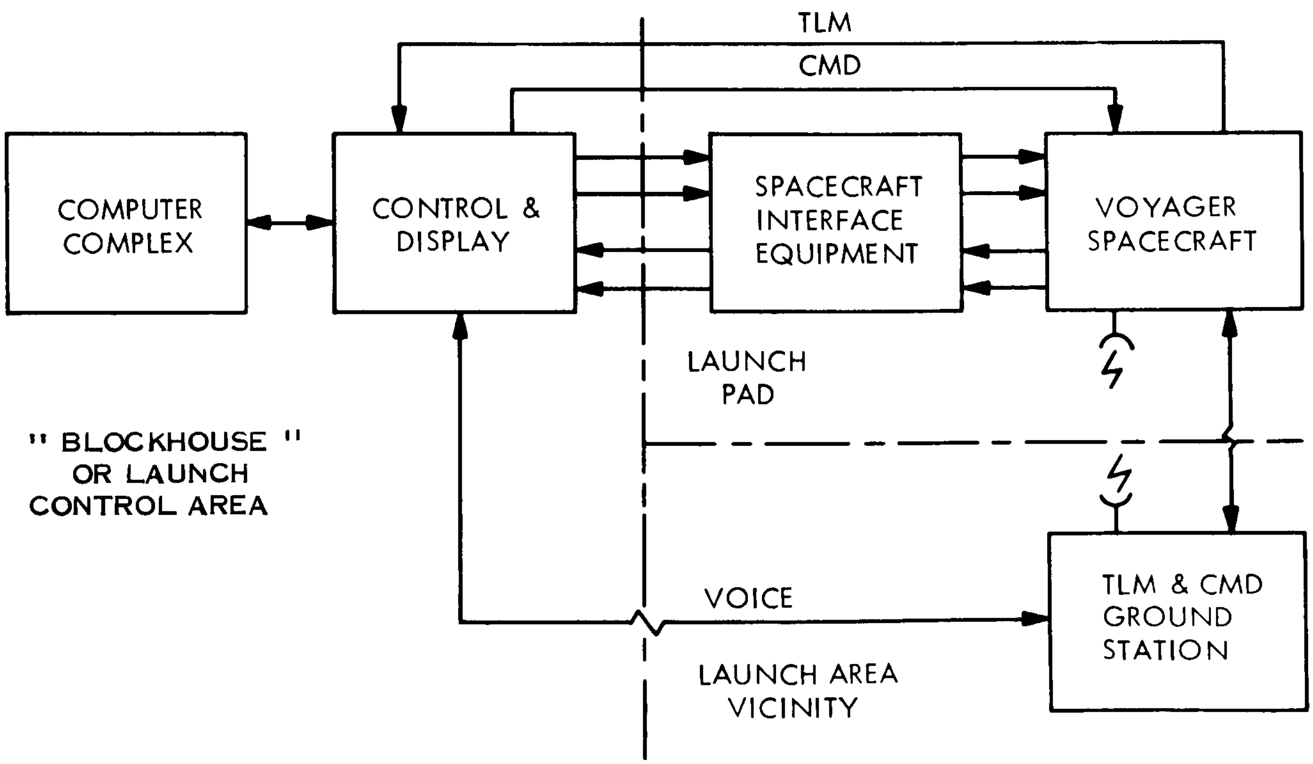


Figure 3. Concept "C"

prior to the advent of launch vehicles requiring a launch complex such as LC 39. The control and display complex is located within the blockhouse and communicates with the SIE over hardwires. To an extent, this can be considered remote operation, but the distance limitations imposed by copper drops, noise, etc., limit the concept to the conventional blockhouse type of launch complex.

The block diagram assumes that the control and display complex performs all of the scaling, processing, etc., of monitored spacecraft data, but this is not essential to the concept. It could also be accomplished at the SIE.

Part of the concept is the utilization of a ground station. The usual elements in a ground station for a fairly complex spacecraft, are telemetry and command ground control consoles and data processing, sufficient in capability to generate simulated commands and process telemetry. This station can be located in the blockhouse, in a nearby facility, or a trailer, etc.

This concept is generally associated with direct operator control and one-for-one relationships (no data compression). It is used primarily in the launch configuration where hardwire and coax lines connect the umbilical functions to the blockhouse, and where end-to-end performance is verified by the ground station. Where the spacecraft complexity is limited, the spacecraft performance characteristics can be determined adequately with a blockhouse system (or where sufficient hardwire test point access exists to a more complex spacecraft).

It should be noted that the blockhouse concept is apparently not too compatible with the physical constraints imposed by LC 39. However, its use for launch control and conditioning of enshrouded spacecrafts mated to the launch vehicle is conceivable due to the selected size of the spacecraft umbilical. The "blockhouse" could be the LCC, and data links used in addition to hardwires.

The block diagram shows a launch operation configuration for this concept. It can be applied equally to spacecraft system test situations at any location, including the spacecraft contractor's factory, MSFC, etc. The block diagram would remain the same in appearance, but the greater test point access and greater depth of test detail would be reflected in correspondingly greater capability in the SIE and controls and displays. Also, the physical location relationships shown in the figure would not constrain the system test configuration for this concept.

The computer, shown in the diagram, is probably more necessary for system test (command and telemetry processing, OSE data processing, etc.) than it is for launch pad support. This is highlighted by the fact that DSIF No. 71, the local operational ground station would be on line and in-lock with the spacecraft as a probable launch criterion. Conceptually, at least, it is reasonable to consider that the operational telemetry and command capability of this ground station could be so applied, perhaps expanded, as to make it unnecessary to physically locate a computer in an actual blockhouse.

3.2.4. Concept "D"

This concept, shown in Figure 4, can be termed an "operational" concept, in that is uses a maximum amount of operational equipment for checkout.

The block diagram shows that the operational ground station, used for flight telemetry and command loop completion, is supplemented by controls and displays analogous to the "blockhouse" concept. This is required because control of critical parameters and monitoring of launch critical signals are functions which should be implemented through the spacecraft's umbilical. The decision as to use of data and control links or hardwire interfacing between the SIE and the launch controls and displays, is dependent upon geographical and economic considerations.

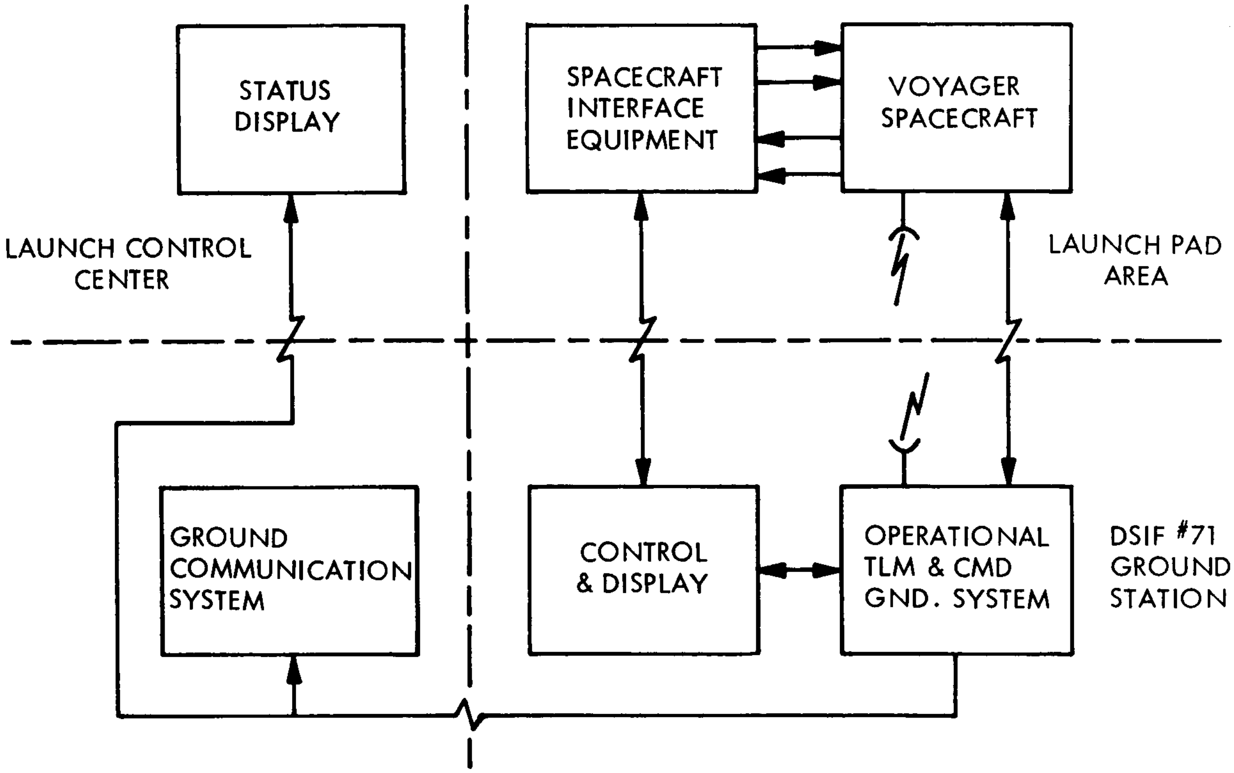


Figure 4. Concept "D"

The main distinction of this concept is the use of the flight operations station, and the operations control center and space network to conduct pre-launch test operations. The flight operations station completes the operational telemetry and command loop to the spacecraft (RF when conditions permit, coax during RF silence periods at pre-launch). Stimuli for testing are not artificial, but are actual commands, while responses to the stimuli and status are obtained through both engineering and, to the degree possible, science telemetry channels.

The essential feature of this concept is that it eliminates, at least at the conceptual level, the uncertainties due to undergoing a transition from a spacecraft-to-OSE interface over to a spacecraft-to-operating system interface during final system checkout. System checkout performed during earlier phases of the ground flow, such as system checkout at a contractor's facility, would require that the flight operations station, control center, and network be simulated. During this phase of checkout, the concept becomes very close to Concept "C", the blockhouse concept, which has been discussed.

An essential element of this "operational" concept is that not only are the command, telemetry, and processing capabilities of the mission operation system used for checkout, but that the executive functions of the checkout and launch teams for the spacecraft would become part of the operations system. There are two significant implications: (a) Incorporation within the local (near the launch site) operations station of data handling and other extra capability needed for system checkout, and (b) that the blockhouse functions, except for spacecraft launch directors displays and communication, can be eliminated by remoting all other umbilical terminal functions to the operating station.

3.2.5. General Conceptual Characteristics

The preceding definition of candidate spacecraft checkout concepts has not addressed two obvious general characteristics; the presence, absence or degree of direct parallel hardware control of the spacecraft, and the degree to which computer equipment (or other hardware for that matter) should be used to make the checkout automatic. These two characteristics are considered to be independent of the essence of the basic checkout concept for most of the cases discussed. Automation and hardwire backup are regarded as features that can be used or not in these several concepts as determined by the policy developed or test requirements of the specific spacecraft type and its mission.

It is recognized that the concepts stated all lend themselves to many variations. The criteria which define a "new" concept, as differentiated from a variation of an "old" concept, have not been established to the mutual understanding of all groups interested in the problem. It is therefore a distinct possibility that concepts which straddle the definitions used here, would nevertheless become of interest for consideration in the 1973 Voyager Spacecraft Program. The use of different concepts for Spacecraft System Testing, and for Launch Support, is an example.

3.2.6. Implementation

The concepts described are intended to be philosophical in order to illustrate the significant aspects of checkout systems for spacecraft. However, each of these concepts has been implemented into an operation system. These systems indicate a design spectrum for test and checkout of launch vehicles and spacecraft. At one end of this spectrum is exclusive hard-line control of the end item with the role of a computer, if one were used, restricted to monitoring system responses. More recently, computers have been allocated the responsibility for pre-programmed subroutines and formatting and transmission of manually initiated command data. Data compression and telemetry decommutation and analysis are additional roles for which a computer is a candidate.

The following illustrates how the concepts have been implemented. Many operational approaches can be described for each concept, but only one is required to illustrate its application.

3.2.7. Concept "A" Implementation

OSE Concept "A" has been implemented on the Apollo Program in the form of Acceptance Checkout Equipment - Spacecraft (ACE-S/C). Systems are operational at KSC, the spacecraft contractor's plants, and at the Manned Space Center. The checkout system now being implemented for a Department of Defense manned space program is another implementation of this concept, although substantial differences exist between the designs and hardware of the two systems.

The following is a brief description of ACE-S/C, intended to illustrate the significant aspects of the concept. Figure 5 is a top level block diagram of this system, as it is implemented at KSC.

The Control Room and Computer Room for each ACE-S/C system is located at the Manned Space Operations Building (MSOB) (for the KSC systems). The test personnel in these rooms can conduct tests on the spacecraft, CSM/LEM, regardless of whether it may be at one of the KSC test sites or at the launch area (LC 39) integrated with the Saturn V on the Mobile Launcher (ML). Upstream commands and downstream data flow between the MSOB and the

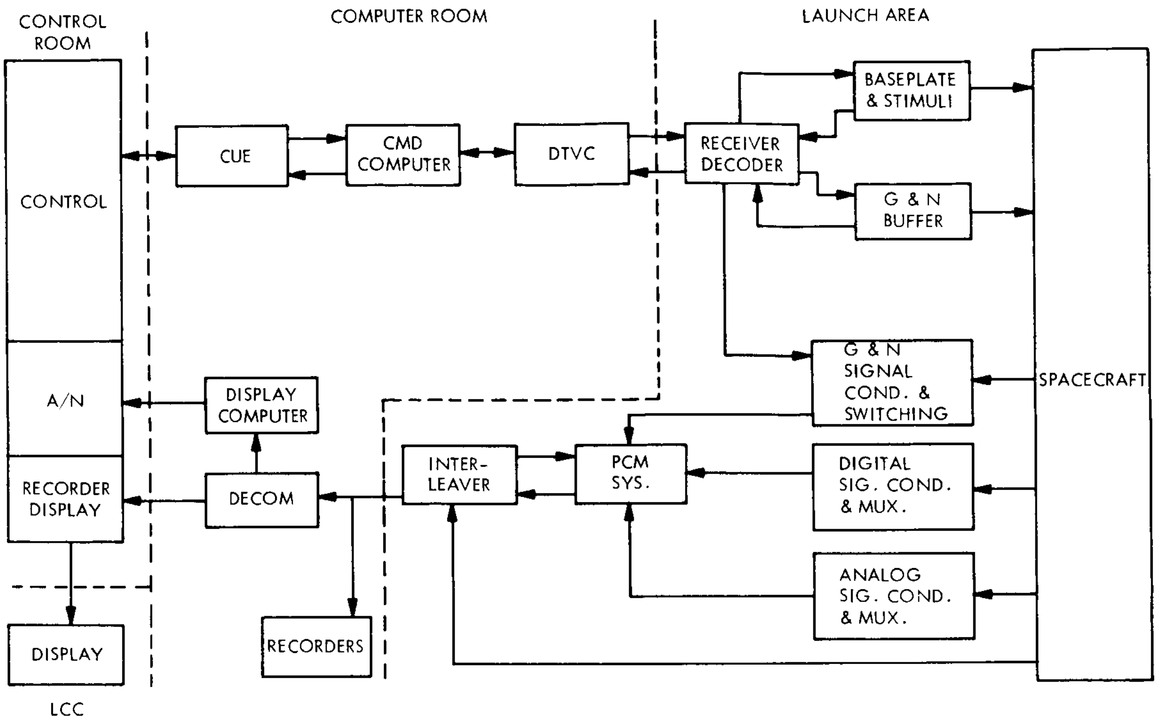


Figure 5. ACE-S/C Block Diagram

and the spacecraft over data/control links. In addition, test data and status are supplied to the spacecraft consoles in the Launch Control Center (LCC) firing rooms, by data link, from the MSOB.

3.2.7.1. Control and Display

The control and display consoles in the control room are general purpose types, and can be configured and allocated according to the requirements of a spacecraft. The present configuration is suited to the Apollo spacecraft. The control capability, available to these console operators, is initiation of discrete commands, preprogrammed computer subroutines, or direct insertion of a digital sequence into the computer. This control is available through the use of three corresponding control module designs used in configuring the consoles.

The displays at these consoles range from alphanumeric CRT displays of engineering data, selectable by the operator, to standardized meter modules, indicator modules, recorders, etc.

3.2.7.2. Computer Complex

The computer complex consists of two general purpose computers with shared memory and high speed programmable decommutation equipment and other peripherals. It performs the dual function of formatting and transmitting operator commands to the parts of the system in the vicinity of the spacecraft and of accepting, decommutating and processing the measurement bit stream for display. The complex is organized such that one computer is a "command" computer while the other is a "display" computer.

Commands, of the three types noted earlier, are input into the computer by a special peripheral, the CUE. The computer formats these commands and transmits them through a special transmission and verification peripheral (DTVC) to the equipment interfacing with the spacecraft. This peripheral verifies that the serial message sent over the data link is correct.

The measurement data sent downstream to the display computer is an interleaved PCM bit stream. It is decommutated and processed for display by the high speed decommutator and display computer. The display computer compares test data to stored limits and, in the case of A/N displays, outputs the test value and the results of the comparison to the display. All raw data is outputted directly to display equipment by the decommutator. The incoming telemetry is recorded for replay or analysis.

3.2.7.3. Vicinity Equipment

The equipment in the vicinity of the spacecraft, which stimulates the spacecraft systems on command and acquires, processes and transmits responses, is functionally divided into a Digital Test Command System (DTCS) and a Digital Test Measurement System (DTMS). These are respectively, the "up" and "down" channels shown on the block diagram of Figure 5. The DTCS basically consists of a receiver decoder which activates standard switching-tree modules (baseplates) and stimuli generators. The DTMS selects, on command, test points and acquires both analog and digital signals to be measured. These are scanned, sampled, multiplexed, and transmitted in interleaved PCM form to the decommutator and display computer for processing and display on control room consoles.

Displays of test status are also transmitted over data link, from the control room to the spacecraft consoles in the LCC.

3.2.8. Concept "B" Implementation

Concept "B" has been implemented in the Saturn ESE program. The system is operational at the KSC (LC-34, LC-37B and LC-39) and in the SDF at the Marshall Space Flight Center.

The following is a brief description of ESE, intended to illustrate the significant aspects of the approach. Figure 6 is a top level functional block diagram of this system.

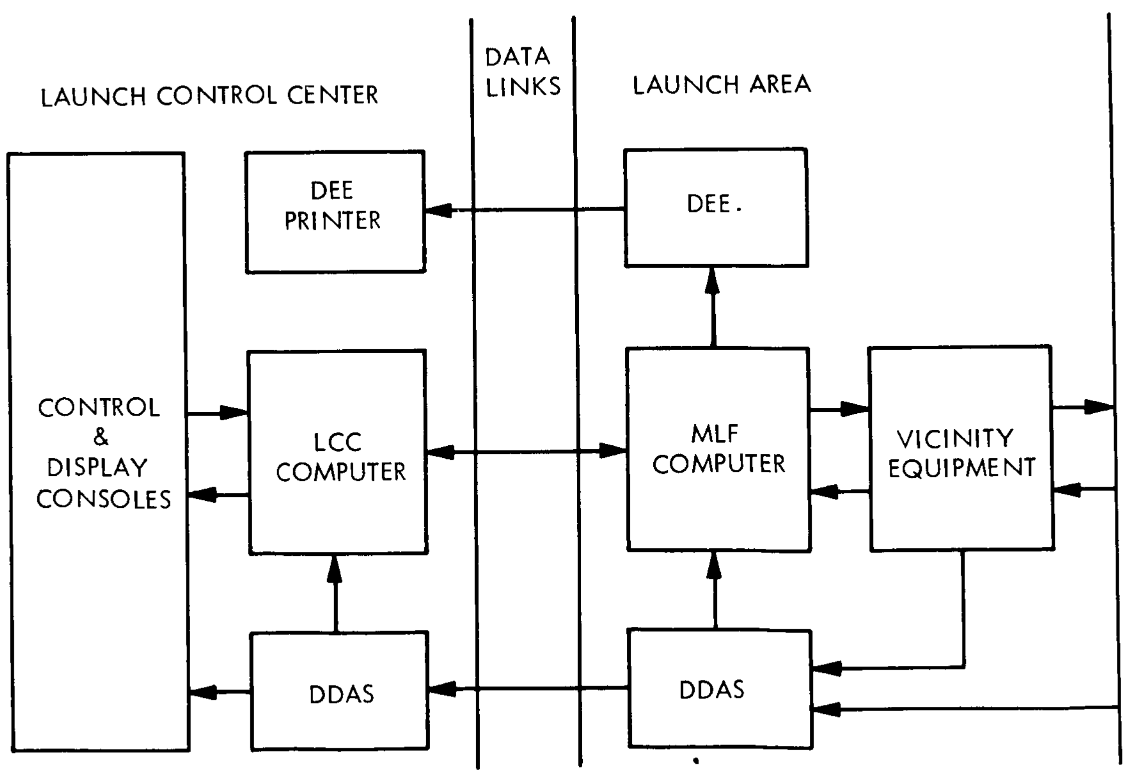


Figure 6. ESE Block Diagram

The equipment associated with this system is located (typically KSC) in the LCC and the Mobile Launch Facility (MLF). Test personnel are located in the LCC during launch operations where they control the stage test, checkout, and propellant loading from this remote site. The number of personnel located in the MLF is kept to a minimum during launch operations, but they may be available during pre-launch tests.

Commands are manually initiated in pre-programmed sequences to the LCC computer. These single-action interrupts are transmitted to the MLF computer where one or more commands are initiated and sent to the vicinity equipment or the stage. Test responses of the vicinity equipment or the stage are returned to the test operator by three methods: (1) as discretes through the return path of the computer data link, (2) as telemetered data via the hardlined Digital Data Acquisition System (DDAS) link, or (3) by the Discrete Event Evaluator (DEE). The DEE is the general-purpose computer in addition to the ML computer that indicates to the test operator any change of state that has occurred in the launch area, rejecting values which do not change. It is one example of the implementation of down-link data compression.

3.2.8.1. Control and Display

The control and display equipments in the LCC are special human-factored consoles designed to control specific subsystems of a stage and of facilities. Each panel utilizes switches to initiate the test commands, meters for analog measurement displays, and indicator lights for discrete actuations. The switches are designed with an on-off-auto capability. The on-off positions are positive interrupts to the computer for test command action. The auto mode is an inactive state for the switch, thereby enabling the pre-programmed test action by the LCC computer. Test commands that deal with stage safing and critical functions bypass the computer and are transmitted to the MLF via the hardwire data link. This mode is also utilized to display responses resulting from these hardlined commands.

The displays at the test consoles consist of recorders, alphanumeric computer - driven CRT's, as well as the aforementioned meters and indicator lights.

3.2.8.2. Computer Complex

The computer complex includes an LCC computer, an MLF computer and the general-purpose DEE computers. The LCC computer receives the manually initiated commands from the test consoles through programmable distributors. The interrupt is formatted and transmitted to the MLF computer for distribution to the stage and its vicinity equipment.

Measurements are formatted and transmitted by the MLF computer to the LCC computer for internal processing or distribution to the test consoles over the data link. All test results inputted to the MLF computer are not necessarily returned to the LCC; however, these test results are sent to the DEE where changes of state are transmitted to a printer for operator notification. The DEE thereby reduces the transmission load between the LCC and MLF computers.

3.2.8.3. Measuring Equipment

The DDAS is used primarily to transmit to the LCC display consoles the telemetered stage information that has not been brought out through the umbilical. This information is made available to the computer complex. An extension of this system augments the information exchange between the LCC and MLF in that a ground DDAS system multiplexes, transmits, and decommutates vicinity equipment, analog and discrete responses, and distributes these data to the computer complex and the test operators' consoles. This system, as in the case of DEE, reduces the test result transmission load between the MLF and LCC computers.

3.2.8.4. Vicinity Equipment

The vicinity equipment of the ESE system is utilized to distribute discretes and generate analog functions for distribution to the stage or facility equipment (such as the propellant loading facility). This equipment is controllable from the MLF computer and, in some special cases, via the hardlined safing link. Programmable patch distributors that house relay modules and special functional equipment are the principal methods of signal generation and distribution.

3.2.8.5. General

The Saturn V ESE system is tailored to the integration requirements of a Saturn V booster, rather than the system test of a spacecraft. Hence, the over-all ESE system includes functions not discussed here, such as the control of propellant loading, with its attendant equipment. These features, which are used for launch sequencing control of Saturn, are not appropriate to this discussion. The launch conditioning and control requirements of an interplanetary spacecraft (as opposed to pre-launch testing) are minor compared to launch control of a large booster system, due to the spacecraft systems being inactive at launch.

3.2.9. Concept C Implementation

Concept C has been implemented on the Mariner program. This system has been operational at KSC to verify the launch readiness of the Mariner Spacecraft.

The following is a brief description of this system intended to illustrate its significant aspects. Figure 7 is a top level functional block diagram of this system.

This approach involves three categories of equipment. Controls and displays located in the blockhouse communicate via hardlines with the SIE located at the pad. A Computer Data System (CDS), remote from the blockhouse, is in communication with the test operators via an audio link.

Telemetry data is partially processed in the blockhouse by the TLM S/S OSE for distribution to meters and displays. The TLM data is also processed at the remote system test area by the CDS (limit checking) and is then printed out to system test consoles (not in the blockhouse).

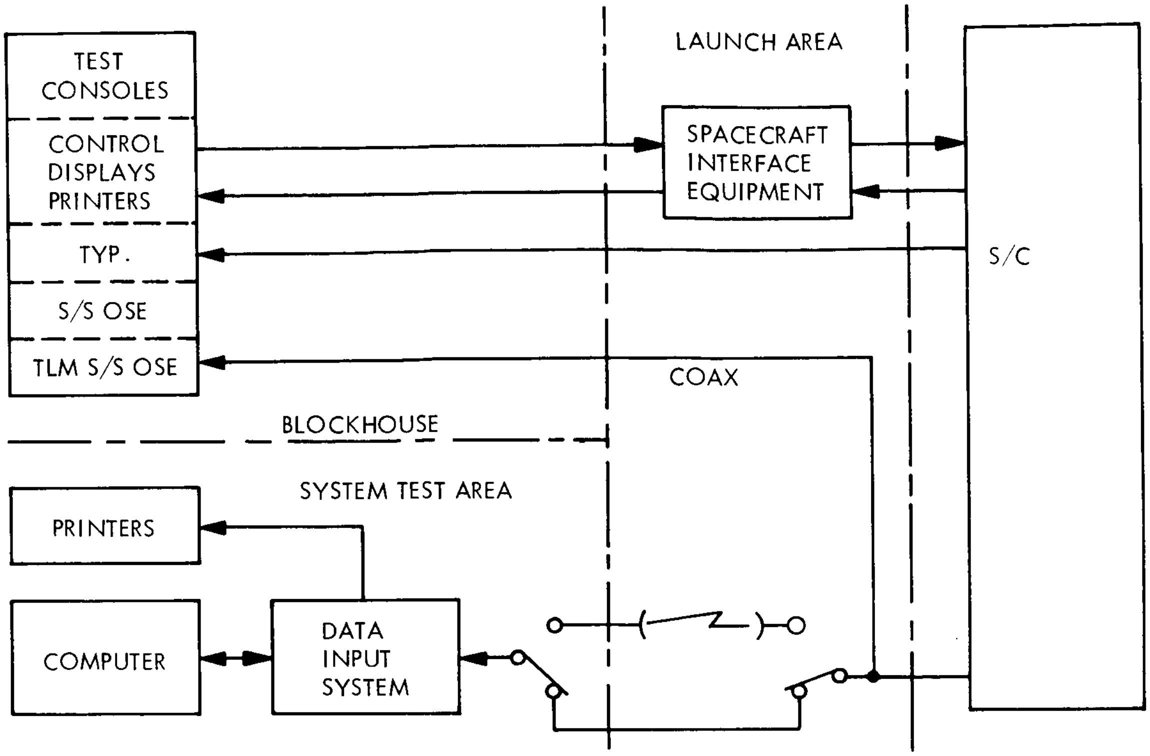


Figure 7. Mariner OSE Block Diagram

3.2.9.1. Control and Display

The test operator has manual control of launch critical functions of the spacecraft from the blockhouse. Critical measurements that are brought out from the umbilical, as well as the indications from the SIE, are displayed on analog meters and indicator lights at the test consoles.

The control and display consoles in the blockhouse are addressed to the corresponding specific subsystems of the spacecraft. The panels making up this equipment were previously used during subsystem and systems test in other test areas.

3.2.9.2. Computer Data System

The CDS consists of a general-purpose computer (Univac 1218), the data input system, and associated displays. One CDS function is data compression of the telemetry data as received from the spacecraft. Changes in the data are available to the test operators via the audio link, from the system test area.

3.2.9.3. Spacecraft Interface Equipment

The SIE consists of portions of test equipment that had to be located near the spacecraft during launch. This equipment is associated with the power, telemetry, and command subsystems.

In the launch configuration, the SIE receives commands from the test consoles in the blockhouse and performs functions such as power application or load computer memory in the spacecraft.

3.2.9.4. System Testing

Figure 7 reflects the OSE configuration at launch and pre-launch, after the shroud was installed on the spacecraft, and test access limited to umbilical and RF functions.

The blockhouse equipment (LCE) was not extensive and consisted of panels removed from the system test complex. These panels are reconfigured to support the less extensive launch support tasks.

While still in spacecraft system test phases, separate groups of control and display equipment, organized on a subsystem basis, interfaced directly (hardwire) with test connectors in the spacecraft's subsystems.

In this configuration, the CDS supported the system test by functioning as the down-link data processor. No direct computer control of up-link functions was provided. Each group of subsystem OSE (the control and display equipment groups in the system test area) had direct manual control over the corresponding spacecraft subsystem.

3.2.10. Concept "D" Implementation

The OSE concept identified as Concept "D" has been implemented, in part, as a Deep Space Instrumentation Facility (DSIF), for example, DSIF 71 at Cape Kennedy, Florida. This facility is used for tracking and communication with deep space vehicles launched from KSC. It has a limited range due to antenna sizing such that only the boost phase data is retrievable, but it is augmented by other stations around the world. The largest facility, (ranging) is located at Goldstone, California.

This system communicates with the spacecraft while it is still on the launch pad through antennas located on the umbilical tower. During RF silence, a hardlined coax could be utilized in lieu of the radio link. A top level functional block diagram of the DSIF as it could be configured to support pre-launch testing, is shown in Figure 8. Included in this diagram

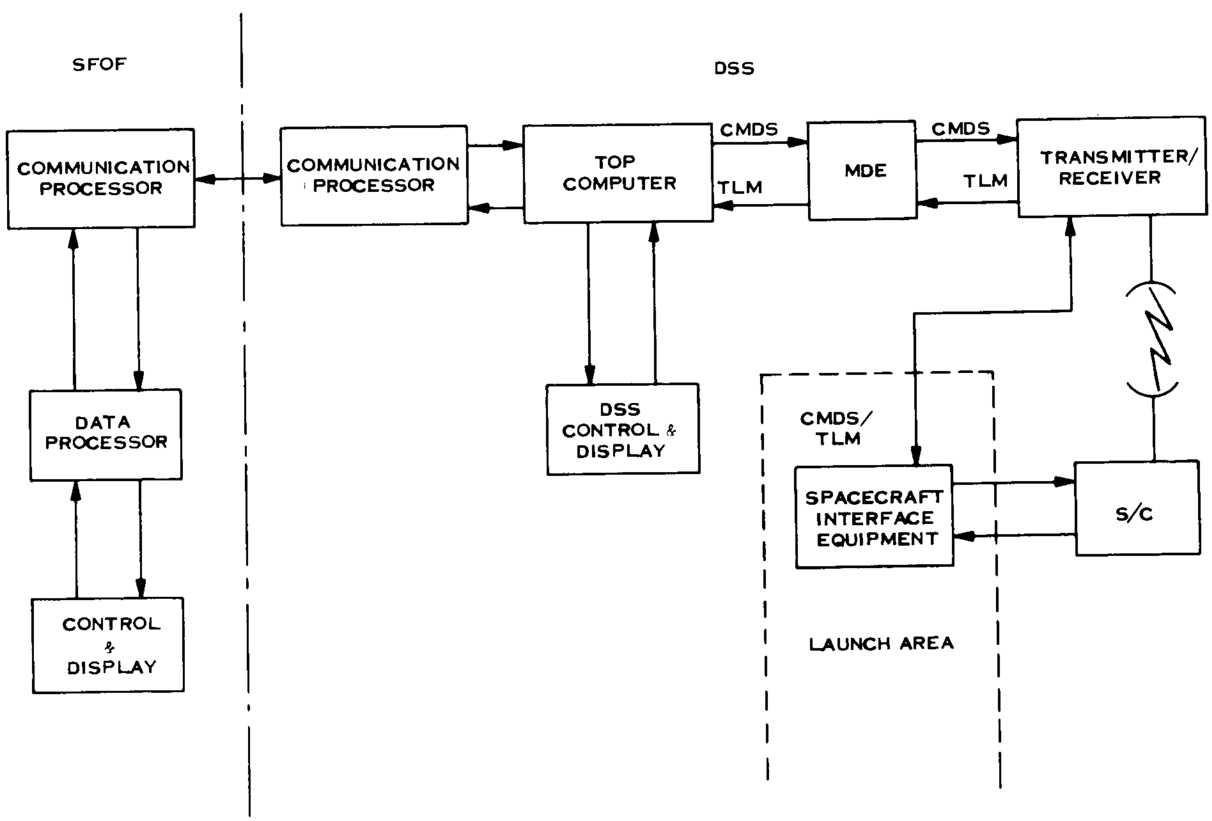


Figure 8. DSIF Block Diagram

is the interface with the Space Flight Operation Facility (SFOF) which is the master control center for all deep space operations.

3.2.10.1. Control and Displays

During preflight operations, commands can be initiated from the test consoles located in the DSIF. These commands would interrupt the Telemetry and Command Processor (TCP) computer to generate the digital format for transmission via the radio link to the spacecraft's receiving antenna. Complete loading of the spacecraft's computer memory can be performed in this way.

In the case where RF silence is imposed, transmission will be replaced by a coax from the transmitter to the SIE whereby the signal is inputted to the spacecraft as an umbilical function.

The receive mode of this concept accepts telemetry (via open loop or coax) and processes this data for display. Typical display devices such as meters, indicator lights, alphanumeric CRT's, and printers can be used to aid the test operator in evaluating the performance of the spacecraft.

This capability of control and display could also be made available in the SFOF; however, the primary control should be from the DSIF during prelaunch operations.

3.2.10.2. Computer Complex

The TCP computer has the primary responsibility for initiating up-link command data from console initiation (either SFOF or DSIF 71) and for receiving, processing, and distributing the MDE decommutated data for display and analysis. This computer is the heart of the DSIF in that it interfaces with both the local control and display and the remote master control and display system in California. It has the capability to decommutate the telemetry data and initiate the digital spacecraft commands upon console action or by computer program.

3.2.10.3. Vicinity Equipment

In normal operations, the DSIF would not require extensive equipment to interface locally with the spacecraft in that the radio link is the prime method for communicating with the spacecraft. However, this could be modified to control and monitor launch critical umbilical functions, or when necessary, to operate during an RF silence condition. Included in this equipment would be switching of discrete and analog signals into the spacecraft umbilical.

3.3. CONCEPT EVALUATION CRITERIA

3.3.1. Performance

The performance capability of an OSE concept, in terms of conducting the required tests and operations on a Voyager spacecraft, is an obvious criterion for selecting that concept most appropriate to Voyager. Though it is obvious, its application to the conceptual level, rather than to a specific design of hardware is difficult. For example, the interface and design criteria given in Section 2 can generally be met in varying degree in the actual design, regardless of the basic concept of the OSE. Considerations such as whether or not there is critical path redundancy, OSE self-check, capability to sample and limit check spacecraft parameters, etc. are almost entirely design dependent, not concept dependent.

Performance capability of one concept with respect to another is not an abstraction, but is nevertheless difficult to assess in a quantitative fashion. In a semi-quantitative sense, it is possible to examine basic aspects of the concepts, compare them against suitability to perform against Voyager spacecraft unique requirements, and emerge with a relative ranking of the suitability of the concept for application to the Voyager checkout and launch control problem.

3.3.2. Availability

In the launch operation, since the launch window is limited, the availability of the OSE used to support the prelaunch and launch operation becomes an important factor. Availability is defined as the probability, as a function of time, that the OSE is functioning correctly at that time. It can be more simply defined as $\frac{\text{MTBF}}{\text{MTBF} + \text{MTTR}}$ or $\frac{\text{Reliability}}{\text{Maintainability} + \text{Reliability}}$

In examining these concepts to determine relative availability, it is assumed that they can be designed with equal maintainability (MTTR). The availability then becomes simply the reliability.

The reliability of a concept is also a difficult aspect to evaluate quantatively. The reliability of an actual system, or of a design can be assessed quantitatively, but reliability numbers cannot fairly be applied to a concept. There are, however, aspects at the conceptual level which have some relation to the probable relative levels of reliability of future implementations. These include:

a. The relative degree of functional redundancy inherent in the concept; that is, the degree of remaining capability when an outage of some function is assumed to have occurred.

b. The extent and complexity of interfaces within the system. An extended (as opposed to integrated) system which has a wide physical dispersion of interfaces can be assumed to promise less reliability than a system with fewer, simpler, less dispersed interfaces. The complexity of these interfaces can also be used as an indication of relative reliability.

c. The "size" of the system that one concept would require with respect to another, and the relative complexity of components can also be used to indicate relative reliability.

The concepts should be evaluated with respect to availability, primarily from the launch support rather than system test configuration viewpoint. The importance of availability during a spacecraft system test is less than it is when a prepared spacecraft is on the pad approaching launch.

3.3.3. Flexibility

The relative flexibility of each concept is a criterion by which it may be evaluated. Flexibility is the inherent capability of the OSE to accommodate changes in the flight article design, test procedures, and interfaces. It is assumed that the system design, based on any selected concept, will have been done with equal degrees of inherent component flexibility; i.e., same degree for parameter value excursions, same degree of preference for infinite control rather than discrete step control, etc.

3.3.4. Technology Impact

The degree to which reduction of an OSE concept to a functioning system will require the development of new components or new techniques, is an inverse relative criterion for evaluation. It is very preferable to have conservative, established, well understood OSE, which uses components and techniques that have established and known characteristics. This makes for greater confidence in indicated test results, as problems can be more readily assigned to the test system or the flight system.

3.3.5. Facilities Impact

The 1973 Voyager is constrained to the unique facilities associated with the Saturn V launch vehicle. The impact of each concept upon these facilities is considered to be an evaluation criterion. The concepts should be evaluated as to the relative degree of adaptability to this established facility, as it is obviously desirable to be able to implement the Voyager OSE system with least impact upon the existing facilities which are needed by other on-going programs using these same facilities. The relative impact upon other facilities, such as the spacecraft contractor's plant, the deep space network and special test sites, is also a factor.

3.3.6. Cost

The cost factors inherent in implementing an OSE system should be an evaluation criterion.

3.3.7. Weighting

The fundamental figures of merit to be used in evaluating OSE concepts, stated above, can be mutually dependent. For example, increase in reliability (desirable) is usually obtained at increases in cost (undesirable), other considerations being equal. Therefore, these figures of merit should be applied in a specific precedence and with a s)ecific weighting which, in turn, are derived from inherent requirements of the total program.

The order of priority and weighting of figures of merit, which are intended for use, follow:

Performance	10
Availability	8
Facilities	8
Flexibility	5
Technology	5
Cost	2

The top priority and maximum weight given to performance is a way of stating that performance of the OSE is an absolute requirement. The compromises and trades between deciding on a given test requirement versus deciding to make the OSE implement it, which may be open as a calculated risk or test policy on certain programs, is precluded from the Voyager program. This is a consequence of having a high operating reliability requirement for a complex flight article, for a uniquely long duration in the hostile environments of the mission.

The other figures of merit are not necessarily as inflexible as performance, since any adverse effect on the total mission or program is a more manageable "judgment" type of decision, for which there is some established precedent. Therefore, the weighting of all of the other factors can be considered tentative and open to discussion and modification. Availability, despite its high suggested weighting, is not considered as absolute as performance, since, assuming that launch control capability is retained, an unavailability of OSE would probably not be considered sufficient reason not to launch if the tail-end of the launch window has arrived.

At the end of the precedence listing, cost has been tentatively allocated least precedence on the assumptions that the order of magnitude of OSE-imposed costs on the program will not vary total program costs too much, regardless of the choice of OSE concepts.

In the event that this assumption is not valid, or that over-all cost cutbacks are imposed, then the precedence and weighting of the "cost" figure of merit would have to be revised, possibly to the extent that it too would have to be treated as an "absolute" rather than "relative" quantity, and no longer be considered a trade parameter.

3.4. CONCEPT EVALUATION

3.4.1. Performance Evaluation

The relative performance suitability of each concept for on-pad spacecraft support is, in essence, the suitability of two characteristics; the compatibility of the concept for implementing remote control of pre-launch and launch operations, and the suitability of the concept in implementing the requirement for by-pass of launch critical functions. Concepts "A", Central Computer Control, and "B", Distributed Computer Control, are well suited to remote control. Concept "B" is perhaps too well suited, as the dual computer complex and its inherent control and command compression characteristics are needlessly powerful for the relatively simple control and monitoring job that can be done on the enshrouded, quiescent spacecraft. (Even if greater access were available on pad, exercise of the spacecraft and excessive monitoring provisions, which are the reason for considering compression schemes, would tend to prejudice the spacecraft's flight readiness status). The two hardwire concepts, "C", the blockhouse approach and "D", the operational approach, on the other hand, are very suitable for hardwire control and monitoring of Voyager's launch critical functions. Therefore, each concept is equally suitable for launch pad implementation, with concept "B" being, perhaps, too much for the job at hand.

Performance capability for detailed system level tests of the unshrouded spacecraft, using all of the direct test point access, as well as the limited umbilical, to determine or verify the exact characteristics of a spacecraft, reduces to relative suitability of each concept for the following:

a. Trouble shooting, by engineering level operators.

b. High frequency and transient analysis.

c. Ability to analyze unexpected interactions or events and compensate quickly to avoid damage.

d. Ability to reconstitute tests for detailed analysis of spacecraft performance, including trend recognition.

The two hardwire concepts, "C" and "D", are considered to be more suitable for this type of operation, although the two remote control concepts "A" and "B" lend themselves more conveniently to test reconstitution and analysis. The relative suitability ranking in descending order is:

A	3
B	2
C	4
D	1

3.4.2. Availability

The concepts can be ordered in terms of relative availability by considering the nature of the computer complexes, umbilical control and display, and gross size and complexity for the case of application to pre-launch and launch operations.

Concept "B" requires a series arrangement of computer complexes in what can be considered a critical path (reliability block diagram) for the control of telemetry and command functions.

Thus, the probable availability of the system can be expected to be lower for this concept. In a similar fashion, concept "A" has its one computer complex as a critical item, where its outage would result in loss of telemetry and command functions (ignoring here the fact that DSIF No. 71 constitutes a back-up).

Insofar as umbilical control and monitoring functions are concerned, concepts "C" and "D" have advantages as a computer outage will not affect these functions. Insofar as outage of any particular console, all the concepts are equivalent.

In terms of gross size and complexity, concept "D", which utilizes the DSIF No. 71 in a check-out mode, enjoys an advantage. This is due to the assumption that DSIF No. 71 being on line is a program launch requirement (except as a last extreme measure at the very end of the launch window). Therefore, this concept does not need a functional duplication of the DSIF station and is, to that extent, capable of being implemented in a less extensive, and hence more inherently reliable design.

The relative ranking of concepts for availability is:

A	2
B	1
C	3
D	4

3.4.3. Flexibility and Technology Impact

The concepts are considered to be essentially equal with respect to inherent characteristics bearing upon both flexibility of application and requirements for component or technique development.

3.4.4. Facilities Impact

Two general areas in which facilities impact of the concepts are considered are KSC facilities associated with use of Saturn V, and other areas such as spacecraft contractor's plant, test sites, etc. The former is of much greater consequence than the latter.

Insofar as impact upon the LC 39 and other KSC facilities, the operational concept "D" requires greater modification than the other concepts, as not only LC 39 is affected, but the DSIF No. 71 station would have to be modified to accommodate the launch control equipment. Concept "A" if it utilized Apollo ACE equipment even though it requires more equipment in the immediate vicinity of the pad than "C", is the concept which could be implemented with least facility impact, if the MSOB at KSC can be utilized for Voyager.

Both concepts "B" and "C" would have more impact on the existing facilities, "B" by having greater requirements for accommodation of equipment in the immediate launch area, and "C" by its requirement for new accommodation of its control/computer room-type of equipment.

The relative ranking of the concepts for minimal facility impact, at KSC, is:

A	4	
B	2.5	essentially equal
C	2.5	essentially equal
D	1	

3.4.5. Cost Evaluation

Although actual cost estimation and comparison of costs in a system trade-off is dependent upon having fairly firm designs as a data base, a relative ranking of costs can be made at the conceptual level by considering the type and extent of hardware and software required to implement a concept.

Concept "C", especially in a configuration to perform system tests, as opposed to pre-launch and countdown, appears to be the least cost approach. This is based on the prediction that computer and software costs would be less than the "remote" concepts, "A" and "B", as the computer complex is not on-line for all operations and the extent of its involvement in up-link control can be less. In addition, it is reasonable to assume that control and display console interface equipment for mating to the computer complex, would also be less in both complexity and scope. Finally, the sensing, stimulation and conversion equipment forming part of the SIE will probably be less costly in relation to the degree to which the system is essentially a manual system, for dominant operating mode.

Concept "D" enjoys these same advantages, but is probably somewhat more costly because the DSIF station must be simulated for system tests, and modified to accommodate blockhouse-type equipment for launch operation type of tests.

Concepts "A" and "B", the two remote operation concepts, appear to have equal cost impact, except that "A" has the advantage in having only one computer complex to implement, and hence less software also.

The relative cost rankings of the concepts assuming complete development, fabrication and implementation cost, are:

A	2
B	1
C	4
D	3

3.4.6. Evaluation Summary

Using the relative weights for the several evaluation criteria and the relative rankings discussed above, the following table indicates the apparent applicability of the candidate concepts:

Table 1. Numerical Selection Matrix

Criteria	Concept Evaluation Matrix								
		Concepts							
		A		B		C		D	
	Wt	Rank	Wt	Rank	Wt	Rank	Wt	Rank	Wt
Performance	10	3	30	2	20	4	40	1	10
Availability	8	2	16	1	8	3	24	4	32
Flexibility	8	2.5	20	2.5	20	2.5	20	2.5	20
Tech Impact	5	2.5	12.5	2.5	12.5	2.5	12.5	2.5	12.5
Fac. Impact	5	4	20	2.5	12.5	2.5	12.5	1	5
Cost	2	2	4	1	2	4	8	3	6
TOTAL			102.5		75		117		85.5

Concept Selection Preference List

1	C	"Blockhouse" concept, hardwire approach
2	A	Central Computer Control Concept
3	D	"Operational" concept, DSIF with blockhouse equipment.
4	B	Distributed Computer Control Concept

The preceding indication of concept applicability to Voyager is only as valid as the weighting factors which were assigned. Since these have been done in a highly intuitive and arbitrary manner, the confidence that may be placed in the results is low. For instance, in evaluating relative costs, total new system development was assumed. A realistic approach would utilize as much existing hardware and design as practicable, but this is not definable at this time. It is therefore considered appropriate to use a deductive approach, oriented to over-all suitability, which would not be as sensitive to errors in weighting factors as the preceding. The following discussion is organized into two distinct sections: one dealing with launch pad applications, and the other dealing with KSC system testing of the spacecraft prior to its integration with lander, shroud, and launch vehicle.

3.4.7. Launch Control Equipment

3.4.7.1. Concept "B", Distributed Computer Control.

This concept's distinctive characteristic is data compression in both the uplink (accomplished by the MLF computer) and in the downlink (accomplished by the DEE's). Other than this, this concept is essentially the same as Concept "A". Note: ACE's decom can perform data compression, but this is done remotely from the spacecraft area whereas ESE does it locally). The question therefore becomes, "is data compression required or desirable for Voyager?"; "LCE or STE as well?" The only possible use for this at the LCE is in the loading of C&S memory. This requires transferring from the LCE to the spacecraft $512 \times 64 \approx 33{,}000$ bits. By having this data programmed and stored in the spacecraft vicinity, it can be loaded into the spacecraft by one command from the remote control location. While this certainly reduces the amount of data to be sent from the remote location to the vicinity equipment, it is an unsatisfactory solution because (a) the data must be preprogrammed and the capability to make last minute changes must be available thus requiring the control capability from a remote area, and (b) there is no problem in sending that data over a data link that handles 10^6 bits per second (bps) with an error rate of 1×10^{-8} that is detectable and correctable such that the eventual error rate approaches zero. The main STE problem in supporting launch complex equipment is in the downlink where a data rate of 500 to 1000 bps will be required. This also is trivial and there is nothing to be gained by data compression in the vicinity of the spacecraft as opposed to doing it remotely.

Therefore, the complexity is not warranted for Voyager and concept "B" can be eliminated directly. This, however, should not rule out the occasional use of decision-making elements in the vicinity equipment in order to reduce the sampling rate of some signals. (Note: the decision and low rate sampling of the input signals would always be required.) An example of this for Voyager might be the sub-bit by sub-bit comparison of transmitted spacecraft command data with detected spacecraft data. The point is - this should be the exception and not the rule and therefore not basic to the over-all concept.

3.4.7.2. Concept "D", (Operational)

This concept is the use of the in-flight ground station or its simulated equivalent as the nucleus of the LCE and the STE.

3.4.7.2.1. DSIF No. 71

Since there is a requirement for the DSIF No. 71 at KSC to be in r-f lock with the spacecraft at the time of launch and since that station can command and monitor the spacecraft, it can inherently perform most of the launch critical functions. Using this station will also bring the DSN into the picture earlier and will therefore make an easier transition of responsibility from KSC to the DSN.

There are, however, several disadvantages as follows:

a. It requires umbilical control and monitoring capability for the launch critical functions. This must be added using one of the other concepts, such as concept "C", blockhouse-type hardwire control and display equipment, preferably located within the DSS.

b. No alternate path is available.

c. Additional TLM display and convenient command capability will be required within the DSIF station.

d. It requires prolonged support from DSIF personnel.

It is felt that these disadvantages outweigh the advantages especially when the alternative is to require the other concepts to use the DSIF No. 71 and SFOF as their backup means for commanding and monitoring the spacecraft during the launch critical phase or any other prior period at KSC.

3.4.7.2.2. DSIF Simulation

An alternative to the above was that the LCE/STE TLM and command functions should be accomplished using a DSIF simulator. This alternative become attractive when there is a considerable amount of MDE or when the spacecraft to MDE compatibility has not been verified adequately prior to this stage.

MDE needed by DSIF stations has been identified as TLM demodulator, command modulator, and software for the TCP. The spacecraft interfaces with the DSIF antennas and its transmitter/receiver which in turn interface with the MDE modulator and demodulator. It is impractical to duplicate this entire path and, therefore, it is impractical to verify the MDE hardware's interface with the transmitter/receiver. Also, the output of the TCP (SFOF side) is the communications terminal equipment which becomes difficult and expensive to duplicate. Because of these holes and since it is felt that the amount of MDE is small, it is not recommended that the LCE nor the STC simulate the DSIF. This can be supported by stating that the MDE spacecraft interface should be verified to the following degree:

a. The PTM vehicle can be used to establish complete DSN/spacecraft compatibility by being used at Goldstone for this purpose.

b. One TCP will be available at the factory for debugging software. This setup will be used off line during system tests.

c. Taped factory system test data (TLM demod input) will be sent to Goldstone for playing thru the entire DSN.

d. Taped command data (output of Goldstone command modulation) will be obtained and used during some portion of the factory system test.

e. DSIF No. 71 will be tied in via hardlines or RF links during systems tests in spacecraft checkout facility at KSC.

The above is not intended to preclude the use of MDE or TCP simulation for system testing or in ACE if that is the cost effective approach. Its intent is to state that it is not required to be done in that manner.

3.4.7.3. Concept "C" (Blockhouse)

This concept involves the use of hardwires on essentially a one-to-one basis per control and monitoring function to connect the umbilical or transfer room equipment to the launch consoles. When it is used in conjunction with a computer, it would take on the characteristics of General Electric's Task B OSE preliminary design.

This concept is attractive with Voyager, especially in view of the MSFC ground rule that hardwire circuits should be used in parallel with data links for critical circuits. All concepts will require hardwire circuits and Voyager launch operations require relatively few wires per spacecraft (12 discrete controls - 3 digital signals (TLM, command, and command received verification), and 20 analog signals) for the launch critical functions. In addition, two control signals and nine analog signals are required for test purposes, but these do not have to be brought back to the remote area.

This concept is shown in a gross level in the block diagrams in the Appendix B, and includes the implementation of the hardwire circuits for one spacecraft.

3.4.7.4. Concept "A", Central Computer Control

This concept is well documented; the ACE implementation is shown also in the appended block diagram. It should be noted that the launch critical functions are hardwired using the same circuits as above except that they are redundant. This last function is performed by the data links.

3.4.7.5. Comparison

Concepts "A" and "C" are the two which are applicable to Voyager launch operations. It is expected that a design level trade study would indicate the following:

a. Performance - Equal

b. Availability - Both more than adequate but hardwire concept better

c. Facilities Impact - Equal

d. Flexibility/Growth - ACE better

e. Technology - Equal

f. Cost - Dependent upon utilizing existing hardware.

3.4.8. System Test Equipment

Based on the foregoing rationale, the two remaining concepts, "A" (the Central Computer Control approach) and "C" (the hardwire or blockhouse approach) should be examined from the point of view of applicability to serve as system test equipment. The choice between these alternatives is determined by their adaptability to this task, as both seem equally applicable to the launch control task.

In comparing the two, concept "A" will be considered independently, instead of being a new application of the existing ACE hardware at KSC. This latter aspect is reported in Volume IV as a separate study.

3.4.8.1. Performance Characteristics

a. Automatic Testing - This is the same for both concepts since essentially identical data flow paths are used.

b. Manual Testing - Concept "C" can do this fully independent of the computer or use the computer as a data analysis aid or for control assistance as desired. Concept "A" requires the computer to be on line. This gives Concept "C" a reliability advantage and a degraded mode advantage, but the performance is probably essentially equal. (Note: in concept "C", the computer can be used for monitoring the switches, etc. of the consoles and can use this information to output remote control-type data.)

c. Troubleshooting or Unplanned Operation Mode - Troubleshooting the OSE will be simpler with the concept "C", since one operator will have controls and inputs/outputs in one location whereas concept "A" will require two separated operators to coordinate their actions and responses using voice communication. Troubleshooting the spacecraft will probably be easier with concept C since two operators would be within sight and hearing (perhaps in different rooms but with windows and phones) whereas in concept "A", operators will be out of sight of each other. Once again, the factor of having the computer in or out of line favors the concept "C" approach for this mode.

d. High Speed Data - This data, which includes all of the RF data as well as pulse measurements of solenoid valve signatures, command output pulse wave shapes, (the wave shape analysis must be obtained locally and cannot be remoted to the control and display consoles). There is no reason why personnel cannot be stationed in the vicinity to take these measurements if the concept "A" approach is used, but the operators would become scattered. These factors favor concept "C" as to performance. The difference is not overwhelming, but one of degree.

3.4.8.2. Availability

As stated above, the main difference is in the manual and troubleshooting mode where concept C can be operated without the computer. This is not only a hardware availability advantage but a software independence advantage also. This latter should be only of significance during the early phases when the software may not be developed to the desired degree. Later in the program, this should be a lesser problem. Concept "C" has superior availability to concept "A".

3.4.8.3. Facilities Impact

The facilities impact can be expressed in relative square feet required at the various equipment locations. The power, lighting, cooling, etc. for these purposes are directly proportionate to square feet also.

	Concept "A"	Concept "C"
Immediate Spacecraft Vicinity	SC	SC
Vicinity Equipment	V	2V
Computer Group	C	C
Display/Control	2V	-

The immediate spacecraft vicinity and computer group are identical for each approach. The vicinity equipment for concept "C" requires twice as much floor space as for concept "A". This is based on the experience that human factoring consoles govern the size of the equipment and that the consoles end up being loosely packed. For the purpose of this trade-off, a factor of 2 to 1 was chosen which may be high.

Concept "A's" control and display groups must therefore be the same size as the concept "C" vicinity equipment, since it contains the human-factored consoles. Concept "C" does not require this equipment, since it is a part of its vicinity equipment.

The results are that concept "A" requires more total floor space than the concept "C" approach (V additional square feet) which implies it needs a greater amount of false floor, lighting, cooling, etc. The main difference is that concept "C" requires more in the vicinity of the spacecraft (within 50 feet) which may be harder to come by.

It should also be noted that when the spacecraft is moved to different test locations within the same major facility, the vicinity equipment must also be moved. This means concept "A" requires less equipment to be moved. With one move considered (support two spacecraft test areas), the total floor space required for each concept would essentially be equal. The net results would appear to be a stand off.

3.4.8.4. Flexibility and Growth

There is nothing inherent in either concept that limits flexibility and growth.

3.4.8.5. Technology Impact

There is nothing inherent in either concept that requires more development, testing, etc. than the other. This is true since all data paths, etc. can be accomplished using identical circuits, techniques, etc.

3.4.8.6. Cost

The concept "C" approach costs less than a new ACE by essentially the cost to regenerate the displays at the remote console group. Software costs are expected to be higher for concept "A", (still assuming design through implementation costs).

3.4.8.7. Compatibility with LCE

Both concepts are inherently compatible with either LCE approach. Concept "C" can be operated in the ACE mode by having the computer use the console switch data (which it is always receiving) in a like manner, i.e., output remote control type signals. Conversely, there is no reason why Concept "A" cannot be hardwired to the vicinity equipment.

These considerations lead to the selection of Concept "C" as the preferred approach for system testing. On the next section, the application of the concept to the Voyager KSC launch program is discussed, and a preferred implementation of a combined concept is shown.

3.5. CONCEPT IMPLEMENTATION APPROACHES

The preceding discussion indicated that concept "C" (the hardwire, "blockhouse" approach) and concept "A" (the approach used by the Apollo program for ACE) were logical concepts to consider for actually implementing an OSE system for checkout and launch control of the Voyager spacecraft. In implementing the preferred concept, the characteristics of the environment and the nature and objectives of the testing should influence the design.

The system tests of the spacecraft, to determine performance levels, capability under conditions of controlled stress or degradation, interactions, etc. will generally be performed in an environment relatively free of requirements for remote operation, and with maximum opportunity to utilize hardwire for access to test points. Under these conditions, there are many fewer constraints on the degree and manner in which the flight hardware can be operated. Therefore, for spacecraft system tests at the factory system test area, factory environmental test chambers, other spacecraft test facilities, and KSC spacecraft system test area, the Voyager system test equipment can be implemented per concept "C" in a straightforward manner. It could be built into trailers for mobility between these locations. Figure 9 is a block diagram showing how the concept could be implemented at these locations.

This approach to the STE design is a direct implementation of concept "C" and is well suited to Voyager in that it takes full advantage of extensive direct hardwire access to the spacecraft. It is best suited to use by engineering oriented personnel to perform this level of test. Each engineer has direct hardwire control over his flight subsystem, and also gets a direct, unprocessed display of data. He also gets processed data which has been first sent to the computer complex. During coordinated system testing, the computer complex should be utilized for control of certain kinds of precision, sequence or time critical inputs under the executive control of the test conductor.

The most desirable approach to implementing the OSE for on-pad prelaunch and launch control, is the approach which has the greatest compatibility with the distinctive physical features

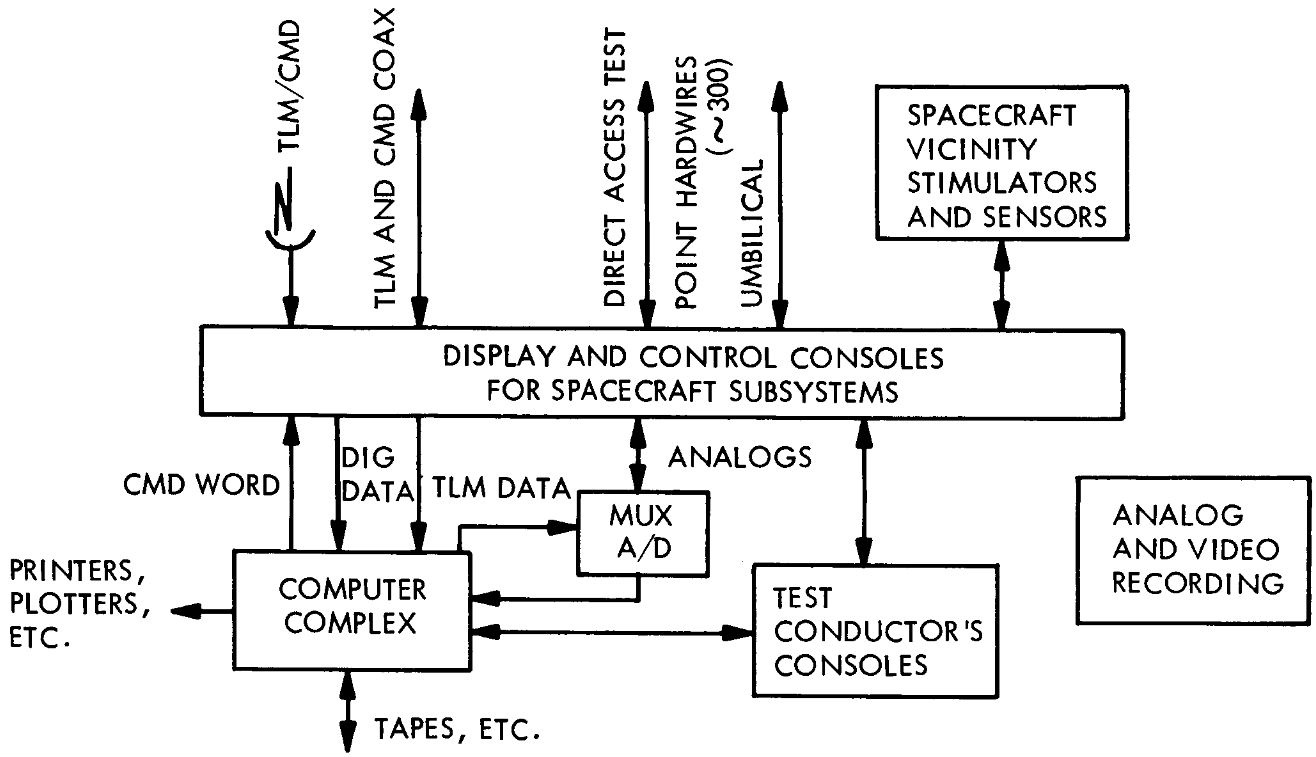


Figure 9. System Test Equipment

of LC 39. Conceptually, it can be considered either a version of concept "A" or "C". One implementation approach is illustrated by the block diagram of Figure 10.

This is an application of the blockhouse concept and uses hardlines for all launch critical control and monitoring. The umbilical is limited to this category. The LC 39 unique feature is that the LCC firing room replaces the "blockhouse", and the umbilical wires (which are variously estimated at 20 to 40) are extended just as in the case of the Saturn V launch vehicle critical hardwired circuits. Its major drawback is that it is dependent upon some external facility for completion of the TLM and command loop to the spacecraft. This may be the operational ground station, DSIF No. 71, or the telemetry and command capability of the system test equipment (exact location at KSC not important). This ground station would have to receive command requests and deliver telemetry data by either data link or by hardwire. Limited analytical capability will exist, unless the LCC functions are tied directly back to the STE, somewhat similar to the General Electric approach generated during Task B. The reason for this is that except for display of telemetry information, the consoles in the LCC

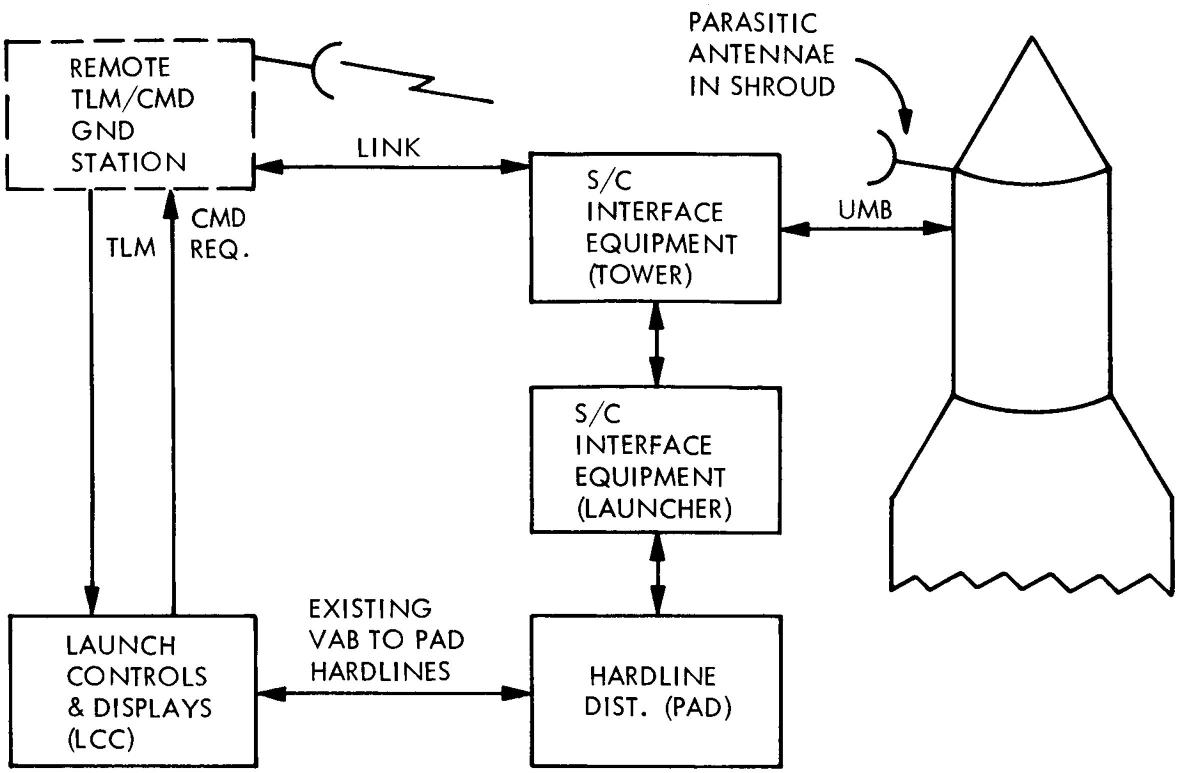


Figure 10. Launch Control Equipment

would have to be sized and allocated in accordance with their critical, hardwire control and monitoring functions rather than in accordance with secondary requirements for maximizing use of analytical capability.

The preferred implementation, which overcomes this drawback somewhat, is shown in Figure 11. Here, the LCC and other spacecraft launch control equipment are made a functional part of a launch system which includes the remote STE.

This principal mode of control and display is through the use of the STE's computer complex, as in the ACE concept. Command requests are sent by link to the STE, which sets up the SIE, issues and checks the command, receives and processes the response (TLM) and drives the LCC displays (alphanumeric computer driven displays. In accordance with launch area practice, all of the launch critical controls and displays are hardwired to the LCC. This is the backup mode.

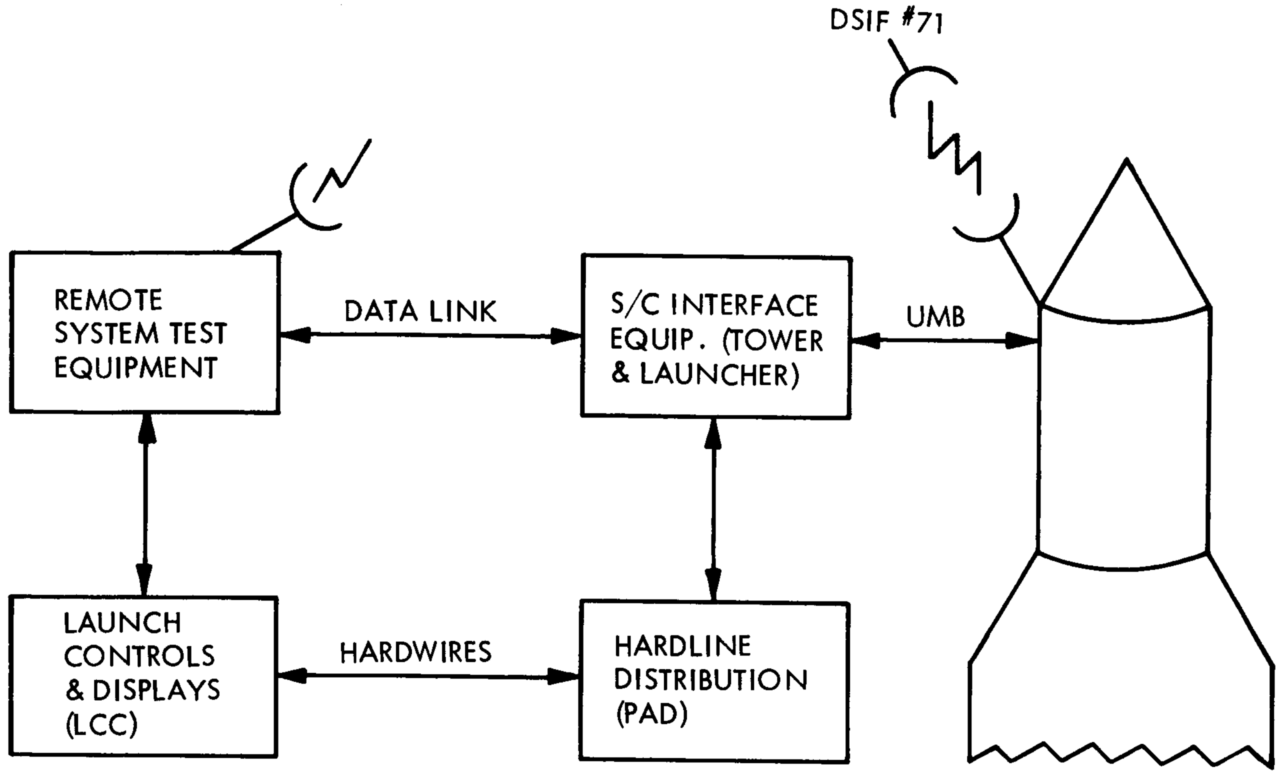


Figure 11. Launch Control Equipment

Additional functions are needed in the SIE (to remote TLM to the STE). Also, the LCC equipment will need data link terminal equipment and computer driven display equipment in addition to the hardwire controls and displays. The DSIF No. 71, shown in Figure 11, provides redundant back up to the STE.

This approach provides for maximum utilization of the OSE capability represented by the STE. It complies completely with the requirement for hardwire backup. It is felt that the added capability to analyze anomalous behavior, detect late appearing trends, etc. inherent in the tie in to the STE, is worth the penalties of tieing up data links and adding equipment to the SIE. This approach is therefore preferred.

It should be noted that the DSIF No. 71 can provide redundancy to this configuration also.

3.6. PHYSICAL DESCRIPTION

It is impractical to attempt a detailed physical description at this time because the spacecraft hardware design is not complete enough to permit an accurate preliminary design of the OSE. However, during Task B in 1965, estimates of equipment quantities were made. These estimates were based on prior experience in designing equipment of similar complexity. It is considered appropriate at this time to extrapolate from the Task B physical description to reflect the changes generated in the spacecraft design during Task D. The quantities remain, however, an estimate based on prior experience rather than detailed preliminary design.

The physical description of the equipment defined in the Task B study report resulted in some 80 racks of equipment required for performing a complete system test. This total included the Computer Data System (CDS), STC ancillary equipment, Test Conductor's Console and all subsystem test equipment which had been integrated into the STC. This hardware approach assumed that the subsystem test equipment was brought into the STC in essentially the same configuration as used during subsystem tests with a few discrete articles of hardware discarded because they were not needed in system test. The functional diagram of this approach appears in Figure 12. This approach results in a larger STC than is really required.

The approach on Task D has been to assume that the STC equipment is designed primarily for system test use and little if any capability is provided for, other than that required to implement the system test requirements. The quantities of equipment required for system test were compared. A rack count of the Task B design was made. A second rack count using the Task B functional block diagram, Figure 12, was made assuming the Task D approach of packaging the functions for system test. The results are tabulated in Table 2. Of course the rack counts for the CDS and other STC peculiar equipment stay essentially the same. These are affected mainly by test support requirements and packaging design and not by differences in the Task B/Task D approach.

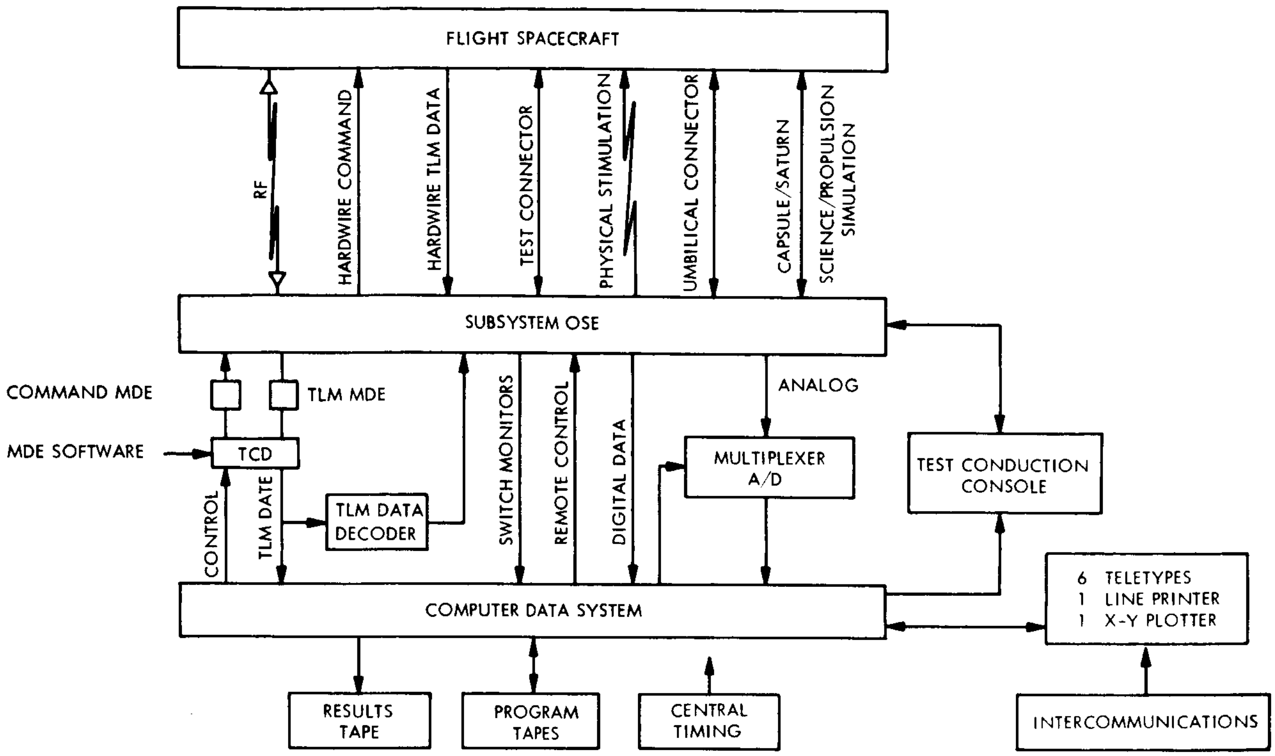


Figure 12. Task B STE Functional Schematic

Table 2. Comparison of STE Equipment Quantities

Type	Task B Racks		Task D* Racks	
CDS	17		15	
System Test Integration				
TCC	5		5	
Ancil. Equip.	8		8	
		30		28
Subsystem Support				
Power	3		2	
Radio	4		4	
TLM	5		} 1	
Data Storage	2			
Command	2			
C&S	4			
G&C	5 + Stimuli		3 + Stimuli	
Propulsion	2			
Pyrotechnic	2		1	
Thermal	1			
Simulators	5		5	

Table 2. Comparison of STE Equipment Quantities (Continued)

Type	Task B Racks	Task D* Racks
TCD	7	-
Science	5	5
Capsule	4	-
	51	21
STE GRAND TOTAL	81 Racks	49 Racks

*Based on Task B design but different packaging in system test.

The subsystem support equipment, the functions of which are accomplished by control and display equipment plus spacecraft interface equipment in the Task D concept, provided the biggest opportunity to consolidate equipment, particularly since most of the subsystem racks contained equipment not needed for system test. Most of the reduction in equipment comes from a change in packaging, but some also comes from a change in requirements. The rationale for reduction follows:

a. Power - Deletion of several test assemblies should result in a reduction from three racks to two.

b. Radio - Radio equipment cannot be readily combined with others so no reduction was assumed.

c. TLM
Command
C&S
Data Storage - Many of these functions will be performed by the CDS and the remaining functions will be combined in one integrated rack of system test equipment. TLM decommutation, command generation, and memory loading are taken over by the CDS. The remaining functions are minimal.

G&C - Stimulation equipment remains unchanged but some extra monitors can be combined elsewhere. Reduction of five racks to three racks.

e. Propulsion
Pyrotechnic
Thermal - Primarily monitoring functions which can easily be combined into one integrated rack.

f. Simulators - These simulate spacecraft interfaces with other systems and are independent of STC packaging approaches.

g. TCP - The computer (CDS) will include the mission dependent functions of the Task B Telecommunications and Command Data Processor (TCP).

h. Science - Five racks of equipment were assumed for support of the science subsystem. There is no change in the assumption.

i. Capsule - The capsule is assumed to use its own STE.

The physical configuration arrived at with this reduction in rack count appears in Figure 13. The LCE is included because it is still assumed that the Task B approach to launch was taken, although the LCC rather than MLF is considered to be the appropriate location for critical function hardwire controls and displays.

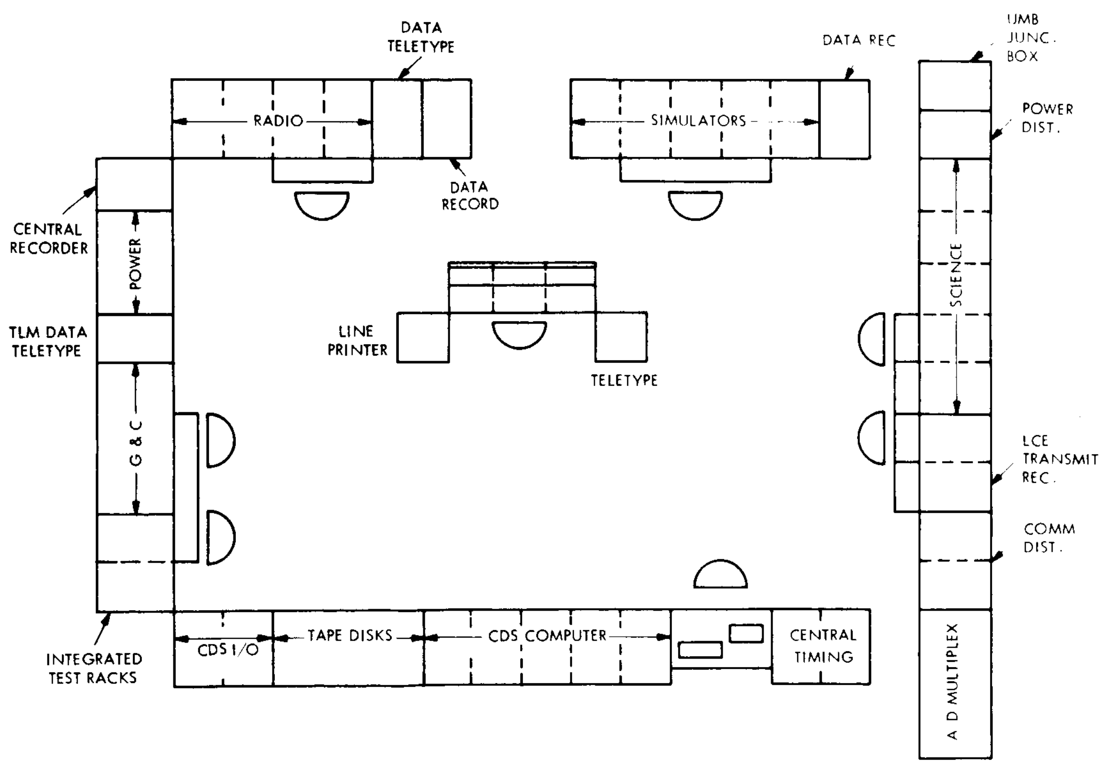


Figure 13. Task D STE Layout Diagram

3.6.1. LCE-STE Interface

Although the functional requirements on the LCE are essentially the same, resulting in similar rack counts of LCE for Task B and Task D, the shift in emphasis on the location of executive control causes changes in the interface between LCE and STE. This is to be expected because launch complex equipment is always sensitive to equipment location and interfaces cannot be completely defined until final decisions are made on the location (building) of all LCE.

In Task B the focal point of control was at the STE location. The Task D studies indicate that the major focus of hardwire critical function control should be at the LCC, in the firing room. This change of focus requires that more information be displayed and that more control capability exist in the firing room. The staff of experts would probably still be located at the STE to review TLM data and offer advice through the intercom systems but actual operations involving "manual" control of umbilical functions would be implemented from the firing room. Since TLM data must be available in the firing room, a data link, similar to the Apollo data link, must exist or the entire STE must be moved to the LCC thus becoming LCE. Either approach is feasible but the LCE/STC interface changes drastically. The most reasonable configuration would probably be to have a remote STE with these interfaces:

a. Redundant voice communications.

b. Data link carrying display TLM data to LCC.

c. Data link carrying control commands to STE.

d. RF link from Spacecraft to STE (backup for DSIF-spacecraft link).

Since most of the equipment capability and most of the operator capability is assigned the STE, it remains the focus of the operation, but the LCE has the direct hardwire control and can override the STE.

4. BENCH TEST EQUIPMENT (BTE)

4.1. DESCRIPTION

A vast amount of important test equipment is described by the generic term "bench test equipment". Bench test equipment is the test equipment designed for use on spacecraft subsystems or portions (bays on Voyager) of subsystems prior to the assembly of the subsystem into the spacecraft. This equipment is very important to the success of a test program. It provides the baseline test data on subsystems, assemblies, and components against which the measurements made in system test will be compared. Bench test equipment has sometimes been called subsystem test equipment, a misnomer because in the context of the Voyager spacecraft system, the test prior to system test may consist of a complete subsystem under test, a few major parts of a subsystem or parts of different subsystems which are mounted on the same mechanical assembly (bay).

4.1.1. SUBSYSTEM TESTING

Figure 14 shows a functional diagram of a typical subsystem showing where the subsystem parts might be positioned on the spacecraft. Most subsystems are scattered throughout the Voyager spacecraft. For example, the power subsystem consists of solar cells located externally, plus batteries and electronics mounted in several equipment bays.

A complete subsystem checkout then would require the assemblage of all these diverse parts connected together with a test harness (the flight harness would not be available prior to system assembly). In actual practice, it may be neither feasible nor desirable to connect all elements of the subsystem together. In cases where all elements are not present, the test equipment must accurately simulate the interface to the remaining equipment. This is extremely important to guarantee consistent results between bench test and system test.

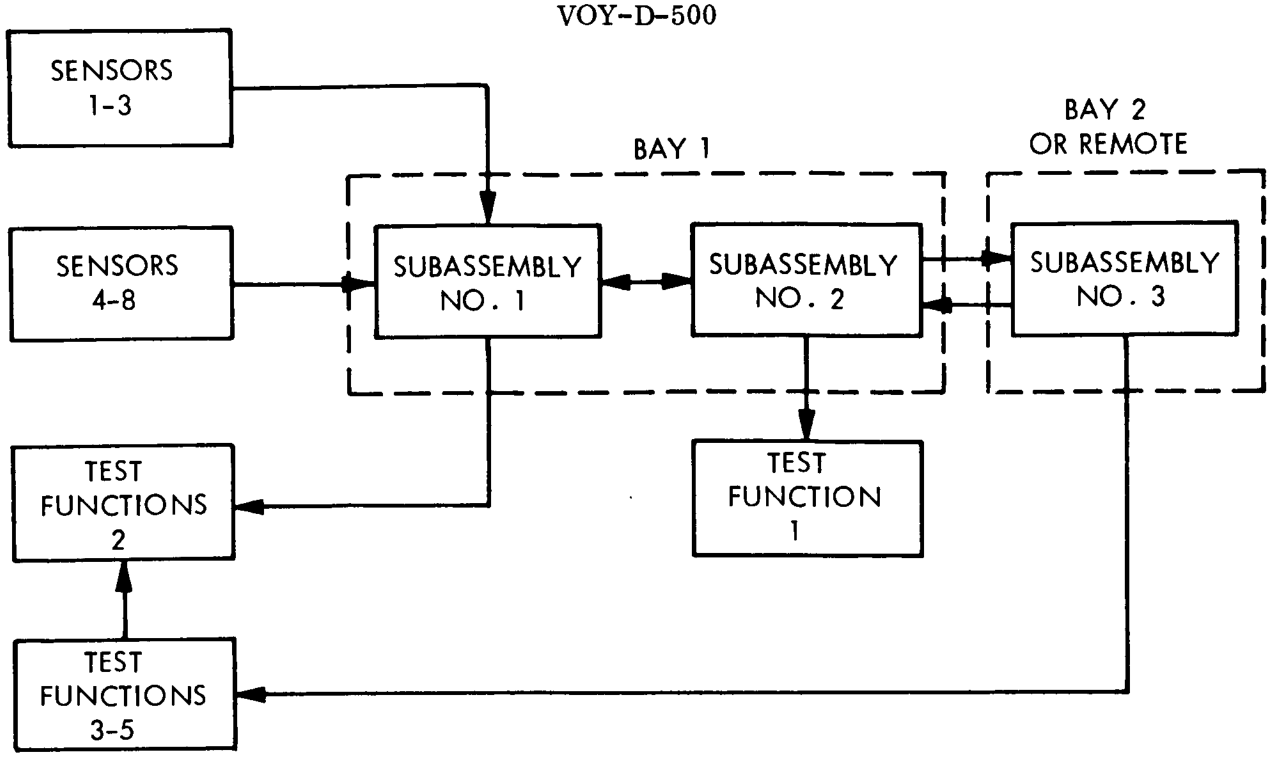


Figure 14. Typical Subsystem

4.1.2. Operation

Using the generic subsystem block diagram in Figure 14, the following illustrates the use of the bench test concept. First, it is assumed that for either assembly or convenience reasons, sensor package 1-3 is not available and, therefore, will not meet the rest of the subsystem until system assembly. This will also be assumed for function package 3-5. Since subassembly 1 and subassembly 2 are on the same bay, they will be connected together with internal bay wiring during test. Subassembly 3 will require a test harness for connection, as will the available sensor function packages. This is shown in Figure 15. BTE assemblies are substituted for the missing sensor and function packages while test harness has been substituted for flight harness.

As at the system level, emphasis will be placed on the use of margin testing. The use of go-no go tests at this level is even less desirable than at the system level. Performance margins must be measured, trends must be detected, and baseline data must be obtained for comparison with component test results and system test results. The BTE must be capable of determining

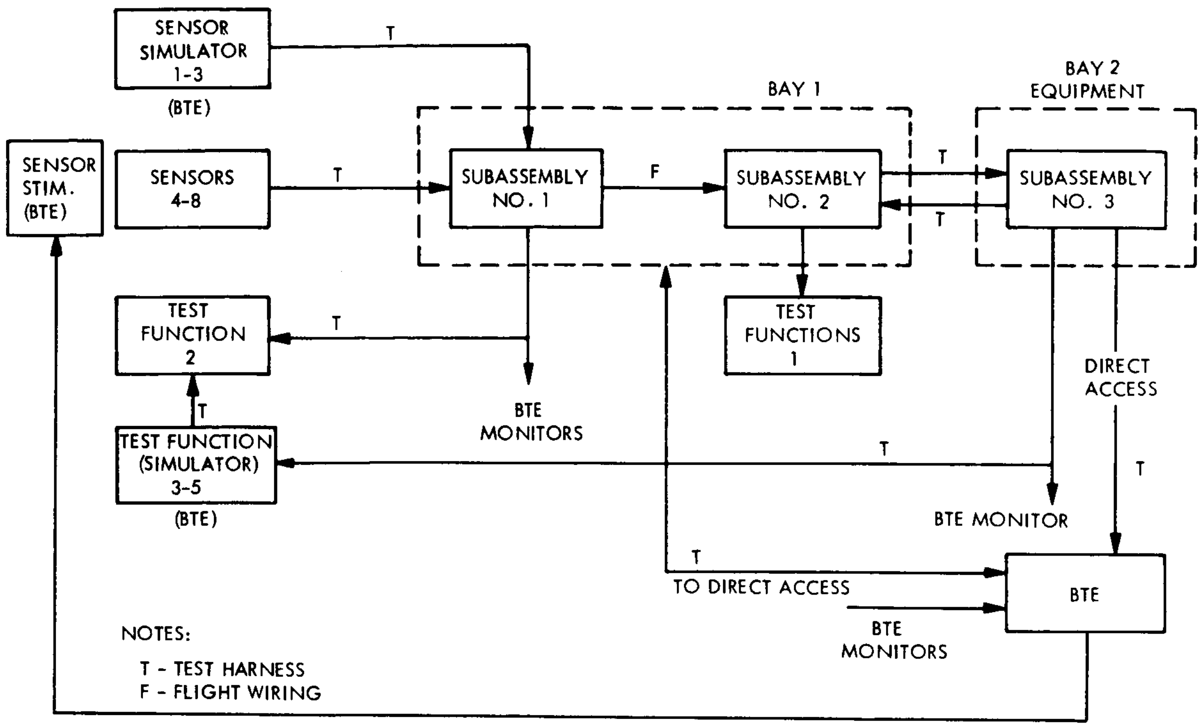


Figure 15. Typical Subsystem Bench Test

the performance characteristics of the subsystem to the extent that the number of test points used in system test may be significantly reduced.

4.1.3. BTE Design

The bench test equipment itself will be defined primarily by the groups working on the flight hardware. It will be particularly oriented to the flight article under test. The equipment will be designed for use by highly skilled technicians who will have a knowledge of how it functions. The bench test equipment will make use of all direct access points available for system test and may even require some additional test points on the flight hardware. Its primary access is through the flight connectors however. The basic design approach will be to use general purpose, off the shelf test equipment wherever possible for simplicity of design and economy. (Ex., Oscilloscopes, RF generator, measuring instrument, rate tables). Where shelf test equipment is not satisfactory, special test units will be designed for specific functions.

Whenever feasible, BTE will be remotely programmable and provide digital outputs so that remote control by computer is possible. A typical set of BTE for a subsystem will contain approximately 50 percent standard test equipment (occasionally with modifications) and 50 percent specially designed test equipment. At least 25 percent of the design will be identical to that used in system test, particularly where direct access connections or umbilical signals are monitored. Of course, not all test units can be rack mounted and although this approach is favored, it will be abandoned where rack mounting will affect test results, i.e., pyrotechnic simulators, sun sensor stimulators, etc.

The next task is to package the required BTE. This is simple if the following rules are followed:

a. Test harness must simulate flight harness.

b. Simulators must have high degree of correlation with flight hardware at nominal points and have as wide a range as necessary.

c. Max use of shelf test equipment.

d. Use of data for future comparison is provided for.

The configuration shown in Figure 16 would result.

The recording capability, degree of automatic control, and use of automatic data analysis techniques (computer) will depend on the individual subsystem. For example, tests on the C&S should be highly automatic and tests on articulation should be essentially manual. Automatic data logging must be traded off in each case against the cost of transposing the data from log sheets to data processing cards. In any event, the data should be available for later comparison with system test data.

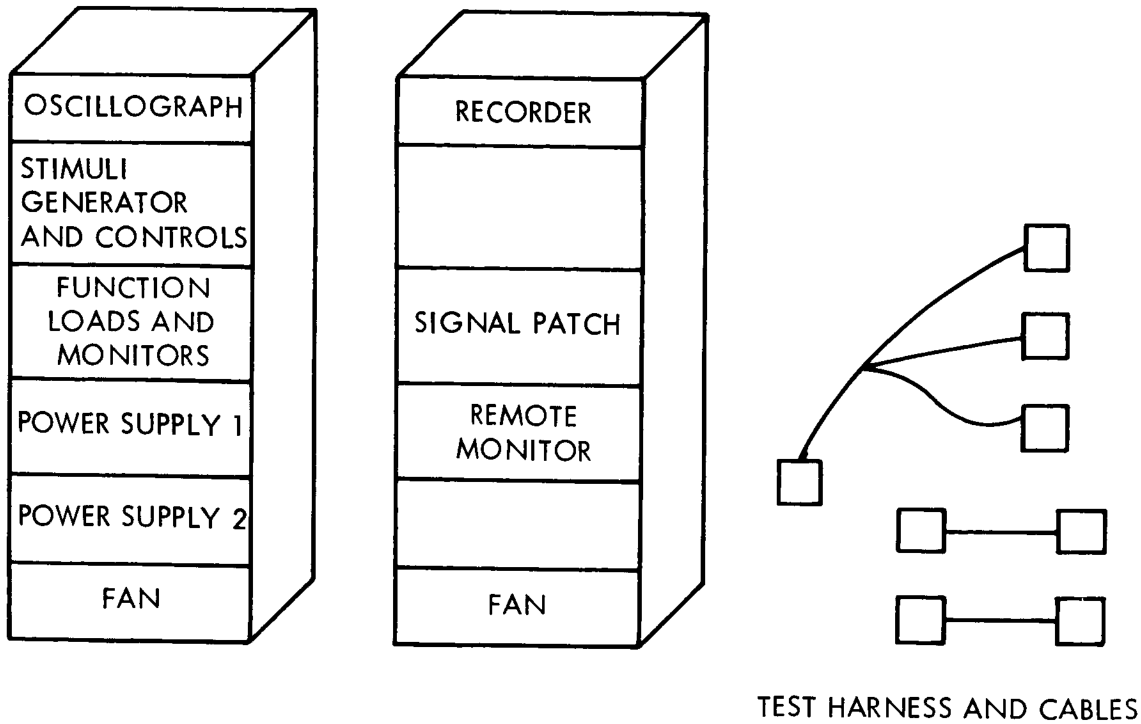


Figure 16. Subsystem Layout Sketch

4.1.4. Fault Isolation

Fault isolation in the flight hardware at the bench test level will depend more on the operators and less on the equipment than at system level, but the equipment must be capable of duplicating a system test malfunction situation for testing a subassembly rejected in system test. Normally, the bench test equipment will check the subsystem and/or its components to a depth greater than the replacement requirements at a particular location.

4.2. COMMONALITY

The BTE design must be as similar as possible to the system test equipment, particularly in the places where the BTE interfaces with the spacecraft. There are several advantages to this approach.

a. Similarity in data gathered - It is easier to relate two data items if they are gathered on the same kind of equipment. Trend analysis capability is enhanced.

b. Similar equipment simplifies operator training and improves the learning curve.

c. It provides some degrees of equipment interchangeability in emergency situations.

d. Simplification of interface requirement by reducing number of different interfaces.

There are also problems created by this approach.

a. BTE must be delivered sooner and if the equipment is to be similar to system test equipment, the requirements for system test equipment must be firmed up sooner.

b. Requirements for the two levels of test are not the same and care must be taken not to prejudice test capability because of a desire for commonality.

The recommended approach to commonality in the design of Voyager BTE is:

a. The design of the system test equipment and its requirements are overriding.

b. Wherever feasible, specific pieces or subassemblies of system test equipment should be incorporated into the Bench Test Equipment.

c. Where incorporation of actual hardware is not possible, similar equipment designs with identical interface circuits will be used.

d. Data should be monitored and presented in a form which is consistent with system tests to simplify future comparison.

4.3. PERFORMANCE REQUIREMENTS

The system test performance requirements are thorough and call for a detailed test of each subsystem in the spacecraft system context. The BTE must be capable of performing all of the tests required for the system situation. (Volume IV, Appendix A, Spacecraft Interface Equipment, lists some of the requirements for the different sets of BTE.) The BTE must also be capable of performing complete tests on subsystem equipment, which must include:

a. Operation in all modes used in system test.

b. Simulation of all system interfaces and signal variation at these interfaces (Extreme care must be taken that the system simulation is adequate to insure that malfunction or marginal operation discovered in system test can be duplicated in subsystem test).

c. Provision for any adjustments or calibration which must be made as late as possible before launch, but cannot be done in system configuration.

5. ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

The ground mission Assembly, Handling, and Shipping Equipment (AHSE) requirements are functional support of the assembly and test shown on the ground mission diagram of Figure 17. AHSE does not include tooling or fixtures required for fabrication operations at the plant level except where such hardware has multiple usage in the plant and the field.

The ground mission of the spacecraft will include the general assembly of the vehicle, support of subsystem and system testing, shipment between major manufacturing, test and field facilities, and the field cycle up to and including encapsulation of the planetary vehicle within the launch vehicle shroud, and transportation of the planetary vehicle to the launch pad. Handling and transportation of the launch vehicle shroud, nose fairing and capsule, except as noted herein, are not presently considered to be within the scope of spacecraft AHSE. Development testing involving the various systems test models such as the Structural Test Model (STM), Thermal Test Model (TTM), the Engineering Development Model (EDM), and the Proof Test Model (PTM) will be supported by the AHSE.

It should be noted that ground mission analysis is based on Flight Acceptance (FA) tests; Development and Proof Test Model (PTM) spacecraft requirements should be treated as complementary increments to be supported by basic FA equipment, supplemented by special purpose test equipment where required.

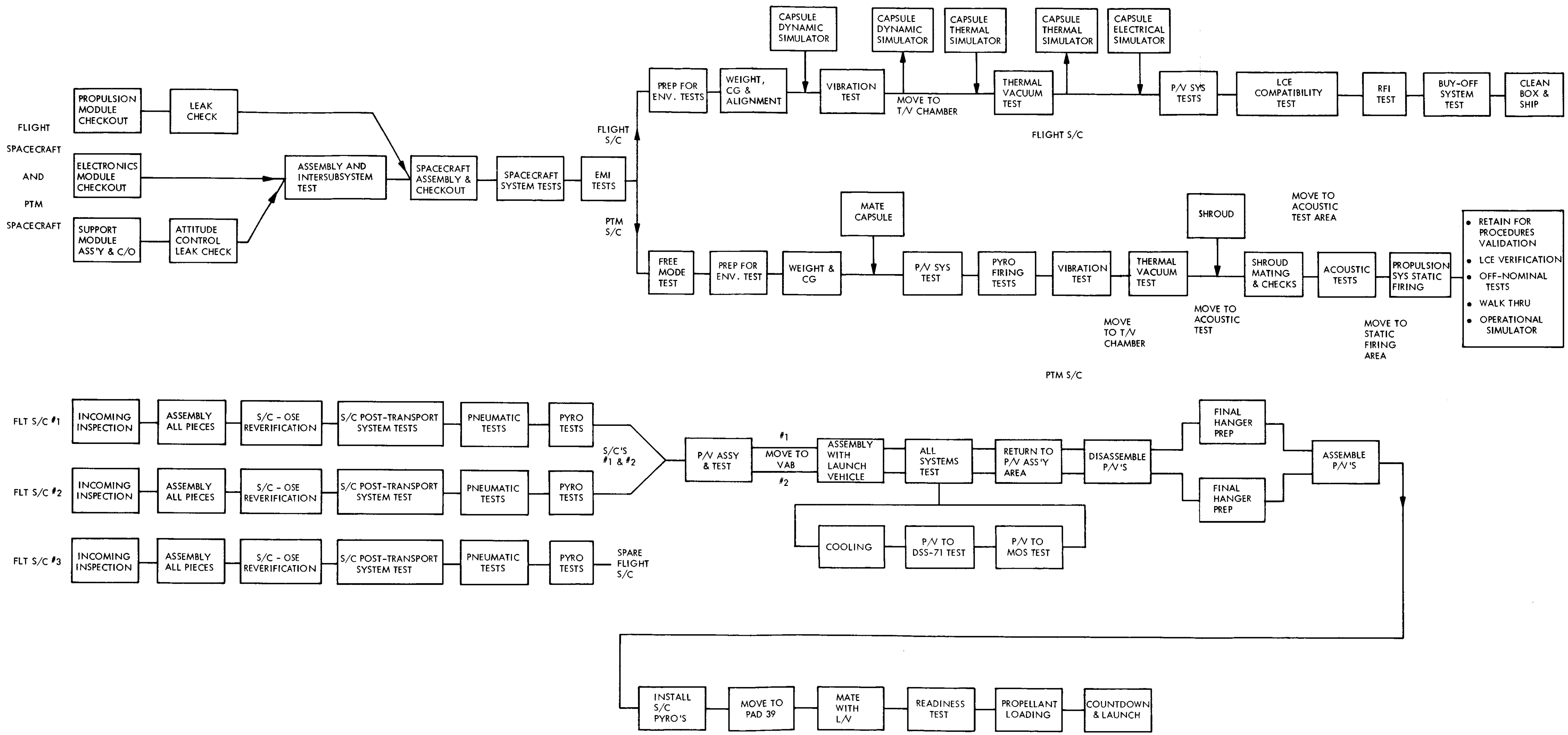


Fig 17-A

Figure 17.8 Ground Missions
(Factory and Field)

5.1. SYSTEM CONCEPT

As a result of analysis based upon system constraints and requirements and the over-all size of the vehicle and its relation to size of facilities required, the preferred approach to AHSE is the development of a basic central station which will satisfy the maximum number of functional requirements and thus minimize vehicle handling and redundancy of design features. The central station concept will permit vehicle assembly, system testing, subsystem testing, weight and lateral center-of-gravity location, free-mode testing, interface compatibility checkout, and final system alignments to be implemented at the one station. Close integration with facility design can permit trenching or ducting of STE cabling, as well as built-in features which improve the economics of station design, as well as reduce testing time. The central station referred to above is designated as the spacecraft assembly fixture.

Test requirements demanding capability beyond that feasible for incorporation in the spacecraft assembly fixture will be satisfied by special purpose test stations to which the spacecraft will be removed for the completion of the test. System rechecks following an environment check will be performed using the spacecraft assembly fixture to which the spacecraft will be returned if recheck is necessary, prior to proceeding to the next special purpose test equipment. Whenever practical, special-purpose tests will be run within the same area as system tests to facilitate environmental controls, (cleanliness, temperature, etc.) and minimize vehicle handling.

AHSE will furnish support for the assembled vehicle, regardless of the test phase or the test location, which again introduces economic advantages in design labor, redundance of equipment, and only one excursion of the learning curve. The general approach to AHSE required for prelaunch operations will follow that outlined above for factory operations. In fact, the KSC central station can be the same item used in the factory.

Review of the transportation methods available for moving the spacecraft from factory to KSC indicates that the fully assembled spacecraft can be shipped by air or water. Disassembly into smaller units for shipment must be minimized because of the potential degradation of the reliability and integrity of the spacecraft system. As a result, the preferred design of shipping equipment can accommodate shipping the assembled spacecraft (less fuel) by air or, as a secondary alternative, by water vessel down the inland water route.

The AHSE required to implement the ground profile is grouped into the following appropriate functional categories:

a. <u>Assembly Equipment</u>

1. <u>Simulators</u>

   Capsule interface simulator
   Propulsion subsystem interface simulator
   Shroud simulator
   Science subsystem test fixture

2. <u>Alignment Sets</u>

   G&C subsystem alignment set
   Solar panel alignment set
   Science payload alignment set
   Medium-gain antenna alignment fixture
   Capsule interface alignment fixture
   Propulsion module alignment set
   Optical tooling kit
   High gain antenna alignment set

3. <u>Assembly Fixtures</u>

   Spacecraft assembly fixture
   High gain antenna assembly fixture
   Medium gain antenna assembly fixture
   Solar panel assembly fixture
   Planet scan platform assembly fixture
   Personnel access stand
   Electronic module assembly fixture
   Propulsion module assembly fixture
   Assembly equipment spacecraft storage stands, non-flight
   Support module assembly fixture

b. <u>Handling Equipment</u>

Spacecraft transporter
Spacecraft protective covers and devices
Transporter prime mover
Spacecrafter support fixture
Spacecraft and planetary vehicle lift sling
Propulsion module lift sling
Planet scan platform handling fixture and sling
Solar panel handling fixture and sling
High gain antenna handling fixture and sling
Medium gain antenna handling fixture and sling
Electronic module lift sling
Spacecraft protective cover purger
Vertical load balancer
Shroud lift adapter
Shroud transporter

c. <u>Shipping Equipment</u>

Shipping container
Shipping barge
Prime mover
Cargo lift trailer
Cargo lift trailer loading ramp
Cargo lift trailer adapter pallet
Transporter aircraft
OSE shipping containers
Spare parts shipping containers

d. <u>Special Purpose Support Equipment</u>

Weight and center-of-gravity determining equipment
Free mode test equipment
Thermal vacuum test equipment
Vibration test equipment
Separation test equipment
Environmental control equipment
Acoustic test equipment
Static firing test equipment

e. <u>Propulsion Service Equipment</u>

Liquid (oxydizer and propellant) equipment
Gas equipment
Leak test equipment

Figure 18 shows the factory and field (KSC) flow of AHSE required by the ground mission. Many of the larger items are conceptually illustrated in this figure.

The flow of Figure 18 illustrates the sequence of assembly, test, and transport operations for a flight model spacecraft. For convenience, the special purpose test equipment, needed to support the additional tests required for the proof test model spacecraft only, are shown on the bottom of the figure.

The flow is influenced by the division of the spacecraft structure into three modules (propulsion module, support module, and electronics module). Figure 18 shows the support and propulsion modules entering into the factory assembly test flow as complete units. Therefore, AHSE shown for these two modules is only that hardware needed at the module handling level.

The flow chart illustrates much of the AHSE and indicates use sequence of these items. Many of these items are conventional in nature, and will not differ in concept or philosophy from analogous AHSE items used for other spacecraft programs. The remainder of this discussion will be addressed to a few items (and categories) of AHSE which reflect requirements or approaches unique to Voyager.

5.2. ASSEMBLY EQUIPMENT

The assembly equipment which most typically reflect considerations unique to the Voyager Spacecraft configuration are (a) the spacecraft assembly fixture, which is first used to assemble the three structural modules into a spacecraft, and (b) the alignment used throughout the spacecraft build-up and test process.

5.2.1. Spacecraft Assembly Fixture (Item 4 of Figure 18)

The fixture will be used for holding the spacecraft and planetary vehicle during final assembly, inspection, repair and tests. The fixture will be primarily of aluminum construction

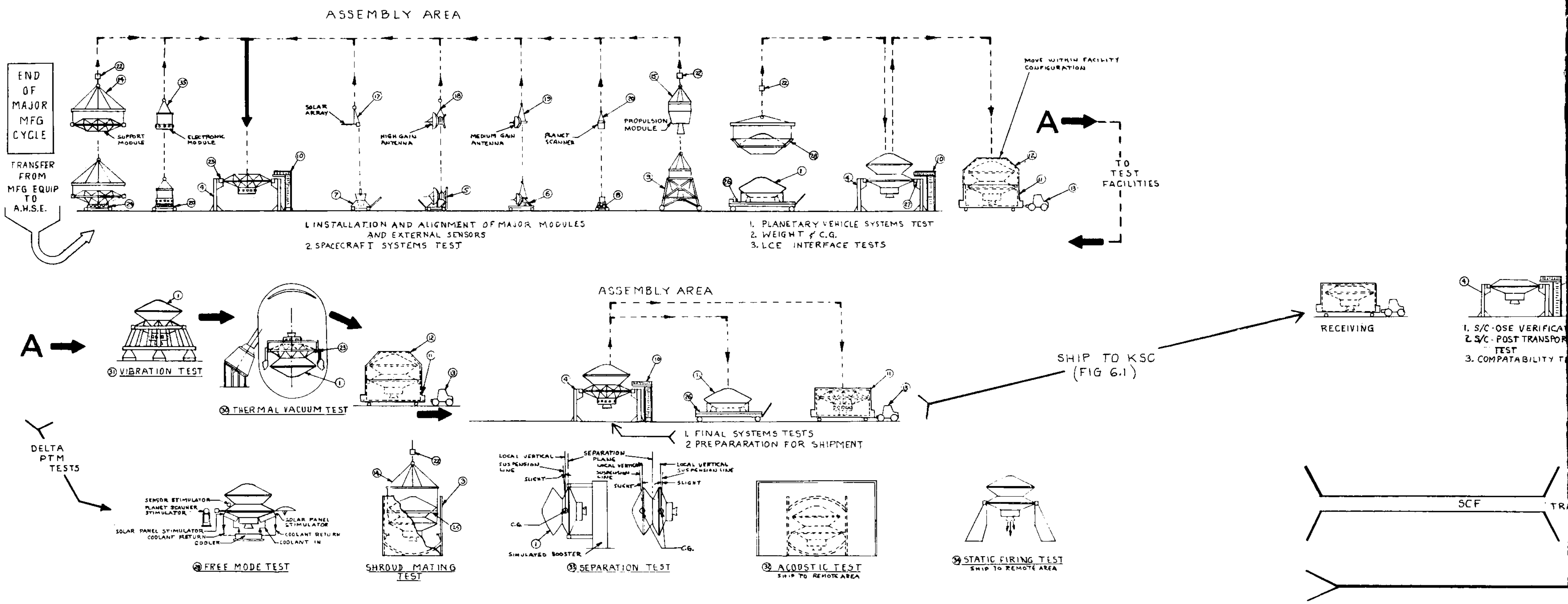
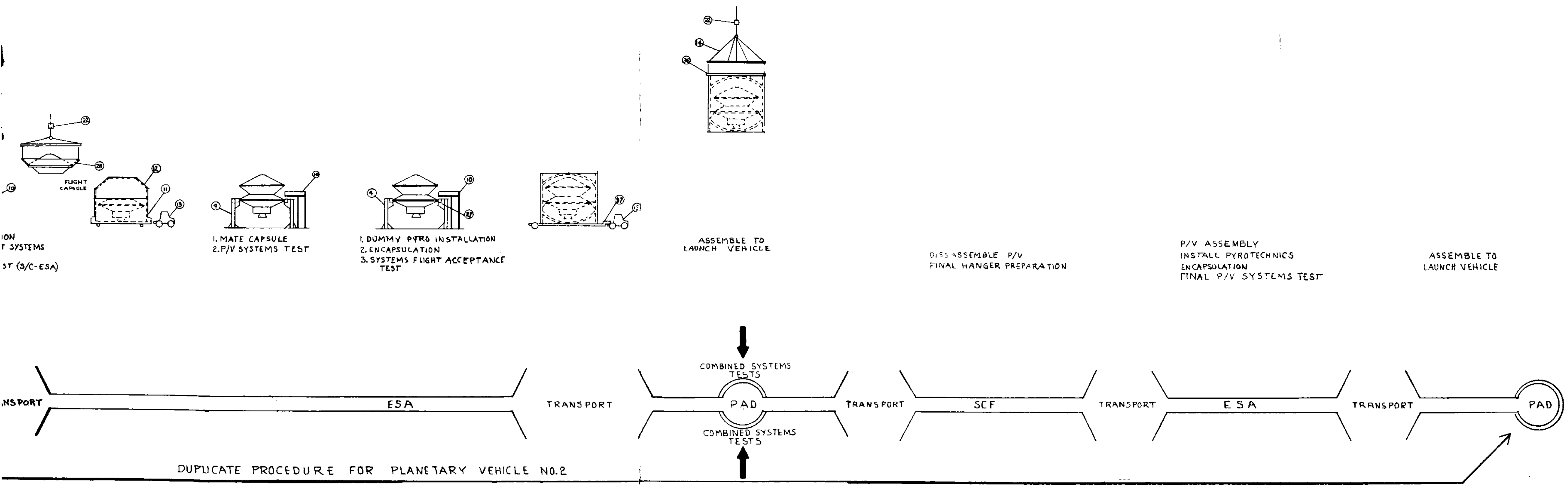


Fig. 18 A

Fig 18 B

FOLDOUT FRAME

FOLDOUT FRAME



1 CAPSULE INTERFACE SIMULATOR
3 SHROUD SIMULATOR
4 S/C ASSEMBLY FIXTURE
5 HGA ASSEMBLY FIXTURE
6 MGA ASSEMBLY FIXTURE
7 SOLAR ARRAY PANEL ASSEMBLY FIXTURE
8 PLANNET SCANNER ASSEMBLY FIXTURE
9 PROPULSION MODULE ASSEMBLY FIXTURE
10 PERSONNEL ACCESS STAND
11 S/C TRANSPORTER
12 S/C PROTECTIVE COVERS AND DEVICES
13 TRANSPORTER PRIME MOVER
14 S/C - P/V LIFT SLING
15 PROPULSION MODULE HANDLING FIXTURE AND SLING
16 PLANET SCANNER HANDLING FIXTURE AND SLING
17 SOLAR ARRAY PANEL HANDLING FIXTURE AND SLING
18 HGA HANDLING FIXTURE AND SLING
19 MGA HANDLING FIXTURE AND SLING
20 ELECTRONIC MODULE HANDLING FIXTURE
21 SHROUD HANDLING SLING
22 PRECISION VERTICAL LOAD BALANCER
23 SPACECRAFT SUPPORT FIXTURE
24 SUPPORT MODULE ASSEMBLY FIXTURE
25 CAPSULE LIFT ADAPTER
26 CAPSULE SIMULATOR SUPPORT STAND
27 WEIGHT AND C.G. EQUIPMENT
28 CAPSULE LIFT SLING - CAPSULE CONTRACTOR
29 FREE MODE TEST FIXTURE
30 THERMAL-VACUUM TEST FIXTURE
31 VIBRATION TEST FIXTURE
32 ACOUSTIC TEST FIXTURE
33 SEPARATION TEST EQUIPMENT
34 STATIC TEST FIXTURE
35 ELECTRONIC MODULE LIFT SLING
36 SHROUD LIFT ADAPTER
37 SHROUD TRANSPORTER

Fig 18 C

Figure 18D AHSE Factory and Field Ground Flow Diagram

FOLDOUT FRAME

70-A FOLDOUT FRAME

FOLDOUT FRAME

with capability of supporting a load of 30,000 pounds. It will be capable of supporting the spacecraft with or without the propulsion module as well as the complete planetary vehicle with the roll axis vertical.

The assembly fixture is required to be compatible with spacecraft-planetary vehicle lift sling (Item 14 of Figure 18), STE and LCE cables, environmental control enclosure, nose fairing simulator and personnel access stands. The assembly fixture contains leveling features for leveling the spacecraft or planetary vehicle to within ± 6 arc seconds. This is needed to permit alignment of spacecraft subsystems. The assembly fixture also has to contain necessary reference markings, required for proper optical alignment of subsystems, weight-indicating devices, such as load cells for weight and c.g. indication of the spacecraft. Four load cells at 10,000 pounds each are needed. The fixture will also contain a mechanism for blocking out cells when weighing is not required. A built-in load cell calibration unit will be used.

The fixture will be a multileg structure, approximately 6 feet high and 20 feet in diameter, and made up of standard structural shapes with the necessary cross-bracing required for rigidity and strength. The structure will be essentially openwork to permit maximum access to the spacecraft. Floor loading will be maintained at acceptable levels so that a special foundation for the assembly fixture will not be required. The weight and readout device will be placed adjacent to the assembly fixture. The leveling device in the fixture will consist of electric motor-driven mechanical jacks with the necessary controls and readouts. Rough leveling will be done automatically; fine leveling will be accomplished by manual operation of controls. The required leveling pads, front-surface mirror mountings and other features required for alignment will be designed into the assembly fixture. Direct line of sight will be maintained at all times. A mobile personnel workstand will surround the assembly fixture. The assembly fixture will be designed such that it is compatible with:

a. Shroud simulator

b. Alignment equipment

c. Thermal control enclosure

d. Capsule simulator

e. Flight capsule

f. Propulsion interface simulator

g. Free mode solar simulators

5.3. ALIGNMENT EQUIPMENT

To the greatest extent possible, the alignment operations will take place at the final assembly level. The spacecraft assembly fixture will be utilized for alignment. Leveling capability and spacecraft axes reference will be built into the assembly fixture to enhance the multiple-use concept. The basic design approach is to use gravity as the common reference between items being aligned and the optical tooling equipment being used. Since the spacecraft orientation for assembly purposes is with the roll axis vertical, the first step in alignment will be to establish the roll axis vertically. This will be accomplished by leveling the separation plane between the flight spacecraft and the shroud. For simplification, this plane will hereinafter be called Reference Plane "A". The spacecraft manufacturing tooling will be used to establish the yaw axis. Through this axis, a vertical reference plane (Reference Plane "B") will be erected by optical tooling procedures. Individual component mounting surfaces will be checked for parallelism of perpendicularity to Reference Plane "A" by using clinometers (angle-reading levels) of appropriate sensitivity. For indexing (or pointing) about the roll axis, optical instruments independently supported on stands will be provided. These instruments will be located relative to component and spacecraft axes by normal optical tooling procedures. Target equipment will be provided. In the case of angular displacements, front surface mirror targets will be used in conjunction with autocollimation. For linear displacements, bifilar targets will be utilized with telescopic instruments having optical micrometers. If the linear displacement tolerance is greater than the range of the optical micrometers (±0.100 inch), optical tooling scales or mechanical staging employing micrometer readouts will be used. The tolerances of those components to be aligned are itemized in Table 3. A typical alignment is the alignment of the propulsion module.

Table 3. Alignment Requirements

COMPONENT	TOLERANCE
Canopus sensor	$\pm$ 15.0 Minutes
Sun sensors	$\pm$ 15.0 Minutes
Gyro and accelerometer package	$\pm$ 10.0 Minutes
Cold gas nozzles	$\pm$ 1.5 Degrees
High gain antenna	$\pm$ 6.0 Minutes
Deployable solar array	$\pm$ 3.0 Degrees
Medium gain antenna	$\pm$ 8.0 Minutes
Flight capsule	30-Min half-cone angle
Propulsion thrust chamber	$\pm$ 12 Minutes
Science payload	Dependent on Science Instruments

The propulsion module engine alignment set will be used to check the perpendicularity of the center lines of the nozzle to Reference Plane "A". The alignment set will provide fixtures and targets to give the capability for optical alignment procedures. The fixture shall have a front surface mirror (of optical tooling quality) mounted such that the surface is perpendicular to the center line of the nozzle within 15 arc seconds. The design approach is to provide plug targets which will fit inside the nozzles to reference the nozzle center line. In order to be able to calibrate the alignment equipment to 15-arc-second accuracy, the mirror will be mounted in an adjustable mount. A calibration fixture will be provided in the optical tooling kit. Before being used, the mirror on the alignment fixture will be adjusted until it is normal to the fixture center line (and therefore also the nozzle center line). This procedure will entail the use of autocollimation. As shown in Figure 19, a theololite and pentaprism will be used

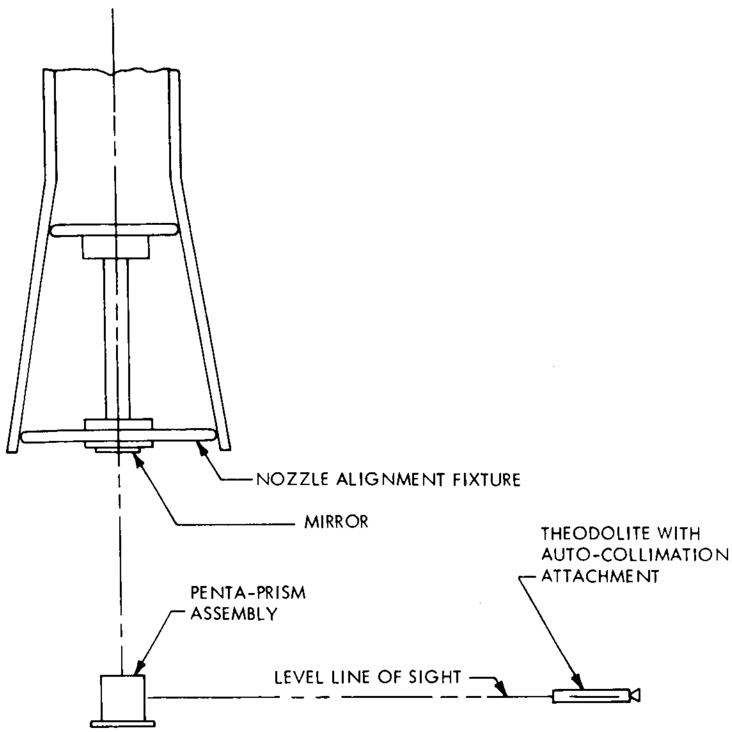


Figure 19. Angular Alignment

to check the angular alignment of the nozzles to the reference plane. Two alignment checks will be made with the axes of the theodolites 90 degrees apart.

5.4. HANDLING EQUIPMENT

The functional handling in which the spacecraft will be involved during the various factory assembly and testing phases is illustrated by Figure 18. A supplemental description of typical handling follows. The following major spacecraft subassemblies will be transferred from the receiving inspection area and the manufacturing equipment in the shop areas to AHSE assembly equipment in the systems test station area.

a. Propulsion module

b. Electronic module

c. Micrometeoroid barrier

d. Support module

e. Solar array panels

f. Planet scan package

g. Science instrumentation

h. Medium-gain antenna

i. High-gain antenna

The spacecraft support fixture (less trunnions) will be attached to the support module at the shroud interface. Utilizing the spacecraft and planetary vehicle lift sling, the support module will be positioned in the spacecraft assembly fixture. Buildup of the support module and subsystem tests will be accomplished while the support module is in the spacecraft assembly fixture. Mating of the electronic module to the support module will be accomplished by moving the electronic module assembly fixture below the support module, raising the electronic module assembly fixture to the correct height and installing the mating hardware. Harnesses, OSE, and other items required will be installed to complete the electronic module configuration. Appropriate subsystem checkout and testing will be performed.

The propulsion module will be mated to the spacecraft in the spacecraft assembly fixture by attaching the propulsion module lift sling to the propulsion module, raising it clear of the propulsion module assembly fixture, transporting it to the spacecraft assembly fixture and lowering it into the spacecraft. The mating hardware will be installed and electrical connections made. The planetary vehicle configuration will be completed by lowering the capsule simulator onto the spacecraft by means of the capsule sling. The mating hardware will be installed and electrical connections made. The scheduled systems test will be conducted (with and without the capsule simulator) in the spacecraft assembly fixture.

For these tests not performed in the spacecraft assembly fixture, such as vibration, thermal vacuum, acoustic and separation, the entire planetary vehicle configuration can be lifted into the spacecraft transporter by the planetary vehicle lift sling, the appropriate protective cover installed and the transporter towed to the remote test site. There, the planetary vehicle lift sling is again utilized to place the planetary vehicle into the test fixture. The planetary vehicle will be returned to the system test area (spacecraft assembly fixture) by reversing the above procedure.

The planetary vehicle-to-shroud interface test will be conducted by lowering the planetary vehicle into the shroud.

After final factory systems tests, the capsule simulator will be returned to its storage area, and the spacecraft shipped to KSC in the transporter.

The tests at KSC will essentially be the same types performed in the factory. A supplemental description of the handling anticipated at KSC follows:

The spacecraft will be packaged in the spacecraft transporter. After unpacking, the spacecraft will be moved to a receiving-cleaning area, where it will be cleaned in compliance with program requirements and subsequently emplaced in the systems test area of the spacecraft checkout facility. The spacecraft will be lifted from transporter by the spacecraft planetary vehicle lift sling and placed on the spacecraft assembly fixture.

KSC flight spacecraft processing will consist of a confidence test flow through the spacecraft checkout facility (SCF) to explosive safe area (ESA) to pad sequence, followed by a servicing phase which flows through the SCF-ESA-pad route and is terminated by lift off.

The incoming and systems tests will be performed at the SCF prior to transfer to the ESA. All tests in the ESA will be conducted in the systems test stations. The planetary vehicle in the encapsulated configuration will be taken to the pad for combined system test, after which it will be taken to the SCF for further systems tests. It will then be returned to the ESA for installation of pyros, weight and balance, encapsulation in the shroud and final planetary vehicle systems tests. All AHSE will be available in the field sequence. For movement between the SCF and ESA, the capsule will remain mated to the spacecraft. The spacecraft transporter will be designed to accommodate the planetary vehicle for land movement. It is intended that AHSE will be transferred between the SCF and the ESA to eliminate equipment redundancy. During the transfer, provisions will be made, such as wrapping and transporting in closed vans, to preserve the cleanliness of the equipment. Current KSC functions require only two items of field peculiar AHSE.

The following material describes some of the Voyager handling equipment needed to implement this flow.

5.4.1. Spacecraft Transporter (Item 11 of Figure 18)

The transporter/canister provides mobility between manufacturing, test, assembly, and checkout facilities in the factory and field for the structural module or propulsion module either separately or together, the spacecraft, and the planetary vehicle (spacecraft with capsule). Further, the transporter/canister will serve a shipping function between the factory and field for the spacecraft. The transporter/canister shall be constructed of steel and aluminum as dictated by strength requirements. When loaded and in use, the pressurization system will maintain a pressure of 0.5 psig above any ambient pressure. The maximum weight of the transporter with cargo shall be less than 25,000 pounds. The maximum overall dimensions shall be such that the transporter will fit within a cylinder 94 feet long and 25 feet in diameter.

The transporter/canister will be constructed of standard structural shapes forming a cylindrical shape on a low bed trailer configuration. The cylindrical portion will be top opening. With the top open, the transporter cargo will be loaded and unloaded vertically with lift slings and overhead cranes. The top will be removable; the bottom portion shall be capable of mating with a protective shipping cover which will provide environmental protection for a spacecraft/capsule configuration.

The transporter will have pneumatic wheels. They will be steerable with the prime mover and be capable of being locked in any rotational position. It shall be towable from either end with commercial truck-tractors or in plant tow vehicles. Maximum towing speed shall not exceed 15 miles per hour. The braking system shall be electric and air, thereby being compatible with either type of prime mover. A demountable fifth-wheel towing attachment for commercially available truck-tractors shall be included.

Environmental control (self-contained) should be provided as part of the transporter. Humidity control will be inherent in the dry nitrogen gas used for maintaining the required pressure. The pressurization system will be able to vent and resupply the transporter due to changes in ambient pressures as will occur with air transport and natural pressure fluctuations.

A self-contained voice communications system will be required such that the two truck drivers will have constant voice contact with a rear guide man and two side guide men.

5.4.2. Spacecraft and Planetary Vehicle Lift Sling (Item 14 of Figure 18)

The spacecraft and planetary vehicle lift sling will be used to lift the spacecraft, planetary vehicle with capsule simulator or flight capsule, shroud empty, or with encapsulated planetary vehicle.

The sling will be made of aluminum and stainless steel with capability for lifting the spacecraft and planetary vehicle. The sling will be compatible with the spacecraft, personnel work stands, the planetary vehicle using either the capsule simulator or flight capsule, and the shroud lift adapter. The lift sling also needs capability of supporting the spacecraft and planetary vehicle by attachment to the spacecraft support fixture trunnions.

The lift sling will be constructed of standard structural shapes with the necessary attachment points for stainless wire rope cables. Quick-release pins will be used to attach the wire rope at the interface points. The sling will attach to removable lift fittings on the planetary vehicle. These fittings will be located eight places at the planetary vehicle-to-shroud interface. The sling will incorporate necessary standoffs to prevent any part of the sling from contacting either the capsule or shroud during encapsulation.

5.5. PROPULSION SERVICE EQUIPMENT

Loading, purging, conditioning, and perhaps proof test verification, will be accomplished at the launch pad, in a mated configuration. As indicated in Volume IV, "Applicability of Apollo Checkout Equipment to Voyager OSE," it is intended to utilize the propellant, oxidizer, and purge facilities now part of the launch complex (for LEM) for Voyager. Hence, the specific Voyager OSE needed to support these operations at KSC are difficult to identify, as the degree of modification required to existing facilities, is not yet firmly identified. The AHSE to support loading, etc., will have the function of completing the spacecraft to launch complex interface, and is not further identified in this report.

Fluid test operations, for both propulsion and gas systems, will be performed at the factory using facility type hardware where possible. Some items which are peculiar to Voyager may have to be used also at KSC and hence are either STE or AHSE, depending upon the item's nature and function.

Leak test equipment is required, although some leak test capability exists at LC 39. The technical approach to leak test, recommended for the Task D spacecraft configuration, is substantially the same as the Task B preliminary design. It is:

a. Through leak testing using He enriched N_2 internally, with an external vacuum applied to adjacent parts.

b. External leak testing with the system pressurized to normal working pressure using He enriched N_2, and use of external gas analyzer probe.

c. Proof pressure testing with conditioned N_2.

Gases instead of liquids should be used to enhance leak detection sensitivity. The procedure and philosophy is considered to be equally applicable to both gas (attitude control) and liquid (propulsion) systems in Voyager.

The degree of repetition of factory level leak and pressure tests at KSC is not firmly established. Hence, the ground mission profile and flow charts (figures 17 and 18, respectively) do not emphasize this facet of ground operations.

5.6. SHIPPING EQUIPMENT

The primary function of the shipping equipment is to support the logistics concepts developed for the entire program. This entails assuring transportability of all equipment to be delivered to a remote destination for either testing or launch. It is further required that the shipping equipment provide the necessary protection to the items being shipped from both natural and induced transportation environmental effects. The specific methods of implementing shipment are as indicated below.

5.6.1. Shipping Configuration

The flight spacecraft shall be shipped fully assembled. This is a prime requirement because of the possible degradation of the reliability of the spacecraft that might result from disassembly and subsequent reassembly. Disassembly would also tend to negate the effectiveness of the final systems testing at the manufacturing site. Compliance with this "no disassembly" concept presents a transportability problem because of the large size of the assembled spacecraft (approximately 22 feet in diameter and 14 feet high).

5.6.2. Mode of Transportation

The size of the fully assembled spacecraft eliminates the possibility of shipment by rail. Air shipment is limited and can be accomplished by specially modified aircraft. Alternate modes of transportation will be over the highway and by water. However, highway transportation shall be kept to an absolute minimum because of the obvious difficulties in moving an item of this size over the road. The shipment will be accompanied by qualified technical personnel to assure constant monitoring of the environmental control system. Figure 20 illustrates the transportation flow.

5.7. SPECIAL-PURPOSE TEST EQUIPMENT

Special performance measurements different from launch required activities are needed to support the Voyager program. This equipment is manned in the test flows for the various test and flight spacecraft. Table 4 summarizes the equipment and the usage it will encounter. Components will receive some of the same environmental inputs as are imposed at the system level. The equipment, however, will be different equipment of universal type as opposed to the system test set-ups which will be specialized for Voyager.

Special-purpose test equipment utilization with tests, vehicles, and locations is shown in Table 5.

Table 4. Special Test Equipment Usage

		SYSTEM LEVEL			
Special Equipment	Component	Development	Qual	Acceptance	
Free mode			0		
Thermal vacuum	0		0	0	
Vibration	0		0	0	
Dynamic separation	0				
Acoustic		0	0		
Separation		0	0		

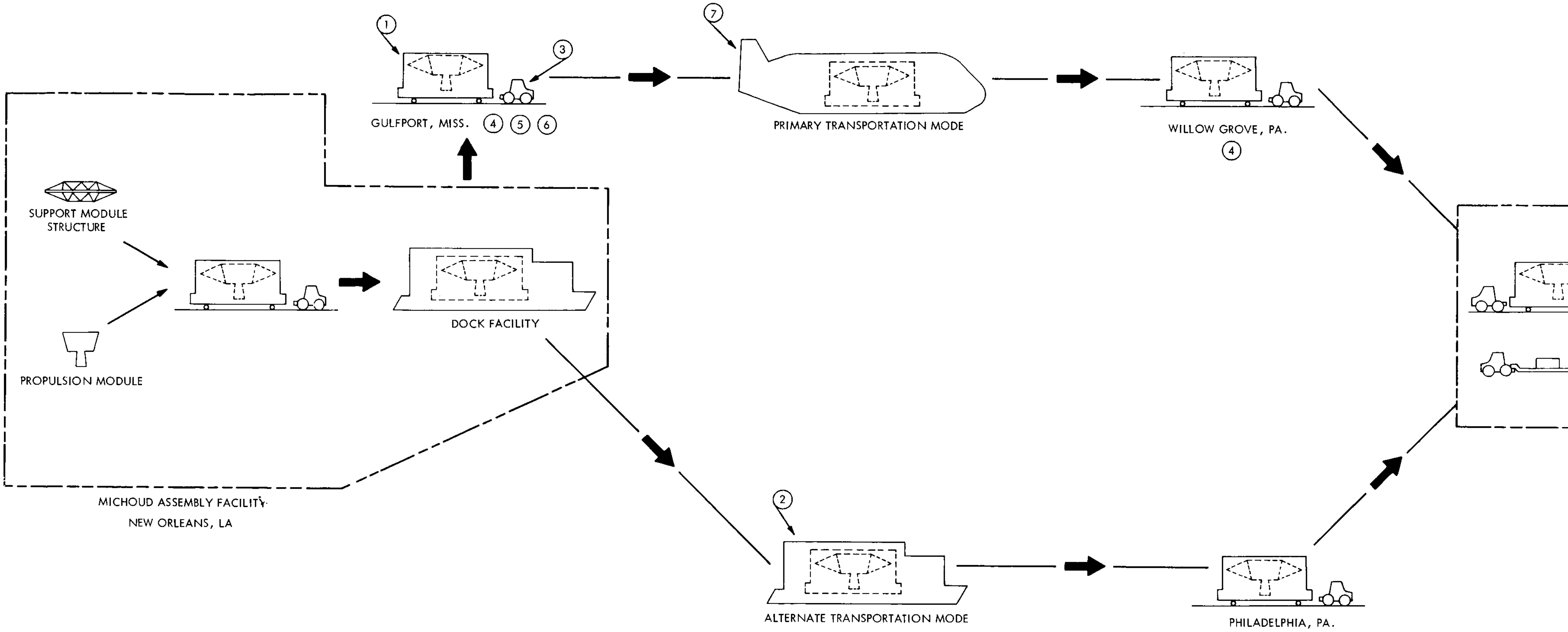
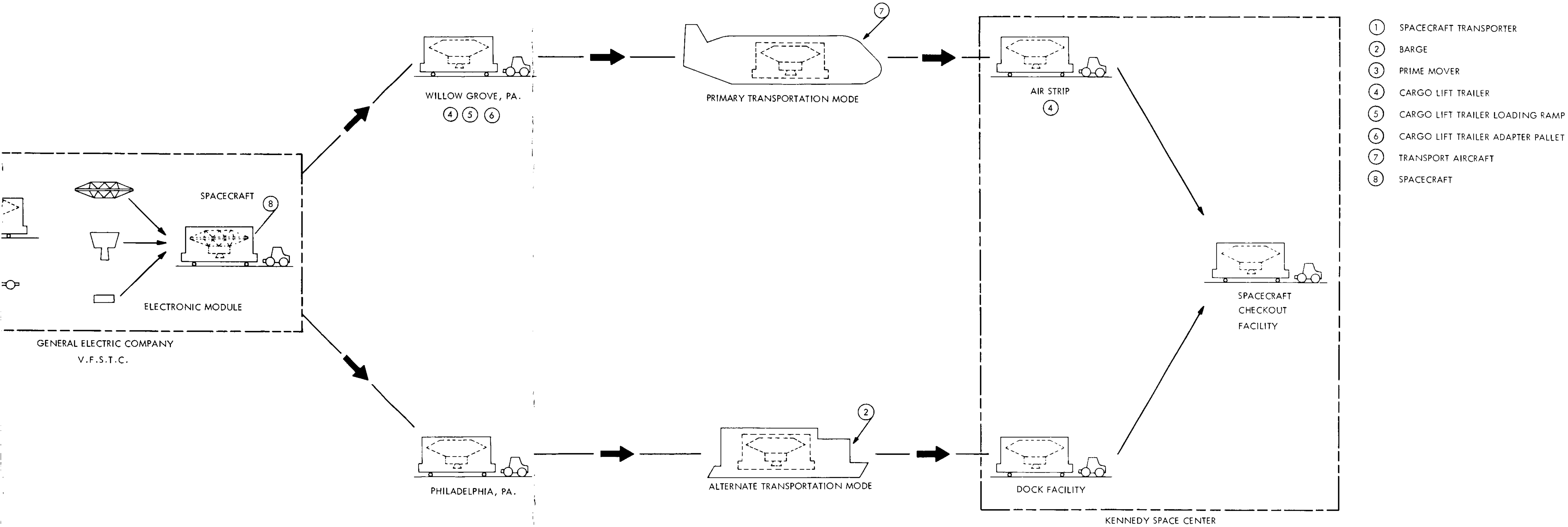


Fig 20-A
83-A
FOLDOUT FRAME

Fig-20-
83-B



B

Fig. 20-C
84-A

Figure 20.D Transportation Flow
Flight Vehicle

FOLDOUT FRAME

FOLDOUT FRAME

Table 5. Special Purpose Test Equipment Utilization

TEST	VEHICLE	LOCATION
Free mode	PTM	SCF, Factory
Thermal vacuum	PTM, FA	Factory
Vibration	PTM, FA, STM	Factory
Separation	STM, LVDM	Factory, MSFC
Acoustic	PTM, STM	Remote
Static firing	PTM, PM	Remote

5.7.1. Free-Mode Test

The free-mode test is performed with the spacecraft completely isolated electrically from system test equipment. Power for operating the vehicle must come from its own batteries and from the solar panels. Communication with the vehicle is via RF. Stimuli will be provided for the various sensors. A capsule simulator will be mounted on the spacecraft for PTM tests.

Equipment used in connection with the free-mode test is comprised of lamp banks simulating sunlight and a means to support the Planetary Vehicle during the test. The free-mode test will be implemented in the spacecraft assembly fixture.

Power required for the lamp banks will be 50 kilowatts in order to produce the equivalent of 1/3 sun (at 1 au) at the solar array. Excess heat energy from the lamps in each bank will be absorbed by gas or liquid coolant. A bank of lamps will be provided for each solar array panel. Approximately 11 kilowatts of electrical energy at each lamp bank will stimulate each solar panels. Since the solar array is most sensitive to light in the near-IR region of the spectrum (peaking at 0.8 micron), the stimulating light should be rich in near-IR, which will have the effect of heating the arrays as well as stimulating them to produce power. Two methods are being considered to limit the temperature of the cells in the solar arrays: (a) bathe the cells in cold gas (e.g., boiled-off CO_2) or (b) pass the radiant energy through a fluid in order to filter the emergent light to a shorter wavelength, less efficient than the near IR.

Of the 11 kilowatts of electrical power supplied to each lamp bank, only 425 watts emerges as useful radiant power, and of this, only 34 watts is converted by the solar array to electrical power in the spacecraft. Sufficient cooling, then, must be supplied to each lamp bank to absorb 500 Btu/min and 24 Btu/min at the solar array, assuming 8 percent efficiency of the solar cells, and 5 percent efficiency of the lamps and reflectors. A 60-ton refrigerating system will cool all the lamp banks. Approximately 25 to 35 pounds per minute of coolant gas would be required to dissipate excess heat at the solar cells.

5.7.2. Thermal-Vacuum Test

The thermal-vacuum test will be conducted in two phases: (a) the thermal model spacecraft will be subjected to hard vacuum, simulated cold space, and simulated sunlight and (b) the flight spacecraft will be subjected to hard vacuum, simulated cold space, and heat energy supplied by thermal sources; the PTM may be tested in either mode. The thermal model will simulate the midcourse correction maneuver, during which time the side of the spacecraft will be subjected to simulated sunlight. All vehicles will be supplied with a thermal simulation of the capsule.

During this test the spacecraft must be supported in the test chamber, and provision must be made for turning the thermal model spacecraft through 90 degrees to simulate either the midcourse correction or direct solar acquisition.

5.7.2.1. Thermal-Vacuum Test Equipment

The thermal model will be placed in the 54-foot thermal-vacuum chamber using the spacecraft transporter for transportation and the spacecraft and planetary vehicle lift sling. The spacecraft will be mounted on the spacecraft support fixture with trunnions attached. Rotation will be effected by a hydraulic cylinder using silicone oil for the operating fluid. A small heating blanket attached to the cylinder will assure reliable operation under conditions of cold space. The pump for the hydraulic fluid will be mounted outside the chamber. The spacecraft will also be positioned on its support within the test chamber by the spacecraft and planetary lift sling and the spacecraft support fixture without trunnions.

The PTM will be placed in the thermal-vacuum chamber using the spacecraft transporter for transportation, the spacecraft and planetary vehicle lift sling and the spacecraft support fixture.

Heat inputs to the PTM will be by means of thermal plates supported in a light aluminum scaffolding erected inside the chamber. The PTM will be supported by an adjustable suspension which will use the spacecraft support fixture and will permit leveling the spacecraft. The scaffolding will have vertically movable sections to permit adjusting the thermal plates to conform to the PTM contour.

5.7.3. Vibration Test

The vibration test will impart a vibration spectrum to the spacecraft with a dynamic capsule simulator attached equivalent to the inputs to the spacecraft by the shroud.

The vibration test equipment will be a fixture for supporting the spacecraft and for transmitting to the spacecraft a vibration environment simulating the various boost phases of the mission. A simulated capsule will be provided for STM and PTM configurations. The simulated capsule will be dynamically similar to the flight article.

Strength of the fixture will be such that a fully loaded spacecraft will be supported uniformly, and deflections of the fixture will be minimized. Vibration inputs can be along the pitch, the yaw, or roll axes. Natural frequency in lowest mode of the fixture will be above 200 cycles per second. Dynamic simulators of the capsule and propulsion propellant and pressurants will be required.

The vibration fixture will provide a mechanical interface similar to the shroud interface with the spacecraft. Vibration inputs to the spacecraft will thus be equivalent to the inputs to the spacecraft by the nose fairing. The direction of input can be along the yaw, pitch, or roll axis.

The vibration facility will be comprised of a bank of vibration machines and oil-film tables to support the fixture. The vibration machines will be connected electrically so that their amplitudes and phases are the same. The vibration fixture will be constructed to permit the shaker heads to provide parallel inputs along the pitch, yaw, and roll axes of the spacecraft. Vibration force can be generated along two axes at once by suitably pivoting the fixture. The fixture will be made of aluminum, which has a higher strength-to-weight ratio than steel. When the spacecraft is vibrated with input along the roll axis, support for the spacecraft and fixture will be provided by the vibration machine heads. The fixture will be provided with pickup points compatible with the spacecraft and planetary vehicle lift sling. The fixture surface mating with the spacecraft will simulate the interface. The first phase of detailed design will include an analytical study of the various concepts with intent to optimize the fixture with respect to vibration in lowest mode and cross-coupling of response.

5.7.3.1. Separation Test

The separation test will test the effectiveness and dynamic response of (a) the planetary vehicle at separation from the shroud interface, and (b) the capsule and the spacecraft at separation from each other. The test will be conducted in two phases: (1) separation of the planetary vehicle from its interface with the shroud and (2) separation of the capsule from the spacecraft.

A group of equipment is required to support operationally separable sections of the planetary vehicle with 5 degrees of freedom during separation of interfacing elements.

The separating sections will be supported by a suspension cable attached at the C.G. point to permit 5 degrees of freedom during the separation. The equipment required for this test will provide means to support the vehicles during test, means to measure energy transfer at separation, and means to detect and measure tip-off motion in other directions in addition to the direction of separation. Similar separation test equipment has been used successfully on past programs. Test equipment will include a dynamic capsule simulator; a shroud simulator; a massive support, which simulates the booster (the booster is assumed infinitely massive); a 1-3/4 inch steel cable to support the STM during test from a pivot 75 feet above the center of gravity of the STM; a divided arc with pointer; and a pivot for the cable at the center of gravity of the STM. (Auxiliary equipment used to measure and record motion of the spacecraft during separation is not included in this description.)

Both of the pivots, one fixed above and one mounted at the C.G. of the planetary vehicle, will be knife-edge pivots, arranged to form a universal joint. This will reduce the friction, always present in ball or roller bearings, to nearly nothing. The lower pivot will be adjustable in three directions to achieve maximum motion sensitivity for pitch or yaw motions. Roll motion will be permitted through twisting in the long cable. The knife-edge form of pivot is deemed best for this application because of its inherent ruggedness (10,000 pounds per inch maximum load for a 90-degree edge) and its low-friction qualities, if hard tool steel is used.

The same equipment will be used during the capsule/spacecraft separation test. Several methods can be used to limit backswing of the separated craft: (a) rubber attached to the cable, (b) moving the support out of the way of the backswing, or (c) moving the upper support forward as the vehicle starts its backswing. A load cell will be incorporated in the support. This will enable the center of gravity of the vehicle along the roll axis to be measured. The support itself will be pivoted at one edge and will have a hydraulic lift at the opposite edge for turning the support from a horizontal to a vertical attitude.

The STM will be mounted on its adapter, previously attached to the support, in a vertical attitude. The assembly will then be pivoted to a vehicle attitude with roll axis horizontal. The center of gravity of the STM will be known. The cable will be attached at this point.

5.7.3.2. Acoustic Test

Equipment will be required to support the PTM in an acoustic chamber in such a manner that it will receive acoustic inputs in the same environment as encountered during powered flight.

The sonic environment will be applied to the exterior of the shroud. Suitable sensing and recording apparatus will determine the response of the planetary vehicle to the applied acoustic forces.

The acoustic test equipment will be used at the acoustic test facility at a remote location. A shroud furnished as GFE, will be provided to support the planetary vehicle in the acoustic chamber. Equipment required for the acoustic test will be the spacecraft and planetary vehicle handling sling, shipping containers for shipment to the acoustic chamber where the test will be conducted, a simulated capsule, a simulated shroud (GFE), and a fixture on which to mount the simulated shroud inside the acoustic chamber. A plane-based aluminum fixture will be used to support the shroud and will simulate the shroud-booster interface. The tie-downs will be located at approximately 3-foot intervals around the periphery of the capsule. Strain-gaged bolts will measure the tension applied to the 1/8 inch tie-down cables.

5.7.3.3. Static Firing Test

This test will provide information for analysis of the vehicle under a static test firing which simulates Mars orbital injection.

The static firing test stand will support the spacecraft in the Z-axis vertical attitude, as required by the test facility. Suitable sensor and recording equipment will provide information for analysis of the vehicle under a static test firing. The static firing test stand mates with the planetary vehicle interface. It will resist both downward and upward loads imposed by the fueled planetary vehicle and by the force generated by firing. Measurement of the force will be accomplished by means of four load cells, each of which will make a record of the thrust. Greatest upward force will be less than 30,000 pounds. Greatest downward force will be a maximum of 20,000 pounds.

The fixture will be made of steel. It will be made in sections to facilitate shipping. It will be assembled by bolting together. Auxiliary tie-downs, if required to withstand the thrust, will be incorporated in the fixture. The fixture will be mated to existing tie-down equipment at the test stand.

APPENDIX A
OSE/MDE REQUIREMENTS

In the evaluation discussion of the candidate Voyager OSE concepts with respect to availability, a basic assumption was made that, maintainability being equal, the availability (probability of the system being up, or available, at time t) becomes the same as the probability that it has not failed. In order to judge the relative reliability (availability) of the four concepts, the approach used was to construct block diagrams representing the critical functions of each concept in the configuration for launch support. In order to do this, with any degree of meaning, assumptions were made as to how these functions would be implemented.

The basic diagrams for the four concepts are shown. These formed the basis (as well as general engineering intuition) for ranking the concepts for relative availability. The diagram for each concept is intended to show each function involved critically, and show the nature of functional redundancy, which is one major aspect of the concept's reliability. They do not show, explicitly, either equipment quantity or complexity.

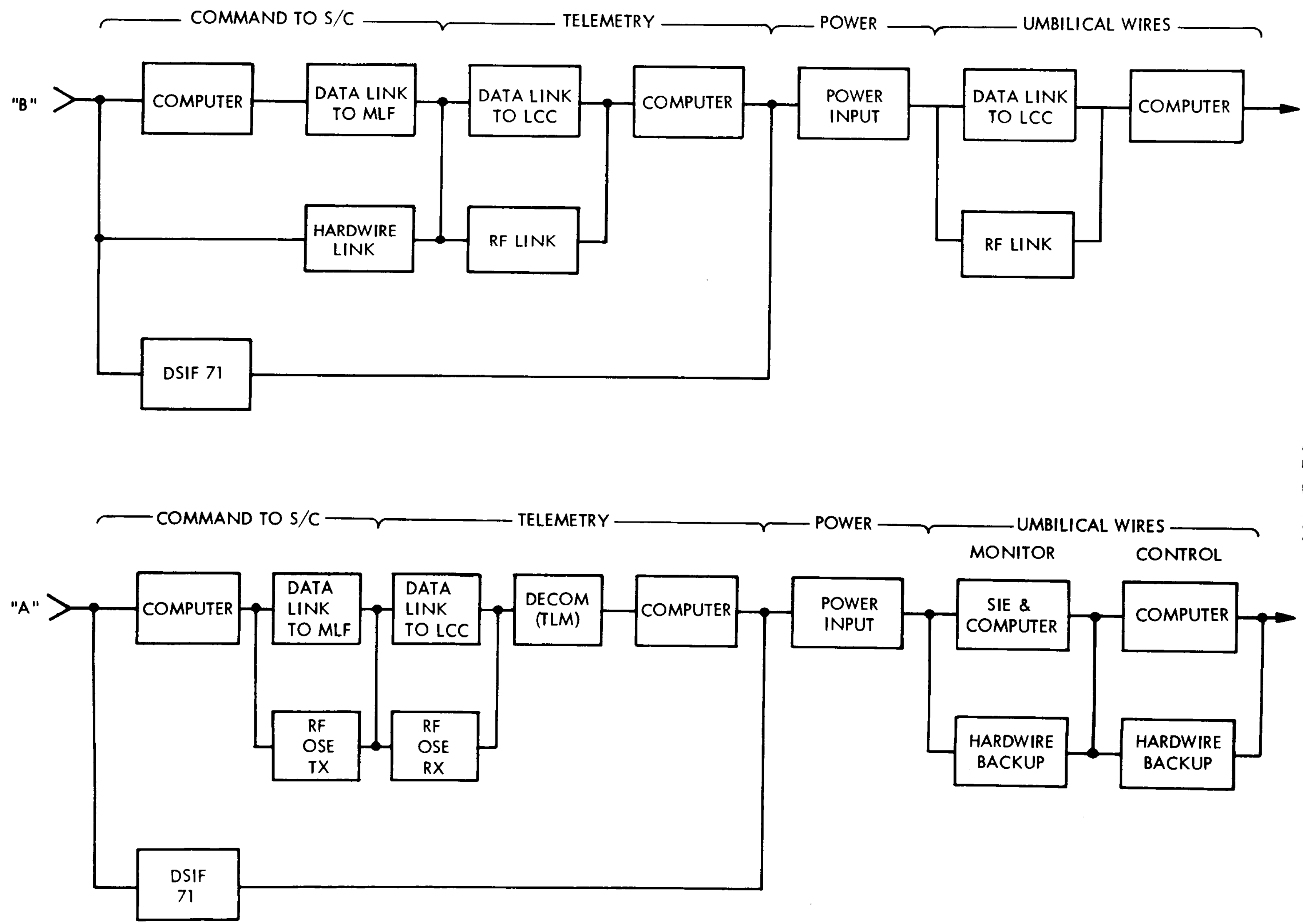


Figure A-1. Concepts "A" and "B" Reliability Block Diagram

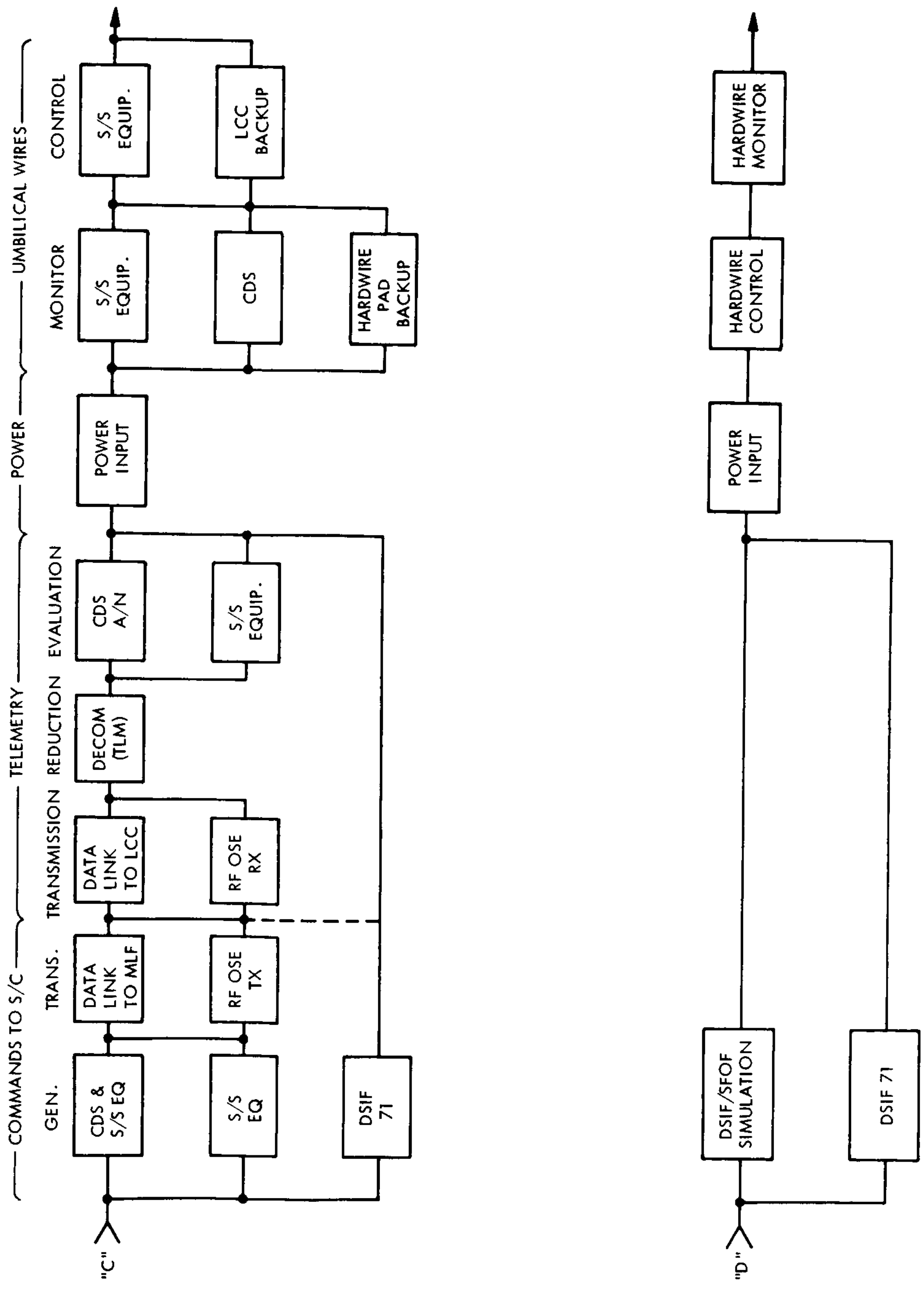


Figure A-2. Concepts "C" and "D" Reliability Block Diagram

APPENDIX B
LAUNCH FUNCTIONS

The nature of prelaunch and launch operation support required of the OSE is quite different from system test phases of the program. This is a consequence of the spacecraft, at this point, being loaded with live pyrotechnics, being enclosed within the shroud, and having access restricted to the RF path and to the umbilical.

The purpose of the OSE at launch is (a) to provide a direct, hardwire control and display path for critical functions, and to set launch conditions, and (b) to provide a coarse indication of system "health." It is used at a point in time when the actual test program has been concluded.

The design approach of the launch control equipment is set by the number and nature of umbilical wires needed to implement the first of these functions. It is assumed that the STE is used, remotely, during prelaunch and launch, to provide end-to-end control via links, and works in conjunction with the launch equipment.

It was considered appropriate to generate first cut approximations of LCE configuration for the two most appropriate LCE concepts. These are shown in schematic form in the following diagrams. They show the first two levels of definition of design approach, for each of the concepts most suitable for Voyager.

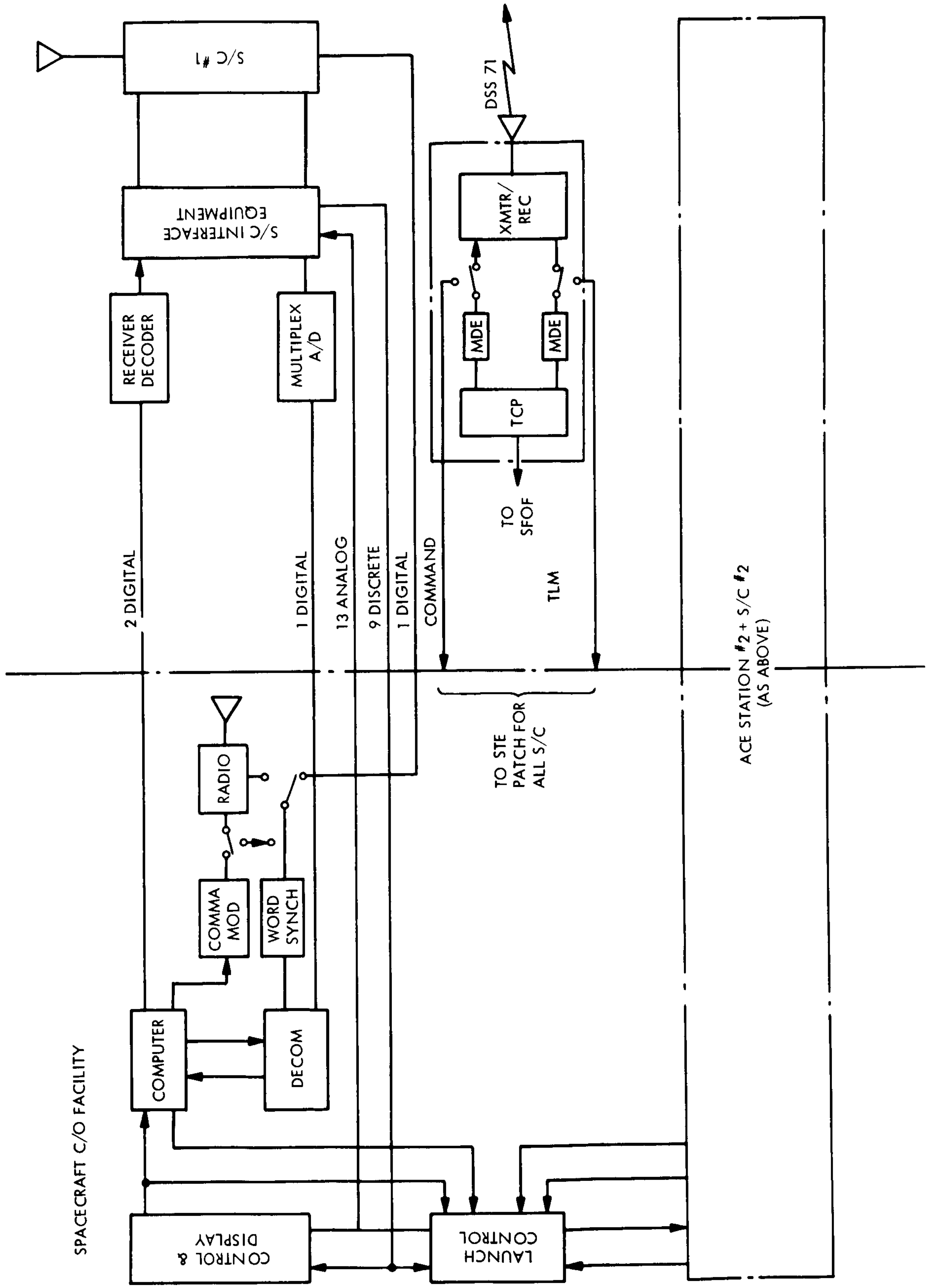


Figure B-1. Concept "A" Block Diagram

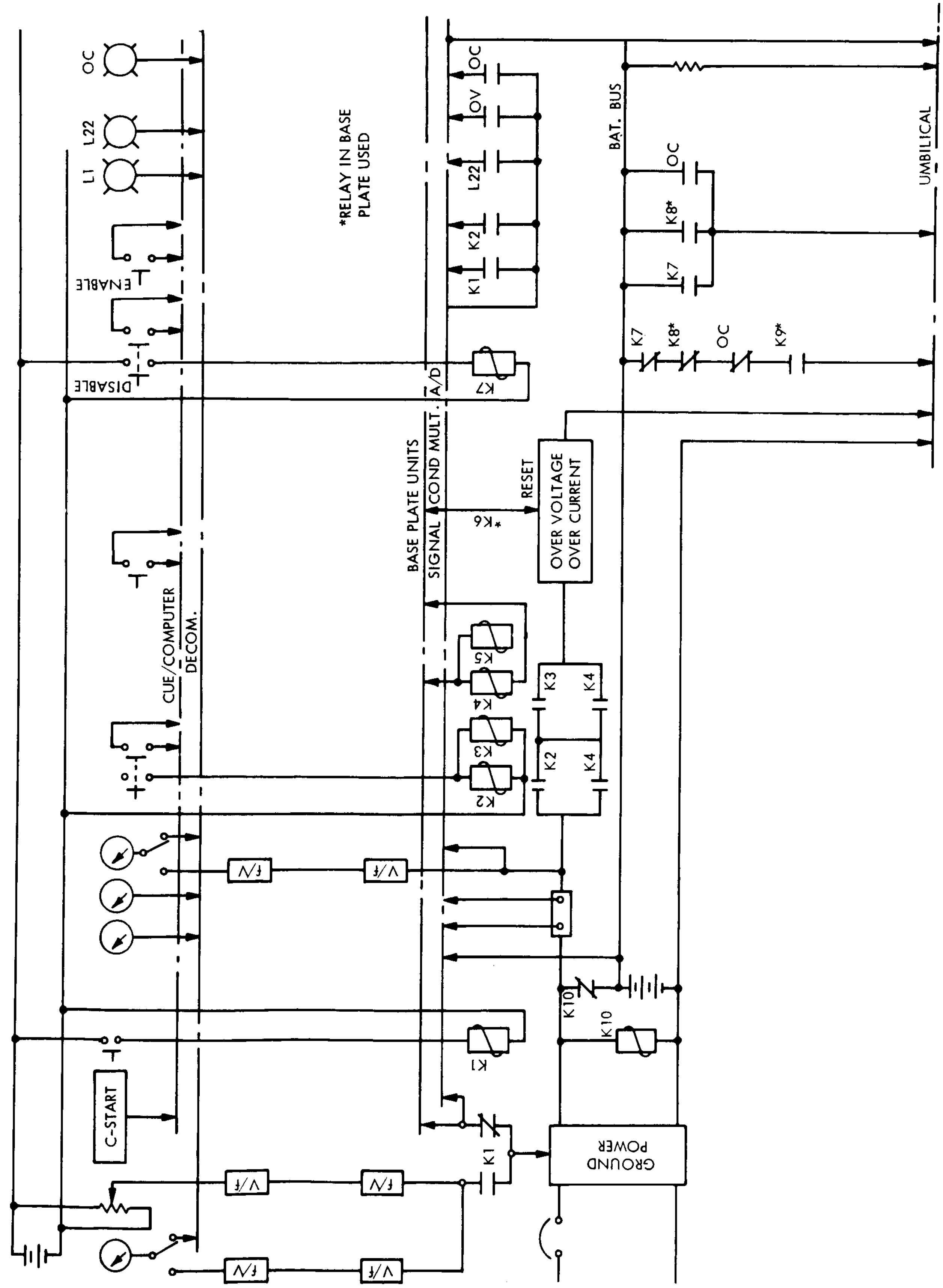


Figure B-2. Concept "A" Power Circuits

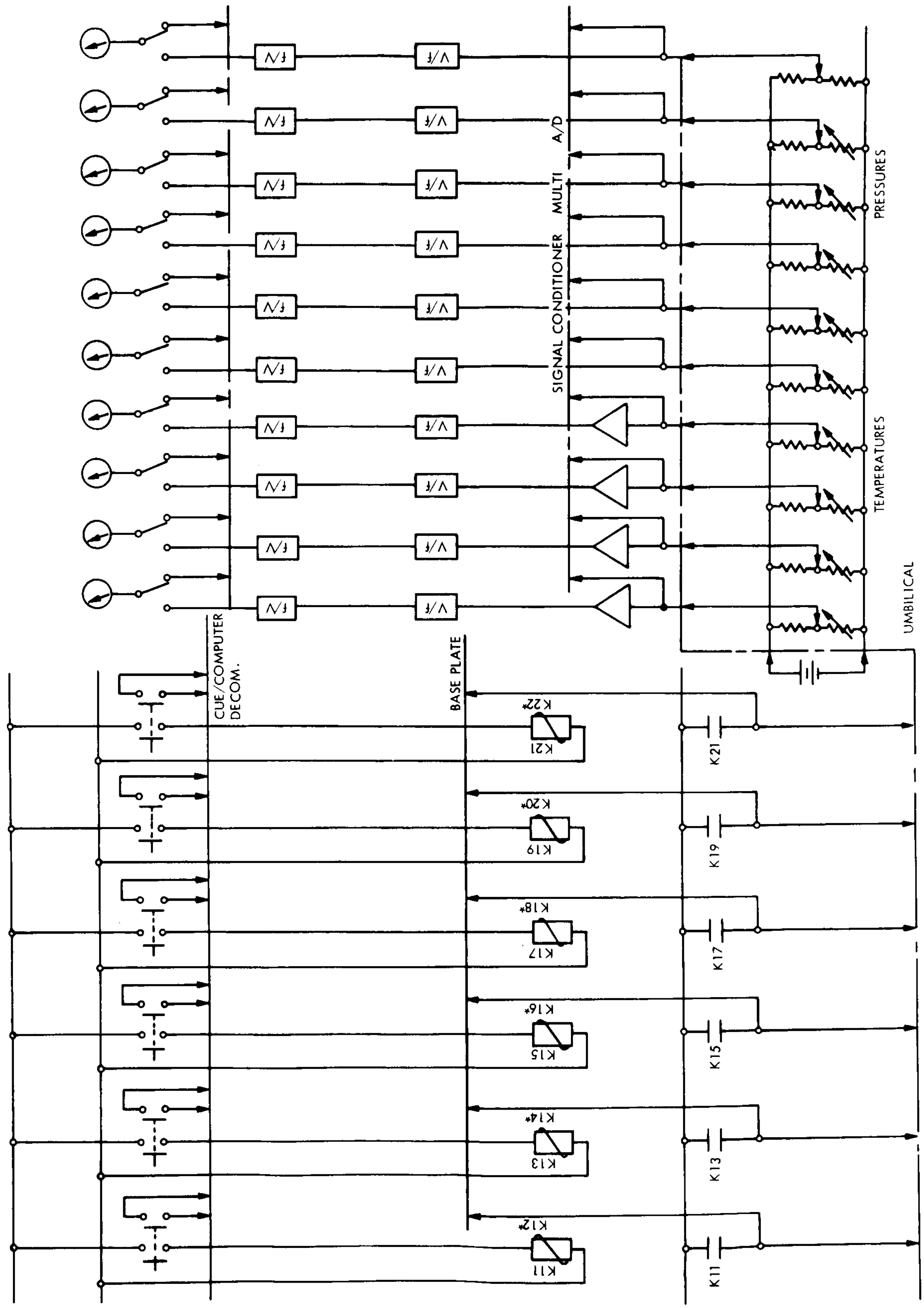


Figure B-3. Concept "A" Propulsion Safing Circuits

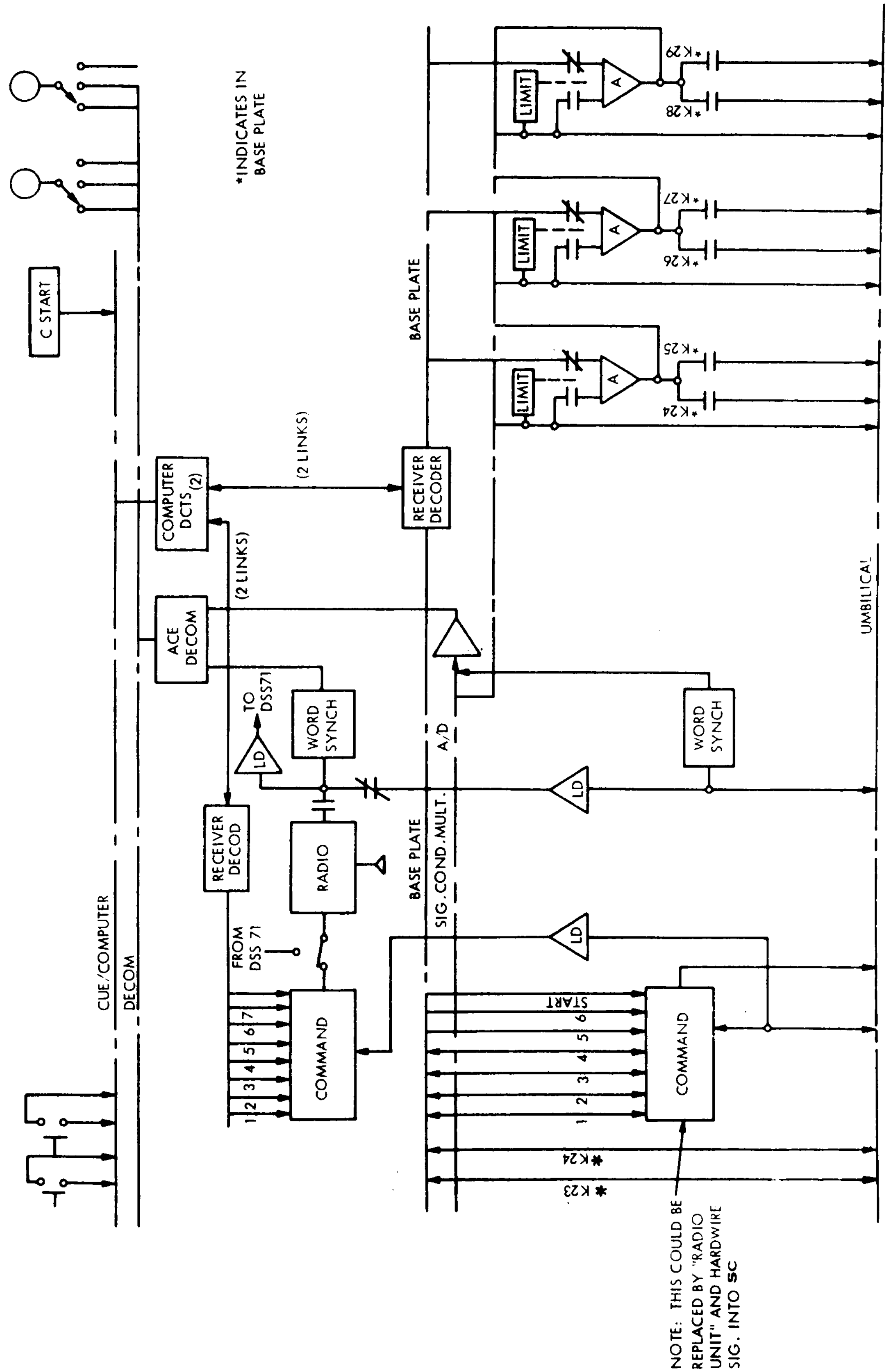


Figure B-4. Concept "A" Command Circuits

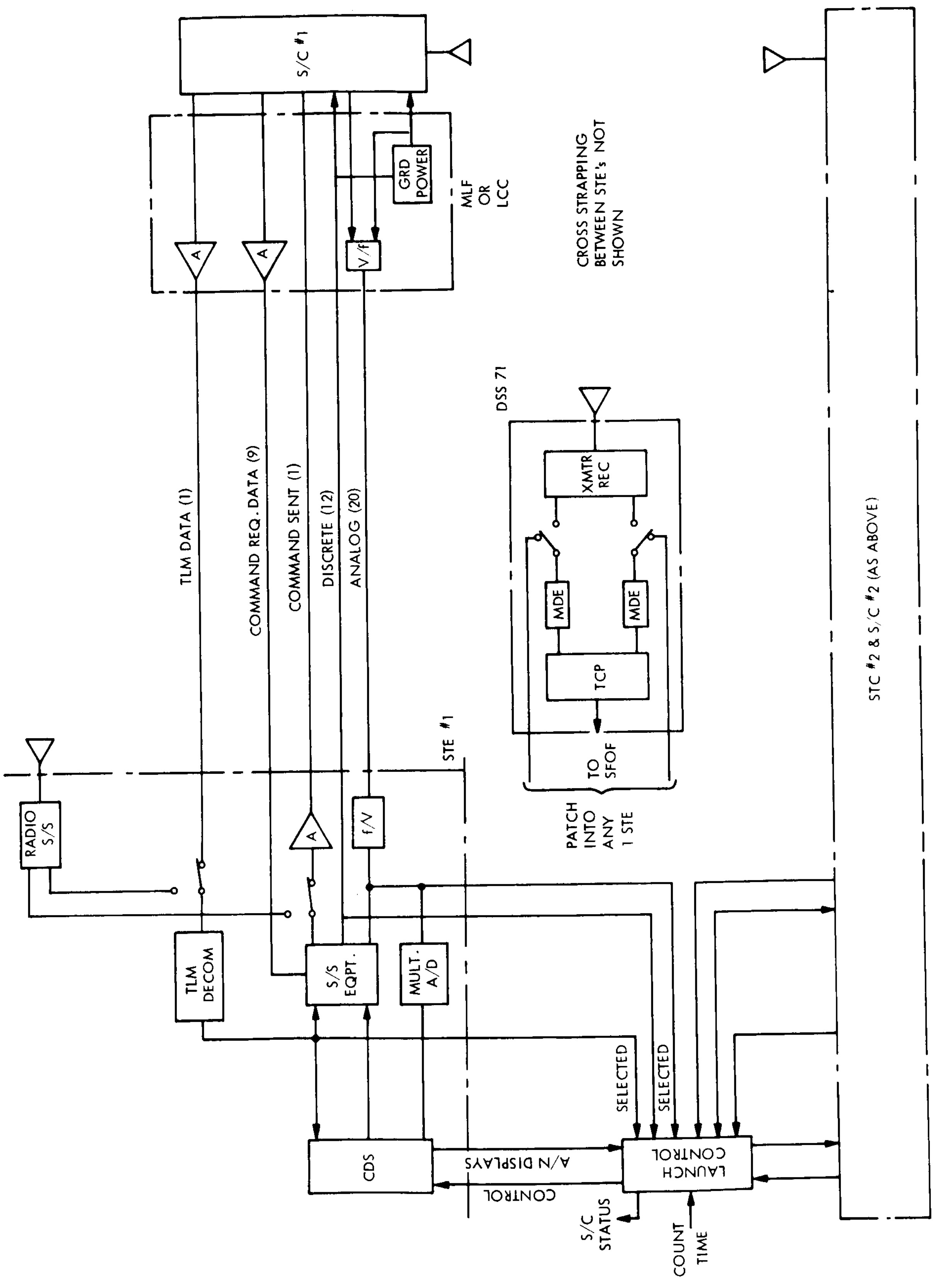


Figure B-5. Concept "C" General Launch Configuration

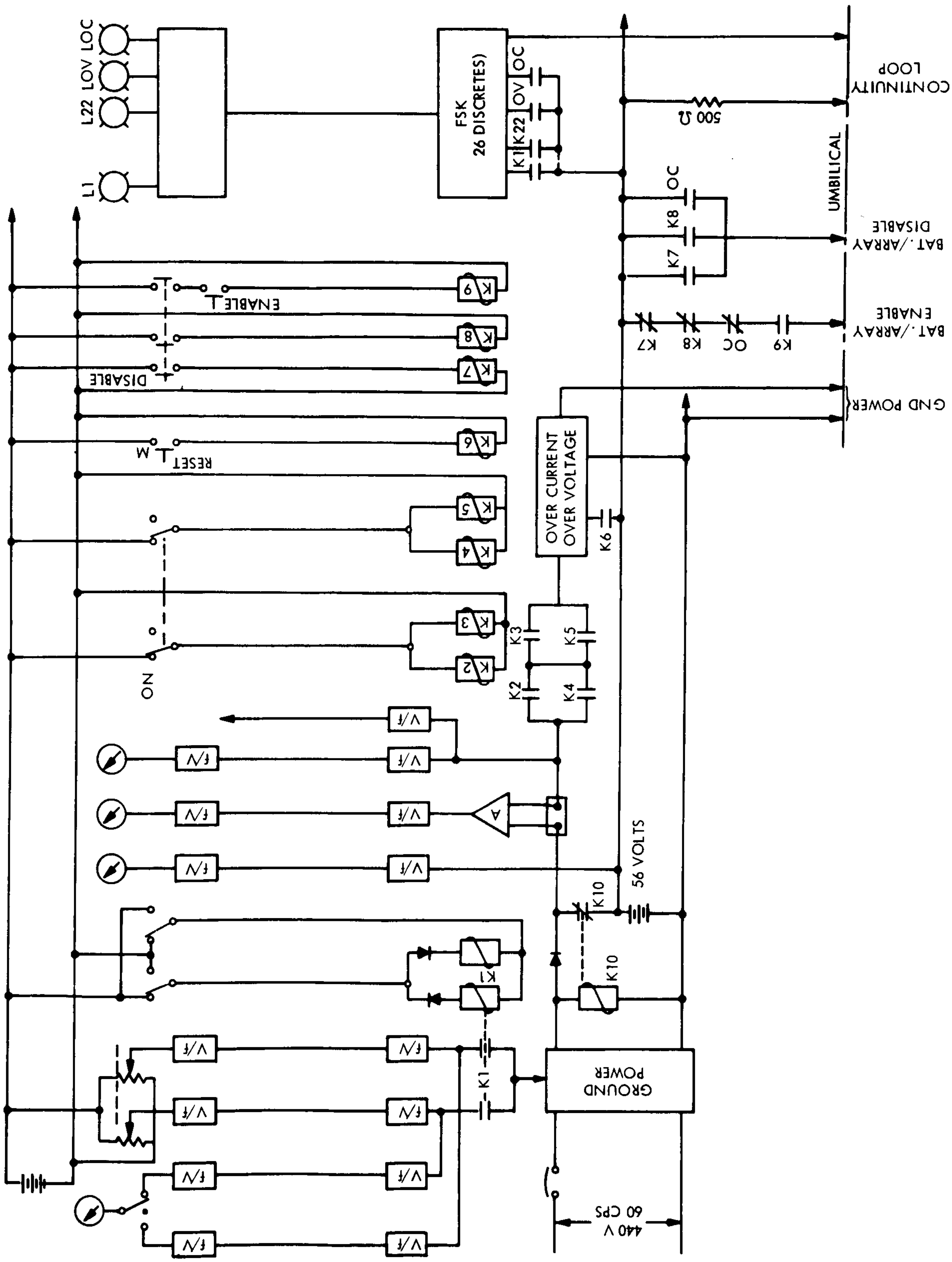


Figure B-6. Concept "C" Power Circuits

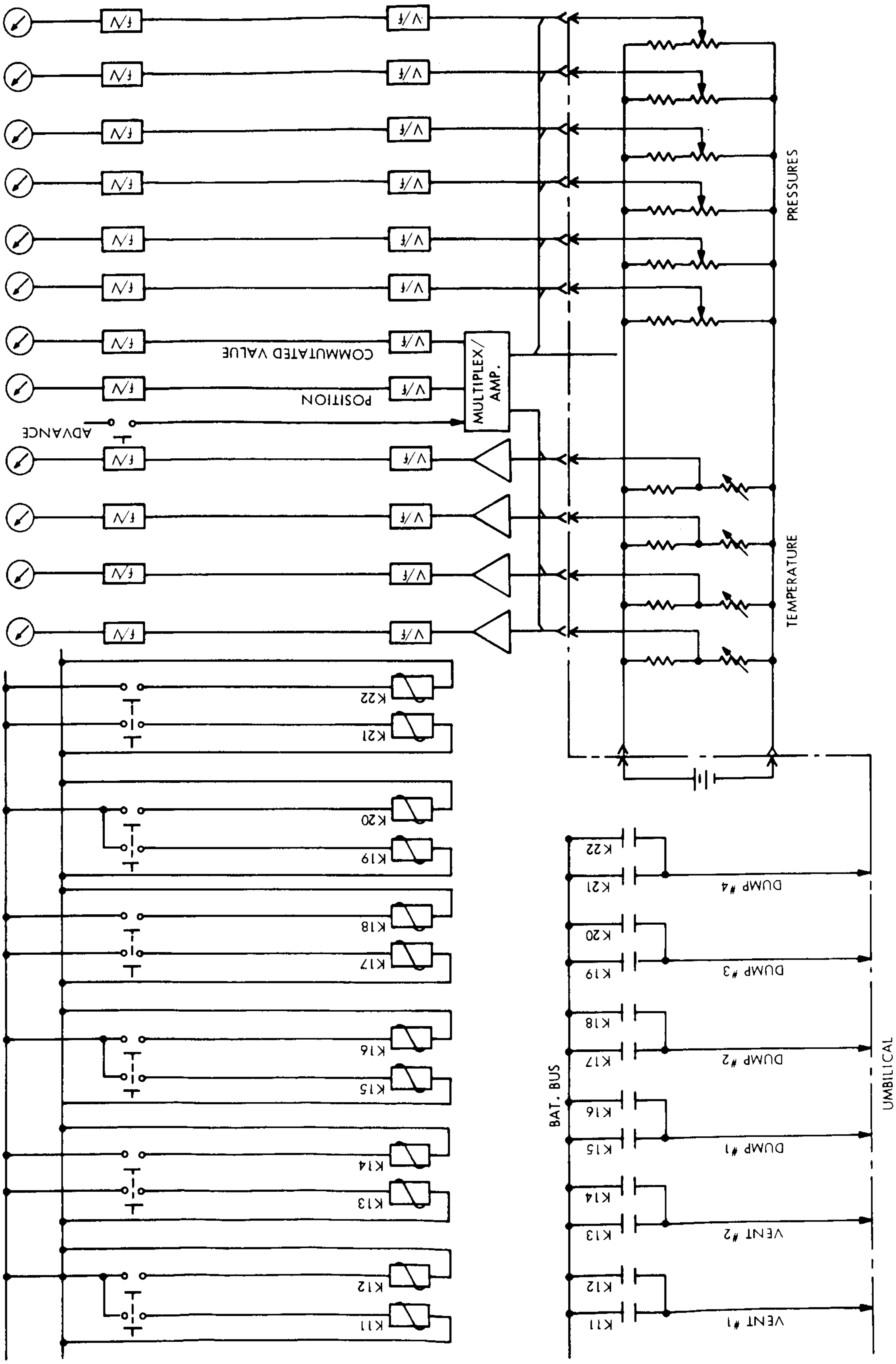


Figure B-7. Concept "C" Propulsion Safing Circuits

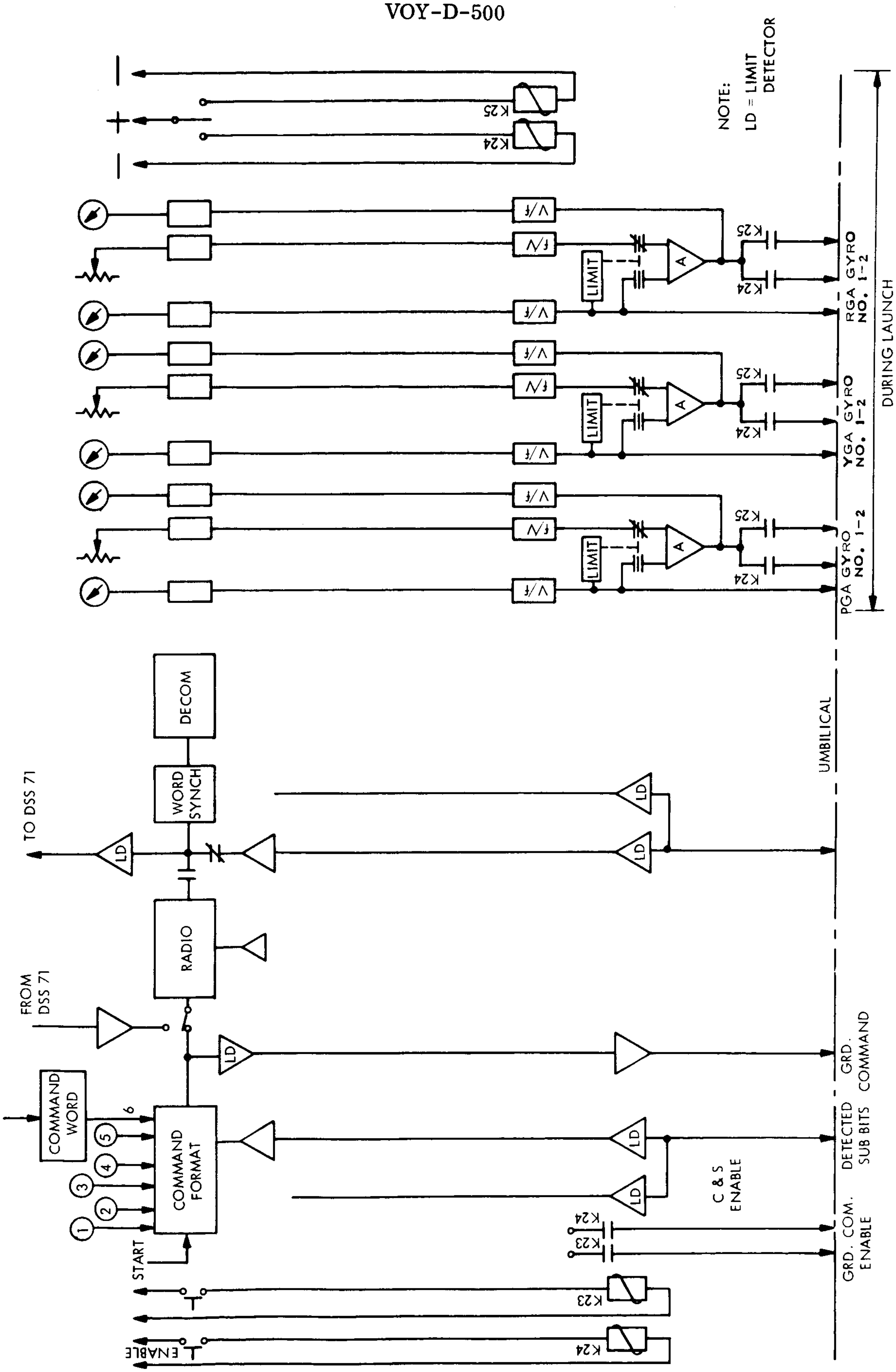


Figure B-8. Concept "C" Command/TLM/"G" and "C" Circuits

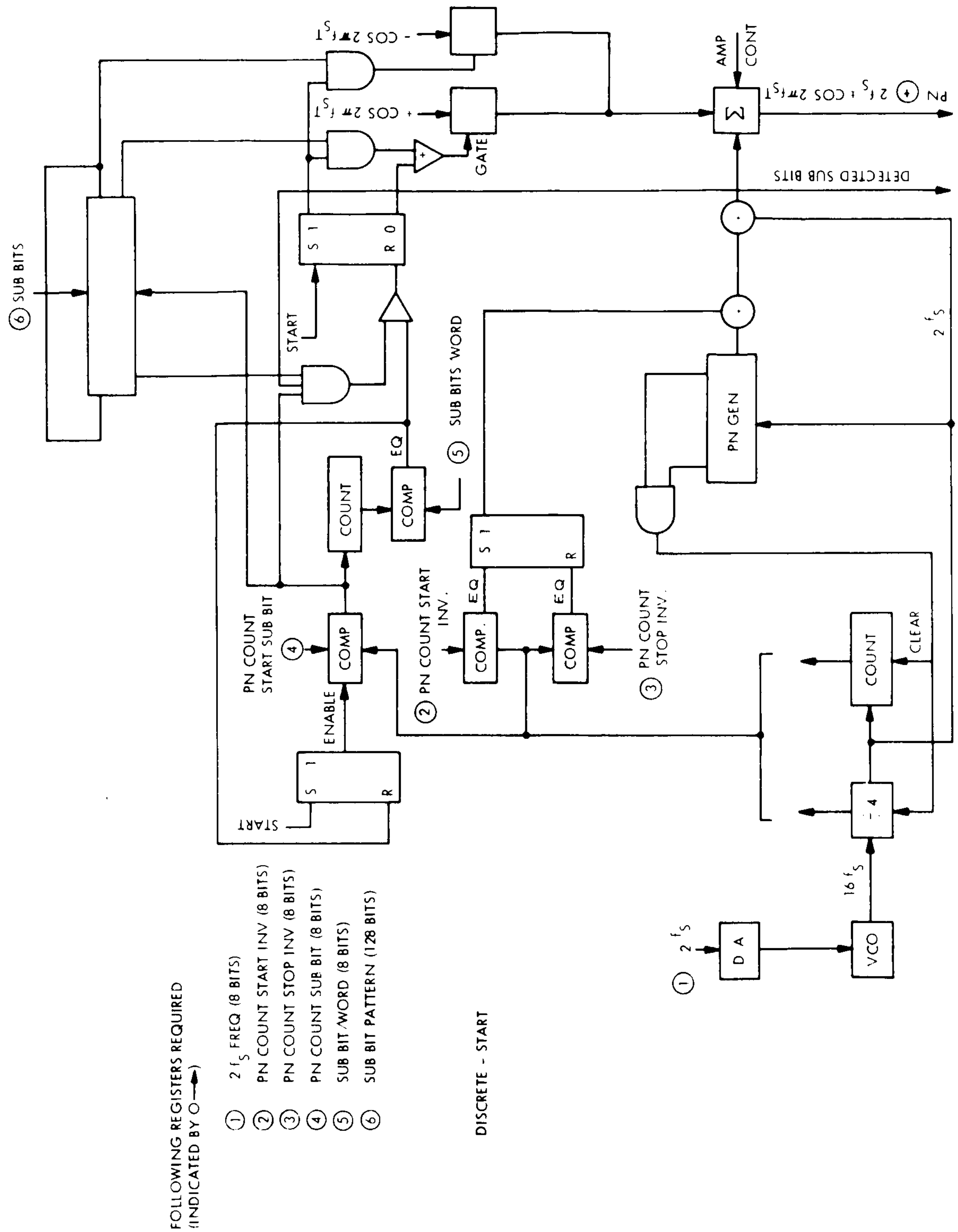


Figure B-9. Concept "C" Command Circuits

SECTION VI
VOY-D-600
MISSION DEPENDENT EQUIPMENT

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2.	Mission Operations	2
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SECTION VI
VOY-D-600
MISSION DEPENDENT EQUIPMENT

1. INTRODUCTION

Mission Dependent Equipment (MDE) is the operational data handling equipment required at Deep Space Network Sites (DSN), KSC and AFETR solely for the support of Voyager. MDE is required primarily for the handling of telemetry data and commands. It interfaces with and operates in conjunction with the general purpose data handling equipment at DSN sites, KSC and AFETR. MDE is part of the Mission Operations System (MOS) which also includes all of the equipment, personnel, and procedures required to conduct the mission operations.

This section contains a statement of the MDE requirements and a description of each item of Mission Dependent Hardware (MDHW) and Mission Dependent Software (MDSW) required to support a Voyager mission in 1973. The material contained in this section is based on the following concepts:

a. The Voyager 1973 Spacecraft is that defined in Section 2 of this report. In particular it has the following characteristics:

1. Telemetry

Low Rate (Engineering) Data: 150 bps

High Rate (Science) Data: 40.5 kbps

Data Compression and Error Correction Coding will be used on the high-rate channel.

Frequency multiplexing will be used to combine the two channels of data

2. Command

Command Rate: 1/2 bps

A psuedo-noise subcarrier will be used for command transmission

b. The Deep Space Network (DSN) is as defined in the Jet Propulsion Laboratories (JPL) Engineering Planning Document No. 283, Revision 2, "Planned Capabilities of the DSN for Voyager 1973," 1 January 1967. That document is the basis for the MDE definition, however the DSN is constantly evolving and as a result each item of MDE is extrapolated to reflect planned developments in the DSN. Information relative to the planned developments is available from two sources: (1) JPL Space Programs Summary, Volume III, "The Deep Space Network," published bi-monthly and (2) JPL Interoffice Memo NAR-67-17, 5 May 1967.

Each of these concepts is susceptible to change as the spacecraft design evolves and the DSN developments are implemented. However, the spacecraft system and the DSN are sufficiently defined that future changes will result in detail changes in the implementation of the MDE rather than changes to the basic concept.

2. MISSION OPERATIONS

The following is a description of the sequence of operations expected to be representative of the Voyager 1973 mission.

Normal space flight operations would require little control since the spacecraft is essentially automatic. There are intervals in the mission, such as just prior to maneuvers, which would require more control activity, but this is generally associated with sending only a few commands associated with timing of the operations. Provisions must be made for sending "emergency" commands to switch to other modes or to use alternate paths, if a principal operating mode appears to be degrading. Even in this case, there is generally not a sense of urgency as there is a long time available for action during most of the mission profile.

The monitoring function is essentially continuous, and the engineering telemetry data is almost always available. This data should be analyzed as received. The data should be extremely repetitive and easily compared. Any deviations should be cause to alert the operators so that they can perform any control functions as required.

Scientific operations will be very minor during transit to Mars. Once in Mars orbit this science data becomes all important and this will make up the preponderance of the data handled by the MOS. The control sequences, once established and set up, will be automatically controlled by the spacecraft, in the normal routine. The data received on the ground should be displayed and preliminary analysis carried out to make sure that the desired coverage is being obtained in a satisfactory manner. If this is not accomplished, some ground control of the spacecraft will have to be initiated.

In order to establish the routines, a description of the operations performed during a typical pass over a Deep Space Station (DSS) follows. This routine holds true for all operations regardless of mission phase, the differences from phase to phase being primarily in timing, number of commands or changes in modes.

2.1 PASS SCHEDULE

The Space Flight Operations Facility (SFOF) will generate the coverage schedule and transmit it to each station. It would include the following:

a. Time schedule for Voyager Spacecraft 1 and 2

b. Acquisition data for each

c. Command data load/sequence for each spacecraft

d. Special tests

e. Make orbit parameter measurements.

2.2 STATION SETUP

The DSS would setup in advance using the data obtained in Paragraph 2.1. In addition, a complete station verification using a telemetery simulator and command verification equipment would be performed periodically. The communication paths to the SFOF would be exercised by transmitting typical data.

2.3 SFOF SETUP

The SFOF will be on-line 24 hours a day and will, therefore, not require equipment setup. It will require running existing, mission independent self test routines. In addition, it will have to prepare the criteria for evaluating the expected data.

2.4 ACQUIRE FIRST SPACECRAFT

Using the data of Paragraph 2.1, the first spacecraft's noncoherent reference signal should be detected and tuned in. The DSS should record and decommutate the data and transmit it to the SFOF. Selected data will be displayed to assist the DSS operators. (Demodulated data is also sent to SFOF by high speed links for decommutation, processing and use at SFOF.)

The SFOF will receive the decommutated data and will perform the following operations continuously:

a. Check spacecraft's health

b. Verify spacecraft is in mode expected

c. Display data as requested by spacecraft engineers in attendance.

This data should be in engineering units. It should be able to be displayed on meter, lights, strip chart recorder, (analog or discrete), and on an alpha numeric display.

2.5 ESTABLISH RF LOCK

The DSS transmitter frequency will be swept thru the spacecraft command frequency. Once lock has been established, as indicated by the spacecraft telemetry signal, the transmitter frequency will be stopped at the nominal frequency and ranging and/or commanding can be accomplished.

2.6 ESTABLISH COMMAND LINK

The DSS transmitted signal can now be modulated with command data to establish PN lock by slaving the spacecraft command decoder clock to the DSS command rate. When this occurs, telemetry will indicate that command PN lock has been satisfied. This fact should be available to local DSS operations as well as the SFOF, since commands can now be sent.

2.7 VERIFY COMMAND SUBSYSTEM

The spacecraft command subsystem's circuits contain accept/reject logic that may be tested prior to sending "real" commands. This would be done to verify rejection of faulty commands received, and would increase ground personnel's confidence that commands can be loaded safely.

2.8 COMMAND SPACECRAFT

Following Paragraph 2.6 the spacecraft can be commanded either in real time (direct) commands or via stored commands. The stored commands are of two basic types. The first type goes into storage and is read out of storage sequentially as to location. The other type of stored command goes to twelve special time-to-go registers each of which controls one hardwired, discrete event. These time-to-go's are always timed by a fixed base. The time reference normally is started at separation and will run until executed. If a new value is loaded, time will start at the end of the new word (loaded in real time).

2.9 OPERATIONS CONTROL

During the above operations and subsequent ones, the operators and equipment in the SFOF (and at MSFC) are observing and comparing the telemetry data to expected values. Upon initial contact, the status should be predictable. As each command (real time or stored) is or should be executed, there should be changes (other than accept) in the spacecraft that can be verified by telemetry.

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designed for deep space exploration at Earth-referenced distances of more than 10,000 miles. The elements of the DSN comprise three major systems: (1) the Deep Space Stations (DSS), (2) the DSN Ground Communication System (GCS), and (3) the Space Flight Operations Facility (SFOF). The DSN is distinct from the other NASA networks such as the Space Tracking and Data Acquisition Network (STADAN).

3.1.1 Space Flight Operations Facility

The SFOF is a control center and data operations complex located at JPL in Pasadena, California. It contains the data handling, recording and processing, display and internal communications equipment as well as area and operations support capabilities needed by technical and operations personnel to perform data and information analysis, interpretation and evaluation, and as required, to determine and implement ground control of spacecraft. The primary function of the SFOF is to provide the means by which mission, flight and DSN control can be exercised by the agencies responsible.

The SFOF provides space and equipment for numerous functions, some being mission dependent or committed to the use of one particular project. The following equipment services are provided:

a. Communications - A wide variety of methods of communication are available within the SFOF and to areas outside the SFOF.

b. Displays - Display equipment consists of both computer driven displays and manually operated status boards. A master display board is maintained in the operations area.

c. Data Processing System - The data processing system is composed of a computer subsystem, a telemetry processing station, data processing control equipment and the data processing system programs.

d. Monitoring System - Monitor information from the DSS, the GCS and the data processing system is displayed on consoles for operations personnel.

e. Simulation Equipment - Hybrid computing equipment is provided for spacecraft simulation.

During normal operations, the SFOF is organized into teams. The DSN control team (personnel supplied by the DSN) operates the network. The project operations team is composed partly of DSN personnel and partly of project personnel. The DSN personnel are grouped according to function and are the interface with the DSN control team. The project personnel operate under the space flight operation director and are divided into three teams:

a. Flight Path Analysis Team

b. Spacecraft Data Analysis Team

c. Space Science Analysis and Command Team.

These teams assist in the definition of the standard mission, determine courses of action during a non-standard mission that will optimize the value of the mission and perform technical liaison required to achieve these objectives.

3.1.2. Ground Communication System

3.1.2.1. Present Ground Communication System

The present GCS configuration includes teletype, voice, high-speed and wideband microwave links. The actual technical control of the design and development activity in the NASCOM is the responsibility of the NASA Goddard Space Flight Center technical communication office. Therefore, advanced configurations of the GCS will be implemented in conjunction with the technological development of the overall NASCOM network except for additional or special requirements peculiar to the DSN.

The present GCS network is described in the following subparagraphs:

a. Teletype - The NASCOM network provides teletype communications between all overseas tracking and data acquisition stations and various computation and control centers. Circuit operating speeds are generally 66 wpm on overseas circuits and 60 wpm on domestic networks.

b. Voice - The voice link capabilities include telephone, commercial and leased facilities and four-wired, non-signaling conferencing networks within and between the DSIF stations and SFOF to permit world-wide voice communication on a party line basis.

c. High Speed Data - The high-speed data subsystem is composed of full duplex landline models which are capable of simultaneous transmission and reception of serial binary data at 2400, 1200, and 600 bits/second.

d. Wideband Microwave Data - A microwave capability between the Goldstone complex and the SFOF provides two wideband channels; a simplex video channel and a duplex data channel. The video channel has 60Hz to 6MHz bandwidth; the data channel has a 300 Hz to 96 kHz bandwidth.

3.1.2.2. GCS in 1973

Comsat Corporation is planning two synchronous satellites. One will serve the Atlantic area and the other, which will be operational in 1967, the Pacific area. By 1973 it is expected that all GCS overseas communications will be satellite derived and will be essentially the same types as are in use today; i.e., voice, TTY, HSDL and some shared wideband capability. Based on the above, the following may be extrapolated as the communication capability of a typical overseas DSS with the SFOF:

a. Teletype: 4 to 6 circuits

b. Voice: 1 to 2 circuits

c. HSDL: 4 circuits, 2400 bps

d. Wideband: 50 kbps one way from DSS to SFOF limited shared usage.

3.1.3 Deep Space Stations

To support Voyager, the Deep Space Instrumentation Facility (DSIF) will consist of one network of three 85-foot antennas and a second network of three 210-foot antennas. These stations are located at approximately 120 degree intervals in longitude with two stations at each such longitude.

A block diagram of a deep space station is shown in the left side of Figure 2. The technical capability of the DSS can be defined in terms of the tracking and communication data which are handled.

a. Tracking Data - The DSS has capability for automatic angle tracking, two-way doppler and precision ranging capability. The tracking data handling subsystem processes all tracking data for transmission to the SFOF via the GCS.

b. Communications - A command system data processing and transmitter phase modulation capability is provided at each DSS. A command verification transmission technique is used whereby the incoming command message is verified and translated by the Telemetry and Command Processor (TCP) into the proper spacecraft language is then transmitted.

   The basic design of the telemetry receiver provides for phase detection of the telemetry spectrum. In addition, interface equipment and the TCP will provide bit and word detection, special correlations and other special requirements.

c. Support and Monitoring Data - The DSS also includes equipment to display received signal strength, recording capability for all telemetry data and selected station parameters, and a status control and monitoring capability. This latter capability is used to monitor tracking and communications system performance and selected parameters. Out-of-tolerance conditions and status reports will be sent to the SFOF periodically.

3.2 MSFC

Mission operations at MSFC will require the capability of receiving and processing spacecraft telemetry data from the DSN ground communication system in order that the MSFC personnel (NASA and contractor), who comprise the Spacecraft Performance Analysis Team (SPAT), can support the operations during periods of relative inactivity (such as interplanetary cruise), without having to be physically at the SFOF in Pasadena. In addition to spacecraft telemetry, the SPAT will require current and projected spacecraft operating modes and status.

These requirements have a rough analogy to the functional requirements that have been implemented at the Huntsville operations support center for the Saturn program. The data acquisition and processing equipment required to tie the Huntsville operations support center into

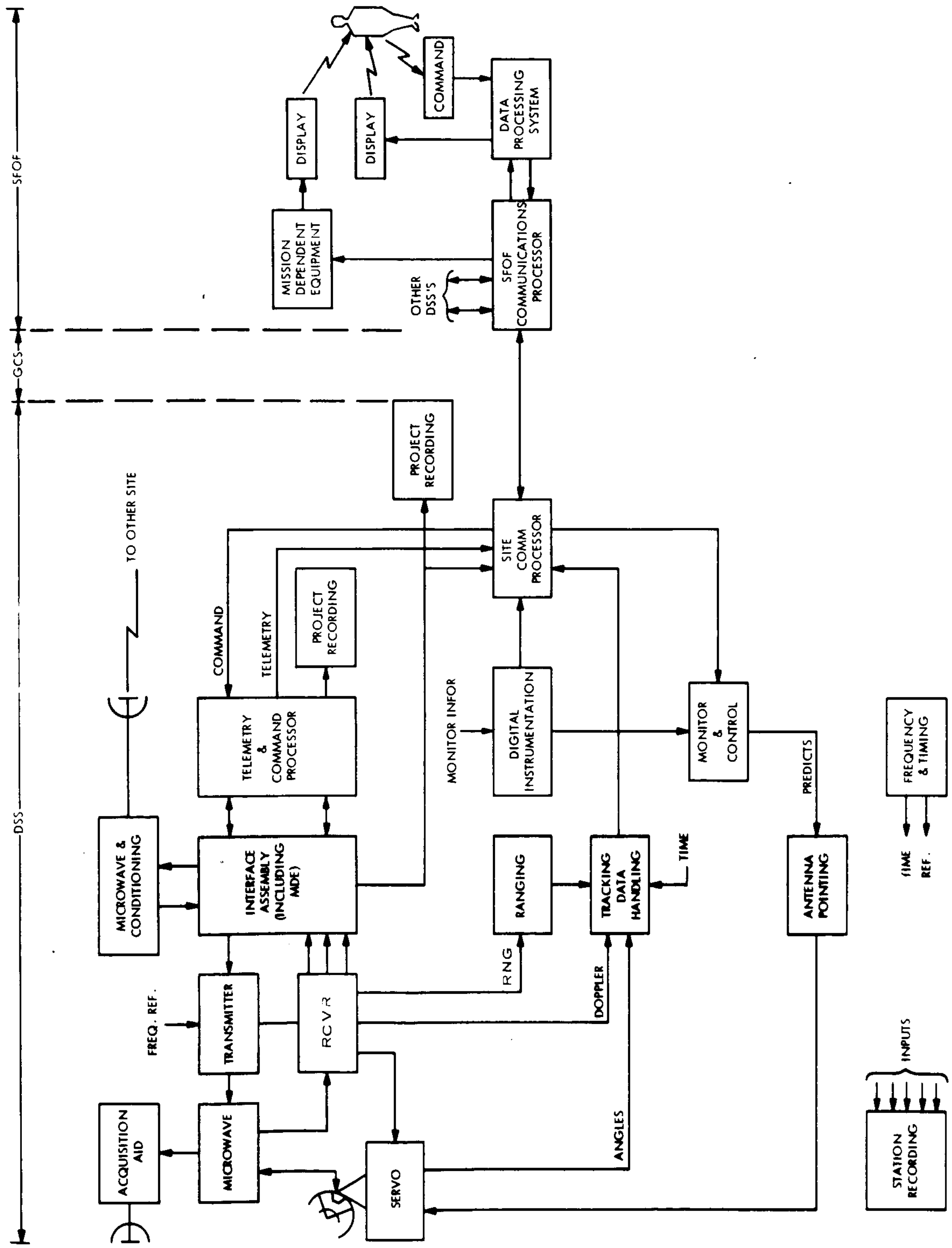

Figure 2. DSN Block Diagram

the ground communication system of the DSN will probably be a general purpose system and hence not be mission dependent. The console configuration and allocation, the controlling software, and the computer program to process telemetry data obtained from the SFOF will probably be mission dependent.

4. FUNCTIONAL REQUIREMENTS

4.1 MISSION DEPENDENT EQUIPMENT

The Mission Dependent Equipment (MDE) required by the DSN for Voyager 1973 is summarized in Table 1. This table identifies each item of MDE by number and name and lists the quantity of each item required. The quantity estimates are based on the following ground rules:

a. All DSSs will be required to process low rate telemetry data.

b. Only the DSSs with 210-foot antennas will need the capability to process high-rate data.

c. Scientific data display equipment will be provided at the SFOF and HOSC but not at the DSSs.

d. DSS72 at Ascension Island does not require command capability. All other stations require this capability.

e. The SFOF must have the capability to process raw data tape recorded at the DSSs.

f. The configuration of the Telemetry Processing Station at the SFOF is the same as that of the TCP at the DSS. If this assumption is not true, the configuration of Items 1 and 6 at the SFOF will be slightly different from the same items at the DSSs; however, the function will be the same.

g. The computation capability at HOSC is sufficient to handle the telemetry data and can be used to drive MDE displays.

h. An adequate data link is available from the SFOF at Pasadena to the HOSC at Huntsville.

Figure 3 represents the mission operation complex from the point of view of MDE. It shows the inter-relations of the MDE and the location of each item.

Table 1. MDE Summary

No.	Name	Deep Space Stations									Space Flight Oper. Facility	Huntsville Oper. Supt. Center	Total
		Echo (12)	Mars (14)	Tidbinbilla (42)	Booroomba (43)	Cebreros (62)	Rio Cofio (63)		KSC (71)	Ascension (72)			
1	Engr. Tlm Demod/Decom	1	1	1	1	1	1	-	1	1	1		9
2	Engr. Tlm Display	1	1	1	1	1	1		1	1			8
3	Science Tlm Demod/Decode		1		1		1				1		4
4	Tlm Simulation	1	1	1	1	1	1		1	1			8
5	Command Modulate and Control	1	1	1	1	1	1		1				7
6	TCP Software	1	1	1	1	1	1		1	1	1		9
10	Scientific Data Display										1	1	2
11	Spacecraft Simulation										1	1	2
12	Tlm Process & Analyze										1		1
13	Command Generate										1		1
14	Trajectory Determination Program										1		1
15	Midcourse Maneuver Program										1		1
16	Orbit Determination Program										1		1
17	Orbit Trim Program										1		1
20	Engr. Tlm Displays											1	1
21	Tlm Process and Analyze											1	1

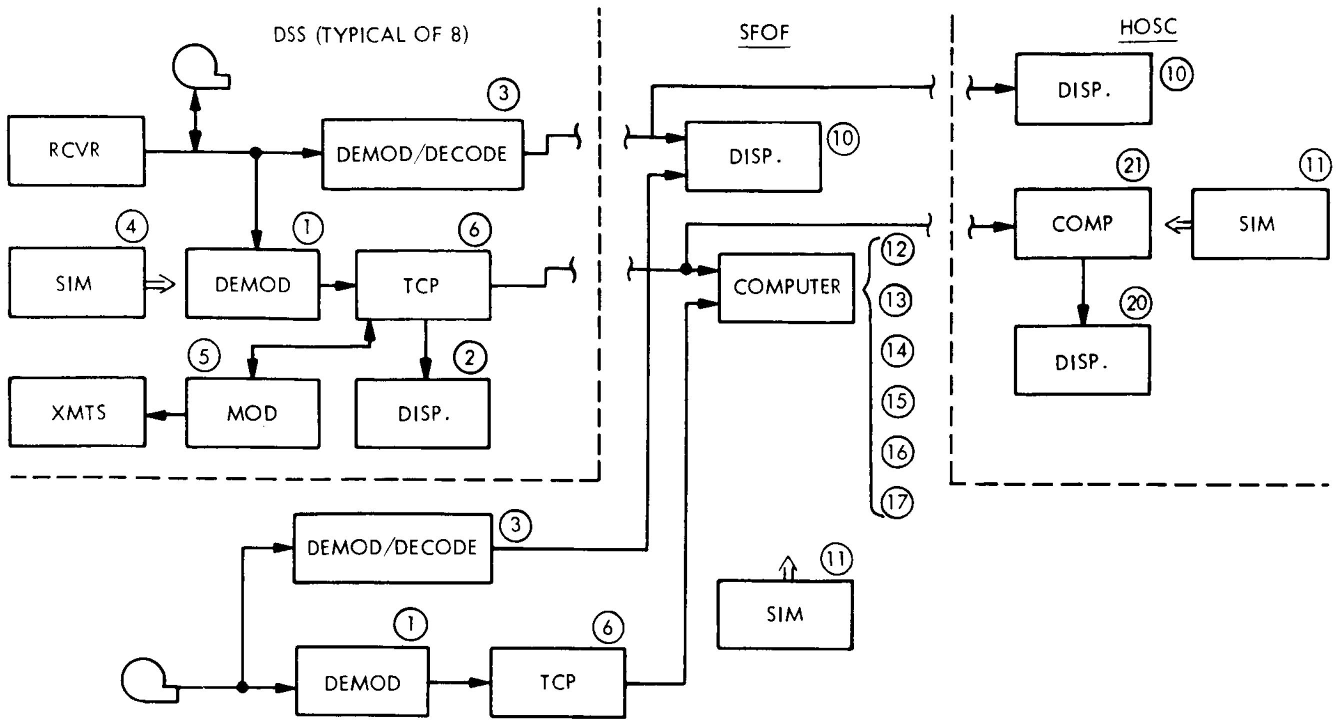

Figure 3. Mission Operation Complex

The data sheets which follow are an item-by-item description of the MDE. Each item is described by name and number and includes a brief statement of the function to be performed and hardware and/or software used (1) in the DSN per EPD283 and (2) in the DSN as extrapolated to 1973. Finally, each description includes remarks relative to the number of each item required and the approximate size of each item.

4.1.1. Number: 1

Name: Engineering Telemetry Demodulator/Decommutator

Function: Operate on the engineering telemetry signal from the telemetry receiver and remove the subcarrier; establish bit, word and frame sync; and identify each word in the signal with a unique number.

Description: (1) The present solution requires the use of mission dependent hardware and mission dependent software in the TCP. The MDHW would remove the subcarrier; establish bit sync, word sync, and frame sync; and present telemetry words in parallel to the TCP. The MDSW in the TCP would establish the subframe syncs (medium speed decks and low speed decks) determine the telemetry mode and generate the identification words.

(2) When the multimission telemetry demodulator is established as mission independent equipment, the MDHW will be eliminated and all mission dependent functions will be accomplished via MDSW.

Remarks: One set is required per DSS and one at the SFOF (total = nine). Hardware for the present solution would be less than one rack of equipment plus the required software.

4.1.2. Number: 2

Name: Engineering Telemetry Displays

Function: Certain spacecraft data is required by the DSS operations to aid them in the performance of their duties. Typical data required for display is static phase error, spacecraft receiver a.g.c., etc.

Description: (1) A certain amount of display hardware will be provided in the form of digital to analog converters with meter and strip chart recorders and lights or numeric displays. These would obtain their input from the TCP where software would be required to control the data that goes to the various displays.

(2) It is possible that, as the DSS data system evolves, general purpose display equipment will be provided as MIE for use with some MDSW.

Remarks: One set is required per DSS (Total = eight). Hardware to implement the present solution would require from one to two relay racks.

4.1.3. Number: 3

Name: Scientific Telemetry Demodulate/Decode

Function: Operate on the scientific data telemetry signal from the telemetry receiver and remove the subcarrier and recover the data from the error correction coding.

Description: (1) This function can be performed best by using hardware to demodulate the signal and recover the data. It would contain an integrate-dump circuit, a phase locked loop to acquire the bit rate, an analog to digital converter and a digital decoder.

(2) It is not anticipated that this solution will change during the time scale of Voyager 1973.

Remarks: One set is required at each DSS with a 210-foot antenna and one at the SFOF (total = four). Each set will consist of approximately one rack of equipment.

4.1.4. Number: 4

Name: Telemetry Simulation Equipment

Function: To supply representative telemetry signals for use in checking operation and status of other elements of the telemetry operational equipment.

Description: (1) This function can be implemented in two ways. First, hardware can be provided which will generate representative signals. Second, prerecorded magnetic tape can be provided for playback from the MIE recorder at the station. In either case equipment would be included to allow the introduction of noise to assess the equipment's performance at various signal-to-noise ratios.

(2) As the Simulation Data Conversion Center (SDCC) at the SFOF develops, responsive telemetry signals will be sent to the DSS's for use as a simulated signal source.

Remarks: One set is required at each DSS (Total = eight). The simulation hardware will consist of approximately one rack of equipment.

4.1.5. Number: 5

Name: Command Modulator and Control

Function: Command Data received from the SFOF via the TCP must be used to modulate a PN subcarrier and sent to the command transmitter. In addition provision must be made for generating spacecraft commands locally.

Description: (1) Hardware will be provided to accept commands from the TCP, generate the PN code and perform the required modulation and verification. This will include a control console to manually insert command messages into the TCP. In addition the TCP will require software to accomplish the command functions.

(2) It is not anticipated that this solution will change as the DSN evolves.

Remarks: One set required per DSS except DSS72 (total = seven). Each set will consist of approximately one rack of equipment.

4.1.6. Number: 6

Name: Telemetry and Command Processor Software

Function: Many of the individual telemetry and command functions require software elements in the TCP. This is an integrated set of computer programs for the TCP to perform all of the mission dependent functions required.

Description: (1) The TCP programs will receive, store, display, format and edit the engineering telemetry data and control its transmission to the SFOF and receive, verify, display, store and control transmission of commands from the SFOF to the spacecraft.

(2) As the multiple mission support capability evolves at the DSN, these programs may have to be integrated into an overall TCP Operating System.

Remarks: One set of programs required per DSS and one at the SFOF (total = nine). Each set will consist of one reel of digital magnetic tape.

4.1.7. Number: 10

Name: Scientific Data Display Equipment

Function: Video data from the spacecraft experiments will be sent to the SFOF from the DSS. Equipment is required to route the data from a particular sensor to the appropriate display device. Display devices will be required for each experiment which cannot be handled using general purpose equipment. For example, some equipment must be provided to produce pictures from the data gathered by the spacecraft photoimaging system. Likewise strip maps must be produced from the scanning radiometer data.

Description: (1) Special purpose equipment will be required for each spacecraft experiment. The exact number and nature of this equipment is not known at this time.

(2) This type of equipment will continue to be special purpose during the life of this project.

Remarks: One set of this equipment is required at the SFOF and one set at HOSC. (total = two). The size of this equipment is not known at this time.

4.1.8. Number: 11

Name: Spacecraft Simulation Equipment

Function: Provide equipment and software for simulation of the spacecraft in support of the Spacecraft Performance Analysis Team (SPAT) during operations.

Description: (1) Mission Dependent Software will be provided for the Simulation Data Conversion Center (SDCC) to simulate the Voyager spacecraft. In addition, spacecraft and selected subsystem models will be provided as necessary. This equipment can be used to support the SPAT during operations and it can also be used for training and checkout prior to launch.

(2) As the SDCC evolves, the amount of software required will increase with an attendant decrease in hardware.

Remarks: One set will be required at the SFOF and a second set will be required at the HOSC. The two sets will not be identical because of the differences between the mission independent equipment at the two sites. The physical make-up of this equipment has not been determined as yet.

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4.1.11 Number: 14

Name: Trajectory Determination Program

Function: Estimate the six injection coordinates and the injection time based on tracking data.

Description: (1) A computer program will be provided to perform these functions.

(2) As the third generation data processing system is procured these programs will be written to be compatible with its operating system.

Remarks: One set will be required at the SFOF. The programs will be provided on digital magnetic tape or tab cards as required.

4.1.12 Number: 15

Name: Midcourse Maneuver Operation Program

Function: Compute the midcourse velocity impulse required to modify the trajectory of the spacecraft in an acceptable way and at a favorable time. Convert the impulse to coordinates and magnitude and generate the necessary spacecraft commands.

Description: (1) A computer program will be provided to perform these functions.

(2) As the third generation data processing system is procured these programs will be written to be compatible with its operating system.

Remarks: One set will be required at the SFOF. The programs will be provided on digital magnetic tape or tab cards are required.

4.1.13 Number: 16

Name: Planetary Orbit Determination Program

Function: Estimate the orbital parameters based on tracking data.

Description: (1) A computer program will be provided to perform these functions.

(2) As the third generation data processing system is procured these programs will be written to be compatible with its operating system.

Remarks: One set will be required at the SFOF. The programs will be provided on digital magnetic tape or tab cards as required.

4.1.14. Number: 17

Name: Planetary Orbit Trim Operation Program

Function: Compute the velocity or impulse required to modify the spacecraft orbit in the desired manner. Convert the impulse to coordinates and magnitude and generate the necessary spacecraft commands.

Description: (1) A computer program will be provided to perform these functions.

(2) As the third generation data processing system is procured these programs will be written to be compatible with its operating system.

Remarks: One set will be required at the SFOF. The programs will be provided on digital magnetic tape or tab cards as required.

4.1.15. Number: 20

Name: Engineering Telemetry Displays

Function: Provide for display of the engineering telemetry data at the Huntsville Operations Support Center at MSFC in near real time. This will allow the engineering personnel at MSFC to support the operations personnel during the mission.

Description: The telemetry data will be analyzed by the HOSC Computer System and sent to the display equipment. The display equipment will consist of digital to analog converters with strip chart recorders and meter and digital displays and lights as required to display the data.

Remarks: One set will be required at HOSC. The nature and size of this equipment has not yet been determined.

4.1.16. Number: 21

Name: Telemetry Processing and Analysis Program

Function: Provide the software required by the HOSC computer system for telemetry processing and display.

Description: A set of computer programs will be provided to operate on the telemetry data received from the SFOF and perform the analysis and processing required to drive the display system. The program will have to be written to operate under the control of the HOSC Computer Operating System.

Remarks: One set will be required at HOSC. The programs will be provided on digital magnetic tape or on tab cards as required.

4.2 OPERATIONAL SUPPORT EQUIPMENT AS MDE

A non-flight spacecraft with the same performance characteristics and configuration as a flight spacecraft, such as the Proof Test Model, may be required at the SFOF and/or MSFC as a test bed for analysis and reconstitution of in-flight anomalies and for physical verification of alternate operation modes which are considered as candidates for working around an anomaly or failure. In this sense, the PTM spacecraft and the OSE, can be considered to be a major item of MDE. The fact that access to the spacecraft, from the OSE, is very great (direct access test points, special connectors and sensors, as well as coax and umbilical hardlines) allows ground failures to be induced so as to duplicate telemetry indications of in-flight troubles. The spacecraft and OSE are an almost ideal tool for the analysis of flight hardware problems. The experience of test personnel, who would have used the same kind of equipment earlier in the program, could substantially augment the SPAT personnel. This approach has been utilized in earlier, less complex, deep space programs. One significant attraction is the economics of the use, as the PTM and its OSE will have already completed their primary function and are available for use as MDE at negligible extra cost to the program.

5. SUMMARY

This section has treated the Telemetry and Command links between the ground and the spacecraft, at a general and functional level.* Design characteristics for Voyager 1973 have been used for purposes of projecting present understanding of mission operations. These characteristics are susceptible to change, however the impact of the changes will affect the detail design requirements rather than the functional requirements. The statement of MDE requirements should, therefore, be considered in this context and used for planning purposes only.

The Deep Space Network has been progressively implemented, in an evolutionary fashion, for some time. Its principal characteristic (insofar as identification of MDE requirements is

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* A more detailed description of the telecommunications aspect of the ground data handling problem can be found in VOY-D-313 of this report.

concerned) is that the general purpose approach, and the use of general purpose equipment, has been progressive. In this fashion, the economic impact of each new program, on MDE costs, has been minimized. It is entirely possible that this evolution will progress to the point that no MDE command or engineering telemetry hardware items will be needed to close the spacecraft SFOF loop in 1973. In this event, MDE hardware would be confined to special simulators, science telemetry hardware, etc. MDE software would tend to remain more substantial. This evolution should also be considered in any use of this preliminary MDE Requirements statement in current planning.

The area of MDE software for the SFOF Computer System is especially subject to change. This is caused by two things: (1) the DSN development plans call for the implementation of a third generation data processing system prior to the Voyager Mission. It follows that a third generation software system will be developed, the nature of which is completely unknown at this time. (2) The MDE software requirements must be based on the Space Flight Operations Plan and the DSN Operations Plan which are not yet available.

6. REFERENCES

1. Engineering Planning Document No. 283, Revision 2, July 1, 1967; "Planned Capabilities of the DSN for Voyager 1973"; Jet Propulsion Laboratory, C.I.T., Pasadena, California.

2. Space Programs Summary, Vol. III, Jet Propulsion Laboratory, C.I.T., Pasadena, California. (The Space Programs Summary is a six-volume bi-monthly publication designed to report on JPL Space Exploration programs and related supporting research and advanced development programs. Vol. III relates to the Deep Space Network.)

3. Milestone Report, "Preliminary MDE Requirements - 1973 Voyager Spacecraft", VOY-P-TM-16, August 18, 1967, General Electric Co. Voyager Project.

4. Jet Propulsion Laboratory Interoffice Memo. NAR-67-17, May 5, 1967.