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FINAL TECHNICAL REPORT

VOYAGER SPACECRAFT

Volume 4
OPERATIONAL SUPPORT EQUIPMENT

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I. INTRODUCTION

This volume is concerned with the approach to and functional descriptions of the major items of operational support equipment associated with all phases of assembly, testing, launch, and mission operations of the Voyager spacecraft.

The work of Task A has been used as the base for a functional analysis of operational requirements. From this analysis, operational requirements were consolidated into end-item requirements so that conceptual designs could be developed. Major differences between the OSE developed under Task A and that described here result from the replacement of the Saturn IB-Centaur or Atlas-Centaur boost vehicle by the Saturn V, and the requirement for the launching of two planetary vehicles as a single payload on the Saturn V. In addition, a higher level of automation is used throughout the spacecraft checkout system, both at the subsystem and system level, permitting more rapid, more repeatable and more reliable testing, better data retrieval, on-line data comparison and thermal analysis.

The operational support equipment includes bench checkout equipment, nine subsystem electrical OSE test sets, mission operations support test equipment, launch complex equipment, and assembly, handling, and shipping equipment. Since bench checkout equipment was excluded from the JPL guidelines, functional descriptions for it are not included here; this equipment is discussed in Volume 3 as it relates to in-process testing. The OSE (except LCE and most AHSE) is designed for use in a system test complex. In addition, MDE and MOSTE is designed for use at the DSIF in support of Voyager flights.

Figure 1 illustrates how all of the EOSE is combined and tested with the Voyager spacecraft at TRW prior to transfer of the spacecraft or any of its models to Goldstone, Capistrano, White Sands, the Eastern Test Range or to DSIF. This checkout and testing concept allows all EOSE configurations and potential problem areas to be evaluated before the OSE is shipped to remote sites. The system test complex is used in essentially the same form at all testing locations.

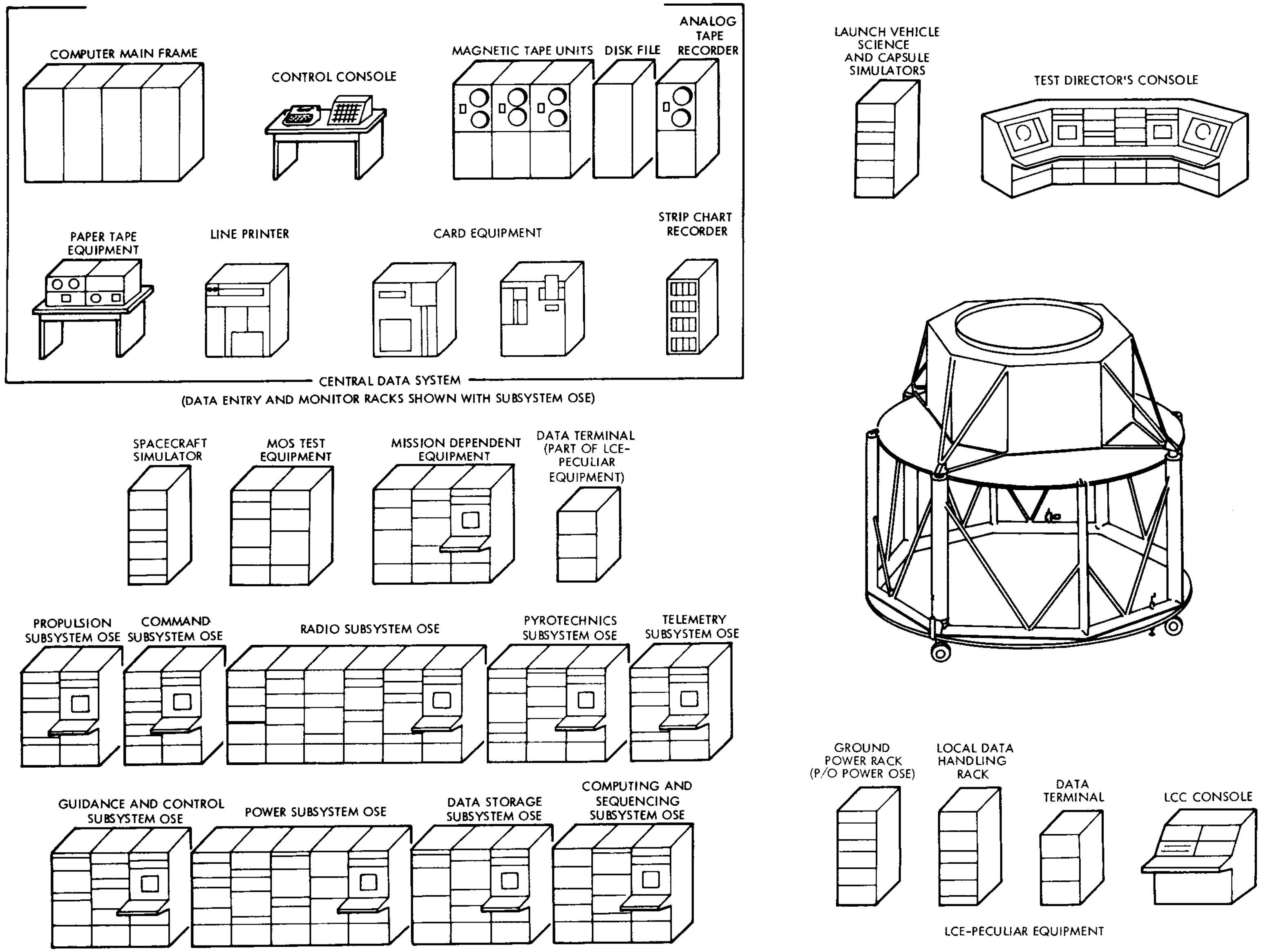


Figure 1. Compatibility Testing

Functional descriptions are included in this volume for each of the major items of OSE. While it is recognized that the mission dependent equipment is not considered part of OSE, we have employed it as a part of the STC in essentially identical form as its DSIF version in order to check MDE-spacecraft compatibility during all system tests. A functional description of the MDE is therefore included in this volume because of its inclusion in the STC.

The content of this volume is organized to present the material in five distinct sections, (1) Objectives and Design Criteria, (2) Design Characteristics and Restraints, (3) System Level Functional Descriptions, (4) Subsystem Level Functional Descriptions, (5) Schedules and Implementation Plan. Because of the significantly different objectives, design criteria, and design approach to EOSE and AHSE, items (1) through (4) are separately presented for these two equipment categories. The schedules and implementation plan for both EOSE and AHSE are presented in a single integrated section.

The requirement that the functional descriptions follow the general format and content of the Mariner functional descriptions does not permit a discussion of the various tradeoffs considered in arriving at the selected OSE baseline configuration. Therefore such tradeoff data has been included in Appendixes A through C. Tradeoffs in the selection of a method of spacecraft transportation are discussed in Appendix E.

II. ELECTRICAL OPERATIONAL SUPPORT EQUIPMENT

This section describes the EOSE test objectives and design criteria, operating characteristics and design constraints, and includes functional descriptions of both system level and subsystem EOSE.

System level EOSE consists of that operational support equipment used during integration, assembly, or test of the Voyager spacecraft and involves the system test complex (STC) used for both subsystem and system testing, launch complex equipment (LCE), and mission operational support test equipment (MOSTE). The STC described here includes the use of mission dependent equipment (MDE) identical to that supplied to the DSIF stations. Thus, a separate functional description has been written on this equipment even though it is recognized that MDE is not considered OSE. Similarly, the central data system (CDS), including the data entry and monitor rack (DE and MR), has been accorded a separate functional description, since it is a major part of the STC.

Subsystem EOSE are provided for each of the 10 spacecraft subsystems. Each subsystem EOSE, in conjunction with the central data system, provides the capability for completely testing its associated spacecraft subsystem prior to integration of the subsystem into the spacecraft. Subsystem EOSE, in conjunction with the central data system, is also used to perform integrated systems tests in the STC.

1. EOSE OBJECTIVES AND DESIGN CRITERIA

This section summarizes the overall purpose and design requirements for the electrical operational support equipment and mission dependent equipment required to support the Voyager 1971 spacecraft test program.

1.1 Objectives

The goal of the EOSE is to provide a tool to verify the design, performance, and flight readiness of the Voyager spacecraft. The EOSE will exercise the flight spacecraft through all of its primary and backup operating modes. It will be capable of verifying spacecraft design performance from fabrication through subsystem and system flight acceptance testing and of detecting spacecraft faults in time to hold a launch for corrective action.

Each subsystem EOSE provides the capability for completely testing its associated spacecraft subsystem before the subsystem is integrated into the spacecraft. In addition, subsystem EOSE is utilized along with a central data system to form the system test complex. Testing of a built-up subsystem will normally be done by the system test complex. The EOSE design will implement the maximum practical utilization of automatic testing techniques.

During the design of each drawer in each subsystem EOSE, consideration was given to the use of that same drawer (or rack) in the bench checkout equipment. This commonality of approach provides for reduced EOSE costs, consistency of test data, and the capability of the subsystem EOSE to provide fault isolation to the replaceable assembly.

The application of portions of the subsystem EOSE to be used as launch complex equipment was also considered during the subsystem EOSE design. Wherever similar functions occur during system testing and launch operations, such as providing power to or control and monitoring of the spacecraft, the subsystem test equipment design is also provided as LCE.

The EOSE will incorporate design and testing features providing safe operating conditions for test personnel, the spacecraft subsystems, and the EOSE units themselves. EOSE designs will be thoroughly reviewed by the Voyager safety office to determine that safety design practices are incorporated in EOSE design. Examples of these practices are:

- Special markings on dangerous or critical equipment areas
- Safeguards to prevent out-of-sequence or out-of-tolerance commands or stimuli from reaching the spacecraft during automatic and manual EOSE operation.
- Cabling connectors keyed to prevent attachment to the wrong sockets.
- Verified test procedures for all test and troubleshooting operations.

Prior to mating an EOSE with a spacecraft subsystem, the EOSE is functionally tested with a spacecraft simulator. Monitors and alarms are provided on each EOSE which indicate faulty EOSE operation. CDS programs are thoroughly tested with the spacecraft simulator before subsystem tests, to search for deleterious command signals. Each EOSE incorporates self-test and fail-safe design features.

1.2 EOSE Design Criteria

1.2.1 EOSE Criteria

Standardized proven circuits are used to the maximum, that is, circuits that have demonstrated reliability on previous programs or through extensive testing. The number of electronic piece-part types employed in the design of the EOSE is held to a minimum. Circuits and subsystems within the EOSE are designed to facilitate inprocess testing at the module and subassembly level in order to detect such anomalies as workmanship errors and miswirings which were not revealed by other methods. Adjustment settings required in the EOSE remain sufficiently stable to permit a complete span of test scheduling without adjustment during the span.

Data obtained from any prelaunch checkout is arranged or tabulated in such a manner that it can be compared readily on a common basis with any subsequent data.

The EOSE incorporates features to protect against damage to itself or the spacecraft caused by failure of the EOSE, spacecraft, or the test facilities. The automatic mode of testing provides sufficient safeguards in its programming to prevent damage resulting from improper sequencing. Conditioning of the spacecraft to a safe mode is provided in case of test facility failure. Temperature alarm monitoring is furnished in the EOSE while the spacecraft or any of its subsystems are in thermal environmental test. Tests on "one shot" devices such as pyrotechnics are included during checkout. Tests to verify status and interfaces to a maximum degree are employed during appropriate in-line or acceptance tests.

The design of EOSE includes flexibility such that the same equipment, with minimum modification, can be used for 1971 and subsequent spacecraft. The design considers multiple use of specific end items for more than one operational requirement. Upgrading capabilities of end items to perform multiple functions is considered a standard design guideline except where costs become excessive, or simple and conservative design is compromised.

The design of the EOSE is such that maintenance operations can be performed safely and efficiently. Access is provided for equipment repair or replacement, for test purposes, for equipment fabrication and assembly, and for inspection. Guards and safety devices are provided for hazardous or delicate hardware. Self-testing of the EOSE is used to assist in troubleshooting both the spacecraft (to determine proper EOSE operation) and the EOSE (to fault isolate).

The EOSE design reflects its intended use of operating in a laboratory-controlled environment. The specific environmental requirements will be established during Phase IB.

All elements of the EOSE are transportable by truck or air. The size, weight, interconnecting, and functional features of the EOSE and cabling are suitable for expeditious take-down, transport, and assembly in support of any testing required at remote test facilities. In general, the time necessary to transport and set up the EOSE will be equal to, or less than, the time necessary to transport and prepare a spacecraft for testing. A validation test will be conducted on the STC and related facility support equipment (without a spacecraft) in such a manner that its complete readiness to support the system test is established and verified.

No test or checkout operation provided by EOSE will inadvertently overstress a spacecraft or its associated subsystems. During type approval qualification tests, parameters exceeding the design limits are imposed on the equipment under test in order to ascertain the proper design tolerances. The EOSE provides artificial stimuli only in those situations where it is not feasible to provide normal system inputs.

1.2.2 Mission Dependent Equipment

Two channels of MDE when installed in a DSIF provide the capability of simultaneous operation with each of the two spacecraft in flight. The

MDE is designed to be completely compatible with the DSIF latest design as planned for 1971.

Mission operations system test equipment is provided with the MDE which is capable of completely verifying proper operation and fault isolating to the provisioned spares replacement level. The MOSTE does not depend on the availability of the DSIF equipment to verify MDE operation.

Emphasis is placed on reliability of design of the MDE in-line operational equipment. Equipment redundancy and testing prior to launch will provide confidence in MDE operation during the mission.

1.2.3 Software

Extreme care will be taken in the design and configuration control of computer programs. Computer program design is centered around system flow charts and detail flow charts. These charts clearly describe each program and will be verified against the test and operating procedures generated for both the EOSE and the spacecraft. The system flow chart provides complete presentation of the program to ensure that all facets of test operations are covered. In addition to the system flow charts, configuration control of the documentation resulting from software design includes:

- Program flow charts
- Subprogram flow charts
- Program acceptance test plan
- Program timing charts
- Program control sequencing decision tables

2. EOSE CHARACTERISTICS AND CONSTRAINTS

2.1 Characteristics

Basic EOSE test capabilities are established by the design requirements provided in the JPL General Specification, "Performance and Design Requirements for the Voyager 1971 Spacecraft System." These requirements have been applied to the preliminary design of all subsystem EOSE, mission dependent equipment, mission operations system test equipment and system level EOSE, which includes the central data system, system test complex and the launch complex equipment.

Subsystem and system EOSE have the capability of automatic operation by means of computers in the central data system. Complete system and subsystem test sequences can be timed and stimuli-controlled. Monitored functions can be displayed via cathode ray tube, discrete status displays, and alphanumeric printout. Some specific tests naturally will require operator intervention, such as varying frequencies or recording information displayed on recorders, X-Y plotters, and oscilloscopes. Each set of subsystem EOSE is capable of providing limited manual test operation with its own control and display equipment. The manual mode of operation is provided primarily as a backup.

Each subsystem EOSE provides the following JPL-specified test capabilities:

- Complete testing of the subsystem, as provided for by the normal subsystem test circuitry.
- Manually controlling its subsystem to any operating condition and in any sequence provided for by the subsystem test circuitry.
- Performing any extensive or complex subsystem test routine expeditiously, correctly, and repetitively.
- Providing subsystem power normally supplied by the flight spacecraft power subsystem.
- Providing the capability to vary flight spacecraft subsystem parameters, or externally supplied signals, for performance and margin testing.
- Isolating trouble to the subassembly replacement level.
- Providing the required grounding networks for instrumentation and power in accordance with the requirements specified in Section 4.1.
- Manually controlling and visually monitoring all flight spacecraft subsystem EOSE interfaces.
- Monitoring and recording all subsystem functions provided by the subsystem test circuitry and all simulated interface functions.

- Providing connecting subsystem simulated interfaces, e.g., necessary signals and loads, required to satisfy the subsystem test requirements.
- Providing automatic self-test capability without interruption of spacecraft operation for isolation of problems to EOSE or flight subsystems.
- Limiting loading effects on a flight spacecraft function to less than 1 percent of the measured function.
- Limiting current to all high power interfaces with the flight spacecraft subsystems or other EOSE.
- Protecting spacecraft in the event of EOSE failure or operator error.
- Alarm monitoring while flight spacecraft or any of its subsystems are under environmental test.

The mission dependent equipment is capable of operating with two spacecraft, individually or together, from the DSIF sites. The primary (inline) functions consisting of command generation, telemetry detection, and computer buffering are provided, as are the secondary (supplementary) functions of command detection and spacecraft status display. Also provided are tertiary (test and maintenance) functions consisting of telemetry detection testing, simulated telemetry data generation, spacecraft simulation, station simulation, and general purpose measurement and calibration.

In the automatic mode of testing, the DCS computer selects the required stimuli or commands provided to the spacecraft by the subsystem EOSE. Any automatic test sequence can be interrupted and specific test routines inserted by the test director. The computer can control and monitor the complete operation of a flight mission. Commands and stimulation can also be provided to the spacecraft manually from each set of subsystem EOSE.

Complete monitoring and display of all spacecraft monitored functions during system tests are provided at the test director's console. These displays consist of cathode ray tubes, indicator lights, and alphanumeric hard copy printout. Each set of subsystem EOSE provides the capability to monitor its subsystem-peculiar functions via displays controlled by the computer during system or subsystem test. During manual operation, the functions peculiar to each subsystem EOSE are displayed via meters, scopes, and special digital displays.

During subsystem testing, each subsystem EOSE provides power to its associated spacecraft subsystem from STC or LCE or power sources.

2.2 Constraints

2.2.1 Human Engineering

Accepted principles of human engineering will be followed, such as:

- Intellectual, physical, and psychomotor capabilities of the intended user.
- Human space limitations for operation and maintenance.
- Visual and auditory perceptual requirements.
- Arrangement and readability of control and instrument panel displays.
- Safety factors minimizing the potential for human error in the operation of equipment.
- Sequence of operational requirements for the operator-served equipment.

2.2.2 Standardization

Standardization of EOSE design has been utilized on previous programs at TRW. Based on this experience, the following principles will be applied.

- Standardization of EOSE is employed throughout the design. Standardization of digital circuit cards via the use of standard integrated circuits mounted on printed circuit cards is included in the design. Standardization of circuit card holders and digital drawers is applied. The EOSE digital circuitry is standardized as to internal pulse characteristics and type of circuitry. Analog circuitry is standardized to the practical limit, insuring that special types of resistors, capacitors, etc., have minimum application.

- Power supplies for all EOSE are standardized as much as possible. Specific voltage and current power supply modules are integrated to provide power for each rack. All EOSE design has considered voltage standardization in its power requirements.

- Standardization of connectors is used throughout the design of the EOSE. The location of connectors on drawers is standardized. All connectors and plugs are RFI protected with tin cans.

- Commercial test equipment is standardized in all EOSE design to the extent that the same type of equipment is used where possible. For example, a standard oscilloscope is used with its plug-in units providing the applicable function. Meters of the same shape and face design are also used.

- Knobs, switches, and indicator lights are standardized as well as their location on panels.

2.2.3 Cabling

EOSE cable design has incorporated proven techniques which provide operator safety, serviceability, and ease of installation.

- EOSE distribution equipment including system cabling, connector junctions, distribution boxes, and miscellaneous in-line equipment contains a minimum of 10 percent spares above design requirements.

- The length of all cabling between the EOSE and the spacecraft or its subsystems is limited to 150 feet for test cables and 1500 feet for umbilical cables, as specified by JPL.

- The following JPL requirements were applied to connector wiring. Connectors in spacecraft to EOSE circuits have pin-socket junctions oriented with sockets on the spacecraft side. Connectors in circuits carrying power have the pin-socket junction oriented with sockets on the power source side. Connectors in circuits carrying power both ways through the connector have oriented pin-socket junctions as determined by analysis. For any single EOSE connector grouping, the connectors are differently keyed.

- The following wire servicing techniques which have proved effective on previous programs is applied. Cable and harness support is provided to protect insulation from cold-flow and abrasion. Assembly interconnect cabling is attached to and supported by the chassis. Connectors are used and mounted such that straight and free engagement of contacts is assured. Interconnecting EOSE cables are comprised of standard wire-service types wherever possible. All service loops which are handled or twisted in normal use are designed to provide the required flexibility. Strain relief is provided for cable conductor interconnects.

2.2.4 Grounding

The grounding and shielding requirements for the EOSE are included in Paragraph 5.3, Section III, Volume 1.

2.2.5 Environmental Requirements

All EOSE is designed using JPL Spacecraft Environmental Specification on Ground Support Equipment (30505B) as a guide.

For STC equipment operating in temperature and humidity controlled areas such as laboratories, ESF, VAB, etc., the design temperature range is 65 to 80°F and 50 per cent relative humidity. Equipment need not be designed to operate during an air conditioner failure; however this capability is desirable. For equipment operating in areas that are not air conditioned, the operating design temperature is 25 to 120°F and 95 per cent relative humidity. The design goal for storage temperature limits for this requirement are 32 to 120°F.

Design requirements for the control of sand and dust, salt spray, rain, and fungus are applicable only for equipment located or operated in an outdoor environment at any of the operational areas.

2.2.6 Racks and Accessories

The following requirements are applicable only to the new equipment purchased for the Voyager program. The standard rack for Voyager EOSE is specified in JPL Specification 30609A. For convenience in handling, rack multiples greater than dual are not used. Paint type and colors are specified in JPL Specification 30600. All equipment in the Voyager EOSE requiring painting is in conformance with this specification.

Each subsystem EOSE contains a writing surface. Panel faces, consoles, racks, rack chassis, and other equipment requiring identification is identified and marked by means of engraving as specified by JPL. It is a requirement that all dials, lights, knobs, switches, and any other indicators and controls be labeled so that the function they perform is clearly indicated.

Sufficient cooling will be provided in the racks to ensure an extended operational life time for the equipment. The louvers or grills are located in either the bottom, front, or back of the rack; they are used in the top only if necessary. Due to the proximity of adjacent racks, louvers, and grills are not placed in the sides of the racks. Fans for air circulation are electrically noise-free.

3. SYSTEM LEVEL FUNCTIONAL DESCRIPTIONS

This section contains functional descriptions for each of the major items of EOSE, the system test complex (STC), launch complex equipment (LCE), and mission operation systems test equipment (MOSTE). The mission dependent equipment (MDE) is also described although in the strict sense it is not OSE. However, it is employed in the STC in essentially identical form as in the DSIF and is therefore relevant to the operation of the STC. Additionally, a functional description of the central data system (CDS), a part of the STC, is included as a separate entity since it operates almost independently in supporting units tests, subsystems tests, and system tests through its data entry and monitor racks (DE and MR). These racks are associated with each of the subsystem test sets (see Section II. 4) in the STC and provide a common interface between the general purpose computer and the subsystem EOSE in the primary automatic test mode. A manual backup mode is also provided in the STC by virtue of an independent telemetry decommutation system in the telemetry subsystem test set.

3.1 System Test Complex

The system test complex (STC) will be capable of testing a complete spacecraft or a complete individual subsystem during all phases of design qualification and flight acceptance testing. It includes the hardware and software to stimulate, control, measure, and record the performance of spacecraft subsystems manually or automatically; to detect and isolate the cause of defective performance in either the spacecraft or any of the STC elements to the provisioned spare replacement level.

The STC includes the nine subsystem EOSE sets, a set of mission-dependent equipment (MDE), the central data system (CDS) (including software), and one each of the following hardware items: test director's console, spacecraft simulator, launch vehicle simulator, capsule simulator, science subsystem simulator, and MOS test equipment.

3.1.1 Capabilities and Configuration

During a systems test the STC operates nine separate subsystem test sets in an integrated sequence. The integrating unit is the central data system computer, which controls the activities of, and accumulates the data from, each of the subsystem test sets. During the test of one or more individual subsystems, only their individual EOSE units are integrated by the computer. Two features of this approach are:

a) Identical test equipment is used throughout all subsystem and system testing, thus providing data on each subsystem directly comparable for trend analysis.

b) The EOSE configuration is highly flexible, allowing fast turn-around from system test to subsystem test and, in fact, simultaneous testing of a subsystem and the balance of the system.

The five main functions performed by the STC are system test, subsystem test (including subsystem retest where required), fault detection and isolation, performance data gathering and record keeping, and trend analysis.

The STC is organized for these functions in a manner making maximum use of the computer's capability while optimizing its interface with the test conductor and subsystem engineers by CRT displays.

Figure 2 pictures the EOSE complement in the STC. The largest number of wire connections are between the subsystem EOSE's and the computer, and between the test sets and the spacecraft. The spacecraft is placed in a clean area separated from the EOSE and personnel. Communication between the test director and the subsystem engineers is by intercom, although direct visibility is possible since they are in the same central room.

During STC operation, commands are originated in the computer and sent to the MDE. The MDE command encoder is equipped for exclusively manual setup and initiation, since there is no requirement for computer-generated commands in the DSIF stations. Therefore, additional logic to provide the ability to accept computer-generated commands is required in the MDE for STC use.

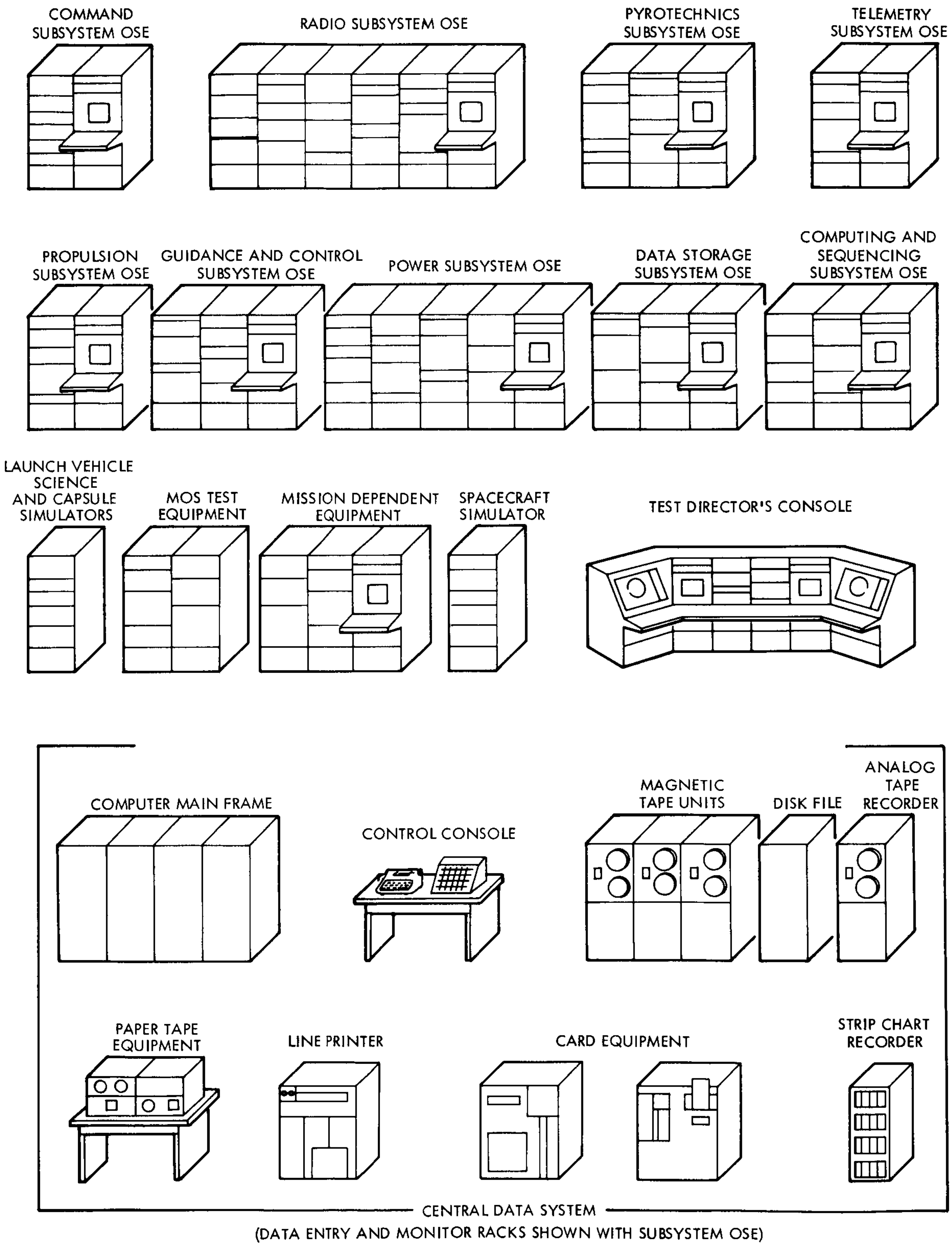


Figure 2. EOSE Complement in the System Test Complex

In addition, the MDE command encoder for STC has the capability to emit direct command words (not PN coded) at a rate faster than 1 bit/sec (probably 20 bits/sec) for insertion into the output command decoder register in the spacecraft (via hardline). This is required to run the four-hour-mission test, and in other cases.

The two-position test director's console is illustrated in Figure 3. During system test, it is anticipated that two positions are occupied by the

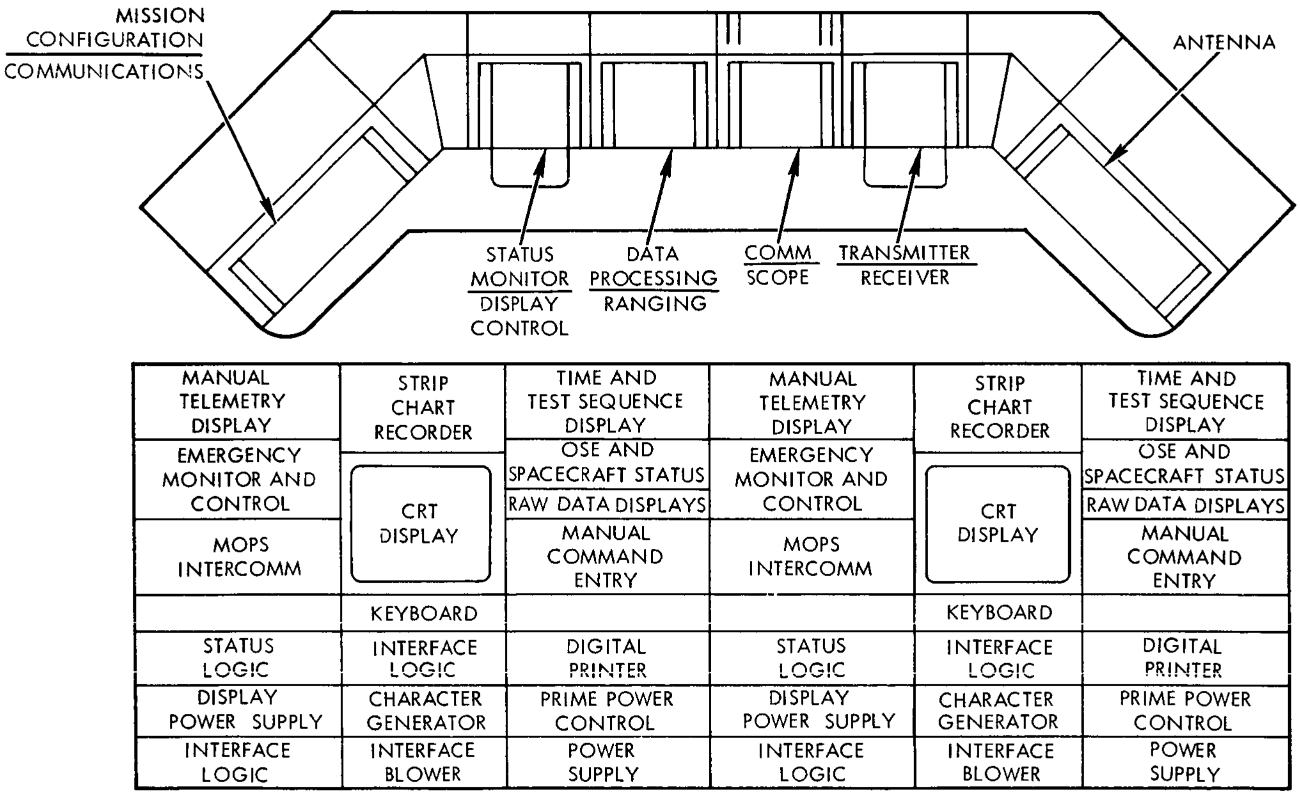


Figure 3. Test Director's Console

test director and his assistant. Although both positions are active, in the sense of having the displays active and the keyboard and switches "on-line," actual use of each depends on the test situation. In addition to allowing convenient shifting of duties and an operating position from which the second man can conveniently function, the second position supplies redundancy.

The individual panels and drawers of the test director's console are mostly duplicates of hardware found in the central data system (Section 3.2) or subsystem EOSE (Section 4) and described there, or are relatively simple items described briefly in the following paragraphs.

a) The CRT display, keyboard, character generator, and interface logic are identical units to those provided in each monitor rack and are described in the central data system description, Section

b) The manual telemetry display is a unit identical to that which is part of each monitor rack (described under CDS, Section 3.2). Telemetry word selection is by panel-mounted digit switches. The display is continuously active, exhibiting discretes and numerics (not engineering units). With the computer on-line, it supplements the CRT display. With the computer off-line, it is the only active telemetry display.

c) The emergency panel contains emergency switches and indicators tied directly to the spacecraft and the power subsystem OSE. They are primarily spacecraft power indications and controls, but may also include temperature, pressure, or other indications and controls.

d) MOPS intercom is identical to that on each monitor rack.

e) The time and test sequence display also appears on each monitor rack. It displays GMT, mission time, and the number of the test sequence currently being executed. The former data is supplied by the facility central time source, the latter by the computer.

f) EOSE and spacecraft status and status logic units are similar to those in the CDS monitor rack, except that they include EOSE status in addition to spacecraft status. The information source for spacecraft status is the CDS computer, and the status logic unit contains decoding circuitry to identify address location (indicator identity) and memory to retain the indication until it is changed by the computer.

g) The manual command entry unit provides coded switch closures to initiate a command to the spacecraft by the command encoder of either the command EOSE or the MDE, depending on the STC conditions. The coding switches are of the same type used on command encoder front panels and interface with the encoder at the same point in the logic as the local switches, which are disabled during system test.

h) A small multi-channel strip chart recorder is provided, with D-A conversion capability. It can be used to record hardline signals from the spacecraft (e.g., bus voltage or current) or can convert digital telemetry data (from the manual telemetry unit) to analog voltage and record.

i) A high speed digital printer provides real-time hard copy printouts of selected subsystem computer-generated information.

j) Raw data displays include an oscilloscope and meters to display, in raw form, hardline special test data.

3.1.2 Functional Interfaces

STC electrical interfaces with the spacecraft are the sum of those listed for the various subsystem EOSE (although in a normal system test, only a portion of the total direct access connections is in use) plus those for the central data system. In general, during a system test, stimuli will be applied at spacecraft sensors (e.g., light sources exciting the optical sensors). Monitoring will use telemetry via RF link or umbilical hardline, supplemented by selected direct access and umbilical connections. Commands will be sent via umbilical or RF link.

During spacecraft subsystem assembly and test, and fault isolation, additional direct access interfaces will be used. As the spacecraft subsystems are integrated, sequential stimulation and termination capabilities of appropriate subsystem EOSE will be required, and appropriate subsystem interface points used. During fault isolation to the provisioned spare replacement level, it may be necessary to utilize direct access points of a lower level than are normally used during system test.

Interface compatibility of the STC with the spacecraft will be verified in all essential respects during subsystem testing of the proof test model. Other interface verifications which will precede interconnection with a flight model spacecraft are interconnection with the spacecraft simulator (which will simulate the umbilical) and marriage to the engineering model spacecraft.

The mechanical aspects of the STC-spacecraft interface are primarily concerned with stimulation of the guidance and control subsystem. This interface is described in Section 4.3.

3.1.3 STC Signal Paths

The primary functional (not a true electrical representation) STC signal paths shown in the simplified block diagram of Figure 4 are:

a) Subsystem OSE-Spacecraft. All stimulation and monitoring of spacecraft functions is done via the subsystem EOSE. These paths terminate at the spacecraft via a combination of RF (coax or air link), umbilical, and direct access test connectors. The block diagram indicates all EOSE connected to the spacecraft, although in a given subsystem test only a portion of the total direct access connections may be in use. For example, commands will frequently be sent via the MDE as part of the continuing compatibility checks; on other occasions, however, it may be necessary to utilize the command EOSE to vary command parameters for design margin testing.

b) Central Data System-Subsystem OSE. Each subsystem test set receives digital control signals from the CDS and delivers its (digital) measurements in the CDS. In addition, operator displays and keyboard entry are located at each subsystem test set; data entry and requests thus flow from each subsystem set to the CDS and display data in the opposite direction.

c) Test Director's Console-CDS. The test director's console has displays and data entry provisions similar to those at each subsystem set and the two-way flow indicated is as described above.

d) Test Director's Console-All Subsystem OSE. Voice intercom between the test director's console and the subsystem test sets is provided.

e) Test Director's Console-Spacecraft and Power Subsystem OSE. Certain emergency control and monitor functions (such as the power bus monitor and spacecraft power disconnect) are connected to the test director's console in addition to the displays generated by the computer. These functions must be available in spite of any conceivable single failure. Therefore, they will be redundantly provided and pass through an absolute minimum of equipment.

f) Telemetry Subsystem EOSE-Other Subsystem EOSE and Test Director's Console. In the manual (computer-down) mode, limited display of telemetry data is accomplished by distribution of telemetry data and sync directly to the monitor rack at each subsystem EOSE and to the test director's console.

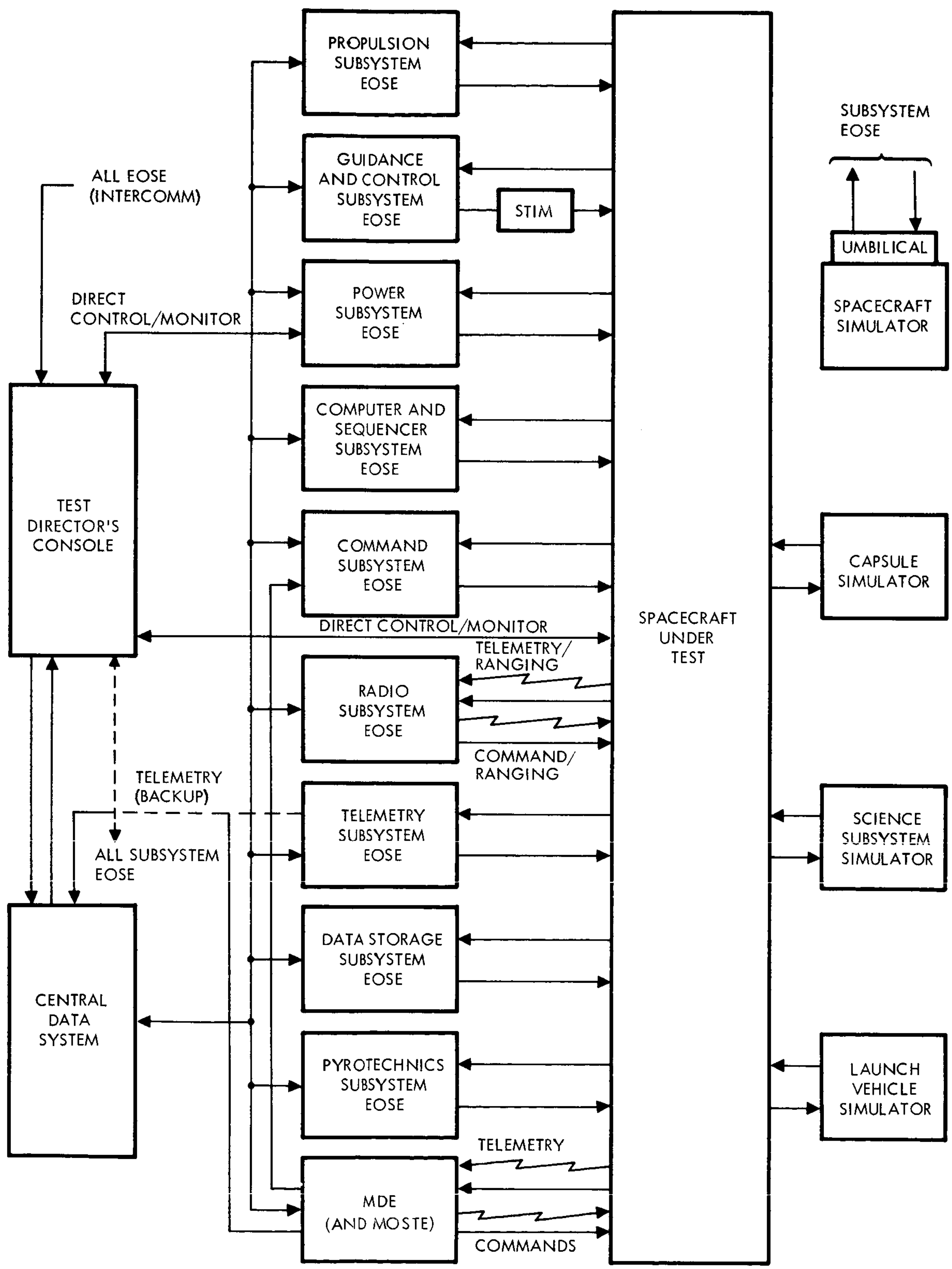


Figure 4. System Test Complex

g) Spacecraft-Science, Capsule, and Launch Vehicle Simulators. The science, capsule, and launch vehicle simulators provide terminations and, to the extent practicable, stimuli to the spacecraft during system test which are exact simulations of those provided by the represented prime system.

h) Spacecraft Simulator-Subsystem EOSE. The spacecraft simulator represents the spacecraft umbilical and, to the extent practical, furnishes all terminations and stimuli normally seen by the EOSE during a system test (for EOSE checkout).

3.1.4 Operation

a. System Test

In the normal (automatic) mode, system test is conducted by the CDS, with manual override by the test director and, at his direction, by the subsystem engineers. The executive program will call up the test routines comprising system test. Each test routine contains the necessary OSE setup commands, spacecraft commands, and instructions for evaluation of spacecraft responses (compare with stored limits, etc.). During system test, a majority of the responses are returned to the CDS via the MDE (or, in backup mode, the telemetry subsystem OSE). The executive program causes the evaluated test results to be sent to the monitor rack, which is a CDS component at each subsystem test set, and to the test director's console.

The display of test results provided is of two types, CRT alphanumerics and engraved legend event indicators. The event indicators are summary or condition type indications (receiver locked, valve open, etc.) while the CRT is a more flexible display which can present detailed and dynamic quantitative information. For example, the CRT can display the measured value of a parameter along with the allowable tolerances, which may be different during different test sequences. This allows the test subsystem engineer to observe the margin existing during any set of test conditions. In addition, particular subsystem test sets and the test director's console include D-A converters and strip chart recorders for use where an analog record of either raw or processed data is more useful.

A digital printer is provided at each operating position, on which hard-copy printouts of relevant real-time information may be produced.

If the test director decides to initiate a test routine (or any other routine) different from the one being executed, he may do so by entering the request on the console CRT by means of the data entry keyboard. The test director or subsystem engineer can also request special displays (e.g., x-y plots) by similar request.

b. Troubleshooting

Fault detection and isolation to the provisioned spare replacement level is provided in the STC. Each subsystem EOSE has special test point monitor capabilities. Automatic isolation to the black-box level is, in most cases, possible by hardline via the spacecraft special test data direct access connectors. The other fault-isolating components required are the CDS routines containing appropriate fault isolation logic. In some cases, it will be necessary to employ portable manual test equipment and the test engineer to isolate a fault.

c. Test Records

One of the important advantages of the automated test is the capability of the computer to prepare and maintain test records. The results of every test performed (subsystem and system) will be recorded on digital magnetic tape, along with the following:

- Identification and status of the spacecraft equipment under test
- Identification of the test
- Status and conditions of STC equipment and test facilities
- Date and time in GMT (to 1 ms)
- STC and spacecraft operating modes
- Alarm messages
- Commands sent to spacecraft

In general, storing these results will be automatic since all this data normally passes through the computer. In exceptional cases, it may be necessary to enter failure date into the file manually by means of the CRT and its keyboard. The CRT keyboard is a convenient method of data entry, since the information to be entered is displayed for check prior to permanent storage in the computer.

d. Trend Presentation and Analysis

The value of the test records, other than as historical documents, rests heavily on the ability to conveniently compare results of identical tests run repeatedly over a period of time. This capability will be provided by the software supplied with the CDS. The quick-response devices for graphically displaying this type of data are the CRT display and the strip charts in the subsystem EOSE and the test director's console. Examples of the types of presentation are:

- Extreme parameter value previously encountered
- Parameter value versus time
- Parameter value as a function of other variables
- Parameter value trends previously found in units which failed

e. Manual Mode

Some STC testing capability is required during computer unavailability. In the present design the backup capability is defined as:

1) The ability to accomplish the EOSE setups normally done by computer command
2) Decommutation of telemetry and presentation of pertinent parameter values (not in engineering units) at each subsystem EOSE and at the test director's console
3) The ability to manually control the spacecraft
4) The ability to record on magnetic tape and strip charts all data from the spacecraft before passing through the computer, and to play this back through the computer later.
5) The ability to directly display special test point data in its raw analog or pulse form at each EOSE.

Item (1) is accomplished by a set of switches at each EOSE. These switches activate, in the manual mode, the normally computer-driven relays which patch together, and set to correct operating mode, the components of the test set to perform a particular test.

Item (2) is mechanized by providing the telemetry subsystem OSE with additional components to accomplish the decommutation process.

Each EOSE monitor rack and the test director's console is equipped with telemetry word selection logic, a limited number of illuminated legends, and numeric memory and display.

Item (3) is provided by EOSE manual switch control of spacecraft control hardlines and manual operability of the command encoder in either the MDE or the command subsystem EOSE.

Item (4) is provided by tape recorders and strip charts connected in parallel to all data sources, with the capability for tape playback through the computer.

Item (5) is provided by direct hardline connections between spacecraft test points and the EOSE.

f. Self-Check

Each EOSE unit has the capability for automatic self-check without interrupting spacecraft operation. The results are entered into the computer, which drives EOSE status indicators on the test director's console.

3.1.5 Simulators

a. Spacecraft Simulator

A spacecraft simulator is provided to present, at a replica spacecraft umbilical, the same interface which will appear at the actual umbilical during system test. The purpose of this simulator is to allow verification of EOSE-spacecraft compatibility and EOSE operation prior to spacecraft marriage and during times when the spacecraft is unavailable.

Umbilical functions will not be defined until Phase IB, but certain types of functions are very likely to be included.

- Telemetry data stream and sync
- Command detector
- Discrete command lines for selected spacecraft functions
- Power input (spacecraft power bus)
- Battery current and voltage monitor signals

- Inverter current and voltage monitor signals

The telemetry signal is generated as part of the self-check hardware of the telemetry subsystem EOSE, and will be brought through an interface circuit in the simulator which is an exact duplicate of the spacecraft circuit. Consideration will be given in Phase IB to including a spacecraft-type PN command detector to evaluate the encoder commands from the EOSE. Termination for the hardline command lines requires only the addition of indicators. The power input terminals will be connected to a load which closely matches spacecraft characteristics. The AC and DC monitor points will be stimulated by appropriate signal generators and power supplies, transforming the impedance, as closely as practical, to the spacecraft values.

b. Launch Vehicle Simulator

The interface between the launch vehicle simulator and spacecraft will be identical to the interface presented by the launch vehicle during system test. At present, this interface is expected to be primarily discrete signals and loads concerned with separation pyrotechnics. This simulator will be designed during Phase IB, when the launch vehicle interface is defined more closely.

c. Other Simulators

During system test, science subsystem and capsule simulators provide the interfaces seen by the spacecraft at the respective connectors when it is in actual operation. The simulation provided is adequate to allow exercising the spacecraft in all operating modes during system test in the absence of the science subsystem or test capsule. The mechanization of the simulator will be defined during Phase IB when interface definition has been accomplished.

3.1.6 Changes from Task A

The differences between the STC described in the foregoing paragraphs and that in the Task A report are primarily in three areas:

- The Task B STC is far more automatic than in Task A. The Task A computer was used for real-time processing of test data, display updating, and off-line reduction. As noted, the computer now directly controls the EOSE, in addition to those tasks.

- The Task B STC also includes a set of subsystem EOSE, while in Task A the STC contained only a system test set (without subsystem test capability).
- The Task B STC contains a set of MDE, while the previous STC did not.

3.2 Central Data System Functional Description

The central data system (CDS) is designed to provide complete subsystem testing capabilities on any test level as a function of available direct access connections. The CDS computer is capable of generating a majority of the required test sequence control signals, as well as reducing, tabulating, recording, and analyzing all pertinent spacecraft and EOSE data. Other computer operational characteristics include go/no-go status evaluation, limit checking, trend analysis, alarm generation, and off-line computational operations. The CDS is programmed to operate in a completely automatic mode in which the computer provides all control and reduction functions; in a semiautomatic mode in which some control and simulation functions are manually stepped; or in a degraded-capability manual mode, in which the computer is not used. The CDS is designed to operate on both the system and subsystem test level configurations and, thereby, is capable of interfacing with one complete set of spacecraft hardware interconnected in any configuration. Computer programs are constructed in a multi-level pyramidal subroutine structure such that individual test sequences or simulated data patterns, can be called up in any testing configuration. Control programs and executive routines are also pyramided such that complete test sequences, partial test sequences, or individual test sequences, can be manually called up, depending on the degree of automation required for a given test configuration.

The CDS is composed of the computer main frame, the peripheral equipment and the data entry and monitor racks as shown in Figure 5. The data entry and monitor racks (DE and MR) are located adjacent to their respective EOSE test sets, while the remainder of the computer units are located in a central computer area.

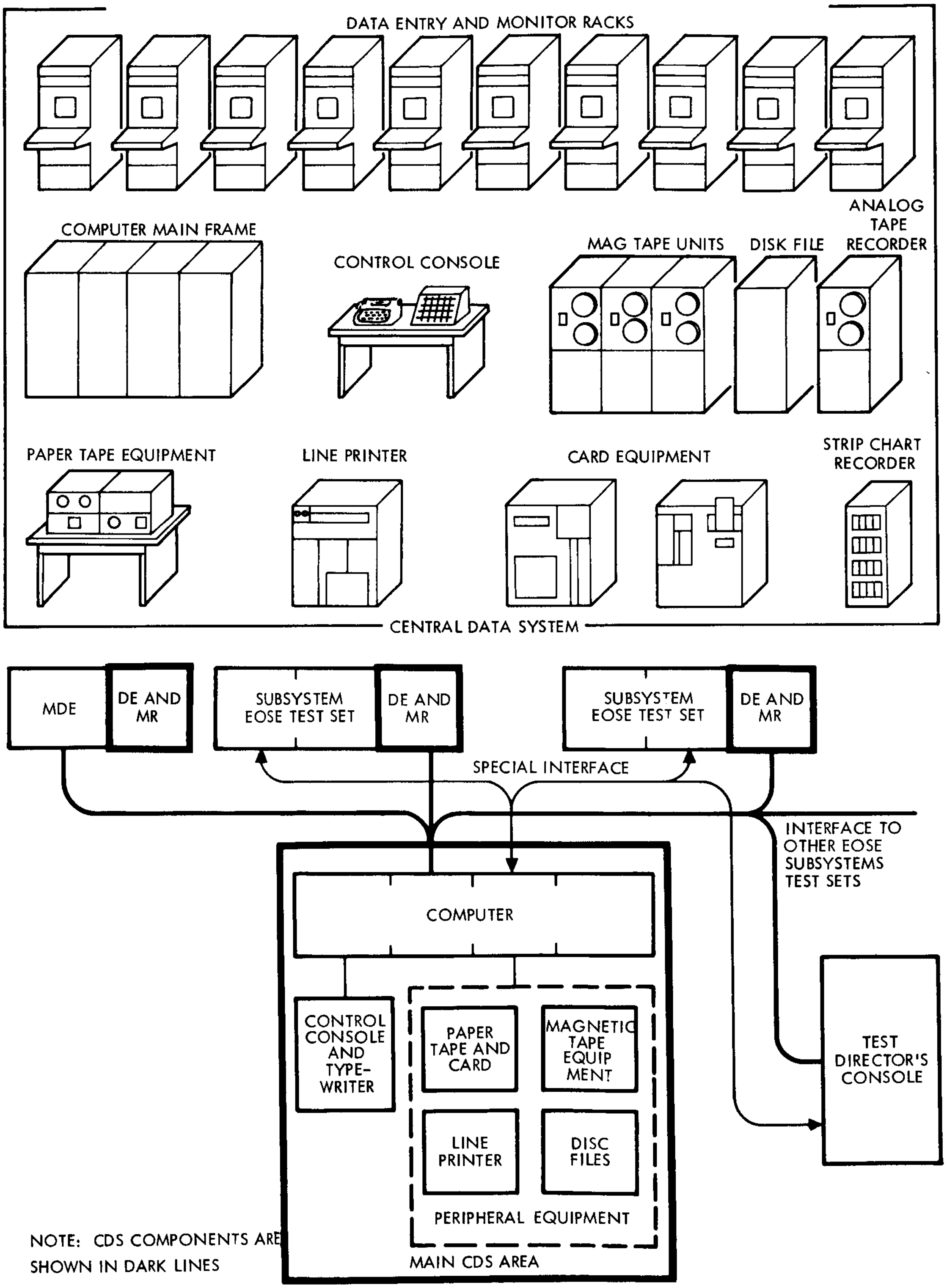


Figure 5. Central Data System EOSE Flow and Rack Layout

3.2.1 Requirements and Design Constraints

To accomplish the EOSE system tasks, the CDS must be capable of performing the following functional requirements.

Centralized Test Logs. The CDS provides centralized means of recording (and time tagging) all spacecraft and EOSE information, together with test sequence data. Recorded information functions are as follows:

- List and identify spacecraft equipment under test
- Identify and time tag tests being performed
- Continually tabulate all spacecraft status data
- Continually tabulate all EOSE status data
- Continually tabulate all self-test data
- Record date and time (GMT)
- Record status and event changes defining spacecraft and EOSE conditions or operating modes.
- Record alarm functions
- Provide complete records of telemetry, command, and direct access data
- Record simulation input data and actuation times
- Record spacecraft-EOSE data evaluated by the CDS
- Provide all necessary off-line recording functions

Real-time Data Records. The CDS provides real-time multiple hard-copy printouts restricted only to the necessary information for real-time evaluation of the spacecraft or subsystem under test. Data suppression and alarm monitoring techniques will be used.

Man-Machine Interface. The CDS provides a man-machine interface with the test director and all subsystem test engineers to allow for call-up of special displays, printouts, simulation programs, and test routines.

Alarm Detection. The CDS provides automatic detection and alarm indications for any calculated abnormal spacecraft or test facility behavior.

Self-Check. The CDS provides self-check capabilities for both the CDS and the subsystem EOSE equipments.

Telemetry Decommutation. The CDS provides decommutation capabilities for the spacecraft, and EOSE simulated telemetry data.

Data Displays. The CDS incorporates the processing capabilities required for updating data displays located in the data entry and monitor racks and the test director's console.

Test Procedure. The CDS is capable of generating multi-level test procedures, sequences, and subroutines for subsystem or system level testing in both the automatic and semiautomatic operational modes.

Master Timing. The CDS provides and distributes master timing information to all EOSE subsystem test sets.

Off Line Data Reduction. The CDS provides the capability of further processing and analysis of previously recorded data.

Program Safeguards. The CDS provides sufficient programming safeguards in the automatic and semiautomatic modes of testing to prevent the occurrence of damage attributed to improper sequencing.

General Purpose Computer. No special purpose or newly-developed hardware or hardware mechanization techniques will be employed within the CDS. Standard modules and techniques will be used in the construction of computer interface hardware.

Universal Interfaces. The CDS/DE and MR interface is designed in a universal manner such that any DE and MR can be used with any subsystem test set by merely changing address matrix cards and engraved indicator terminology. This technique provides great flexibility in operational performance and future equipment changes due to new mission and spacecraft hardware requirements.

Expansion Provisions. Expansion provisions are implemented by modular design techniques and a conservative approach to data transfer hardware design requirements. The CDS data entry and monitor rack interface will be capable of transferring a much larger amount of data than is anticipated for the 1971 mission test operations.

Interface Reliability. All interface units are identically designed, constructed, and tested in order to provide a greater time allocation for reliability calculations, tests, and improvements. Other advantages in this multi-production philosophy consist of better training, documentation, mean time to repair, and spares provisioning, together with the fact that an entire unit can be replaced to limit downtime during critical testing phases.

Human Engineering Factors. The interface unit multi-production philosophy previously discussed also contributes greatly to the human factors requirements since one display and entry device is designed for use by all subsystem test engineers. The man-machine interfaces have been designed to permit maximum flexibility with minimum confusion. Data selection and alarm generation techniques are incorporated to minimize evaluation and decision making time requirements. The CDS also provides a veritable wealth of information to each subsystem test engineer in a form that can be immediately interpreted.

Data Transfer Rates. Various data transfer rates have been selected for CDS interface functions. These rates have been selected to minimize computer main frame display time requirements and yet optimize control and machine data transfer parameters. The following list illustrates some transfer time selections:

- Telemetry data inputs to computer: as required by telemetry data rates on a high priority interrupt basis.
- Discrete computer output control and data signals: a scan process is used with an update and tabulation resolution of 2 ms.
- Discrete spacecraft and EOSE computer status inputs: a scan process is used with an update and tabulation resolution of 2 ms.
- Computer display outputs: status is updated at a slower low priority scan cycle with an update period of greater than 0.1 second.

- CRT display and entry data: by low priority interrupt lines only. Updating only occurs when the respective information has changed its state.
- Line printer output operations: an extremely low priority interrupt system is used utilizing the storage capabilities of disk files.

Programming Techniques. Program techniques shall be provided which result in a Christmas tree test structure. Specific test sequence subroutines become universal in this instance and are used for functional test performance on any test level and in any test sequence. The same support software (symbolic assemblies, utility programs, debugging aids) will be used for program development for all CDS computers. A set of programming standards will be developed before program design is initiated. These standards shall include specifications to be used in modularizing, by subroutines, the total program, program module interfaces, buffer or list instructions, flow diagram conventions, program automation, and documentation. Programming techniques will be of a general purpose nature and will not require the use of special or newly developed techniques.

Multiple Program Usage. Multiple program usage, being defined as providing the use of a portion of, or a total program, during various testing phases will be a basic consideration in program subroutine generation. The majority of the test sequence and reduction operations will be performed on the subsystem functional test level, such that the same specific test sequence can be called up in any test level configuration.

Language. No consideration will be given to the development of any new test-oriented language. Standard forms will be used for all data recording processes regardless of the test level configuration.

Expansion Provisions. Program structure is constructed such that branching operations and insertions, as well as routine group reorganization operations can easily and rapidly be accommodated when spacecraft hardware or mission changes occur. Addressing and indexing structures shall not constrain program development or modification by requiring special program techniques or considerations.

3.2.2 Design Description

From the CDS block diagram (Figure 5) it can be seen that the major computer input-output interfaces include the general categories of peripheral equipment, data link, data entry and monitor racks, MDE, test director's console, and subsystem EOSE test sets. Most CDS equipment is located in the main CDS area. The DE and MR racks are located adjacent to their respective EOSE subsystem test sets. The special interface involves direct computer/EOSE signal transfer functions (i.e., telemetry) and, in one case, a manual telemetry back-up interface which uses the telemetry subsystem test set decommutator outputs to supply visual data indications to all other subsystem EOSE test sets. The test director's console includes various manual control interfaces, both with the computer and with the subsystem EOSE test sets for emergency override purposes.

Data Recording. Centralized data recording is accomplished both in real-time and off-line within the CDS by the peripheral equipment line printer and magnetic tape recording units. The result of this process is a complete set of time tagged records of all pertinent spacecraft, EOSE, and test sequence parameters in both subsystem and system level testing configurations. Initial parameter records are stored on rapid access disk files for later transfer to the line printer and magnetic tape recorder units. This technique is used to assure that all data are recorded regardless of event occurrence densities and data rates. The line printer is continually updated during normal computer duty cycle periods, and momentarily stopped, if main-frame priorities are such that line printer time is not available. This technique establishes a buffer for record operations, thereby giving the recording process the lowest priority. Computer main-frame memory space is conserved since rapid access cycle timing is not required. Magnetic tape recordings are also used for storage to enable efficient and rapid off-line operations to be performed on previously recorded test data.

Test Sequence Control. The CDS computer provides test sequence control functions in both the automatic and semiautomatic operational mode for both system and subsystem testing levels. Control is exercised by the generation of discrete and parallel data information which (through the data entry and monitor racks) operates on subsystem EOSE control actuators and results in the application of, or a reduction of, the required functions. The control programs are capable of operating on the majority of EOSE subsystem control functions (some operations may require manual patching operations or data interpretations). Program subroutine call-ups can be requested such that an entire system or subsystem test procedure can be automatically performed in a continuous process, or such that a single test sequence is intiated.

Data Transfer Operations. Data transfer operations in and out of the CDS computer are accomplished by standard general purpose computer techniques. Standard parallel input and output operations provide the main data flow, while interrupt operations are included for the transfer of time critical information. Since data transfer operations employ standard techniques, specialized computer hardware and software are not required.

EOSE Interface. The EOSE-CDS interface is accomplished remotely to the CDS by the data entry and monitor racks. These racks, all identical in function, provide the major data transfer and man-machine interface functions to each individual subsystem EOSE test set. This rack also includes the major display units and the subsystem test engineer's keyboard for computer input request and data transfer operations.

Simulation Generation. The CDS provides simulation data information to the EOSE subsystem test sets in both the system and subsystem testing phases. The data entry and monitor rack decodes the information and generates EOSE control and data signals which result in the storage of the data information with the test set hardware. The test set hardware provides the conversion and signal conditioning functions under the direction of computer (or manual) control information. This conversion process may consist of a parallel-to-serial conversion operation, clocked asynchronously with respect to the computer clock system. The update data transfer operations are handled on an interrupt basis in this instance to be sure that the data transfer has occurred before shifting operations are continued. Other simulation data is merely stored in a discrete form and applied to the spacecraft as such, after voltage level conditioning has been accomplished. This simulation technique has the basic inherent advantage of being able to call up any complexity of computer program subroutine depending on the test level being performed. Bit error rate measurements, for example, can be efficiently handled by the CDS using this simulation generation philosophy.

Data Displays. The data entry and monitor rack provides the visual display capabilities required for the human evaluation of testing operations and spacecraft-EOSE status. The functionally oriented displays (i.e., status display, and time and test sequence display) are by nature low priority data rate devices which are only required to change status at human visual response rates. The data transfer techniques for these displays, therefore, consist of a low speed update scan process within the CDS computer and as such, requires little main-frame time and no interrupt capabilities. The CRT display and keyboard entry unit operates directly from the computer interface lines and has interrupt capabilities. A buffer memory input is included in these units to eliminate main-frame computer time, which would otherwise be used for refresh purposes. The CRT display provides each subsystem engineer with the capability of selecting or calling up any data contained within the main computer memory, while the functionally oriented displays provide only the relevant information pertaining to the unique spacecraft subsystem.

Alarm Generation. The CDS provides alarm generation to both the data entry and monitor racks and to the test director's console. Alarm generation is a function of the computer evaluation of spacecraft and EOSE data. Alarm routines are programmed in a subsystem-oriented manner, such that the same sequences can be used whether the testing is accomplished on the system or subsystem levels. Alarm indications result from the detection of an abnormal spacecraft condition, which may lead to equipment damage or degradation if the test continues to be performed. EOSE alarm indications may also result when an EOSE failure has occurred, such that proper spacecraft performance evaluation cannot be maintained. CDS alarm indicators are also provided on the test director's console and data entry and monitor racks to indicate a computer failure condition.

Trend Analysis. The basic CDS computer functions of generation, recording, testing, and limit checking are complemented with a trend analysis program technique which evaluates past spacecraft performance with present spacecraft data to determine trends not readily apparent. A spacecraft function verification procedure, therefore, not only establishes that the function is within the prescribed limits, but also that the function trend over a period of time will not cause these limits to be exceeded. Trend analysis is completely a function of the software program.

Sequencing and Requests. The CDS has available in its storage facilities (core, disk files, mag tape, and IBM card) all test sequence and data display subroutines required for spacecraft testing on all levels. It is, therefore, possible to provide a detailed subsystem test sequence call up, during a system testing level operation. Test sequence routines can be called up by any individual subsystem engineer (providing the action has been enabled by the test director and computer), which are not included within the automatically operated test procedure or sequence. These routines may consist of special data requests required for the evaluation of some unusual spacecraft performance status, or may consist of a troubleshooting sequence in the event that a fail condition has been detected. This program philosophy provides the individual subsystem engineers with flexible testing capabilities without interruption of the testing phase.

Control Modes. The CDS software program is constructed in a Christmas tree fashion, thereby allowing the controlling routine to be of a system nature, a subsystem nature, or a troubleshooting nature. Initial control routine selection is based on the test level being performed and the level of automation required. From this point, routine branches can be requested and executed where required for further spacecraft evaluation purposes. The manual control option releases the computer from all control functions, but not from its recording, reduction, and simulation generation functions. A completely manual mode can also be established, which does not require the use of the CDS computer at all. Control mode status is evaluated by the computer executive routine and incorporates manual override features at the test director console.

Off-Line Operations. The CDS provides all of the functional capabilities for off-line reduction operations. In addition to these reduction tasks, the CDS may also be programmed to simulate an entire vehicle testing phase (in degraded form) in an off-line manner from the records generated by a previous test. These techniques can prove useful for both training and spacecraft re-evaluation purposes.

Expansion Provisions. CDS expansion provisions are introduced by the modularized nature of the hardware equipment, the flexibility of the data transfer interface, and the general purpose nature of the software programs. It is obvious at this point that a great amount of flexibility must exist within the CDS to accommodate mission and equipment changes for the various future launch requirements. The data entry and monitor racks, together with the test director's console hardware, have been designed to facilitate reprogramming of functional characteristics without modification of drawer or rack wiring and hardware. Each element has expansion provisions built in, and depends only upon circuit card matrices and engraved legend plates to establish its unique relationship to a given subsystem. This design philosophy, together with the expansion capabilities of the basic general purpose computer, results in a product which is extremely flexible and which possesses almost unlimited expansion capabilities.

Self-Testing Operations. The CDS provides two types of self-test operations: computer self-test, and EOSE self-test. The computer self-test operation consists of the standard test program routines which assure that the basic operational functions of the computer are in a normal status. The EOSE self-test operation consists of discrete signal monitoring and test sequence program generation. The discrete self-test signals are generated within each EOSE subsystem test set and indicate the operational status of various subsystem elements (i.e., oscillators on, power supplied, etc.). The test sequence program generation involves a series of subroutines designed to simulate and monitor various subsystem test set operational units during time periods when these units are not being used in a spacecraft in-line test operation. The computer evaluates the return data with respect to the simulation data generated and produces a discrete status signal indicating operational performance. EOSE test fail conditions are indicated as status discretes and also may be used to actuate alarm generation.

Telemetry Decommutation. Normally, telemetry decommutation (frame sync, word sync, word identification, and data correlation) is performed within the CDS using the telemetry subsystem EOSE test set inputs or the MDE inputs. These inputs have synchronized and converted to parallel bit data formats for computer interface transfer purposes. High priority interrupts are associated with each of these input data groups. A specialized I/O computer buffer is assigned to the telemetry decommutation data transfer tasks with interfaces directly routed to the demodulation and sync equipment. Telemetry data is recorded and continually stored in rapid access memory slot locations. Data outputs from these memory slots to visual displays on the subsystem test sets are generated by a scan process such that any test set can select any telemetry word for display by providing the appropriate EOM address decoding matrix card. Additional display capabilities are presented in the CRT display system.

A manual decommutation data distribution system is also provided to supply pertinent display information during computer fail conditions. This data is continuously displayed on a telemetry display panel located within each data entry and monitor rack and in no way interfaces with the computer. The panel not only serves as a manual back-up device, but also provides unconverted real time telemetry data to each subsystem test set for comparison and evaluation purposes.

Master Timing Distribution. The CDS provides for the distribution of all master time reference signals. The spacecraft master clock, maneuver clock, and alternate timer clock signals are derived from the computer (synchronized by spacecraft clock input data), and distributed to all subsystems requiring this information.

a. Data Entry and Monitor Rack

The data entry and monitor rack includes the following major units (see Figure 6).

- Interface unit (computer)
- Status display unit
- Time and test sequence display unit

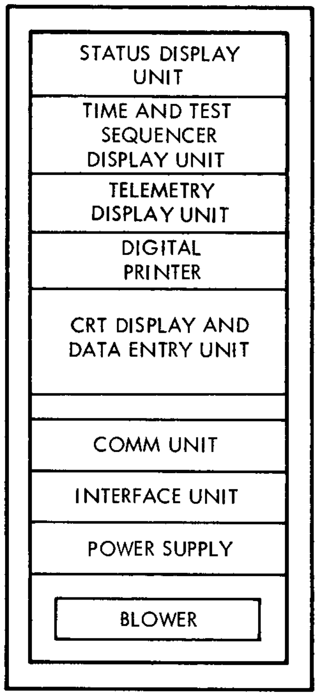


Figure 6. Layout of Data Entry and Monitor Rack

- Digital printer
- Telemetry display unit
- CRT display and data entry unit
- Communications unit
- Power supplies, rack and miscellaneous hardware

The rack is of standard Voyager configuration, dolly-mounted for ease of mobility. A writing surface or shelf is provided for operator convenience The data entry unit keyboard is placed in a slanted position and extends horizontally onto the writing surface. A communications device is located convenient to the operator. Status display indicators are engraved for individual subsystem signal nomenclature. A power control and blower drawer is also included within the system rack.

The data entry and monitor rack, used with each subsystem EOSE test set provides the following functions on both system and subsystem test levels.

- Deliver computer control and spacecraft input data to subsystem EOSE
- Receive spacecraft test results data from the subystem EOSE
- Provide digital data readouts of computer stored spacecraft information in the form of visual data displays, hard copy digital printout, and a number of visual status indicators
- Provide a computer-test engineer communication interface, allowing the subsystem test engineer to initiate computer requests for such things as data display selection, spacecraft input simulation signal requests, and test routine requests
- Provide a means for transferring data information to the computer which cannot be automatically obtained
- Provide a means of transferring EOSE self-test discretes to the central data system
- Provide a communication facility for the subsystem EOSE test set
- Provide all necessary buffering and logic functions for the main computer interface
- Provide a noncomputer-generated telemetry or raw data display of all pertinent spacecraft subsystem data

The data entry and monitor rack (DE and MR) provides the basic computer interface for each EOSE subsystem test set. The rack itself contains display devices, entry devices, and interface devices (see Figure 7). The interface unit provides data buffering in both directions and converts logic levels for OSE compatibility. This unit receives all major computer interface input and output lines. The interface unit provides discrete output control lines for the required addresses, which are used both by the DE and MR and the EOSE test set for data transfer purposes.

The status display unit decodes and displays selected spacecraft and EOSE discrete data in a functionally oriented manner. Storage and indicator amplification circuitry are included within the status display unit to provide the proper human display interface. Digital printer logic is also housed within the status display unit.

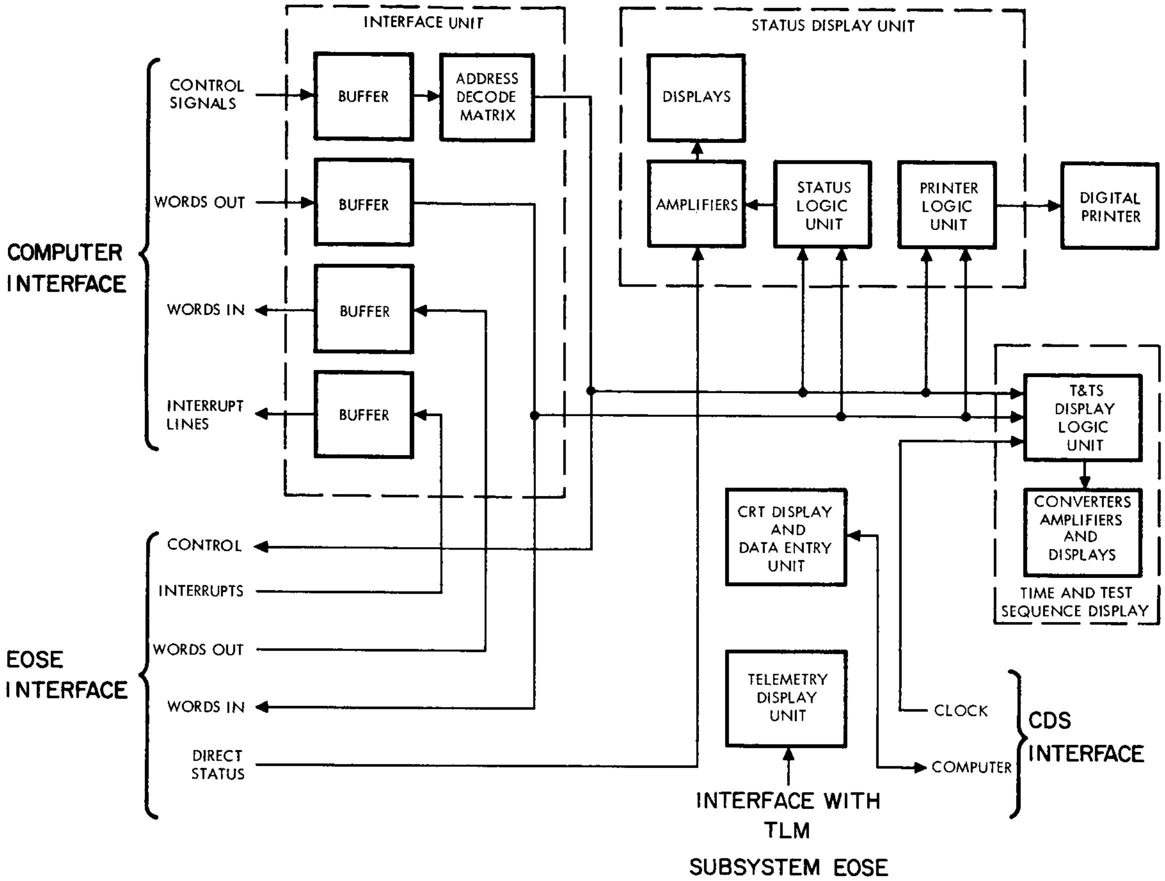


Figure 7. Data Entry and Monitor Rack Block Diagram

The time and test sequence display unit provides decimal displays of main timing information (GMT and mission time) and test sequence data. This unit decodes the input data from the data lines, converts this information to a decimal form, and provides the required display and storage operations. GMT time displays operate independently from the CDS computer and DE and MR interface unit.

A digital printer is included for real-time hard copy printouts of selected subsystem computer-generated information. A communications unit is provided for test coordination purposes. A telemetry display unit is used, which operates independently with respect to the CDS and provides operator displays of selected subsystem spacecraft telemetry data information. This unit directly interfaces with the decommutation equipment included within the telemetry subsystem EOSE test set. The CRT display and data entry keyboard unit provides the main man-machine interface for computer request and data display operations. This unit is directly interfaced with the CDS computer and operates on an interrupt basis.

The DE and MR-EOSE interface allows the computer to communicate with the EOSE such that the computer can direct and partially control the EOSE functions. The EOSE operates typically by applying specific electrical signals or by transferring particular sequences of digital data to the spacecraft subsystems. This communication from the computer to the EOSE is provided through the DE and MR. The DE and MR interprets portions of the computer output information and converts this data into discrete control line signals to the EOSE. Additionally, the DE and MR will transfer a complete word in parallel from the computer to the EOSE. The computer word is used and interpreted by the EOSE to apply specific electrical signals or digital data sequences to the spacecraft subsystems.

This DE and MR-EOSE interface may also allow the EOSE to transfer binary-coded data into the computer. In some cases, the EOSE may be inserting analog voltages (converted to binary coded data in the EOSE) measured at the spacecraft subsystem interface, or possibly the measured values of stimulating signals applied to the spacecraft subsystems by the EOSE. In any case, the data to be transferred into the computer must be presented in some kind of binary-coded form. Multiple word parallel transfers are permissible, one after another at the computer parallel-input transfer rate. All input data transfers from the EOSE are controlled by and passed through the DE and MR to the computer.

Interface Unit. The interface unit provides the main buffering and decoding interface between the computer and the EOSE data devices. The unit contains buffering circuitry for the computer parallel input and output lines as well as the computer address output lines. An address decoding matrix is included to provide discrete control output lines to the DE and MR and EOSE data transfer logic units. Unique program cards are placed within this matrix to enable data transfer operations (input or output) to be directed to the required logic units within the selected subsystem. These cards are identical to other subsystem cards only when identical data information (status data) is permanently required by more than one subsystem.

Status Display Unit. The major function of the status display unit is to update spacecraft and EOSE discrete display indicators to the current information contained within the CDS memory. This information can be derived from reduced telemetry data, spacecraft hardline data, EOSE status hardline data, computer program data, or computer status data. All computer information is available to all subsystem status displays with the selection criteria based only on the interface unit matrix card configuration and the computer software program. The status logic unit provides the update enable and storage logic required to drive the various display amplifiers. Provisions are also made within the status display unit for actuating indicators by hardline subsystem test set inputs. An interface troubleshooting feature is provided if the subsystem hardline illuminates one half of the indicator and the return computer data illuminates the other. A half lit indicator would then show an interface or computer failure. The status display unit also contains the printer logic units which provide enable and storage operations for the digital printer. All status display units are identical with the exception of the unique engraved indicator plates.

Digital Printer. A high speed digital printer is included to provide quick look digital data outputs from any selected computer storage data section. Data selection is again accomplished by the data entry unit keyboard. The printer will be a multiple column alphanumeric device capable of providing printouts of computer program sequences, telemetry data frame sections, hardline spacecraft-computer information, or continuous data word printouts for drift trend analysis. The digital printer receives its actuation and data signals from the status display unit.

Telemetry Display Unit. The telemetry display unit is designed to provide pertinent visual telemetry data indications at each DE and MR in instances where the computer telemetry data transfer may be incorrect or not available. The unit receives word identification and data information in a real-time multiplexed parallel data format from the telemetry subsystem EOSE test set decommutation equipment. The unit logic decodes the identification words and strobes the associated data word or discrete into the appropriate storage register for visual display purposes. This system is not intended to contain complete testing capabilities, but rather to provide degraded subsystem information when computer sources are not available. No printed records result from this unit display.

Time and Test Sequence Display Unit. The time and test sequence display unit provides decimal word displays of GMT time, mission time, and test sequence numerical designations. The unit consists of a logic unit, which functions in an identical manner to the status logic unit, and a converter amplifier and display section. The converter amplifier and display section contains binary to decimal or octal converters along with the associated indicators and drivers. The GMT time display interfaces directly with the independent CDS GMT time unit, not with the computer data transfer lines.

CRT Display and Data Entry Unit. The CRT display and data entry (keyboard) unit consists of commercially available off-the-shelf devices. This device interfaces directly with the computer on an interrupt basis. The CRT display provides data information in alphanumeric form and from any source interfacing with the CDS computer. Display selection is accomplished by the data entry keyboard which is included as a part of the standard unit.

Communications Unit. The communications unit is provided to allow verbal communications to occur between any subsystem test engineer and the test director or other subsystem engineering personnel. This unit is of standard design and construction.

b. General Purpose Computer

The baseline computer used in the CDS is a general-purpose digital computer. The particular computer will be selected in Phase IB and may possess overall capability in excess of that stated here. There exists several machines which possess the desired capability. A selection made at this time, would, due to lack of definition of firm technical requirements, run the risk of not being the best selection, either technically or economically. The computer possesses the following minimal overall capability:

- 16 bit word plus parity bit
- Binary arithmetic

- Instructions with:

  Indexing (without timing penalty)

  Multi-level indirect addressing

  Programmed operators or their equivalent

- Core memory of approximately 30,000 words, all addressable, with:

  ≤1 microsecond access time

  ≤2 microseconds cycle time

- Memory overlap between central processor and input-output with two or more memory banks

- Programmed operators, permitting many special, user-specified instruction codes that can vary from program to program

- Typical execution times (including memory access and indexing) of:

ADD	4 microseconds
MULTIPLY	20 microseconds
DIVIDE	50 microseconds

- Program interchangeability with other computers in the manufacturer's series

- At least 64 channels of priority interrupt

- Memory nonvolatile with power failure; power fail-safe feature permiting saving contents of programmable registers

- Real-time programmable clock with precision of at least 10 milliseconds.

- Two to four communication channels, time-multiplexed with computer operation, providing input-output rates of up to one word per two memory cycle times

- A direct memory access system that allows input-output transfer to occur simultaneously with computer memory access, providing input-output rates of up to one word per memory cycle time

- A capability for one to four direct access communication channels which incorporate the direct memory access system; optional direct memory access connection which may incorporate externally controlled and sequenced equipment into the computer system, thus permitting performance of input-output buffering and control operations by external devices rather than by computer control

- All memory capable of either hardware or program control to assure that a selected portion of memory is protected against possibility of inadvertent change

- Input-output equipment:

  Automatic input/output typewriter

  Control console

  Paper tape reader, paper tape punch

  Magnetic tape units (IBM compatible; binary and BCD)

  Punched card input and output equipment

  Line printer

  Rapid access magnetic disc files

- FORTRAN II and symbolic assembler as part of complete software package

3.2.3 Changes from Task A

The basic changes in design approach since the Task A study report consist of subsystem, other than unit and system testing philosophies, and a much higher degree of automation. Subsystem simulation devices previously contained within the various test sets in general have been replaced by computer-generated simulation programs. This process not only simplifies the EOSE equipment design, but also increases the over-all EOSE testing capabilities. Data selection and display capabilities have been largely increased since the computer has access to a greater amount of correlated spacecraft information.

In the previous central data system, the computer basically functioned as a tabulation and recording device, while in the new configuration, the computer is used as an over-all control and reduction device containing limit checking, status verification, and trend analysis characteristics. The computer now provides test sequences and control functions in several operational modes and in several test level configurations. Recording, tabulating, and time tagging functions have also been operationally increased due to the enlarged quantity of data flow provided. Fault isolation and emergency alarm generation capabilities have also been added.

3.3 Mission Dependent Equipment

3.3.1 General Description

The Voyager mission dependent equipment (MDE) consists of specialized rack-mounted equipment to complement standard DSIF station equipment to enable DSN communication with the Voyager planetary vehicles. Figure 8 is a rack diagram of the MDE/MOS test equipment.

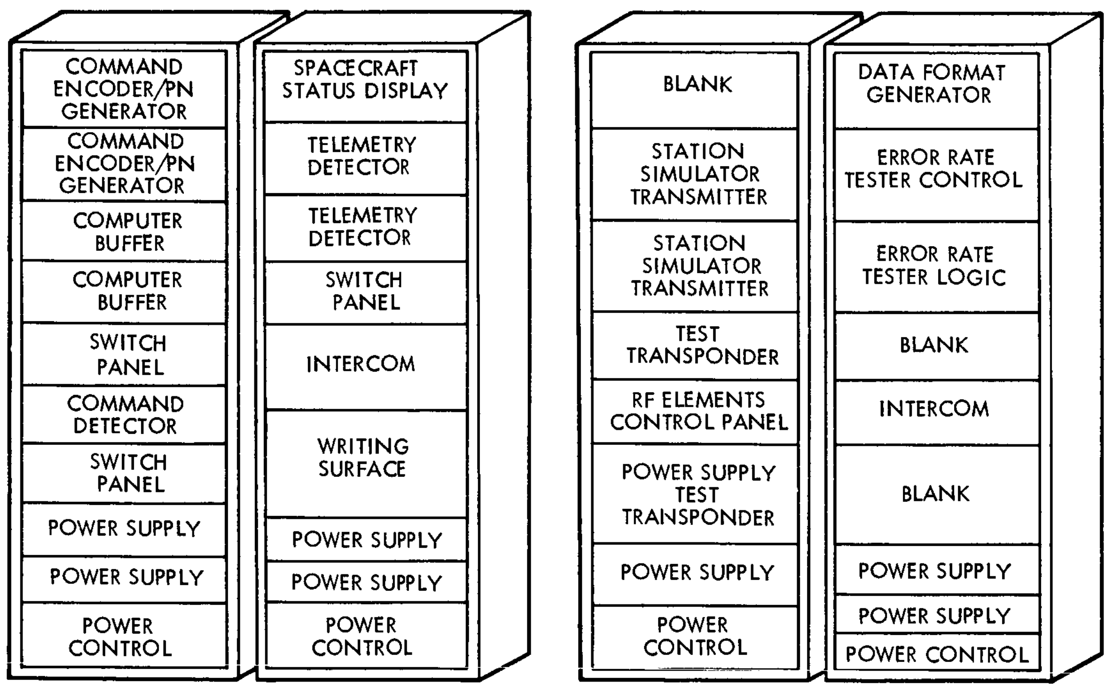


Figure 8. Mission Dependent Equipment/Mission Operations System Rack Layout

The prime in-line functions of the MDE essential to the DSIF link with the Voyager spacecraft are command generation, telemetry detection, and computer buffering. Secondary in-line functions, not essential to the DSIF spacecraft link but desirable for monitoring, are command verification, spacecraft subsystem status display, and data recording. Tertiary functions are those having to do with compatibility and station readiness testing and are not performed in-line. The Voyager MDE is unique, since the command words and telemetry readout for Voyager differ from those of other programs. Operating with the DSIF, commands are entered into the system manually. Programming permits the station computer to decommutate the Voyager telemetry data, to output spacecraft status information to the MDE, to make MDE command checks, and to accept station time signals. Telemetry data, command data, and status data can be typed out on the station computer typewriter or line printer.

The primary and secondary functions are equally necessary for system test of the Voyager spacecraft in a system test complex at the factory or at Cape Kennedy. The tertiary functions are not applicable in system test. In system test, the MDE can automatically generate spacecraft commands incorporated in the test programs of the CDS computer. Typically those commands are generated in a test routine and the spacecraft response is read out on the CDS line printer or CRT displays, while a computer evaluation of spacecraft response determines whether to continue the program tests or to halt the sequence. The objective is to accelerate the complete spacecraft system test. In the automatic mode of operation, manual command entry is available as a backup; the test director can initiate computer-generated or manual commands from the test director's console.

3.3.2 Requirements and Constraints

a. Physical Constraints

The MDE is housed in standard JPL DSIF equipment racks. The test transponder units can also be housed in portable carrying cases. The MDE racks accommodate the trough cabling technique of DSIF, and are designed to accept cooling air at the bottom of each rack. Normal cooling air temperature is to be $55 \pm 10^{\circ}F$. Each drawer weighs no more than 100 pounds and a complete rack of MDE equipment weighs no more than 1000 pounds.

Limited availability of the DSIF for setup and calibration of the MDE imposes a requirement for MDE self-check before it is patched into the DSIF.

b. Electrical Constraints

The DSIF power source is used. Power dissipation limit per rack is 2000 watts maximum.

EMI is controlled so that the MDE does not interfere with other DSIF equipment, nor is it susceptible to EMI from DSIF equipment.

Computer buffering design must assure compatibility with the DSIF telemetry and command data processor.

c. System Requirements

In system test and during a Voyager mission, simultaneous MDE operation with two spacecraft is required. During mission operations, MDE provides visual spacecraft status displays.

MDE for Voyager generates spacecraft commands at 1 bit/sec. These 1-bit/sec PCM NRZ command signals biphase modulate a 511-bit pseudonoise code forming the composite command signal used to phase modulate the DSIF transmitter. Commands manually entered into the MDE for transmission to the spacecraft are computer-evaluated for permissibility. Alternate capability for computer commands is incorporated into MDE for its use as an integral part of the STC.

Operating in the DSIF, the MDE detects telemetry data from the Voyager spacecraft through the DSIF station receiver, transmits it to the station computer for decommutation of data, and provides a command detection function (signals from the station monitor receiver) to enable computer evaluation of the command as it is being transmitted. Telemetry rates up to 15,000 bit/sec are accommodated.

3.3.3 Functional Interfaces

The following are functionally in-line interfaces between Voyager mission dependent equipment and the DSIF (see Figures 9 and 10):

Interface	Nature of Signal
MDE-DSIF transmitter	Composite command signal
MDE-DSIF receiver	Composite telemetry signal
MDE-DSIF monitor receiver	Composite telemetry signal
MDE-DSIF computer	Parallel command signals
	Parallel telemetry signals
	Permissibility request signal
	Permissibility signals

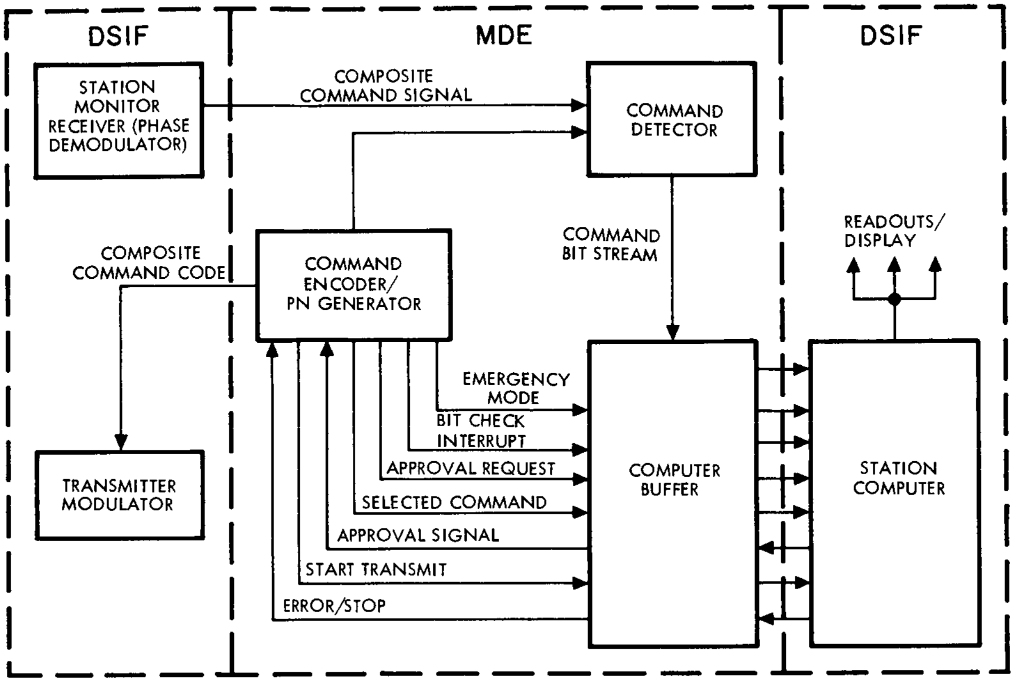


Figure 9. Block Diagram of MDE/DSIF Command Function

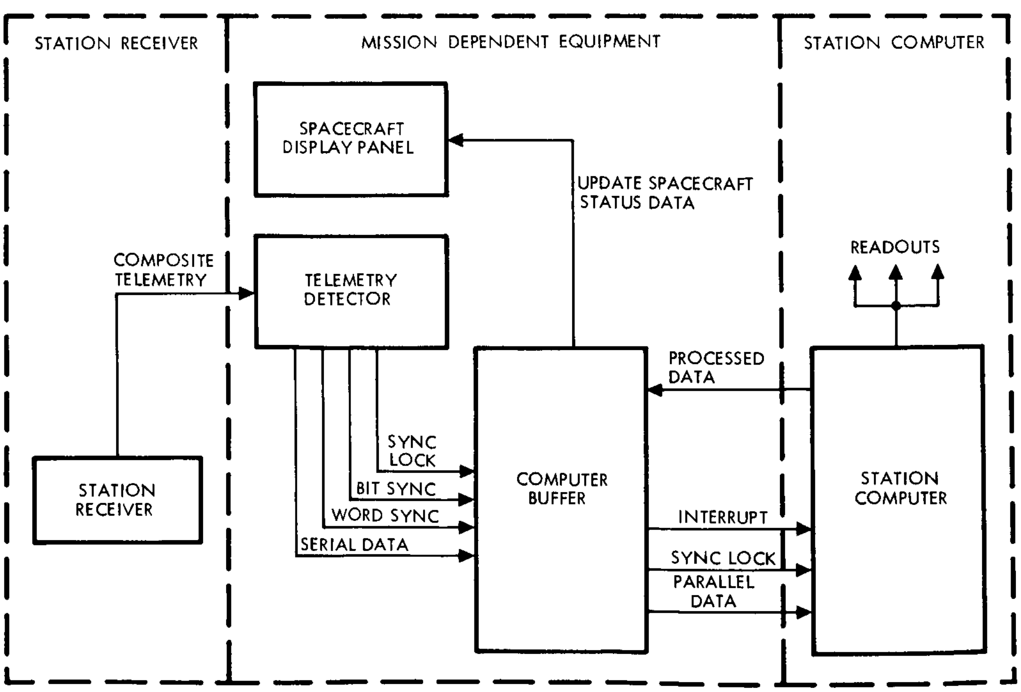


Figure 10. Block Diagram of MDE/DSIF Telemetry

Start-transmit signals
Command bit stream(serial)
Bit check interrupts
Error-stop signals
Emergency mode signals
Spacecraft status signals

To verify proper operation of the in-line functions of MDE prior to mission use, testing and calibrating capabilities of the mission operations system test equipment will be employed. Figure 11 schematically presents the MOSTE-MDE interfaces.

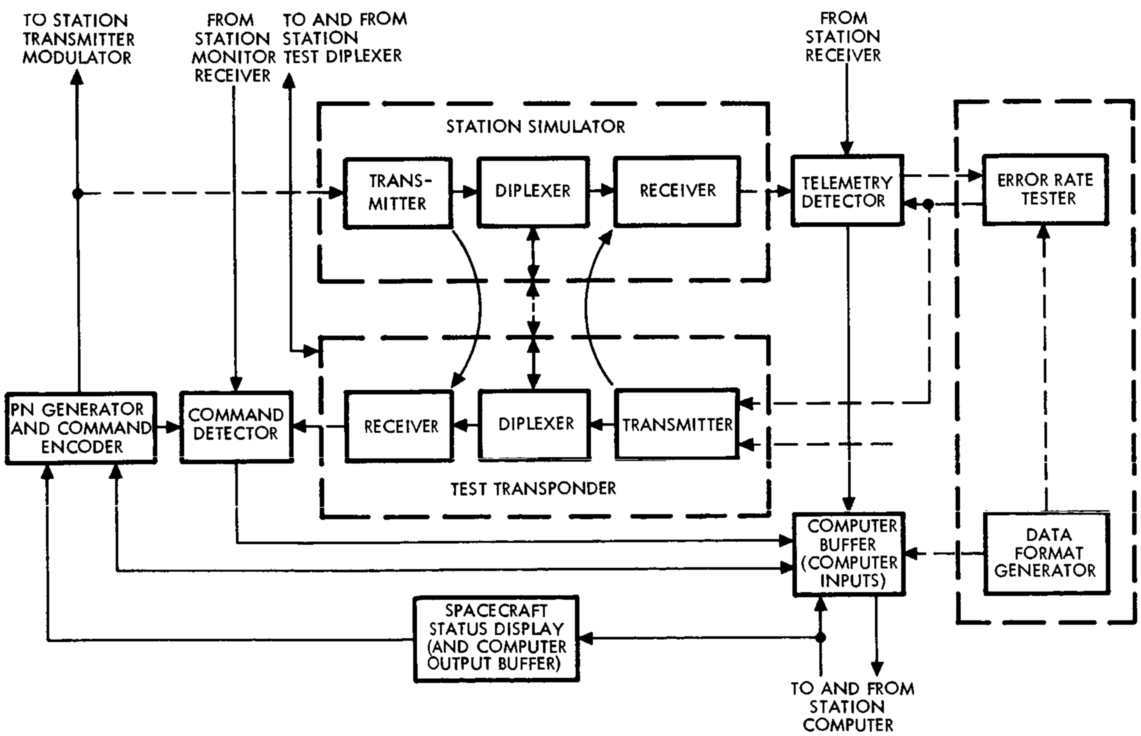


Figure 11. Mission Dependent Equipment/MOSTE

The MDE telemetry detector receives simulated telemetry signals from the MOSTE error rate tester, the data format generator, or the station simulator receiver. Telemetry detector output is the simulated telemetry bit stream returned to the error rate tester for checking. The MDE command encoder outputs a PN-coded command to the MOSTE station simulator transmitter. It is demodulated by the MOSTE test transponder receiver and fed to the MDE command detector for reconstruction. The MDE computer buffer receives MOSTE data format generator signals to allow computer buffer tests and station computer handling of limited telemetry tests.

During system test when radio silence is not required, command and telemetry can be enabled through the RF link from MDE to spacecraft. The MDE commands modulate the transmitter section of the MOSTE station simulator, whose output is received by the spacecraft. The spacecraft telemetry RF signals are received by the MOSTE station simulator receiver and sent to the MDE telemetry detector.

During radio silence, hard-line command and telemetry (through data-links) are used. Automatic commands, without PN-code are initiated by the station computer and sent from the MDE command encoder to the spacecraft command subsystem. Telemetry bit stream and synchronizing signals are fed back from the spacecraft telemetry system (without PN) to the MDE computer buffer.

Since the requirements is for simultaneous command, telemetry, and ranging to each of two Voyager spacecraft, two channels of MDE are required for each DSIF site utilized. Each spacecraft has two command decoders and each decoder is addressable. One MDE command encoder is set up to address either of the command decoders of spacecraft B. These two forms of protection against incorrect spacecraft response (different receiving frequency for each spacecraft, plus spacecraft decoder addressability) are augmented by a third measure, the use of a different PN code format for each spacecraft.

3.3.4 Design Details

Figure 8 illustrates the layout of the MDE. Mission operations system test equipment is also shown in the figure since it closely interacts with the MDE. Figure 11 is a block diagram of MDE and MOSTE interfaces with the DSIF. Command function in the DSIF is shown in the block diagram of Figure 9 and the telemetry function in the DSIF is shown in the block diagram of Figure 10. (See also Section 3.3.5.)

a. PN Generator

The PN generator accepts the command encoder output signal consisting of an NRZ bit stream at a rate of 1 bit/sec. This signal biphase modulates a PN code of 511 bits which is repeated during each command bit. The resulting signal is provided as input to the station transmitter modulator. The PN generator can generate both code formats required by the redundant spacecraft command encoders. PN format selection is made manually.

b. Command Encoder

The command encoder section provides for manual entry of any commands which can be accepted by the Voyager spacecraft. Upon manual initiation, the command is transmitted via the computer buffer to the station computer, where it is checked for permissibility. If the computer check verifies permissibility, this fact is displayed on the command encoder panel and transmission of the command to the station transmitter modulator can then be manually initiated. If the command is not permissible, this fact is displayed on the command encoder panel and transmission of the command is prevented. As the command bit stream is being transmitted, a further check of command transmission is made external to the command encoder. If any bit of the command is found to be in error, the computer will abort the command transmission. (The spacecraft responds only after receiving complete command formats.) A backup mode of operation is provided in which a command may be transmitted without the normal computer checks of the command.

c. Command Detector

The command detector accepts the output of the DSIF station monitor receiver, which is of the same general form as the output of the PN generator: a 511-bit PN code repeated synchronously once each command bit period and biphase modulated by the command bit stream. The command detector recovers the command bit stream from the monitor receiver output signal and applies it to the computer buffer input. The command bit stream is also decoded and displayed on the front panel of the command detector. The unit is mechanized in such a way that the last command received is displayed until operation is interrupted or until a new command is received. The command detector can also accept the output of the test transponder or the direct output of the PN generator.

d. Spacecraft Status Display

The spacecraft status display unit, in addition to having indicators for display of spacecraft status, contains the computer output buffering circuits necessary to operate these displays, to transmit command verification and inhibit signals to the command encoder, and to operate audible and visual alarms.

e. Telemetry Detector

The telemetry detector accepts the phase-demodulated telemetry signal from the DSIF station receiver. The data bit stream, bit synchronization, and word synchronization signals are extracted from the PN signal. Data extraction occurs with an error of less than one in 1000 bits at a signal-to-noise ratio within a few db of theoretical. To initially acquire the telemetry data, the detector voltage controlled oscillator automatically operates in a frequency sweep mode over the necessary range. At a S/N within a few db of theoretical, synchronization will occur within a minimal number of sweeps (approximately two) of the VCO. When synchronization occurs the acquisition mode is automatically terminated.

The telemetry detector provides a data bit stream, a bit synchronization signal, a word bit synchronization signal and a synchronization status signal to the computer buffer.

f. Computer Buffer

The computer buffer accepts the serial telemetry and synchronization outputs of the telemetry detector and at proper intervals interrupts the DSIF station computer to enter the telemetry data and synchronization status signals. The computer buffer accepts the parallel command signal and the verification request discrete signal from the command encoder. When the command request occurs the buffer interrupts the DSIF station computer to enter the command (and verification request). Upon receipt of a pulse generated in the middle of each command bit, the computer buffer transmits an interrupt to the computer signalling the computer to make an error check of the command bit currently being transmitted.

3.3.5 Functional Description

The command encoder/pseudonoise generator drawer is responsible for encoding spacecraft commands manually entered by the switches on the front panel of the drawer. The command generated is PCM 1 bit/sec NRZ containing a maximum of 60 bits. Two functions are combined into the one drawer: the PN generator occupies little space and easily and logically combines with the command encoder.

The PN generator section of the command encoder generates a 511 bits/sec PN code which is repeated synchronously with each command bit. The accuracy of command generation is assured by a scheme of command verification steps, both manually and computer controlled, which ensures the correctness of the command both before and during its transmission by the DSIF station to the spacecraft. Spacecraft decoder address and command are manually set up on the front panel of the command encoder. When the command entry switch is depressed, address and command are stored in an internal register and displayed on the command encoder front panel. Before transmission of the command, the "command verify" switch is depressed, submitting the command to the computer for permissibility check.

If the command is permissible, this fact is displayed on the command encoder panel, at which time command transmission by the station can be initiated manually (again from the command encoder front panel). The 1-bit/sec command biphase modulates the 511-bit PN code and the composite signal is used to phase modulate the DSIF transmitter. The station monitor receiver phase-demodulates a sample of the DSIF RF signal and sends it to the command detector. The detector strips the command bit stream from the composite command signal, displays the command on its panel, and sends it to the computer through the computer buffer. In the computer, the command from the detector is compared bit by bit with the original command in the computer memory. Near the middle of each bit being transmitted, an interrupt signal (at bit sync rate) is sent from the command encoder through the computer buffer to the computer, requesting computer comparison of and concurrence with the bit being transmitted. If the computer detects an error in any bit prior to the last bit of a command, it will, through the buffer, abort the command and cause a "stop" indicator to be lit on the command encoder panel. Since the spacecraft acts only on complete commands, the partial command will be ignored.

A telemetry detector strips the spacecraft telemetry signal after phase demodulation by the DSIF station receiver. The telemetry data stream is sent to the computer via the computer buffer. The station computer decommutates the data and provides a printout on the line printer. Applicable indicators in the spacecraft status display unit are lit by the computer through the computer buffer interface. A given display remains on the spacecraft display unit until changed by the computer.

3.3.6 Changes from Task A

These changes are incorporated in the MDE described here as compared to that in the Task A report:

- One MDE per spacecraft
- No commercial test equipment included in MDE.
- Accommodations for computer-originated program-controlled commands (in the STC configuration of the MDE only).

3.4 Mission Operations System Test Equipment

The over-all function of mission operations system test equipment (MOSTE), in conjunction with a DSIF station computer, is to provide the special circuitry necessary for MDE performance tests during MDE-DSIF integration, for performance verification when not operating with the spacecraft, and for troubleshooting and maintenance. MOSTE provides an RF interface with the spacecraft and plays an important role in MDE readiness and compatibility testing. MOSTE consists of an error rate tester, a data format generator, a station simulator, and a test transponder.

3.4.1 Requirements and Design Constraints

MOSTE is employed when the MDE under test is not in a mission configuration with the DSIF station. MOSTE must operate in the presence of possible electromagnetic radiation from adjacent equipment or from the DSIF equipment. Conversely, MOSTE is designed to prevent conducted or radiated signals from leaving its rack at levels which will interfere with

DSIF functions. Power dissipation in the MOSTE rack is subject to the same limitations as in the MDE racks: 2000 watts maximum per rack. Cabling for the MOSTE rack will be compatible with the trough type cable ducts provided in the DSIF.

The MOSTE supplements the in-line MDE to enable a high level of premission self-test and self-calibration of MDE. The MDE performs its in-line function in the DSIF but does not contain within it the telemetry generation and comparison circuits necessary to evaluate its own detection error rate when it is off-line between missions. These functions are provided by the MOSTE. The MOSTE generates test signals to the MDE, introduces parameter variations to enable the drawer under test to sense threshold or near-threshold conditions, possesses means of measuring and displaying performance errors in a drawer under test, and has flexibility of interfaces so it can be used in alternate configurations, i.e., in the RF loop test as well as in tests where it is connected directly to the MDE telemetry detector. Simulated telemetry signals can be generated by the error rate tester as well as simulated marginal signal-to-noise ratios by mixing a noise source with the telemetry signals. A true rms voltmeter designed into the error rate tester enables measurement of signal and noise levels. A counter circuit is used to select the number of bits which are used as a sample for test. A comparator circuit is provided to detect and display errors. A data format generator permits each individual main-frame or subcommutated word address to be filled with different data.

3.4.2 Functional Interfaces

The data format generator and error rate tester drawers, as well as their power supply and control, are designed to fit standard 19-inch JPL DSIF racks, as shown in Figure 8. Inter-rack cabling and cabling to the DSIF J-box is compatible with the sub-floor troughs provided in the DSIF.

The error rate tester can be used with the data format generator, forming a comprehensive testing and calibrating subsystem for the MDE telemetry detector. (By connecting the test transponder transmitter and the station simulator receiver into this subsystem, a closed-loop RF test can be accommodated.) The MDE telemetry detector strips the data bit stream from the simulated telemetry signals and feeds it back to the MOSTE error rate tester, where the signals are compared and errors displayed. In the RF test case, the output of the error rate tester is used to phase-modulate the MOSTE test transponder transmitter. The MDE telemetry detector then receives the simulated telemetry from the MOSTE station simulator receiver.

The MOSTE data format generator can be connected to the DSIF station computer through the computer buffer. Meaningful data can then be generated and the operating-history recording capabilities of the station are available. The MOSTE test transponder and station simulator RF inputs and outputs interconnect in diplexers. The station test diplexer is available for special RF tests in which it may be necessary to feed RF to the station receiver or receive signals from the station transmitter. This test diplexer is located in the base of the DSIF antenna structure, which results in a loss of 20 db from the MDE and MOSTE.

In system test the spacecraft communications subsystem receives MDE commands radiated by the MOSTE station simulator transmitter. The spacecraft telemetry signals are received in the MDE (telemetry detector) through the MOSTE station simulator receiver.

3.4.3 Equipment Description

Figure 11 is a general block diagram showing MOSTE with MDE as it is in a DSIF station. The dashed lines indicate signal paths for test when MDE is off-line. Figure 8 is a rack layout showing the mission operations system test equipment in its DSIF configuration.

Figure 12 is a block diagram of the data format generator. It allows manual selection of mode, format, and bit rate. Word selection switches are used to address any main-frame word or any subcommutated word. By means of a manually-operated switch register, any possible binary data can be entered in the selected telemetry word. Special words such as frame synchronization and subcommutation identification words are automatically generated. Words not selected for specific content contain a unique code. Alternately, all words (except special words) can be made to contain the contents of the switch register. Correct parity

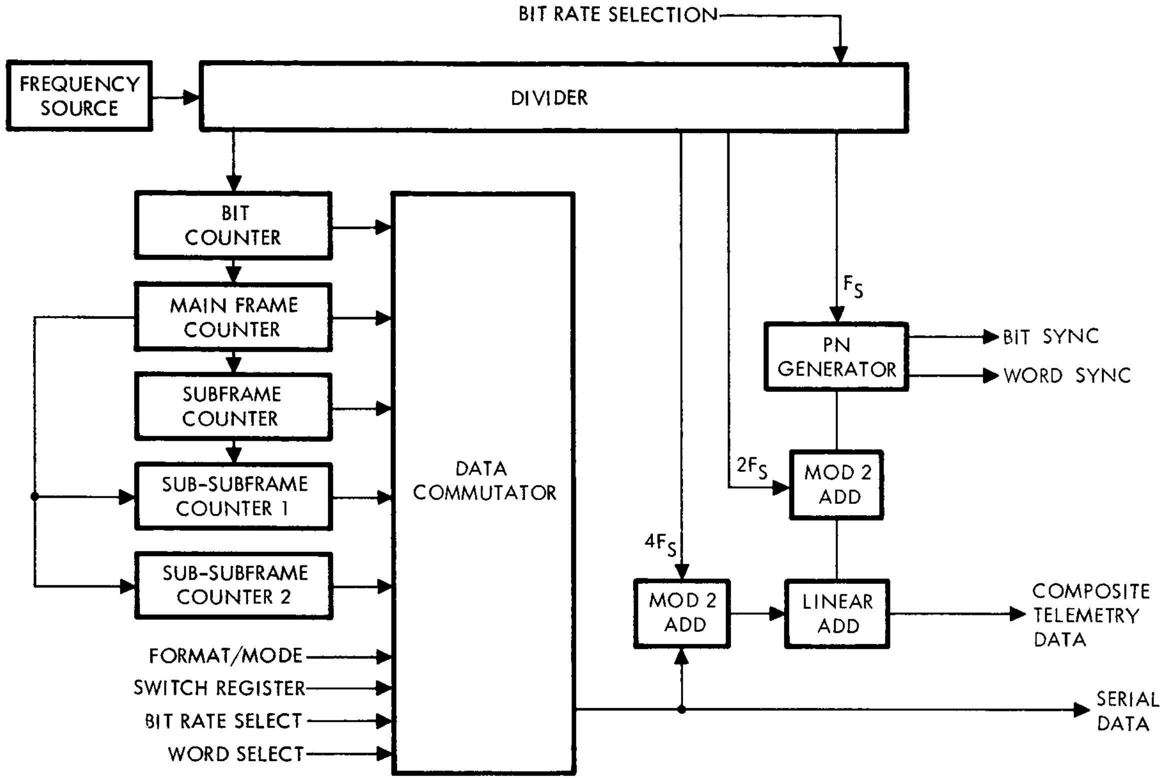


Figure 12. Block Diagram of Data Format Generator

bits can be automatically generated for all words using parity, or incorrect parity bits can be manually selected. The data format generator can be operated at any of the bit rates of the Voyager telemetry data, and any of the Voyager telemetry data formats can be selected.

Figure 13 is a block diagram of the error rate tester. It provides controls for selecting, entering, adjusting, and measuring MDE telemetry detector testing parameters. The error rate tester simulates the spacecraft telemetry signal (having meaningless data words) normally obtained from the DSIF station receiver output. A noise signal may be added to this signal in a known ratio and the combined signal applied to the MDE telemetry detector input. The output of the telemetry detector is returned to the error rate tester, where the detected bit stream is compared bit-by-bit with the simulated telemetry bit stream.

The simulated telemetry output signal of the data format generator or playbacks of previously recorded telemetry data can be selected to

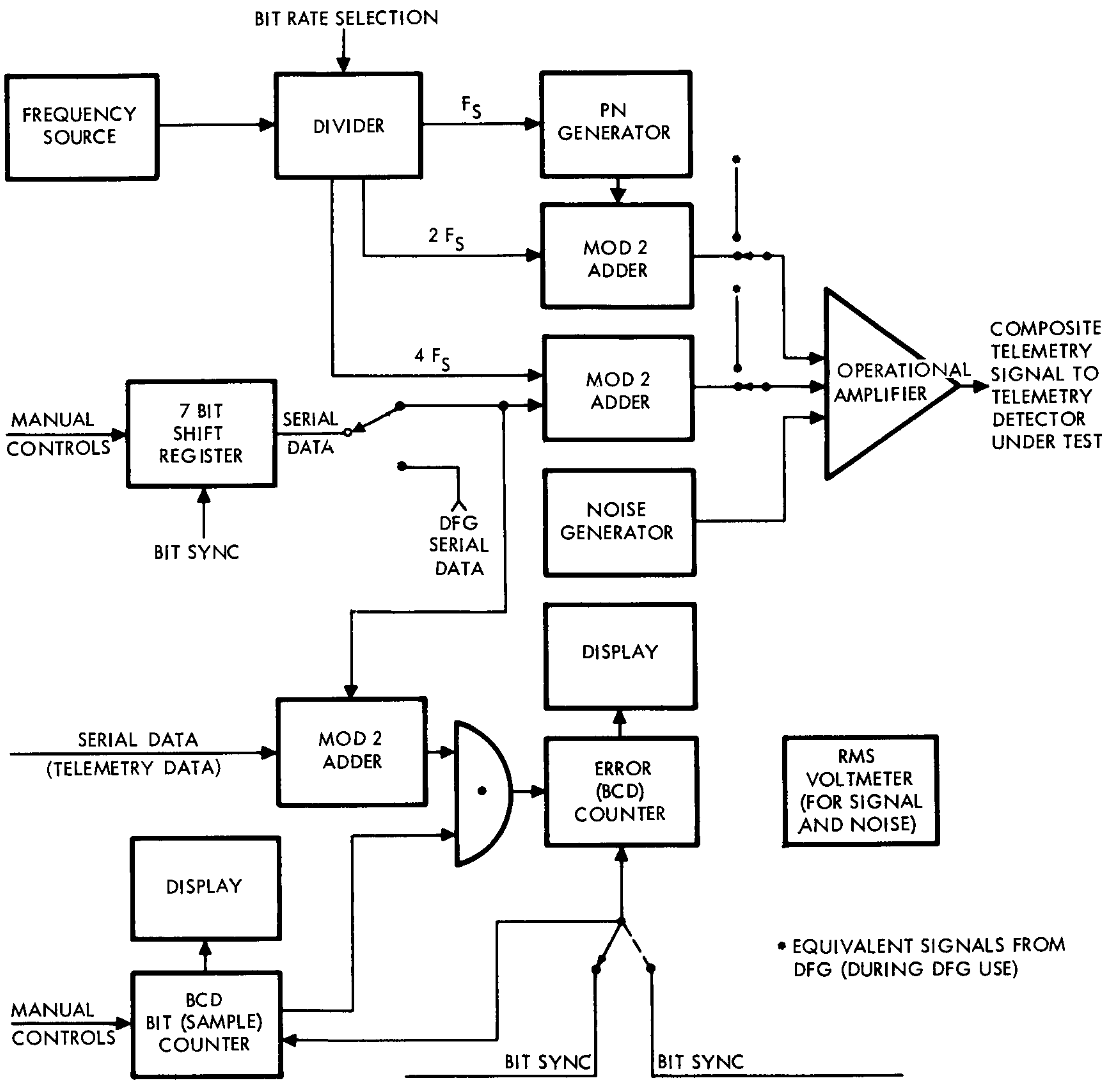


Figure 13. Block Diagram of Error Rate Tester

replace the error rate data generated by the error rate tester. A preselected number of bits are generated and, during the period of their generation, detection errors are counted and displayed, providing an indication of the quality of performance of the telemetry detector. Error sample bit counts of 10^3, 10^4, 10^5, and 10^6 can be selected. Bit rates corresponding to the various spacecraft telemetry bit rates may also be selected. Three signal-to-noise ratios can be selected by direct switch calibration. One of these is at a signal-to-noise ratio corresponding to a theoretical performance of one error in 10^3 bits; the other two are within a few db of that level. Both the signal and the noise generated can be individually adjusted and measured by a true rms voltmeter, with the error count being displayed on an included counter.

When it is desired to include an RF loop test, the simulated telemetry data is fed to the test transponder, where it phase-modulates the transponder in the same manner as in the spacecraft. The test transponder output is then reduced to a level such that the MOSTE station simulator receiver used to receive its signal will have an output with noise, characteristic of small signal reception. In this case, the noise generator in the error rate tester is not used and the signal from the station simulator receiver is directly connected to the comparator of the error rate tester.

3.5 Launch Complex Equipment

The LCE is that group of EOSE which is required to evaluate and condition two spacecraft for simultaneous launch. It includes equipment for automatic control and checkout of the two spacecraft during test mating to the launch vehicle at the VAB, during prelaunch testing on the pad, and during countdown and launch.

The elements comprising the LCE are two system test complexes, two local data handling racks, two data links, a launch control complex console, and a junction box (this last assumed to be furnished by the launch vehicle instrumentation contractor). The STC elements which play active roles in LCE are the test director's console, central data system, mission dependent equipment, and telemetry, power, and radio subsystem EOSE.

3.5.1 Requirements

The major requirements which have been placed on the LCE are:

- Testing via umbilical and RF
- Manual control of spacecraft
- Automatic control of spacecraft during countdown
- Isolate spacecraft test faults
- Supply external power

- Record inputs to spacecraft, spacecraft data, power, interface signals, and external instrumentation data
- Self-test without interruption of spacecraft operation
- Backup emergency power for critical and safety functions
- Limited testing during "RF silence"
- Provide interface equipment for signals routed within and from launch complex
- Interchangeable between different spacecraft.

3.5.2 Functional Interfaces

The spacecraft electrical interfaces with the LCE are the spacecraft umbilical and the RF interface. Exact signal identities will not be defined until Phase IB. The RF interface includes telemetry, ranging, and commands. The umbilical inputs include as a minimum telemetry, power control, voltage/current monitoring, commands (discrete and command detector input), and pyrotechnic safe control.

There are also interfaces of the LCE with the capsule LCE and the launch vehicle support equipment; these remain to be defined in Phase IB.

3.5.3 Design Description

The LCE is derived directly from the STC, using STC elements plus a few LCE-peculiar items. Differences from the STC are:

- Some elements are relocated, giving rise to a need for data links and equipment to condition the hardline signals for transmission over those links.
- Direct access connections to the spacecraft, other than the flight umbilical, are eliminated, thus reducing hardline monitoring.
- Usable system stimuli are reduced, particularly nonelectrical, such as guidance and control stimuli.
- Additional launch-oriented software is provided.
- Reflectors or parasitic antennas are added to direct the RF signal from the shroud window to the STC.

a. Configuration

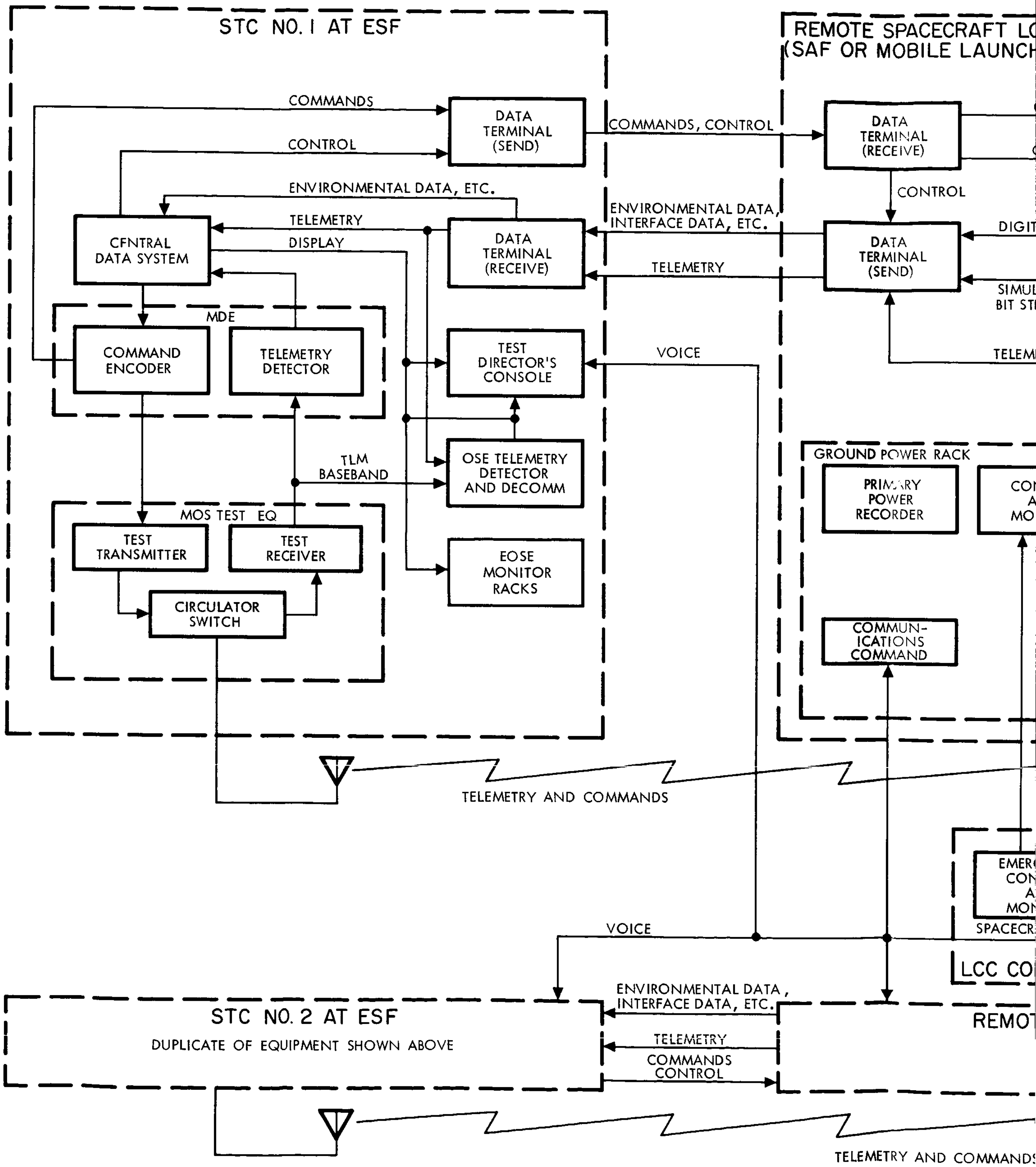
Figure 14 is a simplified block diagram of the LCE; the following discussion refers to that diagram.

Commands are generated in the MDE command encoder in the STC. The encoder incorporates both manual and computer command set-up and initiation. The commands are then transmitted via the radio subsystem OSE transmitter to the spacecraft receiver (or in case of radio silence, via the data link), and thence to the spacecraft command detector. If the computer is down, commands may be manually generated from either the test director's console or the MDE command encoder.

When the RF link is used, telemetry is sent through the MOS TE station receiver simulator, the MDE telemetry detector, and into the CDS computer for processing and CRT display. In addition, the telemetry baseband out of the receiver is decommutated in the telemetry subsystem OSE; the resulting data is sent to the various word selector and display units, primarily for computer-down backup.

During RF silence, the telemetry signal is sent from the remotely located spacecraft to the STC at the explosive safe facility (ESF) via data link. By using the data stream prior to PN coding, the bandwidth handled by the data link is minimized (15 kilobits/sec bit rate versus 270 kilobits/sec when PN coded). Telemetry sync is also sent to facilitate handling the data stream. The data link telemetry stream is fed into the CDS computer and the telemetry subsystem EOSE.

A requirement has been established to record signals to or from interconnecting equipment and external instrumentation data. In addition, other signals which are present on the umbilical and elsewhere at the spacecraft test location and which should be recorded, may not be on spacecraft telemetry. Since the computer should monitor and log these signals in real-time, this data will be telemetered to the ESF via another data link channel. As shown on the block diagram, the analog signals are digitized and scanned with the conditioned discretes in the local data handling rack and put on one data link channel. The bandwidth requirements should, for data of this type, be modest (less than 1 kc).


STC NO. I AT ESF
COMMANDS
CONTROL
DATA TERMINAL (SEND)
COMMANDS, CONTROL
ENVIRONMENTAL DATA, ETC.
TELEMETRY
DISPLAY
CFNTRAL DATA SYSTEM
DATA TERMINAL (RECEIVE)
ENVIRONMENTAL DATA, INTERFACE DATA, ETC.
TELEMETRY
MDE
COMMAND ENCODER
TELEMETRY DETECTOR
TEST DIRECTOR'S CONSOLE
VOICE
TLM BASEBAND
OSE TELEMETRY DETECTOR AND DECOMM
MOS TEST EQ
TEST TRANSMITTER
TEST RECEIVER
CIRCULATOR SWITCH
EOSE MONITOR RACKS
REMOTE SPACECRAFT LC
(SAF OR MOBILE LAUNCH
DATA TERMINAL (RECEIVE)
CONTROL
DATA TERMINAL (SEND)
DIGIT
SIMUL
BIT ST
TELEM
GROUND POWER RACK
PRIMARY POWER RECORDER
CO
A
MO
COMMUN-ICATIONS COMMAND
TELEMETRY AND COMMANDS
EMER
CON
A
MON
SPACECR
LCC CO
VOICE
ENVIRONMENTAL DATA, INTERFACE DATA, ETC.
STC NO. 2 AT ESF
DUPLICATE OF EQUIPMENT SHOWN ABOVE
TELEMETRY
COMMANDS CONTROL
REMOT
TELEMETRY AND COMMANDS

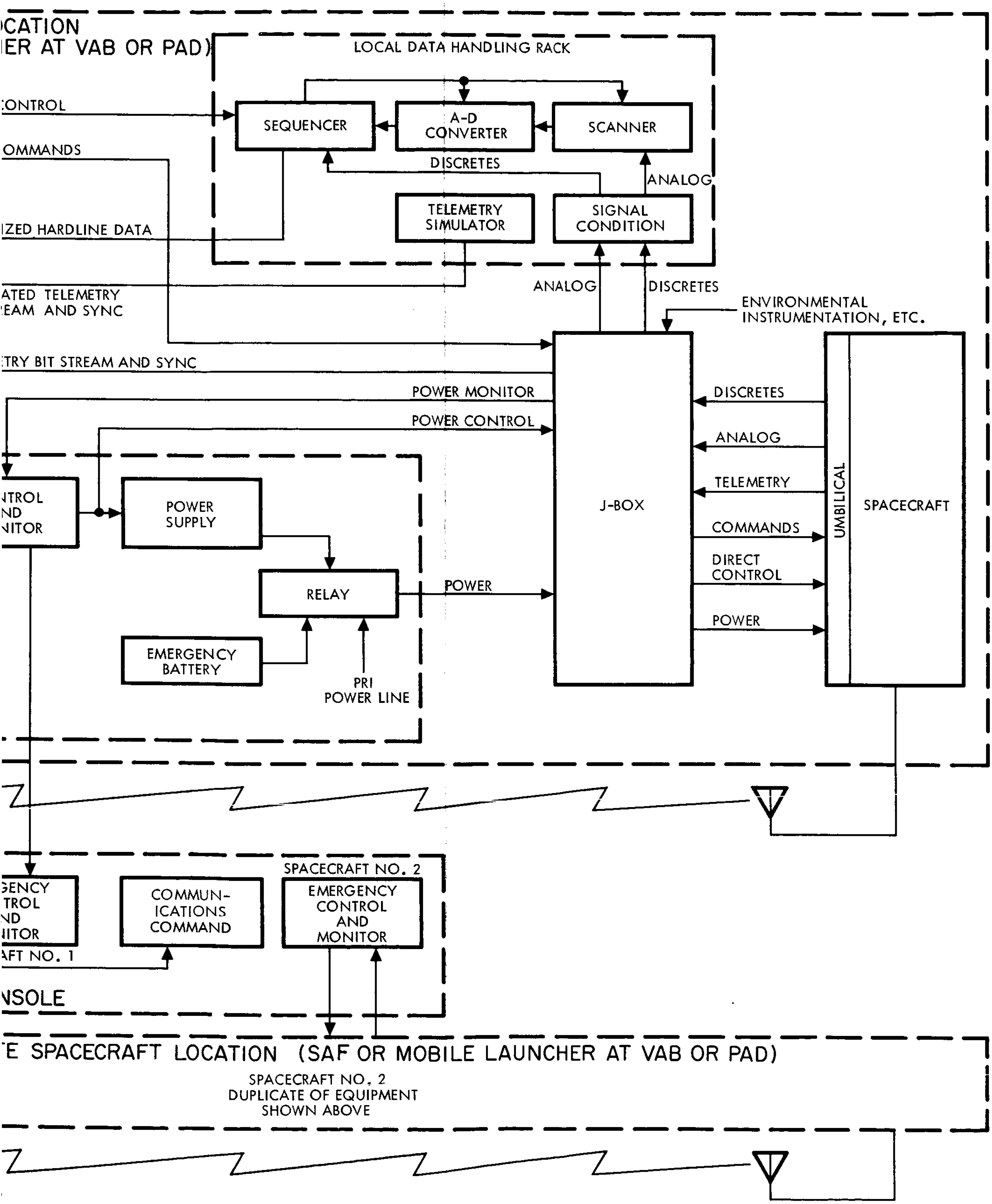


Figure 14. Launch Complex Equipment Block Diagram

2

Ground power is supplied from a rack of the power subsystem EOSE, which is moved from the ESF to the spacecraft location. This rack contains the solar simulator power supply, the emergency battery source, and the prime power source. Control of the power supply is available via the local control and monitor panel and via hardline from the control and monitor panel in the LCC console. The battery supplies emergency control and indication to allow nondamaging shutdown in the event of a main power failure. It is automatically switched in by a relay de-energized by loss of main power. The prime power is recorded as an analog trace on a strip chart rather than as a digitized sample in order to permit the recording of transients.

In addition to control and monitoring of electrical power, other critical functions (pyrotechnic functions, propellant pressures, etc.) are hardlined to the LCC console. These functions are mechanized with extremely high reliability, using redundancy and fail-safe techniques combined with simple control and indication devices. Two of these panels exist, one at the spacecraft location and one at the LCC. They are identical in appearance but somewhat different internally (e.g., drivers).

In addition to the facility MOPS circuits, a separate emergency intercomm between the ESF (test director's console), the LCC and the mobile launcher will be provided.

The equipment blocks and signal paths shown in the block diagram are all duplicated for the second spacecraft except for the junction box, the LCC console, and portions of the voice communications net.

b. Operation

In the automatic mode, the LCE is controlled by the program in the CDS computer, just as for STC system test. The OSE set-up and control portions of the program are sharply reduced, however, since almost all spacecraft monitoring is via telemetry, and the access for stimulation is greatly reduced. Stimulation by electrical command will still be possible, and some sensor checks will be run with simple light fixtures in the shroud prior to sterilization and seal. However, external guidance and control sensor stimulation of the type used in system test will not be possible. Test director manual override capabilities are available, as in STC system test use.

The monitor rack at each subsystem OSE is manned by the subsystem engineer, who observes the cathode ray tube and discrete displays, and can make requests for special displays of computer data.

The spacecraft test conductor is located at a test director's console in the ESF. When both spacecraft are operating, as in simulated or actual launch, each test conductor is at a console, along with his assistant. The test conductors are in voice communication with the LCC console operator.

The LCC console operator monitors critical hardline indicators from both spacecraft and can report to the test conductor on safety status in the event of major ESF-Pad communication failures (such as data link failure during radio silence or spacecraft telemetry failure). He has available the necessary hardline controls to safely shut down the spacecraft, should this action be indicated.

Operation during radio silence is essentially identical to that described above; in fact, the test engineers will not be conscious of the difference in mode, except for two functional differences:

1) Operability of the spacecraft radio subsystem is not demonstrated by testing.

2) PN coding and detection of telemetry is excluded from the test loop.

Faults are detected by computer comparison of telemetered parameters with stored tolerances, as in system test. Since no direct access connections are possible, other than those on the umbilical, the capability to detect marginal internal performance and trends is somewhat less; the extent to which this is the case depends, of course, on the number and distribution of telemetered points.

Display of test results and existence of faults is, as in STC testing, by a combination of cathode ray tube displays and illuminated legend indicators. The recognition of marginal conditions and their probable eventual effect is necessarily dependent on the ability of the subsystem engineers, aided primarily by the computer software and display capability of the CDS.

Fault isolation is accomplished by the self-check capabilities of the EOSE. In particular, the simulated telemetry signal at the mobile launcher is periodically gated into the data link by the computer and the received data compared with the known frame content to verify the data link, the MDE telemetry decommunication, and the computer telemetry processing.

Although it is unlikely that a launch would be attempted without the CDS, LCE manual backup capability has value in allowing continuation of testing. Manual capabilities are essentially those mentioned in the STC description (Section 3.1). They include manually selectable telemetry word display (one word of numerics in non-engineering units and one word of discretes), manual command set-up and initiation and the LCC emergency monitor and control.

c. Elements of LCE

The elements of the LCE are the STC, junction box, local data handling rack, data link, LCC console, and ground power source. The STC has previously been described.

The junction box terminates, and allows access and connection to, all leads from the umbilicals of both spacecraft, environmental sensors related to the spacecraft, etc. Since these leads are part of the total number of leads which emerge at the instrumentation unit, it is assumed that the junction box is the responsibility of the instrumentation unit contractor and that the spacecraft and capsule contractors will be allotted necessary terminals and access. A block diagram of the junction box is shown in Figure 15.

The local data handling rack contains the digital equipment which must be located at the spacecraft. A diagram of this equipment and a rack layout are shown in Figure 16. The function of this equipment is to condition and format the various hardline signals at the mobile launcher which are to be recorded by the CDS computer. These include

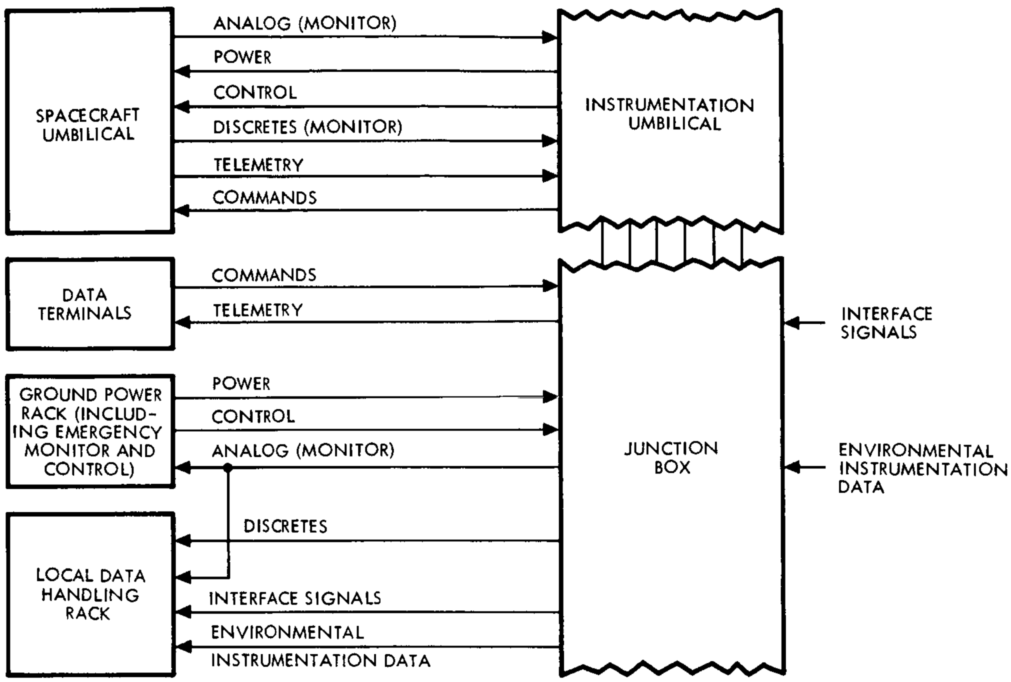


Figure 15. Junction Box Block Diagram

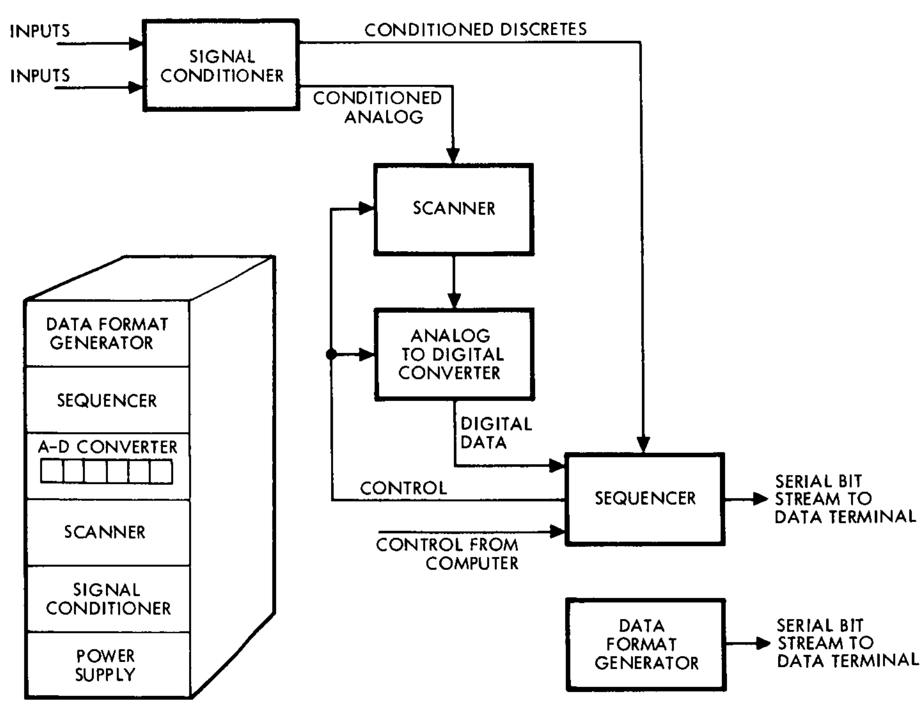


Figure 16. Local Data Handling Rack and Block Diagram

environmental instrumentation outputs and umbilical signals (primarily interface signals to and from the launch vehicle or capsule) which are not on spacecraft telemetry or which are of such importance as to be monitored redundantly to spacecraft telemetry.

The components in the rack, which perform straightforward low-speed digital data acquisition functions, are briefly described below.

- Signal conditioner. Analog signals are converted to nominal amplitude DC for conversion to digital form. Discrete signals are converted to the binary voltage levels required by the transmitting data terminal (see Figure 14).
- Scanner. Conditioned analog signals are scanned to time-share one A-D converter.
- Analog-to-digital conversion. Conditioned analog signals are sequentially converted to binary form, the result being available in an output register.
- Sequencer. The A-D converter output is sampled and combined with the conditioned discretes into a serial bit stream suitable for sending over the data link to the CDE computer. The sequencer also supplies the necessary sync and identification bits.
- TLM simulator. The telemetry simulator is a frame generator of the type described as part of the telemetry subsystem OSE but without PN coding. It is unrelated to the rest of the equipment in the rack and located there for convenience only. Its function is self-check of the data link-MDE demodulator-computer combination.

Data channels are required in both directions between the ESF and the spacecraft location. The limiting bit rate requirements are approximately 15 kilobits/sec for telemetry and approximately 500 bits/sec for command. Other bit rate requirements are expected to be correspondingly low; they will be established in Phase IB, as will be the type of (commerically available) data link to be used.

The LCC console is a simple assembly consisting of three panels: a voice communications panel with selector switches, microphone and phone jacks, and two identical emergency control and monitor panels, arranged as shown in Figure 17. The panel itself consists of lamps and switches hardlined (via appropriate DC driving circuits) to

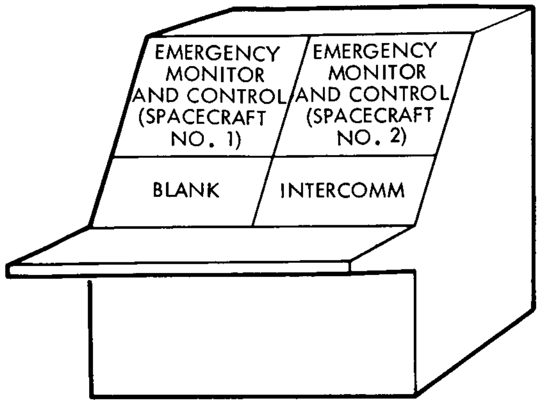


Figure 17. Launch Control Complex Console

the spacecraft umbilical and the ground power source; these hardline connections are indicated in Figure 18.

The ground power rack is one of the power subsystem OSE racks and is moved to the mobile launcher. The rack contains the solar power simulator, emergency battery source, and the power recorder. Added to this rack is a local duplicate of the emergency control and monitor panel of the LCC console, which includes the power supply controls.

3.5.4 Changes from Task A

In both Tasks A and B the LCE is comprised of STC's rearranged geographically with a minimum of LCE-peculiar additional equipment. The geographical arrangement is different from that in Task A and is related to the basic differences in the Voyager operation plans at Cape Kennedy (tandem planetary vehicle rather than two launches, etc.). The Task A plan was to use the SAF as the location for the system test sets with ground power and RF consoles of the system test sets moving to the ESF and pad along with the spacecraft. The current plan is that the three STC's will be located at the ESF; ground power and local data handling, but no RF test set, will be located with the spacecraft. The reasons for the selecting the ESF as the STC location are discussed in Volume 3 of this report, and in Appendix B of this volume.

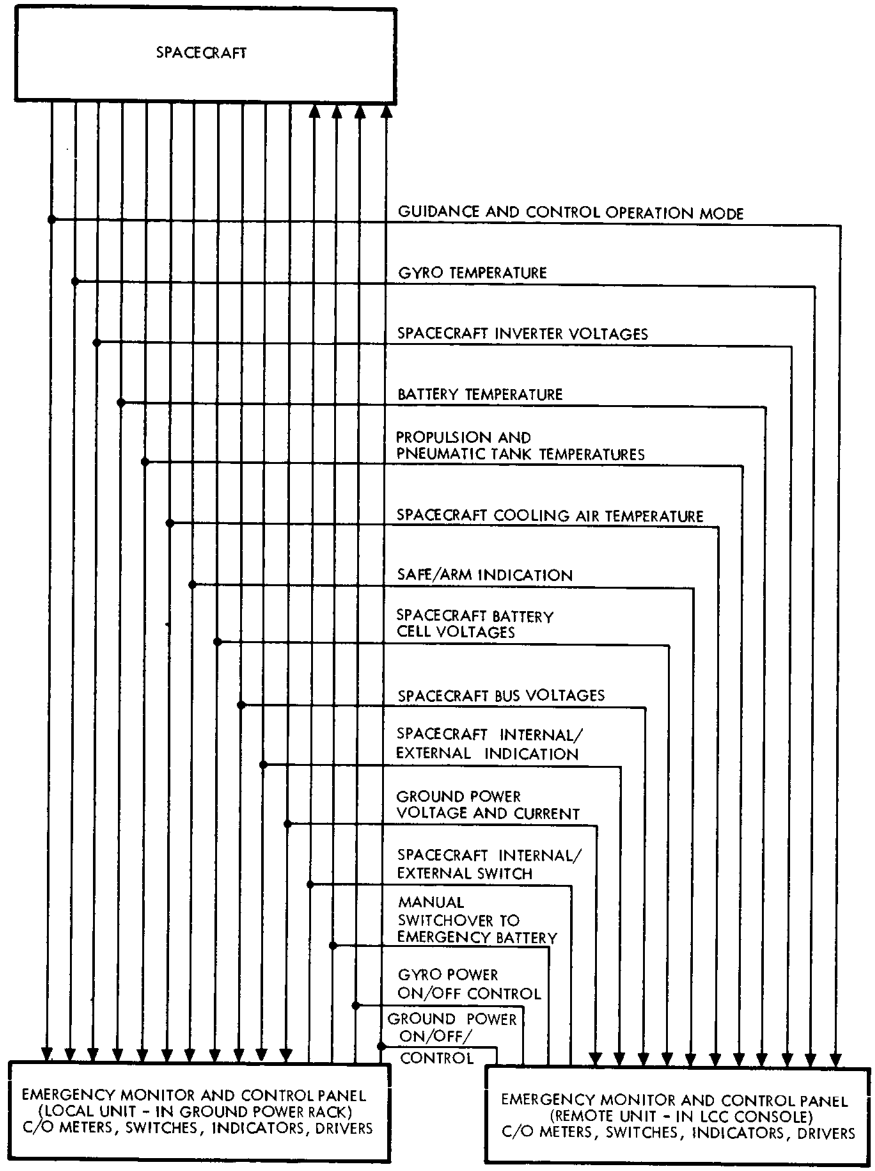


Figure 18. Emergency Monitor and Control Block Diagram

4. SUBSYSTEM LEVEL FUNCTIONAL DESCRIPTIONS

This section contains functional descriptions for each of the subsystem EOSE test sets which are used for end-to-end testing of the corresponding spacecraft subsystem. In addition to their independent application for spacecraft subsystem performance verification, they are used in conjunction with all other subsystem EOSE test sets in the system test complex and in launch complex equipment. Associated with each subsystem test set (whether employed independently or as part of the STC or LCE) is a data entry and monitor rack from the central data system which provides a common interface between the CDS general purpose computer and each of the subsystem test sets. Operation of the subsystem test sets is primarily in the automatic mode with manual backup. When used as part of the STC or the LCE, the manual backup mode provides selected telemetry data generated in the telemetry subsystem EOSE and distributed to the telemetry status panels.

Following in this section are functional descriptions of the EOSE for

- Power subsystem
- Computing and sequencing subsystem
- Guidance and control subsystem
- Radio subsystem
- Telemetry subsystem
- Command subsystem
- Data storage subsystem
- Pyrotechnics subsystem
- Propulsion subsystem

4.1 Power Subsystem EOSE

The power subsystem EOSE will perform functional tests on the spacecraft power subsystem and power distribution circuits, from the detailed subsystem testing phases through integrated spacecraft system tests. It will operate in the automatic mode with a manual backup capability. The power subsystem EOSE will provide simulated solar array voltage and current, variable over the design margin range. Connections between this EOSE and the spacecraft power subsystem will permit testing of the solar panel, power inverter, power control unit, battery, battery control, and the electrical distribution assemblies. Self-test capability is provided by this EOSE during the automatic and manual mode.

4.1.1 Test Objectives and Design Criteria

The power subsystem EOSE will provide the spacecraft power subsystem with the following simulation, stimulation, and control functions:

- Simulated solar array voltage and current: variable 400 and 4000 Hz power to provide for power margin tests on the spacecraft.
- Simulated battery voltage variable between 29 and 42 vdc.
- Inverter and battery loads to simulate the various loads during the different modes of operation.
- Shunt element simulation provided from isolated sources.
- Simulated C and S subsystem commands.
- Battery undervoltage and overvoltage control.
- Light source to stimulate the solar panels.

The following functions will be monitored by the power subsystem EOSE during both subsystem and system level testing:

- Battery cell voltage and temperature
- Inverter output current and voltage under various load conditions
- Telemetry sensors voltage
- Power distribution voltage and control capability
- Sync frequency

Functional interfaces between the power subsystem EOSE and the DCS computer are made through the data entry and monitor rack. This rack accepts the address and data outputs from the computer to control the power EOSE and relays data from the EOSE to the computer.

4.1.2 Operating Characteristics

Figure 19 is a functional block diagram and Figure 20 is a rack layout of the power subsystem EOSE. The following description is related to functions indicated on the block diagram.

- The meter panel contains the various voltmeters and ammeters required to monitor the voltage and current of the simulated

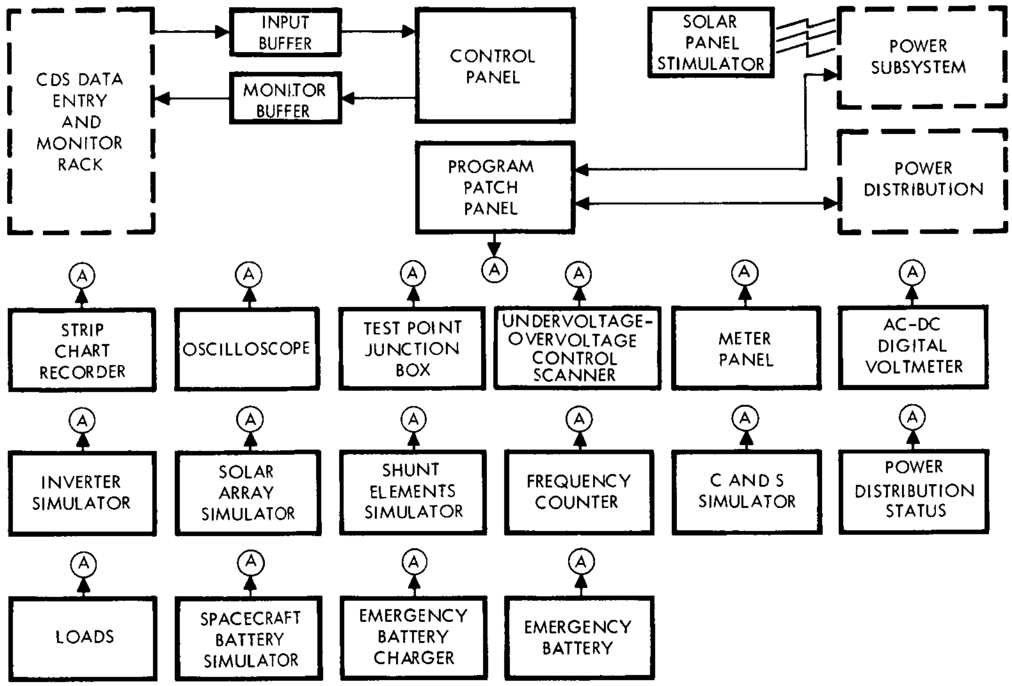


Figure 19. Power Subsystem EOSE Block Diagram

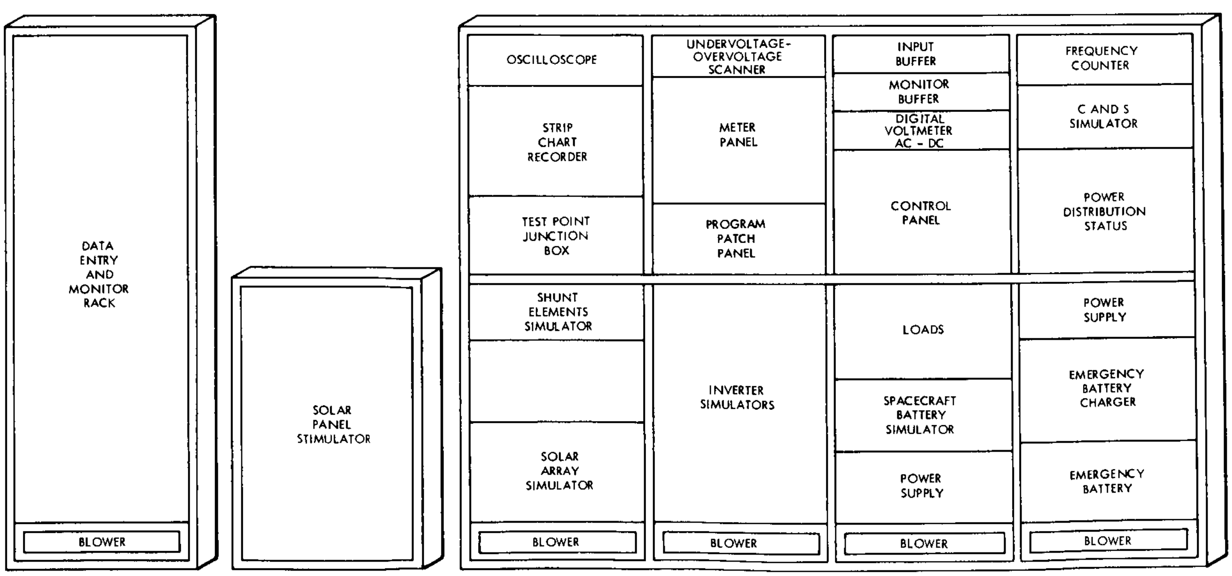


Figure 20. Layout of Power Subsystem EOSE

solar array input to the power subsystem, the output of the inverters and the battery output. Duplicate voltmeters are provided to monitor cell voltage and temperature simultaneously.

- The control panel provides centralized test control of all manual tests. These controls consist of switches which select the applicable tests. Power control for the entire power EOSE is located on the control panel as well as the automatic/manual test mode switch.

- The input buffer panel accepts power subsystem EOSE control inputs and parallel data inputs from the data entry and monitor rack and stores this information in appropriate registers for providing control signals to the control panel.

- The monitor buffer panel converts the information from the digital voltmeter, counter, and the guidance and control subsystem discretes into parallel data outputs to the data entry and monitor rack.

- The load unit contains all of the power loads for the inverters and the battery. The loads are variable to provide complete test capability. The loads simulate the impedances to which the spacecraft power subsystem will be exposed during various modes of operation.

- The solar panel stimulator is a light source used to provide a gross check on the operation of the solar panels. Controls are provided on the mounting fixture which supports the lights.

- The program patch panel consists of relays to provide the connections between the power EOSE and the spacecraft power subsystem. The relays are either controlled by the computer or by manual switches on the panel.

- Shunt element simulators consist of individual isolated power sources which simulate the input impedance, voltage, and current from the solar array which is normally connected to the shunt element assembly in the spacecraft subsystem.

- The C and S simulator provides simulated commands to control the power subsystem in its various modes of operation.

- The power distribution status display indicates the proper operation of the power distribution relays via lights on the panel. This unit also includes the simulated loads for each distribution point.

- The inverter simulators are used to provide the 400 Hz, 2-phase and the 4000 Hz, 1-phase power to the spacecraft in lieu of the spacecraft power subsystem during certain tests. The output of these inverters can be controlled to vary the power application to the spacecraft during marginal tests.

- The battery simulator provides battery power to the spacecraft when the spacecraft battery is not used. Its output is variable over the range of 29 to 42 vdc, to vary the DC power input during marginal tests.

- The emergency battery provides the 50-volt solar array input to the power subsystem in case of facility power failure. The transition to the emergency battery from the solar array simulator power supply is automatic. It allows for an orderly shutdown of the spacecraft power and eliminates the possibility of transients causing deleterious effects on spacecraft circuitry.

- The emergency battery charger is used to insure that the emergency battery is fully charged when not in use.

- The frequency counter is provided to determine that the frequency of the inverter outputs is the same as the clock and sync signal frequencies.

- The AC-DC digital voltmeter is used for analog-digital conversion during automatic operations, being switched to different test points under computer control. It is also used during manual and troubleshooting operations when precise voltage measurements are required. During self-check of the power subsystem EOSE, the digital voltmeter is used to verify EOSE operations.

- The undervoltage-overvoltage control scanner provides the capability of scanning cells in either battery and causes undervoltage and overvoltage alarms to be initiated as well as interruption switches.

- The strip chart recorder is used to provide a permanent record of the battery cell sensing.

- The oscilloscope is provided to monitor signal waveforms during all modes of operation.

- Two separate power supplies are provided for the internal operation of the power EOSE.

- The test point junction box is provided for access to the various monitoring and stimulation points connected to the spacecraft power subsystem. This also provides the capability to monitor critical test points within the EOSE during manual self-tests.

4.2 Computing and Sequencing Subsystem EOSE

The computing and sequencing subsystem EOSE (Figure 21) contains all test circuitry required to test the spacecraft computing and sequencing

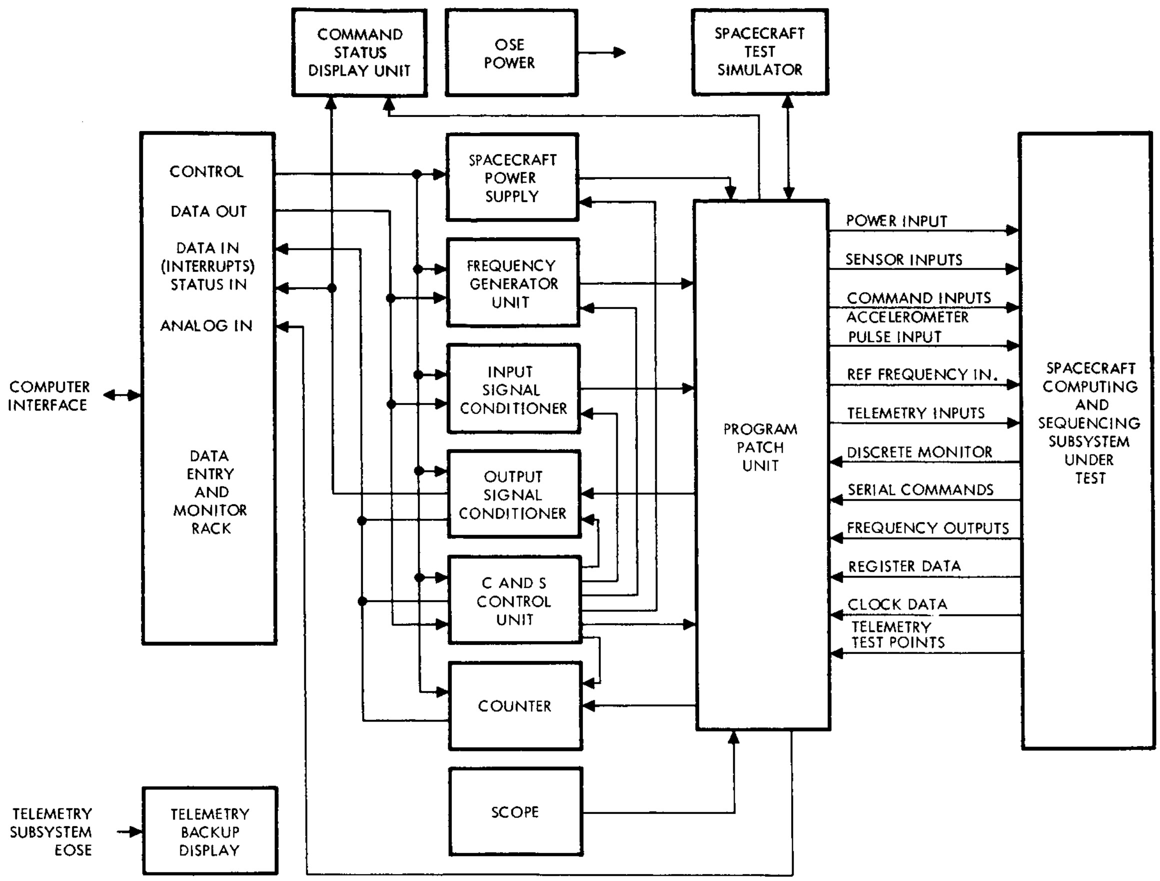


Figure 21. Computing and Sequencing Subsystem EOSE Block Diagram

subsystem from detailed subsystem testing through integrated system testing to the less detailed launch support operational tests. The C and S EOSE can supply all required input simulation signals and monitor all output data signals.

The test set can operate in a completely automatic mode, with the CDS computer providing all control and data reduction functions; in a semiautomatic mode, with some functions manually controlled; or in a decreased capability manual mode, with or without computer connection. Test engineer readout devices allow data presentation selection flexibility in any mode including the computer interface.

The C and S subsystem EOSE operates in conjunction with the CDS data entry and monitor rack in both the subsystem and system level test configurations.

4.2.1 Test Objectives and Design Criteria

a. Test Functions

The computing and sequencing subsystem EOSE performs the following test functions:

Control functions (i.e., generate command input) are accomplished either manually at the C and S subsystem EOSE or automatically by the CDS. Most command actuators operated by the C and S subsystem are not located within the C and S unit. The status of these actuators is monitored by computer correlation techniques operating on all subsystem EOSE test sets.

All data tabulation and recording for permanent records is done at the central data system. Requests for specific tabulation or reduction processes are generated by the subsystem test engineer at the data entry and monitor rack.

Most of the data display functions from the spacecraft and other EOSE are contained on the display panels of the data entry and monitor rack (Figure 22). The status display panel not only displays the computer

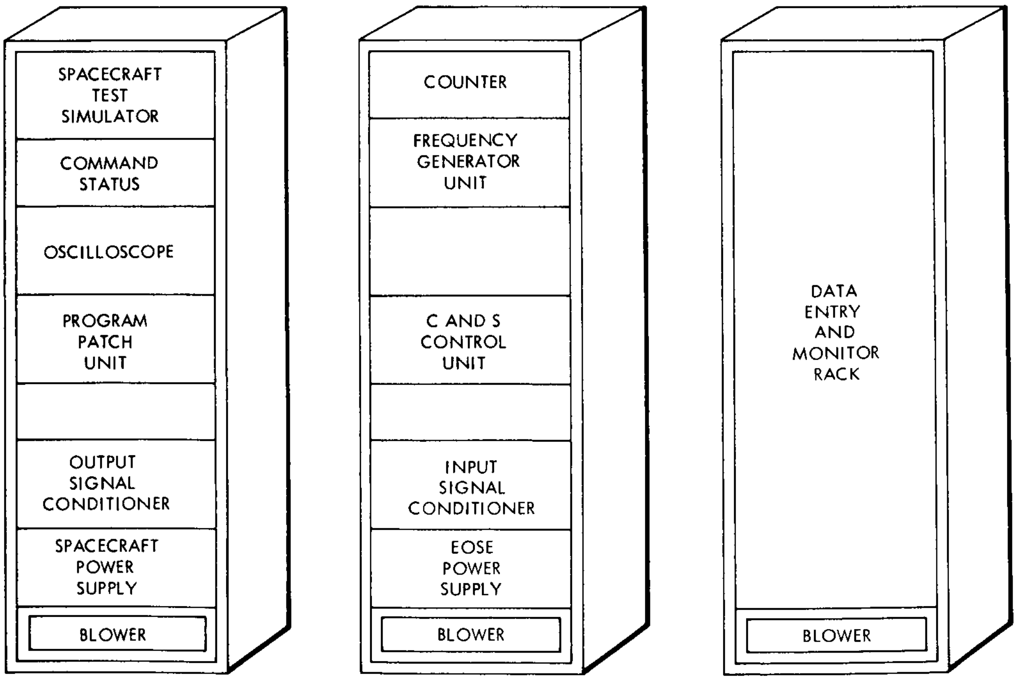


Figure 22. Layout of Computing and Sequencing Subsystem EOSE

discretes, but also has provisions for displaying their direct hardline equivalents. All control functions are displayed on the data storage OSE control panel even if these were actuated automatically.

The selection and display of all memory contents is completely accomplished only in the automatic mode. A single memory location time and mode tag can be displayed manually only when the test configuration allows access to the required direct access connections. The manual selection process is disabled in the prelaunch testing phase. The memory is loaded by a series of computer control commands either applied to the C and S subsystem or the command subsystem (including RF link), depending upon the test configuration. Memory contents are modified by generating the appropriate command sequence either at the C and S subsystem command input channel or at the command subsystem input channel (including RF link) depending on the particular test configuration. A computer-reduced high-speed memory dump can be accomplished in either the subsystem or system test configuration by using the computing and sequencing internal scan operation.

All three time clock registers (master clock, maneuver clock, and orbit timer clock) are continuously monitored by the C and S subsystem EOSE for data transfer to the CDS. This transfer performs two functions: it provides synchronizing data to control the computer generated and distributed time clock data and it is used to test the operation and accuracy of the clock systems themselves. Manual and automatic actuations are included which allow the EOSE to set or reset the various C and S clocks to given conditions when EOSE initial power is applied or when prelaunch condition establishment is required. Time clock (spacecraft master clock, etc.) synchronization routines may not be possible in limited access testing configurations (i.e., prelaunch). In these instances, the computer generates all time clock signals and synchronizes this data with actuations being performed at known mission times.

Serial clock generation is accomplished within the C and S EOSE asynchronously with the computer clock system. Clock start-stop control and rate selection are done either manually at the OSE test set or automatically by the CDS.

Discrete input signal generation (i. e., sensor signals) is accomplished either manually at the C and S subsystem EOSE or automatically by the CDS through the data entry and monitor rack. Manual generation is inhibited during computer controlled test sequences. The signal actuation times are tabulated together with the signal designation at the CDS whether the signal is automatically generated or not. Critically timed discrete signal actuations are accomplished on an interrupt basis.

Discrete output signals (i. e., output commands) are displayed on the C and S EOSE indicators and stored in EOSE flip flops for data transfer to the CDS. In most cases, this data transfer is a normal part of the computer input scan cycle. In other cases, however, where detection timing is critical, interrupt techniques are used to immediately transfer this data to the computer. Each status discrete is logged by the CDS, together with its associated time tag.

Input serial data is generated by the C and S EOSE parallel-to-serial conversion registers, which are clock shifted by the frequency generator outputs. Data can be introduced into the registers only by the CDS, thereby preventing manual command generation. Channel selection, mode selection, and actuation control are accomplished either manually at the EOSE rack or remotely by the CDS.

Serial output is shifted into the serial-to-parallel converter registers by its associated clock input. These registers also serve as a computer buffer and are scanned for parallel data transfer purposes as a part of the normal computer scan cycle. Only one register is required for the serial command output storage and conversion process. Computer interrupt techniques are used for immediate transfer of critical data on a preassigned priority basis.

A spacecraft test simulator is provided which allows the major portions of the computing and sequencing subsystem EOSE to be tested without requiring vehicle interface connections. This simulator will directly replace all vehicle input and output signal lines and will simulate vehicle loads and output characteristics. EOSE self-test fault indications are provided either by test circuitry which provides a display and computer discrete input or by a computer test routine which exercises temporarily off-line EOSE subunits and monitors the output data results.

Programming safeguards are included with the CDS software to prevent damage or forbidden spacecraft actuations resulting from improper sequencing. The process of conditioning the C and S subsystem interface to a safe mode in event of failure and subsequent resumption of facility power is accomplished by the computer-controlled generation of the required command sequences. Manual backup is also provided for establishing gross safe mode conditions in the case of a CDS failure.

b. Interface With Spacecraft

The program patch unit (Figure 21) acts as the major vehicle interface unit and controls the selection of data transfer both into and out of the spacecraft subsystem. The patch unit also provides all necessary hardline test point outputs.

The spacecraft power supply is used to provide the required subsystem power input and monitor point access.

The frequency generator is the main clock generation and synchronization unit within the C and S EOSE, it is used for both spacecraft and OSE signal operations.

The input signal conditioner provides all signal conditioning, serial-parallel conversion and parameter variation functions for spacecraft input signal lines.

The data entry and monitor rack provides the major computer and display interface, consisting of data transfer processing, computer control processing, and display updating.

4.2.2 Operating Characteristics and Constraints

The C and S power supply consists of standard logic and analog supply modules having overload and voltage protection as well as remote sensing capabilities. Primary power controls are included within the supply for subsystem primary power generation and filtering. The spacecraft power supply is a remotely programmable supply with overvoltage protection and remote sensing. The supply is capable of providing voltage variations over the range required for complete parameter variation testing.

The frequency generator unit contains a master crystal-controlled oscillator and various divider chain registers. Automatic and manual output frequency selection is accomplished by providing logical "anding" operations on the counter chains. Clock signal outputs are used both for vehicle input simulation signals and for input and output signal conditioner synchronization.

The input signal conditioner receives digital command and input data in a parallel form, converts these signals to discrete and serial data formats, and conditions the output signals to the parameter requirements of the spacecraft subsystem inputs. Parallel-to-serial converters receive parallel computer data and shift this data out in a serial bit stream at a clock rate determined by the frequency generator unit. Decoded computer commands control the "update converter" operations as well as the "output enable" actuations. All command and discrete signal actuations contain manual entry provisions. Test points are available for C and S EOSE maintenance. Provisions are incorporated which allow the computer to perform a loop test with the input and output signal conditioners. This test consists of applying parallel data to the input signal conditioner, looping the serial output and clock to the output signal conditioner, and supplying the output signal conditioner resultant parallel data back to the computer for comparison with the input.

The output signal conditioner receives spacecraft outputs in a serial or discrete form, changes the signal levels to the standard digital logic levels, converts and stores the information in a computer-compatible parallel word form, and transfers the data to the DE and MR for CDS processing. Serial-to-parallel converters are used to receive the spacecraft data (clocked by spacecraft or EOSE signals) and store it as parallel data for computer output processing. Decoded computer commands control the output data transfer as well as the input enable actuations. Interrupt signals are used for time priority data transfers. Manual operational logic is mechanized by a thumbwheel switch section (storage location selection), a digital comparitor (compares the thumbwheel switch data with the spacecraft memory address register), and a data display and storage register (update to the time and mode tag spacecraft register when comparison enable is generated). Amplifiers and code converters process the register outputs for an octal readout presentation.

The computing and sequencing control unit provides the main controlling functions of the C and S EOSE. This unit receives the DE and MR inputs in a parallel word format, converts the information to single line command voltages, and delivers these signals to the various subsystem EOSE operational units. Each control line command illuminates its appropriate indicator on the control panel. A manual actuation provision is also incorporated on each switch indicator which allows manual control operations when a computer enable condition exists, or when the computer is not connected. The unit output signals control EOSE operational actuations, input and output data transfer, EOSE status data transfer, and program patch unit selection switching.

The command status display unit provides all EOSE monitoring functions for the 256 command discrete outputs. Visual indicators and associated test points for each command output are located on the unit front panel. Pulse shaping and memory circuits are included within this display unit to prolong the command pulse output for visual display and data transfer purposes. Interrupt lines are associated with these signals such that the data can be transferred instantaneously to the CDS for time tagging.

A dual trace oscilloscope is provided within the equipment racks to allow manual monitoring of test and vehicle signal characteristics. A counter (with DVM plug-in) is used to process some vehicle interface signals for computer transfer purposes.

The program patch unit provides the major spacecraft-to-EOSE switching functions. Self-testing operational configurations (spacecraft not connected) are in most cases manually established by means of the patch unit. Hardline test points are contained on the unit panel to allow monitoring of any of the interface signal lines.

The spacecraft simulator is used in self-test operations when vehicle interfaces are not provided. This simulator provides the required spacecraft signal loads and generates many of the spacecraft output signals.

4.2.3 Changes from Task A

Two basic changes in design approach have been taken since the Task A study. Subsystem simulators previously contained within the EOSE have been replaced by computer-generated simulation programs and a higher degree of automation is now provided. These changes simplify the EOSE equipment and increase the over-all testing capabilities. Data display capabilities have been increased, since the computer has access to a greater amount of spacecraft information.

4.3 Guidance and Control Subsystem EOSE

The guidance and control subsystem EOSE performs functional tests on the spacecraft G and C subsystem both before and after spacecraft integration. It is designed to operate in either a manual mode or an automatic mode programmed by an external computer.

4.3.1 Test Objectives and Design Criteria

a. Test Functions

To provide the most comprehensive G and C test possible, the G and C EOSE applies physical stimuli to the G and C sensors wherever practical. The stimuli are designed to perform end-to-end phasing, gross alignment, and gross threshold tests. During system level testing, the spacecraft telemetry subsystem is used wherever possible to minimize the number of hard-line test points required for fault isolation.

b. Performance Requirements

The guidance and control EOSE performs the following functions:

1) Provides both AC and DC power to the spacecraft G and C subsystem

2) Provides various combinations of inputs from the computing and sequencing subsystem, the command subsystem, and sensors to activate the spacecraft guidance and control subsystem's automatic mode switching.

3) Applies stimuli to the spacecraft G and C sensors and monitors the response. The responses are displayed by the G and C EOSE and recorded through the CDS to permit trend analysis. In the manual mode, the test operator evaluates performance; in the automatic mode, the EOSE computer evaluates performance. The following stimuli are applied for phasing, threshold, and alignment evaluation:

- Angular rates and positions to the input axes of the gyros (roll, pitch, yaw).
- Angular rate (torquing signals at system level) to the gyros (roll, pitch, yaw).
- Orient the accelerometer input axes parallel to the earth gravity vector.
- Simulated sunlight to the sun sensors from various fixed angles.
- Star simulator to the Canopus sensor from various fixed positions.
- Mars simulator to the limb-terminator crossing detectors.
- Simulated earth-shine to the earth detector.
- Simulated commands to position the antenna drives.

4) The following G and C outputs will be monitored, displayed, and recorded:

- G and C mode status discretes
- Solenoid valve operation (pressure switches in the pneumatic lines during system level tests; G and C solenoid valve drive signals during subsystem level tests).
- Antenna hinge and shaft position transducers
- Accelerometer pulse repetition rate
- Gyro temperature
- Gyro SMRD signals
- Error signals

- Power supply voltages
- TVC actuator positions

c. Functional Interfaces

The G and C EOSE has four electrical interfaces.

- Facility power is required, conventional 115 volt, 60 cps, single phase power.
- During subsystem testing, the G and C EOSE connects directly to the spacecraft G and C subsystem for all electrical data transfers. During system testing, the G and C EOSE requires a limited number of direct access points to the spacecraft G and C for trouble shooting and fault isolation.
- The G and C EOSE connects to the CDS computer during the automatic mode of operation.
- The G and C EOSE connects to the spacecraft gimbaled structure in order to command various angular rotations during system level tests.

The G and C EOSE mechanical interfaces provide hoods with properly aligned stimuli for the Canopus sensor, earth detector, limb terminator detector, and sun sensor. A mechanical interface is involved in mounting the spacecraft G and C panel on a rotary table or a rate table during subsystem tests.

4.3.2 Operating Characteristics and Contraints

Figure 23 is a block diagram and Figure 24 is a rack layout of the G and C EOSE.

All elements are packaged for mounting in a conventional blower-cooled rack. The various simulators are packaged in separate housings to facilitate their being mounted close to the corresponding spacecraft sensors. The simulators are mounted in a manner which does not cause excessive mechanical loads to be applied to the spacecraft sensors. The rotary table, rate table, and panel mounting fixture are used only during subsystem level testing. The two tables are standard commercial items. The panel mounting fixture is specially designed to attach the G and C equipment mounting panel to the tables during subsystem level tests.

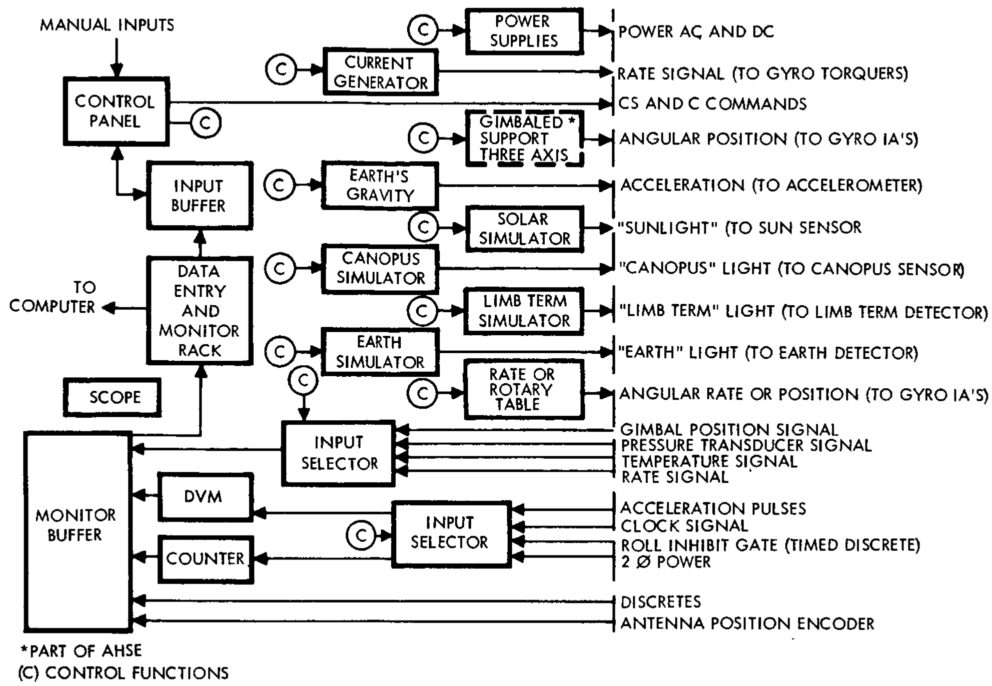


Figure 23. Guidance and Control Subsystem EOSE Block Diagram

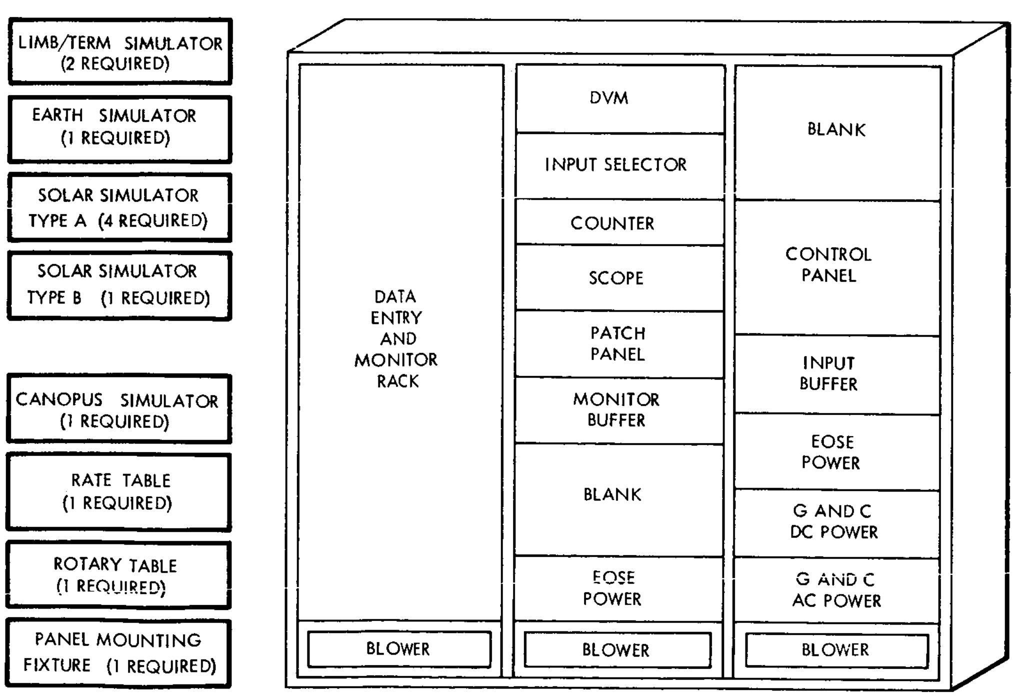


Figure 24. Guidance and Control Subsystem EOSE Rack Layout

The G and C EOSE is designed to operate in both manual and automatic modes. Automatic operation is accomplished by the CDS EOSE, which controls the G and C EOSE and accepts test results from it via the data entry and monitor rack. Devices for data display include status lights, numeric readout, and an oscilloscope to display waveforms for trouble shooting. Data entry and monitor rack displays are also provided in the form of discrete status indications, decimal data displays, and a CRT data display. Data recording is provided by the printer in the EOSE data recording system.

The G and C EOSE contains the following elements:

- The control panel contains appropriate switches, lights, relays, for controlling the operation of the EOSE in the manual mode. A remote/local switch provides for transferring control of the G and C EOSE to the CDS computer. The CDS computer provides inputs to the G and C control panel through the data entry and monitor rack to the input buffer panel.

- The data entry and monitor rack is used to process all data to and from the CDS computer; it contains various displays and data entry provisions. The data entry and monitor rack is described in Section II.3.2.

- The input buffer panel accepts G and C EOSE control inputs and parallel data inputs from the data entry and monitor rack and stores this information in appropriate registers for providing control signals to the control panel.

- The input selector panel contains suitable relays for connecting the input of the digital voltmeter and counter to various monitoring points.

- The monitor buffer panel converts the information from the digital voltmeter, counter, and the guidance and control subsystem discretes into parallel data outputs to the data entry and monitor rack.

- The digital voltmeter performs A-to-D conversions during automatic operation and provides an accurate numeric data display during manual operation.

- The digital counter converts frequency to digital information during automatic operation and provides accurate numeric data display during manual operation.

- The oscilloscope displays various waveforms during trouble-shooting operations.

- The patch panel provides for flexibility in connecting the oscilloscope, digital voltmeter, or counter to various points for trouble shooting.

- The solar simulator consists of two types of units. Type A contains a single lamp; four such units are used for individually stimulating each of the four coarse sun sensors. Type B is used for stimulating the fine sun sensor. Both types of units are designed with light tight hoods which contact the sun sensor housing to provide good control of the light stimulus.

4.4 Radio Subsystem EOSE

The radio subsystem EOSE is used to test and evaluate the performance of the panel-mounted equipment portion of the Voyager spacecraft radio subsystem. The EOSE includes a stimulus and measurement section, an RF section, and bench test accessories and junction box. It is composed of standard commercial test equipment and some special test equipment specifically designed for testing the Voyager radio subsystem. The EOSE is configured so that stimulus and monitor equipment selection and signal routing can be controlled either by the CDS or by a manual control panel. A simplified block diagram of the radio EOSE is shown in Figure 25. The configuration of the rack-mounted equipment is shown in Figure 26. An output is provided for connecting these EOSE racks to an external antenna for radiated RF tests.

Peripheral equipment includes RF bench test accessory items and a junction box for use in individual subsystem test. The data entry and monitor rack provides the test set interface with the CDS and includes status display and permanent recording capabilities.

4.4.1 Test Objectives and Design Criteria

The radio EOSE is required to test the radio subsystem in three distinct modes:

- Independent subsystem checkout, all input-output devices simulated.

- Systems-level test, with the subsystem as part of the operating spacecraft.

- Systems-level test, for individual subsystem checkout.

The integrated radio EOSE provides the following functions:

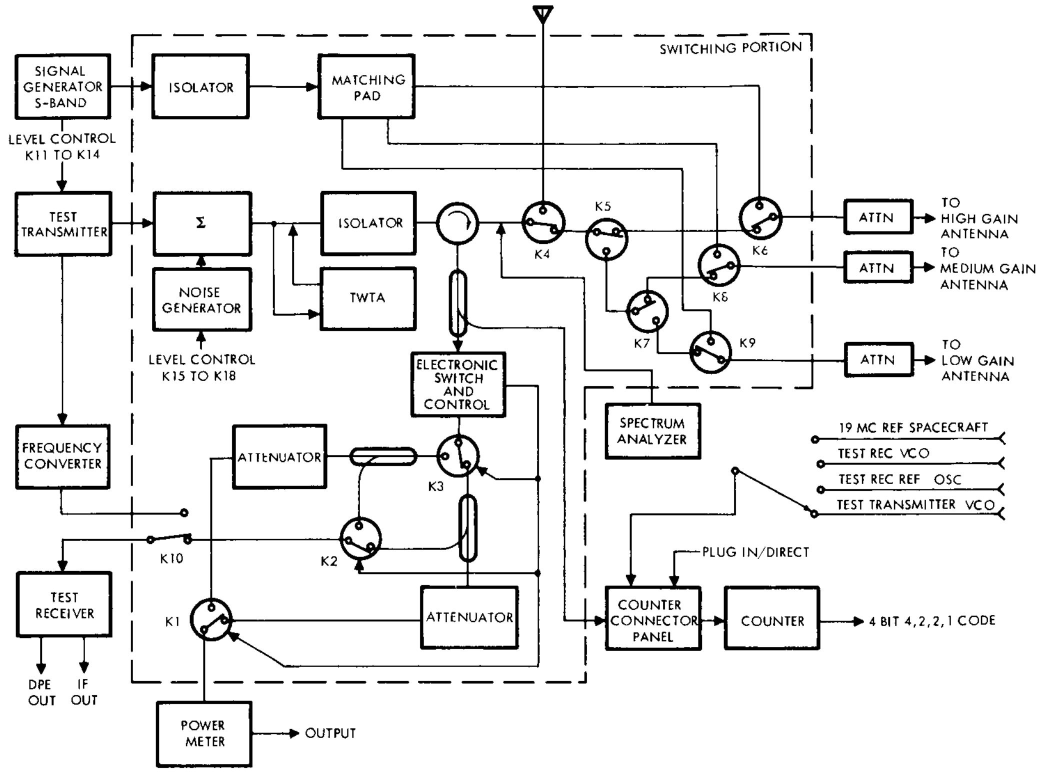


Figure 25a. RF Section of Radio Subsystem Test Set

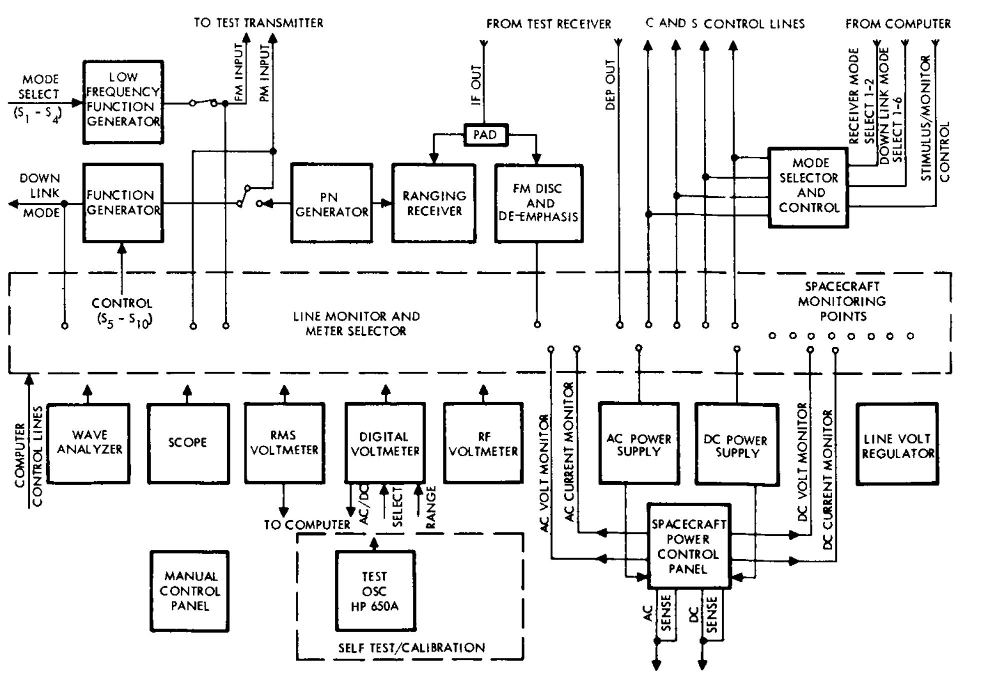


Figure 25b. Measurement and Stimulus Section of Radio Subsystem Test Set

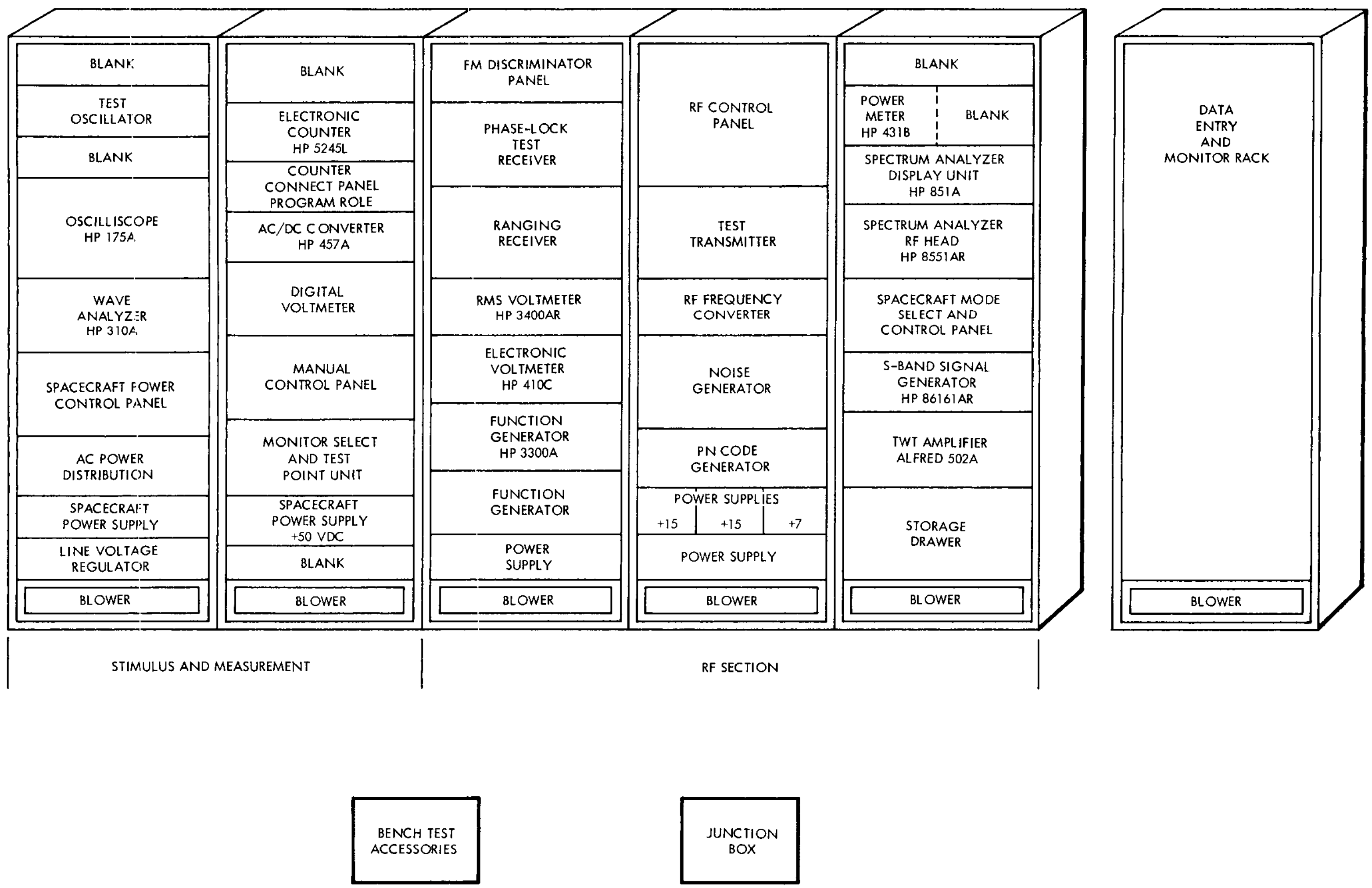


Figure 26. Radio Subsystem Rack Layout

- The uplink RF signal from the test transmitter can be routed to any one of the S-band receivers via hardline as selected by coaxial relays.
- For radiated RF link tests, the test transmitter output is routed to a special test antenna through a TWT power amplifier.
- The test transmitter output level is programmed by external command for sensitivity test of the receivers.
- The noise generator output level is programmed by external command to provide discrete signal-to-noise ratios at the receiver inputs.
- The S-band signal generator output is connected so that an RF signal can be supplied to any one or more of the receivers.
- All RF coaxial relays are connected so that the unswitched side is terminated in a load matched to the line.
- Isolators and a circulator are used to separate the diplexed RF signal and to isolate the test equipment and prime equipment.
- The downlink signal to be monitored by the test set is selected from any one of the three antennas or from the test antenna.
- The downlink signal is routed to the counter, power meter, and test receiver.
- An electronic switch and control is provided for protection of the power meter thermistor mount and test receiver input circuitry. It selects the path attenuation based on the radio subsystem operating power mode.
- Test access is provided for patching the spectrum analyzer with various RF paths.
- Test access is provided for convenient calibration of the attenuation in the RF paths.
- RF closed-loop self-test is provided by the RF frequency converter which converts the test transmitter RF signal to the test receiver frequency.
- Uplink modulation is remotely selectable from the function generator or the PN ranging code generator.
- Downlink modulation stimulus is remotely selectable from the function generator.
- A controlled rate of frequency change of the test transmitter output is provided by the low-frequency function generator.

- Radio subsystem control is provided by the mode selector and control panel which may be controlled by the computer.
- AC and DC power to operate the radio subsystem is supplied and controlled via the spacecraft power control panel.
- The ranging receiver accepts the downlink, range code modulated, IF signal from the test receiver, and correlates it with the uplink ranging code.
- All of the radio subsystem monitor points (except RF), stimulus monitor, and response monitor points are routed to the line monitor and meter selector which is remotely controlled to select any monitor point and route the signal to the appropriate meter.
- The power meter and voltmeter are selected to provide analog outputs which are routed to the digital voltmeter as required.
- The measurement interface with the computer consists of the digital outputs from either the digital voltmeter or from the counter.
- Command data from the command EOSE is routed to the test transmitter phase modulation input during Voyager system test for transmission to the spacecraft.
- The test receiver data output (containing telemetry data from the spacecraft) is routed from radio subsystem EOSE to the telemetry OSE during system test.

In order to perform a rapid evaluation of the operating integrity of the radio subsystem, the test configuration is designed to be basically automatic; manual capability is also provided to permit the performance of more detailed testing, for fault isolation, self-test, and calibration.

A diagram of the interface between the radio subsystem and its EOSE is shown in Figure 27. The radio EOSE interface with other OSE is via the data entry and monitor rack. A diagram of the interface between the radio EOSE and the DE and MR rack is shown in Figure 28.

The radio EOSE operates from standard 115-volt, 60-cycle power source.

4.4.2 Operating Characteristics and Constraints

a. Operation

The phase-lock test receiver provides the link for evaluation of the Voyager S-band transmitter in the end-to-end loop analysis. The

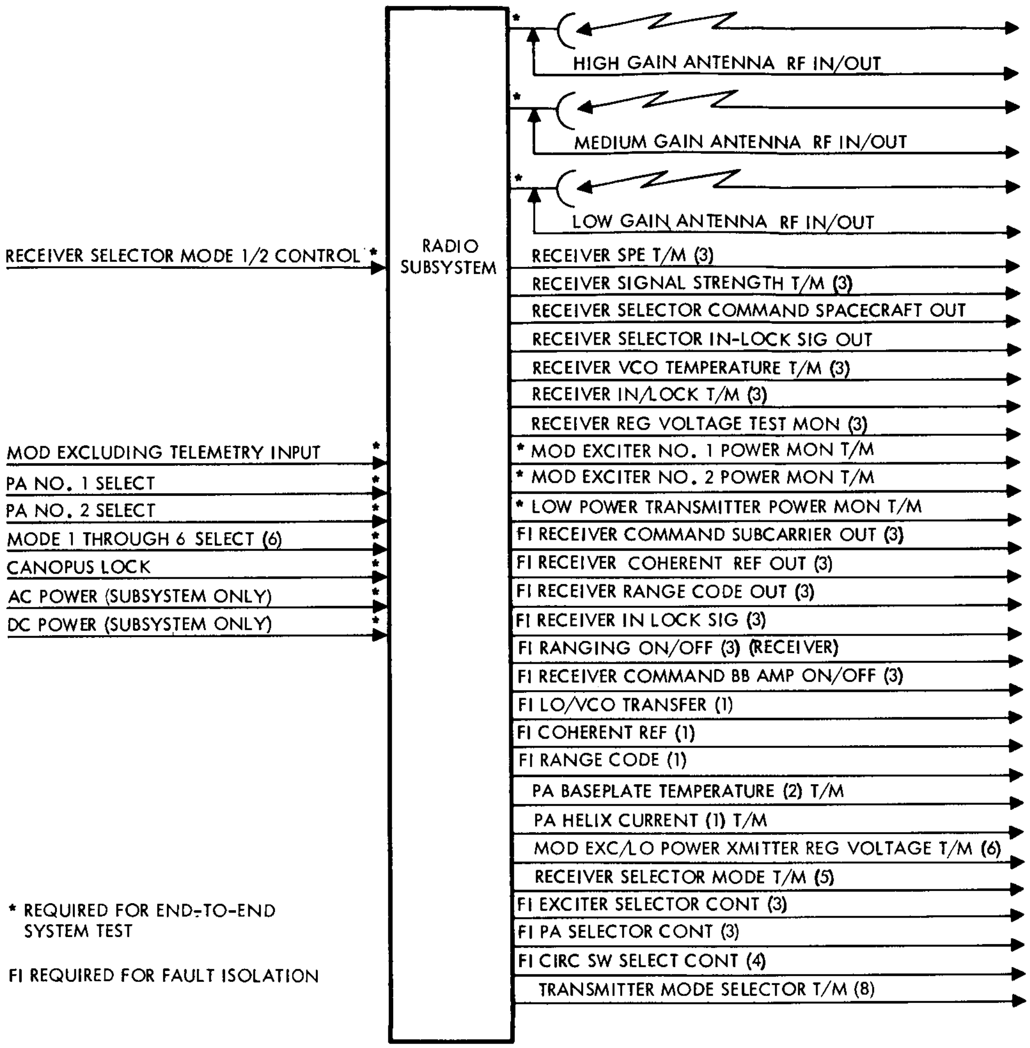


Figure 27. Interface Diagram of Spacecraft Radio Subsystem

receiver, designed to minimize distortion, provides a telemetry data output and an IF output for use by the ranging receiver, and is used in self testing the modulation characteristics of the test transmitter.

The test transmitter provides a stable uplink signal for evaluation of the Voyager S-band receiver in the end-to-end loop analysis with minimized distortion of the transmitted signals. Remote control of the output level is provided for testing the spacecraft receiver dynamic range. The test transmitter provides a phase modulation input for uplink command and ranging and a frequency modulation input for measuring the spacecraft receiver lock-in range.

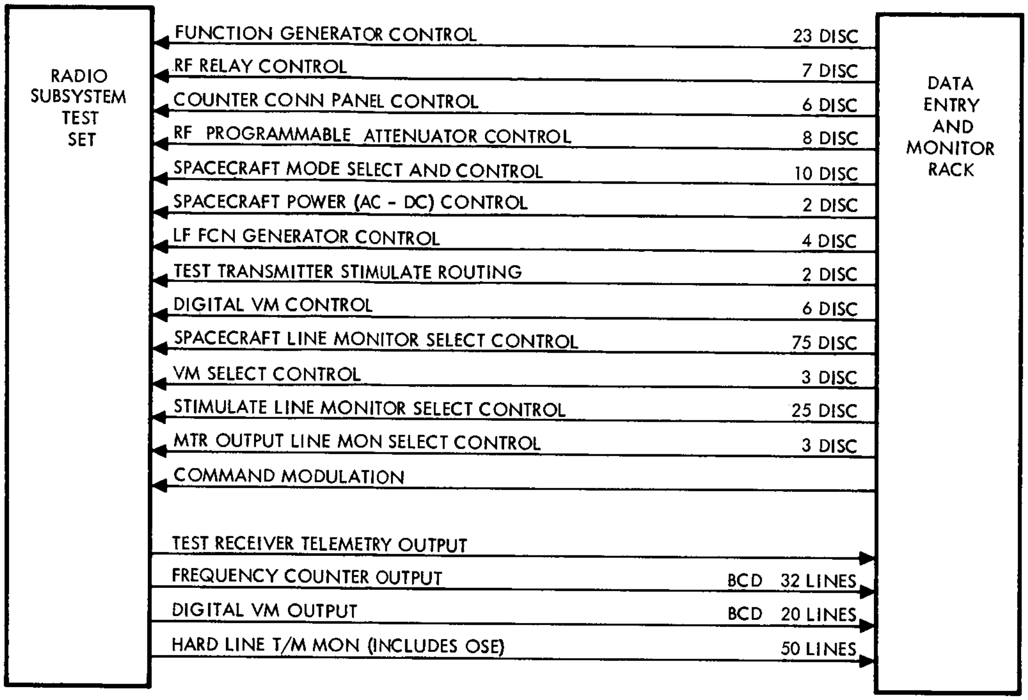


Figure 28. Interface Diagram of Radio Subsystem Test Set and Computer Interface Rack

The PN generator simulates the operational uplink range code modulation for testing the ranging characteristics of the Voyager radio subsystem. The ranging receiver correlates a sample of the uplink range code with the demodulated downlink ranging signal for test of the loop performance of the ranging channel. Correlation results in a measure of the relative ranging delay, i.e., relative to the test set RF closed loop delay.

The frequency converter provides a stable mixing signal for converting the test transmitter output uplink RF signal to the downlink RF frequency for test set closed loop RF test and for zeroing the ranging receiver.

The RF control panel is configured to include the RF hardware required for routing and conditioning the S-band signals in the test set. Signals are routed via relays which can be remotely controlled. The panel includes level-setting attenuators, isolator, circulator, thermistor mount, and protective circuitry to prevent damage to the receiver and the thermistor mount.

The spacecraft mode select and control panel provides the facility for simulating and monitoring the control signals to the radio subsystem normally supplied by other Voyager spacecraft subsystems. The manual control panel provides a switching facility for the test set operator during manual tests, fault isolation, self test, and calibration for selecting relay-controlled signal paths, stimulus, and monitor equipment. The test point panel provides front panel access to test set signals for monitoring during manual tests and for fault isolation of the test set, self test, and calibration. Additionally, the monitor switching and test point panel provides the facility for automatically selecting the line to be monitored and the meter to be used.

The spacecraft power control panel provides the facility for monitoring and controlling the AC and DC power to operate the radio subsystem during individual subsystem test. The 4000 cps power supply supplies the AC power to operate the radio subsystem during individual subsystem test. The AC power distribution panel controls and monitors the distribution of AC primary power within the test set.

The counter connector panel provides the facility for switching the counter input signals and for selecting and controlling direct input to the counter for measuring frequencies below 50 mc or to the plug-in unit for measuring S-band frequencies. The panel operation is controlled remotely.

The FM discriminator panel accepts a modulated IF signal from the test receiver and provides demodulation and de-emphasis. This panel provides the capability for determining the carrier deviation due to command, telemetry data, or ranging modulation. (The test receiver data output does not have sufficient bandwidth for the ranging signal.)

The function generator provides a source of remotely controlled stimuli for test of the uplink command channel, the downlink telemetry channel, and the ranging channel. The junction box provides the electrical and mechanical interface between the radio subsystem flight harness and the test set cables during individual subsystem test. It also provides cable simulation as required. That is, in the various test locations, the interface cables may have different lengths (attenuations). Simplification of computer programming and test set calibration is obtained if the signal levels at the test set can be maintained independent of cable length.

b. Constraints

The radio subsystem EOSE is designed to provide the following input-output signals to the degree of accuracy given.

- The transmitter output power levels of 1 watt and 50 watts are measured with an accuracy of ±0.5 db.
- The output frequencies at the S-band transmitter are measured with an accuracy of one part in 10^7.
- The S-band transmitter deviation sensitivity is measured with an accuracy of ±5 per cent (automatic mode); ±2 per cent (manual mode).
- Modulation linearity of the S-band transmitter is measured with an accuracy of ±3 per cent (automatic); ±1 per cent (manual). Frequency response accuracy, ±5 per cent (automatic); ±3 per cent (manual).
- Intermodulation distortion products resulting from the simultaneous application of two tones to the S-band transmitter input is measured over a dynamic range of 50 db, or better.
- Crosstalk of telemetry data into the ranging channel and ranging into the telemetry data channel is measured over a dynamic range of 50 db, or better.
- The test set measures spurious signals to at least the fifth harmonic of the output frequencies over a dynamic range of 60 db below transponder output power.
- The EOSE measures incidental FM and phase jitter of 36 db (±4 deg) below the transmitter operational deviation of ±4 radians.
- The EOSE exercises all operational and backup modes of the transmitter selector.
- Receiver input signals in the range of at least -141 to -50 dbm are provided with the capability for adding a variable, controlled level of noise.
- Linearity and frequency response of the S-band receiver command and ranging channels are measured with an accuracy of ±3 per cent (automatic); ±2 per cent (manual).

- Two-tone intermodulation distortion is measured over a dynamic range of 50 db, or better.
- The rejection of signals at the frequency of a second planetary vehicle is checked using a variable frequency S-band signal generator with a minimum output level range of -100 to 0 dbm.
- The signal strength which causes the receiver to come into lock is determined with an accuracy of ±1 db. The uplink frequency is measured with an accuracy of ±1 part in 10^7.
- Output frequency is measured with an accuracy of ±3 parts in 10^8. The output amplitude is measured with an accuracy of ±5 per cent.
- The receiver selector switching can be exercised in all of its operational modes.
- The over-all deviation of the S-band transmitter output is measured with an accuracy of ±5 per cent (automatic); ±2 per cent (manual).
- The relative ranging delay is measured with an accuracy of ±6 nanoseconds.
- The AC and DC input power to the S-band subsystem is measured with an accuracy of ±2 per cent.

4.4.3 Changes from Task A

Based on the JPL Task B definition of the system test complex as an integrated combination of the various subsystem EOSE's, a redesign of the radio subsystem EOSE was accomplished to provide for dual operation. Present design provides the necessary configuration changes for integration into the STC and for additional interfaces.

Since the EOSE is capable of testing and monitoring the radio subsystem performance in the subsystem or spacecraft configurations, the use of telemetry data has been optimized and test methods modified, recognizing the limited availability of spacecraft test points in the higher level tests.

Previously, the radio EOSE was automated only to the degree that test results and certain input stimulus measurements were automatically recorded. Under the present configuration the OSE is complemented by the central data system in the three test modes. The test set has now been designed to maximize the use of computer data reduction and input signal stimulation.

4.5 Telemetry Subsystem EOSE

The telemetry subsystem EOSE provides complete subsystem testing capability in both system and subsystem configurations. Testing is performed by simulating the telemetry subsystem input and verifying subsystem operation using the telemetry EOSE. Since the primary output of the spacecraft telemetry subsystem is telemetered data, the telemetry EOSE must demodulate, decommutate, and process the data received from this subsystem. The telemetry EOSE uses the central data system to perform the data processing task and to provide data display on the data entry and monitor rack. The EOSE includes the following major elements:

- Telemetry detector
- Control buffer
- Data format generator
- Telemetry EOSE power supply
- Decommutation and Display Unit
- Printer

A block diagram of the telemetry subsystem EOSE and its functional interfaces with both the spacecraft telemetry subsystem and the central data system is shown in Figure 29.

The telemetry detector demodulates the hardline composite telemetry signal output from the telemetry subsystem, and generates a serial bit stream of reconstructed data. The telemetry detector contains manual selection of one of six telemetry bit rates.

The control buffer serves two functions: to collect and transfer into the central data system computer the reconstructed telemetry data from the telemetry EOSE detector and to generate the simulated inputs required to test the spacecraft telemetry subsystem.

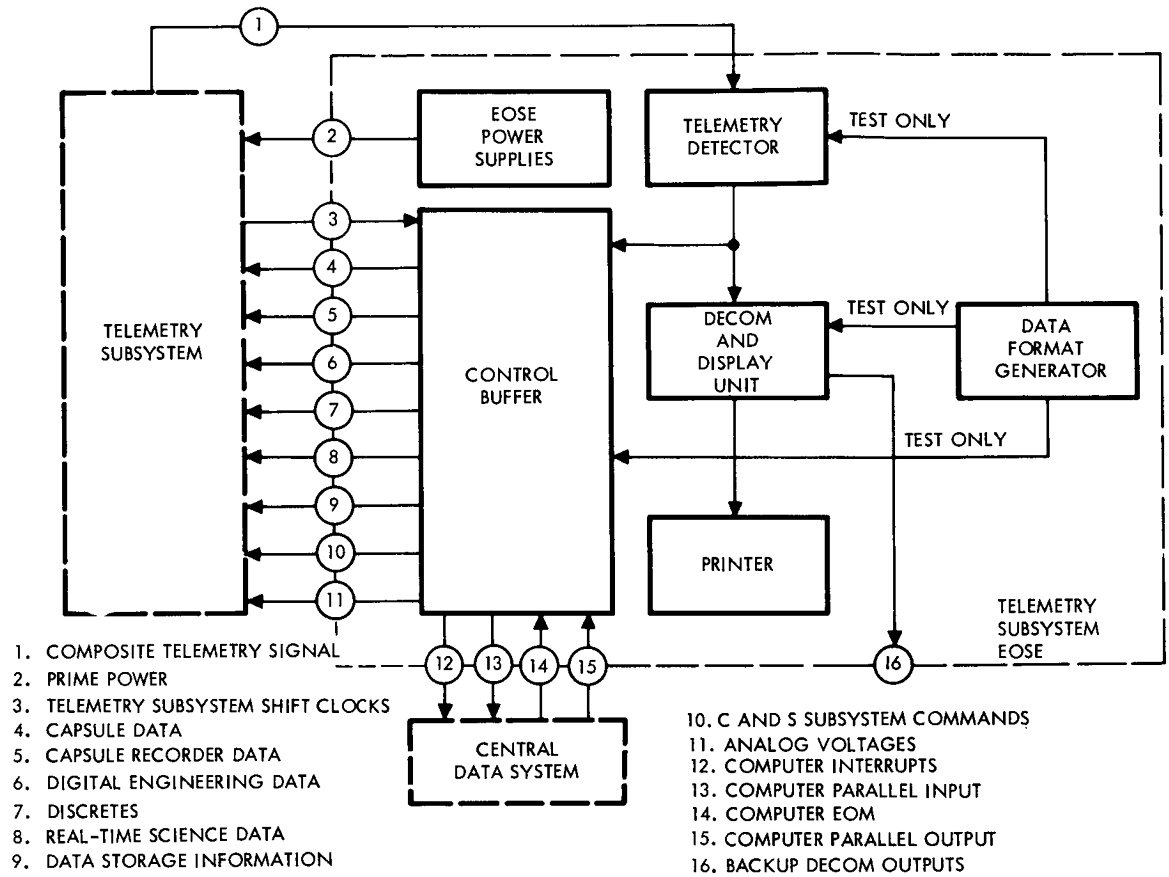


Figure 29. Telemetry Subsystem EOSE Block Diagram

The data format generator simulates the spacecraft telemetry equipment by generating frames of telemetry data in the exact formats of the telemetry subsystem. It is used to determine readiness of the EOSE and central data system to test the telemetry subsystem.

The telemetry EOSE power supplies are used both as the source of DC power for the telemetry EOSE and as the source of prime power for the spacecraft telemetry subsystem under test.

The decommutation and display unit primarily serves as a backup unit when the central data system computer is not available. The unit receives outputs from the telemetry detector and automatically decommutates and formats telemetry words for printing and display. The unit also distributes telemetry words on a common data bus for use in other EOSE.

The printer provides hard copy printout of the telemetry data words. Printing is normally performed with word identification and several telemetry words per column when printing all words. Alternately a single word can be selected for printing and updated as the word arrives on a frame or subframe basis. The selection of the mode of printing is provided on the decommutation and display unit.

The data entry and monitor rack associated with the telemetry test set provides the CDS computer generated display functions. A CRT display and keyboard unit is included for communication with the CDS.

Figure 30 shows a rack layout of the telemetry subsystem, including all of the equipment contained in the telemetry subsystem EOSE for both the automatic decommutation mode and the backup decommutation mode.

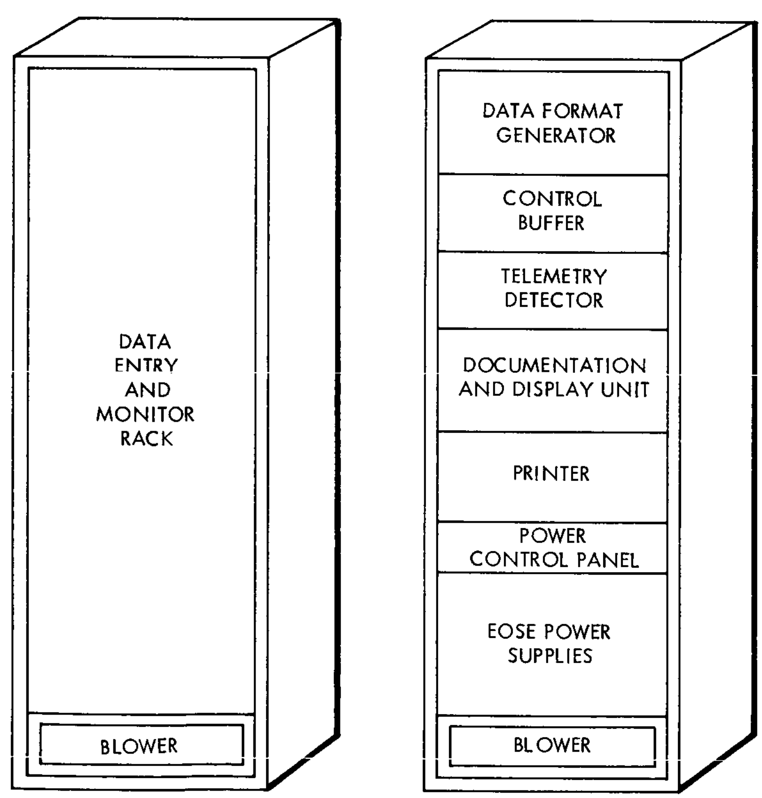


Figure 30. Telemetry Subsystem EOSE Rack Layout

4.5.1 Test Objectives and Design Criteria

a. Test Functions

The telemetry subsystem EOSE simulates all normal telemetry subsystem inputs and demodulates the telemetry subsystem composite telemetry signal output. Extraction of the telemetry data from the composite telemetry signal and comparing the data with known simulated input data is the means of verifying proper spacecraft subsystem operation.

When used with the central data system the telemetry EOSE will demodulate the composite telemetry signal and provide a means of feeding the telemetry data into the central data system computer in word groups. Data processing, display, and recording will be performed by the CDS computer under program control.

Since the telemetry data rate is quite high (approximately 15 kilobits/sec) automatic data processing techniques will be used to provide the processed test results more often than the actual telemetry data. The telemetry EOSE uses the central data system to provide this service.

When the CDS computer is not available, the EOSE has a limited backup capability to decommutate and print all telemetry data words in either octal or decimal form with word identification in decimal. The EOSE also allows for the selection, display, and printing of any single telemetry word with automatic word updating on a frame or subframe basis. The format, mode, and bit rate of the incoming telemetry data is also displayed. In the backup mode the telemetry EOSE provides the following outputs to a common data bus for use in other EOSE subsystems:

- Main frame word number
- Subframe word number
- Sub-subframe (length type 1) word number
- Sub-subframe (length type 2) word number
- Word content (binary and octal)
- Word strobe pulse

b. Performance Requirements

The telemetry EOSE provides test signals to the spacecraft telemetry subsystem and receives and processes signals from the spacecraft telemetry subsystem. The telemetry subsystem EOSE is organized on a functional basis. In most cases the individual equipment functions have been chosen to allow common unit level design with the mission dependent equipment. The telemetry detector and the data format generator are identical to similar equipment in the MDE. The decommutation and display unit is organized to decommutate the serial telemetry data output from the telemetry detector to avoid interfacing with the control buffer. This makes the manual decommutation system independent of the automatic decommutation system.

Telemetry EOSE functional outputs include:

- Simulated C&S subsystem commands. The simulated C and S subsystem commands are six discrete signals which select the mode of operation of the telemetry subsystem.
- Simulated capsule data, both RF and hardline. Simulated capsule data use a hardline connection to the capsule, which supplies asynchronous serial data along with a data clock to the telemetry subsystem, and an RF link from the capsule via an RF receiver, which demodulates the RF data and supplies asynchronous serial data and a data clock to the telemetry subsystem.
- Simulated capsule recorded data. Simulated capsule data is supplied over the hardline normally driven by the capsule via its umbilical. The signal is asynchronous serial data along with a clock. Alternately, this signal can be supplied to the telemetry via the RF link.
- Digital engineering data, serial and discrete. The digital serial engineering data indicates quantitative parameters; the discrete engineering data indicates the on-off status of the elements.
- Real-time science data. Real-time science data consists of serial digital data from the experiments.
- Data storage information. Data storage is binary serial data which was previously derived at a fast rate and stored in the data storage subsystem. The data is subsequently played back at a slower rate into the telemetry subsystem.

- Analog voltages. The analog voltages are signals derived for input to the analog multiplexer of the telemetry subsystem. Approximately 200 simulated inputs are required at levels between 0 and 3 vdc. These inputs are required to test the analog multiplexer and A-D converter.

- CDS computer parallel inputs. The computer parallel input from the EOSE consists of parallel data lines used to transfer binary data into the central data system computer. A line is included in this interface group which signals the EOSE that the computer has completed the data transfer. This interface will generally be used to transfer the reconstructed serial telemetry data output of the telemetry detector into the computer after it has been collected into 14 bit groups.

- CDS computer interrupts. The computer interrupts are generated by the telemetry EOSE to interrupt the computer program operation on a preassigned priority basis. The most common use of this signal in the EOSE is to indicate that telemetry data is available for input to the computer. The computer then enters a subroutine to transfer the data into its memory.

- Backup decommutation outputs. The backup decommutation outputs are those provided on a common data bus for use by other EOSE subsystems. These outputs contain decommutated telemetry data word contents and word identification information.

- Primary power. Primary power to the spacecraft telemetry subsystem and to the telemetry EOSE is provided by a power supply unit in the EOSE.

The telemetry EOSE receives the following inputs:

- Composite spacecraft telemetry signal. The composite telemetry signal is a hardline connection to the spacecraft telemetry data output, which normally feeds the spacecraft radio subsystem. This signal contains time-multiplexed serial telemetry data. This signal is derived in the telemetry subsystem by modulo-two adding the raw serial telemetry data with a square wave data subcarrier and linearly adding the resultant to a PN-coded sync subcarrier.

- CDS computer parallel outputs. The computer parallel outputs are used to transfer binary data or control signals to the EOSE. This set of data lines is accompanied by a strobe signal that can be used to strobe the parallel data into the EOSE.

- CDS computer address outputs. The computer control address outputs are a set of 12 binary data lines used by the computer to specify (address) the particular unit or equipment with which it wishes to communicate.

- Spacecraft telemetry subsystem data clocks. The telemetry subsystem data clocks are bit strobe signals normally fed to the external subsystem, supplying serial binary data to be telemetered. These strobe signals are used to shift the data into the telemetry subsystem.

4.5.2 Operating Characteristics and Constraints

a. Telemetry Detector

The telemetry detector uses coherent code detection techniques in the demodulation of the composite spacecraft telemetry signal. The input to this unit is a hardline connection from the spacecraft telemetry subsystem composite telemetry signal output. The telemetry detector demodulates the signal and provides the reconstructed serial telemetry data as an output. Additionally, the telemetry detector generates both word and bit synchronization pulses and a sync lock discrete signal to indicate its lock or unlock condition.

Figure 31 shows the operation of the unit at a given telemetry bit rate. The blocks marked with the asterisks illustrate those functions

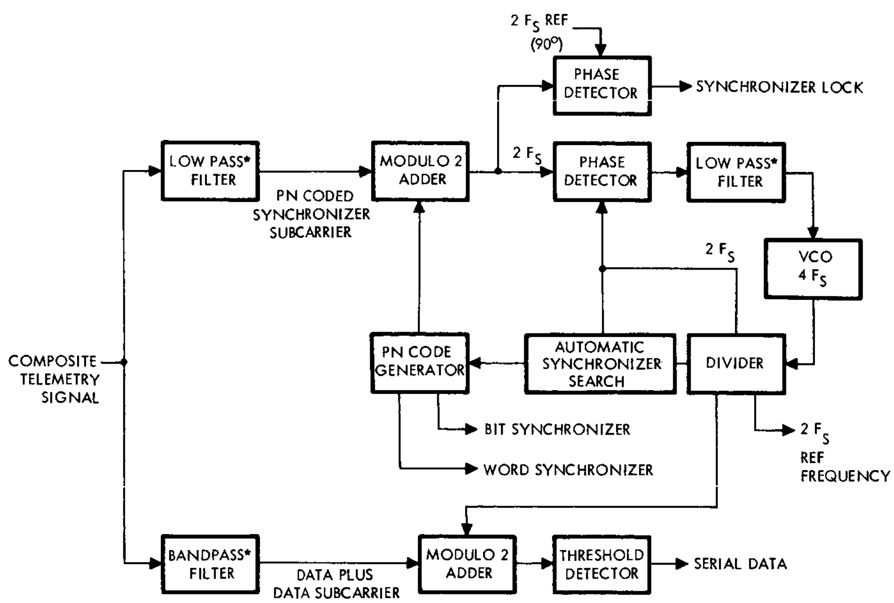


Figure 31. Telemetry Detector Block Diagram

which are frequency-dependent. The frequency-dependent parameters of the blocks are altered when a different telemetry data rate is selected. The unit provides for demodulating the telemetry data at any one of the six Voyager data rates.

The PN code is generated using a six-stage flip-flop shift register with the output stage and the input stage being modulo-two added to form the logic fed into the input stage. This implementation produces the 63-bit repetitive pseudonoise code desired.

The PN code generated in the telemetry detector is precisely synchronized in frequency and bit pattern with that generated in the telemetry subsystem in order to demodulate the data. To allow the spacecraft telemetry detector to lock to the 2 f_s sync subcarrier, a slight frequency offset in the generation of the telemetry detector PN code is allowed until the detector locks to the 2 f_s sync subcarrier. This is provided by the "automatic sync search" circuit in the telemetry EOSE. The logic in the sync search block shifts the telemetry detector PN code incrementally with respect to the PN code generated in the telemetry subsystem until the two codes are in perfect alignment. When the two codes are aligned, the telemetry detector locks to the 2 f_s sync subcarrier; the output of the modulo-two adder is a 2 f_s square wave and the 2 f_s' output of the VCO frequency divider locks to the 2 f_s sync subcarrier. When this lock occurs the automatic sync search logic is disabled.

After the sync subcarrier loop is locked the serial telemetry data can be reconstructed. The data subcarrier is filtered out of the composite telemetry input using a bandpass filter. When the sync subcarrier loop is locked, the 4 f_s' square wave extracted from the VCO divider is synchronously locked to the 4 f_s data subcarrier. By modulo-two adding the 4 f_s square wave with the data-plus-4 f_s data subcarrier, the resultant output is the serial telemetry data. A threshold detector may be used to detect the level of the modulo-two adder output.

Both bit and word sync detection is derived from the PN code generator. The encoding of data is clocked in the telemetry subsystem from seven unique conditions of the PN register. Since the two PN codes are perfectly aligned in frequency and bit pattern in the locked condition, decoding of the PN code shift register states will accurately produce both bit and word synchronization times. By generating pulses each time the register contains the seven unique conditions, the bit sync pulse marks the telemetry bit time. Only one of the seven register states is used to mark the word sync.

The telemetry detector generates a discrete signal which indicates the lock or unlock condition of the sync subcarrier loop. This signal is used by the decommutation equipment as an indicator of either good or bad data output from the telemetry detector. This signal is necessary because both data and sync signals are being issued from the telemetry detector in the unlocked condition. To provide an indication of the stable locked condition of the sync subcarrier loop, a phase detector is used to show the phase relation (and hence lock condition of the loop) between the 2 f_s sync subcarrier and the 2 f_s 90-degree reference output of the divider. When the loop is locked the output of the phase detector will be at its maximum voltage level.

b. Control Buffer

The control buffer performs two functions. It provides a path to and from the central data system computer such that spacecraft telemetry data can be transferred into the computer. It also generates the simulated input signals to be applied to the spacecraft telemetry subsystem.

The control buffer transfers data to the computer via the computer parallel input lines. The information transferred to the computer is telemetry data bits collected in 14-bit groups by the telemetry data shift register (Figure 29). The control buffer provides for selecting either the telemetry detector or the data format generator as the source of input data. The address recognition logic in the control buffer is used to recognize the CDS computer control instructions in specifying the element with which the computer desires to communicate. The address recognition logic reacts to enable gating circuitry which routes the addressed element's output to the computer parallel inputs or enables the computer parallel outputs into the addressed element inputs. It is through the address recognition logic that the computer can turn on either an error light or audible alarm when the CDS computer indicates error conditions. The CDS computer can also generate discrete pulses to the telemetry subsystem event counter via the address recognition logic (control instruction) to simulate a spacecraft event to be counted by the event counter.

Several sets of serial digital data are supplied to the telemetry subsystem as simulated inputs from the control buffer. This data includes digital engineering data, real-time science data, capsule recorder data, and data storage information. These simulated serial data streams are provided by a shift register/counter which supplies successive seven-bit patterns of serial data. In each case the contents of the shift register are incremented after each seven-bit shift to provide varying data intelligence on a per word basis to the telemetry subsystem. In supplying the serial data, known patterns or binary count sequences of data are simulated, rather than having the computer specify the word pattern to be generated for each seven-bit shift. Since the simulated data sequence is known and is different for each input source, the CDS computer can check the telemetry subsystem output to verify that specific word sequences appear in the proper telemetry word slots to verify proper commutation of the data by the telemetry subsystem.

The discrete input signals to the telemetry subsystem are used to report the on-off status of spacecraft elements. Therefore the simulation of these discretes is provided in both the on and off state of each simulated input discrete. To provide discrete signals whose levels change periodically in a known sequence, the parallel output of the digital engineering data shift register-counter is extracted for this purpose. Therefore, the computer can automatically check the states of the status bits reported from the telemetry subsystem with the digital engineering data to verify that the telemetry subsystem can sense both the on and off state of the discretes and commutate the status data in the proper telemetry word slots.

Approximately 200 DC analog voltages are provided as simulated inputs to the telemetry subsystem analog multiplexer from the control buffer. The voltages are spread throughout the 0 to +3 volt range of the A-D converter in several incremental steps to check the resolution of the converter. Approximately eight different voltage levels are simulated with each voltage source, fanning out to approximately 25 inputs to the analog multiplexer. Each voltage source includes fan-out resistors to isolate the input channels of the multiplexer. However, the output resistances used for isolation are sufficiently small as not to degrade the conversion accuracy of the A-D converter.

c. Data Format Generator

The data format generator is used in the telemetry EOSE as a telemetry data simulator providing data in the same modes and formats as the telemetry system. The unit contains front panel controls to select the desired telemetry data format, mode, and bit rate.

It provides two functional simulated outputs. One is identical to the spacecraft telemetry subsystem composite telemetry signal. This signal consists of a 4 f_s square wave data subcarrier modulo-two added to the raw serial telemetry data with the resultant signal linearly added to the PN-coded sync subcarrier. Figure 29 shows the means by which the composite signal is derived. The other functional output consists of serial telemetry data accompanied by bit and word sync pulses and a switch-controlled sync lock signal. These outputs can be used by either the control buffer, decommutation and display unit, or the error rate tester (MDE). These signals simulate the output of the telemetry detector and can be used to check the telemetry EOSE and CDS computer programs or for isolating system malfunctions. When used by the error rate tester, this data can check the telemetry detector output data when the simulated telemetry output of the data format generator is mixed with noise and fed into the telemetry detector to test its error rate.

Serial telemetry data is generated in the data format generator by the data commutator, which serially samples the contents of the main frame word counter, the subframe counter and the two sub-subframe counters and issues the sampled data in seven-bit serial words. Each serial word contains data relating its word number in the main frame or subframe in binary form. Alternately the data commutator can sense the contents of a switch register and generate serial words containing the switch register data in all words except format, mode, identification, and frame synchronization words.

4.6 Command Subsystem EOSE

The command subsystem EOSE permits end-to-end testing of the spacecraft command subsystem in subsystem and system test level configurations. This end-to-end testing is implemented by simulating the normal inputs to the command subsystem and verifying its outputs with the command EOSE operating in conjunction with the central data system. The testing is automatic when the command EOSE is used with the CDS, which provides automatic input simulation and output verification. The command EOSE includes a command encoder, output buffer, frequency counter, and a power supply (AC) for the spacecraft command subsystem. Figure 32 schematically describes the flow of command logic signals. Figure 33 illustrates the command subsystem EOSE rack configuration. The end-to-end testing discussed here is supplemented with subunit test capability to provide increased confidence in the satisfactory status of system functioning.

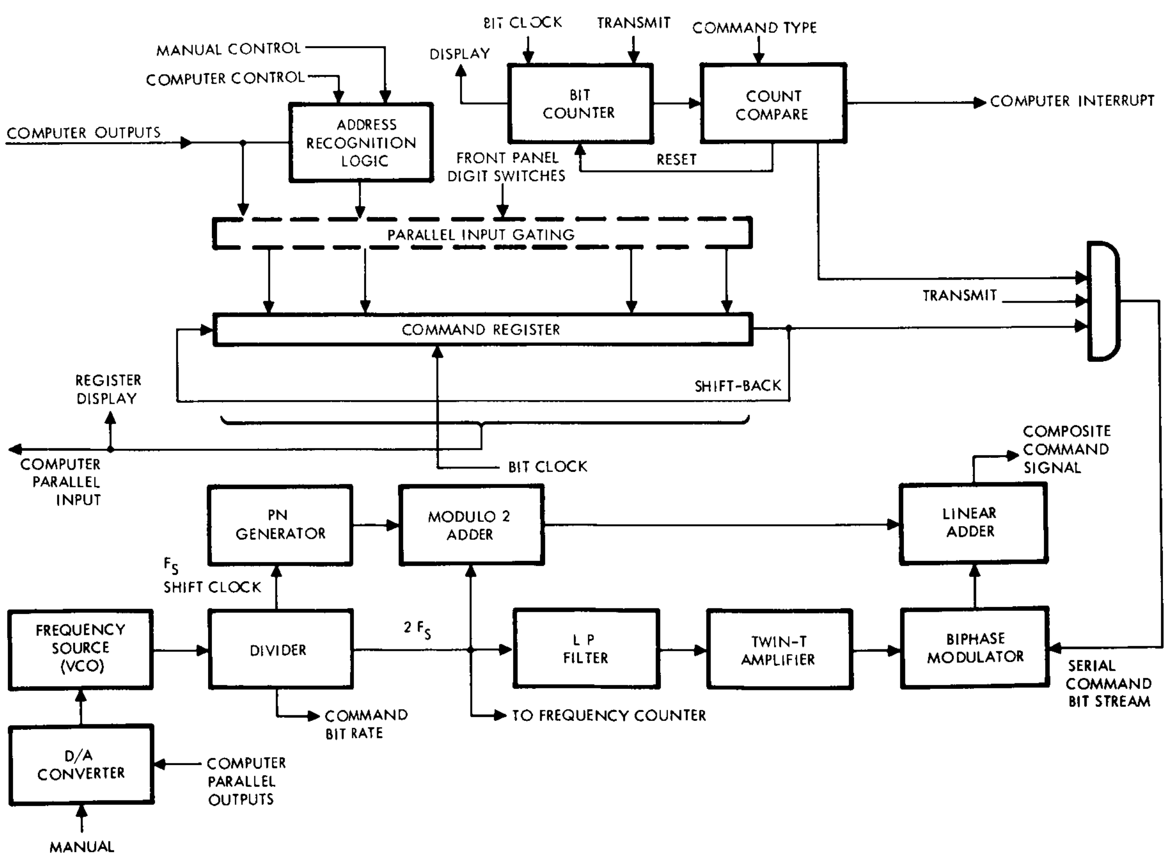


Figure 32. Command Generation Block Diagram

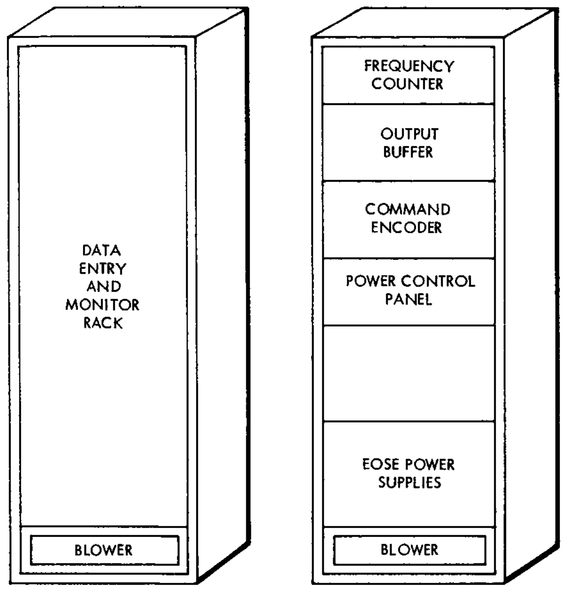


Figure 33. Command Subsystem EOSE Rack Layout

4.6.1 Test Objectives and Design Criteria

a. Test Functions

Flexibility is provided in the selection of command mode, command content, and content verification prior to command generation and command initiation. In the automatic command mode, commands from the central data system are initiated, sequenced, and controlled automatically. The manual command mode permits spacecraft command subsystem testing when the central data system is not available.

During normal spacecraft operation, the telemetry subsystem provides telemetered data to the ground indicating the deviation of the command subsystem $2f_s$ frequency. The command EOSE provides a means of offsetting its $2f_s$ frequency in approximately 0.2-cps increments. To perform the PN code synchronization process, it is necessary to offset the command EOSE $2f_s$ frequency slightly to create a PN code generation timing mismatch; this allows the spacecraft command subsystem to synchronize itself with, and lock to, the DSIF $2f_s$ frequency. Once the spacecraft command subsystem is locked to the command EOSE $2f_s$ frequency, the $2f_s$ frequency can be returned incrementally to nominal. Since the $2f_s$ frequency signal is not directly available operationally, the time between two successive bit sync pulses, as measured by the number of standard timing pulses occurring in that period, is used as a measure of the command subsystem $2f_s$ frequency. The same technique will be used in providing this measurement in the EOSE.

The command EOSE verifies spacecraft command subsystem responses to simulated commands in the manual or automatic control mode.

b. Output Signals

The composite command signal simulates the normal output of the spacecraft radio subsystem to the spacecraft command subsystem. When commanding the spacecraft command subsystem, the data subcarrier, a $2f_s$ sine wave signal, is biphase modulated by the serial command bit stream.

The CDS computer input interface consists of parallel data lines that are used to transfer binary data into the central data system computer. A line is included in this interface group which signals the command EOSE that the computer has completed the parallel data transfer.

The computer interrupt signals are generated by the command EOSE and are used to interrupt the computer program operation on an assigned priority basis. The most common use of computer interrupts in this arrangement is to indicate to the computer that pertinent data is ready for transfer from the command EOSE to the computer. The CDS computer program then automatically enters a subroutine to transfer the available data into its memory.

Primary power for the spacecraft command subsystem is supplied by the command EOSE during subsystem testing. This power interface consists of 4000 cps AC power at approximately 13 watts.

c. Input Signals

The major spacecraft command subsystem outputs to telemetry consist of the command word bits, the detector lock signal, the composite enable signal, the reset signal, and bit sync pulses. The first four of these signals are outputs from a telemetry buffer in the spacecraft decoder; these signals allow the telemetry subsystem to indicate to the ground that a command has been received and properly processed by the command subsystem. During subsystem testing, the command EOSE collects the decoder telemetry buffer output data and transfers this information into the central data system, where it is automatically compared with the selected command. Two such sets of outputs are supplied in the command subsystem because of the redundant command detectors.

Delayed commands to the spacecraft computing and sequencing subsystem, delayed commands to the science subsystem sequencer, direct commands to the science subsystem decoder, and direct serial orders are all outputs from the spacecraft command subsystem input decoder. Each set is duplicated by the redundant input decoder and each set consists of a serial data bit stream, bit sync pulses, plus an enable signal. A direct serial order is a series of data bits each accompanied by a bit sync pulse and a final enable signal. An enable signal indicates that the command subsystem recognized and processed the command as valid.

The direct commands to the controlled elements consist of direct pulse orders. A direct pulse order is a discrete pulse on one of approximately 256 lines. The width of the pulse is approximately 0.5 second.

4.6.2 Operating Characteristics and Constraints

The command encoder generates spacecraft commands of the desired formats to simulate the spacecraft command detector input normally received from the spacecraft radio subsystem. This simulated input is the composite command signal consisting of a PN-modulated sync subcarrier linearly added to the data subcarrier which has been biphase-modulated by the serial command data. The selection of command type and contents can be specified automatically from the computer under program control or by digit switches on the command EOSE's front panel. After the command has been selected and properly verified, it is automatically generated and issued to the spacecraft command subsystem by the command encoder.

The output buffer is an integral part of the command EOSE. It collects spacecraft command subsystem outputs for evaluation. Additionally, the output buffer provides for measuring the deviation of the command subsystem $2f_s$ command data subcarrier frequency in a manner similar to that used by the telemetry subsystem.

The output buffer collects and temporarily stores the serial spacecraft command subsystem data outputs. These outputs are transferred into the central data system for automatic verification and displayed locally for manual evaluation.

The data entry and monitor rack associated with the CDS computer provides computer-generated displays of status, time, and selectable spacecraft information. A TV display and keyboard unit is the main communication device between the test engineer and the CDS.

The command encoder allows for changing the $2f_s$ frequency by varying the frequency of its voltage controlled oscillator. The VCO is driven from a digital to analog converter. The D-A converter output voltage determines the frequency of the signal generated by the VCO and is chosen to obtain the optimum frequency control deviation over a prescribed linear voltage control level compatible with available D-A converters. The VCO output is shaped and counted down to obtain the $2f_s$ sync subcarrier. The exact number of countdown stages used depends on the VCO frequency selected.

The $2f_s$ sync subcarrier square wave is generated by and extracted from the digital countdown divider. By generating both the PN shift clock and the $2f_s$ square wave sync subcarrier from the same divider, the two rates are guaranteed to be synchronous.

The $2f_s$ subcarrier signal is converted to a sinusoid using a low-pass filter and a twin-T feedback amplifier. The twin-T feedback amplifier is tuned to the $2f_s$ frequency and highly attenuates all unwanted frequencies so as to produce a clean, synchronous $2f_s$ sine wave. Although it is desirable for the bandpass of the twin-T feedback amplifier to be as narrow as possible, it must be wide enough to pass the upper and lower limits of the $2f_s$ frequency when it is varied during the PN code synchronization process.

The output buffer employs a similar technique to indicate the $2f_s$ frequency deviation. This unit uses a four-decade binary-coded decimal counter to count the number of pulses between two successive command subsystem bit sync pulses. This measurement can be initiated either manually or automatically and used to indicate where to set the command encoder $2f_s$ frequency during the PN code synchronization process. The EOSE pulse source is supplied in the output buffer. Once the measurement has been initiated the counter automatically starts counting on the next received bit sync pulse and stops on the following bit sync pulse. When the counter is stopped, a computer interrupt signal is generated to indicate to the computer that the results are available.

The output buffer verifies the input decoder outputs by collecting the serial data in a shift register. After the data has been collected in the shift register, its contents are displayed locally and are immediately available to the computer for automatic data comparison.

The command encoder contains a 43-bit command register which can be loaded with command data locally from front panel digit switches or automatically from the central data system. The front panel manual command selection uses digit-switches for each of the command content sections. The digit switch information is loaded directly into the command register through parallel gating.

When the spacecraft command subsystem is decoding a received command, the detected bit command stream and the detector lock signal are provided as outputs to telemetry. To verify these outputs the spacecraft bit sync is used to shift the serial command data into a 43-bit shift register in the output buffer.

The contents of the shift register are displayed locally for manual verification and are transmitted to the computer parallel input lines along with the detector lock signal. The CDS computer receives the command detector output and compares this data with the command data transmitted to the spacecraft command subsystem via the command encoder. The bit sync pulses are also used to generate computer interrupts after a command to the spacecraft command subsystem has been initiated.

Command generation and issuance is started with either a manual or automatic "transmit" signal. The "transmit" signal enables a command "bit time" counter and the serial shifting of the command register. The register output is then gated into the biphase modulator as the modulating signal, causing a phase reversal in the $2f_s$ sinewave data subcarrier for each binary "one" bit in the command bit stream. The bit time counter counts the command bit periods and drives a count comparator which determines the end of the command by comparing the bit counter content with the command type selected. The comparator disables the command register output to the biphase modulator when the proper number of bits have been issued. Meanwhile the output of the command register is shifted back into the command register such that it should contain the same data after the command has been issued as a gross check on register operation.

The command encoder provides for command selection and command generation checking by the central data system computer. The contents of the command register are made available as parallel data to the computer. Following an automatic command selection by the computer, it can address the command register and read its contents as a check to insure that command data was properly loaded into the command register during the automatic command selection cycle. The command encoder also supplies to the computer the serial command bit stream and an interrupt during each command bit time. The serial command bit stream is the same data that is used to modulate the $2f_s$ data subcarrier. Therefore, on receipt of the bit time interrupt, the computer can sense the command bit as a check on the command generation process.

4.7 Data Storage Subsystem EOSE

The data storage subsystem EOSE contains all test circuitry required to test the spacecraft data storage subsystem from subsystem testing through integrated system testing to the less detailed prelaunch testing. The data storage EOSE test set is capable of supplying all required input simulation signals and monitoring all output data signals. A spacecraft test simulator is also provided for pre- and post-test EOSE self-testing.

The data storage EOSE can operate in a completely automatic mode with the computer providing all control and data reduction functions; in a semi-automatic mode, with some control and simulation functions manually controlled; or in a decreased capability manual mode with or without the computer interface. Test engineer readout devices allow complete data presentation flexibility in any mode containing the computer interface.

4.7.1 Test Objectives and Design Criteria

The data storage subsystem EOSE meets the following functional test requirements:

- Provides required subsystem power input.
- Provides synchronizing clock inputs for science and telemetry input data trains.
- Provides simulated science and telemetry input data trains.
- Provides "write" and "read" start-stop command input signals.
- Monitors read and write end of tape signal outputs.
- Monitors "read" telemetry data outputs.
- Monitors "read" telemetry clock outputs.
- Monitors "read" gap signal outputs.
- Monitors telemetry test point outputs.

Control functions (i.e., clock frequency selection and actuation control) are accomplished either manually at the EOSE subsystem test set or automatically by the CDS.

All permanent record data tabulation and recording is done at the central data system. Requests for specific tabulation or reduction processes are generated by the subsystem test engineer at the data entry and monitor rack. Most of the spacecraft and EOSE data display functions are on the display panels of the data entry and monitor racks. The status display panel displays the computer-reduced discretes and has provisions for displaying their direct hardline equivalents. All control functions are displayed on the data storage EOSE control panel during automatic and manual test modes.

Data storage EOSE self-test fault indication is provided either by EOSE circuitry which provides a display and computer discrete input or by a computer test routine which exercises temporarily off-line EOSE subunits and monitors the output data results.

Bit error measurements are performed in the CDS through a bit-by-bit comparison of the playback data with the recorded data.

A spacecraft simulator is provided which allows the major portions of the data storage subsystem EOSE to be tested without requiring vehicle interface connections. This simulator replaces all vehicle input and output signal lines and simulates vehicle loads and output characteristics.

4.7.2 Operating Characteristics and Constraints

a. Operation

The program patch unit (Figure 34) acts as the major interface unit and controls the selection of data transfer both into and out of the spacecraft subsystem. The patch unit also provides all hardline test point outputs. The spacecraft power supply is used to provide the required subsystem power input and monitor point access. The clock rate generator is the main clock generation and synchronization unit within the data storage EOSE. All asynchronous clock signals for both input and output data generation and reduction are provided by the clock rate generator. The input signal conditioner provides all signal conditioning, conversion, and parameter variation functions for spacecraft input signal lines. The output signal conditioner receives and conditions all spacecraft output signal lines. The control unit provides all of the control interfaces for both computer and manual operational configurations. A counter is

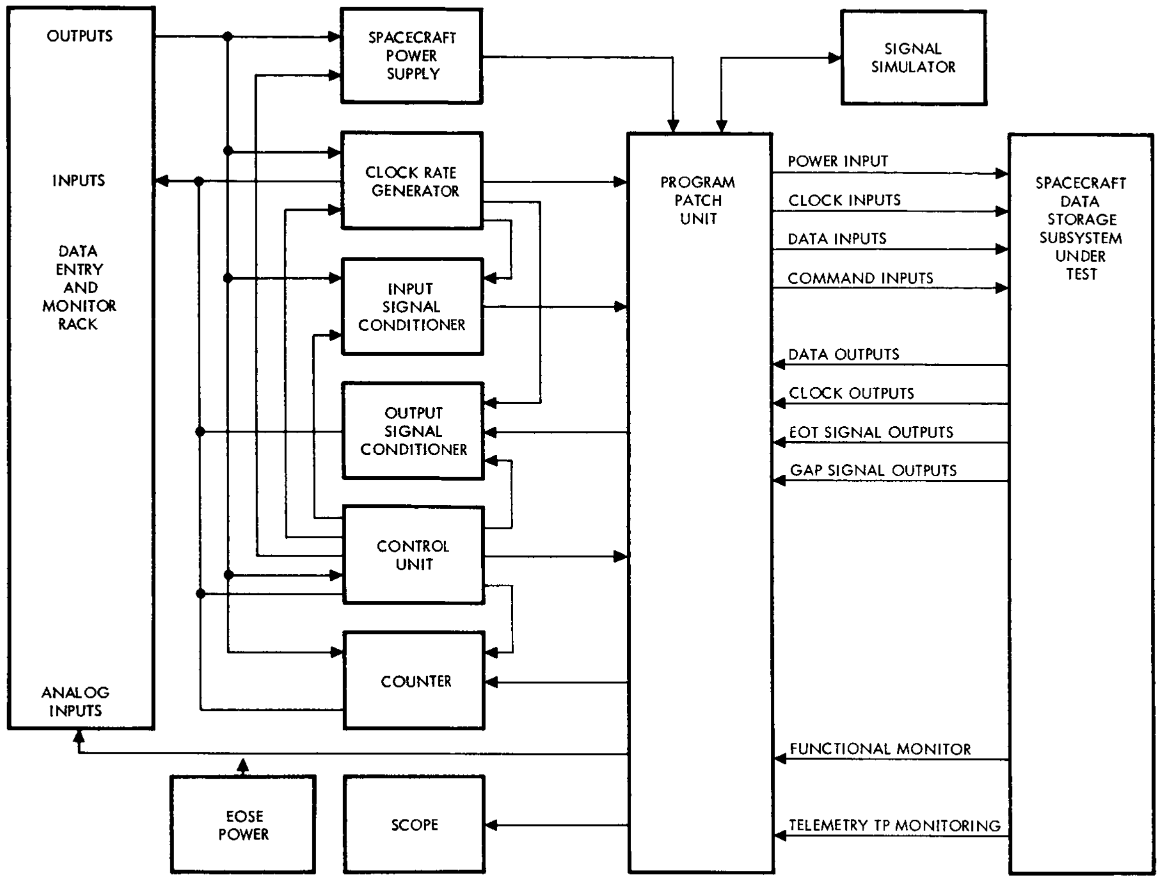


Figure 34. Data Storage Subsystem Test Set Block Diagram

included for frequency measurement purposes and operates in both the manual and automatic operational modes. An oscilloscope is provided for manual signal monitoring purposes. The data entry and monitor rack provides the major computer and man-machine interface. Most displays are located within the data entry and monitor rack. Figure 35 illustrates the rack layout of the data storage test set.

b. Configuration

The data storage EOSE power supply consists of standard supply modules having overload and voltage protection as well as remote sensing capabilities. The spacecraft power supply is a remotely programmable supply with overvoltage protection and remote sensing. The supply is capable of providing voltage variations over the range required for complete parameter variation testing.

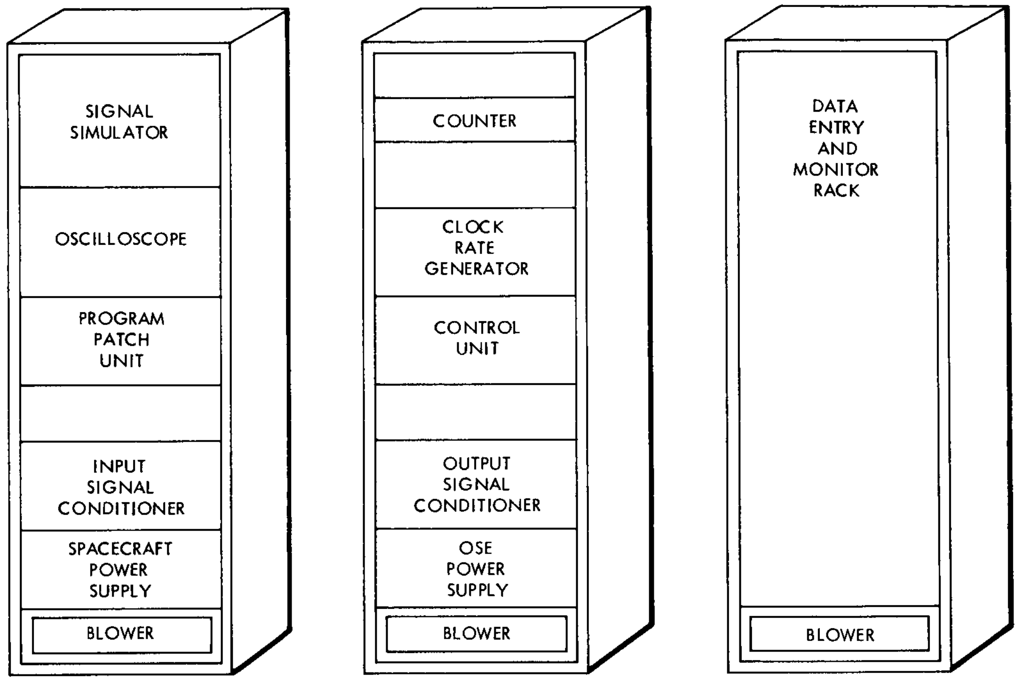


Figure 35. Data Storage Subsystem Test Set Rack Layout

The clock generator unit contains a master crystal-controlled oscillator and various divider chain registers. Automatic and manual output frequency selection is accomplished by providing logical "anding" operations on the counter chains. Two sets of clock signals are provided for each output frequency; one clock signal is used as the spacecraft input, while the other is used to synchronize the input signal conditioner. Gross input and output clock selection is accomplished by the program patch unit.

The input signal conditioner receives digital command and input data in a parallel form, converts these signals to discrete and serial data formats, and conditions the output signals to the parameter requirements of the spacecraft subsystem inputs. Parallel-to-serial converters receive parallel computer data and shift this data out in a serial bit stream at a clock rate determined by the clock rate generator unit. Decoded computer commands control the update and converter switch operations as well as the output enable actuations. All command and discrete signal actuations contain manual entry provisions when enabled by the CDS or in the absence of the CDS. Test points are available for EOSE maintenance. Self-test circuitry is included within the unit to provide discrete status indications to the CDS. Provisions are also incorporated which allow the computer to perform a loop test operation with the input and output signal conditioners. This test consists of applying parallel data to the input signal conditioner, looping the serial output and clock to the output signal conditioners, and sending the output signal conditioner resultant data back to the computer for comparison with the input.

The output signal conditioner receives spacecraft outputs in a serial or discrete form, changes the signal levels to the standard digital logic levels, converts and stores the information in a computer-compatible parallel word form, and transfers the data to the CDS upon computer command. Decoded computer commands control the output transfer and converter "switch" operations, as well as the input enable actuations. Discrete signal processing consists of storing the output information in a flip-flop for computer scan data transfer operations.

The data storage control unit provides the main controlling functions of the data storage EOSE test set. The unit receives the CDS inputs, converts these inputs to single line command voltages, and delivers these command signals to the other sections of the data storage EOSE. Each control line command illuminates its appropriate indicator on the control panel for operator observation. Manual actuators are also incorporated on each switch indicator to allow manual control when a computer enable condition exists or when the computer is not connected. The data storage control unit output signals control EOSE operational actuations, input and output data transfer, EOSE status data transfer, and program patch unit selection switching.

The program patch unit provides the major spacecraft-to-EOSE switching functions. EOSE data and clock rate signals are supplied to the proper recorder input while output data and clock signals are received from the proper recorder output due to the selection switching accomplished within the program patch unit. Self-testing operational configurations (spacecraft not connected) are in most cases established with the use of the patch unit. Hardline test points are contained on the unit panel to allow manual monitoring of any of the interface signal lines.

A dual trace oscilloscope is provided within the data storage EOSE equipment racks to allow manual monitoring of test and vehicle signal characteristics.

A counter (with DVM plug-in) is used to process vehicle interface signals for computer transfer purposes. In the automatic mode, the counter-DVM input leads are automatically switched to the required monitor points by the program patch unit. Parallel outputs are entered into the standard data transfer hardware by commands from the control unit.

4.7.3 Changes From Task A

The basic changes in design since Task A consist of a subsystem rather than unit testing philosophy and a higher degree of automation. Most of the spacecraft subsystem simulation devices previously contained within the data storage EOSE have been replaced by computer-generated simulation programs. This process not only simplifies the EOSE design (i.e., no storage is now required within the data storage EOSE for record-playback comparison), but it also increases the over-all testing capabilities (i.e., the computer is used for detailed bit error rate measurements based on a larger data sample). Data display capabilities have been increased, since the computer has access to a greater amount of spacecraft information.

4.8 Pyrotechnic Subsystem EOSE

The pyrotechnic subsystem EOSE provides discrete commands to spacecraft pyrotechnics via the ordnance initiate circuits, simulated ordnance loads, and monitor lights to indicate ordnance circuit actuation. The subsystem EOSE is operable manually or automatically by the computer via the data entry and monitor rack.

4.8.1 Test Objectives and Design Criteria

The pyrotechnic EOSE performs the following test functions:

- Verify that the safe-arm circuitry provides the proper status indication for both the safe and arm conditions.
- Verify the proper control of prime power application to the spacecraft energy storage circuitry, including proper charge rate.

- Verify that application of prime power within specified tolerances to the spacecraft energy storage circuitry does not result in transients of sufficient magnitude and duration to actuate any electro-explosive devices.
- Verify that current available from the spacecraft energy storage circuitry is within the specified tolerance to fire all ordnance when commanded in a simulated sequence.
- Verify that application of a fire command results in actuation of the commanded ordnance device only of the possible 15 (see Table 1).

Table 1. Fire Commands for Voyager Spacecraft Pyrotechnics

1. Release high-gain antenna
2. Release medium-gain antenna
3. Deploy low-gain antenna
4. Deploy PSP gimbal arm
5. Uncage PSP gimbals
6. Separate capsule emergency cable disconnect
7. Actuate capsule emergency separation (1 and 2)
8. Start midcourse trajectory correction 1
9. Stop midcourse trajectory correction 1
10. Start midcourse trajectory correction 2
11. Stop midcourse trajectory correction 2
12. Start midcourse trajectory correction 3
13. Stop midcourse trajectory correction 3
14. Pressurize propellant tanks
15. Rupture pressurization burst discs
16. Start orbit insertion propulsion
17. Stop orbit insertion propulsion
18. Enable orbit trim propulsion

- Verify absence of stray voltage in the spacecraft circuitry.
- Verify the ability of the spacecraft to recognize and respond to fire commands at the edges of the tolerance limits.

- Verify the ability of the spacecraft to recognize and respond to safe-arm commands.

The Voyager pyrotechnic subsystem connects with the pyrotechnic subsystem EOSE via the 25 spacecraft ordnance initiate circuits. These circuits are stimulated by an interval generator in the EOSE. Load simulation is employed for test purposes; the tests involve only the electrical portion of the pyrotechnic subsystem.

4.8.2 Operating Characteristics and Constraints

As illustrated in Figure 36, the pyrotechnic EOSE consists of an interval generator to generate fire command pulses to trigger the spacecraft ordnance initiate circuits, simulated loads to represent the ordnance

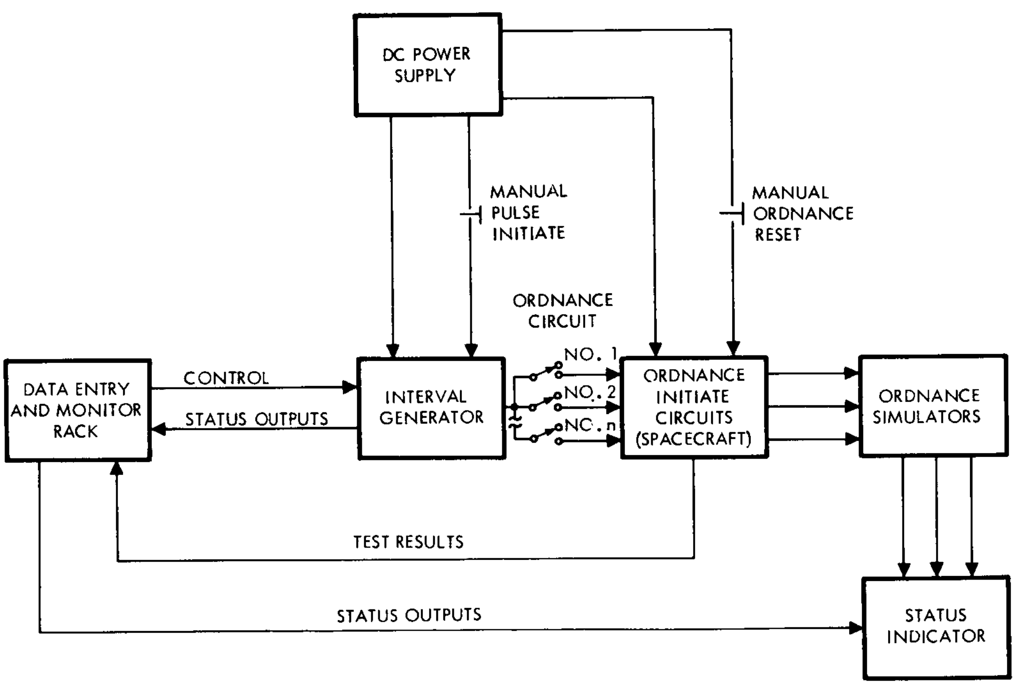


Figure 36. Pyrotechnic Subsystem EOSE Block Diagram

load to the initiate circuits, and threshold sensing to evaluate the current pulses delivered to the explosive devices. In addition, DC and AC power supplies are provided to power both the EOSE and the spacecraft subsystem. A data entry and monitor rack is included to provide the interface between the subsystem EOSE and the central data system computer for automatic operation and status display. An oscilloscope and VTVM are mounted in the rack for manual evaluation of the ordnance signals which are brought to a test jack panel.

The interval generator is normally controlled by the computer via the data entry and monitor rack. It is capable of simultaneously commanding a minimum of five discrete commands. A total of 25 initiated command discretes are provided, all having the following characteristics:

Voltage:	25 to 32 vdc
Duration:	100 ±10 milliseconds
Maximum loading:	100 milliamperes per circuit

In manual mode, the interval generator is set up by front panel control with respect to pulse width, amplitude, and selection of ordnance initiate circuits to be triggered. Pulse initiate and spacecraft circuit reset are also available for manual control.

Ordnance load simulators duplicate the resistive element of the explosive device within 1 per cent and contain a circuit breaker in series with the resistive element which opens on application of a pulse of 5 (+2, -0) amperes with a duration of 50 to 150 milliseconds. The simulators provide indication of circuit breaker actuation. The current through each ordnance simulator is thus evaluated by a threshold sensing device (the circuit breaker), and the data entry and monitor rack and the EOSE status indicator receives indication that the current through the squib commanded to be fired is adequate for the firing. In addition, the EOSE must sense energy present at undesired times, and for this purpose sensors with adjustable thresholds which can be set to a value lower than the breaker threshold are provided across each simulator. These stray pulse indications also interface with both the data entry and monitor rack and the EOSE status indicator panel.

The status indicator provides a group of monitor lights to indicate test results. Latching circuits are provided for discrete commands or responses to provide a positive display. The monitor lights on the status display conform to the following:

- Red = out of tolerance or "no-go" condition
- Green = in tolerance or "go" condition

- Amber or white = general status or test configuration

For manual checking and troubleshooting a VTVM, an oscilloscope, and a strip chart recorder are provided, to be used with the spacecraft test points and ordnance simulator points which are brought to a test jack panel.

The power supplies provide 25 to 50 vdc and 50 volts ±2 per cent square wave at 4.1 kc as prime power input to the spacecraft pyrotechnic subsystem. In addition, EOSE secondary power is delivered from this supply. The spacecraft supply outputs are variable beyond the specified design tolerance of the spacecraft power system and are capable of being modulated with specified noise and transients for subsystem margin testing.

A low current ohmmeter is provided to allow resistance of the bridge wires to be checked.

All pyrotechnic subsystem OSE is mounted in one rack as shown in Figure 37.

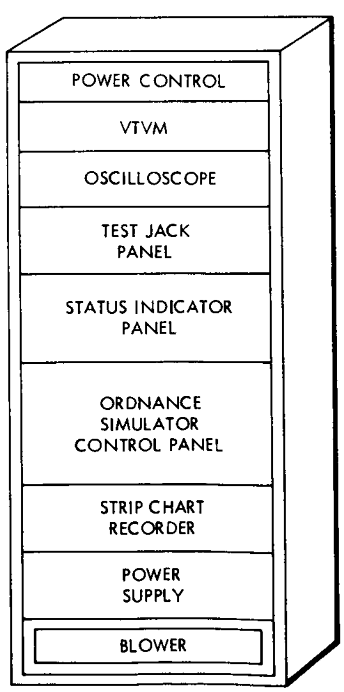


Figure 37. Pyrotechnic Subsystem EOSE Rack Layout

4.9 Propulsion Subsystem EOSE

The propulsion subsystem EOSE provides the capability for automatic and manual functional electrical testing, at both system and subsystem levels, for spacecraft propulsion subsystem cold engine operations.

4.9.1 Test Objectives and Design Criteria

The propulsion subsystem EOSE provides the following test functions:

- Measure equib continuity and resistance (100 ma maximum to prevent inadvertent firing).
- Simulate resistance of solenoids.
- Determine continuity of normally closed contacts.
- Provide analog voltage proportional to transducer output.
- Simulate each transducer sensor voltage output over the sensor output range, to provide calibration flexibility.
- Drive the pintle actuator to each extreme position and determine the response time.
- Include automatic self-test of the EOSE.
- Provide fault isolation in the propulsion subsystem to the provisioned spare replacement levels.

The propulsion subsystem EOSE interfaces with the following equipment or services:

- Primary power from the facility consisting of 115 volt, 60 cps, single-phase power.
- Input-output lines to the CDS data entry and monitor rack which provide automatic control and monitoring.
- Stimulus and monitoring functions between this EOSE and the spacecraft propulsion subsystem squibs, solenoids, pintle actuator, and transducers.

4.9.2 Operating Characteristics and Constraints

Figure 38 is a block diagram and Figure 39 is a rack layout of the EOSE.

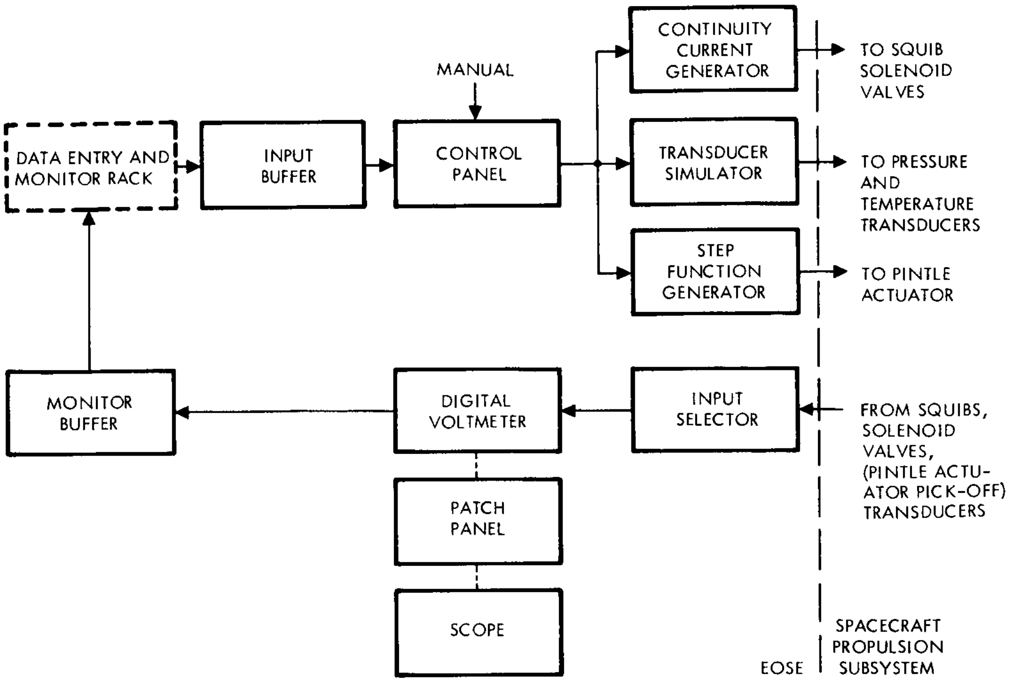


Figure 38. Propulsion Subsystem EOSE Block Diagram

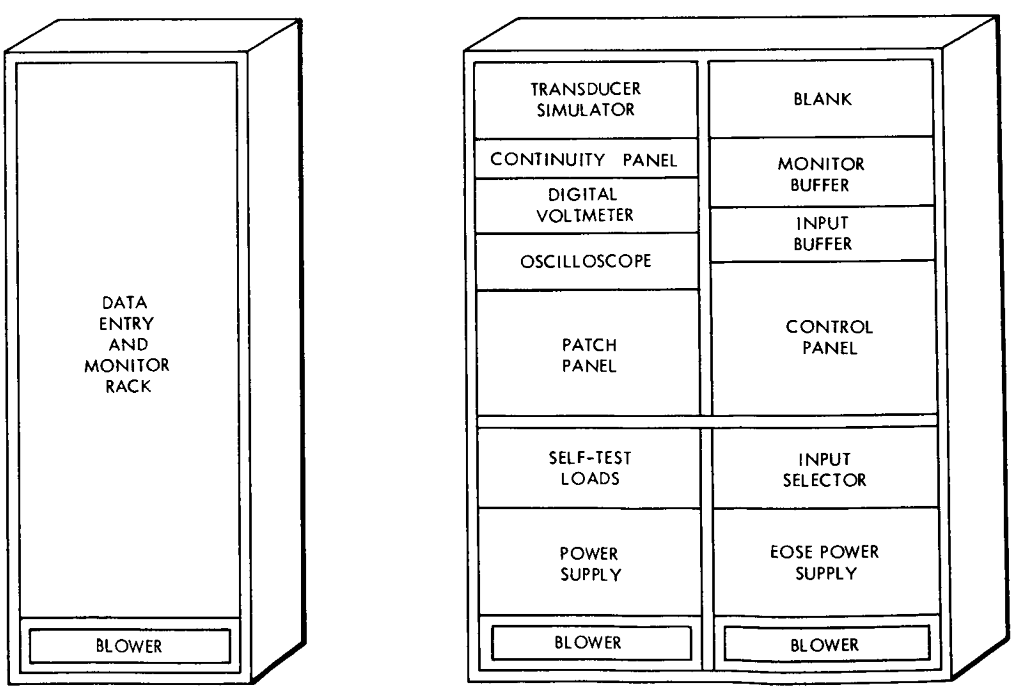


Figure 39. Propulsion Subsystem EOSE Rack Layout

In the automatic mode of operation, propulsion EOSE control is provided either sequentially via the CDS computer program, or via selection of specific tests by keyboard on the data entry and monitor rack (DE and MR). During an automatic test, the computer controls the propulsion EOSE via input lines between the DE and MR and the input buffer where the control is transformed from digital data to discrete control signals.

The control panel is the distribution point for the signals to be applied to the spacecraft propulsion subsystem via the continuity current generator, transducer simulator, and the step function generator. The input buffer also selects the return line from the spacecraft propulsion subsystem by controlling the input selector which, in turn, selects the proper line. The monitored signal is fed to the digital voltmeter, which converts the signal to digital form. The signal is then converted to computer format by the monitor buffer and this data is fed to the DE and MR.

In the manual mode the test functions are selected on the EOSE control panel and routed to the spacecraft propulsion subsystem in the same manner as for automatic operation, except that the oscilloscope is now used to determine the response time of the pintle actuator. In the troubleshooting mode, the patch panel is used to connect the digital voltmeter, oscilloscope, signal simulators, and test points from the spacecraft propulsion subsystem test connector.

The DE and MR provides a decimal readout of the monitored function when operating in the automatic mode. In the manual mode all functions are monitored as a voltage and displayed on the digital voltmeter. The oscilloscope displays both the step function controlling the pintle actuator and its response.

The propulsion EOSE includes the following elements:

- The control panel contains appropriate switches, lights and relays, for controlling the propulsion EOSE. An automatic-manual switch is used for selecting automatic control of the propulsion EOSE (from the CDS computer) or manual control from the EOSE control panel.
- The input buffer panel accepts EOSE control inputs and parallel data inputs from the DE and MR and stores this information in

appropriate registers for providing control signals to the control panel.

- The input selector panel contains suitable relays for connecting the input of the digital voltmeter and the counter to various monitoring points.
- The monitor buffer panel converts information from the digital voltmeter, counter, and the propulsion subsystem discretes into parallel data outputs to the data entry and monitor rack.
- The digital voltmeter performs A to D conversions during automatic operation and provides an accurate numeric data display during manual operation.
- The patch panel contains the current limiters required for protection of the squib circuitry. It also contains the resistor networks required for voltage dividing and signal conditioning. These resistor networks provide the signals for resistance and continuity tests such that the proper decimal readout is easily recognizable (e.g., the digital voltmeter reads out 1.000).
- The power circuit consists of a DC power supply, the output of which is controlled by a silicon controlled rectifier. The silicon controlled rectifier provides the 12 amperes maximum step function required to drive the pintle actuator.
- The transducer simulator circuits provide the capability of matching the impedance of each transducer as it applies the sensor simulator voltage. Voltage dividing networks provide the capability to vary the transducers over their operating range for calibrating and for troubleshooting.
- Two power supplies are used to provide EOSE power. One supplies power for the buffers and input selector; the other provides power for the continuity, transducer and step function circuitry.
- Self-test loads are provided to simulate various propulsion subsystem loads during both automatic and manual EOSE self-tests.

4.10 Science Subsystem EOSE

Science subsystem EOSE is provided GFE and is therefore not detailed in this document. Recognition of interface characteristics of the science subsystem, however, its impact on spacecraft testing, and the integration of such tests into the system test complex procedures are the responsibility of the Voyager spacecraft bus contractor, and these topics are discussed in Volumes 1 and 2. Close liaison is required in early establishment of interface documentation and fabrication and testing of simulators for both the science subsystem interface as presented to the spacecraft and the spacecraft interface as presented to the science subsystem.

III. ASSEMBLY, HANDLING AND SHIPPING EQUIPMENT

Like the EOSE, design of the AHSE began with an analysis of operational flows to derive functional requirements. These functional requirements were then consolidated into end-item requirements so that conceptual designs could be developed. End-item requirements were also correlated to time-phased operational flows for determining quantities, need-dates, and allocations.

The functional analysis conducted for this study (see Volume 3) primarily dealt with spacecraft assembly and checkout, system test, and launch operations. Brief functional descriptions are included here for all major items of AHSE. The equipment list will be expanded and functional descriptions for the smaller items of AHSE will be completed in Phase IB.

The AHSE implementation task breakdown (in IV of this Volume) separates the AHSE into four categories having different design, documentation, and test requirements:

- MSSE (mechanical spacecraft support equipment), spacecraft and subsystem test equipment which leaves a contractor's facility.
- STE (special test equipment), spacecraft and subsystem test equipment which does not leave a contractor's facility.
- ST (special tooling), factory aids and unique subsystem test fixtures which do not leave contractor's facility.
- S/M (simulator or model), simulators or models required for contractor's development testing which does not leave contractor's facility.

The above categories were developed for convenience in identifying various tasks. There is no intention of classifying any item of mechanical support equipment as other than AHSE, in conformance with the AHSE listing of Paragraph 3.12 in the JPL "Performance and Design Requirements, General Specification," dated September 17, 1965.

Three modes of spacecraft transportation are presented: land and sea, air via helicopter, and air via the Very Pregnant Guppy. The equipment required for each mode of transportation is described in the AHSE list, and the tradeoff study for each mode of operation is included in Appendix E. Helicopter and Very Pregnant Guppy modes are recommended in preference to the land and sea modes because of the numerous problems associated with land transportation. Appendix B substantiates this recommendation.

The AHSE design data are presented in the format given in the JPL General Specification:

1) Objectives and Design Criteria
2) Design Characteristics and Restraint
3) System Level Functional Descriptions
4) Subsystem Functional Descriptions

Schedule and implementation information for the AHSE has been integrated with that for the EOSE in Section IV.

Sections 1 and 2 present all of the general design requirements for the AHSE, both the requirements noted in the JPL General Specification and those additional requirements TRW has placed on the design. All unique design requirements are incorporated into each end-item functional description.

A list of all the AHSE derived from this study appears in Tables 4 and 5. Each AHSE requirement noted in Paragraph 3.12 of the General Specification, has been accounted for by an AHSE end-item.

A separate listing of the applicable documents for the AHSE is included in Appendix D of this volume. It is planned that during Phase IB the applicable documents will be extracted from the appendix for inclusion in the functional specifications. The system and subsystem functional descriptions include specific functional design requirement, test requirements, interface definitions, general equipment descriptions, and a preliminary design for each major item of AHSE.

The dimensions shown in the functional descriptions are derived from past experience with similar items, not from analysis, and are preliminary. Because of their simplicity, functional descriptions for slings are not included. However, slings are shown pictorially in the functional descriptions of the sling-interfacing AHSE.

1. OBJECTIVES AND CRITERIA

1.1 Objectives

The AHSE provides the capability for lifting, holding, positioning, aligning, protecting, and transporting the spacecraft and its components and subsystems. It also helps to verify the spacecraft system design concept and confirm the mechanical portion of the flight readiness condition. In achieving the above objectives, the following design policies were followed:

a) Handling fixtures are provided for items which are either too large, too fragile, or too heavy for normal personnel handling. Where precise mating of parts is required, especially when the parts are heavy, a precision slow rate hoist attachment is provided.

b) All AHSE is designed for use by experienced technicians, since the assembly and testing of the spacecraft will be done by technicians who have worked on previous spacecraft. The equipment is compatible with the operational flow for assembly and checkout, spacecraft test, and launch operations of the spacecraft, and with the development test, and manufacturing flow of subsystems.

c) Whenever possible, ground support equipment developed for other programs such as Apollo, LEM, and SIVB is utilized without modification, even if it has excess capabilities. When modifications are require, they are designed to permit easy retromod. The design of new AHSE for Voyager-peculiar requirements has been based on experience and capability developed in other TRW programs.

d) All load-carrying AHSE will be proof-loaded prior to interface with end items and again periodically to verify capability. Operating AHSE will be functionally tested prior to initial use and again periodically to verify proper operation.

d) All test and assembly procedures will be verified with the engineering model before they are used with proof test and flight spacecraft.

1.2 Criteria

The following general design criteria apply to all AHSE. Item-peculiar design criteria are listed in the functional descriptions of the individual items.

Safe operating conditions will prevail for both the equipment being handled and the personnel involved. Specifically, the design of AHSE conforms to the requirements of the General Range Safety Plan, Volume I, and associated Appendix A, and AFETR P30-2. Safety considerations cover interfaces between the AHSE and the operator and between the AHSE and the spacecraft. The following criteria apply:

a) The probability of operator error is kept to a minimum through clearly detailed and logical procedures, clearly visible instructions and caution plagques, sufficient working space, and readily accessible and comfortably operable controls. Noise levels from operating equipment are kept low to permit unambiguous voice commands. Where special hazards appear, additional safeguards such as "dead-man" switches are installed and interlocks to prevent AHSE operation beyond design limits.

b) Design features are incorporated which physically protect the spacecraft and its systems from AHSE failure or malfunction. Materials used in the AHSE present no hazard to the spacecraft during any operational phase. Plating and bearing design take into account potentially degrading metal matings and the various environmental conditions possible. Normal operation of the AHSE will not violate the cleanliness requirements.

c) Shock and vibration damping are provided to protect the spacecraft and its components during checkout, transport, and launch. Reliability of AHSE is insured by standard components, proven design concepts, and conservative design approaches providing easy operation with minimum maintenance. Maintenance can be performed in a safe and comfortable fashion, using standard hand tools. Access is provided for repair and replacement, test, inspection, fabrication, and assembly.

Multiple-use features are incorporated in the design of AHSE wherever possible and feasible. Servicing of different spacecraft of the same design uses interchangeable techniques, within the assemblies down to the subassembly level, without requiring calibration or modification of the parts.

The AHSE planned for use in the clean room is designed for ease of cleaning. The equipment is designed to prevent particle contamination of the clean area by proper surface treatment, materials selection, and avoidance of irregular surfaces. The equipment is also compatible for use within a 100,000-class clean room as specified by Federal Standard 209.

2. CHARACTERISTICS AND RESTRAINTS

2.1 Characteristics

All AHSE for the Voyager spacecraft has the following basic characteristics: Weight and size are as necessary to service the spacecraft and its parts (without constraining spacecraft design in any way); design is simple, and it has complete compatibility (material, functional and magnetic) with the spacecraft; stable adjustment and positioning provisions are included to eliminate readjustment during tests.

2.2 Restraints

All AHSE having electrical components and used to support the spacecraft or its parts during tests use high-quality insulation, eliminating conductive paths to the test item. Magnetic interaction is minimized through use of nonmagnetic materials. The magnetic field strength at the spacecraft magnetometer will be 10 gamma or less when spacecraft and AHSE are combined in magnetic tests. An electrical grounding system is provided as required, compatible with the facility and providing adequate protection to the spacecraft.

No AHSE used in a thermal-vacuum chamber will cause any contamination or function degradation of the spacecraft by outgassing, arcing, spalling, or any other means.

Strength and rigidity requirements are considered at both design (yield) and ultimate load levels. All structures have a positive margin of safety computed in accordance with MIL-HDBK-5 procedures. Figure 40 defines the maximum allowable load envelope for the spacecraft during all operations involving AHSE.

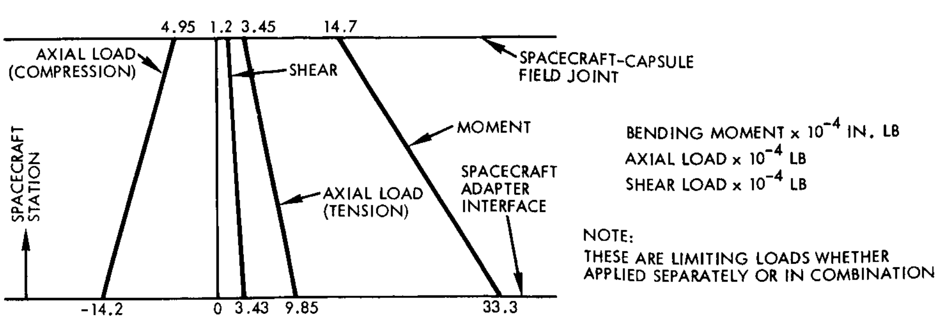


Figure 40. Maximum Allowable Load Envelope for Spacecraft During Transportation and Ground Handling

Limit loads are applied to the AHSE through their structural design centers of gravity and reacted statically. The reactions are appropriate to the design condition and applied conservatively. The design of certain structural components may be dictated by either stiffness or functional limits; but analyses will verify that strength requirements are also satisfied.

All AHSE that will be transported by air is designed to withstand accelerations from emergency landings without any major component breaking loose and without external physical collapse. The ultimate load factors, in accordance with MIL-A-8421B, are shown in Table 2.

Table 2. Emergency Landing Ultimate Load Factors

Direction	Condition		
N_V	4.5	0	0
N_S	0	± 8.0	0
N_L	0	0	± 8.0

Note: The axes of the coordinate system are identified relative to earth and applied to mechanical handling and test equipment in their normal attitude relative to earth. The sign convention refers to direction of acceleration of the mass being handled by the equipment.

N_V = Vertical load, axis vertical relative to earth, positive action down

N_L = Lateral load, axis horizontal relative to earth and in direction of motion of the equipment

N_S = Side load, axis horizontal relative to earth and perpendicular to the direction of motion of the equipment

The loads applied to the spacecraft or its components by the AHSE during movement will not exceed the flight acceptance test level spectra of shock, vibration, and acceleration.

There will be no evidence of excessive deflection or permanent deformation after the AHSE has been proof-loaded. Proof-load values for testing of AHSE items are equivalent to those produced by the design limit load and hazard factors.

The limit load for AHSE is normally the working load (weight of the spacecraft or component and the associated item of mechanical AHSE) multiplied by the limit load factor. Limit load factors to be used for design are listed in Table 3.

AHSE is designed to withstand design loads without permanent deformation or excessive deflection. Excessive deflections are those which would result in unsatisfactory mechanical performance or induce loads in the spacecraft or components that exceed the design loads. The design load is the limit load, multiplied by the hazard factor.

The hazard load factor for all AHSE is considered to be 1.0 except for hoisting equipment, in which case the hazard factor is to be 1.5. Pressure vessels used in AHSE will use a hazard factor of safety of 2.0.

AHSE is designed to withstand ultimate loads without failure. Failure is defined as inability to sustain ultimate load. The ultimate load is the design load multipied by the ultimate factor of safety.

The ultimate factors of safety are as follows:

Factors of Safety

Item	Hazard Factor	Ultimate Factor
All mechanical AHSE (except hoisting equipment)	1.0	2.0
Hoisting equipment (i.e., rotation or tilt fixtures, engine and propulsion handling fixtures, slings)	1.5	2.0
Pressure vessels	2.0	2.0

3. SYSTEM LEVEL FUNCTIONAL DESCRIPTIONS

Table 4 lists the system AHSE. All of the major items in the table (noted by *) are then described, by means of summary page-long

Table 3. AHSE Limit Load Factors

Transporter-flight spacecraft	N_V	+3.0	-2.0	
	$N_L = N_S$	± 1.5	± 1.5	
Environmental cover (spacecraft)	N_V	+3.0	-2.0	Depending on operational sequence, wind loads may also be included.
	$N_L = N_S$	± 1.5	± 1.5	
Environmental covers (other than spacecraft	N_V	+3.0	-2.0	
	$N_L = N_S$	± 1.5	± 1.5	
Shipping containers	N_V	+4.0	-3.0	
	$N_L = N_S$	± 3.0	± 3.0	
Handling dollies	N_V	+2.0	-1.0	
	$N_L = N_S$	± 1.0	± 1.0	
Assembly, handling frames and fixtures	N_V	+4.0	-3.0	Rigidity requirements must be examined
	$N_L = N_S$	± 3.0	± 3.0	
Protective covers	N_V	+4.0	-3.0	
	$N_L = N_S$	± 3.0	± 3.0	
Weighing, c.g., and inertia fixtures	N_V	+2.0		All vertical forces will be assumed to vary in direction from 0 to 10 degrees from nominal rigging position.
	$N_L = N_S$	0	0	

Table 4. System Level AHSE

System		AHSE No.	Item	Source	Quantity
(50 Series)		5001	TRW Electrical Prime	TRW	4
	*	5002	Spacecraft-Planetary Vehicle Sling	TRW	6
	*	5003	Vertical Checkout and Assembly Stand (Mobile)	DAC	6
	*	5004	Hydraset, 1 ton	TRW	4
	*	5005	Hydraset, 5 ton	TRW	4
	*	5006	Hydraset, 10 ton	TRW	4
		5007	Spacecraft Work Stands	TRW	6
		5008	Mechanics Tool Kit	TRW	30
	*	5009	Hydraset, 20 ton	TRW	
		5010	Capsule (Test)	GFE/JPL	3
		5011	Capsule Shipping and Handling Dolly with Environmental Cover	GFE/JPL	3
		5012	Capsule Transporter and Hoist Sling	GFE/JPL	1
	*	5013	Planetary Vehicle Inverter	TRW	1
	*	5014	Component Alignment Instruments	TRW	2
	*	5015	Alignment Optical Instruments	TRW	2
	*	5016	Equipment Kit, Mass Properties	DAC	2
		5017	Miscellaneous Shipping Container	TRW	6
	*	5018	Magnetic Test Fixture	TRW	1
	*	5019	Vibration Machine Adapter	TRW	1
	*	5020	Special Appendage Deployment Equipment	TRW	1
	*	5021	Thermal-Vacuum Test Adapters	TRW	1
		5022	Thermal-Vacuum Test Instrumentation	TRW	1
	*	5023	Free Mode Test Adapter	TRW	1
		5024	Shroud-Spacecraft Clearance Measuring Instrument	TRW	1

*Functional Description included in this volume.

Table 4. System Level AHSE (Continued)

System	AHSE No.	Item	Source	Quantity
(50 Series)	* 5025	Sterilization Pressure Dome	DAC	2
	5026	Tag Lines	TRW	9
	* 5027	Flight Shroud Planetary Vehicle Transporter	GFE/DAC	1
	5028	Flight Shroud Section Sling	GFE	1
	* 5029	Flight Shroud Planetary Vehicle Cover	DAC	3
	* 5030	Flight Shroud Planetary Vehicle Hoist Beam	DAC	4
	5031	Flight Shroud Assembly Fixture	GFE	1
	* 5032	Hoist Kit, Spacecraft Shipping Container (DSV-4B-303)	GFE/DAC	1
	* 5033	Instrumentation Unit (SC) (1B57308)	GFE/DAC	1
	* 5034	Sterilization Unit	DAC	1
	5035	Saturn 5-Booster Simulator	GFE	1
	* 5036	Equipment Mounting Panel Handling Fixture	TRW	3
	5037	Equipment Mounting Panel Hoist Sling	TRW	2
	* 5038	Equipment Mounting Panel Installation Fixture	TRW	3
	5039	Equipment Mounting Panel Shipping Container	TRW	3
	5040	Test Capsule Shipping Container	GFE/JPL	3
	* 5041a	Spacecraft Transporter Modified SIVB Transporter (DSV-4B-300)	GFE/DAC	1
	* 5042a	Instrumentation Trailer, SIVB, Modified (NASA 5146-1)	GFE/NASA	1

a - AHSE required for road and sea transporation of the spacecraft

Table 4. System Level AHSE (Continued)

System	AHSE No.	Item	Source	Quantity
(50 Series)	*5043a	Handling and Support Kit Spacecraft Shipping Container Mod. SIVB Equipment (DSV-4B-462)	GFE/DAC	1
	*5044a	Air Conditioning Unit (new item)	DAC	1
	*5045a	Transporter Cradles, Shipping Containers, SIVB Mod. (DSV-4B-301)	GFE/DAC	1
	5046a	Purge Unit, SIVB Mod. (DSV-4B-1865)	GFE/DAC	1
	*5047a	Generator Trailer, (NASA 5145-9)	GFE/NASA	1
	*5048a	Transporter Prime Mover, SIVB	GFE/DAC	1
	*5049b	Roller Kit, SIVB (DSV-4B-1863)	GFE/DAC	1
	*5050b	Air Carry Support Kit, SIVB Mod. (DSV-4B-1859)	GFE/DAC	1
	*5051b	Tie-Down Kit, SIVB Mod. (DSV-4B-1861)	GFE/DAC	1
	*5052b	Access Kit, SIVB Mod. (DSV-4B-1860)	GFE/DAC	1
	*5053b	Cargo Life Trailer, SIVB	GFE/NASA	1
	5054a	AKD Barge Tie-Down Kit	GFE/DAC	1
	*5055c	Miscellaneous Handling and Rigging Kit, Helicopter (new item)	DAC	1
	*5056c	Instrumentation Kit, Helicopter (new item)	DAC	1
	*5057	Magnetic Facility Adapter	TRW	1
	*5058a, b, and c	Spacecraft Shipping Container	DAC	4
	5059	Spacecraft Shipping Container Sling	DAC	3

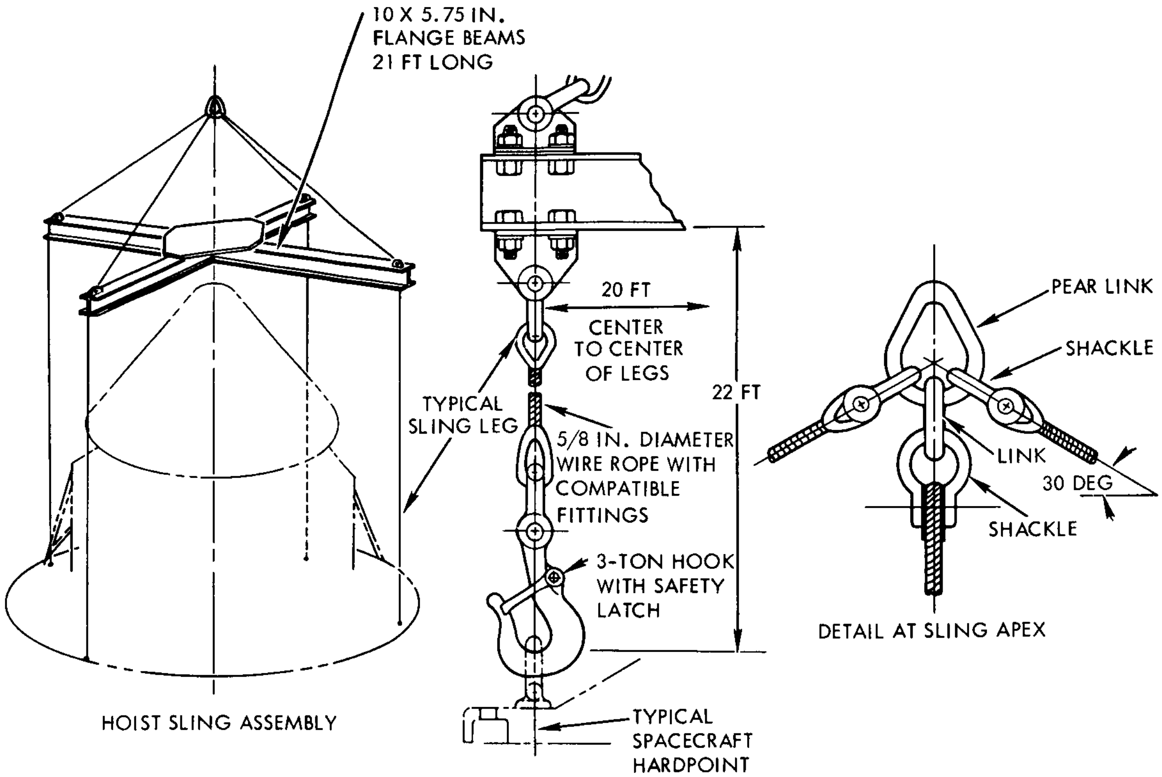
a - AHSE required for road and sea transportation of the spacecraft
b - AHSE required for aircraft (VPG) transportation of the spacecraft
c - AHSE required for helicopter transportation of the spacecraft

functional descriptions, except for GFE items. Where known, however, part numbers are given for those items of GFE. The table lists the sources, where known, of the GFE items. Thus GFE/DAC means that an existing item provided by Douglas Aircraft Company for another program can also be used for Voyager.

All of the system AHSE in the AHSE list in the "Performance and Design Requirements, General Specification, " 17 September 1965, is included in Table 4 , and, with the exception of two items, one-page functional descriptions are included. The two exceptions are the spacecraft-shroud clearance measuring device and the science-spacecraft test fixture. For the first, a commercial expansion gage is planned for measuring the clearance. For the second, whose function is to provide a test base for proper location of the science, one of the model spacecraft will be provided; the particular model used (thermal, structural, separation, etc.) will be a function of when this science test base is required. In the event that the plans show that none of the models will be available at the correct time, a simulated spacecraft will be provided.

Although a few of the GFE/DAC items require minor modifications to accommodate the Voyager configuration, e.g., repositioning the cradles on the transporter, installing a flat plate on the transporter, most of the GFE/DAC items can be used without modification.

SPACECRAFT-PLANETARY VEHICLE SLING, AHSE 5002



Functional Requirements. A hoist sling is required for handling the spacecraft and planetary vehicle, by means of overhead cranes or hoists.

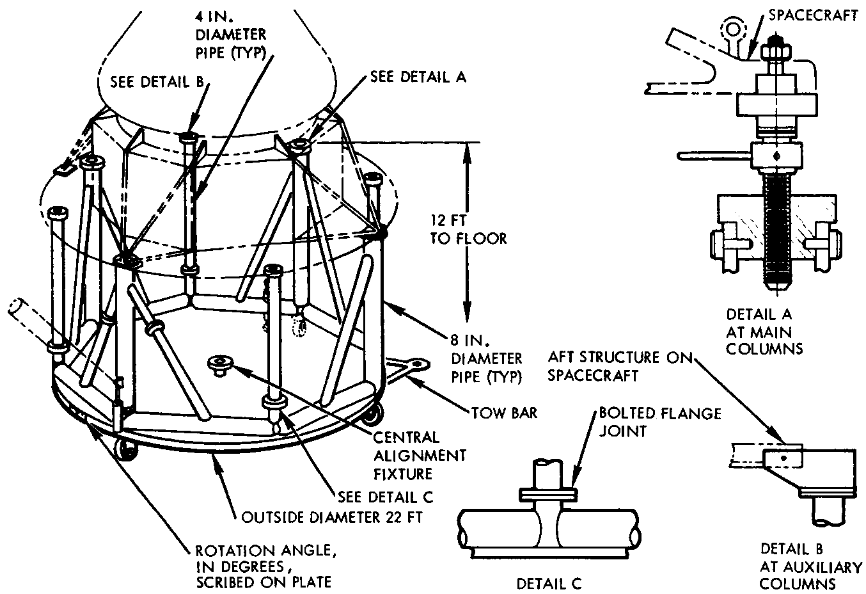
Design Requirements. The sling is designed to carry a load of 11,500 pounds. Standard cables and fittings are used throughout the assembly.

Description. The hoist sling consists of a spreader structure, in the form of a cross, and wire rope legs. Four upper legs attach to the extreme ends of the spreader and meet at a common link. Four lower legs are suspended from the extreme ends of the spreader and terminate in hooks with safety latches. The sling components are corrosion-resistant; the wire rope is vinyl coated.

Test Requirements. Fit checks, functional tests, and proof load tests are required to demonstrate the slings capability.

Interface Requirements. The sling is compatible with the planetary vehicle and hoists it from the aft end. Hooks on the sling legs interface with eyebolts at the structure hardpoints. The uppermost lifting link is compatible with facility crane hooks and hydrasets.

VERTICAL CHECKOUT AND ASSEMBLY STAND, MOBILE, AHSE 5003



Functional Requirements. The vertical checkout and assembly stand is required to support a completely assembled planetary vehicle in a vertical attitude or to support a spacecraft alone.

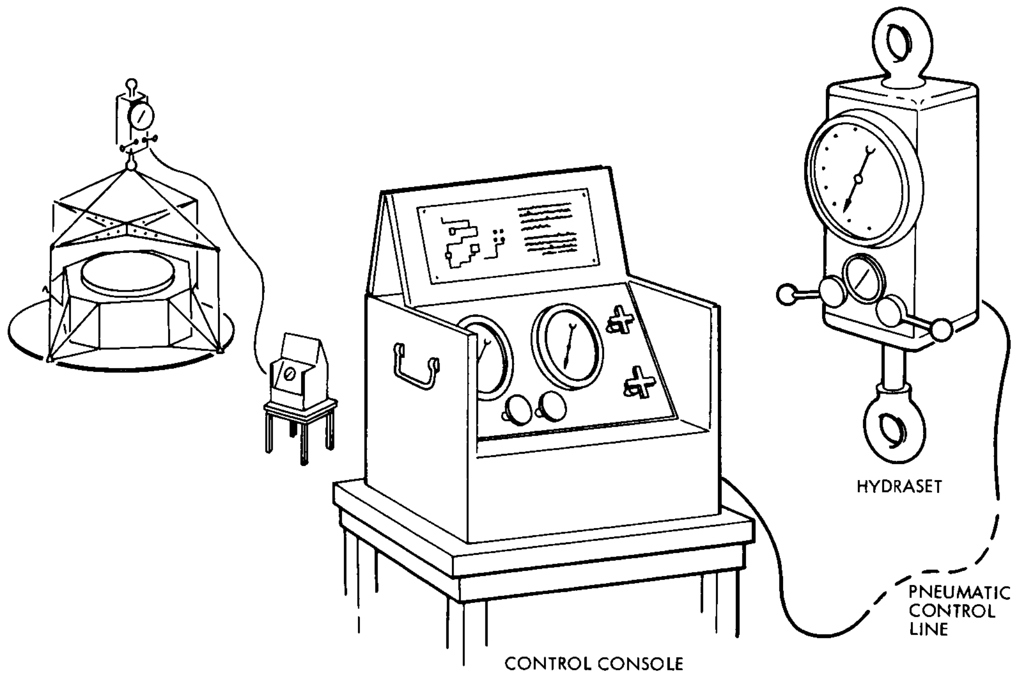
Design Requirements. The stand must support a load of 11,500 pounds. The stand may be towed, unloaded, for in-plant movement. Leveling jacks are provided in the stand base. For alignment a central fixture is attached to the base and angular readouts are provided. A means of attachment is provided for the spacecraft structure and the complete planetary vehicle. The construction of the stand allows the antenna to be deployed without interference. The stand can be rotated about its vertical axis and tilted.

Description. The vertical checkout and assembly stand consists of an octagonal base with vertical columns constructed from pipe. Mobility is provided by three caster assemblies mounted on the underside of the stand. A tow bar is provided. At the stand center is a support bearing which allows complete rotation, the load being supported by three air bearings in the base. Angular readout marks are scribed on the periphery of a circular baseplate and a central alignment fixture is provided. Of the four main columns which support the spacecraft one is hinged at its base to deploy the antenna. The four auxiliary columns which attach to the spacecraft structure are removable.

Test Requirements. Fit checks, functional tests, and proof load tests are required on the stand.

Interface Definitions. The stand interfaces with the spacecraft hardpoints and is compatible with the following items: TRW prime mover AHSE 5001, spacecraft work stands AHSE 5007, Appendage deployment equipment AHSE 5020.

HYDRASETS, AHSE 5004, 5005, 5006, 5009



Functional Requirements. Hydrasets are required for precision control during all hoisting operations, especially those involving mating of heavy components and the movement of the spacecraft.

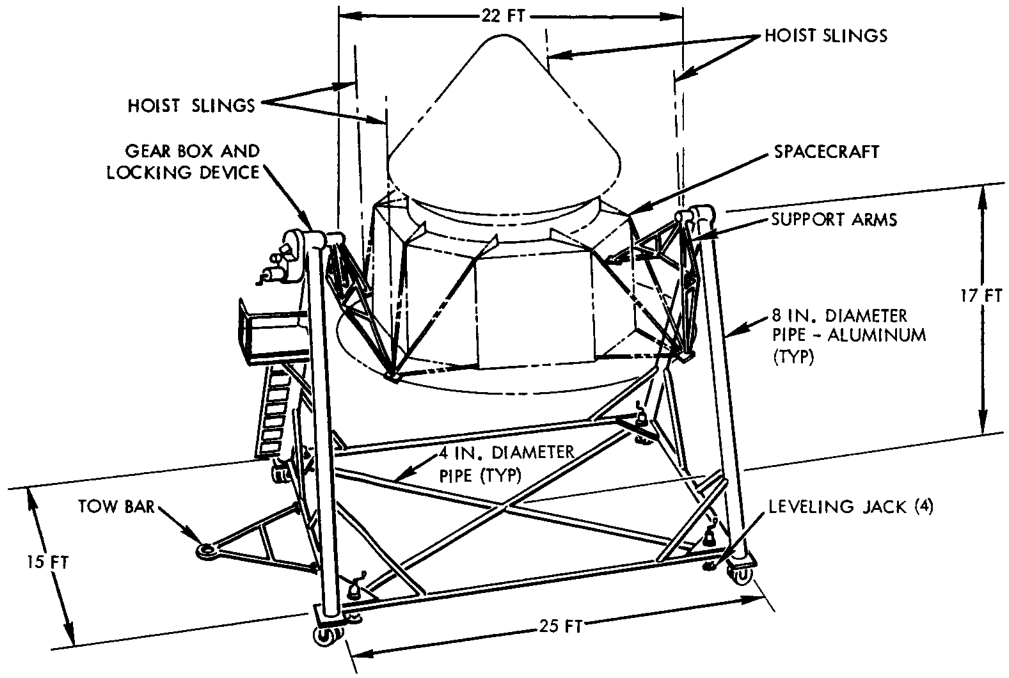
Design Requirements. The hydrasets must provide precision control for all hoisting operations, with both direct and remote actuation. Instruments of 1-, 5-, 10-, and 20-ton capacity must be provided.

Description. The hydraset is a hydraulic-pneumatic lifting device which provides precise control of initial and terminal hoist movement. It consists of a hydraulic cylinder with manual controls and a remote pneumatic control console.

Test Requirements. Functional testing at the rated load of each unit will be required.

Interface Description. The hydrasets interface with the facilities hoists and all handling and hoisting fixtures and slings.

PLANETARY VEHICLE INVERTER, AHSE 5013



Functional Requirements. The planetary vehicle inverter is required to provide for planetary vehicle rotation from the upright to the inverted position.

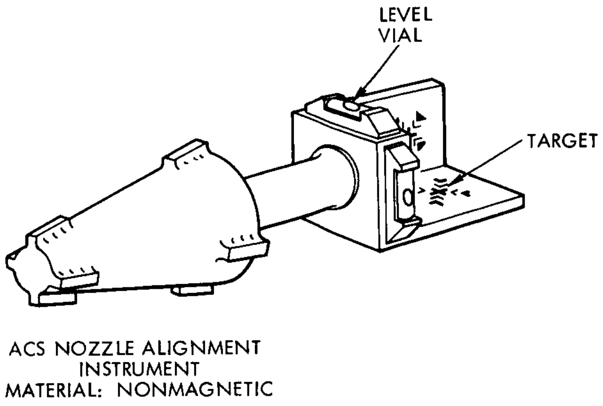
Design Requirements. The planetary vehicle shall be rotated at a slow rate to assure that it is not adversely affected. Attachment points for the planetary vehicle shall be such that safe handling is inherent.

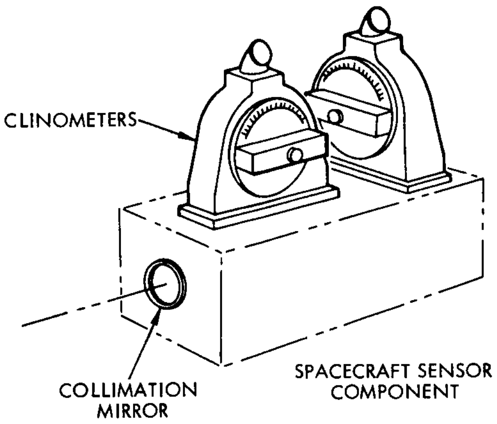
Description. The planetary vehicle inverter is composed of a basic "A" frame constructed of nonmagnetic 8-inch diameter aluminum pipe and 4-inch diameter support members. The planetary vehicle is supported by two sets of arms so that it will rotate about its center of gravity in the horizontal axis. The support arms are attached to the "A" frame at bearing points and are rotated by a gear box which also serves as a locking and positioning device. A tow bar and set of caster wheels at each member of the "A" frame provide mobility. Four jacks level and secure the inverter. (Same basic design as AHSE 5057).

Test Requirements. The equipment requires testing to insure that it can support the required loads and that the planetary vehicle will fit and rotate without interference.

Interface Definitions. The planetary vehicle inverter interfaces with and must be compatible with the planetary vehicle and the assembly and checkout facility.

COMPONENT ALIGNMENT INSTRUMENTS, AHSE 5014





Functional Requirements. Many spacecraft components (e.g., the control system sensors, the propulsion system) require careful alignment to the spacecraft reference system. A fixture for mounting a target, a collimation mirror, a level, or a clinometer to each of these components is provided for this purpose.

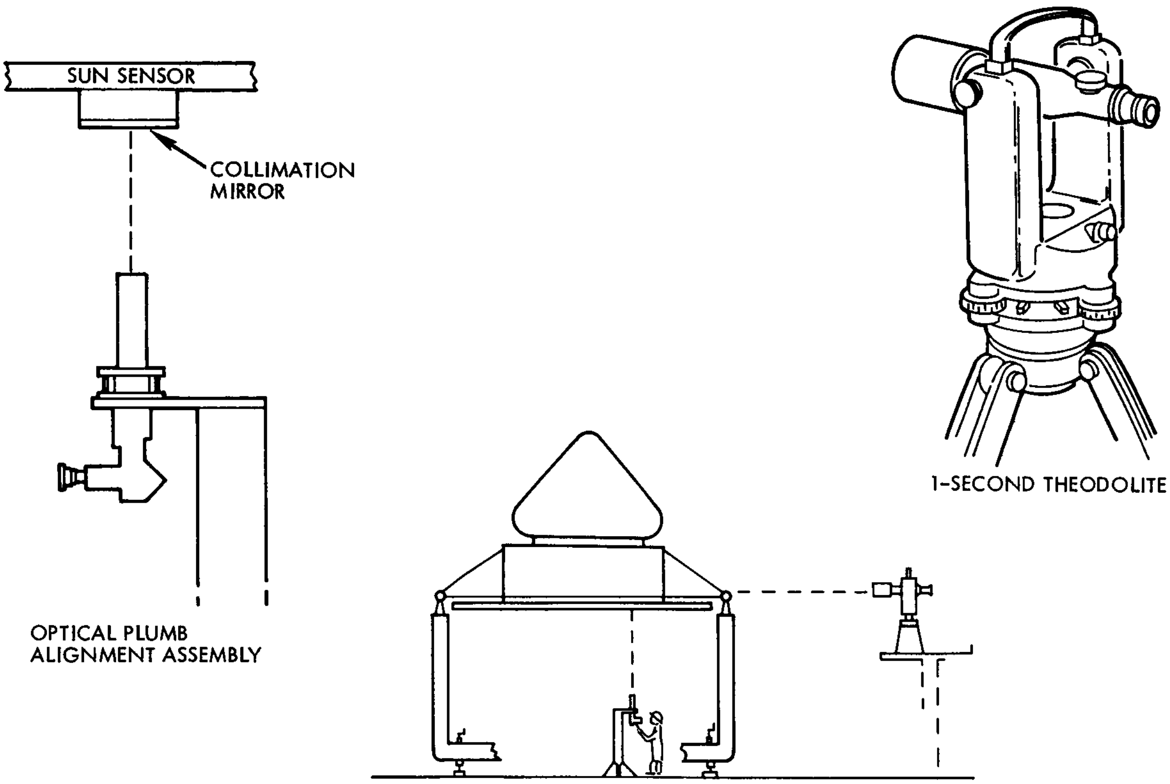
Design Requirements. All component alignment instruments must attach easily and accurately without affecting the alignment of the components on which they are mounted. Each component alignment instrument must be capable of measuring the alignment of its component to an accuracy one order of magnitude greater than the required alignment. All component alignment instruments must use commercially available targets, levels, or mirrors mounted on specially designed and fabricated fixtures.

Description. Each component alignment instrument consists of a special fixture to adapt standard commercial levels, clinometers, targets, or collimating mirrors to a specific spacecraft component. Two typical devices are shown.

Test Requirements. The test requirements of each instrument will be established by the accuracy required of it and the nature of its physical interface with the spacecraft component. The functional accuracy of each instrument is maintained by normal calibration procedures.

Interface Definition. Each component alignment instrument has a mechanical interface with its component and its supporting fixture.

ALIGNMENT OPTICAL INSTRUMENTS, AHSE 5015



Functional Requirements. The most critical alignment requirements on the Voyager spacecraft are angular. These angles must be measured to a maximum accuracy of 20 seconds of arc by optical instruments in conjunction with permanent references on the assembly, checkout, and alignment fixture (AHSE 5003) and on the spacecraft.

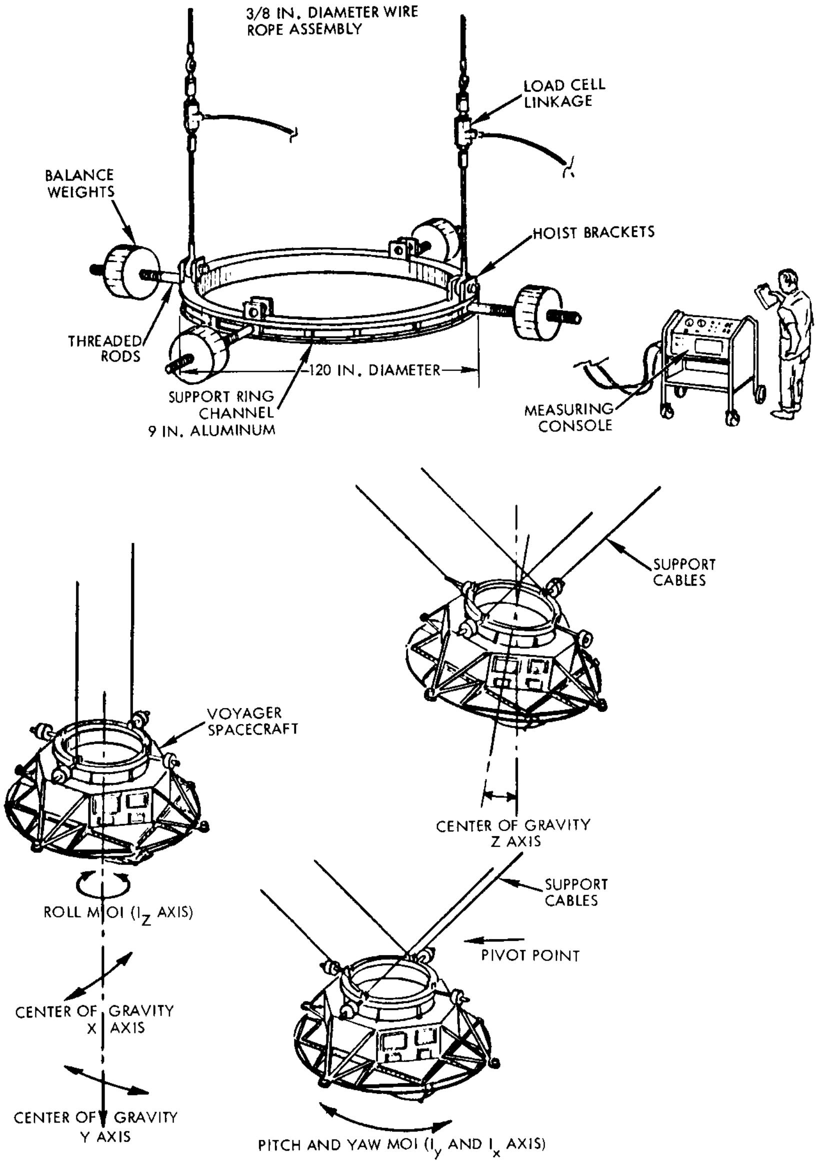
Design Requirements. The alignment optical instruments must be standard commercial instruments with standard fittings and with pointing accuracies of about one second of arc.

Description. The alignment optical instruments consist of commercially available one-second theodolites (e.g. the K and E KE 2 industrial model) and optical plumb alignment assemblies (e.g., the K and E model 71-5160).

Test Requirements. Normal optical bench calibration procedures will satisfy all test requirements.

Interface Definition. The alignment optical instruments have optical interfaces with the permanent basic alignment reference system and with the component alignment instruments (AHSE 5014).

EQUIPMENT KIT, MASS PROPERTIES, AHSE 5016



Functional Requirements. Mass properties of the Voyager spacecraft will be measured to obtain weight, center of gravity, and moments of inertia.

Design Requirements. The mass properties equipment kit must provide a rigid, free-swinging support for the spacecraft. Measurements taken must be to the following accuracies:

	Requirement	Target	Capability
Weight, percent	0.5	0.25	0.1
Center of gravity, in.	0.3	0.1	0.05
Moments of inertia, percent	3.0	1.0	

Optimum tare weight of the mass properties equipment kit is less than 10 percent of the spacecraft dry weight. Load-cells, cables, swivels, and pivots must be compatible with the measuring system accuracy and functional requirements. The load-cell capacity is 4000 pounds and is capable of withstanding a minimum of 20 percent overload without permanent adverse effect. Ultimate strength of the load-cell is 250 percent of the rated maximum load.

The load-cell readout equipment:

a. Provides a minimum of two load-cell channels

b. Confines instrument nonlinearity without load-cell to within ±0.0005 percent of full scale.

c. Confines zero instability to less than ±0.01 percent of full scale

d. Provides readout in percent of load-cell capacity in 1 and 10 percent, add-to-read switch settings. The minimum reading is equivalent to 0.1 pound when used with a 10,000-pound capacity load-cell.

A load simulator to set the span and zero reference of the measuring instrumentation is required. The load simulator consists of a calibrated resistor substitution bridge, linear to ±0.025 percent of reading.

All equipment must satisfy the above requirements while operating in an ambient temperature range of 70 ±30 $^{\circ}$F, and a relative humidity range of 20 to 90 percent. Maximum load-cell creep and hysteresis must not exceed 0.5 percent of the applied load.

Compensation must be provided (from 25 to 125 $^{\circ}$F) to limit the: (1) effect on sensitivity (output) to +0.0013 percent of load/$^{\circ}$F, and (2) effect of zero to +0.0025 percent of full scale/$^{\circ}$F.

Load-cells must be selected so that the output curve of mv/v versus load of each cell (maximum nonlinearity) is within the locus of points distant from a straight line by ±0.15 percent of full scale (2 mv/v output).

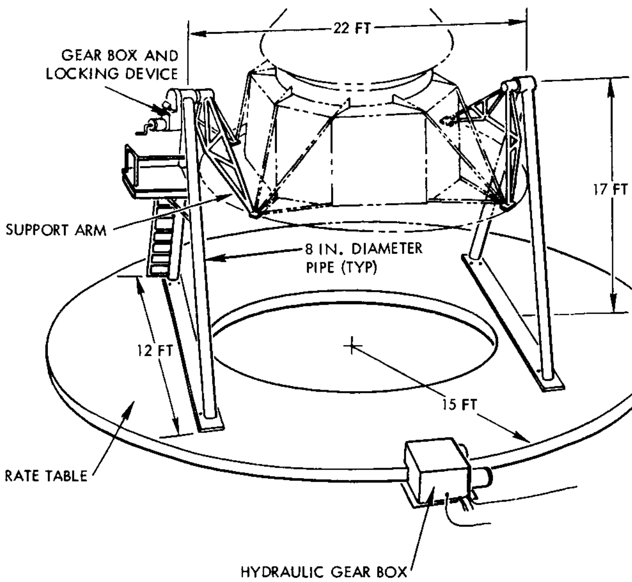
Cables must have minimum moisture-absorption tendency, all conductors of compound F per MIL-C-13777. The cables are neoprene-jacketed, exterior-type high flexible, nonflammable, and resistant to oils, solvents, chemicals, moisture, and aging. They will be 50 feet in length and shielded with tinned copper braid of 90 percent maximum coverage with capacitance stability approximately constant for eight wire leads over the full temperature range.

Description. The mass properties equipment kit consists of two major components: a support ring and weighing-suspending equipment. The support ring, a rigid circular aluminum channel structure bolted to the spacecraft-capsule adapter ring, fastenes to the capsul mounting holes in the adapter ring. Four hoist brackets at 90-degree intervals on the upper side of the support ring attach to two wire rope cables for MOI measurements or a set of slings and load-cell linkages for weighing. Four threaded rods are fastened to the periphery of the ring at the hoist bracket locations to accept adjustable position weights for center of gravity and MOI determination. The weighing equipment consists of load-cell linkages and a measuring console. Each linkage consists of a Miller-type swivel, 4000-pound load cell, and a wire rope cable assembly. The readout equipment consists of a portable electronic weight measuring kit.

Test Requirements. The hoisting equipment will be subjected to a static proof loading test to demonstrate design and fabrication adequacy. Applicable kit components will be functionally tested to demonstrate performance.

Interface Definition. The mass properties equipment kit mechanically interfaces with mounting holes on the spacecraft-capsule adapter ring and hooks on overhead cranes and hoists at the assembly and checkout facility.

MAGNETIC TEST FIXTURE, AHSE 5018



Functional Requirements. The magnetic test fixture is required to provide for the Voyager planetary vehicle rotation about the vertical axis, at a given rate in either direction, and rotation and locking about the horizontal axis for the perm-deperm phase of magnetic testing.

Design Requirements. All material is nonmagnetic including the gear boxes and bearings.

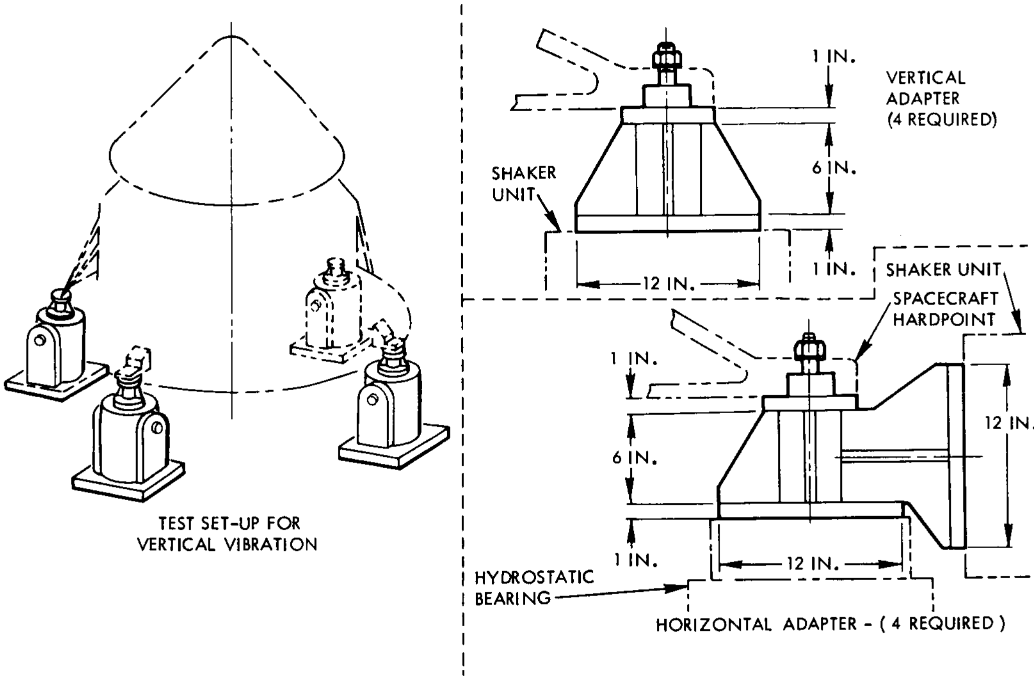
Description. The magnetic test fixture is composed of a basic "A" frame of 8-inch diameter aluminum pipe which can pivot the spacecraft about its horizontal axis. The spacecraft is attached to the "A" frame structure by interchangeable support arms so as to maintain the center of gravity of the spacecraft along the axis of rotation. (Same basic design as AHSE 5013 and 5057.)

Horizontal rotation is by a hand or power operated gear box which also serves as a locking device. Rotation about the vertical axis is by mounting the "A" frame structure on a hydraulically-driven thrust bearing. The thrust bearing acts as a rate table since the velocity and direction can be controlled as desired. The hydraulic power supply is located at a distance from the magnetic test envelope so as not to cause any magnetic interferences. The rate table controls are located near the trust bearing for convenience.

Test Requirements. The magnetic test fixture requires testing to insure that it is capable of supporting the required loads without excessive deflection, rotates at the required rates under load, does not exceed magnetic moment requirements, and the vehicle will fit and rotate without interference.

Interface Definitions. The magnetic test fixture interfaces with and must be compatible with the planetary vehicle and the magnetic test facility.

VIBRATION MACHINE ADAPTERS, AHSE 5019



Functional Requirements. Vibration machine adapters will be used for securing a completed assembled spacecraft to a vibration machine for testing in both vertical and horizontal directions.

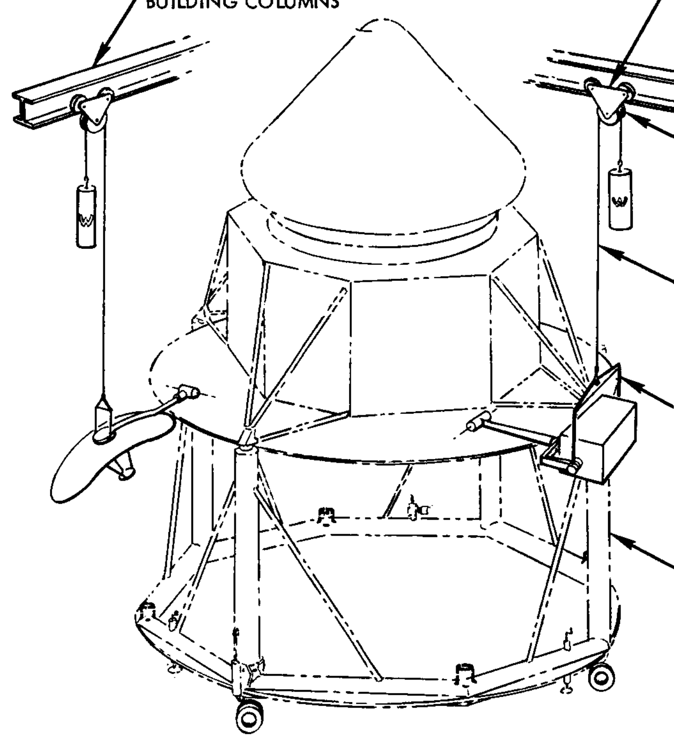
Design Requirements. The adapters are designed to support a load of 11,500 pounds.

Description. The vertical vibration adapters consist of four steel pads with fasteners on their upper surfaces for attaching to the spacecraft handpoints and holes in their lower surfaces for securing to vertical shaker units. The horizontal vibration adapters have hole patterns in their lower surfaces for securing to hydrostatic bearings. The adapters for horizontal vibration will have a means for connecting to horizontal shaker units.

Test Requirements. The adapters require fit checks, functional tests, and proof load tests.

Interface Definitions. Two adapter configurations are required, one for longitudinal vibration and the other for lateral vibration. For longitudinal vibration, adapters connect four shaker units to the aft end of the spacecraft at its hard points. For lateral vibration the spacecraft is supported on four hydrostatic bearings by means of adapters connecting to horizontally-mounted shaker units.

SPECIAL APPENDAGE DEPLOYMENT EQUIPMENT, AHSE 5020



Functional Requirements. Special appendage deployment equipment is required to support the spacecraft appendages (booms and antennas) during deployment tests. These appendages are deployed in a simulated zero-g field using live ordnance for initiating deployment action.

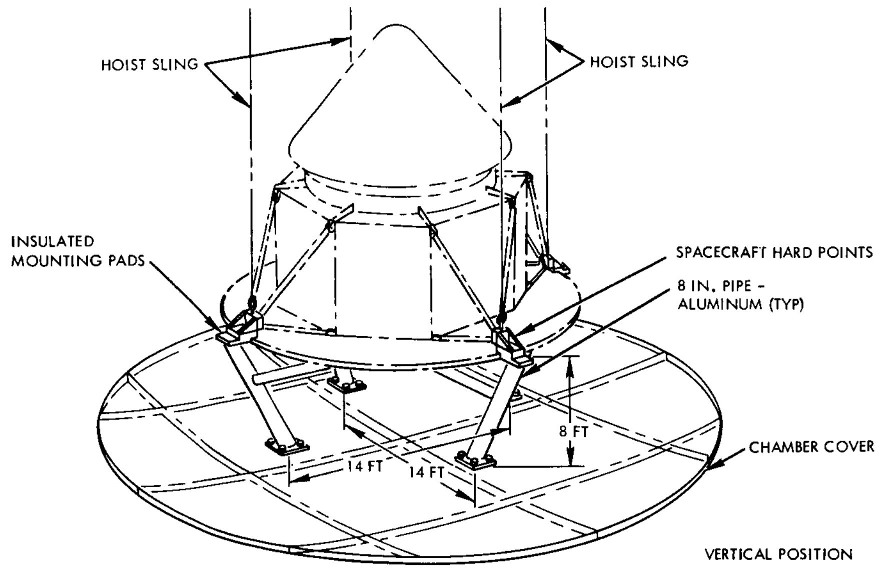
Design Requirements. Each appendage has a counterweight system that equalizes the moment generated during deployment.

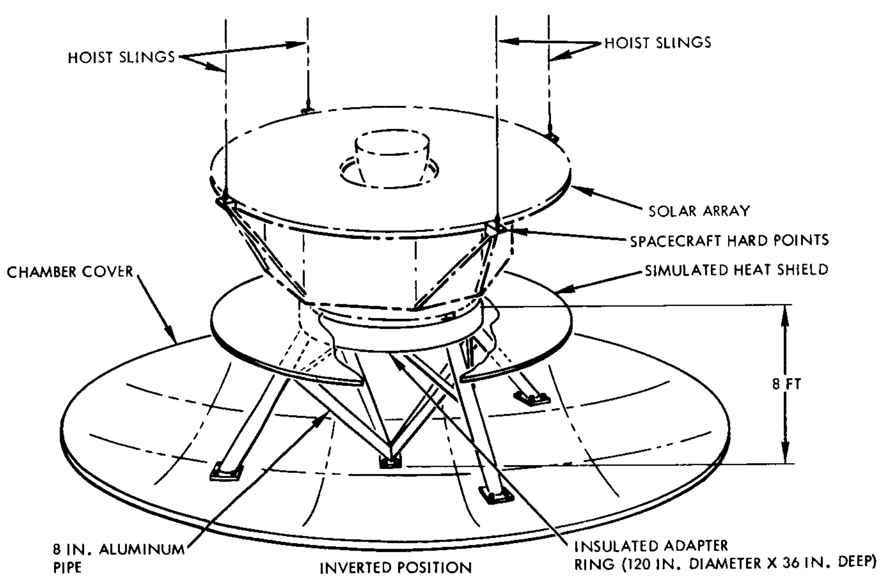
Description. The equipment consists of overhead-mounted tracks on which trolleys are free to move in two directions; one trolley is required for each appendage. Each trolley carries a pulley fitted with a ball bearing. A support cable attaches to the appendage being tested, rides over the pulley, and terminates in a counterweight.

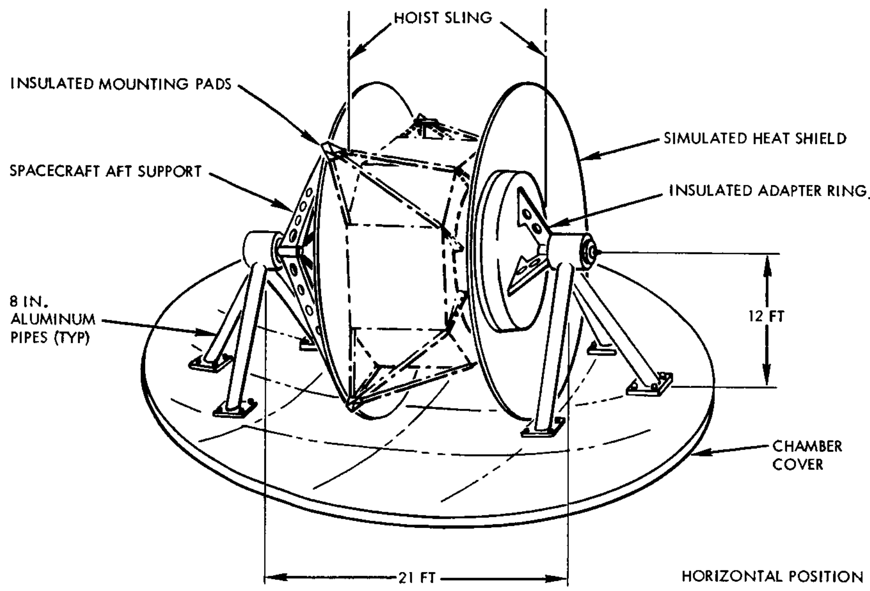
Test Requirements. The equipment will have fit checks, functional tests, and proof-load tests.

Interface Definitions. The special appendage deployment equipment supports the individual booms and antennas being deployed. No other contact is made with the spacecraft.

THERMAL VACUUM TEST ADAPTERS, AHSE 5021







Functional Requirements. The thermal vacuum test adapters are required for positioning the planetary vehicle on three axes during thermal-vacuum tests.

Design Requirements. All material must be selected for low temperature, outgassing, radiation, and conduction characteristics. Shields and/or paint must be utilized to reduce radiation effects. Heaters must be used to equalize the effect of conduction between the planetary vehicle and the thermal-vacuum test adapters.

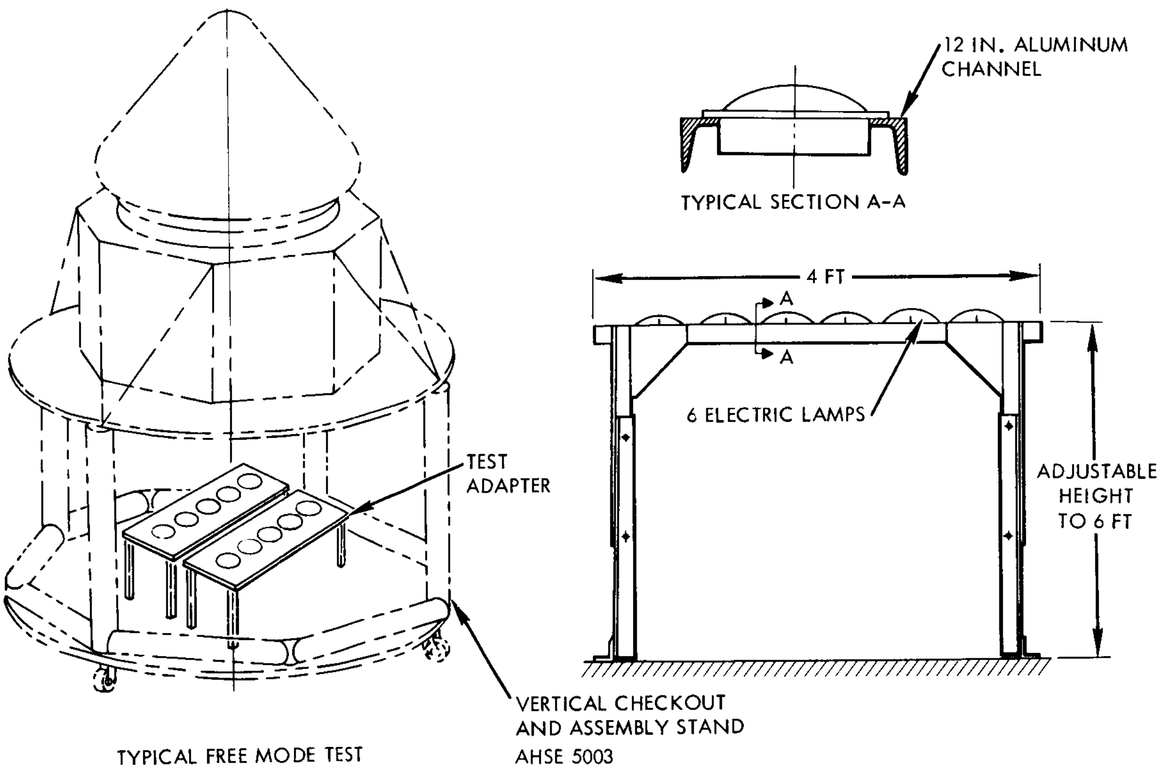
Description. Three separate fixtures, to support the planetary vehicle from the thermal vacuum cover on each of its three test positions (vertical inverted, and horizontal) are made of 8-inch diameter aluminum pipe and will be attached to the chamber cover with mounting pads. Insulated mounting pads or adapters between the basic structure and the spacecraft reduce heat conduction. Heater elements are used to equalize thermal conditions.

A simulated heat shield with a spacecraft insulated adapter ring is utilized in the inverted and horizontal positions. Heaters in the spacecraft insulated adapter ring equalize the thermal conditions. With the spacecraft in the horizontal position, the insulated adapter ring acts as the forward support section attaching to the capsule adapter hardpoints. Shields and/or paint are utilized to avoid radiation where necessary. The spacecraft or planetary vehicle is positioned on the planetary vehicle inverter, AHSE 5013, before it is installed in the thermal-vacuum test adapters.

Test Requirements. Tests must be performed on the thermal-vacuum test adapters to demonstrate design load and structural requirements, thermal conductivity and radiation effects, and compatibility requirements.

Interface Definitions. The thermal-vacuum test adapters are compatible with the spacecraft, planetary vehicle, and the thermal vacuum chamber and cover.

FREE MODE TEST ADAPTER, AHSE 5023



TYPICAL FREE MODE TEST
SET-UP

Functional Requirements. An adapter is required to illuminate the spacecraft solar arrays for free mode tests. These tests will demonstrate the capability of the electrical power system to function with sunlight as its only source of power.

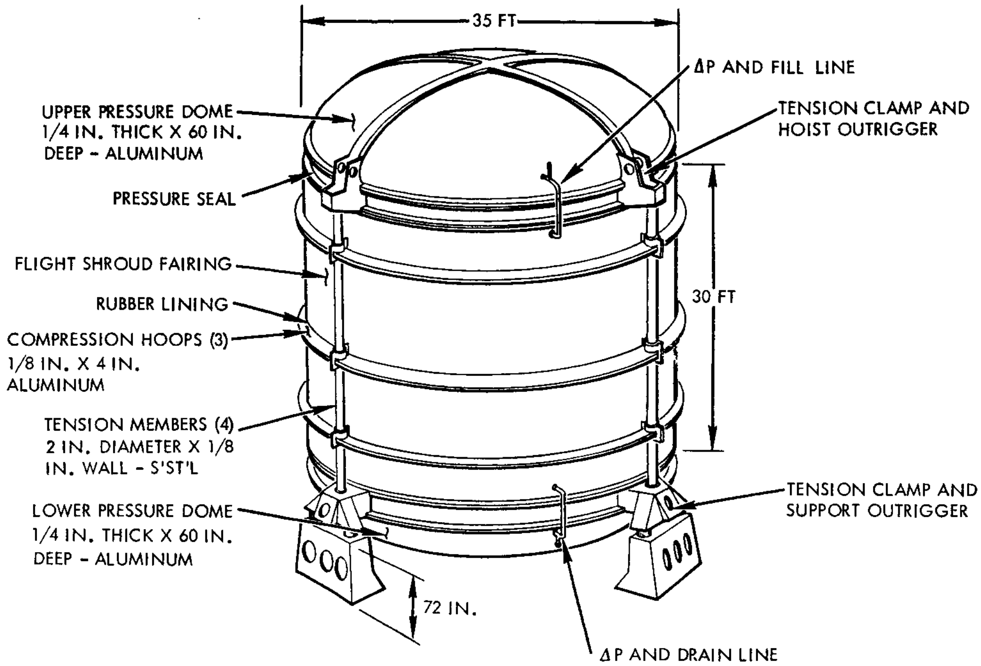
Design Requirements. The free mode test adapter provides simulated sunlight to illuminate the solar array panels. The solar array panel assembly is doughnut shaped with a 20-foot external diameter and a 5-foot inside diameter.

Description. The adapter consists of a channel-shaped section supported on adjustable angle legs. The channel has holes for mounting electrical lamps.

Test Requirements. The adapter requires fit checks and functional tests.

Interface Definitions. The free mode test adapter is compatible with the vertical checkout and assembly stand, AHSE 5003.

STERILIZATION PRESSURE DOME, AHSE 5025



Functional Requirements. A sterilization pressure dome is required to provide auxiliary paths for the pressurization loads imposed upon the booster nose fairing (flight shroud) and its biological barrier end seals during sterilization to prevent the rupture of the end seals and shroud.

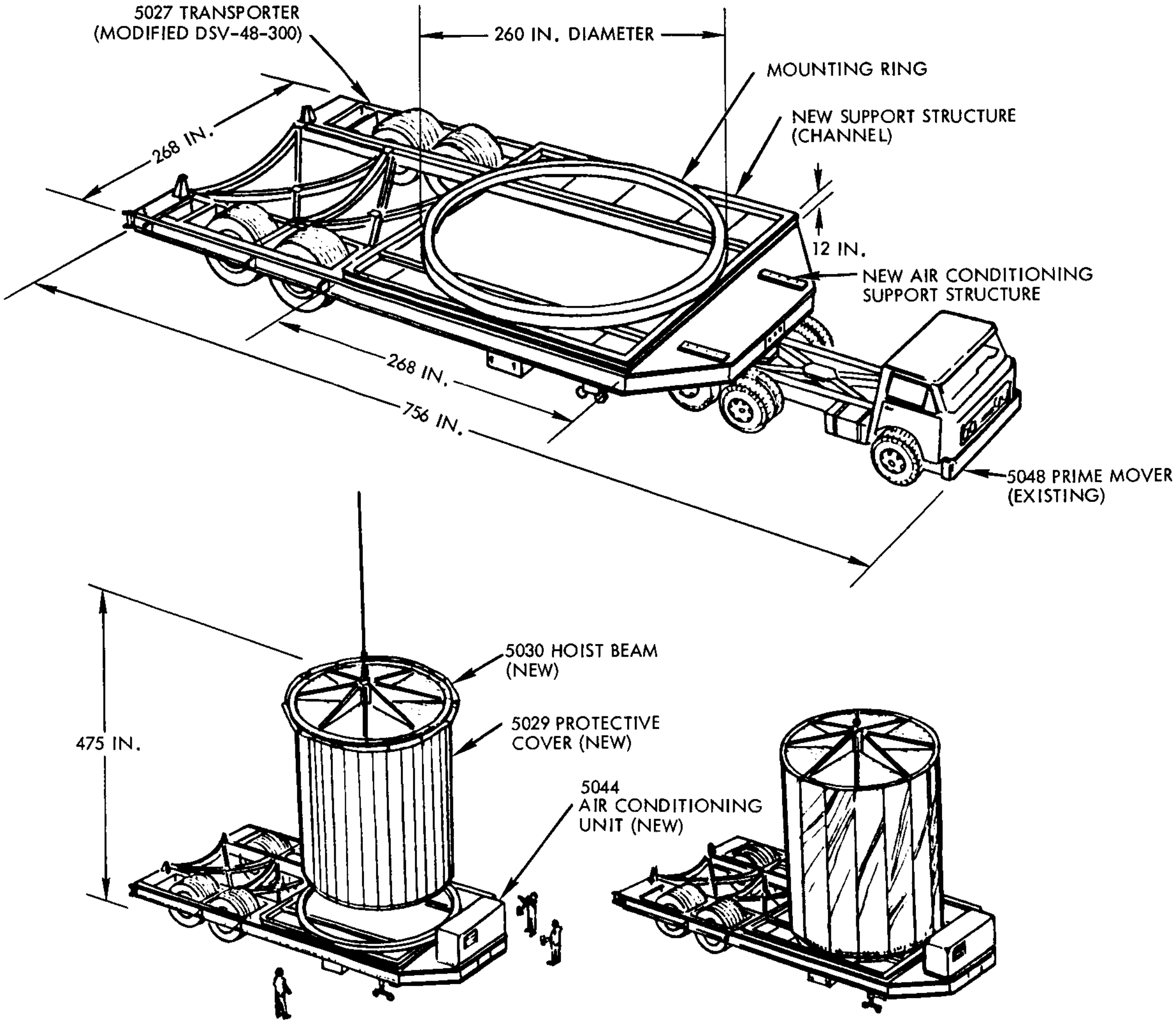
Design Requirements. The domes must be made of lightweight material either light allow sheet or fiberglas skin over light allow formers and stringers. All loads result from a 2-psig internal pressure on the entire assembly.

Description. The flight shroud and biological barrier end seals are pressurized through two domes sealing each end. The loads will be absorbed through four tension members constructed of 2-inch diameter stainless steel securing the domes to one another. Three compressions hoops are equally spaced along the tension members and cushioned from the shroud with rubber to absorb eht ehoop stresses generated. The entire assembly is mounted vertically on four pedestals.

Test Requirements. The upper and lower domes must be proof-pressure tested and leak checked. The entire assembly must be fit checked with the fairing to demonstrate compatibility and sealing capability.

Interface Definition. The sterilization pressure dome assembly interfaces with and must be compatible with the shroud, biological barrier end seals, and test facility.

TRANSPORTER GROUP, FLIGHT SHROUD-PLANETARY VEHICLE, AHSE: 5027, 5029, 5030, 5044 and 5048



Functional Requirements. Transportation handling and environmental protection must be provided the flight shroud-planetary vehicle combination at and between the VAB and ESA buildings and the launch pad at Cape Kennedy.

Design Requirements. The transporter is designed in accordance with MIL-M-008090D, Type II mobility. The transporter will not impose excess shock and vibration loads or resonant frequencies to the flight shroud-planetary vehicle combination during transportation. It is capable of supporting the fully-loaded (including all propellants and arming devices) combination during transit, and provides positive grounding.

The hoist beam assembly lifts the fully-loaded combination for loading and unloading operations to and from the transporter and during mating to the launch vehicle. It also serves as an environmental cover.

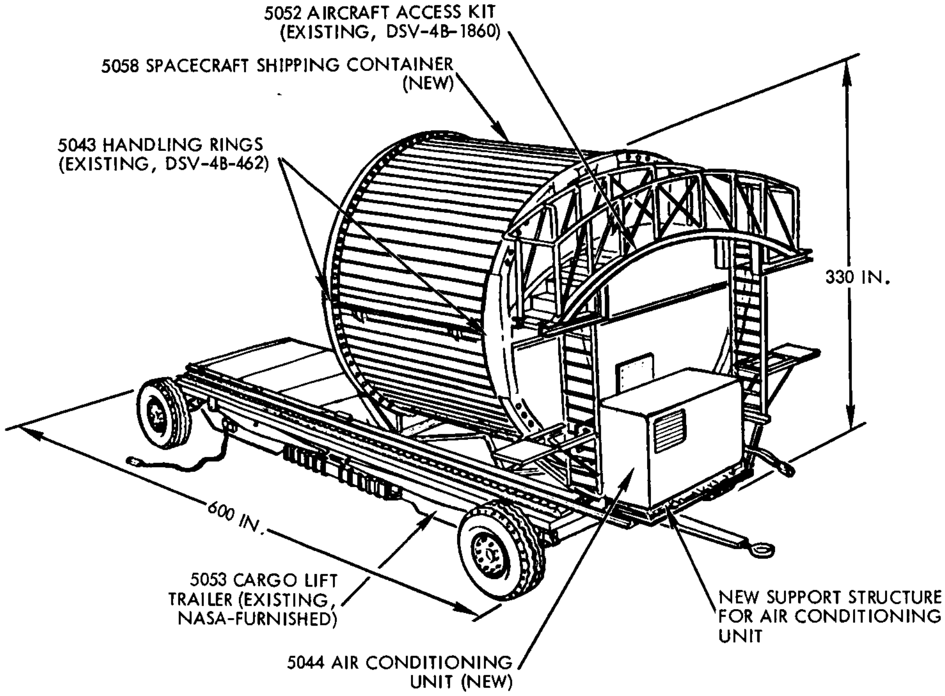
The protective cover (AHSE 5029) consists of a series of flaps that extend vertically from the hoist beam structure to the bottom of the flight shroud-planetary vehicle combination. The flaps attach to one another and to the transporter mounting ring by a series of quick release fasteners. Each flap is stored in a rolled up condition beneath the hoist beam structure. When in place, these flaps form a relatively tight weather barrier around the assembly under which conditioned air can be circulated.

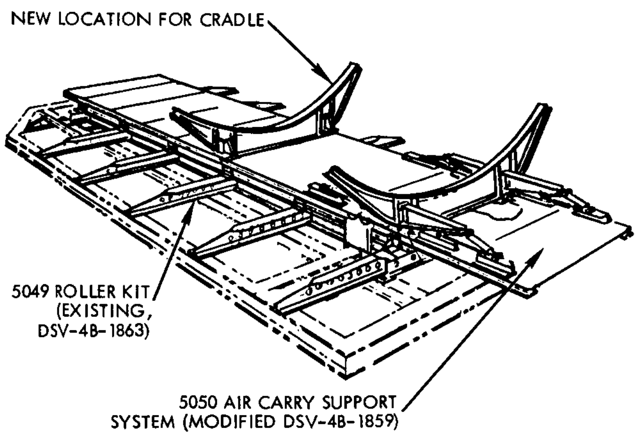
Description. The flight shroud-planetary vehicle transporter group consists of a modified Saturn S-IVB transporter (AHSE 5027, Saturn model DSV-4B-300), a circular hoist beam (AHSE 5030), a protective cover, (AHSE 5029) an air-conditioning unit (AHSE 5044) and a prime mover (AHSE 5048). Flight shroud-planetary vehicle load size and weight allows use of the Saturn S-IVB transporter with minor modifications. Modifications consist of adding a mounting rind and a structural framework to distribute the flight shroud-planetary vehicle loads to the main load-carrying members on the transporter.

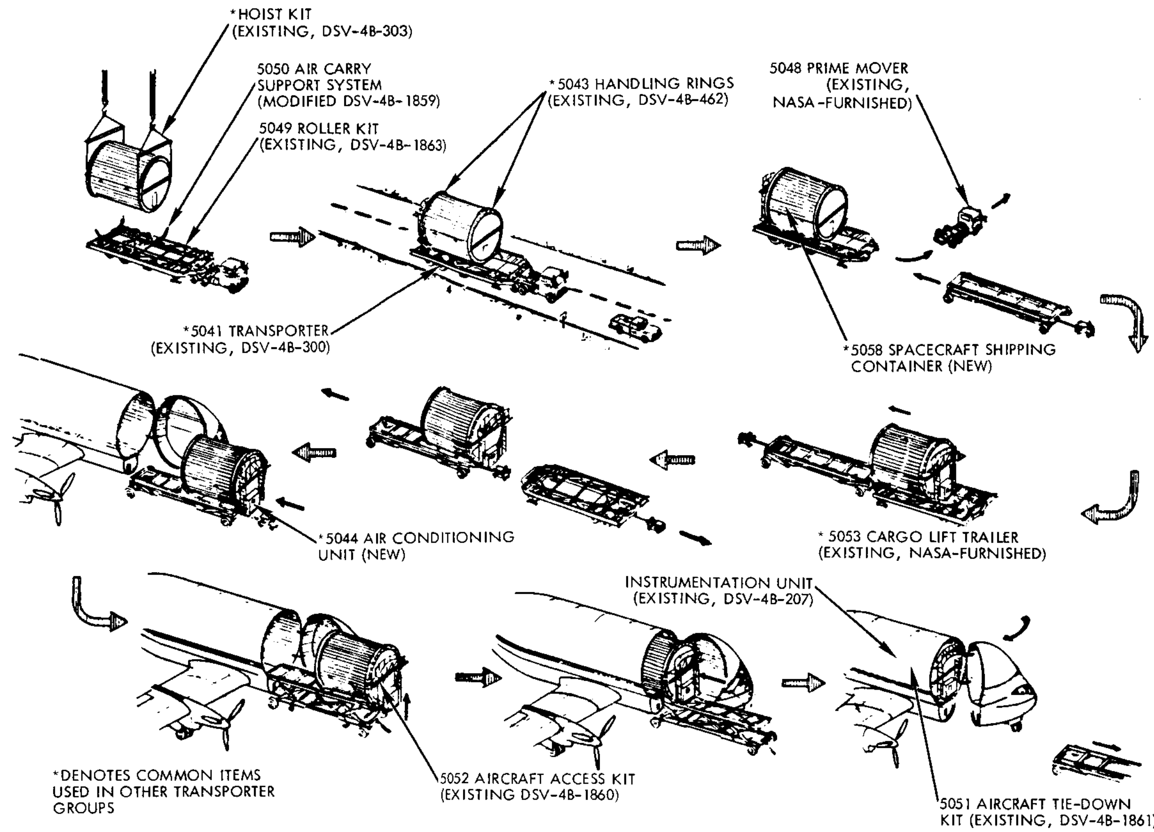
Test Requirements. The flight shroud-planetary vehicle transporter group will be functionally and proof load tested. The protective cover seal design will be tested with the air-conditioning unit to demonstrate capability to meet the environmental conditioning requirements beneath the cover under ambient ranges normally expected at Cape Kennedy.

Interface Definition. The flight shroud-planetary vehicle transporter group interfaces with the field joint mating holes of the flight shroud for both mounting the assembly to the transporter mounting ring and for attaching the hoist beam and protective cover assembly to the flight shroud-planetary vehicle.

TRANSPORTER GROUP, SUPER GUPPY
AHSE 5032, 5033, 5043, 5044, 5049, 5050, 5051, 5052, 5053







Functional Requirements. The spacecraft may be transported by suitable cargo aircraft to cross-country destinations. The spacecraft will be complely assembled (dry) and mounted and sealed within a shipping container while being transported. Stabilized support, handling, and transfer loading equipment must be provided for loading and unloading the aircraft with the spacecraft shipping container. The spacecraft must be environmentally controlled while being transferred into, secured within, and transported by the aircraft. All loads imposed upon the spacecraft during these operations must be monitored and recorded.

Design Requirements. The handling equipment and transfer procedures must not impose excess shock and vibration loads or resonant frequencies on the spacecraft. The transported group functions at altitudes from sea level to 30,000 feet. A temperature, humidity, and cleanliness-controlled environment in accordance with Federal Standard No. 209, Class 100,000, is provided within the spacecraft shipping container. The air conditioning equipment is mounted so that it remains connected to the shipping container during all handling operations. An instrumentation unit monitors and records the load history imposed on the spacecraft.

Description. The Super Guppy transporter group consists of a Super Guppy aircraft; a cargo-lift trailer (AHSE 5053); a roller kit (AHSE 5049); an air-carry support system (AHSE 5050); an aircraft access kit (AHSE 5052); an aircraft tie-down kit (AHSE 5051); handling rings (AHSE 5043);

a hoist kit (AHSE 5032); an instrumentation kit (AHSE 5033); an air conditioning unit (AHSE 5044); an umbilical assembly; and a spacecraft shipping container (AHSE 5058). The size and weight of the Voyager spacecraft allows use of most of the Saturn S-IVB Stage SG transporting equipment with minor modifications.

The cargo-lift trailer (AHSE 5053) is government-furnished equipment and requires no modification. The aircraft access kit (AHSE 5052, Saturn model DSV-4B-1860) and the aircraft tie-down kit (AHSE 5051, Saturn model DSV-4B-1861) require no modification. The Saturn S-IVB stage handling rings (AHSE 5043, Saturn model DSV-4B-462) and the Saturn S-IVB stage hoist kit (AHSE 5032, Saturn model DSV-4B-303) require no modification.

The Saturn S-IVB roller kit (AHSE 5049, Saturn model DSV-4B-1863) allows for transfer of the spacecraft shipping container from the Saturn S-IVB transporter (AHSE 5027, Saturn model DSV-4B-400) to the cargo lift trailer. The roller kit is modified to accommodate new cradle assembly locations and an air-conditioning unit for the spacecraft shipping container.

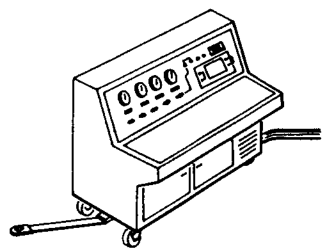
The air-carry support system (AHSE 5050, Saturn model DSV-4B-1859) supports the spacecraft shipping container on the road transporter, the cargo-lift trailer, and in the Super Guppy. The air-carry support system is modified to accommodate new cradle locations and the air conditioning unit for the spacecraft shipping container.

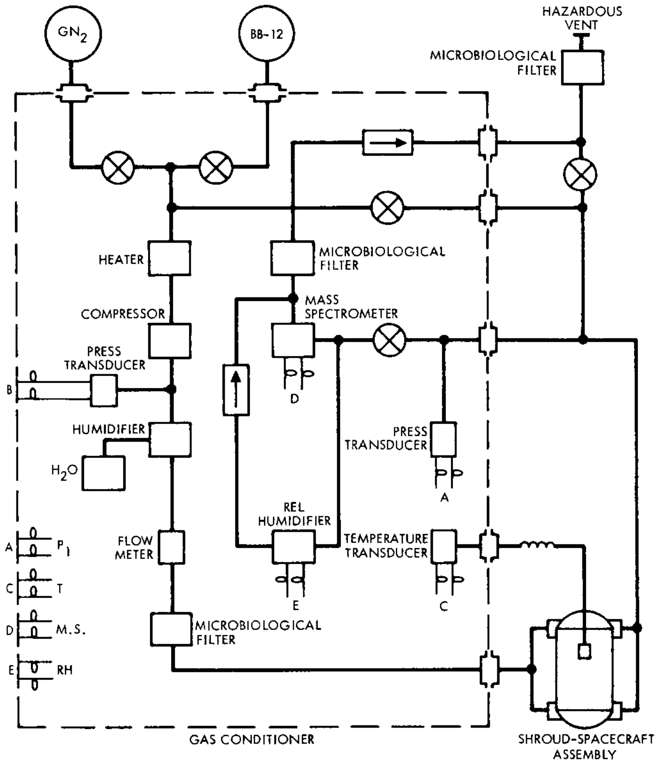
The Saturn S-IVB Stage instrumentation unit (AHSE 5033, Saturn model DSV-4B-207) consists of an acceleration switch alarm; a multiple-channel strain-gage type signal conditioner; humidity and temperature indicators; a tape recorder, and a patch panel for varying the channels on the strain-gage signal conditioner. The instrumentation unit probably will require no modification.

Test Requirements. The transporter group is functionally and proof-load tested, and the environmental conditioning equipment is qualified to assure its performance within the ambient conditions it experiences during air transportation.

Interface Requirements. The Super Guppy transportation group interfaces with the spacecraft shipping container, the Super Guppy loading rails, and the land transporter equipment group.

SPACECRAFT STERILIZATION UNIT, AHSE 5034


CONTROL CONSOLE



Functional Requirements. The sterilization unit is required to purge each shroud-spacecraft assembly with sterile warm GN_2, charge it with a sterilizing gas mixture, and repurge it with sterile GN_2.

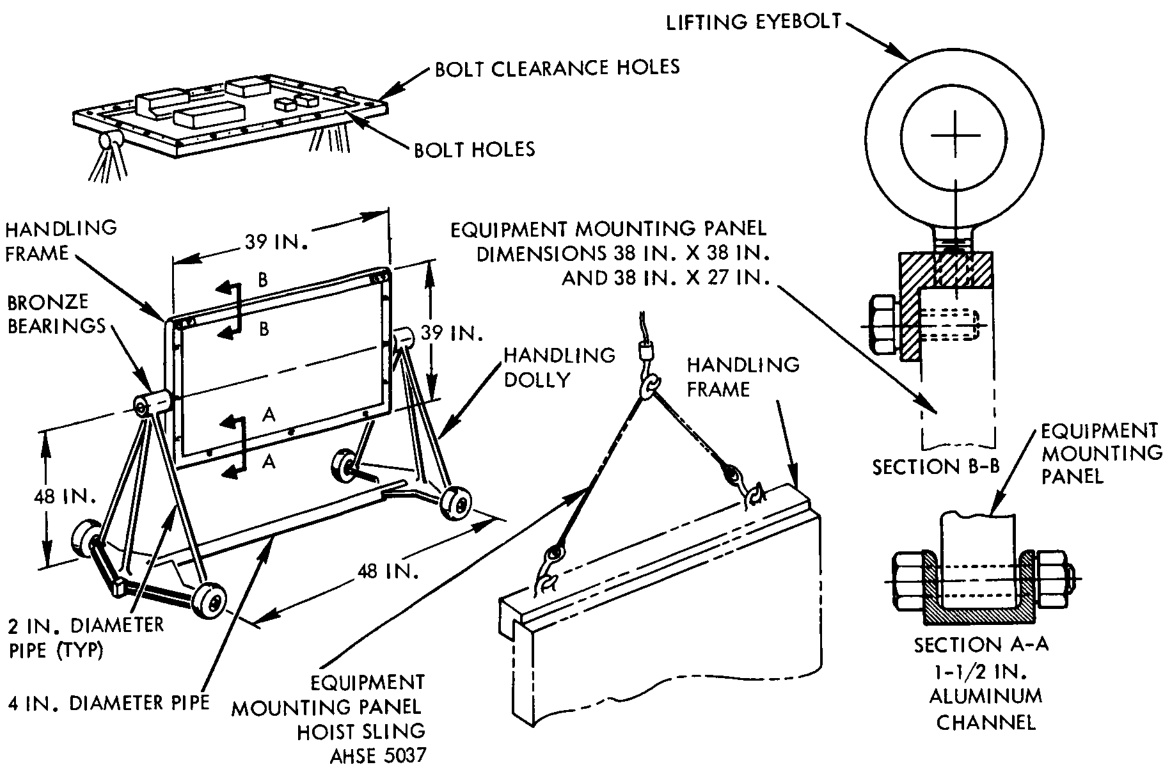
Design Requirements. The sterilization unit must include microbiological filters in all gas lines leading into or out of the sealed shroud. It must heat and charge the shroud with sterile gaseous nitrogen, circulate a properly humidified (50 ± 5% RH) sterilizing gas at a controlled temperature (100 ± 5°F) and pressure (2 ± 10%) for at least 12 hours, and purge the shroud with sterile cool gaseous nitrogen. The unit must monitor, regulate, and record the shroud internal temperature, pressure, and humidity, and the composition of the sterilizing gas. It must be remotely controlled and include redundant safety features to prevent hazard to operating personnel.

Description. The sterilization unit consists of a remote control and monitoring console, a gas control and conditioning unit, and quick-acting connections to the shroud-spacecraft assembly and sources of LN_2 and Freon-ethylene oxide (88-12).

Test Requirements. The sterilization unit will be leak and pressure tested and functionally tested.

Interface Definition. The sterilization unit has pneumatic and instrumentation interfaces with the shroud-spacecraft assembly.

EQUIPMENT MOUNTING PANEL HANDLING FIXTURE, AHSE 5036



Functional Requirements. A fixture is required to support and handle the equipment mounting panels and to provide in-plant mobility.

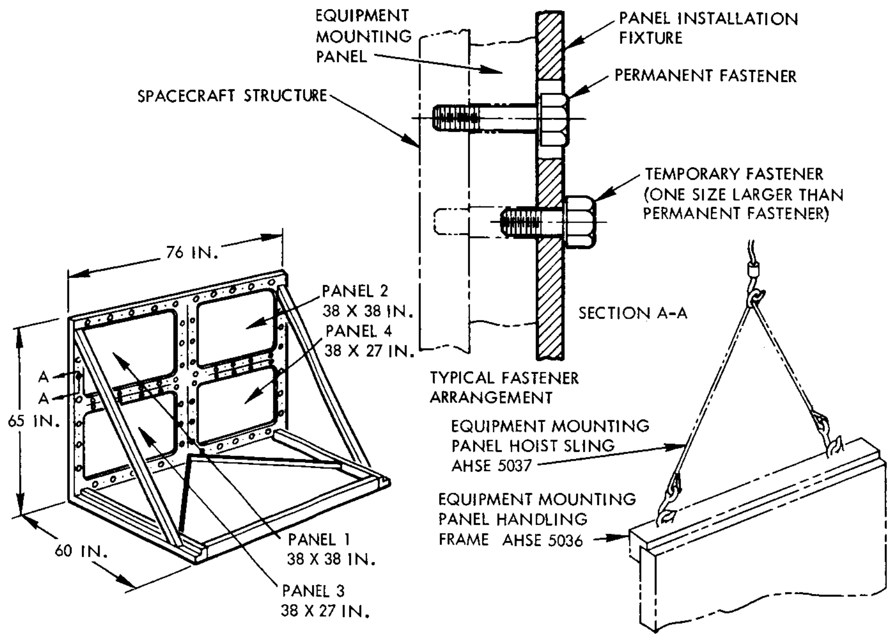
Design Requirements. The handling fixture must support individual equipment panels and adjust to panels of varying outside dimensions. It must rotate the panels 360 degrees and hold them at any angle during rotation.

Description. The fixture consists of a tubular structure mounted on four rubber tired wheels. The structure carries bronze bearings on which a panel holding frame rotates. This frame surround the panel and supports it at its outer edges. Eyebolts are provided for handling the panels by means of a hoist sling. Bolt clearance holes are provided for transfer of panels to the equipment mounting panel installation fixture (AHSE 5038).

Test Requirements. The fixture requires fit checks and functional tests.

Interface Definitions. The fixture is compatible with two sizes of panels 38 x 38 x 1 inch and 38 x 27 x 1 inch. It interfaces with the equipment mounting panel hoist sling (AHSE 5037), and is used in conjunction with the equipment mounting panel installation fixture (AHSE 5038).

EQUIPMENT MOUNTING PANEL INSTALLATION FIXTURE, AHSE 5038



Functional Requirements. A fixture is required to hold four equipment mounting panels in a position similar to their ultimate spacecraft position. The fixture must be capable of being manually adjusted and secured on top of the spacecraft aft equipment mounting module.

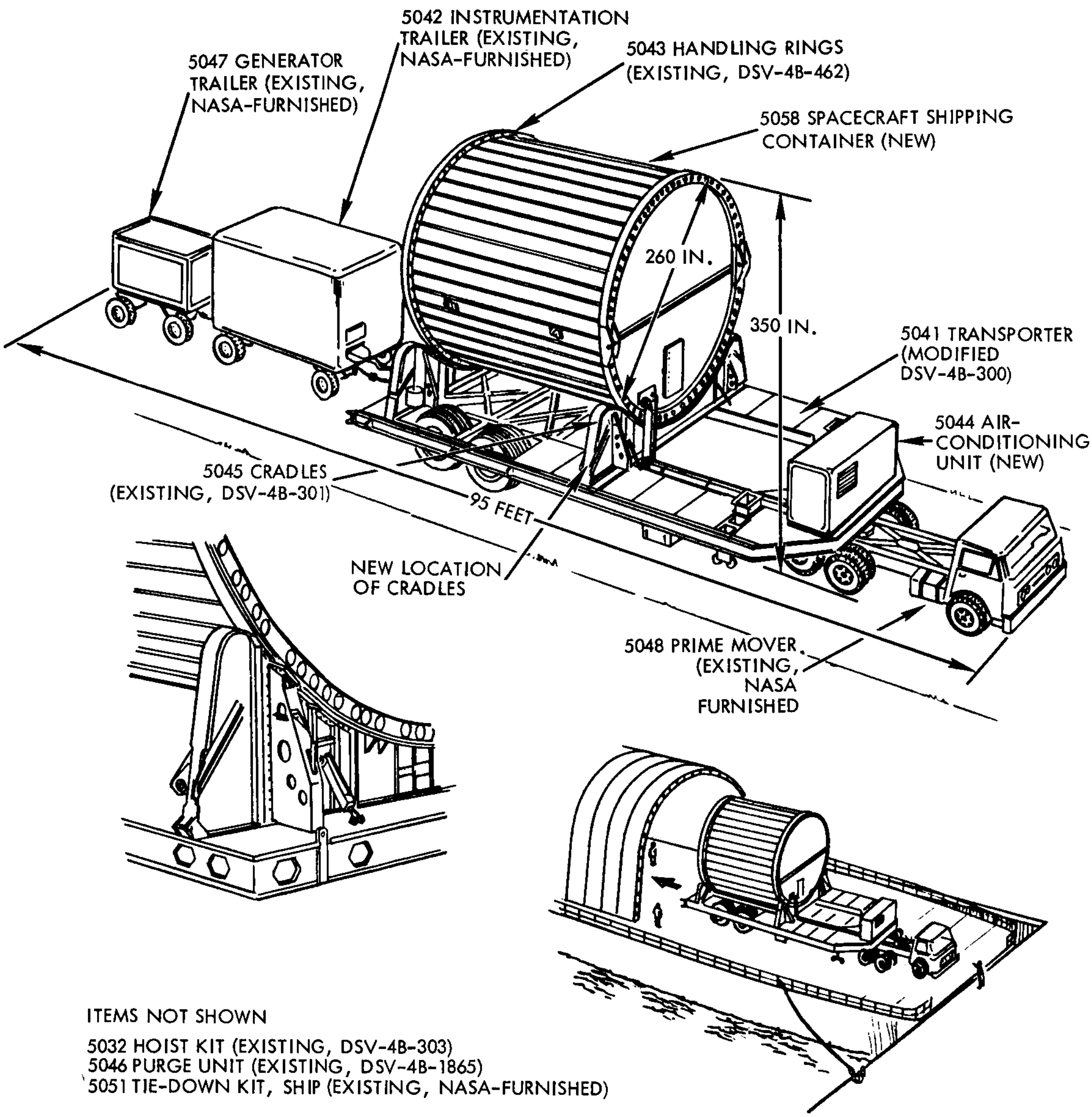
Design Requirements. The fixture must support four equipment mounting panels. Its base is stable and the center of gravity of the loaded fixture is located in a manner to prevent overturning. Skid surfaces are provided on the lower surface where contact is made with the spacecraft. Connection provisions allow the panels to be simultaneously secured to the spacecraft and equipment mounting panels.

Description. The fixture consists of a lightweight structural frame mounted on skids which are lined with Teflon or similar soft material. The frame is held in a vertical attitude by bracing members Two hole patterns are provided to match holes in the equipment mounting panels, one set for bolting the panels to the fixture and one set of clearance holes to allow attachment of the panels to the spacecraft structure.

Test Requirements. The fixture requires fit checks and functional tests to demonstrate its capability.

Interface Definitions. The fixture interfaces with a set of four equipment mounting panels and is compatible with the equipment mounting panel handling fixture (AHSE 5036). An equipment mounting panel hoist sling (AHSE 5037), is used in conjunction with the installation fixture for handling the panels.

TRANSPORTER GROUP, LAND AND SEA, AHSE 5041, 5042, 5043, 5044, 5045, 5047, 5048, 5058



Functional Requirements. The land and sea transporter group provides means of transporting the fully assembled (dry) spacecraft, mounted in its shipping container, between various assembly, checkout, test, and integration facilities in southern California, New Mexico, and Cape Kennedy.

Design Requirements. The equipment and operating procedures must not impose excess shock and vibration loads or resonant frequencies on the spacecraft during transit, loading, or unloading. The transporter must be designed in accordance with MIL-M-008090D Type II mobility for over-the-road speeds of up to 20 mph. The transporter group must provide a temperature, humidity, and cleanliness controlled environment in accordance with Federal Standard No. 209, Class 100,000, within the spacecraft shipping container. Equipment must be provided to purge the interior of the spacecraft shipping container to a $-20^{\circ}F$ dew point. A mobile instrumentation unit must be provided for shock, vibration, temperature, and humidity monitoring and recording. A mobile generator trailer must be furnished with primary and secondary power sources. A means must be provided to mount and stabilize the land transporter to a NASA-supplied water carrier during sea shipment.

Description. The equipment consists of a transporter (AHSE 5041); cradles (AHSE 5045); handling rings (AHSE 5043); a hoist kit (AHSE 5032); an instrument trailer (AHSE 5042); a generator trailer (AHSE 5047); an air-conditioning unit (AHSE 5044); a purge unit (AHSE 5046); an AKD barge tie-down kit (AHSE 5051); a transporter prime mover (AHSE 5048); and a spacecraft shipping container (AHSE 5058). The Voyager spacecraft size and weight allows the use of Saturn S-IVB land and sea transporting equipment with minor modifications and additions.

The frame of the Saturn S-IVB transporter (AHSE 5041, Saturn model DSV-4B-300) is modified to support the Voyager spacecraft shipping container by relocating the cradle support installation points.

The Saturn S-IVB stage cradles (AHSE 5045, Saturn model DSV-4B-301) can be used without modification except that the location of the cradles on the transporter will be changed.

The Saturn S-IVB handling rings (AHSE 5043, Saturn model DSV-4B-462) and hoist kit (AHSE 5032, Saturn model DSV-4B-303) need no modification.

The instrumentation trailer (AHSE 5042, NASA model 5146-1) and generator trailer (AHSE 5047, NASA model 5146-9) for land and sea transportation are both GFE and probably require no modifications.

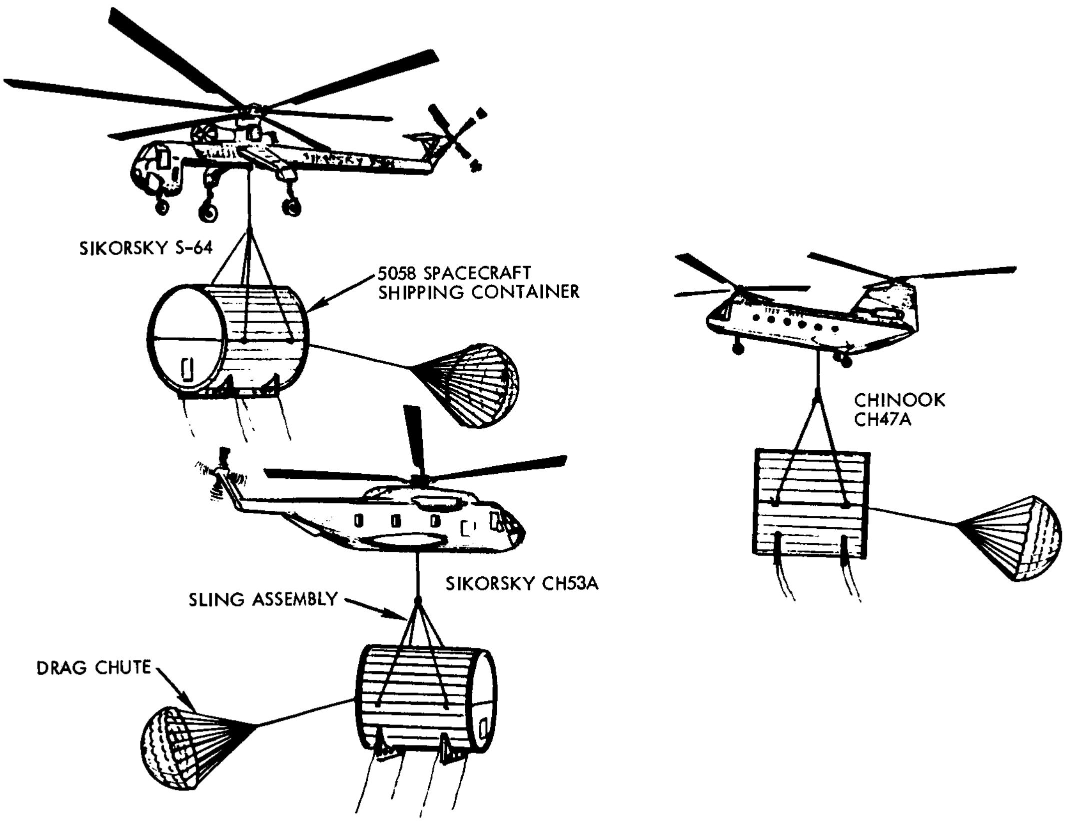
The air conditioning unit (AHSE 5044) maintains a temperature of $72 \pm 5^{\circ}F$, a relative humidity of $30 \pm 5\%$, and a total dust particle count within the spacecraft shipping container in accordance with Class 100,000 clean area specified in Federal Standard No. 209. The air-conditioning unit is mounted on the bed of the transporter and is a new equipment item.

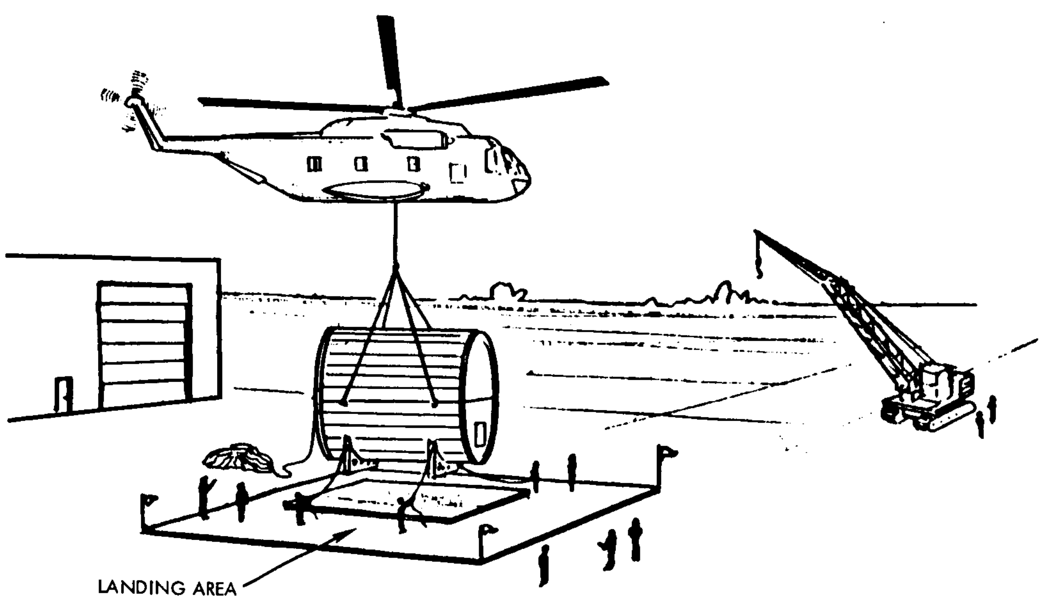
The Saturn S-IVB purge unit (AHSE 5046, Saturn model DSV-4B-1865) requires no modification. The AKD barge tie-down kit (AHSE 5054, NASA models 90M-01901, 2, 9) is GFE and needs no modification. The Saturn transporter prime mover (AHSE 5048) is GFE and requires no modification.

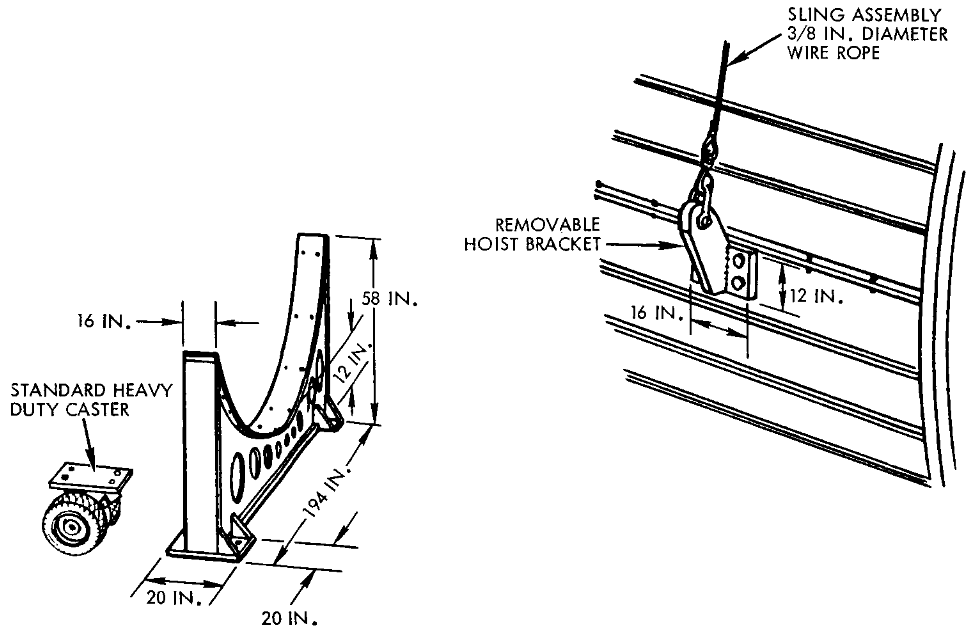
Test Requirements. The transporter group will be functionally and proof-load tested.

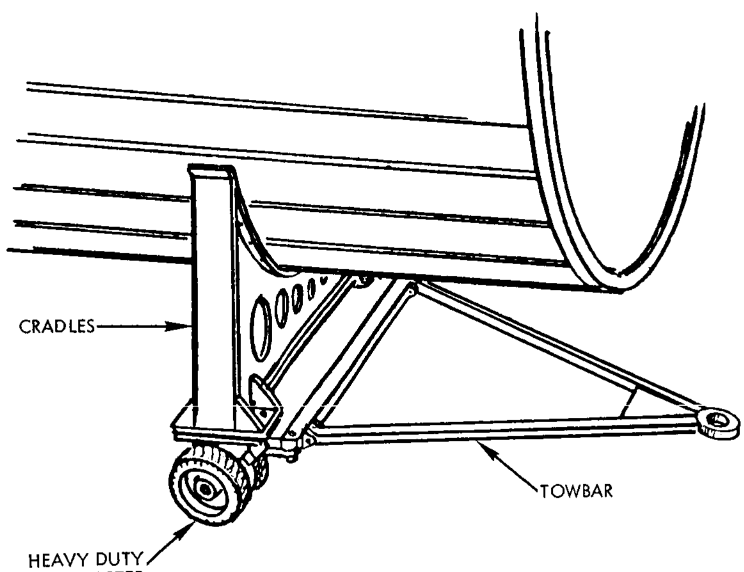
Interface Requirements. The group interfaces with the spacecraft shipping container, commercial handling equipment for loading commercial cargo ships and barges, and any necessary facilities equipment.

TRANSPORTER GROUP, HELICOPTER, 5055 and 5056, 5058









Functional Requirements. The spacecraft will be transported by helicopter over distances generally not to exceed 60 miles and at an altitude less than 10,000 feet. The spacecraft will be completely assembled (dry) and mounted and sealed within a shipping container. Stabilized support and limited mobility for the spacecraft shipping container must be provided for its movement between facility work stations and the helicopter loading point.

Design Requirements. The transporter group equipment and procedures must not impose excess shock and vibration loads or resonant frequencies on the spacecraft during transit, loading, or unloading. Shock is reduced by lowering the spacecraft at a rate not exceeding 2 ft/sec onto a shock-mounted facility platform or resilient pad. The shipping container has steerable wheel-mounted stabilizing assemblies. An instrumentation unit in the helicopter monitors and records shock and vibration imposed on the spacecraft during transportation. It operates from aircraft power, with a secondary self-contained power source.

Description. The helicopter transportation group consists of a Sikorsky 64-A, Sikorsky CH-53A, or a Chinook helicopter; a handling and rigging kit (AHSE 5055); an instrumentation unit (AHSE 5056); and a spacecraft shipping container (AHSE 5058). The handling and rigging kit consists of a 25,000-pound nylon bungee rope, a low response hoist sling, a drag chute, tag lines, removable shipping container cradles and brackets, and casters which attach to the cradles.

During helicopter liftoff and touchdown and intra-facility mobility the shipping container is stabilized by the removable under-cradles and the rubber-tired caster assemblies. These cradles and the hoist brackets are bolted to the surface of the shipping container at external stations coincident with the main internal structural beams.

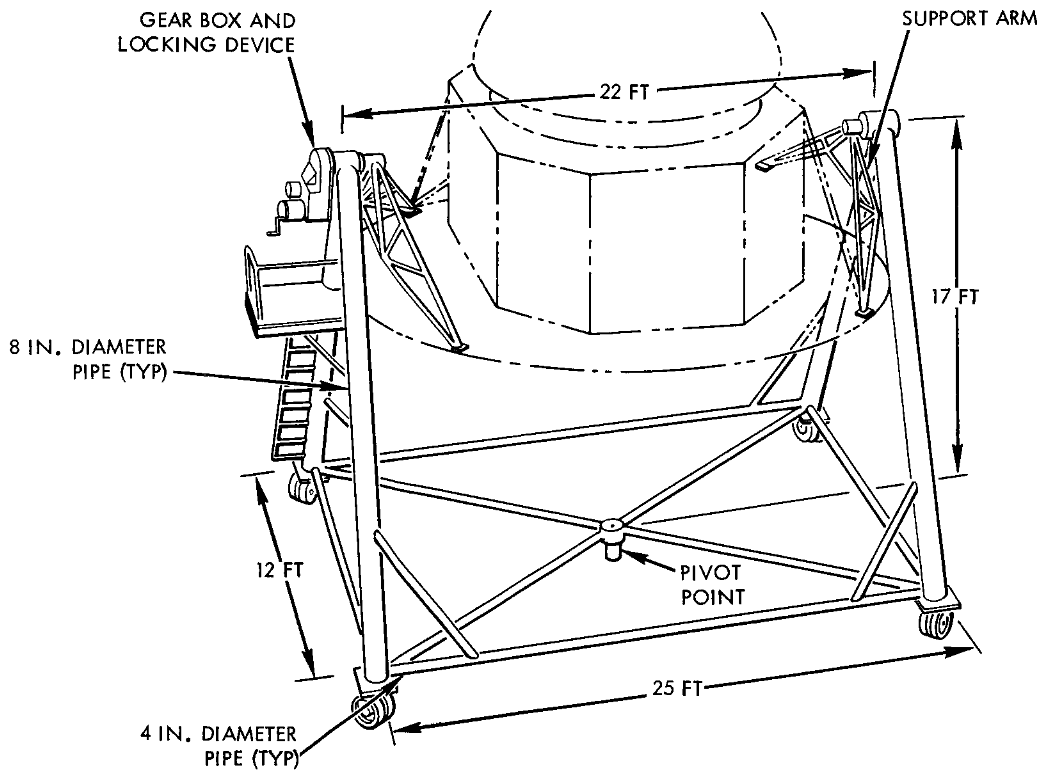
A removable drag chute is attached to the shipping container to provide aerodynamic stability and reduce wind load vibrations during transit.

The instrumentation unit (AHSE 5056) consists of an acceleration switch alarm, a multiple channel, gage-type signal conditioner, a tape recorder, and a patch panel for varying the channels on the strain gage signal conditioner.

Test Requirements. The transportation group will be functionally and proof-load tested. The group will be qualified with the helicopter during development phaes by type approval tests using instrumented dummy loads.

Interface Definition. The helicopter transportation group interfaces with the spacecraft shipping container, the facilities area, a towing vehicle, and any necessary ground hoisting and rigging equipment.

MAGNETIC FACILITY ADAPTER, AHSE 5057



Functional Requirements. The magnetic facility adapter is required to support and rotate the Voyager planetary vehicle about two axes and lock it in any position for magnetic tests.

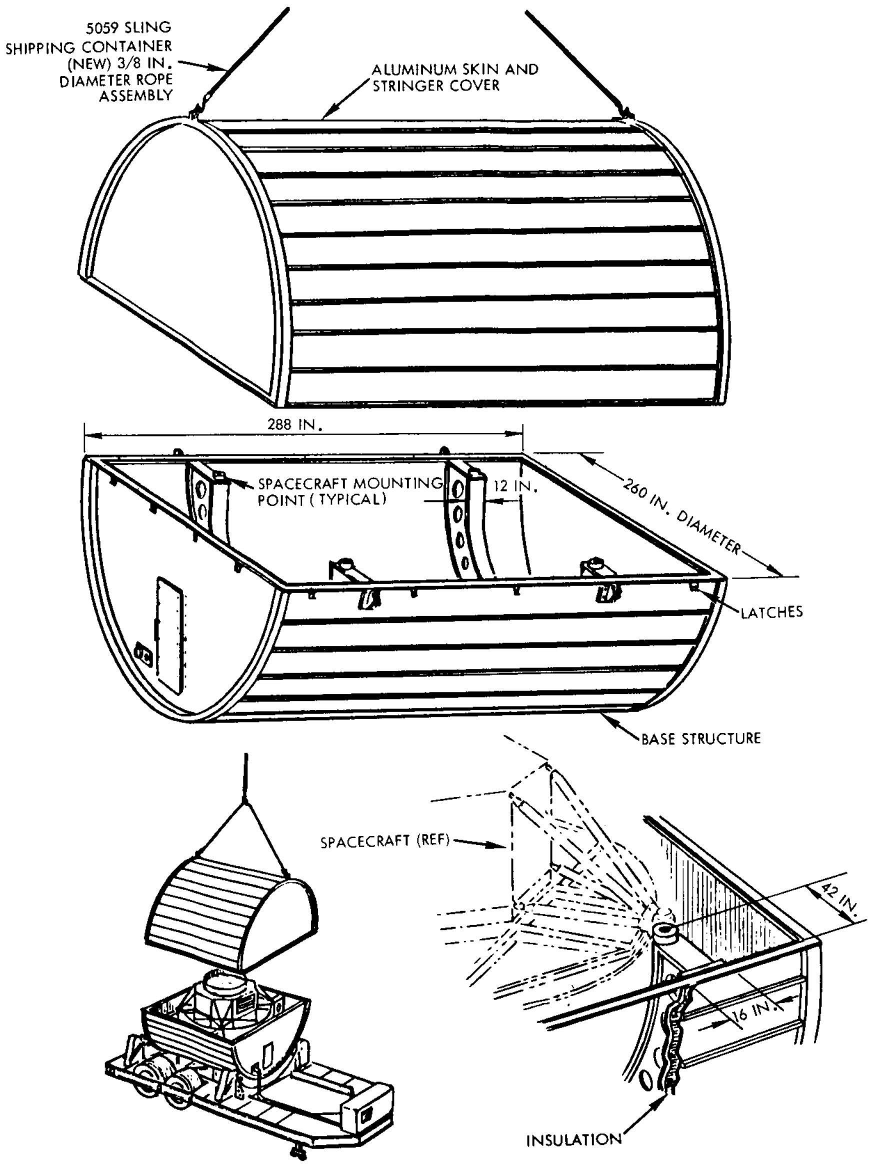
Design Requirements. All material must be nonmagnetic including the gear box, bearings, and attaching points.

Description. The magnetic facility adapter is composed of a basic "A" frame constructed of 8-inch diameter aluminum pipe which pivots about a spindle on its center line at the base. The planetary vehicle is supported by two sets of support arms so that it will rotate about its center of gravity in the horizontal axis. The support arms are attached to the "A" frame at bearing points and are rotated by a gear box which also serves as a locking and positioning device (Same basic design as AHSE 5013.)

Test Requirements. The magnetic facility adapter requires testing to insure that it is capable of supporting the required loads, does not exceed magnetic requirements, and that the planetary vehicle will fit and rotate without interference.

Interface Definitions. The magnetic facility adapter interfaces with and must be compatible with the planetary vehicle and the magnetic test facility.

CONTAINER, SHIPPING, SPACECRAFT, AHSE 5058



Functional Requirements. The spacecraft, during storage and shipment must be maintained in its flight attitude and protected.

Design Requirements. The shipping container protects the spacecraft during transportation and storage and mounts it in its flight attitude. It provides environmental protection to meet temperature, humidity, and cleanliness-controlled environments in accordance with Federal Standard 209 for Class 100,000 cleanliness. The shipping container provides mounting holes on each end for attaching the Saturn S-IVB forward and aft handling rings (Model DSV-4B-462). These handling rings (AHSE 5053) are used to mount the shipping container on the land and sea transporter (AHSE 5041 and 5045) and on the air-carry support system (AHSE 5050). The shipping container must be compatible with all transportation modes (e.g., air, land, or sea transporter groups).

Description. The spacecraft shipping container consists of two cylindrical half-sections: a base structure and a cover, totaling 260 inches in diameter and 288 inches in length. The two half-sections are closed at their ends and joined by a series of latches forming an airtight and watertight seam.

Test Requirements. The shipping container will be proof-load tested to demonstrate its design and fabrication adequacy, and will be functionally tested to demonstrate its required performance in supporting the spacecraft and maintaining required internal environment.

Interface Definition. The spacecraft shipping container interfaces with the four mounting fixtures on the spacecraft outriggers which mate to the space vehicle flight shroud; the mounting holes on the forward and aft handling rings (AHSE 5053); the inlet and outlet ducts on the portable air conditioning unit (AHSE 5054) used during land, sea, and Super Guppy transportation; the electrical cables on the instrumentation trailer (AHSE 5042) which monitor and record the load history on the spacecraft during transportation by land, sea, and air; and the shipping container sling (AHSE 5059) used for handling the container without the forward and aft handling rings attached.

4. SUBSYSTEM FUNCTIONAL DESCRIPTIONS

Table 5 covers only those items of subsystem AHSE required for system assembly, checkout, and test. The subsystem AHSE that is peculiar to individual subsystem development, test, and fabrication will be defined in detail in Phase IB.

As in the system level functional descriptions, only major subsystem AHSE noted in the general specification are included in the set of subsystem functional descriptions (noted by * in Table 5). Subsystem AHSE which are GFE are noted with part numbers where possible, without functional descriptions.

A major number of existing items of AHSE for the LEM (identified as GFE/GAEC in the table) have capabilities beyond Voyager needs. Our approach to AHSE makes maximum use of these items to take advantage of existing designs, but only the Voyager-peculiar portions of these items are specified as requirements.

Table 5. Subsystem Assembly, Handling, and Shipping Equipment

Subsystem	AHSE No.	Item	Source	Quantity
S-Band Communication	* 5101	S-Band Subsystem Electronics Shipping Container	TRW	3
	* 5102	High-Gain Antenna Container	TRW	3
	5103	High-Gain Antenna Sling	TRW	3
	* 5104	Medium-Gain Antenna Container	TRW	3
	5105	Medium-Gain Antenna Sling	TRW	3
	* 5106	Low-Gain Antenna Container	TRW	3
Capsule Relay Link	* 5201	Relay Link Antenna Shipping Container	TRW	3
Command	* 5301	Command Subsystem Shipping Container	TRW	3
Computing and Sequencing	* 5401	Computer and Sequencer System Shipping Container	TRW	3
Telemetry	* 5501	Telemetry Subsystem Shipping Container	TRW	3
Data Storage	* 5601	Data Storage Subsystem Shipping Container	TRW	3
Guidance and Control	* 5701	Guidance and Control Subsystem Shipping Container	TRW	3
	5702	Guidance and Control Subsystem Shipping Container Sling	TRW	3
	* 5703	Reaction Control Pressure Vessel Handling Fixture	TRW	6
	* 5704	Reaction Control Pressure Vessel Handling Sling	TRW	3
	5705	Pneumatic Test Set	DAC	2
Power	* 5801	Power Subsystem Shipping Container	TRW	3
	* 5802	Solar Array Mounting Fixture	TRW	3
	* 5803	Solar Array Handling Dolly	RCA	3
	* 5804	Solar Array H isting Sling	RCA	3
	* 5805	Solar Array Protective Covers	RCA	4
	5806	Dummy Solar Arrays	RCA	6
	5807	Solar Array Checkout Kit	TRW	2
	* 5808	Solar Array Shipping Container	RCA	4
	* 5809	Solar Array Handling Frame	RCA	3
Cabling	* 5901	Spacecraft Harness Assembly Shipping Container	TRW	3
Structure and Mechanical	6001	Aft Equipment Module Protective Cover Sling	TRW	3
	6002	Aft Equipment Module Lifting Sling	TRW	3
	* 6003	Aft Equipment Module Shipping Container	TRW	3
	* 6004	Aft Equipment Module Protective Cover	TRW	6

*Functional descriptions included in this volume.

Table 5. Subsystem Assembly, Handling, and Shipping Equipment (Continued)

Subsystem	AHSE No.	Item	Source	Quantity
	* 6005	Aft Equipment Module Dolly	TRW	3
	6006	Flight Capsule Interstage Structure Shipping Container	DAC	3
	6007	Flight Capsule Interstage Structure Shipping Container Sling	DAC	2
	* 6008	Apt Equipment Module Shipping Fixture	DAC	2
Temperature Control	6101	Louvers Shipping Container	TRW	3
	* 6102	Louvers Installation and Handling Devices	TRW	3
	* 6103	Louvers Protective Covers	TRW	6
	6104	Temperature Control Subsystem Testing Kit	TRW	3
	6105	Temperature Control Subsystem Module Shipping Container	TRW	3
	6106	Temperature Control Subsystem Module Installation Devices	TRW	3
	6107	Insulation Shipping Container	TRW	3
	6108	Louvers Sling	TRW	3
Pyrotechnic	* 6201	Ordnance Checkout Kit and Handling Case	DAC	2
	6202	Pyrotechnic Subsystem Shipping Container	DAC	3
	6203	Pyrotechnic Subsystem Shipping Container Sling	DAC	3
Fixed Science Packages	6401	Fixed Science Package Shipping Containers	GFE/JPL	3
	6402	Fixed Science Package Assembly and Handling Fixtures	GFE/JPL	3
	6403	Fixed Science Package Slings	GFE/JPL	3
	6404	Science Subsystem Spacecraft Fixture	TRW	3
LEM-D Propellant Retropropulsion	* 6501	Engine Test Facility Adapter	TRW	1
	6502	Pyrotechnic Initiator Test Set (410-62050)	GFE/GAEC	3
	6503	Portable Clean Environment Kit (420-13130)	GFE/GAEC	3
	6504	Engine Firing Control Station (410-62220)	GFE/GAEC	3
	6505	Thrust Vector Control Station (410-62250)	GFE/GAEC	3
	6506	Descent Stage Propellant Tank Dolly (420-63980)	GFE/GAEC	3
	6507	Helium Distribution Unit Controller (410-64018)	GFE/GAEC	3
	6508	Propellant Loading Control Assemby (410-64020)	GFE/GAEC	3
	6509	Descent Stage Engine Installation Dolly (420-63400)	GFE/GAEC	3
	6510	Helium Components Test Stand (430-62110)	GFE/GAEC	3

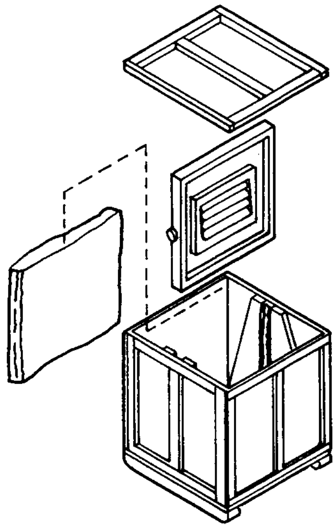
Table 5. Subsystem Assembly, Handling, and Shipping Equipment (Continued)

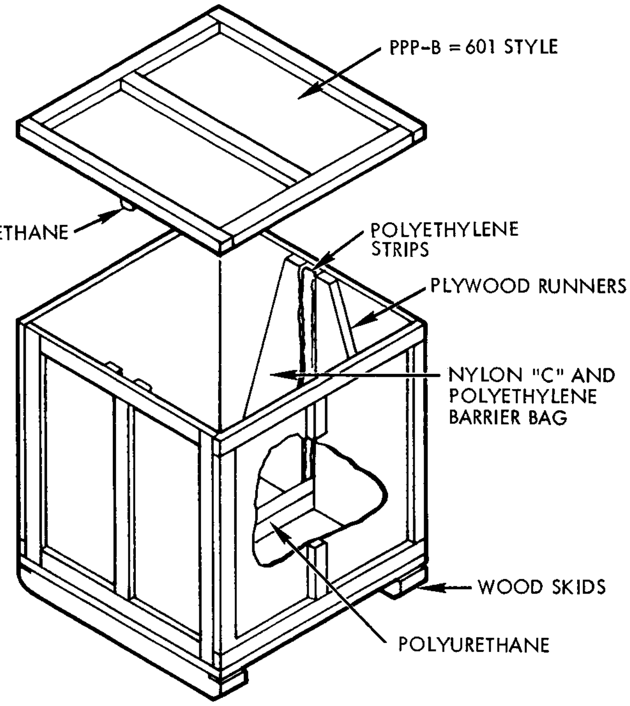
Subsystem	AHSE No.	Item	Source	Quantity
	6511	Ascent/Descent Propellant System Checkout Unit (430-62170)	GFE/GAEC	3
	6512	Propulsion Systems Checkout Cart (430-62180)	GFE/GAEC	e
	6513	Halogen Leak Detector (430-62350)	GFE/GAEC	3
	6514	Helium-Hydrogen Mass Spectrometer Leak Detector (430-82720)	GFE/GAEC	3
	6515	Propulsion Systems Portable Checkout Unit (430-62260)	GFE/GAEC	3
	6516	Helium Pressure Distribution Unit (430-64430)	GFE/GAEC	3
	6517	Fuel Loading Control Assembly (430-64430)	GFE/GAEC	3
	6518	Oxidizer Loading Control Assembly (430-64450)	GFE/GAEC	3
	6519	Descent Stage Propellant Tank Installation Fixture (420-63150)	GFE/GAEC	3
	6520	Pressure Maintenance Unit (430-64500)	GFE/GAEC	3
	6521	Oxidizer Transfer and Conditioning Unit (430-94002)	GFE/GAEC	3
	6522	Fuel Transfer and Conditioning Unit (430-94008)	GFE/GAEC	3
	6523	Helium Transfer and Conditioner Unit (430-94009)	GFE/GAEC	3
	6524	Helium Booster Cart (430-94022)	GFE/GAEC	3
	6525	Fuel Ready Storage Unit (430-94058)	GFE/GAEC	3
	6526	Oxidizer Ready Storage Unit (430-94059)	GFE/GAEC	3
	6527	Fuel Vapor Disposal Unit (430-94060)	GFE/GAEC	3
	6528	Oxidizer Vapor Disposal Unit	GFE/GAEC	3
	6529	Helium Storage Trailer (430-94062)	GFE/GAEC	3
	6530	B-377PG Transportation Kit (420-13020)	GFE/GAEC	3
	6531	Descent Stage Fitting Assembly (420-13020)	GFE/GAEC	3
	6532	Descent Stage Protective Cover (420-13480)	GFE/GAEC	6
	6533	Descent Stage Handling Dolly (420-13550)	GFE/GAEC	3
	6534	Descent Stage Support Stand (420-13700)	GFE/GAEC	3
	6535	Level Loading Cargo Lift Trailer (420-63250)	GFE/GAEC	3
	6536	Auxiliary Crane Control (420-13060)	GFE/GAEC	3

Table 5. Subsystem Assembly, Handling, and Shipping Equipment (Continued)

Subsystem	AHSE No.	Item	Source	Quantity
	6537	Console, Liquid Leveling Remote Control (LDW-410-1080)	GFE/GAEC	3
	6538	Console, Test Conductor (410-62900)	GFE/GAEC	3
	6539	Sling, D/S Propulsion Tank Assembly (LDW-420-63160)	GFE/GAEC	3
	6540	Cover, Protective D/S Engine (LDW-420-63167)	GFE/GAEC	6
	6541	Cover, D/S Engine Skirt (LDW-420-63169)	GFE/GAEC	6
	6542	Fixture, Helium Tank Handling (LDW-420-63380)	GFE/GAEC	3
	6543	Sling, Spherical Tanks (LDW-420-63399)	GFE/GAEC	3
	6544	Dolly, D/S Engine Handling (LDW-420-63400)	GFE/GAEC	3
	6545	Plug, D/S Engine (LDW-420-63420)	GFE/GAEC	3
	6547	Adapter, D/S Propellant Tank (LDW-420-63960)	GFE/GAEC	3
	6548	Drain Plug, D/S Engine (LDW-420-64190)	GFE/GAEC	6
	6549	Support Stand D/S (LDW-420-1460)	GFE/GAEC	3
	6550	Work Stand, D/S (LDW-420-1480)	GFE/GAEC	3
	6551	Sling, D/S Propulsion Tank Handling Fixture (LDW-420-1020)	GFE/GAEC	3
	6552	Support Stand D/S Engine (LDW-420-1050)	GFE/GAEC	3
	6553	Dolly, Propulsion Tank (LDW-420-6250)	GFE/GAEC	3

SHIPPING CONTAINER, SUBSYSTEM, TYPICAL,
AHSE 5101, 5301, 5401, 5501, 5601, 5701, 5801, and 5901





Functional Requirements. The shipping containers must protect the subsystem during general handling, shipment, and storage.

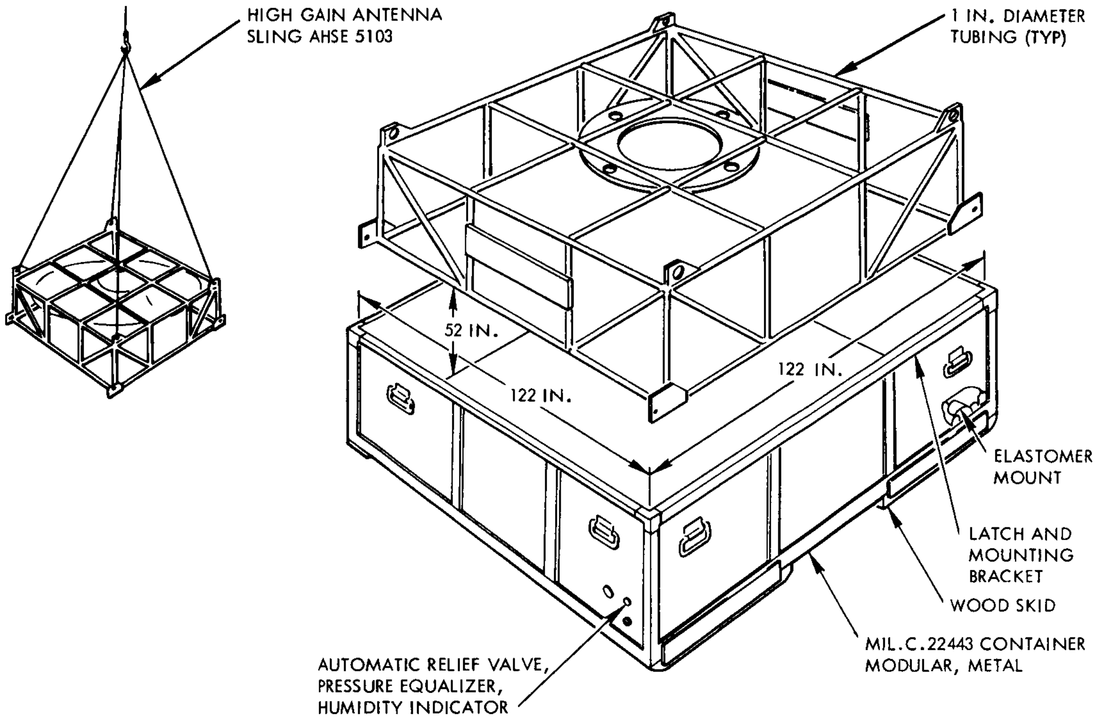
Design Requirements. The containers must protect the subsystem from physical damage and environmental contamination during surface and air transportation and storage. Their weights and sizes must be the minimum necessary to provide the desired protection. Desiccants conforming to MIL-D-3716 must be used for humidity control. Breathing provisions must be incorporated for air transportation with pressure differentials from sea level to 30,000 feet, but must not compromise the cleanliness of the subsystem. The containers must be capable of being transported by rail, truck, or air, and must be reusable.

Description. The shipping containers consist of environmental covers (barrier material), shock mitigating systems, and external protective containers. The subsystems will be completely encapsulated and supported in such a manner that the load is distributed equally.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded containers as well as flat drop, rotational drop, and pendulum impact tests.

Interface Definitions. The shipping containers interface with the subsystems of course, but have no interface with other OSE.

SHIPPING CONTAINER, HIGH-GAIN ANTENNA, AHSE 5102



Functional Requirements. The high-gain antenna shipping container must protect the antenna during handling, shipment, and storage.

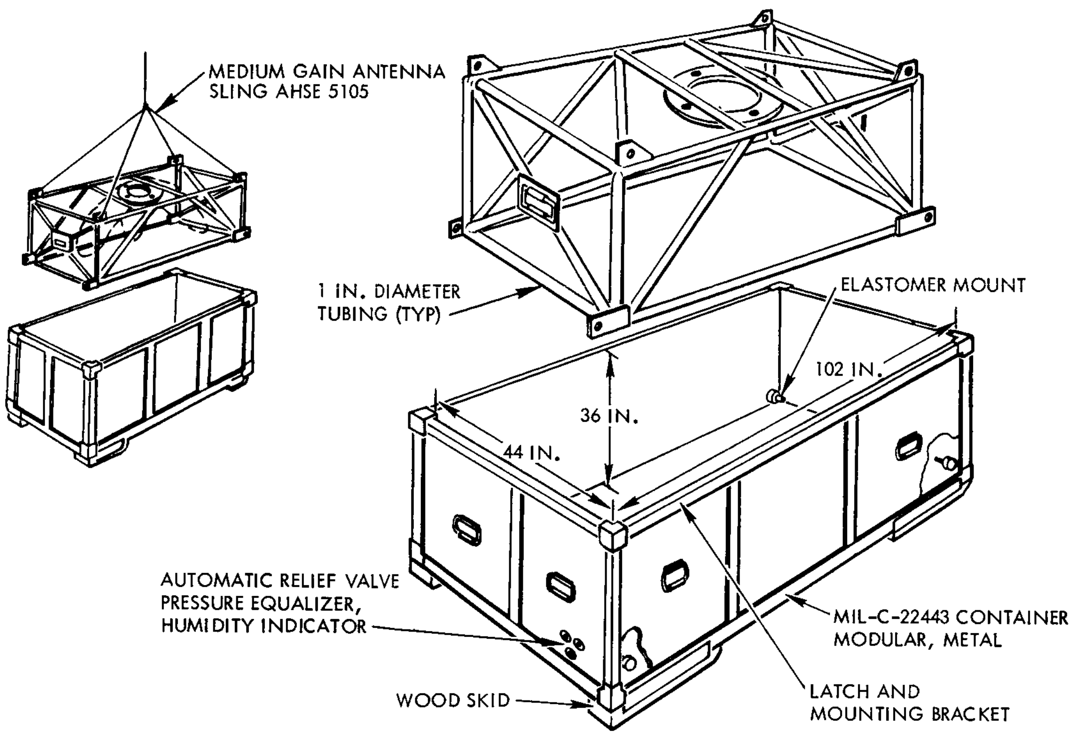
Design Requirements. The shipping container must protect the high-gain antenna from physical damage and particulate contamination during surface and air transportation and storage Its weight and size must be the minimum necessary to provide the desired protection. Shock and vibration isolation must be provided. Desiccants conforming to MIL-D- 3716 must be used for humidity control. Breathing provisions must be incorporated for air transportation to equalize pressure differentials from sea level to 30,000 feet, but must not compromise the cleanliness of the antenna. The shipping container must be reusable.

Description. The shipping container consists of a shock-mitigating system and an exterior metal modular protective container conforming to MIL-C-22443. The antenna is hard mounted to a fixture, which in turn is shock mounted to the container on elastomer mounts.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and pendulum impact tests.

Interface Definition. The shipping container is used to store and transport the high-gain antenna but has no physical or electrical interface with other OSE.

SHIPPING CONTAINER, MEDIUM-GAIN ANTENNA
AHSE 5104



Functional Requirements. The medium-gain antenna shipping container must protect the antenna during handling, shipment, and storage.

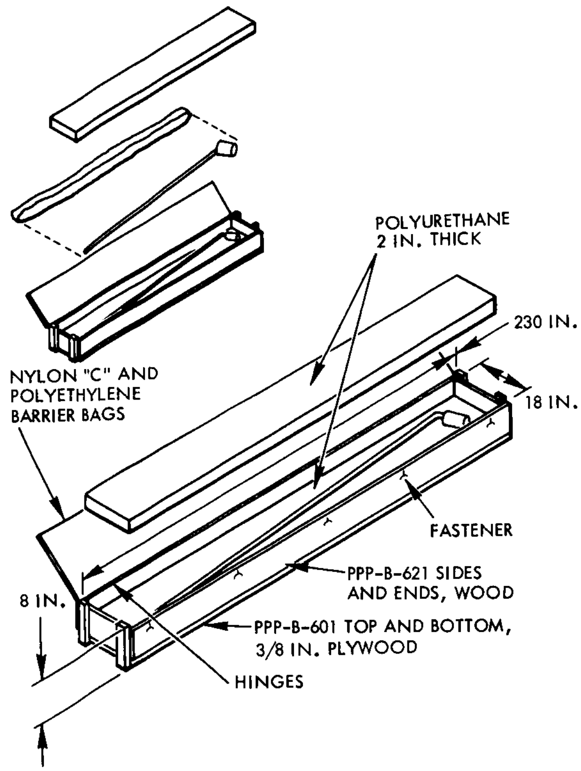
Design Requirements. The shipping container must protect the medium-gain antenna from physical damage and particulate contamination during surface and air transportation and storage. Its weight and size must be the minimum necessary to provide the desired protection. Shock and vibration isolation must be provided. Desiccants conforming to MIL-D-3716 must be used to control humidity. Breathing provisions must be incorporated for air transportation to equalize pressure differentials from sea level to 30,000 feet but must not compromise the cleanliness of the antenna. The shipping container must be reusable.

Description. The shipping container consists of a shock-mitigating system and an exterior metal modular protective container conforming to MIL-C-22443. The antenna is hard mounted to the handling fixture, which in turn is shock mounted to the container on elastomer mounts.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and pendulum impact tests.

Interface Definitions. The shipping container is used to store and transport the medium-gain antenna but has no physical or electrical interface with other OSE.

SHIPPING CONTAINER, LOW-GAIN ANTENNA, AHSE 5106



Functional Requirements. The low-gain antenna shipping container must protect the antenna during handling, shipment, and storage.

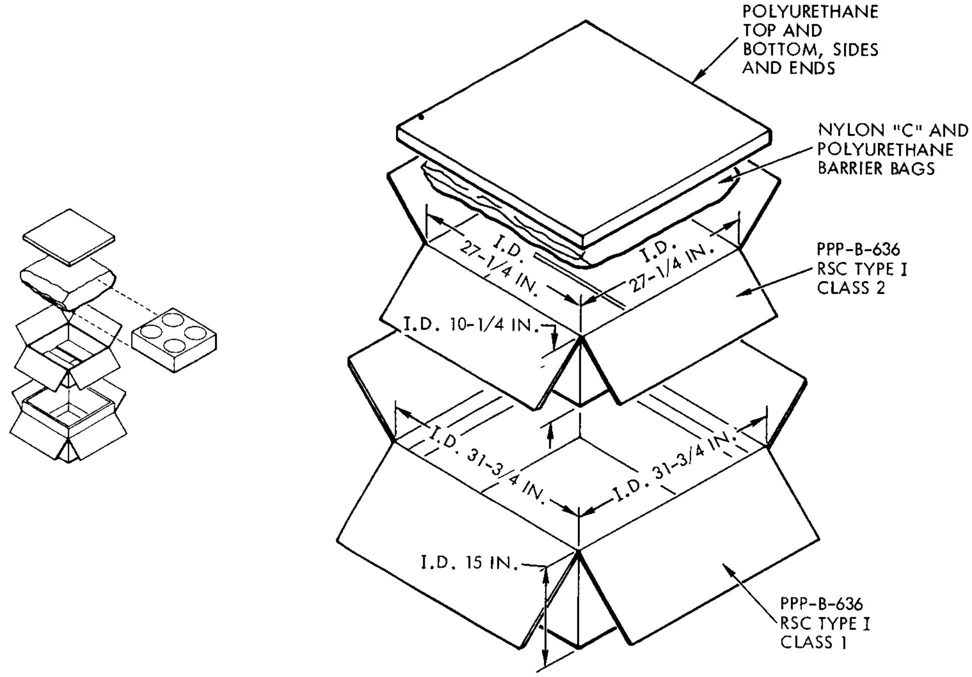
Design Requirements. The shipping container must protect the low-gain antenna from physical damage and environmental contamination during surface and air transportation and storage. Its weight and size must be the minimum necessary to provide the desired protection. Shock and vibration isolation must be provided. Desiccants conforming to MIL-D-3716 must be used to control humidity. Breathing provisions must be incorporated for air transportation to equalize pressure differentials from seal level to 30,000 feet without compromising the cleanliness of the antenna. The shipping container must be reusable.

Description. The shipping container consists of a shock-mitigating system, an environmental cover (barrier material) and an exterior shipping container The bagged antenna and boom are nested in a manner which distributes the load evenly.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and inclined impact tests.

Interface Requirements. The shipping container is used to store and transport the low-gain antenna but has no physical or electrical interface with other OSE.

SHIPPING CONTAINER RELAY-LINK ANTENNA AHSE 5201



Functional Requirements. The relay-link antenna container must protect the antenna during handling, shipment and storage.

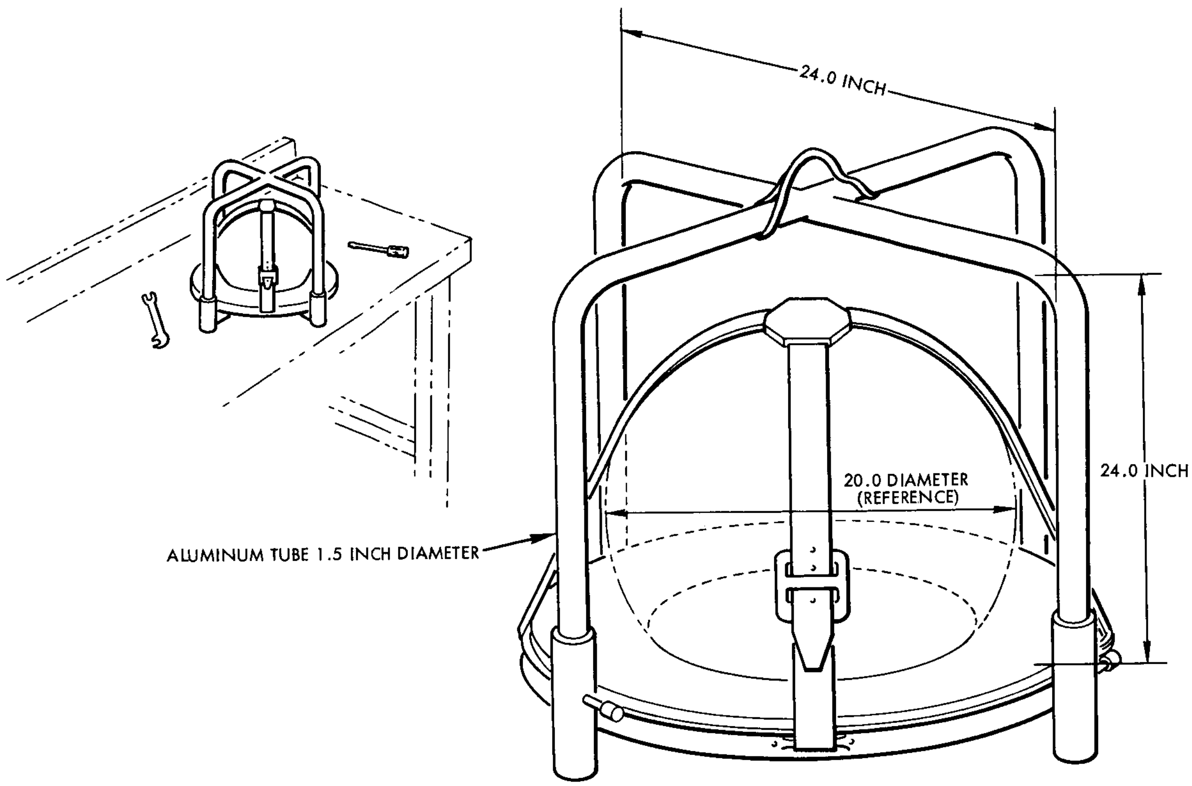
Design Requirements. The shipping container must protect the antenna from physical damage and environmental contamination during surface and air transportation and storage. Desiccants conforming to MIL-D-3716 must be used to control humidity. The antenna must be protected from particulate contamination by a nylon "C" clean barrier bag and one or more polyethylene bags. Breathing provisions must be incorporated for air transportation to equalize pressure differentials from sea level to 30, 000 feet, but must not compromise the cleanliness of the antenna. The container must be capable of being transported by rail, truck, or air. The shipping container must be reusable.

Description. The shipping container consists of an environmental cover (barrier material), a shock mitigating system, and an external protective cover. The bagged antenna and boom are completely encapsulated in polyurethane foam which nests the antenna in such a manner that the load is evenly distributed.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and inclined impact tests.

Interface Requirements. The shipping container is used to store and transport the antenna but has no physical or electrical interface with other OSE.

REACTION CONTROL PRESSURE VESSEL
HANDLING FIXTURE, AHSE 5703



Functional Requirements. A reaction control pressure vessel handling fixture is required for support and protection of individual vessels during testing, storage, and intra-plant transportation.

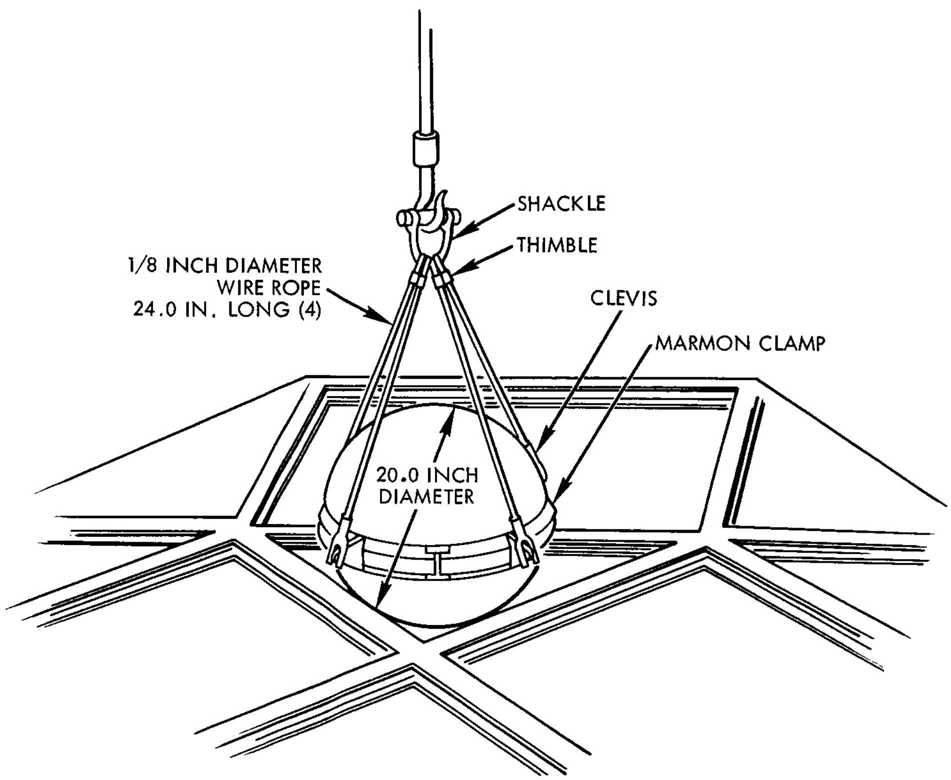
Design Requirements. The handling fixture provides a continuous support base for the 20-inch diameter spherical pressure vessel. The supporting cradle and straps provide adequate bearing surface to prevent deformation of the vessel.

Description. The fixture consists of an aluminum sheet metal cradle, lined with a cushioning material. The cradle is supported by stiffeners welded to four tubular legs. Two fabric strap assemblies, attached to the upper surface of the cradle, contain a buckle-type clinching mechanism for retention of the vessel sphere in the cushioned cradle. A tubular cage structure is attached to the four supporting legs with quick-disconnect fittings.

Test Requirements. The fixture will have fit checks and functional tests to demonstrate its capability.

Interface Definition. The handling fixture interfaces with the vessel and is used compatibly with the handling sling.

REACTION CONTROL PRESSURE VESSEL HANDLING SLING, AHSE 5704



Functional Requirements. A reaction control pressure vessel handling sling is required to lift individual pressure vessels by cranes or hoists during assembly, testing, and mating operations.

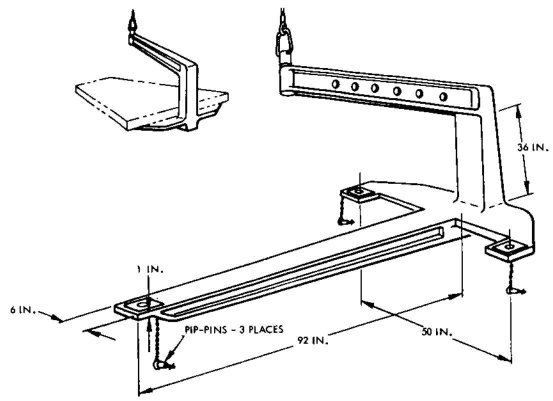
Design Requirements. The handling sling must carry approximately 20 pounds. Standard fittings are used for cable ends

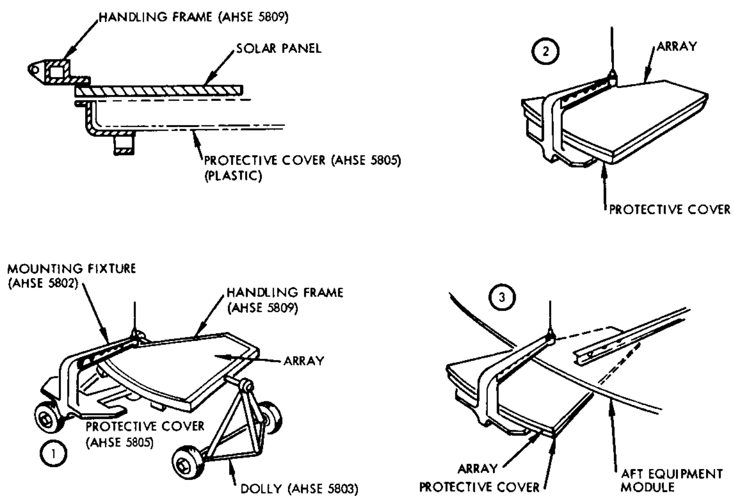
Description. The handling sling consists of a Marman clamp ring that is padded and mechanically limited to a minimum bearing pressure. Three attach brackets are welded to the clamp body at 120-degree intervals. A three-leg cable sling connects to the brackets. The cables are attached to a shackle for hoist hook attachment. The cables, shackle, end fittings, and handling ring surfaces are coated with vinyl to prevent scratching the vessel. The cables are corrosion resistant.

Test Requirements. The sling will have fit checks, functional tests, and proof-load tests to demonstrate its capability.

Interface Definition. The sling is compatible with hooks of overhead hoists or portable floor hoist. It is designed to function with the reaction control pressure vessel handling fixture (AHSE 5703).

SOLAR ARRAY MOUNTING FIXTURE, AHSE 5802





Functional Requirements. The function of the solar array mounting fixture is to attach the array segments to the hoist for transfer from the handling frame (AHSE 5809) to the spacecraft.

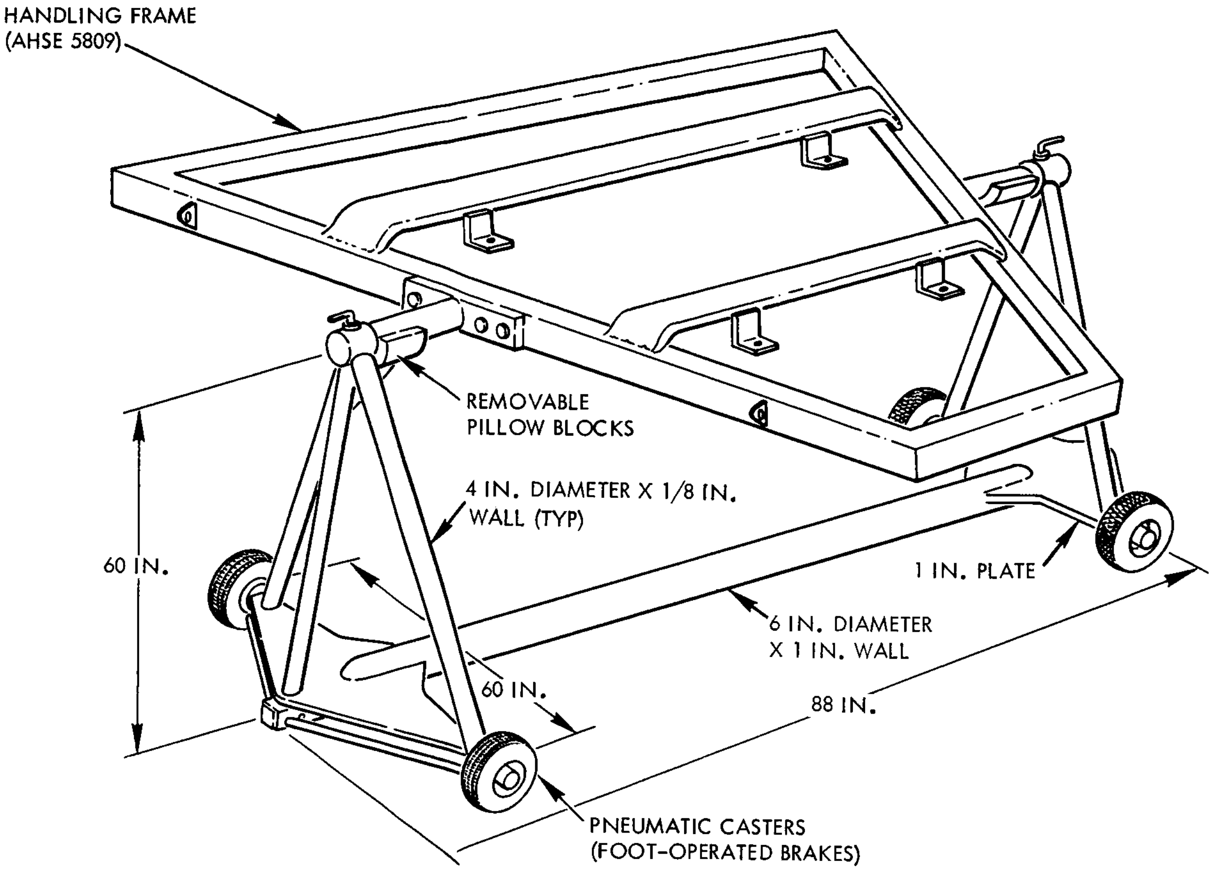
Design Requirements. The mounting fixture must attach to the solar array protective covers (AHSE 5805) with quick-release devices and be capable of supporting the arrays in position for attachment to the spacecraft structure. It must be provided with a swivel for attachment to the hydraset (AHSE 5004).

Description. The mounting fixture is a hoisting adapter of aluminum extrusion, offset to allow installation of the array segments from above.

Test Requirements. The mounting fixture will be load tested and functionally tested with dummy arrays.

Interface Definition. The mounting fixture interfaces with the solar array protective covers (AHSE 5805) and with the hydraset (AHSE 5004).

SOLAR ARRAY HANDLING DOLLY, AHSE 5803



Functional Requirements. A solar array dolly is required for in-plant transport, positioning, and rotation of the solar array segments.

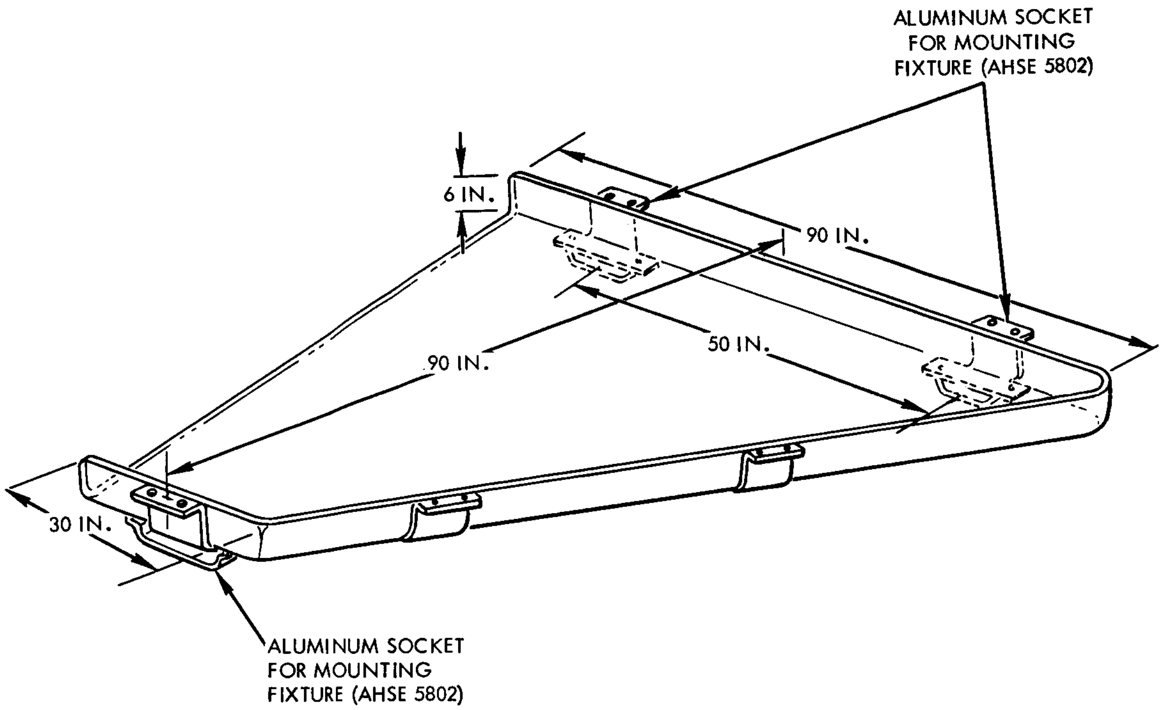
Design Requirements. The dolly must provide unrestricted access to both the top and bottom of the solar array panel. It must be capable of rotating the solar array 360 degrees about its transverse axis and of locking in any position. The steerable pneumatic-tired casters must have foot-operated parking brakes and be on a broad base to provide stability. The dolly must allow the installation of the solar array protective covers (AHSE 5805) and provide clearance for use of the solar array mounting fixture (AHSE 5802).

Description. The solar array handling dolly is a lightweight, steerable, open frame structure of aluminum, capable of positioning the solar array for assembly, test, and integration operations, and compatible with all associated AHSE.

Test Requirements. Each dolly will be functionally tested and load tested with an assembly handling frame (AHSE 5809).

Interface Definition. The solar array handling dolly has a mechanical interface with the solar array handling frame (AHSE 5809).

SOLAR ARRAY PROTECTIVE COVER, AHSE 5805



Functional Requirements. The solar array covers will protect the solar arrays during handling, storage, shipment, and spacecraft integration.

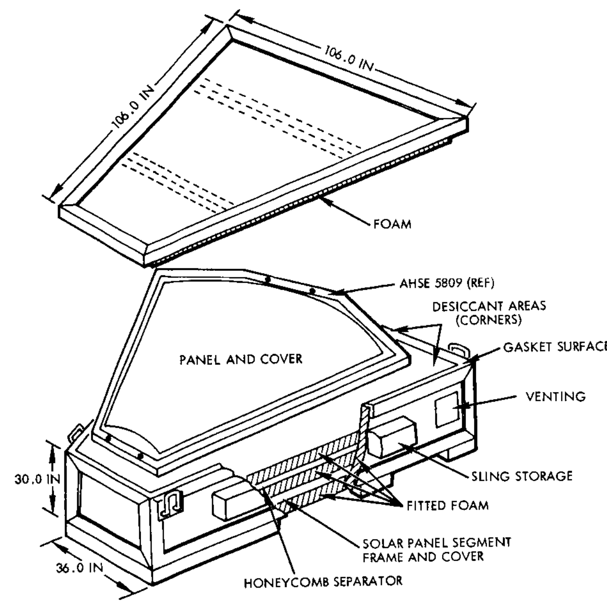
Design Requirements. The protective covers must mount to the solar array structure without interference with the solar array-spacecraft interface, the solar cells, or associated AHSE. They must be rigid enough to support the load of the arrays during installation of the arrays with the solar array mounting fixture (AHSE 5802). The covers must be transparent to allow for inspection and functional tests. Air vents must be provided to avoid overheating.

Description. The solar array protective covers are molded transparent plastic covers of ribbed construction. The attachment points (to the solar array structure) are reinforced with metal as required.

Test Requirements. The covers will be fit checked and load tested before use.

Interface Definition. The protective covers have mechanical interfaces with the solar array structure and the solar array mounting fixture (AHSE 5802).

SHIPPING CONTAINER, SOLAR ARRAY
AHSE 5808



Functional Requirements. A solar array shipping container is required to protect the solar array segments (AHSE 5808) during handling, shipment, and storage.

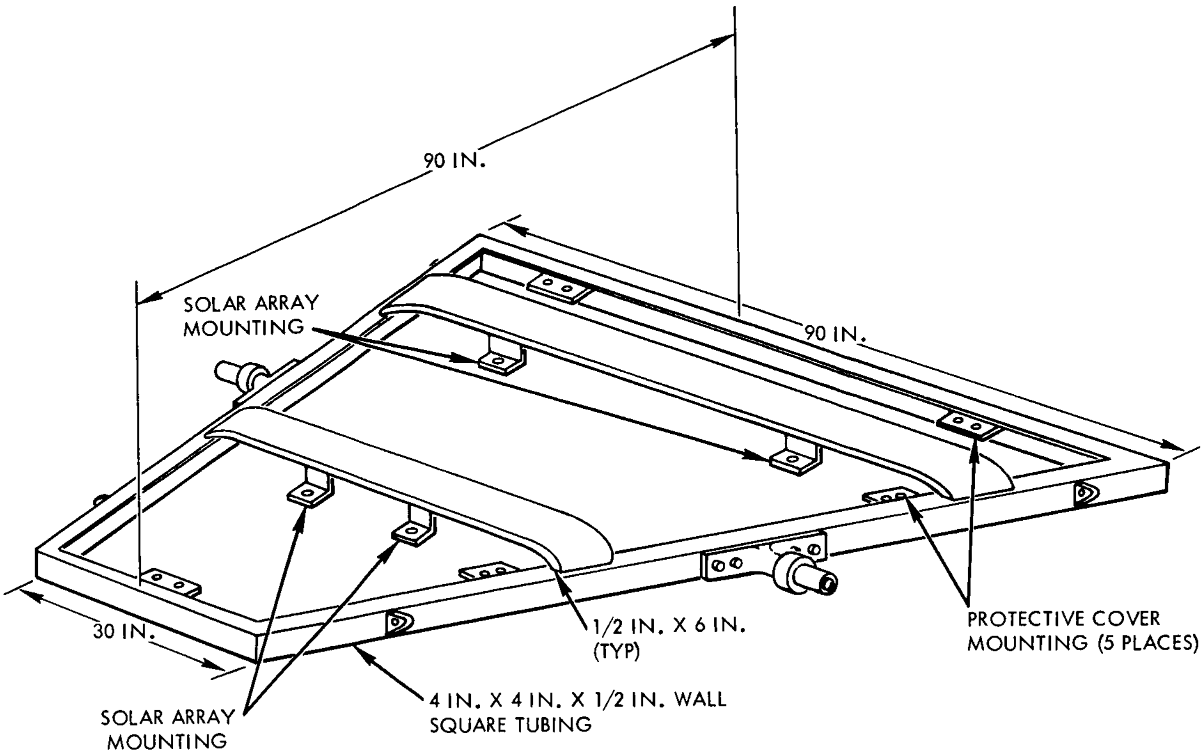
Design Requirements The shipping container must protect the array segments from physical damage and environmental contamination during surface and air transportation and storage. Their weight and size must be the minimum necessary to provide the desired protection. Desiccants conforming to MIL-D-3716 must be used to control humidity. Breathing provisions must be incorporated for air transportation with pressure differentials from sea level to 30,000 feet but must not compromise the cleanliness of the solar array segments. The shipping containers must be reusable. A sling, suitable for removing the cover from the container, and for removing the array segments from the container, must be provided.

Description. The shipping container consists of an environmental cover (barrier material), a shock mitigating system, and an external protective container. A sling to remove the container cover and the array segments is stored on the exterior of the container. Each container holds two segments.

Test Requirements. Shock and vibration tests must be performed on the proof-loaded container as well as flat drop, rotational drop, and pendulum impact tests. Proof-loading tests must be performed on the sling

Interface Definition. The shipping container is used to store and transport the subsystem but has no physical or electrical interface with other operating support equipment. The sling has a mechanical interface with the solar array segment handling frames (AHSE 5809) and the hydraset (AHSE 5004).

SOLAR ARRAY HANDLING FRAME, AHSE 5809



Functional Requirements. The solar arrays will require structural support during assembly, testing, and inspection. It is the function of the handling frame to provide this support and adapt the arrays to the solar array handling dolly (AHSE 5803).

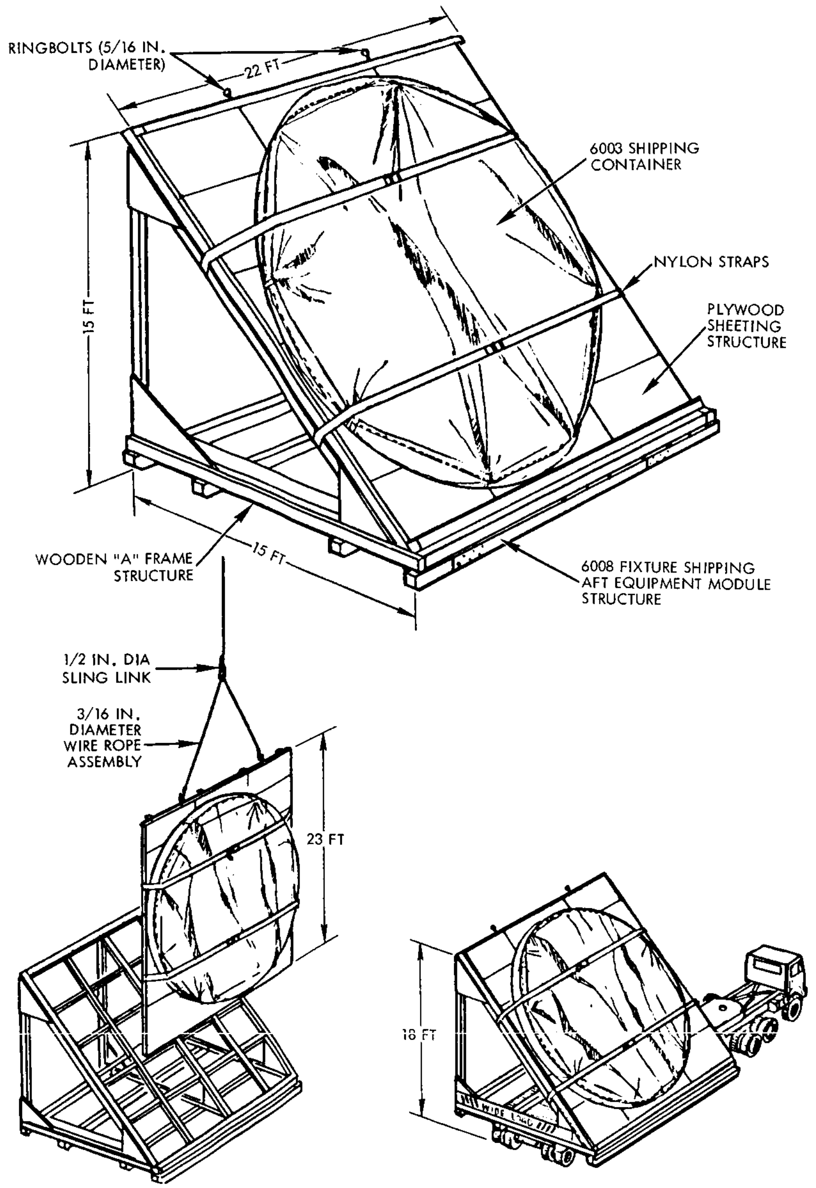
Design Requirements. The solar array handling frame must be sufficiently stiff and must be mounted in such a way as to prevent physical loading of the solar array structure. It must allow unrestricted access to both surfaces of the array and allow installation of the solar array mounting fixture (AHSE 5802) and the solar array protective cover (AHSE 5805).

Description. The solar array handling frame consists of a welded aluminum framework which attaches to and supports the solar array structure at the solar array-spacecraft interface, and mounts in the handling dolly (AHSE 5803) on trunnions.

Test Requirements. Each handling frame will be proof-load tested and functionally tested and fit-checked with a solar array handling dolly (AHSE 5803).

Interface Definition. The solar array handling frame has a mechanical interface with the solar array handling dolly (AHSE 5803) and with the solar array sling (AHSE 5808).

FIXTURE, SHIPPING, AFT EQUIPMENT MODULE STRUCTURE, AHSE 6003 and 6008



Functional Requirements. The aft equipment module structure must be environmentally protected during shipment from the point of manufacture to various assembly and test facilities. In addition, it must have a shipping fixture, rigidly supporting it against torsional or racking loads.

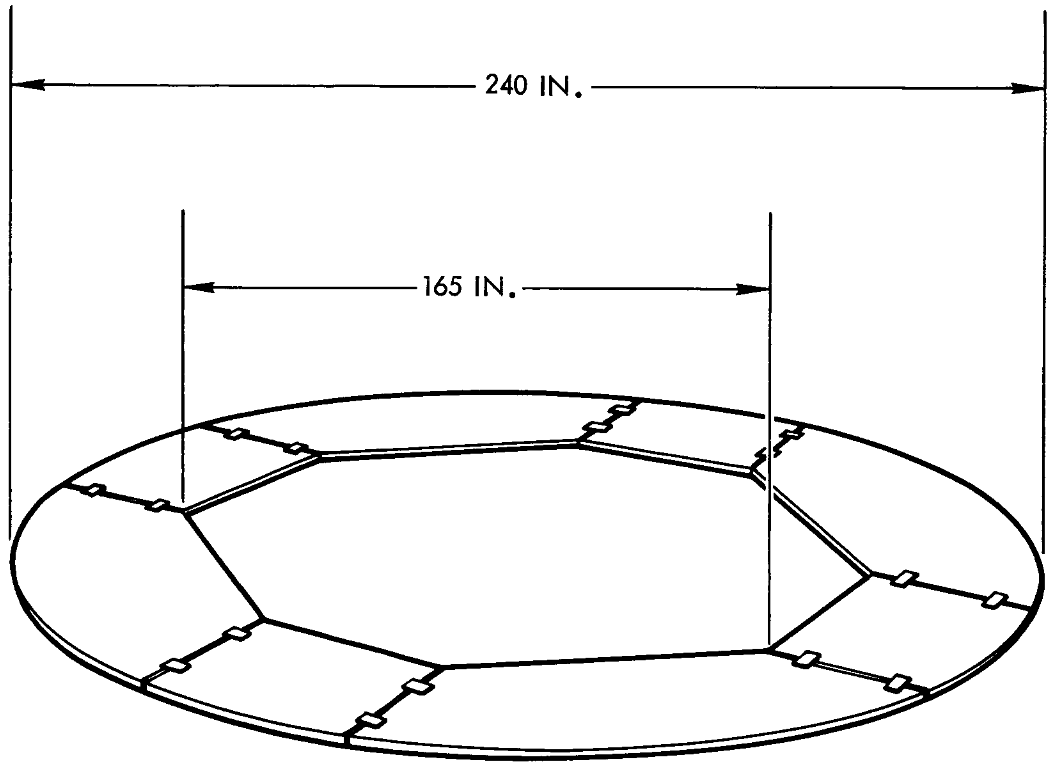
Design Requirements. The shipping fixture must be reusable and protect the module structure during transportation.

Description. The shipping fixture consists of a 45-degree angle wooden "A" frame and a removable plywood sheeting structure. The "A" frame is fabricated from 4 x 6 and 4 x 4-foot skids and headed with splicing, conforming to MIL-HBK-710, to prevent skid deflection during transportation. A desiccated, peripherally zippered, reusable waterproof vinyl cover (protective barrier) is placed around the aft equipment module structure. All mounting bolts which must pass through this barrier are gasketed in accordance with MIL-P-116D. The fixutre is attached by bolts to the bottom of the aft equipment module structure through the solar array attach holes.

Test Requirements. The shipping fixture will be functionally and proof-load tested, and the protective barrier will be functionally tested.

Interface Requirements. The shipping fixture interfaces with the aft equipment module structure and the aft equipment module structure shipping fixture sling (AHSE 6001). It is rigidly fastened to the bed of shipping container (AHSE 6003). It is rigidly fastened to the bed of a standard low-boy flat-bed commercial trailer during interfacility shipment.

AFT EQUIPMENT MODULE PROTECTIVE COVER,
AHSE 6004



Functional Requirements. An aft equipment module protective cover is required to protect the module from damage during assembly and integration of the spacecraft.

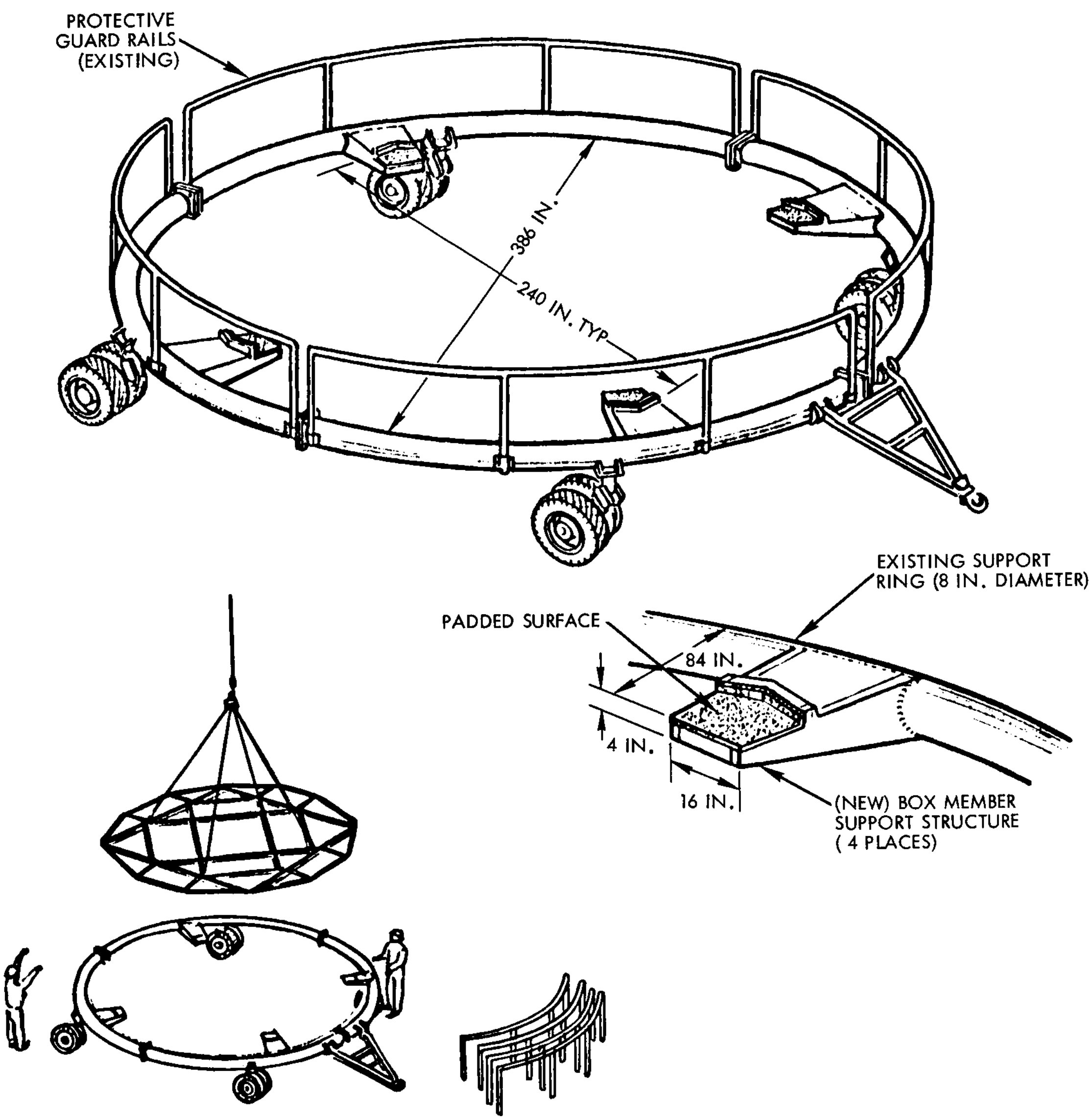
Design Requirements. All material must be nonmagnetic, lightweight, and capable of absorbing high-impact loads. Each section must be capable of supporting and absorbing design loads separately.

Description. The protective cover is made in eight individual sections, any one of which can be removed without disturbing the integrity of the remaining section. The basic structural material is a half-inch aluminum honeycomb. The protective cover is supported by selected surfaces and support members on the spacecraft and the vertical checkout and assembly stand (AHSE 5003).

Test Requirements. A series of structural and load carrying tests will be performed to demonstrate design requirements.

Interface Definitions. The aft equipment module protective cover must be compatible with the spacecraft and the vertical checkout and assembly stand (AHSE 5003).

DOLLY, AFT EQUIPMENT MODULE STRUCTURE,
AHSE 6005



Functional Requirements. The spacecraft aft equipment module structure must be supported and protected, within a specific facility during and between various assembly and test operations.

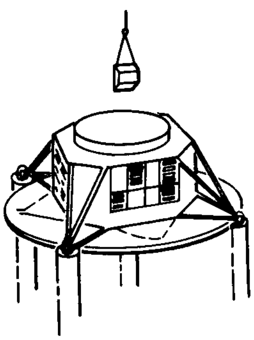
Design Requirements. The spacecraft aft equipment module structure dolly provides support and transportability for the aft equipment module structure during assembly and test operations. The dolly provides physical protection around the periphery of the structure to prevent inadvertent contact with objects. Jack pads, a shock attenuation system, and a manual braking system for the dolly are provided. The dolly mounts on pneumatic tires not less than 4.00 x 8 in size and can be towed at speeds up to 5 mph. Working load of the dolly will not exceed 2000 pounds. Mobility of the dolly will be in accordance with Type I, Class 2 in MIL-M-00809D.

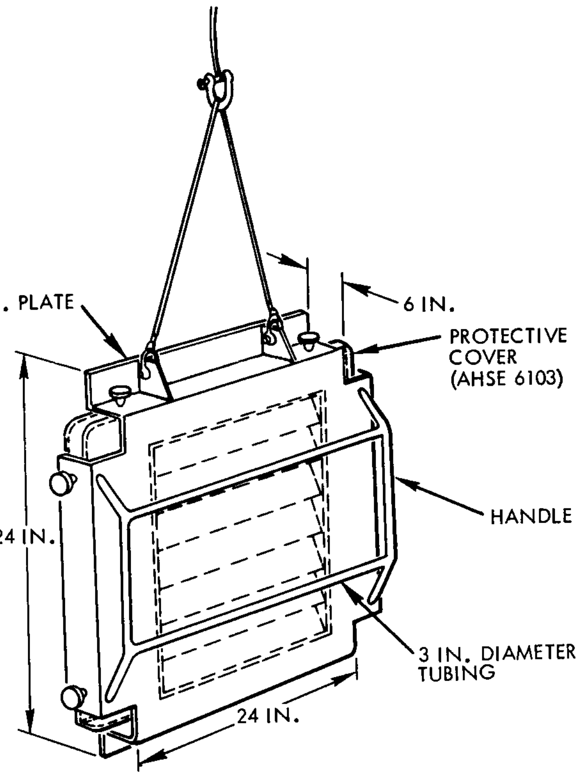
Description. The dolly for the spacecraft aft equipment module structure consits of a modified version of the existing Saturn S-IVB/IB interstage structure handling kit and dolly (DSV-4B-352). The modifications consist of the addition of four box member support structures with padded surfaces to support the aft equipment module structure at the bottom flanges of the four interconnecting corners of the main structural beams. A towbar assembly is provided for moving the dolly within a facility to various work stations.

Test Requirements. The aft equipment module structure dolly will undergo applicable tests for Type I, Class 2 mobility, as specified in MIL-M-00809D. It will be functionally tested to demonstrate required performance and proof-load tested to demonstrate design and fabrication adequacy.

Interface Definitions. The dolly interfaces with the bottom side of the aft equipment module structure at the four corners of the main structural beam assembly and industrial mules or electric tractors at the towbar attach point.

TEMPERATURE CONTROL SYSTEM INSTALLATION DEVICE
AHSE 6102





Functional Requirements. The temperature control louver installation device must provide physical support to the louver structure during testing and assembly on the spacecraft.

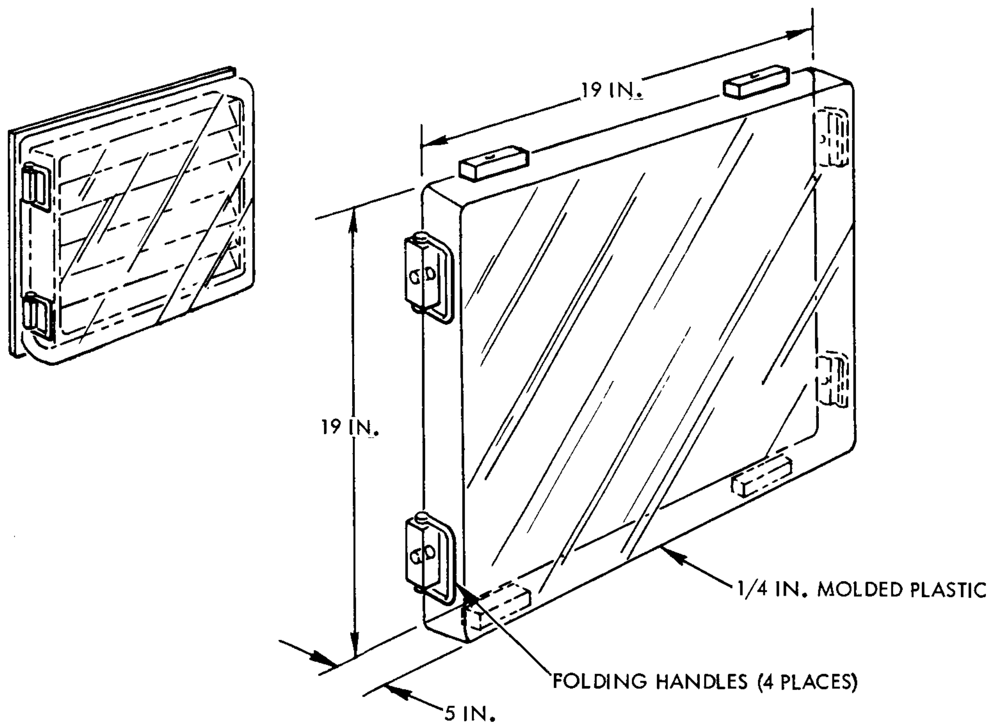
Design Requirements. The temperature control louver installation device must attach to the louver assembly without interference with the louver-spacecraft interface and without distorting the louver assembly structure. The louver installation fixture must provide an attachment point for a hydraset from which the louver assembly can be suspended in its installed position while it is being attached to the spacecraft.

Description. The louver installation fixture consists of a rigid, lightweight, nonmagnetic structure with quick release attachments to the louver structure. A hoist and sling attachment point as well as carrying handles is provided. This concept, based on the Mariner units, is shown.

Test Requirements. Functional testing and load testing will be performed on the fixture before use.

Interface Definition. The louver installation fixture has mechanical interfaces with the louver assemblies and hydraset.

TEMPERATURE CONTROL LOUVER PROTECTIVE COVER
AHSE 6103



Functional Requirement. The temperature control louver protective covers are required to protect the louvers during spacecraft assembly and test.

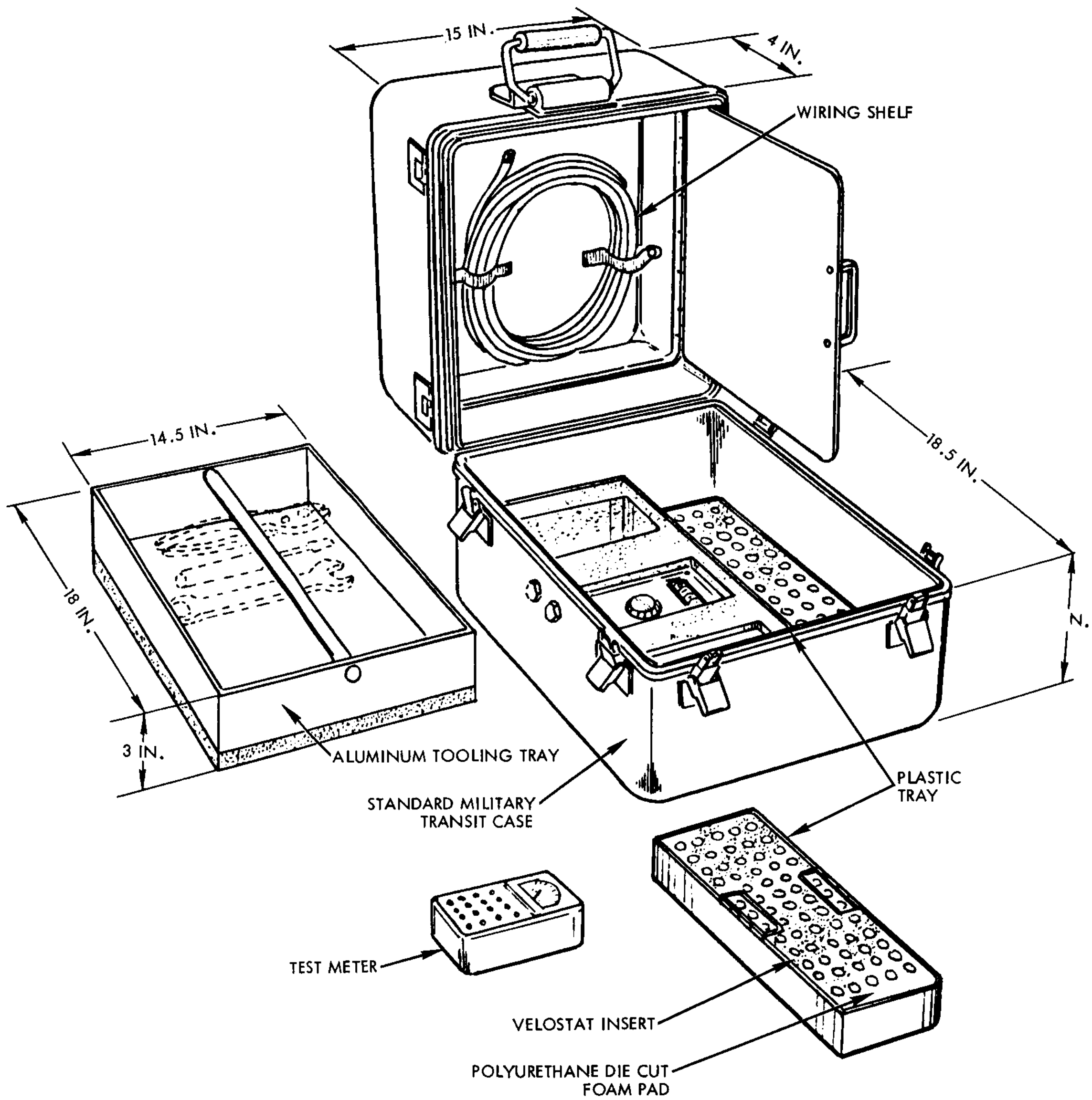
Design Requirement. The temperature control louver protective covers must attach, with a quick disconnect device, to the louver frame assembly without interference with the louver frame-spacecraft interface. The covers must be transparent and provide clearance for attachment of the louver installation and handling device (AHSE 6106). The protective covers must not interfere with the normal operation of the louvers as they respond to temperature changes.

Description. The protective covers are 1/4-inch-thick moded acrylic or other transparent plastic. Attachment points are reinforced with metal as necessary. Folding handles are provided for installation and removal of the covers.

Test Requirements. All temperature control louver protective covers will be fit checked and functionally tested.

Interface Definition. The protective covers have a mechanical interface with the louver frame assembly.

HANDLING CASE, ORDNANCE CHECKOUT AND ARMING KIT, AHSE 6201



Functional Requirements. Positive accountability and installation and checkout capability is required for all spacecraft electroexplosives, simulators, and shorting plug devices. Means for transporting these devices and pyrotechnic circuit test instruments to and from the ESA is also required.

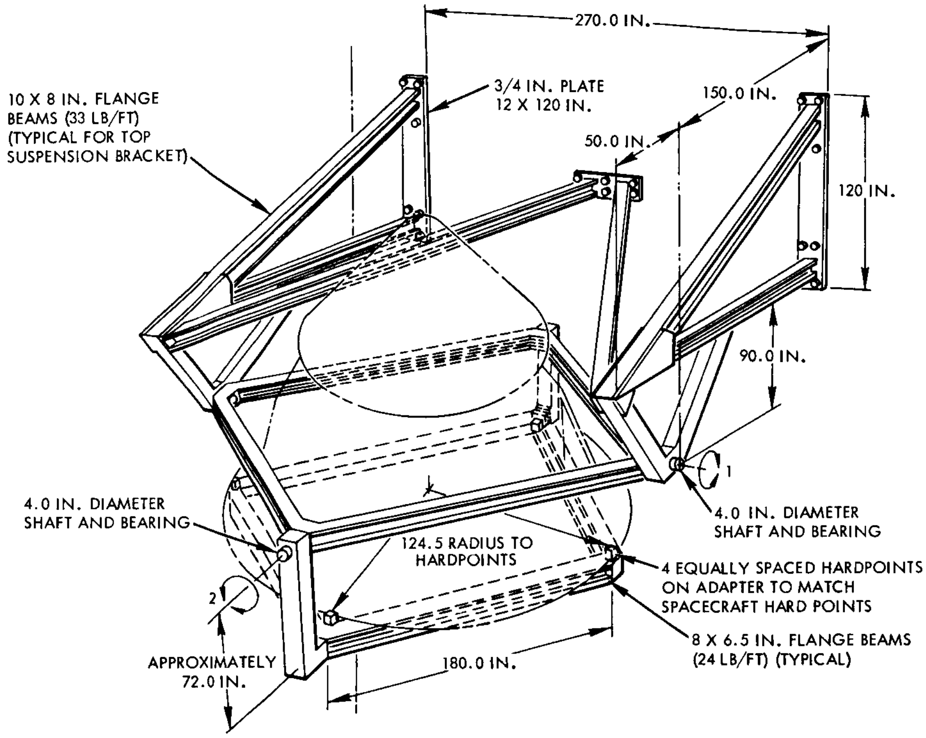
Design Requirements. The arming kit handling case must be portable, equipped with carrying handles, and provided with recesses for visual access to all ordnance and arming devices, test instrumentation, and tools required to test pyrotechnic circuits and to arm the spacecraft. Conductive material must be used for all components of the handling case, and the case must be provided with means for positive grounding.

Description. The handling case consists of an aluminum wiring shelf, an aluminum tooling try, two die-cut polyurethane foam pads, two plastic trays, and a standard aluminum military transit case. The wiring shelf, of sheet aluminum, is hinged and latched to the top of the handling case. The depth of the shelf is 3 inches minimum. The tooling tray, also sheet aluminum, is provided with an aluminum handle across the entire length of the tray. A 2.4-pound density polyurethane foam pad is bonded to the bottom of the tray. A carbonized conductive polyethylene film (Velostat) is placed around each die-cut foam pad to prevent static charge buildup.

Test Requirements. The handling case will be drop-shock tested and tested to demonstrate static charge inhibition and ground continuity.

Interface Requirements. The handling case has no physical or electrical interface with other operating equipment, but provide mounting recesses for all spacecraft ordnance devices and checkout equipment.

ENGINE TEST FACILITY ADAPTER, AHSE 6501



Functional Requirements. An engine test facility adapter is required to adapt the spacecraft to a test stand for tests on the engine. These tests consist of suspending the spacecraft in a test chamber and determining interactions between the propulsion, structure, and attitude control subsystems during propulsion firings.

Design Requirements. The spacecraft is allowed to pivot about two axes under the influence of the engine during firing. The adapter is designed to support a suspended load of 11,500 pounds and the propulsion thrust load of 10,000 pounds.

Description. The adapter consists of a gimballing system to allow the spacecraft to pivot on two axes. A structural frame is attached to the aft end of the spacecraft at its hardpoints and has bearing shafts connected to it. This frame with attached spacecraft is suspended from an upper structure located at the spacecraft center of gravity. The upper structure carries bearing shafts which mate with suspension brackets on the engine test stand.

Test Requirements. The adapter will have fit checks, functional tests, and proof-load tests.

Interface Definition. The engine test facility adapter interfaces with hardpoints on the aft end of the spacecraft and with the engine test stand.

IV. OSE IMPLEMENTATION PLAN

1. INTRODUCTION

The purpose of this implementation plan is to identify and organize the activities required to develop, manufacture, test, and document the operational support equipment used for the Voyager spacecraft. The plan includes the implementation approach, critical areas, milestone schedules, task flows, detailed task descriptions, and quantity and utilization. The work breakdown structure provides a detailed listing of tasks planned for Phase IB and a general task breakdown for tasks in Phase II. This is in consonance with the prescribed approach (given in "Performance and Design Requirements for the Voyager 1971 Spacecraft System, General Specification," dated September 17, 1965) of developing detailed task descriptions for Phase II as a portion of the Phase IB effort.

2. APPROACH

The initial Phase IB effort will concentrate on developing a complete list of OSE, i.e., all OSE required for system and subsystem development, manufacturing, test and support. Preliminary designs and specifications will then be generated. Preliminary design specifications (Part I CEI's) will be completed by October 1966 for submittal to JPL, with approved design specifications (Part I CEI's) completed by the end of Phase IB. Also by the end of Phase IB, all OSE will be quantified, scheduled, and allocated. Development of a Phase II work package and cost plan, which contains Phase II task descriptions and plans, will also be part of the Phase IB effort.

The initial Phase II effort will include detailed design of OSE and preparation of procedures and equipment specifications (Part II CEI's). The designs will be based on the Phase IB preliminary designs. Procedures to be generated include both test and operational. The procedures and designs will be reviewed at the CDR. Fabrication of EOSE starts 3 months and of AHSE 7 months after Phase II go-ahead. Deliveries are scheduled to support all subsystem development test and fabrication as well as spacecraft assembly and test activities.

3. CRITICAL AREAS

Certain critical areas exist in the development of OSE for support of the Voyager spacecraft project. The following areas will be given special attention, and through the methods indicated, the possibility of these adversely affecting the schedule will be precluded.

3.1 Spacecraft Test Requirements

It is necessary that spacecraft functional test requirements for each of the types and levels of test be defined in a timely and accurate manner. During the first part of Phase IB, additional or changed test requirements data beyond that established in Phase IA will be made firm before the detailed OSE design starts. Close and continuous liaison with spacecraft subsystem and system designers will continually communicate test requirements data to OSE engineers.

3.2 Spacecraft-OSE Interfaces

Preliminary definition of the spacecraft-OSE interfaces has been made during Phase IA. Detailed definition of these interfaces and of the man-machine interface will be completed during Phase IB. Extensive control is planned for the interfaces between checkout equipment and spacecraft, checkout equipment and experiments, and checkout equipment and personnel. Interface definition control and evaluation will be assisted by interface criteria documentation which will:

- Identify interfaces through block diagrams and lists
- Identify interface types
- Define interface format, content, and notation in a standardized manner.
- Identify the interface participants and those responsible for preparing, distributing, and maintaining the interface documentation.
- Define review and approval methods

Potential interface problems will be minimized by a systematic procedure of identification, interface definition, definition of the impact of equipment design changes, and the maintenance of system compatibility through tightly controlled engineering data management.

3.3 Spacecraft Test Point Access

The provision of adequate test access to spacecraft subsystems and units has been considered during the preliminary design work of Phase IA. The work will be continued during Phase IB, during which the testing process for all levels from assembly through system will be considered such that the proper tradeoff can be made between access to sufficient but minimal data points and the possible provision of an excess over that required for proper flight system status evaluation. To aid in this goal test engineers will participate in the spacecraft design tradeoff studies.

3.4 Long-Lead Procurement

Some material or equipment included in the OSE involves an unavoidable long procurement cycle. In these cases, advanced material releases are required prior to completion of the design, design review, and drawing release procedures. To avoid schedule slippages, these long-lead items have been identified and their procurement will be continuously monitored.

3.5 Configuration Control

One of the prime source of problems in a complex system development is that of control of the end item configuration. Control implies the timely communication of configuration information on the end item to all affected parties as well as continual direction and monitoring of the adherence of each item to functional requirements and standardization. This will be achieved through adequate planning, culminating in the preparation and maintenance of the minimum configuration control documentation consistent with achieving this objective.

4. MILESTONE SCHEDULE

The policy used in scheduling Phase IB and Phase II activities as shown on the OSE summary schedule (Figure 41) was to provide a set of initial OSE for verification test (interface and operational) with the engineering model spacecraft before OSE is provided for the proof test model and flight spacecrafts. The schedule allows for design changes, as developed from the engineering model verification tests, to be integrated into the OSE prior to PTM and flight spacecraft OSE fabrication.

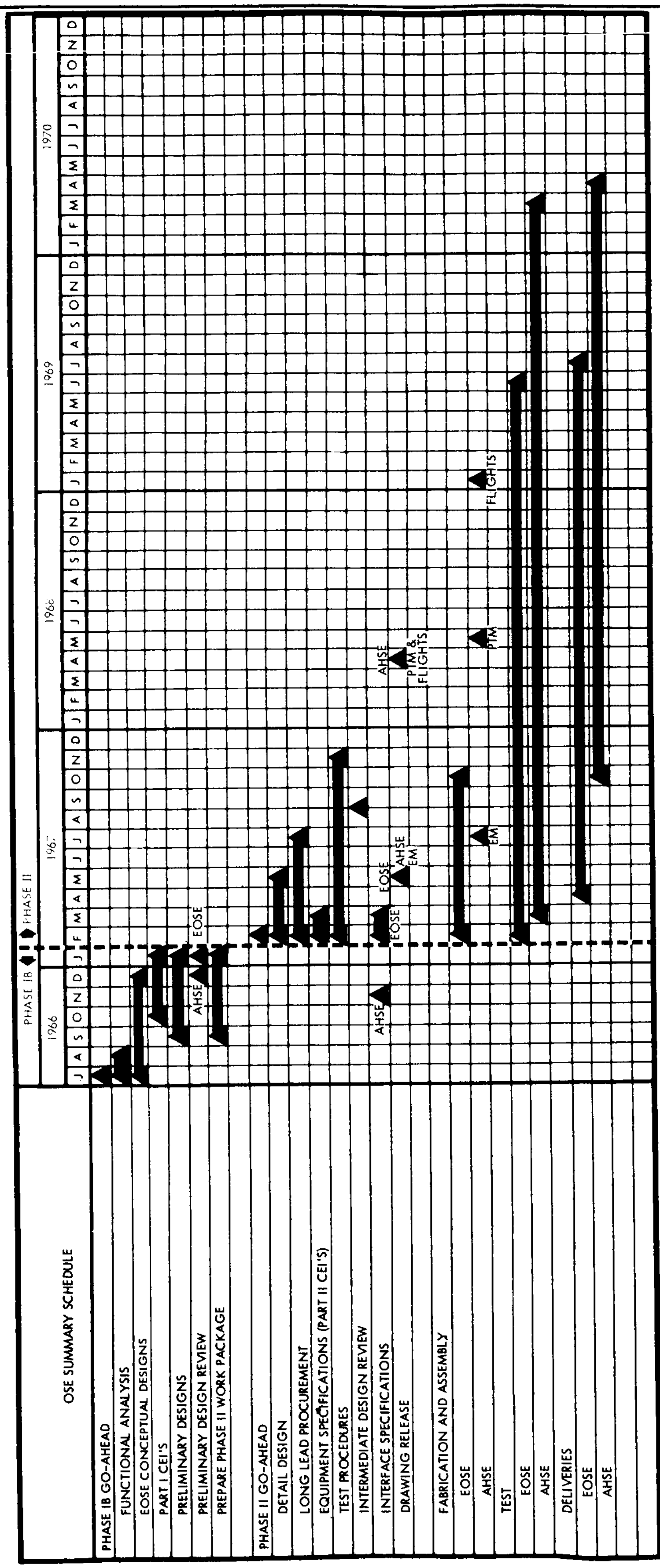


Figure 41. OSE Summary Schedule

Drawing releases, procedures, specifications, and all other activities are scheduled in consonance with top program milestones and general requirements, i.e., spacecraft launch dates, October 1966 preliminary specification delivery in Phase IB, final specification delivery at the end of Phase IB, complete design layouts by the end of Phase IB, etc.

An EOSE milestone schedule is given in Figure 42, indicating the primary phases of EOSE development as a function of time. The sequential list of major milestone items indicates the estimated time span for each of the major tasks during EOSE development. The AHSE milestone schedule (Figure 43) depicts detailed AHSE milestones, the time span for which the AHSE is required for each operation: assembly and checkout, system test, launch operation, magnetic facility testing, vibration, thermal vacuum, White Sands, Goldstone and facility checkout. The specific items of AHSE to be delivered are tabulated in Section III of this document. To be noted is that the schedule does not imply delivery of complete sets of AHSE for each spacecraft.

5. DEVELOPMENT FLOWS AND TASK DESCRIPTIONS

5.1 EOSE Development Flow

The activities of the EOSE development phases are shown on the development program network of Figure 44, which indicates input requirements and tasks performed for each specific development event and relates specific EOSE development tasks. Since the diagram is at the first level of detail, each event shown can be subdivided into many subordinate events. During the preparation of the EOSE program plan in Phase IB, the subordinate activities related to design, manufacture, and test of OSE will be defined. The format used is compatible with that in the task flows of Volume 3 of this report.

5.2 EOSE Tasks

The EOSE task descriptions present the various activities to be performed in implementing the development of the Voyager EOSE. The tasks are analysis, design, manufacture, testing and sustaining engineering. The discussion follows the time planning of the various activities shown in the EOSE subsystem task flow of Figure 44.

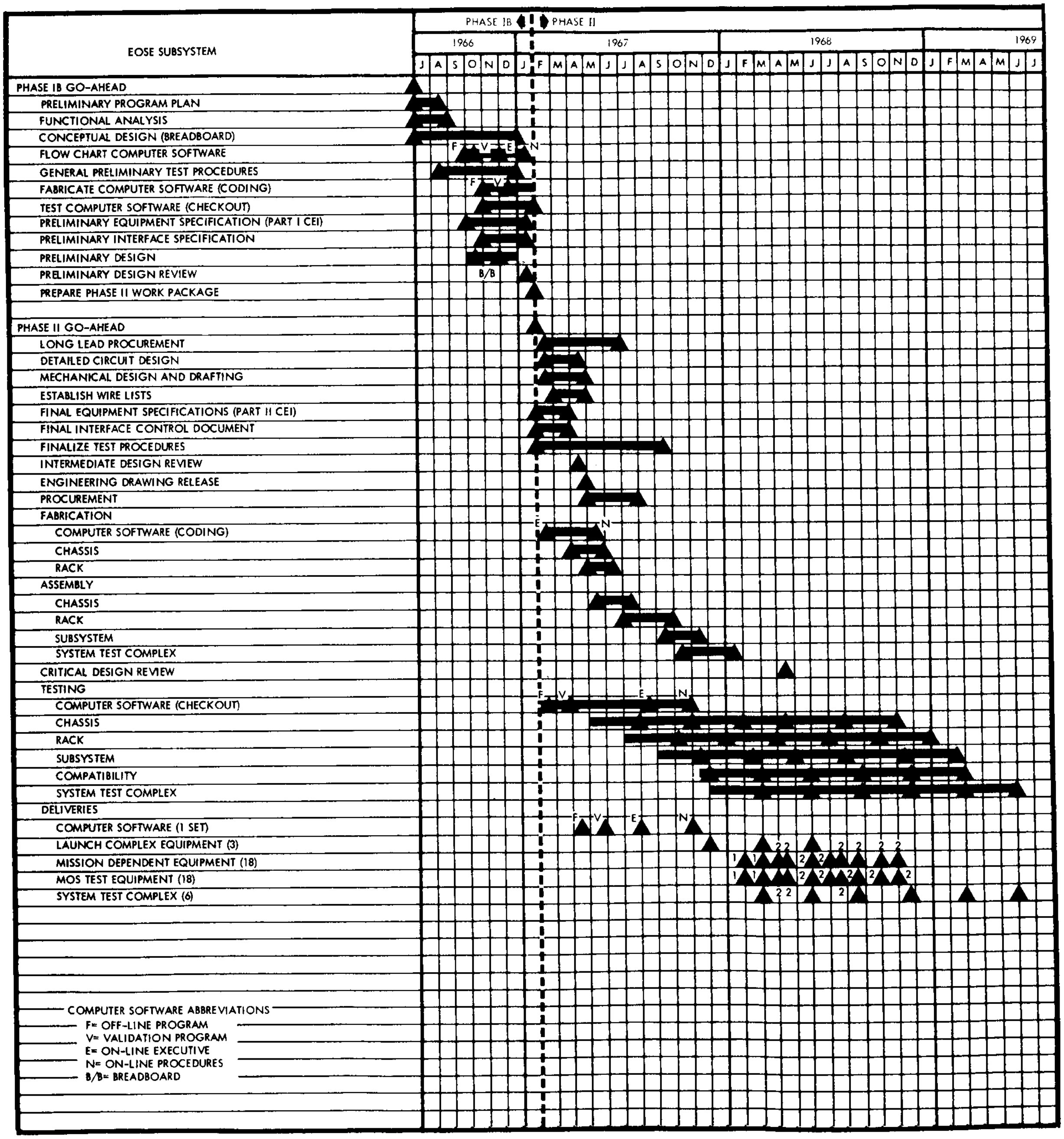


Figure 42. EOSE Milestone Schedule

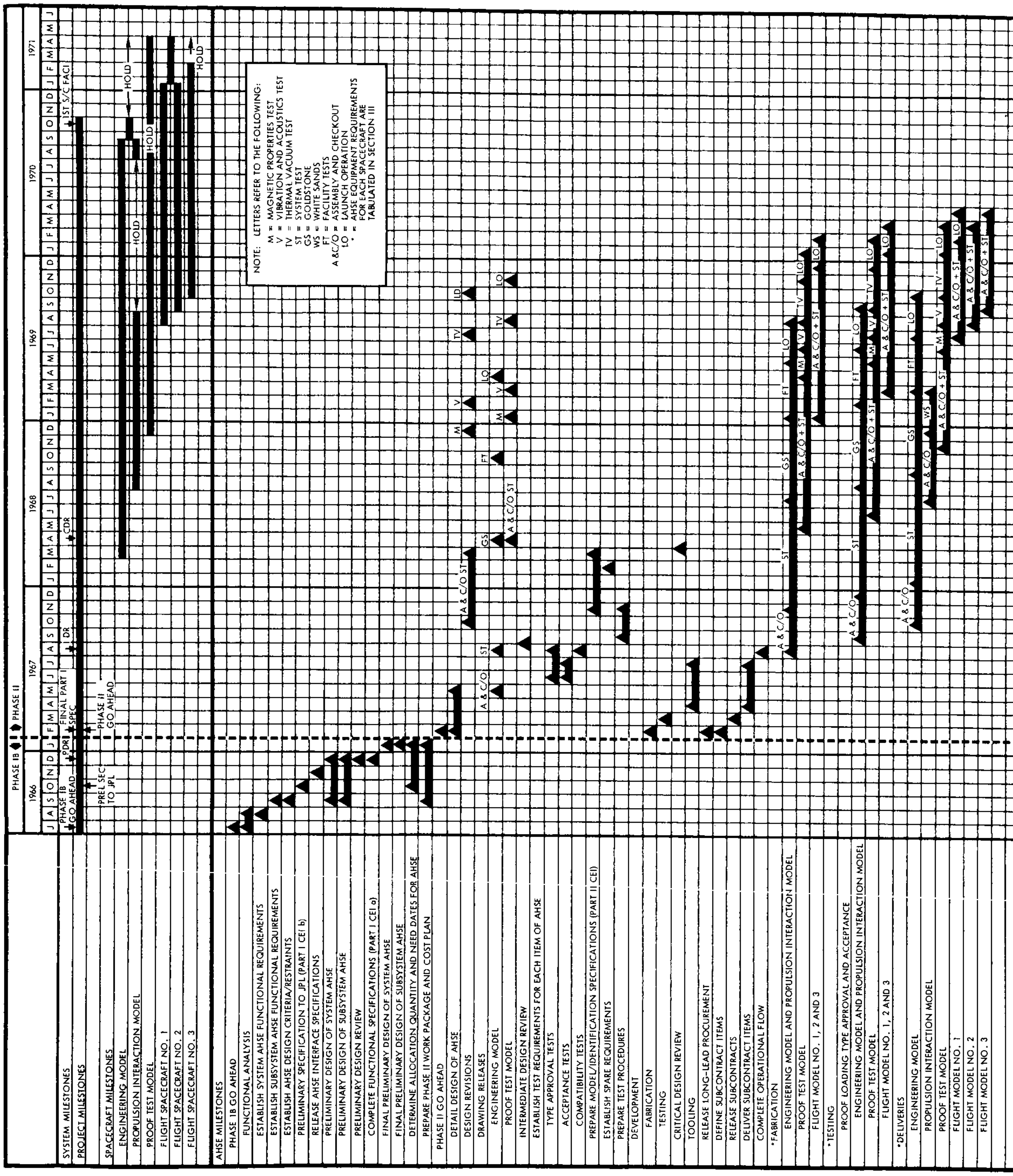
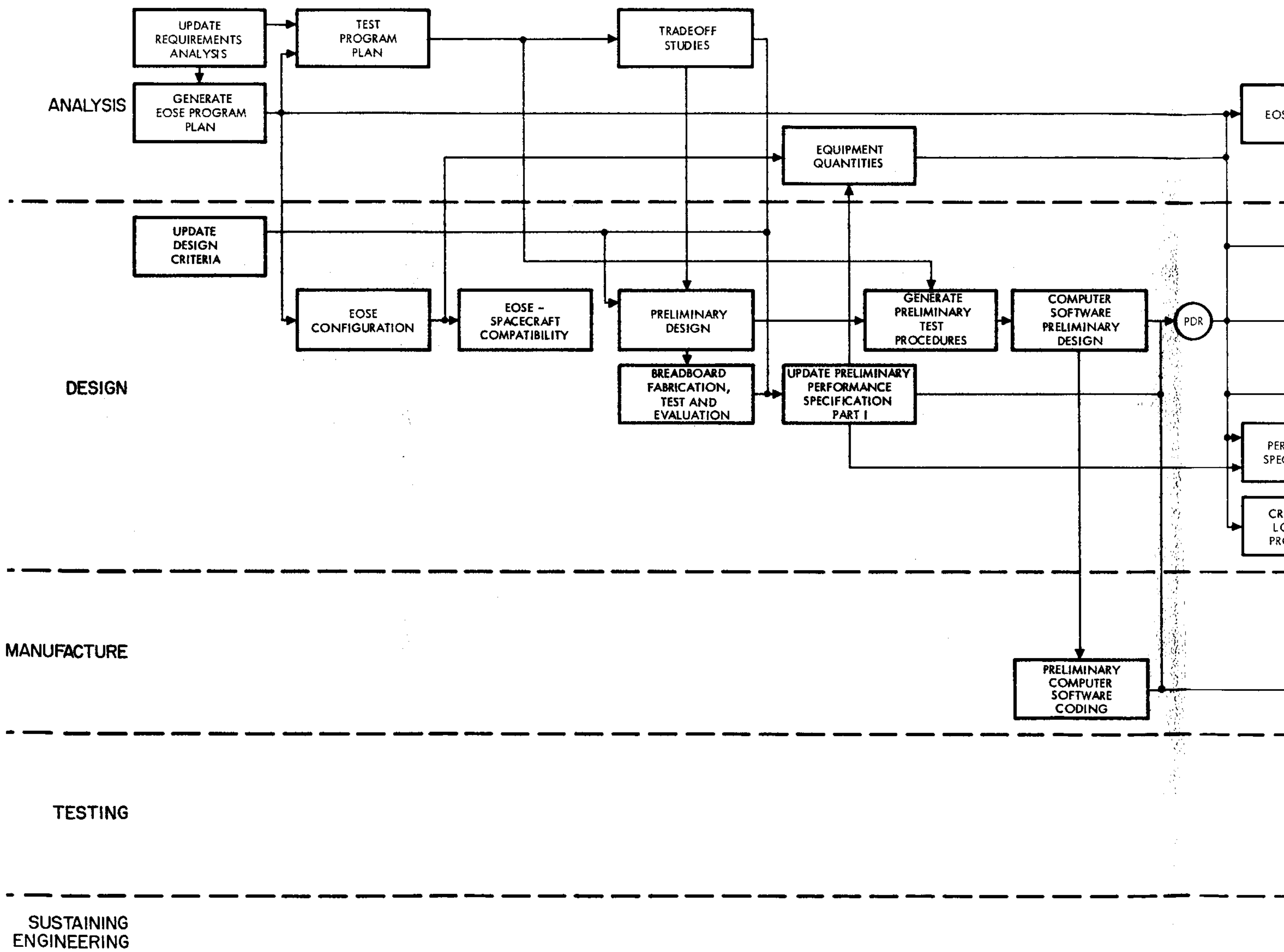


Figure 43. AHSE Milestone Schedule


ANALYSIS
DESIGN
MANUFACTURE
TESTING
SUSTAINING ENGINEERING
UPDATE REQUIREMENTS ANALYSIS
TEST PROGRAM PLAN
TRADEOFF STUDIES
GENERATE EOSE PROGRAM PLAN
EQUIPMENT QUANTITIES
UPDATE DESIGN CRITERIA
EOSE CONFIGURATION
EOSE - SPACECRAFT COMPATIBILITY
PRELIMINARY DESIGN
GENERATE PRELIMINARY TEST PROCEDURES
COMPUTER SOFTWARE PRELIMINARY DESIGN
PDR
BREADBOARD FABRICATION, TEST AND EVALUATION
UPDATE PRELIMINARY PERFORMANCE SPECIFICATION PART I
PRELIMINARY COMPUTER SOFTWARE CODING

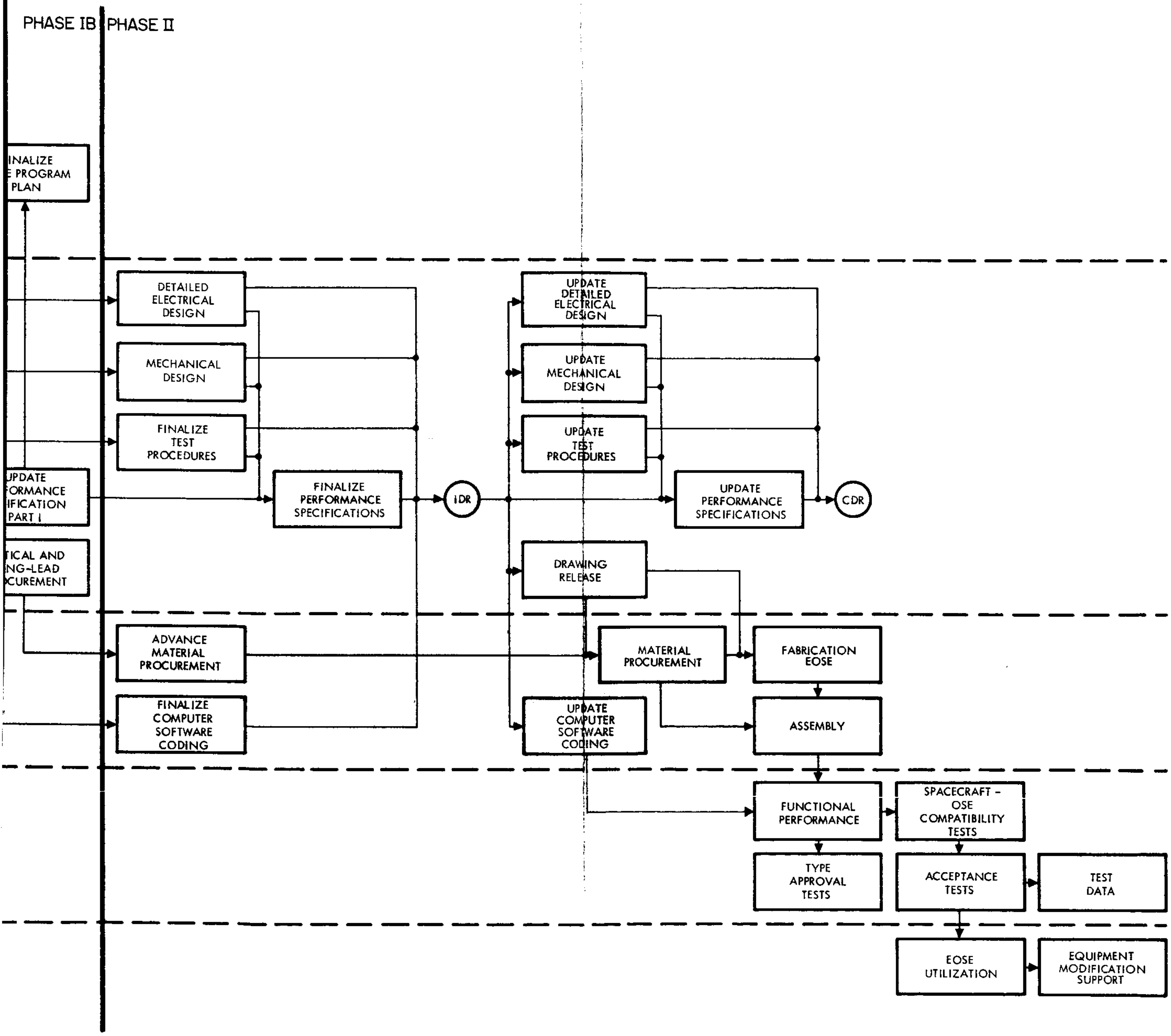


Figure 44. Design and Development of EOSE

2

5.2.1 Analysis

a. Requirements Analysis

The initial EOSE requirements analysis has been completed during Phase IA. Based upon the program guidelines, ground rules, specifications from JPL, and spacecraft preliminary design, the definition of spacecraft requirements and constraints on the EOSE and the EOSE impact on the spacecraft system design have been partially completed. This analysis will continue during Phase IB and, as an iterative process, during Phase II.

Predicted on these same program guidelines and spacecraft over-all test requirements and constraints, the preliminary policy for spacecraft testing has been established, specifying in general what data is to be acquired at each level and location of spacecraft test, how the resulting information is to be handled, and the test techniques to be applied. Final establishment of the test philosophy with required details will be completed in Phase IB.

Following are general areas for continued requirements analysis:

Maintainability. Through additional analysis in self-test capability, multiple use of circuits, human factors, modular construction, and standardization of panel design, maintainability will be enhanced and a minimum mean time to repair will be achieved.

Reliability. Maximum reliability within the constraint of cost will be achieved in the EOSE through further analysis of design simplicity incorporating a minimum of components, minimum component complexity for a given function, proven techniques, circuits, and components, component derating and redundancy of circuits and components when this is established as required.

Environmental. Continued analysis of environmental conditions incurred in EOSE transportation will define the limiting requirements for equipment design. Analysis of temperature, humidity, pressure, shock, vibration, electromagnetic interference, and magnetic field interference at each of the operational locations will be completed as well as those for transportation.

Human Factors. In addition to the maintainability requirements which invoke human factors design, a prime factor to be considered is operability in order that correct communication between man and equipment is achieved. Additional analysis of human factors will provide equipment which is easy and convenient to use.

Packaging. Other than the factor of maintainability, primary packaging concepts will be determined by transportability and best functional performance. Considerations of electrical noise, cross modulation, EMI, and magnetic field interference are paramount in packaging analysis.

Safety. As indicated in the EOSE design criteria of this volume, the safety of equipment and people is also inportant. Specific requirements for safety will be established early in the program as part of safety analysis.

b. EOSE Program Plan Analysis

A preliminary EOSE program plan will be completed early in Phase IB, updated periodically, frozen concurrently with the generation of the equipment hardware and software specifications. Analysis performed in generating this plan will identify all aspects of the EOSE program and will include the following separate subordinate plans analysis:

Configuration Control Plan. For configuration control items to be analyzed are the methods for establishing and maintaining EOSE configuration, use of configuration control documentation, integration engineering, and the communication of this information utilizing computer tab run techniques.

Logistic Plan. Studies in logistics will be aimed at a defining of the spares requirement and training needs for the EOSE at each of the test sites.

Manufacturing Plan. The manufacturing process for EOSE will be described in detail from material procurement through the fabrication steps to equipment assembly and preparation for checkout and calibration.

EOSE Test Program Plan. The work in development testing, fabrication checkout, qualification test, and acceptance tests will be specified. The purpose of this analysis is to assure by tests that the

EOSE is operating properly and can evaluate spacecraft performance status and detect malfunctions. Detailed analysis is required to assure the establishment of a test program capable of spacecraft fault isolation to a replaceable assembly.

Maintenance Plan. The maintenance activities required to minimize the EOSE down-time will be defined.

Launch Operations. Studies will include the installation of EOSE at the field site, the prelaunch evaluation tests, and the actual launch activity.

c. Test Program Plan Analysis

The data needed to develop the test plan requirements, as well as the specific application of this data, will be generated for the following tests:

Spacecraft Design and Development Tests. Analysis of test steps required to evaluate the adequacy of the design of individual spacecraft components prior to their release for fabrication will establish requirements for this part of the test plan.

In-Process and Assembly Tests. Analysis of test steps required during the fabrication process on subassemblies, assemblies, and subsystems (panels) as well as during the spacecraft assembly and integration phases will yield additional test requirements. Consideration of the spacecraft-EOSE compatibility tests will be included in this analysis.

Qualification Tests. Analysis will be undertaken of tests which stress the spacecraft beyond the design limits in order to prove the design margin of safety. Analysis will also stress the additional flexibility and measurement range extension to be provided in the EOSE to support these tests.

Acceptance Tests. Functional tests on the spacecraft units and integrated system will be analyzed to assure that they prove adequate functional performance under flight environmental conditions.

Pre-Launch Field Tests. Analysis will be continued of tests of an assembly and acceptance nature in the launch area at the explosive safe facility, vertical assembly building, and at the launch pad to evaluate functional integrity of the spacecraft.

d. Tradeoff Studies

Several tradeoff studies have been started in Phase IA and will be completed during Phase IB. It is important that the EOSE be considered an integral portion of the overall Voyager system in that EOSE functional requirements will affect the spacecraft design, particularly in obtaining the optimum quantity evaluation information. Each of the tradeoff studies must be concerned specifically with testing at the integrated system, subsystem, and assembly levels in that requirements at these levels may be substantially different. The following factors are included in the tradeoff decisions being made:

Degree of Spacecraft Performance Evaluation. Establishment of the ratio of spacecraft functions tested to those left untested, maximized within the constraints of test data point availability, constitutes a tradeoff between current and future performance confidence and spacecraft statistical reliability.

Level of Fault Isolation. Studies involve determination of a specific level of fault isolation capability and tradeoff between test data point availability, maintenance philosophy, and EOSE capability.

Degree of Automation. Analysis will determine the degree to which the test operations of program control, stimuli generation, and data handling should be carried. This is a tradeoff between time allowed or required for test, operator, and maintenance personnel skill level, requirements for equipment simplicity, flexibility, and repeatability, quantity and rate of spacecraft testing, amount of data to be handled and processed, degree of manual override capability, and other factors.

Method of Test Data Presentation. Considerations here will determine the optimum method of test data presentation, involving tradeoff between various display methods and hard copy or printed test records. The extent of go-no-go indication is included. Digital versus analog, direct measurement versus comparison measurement, permanent recording versus transient recording, and type of recording for future reference and data analysis are also included in the tradeoff.

Alternate Test Methods. Studies will involve various methods of acquiring data such as single stimuli, transfer functional analysis, and continuous monitor, each under nominal or marginal testing conditions.

Test Accessibility. A tradeoff must be made between the accessibility and test data points required for proper spacecraft evaluation and the complexity or other constraints imposed on the spacecraft to achieve this accessibility. The tradeoff includes requirements for isolation of test data points from spacecraft circuitry.

Multiple Use of Equipment. A tradeoff study will be completed in evaluating the quantity of EOSE required as a function of the quantity, schedule, and location of spacecraft test operations, to minimize the quantity of EOSE through multiple use. The study will include sequential time sharing, central data system multiplexing, equipment automation, and refurbishing equipment for subsequent spacecraft models. The impact of spacecraft schedule and, in particular, the launch window impact on EOSE requirements form a major part of this tradeoff.

EOSE Cost. A major factor in the EOSE tradeoffs to be completed during Phase IB is that of cost. A value engineering effort initiated during Phase IA will be continued for the purpose of making an efficient cost tradeoff with each of the factors just discussed. Value engineering check points occurring during the development of the EOSE include: the establishment of equipment capabilities, establishment of equipment quantities, the EOSE requirements on spacecraft design during the EOSE final program plan generation, and the design reviews.

e. Equipment Quantities Analysis

The next phase of the program involves the final determination of EOSE quantity needs. Predicated on the final schedule and quantity of spacecraft, as well as the test philosophy, the quantity of supporting equipment for each of the areas of test will be changed if necessary from that established during Phase IA. The following considerations are important in making EOSE quantity decisions.

Test Area Requirements. Studies will involve requirements for separate equipment to support each of the test sites of factory assembly, environmental test, magnetic test site, launch complex assembly, and the launch site.

Test Area Schedule. An analysis will be made of EOSE quantity as a function of the schedule applicable to each of the test areas and the time-sharing or sequential testing possibilities which may be applied to these areas.

Spacecraft Subsystem Quantities. Consideration will be given to the specific quantities of spacecraft equipment as a prime factor in determining EOSE quantity since it has a direct bearing on the test area schedule.

Spacecraft Subsystem Test Time. Studies in test time are required since test time and hence EOSE quantities are functions of equipment set up, response time of the equipment under test, the degree of EOSE automation, and equipment tear down time.

Utilization Factor. EOSE quantities depend upon the percentage utilization factor of individual items. The higher this factor, the fewer the quantity of specific items. Therefore, studies will be made so that the utilization factor may be increased through proper spacecraft equipment test scheduling, maximum EOSE operability, and sufficient flexibility and mobility of the EOSE.

Cost. As part of continued cost tradeoff studies, consideration has been given to the desirability of establishing an additional test station on the basis that the cost for such a station is less than that incurred by an increase in manpower or material costs through a schedule delay in spacecraft testing operations. Additional effort is required in this area.

Spares. Spare requirements are determined primarily through the maintenance philosophy and plan. The spare requirement is also a function of the packaging techniques. Through adequate module replacement capability, a reduction in equipment downtime is achieved. From continuing analysis of the functional flow diagrams, EOSE end item requirements lists will be developed, broken down to generic groupings. The total allocation for each EOSE item, its location, and whether it will be shared by two or more sites will be established as part of this study.

5.2.2 Design

a. Phase IB

Further identification will be accomplished for each type of EOSE required along with generation of updated design and performance requirement specifications for both hardware and software at the system and subsystem equipment levels.

The criteria to be used by the design engineer will be stated explicitly for each item of equipment. EOSE end item design criteria will be developed from the EOSE design requirement documents. The purpose of the design criteria is to provide controllable documents for each EOSE end item to support the design reviews. In addition, engineering design personnel will have an approved controlled design criteria document for reference. It will consist of individual, written descriptions for each specified item of EOSE. The purpose and use of the specific item will be covered. Spacecraft power and utility requirements affecting the EOSE design will be listed as to power, type, regulation, total load, reserve factors, and duration of use. The design criteria will agree with the decisions made during the requirements analysis tradeoff studies and as defined in the support plans for maintenance, reliability, quality assurance, and test.

In the interest of equipment compatibility and standardization, the preliminary end item equipment specification will define the preliminary end item configurations as a guide to the design engineer. The end item configuration will be consistent with the requirements established during the analysis and with those defined in the maintenance and configuration control planning documents.

Design work in the areas of subsystem test sets, system test complex, launch complex equipment, and mission dependent equipment will be carried forward in Phase IB including breadboarding of certain critical items to permit proper definition of specifications and to assure confidence in associated cost estimates. Emphasis on the establishment of general test procedure documentation is to permit early development of computer software. The following sequence of design tasks will be performed in the development of subsystem designs.

EOSE design layout drawings will be prepared for each item of EOSE that has been outlined in the design criteria. These preliminary drawings, accommodating all constraints and fixed spacecraft dimensions, will be prepared and issued for TRW review, coordination, and preliminary approval. A major purpose of the preliminary drawings will be to obtain preliminary agreement on each EOSE item and to enable interface checking on the interacting EOSE items. Following review, final detailed layout drawings will be produced. Alternate design solutions to problem areas will be developed. Each solution will undergo detailed review and trade-off analysis as to its fulfillment of the design criteria. To minimize the design effort time, judicious use of previously designed circuitry or complete units from previous programs will be included to the maximum extent possible. The use, where practicable, of general purpose commercial equipment and the application of developed and proven techniques EOSE will help keep program costs down and assure compliance with delivery schedules.

EOSE functional drawings will be prepared and supplemented by functional descriptions as required in order to document the conceptual design in preparation for the preliminary design review on the EOSE. Documentation will be sufficient to identify the equipment plan and outline the functional capabilities and the way these satisfy the established spacecraft test requirements.

Development testing activities in the EOSE program will be started during Phase IB but will be limited to critical areas not already evaluated. Development testing will be accomplished during the EOSE design phase and will include performance tests.

Mechanical design, including rack and panel layout, interconnecting cabling, and handling equipment will be initiated and coordinated with human factor design to assure the correct man-machine communication and test control efficiency. Subsystem layouts of the individual approved EOSE items will be initiated also.

Detailed test procedures will be initiated for the EOSE qualification and acceptance tests. Completion of test to these procedures, written to satisfy the requirements stated in the test program plan, will provide assurance that the EOSE is in the proper status for connection to the spacecraft or its subassemblies.

General studies conducted during Phase IA will be completed in the selection of a computer as the basis of the proposed automated checkout system. Flow charting of preliminary test procedures will proceed following computer selection.

Design audits and the preliminary design review will be accomplished during Phase IB. The preliminary design review will follow definition of EOSE functional requirements and the generation of EOSE functional block diagrams.

Specifications will be updated which fully define performance requirements for compatibility with spacecraft and other interfacing equipment.

Soon after the start of EOSE design, a parts analysis will proceed, involving the review of preferred parts lists for application of these parts to the EOSE. A list for advanced material procurement will be generated including only that equipment of low-risk use and long-lead procurement. This list will be made available as one of the last activities in Phase IB so that advanced material procurement can be undertaken immediately upon authorization in Phase II.

b. Phase II

The design activities during Phase II will in all cases continue the initial steps taken in Phase IB.

Power design, logic design, and wire lists will be completed and included in the design review package assembled prior to the critical design review. Detailed drafting and mechanical design will be completed, and test procedures will be completed with concurrent coding of computer software. Final specifications will be prepared which define the end item configuration of the EOSE to include interface definitions and control, both internal and external, for compatibility with spacecraft and other interfacing equipment

The intermediate design review will follow the completion of fabrication drawings and their preliminary release, for which the completion of

EOSE design and development testing are requirements. Design documentation will be prepared as required to describe the EOSE design status at these phases. Requirements for changes resulting from the design review will be reflected in EOSE design drawings and documentation. The critical design review for EOSE will be held at essentially the same time as the spacecraft subsystem CDR's to acknowledge the incorporation of all design changes to the EOSE determined during comprehensive testing of the engineering model spacecraft.

5.2.3 Manufacturing

Electrical and mechanical manufacturing activities are performed exclusively in Phase II except for some computer software coding. Design and vendor control drawings will outline in detail the components required and will contain all applicable specifications controlling fabrication, manufacture, quality control, and documentation. Test equipment required to support in-process and fabrication checkout tests is primarily of the general purpose, capital equipment type. The following are the specific tasks to be completed in the EOSE manufacture.

a. Preliminary Computer Software

During Phase IB, coding of the relatively simple off-line programs will be completed to permit concentration during Phase II on the more complex on-line processors and the on-line executive program.

b. Advance Material Procurement

Placement of orders for material on the list of critical long lead procurement items will be completed in Phase IB to assure receipt in time to meet fabrication and assembly schedules.

c. Drawing Release

Preliminary and final drawing releases will be accomplished prior to and following the intermediate design review. Advanced release, as previously stated, is necessary on some equipment to preclude delays in material procurement and fabrication. Drawing release for fabrication constitutes that point in the design where formal approval of JPL is required for any subsequent engineering changes.

d. Material Procurement

The final material procurement activities will follow the critical design review and the preliminary and final release of drawings. Parts and materials will be selected from the preferred parts list or identified under material control drawings.

e. Fabrication Planning

A fabrication plan will be initiated before the release of drawings as detailed in the Manufacturing Plan. It is planned that fabrication will be accomplished under the cognizance of the engineers who are responsible for the design. Fabrication requirements, therefore, may be determined well ahead of drawing release dates. An expedient fabrication schedule is imperative in that EOSE is required by the time Voyager flight vehicle equipment is ready for test even though EOSE design completion must await the firm vehicle design. As a part of the fabrication planning, a determination will be made in detail where each part will be fabricated and the extent to which each part will be inspected and tested. Materials and parts will be subjected to incoming inspection and test in accordance with Voyager quality assurance practice.

f. Detailed Part Fabrication

Individual components and modules will be procured or fabricated as required, and will be subjected to in-process tests and inspection. Extensive use of plug-in modules and circuit cards is anticipated, and verification of functional performance of these items will be made prior to next higher assembly.

g. Subassembly Fabrication

The fabrication of drawers will be scheduled so that those drawers required for testing of other or subsequent drawers will be available first. During integration of EOSE modules into drawers, each functional group will be examined for interference as evidenced by performance degradation. In-process tests and inspections will be employed to validate this integration, including electrical and mechanical interfaces. Any new interface problems becoming evident at this time will be corrected.

h. Assembly

Fabricated drawers will be integrated along with general purpose equipment into specific EOSE racks with in-process tests and inspections employed as required. Integration and interface testing during this activity will verify proper functional operation of constituent assemblies. Following fabrication of assemblies, the final debugging operations and proving of the fabrication integrity will be done on completely assembled equipment. Any final manufacturing errors will be corrected and the equipment adjusted and calibrated in preparation for the qualification and acceptance tests to follow.

i. Computer Software Coding

Coding activity started in Phase IB will be completed in Phase II. Following final debugging this software will become an essential tool along with the computer in performing the various tests on EOSE, the spacecraft, and the mission dependent equipment.

j. EOSE Instructions

Supporting documentation in the form of hardware and software instructions, test programs, manuals, training plans, and materials will be generated as required to supplement the EOSE. The documentation will be the minimum required to efficiently utilize the EOSE and to meet program specified requirements.

5.2.4 Testing

Before it is connected to a subsystem of the spacecraft, a complete acceptance performance test will be conducted on each EOSE test set to ensure that no damage to development or flight hardware can occur. The performance test will be designed to check all interface and operating parameters. The accuracy of all stimuli parameters, and simulated loads, as well as the accuracy of measurement and recording functions will be determined under various specified conditions such as high and low line voltage and ambient temperature. This test will be conducted at the component (chassis) level as well as at the system level. The accuracy of self-test loops will be demonstrated and interface impedances measured. The adequacy of protective fail-safe circuits will be verified.

Environmental tests appropriate to the particular EOSE item will then be performed. Each item will be fully tested under the environmental conditions in which it will be used and then inspected and tested for indications of degradation.

In-process tests will be accomplished on modules and cards of the EOSE during manufacture and on assemblies of these items in concurrence with quality assurance requirements. Through these tests, confidence is acquired that equipment has been fabricated to quality standards and that no manufacturing defects exist.

Functional performance tests will be made on each element of the EOSE as required at various stages during manufacturing and through acceptance tests. Such performance tests are also run following any calibration adjustment or to verify that the EOSE is suitable for formal test or utilization with the spacecraft.

Certain EOSE will be subjected to type approval tests in accordance with test requirements. The tests will follow a JPL-approved test procedure. The equipment will be subjected to environmental stress and its capability to withstand the rigors of these environments will be evaluated before, during, and following the test through measurements of its functional performance.

Compatibility tests follow fabrication of the EOSE and any subsequent modification, to verify electrical and mechanical compatibility with the spacecraft and interfacing equipment. These tests verify interface connections, ability of the EOSE to control and monitor the spacecraft, and ability to protect against possible degradation of the spacecraft. The first tests are accomplished with simulated spacecraft models and circuits. followed by connection to the spacecraft in planned steps. This procedure is also applicable to establishing the compatibility between the subsystem EOSE and spacecraft subsystems.

Each of the fabricated EOSE drawers, subsystem test sets, and system test complexes will be subjected to acceptance tests, including those which have undergone qualification tests. Acceptance tests will be performed in accordance with established test plan, as defined by the program guidelines, and will be run according to JPL-approved acceptance test procedures. Acceptance tests will evaluate the final operating integrity of the EOSE prior to use with the spacecraft and other interfacing equipment.

During each of the tests imposed on the EOSE, test data will be recorded for detailed evaluation of EOSE status. This test data will be retained for later comparative analysis and reliability evaluation. Test data will be taken by equipment integrated into the EOSE and by supplementary capital equipment connected to and calibrated with the EOSE.

5.2.5 Sustaining Engineering

The EOSE at the subsystem and system test levels will be utilized in support of factory spacecraft testing operations. The development, in-process fabrication, integration, qualification, and acceptance testing operations on the spacecraft will be supported by this equipment. The EOSE to support field test and launch operations of the spacecraft will be shipped to the field and installed sometime before the launch activity. Following installation, the equipment will be calibrated and retested to verify that no degradation has developed during shipping and handling. The field spacecraft activities which require this support include: the SAF tests, the ESF tests, VAB tests, the on-stand flight readiness evaluation, and the launch control operations.

As a direct consequence of changes to the spacecraft, modifications may be required to the EOSE used at each level of test. The modification process will follow this EOSE implementation plan in similar sequence, but necessarily of abbreviated scope. Following a definition of EOSE modification requirements, an engineering change order will be written to initiate detailed design effort on the change. A modification kit will be procured and shipped to the EOSE location for installation in the affected equipment. Documentation changes and additions will parallel the build-up of the modification kit. Installation instructions will be generated, the model specification on the equipment will be revised, and test procedures for checkout and acceptance of the revised equipment will be prepared. Following installation of the modification kit, the acceptance test procedure will be accomplished under the cognizance of quality assurance as before. Where required, other test procedures may be run. Records of the test results following modification will be retained with the other EOSE test records.

Sustaining engineering on the Voyager EOSE will continue under the cognizance of quality assurance, as have all activities prior to delivery of the equipment. Configuration control will be maintained, in accordance with the configuration control plan, over all activities involving the EOSE following equipment delivery to the customer.

It is anticipated that the flexibility and capability initially designed into the EOSE will be adequate to preclude modifications required by the majority of the possible changes to the spacecraft design.

5.3 AHSE Development Flow

The AHSE task flow shown in Figure 45 depicts the relationship of tasks to be performed in Phases IB and II. The flow indicates only first level details; additional details for each event shown will be provided during Phase IB. The format is identical to that used in the task flows of Volume 3 of this report.

5.4 AHSE Tasks

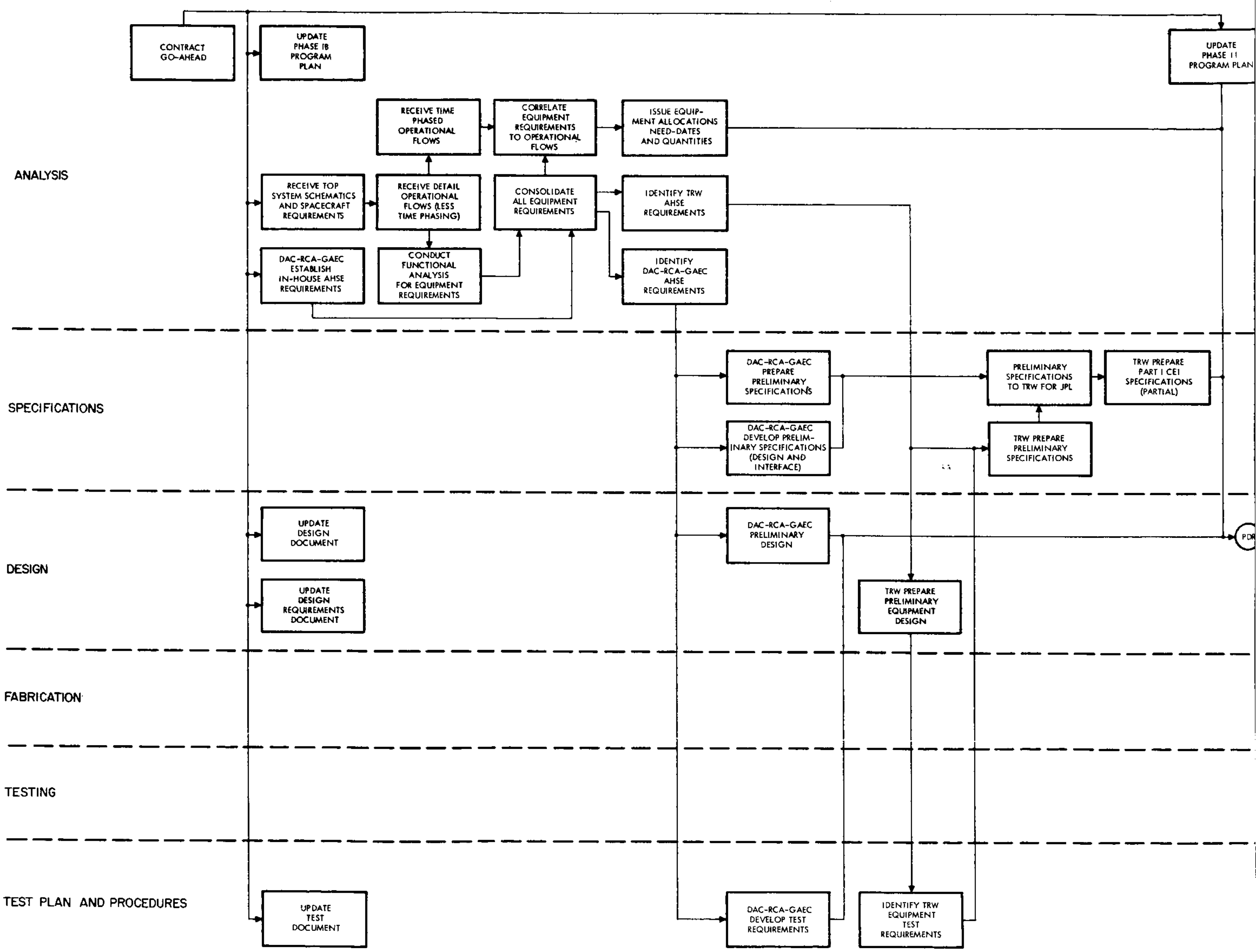
This section breaks down AHSE activities of Phase IB and Phase II by operational sequence and by responsible sectors. It describes and categorizes all work necessary to assure that functionally adequate AHSE, in adequate quantities, will be at the right place when needed.

Phase IB tasks begin with functional analyses, move through preliminary designs and reviews, and after establishing Phase II work packages, end with the development of Part I preliminary specifications for all configured end items. Phase II tasks begin with detailed designs of end items, establish procedures for test and fabrication, and analyze quantities and allocations.

5.4.1 Phase IB

a. Functional Analysis

The detailed operational flows for each spacecraft from assembly and checkout through system test and launch operations will


ANALYSIS
SPECIFICATIONS
DESIGN
FABRICATION
TESTING
TEST PLAN AND PROCEDURES
CONTRACT GO-AHEAD
UPDATE PHASE IB PROGRAM PLAN
UPDATE PHASE II PROGRAM PLAN
RECEIVE TIME PHASED OPERATIONAL FLOWS
CORRELATE EQUIPMENT REQUIREMENTS TO OPERATIONAL FLOWS
ISSUE EQUIPMENT ALLOCATIONS NEED-DATES AND QUANTITIES
RECEIVE TOP SYSTEM SCHEMATICS AND SPACECRAFT REQUIREMENTS
RECEIVE DETAIL OPERATIONAL FLOWS (LESS TIME PHASING)
CONSOLIDATE ALL EQUIPMENT REQUIREMENTS
IDENTIFY TRW AHSE REQUIREMENTS
DAC-RCA-GAEC ESTABLISH IN-HOUSE AHSE REQUIREMENTS
CONDUCT FUNCTIONAL ANALYSIS FOR EQUIPMENT REQUIREMENTS
IDENTIFY DAC-RCA-GAEC AHSE REQUIREMENTS
DAC-RCA-GAEC PREPARE PRELIMINARY SPECIFICATIONS
PRELIMINARY SPECIFICATIONS TO TRW FOR JPL
TRW PREPARE PART I CEI SPECIFICATIONS (PARTIAL)
DAC-RCA-GAEC DEVELOP PRELIMINARY SPECIFICATIONS (DESIGN AND INTERFACE)
TRW PREPARE PRELIMINARY SPECIFICATIONS
UPDATE DESIGN DOCUMENT
DAC-RCA-GAEC PRELIMINARY DESIGN
UPDATE DESIGN REQUIREMENTS DOCUMENT
TRW PREPARE PRELIMINARY EQUIPMENT DESIGN
UPDATE TEST DOCUMENT
DAC-RCA-GAEC DEVELOP TEST REQUIREMENTS
IDENTIFY TRW EQUIPMENT TEST REQUIREMENTS

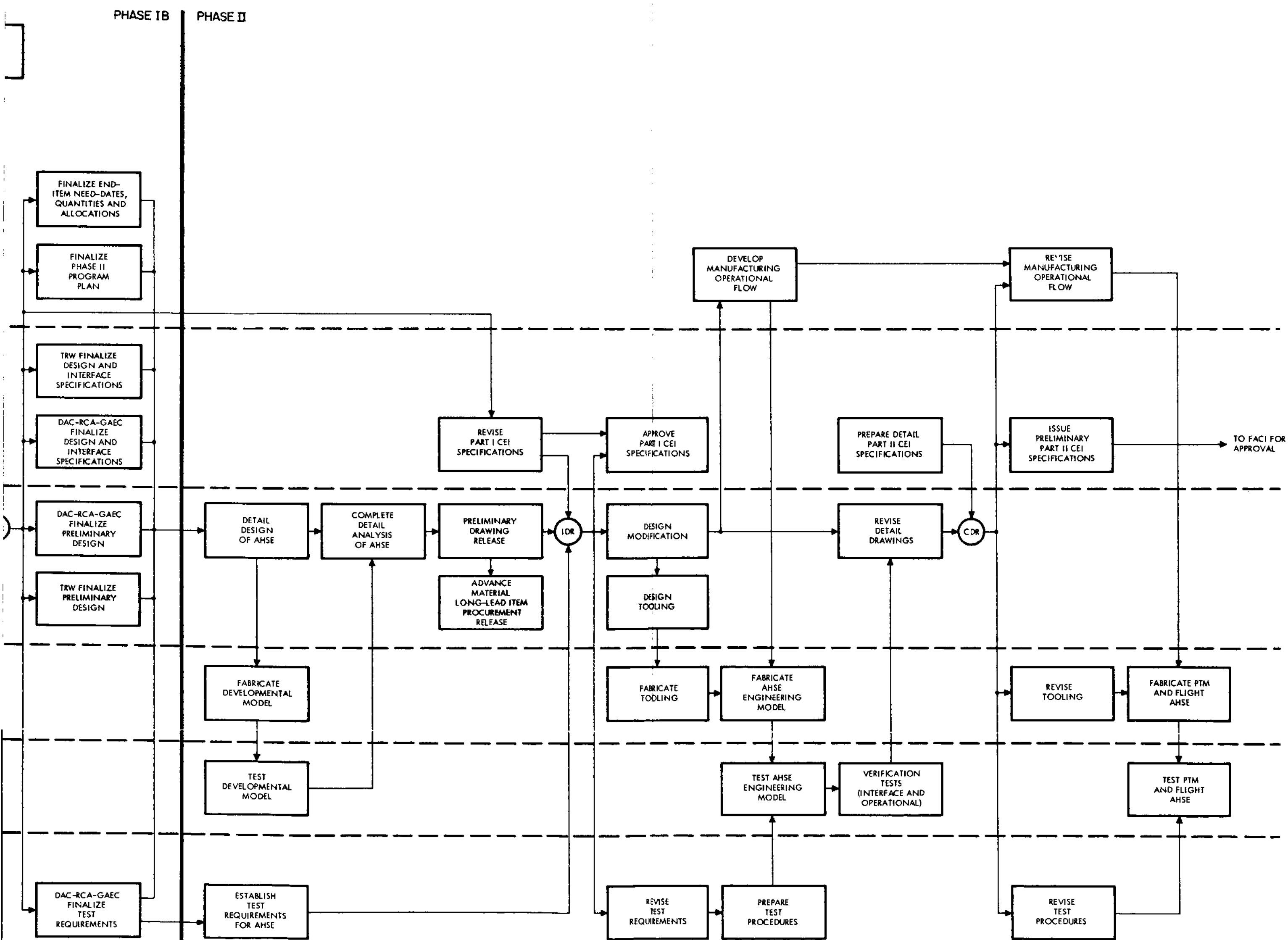


Figure 45. Design and Development of AHSE

2

be analyzed to determine specific functional requirements in terms of equipment and facilities. The equipment will be identified by type: i.e., mechanical spacecraft support equipment, special test equipment, special tooling and simulator/model, distinguishing those which do and do not leave the TRW facility. Figure 46 shows the form used to record the functional

DETAILED FUNCTIONAL FLOW

OPER NO.	TASK DESCRIPTION	FACILITY REQUIRED	REQUIRED OSE LISTED BY EOSE AND AHSE NUMBERS (SEE APPENDIX A FOR LIST)	APPLICABLE SPACECRAFT					
				EM	PTM	PIM	FL1	FL2	FL3

Figure 46. Voyager Form for Detailed Functional Flow

analyses, correlating each operation, its support equipment, facility requirements, and type designation. This form and the others described here have been developed during the Phase IA effort as tools to systemalize the approach to all AHSE activities.

With functional requirements established for AHSE and facilities, contractors will be identified for each item and conceptual design requirements defined. The form shown in Figure 47 will be used to list all AHSE

Type of Equipment ____________ Date ____________

Page ____ of ____ Rev. Ltr. ____________

Item No.	Nomenclature	Functional Requirements	Oper. Flow No.	Conceptual Design Requirements	Contractor
5001	TRW Electrical Prime Mover	Provide towing of vertical checkout and assembly stand and other AHSE within the assembly and checkout area	1-178	Must be electrical type due to clean room requirements	TRW
5003	Vertical Checkout and Assembly Stand	Provide a stand for assembly and checkout of the spacecraft-planetary vehicle. Also provide for spacecraft-planetary vehicle alignment capabilities	1-178	Must be rigid and allow for deployment of antennas. Must also be mobile and capable of supporting the aft equipment module as well as the spacecraft-planetary vehicle	DAC
5018	Magnetic Test Fixture	Support the planetary vehicle during magnetic testing	28-33, 67-105	Capable of rotating the planetary vehicle about either axis, nonmagnetic material shall be used	TRW

Figure 47. Voyager Form for AHSE Functional and Conceptual Design Requirements

and correlate each functional and/or conceptual design requirement with a particular contractor or subcontractor. The form shown in Figure 48 is used to correlate specific AHSE with detailed operational flows, and to incorporate identification numbers and nomenclatures.

b. Preliminary Design

After the analyses of load, material, strength, dynamics, weight, reliability, maintainability, human factors, and safety, conceptual designs of AHSE will be developed. Upon review and approval, these designs will go into preliminary layouts, which will in turn be reviewed and approved for generation of Part I CEI specifications.

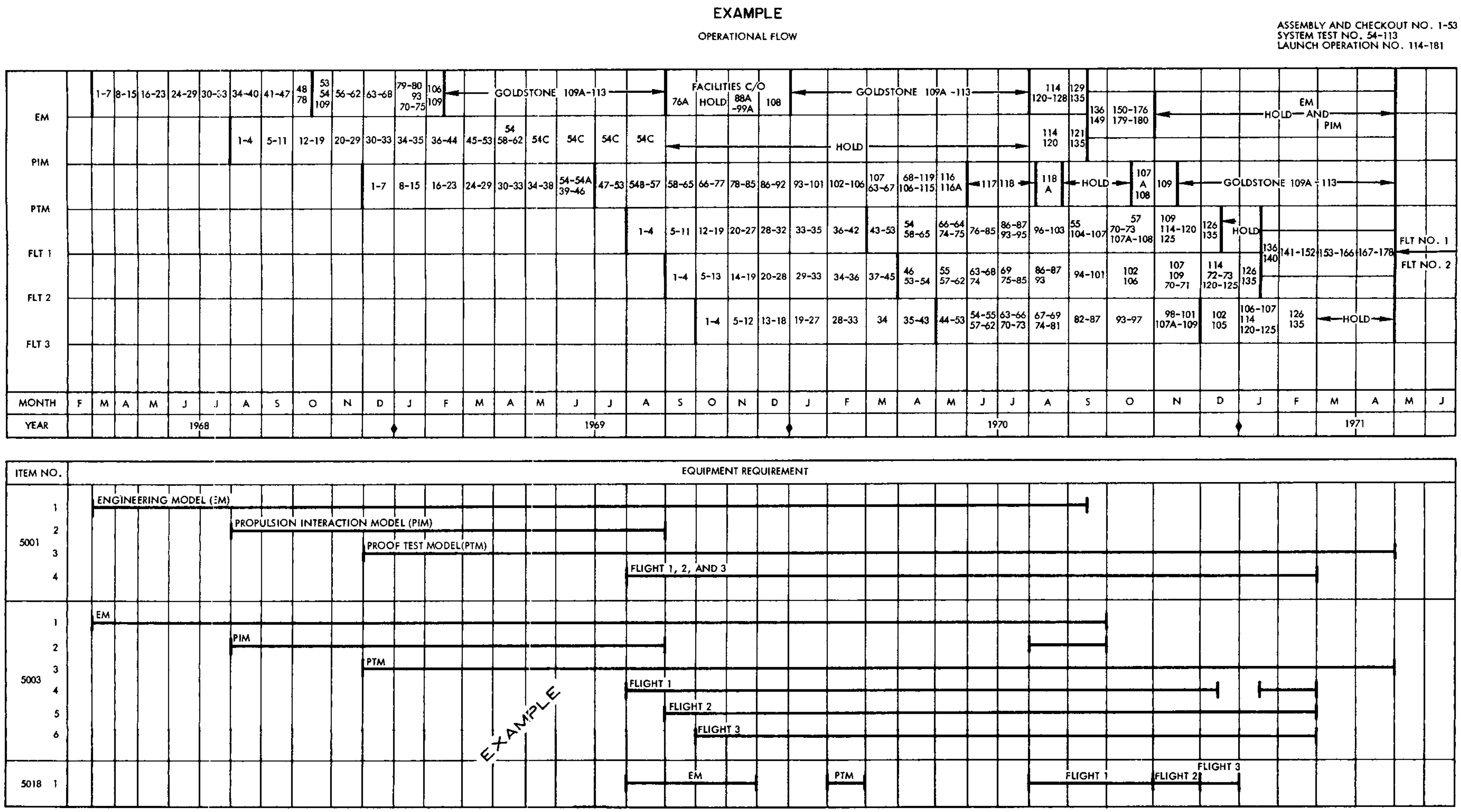


Figure 48. Form for Correlation of AHSE with Operational Flow

c. Allocations, Quantity, and Need-Dates

AHSE and facility identifications will be correlated with time-phased operational flows for analysis of quantities and need-dates. Figure 49 identifies quantities, need-dates, and contractor responsibilities, and identifies allocations.

d. Scheduling Activities

After generating schedules for design and development, test, and manufacturing of each item of AHSE, PERTs will be developed to satisfy quantities and need-dates.

e. Preliminary Design Review

A preliminary design review will be conducted on all documents generated in previous efforts, in order to issue revised preliminary designs, allocations, PERT's, and specification of Part I CEI's.

f. Phase II Work Package and Cost Plan

Part I, CEI specifications will be analyzed for breakdown into Phase II work packages and cost plans, including subwork packages for functional elements of TRW and subcontractors, and coordination of data inputs from each.

g. Subcontractor Liaison

Tasks required for proper liaison with subcontractors are analagous to those listed above.

h. Management Activities

Throughout Phase IB, supervision of AHSE activities will take many forms and utilize many functional elements of TRW. Tasks providing guidance and decisions for conduct of the Phase IB program include the following:

- Set up forms and procedures for all Phase IB functions
- Continuously monitor progress of Phase IB tasks
- Monitor expenditures for Phase IB
- Allocate funding budgets to functional elements of TRW
- Conduct design audits

AHSE REQUIREMENTS

		1968											1969												1970												1971						
NO.	NOMENCLATURE	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	J	J	A	S	O	N	D	J	F	M	A	M	TOTAL	CONTRACTOR
5001	TRW ELECTRICAL	1					1				1								1																							4	TRW
	PRIME MOVER																																										
5003	VERTICAL CHECKOUT	1					1				1								1	1	1																					6	DAC
	AND ASSEMBLY STAND																																										
5018	MAGNETIC TEST FIXTURE																		1																							1	TRW

AHSE ALLOCATIONS

AHSE NO.	SER NO.	NOMENCLATURE	ALLOCATION					
			EM	PTM	PIM	FLIGHT NO. 1	FLIGHT NO. 2	FLIGHT NO. 3
5001	1	TRW ELECTRICAL PRIME MOVER	3/68					
5001	2	TRW ELECTRICAL PRIME MOVER		12/68				
5001	3	TRW ELECTRICAL PRIME MOVER			8/68			
5001	4	TRW ELECTRICAL PRIME MOVER				8/69	9/69	10/69
5003	1	VERTICAL CHECKOUT AND ASSEMBLY STAND	3/68					
5003	2	VERTICAL CHECKOUT AND ASSEMBLY STAND		12/68				
5003	3	VERTICAL CHECKOUT AND ASSEMBLY STAND			8/68			
5003	4	VERTICAL CHECKOUT AND ASSEMBLY STAND				8/69		
5003	5	VERTICAL CHECKOUT AND ASSEMBLY STAND					9/69	
5003	6	VERTICAL CHECKOUT AND ASSEMBLY STAND						10/69
5018	1	MAGNETIC TEST FIXTURE	8/69	2/70		8/70	11/70	12/70

EXAMPLE

Figure 49. Form for AHSE Requirements and Allocations

- Review cost audits and PERT
- Coordinate PDR support requirements
- Conduct PDR
- Establish Phase IB Task

i. Development Preliminary Specifications

The final task in Phase IB is the forwarding to JPL of the preliminary specification of all AHSE end items, in appropriate format, and including completion schedules and required data on performance, design and construction, and quality assurance provisions.

5.4.2 Phase II

Phase II of the AHSE activities, like all other parts of the Voyager spacecraft project, takes the plans generated in Phase IB and utilizes them as the basis for developing the hardware. The tasks break down into design, test, manufacture, and management.

a. Design

End items fully specified during Phase IB will go into detailed design, producing detailed drawings and a drawing breakdown for each end item. Preliminary equipment/identification specifications (Part II CEI's) will be prepared for each item as appropriate. Spares and maintenance requirements for each item will also be noted. Critical design review of AHSE will be conducted to convert detailed designs into production drawings, incorporating changes in specifications and the maintenance spares plan.

b. Test Activities

Detailed test procedures covering proof-testing, interface compatibility testing, and acceptance testing, will be generated for each AHSE end item. The test procedures will include check list, data, and signature provisions, and identify the step-by-step functions required. Tests required to develop and verify designs will be conducted prior to the CDR. Otherwise CDR will make the test procedures firm.

c. Fabrication Activities

Fabrication tasks cover the design and fabrication of application aids as well as fabricating the end items. Included in the effort are detailed fabrication and assembly, review of test data, and necessary paper work required for delivery of end items.

d. Management Activities

The management tasks performed in Phase IB will be continued in Phase II, with the intermediate design review and the critical design review marking special tasks.

6. QUANTITY AND UTILIZATION

Table 6 is a tabulation of EOSE quantities, categorized by type and location of equipment. The AHSE quantities are as noted in the equipment list in Section III. These quantities were derived from correlating the equipment required to the operational flow. Examples of the correlated equipment required and allocation data are included in Figures 48 and 49. The quantities noted in the equipment list are limited to the equipment required for spacecraft assembly and checkout, system test, and launch operations. The AHSE required for subassembly development, assembly, manufacturing, and testing are not included in the list. It is planned that the subsystem-peculiar equipment, i.e., equipment requirements for subsystem development, etc., as well as subsystem requirements of some of the items given in this list will be integrated into a composite AHSE equipment list during Phase IB.

Table 6. Estimated Quantities of EOSE

		STC	LCE[a]	MDE	MOSTE
TRW	EM	1	1	1[b]	1[b]
	PTM	1	—	1[b]	1[b]
	Flight 1, 2, 3	3	—	1[b]	1[b]
	Spare	1	—	1[b]	1[b]
Eastern Test Range (ETR)		—[c]	2	—	—
DSIF		—	—	12[d]	12[d]
Total		6	3	18	18

[a]LCE-peculiar equipment only; the STC that is part of each LCE is included in the STC column.

[b]Part of STC.

[c]STC identified with spacecraft; no additional STC required at ETR.

[d]Madrid, Canberra, Johannesburg, Goldstone, Woomera.

APPENDIX A

FACTORS INFLUENCING CHOICE OF BASELINE STC CONFIGURATION

Voyager spacecraft system test requirements, either established by JPL or arising from the details of the spacecraft design adopted, strongly influenced the design of the STC. However, many areas of choice remained during alternatives which would satisfy these requirements, and therefore tradeoff analyses were needed. In arriving at the baseline configuration described in Section 3. 3. 1 of this volume, three major choices were made, regarding:

- Extent of automation
- Type and amount of backup
- Whether to include a set of MDE and MOSTE

1. AUTOMATION

The need for a computer in the STC was never at issue. The required tasks which the computer can readily handle are:

a) Decommutation of telemetry data, comparison of that data with limits, conversion to engineering units including calibration correction, and driving displays

b) Organizing and maintaining a test log, including telemetry, hardline data, commands, and manually entered data, and formatting the log data for printout and more complex (CRT) display

c) Generating simulated data for test use and comparing the same data from the spacecraft for error rate testing (e. g., loading, unloading, and comparing sequencer memory and data stored in tape recorders)

d) Fault isolation logic

e) Checking of spacecraft commands for forbidden combinations of commands activated during any particular test sequence.

f) Trend analysis of data

Although the tasks could be done without a general-purpose computer, completing them on a real-time basis would clearly require an excessive number of personnel and large amounts of special purpose logic. The choice was, rather, given a general-purpose computer what level of emphasis should be placed on manual versus automated approaches to test sequencing, patching of EOSE measuring and stimulation equipment into test configurations, and logging test measurements and evaluating them for status presentations.

A number of subsystem and system test parameters needed to be considered with respect to this choice of automatic versus manual control. The effects on each parameter of the two alternatives are simply listed in Table A-1. In general, the comparison seemed to indicate that automatic checkout is superior to manual in that the testing performed is faster (encouraging more exhaustive and more frequent testing), more dependable with respect to the way it is performed and recorded, and less likely to result in spacecraft damage from procedural errors. It is inferior in that unexpected conditions are more likely to go unrecognized, total program costs attributable to system test are likely to be higher (in spite of saving test man-hours), and automatic test equipment is more difficult to produce on a short schedule. Equipment reliability (as distinct from total test reliability) is worse for the automatic equipment, by virtue of the difference in component population, although measures can be taken to combat this problem (backup modes, redundancy, conservative logic design, etc.).

The answer to this question of automatic versus manual testing rests on whether the probability of mission success is increased by the advantages of automatic testing and, if it is, how the disadvantages can be minimized.

It is reasonable to assume that chances for mission success are improved if more tests of a meaningful nature are performed more frequently (since wearout is not a problem with present spacecraft electronic subsystem designs). With more tests being performed, more data is gathered, permitting better statistical and trend analyses. Also more types of tests are performed more frequently when they are short and easy to perform. Determination of a quantitative increase in the probability of mission success for each increase in testing quality or amount is essentially impossible; but a qualitative increase is apparent.

Table A-1. Automatic Versus Manual Testing Tradeoffs

Parameter	Automatic	Manual
Testing speed	Limitations in this case will only be transient settling times in spacecraft and OSE, and in command times when RF commands are used.	Limited by operator speed—much slower than automatic.
Test condition repeatibility	Limited only by stability of test equipment. Requires configuration control of software to same degree as hardware.	Limited by care exercised by operator. Can be controlled by discipline in use of written procedures
Requirements on operating personnel	Reduces actions required, but frequently encounters resistance to use in place of familiar manual methods, especially if initial integration encounters problems.	Increases number of personnel required, but requires shorter time to build up confidence of experienced personnel in test methods. Test personnel qualifications required are higher.
Test documentation	Excellent, if analysis preceding software design is accurate in predicting operational conditions and procedures.	One of the major difficulties of manual test systems. Discipline in test result reporting must be constantly monitored. Tendency not to record transient or unexplainable events.
Flexibility	In practice less flexible than manual because of additional problem of unforeseen effects of program changes.	Difficulties in implementing changes in test procedures or test equipment dependent on change control procedures in effect.
Spacecraft damage potential	Little danger. Reaction time shorter than manual and shutdown procedures more reliable.	Depends entirely on skill, alertness, and reaction time of operators. Reaction time inevitably longer than automatic.
Fault isolation ability	Much faster, but accuracy depends on skill in analysis of failure modes and symptoms, which is done in parallel with spacecraft equipment development.	Depends on skill of operators, but improves rapidly with time, as operators gain experience with spacecraft.
Reliability	Equipment reliability is worse because more equipment of greater complexity is involved, but total test process reliability may be better because of reduced opportunities for human error.	Equipment reliability better because equipment is simpler, but fault may go unrecognized longer because selfcheck is not automatic. Human error a greater problem.
Recognition of unexpected conditions	Depends entirely on skill of system designer—usually system is limited in this respect.	Depends on skill and alertness of operators, but normally much better than automatic system.
Development cost	Substantially greater; software costs can equal computer equipment costs.	Less, expecially if tests can be configured to use commercial equipment.
Total program cost	Higher, but difference from manual reduced by lower testing time and fewer operating personnel.	Probably lower than automatic, in spite of increased manhours and level of personnel per test, unless number of spacecraft is large.
Development schedule	Longer, and more difficult to compress, because people needed are more skilled and must be versed in total-system details. Integration with spacecraft normally takes longer.	Tends to be more easily separable into parallel segments, and design is less critically dependent on exact test procedures to be used.

The problem of decreased checkout equipment reliability is resolved in one manner in the following section; on backup. In addition, design techniques are used which should alleviate reliability problems (particularly in digital design and components, the area of most of the increased complexity).

The final question is whether the increase in probability of mission success is worth the incremental cost, assuming that resources are available to make the development cycle fit into the over-all schedule. Schedule time is adequate for development of the automatic equipment, but available launch window time is quite limited. Hence it appears that mission success can be significantly improved by shortening launch pad checkout, repair, and replace time. The selected baseline STC configuration therefore includes automatic checkout.

2. BACKUP

The types of failures to be expected in an automatic checkout system are:

a) Failure in one of the subsystem EOSE components (stimuli generator or measuring device) or cables to the spacecraft

b) Failure in an in-line component of computer hardware

c) Failure in computer software (the program encountering an unforseen combination of conditions not uncovered during debugging)

d) Failure at the interface between the computer and subsystem EOSE

e) Failure in the ability of the computer to communicate with the operator (i. e., the displays on data entry devices)

It is obvious that all these types of failures do not have the same consequences, and that the kinds of consequences depend on the use being made of the OSE at the time. If we consider, for example, the case of system test being conducted at the time of failure, failures of types (b), (c), or (e) are clearly the most serious. With the other types of failures, testing can continue, skipping the subsystem tests affected. (Note, however, that some subsystem EOSE functions are central to almost any phase of system test; for example telemetry detection.) Relative to these three types of failures there can be either:

a) No backup, operation not possible without repair, system test delay accepted until failure is repaired

b) Redundant elements, providing a backup identical element for each element (or selected critical elements) with automatic or manual switchover

c) Backup to allow manual operation in a degraded mode until the fault can be repaired

Items (b) and (c) are not mutually exclusive, and some combination of them is possible. However, in evaluating possible baseline configurations, all three are considered separately against three major tradeoff parameters: first, the delay in system testing which will result from a failure; second, the value of test results taken during the time the failure exists; third, the incremental cost of implementing each alternative.

The performance of the alternatives with respect to these three parameters is summarized in Table A-2. The descriptions there indicate that the conclusions reached will depend on the details of the system design, and therefore are likely to change as the design evolves. For example, it may be practical to consider switching a computer from one STC to another if the computers for more than one STC are located in the same room and one of the STC's is not in use.

Manual backup is effective in allowing the continuation of degraded mode testing during computer failure and display failure, since it would include (at least) display of one or more selectable telemetry words. Another factor, not shown in the table, is the benefit of the manual backup mode in facilitating checkout of the subsystem EOSE without special test fixtures.

The cost of adding manual mode capability is estimated at around 5 per cent of the cost of an STC, while that of a redundant computer plus switchover is estimated at around 20 per cent. For the baseline STC design the following conclusions have been reached:

a) A completely redundant computer system is excessively expensive for the backup capability provided.

Table A-2. Backup Tradeoffs

Parameter	No Backup	Redundant Elements	Manual Mode Backup
Delay in testing	Maximum, testing delayed until repair is effected	Depends on switch-over. If automatic, minimum delay (but automatic switch-over capability can itself contribute unreliability if series elements are required)	Intermediate. Testing proceeds at slower manual rate.
Utility of test results	None	Maximum, with exception noted above, since testing quickly resumed in normal mode.	Depends on discipline of operators in recording results. Data can (to an extent dependent on program design) be manually entered into computer test log off-line.
Cost	None	Maximum. Redundant element cost plus design and fabrication of switchover means, if automatic (possibility of sharing redundant elements between several STC's).	Intermediate. Depends on the computer functions which must be manually performed and whether the capability to perform them involves additional equipment.

b) Some degree of manual capability is highly cost-effective, since it will help respond to four of the five types of failures considered (computer hardware, software, EOSE-computer interface, and display).

c) The type of manual capability required includes:

- Telemetry decommutation and display — one word at a time is adequate, and conversion to engineering units (being inordinately expensive) should not be included.

- Command encoding — this is a requirement for MDE in any event and involves essentially no additional expense.

- Manual setup and display of stimulation and measuring instruments—this involves little additional expense; in general, it requires provision of switches and "or" gates to activate relays in EOSE which are normally picked by computer-operated discrete lines, and inclusion of numeric displays on digital voltmeters (A-D converters).

3. MISSION-DEPENDENT EQUIPMENT

The baseline STC configuration is comprised of the subsystem EOSE, including the command and telemetry subsystem OSE. A point which needs consideration is whether, in addition, a partial or complete set of mission-dependent equipment (MDE) and mission operations system test equipment (MOSTE) should be included in the STC for use during system test.

The reason for considering this implementation is the overriding importance of the MDE-spacecraft compatibility. The importance of continuing proof of compatibility is stressed in both the Voyager specification and the Mariner C specification. The Mariner C specification specifically requires mission-dependent equipment as part of the system test complex.

Since the Voyager program provides for a period of DSN compatibility testing of the engineering model spacecraft with the Goldstone DSIF, the question which should be posed is whether that test as a validation of MDE-spacecraft compatibility makes unnecessary (or not cost-effective) the use of MDE in system test operations.

The benefits derived from including MDE in the STC are:

a) The probability that using MDE with several different spacecraft in many different test environments will reveal interactions and real or potential problems in the MDE-spacecraft interface which would not be revealed in engineering model tests at Goldstone.

b) The hundreds of hours of additional operating time on the STC MDE's will furnish valuable experience relative to reliability, maintainability, and operability of the MDE. This should lead to substantially increased confidence in MDE availability during the period (of over one year) of the Voyager mission flight.

c) The increase in availability of the STC, since telemetry demodulation, command generation, and ground RF reception and transmission will be effectively backed up by redundant capability.

The above benefits appear to outweigh the slight increase in anticipated STC operating costs (added design costs are negligible, since the MDE modifications for STC use should be minor). Therefore, we have included the MDE in our baseline STC configuration.

APPENDIX B

FACTORS INFLUENCING CHOICE OF BASELINE LAUNCH COMPLEX EQUIPMENT CONFIGURATION

In determining the configuration to be adopted for the launch complex equipment (LCE), four major criteria have been applied:

a) The extent to which the LCE requirements (of Section 3.5.2) can be met.

b) The extent to which the flow of launch site activities are facilitated.

c) The extent to which additional equipment is required beyond that in the STC.

d) The extent to which the LCE meshes smoothly with the established operating procedures and equipment interfaces at Kennedy Space Center.

Of the four criteria, the last is probably the least well defined at this time, since some of the facilities to be used for Voyager are not yet fully defined, and detailed procedures and plans for using existing facilities for Voyager need greater definition.

In considering LCE design the basic problem is determining the optimum geographical distribution of STC equipment. As a percentage of an STC, the additional (LCE-peculiar) equipment needed to satisfy (a) and (b) above is quite small. In comparison with factory test operations, launch checkout imposes more severe limitations on access to the spacecraft (both electrical and by personnel) and presents new operating interfaces with capsule, launch vehicle, and launch operations personnel. From an electrical stimulus and monitor standpoint, there are almost no operations to be performed which are not inherent in the STC capability. Thus the general approach of using STC equipment, appropriately arranged and augmented with a small amount of LCE-peculiar equipment is completely feasible.

The most important implementation choices to be made within this general approach are the following:

a) STC's moved with spacecraft or stationary and connected via RF or data link.

b) If stationary, where should STC's be located.

c) Should test director's console remain with STC or be moved to LCC for VAB and on-pad phases.

d) RF test capability during RF silence (i.e., location of RF OSE).

These choices are not independent; conclusions in one area affect those following.

The question of whether the STC's should be moved with the spacecraft or remain stationary (with data link to spacecraft location) arises because the launch site operation plans require that complete or partial systems tests be run at the SAF (hangar AO), ESA, VAB, and pad. Although the STC must be capable of being transported and set up in a time equal to or less than that required to move and set up the spacecraft, it is clearly not desirable to complicate launch operations by moving the STC unless a net advantage occurs from doing so.

The determining factor in choosing a location is the relative time the spacecraft will spend at each site and the degree of testing and troubleshooting required at each. The testing and troubleshooting is important because of the difficulties of extensive hardline access at remote locations. Current launch site plans call for the three spacecraft to be located most of the time in an annex to the ESA, because the ESA is planned as a clean facility while the SAF is not. Thus, an ESA location would place the STC's near the spacecraft the greatest percentage of the time.

Still remaining is the question of whether to transport the STC's along with the spacecraft to the SAF, the VAB, and the mobile launcher or LCC. In the case of the SAF, the spacecraft are expected to be there only one at a time for short periods, and the testing there is basically an abbreviated checkout to verify that no transport damage has occurred.

During the period at the VAB, integrated system tests are conducted for which the STC hardline access would be of value; if the STC is remote, a modest data acquisition and transmission link is required to sample, convert, and transmit to the STC analog and discrete functions from the spacecraft umbilical and environmental instrumentation. In addition, a telemetry/command channel is required during radio silence (this requirement is discussed below in the section on capability during radio silence).

When the mobile launcher is at the launch pad, a VAB/LCC or ESA location of the STC is indicated. Thus, transmission of hardline data as described above would be a requirement, regardless of whether the STC's are moved or not.

Summarizing the various tradeoffs, moving the STC's eliminates the need for hardline data acquisition and transmission at the SAF and VAB but not when the mobile launcher is at the launch pad; therefore design costs for data transmission will be incurred in any event. Moving the STC's would eliminate the need for transmission of telemetry data and command signals over a data link, since RF silence during the short time at the pad would not be a problem. On the other hand, locating the STC's permanently at the ESA places them near all three Voyager spacecraft the majority of the time, and eliminates the reliability hazards of moving the STC and the cost of transportation and re-setup. The optimum course then (in light of the present launch operation plans) is to locate the STC's permanently at the ESA, moving only that equipment which must be near the spacecraft (power rack, data acquisition, and data link transmission equipment).

With the ESA as the "permanent" location of the STC's for all phases of Voyager operations at KSC, the choice to be considered now is whether to maintain the test director's console (TDC) at the ESA for all operations, or move it to the LCC for prelaunch and launch operations at the VAB or launch pad. A third possibility, more expensive in equipment, is to provide consoles at both the ESA and the LCC.

There are two major factors influencing the selection between the above alternatives. The first concerns the functional requirements for monitor and control by the spacecraft test engineer located in the LCC. The second concerns the equipment required to handle the data interface between the console and the rest of the STC over the distance from the ESA to the LCC.

The assignment of operational personnel tasks during prelaunch and launch are only tentative at this time; the present concept, however, calls for the spacecraft test director and his assistant to be equipped with the STC computer-driven displays, computer-entry keyboard, and other devices (printer, etc.). Another aid to the director will be consultation with the subsystem engineers; this can be via voice intercomm, although situations may arise for which personal contact is more advantageous. If he is at the STC, the spacecraft test director would, of course, be able to summon subsystem engineers from their monitor racks if desired. On the other hand, if he is at the LCC, he would be in contact with the boost vehicle launch director.

The STC interface between the central data system and the director's console is essentially a parallel data interface of between 50 and 100 lines, some of which (data input/output) are relatively fast (0.5 to 1 microsecond rise time required). It is clear that driving such a large quantity of wideband data lines over the relatively long distance from the ESA to the LCC is not a practical solution, and that parallel-to-serial transformation would be required to minimize the number of channels needed. In addition, the present CRT display memory plan calls for the display memory for all displays to be a modularized central buffer. A separate TDC buffer would be required if the TDC were remotely located. It appears clear that a substantial amount of equipment would be involved if the TDC is remotely located, and the commonality of the data entry and monitor rack and TDC interfaces with the computer would very probably be lost. Therefore, locating the test director's console at the ESA is indicated, since the benefits gained by locating the spacecraft test director at the LCC do not appear to warrant the added expense and complexity of such a remote interface.

An interfacing question is the type of facilities to be provided the spacecraft test engineer at the LCC. One task envisioned for the LCC engineer is answering to the boost vehicle launch director for spacecraft status. This requires he have reliable communications with the spacecraft launch director. To insure reliability an emergency communications channel redundant to the established facility channel is provided. A second task for the LCC engineer is emergency control and monitoring of both flight spacecraft. Assigning this task to the LCC engineer has two advantages. First, the shorter distance between the launch pad and LCC as compared with estimated pad-to-ESA distance, and the already provided launcher-LCC hardlines allow simpler, more redundant and more reliable means of spacecraft monitor and control. Second, the emergency control and the monitor point is close to the launch vehicle and capsule emergency control and monitor points and to the launch director.

The conclusion reached, then, is that the test director's console should remain at the ESA during launch; the spacecraft engineer at the LCC should be furnished with redundant communication capability to the spacecraft launch director at the ESA, and hardline emergency monitor and control to both spacecraft.

The choice to be considered in the radio subsystem OSE relates to the test capability which should be retained during RF silence. If the radio test set is kept close to the spacecraft (along with the ground power unit and the local data handling equipment), tests and monitoring during RF silence can include the spacecraft receiver and the transmitter functions. This then would provide for more complete testing of the spacecraft during RF silence. Coupling to the spacecraft is accomplished by means of relatively RF-tight hoods (with an internal pickup probe) to the spacecraft fastened to the shroud over the RF windows. The telemetry/ranging basebands can then be transmitted to, and the command/ranging basebands received from, the STC. However, a disadvantage is that a long video transmission system of 2 to 3 Mc bandwidth would be required.

The primary basis for the present configuration, which retains the radio test set at the STC, is that the frequency and duration of RF silence should be relatively limited and hence does not warrant the additional expense of a remote radio OSE-CDS data link. The telemetry data link provided for radio silence is therefore a narrow bandwidth link which handles only the basic 15 kilobits/sec telemetry data stream from the umbilical, rather than the telemetry baseband. Therefore, during RF silence the spacecraft radio subsystem would not be operated.

APPENDIX C

DATA ENTRY AND MONITOR RACK TRADEOFFS

The data entry and monitor rack (DE and MR) functions as the major EOSE test set-CDS interface device. The final configuration is to be such that the DE and MR can make requests for data from the computer and such that all computer data and status information can be displayed on the DE and MR without requiring additional signal processing capabilities. As such, the rack must be capable of display and request functions when separated from the subsystem EOSE. The rack should also be so implemented that a common rack interface is established with the CDS which does not limit data transfer operations or the number of racks being used. Signal conversion operations (i.e., parallel to serial) are accomplished within the individual EOSE items since these operations will vary in function between subsystem test sets.

1. UNIVERSAL ADDRESS INTERFACE

The basic operation of the DE and MR consists of delivering data to its own displays, to the associated subsystem EOSE, and to the CDS upon generation of the unique address associated with that data. This address can be implemented in either of two configurations. The first method (display-oriented) consists of generating an address peculiar to the data group being updated by a particular DE and MR register. This implementation requires that data to be transferred must be stored in individual memory locations which functionally correspond to individual subsystem display registers. Thus, memory slot N would contain 24 discrete bits of information which during each display updating cycle would be transferred to the corresponding 24 discrete display units on a particular subsystem test set panel. The second method (data-oriented) consists of storing data in the sequence of its generation. A particular set of 24 bits of information in this instance might contain discrete information sequentially obtained from the telemetry link which is to be transferred in total or in part to all subsystem EOSE display panels, the particular transfers involved being determined by routing address information associated with the stored data.

The display-oriented method simplifies the design of the EOSE logic and reduces the number of addresses used by each DE and MR. The basic disadvantage of this method, however, is that a greater amount of main-frame computer time is required to separate the incoming data into the desired display format. In the discrete operation previously discussed, for example, one memory word may require rewriting 24 times during a normal data reduction cycle. Another disadvantage is limited flexibility, since the relation between the memory and the display equipment becomes fixed early in the program. Changes in subsystem data requests thus become difficult to make and the addition of another data entry and monitor rack becomes almost impossible.

The data-oriented method therefore seems to be the better choice. Although increased logical complexity within the data entry and monitor racks is required, this disadvantage is overshadowed by the increased display flexibility and the decreased computer main-frame time requirements. Data storage duplication is also eliminated since any number of data entry and monitor racks can display the same data, provided only that they all contain the same address capabilities. Data display flexibility and expansion are almost unlimited with this approach, since all computer data is available to all subsystem test set racks. Additional data entry and monitor racks can be supplied at any time without changing the computer program.

2. GENERAL DESIGN TRADEOFFS

Each subsystem test set will include an interface unit for computer discrete control and display functions. These functions include such things as spacecraft telemetry status displays, spacecraft hardline status displays, test equipment status displays, computer status displays, and control status indicators. The unit as described is concerned only with discrete signal interfaces between the computer and subsystem EOSE.

Figure C-1 illustrates a modular approach to the interface unit design, consisting of three basic blocks: the address matrix, the status indicator units, and the control units. The proposed configuration is designed so

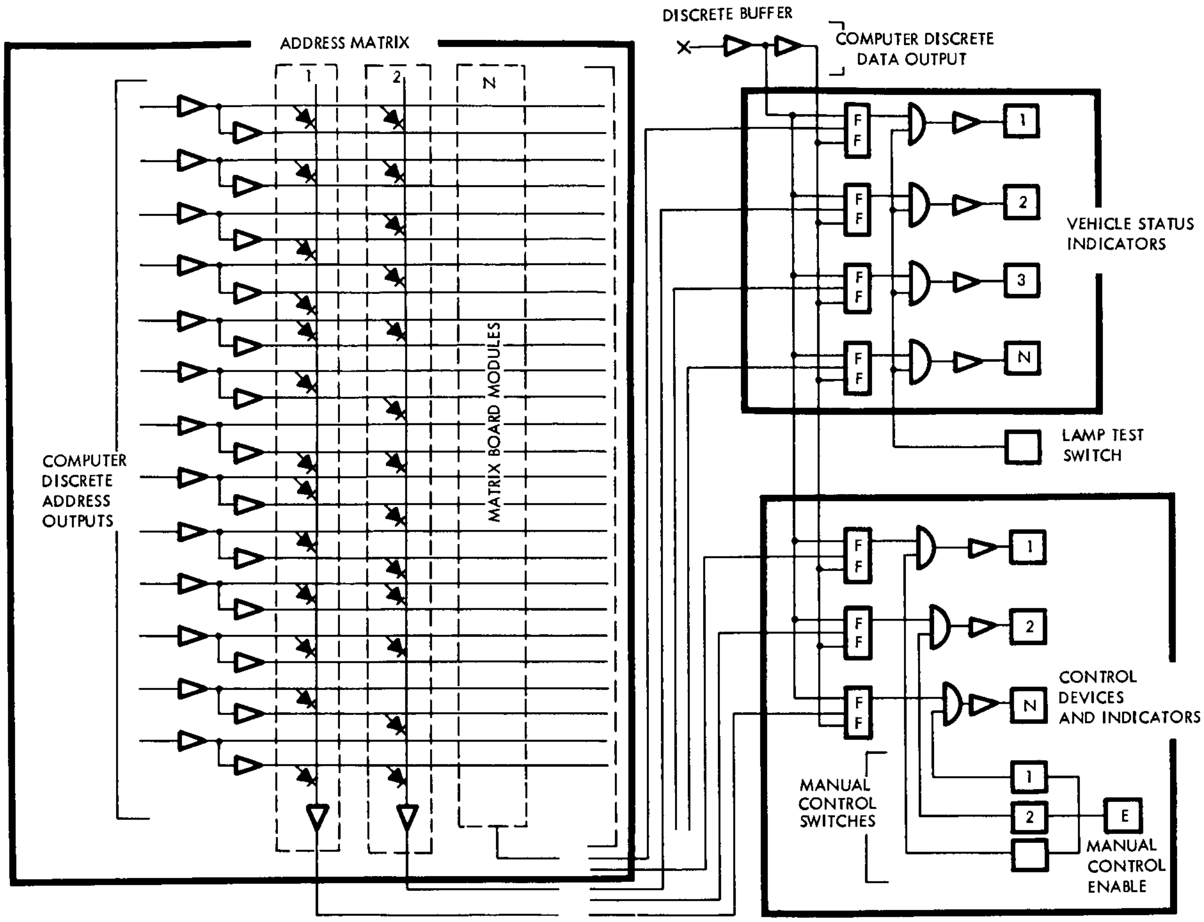


Figure C-1. Computer-Subsystem EOSE Interface Unit for Discrete Control and Display

that operational functions can be increased by adding additional matrix, display, or control cards. The basic unit therefore is identical for all subsystem OSE units. The matrix boards can be produced from the same basic card, requiring only a change in diode location for address reprogramming. An alternate approach consists of using universal matrix cards, with the required address programming within the drawer wiring. This method, however, would make each drawer unique and complicate the process of reprogramming specific address functions. The front panels of each subsystem unit can also have a common design. Unique display terminology (label inserts) would be supplied to each subsystem test unit in order to maintain the proper man-machine relationship. The control and display functions would be modularized so that the number of cards required would be proportional to the number of functions needed. This approach then requires design of only one interface drawer. The drawers become unique to their respective subsystems only when the required matrix boards and display legends are added to the assembly.

The address matrix consists of 24 buffer inverters and the required number of matrix board modules. The buffer inverters (see Figure C-1) form a 24-line matrix from the 12-bit computer discrete address inputs. Each individual matrix board module consists of 12 diodes and an output buffer. The diodes are placed on the individual boards so that an output gate pulse is generated when the computer input address matches the matrix diode configuration. The matrix board output signal is used to enable the appropriate display or command flip flop to update the appropriate device to the information contained on the computer discrete data output line. All of the discrete signals are time multiplexed on the discrete output line in synchronism with the address data on the computer discrete address lines.

The status indicator units consist of updating flip flops and associated amplifiers and indicator bulbs. Each individual flip flop is enabled by its associated matrix output signal and sets or resets itself to the data contained on the computer discrete data line. The relationships between address and data timing are not known at this time, but it can be assumed that one of the two signals will provide enable levels to clock the other. An OR gate at each flip flop output provides lamp test signal actuation. Card configurations can be designed so that one or two display units can be placed on one card. The chassis wiring is such that cards can be added or removed depending on the particular subsystem interface unit requirements.

The control units operate in the same manner as the status indicator units. A provision is made (OR gates) to allow manual control operations when the manual control enable switch is active. A priority system (AND-NOT gate) must be established for this switch such that manual operations cannot be performed during an automatic testing sequence.

The functional interface between the OSE and the central data system proposed for most controls and data transfer functions consists of initiating continuous computer I/O scan programs during which time all display and control functions, together with all computer input functions, are sequentially enabled and updated at a fixed multiplexing rate. This technique results in minimum programming complexity and OSE-computer cabling. The only disadvantage of this system lies in the fact that the actuation and control times will not be based on priority timing but rather segmented into finite time increments. These time increments, however, can be made quite small. Program alterations can also be provided, should greater time resolution be required for certain functions.

Figure C-2 illustrates the OSE interface unit configuration. The interfacing signals are composed of the computer parallel output lines (24 bits plus clock), the computer parallel input lines (24 bits plus clock), and the computer matrix address output lines (12 bits plus clock). All required functions can be accomplished with these lines, except for any special high rate input and output signals. The address matrix is used to decode the computer time-multiplexed address outputs to provide enable (control) signals to the selected device or register within the selected subsystem OSE test set. The control lines (one line for each 24 bits to be read into or out of the computer) enables their respective display, control, or data transferring device to update the data contained in the 24-bit parallel data input buffer, or to transfer information to the 24-bit parallel data output buffer. The computer data inputs therefore can be routed either to the digital printer, the time and test sequence display, the discrete status display, the discrete control devices, or the parallel and/or serial vehicle input devices.

Since the scan method of data transfer will be employed, the data acquisition rate depends on the entire cycle time. In most computers a total of five memory cycle times are required for a 24-bit word input transfer and four memory cycle times for a 24-bit word output transfer. For a memory cycle time of 2 microseconds, a 24-bit parallel input takes 10 microseconds, and a 24-bit parallel output takes 8 microseconds. As an example of multiplexing rate capabilities, let us assume that the follow-

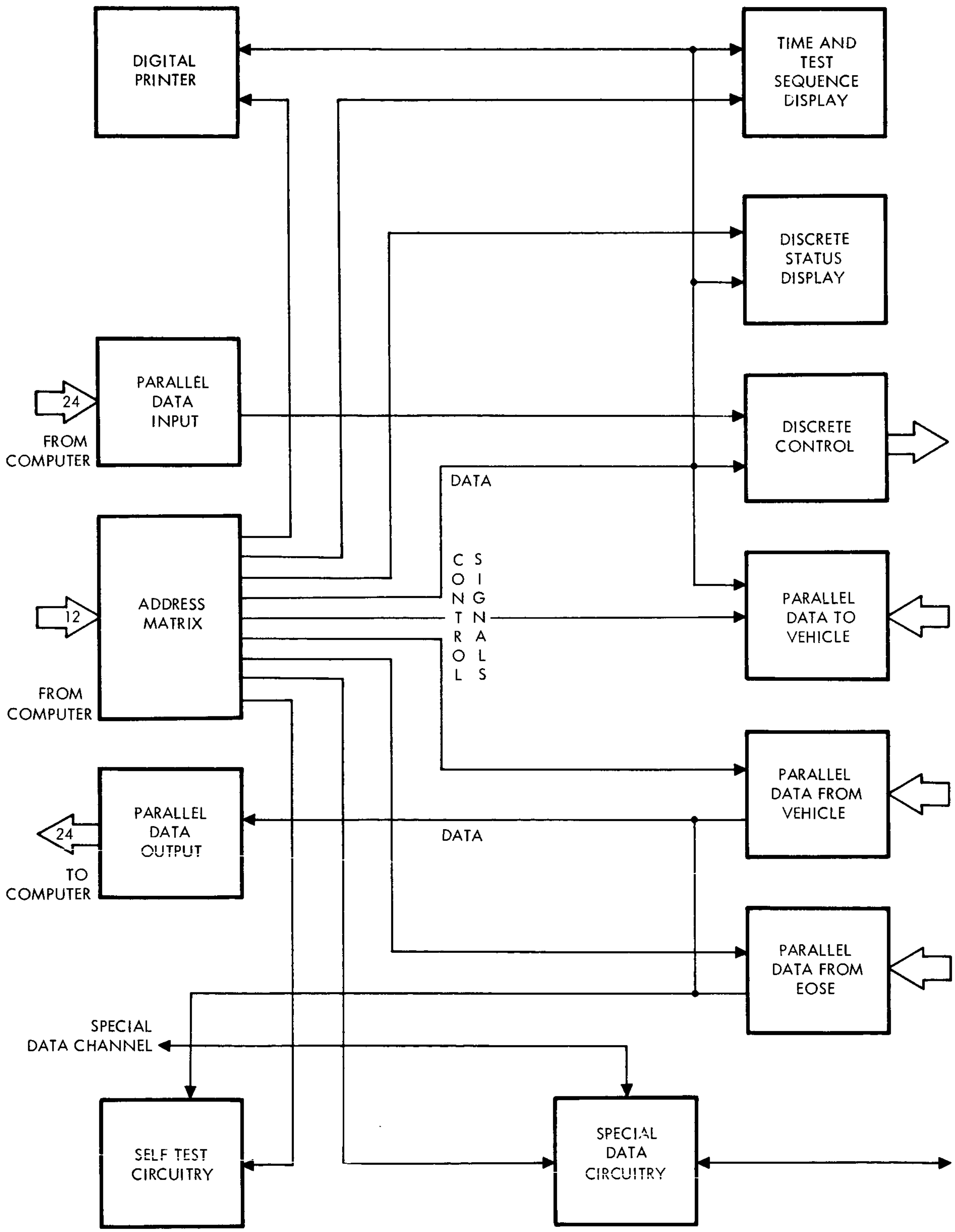


Figure C-2. Configuration of OSE Interface Unit ing signal transfers are required for a normal vehicle test phase. (PCM, recorder inputs, etc., are independent of this data multiplexing.)

Computer Inputs

240 discrete signals (as 24-bit words)	=	100 microseconds
20 data word inputs of 24 bits each	=	200 microseconds

Computer Outputs

240 discrete displays (as 24-bit words)	=	80 microseconds
96 discrete control operations (as 24-bit words)	=	32 microseconds
20 data word outputs of 24 bits each	=	160 microseconds

Total Cycle Time = 572 microseconds

Therefore, a complete cycle time of about 1 millisecond seems readily attainable, whereas even a cycle time of 5 milliseconds would probably be adequate for vehicle testing purposes.

3. DIGITAL DATA PRINTER

A digital data printer is included within each data entry and monitor rack to provide real-time hard copy outputs of specific subsystem data. Data selection for this device is display-oriented with a given memory location being unique to an individual printer. Data transfer to the printer is done in the normal scan process with one 24-bit data word containing four alpha-numeric characters (6 bits/character). The data entry and monitor rack electronics store the four characters and provide serial character outputs to the digital printer upon printer control requests. The printer itself must be of high speed to reduce computer storage requirements. A tentative operating speed of 1000 characters per second has been selected to allow the printer to track in real time the majority of data inputs. Other printer-imposed requirements of a general nature include high reliability and low noise generation.

4. CRT DISPLAY AND DATA ENTRY UNIT

A commercial TV display and keyboard entry device is included for presenting data. This device allows each subsystem test engineer to select for display any information contained within the CDS computer memory. The device therefore becomes a universal interface device and is in no way functionally oriented to a particular subsystem or system operational task.

A storage buffer is required for regenerative display operations. This requirement is imposed to limit the amount of main-frame computer time necessary to update and maintain these display units. The keyboard section of this device is used as the main communication path from the test personnel to the CDS computer. Data display formats, simulation generation requests, and test sequence requests are handled by this keyboard unit.

APPENDIX D

AHSE APPLICABLE DOCUMENTS

TRW Systems Process Specification	PR 8-1
Federal Standard 209	"Clean Room and Work Station Requirements, Controlled Environment"
MIL-HDBK-5	Safety Margins
MIL-A-8421	Ultimate Load Factors
MIL-STD-129	"Marketing for Shipment and Storage"
MS-33586	"Metals, Definition of Dissimilar"
MIL-D-3716-A, Amend. 2	"Desiccants, Activated, for Dynamic Dehumidification"
MIL-E-5556-A, Amend. 1	"Enamel, Camouflage, Quick Dry"
MIL-M-008090	"Mobility Requirements, Ground Support Equipment, General Specification for"
MIL-C-13777-D, Amend. 1	"Cable, Special Purpose, Electrical, General Specification for"
MSFC-SPEC-164	"Cleanliness of Components for Use in Oxygen, Fuel, and Pneumatics Systems, Specification for"
PPP-B-621-A, Amend. 2	"Box, Wood, Nailed and Lock Corner"
MIL-D-3464-B	"Desiccants, Activated, Bagged, Packaging Use and Static Dehumidification"
MIL-C-9959, Amend. 1	"Container, Flexible, Reusable, Water-Vapor Proof"
MIL-B-26195-A	"Boxes, Wood Cleated, Skidded, Load Bearing Base"
PPP-B-601-A, Amend. 2	"Boxes, Wood, Cleated, Plywood"
MIL-P-116-D	"Preservation, Methods of"
PPP-B-636-C	"Box, Fiberboard"

MIL-P-9024-B and C	"Packaging, Air Weapons Systems, Specifications and General Design Requirements for "
MIL-STD-1186	"Cushioning, Anchoring, Bracing, Blocking, and Water-Proofing, with Appropriate Test Methods"
ICC Tariff No. 15	"Regulation for Transportation of Explosives and Other Dangerous Articles"
A.F. Manual 71-4	"Packaging and Handling of Dangerous Material for Military Aircraft"
MIL-B-131	"Barrier Material, Water-Vapor-Proof, Flexible"
MIL-P-27401-B	"Propellant Pressurizing Agent, Nitrogen"
AFETR P80-2	"General Range Safety Plan." Volume I and associated Appendix A

APPENDIX E

TRANSPORTATION TRADEOFFS

The maximum allowable Voyager spacecraft envelope, as specified in JPL "Voyager 1971 Preliminary Mission Description," October 15, 1965, is 20 feet in diameter and 17-1/3 feet in length. The maximum weight of the assembled 1977 spacecraft, less propellants and gas, will be approximately 6400 pounds. The shipped configuration is further increased beyond the dimensions of the spacecraft by an enclosing shipping container, which provides environmental control (e.g., temperature, humidity, and cleanliness), and shock attenuation to protect the spacecraft from transport and handling loads. The entire shipped configuration may approach a weight of 9400 pounds and assume the shape of a cylindrical body 260 inches in diameter and 19 to 24 feet long. Because of spacecraft structural, dynamic, and static-load design criteria adopted in this study, it was decided that shipping the spacecraft horizontally would complicate the transportation shock and vibration load mitigation systems. A vertical shipping attitude was therefore used as a constraint, and imposed the maximum size package for consideration.

The weight involved does not create major problems; the dimensions do. But these dimensions are so similar to those of the Saturn S-IVB interstage structure that the direct use of Saturn S-IVB stage-handling and transportation equipment and procedures appears feasible.

Long distance or cross-country transportation requirements can be satisfied only by air or water shipment; the Saturn program demonstrated that cross-country overland transportation of this size load is impractical.

Cross-country air transportation for the Voyager program operational period can rely on the Super Guppy aircraft, or possibly the C-5A if it is operational. The Super Guppy presumably will have been completely proven and qualified, with procedures and equipment established. Accordingly, further discussion of this mode appears unnecessary.

Similarly, shipment by waterway of the packaged spacecraft may rely on existing NASA-operated naval craft, such as the AKD presently equipped and used to ship Saturn flight stages. Furthermore, sea shipment of the spacecraft for short distances between facilities located near the southern California coast is also possible using NASA-operated barges or the AKD. However, this would only provide a partial solution to the short-distance logistic problem and would require employment of additional transport modes between the sea-carrier loading dock and the spacecraft assembly or test facilities, some distance inland.

The local short distance transport of point-to-point movement between facilities can be accomplished only by surface (over-the-road) movement or air (by helicopter airlift), since neither water shipment nor conventional aircraft can satisfy the facility-to-facility movement requirements. Accordingly, the former two modes were evaluated in this study. Either method is considered feasible and each could additionally provide movement between assembly and test facilities and the loading points for cross-country carriers.

More detailed study is required to establish precise quantitative comparison figures, particularly with respect to cost, development, and qualification requirements. Costs for road transportation can be estimated rather closely from previous experience, but the helicopter-transport costs depend on the type of equipment available and the specific service contract arrangements with the carrier. Since contractual arrangements can vary extensively, the costs cannot be estimated at this time, except over a broad range of possibilities, and the development, equipment, and procedures qualification requirements can be predicted accurately only after detailed design and test planning have proceeded further. However, on the basis of the comparisons made to date, the helicopter transport mode has several distinct advantages. These advantages are shown on the summary matrix chart (Table E-1) and its accompanying discussion.

Although many other factors may be compared in evaluating the candidate transport modes, the factors evaluated appear to be the major areas of impact on a final selection. These preliminary studies must be continued, but they currently indicate a considerable margin of preference for the helicopter-transport mode.

Table E-1. Summary Matrix Chart

	Desirability Rating		Relative Weight Factor	Summary Rating	
	Land	Air		Land	Air
Cost (summary of (a) to (f)	3	5	10	30	50
a) **Equipment required**	(3)	(8)	(10)	(30)	(80)
b) **Time and manpower**	(2)	(9)	(6)	(12)	(54)
c) **Route preparation**	(2)	(10)	(8)	(16)	(80)
d) **Special permit fees**	(5)	(7)	(1)	(5)	(7)
e) **Leasing or rental contract**	(10)	(0)	(2)	(20)	(0)
f) **Development and qualification**	(8)	(4)	(8)	(64)	(32)
Route preparation problems	2	10	8	16	80
Time factors	2	9	3	6	27
Equipment complexity	4	7	6	24	42
Development and qualification requirements	9	2	8	72	16
Summary Total Desirability Rating				148	215

Notes:
1) The number assigned to the desirability of each mode, plus the parametric weighting of each comparison factor, are best estimates at this time.
2) Assigned numbers for relative desirability (desirability rating) or degree of importance (relative weight factor) indicate greater desirability or importance by higher numerical values on a 0 to 10 scale.
3) Numbers in parentheses (shaded area of chart) show rating of subelements of the basic comparison factor and do not directly appear in the totals.
4) Summary ratings are reached by multiplying the desirability rating by the relative weight factor.

1. COST

Cost elements can be grouped into the following main contributing categories: (a) amount of equipment required; (b) time and manpower expended in support of each move; (c) route preparation and maintenance; (d) fees for special permits; (e) leasing, rental, or service contract costs for a commercial carrier; and (f) development and qualification requirements. Problems associated with these contributing factors beyond the immediate aspects of cost are further discussed in subsequent sections.

As detailed in Section III of this volume, the amount and complexity of AHSE required for road transportation far exceeds that predicted for helicopter airlift, even assuming the use of developed Saturn road transport and handling equipment. If entirely new Voyager road equipment is designed, developed, and qualified, the cost differential will be even greater.

Time and supporting manpower required for road shipment will exceed that required for airlift. Airlift will require only ground crews at each terminal point to load and unload the helicopter. Airlift transit time for anticipated distances will usually be less than one hour. Manpower required for road shipment includes escort crews, maintenance personnel, and transport crews accompanying each shipment, along with loading and unloading crews at the terminal points. Each road shipment may require up to 24 hours because of conflicting ordinances concerning the legal hours of oversize movement.

Ground-route preparation and maintenance for moving loads of this size has proven to be costly in the past. Preliminary surveys, relocation of structures and utilities along the selected route, and continual resurvey and maintenance during the course of its use are required, and in congested urban areas generally costs several thousand dollars per mile. Comparatively, the costs for preparing an air route will be low, consisting mainly of landing-pad preparation at each site. Existing paved apron areas at the various assembly and test facilities may be used directly, or with little modification.

Special permits must be obtained from each political division transversed each time an oversize load is transported across public highways. Costs of these permits are usually not excessive, but do involve the additional costs of planning and acquisition. It is uncertain at this time whether special permits would be required from political entities for permission to overfly their geographical areas with a helicopter carrying a suspended load of this nature, but it is assumed that they would not.

The required road-transport-equipment group discussed in Section III is assumed to be Voyager equipment and, therefore, not subject to any leasing, rental, or service-contract costs. Conversely, the transporting helicopter will probably not be part of the Voyager program equipment inventory. Therefore, costs will be incurred from renting, leasing, or negotiating a service contract with a commercial helicopter carrier. Preliminary talks with Los Angeles Airways (which presently is doing similar jobs) have suggested that such fees would be moderate. However, these costs have been difficult to ascertain with any degree of accuracy, because they are entirely dependent on the type of service-contract required.

Assuming direct application of existing Saturn transportation and handling equipment, costs for development and qualification may be considered moderate. On the other hand, it is expected that considerable development and qualification costs will be incurred in the use of helicopter airlifting. These costs include performance demonstration flights using dummy loads. It should be kept in mind, however, that before 1971 considerable development and qualification of the basic transport mode will probably have been completed.

In sum, the above points suggest that the total cost of transporting the Voyager spacecraft by helicopter airlift will be considerably less than by road during the 1971 to 1977 period.

2. ROUTE PREPARATION PROBLEMS

Before the shipment of an oversize load by highway, preliminary surveys of all potential routes must be conducted in cooperation with representatives from all affected public-utility and private companies and with representatives of city, county, and state agencies. Previous experience has shown that selected routes require extensive alterations and must be continually resurveyed and traversed prior to every shipment to ensure that no alterations (e.g., construction work, new utility installations) have occurred between the time of original route preparation and actual shipment. Freeways usually cannot be used for transporting oversize loads because of restrictive overhead clearance of signs and overpasses, limited turn radii of on- and off-ramps, and the necessity of blocking several lanes of traffic simultaneously. Additionally, in California, conflicting ordinances govern transport of oversize loads: state highways or state sign routes may be used only from sunrise to sunset, whereas county and city roads and streets may be used only from midnight to 6 a.m. These constraints pose difficult scheduling problems. Virtually every foot of the selected route must be checked to determine the feasibility of passage. In most cases, a transporter carrying a dummy load must be periodically dispatched to re-establish route suitability. In some cases, guide lines must be painted on the road surfaces to guide drivers during actual shipment. Additionally, scheduled bus service on selected routes may require rerouting during the shipping period, and other traffic temporarily detoured or stopped. Local ordinances also require the escort services or police, fire protection, and maintenance vehicles during shipment of loads exceeding 18 feet in width and 18-1/2 feet in height. There are numerous other detailed problems and procedures which further complicate road shipment.

Route preparation associated with helicopter airlift will consist essentially of FAA approval of the selected flight path, and any overfly approval permits required of local political entities. In addition, as previously discussed, there will probably be some moderate facility preparation at the point of helicopter load pickup or landing.

3. TIME FACTORS

The time required for road shipment, as suggested above, may be as long as 24 hours or more for short-distance trips, because of conflicting state and local ordinances as well as restrictive speed limits. Although traffic ordinances may permit speeds of up to 45 mph, experience gained on other programs has shown that speeds of 10 to 20 mph are maximum because of shock-load limits and speed limitations imposed by route negotiation problems.

Helicopter-transport time, from pickup to touchdown of the load, will seldom exceed one hour for the local transportation requirements anticipated for Voyager. If test and other operations schedules are critical, it is obvious that the airlift mode has distinctive advantages over the surface mode.

4. EQUIPMENT COMPLEXITY

Section III indicates the equipment required for both the land-transporter and the helicopter-transporter groups. It is evident that considerably more program equipment will be required for land then for air transportation. In fact, the helicopter-transportation group supplied as program equipment consists essentially of a simple rigging kit and an airborne instrumentation unit consisting of standard commercial components packaged for airborne operation. Land transportation equipment includes a transporter, shipping container adapter cradles, shipping container handling rings, a shipping container hoist kit, an instrumentation trailer, a generator trailer, an air conditioning unit, and a tractor prime mover. Compared to the helicopter transport group, these equipment items are large and complex.

5. DEVELOPMENT AND QUALIFICATION REQUIREMENTS

Again assuming the use of Saturn equipment modified as necessary for Voyager application, only minor development and qualification of the road equipment will be required. Such development and qualification will consist essentially of a functional demonstration of the modified equipment's capability to transport the Voyager spacecraft in a safe manner. If entirely new Voyager-program land transport AHSE is defined, the development and qualification of this equipment will be extensive and will exceed that required for the proposed helicopter-transportation AHSE group in time and cost.

Some new development and qualification of the helicopter transport equipment will be required. However, if similar application of helicopters to this type of service has been commercially developed prior to 1971, it is probable that only the Voyager load-rigging and suspension system will require a development and testing program. Operational procedures will require development, and perhaps the public will need education to accept overflights of helicopters carrying suspended loads in congested urban areas. Thus, unless considerable previous development of similar helicopter application has occurred it must be assumed that a greater development effort will be required for the helicopter concept, especially if ground-transportation requirements can be satisfied by already qualified equipment.